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RESEARCH MEMORANDUM

for the

Air Materiel Command, Army Air Forces

PRELIMINARY RESULTS OF AN ALTITUDE-WIND-TUNNEL INVESTIGATION OF A TG-100A GAS TURBINE-PROPELLER ENGINE

V - COMBUSTION-CHAMBER CHARACTERISTICS

By Robert M. Geisenheyner, and Joseph J. Berdysz

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PRELIMINARY RESULTS OF AN ALTITUDE-WIND-TUNNEL INVESTIGATION OF A TG-100A GAS TURBINE-PROPELLER ENGINE

V - COMBUSTION-CHAMBER CHARACTERISTICS

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SUMMARY

An investigation to determine the performance and operational characteristics of the TG-100A gas turbine-propeller engine was conducted in the Cleveland altitude wind tunnel. As part of this investigation, the combustion-chamber performance was determined at pressure altitudes from 5000 to 35,000 feet, compressor-inlet ram-pressure ratios of 1.00 and 1.09, and engine speeds from 8000 to 13,000 rpm. Combustion-chamber performance is presented as a function of corrected engine speed and corrected horsepower.

For the range of corrected engine speeds investigated, over-all total-pressure-loss ratio, cycle efficiency, and the fractional loss in cycle efficiency resulting from pressure losses in the combustion chambers were unaffected by a change in altitude or compressor-inlet ram-pressure ratio. The scatter of combustion-efficiency data tended to obscure any effect of altitude or ram-pressure ratio. For the range of corrected horsepowers investigated, the total-pressure-loss ratio and the fractional loss in cycle efficiency resulting from pressure losses in the combustion chambers decreased with an increase in corrected horsepower at a constant corrected engine speed. The combustion efficiency remained constant for the range of corrected horsepowers investigated at all corrected engine speeds.

INTRODUCTION

An investigation to determine the performance and operational characteristics of the TG-100A gas turbine-propeller engine was conducted in the NACA Cleveland altitude wind tunnel at the request of the Air Materiel Command, Army Air Forces. The performance characteristics of the component parts of the engine were determined in addition to the evaluation of the over-all performance characteristics. Various phases of the investigation are reported in references 1 to 4.

The combustion-chamber performance is presented herein to show the effect of engine operating conditions on combustion-chamber over-all total-pressure loss, combustion efficiency, cycle efficiency, and the fractional loss in cycle efficiency due to combustion-chamber pressure losses. Data were obtained for pressure altitudes from 5000 to 35,000 feet, compressor-inlet ram-pressure ratios of 1.00 and 1.09, and engine speeds from 8000 to 13,000 rpm.

DESCRIPTION OF COMBUSTION CHAMBER

The TG-100A gas turbine-propeller engine has nine cylindrical counterflow combustion chambers (fig. 1). The air leaving the last stage of the compressor is turned 180° before entering the combustion chambers and passing into annular spaces between the casings and the liners. The casing (fig. 2(a)) is 6 inches in diameter and $14\frac{1}{2}$ inches long and contains a removable liner (fig. 2(b)) that divides the combustion zone from the passage for the entering air flow. A series of openings in the wall of the liner allows air to pass into the combustion zone, where it is mixed with fuel sprayed from an atomizing nozzle located in the center of the combustion-chamber dome. The fuel is ignited by spark plugs located in the dome of two of the combustion chambers; fuel in the other combustion chambers is ignited through cross-fire tubes.

ENGINE INSTALLATION AND INSTRUMENTATION

The TG-100A engine was installed in a streamlined wing nacelle, which was mounted in the 20-foot-diameter test section of the Cleveland altitude wind tunnel (fig. 3). A sectional drawing of the engine showing the location of measuring stations is given in figure 4. A complete description of the engine, the installation and the instrumentation is presented in reference 1.

The instrumentation used in this analysis was located at the compressor inlet (station 2), the compressor outlet (station 3), the turbine inlet (station 5), and the tail-pipe-nozzle outlet (station 8). Because the instrumentation at the compressor elbow, (station 4), was inadequate for determining pressure losses across the combustion chambers, the instrumentation at the compressor outlet (station 3), was used.

The compressor-outlet instrumentation consisted of three rakes, located 120° apart, with three total-pressure tubes and two thermocouples each. The location of all instrumentation at this station, including five wall static tubes and two probe static tubes, is shown in figure 5. Turbine-inlet instrumentation consisted of five total-pressure tubes and five wall static tubes located as shown in figure 6. The tail-pipe rake at station 8 (fig. 7) contained 16 total-pressure tubes, three static-pressure tubes, and six thermocouples and was located with the plane of measurement 1 inch upstream of the tail-pipe-nozzle outlet.

PROCEDURE

The investigation was conducted at pressure altitudes ranging from 5000 to 35,000 feet and at compressor-inlet ram-pressure ratios of 1.00 and 1.09. At each altitude and compressor-inlet ram-pressure ratio, the engine speed was varied from 8000 to 13,000 rpm. The engine power measured at the torquemeter ranged from 70 to 1050 horsepower. Ambient temperatures were maintained at approximately NACA standard altitude conditions. Average values of pressure and temperature used in this analysis were taken from references 1 and 3.

SYMBOLS

The following symbols are used in this report:

A	cross-sectional area, square feet
c_p	specific heat of gas at constant pressure, Btu per pound °R
$c_{p,j}$	average specific heat at constant pressure from station 9 to station 0, Btu per pound °R
f	fuel-air ratio

g	acceleration due to gravity, 32.2 feet per second per second
ghp	horsepower loss in high-speed reduction gear
H	enthalpy, Btu per pound
H_a	enthalpy of air, Btu per pound
H_f	enthalpy of fuel, Btu per pound
h_f	heating value of fuel, 18,700 Btu per pound
h_p	shp + ghp
J	mechanical equivalent of heat, 778 foot-pounds per Btu
N	engine speed, rpm
P	total pressure, pounds per square foot absolute
$(\Delta P_T)/P_3$	over-all total-pressure-loss ratio across combustion chambers due to friction and momentum changes $\left(\frac{P_3-P_5}{P_3}\right)$, pounds per square foot
p	static pressure, pounds per square foot absolute
R	gas constant, 53.4 foot-pounds per pound $^{\circ}R$
shp	shaft horsepower measured at torquemeter
T	total temperature, $^{\circ}R$
T_i	indicated temperature, $^{\circ}R$
t	static temperature, $^{\circ}R$
W_a	air flow, pounds per second
W_f	fuel flow, pounds per hour
W_g	gas flow, pounds per second
γ	ratio of specific heats for gases

δ ratio of compressor-inlet total pressure to static pressure of NACA standard atmosphere at sea level

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

η cycle efficiency

$\Delta\eta$ cycle-efficiency loss

η_b combustion efficiency

Subscripts:

0 tunnel-test-section free air stream

2 compressor inlet

3 compressor outlet

5 turbine inlet

8 tail-pipe-nozzle outlet

9 station in jet where static pressure first reaches free-stream static pressure

The data were generalized to NACA standard sea-level conditions by the following parameters:

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

$hp/(\delta\sqrt{\theta})$ corrected horsepower

METHODS OF CALCULATION

Gas flow and air flow. - The gas flow at the tail-pipe-nozzle outlet was obtained from

$$W_{g,8} = p_8 A_8 \sqrt{\frac{2\gamma_8 g}{(\gamma_8-1)R_8 t_8}} \sqrt{\left(\frac{P_8}{p_8}\right)^{\frac{\gamma_8-1}{\gamma_8}} -1}$$

The air flow was then determined from

$$W_a = W_g - (W_f/3600)$$

Turbine-inlet temperature. - Turbine-inlet temperature was calculated from the enthalpy drop through the turbine and the enthalpy at the tail-pipe-nozzle outlet:

$$H_5 = \left(\frac{shp + ghp}{W_g\ J}\right) + \left(H_3 - H_2\right) + H_8$$

$$T_5 = \frac{H_5}{c_{p,5}}$$

An integrated value of $c_{p,5}$ was used in this equation. The value of ghp used in calculating H_5 was estimated to vary from 50 horsepower at an engine speed of 13,000 rpm to 25 horsepower at 8000 rpm.

Jet-gas temperature. - The gas temperature in the jet was determined from temperature and pressure measurements at the tail-pipe-nozzle outlet. For a thermocouple recovery factor of 0.85, the total temperature at the nozzle outlet was calculated from

$$T_8 = \frac{T_{i,8}\left(\frac{P_8}{p_8}\right)^{\frac{\gamma_8-1}{\gamma_8}}}{1 + 0.85\left[\left(\frac{P_8}{p_8}\right)^{\frac{\gamma_8-1}{\gamma_8}} - 1\right]}$$

If the total pressure, the total temperature, and the ratio of specific heats at the nozzle outlet (station 8), and in the jet, (station 9), are assumed to be equal,

$$t_9 = T_8\left(\frac{p_0}{P_8}\right)^{\frac{\gamma_8-1}{\gamma_8}}$$

METHODS OF ANALYSIS

Combustion efficiency. - Combustion efficiency is defined as the ratio of the actual increase in enthalpy of the gas to the theoretical increase that would result from complete combustion of the fuel charge:

$$\eta_b = \frac{(H_{a,5}-H_{a,3}) + (H_{f,5}-H_{f,3})f}{fh_f}$$

Cycle efficiency. - Cycle efficiency was determined according to the standard definition

$$\eta = \frac{\text{heat supplied by source - heat rejected to sink}}{\text{heat supplied by source}}$$

$$\eta = \frac{(H_5-H_3) - c_{p,j}(t_9-t_0)}{H_5-H_3}$$

Loss in cycle efficiency. - The expression for loss in cycle efficiency resulting from pressure losses in the combustion chamber was calculated by the method given in reference 5:

$$\Delta\eta = \frac{c_{p,j}\, t_9 \left[1 - \left(\frac{P_5}{P_3}\right)^{\frac{\gamma-1}{\gamma}}\right]}{H_5-H_3}$$

where γ is the average value between stations 5 and 9.

RESULTS AND DISCUSSION

The presentation of combustion-chamber performance includes the evaluation of over-all loss in total pressure across the combustion chambers, combustion efficiency, cycle efficiency, and loss in cycle efficiency resulting from pressure losses in the combustion chambers.

The over-all total-pressure loss across the combustion chambers may be considered as the sum of two components: the loss due to friction and that due to the addition of heat by combustion, or momentum-pressure loss. Reference 5 gives a method of determining these components by assuming that all the friction loss occurs before the gas is heated. This assumption is valid for a through-flow type of combustion chamber. In a counterflow combustion chamber, however, the assumption is invalid because preheating occurs during the flow of the entering air along the outside of the liner or basket. Consequently, the method of reference 5 is not applicable to the counterflow type of combustion chamber and only over-all pressure losses are considered in this report.

Pressure-Loss Ratio

Over-all total-pressure-loss ratio $(\Delta P_T)/P_3$ for the range of corrected engine speeds investigated is presented in figure 8(a) for altitudes from 5000 to 35,000 feet at static conditions and a tail-pipe temperature of approximately 1500° R. A similar comparison of pressure-loss ratio is given in figure 8(b) for compressor-inlet ram-pressure ratios of 1.00 and 1.09 at an altitude of 25,000 feet. Changes in altitude or ram-pressure ratio did not appreciably affect the total-pressure-loss ratio across the combustion chamber. The total-pressure-loss ratio was 0.025 for the range of corrected engine speeds investigated. The variation of total-pressure-loss ratio with corrected horsepower is shown in figure 9 for various corrected engine speeds at an altitude of 5000 feet and static conditions. At a constant corrected engine speed, the total-pressure-loss ratio decreased with an increase in corrected horsepower. For a constant corrected horsepower, the total-pressure-loss ratio increased as the corrected engine speed increased.

Combustion Efficiency

The relation of combustion efficiency to corrected engine speed is presented in figure 10(a) for altitudes from 5000 to 35,000 feet at a ram-pressure ratio of 1.00 and in figure 10(b) for ram-pressure ratios of 1.00 and 1.09 at an altitude of 25,000 feet. The scatter of these data tended to obscure any effect of altitude or ram-pressure ratio on combustion efficiency. The variation of combustion efficiency with corrected horsepower is shown in figure 11 for various corrected engine speeds at an altitude of 5000 feet and static conditions. For the range of corrected horsepowers investigated, varying the corrected engine speed had no appreciable effect on combustion efficiency. Values of combustion efficiency over 1.00 are considered the result of inaccuracies in measurement of turbine-outlet temperature and shaft horsepower.

Cycle Efficiency

Cycle efficiency as a function of corrected engine speed at a tail-pipe temperature of approximately 1500° R is shown in figure 12(a) for altitudes from 5000 to 35,000 feet at static conditions and in figure 12(b) for ram-pressure ratios of 1.00 and 1.09 at an altitude of 25,000 feet. As the corrected engine speed was increased from 8170 to 14,180 rpm, the cycle efficiency was raised from approximately 0.04 to 0.17. Changes in altitude or ram-pressure ratio had no apparent effect on cycle efficiency. The variation of cycle efficiency with corrected horsepower at various corrected engine speeds and static conditions is shown in figure 13. Cycle efficiency increased with increasing corrected horsepower from 0.03 to 0.16 over the range of the investigation. At a constant corrected horsepower, a change in engine speed had no effect on cycle efficiency.

Loss in Cycle Efficiency

Cycle-efficiency losses that resulted from combustion-chamber pressure losses are presented as fractional loss in cycle efficiency $(\Delta\eta)/\eta$ for the range of corrected engine speeds investigated at altitudes from 5000 to 35,000 feet and static conditions at a tail-pipe temperature of approximately 1500° R (fig. 14(a)). Fractional loss in cycle efficiency decreased with an increase in corrected engine speed over the entire range of the investigation. The change in altitude had no appreciable effect on $(\Delta\eta)/\eta$ for the range of corrected engine speeds. Increasing the ram-pressure ratio from 1.00 to 1.09 reduced the fractional loss in cycle efficiency approximately 0.01 for the range of corrected engine speeds investigated (fig. 14(b)). The fractional loss in cycle efficiency for a range of corrected horsepowers at an altitude of 5000 feet and static conditions is shown in figure 15. For a constant corrected engine speed, $(\Delta\eta)/\eta$ decreased with an increase in corrected horsepower. At a constant corrected horsepower, $(\Delta\eta)/\eta$ increased with an increase in engine speed.

SUMMARY OF RESULTS

An investigation of counterflow combustion chambers operating in a TG-100A gas turbine-propeller engine over a range of pressure altitudes from 5000 to 35,000 feet and ram-pressure ratios of 1.00 and 1.09 gave the following results:

1. Total-pressure-loss ratio was unaffected by changes in altitude at a constant tail-pipe temperature and remained at a value of approximately 0.025 for the range of operating conditions investigated. A change in ram-pressure ratio from 1.00 to 1.09 had no appreciable effect on total-pressure-loss ratio. At a constant corrected engine speed, the total-pressure-loss ratio was reduced as the corrected horsepower increased.

2. The scatter of data tended to obscure any effect of altitude or ram-pressure ratio on combustion efficiency. Varying the corrected horsepower at a constant corrected engine speed had no effect on combustion efficiency.

3. At a constant tail-pipe temperature, the cycle efficiency increased with increasing corrected engine speed, but was unaffected by a change in altitude or ram-pressure ratio. At a constant corrected engine speed, the cycle efficiency increased as the corrected horsepower increased. At a constant corrected horsepower, however, changes in corrected engine speed had no effect on cycle efficiency.

4. Fractional loss in cycle efficiency, the result of pressure losses in the combustion chambers, decreased with an increase in corrected engine speed and was not appreciably affected by a change in altitude or ram-pressure ratio. At a constant corrected engine speed, the fractional loss in cycle efficiency decreased with increasing corrected horsepower.

Flight Propulsion Research Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio, November 25, 1947.

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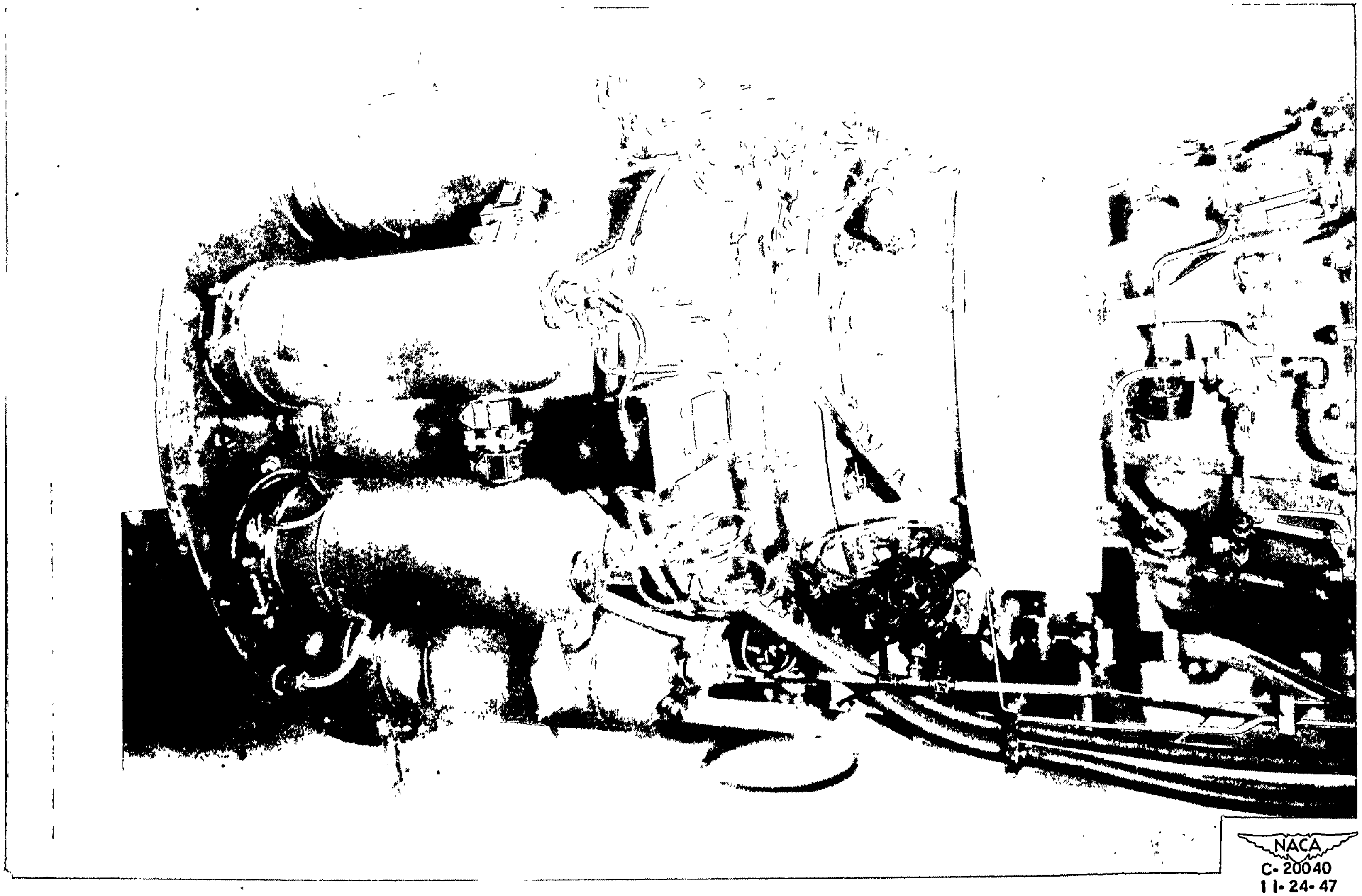
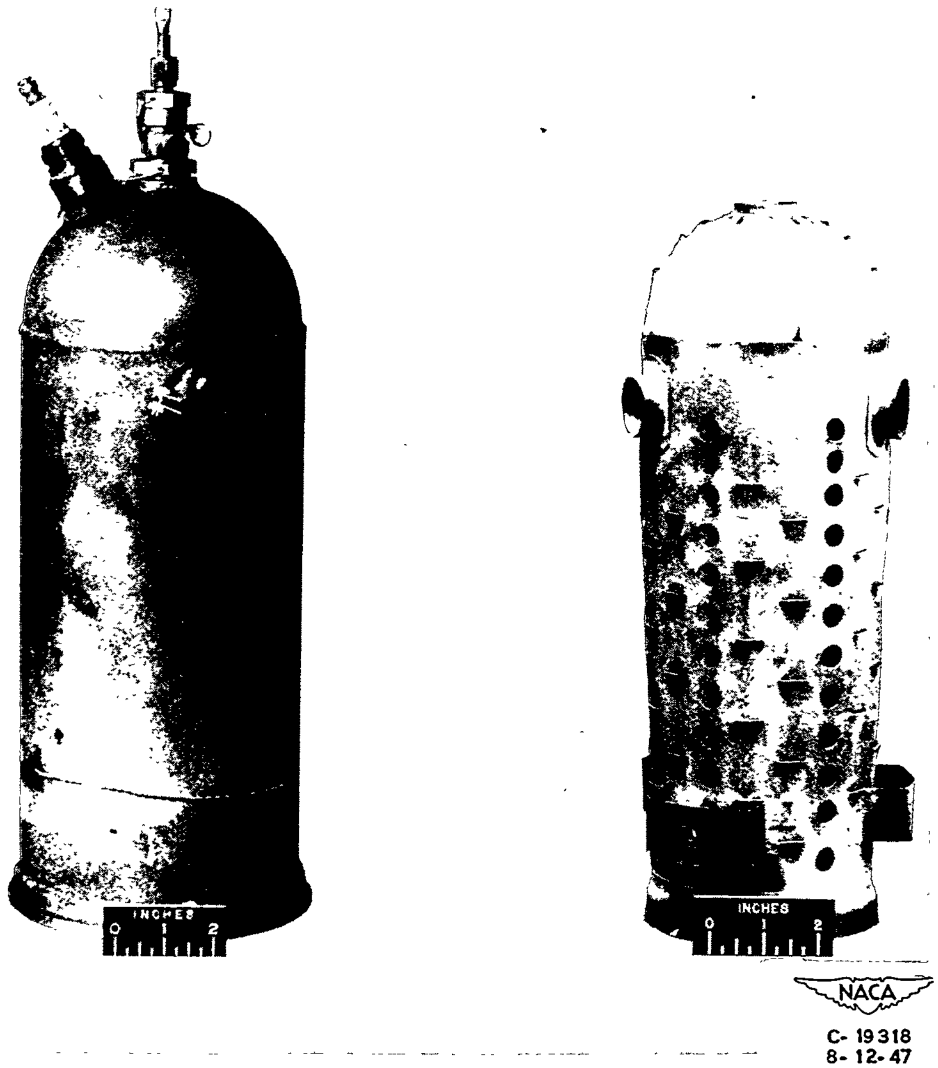


Figure 1. - Side view of TG-100A gas turbine-propeller engine showing installation of counterflow combustion chambers.



(a) Casing with spark plug installed.

(b) Liner.

Figure 2. - Counterflow combustion chamber of TG-100A gas turbine-propeller engine.

Figure 3. - Front view of TG-100A gas turbine-propeller engine in altitude wind tunnel.

Station

2 Compressor inlet
3 Compressor outlet
4 Compressor elbow
5 Turbine inlet
6 Turbine outlet.
7 Exhaust-cone outlet
8 Tail-pipe-nozzle outlet

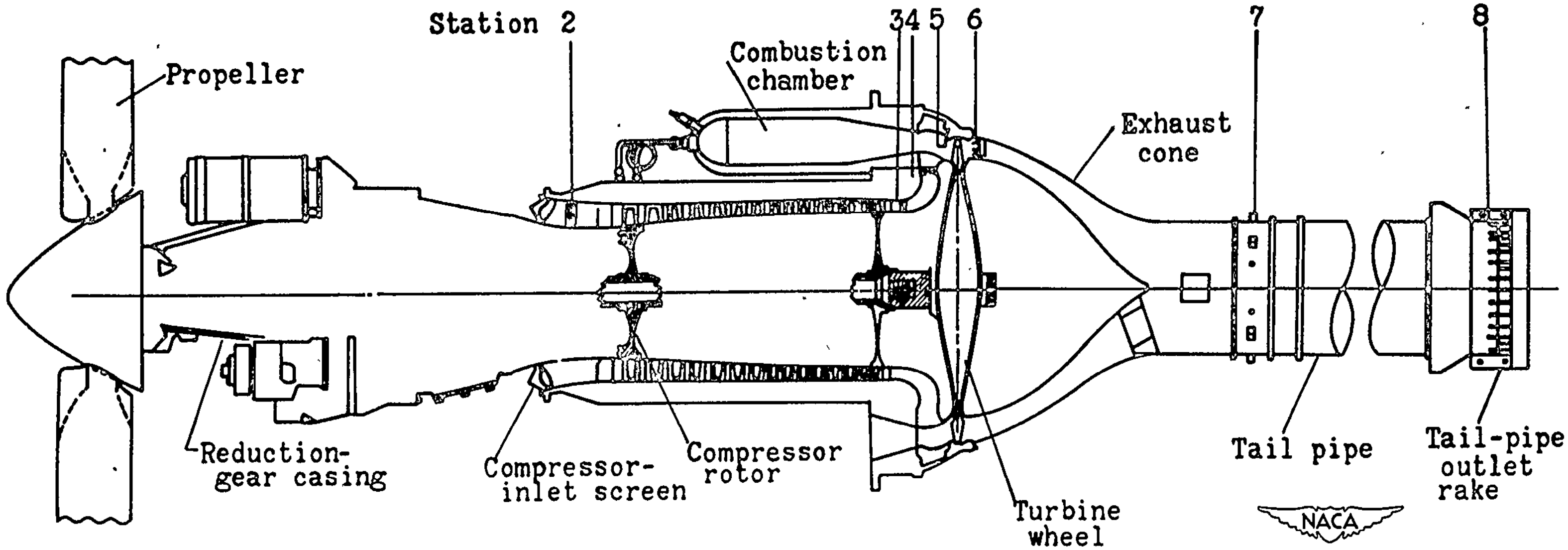


Figure 4. - Side view of TG-100A engine showing location of measuring stations.

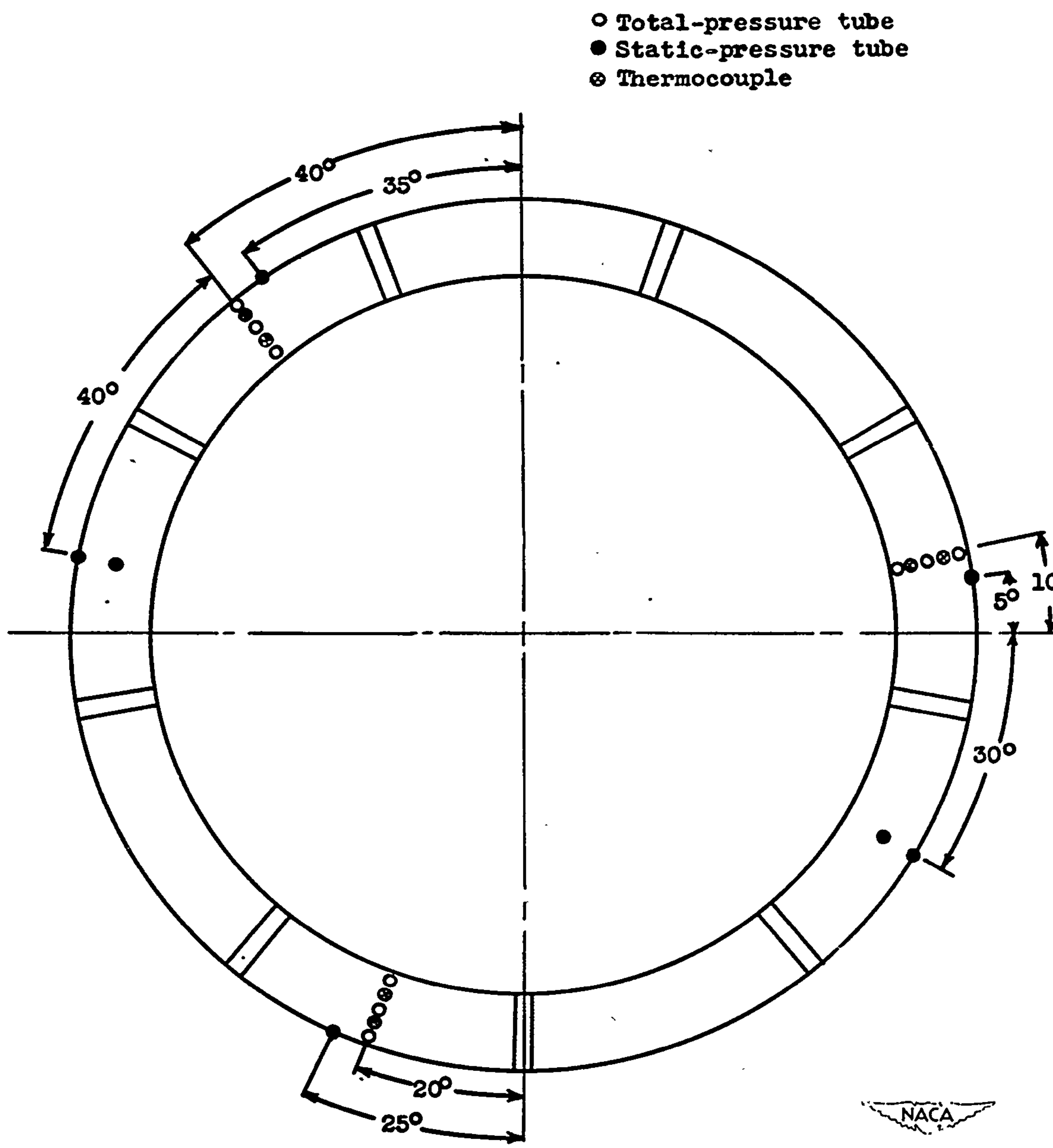


Figure 5. – Location of instrumentation at compressor outlet, looking aft, station 3.

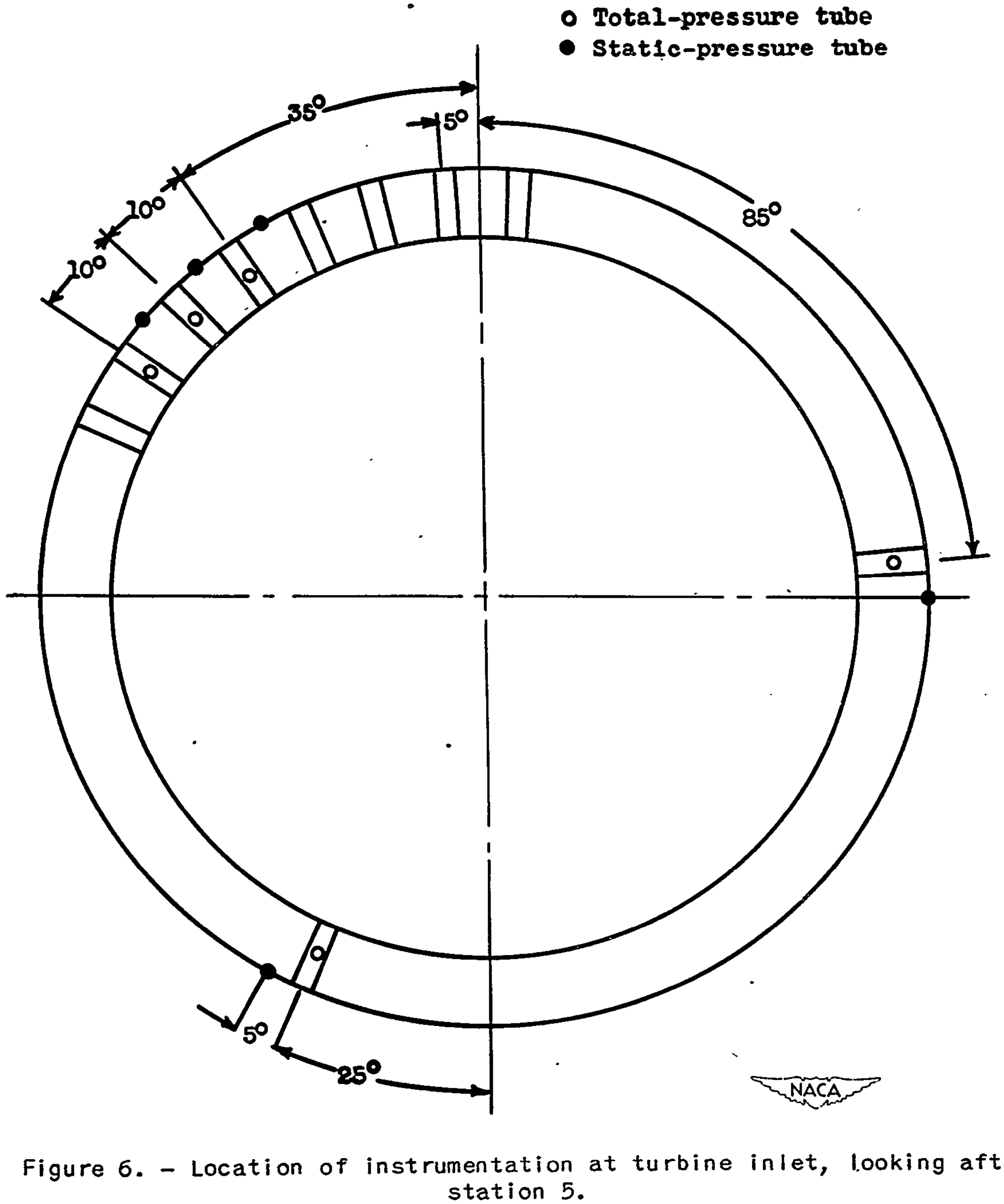


Figure 6. - Location of instrumentation at turbine inlet, looking aft, station 5.

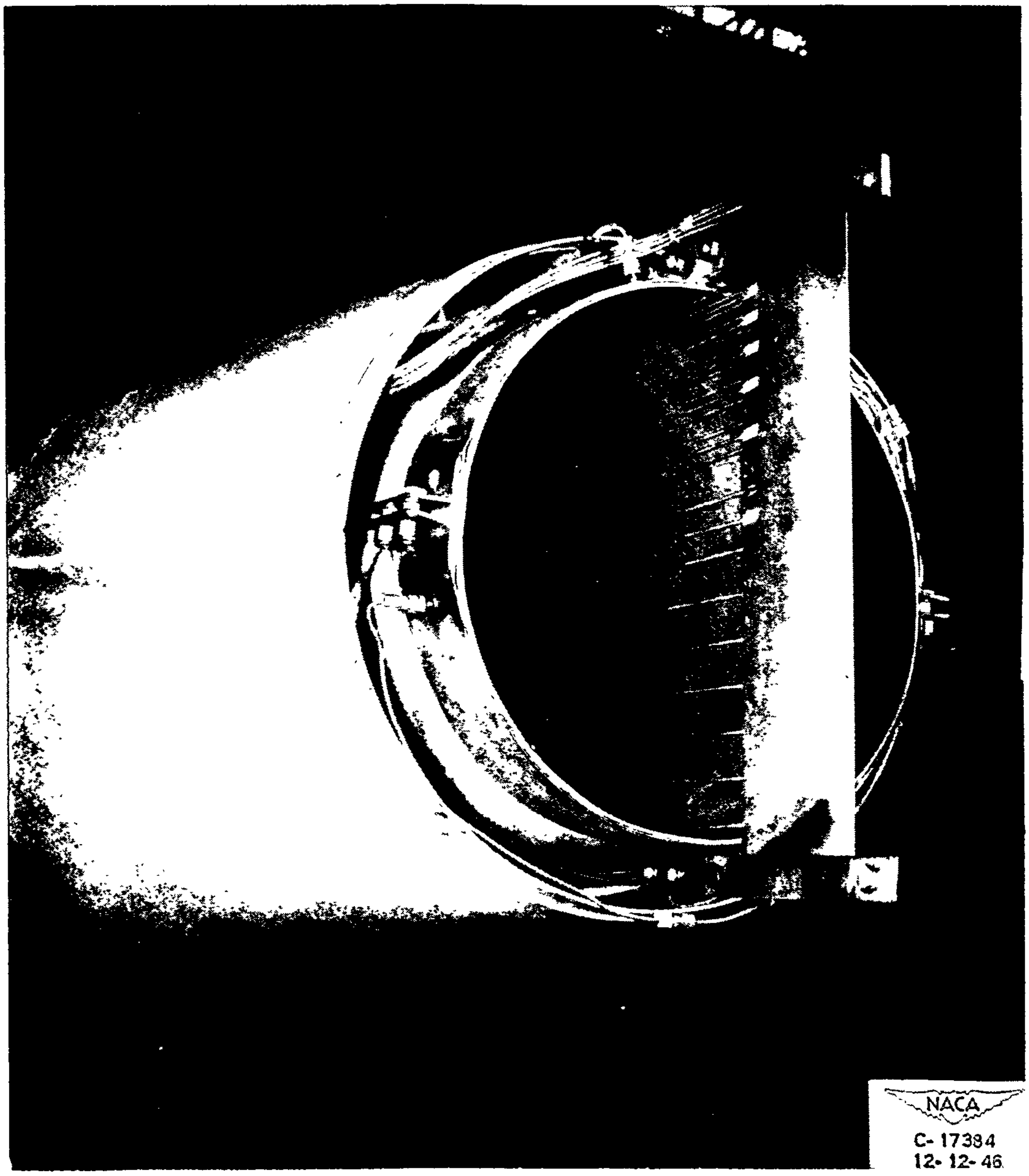
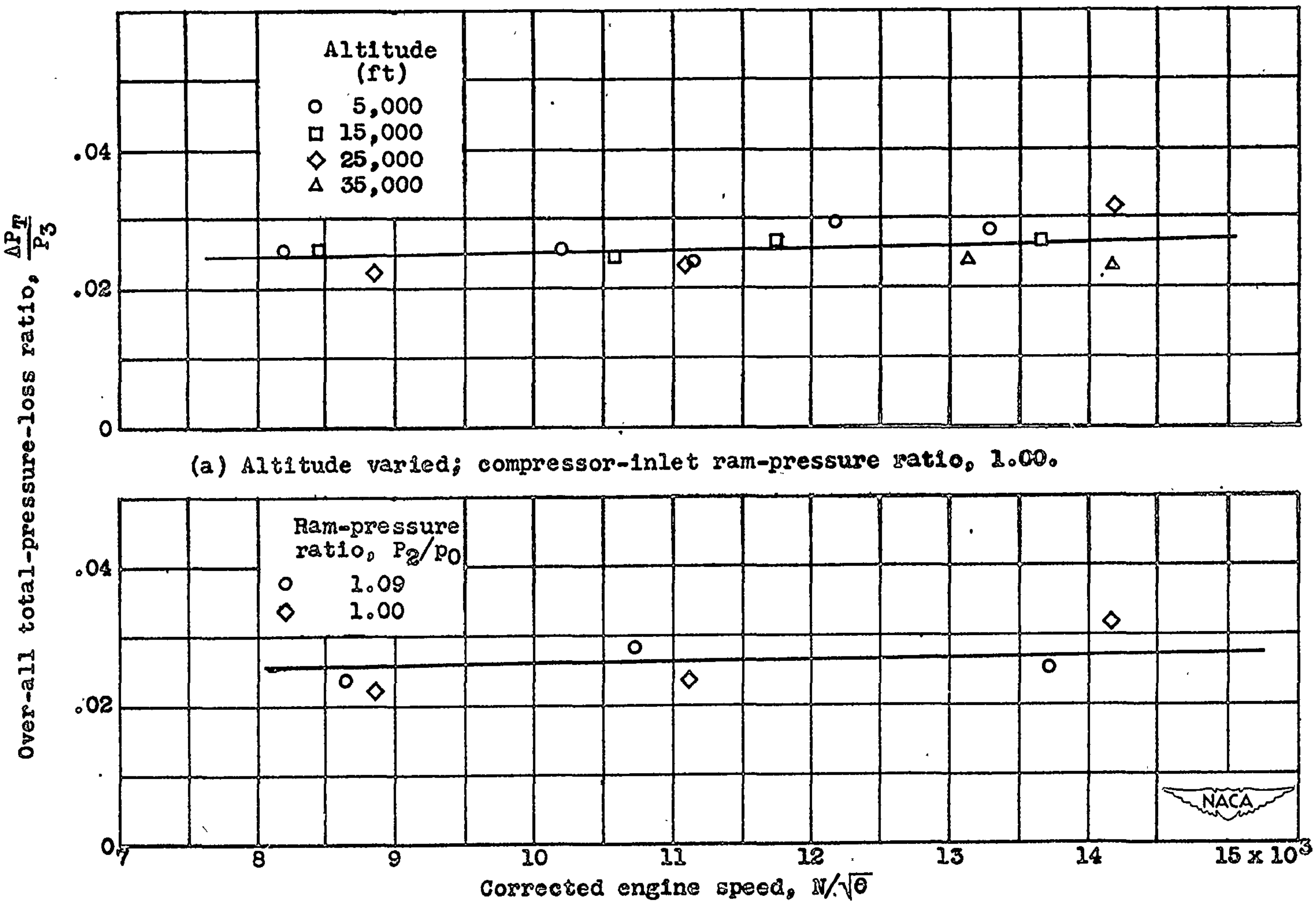


Figure 7. - Instrumentation at tail-pipe-nozzle outlet, station 8.



(a) Altitude varied; compressor-inlet ram-pressure ratio, 1.00.

(b) Compressor-inlet ram-pressure ratio varied; altitude, 25,000 feet.

Figure 8. - Effect of corrected engine speed on over-all total-pressure-loss ratio for various altitudes and compressor-inlet ram-pressure ratios. Tail-pipe temperature, 1500° R.

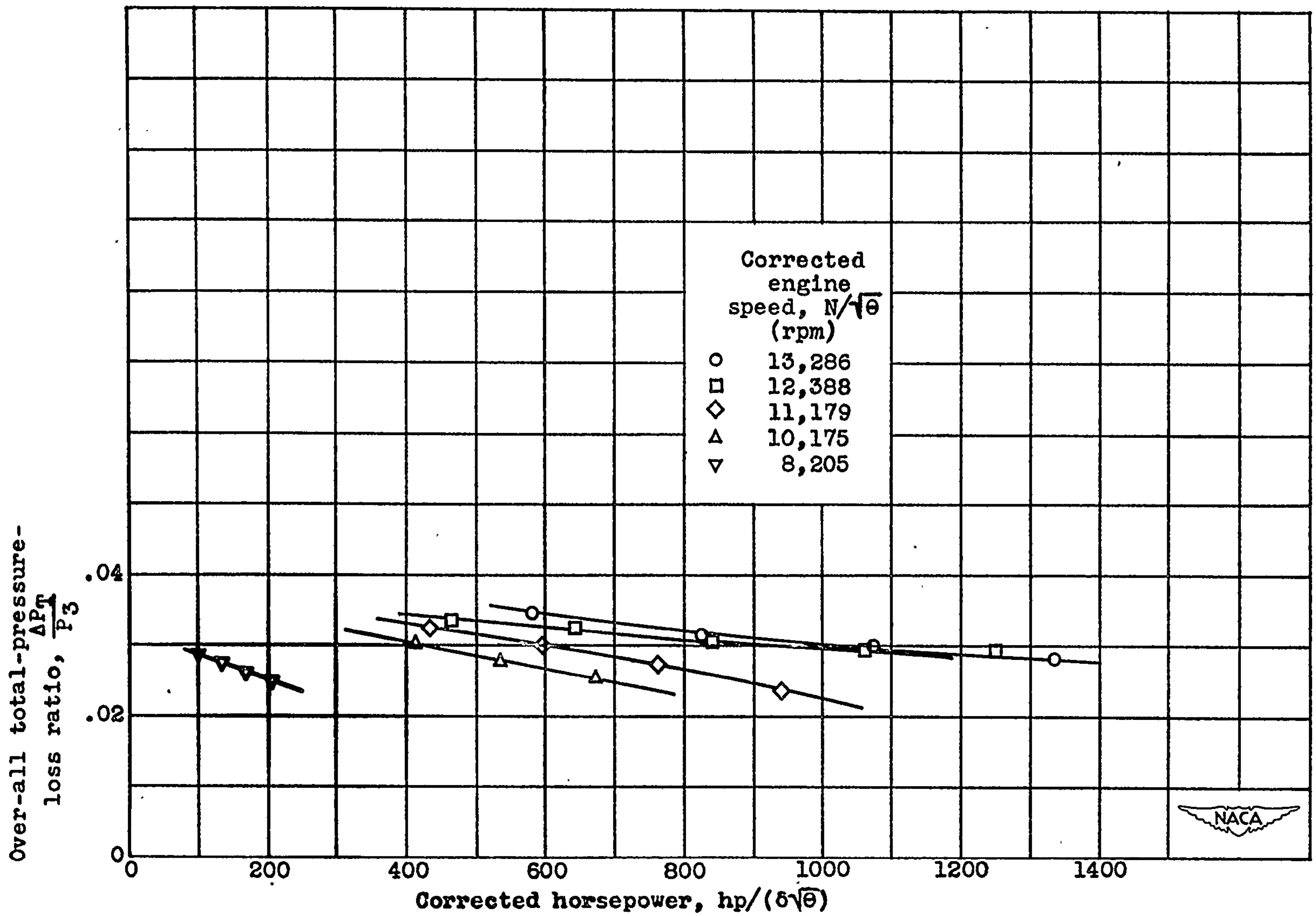
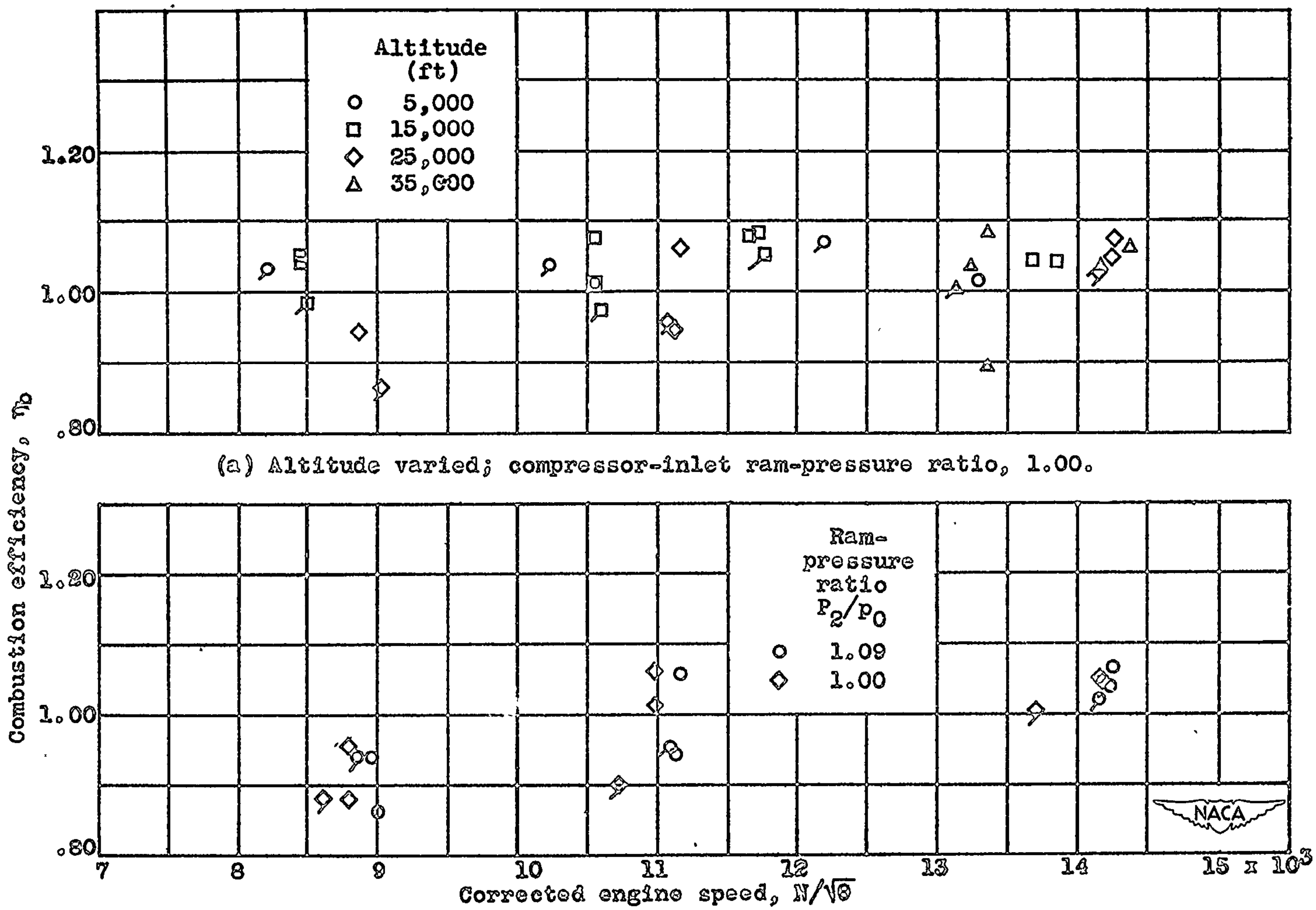


Figure 9. - Effect of corrected horsepower on over-all total-pressure-loss ratio for various corrected engine speeds. Altitude, 5000 feet; compressor-inlet ram-pressure ratio, 1.00.



(b) Compressor-inlet ram-pressure ratio varied; altitude, 25,000 feet.

Figure 10. - Effect of corrected engine speed on combustion efficiency for various altitudes and compressor-inlet ram-pressure ratios. (Tailed points indicate tail-pipe temperature of 1500° R.)

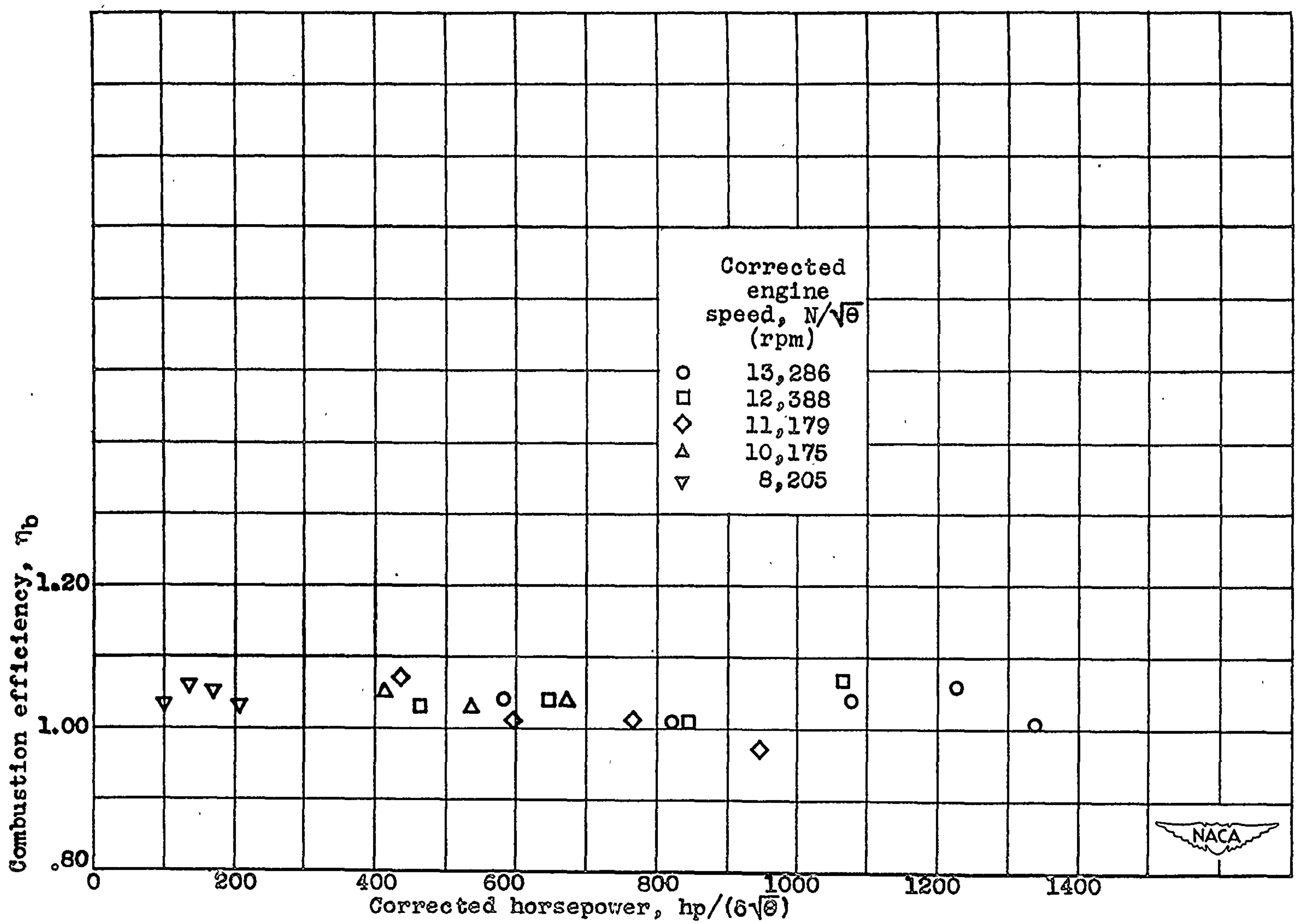
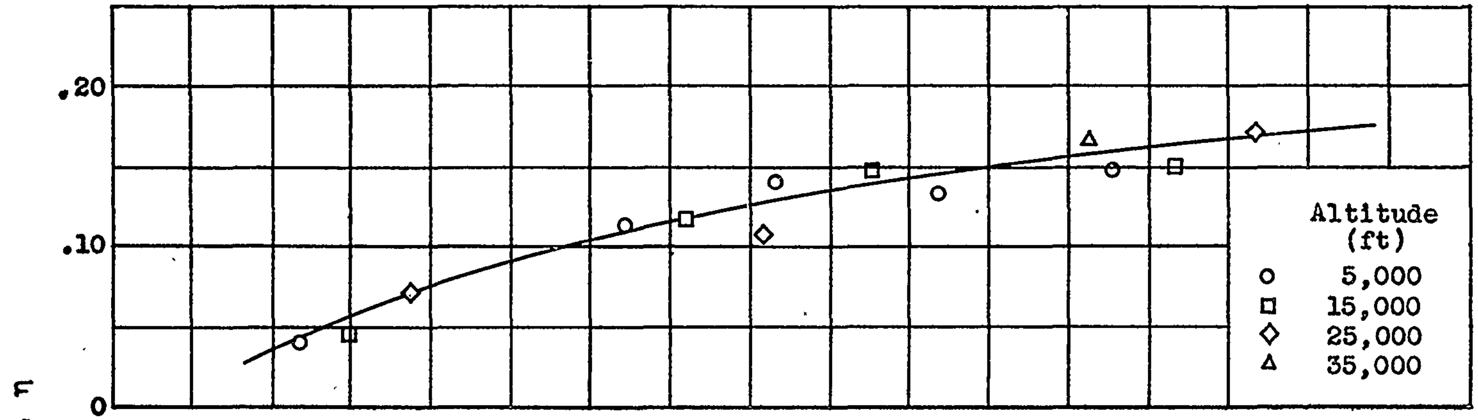
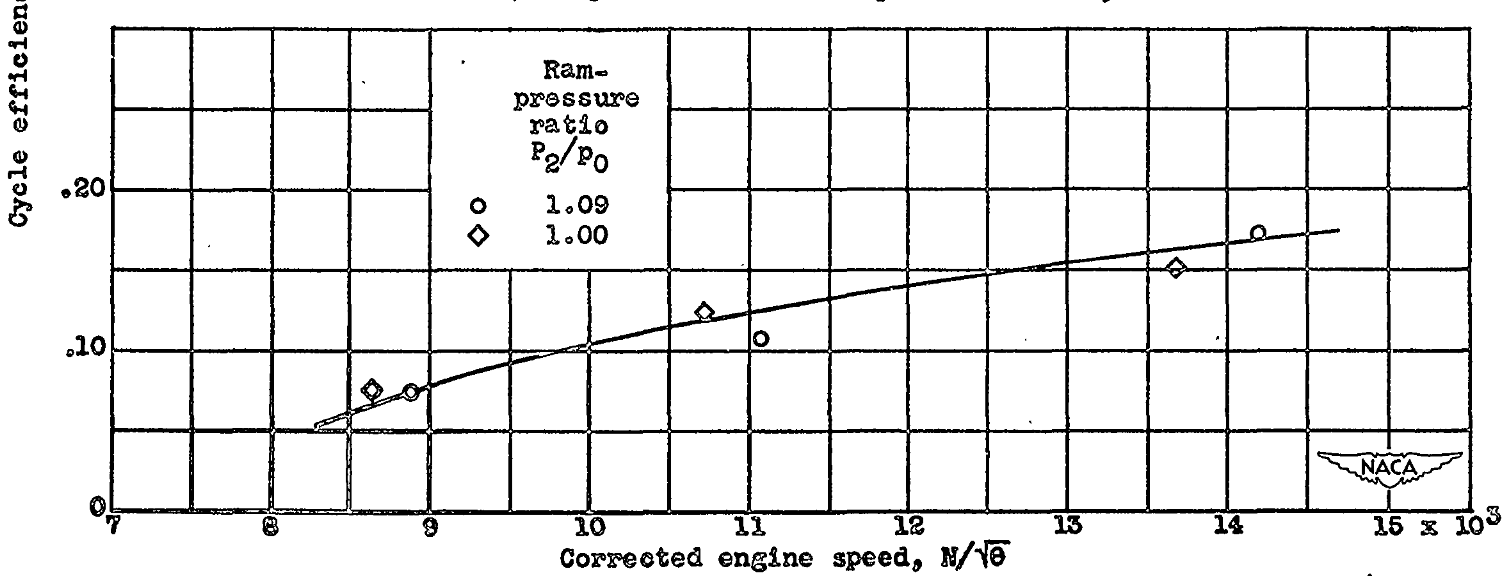


Figure 11. - Effect of corrected horsepower on combustion efficiency for various corrected engine speeds. Altitude, 5000 feet; compressor-inlet ram-pressure ratio, 1.00.



(a) Altitude varied; compressor-inlet ram-pressure ratio, 1.00.



(b) Compressor-inlet ram-pressure ratio varied; altitude, 25,000 feet.

Figure 12. - Effect of corrected engine speed on cycle efficiency for various altitudes and compressor-inlet ram-pressure ratios. Tail-pipe temperature, 1500° R.

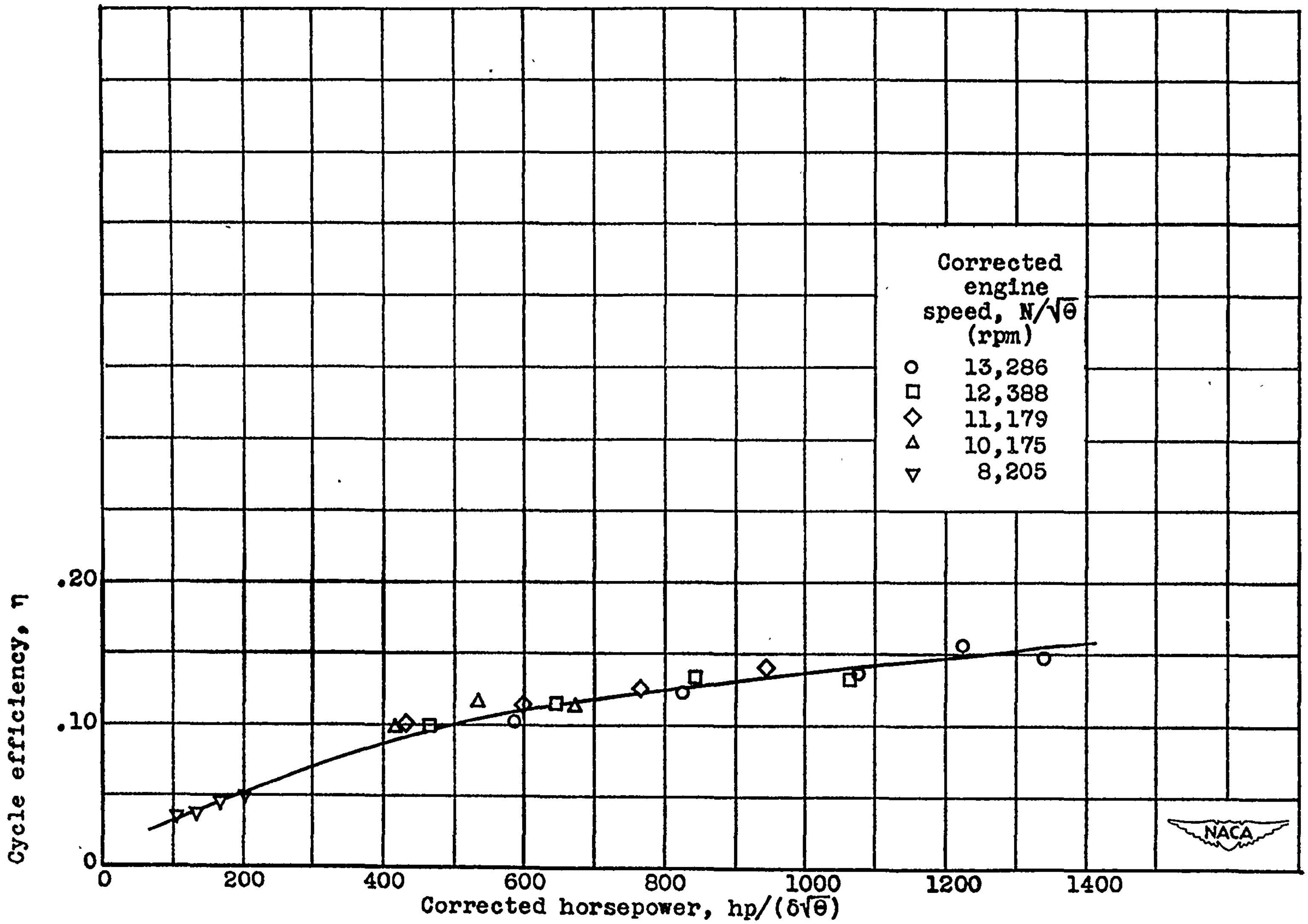
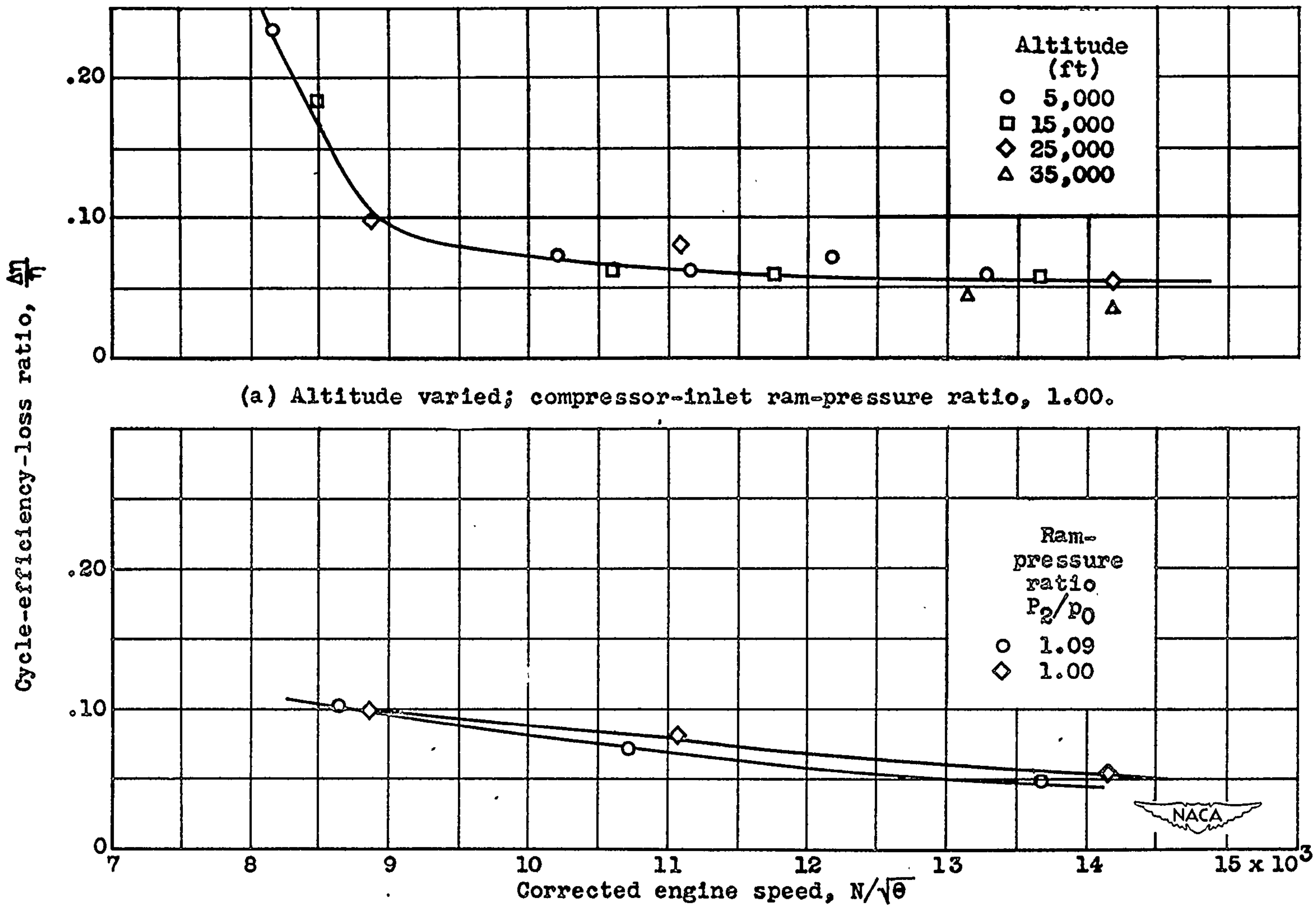


Figure 13. - Effect of corrected horsepower on cycle efficiency for various corrected engine speeds. Altitude, 5000 feet; compressor-inlet ram-pressure ratio, 1.00.



(a) Altitude varied; compressor-inlet ram-pressure ratio, 1.00.

(b) Compressor-inlet ram-pressure ratio varied; altitude, 25,000 feet.

Figure 14. - Effect of corrected engine speed on loss in cycle-efficiency ratio for various altitudes and compressor-inlet ram-pressure ratios. Tail-pipe temperature, 1500° R.

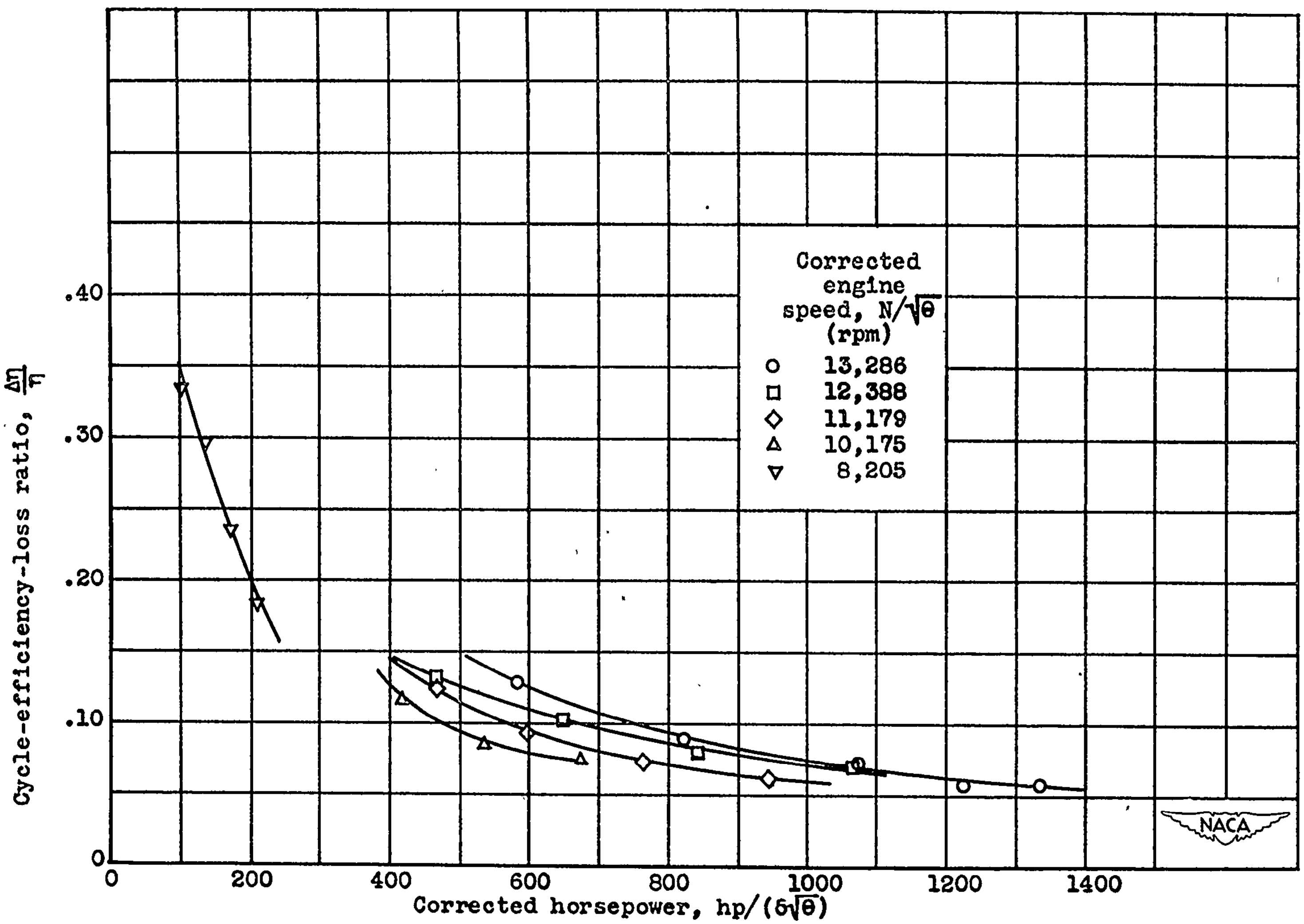


Figure 15. - Effect of corrected horsepower on loss in cycle-efficiency ratio for various corrected engine speeds. Altitudè, 5000 feet; compressor-inlet ram-pressure ratio, 1.00.