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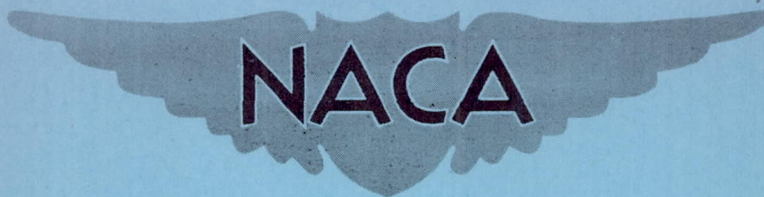
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RESEARCH MEMORANDUM

ALTITUDE WIND TUNNEL INVESTIGATION OF XJ34-WE-32 ENGINE
PERFORMANCE WITHOUT ELECTRONIC CONTROL

By Harry E. Bloomer, William J. Walker
and George L. Pantages

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RESEARCH MEMORANDUM

ALTITUDE WIND TUNNEL INVESTIGATION OF XJ34-WE-32 ENGINE

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SUMMARY

An investigation was conducted in the NACA Lewis altitude wind tunnel to evaluate the performance characteristics of an XJ34-WE-32 turbojet engine which was equipped with an afterburner, a variable-area exhaust nozzle, and an integrated electronic control. The data were obtained with the afterburner and electronic control inoperative. Performance data were obtained at altitudes from 5000 to 55,000 feet and flight Mach numbers from 0.28 to 1.06 for a complete range of operable engine speeds at each of four fixed positions of the variable-area exhaust nozzle.

The variation of generalized values of jet thrust, net thrust, and air flow with corrected engine speed were adequately defined by a single curve for altitudes up to 40,000 feet at a flight Mach number of 0.528. Generalized values of fuel flow and performance variables dependent upon fuel flow varied with changes in altitude at a given flight Mach number. Engine pumping characteristics, from which engine performance can be predicted for corrected engine speeds of 11,500 and 12,500 rpm over a wide range of Reynolds number index are presented, and two methods of thrust modulation from 70 to 100 percent of maximum thrust are compared. The results indicate that the specific fuel consumption was essentially the same for thrust modulation obtained by varying engine speed at constant exhaust-nozzle area and by varying exhaust-nozzle area at constant engine speed.

INTRODUCTION

As a part of the comprehensive investigation of the XJ34-WE-32 engine conducted in the NACA Lewis altitude wind tunnel, the over-all performance was determined over a range of altitudes and flight Mach numbers. Other phases of the investigation are reported in reference 1.

The performance data presented herein were obtained at four fixed settings of the variable-area exhaust nozzle and with the afterburner

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and electronic control inoperative. Data were obtained at altitudes from 5000 to 55,000 feet and flight Mach numbers from 0.28 to 1.06. The results are given in tables and also in graphical form to show the trends of engine performance associated with changes of altitude, flight Mach number, and exhaust-nozzle area.

APPARATUS AND PROCEDURE

Engine

The XJ34-WE-32 engine, with afterburner inoperative, has a static sea-level thrust rating of 3370 pounds at an engine speed of 12,500 rpm and an average turbine-inlet temperature of 1525° F. At this operating condition, the air flow is approximately 58 pounds per second. The engine has an 11-stage axial-flow compressor, a double annular combustor, a two-stage turbine, and an integral afterburner. The over-all length of the engine is 185 inches and the maximum diameter is 27 inches at the afterburner. The total weight of the engine and accessories is 1558 pounds. The engine is equipped with an electronic control which provides thrust regulation throughout the unaugmented and afterburning regions by means of a single thrust-selector lever. A mixer-vane assembly was installed at the compressor discharge because of a temperature-inversion problem at the turbine.

Installation

The engine and afterburner were mounted on a wing section that spanned the 20-foot-diameter test section of the altitude wind tunnel (fig. 1). 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 inlet of the engine. Engine thrust and drag measurements by the tunnel balance scales were made possible by the frictionless slip joint located in the duct upstream of the engine.

Instrumentation for measuring pressures and temperatures was installed at various stations in the engine (fig. 2).

Procedure

Pertinent engine-performance data were obtained over the range of flight conditions listed in the following table:

Altitude (ft)	Flight Mach number			
	0.28	0.53	0.79	1.06
5,000	x			
10,000		x		
25,000	x	x	x	x
40,000		x	x	x
47,000		x		
55,000		x	x	

At most of the flight conditions listed, data were obtained over a wide range of engine speeds at the full open, full closed, and at two intermediate exhaust-nozzle areas corresponding to projected nozzle areas of 153, 164, 192, and 274 square inches. Data were not obtained, however, when the combination of nozzle area and engine operating conditions was such that excessive turbine temperatures resulted.

In order to set up these various flight conditions, the air flow through the make-up air duct was throttled from approximately sea-level pressure to the total pressure that corresponded to the desired flight Mach number at a given altitude. The tunnel, into which the engine exhausted, was set at the desired altitude ambient pressure. In the calculation of flight Mach number, complete ram-pressure recovery was assumed. The temperature of the inlet air approximated NACA standard values except that the minimum temperature obtained was 440° R. The fuel used was MIL-F-5572, grade 80 (ANF-48b), clear gasoline, having a lower heating value of 19,000 Btu per pound and a hydrogen-carbon ratio of 0.186.

The methods of calculation and the symbols used herein are given in the appendix.

RESULTS AND DISCUSSION

Values of the variables which are descriptive of engine performance are tabulated in table I along with the engine-operating and simulated-flight conditions.

During the investigation, the engine was sometimes operated at compressor pressure ratios that caused the compressor to operate in a mild-stall condition. Because of this phenomenon, the engine performance variables are affected and apparent discontinuities appear in the data. In general, this stall operation occurred in the engine-speed range from 10,000 to 12,500 rpm at altitudes from 25,000 to 55,000 feet

and, of course, was most prevalent with the smaller exhaust-nozzle areas. The specific conditions at which stall influenced the performance are given in the following table:

Altitude (ft)	Flight Mach number	Engine-speed range (rpm)	Exhaust-nozzle projected area (sq in.)
25,000	0.28	10,000 - 11,000	153
25,000	.53	11,500 - 11,750	153
40,000	.53	10,000 - 12,500	153
40,000	.79	10,500 - 11,500	153
40,000	1.06	11,400 - 11,500	153
47,000	.53	Below 11,000	164
55,000	.53	All points taken	192
55,000	.79	Below 11,500	192

The use of an electronic control which schedules open exhaust nozzle until rated engine speed is attained would permit the engine to skirt all stall regions encountered during the investigation.

Generalized Performance

Engine-performance data have been generalized to NACA standard sea-level conditions by use of the conventional factors δ_T and θ_T , which are defined in the appendix. Generalized performance variables for all flight conditions investigated are given in table I. The effectiveness of the correction factors in correlating data obtained at various flight conditions to a single curve is shown in figures 3 to 9. Changes in component efficiencies such as those associated with variations in Reynolds number which accompany changes in altitude or flight speed will, of course, lessen the possibility of defining generalized performance by a single curve.

Effect of altitude. - The corrected performance data, obtained at a flight Mach number of 0.528 and at altitudes from 10,000 to 55,000 feet, are presented in figures 3 to 8 to show the effect of altitude on the corrected engine performance variables when the variable-area exhaust nozzle is in each of four fixed positions. The corrected values of jet thrust (fig. 3) and net thrust (fig. 4) reduce to a single curve for altitudes from 10,000 to 40,000 feet for all exhaust-nozzle sizes. A further increase in altitude resulted in higher values of the corrected thrusts. This increase in thrust is traceable to the reduction in compressor efficiency with altitude which requires a higher turbine-inlet temperature to sustain a given corrected engine speed. Inasmuch as compressor pressure ratio is a function of the turbine-inlet temperature, the thrust is increased notwithstanding the slight decrease in air flow shown in figure 5. Corrected values of air flow reduced to a single curve for all altitudes up to 40,000 feet for the variable-area exhaust nozzle in the wide-open position. For the two intermediate

positions of the nozzle, the air flow reduced to a single curve only for altitudes up to 25,000 feet. Any further increase in altitude reduced the air flow throughout the engine-speed range. For the smallest exhaust-nozzle area, however, the generalized air flow reduced to a single curve, within the range of data scatter, for altitudes from 10,000 to 40,000 feet, the highest altitude investigated. The aforementioned reductions in air flow with increasing altitude are probably due to changes in the internal-flow conditions caused by lower Reynolds numbers at the higher altitudes.

Because of large changes in combustion efficiency with altitude, the parameters that are dependent upon fuel flow did not reduce to a single curve for any engine speed or altitude at which data were taken. Corrected fuel flow (fig. 6) and corrected specific fuel consumption (fig. 7) increased with altitude throughout the range of corrected engine speeds. These trends are the result of lower engine combustion efficiencies caused by low pressures in the combustor at higher altitudes.

Corrected exhaust-gas total temperature (fig. 8) also increased with altitude throughout the corrected engine-speed range. This trend is due to reductions in compressor and turbine efficiencies with altitude that require higher temperatures to maintain a given corrected engine speed.

Effect of flight Mach number. - With the exception of corrected air flow, a single-curve correlation of generalized performance variables obtained over a range of flight Mach numbers is precluded by variations in engine pressure ratio, combustion efficiency, and Reynolds number effects on component efficiencies. The effect of flight Mach number on the variation of corrected air flow with corrected engine speed is presented in figure 9 for an altitude of 25,000 feet. Data showing the effect of flight Mach number on other performance variables are included in table I. Corrected air flow reduced to a single curve at the higher engine speeds and diverged slightly at the lower engine speeds for the three largest exhaust-nozzle areas. The greater separation of the corrected air-flow curves for the small nozzle area probably is the result of localized regions of stall within the compressor that result from the proximity of the engine operating lines to the compressor stall line. This trend of reduced air flow during stall is evidenced by the two data points obtained in the stall region.

From the data of figures 3 to 8, performance within the range of the investigation can be determined for operation at a flight Mach number of 0.528. In order to permit calculation of engine performance at other flight Mach numbers, engine performance is presented in terms of pumping characteristics, which are discussed in the following section.

Pumping Characteristics

Engine performance is presented in figures 10 to 12 in terms of engine total-pressure ratio, engine total-temperature ratio, corrected air flow, corrected fuel flow, and Reynolds number index for corrected engine speeds of 12,500 and 11,500 rpm. (The relation between Reynolds number index, altitude, and flight Mach number is shown in fig. 13.) From the data presented, complete engine performance may be computed at any flight condition within the range of Reynolds number indices covered by these data provided that losses in the tail pipe and the exhaust nozzle are known.

The data presented in figure 10 indicate that the critical Reynolds number index was about 0.60 at the temperature ratios and the corrected engine speeds investigated. As the Reynolds number index was reduced below the critical, the engine pressure ratio decreased rapidly. This reduction in engine pressure ratio is associated with the reduction in component efficiencies at low Reynolds numbers. This same trend is evident for corrected air flow (fig. 11). The reduction in air flow, however, is probably due to a reduction in effective-flow area caused by an increasing boundary-layer thickness or flow separation in the compressor passages. Air flow for different temperature ratios reduced to a single curve at a constant corrected engine speed of 12,500 rpm because of choking in the first stage of the compressor. However, the air flows for different temperature ratios at a constant corrected engine speed of 11,500 rpm, where the compressor is not choked, do not reduce to a single curve.

As a matter of convenience, the corrected fuel flow is presented as a function of Reynolds number index in figure 12. Although Reynolds number index is not intended to be a basis for generalizing combustion data, the correlation obtained is adequate for presentation of the fuel-flow results. The rapid increase in fuel flow at the low Reynolds number indices is obviously a result of low combustion efficiency which is associated with high altitude flight conditions. From these curves, air flow, fuel flow, and total pressure can be determined at the turbine outlet for any flight condition within the range of Reynolds number indices covered. With these values and an average over-all tail-pipe pressure loss, of 0.065 of the turbine-outlet total pressure as determined in this investigation, jet thrust can be calculated by using equation (7) in the appendix. The over-all engine performance for other tail-pipe or inlet-duct configurations may also be readily obtained if the pressure-loss characteristics of these configurations are known. This method may be extended to the lower engine-speed range by construction of similar plots from the data in table I.

Effect of Method of Engine Operation on Performance

The engine performance variables in ungeneralized form are presented in figures 14 to 17. These data have been adjusted to compensate for experimental deviation from standard NACA inlet temperature and pressure conditions by the use of the factors δ_{adj} and θ_{adj} defined in the appendix.

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The variation of net thrust and specific fuel consumption with turbine-outlet temperature for altitudes of 10,000 and 25,000 feet at a Mach number of 0.528, shown in figure 14, demonstrates conditions of engine speed and turbine-outlet temperature for maximum thrust and minimum specific fuel consumption. The value and location of the maximum engine speed for each operating line is indicated. Maximum thrust occurs at maximum engine speed and limiting turbine-outlet temperature for any given nozzle size. At this maximum thrust condition, the specific fuel consumption was slightly higher than the minimum value obtainable. It should be noted that with the smallest exhaust-nozzle size, rated engine speed cannot be reached at either altitude because of turbine temperature limitations. Rated engine speed is reached before the turbine temperature limit when the three larger nozzle sizes are used. Also it should be noted that, whereas the slope of the thrust curve is always positive, thus indicating larger thrusts for higher temperatures, the specific fuel consumption curve reaches a minimum value before the limiting temperature is reached. Therefore, there exists for each flight condition a different engine speed and exhaust-nozzle area at which minimum specific fuel consumption (at reduced thrust) may be obtained. These points are discussed in more detail in the following paragraphs.

The variation of net thrust with altitude at a constant flight Mach number of 0.528 is shown in figure 15(a). The data show performance results at rated engine speed with thrust variations obtained by changes in exhaust-nozzle area. The circular symbols represent maximum thrust points at rated engine speed and maximum turbine temperature limit. These data were taken from cross-plots of data similar to that shown in figure 14. The other symbols represent points at 90, 80, and 70 percent of the maximum thrusts; these thrusts and the accompanying specific fuel consumptions, presented in figure 15(b), were interpolated at rated speed and larger exhaust-nozzle areas. The specific fuel consumption did not change significantly with the thrust level.

Another way of modulating thrust is by keeping a constant exhaust-nozzle size and changing engine speed. Figure 15(c) shows the engine speeds required to produce 90, 80, and 70 percent of maximum thrust with a fixed exhaust-nozzle area of 164 square inches. Figure 15(d) shows the variation with altitude of specific fuel consumption for

constant exhaust-nozzle area operation at these engine speeds. Again, as thrust is reduced to as little as 70 percent of maximum thrust by lowering engine speed, the specific fuel consumption remains practically constant for the given altitudes. Comparing this mode of operation with the method of constant engine speed and varying nozzle area fail to disclose any significant difference in specific fuel consumption within this thrust range.

The effect of flight Mach number at 25,000 feet, with the same variables presented in figure 15, is presented in figure 16. Again, for the various flight Mach numbers shown, there is little difference in performance for the two methods of thrust modulation at any flight Mach number.

CONCLUDING REMARKS

Complete engine-performance data were obtained for operation over a wide range of engine speeds and with four fixed exhaust-nozzle areas at simulated altitudes as high as 55,000 feet and flight Mach numbers as high as 1.06. Results obtained at a flight Mach number of 0.528 for altitudes from 10,000 to 55,000 feet were generalized by the use of the correction factors δ_T and θ_T . Jet thrust, net thrust, and air flow in general reduced to a single curve as a function of corrected engine speed for a given flight Mach number and altitudes up to about 40,000 feet; however, parameters involving fuel flow failed to reduce to a single curve. For operation over a range of flight Mach numbers from 0.284 to 1.055 at a constant altitude of 25,000 feet, only corrected air-flow values tended to reduce to a single curve. Engine performance at speeds of 11,500 and 12,500 rpm may readily be calculated, however, for a range of either flight Mach numbers or altitudes by the use of engine pumping curves presented herein. All the data obtained are also given in tabular form thereby permitting the construction of pumping-characteristic curves for a wide range of engine speeds.

Two methods of thrust modulation, (a) varying engine speed at constant exhaust-nozzle area and (b) varying exhaust-nozzle area at constant (rated) engine speed, were compared. For thrust loads from maximum to 70 percent of maximum at a given flight condition, the specific fuel consumption was essentially independent of the mode of operation over the entire range of flight conditions simulated.

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APPENDIX - CALCULATIONS

Symbols

The following symbols are used in the calculations and on the figures:

A	cross-sectional area, sq ft
B	thrust-scale reading, lb
C_V	velocity coefficient, ratio of scale jet thrust to rake jet thrust
D	external drag of installation, lb
D_r	drag of exhaust-nozzle survey rake, lb
F_j	jet thrust, lb
F_n	net thrust, lb
g	acceleration due to gravity, 32.2 ft/sec ²
M	Mach number
N	engine speed, rpm
P	total pressure, lb/sq ft absolute
p	static pressure, lb/sq ft absolute
R	gas constant, 53.4 ft-lb/(lb)(°R)
T	total temperature, °R
t	static temperature, °R
V	velocity, ft/sec
W_a	air flow, lb/sec
W_f	fuel flow, lb/hr
W_g	gas flow, lb/sec
γ	ratio of specific heat for gases

δ_T	ratio of compressor-inlet absolute total pressure to absolute static pressure of NACA standard atmosphere at sea level
δ_{adj}	ratio of compressor-inlet absolute total pressure to total pressure of NACA standard atmosphere at altitude flight condition
θ_T	ratio of compressor-inlet absolute total temperature to absolute static temperature of NACA standard atmosphere at sea level
θ_{adj}	ratio of compressor-inlet absolute total temperature to total temperature of NACA standard atmosphere at altitude flight condition
ϕ	ratio of kinematic viscosity of air at compressor inlet to viscosity of NACA standard atmosphere at sea level

Subscripts:

a	air
f	fuel
i	indicated
s	scale
0	free-stream conditions
1	inlet duct at frictionless slip joint
2	compressor-inlet annulus
5	turbine outlet
7	exhaust-nozzle inlet
8	exhaust nozzle, $1\frac{3}{8}$ -in. forward of fixed portion of exhaust nozzle

Methods of Calculation

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

$$M_0 = \sqrt{\frac{2}{\gamma_1 - 1} \left[\left(\frac{P_1}{P_0} \right)^{\frac{\gamma_1 - 1}{\gamma_1}} - 1 \right]} \quad (1)$$

Airspeed. - The following equation was used to calculate the equivalent airspeed

$$V_0 = M_0 \sqrt{\gamma g R T_1 \left(\frac{P_0}{P_1} \right)^{\frac{\gamma_1 - 1}{\gamma_1}}} \quad (2)$$

Temperature. - Static temperatures were determined from indicated temperatures with the following relation

$$t = \frac{T_i}{1 + 0.85 \left[\left(\frac{P}{P} \right)^{\frac{\gamma - 1}{\gamma}} - 1 \right]} \quad (3)$$

where 0.85 is the impact recovery factor for the type of thermocouple used. Total temperature was calculated from the adiabatic relation between temperatures and pressures.

Air flow. - Air flow was determined from pressure and temperature measurements in the engine-inlet air duct by use of the equation

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

Gas flow. - The total weight flow through the engine was calculated as follows:

$$W_{g,5} = W_{a,1} + \frac{W_f}{3600} \quad (5)$$

Jet thrust. - The jet thrust of the installation was determined from the balance-scale measurements by using the following equation:

$$F_{j,s} = B + D + D_r + \frac{W_{a,1} V_1}{g} + A_1 (p_1 - p_0) \quad (6)$$

The last two terms of this expression represent the momentum and pressure forces on the installation at the slip joint in the inlet-air duct. The external drag of the installation was determined with the engine inoperative. Drag of the water-cooled exhaust-nozzle survey rake was measured by an air-balance piston mechanism.

Scale net thrust was obtained by subtracting the equivalent free-stream momentum of the inlet air from the scale jet thrust:

$$F_{n,s} = F_{j,s} - \frac{W_{a,1} V_0}{g}$$

Jet thrust. - If it is assumed that there is complete expansion and that there are no losses in the exhaust system,

$$F_j = \frac{W_a \left(1 + \frac{W_f}{W_a} \right)}{g} \sqrt{\frac{2\gamma_5 g R T_5}{(\gamma_5 - 1)} \left[1 - \left(\frac{p_0}{p_5} \right)^{\frac{\gamma_5 - 1}{\gamma_5}} \right]} \quad (7)$$

REFERENCES

1. Sobolewski, A. E., and Farley, J. M.: Steady-State Engine Windmilling and Engine Speed Decay Characteristics of an Axial-Flow Turbojet Engine. NACA RM E51106, 1951.

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SIMULATED-FLIGHT CONDITIONS WITH MIXER VANES INSTALLED - Continued



Table with columns: Engine total-temperature ratio T5/T2, Altitude Wt, Fuel flow (lb/hr) [Corrected, Adjusted], Turbine-outlet total pressure P5, Specific fuel consumption [lb/hr, Altitude, Corrected, Adjusted], Exhaust gas total temperature [Altitude, Corrected, Adjusted], Cor-rected engine speed N, Ad-justed engine speed N, Run. Includes a sub-header '(d) Exhaust-nozzle area, 274 square inches.' and a dense grid of data points.

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SIMULATED-FLIGHT CONDITIONS WITH MIXER VANES INSTALLED - Concluded

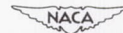


Table with columns: Engine total-temperature ratio (T5/T2), Fuel flow (lb/hr) (Corrected, Adjusted), Turbine-outlet total pressure (P5), Specific fuel consumption (lb/hr) (Altitude, Corrected, Adjusted), Exhaust gas total temperature (T8), Corrected engine speed (N), Adjusted engine speed (N), and Run. Sub-caption: (e) Miscellaneous points, exhaust-nozzle area given.

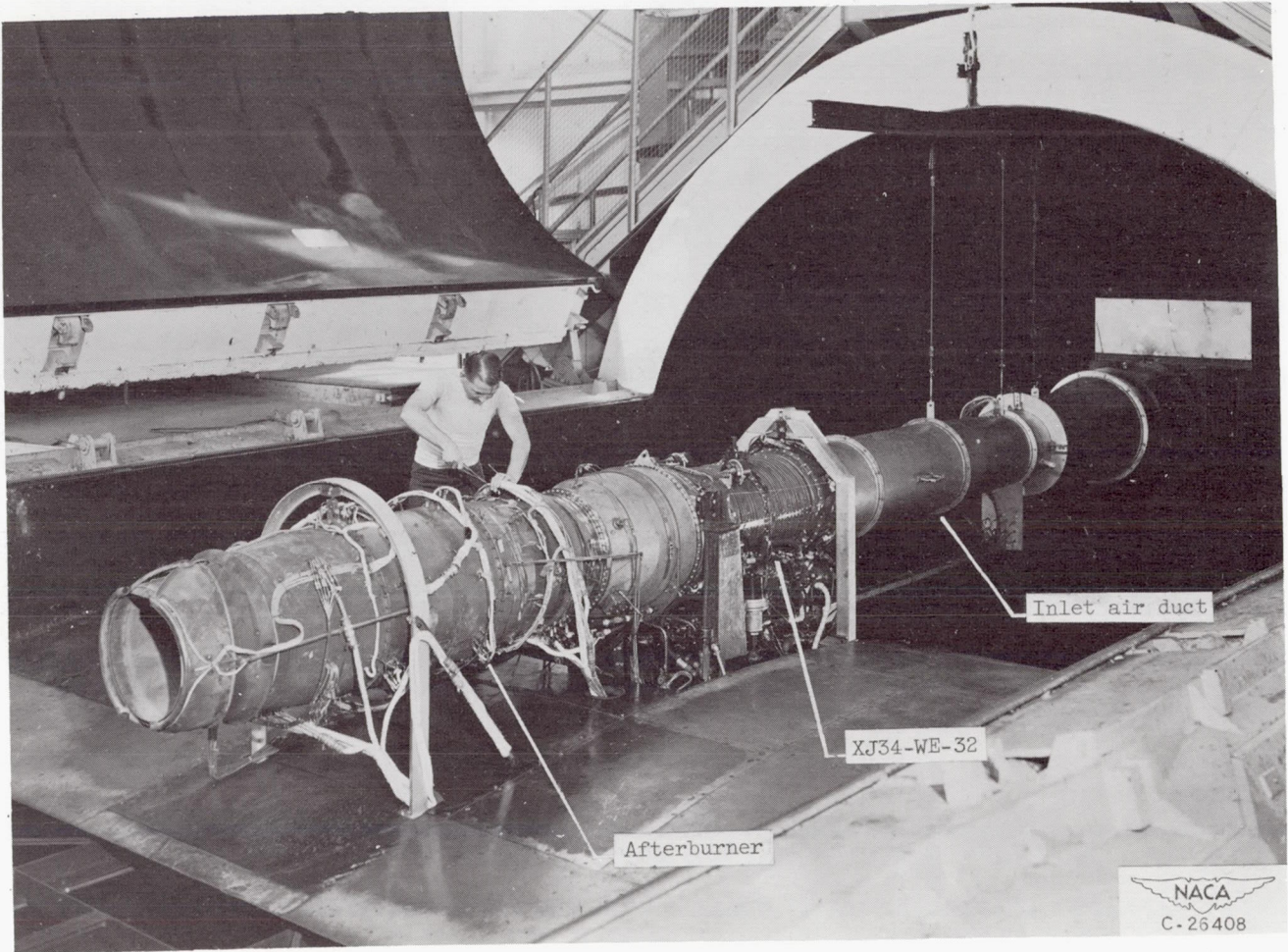


Figure 1. - Installation of XJ34-WE-32 in altitude wind tunnel.

Station	Total pressure tubes	Static pressure tubes	Thermo-couples
1	17	5	9
2	16	10	8
3	15	3	3
4	5	--	--
5	21	6	36
7	30	20	30
8	26	11	16

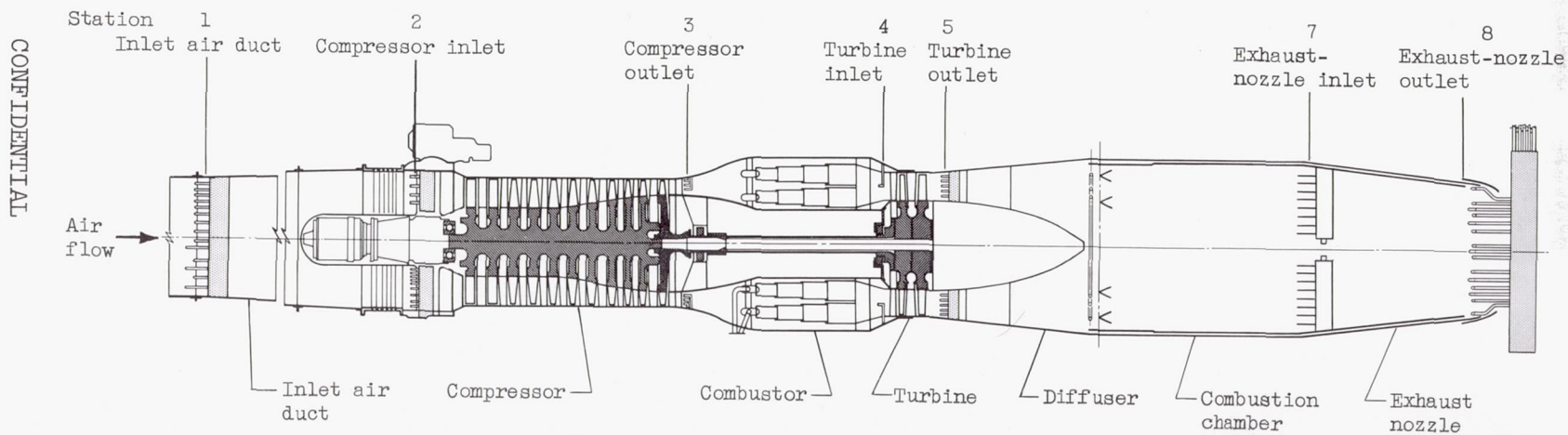
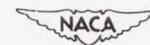


Figure 2. - Cross section of engine showing location of instrumentation.



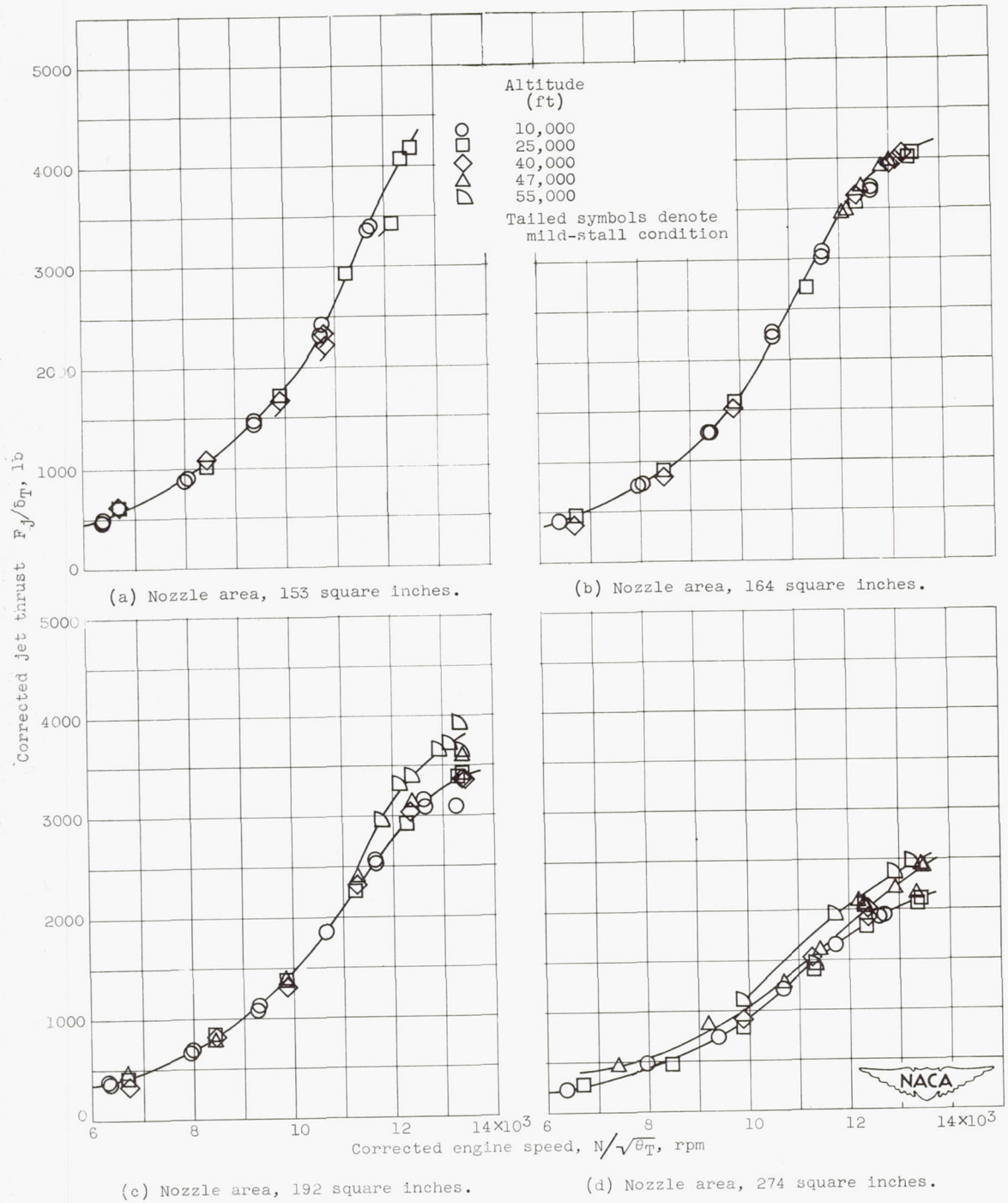


Figure 3. - Effect of altitude on variation of corrected jet thrust with corrected engine speed at flight Mach number of 0.528.

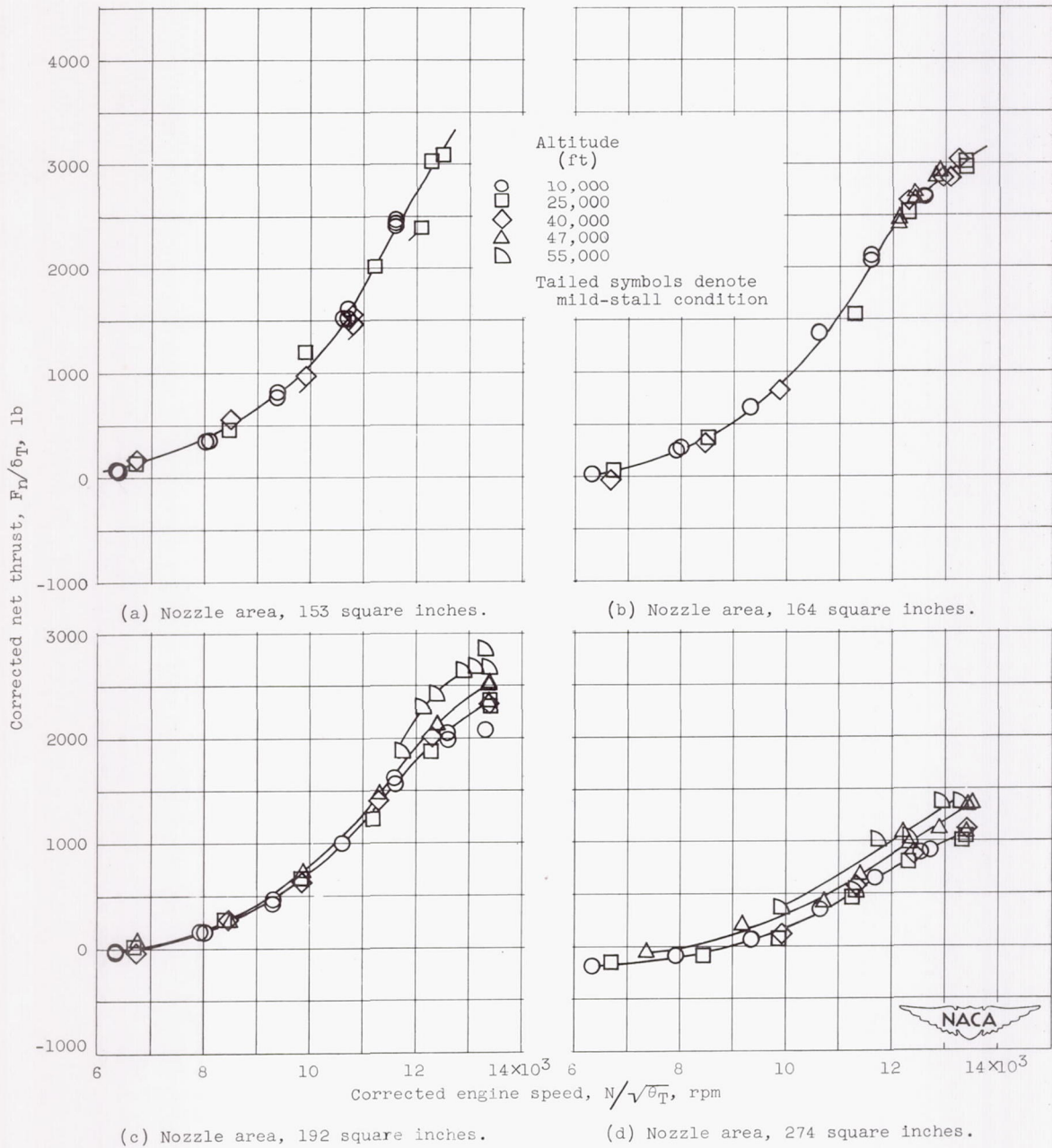


Figure 4. - Effect of altitude on variation of corrected net thrust with corrected engine speed at flight Mach number of 0.528.

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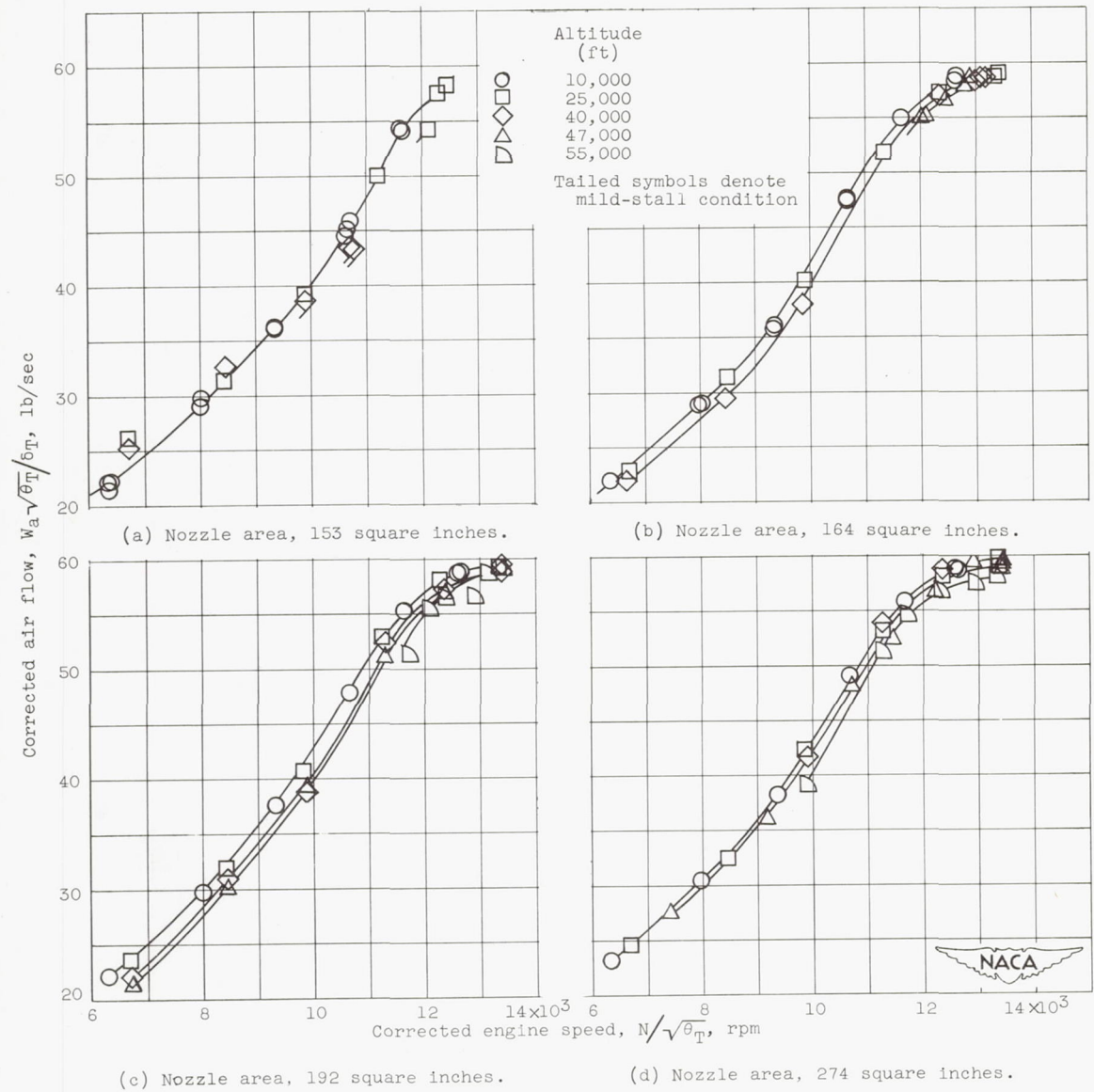


Figure 5. - Effect of altitude on variation of corrected air flow with corrected engine speed at flight Mach number of 0.528.

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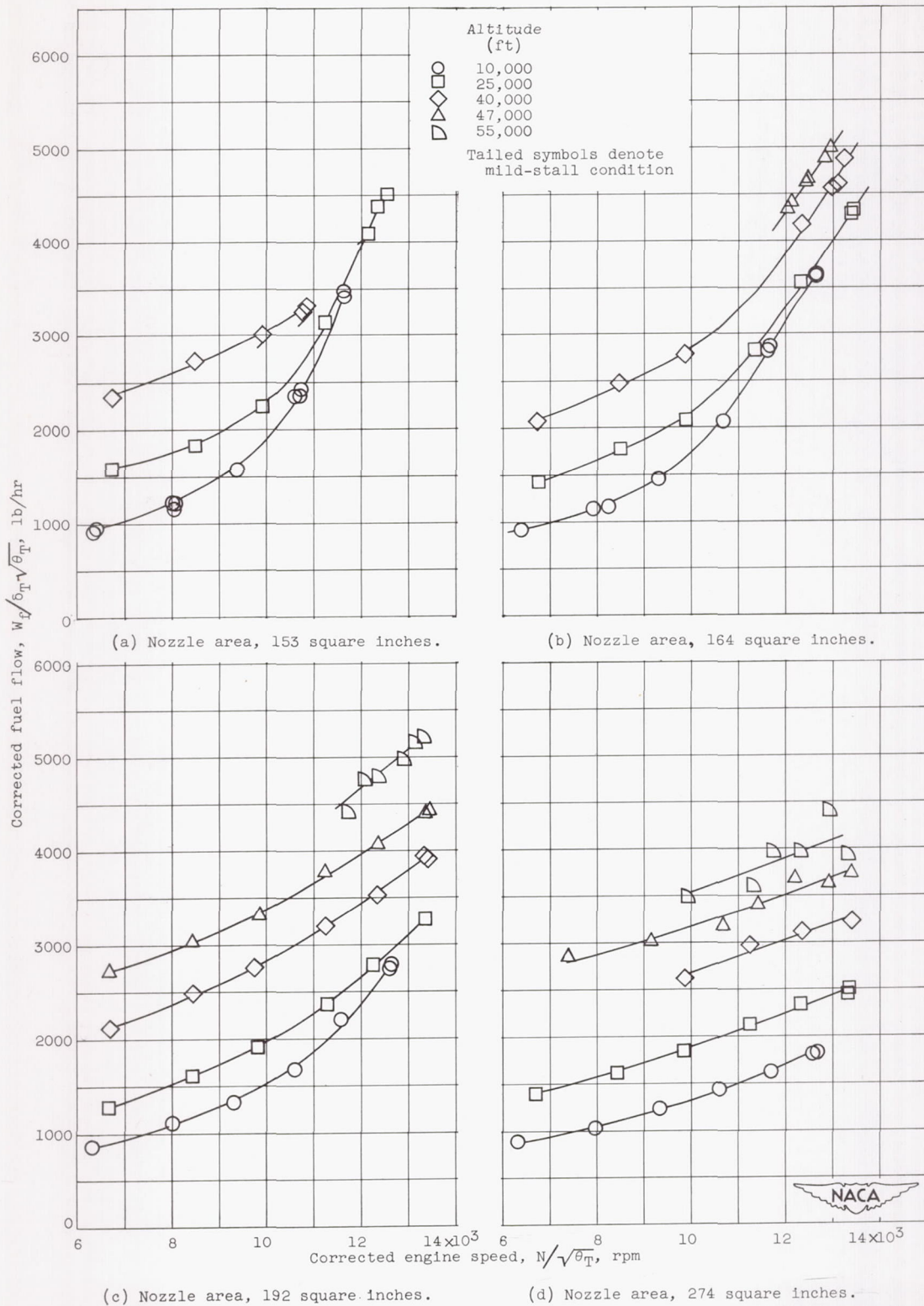


Figure 6. - Effect of altitude on variation of corrected fuel flow with corrected engine speed at flight Mach number of 0.528.

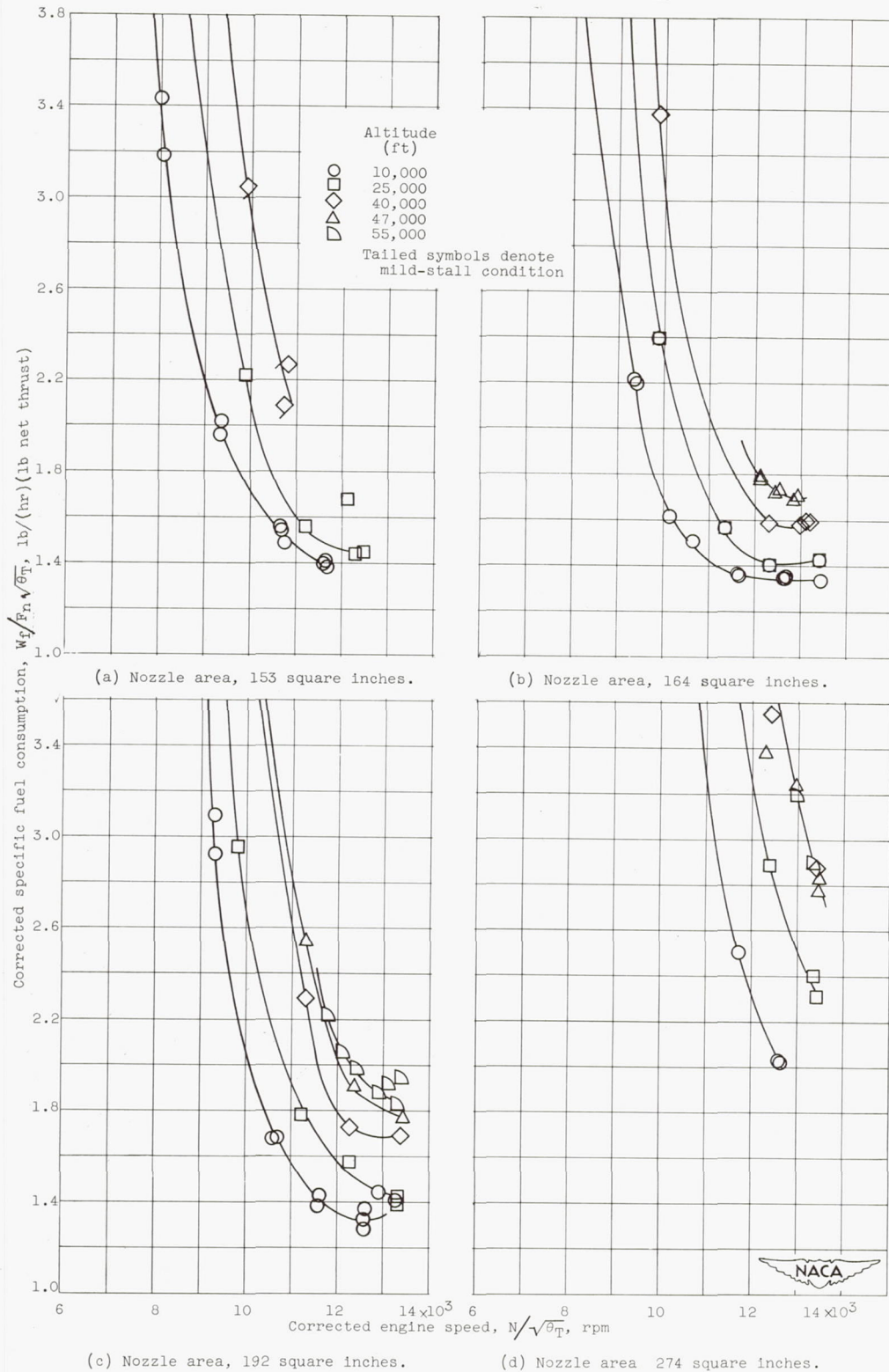


Figure 7. - Effect of altitude on variation of corrected specific fuel consumption with corrected engine speed at flight Mach number of 0.528.

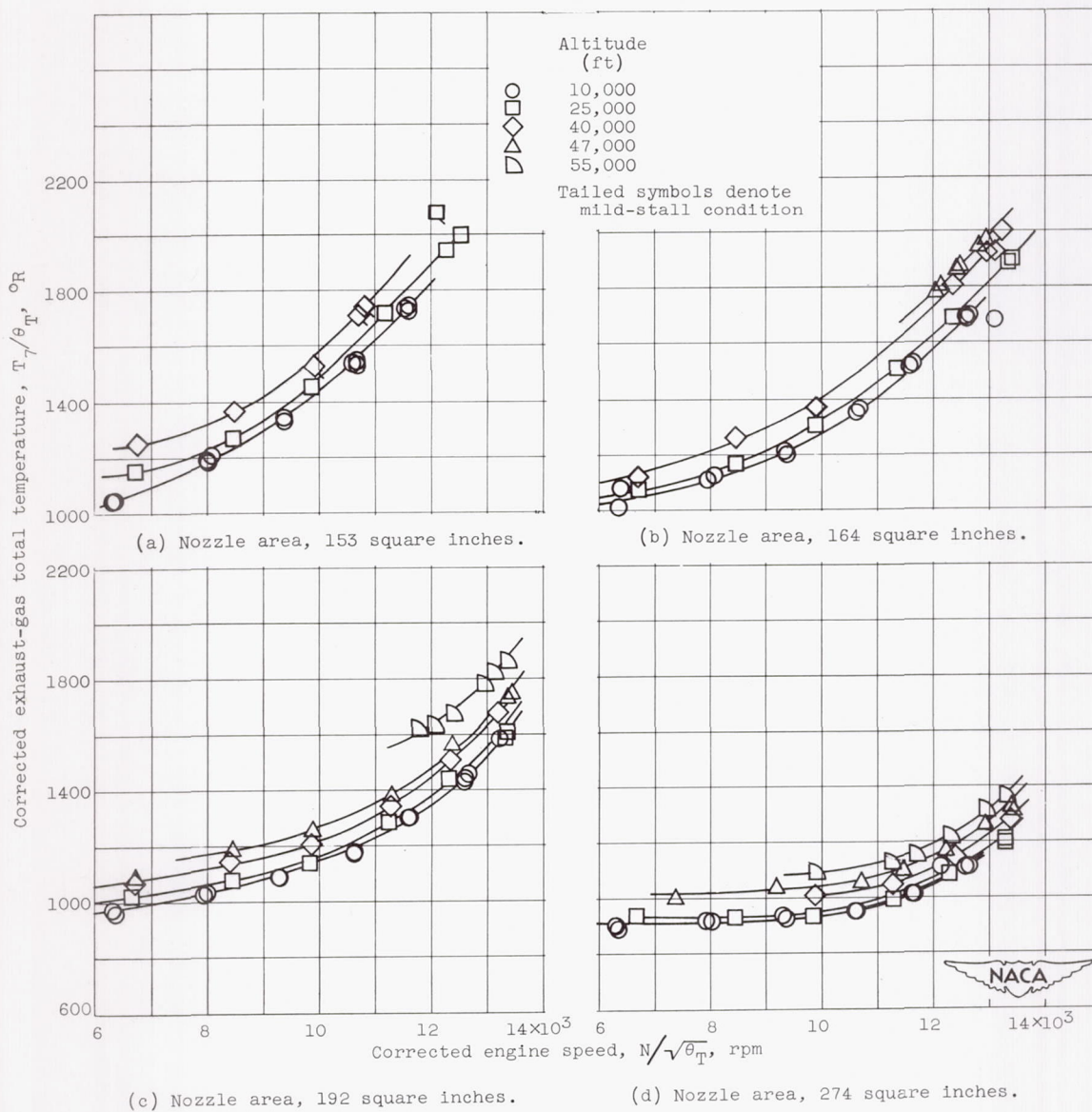


Figure 8. - Effect of altitude on variation of corrected exhaust-gas total temperature with corrected engine speed at flight Mach number of 0.528.

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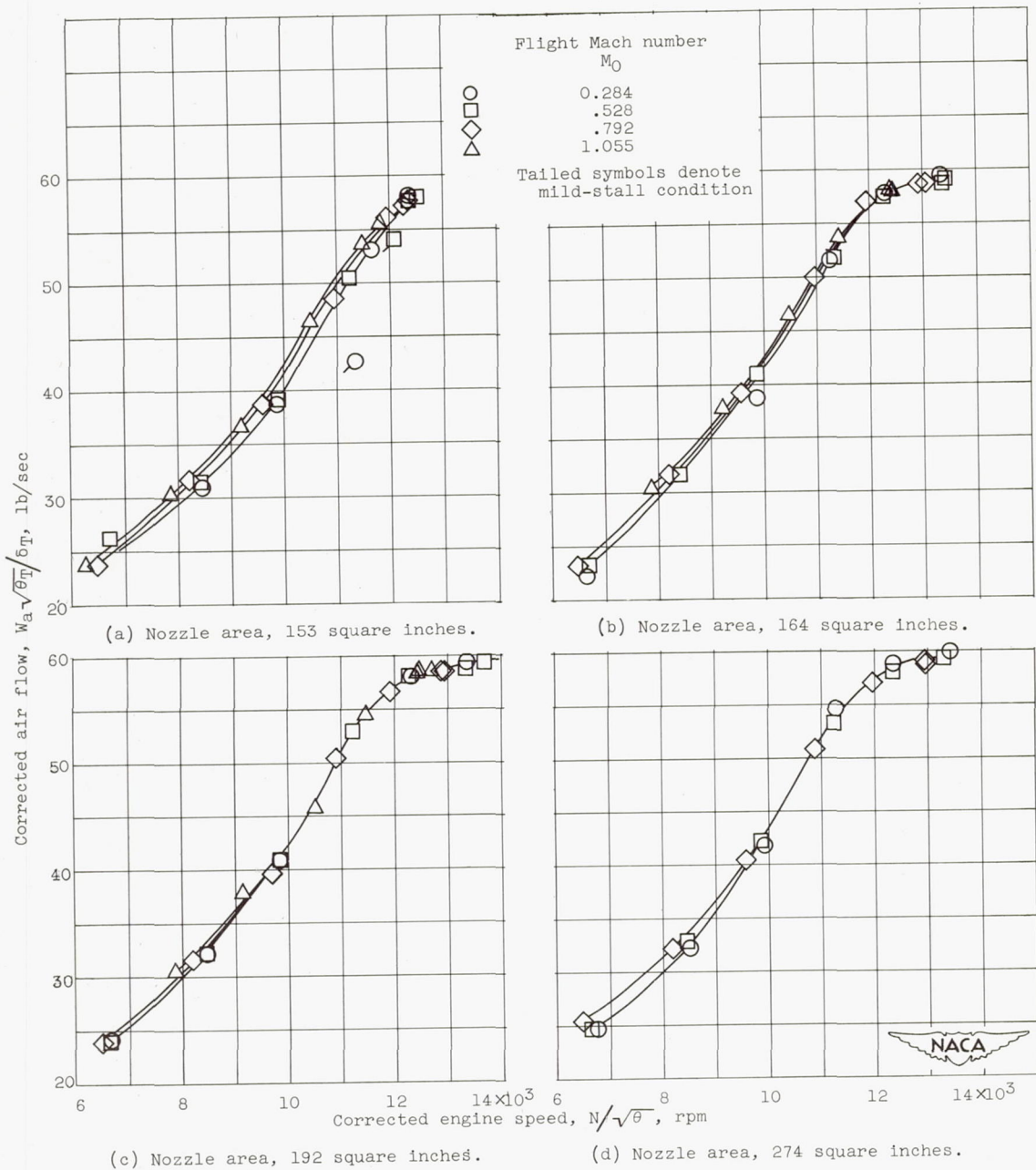


Figure 9. - Effect of flight Mach number on variation of corrected air flow with corrected engine speed at altitude of 25,000 feet.

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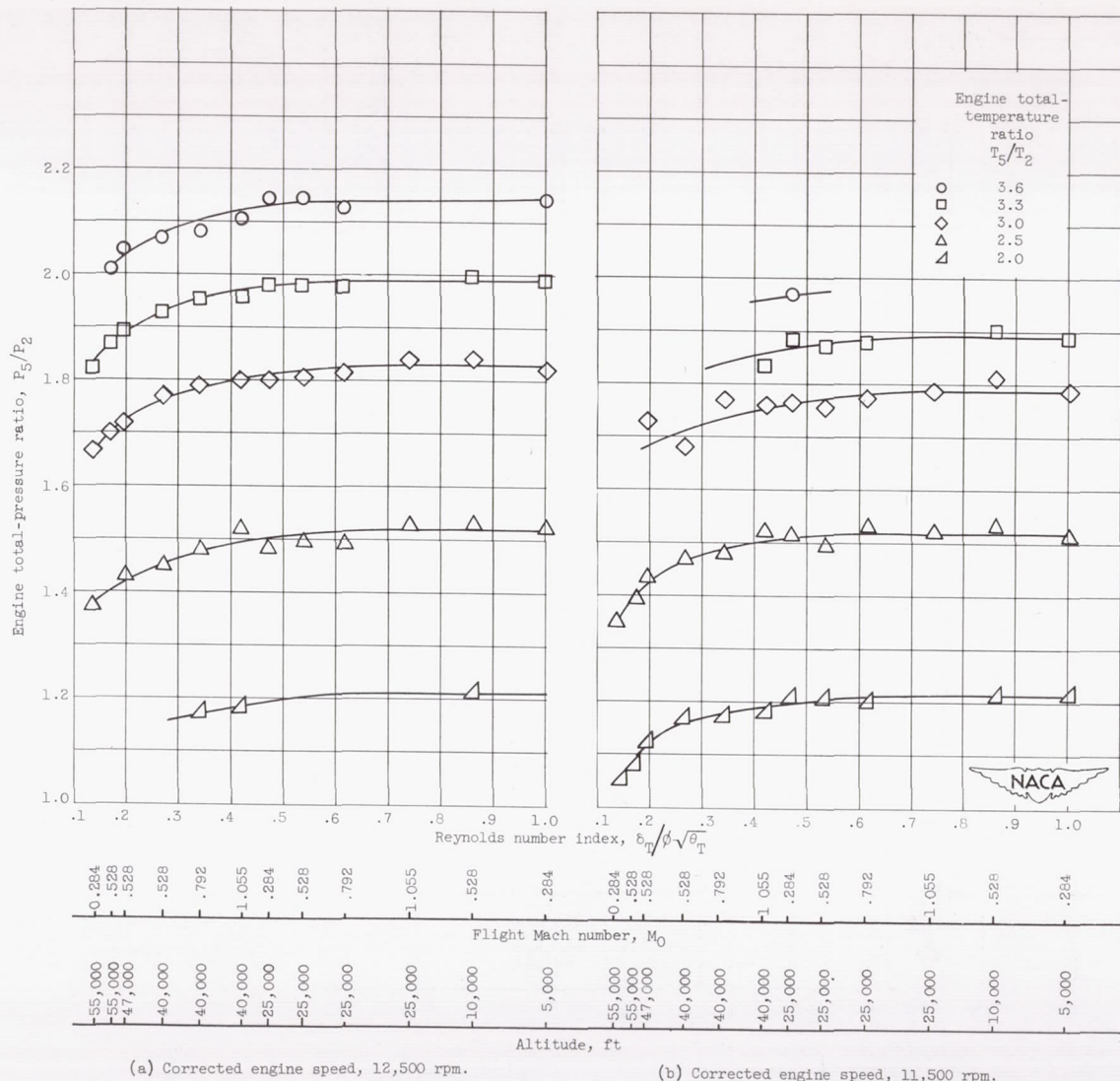


Figure 10. - Variation of engine total-pressure ratio with Reynolds number index for various engine total-temperature ratios.

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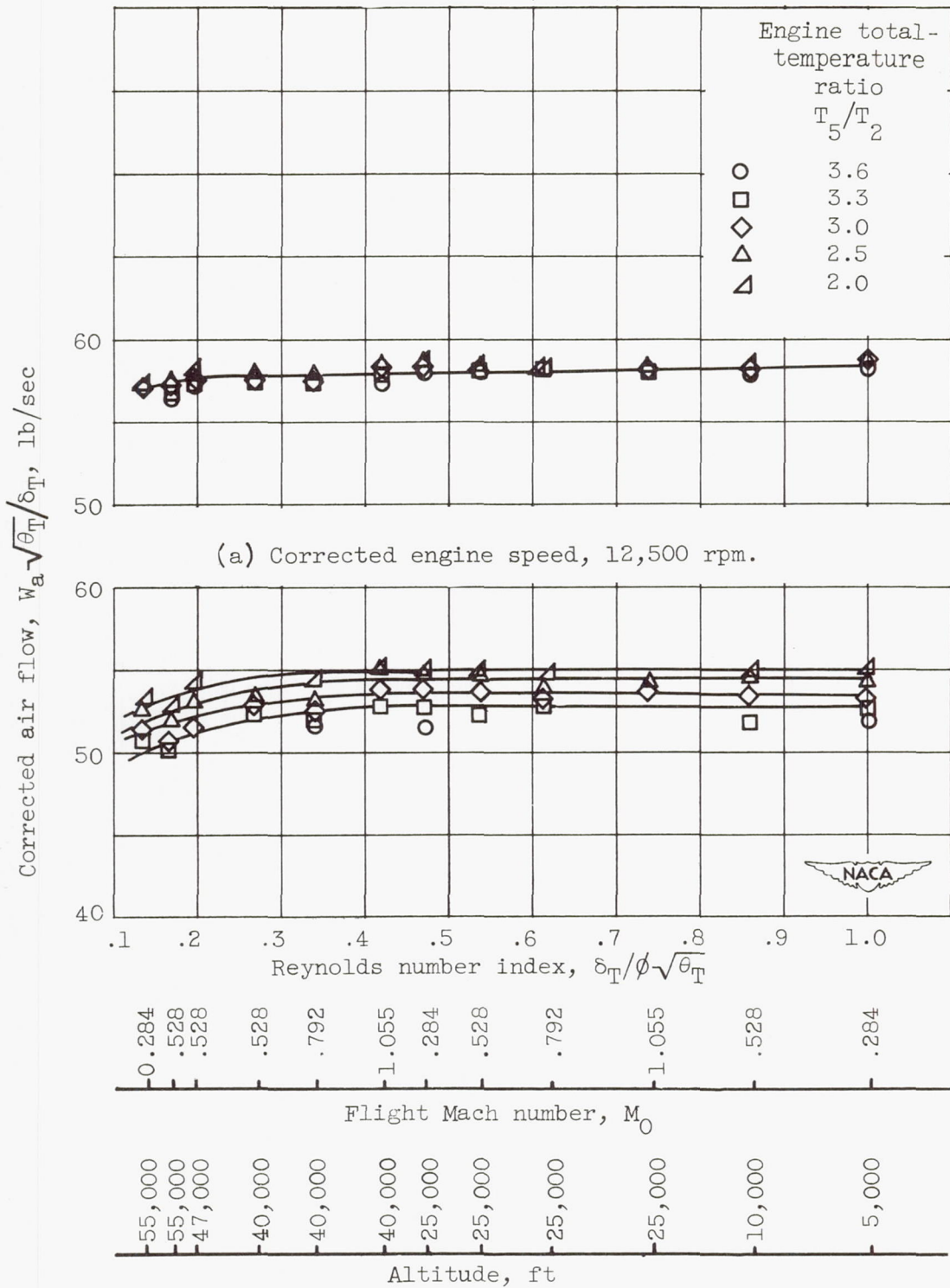


Figure 11. - Variation of corrected air flow with Reynolds number index for various engine temperature ratios.

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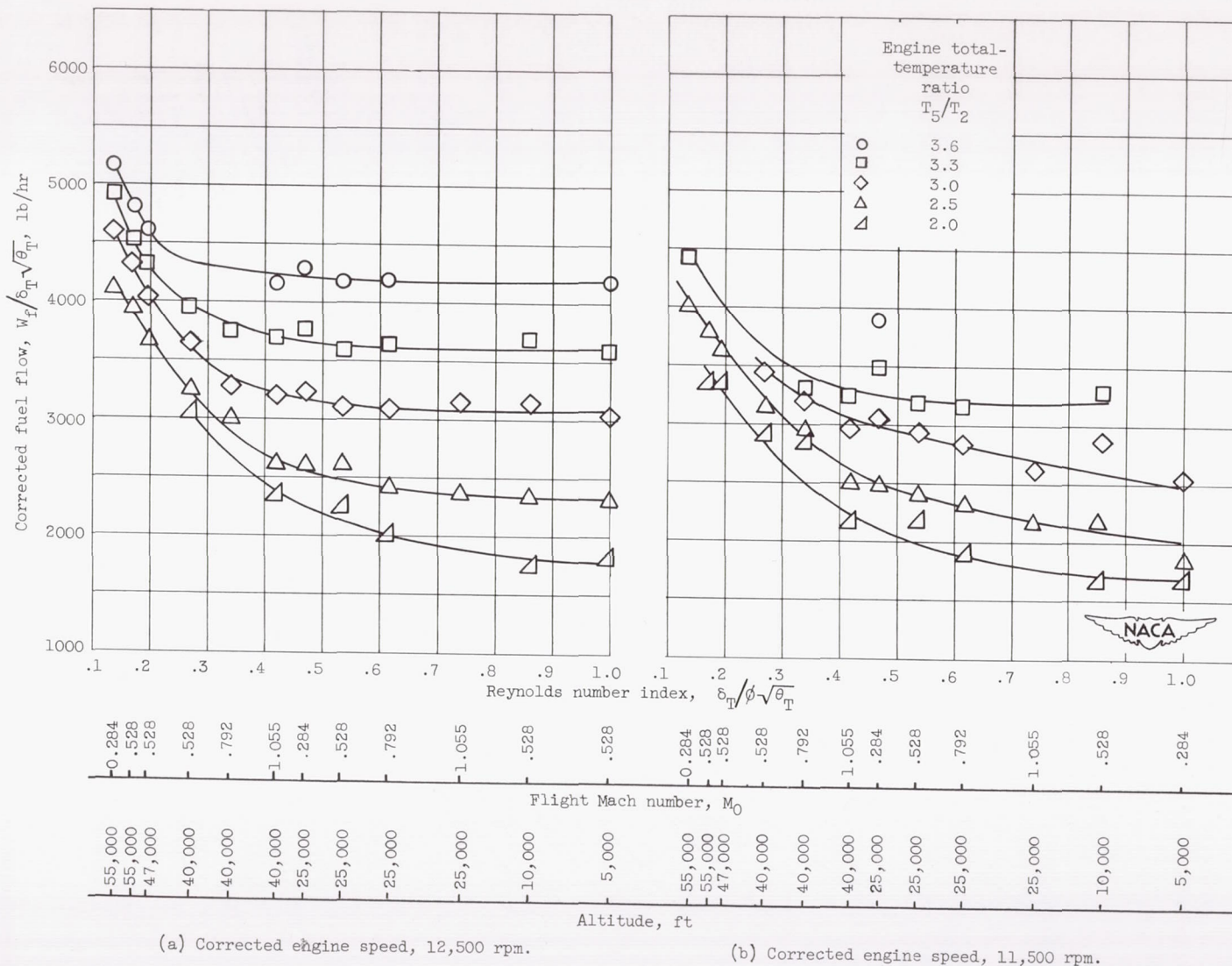


Figure 12. - Variation of corrected fuel flow with Reynolds number index for various engine total-temperature ratios.

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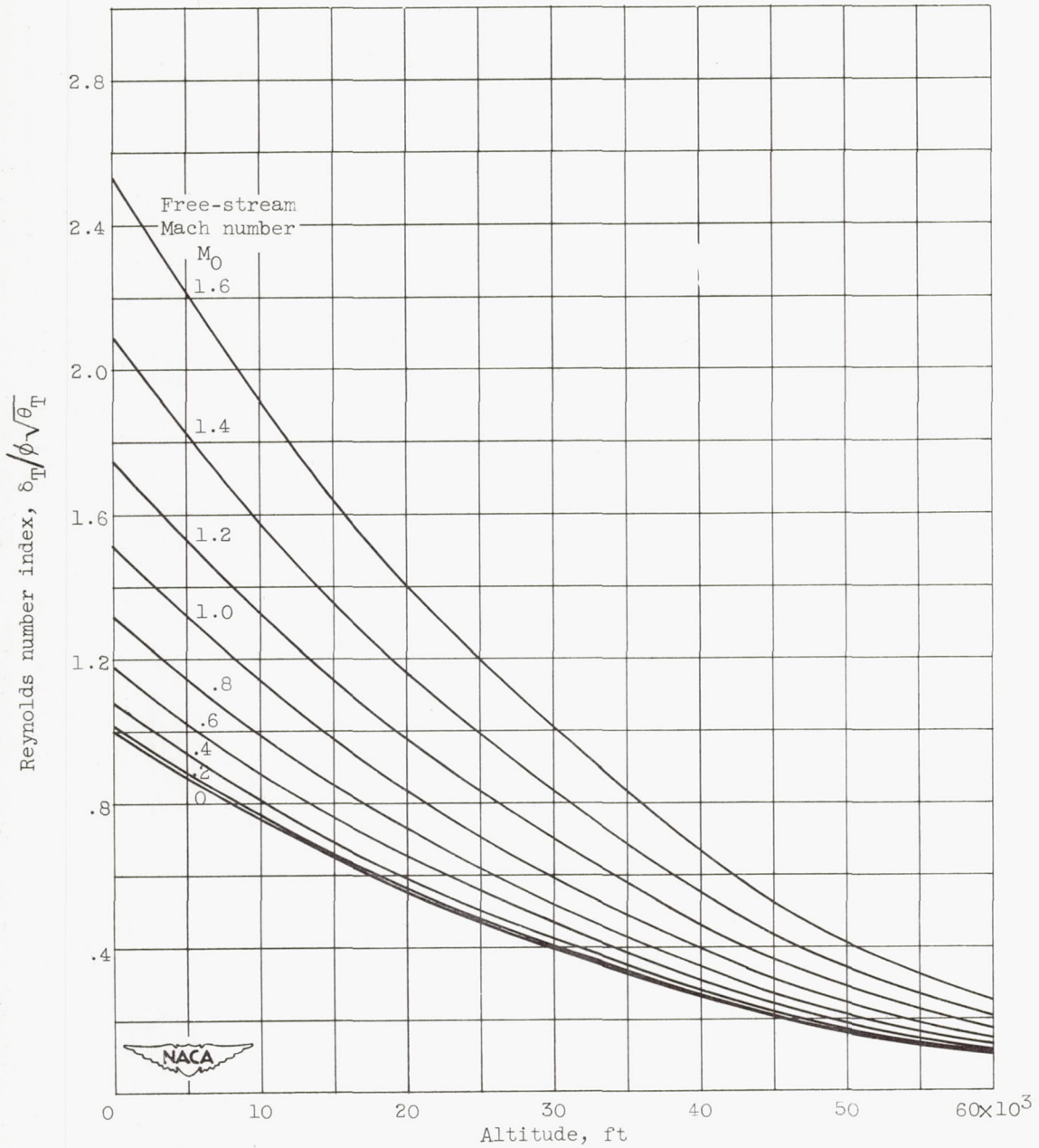


Figure 13. - Chart for evaluating Reynolds number index at altitude for flight Mach numbers varying from 0 to 1.6.

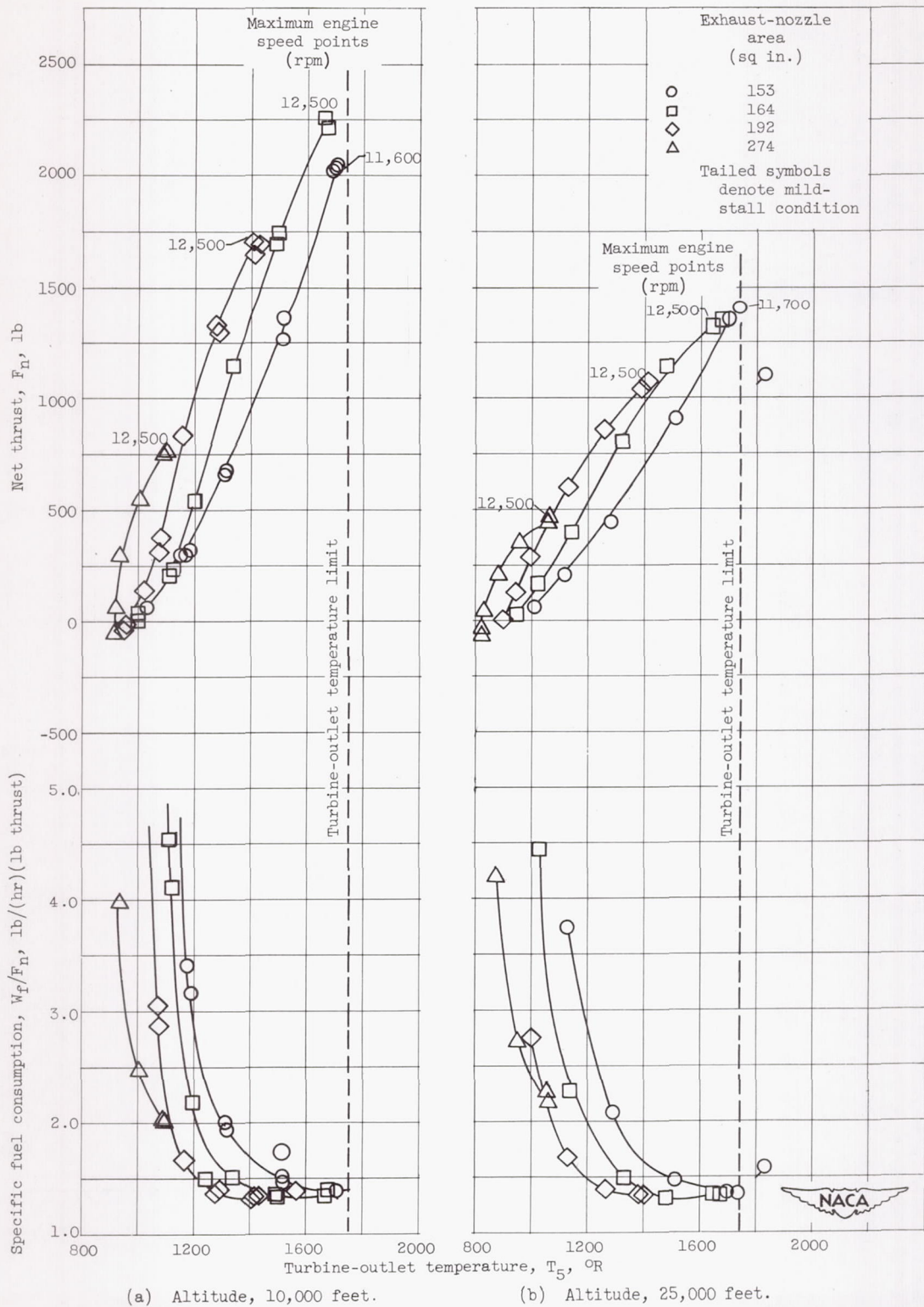
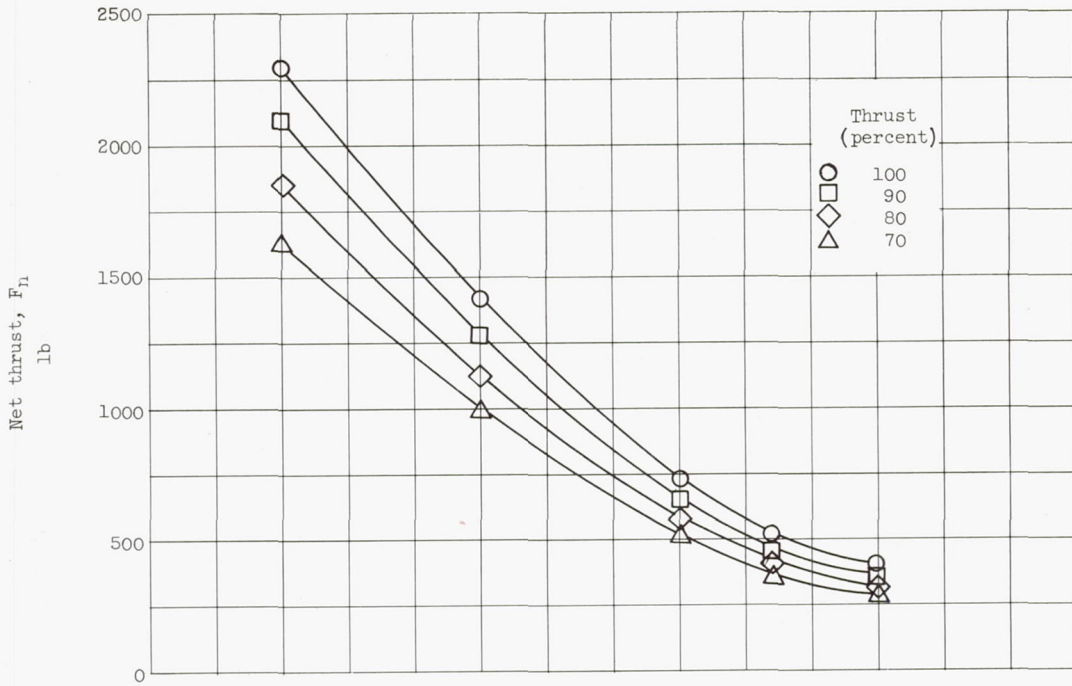
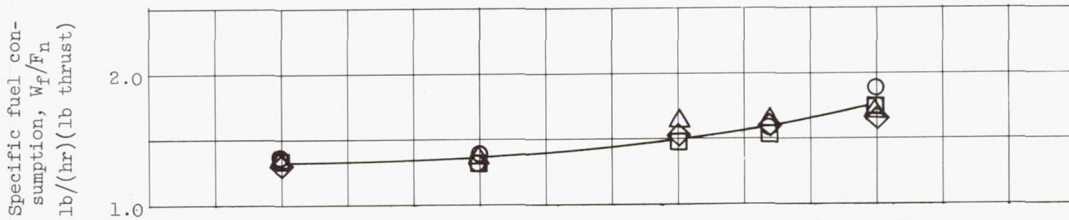


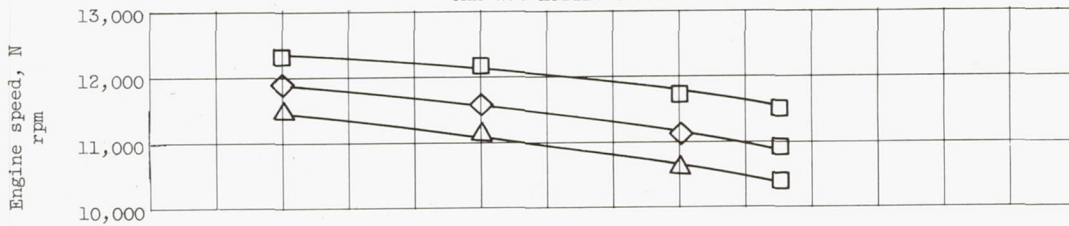
Figure 14. - Variation of specific fuel consumption and net thrust with turbine-outlet temperature for four nozzle areas at flight Mach number of 0.528.



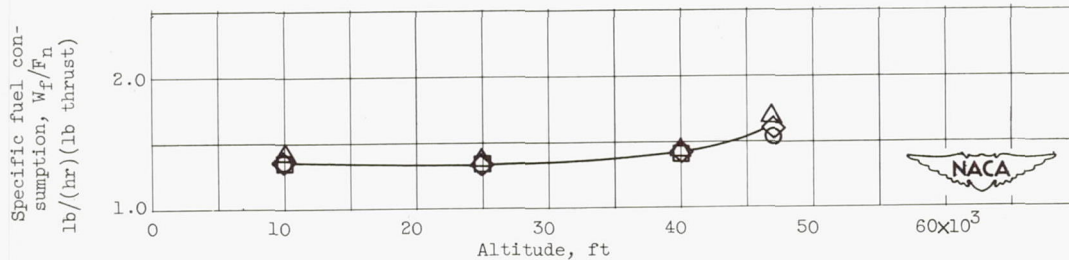
(a) Net thrust values obtained with both methods shown in (b) and (d).



(b) Specific fuel consumption obtained at rated engine speed and with varying exhaust-nozzle size.

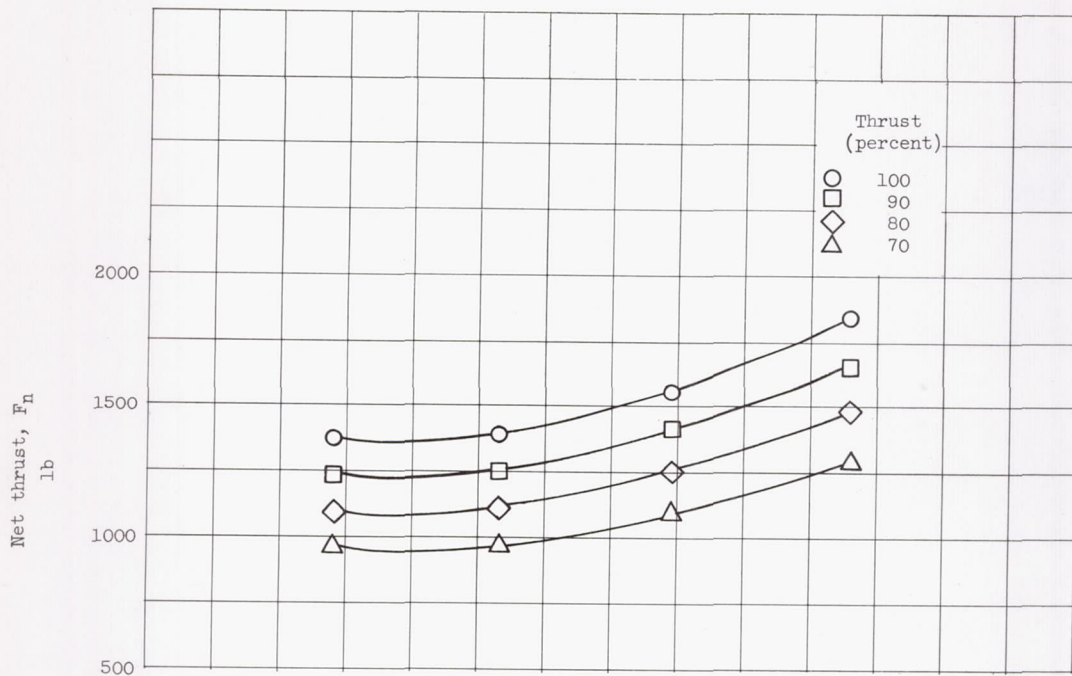


(c) Variation of engine speed at constant exhaust-nozzle area.

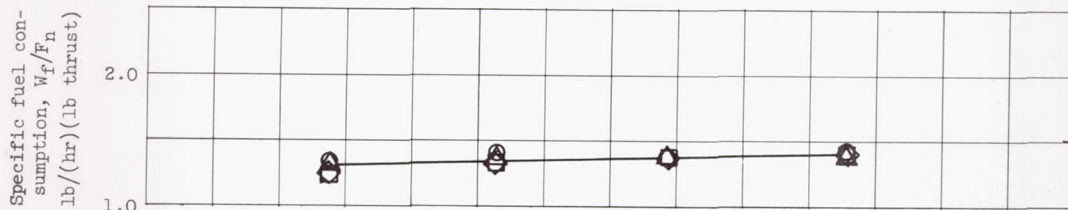


(d) Variation of specific fuel consumption at constant exhaust-nozzle area.

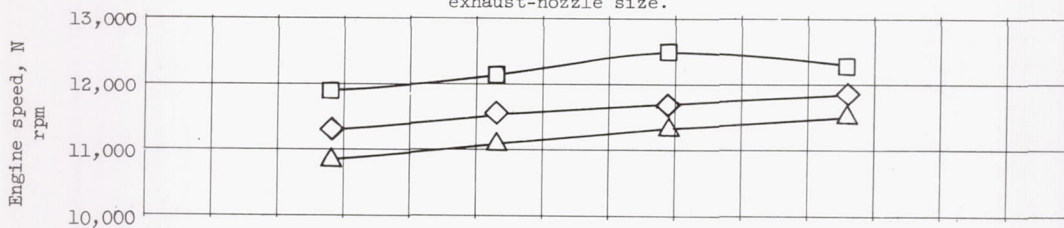
Figure 15. - Variation of engine variables with altitude at flight Mach number of 0.528.



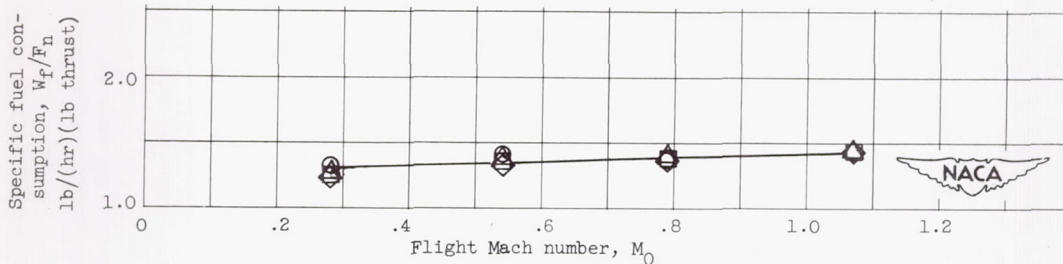
(a) Net thrust values obtained with both methods shown in (b) and (d).



(b) Specific fuel consumption obtained at rated engine speed and with varying exhaust-nozzle size.



(c) Variation of engine speed at constant exhaust-nozzle area.



(d) Variation of specific fuel consumption at constant exhaust-nozzle area.

Figure 16. - Variation of engine variables with flight Mach number at altitude of 25,000 feet.

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