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RESEARCH MEMORANDUM

THEORETICAL PERFORMANCE OF JP-4 FUEL AND LIQUID OXYGEN AS A ROCKET PROPELLANT

I - FROZEN COMPOSITION

By Vearl N. Huff and Anthony Fortini

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Cleveland, Ohio

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

Theoretical rocket performance for frozen composition during expansion was calculated for the propellant combination JP-4 fuel and liquid oxygen at two chamber pressures and several pressure ratios and oxidant-fuel ratios.

The parameters included are specific impulse, combustion-chamber temperature, nozzle-exit temperature, molecular weight, characteristic velocity, coefficient of thrust, ratio of nozzle-exit area to throat area, specific heat at constant pressure, isentropic exponent, viscosity, and thermal conductivity. A correlation is given for the effect of chamber pressure on several of the parameters.

INTRODUCTION

A continuing interest in hydrocarbon fuels and liquid oxygen as rocket propellants is assured by favorable logistics and relatively high specific impulse. Theoretical performance of several hydrocarbons with liquid oxygen has been reported in the literature, for example, in references 1 to 3.

Additional computations were made for the propellant combination JP-4 fuel and liquid oxygen at the NACA Lewis laboratory between 1953 and 1955 as required for theoretical and experimental programs. These data were computed for both frozen and equilibrium composition during expansion.

The present report presents the data for frozen composition during expansion for two chamber pressures and a wide range of oxidant-fuel ratios and pressure ratios. A correlation is given which permits the determination of specific impulse, characteristic velocity, ratio of nozzle-exit area to throat area, combustion-chamber temperature, and nozzle-exit temperature for a wide range of chamber pressure.

SYMBOLS

The following symbols are used in this report:

A	nozzle area, sq in.
a	local velocity of sound (velocity of flow at throat), ft/sec
C_F	coefficient of thrust; $C_F = \frac{g_c I}{c^*} = \frac{F}{P_c A_t}$
C_p^o	molar specific heat at constant pressure, cal/(mole)(°K)
c_p	specific heat at constant pressure, $\frac{\sum_i n_i (C_p^o)_i}{M(1 - n_k)}$, cal/(g)(°K)
c_v	specific heat at constant volume
c^*	characteristic velocity, $g_c P_c A_t/w$, ft/sec
F	thrust, lb
$f_1, f_2, \ldots$	functions
g_c	gravitational conversion factor, 32.174 (lb mass/lb force)(ft/sec^2)
H_T^o	sum of sensible enthalpy and chemical energy, cal/mole
h	sum of sensible enthalpy and chemical energy per unit mass, $\frac{\sum_i n_i (H_T^o)_i}{M(1 - n_k)}$, cal/g
I	specific impulse, lb force-sec/lb mass
k	coefficient of thermal conductivity, cal/(sec)(cm)(°K)

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M	molecular weight, $\frac{\sum_i n_i M_i}{1 - n_k}$, g/g-mole or lb/lb-mole
n	mole fraction
n_{c^*}	characteristic-velocity exponent, $\left(\frac{\Delta \log c^*}{\Delta \log P_c}\right)$
n_I	specific-impulse exponent for fixed pressure ratio, $\left(\frac{\Delta \log I}{\Delta \log P_c}\right)_{P_c/P}$
n_T	temperature exponent for fixed pressure ratio, $\left(\frac{\Delta \log T}{\Delta \log P_c}\right)_{P_c/P}$
n_ε	area-ratio exponent for fixed pressure ratio, $\left(\frac{\Delta \log \varepsilon}{\Delta \log P_c}\right)_{P_c/P}$
O/F	oxidant-to-fuel weight ratio
P	static pressure (sum of partial pressures), lb/sq in.
p	partial pressure, lb/sq in.
R	universal gas constant (consistent units)
r	equivalence ratio, ratio of four times the number of carbon atoms plus the number of hydrogen atoms to two times the number of oxygen atoms $\frac{4(C) + (H)}{2(O)}$
S_T^o	entropy at a pressure of 1 atmosphere, cal/(mole)(^{o}K)
s	entropy per unit mass, $\frac{\sum_i n_i (S_T^o)_i}{M(1 - n_k)} - \frac{R \sum_j p_j \ln p_j/14.696}{PM}$, cal/(g)(^{o}K)
T	temperature, ^{o}K
w	mass-flow rate, lb/sec
γ	isentropic exponent, $\left(\frac{\partial \log P}{\partial \log \rho}\right)_s$

ε ratio of nozzle area to throat area, A/A_t

ρ density, lb/cu in.

μ absolute viscosity, poises = g/(cm)(sec)

Subscripts:

c combustion chamber

e nozzle exit

i product of combustion including both gaseous and solid phases

j gaseous product of combustion

k solid product of combustion (graphite)

o conditions at 0° K

P constant pressure

P_c/P constant pressure ratio

s constant entropy

t nozzle throat

1 reference point

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CALCULATION OF PERFORMANCE DATA

Performance data were obtained for two chamber pressures for a range of equivalence ratios and pressure ratios. Frozen composition during expansion was assumed.

The computations were carried out by means of the method described in reference 4 with modifications to adapt it for use with an IBM card-programmed electronic calculator. The machine was operated with floating-decimal-point notation and eight significant figures. The successive approximation process used in the calculations was continued until seven-figure accuracy was reached in the desired values of the assigned parameters (mass balance and pressure).

Assumptions

The calculations were based on the following usual assumptions: perfect gas law, adiabatic combustion at constant pressure, isentropic expansion, no friction, homogeneous mixing, and one-dimensional flow. The products of combustion were assumed to be graphite and the following ideal gases: atomic carbon C, methane CH_4, carbon monoxide CO, carbon dioxide CO_2, atomic hydrogen H, hydrogen H_2, water H_2O, atomic oxygen O, oxygen O_2, and the hydroxyl radical OH. The combustion products are assumed to be completely expanded within the exit nozzle; that is, ambient pressure equals exit pressure.

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The graphite was assumed to be finely divided and to have the temperature and velocity of the gases during the flow process.

Initial Data

Thermodynamic data. - The thermodynamic data for all combustion products except graphite, methane, and water were taken from reference 4. Data for graphite were taken from reference 5, and for water from reference 6. Data for methane were determined by the rigid-rotator - harmonic-oscillator approximation using spectroscopic data from reference 7. The base used in this report for assigning absolute values to enthalpy is the same as in reference 4.

The heat of sublimation of graphite at 298.16° K was taken to be 171.698 kilocalories per mole (ref. 8).

Physical and thermochemical data. - The properties of the fuel used in these calculations are typical of the JP-4 fuel delivered to this laboratory over a period of 2 years. The JP-4 fuel was assumed to have a hydrogen-to-carbon weight ratio of 0.163 (atom ratio of 1.942), a lower heat of combustion value of 18,640 Btu per pound and a specific gravity of 0.769. Additional properties of jet fuels may be found in reference 9.

Several properties of the oxidant taken from references 4, 8, and 10 are listed in table I.

Viscosity data. - The viscosity data for the individual combustion products were either taken from the literature when available, or estimated.

The viscosity data for CO, CO_2, CH_4, H_2, and O_2 were calculated by the method of reference 11 using the values of the constants from table 1A of that reference.

The viscosities of C, O, H, and OH were calculated by the method of reference 12, which assumes that the logarithm of viscosity is a linear function of the logarithm of the temperature.

The viscosity of H_2O was calculated from the modified Sutherland equation given in reference 13.

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Computation of Combustion Conditions

A combustion pressure was assigned (300 or 600 lb/sq in. abs). At this assigned pressure, the composition n_i, enthalpy h (including both chemical and sensible energy), and entropy s, were determined for three temperatures at 100° K intervals. The temperatures were chosen to band the value of enthalpy for the propellant mixture h_c. The formulas (ref. 4) used to calculate h and s are

$$h = \frac{\sum_i n_i (H_T^O)_i}{M(1 - n_k)} \tag{1}$$

$$s = \frac{\sum_i n_i (S_T^O)_i}{M(1 - n_k)} - \frac{1.98718 \sum_j p_j \ln p_j/14.696}{PM} \tag{2}$$

Combustion composition corresponding to h_c was obtained by ordinary three-point interpolation of composition as a function of h. Entropy s_c corresponding to h_c was obtained by means of a three-point - three-slope interpolation of s as a function of h. The slope was obtained by means of the thermodynamic relation

$$\left(\frac{\partial s}{\partial h}\right)_P = \frac{1}{T} \tag{3}$$

It is convenient to treat the products of combustion (sometimes a mixture of solid graphite and ideal gases) as a single homogeneous fluid. Therefore, the molecular weight of the combustion products M is defined as the weight of a sample (including gases and solid graphite) divided by the number of moles of gas and was computed by

$$M = \frac{\sum_i n_i M_i}{1 - n_k} \tag{4}$$

This value of M is suitable for use in the gas law

$$P = \frac{\rho RT}{M} \tag{5}$$

provided the solid phase is included in the density. Such a fluid will exhibit ideal properties as long as the volume of the gases is large with respect to the volume of the solid phase. The procedure is also consistent with the assumption that the solid particles are small enough to be considered gas molecules of extremely large molecular weight.

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Computation of Exit Conditions

Calculation of parameters at assigned temperatures. - Exit temperatures were selected at 300° or 400° K intervals to cover the range of pressure ratios from 1 to 1500. At these selected temperatures, the following data were computed assuming isentropic expansion and frozen composition: pressure, enthalpy, specific heat at constant pressure, isentropic exponent, absolute viscosity, thermal conductivity, nozzle-area ratio, coefficient of thrust, and specific impulse.

Interpolation of throat pressure. - A cubic equation in terms of ln P was derived from the following function and its first derivative using the data at two assigned temperatures:

$$\text{function, } f_1 = \ln f_2 = \ln\left(\frac{h}{R} + \frac{\gamma T}{2M} - \frac{h_o}{R}\right)$$

$$\text{first derivative, } \frac{df_1}{d \ln P} = \frac{T}{2Mf_2}\left(\gamma + 1 + \frac{d\gamma}{d \ln P}\right)$$

(Values for $d\gamma/d \ln P$ were found by a numerical method.)

The two temperatures were selected to band the throat temperature. The pressure at the throat was found by interpolating ln P as a function of f_1 for the point $f_1 = \ln\left(\frac{h_c}{R} - \frac{h_o}{R}\right)$. At this point the velocity of flow equals the velocity of sound.

Interpolation of enthalpy. - Enthalpies were interpolated for a series of pressures including the throat pressure by means of quartic equations in terms of ln P. Each of the quartic equations used was derived from data at two successive assigned temperatures and used to interpolate those points within the temperature interval. The data used in forming each quartic were the following function at one of the assigned temperatures and its first and second derivatives at both assigned temperatures:

$$\text{function, } f_3 = \frac{h}{R}$$

$$\text{first derivative, } \frac{df_3}{d \ln P} = \frac{T}{M}$$

$$\text{second derivative, } \frac{d^2 f_3}{(d \ln P)^2} = \frac{T}{M}\left(\frac{\gamma - 1}{\gamma}\right)$$

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Interpolation of temperature. - Temperatures were interpolated for a series of pressures including the throat pressure by means of cubic equations in terms of $\ln P$. Each of the cubic equations used was derived from data at two successive assigned temperatures and used to interpolate those points within the temperature interval. The data used in forming each cubic were the following function and its derivative at both assigned temperatures:

$$\text{function, } f_4 = \ln T$$

$$\text{first derivative, } \frac{df_4}{d \ln P} = \frac{\gamma - 1}{\gamma}$$

Interpolation of specific heat. - Specific heats were interpolated for a series of pressures including the throat pressure by means of cubic equations in terms of $\ln P$. Each of the cubic equations used was derived from values of specific heat for four successive assigned temperatures and used to interpolate those points within the interval of the two middle temperatures.

Accuracy of interpolation. - The errors due to interpolation were checked for several cases. The values presented for enthalpy, entropy, and specific impulse appear to be correctly computed to all figures tabulated, while the remaining parameters may in some cases be in error by one or two figures in the last place tabulated. However, because of uncertainties in thermodynamic data used, all values are probably tabulated to more places than are entirely significant.

Formulas

The formulas used in computing the various performance parameters are as follows:

Specific impulse, lb force-sec/lb mass

$$I = 294.98\sqrt{\frac{h_c - h_e}{1000}} \tag{6}$$

Throat area per unit mass-flow rate, (sq in.)(sec)/lb

$$\frac{A_t}{w} = \frac{2781.6\ T_t}{P_t M_t a} \tag{7}$$

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Characteristic velocity, ft/sec

$$c^* = g_c P_c (A_t/w) = 32.174\ P_c (A_t/w) \tag{8}$$

Coefficient of thrust

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$$C_F = \frac{g_c I}{c^*} = \frac{32.174\ I}{c^*} \tag{9}$$

Nozzle area per unit mass flow rate, (sq in.)(sec)/lb

$$\frac{A}{w} = \frac{86.455\ T}{PMI} \tag{10}$$

Ratio of nozzle area to throat area

$$\varepsilon = \frac{A/w}{A_t/w} \tag{11}$$

Specific heat at constant pressure, cal/(g)(^{o}K)

$$c_p = \frac{\sum_i n_i (C_p^o)_i}{M(1 - n_k)} \tag{12}$$

Isentropic exponent (when the composition is frozen)

$$\gamma = \left(\frac{\partial \ln P}{\partial \ln \rho}\right)_s = \frac{c_p}{c_p - \frac{R}{M}} = \frac{c_p}{c_v} \tag{13}$$

Absolute viscosity, poises

$$\mu = \frac{PM}{\sum_j \frac{p_j}{\mu_j/M_j}} \tag{14}$$

Coefficient of thermal conductivity, cal/(sec)(cm)($^{\circ}$K)

$$k = \mu\left(c_p + \frac{5}{4}\frac{R}{M}\right) \tag{15}$$

The values of viscosity and thermal conductivity for mixtures of combustion gases calculated by means of equations (14) and (15) are only approximate. When more reliable transport properties for the various products of combustion become available, a more rigorous procedure for computing the properties of mixtures may also be justified. When solid graphite was present among the combustion products, it was omitted from equation (14). 4021

THEORETICAL PERFORMANCE DATA

Tables

The calculated values of the performance parameters are given in tables II to VI. The properties of gases in the combustion chamber and the characteristic velocity are given in table II for each chamber pressure and equivalence ratio. Table III presents the values of performance parameters at assigned temperatures and constant entropy. These values were computed directly and used to interpolate properties for assigned pressure ratios. The first temperature for each equivalence ratio is greater than the combustion temperature and represents an isentropic compression from combustion conditions. The data for this temperature were used for interpolation. The values of viscosity and thermal conductivity of the mixture are also given in this table as functions of temperature.

The performance parameters for small pressure ratios from 1 to 8 are given in table IV. These properties permit computations within the rocket nozzle and for finite combustion-chamber diameters. Properties at the throat may be found where $\varepsilon = 1.000$. The values adjacent to the throat correspond to pressures 1.2 and 0.8 times the throat pressure.

The performance parameters for pressure ratios from 10 to 1500 are given in table V. This table gives sufficient data to permit interpolation of complete data for any pressure ratio within the range tabulated.

The specific impulse and area-ratio values for expansion from chamber pressure to 1 atmosphere are summarized in table VI. The maximum values calculated for specific impulse for chamber pressures of 600 and 300 pounds per square inch absolute are 271.8 and 250.4, respectively, at 31.98 weight percent fuel.

Curves

The performance parameters are plotted in figures 1 to 5 for chamber pressures of 300 and 600 pounds per square inch absolute.

Curves of specific impulse are presented in figure 1 for pressure ratios from 10 to 1500 as functions of weight percent fuel. The maximum values occur at about 31.98 weight percent fuel. The exponent n_I is also shown.

Curves of combustion temperature and exit temperature for pressure ratios from 10 to 1500 are plotted in figure 2 as functions of weight percent fuel. The exponent n_T is also shown.

Curves of the ratio of nozzle area to throat area are plotted in figure 3 for pressure ratios from 10 to 1500 as functions of weight percent fuel. The exponent n_ε is also shown.

Figure 4 gives the curves for coefficient of thrust for pressure ratios from 10 to 1500 as functions of weight percent fuel.

Figure 5 presents curves of molecular weight and characteristic velocity as functions of weight percent fuel. Also shown is the exponent n_{c^*}.

Effect of solid graphite. - The theoretical calculations of equilibrium composition in the combustion chamber showed that solid graphite was not present for the equivalence ratios of 1 to 2 (weight percent fuel, 22.71 to 37.01) and was present for equivalence ratios of 3. The appearance of solid graphite did affect the values of the thermodynamic parameters and resulted in a break in the performance data in the region of equivalence ratios between 2 and 3. The performance at an equivalence ratio of 3 was not plotted in figures 1 to 5 but is presented in tables II to VI.

Chamber-Pressure Effect

The logarithms of the parameters I, T, ε, and c^* are very nearly linear with the logarithm of chamber pressure for a fixed equivalence ratio and pressure ratio. This linearity permits the data to be correlated by means of exponents according to the following equations:

$$n_I = \left(\frac{\Delta \log I}{\Delta \log P_c}\right)_{P_c/P} \tag{16}$$

$$n_T = \left(\frac{\Delta \log T}{\Delta \log P_c}\right)_{P_c/P} \tag{17}$$

$$n_\varepsilon = \left(\frac{\Delta \log \varepsilon}{\Delta \log P_c}\right)_{P_c/P} \tag{18}$$

$$n_{c^*} = \left(\frac{\Delta \log c^*}{\Delta \log P_c}\right) \tag{19}$$

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Equations (16) to (19) may be written as

$$I = I_1 \left(\frac{P_c}{P_{c,1}}\right)^{n_I} \tag{20}$$

$$T = T_1 \left(\frac{P_c}{P_{c,1}}\right)^{n_T} \tag{21}$$

$$\varepsilon = \varepsilon_1 \left(\frac{P_c}{P_{c,1}}\right)^{n_\varepsilon} \tag{22}$$

$$c^* = c_1^* \left(\frac{P_c}{P_{c,1}}\right)^{n_{c^*}} \tag{23}$$

where $P_{c,1}$ may be selected to be either 300 or 600 pounds per square inch absolute provided that I_1, T_1, ε_1, and c_1^* are the corresponding values for the chamber pressure selected.

The data of tables II and V were used in equations (16) to (19) to calculate exponents which are also shown in the tables and are plotted in figures 1, 2, 3, and 5.

To illustrate the use of these exponents, suppose it is desired to obtain the value of specific impulse for a chamber pressure of 450 pounds per square inch absolute and a pressure ratio of 30.62 (exit pressure, 1 atm) for an equivalence ratio r of 1.5 (30.59 weight percent fuel). From figure 1(a) and table V, the value of I at this pressure ratio and equivalence ratio (but for a chamber pressure of 300 lb/sq in. abs) is 261.5 and the value of n_I is 0.0142. From equation (20),

$$I = 261.5 \left(\frac{450}{300}\right)^{0.0142}$$

$$= 261.5\ (1.00577)$$

$$= 263.0$$

A comparison of the parameters obtained by means of the chamber-pressure correlation and by a direct calculation for two examples is given in the following table ($r = 1.5$; 30.59 weight percent fuel):

Parameter	P_c = 450 lb/sq in. abs P_e = 1 atm			P_c = 1200 lb/sq in. abs P_e = 1 atm		
	Estimated by corre-lation	Direct calcu-lation	Error	Estimated by corre-lation	Direct calcu-lation	Error
I	263.04	263.09	0.05	290.40	290.25	0.15
T_c	3482.7	3482.9	.1	3605.2	3600.4	4.8
T_e	1815.4	1815.5	.2	1560.5	1558.0	2.5
ε	4.643	4.641	.002	9.429	9.408	.021
c^*	5762.5	5762.7	.2	5838.2	5835.0	3.2

It is expected that values estimated for other equivalence ratios and pressure ratios will have small errors of the order of magnitude shown in the preceding table. A possible exception might occur when the value of the exponent is changing rapidly such as in the region when solid graphite first appears.

SUMMARY OF RESULTS

A theoretical investigation of the performance of JP-4 fuel with liquid oxygen as an oxidant was made for the following conditions: (1) equivalence ratios from 1 to 3, (2) chamber pressures of 300 and 600 pounds per square inch, (3) pressure ratios from 1 to 1500, and (4) frozen composition during expansion.

The results of the investigation are as follows:

1. The maximum values of specific impulse for chamber pressures of 600 and 300 pounds per square inch absolute (40.83 and 20.41 atm) and and exit pressure of 1 atmosphere were 271.8 and 250.4, respectively, at 31.98 weight percent fuel.

2. The data presented in this report permit interpolation of complete performance data for any equivalence ratio from 1.00 to 2.00, chamber pressure from 150 to 1200 pounds per square inch absolute, and pressure ratio up to 1500.

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Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
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TABLE I. - PROPERTIES OF LIQUID OXYGEN

Molecular weight, M	32.00
Density, g/cc	[a]1.1415
Freezing point, $^{\circ}$C	[b]-218.76
Boiling point, $^{\circ}$C	[b]-182.97
Enthalpy required to convert liquid at boiling point to gas at 25° C, kcal/mole	[c]3.080
Enthalpy of vaporization, kcal/mole	[d]1.630
Enthalpy of fusion, kcal/mole	[e]0.106

[a]At -182.0° C; ref. 10.

[b]Ref. 8.

[c]Ref. 4.

[d]At -182.97° C; ref. 8.

[e]At -218.76° C; ref. 8.

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TABLE II. - THERMODYNAMIC PROPERTIES OF COMBUSTION GASES FOR JP-4 FUEL AND LIQUID OXYGEN

Equivalence ratio, r, $\frac{4(C)+(H)}{2(O)}$	Percent fuel by weight	Oxidant to fuel weight ratio, O/F	Temperature, T, °K	Temperature exponent, n_T	Molecular weight, M	Enthalpy, h, cal/g (a)	Entropy, s, $\frac{cal}{(g)(^{o}K)}$	Specific heat, c_p, $\frac{cal}{(g)(^{o}K)}$ (b)	Isentropic exponent, γ (b)	Characteristic-velocity exponent, n_{c*} (b)	Characteristic velocity, c*, ft/sec (b)
Combustion-chamber pressure, 300 lb/sq in. abs											
1.00	22.71	3.403	3507	0.0426	25.24	2531.6	2.6273	0.449	1.212	0.0157	5415
1.20	26.07	2.836	3523	.0423	23.80	2901.1	2.7391	.469	1.217	.0157	5582
1.30	27.64	2.618	3511	.0411	23.14	3074.1	2.7889	.478	1.219	.0153	5647
1.40	29.15	2.431	3482	.0386	22.50	3239.9	2.8349	.486	1.222	.0146	5697
1.50	30.59	2.269	3433	.0353	21.88	3399.0	2.8773	.493	1.226	.0133	5732
1.60	31.98	2.127	3363	.0309	21.27	3551.6	2.9160	.500	1.230	.0119	5746
1.80	34.59	1.891	3160	.0208	20.09	3839.4	2.9826	.512	1.239	.0080	5716
2.00	37.01	1.702	2900	.0114	18.99	4105.8	3.0351	.521	1.251	.0045	5613
Combustion-chamber pressure, 600 lb/sq in. abs											
1.00	22.71	3.403	3612	0.0426	25.48	2531.6	2.5729	0.451	1.209	0.0157	5475
1.20	26.07	2.836	3628	.0423	24.03	2901.1	2.6815	.470	1.213	.0157	5643
1.30	27.64	2.618	3612	.0411	23.36	3074.1	2.7297	.479	1.216	.0153	5707
1.40	29.15	2.431	3576	.0386	22.70	3239.9	2.7740	.487	1.219	.0146	5755
1.50	30.59	2.269	3518	.0353	22.05	3399.0	2.8146	.494	1.223	.0133	5785
1.60	31.98	2.127	3436	.0309	21.41	3551.6	2.8515	.501	1.227	.0119	5794
1.80	34.59	1.891	3205	.0208	20.17	3839.4	2.9142	.513	1.238	.0080	5747
2.00	37.01	1.702	2923	.0114	19.03	4105.8	2.9627	.522	1.250	.0045	5630
3.00	46.85	1.134	1657		15.49	5188.4	3.0102	.542	1.310		4618

[a]The base used for enthalpy is given in reference 4.

[b]Parameter based on frozen composition.

TABLE III. - THEORETICAL ROCKET PERFORMANCE AT ASSIGNED TEMPERATURE FOR JP-4 FUEL AND LIQUID OXYGEN WITH FROZEN COMPOSITION DURING ISENTROPIC PROCESS

(a) Chamber pressure, 300 pounds per square inch absolute.

Temperature, T, °K	Pressure, P, lb/sq in. abs	Enthalpy, h, cal/g	Isentropic exponent, γ	Specific heat, c_p, $\frac{cal}{(g)(°K)}$	Absolute viscosity μ, micropoises	Thermal conductivity k, $\frac{cal}{(cm)(sec)(°K)}$	Area ratio, ϵ	Thrust coefficient, C_F	Specific impulse, I, $\frac{lb\text{-}sec}{lb}$
r = 1.00; percent fuel[a] = 22.71; O/F[b] = 3.405									
3600	348.310	2573.3	1.212	0.4505	924	0.00051	-------	-------	-------
3200	178.280	2394.2	1.215	.4449	856	.00047	1.00	0.650	109.3
2800	84.332	2217.6	1.219	.4380	785	.00042	1.23	.982	165.3
2400	36.079	2044.1	1.225	.4294	709	.00037	1.97	1.224	206.0
2000	13.522	1874.5	1.232	.4182	629	.00032	3.78	1.421	239.1
1600	4.224	1710.1	1.243	.4028	541	.00027	8.65	1.588	267.4
1200	1.009	1553.2	1.261	.3807	444	.00021	24.89	1.734	291.8
900	.262	1442.2	1.282	.3581	361	.00017	68.19	1.829	307.9
600	.045	1338.9	1.313	.3301	266	.00011	254.84	1.914	322.2
r = 1.20; percent fuel = 26.07; O/F = 2.836									
3600	338.520	2937.1	1.216	0.4699	910	0.00052	-------	-------	-------
3200	175.160	2750.2	1.219	.4641	843	.00048	1.00	0.661	114.6
3000	122.520	2657.8	1.221	.4607	808	.00046	1.06	.839	145.5
2800	83.852	2566.0	1.224	.4569	773	.00043	1.23	.984	170.2
2600	55.996	2475.0	1.226	.4527	736	.00041	1.51	1.110	192.6
2400	36.359	2384.9	1.229	.4480	699	.00039	1.96	1.222	211.9
2200	22.858	2295.9	1.232	.4426	660	.00036	2.63	1.323	229.5
2000	13.840	2208.0	1.237	.4363	619	.00033	3.70	1.416	245.6
1800	8.017	2121.4	1.242	.4289	577	.00031	5.41	1.501	260.5
1600	4.405	2036.5	1.248	.4200	533	.00028	8.32	1.581	274.3
1200	1.078	1872.9	1.267	.3965	437	.00022	23.37	1.724	299.1
900	.287	1757.4	1.288	.3729	356	.00017	62.53	1.818	315.5
600	.050	1649.7	1.320	.3443	262	.00012	226.75	1.902	330.0
r = 1.30; percent fuel = 27.64; O/F = 2.618									
3600	345.110	3116.9	1.218	0.4789	904	0.00053	-------	-------	-------
3200	179.640	2926.5	1.222	.4730	838	.00049	1.00	0.646	113.3
2800	86.581	2738.7	1.226	.4657	768	.00044	1.21	.973	170.2
2400	37.829	2554.1	1.232	.4567	694	.00039	1.90	1.212	212.7
2200	23.882	2463.4	1.235	.4511	656	.00037	2.55	1.314	230.5
2000	14.527	2373.8	1.239	.4447	616	.00034	3.56	1.407	246.9
1800	8.458	2285.6	1.245	.4371	574	.00031	5.19	1.493	261.9
1600	4.673	2199.0	1.251	.4280	530	.00028	7.92	1.572	275.9
1200	1.159	2032.4	1.270	.4040	435	.00022	21.96	1.716	301.1
900	.312	1914.7	1.292	.3799	354	.00017	58.00	1.810	317.6
600	.056	1804.9	1.324	.3512	261	.00012	206.97	1.894	332.3
r = 1.40; percent fuel = 29.15; O/F = 2.431									
3600	360.700	3297.5	1.221	0.4875	899	0.00054	-------	-------	-------
3200	189.010	3103.7	1.225	.4815	833	.00049	1.01	0.615	108.9
2800	91.760	2912.5	1.229	.4742	764	.00045	1.18	.953	168.2
2400	40.426	2724.6	1.234	.4649	691	.00040	1.82	1.196	211.7
2000	15.673	2541.0	1.242	.4527	613	.00034	3.37	1.393	246.6
1600	5.100	2363.1	1.254	.4356	527	.00029	7.39	1.560	276.2
1200	1.283	2193.5	1.274	.4112	433	.00023	20.17	1.704	301.8
900	.350	2073.7	1.296	.3867	353	.00018	52.54	1.799	318.6
600	.063	1961.9	1.327	.3580	260	.00012	184.35	1.883	333.5
r = 1.50; percent fuel = 30.59; O/F = 2.269									
3600	388.450	3481.5	1.224	0.4959	894	0.00055	-------	-------	-------
3200	205.010	3284.3	1.228	.4897	829	.00050	1.04	0.561	99.9
2800	100.330	3089.9	1.232	.4822	761	.00045	1.13	.921	164.0
2400	44.604	2898.8	1.238	.4728	688	.00040	1.72	1.171	208.6
2000	17.476	2712.1	1.246	.4604	610	.00035	3.11	1.372	244.5
1600	5.758	2531.2	1.258	.4430	526	.00029	6.73	1.543	274.8
1200	1.471	2358.6	1.277	.4182	432	.00023	18.04	1.689	300.9
900	.407	2236.8	1.300	.3935	352	.00018	46.27	1.785	318.0
600	.075	2123.0	1.331	.3648	260	.00012	159.48	1.870	333.2
r = 1.60; percent fuel = 31.98; O/F = 2.127									
3600	432.810	3670.8	1.228	0.5039	891	0.00055	-------	-------	-------
3200	230.200	3470.5	1.231	.4976	827	.00051	1.13	0.471	84.0
2800	113.650	3272.9	1.236	.4900	758	.00046	1.08	.872	155.7
2400	51.032	3078.7	1.241	.4804	686	.00041	1.58	1.136	202.9
2000	20.226	2889.0	1.250	.4678	609	.00036	2.81	1.344	240.1
1600	6.755	2705.2	1.262	.4502	524	.00030	5.96	1.520	271.4
1200	1.755	2529.8	1.282	.4251	431	.00023	15.66	1.670	298.2
900	.493	2405.9	1.304	.4003	352	.00018	39.50	1.768	315.7
600	.093	2290.1	1.336	.3718	260	.00013	133.60	1.855	331.3
r = 1.80; percent fuel = 34.59; O/F = 1.891									
3200	319.980	3859.7	1.239	0.5128	824	0.00052	-------	-------	-------
2800	160.990	3656.1	1.244	.5049	756	.00048	1.00	0.711	126.3
2400	73.858	3456.0	1.250	.4950	684	.00042	1.29	1.028	182.6
2000	30.011	3260.5	1.258	.4820	607	.00037	2.16	1.263	224.4
1600	10.319	3071.0	1.271	.4641	524	.00031	4.36	1.455	258.6
1200	2.776	2890.2	1.291	.4387	431	.00024	10.93	1.618	287.4
900	.804	2768.2	1.314	.4140	352	.00019	26.59	1.723	306.2
600	.156	2642.1	1.344	.3863	261	.00013	86.41	1.817	322.3
r = 2.00; percent fuel = 37.01; O/F = 1.702									
3200	490.890	4262.9	1.248	0.5274	824	0.00054	-------	-------	-------
2800	251.760	4053.5	1.252	.5192	756	.00049	1.29	0.387	67.4
2400	118.050	3847.8	1.259	.5091	684	.00044	1.06	.859	149.3
2000	49.185	3646.7	1.268	.4958	607	.00038	1.59	1.146	199.9
1600	17.416	3451.8	1.281	.4777	524	.00032	3.02	1.367	238.6
1200	4.850	3265.5	1.301	.4524	432	.00025	7.17	1.550	270.4
900	1.446	3133.4	1.324	.4281	354	.00020	16.75	1.668	290.9
600	.291	3008.9	1.353	.4015	264	.00014	52.28	1.771	308.9

[a]Fuel in propellant, percent by weight.
[b]Oxidant-to-fuel ratio, by weight.

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TABLE III. - Concluded. THEORETICAL ROCKET PERFORMANCE AT ASSIGNED TEMPERATURES FOR JP-4 FUEL AND LIQUID OXYGEN WITH FROZEN COMPOSITION DURING ISENTROPIC PROCESS

(b) Chamber pressure, 600 pounds per square inch absolute.

Temperature, T, °K	Pressure, P, lb/sq in. abs	Enthalpy, h, cal/g	Isentropic exponent, γ	Specific heat, c_p, $\frac{cal}{(g)(^oK)}$	Absolute viscosity μ, micropoises	Thermal conductivity k, $\frac{cal}{(cm)(sec)(^oK)}$	Area ratio, ε	Thrust coefficient, C_F	Specific impulse, I, $\frac{lb\text{-}sec}{lb}$
r = 1.00; percent fuel = 22.71; O/F = 3.403									
4000	1085.000	2707.3	1.207	0.4554	986	0.00055	-------	-------	-------
3600	588.250	2526.0	1.209	.4507	922	.00051	3.33	0.129	22.0
3200	299.050	2346.8	1.212	.4450	854	.00046	1.01	.745	126.2
2800	140.380	2170.2	1.217	.4381	783	.00042	1.35	1.042	177.3
2400	59.544	1996.6	1.222	.4295	708	.00037	2.23	1.268	215.2
2000	22.098	1826.9	1.229	.4182	627	.00032	4.37	1.455	247.6
1600	6.824	1662.6	1.240	.4027	540	.00027	10.20	1.616	275.0
1200	1.609	1505.7	1.258	.3804	443	.00021	29.87	1.756	298.8
900	.412	1394.8	1.279	.3577	361	.00016	83.01	1.848	314.5
600	.069	1291.7	1.310	.3293	265	.00011	315.28	1.930	328.5
r = 1.20; percent fuel = 26.07; O/F = 2.836									
4000	1047.500	3076.8	1.211	0.4748	971	0.00056	-------	-------	-------
3600	573.730	2887.8	1.213	.4700	907	.00052	2.26	0.195	34.1
3200	294.950	2700.9	1.217	.4641	841	.00048	1.01	.753	132.0
2800	140.180	2516.7	1.221	.4570	771	.00043	1.34	1.043	182.9
2400	60.293	2335.6	1.226	.4480	697	.00038	2.21	1.265	221.8
2000	22.739	2158.6	1.234	.4362	618	.00033	4.26	1.449	254.2
1600	7.161	1987.3	1.245	.4198	532	.00028	9.75	1.608	282.0
1200	1.731	1823.8	1.264	.3961	436	.00022	27.86	1.746	306.2
900	.455	1708.4	1.286	.3722	355	.00017	75.57	1.837	322.2
600	.079	1601.0	1.317	.3434	262	.00012	278.10	1.918	336.3
r = 1.30; percent fuel = 27.64; O/F = 2.618									
4000	1068.800	3261.0	1.213	0.4839	965	0.00057	-------	-------	-------
3600	588.850	3068.4	1.216	.4789	902	.00053	3.42	0.126	22.4
3200	304.680	2878.0	1.219	.4730	836	.00048	1.01	.737	130.6
2800	145.860	2690.2	1.224	.4657	766	.00044	1.32	1.031	182.2
2400	63.247	2505.7	1.229	.4566	693	.00039	2.14	1.254	222.4
2000	24.081	2325.3	1.237	.4445	614	.00034	4.07	1.439	255.3
1600	7.669	2150.7	1.248	.4277	529	.00028	9.22	1.598	283.5
1200	1.881	1984.2	1.267	.4035	434	.00022	25.94	1.736	308.0
900	.501	1866.7	1.289	.3791	353	.00017	69.39	1.827	324.1
600	.088	1757.2	1.321	.3502	260	.00012	251.02	1.909	338.5
r = 1.40; percent fuel = 29.15; O/F = 2.431									
3600	622.770	3251.6	1.219	0.4875	897	0.00054	-------	-------	-------
3200	324.560	3057.8	1.222	.4814	831	.00049	1.00	0.704	125.9
2800	156.620	2866.7	1.227	.4740	763	.00044	1.27	1.008	180.2
2400	68.535	2678.8	1.232	.4648	690	.00040	2.03	1.235	221.0
2000	26.369	2495.3	1.240	.4524	611	.00034	3.81	1.423	254.5
1600	8.504	2317.5	1.252	.4353	527	.00029	8.49	1.584	283.3
1200	2.118	2148.1	1.271	.4106	432	.00022	23.49	1.723	308.2
900	.573	2028.5	1.293	.3860	352	.00017	61.87	1.815	324.7
r = 1.60; percent fuel = 31.98; O/F = 2.127									
3600	772.670	3634.2	1.226	0.5038	890	0.00055	-------	-------	-------
3200	409.370	3433.9	1.229	.4975	826	.00051	1.04	0.562	101.2
2800	201.230	3236.4	1.234	.4898	758	.00046	1.13	.920	165.6
2400	89.929	3042.4	1.240	.4802	685	.00041	1.71	1.169	210.5
2000	35.451	2852.7	1.248	.4675	608	.00035	3.08	1.369	246.6
1600	11.767	2669.1	1.260	.4497	524	.00031	6.60	1.539	277.1
1200	3.035	2494.0	1.280	.4245	431	.00023	17.54	1.685	303.4
900	.847	2370.2	1.303	.3995	351	.00018	44.60	1.780	320.6
600	.158	2254.6	1.334	.3709	260	.00013	152.15	1.866	335.9
r = 1.80; percent fuel = 34.59; O/F = 1.891									
3600	1102.400	4043.1	1.234	0.5192	888	0.00057	-------	-------	-------
3200	594.950	3836.7	1.238	.5126	824	.00052	5.09	0.085	15.2
2800	298.610	3633.2	1.243	.5047	756	.00047	1.01	.750	133.9
2400	136.620	3433.2	1.249	.4948	684	.00042	1.35	1.052	188.0
2000	55.344	3237.8	1.257	.4818	607	.00037	2.27	1.281	228.2
1600	18.965	3048.5	1.270	.4637	523	.00031	4.63	1.469	262.3
1200	5.082	2867.8	1.290	.4382	431	.00024	11.69	1.628	290.8
900	1.466	2740.0	1.313	.4135	352	.00019	28.58	1.731	309.3
600	.284	2620.0	1.343	.3857	261	.00013	93.35	1.823	325.7
r = 3.00; percent fuel = 46.85; O/F = 1.134									
1800	852.500	5266.2	1.305	0.5491	567	0.00040	-------	-------	-------
1600	517.330	5157.4	1.312	.5389	526	.00037	1.39	0.362	51.9
1400	297.010	5050.8	1.321	.5272	482	.00033	1.00	.762	109.4
1200	158.910	4946.6	1.333	.5137	436	.00029	1.21	1.010	145.0
1000	77.401	4845.4	1.346	.4987	387	.00025	1.74	1.204	172.2
900	51.524	4795.9	1.354	.4906	361	.00023	2.21	1.288	184.2
600	11.380	4652.3	1.379	.4666	275	.00017	5.70	1.505	216.0
400	2.662	4560.3	1.393	.4545	206	.00013	14.99	1.629	233.2

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TABLE IV. - THEORETICAL ROCKET PERFORMANCE FOR PRESSURE RATIOS BETWEEN 1 AND 8 FOR JP-4 FUEL AND LIQUID OXYGEN WITH FROZEN COMPOSITION DURING ISENTROPIC PROCESS

(a) Chamber pressure, 300 pounds per square inch absolute.

Pressure ratio, P_c/P	Pressure, P, lb/sq in. abs	Temperature, T, °K	Enthalpy, h, cal/g	Specific heat, c_p, $\frac{cal}{(g)(^oK)}$	Isentropic exponent, γ	Thrust coefficient, C_F	Area ratio, ε	Specific impulse, I, $\frac{lb\text{-}sec}{lb}$
r = 1.00; percent fuel[a] = 22.71; O/F[b] = 3.403								
1.000	300.00	3507	2531.6	0.449	1.212	------	------	------
1.020	294.11	3495	2526.1	.449	1.213	0.129	3.329	21.8
1.040	288.47	3483	2520.8	.449	1.213	.182	2.406	30.6
1.200	249.99	3397	2482.0	.448	1.213	.390	1.263	65.7
1.480	202.11	3272	2426.2	.446	1.214	.569	1.032	95.8
1.781	168.43	3168	2380.0	.444	1.215	.682	1.000	114.8
2.230	134.74	3045	2325.4	.442	1.217	.796	1.030	133.9
4.000	75.00	2742	2192.0	.437	1.220	1.021	1.299	171.9
8.000	37.50	2417	2051.4	.430	1.224	1.214	1.926	204.4
r = 1.20; percent fuel = 26.07; O/F = 2.836								
1.000	300.00	3523	2901.1	0.469	1.217	------	------	------
1.020	294.11	3511	2895.3	.469	1.217	0.130	3.333	22.5
1.040	288.47	3499	2889.6	.469	1.217	.182	2.409	31.6
1.200	249.99	3411	2848.4	.467	1.218	.391	1.265	67.8
1.490	201.83	3283	2788.6	.465	1.219	.570	1.032	99.0
1.784	168.20	3177	2739.4	.464	1.220	.684	1.000	118.6
2.230	134.55	3051	2681.4	.462	1.221	.797	1.030	138.3
4.000	75.00	2743	2540.2	.456	1.224	1.022	1.296	177.2
8.000	37.50	2414	2391.1	.448	1.229	1.214	1.919	210.7
r = 1.30; percent fuel = 27.64; O/F = 2.618								
1.000	300.00	3511	3074.1	0.478	1.219	------	------	------
1.020	294.11	3498	3068.2	.477	1.219	0.130	3.336	22.8
1.040	288.47	3486	3062.4	.477	1.219	.182	2.410	32.0
1.200	249.99	3397	3020.1	.476	1.220	.391	1.265	68.6
1.490	201.65	3268	2958.6	.474	1.221	.571	1.032	100.3
1.785	168.03	3161	2908.2	.472	1.222	.685	1.000	120.1
2.230	134.43	3035	2848.9	.470	1.223	.798	1.030	140.0
4.000	75.00	2727	2704.6	.464	1.227	1.022	1.295	179.3
8.000	37.50	2396	2552.3	.457	1.232	1.214	1.915	213.1
r = 1.40; percent fuel = 29.15; O/F = 2.431								
1.000	300.00	3482	3239.9	0.486	1.222	------	------	------
1.020	294.11	3469	3233.9	.486	1.222	0.130	3.338	23.0
1.040	288.47	3457	3227.9	.486	1.222	.183	2.412	32.3
1.200	249.99	3368	3184.8	.484	1.223	.391	1.266	69.3
1.490	201.44	3238	3121.8	.482	1.224	.573	1.032	101.4
1.787	167.86	3131	3070.5	.480	1.225	.686	1.000	121.4
2.230	134.29	3005	3010.1	.478	1.227	.799	1.030	141.4
4.000	75.00	2697	2863.6	.472	1.230	1.022	1.293	181.0
8.000	37.50	2366	2708.8	.464	1.235	1.214	1.910	215.0
r = 1.50; percent fuel = 30.59; O/F = 2.269								
1.000	300.00	3433	3399.0	0.493	1.226	------	------	------
1.020	294.11	3421	3392.8	.493	1.226	0.130	3.342	23.2
1.040	288.47	3409	3386.8	.493	1.226	.183	2.415	32.6
1.200	249.99	3320	3343.1	.492	1.227	.392	1.267	69.7
1.490	201.19	3189	3278.9	.490	1.228	.574	1.032	102.2
1.789	167.65	3083	3226.9	.488	1.229	.687	1.000	122.3
2.240	134.12	2957	3165.8	.485	1.230	.800	1.030	142.4
4.000	75.00	2650	3017.9	.479	1.234	1.022	1.291	182.1
8.000	37.50	2321	2861.6	.471	1.239	1.214	1.905	216.2
r = 1.60; percent fuel = 31.98; O/F = 2.127								
1.000	300.00	3363	3551.6	0.500	1.230	------	------	------
1.020	294.11	3350	3545.4	.500	1.230	0.130	3.346	23.2
1.040	288.47	3338	3539.4	.500	1.230	.183	2.417	32.7
1.200	249.99	3250	3495.3	.498	1.231	.392	1.268	70.0
1.490	200.89	3119	3430.3	.496	1.232	.575	1.032	102.8
1.792	167.42	3014	3378.0	.494	1.233	.688	1.000	122.9
2.240	133.93	2889	3316.5	.492	1.234	.801	1.030	143.0
4.000	75.00	2586	3168.4	.485	1.238	1.022	1.289	182.6
8.000	37.50	2260	3011.7	.476	1.244	1.214	1.898	216.8
r = 1.80; percent fuel = 34.59; O/F = 1.891								
1.000	300.00	3160	3839.4	0.512	1.239	------	------	------
1.020	294.11	3148	3833.2	.512	1.240	0.131	3.355	23.2
1.040	288.47	3137	3827.2	.512	1.240	.184	2.424	32.6
1.200	249.99	3051	3783.4	.510	1.241	.393	1.270	69.8
1.500	200.19	2922	3717.7	.507	1.242	.579	1.031	102.9
1.798	166.83	2820	3666.0	.505	1.243	.691	1.000	122.8
2.250	133.46	2699	3605.1	.503	1.245	.804	1.029	142.8
4.000	75.00	2407	3459.7	.495	1.250	1.023	1.283	181.8
8.000	37.50	2093	3305.6	.485	1.256	1.213	1.882	215.5
r = 2.00; percent fuel = 37.01; O/F = 1.702								
1.000	300.00	2900	4105.8	0.521	1.251			
1.020	294.11	2889	4099.8	.521	1.251	0.131	3.366	22.8
1.040	288.47	2878	4094.0	.521	1.251	.184	2.432	32.1
1.200	249.99	2796	4051.5	.519	1.253	.394	1.274	68.8
1.500	199.35	2671	3986.7	.516	1.254	.584	1.031	101.8
1.806	166.12	2574	3936.7	.514	1.256	.695	1.000	121.3
2.260	132.90	2459	3877.9	.511	1.258	.807	1.029	140.8
4.000	75.00	2185	3739.0	.502	1.263	1.024	1.277	178.6
8.000	37.50	1888	3591.5	.491	1.271	1.213	1.864	211.5

[a]Fuel in propellant, percent by weight.

[b]Oxidant-to-fuel ratio, by weight.

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TABLE IV. - Concluded. THEORETICAL ROCKET PERFORMANCE FOR PRESSURE RATIOS BETWEEN 1 AND 8 FOR JP-4 FUEL AND LIQUID OXYGEN WITH FROZEN COMPOSITION DURING ISENTROPIC PROCESS

(b) Chamber pressure, 600 pounds per square inch absolute.

Pressure ratio, P_c/P	Pressure, P, lb/sq in. abs	Temperature, T, ^{o}K	Enthalpy, h, cal/g	Specific heat, c_p, $\frac{cal}{(g)(^{o}K)}$	Isentropic exponent, γ	Thrust coefficient, C_F	Area ratio, ϵ	Specific impulse, I, $\frac{lb\text{-}sec}{lb}$
r = 1.00; percent fuel = 22.71; O/F = 3.403								
1.000	600.00	3612	2531.6	0.451	1.209	-------	-------	-------
1.020	588.24	3600	2526.0	.451	1.209	0.129	3.326	22.0
1.040	576.92	3588	2520.6	.451	1.209	.182	2.404	31.0
1.200	500.00	3500	2481.0	.449	1.210	.390	1.263	66.3
1.480	404.73	3374	2424.4	.448	1.211	.568	1.032	96.6
1.779	337.27	3268	2377.2	.446	1.212	.681	1.000	115.9
2.220	269.82	3143	2321.4	.444	1.213	.795	1.030	135.2
4.000	150.00	2833	2184.7	.439	1.216	1.021	1.301	173.7
8.000	75.00	2502	2040.7	.432	1.220	1.215	1.931	206.7
r = 1.20; percent fuel = 26.07; O/F = 2.836								
1.000	600.00	3628	2901.1	0.470	1.213	-------	-------	-------
1.020	588.24	3616	2895.2	.470	1.213	0.130	3.330	22.7
1.040	576.92	3604	2889.4	.470	1.213	.182	2.406	31.9
1.200	500.00	3514	2847.3	.469	1.214	.390	1.264	68.4
1.485	404.14	3384	2786.6	.467	1.215	.569	1.032	99.8
1.782	336.79	3276	2736.4	.465	1.216	.683	1.000	119.7
2.226	269.42	3149	2677.2	.463	1.217	.796	1.030	139.6
4.000	150.00	2834	2532.4	.458	1.220	1.021	1.298	179.1
8.000	75.00	2498	2379.8	.450	1.225	1.214	1.925	213.0
r = 1.30; percent fuel = 27.64; O/F = 2.618								
1.000	600.00	3612	3074.1	0.479	1.216	-------	-------	-------
1.020	588.24	3599	3068.1	.479	1.216	0.130	3.332	23.0
1.040	576.92	3587	3062.1	.479	1.216	.182	2.408	32.3
1.200	500.00	3497	3019.0	.477	1.217	.390	1.264	69.3
1.490	403.76	3366	2956.6	.476	1.218	.570	1.032	101.1
1.783	336.46	3258	2905.2	.474	1.219	.684	1.000	121.2
2.230	269.17	3129	2844.6	.472	1.220	.797	1.030	141.3
4.000	150.00	2814	2696.9	.466	1.223	1.021	1.297	181.2
8.000	75.00	2477	2541.0	.459	1.228	1.214	1.920	215.4
r = 1.40; percent fuel = 29.15; O/F = 2.431								
1.000	600.00	3576	3239.9	0.487	1.219	-------	-------	-------
1.020	588.24	3563	3233.7	.487	1.219	0.130	3.336	23.2
1.040	576.92	3551	3227.7	.487	1.219	.182	2.410	32.6
1.200	500.00	3461	3183.8	.486	1.220	.391	1.265	69.9
1.490	403.30	3329	3119.9	.484	1.221	.571	1.032	102.2
1.785	336.08	3220	3067.6	.482	1.222	.685	1.000	122.4
2.230	268.88	3092	3006.0	.480	1.223	.798	1.030	142.7
4.000	150.00	2778	2856.1	.474	1.227	1.022	1.295	182.7
8.000	75.00	2441	2697.9	.466	1.231	1.214	1.915	217.2
r = 1.60; percent fuel = 31.98; O/F = 2.127								
1.000	600.00	3436	3551.6	0.501	1.227	-------	-------	-------
1.020	588.24	3423	3545.3	.501	1.227	0.130	3.343	23.4
1.040	576.92	3411	3539.2	.501	1.227	.183	2.416	32.9
1.200	500.00	3321	3494.5	.499	1.228	.392	1.267	70.5
1.490	402.13	3189	3428.6	.497	1.230	.574	1.032	103.4
1.790	335.11	3082	3375.6	.495	1.231	.687	1.000	123.8
2.240	268.08	2956	3313.0	.493	1.232	.800	1.030	144.1
4.000	150.00	2648	3162.2	.486	1.236	1.022	1.290	184.1
8.000	75.00	2317	3002.6	.478	1.241	1.214	1.902	218.6
r = 1.80; percent fuel = 34.59; O/F = 1.891								
1.000	600.00	3205	3839.4	0.513	1.238	-------	-------	-------
1.020	588.24	3193	3833.1	.513	1.238	0.130	3.354	23.3
1.040	576.92	3181	3827.0	.512	1.238	.183	2.423	32.8
1.200	500.00	3095	3782.8	.511	1.239	.393	1.270	70.2
1.500	400.60	2965	3716.6	.508	1.241	.579	1.031	103.3
1.797	333.83	2862	3664.3	.506	1.242	.691	1.000	123.4
2.250	267.07	2740	3602.7	.503	1.243	.803	1.029	143.5
4.000	150.00	2445	3455.5	.496	1.248	1.023	1.284	182.8
8.000	75.00	2128	3299.6	.486	1.254	1.213	1.885	216.7
r = 3.00; percent fuel = 46.85; O/F = 1.134								
1.000	600.00	1657	5188.4	0.542	1.310	-------	-------	-------
1.020	588.24	1650	5184.2	.542	1.310	0.133	3.422	19.1
1.040	576.92	1642	5180.1	.541	1.311	.187	2.471	26.9
1.200	500.00	1587	5150.4	.538	1.313	.400	1.289	57.5
1.540	390.21	1496	5101.4	.533	1.317	.606	1.028	87.0
1.845	325.18	1431	5067.2	.529	1.320	.715	1.000	102.7
2.310	260.13	1355	5027.3	.524	1.324	.825	1.027	118.4
4.000	150.00	1183	4937.8	.512	1.334	1.029	1.246	147.6
8.000	75.00	992	4841.4	.498	1.347	1.211	1.776	173.8

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TABLE V. - THEORETICAL ROCKET PERFORMANCE FOR PRESSURE RATIOS BETWEEN 10 AND 1500 FOR JP-4 FUEL AND LIQUID OXYGEN WITH FROZEN COMPOSITION DURING ISENTROPIC PROCESS

(a) Chamber pressure, 300 pounds per square inch absolute.

Pressure ratio, P_o/P	Pressure, P, lb/sq in. abs	Temperature, T, ^oK	Temperature exponent, n_T	Enthalpy, h, cal/g	Specific heat, c_p, $\frac{cal}{(g)(^oK)}$	Isentropic exponent, γ	Thrust coefficient, C_F	Area-ratio exponent, n_ε	Area ratio, ε	Specific-impulse exponent, n_I	Specific impulse, I, lb-sec/lb
r = 1.00; percent fuel[a] = 22.71; O/F[b] = 3.403											
10	30.00	2320	0.0509	2009.8	0.427	1.226	1.266	0.0049	2.22	0.0161	213.1
15	20.00	2152	.0525	1938.4	.423	1.229	1.350	.0062	2.89	.0165	227.2
20	15.00	2039	.0537	1891.0	.419	1.231	1.403	.0072	3.52	.0167	236.1
30	10.00	1889	.0555	1828.3	.414	1.235	1.470	.0086	4.66	.0170	247.4
40	7.50	1788	.0567	1786.7	.411	1.237	1.513	.0096	5.72	.0172	254.6
60	5.00	1654	.0586	1731.7	.405	1.241	1.567	.0118	7.65	.0175	263.8
80	3.75	1563	.0599	1695.3	.401	1.244	1.603	.0123	9.44	.0178	269.8
100	3.00	1496	.0610	1668.4	.398	1.247	1.628	.0133	11.11	.0179	274.1
150	2.00	1380	.0631	1622.6	.392	1.252	1.671	.0150	14.98	.0182	281.2
200	1.50	1302	.0646	1592.2	.387	1.255	1.699	.0163	18.53	.0185	285.9
300	1.00	1198	.0668	1552.3	.381	1.261	1.734	.0182	25.06	.0188	291.9
400	.75	1128	.0684	1526.0	.376	1.265	1.758	.0196	31.05	.0190	295.8
600	.50	1035	.0708	1491.5	.369	1.271	1.787	.0217	42.03	.0193	300.8
800	.37	973	.0725	1468.7	.364	1.276	1.807	.0232	52.12	.0195	304.1
1000	.30	927	.0738	1452.0	.360	1.280	1.821	.0244	61.58	.0196	306.5
1500	.20	848	.0763	1423.7	.354	1.286	1.845	.0266	83.38	.0199	310.5
r = 1.20; percent fuel = 26.07; O/F = 2.836											
10	30.00	2315	0.0505	2347.1	0.446	1.230	1.266	0.0049	2.21	0.0161	219.6
15	20.00	2145	.0521	2271.6	.441	1.234	1.349	.0061	2.88	.0164	234.0
20	15.00	2031	.0533	2221.5	.437	1.236	1.402	.0071	3.50	.0166	243.2
30	10.00	1879	.0550	2155.4	.432	1.240	1.468	.0085	4.63	.0169	254.7
40	7.50	1777	.0562	2111.5	.428	1.242	1.511	.0095	5.68	.0171	262.1
60	5.00	1641	.0581	2053.7	.422	1.247	1.565	.0111	7.59	.0174	271.7
80	3.75	1549	.0595	2015.4	.417	1.250	1.600	.0122	9.35	.0177	277.8
100	3.00	1482	.0605	1987.1	.414	1.253	1.626	.0131	11.00	.0178	282.2
150	2.00	1364	.0626	1939.0	.407	1.258	1.668	.0149	14.81	.0181	289.4
200	1.50	1286	.0641	1907.2	.402	1.262	1.695	.0162	18.30	.0183	294.1
300	1.00	1181	.0663	1865.4	.395	1.268	1.730	.0181	24.71	.0186	300.2
400	.75	1111	.0679	1837.9	.390	1.272	1.753	.0195	30.59	.0188	304.2
600	.50	1018	.0702	1801.9	.383	1.279	1.783	.0215	41.34	.0191	309.3
800	.37	956	.0719	1778.2	.378	1.284	1.802	.0230	51.20	.0193	312.6
1000	.30	909	.0733	1760.9	.374	1.288	1.816	.0242	60.43	.0195	315.0
1500	.20	830	.0758	1731.5	.367	1.295	1.839	.0264	81.68	.0198	319.0
r = 1.30; percent fuel = 27.64; O/F = 2.618											
10	30.00	2297	0.0489	2507.4	0.454	1.233	1.265	0.0047	2.20	0.0156	222.1
15	20.00	2127	.0505	2430.4	.449	1.237	1.349	.0059	2.87	.0159	236.7
20	15.00	2012	.0516	2379.3	.445	1.239	1.401	.0068	3.48	.0162	245.9
30	10.00	1860	.0532	2311.9	.440	1.243	1.467	.0082	4.61	.0165	257.5
40	7.50	1758	.0544	2267.2	.435	1.246	1.510	.0092	5.65	.0167	265.0
60	5.00	1622	.0562	2208.4	.429	1.250	1.564	.0107	7.55	.0170	274.5
80	3.75	1531	.0575	2169.5	.424	1.254	1.599	.0118	9.29	.0172	280.6
100	3.00	1463	.0586	2140.8	.421	1.257	1.624	.0127	10.93	.0173	285.0
150	2.00	1346	.0606	2091.9	.414	1.262	1.666	.0144	14.70	.0176	292.3
200	1.50	1267	.0620	2059.7	.409	1.266	1.693	.0156	18.16	.0178	297.1
300	1.00	1163	.0642	2017.4	.401	1.272	1.728	.0175	24.49	.0181	303.2
400	.75	1093	.0657	1989.5	.396	1.277	1.750	.0188	30.29	.0183	307.2
600	.50	1000	.0680	1953.1	.388	1.284	1.780	.0208	40.90	.0186	312.3
800	.37	938	.0696	1929.2	.383	1.289	1.798	.0222	50.61	.0188	315.6
1000	.30	892	.0709	1911.7	.379	1.293	1.812	.0234	59.71	.0189	318.0
1500	.20	813	.0733	1882.0	.372	1.300	1.835	.0255	80.61	.0192	322.1
r = 1.40; percent fuel = 29.15; O/F = 2.431											
10	30.00	2267	0.0458	2663.2	0.074	1.237	1.265	0.0043	2.20	0.0148	224.0
15	20.00	2097	.0473	2585.1	.456	1.240	1.348	.0055	2.86	.0151	238.7
20	15.00	1983	.0483	2533.3	.452	1.243	1.400	.0063	3.47	.0153	248.0
30	10.00	1831	.0499	2465.0	.446	1.247	1.466	.0076	4.59	.0155	259.7
40	7.50	1729	.0510	2419.8	.442	1.250	1.509	.0085	5.62	.0157	267.1
60	5.00	1594	.0527	2360.4	.435	1.254	1.562	.0099	7.50	.0160	276.6
80	3.75	1503	.0539	2321.0	.430	1.258	1.597	.0110	9.22	.0162	282.8
100	3.00	1435	.0549	2292.1	.427	1.261	1.622	.0118	10.84	.0164	287.2
150	2.00	1319	.0567	2242.8	.419	1.267	1.663	.0134	14.57	.0166	294.6
200	1.50	1241	.0581	2210.3	.414	1.271	1.690	.0145	17.99	.0168	299.3
300	1.00	1137	.0601	2167.8	.406	1.278	1.725	.0163	24.23	.0171	305.4
400	.75	1068	.0615	2139.8	.401	1.282	1.747	.0175	29.95	.0173	309.4
600	.50	976	.0636	2103.2	.393	1.290	1.776	.0193	40.39	.0175	314.5
800	.37	914	.0651	2079.2	.388	1.295	1.795	.0207	49.93	.0177	317.8
1000	.30	869	.0663	2061.7	.384	1.299	1.808	.0217	58.87	.0178	320.2
1500	.20	791	.0684	2032.0	.377	1.306	1.831	.0236	79.37	.0181	324.2

[a]Fuel in propellant, percent by weight.

[b]Oxidant-to-fuel ratio, by weight.

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TABLE V. - Continued. THEORETICAL ROCKET PERFORMANCE FOR PRESSURE RATIOS BETWEEN 10 AND 1500 FOR JP-4 FUEL AND LIQUID OXYGEN WITH FROZEN COMPOSITION DURING ISENTROPIC PROCESS

(a) Concluded. Chamber pressure, 300 pounds per square inch absolute.

Pressure ratio, P_c/P	Pressure, P, lb/sq in. abs	Temperature, T, °K	Temperature exponent, n_T	Enthalpy, h, cal/g	Specific heat, c_p, cal/(g)(°K)	Isentropic exponent, γ	Thrust coefficient, C_F	Area-ratio exponent, n_ϵ	Area ratio, ϵ	Specific-impulse exponent, n_I	Specific impulse, I, lb-sec/lb
r = 1.50; percent fuel = 30.59; O/F = 2.269											
10	30.00	2223		2815.6	0.468	1.241	1.265		2.19		225.3
15	20.00	2054		2736.9	.462	1.244	1.347		2.85		240.0
20	15.00	1940		2684.7	.458	1.247	1.399		3.45		249.3
30	10.00	1790		2616.1	.452	1.251	1.465		4.56		261.0
40	7.50	1689		2570.6	.447	1.255	1.507		5.58		268.5
60	5.00	1554		2511.0	.441	1.260	1.560		7.44		278.0
80	3.75	1464		2471.5	.436	1.263	1.595		9.15		284.1
100	3.00	1397		2442.5	.431	1.267	1.619		10.74		288.5
150	2.00	1282		2393.2	.424	1.273	1.661		14.42		295.8
200	1.50	1205		2360.7	.419	1.277	1.687		17.78		300.6
300	1.00	1103		2318.3	.411	1.284	1.721		23.93		306.6
400	.75	1034		2290.4	.405	1.289	1.743		29.54		310.6
600	.50	944		2254.0	.397	1.296	1.772		39.78		315.6
800	.37	883		2230.2	.392	1.302	1.790		49.14		318.9
1000	.30	838		2212.7	.388	1.306	1.803		57.88		321.3
1500	.20	762		2183.3	.381	1.313	1.826		77.92		325.2
r = 1.60; percent fuel = 31.98; O/F = 2.127											
10	30.00	2163	0.0366	2965.7	0.473	1.246	1.264	0.0034	2.18	0.0121	225.8
15	20.00	1996	.0378	2886.4	.468	1.250	1.347	.0043	2.83	.0123	240.5
20	15.00	1884	.0386	2834.2	.463	1.253	1.398	.0049	3.43	.0125	249.8
30	10.00	1735	.0398	2766.3	.457	1.257	1.464	.0059	4.53	.0127	261.4
40	7.50	1635	.0407	2721.1	.452	1.261	1.505	.0067	5.54	.0128	268.8
60	5.00	1503	.0420	2661.3	.445	1.266	1.558	.0078	7.37	.0131	278.3
80	3.75	1414	.0430	2622.2	.440	1.270	1.592	.0086	9.05	.0132	284.4
100	3.00	1348	.0438	2593.8	.435	1.273	1.617	.0093	10.63	.0133	288.7
150	2.00	1235	.0452	2544.6	.428	1.280	1.657	.0105	14.24	.0135	296.0
200	1.50	1159	.0463	2512.7	.422	1.284	1.684	.0114	17.55	.0137	300.7
300	1.00	1059	.0478	2470.6	.414	1.291	1.717	.0127	23.57	.0139	306.7
400	.75	992	.0489	2443.3	.408	1.297	1.739	.0137	29.07	.0140	310.6
600	.50	903	.0505	2407.4	.401	1.304	1.767	.0151	39.08	.0142	315.6
800	.37	844	.0517	2383.2	.395	1.310	1.785	.0161	48.21	.0143	318.8
1000	.30	801	.0525	2366.6	.391	1.314	1.798	.0169	56.73	.0144	321.1
1500	.20	726	.0541	2337.8	.384	1.321	1.820	.0183	76.25	.0146	325.0
r = 1.80; percent fuel = 34.59; O/F = 1.891											
10	30.00	2000	0.0239	3260.4	0.482	1.258	1.263	0.0022	2.16	0.0081	224.4
15	20.00	1839	.0246	3183.5	.476	1.263	1.345	.0027	2.80	.0083	238.9
20	15.00	1732	.0252	3132.7	.471	1.266	1.396	.0032	3.38	.0084	248.0
30	10.00	1589	.0259	3066.1	.464	1.271	1.460	.0038	4.45	.0085	259.4
40	7.50	1494	.0265	3022.2	.458	1.275	1.501	.0043	5.43	.0086	266.7
60	5.00	1368	.0274	2964.9	.450	1.281	1.553	.0050	7.21	.0087	275.9
80	3.75	1284	.0280	2927.1	.445	1.286	1.586	.0055	8.83	.0088	281.7
100	3.00	1221	.0285	2899.5	.440	1.290	1.610	.0060	10.34	.0089	286.0
150	2.00	1114	.0294	2852.7	.432	1.297	1.649	.0067	13.81	.0090	293.0
200	1.50	1042	.0300	2822.0	.426	1.302	1.675	.0073	16.98	.0091	297.5
300	1.00	948	.0310	2782.2	.418	1.310	1.707	.0081	22.72	.0092	303.3
400	.75	885	.0317	2756.1	.413	1.315	1.728	.0087	27.94	.0093	307.0
600	.50	803	.0326	2722.3	.405	1.323	1.755	.0096	37.42	.0094	311.8
800	.37	748	.0333	2700.2	.400	1.328	1.772	.0101	46.03	.0095	314.8
1000	.30	707	.0338	2684.2	.396	1.332	1.785	.0106	54.06	.0096	317.0
1500	.20	639	.0347	2657.2	.390	1.340	1.805	.0114	72.38	.0097	320.7
r = 2.00; percent fuel = 37.01; O/F = 1.702											
10	30.00	1800		3548.5	0.487	1.273	1.262		2.13		220.2
15	20.00	1649		3475.3	.480	1.279	1.343		2.76		234.2
20	15.00	1548		3427.2	.475	1.283	1.393		3.33		243.0
30	10.00	1415		3364.4	.467	1.289	1.456		4.36		254.0
40	7.50	1326		3323.1	.461	1.294	1.496		5.30		261.0
60	5.00	1209		3269.4	.453	1.300	1.547		7.02		269.8
80	3.75	1130		3234.2	.447	1.306	1.579		8.57		275.4
100	3.00	1073		3208.5	.443	1.310	1.602		10.02		279.4
150	2.00	974		3165.1	.434	1.317	1.640		13.32		286.1
200	1.50	908		3136.8	.429	1.323	1.665		16.32		290.4
300	1.00	822		3100.1	.421	1.331	1.696		21.75		295.8
400	.75	765		3076.3	.416	1.336	1.716		26.68		299.3
600	.50	690		3045.4	.409	1.343	1.741		35.58		303.8
800	.37	641		3025.4	.405	1.348	1.758		43.64		306.6
1000	.30	605		3010.8	.402	1.352	1.769		51.14		308.7
1500	.20	544		2986.5	.397	1.358	1.789		68.23		312.1

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TABLE V. - Continued. THEORETICAL ROCKET PERFORMANCE FOR PRESSURE RATIOS BETWEEN 10 AND 1500 FOR JP-4 FUEL AND LIQUID OXYGEN WITH FROZEN COMPOSITION DURING ISENTROPIC PROCESS

(b) Chamber pressure, 600 pounds per square inch absolute.

Pressure ratio, P_c/P	Pressure, P, lb/sq in. abs	Temperature, T, °K	Temperature exponent, n_T	Enthalpy, h, cal/g	Specific heat, c_p, $\frac{cal}{(g)(^{\circ}K)}$	Isentropic exponent, γ	Thrust coefficient, C_F	Area-ratio exponent, n_ε	Area ratio, ε	Specific-impulse exponent, n_I	Specific impulse, I, lb-sec/lb
r = 1.00; percent fuel = 22.71; O/F = 3.403											
10	60.00	2403	0.0509	1998.0	0.430	1.222	1.266	0.0049	2.22	0.0161	215.5
15	40.00	2232	.0525	1924.8	.425	1.225	1.350	.0062	2.90	.0165	229.8
20	30.00	2117	.0537	1876.0	.422	1.227	1.404	.0072	3.53	.0167	238.8
30	20.00	1963	.0555	1811.5	.417	1.230	1.471	.0086	4.69	.0170	250.3
40	15.00	1860	.0567	1768.6	.413	1.232	1.514	.0096	5.76	.0172	257.7
60	10.00	1722	.0586	1718.0	.408	1.236	1.569	.0112	7.71	.0175	267.0
80	7.50	1629	.0599	1674.5	.404	1.239	1.605	.0123	9.52	.0178	273.1
100	6.00	1561	.0610	1646.7	.401	1.242	1.631	.0133	11.21	.0179	277.5
150	4.00	1441	.0631	1599.3	.395	1.246	1.674	.0150	15.13	.0182	284.8
200	3.00	1361	.0646	1567.9	.390	1.250	1.702	.0163	18.74	.0185	289.6
300	2.00	1255	.0668	1526.5	.384	1.255	1.738	.0182	25.37	.0188	295.7
400	1.50	1183	.0684	1499.2	.379	1.259	1.761	.0196	31.47	.0190	299.7
600	1.00	1087	.0708	1463.3	.373	1.265	1.792	.0217	42.67	.0193	304.9
800	.75	1024	.0725	1439.6	.368	1.269	1.812	.0232	52.96	.0195	308.2
1000	.60	976	.0738	1422.3	.364	1.273	1.826	.0244	62.63	.0196	310.7
1500	.40	894	.0763	1392.7	.357	1.279	1.850	.0266	84.94	.0199	314.8
r = 1.20; percent fuel = 26.07; O/F = 2.836											
10	60.00	2398	0.0505	2334.6	0.448	1.226	1.266	0.0049	2.22	0.0161	222.0
15	40.00	2224	.0521	2257.2	.443	1.229	1.350	.0061	2.89	.0164	236.7
20	30.00	2107	.0533	2205.7	.440	1.232	1.403	.0071	3.51	.0166	246.0
30	20.00	1952	.0550	2137.7	.434	1.235	1.470	.0085	4.66	.0169	257.7
40	15.00	1847	.0562	2092.5	.431	1.238	1.512	.0095	5.71	.0171	265.3
60	10.00	1708	.0581	2032.9	.425	1.242	1.567	.0109	7.65	.0174	274.9
80	7.50	1615	.0595	1993.4	.420	1.245	1.602	.0122	9.43	.0177	281.0
100	6.00	1545	.0605	1964.3	.417	1.247	1.628	.0131	11.10	.0178	285.5
150	4.00	1425	.0626	1914.5	.410	1.252	1.671	.0148	14.96	.0181	293.0
200	3.00	1344	.0641	1881.6	.406	1.256	1.698	.0162	18.51	.0183	297.9
300	2.00	1237	.0663	1838.4	.399	1.262	1.734	.0181	25.02	.0186	304.1
400	1.50	1165	.0679	1809.8	.394	1.266	1.757	.0195	31.00	.0188	308.2
600	1.00	1069	.0702	1772.4	.386	1.272	1.787	.0215	41.96	.0191	313.4
800	.75	1004	.0719	1747.8	.381	1.277	1.806	.0230	52.02	.0193	316.8
1000	.60	957	.0733	1729.7	.377	1.281	1.820	.0242	61.46	.0195	319.3
1500	.40	875	.0758	1699.0	.370	1.288	1.844	.0264	83.19	.0198	323.4
r = 1.30; percent fuel = 27.64; O/F = 2.618											
10	60.00	2377	0.0489	2495.0	0.456	1.229	1.266	0.0047	2.21	0.0156	224.5
15	40.00	2202	.0505	2416.0	.451	1.232	1.349	.0059	2.88	.0159	239.3
20	30.00	2086	.0516	2363.5	.447	1.235	1.402	.0068	3.50	.0162	248.7
30	20.00	1930	.0532	2294.3	.442	1.238	1.469	.0082	4.64	.0165	260.5
40	15.00	1826	.0544	2248.3	.438	1.241	1.511	.0092	5.69	.0167	268.1
60	10.00	1686	.0562	2187.8	.432	1.245	1.566	.0107	7.60	.0170	277.7
80	7.50	1593	.0575	2147.7	.427	1.249	1.601	.0118	9.37	.0172	283.9
100	6.00	1523	.0586	2118.1	.424	1.251	1.626	.0127	11.02	.0173	288.4
150	4.00	1403	.0606	2067.6	.417	1.256	1.668	.0144	14.84	.0176	295.9
200	3.00	1323	.0620	2034.3	.412	1.260	1.696	.0156	18.36	.0178	300.8
300	2.00	1216	.0642	1990.5	.405	1.266	1.731	.0175	24.79	.0181	307.1
400	1.50	1144	.0657	1961.6	.399	1.271	1.754	.0188	30.69	.0183	311.1
600	1.00	1048	.0680	1923.9	.392	1.277	1.784	.0208	41.49	.0186	316.4
800	.75	984	.0696	1899.0	.387	1.282	1.803	.0222	51.40	.0188	319.8
1000	.60	937	.0709	1880.8	.382	1.286	1.817	.0234	60.68	.0189	322.2
1500	.40	855	.0733	1849.9	.375	1.293	1.840	.0255	82.04	.0192	326.4

TABLE V. - Concluded. THEORETICAL ROCKET PERFORMANCE FOR PRESSURE RATIOS BETWEEN 10 AND 1500 FOR JP-4 FUEL AND LIQUID OXYGEN WITH FROZEN COMPOSITION DURING ISENTROPIC PROCESS

(b) Concluded. Chamber pressure, 600 pounds per square inch absolute.

Pressure ratio, P_c/P	Pressure, P, lb/sq in. abs	Temperature, T, °K	Temperature exponent, n_T	Enthalpy, h, cal/g	Specific heat, c_p, $\frac{cal}{(g)(°K)}$	Isentropic exponent, γ	Thrust coefficient, C_F	Area-ratio exponent, n_ϵ	Area ratio, ϵ	Specific-impulse exponent, n_I	Specific impulse, I, lb-sec/lb
r = 1.40; percent fuel = 29.15; O/F = 2.431											
10	60.00	2341	0.0458	2651.2	0.463	1.233	1.265	0.0043	2.20	0.0148	226.3
15	40.00	2167	.0473	2571.3	.458	1.236	1.349	.0055	2.87	.0151	241.2
20	30.00	2050	.0483	2518.2	.454	1.239	1.401	.0063	3.49	.0153	250.6
30	20.00	1895	.0499	2448.1	.449	1.243	1.467	.0076	4.61	.0155	262.5
40	15.00	1791	.0510	2401.7	.444	1.245	1.510	.0085	5.65	.0157	270.1
60	10.00	1653	.0527	2340.6	.438	1.250	1.564	.0099	7.55	.0160	279.7
80	7.50	1560	.0539	2300.2	.433	1.253	1.599	.0110	9.29	.0162	286.0
100	6.00	1491	.0549	2270.4	.429	1.256	1.624	.0118	10.93	.0164	290.5
150	4.00	1372	.0567	2219.6	.422	1.262	1.666	.0134	14.71	.0166	298.0
200	3.00	1292	.0581	2186.0	.417	1.266	1.693	.0145	18.17	.0168	302.8
300	2.00	1185	.0601	2142.1	.410	1.272	1.728	.0163	24.50	.0171	309.1
400	1.50	1114	.0615	2113.1	.404	1.277	1.751	.0175	30.31	.0173	313.1
600	1.00	1020	.0636	2075.3	.396	1.284	1.780	.0193	40.93	.0175	318.3
800	.75	956	.0651	2050.4	.391	1.289	1.799	.0207	50.65	.0177	321.7
1000	.60	910	.0663	2032.2	.387	1.293	1.812	.0217	59.76	.0178	324.2
1500	.40	829	.0684	2001.3	.379	1.300	1.835	.0236	80.68	.0181	328.3
r = 1.60; percent fuel = 31.98; O/F = 2.127											
10	60.00	2218	0.0366	2955.7	0.475	1.243	1.265	0.0034	2.18	0.0121	227.7
15	40.00	2048	.0378	2875.4	.469	1.247	1.347	.0043	2.84	.0123	242.6
20	30.00	1935	.0386	2822.2	.465	1.249	1.399	.0049	3.44	.0125	251.9
30	20.00	1783	.0398	2752.3	.459	1.254	1.465	.0059	4.55	.0127	263.7
40	15.00	1682	.0407	2706.1	.454	1.257	1.506	.0067	5.56	.0128	271.2
60	10.00	1547	.0420	2645.3	.447	1.262	1.559	.0078	7.41	.0131	280.8
80	7.50	1457	.0430	2605.2	.442	1.266	1.594	.0086	9.11	.0132	287.0
100	6.00	1390	.0438	2575.8	.438	1.269	1.618	.0093	10.70	.0133	291.4
150	4.00	1274	.0452	2525.6	.430	1.275	1.659	.0105	14.35	.0135	298.8
200	3.00	1197	.0463	2492.7	.424	1.280	1.686	.0114	17.69	.0137	303.6
300	2.00	1094	.0478	2449.6	.416	1.287	1.720	.0127	23.78	.0139	309.7
400	1.50	1026	.0489	2421.3	.411	1.292	1.742	.0137	29.35	.0140	313.6
600	1.00	935	.0505	2384.4	.403	1.300	1.770	.0151	39.49	.0142	318.7
800	.75	875	.0517	2360.2	.397	1.305	1.788	.0161	48.75	.0143	322.0
1000	.60	830	.0525	2342.6	.393	1.309	1.801	.0169	57.40	.0144	324.3
1500	.40	754	.0541	2312.8	.386	1.317	1.823	.0183	77.23	.0146	328.3
r = 1.80; percent fuel = 34.59; O/F = 1.891											
10	60.00	2033	0.0239	3253.8	0.483	1.256	1.264	0.0022	2.16	0.0081	225.7
15	40.00	1871	.0246	3175.9	.477	1.261	1.345	.0027	2.80	.0083	240.3
20	30.00	1762	.0252	3124.4	.472	1.264	1.396	.0032	3.39	.0084	249.4
30	20.00	1618	.0259	3056.9	.465	1.269	1.461	.0038	4.46	.0085	260.9
40	15.00	1522	.0265	3012.4	.459	1.273	1.502	.0043	5.45	.0086	268.2
60	10.00	1394	.0274	2954.2	.452	1.279	1.554	.0050	7.23	.0087	277.5
80	7.50	1309	.0280	2915.9	.446	1.284	1.587	.0055	8.86	.0088	283.5
100	6.00	1245	.0285	2887.8	.442	1.287	1.611	.0060	10.39	.0089	287.7
150	4.00	1137	.0294	2840.3	.433	1.294	1.651	.0067	13.88	.0090	294.8
200	3.00	1064	.0300	2809.1	.428	1.299	1.676	.0073	17.06	.0091	299.4
300	2.00	969	.0310	2768.5	.420	1.307	1.709	.0081	22.84	.0092	305.2
400	1.50	905	.0317	2742.0	.414	1.312	1.730	.0087	28.11	.0093	309.0
600	1.00	821	.0326	2707.5	.406	1.320	1.757	.0096	37.67	.0094	313.8
800	.75	765	.0333	2685.1	.401	1.326	1.774	.0101	46.36	.0095	316.9
1000	.60	724	.0338	2668.7	.397	1.330	1.787	.0106	54.46	.0096	319.2
1500	.40	654	.0347	2641.2	.391	1.337	1.808	.0114	72.96	.0097	322.9
r = 3.00; percent fuel = 46.85; O/F = 1.134											
10	60.00	936		4813.8	0.494	1.351	1.258		2.02		180.5
15	40.00	842		4767.6	.486	1.359	1.333		2.57		191.3
20	30.00	780		4737.7	.481	1.364	1.380		3.06		198.0
30	20.00	700		4699.3	.474	1.371	1.437		3.96		206.3
40	15.00	647		4674.4	.470	1.375	1.473		4.76		211.5
60	10.00	579		4642.6	.465	1.381	1.518		6.20		217.9
80	7.50	535		4622.0	.462	1.384	1.547		7.49		222.0
100	6.00	503		4607.2	.460	1.387	1.567		8.69		224.9
150	4.00	449		4582.5	.457	1.391	1.600		11.39		229.6
200	3.00	414		4566.6	.455	1.393	1.621		13.83		232.6

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TABLE VI. - THEORETICAL ROCKET PERFORMANCE FOR COMPLETE EXPANSION TO EXIT PRESSURE OF 1 ATMOSPHERE FOR JP-4 FUEL AND LIQUID OXYGEN

[Frozen composition during isentropic process.]

Equivalence ratio, r, $\frac{4(C)+(H)}{2(O)}$	Percent fuel by weight	Oxidant-to-fuel weight ratio, O/F	Combustion temperature, T_c, °K	Exit temperature, T_e, °K	Characteristic velocity, c*, ft/sec	Thrust coefficient, C_F	Area ratio, ε	Specific impulse, I, $\frac{\text{lb-sec}}{\text{lb}}$
Combustion-chamber pressure, 300 lb/sq in. abs								
1.00	22.71	3.403	3507	2032	5415	1.406	3.57	236.7
1.20	26.07	2.836	3523	2023	5582	1.405	3.55	243.8
1.30	27.64	2.618	3511	2004	5647	1.405	3.53	246.5
1.40	29.15	2.431	3482	1975	5697	1.404	3.52	248.6
1.50	30.59	2.269	3433	1933	5732	1.403	3.50	249.9
1.60	31.98	2.127	3363	1876	5746	1.402	3.48	250.4
1.80	34.59	1.891	3160	1724	5716	1.399	3.43	248.6
2.00	37.01	1.702	2900	1541	5613	1.396	3.37	243.6
Combustion-chamber pressure, 600 lb/sq in. abs								
1.00	22.71	3.403	3612	1853	5475	1.517	5.84	258.2
1.20	26.07	2.836	3628	1840	5643	1.515	5.80	265.8
1.30	27.64	2.618	3612	1818	5707	1.514	5.77	268.6
1.40	29.15	2.431	3576	1784	5755	1.513	5.73	270.6
1.50	30.59	2.269	3518	1737	5785	1.511	5.69	271.7
1.60	31.98	2.127	3436	1675	5794	1.509	5.64	271.8
1.80	34.59	1.891	3205	1515	5747	1.504	5.52	268.7
2.00	37.01	1.702	2923	1333	5630	1.499	5.39	262.3
3.00	46.85	1.134	1657	644	4618	1.476	4.83	211.8

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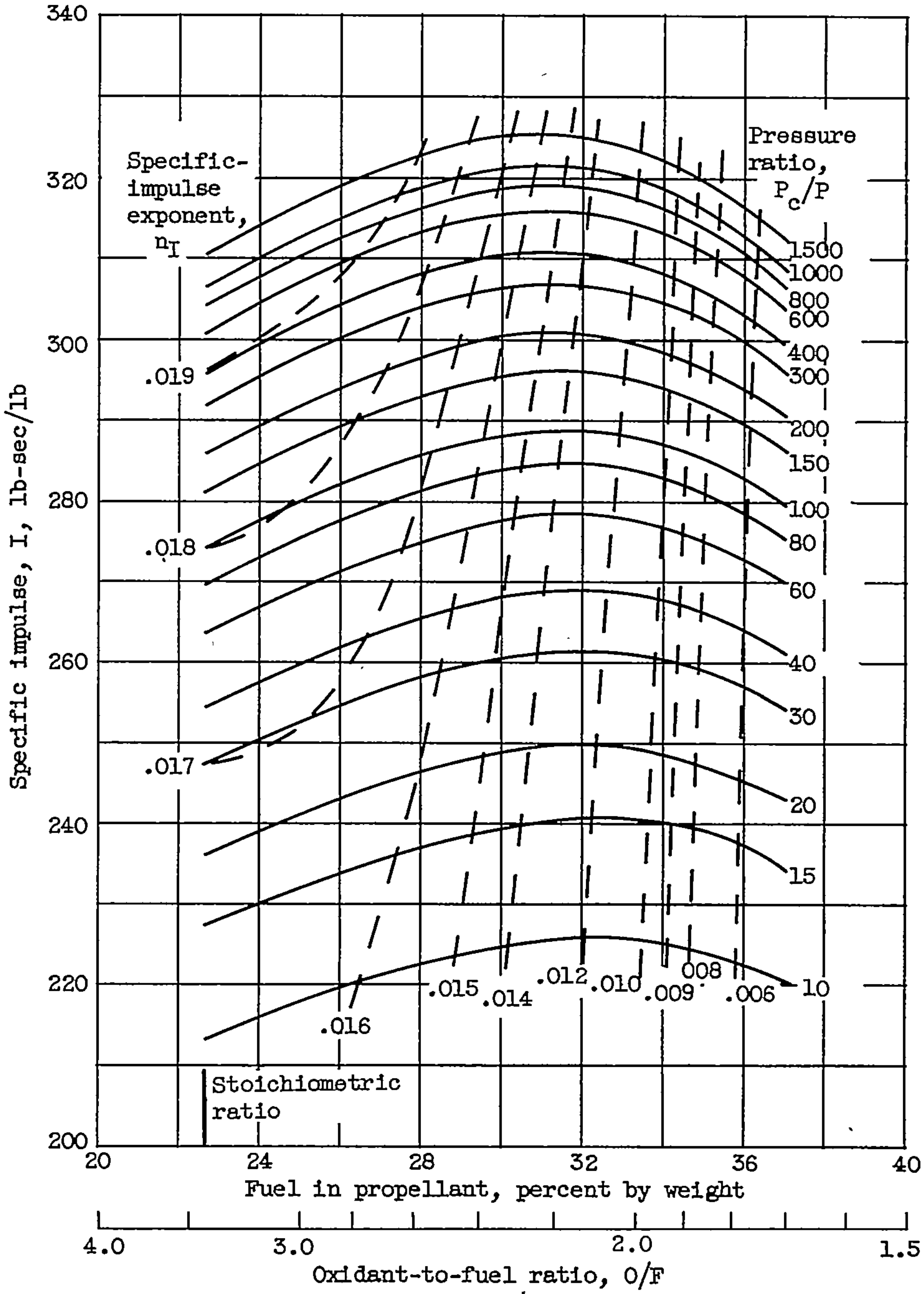
CO-4 back



(a) Chamber pressure, 300 pounds per square inch absolute.

Exponent n_I for use in equation $I = I_{300}\left(\frac{P_c}{300}\right)^{n_I}$.

Figure 1. - Theoretical specific impulse of JP-4 fuel with liquid oxygen. Frozen composition during isentropic expansion to pressure ratio indicated.

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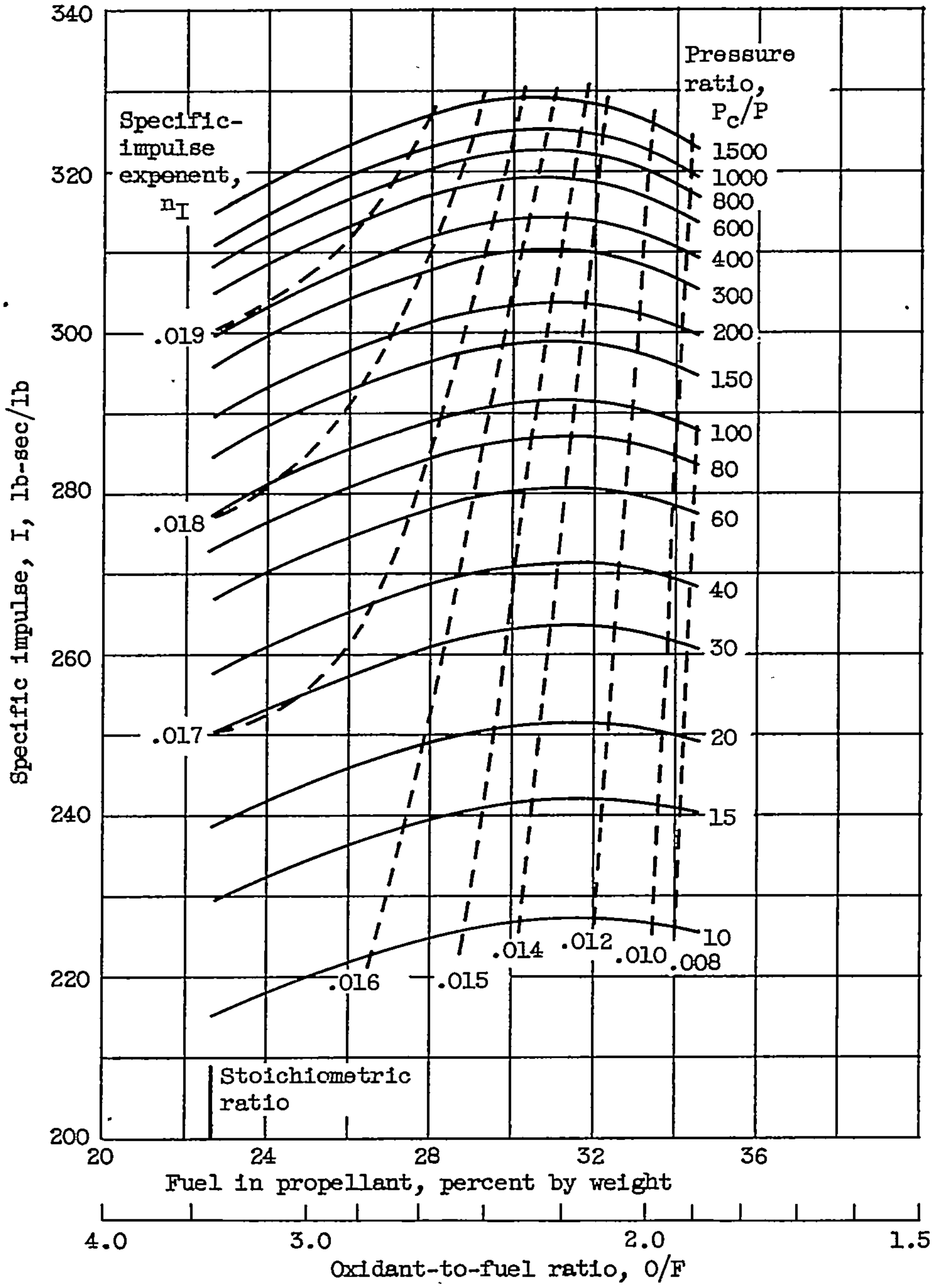

(b) Chamber pressure, 600 pounds per square inch absolute.

Exponent n_I for use in equation $I = I_{600}\left(\frac{P_c}{600}\right)^{n_I}$.

Figure 1. - Concluded. Theoretical specific impulse of JP-4 fuel with liquid oxygen. Frozen composition during isentropic expansion to pressure ratio indicated.

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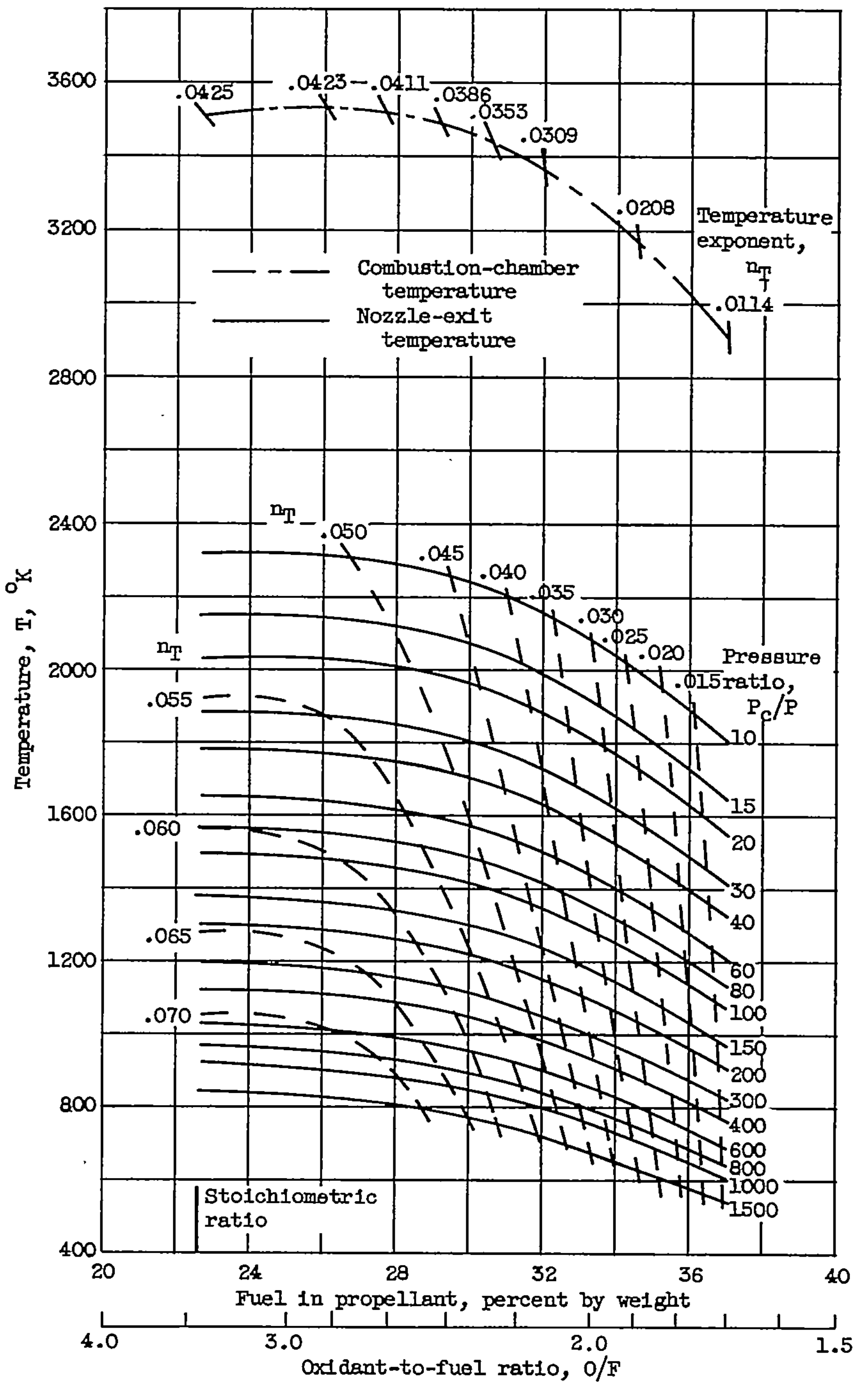

(a) Chamber pressure, 300 pounds per square inch absolute.

Exponent n_T for use in equation $T = T_{300}\left(\frac{P_c}{300}\right)^{n_T}$.

Figure 2. - Theoretical combustion-chamber temperature and nozzle-exit temperature of JP-4 fuel with liquid oxygen. Frozen composition during isentropic expansion to pressure ratio indicated.

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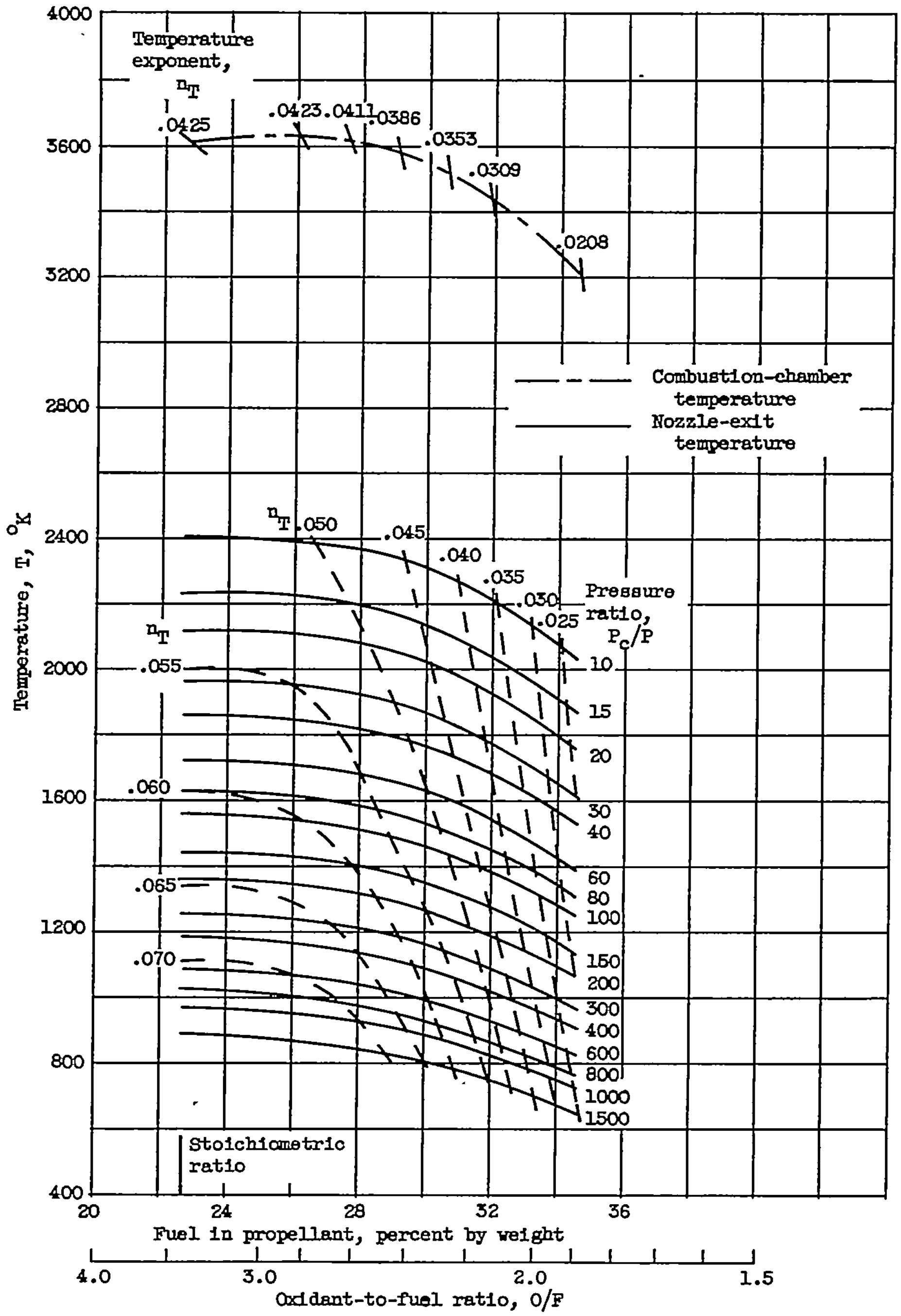

(b) Chamber pressure, 600 pounds per square inch absolute. Exponent n_T for use in equation $T = T_{600}\left(\frac{P_c}{600}\right)^{n_T}$.

Figure 2. - Concluded. Theoretical combustion-chamber temperature and nozzle-exit temperature of JP-4 fuel with liquid oxygen. Frozen composition during isentropic expansion to pressure ratio indicated.

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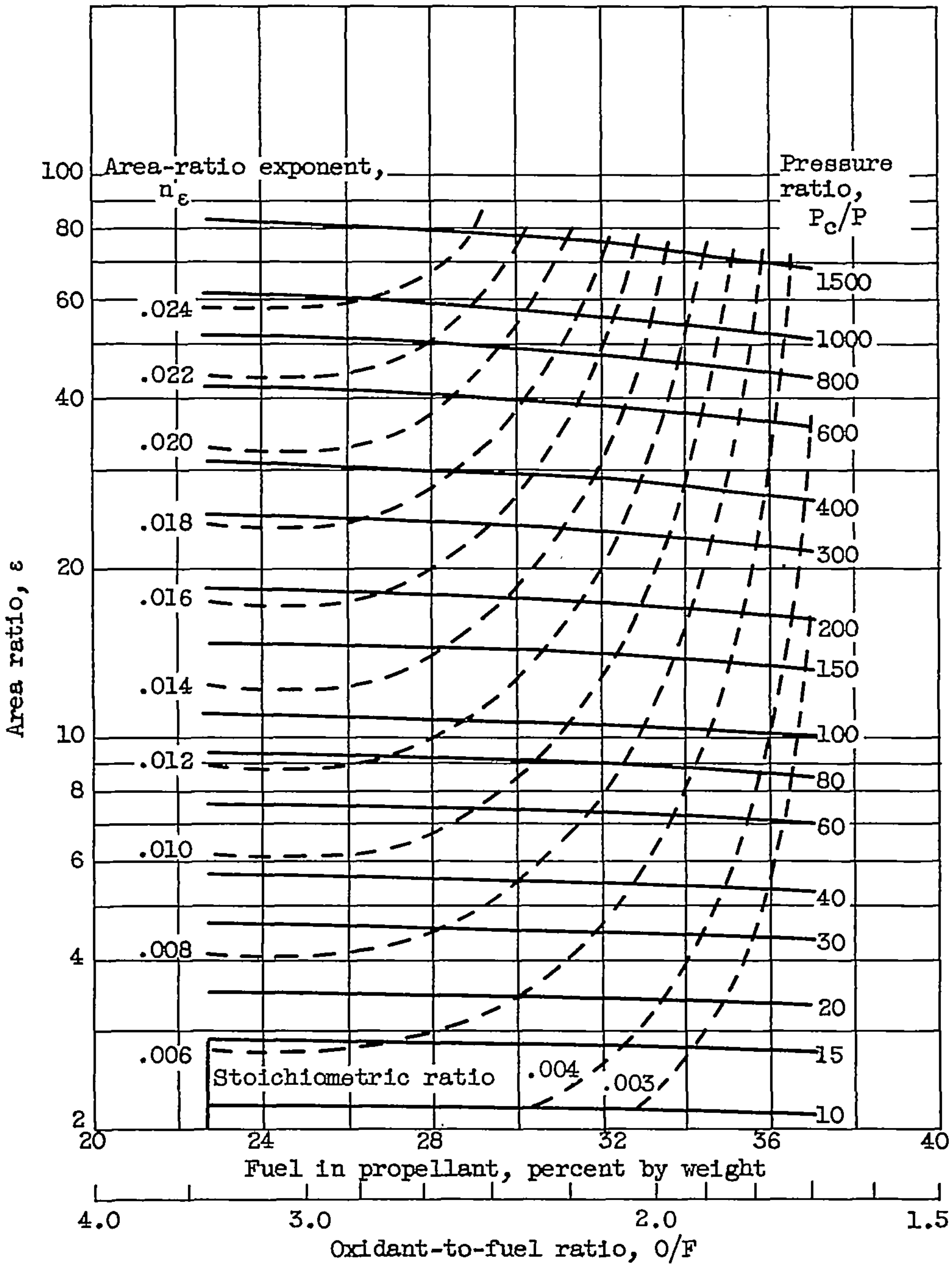

(a) Chamber pressure, 300 pounds per square inch absolute.

Exponent n_ε for use in equation $\varepsilon = \varepsilon_{300}\left(\frac{P_c}{300}\right)^{n_\varepsilon}$.

Figure 3. - Theoretical ratio of nozzle area to throat area for JP-4 fuel with liquid oxygen. Frozen composition during isentropic expansion to pressure ratio indicated.

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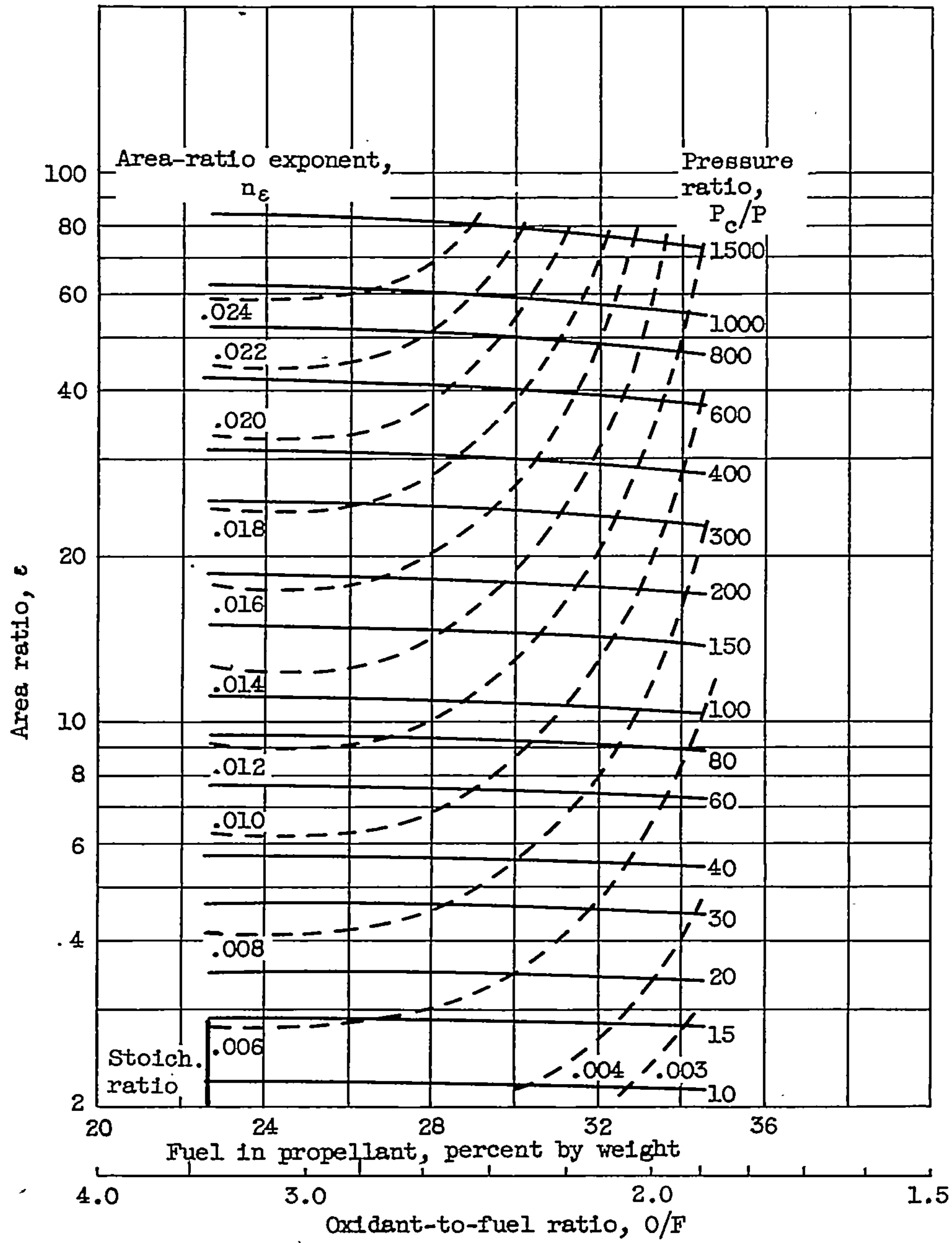
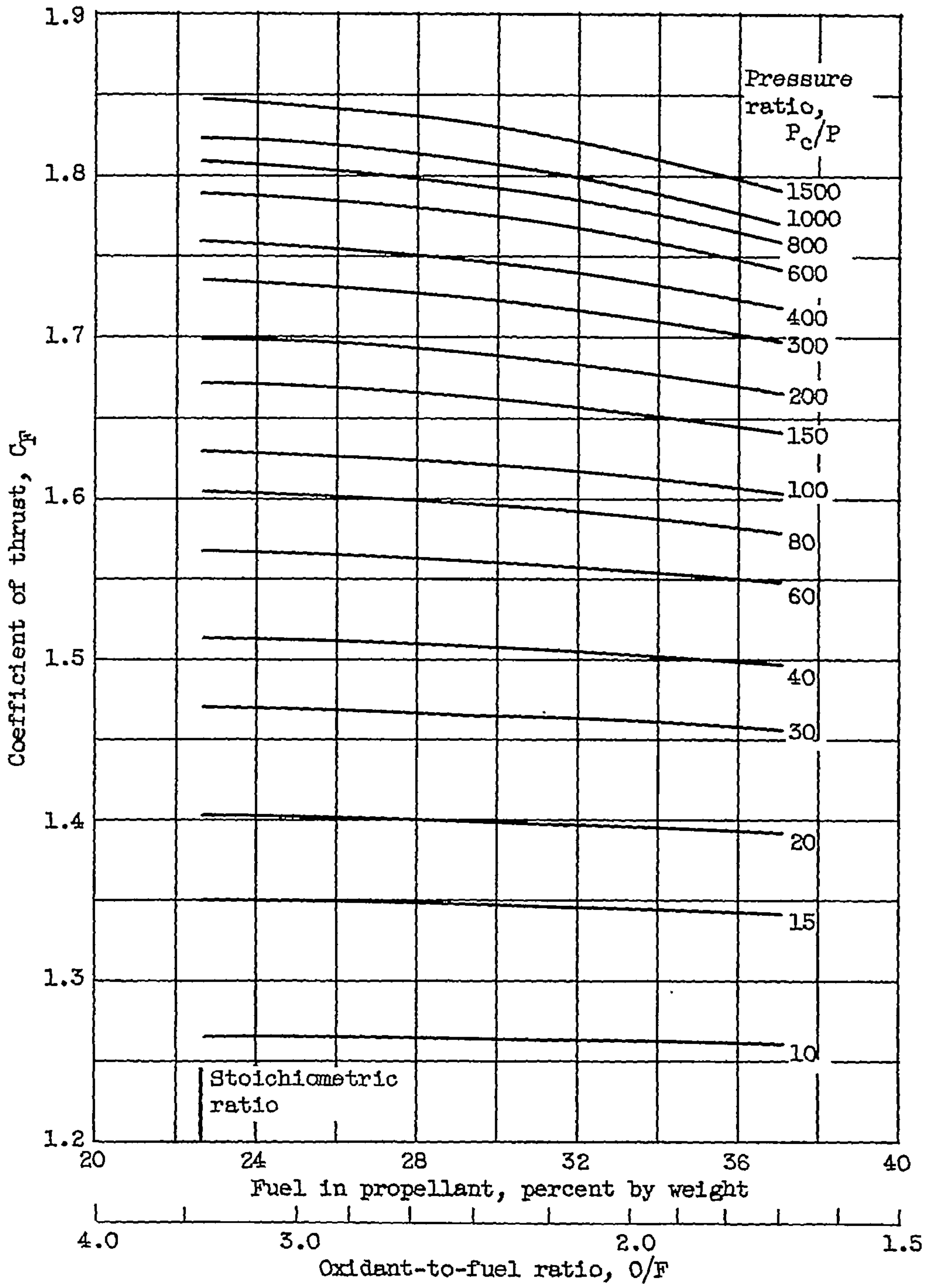


(b) Chamber pressure, 600 pounds per square inch absolute.

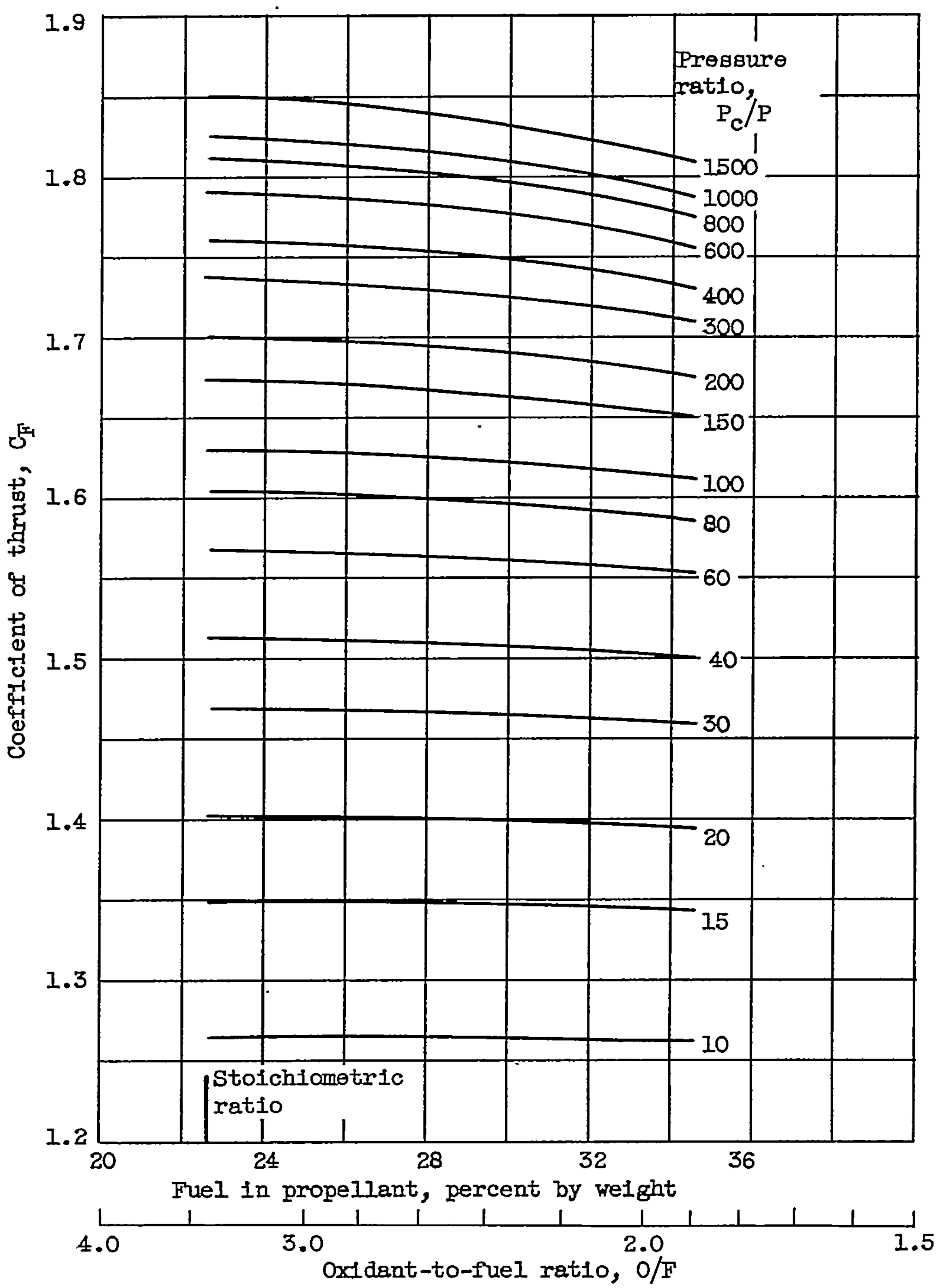
Exponent n_ε for use in equation $\varepsilon = \varepsilon_{600}\left(\frac{P_c}{600}\right)^{n_\varepsilon}$.

Figure 3. - Concluded. Theoretical ratio of nozzle area to throat area for JP-4 fuel with liquid oxygen. Frozen composition during isentropic expansion to pressure ratio indicated.



(a) Chamber pressure, 300 pounds per square inch absolute.

Figure 4. - Theoretical coefficient of thrust for JP-4 fuel with liquid oxygen. Frozen composition during isentropic expansion to pressure ratio indicated.



(b) Chamber pressure, 600 pounds per square inch absolute.

Figure 4. - Concluded. Theoretical coefficient of thrust for JP-4 fuel with liquid oxygen. Frozen composition during isentropic expansion to pressure ratio indicated.

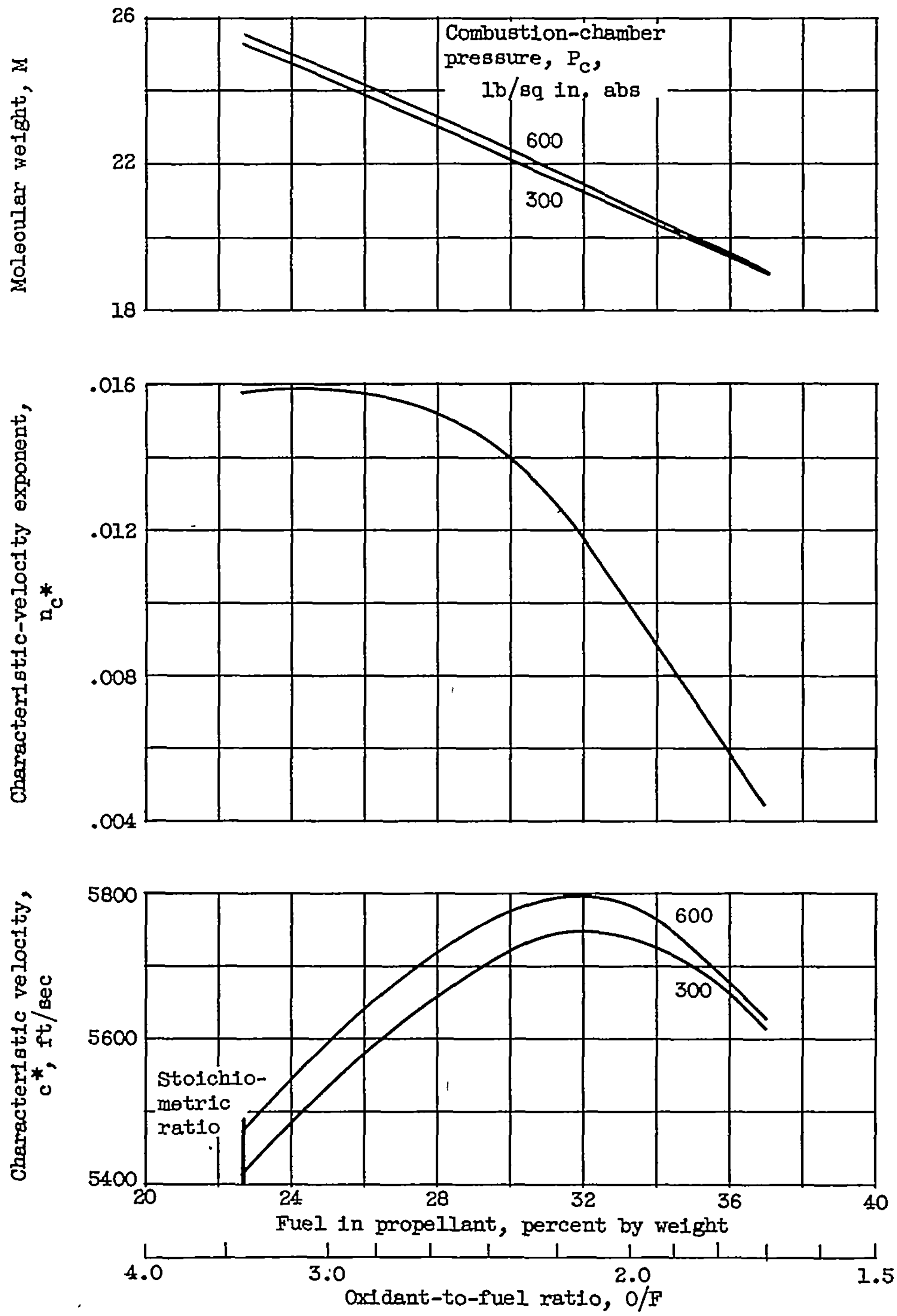


Figure 5. - Theoretical molecular weight, characteristic-velocity exponent and characteristic velocity. Exponent n_{c^*} for use in equation $c^* = c^*_{300}\left(\frac{P_c}{300}\right)^{n_{c^*}}$. Frozen composition during isentropic expansion from chamber pressure indicated.