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RESEARCH MEMORANDUM

ALTITUDE-WIND-TUNNEL INVESTIGATION OF PERFORMANCE CHARACTERISTICS OF A J47D PROTOTYPE (RX1-1) TURBOJET ENGINE WITH VARIABLE-AREA EXHAUST NOZZLE

CLASSIFICATION CHANGED

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Cleveland, Ohio

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ALTITUDE-WIND-TUNNEL INVESTIGATION OF PERFORMANCE CHARACTERISTICS OF A J47D PROTOTYPE (RX1-1) TURBOJET ENGINE WITH VARIABLE-AREA EXHAUST NOZZLE

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SUMMARY

An investigation has been conducted in the NACA Lewis altitude wind tunnel to evaluate the performance of a prototype model of the J47D (RX1-1) turbojet engine equipped with afterburner, variable-area exhaust nozzle, and integrated electronic control. All the data presented in this report were obtained with the afterburner inoperative. The engine was operated over a range of simulated flight conditions from 5000 to 55,000 feet, flight Mach numbers from 0.19 to 1.12, and a complete range of engine speeds using exhaust-nozzle areas scheduled by the control. Data are presented to show the effects of altitude at a flight Mach number of 0.19 and of flight Mach number at an altitude of 25,000 feet. Performance maps over the complete range of exhaust-nozzle areas were obtained at altitudes of 15,000 and 45,000 feet at a flight Mach number of 0.19 to determine the exhaust-nozzle-area schedule giving optimum values of specific fuel consumption. Performance of the engine at rated thrust (unaugmented) is also given for altitudes up to 60,000 feet at flight Mach numbers from 0.51 to 1.71.

For operation with the scheduled exhaust-nozzle area, the minimum specific fuel consumption was 1.15 pounds of fuel per hour per pound of net thrust and occurred at about 7200 rpm at altitudes from 5000 to 25,000 feet and a flight Mach number of 0.19. Some improvement in specific fuel consumption was possible at relatively high power levels by the use of a revised exhaust-nozzle-area schedule; however, such a schedule would probably penalize the acceleration characteristics of the engine. At flight Mach numbers above about 0.70, it was not possible to attain limiting turbine-outlet temperature with the nozzle fully closed. At an altitude of 25,000 feet and a flight Mach number of 0.92, analysis indicates that the net thrust could be increased approximately 6 percent if the exhaust-nozzle area would be sufficiently decreased to give limiting turbine-outlet temperature.

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INTRODUCTION

An investigation has been conducted in the NACA Lewis altitude wind tunnel to investigate the performance of a J47D prototype (RX1-1) turbojet engine over a wide range of simulated flight conditions. This engine was equipped with an afterburner, a variable-area exhaust nozzle, and an integrated electronic control. Inasmuch as some installations of this basic engine are contemplated with fixed-area exhaust nozzles, the first phase of the program (reference 1) was a complete evaluation of performance using a fixed-area exhaust nozzle.

The results presented herein were obtained during the second phase of the program and show the performance of the engine equipped with afterburner and variable-area exhaust nozzle, but with the afterburner inoperative. Two types of operation were investigated: (1) the performance was evaluated using the control, which provided a definite schedule of exhaust-nozzle area and engine speed with thrust-selector position and (2) the performance was evaluated for operation over a wide range of exhaust-nozzle areas at constant engine speed. These data were obtained at several values of engine speed to cover the range of practical interest.

Operation of the first type, referred to as scheduled operation, was conducted at altitudes from 5000 to 55,000 feet at a flight Mach number of 0.19 and at flight Mach numbers from 0.19 to 0.92 at an altitude of 25,000 feet to illustrate the effects of both variables. Unscheduled operation (with various constant exhaust-nozzle areas) was conducted at a flight Mach number of 0.19 and at altitudes of 15,000 and 45,000 feet. Unscheduled performance data were also obtained at an altitude of 25,000 feet for a more typical flight Mach number of 0.71. In addition, the investigation was extended to higher flight Mach numbers over a range of altitudes for the rated thrust condition. Flight Mach numbers as high as 1.71 were simulated at altitudes from 45,000 to 60,000 feet.

Data are presented in graphical form to show the trends of engine performance associated with changes of altitude, flight Mach number and exhaust-nozzle area. Also, all the over-all engine performance data obtained during this phase of the investigation are given in tabular form.

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APPARATUS AND PROCEDURE

Engine

Although the experimental prototype engine used for these tests carried no official manufacturer's rating, the rating of the J47D engine (with the afterburner not operating) is a minimum static sea-level thrust of 5700 pounds at 7950 rpm, and a turbine-outlet temperature of 1250° F. Air flow at rated conditions is given as 99 pounds per second with screens retracted. During this program, turbine-discharge temperatures as high as 1300° F were investigated. This engine was aerodynamically similar to the D series engine. The engine has a 12-stage axial-flow compressor with a pressure ratio of about 5.1 at rated speed and temperature, eight cylindrical direct-flow-type combustors, a single-stage impulse turbine, a tail-pipe burner, and a variable-area exhaust nozzle. Over-all length of the engine and afterburner is approximately 217 inches, the maximum diameter is about 37 inches, and the total weight about 3000 pounds.

The engine was controlled by an electronic system which provided thrust regulation throughout the unaugmented and afterburning regions by means of a single thrust-selector lever. Exhaust-nozzle projected area and engine speed were scheduled as functions of thrust-selector position (fig. 1). The schedule chosen represents the manufacturer's compromise between a schedule giving optimum steady-state performance and one giving optimum acceleration characteristics. Exhaust-nozzle position was indicated by potentiometer voltage and the relation between the projected nozzle area and the voltage is given in figure 2. For convenience, the numbers used to designate nozzle position correspond to the voltage readings.

Installation

The engine was mounted on a wing in the tunnel test section (fig. 3). Dry refrigerated air was supplied to the engine from the tunnel make-up air system through a duct connected to the engine inlet. Throttle valves were installed in the duct to permit regulation of the pressure at the face of the engine. Engine thrust and drag measurements by the tunnel balance scales were made possible by a frictionless slip joint located in the duct ahead of the engine.

Instrumentation for measuring pressures and temperatures was installed at various stations in the engine (fig. 4).

Procedure

Engine performance data with the scheduled values of exhaust-nozzle area were obtained over a wide range of thrust-selector settings at the following flight conditions:

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Altitude (ft)	Flight Mach number
5,000	0.19
15,000	0.19, 0.51, 0.72
25,000	0.19, 0.51, 0.71, 0.92
35,000	0.19, 0.71, 1.12
45,000	0.19
55,000	0.19

The air flow through the make-up air duct was throttled from approximately sea-level pressure to a total pressure at the engine inlet corresponding to the desired flight Mach number at a given altitude. In the calculation of flight Mach number, complete ram pressure recovery was assumed. Engine inlet-air temperatures were held at approximately NACA standard values for each flight condition except for low flight Mach number at altitudes above 25,000 feet. Engine inlet-air temperatures below 430° R were unobtainable. The fuel used was MIL-F-5624 (AN-F-58) with a lower heating value of 18,900 Btu per pound.

In order to obtain engine performance maps, the exhaust-nozzle area was manually adjusted to permit operation over a wide range of engine pressure ratios at constant engine speed. From five to eight exhaust-nozzle areas were used at each of the following conditions:

Altitude (ft)	Flight Mach number	Engine speed, rpm									
15,000	0.19	4091	5114		5944		6643		7386		7955
25,000	.71								7386	7692	7955
45,000	.19			5445	5944	6294	6643	6993	7386	7692	7955

The range of the investigation was extended to much higher flight Mach numbers and altitudes by reproducing the engine-inlet pressures and temperatures that would exist at these flight conditions and maintaining the static pressure in the tunnel test section low enough to produce choking of the exhaust nozzle. This procedure removes the limitations imposed by the tunnel exhauster equipment, and hence extends the range of available operating conditions. Because at high engine speeds sonic flow exists in the exhaust nozzle, conditions within the engine are unaffected by changes in the tunnel pressure. Consequently, engine performance is unaffected except for thrust, and the true value of thrust may be calculated by the use of equation (7) of the appendix.

An experimental check on the validity of the preceding method of operation was made by simulating a given flight condition and engine operating condition using three different pressures in the tunnel test section. One of these pressures corresponded to the actual ambient static pressure that would exist at the altitude being simulated. Net thrust values agreed within 2 percent and specific fuel consumption values within 1 percent.

Using this method, the performance of the engine at rated conditions of speed and temperature was obtained at the following conditions:

Simulated flight conditions			Inlet-air temperature (°F)
Altitude (ft)	Ram pressure ratio	Flight Mach number	
30,000	1.88	1.00	33
35,000	1.88	1.00	11
35,000	2.58	1.25	57
40,000	3.66	1.50	111
45,000	1.88	1.00	11
45,000	2.58	1.25	56
45,000	3.66	1.50	111
45,000	5.00	1.71	163
55,000	1.88	1.00	11
55,000	2.58	1.25	56
55,000	3.66	1.50	111
55,000	5.00	1.71	163
60,000	2.58	1.25	56
60,000	3.66	1.50	111
60,000	5.00	1.71	163

Although the range of flight conditions that may be investigated is greatly extended by this method, there are several factors which impose new limits. The requirement that the exhaust nozzle be choked restricts the range of engine operation to high thrust levels, particularly at low flight Mach numbers. Another restriction in the present case is the manufacturer's requirement that the engine inlet-air temperature not exceed 165° F. A third restriction on the application of this method lies in the assumption of complete ram pressure recovery. This assumption results in inlet conditions that are not precisely duplicated in actual installations, but for moderate flight Mach numbers (1.00 to 1.60) the errors are not large and engine performance obtained at a given set of inlet conditions may be presented in terms of a specific flight condition. At flight Mach numbers above about 1.60 however, not only does the pressure recovery of any particular diffuser type decrease considerably, but also the performance of various types varies appreciably. A diffuser efficiency must therefore be assumed both in computing the flight condition corresponding to a given set of engine inlet conditions and in computing net thrust from jet thrust. The applicability of data intended to simulate flight Mach numbers above 1.60 is therefore reduced. Accordingly, it should be noted that the values given in figure 14 for a flight Mach number of 1.71 do not correspond exactly to values which would be obtained in an actual installation.

The thrust data obtained using this method of operation are based on survey-rake measurements 1 inch upstream of the exhaust-nozzle outlet. Inasmuch as the total-pressure reduction from the nozzle inlet to the survey rake has been found to be less than 1 percent, most of the thrust loss associated with the exhaust nozzle occurs downstream of the measuring rake. Rake-thrust values therefore closely approximate the thrust ideally available. All other thrust values presented (obtained by simulating altitude pressures in the tunnel) are calculated from balance-scale measurements, which include losses associated with the exhaust nozzle. The ratio of these thrusts, defined as the effective velocity coefficient of the exhaust nozzle, is given in figure 5 to indicate the departure of the rake-thrust values from the thrusts actually obtainable. Comparable performance of a fixed-area exhaust nozzle is also shown by the broken line. These data, however, do not extend to sufficiently high values of exhaust-nozzle pressure ratio to permit direct application to the data obtained at the higher flight Mach numbers. The methods of calculation and the symbols used are given in the appendix.

RESULTS AND DISCUSSION

All data obtained in this phase of the investigation, including operation with scheduled and nonscheduled values of exhaust-nozzle

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area, are compiled in table I. Because the exhaust-nozzle area was varied, the derivations of reference 2 for the generalized performance of turbojet engines are not applicable, and accordingly, the data have been presented only in ungeneralized form. Also, the customary procedure of adjusting the data to NACA standard altitude conditions is impossible; therefore, average equivalent ambient-air temperatures (equation (3), appendix) are given in figure legends to indicate the departure from standard temperatures.

Scheduled Operation

Effect of altitude. - Engine performance data obtained at a flight Mach number of approximately 0.19 at altitudes from 5000 to 55,000 feet are presented in figure 6 for operation with the scheduled values of exhaust-nozzle area. Marked discontinuities, apparent in most of the performance curves, occur at the point where rated engine speed is reached; further change in performance is the result of changes in exhaust-nozzle area only. For clarity, arrows and symbols have been used in some of the figures to denote the point corresponding to the maximum setting of the thrust selector (without afterburning). At low altitudes, large thrust changes may be made almost instantaneously by closing the exhaust nozzle with the engine operating at rated speed. For example, at 5000 feet a thrust change of 22 percent of rated thrust (rated speed and limiting turbine-outlet temperature) is possible. At 55,000 feet, however, no appreciable thrust modulation is possible because the maximum exhaust-nozzle area scheduled at rated speed is required to prevent excessive turbine-outlet temperature. The larger exhaust-nozzle areas required at high altitudes result from decreases in component efficiencies associated with operation at reduced Reynolds number.

As would be expected, engine net thrust, fuel flow, and air flow decreased consistently with increasing altitude (figs. 6(a) to 6(c)). The specific fuel consumption (fig. 6(d)) changed very little as altitude was increased from 5000 to 25,000 feet; however, significant increases occurred at higher altitudes, reflecting decreased component efficiencies. Minimum specific fuel consumption values varied from 1.15 pounds of fuel per hour per pound of net thrust at 5000 feet to 1.48 at 55,000 feet. The engine speeds at which these minimum values occurred were 7200 and 7955 rpm, respectively. At rated speed, decreasing the exhaust-nozzle area slightly reduced the specific fuel consumption.

At altitudes above 15,000 feet, both engine fuel-air ratio and turbine-outlet total temperature increased with altitude as a result of decreasing component efficiencies (figs. 6(e) and 6(f)).

Losses with inoperative afterburner. - One effect of tail-pipe pressure losses associated with the inoperative afterburner may be seen by a comparison of specific fuel consumption values. With the afterburner installed, a specific fuel consumption of 1.225 pounds of fuel per hour per pound of thrust was obtained at rated thrust, an altitude of 25,000 feet, and a flight Mach number of 0.19. With a standard tail pipe and fixed conical exhaust nozzle, a specific fuel consumption of 1.178 pounds of fuel per hour per pound of net thrust was obtained at the same operating conditions (reference 1). Net thrust values at these operating conditions were reduced 2.4 percent by installation of the afterburner. Inasmuch as the exhaust nozzle losses were about the same for both fixed and variable-area nozzles (fig. 5) the loss in performance is due to greater pressure losses in the afterburner than in the standard engine tail pipe. It should be noted that total pressure loss in the standard tail pipe was high (4.5 percent) due to the very short over-all length and the consequent rapid expansion required.

Effect of flight Mach number. - In order to illustrate the effects of flight Mach number on engine performance, data are given in figure 7 for operation at Mach numbers of 0.19, 0.51, 0.71, and 0.92 at an altitude of 25,000 feet. Standard inlet-air temperatures were obtained at the two highest Mach numbers, whereas low temperatures corresponding to the two lowest Mach numbers could not be obtained.

As the flight Mach number was raised, the energy of the inlet air increased and the pressure drop across the engine increased. Accordingly, the engine approached the windmilling condition and less heat release from the fuel was required per pound of air to sustain a given engine speed. This reduction in heat release per pound of air is indicated by the decrease in fuel-air ratio (fig. 7(e)) and turbine-outlet total temperature (fig. 7(f)) with increasing Mach number. Over the entire speed range, the air flow increased as the flight Mach number was raised (fig. 7(c)). At engine speeds above 6650 rpm, the increase in air flow with Mach number outweighed the reduction in heat release per pound, requiring an increase in fuel flow; the converse was true at lower engine speeds (fig. 7(b)). Inasmuch as net thrust (fig. 7(a)) is directly related to energy added to the working fluid in passing through the engine, the trends of net thrust reflect those of fuel flow with due allowances for changes in component efficiencies.

At all engine speeds, the specific fuel consumption based on net thrust increased with an increase in flight Mach number (fig. 7(d)). Also, as engine speed decreased, the specific fuel consumption increased at a faster rate at the higher flight Mach numbers, reflecting the more rapid thrust decrease shown in figure 7(a). The minimum specific fuel consumption was 1.15 pounds per hour per pound of net thrust and

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occurred at 7200 rpm and a flight Mach number of 0.19. At a flight Mach number of 0.92 the minimum specific fuel consumption was 1.40 and occurred at rated speed.

The data in figure 7(f) and table I show that at flight Mach numbers above approximately 0.70 with the exhaust nozzle fully closed, limiting turbine-outlet temperature was not attained. Analysis indicates that if the nozzle were able to close far enough to give limiting turbine-outlet temperature at a flight Mach number of 0.92, the net thrust would be increased approximately 6 percent.

Nonscheduled Operation

Effect of exhaust-nozzle area. - The preceding discussion, which illustrated the effects of altitude and flight Mach number, was confined to engine operation with scheduled values of exhaust-nozzle area. A complete description of engine performance also requires that the effect of exhaust-nozzle area be given. This effect is shown in figures 8 and 9 for altitudes of 15,000 and 45,000 feet at a flight Mach number of 0.19 and in figure 10 for an altitude of 25,000 feet and a flight Mach of 0.71. Net thrust, fuel flow, air flow, specific fuel consumption, fuel-air ratio, and turbine-outlet total temperature are presented as functions of engine speed for several constant values of exhaust-nozzle area. These data were obtained by variations in nozzle area at several fixed values of engine speed; constant values of exhaust-nozzle area were obtained by cross plotting.

At an altitude of 15,000 feet and a flight Mach number of 0.19, net thrust, fuel flow, fuel-air ratio, and turbine-outlet total temperatures all decreased as the exhaust-nozzle area increased (figs. 8(a), 8(b), 8(e), and 8(f)). Engine air flow (fig. 8(c)) was unaffected by nozzle area in the range of engine speeds for which data were obtained because the compressor characteristic curves are essentially vertical in this region of operation.

At rated engine speed (fig. 8(a)), a thrust regulation of about 39 percent of maximum thrust was obtainable almost instantaneously by changing the nozzle area. At the same flight Mach number at 45,000 feet, a thrust regulation of 32 percent of maximum thrust was obtainable (fig. 9(a)). The thrust regulation increased to 51 percent at an altitude of 25,000 feet and a flight Mach number of 0.71 (fig. 10(a)). An appreciable degree of thrust regulation is therefore possible over a wide range of flight conditions by varying exhaust-nozzle area. Such regulation of thrust is extremely important for a "wave off" or other condition where a rapid acceleration is required. This change in thrust may be made much more rapidly by varying nozzle area rather than engine speed, particularly at high altitude where the density of the working fluid is reduced.

Except for the range of engine speeds between 6400 and 7955 rpm, the specific fuel consumption (fig. 8(d)) increased as exhaust-nozzle area increased. At speeds between 6400 and 7955 rpm, the specific fuel consumption was unaffected at nozzle positions up to 8 which corresponds to about 110 percent of the area in the closed position. Further increases in nozzle area caused the specific fuel consumption to increase.

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It should be noted that most of the variables are much less sensitive to changes in nozzle area near the open position than near the closed position. Inasmuch as the turbine-outlet area is approximately 2.87 square feet, further increases in exhaust-nozzle area above the maximum investigated (3.20 sq ft) would obviously have little effect on engine performance.

The conventional methods of performance generalization, such as discussed in reference 2, may be used in conjunction with the constant-area data of figure 8 to predict engine performance for a range of altitudes up to 25,000 or 30,000 feet with reasonable accuracy. These generalization methods, however, do not account for changes in performance due to changes in Reynolds number or combustion efficiency. Data obtained at 45,000 feet are therefore presented in figure 9 to show the complete performance at a low value of Reynolds number and to provide a more accurate basis for performance prediction at high altitues. Typical effects of Reynolds number changes on compressor performance are given in reference 3 for several engines. The effects of variations in Reynolds number on several engine performance variables are shown for a given engine in reference 4.

The data obtained at 45,000 feet and a flight Mach number of 0.19 (fig. 9) exhibit the same trends with changes in nozzle area as the data obtained at 15,000 feet. A consequence of the Reynolds number effect, coupled with a change in corrected engine speed, is indicated by a comparison of turbine-outlet temperatures at 15,000 and 45,000 feet (figs. 8(f) and 9(f)). Extrapolation of the data shows that for operation at standard NACA inlet-air temperatures, an increase of about 10 percent in projected exhaust-nozzle area would be required to maintain limiting turbine-outlet temperature at rated speed as the altitude was increased from 15,000 to 45,000 feet at a flight Mach number of 0.19.

The data of figure 10, obtained at an altitude of 25,000 feet and a flight Mach number of 0.71, are more representative of a typical flight condition than the data of figures 8 and 9. Although the range of the data of figure 10 is small, the region of practical interest is presented. Trends of the data are similar to those previously shown; however, the turbine-outlet temperatures are lower for a given exhaust-nozzle area, and as noted in connection with figure 7(f) the exhaust-nozzle area could not be reduced enough to give limiting turbine-outlet temperature or maximum thrust. The minimum specific fuel consumption obtained, 1.3 pounds of fuel per hour per pound of net thrust, occurred at a speed of 7386 rpm using a projected exhaust-nozzle area slightly larger than the minimum. At this condition, the thrust was about 70 percent of the maximum obtained.

Over-all performance maps. - From the same data used to plot figures 8 and 9, performance maps have been constructed to provide a comprehensive picture of engine performance at each of the two flight conditions and to show the exhaust-nozzle schedule which would result in the minimum specific fuel consumption at any desired thrust level (figs. 11 and 12). Solid lines are used to denote constant values of net thrust and broken lines to designate contours of constant specific fuel consumption. Also shown on these maps are the nozzle schedule used and a nozzle schedule that would result in the minimum specific fuel consumption at any given value of net thrust. A symbol is used to denote the rated thrust condition.

At low thrust levels, the lines of constant thrust are almost parallel with the lines of constant specific fuel consumption. Consequently, the specific fuel consumption is not greatly affected by the combination of engine speed and area used to obtain a given thrust. At high thrust levels, however, the thrust and fuel consumption curves are not parallel. At altitudes of 15,000 and 45,000 feet (figs. 11 and 12), the specific fuel consumption at constant thrust is decreased as the area and engine speed are both reduced.

From the previous discussion it may be noted that insofar as minimum specific fuel consumption is concerned there is little to be gained at either flight condition by departure from the exhaust-nozzle schedule used at low thrust levels. However, an appreciable improvement is possible at high thrust levels. For example, at a thrust level of 2400 pounds (fig. 11), the specific fuel consumption could be reduced from about 1.17 to 1.11 pounds per hour per pound of net thrust, a reduction of about 5 percent, by reducing engine speed from 7800 to 7000 rpm and reducing the nozzle area. Similarly at 45,000 feet and a representative cruise thrust level of 700 pounds, a reduction of about 5 percent in specific fuel consumption is possible by a departure from the nozzle schedule used and a reduction in engine speed from 7600 to 6800 rpm. These improvements are primarily the result of a shift in the compressor and turbine operating points to more efficient regions of operation. Although these departures from the schedule maintained by the control would result in improvements in specific fuel consumption, the time required for acceleration to rated thrust would be materially increased because of the much larger change required in rotor speed. This increase is particularly apparent at high altitudes where the excess power available for acceleration is reduced in proportion to the air density while the rotor mass to be accelerated remains constant. 2135

The data obtained at 25,000 feet and a flight Mach number of 0.71 were insufficient to permit construction of a map similar to figures 11 and 12; however, the optimum schedule from a performance viewpoint may be seen in figure 13 where specific fuel consumption is plotted against net thrust. Throughout the range covered by the data, the lowest specific fuel consumption was obtained at the lowest of the three engine speeds. Concomitantly, at any constant net thrust, the smallest nozzle areas used gave the optimum specific fuel consumption and, in addition, the trends of the data are such that a slight improvement might be expected if the engine speed were reduced even further. It may be concluded from figures 11 to 13 that some improvement in specific fuel consumption was possible at relatively high power levels by the use of a revised nozzle schedule; however, the engine acceleration characteristics would probably be penalized.

Operation at rated thrust. - Net thrust and specific fuel consumption based on net thrust are shown as functions of flight Mach number in figure 14 for operation at the rated thrust condition. For the range of flight Mach numbers covered, the increase in thrust with Mach number appears to be almost linear. As would be expected, specific fuel consumption also increased with flight Mach number; the increase, however, was most rapid at the highest Mach number. Scatter of the specific fuel consumption data precludes any conclusions as to the effect of altitude.

SUMMARY OF RESULTS

The following results were obtained from an investigation of a J47D (RX-1) turbojet engine in the NACA Lewis altitude wind tunnel:

1. For operation with the exhaust-nozzle schedule specified by the manufacturer, the minimum specific fuel consumption was 1.15 pounds of fuel per hour per pound of net thrust and occurred at an engine speed of about 7200 rpm, altitudes from 5000 to 25,000 feet, and a flight Mach number of 0.19.

2. At flight Mach numbers above about 0.70, it was impossible to attain limiting turbine-outlet temperature with the nozzle fully closed.

Analysis indicates that the net thrust could be increased approximately 6 percent at an altitude of 25,000 feet and a flight Mach number of 0.92 if the exhaust-nozzle area were decreased enough to give limiting turbine-outlet temperature.

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3. Although some improvement in specific fuel consumption was possible at relatively high power levels by the use of a revised exhaust-nozzle-area schedule, the acceleration-characteristics of the engine would probably be penalized by use of the revised schedule.

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APPENDIX - METHODS OF CALCULATIONS

Symbols

The following symbols were used in this report: 2135

A cross-sectional area, sq ft

B thrust-scale reading, lb

C_E exhaust-nozzle effective jet-velocity coefficient (ratio of actual jet thrust to ideal jet thrust after expansion to free-stream static pressure)

C_T ratio of hot exhaust-nozzle area to cold exhaust-nozzle area

D external drag of installation, lb

D_r exhaust-nozzle tail-rake drag, lb

F_j jet thrust, lb

F_n net thrust, lb

f/a fuel-air ratio

g acceleration due to gravity, 32.2 ft/sec^2

M_0 flight Mach number

N engine speed, rpm

P total pressure, lb/sq ft absolute

p static pressure, lb/sq ft absolute

R gas constant, 53.3 ft-lb/(lb)($^{\circ}$R)

T total temperature, $^{\circ}$R

T_i indicated temperature, $^{\circ}$R

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

V velocity, ft/sec

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W_a air flow, lb/sec

W_f fuel flow, lb/hr

W_f/F_n specific fuel consumption based on net thrust, lb/(hr)(lb net thrust)

γ ratio of specific heats

Subscripts:

0 free-air stream

1 engine inlet

6 turbine outlet

8 1 in. upstream of exhaust-nozzle outlet

e equivalent ambient

r rake measurement

s scale measurement

x inlet duct 6 in. upstream from frictionless slip-joint flange

y inlet duct $28\frac{3}{4}$ in. downstream from frictionless slip joint

Calculations

Flight Mach number. - The flight Mach number, assuming complete ram pressure recovery, was calculated from the expression

$$M_0 = \sqrt{\frac{2}{\gamma_1 - 1}\left[\left(\frac{P_1}{p_0}\right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1\right]} \tag{1}$$

Temperature. - Total temperature was determined, using a calibrated thermocouple with an impact-recovery factor of 0.85, from the indicated temperature by use of the equation

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

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Equivalent ambient-air temperature. - Equivalent ambient-air temperature was obtained from the adiabatic relation of pressures and temperatures,

$$t_e = \frac{T_1}{\left(\frac{P_1}{p_0}\right)^{\frac{\gamma_1-1}{\gamma_1}}} \tag{3}$$

Engine air flow. - Engine air flow was calculated from pressure and temperature measurements obtained at the engine inlet (station 1) by

$$W_{a,1} = A_1 p_1 \sqrt{\frac{2\gamma_1 g}{t_1 R(\gamma_1 - 1)}\left[\left(\frac{P_1}{p_1}\right)^{\frac{\gamma_1-1}{\gamma_1}} - 1\right]} \tag{4}$$

Air flows calculated from measurements in the tail pipe agreed with those obtained at the engine inlet within 1 percent at all operating conditions.

Thrust. - It was possible to obtain thrust by both balance-scale measurements and from measurements of temperature and pressure at the exhaust-nozzle outlet. The ratio of scale to rake thrust, termed the effective jet-velocity coefficient, is given in figure 5.

Jet thrust determined from the balance-scale measurements was calculated as

$$F_{j,s} = D + B + D_r + \frac{W_{a,1}V_y}{g} + A_x(p_x - p_0) \tag{5}$$

Then, net thrust is

$$F_{n,s} = F_{j,s} - \frac{W_{a,1}}{g} V_e \tag{6}$$

The last two terms of equation (5) represent the momentum and the pressure forces on the installation at the slip joint in the inlet-air duct. The drag of the installation was determined with the engine inoperative and with a blocking plate installed in the inlet to prevent air flow through the engine.

The rake thrust, which is the ideal thrust available, is

$$F_{j,r} = \frac{2C_T A_9 p_9 \gamma_9}{\gamma_9 - 1}\left[\left(\frac{p_9}{P_9}\right)^{\frac{\gamma_9 - 1}{\gamma_9}} - 1\right] + C_T A_9 (p_9 - p_0) \qquad (7)$$

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TABLE I - ENGINE PERFORMANCE DATA FOR J47D (RX1-1)

(a) With electronic

Run	Altitude (ft)	Ram pressure ratio P_1/p_0	Flight Mach number M_0	Thrust-selector position (deg)	Nozzle position (d.-c. volts)	Tunnel static pressure p_0 (lb/sq ft abs.)	Equivalent ambient-air temperature t_e (°R)	Engine speed N (rpm)	Engine-inlet indicated temperature $T_{i,1}$ (°R)	Jet thrust $F_{j,s}$ (lb)	Net thrust $F_{n,s}$ (lb)
1	5000	1.020	0.169	90	5.0	1745	501	7955	502	5141	4647
2		1.021	.173	85	6.5	1747	501	7955	502	5018	4512
3		1.021	.173	80	8.0	1764	502	7955	503	4618	4104
4		1.021	.173	75	9.0	1761	500	7955	501	4340	3827
5		1.022	.176	70	11.0	1757	501	7955	502	4134	3617
6		1.021	.173	62	11.0	1754	501	7692	502	3817	3315
7		1.023	.180	55	11.3	1754	504	7386	505	3418	2903
8		1.023	.180	48	12.0	1749	500	6993	502	2887	2399
9		1.027	.195	28	19.0	1756	501	5944	504	1489	1058
10		1.030	.206	22	19.0	1749	493	5114	497	977	610
11		1.025	.188	22	20.0	1766	500	5114	503	939	614
12		1.033	.216	17	20.0	1772	499	4091	503	432	167
13		1.028	.199	13	20.0	1759	500	3147	503	260	81
14		1.028	.199	--	20.0	1759	497	2046	500	104	-17
15	15,000	1.023	0.180	90	5.5	1192	469	7955	470	3692	3329
16		1.023	.180	80	8.0	1188	470	7955	471	3417	3057
17		1.023	.180	70	11.0	1191	470	7955	471	3009	2646
18		1.026	.192	62	11.0	1188	470	7692	471	2718	2334
19		1.024	.184	48	12.5	1189	469	6993	471	2100	1754
20		1.027	.195	32	19.0	1194	470	5944	473	1095	790
21		1.027	.195	23	19.0	1190	468	5114	472	666	427
22		1.028	.199	17	19.0	1189	468	4091	472	225	57
23		1.029	.203	13	19.0	1191	471	3147	475	191	68
24		1.029	.203	--	19.0	1190	468	2448	472	116	25
25		1.192	.507	90	4.5	1190	472	7955	494	4409	3259
26		1.196	.512	80	8.0	1184	471	7955	494	3733	2567
27		1.195	.511	70	10.0	1185	471	7955	494	3407	2243
28		1.191	.506	60	11.0	1192	471	7692	493	3136	1998
29		1.195	.511	48	12.0	1192	470	6993	493	2477	1393
30		1.197	.514	28	19.0	1197	469	5944	493	1214	332
31		1.196	.512	22	20.0	1195	469	5114	494	725	20
32		1.202	.520	17	20.0	1197	469	4091	494	345	-139
33		1.199	.516	--	20.0	1197	470	3147	495	156	-175
34		1.405	.715	90	4.0	1197	465	7955	510	----	----
35		1.403	.713	80	7.5	1190	467	7955	512	----	----
36		1.407	.716	70	10.0	1188	467	7955	513	----	----
37		1.409	.718	60	11.0	1186	468	7692	514	----	----
38		1.401	.711	48	12.0	1197	469	6993	514	----	----
39		1.407	.716	29	18.5	1190	466	5944	513	----	----
40		1.408	.717	18	19.5	1193	468	5114	515	----	----
41		1.412	.720	--	20.0	1193	468	4091	515	----	----
42	25,000	1.024	0.184	90	6.0	782	457	7955	458	2475	2229
43		1.023	.180	85	6.5	781	458	7955	459	2458	2220
44		1.023	.180	80	8.0	781	460	7955	461	2236	2000
45		1.023	.180	75	9.0	783	462	7955	463	2138	1899
46		1.023	.180	72	10.5	782	461	7955	462	2031	1792
47		1.022	.176	63	11.5	784	460	7692	461	1821	1591
48		1.024	.184	49	12.0	782	460	6993	462	1428	1197
49		1.024	.184	30	18.5	781	462	5944	464	753	561
50		1.027	.195	23	20.0	781	461	5114	464	447	288
51		1.028	.199	17	20.0	783	461	4091	465	222	119
52		1.029	.203	13	20.0	781	461	3147	465	----	----
53		1.032	.212	--	20.0	780	462	2046	466	52	-5
54		1.198	.515	90	5.0	774	442	7955	463	3082	2303
55		1.198	.515	90	5.0	772	443	7955	464	3056	2286
56		1.190	.505	80	8.5	777	443	7955	464	2724	1973
57		1.193	.509	70	11.0	774	443	7955	464	2435	1680
58		1.192	.507	62	11.5	775	441	7692	462	2215	1465
59		1.201	.519	48	12.3	776	439	6993	462	1742	1018

TURBOJET ENGINE EQUIPPED WITH VARIABLE-AREA NOZZLE

control operative.

Engine-inlet air flow $W_{a,1}$ (lb/sec)	Fuel flow W_f (lb/hr)	Specific fuel-consumption based on net thrust $W_f/F_{n,s}$ (lb/(hr)(lb thrust))	Fuel-air ratio f/a	Turbine-outlet total temperature, T_6 (°R)	Turbine-outlet total pressure P_6 (lb/sq ft abs.)	Engine total-pressure ratio P_9/P_1	Engine total-temperature ratio T_6/T_1	Run
85.88	5550	1.194	0.0184	1751	3666	1.930	3.474	1
85.69	5315	1.178	.0177	1711	3604	1.890	3.395	2
86.87	4770	1.162	.0156	1600	3415	1.760	3.168	3
86.86	4485	1.172	.0147	1533	3270	1.675	3.048	4
86.19	4235	1.171	.0140	1486	3176	1.621	2.948	5
85.04	3790	1.143	.0127	1408	3041	1.561	2.794	6
83.46	3340	1.151	.0114	1320	2892	1.479	2.604	7
79.37	2860	1.192	.0102	1239	2708	1.391	2.458	8
64.78	1675	1.583	.0073	1036	2221	1.157	2.051	9
52.54	1242	2.036	-----	----	2033	1.088	-----	10
50.64	1227	1.998	.0069	995	2037	1.084	1.974	11
36.07	1020	6.108	.0080	1067	1931	1.030	2.117	12
26.41	823	10.160	.0088	1129	1851	1.013	2.240	13
17.87	515		.0081	1100	1808	.995	2.196	14
61.09	3960	1.190	0.0185	1746	2564	1.952	3.699	15
60.51	3500	1.145	.0165	1623	2395	1.819	3.431	16
61.02	3100	1.172	.0145	1509	2233	1.676	3.190	17
60.42	2690	1.153	.0127	1386	2121	1.590	2.930	18
56.85	1975	1.126	.0099	1191	1895	1.426	2.523	19
47.34	1185	1.500	.0071	985	1554	1.179	2.078	20
37.15	870	2.037	.0066	948	1401	1.101	2.008	21
25.69	758	13.30	.0083	1002	1311	1.049	2.123	22
18.34	635	9.338	.0098	1067	1247	1.008	2.246	23
13.67	518	20.72	.0107	1081	1234	.998	2.290	24
68.50	4310	1.322	.0180	1719	2867	1.891	3.466	25
68.83	3500	1.363	.0144	1521	2522	1.631	3.067	26
68.91	3190	1.422	.0132	1431	2381	1.520	2.885	27
68.02	2825	1.414	.0118	1339	2269	1.445	2.705	28
64.23	2090	1.500	.0093	1159	2041	1.299	2.346	29
51.99	1052	3.169	.0057	904	1586	1.025	1.830	30
41.76	723	36.15	.0049	839	1429	.957	1.698	31
28.29	550		.0055	824	1319	.896	1.668	32
19.41	364		.0053	776	1274	.877	1.568	33
80.49	4670	-----	.0165	1674	3257	1.808	3.270	34
79.77	3950	-----	.0141	1525	2942	-----	2.967	35
79.33	3445	-----	.0124	1411	2710	1.468	2.740	36
78.31	2970	-----	.0108	1307	2556	1.379	2.533	37
72.41	2065	-----	.0081	1111	2232	1.197	2.153	38
56.85	882	-----	.0042	832	1653	.907	1.619	39
46.33	497	-----	.0030	741	1432	.814	1.436	40
-----	270	-----	-----	----	1315	.763	-----	41
40.90	2730	1.225	0.0191	1796	1726	1.990	3.904	42
40.55	2675	1.205	.0188	1791	1726	1.992	3.885	43
40.44	2380	1.190	.0168	1648	1606	1.854	3.559	44
40.48	2270	1.195	.0160	1577	1556	1.785	3.391	45
40.46	2150	1.200	.0151	1535	1498	1.704	3.308	46
39.89	1875	1.179	.0134	1407	1425	1.618	3.039	47
38.30	1385	1.157	.0103	1198	1280	1.458	2.587	48
31.81	850	1.515	.0076	980	1038	1.195	2.108	49
24.98	645	2.240	.0073	937	941	1.118	2.019	50
15.88	570	4.790	.0101	993	867	1.046	2.135	51
-----	480	-----	-----	----	834	1.019	-----	52
8.20	300		.0104	1073	616	1.001	2.303	53
47.28	3065	1.331	.0185	1768	1959	1.961	3.802	54
46.64	3065	1.341	.0157	1739	1948	1.952	3.732	55
46.39	2590	1.313	.0159	1579	1785	1.774	3.388	56
46.26	2330	1.387	.0143	1459	1651	1.609	3.131	57
46.15	2010	1.372	.0124	1329	1560	1.523	2.864	58
43.74	1445	1.419	.0094	1127	1396	1.344	2.434	59

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TABLE I - ENGINE PERFORMANCE DATA FOR J47D (RX1-1) TURBOJET

(a) With electronic control

Run	Altitude (ft)	Ram pressure ratio P_1/p_0	Flight Mach number M_0	Thrust-selector position (deg)	Nozzle position (d.-c. volts)	Tunnel static pressure p_0 (lb/sq ft abs.)	Equivalent ambient-air temperature t_e (°R)	Engine speed N (rpm)	Engine-inlet indicated temperature $T_{1,1}$ (°R)	Jet thrust $F_{j,s}$ (lb)	Net thrust $F_{n,s}$ (lb)
60	25,000	1.202	0.520	31	18.5	776	439	5944	462	910	295
61		1.204	.522	22	19.5	779	439	5114	463	531	37
62		1.195	.511	21	19.5	781	440	4091	463	263	-84
63		1.397	.708	90	5.0	776	424	7955	465	3745	2518
64		1.397	.708	80	8.5	774	428	7955	469	3327	2110
65		1.391	.704	70	11.0	782	429	7955	470	3029	1813
66		1.394	.706	62	11.0	781	430	7692	471	2744	1534
67		1.401	.711	47	12.0	779	430	6993	473	2124	981
68		1.391	.704	29	19.0	782	432	5944	474	1031	88
69		1.399	.710	18	19.5	781	432	5114	475	572	-180
70		1.403	.713	--	19.5	779	423	4091	466	294	-257
71		1.765	.939	90	3.5	790	418	7955	490	----	----
72		1.732	.922	90	3.5	786	435	7955	507	4751	2837
73		1.730	.921	85	5.5	782	438	7955	510	----	----
74		1.720	.916	80	7.5	779	432	7955	502	4184	2309
75		1.734	.923	75	9.0	786	437	7955	509	4026	2117
76		1.736	.924	70	10.5	779	430	7955	502	3836	1924
77		1.731	.922	63	10.5	788	434	7692	506	3534	1848
78		1.730	.921	49	12.5	781	432	6993	503	2614	868
79		1.727	.920	28	19.0	784	436	5944	509	1246	-182
80		1.734	.923	10	19.0	779	428	5280	500	764	-410
81	35,000	1.020	0.169	90	7.8	498	453	7955	454	1470	1329
82		1.020	.169	80	8.0	494	451	7955	452	1499	1359
83		-----	----	70	10.0	496	---	7955	450	----	----
84		-----	----	62	11.0	498	---	7692	451	----	----
85		1.020	.169	50	12.5	498	448	6993	450	932	799
86		1.022	.176	30	18.5	498	449	5944	451	492	377
87		1.028	.199	21	19.6	494	447	5114	451	279	178
88		1.034	.219	16	19.6	494	447	4091	451	169	91
89		1.394	.706	90	5.0	498	411	7955	450	2541	1757
90		1.400	.711	80	7.2	493	410	7955	449	2296	1508
91		1.393	.705	70	10.0	491	408	7955	447	2033	1262
92		1.385	.699	64	10.6	496	410	7692	448	1864	1108
93		1.389	.702	50	11.5	493	409	6993	447	1441	713
94		1.388	.701	30	18.0	500	411	5944	450	755	138
95		1.394	.706	16	19.5	493	408	5114	448	422	-92
96		1.404	.714	10	19.5	495	408	4273	450	229	-166
97		2.193	1.122	90	4.5	497	397	7955	495	4196	2394
98		2.176	1.116	80	7.5	500	402	7955	500	3648	1847
99		2.178	1.117	70	10.0	501	402	7955	500	3377	1591
100		2.171	1.114	62	11.0	498	403	7692	501	3083	1335
101		2.204	1.126	48	11.5	496	401	6993	501	2425	762
102		2.187	1.120	28	19.6	503	393	5944	490	1306	-102
103		2.193	1.122	20	19.6	502	391	5114	499	701	-419
104	45,000	1.039	0.235	90	9.5	308	443	7955	446	914	792
105		1.023	.180	85	11.0	310	444	7955	445	905	811
106		1.026	.192	70	11.0	303	442	7955	443	852	755
107		1.032	.212	80	12.0	310	441	7955	443	867	757
108		1.026	.192	62	11.5	303	443	7692	444	789	692
109		1.026	.192	55	11.5	310	444	7386	446	723	627
110		1.033	.216	48	12.2	306	443	6993	446	615	511
111		1.026	.192	32	17.5	311	448	5944	450	329	254
112		1.032	.212	24	20.0	310	446	5114	450	187	120
113	55,000	1.032	0.212	90	13.0	190	449	7955	452	504	438
114		1.026	.192	85	12.5	191	453	7955	455	510	451
115		1.032	.212	80	14.0	190	449	7955	452	497	431
116		1.037	.228	62	11.5	190	449	7692	453	478	410
117		1.032	.212	40	12.0	189	450	6993	453	366	307

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ENGINE EQUIPPED WITH VARIABLE-AREA NOZZLE - Continued

operative - Concluded.

Engine-inlet air flow $W_{a,1}$ (lb/sec)	Fuel flow W_f (lb/hr)	Specific fuel-consumption based on net thrust $W_f/F_{n,s}$ (lb/(hr)(lb thrust))	Fuel-air ratio f/a	Turbine-outlet total temperature, T_6 (^oR)	Turbine-outlet total pressure P_6 (lb/sq ft abs.)	Engine total-pressure ratio P_9/P_1	Engine total-temperature ratio T_6/T_1	Run
37.10	759	2.573	0.0058	870	1093	1.069	1.879	60
29.62	512	13.84	.0049	788	965	.974	1.702	61
21.27	421		.0056	788	870	.907	1.702	62
53.80	3405	1.352	.0176	1712	2225	1.905	3.666	63
53.19	2885	1.367	.0151	1543	2018	1.707	3.276	64
53.48	2545	1.404	.0132	1421	1864	1.541	3.011	65
53.03	2170	1.415	.0114	1295	1747	1.435	2.738	66
49.80	1520	1.549	.0085	1090	1549	1.259	2.300	67
41.62	677	7.693	.0045	808	1137	.945	1.701	68
32.96	376		.0032	695	966	.837	1.463	69
24.34	241		.0027	638	887	.786	1.369	70
-----	3980	-----	-----	1678	2738	1.828	-----	71
65.31	3840	1.354	.0168	1703	2661	1.823	3.346	72
64.73	3705	-----	.0163	1675	2596	1.785	3.271	73
64.66	3150	1.364	.0139	1501	2343	1.594	2.978	74
64.90	2955	1.396	.0130	1445	2259	1.497	2.828	75
65.46	2775	1.442	.0121	1386	2163	1.425	2.750	76
64.47	2485	1.508	.0110	1290	2045	1.333	2.539	77
59.86	1564	1.802	.0074	1056	1707	1.104	2.091	78
48.81	573		.0033	761	1209	.789	1.492	79
40.35	277		.0019	635	1024	.704	1.267	80
25.81	1720	1.294	0.0191	1766	1059	1.906	3.873	81
25.61	1695	1.247	.0189	1747	1052	1.907	3.848	82
-----	1450	-----	-----	----	----	-----	-----	83
-----	1287	-----	-----	----	----	-----	-----	84
24.45	968	1.212	.0112	1221	825	1.463	2.707	85
20.31	615	1.631	.0086	985	667	1.198	2.179	86
15.81	490	2.753	.0088	930	594	1.118	2.062	87
11.04	480	5.275	.0123	1009	548	1.043	2.237	88
35.93	2405	1.369	.0191	1783	1487	1.968	3.945	89
35.97	2070	1.373	.0164	1621	1369	1.810	3.594	90
35.53	1750	1.387	.0140	1462	1225	1.605	3.256	91
35.08	1530	1.381	.0124	1342	1160	1.501	2.982	92
33.64	1103	1.547	.0093	1106	1013	1.307	2.463	93
28.51	547	3.964	.0054	514	756	.964	1.805	94
23.67	299		.0036	684	629	.860	1.523	95
17.99	194		.0030	640	569	.793	1.422	96
52.91	3250	1.358	.0175	1705	2184	1.862	3.431	97
52.87	2590	1.402	.0139	1507	1912	1.597	3.002	98
52.40	2285	1.436	.0124	1382	1749	1.428	2.753	99
51.29	1960	1.468	.0109	1285	1631	1.334	2.555	100
48.44	1288	1.689	.0076	1051	1393	1.107	2.089	101
41.62	484		.0033	738	960	.739	1.503	102
33.14	425		.0036	673	719	.597	1.373	103
16.19	1130	1.427	0.0199	1759	657	1.869	3.926	104
16.16	1088	1.342	.0192	1737	645	1.864	3.886	105
15.70	1036	1.372	.0188	1708	626	1.823	3.838	106
16.25	1020	1.347	.0179	1692	628	1.772	3.802	107
15.70	890	1.286	.0161	1573	583	1.701	3.527	108
15.56	768	1.225	.0140	1444	577	1.642	3.230	109
14.99	689	1.348	.0130	1316	541	1.541	2.944	110
12.13	475	1.870	.0111	1060	440	1.266	2.350	111
9.77	392	3.267	.0113	991	380	1.138	2.202	112
9.57	659	1.505	0.0196	1734	380	1.730	3.828	113
9.51	647	1.435	.0194	1750	378	1.724	3.828	114
9.57	635	1.473	.0190	1727	381	1.724	3.812	115
9.29	645	1.573	.0198	1704	370	1.675	3.753	116
8.60	541	1.762	.0180	1508	341	1.585	3.322	117

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TABLE I -- ENGINE PERFORMANCE DATA FOR J47D (RX1-1) TURBOJET

(b) With electronic

Run	Altitude (ft)	Ram pressure ratio P_I/p_0	Flight Mach number M_0	Nozzle position (d.-c. volts)	Tunnel static pressure p_0 (lb/sq ft abs.)	Equivalent ambient-air temperature t_e (°R)	Engine speed N (rpm)	Engine-inlet indicated temperature $T_{i,1}$ (°R)	Jet thrust $F_{j,s}$ (lb)	Net thrust $F_{n,s}$ (lb)
1	15,000	1.020	0.169	5.5	1187	476	7955	477	3601	3266
2		1.024	.184	7.5	1186	473	7955	474	3462	3092
3		1.023	.180	8.0	1191	470	7955	471	3438	3075
4		1.024	.184	10.0	1188	471	7955	472	3134	2763
5		1.023	.180	11.5	1193	472	7955	473	2889	2525
6		1.025	.188	15.0	1188	471	7955	472	2673	2292
7		1.022	.176	34.5	1190	472	7955	473	2300	1944
8		1.024	.184	3.5	1188	474	7386	475	3375	3017
9		1.024	.184	5.0	1190	475	7386	476	3089	2729
10		1.023	.180	5.5	1190	474	7386	475	2973	2622
11		1.024	.184	8.5	1187	474	7386	475	2626	2268
12		1.024	.184	14.0	1186	473	7386	474	2262	1904
13		1.024	.184	34.5	1188	475	7386	476	1896	1538
14		1.026	.192	3.5	1183	476	6643	479	2626	2288
15		1.026	.192	6.0	1186	473	6643	475	2393	2051
16		1.024	.184	8.0	1186	472	6643	474	2180	1853
17		1.024	.184	10.0	1188	474	6643	476	2005	1677
18		1.024	.184	14.0	1190	476	6643	478	1739	1408
19		1.024	.184	34.5	1192	476	6643	478	1444	1115
20		1.027	.195	3.5	1193	473	5944	476	1878	1580
21		1.026	.192	5.5	1188	472	5944	474	1722	1425
22		1.025	.188	7.5	1191	473	5944	475	1556	1268
23		1.025	.188	14.0	1190	473	5944	475	1413	1125
24		1.026	.192	15.0	1190	473	5944	475	1166	869
25		1.027	.195	34.5	1187	473	5944	476	1033	733
26		1.028	.199	3.5	1187	470	5114	473	1135	892
27		1.030	.206	5.5	1188	469	5114	473	992	743
28		1.027	.195	9.0	1189	469	5114	473	901	662
29		1.027	.195	11.0	1190	470	5114	474	790	551
30		1.027	.195	14.0	1190	470	5114	474	742	503
31		1.018	.160	34.5	1190	471	5114	472	663	462
32		1.030	.206	3.5	1190	468	4091	472	550	387
33		1.029	.203	5.5	1186	465	4091	469	514	348
34		1.029	.203	8.0	1188	467	4091	471	456	284
35		1.029	.203	9.5	1188	467	4091	471	419	253
36		1.032	.212	13.5	1187	468	4091	472	397	216
37		1.031	.210	19.5	1182	468	4091	472	378	207
38		1.029	.203	34.5	1190	467	4091	471	336	168
39	25,000	1.398	0.709	3.5	779	429	7955	470	3791	2570
40		1.391	.704	5.0	788	434	7955	475	3621	2396
41		1.405	.715	6.0	781	432	7955	474	3384	2139
42		1.387	.702	8.5	788	433	7955	474	3120	1910
43		1.387	.702	12.5	783	433	7955	474	2812	1609
44		1.386	.701	34.5	783	434	7955	474	2409	1233
45		1.390	.704	3.5	785	426	7692	466	3632	2427
46		1.393	.706	4.0	784	430	7692	471	3473	2275
47		1.400	.711	7.0	778	430	7692	471	3201	1990
48		1.393	.706	7.5	781	430	7692	471	3014	1813
49		1.394	.706	10.0	779	430	7692	471	2768	1568
50		1.398	.709	20.0	787	426	7692	467	2347	1144
51		1.392	.705	34.0	780	426	7692	466	2294	1096
52		1.409	.718	3.5	776	430	7386	472	3328	2116

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ENGINE EQUIPPED WITH VARIABLE-AREA NOZZLE - Continued

control inoperative.

Engine-inlet air flow $W_{a,1}$ (lb/sec)	Fuel flow W_f (lb/hr)	Specific fuel-consumption based on net thrust $W_f/F_{n,s}$ (lb/(hr)(lb thrust))	Fuel-air ratio f/a	Turbine-outlet total temperature, T_6 (°R)	Turbine-outlet total pressure P_6 (lb/sq ft abs.)	Engine total-pressure ratio P_9/P_1	Engine total-temperature ratio T_6/T_1	Run
59.66	3895	1.193	0.0186	1751	2531	1.944	3.656	1
60.64	3660	1.184	.0171	1676	2468	1.883	3.521	2
61.03	3605	1.172	.0169	1650	2433	1.852	3.488	3
60.90	3265	1.182	.0153	1563	2294	1.732	3.297	4
61.03	3015	1.194	.0141	1475	2180	1.625	3.105	5
61.26	2800	1.222	.0130	1405	2070	1.525	2.964	6
61.11	2585	1.330	.0120	1338	1921	1.377	2.817	7
58.63	3375	1.119	.0164	1627	2494	1.922	3.411	8
58.75	3060	1.121	.0148	1521	2361	1.805	3.182	9
58.71	2920	1.114	.0141	1470	2292	1.752	3.082	10
58.59	2545	1.122	.0124	1351	2106	1.601	2.832	11
58.63	2225	1.169	.0108	1248	1945	1.451	2.622	12
58.63	1955	1.271	.0095	1166	1794	1.301	2.439	13
52.86	2585	1.130	.0139	1447	2221	1.717	3.015	14
53.66	2325	1.134	.0123	1345	2089	1.610	2.826	15
53.62	2090	1.128	.0111	1258	1983	1.519	2.648	16
53.60	1950	1.163	.0103	1206	1889	1.444	2.528	17
53.98	1710	1.214	.0090	1121	1782	1.338	2.340	18
53.64	1500	1.345	.0079	1048	1651	1.219	2.188	19
46.20	1920	1.215	.0118	1293	1943	1.498	2.711	20
46.59	1757	1.233	.0107	1228	1851	1.436	2.585	21
46.13	1590	1.254	.0098	1157	1767	1.366	2.431	22
46.08	1445	1.284	.0089	1095	1689	1.302	2.300	23
46.56	1248	1.436	.0076	1009	1570	1.199	2.120	24
46.46	1131	1.543	.0069	964	1518	1.157	2.021	25
37.03	1301	1.459	.0100	1165	1641	1.289	2.458	26
36.64	1214	1.634	.0094	1109	1577	1.231	2.345	27
37.11	1102	1.665	.0084	1047	1531	1.201	2.214	28
37.13	1053	1.911	.0080	1015	1482	1.162	2.141	29
37.13	968	1.924	.0074	984	1453	1.137	2.076	30
38.02	897	1.942	.0067	936	1389	1.097	1.979	31
23.97	941	2.432	.0111	1151	1404	1.122	2.439	32
24.91	936	2.690	.0106	1131	1402	1.120	2.412	33
25.82	872	3.070	.0095	1083	1368	1.088	2.299	34
24.93	836	3.304	.0095	1052	1340	1.072	2.234	35
25.87	793	3.671	.0087	1017	1323	1.054	2.155	36
24.77	758	3.662	.0086	999	1303	1.039	2.117	37
25.18	758	4.512	.0085	988	1297	1.035	2.098	38
54.54	3445	1.340	0.0180	1737	2272	1.948	3.680	39
54.88	3230	1.348	.0168	1654	2176	1.847	3.468	40
55.03	2925	1.367	.0151	1557	2056	1.730	3.271	41
54.51	2670	1.398	.0139	1463	1923	1.597	3.073	42
54.17	2425	1.507	.0127	1364	1770	1.449	2.866	43
52.97	2155	1.748	.0116	1288	1601	1.217	2.706	44
54.56	3190	1.314	.0167	1632	2217	1.903	3.487	45
53.77	2980	1.310	.0158	1580	2145	1.830	3.340	46
53.92	2635	1.324	.0139	1460	1993	1.687	3.087	47
53.90	2465	1.360	.0130	1397	1900	1.597	2.953	48
53.77	2235	1.425	.0118	1310	1762	1.454	2.770	49
53.92	1975	1.726	.0104	1217	1586	1.235	2.595	50
54.08	1950	1.779	.0102	1211	1560	1.189	2.588	51
53.46	2790	1.319	.0149	1524	2091	1.778	3.215	52

TABLE I - ENGINE PERFORMANCE DATA FOR J47D (RX1-1) TURBOJET

NACA

(b) With electronic control

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Run	Altitude (ft)	Ram pressure ratio P_1/p_0	Flight Mach number M_0	Nozzle position (d.-c. volts)	Tunnel static pressure p_0 (lb/sq ft abs.)	Equivalent ambient-air temperature t_e (°R)	Engine speed N (rpm)	Engine-inlet indicated temperature $T_{i,1}$ (°R)	Jet thrust $F_{j,s}$ (lb)	Net thrust $F_{n,s}$ (lb)
53	25,000	1.397	0.708	5.5	781	425	7386	466	3157	1970
54		1.402	.712	6.5	779	425	7386	466	2984	1794
55		1.396	.707	8.5	781	425	7386	466	2734	1575
56		1.389	.702	10.0	786	428	7386	468	2521	1366
57		1.384	.699	34.5	787	430	7386	470	2096	940
58	45,000	1.022	0.176	9.0	317	440	7955	441	942	848
59		1.022	.176	11.0	313	440	7955	441	844	752
60		1.016	.152	12.5	313	440	7955	440	762	684
61		1.026	.192	15.5	303	440	7955	441	694	599
62		1.030	.206	33.0	303	439	7955	441	625	521
63		1.036	.226	7.0	306	436	7692	439	956	841
64		1.023	.180	8.5	306	438	7692	440	835	745
65		1.029	.203	11.0	308	434	7692	436	780	675
66		1.029	.203	14.2	306	437	7692	440	718	617
67		1.029	.203	34.5	308	436	7692	439	603	500
68		1.032	.212	6.0	310	434	7386	437	934	826
69		1.026	.192	6.8	310	438	7386	440	850	756
70		1.029	.203	8.0	310	434	7386	437	798	697
71		1.029	.203	10.0	310	433	7386	436	722	621
72		1.029	.203	10.0	310	432	7386	435	724	621
73		1.026	.192	12.0	304	433	7386	435	658	564
74		1.030	.206	12.5	303	434	7386	437	664	562
75		1.033	.216	34.5	306	436	7386	439	550	440
76		1.029	.203	3.5	308	437	6993	440	927	830
77		1.026	.192	5.0	307	440	6993	442	854	762
78		1.029	.203	9.0	307	437	6993	440	642	543
79		1.030	.206	12.5	304	439	6993	442	609	511
80		1.029	.203	34.5	306	439	6993	442	471	372
81		1.032	.212	3.5	310	430	6643	433	809	706
82		1.029	.203	6.0	308	434	6643	437	669	574
83		1.023	.180	8.5	303	433	6643	435	601	521
84		1.030	.206	13.0	301	433	6643	436	514	421
85		1.023	.180	13.5	306	435	6643	437	503	420
86		1.033	.216	34.5	303	432	6643	435	419	317
87		1.033	.216	3.5	303	432	6294	435	665	574
88		1.029	.203	4.0	308	436	6294	439	639	550
89		1.026	.192	10.0	308	434	6294	436	497	413
90		1.030	.206	16.0	296	434	6294	437	384	298
91		1.022	.176	34.5	315	434	6294	436	398	316
92		1.029	.203	3.5	306	442	5944	445	530	449
93		1.023	.180	6.0	308	435	5944	437	---	---
94		1.023	.180	6.5	307	438	5944	440	469	397
95		1.026	.192	9.0	307	435	5944	437	413	336
96		1.026	.192	14.0	306	438	5944	440	353	274
97		1.023	.180	19.0	308	445	5944	447	310	237
98		1.023	.180	34.5	308	440	5944	442	296	221
99		1.035	.223	3.5	310	437	5455	440	436	356
100		1.026	.192	5.0	313	438	5455	440	398	326
101		1.026	.192	7.0	310	438	5455	440	360	290
102		1.019	.164	10.0	313	440	5455	441	263	203
103		1.029	.203	13.5	313	438	5455	441	253	177
104		1.029	.203	33.0	310	438	5455	441	225	149

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ENGINE EQUIPPED WITH VARIABLE-AREA NOZZLE - Concluded

inoperative - Concluded.

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Engine-inlet air flow $W_{a,1}$ (lb/sec)	Fuel flow W_f (lb/hr)	Specific fuel-consumption based on net thrust $W_f/F_{n,s}$ (lb/(hr)(lb thrust))	Fuel-air ratio f/a	Turbine-outlet total temperature, T_6 (°R)	Turbine-outlet total pressure P_6 (lb/sq ft abs.)	Engine total-pressure ratio P_9/P_1	Engine total-temperature ratio T_6/T_1	Run
53.39	2560	1.299	0.0136	1436	2027	1.726	3.068	53
53.16	2335	1.302	.0125	1355	1917	1.620	2.895	54
52.16	2125	1.349	.0116	1280	1799	1.509	2.735	55
52.18	1945	1.424	.0106	1207	1699	1.401	2.568	56
52.43	1665	1.771	.0090	1123	1474	1.135	2.379	57
16.66	1139	1.343	0.0193	1775	794	2.250	4.007	58
16.28	1010	1.343	.0175	1670	726	2.063	3.770	59
16.03	933	1.364	.0164	1576	680	1.934	3.566	60
15.50	866	1.446	.0158	1523	623	1.772	3.438	61
15.79	812	1.559	.0145	1466	564	1.564	3.309	62
16.01	1093	1.300	.0195	1772	678	1.968	4.027	63
15.62	954	1.281	.0174	1633	632	1.843	3.703	64
16.32	902	1.336	.0157	1540	595	1.694	3.516	65
15.61	830	1.345	.0151	1483	557	1.584	3.363	66
16.01	745	1.490	.0132	1370	528	1.416	3.114	67
16.10	1049	1.270	.0186	1713	677	1.956	3.911	68
15.38	950	1.257	.0176	1628	639	1.852	3.692	69
15.78	890	1.277	.0160	1560	618	1.781	3.562	70
15.80	825	1.329	.0148	1454	574	1.636	3.327	71
16.09	815	1.312	.0144	1454	584	1.658	3.335	72
15.39	775	1.374	.0143	1403	552	1.587	3.218	73
15.64	775	1.379	.0141	1393	551	1.587	3.180	74
15.96	672	1.527	.0119	1278	504	1.361	2.905	75
15.03	1042	1.255	.0198	1765	707	2.088	4.003	76
15.01	950	1.247	.0180	1664	664	1.969	3.757	77
15.40	734	1.352	.0135	1352	549	1.579	3.063	78
14.92	684	1.339	.0130	1309	535	1.537	2.955	79
15.31	600	1.613	.0111	1184	488	1.340	2.673	80
15.30	912	1.292	.0169	1600	663	1.934	3.687	81
14.78	778	1.355	.0150	1430	585	1.713	3.265	82
13.98	692	1.328	.0140	1322	531	1.574	3.032	83
14.26	618	1.468	.0123	1209	480	1.390	2.767	84
14.40	600	1.429	.0118	1207	491	1.399	2.756	85
14.74	540	1.703	.0104	1124	452	1.252	2.578	86
13.32	770	1.341	.0164	1476	585	1.751	3.385	87
13.79	756	1.375	.0156	1439	585	1.722	3.270	88
13.79	598	1.448	.0123	1208	498	1.443	2.764	89
13.19	513	1.721	.0110	1103	440	1.298	2.518	90
14.60	503	1.592	.0097	1069	448	1.233	2.446	91
12.55	660	1.470	.0149	1381	539	1.610	3.096	92
12.67	623	-----	-----	----	---	-----	-----	93
12.58	602	1.516	.0135	1264	501	1.490	2.866	94
12.68	544	1.619	.0121	1168	470	1.384	2.667	95
12.95	490	1.788	.0107	1078	429	1.248	2.444	96
12.53	460	1.941	.0104	1044	425	1.229	2.330	97
12.96	453	2.050	.0099	1020	417	1.203	2.302	98
11.31	572	1.607	.0143	1323	565	1.679	3.000	99
11.77	558	1.712	.0134	1273	547	1.620	2.887	100
11.41	517	1.783	.0128	1194	504	1.506	2.707	101
11.38	471	2.320	.0116	1121	466	1.382	2.536	102
11.82	433	2.446	.0103	1047	435	1.264	2.369	103
11.84	407	2.732	.0097	1002	409	1.194	2.267	104

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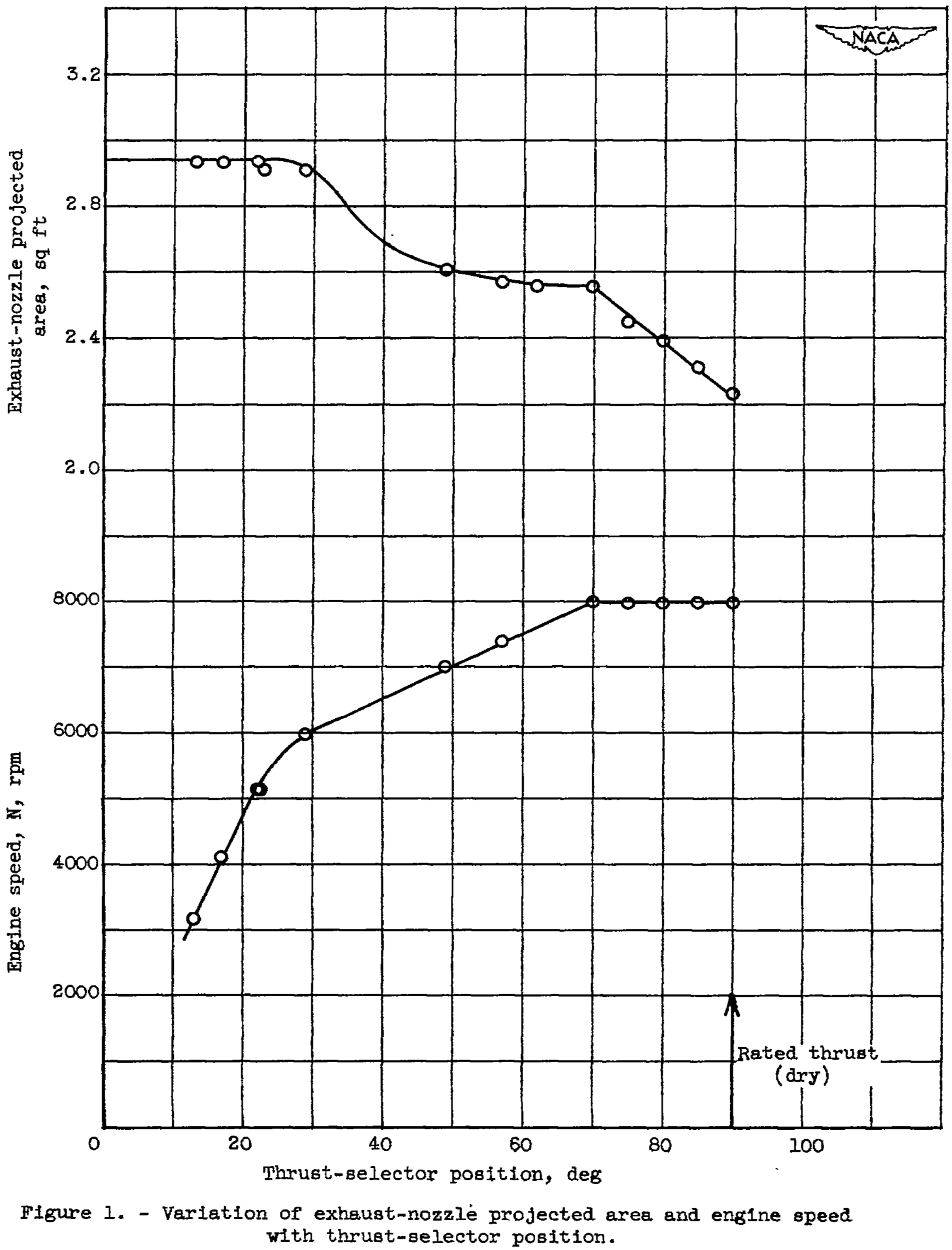


Figure 1. - Variation of exhaust-nozzle projected area and engine speed with thrust-selector position.

[redacted]

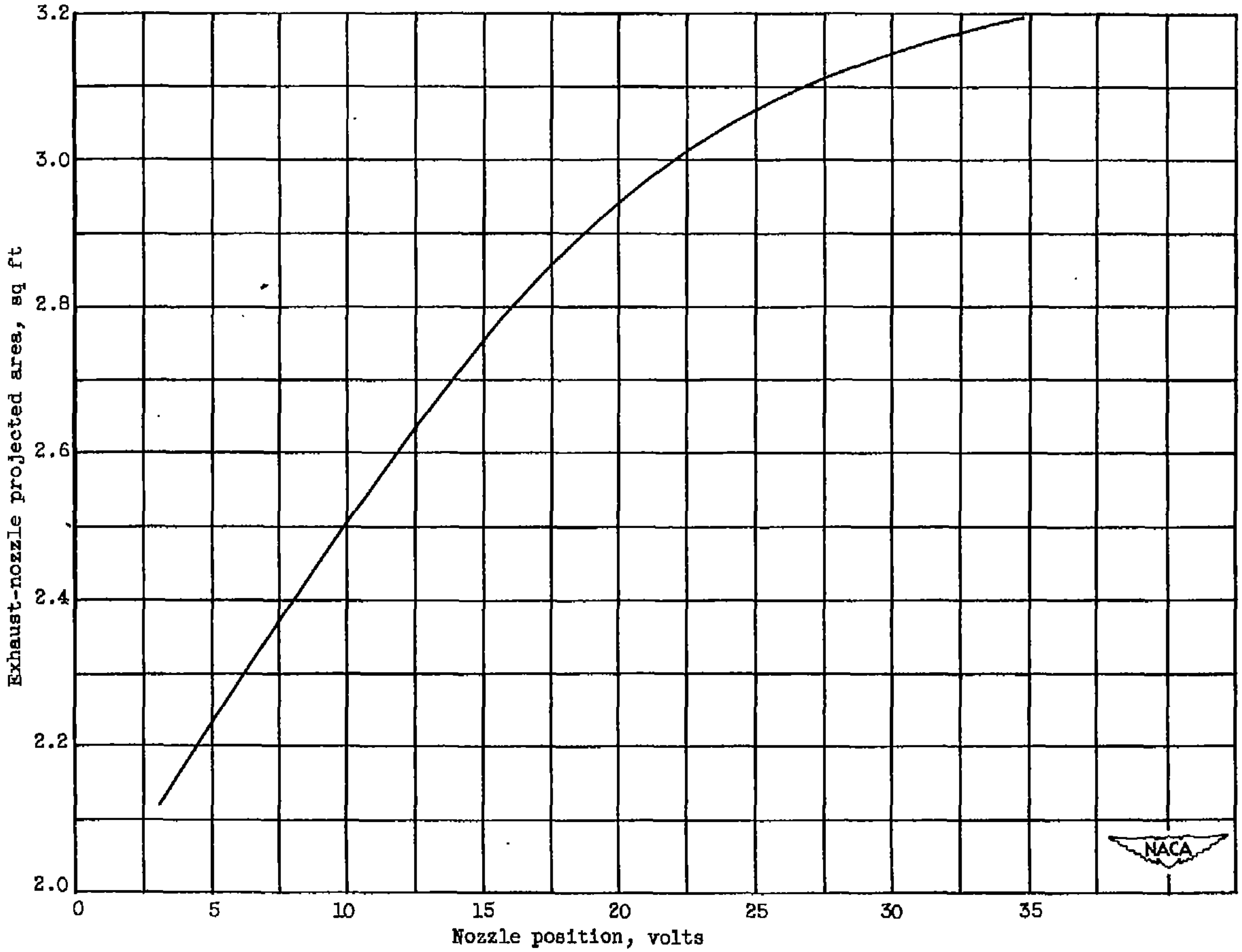


Figure 2. - Calibration curve giving projected nozzle area as function of nozzle position.

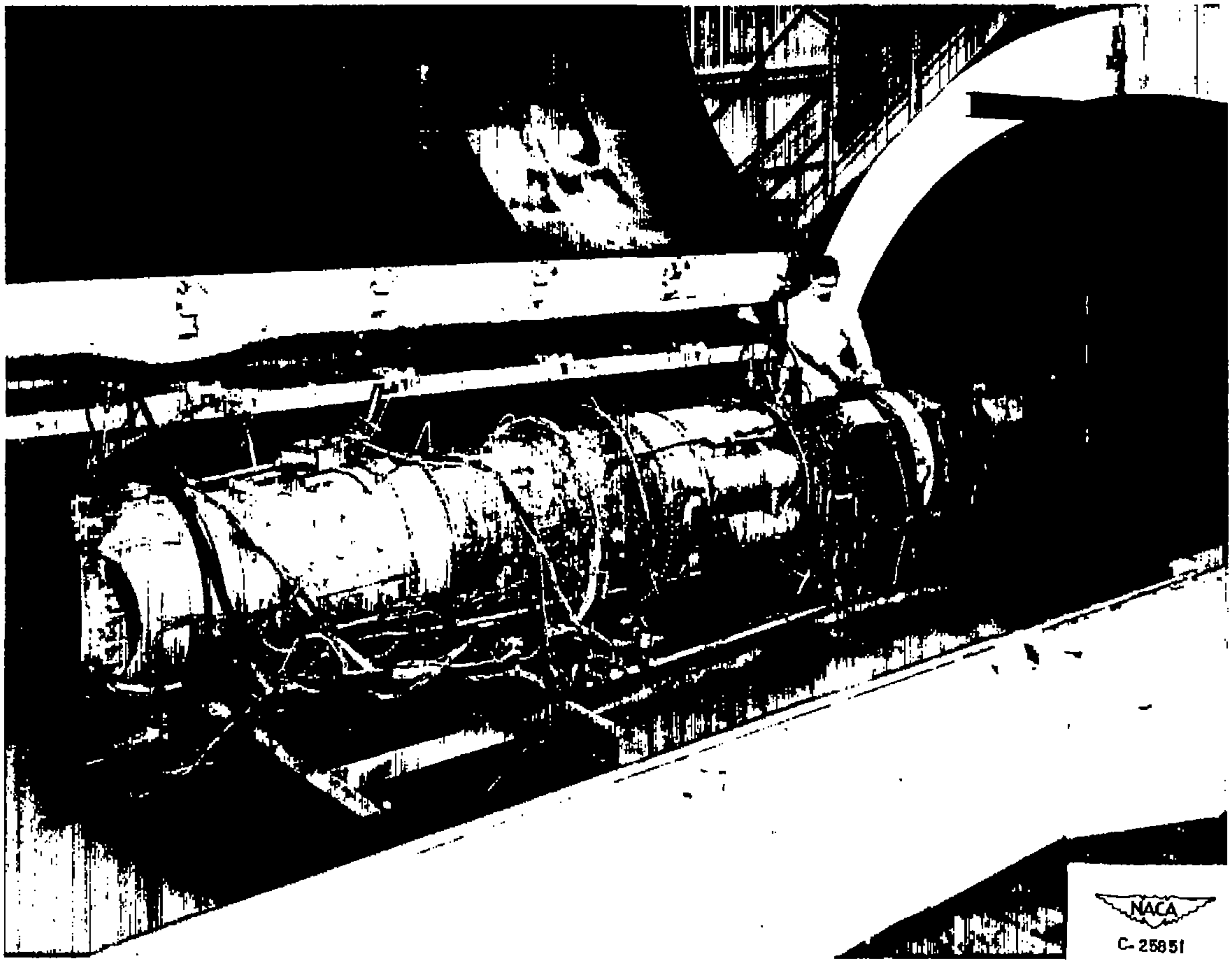
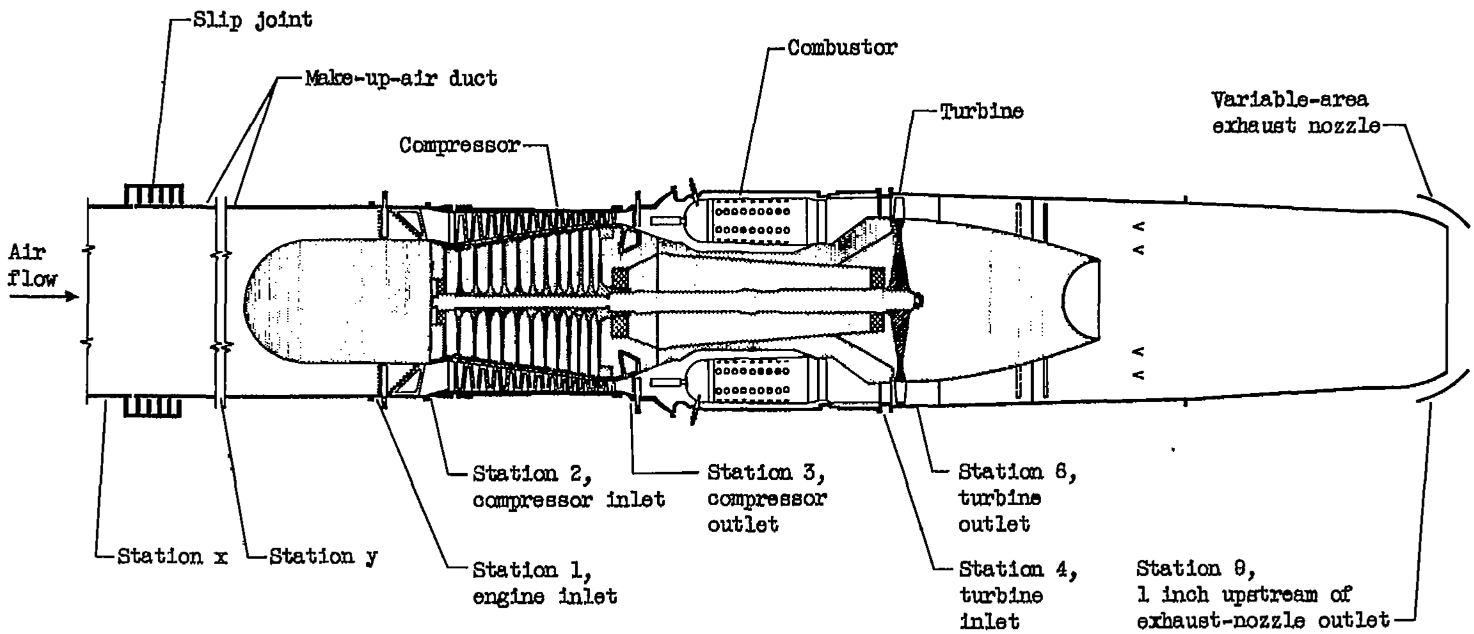


Figure 3. - J47D (RX1-1) turbojet engine installed in test section of altitude wind tunnel.



Station	Total-pressure tubes	Static-pressure tubes	Wall static-pressure orifices	Thermo-couples
x	0	0	4	0
y	1	2	4	2
1	32	8	5	4
2	6	0	2	0
3	20	0	4	6
4	5	0	0	0
6	30	0	4	24
8	20	3	4	10

Figure 4. - Cross section of engine installation showing stations at which instrumentation was installed.

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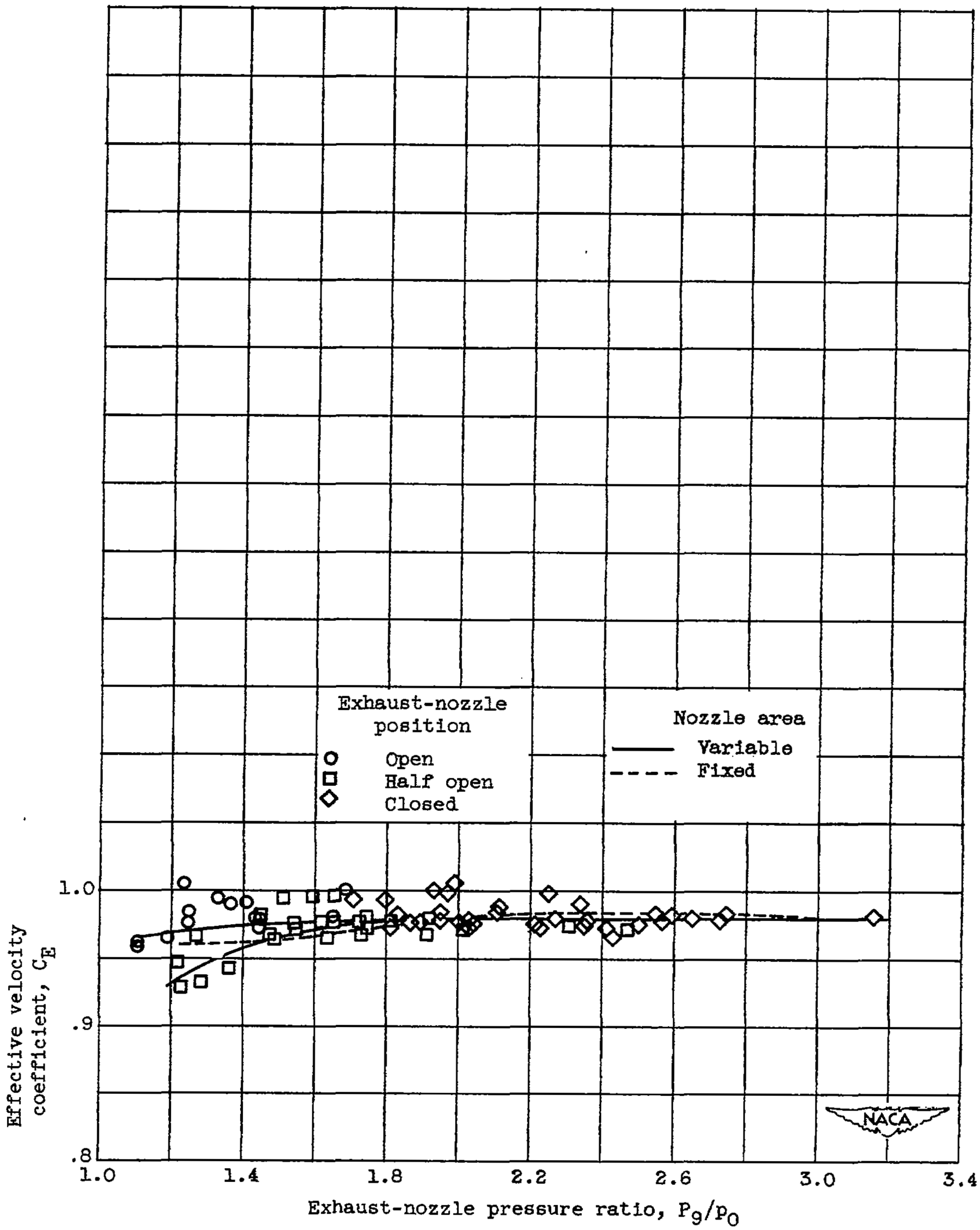

Figure 5. - Variation of exhaust-nozzle effective velocity coefficient with exhaust-nozzle pressure ratio.

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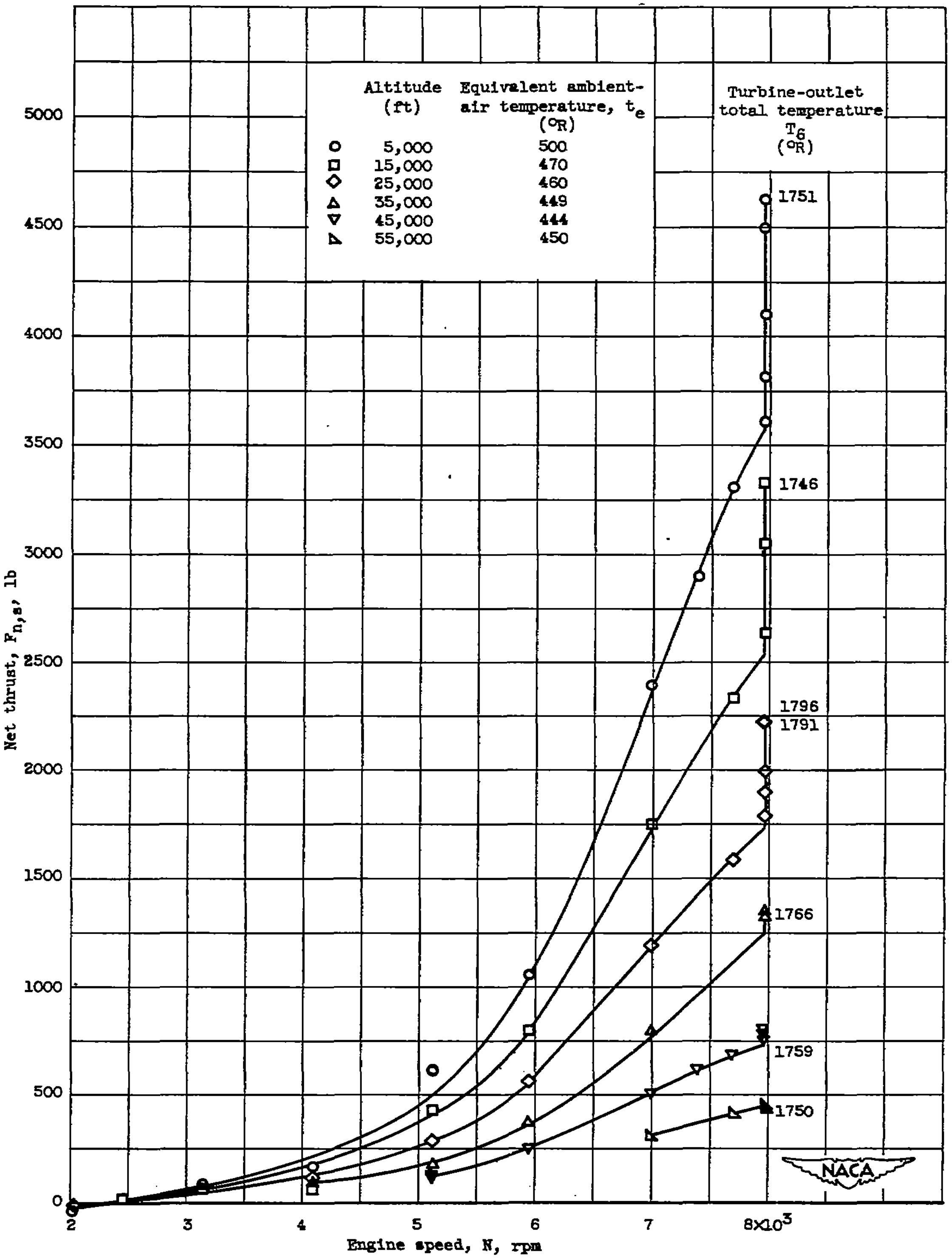
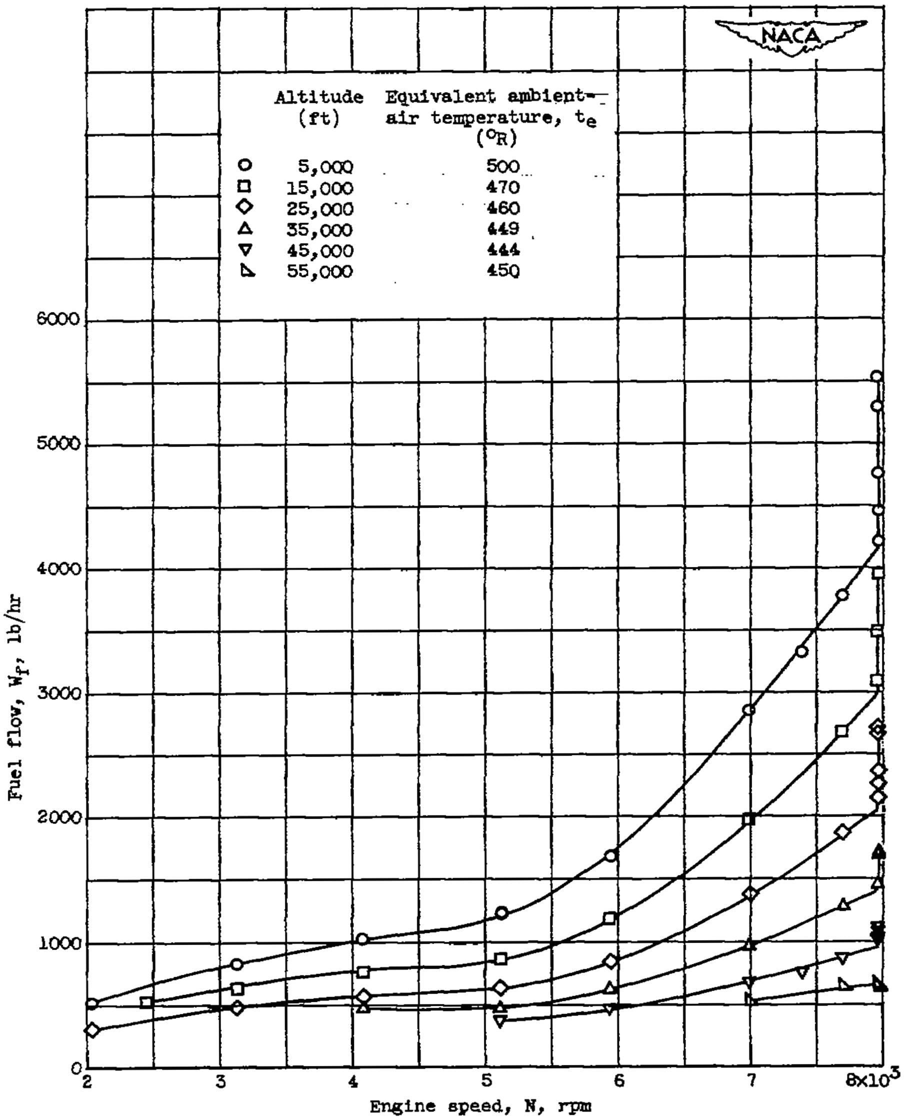


(a) Net thrust.

Figure 6. - Effect of altitude on variation of engine performance with engine speed at flight Mach number of 0.19.



(b) Fuel flow.

Figure 6. - Continued. Effect of altitude on variation of engine performance with engine speed at flight Mach number of 0.19.

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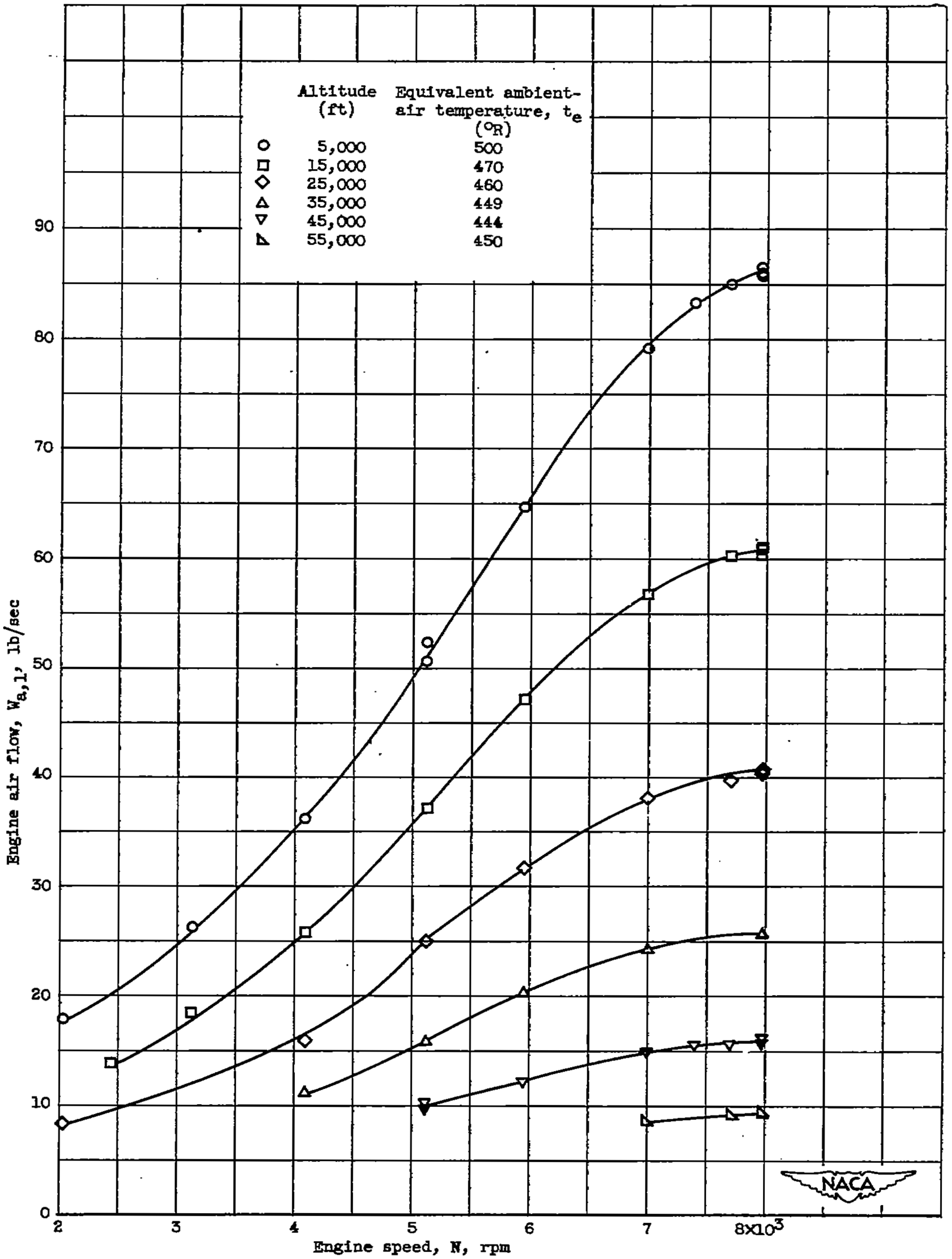

(c) Engine air flow.

Figure 6. - Continued. Effect of altitude on variation of engine performance with engine speed at flight Mach number of 0.19.

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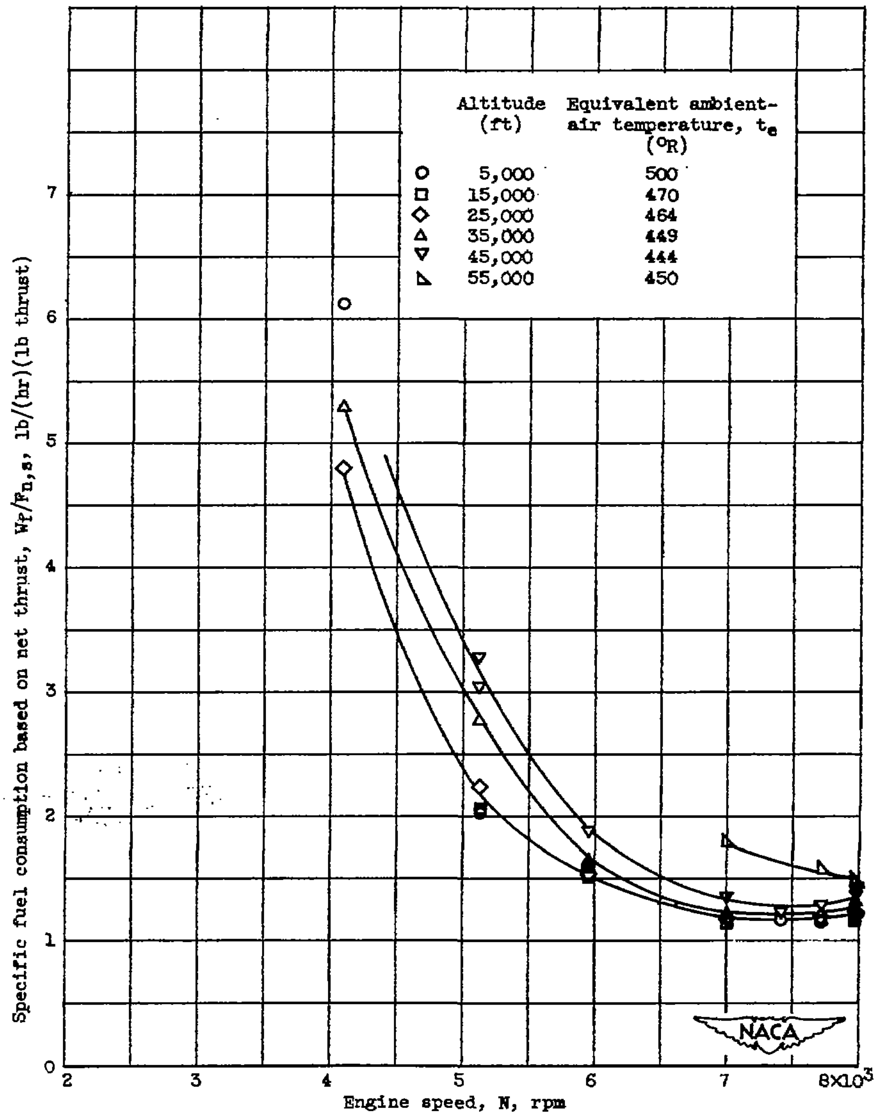

(d) Specific fuel consumption.

Figure 6. - Continued. Effect of altitude on variation of engine performance with engine speed at flight Mach number of 0.19.

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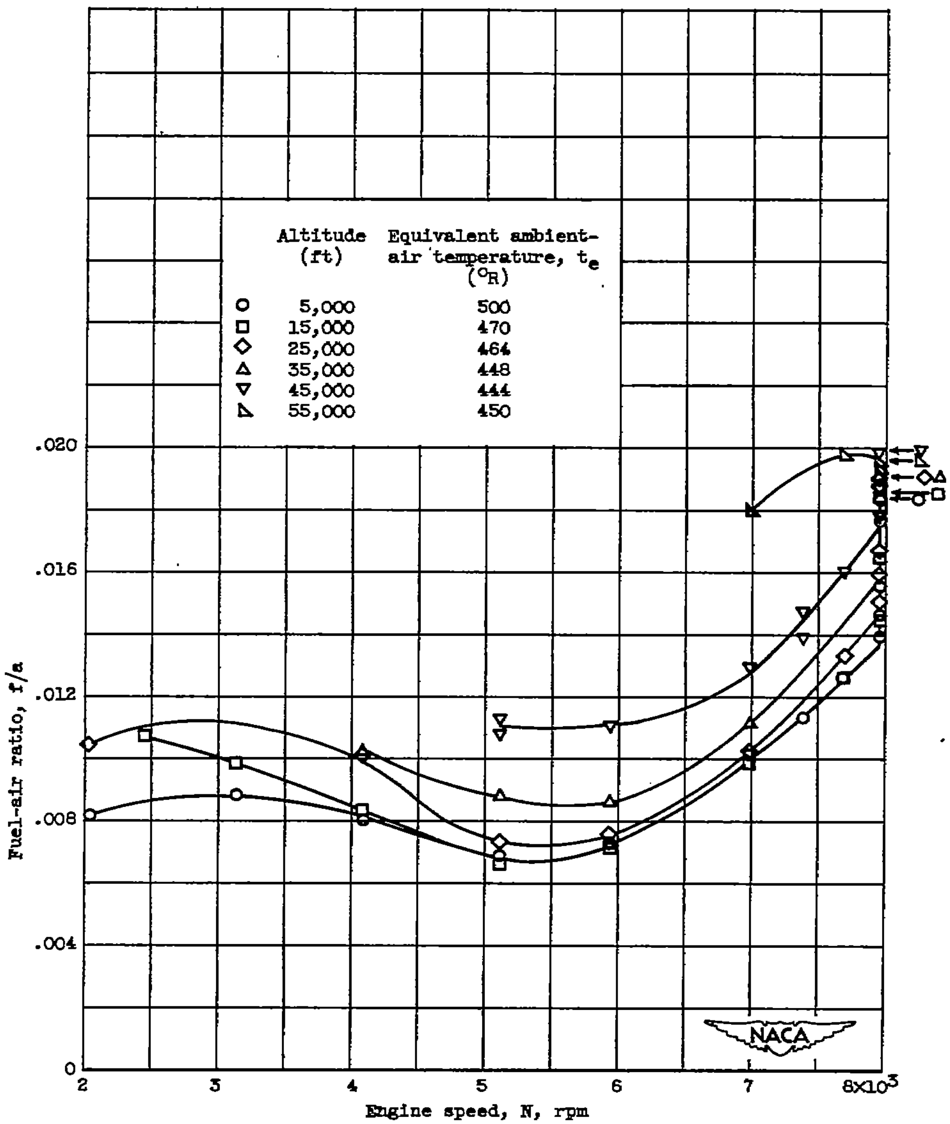
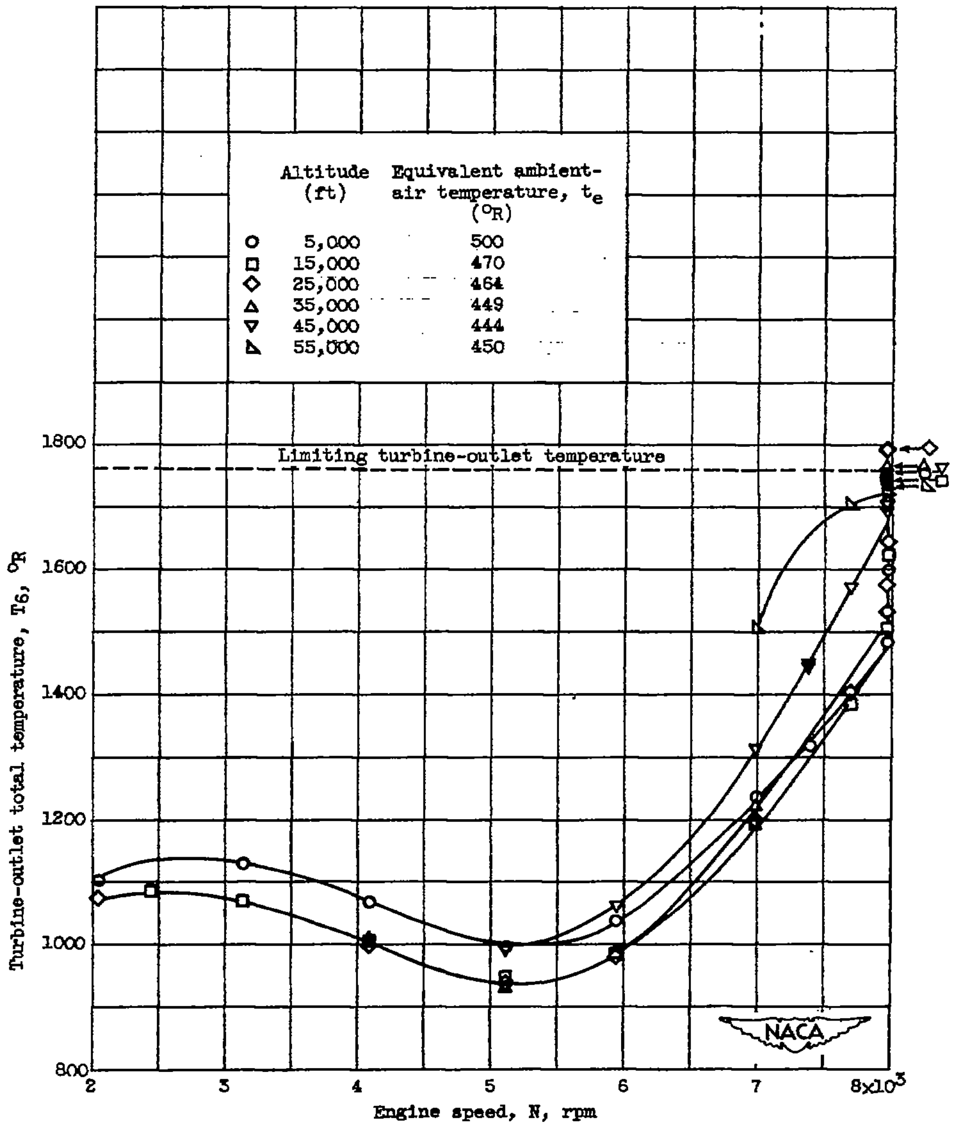


(e) Fuel-air ratio.

Figure 6. - Continued. Effect of altitude on variation of engine performance with engine speed at flight Mach number of 0.19.

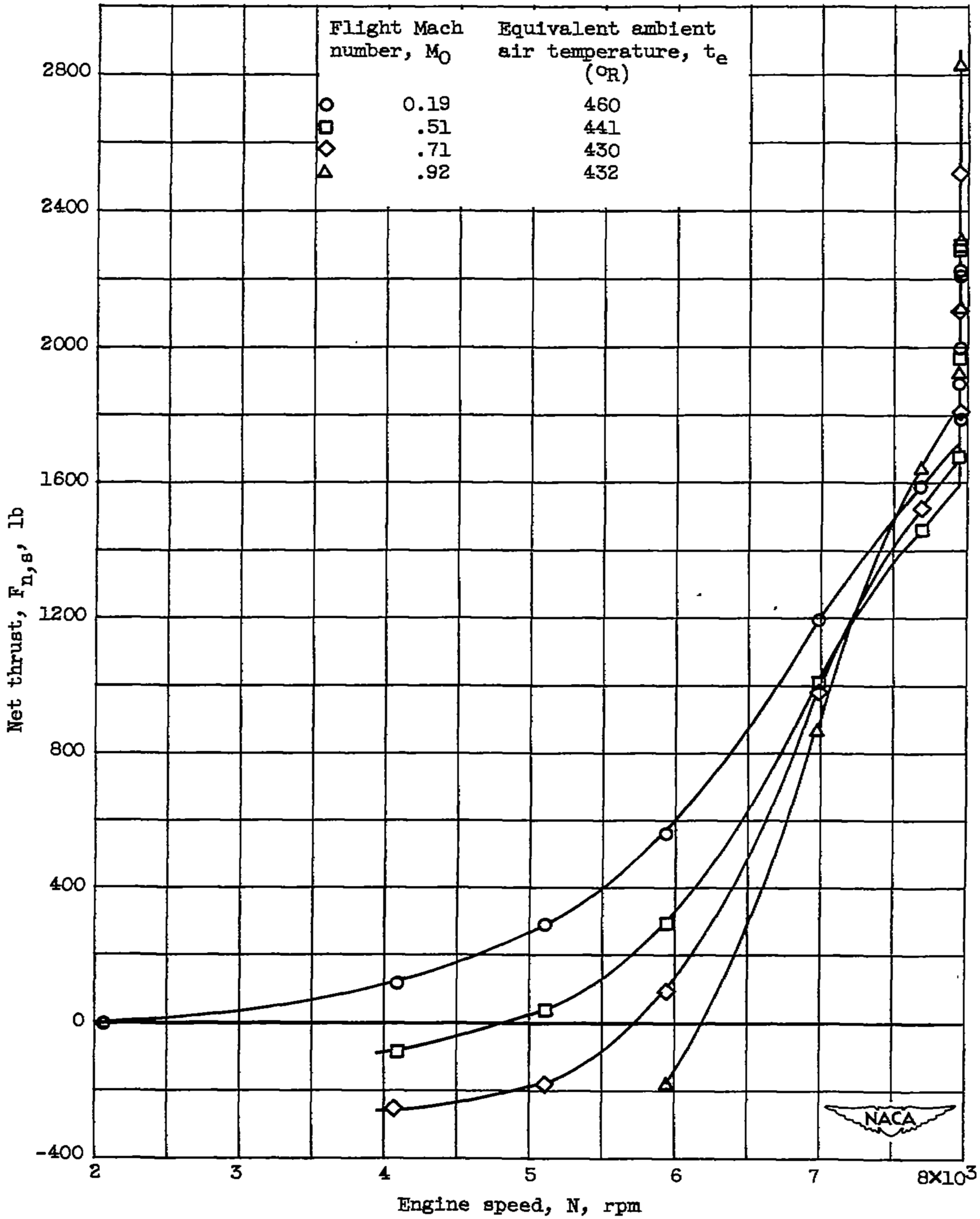
2135



(f) Turbine-outlet gas temperature.

Figure 6. - Concluded. Effect of altitude on variation of engine performance with engine speed at flight Mach number of 0.19.

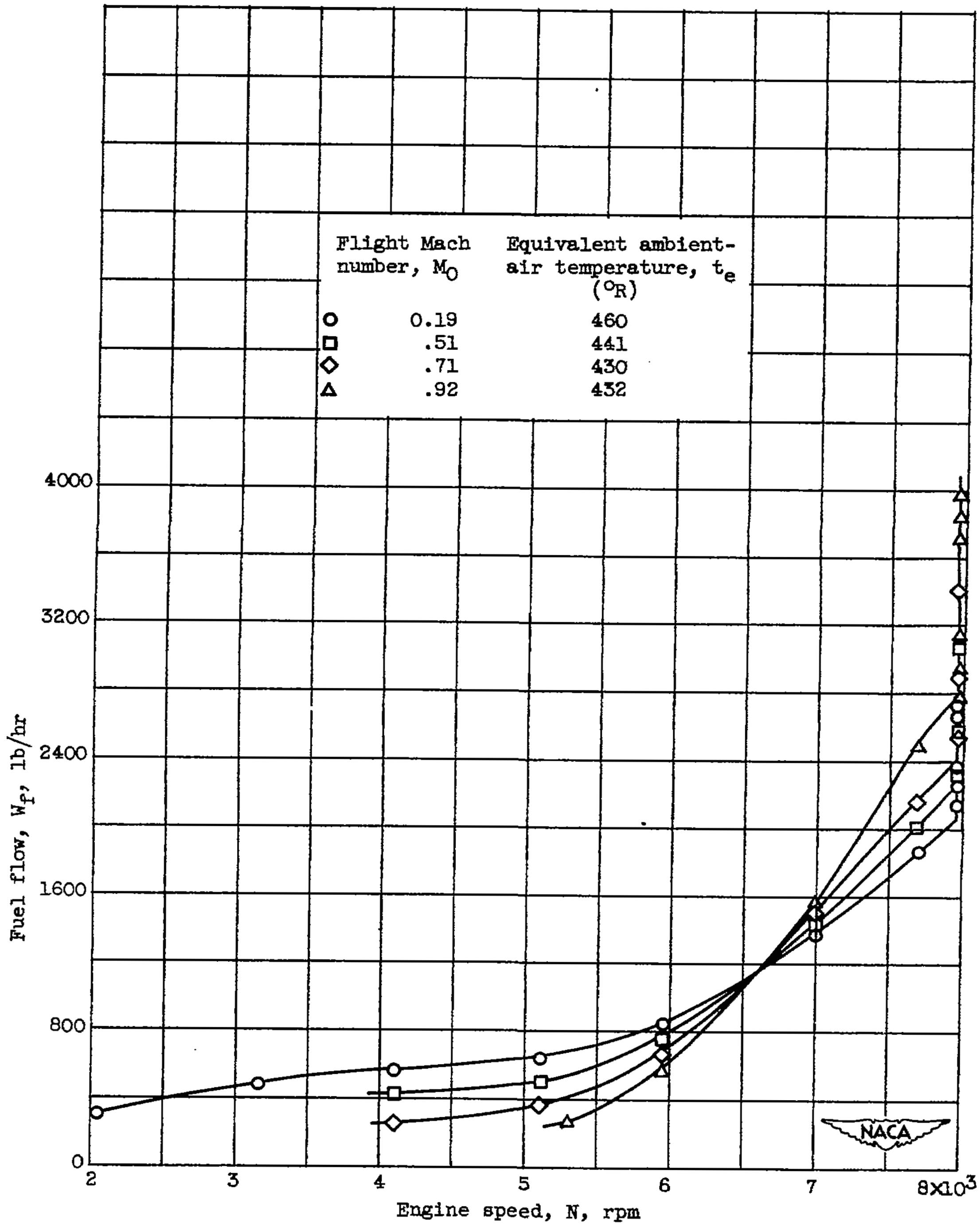
2135



(a) Net thrust.

Figure 7. - Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

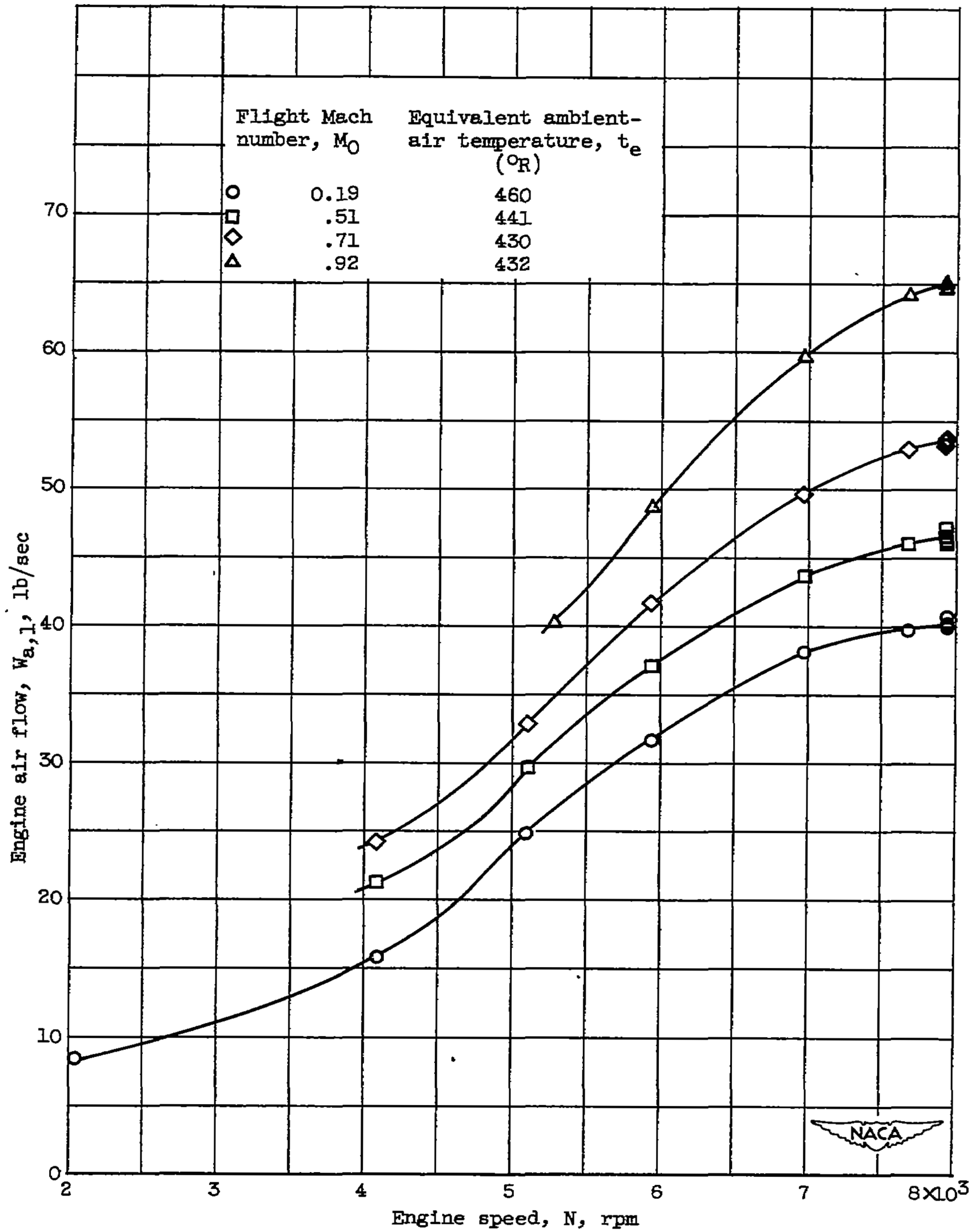
2135



(b) Fuel flow.

Figure 7. - Continued. Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

2135



(c) Engine air flow.

Figure 7. - Continued. Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

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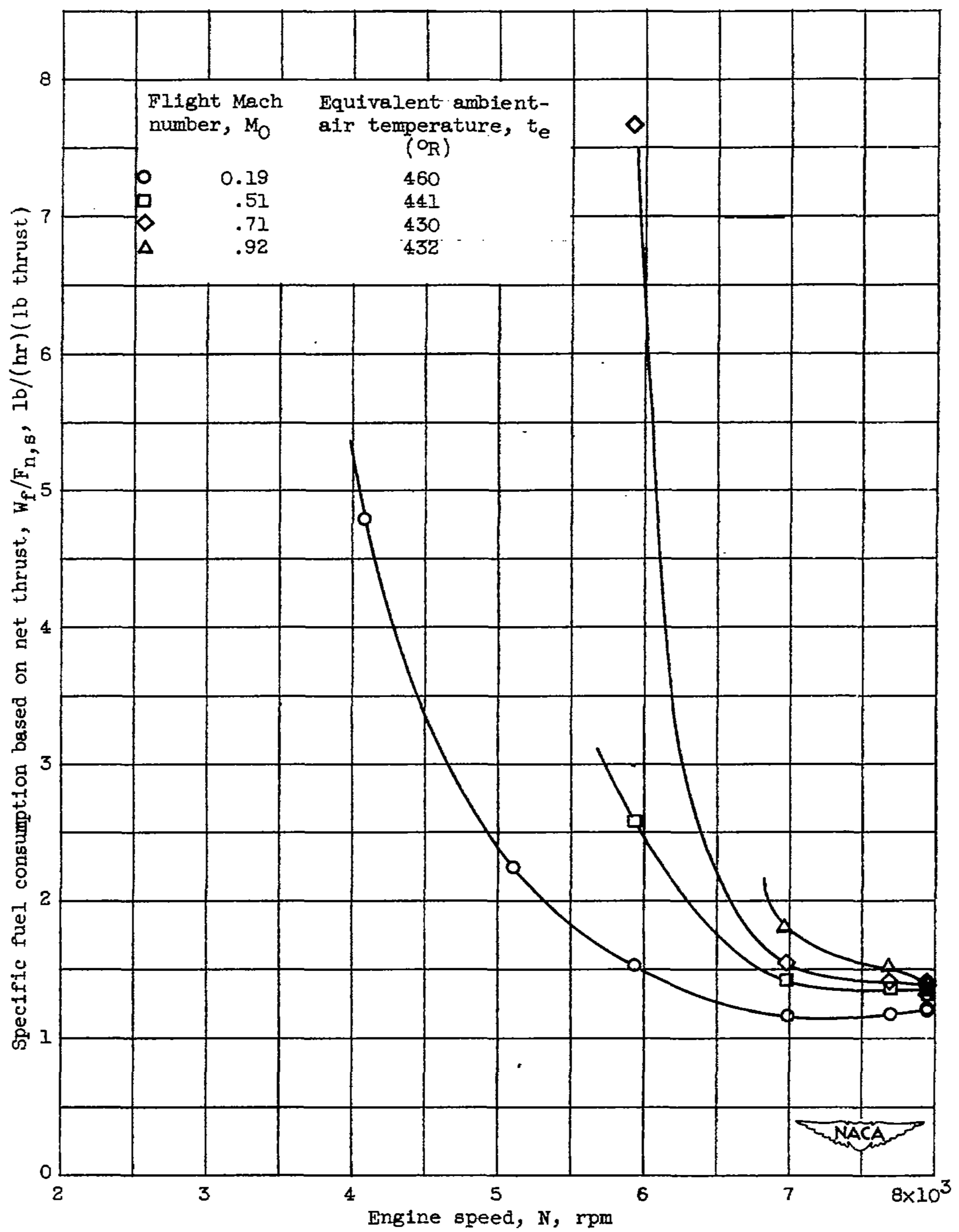
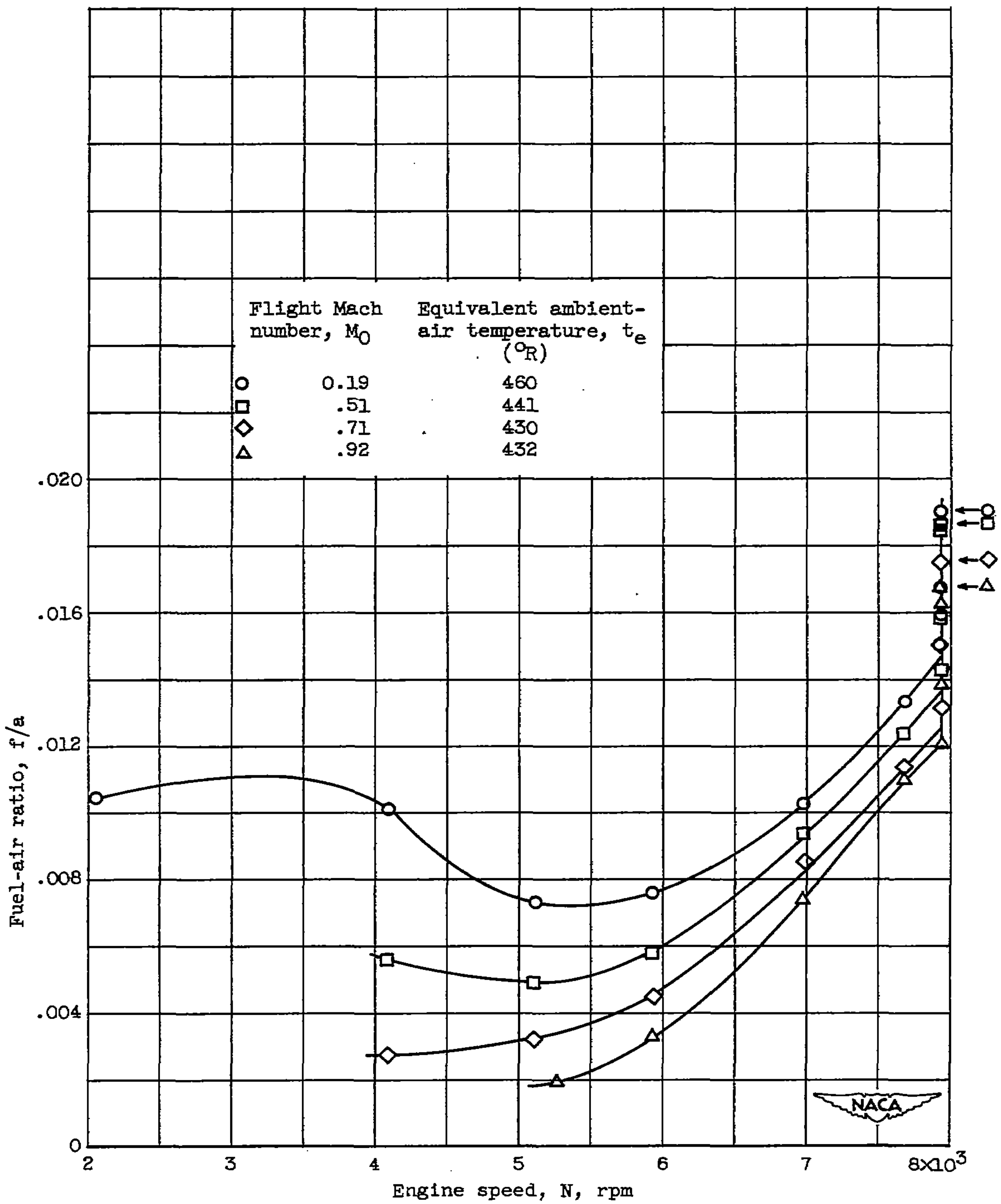
2135



(d) Specific fuel consumption.

Figure 7. - Continued. Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

2135

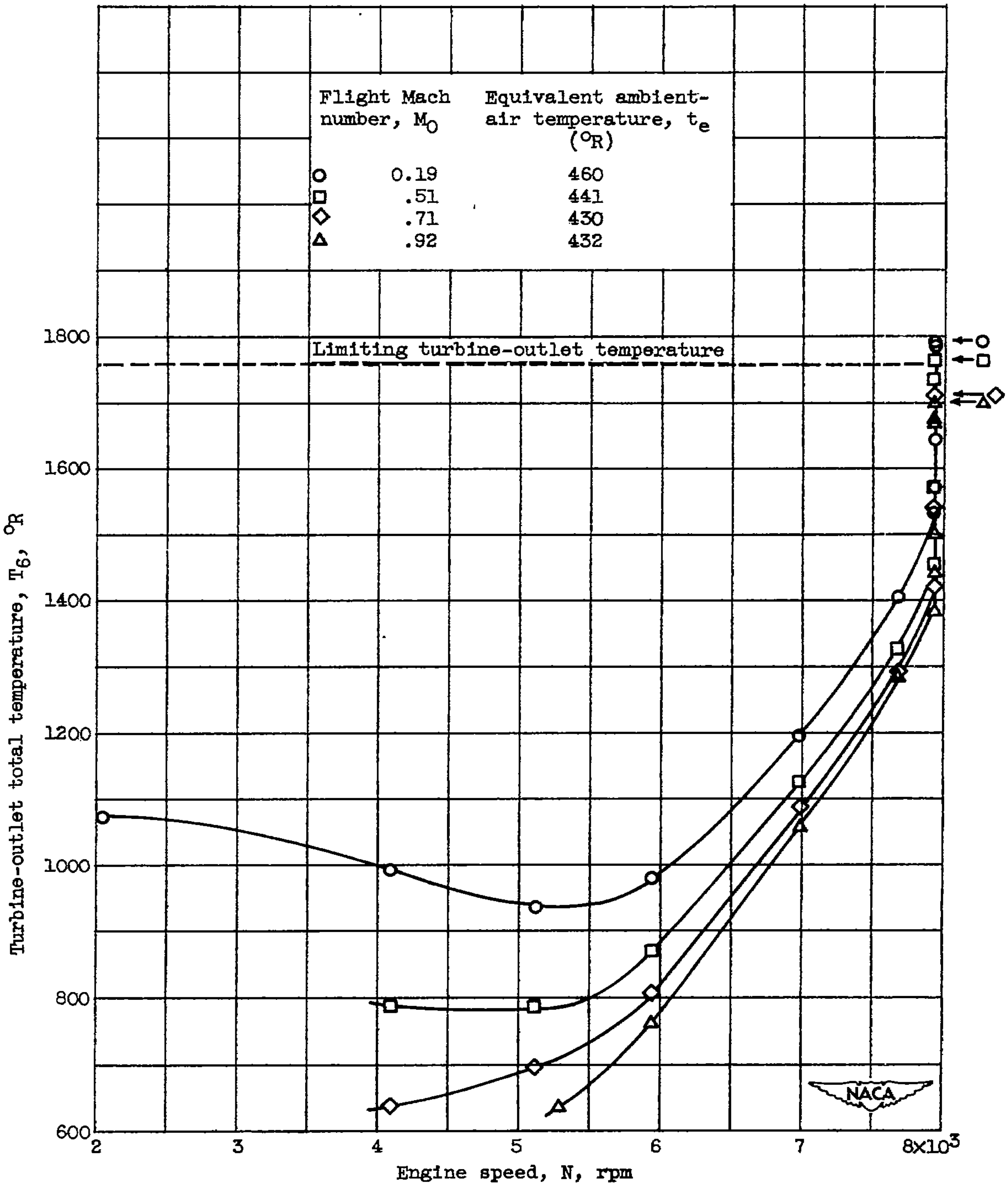


(e) Fuel-air ratio.

Figure 7. - Continued. Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

[redacted]

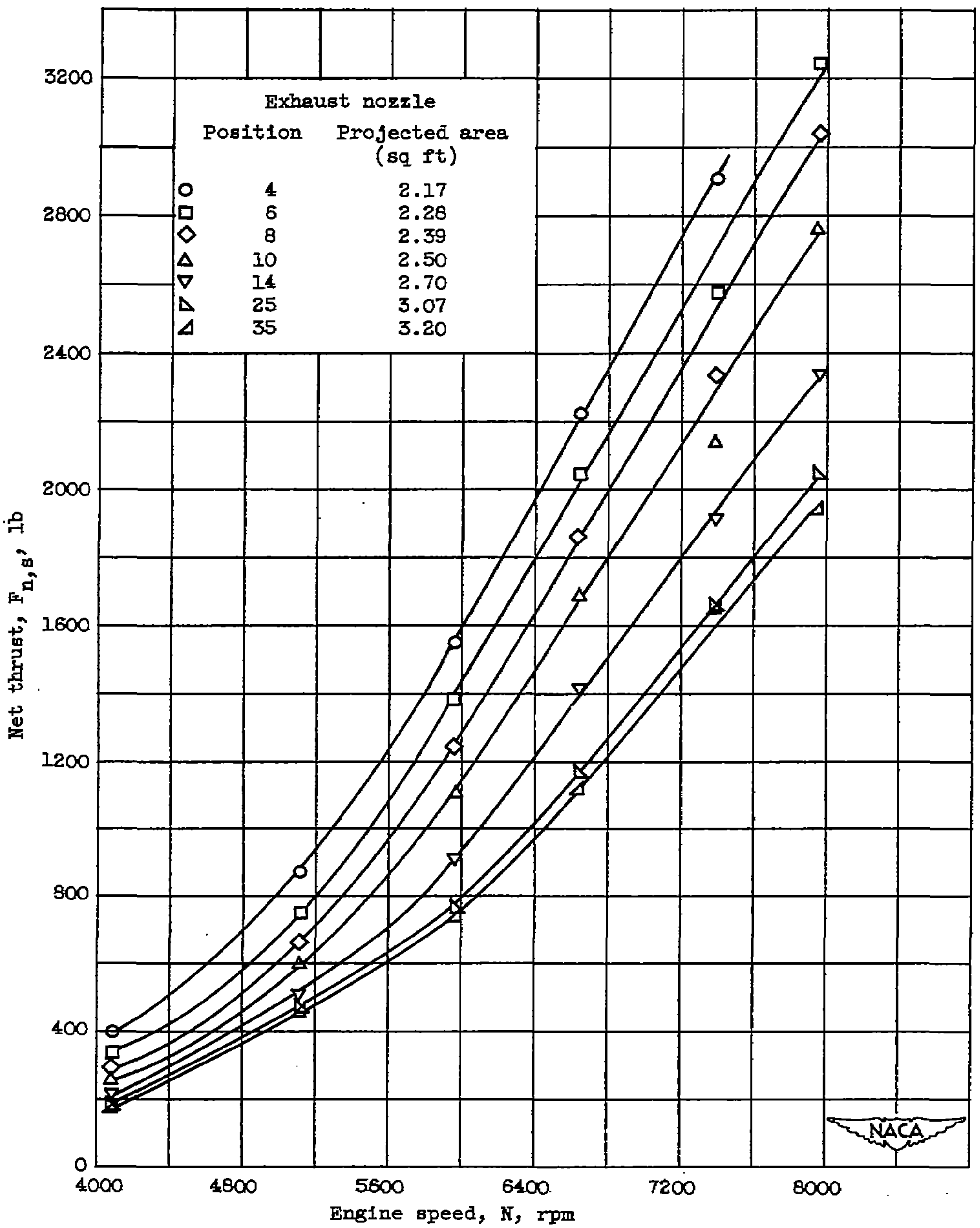
2135



(f) Turbine-outlet gas temperature.

Figure 7. - Concluded. Effect of flight Mach number on variation of engine performance with engine speed at altitude of 25,000 feet.

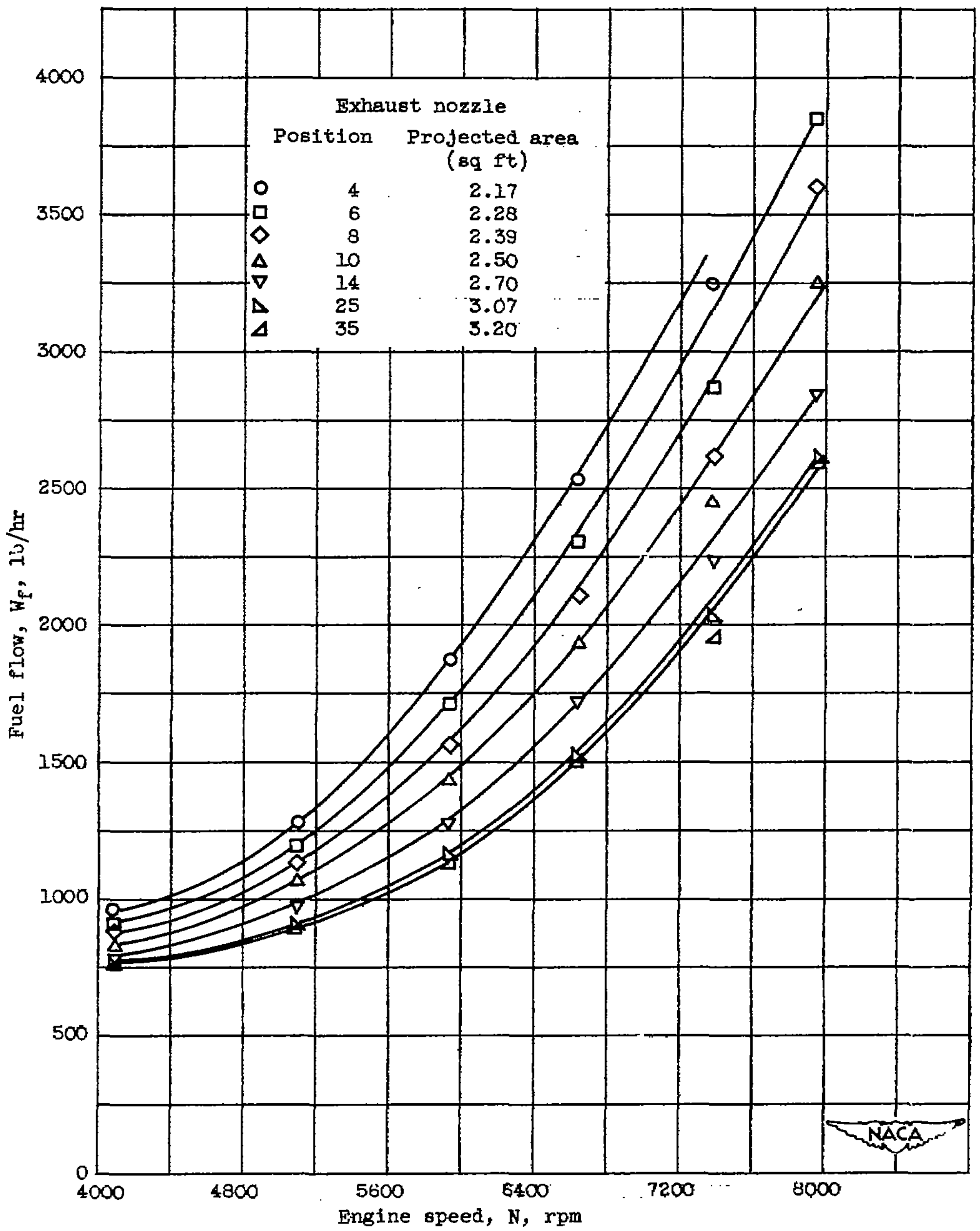
2135



(a) Net thrust

Figure 8. - Effect of exhaust-nozzle area on engine performance at altitude of 15,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 471° R.

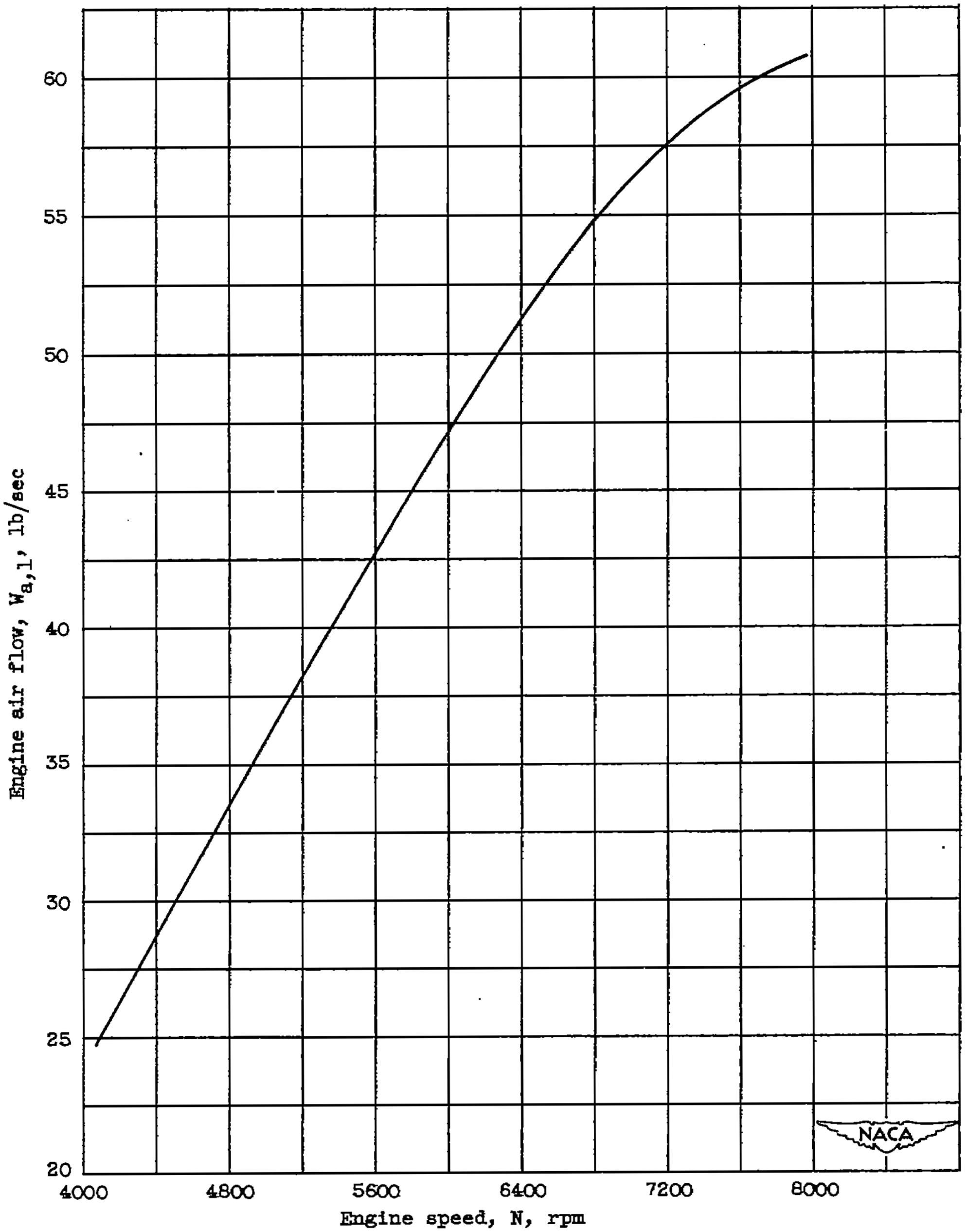
2135



(b) Fuel flow.

Figure 8. - Continued. Effect of exhaust-nozzle area on engine performance at altitude of 15,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 471° R.

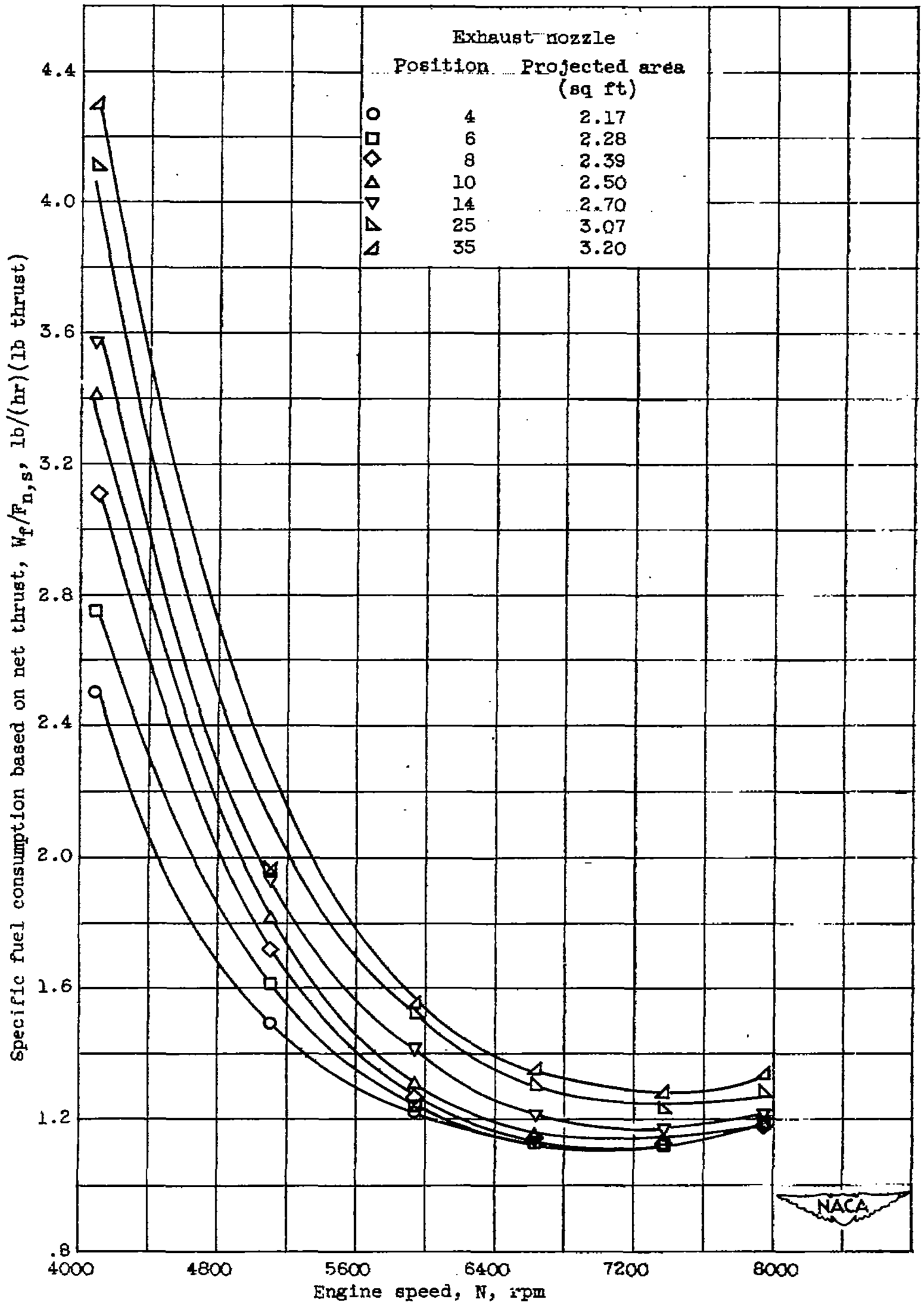
2135



(c) Engine air flow. (Curve valid for all nozzle positions.)

Figure 8. - Continued. Effect of exhaust-nozzle area on engine performance at altitude of 15,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 471° R.

2135

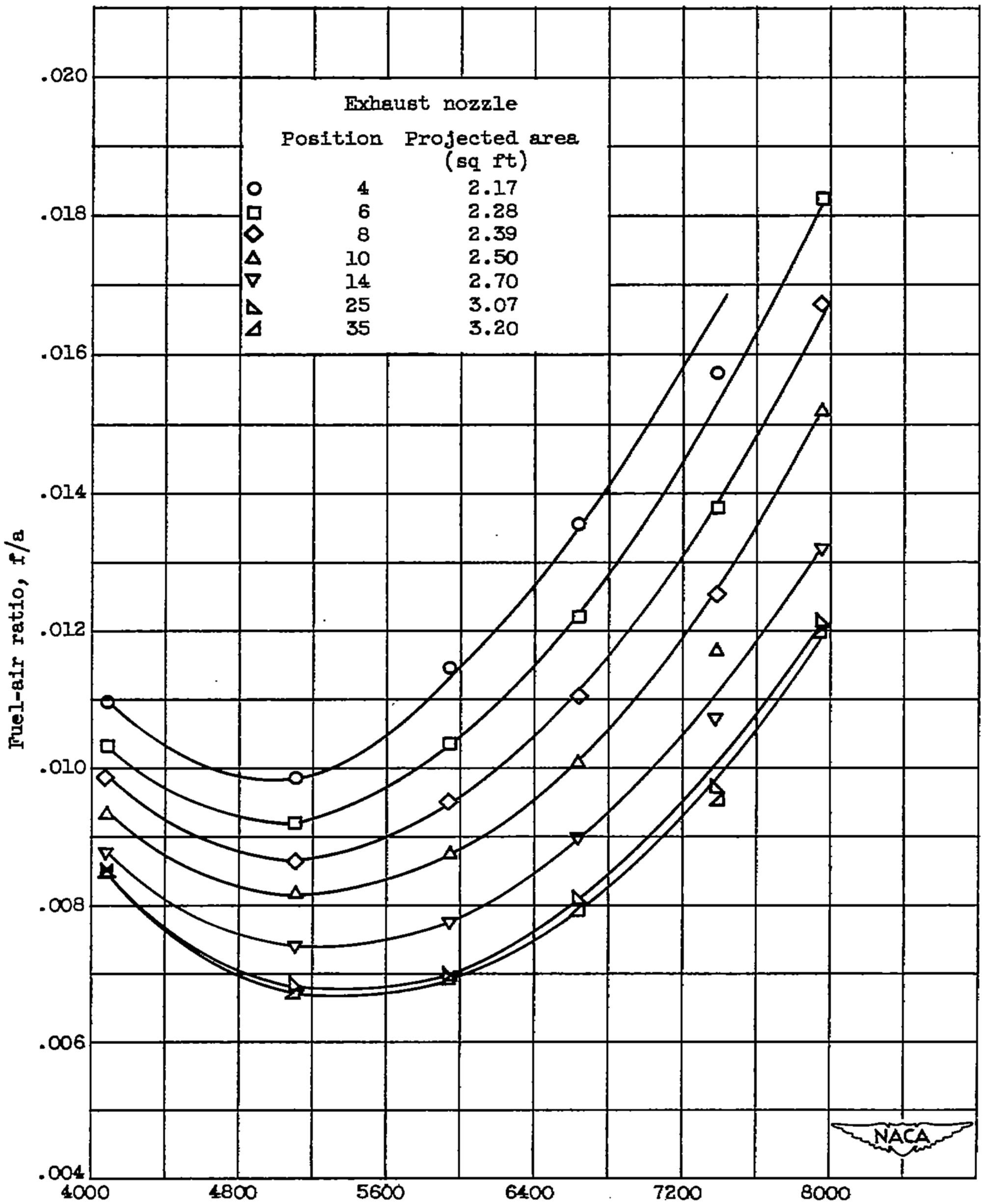


(d) Specific fuel consumption.

Figure 8. - Continued. Effect of exhaust-nozzle area on engine performance at altitude of 15,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 471° R.

[redacted]

2135

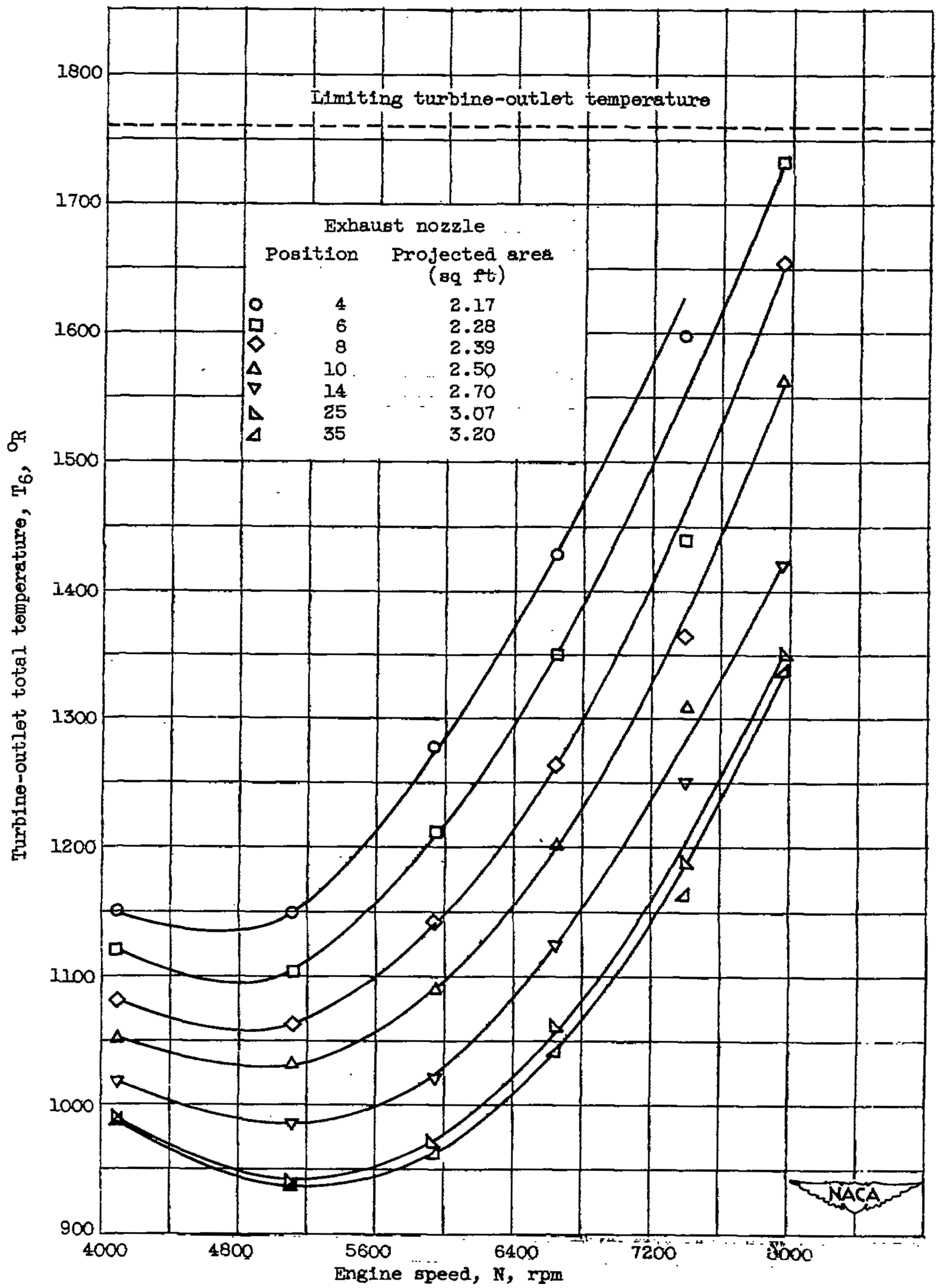


(e) Fuel-air ratio.

Figure 8. - Continued. Effect of exhaust-nozzle area on engine performance at altitude of 15,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 471° R.

[redacted]

2135



(f) Turbine-outlet gas temperature.

Figure 8. - Concluded. Effect of exhaust-nozzle area on engine performance at altitude of 15,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 471° R.

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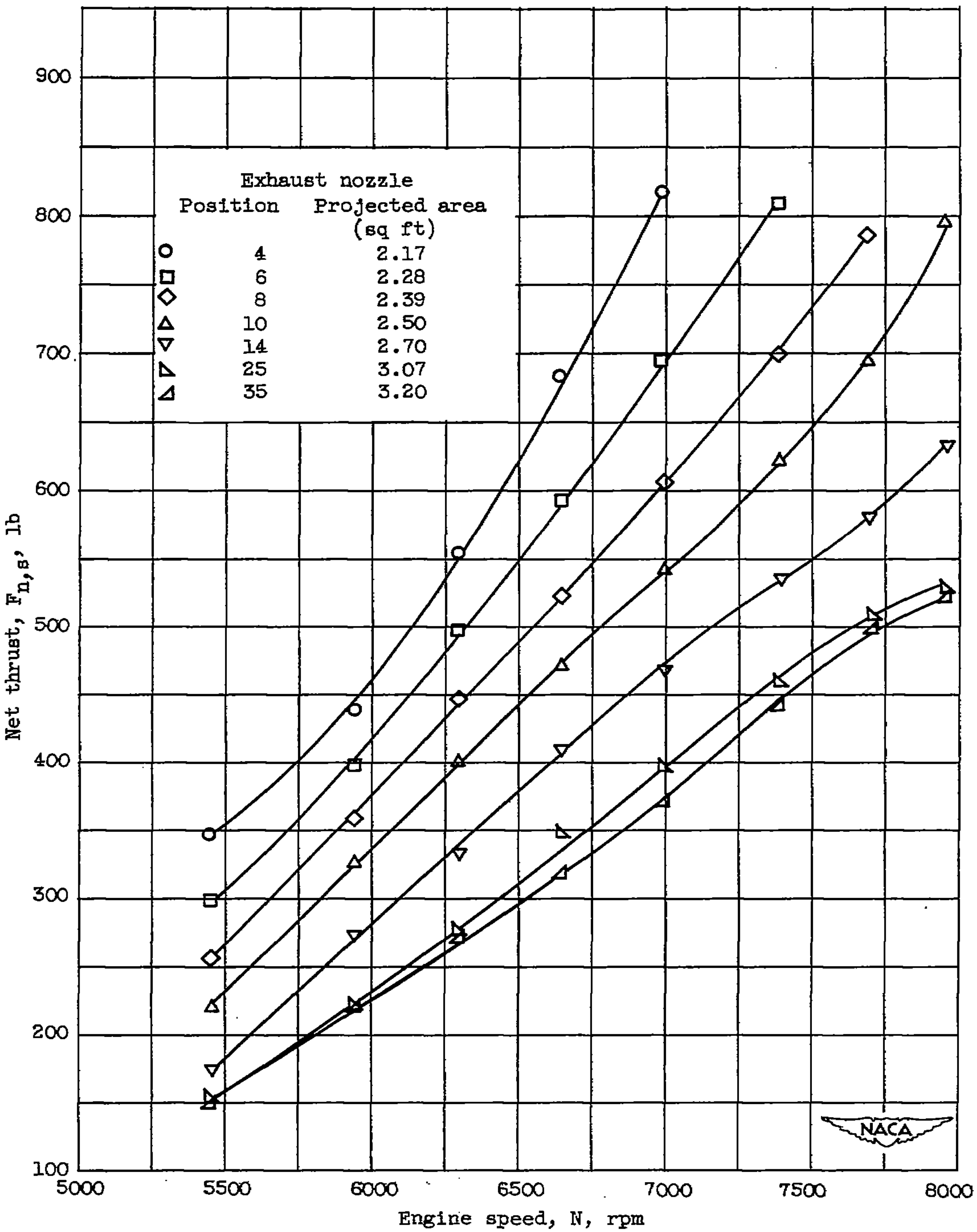
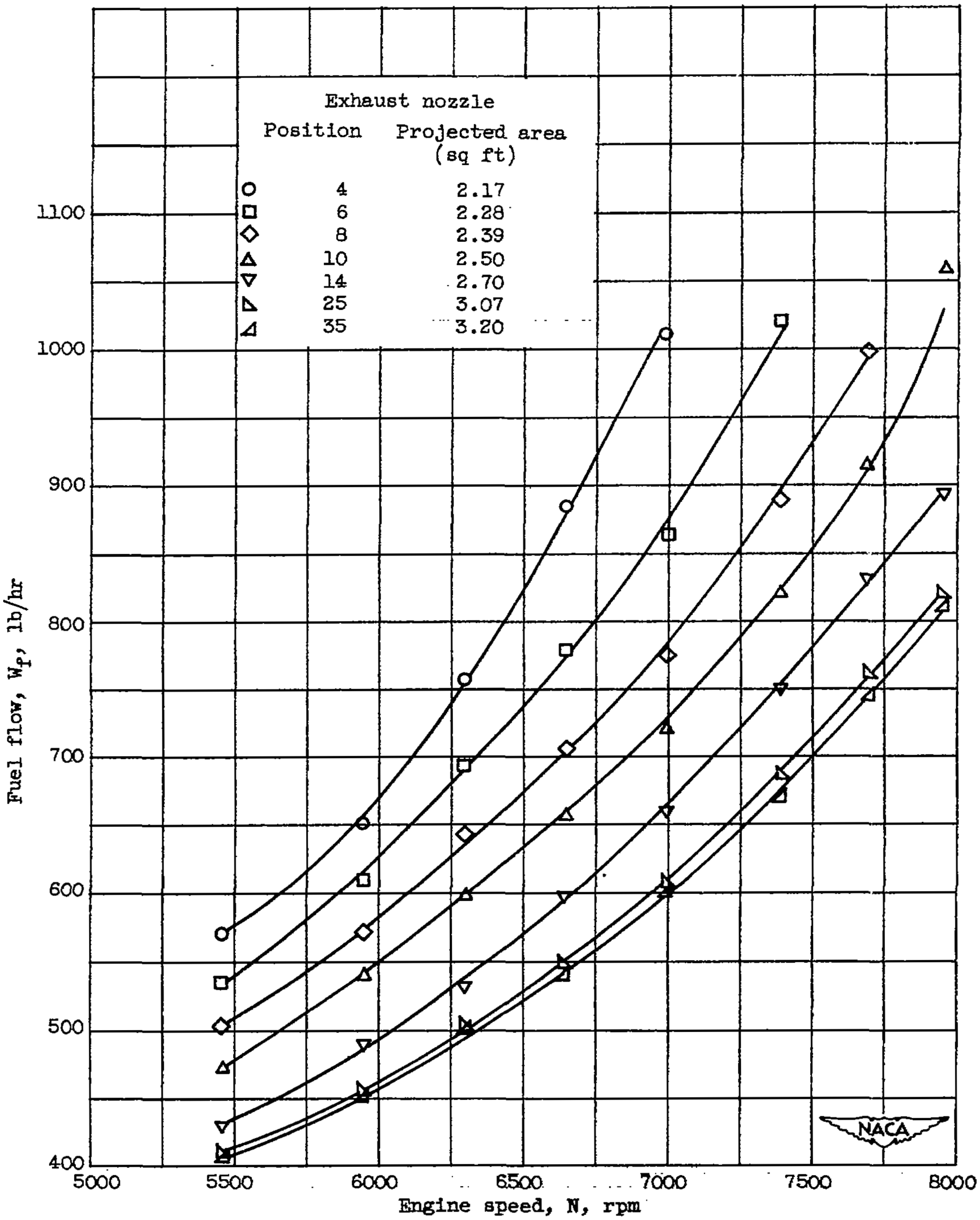
2135



(a) Net thrust.

Figure 9. - Effect of exhaust-nozzle area on engine performance at altitude of 45,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 437° R.

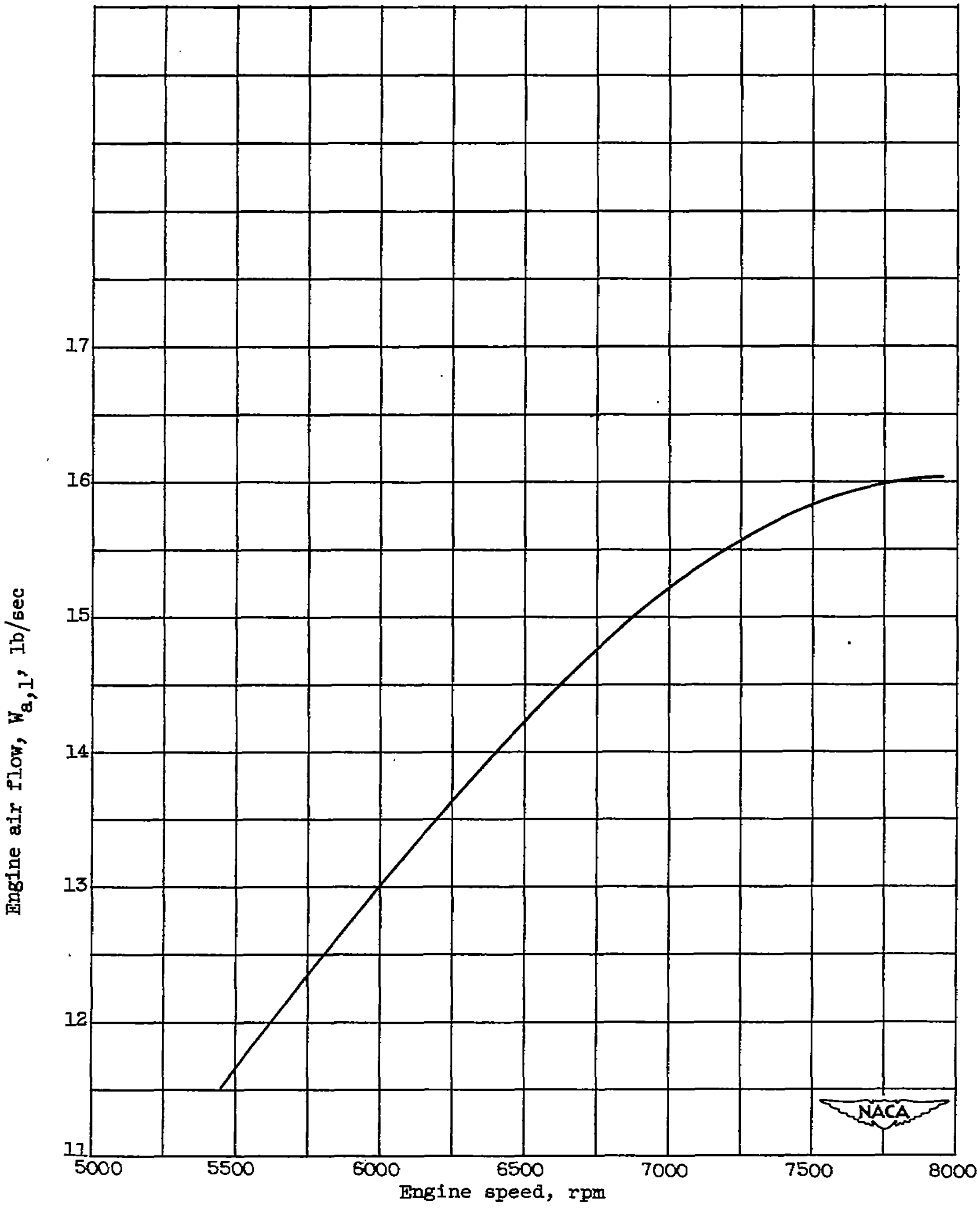
2135



(b) Fuel flow.

Figure 9. - Continued. Effect of exhaust-nozzle area on engine performance at altitude of 45,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 437° R.

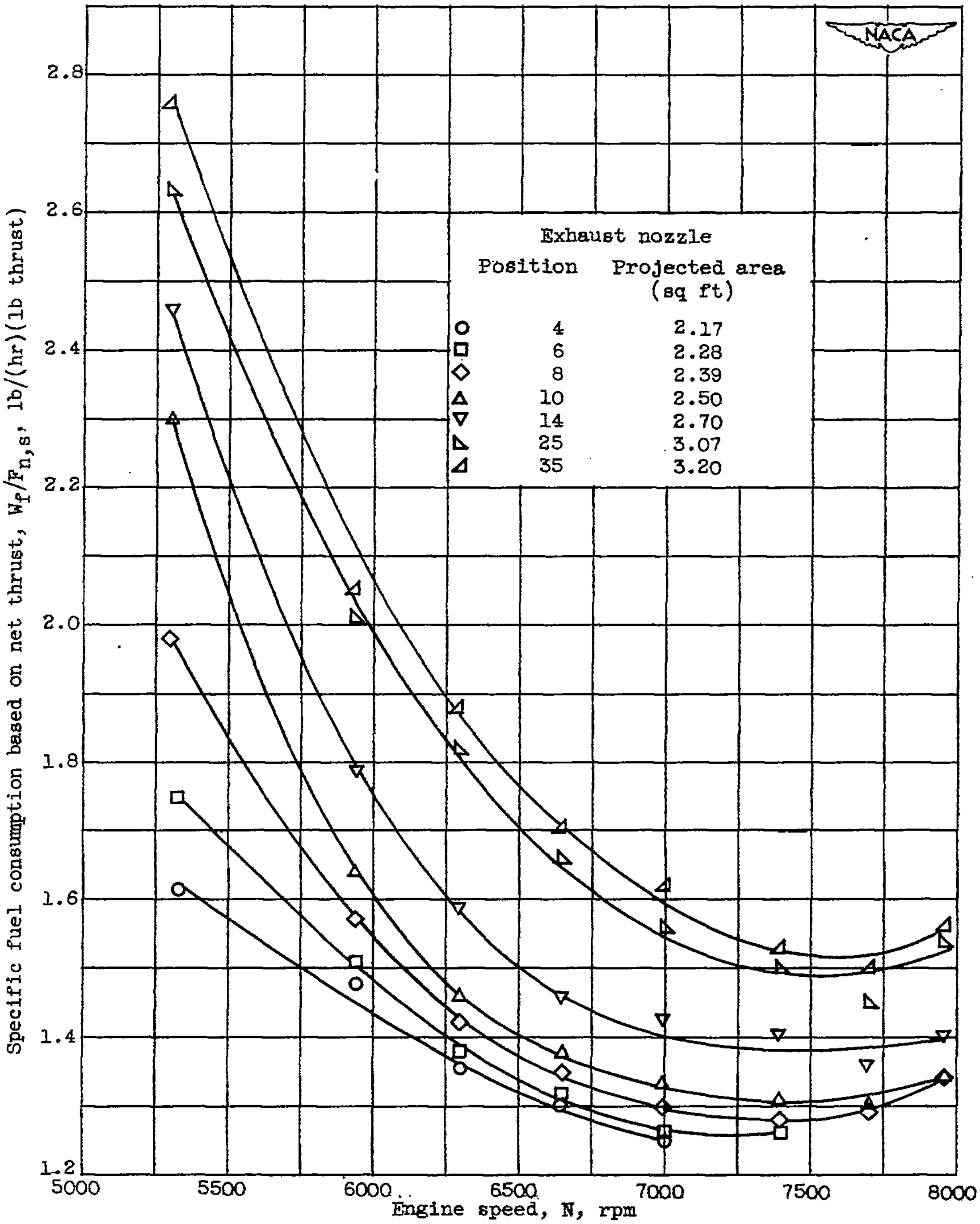
2135



(c) Engine air flow. (Curve valid for all nozzle positions.)

Figure 9. - Continued. Effect of exhuast-nozzle area on engine performance at altitude of 45,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 437° R.

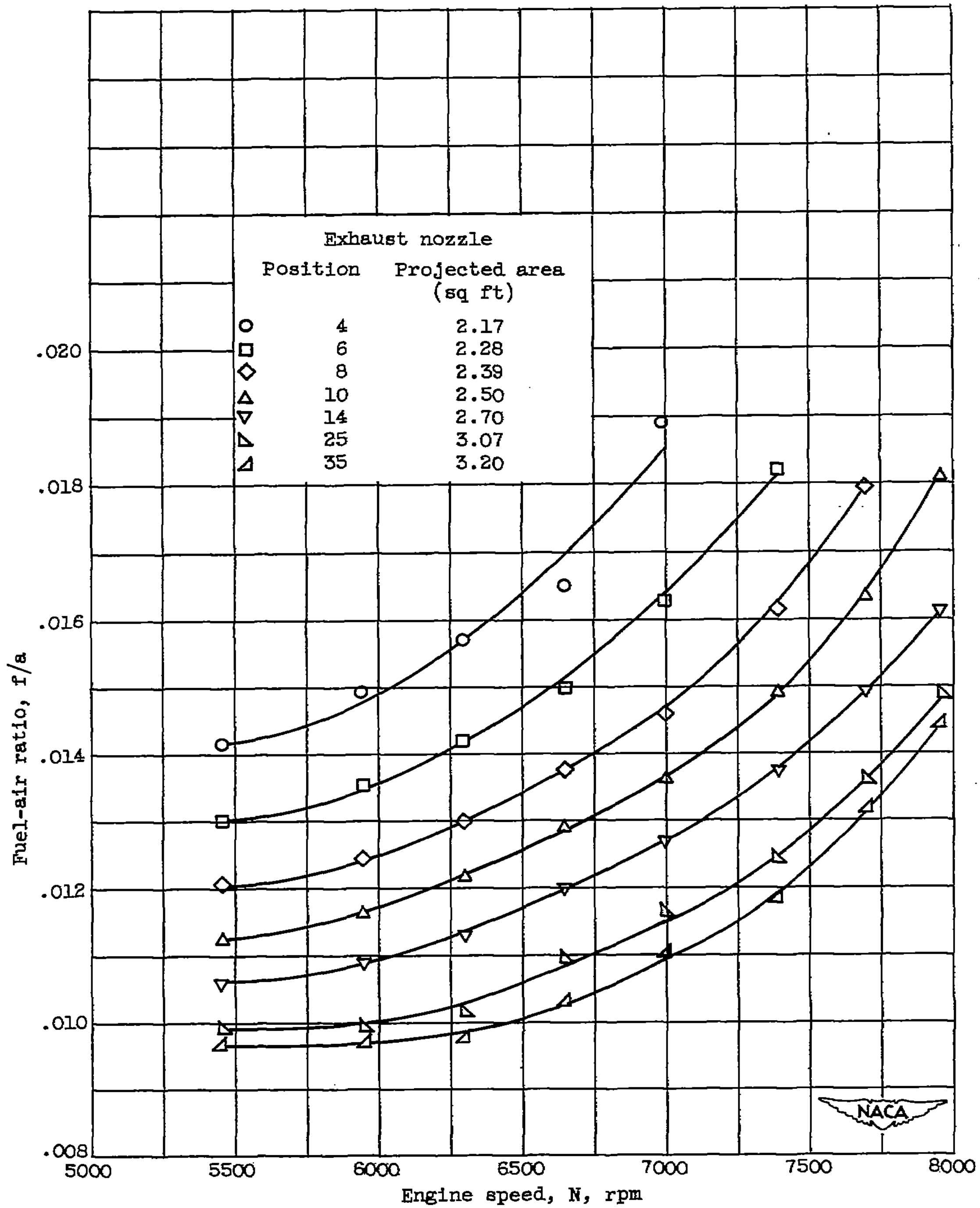
2135



(d) Specific fuel consumption.

Figure 9. - Continued. Effect of exhaust-nozzle area on engine performance at altitude of 45,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 437° R.

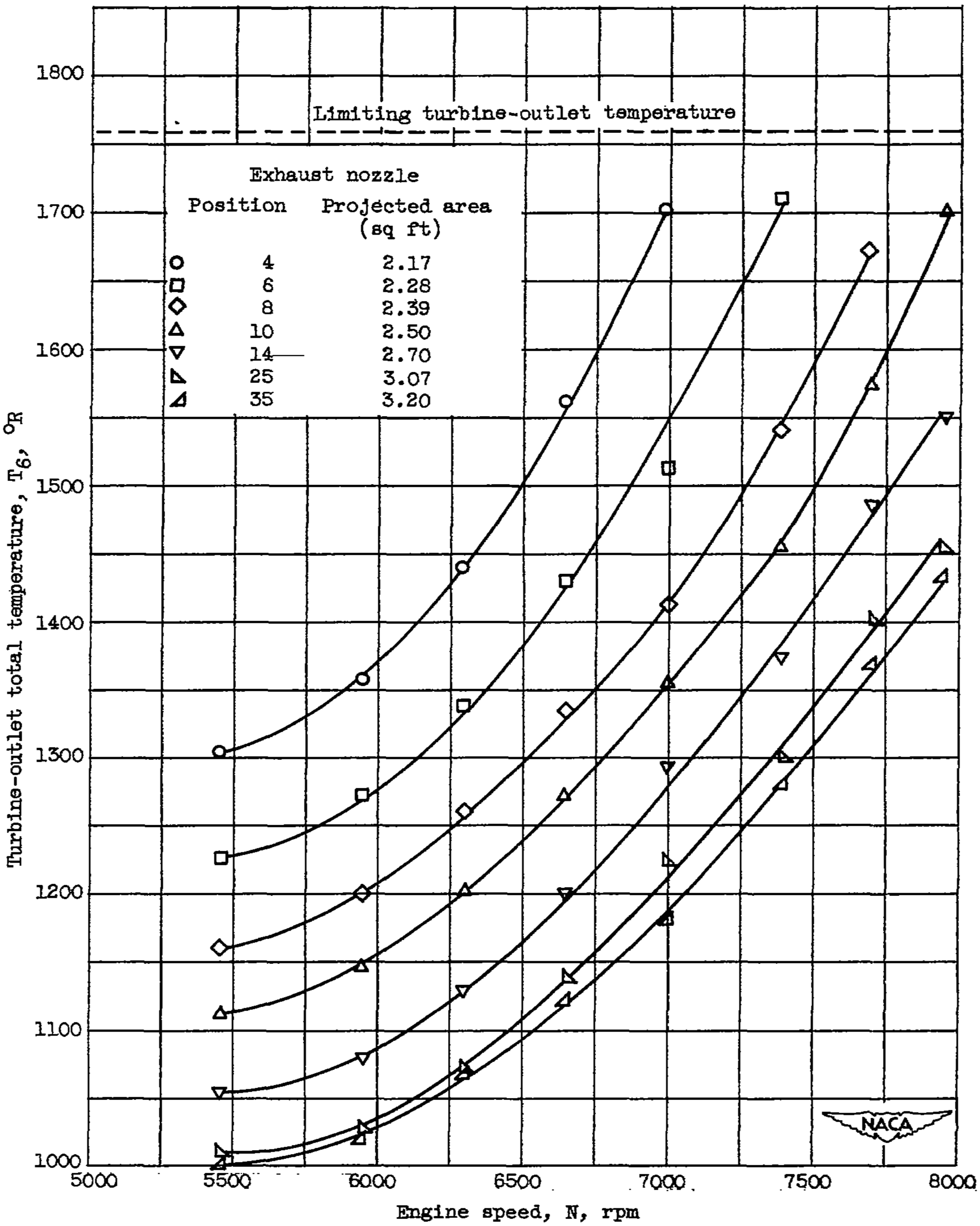
2135



(e) Fuel-air ratio.

Figure 9. - Continued. Effect of exhaust-nozzle area on engine performance at altitude of 45,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 437° R.

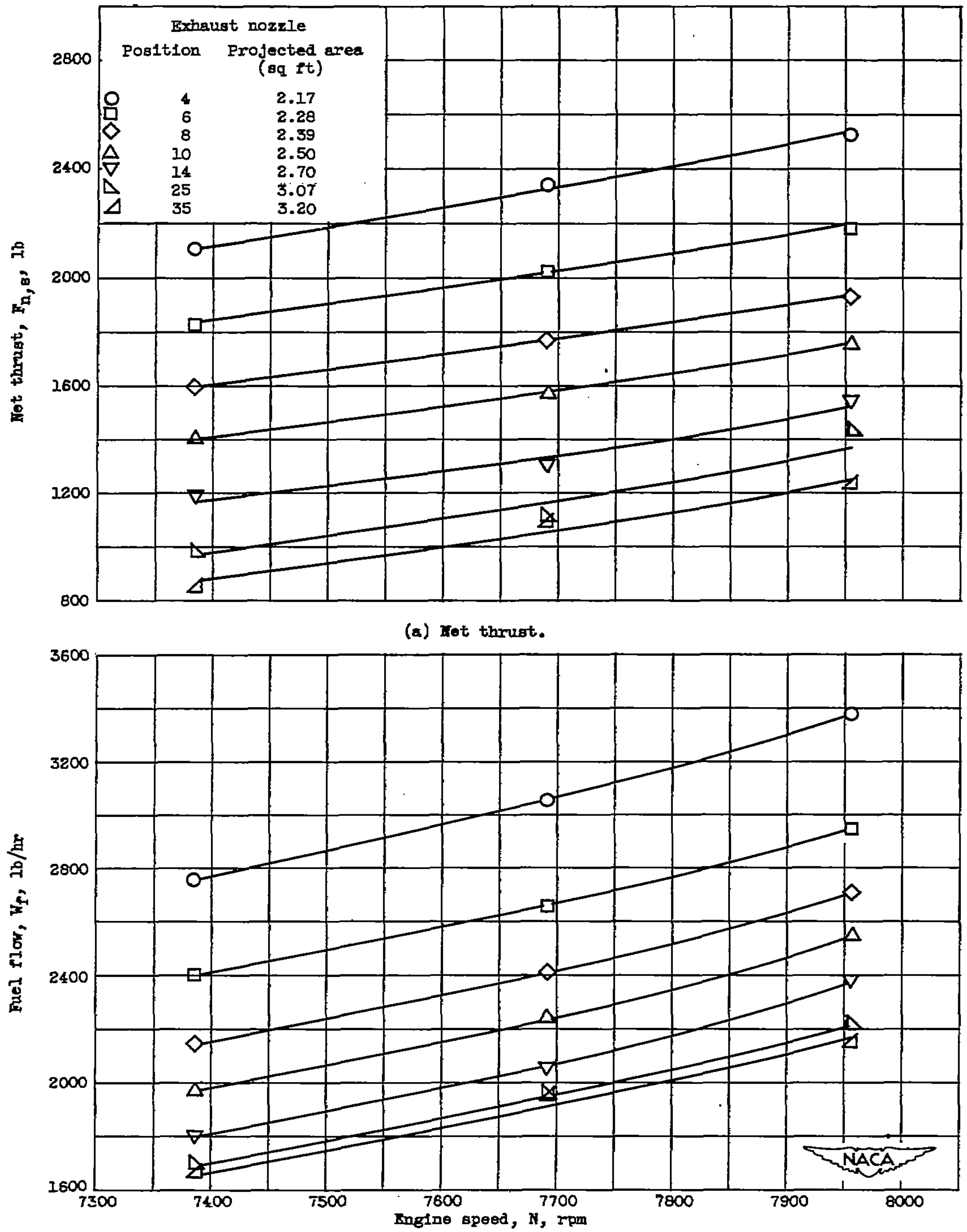
2135



(f) Turbine-outlet gas temperature.

Figure 9. - Concluded. Effect of exhaust-nozzle area on engine performance at altitude of 45,000 feet and flight Mach number of 0.19. Equivalent ambient-air temperature, 437° R.

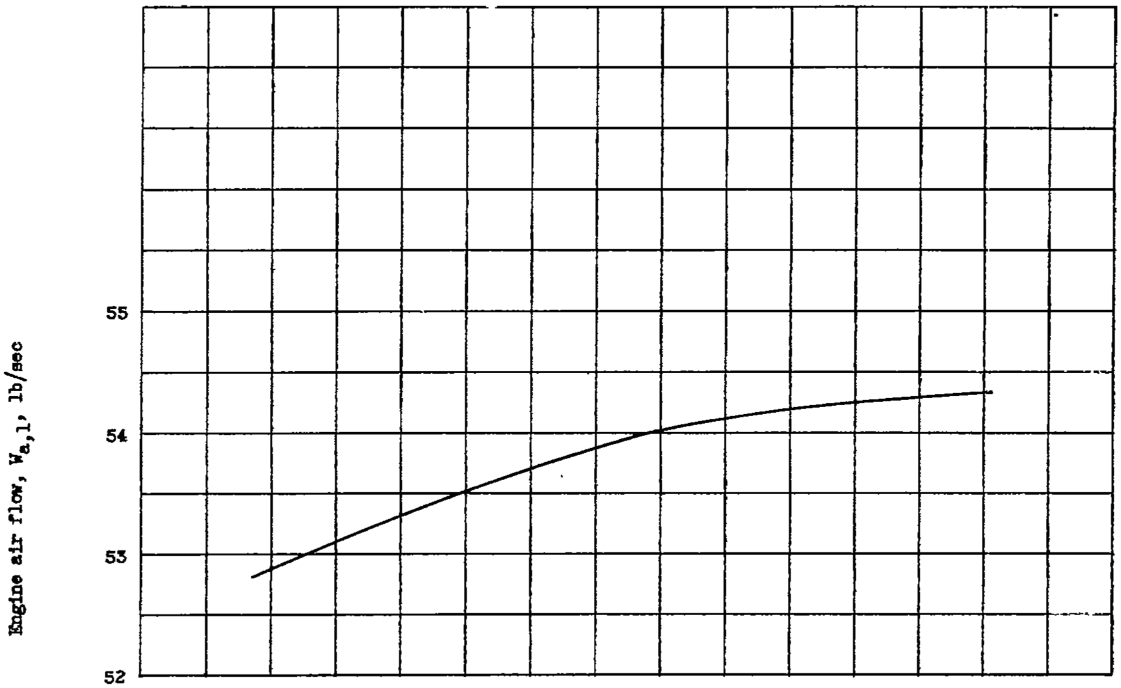
[redacted]



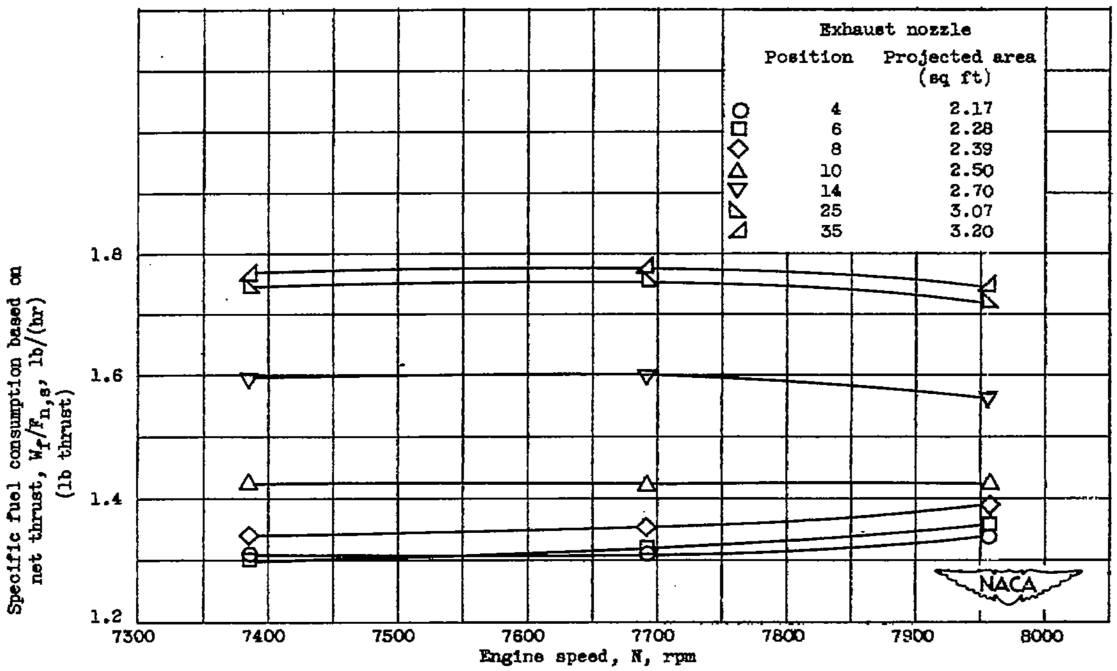
(a) Net thrust.

(b) Fuel flow.

Figure 10. - Effect of exhaust-nozzle area on engine performance at altitude of 25,000 feet and flight Mach number of 0.71. Equivalent ambient-air temperature, 429° R.



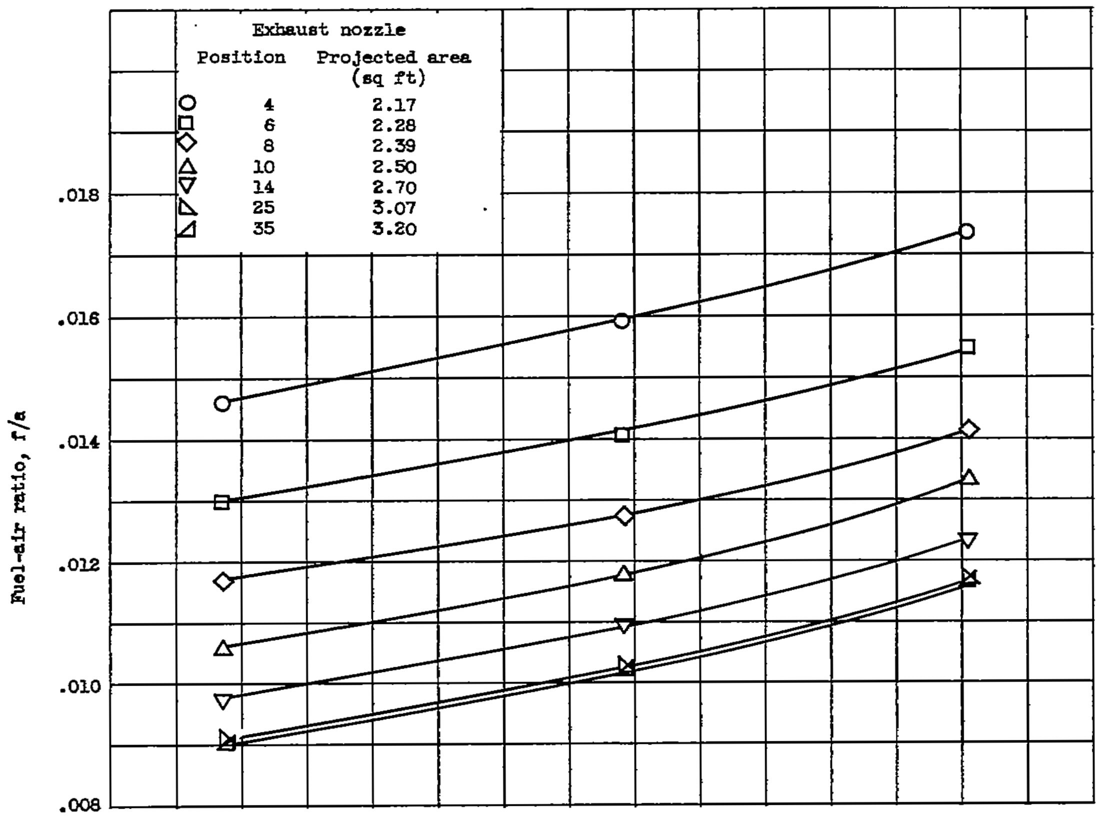
(c) Engine air flow. (Curve valid for all nozzle positions.)



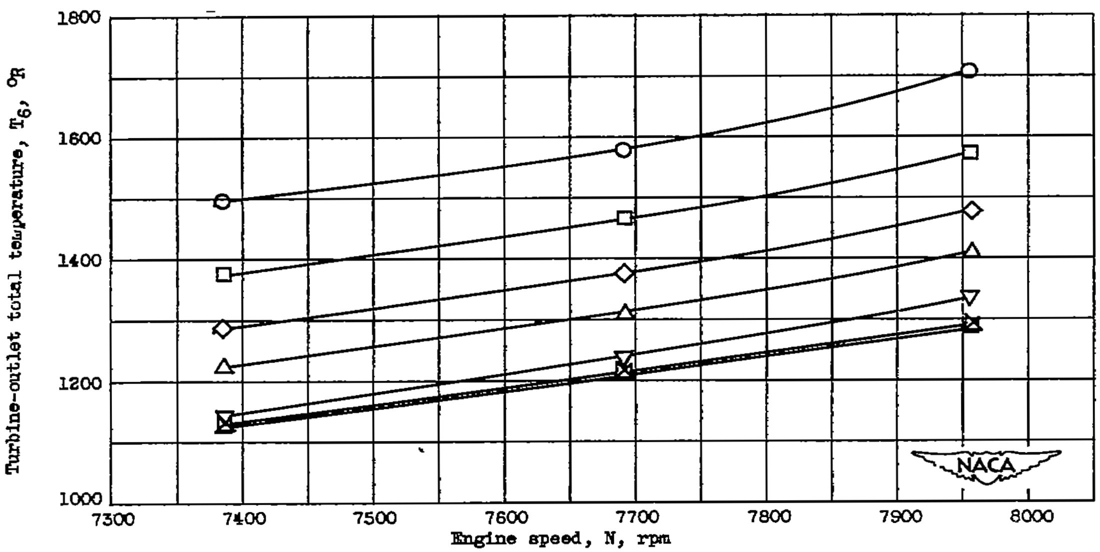
(d) Specific fuel consumption.

Figure 10. - Continued. Effect of exhaust-nozzle area on engine performance at altitude of 25,000 feet and flight Mach number of 0.71. Equivalent ambient-air temperature, 429° R.

[redacted]



(e) Fuel-air ratio.



(f) Turbine-outlet gas temperature.

Figure 10. - Concluded. Effect of exhaust-nozzle area on engine performance at altitude of 25,000 feet and flight Mach number of 0.71. Equivalent ambient-air temperature, 429° R.

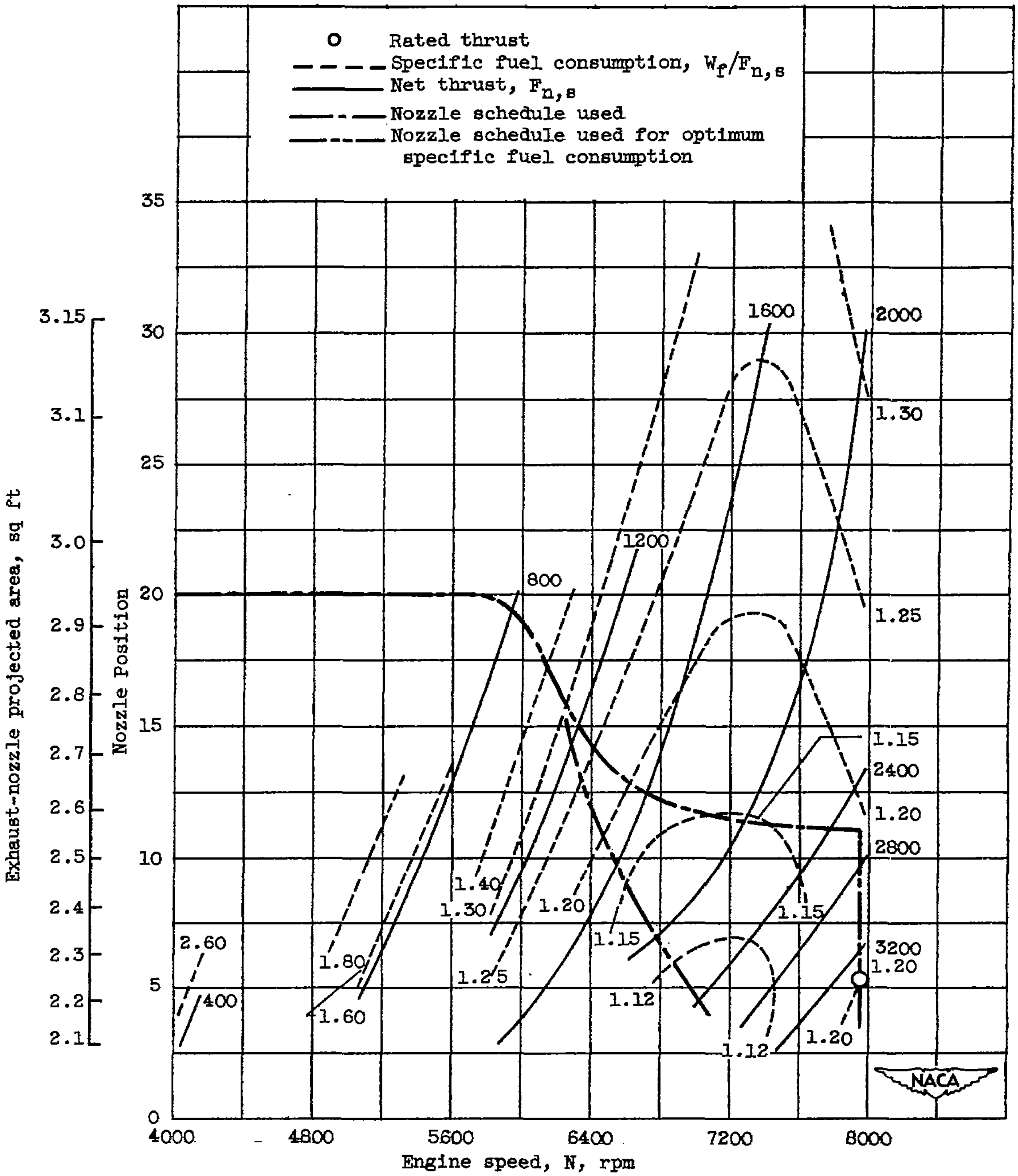


Figure 11. - Performance map at altitude of 15,000 feet and flight Mach number of 0.19.

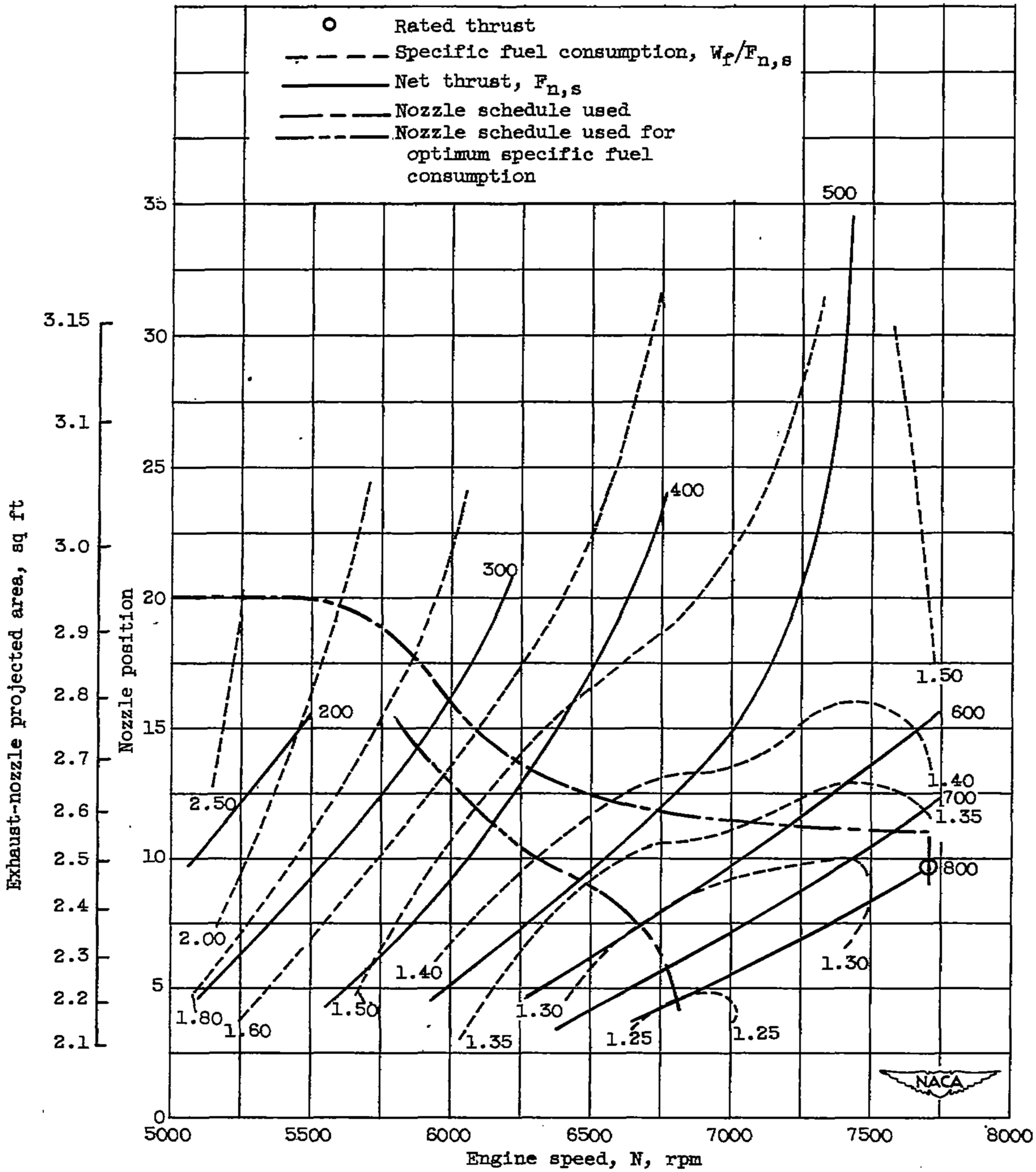


Figure 12. - Performance map at altitude of 45,000 feet and flight Mach number of 0.19.

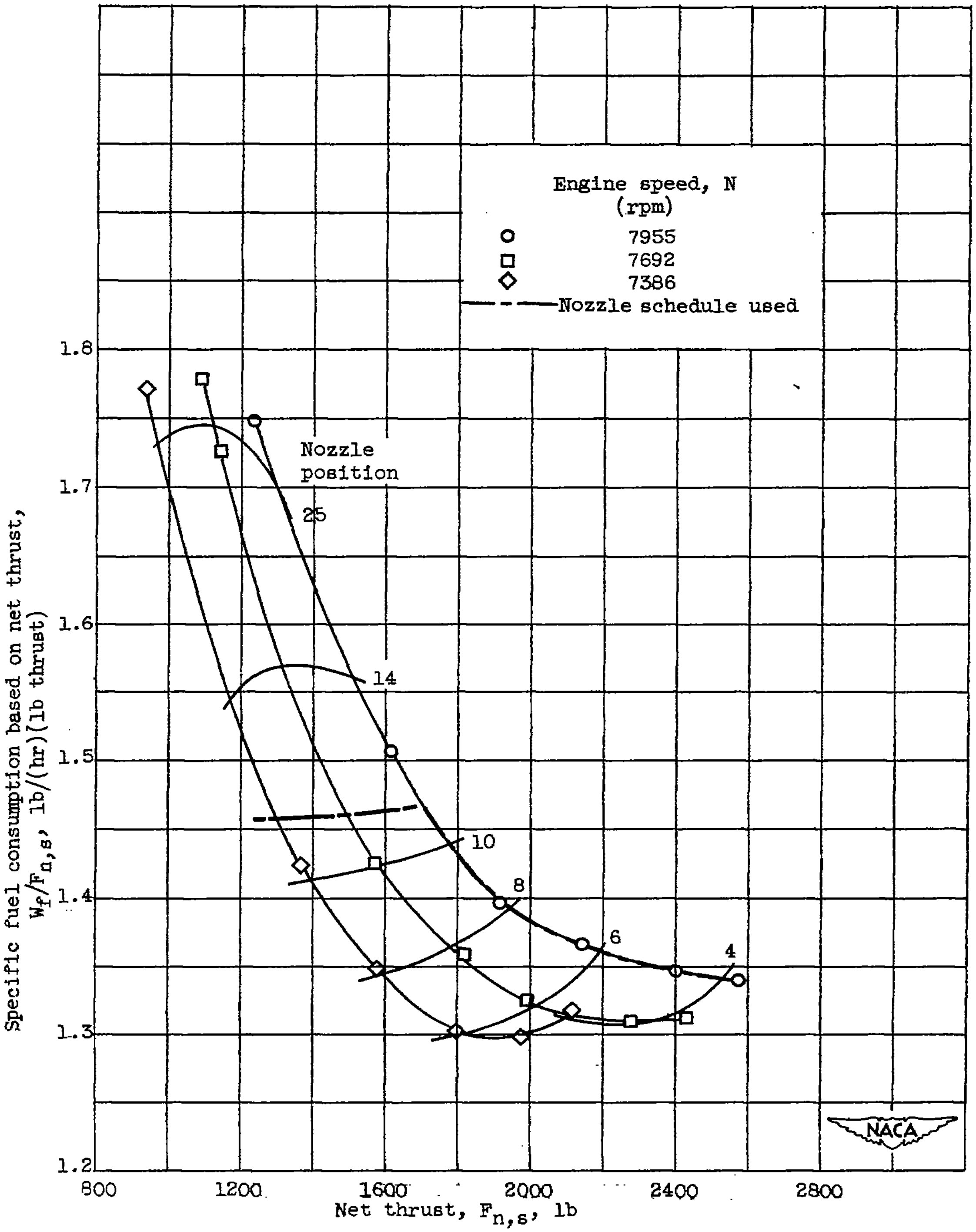
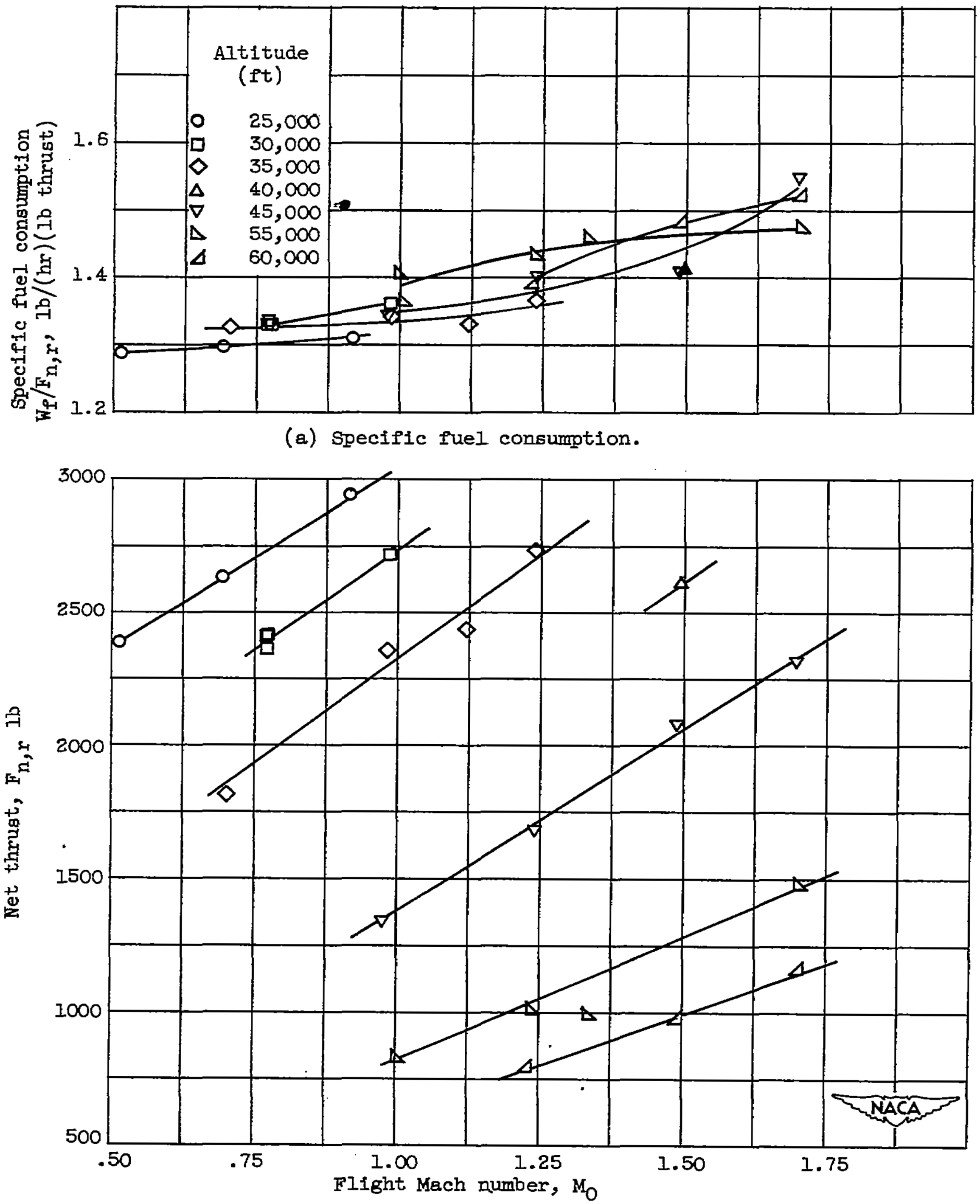


Figure 13. - Effect of engine speed on variation of specific fuel consumption with net thrust at altitude of 25,000 feet and flight Mach number of 0.71.



(a) Specific fuel consumption.

(b) Net thrust.

Figure 14. - Performance at maximum thrust.

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