# Orbit Transfer Vehicle (OTV) Advanced Expander Cycle Engine Point Design Study

Contract NAS 8-33574
Final Report
Volume I: Executive Summary
December 1980

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

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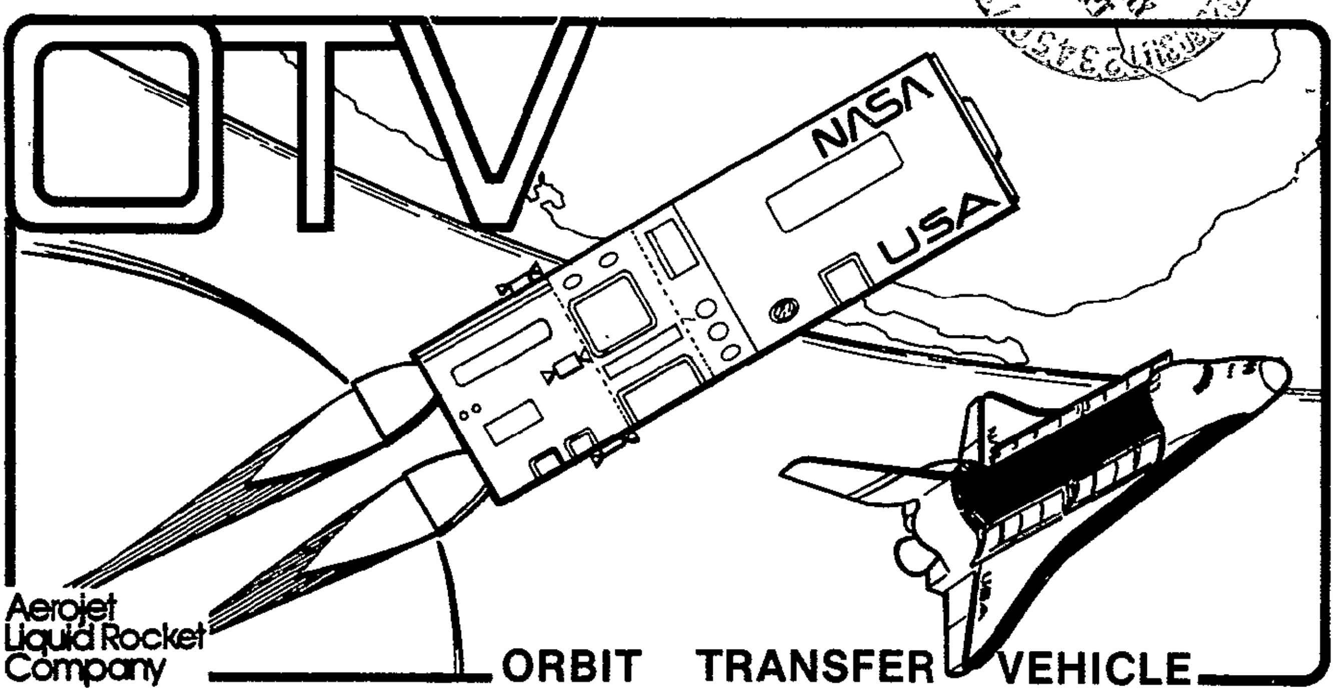


Aerojet
Liquid Rocket
Company

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# ORBIT TRANSFER VEHICLE (OTV)
# ADVANCED EXPANDER CYCLE ENGINE
# POINT DESIGN STUDY

Final Report

Contract NAS 8-33574

## VOLUME I: EXECUTIVE SUMMARY

Prepared For:

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## FOREWORD

This final report is submitted for the Orbit Transfer Vehicle (OTV) Advanced Expander Cycle Engine Point Design Study per the requirements of Contract NAS 8-33574, Data Procurement Document No. 578, Data Requirement No. MA-05. This work was performed by the Aerojet Liquid Rocket Company (ALRC) for the NASA/Marshall Space Flight Center.

The study consisted of the generation of a performance-optimized engine system design for an advanced LOX/Hydrogen expander cycle engine. The designs of the components and engine were prepared in sufficient depth to calculate engine and component weights and envelopes, turbopump efficiencies and recirculation leakage rates, and engine performance. Engine control techniques were established, and new technology requirements were identified.

The NASA/MSFC COR was Mr. D. H. Blount. The ALRC Program Manager was Mr. L. B. Bassham, and the Study Manager was Mr. J. A. Mellish.

The final report is submitted in two volumes:

Volume I: Executive Summary
Volume II: Study Results

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# I. INTRODUCTION

## A. BACKGROUND

The Space Transportation System (STS) includes an orbit transfer vehicle (OTV) that is carried into low-Earth orbit (LEO) 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 the DoD have been studying various types of OTV's in recent years. Data have been accumulated from the analyses of the various concepts, operating modes, and projected missions. With the inclusion of man in these transportation scenarios, it becomes necessary to reach for the safest and most fully optimized propulsion stage.

The purpose of this study was to generate a performance-optimized engine system design for a man-rated, advanced LOX/Hydrogen expander cycle engine. This concept was originally conceived by ALRC on the OTV Phase A Engine Study, Contract NAS 8-32999, and recommended to NASA in October 1978. The recommendation was based upon a thorough evaluation of the vehicle's performance, envelope, reusability, and man-rating requirements, along with a desire to reduce the development risk of the OTV engine. Our engine cycle recommendation was approved by NASA and led to further evaluations of the advanced expander cycle engine, including this point design effort.

## B. ORBIT TRANSFER VEHICLE CHARACTERISTICS

The Manned Orbit Transfer Vehicle (MOTV) 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. This vehicle is planned to be a cryogenic stage, with the baseline design mission being a four-man, 30-day sortie to geosynchronous orbit (GEO). The required round trip payload to GEO and return to low-Earth orbit (LEO) is 13,000 lbm. 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 rendezvous with the Orbiter in LEO. 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 objectives of the OTV Advanced Expander Cycle Engine Point Design Study were to (1) generate a performance-optimized engine system design for an advanced LOX/Hydrogen expander cycle engine; (2) provide sufficient design and analysis of the engine and components to produce accurate engine and component weights and envelopes, turbopump efficiencies and recirculation leakage rates, and engine performance; (3) establish engine control techniques; and (4) identify new technology requirements.

Specific study objectives were to accomplish the following:

- Prepare detailed computer models of the engine to predict both the steady-state and transient operation of the engine system.

- Prepare mechanical design layout drawings of the following components:

  - Thrust chamber and nozzle
  - Extendible nozzle actuating mechanism and seal
  - LOX turbopump
  - LOX boost pump
  - Hydrogen turbopump
  - Hydrogen boost pump
  - Propellant control valves

- Perform the necessary heat transfer, stress, fluid flow, dynamic and performance analysis to support the mechanical design.

- Determine effective control points and methods to control the engine operation through start and shutdown transients as well as steady-state operation. These include thrust and mixture ratio control.

II, Study Objectives and Scope (cont.)

- Determine optimum actuation drive methods for engine control elements.

- Define controller requirements.

- Prepare an engine configuration layout drawing to show the spatial arrangement of the various engine components with regard to system effectiveness, safety, and the impact upon maintainability as well as engine performance.

- Prepare an engine data summary to include the engine and component layout drawings, the performance and life predictions, and the engine and component weights and physical envelopes.

- Identify any new technology required to perform detailed design, construction and testing of the engine.

- Prepare and deliver computer software/documentation for the steady-state and transient engine models.

- Prepare a final report at the completion of the study which documents the technical details and programmatic assessments resulting from the study. The final report is submitted in two volumes:

  Volume I: Executive Summary
  Volume II: Study Results

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

II, Study Objectives and Scope (cont.)

- Task I - Steady-State Computer Model
- Task II - Heat Transfer, Stress, and Fluid Flow Analysis
- Task III - Component Mechanical Design and Assembly Drawings
- Task IV - Engine Transient Simulation Computer Model
- Task V - Engine Control
- Task VI - Engine Configuration Layout
- Task VII - Engine Data Summary
- Task VIII - Technology Requirements
- Task IX - Computer Software Documentation
- Task X - Reporting and Performance Reviews

## III. ENGINE REQUIREMENTS AND PRINCIPAL ASSUMPTIONS

### A. REQUIREMENTS

The requirements for the manned orbit transfer vehicle (MOTV) engine were derived from numerous NASA in-house and contracted studies and are summarized on Table I. The engine thrust level of 15,000 lbF was selected by NASA for the OTV engine point design studies.

In addition to the specified requirements, our Phase A OTV Engine Study identified design impacts resulting from the man-rating, safety, and reliability requirements. These are summarized on Table II. A multiple-engine installation is necessary to meet the crew safety requirements. The series-redundant main propellant valves are required to assure that the engine will shut down. They also inhibit leakage of propellant through the engine into the Orbiter's payload bay. Redundant spark ignition is required to assure that the engine will start on all burns. Dual-coils are required to assure that the actuator will function and provide sufficient force to open critical valves.

All of the specified and identified requirements were incorporated in our point design.

### B. ASSUMPTIONS AND GUIDELINES

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

1. All engine designs and characteristics will be compatible with the OTV requirements and will be based on 1980 technology.

## TABLE I

## OTV ENGINE POINT DESIGN REQUIREMENTS

Rated Vacuum Thrust: 15,000 lb

Propellants: Hydrogen and Oxygen

Power Cycle: Expander

Technology Base: 1980 State-Of-The-Art

Engine Mixture Ratio: Nominal = 6.0 Range = 6.0 to 7.0

Propellant Inlet Conditions:

| | | $H_2$ | $O_2$ |
|---|---|---|---|
| Boost Pump | NPSH, ft | 15 | 2 |
| | Temp., °R | 37.8 | 162.7 |

Service Life Between Overhauls: 300 Cycles or 10 Hours

Service-Free Life: 60 Cycles or 2 Hours

Engine Nozzle: Contoured Bell with Extendible/Retractable Section

Maximum Engine Length with Nozzle Retracted: 60 in.

Gimbal Angle: +15°, -6° Pitch

±6° Yaw

Provide Gaseous Hydrogen & Oxygen Tank Pressurization

Man-Rated with Abort Return Capability

Meet Orbiter Safety and Environmental Criteria

Max $P_c$ Deviations: ±5% of Steady-State Pressure

Adaptable to Extended Low-Thrust Operation (∿1.5K lbF)

## TABLE II

## MAN-RATING, SAFETY, AND RELIABILITY-IMPOSED REQUIREMENTS

Engine Should Be Designed For A Multi-Engine Installation (Preferably Twin Engines

Series-Redundant Main Propellant Valves Required

Redundant Spark Igniter Required

Dual Coils Will Be Used On All Valves Identified By FMEA As Single Point Failures

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

3. The engine and OTV will be designed for return to Earth in the Shuttle and subsequent reuse. 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.

## C. 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 with demonstrated adherence to the factors specified below. Where structural testing is not feasible, more conservative structural design factors will be supplied by the procuring agency.

1. The structures shall not experience gross yielding (total net section) 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.

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

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

4. The structure shall be capable of withstanding at least 4 times the limit life based on lower-bound fatigue property data.

5. 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.

## IV. SUMMARY OF STUDY RESULTS

This section summarizes the significant achievements or results obtained in the conduct of each major study technical task.

It should be noted that, due to schedule and funding limitations, many of the tasks and subtasks were conducted in parallel. Therefore, it was not possible to conduct design iterations, nor could the results of all tasks and subtasks be entirely incorporated into the engine designs and data presented herein. Additional design definition should be prepared in future study efforts to further optimize the advanced expander cycle engine.

### A. TASK I - STEADY-STATE COMPUTER MODEL

The objective of this task was to provide a computer model of the engine system steady-state operation.

An existing ALRC computer model was modified to simulate the ALRC advanced expander cycle engine. This modified computer model, designated OTVMOD7, is a FORTRAN computer program which performs the engine cycle power balance and performance predictions for design and off-design operation of the engine. The off-design operation encompasses a mixture ratio range from 5 to 10 and thrust levels from the nominal 15,000 lbF to low-thrust operation at 1.5K lbF. This model was delivered to NASA/MSFC as part of Task IX. The user's manual for this program was issued as a separate report under this contract.

The computer model is compatible with the Univac 1108 computer and requires approximately 10 seconds to run each case.

A sample program output is presented in Table III. The output displays engine performance, envelope, and weight data on one page and the engine pressure schedule and power balance parameters on the second page. The

## TABLE III

## OTV EXPANDER CYCLE ENGINE STEADY-STATE MODEL SAMPLE OUTPUT

### ENGINE PERFORMANCE

| | |
|---|---|
| 1.THRUST(LBF) | 15000.00 |
| 2.CHAMBER PRESSURE(PSIA) | 1200.02 |
| 3.MIXTURE RATIO | 6.00 |
| 4.TOTAL FLOWRATE(LBM/SEC) | 31.55 |
| 5.LOX FLOWRATE(LBM/SEC) | 27.04 |
| 6.FUEL FLOWRATE(LBM/SEC) | 4.51 |
| 7.ISP ODE(SECONDS) | 486.11 |
| 8.NOZZLE EFFICIENCY | .9929 |
| 9.ENERGY RELEASE EFFICIENCY | 1.0000 |
| 10.KINETIC EFFICIENCY | .9957 |
| 11.BOUNDARY LAYER LOSS (LB) | 163.32 |
| 12.ISP DELIVERED(SECONDS) | 475.40 |
| 13.THRUST TO WEIGHT RATIO | 26.11 |

### ENGINE SIZE (IN AND IN**2)

| | |
|---|---|
| 1.GIMBAL LENGTH | 2.390 |
| 2.INJECTOR LENGTH | 4.820 |
| 3.CHAMBER LENGTH | 18.000 |
| 4.NOZZLE LENGTH | 84.390 |
| 5.ENG LENG STOWED | 60.000 |
| 6.ENG LENG DEPLYD | 109.600 |
| 7.EXIT DIAMETER | 58.163 |
| 8.THROAT RADIUS | 1.395 |
| 9.AREA RATIO | 435. |
| 10.CU AREA RATIO | 11. |
| 11.TUBE BNDL AREA RATIO | 172. |
| 12.PERCENT BELL | 81.80 |
| 13.X/RT | 60.49 |
| 14.PERCENT RAD | 108.00 |

### ENGINE WEIGHTS (LBM)

| | |
|---|---|
| 1.GIMBAL | 3.30 |
| 2.INJECTOR | 30.60 |
| 3.CHAMBER | 47.30 |
| 4.COPPER NOZZLE | 27.00 |
| 5.TUBE BNDLE NOZZL | 38.40 |
| 6.RAD NOZZLE | 80.00 |
| 7.NOZZLE DEPLOY SYS | 72.00 |
| 8.FUEL RICH PREBRNRS | .00 |
| 9.VALVES AND ACTUATORS | 72.70 |
| 10.LOX BOOST PUMP | 5.60 |
| 11.LH2 BOOST | 8.50 |
| 12.LOX TPA (HI SPD) | 26.90 |
| 13.LH2 TPA (HI SPEED) | 26.30 |
| 14.MISC. VALVES | 12.60 |
| 15.LINES | 37.00 |
| 16.IGNITION SYSTEM | 9.20 |
| 17.ENGINE CONTROLLER | 35.00 |
| 18.MISCELLANEOUS | 37.00 |
| 19.HEAT EXCHANGER | 5.00 |
| 20.TOTAL ENGINE WEIGHT | 574.40 |

## TABLE III (cont.)

POWER BALANCE
EXPANDER CYCLE:
SERIES TURBINES
NO REHEAT
BOOST PUMPS

PRESSURE SCHEDULE (PSIA)

| | FUEL CIRCUIT | LOX CIRCUIT |
|---|---|---|
| 1.PUMP INLET | 49.09 | 48.08 |
| 2.PUMP PRESSURE RISE | 2510.92 | 1439.66 |
| 3.PUMP DISCHARGE | 2560.00 | 1487.74 |
| 4.LINE PRESSURE DROP | 10.08 | 25.19 |
| 5.VALVE INLET | 2549.93 | 1462.55 |
| 6.VALVE PRESSURE DROP | 24.82 | 14.73 |
| 7.VALVE OUTLET | 2525.11 | 1447.82 |
| 8.LINE PRESSURE DROP | 30.23 | 15.11 |
| 9.COOLANT JACKET INLET | 2494.88 | -- |
| 10.COOLANT JACKET PRESSURE DROP | 92.00 | -- |
| 11.COOLANT JACKET OUTLET | 2402.88 | -- |
| 12.LINE PRESSURE DROP | 30.23 | -- |
| 13.FUEL CIRCUIT TURBINE INLET | 2372.65 | -- |
| 14.FUEL CIRCUIT TURBINE PRESSURE RAT. | 1.568 | -- |
| 15.FUEL CIRCUIT TURBINE EXIT | 1551.55 | -- |
| 15A.BETWEEN TURBINES PRESSURE DROP | 48.14 | -- |
| 15B.OX CIRCUIT TURBINE INLET | 1503.41 | -- |
| 15C.OX CIRCUIT TURBINE PRESSURE RAT. | 1.132 | -- |
| 15D.OX CIRCUIT TURBINE EXIT | 1362.40 | -- |
| 16.LINE PRESSURE DROP | 34.09 | -- |
| 17.TCA INJECTOR INLET | 1328.31 | 1432.70 |
| 18.TCA INJECTOR PRESSURE DROP | 110.41 | 214.80 |
| 19.TCA INJECTOR FACE | 1217.90 | 1217.90 |
| 20.TCA PRESSURE DROP | 17.88 | 17.88 |
| 21.CHAMBER PRESSURE | 1200.02 | 1200.02 |

FLOWRATES (LBM/SEC)
TEMP DROP (DEGREES R)
CP(BTU/LBM-R)
(FC=FUEL CIRCUIT)
(OC=OX CIRCUIT)
(T-S=TOTAL TO STATIC TEMP)
(T=TOTAL TEMP)

| | |
|---|---|
| 1.FC TURBINE FLOW | 4.24 |
| 2.OC TURBINE FLOW | 4.24 |
| 3.FC TURB T DROP(T-S) | 64.01 |
| 4.OC TURB T DROP(T-S) | 16.74 |
| 5.FC TURB INLET T(T) | 535.00 |
| 6.FC TURB EXIT T(T) | 485.84 |
| 7.OC TURB IN T(T) | 485.84 |
| 8.OC TURB EXIT T(T) | 474.67 |
| 9.DRIVE GAS CP | 3.652 |
| 10.DRIVE GAS GAMMA | 1.395 |
| 11.BYPASS FLOW | .27 |

HORSEPOWERS
AND EFFICIENCIES

(FC=FUEL CIRCUIT)
(OC=OX CIRCUIT)
(T-S=TOTAL TO STATIC TEMP)
(T=TOTAL TEMP)

| | |
|---|---|
| 1.FC TURB HORSEPOW | 1076.05 |
| 2.OC TURB HORSEPOW | 244.34 |
| 3.FL PUMP SHP | 1076.05 |
| 4.OX PUMP SHP | 244.34 |
| 5.FC TURB EFF | .768 |
| 6.OC TURBINE EFF | .667 |
| 7.FUEL PUMP EFF | .633 |
| 8.OX PUMP EFF | .615 |
| 9.OX FLOW | 27.04 |
| 10.TOTAL FUEL FLO | 4.51 |

output shown on the table is an updated baseline that reflects the modifications that resulted from the study.

In addition to this power balance model, more detailed steady-state evaluations can be conducted with the engine transient computer model described in Section IV,D. This model has more detailed analytical descriptions and simulations of all of the engine components incorporated in it. The steady-state condition is just a special case of the transient condition in this model.

The steady-state power balance model (OTVMOD7) is recommended for preliminary design studies, while the steady-state case should be evaluated on the transient model during detailed engine design studies.

## B. TASK II - HEAT TRANSFER, STRESS, AND FLUID FLOW ANALYSIS

The objective of this task was to provide the analyses required to support the engine and engine component mechanical design parameters.

The advanced expander cycle engine (AEC) coolant flow schematic is shown on Figure 1. This coolant scheme was selected as a result of optimization studies conducted for the Phase A and Phase A Extension OTV Engine Study work. Eighty-five (85) percent of the hydrogen flow is used to cool the chamber in a single pass from an area ratio of 10.6:1 to the injector end. Fifteen (15) percent of the hydrogen is used to cool the fixed nozzle in parallel with the chamber in a two-pass tube bundle. The temperature data on the figure is shown for the design point thrust and mixture ratio of 15,000 lb and O/F = 6.0. The thermal analysis results are summarized on Table IV for

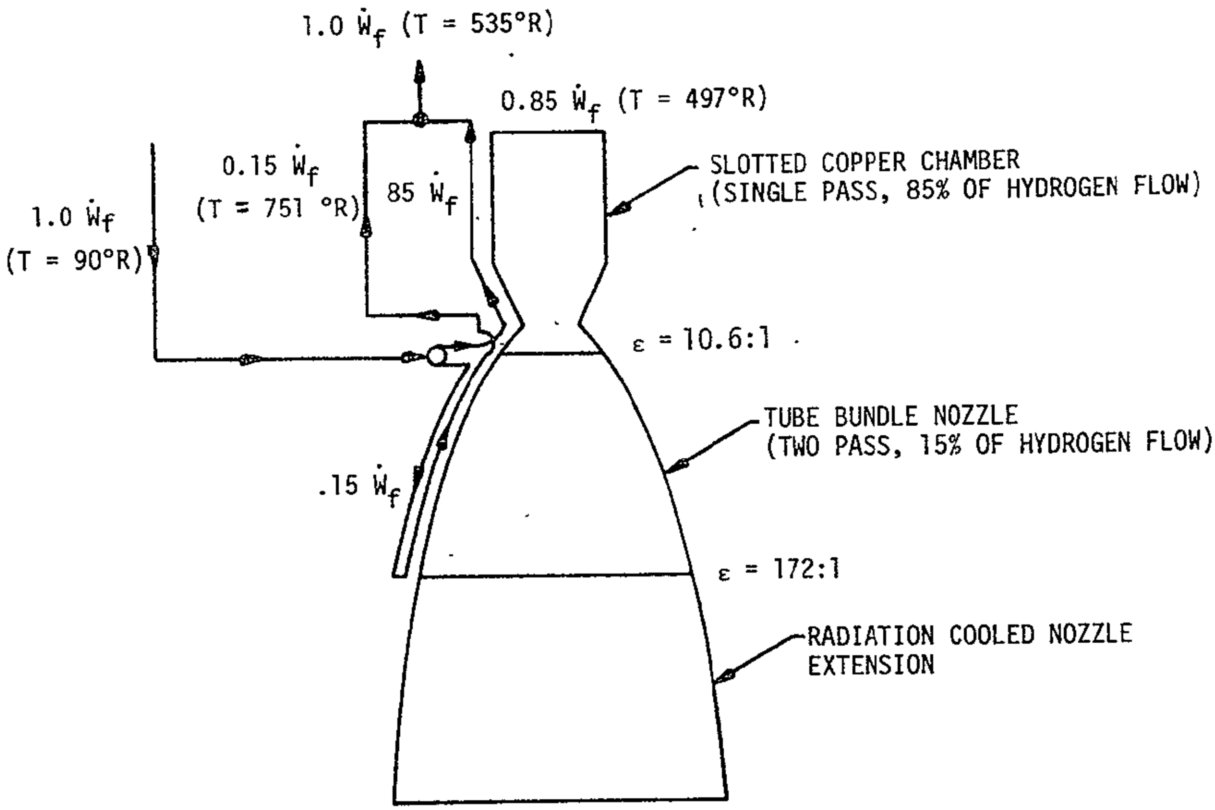



Figure 1. AEC Engine Coolant Flow Schematic

## TABLE IV

## AEC THERMAL ANALYSIS DESIGN & OFF-DESIGN O/F SUMMARY AT RATED THRUST

| | MIXTURE RATIO | |
|---|---|---|
| | 6.0 | 7.0 |
| COMBUSTION CHAMBER COOLANT FLOW RATE, LB/SEC | 3.816 | 3.358 |
| SLOTTED COPPER CHAMBER AREA RATIO | 10.6 | 10.6 |
| CHAMBER PRESSURE DROP, PSIA | 92 | 76 |
| COOLANT INLET TEMPERATURE, °R | 90 | 90 |
| CHAMBER COOLANT TEMPERATURE RISE, °R | 407 | 431 |
| FIXED TUBE BUNDLE NOZZLE FLOWRATE LB/SEC. | 0.674 | 0.592 |
| TUBE BUNDLE NOZZLE AREA RATIO | 172 | 172 |
| TUBE BUNDLE COOLANT PRESSURE DROP, PSIA | 11 | 8 |
| TUBE BUNDLE COOLANT TEMPERATURE RISE, °R | 661 | 672 |
| TURBINE INLET TEMP., °R | 535 | 557 |

the design and off-design mixture ratio conditions. Pressure drop data shown is for the losses in the channels or tubes only and does not include the manifold losses.

The thermal analyses have also shown that the chamber life at low-thrust operation is not penalized and is, in fact, better. The tube bundle must be designed for the low-thrust operating point in order to meet the life requirement. This results in smaller, higher pressure drop tubes. The higher pressure drop tubes are not a problem and do not penalize the engine at rated thrust because the tube bundle pressure drop is on the order of 10 psi compared to a chamber coolant pressure drop in excess of 90 psi.

The thermodynamic, hydraulic, and performance analyses resulted in recommending the injector and combustion chamber parameters shown on Table V to support the mechanical design.

Analysis also showed the feasibility of a 10% of rated thrust operating point through engine "kitting." The recommended TCA "kits" are smaller diameter oxidizer coaxial injection elements and an orifice in the hydrogen line downstream of the thrust chamber coolant jacket exit. The new oxidizer injection elements are required to avoid chugging instability. The orifice is required to maintain the coolant jacket exit pressure well above the critical pressure of hydrogen to avoid the problems associated with two-phase coolant flow.

Stress and low cycle fatigue analyses were performed to substantiate the structural adequacy of the baseline chamber design. The chamber channel geometries were obtained from thermal analysis and are summarized below.

## TABLE V

## OTV THRUST CHAMBER DESIGN RECOMMENDATIONS

### INJECTOR

| | | |
|---|---|---|
| Injection Element Type: | | Swirl Coaxial Element |
| Coaxial Element Quantity: | | 84 |
| No. Rows: | | 4 (30 + 24 + 18 + 12 = 84) |
| Oxidizer Metering Orifice Dia. = | | .100 (in.) |
| Oxidizer Swirl Cone Angle | = | 30° half angle |
| Oxidizer Element Tip OD / Fuel Annulus ID | = | .150 (in.) |
| Fuel Annulus OD | = | .200 (in.) |
| Oxidizer Element Tip Recess | = | .100 (in.) |

### COMBUSTION CHAMBER

| | | |
|---|---|---|
| Chamber Length, L' | = | 18 (in.) |
| Chamber Diameter | = | 5.34 (in.) |
| Throat Diameter | = | 2.79 (in.) |
| Upstream Radius = 1.0 RT | = | 1.395 (in.) |
| Convergent Inlet Radius = 3.0 RT | = | 4.185 (in.) |
| Alpha Inlet Angle, $\alpha_i$ | = | 20° |
| Downstream Radius = 1.0 RT | = | 1.395 |
| Downstream Throat Tangency Angle, $\theta_t$ | = | 41° |

- <u>Throat</u>
  Slot Width = 0.04 in.
  Slot Depth = 0.121 in.
  Web (land) Thickness = 0.04 in.
  Wall Thickness = 0.030 in.

- <u>Cylindrical Section</u>
  Slot Width = 0.070 in.
  Slot Depth = 0.256 in.
  Web (land) Thickness = 0.081 in.
  Wall Thickness = 0.030 in.

The results indicate that the zirconium copper liner is structurally adequate to sustain the predicted differential pressures. The electroformed nickel closeout is assumed to carry all pressure hoop membrane loads, and the required thickness was determined on that basis. Minimum closeout thickness is 0.035 in. for the throat region and 0.070 in. for the cylindrical section.

The low cycle fatigue prediction is based on an elastic/plastic iterative plane strain solution of wall sections taken from the throat and cylindrical regions. The service life ($N_f$) predicted for the chamber is based on the maximum effective strain, determined for either section, using a factor of 4 on the lower-bound design curve for zirconium-copper. Results determined from this investigation are presented below.

| Location | $T_{GS}$ (°F) | $T_{BS}$ (°F) | EFNI t(IN) | $\epsilon_T$ (%) | $N_f$ Cycles | $N_{REQ}$ Cycles | $K_E$ |
|---|---|---|---|---|---|---|---|
| Throat | 615 | -219 | .035 | 1.11 | 460 | 300 | 1.4 |
| Cylinder | 757 | 33 | .070 | 1.265 | 350 | 300 | 1.87 |

$T_{GS}$ = gas-side temperature
$T_{BS}$ = backside temperature
EFNIt = Electroformed Nickel Closeout Thickness
$\epsilon_T$ = Total Strain
$N_f$ = Number of Cycles (includes factor of 4)
$K_E$ = Strain Concentration Factor

The results show that the chamber meets the service life requirement.

The hydraulic design analyses results for the main oxygen pump are summarized on Table VI. The main pump is sized for an inlet pressure of 48 psia which is delivered by the boost pump. The inducer has a 10° inlet angle which results from the inlet flow coefficient and the incidence to blade angle ratio. Even with a boost pump, the main pump inducer must operate at a suction specific speed of 21,000 $(RPM)(GPM)^{1/2}/(FT)^{3/4}$. At this value, the head loss is small, and the rotating speed is limited to 34,720 RPM.

The main fuel pump raises the $LH_2$ pressure from 51 psia to 2473 psia and delivers a flowrate of 4.49 lb/sec (456 GPM). The turbopump consists of three centrifugal pump stages and an axial flow inducer driven by a two-stage, warm-gas hydrogen turbine. The design point speed is 90,000 RPM, slightly below the bearing DN limit speed of 100,000 RPM which can be obtained with 20 mm bearings. The three main stages provide an $N_s$ value near 1,000, which results in high efficiency without excess complexity. The high-speed inducer provides a high static inlet pressure to avoid vapor generation at the centrifugal stage inlet caused by the high enthalpy fluid returning to the inlet. Pertinent hydraulic design parameters for the baseline main hydrogen turbopump are summarized on Table VII.

Design point analyses were also conducted to determine the design parameters and performance of the fuel and oxygen pump turbines. The results are summarized below and on Table VIII.

TABLE VI

$LO_2$ MAIN PUMP DESIGN PARAMETERS

| | Inducer | | Impeller |
|---|---|---|---|
| Speed, RPM | 34,720 | | 34,720 |
| Inlet Pressure, psia | 48.0 | | 270 |
| Inlet Temperature, °R | 164.3 | | - |
| NPSH, ft. | 66 | | 380 |
| Flow, GPM | 196.2 | | 226 |
| Specific Speed, RPM * $GPM^{1/2}$ * $Ft^{-3/4}$ | 5,000 | | 1,500 |
| Suction Specific Speed, RPM * $GPM^{1/2}$ * $Ft^{-3/4}$ | 21,000 | | 6,000 |
| Inlet Diameter, in. | 1.77 | | 1.77 |
| Exit Diameter, in. | 1.77 | | 2.85 |
| Inlet Blade Angle, Degree | 10 | | - |
| Exit Blade Angle, Degree | 10 | | 25 |
| Head Coefficient | .2 | | .49 |
| Flow Coefficient Inlet | .114 | | .20 |
| Flow Coefficient Exit | .17 | | .12 |
| No. Vanes | 4 | | 9 |
| Head Rise, ft. | 450 | | 2,390 |
| Efficiency, % | 75 | | 71 |
| Delivered Weight Flow, lb/sec | | 30.1 | |
| Delivered Head, ft. | | 2,840 | |
| Combined Efficiency, % | | 61.5 | |
| Required Horsepower, SHP | | 252.5 | |

TABLE VII

BASELINE $LH_2$ MAIN PUMP DESIGN PARAMETERS

| Inducer | Baseline |
|---|---|
| Head Rise, ft. | 3,000 |
| Flow, GPM | 547 |
| Efficiency, % | 80 |
| Specific Speed, RPM * GPM $^{1/2}$ * FT $^{-3/4}$ | 5,202 |
| Suction Specific Speed, RPM * GPM $^{1/2}$ * FT $^{-3/4}$ | 11,238 |
| Tip Diameter, in. | 1.90 |
| Power, HP | 36.5 |
| Head Coefficient | .17 |
| Blades | 4 |
| **Stage I** | |
| Head Rise, ft. | 25,302 |
| Flow, GPM | 572 |
| Efficiency, % | 66.5 |
| Specific Speed, RPM * GPM $^{1/2}$ * FT $^{-3/4}$ | 1,072 |
| Tip Diameter, in. | 3.37 |
| Power, HP | 374 |
| Head Coefficient | .464 |
| Blades | 10 |
| **Stage II/III** | |
| Head Rise, ft. | 25,302 |
| Flow, GPM | 478 |
| Efficiency, % | 65.0 |
| Specific Speed, RPM * GPM $^{1/2}$ * FT $^{-3/4}$ | 981 |
| Tip Diameter, in. | 3.37 |
| Power, HP | 331 |
| Head Coefficient | .464 |
| Blades | 10 |
| **Overall** | |
| Head Rise, ft. | 78,910 |
| Flow, GPM | 456 |
| Efficiency, % | 59.8 |
| Power, HP | 1,072 |

## TABLE VIII

## TURBINE DESIGN POINT PERFORMANCE

### $LH_2$ TPA Gas Turbine

| Parameter | First Stage | Second Stage | Both Stages | Overall Turbine, Including Manifolds |
|---|---|---|---|---|
| Work, BTU/lb | 87.7 | 87.7 | 175.4 | 175.4 |
| Flowrate, lb/sec | 4.08 | 4.08 | 4.08 | 4.08 |
| Static Pressure Ratio | 1.241 | 1.236 | 1.505 | 1.543 |
| Stage Loading Factor | 0.7 | 0.7 | - | - |
| Stage Flow Coefficient | 0.308 | 0.308 | - | - |
| Velocity Ratio | 0.555 | 0.557 | 0.398 | 0.387 |
| Static Efficiency, % | 80. | 80.7 | 82.2 | 77.7 |

The estimated disk friction loss is 11.3 HP. When this loss is included, the overall turbine efficiency is 76.8%.

### $LO_2$ TPA Gas Turbine

| Parameter | Stage | Overall Turbine, Including Manifolds |
|---|---|---|
| Work, BTU/lb | 40.8 | 40.8 |
| Flowrate, lb/sec | 4.15 | 4.15 |
| Static Pressure Ratio | 1.105 | 1.130 |
| Stage Loading Factor | 0.60 | - |
| Stage Flow Coefficient | 0.328 | - |
| Velocity Ratio | 0.533 | 0.483 |
| Static Efficiency, % | 82.3 | 67.7 |

The estimated disk friction loss is 3.86 HP. When this loss is included, the overall turbine efficiency is 66.7%.

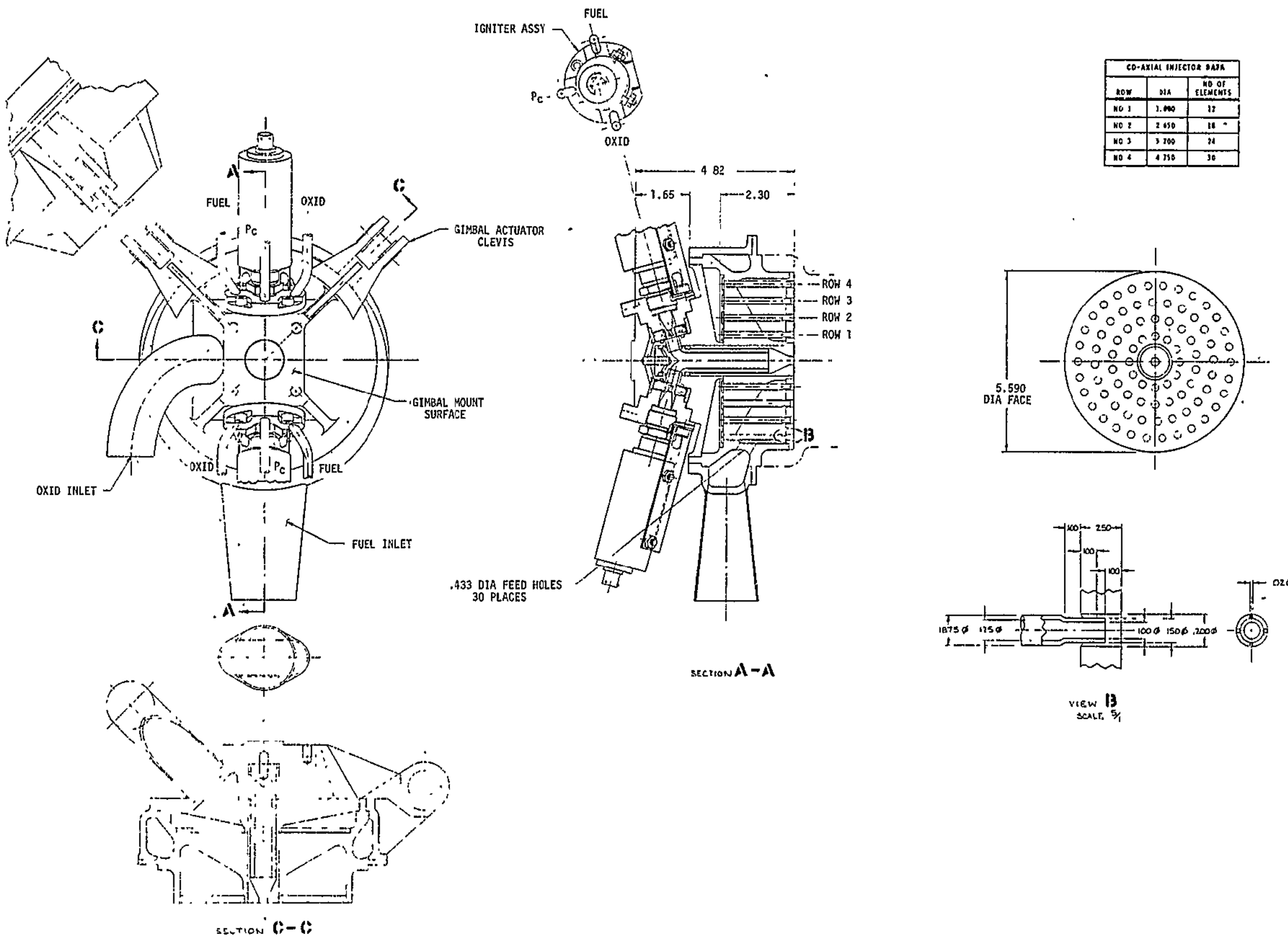
| Hydrogen Pump Turbine | Oxygen Pump Turbine |
|---|---|
| Two-Stage Reaction Turbine | Single-Stage Reaction Turbine |
| ° Pitch Diameter = 3.31 inches | ° Pitch Diameter = 5.545 inches |
| ° Overall Velocity Ratio = 0.387 | ° Overall Velocity Ratio = 0.483 |
| ° Estimated Static Efficiency = 76.8% | ° Estimated Static Efficiency = 66.7% |

## C. TASK III - COMPONENT MECHANICAL DESIGN AND ASSEMBLY DRAWINGS

The primary objective of this task was to provide mechanical design and assembly drawings of the major engine components. The engine designs were prepared in only sufficient depth to reveal manufacturing difficulties, allow leakage and cooling flows to be assessed, calculate weights, and determine technology requirements. Therefore, the resulting designs are not firm, and further iterations in the designs and data can be expected as the advanced expander cycle engine design matures.

### 1. Igniter/Injector Assembly

The OTV ignition system employs redundant igniters to meet the man-rating requirement; these are shown mounted on the OTV injector in Figure 2. Each igniter is a small thruster which can accept either liquid two-phase, or gaseous propellants. These are ignited by using a very low energy spark. The igniter produces a hot gas torch of sufficient energy to provide reliable rapid main stage ignition. The capacitance discharge power supply is integral with the spark plug. The igniter assembly is located in the injector oxidizer manifold cover plate directly below the gimbal mount surface. The igniter design is based upon concepts which were demonstrated




| CO-AXIAL INJECTOR DATA | | |
|---|---|---|
| ROW | DIA | NO OF ELEMENTS |
| NO 1 | 1.000 | 12 |
| NO 2 | 2 650 | 18 |
| NO 3 | 3 700 | 24 |
| NO 4 | 4 750 | 30 |

Figure 2. Igniter/Injector Assembly

under the integrated thruster assembly (ITA) and extended temperature range (ETR) programs conducted for NASA/LeRC. The igniter flows during steady-state operation to keep the plenum cool.

A layout of the injector configuration is also shown on Figure 2. The injector uses coaxial elements because this element type has an extensive history of operation with $GH_2/LO_2$ propellants over a broad range of thrusts and chamber pressure. The injection pattern is a four-row array of 84 elements uniformly distributed over the injector face. The oxidizer tubes, which are recessed approximately one tube diameter into the faceplate, are held concentrically within the fuel discharge orifice by four small tabs integral with the faceplate.

Since the hydrogen is injected as a gas and the cryogenic LOX immediately flashes into a gas upon injection into the hot combustion chamber, combustion is expected to occur very close to the injector face. This results in the need for injector face cooling. Regenerative cooling of the injector face, coupled with discrete face fuel film cooling, provides the most reliable face cooling. The injector faceplate material is a laminate of photoetched copper face platelets brazed to a structural steel backup plate. The extremely accurate photoetched flow control passages assure uniform flow across the entire injector face. The faceplate concept precludes problems associated with flow control and flow distribution encountered with Rigimesh, a stainless steel wire material used on a variety of rocket engines with varied success.

Although combustion instability problems are not expected, a resonator cavity is provided around the injector periphery. The cavity is designed so that dynamic pressure oscillations do not exceed $\pm$ 5%.

2. Chamber and Tube Bundle Nozzle

A design layout for the combustion chamber is illustrated in Figure 3. The chamber ID of 5.34 in. is cylindrical for 13.54 in. and then converges at a 30° half angle to a throat diameter of 2.79 in. The thrust chamber gas-side wall contains 113 coolant slots which are equally spaced in a zirconium copper liner. The coolant slots are closed out with electroformed nickel. The coolant enters the chamber at an area ratio of 10.6. The inlet manifold is located 4.20 in. below the throat. The coolant flows axially for a distance of 22.20 in. toward the forward end of the chamber. The coolant is collected in a manifold outboard of the resonator cavities and exits radially into the chamber's coolant outlet manifold.

The selection of thrust chamber geometry (contraction ratio and combustor length) is influenced by engine cycle considerations. Optimization studies during the Phase A engine work resulted in the selection of a 3.66 contraction ratio and a combustor length, L', of 18 in.

Twelve resonator cavities are located at the forward end of the thrust chamber and are bounded by the injector body outside diameter and chamber inside diameter. The coolant passages in line with the 12 partitions between the resonator cavities will extend partially through the partition to regeneratively cool the partition.

A conical support structure surrounds the entire chamber to provide a load path for gimbal-induced loads from the nozzle extension as well as a means of attachment for engine components. The conical support structure has a circumferential channel located at its inside diameter approximately midway along its length to provide additional structural rigidity. Openings in the cone provide access for bolting components and routing propellant lines.

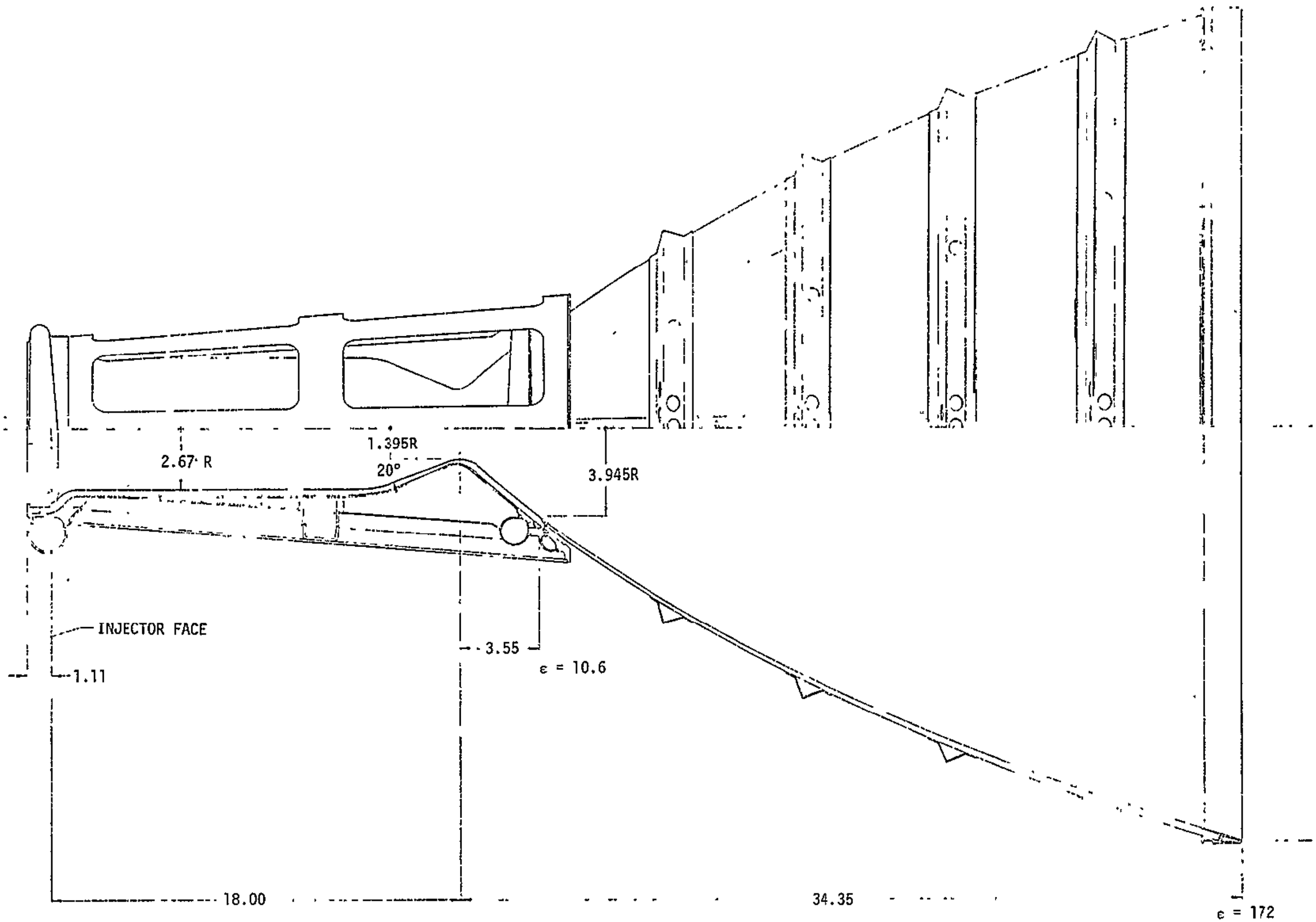



Figure 3. Chamber and Tube Bundle Nozzle

As shown on Figure 3, the regeneratively cooled tube bundle nozzle is physically attached to the chamber at an area ratio of 10.6:1. A two-pass tube bundle configuration was selected. A total of 326 coolant tubes are spaced equally around the nozzle contour. The tubes taper from 0.089 in. OD with a wall thickness of 0.007 in. at the forward end to 0.361 in. OD with a wall thickness of 0.010 in. at the aft end. A-286 has been preliminarily selected as the tube bundle material in this design although further study is required.

### 3. Radiation-Cooled Nozzle Extension and Deployment Mechanism

The radiation cooled nozzle extension assembly is shown as part of the engine assembly layout (see Figure 10). It consists of a contoured, radiation cooled nozzle, an extension/retraction mechanism, and a mechanical drive system.

Design of the radiation cooled nozzle is based upon the OMS engine nozzle design. The nozzle material is C 103 columbium alloy. An oxidation resistant coating (an aluminide or silicide slurry) is required on all surfaces of the nozzle.

The radiation cooled nozzle extension that is exposed to the hot gases is 49.6 in. long. A thin wall cylindrical ring assembly, approximately 10.4 in. long and containing the nozzle attachment flange, is an integral part of the nozzle. This ring assembly is not exposed to the hot products of combustion. Its function is to permit extension of the nozzle flange up to the deployment ring where it is bolted to the extension/retraction mechanism. The radiation-cooled nozzle incorporates a bolt-on flange and can be removed and replaced from the aft end.

The extendible nozzle is aligned directly with the fixed nozzle. A spring-loaded nozzle centering device, consisting of a one piece spring positioned in a groove on the lower support ring for the extension/retraction mechanism, is used. A second centering spring is positioned just upstream of the diametric seal located on the tip of the regeneratively cooled nozzle. This spring assures final centering of the nozzle and a more uniform loading on the seal.

The guide and support system concept consists of three tubular steel screw shafts located axially 120° apart around the fixed portion of the nozzle. Each shaft is mounted in sleeve bearings located in the lower tube bundle "V" band support. The other end of the threaded shaft is mounted in a bearing in the gearbox which is rigidly attached to the upper support ring.

A small, reversible 28-volt DC drive motor suitable for vacuum operation, having integral spur reduction gear sets and provisions for three separate power takeoffs, is mounted on the engine structure just below the gimbal plane. A flexible drive shaft transmits power to each screw shaft. The motor contains a mechanical lock that is automatically activated to prevent movement of the drive train components whenever the nozzle is in the extended or retracted position. The motor also contains a tool attachment to manually extend or retract the nozzle if necessary.

### 4. $LH_2$ Boost Pump

The $LH_2$ boost pump is illustrated in Figure 4. It is driven by a partial admission hydraulic tip turbine. The turbine is driven in turn by hydrogen flow obtained from the first stage of the main hydrogen turbopump. The hydrogen leaving the tip turbine is mixed with the boost pump

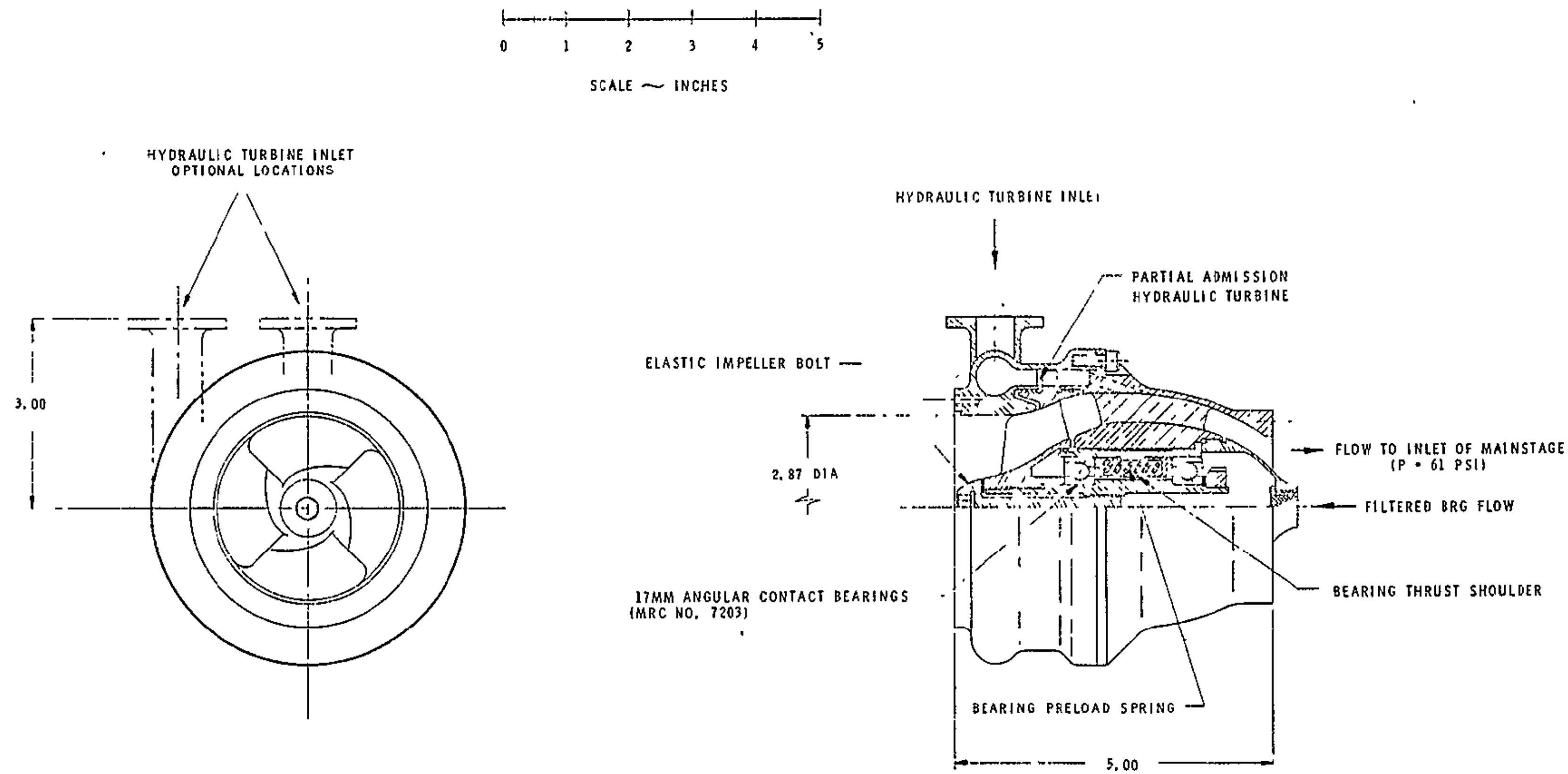



Figure 4. $LH_2$ Boost Pump

through-flow and then reenters the main hydrogen pump. The boost pump discharge flow is axial to facilitate close coupling of the boost and main turbopump assemblies in the engine package.

The hydraulic turbine is hub-mounted and yields an efficiency of 66% with a head change of 7136 feet and with 129 GPM of flow.

The boost pump operates at an NPSH of 15 ft and delivers a discharge pressure of 50 psia. It is 78% efficient.

5. $LO_2$ Boost Pump

The low-speed $LO_2$ boost pump is illustrated in Figure 5. Like the $LH_2$ boost pump, it is also driven by a partial admission hydraulic tip turbine. The turbine is driven by $LO_2$ flow from the main pump discharge. The $LO_2$ boost pump exit flow is axial to facilitate close coupling of the boost and main turbopump assemblies in the engine package.

The NPSH of the boost pump is 2 ft, and the discharge pressure is 57 psia. Boost pump efficiency is 66%.

The hydraulic turbine has an efficiency of 52% and is driven by 16.7 GPM of flow.

6. $LH_2$ TPA

The main high-speed hydrogen turbopump assembly is shown on Figure 6. The pump is a three-stage machine that is driven by a two-stage turbine. The pump discharge pressure is 2531 psia and it operates at a speed of 90,000 RPM. The turbine inlet temperature is 535°R, which is a benign operating environment considered desirable for this man-rated application.

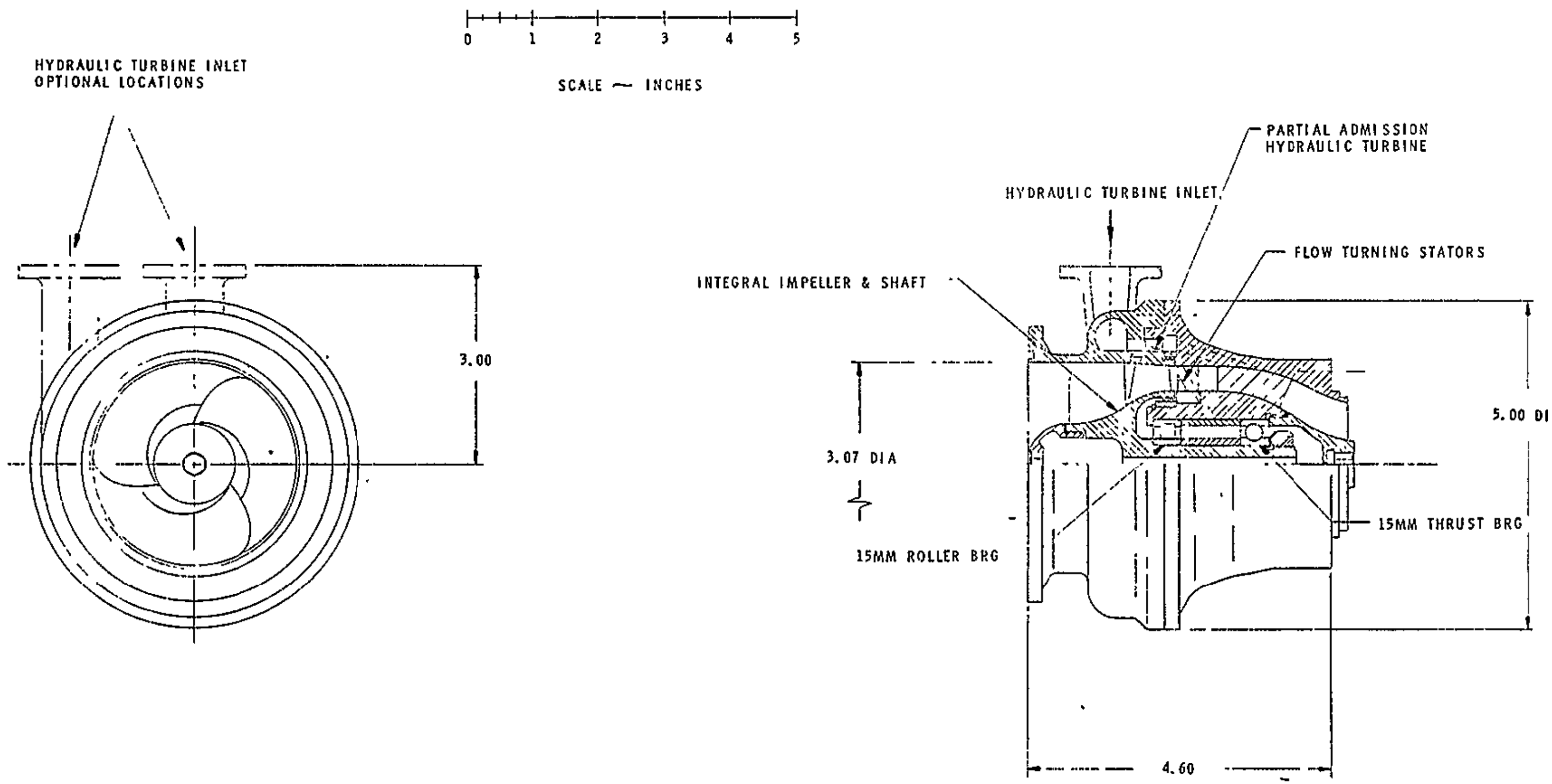



Figure 5. $LO_2$ Boost Pump

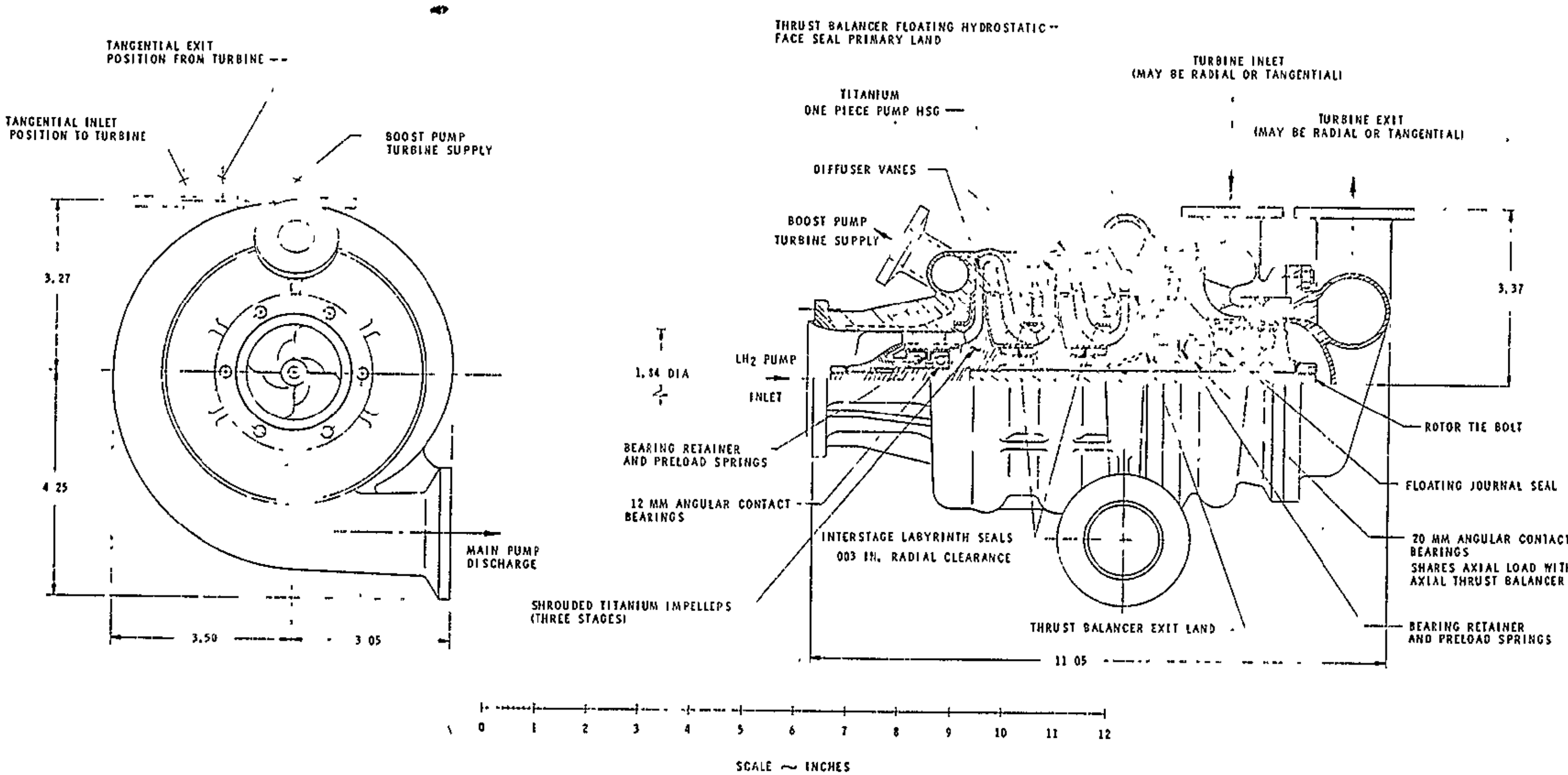



Figure 6. $LH_2$ TPA

The pump has a one-piece titanium housing which is flanged to a Nitonic-50 turbine housing. The pump impeller and inducer are made of titanium. The pump diffuser vanes and crossover passages are encased in removable disks. The rotating assembly is a built-up construction with an elastic tie bolt to maintain structural integrity. The rotating assembly is supported on a set of angular contact bearings between the inducer and first impeller and between the third impeller and the first turbine disk. The 10 mm pump end-bearing set supports radial loads but permits axial motion. The 20 mm turbine bearing set provides radial support and axial restraint. A hydraulic thrust balancer is located on the backside of the third impeller. Pressure from the third stage impeller operates the balancer, and the exit flow returns to the second impeller inlet. To prevent turbine gas from entering the bearing cavity, a high-pressure turbine seal controls leakage to the turbine, and an adjacent seal controls flow to the bearings.

The pump impellers are shrouded with straight labyrinths at the front shroud impeller inlet and on the shaft between each impeller. The front of all three impellers is identical. The back of the third stage differs from the back of the first and second stage by the axial thrust balancer.

7. $LO_2$ TPA

The main high-speed oxygen turbopump assembly is shown on Figure 7. The pump is a single stage machine that is driven by a single-stage turbine.

The main pump is made up of two pumping elements: an inducer and an impeller, directly connected together. The pump, which is suction specific speed limited, has a discharge pressure of 1487 psia and runs at a speed

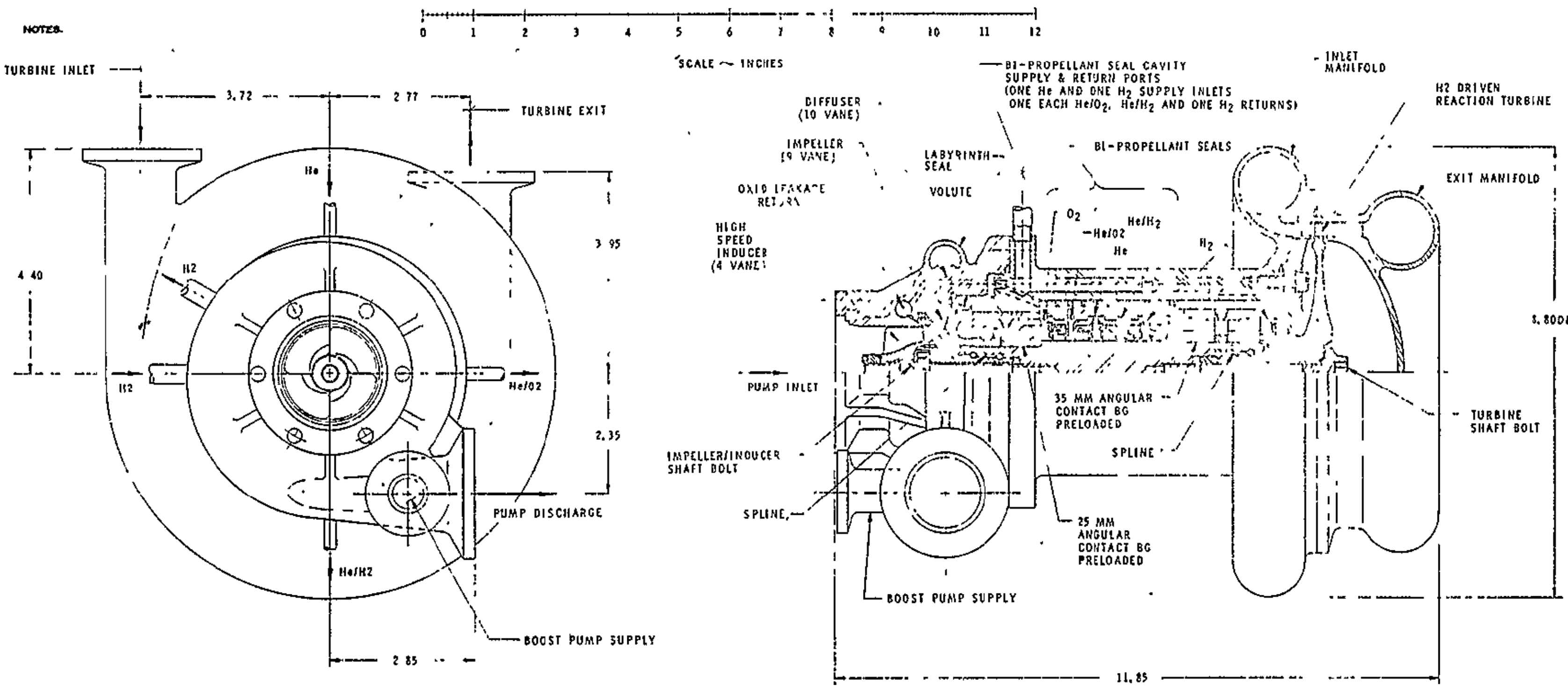



Figure 7. $LO_2$ TPA

of 34,720 RPM. The inducer, which can run at a relatively high speed without cavitating, provides sufficient head to the impeller to keep it from cavitating. The head split is approximately 15% to 85%.

The seal package between the oxygen in the pump and the heated hydrogen in the turbine is a conventional design using a helium purge. This seal package consists of five circumferential type shaft seals. Helium is introduced between the second and third seal (counting from the pump end bearings) at a pressure sufficiently high so that helium leaks under both seals and mixes with oxygen on the pump side seal and hydrogen on the turbine side seal.

The selected turbopump shaft bearings are a pair of back-to-back sets of deep-groove, angular contact bearings which are located at each end. These bearing sets are preloaded axially with springs so that there is no radial looseness. The turbine end pair are 35mm and are locked in the housing so that one will carry axial thrust in one direction and the other in the opposite direction. The pump end pair (25mm) are allowed to move axially in the housing and carry only radial load.

The turbine end bearing(s) are lubricated with liquid hydrogen. The liquid hydrogen is tapped off the second stage of the main hydrogen pump, flowed through the bearings, and returned to the inlet of the main hydrogen pump second stage.

The pump end bearings take their flow from the impeller backside labyrinth leakage flow through the bearings. This flow is returned to a manifold which feeds the flow between the pump inducer and pump impeller.

8. Shutoff Valve

Series-redundant main propellant valves are recommended to assure that the engine will shut down. The design shown on Figure 8 is very similar to that used on the recently developed OMS engine. This valve design incorporates series-redundant balls and actuators. The valve is pneumatically actuated to the open position and spring closed. The pilot valve, connectors, and actuators can be rotated to accommodate the engine packaging and to reduce the engine envelope.

The pneumatic actuation system consists of gaseous nitrogen pressurization to provide the opening force and of a spring for the closing force.

9. Modulating Valve

Modulating valves are required in the line bypassing both turbines for thrust control and in the line bypassing one of the turbines for mixture ratio control. The valve design for both of these applications is shown on Figure 9. The poppet valve design uses redundant electric motor actuation which was incorporated in the design to provide a "fail-closed" capability.

## D. TASK IV - ENGINE TRANSIENT SIMULATION COMPUTER MODEL

The primary objective of this task was to formulate a computer program to simulate the transient behavior of the cryogenic $O_2/H_2$ expander cycle engine.

Existing models for this application were evaluated to determine the feasibility of tailoring an existing model to simulate the OTV Advanced

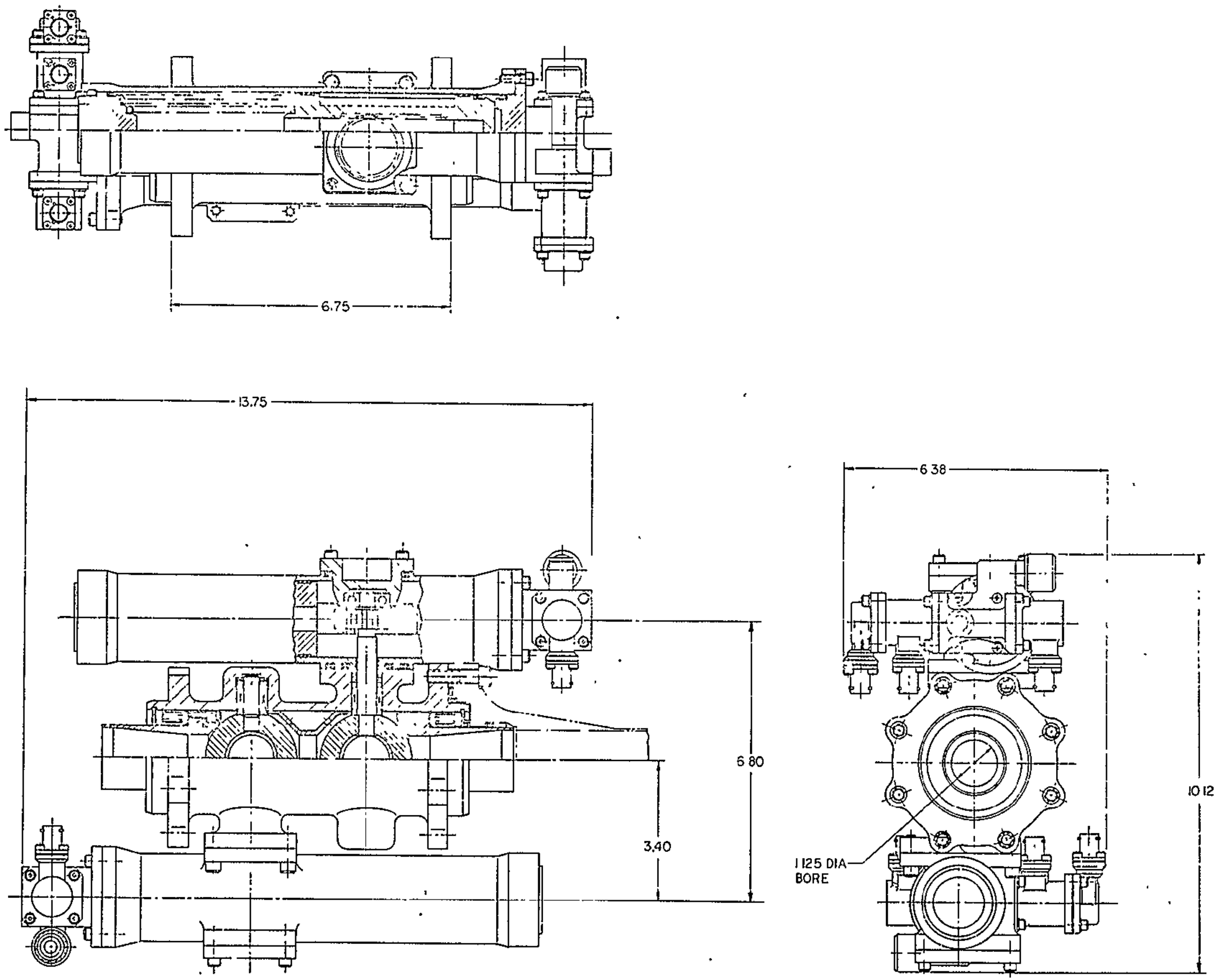



Figure 8. Shutoff Valve

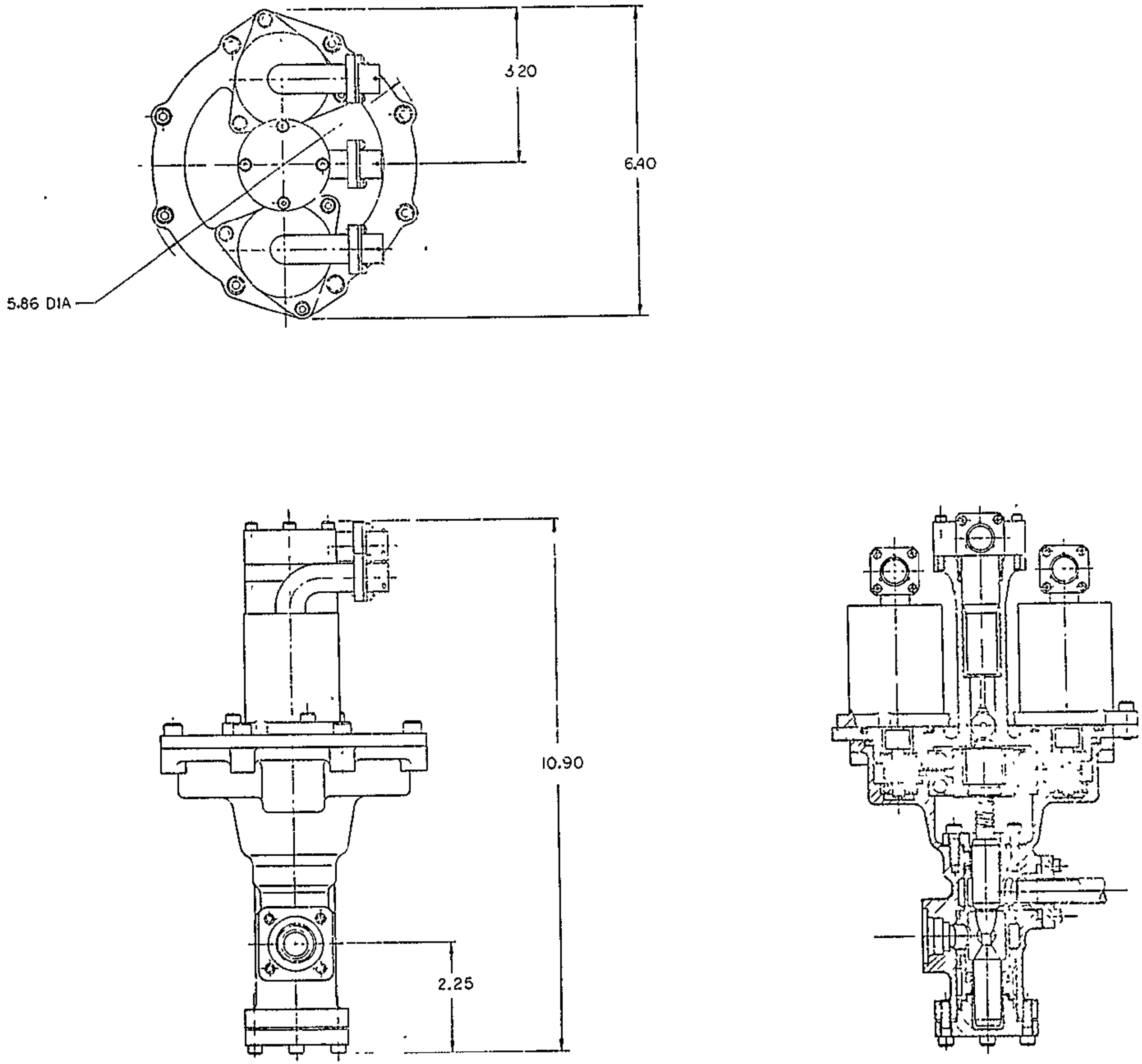



Figure 9. Modulating Valve

Expander Cycle Engine. All existing models were found to have serious shortcomings. Based upon this evaluation, a new model was developed. A description of this model and a user's manual are provided in a separate Task IV report for this contract. The input and output examples are too lengthy to be included herein and, hence, the program is described briefly in the following paragraphs.

The program has been designated as version number three of the Liquid Engine Transient Simulation program (LETS-3). The steady-state condition is also a special case of this transient model. The computer program is flexible but rather complex. It is intended for use in detailed design and development efforts.

The program solves the transient and steady-state equations describing the combustion, fluid flow, and heat transfer associated with cryogenic $O_2/H_2$ rocket engine systems, including the chilldown phase. This program can be used not only to simulate the effect of engine system component location and characteristics but also to define engine system start and steady-state requirements (i.e., valve location and sequencing for safe operation, steady- state operating point, power balance, and system pressure schedule).

The program has been designed with a maximum of flexibility to facilitate modeling of a variety of engine systems, including both pump and pressure-fed ones. The entire engine system, including tankage, can be modeled. The computer program is structured so that engine system descriptions and changes are made through input. No program changes are required to achieve modeling of a variety of engine systems. Engine system simulation flexibility is accomplished by linking together a library of component subroutines which include lines, pumps, valves, turbines, combustors, etc. Component linkage is accomplished entirely through input. The component subroutines contain equations and logic for simulating their transient and steady-state behavior. All program subroutines are written in FORTRAN IV language.

Engine operation requires that start and shutdown transients occur in a safe and predictable manner. Operation in modes or regimes which could cause damage must be avoided. Typical rocket engine start problems are delayed ignition, possibly resulting in extremely high-pressure spikes; manifold contamination caused by the backflow of a propellant; low frequency chugging producing high thermal and mechanical loads; excessive pressure overshoot; unstable fuel flow in the regeneratively cooled thrust chamber circuit; flow variation due to injector thermal effects; pump cavitation; and pump stall. Typical shutdown problems are high thermal loads resulting from off-mixture ratio operation, chugging, and manifold contamination. These transient problems are solved by the selection of appropriate valve locations, valve sequences, control methods, provision of an effective ignition system, and the use of adequate purges. This computer program will enable these selections to be made during the design phase of the engine development.

The LETS-3 computer program has been developed to operate on the Univac 1108 computer. Core requirements for the program are as follows:

| I Bank | D Bank | Total |
|---|---|---|
| 18,200 | 42,000 | 60,200 |

Both printed and plotted outputs are available.

## E. TASK V - ENGINE CONTROL

The objectives of this task were to (1) determine effective control points and methods to achieve thrust and mixture ratio control;

(2) determine suitable actuation systems; and (3) define controller requirements.

Engine operation is described in Section IV,G of this report, and the location of the valves is shown on the cycle schematic (Figure 11) in that section. The turbine flow control valve should be relocated to bypass the oxygen pump turbine. This study result was obtained too late to be incorporated into the design.

Two basic valve configurations were selected for the main propellant valves and were presented as Figure 8 and 9 in Section IV,C. These valves are (1) a pneumatically operated on-off ball valve for propellant flow control and (2) an electric motor-driven modulating poppet valve for turbine speed control. Both valve configurations are based upon Aerojet Liquid Rocket Company designs which have performed successfully on the Orbital Maneuvering System (OMS), used on the Space Shuttle, the Space Propulsion Subsystem (SPS), used on Apollo, and the Titan family engines.

Two versions of the ball valve are used in the engine. Single valves are used for the fuel and oxidizer start bypass valves, and series-redundant versions are used for the main fuel and oxidizer propellant shutoff valves.

The modulating valve is used for turbine bypass and turbine flow control functions.

The primary functions of the fuel shutoff valve are to terminate fuel flow at engine shutdown and to prevent flow through the turbines during the tank head idle mode. To provide these functions, the valve is a normally closed on-off valve. To provide high shutoff and leakage reliability, the valve is series-redundant and is fail-safe to the closed position in the event of electrical power loss. This valve is located downstream of the turbine bypass and $GH_2$ start bypass lines to provide the shutoff function during tank head idle.

The main purpose of the oxidizer shutoff valve is to terminate oxidizer flow at engine shutdown. The valve is located as close to the injector as possible to minimize the residual oxidizer in the system downstream of the valve at engine shutdown. This valve is a normally closed on-off valve and will be fail-safe to the closed position in the event of electrical power loss. To provide high reliability in the shutoff mode, the valve is series-redundant.

Two valves are required in the turbine bypass circuit. The $GH_2$ start bypass valve is an on-off valve that bypasses all the hydrogen flow during tank head idle mode operation. The turbine bypass valve is used to control the amount of flow bypassing the turbine during steady-state operation and to control engine thrust. Nominally, 6% of the hydrogen flow bypasses the pump turbines.

The main function of the $GO_2$ start bypass valve is to control the flow of gaseous oxygen from the heat exchanger to the injector during tank head idle. The valve is also required to remain closed at engine shutdown to prevent bypassing oxidizer flow around the $LO_2$ shutoff valve.

The turbine flow control valve is used to provide the engine mixture ratio control. During the course of this study, analysis has shown that the turbine flow control valve should be placed in a line bypassing the oxygen pump turbine rather than the fuel pump turbine (see Figure 11 in Section IV,G). Schedule and funding limitations did not permit another design iteration to incorporate this feature. Approximately twice as much oxygen turbine bypass flow is required to obtain the same mixture ratio variation as for fuel pump turbine bypass flow. However, the effect upon the engine cycle power balance is only about one-half as much.

An actuator trade study was conducted to assess reliability, complexity, weight, safety, control precision, and state-of-the-art parameters. Pneumatic, hydraulic, and electric actuation systems (and various combinations thereof) were considered. Based upon this trade study, the pneumatically actuated shutoff valves and electric motor modulating valve actuation were selected.

The pneumatic actuation system selected for the shutoff valves consists of gaseous nitrogen pressurization to provide the opening force and of a spring for the closing force. This system was selected for the following reasons:

- The high-pressure system minimizes the actuator size.
- The valve will fail to the closed position.
- The system is state-of-the-art and is used on the OMS engine.
- No materials compatibility problems are presented.
- No additional propellant leak paths are provided.

Electric motor actuation was selected for the modulating valves because the system is a straightforward, simple, state-of-the-art design for a system requiring continuous actuation capability. This system is used on both the thrust (turbine bypass) and mixture ratio (turbine flow control) control valves.

The modulating valves are part of a closed-loop control system which includes the electronic controller. Valve position is set by the controller and the output verified by the various engine parameters (flows, turbine speeds, chamber pressure, valve positions, etc.) monitored by the controller. By trimming the turbine speeds relative to each other and to the total flow requirements, the engine maintains the required mixture ratio and thrust level.

Engine start and shutdown sequences and a control logic were defined to establish the engine controller requirements. These are presented in Volume II of this contract. The controller power requirement was estimated to be watts. This estimate was made by comparing the SSME and OTV engine control requirements and considering the advancements in the electronics state of the art.

## F. TASK VI - ENGINE CONFIGURATION LAYOUT

The primary objective of this task was to provide an engine configuration layout drawing showing the packaging relationship of the primary engine components.

The engine assembly layout drawing is presented on Figure 10 which shows the engine top and side views. The engine is 60 in. long with the extendible nozzle in the stowed position. This length is measured from the top of the gimbal block to the end of the tube bundle nozzle. The engine is 109.6 in. long and has an area ratio of 435:1 with the extendible nozzle deployed. Approximately 10.4 in. of potentially available deployed length is lost in the area of the extendible nozzle deployment mechanism and attachment plane. Further design refinements could increase the deployed length to a maximum of 120 in., with a resulting area ratio of 473:1 and a performance increase of 1.8 seconds over the data shown in Table IX in Section IV,G in this report.

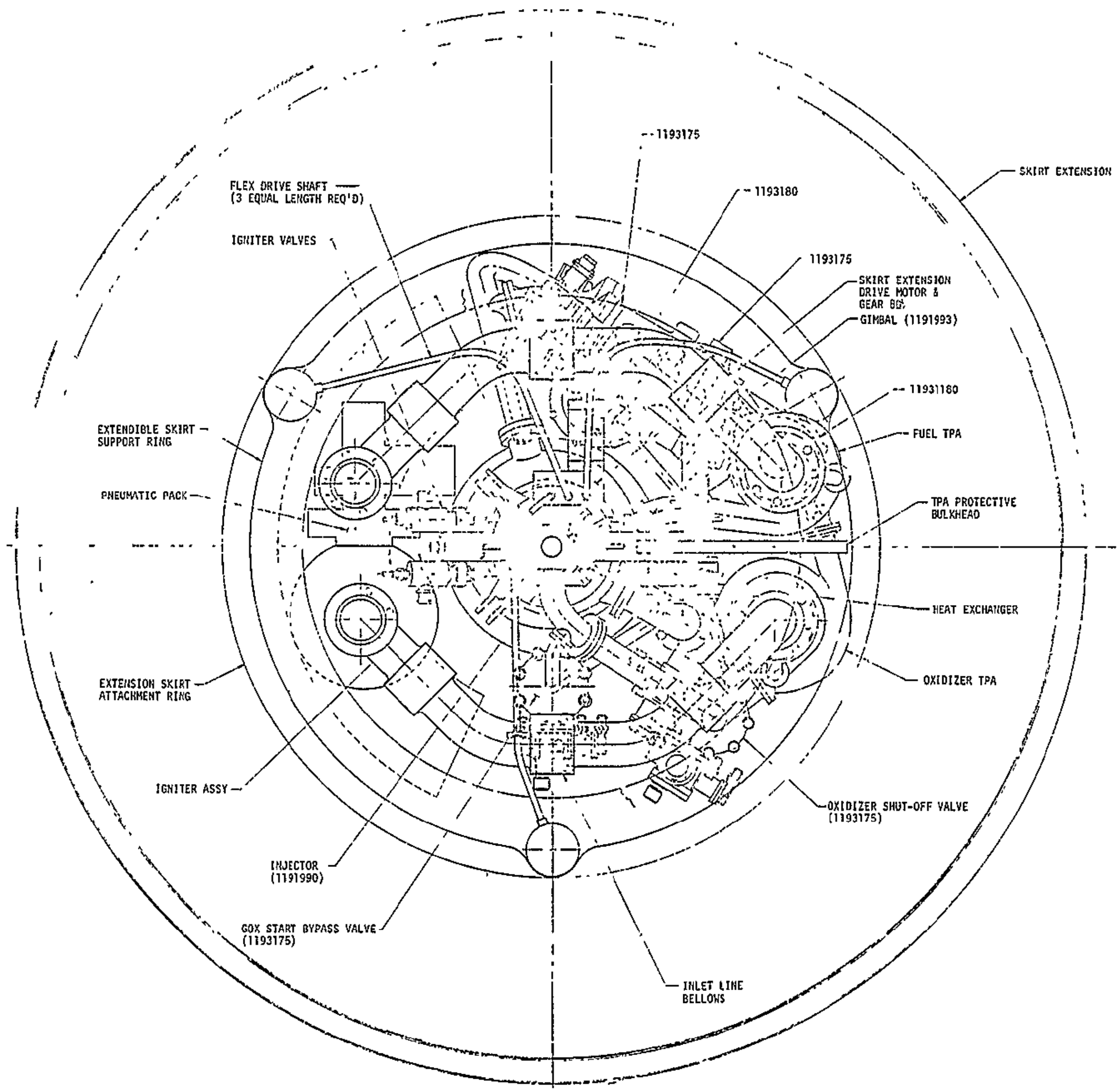



Figure 10. Engine Layout (Sheet 1 of 2)

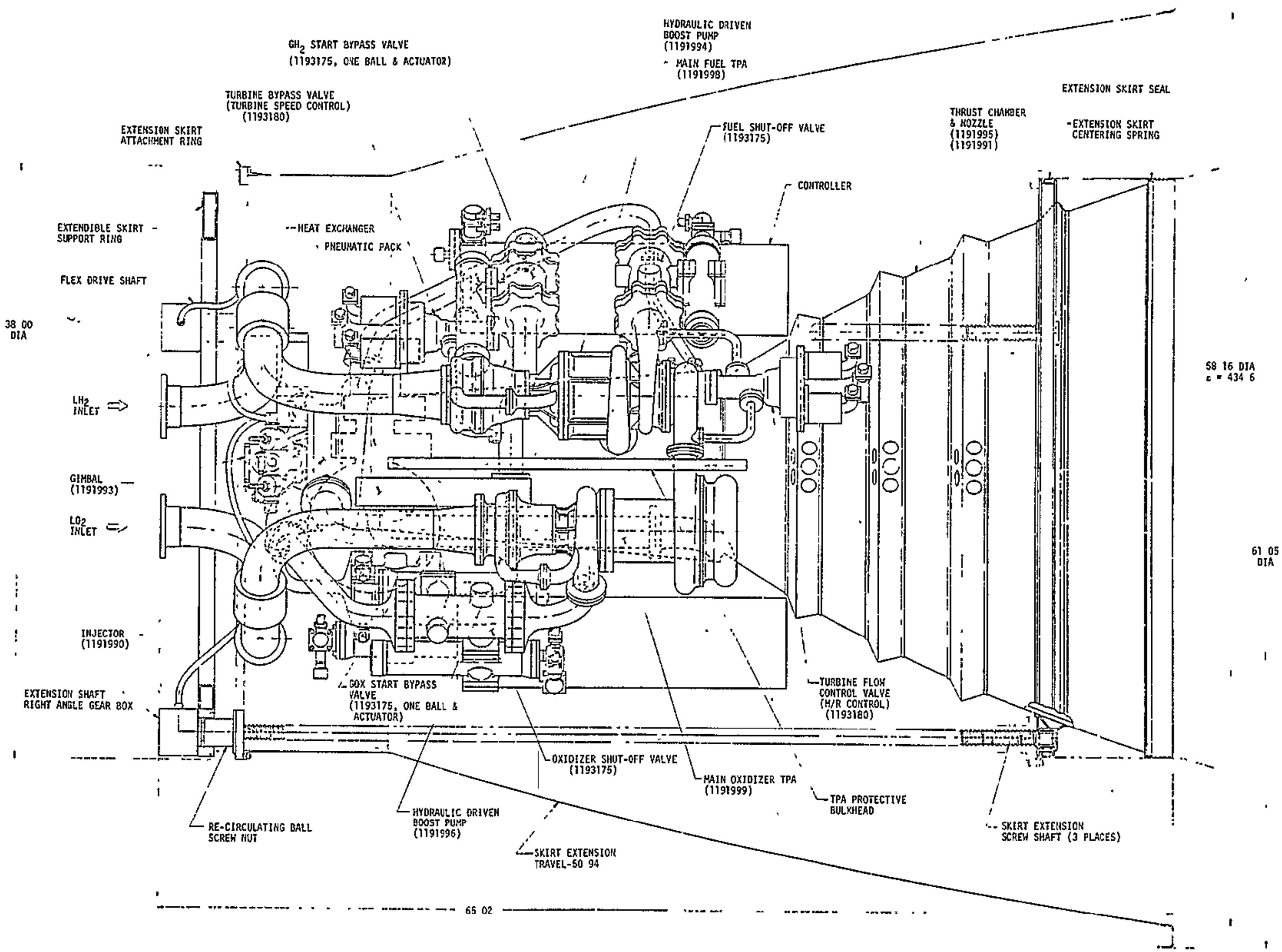



Figure 10. Engine Layout (Sheet 2 of 2)

G. TASK VII - ENGINE DATA SUMMARY

The primary objective of this task was to prepare a document which summarizes the engine performance, weight, envelope, and service life data and presents the engine and component layout drawings. This document was submitted as a Task VII, Engine Data Summary, report for this contract.

Engine and component layout drawings are presented in other sections of this report. The remaining data is summarized herein.

Based upon the results of design analyses, engine sensitivities, cycle optimization, and thrust chamber geometry optimization conducted in conjunction with both this Point Design Study and the OTV Phase A Engine Study, an engine with the characteristics summarized in Table IX was selected as a representative 1980 technology baseline. The data is presented for both nominal mixture ratio (6.0) and off-design mixture ratio (7.0) operation. The engine length with the extendible nozzle in the stowed position is 60 in. With the extendible nozzle deployed, the engine length is 109.6 in. The chamber pressure of 1200 psia was selected on the basis of cycle optimization and trade-off studies which evaluated specific impulse and weight changes with chamber pressure. Further optimization and tradeoffs are planned in future work, and some changes in operating chamber pressure and performance are anticipated.

The $O_2/H_2$ expander cycle engine uses a series turbine drive cycle which is shown on Figure 11. The engine uses hydraulically driven boost pumps, with the flow tapped off the main pump stages. Fuel flows from the pump discharge to the thrust chamber where 85% of the hydrogen flow is used to cool the slotted copper chamber in a single pass from an area ratio of 10.6:1 to the injector head end. Fifteen (15) percent of the hydrogen is used to

TABLE IX

ADVANCED EXPANDER CYCLE ENGINE DATA SUMMARY

Rated Vacuum Thrust = 15,000 lb
Stowed Length = 60 in.

| | Engine Mixture Ratio | |
|---|---|---|
| | 6.0 | 7.0 |
| Vacuum Thrust, lb | 15,000 | 15,000 |
| Vacuum Specific Impulse, sec. | 475.4 | 471.0 |
| Total Flowrate, lb/sec | 31.56 | 31.85 |
| Mixture Ratio | 6.0 | 7.0 |
| Oxygen Flowrate, lb/sec | 27.05 | 27.87 |
| Hydrogen Flowrate, lb/sec | 4.51 | 3.98 |
| Chamber Pressure, psia | 1,200 | 1,162 |
| Nozzle Area Ratio | 435 | |
| Nozzle Exit Diameter, in. | 58.2 | |
| Engine Length, in. | | |
| Extendible Nozzle Stowed | 60.0 | |
| Extendible Nozzle Deployed | 109.6 | |
| Engine Dry Weight, lb | 574 (1) | |

(1) Calculated from preliminary component layout drawings where available. Further design definition required.

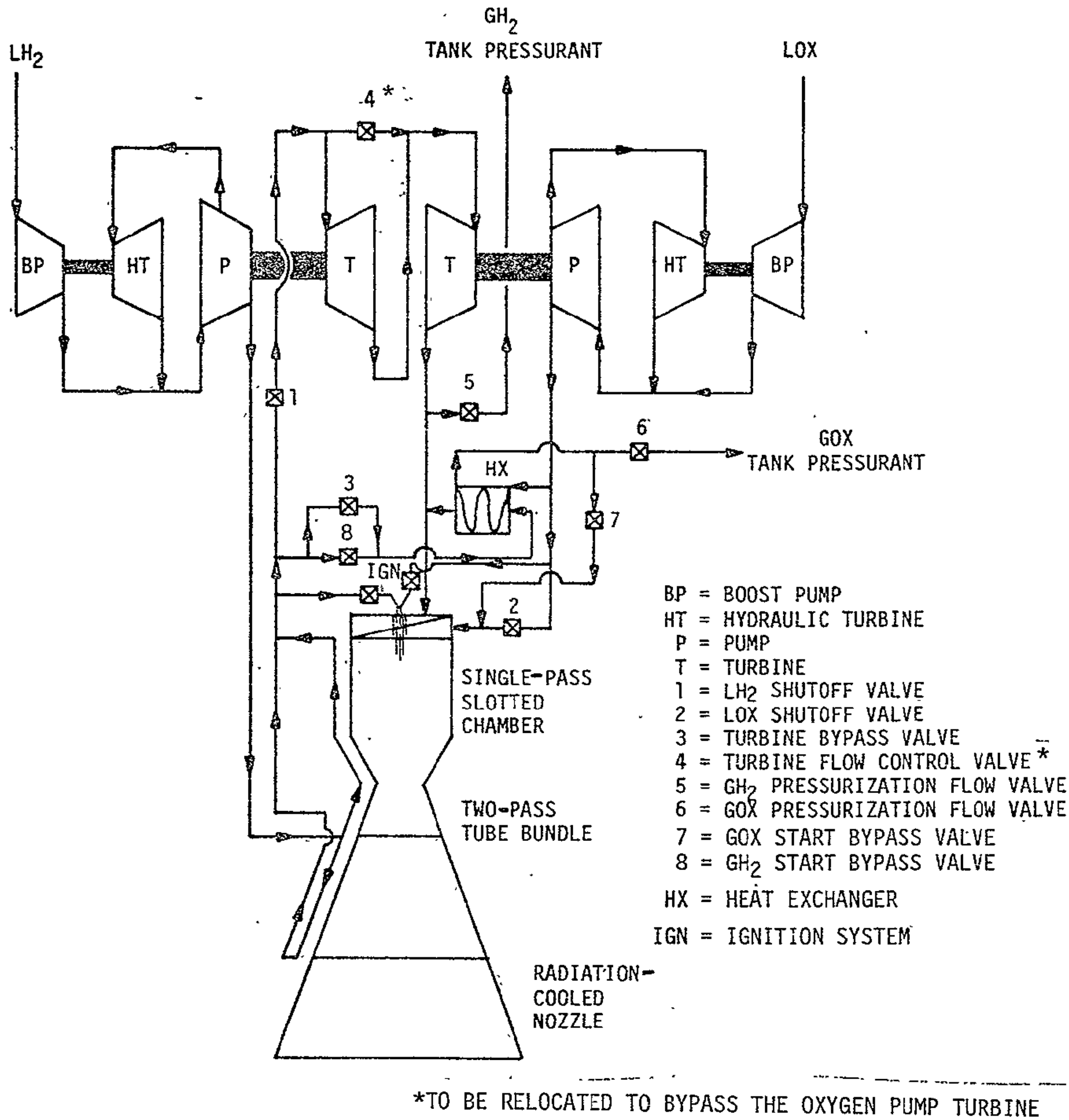



Figure 11. Baseline Advanced Expander Cycle Engine Flow Schematic

cool the tube bundle nozzle in two passes from an area ratio of 10.6:1 to the end of the fixed nozzle ($\varepsilon$ = 172:1) and return. The coolant flows are merged, and 6% of the total engine hydrogen flow is used to bypass both turbines to provide cycle power balance margin and thrust control. The remaining hydrogen flow first drives the fuel pump turbine and then drives the oxidizer pump turbine. After driving the oxidizer pump turbine, a small amount of heated hydrogen is tapped off for hydrogen tank pressurization. The remaining hydrogen flow is then injected into the combustion chamber.

At rated thrust operation, oxidizer flows from the main pump discharge directly to the thrust chamber and is injected in a liquid state. A small amount of oxidizer is tapped off and heated by the hydrogen turbine bypass flowrate in a heat exchanger to provide LOX tank pressurization.

The engine is also capable of operating in a tank head idle mode and is adaptable to extended low-thrust operation at a thrust level of 1.5K lb.

The purpose of the tank head idle mode is to thermally condition the engine without non-propulsive dumping of propellants. This is a pressure-fed mode of operation at a thrust level of approximately 50 lbs and a vacuum specific impulse estimated at 400 sec. During this mode of operation, the main fuel and oxygen valves (numbers 1 and 2 on the schematic) are closed. All of the fuel bypasses the turbines through valve number 8 so that the pumps are not rotating. The heat exchanger in the turbine bypass line gasifies the oxygen which then flows through valve number 7 to the chamber. Tank pressurization is not supplied during this operating mode, and valves 5 and 6 are closed. The pressurization valves are opened as the engine is brought up to steady-state, full-thrust operation.

The OTV point design engine is adaptable to operation at 10% of rated thrust (i.e., 1.5K lbF) with minor modifications. This low-thrust operating point is a dedicated condition, and the engine is not required to operate at both the 15K and 1.5K thrust levels on the same mission. To operate at low thrust, the oxidizer injection elements must be changed to one of smaller size and an orifice must be installed in the line downstream of the chamber coolant jacket.

Engine system delivered specific impulse for design and off-design operation is presented on Table X.

The engine has been designed for 1200 thermal cycles and 10 hours accumulated run time. Therefore, all of the component designs, illustrated in Section IV,C, are based on the minimum service life requirement (300 cycles or 10 hours) with a safety factor of 4 applied to lower-bound data. This service life is not predicted to be reduced when the engine is operated at mixture ratios between 6.0 and 7.0. Similarly, low-thrust operation (i.e., 1500 lbF) at mixture ratios between 6.0 and 7.0 is not predicted to reduce this service life.

The advanced expander cycle engine weight breakdown by component is shown on Table XI. Weights have been either calculated from the preliminary component layout drawings or estimated. Estimations are based upon scaling the weights of similar actual components or the weight data from other engine studies. Further design definition is required to establish a good baseline target weight.

TABLE X

ADVANCED EXPANDER CYCLE ENGINE
PERFORMANCE AT DESIGN AND
OFF-DESIGN O/F
RATED AND LOW-THRUST OPERATION

| THRUST, LB | ENGINE MIXTURE RATIO | THRUST CHAMBER PRESSURE, PSIA | ENGINE DELIVERED VACUUM SPECIFIC IMPULSE, SEC. | FLOWRATES, LB/SEC | |
|---|---|---|---|---|---|
| | | | | Fuel | OX |
| 15000 | 6.0 | 1200 | 475.4 | 4.51 | 27.05 |
| 15000 | 6.5 | 1180 | 474.9 | 4.21 | 27.37 |
| 15000 | 7.0 | 1162 | 471.0 | 3.98 | 27.87 |
| 1500 | 6.0 | 125 | 459.7 | .466 | 2.80 |
| 1500 | 7.0 | 121 | 451.7 | .415 | 2.91 |

Note: Injector elements are modified for the low-thrust condition.

TABLE XI

ADVANCED EXPANDER CYCLE ENGINE
WEIGHT DATA

| | COMPONENT | WEIGHT, LB | |
|---|---|---|---|
| 1. | Gimbal | 3.3 | Calculated |
| 2. | Injector | 30.6 | Calculated |
| 3. | Chamber | 47.3 | Calculated |
| 4. | Copper Nozzle | 27.0 | Calculated |
| 5. | Tube Bundle Nozzle | 38.4 | Calculated |
| 6. | Radiation Nozzle | 80.0 | Calculated |
| 7. | Nozzle Deployment System | 72.0 | Calculated |
| 8. | Valves and Actuators | 72.7 | Calculated |
| 9. | LOX Boost Pump | 5.6 | Calculated |
| 10. | $LH_2$ Boost Pump | 8.5 | Calculated |
| 11. | LOX TPA (HI SPD) | 26.9 | Calculated |
| 12. | $LH_2$ TPA (HI SPEED) | 26.3 | Calculated |
| 13. | Misc. Valves & Pneumatic Pack | 12.6 | Estimated |
| 14. | Lines | 37.0 | Estimated |
| 15. | Ignition System | 9.2 | Calculated |
| 16. | Engine Controller | 35.0 | Estimated |
| 17. | Miscellaneous | 37.0(1) | Estimated |
| 18. | Heat Exchanger | 5.0 | Estimated |
| | Total Engine Weight | 574.4 | |

(1) Miscellaneous includes: Electrical harness, 12.5 lb; service lines, 6.5 lb; TPA protective bulkhead, 0.4 lb; attachment hardware, 15.0 lb; and instrumentation, 2.6 lb.

## H. TASK VIII - TECHNOLOGY REQUIREMENTS

The objective of this task was to identify any new technology required to perform the detailed design, construction, and testing of the advanced expander cycle engine.

A list of critical technology requirements for this engine was prepared as a result of this study, the Phase A engine study, and ALRC in-house efforts. We first recommended this point design study and a thrust chamber technology program in October 1978. Both of these recommendations have been pursued by NASA. We also submitted an Advanced Expander Cycle Engine Critical Component Technology and Experimental Engine Plan to NASA/MSFC in February 1980.

The three design drivers which the technology activities should address are as follows:

- Engine Turbopump Drive Power (P)
- Development and Operational Risk Reduction (R)
- Engine Performance, Impulse (I)

Power technology activities (P) are aimed at assuring or increasing the power available to the turbopump. The results of these programs are used to verify that the engine will operate at the selected design point chamber pressure, has the capability of operating at a higher pressure and performance level, or can accept greater component performance margins or tolerances. As a result, overall program economies are achieved by guaranteeing that the engine operating design point can be reached or exceeded (growth). This saves development dollars because large variations in costs result from parallel resolution of small instant problems.

Risk reduction technologies (R) allow the solution of design deficiencies at the technology level prior to committing to a design specification and entering the engine development program. Making decisions at this point provides for design iterations and decisions to be made at the low expenditure level of the overall program. Risk reduction solutions provide for higher confidence in the engine operation, which is consistent with man-rating design philosophy.

Performance technology programs (I) are those which are geared to guarantee the performance level of the engine prior to a commitment to the specification. These programs provide a high confidence in the performance position so that payloads can be firmed up at reasonable levels prior to DDT&E. The level of performance growth is also determined by these programs.

Twenty-four technology programs were identified and are summarized on Figure 12. These programs support the following key decision points and/or engine design and development logic which are also displayed on the figure.

(1) Engine Power Balance
(2) Throttling Power Balance
(3) Cycle Optimization
(4) Performance Optimization
(5) Nozzle Extension Decision
(6) Experimental Engine Design Decision
(7) Experimental Engine Fabrication and Test

Table XII also lists the twenty-four suggested programs along with a priority designation. The programs designated as priority "A" are those which have a major impact on the expander cycle engine design. As a minimum, the critical technology activity should address these items. We recommended

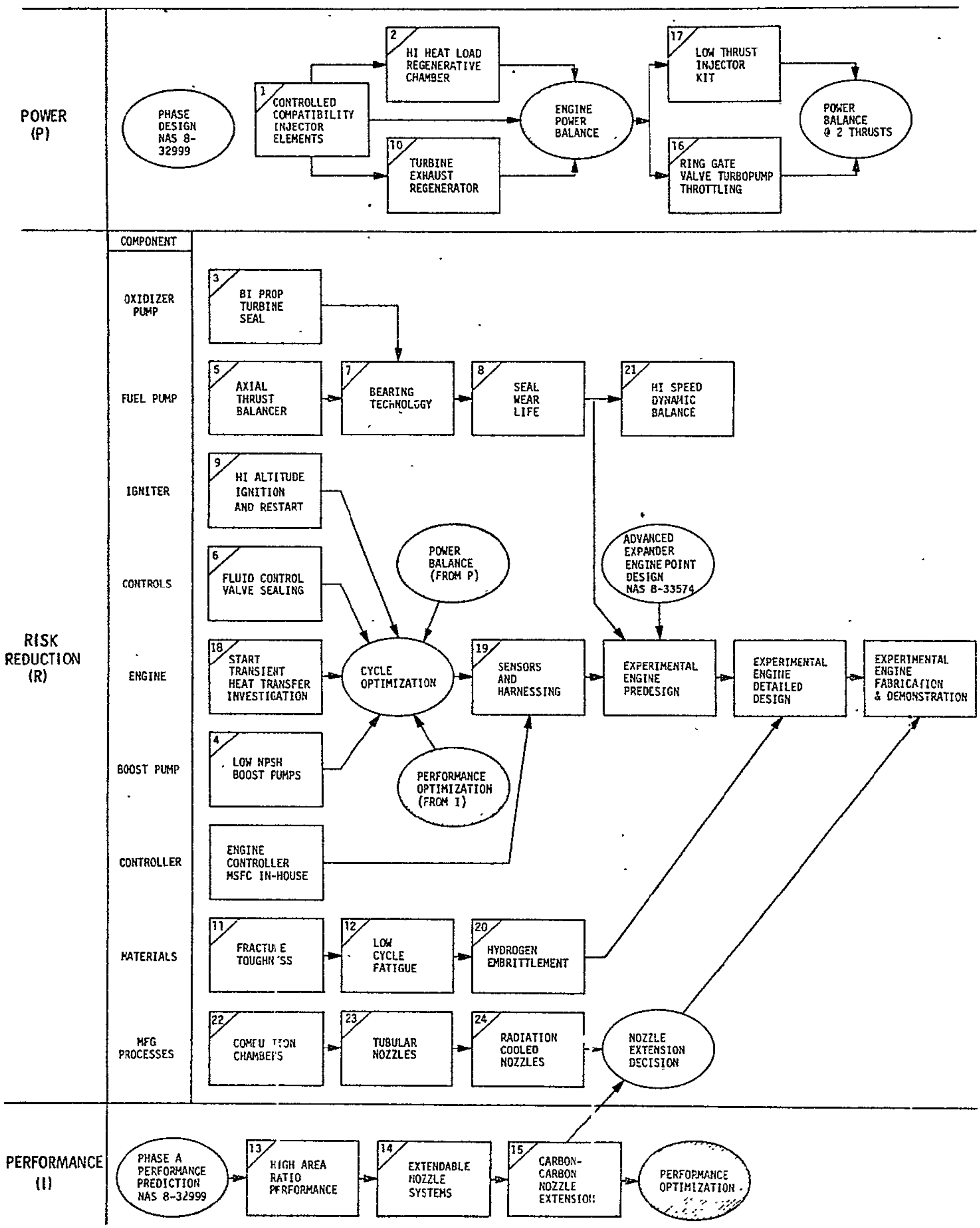



Figure 12. Critical Component Technology Program Logic

TABLE XII

COMPONENT TECHNOLOGY PROGRAM PRIORITIES

| | PROGRAM | CATEGORY | PRIORITY |
|---|---|---|---|
| 1. | Controlled Compatibility Injector Elements | P | A |
| 2. | High Heat Load Regenerative Chamber | P | A |
| 3. | Oxidizer Turbopump Bipropellant Seal | R | A |
| 4. | Low NPSH Boost Pumps | R | A |
| 5. | Axial Thrust Balancer | R | A |
| 6. | Fluid Control Valve Sealing | R | A |
| 7. | Turbopump Bearing Technology | R | A |
| 8. | Turbopump Seal Technology | R | A |
| 9. | Igniter Development | R | B |
| 10. | Regenerator Development | P | B |
| 11. | Fracture Toughness Testing of Structural Alloys in Gaseous Hydrogen | R | B |
| 12. | Predictive Analysis of Low Cycle Fatigue Life for OTV Alloys of Construction | R | B |
| 13. | High Area Ratio Nozzle Performance | I | B |
| 14. | Extendible Nozzle Systems | I | B |
| 15. | Carbon-Carbon Nozzle Extensions | I | C |
| 16. | Ring Gate Valve Turbopump Throttling | P | C |
| 17. | Low-Thrust Injector Kit | P | C |
| 18. | Start Transient Heat Transfer Coefficient Investigation | R | C |
| 19. | Sensors and Harnessing | R | C |
| 20. | Hydrogen Embrittlement Study of Columbium Alloys in OTV Radiation Cooled Nozzle Environment | R | C |
| 21. | High-Speed Dynamic Balancing | R | C |
| 22. | Combustion Chamber Manufacturing Processes | R | C |
| 23. | Tubular Nozzle Manufacturing Process | R | C |
| 24. | Radiation-Cooled Nozzle Welding | R | C |

grouping programs 1, 2, and 9 into a single program. NASA/MSFC is currently evaluating proposals on an Expander Cycle Thrust Chamber Verification program which has objectives similar to those of our suggested technologies (i.e., verify experimentally the analytically determined design and operating characteristics of an expander cycle thrust chamber). Program 10 could also be an added option to complete the evaluation of the expander cycle engine power balance.

Programs designated as priority "B" are those which have a definite bearing on the cost effectiveness of the OTV engine development program.

Priority "C" programs represent either expansion of the application of the expander cycle engine or activities which directly influence the production aspects of the engine program.

The objectives and justifications for each of the suggested technologies are presented in Volume II, Study Results, of this final report.

## I. TASK IX - COMPUTER SOFTWARE DOCUMENTATION

As part of this program, two engine computer models were delivered to NASA/MSFC. One of the programs is the Task I - Steady-State Computer Model, and the second is the Task IV - Engine Transient Simulation Computer Model.

Documentation submitted on these computer models included:

- User's Manual
- FORTRAN program listing
- Program flow charts
- Sample inputs and outputs

At the request of the NASA COR, a FORTRAN card deck was submitted for the steady-state model. The transient model was submitted on tape. Both programs are compatible with a Univac 1108 system.

## V. CONCLUSIONS AND RECOMMENDATIONS

The following conclusions were derived from this and related study efforts.

- A new engine is requird to meet the OTV performance, man-rating, and life requirements.
- The advanced expander cycle engine is a high-performance, low-risk, low-cost option.
- The benign turbine operating environment of an expander cycle reduces the engine development risk and cost.
- Further design definition of the expander cycle engine and its components is required.
- The performance of the expander cycle engine can be increased by removing the 1980 state-of-the-art technology requirement.
- Engine operation at 10% of rated thrust is feasible by "kitting" the engine.
- Experimental verification of low-thrust operation is required.

Based upon the conclusions, the followng recommendations for future work are made:

- Expander cycle engine component critical technology programs should be initiated to:
  - Reduce risk
  - Verify power balance
  - Verify performance

V, Conclusions and Recommendations (cont.)

- Component technology should address both high- and low-thrust operation
- Continue point design studies to optimize the advanced expander cycle engine
- Conduct detailed design analysis of a breadboard advanced expander cycle engine
- Fabricate and test a breadboard expander cycle engine and its components