NASA Technical Memorandum 81420

(NASA-TM-81420) ANALYTICAL INVESTIGATION OF TWO HYDROGEN OXYGEN ROCKET ENGINE SYSTEMS FOR LOW-THRUST APPLICATION (NASA) 17 p HC A02/MF A01 CSCL 21H

N80-17138

Unclas
G3/20 47111

ANALYTICAL INVESTIGATION OF TWO HYDROGEN OXYGEN ROCKET ENGINE SYSTEMS FOR LOW-THRUST APPLICATION

Dean D. Scheer
Lewis Research Center
Cleveland, Ohio

Prepared for the
1980 JANNAF Propulsion Meeting
Monterey, California, March 11-13, 1980

UNCLASSIFIED

ANALYTICAL INVESTIGATION OF TWO HYDROGEN-OXYGEN ROCKET ENGINE SYSTEMS FOR LOW-THRUST APPLICATION (U)*

Dean D. Scheer
NASA Lewis Research Center
Cleveland, Ohio 44135

E-343

ABSTRACT

Two hydrogen-oxygen rocket engine system concepts were analyzed parametrically over a thrust range from 100 to 1000 pounds and a chamber pressure range from 175 to 1000 psia. Both concepts were regeneratively cooled with hydrogen and were pump-fed by electric motor driven positive displacement pumps. Electric power was provided by either a turboalternator (turboalternator concept) or some means external to the engine system (auxiliary power concept). The computer program used to conduct the analyses along with the design characteristics of the major engine system components are briefly described. The feasible design range of the systems over the parametric range of thrust is discussed in terms of allowable chamber pressure considering the constraints of thrust chamber cooling and cycle power. Engine system estimated performance, mass, and dimensional envelope parametric data within the feasible design range are presented.

INTRODUCTION

A number of mission studies have forecast a need for large space structures in geosynchronous equatorial orbit (GEO). As envisioned, these structures would be launched to low earth orbit (LEO) in a packaged condition using the Space Shuttle and subsequently transferred to GEO using a high energy space propulsion system. There are two options available for placement of these types of payloads in GEO. In the first option, the LEO-to-GEO transfer would be accomplished with the payload in the packaged condition, followed by manned or automated deployment and assembly in GEO. Either high or low thrust could be used for the transfer. In the second option, manned or automated deployment and assembly would be carried out in LEO, followed by a LEO-to-GEO transfer with the payload in the deployed condition. Here, low thrust would be required in order to maximize deliverable area for a given payload mass.

Since the early 1970's the DOD and NASA have funded a number of studies which examined chemical rocket propulsion systems suitable for the high thrust option noted above. Considerable effort has also been conducted on very low-thrust solar-electric propulsion systems which have application for missions in which extended LEO-to-GEO transfer times are acceptable. These extensive study efforts have provided the mission analyst with propulsion system performance data at the extremes of the orbit-transfer thrust range of interest. In an attempt to provide a more comprehensive propulsion system data base for use in future orbit-

*Approved for public release; distribution unlimited.

transfer studies the NASA Lewis Research Center in 1979 funded two contracts (NAS3-21940 and NAS3-21941) to generate low-thrust chemical propulsion system parametric and preliminary design data. The scope of the contracted effort includes several candidate propellant combinations and a number of pressure-fed and pump-fed engine system concepts. Prior to the contracts, in-house analyses were conducted in order to define some of the guidelines for the contract effort and to provide preliminary information on engine system performance and technology needs. Results of the in-house analyses are reported herein.

The scope of the in-house work was limited to hydrogen-oxygen propellants at a mixture ratio (O/F) of 6.0. Two candidate engine system concepts were studied. Both concepts were regeneratively cooled with hydrogen and were pump-fed by electric motor driven positive displacement pumps. The concepts differed only in the assumed method of providing power for the pump-drive motors.

DESCRIPTION OF ENGINE SYSTEMS

The engine concepts selected for study are illustrated in Fig. 1. The concept shown in Fig. 1(a) (herein called the turboalternator concept) uses the so-called expander turbine drive cycle to provide power requirements for the propellant feed system. Propellants are pumped from low-pressure storage tanks by separately driven tank-mounted hydrogen and oxygen pumps. The pumps were assumed to be positive displacement driven by electric motors. Accumulators were included in each propellant feed system to reduce the possibility of flow and pressure pulsations downstream of the positive displacement pumps. Electric power for the pump drive motors is provided by an alternator which is driven by a turbine using gaseous hydrogen from the thrust chamber regenerative cooling jacket as the working fluid. A turbine by-pass valve is used to provide turbine speed control. The hydrogen by-pass flow and turbine exhaust flow are combined downstream of the turbine discharge, injected into the combustion chamber and burned with oxygen. Combustion products are expanded in the nozzle which was assumed to be radiation cooled downstream of the regenerative cooling jacket.

In the concept shown in Fig. 1(b) (herein called the auxiliary power concept) it was assumed that a fuel cell/inverter system or some other means external to the engine system was available to provide the power necessary for the pump-drive motors. Thus the hydrogen exiting from the regenerative cooling jacket flows directly to the thrust chamber. With this exception, the propellant feed systems are the same as those described for the turboalternator concept.

METHOD OF ANALYSIS

A computer program was written to calculate design point steady-state characteristics of the engine systems shown in Fig. 1 over a thrust range from 100 to 1000 pounds and a chamber pressure range from 175 to 1000 psia. A simplified logic diagram for the computer program is shown in Fig. 2.

Required inputs to the program are thrust level, chamber pressure, theoretical one dimensional equilibrium (ODE) specific impulse, characteristic velocity, combustion temperature and concept option. With these inputs the program calculates delivered specific impulse, hydrogen and oxygen mass flow rates, thrust chamber and nozzle geometry and the heat loads in the regeneratively cooled nozzle and chamber. The program then assumes a value of inlet pressure to the regenerative cooling jacket and calculates the enthalpy of the hydrogen coolant at the chamber throat station and the chamber exit station. A throat station pressure is assumed and hydrogen properties at the throat as a function of enthalpy and assumed pressure are determined and used in calculating coolant average velocity, pressure drop, and pressure at the throat station. Iterations are performed until the assumed pressure is equal to the calculated pressure within a specified tolerance. After convergence, the same procedure is used in the analysis of the cooling process from the throat station to the chamber exit station. This routine also includes flags which terminate execution if the average coolant velocity or coolant exit temperature exceed specified limiting values.

In the turboalternator concept option the pressure drops in the lines and valves, head rise and power requirements of the pumps, and turbine pressure drop required to provide the power are then calculated. The calculated turbine discharge pressure is compared with the required injector pressure and iterations, using inlet pressure to the jacket as the independent variable, are performed until turbine discharge pressure and required injector pressure are matched within a specified tolerance. After a balance is achieved the program calculates the mass of the engine system. If a balance cannot be achieved for the given input within a specified number of iterations, program execution is terminated.

In the auxiliary power concept option, the hydrogen pressure at the chamber exit station is matched with required injector pressure, and the turbine routine is by-passed. With these exceptions, the analysis procedure is the same as for the turboalternator option.

ENGINE SYSTEM SIMULATION

In order to formulate the computer program and conduct the analysis it was necessary to select and simulate components and to assume certain operating conditions in the engine systems. In the following paragraphs the design characteristics of the major components and the assumed operating conditions are discussed. It should be noted that in some cases the characteristics and assumptions significantly affect the absolute values of results and that there was no attempt made in the study effort to optimize the concepts by considering alternate types of pumps, turbines, etc.

ENGINE SYSTEM PERFORMANCE

A mixture ratio (O/F) of 6.0 and a nozzle expansion area ratio of 400:1 were selected as being representative of requirements for a future low-thrust hydrogen-oxygen propulsion system. Theoretical one dimensional equilibrium (ODE) specific impulse over the parametric study range was determined using the computer program described in Ref. 1. Propellant enthalpies at the engine system inlet were assumed to be those corresponding to liquid oxygen and liquid hydrogen at their normal boiling points. Losses considered in determining delivered performance were those associated with energy release, kinetics, divergence, and boundary layer. Energy release efficiency was assumed constant at 98 percent over the parametric study range and divergence efficiency of the 400:1 nozzle was assumed constant at 99.55 percent. Kinetic losses which vary with thrust and chamber pressure were based on results using the computer program described in Ref. 2. Boundary layer losses which also vary with thrust and chamber pressure were determined using the scaling equation given in Ref. 3. The boundary layer loss for a 400:1, 90 percent bell nozzle given in Ref. 4 was used as the base point value in the scaling equation.

THRUST CHAMBER

The regenerative portion of the thrust chamber and 90 percent bell nozzle was assumed to be copper with milled channels and a nickel closeout. Heat transfer, coolant velocity and coolant pressure drop scaling equations as functions of thrust and chamber pressure were derived based on results from a number of detailed thrust chamber design cases using a LeRC in-house computer program. The regeneratively cooled portion of the thrust chamber and nozzle was assumed to extend to an area ratio at which the gas side wall temperature was sufficiently low to permit attachment of a radiation cooled, coated columbium nozzle extension. A gas side wall temperature of 2910° R was assumed as the allowable transition temperature based on information in Ref. 5. Extrapolated design data, also from Ref. 5, were used to determine the variation in attach area ratio with chamber pressure. The effect of thrust level on attach area ratio was neglected.

A hot gas wall temperature of 1260° R (maximum temperature at local location) was assumed in the analysis of the regeneratively cooled portion of the thrust chamber. The assumed temperature was estimated from thrust chamber stress analysis results given in Refs. 4 and 6, and reflects an average of the design wall temperatures for the two reference applications. Results from the detailed thrust chamber heat transfer design cases indicated that a hydrogen temperature of 800° R at the cooling jacket exit and an average coolant velocity of 700 ft/sec were representative of the coolant temperature and velocity limits associated with the assumed hot gas wall temperature.

PUMPS AND MOTORS

The pumps were assumed to be positive displacement vane pumps operating at constant speeds of 8000 rpm (hydrogen) and 3000 rpm (oxygen). Speed magnitudes were selected to be compatible with surface rubbing velocity limits in hydrogen and oxygen reported in Ref. 7. Pump efficiency as a function of specific speed was derived from design data given in Ref. 8. The pumps were assumed to be driven by 115 volt, 3 phase, 400 hertz electric motors with efficiencies of 92 percent (hydrogen drive) and 78 percent (oxygen drive). Efficiency magnitudes were based on information in Ref. 9 and were assumed constant over the parametric study range.

TURBOALTERNATOR

The turboalternator speed was established at 24 000 rpm consistent with a two-pole alternator providing 400 hertz power. Alternator efficiency was assumed constant at 92 percent over the parametric study range. Because of the relatively low hydrogen flow magnitudes available as working fluid, a partial admission turbine was selected to drive the alternator. It was assumed that 10 percent of the total hydrogen flow was by-passed in order to provide turbine control. Turbine admission was selected at 12 percent. Basic turbine efficiency as a function of isentropic velocity ratio was taken from test results on a 12 percent admission turbine given in Ref. 10. Basic turbine efficiency was modified to include tip clearance effects based on data presented in Ref. 11.

ENGINE SYSTEM MASS AND ENVELOPE

Scaling equations which related the mass of the engine system components to thrust and chamber pressure were derived based on historical mass trends and/or estimates from conceptual sketches. Actual mass data for low-thrust hydrogen-oxygen components of the type assumed in the study effort are essentially nonexistent and it is important to note that the mass derivations typically involved extrapolations from outside the parametric study range. Scaling equations which related engine length and nozzle exit diameter to thrust and chamber pressure were derived using the fundamental equations which define thrust chamber and nozzle geometry as a function of thrust chamber throat area.

ANALYSIS RESULTS AND DISCUSSION

The initial objective of the study effort was to determine the feasible design range of the engine concepts in terms of thrust and chamber pressure and to provide preliminary performance, mass, and dimensional envelope data within the range for use in mission studies. In conducting the analysis, an upper limit on thrust was selected at 1000 pounds which corresponds to a maximum thrust-to-mass ratio of approximately 0.05 for a hydrogen-oxygen orbit-transfer-stage (including payload) having a start-burn mass of 60 000 pounds. A lower limit on chamber pressure was selected at 175 psia in order to ensure that hydrogen pressures upstream of the injector would be above critical pressure thus avoiding possible problems with boiling heat trnsfer in the cooling jacket. With these boundaries fixed, the lower limit on thrust and the upper limit on chamber pressure were then determined to establish the feasible design range of the engine concepts.

TURBOALTERNATOR CONCEPT

Figure 3 shows the feasible design range and thrust chamber cooling limits of the turboalternator concept. At a chamber pressure of 175 psia, design point thrust levels down to 200 pounds were feasible with the lower thrust limit being established by the thrust chamber coolant exit temperature limit of 800° R. As design point chamber pressure was increased from 175 psia, it was necessary to increase thrust as shown to avoid exceeding the 800° R coolant exit temperature limit. At a thrust level of 430 pounds and a chamber pressure of 420 psia, the concept was power limited with only small increases in maximum chamber pressure being possible in the thrust range from 430 to 1000 pounds.

The power limit at a thrust level of 430 pounds is illustrated in Fig. 4. The pressure and temperature trends are typical for an expander cycle engine. The figure shows that as chamber pressure was increased from 175 psia, the coolant exit temperature also increased and reached the limiting value of 800° R at a chamber pressure of 420 psia. The increased energy content of the hydrogen, however, was not sufficient to provide cycle power and increases in turbine pressure ratio (turbine inlet pressure-to-turbine discharge pressure) were necessary to achieve power balances. Note that at the higher chamber pressures, the required turbine pressure ratio increased rapidly. At chamber pressures above 420 psia, the turbine was unable to provide the necessary power regardless of pressure ratio magnitude.

Predicted performance for the turboalternator concept is given in Fig. 5. Note that the slight increase in maximum chamber pressure which was attainable as thrust was increased from 430 to 1000 pounds has not been considered in the analysis, i.e., the maximum chamber pressure for which parametric data were generated was 420 psia. As indicated in Fig. 5, delivered specific impulse increases with increasing thrust and increasing chamber pressure from 437 lbf-sec/lbm at a thrust of 200 pounds and chamber pressure of 175 psia to 452 lbf-sec/lbm at a thrust of 1000 pounds and chamber pressure of 420 psia. The increase in specific impulse occurs primarily because of a decrease in kinetics loss as thrust and chamber pressure are increased.

Engine system mass and dimensional envelope parametric data over the feasible design range are shown in Fig. 6. Figure 6(a) shows that engine system mass increases with increasing thrust and increasing chamber pressure with thrust having the greater influence. The curves in Fig. 6(a) also indicate that engine system mass is approaching a minimum value at a chamber pressure of 175 psia. Figure 6(b) shows the familiar trends of increased engine length and increased nozzle exit diameter as thrust is increased at a given chamber pressure or chamber pressure is decreased at a given thrust. In both cases the chamber throat diameter must be increased to accommodate the propellant mass flow which, for a given nozzle are ratio, results in a longer nozzle and a larger nozzle exit diameter.

AUXILIARY POWER CONCEPT

Figure 7 shows the feasible design range of the auxiliary power concept. At a chamber pressure of 175 psia, design point thrust levels down to 200 pounds were feasible with the 200 pound level being established by the coolant exit temperature limit of 800° R. This boundary of the feasible design range is the same as that determined for the turboalternator concept. Also, as chamber pressure was increased from 175 psia, it was necessary to increase thrust in the same manner as the turboalternator concept in order to maintain the 800° R coolant exit temperature. Comparison of the cooling limit boundary for the auxiliary power concept (Fig. 7) with the cooling limit boundary for the turboalternator concept (Fig. 3) indicates that, from a cooling standpoint, the auxiliary power concept was restricted to lower chamber pressures as thrust was increased above approximately 350 lbf. This occurred because of the lower coolant inlet pressures in the auxiliary power concept and the resultant effect on hydrogen density and velocity in the cooling jacket. Higher chamber pressures than those shown in Fig. 7 could have been achieved by increasing coolant inlet pressure with a corresponding increase in auxiliary power requirements. This area however, was not examined in the study effort.

Predicted performance for the auxiliary power concept is given in Fig. 8. With the exception of changes and additions associated with the different feasible design range, the specific impulse magnitudes shown are the same as for the turboalternator concept. Rigorous analyses would give a slightly higher specific impulse for the auxiliary power concept because pre-combustion propellant pressures and resultant increase in enthalpies would be provided by a source external to the engine system. The enthalpy increases, however, are very small and a cursory analysis indicated the effect on specific impulse would be much less than 1 lbf-sec/lbm.

Engine system mass and dimensional envelope parametric data are shown in Fig. 9. Engine mass (Fig. 9(a)) trends are the same as for the turboalternator concept with mass magnitudes lower primarily because of the elimination of the turboalternator although some decrease in mass was evident because of the reduced system pressures associated with the auxiliary power concept. Engine mass magnitudes do not include mass associated with the auxiliary power system. Engine dimensions shown in Fig. 9(b) are the same as for the turboalternator concept with the exception of the changes associated with the different feasible design range.

The total power requirements for the pumps of the auxiliary power concept over its feasible design range are shown in Fig. 10. At a given chamber pressure, the figure indicates a small decrease in power-to-thrust ratio as thrust is increased. This favorable trend is due primarily to increases in efficiency of the pumps at the higher thrust levels.

SUMMARY OF RESULTS

Two hydrogen-oxygen rocket engine system concepts were analyzed parametrically over a thrust range from 100 to 1000 pounds and a chamber pressure range from 175 to 1000 psia. Both concepts were regeneratively cooled with hydrogen and were pump-fed by electric motor driven positive displacement pumps. Electric power for the pump-drive motors was assumed to be provided by either a turboalternator in an expander cycle (turboalternator concept) or some means external to the engine system (auxiliary power concept). Major results of the analyses are summarized as follows:

1. At the minimum study chamber pressure of 175 psia, design point thrust levels down to 200 pounds were feasible with both concepts.

2. For the turboalternator concept, the upper limit on chamber pressure between 200 pounds thrust and 430 pounds thrust was thrust dependent and was established by thrust chamber cooling restrictions. Above 430 pounds thrust the concept was power limited to approximately 420 psia chamber pressure.

3. For the auxiliary power concept, the upper limit on chamber pressure between 200 pounds thrust and 1000 pounds thrust was thrust dependent and was established by thrust chamber cooling restrictions. The maximum attainable chamber was approximately 550 psia at a thrust level of 1000 pounds.

4. Delivered specific impulse was thrust and chamber pressure dependent. For the turboalternator concept, impulse ranged from 437 lbf-sec/lbm at a thrust of 200 pounds and chamber pressure of 175 psia to 452 lbf-sec/lbm at a thrust of 1000 pounds and a chamber pressure of 420 psia. For the auxiliary power concept, impulse ranged from 437 lbf-sec/lbm at a thrust of 200 pounds and a chamber pressure of 175 psia to 455 lbf-sec/lbm at a thrust of 1000 pounds and a chamber pressure of 550 psia.

5. Engine system mass was thrust and chamber pressure dependent. For the turboalternator concept, mass ranged from 53 lbm at a thrust of 200 pounds and a chamber pressure of 175 psia to 180 lbm at a thrust of 1000 pounds and a chamber pressure of 420 psia. For the auxiliary power concept, mass ranged from 39 lbm at a thrust of 200 pounds and a chamber pressure of 175 psia to 151 lbm at a thrust of 1000 pounds and a chamber pressure of 550 psia.

REFERENCES

1. Gordon, S.; and McBride, B. J.: Computer Program for Calculation of Complex Chemical Equilibrium Compositions, Rocket Performance, Incident and Reflected Shocks, and Chapman-Jouguet Detonations. NASA SP-273, 1976.

2. Bittker, D. A.; and Scullin, V. J.: General Chemical Kinetics Computer Program for Static and Flow Reactions, with Application to Combustion and Shock Tube Kinetics. NASA TN D-6586, 1972.

3. JANNAF Rocket Engine Performance Prediction and Evaluation Manual CPIA-Publ-246, John Hopkins Univ., 1975.

4. Yost, M. C.; Marker, H. E.; and Dennies, P. C.: Advanced Thrust Chamber Technology. (R76-116, Rocketdyne, Canoga Park, Calif.; NASA Contract NAS3-17825.) NASA CR-135221, 1977.

5. Mellish, J. A.: Advanced Engine Study for Mixed-Mode Orbit-Transfer Vehicles. NASA CR-159491, 1978.

6. Paster, R. D.: Hydrogen-Oxygen APS Engines. Volume I: High-Pressure Thruster. (R-8837-1-VOL-1, Rocketdyne, Canoga Park, Calif.; NASA Contract NAS3-14352.) NASA CR-120805, 1973.

7. Gluzek, F.; et al.: Liquid Oxygen/Liquid Hydrogen Boost/Vane Pump for the Advanced Orbit Transfer Vehicles Auxiliary Propulsion System. NASA CR 159648, 1979.

8. Balje, O. E.: A Study on Design Criteria and Matching of Turbomachines. J. Eng. for Power, Series A, vol. 84, 1962, pp. 83-114.

9. Nichols, J. F.: Study of Liquid Oxygen/Liquid Hydrogen Auxiliary Propulsion Systems for the Space Tug. NASA CR-134790, 1975.

10. Klassen, H. A.: Cold-Air Investigation of Effects of Partial Admission on Performance of 3.75 Inch Mean Diameter Single Stage Axial Flow Turbine. NASA TN D-4700, 1968.

11. Anon.: Liquid Rocket Engine Turbines. NASA SP-8110, 1974.

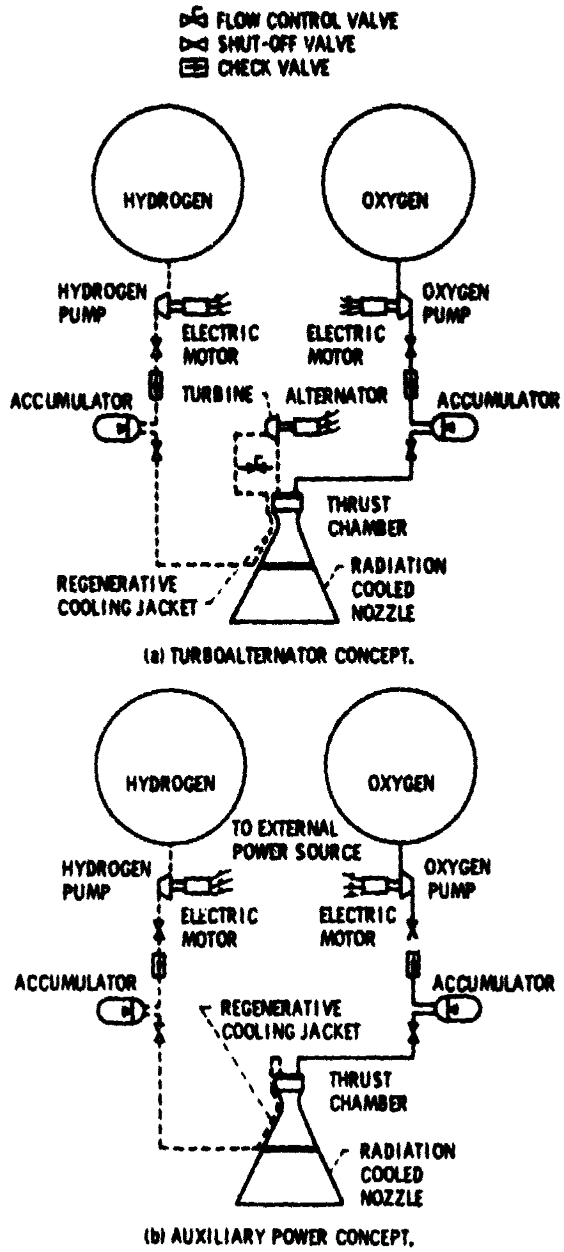

Figure 1. - Schematic drawings of engine system concepts.

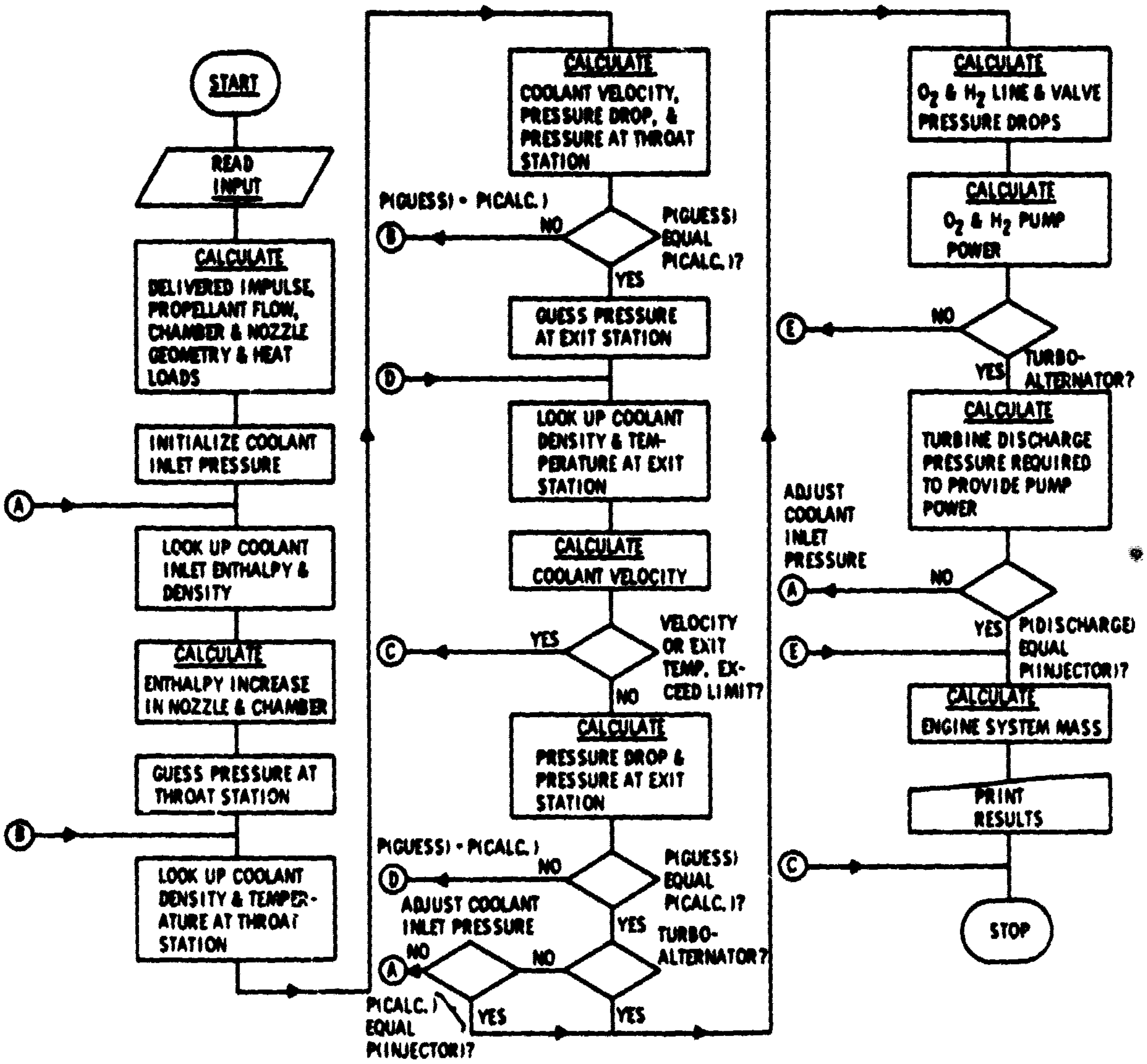

Figure 2. - Computer simulation logic diagram.

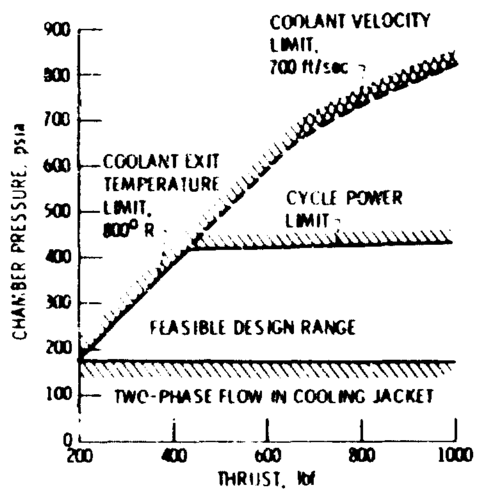

Figure 3. - Feasible design range-turboalternator concept. Maximum hot gas wall temperature - 1260° R; coolant exit temperature ≤ 800° R; coolant average velocity ≤ 700 ft/sec.

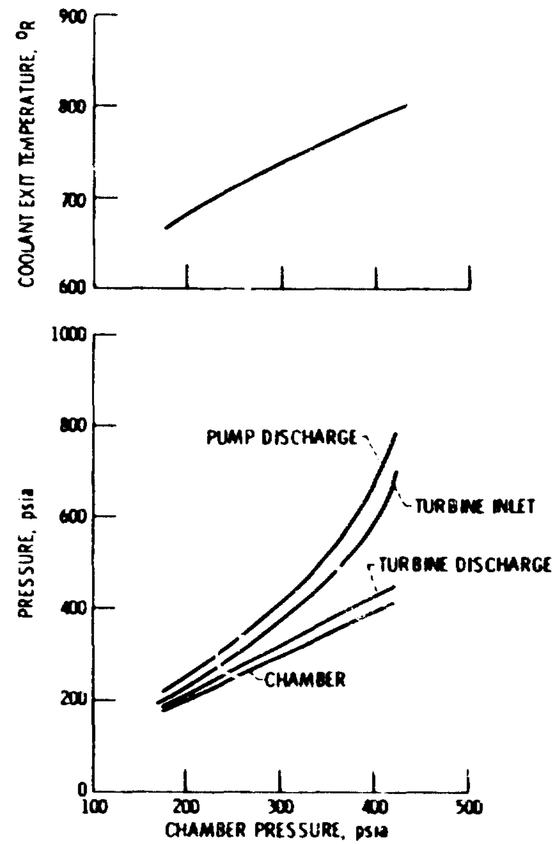

Figure 4. - Effect of chamber pressure on hydrogen feed system pressures and temperature-turboalternator concept. Thrust - 430 lbf.

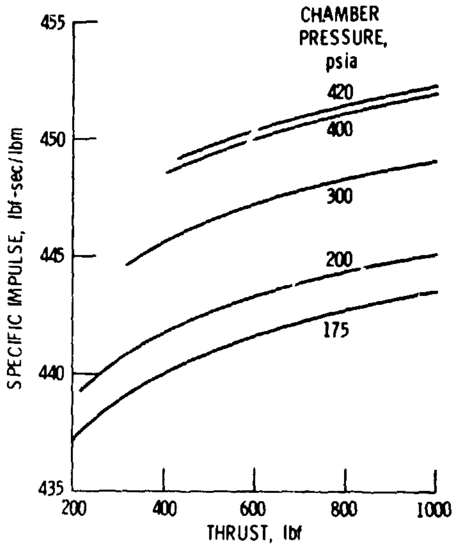

Figure 5. - Effect of thrust and chamber pressure on performance-turboalternator concept. Propellants - Hydrogen/oxygen; mixture ratio (O/F) = 6.0; area ratio = 400:1.

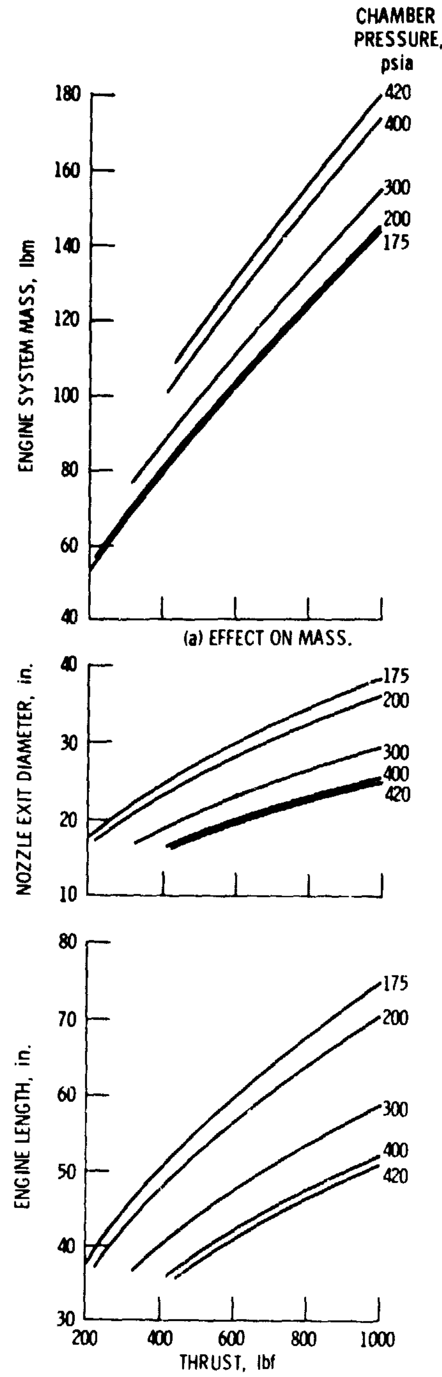

Figure 6. - Effect of thrust and chamber pressure on mass and dimensional envelope-turboalternator concept.

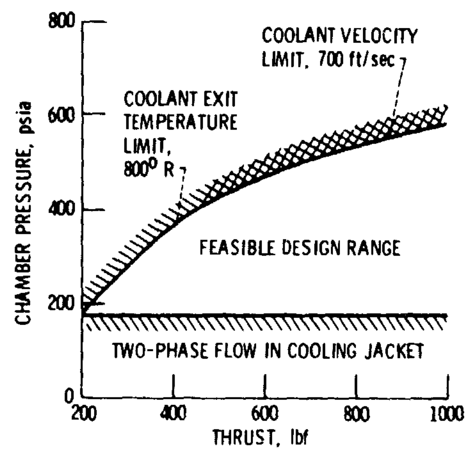

Figure 7. - Feasible design range-auxiliary power concept. Maximum hot gas wall temperature • 1260° R; coolant exit temperature ≤ 800° R; coolant average velocity ≤ 700 ft/sec.

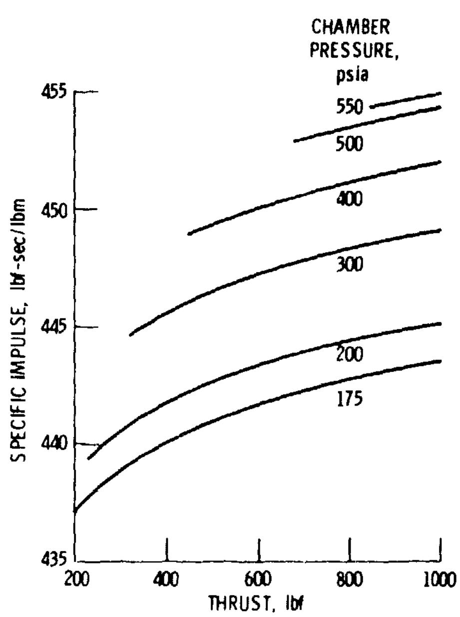

Figure 8. - Effect of thrust and chamber pressure on performance-auxiliary power concept. Propellants - hydrogen/oxygen; mixture ratio (O/F); area ratio - 400:1.

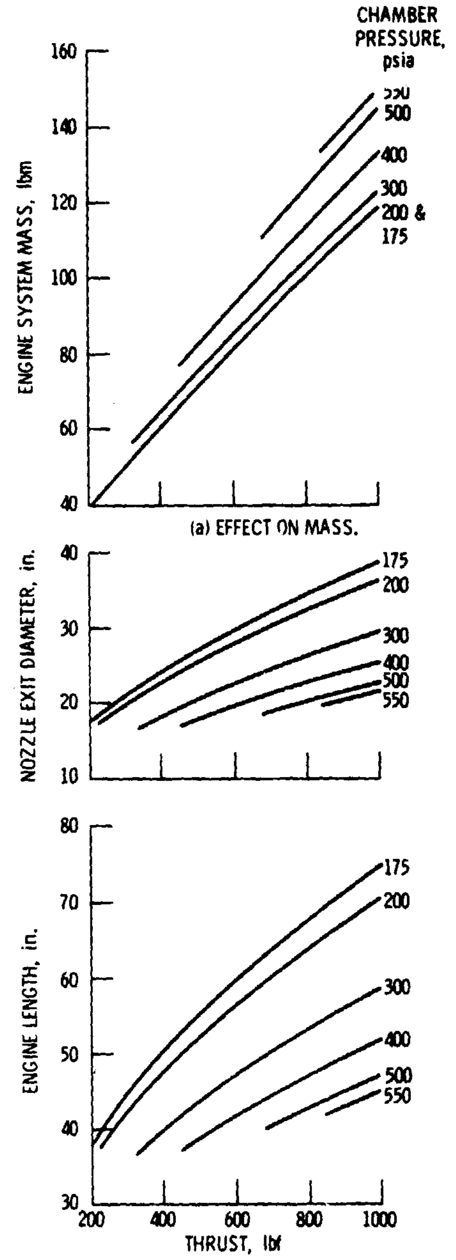

Figure 9. - Effect of thrust and chamber pressure on mass and dimensional envelope-auxiliary power concept.

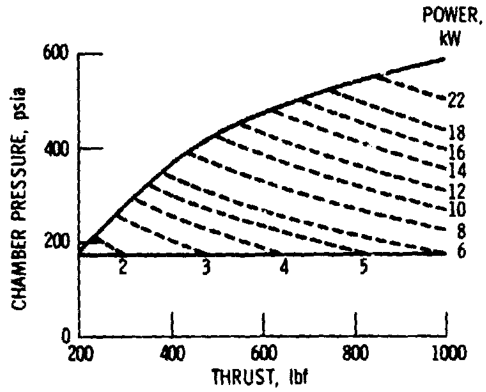

Figure 10. - Propellant feed system power requirements-auxiliary power concept.