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RESEARCH MEMORANDUM

ANALYTICAL COMPARISON OF A STANDARD TURBOJET ENGINE, A TURBOJET ENGINE WITH A TAIL-PIPE BURNER, AND A RAM-JET ENGINE

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY

The calculated performance of a standard turbojet engine, a turbojet engine whose thrust is augmented by tail-pipe burning, and a ram-jet engine are compared. Computations for the performance of the turbojet engines are based on an analytical extension of experimental data obtained from an investigation of a turbojet engine incorporating an 11-stage axial-flow compressor and a single-stage impulse turbine in the altitude wind tunnel. The three types of engine are considered to be operating at maximum output for any given set of flight conditions.

The three engines are compared on the basis of net thrust per unit frontal area and specific fuel consumption at an altitude of 30,000 feet for flight Mach numbers up to 2.0. The effect of changes in altitude from sea level to 40,000 feet were calculated at Mach numbers of 1.2 and 1.6.

At static conditions, tail-pipe burning increased the net thrust of the standard turbojet engine about 60 percent. At flight speeds greater than the speed of sound, the net thrust was more than doubled. Greatest percentage increases in net thrust with tail-pipe burning could be obtained at low altitudes and high flight Mach numbers. The net thrust per unit frontal area of the augmented turbojet engine and the ram-jet engine were equal at a flight Mach number of 1.2. At this flight speed the ram jet had a specific fuel consumption based on net thrust approximately 45 percent greater than the augmented turbojet.

INTRODUCTION

Because of temperature limitations of the turbine, the fuel burned in the combustion chamber of a turbojet engine is limited to an amount corresponding to a mixture less than one-third stoichiometric. When additional fuel is burned in the tail pipe, the over-all fuel-air ratio of the engine may be raised to stoichiometric and the thrust developed by the engine greatly augmented. Experimental investigations (reference 1) have shown that in some cases the thrust can be more than doubled by means of tail-pipe burning. The effectiveness of tail-pipe burning is particularly good at high flight speeds.

Operation with the tail-pipe burner so increases the thrust of the turbojet engine that it is comparable with that of a ram-jet engine at supersonic flight speeds. An evaluation of the thrust per unit frontal area and specific fuel consumption of these two propulsion systems has therefore been made.

A comparison is made of a standard turbojet engine, a turbojet engine whose thrust is augmented by tail-pipe burning, and a ram-jet engine. Engine performance was calculated at an altitude of 30,000 feet for flight Mach numbers from 0 to 2.0. The effect of changes in altitude from sea level to 40,000 feet on net thrust per unit frontal area and specific fuel consumption were calculated for Mach numbers of 1.2 and 1.6.

The performance characteristics for the standard turbojet engine and its components were obtained from an investigation of a standard turbojet engine incorporating an 11-stage axial-flow compressor and a single-stage impulse turbine in the Cleveland altitude wind tunnel. The results of this investigation were extended from actual operating conditions to flight conditions of higher Mach numbers and lower altitude. The performance characteristics for the ram-jet engine were computed entirely from theoretical considerations and on the assumption that the burner-inlet velocity was constant.

BASES OF ANALYSIS

The stations used in the performance analysis of the three engines are indicated in figure 1. Corresponding stations in the standard turbojet and the augmented turbojet engines have the same numbers. Only four stations are required for the analysis of the ram jet.

Standard turbojet engine. - The analysis of the standard turbojet engine incorporating an 11-stage axial-flow compressor and a single-stage impulse turbine (TG-180) is based on experimental data taken at simulated altitudes from 5000 to 40,000 feet and flight Mach numbers from 0 to 0.88. These data have been extended to simulated sea-level conditions and flight Mach numbers up to 2.0 by means of the following assumptions:

1. The engine is operated at limiting conditions of 7600 rpm and a turbine-inlet temperature of 2000° R, which approximate the physical limitations of the turbine set by the manufacturer. The tail-pipe nozzle outlet area is adjusted to maintain limiting conditions of the engine.

2. Corrected air flow $W\sqrt{\theta_2}/\delta_2$ is a function of corrected compressor speed $N/\sqrt{\theta_2}$ and is independent of altitude and ram-pressure ratio P_2/p_0 (fig. 2). (All symbols are defined in the appendix.)

3. Compressor efficiency η_c is a function of only corrected compressor speed (fig. 3).

4. The compressor pressure coefficient ψ is a function of only corrected compressor speed (fig. 3).

5. The combustion chamber has a combustion efficiency of 100 percent. The total-pressure ratio across the combustion chamber P_4/P_3 is a function of only corrected compressor speed (fig. 4).

6. Turbine efficiency η_t based on total outlet pressure is a function of corrected turbine speed $N/\sqrt{\theta_4}$ and is independent of altitude, ram-pressure ratio, and tail-pipe nozzle outlet area. Because the engine speed and turbine-inlet temperature are fixed, the turbine efficiency has a single value of 0.803.

7. The diffuser and the tail-pipe nozzle are 100-percent efficient.

8. The enthalpy rise through the compressor is equal to the enthalpy drop through the turbine. Accessory power, bearing friction, and thermal radiation are neglected.

9. The mass flow of gas in the tail pipe is taken equal to the mass flow of air at the compressor inlet because the mass of air used for cooling and discharged from the engine was approximately equal to the mass of fuel burned.

10. A convergent nozzle is used at the end of the tail pipe.

11. The engine completely assembled with nacelle has an outside diameter of 40 inches.

Turbojet engine with tail-pipe burner. - The greatest gain in thrust with tail-pipe burning is obtained when the engine is operated at limiting conditions (engine speed, 7600 rpm; turbine-inlet temperature, 2000° R). The tail-pipe nozzle outlet area is so adjusted that engine limiting conditions are maintained as the flight Mach number and fuel flow are varied. In addition to the assumptions listed for the standard turbojet engine, the following assumptions are necessary for the analysis of the turbojet engine with a tail-pipe burner:

1. The over-all fuel-air ratio for the engine and the tail-pipe burner is stoichiometric.

2. Combustion in the tail-pipe burner is 100 percent efficient. The inlet velocity and the temperature-rise ratio across the tail-pipe burner produce negligible momentum-pressure losses.

Ram-jet engine. - Performance analysis of the ram-jet engine is based on theoretical considerations. The following specific assumptions have been made to facilitate a comparison of ram-jet and turbojet engines:

1. The ram-jet engine operates with no pressure losses other than the momentum-pressure loss associated with combustion.

2. The burner-inlet diameter is 36 inches and the outside diameter complete with nacelle is 38 inches.

3. The ram jet is operated with stoichiometric fuel-air ratio.

4. The burner-inlet velocity is held constant at 175 feet per second by varying the tail-pipe nozzle outlet area. This value for burner-inlet velocity was chosen as a maximum value for efficient combustion from the results of the investigation discussed in reference 2.

5. A convergent nozzle is used to discharge the burned gases from the ram-jet engine.

Turbojet and ram-jet engines. - The fuel used in all three engines is assumed to have a lower heating value of 18,600 Btu per pound, a hydrogen-carbon ratio of 0.170, and a stoichiometric fuel-air ratio of 0.068. For all flight Mach numbers for which the total-pressure ratio across the turbojet engine is greater than unity, the 100 percent efficient pressure recovery assumed for the inlet diffuser of all three engines provides a greater advantage for the ram-jet engine than for the turbojet engines.

RESULTS AND DISCUSSION

Effect of Flight Mach Number on Engine Performance

The performance of a standard turbojet engine, a turbojet engine with a tail-pipe burner, and a ram-jet engine was calculated over a range of flight Mach numbers from 0 to 2.0 at an altitude of 30,000 feet for operation at limiting conditions.

Standard turbojet engine. - The performance characteristics of a standard turbojet engine were determined on the basis of the preceding assumptions and the derivations in the appendix. Jet and net thrust per unit frontal area, total-pressure and total-temperature ratios across the engine, and fuel-air ratio are shown in figure 5.

The jet thrust per unit frontal area F_j/A_f at an altitude of 30,000 feet increases from 200 pounds per square foot at a flight Mach number M_0 of 0 to 1205 pounds per square foot at a flight Mach number of 2.0. A minimum net thrust per unit frontal area F_n/A_f of 183 pounds per square foot occurs at a flight Mach number of about 0.3 and reaches a maximum of approximately 413 pounds per square foot at a flight Mach number above 2.0. Increased flight speed results in a decrease in fuel-air ratio f from 0.0186 at a flight Mach number of 0 to 0.0143 at a flight Mach number of 2.0.

As the flight Mach number increases, the total temperature at the compressor inlet T_2 increases and the compressor Mach number decreases for a fixed engine speed. As a result the total-pressure ratio across the engine P_5/P_2 decreases as the flight Mach number increases. Because the compressor-inlet temperature rises with increasing flight Mach number, the total-temperature ratio across the engine T_5/T_2 falls, as shown in figure 5.

For the standard turbojet engine, the compressor characteristics are such that the ratio between compressor pressure coefficient ψ and compressor efficiency η_c is constant over a wide range of corrected compressor speeds (fig. 3). In the appendix it is proved that, if the ratio ψ/η_c is constant, the total-temperature drop across the turbine ΔT_t, and therefore the total-temperature rise across the compressor ΔT_c, and the work put into the compressor by the turbine per pound of air are all constant. With a fixed turbine-inlet temperature and turbine temperature drop, the turbine-outlet temperature is constant. Another effect of the constancy of ψ/η_c is that the tail-pipe nozzle outlet area must be constant to maintain limiting conditions in the engine.

Turbojet engine with tail-pipe burner. - Limiting conditions for the turbojet engine with a tail-pipe burner are the same as for the standard turbojet engine with the added condition that the over-all fuel-air ratio is stoichiometric. The jet thrust F_j for the turbojet engine with a tail-pipe burner was computed from equation (19) in the appendix. The values of tail-pipe nozzle outlet velocity V_{10} and tail-pipe nozzle outlet area A_{10} change appreciably from the values in equation (11) because of tail-pipe burning. Equations (10) and (15) show that, aside from changes in the ratio of specific heats at the tail-pipe nozzle outlet γ_{10}, both A_{10} and V_{10} vary with the square root of the total temperature at station 10 $\sqrt{T_{10}}$. Accordingly, tail-pipe burning increases the jet thrust developed by the standard turbojet engine by a factor approximately equal to the square root of the temperature ratio across the tail-pipe burner $\sqrt{\tau}$.

Considerable increase in net thrust per unit frontal area available from tail-pipe burning, especially at high Mach numbers, is shown in figure 6 and the following table:

Mach number	Net thrust per unit frontal area (lb/sq ft)		Increase in net thrust (percent)
	Standard turbojet	Turbojet with tail-pipe burner	
0	200	316	58
1.2	263	560	113
2.0	400	1180	195

The percentage increase in net thrust presented in this table is not comparable with that given in reference 1. The engine investigated in reference 1 and the engine used as the basis of this analysis differed considerably in performance. The net thrust developed by the standard engine of this analysis was about 26 percent higher than the net thrust of the engine of reference 1 for a range of flight Mach numbers from 0.2 to 0.6. This difference in net thrust is attributed to the pressure drop in the tail-pipe burner (reference 1) and to small differences in the performance characteristics of the compressor and the turbine and in the area of the tail-pipe nozzle.

The specific fuel consumption based on net thrust for the three engines operating under the same conditions as for figure 6 is shown in figure 7. The specific fuel consumption for the standard turbojet engine increases from about 0.96 pound of fuel per hour per pound of net thrust at a flight Mach number of 0 to 1.68 pounds per hour per pound of net thrust at a flight Mach number of 2.0. The specific fuel consumption for the engine with tail-pipe burning averages about twice that of the standard engine, varying from 2.28 to 2.68 over the same range of flight Mach numbers.

A comparison of figures 6 and 7 gives the following information: At a flight Mach number of 0, tail-pipe burning increases the net thrust 58 percent or from 200 to 316 pounds per unit frontal area. Under the same conditions the specific fuel consumption increases from 0.96 to 2.28 or about 138 percent. At a flight Mach number close to 1.0, tail-pipe burning increases both the thrust and specific fuel consumption about 100 percent.

Comparison of ram-jet and turbojet engines. - For a comparison of the ram-jet engine with the standard turbojet and the augmented turbojet engines, the assumption was made that all three engines were running at limiting conditions. The limiting conditions for the ram jet consist of a constant burner-inlet velocity of 175 feet per second and provision of sufficient fuel to give a stoichiometric fuel-air ratio.

Figure 6 shows that both the ram-jet and the standard turbojet engines produce a net thrust per unit frontal area of 210 pounds per square foot at a flight Mach number of 0.79 and that the ram jet and the augmented turbojet both produce a net thrust of 570 pounds per unit frontal area at a Mach number of 1.22.

At a flight Mach number of 2.0, the ram jet develops a net thrust per unit frontal area of 1670 pounds per square foot, which is 41 percent greater than that produced by the augmented turbojet engine and the specific fuel consumption of the ram jet exceeds that of the augmented turbojet by 7 percent. Although the net thrusts of these two engines are equal at a Mach number of 1.22, the specific fuel consumption is about 45 percent higher for the ram jet (3.62) than for the augmented turbojet (2.50). (See fig. 7.)

Additional calculations show that the pressure ratio across the standard turbojet engine is 1.02 at a Mach number of 2.6. At this Mach number the net thrust per pound of air developed by the augmented turbojet and the ram jet should be nearly equal if both are burning sufficient fuel to consume all the oxygen in the air. Calculations show that at a Mach number of 2.6 and an altitude of 30,000 feet, the net thrust per pound of air for the augmented turbojet engine is 86.5 pounds of thrust per pound of air per second as compared with 83.5 for the ram jet. The discrepancy is attributed to departure of the value of the pressure ratio across the engine from unity and to the momentum-pressure loss in the ram jet.

Effect of Altitude on Engine Performance

The effect of changes in altitude on net thrust and specific fuel consumption based on net thrust are shown in figures 8 and 9, respectively, at flight Mach numbers of 1.2 and 1.6 for the three engines. At each Mach number the net thrust per unit frontal area for the standard turbojet engine decreases about 67 percent for an increase in altitude from sea level to 40,000 feet. The net thrust decreases 72 percent for the augmented turbojet engine and 76 percent for the ram-jet engine over the same range in altitude.

The effect of tail-pipe burning on net thrust decreases as the altitude increases (fig. 8). At a flight Mach number of 1.2 (fig. 8(a)), for example, tail-pipe burning increases the net thrust 134 percent at sea level, whereas the increase at 40,000 feet is only 108 percent.

The specific fuel consumption (fig. 9) for the standard turbojet engine and for the turbojet engine with a tail-pipe burner at Mach numbers of 1.2 and 1.6 decreases about 12 percent as the altitude increases from 0 to 40,000 feet. The specific fuel consumption for the ram jet is nearly constant for all altitudes from sea level to 40,000 feet at flight Mach numbers of 1.2 and 1.6.

CONCLUSIONS

An analysis was made of the computed performance of a standard turbojet engine, a turbojet engine augmented by tail-pipe burning, and a ram-jet engine. It is concluded that a turbojet engine whose thrust is augmented by burning fuel in the tail pipe is an advantageous system for the propulsion of aircraft at subsonic and low supersonic speeds. At these speeds, the augmented turbojet engine gives higher thrusts per unit frontal area than the standard turbojet engine and has a higher thrust per unit frontal area and a lower specific fuel consumption than the ram-jet engine. Greatest percentage increases in net thrust by means of tail-pipe burning are obtained at low altitudes and high flight Mach numbers.

At static conditions, burning sufficient fuel in the tail pipe to raise the fuel-air ratio to stoichiometric increases the net thrust of the turbojet engine about 60 percent and at flight velocities above the speed of sound the net thrust of the augmented turbojet engine is more than double that for the standard turbojet engine. A flight Mach number of about 1.2 must be reached before the net thrust per unit frontal area for a ram-jet engine is equal to that of an augmented turbojet engine and at this flight speed the ram jet has about a 45 percent higher specific fuel consumption based on net thrust. At a Mach number of 2.0 the specific fuel consumption of the ram-jet engine exceeds that of an augmented turbojet engine by 7 percent, whereas the net thrust per unit frontal area is 40 percent greater.

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APPENDIX - ANALYSIS OF THREE JET-PROPULSION ENGINES

Symbols

The following symbols were used in the analysis:

A	cross-sectional area, sq ft
a	velocity of sound in gas, ft/sec
c_p	specific heat at constant pressure, Btu/lb $^{\circ}$R
D	diameter, ft
F_j	jet thrust, lb
F_n	net thrust, lb
f	fuel-air ratio in combustion chamber of standard turbojet engine
g	acceleration of gravity, 32.2 ft/sec^2
k_1,k_2,k_3,k_4	constants
M	Mach number
N	engine speed, rpm
n	number of stages on compressor
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
R	universal gas constant, 53.4 ft-lb/lb $^{\circ}$R
T	total temperature, $^{\circ}$R
t	static temperature, $^{\circ}$R
V	velocity, ft/sec
W	mass flow of air, lb/sec

γ	ratio of specific heats
δ	pressure correction factor P/2116 (total pressure divided by NACA standard sea-level pressure)
η	efficiency
θ	temperature correction factor T/519 (total temperature divided by temperature of air at NACA standard sea-level conditions)
ρ	static density, lb/cu ft
τ	tail-pipe-burner temperature ratio, T_8/T_7
ψ	average compressor pressure coefficient per stage, ratio of isentropic increase in energy of air across compressor to isentropic increase in energy it would have moving at rotor tip speed divided by number of compressor stages

Subscripts for turbojet engines:

c	compressor
f	frontal
t	turbine
0	ambient or free stream
2	compressor inlet
3	compressor outlet
4	turbine inlet
5	turbine outlet
7	tail-pipe-burner inlet
8	tail-pipe-burner outlet
10	tail-pipe-nozzle outlet

Subscripts for ram jet:

f	frontal
0	ambient or free stream
1	diffuser inlet
2	burner inlet
3	burner outlet
4	tail-pipe-nozzle outlet

Equations for Analysis

Standard turbojet engine. - The mathematical analysis for the standard turbojet engine follows the assumptions listed in BASES OF ANALYSIS. The pressure and temperature ratios across the engine were derived from the assumptions that the enthalpy rise across the compressor is equal to the enthalpy drop across the turbine and that the mass flow of air in the compressor is equal to the mass flow of gas in the turbine.

$$W\ c_{p,c}\ (T_3 - T_2) = W\ c_{p,t}\ (T_4 - T_5) \tag{1}$$

and from the definition of compressor efficiency, turbine efficiency, and pressure coefficient given in the following equations:

$$\eta_c = \frac{T_2\left[\left(\frac{P_3}{P_2}\right)^{\frac{\gamma_c-1}{\gamma_c}} - 1\right]}{T_3 - T_2} \tag{2}$$

$$\eta_t = \frac{T_4 - T_5}{T_4\left[1 - \left(\frac{P_5}{P_4}\right)^{\frac{\gamma_t-1}{\gamma_t}}\right]} \tag{3}$$

$$\psi = \frac{2}{n\ (\gamma_c - 1)}\ \frac{\left[\left(\frac{P_3}{P_2}\right)^{\frac{\gamma_c-1}{\gamma_c}} - 1\right]}{M_c^{\,2}} \tag{4}$$

from which

$$\psi = \frac{\left[\left(\frac{P_3}{P_2}\right)^{\frac{\gamma_c-1}{\gamma_c}} - 1\right]}{k_1 \frac{N^2}{\theta_2}} \tag{5}$$

where

$$k_1 = \frac{n\ (\gamma_c - 1)\ \pi^2\ D_c^{\ 2}}{7200 \times 519\ \gamma_c\ g\ R} \tag{6}$$

When equations (2) and (3) are inserted in equation (1)

$$\frac{c_{p,c}\ T_2 \left[\left(\frac{P_3}{P_2}\right)^{\frac{\gamma_c-1}{\gamma_c}} - 1\right]}{\eta_c} = c_{p,t}\ \eta_t\ T_4 \left[1 - \left(\frac{P_5}{P_4}\right)^{\frac{\gamma_t-1}{\gamma_t}}\right]$$

and by means of equation (5)

$$\left(\frac{P_5}{P_4}\right)^{\frac{\gamma_t-1}{\gamma_t}} = 1 - \frac{c_{p,c}\ T_2\ \psi\ k_1\ N^2}{c_{p,t}\ T_4\ \eta_c\ \eta_t\ \theta_2} \tag{7}$$

But

$$\frac{P_5}{P_4}\frac{P_4}{P_3}\frac{P_3}{P_2} = \frac{P_3}{P_2}\frac{P_4}{P_3}\left(1 - \frac{c_{p,c}\ T_2\ \psi\ k_1\ N^2}{c_{p,t}\ T_4\ \eta_c\ \eta_t\ \theta_2}\right)^{\frac{\gamma_t}{\gamma_t-1}}$$

and again using equation (5)

$$\frac{P_5}{P_2} = \frac{P_4}{P_3}\left(1 + \frac{\psi\ k_1\ N^2}{\theta_2}\right)^{\frac{\gamma_c}{\gamma_c-1}} \left(1 - \frac{c_{p,c}\ T_2\ \psi\ k_1\ N^2}{c_{p,t}\ T_4\ \eta_c\ \eta_t\ \theta_2}\right)^{\frac{\gamma_t}{\gamma_t-1}} \tag{8}$$

When computations were made using equation (8), γ_c was taken as 1.4 and the value of γ_t was taken corresponding to the turbine-inlet temperature at a fuel-air ratio of 0.015.

The relation between the turbine-outlet temperature and the compressor-inlet temperature was derived from equations (1) and (2) rewritten in the form

$$\frac{T_5}{T_2} = \frac{T_4}{T_2} - \frac{c_{p,c}\left[\left(\frac{P_3}{P_2}\right)^{\frac{\gamma_c-1}{\gamma_c}} - 1\right]}{c_{p,t}\ \eta_c}$$

or by equation (5)

$$\frac{T_5}{T_2} = \frac{T_4}{T_2} - \frac{c_{p,c}}{c_{p,t}}\ \frac{k_1\ \psi\ N^2}{\eta_c\ \theta_2} \tag{9}$$

The tail-pipe nozzle outlet area was calculated from the relation

$$A_{10} = \frac{W}{\rho_{10}\ V_{10}}$$

Because the product of the pressure across the engine P_5/P_2 and the ram-pressure ratio P_2/p_0 corresponding to the flight Mach number was always greater than critical, the velocity of the gas at the tail-pipe-nozzle outlet was sonic or

$$V_{10} = a_{10}$$

From the isentropic relations between total and static temperatures and pressures at a Mach number of 1.0, the equation for tail-pipe nozzle outlet area becomes

$$A_{10} = \sqrt{\frac{R\ T_{10}}{g\ \gamma_{10}}}\ \frac{W}{P_{10}}\left(1 + \frac{\gamma_{10} - 1}{2}\right)^{\frac{\gamma_{10}+1}{2(\gamma_{10}-1)}} \tag{10}$$

Jet thrust and net thrust were computed from the following relations:

$$F_j = \frac{W}{g} V_{10} + A_{10} (p_{10} - p_0) \tag{11}$$

$$F_n = \frac{W}{g} (V_{10} - V_0) + A_{10} (p_{10} - p_0) \tag{12}$$

In these equations V_0, V_{10}, and p_{10} are evaluated by means of the following relations:

$$V_0 = M_0 \sqrt{\gamma_0 \, g \, R \, t_0} \tag{13}$$

$$p_{10} = \frac{P_{10}}{\left(1 + \frac{\gamma_{10} - 1}{2}\right)^{\frac{\gamma_{10}}{\gamma_{10} - 1}}} \tag{14}$$

$$V_{10} = \sqrt{\frac{2\gamma_{10} \, g \, R \, T_{10}}{\gamma_{10} + 1}} \tag{15}$$

In equations (14) and (15), γ_{10} was taken as that value corresponding to the static temperature at station 10 and a fuel-air ratio of 0.015. The actual fuel-air ratio was found from known values of T_5 and T_2 by means of figure 5 in reference 3. The specific fuel consumption based on net thrust was computed as follows:

$$\text{specific fuel consumption} = \frac{3600 \, W_f}{F_n} \tag{16}$$

Over a range of corrected compressor speeds from about 6350 to 8550 rpm, the turbine-outlet total temperature and the tail-pipe nozzle outlet area required to maintain limiting conditions on the engine were found to be constant. The temperature drop across the turbine was given by

$$\Delta T_t = T_4 - T_5 = \frac{519 \, k_1 \, c_{p,c} \, \psi \, N^2}{c_{p,t} \, \eta_c} \tag{17}$$

At limiting conditions all the variables on the right-hand side of equation (17) were constant except ψ and η_c. The ratio ψ/η_c

was constant over a range of corrected compressor speeds from 6250 to 8500 rpm as shown in figure 3. Therefore, the temperature drop across the turbine was constant and, because the turbine-inlet temperature T_4 had been assumed constant, the turbine-outlet temperature T_5 was also constant.

With T_4 and η_t assumed fixed and ΔT_t shown to be constant, P_5/P_4 was fixed. At limiting conditions there was always sonic velocity at the turbine-nozzle throat, so that $W\sqrt{T_4}/P_4$ was fixed and

$$\frac{W}{P_4} = k_2 \quad \text{and} \quad \frac{W}{P_5} = k_3$$

Equation (10) could then be written

$$A_{10} = k_4 \sqrt{\frac{T_{10}}{\gamma_{10}}} \left(1 + \frac{\gamma_{10} - 1}{2}\right)^{\frac{\gamma_{10}+1}{2(\gamma_{10}-1)}} \tag{18}$$

For a fixed tail-pipe temperature, the tail-pipe nozzle outlet area was constant as long as the ratio ψ/η_c was constant.

Turbojet engine with tail-pipe burner. - The final total temperature of the gas after burning sufficient fuel in the tail pipe to raise the fuel-air ratio for the engine and the tail-pipe burner to stoichiometric was computed from the charts of reference 4. The tail-pipe nozzle outlet area was given by the relation in equation (10).

Jet thrust and net thrust were computed from the following relations:

$$F_j = \frac{W}{g} (1.06775 - f)\, V_{10} + A_{10} (p_{10} - p_0) \tag{19}$$

$$F_n = F_j - \frac{W}{g} V_0 \tag{20}$$

Specific fuel consumption based on net thrust was found by

$$\text{specific fuel consumption} = \frac{244\ W}{F_n} \tag{21}$$

<u>Ram-jet engine.</u> - The momentum-pressure losses in the burner were determined from charts developed from equations given in reference 5. The total temperature after combustion with stoichiometric fuel-air ratio was obtained from the Hottel charts of reference 4.

For the case where the sonic velocity of the gas in the tail-pipe nozzle was not encountered, the area was defined by the relations

$$A_4 = \frac{W}{\rho_4 V_4}$$

$$t_4 = \frac{T_4}{\left(1 + \frac{\gamma_4 - 1}{2} M_4{}^2\right)} \qquad (22)$$

$$M_4 = \sqrt{\frac{2}{\gamma_4 - 1}\left[\left(\frac{P_4}{p_4}\right)^{\frac{\gamma_4-1}{\gamma_4}} - 1\right]}$$

and p_4 was equal to the atmospheric static pressure. The formulas used for jet thrust and net thrust for subsonic tail-pipe velocity are

$$F_j = \frac{1.06775\ W}{g} M_4 \sqrt{\gamma_4\ g\ R\ T_4 \left(\frac{p_4}{P_4}\right)^{\frac{\gamma_4-1}{\gamma_4}}} \qquad (23)$$

$$F_n = \frac{1.06775\ W}{g} M_4 \sqrt{\gamma_4\ g\ R\ T_4 \left(\frac{p_4}{P_4}\right)^{\frac{\gamma_4-1}{\gamma_4}}} - \frac{W}{g} V_0 \qquad (24)$$

After sonic conditions were reached in the tail pipe, the jet thrust was given by

$$F_j = \frac{W}{g}\ 1.06775\ V_4 + A_4\ (p_4 - p_0) \qquad (25)$$

and the net thrust by

$$F_n = \frac{W}{g} 1.06775 V_4 - \frac{W}{g} V_0 + A_4 (p_4 - p_0) \tag{26}$$

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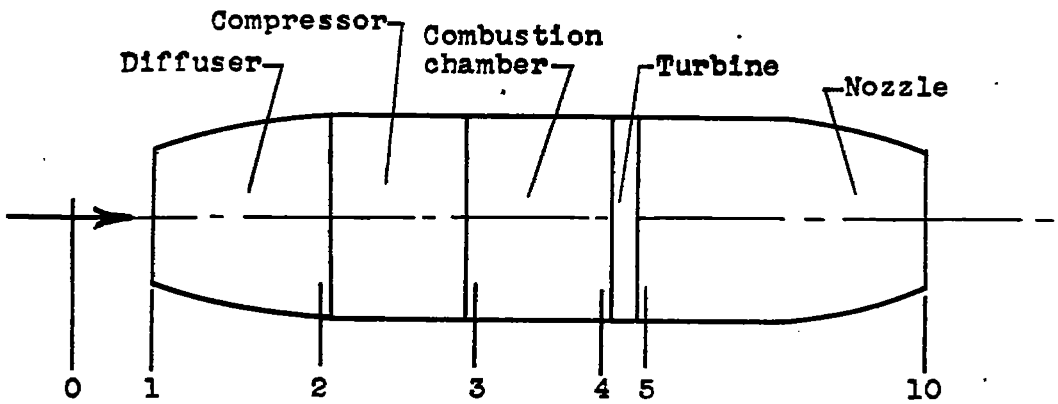
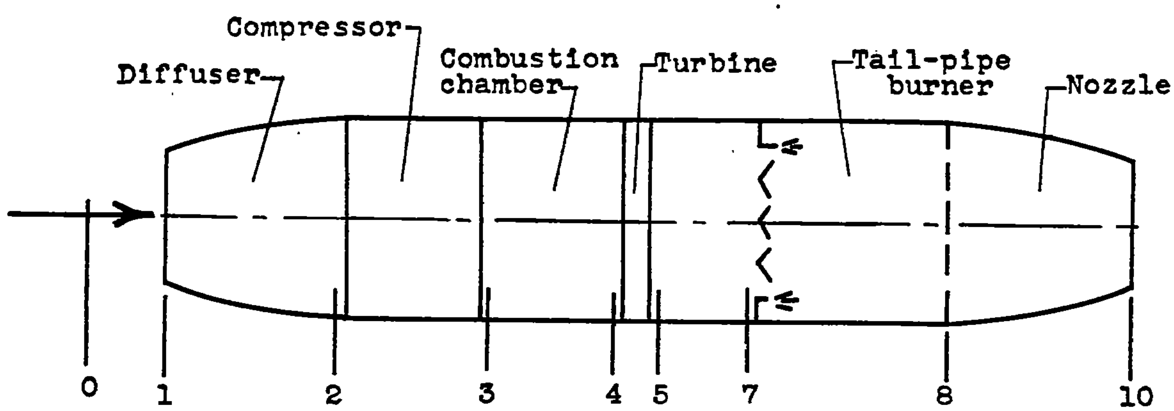
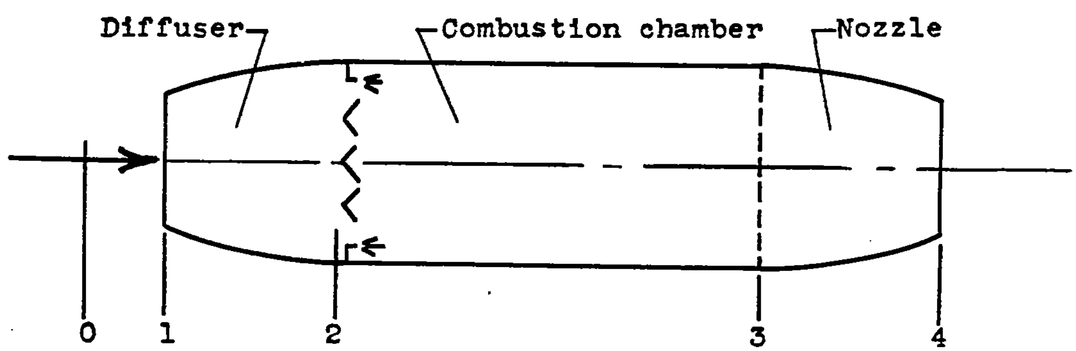


(a) Standard turbojet engine.



(b) Turbojet engine with tail-pipe burner.



(c) Ram-jet engine.

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Figure 1. - Stations used in analysis of three types of jet-propulsion engine.

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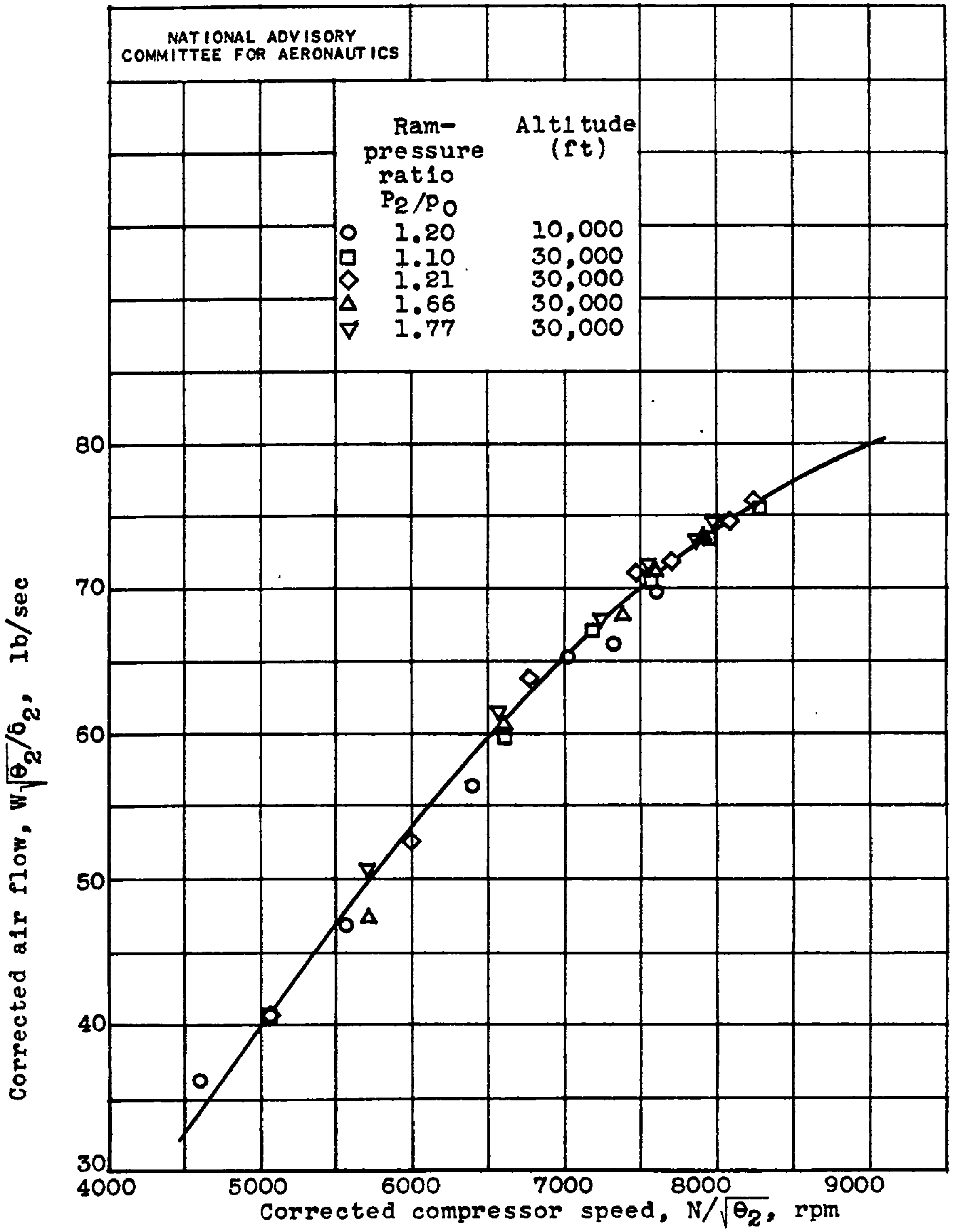


Figure 2.- Relation between corrected air flow and corrected compressor speed. Air flow and compressor speed corrected to NACA standard atmospheric conditions at sea level.

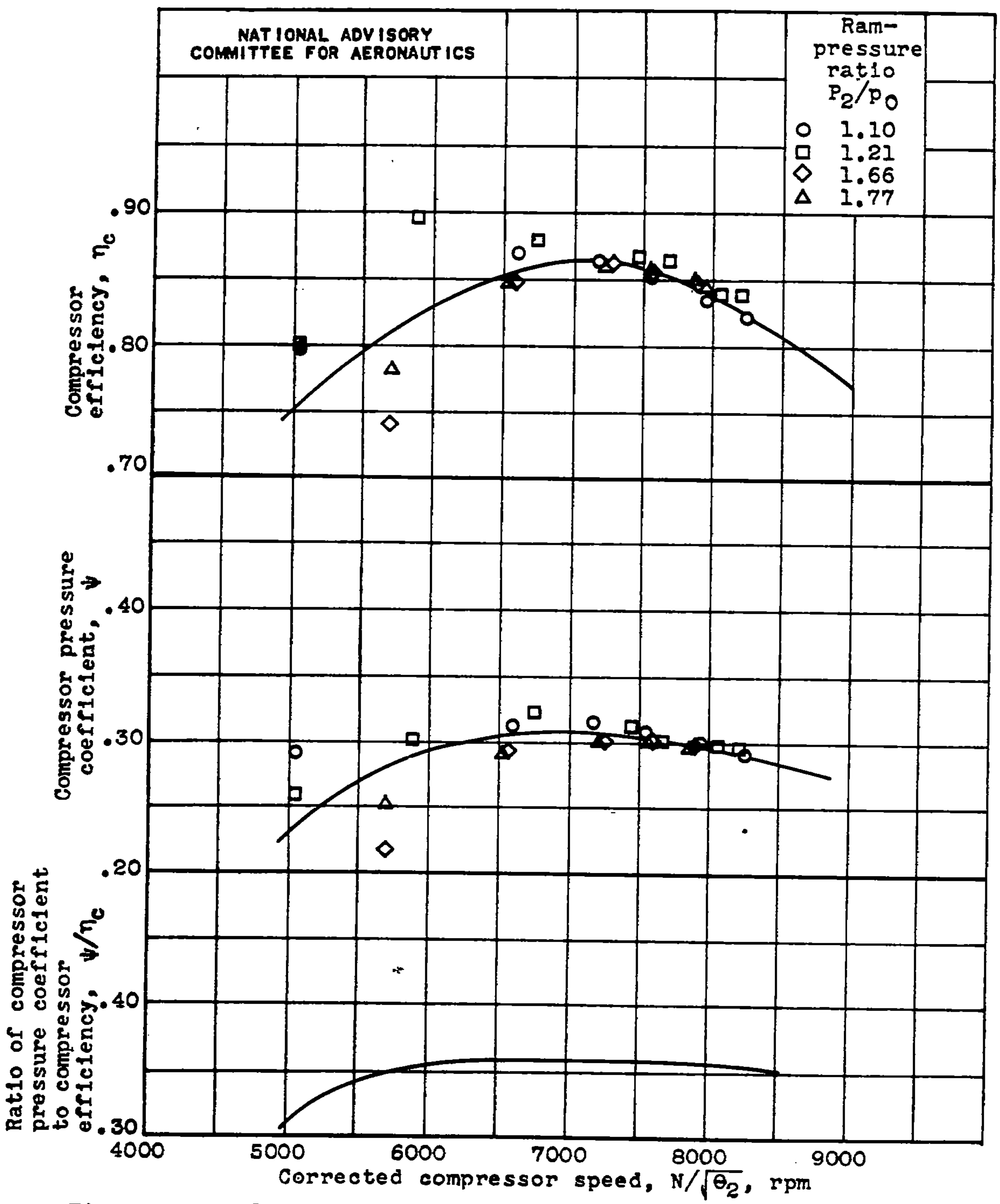


Figure 3.- Relation among compressor efficiency, compressor pressure coefficient, ratio of compressor pressure coefficient to compressor efficiency, and corrected compressor speed. Compressor speed corrected to NACA standard atmospheric conditions at sea level.

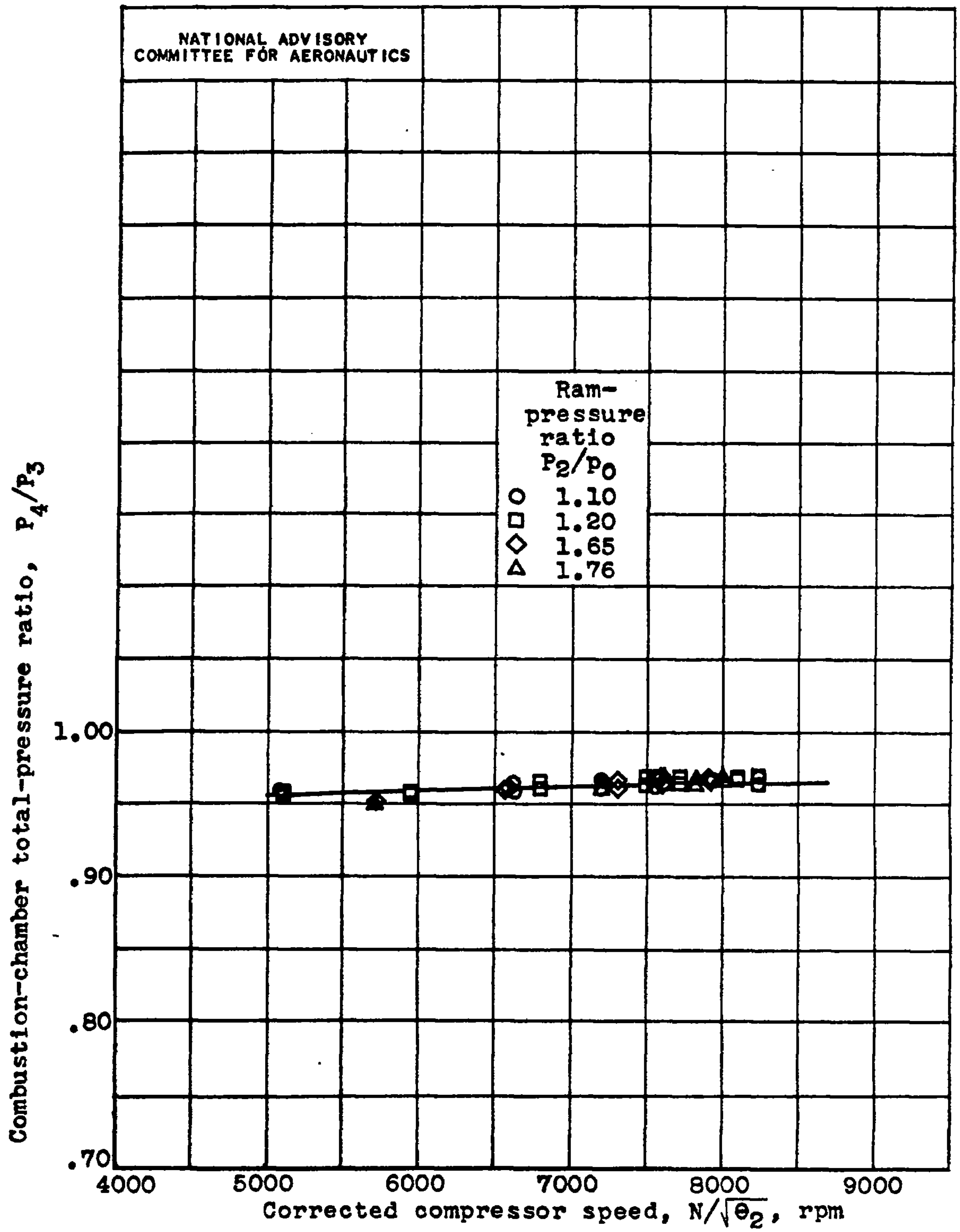


Figure 4.- Relation between combustion-chamber total-pressure ratio and corrected compressor speed. Altitude, 30,000 feet. Compressor speed corrected to NACA standard atmospheric conditions at sea level.

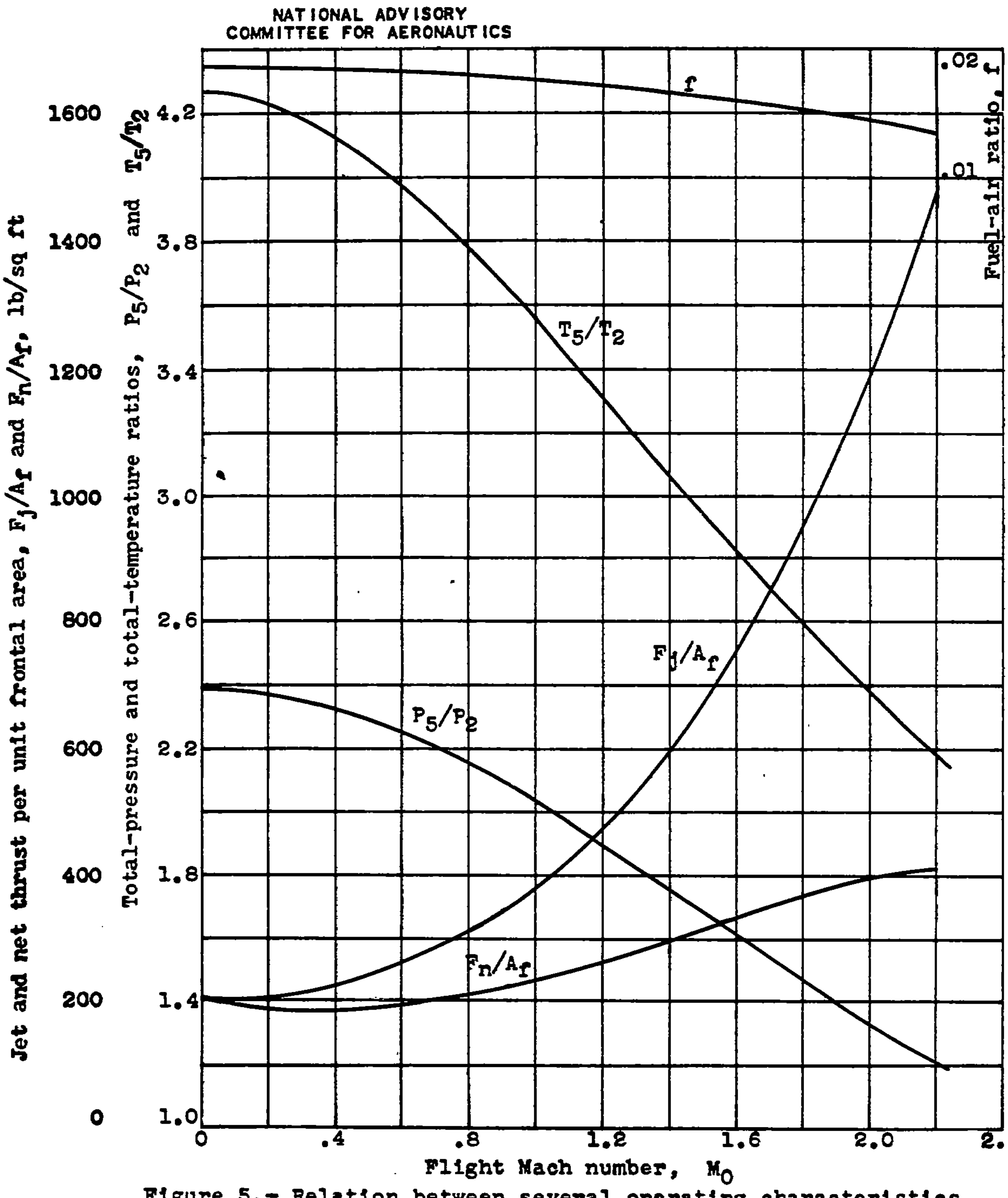


Figure 5.- Relation between several operating characteristics of standard turbojet engine and flight Mach number. Engine operating at limiting conditions; altitude, 30,000 feet; 100-percent ram recovery.

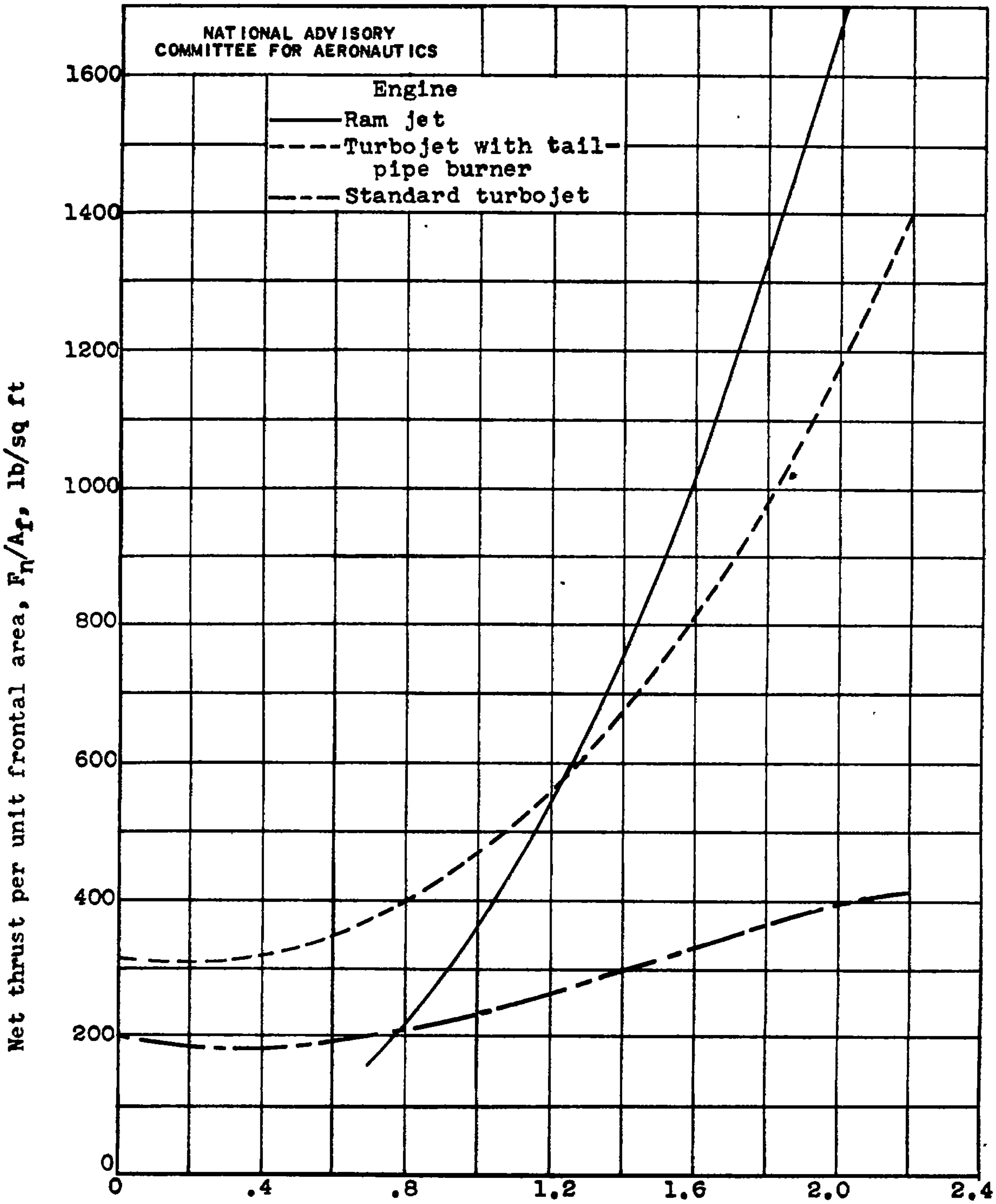


Figure 6.- Relation between net thrust per unit frontal area and flight Mach number for three jet-propulsion engines. Engines operating at respective limiting conditions; altitude, 30,000 feet; 100-percent ram recovery.

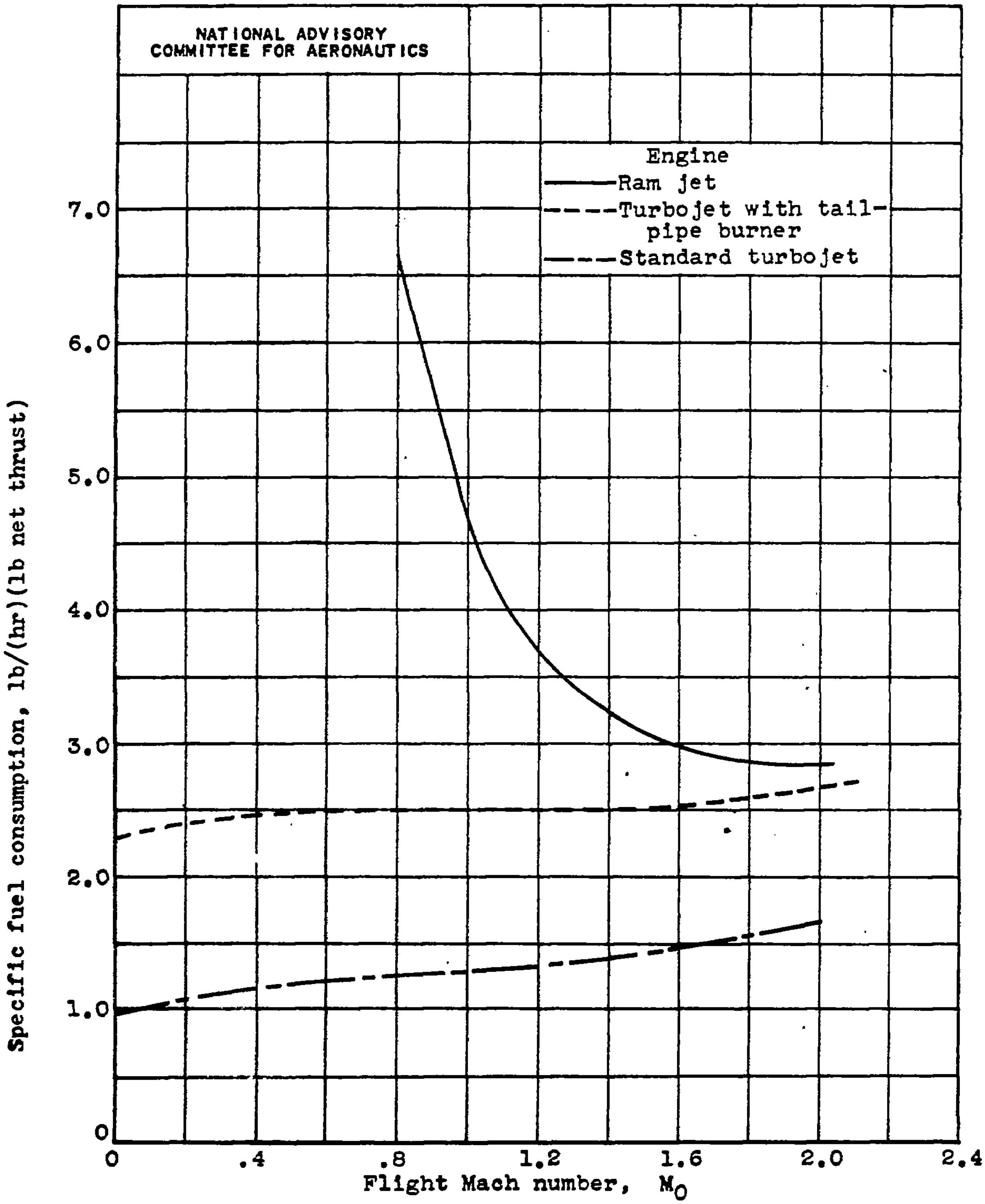
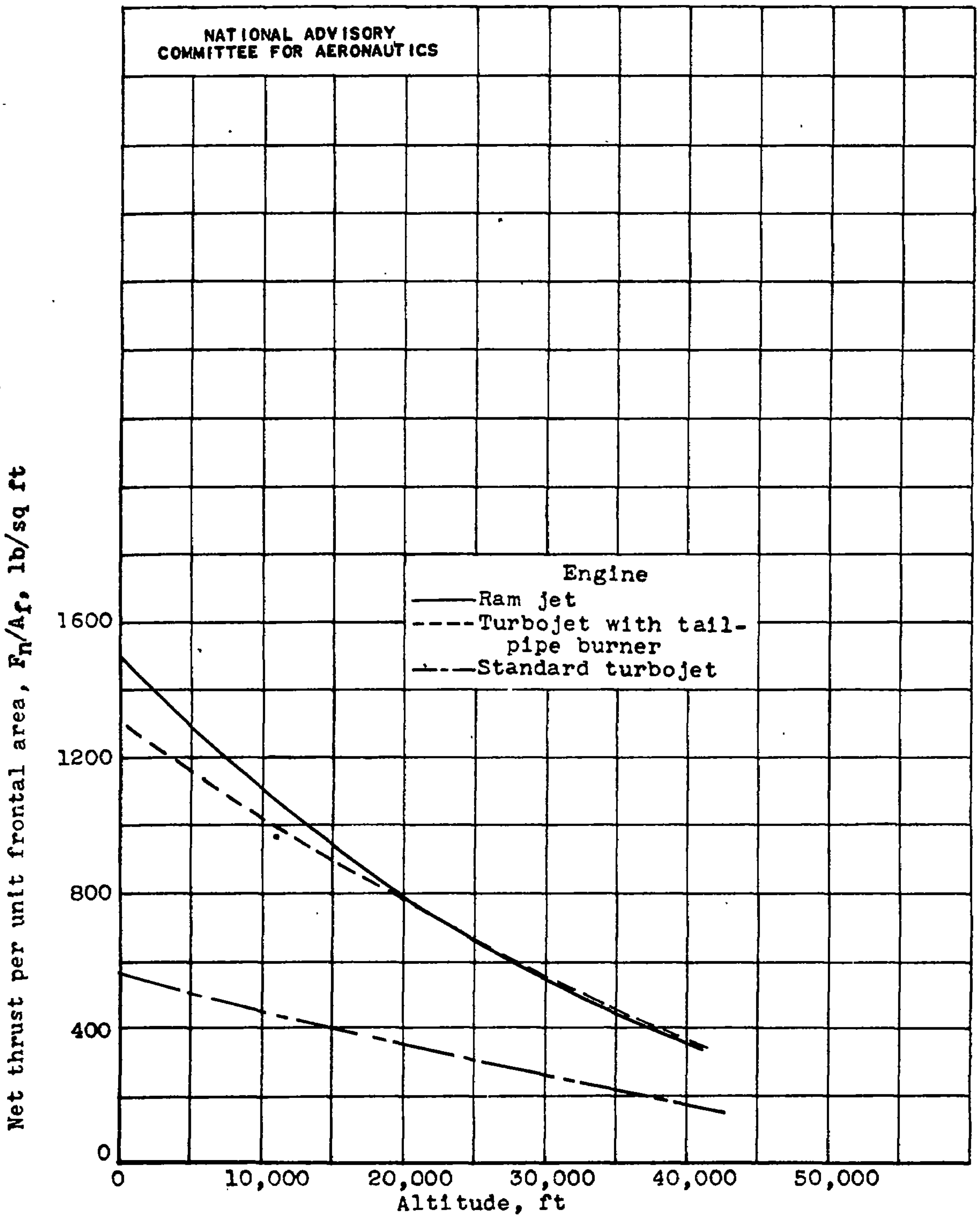


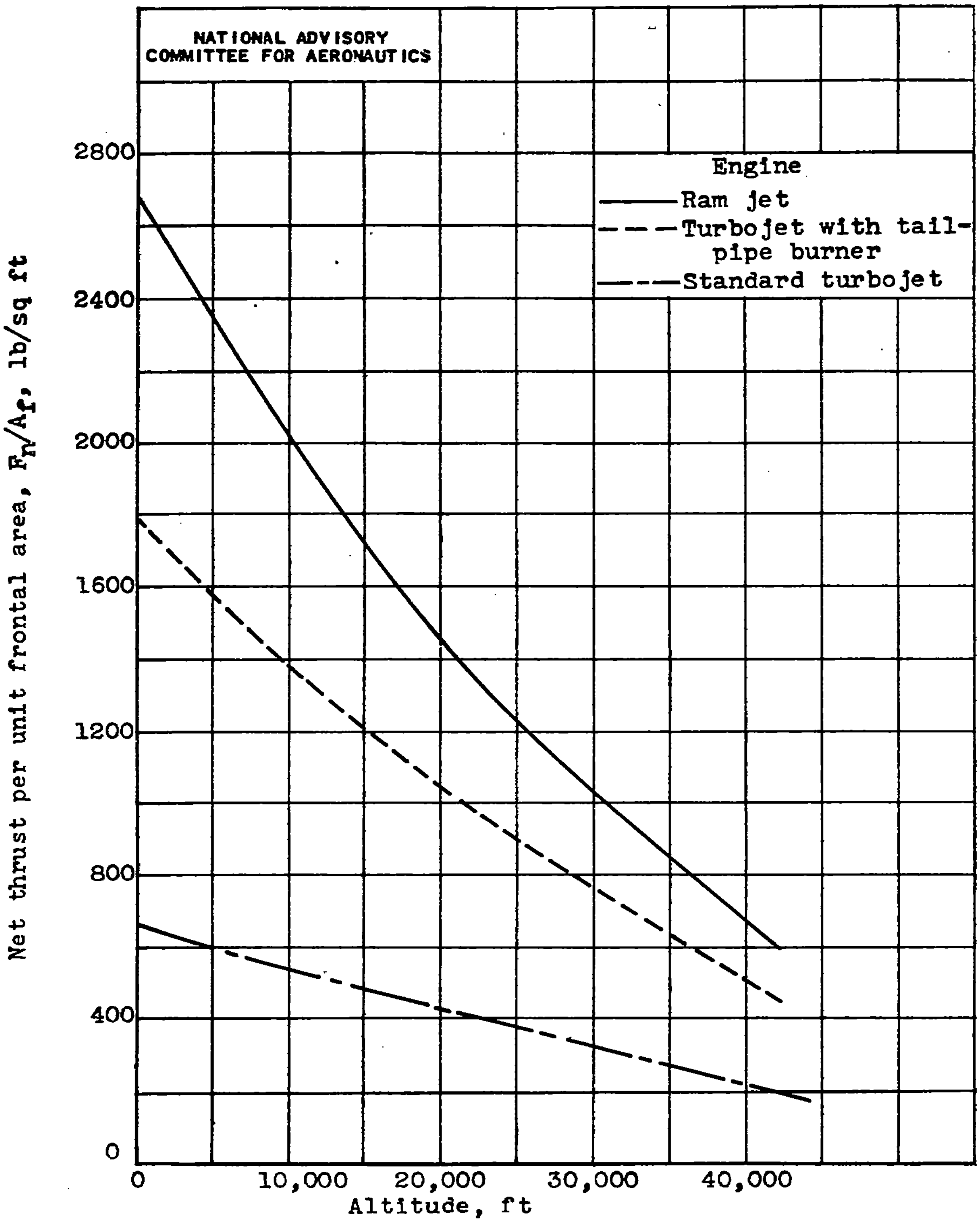
Figure 7.- Relation between specific fuel consumption based on net thrust and flight Mach number for three jet-propulsion engines. Engines operating at respective limiting conditions; altitude, 30,000 feet; 100-percent ram recovery.

[redacted]



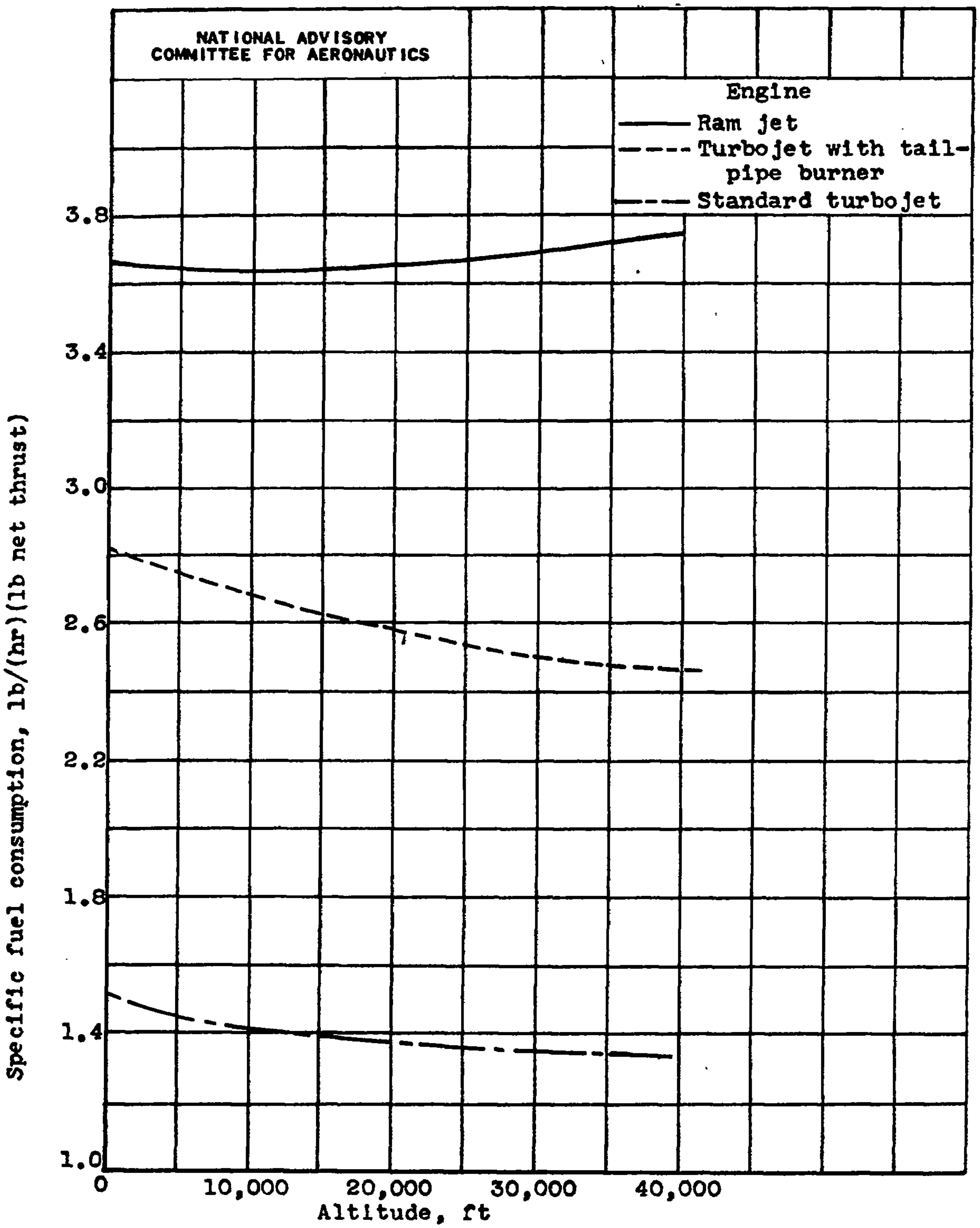
(a) Flight Mach number M_0, 1.2.

Figure 8.- Relation between net thrust per unit frontal area and altitude for three jet-propulsion engines. Engines operating at respective limiting conditions; 100-percent ram recovery.



(b) Flight Mach number M_0, 1.6.

Figure 8.- Concluded. Relation between net thrust per unit frontal area and altitude for three jet-propulsion engines. Engines operating at respective limiting conditions; 100-percent ram recovery.



(a) Flight Mach number M_0, 1.2.

Figure 9.- Relation between specific fuel consumption based on net thrust and altitude for three jet-propulsion engines. Engines operating at respective limiting conditions; 100-percent ram recovery.

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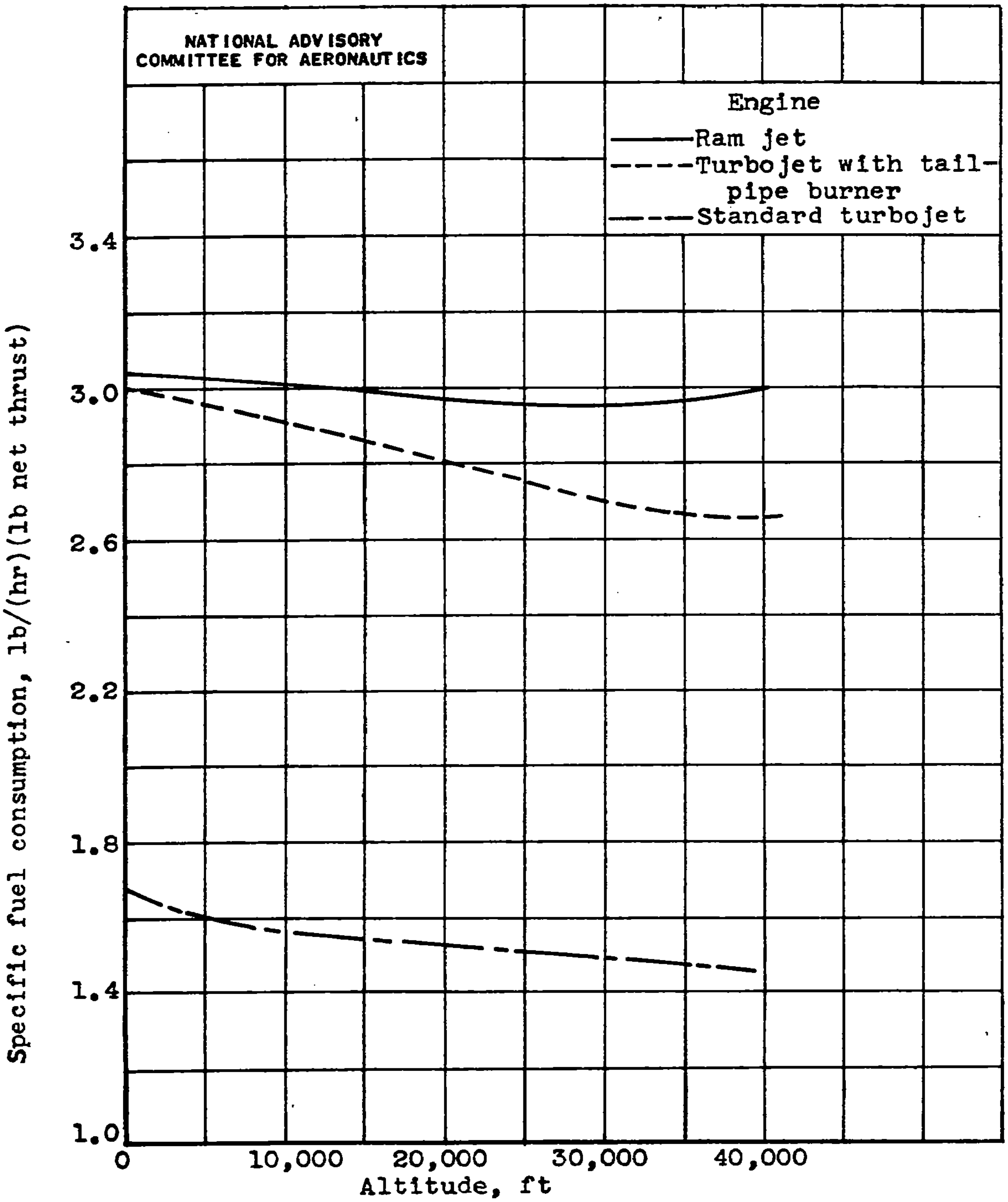

(b) Flight Mach number M_0, 1.6.

Figure 9.- Concluded. Relation between specific fuel consumption based on net thrust and altitude for three jet-propulsion engines. Engines operating at respective limiting conditions; 100-percent ram recovery.