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RESEARCH MEMORANDUM

ALTITUDE-WIND-TUNNEL INVESTIGATIONS OF THRUST AUGMENTATION OF A TURBOJET ENGINE

I - PERFORMANCE WITH TAIL-PIPE BURNING

By W. A. Fleming and R. O. Dietz

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SUMMARY

Thrust augmentation of a TG-180 turbojet engine by burning fuel in the tail pipe has been investigated in the Cleveland altitude wind tunnel. The engine thrust and the fuel consumption were determined for a wide range of simulated flight conditions and tail-pipe fuel flows. The investigation was particularly directed toward evaluation of thrust augmentation for high-speed and high-altitude flight. The engine tail pipe was modified for the investigation to reduce the gas velocity at the inlet of the tail-pipe combustion chamber.

The thrust of the standard TG-180 engine was increased 109 percent by tail-pipe burning when investigated under conditions corresponding to a Mach number of 1.18 at a simulated altitude of 30,000 feet. At these simulated flight conditions the specific fuel consumption, defined as the pounds of fuel burned in the engine and tail pipe per pound of net thrust, increased 67 percent above the specific fuel consumption for the standard engine without tail-pipe burning. The cycle efficiency and the thrust increment from tail-pipe burning decreased with a decrease in simulated flight speed. At a simulated altitude of 30,000 feet and a flight Mach number of 0.7, the thrust was augmented 95 percent and the specific consumption increased 91 percent above the consumption of the standard engine.

The general trends of the experimental values were in agreement with values calculated from theoretical equations.

INTRODUCTION

Thrust augmentation of turbojet engines to provide military ratings is of extreme importance in increasing their usefulness and range of application. The burning of fuel in the high-pressure region of the tail pipe provides a practical cycle for increasing the thrust of the jet engine without increasing the temperature or stresses in the turbine buckets or otherwise disturbing the normal cycle of the engine operation provided that the tail pipe is equipped with an adjustable-area nozzle.

An investigation of thrust augmentation by tail-pipe burning has therefore been conducted in the Cleveland altitude wind tunnel to determine whether the theoretically predicted performance of tail-pipe burner installations could be practically achieved. Particular attention was directed toward the evaluation of performance at high speeds and altitudes.

The most important requirements for the ideal tail-pipe burner are:

1. Maximum thrust at high efficiency

2. Wide range of stable burner operation

3. Minimum thrust loss for operation without tail-pipe burning

4. Minimum change in over-all dimensions of the engine

5. Adequate tail-pipe cooling

6. Light weight

The fulfillment of these objectives introduces numerous research problems. In this investigation attention was concentrated on the attainment of the first two requirements.

The standard tail pipe of the TG-180 turbojet engine was replaced with a larger tail pipe designed to provide favorable conditions for combustion; no particular consideration was given to size and weight of this installation. A series of interchangeable, fixed-area nozzles was used because no variable-area nozzle was available. The investigation was made at ram-pressure ratios between 1.045 and 2.35, corresponding to flight Mach numbers from 0.25 to 1.18, and at simulated altitudes of 5000 and 30,000 feet. Air was supplied to the engine through a duct at pressures corresponding to conditions at each simulated altitude and airspeed.

Performance results of the tail-pipe burning investigation on the TG-180 turbojet engine and a comparison of experimental results with theoretical calculations are presented.

SYMBOLS

The following symbols are used in the calculations:

A	cross-sectional area, square feet
a	speed of sound, feet per second
B	thrust scale reading, pounds
C_D	external drag coefficient of installation (determined from power-off tests)
c_p	specific heat of gas at constant pressure, Btu per pound per $^{\circ}F$
F_j	jet thrust, pounds
F_n	net thrust, pounds
g	acceleration of gravity, feet per second per second
J	mechanical equivalent of heat, foot-pounds per Btu
M	Mach number
m	mass flow, slugs per second
P	total pressure, pounds per square foot absolute
P_1/p_0	ram-pressure ratio
p	static pressure, pounds per square foot absolute
q	dynamic pressure, pounds per square foot
R	gas constant
S	wing-section area, square feet
T	total temperature, $^{\circ}R$
T_i	indicated temperature, $^{\circ}R$
t	static temperature, $^{\circ}R$

V velocity, feet per second

W_a air flow, pounds per second

W_f total fuel consumption, pounds per hour

W_{fe} turbojet-engine fuel consumption, pounds per hour

W_{ft} tail-pipe fuel consumption, pounds per hour

W_g exhaust gas flow, pounds per second

W_f/F_n specific fuel consumption based on net thrust and total fuel consumption, pounds per hour pound thrust

f/a fuel-air ratio based on total fuel flow to engine and tail pipe

γ ratio of specific heats for gases

ρ mass density of gas, slugs per cubic foot

τ total-temperature ratio across tail pipe, T_{10}/T_6

Subscripts:

g exhaust gas

j exhaust jet at vena contracta

r inlet duct at survey rake, station r

x inlet duct at slip joint, station x

0 tunnel test-section free-air stream

1 cowl inlet

6 diffuser inlet

10 tail-pipe nozzle outlet

FUNDAMENTALS OF TAIL-PIPE BURNING

The jet thrust of the turbojet engine is equal to the product of the mass rate of gas flow and the jet velocity. Thrust augmentation of a turbojet engine by burning fuel in the tail pipe results in an increase of the final jet velocity. The value of the final jet velocity is given by

$$V_j = a_j \sqrt{\left[\left(\frac{P_{10}}{p_0}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] \frac{2}{\gamma - 1}} \qquad (1)$$

Inasmuch as the speed of sound of the jet a_j is proportional to the square root of the absolute temperature of the jet, it follows that the jet thrust is also proportional to the square root of the absolute temperature of the jet. Maximum final jet temperatures are reached when sufficient fuel is added in the tail pipe to burn completely all the oxygen in the air passing through the engine. Maximum thrust is obtained when the engine is operated at the maximum allowable engine speed and turbine-outlet temperature. These operating conditions correspond to maximum total pressures of the gas at the discharge of the turbine.

As the amount of fuel burned in the tail pipe is varied, the tail-pipe nozzle area must be changed to maintain maximum allowable engine conditions. An expression for jet thrust involving jet area and jet Mach number is

$$F_j = \gamma p_0 A_j M_j^2 \qquad (2)$$

and the Mach number squared is

$$M_j^2 = \frac{2}{\gamma - 1}\left[\left(\frac{P_{10}}{p_0}\right)^{\frac{\gamma-1}{\gamma}} - 1\right] \qquad (3)$$

Jet Mach number M_j is principally a function of the total-pressure ratio P_{10}/p_0, which remains essentially constant at fixed engine and flight conditions and, if losses are disregarded, is independent of the jet temperature. At the maximum allowable engine operating conditions, the jet thrust is therefore proportional to the jet area (equation (2)). Because the jet thrust at fixed engine operating conditions is proportional to the square root of the jet temperature, it follows that the jet area required to maintain maximum allowable engine operating conditions will be proportional to the square root of the jet temperature.

INSTALLATION FOR TAIL-PIPE BURNING

IN TG-180 TURBOJET ENGINE

The TG-180 engine has an 11-stage axial-flow compressor, eight cylindrical combustion chambers, a single-stage turbine, and an exhaust nozzle. The over-all length of the standard engine is 14 feet and the maximum diameter is 36 inches.

The standard tail pipe was replaced by a modified tail pipe, which was lengthened to include a diffuser, a 5-foot combustion chamber, a reducer section, and a tail-pipe nozzle. A sketch of the installation is shown in figure 1. The diffuser (fig. 2) was designed to reduce the average gas speed to approximately 300 feet per second.

The flame holder used in the tail pipe (fig. 3) consists of vertical and horizontal V-type gutters and is a modification of the type found to be satisfactory in altitude wind tunnel investigations of an NACA 20-inch-diameter ram jet. Fuel was sprayed normal to the direction of gas flow through small holes in seven horizontal spray bars, which were attached to the upstream side of the horizontal V-type gutters in the flame holder and to a manifold outside the wall of the flame-holder section (fig. 3(a)). Fuel was supplied to the spray bars at pressures from 5 to 175 pounds per square inch, depending on the operating condition. The tail-pipe fuel was ignited either by means of propane blown over a spark plug or by rapid acceleration of the engine.

A circular combustion chamber 34 inches in diameter and 5 feet in length was located immediately downstream of the flame holder. A reducer section and an exhaust nozzle converged to form the desired outlet area. Downstream of the flame holder the tail pipe was wrapped with 0.75-inch-diameter copper tubing. Water was circulated through this tubing to prevent excessive shell temperature.

WIND-TUNNEL INSTALLATION AND TEST PROCEDURE

The TG-180 turbojet engine was suspended from a wing section installed in the 20-foot-diameter test section of the altitude wind tunnel (fig. 4). The installation was simplified by omitting the cowling. Dry refrigerated air was supplied to the engine through a duct from the tunnel make-up air system (fig. 5). A frictionless sealed slip joint in the inlet-air duct 40 feet upstream of the engine inlet made it possible to measure the thrust with the tunnel balance scale system. The air was throttled from approximately sea-level pressure to the desired pressure at the engine inlet while the pressure in the wind-tunnel test section was maintained at the desired altitude. The temperature of the engine inlet air was regulated to the approximate NACA standard temperature corresponding to the simulated flight speed and altitude.

Preliminary calibration runs were made with a standard engine equipped with a $16\frac{3}{4}$-inch-diameter nozzle to provide a basis for evaluating the changes in performance resulting from tail-pipe burning.

The investigations were conducted at simulated altitudes of 5000 and 30,000 feet and ram-pressure ratios from 1.045 to 2.35, which correspond to flight Mach numbers from 0.25 to 1.18. At each simulated flight condition, the TG-180 engine was operated at a speed of 7600 rpm and data were obtained at various fuel flows throughout the operable range of the tail-pipe combustion chamber. The minimum fuel flow was determined by combustion blow-out in the tail pipe and the maximum fuel flow was determined by the limiting turbine-discharge temperature (1220° F). Tail-pipe nozzles larger than standard were needed to permit high fuel flows for the tail-pipe burning without exceeding permissible turbine temperatures. Because no continuously variable outlet-area nozzle was available, fixed nozzles of 19-, $19\frac{3}{4}$-, and 21-inch diameters were substituted.

A survey rake was mounted in the inlet duct upstream of the engine inlet (fig. 5) to measure the engine air flow. Pressures and temperatures of the gases were measured at 10 stations in the engine (fig. 1). The tail-pipe rake, station 10, was so mounted that it could be retracted from the nozzle outlet. Measurements were made with this rake only for conditions with no tail-pipe burning because of the high gas temperatures when fuel was burning in the tail pipe. Thrust was determined from the balance scales for all the test conditions for which data are presented. The methods used to determine thrust and air flow from these measurements are given in appendix A. Kerosene (AN-F-32) was burned in the engine and 62-octane unleaded gasoline was burned in the tail-pipe combustion chamber. The fuel flow to each component was measured by calibrated rotameters.

DISCUSSION OF RESULTS

Data obtained in this investigation are presented in figures 6, 7, and 8 for three tail-pipe nozzle diameters, two altitudes, and five ram-pressure ratios, respectively. Each of these figures present (a) jet thrust, (b) net thrust, (c) engine fuel consumption, (d) specific fuel consumption based on net thrust, (e) air flow, (f) total fuel-air ratio, and (g) tail-pipe total-pressure ratio as functions of tail-pipe fuel consumption. A limit line is drawn in some figures to show the conditions at which the maximum allowable turbine-outlet temperature (1220° F) was attained.

The engine with the standard tail pipe and nozzle operated at approximately limiting tail-pipe temperatures at low altitude and low airspeed conditions. At 30,000 feet and at the high ram-pressure ratios at which the investigations were conducted, the tail-pipe temperature was lower than the maximum allowable temperature. A variable-area nozzle should be used to maintain limiting tail-pipe temperature. Because present installations of jet engines do not use a variable-area nozzle, the results of the tail-pipe burning tests have been compared with the engine using a standard nozzle with a fixed diameter of $16\frac{3}{4}$ inches.

The significant results of this investigation were obtained at conditions where the turbine-outlet temperatures reached the limiting value of 1220° F, as indicated by the dashed lines in figures 6 to 8. The succeeding discussion will be confined to the results obtained at these conditions.

The jet thrust at conditions of limiting turbine-outlet temperature and maximum engine speed increased in direct proportion to the increase in tail-pipe nozzle area, as shown in figure 9. The straight line on the graph shows the ideal thrust calculated by equations (2) and (3). The jet thrust obtained was about 17 percent less than the value computed by theory. The difference between theoretical and experimental results is attributed to friction and combustion-pressure losses in the tail pipe.

The loss in total pressure in the modified tail pipe was determined from the difference in total pressure at the diffuser inlet (station 6) and the total pressure at the tail-pipe nozzle outlet (station 10). The losses were measured when the engine was equipped with a standard $16\frac{3}{4}$-inch-diameter nozzle and no burning was present in the tail pipe. The cold friction coefficient $(P_6 - P_{10})/q_6$ was 0.7 q_6, which corresponded to a total pressure loss of 7 percent. The losses were smaller in the standard tail pipe than in the modified tail pipe; the data for the standard engine in figure 9 were therefore expected to fall between the theoretical and experimental curve. As previously mentioned, during operation at high simulated altitudes and high ram-pressure ratios, the limiting temperatures were not reached with the standard $16\frac{3}{4}$-inch-diameter tail-pipe nozzle, which explains why the thrust for the standard engine was lower than expected.

The relation among jet thrust, total temperature of the jet, and total-temperature ratio τ is shown in figure 10. The values of total temperature were computed by equation (B-5) in appendix B. Combustion temperatures of 3360° R were reached (fig. 10). Figure 11 shows the relation among tail-pipe fuel consumption, total fuel-air ratio, and total-temperature ratio across the tail pipe. A dashed line has been drawn showing the theoretical temperature ratio assuming 100-percent combustion efficiency and a heating value of the fuel equal to 19,000 Btu per pound. The theoretical temperature ratio did not increase linearly with the fuel-air ratio because of the variation in the specific heat of the gas with gas temperature. The mean value of tail-pipe combustion efficiency computed from this figure for results obtained at an altitude of 30,000 feet and a ram-pressure ratio of 1.66 is about 70 percent. At ram-pressure ratios of 1.045 and 2.35 combustion efficiencies of approximately 65 and 83 percent, respectively, were obtained. The increased combustion efficiency at high ram-pressure ratios results from the higher pressure in the tail pipe at the high ram conditions and is in general agreement with results obtained in other combustion studies.

The maximum total fuel-air ratio attained at a ram-pressure ratio of 1.66 and an altitude of 30,000 feet was 0.057 with a 21-inch-diameter tail-pipe nozzle. Fuel-air ratios closer to stoichiometric with correspondingly greater thrusts might have been obtained with a slightly larger tail-pipe nozzle.

The increase in thrust resulting from tail-pipe burning results in higher specific fuel consumptions than for the standard engine (fig. 12). When fuel was burned in the tail pipe at 30,000 feet and a ram-pressure ratio of 1.66, the net thrust was 3020 pounds with a net thrust specific fuel consumption of 2.50 pounds per hour per pound of net thrust as compared with a net thrust of 1530 pounds and a net thrust specific fuel consumption of 1.36 pounds per hour per pound of net thrust for the standard engine without tail-pipe burning at the same flight conditions.

The effect of altitude on tail-pipe burning was obtained from tests with a 21-inch-diameter tail-pipe nozzle at low ram-pressure ratios. The total fuel-air ratios and the tail-pipe total-pressure ratios P_6/p_0 at limiting tail-pipe temperatures were much lower at 5000 feet than at 30,000 feet (figs. 7(f) and 7(g)). The higher combustion efficiencies at low altitudes were primarily responsible for the lower observed fuel-air ratios at 5000 feet. The total-pressure ratio across the tail pipe increased at higher altitudes owing to the increase in the compression ratio of the engine compressor as the inlet temperature was lowered. As a result of the higher tail-pipe total-pressure ratios at the higher altitudes, larger percentage increases in thrust available from tail-pipe burning were possible at 30,000 feet than at 5000 feet (fig. 13).

Increasing the ram-pressure ratio increased the mass flow of gases through the engine and the total-pressure ratio P_6/p_0 across the tail pipe (figs. 8(e) and 8(g)). A cross plot showing the variation of tail-pipe total-pressure ratio with ram-pressure ratio at limiting turbine-outlet temperatures is shown in figure 14. Experimental and theoretical values of the variation of jet thrust with tail-pipe total-pressure ratio at limiting turbine-outlet temperatures and at an altitude of 30,000 feet are shown in figure 15. The theoretical results were calculated by means of equations (2) and (3). The experimental values are lower than the theoretical values owing to friction and combustion pressure losses in the tail pipe.

The measured values of net thrust at an altitude of 30,000 feet for the engine with a standard $16\frac{3}{4}$-inch-diameter nozzle and for the engine equipped with the tail-pipe combustion chamber and a 21-inch-diameter nozzle are shown in figure 16. The values given for the engine with tail-pipe burner represent a limiting temperature of 1220° F at the turbine outlet. The results are replotted in figure 17 to show the percentage increase in net thrust attributable to tail-pipe burning.

The net thrust was increased 71 percent at a Mach number of 0.25, 95 percent at 0.7, and 109 percent at 1.18. A further increase in thrust is believed possible by so increasing the nozzle size that the effective total fuel-air ratio of the engine is brought up to the stoichiometric value.

Specific fuel consumptions based on net thrust that correspond to the thrust results of figure 16 are shown in figure 18. The specific fuel consumption increased 126 percent at a Mach number of 0.25, 91 percent at 0.7, and 67 percent at 1.18. The improvement in efficiency and thrust augmentation with increase in airplane speed is clearly demonstrated.

SUMMARY OF RESULTS

Results from an investigation in the Cleveland altitude wind tunnel of thrust augmentation of a TG-180 turbojet engine with tail-pipe burning were as follows:

1. At an altitude of 30,000 feet, the net thrust of the standard engine was increased 71 percent at a Mach number of 0.25, 95 percent at a Mach number of 0.7, and 109 percent at a Mach number of 1.18. The corresponding increases in specific fuel consumption were 126, 91, and 67 percent.

2. As a result of the higher tail-pipe total-pressure ratios at the higher altitudes, larger percentage increases in thrust available from tail-pipe burning were possible at 30,000 feet than at 5000 feet.

3. An adjustable-area tail-pipe nozzle is required in order to obtain benefit from tail-pipe burning. At an altitude of 30,000 feet and a ram-pressure ratio of 1.66, the optimum diameter with tail-pipe burning was slightly larger than 21 inches, as compared with the standard nozzle diameter of $16\frac{3}{4}$ inches.

4. At an altitude of 30,000 feet, the combustion efficiency of the tail-pipe combustion chamber varied from 65 percent with the low pressure in the tail pipe, which is associated with a ram-pressure ratio of 1.045, to 83 percent with the higher tail-pipe pressure, which is associated with a ram-pressure ratio of 2.35.

5. The engine performance characteristics experimentally determined with tail-pipe burning were compared with those calculated from theoretical equations and the general trends of the experimental and theoretical values were in agreement. The jet thrust obtained with tail-pipe burning was 17 percent lower than the jet thrust estimated from a simple analysis that did not include losses.

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APPENDIX A

METHODS OF CALCULATION

Temperature

A cold calibration of a sample thermocouple up to a Mach number of about 0.8 showed that the thermocouple measured the static temperature plus approximately 85 percent of the adiabatic temperature rise owing to the impact of the air on the thermocouple. Static temperature may be determined from indicated temperature by applying this factor to the adiabatic relation between temperature and pressure in the following manner:

$$t = \frac{T_i}{1 + 0.85\left[\left(\frac{P}{p}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]} \tag{A-1}$$

and the total temperature

$$T = t\left(\frac{P}{p}\right)^{\frac{\gamma-1}{\gamma}} = \frac{T_i\left(\frac{P}{p}\right)^{\frac{\gamma-1}{\gamma}}}{1 + 0.85\left[\left(\frac{P}{p}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]} \tag{A-2}$$

Air Flow

The air flow through the engine was determined from pressure and temperature measurements obtained with a vertical survey rake installed in the inlet duct $11\frac{1}{4}$ feet ahead of the engine inlet (fig. 5). Air flow was calculated by

$$W_a = \rho_r A_r V_r g = \frac{p_r A_r}{R} \sqrt{\frac{2Jgc_p}{t_r}\left[\left(\frac{P_r}{p_r}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]} \tag{A-3}$$

The static temperature in equation (A-3) was obtained by use of equation (A-1).

Jet Thrust

Jet thrust was determined from the balance-scale measurements by combining the forces on the installation in the following equation:

$$F_j = B + C_D q_0 S + \frac{W_a V_x}{g} + A_x(p_x - p_0) \qquad \text{(A-4)}$$

The second term in the right-hand side of equation (A-4) represents the external drag of the installation and the third and fourth terms combined represent the force on the installation at the frictionless slip joint in the inlet-air duct.

Equivalent Airspeed

Inasmuch as all calculations are based on 100-percent ram recovery, the equivalent airspeed corresponding to the ram-pressure ratio at the engine inlet can be expressed by

$$V_0 = \sqrt{2Jgc_p T_{i1} \cdot \left[1 - \left(\frac{p_0}{P_1}\right)^{\frac{\gamma-1}{\gamma}}\right]} \qquad \text{(A-5)}$$

Because the adiabatic temperature rise due to the cowl-inlet velocity was low, the equivalent free-stream total temperature can be assumed equal to the cowl-inlet indicated temperature. The use of this assumption introduces an error in airspeed of less than 1 percent.

Net Thrust

When equations (A-3), (A-4), and (A-5) are combined, the equivalent free-stream momentum of the inlet air may be subtracted from the jet thrust and the following equation for net thrust is obtained:

$$F_n = F_j - \frac{W_a V_0}{g} \qquad \text{(A-6)}$$

APPENDIX B

DERIVATION OF EQUATION FOR ESTIMATING GAS TEMPERATURE FROM THRUST AND AIR FLOW

In all cases covered in this report, the jet velocity was supersonic. It is therefore assumed that sonic velocity exists at the outlet of the tail-pipe nozzle, station 10. The jet thrust F_j is given by

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

The velocity V_{10} equals the sonic velocity a_{10}; consequently,

$$F_j = \frac{W_g}{g} a_{10} + A_{10}(p_{10} - p_0) \qquad \text{(B-1)}$$

The pressure p_{10} at the nozzle outlet can be eliminated by the equation of continuity

$$W_g = \rho_{10} A_{10} V_{10} g$$

$$= \frac{\gamma g p_{10} A_{10}}{a_{10}} M_{10}$$

But $M_{10} = 1.0$; therefore

$$p_{10} = \frac{W_g a_{10}}{\gamma g A_{10}} .$$

Equation (B-1) can then be reduced to

$$F_j = \frac{W_g a_{10}}{g} \left(1 + \frac{1}{\gamma}\right) - A_{10} p_0$$

Solving for a_{10},

$$a_{10} = \frac{F_j + p_0 A_{10}}{W_g} \left(\frac{\gamma g}{\gamma + 1}\right) \tag{B-2}$$

The static temperature at station 10 is related to a_{10} by

$$t_{10} = \frac{a_{10}^2}{\gamma g R} \tag{B-3}$$

and the relation of total to static temperature at station 10, where $M = 1.0$, is

$$\frac{T}{t} = 1 + \frac{\gamma - 1}{2} = \frac{1 + \gamma}{2} \tag{B-4}$$

When equations (B-2), (B-3), and (B-4) are combined, the following expression for the total temperature T_{10} is obtained:

$$T_{10} = \frac{(F_j + p_0 A_{10})^2}{W_g^2} \frac{\gamma g}{2R(\gamma + 1)} \tag{B-5}$$

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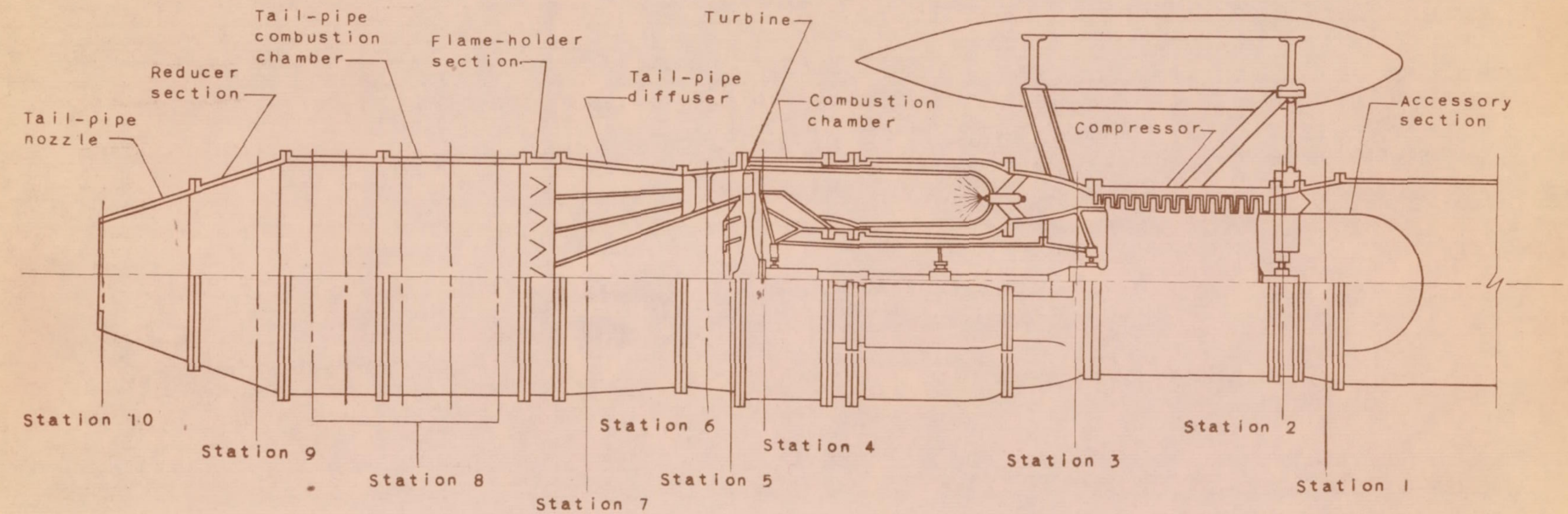

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Figure 1. - Installation of TG-180 turbojet engine with tail-pipe combustion chamber showing relation of component parts and measuring stations in engine.

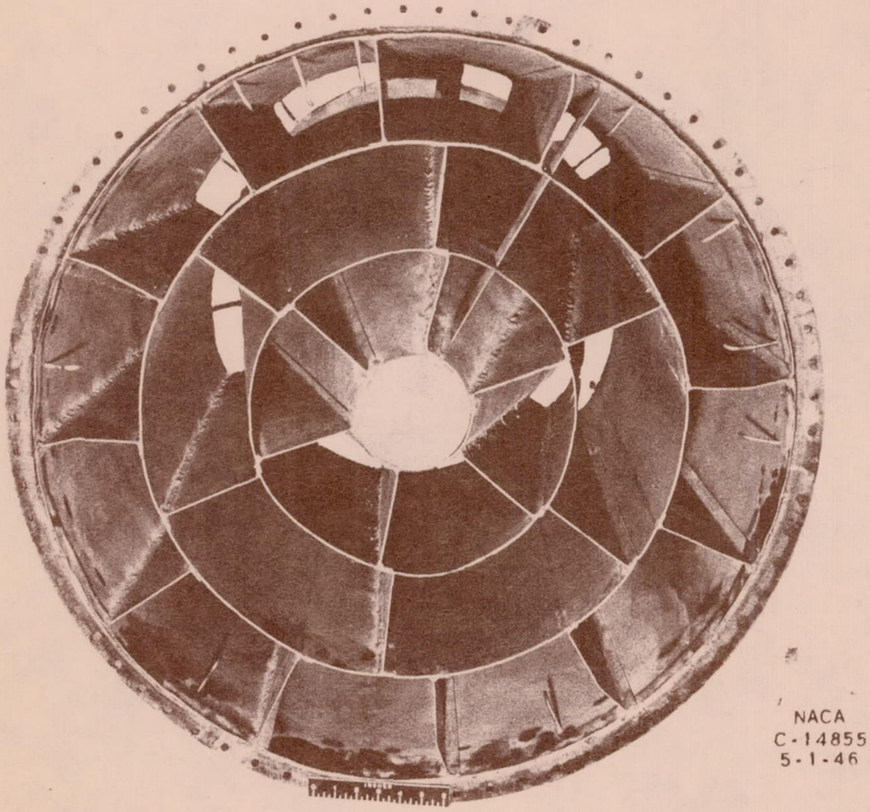

Figure 2. - Downstream end of diffuser for modified tail-pipe of TG-180 turbojet engine.

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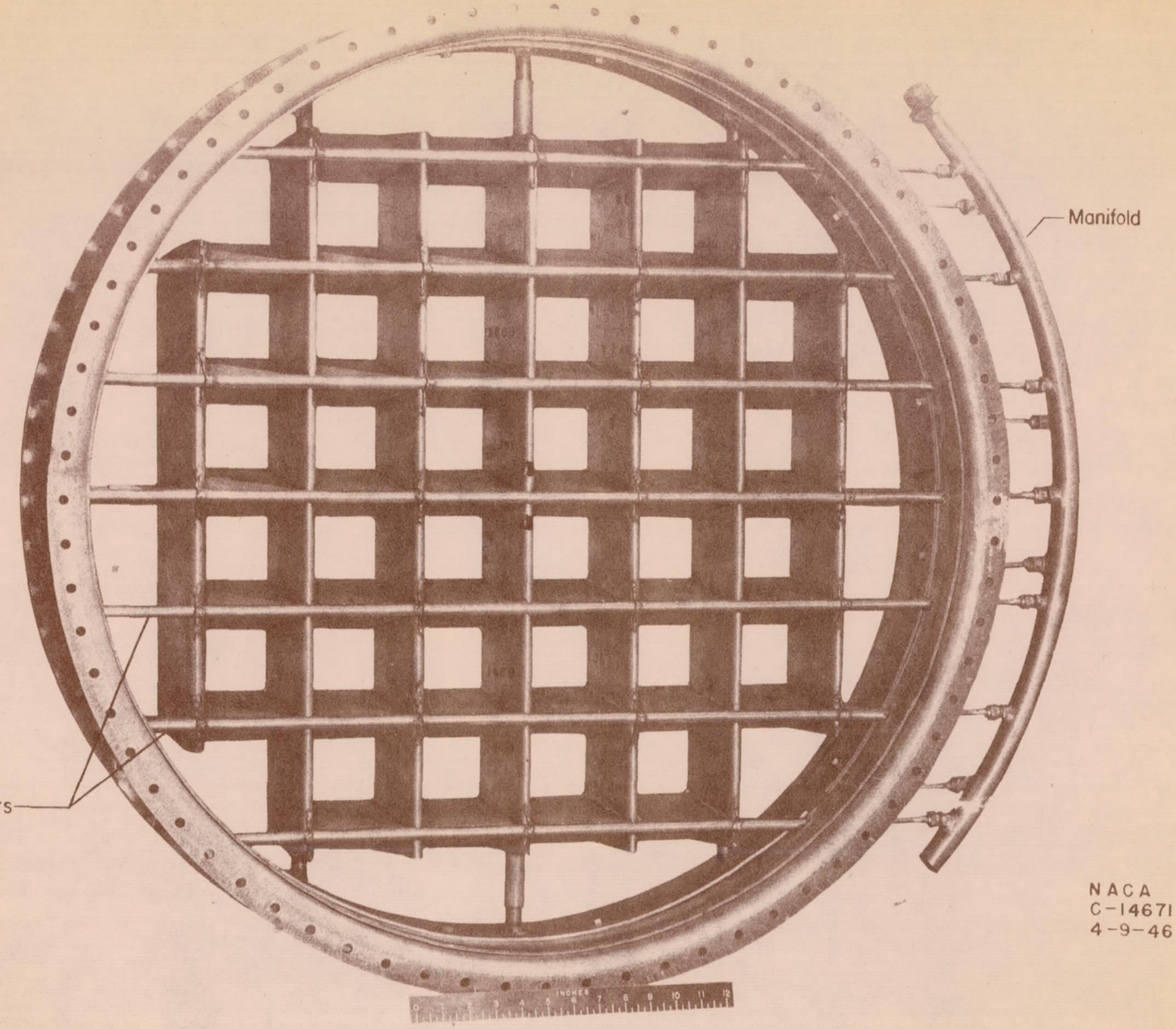

(a) Upstream view of flame holder.

Figure 3. - Flame holder installed at inlet of tail-pipe combustion chamber of TG-180 turbojet engine.

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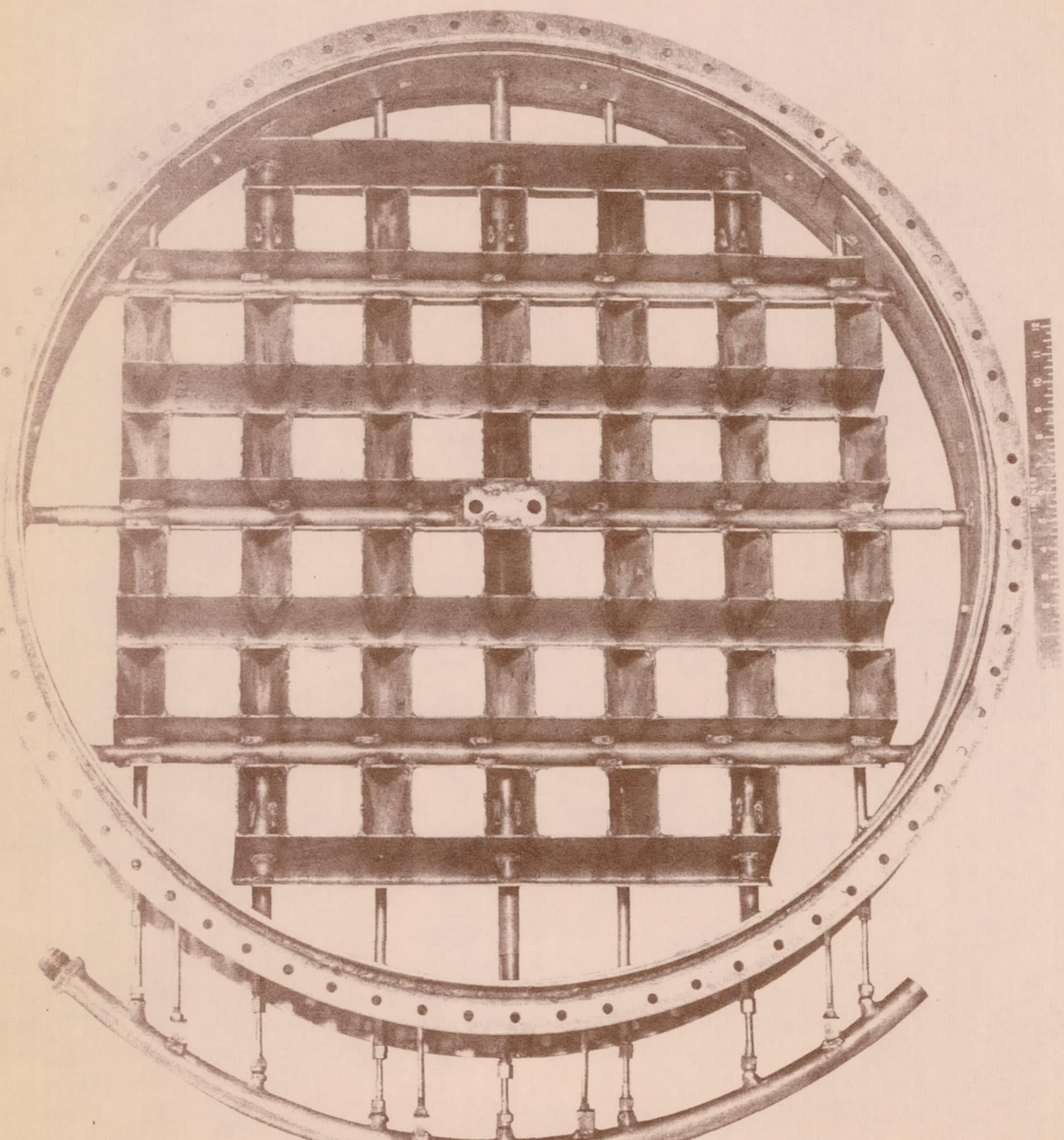

(b) Downstream view of flame holder.

Figure 3. - Concluded. Flame holder installed at inlet of tail-pipe combustion chamber of TG-180 turbojet engine.

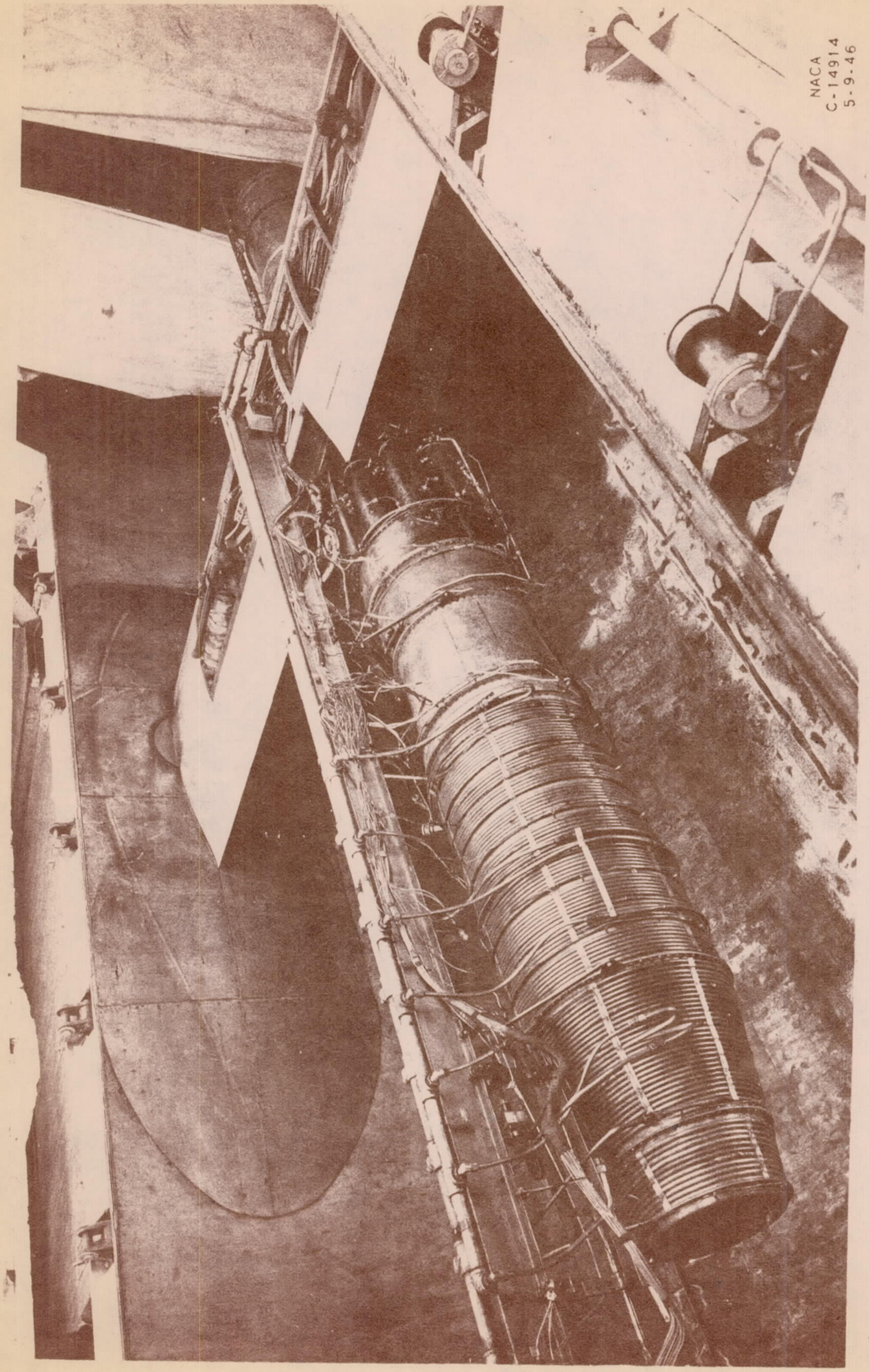

Figure 4. - Wind-tunnel installation of TG-180 turbojet engine with tail-pipe combustion chamber.

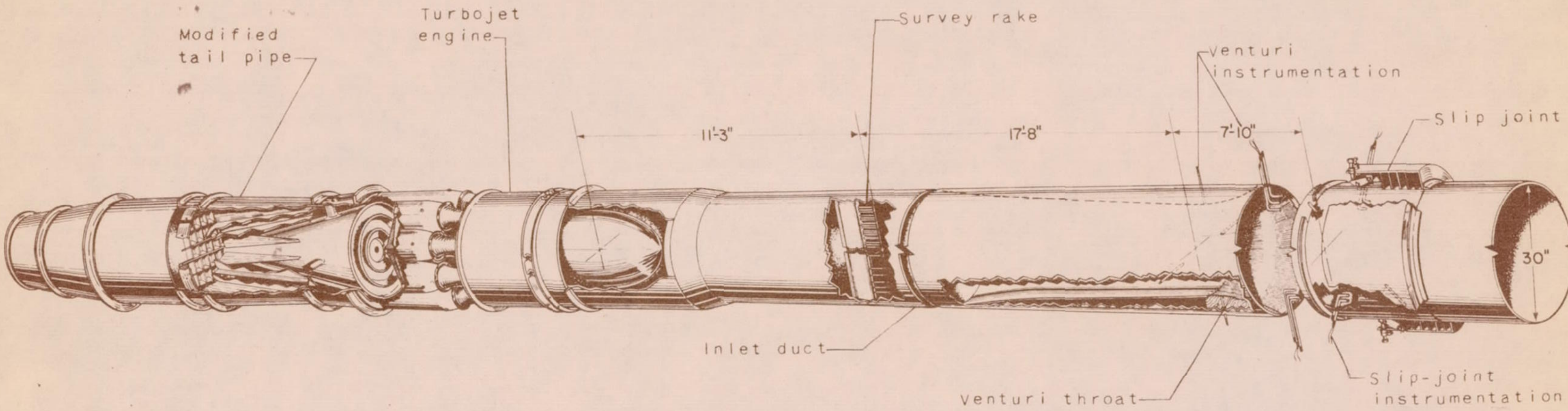

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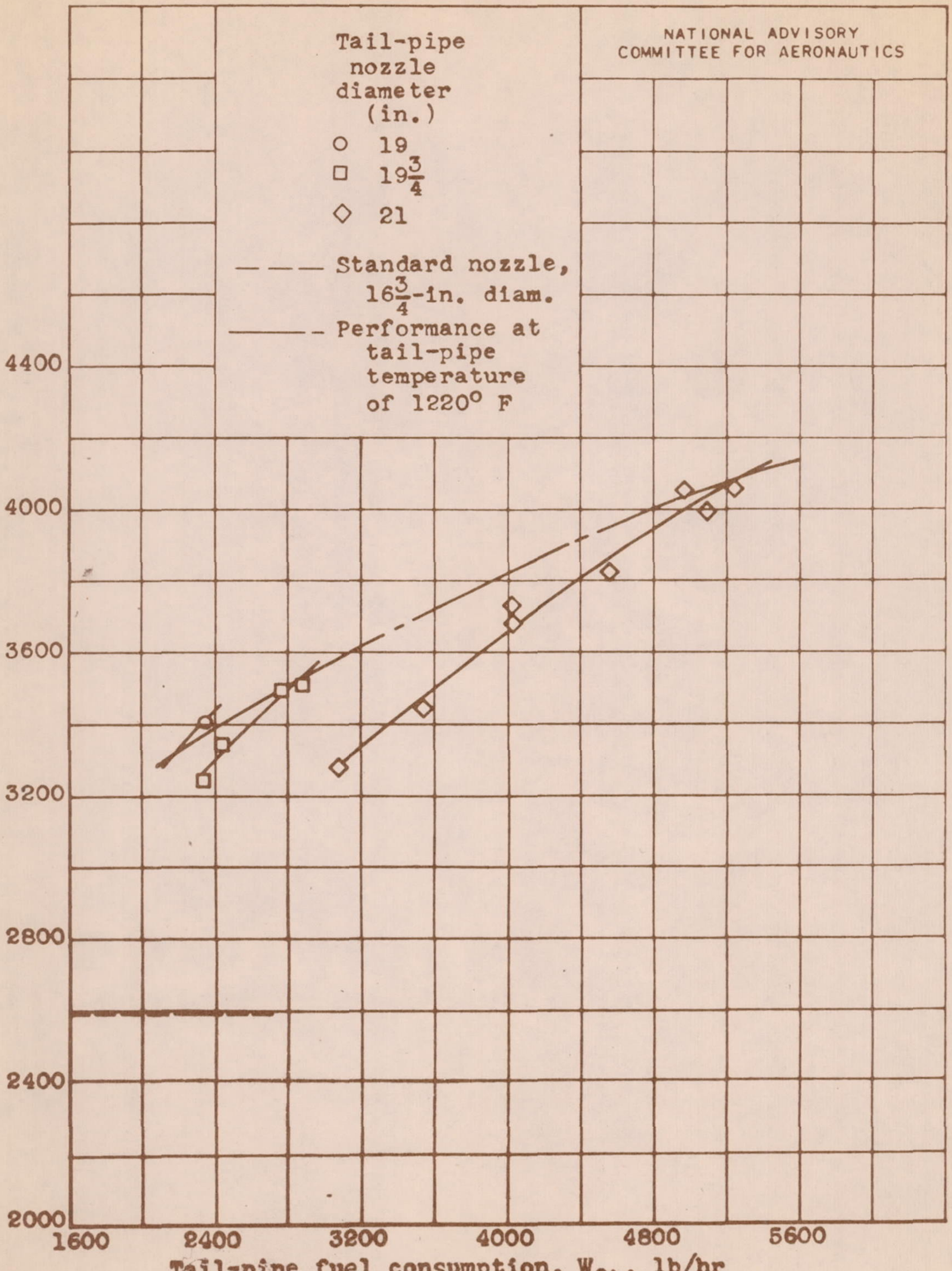
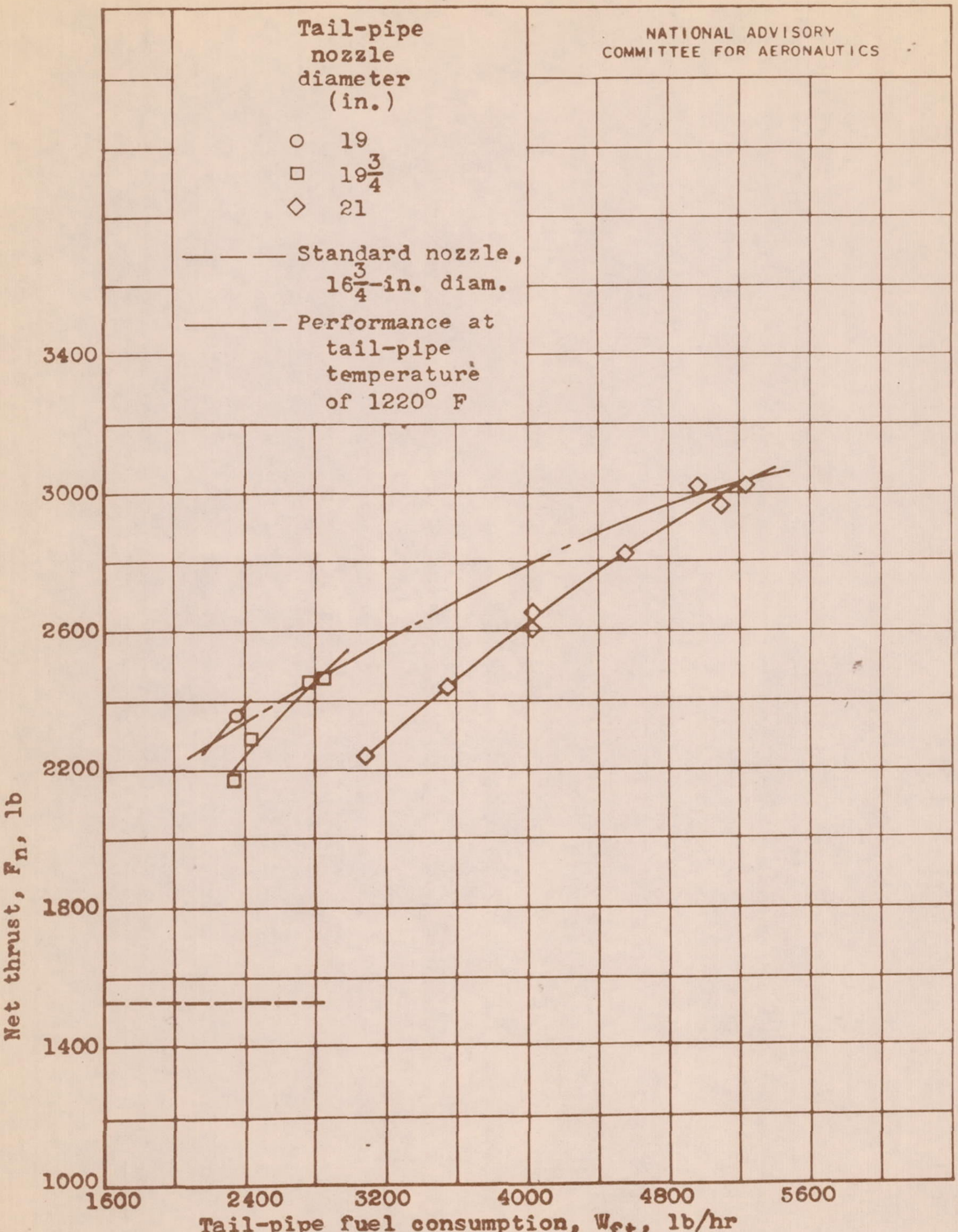
Figure 5. - Installation of TG-180 turbojet engine showing duct from tunnel make-up air system to engine inlet for supplying air at ram pressures.



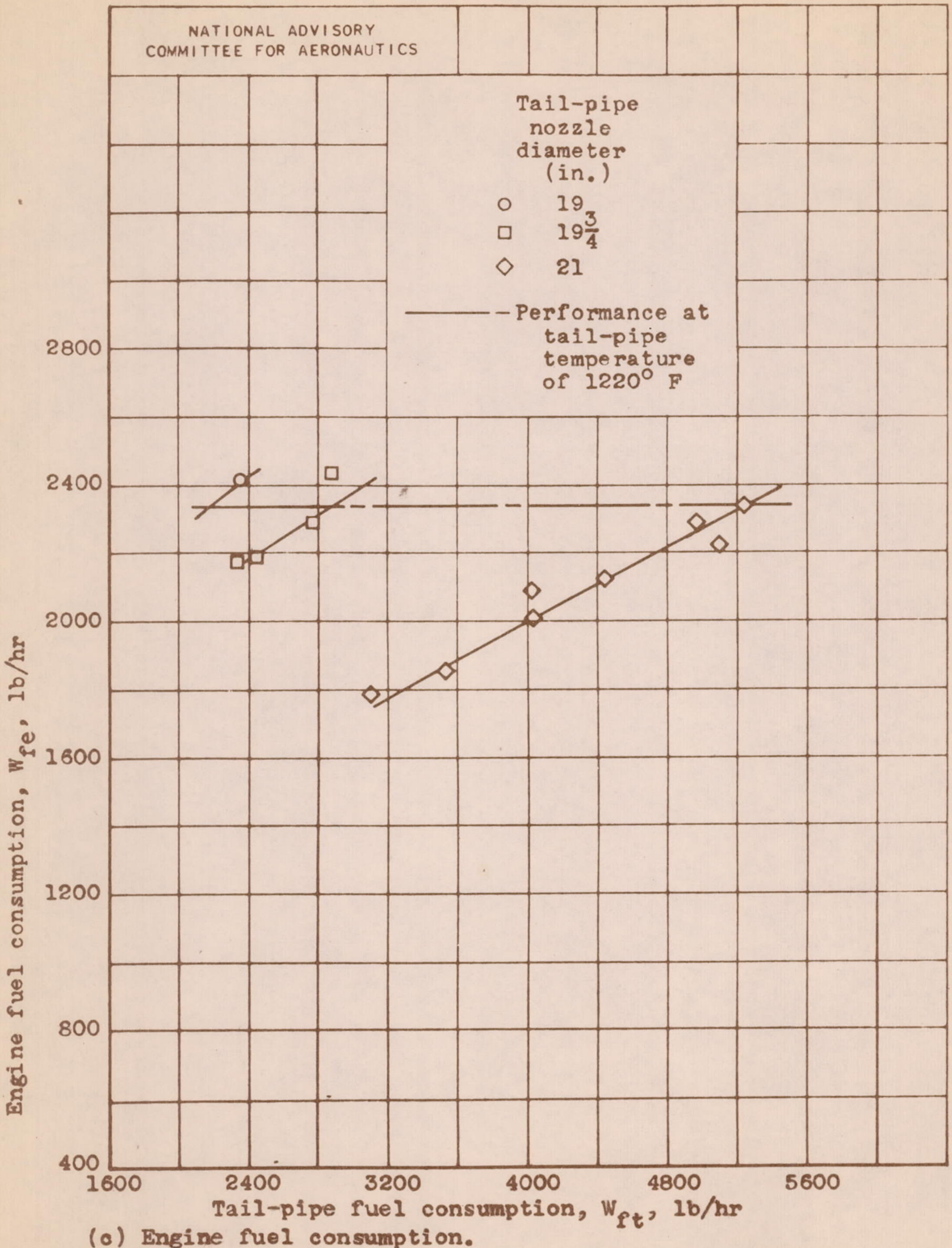
(a) Jet thrust.

Figure 6.- Variation of turbojet engine performance with tail-pipe fuel consumption for three tail-pipe nozzle diameters. Engine speed, 7600 rpm; altitude, 30,000 feet; ram-pressure ratio, 1.66.



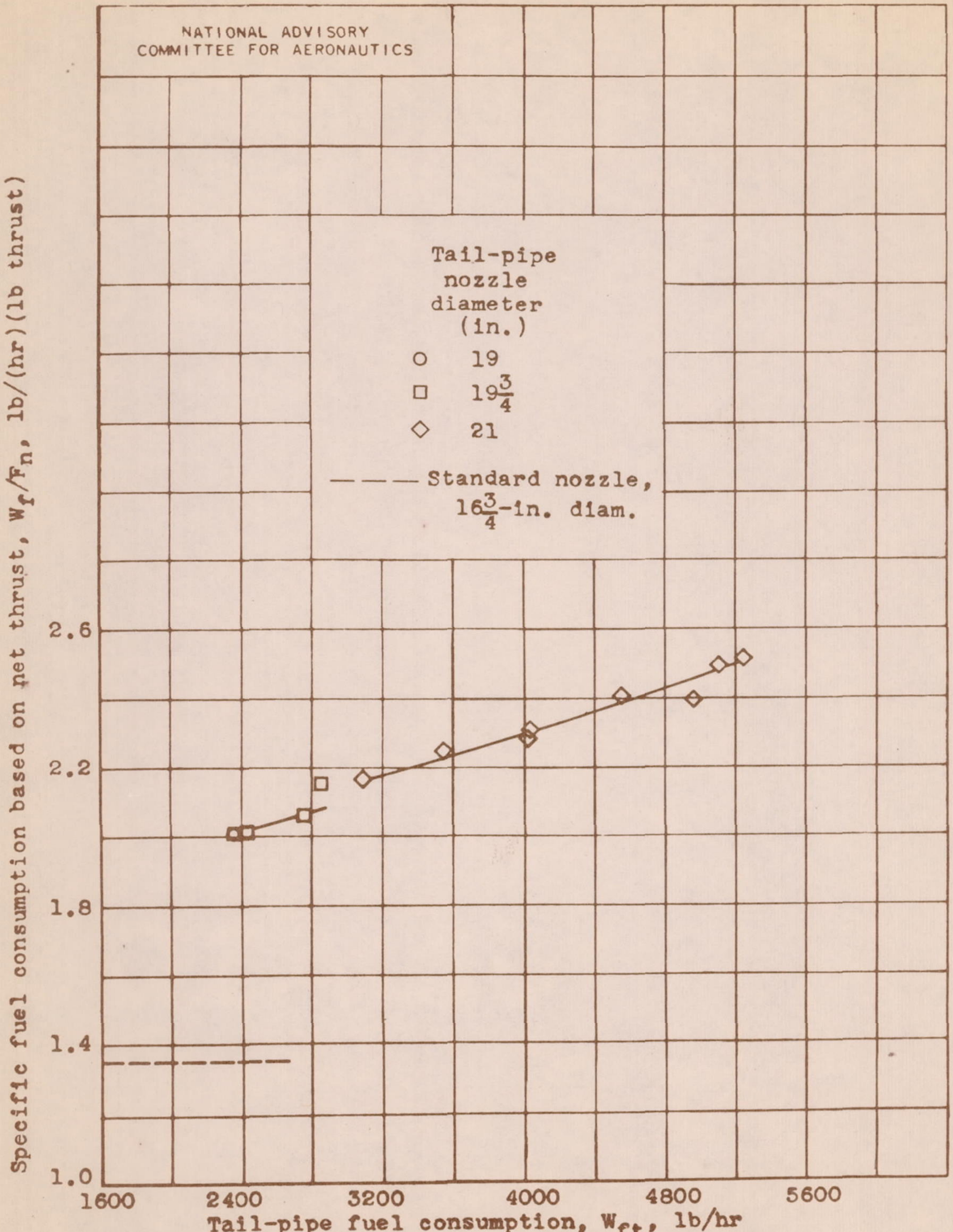
(b) Net thrust.

Figure 6.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for three tail-pipe nozzle diameters. Engine speed, 7600 rpm; altitude, 30,000 feet; ram-pressure ratio, 1.66.



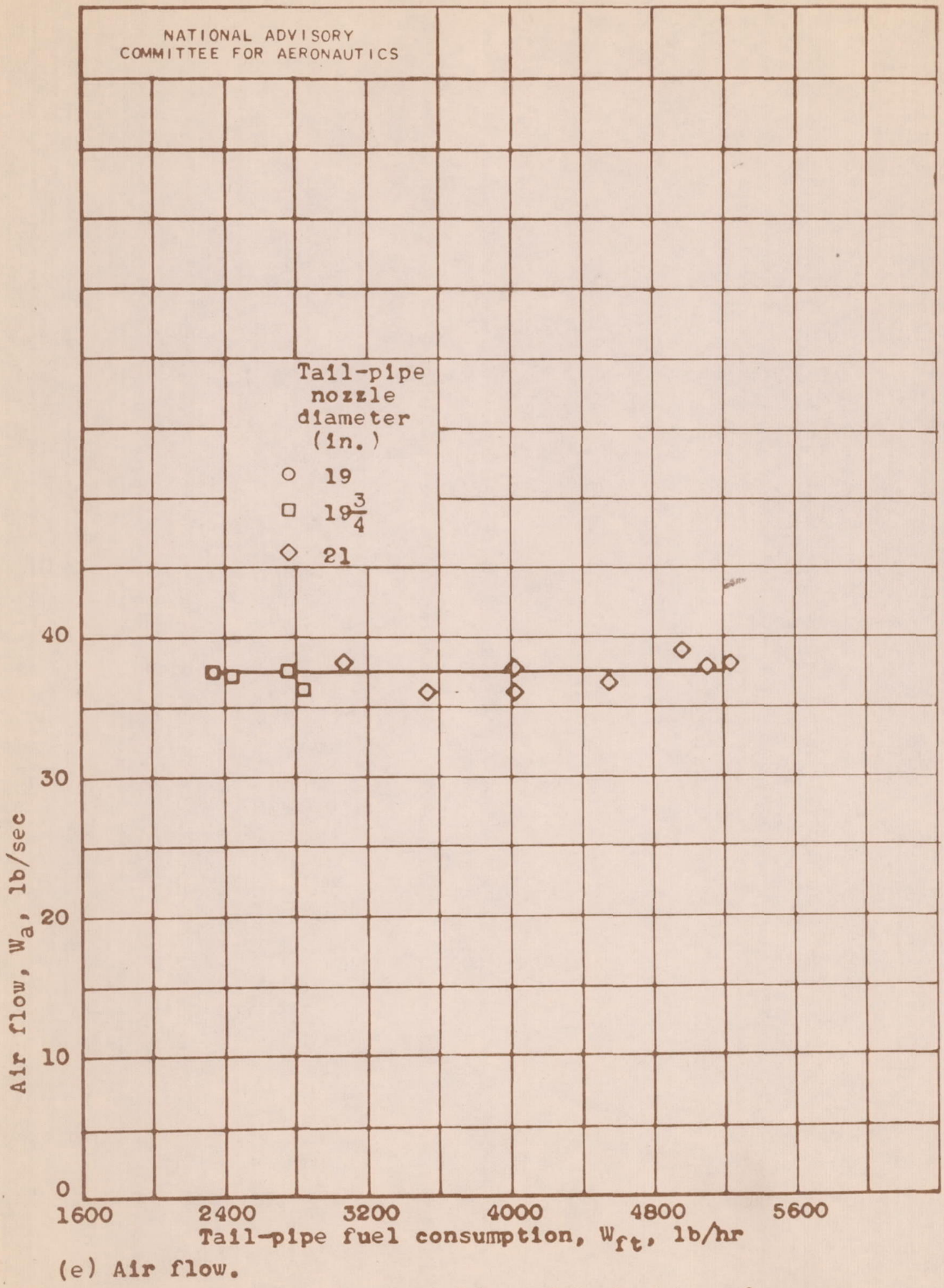
(c) Engine fuel consumption.

Figure 6.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for three tail-pipe nozzle diameters. Engine speed, 7600 rpm; altitude, 30,000 feet; ram-pressure ratio, 1.66.



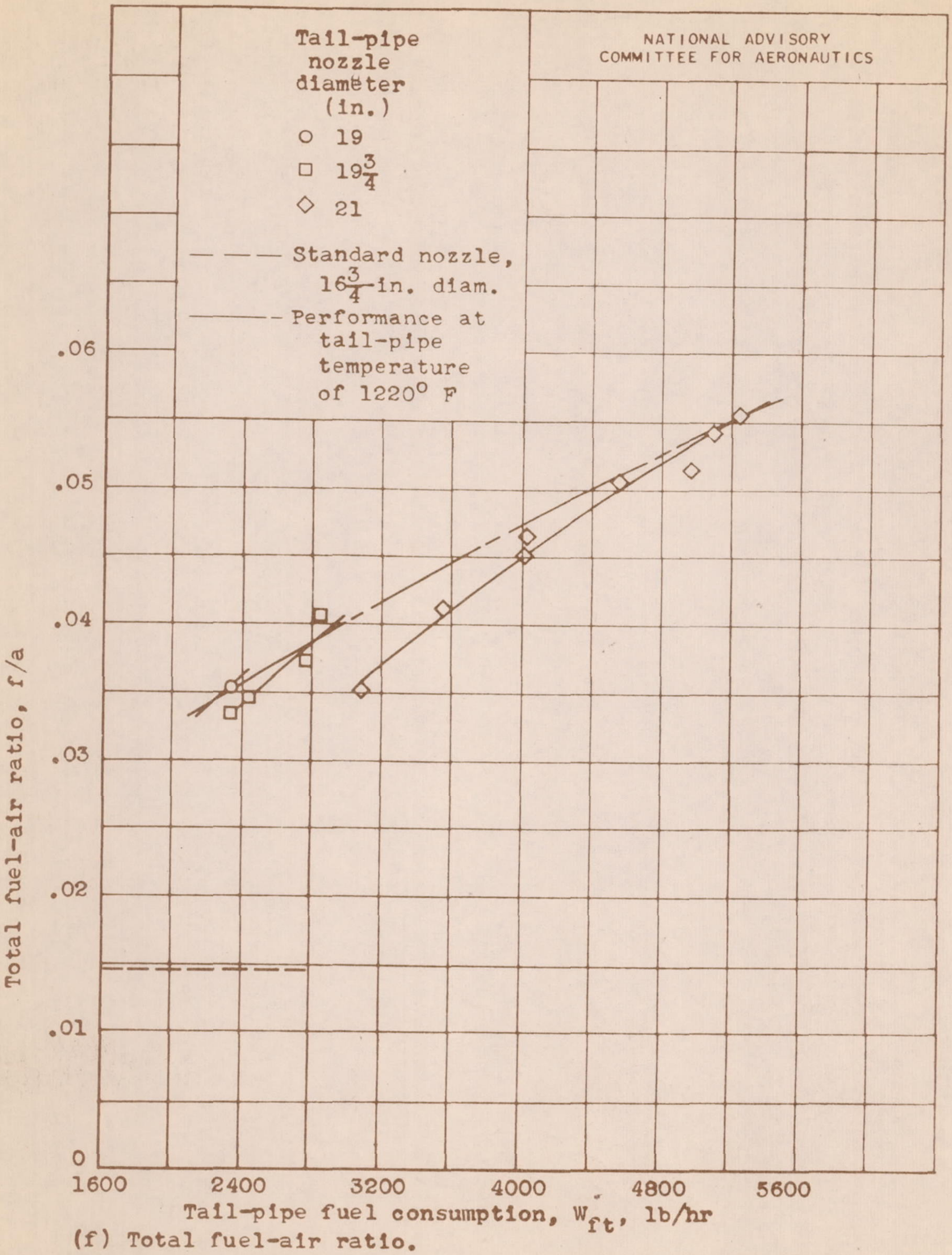
(d) Specific fuel consumption based on net thrust.

Figure 6.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for three tail-pipe nozzle diameters. Engine speed, 7600 rpm; altitude, 30,000 feet; ram-pressure ratio, 1.66.



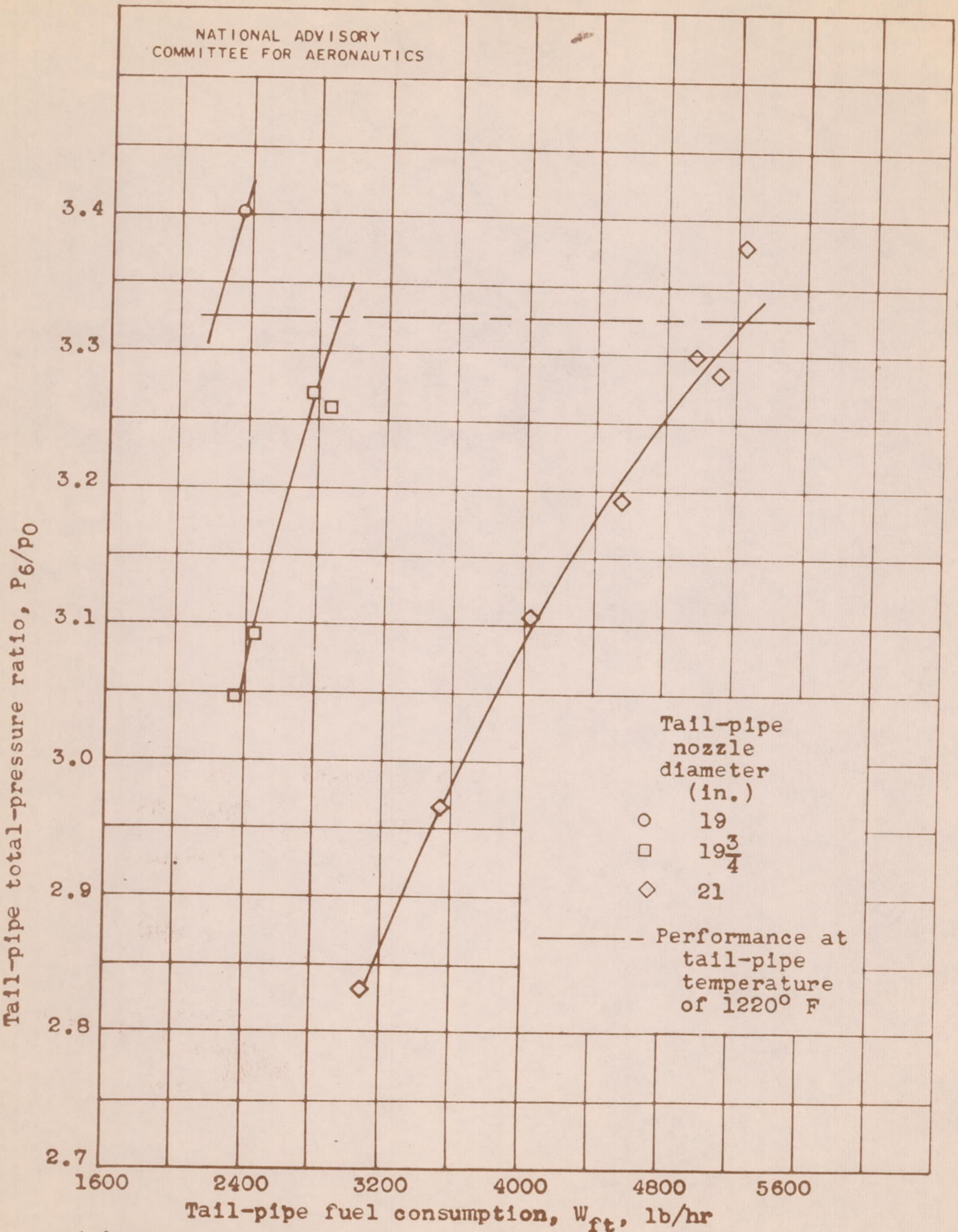
(e) Air flow.

Figure 6.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for three tail-pipe nozzle diameters. Engine speed, 7600 rpm; altitude, 30,000 feet; ram-pressure ratio, 1.66.



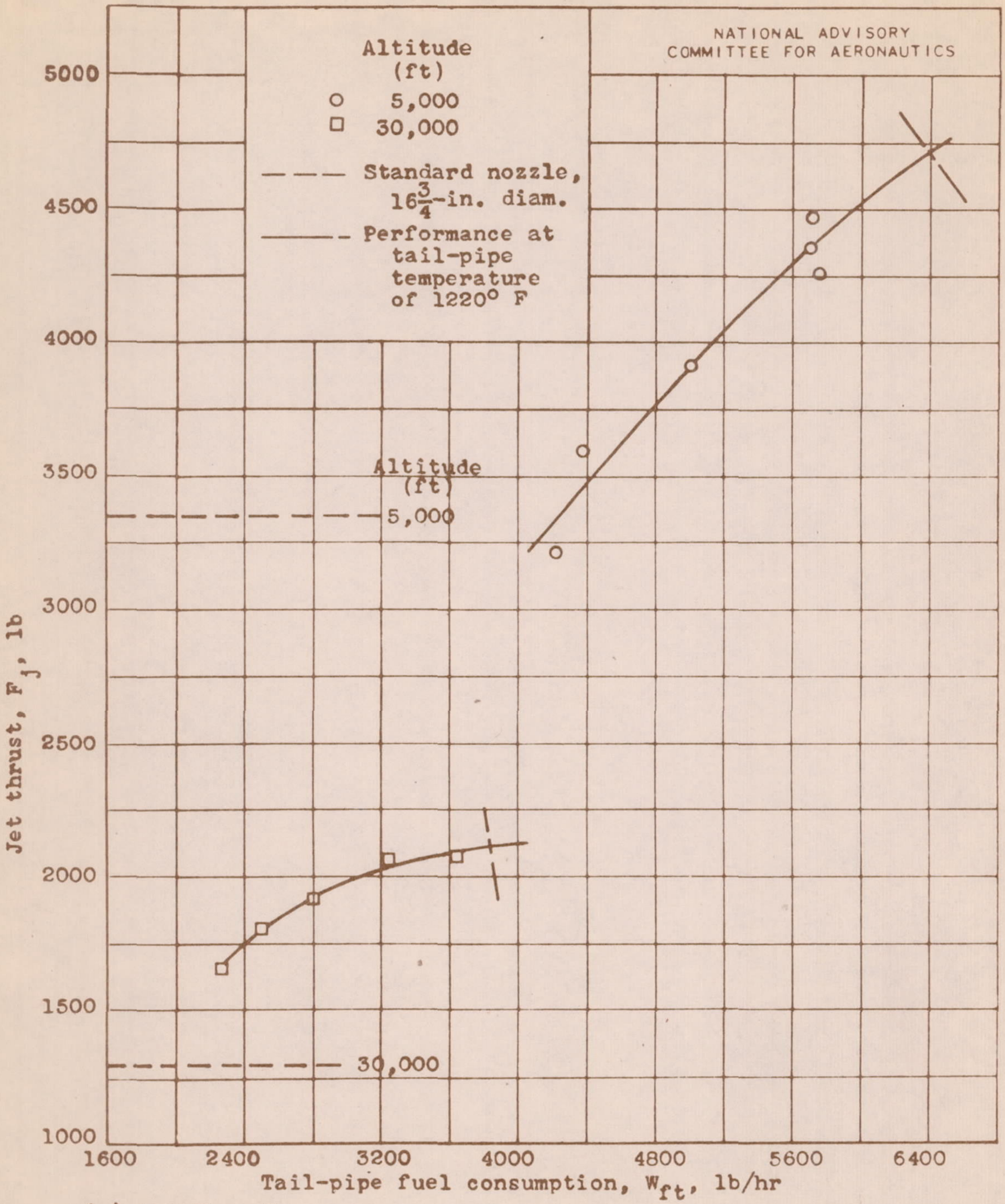
(f) Total fuel-air ratio.

Figure 6.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for three tail-pipe nozzle diameters. Engine speed, 7600 rpm; altitude, 30,000 feet; ram-pressure ratio, 1.66.



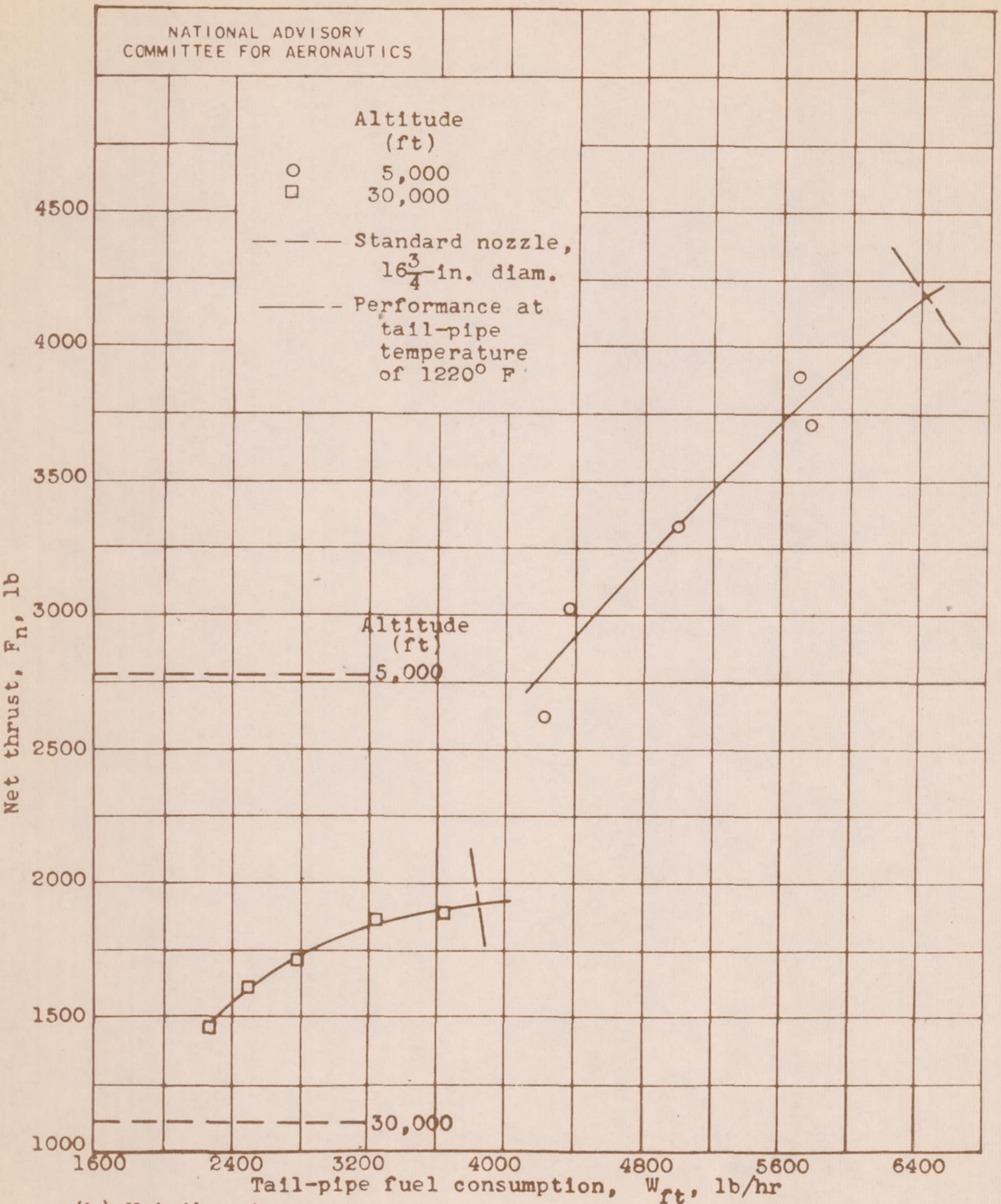
(g) Tail-pipe total-pressure ratio.

Figure 6.- Concluded. Variation of turbojet engine performance with tail-pipe fuel consumption for three tail-pipe nozzle diameters. Engine speed, 7600 rpm; altitude, 30,000 feet; ram-pressure ratio, 1.66.



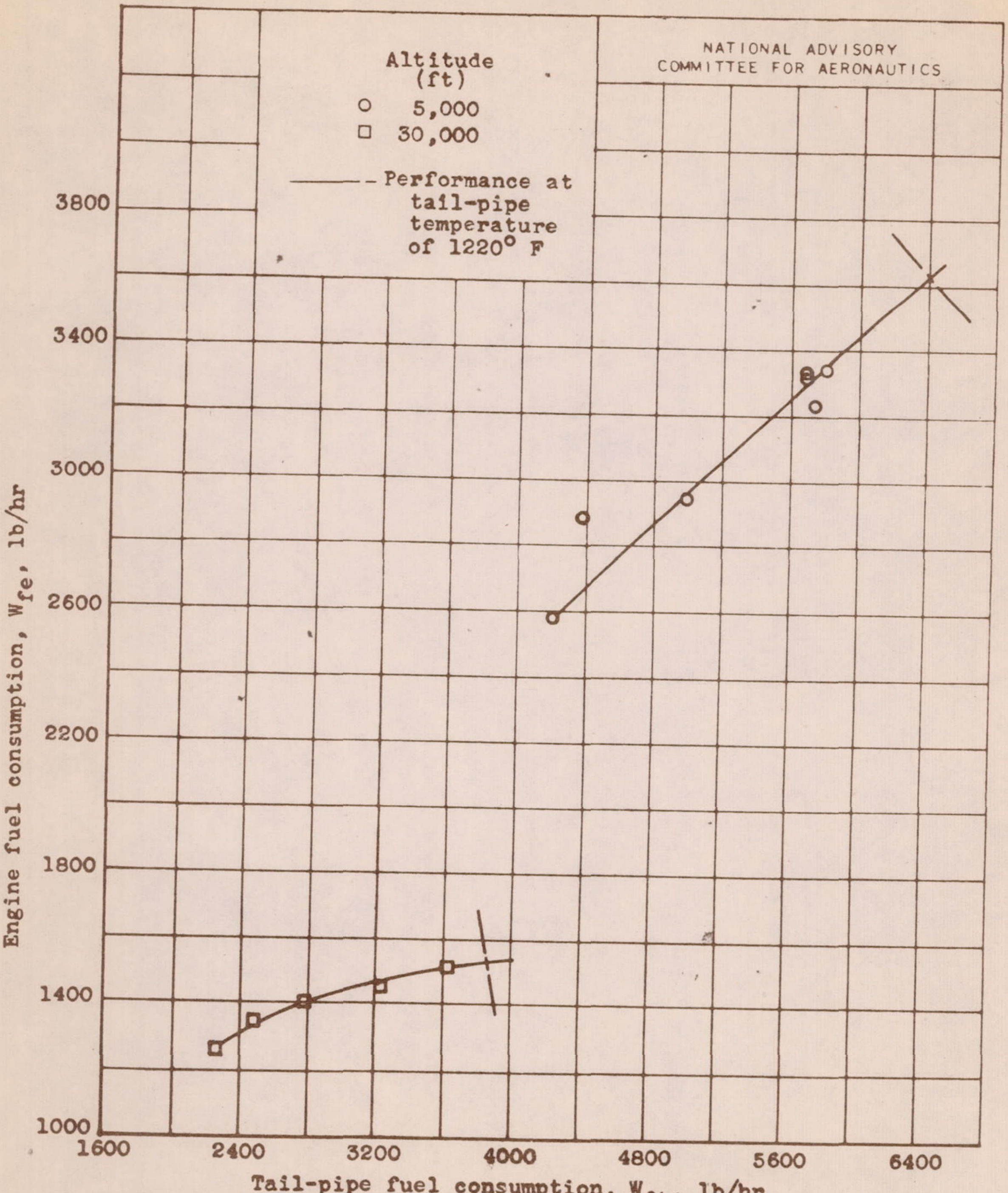
(a) Jet thrust.

Figure 7.- Variation of turbojet engine performance with tail-pipe fuel consumption for two altitudes. Engine speed, 7600 rpm; ram-pressure ratio, 1.045; 21-inch-diameter tail-pipe nozzle.



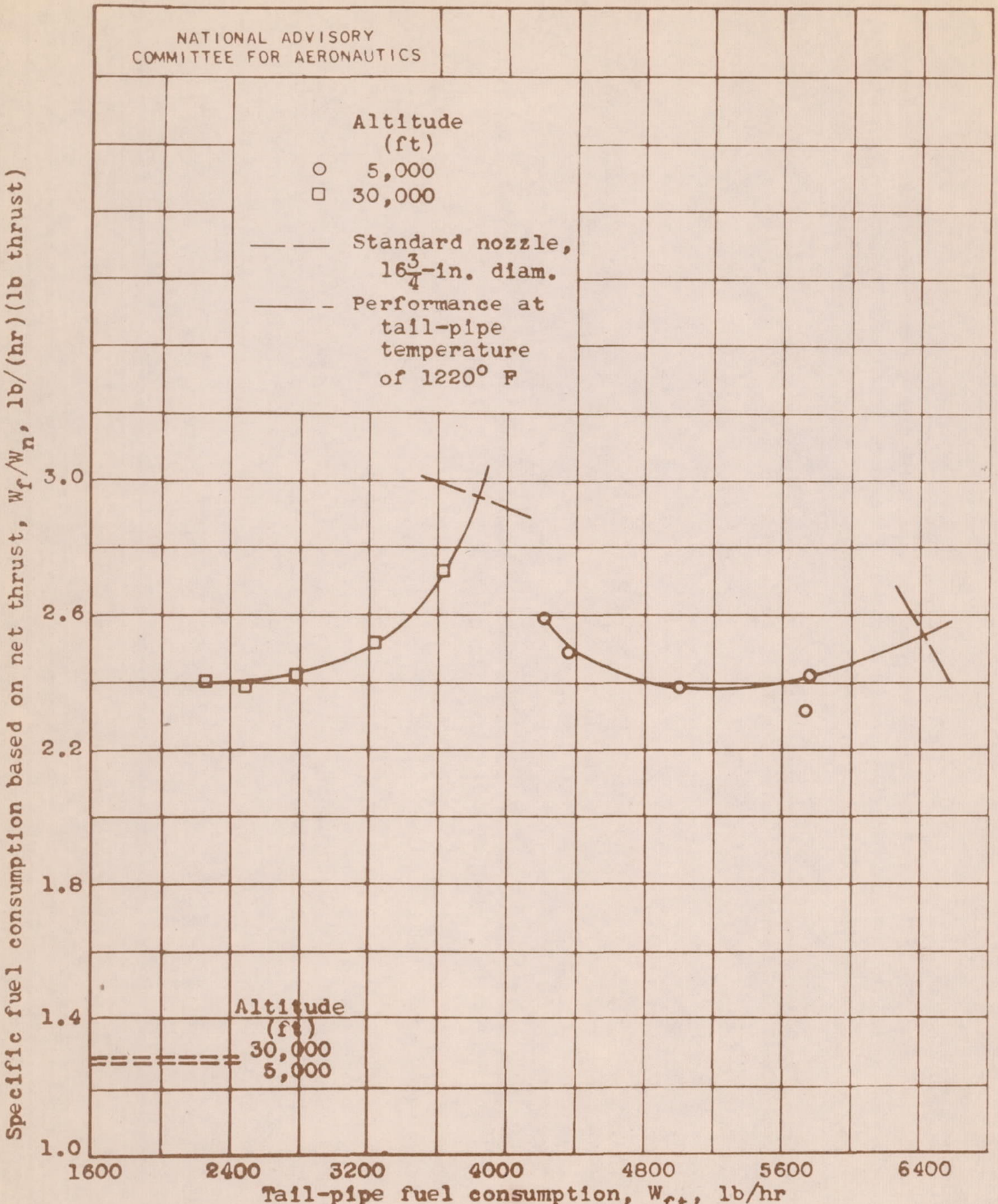
(b) Net thrust.

Figure 7.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for two altitudes. Engine speed, 7600 rpm; ram-pressure ratio, 1.045; 21-inch-diameter tail-pipe nozzle.



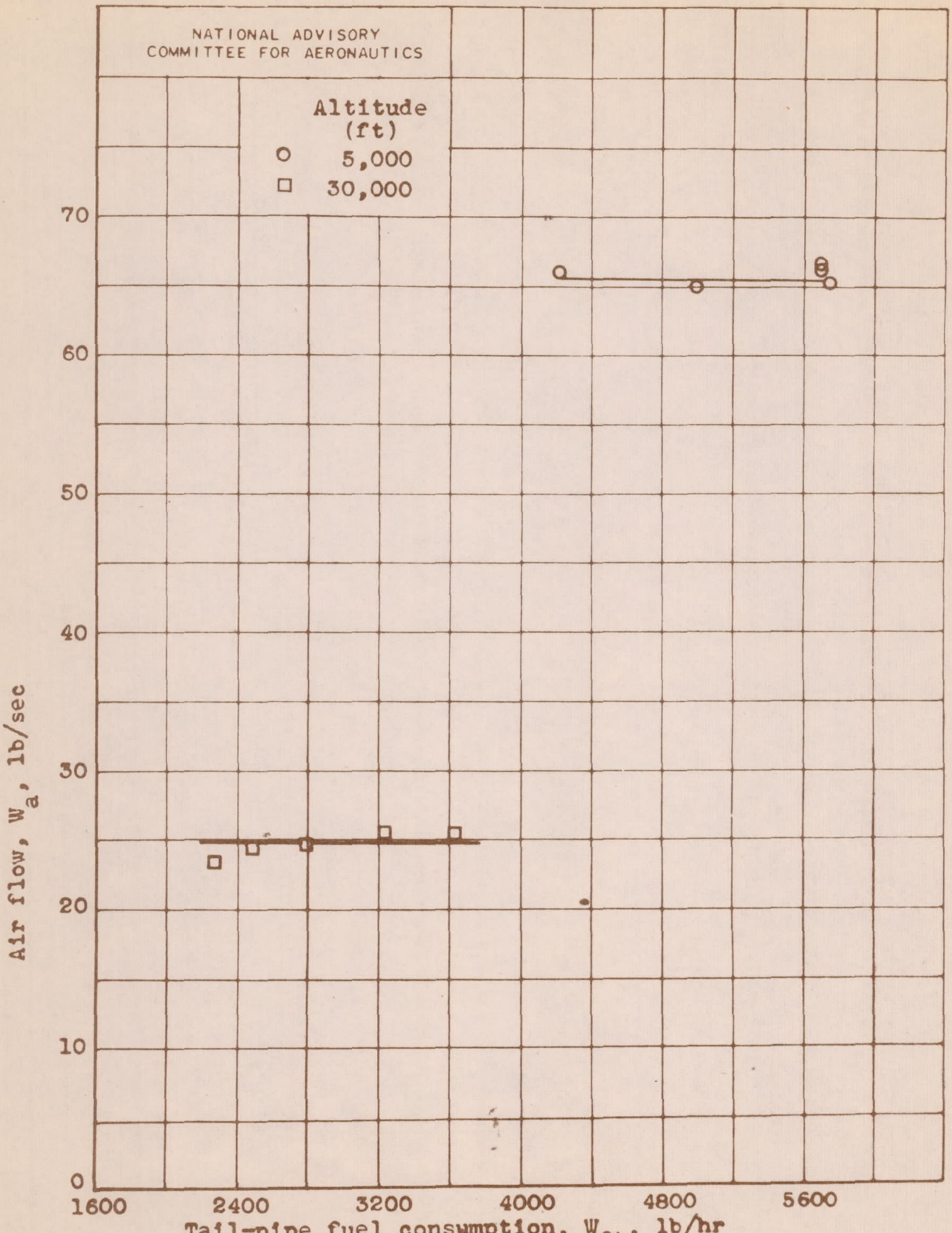
(c) Engine fuel consumption.

Figure 7.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for two altitudes. Engine speed, 7600 rpm; ram-pressure ratio, 1.045; 21-inch-diameter tail-pipe nozzle.



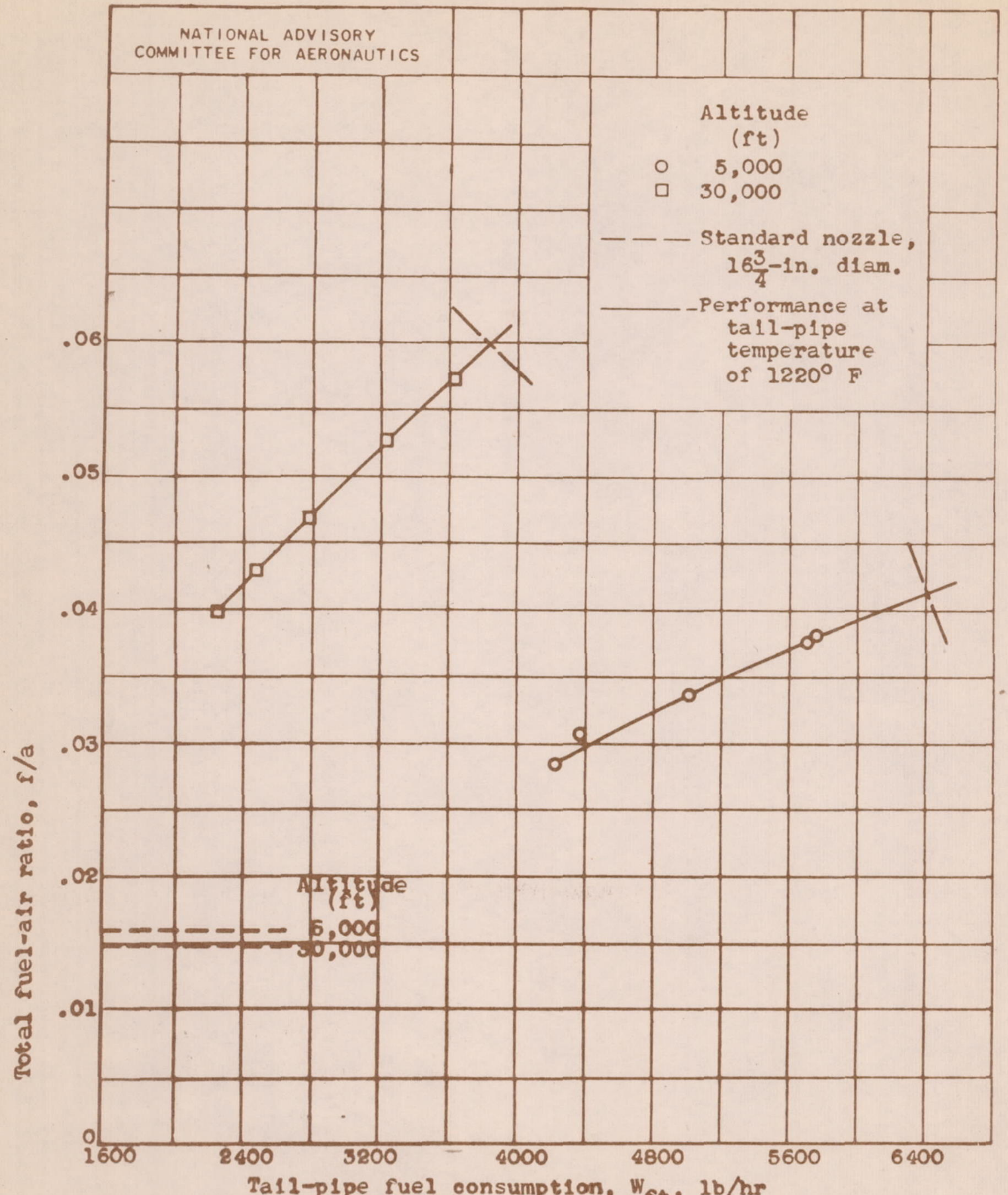
(d) Specific fuel consumption based on net thrust.

Figure 7.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for two altitudes. Engine speed, 7600 rpm; ram-pressure ratio, 1.045; 21-inch-diameter tail-pipe nozzle.



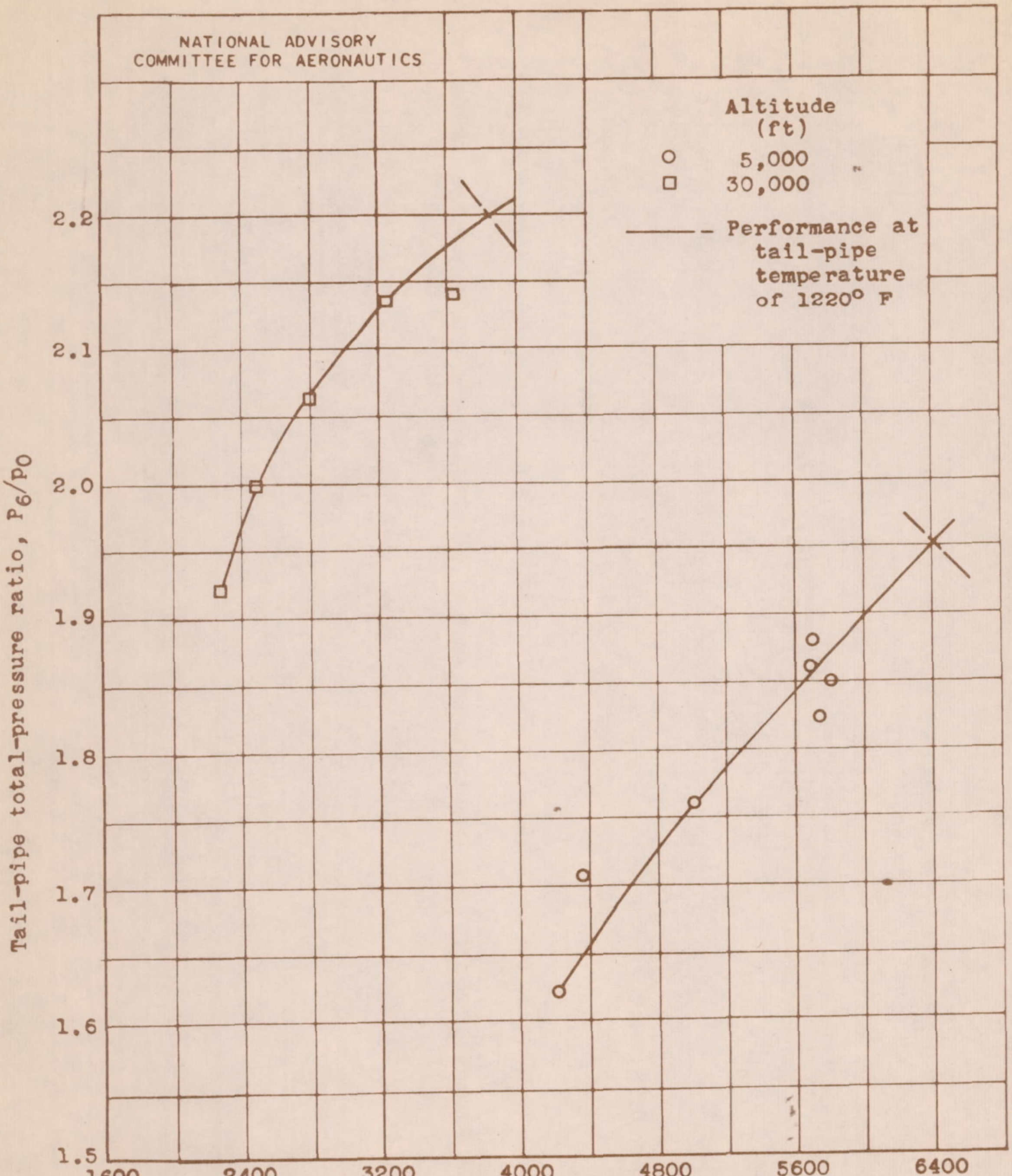
(e) Air flow.

Figure 7.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for two altitudes. Engine speed, 7600 rpm; ram-pressure ratio, 1.045; 21-inch-diameter tail-pipe nozzle.



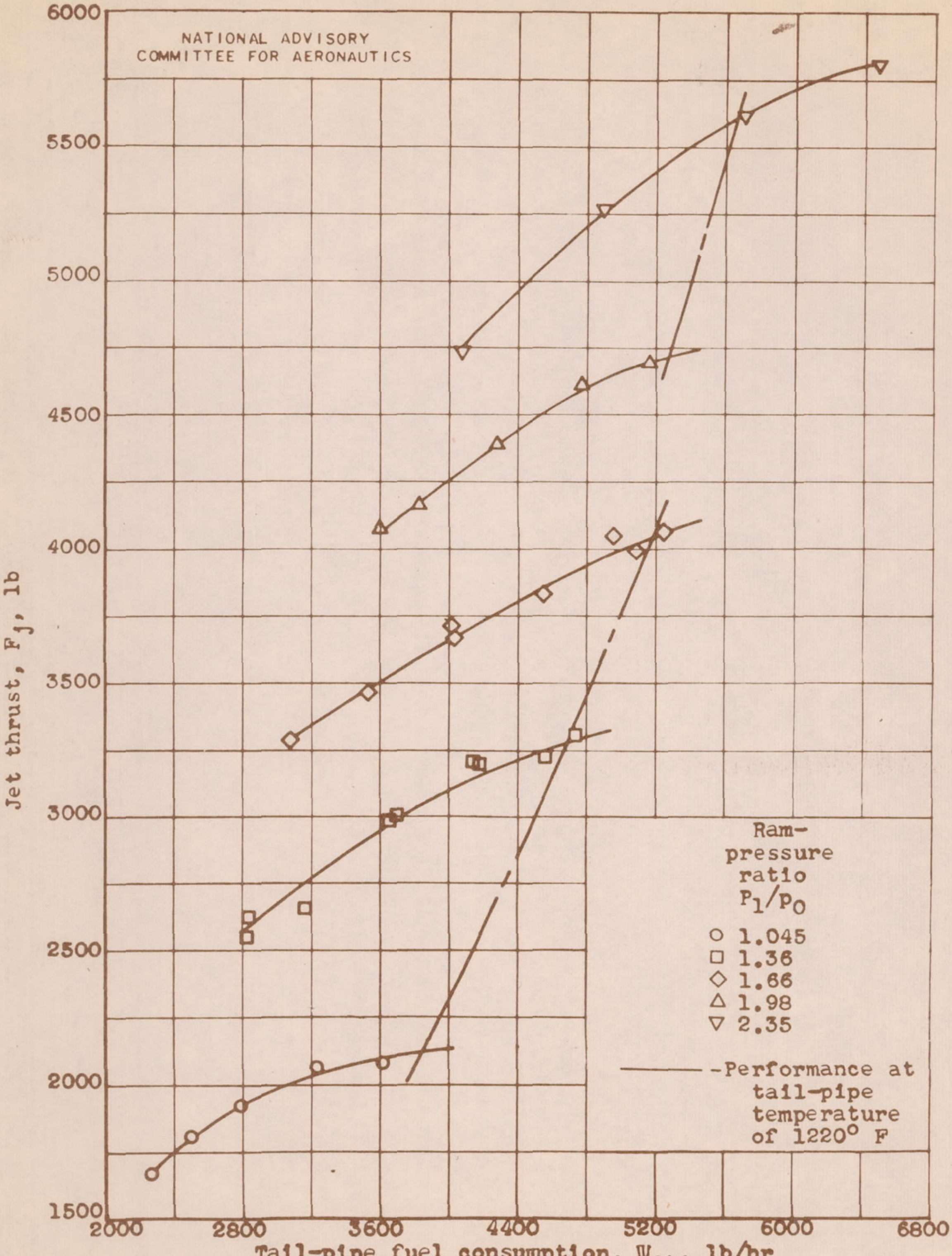
(f) Total fuel-air ratio.

Figure 7.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for two altitudes. Engine speed, 7600 rpm; ram-pressure ratio, 1.045; 21-inch-diameter tail-pipe nozzle.



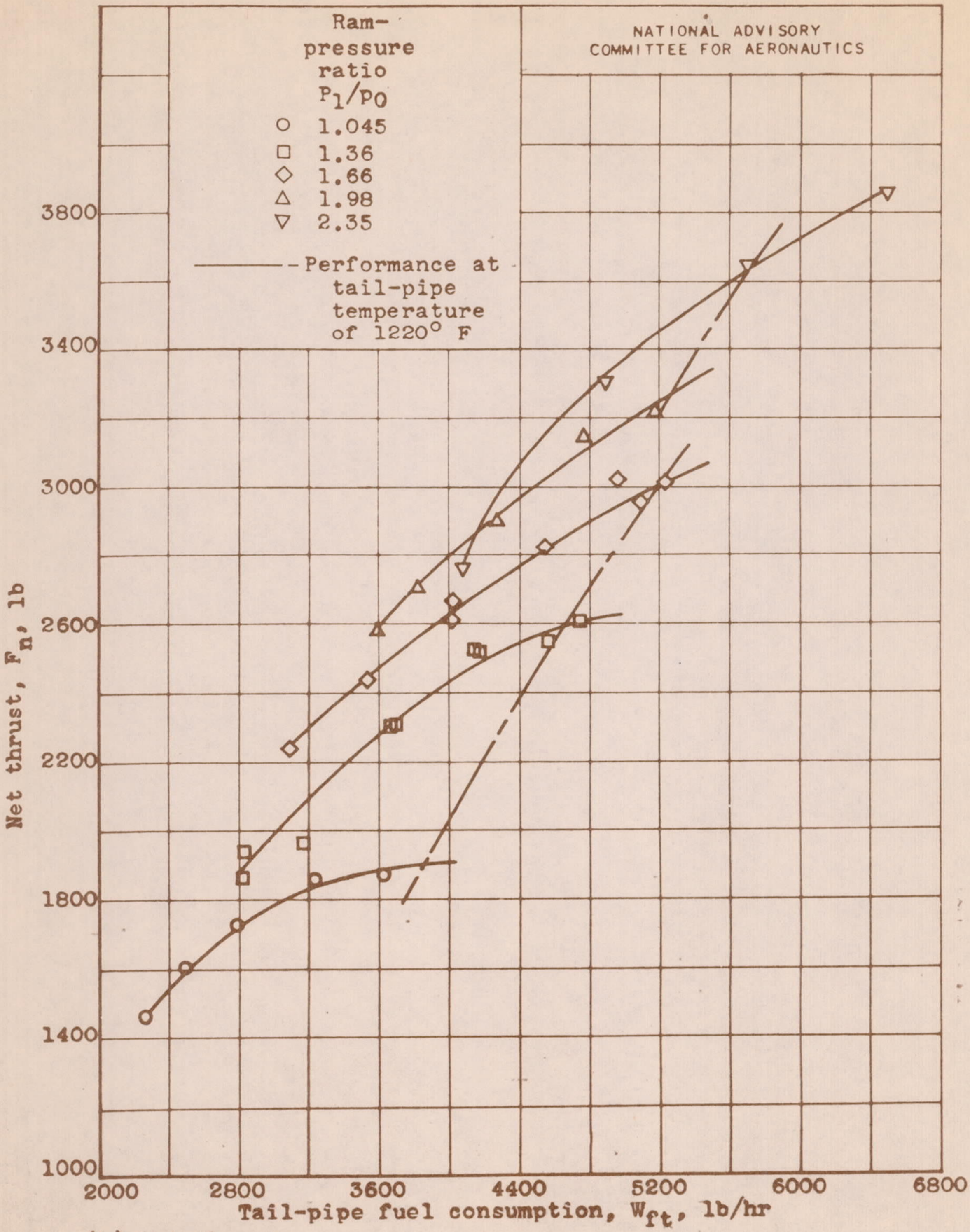
(g) Tail-pipe total-pressure ratio.

Figure 7.- Concluded. Variation of turbojet engine performance with tail-pipe fuel consumption for two altitudes. Engine speed, 7600 rpm; ram-pressure ratio, 1.045; 21-inch-diameter tail-pipe nozzle.



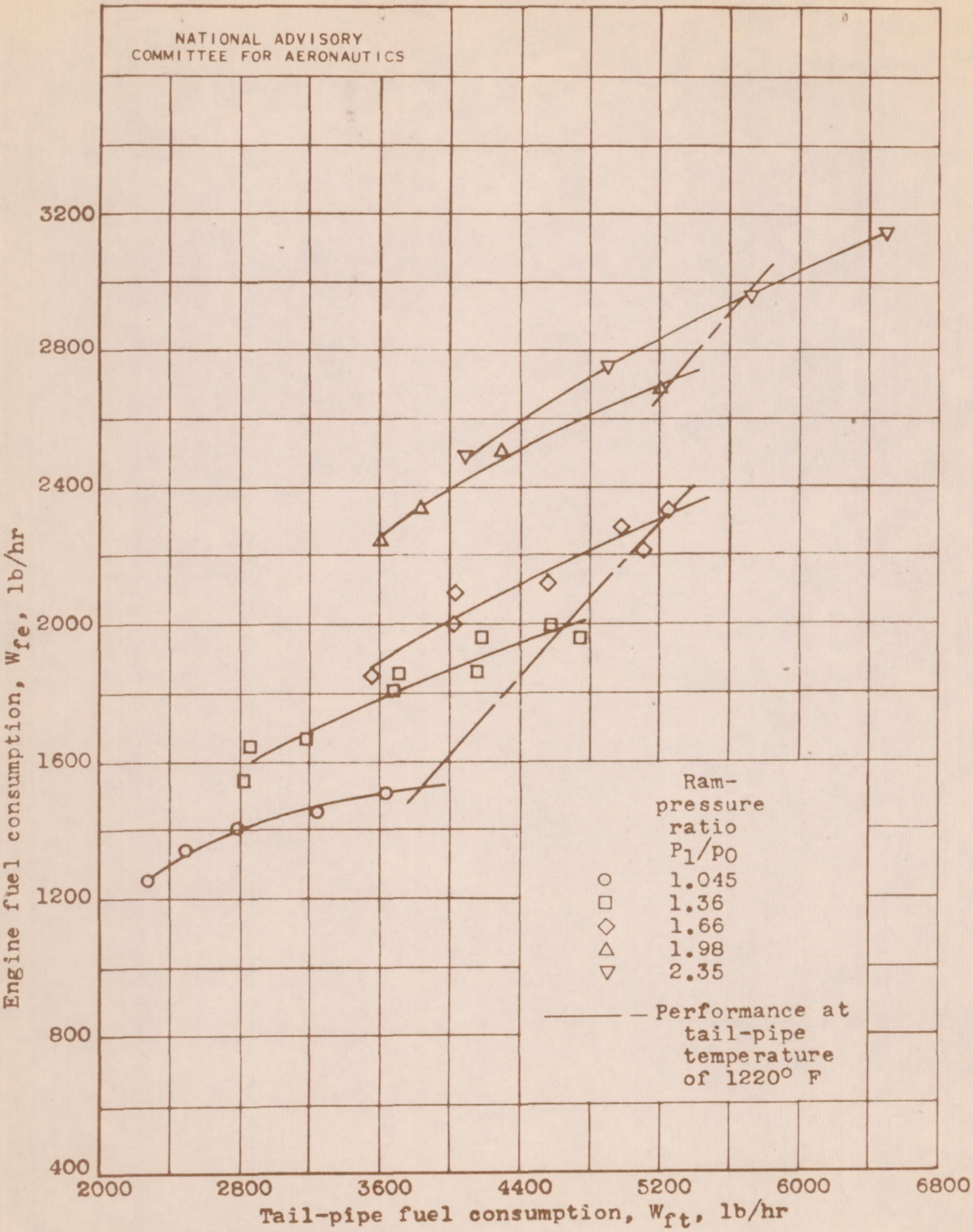
(a) Jet thrust.

Figure 8.- Variation of turbojet engine performance with tail-pipe fuel consumption for five ram-pressure ratios. Altitude, 30,000 feet; 21-inch-diameter tail-pipe nozzle.



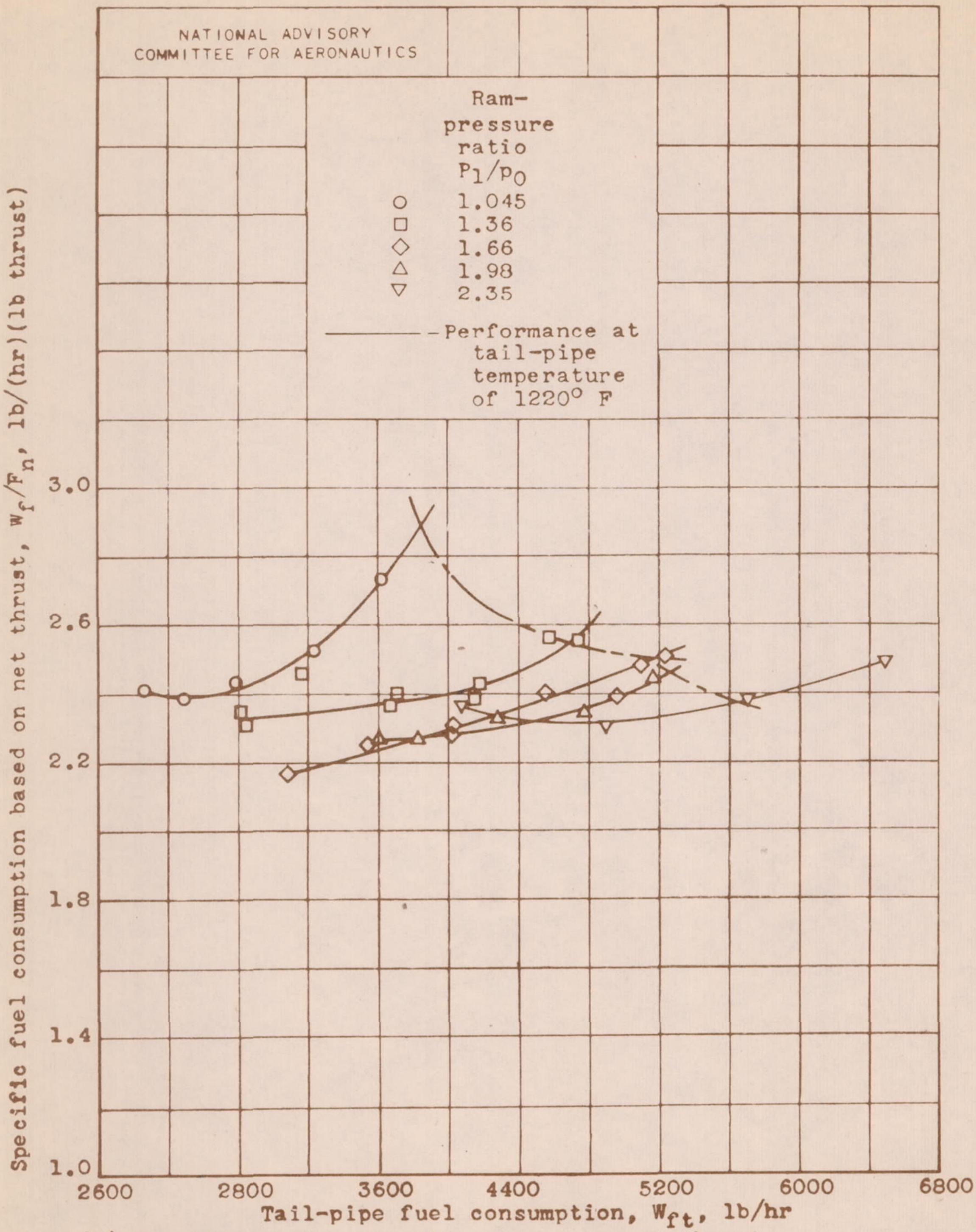
(b) Net thrust.

Figure 8.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for five ram-pressure ratios. Altitude, 30,000 feet; 21-inch-diameter tail-pipe nozzle.



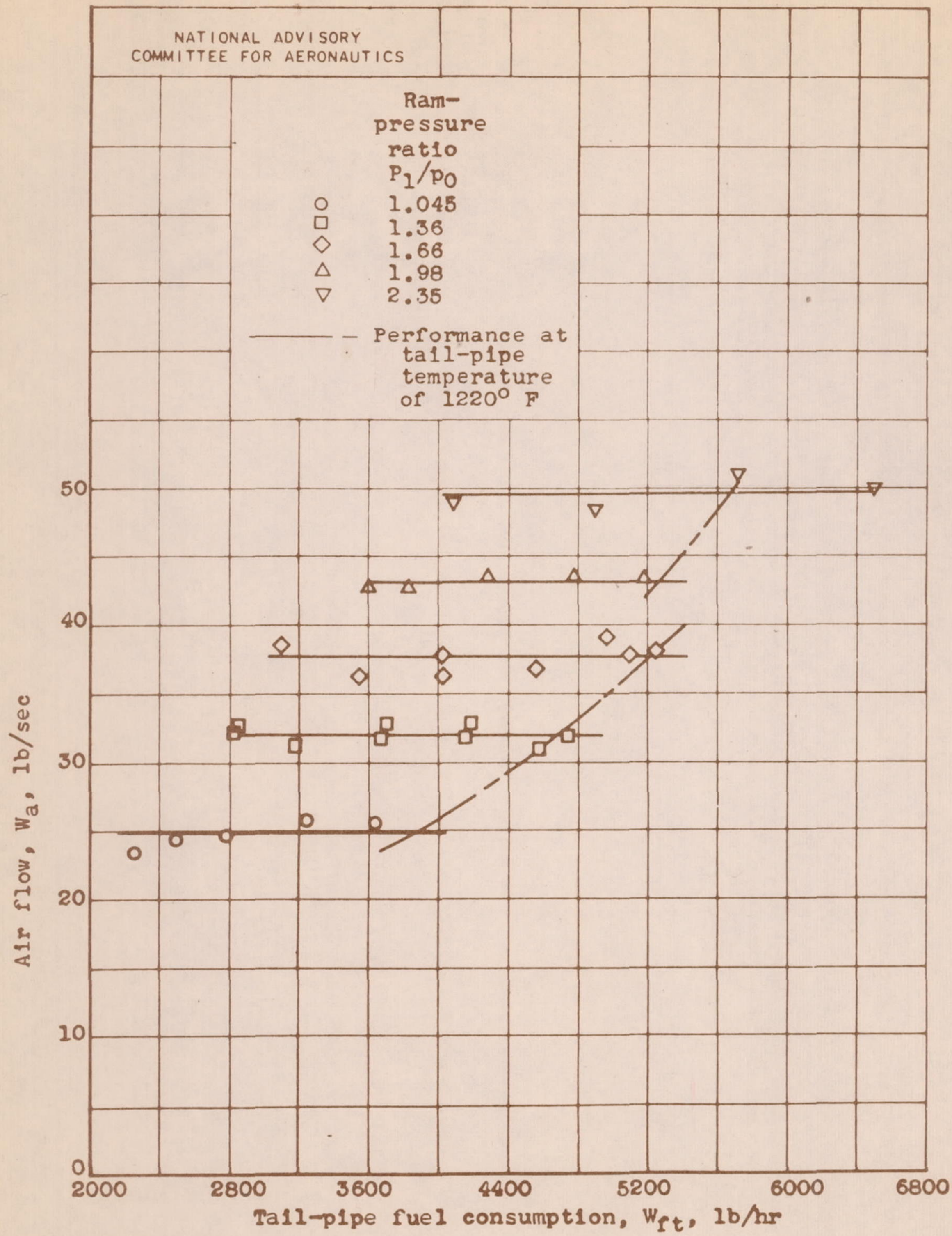
(c) Engine fuel consumption.

Figure 8.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for five ram-pressure ratios. Altitude, 30,000 feet; 21-inch-diameter tail-pipe nozzle.



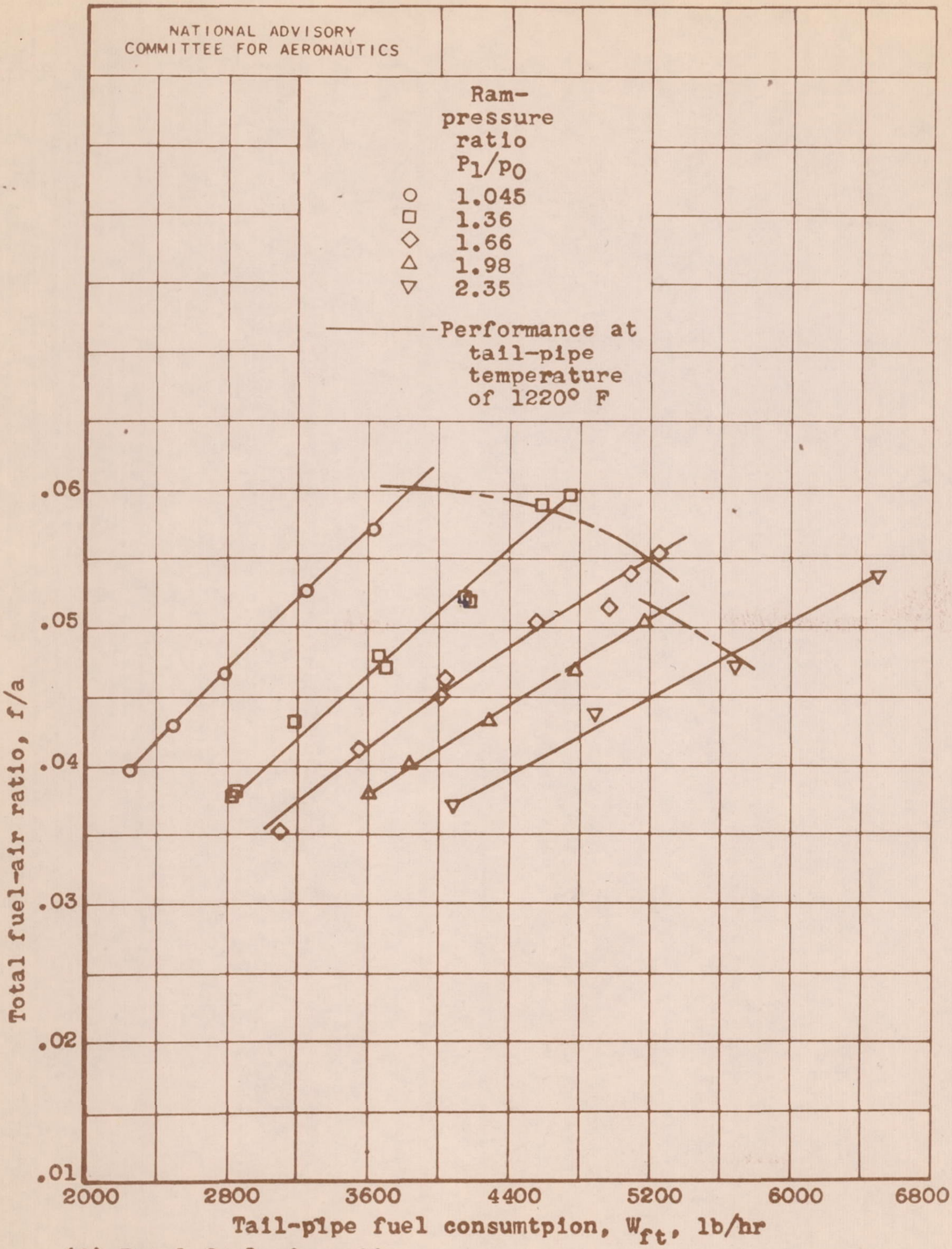
(d) Specific fuel consumption based on net thrust.

Figure 8.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for five ram-pressure ratios. Altitude, 30,000 feet; 21-inch-diameter tail-pipe nozzle.



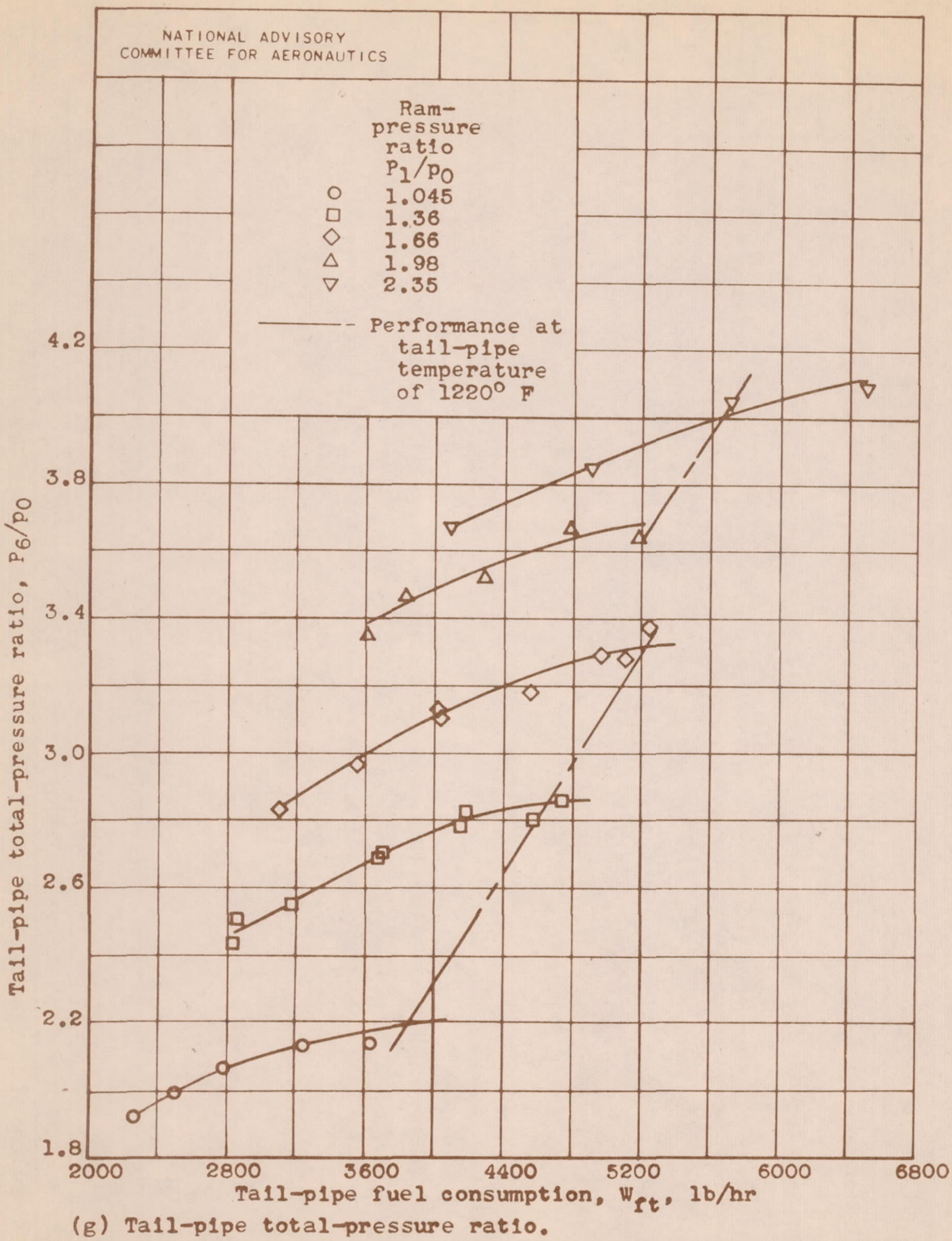
(e) Air flow.

Figure 8.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for five ram-pressure ratios. Altitude, 30,000 feet; 21-inch-diameter tail-pipe nozzle.



(f) Total fuel-air ratio.

Figure 8.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for five ram-pressure ratios. Altitude, 30,000 feet; 21-inch-diameter tail-pipe nozzle.



(g) Tail-pipe total-pressure ratio.

Figure 8.- Continued. Variation of turbojet engine performance with tail-pipe fuel consumption for five ram-pressure ratios. Altitude, 30,000 feet; 21-inch-diameter tail-pipe nozzle.

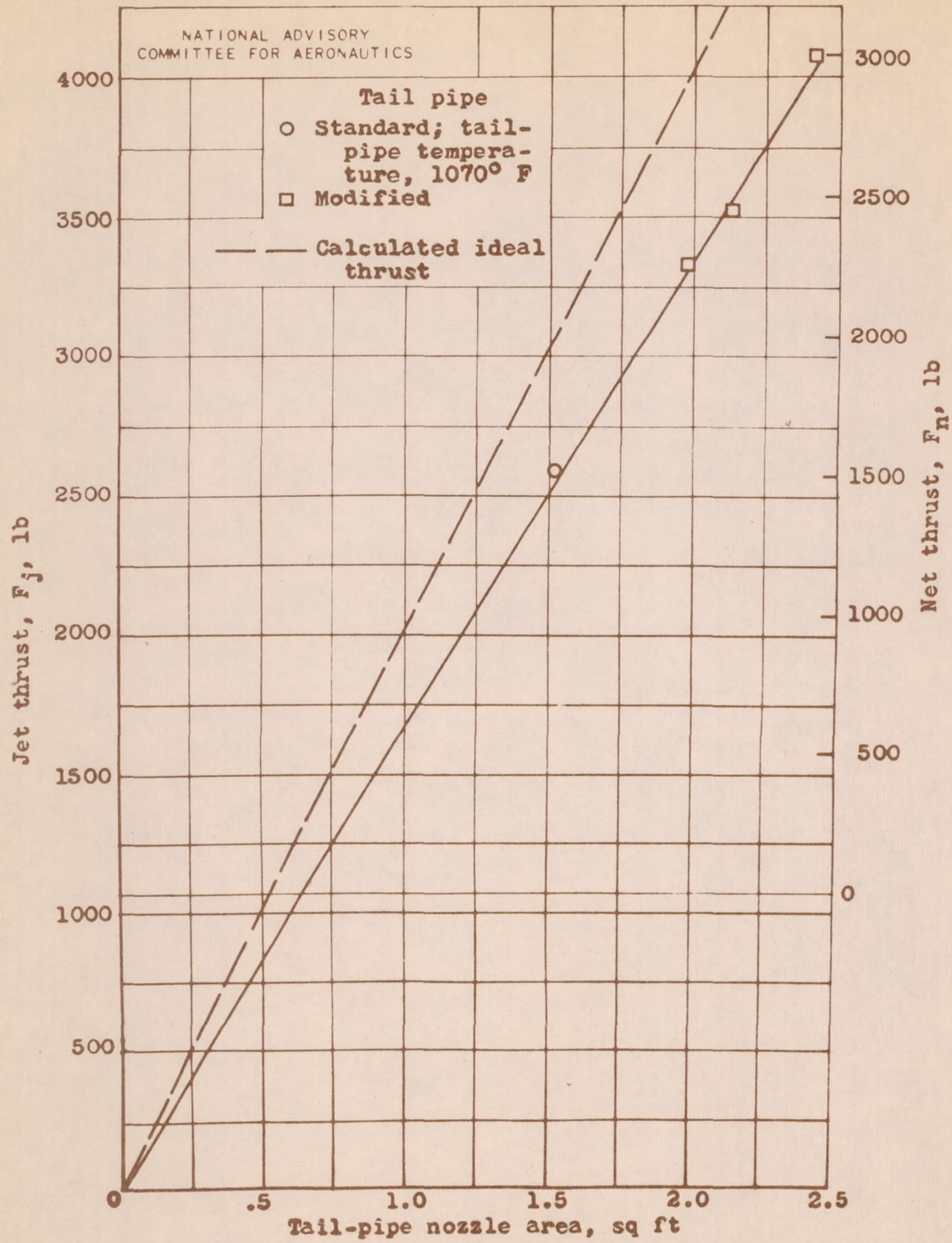


Figure 9.- Relation between nozzle area and thrust of turbojet engine with tail-pipe burning. Engine speed, 7600 rpm; turbine-outlet temperature, 1220° F; ram-pressure ratio, 1.66; altitude, 30,000 feet. Value of γ was assumed equal to 1.27 for calculation of theoretical thrust.

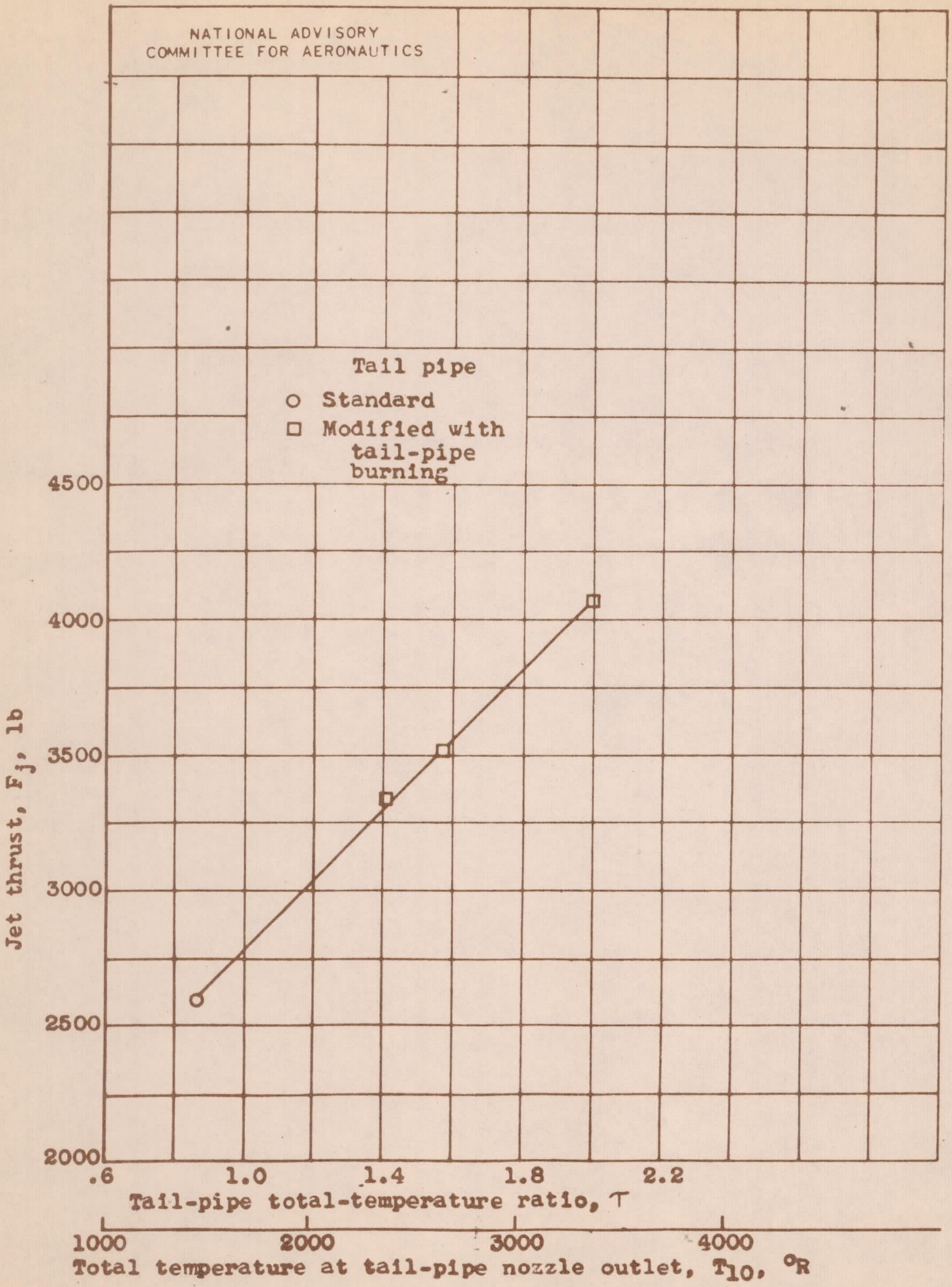


Figure 10.- Relation between jet thrust and total temperature of gases leaving tail pipe of turbojet engine with tail-pipe burning. Engine speed, 7600 rpm; turbine-outlet temperature, 1220° F; ram-pressure ratio, 1.66; altitude, 30,000 feet.

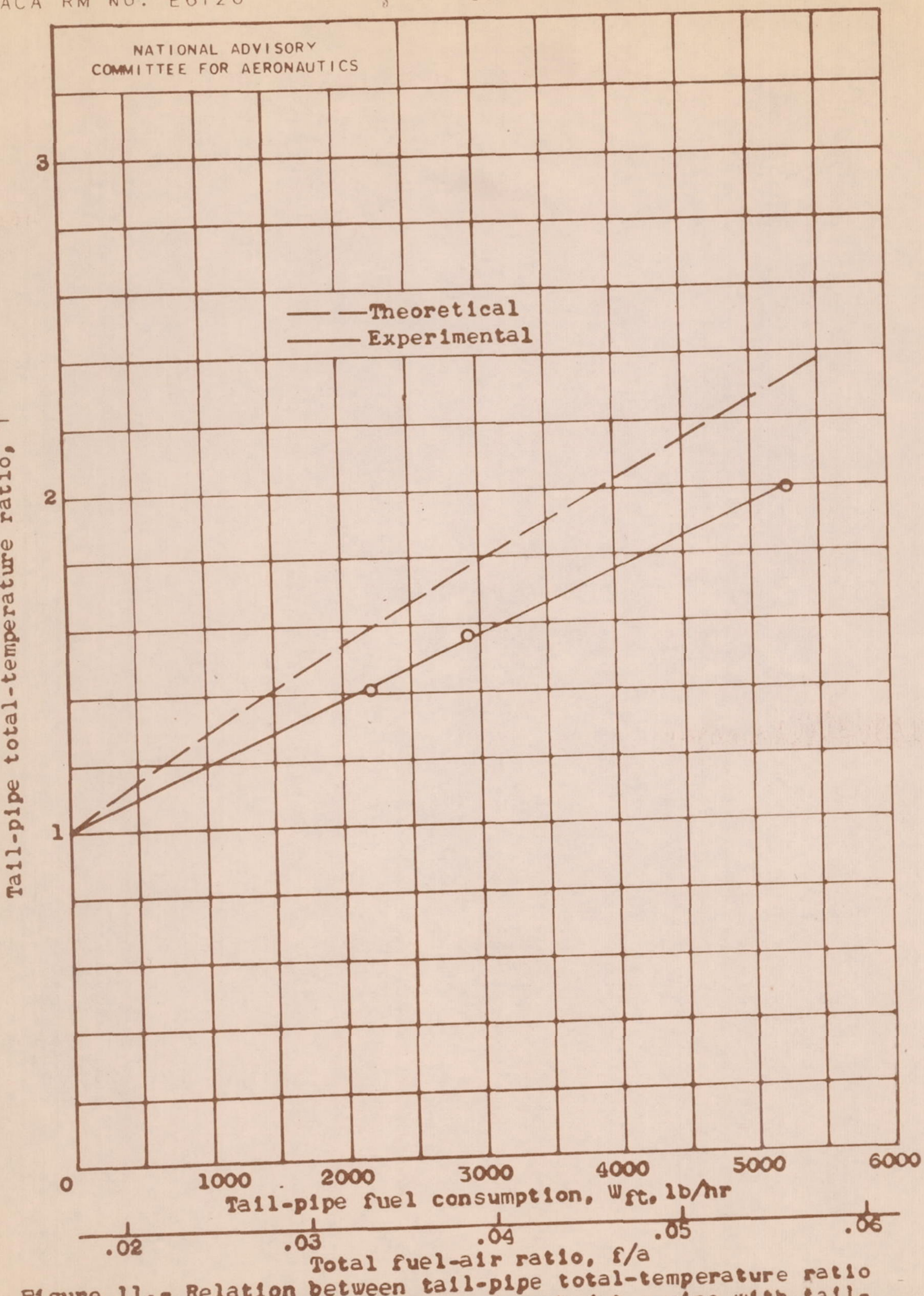


Figure 11.- Relation between tail-pipe total-temperature ratio and tail-pipe fuel consumption of turbojet engine with tail-pipe burning. Engine speed, 7600 rpm; ram-pressure ratio, 1.66; turbine-outlet temperature, 1220° F; altitude, 30,000 feet.

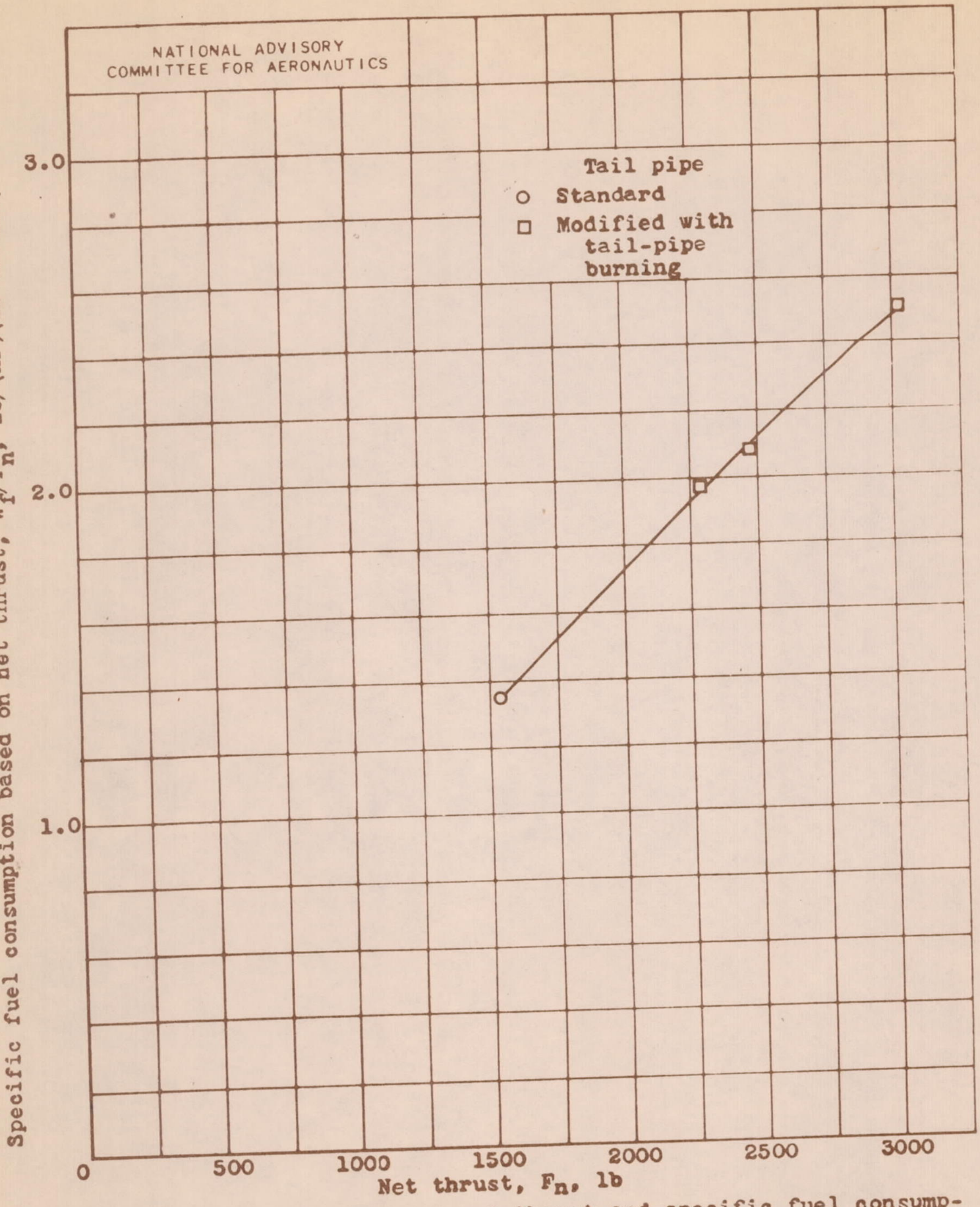


Figure 12.- Relation between net thrust and specific fuel consumption based on net thrust of turbojet engine with standard tail pipe and with modified tail pipe with tail-pipe burning. Engine speed, 7600 rpm; turbine-outlet temperature, 1220° F; ram-pressure ratio, 1.66; altitude, 30,000 feet.

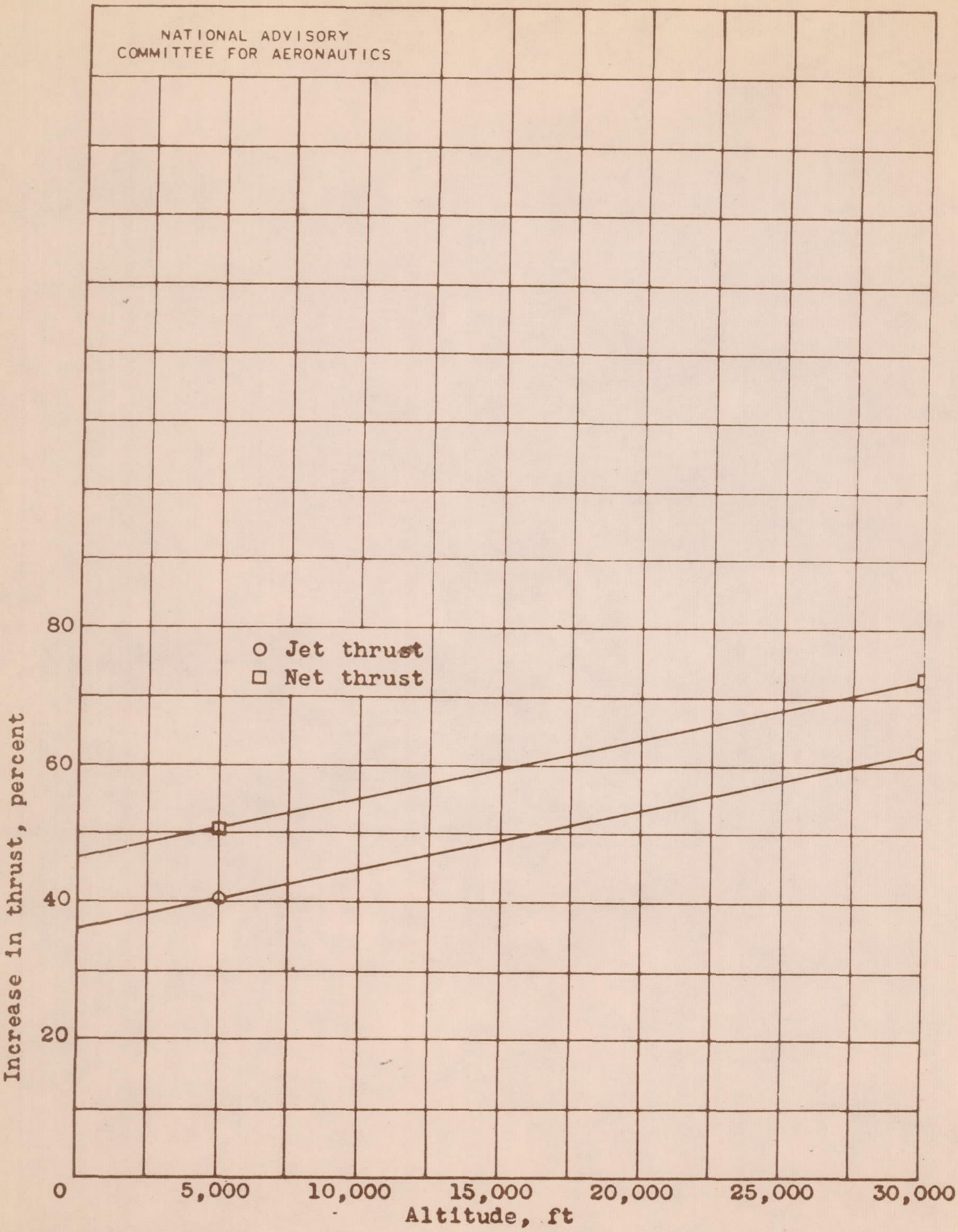


Figure 13.- Effect of altitude on percentage increase in thrust from tail-pipe burning with turbojet engine. Engine speed, 7600 rpm; turbine-outlet temperature, 1220° F; ram-pressure ratio, 1.045; 21-inch-diameter tail-pipe nozzle.

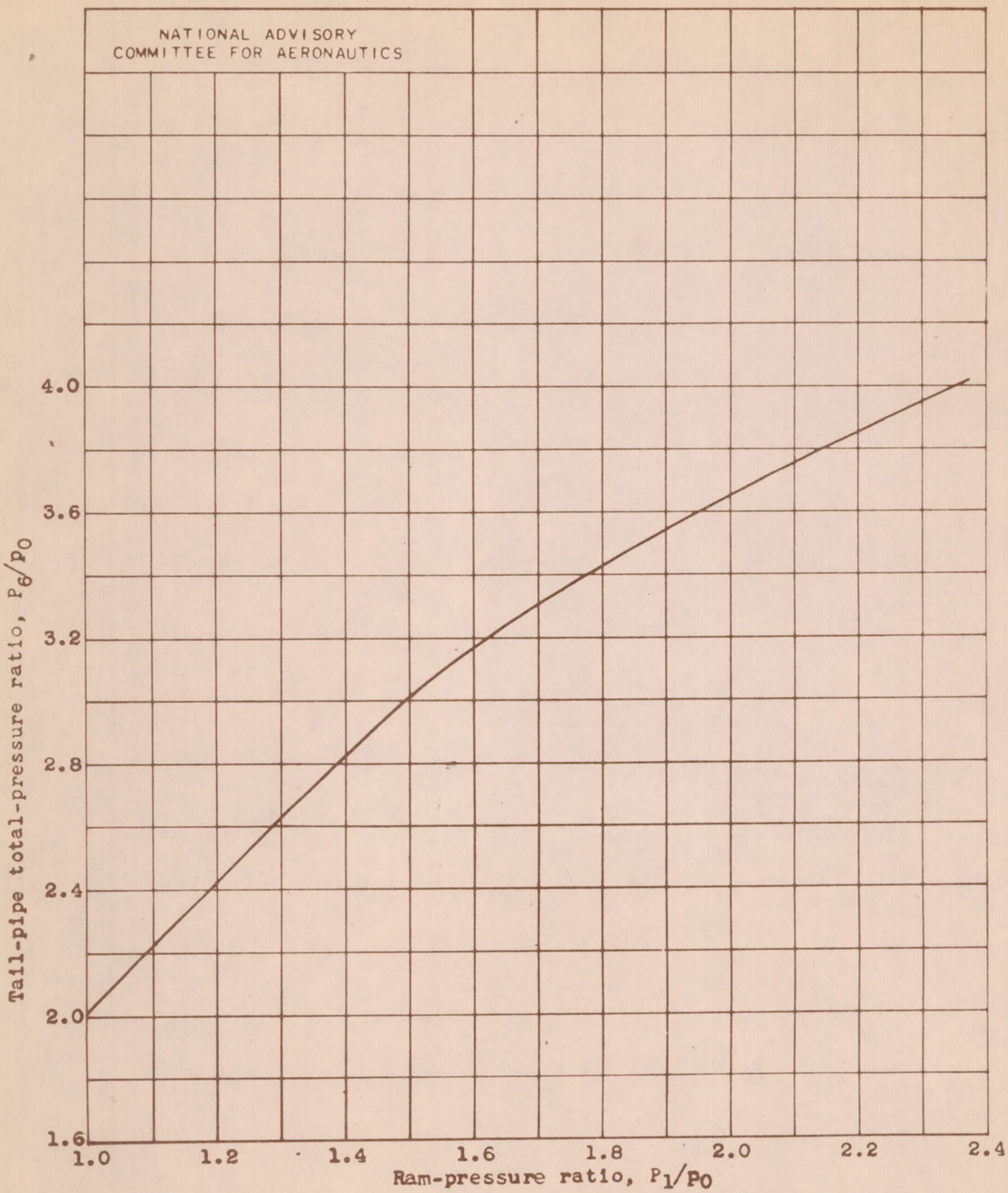


Figure 14.- Relation between tail-pipe total-pressure ratio and ram-pressure ratio for turbojet engine with tail-pipe burning. Engine speed, 7600 rpm; turbine-outlet temperature, 1220° F; altitude, 30,000 feet.

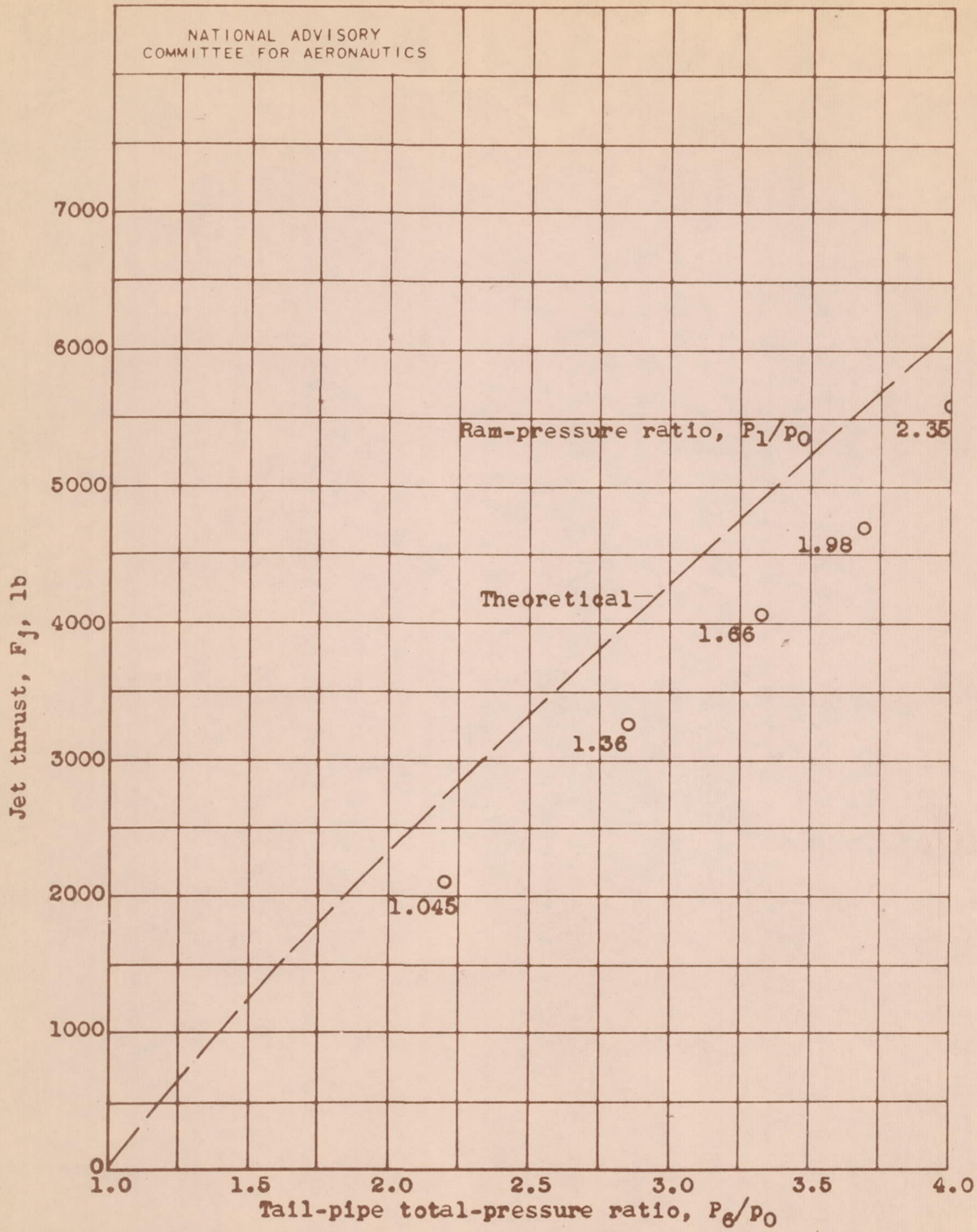


Figure 15.- Effect of ram-pressure ratio on relation between tail-pipe total-pressure ratio and jet thrust of turbojet engine with tail-pipe burning. Engine speed, 7600 rpm; turbine-outlet temperature, 1220° F; altitude, 30,000 feet.

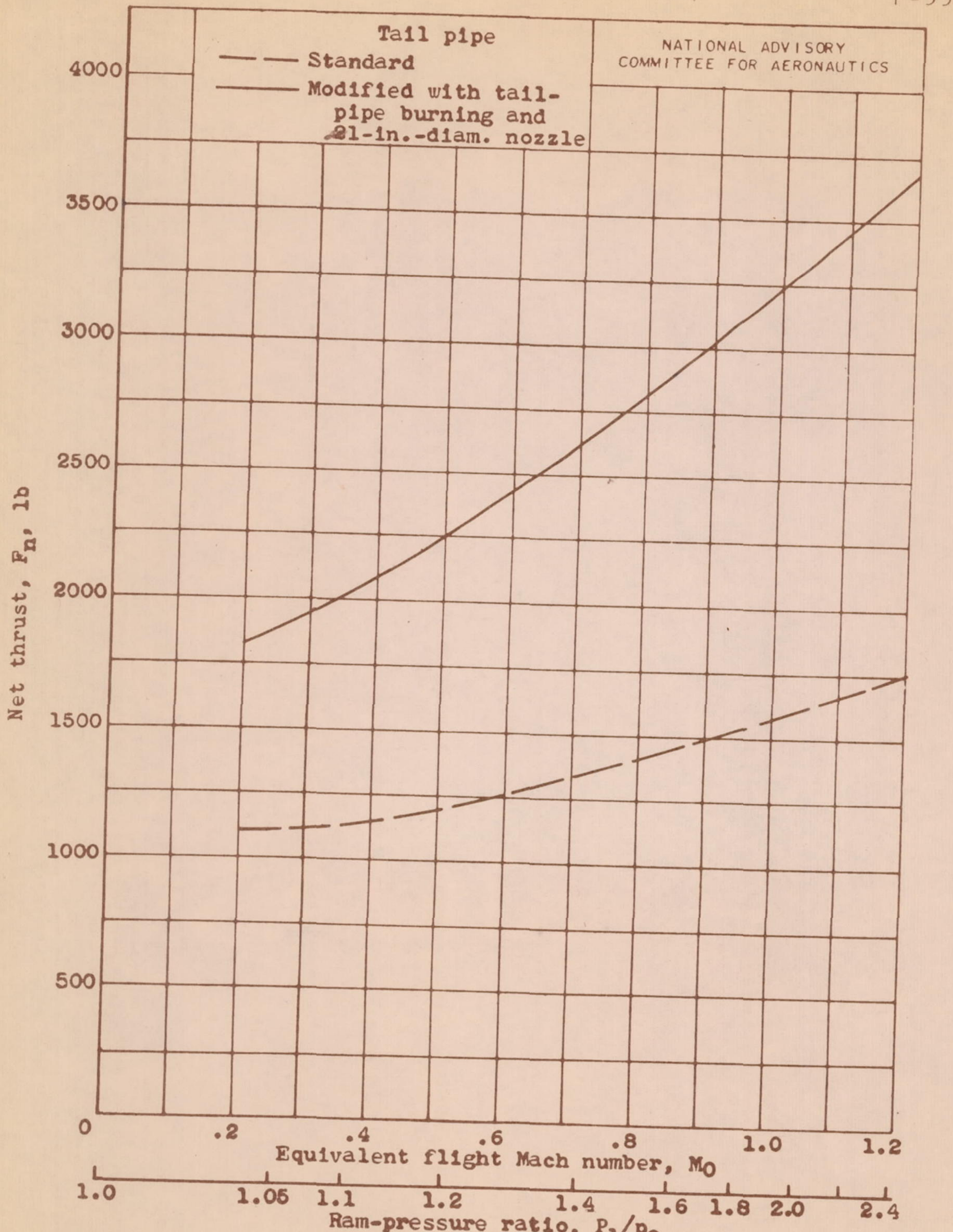


Figure 16.- Relation between equivalent flight Mach number and net thrust of turbojet engine with standard tail pipe and with modified tail pipe and tail-pipe burning. Engine speed, 7600 rpm; turbine-outlet temperature, 1220° F; altitude, 30,000 feet.

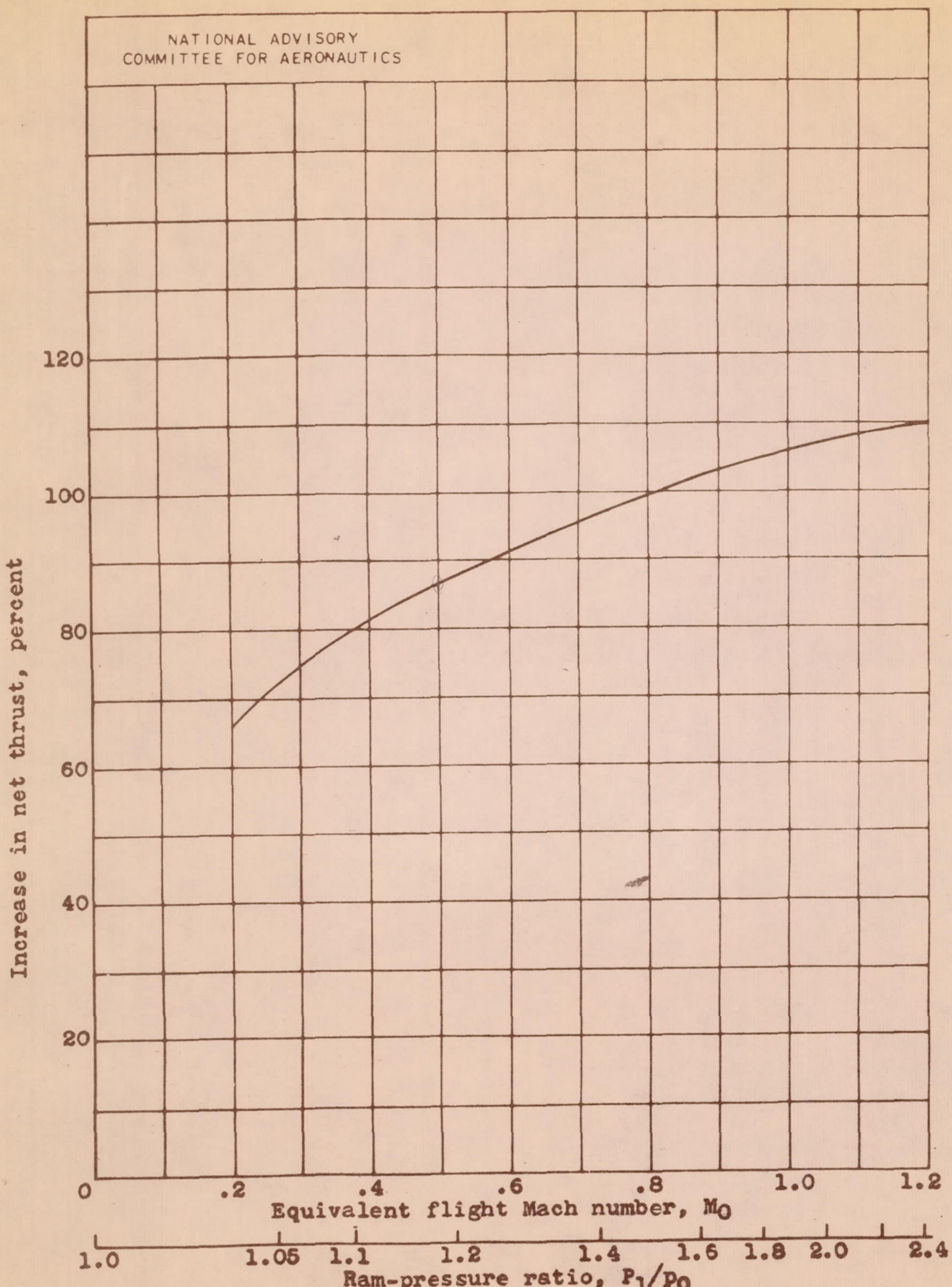


Figure 17.- Relation between equivalent flight Mach number and percentage increase in net thrust resulting from tail-pipe burning with turbojet engine. Engine speed, 7600 rpm; turbine-outlet temperature, 1220° F; altitude, 30,000 feet.

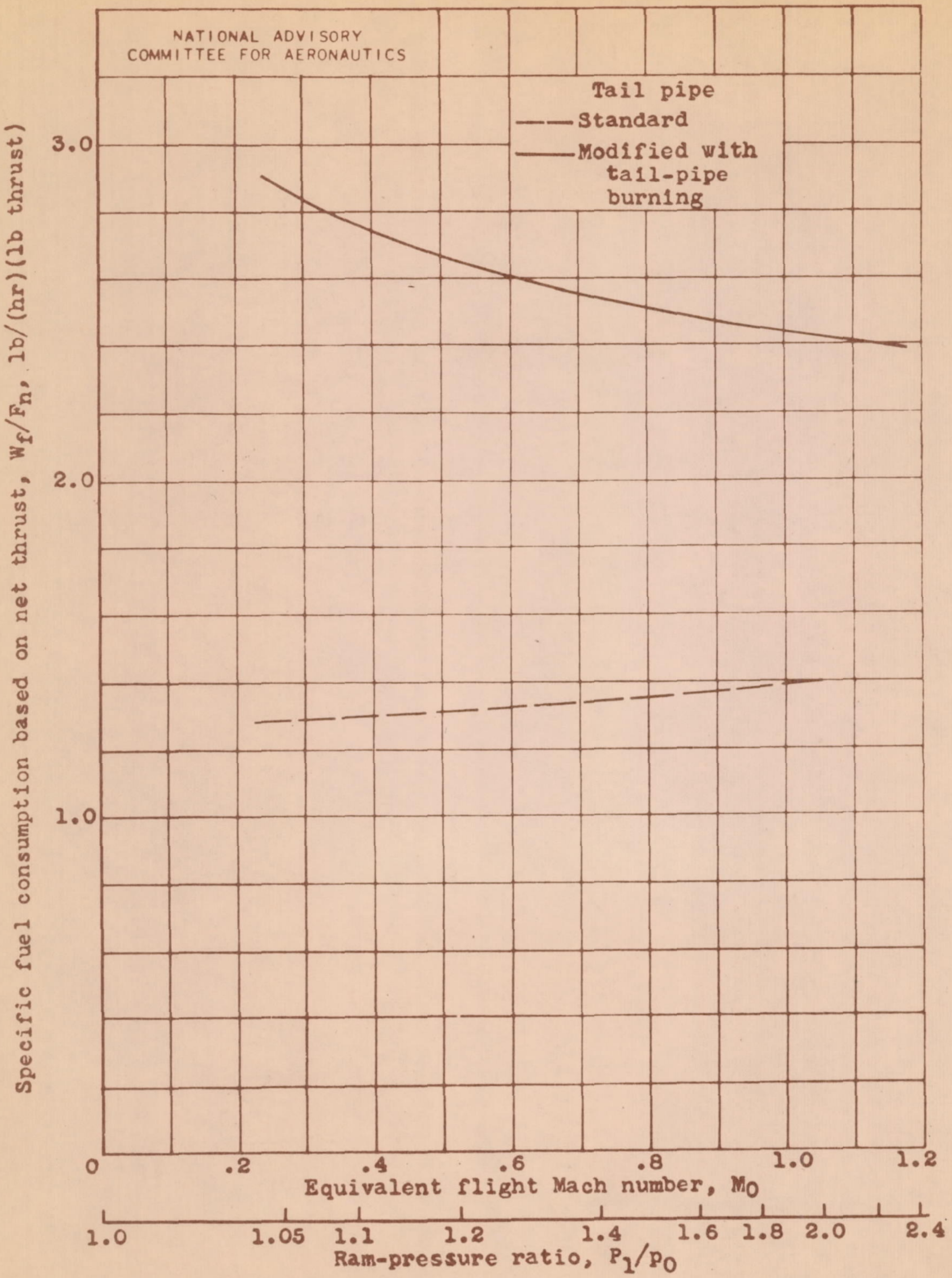


Figure 18.- Relation between equivalent flight Mach number and specific fuel consumption based on net thrust for turbojet engine with standard tail pipe and modified tail pipe with tail-pipe burning. Engine speed, 7600 rpm; turbine-outlet temperature, 1220° F; altitude, 30,000 feet.