RESEARCH MEMORANDUM

PRELIMINARY RESULTS OF AN ALTITUDE-WIND-TUNNEL
INVESTIGATION OF AN AXIAL-FLOW GAS
TURBINE-PROPELLER ENGINE

I - PERFORMANCE CHARACTERISTICS

By Martin J. Saari and Lewis E. Wallner

Flight Propulsion Research Laboratory
Cleveland, Ohio

NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

WASHINGTON
August 2, 1948
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

PRELIMINARY RESULTS OF AN ALTITUDE-WIND-TUNNEL INVESTIGATION

OF AN AXIAL-FLOW GAS TURBINE-PROPELLER ENGINE

I - PERFORMANCE CHARACTERISTICS

By Martin J. Saari and Lewis E. Wallner

SUMMARY

A preliminary investigation of an axial-flow gas turbine-propeller engine has been conducted in the Cleveland altitude wind tunnel. Performance data were obtained for engine speeds from 8000 to 13,000 rpm at altitudes from 5000 to 35,000 feet and compressor-inlet ram-pressure ratios from 1.00 to 1.17. In order to conserve turbine life, the maximum exhaust-gas temperatures were limited to values from 500 to 1000°F below the normal continuous temperature rating. A tabulation of performance data is presented together with curves of performance characteristics and a brief discussion of the results. A complete description of the instrumentation of the installation is given.

For a constant compressor-inlet ram-pressure ratio, the performance data obtained at different altitudes and engine speeds were generalized to standard sea-level conditions with reasonable accuracy. The specific fuel consumption at a given corrected engine speed decreased as the corrected shaft horsepower was increased. At a corrected engine speed of 13,000 rpm, the specific fuel consumption based on shaft horsepower decreased from 1.41 at a corrected shaft horsepower of 700 to 0.86 at a corrected shaft horsepower of 1800. At each altitude and engine speed an increase in shaft horsepower of 100 percent was accompanied by an increase in jet thrust of about 10 percent. When the engine speed was changed from 10,000 to 13,000 rpm at a constant turbine-inlet temperature, the jet thrust increased at a greater rate than the shaft horsepower. At a turbine-inlet temperature of 2000°F, this change in engine speed was accompanied by an increase in jet thrust of about 115 percent and an increase in shaft horsepower of 55 percent. In order to obtain an optimum ratio of shaft
horsepower to the kinetic energy of the jet at operating conditions other than the design condition, a variable-area tail-pipe nozzle is required. An increase in compressor-inlet ram-pressure ratio from 1.00 to 1.13 did not appreciably affect the engine performance.

INTRODUCTION

An investigation has been conducted in the Cleveland altitude wind tunnel to determine the altitude performance and operational characteristics of an axial-flow gas turbine-propeller engine. Detailed pressure and temperature measurements were taken at eight stations in order to evaluate the over-all and component performance of the engine. Performance data were obtained at altitudes from 5000 to 35,000 feet and compressor-inlet ram-pressure ratios from 1.00 to 1.17, corresponding to airspeeds from 0 to 317 miles per hour. The engine was operated at speeds from 8000 to 13,000 rpm. The tunnel temperatures and pressures were maintained at approxi-mately NACA standard altitude conditions.

A tabulation of performance data is presented together with performance characteristic curves. The effects of engine speed, altitude, and ram-pressure ratio on engine performance character-istics are shown and discussed. The data obtained at several altitudes have been generalized and the methods employed in obtaining generalized performance curves are presented.

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 per \(^\circ\)F
- \( F_j \) jet thrust, pounds
- \( f/a \) fuel-air ratio
- \( g \) acceleration due to gravity, feet per second per second
- \( H \) enthalpy, Btu per pound
- \( J \) mechanical equivalent of heat, foot-pounds per Btu
N  engine speed, rpm
P  total pressure, pounds per square foot absolute
p  static pressure, pounds per square foot absolute
R  gas constant
shp shaft horsepower measured at torquemeter
ghp horsepower loss in high-speed reduction gear
hp  shp + ghp
T  total temperature, °R
T_i indicated temperature, °R
t  static temperature, °R
V  velocity, feet per second
W_a air flow, pounds per second
W_f fuel flow, pounds per hour
W_g gas flow, pounds per second
W_f/shp specific fuel consumption, pounds per shaft horsepower-hour
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

Subscripts:
0  tunnel test-section air stream
1  wing-duct inlet
2  compressor inlet
3  compressor outlet
5  turbine inlet
8  tail-pipe-nozzle outlet

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

\( F_j/\bar{\theta} \)  corrected jet thrust, pounds
\( N/\sqrt{\bar{\theta}} \)  corrected engine speed, rpm
\( (W_a\sqrt{\bar{\theta}})/\bar{\theta} \)  corrected air flow, pounds per second
\( W_f/(\bar{\theta}\sqrt{\bar{\theta}}) \)  corrected fuel flow, pounds per hour
\( hp/(\bar{\theta}\sqrt{\bar{\theta}}) \)  corrected horsepower
\( T_5/\bar{\theta} \)  corrected turbine-inlet temperature

ENGINE, INSTALLATION, AND INSTRUMENTATION

Engine

The T51 gas turbine-propeller engine investigated has a 14-stage axial-flow compressor, nine cylindrical combustion chambers, a single-stage turbine, an exhaust cone, and a two-stage planetary reduction gear. A sectional drawing of the engine showing the location of these components is given in figure 1. The engine has a maximum over-all diameter of 37 inches and an over-all length of 116 inches with a straight exhaust cone and without a propeller. A tail pipe 96 inches in length and 14 inches in diameter was used in this investigation. The dry weight of the engine, including piping and all accessories, is approximately 1980 pounds.

The operating limits of the engine for static sea-level conditions as established by the manufacturer are:
Turbine speed, rpm:
  Maximum overspeed ........................................ 13,300
  Normal rated ............................................. 13,000
  Idling ..................................................... 10,000
Exhaust-gas temperatures (at exhaust-cone outlet), °F:
  Military rating 5 minutes .................................. 1265
  Normal continuous rating ................................ 1170
  Starting and acceleration ................................ 1600
Bearing temperatures, °F ................................... 250
Vibration, in.:
  At turbine frequency ..................................... 0.004
  At propeller frequency ................................... 0.025

Air enters the engine through a screened annular inlet surrounding the aft gear casing and passes into the first stage of the compressor through a set of inlet guide vanes. The air travels through the 14-stage compressor with a resultant increase in pressure and temperature. Small quantities of air are bled through the compressor casing from the fifth and sixth stator stages to cool the turbine wheel and pressurize the compressor balance piston chamber at the front of the compressor. Air is discharged from the compressor through two sets of guide vanes and turned 180° before entering the combustion chambers.

Most of the air flows directly into the combustion chambers and a small part is directed through the hollow turbine-nozzle vanes before entering the combustion zone. The combustion chambers are the counter-flow type and consist of an outer shell and a perforated inner liner. A duplex fuel nozzle is located in the dome of each combustion chamber. Spark plugs are installed in two of the combustion chambers and ignition in the other chambers is accomplished through cross-fire tubes. Quick-disconnect clamps connect the combustion chambers to the main frame.

Products of combustion leaving the combustion chambers pass through transition sections to the annular turbine-nozzle where the gases are accelerated. A large part of the energy of the high-velocity gases is absorbed by the turbine to drive the propeller shaft, the compressor, the reduction gears, and accessories. The gases are discharged from the turbine into an annular exhaust cone and through a tail pipe, where the remaining energy is utilized as jet thrust.

The turbine shaft passes through the compressor rotor and is supported by two journal-sleeve bearings. A splined sleeve coupling connects the turbine shaft with the reduction-gear assembly. The reduction gear consists of two planetary gear systems in series,
with a speed reduction ratio of 11.3513:1. The ring gear of the high-speed stage is the floating type and its motion is restrained by six hydraulic pistons that provide a means of determining the shaft torque. The propeller shaft is connected to the low-speed planetary gear and is supported at the forward end by the main thrust ball bearing. Accessories are driven from a ring gear attached to the low-speed planetary cage.

Installation

The engine was installed in a streamline nacelle-wing combination, which was supported in the wind-tunnel test section by the tunnel balance frame. The axial center line of the nacelle coincided with the chord line of the wing and with the longitudinal center line of the tunnel test section. (See figs. 2 to 4.) The wing had an NACA 65,2-014 airfoil section with a chord length of 15 feet. Maximum diameter of the nacelle was 42.7 inches. When completely faired, the nacelle had a fineness ratio of 6.86. The engine was supported in the nacelle by two self-aligning ball and socket mounts located on each side of the forward gear casing and by a tie bolt on the bottom of the turbine main frame.

The engine was equipped with a Hamilton-Standard four-blade super hydromatic propeller (hub design, 4260) that is 12 feet, 7 inches in diameter. The propeller included a self-contained governor assembly and a blade-angle indicating mechanism. The controls were modified to provide either manual or constant-speed operation of the propeller. The maximum rate of pitch change as rated by the manufacturer is 45° per second. In order to set accurately the test conditions, the propeller was operated by means of the manual control and restrictions were inserted in the control oil passages to reduce the manual rate of pitch change. For this investigation the low- and high-blade angle stops were set at 4° and 46°, respectively. The minimum flight blade angle stop was set at 12° and the feathering blade angle stop was set at 82°.

Air was supplied to the engine from the tunnel air stream through two wing ducts with leading-edge inlets in the propeller slipstream. The centers of the inlet were located along the wing span at about 60 percent of the propeller radius. (See figs. 4 and 5.) The air entering the wing ducts was turned 90° with the aid of guide vanes. A small part of the air was diverted from each duct for cooling the engine nacelle chamber. At the engine, the wing ducts were joined to form an annulus around the aft reduction-gear casing (fig. 6). This annulus was attached to the compressor-inlet flange. (See fig. 3.) Splitter plates were inserted vertically between the two halves of the annulus to reduce rotational flow.
Instrumentation

Pressure and temperature measurements were taken at eight stations throughout the installation. Details of the instrumentation installation at each measuring station are given in the appendix.

PROCEDURE

Performance characteristics of the engine were obtained at altitudes from 5000 to 35,000 feet, compressor-inlet ram-pressure ratios from 1.00 to 1.17, and a range of shaft horsepowers at engine speeds from 8000 to 13,000 rpm. In order to lengthen the turbine life, the engine was not operated at maximum power for any of the conditions presented.

Jet thrust was calculated from pressure and temperature measurements at the tail-pipe-nozzle outlet. Shaft horsepower was determined from torquemeter pressure measurements. The shaft power determined from the torquemeter pressure is greater than the propeller shaft power by an amount equal to the sum of the low-speed reduction gear loss and the power required to drive the accessories. The values of shaft horsepower presented in the uncorrected performance data were measured at the torquemeter. For the generalized data, however, the power loss in the high-speed reduction gear was added to the torquemeter power. Thus the corrected horsepower represents the total turbine power less the power required to drive the compressor.

Pressures were measured on water and mercury manometers and were photographically recorded. Temperatures were measured and recorded by two self-balancing potentiometers. The engine speed was determined with a tachometer and the exhaust-gas temperature for setting limiting test conditions was indicated by two thermocouples at station 7 (fig. 1) that were connected in parallel to a temperature gage on the engine panel.

METHOD OF CALCULATIONS

Temperatures

The tail-pipe-nozzle outlet and compressor-inlet temperatures were calculated from the indicated temperature, using a thermo-couple recovery factor of 0.85, and respective values of pressure, temperature, and ratio of specific heats.
\[
T = \frac{\gamma - 1}{\gamma} T_1 \left( \frac{P}{P_1} \right)^{\gamma - 1} \left[ \frac{\gamma - 1}{\gamma} \left( \frac{P}{P_1} \right) - 1 \right]
\]

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

The enthalpy drop through the turbine included the power required to drive the compressor, the shaft power measured at the torquemeter, and the power loss in the high-speed reduction gear:

\[
H_5 = \left( \frac{\text{shp} + \text{shp}}{W_g} \right) \frac{550}{\gamma} + \left( H_3 - H_2 \right) + H_8
\]

\[
T_5 = \frac{H_5}{c_p \gamma}
\]

An integrated value of \( c_p \) was used in this equation. The gear horsepower used in calculating \( H_5 \) represents the power loss in the high-speed reduction gear and was estimated to vary from 50 horsepower at an engine speed of 13,000 rpm to 25 horsepower at 8000 rpm.

**Velocity**

The tunnel velocity was determined from the following relation:

\[
v_0 = \sqrt{\frac{2 \gamma R_0 T_0}{\gamma - 1} \left[ 1 - \left( \frac{P_0}{P_0} \right)^{\gamma - 1} \right]}
\]
Thrust

The jet thrust of the engine was calculated from the following equation, which was derived in reference 1:

\[
F_J = \frac{2\gamma_8}{\gamma_8 - 1} p_8 A_8 \left[ \frac{\gamma_8}{\left(\frac{P_8}{p_8}\right)^{\frac{\gamma_8}{\gamma_8 - 1}}} - 1 \right] + A_8 \left( P_8 - P_0 \right)
\]

Air Flow

Engine air flow was determined from measurements of the compressor inlet and at the tail-pipe-nozzle outlet. The air flow presented in the tabulation data and used in the performance calculations was obtained from measurements at the compressor inlet because of more comprehensive instrumentation at that station. The air flow at the compressor inlet was calculated from the following relation:

\[
W_a,2 = p_2 A_2 \sqrt{\frac{2\gamma g}{(\gamma - 1) R_2 t_2}} \sqrt{\left(\frac{P_2}{p_2}\right)^{\frac{\gamma - 1}{\gamma}}} - 1
\]

Gas flow at the tail-pipe-nozzle outlet was obtained from the following relation:

\[
W_g = p_8 A_8 \sqrt{\frac{2\gamma_8 g}{(\gamma_8 - 1) R_8 t_8}} \sqrt{\left(\frac{P_8}{p_8}\right)^{\frac{\gamma_8 - 1}{\gamma_8}}} - 1
\]

The air flow was then found from

\[
W_a = W_g - \left(\frac{W_f}{3600}\right)
\]
RESULTS AND DISCUSSION

Preliminary performance data are presented in table I. In the following sections, the effects of altitude, engine speed, and ram-pressure ratio on engine performance are discussed; generalized performance data and the methods employed in obtaining generalized performance curves are presented.

On several occasions oil entered the compressor air passage. Oil and dirt that adhered to the blades may have had some adverse effect on the compressor performance.

Altitude. - Performance data obtained at altitudes from 5000 to 35,000 feet, an engine speed of 13,000 rpm, and a compressor-inlet ram-pressure ratio of 1.00 are shown in figures 7 to 12. The calculated turbine-inlet temperature (fig. 7) increased linearly with shaft horsepower at each altitude. Using these values of turbine-inlet temperature, constant temperature contours are plotted in figures 8, 9, 11, and 12. From the data presented in these figures, the performance of the engine at an engine speed of 13,000 rpm and a ram-pressure ratio of 1.00 can be estimated for the range of altitudes investigated for turbine-inlet temperatures between 1700° and 2100° R.

At a given turbine-inlet temperature, the fuel flow decreased as the altitude was increased (fig. 8). The data shown in figure 9 indicate that the lowest specific fuel consumption at each turbine-inlet temperature occurred at an altitude of approximately 20,000 feet. The engine air flow remained constant at each altitude as the shaft horsepower was changed. (See fig. 10.) At constant turbine-inlet temperatures above 1800° R, the fuel-air ratio did not change appreciably with changes in altitude (fig. 11). At each altitude the jet thrust increased about 10 percent for an increase in shaft horsepower of 100 percent (fig. 12).

Engine speed. - The effect of engine speed on engine performance at an altitude of 5000 feet and a ram-pressure ratio of 1.00 is shown in figures 13 to 18. Constant temperature contours are plotted in figures 14, 15, 17, and 18, using the calculated values of turbine-inlet temperature in figure 13. With these contours superimposed on the data, the engine performance can be estimated for any engine speed at an altitude of 5000 feet and a ram-pressure ratio of 1.00.

The calculated turbine-inlet temperatures shown in figure 13 increase linearly with shaft horsepower at each engine speed. The temperature lines for constant engine speeds intersect owing to the change in the ratio of shaft horsepower to the kinetic energy of the jet as the engine speed is varied. The tail-pipe-nozzle area for
this type of engine is so selected that the ratio of shaft horsepower to the kinetic energy of the jet gives optimum economy at the design operating conditions. In order to obtain an optimum ratio of shaft horsepower to jet thrust power for operating conditions other than the design condition, a variable-area tail-pipe nozzle is therefore required.

At each engine speed the fuel flow (fig. 14), the fuel-air ratio (fig. 17), and the jet thrust (fig. 18) increase linearly with shaft horsepower, whereas the air flow (fig. 16) remains constant as the shaft horsepower is changed. An increase in shaft horsepower reduced the specific fuel consumption at each engine speed (fig. 15). As the engine speed is raised at a constant turbine-inlet temperature, the rate of change in fuel flow with shaft horsepower decreases as the turbine-inlet temperature is increased (fig. 14). The significance of this relation is shown in figure 15, in which the specific fuel consumption at each engine speed decreases as the turbine-inlet temperature increases. At a turbine-inlet temperature of 2000° R, the specific fuel consumption was about 1.06 and was not noticeably changed by variations in engine speed. At a turbine-inlet temperature of 1700° R, however, the specific fuel consumption increased sharply from 1.52 at an engine speed of 10,000 rpm to 1.87 at 13,000 rpm.

An increase in engine speed at a constant turbine-inlet temperature reduced the fuel-air ratio at engine speeds up to 12,000 rpm (fig. 17). An increase in engine speed from 12,000 to 13,000 rpm did not change the fuel-air ratio in a manner consistent with the data obtained at lower speeds. The shaft horsepower, however, does not represent the total power output of the engine in that the kinetic energy of the jet is not included. As the engine speed was increased from 10,000 to 13,000 rpm at a constant turbine-inlet temperature, the jet thrust increased at a much greater rate than the shaft horsepower. (See fig. 18.) Because the change in shaft horsepower is small, the fuel-air ratios for engine speeds of 13,000 and 12,000 rpm are nearly the same (fig. 17).

Changing the engine speed from 10,000 to 13,000 rpm increased the jet thrust about 115 percent at each value of turbine-inlet temperature. The same increase in engine speed, however, resulted in an increase in shaft horsepower of only 7 percent at a turbine-inlet temperature of 1700° R and 55 percent at 2000° R (fig. 18).

Ram-pressure ratio. - The effect on engine performance of an increase in compressor-inlet ram-pressure ratio from 1.00 to 1.13 at an altitude of 25,000 feet and an engine speed of 13,000 rpm is shown in figures 19 to 24. The small change in
ram-pressure ratio had no apparent effect on the relation between shaft horsepower and turbine-inlet temperature (fig. 19). An increase in ram-pressure ratio resulted in a slight decrease in fuel flow, specific fuel consumption, and fuel-air ratio (figs. 20, 21, and 23, respectively), whereas the jet thrust increased slightly (fig. 24). The increase in ram-pressure ratio did not appreciably change the air flow (fig. 22). Because of data scatter, the fuel-air ratios presented in figure 23 were obtained from faired values of fuel flow (fig. 20) and air flow (fig. 22).

Generalized engine performance. - The pressure and temperature factors 5 and \( \theta \) have been applied to the performance data to determine whether the engine performance obtained at several altitudes could be generalized. The generalized performance parameters used are corrected engine speed \( N/\sqrt{\theta} \), corrected fuel flow \( W_f/(\theta \sqrt{\theta}) \), corrected air flow \( W_a/(\theta \sqrt{\theta}) \), corrected horsepower \( \text{hp}/(\theta \sqrt{\theta}) \), and corrected turbine-inlet temperature \( T_5/\theta \). These parameters were developed from concepts of flow similarity in reference 2.

The performance data obtained at several altitudes and engine speeds at a constant compressor-inlet ram-pressure ratio were generalized to standard sea-level conditions as shown in figures 25 to 34. Representative generalized data showing the variation of corrected fuel flow with corrected horsepower are presented in figure 25 for altitudes from 5000 to 35,000 feet, a compressor-inlet ram-pressure ratio of 1.00, and corrected engine speeds from 8200 to 14,200 rpm. Similar data showing the variation of corrected turbine-inlet temperature with corrected fuel flow for the same range of corrected engine speeds are given in figure 26. The generalized data presented in figures 25 and 26 were cross-plotted in figure 27, which shows the variation of corrected fuel flow with corrected engine speed for selected constant values of corrected horsepower and corrected turbine-inlet temperature.

The relation between corrected horsepower and corrected turbine-inlet temperature is presented in figure 28 for several corrected engine speeds. These values were cross-plotted in figure 29 to show the variation of corrected horsepower with corrected engine speed for constant corrected turbine-inlet temperatures. The reason for the decrease in corrected horsepower at the higher values of corrected engine speed has not been determined. A combination of the data in figures 27 and 29 gave two families of curves that show the variation of specific fuel consumption with corrected engine speed for constant corrected values of horsepower and turbine-inlet temperature (fig. 30). For a given engine speed, the specific fuel consumption decreased as the horsepower and turbine-inlet temperature increased.
At a corrected engine speed of 13,000 rpm, the specific fuel consumption based on shaft horsepower decreased from 1.41 at a corrected horsepower of 700 to 0.86 at 1800. At each constant corrected turbine-inlet temperature, the best fuel economy was obtained at a corrected engine speed of about 12,000 rpm. However, the specific fuel consumption based on shaft horsepower does not represent the over-all specific fuel consumption because the jet thrust has not been included.

The variation of corrected jet thrust with corrected horsepower and corrected turbine-inlet temperature for a range of corrected engine speeds is shown in figures 31 and 32, respectively. These data are cross-plotted in figure 33 to show the relation between corrected jet thrust and corrected engine speed for constant corrected values of horsepower and turbine-inlet temperature. The jet thrust increased as the engine speed and the horsepower were increased.

Corrected engine air flows obtained from measurements at the compressor inlet are shown in figure 34. These data were obtained at altitudes from 5000 to 35,000 feet at a compressor-inlet ram-pressure ratio of 1.00 for the range of shaft horsepower investigated. The values of air flow calculated from pressure and temperature measurements at the tail-pipe-nozzle outlet were approximately 4 percent lower than those obtained from measurements at the compressor inlet.

**SUMMARY OF RESULTS**

An investigation of an axial-flow gas turbine-propeller engine in the Cleveland altitude wind tunnel at altitudes from 5000 to 35,000 feet, ram-pressure ratios from 1.00 to 1.13, and engine speeds from 8000 to 13,000 rpm gave the following results:

1. Performance data obtained at several altitudes and engine speeds for a constant compressor-inlet ram-pressure ratio were generalized to standard sea-level conditions with reasonable accuracy.

2. The specific fuel consumption at a given corrected engine speed decreased as the corrected horsepower increased. At a corrected engine speed of 13,000 rpm, the specific fuel consumption based on shaft horsepower decreased from 1.41 at a corrected horsepower of 700 to 0.86 at 1800.
3. At each altitude and engine speed, an increase in shaft horsepower of 100 percent was accompanied by an increase in jet thrust of about 10 percent. As the engine speed was increased from 10,000 to 13,000 rpm at a constant turbine-inlet temperature, the jet thrust increased at a greater rate than the shaft horsepower. At a turbine-inlet temperature of 2000°C, this change in engine speed was accompanied by increases of 115 percent in jet thrust and 55 percent in shaft horsepower.

4. In order to obtain an optimum ratio of shaft horsepower to the kinetic energy of the jet at operating conditions other than the design condition, a variable-area tail-pipe nozzle is required.

5. An increase in compressor-inlet ram-pressure ratio from 1.00 to 1.13 had little effect on the engine performance.

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

Instrumentation was installed at eight measuring stations throughout the engine (fig. 1) to obtain the detailed pressure and temperature measurements from which the over-all and component engine performance could be calculated. Photographs and sectional drawings showing the instrumentation installed at each measuring station are presented in figures 35 to 41. The number of total- and static-pressure tubes and thermocouples installed at each station and the flow area at each station are given in table II. In addition to the pressure and temperature measurements taken at the eight stations, measurements were taken at other locations throughout the engine and installation.

Temperatures were measured with iron-constantan thermocouples at stations 1 to 4 and chromel-alumel thermocouples at the other stations. These temperatures were recorded on self-balancing potentiometers. Pressures were measured with water or mercury Alkazene manometers, depending upon the magnitude of the pressure at the measuring station, and were photographically recorded. Transverse vibrations of the engine in the horizontal and vertical planes were transmitted by vibration pickups located at the front bearing support, the front flange of the compressor, and the turbine flange. The vibration transmitters were connected to a vibration indicator in the control room through a selector switch.

REFERENCES


<table>
<thead>
<tr>
<th>Run</th>
<th>Altitude (ft)</th>
<th>Engine speed (rpm)</th>
<th>Shaft horsepower</th>
<th>Static atmospheric pressure, ( \text{Po/Po}_0 )</th>
<th>Normal static pressure, ( \text{Pn/Po}_0 )</th>
<th>Temperature, ( \text{Temp/Po}_0 )</th>
<th>Air flow, ( \text{Wm} )</th>
<th>Air flow ratio, ( \text{f_a} )</th>
<th>Peak air ratio, ( \text{f_p} )</th>
<th>Compressor pressure ratio, ( \text{r} )</th>
<th>Turbine pressure ratio, ( \text{r_t} )</th>
<th>Turbine nozzle angle, ( \beta )</th>
<th>Turbine nozzle cooling air flow, ( \text{Wm} )</th>
<th>Corrected corrected engine horsepower, ( \text{HP} )</th>
<th>Corrected corrected engine horsepower, ( \text{HP}/\text{lbm} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>5,000</td>
<td>13,000</td>
<td>425</td>
<td>0.99</td>
<td>0.91</td>
<td>1780</td>
<td>504</td>
<td>775</td>
<td>1.025</td>
<td>17.944</td>
<td>0.129</td>
<td>1.783</td>
<td>0.130</td>
<td>3.10</td>
<td>0.10</td>
</tr>
<tr>
<td>2</td>
<td>6,000</td>
<td>13,000</td>
<td>519</td>
<td>0.99</td>
<td>0.91</td>
<td>1780</td>
<td>501</td>
<td>970</td>
<td>1.045</td>
<td>16.04</td>
<td>0.144</td>
<td>1.783</td>
<td>0.130</td>
<td>3.10</td>
<td>0.10</td>
</tr>
<tr>
<td>3</td>
<td>7,000</td>
<td>13,000</td>
<td>619</td>
<td>0.99</td>
<td>0.91</td>
<td>1780</td>
<td>501</td>
<td>970</td>
<td>1.045</td>
<td>16.04</td>
<td>0.144</td>
<td>1.783</td>
<td>0.130</td>
<td>3.10</td>
<td>0.10</td>
</tr>
<tr>
<td>4</td>
<td>8,000</td>
<td>13,000</td>
<td>719</td>
<td>0.99</td>
<td>0.91</td>
<td>1780</td>
<td>501</td>
<td>970</td>
<td>1.045</td>
<td>16.04</td>
<td>0.144</td>
<td>1.783</td>
<td>0.130</td>
<td>3.10</td>
<td>0.10</td>
</tr>
</tbody>
</table>

TABLE I - PERFORMANCE DATA
<table>
<thead>
<tr>
<th>Location</th>
<th>Station</th>
<th>Area (sq in.)</th>
<th>Thermocouples</th>
<th>Pressure tubes Total</th>
<th>Probe static</th>
<th>Wall static</th>
</tr>
</thead>
<tbody>
<tr>
<td>Upper lip, left wing duct</td>
<td>1</td>
<td>82.4</td>
<td>2</td>
<td>26</td>
<td>2</td>
<td>17</td>
</tr>
<tr>
<td>Lower lip, left wing duct</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Left wing-duct inlet</td>
<td>1</td>
<td>82.4</td>
<td>2</td>
<td>26</td>
<td>2</td>
<td>6</td>
</tr>
<tr>
<td>Right wing-duct inlet (fig. 35)</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wing cooling-air bleed,</td>
<td>2.85</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td></td>
</tr>
<tr>
<td>left forward</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wing cooling-air bleed,</td>
<td>2.85</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td></td>
</tr>
<tr>
<td>left aft</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wing cooling-air bleed,</td>
<td>2.85</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td></td>
</tr>
<tr>
<td>right forward</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Wing cooling-air bleed,</td>
<td>2.85</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td></td>
</tr>
<tr>
<td>right aft</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Compressor inlet (fig. 36)</td>
<td>2</td>
<td>94.7</td>
<td>6</td>
<td>15</td>
<td>6</td>
<td>9</td>
</tr>
<tr>
<td>Compressor stators</td>
<td>2,3</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>14</td>
</tr>
<tr>
<td>Balance piston cooling</td>
<td>.30</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td></td>
<td>1</td>
</tr>
<tr>
<td>air ducts</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Compressor outlet (fig. 37)</td>
<td>3</td>
<td>48.5</td>
<td>9</td>
<td>12</td>
<td>2</td>
<td>5</td>
</tr>
<tr>
<td>Compressor-outlet elbow (fig. 37)</td>
<td>4</td>
<td></td>
<td></td>
<td>4</td>
<td>2</td>
<td></td>
</tr>
<tr>
<td>Turbine inlet (fig. 38)</td>
<td>5</td>
<td>25.0</td>
<td></td>
<td>5</td>
<td></td>
<td>5</td>
</tr>
<tr>
<td>Turbine cooling-air baffles</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Turbine cooling-air ducts</td>
<td>.60</td>
<td>1</td>
<td>1</td>
<td>1</td>
<td></td>
<td>1</td>
</tr>
<tr>
<td>Turbine outlet (fig. 39)</td>
<td>6</td>
<td>79.0</td>
<td>9</td>
<td>9</td>
<td>3</td>
<td>3</td>
</tr>
<tr>
<td>Exhaust-cone outlet (fig. 40)</td>
<td>7</td>
<td>154.0</td>
<td>4</td>
<td>4</td>
<td>2</td>
<td></td>
</tr>
<tr>
<td>Tail-pipe surface</td>
<td>7,8</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>3</td>
</tr>
<tr>
<td>Tail-pipe outlet surface</td>
<td>8</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td>4</td>
</tr>
<tr>
<td>Tail-pipe outlet rake (fig. 41)</td>
<td>8</td>
<td>154.0</td>
<td>6</td>
<td>16</td>
<td>3</td>
<td></td>
</tr>
<tr>
<td>Fuselage</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Engine external</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Bearings</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Fuel supply</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Starter</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>
Figure 1. - Side view of axial-flow gas turbine-propeller engine showing location of measuring stations.

Station
1 Wing-duct inlet (fig. 5)
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 2. - Top view of axial-flow gas turbine-propeller engine in altitude wind tunnel.
Figure 3. Installation of axial-flow gas turbine-propeller engine in nacelle.
Figure 4. - Front view of axial-flow gas turbine-propeller engine in altitude wind tunnel.
Figure 5. Sketch of axial-flow gas turbine-propeller engine installation showing location of wing ducts and inlets.
(a) Front view.  

(b) Rear view.  

Figure 6. - Induction duct annulus.
Figure 7.- Effect of shaft horsepower and altitude on turbine-inlet temperature. Engine speed, 13,000 rpm; compressor-inlet ram-pressure ratio, 1.00.
Figure 8.— Effect of shaft horsepower and altitude on engine fuel flow. Engine speed, 15,000 rpm; compressor-inlet ram-pressure ratio, 1.00.
Figure 9. - Effect of shaft horsepower and altitude on specific fuel consumption. Engine speed, 13,000 rpm; compressor-inlet ram-pressure ratio, 1.00.
Figure 10.- Effect of shaft horsepower and altitude on engine air flow. Engine speed, 13,000 rpm; compressor-inlet ram-pressure ratio, 1.00.
Figure 11.—Effect of shaft horsepower and altitude on fuel-air ratio. Engine speed, 13,000 rpm; compressor-inlet ram pressure ratio, 1.00.
Figure 12.— Effect of shaft horsepower and altitude on jet thrust. Engine Speed, 15,000 rpm; compressor-inlet ram-pressure ratio, 1.00.
Figure 13.—Effect of shaft horsepower and engine speed on turbine-inlet temperature. Altitude, 5000 feet; compressor-inlet ram-pressure ratio, 1.00.
Figure 14.—Effect of shaft horsepower and engine speed on fuel flow. Altitude, 5000 feet; compressor-inlet ram-pressure ratio, 1.00.
Figure 15.—Effect of shaft horsepower and engine speed on specific fuel consumption. Altitude, 5000 feet; compressor-inlet ram-pressure ratio, 1.00.
Figure 16.- Effect of shaft horsepower and engine speed on engine air flow. Altitude, 5000 feet; compressor-inlet ram-pressure ratio, 1.00.
Figure 17.— Effect of shaft horsepower and engine speed on fuel-air ratio. Altitude, 5000 feet; compressor-inlet ram-pressure ratio, 1.00.
Figure 18.—Effect of shaft horsepower and engine speed on jet thrust. Altitude, 5000 feet; compressor-inlet ram-pressure ratio, 1.00.
Figure 19.—Effect of shaft horsepower and compressor-inlet ram-pressure ratio on turbine-inlet temperature. Engine speed, 13,000 rpm; altitude, 25,000 feet.
Figure 20.- Effect of shaft horsepower and compressor-inlet ram-pressure ratio on fuel flow. Engine speed, 13,000 rpm; altitude, 25,000 feet.
Figure 21. - Effect of shaft horsepower and compressor-inlet ram-pressure ratio on specific fuel consumption. Engine speed, 13,000 rpm; altitude, 25,000 feet.
Figure 22.- Effect of shaft horsepower and compressor-inlet ram-pressure ratio on engine air flow. Engine speed, 15,000 rpm; altitude, 25,000 feet.
Figure 23. Effect of shaft horsepower and compressor-inlet ram-pressure ratio on fuel-air ratio. Engine speed, 13,000 rpm; altitude, 25,000 feet.
Figure 24. - Effect of shaft horsepower and compressor-inlet ram-pressure ratio on jet thrust. Engine speed, 15,000 rpm; altitude, 25,000 feet.
Figure 25.— Effect of corrected horsepower and corrected engine speed on corrected fuel flow. Compressor-inlet ram-pressure ratio, 1.00.
Figure 26. Effect of corrected fuel flow and corrected engine speed on corrected turbine-inlet temperature. Compressor-inlet ram-pressure ratio, 1.00.
Figure 27.- Effect of corrected engine speed and corrected horsepower on corrected fuel flow. Compressor-inlet ram-pressure ratio, 1.00.
Figure 28. - Effect of corrected horsepower and corrected engine speed on corrected turbine-inlet temperature. Compressor-inlet ram-pressure ratio, 1.00.
Figure 29.- Effect of corrected engine speed and corrected turbine-inlet temperature on corrected horsepower. Compressor-inlet ram-pressure ratio, 1.00. (Dashed portion of curves extrapolated.)
Figure 30.— Effect of corrected engine speed, corrected horsepower, and corrected turbine-inlet temperature on specific fuel consumption. Compressor-inlet ram-pressure ratio, 1.00.
Figure 51.- Effect of corrected horsepower and corrected engine speed on corrected jet thrust. Compressor-inlet ram-pressure ratio, 1.00.
Figure 32.- Effect of corrected turbine-inlet temperature and corrected engine speed on corrected jet thrust. Compressor-inlet ram-pressure ratio, 1.00.
Figure 35.— Effect of corrected engine speed, corrected horsepower, and corrected turbine-inlet temperature on corrected jet thrust. Compressor-inlet ram-pressure ratio, 1.00.
Corrected air flow \( \frac{W_a \sqrt{\phi}}{\phi} \), lb/sec

Figure 34. Effect of corrected engine speed and altitude on corrected engine air flow. Compressor inlet temperature ratio, 1.00.
(a) Location of instrumentation.

Figure 35. Instrumentation at wing-duct inlet, station 1.
(b) Front view showing survey rakes.

Figure 35. - Concluded. Instrumentation at wing-duct inlet, station I.
(a) Location of instrumentation.

Figure 36.- Instrumentation at compressor inlet, station 2.
(b) Installation of pressure and temperature rakes.

Figure 36. - Continued. Instrumentation at compressor inlet, station 2.
(c) Detail sketch of total-pressure, static-pressure, and thermocouple rakes.

Figure 36.—Concluded. Instrumentation at compressor inlet, station 2.
(a) Location of instrumentation at station 3.

Figure 37.- Instrumentation at compressor outlet, station 3, and compressor elbow, station 4.
(b) Installation of pressure and temperature rakes at station 3.

Figure 37. - Continued. Instrumentation at compressor outlet, station 3, and compressor elbow, station 4.
(c) Detail sketch of instrumentation at stations 3 and 4.

Figure 37.- Concluded. Instrumentation at compressor outlet, station 3, and compressor elbow, station 4.
(a) Location of instrumentation looking aft.

Figure 38.- Instrumentation at turbine inlet, station 5.
(b) Installation of total-and static-pressure tubes.

Figure 38. — Concluded. Instrumentation at turbine inlet, station 5.
(a) Location of instrumentation looking aft.

Figure 39.- Instrumentation at turbine outlet, station 6.
(b) Installation of total-pressure rakes and static-pressure tubes.

Figure 39. - Continued. Instrumentation at turbine outlet, station 6.
(c) Detail sketch of total-pressure rake and wafer static tube.
Figure 39—Concluded. Instrumentation at turbine outlet, station 6.
Figure 40. - Location of instrumentation at exhaust-cone outlet, station 7.
(a) Installation of tail-pipe-outlet rake.

Figure 41. - Instrumentation at tail-pipe-nozzle outlet, station 8.
(b) Detail sketch of tail-pipe-outlet rake.

Figure 41. - Concluded. Instrumentation at tail-pipe-nozzle outlet, station 8.