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# In-Situ Propellant Rocket Engines for Mars Mission Ascent Vehicle

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# IN-SITU PROPELLANT ROCKET ENGINES FOR MARS MISSIONS ASCENT VEHICLE

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## Abstract

When contemplating the human exploration of Mars, many scenarios using various propulsion systems have been considered. One propulsion option among them is a vehicle stage with multiple, pump-fed rocket engines capable of operating on propellants available at Mars. This reduces the Earth launch mass requirements, resulting in economic and payload benefits. No plentiful sources of hydrogen on Mars have been identified on the surface of Mars, so most commonly used high-performance liquid fuels, such as hydrogen and hydrocarbons, can be eliminated as possible in-situ propellants. But 95 percent of the Martian atmosphere consists of carbon dioxide, which can be converted into carbon monoxide and oxygen. The carbon monoxide–oxygen propellant combination is a candidate for a Martian in-situ propellant rocket engine. This report analyzes the feasibility of a pump-fed engine cycle using the propellant combination of carbon monoxide and oxygen.

## I. Introduction

Human exploration of Mars is a key portion of NASA's Space Exploration Initiative. For Mars exploration many vehicle and engine configurations have been investigated. Of these configurations, one viable option uses in-situ propellants (propellants produced from resources available at the mission site), which can result in lower Earth launch mass, up to 30 percent over 10 missions in a 20-year period[1] and cost savings due to the decrease in the number of launches required, including launches of propellant production equipment.[2]

Several feasible in-situ propellant combinations have been identified by Ramohalli et al.[3]:

(1) Producing only oxygen at Mars and bringing the fuel from Earth
(2) Producing carbon monoxide as the fuel and oxygen as the oxidizer
(3) Producing methane as the fuel and oxygen as the oxidizer.

The first and third concepts require that all or some of the fuel be brought to Mars from Earth. The major advantage of the second concept is that both oxygen and carbon monoxide can be manufactured from carbon dioxide, which makes up about 95 percent of the Mars atmosphere and is the one thoroughly known and readily available Martian resource. An in-situ propellant rocket engine will be useful for lifting a vehicle off the Martian surface to a vehicle orbiting Mars or Earth. In most Mars ascent vehicle studies the rocket engines are treated as "black boxes" of a given weight and thrust. In fact, a carbon monoxide and oxygen propelled rocket engine has not been designed in detail. Hence, a more detailed investigation can determine if an in-situ propellant rocket engine can, indeed, be designed that will meet the requirements of a journey from the Martian surface to an orbit around Mars or Earth.

This analysis investigates the possibility of using carbon monoxide and oxygen propellants in a rocket engine that can be used for a Mars mission ascent vehicle. Multiple, pump-fed, regeneratively cooled, rocket engines were considered for liftoff for the following reasons: (1) multiple engines for redundancy and engine-out capability; (2) pump-fed engines for higher chamber pressure and performance; (3) regenerative cooling for maximum heat extraction to the working fluid. Because of its simplicity, a full expander cycle was chosen using supercritical oxygen as the coolant and the working fluid.

## II. Description of the Expander Cycle

Figure 1 is a schematic of a full expander cycle for $CO$-$O_2$ propellants, where oxygen is the turbine working fluid. Propellants are pumped from low-pressure storage tanks by fuel and oxidizer pumps. The oxygen is pumped to the required pressure by the oxidizer pump. It then cools the combustion chamber, thereby picking up energy, and expands through the turbines to drive the propellant pumps. All of the oxygen eventually is injected into the thrust chamber through the injector. The carbon monoxide is supplied to the thrust chamber assembly by the fuel pump and then injected directly into the combustion chamber. Thrust control is achieved by using a turbine bypass valve. The majority of the oxygen flows through the turbines, while a small amount is bypassed. The bypass flow is mixed with the turbine flow upstream of the injector.

## III. Rocket Engine Analysis Procedure

In the expander cycle the available turbine power is dependent on the heat picked up in the cooling jacket by the working fluid and also on the pressure ratio across the turbine. The necessary pump power is dependent on the total pressure required in the cycle for the working fluid and the chamber pressure for the other propellant. At low chamber pressures abundant turbine power is available to drive the pumps. At high chamber pressures the pump power requirements increase beyond the available turbine power. The maximum chamber pressure at which the cycle will still function is where the available turbine power is slightly more than the required pump power.

To determine the maximum chamber pressure for this application, the turbine power and pump power were calculated over a range of chamber pressures, keeping the combustion chamber wall temperature below 445 K (800 °R). A total thrust level of 1340 kN (300,000 $lb_f$) was chosen to enable the vehicle to reach orbit around Mars. To determine the cooling-jacket pressure losses, the heat flux from the combustion gases was balanced with the heat flux to the coolant. The heat flux to the coolant is dependent on the cooling-jacket geometry and the combustion chamber geometry. The cooling-jacket geometry was calculated for minimum pressure drop for a given chamber pressure. The procedures and computer programs used for this analysis are shown in the flow chart in Fig. 2 and are described in the sections that follow.

## Thrust Chamber Calculations

The Complex Chemical Equilibrium Composition (CEC) computer program[4] is used to determine the characteristic exhaust velocity ($C^*$), the combustion temperature, the thrust coefficient ($C_f$), and the specific impulse ($I_{sp}$), for a given chamber pressure, mixture ratio, and area ratio. From this CEC output the thrust chamber throat radius can be calculated from the following equations[5]:

$$C_f = \frac{F}{A_t P_c} \quad \text{or} \quad A_t = \frac{F}{P_c C_f} \tag{1}$$

where $C_f$ is the thrust coefficient, F is the thrust, $P_C$ is the chamber pressure, and $A_t$ is the area of the throat.

The Rao Method Optimum Nozzle Contour Program[6] was used to optimize the thrust chamber nozzle from the throat to the exit. This program can be used to calculate a supersonic exhaust nozzle contour for a given nozzle area ratio that gives maximum thrust for its length. One of the four options to determine the nozzle contour is to use calculus of variations for an ideal gas, constant $\gamma$ expansion. The calculus of variations established the geometric relationships using the method of characteristics for an optimum nozzle contour. The output from the Rao Method Optimum Nozzle Contour Program gives the nozzle mass flow and the corresponding nozzle contour. The output also includes the Mach number, specific impulse, gas pressure, gas density, and gas temperature along the contour of the nozzle for each axial location.

The combustion chamber geometry was determined using equations from Huzel and Huang[5]. The throat area is generally the starting point in designing the combustion chamber. To calculate the combustion chamber length for a cylindrical combustion chamber, Eq. (2) was used for an approximate value of the combustion chamber volume, which is typically defined as the space from the injector face to the nozzle throat plane (see Fig. 3),

$$V_c = A_t[L_c \epsilon_c + 0.333 r_t \cot\theta(\epsilon_c^{0.333}c - 1)] \tag{2}$$

where $\epsilon_c$ is the chamber contraction area ratio, $\theta$ is the chamber contraction angle, $A_t$ is the throat area, $r_t$ is the throat radius, and $L_c$ is the length of the combustion chamber. The theoretical required chamber volume is proportional to the mass flow rate of the propellants, $m_{tc}$, the average specific volume, V, and the stay time necessary for efficient combustion, $t_s$ (Ref. 5):

$$V_c = m_c V t_s \tag{3}$$

Substituting Eq. (3) into Eq. (2) and solving for the chamber length, $L_c$, gives

$$L_c = \frac{1}{\epsilon_c}\left[\frac{m_{tc} V t_s}{A_t} - 0.333 r_t \cot\theta(\epsilon_c^{0.333} - 1)\right] \tag{4}$$

The contour of the combustion chamber from the throat to the cylindrical section was calculated assuming a 15° contraction angle, $\theta$, and an upstream throat radius of curvature of $1.0 r_t$.

## Cooling-Jacket Calculations

An unpublished, in-house computer code was used to predict the heat flux from the combustion gases to the coolant. This computer code uses the CEC computer program[4] for the hot-gas-side calculations and the FLUID program[7] to obtain coolant-side properties. In the code the hot-gas-side properties are evaluated at an axial station, then the coolant-side properties are evaluated. A heat flux balance between the two sides is then undertaken. When the coolant-side and hot-gas-side heat flux is balanced, the next axial station is evaluated. Pressure drop and temperature rise for the coolant are determined from station to station and are calculated for the total cooling jacket.

## Pump Power Calculations

The power necessary to drive the pump is calculated from the following equation:

$$\text{Power} = \frac{m_p H_p}{\eta_p} \tag{5}$$

where $m_p$ is the flow rate of the fluid through the pump, $H_p$ is the pump headrise, and $\eta_p$ is the pump efficiency. The pump headrise is calculated based on the average density across the pump $\rho_{av}$ and the pressure rise across the pump $\Delta P$ required by the system characteristics

$$H_p = \frac{\Delta P}{\rho_{av}} \tag{6}$$

The pump efficiency can be estimated from the Worthington efficiency curves (e.g., see Ref. 8), which is a function of the specific speed $N_s$ and the pump flow rate $m_p$. The specific speed $N_s$ is calculated from

$$N_s = \frac{N(m_p)^{0.5}}{(H_p)^{0.75}} \tag{7}$$

These calculations were undertaken for both the fuel pump and the oxidizer pump under operating conditions pertinent to the various combustion chamber configurations being considered. For this analysis pump efficiencies varied from 60 to 85 percent.

### Turbine Power Calculations

The available turbine power was calculated using an unpublished turbine design computer code developed at Lewis. The code calculates design characteristics for single- or two-stage reaction or impulse turbines. The code includes two design options: (1) the turbine inlet and discharge pressure are known, from which the program calculates the basic turbine design parameters and the output power; and (2) the turbine inlet pressure and the required power are known, from which the program calculates the basic parameters and the discharge pressure necessary to provide the required power. The design code can use a variety of working fluids, including hydrogen, oxygen, nitrogen, and hydrogen-oxygen combustion products. After reading the input file, the program determines inlet properties of the working fluids and estimates a discharge temperature for design option 1. The volume flow at the turbine rotor discharge is then determined and used in the specific speed, specific diameter, and aerodynamic loss analyses. An inner loop minimizes turbine leaving loss using blade speed as the independent variable. If the optimum blade speed exceeds the allowable speed, aerodynamic losses are determined for the maximum speed condition. After aerodynamic losses have been determined, the program calculates disk friction and clearance losses, overall efficiency, and output power. For design option 1 discharge temperature is calculated and compared with the estimated value. Iterations are then performed until the estimated temperature is equal to the calculated temperature within a specified tolerance.

## VI. Results

A possible Mars ascent vehicle (MAV) rocket engine was analyzed that would ascend from the Martian surface and orbit Mars or Earth. An expander cycle was chosen for the rocket engine cycle, using one single-stage 50-percent-reaction turbine to drive both the fuel and the oxidizer pumps and using single-stage centrifugal pumps to pump the fuel and the oxidizer. Since the densities of carbon monoxide and oxygen are similar, a single turbine on a single shaft could be used for both pumps at the same speed. The oxidizer, oxygen, was used as the working fluid and thrust chamber coolant. Four engines, each at a thrust level of 334 kN (75,000 lbf), were chosen to meet the vehicle requirements to reach an orbit around Mars or Earth.

To determine the maximum chamber pressure at which the pump power requirements were equal to or less than the available turbine power, the chamber pressure was varied from 1.72 to 13.79 MPa (250 to 2000 psi). A propellant mixture ratio of 0.55 was chosen because it gave the maximum specific impulse for this range of pressures at the maximum nozzle exit area ratio. The exit pressure at the nozzle exit was kept above the Martian atmospheric pressure of 690 Pa (0.1 psia).

Engine expander cycles were designed at each chamber pressure using the procedure shown in Fig. 2. Output from the CEC program[4] was iterated upon to determine the largest nozzle exit area ratio that would give an exit pressure of 690 Pa (0.1 psi). The maximum exit area ratio gives the maximum specific impulse for a given chamber pressure. These exit area ratios are given in Table I for each chamber pressure, as well as the corresponding specific impulse. From the CEC output, throat radii were calculated for each chamber pressure (see Table II). The Rao Optimization Program[6] gave optimum mass flow rates and optimum nozzle contours. The mass flow rates in Table II assumed 94 percent characteristic exhaust velocity efficiency $\eta_{C*}$. After determining the chamber contour, the rocket engine heat-transfer evaluation program was used to calculate the minimum coolant inlet pressure necessary to keep the chamber wall temperature below 445 K (800 °R).

The carbon monoxide pump power was calculated such that the pump outlet pressure was 15 percent greater than the chamber pressure, which allowed for the pressure drop across the injector and the main carbon monoxide valve. The oxygen pump power was calculated such that the pump outlet pressure was 1 percent greater than the coolant inlet pressure, which allowed for pressure losses in

TABLE I. SPECIFIC IMPULSE AND AREA RATIO AT EACH CHAMBER PRESSURE

[Exit pressure (Mars atmospheric pressure), 690 Pa (0.1 psi).]

| Chamber pressure, MPa | Chamber pressure, psi | Area ratio | Specific impulse, sec |
|---|---|---|---|
| 1.72 | 250 | 228 | 288 |
| 2.07 | 300 | 264 | 290 |
| 3.45 | 500 | 397 | 297 |
| 5.17 | 750 | 545 | 302 |
| 6.89 | 1000 | 684 | 305 |
| 8.62 | 1250 | 816 | 308 |
| 10.34 | 1500 | 942 | 310 |
| 12.07 | 1750 | 1064 | 311 |
| 13.79 | 2000 | 1182 | 312 |

TABLE II. THROAT RADIUS AND NOZZLE MASS FLOW AT EACH CHAMBER PRESSURE

[Characteristic exhaust velocity efficiency, $\eta_{C^*}$, 94 percent.]

| Chamber pressure | Throat radius | | Mass flow rate | |
|---|---|---|---|---|
| MPa | cm | in. | kg/sec | $lb_m$/sec |
| 1.72 | 17.155 | 6.754 | 127.15 | 280.3 |
| 2.07 | 15.588 | 6.137 | 125.92 | 277.6 |
| 3.45 | 11.966 | 4.711 | 122.44 | 269.9 |
| 5.17 | 9.715 | 3.825 | 120.10 | 264.8 |
| 6.89 | 8.379 | 3.299 | 118.45 | 261.1 |
| 8.62 | 7.475 | 2.943 | 117.34 | 258.7 |
| 10.34 | 6.810 | 2.681 | 116.45 | 256.7 |
| 12.07 | 6.294 | 2.478 | 115.74 | 255.2 |
| 13.79 | 5.880 | 2.315 | 115.16 | 253.9 |

TABLE III. DIMENSIONS FOR THE CHAMBER CONTOUR AND COOLANT PASSAGES FOR CHOSEN CONFIGURATION

[Chamber pressure, 4.14 MPa; maximum wall temperature, 400 K; nozzle exit pressure, 1.79 kPa; $C^*$ efficiency, 94 percent; thrust, 334 kN; coolant flow rate, 50.78 kg/sec; coolant inlet pressure, 13.79 MPa; coolant exit pressure, 9.8 MPa.]

| Axial distance from throat, cm | Contoured chamber diameter, cm | Coolant passage height, cm | Coolant passage width, cm | Coolant passage aspect ratio | Number of coolant passages |
|---|---|---|---|---|---|
| -112.0 | 41.45 | 0.833 | 0.104 | 8 | 630 |
| -97.29 | 41.45 | .833 | .104 | 8 | 630 |
| -82.56 | 41.45 | .833 | .104 | 8 | 630 |
| -67.82 | 41.45 | .833 | .104 | 8 | 630 |
| -53.09 | 41.45 | .833 | .104 | 8 | 630 |
| -38.35 | 41.45 | .833 | .104 | 8 | 630 |
| -34.37 | 39.86 | .833 | .104 | 8 | 600 |
| -30.31 | 39.33 | .833 | .104 | 8 | 600 |
| -21.24 | 34.47 | .833 | .104 | 8 | 525 |
| -12.17 | 29.61 | .833 | .104 | 8 | 450 |
| -3.097 | 24.75 | .691 | .086 | 8 | 450 |
| -1.562 | 24.14 | .671 | .084 | 8 | 450 |
| 0.00 | 23.93 | .671 | .084 | 8 | 450 |
| 2.754 | 24.11 | .671 | .084 | 8 | 450 |
| 3.889 | 25.93 | .731 | .091 | 8 | 450 |
| 5.728 | 28.94 | .813 | .102 | 8 | 450 |
| 7.909 | 32.58 | .914 | .114 | 8 | 450 |
| 10.53 | 36.98 | 1.036 | .129 | 8 | 450 |
| 13.76 | 42.36 | 1.178 | .147 | 8 | 450 |
| 17.83 | 49.02 | 1.361 | .170 | 8 | 450 |
| 27.29 | 63.71 | .894 | .112 | 8 | 900 |
| 39.74 | 81.36 | 1.138 | .142 | 8 | 900 |
| 73.96 | 122.2 | 1.708 | .213 | 8 | 900 |
| 121.97 | 167.1 | 2.337 | .292 | 8 | 900 |
| 177.94 | 208.2 | 2.906 | .363 | 8 | 900 |
| 227.0 | 237.5 | (a) | (a) | (a) | (a) |
| 280.1 | 263.8 | (a) | (a) | (a) | (a) |
| 343.5 | 289.7 | (a) | (a) | (a) | (a) |
| 422.4 | 315.0 | (a) | (a) | (a) | (a) |
| 447.0 | 321.6 | (a) | (a) | (a) | (a) |

[a]No coolant channels existed at these axial location because of low heat flux at the higher area ratios (radiation cooling).

the lines and across the main oxygen valve. The turbine power was calculated such that the turbine inlet pressure was 10 percent less than the coolant outlet pressure, accounting for the pressure losses across the turbine bypass valve. The turbine outlet pressure was determined from the chamber pressure and the injector pressure drop. Real fluid properties were used for both the carbon monoxide and oxygen calculations. The pump power and the turbine power were plotted as functions of chamber pressure in Fig. 4. The pump power line and the turbine line cross at a chamber pressure around 4.00 MPa (580 psi). A chamber pressure of 4.14 MPa (600 psi), slightly above the intersection in Fig. 4, was chosen for the final analysis, as fine-tuning of the configuration allowed for a slightly higher chamber pressure.

In the final analysis the process shown in Fig. 2 was used with some additional iterations. The final nozzle exit area ratio was 210, which resulted in an exit pressure of 1.79 kPa (0.26 psi), slightly above atmospheric pressure to assure no overexpansion of the nozzle if run in a throttled mode. Figure 5 shows the final combustion chamber contour. The cooling-jacket geometry was parametrically varied until a configuration was determined that resulted in 10 percent more available turbine power than required pump power. This 10 percent power surplus allowed for a small portion of flow to bypass the turbines, resulting in thrust control. The coolant passages were uniformly spaced circumferentially. The height, width, and number of coolant passages were varied in the axial direction for a given chamber contour and were also varied for the various cooling-jacket geometries for optimum cooling of the chamber wall. Beyond an area ratio of 75, the chamber nozzle was assumed to be radiation cooled because the heat flux was below 258 $kW/m^2$ (0.158 $Btu/in.^2$ sec) and regenerative cooling was not necessary to keep the wall temperature below 445 K (800 °R). Table III gives the dimensions for the chamber contour and for the coolant passages in the coolant jacket at several axial locations, as determined using the rocket engine heat-transfer evaluation program. The final configuration gave a 10-percent power surplus and a maximum combustion chamber wall temperature of 400 K (720 °R). Figure 6 shows the expander cycle schematic with the pressures, temperatures, and flow rates for the final configuration. The supercritical oxygen coolant provided sufficient heat capacity to cool the combustion chamber and to drive the turbines.

## VII. Concluding Remarks

One viable rocket propulsion system for a Mars ascent vehicle (MAV) is multiple, pump-fed, regeneratively cooled, rocket engines using the in-situ propellant combination of carbon monoxide and oxygen. The full expander cycle, which is simple and efficient, can be used for an in-situ propellant rocket engine for a MAV, using supercritical oxygen as the working fluid. The supercritical oxygen can provide sufficient heat capacity to cool the combustion chamber and to drive the turbines up to a chamber pressure of 13.79 MPa (2000 psia). The configuration was designed at a chamber pressure of 4.14 MPa (600 psia) to optimize power usage and at a mixture ratio of 0.55 to optimize specific impulse. Four engines, each at a thrust level of 334 kN (75,000 $lb_f$), were chosen to meet vehicle requirements to reach an orbit around Mars or Earth. A 10 percent power surplus was included in the system to allow for a small portion of the flow to bypass the turbines.

## References


1. Galecki, D.L., "In-Situ Propellant Advantages for Fast Transfer to Mars," AIAA Paper 88-2901, July 1988.

2. Duke, M.B., and Keaton, P.W., eds., "Manned Mars Missions, Working Group Papers," NASA TM-89320 and TM-89321, 1986.

3. Ramohalli, K., et al., "Some Aspects of Space Propulsion with Extraterrestrial Resources," J. Spacecraft, Vol. 24, No. 3, May-June 1987, pp. 236–244.

4. McBride, B.S., and Gordon, S., "Computer Program for Calculation of Complex Chemical Equilibrium Compositions, Rocket Performance, Incident and Reflected Shocks, and Chapman-Jouguet Detonations," NASA SP-273, Interim Revision, 1976.

5. Huzel, D.K., and Huang, D.H., "Design of Liquid Propellant Rocket Engines," NASA SP-125, 1971.

6. Nickerson, G.R., Dang, A.L., and Dunn, S.S., "The Rao Method Optimum Nozzle Contour Program," NAS8-36863, 1988. (A copy of this program can be obtained through CPIA.)

7. Fessler, T.E.: "FLUID: A Numerical Interpolation Procedure for Obtaining Thermodynamic and Transport Properties," NASA TM X-3572, 1977.

8. Shepherd, D.G.: Principles of Turbomachinery, The MacMillan Co., 1956.

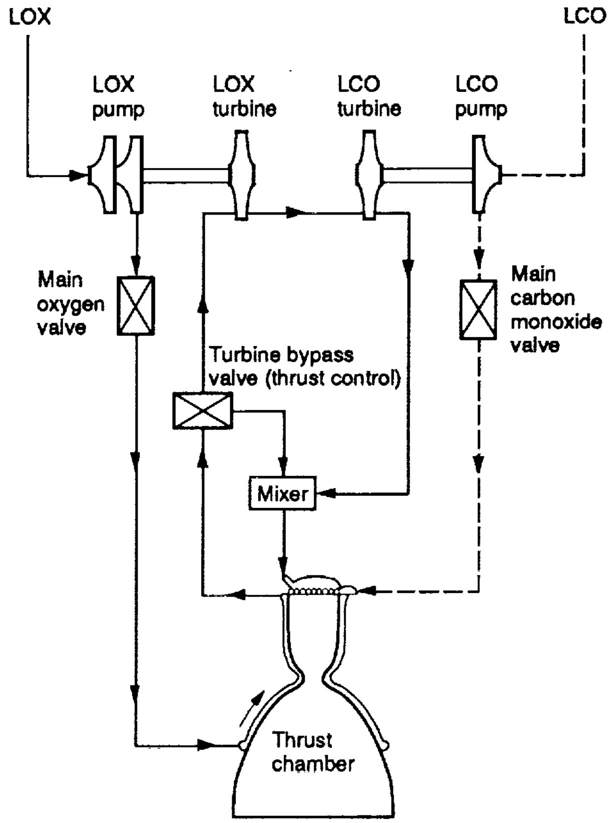


Figure 1.—LOX-LCO expander cycle.

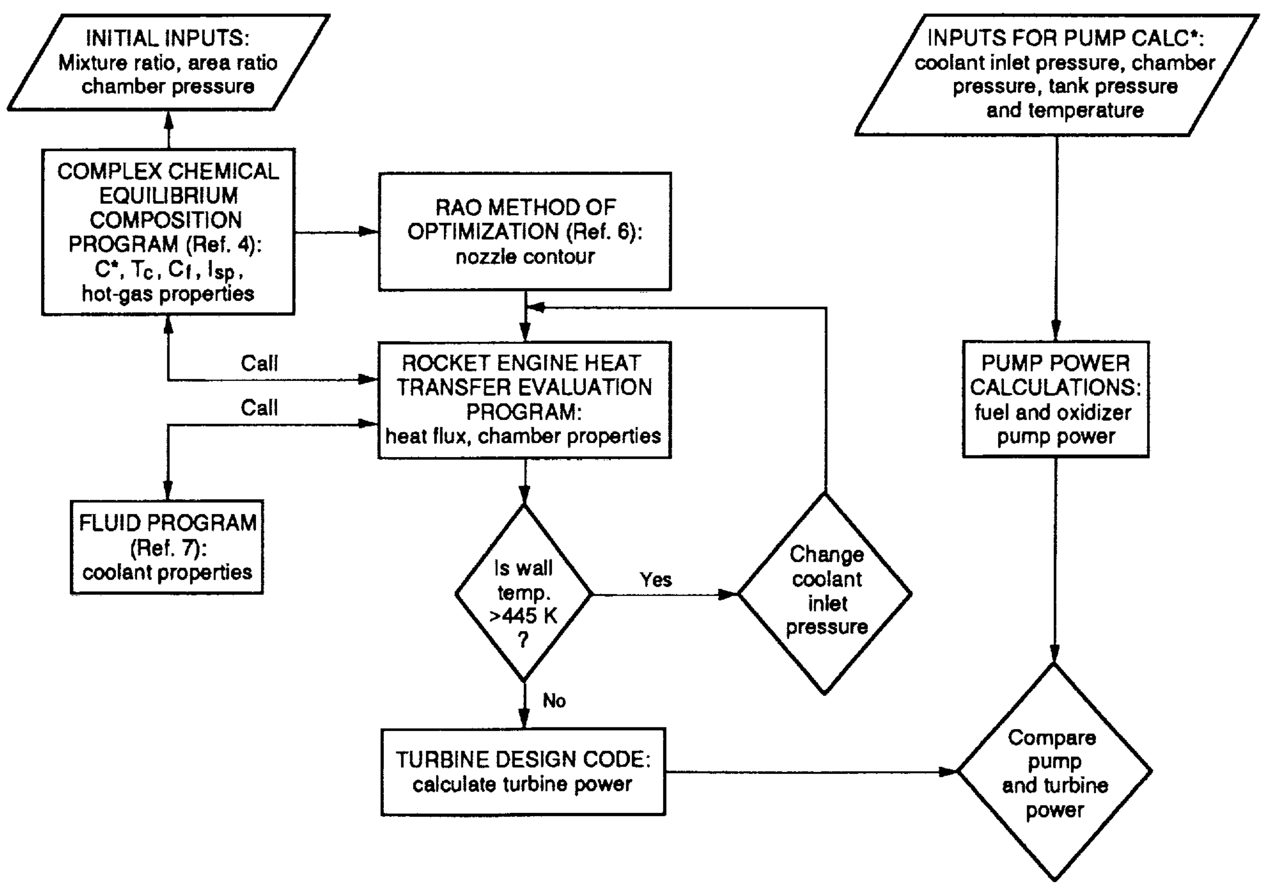


Figure 2.—Flowchart of computer programs used in this analysis.

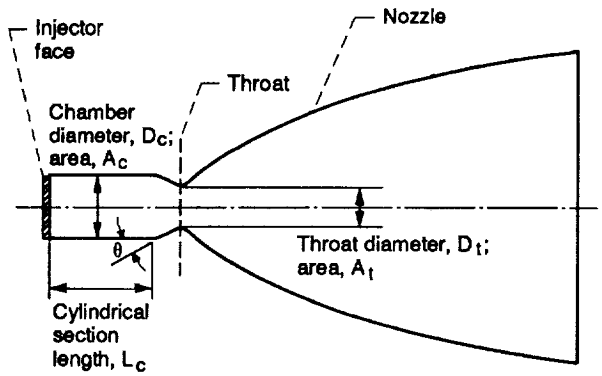


Figure 3.—Typical thrust chamber contour. Chamber contraction area ratio, $E_c = A_c/A_t$.

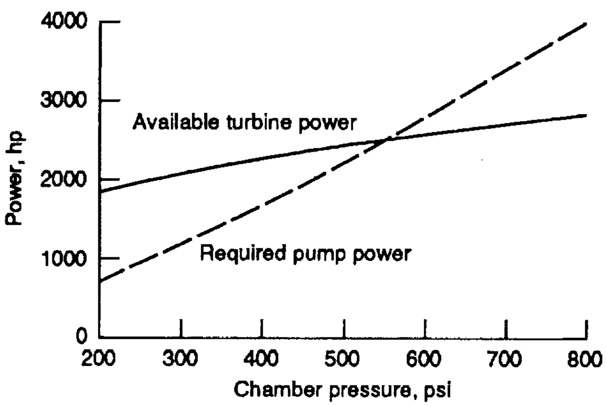


Figure 4.—Pump and turbine power as function of chamber pressure.

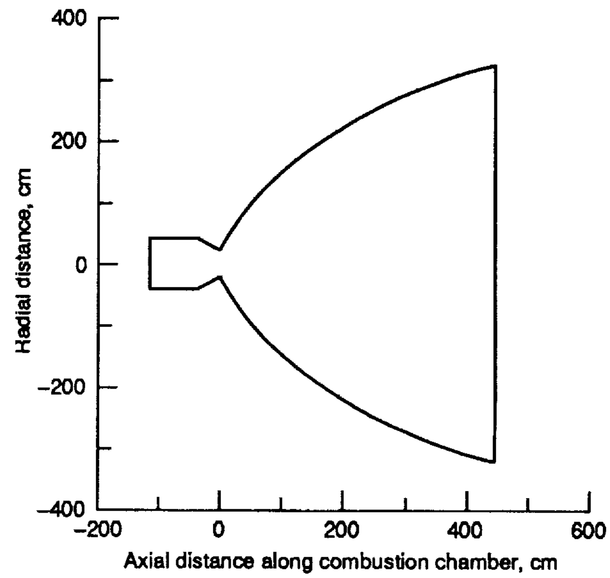


Figure 5.—Final thrust chamber contour.

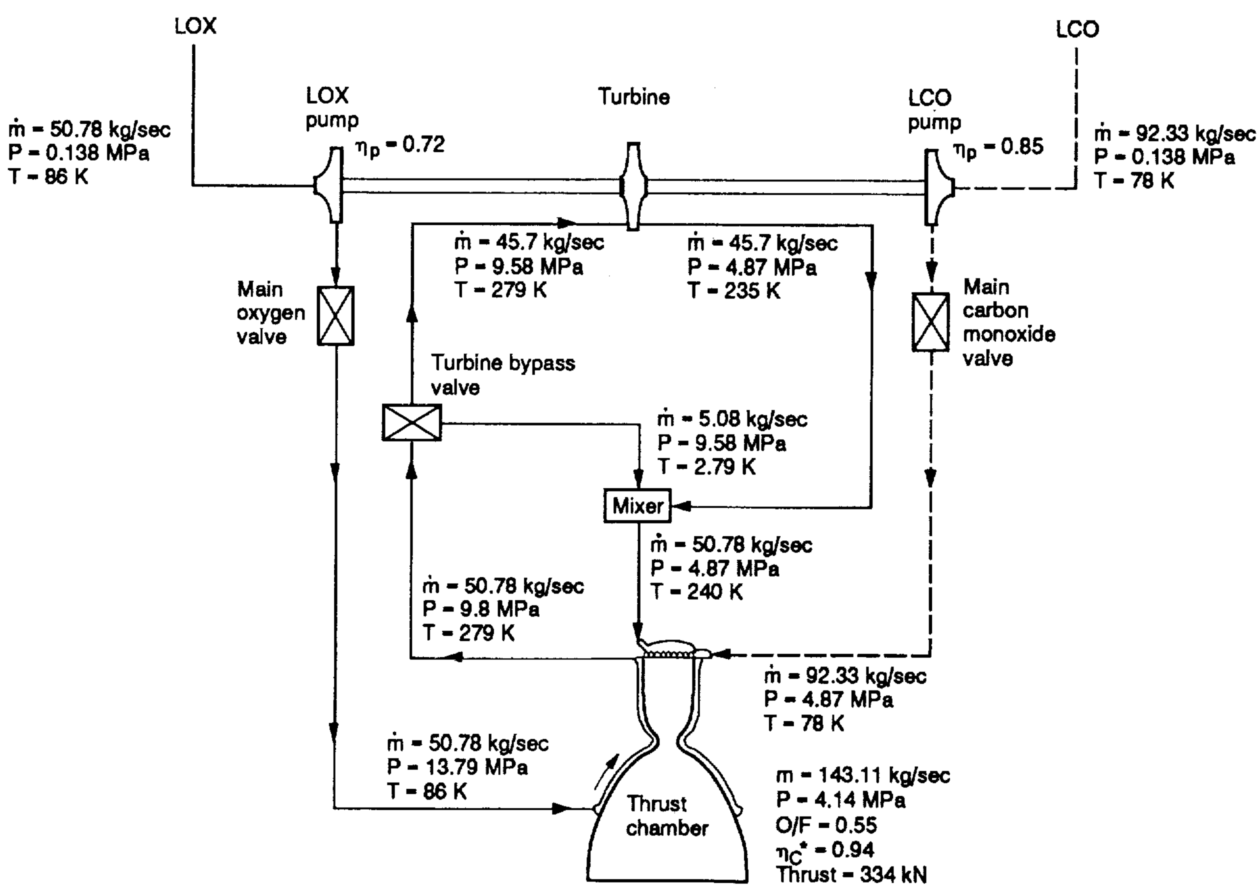


Figure 6.—LOX-LCO expander cycle with run conditions.

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16. Abstract

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