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ORBIT TRANSFER VEHICLE
ENGINE STUDY, PHASE A - EXTENSION
(Volume I: Executive Summary)

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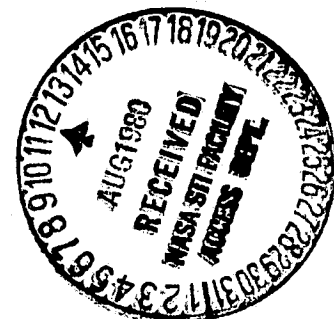
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prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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ORBIT TRANSFER VEHICLE

ENGINE STUDY, PHASE A - EXTENSION

(VOLUME I: EXECUTIVE SUMMARY)

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FOREWORD

This report presents the results of studies conducted during the Extension Period of the Orbital Transfer Vehicle Engine Study, Phase A - Contract. Engine trade studies and system analyses leading to a baseline engine selection for the Advanced Expander Cycle Engine Point Design Study were conducted by Rocketdyne, a Division of Rockwell International, under contract NAS8-32996--administered by Marshall Space Flight Center of the National Aeronautics and Space Administration (NASA). The NASA Contracting Officer's Representative and Alternate Contracting Officer's Representative were Mr. Dale Blount and Mr. James F. Thompson, respectively. Mr. H. G. Diem was the Rocketdyne Program Manager. The technical effort was conducted under the direction of Mr. A. J. Sobin, Study Manager.

The Final Report is submitted in three parts:

- Vol. I , Executive Summary
- Vol. II-A, Study Results
- Vol. II-B, Programmatic and Cost

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INTRODUCTION AND SUMMARY

The Orbital Transfer Vehicle (OTV) in which the oxygen/hydrogen OTV engine will be utilized is part of the planned space transportation system (STS). The OTV, carried into a low earth orbit by the Space Shuttle Orbiter, will extend the operating regime of the STS beyond the capability of the basic Space Shuttle, to include higher altitude orbits, geosynchronous orbits, and other space missions. For the past several years, NASA, the Department of Defense, and various vehicle and propulsion system contractors, including Rocketdyne, have been studying OTV-related requirements, missions, vehicle configurations, and engine systems for this application. The goals of the vehicle and propulsion system studies have been the definition of and technology development for a high-performance, low operating cost, reusable long-life space system with the required operational versatility and payload retrieval capability. The basic OTV user requirements for the OTV propulsion system are shown in Fig. 1, together with an artist conception of the vehicle.

Two concepts have been defined by NASA as potential designs for the Orbital Transfer Vehicle: these are the all-propulsive orbit transfer vehicle (APOTV) and the aero-maneuvering orbit transfer vehicle (AMOTV). The baseline design of these preliminary studies had assumed an uprated shuttle orbiter with 100,000-pound payload capability to low earth orbit. As an interim vehicle, the current 65,000-pound payload capability shuttle orbiter would be used.

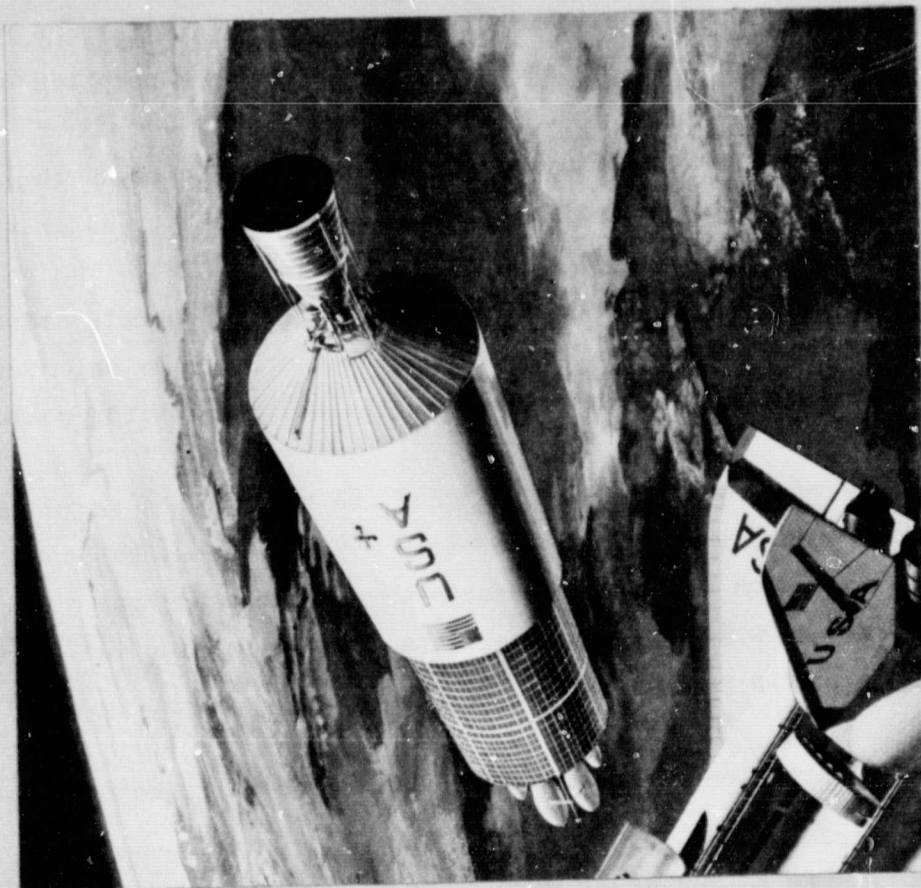
The baseline mission defined by NASA for the OTV Phase A Engine Studies was a four-man, 30-day sortie from low earth orbit to geosynchronous orbit and return to low earth orbit. A reusable oxygen-hydrogen upper stage with the high performance which these propellants offer is essential to achieving a defined round-trip payload of 13,000 pounds.

The Phase A Extension OTV Engine Study performed by Rocketdyne has resulted in:

- (1) Performance optimization of advanced expander cycle engines in the 10 to 20K-pound thrust range.
- (2) Selection of a recommended advanced expander engine configuration based on maximized performance and minimized mission risk, and definition of the components for this configuration.

HIGH THRUST OTV USER REQUIREMENTS

- HIGH PERFORMANCE
- OPERATIONAL FLEXIBILITY
- MULTIPLE REUSE
- QUICK TURNAROUND
- MANRATED
- MINIMUM COST



SC235-617A

Figure 1

- (3) Characterization of the low thrust adaptation requirements and performance for the staged combustion engine,
- (4) Generation of a suggested safety and reliability approach for OTV engines independent of engine cycle,
- (5) Definition of program risk relationships between expander and staged combustion cycle engines, and
- (6) Development of schedules and costs for the DDT&E, production, and operation phases of the 10K-pound thrust expander engine program.

The Phase A Extension OTV Engine Study concluded that an advanced expander cycle engine is a viable contender for selection as the OTV main engine. This appraisal of the expander is based on performance optimization, reliability and development risk assessment, and on a comparative cost evaluation. In addition, the gain in simplicity of the expander cycle engine configuration over the staged combustion engine configuration and the ability to directly apply to the expander the technology developed from the staged combustion engine programs make the advanced expander an attractive choice for OTV propulsion.

The expander engine configured in Phase A Extension (Task 8) studies formed the initial baseline on which studies being conducted under the Advanced Expander Engine Point Design Contract (NAS8-33568) are based. Detail component analysis and design under the latter contract has resulted in updated component performance efficiencies and, as a consequence, updated engine operating parameters.

To accomplish the effort described in the Statement of Work (SOW), the following five technical study tasks were defined:

- Task 8 Advanced Expander Cycle Engine Optimization
- Task 9 Alternate Low Thrust Capability
- Task 10 Safety, Reliability and Development Risk Comparison
- Task 11 10K Expander Engine Programmatic
- Task 12 Vehicle Systems Studies Support

This report is organized by study tasks and summarizes the results of these tasks. The work was performed during a 9-month period from July 1979 through May 1980.

TASK 8. ADVANCED EXPANDER CYCLE ENGINE OPTIMIZATION

OTV ENGINE STUDY OBJECTIVES AND GROUND RULES

The OTV Engine Phase A Extension Study has as one of its objectives the definition of a performance optimized engine system for an advanced expander cycle engine. The scope of this study included the in-depth analysis and design efforts necessary to accurately provide definition of engine and component configurations, fabrication procedures, weight and envelope requirements, control system technique, turbopump performances, and areas of technology development. The design philosophy employed during the study maintained and built upon the inherent simplicity, reliability, and system safety furnished by the basic expander concept.

The requirements for the main propulsion system are summarized in Table 1. The main engine is defined as a high performance bell nozzle, pump-fed system utilizing oxygen/hydrogen propellants and 1980 technology. The maximum retracted length is 60 inches. The engine must be man-rated with crew safety (shuttle and OTV) and abort return capability important considerations. For the mission defined, thrust optimizes at a value of 15,000 pounds and mixture ratio at a value of 6:1. It is desirable that the engine be adaptable through component kitting to extended low thrust operation at 1800 pounds thrust. The engine must also be capable of mixture ratio excursion between 6 and 7. Life, gimbaling, and other operational requirements are also listed in Table 1. Structural design criteria are provided in Table 2.

ENGINE OPTIMIZATION

The process of selecting component concepts, subsystem arrangements and final engine configuration was accomplished through the extensive use of optimization, parametric, and trade studies. Optimization studies provided the means for maximization of engine power within current state-of-the-art hardware concepts and led to the selection of optimized engine geometry,

TABLE 1 ENGINE REQUIREMENTS

- EXPANDER CYCLE
 - HIGH PERFORMANCE O_2/H_2
 - BELL NOZZLE, PUMP-FED SYSTEM
 - 1980 TECHNOLOGY
 - 60-INCH RETRACTED LENGTH
- MAN RATED/SHUTTLE CARGO
- OPERATING CHARACTERISTICS
 - MIXTURE RATIO, 6:1 TO 7:1
 - THRUST (VACUUM), 15,000 POUNDS
 - IDLE MODE
 - AUTOGENOUS PRESSURIZATION
 - LOW NPSH (2 FEET O_2 , 15 FEET H_2)
 - GIMBAL:
 - +15 } DEGREE PITCH PLANE
 - 6 }
 - ±6 DEGREES YAW PLANE
 - CHAMBER PRESSURE OSCILLATIONS, ±5%
- LONG LIFE
 - SERVICE FREE
 - 60 CYCLES OR 2 HOURS
 - BETWEEN OVERHAULS
 - 300 CYCLES OR 10 HOURS

TABLE 2 STRUCTURAL REQUIREMENTS

- NO NET SECTION YIELDING
 - 1.1 X LIMIT LOAD
- NO FAILURE
 - 1.4 X LIMIT LOAD
- NO FAILURE PRESSURE VESSELS
 - 1.5 X LIMIT PRESSURE
- LIMIT LOAD: MAX. PREDICTED
 - RESULTANT LOAD DURING DESIGN LIFE
- STRUCTURE CAPABILITY
 - LIMIT LIFE X 4
- PRESSURE VESSELS PROOFED TO 1.2 AT DESIGN
 - ENVIRONMENT CONDITION

cycle configuration, and delivered payload. The configuration optimization areas are shown in Table 3, which describes studies in turbomachinery, coolant routing and engine control. Parametric studies include detailed power and flow balances at thrust levels of 10,000, 15,000, and 20,000 pounds thrust for engine mixture ratios of 6.0, 6.5, and 7.0, and area ratios of 200 to 1400.

While the parametric studies provided performance trends with engine geometric parameters, the trade studies provided hardware configurations conforming with 1980 technology. Each component was assessed by the trade disciplines listed in Table 4 to determine the most applicable concept for use in the expander engine.

POWER CYCLE CONFIGURATION OPTIMIZATION

Four major areas of power cycle configuration optimization were examined: main turbine arrangement, cycle energy source, high pressure pump design, and boost pump drive. Cycle configurations resulting from the first two areas were parallel main turbines, series main turbines, turbine gas regeneration, and turbine gas reheat. The parallel and series turbine cycles are illustrated in Fig. 2. These cycle configurations were evaluated in conjunction with conventional combustors and with thermally-enhanced combustors.

Main Turbine Arrangement: For the main turbine arrangements of series and parallel, an engine cycle optimizer code was allowed to optimize engine specific impulse under the following constraints: turbine/pump geometry, and bearing DN compatible with 1980 technology; combustor length limits of 10 to 20 inches; coolant jacket bulk temperature limits of 1000 and 1260R, respectively for combustor and nozzle; and certain flow and pressure drop margins. The optimizer was allowed to vary nozzle expansion area ratio, pump speed, chamber pressure, pressure ratio, and combustor length to arrive at an optimum delivered specific impulse and a corresponding chamber pressure. Results for the series versus parallel turbine case, for a thrust level of 15,000 pounds, can be seen in Fig. 3.

**TABLE 3. OTV Expander Cycle
Configuration Optimization**

- TURBOMACHINERY
 - LOW PRESSURE PUMP DRIVE
 - GASEOUS H₂
 - HYDRAULIC
 - GEARED
 - MAIN TURBINE ARRANGEMENT
 - SERIES vs PARALLEL
 - TURBINE GAS REHEAT
 - TURBINE GAS REGENERATION
 - MAIN PUMP TYPE
 - 1 - 3 STAGES, CENTRIFUGAL
- THRUST CHAMBER COOLING
 - SPLIT-FLOW REGENERATIVE
 - FULL-FLOW REGENERATIVE
 - DUMP COOLANT PRESSURE
- ENGINE CONTROL
 - THRUST
 - MIXTURE RATIO
- IDLE MODE HEAT EXCHANGER LOCATION

Table 4. Component Trade Areas

- PERFORMANCE
- STATE-OF-THE-ART ASSESSMENT
- CYCLE LIFE
- MANUFACTURING AND HANDLING
- COMPLEXITY, SAFETY, RELIABILITY
- RESPONSE, START LOSSES
- OFF-DESIGN OPERATION
- COSTS, DEVELOPMENT AND PRODUCTION
- MAINTAINABILITY

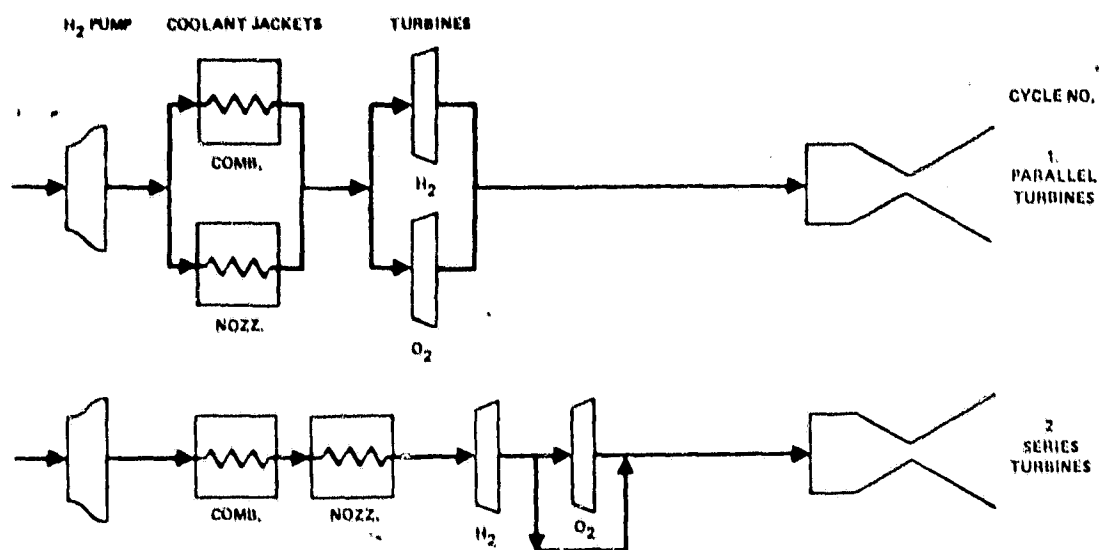


FIGURE 2. Expander Cycle Main Turbine Arrangements

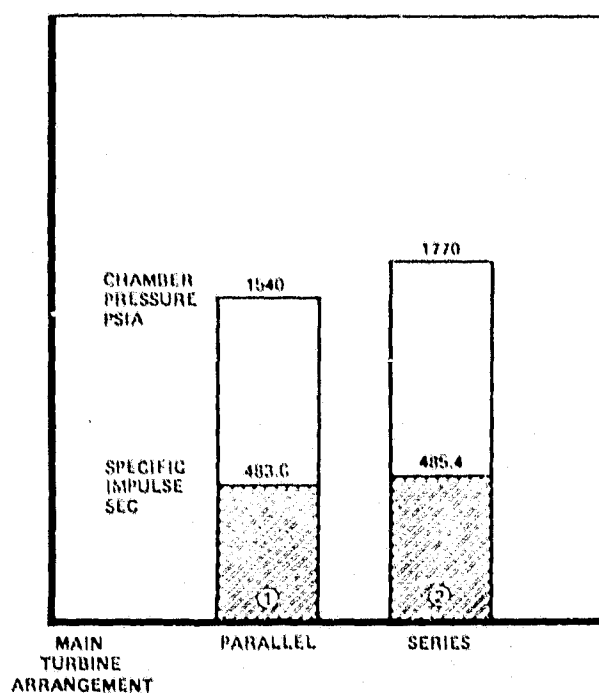


Figure 3. Chamber Pressure and Specific Impulse for Parallel and Series Turbine Cycles.

The series turbine arrangement attains approximately 230 psi higher in chamber pressure than the parallel turbine arrangement. The reason for this is that more flow is available to each turbine in the series configuration than in the parallel arrangement. The higher flowrate allows higher turbine power. It allows also higher volume flow in the turbines and, therefore, improved turbine efficiencies and higher turbine arc of admission. The higher chamber pressure in the series turbine arranged engine results in higher nozzle expansion area ratio and thus approximately 2 seconds higher specific impulse.

Turbine Gas Reheat and Regeneration: The same optimization procedure and constraints used in the main turbine arrangement studies were used to optimize the turbine gas reheat and turbine gas regeneration concepts (Fig. 4) utilizing a thermally enhanced chamber. Turbine gas regeneration uses a heat exchanger to accumulate heat in the driver fluid (hydrogen) thus allowing a higher turbine temperature for the same coolant jacket heat input to the hydrogen. The turbine gas regenerator was placed immediately upstream or downstream of the combustor coolant jacket. Turbine gas reheat provides a means of boosting the driver fluid temperature after a turbine prior to entry into the next turbine. Results shown in Fig. 5 indicate that the turbine gas reheat cycle (No. 3) attains approximately the same chamber pressure and specific impulse as the turbine gas regeneration cycle (No. 4). The optimization of each of these two cycles results in very high and nearly identical nozzle expansion area ratio, and nozzle length; and therefore, similar specific impulse. The chamber pressure is determined by the maximum combustor coolant jacket hydrogen bulk temperature limit of 1000 R.

Since both configurations have thermally enhanced combustors and combustor length is allowed to increase in both, the bulk temperature limits of 1000 R are reached in both configurations, resulting in equal chamber pressures. With nearly equal combustor length in each configuration, equal specific impulse is obtained with equal total length restraints (112 inches extended).

With the regenerator placed between the combustor and nozzle coolant

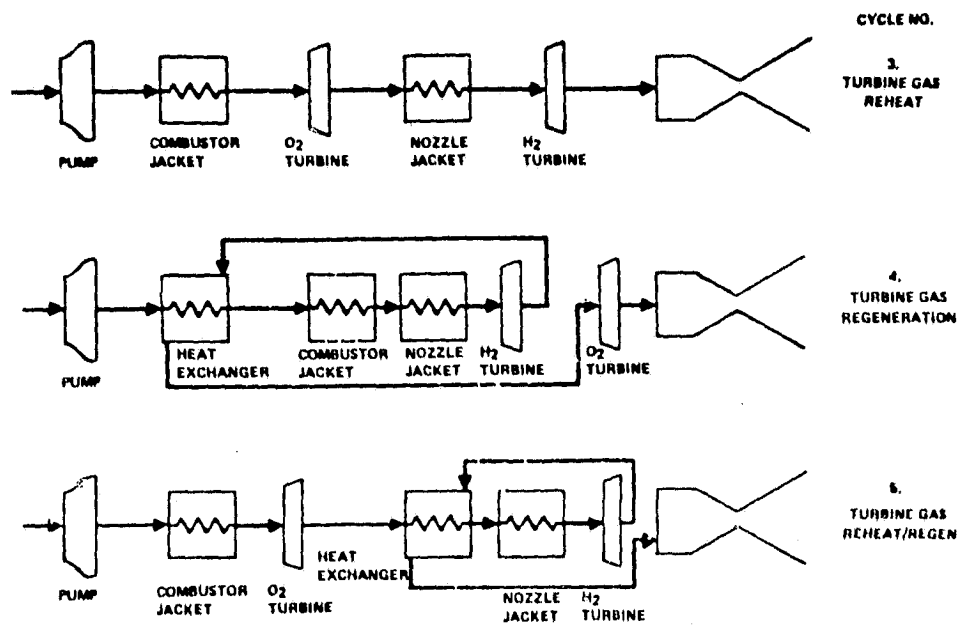


Figure 4. Expander Engine Turbine Gas Reheat and Regeneration Schematics

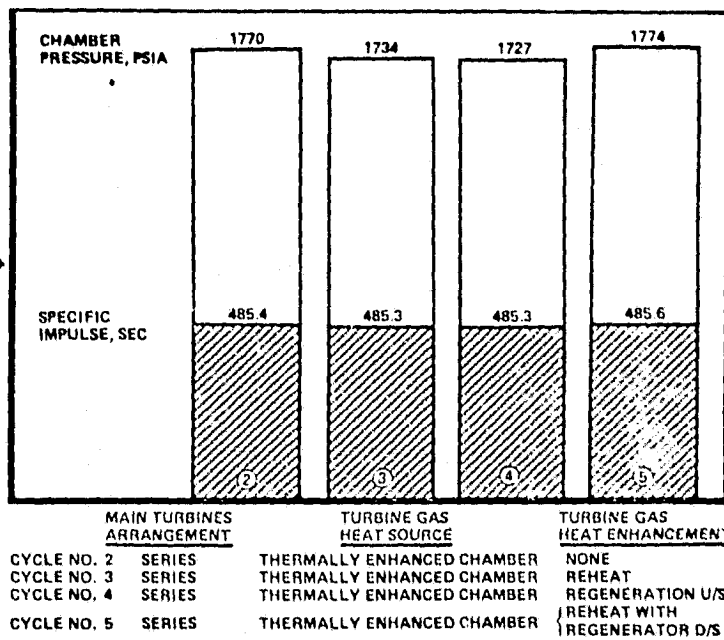


Figure 5. Main Turbine Arrangement and Turbine Gas Reheat and Regeneration at 15,000 Pounds Thrust

jackets, cycle No. 5, the chamber pressure attained is 1774 psia, 38 psia higher than cycle No. 4. Regeneration is performed in the case of cycle No. 5 with the nozzle coolant jacket and since this component has higher bulk temperature limits, the turbine inlet temperature attained is 1225 R, 100 degrees higher than in cycle No. 4. The higher chamber pressure results in higher expansion area ratio but only a slightly higher specific impulse (0.1 sec). The latter is the result of the nearly flat trend of specific impulse with area ratio at these high chamber pressures, and the fact that specific impulse at a given area ratio does not increase significantly with chamber pressure.

From the above examples, it follows that the maximum hydrogen bulk temperature prior to entry into the turbine determines the chamber pressure for a turbine flow limited cycle like the expander topping cycle. The bulk temperature limit is set by the wall temperature limit of the component just upstream of the turbine and is reflected in the hydrogen bulk temperature limits of 1000 and 1260 R, respectively, for the combustor and nozzle coolant jackets. As long as sufficient thermal enhancement is provided in the combustor to achieve the imposed bulk temperature limit, any of the cycles defined will reach a maximum specific impulse nearly equal in every case. The thermally enhanced chamber is thus capable of delivering a specific impulse of 485.5 seconds at a thrust of 15,000 pounds.

Combustor Length Effects. The preceding cases demonstrated that all systems, where sufficient heat loads are available to achieve a given bulk temperature limit, will attain the same performance level. This is always true for cases where the combustion length is optimized. A study was conducted to define the performance parameters of the above configurations as a function of length. Results have indicated that at combustor lengths other than optimum turbine gas regeneration can be beneficial in terms of attainable chamber pressure (Fig. 6) and engine delivered specific impulse (Fig. 7).

A pictorial representation of the engine concepts considered in the combustor length studies is presented in Fig. 8, grouped according to technology level requirements. Specific impulse and engine weight (the parameters which influence payload directly) are shown as well as chamber pressure and area ratio (which affect engine development testing). In the group of smooth-chamber engines constituting state-of-the-art technology,

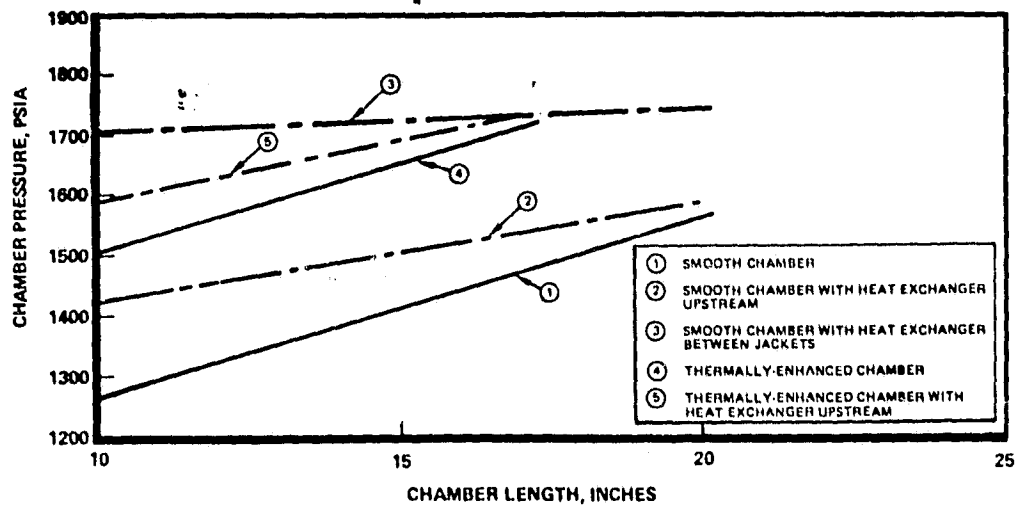


Figure 6. Chamber Pressure Variation with Chamber Length

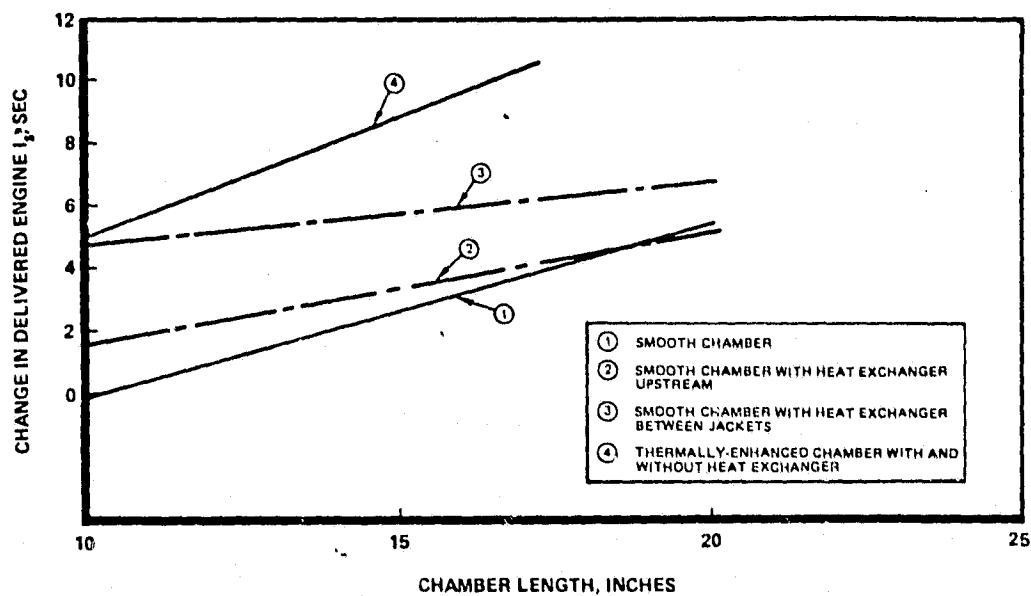


Figure 7. Performance Variation with Chamber Length

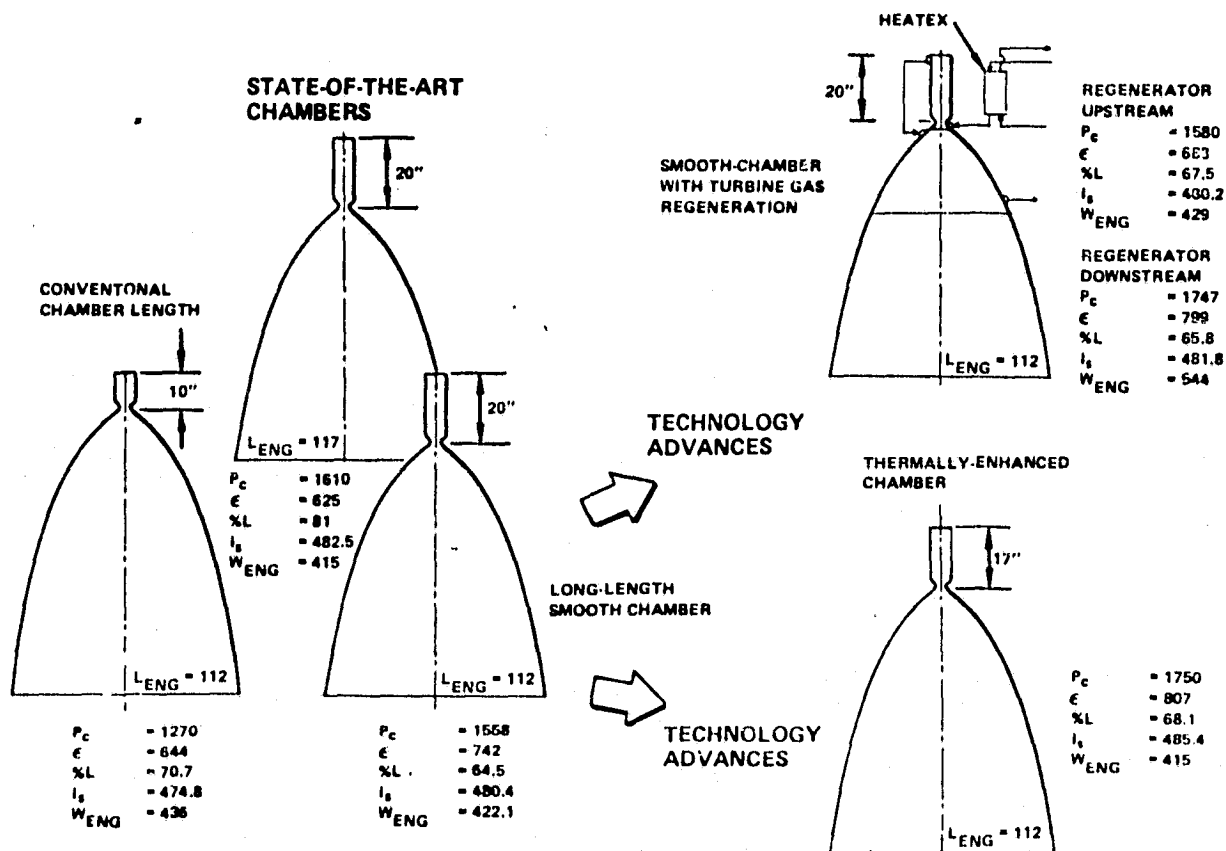


Figure 8. Performance Optimization Studies at 15,000 Pounds Thrust

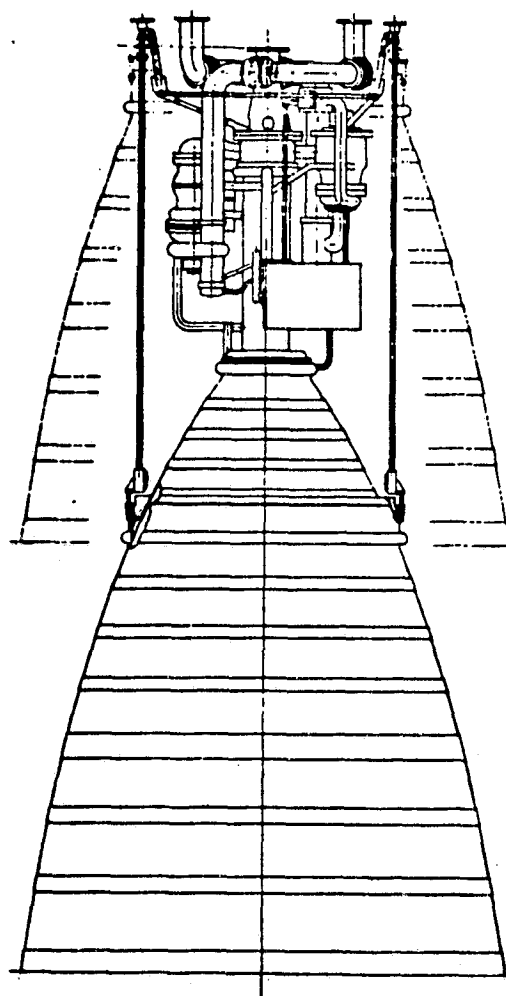
it is seen that a gain of approximately 8 seconds has been achieved by increasing combustor length above that of a conventional engine; with a reasonable nozzle expansion area ratio (625). No technology advances are required to obtain the quoted performance. Chamber pressure (power) margins are provided for contingencies arising from engine-to-engine component performance variations, heat transfer, and system pressure drop.

Larger gains in performance can be obtained if technology advances are considered in the areas of improved heat exchangers for turbine gas regeneration and in heat transfer enhancement in the chamber. A moderate gain of 1.4 seconds is indicated for the turbine gas regenerative cycle over the performance of the state-of-the-art concepts. Technology advances are required to reduce the size and weight of the heat exchanger so as to maximize the conversion of this performance advantage into payload advantages. With the present calculated heat exchanger weight, a loss in payload (22 pounds) results. Exchange factors of 70 pounds of payload per second of specific impulse and 1 pound of payload per pound of engine weight have been used from Phase A OTV Engine Study results. A turbine gas regenerator can also be added to an existing engine whose thrust falls below the specified minimum. This modification will allow the engine to meet design thrust requirements.

A more significant gain in performance of 5 seconds and a reduction in engine weight are obtained with the thermally enhanced chamber configuration over the state-of-the-art chambers for a corresponding gain in payload of 357 pounds. This concept, however, requires test verification before it can be implemented; therefore, it is included in the category requiring technology advanced beyond the current 1980 level. Because of its potential, this concept is considered an alternate for future growth capability of the selected smooth-wall chamber concept.

SELECTED ENGINE CONFIGURATION

The optimization and system trade studies resulted in the design characteristics of the baseline engine configuration as presented in Fig. 9.



BASELINE DESIGN CHARACTERISTICS

● FULL THRUST (VAC), LB	15,000
● MIXTURE RATIO	6
● CHAMBER PRESSURE, PSIA	1610
● EXPANSION AREA RATIO	625
● SPECIFIC IMPULSE, SEC	482.5
● PUMP DISCHARGE PRESSURE, PSIA	
● OXYGEN	2682
● HYDROGEN	4821
● PUMP SPEEDS, RPM	
● OXYGEN	58,100
● HYDROGEN	107,140
● TURBINE INLET TEMP, R	955
● RETRACTED LENGTH, IN.	80
● EXTENDED LENGTH, IN.	117
● ENGINE DRY WEIGHT, LB	415
● TECHNOLOGY BASIS	1980

DESIGN FEATURES

- FULL-FLOW REGEN.COOLING
- SERIES MAIN TURBINES
- GH₂-DRIVEN LOW PRESSURE PUMPS
- SMOOTH-WALL 20 IN. LONG COMBUSTION CHAMBER
- AUTOGENOUS TANK PRESSURIZATION

Figure 9. Performance-Optimized Advanced Expander Cycle Engine
Baseline Design Characteristics

This selection resulted from specific impulse performance and weight optimization of the engine in the 10,000 to 20,000 pound thrust range. Selection of component design and performance levels was made to provide design margins that will reduce engine development risks. Although, for the engine length selected, specific impulse performance optimizes at an area ratio of 900:1, the recommended design utilizes a nozzle with expansion area ratio of 625:1. The lower value facilitates altitude testing in idle mode, reduces test and fabrication costs, and results in a lower engine weight.

High Pressure Pump Designs

Both the fuel and oxidizer high pressure pump designs present the opportunity to apply proven pump technology developed in current programs. The pump speeds required to provide maximum efficiency in the utilization of turbine derived power are 107,000 rpm for the hydrogen pump and 58,111 rpm for the oxidizer pump. These speeds are consistent with 20 mm diameter bearings, which have received extensive testing in the Advanced Space Engine (ASE) fuel and oxidizer pump technology development program. The hydrogen pump speed is within current bearing DN limits, thus complying with the 1980 technology ground rule. This ground rule is also complied with at the pump-to-turbine interface seal on the high pressure oxidizer pump where current seal configurations (ASE and SSME) provide a proven concept for OTV application.

Because of the high torque required to drive both main pumps from a single turbine and the resultant life impact on a gear drive concept, a single turbine with geared drives was ruled out for the high pressure turbomachinery. Similar life considerations were used to rule out dual turbines with synchronizing gears even though this scheme may be advantageous for ease of engine control during engine transient buildup and decay.

Boost Pump Drive

The impact of boost pump drive method upon overall engine performance is small because of the low horsepower requirements for these pumps; therefore,

other factors are more influential in the selection of the boost pump drive.

Preliminary selection of gaseous hydrogen turbine drives was made for the hydrogen and oxygen boost pumps based on trade studies which considered performance, transient characteristics, flexibility of design, complexity, packaging, and testing flexibility. The other boost pump drives being considered are: electric, hydraulic, and geared.

Cooling Circuit

The cooling circuit selected is a full-flow regenerative scheme depicted in Fig. 10. All of the hydrogen flow, except for pump seal leakage and that used in supplemental cooling, is used in cooling the thrust chamber and nozzle in series arrangement. This hydrogen flow enters the combustor coolant jacket from the pump outlet at an expansion area ratio of 6:1 and it exits at the injector plane. From here, it is routed externally and enters the nozzle coolant distributing manifold at an expansion area ratio of 6:1. The nozzle is then cooled in a one and one-half pass cooling scheme with the coolant exiting at an area ratio of 100. This circuit allows for the highest fuel turbine inlet temperature limits compatible with safe nozzle wall temperatures.

Injector

Coaxial injection elements and a transpiration cooled Rigimesh faceplate have been selected for the baseline engine design. This configuration offers proven performance, reliability, and complete fabrication experience for minimum iteration during design and development.

Combustion Chamber

The combustion chamber is configured with a smooth hot wall and a length of 20 inches. The hot wall is regeneratively cooled with hydrogen flowing in axial passages from the throat region toward the injector face. The

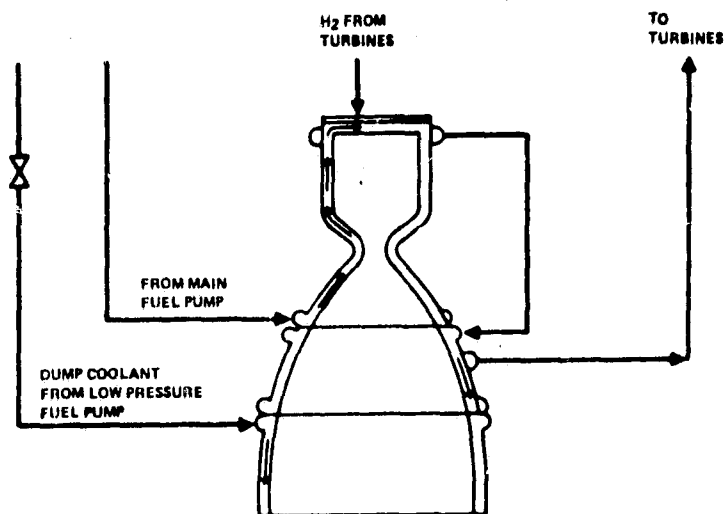


Figure 10. Full-Flow Regenerative Cooling

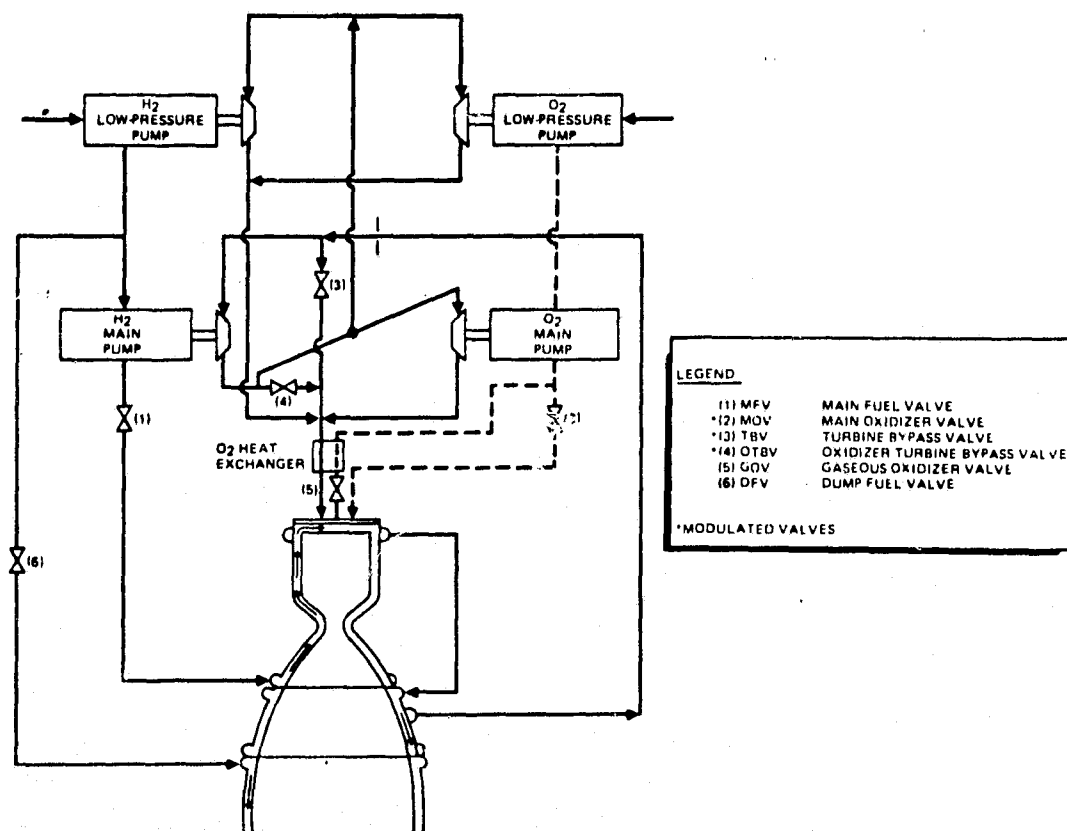


Figure 11. Advanced Expander Engine Control Schematic

coolant channel geometry favors a contraction ratio of 4, which allows doubling the number of passages from the throat to the combustion chamber. The chamber configuration represents 1980 technology.

Fixed Nozzle

Configuration trade study results for the fixed portion of the nozzle reveal a highly developed state of current technology in this area. The selected design configuration has a thorough development basis in performance, reliability, fabrication, and maintainability. It consists of a NARloy-Z channel cooled configuration in the high heat flux region near the combustor-nozzle attach plane and a thin wall A286 brazed lightweight tube structure from an expansion ratio of 6 to 210.

Extendible Nozzle

Because cooled metallic nozzles are considered current state-of-the-art, a hydrogen dump-cooled nozzle was selected for the initial baseline engine configuration. The retractable nozzle extends from an area ratio of 210 to 625. Cooling of this nozzle is feasible at all engine mixture ratios and only requires at off-design different coolant flow splits with other thrust chamber components than at on-design mixture ratio. No structural problems are expected with this type nozzle during start and shutdown transients, full thrust, or during normal handling.

Refractory metallic and carbon-carbon radiation cooled nozzles were examined for application to the extendible nozzle. The Space Shuttle Orbital Maneuvering System 6000-pound thrust engine uses a columbium nozzle that has a disilicide coating to prevent oxidation. Exploratory development testing has been conducted by industry on carbon-carbon nozzles for application to both solid and liquid propellants.

Engine Control System

To enable control of thrust and mixture ratio level during mainstage and start, a simple three-valve closed-loop control system was defined. The valves required in this control system are shown schematically in Fig. 11. The control points selected have the capability to fully control the engine thrust and mixture ratio over the full range of required engine operating conditions. The control modulates the areas of the main oxidizer valve, the turbine bypass valve, and the oxidizer turbine bypass valve to achieve independently the proper balance of propellant flows for independent control of thrust and mixture ratio.

The thrust and mixture ratio levels are variable upon vehicle command. Closed-loop control is required to maintain engine thrust and mixture ratio, within anticipated values of ± 5 and ± 2 percent, respectively, of the commanded mainstage values. This type of precision is similar to Centaur upper stage engine thrust and mixture ratio precision requirements of ± 2 percent. The thrust accuracy requirement has been relaxed somewhat for the OTV from that required in the Centaur vehicle. The trade studies conducted during the selection of the control system confirmed previous work and concluded that for a reusable manned OTV, a closed loop controlled propulsion system is required to provide necessary accuracy, life, and reliability.

ENGINE PERFORMANCE PARAMETRICS

Engine performance, weight, and envelope trends were derived for the advanced expander cycle engine as functions of thrust, engine retracted length, and nozzle area ratio. These data were used in the selection of the engine cycle configuration, and were also furnished to the vehicle contractors for use in current OTV selection studies. In addition, payload calculations were performed to aid in the optimization and final selection of the engine geometry. For the payload studies, the mission and vehicles defined in NASA TMS 73394, "Orbit Transfer Systems with Emphasis on Shuttle Applications 1986-1991," were used.

Predicted parametric specific impulse trends with thrust and nozzle expansion area ratio are shown in Fig. 12 for an engine mixture ratio of 6:1

at a retracted engine length of 60 inches. At the fixed engine length, the lower thrust engines with attendant lower throat area achieve the higher nozzle expansion area ratios. Thus, the lower thrusts provide the highest engine specific impulses.

Peak specific impulse at each thrust is achieved through tradeoff of ideal specific impulse and nozzle nonideal losses as nozzle area ratio is increased. The tradeoff results in an optimum value of nozzle area ratio at each thrust, which with the fixed nozzle length establishes the nozzle percent length.

Area ratio can be reduced below the optimum value without affecting specific impulse significantly. Reducing area ratio below the peak specific impulse value results in a nozzle configuration which facilitates simulated altitude testing at idle mode conditions, reduces nozzle flow separation tendencies, reduces test and fabrication costs, results in lower engine weight, and leads to payload optimization. The selected 15,000 pound thrust engine makes full use of the above advantages and of recent design study results wherein optimization of gimbal plane-to-injector distances and nozzle retracting mechanism lengths led to better utilization of available engine length. The latter led to an engine length selection of 117 in. instead of the 112 in. used in the parametrics (Fig. 12).

Parametric off-design mixture ratio performance data for the parametric array of engine configurations is shown in Fig. 13; weights and envelope data appear in Fig. 14 and 15, respectively. Using a simplified mission analysis program, the engine weight and performance data were utilized to predict the payloads to be expected for orbital transfer vehicles associated with a Space Shuttle of 100,000 pounds gross weight capability. The results normalized to the maximum payload obtained are shown in Fig. 16 and 17 for the All-Propulsive Orbital Transfer Vehicle (APOTV) and the Aero Maneuvering Orbital Transfer Vehicle (AMOTV), respectively. The APOTV uses main propulsion to supply all necessary delta-V, while the AMOTV employs aerodynamic forces to supply part of the delta-V. As seen in both Fig. 16 and 17, peaks occur in the payload performance

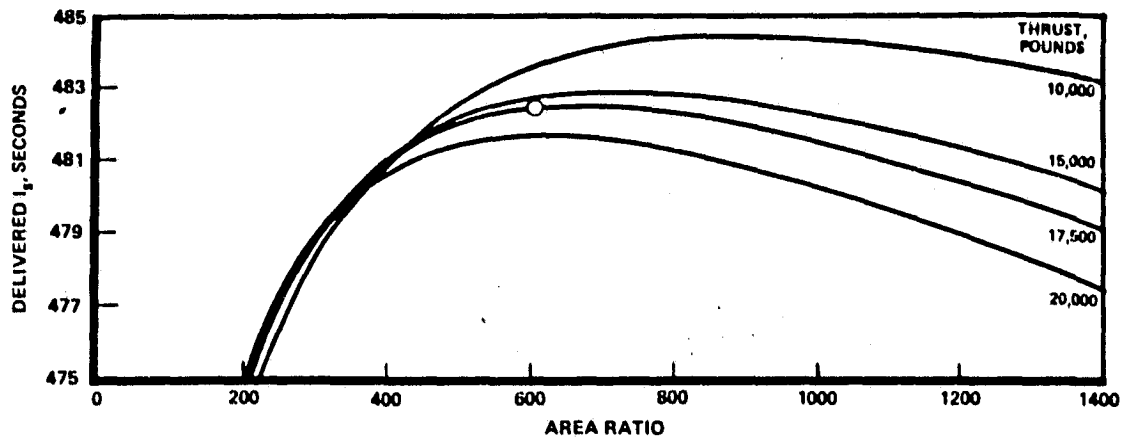


Figure 12. Expander Cycle Specific Impulse vs. Area Ratio
(MR = 6, Engine Length = 112 Inches)

Legend: ○ Selected engine with engine length of 117 inches and 15,000 pound thrust

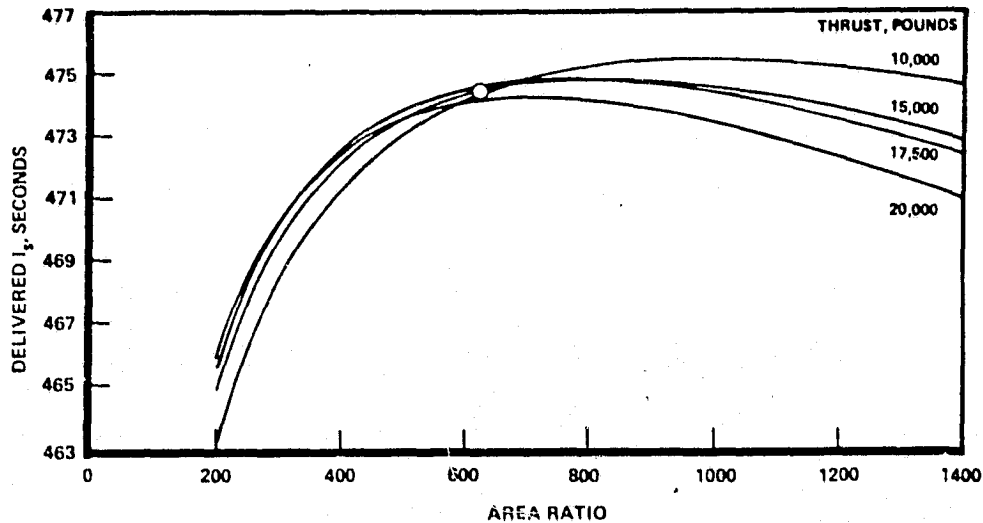


Figure 13. Expander Cycle Specific Impulse vs. Area Ratio
(MR = 7, Engine Length = 112 Inches)

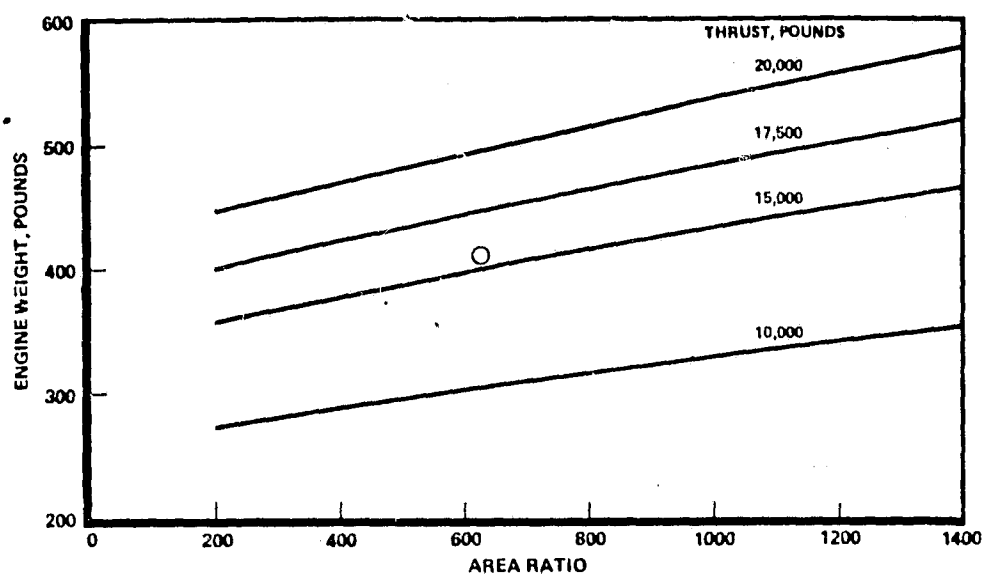


Figure 14. Expander Cycle Engine Weight for MR = 6

Legend: ○ Selected engine with engine length of 117 in. and 15,000 lb thrust

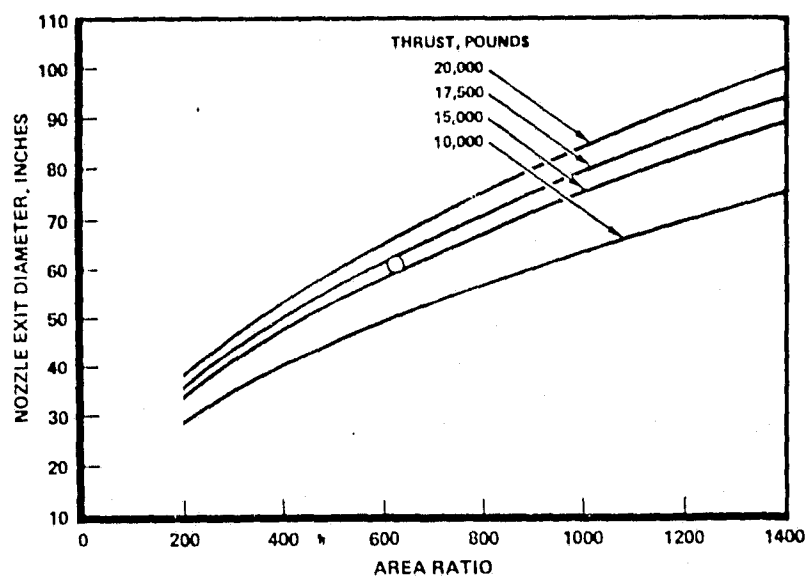


Figure 15. Expander Cycle Exit Diameter vs. Area Ratio
(Engine Length = 112 Inches)

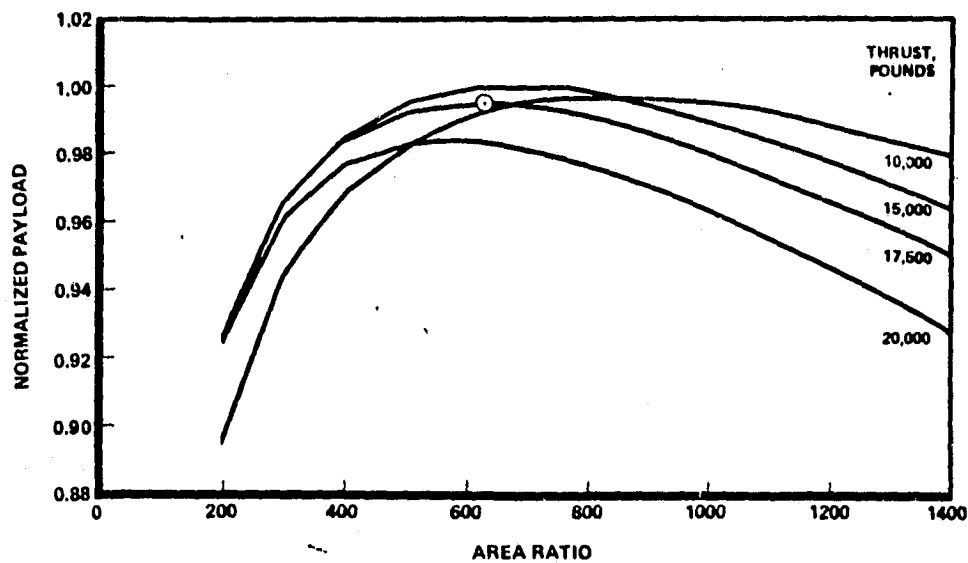


Figure 16. APOTV, Expander Cycle Normalized Payload for MR = 6

Legend; ○ Selected engine with engine length of 117 in. and 15,000 lb thrust

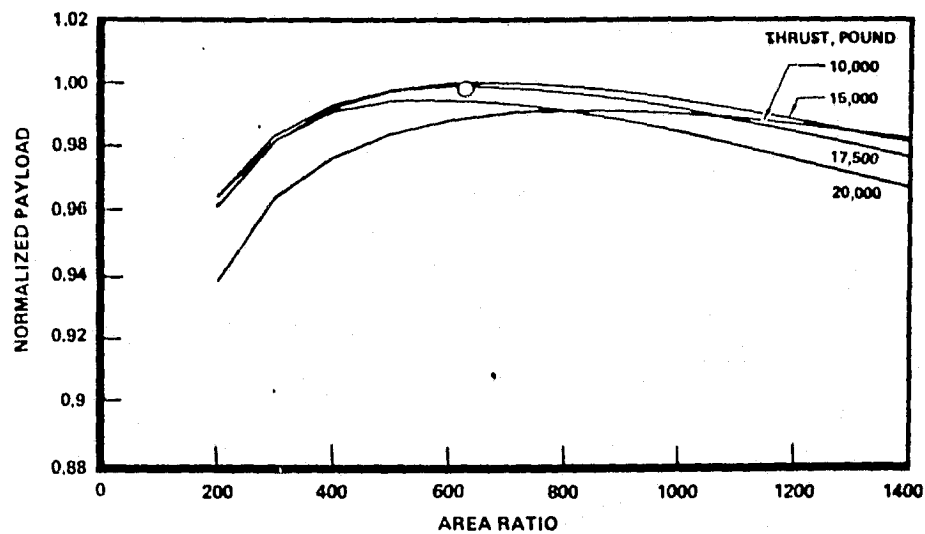


Figure 17. AMOTV, Expander Cycle Normalized Payload for MR = 6.

curves. These maximum values are induced by the specific impulse performance and engine weight trends with area ratio. The engine weight increase with area ratio reduces the optimum area ratio values obtained previously for peak specific impulse. Maximum payloads occur at 15,000 pounds thrust and are caused by the interaction of gravity losses and engine weight changes with thrust. Peak payload at 15,000 pounds thrust occurs at a nozzle area ratio near 650. At 15,000 pounds thrust payload is not sensitive to area ratio changes in the 625 to 700 range. Payload studies thus provide the final basis for selection of the nozzle area ratio (625:1).

TASK 9. ALTERNATE LOW THRUST CAPABILITY

The feasibility and design impact on the OTV staged combustion cycle engine of adapting the engine to operate for extended periods in the 1-2K lb. thrust range was investigated.

LOW THRUST OPERATION AND PERFORMANCE

The OTV engine digital computer transient model was used to analyze engine operation at low thrust conditions. Pump operation and feed system stability were examined using the OTV engine start transient model. Heat exchanger flowrate requirements and thrust chamber coolant flowrate requirements were established analytically. LOX injector flowrate requirements for stability and thrust chamber combustion performance were assessed using ASE and J-2S test experience.

The objectives of the OTV computer model runs (Table 5) were to establish steady-state chamber pressure and mixture ratio conditions, valve scheduling and valve positioning required for efficient combustion of propellants in the chamber and for stable system operation. The pumps were assumed to be thermally conditioned to saturated propellant temperatures of 162.7R for the LOX and 37.8R for the hydrogen. Propellants were made available at tank pressure corresponding to NPSH values of 2 and 15 for the oxygen and hydrogen, respectively, at temperatures prescribed above. These conditions are all based on an assumed period of tank head idle mode operation prior to the low thrust operation.

The results of the study of extended low thrust operation of the OTV staged combustion engine indicate that long-life, high performance operation of the engine in the low thrust mode (1-2K level) can be achieved with minimum modification (kitting) of the engine.

TABLE 5. System Operation and Stability Objectives
and Conditions - OTV Start Model Runs

OBJECTIVES

Attainable steady-state chamber pressure

Attainable steady-state mixture ratio

Effect of main oxidizer valve (MOV) opening
on mixture ratio

Effect of pump H/Q map characteristics

Effect of fuel pump recirculation

Effect of turbine bypass for thrust control

OTV ENGINE START CONDITIONS

Propellants in tanks at saturated conditions

Pumps thermally conditioned to cryogenic temperature

LOX heat exchanger flow and temperature modeled

Injector/dome heat transfer not modeled

Modifications to the OTV 20K staged combustion engine which were considered were:

- (1) Removal of the preburner injector, since in pump-fed idle the engine is run in an expander mode and does not require a preburner. Removal of the preburner injector increases the available turbine inlet pressure to the drive turbines. The preburner LOX line and valve can be removed in order to simplify the system.
- (2) Modification of the main injector to increase the oxidizer side pressure drop so that potential feed system coupled instability is avoided. Two approaches are possible; one is to use a LOX post insert which incorporates an orifice to increase oxidizer side pressure drop, a second approach is to braze on new LOX posts which incorporate a smaller flow area.
- (3) The use of fuel pump recirculation in order to avoid the positive slope region of the pump H-Q curve. This modification would add a line and valve from the pump discharge to the pump inlet in order to recirculate hydrogen and keep the pump at a higher flowrate. Computer model studies have indicated that this is an effective method to avoid pump instability problems when operating at low flow conditions. The computer modeling has included the effect of propellant heating which occurs when pump recirculation is used.
- (4) Modifications to the hydrogen pump could be considered in order to have a pump H-Q curve without a positive slope region at low flow conditions. A modified fuel pump can be derived by a variety of changes, such as: (1) increased sweepback of the impeller blades to make the H-Q curve have more negative slope, (2) decrease the impeller O.D. to allow the pump speed to increase and operate at a more favorable Q/N, (3) decrease the tip

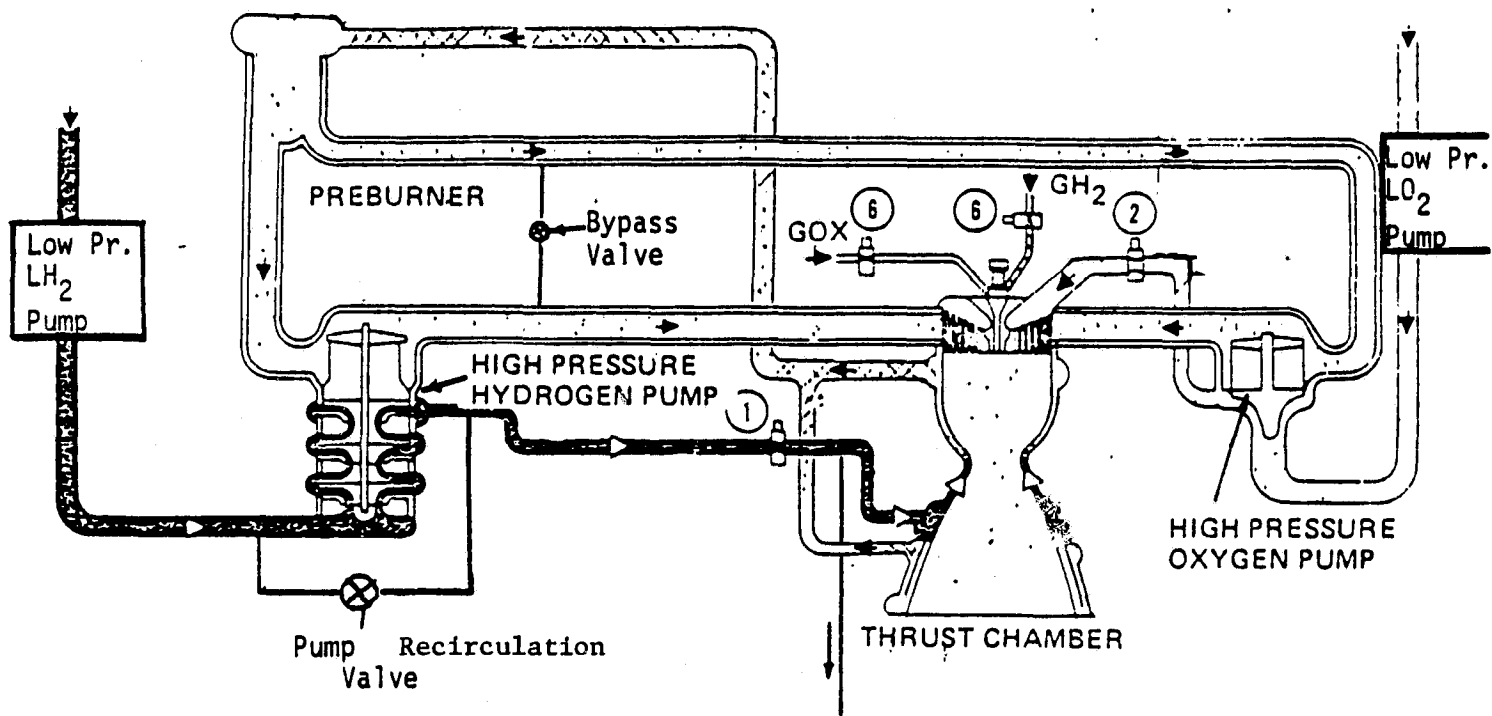
width of the impeller in order to modify the H-Q characteristics, and (4) redesign the first stage cross-over diffuser to avoid stall at low flowrates.

Each of these modifications were considered and evaluated by means of the OTV staged combustion transient computer model.

The modifications which are recommended for kitting of the OTV staged combustion engine are: (1) removal of the preburner injector, (2) modification of the main injector to improve feed system stability, and (3) the use of fuel pump recirculation to avoid fuel pump operation in the positive slope region of the H-Q map. The modifications are shown schematically in Figure 18. Modification to the fuel pump, such as changes to the impellers, would require more extensive fuel pump development and could be considered as alternatives to the use of fuel pump recirculation.

Flow instabilities were not present in the oxidizer system during simulation of low thrust operation and based upon the computer simulation no changes to the oxidizer pump and feed system are recommended.

The attainable specific impulse at high and low thrust is shown in Table 6 for the OTV staged combustion engine which resulted from the OTV engine Phase A study. This engine is a 20K thrust staged combustion system operating at 2000 psia chamber pressure in the high thrust mode with an expansion area ratio of 400:1. The specific impulse projected for low thrust operation includes the effects of increased reaction kinetic losses and the expected energy release efficiency at reduced flow conditions.



VALVE	
①	MAIN FUEL
②	MAIN OXIDIZER
⑥	IGNITERS

LEGEND	
LH ₂	
GH ₂	
LO ₂	
GOX	
HOT GAS	(MAIN STAGE)

Figure 18. Modified OTV Staged Combustion Engine Schematic

TABLE 6. ATTAINABLE THRUST AND MIXTURE RATIO
AT BOTH THRUST EXTREMES

Engine System - 20K Vacuum Thrust Staged Combustion

Cycle, $P_c = 2000$ psia, $\epsilon = 400$

Mixture Ratio (O/F) = 6:1

	<u>High Thrust Mode</u>	<u>Low Thrust Mode</u>
Thrust, lbf	20,000	1,610*
Chamber pressure, psia	2,000	166
Mixture ratio, engine	6.0	6.0
Specific impulse, engine, sec	477.1	455.9
Engine fuel flowrate, lb/sec	5.99	.5022
Engine oxidizer flowrate, lb/sec	35.93	3.029
Main fuel pump speed, RPM	95,000	22,722
Main oxidizer pump speed, RPM	70,000	17,271
Main fuel pump discharge pressure, psia	4,563	268
Main oxidizer pump discharge pressure, psia	4,852	386
Main fuel pump recirculation flowrate, percent	0	64%

*Other points of operation below 1.6K
are feasible.

EXPECTED LIFE AND RELIABILITY

The low thrust adaptation of the 20K lb. thrust staged combustion engine will require stage burn times of 6 to 7 hours. These are in the order of 12 times longer than those for the basic engine to accomplish the same mission. However, the lower pressure, thermal load and strain levels, and rotational speeds will result in longer low cycle, fatigue, and creep limited life of engine components during low thrust operation as compared to high thrust operation. An engine life increase is expected for low thrust operation above the 10-hour life for full thrust operation. The exact magnitude of life increase would require a more detailed analysis of thrust chamber and pump designs.

The longer burn time could decrease mainstage reliability if the failure rate stayed constant. However, it is expected that failure rates of components operating at the low thrust design point would be lower than at the 20K lb. thrust point due to lower stresses and strains; many components would be operated at essentially derated conditions. While it is not possible to analytically relate failure rates for operation at low thrust to operation at high thrust, the increased margins of safety would imply that a decrease by a factor of 10 or more is achievable. If operation at the low thrust mode requires more start and stop cycle, however, the lower stresses may not be the most significant factor in the probability of a successful start and additional attention during development should be placed on achieving a high start reliability. Full duration engine tests represent one of the major DDT&E cost items.

DDT&E COSTS

The DDT&E costs for a 20K lb. thrust staged combustion engine, modified for low thrust operation, were estimated based on the following ground-rules:

1. The low thrust engine development is assumed to be preceded by the full development of a rated high thrust engine. The low thrust engine development, therefore, will build on the basis of a rated thrust engine whose operating characteristics have been firmly established by detailed component and engine tests.
2. Engine conversion from rated thrust to low thrust is performed using the recommended modifications to the existing engine components as described earlier.

The ROM cost estimate for DDT&E to convert a 20K lb. thrust staged combustion engine to a 1 to 2K lb. thrust high performance, long burn duration engine is approximately \$12.75 million (1980) dollars. The majority of the cost, about 61%, is incurred due to engine testing.

ENGINE WEIGHT IMPACT

The conversion of a high thrust staged combustion engine to a low thrust engine will not influence engine weight significantly.

IDENTIFICATION OF NEW TECHNOLOGY

New technology required to enable design and construction of a staged combustion engine capable of efficient operation at the low thrust mode has been identified as a part of this task. The major required modifications to the basic staged combustion engine are in the fuel pump to allow operation at low thrust and possible modification of the injector to prevent feed system coupled instability. The technology program could use a modified MK 48-F liquid hydrogen turbopump to define the limits of rocket engine turbopump operation, due to throttling, down to low thrust levels and with wide variation in mixture ratio.

TAST 10: SAFETY, RELIABILITY AND DEVELOPMENT RISK COMPARISON

The OTV propulsion system must be designed as a highly reliable and safe system due to its incorporation in and service for a manned vehicle. In this task, the impact of safety and reliability on propulsion system selection, engine design, test philosophy and engine cycle were investigated. In addition, a development risk comparison was performed to study any relative advantages of the advanced expander versus those of the staged combustion cycle engines.

The task objectives were accomplished by investigating the following four areas:

- (1) Approach to safety and reliability of past and current rocket engines for manned space vehicles,
- (2) Major issues in manrating, safety and reliability for OTV propulsion,
- (3) Suggested safety and reliability approach for OTV engines,
- (4) Development risk comparison between advanced expander and staged combustor.

PAST AND CURRENT ENGINES FOR MANNED SPACE PROGRAMS

Prior to the Space Shuttle, emphasis was mainly on high reliability for safety. Numerical reliability goals were imposed on the engines by specifications and were either to be demonstrated (H-1, F-1, J-2) or were given as an allocation goal without demonstration (SPS, LEM engines), see Table 7. Engine safety was not addressed explicitly as a design criterion. Engine out capability early in the mission was limited to the Saturn 1-B first-stage powered by eight H-1's. The first and second Saturn V stages had engine out capability only in the late boost phases of each stage.

The engines for the Space Shuttle Orbiter do not require quantitative reliability demonstration. Emphasis is on safety as a total program commitment and as an explicit design objective reflected in specifications

TABLE 7: RELIABILITY CRITERIS FOR MANRATED VEHICLES

		ENGINE RELIABILITY GOAL	VERIFICATION
EXPENDABLE ENGINES	• MERCURY ATLAS	.85	TEST DEMONSTRATION: 100 TESTS; 90% CONF
	• GEMINI TITAN I		
	• SATURN BOOSTERS	.99	TEST DEMONSTRATION: 69 TESTS, 50% CONF
	H-1		
	F-1 J-2		
REUSABLE	• APOLLO		
	SPS	.9999	{ DESIGN VERIFICATION TESTS QUAL TESTS CERTIFICATION
	LEM DESCENT	.9991	
	LEM ASCENT	.999982	
	• SPACE SHUTTLE		
	SSME OMS RCS	NO NUMERICAL GOAL	{ DESIGN VERIFICATION SPEC'S LIFE REQUIREMENTS CERTIFICATION

and program plans. Reliance is placed on engine out capability during mission abort for crew safety, and greater engine component redundancy than in the past.

The investigation showed that various reliability and safety approaches existed for past and current manned space vehicle engines. These approaches can be summarized as follows:

Past Engines for Mercury, Gemini, Saturn, Apollo Programs

- . Emphasis on high reliability; engine safety not explicitly addressed
- . In some programs, numerical goals for reliability were demonstrated
- . Few redundant engine components
- . Limited engine out capability

Current Engines for Space Shuttle Program

- . Emphasis on safety as a total program commitment
- . Concentration on assessment and confirmation of engineering design for reliability
- . Qualitative treatment of reliability
- . Engine out capability for safe return considered important
- . Redundancy of selected engine components

OTV PROPULSION SYSTEM SAFETY AND RELIABILITY ISSUES

The following points are addressed as major issues for OTV propulsion manrating, safety and reliability:

1. Do current NASA specifications for the STS and for previous manned space programs adequately cover safety criteria for OTV propulsion, specifically during space flight?
2. Which OTV propulsion system configurations are best for mission success and for crew safety from the standpoint of highest success probability?

3. How is engine reliability affected by:
 - a. The number of engine tests,
 - b. Demonstration requirements of reliability goals,
 - c. Engine component redundancy, and
 - d. Engine cycle?
4. What is a viable OTV engine design approach for safety and reliability?

Each of these issues is treated in detail in Vol. II-A of the Phase A Extension Final Report and resulted in the formulation of a suggested safety and reliability approach outlined in the following section.

SUGGESTED SAFETY AND RELIABILITY APPROACH FOR OTV ENGINES

Based on the task investigation, the following safety and reliability approach for OTV engines is suggested:

Use Current STS Propulsion System Approach for Safety and Reliability

This approach emphasizes design-to safety and reliability, featuring thorough safety and reliability analyses of the design and extensive design verification by test or analysis. Demonstration of numerical reliability goals (as done for H-1, F-1, J-2) is not considered cost effective.

Best Propulsion System Concept With Regard to Mission Success and Crew Safety

Mission Success: Two main engines, with either engine capable of mission completion after the first burn occurred with both engines.

Crew Safety: Two engines with backup propulsion. Either main engine or the backup propulsion are capable of safely returning the OTV and its crew from any point in the mission.

The capability of a backup propulsion for crew safety needs to be investigated and firmly established.

Engine Design Features for Safety and Reliability

The OTV engines should have specific features to enhance safety and reliability. Major features are as follows:

Selected Component Redundancy: Safety critical components should be made redundant. This includes the controller, igniters and critical flight instrumentation.

Fail-Safe Components: Fail-safe features, such as valve closure upon loss of signal, should be incorporated wherever they improve safety and do not degrade mission success.

Protection against Common Mode Failures: Design protection against common mode and non-independent failures should be provided to the greatest possible extent. This includes separation, proper placement, and hardening of vital components.

Engine Checkout and Health Monitoring System: The engine design should include a checkout and health monitoring system which automatically checks for failures, alerts the crew if redlines are exceeded, and shuts down the engine prior to the occurrence of hazardous failures.

Engine Cycle Preference: The advanced expander engine was found to be slightly more reliable than the staged combustion engine at the same development stage. However, this difference is not considered to be significant enough to make an engine cycle selection on the basis of reliability alone.

OTV ENGINE DEVELOPMENT RISK

A development risk assessment was performed for the OTV engines with the following two purposes:

- a) To establish a relative risk ranking with respect to engine cycle; i.e., advanced expander and staged combustor,
- b) To investigate the influence of slipped and advanced versions of the nominal schedule upon development risk.

A methodology was developed in which the structured, subjective responses of cognizant component engineers were converted into quantitative probability and cost estimates.

Results of Development Risk Assessment

Figure 19 shows the relative DDT&E costs for the 20K staged combustor and for the 10K advanced expander as a function of DDT&E schedule change. Since both engine configurations have practically the same development schedule, the cost-versus-schedule curves are of the same shape. The 20K staged combustor cost at the nominal schedule, i.e., at the schedule presented in Task 6 results of the Phase A study, is normalized to one. The 10K advanced expander has an about 5% lower DDT&E cost, as detailed in Vol. II-B. The costs for both engine configurations do not change significantly when the schedule is shortened by six months, but increase rapidly when the schedule is lengthened.

The development risk, shown in Fig. 20, increases substantially if the schedule is shortened, e.g. more than doubles for a 6-month schedule compression, and decreases if the schedule is lengthened. Fig. 20 also indicates that the development risk of the expander is less than that of the staged combustor at the same schedule.

The reason why the risk increases with a compressed schedule is explained by Fig. 21, which shows the estimated development success probability-versus-schedule changes. For shortened schedules, both staged combustor and expander have significantly decreased success probabilities which account for the higher development risks noted above. Main reasons

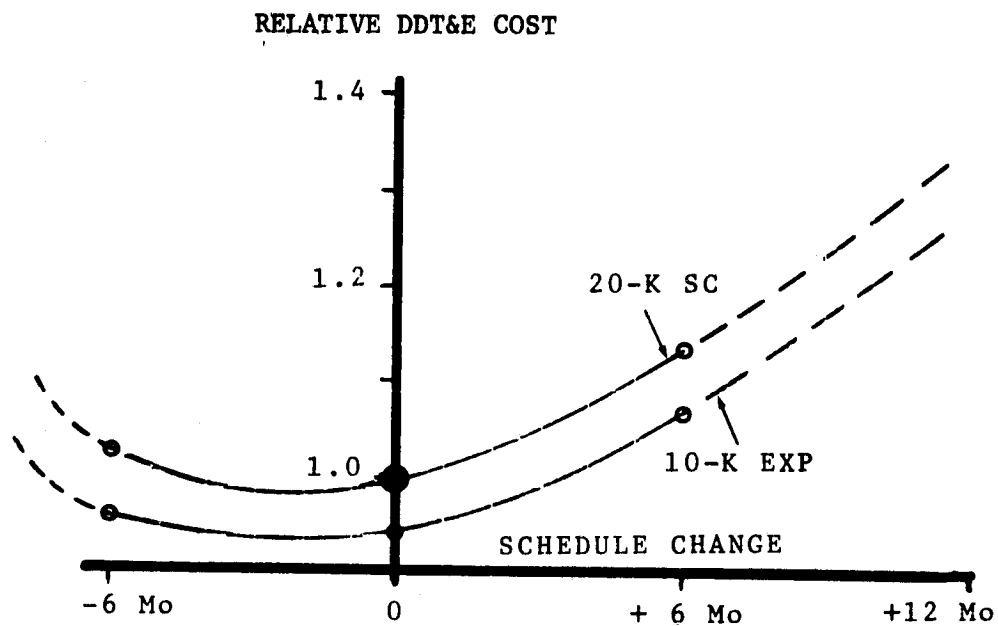


Figure 19. Relative DDT&E Cost Comparison

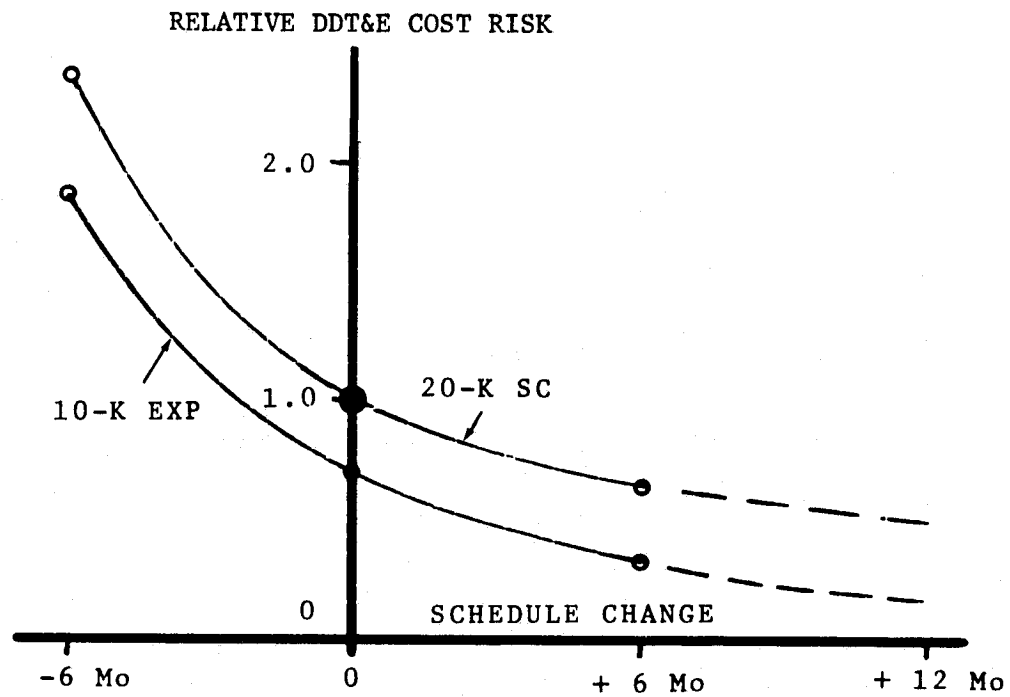


Figure 20. Relative DDT&E Cost Risk Comparison

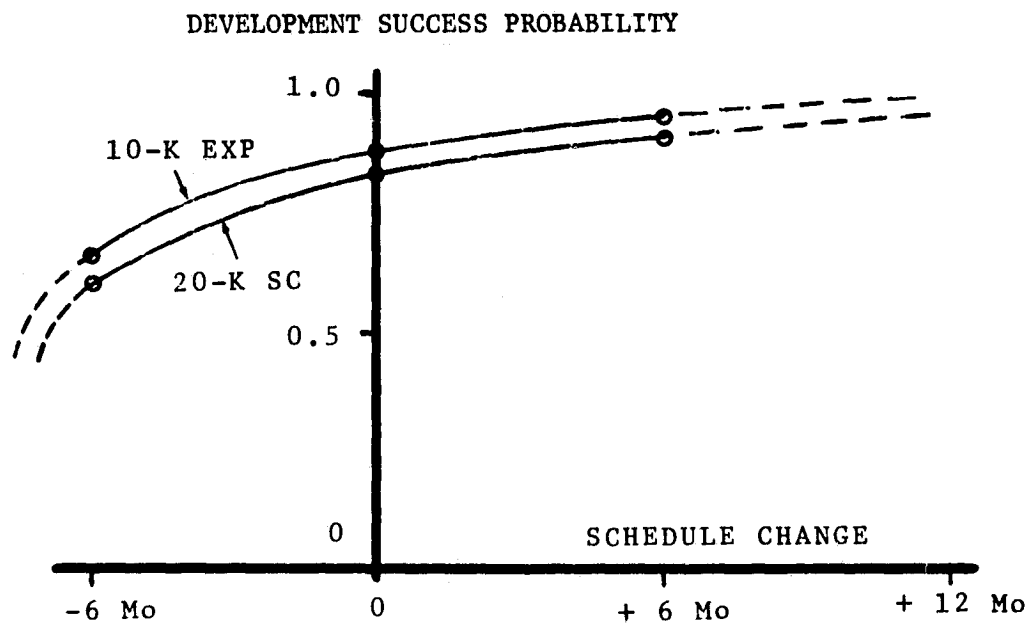


Figure 21. Development Success Probability Comparison

for the lower success probabilities are that any shortening of the schedule would lead to anticipated difficulties in material procurement, crowding of test schedules, and overloading of development efforts. Fig. 21 also shows that the staged combustor has an about 5% lower success probability compared to the expander at the same development schedule.

The results of the development risk assessment can be summarized as follows:

- . The estimated development risk for the 10K-pound thrust advanced expander is less than that for the 20K-pound thrust staged combustor.
- . The development risk would be reduced by 1/3 for the staged combustor, and by 1/2 for the expander, if the program schedule would be lengthened by 6 months. The associated program cost increase would amount to about 14%.
- . Shortening the program schedule would produce practically no change in program costs, but would increase program risk substantially.
- . Major risk drivers for both engine configurations were found to be: turbomachinery, controls, and engine testing.

The following qualifications are made to the result of the development risk assessment investigation: though the assessment results are in quantitative form, they are based on informed, but subjective opinions, and should only be used as trend indications and as general guide for judging development risk. The survey form for the cognizant component engineers was problem-finding oriented and the responses are correspondingly conservative. This is exhibited best by the estimated values for success probability at nominal schedule which are less than one. In contrast, the original cost and schedule estimates presented in Tasks 6 and 7 results of the Phase A Study were success oriented. The lower than 100% program development success probability determined during the Phase A extension investigation was one of the factors which led to the reevaluation of the development costs for the 20K staged combustor as discussed in Volume II-B.

TASK 11. COST AND PLANNING COMPARISON

The Task 11 effort was directed toward a schedule and cost comparison between a 10K lb. thrust advanced expander engine and a 20K lb. thrust staged combustion engine. For the advanced expander, a six-year development program is anticipated, similar to that of the staged combustor. The program is assumed to start with the authorization to proceed (ATP) with the C-D phase in the last quarter of calendar year 1 and extends through year 7 with initial operational capability (IOC) at the end of that year. The designation of numerical schedule years was preferred for this task, since it avoids tying schedules to specific actual calendar years. This makes the application of the programmatic results more general.

Both expander and staged combustor engines are assumed to exist as breadboard engines prior to the start of the C-D phase, i.e., an approximately equal start time in terms of development status is assumed for both engines. A phase B design and analysis effort of sufficient depth and scope is presupposed to permit design layouts which will lead to long lead item procurement concurrently with the start of the C-D phase. Two 10K lb. thrust expander engines (with area ratio of 600) are assumed to be required per OTV vehicle, versus one 20K lb. thrust staged combustion engine (with area ratio of 400).

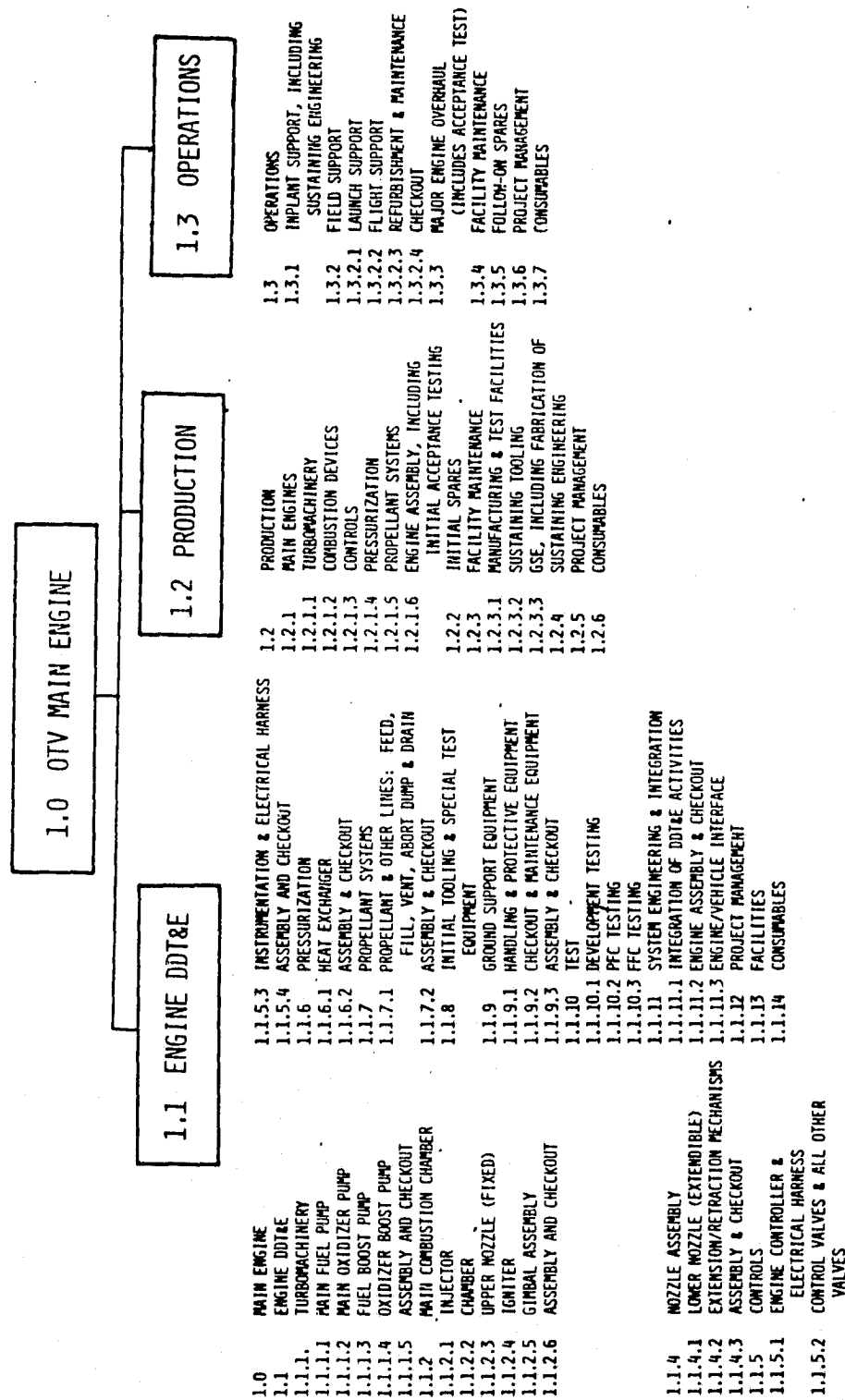
WORK BREAKDOWN STRUCTURE

The work breakdown structure (WBS) for the expander, Figure 22, is identical to that of the staged combustor developed during Phase A, except for the deletion of the preburner, WBS line item 1.1.3.

PROGRAM SCHEDULE

The DDT&E program schedule for the 10K advanced expander is shown in Figure 23. This schedule is almost identical to the schedule for the

FIGURE 22: WORK BREAKDOWN STRUCTURE



staged combustor, developed during Phase A, with the exception of slightly shorter turbopump test schedules and deleted preburner tests. The shortened turbopump test schedules are due to the more benign HPOTP fluid conditions of the expander compared to those of the staged combustor, leading to reduced test numbers. This is also reflected in Fig. 24 which depicts major development testing for the 10K expander. The number and total calendar time duration of expander engine system tests is the same as those for the staged combustor.

FIGURE 23: 10K EXPANDER ENGINE DDT & E PROGRAM SCHEDULE

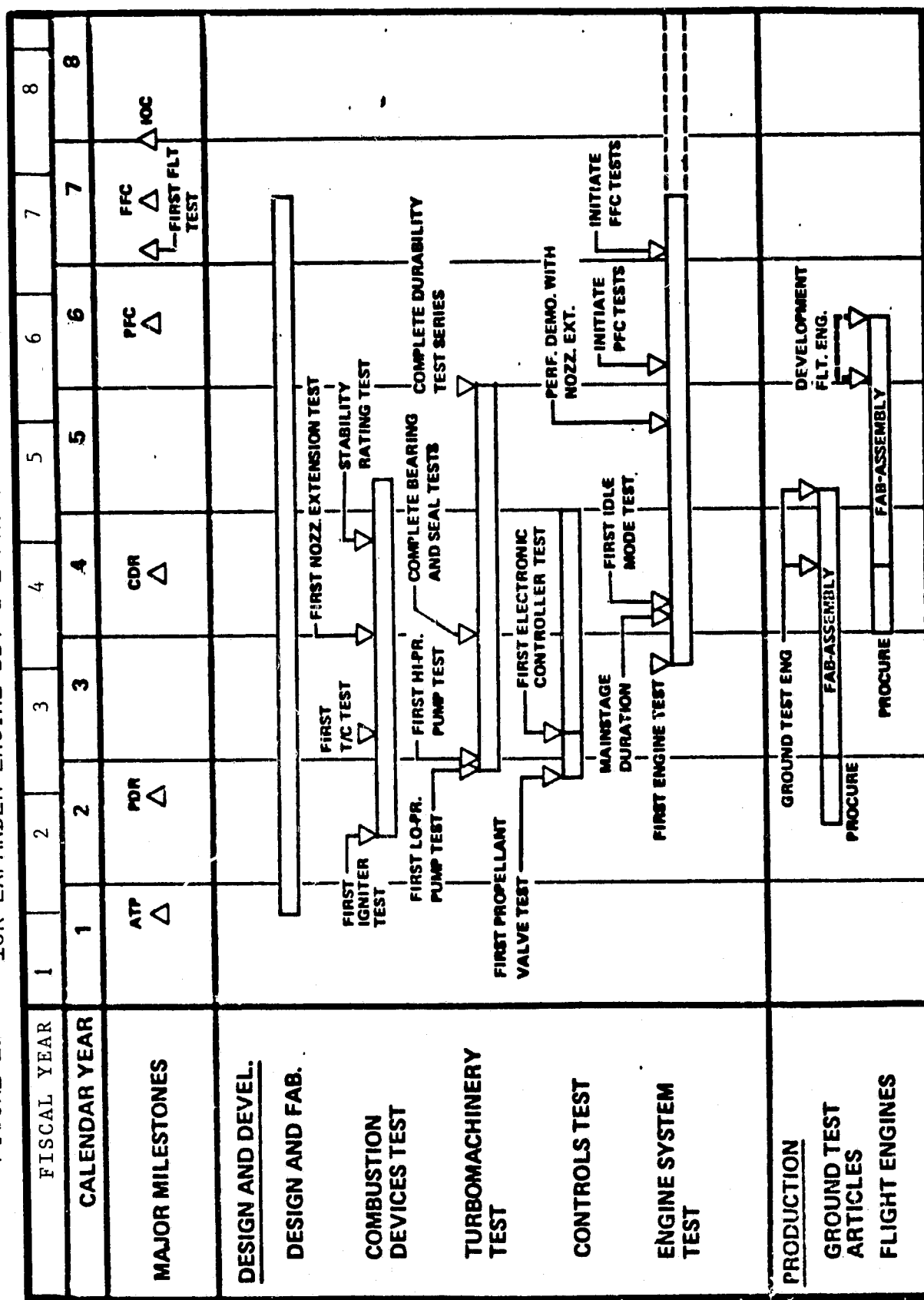
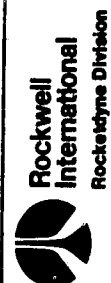
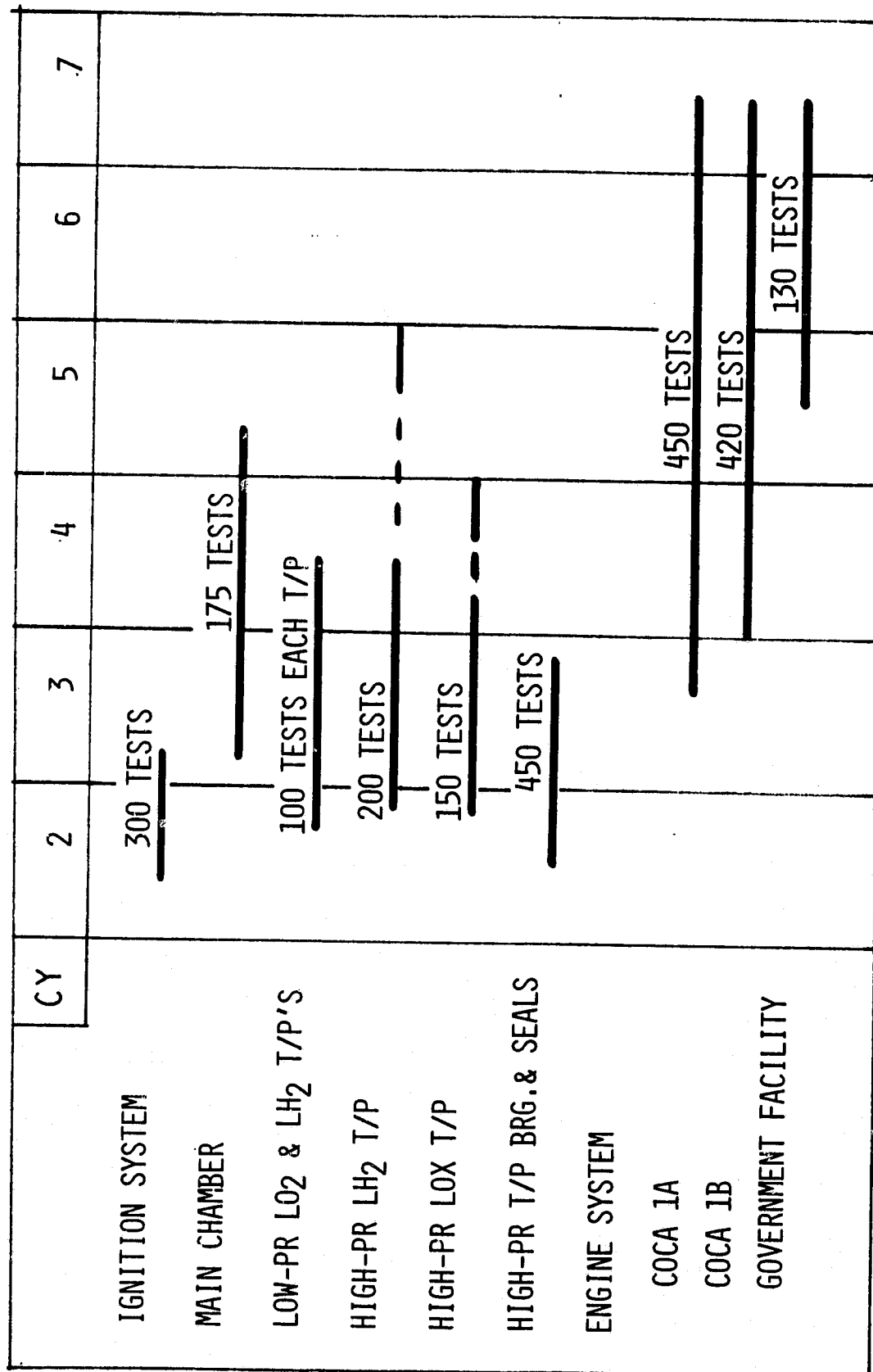


FIGURE 24: 10K EXPANDER ENGINE MAJOR DEVELOPMENT TESTING



COST ESTIMATE

A complete cost estimate for the 10K lb. thrust advanced expander engine system was developed. The cost estimate includes all necessary design, development, test, and evaluation efforts necessary for the complete OTV engine system. The major categories of cost considered were the engineering effort, fabrication, material, component test, engine system test, and associated functions, such as quality control, reliability, and program management. Each of these factors and categories were included in arriving at the OTV engine cost estimate. Details and results of the cost evaluation effort are presented separately in Volume II-B of this report.

The 10K advanced expander DDT&E costs were estimated using a WBS line item-by-item incremental costing approach with the WBS element costs for the staged combustor as reference. During the extension phase of the contract, it was recognized that the DDT&E costs for the 20K staged combustor previously given in Phase A, Task 7, were too low, mainly in the areas of turbomachinery, main combustion chamber, controls, and engine system test. Reevaluation of the DDT&E costs in these areas resulted in a DDT&E cost increase due to the following specific considerations:

1. The stringent reliability and safety requirements for the OTV engines require that a large amount of component and engine system tests need to be conducted at simulated mission conditions, specifically, at long firing durations (~2000 sec for a 20K lb. thrust engine), and with multiple starts (6 or more). These extensive mission simulation tests are necessary to develop confidence in the manrating capabilities of the OTV engines.
2. More component development work is foreseen than previously anticipated, based on ASE and SSME development experience obtained since issuance of the Phase A results. Specific attention areas are:

- a. Turbopump seals and bearings.
 - b. Low flow mixture ratio engine control
 - c. Control responses (e.g., for fast acting component redundancy management).
 - d. Preburner gas uniformity
 - e. Material procurement for test components
3. Additional government test facility modifications are foreseen due to concerns in the following areas:
- a. The OTV engine gimbal requirement of 15 degrees exceeds the AEDC gimbal envelope limit at 6 degrees.
 - b. Low enough engine exhaust pressure during start simulation are of concern. Only White Sands test facilities offer an altitude capability which is sufficiently representative for engine space starts.
 - c. Test facilities at AEDC and White Sands have some tentative test schedules for other engine programs, such as MX Stage IV and other O_2/H_2 engines. This test schedule conflict may lead to schedule delay or test stand modifications for the OTV engine.

Based on the revised staged combustor cost estimate the DDT&E ROM costs for the 10K advanced expander engine were determined for each component. Differences in thrust level, engine cycle and testing requirements were considered. Table 8 summarizes differences between the two engines influencing the DDT&E cost estimate.

TABLE 8. MAJOR DIFFERENCES AFFECTING DDT&E COST OF 10K EXPANDER

(BASIS: 20K STAGED COMBUSTOR)

● DECREASED THRUST	➔	SMALLER GEOMETRY	<u>COST EFFECT</u> DECREASE
● PREBURNER	➔	DELETED	DECREASE
● HPOTP	➔	LOWER DISCHARGE PRESSURE (-50%)	DECREASE
	➔	LOWER TURBINE TEMPERATURE(-50%)	
	➔	LOWER SPEED (-60%)	
● HPFTP	➔	LOWER DISCHARGE PRESSURE (-10%)	DECREASE
	➔	LOWER TURBINE TEMPERATURE(-50%)	
● CONTROLS	➔	LESS CRITICAL TIMING PROBLEMS	DECREASE
● HIGHER €	➔	LARGER EXIT DIAMETER	INCREASE
	➔	LARGER NUMBER OF TUBES	
	➔	SOME FACILITY MODIFICATIONS	
OVERALL EFFECT:			DECREASE BY 6%

SUMMARY AND RECOMMENDATIONS

A 1980 state-of-the-art advanced expander engine was configured that delivers near-optimum payload performance and exhibits the simplicity, reliability, and life characteristics required of the OTV mission. Engine optimization studies indicate that the performance of the conventional expander cycle engine can be improved substantially through combustion chamber surface area enhancement, thermal enhancement, turbine regeneration, and power cycle efficiency enhancement procedures. A large portion of the performance increase can be achieved with 1980 state-of-the-art component concepts while additional improvements require some degree of technology advancement in the areas of combustor heat transfer augmentation and of turbine gas regenerators.

An alternate low-thrust capability study has shown that the 20K lb. thrust staged combustion engine can be adapted to operate at extended periods in the 1 to 2K lb. thrust regime. The study results indicate that long-life, high performance operation in the low thrust mode can be achieved with minimum modifications (kitting) of the engine.

Reliability and safety studies showed that the main contribution to mission success and crew safety stems from propulsion system redundancy. However, the studies also show that significant reliability can be gained from engine component redundancy, specific engine design features ensuring safety and reliability, including an engine health monitoring system, and from a total program approach to safety and reliability, such as that presently applied to the Space Shuttle.

A comparison of the 10K lb. thrust advanced expander with the 20K lb. staged combustor indicates that the expander has some advantages in the areas of reliability, development program risk, and program cost.

The summary and recommendations of the OTV Phase A Extension engine study just completed are shown in Table 9.

TABLE 9. SUMMARY AND RECOMMENDATIONS

1. Continue technology programs for OTV staged combustion cycle with emphasis upon breadboard demonstration program.
2. The staged combustor cycle offers the highest performance engine system for OTV and the engine design is based entirely on current technology.
3. The advanced expander cycle engine is a viable contender for OTV propulsion and is attractive due to its simplicity. It is, therefore, recommended to establish an advanced expander breadboard engine program.
4. Establish a technology program for demonstration of the advanced expander component performance improvement potential.
5. Consider a trade-off study to determine the benefits on the total OTV program level of higher staged combustor performance versus the higher reliability, lower development risk and lower cost of the advanced expander.
6. Further studies of engine design to improve accessibility for inspection and component removal are recommended for both cycles.
7. Demonstrate experimentally the low thrust capability of the staged combustion engine components and of the engine system.
8. Capitalize on SSME and ASE technology base and experience in the design of a new OTV engine.