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RESEARCH MEMORANDUM

ALTITUDE-WIND-TUNNEL INVESTIGATION OF A 3000-POUND-THRUST AXIAL-FLOW TURBOJET ENGINE

II - ANALYSIS OF COMPRESSOR PERFORMANCE

By Robert O. Dietz, Jr,, Joseph J. Berdysz, and Ephraim M. Roward

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ALTITUDE-WIND-TUNNEL INVESTIGATION OF A 3000-POUND-THRUST AXIAL-FLOW TURBOJET ENGINE

II - ANALYSIS OF COMPRESSOR PERFORMANCE

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SUMMARY

An investigation was conducted in the NACA Cleveland altitude wind tunnel to determine the performance of an original and a modified compressor operating as an integral part of a turbojet engine. Compressor performance data were obtained while the turbojet engine was run over its full operable range of engine speeds at various simulated altitudes and flight Mach numbers. The use of four different exhaust-nozzle-outlet areas on the engine extended the range of compressor operation.

Operating lines for both the original and the modified compressors lay on the high air-flow side of the region of maximum compressor efficiency. Over most of the normal operating range of corrected engine speeds, compressor efficiency decreased as (a) the altitude was raised, (b) the flight Mach number was raised, (c) the exhaust-nozzle-outlet area was increased. Variations in the compressor performance characteristics with flight conditions were attributed to the effects of changes in Reynolds number.

A more nearly uniform radial distribution of the axial velocity at the compressor outlet and slightly higher stator-stage static-pressure ratios were obtained with the modified compressor than with the original compressor. Higher maximum efficiencies were obtained with the modified than with the original compressor.

INTRODUCTION

Performance data for two 11-stage axial-flow compressors have been obtained from an investigation of a complete turbojet engine having a thrust rating of 3000 pounds over a range of simulated flight conditions in the NACA Cleveland altitude wind tunnel. The use of four different exhaust-nozzle-outlet areas extended the

range of engine-component operations over which performance data oould be obtained. An analysisof the turbine performance data obtained in this investigation is presented in reference 1.

The original compressor and a modified compressor each operating as an integral part of the engine were investigated and the results are presented herein. Compressor characteristics were Investigated to determine the effects of variations in altitude, flight Mach number, and exhaust-nozzle-outlet area on compressor performance and to determine how effectively the compressors were used in the engine. Compressor performance is presented as a function of air flow and engine speed, both of which are corrected to sea-level conditions. Performance characteristics of the original and modified compressors are compared and the effect of Reynolds number on both compressors is shown.

COMPRESSORS

The original and modified compressors are 11-stage axial-flow types and handle about 58.5 pounds of air per second at the rated engine speed Of 12,500 rpm with standard sea-level conditions at the Inlet. At this condition the compressor pressure *ratio* is about 3.8.

Each compressor ha8 five principal components: inlet section, split stator casing, rotor, blading, and annular diffuser (fig. 1). Both compressors are similar with the exception of the eleventh-rotor-stage blading. In the revised configuration, the compressor was modified by reducing the load on the eleventh-stage rotor blades in order to obtain a more nearly uniform velocity distribution at the compressor outlet. This reduction was accomplished by twisting the blades in the direction of reduced angle of attack 3° at the midspan and 6° at the tip.

The length of the 11-stage rotor (fig. 1) from the front face Of the first-stage rotor disk to the rear face Of the eleventh-stage rotor disk is 29.7 inches and the blading has *a* constant outside diameter of 16.9 inches. The hub diameter increases from 9.4 inches at the first and second stages to 14.0 inches at the tenth and eleventh stages.

Air enters the compressor through a single row Of inlet guide vanes, which are fitted in the inlet section. After being compressed, the sir is discharged through two rows Of straightening vanes into the annular diffuser. The stator blading, the straightening vanes, and the annular diffuser are parts of the split stator

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casing assembly (fig. 2). The stator blades are completely Shrouded. Labyrinth air seals that prevent leakage around the stator blades are formed by the inner shrouds, which fit into grooves in the compressor rotor.

INSTALLATION AND INSTRUMENTATION

The 24C turbojet engine has an 11-stage axial-flow compressor (described in the preceding section), a double-annulus combustion chamber, a two-stage reaction turbine, a tail pipe, and an exhaust nozzle. The rated thrust of approximately 3000 pounds is obtained under sea-level etatio conditions at a corrected engine speed of 12,500 rpm. The fuel flow at this condition is approximately 3200 pounds per hour.

The engine was installed in a wing section that extended across the 20-foot-diameter teat section of the altitude wind tunnel (fig. 3). Compressor-inlet pressures corresponding to flight at high speeds were obtained by introducing from the tunnel make-up air system dry refrigerated air that was throttled from approximately sea-level pressure to the desired pressure at the engine inlet. A frictionless slip joint in the make-up air duct (fig. 3) permitted the measurement of thrust with the wind-tunnel balance scale system.

A different engine configuration was used with each of the two compressors installed. The blocking area of the screens at the combustion-chamber inlet was reduced in the modified engine. The combustion chamber in the modified engine differed from the original combustion chamber in that the shape and location of the sir-inlet holes in the liner were changed to improve mixing in the secondary combustion Zone. The total area of the air-inlet holes was the same in both combustion chambers. No change was made in the engine turbine installation. Changes in the modified engine made possible an increase in the maximum allowable turbine-outlet gas temperature and a decrease in the exhaust-nozzle-outlet area at which limiting turbine-outlet temperatures were obtained from 183 to 171 square inches.

Four engine exhaust-nozzle configurations were used with each caupressor. With one configuration, exhaust gases were discharged from the straight tail pipe, which had an area of 330 inches. With the other three configurations, exhaust nozzles 20 inches in length were installed. These nozzles tapered uniformly from an area of 330 square inches to outlet areas of 232, 189, and 171 inches for the engine with the modified compressor and to 280, 232, and 183 square inches for the engine with the original compressor.

Instrumentation for the measurement of pressures and temperatures wss installed at several stations through the engine (fig. 4). A cross section of the plane of survey at the compressor inlet (station 2) showing the location of the compressor-inlet instrumentation is shown in figure 5. Compressor stator-stage static pressures were measured by wall orifices located between the stator blades of each stage. Several of these wall orifices srs shown in figure 2. Location of the instrumentation in the plane of survey at the compressor outlet (station 4) is shown in figure 6. All instrumentation was installed in a fixed position aud could not be rotated to measure whirl velocity components.

RANGE OF INVESTIGATION

Caupressor performance data were obtained for engine operation at simulated altitudes of 15,000, 25,000, 35,000, aud 45,000 feet and a flight Mach number of 0.53. At a simulated altitude of 25,000 feet, the engine was operated at flight Mach numbers of 0.53, 0.86, and 1.08. At each simulated flight condition, the engine was run over the full operable range of speeds. This procedure wss employed with each of four exhaust-nozzle configurations for both compressors.

RESULTS AND DISCUSSION

The compressor performance data presented represent a wide range of compressor operation. Data obtained with the original and modified compressors operating as integral parts of the turbo-Jet engine at various simulated flight conditions and with four exhaust-nozzle configurations were plotted in the form of compressor operating lines; that is, compressor pressure ratio was plotted as a function of corrected air flow and corrected engine speed. Typical examples of these curves are given in figure 7. These data were also plotted to determine the relation between compressor efficiency (equation (1) of the appendix) aud corrected air flow (fig. 8). Curves containing the data for all operating conditions were used to make composite plots of compressor performance characteristics, au example of which is shown in figure 9.

With both original and modified compressors, a change in altitude caused the efficiency contour8 and lines of constant corrected engine speed to shift with respect to the basic parameters of the compressor-characteristic curve (fig. 9). A similar shift of the efficiency contours aud lines of corrected engine

speed was caused by a variation in flight Mach number (fig. 10). Scatter of the points taken from cross plots of the data for the modified. compressor precludes separation of the constant corrected-engine-speed lines (fig. 10(b)). For clarity, some Intermediate efficiency contours snd the contour of maximum efficiency are omitted in figures 9 and 10.

An increase in altitude caused a change in the compressor characteristic curves that was qualitatively similar to the change produced by a decrease in flight Mach number. Roth changes cause decreases in the Reynolds number of the air flowing through the compressor. The change in Reynolds number for the range of altitudes presented is more than twice that for the range of flight Mach numbers. Figures 9 and 10 show that the change in the characteristic curves is commensurate with the change in Reynolds number. The effects of changes In altitude and flight Mach number on the compressor-performance characteristics may be attributed to the changes in the aerodynamic characteristics of the compressor blading that accompany changes in Reynolds number.

A comparison of the performance characteristics of the original and modified compressors at an altitude of 25,000 feet and a flight Mach number of 0.55 is presented in figure 11. Compressor operating lines with the minimum-area exhaust nozzle for each compressor are superimposed on these characteristic curves. The operating line for the modified compressor is slightly to the left of the operating line for the original compressor. The change in the compressor characteristics is such that the maximum efficiency obtainedwith the modified compressor was about 1 percent higher than that obtained with the original compressor.

Typical examples of the radial distribution of the axial velocity at the compressor outlet for both the original and modified compressors show that the attempt to reduce the nonuniformity of the axial velocity at the compressor outlet was successful (fig. 12).

The ratio of the static pressure at any stage to the static pressure at the compressor inlet (station 2) is slightly higher for the modified compressor than for the original compressor (fig. 13). For the first stage, the static pressure was less than the static pressure at the inlet because of the drop in static pressure due to increased air velocity through the inlet guide vanes.

The previous discussion has applied to compressor characteristics in general. Effects of variations in altitude, flight

Mach number, and exhaust-nozzle-outlet area on the operation of the modified compressor in the engine are shown in figures 14 to 18. The relative changes in the position of the operating lines and the efficiency contours were such that, at a given corrected engine speed, adecrease in efficiency was experienced as the altitude was raised throughout the full range of corrected engine speed (figs. 14 and 15).

The relative positions of the efficiency contours snd the operating lines in figures 16 and 17 show that the compressor efficiency decreased as the flight Mach number was incressedfor corrected engine speeds below 11,200 rpm. Increasing the exhaust-nozzle-outlet area resulted in a decrease in compressor efficiency (figs. 18 and 8).

Compressor efficiencies at engine speeds greater than 0.8 rated speed, (10,000 rpm) varied from 0.80 to 0.86. Maximum efficiencies occurred between corrected airflows of 51.0 and 55.5 pounds per second, corresponding to corrected engine speeds of 11,100 and 12,000 rpm, respectively.

This investigation indicated that the compressor operating lines fell. on the high air-flow side of the region of maximum efficiency (figs. 14, 16, aud 18). The operating line for the compressor in the engine with the minimum exhaust-nozzle-outlet area fell closest to the region of maximum efficiency (fig. 18).

SUMMARY OF RESULTS

From an investigation of a complete turbojet engine having a thrust rating of 3000 pounds in the NACA Cleveland altitude wind tunnel under simulated conditions of altitude and flight Mach number, the following results of compressor performance were obtained:

1. The compressor operating lines fell on the high air-flow side of the region of maximum efficiency. The operating line for the compressor in the engine with the minimum exhaust-nozzle-outlet area fell closest to the region of maximum efficiency.

2. Over most of the normal operating range of corrected engine speeds, compressor efficiency decreased as (a) the altitude was raised, (b) the flight Mach number was raised, (c) the exhaust-nozzle-outlet area was increased.

3. Variations in the compressor performance characteristics with altitude or flight Mach number were attributed to changes in aerodynamic characteristics of the caupressor blading that accompany changes in Reynolds number.

4. Modification of the eleventh-stage compressor rotor blading resulted in a more nearly uniform radial distribution of the axial velocity at the compressor outlet and slightly Increased the stator-stage static-pressure ratio. Higher maximum efficiencies were obtained with the modified canpressor than with the original compressor.

5. Compressor efficiencies at engine speeds greater than 0.8 rated speed (10,000 rpm) varied from 0.80 to 0.86. Maximum efficiencies occurred between corrected air flows of 51.0 and 56.5 pounds per second, corresponding to corrected engine speeds of 11,100 and 12,000 rpm, respectively.

Flight Propulsion Research Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio.

APPENDIX - METHOD OF CALCULATION

Symbols

Ths symbols used in this analysis are:

c_p specific heat at constant pressure, Btu/(lb)(°R)

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

J mechanical equivalent of heat, 778 ft-lb/Btu

M_0 flight Mach number

N engine speed, rpm

P total pressure, lb/sq ft absolute

P static pressure, lb/sq ft absolute

T total temperature, °R

V axial velocity, ft/sec

W_a air flow, lb/sec

γ ratio of specific heat at constant pressure to specific heat at constant volume

8 ratio of compressor-inlet total Pressure to NACA standard eea-level pressure

η_c compressor efficiency

θ ratio of compressor-inlet absolute total temperature to NACA standard sea-level absolute temperature

Subscripts:

2 compressor inlet

3 compressor stator stages

4 canpressor outlet

c canpressor

The stations to which the numerical subscripts refer are shown in figure 4.

The following parameters are used in generalizing the results to NACA standard atmospheric conditions at sea level:

$\frac{W_a\sqrt{\theta}}{\delta}$ corrected air flow, lb/sec

$\frac{N}{\sqrt{\theta}}$ corrected engine speed, rpm

Calculations

Static temperature was calculated from indicated temperature by assuming a thermocouple impact recovery factor of 0.85. Total temperatures at the compressor inlet and at the compressor outlet were assumed to be equal to the corresponding indicated temperatures because the average velocities were relatively low at these stations. This assumption introduced an error of less than 2° F.

Air flow was computed from the pressures and the temperatures measured at ths cowl inlet (station 1) by means of the continuity equation.

Compressor efficiency. - The following equation was used to calculate the adiabatic temperature-rise compressor efficiency:

$$\eta_c = \frac{\left(\frac{P_4}{P_2}\right)^{\frac{\gamma_c - 1}{\gamma_c}} - 1}{\frac{T_4}{T_2} - 1} \tag{1}$$

where γ_c is the ratio of specific heats corresponding to the average total temperature of the air flowing through the compressor.

Compressor-outlet axial velocity. - The axial velocities at the compressor outlet were calculated from the relation

$$V = \sqrt{2Jgc_pT_4\left[1 - \left(\frac{p_4}{P_4}\right)^{\frac{\gamma_4-1}{\gamma_4}}\right]} \qquad (2)$$

where

P_4 total pressure obtained by averaging pressure reading8 of instrumentation located at each integral distance from inner wall of compressor-outlet annulus

T_4 average of all indicated temperatures measured at station 4

p_4 average of all static pressures measured at station 4

Average values of the indicated temperature and the static pressure were used because the temperature and static-pressure distributions were relatively uniform.

REFERENCE

1. Conrad, Earl W., Dietz, Robert O., Jr., and Golladay, Richard L.: Altitude-Wind-Tunnel Investigation of a 3000-Pound-Thrust Axial-Flow Turbojet Engine. I - Analysis of Turbine Performance. NACA RM No. E8A23, 1948.

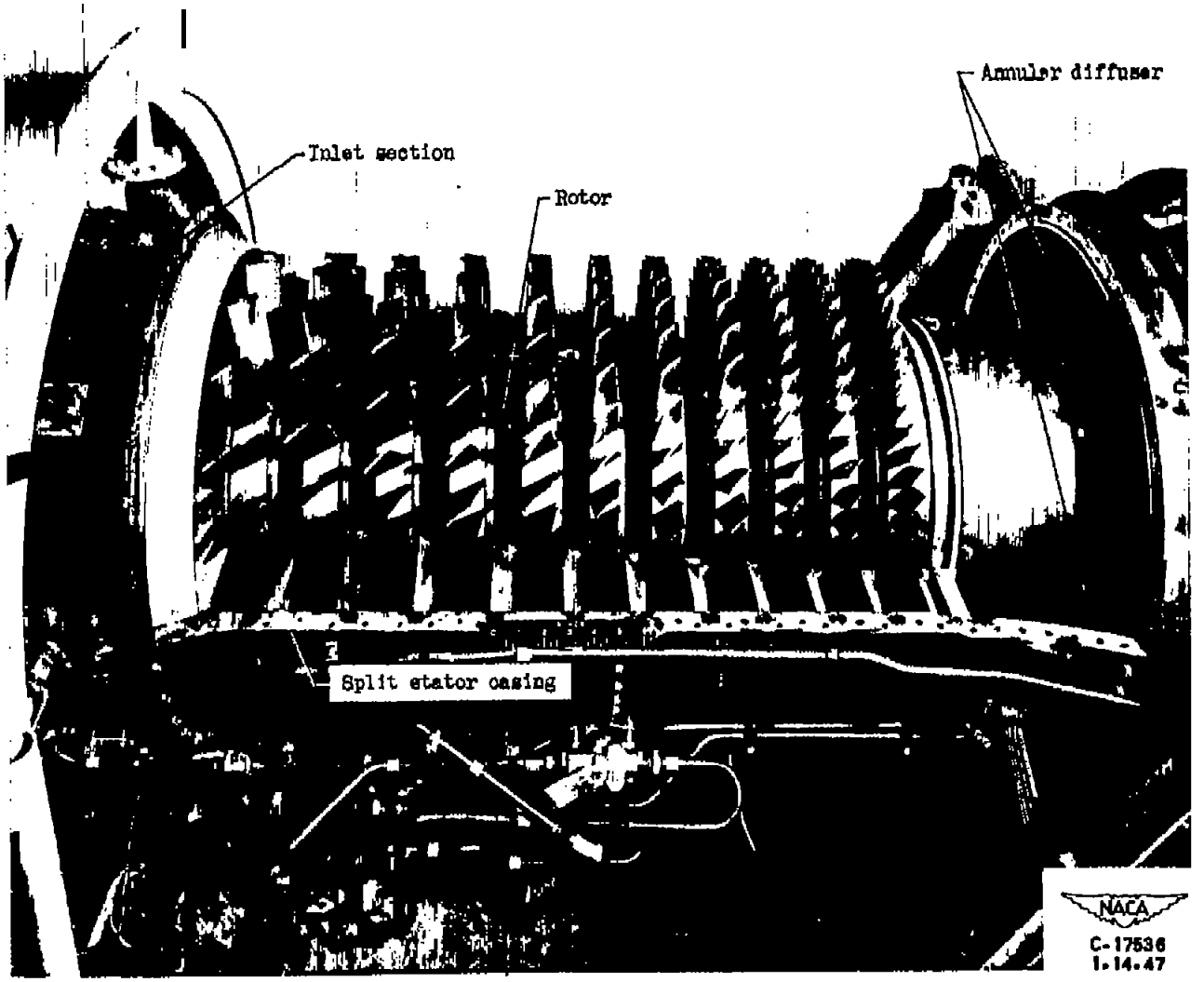


Figure 1. -Compressor installation with [illegible] stator casing removed.

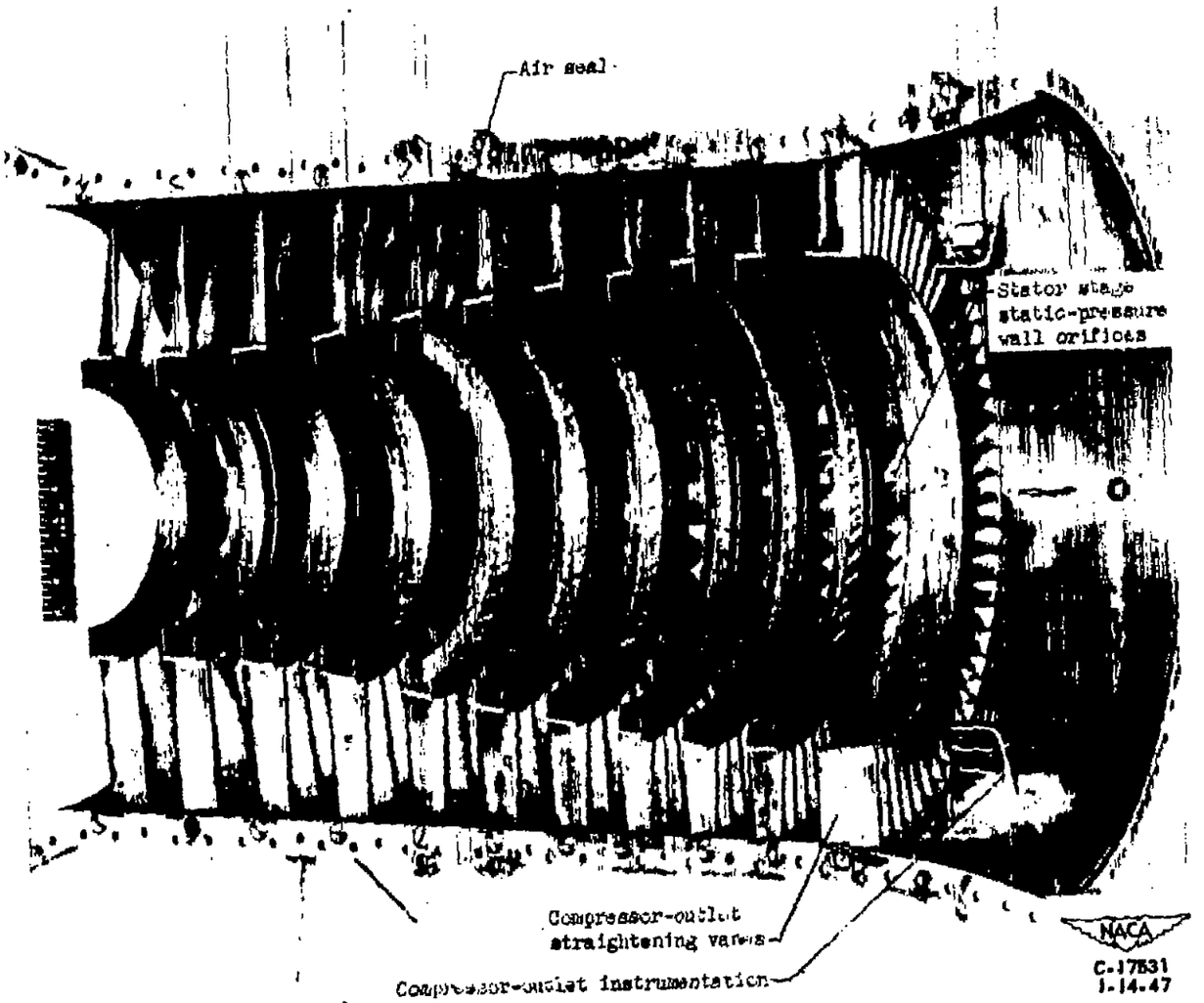


Figure 2. - Interior of compressor-casing assembly showing shrouded stator blading and two rows of outlet straightening vanes.

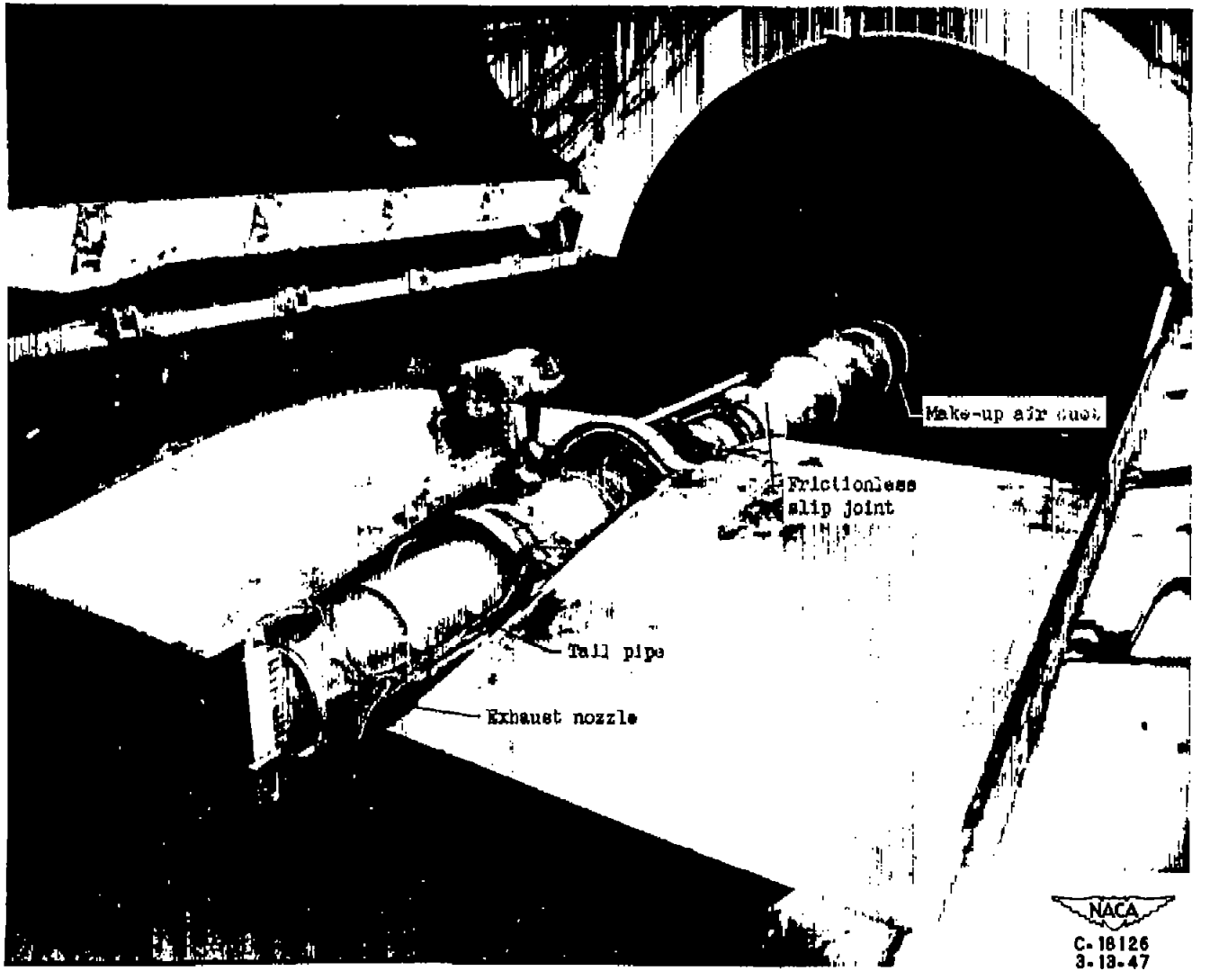
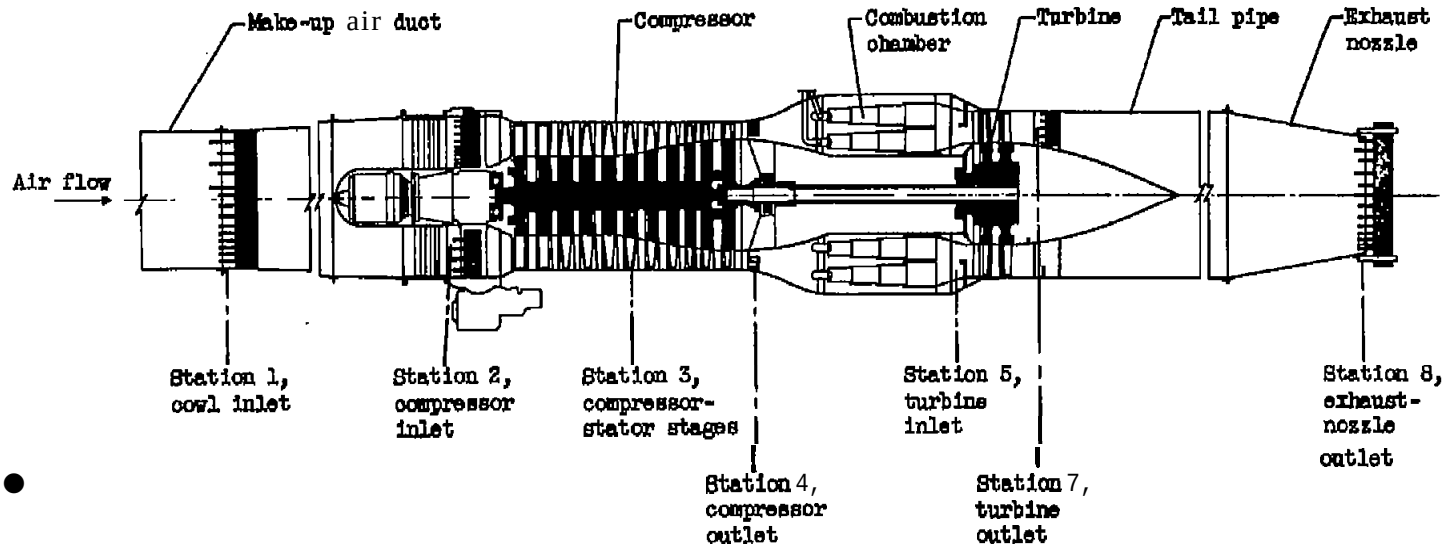


Figure 3. - Installation of turbojet engine in wing section in 20-foot-diameter test section of altitude wind tunnel.



Station	Total-pressure tubes	Static-pressure tubes	Static-pressure wall orifices	Thermocouples
1	16	3	4	6
2	16	8	4	6
3			10	
4	15	6	6	6
5	4			
7	10		3	55
8	8	3	4	16

Figure 4. - Cross section or turbojet engine showing stations at which instrumentation was installed.

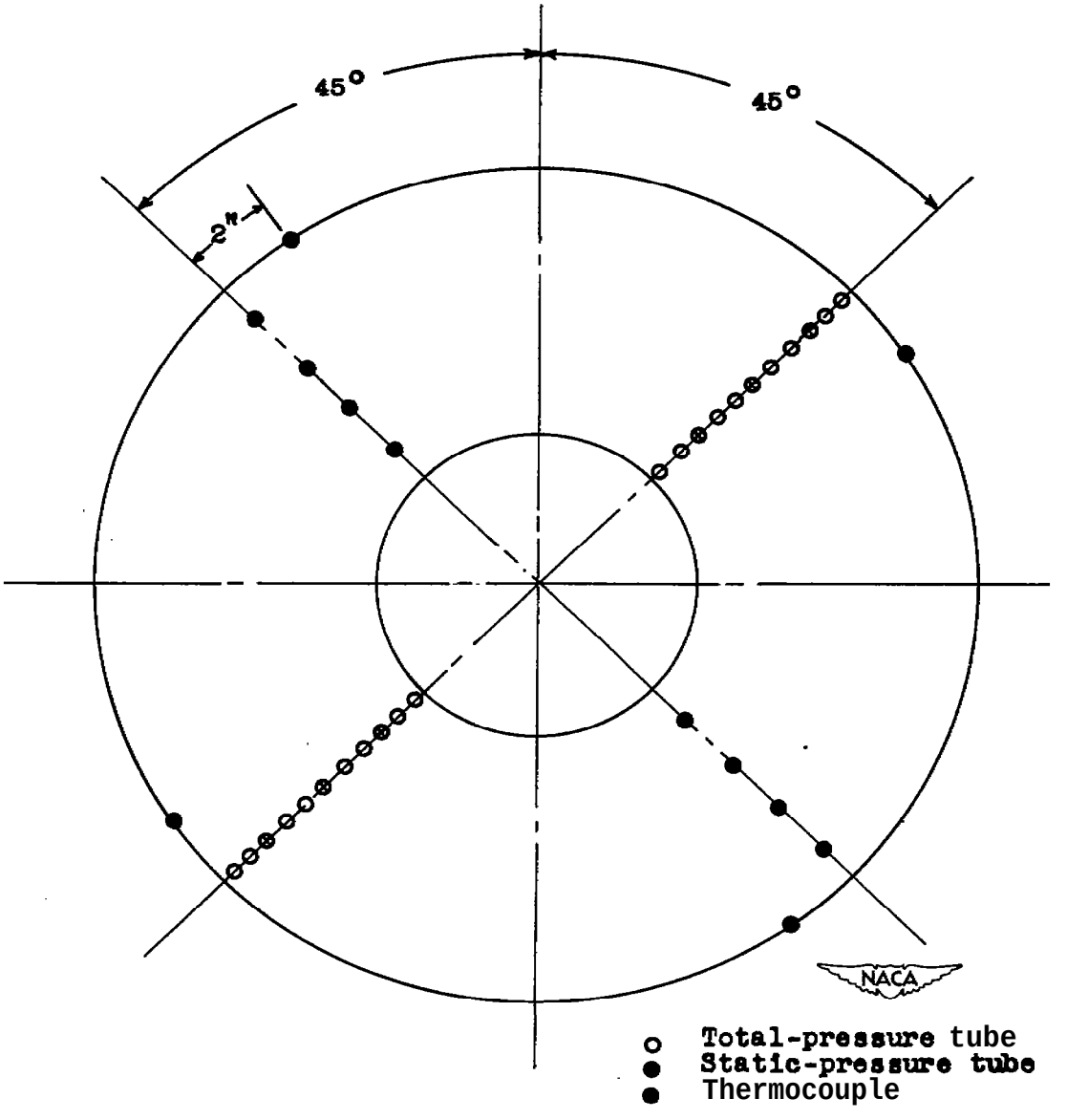


Figure 5. - Instrumentation at compressor inlet, station 2, $\frac{3}{4}$ inch behind rear flange of oil cooler.

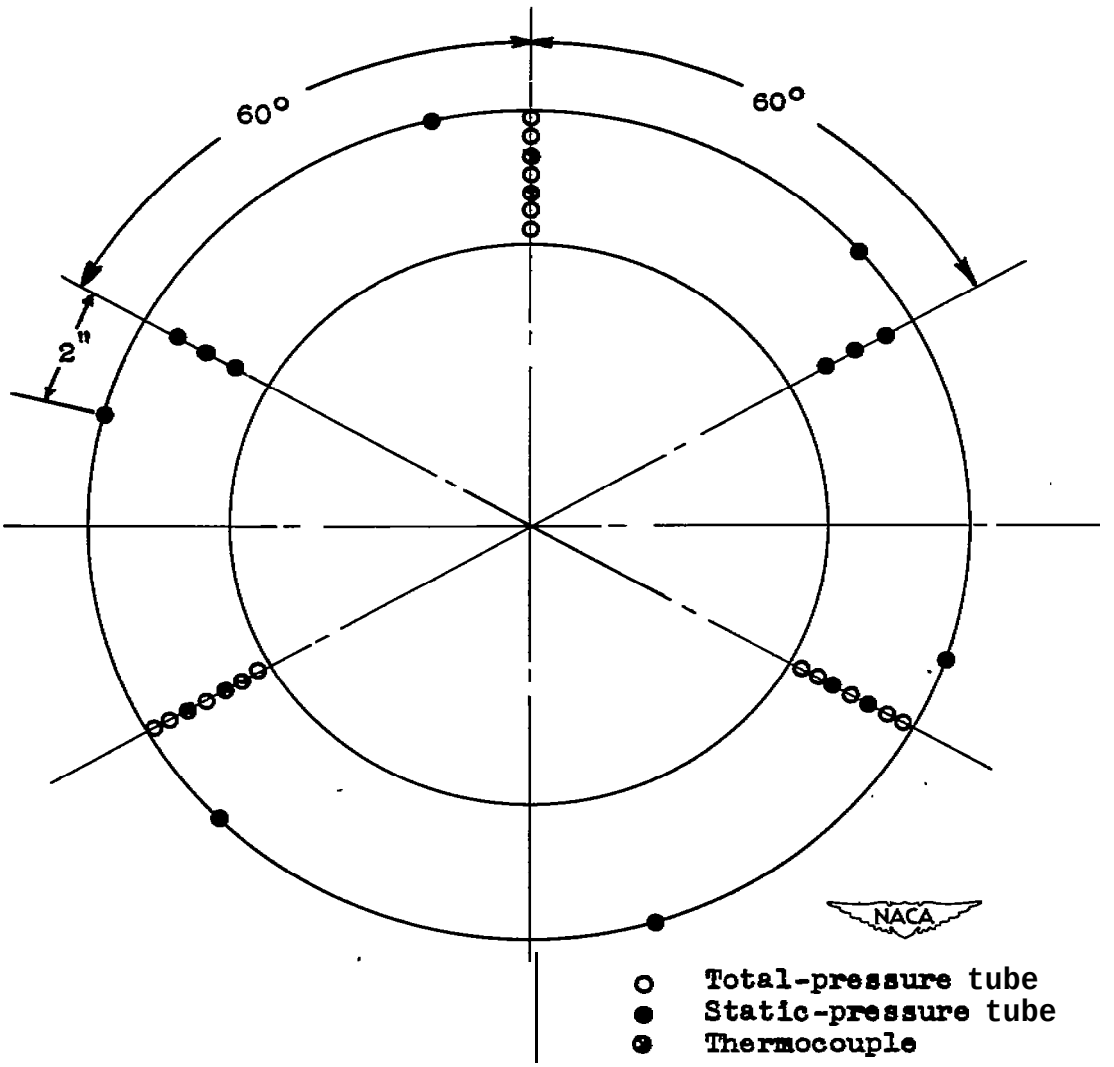
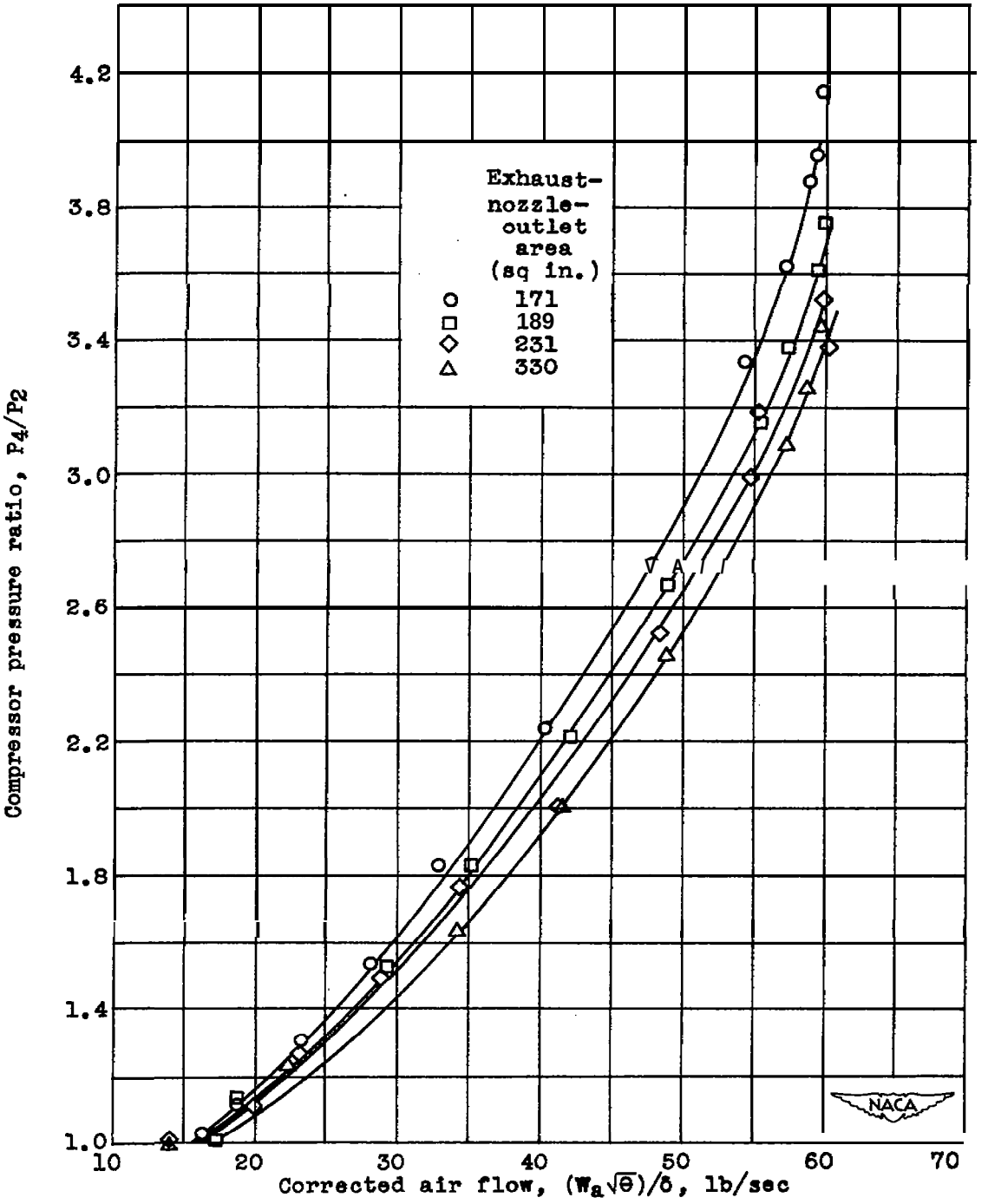
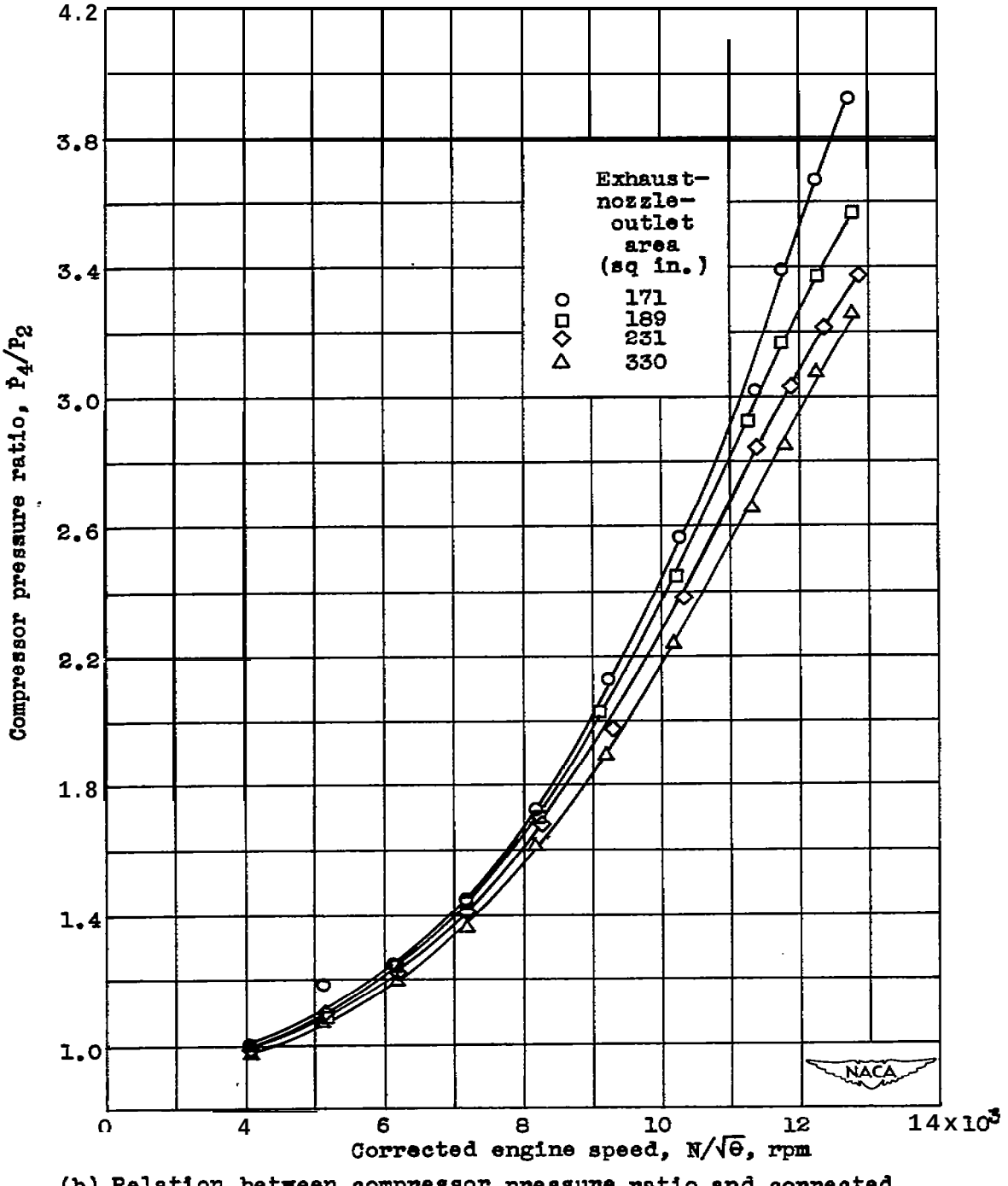


Figure 6. - Instrumentation at compressor outlet, station 4, $1\frac{1}{2}$ inches behind trailing edge of compressor-outlet straightening vanes.



(a) Relation between compressor pressure ratio and corrected airflow.

Figure 7. - Effect of exhaust-nozzle-outlet area on compressor operating line for modified compressor. Altitude, 25,000 feet; flight Mach number, 0.53.



(b) Relation between compressor pressure ratio and corrected engine speed.

Figure 7. - Concluded. Effect of exhaust-nozzle-outlet area on compressor operating line for modified compressor. Altitude, 25,000 feet; flight Mach number, 0.53.

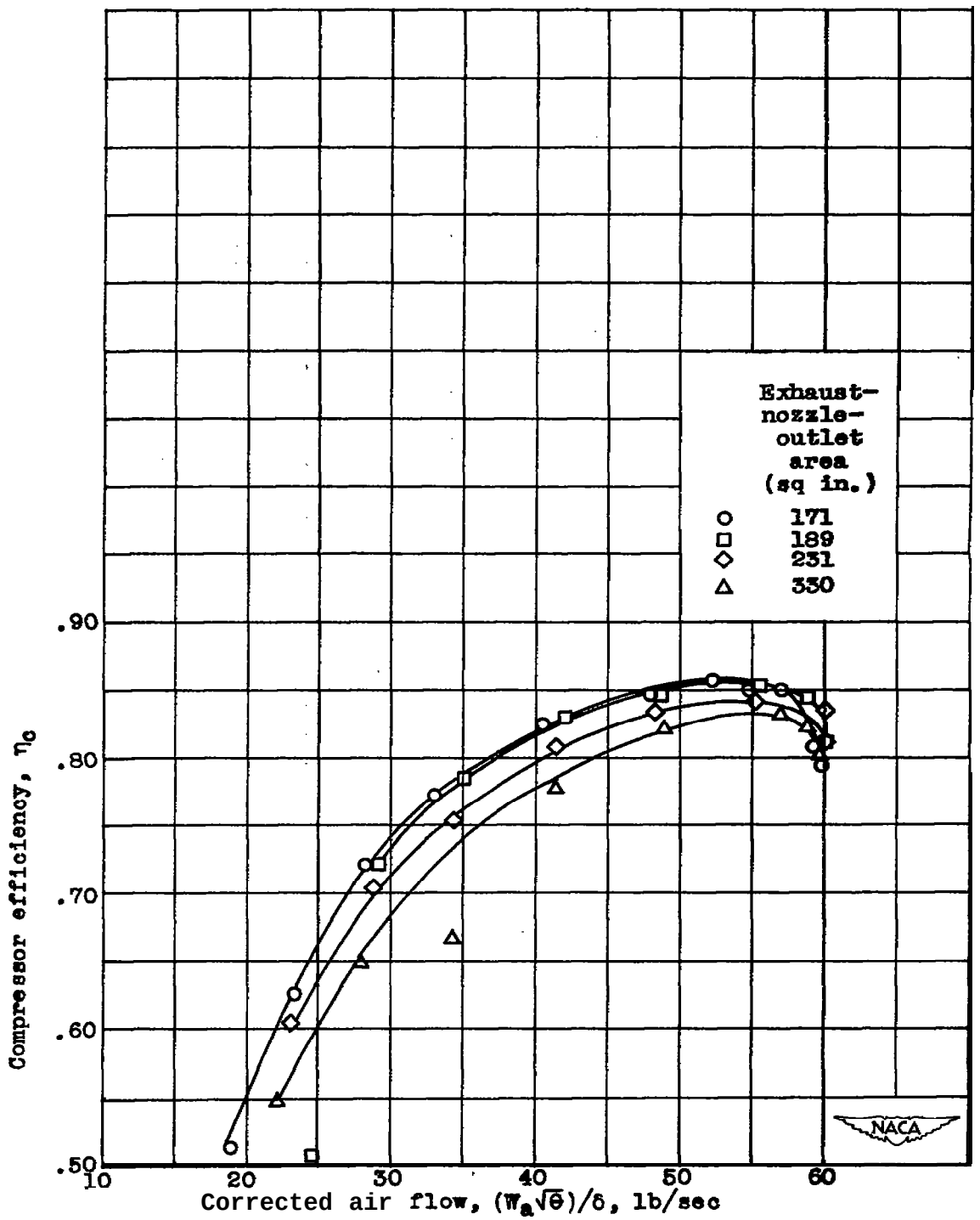
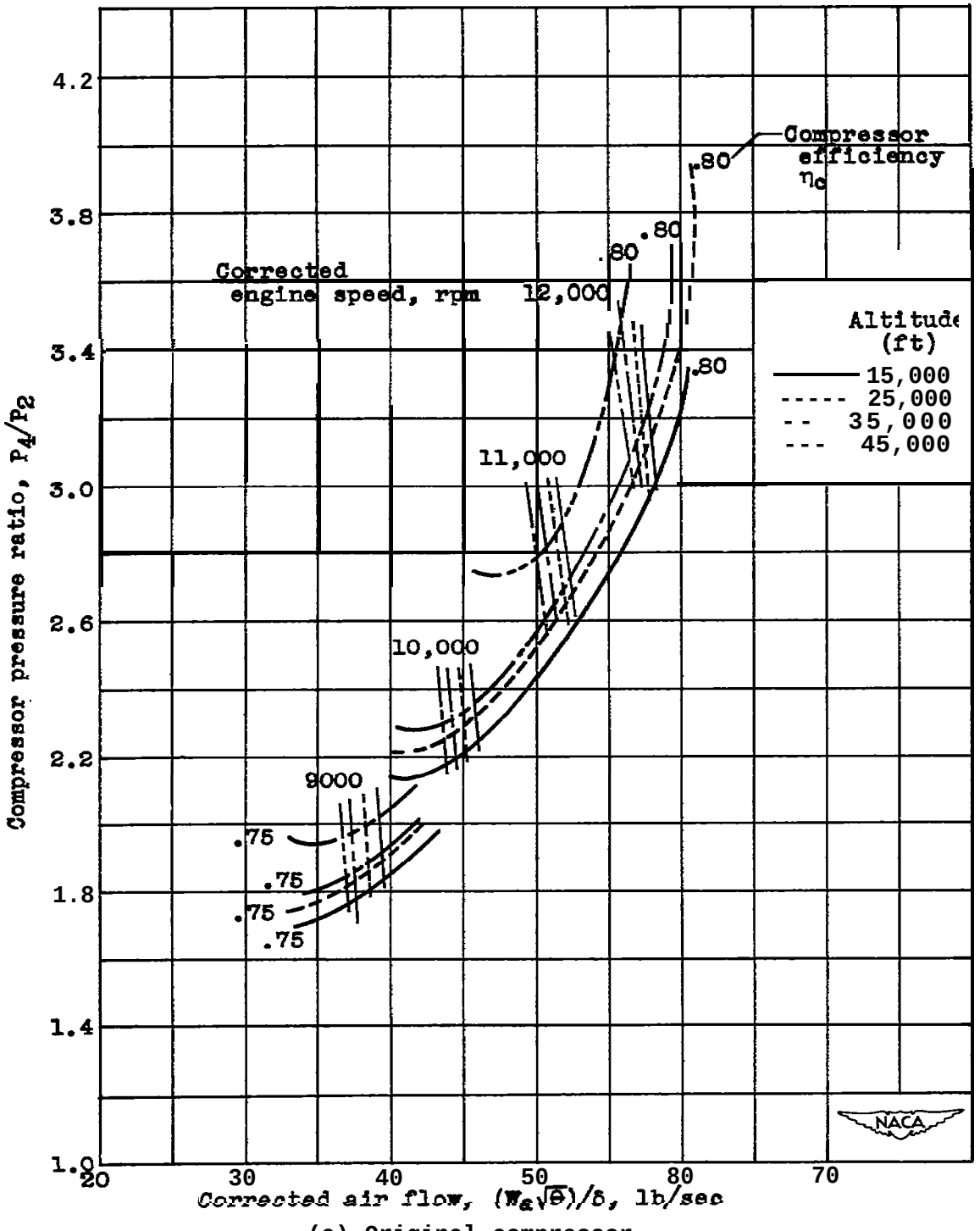
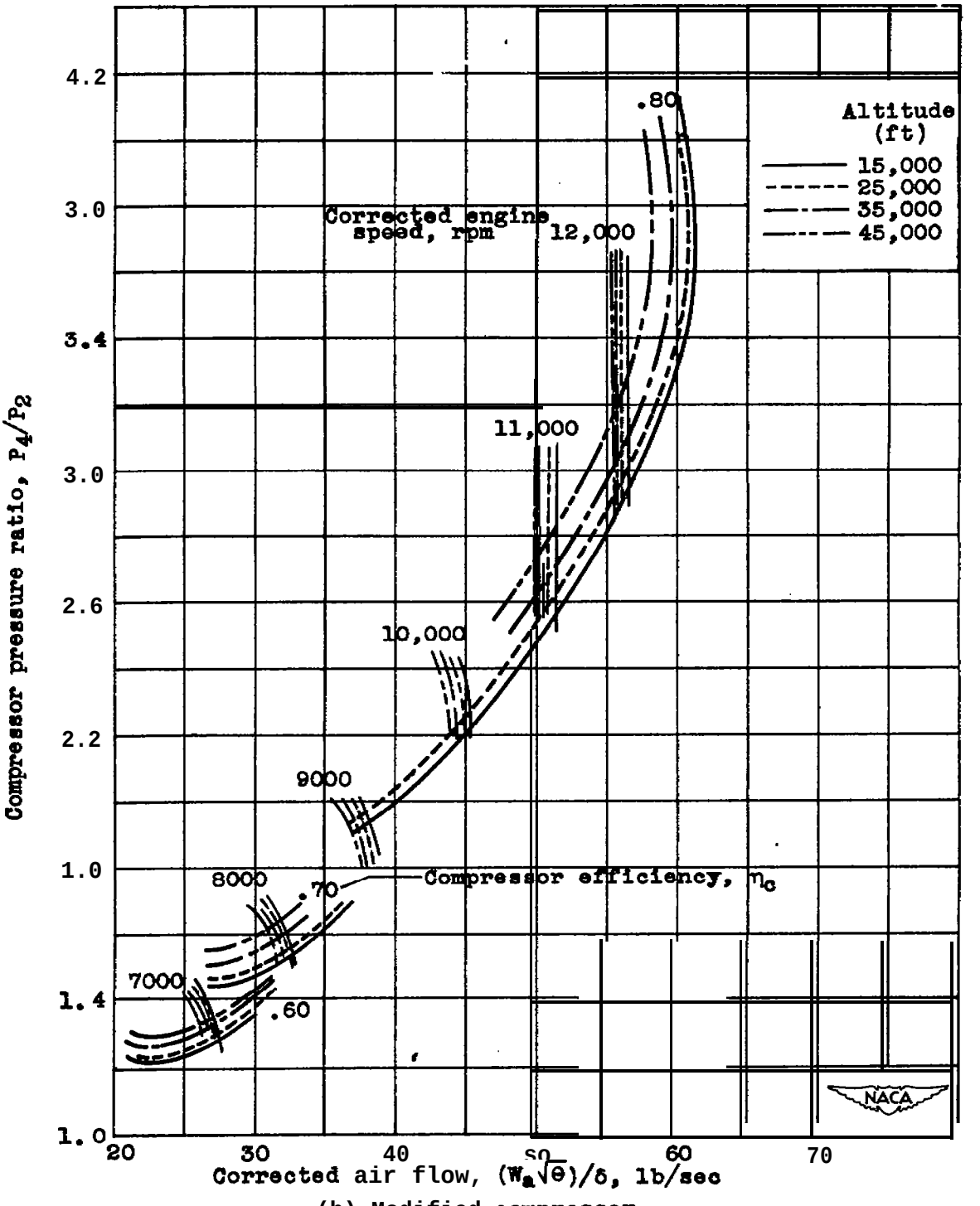


Figure 8. - Relation between compressor efficiency and corrected air flow of modified compressor for various exhaust-nozzle-outlet areas. Altitude, 25,000 feet; flight Mach number, 0.53.



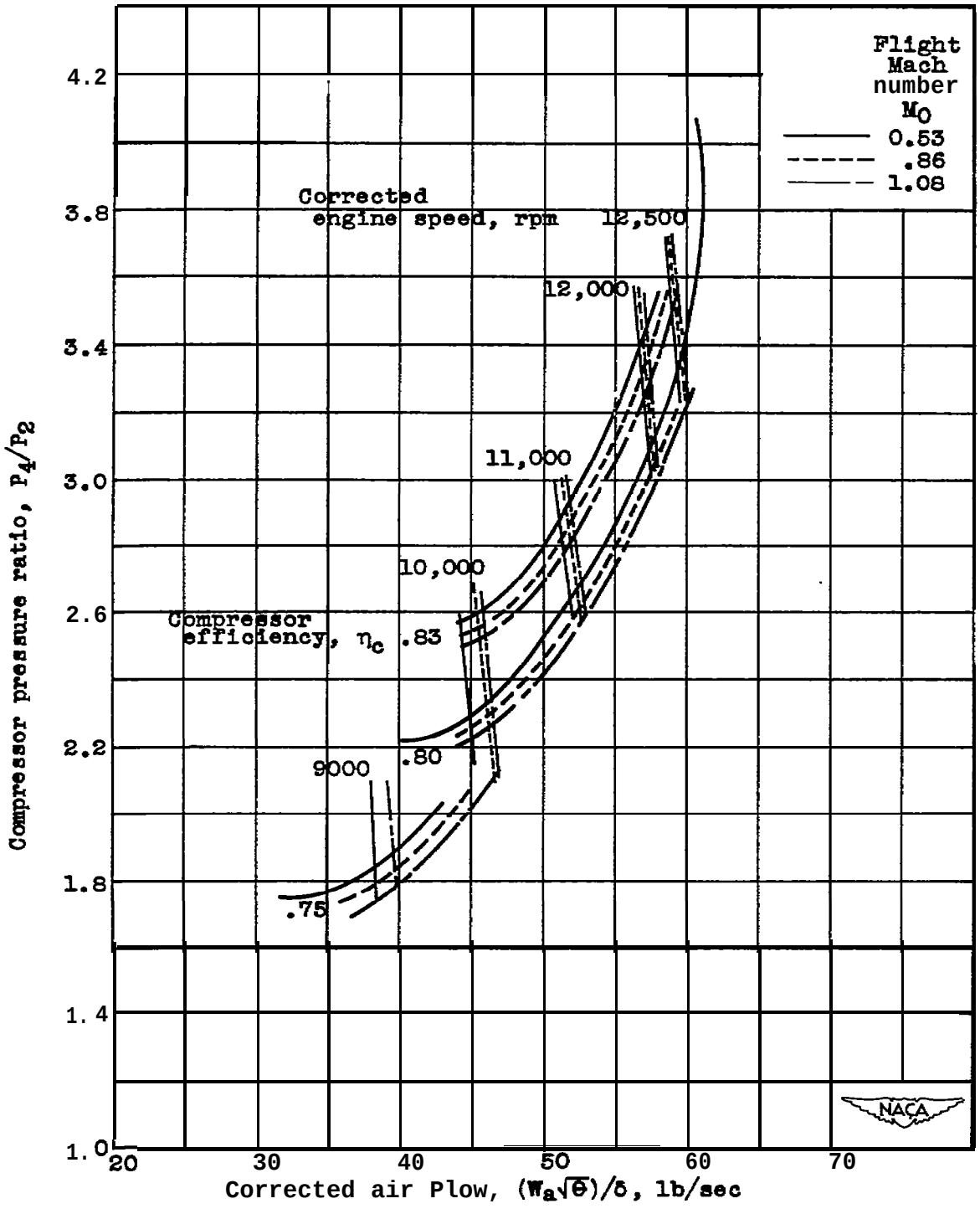
(a) Original compressor.

Figure 9. - Effect of altitude on compressor performance characteristics. Flight Mach number, 0.55.



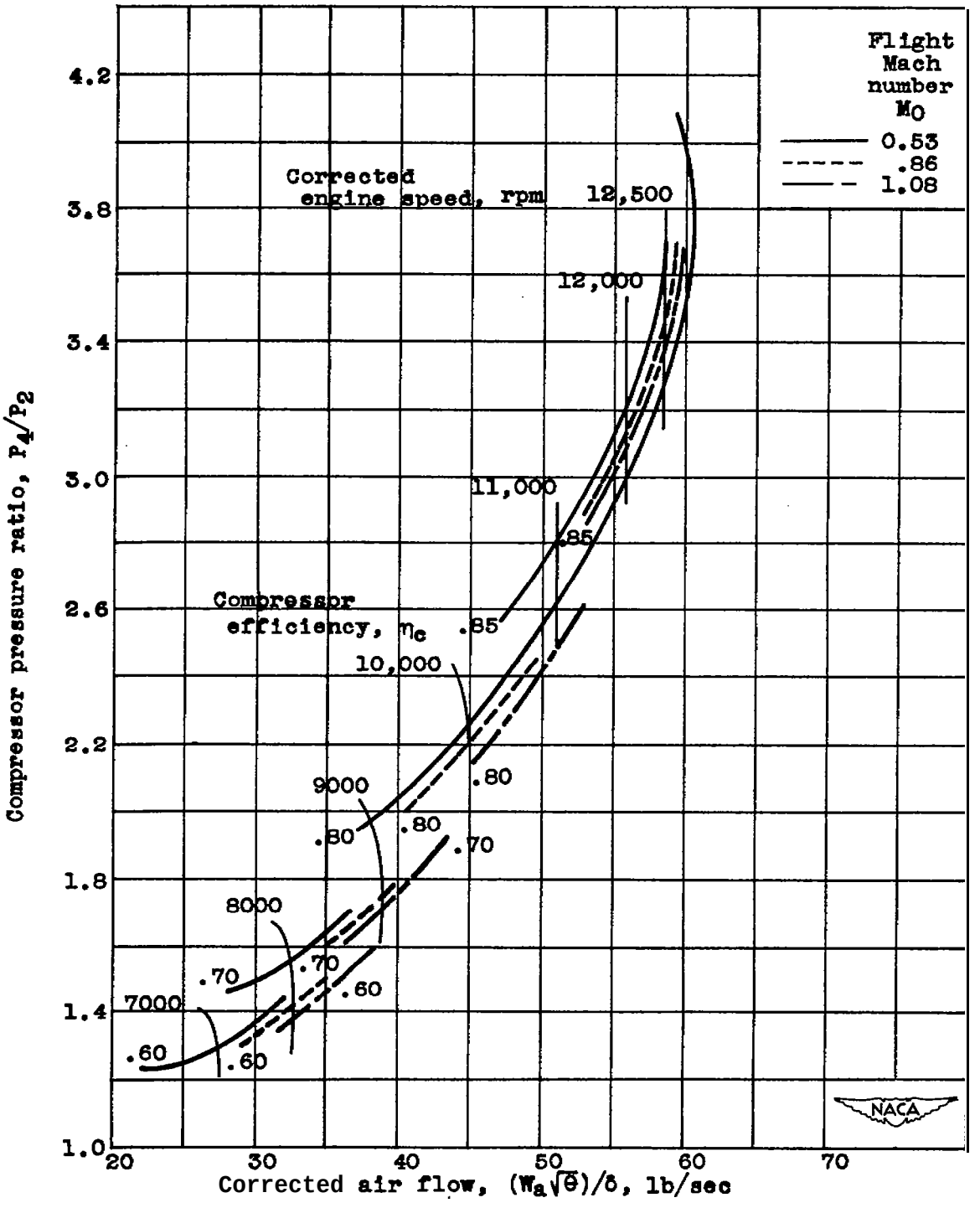
(b) Modified compressor.

Figure 9. - Uonoluded. Effect of altitude on compressor performance characteristics. Flight Mach number, 0.53.



(a) Original compressor.

Figure 10. - Effect of flight Mach number on compressor performance characteristics. Altitude, 25,000 feet.



(b) Modified compressor.

Figure 10. - Conoluded. Effect of flight Mach number on compressor performance characteristics. Altitude, 25,000 feet.

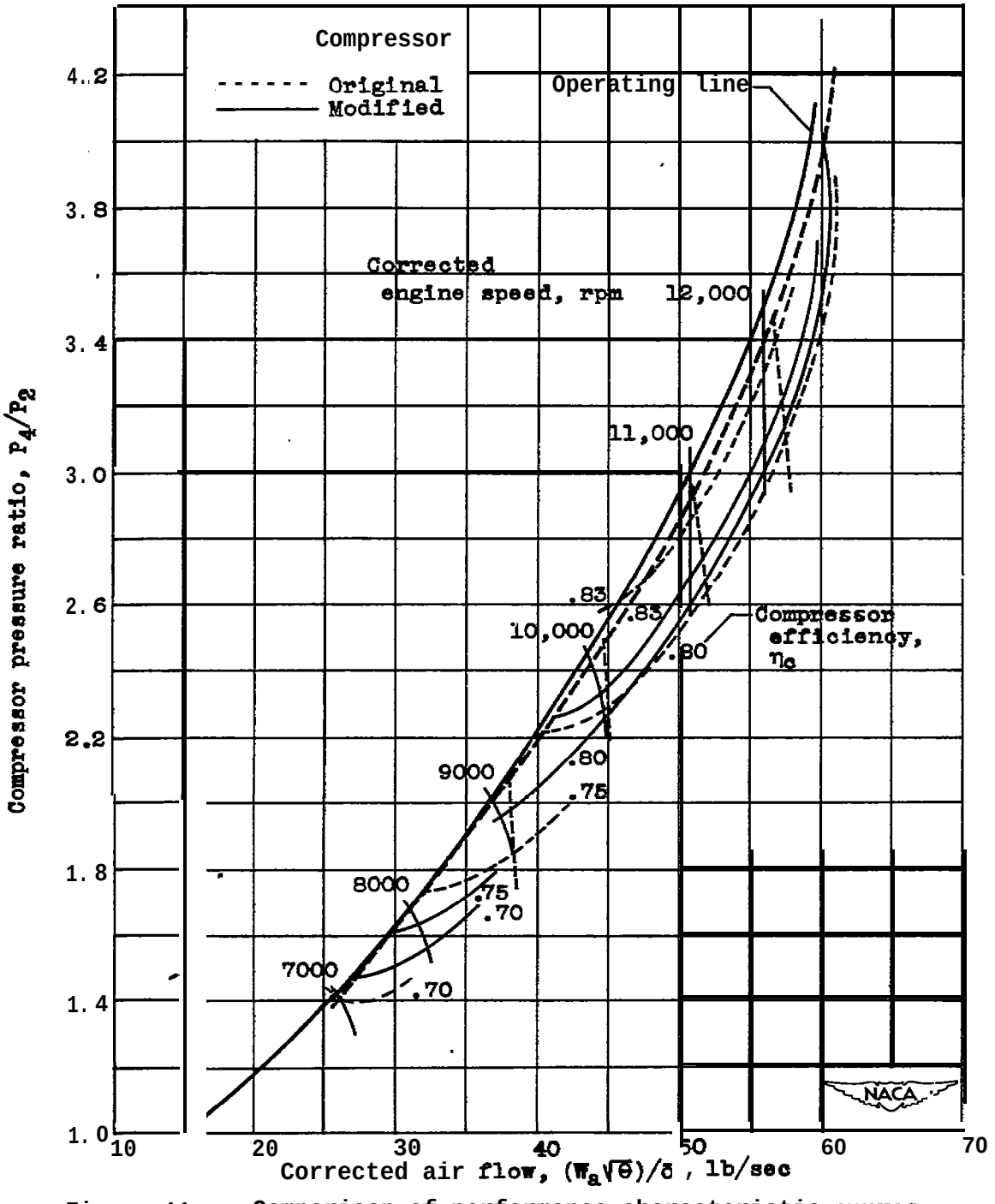
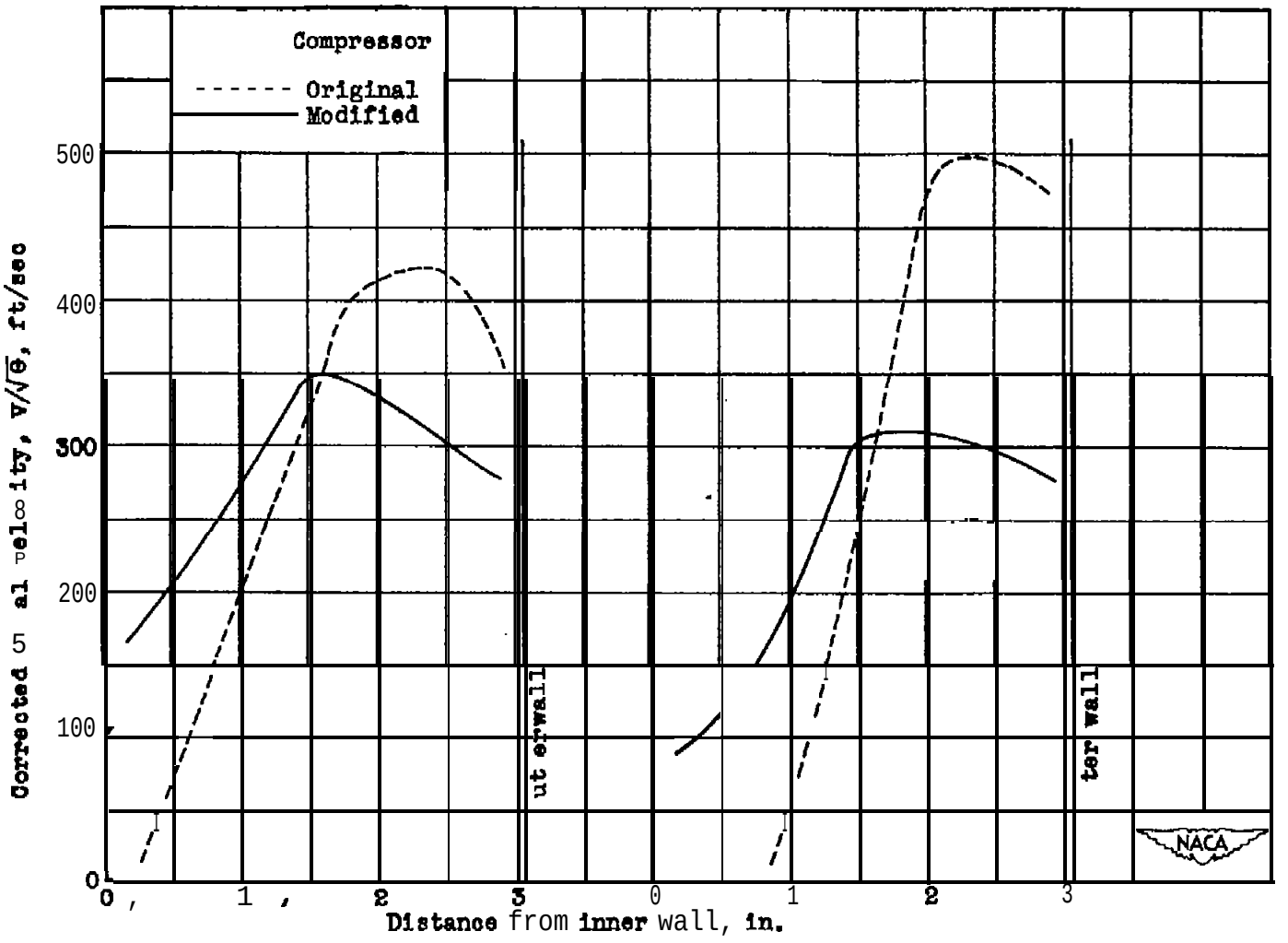
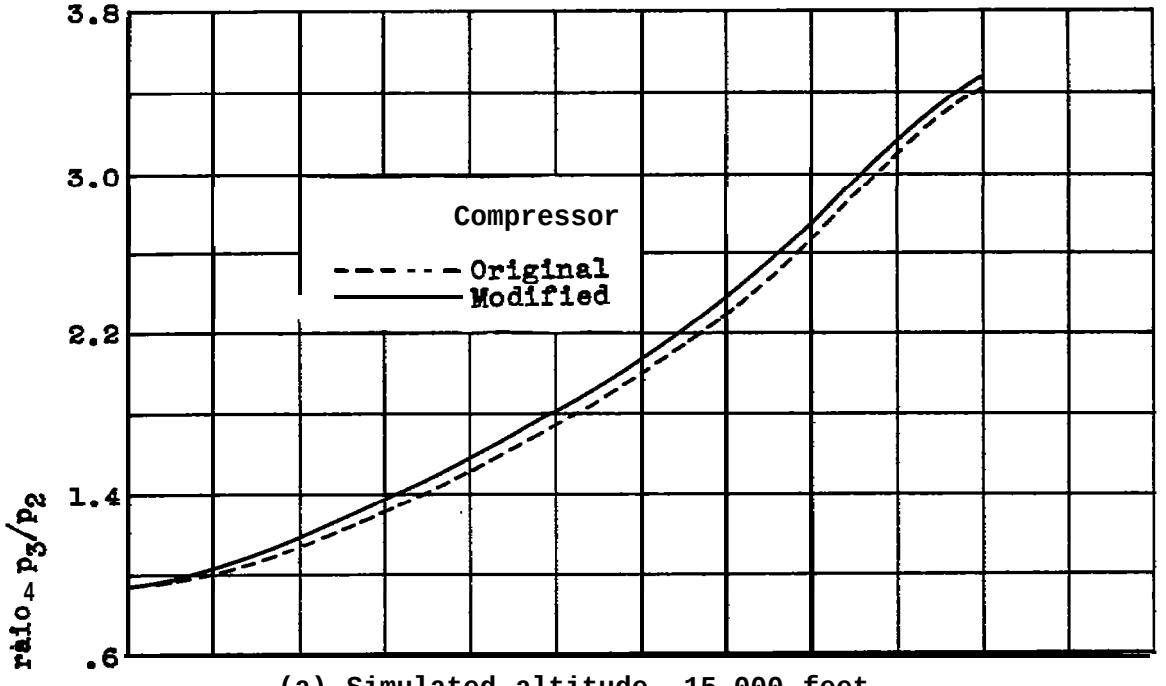


Figure 11. - Comparison of performance characteristic curves for original and modified compressors with operating lines for minimum exhaustnozzle-outlet area superimposed. Simulated altitude, 25,000 feet; flight Mach number, 0.53.

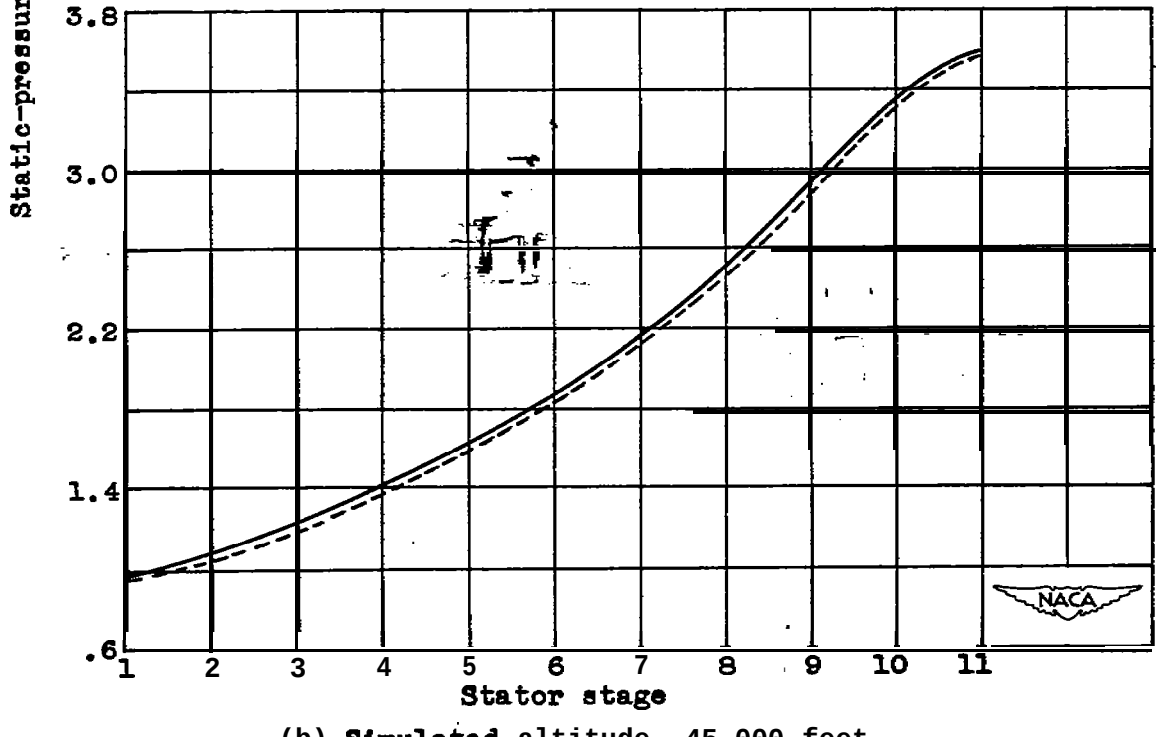


(a) Altitude, 15,000 feet. (b) Altitude, 45,000 fast.'

'Figure 12. - Radial distribution of corrected axial velocity at outlet of original and modified compressors. Corrected engine speed, 32,000 rpm; flight Mach number, 0.53.



(a) Simulated altitude, 15,000 feet.



(b) Simulated altitude, 45,000 feet.

Figure 13. - Stator-stage static-pressure ratio for original and modified compressors. Corrected engine speed, 12,000 rpm; flight Mach number, 0.53; minimum exhaust-nozzle-outlet area.

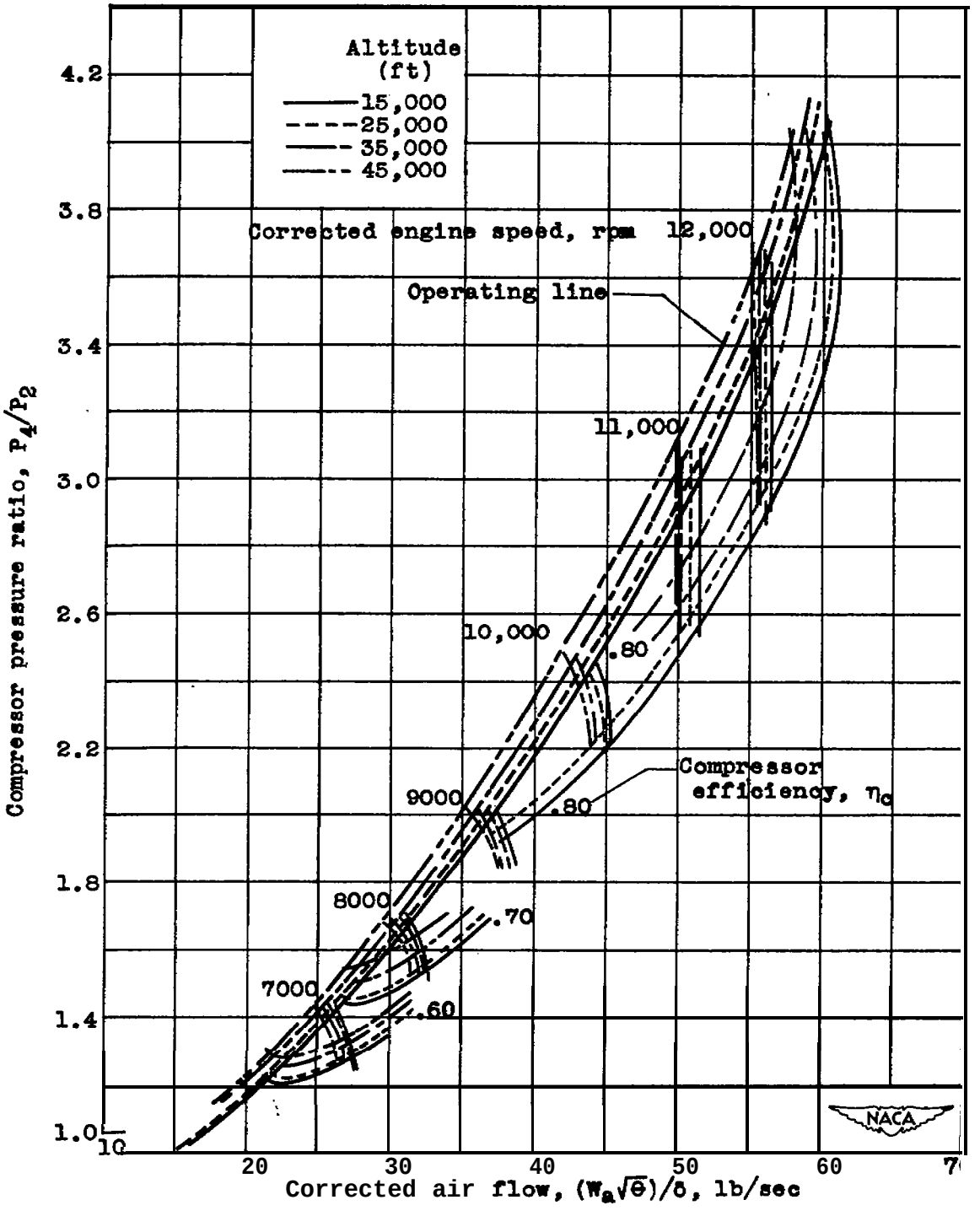


Figure 14. - Performance characteristic curve8 and operating lines for modified compressor at various altitudes. Flight Mach number, 0.53; exhaust-nozzle-outlet area, 171 square inches.

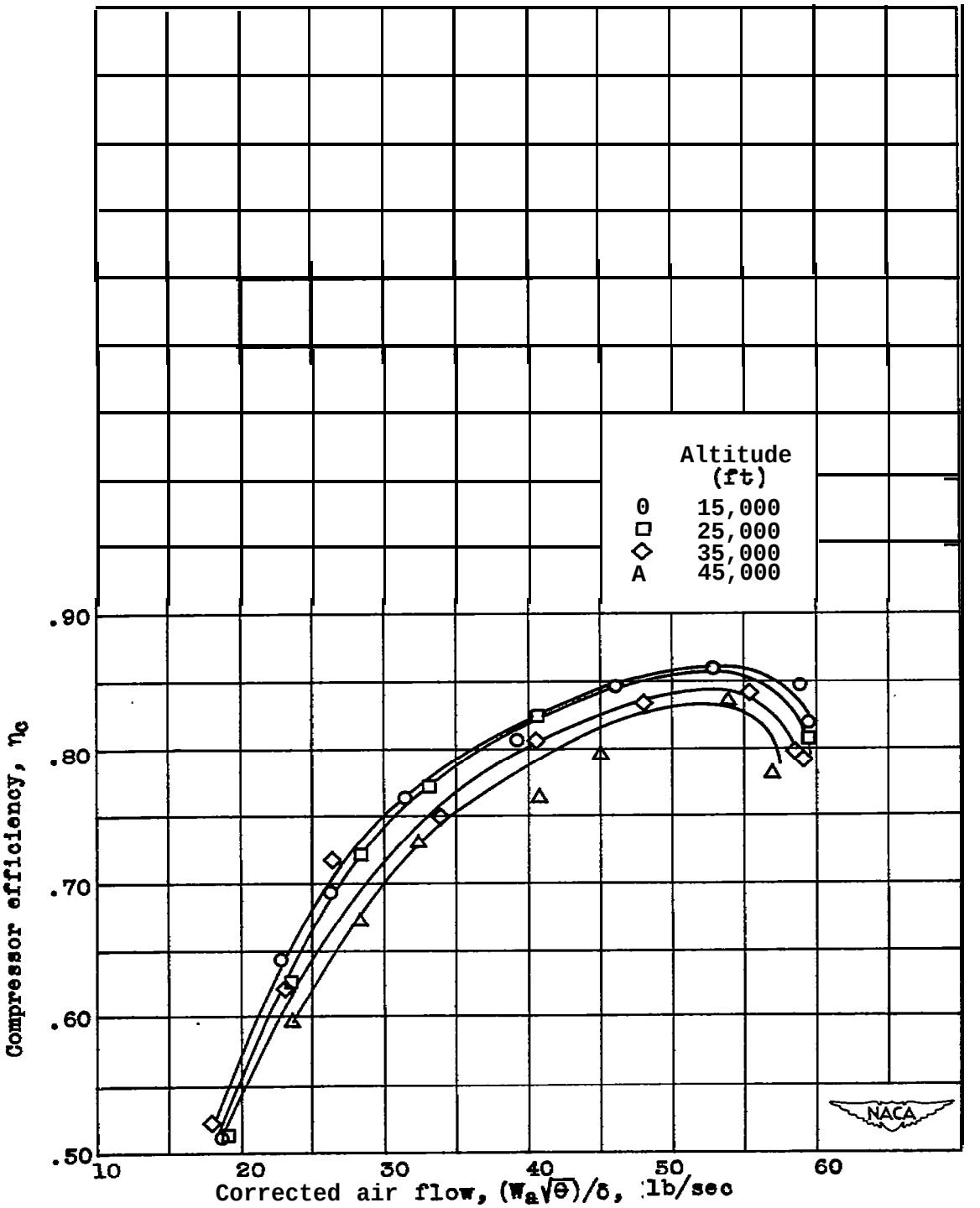


Figure 15. - Relation between compressor efficiency and corrected air flow for modified compressor at various altitudes. Flight Mach number, 0.53; exhaust-nozzle-outlet area, 171 square inches.

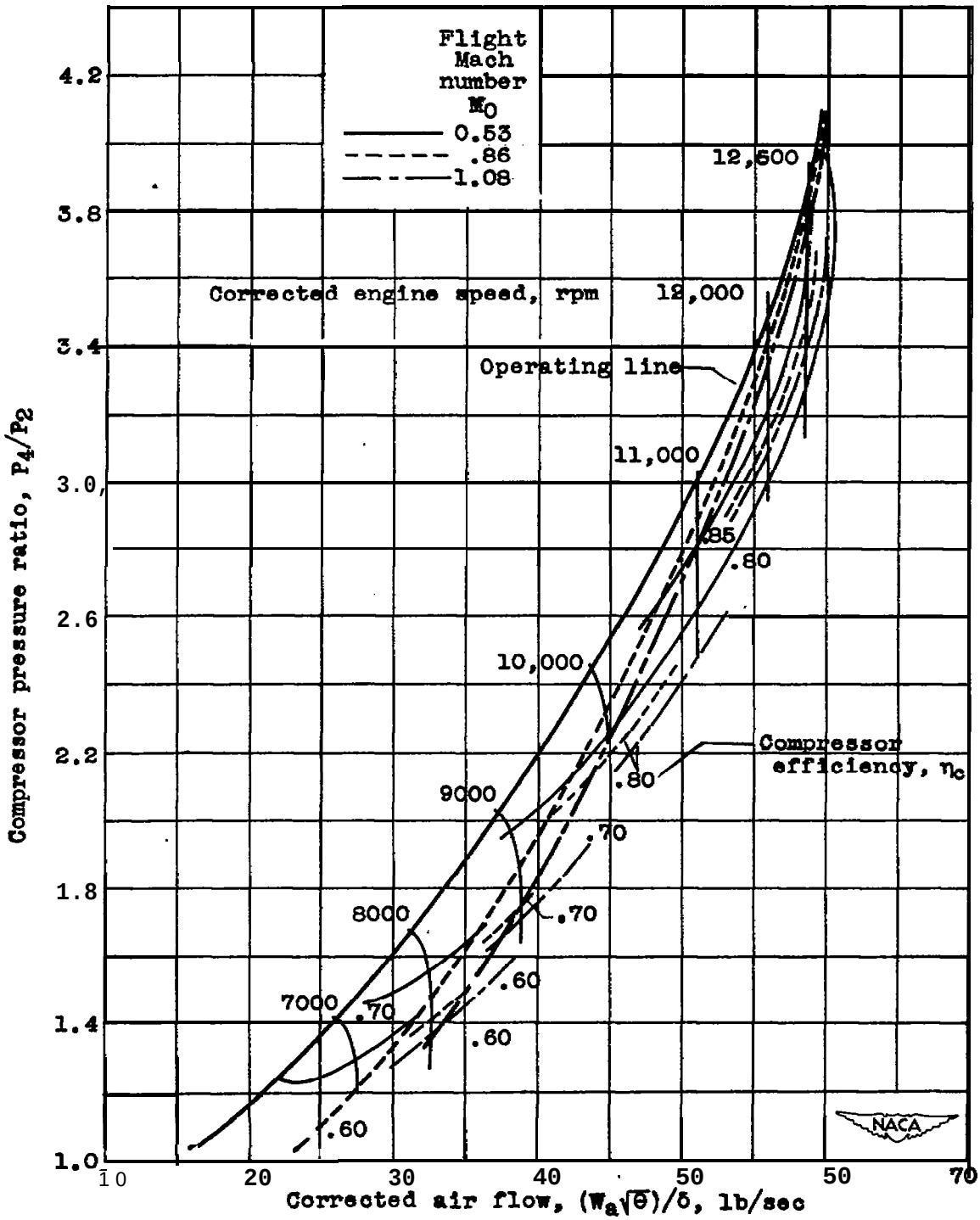


Figure 16. - Performance characteristic curves and operating lines for modified compressor at various flight Mach numbers. Simulated altitude, 25,000 feet; exhaust-nozzle-outlet area, 171 square inches.

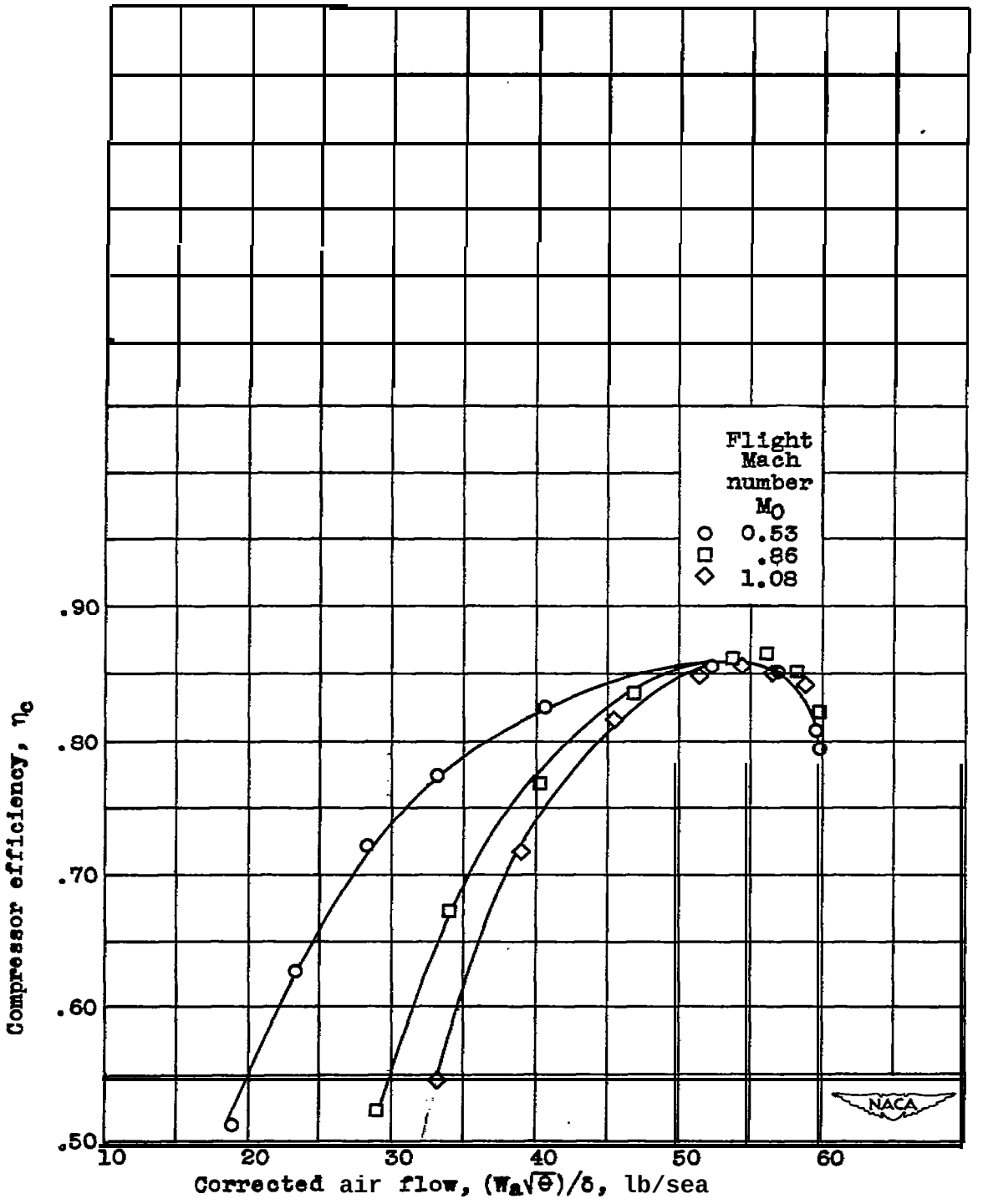


Figure 17. - Relation between compressor efficiency and corrected air flow for modified compressor at various flight Mach numbers. Simulated altitude, 25,000 feet; exhaust-nozzle-outlet area, 171 square inches.

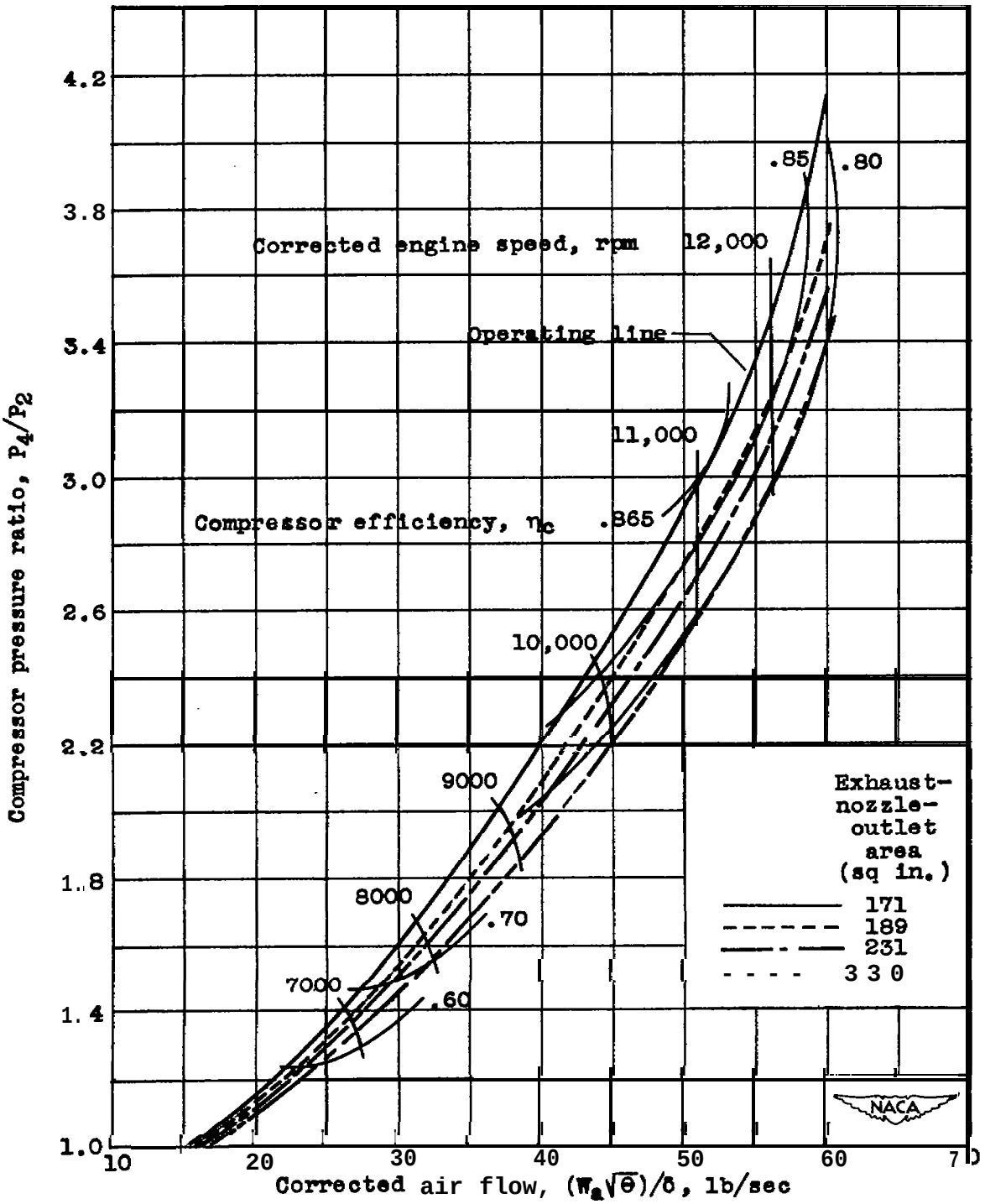


Figure 18. - Performance characteristic curve for modified compressor with operating lines for various exhaust-nozzle-outlet areas. Simulated altitude, 25,000 feet; flight Mach number, 0.53.