Y3.N21/5:6/2697

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NACA TN 2697

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 2697

METHOD AND GRAPHS FOR THE EVALUATION OF AIR-INDUCTION SYSTEMS

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Washington
May 1952

BUSINESS, SCIENCE
& TECHNOLOGY DEP'T.

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SUMMARY

Graphs have been developed for rapid evaluation of air-induction systems from considerations of their aerodynamic-performance parameters in combination with power-plant characteristics. The graphs cover the range of supersonic Mach numbers up to 3.0. Examples are presented for an air-induction system and engine combination at two Mach numbers and two altitudes in order to illustrate the method and application of the graphs. The examples show that jet-engine characteristics impose restrictions on the use of fixed inlets if the maximum net thrusts are to be realized at all flight conditions.

INTRODUCTION

In order to obtain a true indication of the worth of a given air-induction system as a component of a propulsive unit, it is necessary to employ an evaluation parameter that represents a summation of all the gains and penalties resulting from the use of that particular system. Such a parameter should consider not only the aerodynamics of the entire installation but also such factors as the weight, mechanical complexity, tactical purpose of the aircraft, and many others. Obviously, such a universal parameter is difficult to derive and even more difficult to apply. For this reason, it is convenient to make a partial evaluation based on the aerodynamic considerations before attempting a general evaluation. In such a case, the net thrust or the net thermal efficiency can be used as figures of merit because they provide a measure of the aerodynamic and thermodynamic qualities of the installation. The net thrust represents the force remaining after subtraction of the drag chargeable to the propulsive system from the thrust that it develops. The net thermal efficiency may be obtained from the net thrust, the flight velocity, and the rate of fuel consumption.

The maximum net thrust and thermal efficiency attainable with a jet-engine installation depend greatly on the performance of the air-induction system employed. The characteristics of air-induction systems are usually presented in terms of total-pressure recovery, external drag coefficient, and mass-flow ratio. Unless all three of these parameters for one system excel those for another at supersonic speeds, it is difficult to choose

the better system because of the interdependence of the engine and induction-system parameters. Because of this interdependence, it is necessary to combine the induction system and power-plant characteristics so as to obtain a single figure of merit for the complete installation. By comparing the figures of merit, it is possible to establish the relative aerodynamic worth of each of the air-induction systems considered when they are used with a given engine.

The effects of changes in various parameters on the over-all performance of propulsive systems have been evaluated in the past (see references 1, 2, and 3); however, the scope of each of these investigations was limited because the magnitudes of changes due to variations in parameters were determined for specific engines or specific installations. Therefore, the results have quantitative significance for the assumed installations only and cannot be applied directly to propulsive systems having component characteristics different from those used in the analyses.

The purpose of this report is to present a method of evaluation of various air-induction systems when combined with arbitrary jet engines, and to present graphs that were developed to permit rapid determination of the thrust coefficients of a wide variety of jet-propulsion systems (ram jets, turbojets with afterburning, ducted fans, etc.). The evaluation is based on considerations of the air handling qualities of the induction systems and the component characteristics of engines. The method allows selection of the aerodynamically optimum combination of an induction system and engine for a particular set of flight conditions. Thus, it provides the initial solution in the more general problem that considers the relative merit of systems for a range of flight conditions.

NOTATION

A cross-sectional area at any point of stream tube containing the air flowing through the propulsive system, square feet

a speed of sound, feet per second

C_{D_E} external drag coefficient of propulsive system based on maximum frontal area of installation $\left(\frac{D_E}{q_o S_R}\right)$, dimensionless

C_{D_E}' external drag coefficient of propulsive system based on free-stream cross-sectional area of stream tube entering the inlet $\left(\frac{D_E}{q_o A_o}\right)$, dimensionless

C_{F_i} internal thrust coefficient based on maximum frontal area of installation $\left(\frac{F_i}{q_o S_R}\right)$, dimensionless

C_{F_i}' internal thrust coefficient based on the free-stream cross-sectional area of stream tube entering the inlet $\left(\frac{F_i}{q_o A_o}\right)$, dimensionless

C_{F_N}' net thrust coefficient based on the free-stream cross-sectional area of stream tube entering the inlet $(C_{F_i}' - C_{D_E}')$, dimensionless

$\left(C_{F_N}\right)_P$ net thrust coefficient based on maximum frontal area of engine, dimensionless

D_E external drag force chargeable to propulsive system $\left[D_{(B+S)} - D_B + D_m\right]$, pounds

D_B pressure and friction drag forces acting on the basic body shape (fuselage) without an air inlet, pounds

D_m total momentum of the incoming mass of air at the entrance station less the total momentum of the same mass of air in the free stream $[(p_1 A_1 + m_1 V_1) - (p_o A_o + m_1 V_o)]$, pounds

$D_{(B+S)}$ pressure and friction drag forces acting on the external surface of the combined basic body and the propulsive system, pounds

F_N net thrust force $(F_i - D_E)$, pounds

F_i internal thrust force (rate of momentum change of internal flow between free-stream and the tail-pipe exit where static pressure is assumed equal to the free-stream static pressure), pounds

g acceleration due to gravity, feet per second squared

H total pressure, pounds per square foot

h specific enthalpy, Btu per pound

J mechanical equivalent of heat, 778 foot-pounds per Btu

L.H.V. lower heating value of fuel, Btu per pound

M Mach number, dimensionless

m mass-flow rate, slugs per second

$\frac{m_1}{m_o}$ mass-flow ratio $\left(\frac{\rho_1 V_1 A_1}{\rho_o V_o A_1}\right)$

N_A actual rotational speed of engine, revolutions per minute

N_C corrected rotational speed $\left(N_A\sqrt{\frac{1.4 \times 519}{\gamma T}}\right)$ (where γ and T correspond to stagnation conditions at the inlet of the unit under consideration), revolutions per minute

p static pressure, pounds per square foot

q dynamic pressure, pounds per square foot

S_P maximum frontal area of power plant, square feet

S_R maximum frontal area of installation, square feet

t static temperature, degrees Rankine

T total temperature, degrees Rankine

V speed, feet per second

W weight-flow rate, pounds per second

γ ratio of specific heats

Δ increment

δ pressure correction factor $\left(\frac{H}{p_r}\right)$

η net thermal efficiency of propulsive system (net thrust times the flight speed divided by energy input rate)

$\eta_c\eta_t$ combined adiabatic efficiency of compressor and turbine

$\sqrt{\frac{1}{\theta}}$ temperature-correction factor $\left[\left(\frac{1.40 \times 519}{\gamma T}\right)^{\frac{1}{2}}\right]$

ρ mass density, slugs per cubic foot

Subscripts

A actual

a air

c compressor

cc corrected compressor parameter

ct corrected turbine parameter

D	drag
E	external
F	thrust
f	fuel
H	stagnation conditions
i	internal (within boundaries of stream tube entering the inlet)
M	conditions corresponding to flight Mach number
N	net
opt	optimum conditions (conditions of best performance)
P	power plant
r	at standard sea-level static conditions
R	reference (such as frontal area, used in drag coefficient)
t	turbine
0,1,2, 3,...11	station, as shown in figure 1

METHOD

The method of evaluation consists of determining the maximum net thrust coefficient based on the frontal area of the engine at various conditions of flight. The present report considers primarily the net thrust coefficient, because the net thermal efficiency depends directly on the net thrust coefficient (see equation (A8) of appendix A); the evaluation at a given flight condition leads to the same conclusions regardless of which of the two parameters is used. The net thermal efficiency is useful in evaluating complete flight plans when range and endurance must be considered.

In order to evaluate an induction system, the following information must be available:

1. The induction-system characteristics H_3/H_0, m_1/m_0, and C_{D_E} (which represents the total drag chargeable to the propulsive system) for various Mach numbers and ratios of inlet area to frontal area of the installation

2. The engine characteristics, such as the exhaust-to-inlet pressure and temperature ratios and the air handling capacity (volume of air consumed per second)[1]

With this information, an inlet size can be selected for the propulsion system such that it operates at any desired mass-flow ratio and provides a constant volume of air as required by the engine. The inlet size and the mass-flow ratio in turn determine the external drag of the system. By the use of the induction-system and engine-performance data corresponding to the operating condition, it is possible to calculate the net thrust coefficient.

In general, the method entails the following steps:

1. The thrust coefficient C_{F_i}', based on the free-stream area A_o of the air required by the engine, is calculated for the given engine as a function of total-pressure recovery for the flight Mach number and altitude. This computation is performed using the induction-system characteristics (H_3/H_o as a function of m_1/m_o), the graphs presented in this report, and equation (A3) of appendix A.

2. The drag coefficient attributable to the propulsive system at various mass-flow ratios, C_{D_E}, is transformed to C_{D_E}' by multiplying C_{D_E} by the ratio S_R/A_o. (See equation (A4) of appendix A.)

3. The net thrust coefficient C_{F_N}' (based on A_o) at various mass-flow ratios is obtained by subtracting the drag coefficient C_{D_E}' (step 2) from the thrust coefficient C_{F_i}' (step 1). (See appendix A, equation (A6).)

4. The net thrust coefficient $(C_{F_N})_P$ based on the frontal area of the engine is obtained by transforming C_{F_N}' (step 3) by means of equation (A7) of appendix A.

After these calculations are performed for the range of operational mass-flow ratios, the maximum net thrust coefficient attainable with the given induction-system and engine combination is found from the plot of $(C_{F_N})_P$ as a function of m_1/m_o.

Determination of the Maximum Net Thrust Coefficient

Many air-induction systems produce highest total-pressure recoveries at mass-flow ratios less than the maximum attainable at a given supersonic

[1]The ram-jet and turbojet engines operate essentially with constant-volume flow since the velocity past the flame holder and at the compressor intake must remain constant at the maximum allowable values. This condition is required in order to obtain the necessary thrust with an engine of the least frontal area.

Mach number. As the mass-flow ratio is reduced from its maximum value until the maximum total-pressure recovery is attained, the thrust coefficient C_{F_i}' increases due to rising recovery and so does the drag coefficient C_{D_E}' due to increasing additive drag (see reference 4). Thus, the total-pressure recovery and drag have opposite effects on the net thrust coefficient C_{F_N}' within the range of mass-flow ratios limited by conditions of the least drag and the highest total-pressure recovery. To attain the maximum net thrust coefficient it is necessary, therefore, to provide the air required by the engine at an optimum mass-flow ratio that provides the best compromise between thrust and drag.

Determination of internal thrust coefficient, C_{F_i}'.- When calculations of the net thrust coefficient are made in order to find the optimum mass-flow ratio, it is convenient to consider the flow field existing about a propulsive system to be divided in two parts, internal and external, the boundary being that of the stream tube surrounding the air which enters the system. The flows within these two regions may be analyzed separately, and the results can be combined to obtain a figure of merit for the complete system. It is shown in appendix A (equation (A3)) that when the thrust due to internal flow is expressed in coefficient form (C_{F_i}') based on the free-stream area of the stream tube entering the induction system (A_o), the significance of the quantity of air required by the engine disappears. Then, the magnitude of C_{F_i}' depends only on the ratio of the exhaust and flight velocities. (The contribution of the mass of fuel burned, W_f/W_a, to the thrust, C_{F_i}', is usually on the order of only 5 percent at rated conditions.) This fact makes it possible to isolate the effects of the total-pressure recovery on the thermodynamic cycle which determines the magnitude of the internal thrust coefficient C_{F_i}' attainable with given engine characteristics at a fixed Mach number and altitude. To reduce computational effort and to make the determination of the internal thrust coefficient universal, graphs based on the thermodynamic cycles of jet engines have been developed.

The processes undergone by the internal air flow from one station of a propulsion system to another have been represented graphically in figures 2 to 4 using the tables and methods of reference 5. (See example 16 of reference 5 for illustration of the method of solution for various states of the gas.) The subscript numerals designate the stations shown in figure 1. The assumptions used are independent of any actual installation. They are as follows:

1. For a turbojet engine, the specific enthalpy change through the turbine is equal to the specific enthalpy rise in the compressor (calculated as if the compression were isentropic) divided by the product of the adiabatic efficiencies (the combined efficiency) of the two units.

2. The air-fuel ratios are those necessary to maintain the assigned combustion temperature by complete combustion of a fuel with the lower heating value (L.H.V.) equal to 19,000 Btu per pound.

3. The flow in the exhaust nozzle is isentropic; it has the properties of air at the exhaust temperatures and leaves the tail pipe at free-stream static pressure at all flight conditions. (This assumption requires an adjustable exhaust nozzle with variable throat and exit areas A_{10} and A_{11}.)

Flight and exhaust velocities can be determined from figure 2 for ram jets and also for turbojets if the engine performance is available in the generalized form (H_9/H_3 and T_9/T_3) suggested in reference 6. If not, figures 3 to 5 must be used to find the effects of engine operation on the fluid conditions at the engine outlet or at the entrance of the exhaust nozzle. Quadrant I of figure 2 presents the variation of the free-stream Mach number with the speed of flight at various altitudes. Quadrant II yields the ratio of recovered total pressure to free-stream static pressure (H_3/p_0) as a function of total-pressure recovery (H_3/H_0) of the air-induction system. Quadrant III shows the highest exhaust Mach number obtainable with the available pressure ratio (H_9/p_0), which includes mechanical compression due to the power plant, as a function of the total temperature of the exhaust. The last quadrant provides the exhaust velocity corresponding to the exhaust Mach number and the total temperature T_9 when $p_{11} = p_0$. A correction for the exhaust-nozzle losses can be applied to the exhaust velocity, which was calculated on the assumption of isentropic flow, if the actual-to-theoretical jet-speed ratio is known. (See reference 4.)

The temperature graph, from which the total temperature at various stations throughout the propulsive system can be determined, is shown in figure 3. The effects of altitude, flight Mach number, mechanical compression, and burning of the fuel are included in this figure. Quadrant IV can be used to find the temperatures or the amounts of fuel consumed in afterburning as well as in the main combustion chambers. The effects of incomplete combustion or of a different fuel heating value on fuel consumption can be taken into account by direct ratios of combustion efficiencies or of the heating values. The temperature graph is used in conjunction with the compressor-turbine graph shown in figure 4 from which the turbine expansion ratio necessary to drive the compressor can be determined. This figure also provides the resulting total temperature at the turbine outlet (T_6) for a given inlet temperature.

Figure 5 presents the variation of the temperature correction factor with temperature at the compressor or turbine inlet. This figure is used to calculate the engine operational conditions from the performance parameters corrected to standard sea-level conditions. The effect of temperature on the ratio of specific heats for air has been included in the temperature correction factor for the compressor because the stagnation temperature of the free stream varies sufficiently to cause error if it were neglected.

The graphs described above allow determination of the velocities and air-fuel rates that must be known in order to calculate the internal thrust

thrust coefficient C_{F_i}'. The coefficient C_{F_i}' is computed by substituting these quantities in equation (A3) of appendix A.

Determination of the external drag coefficient C_{D_E}'.- The total drag chargeable to a propulsive unit consists of drag attributable to the induction system, to the modification of the airframe necessary to house the engine, and to the interference of the pressure field of the propulsive system with other components of the aircraft. The magnitude of drag chargeable to the induction system depends on the amount of diffusion ahead of the inlet, which is a function of mass-flow ratio (see reference 7), and on the geometric proportions of the induction system which may be described, in general, by the length of the induction system and the ratio of inlet area to the frontal area of the installation (A_1/S_R). The inlet area necessary to provide the air required by the engine depends on the inlet mass-flow ratio; if the optimum mass-flow ratio is unknown, the value of the ratio A_1/S_R necessary for optimum operation is also unknown. Thus, when the optimum mass-flow ratio is being calculated, it is necessary to compute the variation of the net thrust coefficient $(C_{F_N})_P$ with mass-flow ratio for several values of A_1/S_R which are between the value for the inlet operating at the maximum mass-flow ratio and that required at the mass-flow ratio for maximum total-pressure recovery.

In practice, induction-system characteristics are usually obtained from tests with models having a fixed A_1/S_R ratio. If data for various A_1/S_R ratios are unavailable, it is necessary to estimate the effects of A_1/S_R ratio on the drag and total-pressure recovery. Since the A_1/S_R ratio approximately establishes the angle of inclination of the external surface of the induction system with respect to the flow direction, the surface-pressure drag is affected the most by changes in inlet size. Since the surface-pressure drag usually represents only a small portion of the total drag of a propulsive system and the angle of surface inclination changes little with moderate variations in A_1/S_R ratio, the effects of the ratio A_1/S_R can often be neglected.

Determination of Inlet Size For Optimum Operation

To develop maximum thrust at a given flight condition with an engine operating with a fixed volumetric capacity, it is necessary to match the engine and the induction system so that the latter operates at the optimum mass-flow ratio. This condition is attained when the induction-system inlet area (A_1) is such that the cross-sectional area of the free-stream tube entering the inlet (A_0) is equal to that required by the engine. For a fixed volumetric capacity, the area A_0 required by the engine varies directly with the total-pressure recovery at the exit of the induction system. This area (A_0) can be found using relations given in appendix B

(see equations (B1) and (B8)) and the engine characteristics. The inlet area (A_1) can be found from the equation defining the mass-flow ratio, that is, $m_1/m_0 = A_0/A_1$.

The engine air requirements change with Mach number and altitude, and the optimum mass-flow ratio usually changes with Mach number. Thus, it is unlikely that any one propulsive system will develop the maximum net thrust at all flight conditions unless the inlet area is adjustable. If the engine air requirements are adjusted to the characteristics of the induction system having a constant inlet area by means of engine-speed control, the propulsive system will not develop the attainable maximum net thrust at off-design flight conditions.[2] Appendix C presents the relations between the mass-flow ratio, the total-pressure recovery, and the engine air requirements at various engine speeds; these relations provide the information necessary for computation of engine performance at part-throttle operation.

ILLUSTRATIVE EXAMPLES

Application of Graphs

The use of the graphs can be demonstrated best by illustrative examples. For this purpose a ram jet and a turbojet with and without afterburning have been selected since they represent the presently popular types of propulsion systems. When the charts are used for determining the internal thrust coefficients of actual installations, the actual induction system and engine characteristics are used; in the present report, these characteristics have been assumed. The computations were performed for various total-pressure recoveries (H_3/H_0 from 1.0 to 0.4) and Mach numbers (M_0 from 1.0 to 3.0) in order to obtain data for the example induction-system calculations presented later in this report.

Ram jet.- The steps of solution, presented in tabular form, for a ram jet flying at a Mach number 2.0 in the isothermal region of the atmosphere are as follows (the assumed values are indicated by asterisks):

[2]Since the engine ordinarily runs at the maximum speed allowable for continuous operation (rated rpm), the speed control can only reduce the speed. Thus, the weight of air handled per second and the operating temperature decrease with the result that thrust decreases.

Item	Source	Units	Quantity
V_o	Fig. 2, quadrant I	$\frac{ft}{sec}$	1940
H_3/H_o	Induction system characteristics	None	0.80*
H_3/p_o	Fig. 2, quadrant II	None	6.25
T_{o-3}	Fig. 3, quadrants I and II	^oR	708
T_9	Operating temperature	^oR	3000*
$\left(\frac{W_a}{W_f}\right)_8$	Fig. 3, for T_9 and T_{o-3}	$\frac{\text{lb air}}{\text{lb fuel}}$	29.6
H_9/p_o	$\left(\frac{H_3}{p_o}\right)\left(\frac{H_9}{H_3}\right)$	None	6.25
V_{11}	Fig. 2, quadrants III and IV for H_9/p_o and T_9	None	3913
C_{F_i}'	$2\left[\left(1 + \frac{W_f}{W_a}\right)\frac{V_{11}}{V_o} - 1\right]$	None	2.170

The loss of total pressure between stations 3 and 9 was neglected. However, such loss can be taken into account by reducing the value of H_9/H_3 by the amount of loss.

Figure 6 shows the variation of the internal thrust coefficient C_{F_i}' with the total-pressure recovery and Mach number computed using the procedure outlined in the sample calculation. The temperature of combustion was assumed to be 3000° R throughout the range of Mach numbers. It must be remembered in the use of this figure that, since a fixed temperature was used, either the size of the inlet or the mass-flow ratio must reduce with decreasing total-pressure recovery or the critical area of the exhaust nozzle must be increased to compensate for the greater specific volume of the air handled.

Turbojet.- In order to find the variation of the internal thrust coefficient for the turbojet without and with afterburning, the generalized characteristics of the engine shown in figure 7 were used. Constant actual speed N_A of 12,500 rpm (rated speed at standard intake conditions) was assumed for the entire range of operation in both cases. The compressor characteristics were selected so that the pressure ratio at Mach number 1.4 would be 6.25. (The reasons for selecting this pressure ratio are discussed in appendix D.) The effects of Reynolds number index (the ratio

of Reynolds number to Mach number, see reference 6) on the compressor characteristics have been neglected. The turbine characteristics necessary to satisfy the engine operational requirements were determined using figures 3, 4, and 5 upon assumption of the variation of the adiabatic efficiency η_t with corrected turbine speed for sea-level static conditions. The operational limits were fixed by the assumptions of: (1) the combustion temperature (2000° R) at rated engine speed; (2) the equality of actual rotational speeds of compressor and turbine; and (3) the equality of compressor and turbine pressure ratios at the idling speed.

The internal thrust coefficients for the turbojet with afterburning were computed on the assumption that the total fuel consumption (engine plus afterburner) per pound of air is equal to that of the ram jet at the same Mach number. The steps of solution for the turbojet without and with afterburning are given in chronological order in table I which contains the computations for flight at Mach number 2.0 in the isothermal region of the atmosphere.

As in the case of the ram jet, the losses of total pressure due to combustion and friction $\left(\frac{H_5}{H_4}, \frac{H_7}{H_6}, \frac{H_8}{H_7} \text{ and } \frac{H_9}{H_8}\right)$ were neglected. Such losses, however, can be readily accounted for when H_9/p_0 is computed.

The variation of the internal-thrust coefficient C_{F_i}' with total-pressure recovery computed for a range of Mach numbers is shown in figure 8. The variations of the internal thrust coefficients with Mach number and total-pressure recovery were calculated for the engine characteristics pertaining to a fixed actual engine speed. Thus, as the total-pressure recovery decreases at a given Mach number and altitude, the size of the air inlet or the mass-flow ratio must be reduced for a fixed-size engine.

Calculation of Optimum Mass-Flow Ratio And Inlet Area

To illustrate the calculation of the maximum net thrust coefficient and of the required inlet area, the assumed characteristics of an air-induction system will be combined with the assumed engine characteristics shown in figure 7. In addition, the flight conditions of the example calculations will be selected so as to demonstrate the effects of Mach number and altitude on the maximum net thrust coefficient, the optimum mass-flow ratio, and the inlet area necessary for optimum performance.

Example 1.- Flight is in the isothermal region of the atmosphere at Mach number 2.5. The engine operates at the rated actual speed and uses afterburning to the extent that the total fuel consumption is equal to that of a ram jet having a 3000° R combustion temperature T_{8-9}. (See figs. 1 and 6.)

Example 2.- Flight is in the isothermal region of the atmosphere at Mach number 1.2. The engine operates at the rated actual speed and uses no afterburning.

Example 3.- Flight is at sea level at Mach number 1.2. The engine operates at the rated actual speed and uses no afterburning.

To simplify presentation, the effects of the ratio of inlet area to the maximum frontal area of the installation (A_1/S_R) on the drag coefficient and total-pressure recovery will be neglected; that is, the characteristics for a fixed (A_1/S_R) ratio will be used to find the optimum mass-flow ratio. The engine frontal area S_P will be assumed equal to 70 percent of S_R, and $A_1/S_R = 0.20$ will be used in all examples; it will also be assumed that the engine operates at a constant speed (12,500 rpm) at all times, and that its size is such that it requires $A_0 = 1$ square foot when $H_3/H_0 = 1.0$ and $M_0 = 1.0$ at sea level, that is $(W_a)_r = 54.6$ lb/sec.

The assumed characteristics of the induction system to be evaluated are shown in figures 9 and 10. This system, designed for $M_0 = 2.5$, uses one oblique shock wave ahead of the entrance and a normal shock wave just inside the inlet at the maximum mass-flow ratio. The A_1/S_R ratio is typical of induction systems of the side-scoop type. At $M_0 = 1.2$, the maximum mass-flow ratio is less than at $M_0 = 2.5$ because the normal shock wave is well ahead of the entrance and a considerable portion of flow is deflected past the inlet. For the same reason, the minimum external drag coefficient C_{D_E} is higher as a result of a greater additive drag coefficient (see reference 7). The total-pressure recovery is higher at $M_0 = 1.2$ because the normal shock wave occurs at a lower Mach number.

In all cases the solution follows the outline given in the section entitled "Method" of this report.

Step 1: The variation of the internal thrust coefficient C_{F_i}' with H_3/H_0 at M_0 and flight altitude is calculated. Figure 11 shows the internal thrust coefficient C_{F_i}' for example 1 obtained from the cross plot of data of figure 8 for $M_0 = 2.5$.

Step 2: The external drag coefficients shown in figure 9, when converted to A_0 using equation (A4) of appendix A and $A_1/S_R = 0.20$, assume the values shown in figure 11.

Step 3: The net thrust coefficient C_{F_N}' is obtained by subtracting C_{D_E}' from C_{F_i}' of figure 11.

Step 4: The net thrust coefficient $(C_{F_N})_P$, obtained using the relation

$$(C_{F_N})_P = C_{F_N}' \left(\frac{A_1}{S_P}\right)\left(\frac{m_1}{m_0}\right) = C_{F_N}' \left(\frac{A_1}{S_R}\right)\left(\frac{m_1}{m_0}\right)\left(\frac{S_R}{S_P}\right) = 0.286\, C_{F_N}' \left(\frac{m_1}{m_0}\right)$$

and the values found in step 3, is shown in figure 12 for example 1.

Similar calculations for examples 2 and 3 yield the results shown in figure 13. In the case of example 3, curves similar to those of figure 8 for sea-level flight must be used.

The free-stream-tube area necessary to supply the air required by the turbojet engine of the illustrative examples is shown in figure 14 for various Mach numbers and total-pressure recoveries at two altitudes. The curves shown were obtained using equation (B8) of appendix B. The inlet areas required at each of the three flight conditions to produce maximum net thrust coefficients $(C_{F_N})_P$ were obtained by dividing the free-stream areas A_o, required by the engine when the total-pressure recoveries are equal to those at $\left(\frac{m_1}{m_o}\right)_{opt}$, by the respective $\left(\frac{m_1}{m_o}\right)_{opt}$

The results of the above calculations are summarized in the following table:

Example	$\left(\frac{m_1}{m_o}\right)_{opt}$	Maximum $(C_{F_N})_P$	Optimum inlet area $(A_1)_{opt}$ (sq ft)
1	1.65	0.750	0.794
2	.94	.450	1.075
3	.94	.149	.915

CONCLUDING REMARKS

The method and graphs allow rapid evaluation of air-induction systems from consideration of their aerodynamic parameters in combination with engine characteristics. In addition, the method and graphs simplify considerably the problem of determining the optimum mass-flow ratio to be used in cases when the maximum total-pressure recovery does not occur at the maximum mass-flow ratio.

The illustrative examples indicate that the inlet area required for optimum performance must change with altitude at a fixed Mach number as well as with Mach number at a fixed altitude. Thus, it is unlikely that any one propulsive system will develop the attainable maximum net thrust throughout the range of Mach numbers and altitudes unless some means of varying the inlet area are employed.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif., Feb. 19, 1952

APPENDIX A

RELATIONS DESCRIBING NET THRUST COEFFICIENT AND NET THERMAL EFFICIENCY

The internal thrust force and the internal thrust coefficient are given by the following relations when the exhaust pressure is equal to free-stream static pressure:

$$F_i = \frac{W_a}{g}\left[\left(1 + \frac{W_f}{W_a}\right)V_{11} - V_o\right] = \rho_o A_o V_o^2\left[\left(1 + \frac{W_f}{W_a}\right)\frac{V_{11}}{V_o} - 1\right] = 2q_o A_o\left[\left(1 + \frac{W_f}{W_a}\right)\frac{V_{11}}{V_o} - 1\right] \tag{A1}$$

$$C_{F_i} = \frac{F_i}{q_o S_R} = 2\,\frac{A_o}{S_R}\left[\left(1 + \frac{W_f}{W_a}\right)\frac{V_{11}}{V_o} - 1\right] \tag{A2}$$

where W_f/W_a represents the total amount of fuel consumed per pound of air handled. When based on the free-stream area of the stream tube entering the duct, the internal thrust coefficient is given by

$$C_{F_i}' = C_{F_i}\frac{S_R}{A_o} = 2\left[\left(1 + \frac{W_f}{W_a}\right)\frac{V_{11}}{V_o} - 1\right] \tag{A3}$$

Similarly,

$$C_{D_E}' = C_{D_E}\frac{S_R}{A_o} = C_{D_E}\frac{S_R}{A_1\dfrac{m_1}{m_o}} \tag{A4}$$

where $\dfrac{m_1}{m_o} = \dfrac{\rho_1 V_1 A_1}{\rho_o V_o A_1}$ by definition. By continuity, $\rho_1 V_1 A_1 = \rho_o V_o A_o$, and thus $\dfrac{m_1}{m_o} = \dfrac{\rho_o V_o A_o}{\rho_o V_o A_1} = \dfrac{A_o}{A_1}$.

The net thrust coefficient based on A_o is given by

$$C_{F_N}' = C_{F_i}' - C_{D_E}' \tag{A5}$$

thus,

$$C_{F_N}' = 2\left[\left(1 + \frac{W_f}{W_a}\right)\frac{V_{11}}{V_o} - 1\right] - C_{D_E}\frac{S_R}{A_1\dfrac{m_1}{m_o}} \tag{A6}$$

The net thrust coefficient based on the engine frontal area is related to C_{F_N}' by the following relation:

$$(C_{F_N})_P = C_{F_N}'\left(\frac{A_1}{S_P}\right)\left(\frac{m_1}{m_o}\right) = C_{F_N}'\left(\frac{A_1}{S_R}\right)\left(\frac{S_R}{S_P}\right)\left(\frac{m_1}{m_o}\right) \tag{A7}$$

The net thermal efficiency of a propulsive system is given as

$$\eta = \frac{F_N V_o}{[(L.H.V.)W_f]J}$$

where

$$F_N = C_{F_N}' \frac{1}{2} \rho_o V_o^2 A_o = \frac{1}{2} C_{F_N}' (\rho_o V_o A_o) V_o = \frac{1}{2} C_{F_N}' \frac{W_a}{g} V_o$$

thus

$$\eta = \frac{1}{2} \frac{C_{F_N}' V_o^2}{(L.H.V.)gJ} \frac{W_a}{W_f} \tag{A8}$$

where W_a/W_f is the weight ratio of air to fuel for the complete installation.

If fuel is added at stations A and B and the local W_a/W_f ratios are known, it can be shown that

$$\left(\frac{W_a}{W_f}\right)_{total} = \frac{\left[\left(\frac{W_a}{W_f}\right)_A + 1\right]\left[\left(\frac{W_a}{W_f}\right)_B + 1\right]}{\left(\frac{W_a}{W_f}\right)_A + \left(\frac{W_a}{W_f}\right)_B + 2} - 1$$

since for one pound of mixture (see reference 5)

$$\left(\frac{W_f}{W_a + W_f}\right)_{total} = \left(\frac{W_f}{W_a + W_f}\right)_A + \left(\frac{W_f}{W_a + W_f}\right)_B$$

Similarly, solving for the air-to-fuel ratio necessary to make up a total ratio,

$$\left(\frac{W_a}{W_f}\right)_B = \frac{\left[\left(\frac{W_a}{W_f}\right)_A + 1\right]\left[\left(\frac{W_a}{W_f}\right)_{total} + 1\right]}{\left(\frac{W_a}{W_f}\right)_A - \left(\frac{W_a}{W_f}\right)_{total}} - 1 \tag{A9}$$

APPENDIX B

RELATION BETWEEN ENGINE REQUIREMENTS AND THE OPTIMUM INLET AREAS AT VARIOUS FLIGHT CONDITIONS

The corrected air flow given by the relation

$$(W_a)_{cc} = \left(W_a \frac{\sqrt{\theta_3}}{\delta_3}\right)_M$$

can be plotted in the form of a ratio to the rated corrected value

$$(W_a)_r = \left(W_a \frac{\sqrt{\theta_3}}{\delta_3}\right)_r$$

Thus

$$\frac{(W_a)_{cc}}{(W_a)_r} = \frac{\left(W_a \frac{\sqrt{\theta_3}}{\delta_3}\right)_M}{\left(W_a \frac{\sqrt{\theta_3}}{\delta_3}\right)_r}$$

or

$$(W_a)_M = \left(W_a \frac{\sqrt{\theta_3}}{\delta_3}\right)_r \frac{(W_a)_{cc}}{(W_a)_r} \left(\frac{\delta_3}{\sqrt{\theta_3}}\right)_M \tag{B1}$$

where $(W_a)_M = g\rho_o V_o A_o$ is the actual weight rate of air flow through the engine at a given Mach number. For any two Mach numbers M_1 and M_2 the weight rates are thus related by the ratio

$$\frac{(W_a)_{M_1}}{(W_a)_{M_2}} = \frac{\left[\frac{(W_a)_{cc}}{(W_a)_r}\left(\frac{\delta_3}{\sqrt{\theta_3}}\right)\right]_{M_1}}{\left[\frac{(W_a)_{cc}}{(W_a)_r}\left(\frac{\delta_3}{\sqrt{\theta_3}}\right)\right]_{M_2}} \tag{B2}$$

for a given engine.

The weight flow in the free stream through an area equal to that of inlet passage is given by the relation $g\rho_o V_o A_1 = g\rho_o M_o A_1 a_o$ and the weight entering the inlet is given by

$$(W_a)_M = \left[(g\rho_o M_o A_1 a_o)\frac{m_1}{m_o}\right]_M \tag{B3}$$

At any two Mach numbers M_1 and M_2

$$\frac{(Wa)_{M_1}}{(Wa)_{M_2}} = \frac{\left(g\rho_o M_o A_1 a_o \frac{m_1}{m_o}\right)_{M_1}}{\left(g\rho_o M_o A_1 a_o \frac{m_1}{m_o}\right)_{M_2}} \tag{B4}$$

If the two static pressures and temperatures are equal (fixed standard altitude)

$$\frac{(Wa)_{M_1}}{(Wa)_{M_2}} = \frac{\left(A_1 M_o \frac{m_1}{m_o}\right)_{M_1}}{\left(A_1 M_o \frac{m_1}{m_o}\right)_{M_2}} \tag{B5}$$

The flow through the inlet must be equal to the flow through the engine for a matched condition. Thus,

$$\frac{\left(A_1 M_o \frac{m_1}{m_o}\right)_{M_1}}{\left(A_1 M_o \frac{m_1}{m_o}\right)_{M_2}} = \frac{\left[\frac{(Wa)_{cc}}{(Wa)_r}\frac{\delta_3}{\sqrt{\theta_3}}\right]_{M_1}}{\left[\frac{(Wa)_{cc}}{(Wa)_r}\frac{\delta_3}{\sqrt{\theta_3}}\right]_{M_2}}$$

since $\delta_3 = \frac{H_3}{p_o}\frac{p_o}{p_r}$ and $\frac{p_o}{p_r}$ = constant for a fixed altitude,

$$\frac{(A_1)_{M_2}}{(A_1)_{M_1}} = \frac{\left[M_o\left(\frac{m_1}{m_o}\right)_{opt}\right]_{M_1}}{\left[M_o\left(\frac{m_1}{m_o}\right)_{opt}\right]_{M_2}} \frac{\left[\frac{(Wa)_{cc}}{(Wa)_r}\frac{H_3}{p_o}\sqrt{\frac{1}{\theta_3}}\right]_{M_2}}{\left[\frac{(Wa)_{cc}}{(Wa)_r}\frac{H_3}{p_o}\sqrt{\frac{1}{\theta_3}}\right]_{M_1}} \tag{B6}$$

or

$$\frac{(A_1)_{M_2}}{(A_1)_{M_1}} = \frac{\left[M_o\left(\frac{m_1}{m_o}\right)_{opt}\right]_{M_1}}{\left[M_o\left(\frac{m_1}{m_o}\right)_{opt}\right]_{M_2}} \frac{\left[\frac{(Wa)_{cc}}{(Wa)_r}\frac{H_3}{H_o}\frac{H_o}{p_o}\sqrt{\frac{1}{\theta_3}}\right]_{M_2}}{\left[\frac{(Wa)_{cc}}{(Wa)_r}\frac{H_3}{H_o}\frac{H_o}{p_o}\sqrt{\frac{1}{\theta_3}}\right]_{M_1}}$$

The values of $\frac{(Wa)_{cc}}{(Wa)_r}$ are given by the engine (compressor) characteristics for corrected compressor speed N_{cc}, which may be found using figures 3 and 5. The rest of the terms are determined from the inlet characteristics and flight conditions with the help of figures 2, 3, and 5.

For flight at different altitudes and Mach numbers it can be shown, similarly to equation (B6), that

$$\frac{(A_1)_{M_2}}{(A_1)_{M_1}} = \frac{\left[M_o\left(\frac{m_1}{m_o}\right)_{opt}\frac{p_o}{H_o}\right]_{M_1}\left[\frac{(Wa)_{cc}}{(Wa)_r}\frac{H_3}{H_o}\frac{a_o}{a_H}\right]_{M_2}}{\left[M_o\left(\frac{m_1}{m_o}\right)_{opt}\frac{p_o}{H_o}\right]_{M_2}\left[\frac{(Wa)_{cc}}{(Wa)_r}\frac{H_3}{H_o}\frac{a_o}{a_H}\right]_{M_1}} \tag{B7}$$

where $a_H = \sqrt{\gamma_3 g R T_3}$ is the speed of sound at stagnation conditions.

The free-stream area A_O required by the engine at various flight conditions can be obtained using equation (B7) by letting $\left(\frac{m_1}{m_O}\right)_{opt} = 1$, and $A_1 = A_O$. Thus

$$\frac{(A_O)_{M_2}}{(A_O)_{M_1}} = \frac{\left(M_O \frac{p_O}{H_O}\right)_{M_1} \left[\frac{(W_a)_{cc}}{(W_a)_r} \frac{H_3}{H_O} \frac{a_O}{a_H}\right]_{M_2}}{\left(M_O \frac{p_O}{H_O}\right)_{M_2} \left[\frac{(W_a)_{cc}}{(W_a)_r} \frac{H_3}{H_O} \frac{a_O}{a_H}\right]_{M_1}} \tag{B8}$$

where $\frac{p_O}{H_O} = \frac{1}{H_3/p_O}$ when $\frac{H_3}{H_O} = 1.0$. Quadrant II of figure 2 gives $\frac{H_3}{p_O}$ for M_O.

$$\frac{a_O}{a_H} = \frac{a_O}{a_r}\frac{a_r}{a_H} = \frac{a_O}{a_r}\sqrt{\frac{1.4\ 519}{\gamma\ T_H}} = \frac{a_O}{a_r}\sqrt{\frac{1}{\theta}}$$

where

$\frac{a_O}{a_r}$ is the ratio of speed of sound at static temperature for flight altitude to that at standard sea-level conditions. This ratio can be found from quadrant I of figure 2.

$\sqrt{\frac{1}{\theta}}$ is given in figure 5 for $T_H \equiv T_{O-3}$ which can be found from quadrants I and II of figure 3.

APPENDIX C

RELATION BETWEEN INLET MASS-FLOW RATIO AND THE ENGINE AIR REQUIREMENTS AT A FIXED MACH NUMBER AND ALTITUDE

The mass-flow ratio, by definition, is given by

$$\frac{m_1}{m_o} = \frac{\rho_1 V_1 A_1}{\rho_o V_o A_1}$$

The mass of air flowing through the inlet is then

$$\rho_1 V_1 A_1 = \frac{m_1}{m_o} \rho_o V_o A_1$$

The weights of air flowing through the inlet and the engine are equal and are given by

$$W_a = \frac{m_1}{m_o} g\rho_o V_o A_1 = (W_a)_{cc} \left(\frac{\delta_3}{\sqrt{\theta_3}}\right)_M$$

and

$$\left(\frac{m_1}{m_o}\right)_{opt} g\rho_o V_o A_1 = \left[(W_a)_{cc} \left(\frac{\delta_3}{\sqrt{\theta_3}}\right)_M\right]_{opt}$$

Thus, the ratio of mass flows at different m_1/m_o is given by

$$\frac{\frac{m_1}{m_o}}{\left(\frac{m_1}{m_o}\right)_{opt}} = \frac{\left[(W_a)_{cc} \left(\frac{\delta_3}{\theta_3}\right)_M\right](W_a)_r}{\left[(W_a)_{cc} \left(\frac{\delta_3}{\sqrt{\theta_3}}\right)_M\right]_{opt}(W_a)_r}$$

or

$$\frac{\frac{m_1}{m_o}}{\left(\frac{m}{m_o}\right)_{opt}} = \frac{\left[\frac{(W_a)_{cc}}{(W_a)_r}\left(\frac{\delta_3}{\theta_3}\right)_M\right]}{\left[\frac{(W_a)_{cc}}{(W_a)_r}\left(\frac{\delta_3}{\sqrt{\theta_3}}\right)_M\right]_{opt}} \tag{C1}$$

Solving for $\frac{(W_a)_{cc}}{(W_a)_r}$ and rearranging terms, one obtains

$$\left.\frac{(W_a)_{cc}}{(W_a)_r}\right|_{\frac{m_1}{m_o}} = \left[\frac{(W_a)_{cc}}{(W_a)_r}\right]_{opt} \frac{\frac{m_1}{m_o}}{\left(\frac{m_1}{m_o}\right)_{opt}} \frac{\left[\left(\frac{\delta_3}{\sqrt{\theta_3}}\right)_M\right]_{opt}}{\left(\frac{\delta_3}{\sqrt{\theta_3}}\right)_M} \tag{C2}$$

For a given Mach number and altitude the total temperature T_{o-3} is constant, hence,

$$\left[\left(\sqrt{\theta_3}\right)_M\right]_{opt} = \left(\sqrt{\theta_3}\right)_M$$

By definition,

$$\delta_3 = H_3/p_r$$

or

$$\delta_3 = \frac{H_3}{H_o}\frac{H_o}{p_r} = \frac{H_3}{H_o}\frac{H_o}{p_o}\frac{p_o}{p_r}$$

where $\frac{H_o}{p_r}$ = constant for a given M_o and altitude. Thus,

$$\left.\frac{(W_a)_{cc}}{(W_a)_r}\right|_{\frac{m_1}{m_o}} = \left[\frac{(W_a)_{cc}}{(W_a)_r}\right]_{opt} \frac{\frac{m_1}{m_o}}{\left(\frac{m_1}{m_o}\right)_{opt}} \frac{\left(\frac{H_3}{H_o}\right)_{opt}}{\left(\frac{H_3}{H_o}\right)_{\frac{m_1}{m_o}}} \tag{C3}$$

Using $\left.\frac{(W_a)_{cc}}{(W_a)_r}\right|_{\frac{m_1}{m_o}}$, the compressor corrected speed N_{cc} can be found from the engine characteristics. The actual engine speed N_A can be found, once $\sqrt{\theta_3}$ is known, since $N_A = N_{cc}\sqrt{\theta_3}$.

APPENDIX D

SELECTION OF COMPRESSOR PRESSURE RATIO

The compressor pressure ratio H_4/H_3, as used in this report, has been selected using the maximum obtainable internal thrust coefficient as a criterion. Figures 2, 3, and 4 were used together with the equation for the internal thrust coefficient C_{F_i}' (appendix A, equation (A3)). The calculations were based on the following principles:

1. In order to segregate the effects of variation of engine parameters from the effects of altitude and pressure recovery, the conditions of the isothermal region of the atmosphere were used and isentropic recovery was assumed.

2. The combined efficiency of the compressor and turbine $\eta_c\eta_t$ was assumed to be 100 percent in all computations, and a combustion temperature of 2000° R was used.

3. The internal thrust coefficient C_{F_i}' was determined for assumed Mach numbers of flight. The exhaust nozzle was considered 100 percent efficient.

The results of these calculations are shown in figure 15 for the turbojet without afterburning. The points of maximum internal thrust coefficients have been joined by a curve which indicates the compressor pressure ratios necessary to obtain optimum operation. The data of figure 15 are idealized since the variation in compressor efficiency with temperature was not included ($\eta_c\eta_t$ was assumed to be 100 percent).

The optimum pressure ratios, however, are not affected appreciably by the combined efficiency; the thrust coefficients, on the contrary, are largely dependent on efficiency of every component of an installation. Pressure ratios of the compressor of figure 7 at various Mach numbers also are shown in figure 15. The value of $H_4/H_3 = 6.25$ at $M_0 = 1.4$ was selected as one that would produce approximately maximum internal thrust coefficient without afterburning at Mach numbers less than 2.0. Figure 16, which presents data similar to those of figure 15 but with afterburning, shows that the same engine using afterburning is capable of producing nearly maximum internal thrust coefficients in the range of Mach numbers between 2.0 and 3.0. Again, the amount of afterburning was controlled so that total fuel consumption would be equal to that of the ram jet of figure 6.

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TABLE I.- SAMPLE CALCULATIONS FOR A TURBOJET ENGINE WITH AND WITHOUT AFTERBURNING

Item	Source	Units	Turbojet without afterburning	Turbojet with afterburning
V_o	Fig. 2, quadrant I	$\frac{ft}{sec.}$	1940	1940
H_3/H_o	Induction-system characteristics	None	0.80[a]	0.80[a]
H_3/p_o	Fig. 2, quadrant II	None	6.25	6.25
T_{o-3}	Fig. 3, quadrants I and II	^oR	708	708
$\sqrt{\frac{1}{\theta_3}}$	Fig. 5, for T_3	None	0.858	0.858
N_A	Actual engine speed	rpm	12,500[a]	12,500[a]
N_{cc}	$N_A\sqrt{\frac{1}{\theta_3}}$	rpm	10,720[a]	10,720[a]
η_c	Compressor characteristics for N_{cc}, fig. 7	None	0.900[a]	0.900[a]
H_4/H_3	Compressor characteristics for M_{cc}, fig. 7	None	4.56[a]	4.56[a]
T_4	Fig. 3 for H_4/H_3 and T_{o-3}	^oR	1060	1060
T_5	Engine characteristics for N_{cc}, fig. 7	^oR	2000[a]	2000[a]
$\sqrt{\frac{1}{\theta_5}}$	Fig. 5 for T_5	None	0.522	0.522
N_{ct}	$N_A\sqrt{\frac{1}{\theta_5}}$	rpm	6530	6530
η_t	Turbine characteristics for N_{ct}, fig.7	None	0.900[a]	0.900[a]
H_5/H_6	Fig. 4 for T_3, $\eta_c\eta_t$, and T_5	None	2.50	2.50
T_6	Fig. 4 for $\frac{H_5}{H_6}$ and T_5	^oR	1584	1584
$\left(\frac{W_a}{W_f}\right)_5$	Fig. 3 for T_4 and T_5	$\frac{lb\ air}{lb\ fuel}$	77.06	77.06
$\left(\frac{W_a}{W_f}\right)$ total at sta. 8	Assumed equal to that of ram jet operating at 3000° R	$\frac{lb\ air}{lb\ fuel}$	- - -	29.6[a]
$\left(\frac{W_a}{W_f}\right)$ added at sta. 8	Equation (A8) of appendix A	$\frac{lb\ air}{lb\ fuel}$	- - -	49.29
T_9	Fig. 3, quadrant IV, for T_6 and $\left(\frac{W_a}{W_f}\right)$ added at sta. 8	^oR	1584	2926
H_9/p_o	$\left(\frac{H_3}{p_o}\right)\left(\frac{H_4}{H_3}\right)\left(\frac{H_5}{H_4}\right)\left(\frac{H_6}{H_5}\right)\left(\frac{H_7}{H_6}\right)\left(\frac{H_8}{H_7}\right)\left(\frac{H_9}{H_8}\right)$	None	11.30	11.30
V_{11}	Fig. 2, quadrants III and IV, and $\frac{H_9}{p_o}$ and T_9	$\frac{ft}{sec}$	3118	4297
C_{F_i}'	$2\left[\left(1+\frac{W_f}{W_a}\right)\frac{V_{11}}{V_o}-1\right]$	None	1.256	2.520

[a]Values for assumed condition of operation.

NACA

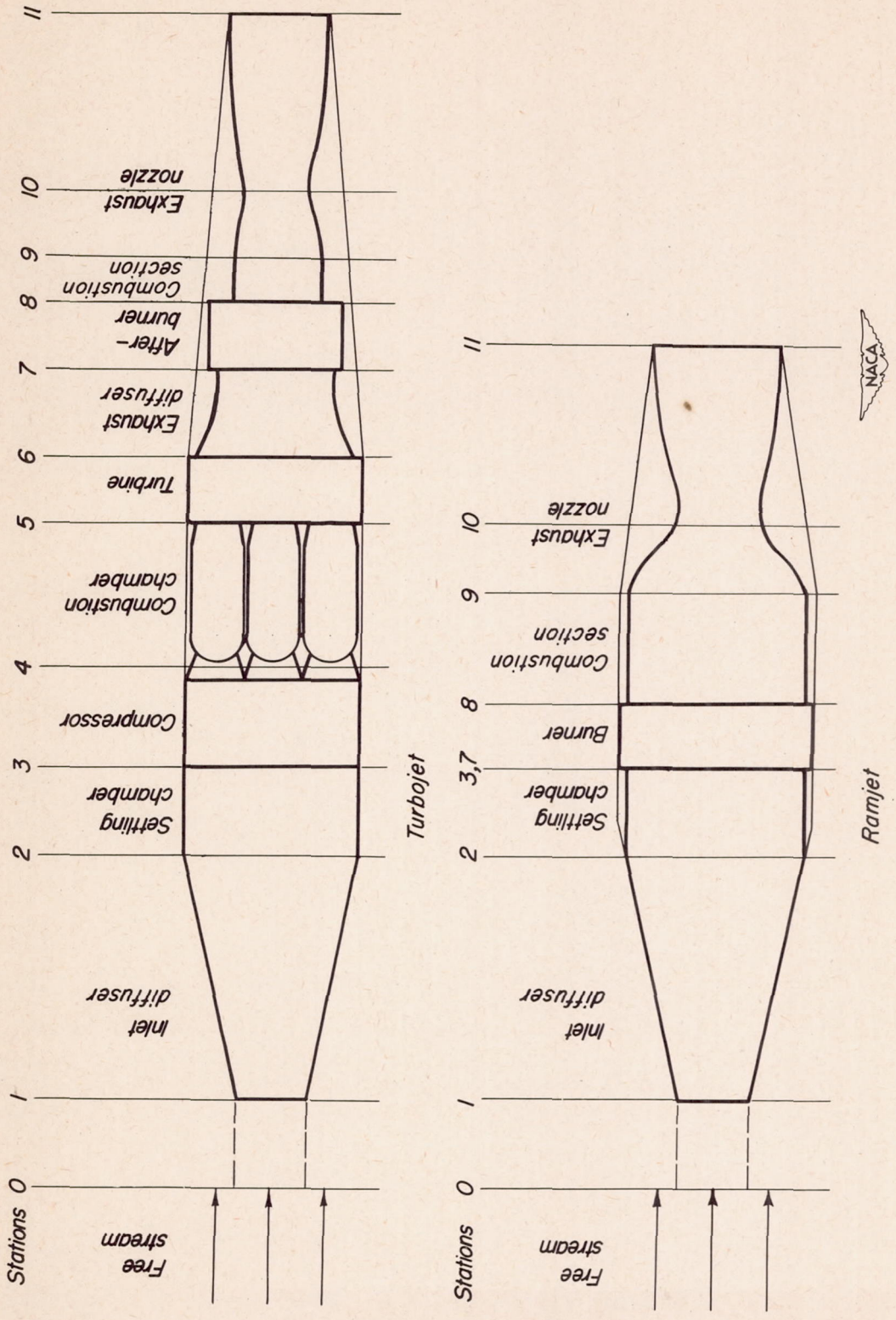


Figure 1.- Schematic representation of propulsive systems.

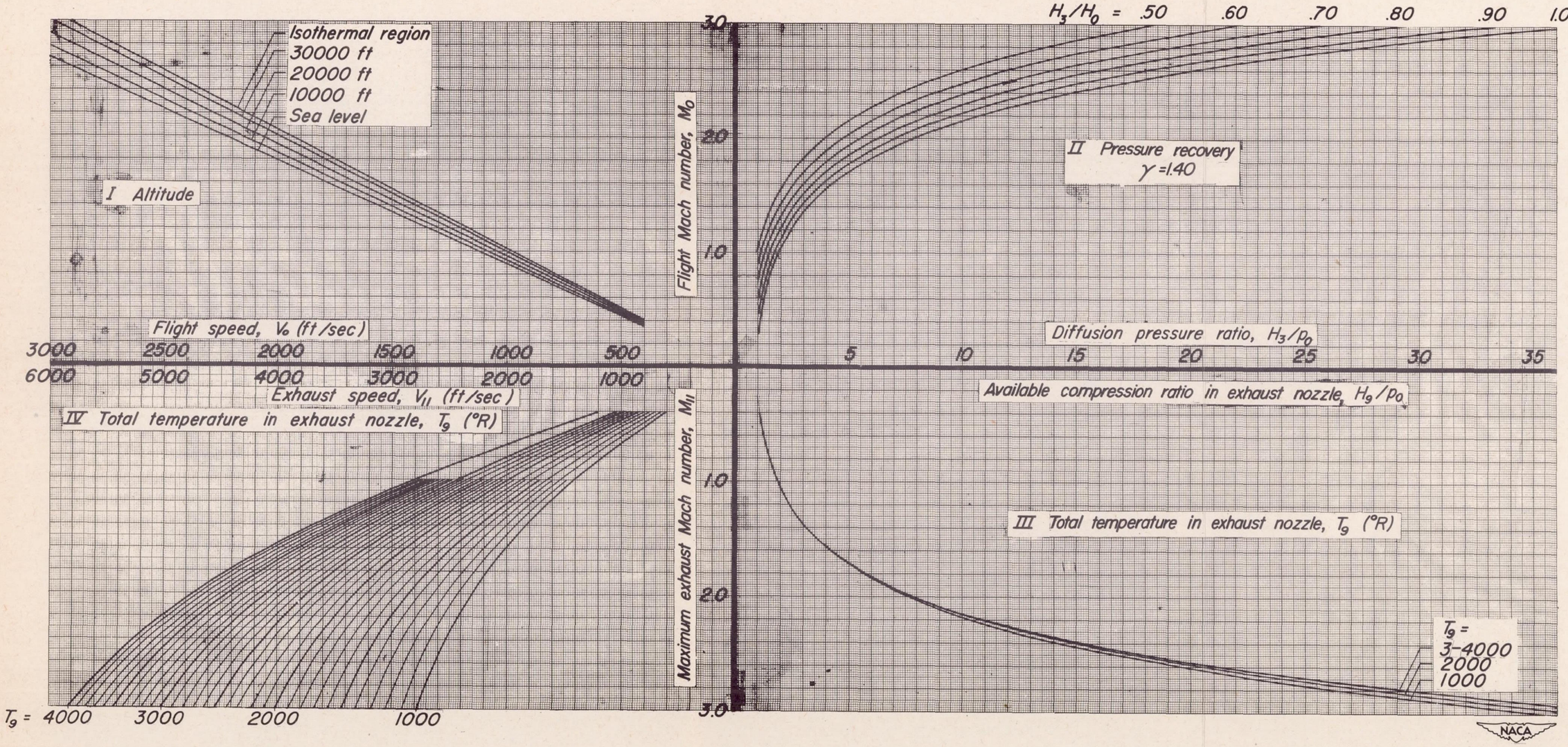


Figure 2.- Velocity graph.

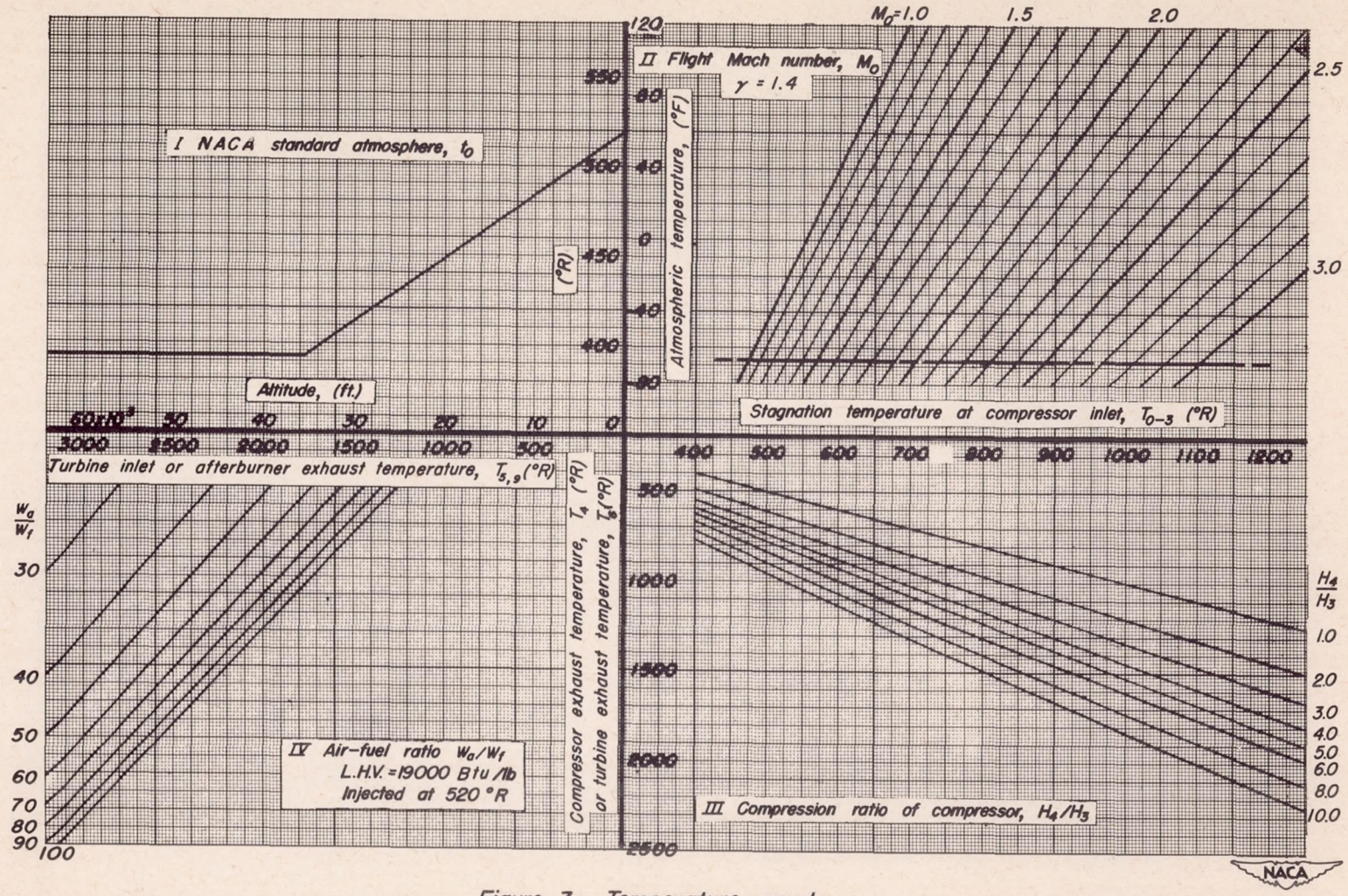


Figure 3. – Temperature graph.

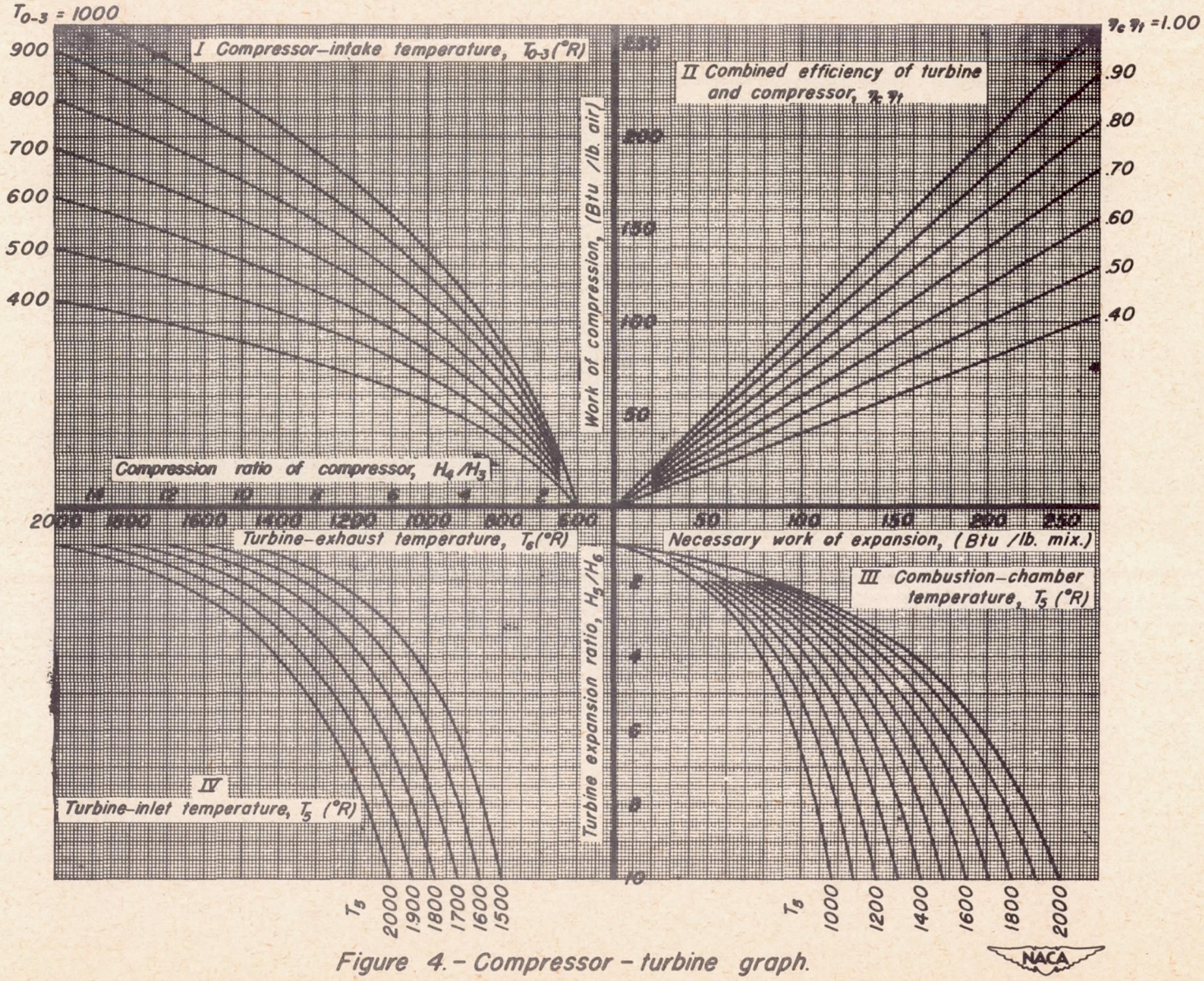


Figure 4. – Compressor – turbine graph.

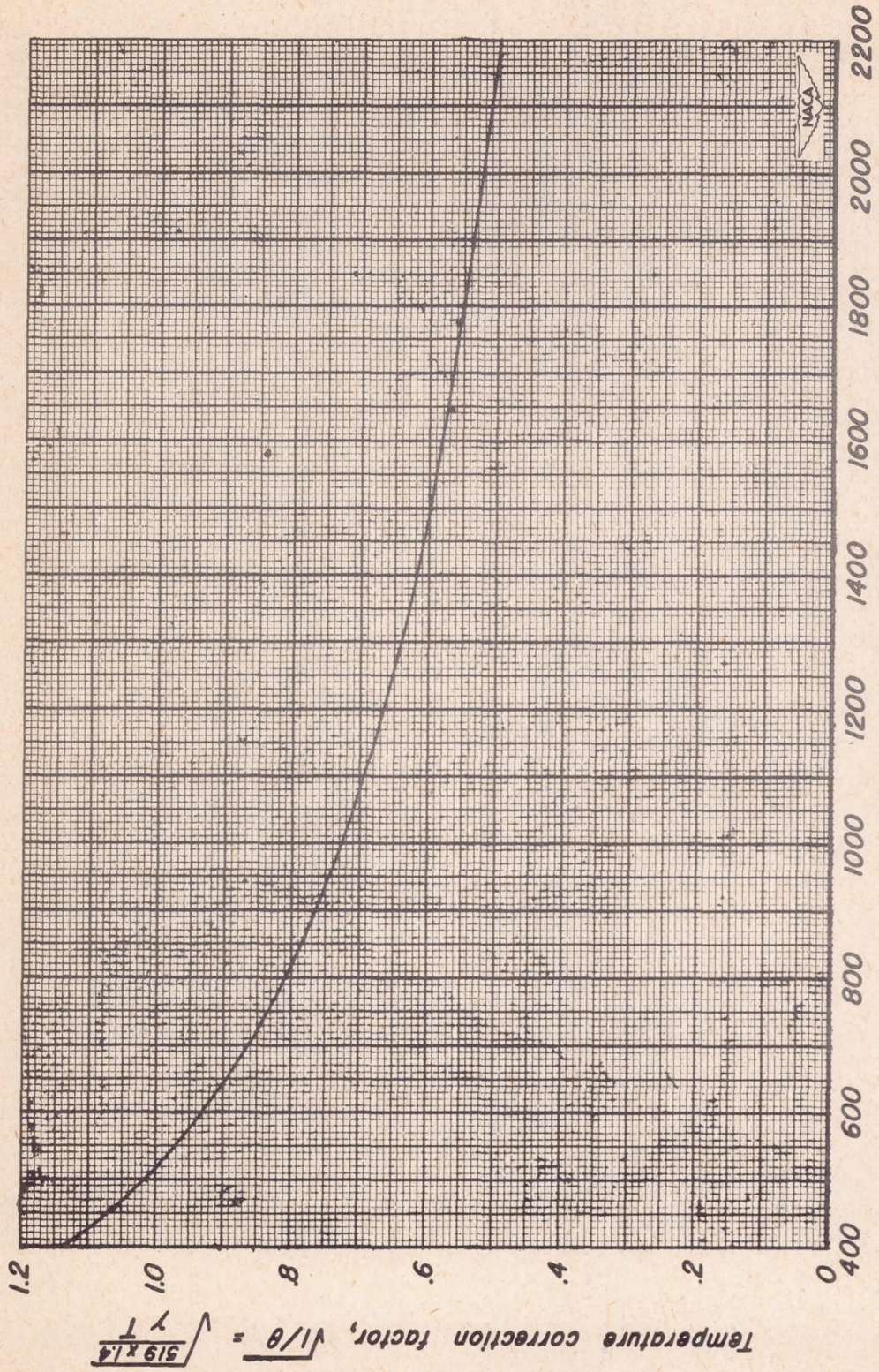


Figure 5.- Temperature correction graph.

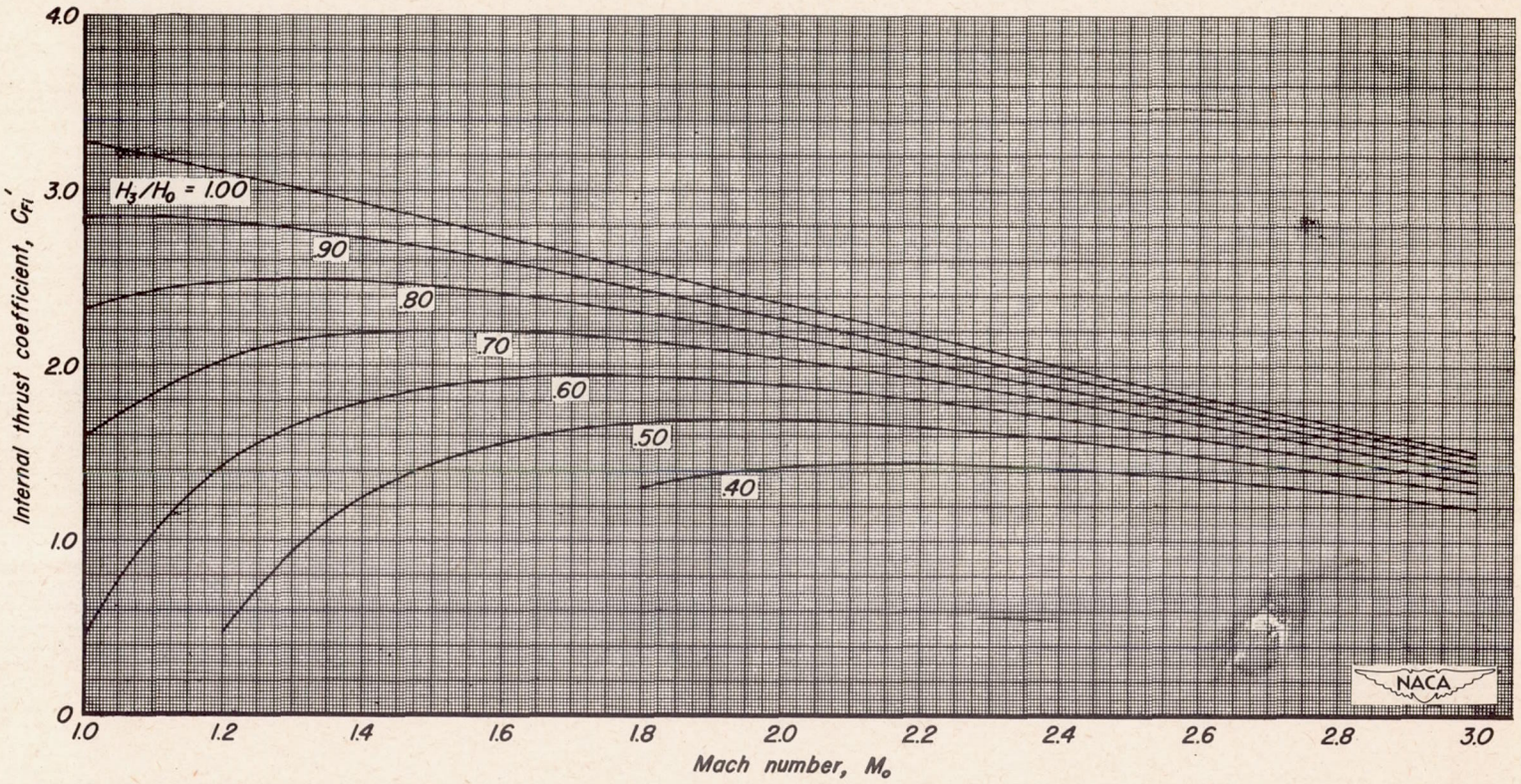


Figure 6.- Effects of total-pressure recovery on internal thrust coefficient of a ram jet operating at 3000° Rankine in the isothermal region of atmosphere.

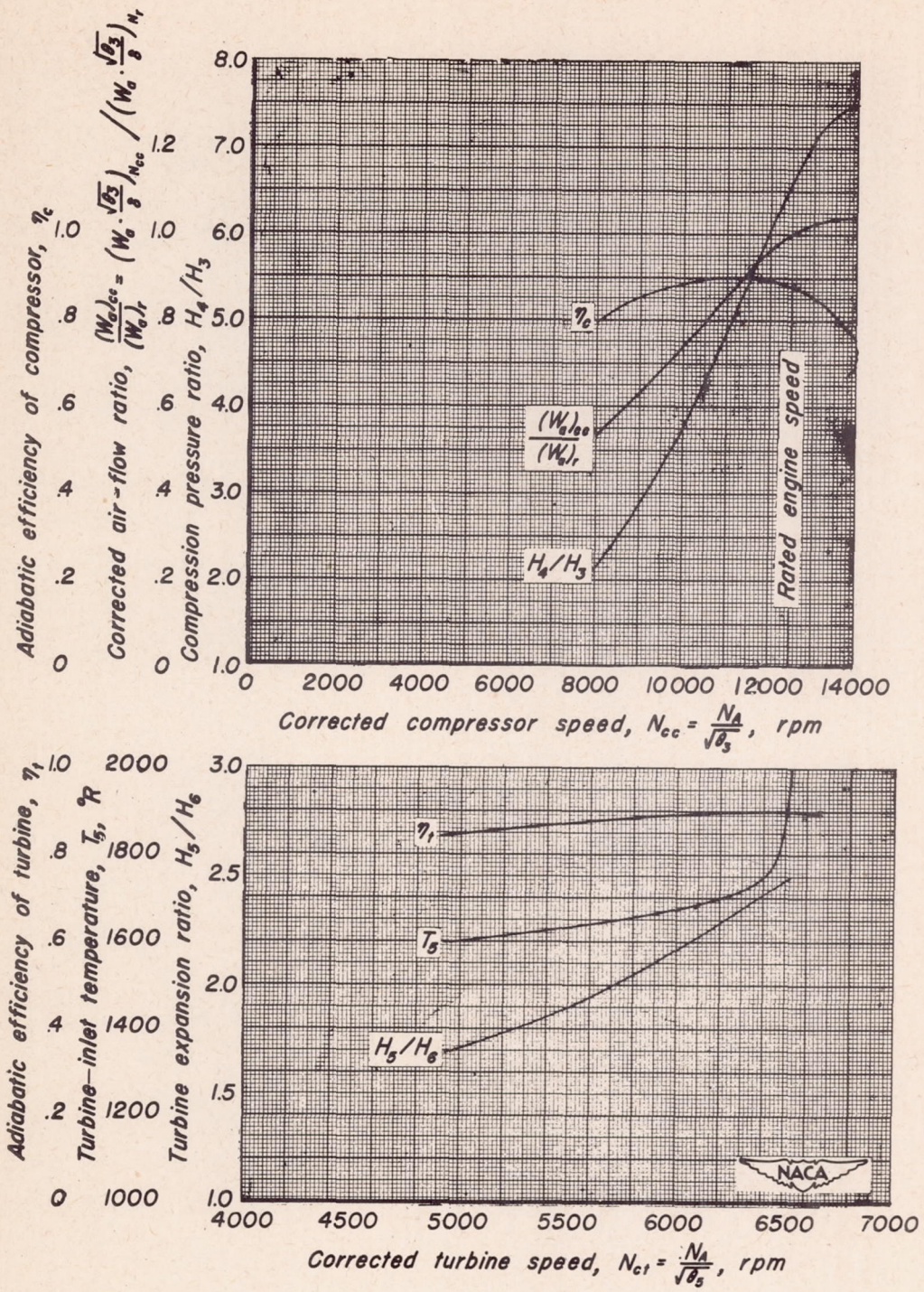


Figure 7. – Turbojet–engine characreristics corrected to standard sea-level static atmospheric conditions.

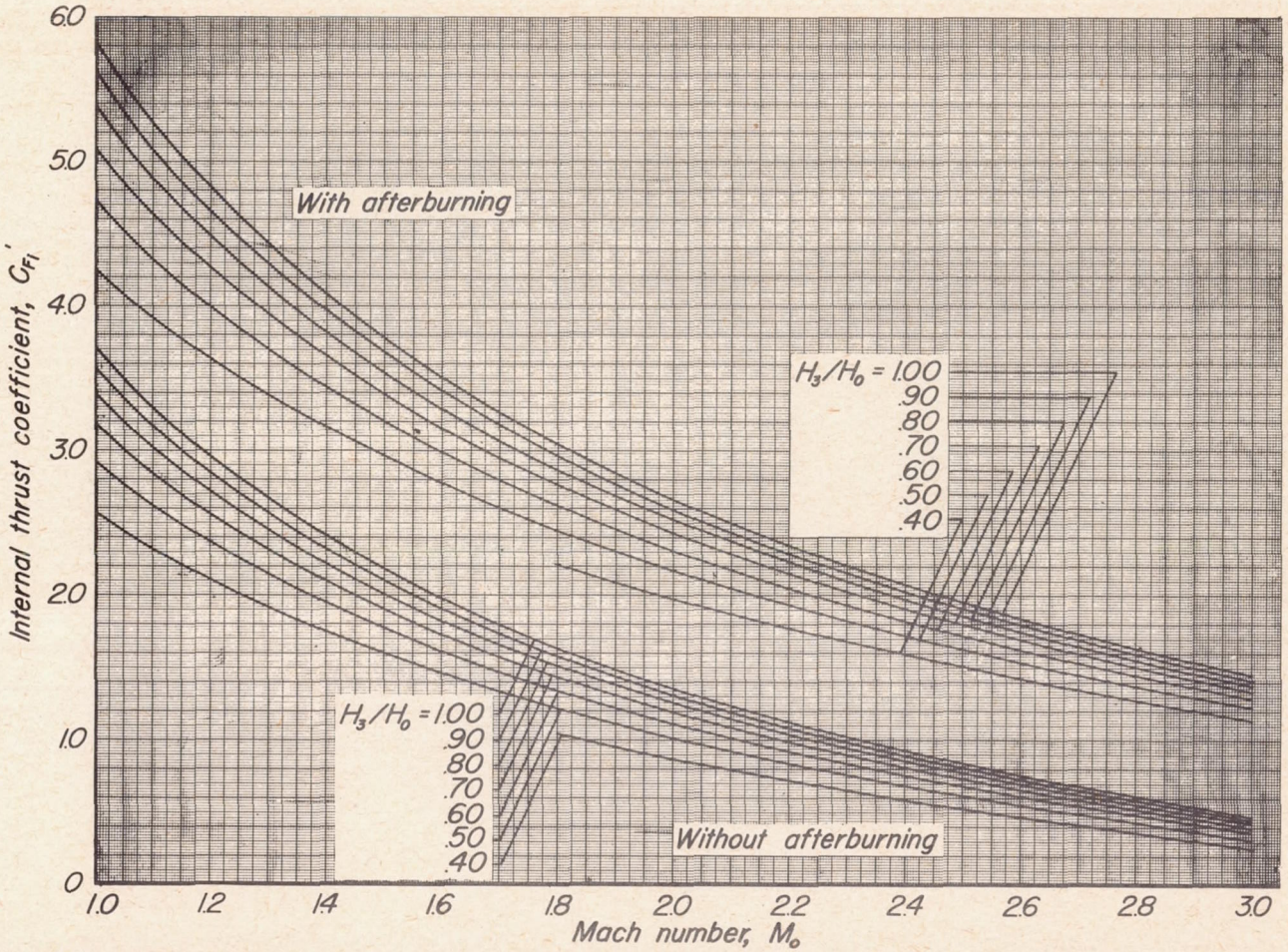


Figure 8.- Effects of total-pressure recovery on internal thrust coefficient of turbojet of figure 7 operating at 2000° Rankine without afterburning and the same turbojet with afterburning having a total air-to-fuel ratio equal to that of the ram jet of figure 6.

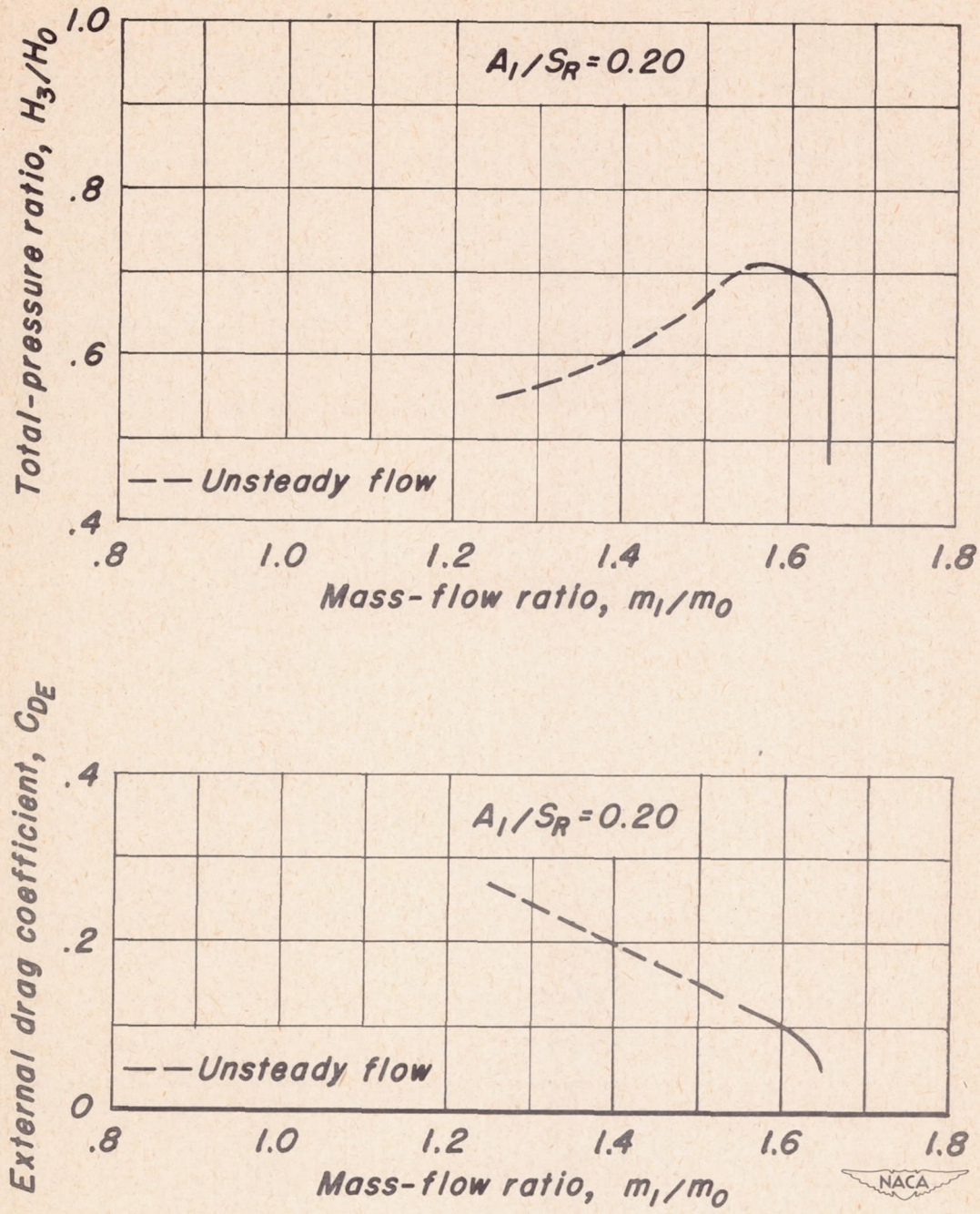


Figure 9.- Induction-system characteristics at Mach number 2.5.

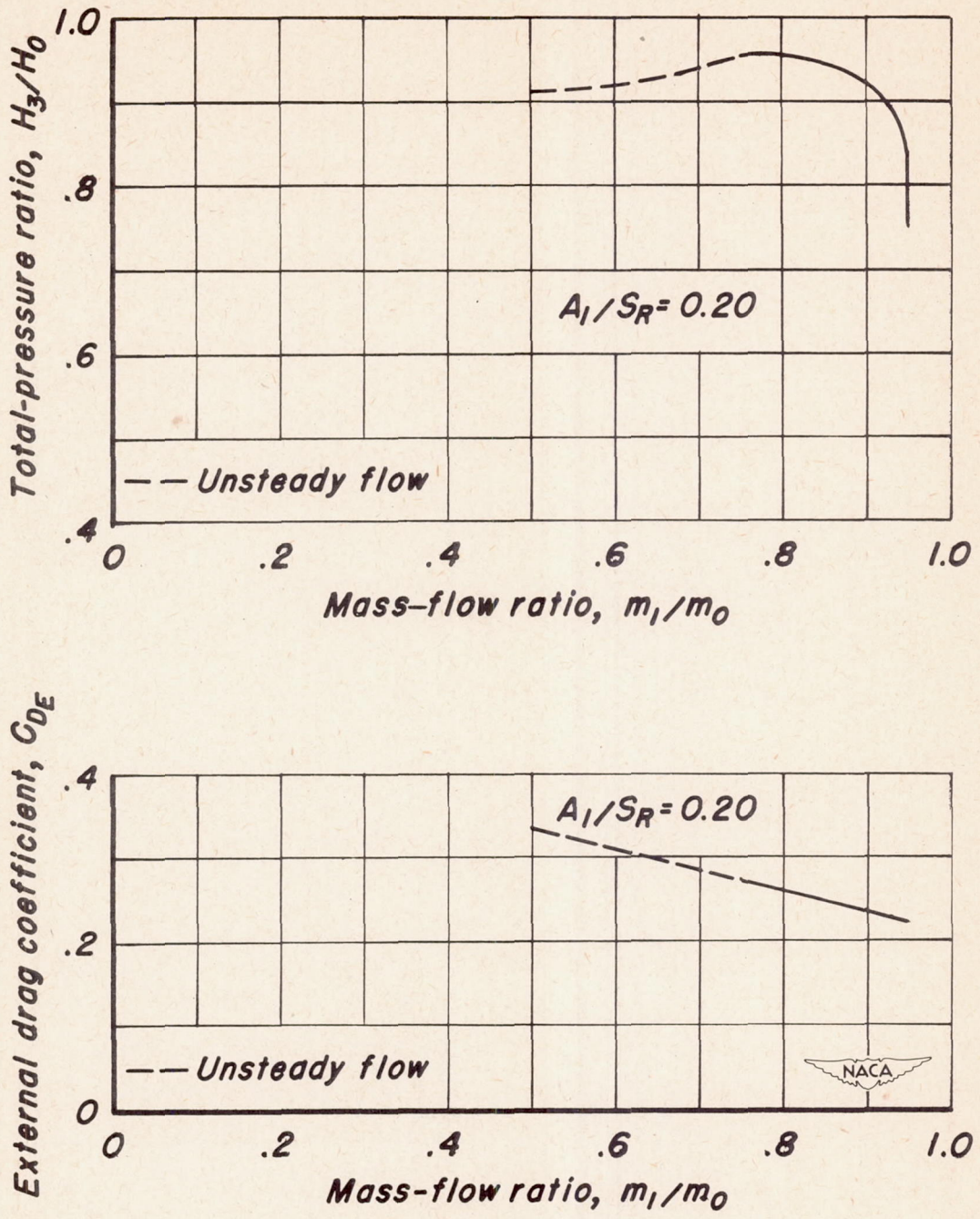


Figure 10.- Induction-system characteristics at Mach number 1.2.

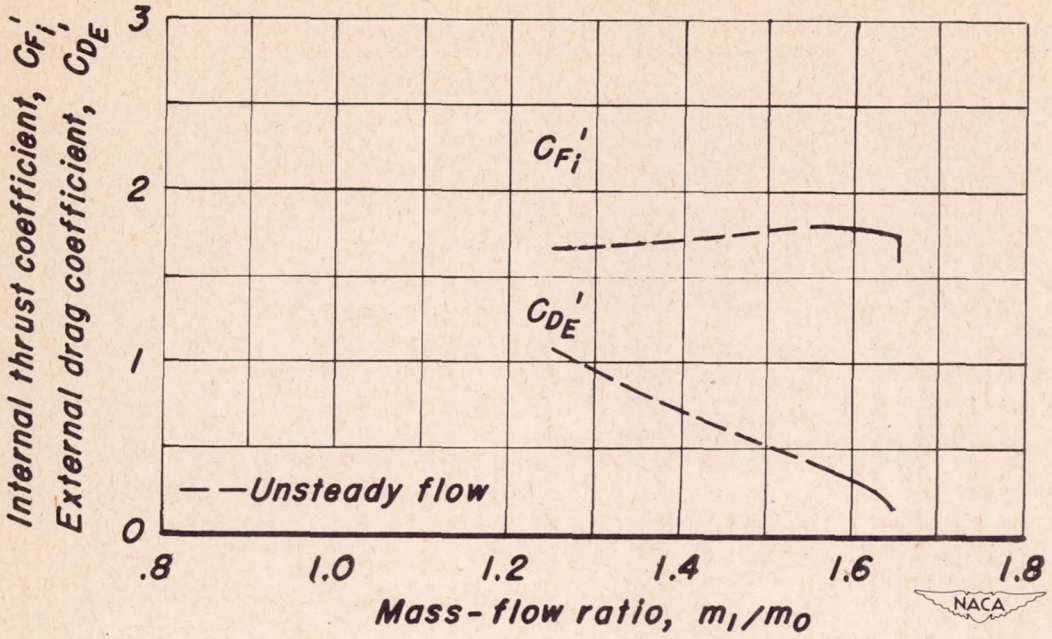


Figure 11.- Effects of mass-flow ratio on thrust and drag coefficients.

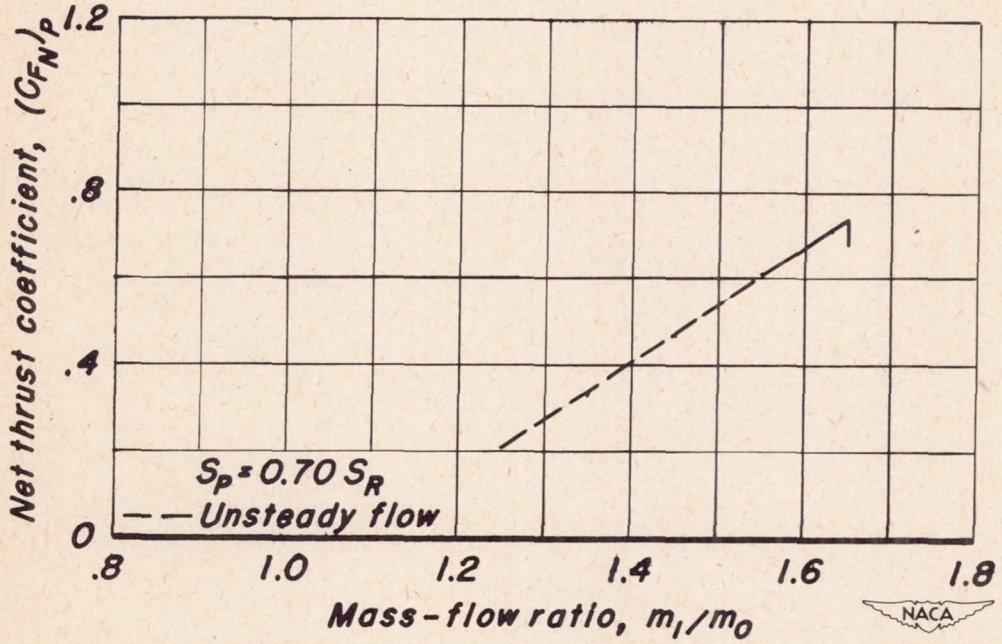


Figure 12.- Effects of mass-flow ratio on net thrust coefficient at Mach number 2.5 in the isothermal region of the atmosphere.

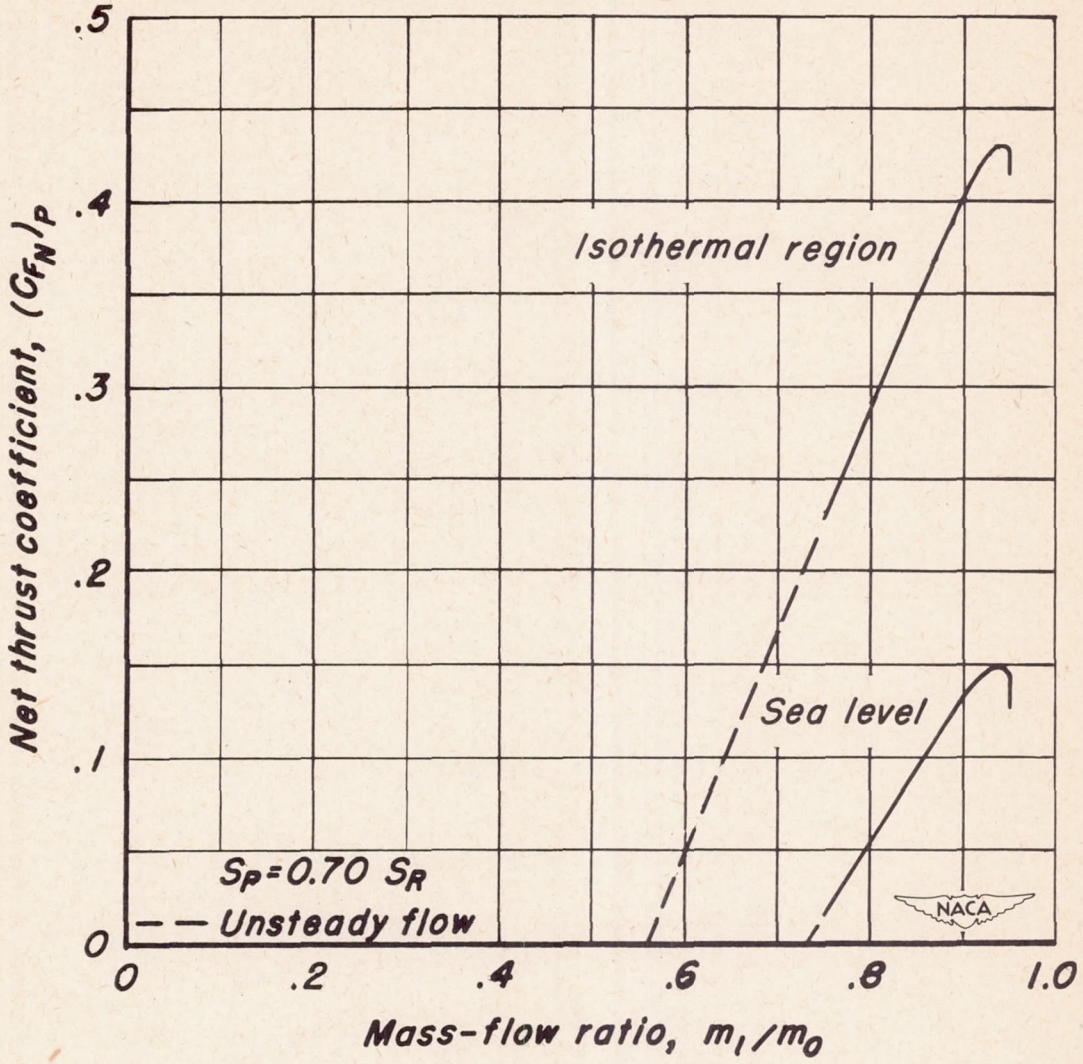
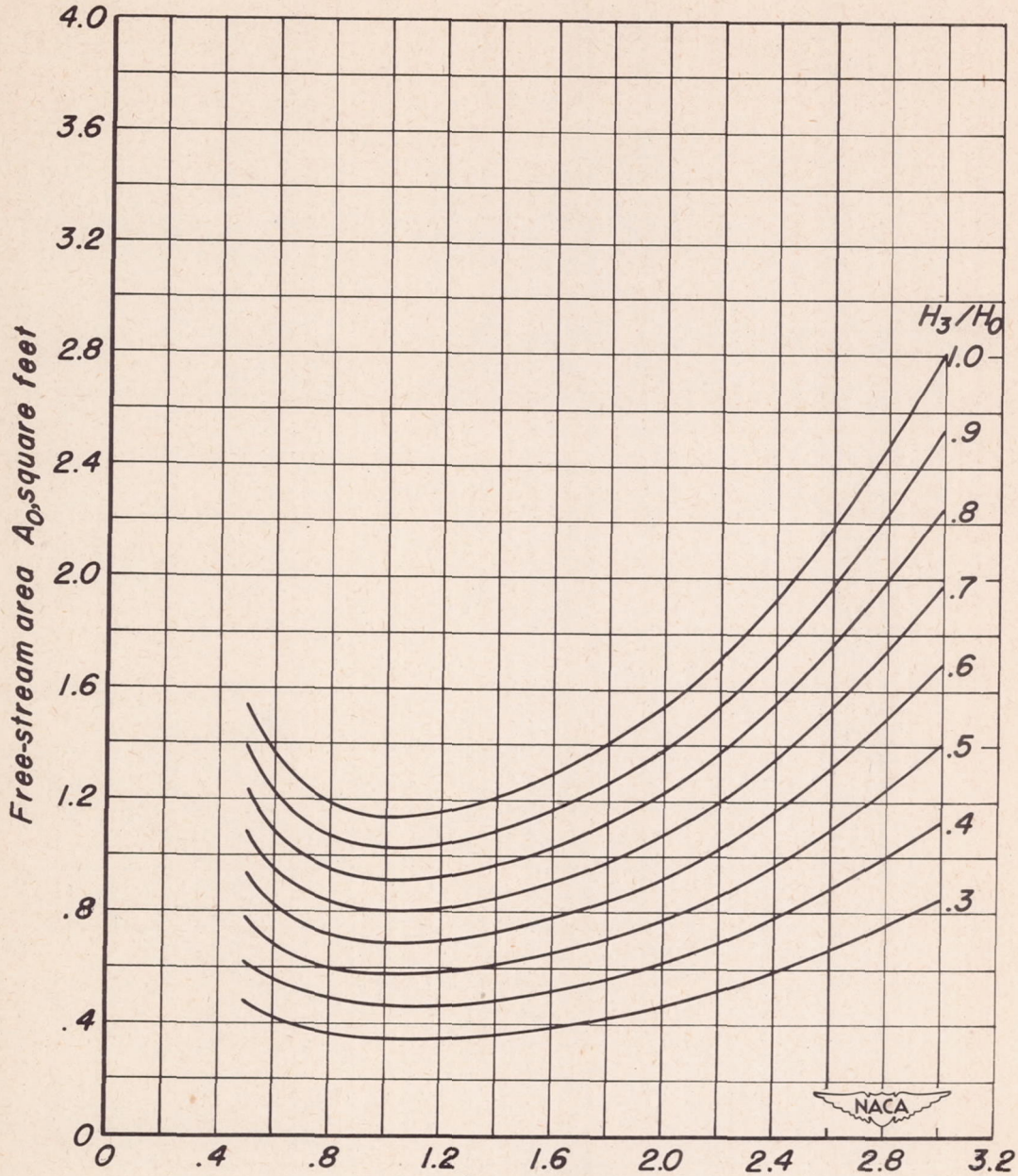
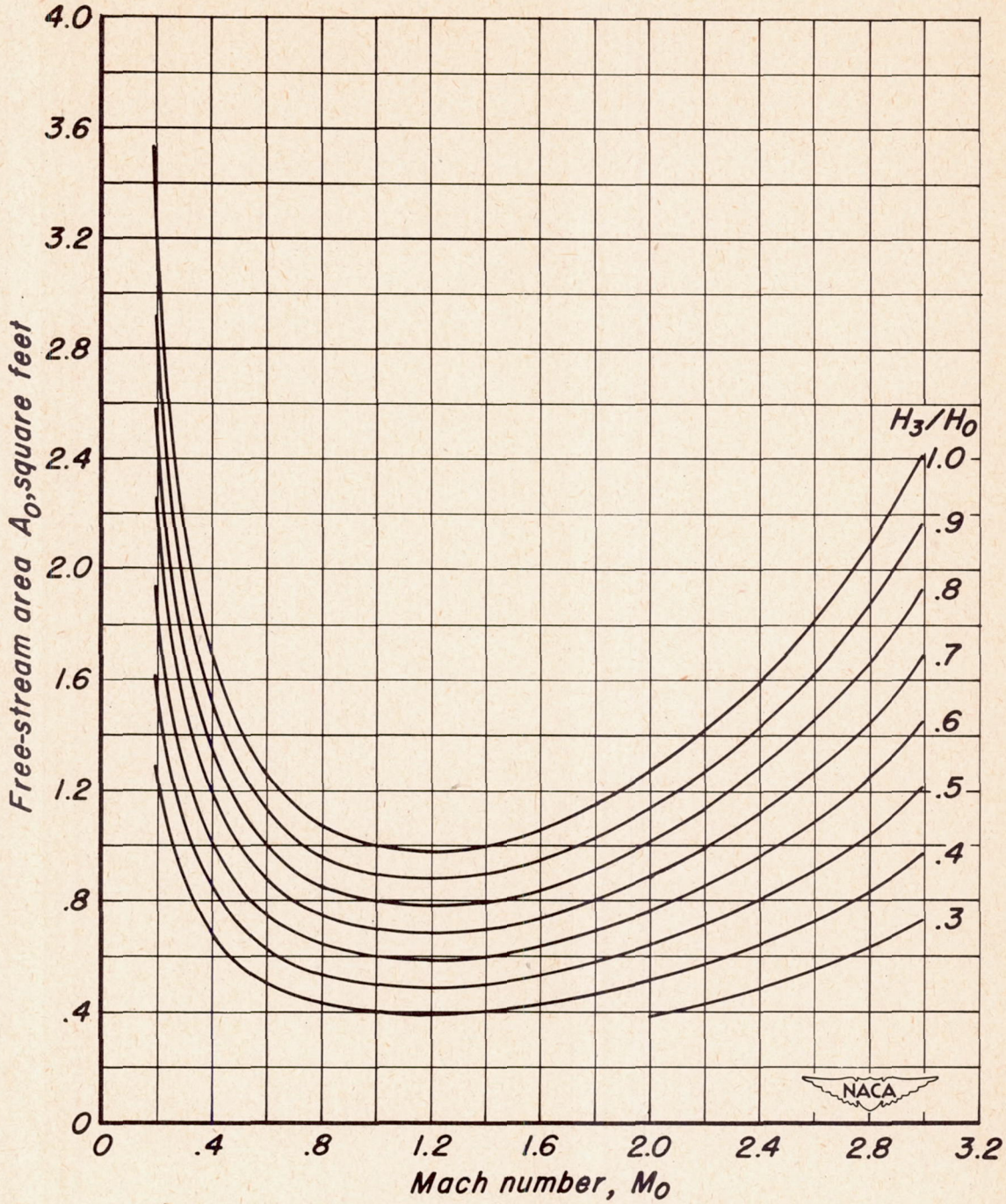


Figure 13.- Effects of mass-flow ratio on net thrust coefficient at Mach number 1.2.



(a) Isothermal region.

Figure 14.- Free-stream area required by the turbojet engine of the illustrative examples.



(b) Sea level, standard atmosphere.

Figure 14.- Concluded.

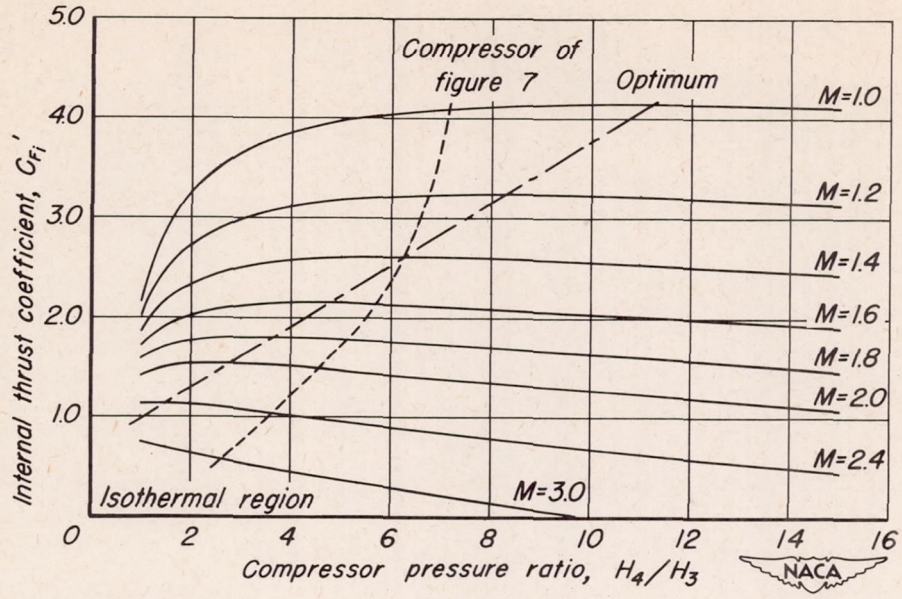


Figure 15.- Effects of compressor pressure ratio on internal thrust coefficient of turbojet without afterburning.

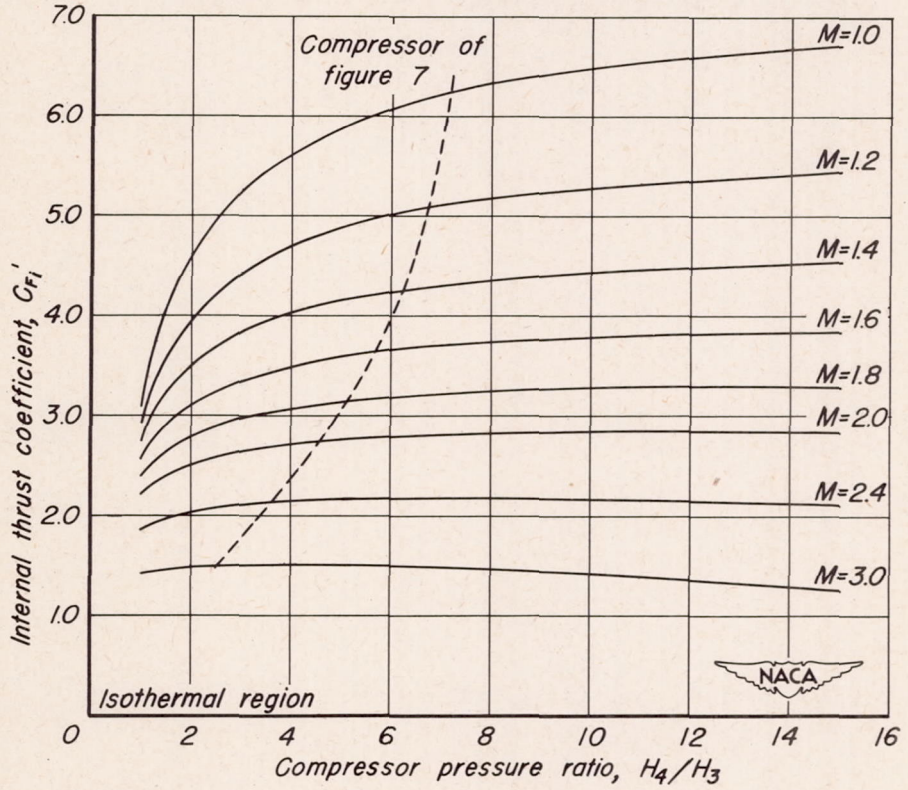


Figure 16.- Effects of compressor pressure ratio on internal thrust coefficient of turbojet with afterburning.