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Computational Method To Predict Thermodynamic, Transport, and Flow Properties for the Modified Langley 8-Foot High-Temperature Tunnel

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Abstract

The Langley 8-Foot High-Temperature Tunnel (8-ft HTT) is used to test components of hypersonic vehicles for aerothermal loads definition and structural component verification. The test medium of the 8-ft HTT is obtained by burning a mixture of methane-air under high pressure; the combustion products are expanded through an axisymmetric conical-contoured nozzle to simulate atmospheric flight at Mach 7. This facility has been modified to raise the oxygen content of the test medium to match that of air and to include Mach 4 and Mach 5 capabilities. These modifications will facilitate the testing of hypersonic air-breathing propulsion systems for a wide range of flight conditions. A computational method to predict the thermodynamic, transport, and flow properties of the equilibrium chemically reacting oxygen-enriched methane-air combustion products has been implemented in a computer code. This code calculates the fuel, air, and oxygen mass flow rates and test section flow properties for Mach 7, Mach 5, and Mach 4 nozzle configurations for given combustor and mixer conditions. Salient features of the 8-ft HTT are described, and some of the predicted tunnel operational characteristics are presented in the carpet plots to assist users in preparing test plans.

Introduction

The Langley 8-Foot High-Temperature Tunnel (8-ft HTT), which became operational in the 1960's, primarily has been used to define thermal and structural loads on large models at hypersonic speeds. Although many other hypersonic facilities were abandoned in the early 1970's, this tunnel is still operational and qualifies as a unique national facility. A resurgent interest exists in the development of air-breathing hypersonic vehicles that can fly at hypersonic speeds within the atmosphere. This interest is evident from the current development program of the National Aero-Space Plane (NASP), which is envisioned to be a hypersonic aircraft that can fly directly into orbit from a conventional runway (refs. 1 and 2).

To develop and improve advanced propulsion systems required for vehicles such as NASP, test facilities must simulate conditions that are encountered by the vehicle during its mission. Although the high-temperature test medium of the methane-air combustion products produced in the 8-ft HTT does not have the oxygen content of air, it has been useful for aerothermal loads definition and structural component verification. However, engine testing requires oxygen enrichment because most of the oxygen available in the air is used in the combustion of methane and the test medium cannot support further combustion in the engine. The need for large-engine test facilities to develop air-breathing engines for hypersonic flight prompted the modification of the 8-ft HTT. The modified facility can be operated in the oxygen-enriched combustion mode to produce 21 percent oxygen by volume in the test section or in the no-oxygen-enriched mode. The addition of Mach 4 and Mach 5 nozzle configurations will also complement the existing Mach 7 capability.

Figure 1 illustrates the operational envelopes, in terms of Mach number and pressure altitude, for the 8-ft HTT and two other facilities (the Aero Propulsion Test Unit (APTU) and the Aero Propulsion System Test Facility (ASTF) of the U.S. Air Force Arnold Engineering Development Center (AEDC)). Also superimposed around the facility operational envelope is a typical air-breathing engine operational envelope. The upper altitude limit is imposed by the ability of the engine to sustain combustion, and the lower altitude limit is imposed by the ability of the structure to survive the aerothermal loads. The inclusion of Mach 4 and Mach 5 capabilities enhances the testing envelope of the 8-ft HTT in the turbojet and ramjet operating ranges. However, the lower altitude limit of the facility with oxygen enrichment is restricted because the LOX (liquid oxygen) tank pressure limit is 2300 psia, which limits the combustor pressure to 2000 psia.

This report describes a computational method to predict test section flow properties at the nominal Mach numbers of 7, 5, and 4 for the 8-ft HTT. The flow is assumed to be in chemical equilibrium. Salient features of the 8-ft HTT are discussed to make the code requirements clear. The tunnel operational characteristics are presented for both the methane-air and methane-air-oxygen modes.

Symbols

A	nozzle throat cross-sectional area, ft^2
a	acoustic velocity, ft/sec
C_p	molar heat capacity of species, Btu/mole-°R
c_p	specific heat capacity of gaseous mixture, Btu/lbm-°R
FSA	flow survey apparatus
g	acceleration due to gravity, ft/sec^2

H	sensible enthalpy of species, Btu/mole
h	enthalpy of gaseous mixture, Btu/lbm
J	mechanical equivalent of heat, 778.26 ft-lbf/Btu
K	thermal conductivity, Btu/ft-sec-°R
$K_{p,j}$	equilibrium constant in terms of partial pressures (eqs. (A7) to (A12))
k	Boltzmann constant, Btu/molecule-°R
LOX	liquid oxygen
M	Mach number
m	mass flow rate, lbm/sec
MW	average molecular weight of gaseous mixture, lbm/mole, $1/\sigma$
N_{Pr}	Prandtl number
N_{Re}	unit Reynolds number, ft^{-1}
p	pressure, psia
PTC	combustor pressure, psia
PTM	mixer pressure, psia
Q	heat of formation, Btu/mole
q	dynamic pressure, psi
$\dot{q}_s$	stagnation point heat-transfer rate, Btu/ft^2-sec
R	universal gas constant, 1.9858 Btu/mole-°R
R_n	radius of spherically blunt body, 1 ft
R_{sp}	specific gas constant for mixture, R/MW, Btu/lbm-°R
S	entropy of species, Btu/mole-°R
s	entropy of gaseous mixture, Btu/lbm-°R
T	temperature, °R
TTC	combustor temperature, °R
TTM	mixer temperature, °R
V	velocity, ft/sec
x	proportion of oxygen in gaseous mixture by volume
y	moles of oxygen per mole of fuel
z	moles of air per mole of fuel
γ	ratio of specific heats
ε	convergence criteria
λ	Lennard-Jones force constant
μ	viscosity of gaseous mixture, lbm/ft-sec
ζ	collision diameter of molecule, Å
ρ	mass density of mixture, lbm/ft^3
σ	mole number of mixture, mole/lbm of mixture, $\Sigma\, \sigma_i$
$\sigma_{\mathrm{C}}, \sigma_{\mathrm{H}}, \sigma_{\mathrm{O}}, \sigma_{\mathrm{N}}$	moles of elements C, H, O, and N per lbm of mixture, mole/lbm
Ω	collision integral

Subscripts:

a	air
c	combustor air
e	effective
f	fuel
i	chemical species index; pre-combustion species
j	chemical reaction index; post-combustion species
m	mixer air
ox	molecular oxygen
r	reference
s	sum of fuel, total air, and oxygen
t	total
u, d	upstream and downstream sections of transpiration-cooled nozzle (fig. 8)
1,2	locations shown in figure 8

Superscript:

$'$	computed value

Test Facility

General Description

A schematic of the 8-ft HTT is given in figure 2. The tunnel is a large, hypersonic blowdown facility that simulates true-temperature flight at altitudes and Mach numbers shown in figure 1. The facility obtains its high-energy test medium by burning a mixture of methane and air under high pressure in the combustor. The combustion products are expanded to the test stream Mach number through an axisymmetric conical-contoured nozzle that has an exit diameter of 8 ft. The free-jet flow passes through the test chamber and enters a straight-tube supersonic diffuser. The flow then is pumped by a single-stage annular air ejector into a mixing tube and exhausted to the atmosphere through a subsonic diffuser.

A view of the tunnel test section from the nozzle exit during the entry of a model into the test stream is shown in figure 3. The model is stored in a pod beneath the test stream to protect it from tunnel start-up loads. Once the flow conditions are established, the model is inserted into the stream using a hydraulically actuated elevator. Model insertion time from the edge of the test core to the tunnel centerline is approximately 1 sec. An array of quartz lamps, located in the pod, is used to radiantly heat the models, if required, before and after exposure of the model to the test stream. A flow survey apparatus (FSA), which can obtain measurements of the test section flow properties and can survey the flow at various axial locations along the test section, is shown in its stowed position. This FSA is instrumented with a total of 37 probes, including 13 pitot, 11 static pressure, and 13 total-temperature probes.

The combustor operates at total temperatures ranging from 2400°R to 3600°R. For the Mach 7 nozzle configuration with no oxygen enrichment, the free-stream dynamic pressure ranges from 1.74 psi to 12.5 psi, and the free-stream unit Reynolds number ranges from 0.3×10^6 per ft to 2.2×10^6 per ft. The maximum run time of the tunnel is approximately 120 sec, and it is primarily restricted by the air storage capacity of a local bottle field that is the common source for the combustor and the ejector.

Figure 1 shows that the test conditions for the Mach 7 nozzle configuration are at the lower end of the scramjet range. To obtain test conditions for lower Mach numbers and pressure altitudes, the nozzle must be reconfigured. A dual-throat-mixer concept similar to that used to convert the AEDC Von Karman Gas Dynamics Facility from a Mach 10 tunnel to a Mach 4 true-temperature tunnel (ref. 3) was employed to reduce the cost by utilizing the maximum portion of the existing facility. (Note that the Mach 10 tunnel is not a true-temperature tunnel.) However, in the present facility, gas dynamics rather than mechanical means (ref. 3) are used to mix the hot combustion products with air. A similar concept has been used to provide increased capabilities for the Langley Arc-Heated Scramjet Test Facility (ref. 4).

The actual Mach number for the Mach 7 nozzle configuration varies from 5.8 to 7.3 (fig. 1). This Mach number variation is caused by the water condensation in the nozzle (ref. 5). Condensation is found to be maximum at low temperatures and high pressures and minimum at high temperatures and low pressures. In the future, the combustor temperature upper limit of 3600°R may be increased to 4000°R with an improved thermal liner for the combustor.

Oxygen Enrichment

The 8-ft HTT has been modified to include oxygen enrichment for facilitating scramjet engine testing. A schematic of the combustor, which shows the injection location and path of the LOX along with the fuel and air supply, is given in figure 4. High-pressure air from the 6000 psi bottle field is introduced at the upstream end of the combustor. The combustor consists of a carbon steel outer pressure vessel that is protected by a 316 Stainless-Steel outer liner and a 201 Nickel inner liner. The supply air passes downstream through the annulus between the pressure vessel and stainless-steel liner, and then it turns 180° upstream in the annulus between the outer and inner liners. The LOX enters the annulus between the outer and inner liners near the end of the LOX injection mixing ramp and mixes with the upstream air before it exits the annulus. The flow once again turns 180° and passes across the methane injectors where the oxygen-enriched air mixes with methane and burns. The combustion gas then flows through the nozzle throat and expands into the test section.

Because the LOX run tank pressure is limited to 2290 psia, the combustor pressure is limited to 2000 psia. Therefore, the facility that operates with oxygen enrichment is limited to a maximum dynamic pressure of approximately 1800 psf or minimum pressure altitudes of about 60 000 ft at $M = 4$, 78 000 ft at $M = 5$, and 90 000 ft at $M = 7$.

Air-Transpiration-Cooled Nozzle Throat

Prior to the facility modification, the nozzle throat was air-film cooled to prevent it from overheating because of the high-temperature combustion products. The water-cooled nozzle approach section, the throat, and a portion of the nozzle expansion section have been replaced with an air-transpiration-cooled section (fig. 5). This section is approximately 7.2 ft long and has a maximum internal diameter of

36 in. and a throat diameter of 5.6 in. A platelet concept has been used to fabricate this section in which the air passages are photoetched on thin sheets of varying thicknesses and bonded together. This air-transpiration-cooling approach is expected to reduce the coolant air mass flow rate by 50 percent of the amount that was previously used just to air-film cool the throat. This reduction in the coolant flow should produce a larger, uniform-temperature test section core. The coolant flows at the upstream and downstream locations of the nozzle throat are controlled such that they are proportional to the combustor pressure. However, the optimum values of the coolant flows will be determined experimentally so that the boundary layer does not separate from the nozzle wall but cools the nozzle effectively. Reference 6 gives more information concerning the design details of the air-transpiration-cooled section.

Alternate Mach Number

Schematics of the nozzle modifications to include Mach 4 and Mach 5 capabilities are shown in figure 6. As explained in the section entitled "General Description," a dual-throat-mixer concept has been selected to preclude modifications to the combustor and maintain the 8-ft nozzle exit diameter for Mach 4 or Mach 5 operation. For these Mach numbers, the removable section of the Mach 7 nozzle is replaced with a mixer and the Mach 4 or Mach 5 nozzle insert that joins the 18.4-ft-long fixed nozzle section as shown in figure 6. Air at ambient temperature is injected at various locations of the mixer to decrease the temperature and momentum of the hot gas and increase the mass flux in the mixer. The gas then will be expanded through the Mach 4 or Mach 5 nozzle to simulate the true temperature of flight at realistic altitudes. The gas temperature in the mixer is 1640°R and 2350°R for the Mach 4 and Mach 5 configurations, respectively.

Flow Computational Method

General Description of Code

The computational sequence of the computer code to calculate the thermodynamic, transport, and flow properties for the 8-ft HTT is described in the flowchart in figure 7. This code, called HTT, calculates the test stream flow properties and the mass flow rates of fuel, air, and oxygen at various locations of the 8-ft HTT for the Mach 7, Mach 5, and Mach 4 nozzle configurations (fig. 8). These mass flow rates will be monitored and controlled during the tunnel operation.

The combustion products are assumed to be in chemical equilibrium and to be composed of 4 elements (C, H, O, and N) and 10 reacting species (H_2O, CO_2, CO, O_2, H_2, N_2, H, O, OH, and NO) which obey the perfect gas law. First, the proportions of fuel, air, and oxygen are determined using an iterative procedure such that the combustion products attain the specified percentage of oxygen by volume and temperature at a given pressure. Note that the methane-air mode is calculated by setting the oxygen volume to the amount contained in air. Calculating the temperature of the combustion products at a specified pressure requires the mole numbers σ_C, σ_H, σ_O, and σ_N of the gas supply. These mole numbers are used to determine the chemical composition by solving a set of simultaneous equations which represent the chemical reaction and elemental mass balance. The thermodynamic properties of the mixture are then computed using thermodynamic equations and chemical tables. The temperature of the combustion products is calculated assuming that the enthalpy of postcombustion and precombustion products is constant for a given combustor condition. The proportion of the gas supply is adjusted until the calculated temperature converges with the specified temperature. The actual quantities of fuel, air, and oxygen are then calculated.

Next, the test stream flow properties are computed using either the FSA data or isentropic expansion procedure. In the first method, the values of test stream static pressure p_1 and pitot pressures $p_{t,2}$ are obtained from the FSA data. These values are used in the governing equations to obtain all the other flow properties. The flow properties for the Mach 7 nozzle configuration are computed using the FSA data, which are available from measurements obtained prior to the tunnel modification. Calculations for the Mach 4 and Mach 5 nozzle configurations are done assuming isentropic expansion because the FSA data are not presently available. For this method, p_1 and $p_{t,2}$ are calculated by first isentropically expanding the gas to the desired test stream Mach number and then solving the governing equations. Once the values for p_1 and $p_{t,2}$ are obtained, the calculation procedure for all the other test stream flow properties is the same for both these methods. Finally, the transport properties of the mixture are calculated.

Combustor Conditions

Gas mixture. The calculation method for the proportions of fuel, air, and oxygen enrichment to produce the combustion products at specified combustor pressure PTC and combustor temperature TTC with the desired volumetric proportion of oxygen x is illustrated by the initial portion of the computer flowchart in figure 7. The chemical equation for methane-air with oxygen enriched combustion is

$$CH_4 + z(0.21O_2 + 0.79N_2) + yO_2 = CO_2 + 2H_2O + (y + 0.21z - 2)O_2 + (0.79z)N_2 \quad (1)$$

This equation represents the chemical reaction for 1 mole of methane fuel with z moles of air and y moles of oxygen enrichment. The air is assumed to have a nitrogen to oxygen volumetric ratio of 79 to 21. The numerical coefficient of O_2, $y + 0.21z - 2$, in the right-hand side of the equation corresponds to the amount of oxygen present in the combustion products.

The proportions of oxygen and air in terms of methane fuel can be used to define the elemental composition of σ_C, σ_H, σ_O, and σ_N, respectively. For CH_4

$$\sigma_H/\sigma_C = 4 \quad (2)$$

The mole ratios of oxygen to carbon and nitrogen to carbon from the left-hand side of equation (1) are

$$\sigma_O/\sigma_C = 2(y + 0.21z) \quad (3)$$

and

$$\sigma_N/\sigma_C = 2(0.79z) \quad (4)$$

The four elemental constants of σ_C, σ_O, σ_H, and σ_N per unit mass of mixture are related by the following elemental mass balance equation:

$$12\sigma_C + \sigma_H + 16\sigma_O + 14\sigma_N = 1 \quad (5)$$

where the numerical coefficients of σ_C, σ_H, σ_O, and σ_N are the atomic weights of C, H, O, and N, respectively. An expression for σ_C is obtained by combining equations (2) to (4) with equation (5) as

$$\sigma_C = 1/(16 + 32y + 28.84z) \quad (6)$$

The proportion of each element supplied to the combustor is a function of moles of oxygen and air per mole of methane fuel. This proportion can be resolved by specifying the volumetric ratio x_c of the oxygen to the total of the products in the combustor defined by the following equation:

$$x_c = (y + 0.21z_c - 2)/(y + z_c + 1) \quad (7)$$

The numerator of this ratio is the coefficient of oxygen in the combustor which is contained in the third term of the right-hand side of the general equation (1) with z_c instead of z. The denominator is the total moles of the reacting combustor gas in the equation. In general, x and z are defined to describe the gas mixture that is produced in the test stream. The subscripted parameters x_c and z_c (fig. 7) correspond to the combustor gas that is directly expanded to the test stream for the Mach 7 case or the mixer for the Mach 4 and Mach 5 cases.

As indicated by the flowchart in figure 7, the gas composition supplied to the combustor is determined by specifying the required proportion of oxygen in the test section x (which is equal to x_c for the Mach 7 case) and by assuming an initial value for z_c to compute y from equation (7). The combustor elemental mole numbers are computed using the combustor air moles z_c instead of z in the general equations (2) to (4) and (6). The chemical composition of the gas is then computed as described in appendix A; 10 species and 6 chemical reactions (eqs. (A1) to (A6)) are considered, and the species are assumed to be in chemical equilibrium. The chemical composition is computed for a specified combustor pressure and temperature by simultaneously solving the set of equations (A7) to (A16) (see ref. 7 for details). The temperature of the combustion products is then computed as explained in appendix B. Iterations are made on z_c until the computed temperature of the combustion products converges on the specified combustor temperature (fig. 7). This iterative procedure will give the correct values of z_c and y and the corresponding chemical and thermodynamic properties of the combustor gas. For the methane-air case, no oxygen enrichment exists, and the computation is done with y set to zero.

Mass flow rates. The transpiration-cooled air is injected upstream and downstream of the Mach 7 nozzle throat, respectively. The coolant mass flow rates are controlled at the designed level (which is proportional to PTC) and are defined by the respective equations:

$$m_u = 0.021(\mathrm{PTC}) \quad (8)$$

$$m_d = 0.017(\mathrm{PTC}) \quad (9)$$

These correlations may be modified to include TTC based upon the future optimization analysis of the nozzle cooling performance.

The mass flow rates of combustor air m_c and oxygen m_{ox} per unit mass flow rate of fuel are determined from the computed values of z_c and y, respectively, by using the following equations:

$$m_c/m_f = z_c(\mathrm{MW}_a/\mathrm{MW}_f) \quad (10)$$

$$m_{ox}/m_f = y(MW_{ox}/MW_f) \qquad (11)$$

However, to compute the actual mass flow rates of each gas, another equation that relates these mass flow rates is developed assuming that the Mach 7 nozzle is choked.

First, a nozzle throat annular area that corresponds to a choked m_u (eq. (8)) is computed assuming a total pressure of PTC and a total temperature of 1360°R, which is the nozzle design inner surface temperature. Next, an effective throat area, which is the Mach 7 nozzle geometrical throat area minus the annular area that corresponds to the coolant flow m_u, is calculated. Finally, the choked mass flow rate through the effective area m_e is calculated using PTC and TTC of the combustor gases. This mass flow rate m_e is the sum of fuel, air, and oxygen which are supplied to the combustor, as indicated by the following expression:

$$m_e = m_f + m_c + m_{ox} \qquad (12)$$

The mass flow rates of m_f, m_c, and m_{ox} are calculated by solving equations (10) to (12).

Mixer Conditions

For Mach 4 and Mach 5 nozzle configurations, the mixer supplies additional air to achieve flight-temperature simulation and higher mass flow rates for low-altitude simulation. The airflow rate to the mixer m_m and pressure PTM are calculated to provide a specified mixer temperature TTM. The gas temperatures in the mixer for the Mach 4 and Mach 5 cases were assumed to be the design values of 1640°R and 2350°R (for flight-temperature simulation), respectively. These temperatures produce a test stream static temperature of approximately 400°R at the nozzle exit. However, the code can be run for a range of mixer temperatures.

The computation procedure for the mixer flow quantities (fig. 7) is similar to that explained in the section entitled "Mass Flow Rates" for computing the combustor flow quantities. For the combustor analysis, z_c moles are represented by m_c, but for the mixer, z_t moles of air are represented by the total airflow m_t, which is the sum of combustor air m_c, the transpiration-coolant air m_u and m_d, and the mixer air m_m. The total mass flow rate through the second throat is the sum of the fuel and the oxygen supplied in the combustor and the total airflow rate, which is

$$m_s = m_f + m_{ox} + m_t \qquad (13)$$

The resulting mixture pressure is computed, assuming the second throat is choked, as

$$\mathrm{PTM} = \left[m_s\sqrt{(\mathrm{TTM})}\right] \Big/ (0.532A) \qquad (14)$$

The value of z_t is adjusted, and the gas mixture composition and properties are computed until convergence on TTM is obtained utilizing the same procedure used for the combustor. The value of y for the mixer calculations corresponds to the LOX supplied to the combustor.

For normal operations in which the desired oxygen content of the test stream x is 0.21, the combustor gas produced at $x_c = 0.21$ is mixed with air (with the same oxygen content) so that $x' = x$. However, if a different oxygen content is desired, an additional iteration is required to adjust x_c, and the whole computational procedure is repeated using z_t instead of z_c in equation (7) until the desired oxygen content is obtained (fig. 7).

Test Stream Properties

As indicated by the latter portion of the computer flowchart in figure 7, the test stream properties are computed from data correlations measured using the tunnel FSA or assuming isentropic expansion of the combustion products to the test stream Mach number. When using FSA data, p_1 and $p_{t,2}$, respectively, are linearly correlated to TTC and PTC. In the latter case, p_1 and $p_{t,2}$ are calculated by isentropically expanding the gas to the desired Mach number and by solving the continuity, momentum, energy, and state equations. The other test stream properties, on both sides of the normal shock, are computed by assuming that the enthalpies $h_{t,1}$ and $h_{t,2}$ are equal and by solving the governing equations. The chemical composition, thermodynamic properties, and other stream properties are calculated by methods described in appendices A, B, and C, respectively.

Flow survey apparatus data or isentropic expansion. The test stream properties, which are computed using FSA measurements of p_1 and $p_{t,2}$ (fig. 8), include the losses in the tunnel. The measured $p_{t,2}$ and p_1, at a location 84 in. downstream of the Mach 7 nozzle exit, are correlated to TTC and PTC as

$$p_{t,2}/\mathrm{PTC} = (1.2856 \times 10^{-6})\mathrm{TTC} + 0.00277 \qquad (15)$$

and

$$p_1/p_{t,2} = (-4.00 \times 10^{-6})\mathrm{TTC} + 0.03080 \qquad (16)$$

These correlations for the Mach 7 nozzle configuration, which are from FSA data that were obtained before the tunnel modification, will be corrected, if necessary, once FSA data from the modified tunnel are available.

In the case of the isentropic expansion procedure, p_1 and $p_{t,2}$ are calculated by isentropically expanding the combustion products to the required test stream Mach number and by solving the mass, momentum, energy equations, and state equations. Because of the unavailability of FSA data for Mach 5 and Mach 4 nozzle configurations, the isentropic expansion procedure is used to compute the values of p_1 and $p_{t,2}$. However, when FSA data become available, the correlation for p_1 and $p_{t,2}$ can be expressed in terms of TTM and PTM, similar to equations (15) and (16), and can be incorporated into the code. From these computed p_1 and $p_{t,2}$ values, the calculation method for all the flow properties upstream and downstream of the shock is described in the following sections.

Computation of flow properties. The value of $p_{t,2}$ is obtained from equation (15) for given combustor conditions. The temperature of the combustion products $T_{t,2}$ at $p_{t,2}$ is calculated as explained in appendix B. The gas density at the stagnation conditions of $p_{t,2}$ and $T_{t,2}$ is computed from the state equation. The gas is isentropically expanded from $p_{t,2}$ and $T_{t,2}$ to p_2 at an initially assumed value of T_2 (fig. 7), and all the gas properties at p_2 and T_2 are determined as explained subsequently.

First, a new static pressure p_2' is calculated as

$$p_2' = p_2 - dp$$

where the incremental pressure dp is calculated from

$$dp = -p_2(s_2 - s_{t,2})/(R\sigma)$$

The term p_2' is substituted for p_2, and the iteration is continued until dp satisfies a convergence criteria. The gas composition and the thermodynamic properties are calculated at p_2 and T_2, and the flow properties ρ_2 and V_2 are calculated from

$$\rho_2 = p_2/R\sigma T_2 \tag{17}$$

and

$$V_2 = [2J(h_{t,2} - h_2)]^{1/2} \tag{18}$$

Next, V_1 and ρ_1, which are obtained by solving the mass and momentum equations across a normal shock, are given by

$$V_1 = [(p_2 - p_1)/\rho_2 V_2] + V_2 \tag{19}$$

and

$$\rho_1 = \rho_2(V_2/V_1) \tag{20}$$

The value of T_1 is calculated from the state equation using the values p_1 and ρ_1 obtained from equations (16) and (20), respectively, and an initial assumption for σ. The chemical composition and a new value of σ are computed at p_1 and T_1, and T_1 is updated until it converges on σ. The enthalpy of gaseous mixture h_1 at p_1 and T_1 is determined, and a differential enthalpy is calculated as

$$dh = h_{t,2} - \left(h_1 + 0.5V_1^2/J\right)$$

The initially assumed T_2 in the outermost loop of this subroutine is adjusted, and the entire computation is continued until the value of dh is less than a convergence criteria ε (fig. 7). The chemical composition, thermodynamic properties, and transport properties for the converged solution are calculated as described in appendices A, B, and C, respectively.

The Mach number, unit Reynolds number, dynamic pressure, and stagnation heat-transfer rate are calculated. The Mach number is defined as

$$M = V/a \tag{21}$$

where $a = \sqrt{g(dp/d\rho)}$. In this equation, dp is calculated as the difference of static pressures corresponding to two temperatures of $T_t - dT$ and $T_t + dT$ at p_t. The value for dT is arbitrarily chosen to be 1°R. The difference in the densities $d\rho$, corresponding to dp for this change in T_t, is determined from the calculated chemical composition and the state equation.

The unit Reynolds number is calculated from

$$N_{\mathrm{Re}} = \rho V/\mu \tag{22}$$

The dynamic pressure is calculated from

$$q = (0.5\rho V^2)/144 \tag{23}$$

The stagnation heat-transfer rate $\dot{q}_s$ for a spherically blunted body of a unit radius of 1 ft is calculated using the Fay and Riddell equation (ref. 8). The quantities of c_p, h, s, ρ, and μ, which are required for the calculation of $\dot{q}_s$, are computed for a wall temperature of 540°R.

Results and Discussion

Mach 7 Configuration

The carpet plots of the flow properties for the 21-percent oxygen enrichment and no-enrichment (no-LOX) cases are shown in figure 9 for the Mach 7 nozzle configuration. The combustor operating ranges for PTC and TTC are 600 psia to 3500 psia and 2500°R to 4000°R, respectively. However, for the oxygen enrichment case (shown by the dashed lines), PTC is limited to 2000 psia because of the oxygen run tank pressure limit. In the cross plots of p_1 versus T_1 (fig. 9(a)), T_1 is slightly higher for the oxygen enrichment case compared with the no-LOX case. This difference increases with increasing TTC. The variation of T_1 with combustor pressure is insignificant for both the LOX and no-LOX cases. Note from equation (16) that p_1 is assumed to be the same for both the LOX and no-LOX cases, and it will be updated, if necessary, once the tunnel survey measurements are available. The difference in the properties of V_1, ρ_1, M_1, N_{Re}, $\dot{q}_s$, and q_1, shown in figures 9(b) to 9(d), for the LOX and no-LOX cases is very small. The Mach number variation of approximately 6.1 to 7.2 in figure 9(c) is because of water condensation. This variation is accounted for by the FSA measurements in the test section.

The test stream gas composition, in terms of mole fraction σ_i/σ, is given in table I for both the LOX and no-LOX cases for the 21-percent oxygen enrichment. Note that only the major species of H_2O, CO_2, O_2, and N_2 are listed in this table. The calculated free-stream static temperature is less than the chosen reference temperature of $T_r = 536.67$°R, and the minor species are not considered for temperatures below T_r (appendix A). The gas composition is listed for the combustor design temperature of TTC = 3560°R. The composition is affected by temperature, but it does not show significant change within the operating range of the combustor pressure of 600 psia to 3500 psia. The calculated value of the test stream gas constant R_{sp} at this design temperature, for both the LOX and no-LOX cases for the 21-percent oxygen enrichment, is 0.070 Btu/lbm-°R, and the corresponding values of the ratio of specific heats γ are 1.38.

Mach 5 and Mach 4 Configurations

The carpet plots for the Mach 5 and Mach 4 nozzle configurations for a combustor pressure range of 600 psia to 3500 psia and combustor temperature of 3560°R are shown in figures 10 and 11, respectively. The mixer design temperatures are 2350°R and 1640°R for the Mach 5 and Mach 4 nozzle configurations, respectively. The plots are shown for the mixer design temperatures and for a few more mixer temperatures. A very small variation exists in the flow properties between the LOX and no-LOX cases. The variation observed in the plot p_1 versus T_1 for the Mach 7 case (fig. 9(a)) is not seen in similar plots for the Mach 5 and Mach 4 cases (figs. 10(a) and 11(a)), which is mainly because of the dominant effect of air added in the mixer.

Figures 12 and 13 show the test stream properties in terms of PTM and TTM for the Mach 5 and Mach 4 cases, respectively. The plots in these figures can be used to estimate the test conditions from the mixer conditions alone. Correlations of combustor and mixer conditions for the Mach 5 and Mach 4 cases are shown in figures 14 and 15, respectively. Note that no significant variation exists in these plots between the LOX and no-LOX cases. Some of the tunnel operational range and restrictions can be understood from these plots. For example, at TTM = 2350°R (the temperature required for true-temperature flight simulation) and PTM = 300 psia, which is shown by the bold line in figure 14(b), the combustor can be operated at different total temperatures. However, for this mixer condition and for TTC = 3000°R, the combustor pressure is above 2000 psia (fig. 14(a)). The combustor cannot be operated at this temperature with the LOX because the LOX run tank pressure upper limit is 2000 psia. Figure 14(b) also shows that for the Mach 5 case at TTM = 2350°R, the combustor cannot be operated at TTC = 2500°R because the transpiration coolant flows (m_u and m_d in fig. 8) cool the combustor gas slightly below TTM = 2350°R. As a result, the code computes negative values of the mixer airflow m_m, as shown in the plot in figure 14(b). However, the mixer will require air addition for cooling.

Typical test stream gas composition for the Mach 5 and Mach 4 cases is given in table I at the mixer design temperatures. The gas composition is not affected by PTC and TTC, but it changes with TTM. Similar to the Mach 7 case, the minor species are set to zero for the Mach 5 and Mach 4 cases because the test stream static temperature is below the reference temperature of $T_r = 536.67$°R. The values of R_{sp} are 0.070 Btu/lbm-°R and 0.069 Btu/lbm-°R, for the Mach 5 and Mach 4 nozzle configurations, respectively, for both the LOX and no-LOX cases. The corresponding value of ratio of specific heats γ is 1.39.

Concluding Remarks

A computer program, HTT, has been developed which calculates the thermodynamic, transport, and flow properties of equilibrium chemically reacting oxygen-enriched methane-air combustion products. This program is tailored to compute the various test stream flow properties and the mass flow requirements of the Langley 8-Foot High-Temperature

Tunnel for the nominal Mach numbers of 7, 5, and 4 with and without oxygen enrichment. However, this computer program can be applied with only minor program modifications to other facilities that use methane-air-oxygen combustion products as the test medium. The option to compute the test section flow using flow survey data or an isentropic expansion procedure is available in this code. The flow properties and operational characteristics for the expected operating envelope of the modified tunnel are presented in the carpet plots to assist tunnel users in preparing test plans.

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May 20, 1992

Appendix A

Chemical Composition

The equilibrium chemical composition of the oxygen-methane-air combustion products is considered the unique function of partial pressures p_i and temperature T. These combustion products are thought to be thermally perfect and composed of 4 elements, namely C, H, O, and N, and 10 chemically reacting species numbered from 1 to 10 in the following order: H_2O, CO_2, CO, O_2, H_2, N_2, H, O, OH, and NO. The six independent chemical reactions are expressed as

$$CO + H_2O = CO_2 + H_2 \tag{A1}$$

$$2CO_2 = 2CO + O_2 \tag{A2}$$

$$H_2 + O_2 = 2OH \tag{A3}$$

$$H_2 = 2H \tag{A4}$$

$$O_2 = 2O \tag{A5}$$

$$O_2 + N_2 = 2NO \tag{A6}$$

These reactions are represented, respectively, in terms of equilibrium constants $K_{p,j}$ and mole numbers σ_i as

$$K_{p,1} = \sigma_2\sigma_5/\sigma_1\sigma_3 \tag{A7}$$

$$K_{p,2} = \sigma_3^2\sigma_4/\sigma_2^2 \tag{A8}$$

$$K_{p,3} = \sigma_9^2/\sigma_4\sigma_5 \tag{A9}$$

$$K_{p,4} = \sigma_7^2/\sigma_5 \tag{A10}$$

$$K_{p,5} = \sigma_8^2/\sigma_4 \tag{A11}$$

$$K_{p,6} = \sigma_{10}^2/\sigma_4\sigma_6 \tag{A12}$$

The expression for $K_{p,j}$ is given by

$$\log\ K_{p,j} = B_1/T + B_2 + B_3T + B_4T^2 + B_5T^3$$

where the constants B_1 to B_5 are incorporated in the code; these constants were obtained from reference 7 for the temperature ranges from 360°R to 1800°R, 1800°R to 5400°R, and 5400°R to 10 800°R. The units of $K_{p,j}$ for reactions expressed by equations (A1), (A3), and (A6) are nondimensional, and those for equations (A2), (A4), and (A5) are given in atmospheres.

The four elemental balance equations are

$$\sigma_H = 2\sigma_1 + 2\sigma_5 + \sigma_7 + \sigma_9 \tag{A13}$$

$$\sigma_O = \sigma_1 + 2\sigma_2 + \sigma_3 + 2\sigma_4 + \sigma_8 + \sigma_9 + \sigma_{10} \tag{A14}$$

$$\sigma_N = 2\sigma_6 + \sigma_{10} \tag{A15}$$

$$\sigma_C = \sigma_2 + \sigma_3 \tag{A16}$$

The composition is calculated by solving the six independent chemical equilibrium equations (eqs. (A7) to (A12)), which simultaneously represent the chemical reactions (eqs. (A1) to (A6)) and four elemental balance equations (eqs. (A13) to (A16)).

For temperatures below $T_r = 536.67$°R, only the major species of σ_1, σ_2, σ_4, and σ_6, which correspond to H_2O, CO_2, O_2, and N_2, respectively, are assumed to be present in the mixture. The remaining minor species are set to zero in equations (A13) to (A16) to give

$$\sigma_H = 2\sigma_1 \tag{A17}$$

$$\sigma_O = \sigma_1 + 2\sigma_2 + 2\sigma_4 \tag{A18}$$

$$\sigma_N = 2\sigma_6 \tag{A19}$$

$$\sigma_C = \sigma_2 \tag{A20}$$

The major species are directly calculated from these equations once the elemental composition is known.

Appendix B

Thermodynamic Properties and Temperature

Thermodynamic Properties

The thermodynamic properties, namely specific heat c_p at constant pressure, enthalpy h, and entropy s of the mixture at a given temperature and pressure, are calculated.

The molar heat capacities of the species C_p at constant pressure are given in terms of temperature as

$$C_{p,i} = A_1 + A_2T + A_3T^2 + A_4T^3 + A_5T^4 \quad \text{(B1)}$$

The C_p values for the species are obtained from reference 9 for four different temperature ranges from 180°R to 720°R, 720°R to 1800°R, 1800°R to 5400°R, and 5400°R to 10 800°R. These values have been fitted with a fourth-degree polynomial to give the coefficients A_1 to A_5 of equation (B1).

The value of c_p is calculated using the equation

$$c_p = \Sigma C_{p,i}\sigma_i \quad \text{(B2)}$$

The enthalpy of the species H_i, minus the reference enthalpy $H_{i,r}$ at $T_r = 536.67$°R, is

$$H_i - H_{i,r} = \int_{T_r}^{T} C_{p,i}\, dT \quad \text{(B3)}$$

The enthalpy of the gaseous mixture h is computed using the equation

$$h = \Sigma\sigma_i \left[(H_i - H_{i,r}) + Q_{i,r}\right] \quad \text{(B4)}$$

where the values of $Q_{i,r}$ (the heat of formation of the pure species) are referenced to $T_r = 536.67$°R and are obtained from reference 9 and incorporated into the code. The entropy of the species S is given by

$$S_i = \left[\int_{T_r}^{T} C_{p,i}\, dT/T + S_{i,r} - R\ln(p_i/p_r)\right] \quad \text{(B5)}$$

where $S_{i,r}$ is the reference entropy of the species at $T_r = 536.67$°R and $p_r = 1$ atmosphere, which are obtained from reference 9 and are incorporated into the code. The value of h is calculated from

$$s = \Sigma\sigma_i S_i \quad \text{(B6)}$$

Temperature of Combustion Products

The temperature of the combustion products is defined at a given pressure for a given fuel and oxidizer and at an initial temperature of the precombustion species. An initially assumed temperature is iterated until convergence is obtained by using the following equation:

$$\begin{aligned}\Sigma\sigma_i &\left[(H_i - H_{i,r}) + Q_{i,r}\right] \\ &= \Sigma\sigma_j \left[(H_j - H_{j,r}) + Q_{j,r}\right] \quad \text{(B7)}\end{aligned}$$

The right-hand side refers to the precombustion species and the left-hand side refers to the combustion gas mixture. The initial temperature of the precombustion species is chosen to be 536.67°R.

Appendix C

Transport Properties

The viscosities of all of the species are calculated using the following equation (ref. 10):

$$\mu_i = 1/1.4882\left[(26.69)\sqrt{(\mathrm{MW}_i T)}\right] \Big/ (\zeta^2\Omega) \quad \text{(C1)}$$

The numerical value of 1/1.4882 is the conversion factor from kg/m-sec to lbm-ft-sec. Values of the collision integral Ω are obtained from reference 10, in which Ω is tabulated as a function of kT/λ. Values of ζ and λ/k for the species are also obtained from reference 10. The viscosity of the gaseous mixture μ is calculated using the following equation (refs. 10 and 11):

$$\mu = \mu_i/\Sigma\phi_{i,j}\sigma_j/\sigma_i \quad \text{(C2)}$$

where

$$\phi_{i,j} = \frac{\left[1 + (\mu_i/\mu_j)^{1/2}(\mathrm{MW}_j/\mathrm{MW}_i)^{1/4}\right]^2}{\left\{2\sqrt{2}[1 + (\mathrm{MW}_i/\mathrm{MW}_j)]^{1/2}\right\}} \quad \text{(C3)}$$

The thermal conductivity K is given by the Eucken equation (refs. 10 and 11) as

$$K = [c_p + (5/4)R\sigma]\mu \quad \text{(C4)}$$

The Prandtl number N_{Pr} is calculated from

$$N_{\mathrm{Pr}} = \mu c_p/K \quad \text{(C5)}$$

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Table I. Typical Test Stream Gas Composition in Terms of Mole Fraction σ_i/σ

Conditions	Mole fraction of species			
	H_2O	CO_2	O_2	N_2
Mach 7 nozzle configuration; TTC = 3560°R				
No LOX	0.15200	0.07599	0.04206	0.73000
LOX	.15420	.07710	.21000	.55870
Mach 5 nozzle configuration; TTM = 2350°R				
No LOX	0.08314	0.04157	0.11810	0.75720
LOX	.08358	.04179	.21000	.66460
Mach 4 nozzle configuration; TTM = 1640°R				
No LOX	0.04778	0.02389	0.15720	0.77110
LOX	.04792	.02396	.21000	.71810

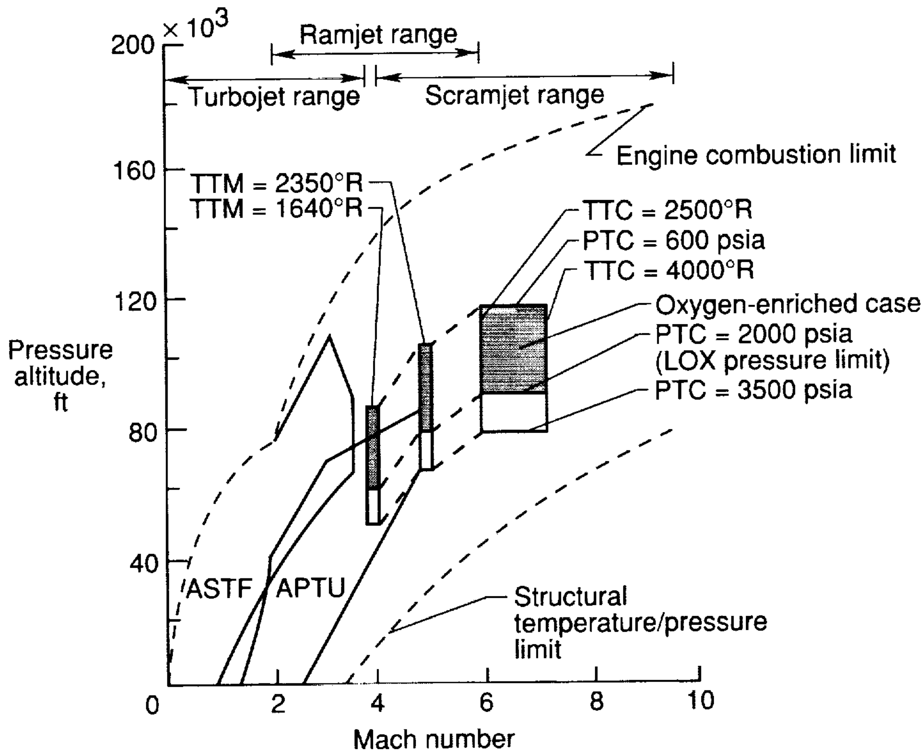

Figure 1. Operational envelopes for propulsion facilities.

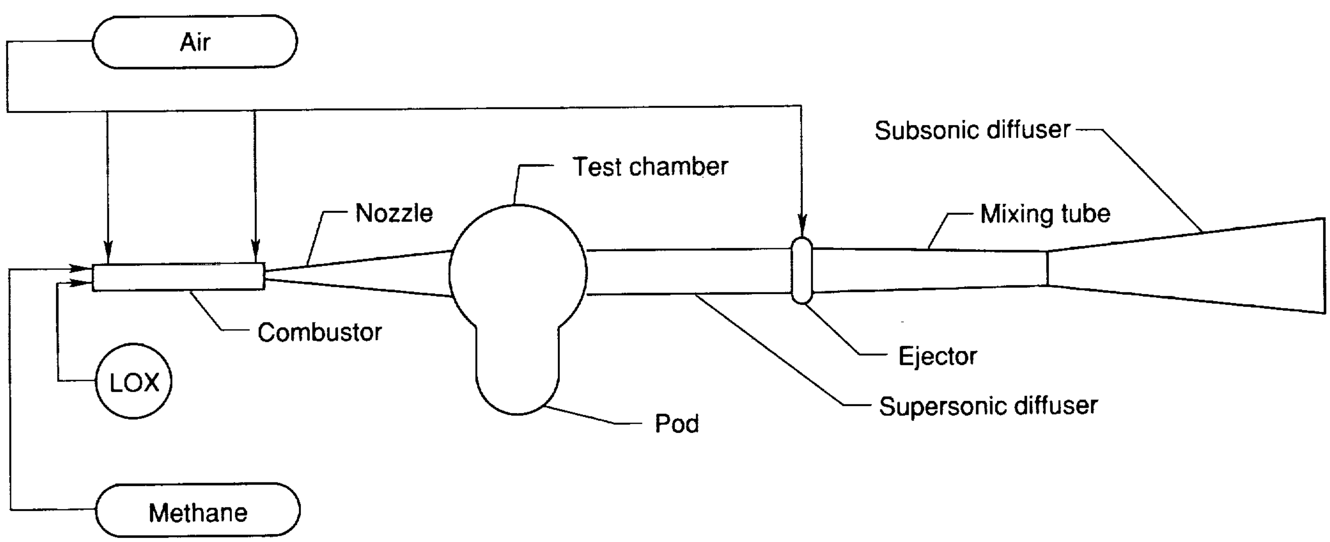

Figure 2. Schematic of Langley 8-Foot High-Temperature Tunnel.

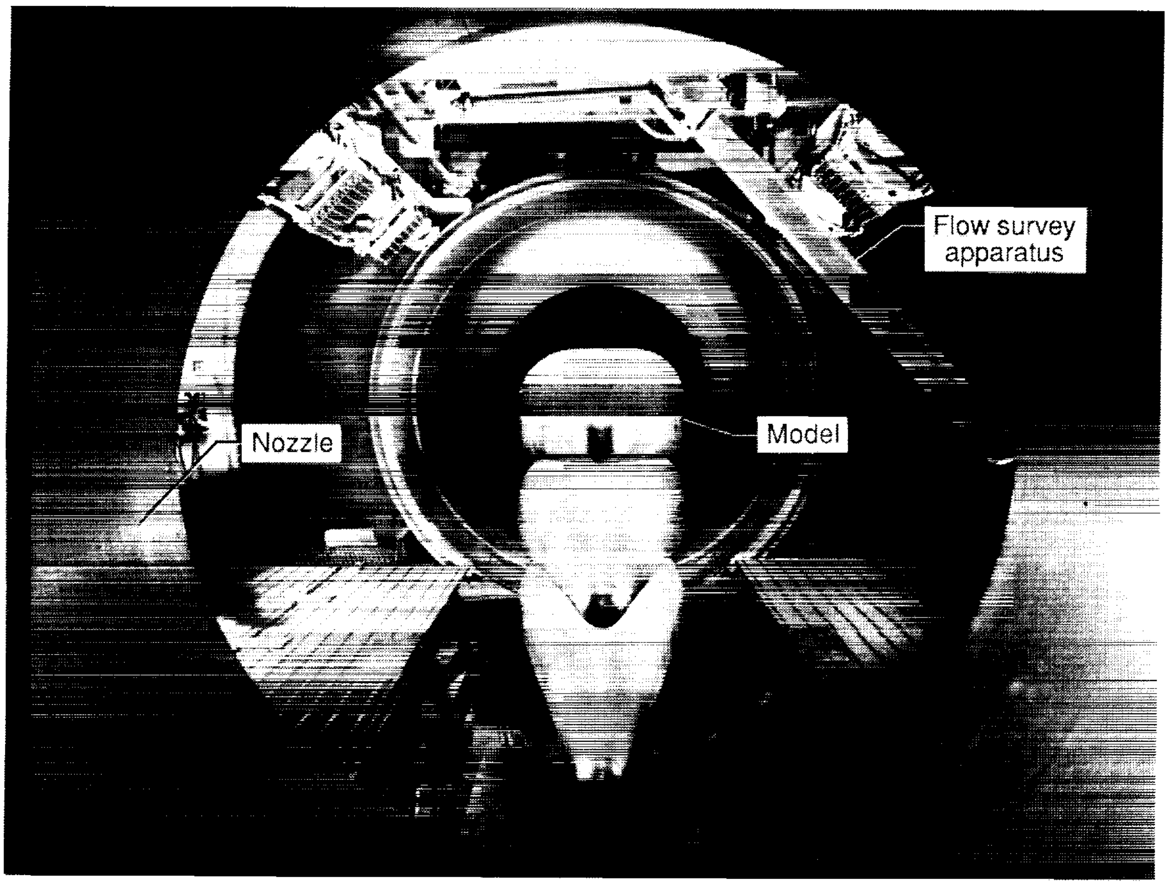

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Figure 3. Triple exposure of model during its entry into test section with flow survey apparatus in stowed position.

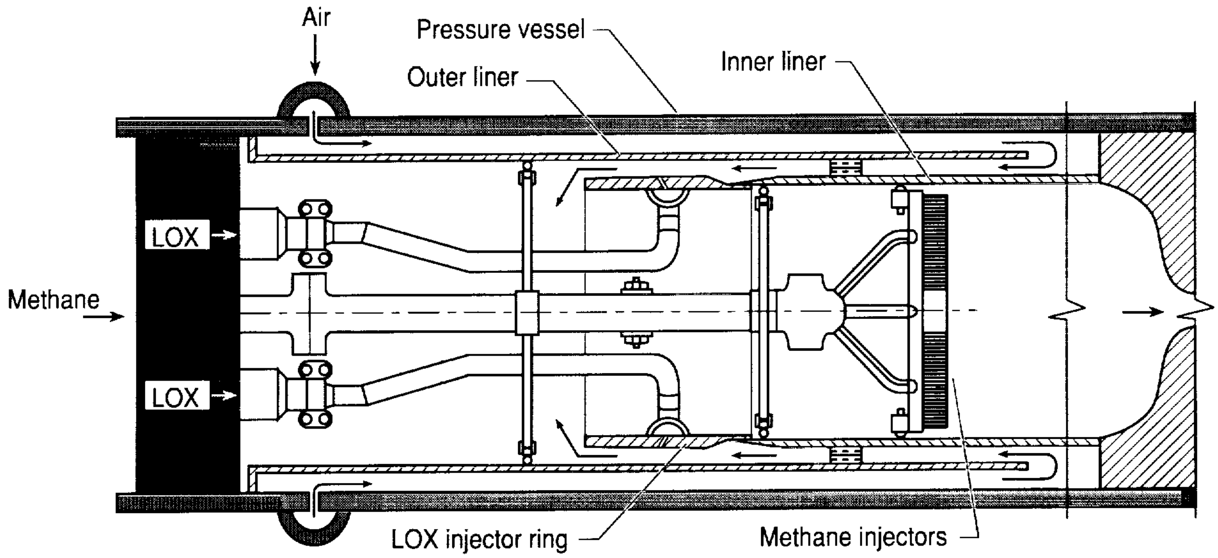


Figure 4. Langley 8-Foot High-Temperature Tunnel combustor. (Schematic is not drawn to scale; horizontal distances have been shortened.)

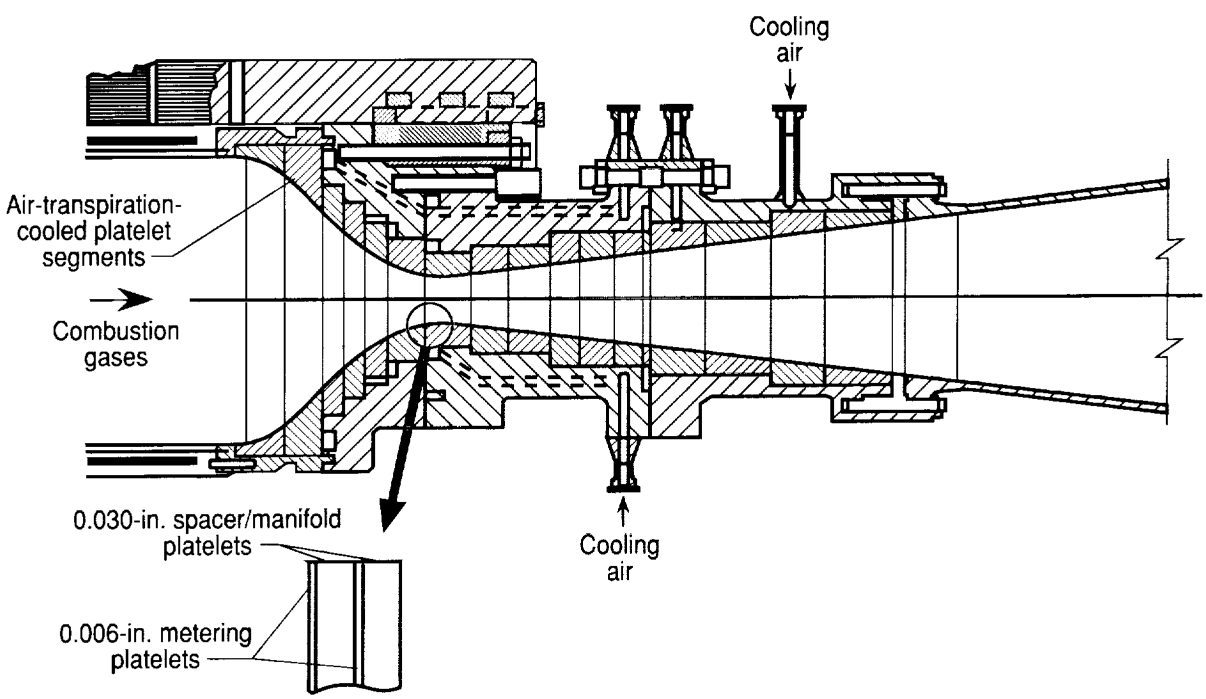
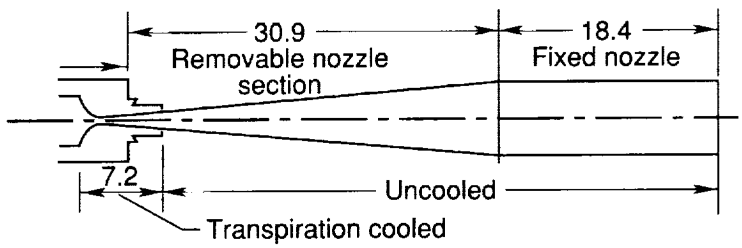
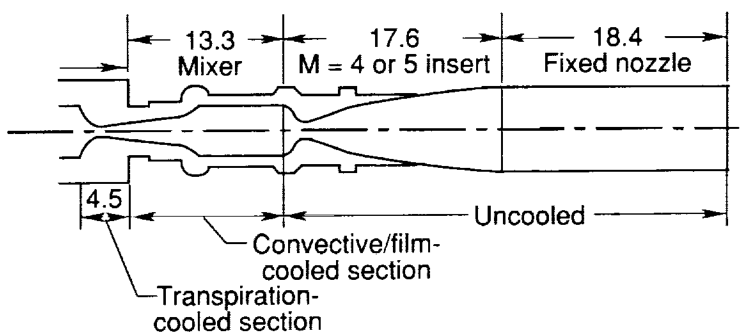


Figure 5. Air-transpiration-cooled nozzle section.



(a) Mach 7 configuration.



(b) Mach 4 and Mach 5 configurations.

Figure 6. Modified nozzles for alternate Mach number capability. All linear dimensions are given in feet.

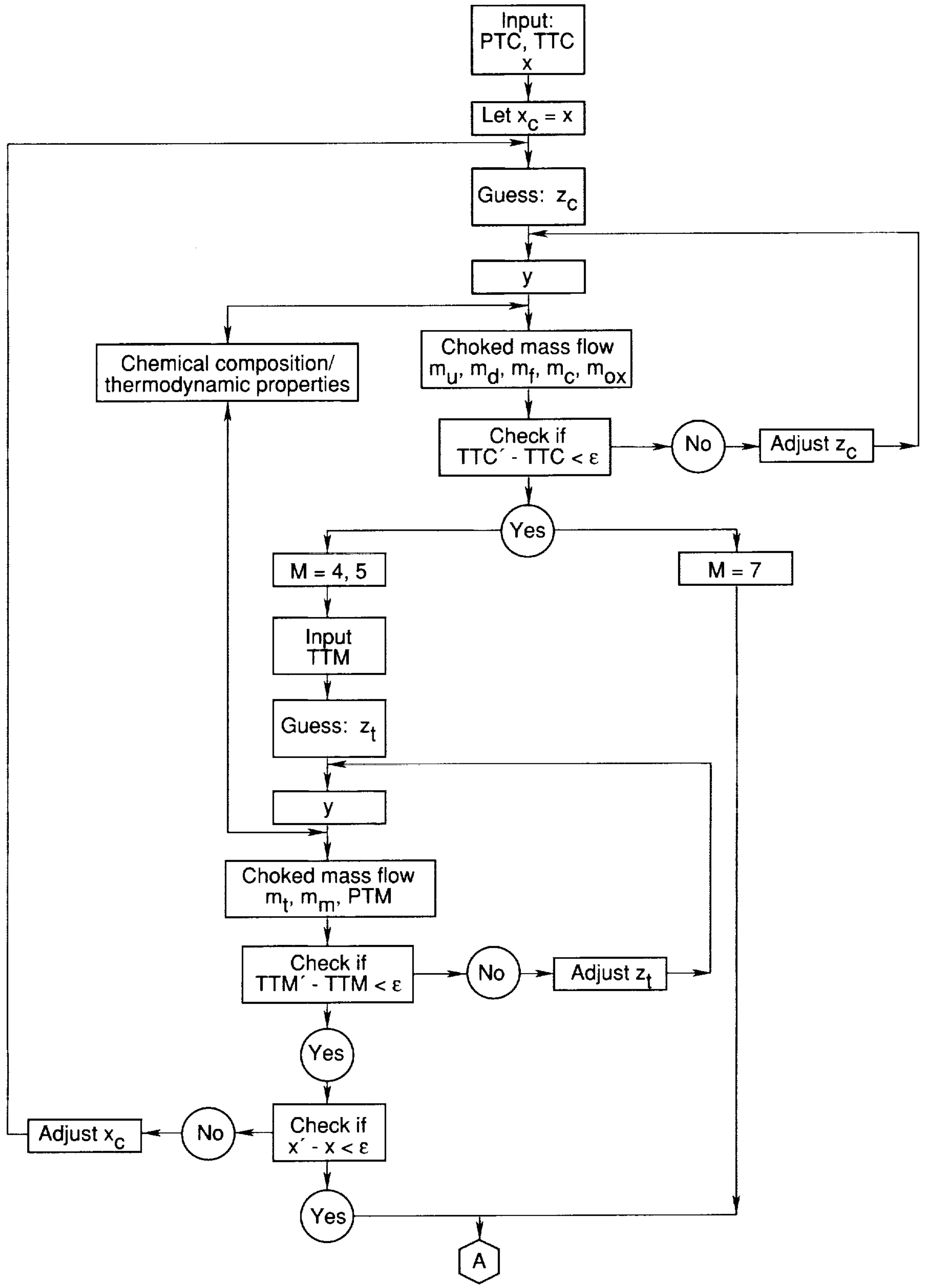


Figure 7. Computational sequence of HTT code with oxygen enrichment.

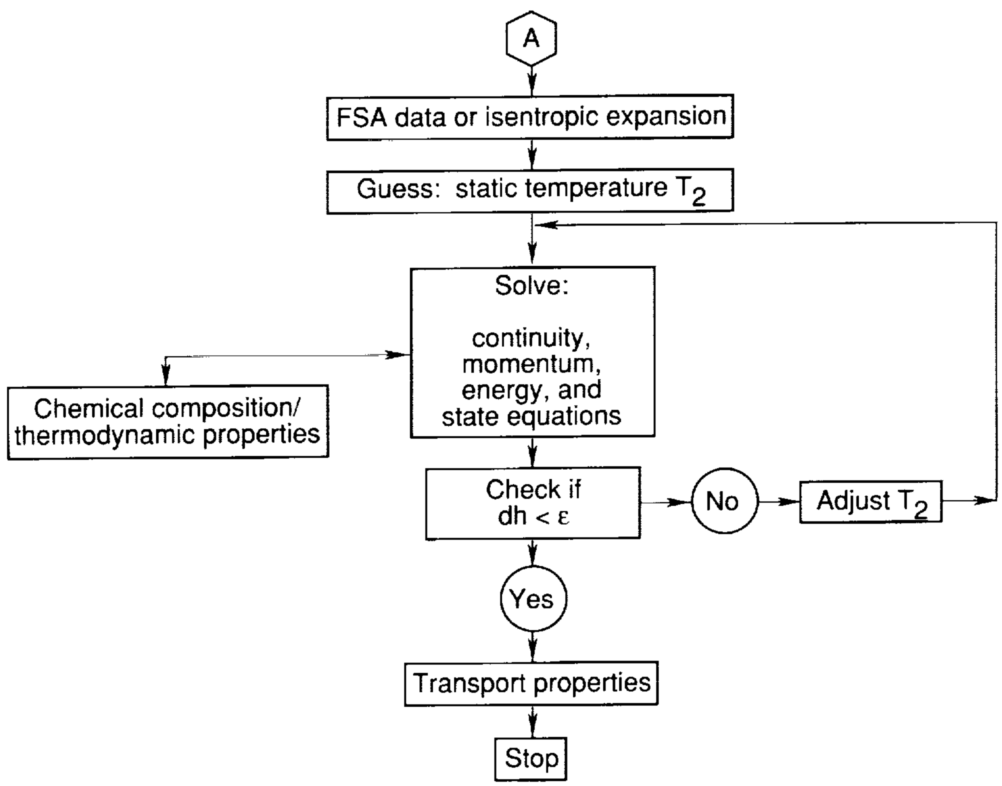


Figure 7. Concluded.

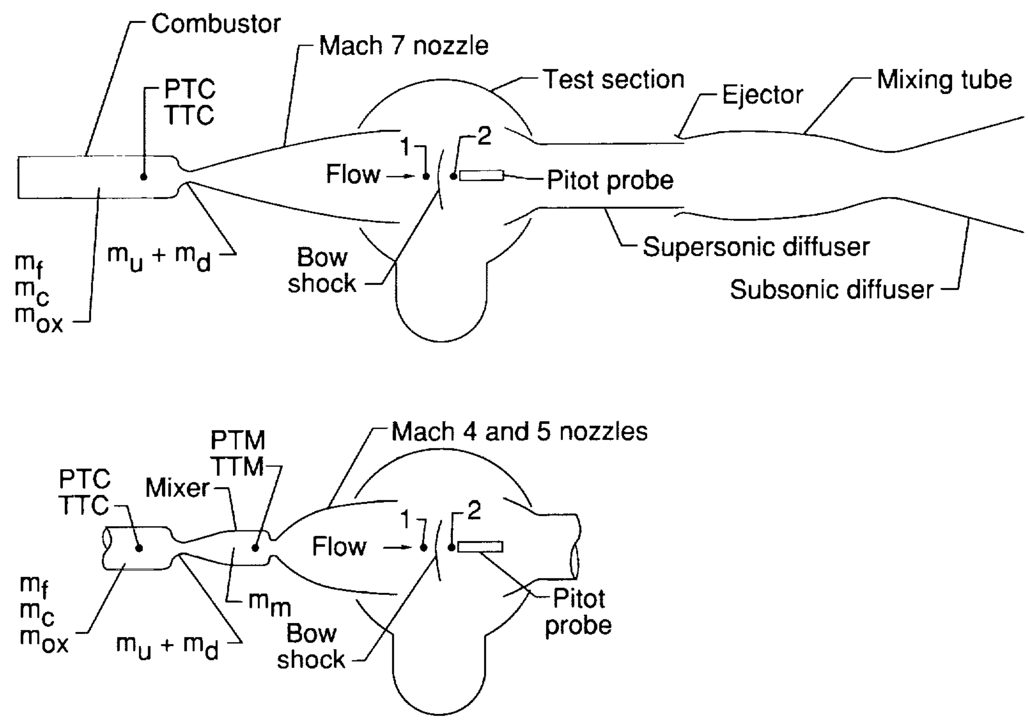
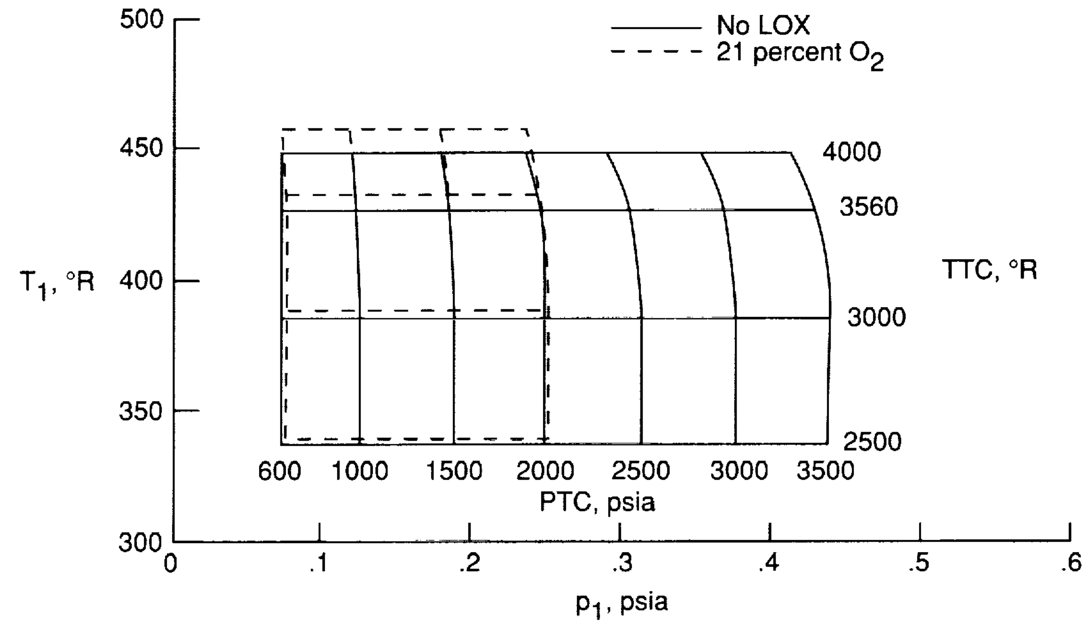
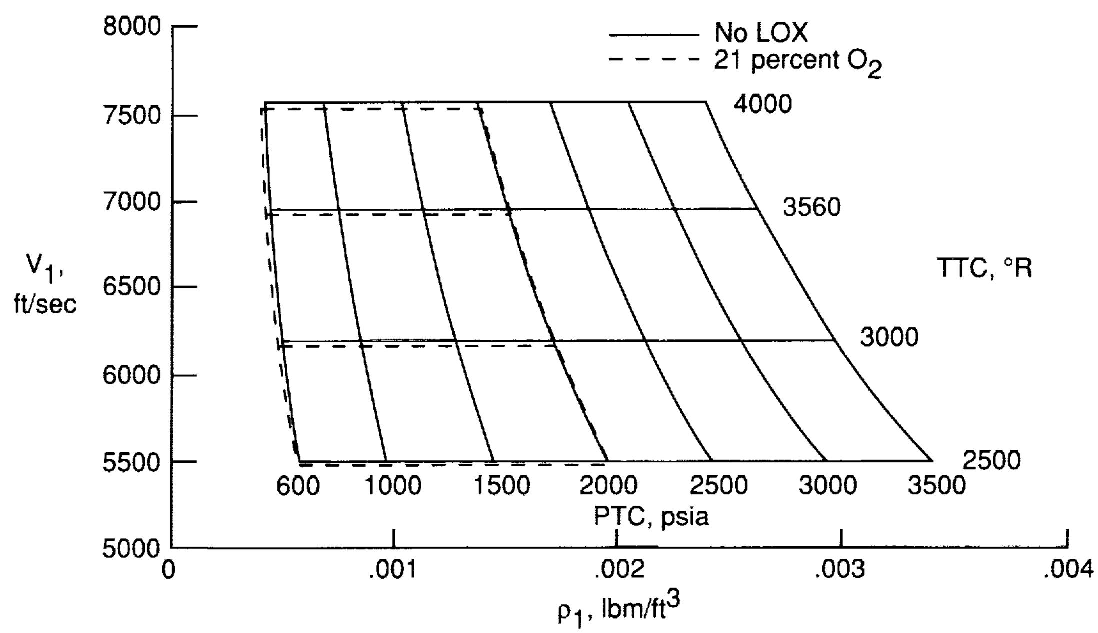


Figure 8. Locations of flow variables for flow computations.

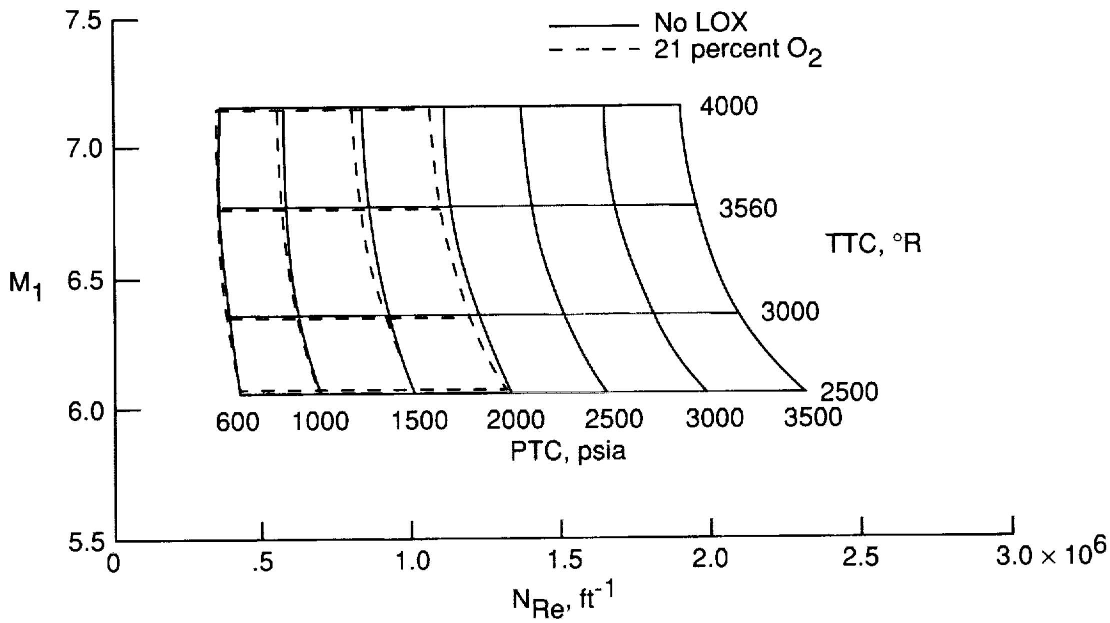


(a) Pressure and temperature.

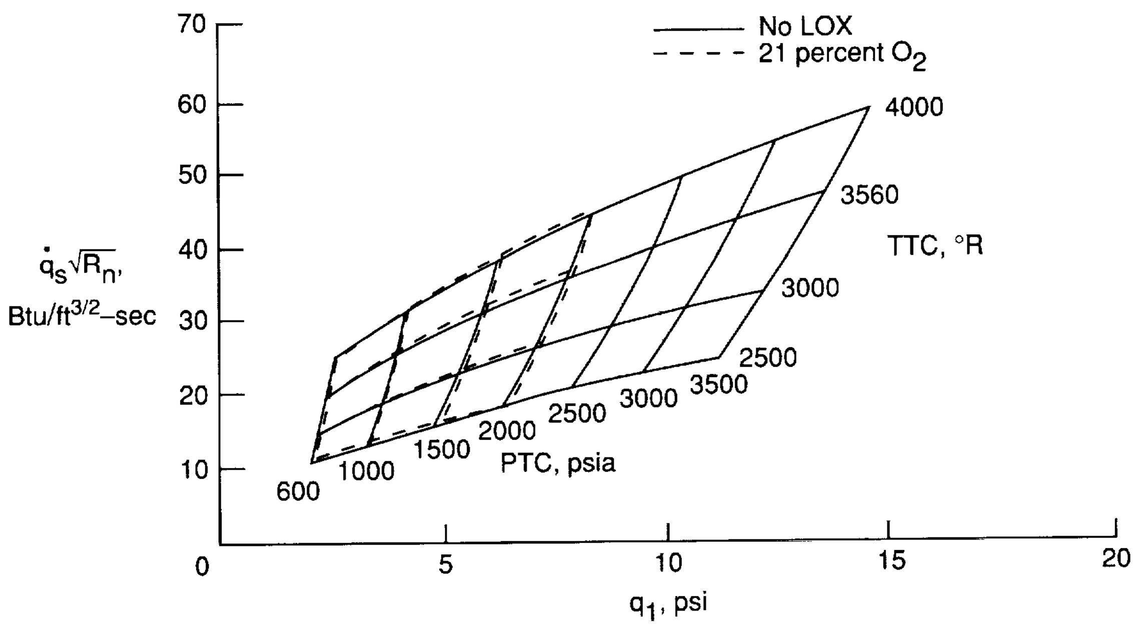


(b) Density and velocity.

Figure 9. Test flow characteristics for Mach 7 nozzle configuration with and without oxygen enrichment.

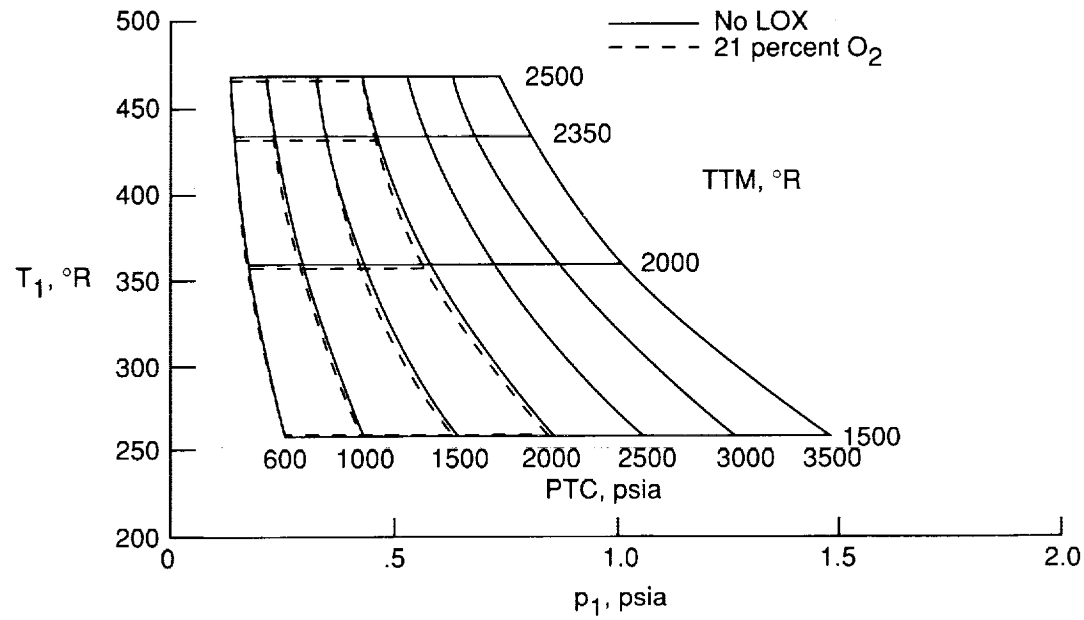


(c) Unit Reynolds and Mach numbers.

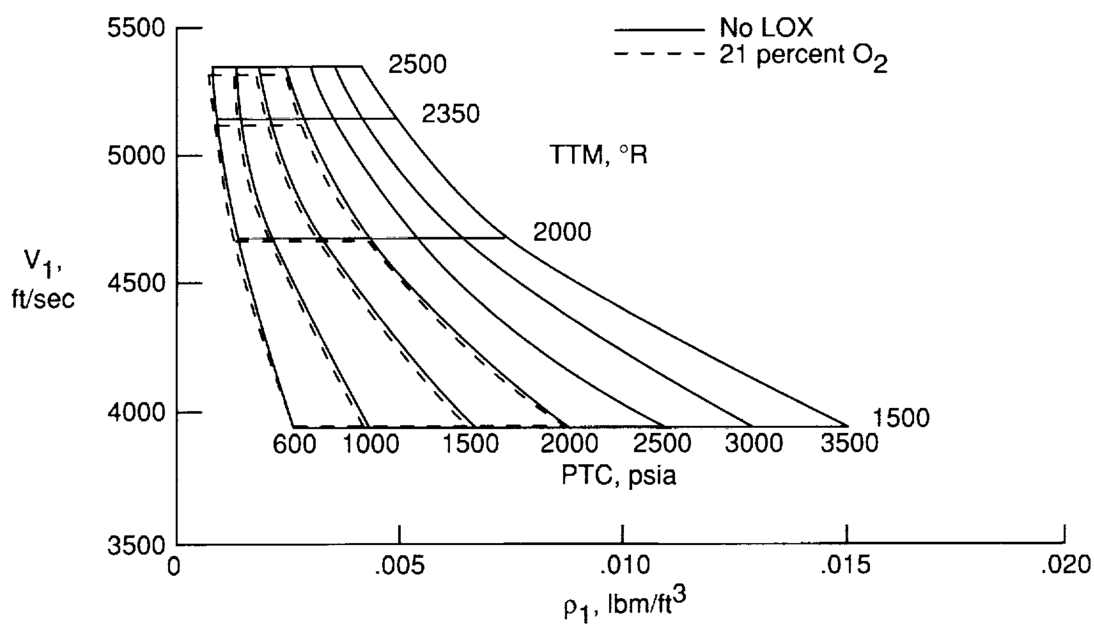


(d) Dynamic pressure and stagnation point heating.

Figure 9. Concluded.

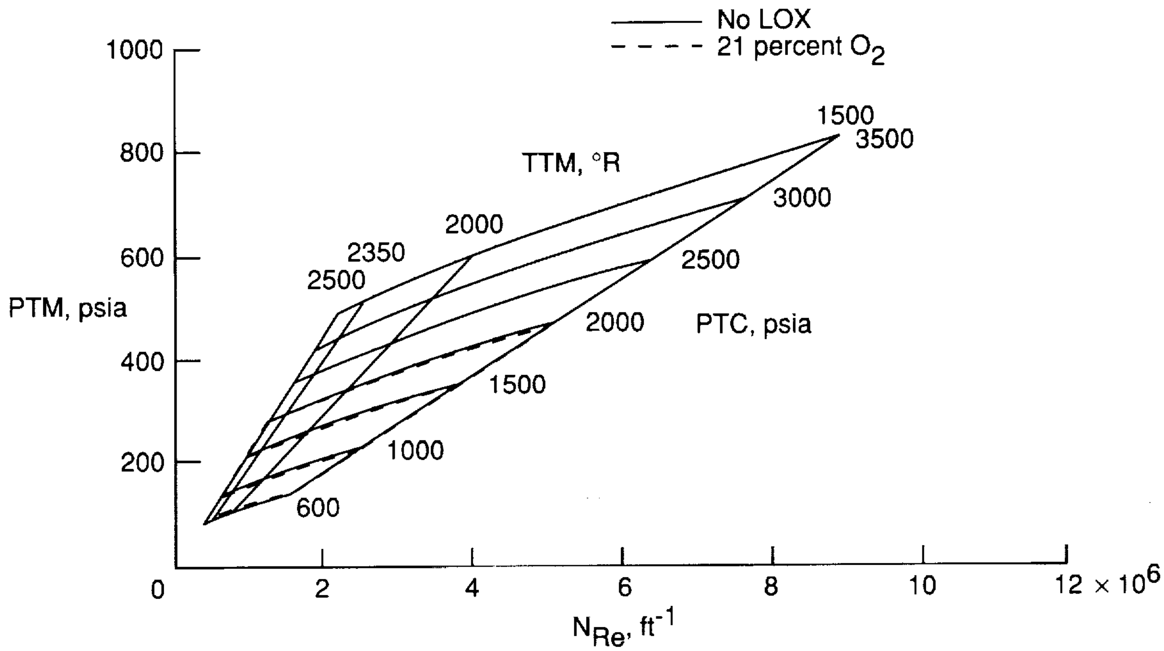


(a) Pressure and temperature.

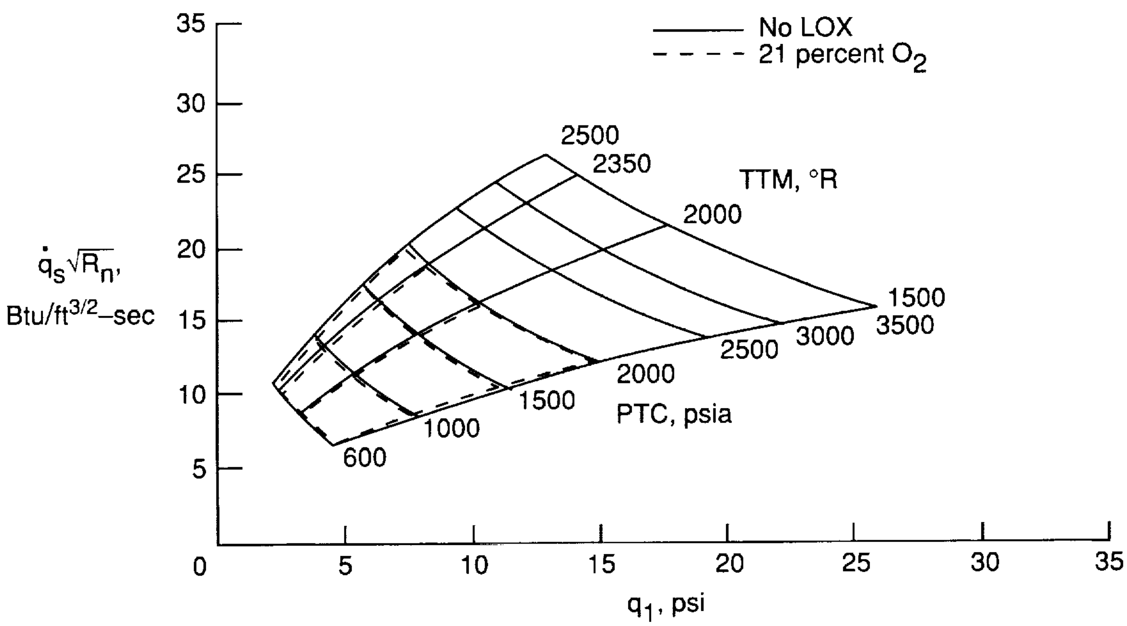


(b) Density and velocity.

Figure 10. Test flow characteristics for Mach 5 configuration with and without oxygen enrichment for TTC = 3560°R.

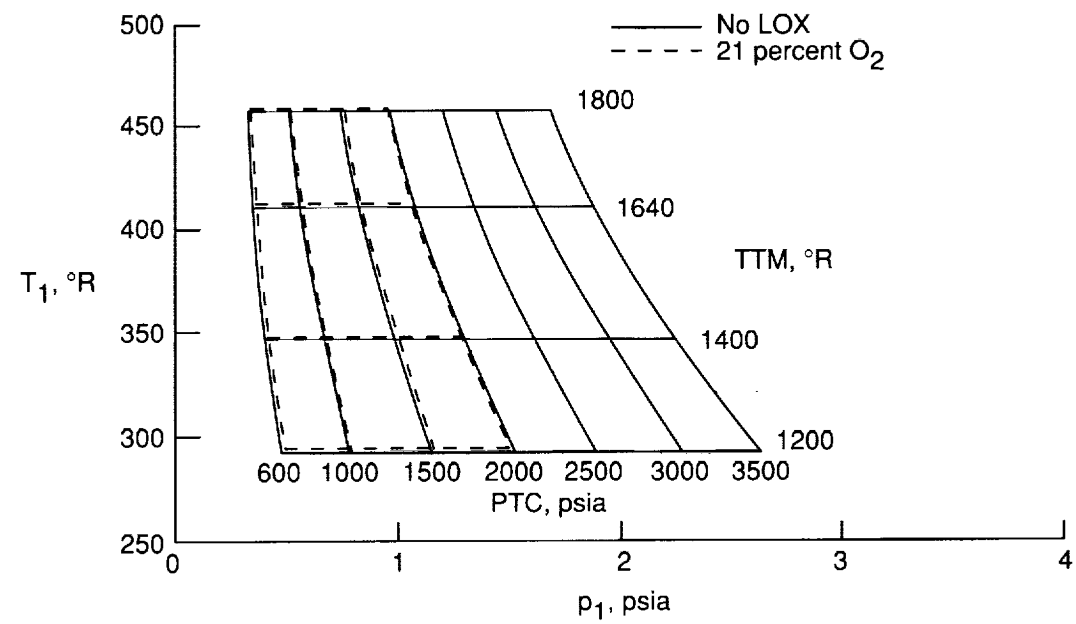


(c) Unit Reynolds number and mixer pressure.

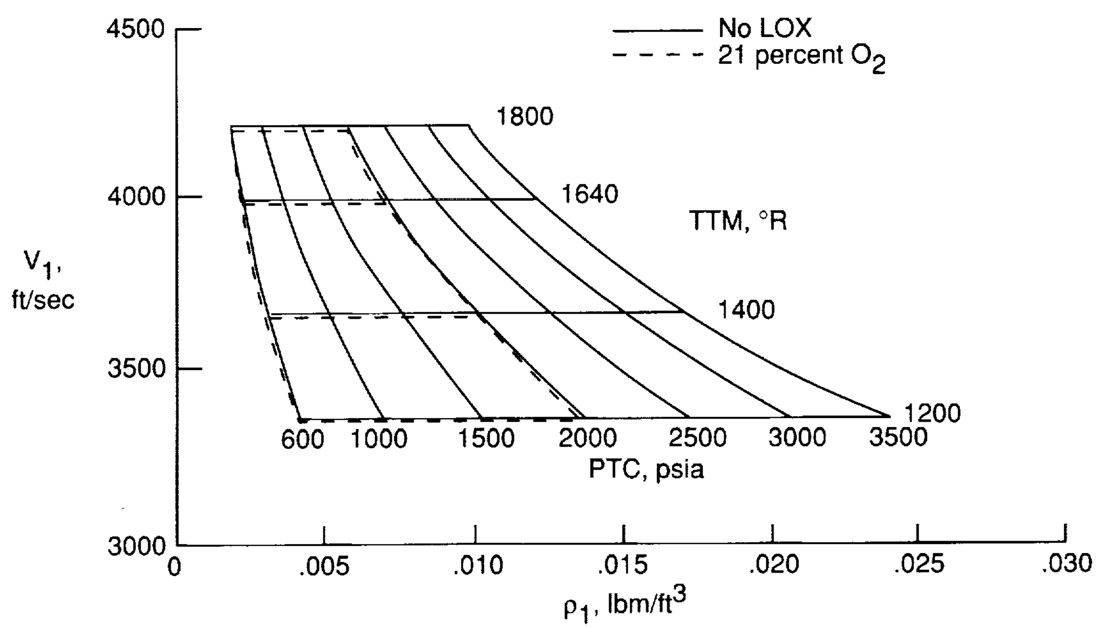


(d) Dynamic pressure and stagnation point heating.

Figure 10. Concluded.

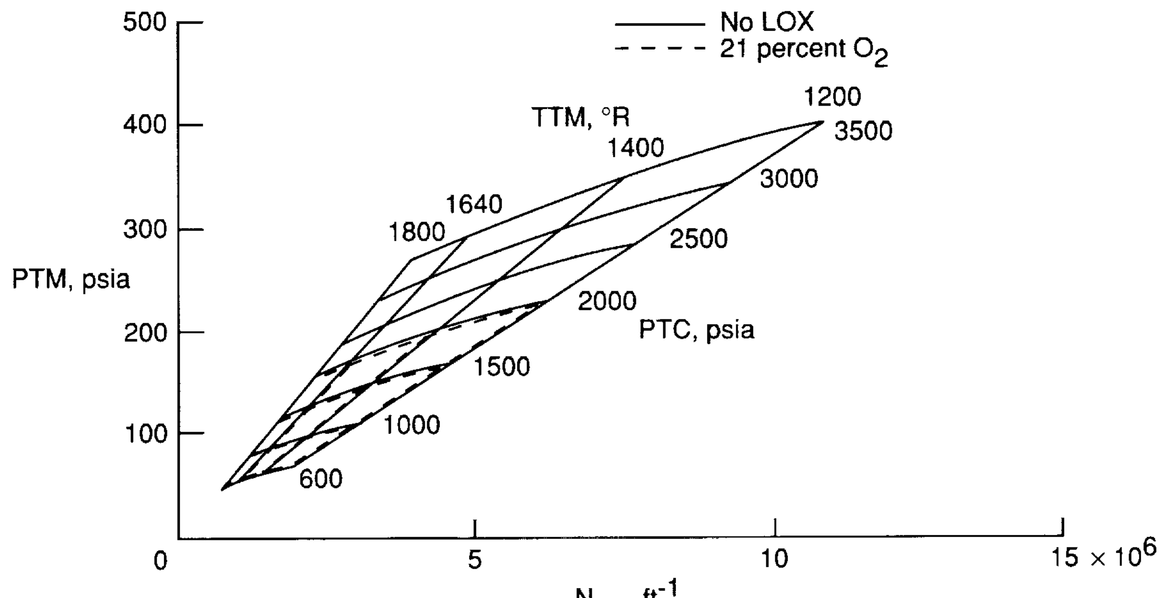


(a) Pressure and temperature.

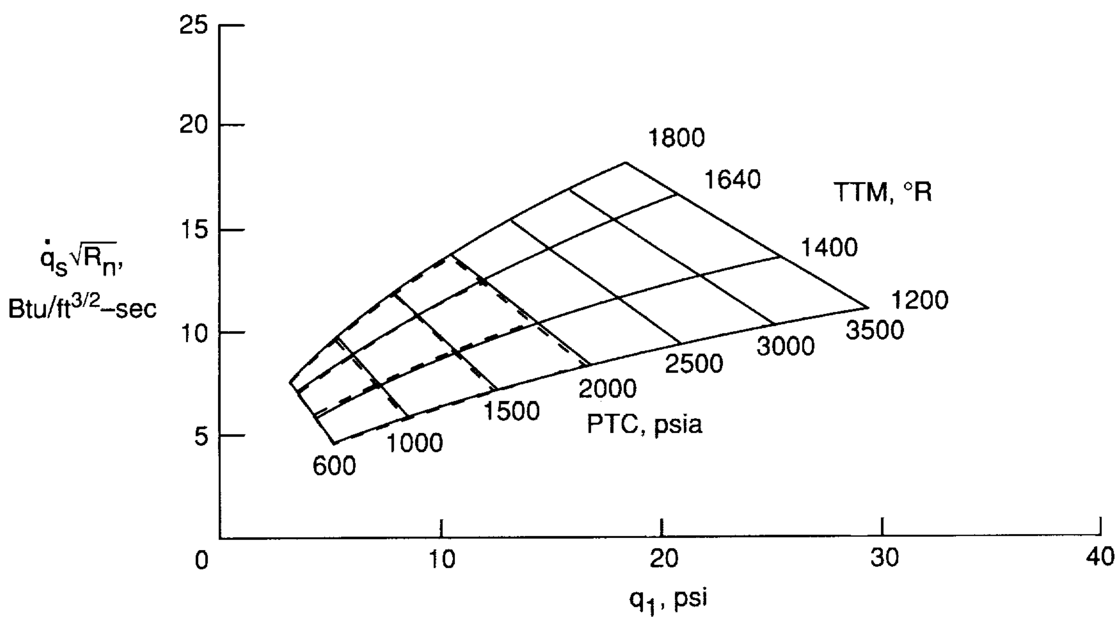


(b) Density and velocity.

Figure 11. Test flow characteristics for Mach 4 configuration with and without oxygen enrichment for TTC = 3560°R.

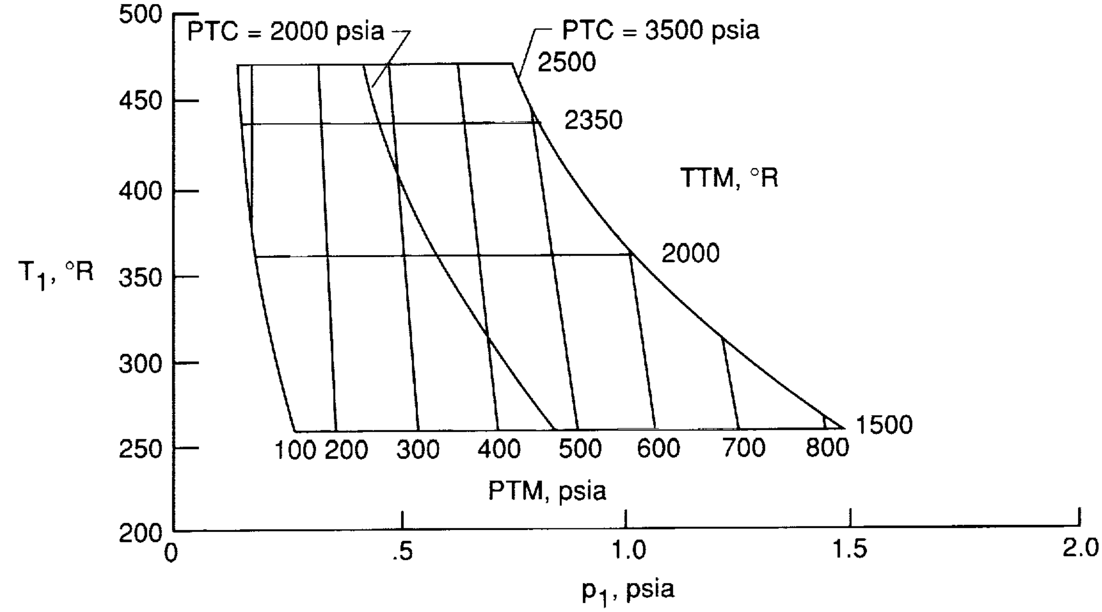


(c) Unit Reynolds number and mixer pressure.

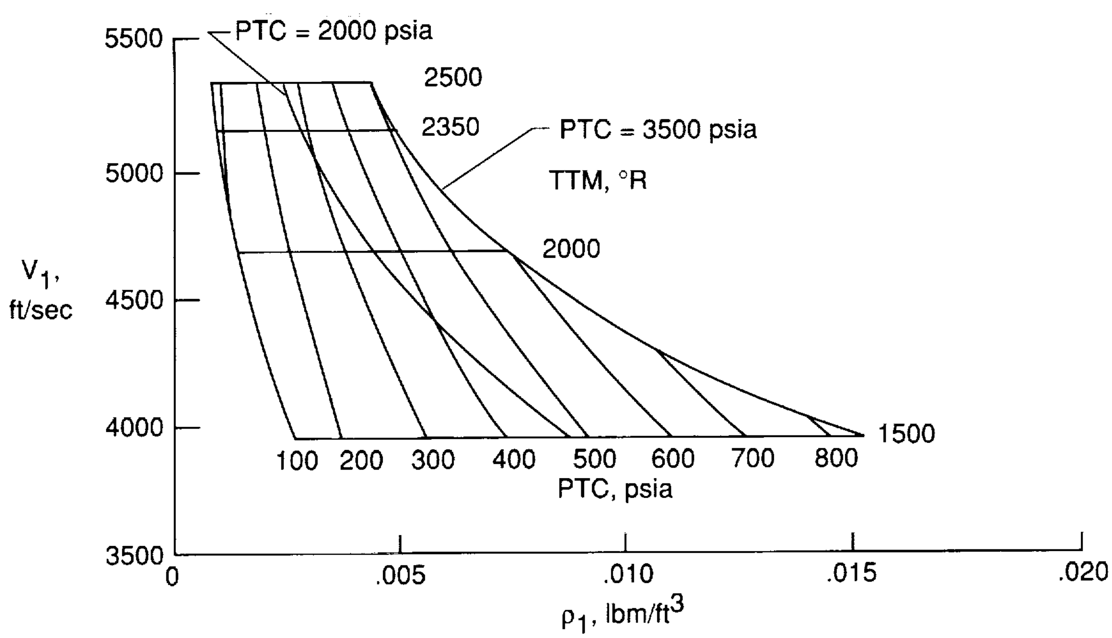


(d) Dynamic pressure and stagnation point heating.

Figure 11. Concluded.

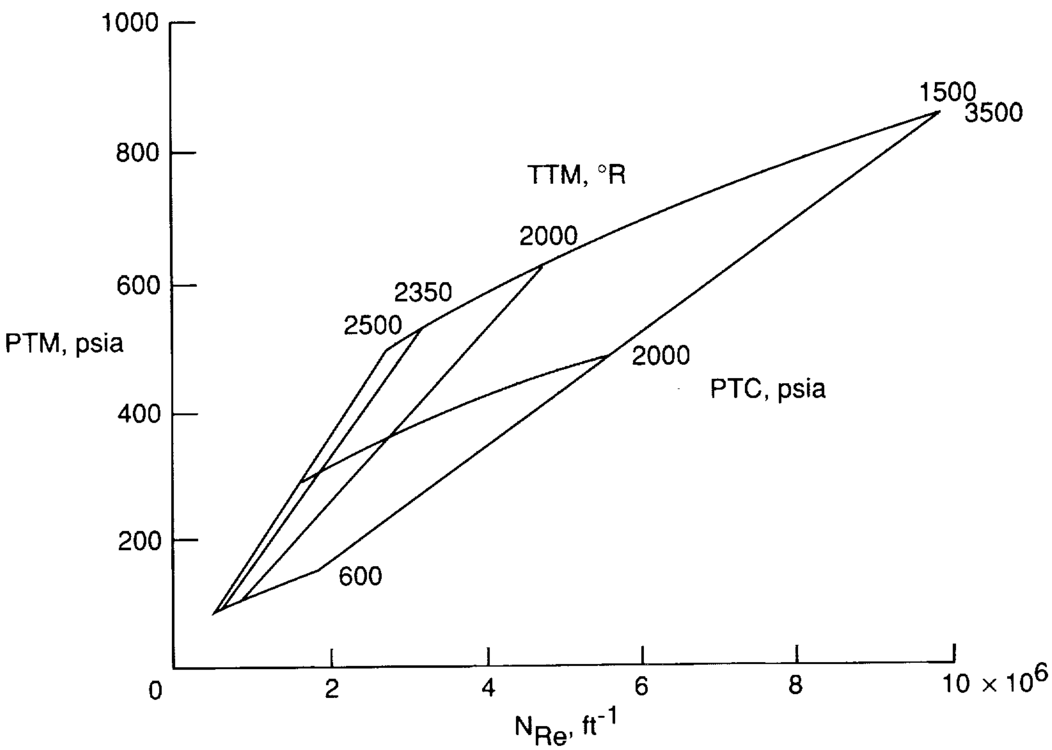


(a) Pressure and temperature.

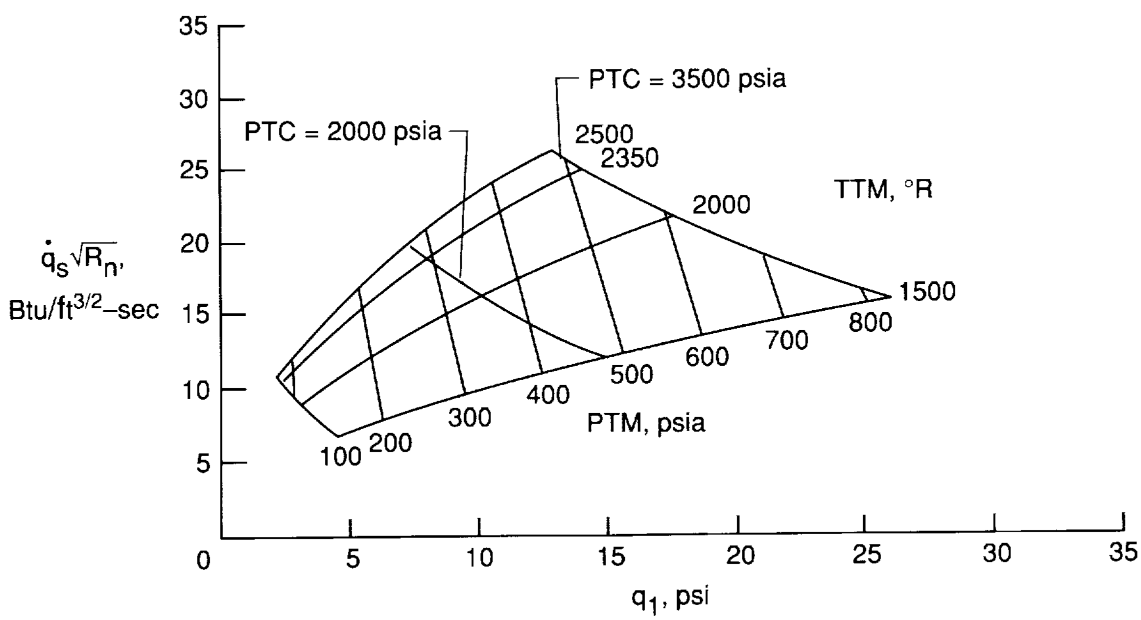


(b) Density and velocity.

Figure 12. Test stream properties for Mach 5 configuration with and without oxygen enrichment.

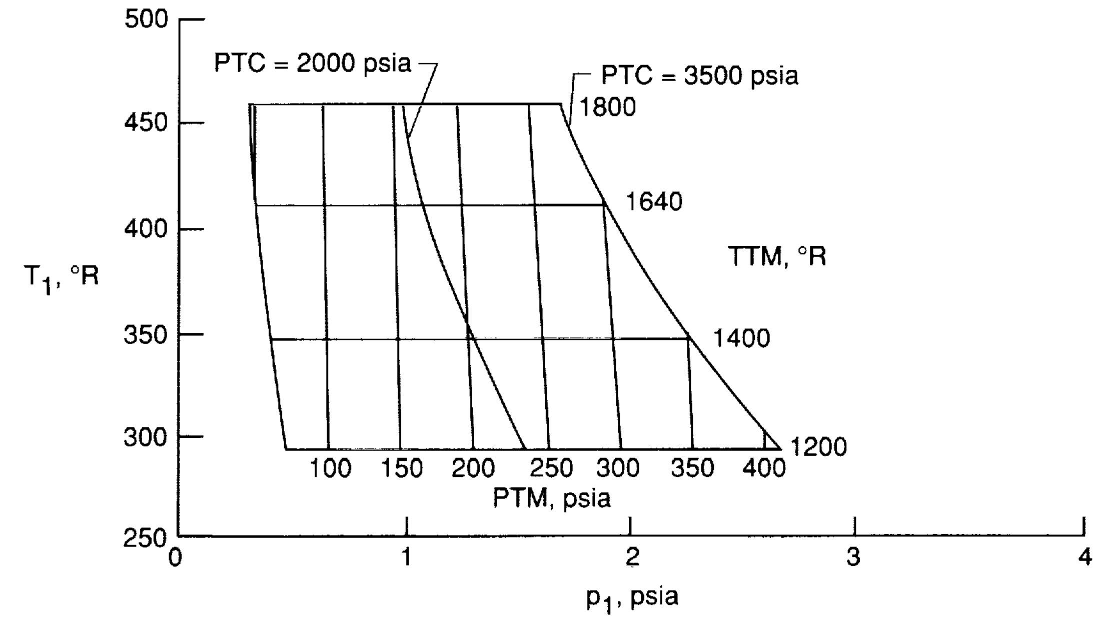


(c) Unit Reynolds number and mixer pressure.

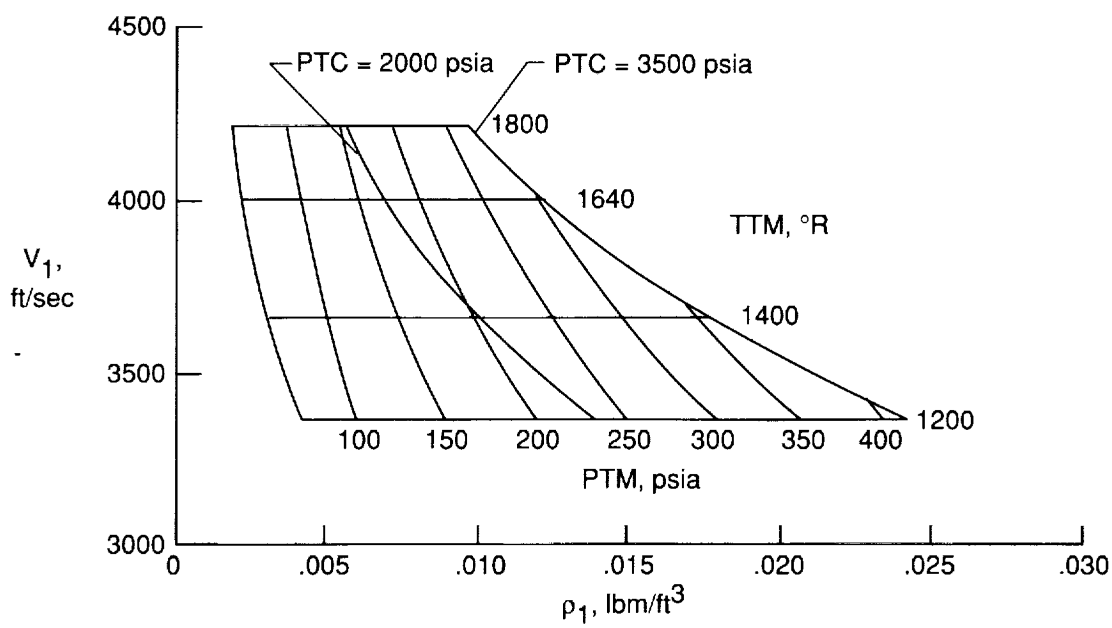


(d) Dynamic pressure and stagnation point heating.

Figure 12. Concluded.

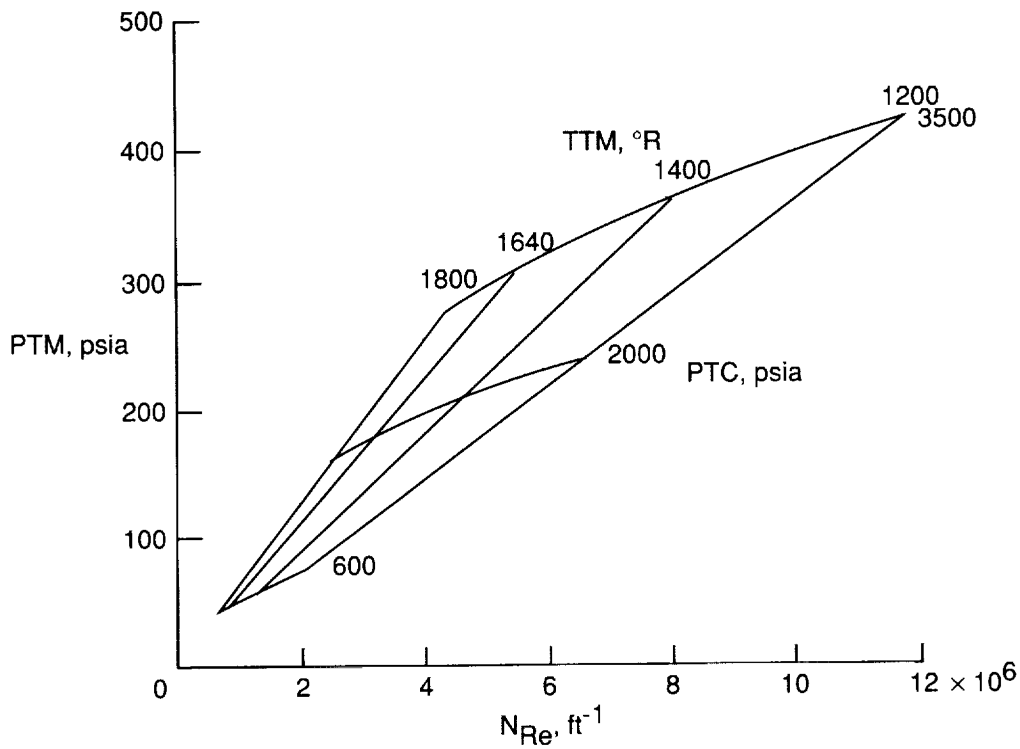


(a) Pressure and temperature.

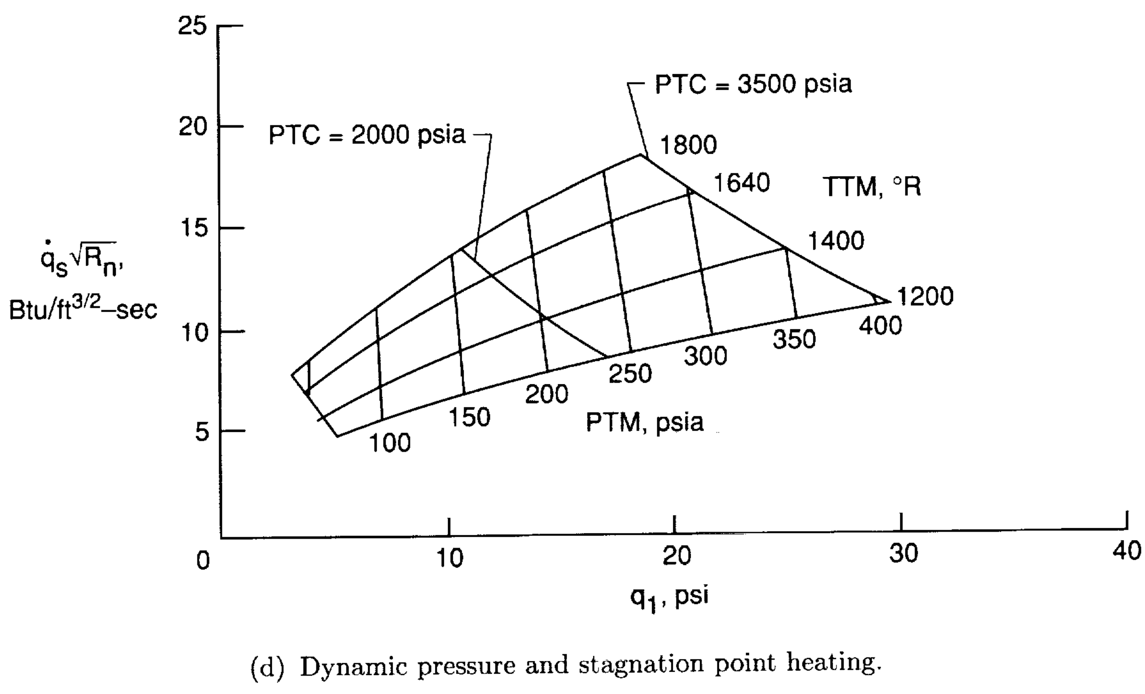


(b) Density and velocity.

Figure 13. Test stream properties for Mach 4 configuration with and without oxygen enrichment.

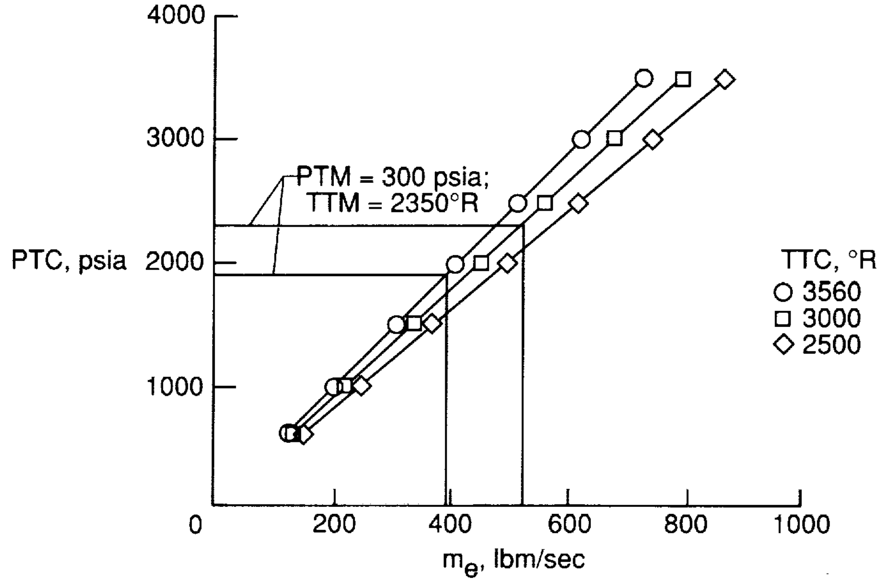


(c) Unit Reynolds number and mixer pressure.

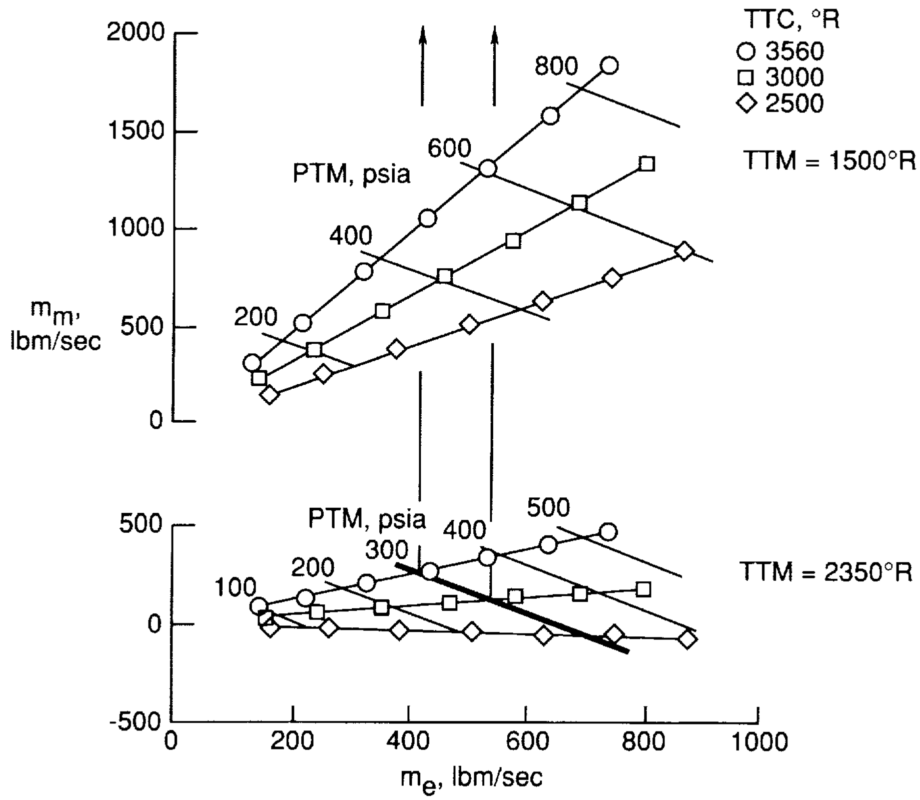


(d) Dynamic pressure and stagnation point heating.

Figure 13. Concluded.

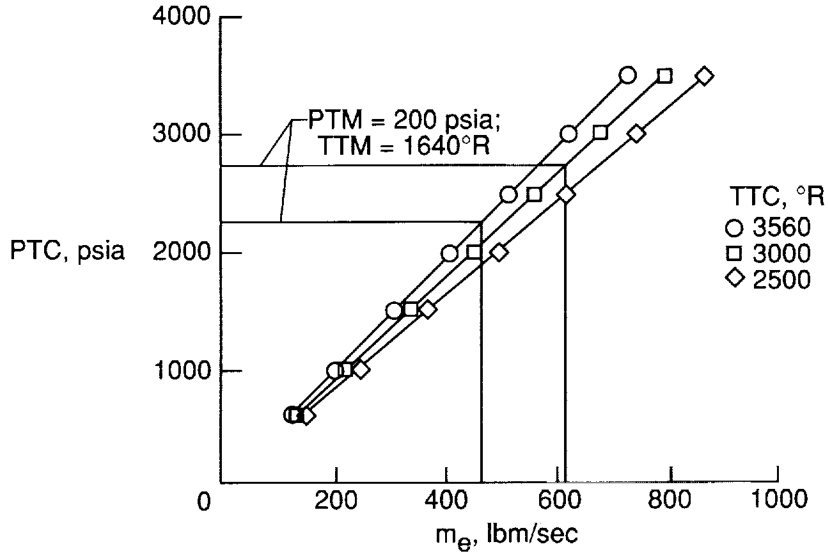


(a) Combustor conditions.

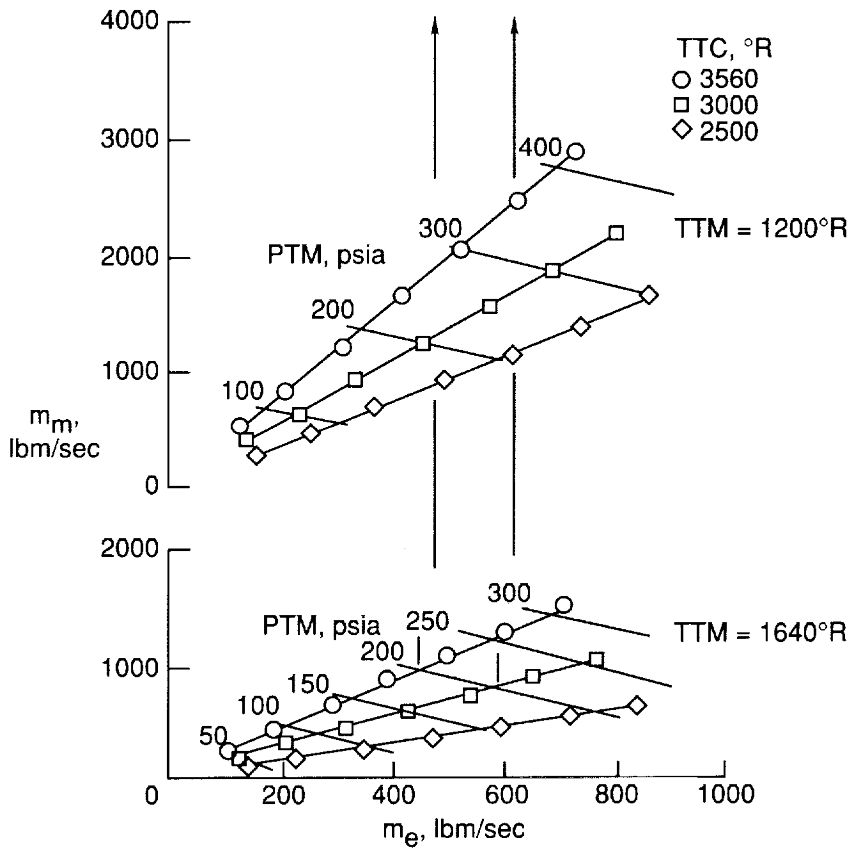


(b) Mixer conditions.

Figure 14. Correlation of combustor and mixer conditions for Mach 5 operation.



(a) Combustor conditions.



(b) Mixer conditions.

Figure 15. Correlation of combustor and mixer conditions for Mach 4 operation.

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12a. DISTRIBUTION/AVAILABILITY STATEMENT
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13. ABSTRACT *(Maximum 200 words)*
The Langley 8-Foot High-Temperature Tunnel (8-ft HTT) is used to test components of hypersonic vehicles for aerothermal loads definition and structural component verification. The test medium of the 8-ft HTT is obtained by burning a mixture of methane-air under high pressure; the combustion products are expanded through an axisymmetric conical-contoured nozzle to simulate atmospheric flight at Mach 7. This facility has been modified to raise the oxygen content of the test medium to match that of air and to include Mach 4 and Mach 5 capabilities. These modifications will facilitate the testing of hypersonic air-breathing propulsion systems for a wide range of flight conditions. A computational method to predict the thermodynamic, transport, and flow properties of the equilibrium chemically reacting oxygen-enriched methane-air combustion products has been implemented in a computer code. This code calculates the fuel, air, and oxygen mass flow rates and test section flow properties for Mach 7, Mach 5, and Mach 4 nozzle configurations for given combustor and mixer conditions. Salient features of the 8-ft HTT are described, and some of the predicted tunnel operational characteristics are presented in the carpet plots to assist users in preparing test plans.

14. SUBJECT TERMS
Air-methane-oxygen combustion; Hypersonic; Equilibrium chemically reacting; Thermodynamic and transport; High-temperature tunnel

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