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RESEARCH MEMORANDUM

ANALYSIS OF RAM-JET ENGINE PERFORMANCE INCLUDING EFFECTS OF COMPONENT CHANGES

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NATIONAL ADVISORY COMMITTEE
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SUMMARY

Ram-jet engine performance data are presented over a range of engine design variables to aid in the selection and evaluation of a ram-jet engine configuration with particular emphasis on one suitable for a long-range supersonic missile. Calculated design-point performance of engines using JP-4 fuel is presented for a wide range of engine total-temperature ratios and combustion-chamber-inlet Mach numbers for flight Mach numbers from 1.5 to 4.0. The results include engine thrust, drag, fuel consumption, and area ratios, and are presented both with and without nacelle drag included. Maximum engine fuel specific impulse (including nacelle drag) is 1600 to 1700 pound-seconds per pound and occurs at a flight Mach number of 2.5 to 3.0. Over-all engine efficiency, however, continues to increase to a flight Mach number of 4.0, where it is 35 to 40 percent.

Important gains in both thrust coefficient and specific impulse may be achieved by improving the diffuser pressure recovery. Changes in flameholder pressure loss and combustor length have only small effects on engine performance. That these factors, however, influence combustion efficiency is significant, because the specific impulse varies directly with the efficiency, although the thrust coefficient is practically unaffected. Engine performance is very sensitive to changes in nozzle performance, a 1-percent variation in velocity coefficient often producing a 3-percent variation in engine thrust and specific impulse. Some underexpansion of the exhaust gases is desirable to reduce nacelle drag whenever the nozzle-exit diameter exceeds that of the combustion chamber.

With a fixed-geometry configuration, a ram-jet engine does not operate satisfactorily at off-design conditions. Somewhat better thrust can be obtained with the added complication of a translating-spike diffuser, although the specific impulse is poorer. Use of a movable plug to vary the throat area of the exhaust nozzle yields both thrust and specific impulse approaching those of a continuously variable geometry engine.

INTRODUCTION

The suitability of the ram-jet engine for the propulsion of high-speed aircraft has been generally recognized and accepted. The selection of the proper engine for such applications must be based on many factors relating to the airframe-engine combination. The purpose of this report is to present calculated ram-jet engine performance data over a range of engine design parameters in order to aid in the selection and evaluation of an engine design. General emphasis is placed on designs suitable for a long-range supersonic missile. A second purpose is to illustrate the relative importance of the different engine parameters that influence the design of the ram-jet engine. Many other thermodynamic cycle studies of ram-jet engines are presented in the literature (e.g., refs. 1 and 2). This report presents a wider range of operating conditions, uses somewhat more advanced, but realistic, component characteristics, and demonstrates the effect of changes in these component characteristics.

The engine performance data are presented such that they may be used either with or without nacelle drag included and are therefore suitable for aircraft configurations with either internal or external engine installations.

The report has three major sections:

(1) General design-point engine performance is presented for a wide range of engine total-temperature ratios and combustion-chamber-inlet Mach numbers for flight Mach numbers from 1.5 to 4.0. The fuel used is JP-4, and nominal assumptions are used for the component characteristics.

(2) The sensitivity of these design-point results to changes in the nominal assumptions is indicated by showing the effect of varying the different component parameters, one at a time, on the performance of selected engines. The parameters investigated are diffuser pressure recovery, flameholder pressure loss, combustion efficiency, nozzle velocity coefficient, nozzle expansion ratio, nozzle jet-deflection angle, and altitude. Calculations were also made comparing a high-energy fuel (pentaborane) with JP-4.

(3) Off-design performance is presented for an engine designed for efficient cruising at a flight Mach number of 3.5. Configurations having fixed geometry, continuously variable geometry, and two types of practically variable geometry are compared. Off-design performance is not emphasized, because results obtained in missile studies indicate that self-boosting capabilities are not important for ram-jet-powered long-range missiles using rocket boosters.

ANALYSIS

The symbols used in this report are defined in appendix A.

A schematic diagram of a ram-jet engine is shown in figure 1. High-speed air enters the engine at station 1 and is decelerated to a low velocity at 2. Fuel is added, ignition takes place, and combustion is stabilized at the flameholders, between stations 2 and 3. Combustion occurs in a constant-area duct from stations 3 to 4, and the hot gases are expanded and discharged through a convergent-divergent nozzle (stations 4 to 6).

The performance calculations were made on the basis of one-dimensional flow, using the equations of state, continuity, and conservation of momentum and energy. Reference 2 presents equations similar to those used in the present analysis, in which the values of γ are based on the gas static temperature and composition at each station. The problem of specifying the gas properties is discussed more fully in appendix B, and the assumed engine geometry and methods used in calculating the engine drags are detailed in appendix C. Engine performance data, which are generally presented for an altitude of 70,000 feet, may be used for any altitude in the isothermal region of the atmosphere with negligible error.

Engine performance is presented in this report in terms of the following:

(1) Propulsive thrust coefficient C_T, defined as engine thrust minus nacelle drag per unit cross-sectional area, divided by free-stream incompressible dynamic pressure. The cross-sectional area used is the diffuser capture area or the combustion-chamber frontal area, whichever is larger. This coefficient is a measure of the engine size required to produce a given amount of propulsive thrust.

(2) Specific impulse I, defined as engine thrust minus nacelle drag divided by engine fuel-flow rate. At any given flight speed, this parameter is a measure of the efficiency with which thrust is produced.

Similarly, the net-thrust coefficient and specific impulse C_F and I_F are defined as above except that nacelle drag is not included.

Most of the various engine parameters fall into two groups: (A) those of major importance that cannot be finally specified without a complete flight analysis, and (B) those that can be realistically chosen from an engine study alone, or that are limited by what can practically be achieved. These major variables (group A) were taken as flight Mach number, combustion-chamber-inlet Mach number, and engine total-temperature ratio. A complete set of design-point performance calculations was obtained for different values of these variables, based on nominal assumptions made for the component parameters of group B. The range of calculations for these major variables included flight Mach numbers from 1.5 to 4.0, combustion-chamber-inlet Mach numbers from 0.125 to 0.225, and engine total-temperature ratios corresponding to fuel-air ratios of approximately 0.01 to stoichiometric, except where limited by thermal choking.

The following assumptions were used for the variables of group B in the design-point calculations:

(1) Engines were considered with both single- and double-cone diffusers. The design-point engines operated critically (i.e., with the normal shock located at the inlet lip). For the single-cone diffuser, cone angle was varied with design flight speed to achieve maximum pressure recovery. For the two-cone diffuser, the cone angles were not selected for maximum pressure recovery but chosen to permit use of a low-drag cowl (low lip angle). Figure 2 shows the assumed variation of pressure recovery with flight Mach number for the two diffuser types. The illustrated single-cone values are in good agreement with the experimental data for similar inlets reported in reference 3. Reference 4 shows that, in the present state of inlet development, the engine performance obtained with the single-cone inlet with low drag cowl is as good as that afforded by more elaborate diffusers such as the isentropic spike.

(2) Flameholder total-pressure loss was taken as twice the incompressible dynamic pressure at station 2. Combustion of the fuel (JP-4 with a lower heating value H of 18,640 Btu/lb) took place from stations 3 to 4 with an assumed efficiency of 0.90. The resulting relation between τ (engine total-temperature ratio) and fuel-air ratio f/a for different flight Mach numbers is shown in figure 3. Reference 5 reports the achievement of about 0.95 efficiency with the same amount of flameholder loss in tests of a 16-inch combustor at a combustor pressure of about 1 atmosphere.

(3) The nozzle velocity coefficient was taken as 0.975. Values of this magnitude have been obtained experimentally for convergent-divergent nozzles at nozzle pressure ratios P_4/p_6 of about 15 (ref. 6). At flight Mach numbers of 2.5 and higher, the nozzle-exit diameter is generally the largest diameter of the engine. To reduce nacelle drag in these cases, the nozzle expansion ratio was made less than that required for complete expansion of the gases to ambient pressure. From unpublished

data obtained in long-range missile studies showing the effect of expansion ratio on missile range, the optimum expansion ratio is roughly generalized by the empirical formula

$$\frac{A_6}{A_3} = 1 + 0.55\left[\left(\frac{A_6}{A_3}\right)_{p_6=p_0} - 1\right] \tag{1}$$

where $\left(\frac{A_6}{A_3}\right)_{p_6=p_0}$ is the ratio of nozzle-exit to combustion-chamber area for complete expansion. This expression was used for the design-point engine calculations whenever the nozzle-exit diameter exceeded that of the combustion chamber. In all other cases the nozzle was made completely expanding.

The sensitivity of the design-point results to changes in these assumed values of the various component parameters was indicated by calculating the effect of varying these parameters, one at a time, at flight Mach numbers of 2.5 and 3.5. At each speed, two values of τ were considered, a low value for good cruising performance and a higher value to give increased thrust for acceleration.

The off-design performance of engines designed for cruising at a flight Mach number of 3.5 was also evaluated. Engines equipped with the following features were considered:

(1) Continuously variable diffuser and nozzle

(2) Variable-throat-area nozzle with fixed diffuser

(3) Translating-spike diffuser with fixed nozzle

(4) Fixed diffuser and nozzle

RESULTS AND DISCUSSION

Design-Point Performance

The calculated design-point values of propulsive thrust coefficient and specific impulse are shown in figure 4 for the single-cone diffuser as functions of flight Mach number M_0, ratio of combustion-chamber-exit to -inlet total temperature τ, and combustion-chamber-inlet Mach number M_2. These data, as well as engine area ratios and drag coefficients, are listed in table I. Similar data are listed in table II for the two-cone diffuser. The values of C_T and I include nacelle drag. The corresponding values without nacelle drag may be obtained by the relations

$$C_F = C_T + C_D \tag{2}$$

$$I_F = I\frac{C_F}{C_T} \tag{3}$$

Performance of engines having velocity coefficients other than 0.975 may be calculated from the following formula:

$$C_T = \frac{C_V}{0.975}\left[(C_T)_{0.975} + C_D + 2\left(\frac{A_1}{A_3}\right)\right] - C_D - 2\left(\frac{A_1}{A_3}\right) \tag{4}$$

which is based on the assumption that the jet thrust is directly proportional to the velocity coefficient and that the nacelle drag does not change. The change in I is directly proportional to the change in C_T.

Performance can also be computed for values of combustion efficiency other than 0.90. At any constant value of τ, the thrust coefficient remains essentially constant with changes in combustion efficiency, and specific impulse and fuel-air ratio are given by

$$I = \frac{\eta_c}{0.90} I_{0.90} \tag{5}$$

$$\frac{f}{a} = \frac{\eta_c}{0.90}\left(\frac{f}{a}\right)_{0.90} \tag{6}$$

The effect of changes in diffuser pressure recovery may be approximated by the following expressions:

$$C_T = \frac{F - D}{q_0 A_3} = \left(C_T' + \frac{A_6/A_3}{\frac{\gamma}{2} M_0^2} + C_D\right)\frac{P_2/P_0}{(P_2/P_0)'} + \frac{A_6/A_3}{\frac{\gamma}{2} M_0^2}\left(\frac{p_6}{p_0} - 1\right) - C_D \tag{7}$$

$$I = I' \frac{C_T}{C_T'} \frac{(P_2/P_0)'}{P_2/P_0} \tag{8}$$

$$A_1/A_3 = \frac{(A_1/A_3)'(P_2/P_0)}{(P_2/P_0)'} \tag{9}$$

in which $(P_2/P_0)'$ denotes the single-cone pressure recoveries given in figure 2, and C_T' , I', and $(A_1/A_3)'$ are the values listed in table I. Equations (7) to (9) are based on the assumptions that the inlet capture area is varied with pressure recovery and nacelle drag and nozzle exit area are constant. The data of table I and equations (7) to (9) were used to compute the performance presented in table II for the low-cowl-drag two-cone diffuser. (Note that eqs. (7) and (8) require that C_T be based on A_3.)

The remaining discussion, except where noted, is based on the low-cowl-drag single-cone diffuser. Although the actual level of performance may be somewhat different with other diffuser types, all the trends are expected to be the same.

Figure 4 shows that high thrusts are obtained at the high values of τ and maximum specific impulses at intermediate values of τ. Raising τ (at a constant M_0 and M_2) increases the exit momentum of the gases, mainly because of the higher jet velocity and, to a lesser degree, because of the increased fuel mass flow. However, as τ is raised, the fuel flow increases at a greater rate than the jet thrust, so that, after the constant loss of the inlet momentum drag is sufficiently overcome, the specific impulse reaches a maximum and then decreases. When nacelle drag is included, the value of τ for maximum specific impulse is raised.

The effect of flight Mach number on over-all engine efficiency and specific impulse is shown in figure 5, in which the value of τ is varied to provide maximum I and E at each flight speed. Combustion-chamber-inlet Mach number is generally 0.200, except for M_0 above 3.5, where it was necessary to reduce M_2 to prevent the diffuser-inlet diameter from exceeding that of the combustion chamber. Maximum I (1600 lb-sec/lb including nacelle drag) occurs at M_0 of 2.5. This flight Mach number, however, may not be optimum for a missile, because missile range is more nearly related to the over-all engine efficiency, which in turn is proportional to the product of specific impulse and flight velocity rather than to specific impulse alone. Maximum E of the order of 0.35 is realized at M_0 near 4, while the highest efficiency obtainable at M_0 of 2.5 is only 27 percent. For engines with a two-cone diffuser, the maximum values of I and E are 1700 and 40, respectively. (Still higher values of E would be expected for M_0 greater than 4.)

The effect of combustion-chamber-inlet Mach number M_2 is indicated in figure 4 but is more readily apparent in a cross plot of some of these data (fig. 6). The thrust coefficient increases with M_2 because of the essentially linear increase in air flow to the point where the inlet capture area becomes equal to the combustion-chamber area. The value of M_2 at which these areas are equal is higher than the region of interest shown in figure 6, but is a function of the flight speed and the diffuser pressure recovery. Raising M_2 also increases both the flameholder pressure loss and the momentum pressure loss due to heat addition. If nacelle drag is not included, the specific impulse decreases with increasing values of M_2. However, higher values of M_2 reduce the diameter of the combustor and nozzle relative to the diffuser capture area and so result in lower nacelle drag per pound of air (provided the capture area remains smaller than the combustion-chamber area). Because of these two opposing effects, the specific impulse including drag is fairly insensitive to variations in M_2 for the conditions of figure 6.

Effect of Variations in Design-Point Assumptions

Diffuser pressure recovery. - Diffuser total-pressure ratio is used in this report as a measure of the efficiency with which the diffuser converts the kinetic energy of the captured airstream to pressure. Lines of constant kinetic-energy efficiency superimposed on the curve of pressure recovery against flight Mach number (fig. 2) show that the lower numerical values of total-pressure ratio at high flight Mach numbers do not necessarily mean lower diffuser efficiency.

The nominal diffuser assumed for the design-point calculations is an oblique-shock inlet with a single-cone spike centerbody. Figure 7 shows the effect on engine performance of changes in the assumed values of pressure recovery. (This performance is based on the drag of a low-angle cowl at all pressure recoveries.) The engine air flow per unit combustion-chamber area increases linearly with pressure recovery, so that the propulsive thrust coefficient (based on combustion-chamber area) also increases nearly linearly. As pressure recovery is increased, the diffuser capture area enlarges relative to the combustion chamber in order to handle these larger air flows at constant M_2. At high flight Mach numbers and high pressure recoveries, the resulting capture area often becomes greater than the combustion-chamber area. The size of the engine required to produce a given thrust is then indicated by basing the thrust coefficient on diffuser capture area. At a flight Mach number of 2.5, the combustion chamber is always the larger for the range of pressure recoveries considered. At a flight Mach number of 3.5 and M_2 of 0.20, the capture area becomes the larger at pressure recoveries greater than 0.43, which causes the sharp break in thrust coefficient observed at this point in figure 7. The engine specific impulse increases with increasing pressure recovery because of the higher pressure ratio across the exhaust nozzle. The higher air flow also improves the specific impulse because of the lower nacelle drag per pound of air.

In general, however, diffuser designs that result in improved pressure recoveries have associated with them high engine-cowl pressure drags. Consequently, the gain in engine performance resulting from improved pressure recovery may be largely offset by the resulting engine drag increase. Figure 8 shows engine performance at a flight Mach number of 3.5 as a function of both pressure recovery and engine nacelle drag coefficient. The dotted line repeats the low-angle-cowl drag values from figure 7 and represents the best performance attainable at any value of pressure recovery. Figure 8 indicates the penalties in drag rise that are acceptable to obtain better engine performance as a result of improved pressure recovery. Calculations based on the experimentally measured pressure recovery and cowl drags reported in reference 3 confirm the conclusion that currently available high-recovery inlets do not yield better over-all engine performance than does the single-cone type. Although the single-cone inlet was used to give performance representative of that available with other current inlet types, the advanced inlets have greater potentialities for improvement, as indicated in figure 8. Other factors must also be considered, of course, in comparing different diffuser designs. For example, a single-cone inlet may be easier to design and manufacture and is less sensitive to angle of attack than are more elaborate types. On the other hand, the higher pressure provided by an advanced inlet may increase combustion efficiency and prevent blowout.

The effects of variations discussed in the following sections are based on the use of a single-cone diffuser.

Combustion efficiency. - If τ is held constant in an engine, variations in combustion efficiency have only a negligible effect on engine thrust. However, fuel flow and hence specific impulse are directly proportional to the combustion efficiency. The great importance of this effect lies in the fact that the range of a ram-jet missile varies directly with the specific impulse, if all other factors do not change.

Flameholder pressure loss. - The purpose of the flameholder is to ensure the ignition and efficient burning of a fuel-air mixture moving at several hundred feet per second when the laminar flame speed of the mixture may be in the order of only 5 feet per second. Increased flow blockage and turbulence often improve the combustion efficiency but introduce pressure losses detrimental to the thrust output of the engine. A compromise is often necessary between these opposing factors. The change in propulsive thrust and specific impulse with the flameholder cold-flow pressure-drop coefficient is indicated in figure 9 for a constant combustion efficiency.

Fuel type. - Another combustor variable that may be changed to improve performance is the fuel used. High-energy fuels permit raising both thrust and specific impulse, but they are generally more expensive than hydrocarbon fuels, or they may have other undesirable characteristics such as pumping or storage problems. Calculations were made for pentaborane (B_5H_9) as a typical high-energy fuel frequently mentioned for ram-jet applications. Figure 10 shows the propulsive thrust coefficient and specific impulse of an engine designed for flight Mach number of 3.5 for pentaborane and JP-4. These calculations for B_5H_9 were made with the assumption of equilibrium composition of the exhaust gases and with expansion to the same area assumed with JP-4. Data for these calculations were taken from reference 7. These curves are for a combustion efficiency of 0.95 for the pentaborane and 0.90 for the JP-4 fuel.

For the same thrust coefficient, a specific impulse with pentaborane of more than 150 percent of that with JP-4 is indicated at low fuel-air ratios up to those that give maximum specific impulse. The improvement is less at high fuel-air ratios that give near maximum thrust coefficient.[a]

Nozzle area ratio. - The effect of nozzle area ratio is presented in figure 11. Very little loss in propulsive thrust coefficient and specific impulse is suffered by cutting back the nozzle area as much as 30 percent from that required for complete expansion. In fact, for smaller amounts of underexpansion, gains of 1 or 2 percent may be realized, because reducing these areas reduces the external nacelle drag sufficiently to compensate for the lower internal thrust. In addition, since the nozzle was assumed to have a velocity coefficient less than 1.0, a small amount of underexpansion results in a very small increase in internal thrust. It is sometimes proposed that the nozzle-exit area not be permitted to exceed the combustion-chamber area. This condition ($A_6 = A_3$) is marked on the curves of figure 11. It is apparent that this amount of underexpansion results in appreciable losses, particularly at the higher flight speed. The performance of a convergent nozzle ($A_6 = A_5$) is seen to be very poor at both speeds.

Nozzle velocity coefficient. - The velocity coefficient (defined as the actual velocity at the nozzle exit divided by the ideal isentropic velocity at the nozzle exit for the same pressure ratio) is used to indicate the amount of the nozzle internal flow losses. These losses, which reduce the total pressure, are due to shocks, turbulence, and viscous effects within the gas stream and to wall friction at the gas boundaries. The effect on engine performance of a variation in the nozzle

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[a]Since the calculations for figure 10 were completed, revised data have become available for the combustion products of B_5H_9. The curve presented, therefore, is only indicative of the general improvement possible with B_5H_9; the absolute magnitudes may be somewhat in error. (Ref. 8 presents charts and tables which incorporate these revised combustion data and which may be used for cycle calculations with pentaborane fuel.)

velocity coefficient from the assumed value of 0.975 is indicated in figure 12. A 1-percent change in velocity coefficient changes the thrust and specific impulse from 2 to 3 percent for the indicated values of Mach number and τ.

There are other sources of thrust losses through the nozzle that do not affect total-pressure loss. In addition to the flow losses treated through the application of a nozzle velocity coefficient, the assumption of one-dimensional flow implies that all exhaust gases are discharged axially and that there are no radial gradients in velocity. Neither of these implications is necessarily true. A nonuniform temperature distribution at the combustor exit would result in radial velocity gradients. Calculations indicate that all reasonable temperature distributions, such as a parabolic profile, result in thrust losses of less than 2 percent. Losses due to nonaxial discharge from a conical nozzle with a half-angle of 15° would be of the order of 1.5 percent. Use of a smaller angle or changes in nozzle contour would reduce this loss, although possibly at the expense of increased manufacturing cost and nozzle length.

Nozzle jet-deflection angle. - An interesting possibility for improving the performance of a ram-jet missile lies in turning the jet thrust of the engine downward. This slightly decreases the forward thrust and specific impulse but provides some lift, thereby permitting the use of a smaller wing and lowering the missile weight and drag. Figure 13 presents the effect of jet-deflection angle on engine performance, in which $C_{T,v}$ represents the component of vertical thrust divided by the free-stream dynamic pressure q_0 and the engine cross-sectional area A_m. These data are based on the assumption that nacelle drag does not change with deflection angle.

Altitude. - In the stratosphere (between 35,332 and 105,000 ft), changing flight altitude has only a small effect on propulsive thrust coefficient and specific impulse through the Reynolds number effect on nacelle skin-friction drag coefficient. Below the tropopause, in addition to this Reynolds number effect, the changing ambient temperature significantly affects ram-jet performance. In this region more fuel is required to maintain a design τ as the altitude is reduced. This extra mass addition, although it raises the thrust coefficient slightly, lowers the specific impulse considerably. This same increase in fuel consumption is felt by the over-all engine efficiency; however, the increased flight velocity (at constant flight Mach number) reduces the magnitude of the effect. These considerations combine to produce the variations in propulsive thrust coefficient, specific impulse, and over-all efficiency shown in figure 14. Also important is the effect of flight altitude (not shown) on combustion efficiency through changes in ambient pressure and temperature, which in turn establish the temperature, pressure, and velocity at the combustor inlet.

Off-Design Performance

All the previous discussion has been concerned with a continuously variable-geometry engine or a series of fixed-geometry design-point engines. Thus, it is implied that the inlet capture area is sized to avoid subcritical spillage, the diffuser cone angle is selected for optimum pressure recovery, the diffuser spike can be translated for correct positioning of the oblique shock upon the cowl lip, and the nozzle-throat area and area ratio are optimum. A practical engine incorporating such variable components is not yet available.

This invariance of geometry is of no concern if the engine can always be operated at its design or cruise point. Design-point engine operation is possible for a ram-jet missile that cruises along a Breguet flight path, provided the missile is fully boosted to its cruising Mach number and altitude by some other means. Even after starting cruise flight, however, some corrective action may be required and off-design engine operation may be necessary. Moreover, ram-jet thrust may be desired during the boost phase of the flight. In order to include engine performance for these flight conditions, some off-design engine calculations were also made. Because it was desired to indicate trends rather than absolute magnitudes, a constant value of γ of 1.30 for the exhaust gas was used for ease of computation of the off-design performance.

Figure 15 shows the propulsive thrust coefficient and specific impulse of a fixed-geometry engine designed to operate at M_0 of 3.5, M_2 of 0.200, and τ of 2.25. Because the combustor must now operate over a wide range of flight conditions, the combustion efficiency (0.87) was assumed to be slightly lower than that for the design-point case (0.90). Along each line of constant M_0, the parameter τ is raised to increase the thrust coefficient. For any given τ there is a single unique value of M_2 due to the choked fixed-nozzle throat area. At M_0 of 3.5 and values of τ below 2.25, M_2 is greater than 0.2 and the diffuser operates supercritically, with a severe loss in pressure recovery and a consequent adverse effect on thrust and specific impulse. As τ is raised by burning more fuel, M_2 is reduced to its design value, and the diffuser then operates critically, with the normal shock positioned at the diffuser lip. This condition corresponds to the sharp break in the curve. Further increase of τ causes subcritical diffuser operation. Although the external shock or "bow wave" so generated does not necessarily reduce the diffuser pressure recovery severely, it spills air that would normally enter the engine and causes large additive-drag losses. With subcritical operation and no loss in pressure recovery, a small gain in thrust is obtainable over critical operation, but the specific impulse drops markedly. In addition to inefficient operation, subcritical operation often results in instability of flow or "buzzing," which can, in severe cases, even blow out the combustor flame or damage the diffuser structure. In general, then, subcritical operation is undesirable.

For speeds less than design, best engine performance is also generally obtained with the value of τ chosen to give critical diffuser operation. However, as flight speed is reduced, the value of M_2 for critical operation is raised.

The off-design performance of several engines designed for efficient cruising at a flight Mach number of 3.5 and incorporating various types of geometry variation is shown in figure 16 as a function of flight Mach number. Performance is shown for the engines operating at their maximum thrust condition. Also included are data for critical operation of a fixed-geometry engine obtained by cross-plotting the peaks of the curves from figure 15. The performance of an engine with both continuously variable inlet diffuser and exit nozzle is obtained with a wide-open exhaust nozzle and a stoichiometric fuel-air ratio. Extremely large penalties in thrust are suffered with the fixed-geometry engine, with a thrust at flight Mach number 2.5 of only 15 percent of that available from an engine equipped with a continuously variable inlet and outlet. These large thrust losses are mainly due to the necessity of reducing τ to prevent subcritical operation. The ability to burn more fuel without being forced into the subcritical region explains why the continuously variable engine can produce more thrust than the fixed-geometry engine even at the design Mach number of 3.5.

Equipping an engine with either a movable-spike inlet or a variable-area exit nozzle, both of which are currently feasible, results in considerable gain over fixed-geometry engine performance. With a movable-spike inlet, the spike is translated axially so that all air spillage occurs behind an oblique shock. The flow behind the oblique shock remains supersonic, and the additive drag is not as severe as in the case of a bow wave. This spillage permits τ to be increased without causing subcritical operation. Thrust increases over the fixed configuration of 50 to 100 percent are possible, but the specific impulse is very low.

Thrust levels approaching the continuously variable case with about the same specific impulse can be achieved with an engine having a fixed inlet and a movable-plug nozzle. Although the nozzle throat area is variable, the nozzle-exit diameter is fixed and the nozzle expansion ratio cannot be independently chosen. At a flight Mach number of 2.5, the thrust is 74 percent that of the continuously variable engine.

CONCLUDING REMARKS

The calculated design-point performance of ram-jet engines using JP-4 fuel is presented for a wide range of engine total-temperature ratios and combustor-inlet Mach numbers for flight Mach numbers of 1.5 to 4.0. The results, which include engine thrust, drag, fuel consumption, and area ratios, are given in both graphical and tabular form. Maximum engine specific impulse (including nacelle drag) is approximately 1600 to 1700 pound-seconds per pound and occurs at a flight Mach number of about 2.5 to 3.0. Over-all engine efficiency, however, continues to increase to a flight Mach number of 4.0.

Calculations are also presented which indicate the sensitivity of the design-point results to changes in diffuser pressure recovery, flameholder pressure loss, combustion efficiency, fuel type, nozzle expansion ratio, nozzle velocity coefficient, nozzle jet-deflection angle, and altitude. Significant gains in both thrust coefficient and specific impulse may be achieved by improving the diffuser pressure recovery. However, presently available inlet designs that provide high recovery also have high cowl pressure drags which largely offset this potential gain. Changes in flameholder pressure loss have only small effects on engine performance. This factor may, however, influence combustor efficiency and the resulting range. Engine performance is very sensitive to changes in nozzle performance, a 1-percent variation in velocity coefficient often producing a 3-percent variation in engine thrust and specific impulse. Some underexpansion of the exhaust gases is desirable to reduce nacelle drag whenever the nozzle-exit diameter exceeds that of the combustion chamber.

Satisfactory off-design operation of a ram-jet engine is not possible with a fixed-geometry configuration. Somewhat better thrust can be obtained with the added complication of a translating-spike diffuser, although the specific impulse is poorer. Use of a movable plug to vary the throat area of the nozzle yields both thrust and specific impulse approaching those of a continuously variable-geometry engine. In considering engines designed for good cruise performance at flight Mach number 3.5 but operating at flight Mach number 2.5, the maximum thrusts are 15, 37, and 74 percent of that available with continuously variable geometry for engines with fixed-geometry, translating-spike diffuser, and movable-plug nozzle, respectively.

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APPENDIX A

SYMBOLS

The following symbols are used in this report:

A	area, sq ft
A_m	diffuser capture area A_1 or combustion-chamber area A_3, whichever is larger, sq ft
C_D	nacelle drag coefficient, D/q_0A_m
C_F	net-thrust coefficient, F/q_0A_m
C_T	propulsive thrust coefficient, $C_F - C_D$
C_V	nozzle velocity coefficient
D	drag, lb
E	over-all engine efficiency, $(F - D)V_0/JHw_f$
F	net thrust, $m_6V_6 - m_0V_0 + A_6(p_6 - p_0)$, lb
f/a	fuel-air ratio
H	lower heating value of fuel, Btu/lb
I	fuel specific impulse, $(F - D)/w_f$, lb-sec/lb
I_F	fuel specific impulse not including drag, F/w_f, lb-sec/lb
J	mechanical equivalent of heat, 778 ft-lb/Btu
M	Mach number
m	mass-flow rate, slugs/sec
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
q	incompressible dynamic pressure, $\rho V^2/2$, lb/sq ft
R	gas constant, 53.4 ft-lb/(lb)($^{\circ}$R)
T	total temperature, $^{\circ}$R

t static temperature, $^{\circ}R$

V velocity, ft/sec

w_f fuel-flow rate, lb/sec

γ ratio of specific heat at constant pressure to specific heat at constant volume

η_c combustion efficiency, $w_{f,id}/w_{f,ac}$

λ jet-deflection angle, deg

ρ density, lb/cu ft

τ engine total-temperature ratio, T_4/T_3

Subscripts:

ac actual

ef effective

fr friction

id ideal

nom nominal

v vertical

0 free stream

1 diffuser inlet

2 combustion-chamber inlet (upstream of flameholder)

3 combustion-chamber inlet (downstream of flameholder)

4 combustion-chamber exit

5 nozzle throat

6 nozzle exit

APPENDIX B

THERMODYNAMIC ASSUMPTIONS

Assigning correct gas properties and maintaining high computational accuracy are very important in any ram-jet engine analysis, because small changes in the calculated jet thrust may be magnified 3 or 4 times in the net propulsive thrust. Specification of the gas properties involves realistic choice of γ, R, and τ as functions of temperature, fuel-air ratio, and pressure (if there is appreciable dissociation).

Data for octane (ref. 2), which include dissociation effects, were converted to JP-4 and used in constructing figure 3, which gives τ as a function of M_0 and f/a for an ambient temperature of 392° R. Similar curves were constructed for altitudes under the tropopause where the ambient temperature is different from 392° R. The very high temperatures plus the changes of composition due to burning cause the γ of the combustion gases to vary appreciably from 1.40 if equilibrium is reached. These equilibrium values, as a function of temperature and f/a and for a pressure of 1 atmosphere, were obtained from reference 9, which includes a very appreciable effect of dissociation. Molecular equilibrium is not necessarily maintained during the expansion of the gas through the nozzle. However, considerations of heat-capacity lag and chemical-reaction rates indicated that the process is probably closer to equilibrium than to "frozen" conditions. After trying several different methods, it was found that specifying an effective γ for the nozzle by

$$\gamma_{ef} = \frac{1}{2}\left(\gamma_4 + \gamma_6\right)$$

gave best results as checked by equilibrium calculations using references 10 and 11. The deviations in net thrust were generally found to be less than 3 percent. According to reference 9, this means of specifying γ_{ef} is also best for use in the isentropic equation

$$\frac{t}{T} = \left(\frac{p}{P}\right)^{\frac{\gamma-1}{\gamma}}$$

The value of R for the airstream was taken as 53.4 ft-lb/(lb)(°R). The same value was used for the exhaust gas with small error (ref. 12).

APPENDIX C

ENGINE DRAG

The engine thrust is defined in terms of the total-momentum changes occurring between the free-stream tube ahead of the engine and the exhaust at the end of the engine. The drag must, therefore, include all the external forces acting on the stream tube and engine between the same two points. These forces are composed of pressure forces acting perpendicular to the stream tube and engine, and a friction force acting parallel to the engine nacelle due to the viscosity of the air. The pressure (or wave) drag may be further broken down into the part acting directly on the nacelle and the additional, imaginary part acting on the stream tube because of the way the thrust was defined. The latter force is termed additive drag, and was calculated by the method of reference 13.

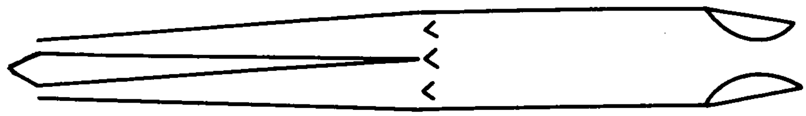
The nacelle friction drag was calculated from the following equation, which is based on the flat-plate formula of reference 14:

$$C_{D,fr} = \frac{A_W}{A_3} \frac{0.0306\ K}{Re^{1/7}\left(1 + \frac{\gamma - 1}{4} M_0^2\right)^{5/7}}$$

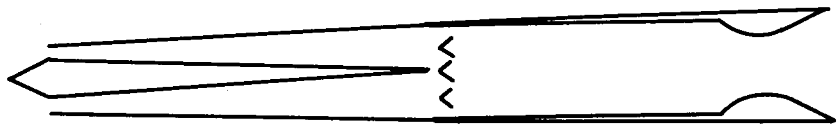
where A_W is nacelle skin area, K is a shape factor taken as 1.05 for a cylindrical nacelle, and the Reynolds number Re is based on free-stream conditions with a nominal length of 30 feet and a nominal altitude of 70,000 feet.

The nacelle pressure drag was obtained from the linearized theory from reference 15. For the low-speed cases when the exit area was smaller than combustion-chamber area, data for boattail drag for a 7.04° cone were used from the same source. The diffuser cowl was assumed conical, with no added pressure drag incurred from a curved lip.

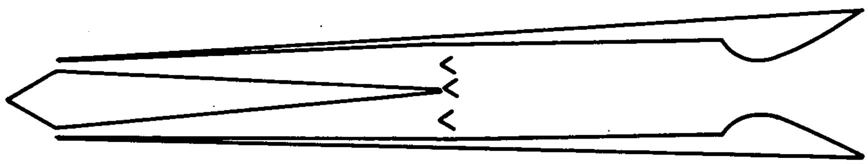
The engine was assumed to have a fineness ratio (length divided by combustion-chamber diam.) of 9, the diffuser being nominally 4, combustion chamber 3, and nozzle 2. The appearance of the engine as a function of flight speed is as shown in the following sketch (not to scale):



Low M_0 (1.5 - 2.0)



Medium M_0 (2.0 - 3.0)



High M_0 (3.0 - 4.0)

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TABLE I. - DESIGN-POINT RAM-JET ENGINE PERFORMANCE

WITH SINGLE-CONE DIFFUSER

Flight Mach number, M_0											
1.5						2.0					
Combustion-chamber-inlet Mach number, M_2 = 0.150											
A_1/A_3 = 0.287						A_1/A_3 = 0.382					
τ	C_T	I	C_D	A_6/A_3	A_5/A_3	τ	C_T	I	C_D	A_6/A_3	A_5/A_3
2.0	0.023	152	0.186	0.468	0.395	2.5	0.291	1274	0.118	0.750	0.450
3.0	.210	907	.175	.585	.499	3.0	.428	1372	.009	.840	.499
4.0	.383	1054	.161	.699	.590	3.5	.551	1400	.098	.925	.548
5.0	.533	1080	.143	.812	.690	4.0	.670	1393	.086	1.005	.595
6.0	.698	1056	.122	.931	.794	4.5	.780	1360	.087	1.085	.643
7.0	.882	998	.109	1.060	.909	5.0	.880	1307	.088	1.174	.696
7.32	.945	975	.109	1.106	.945	5.5	.990	1227	.090	1.275	.751
						5.96	1.096	1139	.092	1.380	.807
Combustion-chamber-inlet Mach number, M_2 = 0.175											
A_1/A_3 = 0.334						A_1/A_3 = 0.444					
2.0	0.055	418	0.175	0.550	0.465	2.5	0.365	1426	0.104	0.874	0.530
3.0	.282	1043	.158	.694	.590	3.0	.524	1508	.086	.985	.593
4.0	.471	1139	.138	.825	.713	3.5	.660	1488	.084	1.086	.654
5.0	.635	1106	.113	.970	.843	4.0	.780	1438	.085	1.180	.720
[a]6.0	.780	984	.109	1.124	1.000	4.5	.895	1369	.087	1.287	.790
						5.0	1.007	1288	.089	1.400	.863
						5.5	1.122	1197	.092	1.527	.940
						[a]5.85	1.206	1130	.094	1.618	1.000
Combustion-chamber-inlet Mach number, M_2 = 0.200											
A_1/A_3 = 0.379						A_1/A_3 = 0.505					
2.5	0.225	996	0.150	0.726	0.620	2.25	0.347	1433	0.091	0.941	0.581
3.0	.344	1131	.137	.808	.697	2.50	.440	1498	.081	1.009	.620
3.5	.454	1177	.121	.896	.779	2.75	.520	1527	.082	1.072	.660
4.0	.550	1178	.103	.986	.865	3.00	.598	1519	.083	1.139	.700
4.5	.625	1136	.100	1.080	.952	3.25	.672	1502	.084	1.194	.743
[a]4.77	.659	1065	.101	1.132	1.000	3.75	.804	1448	.087	1.324	.832
						4.25	.927	1371	.090	1.460	.929
						[a]4.60	1.006	1312	.093	1.560	1.000
Combustion-chamber-inlet Mach number, M_2 = 0.225											
A_1/A_3 = 0.424						A_1/A_3 = 0.564					
2.50	0.266	1079	0.130	0.835	0.719	2.25	0.399	1469	0.081	1.078	0.675
2.75	.348	1151	.122	.876	.766	2.50	.488	1492	.083	1.152	.721
3.00	.400	1180	.111	.931	.819	2.75	.574	1497	.084	1.226	.770
3.25	.452	1177	.099	.988	.870	3.00	.657	1481	.086	1.302	.822
3.50	.499	1144	.097	1.040	.928	3.25	.733	1457	.087	1.378	.879
[a]3.80	.540	1047	.099	1.110	1.000	3.50	.801	1423	.089	1.456	.940
						[a]3.72	.863	1377	.091	1.530	1.000

[a] Thermal choking

TABLE I. - Continued. DESIGN-POINT RAM-JET ENGINE PERFORMANCE

WITH SINGLE-CONE DIFFUSER

Flight Mach number, M_0											
2.5						3.0					
Combustion-chamber-inlet Mach number, M_2 = 0.150											
A_1/A_3 = 0.505						A_1/A_3 = 0.626					
τ	C_T	I	C_D	A_6/A_3	A_5/A_3	τ	C_T	I	C_D	A_6/A_3	A_5/A_3
2.0	0.284	1441	0.076	0.960	0.397	2.00	0.390	1522	0.061	1.177	0.398
2.5	.469	1555	.072	1.052	.453	2.25	.507	1565	.062	1.235	.429
3.0	.638	1546	.073	1.125	.501	2.50	.620	1566	.063	1.290	.456
3.5	.795	1510	.074	1.193	.552	2.75	.730	1550	.064	1.344	.481
4.0	.960	1432	.075	1.275	.606	3.00	.834	1516	.065	1.402	.506
4.5	1.132	1329	.077	1.371	.661	3.25	.951	1474	.066	1.466	.537
4.80	1.250	1249	.078	1.436	.697	3.50	1.079	1419	.069	1.540	.559
						3.96	1.349	1300	.073	1.701	.619
Combustion-chamber-inlet Mach number, M_2 = 0.175											
A_1/A_3 = 0.586						A_1/A_3 = 0.727					
2.0	0.348	1512	0.070	1.062	0.468	2.00	0.458	1537	0.058	1.305	0.472
2.5	.554	1594	.071	1.138	.534	2.25	.591	1576	.061	1.370	.507
3.0	.749	1565	.073	1.246	.598	2.50	.722	1574	.063	1.430	.538
3.5	.934	1508	.075	1.338	.665	2.75	.851	1551	.066	1.499	.571
4.0	1.125	1406	.077	1.432	.734	3.00	.972	1515	.067	1.567	.606
4.5	1.302	1299	.079	1.535	.810	3.25	1.105	1462	.069	1.644	.641
4.80	1.420	1212	.081	1.600	.860	3.50	1.249	1402	.072	1.732	.674
						3.96	1.546	1266	.078	1.942	.747
Combustion-chamber-inlet Mach number, M_2 = 0.200											
A_1/A_3 = 0.666						A_1/A_3 = 0.826					
2.0	0.400	1537	0.071	1.158	0.545	2.00	0.520	1531	0.065	1.425	0.545
2.5	.630	1597	.072	1.262	.625	2.25	.657	1561	.067	1.502	.590
3.0	.850	1566	.074	1.363	.706	2.50	.801	1555	.068	1.578	.633
3.5	1.052	1501	.076	1.470	.793	2.75	.953	1533	.070	1.652	.673
4.0	1.250	1396	.079	1.589	.888	3.00	1.091	1496	.072	1.727	.714
[a]4.45	1.420	1287	.082	1.710	1.000	3.25	1.239	1444	.074	1.822	.763
						3.50	1.390	1375	.076	1.908	.811
						3.96	1.692	1220	.084	2.162	.918
Combustion-chamber-inlet Mach number, M_2 = 0.225											
A_1/A_3 = 0.745						A_1/A_3 = 0.923					
2.00	0.439	1519	0.071	1.240	0.625	2.00	0.589	1551	0.053	1.553	0.629
2.25	.565	1556	.073	1.301	.675	2.25	.748	1556	.055	1.628	.680
2.50	.689	1565	.074	1.360	.722	2.50	.900	1544	.072	1.714	.730
2.75	.810	1555	.075	1.421	.775	2.75	1.052	1513	.074	1.817	.787
3.00	.920	1527	.076	1.486	.829	3.00	1.197	1465	.076	1.901	.840
3.25	1.028	1487	.077	1.551	.897	3.25	1.351	1410	.078	2.022	.901
[a]3.66	1.201	1400	.080	1.669	1.000	[a]3.58	1.584	1314	.083	2.168	1.000

[a] Thermal choking

TABLE I. - Concluded. DESIGN-POINT RAM-JET ENGINE PERFORMANCE

WITH SINGLE-CONE DIFFUSER

Flight Mach number, M_0											
3.5						4.0					
Combustion-chamber-inlet Mach number, M_2 = 0.125											
A_1/A_3 = 0.618						A_1/A_3 = 0.743					
τ	C_T	I	C_D	A_6/A_3	A_5/A_3	τ	C_T	I	C_D	A_6/A_3	A_5/A_3
1.75	0.271	1349	0.055	1.209	0.303	1.50	0.162	954	0.047	1.362	0.281
2.00	.388	1411	.056	1.275	.329	1.75	.334	1255	.050	1.457	.308
2.25	.505	1428	.057	1.341	.355	2.00	.493	1324	.052	1.543	.334
2.50	.628	1422	.058	1.406	.376	2.25	.654	1334	.054	1.627	.356
2.75	.759	1398	.059	1.488	.400	2.50	.838	1310	.058	1.804	.383
3.00	.900	1353	.062	1.615	.426	2.75	1.047	1218	.064	2.048	.412
3.25	1.059	1276	.064	1.744	.451						
Combustion-chamber-inlet Mach number, M_2 = 0.150											
A_1/A_3 = 0.740						A_1/A_3 = 0.890					
1.75	0.341	1375	0.054	1.358	0.370	1.50	0.213	1045	0.049	1.536	0.399
2.00	.470	1441	.056	1.437	.401	1.75	.403	1263	.052	1.653	.373
2.25	.617	1450	.058	1.515	.430	2.00	.592	1329	.055	1.762	.403
2.50	.755	1442	.060	1.606	.459	2.25	.784	1329	.058	1.880	.435
2.75	.908	1412	.062	1.695	.487	2.50	1.001	1314	.063	2.086	.466
3.00	1.076	1360	.066	1.856	.519	2.75	1.254	1217	.071	2.356	.501
3.25	1.255	1266	.070	2.015	.549						
Combustion-chamber-inlet Mach number, M_2 = 0.175											
A_1/A_3 = 0.858						A_1/A_3 = 1.032					
1.75	0.389	1410	0.058	1.524	0.438	1.50	0.244	1035	0.055	1.730	0.401
2.00	.553	1452	.060	1.609	.474	1.75	.467	1263	.057	1.844	.438
2.25	.720	1462	.062	1.691	.507	2.00	.687	1329	.061	1.986	.477
2.50	.887	1450	.064	1.805	.544	2.25	.906	1335	.064	2.125	.513
2.75	1.060	1415	.067	1.912	.578	2.50	1.158	1310	.069	2.352	.551
3.00	1.240	1353	.072	2.104	.620	2.75	1.449	1213	.078	2.680	.598
3.25	1.440	1250	.077	2.284	.662						
Combustion-chamber-inlet Mach number, M_2 = 0.200											
A_1/A_3 = 0.976											
1.75	0.428	1375	0.063	1.665	0.506						
2.00	.610	1420	.065	1.768	.551						
2.25	.800	1436	.067	1.871	.594						
2.50	.995	1426	.069	1.991	.638						
2.75	1.195	1394	.072	2.118	.683						
3.00	1.400	1340	.077	2.311	.732						
3.25	1.634	1240	.084	2.560	.790						

TABLE II. - DESIGN-POINT RAM-JET ENGINE PERFORMANCE WITH TWO-CONE DIFFUSER

[Note: Values of C_D, A_6/A_3, and A_5/A_3 are assumed to be the same as those in table I.]

Flight Mach number, M_0								
2.0			2.5			3.0		
Combustion-chamber-inlet Mach number, M_2 = 0.150								
A_1/A_3 = 0.392			A_1/A_3 = 0.557			A_1/A_3 = 0.758		
τ	C_T	I	τ	C_T	I	τ	C_T	I
2.5	0.309	1319	2.0	0.344	1581	2.00	0.525	1690
3.0	.451	1406	2.5	.344	1652	2.25	.668	1704
3.5	.577	1429	3.0	.550	1621	2.50	.810	1684
4.0	.700	1417	3.5	.738	1571	2.75	.943	1652
4.5	.814	1382	4.0	.913	1483	3.00	1.071	1607
5.0	.918	1327	4.5	1.097	1372	3.25	1.215	1555
5.5	1.032	1245	4.80	1.289	1287	3.50	1.373	1491
5.96	1.141	1155				3.96	1.706	1358
Combustion-chamber-inlet Mach number, M_2 = 0.175								
A_1/A_3 = 0.456			A_1/A_3 = 0.646			A_1/A_3 = 0.880		
2.5	0.386	1469	2.0	0.416	1639	2.00	0.611	1692
3.0	.550	1541	2.5	.645	1683	2.25	.774	1705
3.5	.691	1516	3.0	.863	1635	2.50	.736	1684
4.0	.815	1462	3.5	1.069	1565	2.75	1.095	1648
4.5	.934	1391	4.0	1.283	1453	3.00	1.244	1601
5.0	1.050	1308	4.5	1.480	1339	3.25	1.408	1545
5.5	1.170	1215	4.80	1.612	1248	3.50	1.586	1470
[a]5.85	1.257	1146				3.96	1.954	1321
Combustion-chamber-inlet Mach number, M_2 = 0.200								
A_1/A_3 = 0.519			A_1/A_3 = 0.735			A_1/A_3 = 1.100		
2.25	0.368	1479	2.0	0.476	1658	2.00	0.691	1680
2.50	.464	1538	2.5	.761	1678	2.25	.860	1687
2.75	.547	1563	3.0	.977	1632	2.50	1.037	1663
3.00	.627	1552	3.5	1.203	1556	2.75	1.224	1626
3.25	.704	1532	4.0	1.424	1442	3.00	1.394	1579
3.75	.841	1474	[a]4.45	1.615	1327	3.25	1.577	1518
4.25	.969	1395				3.50	1.763	1440
[a]4.60	1.051	1334				3.96	2.139	1274
Combustion-chamber-inlet Mach number, M_2 = 0.225								
A_1/A_3 = 0.579			A_1/A_3 = 0.822			A_1/A_3 = 1.118		
2.25	0.422	1514	2.00	0.521	1633	2.00	0.695	1688
2.50	.515	1532	2.25	.661	1651	2.25	.870	1670
2.75	.604	1533	2.50	.800	1647	2.50	1.040	1644
3.00	.690	1514	2.75	.935	1627	2.75	1.208	1604
3.25	.768	1487	3.00	1.058	1542	3.00	1.368	1546
3.50	.839	1452	3.25	1.178	1545	3.25	1.539	1482
[a]3.72	.893	1388	[a]3.66	1.372	1450	[a]3.58	1.796	1376

[a]Thermal choking.

TABLE II. - Concluded. DESIGN-POINT RAM-JET

ENGINE PERFORMANCE WITH TWO-CONE DIFFUSER

[Note: Values of C_D, A_6/A_3, and A_5/A_3 are assumed to be the same as those in table I.]

Flight Mach number, M_0					
3.5			4.0		
Combustion-chamber-inlet Mach number, M_2 = 0.125					
A_1/A_3 = 0.834			A_1/A_3 = 1.048		
τ	C_T	I	τ	C_T	I
1.75	0.435	1602	1.50	0.284	1243
2.00	.595	1604	1.75	.520	1452
2.25	.756	1584	2.00	.738	1472
2.50	.926	1552	2.25	.959	1452
2.75	1.106	1509	2.50	1.214	1410
3.00	1.303	1451	2.75	1.507	1302
3.25	1.523	1359			
Combustion-chamber-inlet Mach number, M_2 = 0.150					
A_1/A_3 = 0.999			A_1/A_3 = 1.256		
1.75	0.535	1597	1.50	0.300	1311
2.00	.713	1619	1.75	.518	1445
2.25	.915	1593	2.00	.735	1468
2.50	1.106	1564	2.25	.955	1441
2.75	1.317	1517	2.50	1.206	1409
3.00	1.551	1452	2.75	1.501	1296
3.25	1.801	1346			
Combustion-chamber-inlet Mach number, M_2 = 0.175					
A_1/A_3 = 1.158			A_1/A_3 = 1.456		
1.75	0.525	1631	1.50	0.304	1287
2.00	.720	1621	1.75	.530	1432
2.25	.918	1598	2.00	.754	1459
2.50	1.117	1566	2.25	.978	1440
2.75	1.323	1515	2.50	1.237	1399
3.00	1.542	1443	2.75	1.538	1288
3.25	1.783	1327			
Combustion-chamber-inlet Mach number, M_2 = 0.200					
A_1/A_3 = 1.318					
1.75	0.507	1589			
2.00	.697	1584			
2.25	.895	1567			
2.50	1.099	1538			
2.75	1.309	1490			
3.00	1.526	1426			
3.25	1.775	1315			

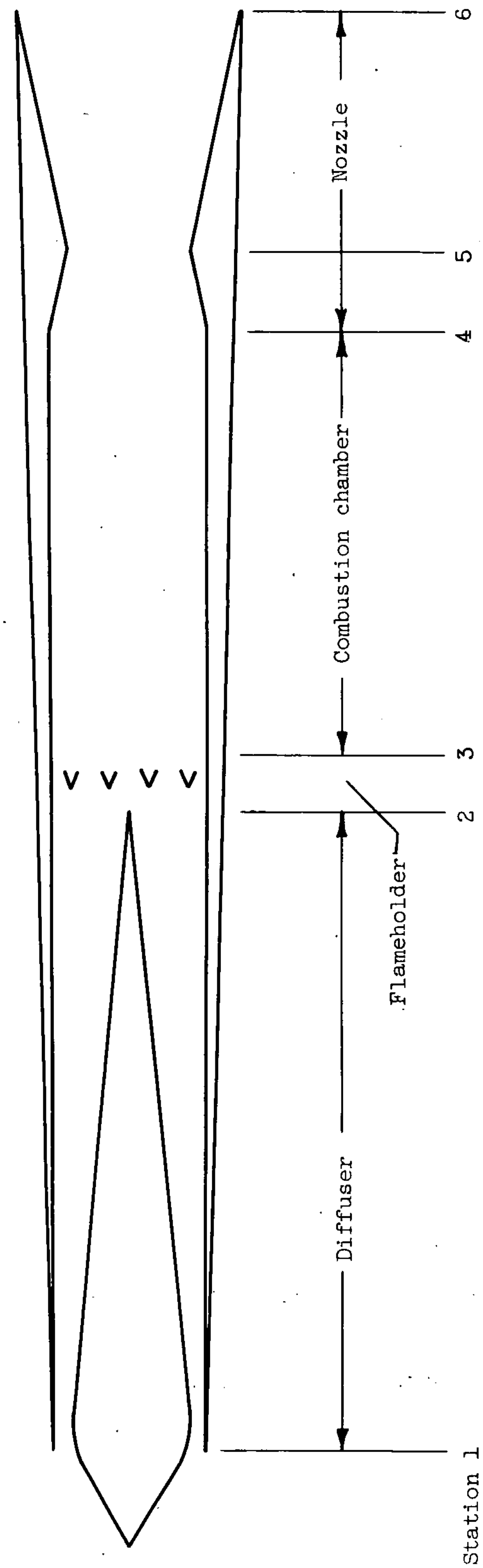


Figure 1. - Schematic diagram of typical ram-jet engine.

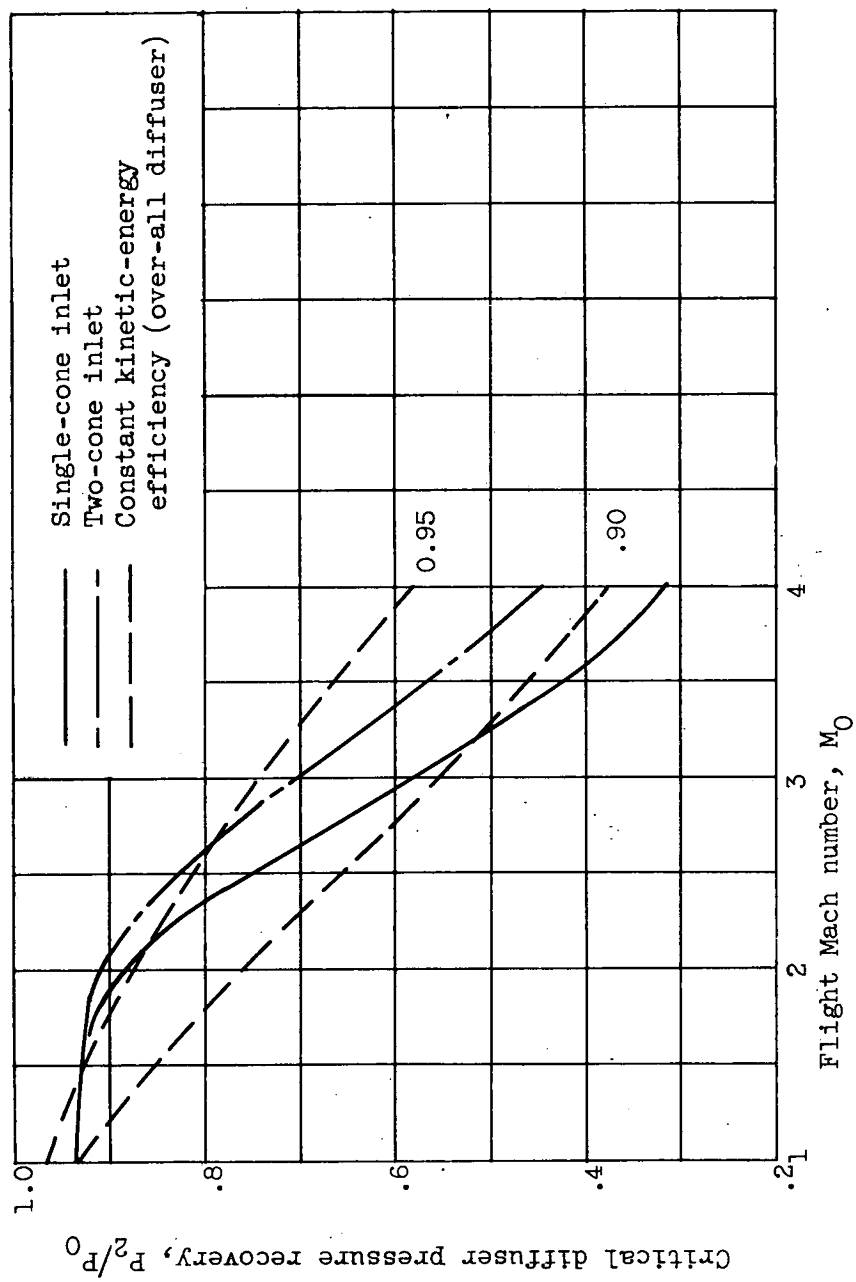


Figure 2. - Effect of flight Mach number on critical diffuser pressure recovery.

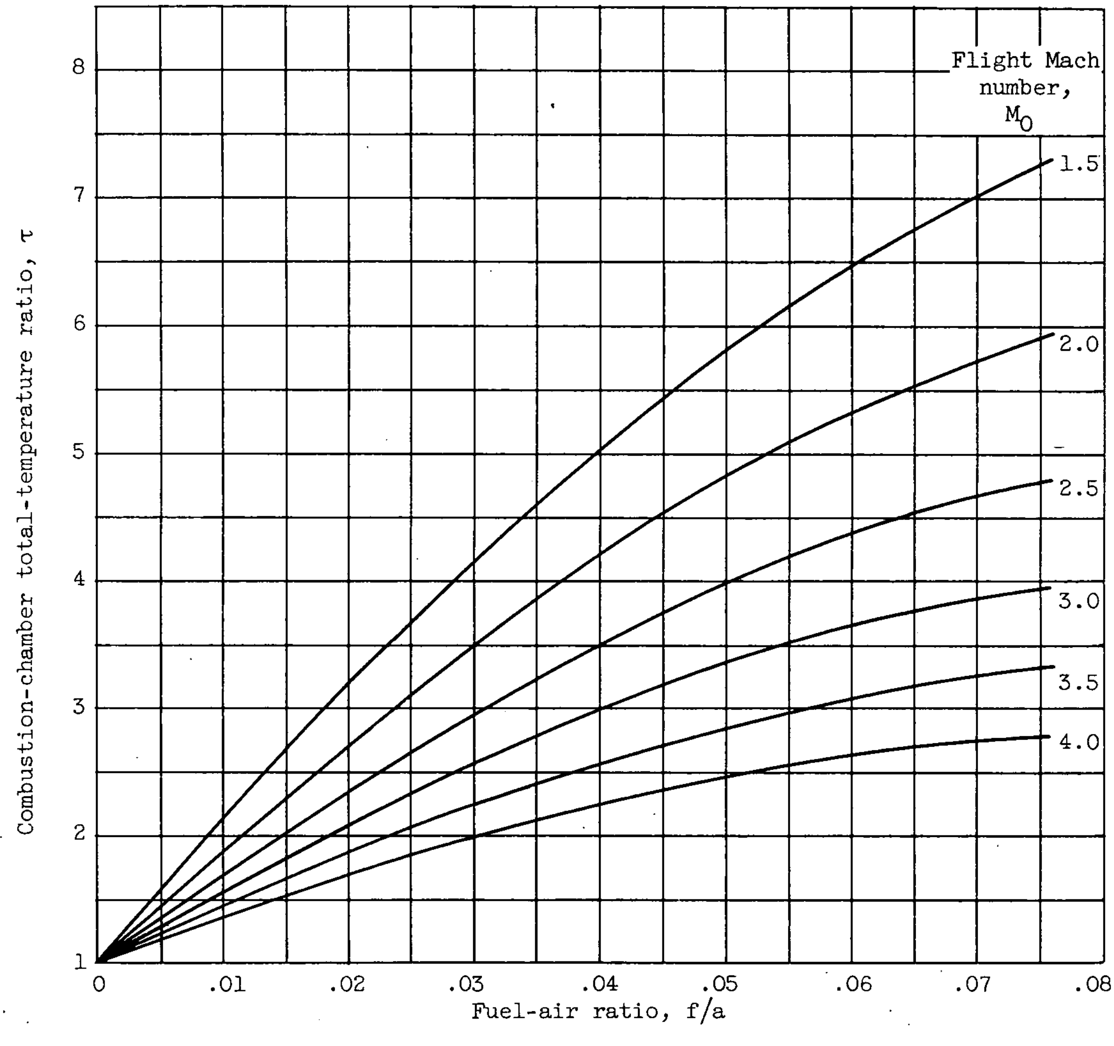
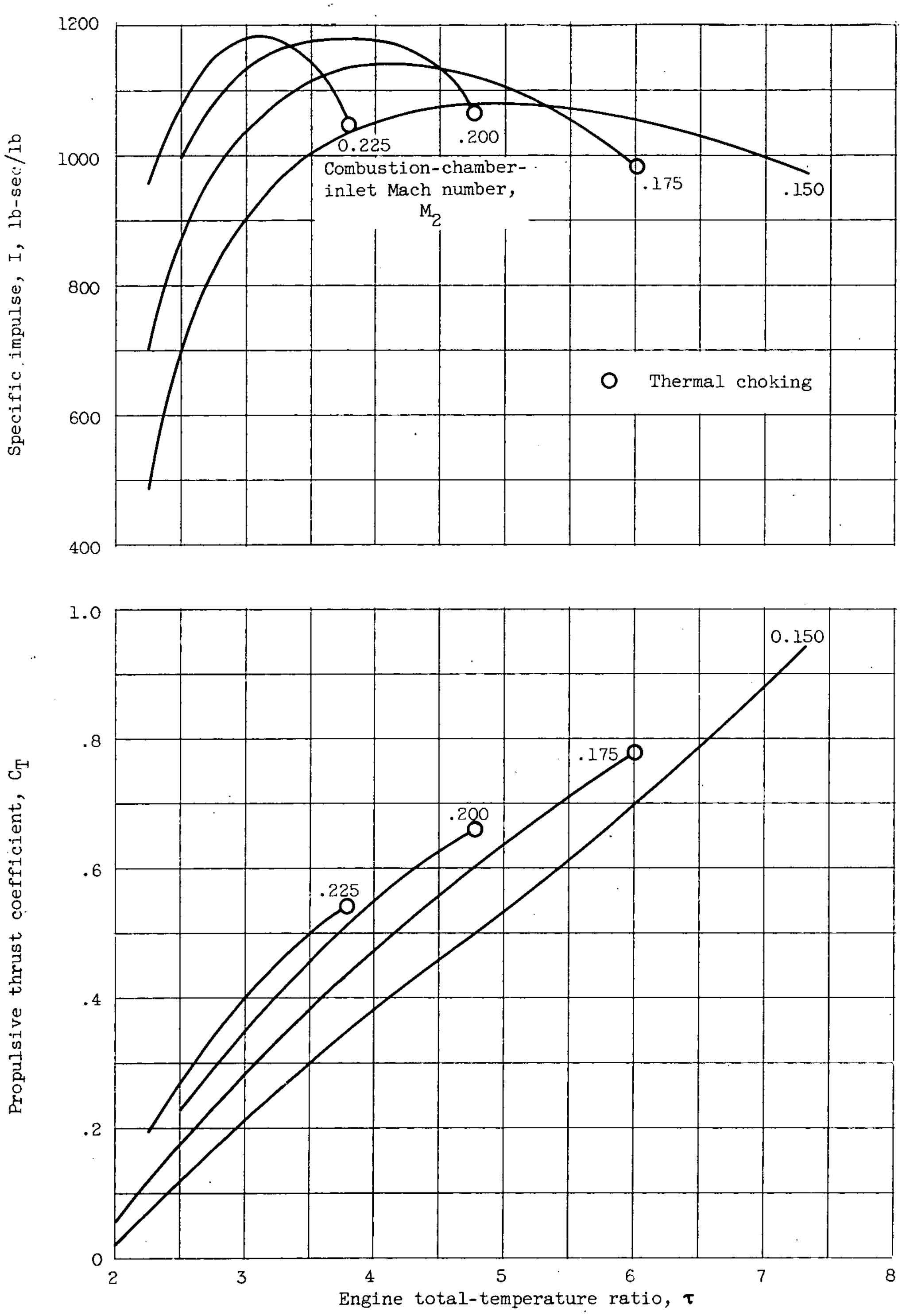
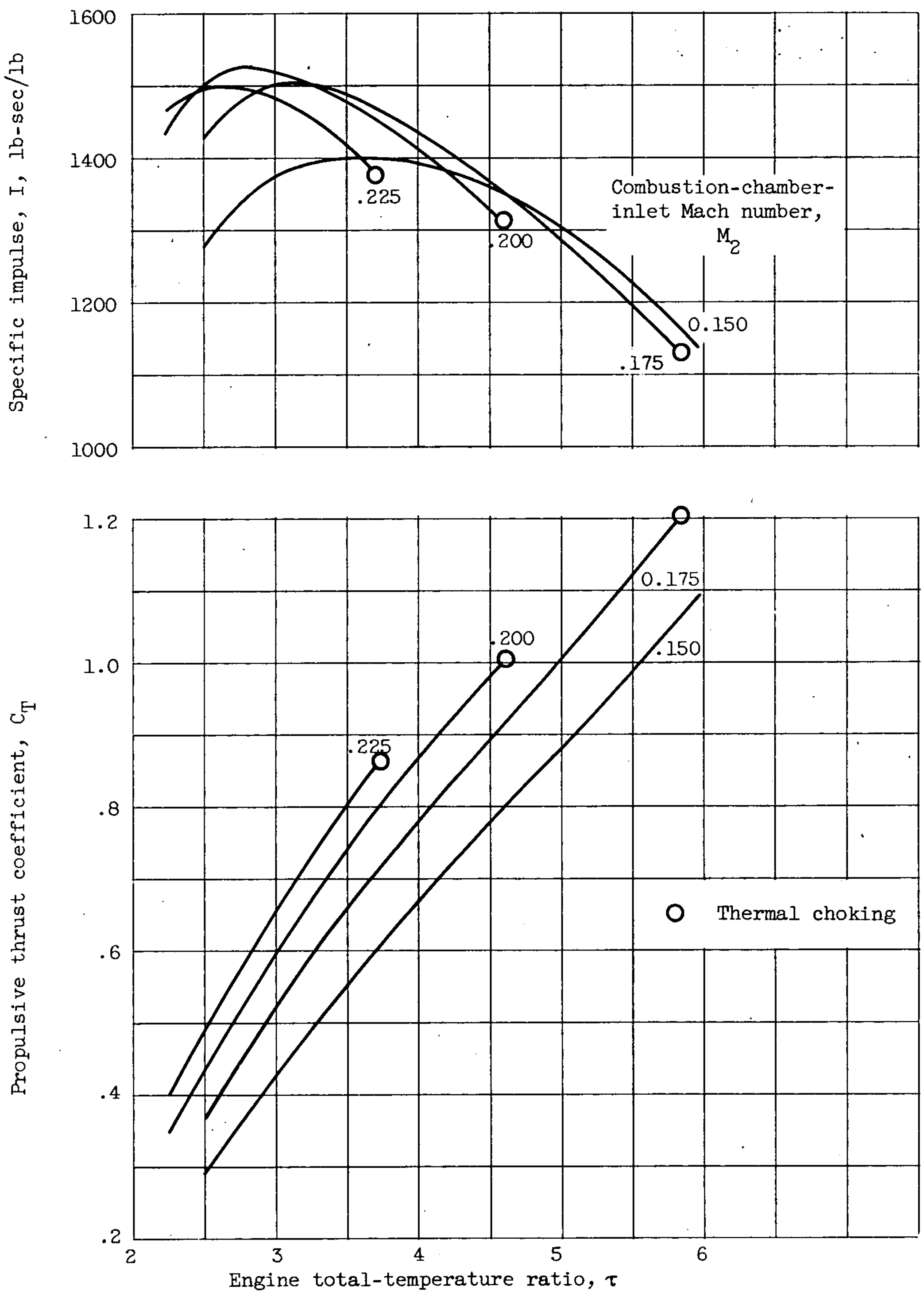


Figure 3. - Effect of fuel-air ratio on combustion-chamber total-temperature ratio for JP-4 fuel (heating value, 18,640 Btu/lb). Ambient temperature, 392° R; combustion efficiency, 0.90.



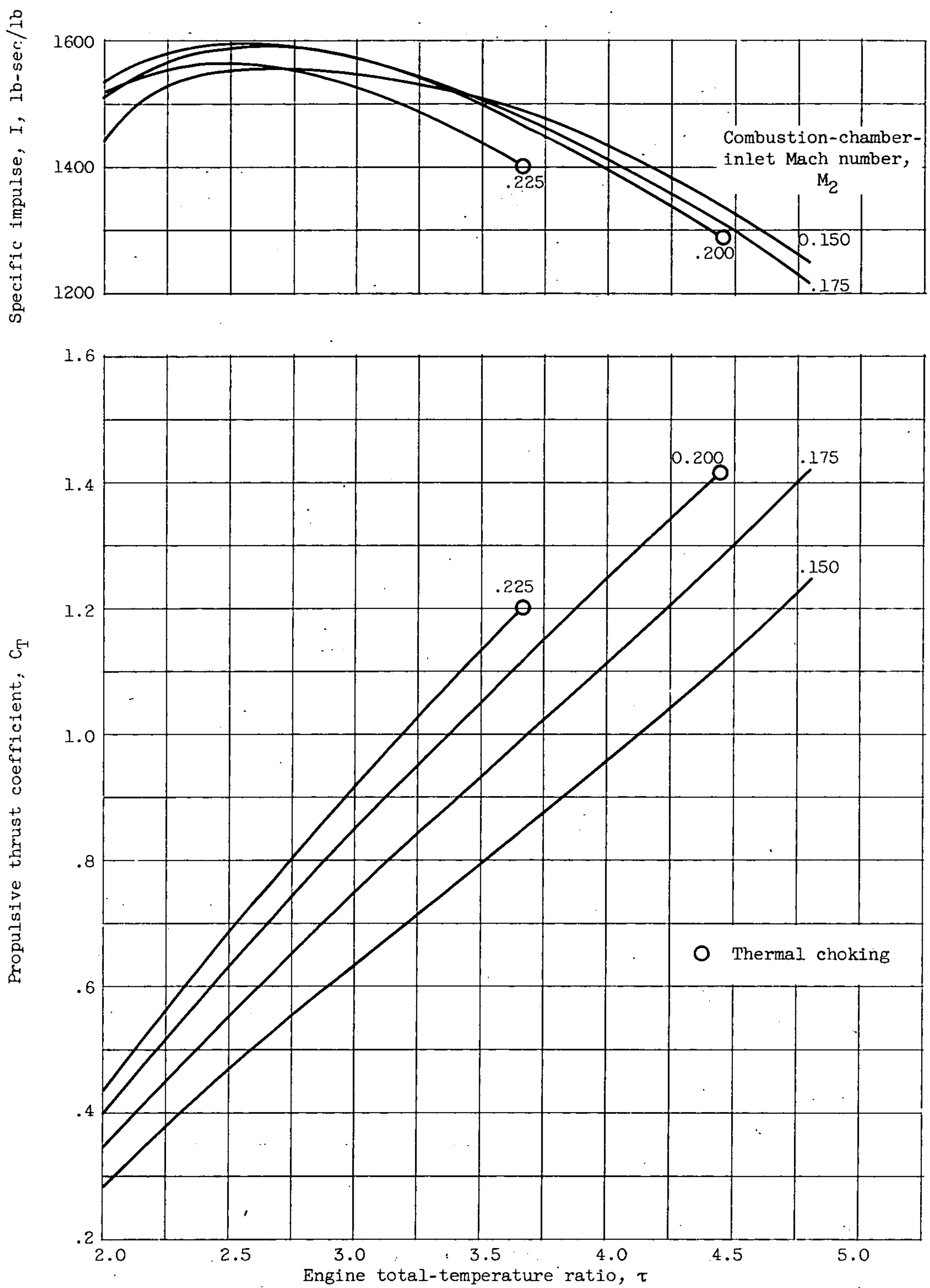
(a) Flight Mach number, 1.5.

Figure 4. - Design-point performance of ram-jet engine with single-cone diffuser.



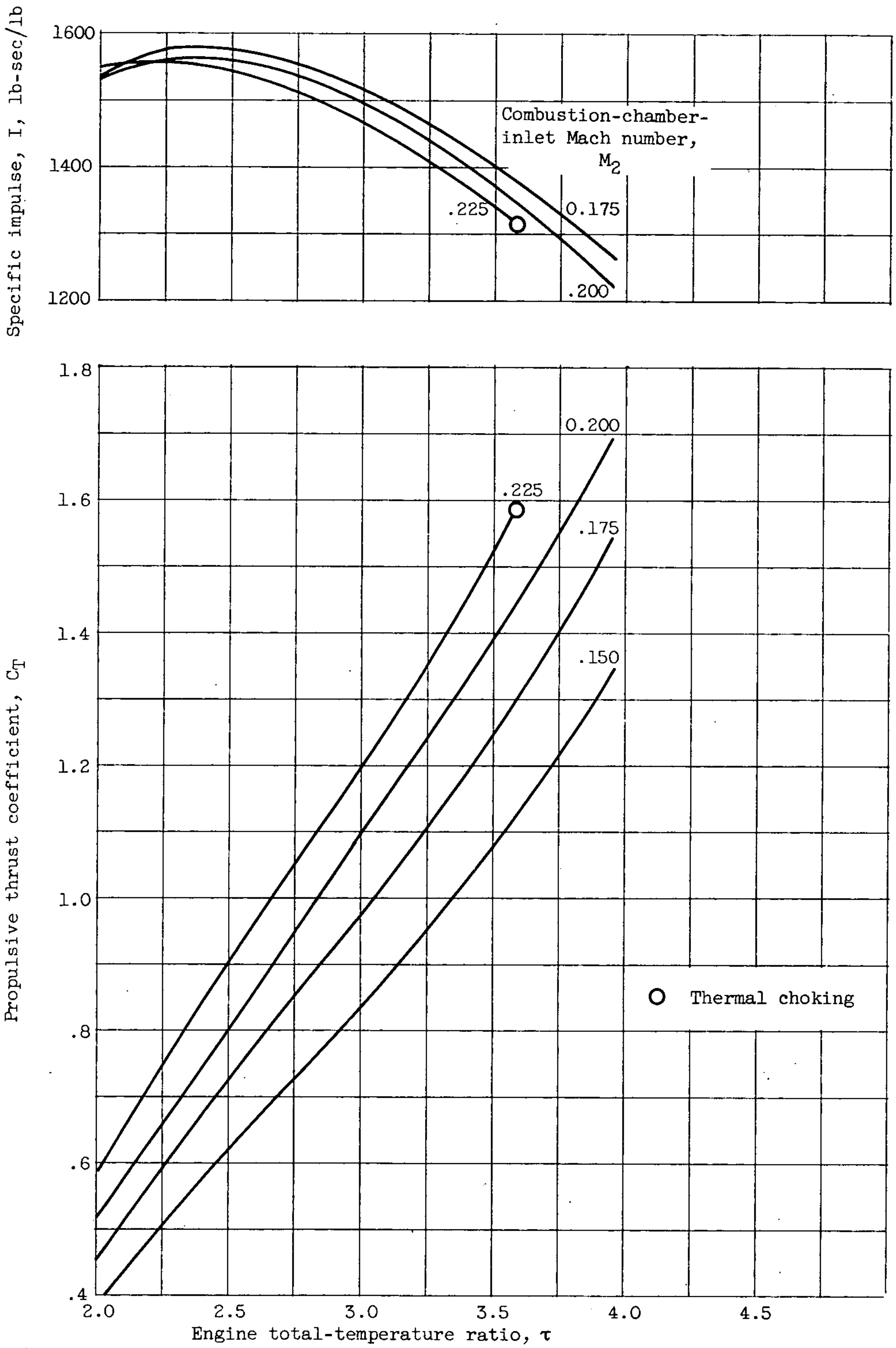
(b) Flight Mach number, 2.0.

Figure 4. - Continued. Design-point performance of ram-jet engine with single-cone diffuser.



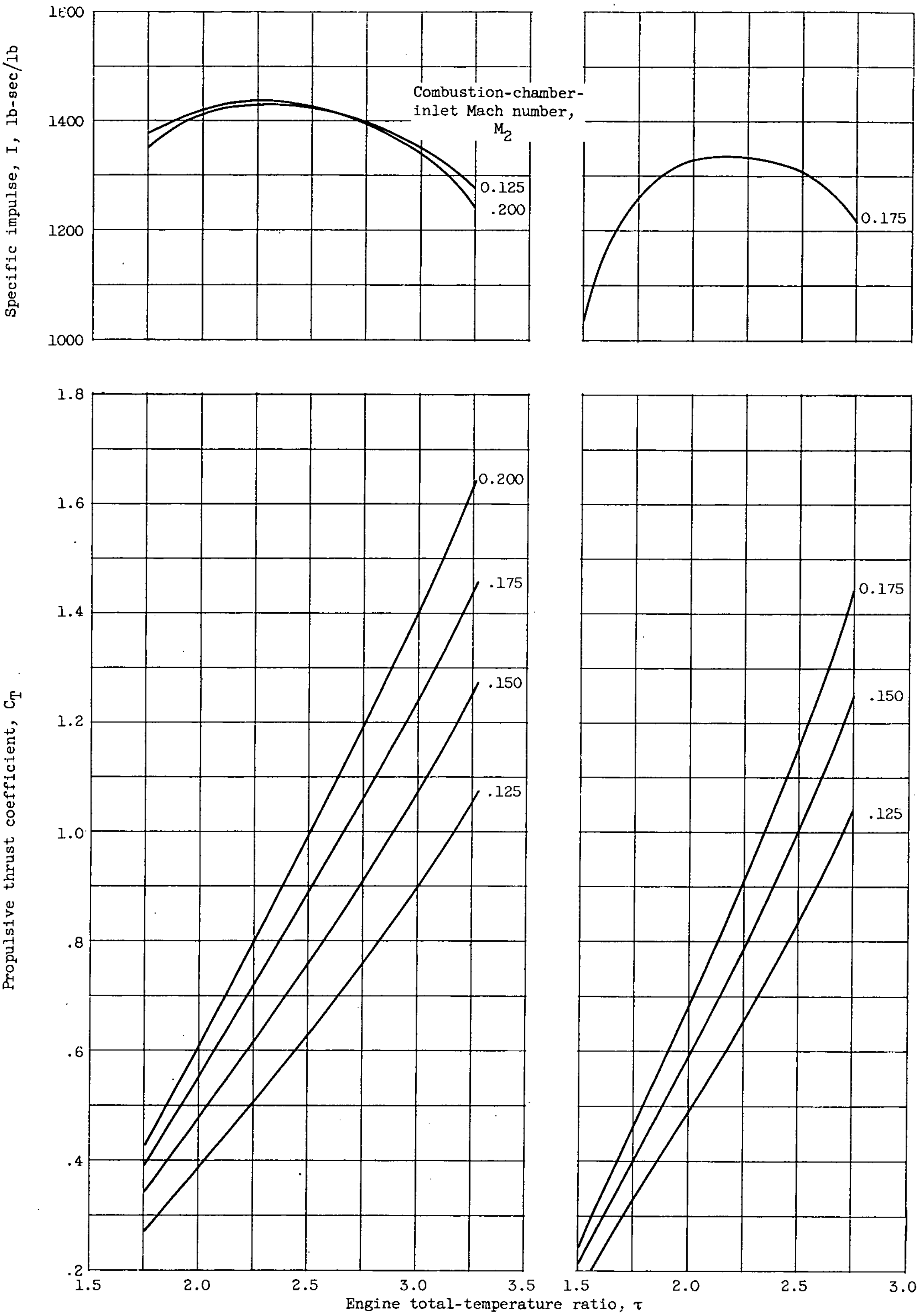
(c) Flight Mach number, 2.5.

Figure 4. - Continued. Design-point performance of ram-jet engine with single-cone diffuser.



(d) Flight Mach number, 3.0.

Figure 4. - Continued. Design-point performance of ram-jet engine with single-cone diffuser.



(e) Flight Mach number, 3.5.

(f) Flight Mach number, 4.0.

Figure 4. - Concluded. Design-point performance of ram-jet engine with single-cone diffuser.

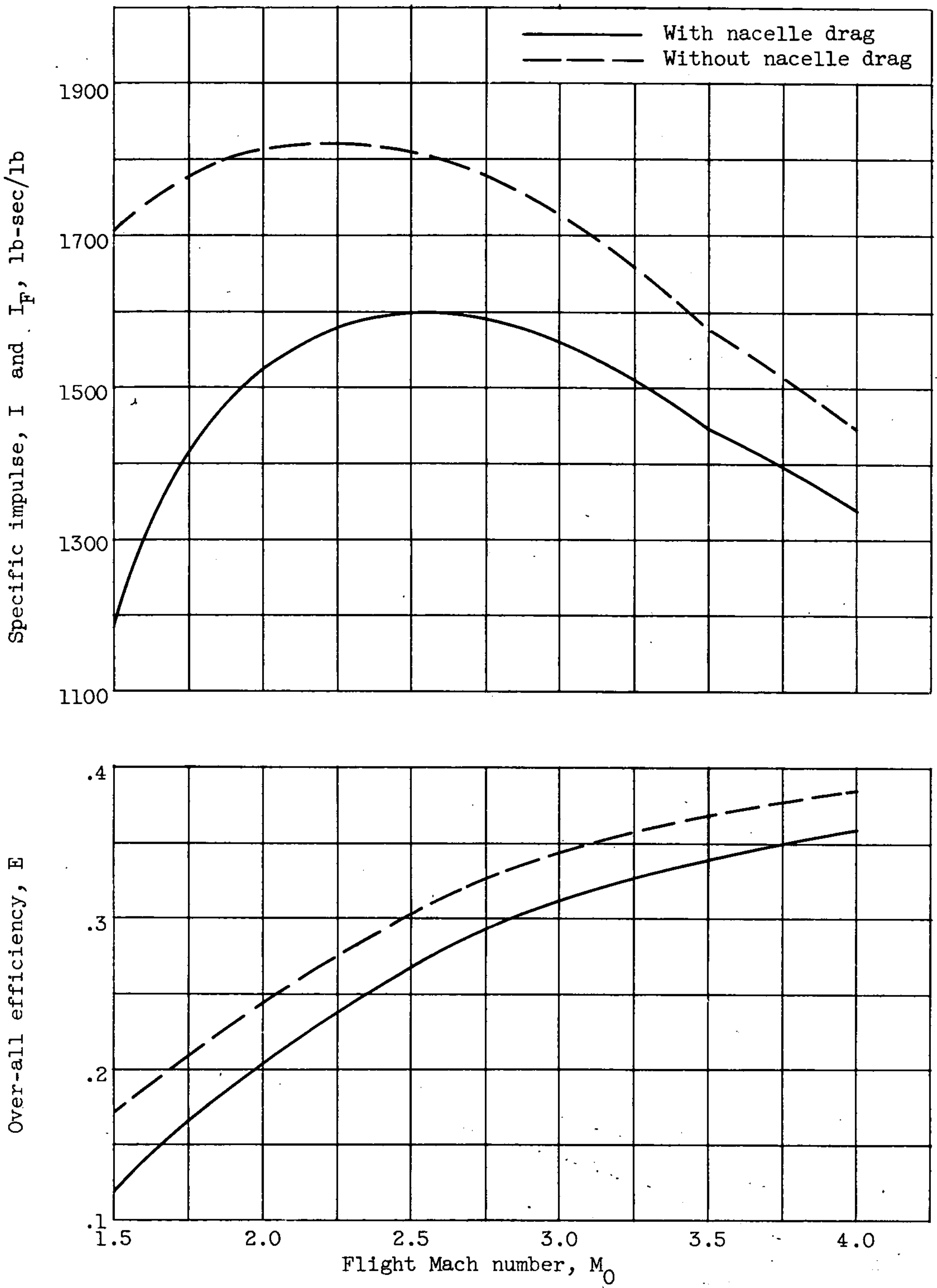
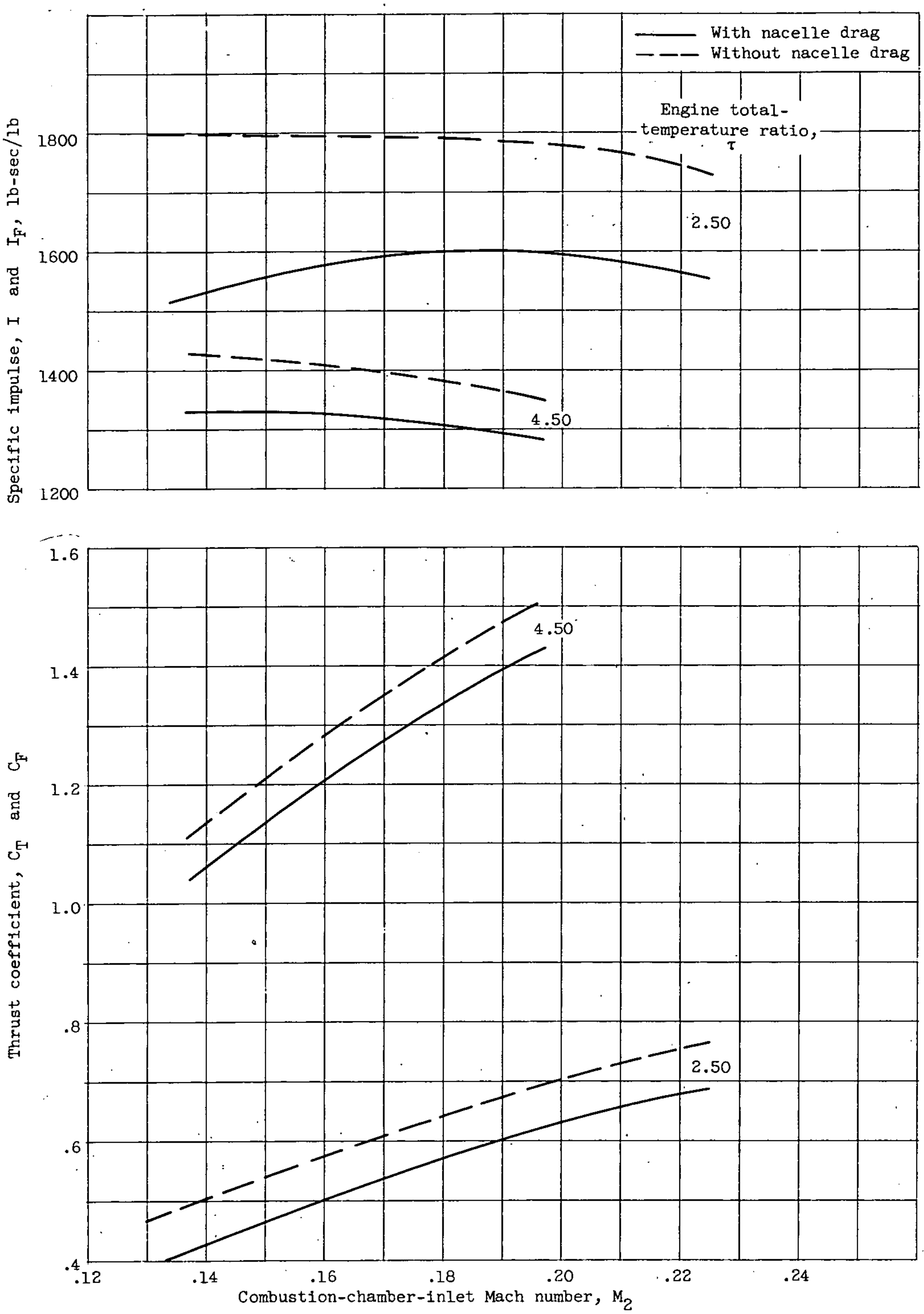
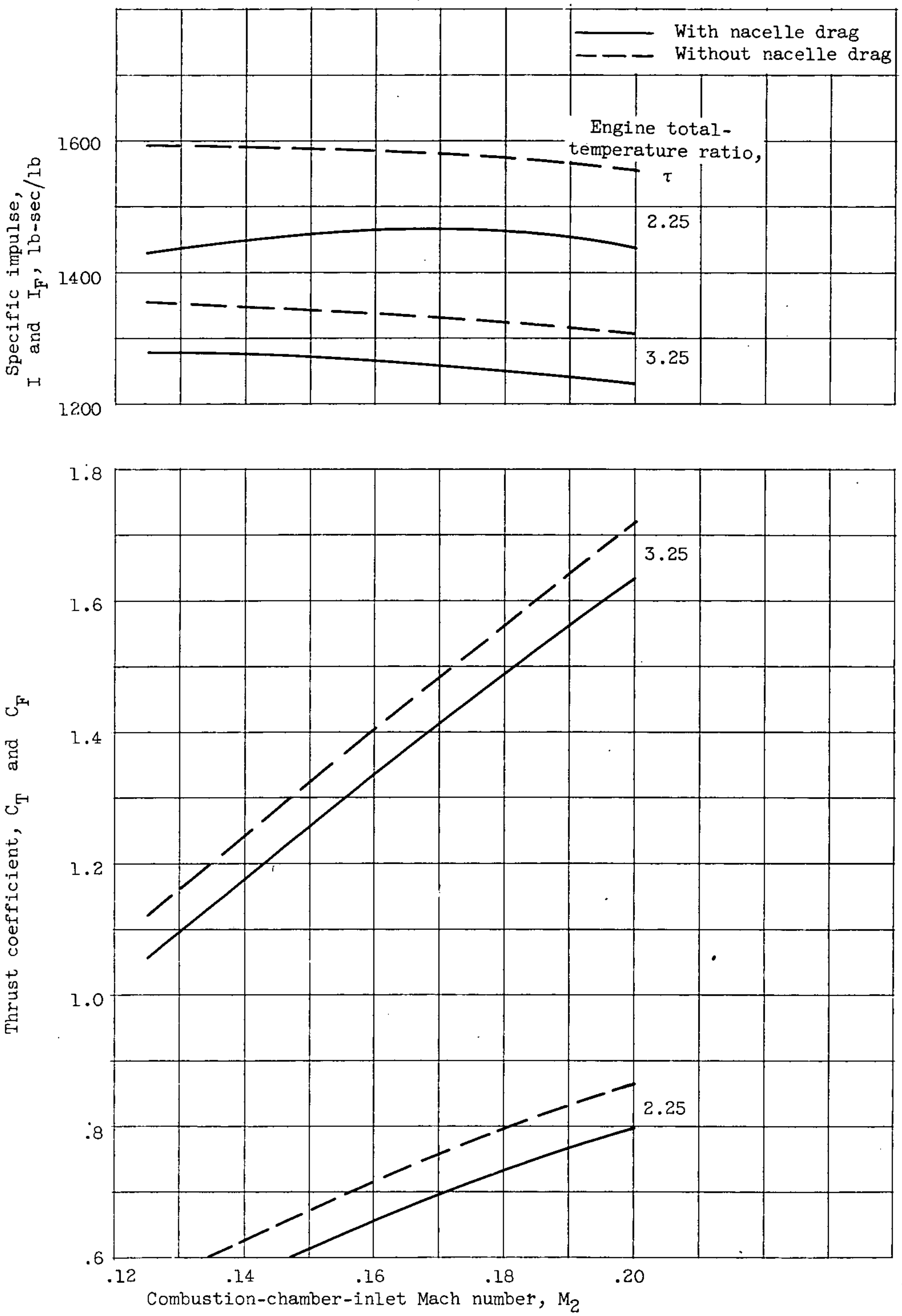


Figure 5. - Effect of flight Mach number on design-point performance of ram-jet engine. Engine total-temperature ratio chosen for maximum efficiency.



(a) Flight Mach number, 2.5.

Figure 6. - Effect of combustion-chamber-inlet Mach number on ram-jet engine performance.



(b) Flight Mach number, 3.5.

Figure 6. - Concluded. Effect of combustion-chamber-inlet Mach number on ram-jet engine performance.

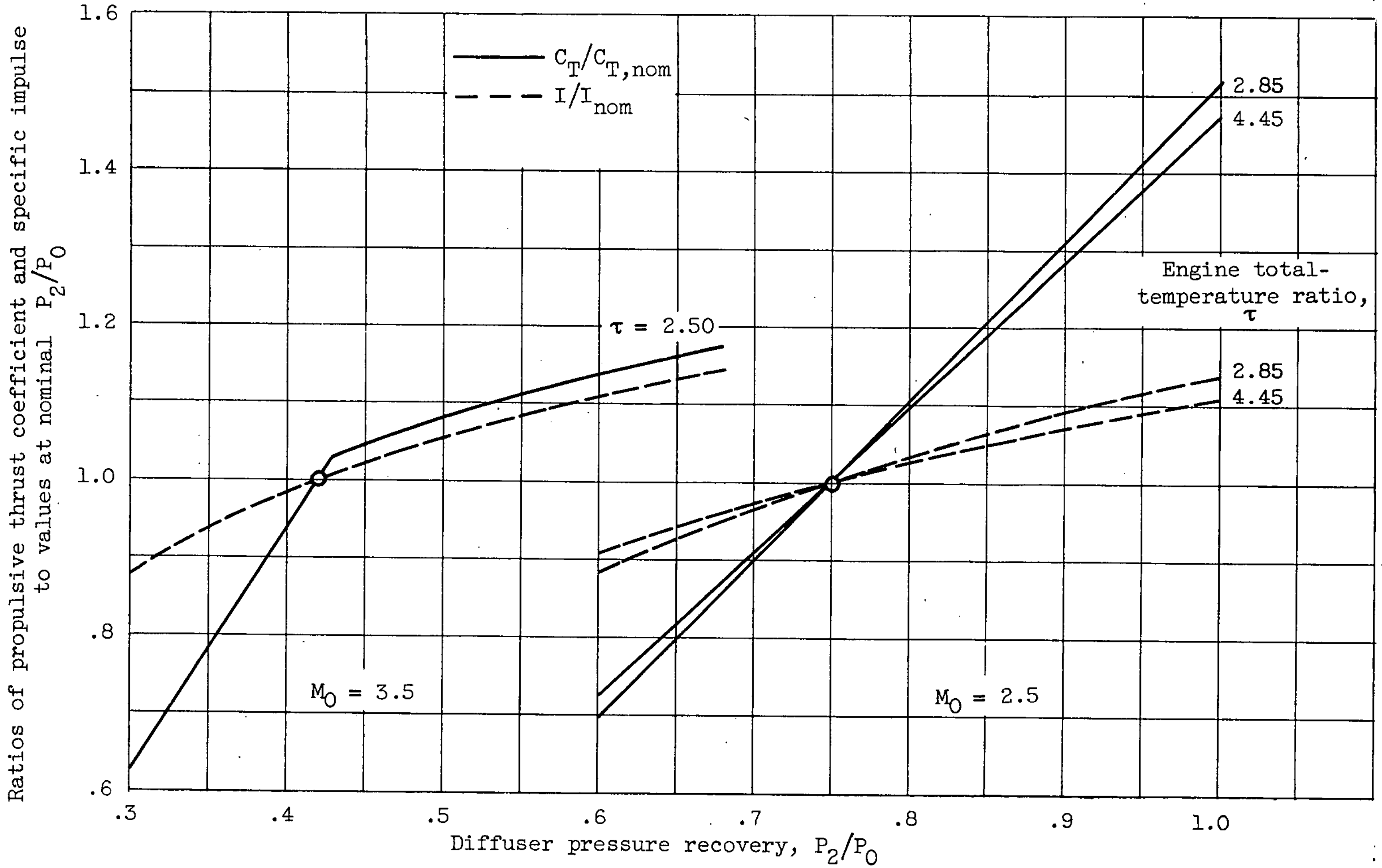


Figure 7. - Effect of diffuser pressure recovery on ram-jet engine performance. Combustion-chamber-inlet Mach number, 0.200.

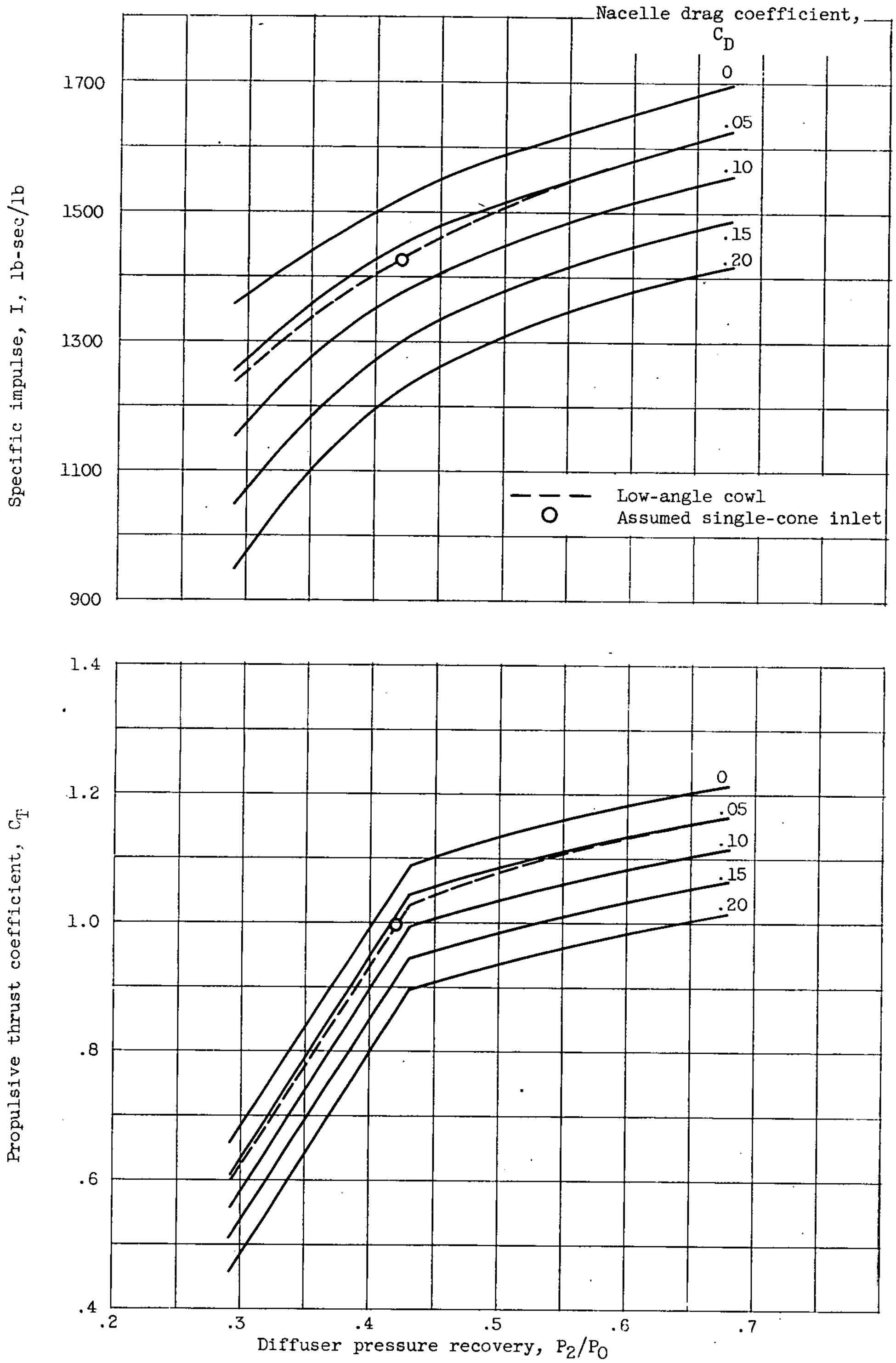


Figure 8. - Effect of diffuser pressure recovery and engine drag coefficient on ram-jet engine performance. Flight Mach number, 3.5; combustion-chamber-inlet Mach number, 0.200; engine total-temperature ratio, 2.50.

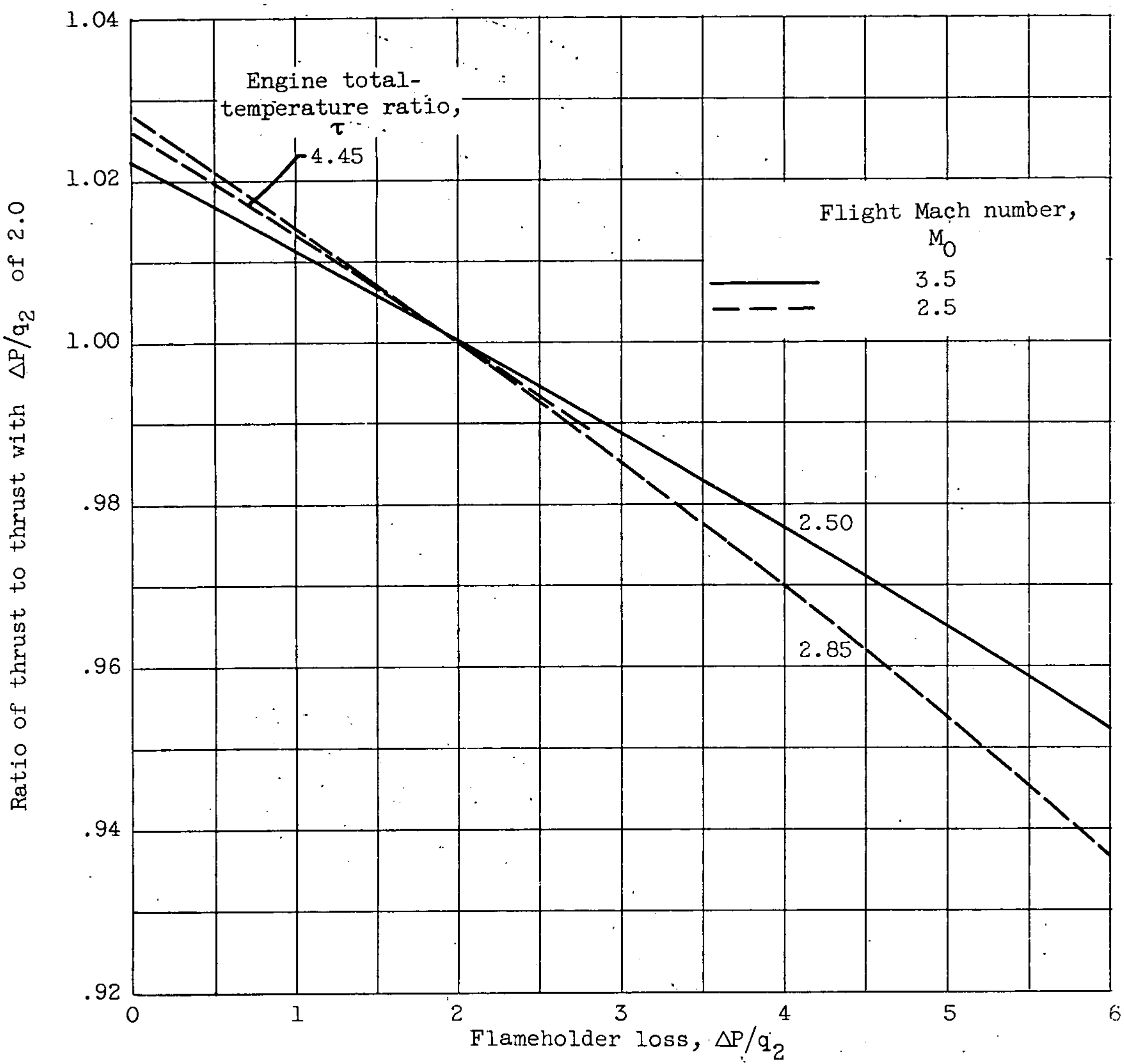


Figure 9. - Effect of flameholder pressure loss on ram-jet engine performance. Combustion-chamber-inlet Mach number, 0.200; combustion efficiency, 0.90. (Specific impulse is affected in same proportion as thrust.)

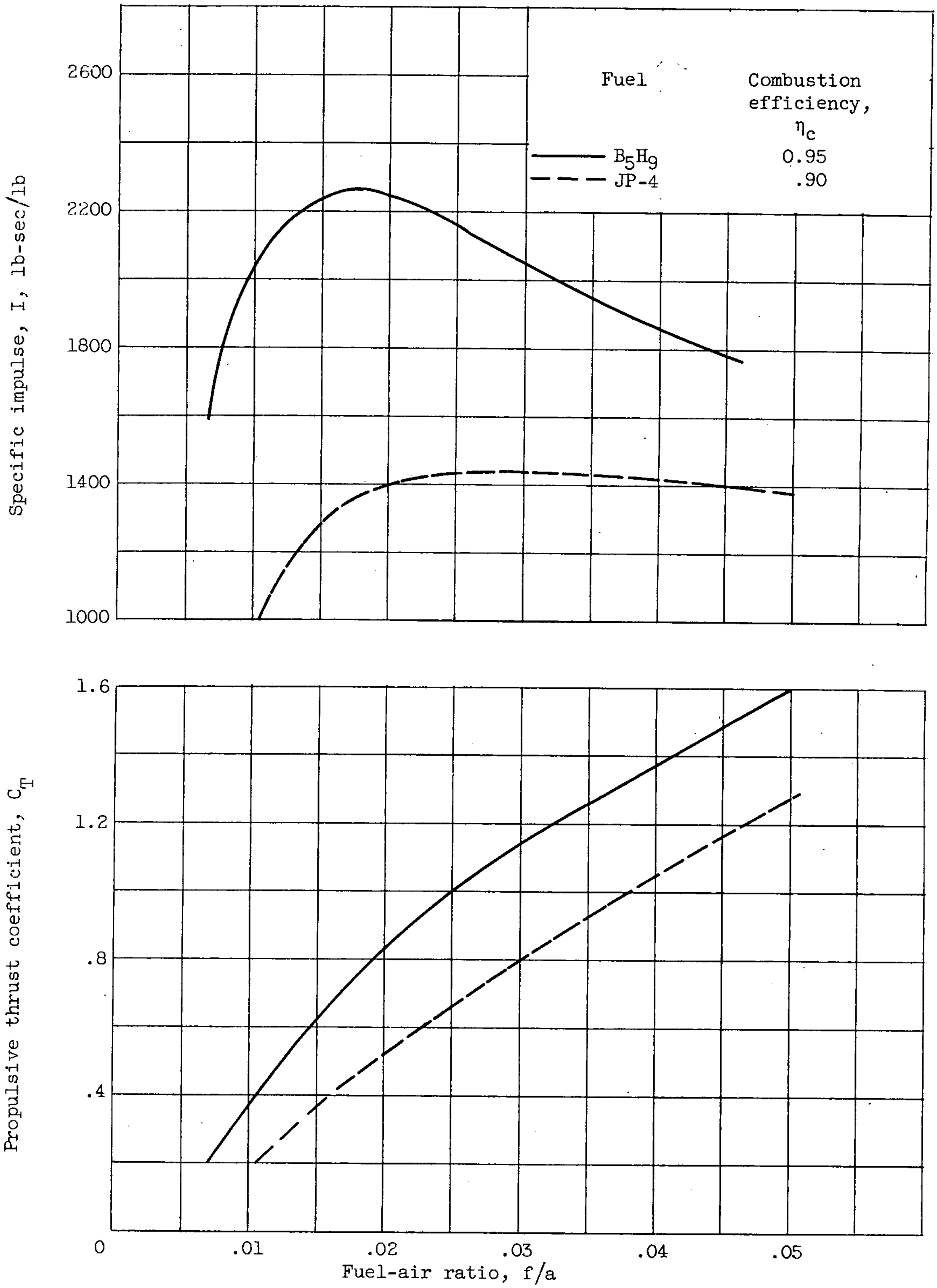


Figure 10. - Comparison between performance of engines using pentaborane and JP-4 fuel. Flight Mach number, 3.5; combustion-chamber-inlet Mach number, 0.200.

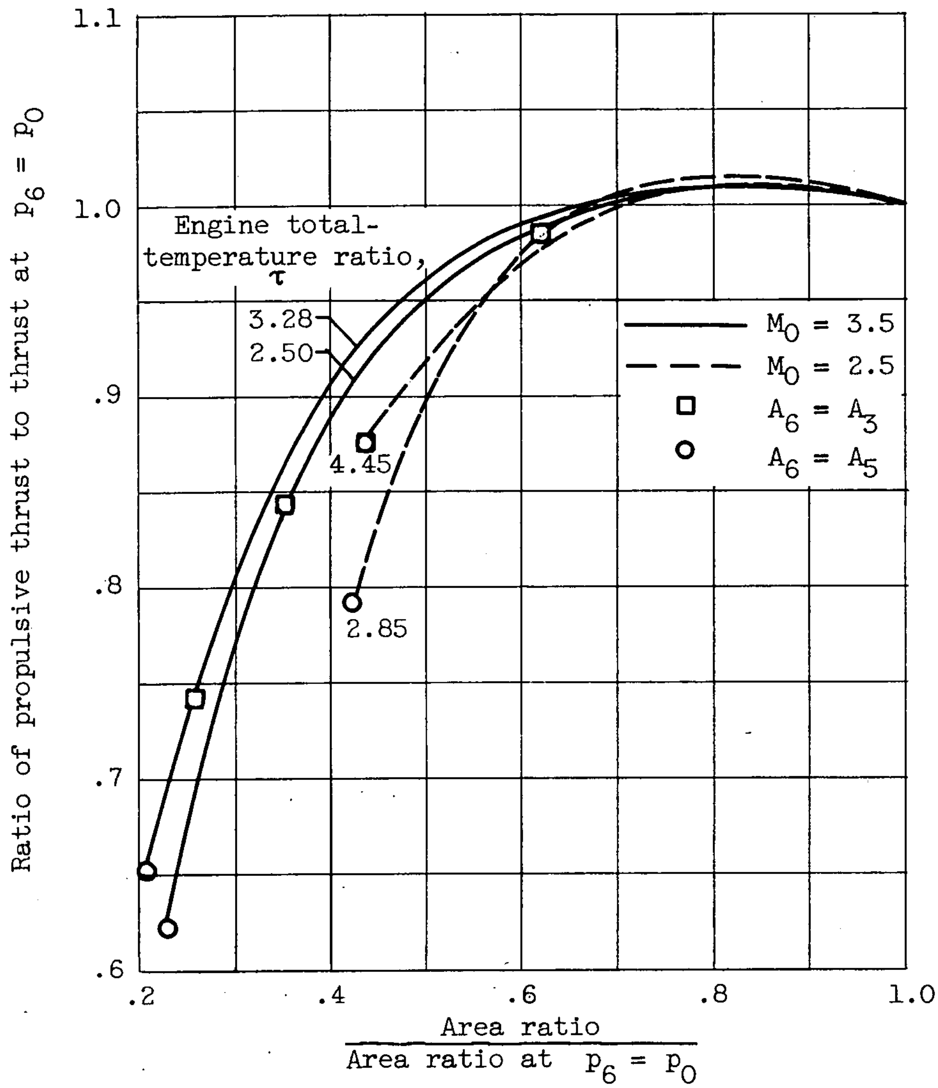


Figure 11. - Effect of nozzle area ratio on ram-jet engine performance. Combustion-chamber-inlet Mach number, 0.200. (Specific impulse is affected in same proportion as thrust.)

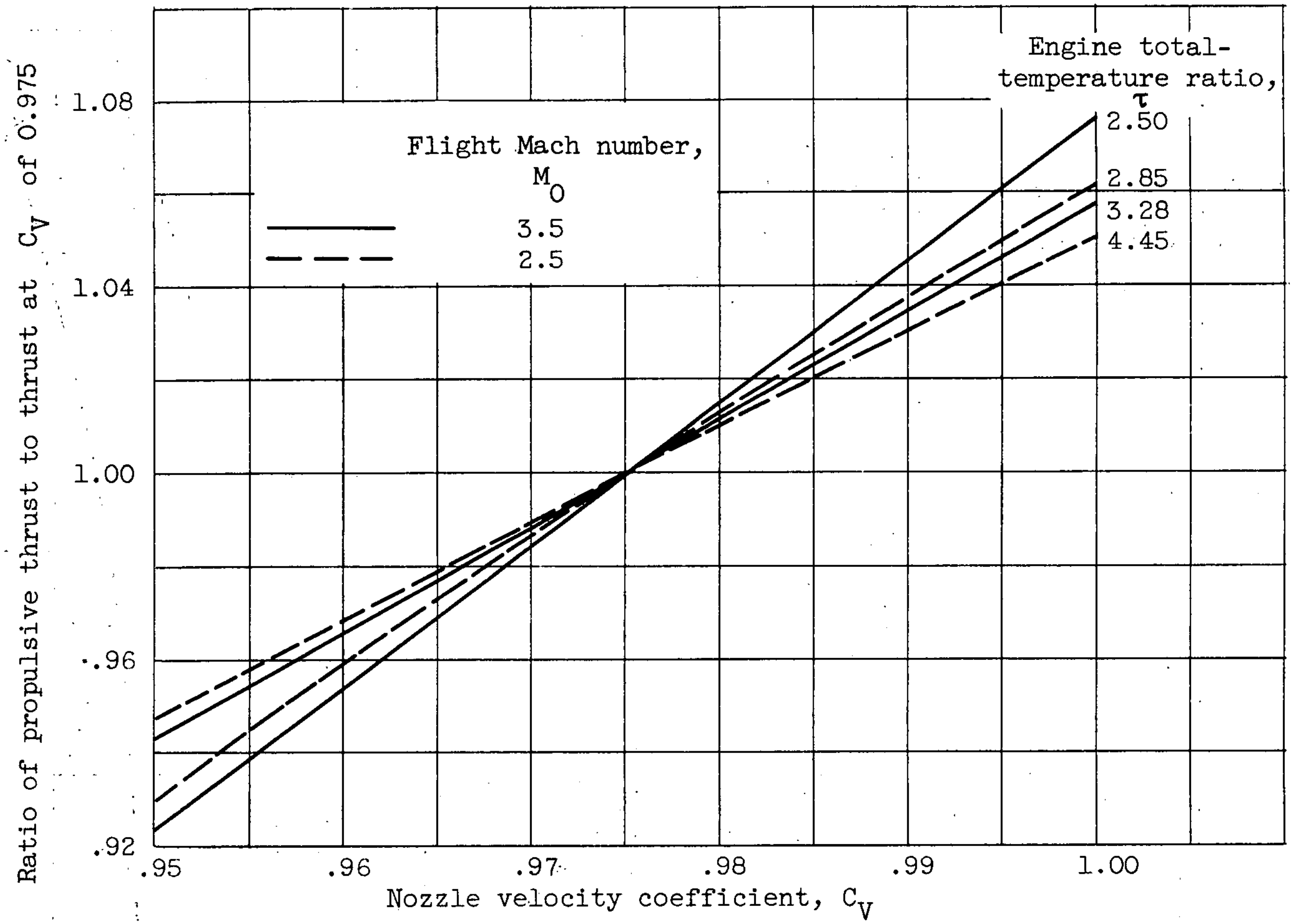


Figure 12. - Effect of nozzle velocity coefficient on ram-jet engine performance. Combustion-chamber-inlet Mach number, 0.200. (Specific impulse is affected in same proportion as thrust.)

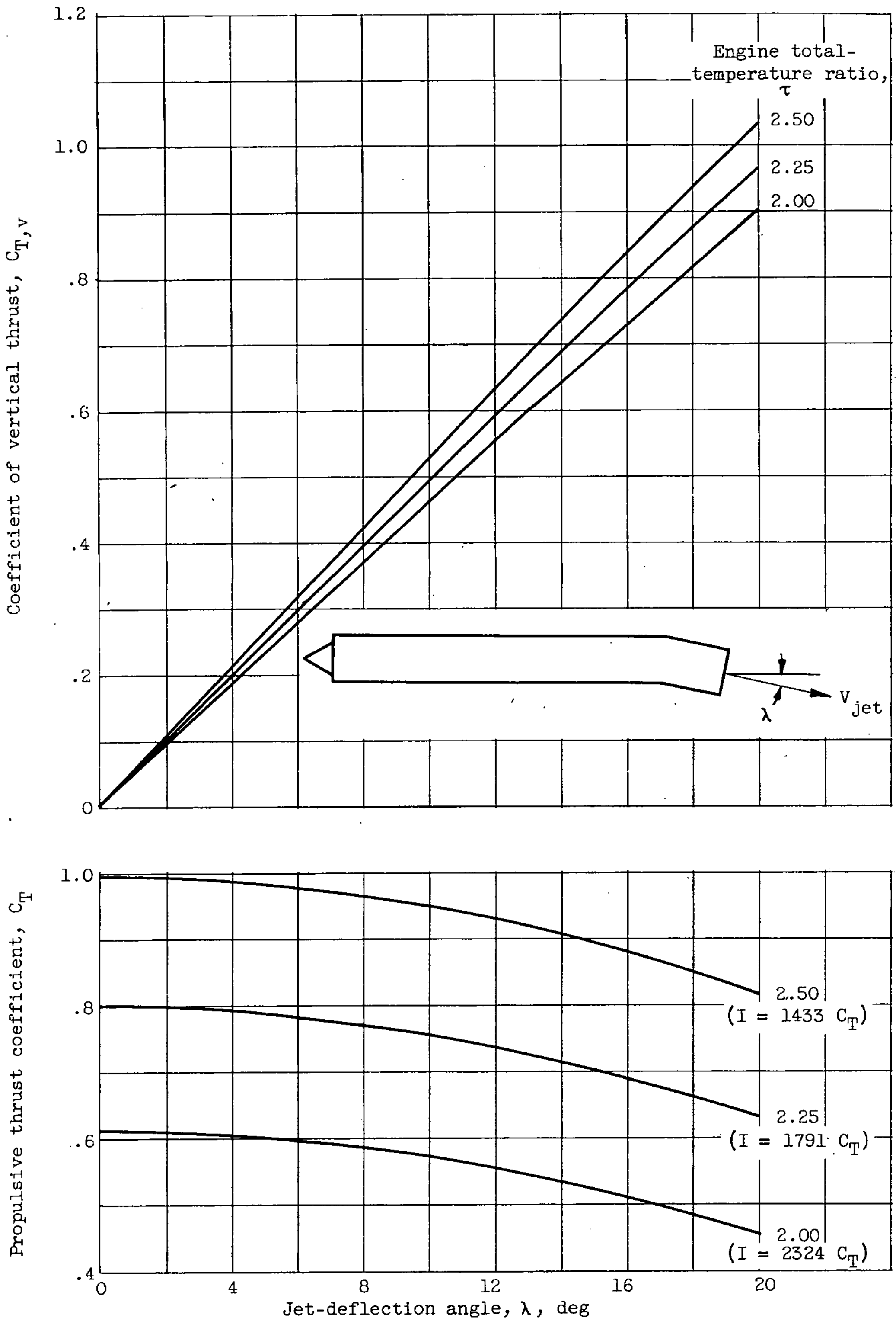


Figure 13. - Effect of jet-deflection angle on ram-jet engine performance. Flight Mach number, 3.5; combustion-chamber-inlet Mach number, 0.200.

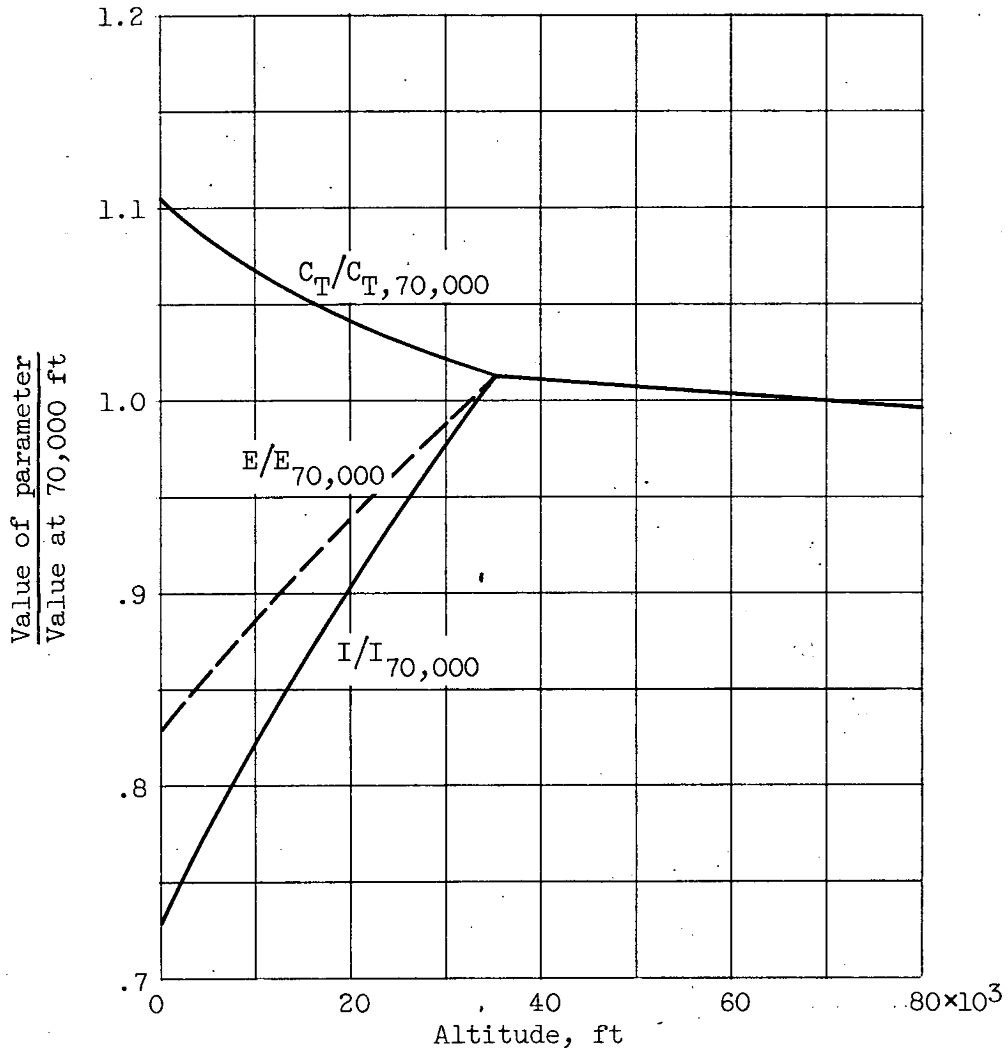


Figure 14. - Effect of altitude on ram-jet engine performance. Flight Mach number, 3.5; combustion-chamber-inlet Mach number, 0.200; engine total-temperature ratio, 2.50.

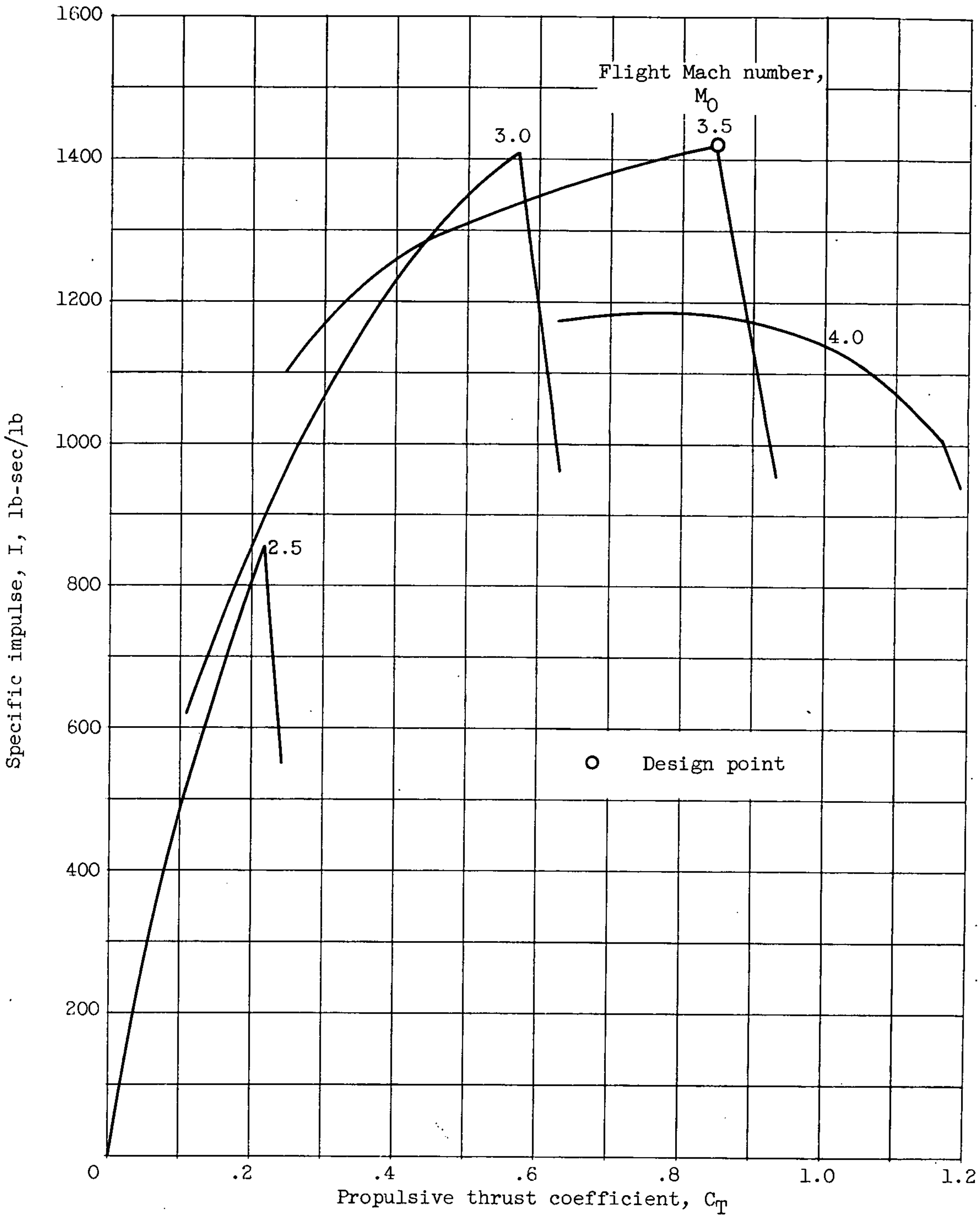


Figure 15. - Off-design performance of fixed-geometry ram-jet engine designed for flight Mach number of 3.5, combustion-chamber-inlet Mach number of 0.200, and engine total-temperature ratio of 2.25. Combustion efficiency, 0.87; ratio of specific heats for exhaust gases, 1.30.

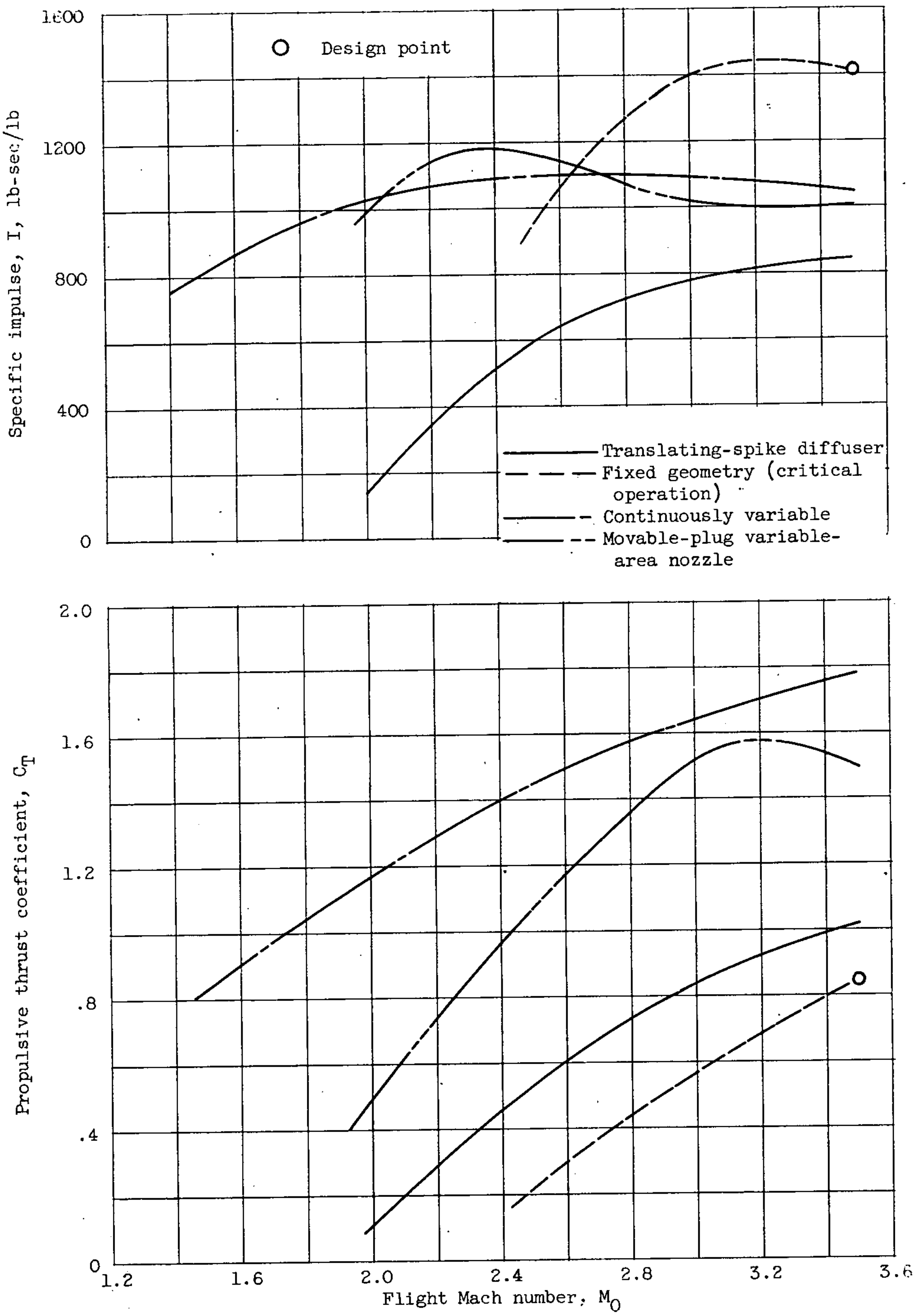


Figure 16. - Off-design performance of various fixed- and variable-geometry ram-jet engines operating at maximum thrust. Engines designed for flight Mach number of 3.5, combustion-chamber-inlet Mach number of 0.200, and engine total-temperature ratio of 2.50. Combustion efficiency, 0.87; ratio of specific heats for exhaust gases, 1.30.