RESEARCH MEMORANDUM

ALTITUDE PERFORMANCE CHARACTERISTICS OF TAIL-PIPE BURNER WITH CONVERGING CONICAL BURNER SECTION ON J47 TURBOJET ENGINE

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Information so classified may be imparted only to persons in the military and naval services of the United States, appropriate civilian officers and employees of the Federal Government who have a legitimate interest therein, and to United States citizens of known loyalty and discretion who of necessity must be informed thereof.
An investigation of turbojet-engine thrust augmentation by means of tail-pipe burning was conducted in the NACA Lewis altitude wind tunnel. Performance data were obtained with a tail-pipe burner having a converging conical burner section installed on an axial-flow-compressor type turbojet engine over a range of simulated flight conditions and tail-pipe fuel-air ratios with a fixed-area exhaust nozzle.

A maximum tail-pipe combustion efficiency of 0.86 was obtained at an altitude of 15,000 feet and a flight Mach number of 0.23. At an altitude of 25,000 feet, a tail-pipe burner-inlet temperature of 1700° R, and rated engine speed, the ratio of augmented thrust to normal thrust increased from 1.3 at a flight Mach number of 0.23 to 1.4 at a flight Mach number of 0.76. Tail-pipe burner operation was possible up to an altitude of 45,000 feet at a flight Mach number of 0.23.

An extensive research program on thrust augmentation of turbojet engines has been conducted in the NACA Lewis altitude wind tunnel to determine the effect of various component designs on the performance and operational characteristics of several tail-pipe burner installations. The effect of flame-holder and fuel-system design on the burner performance and the effect of altitude and flight Mach number on over-all performance with a fixed-area exhaust nozzle are reported in references 1 to 4. Altitude performance characteristics of a tail-pipe burner having a converging conical burner section are presented in this report.
Performance data were obtained with the converging-burner configuration installed on a J47 turbojet engine over a range of simulated flight conditions and tail-pipe fuel-air ratios with a fixed-area exhaust nozzle. The tail-pipe burner was designed for use in a particular airplane. The fuselage dimensions limited the maximum diameter of the burner and necessitated the use of a converging combustion chamber. The burner configuration discussed in this report is a modified version of a tail-pipe burner whose performance is reported in reference 2. This modification resulted from the greater air flow of the J47 turbojet engine used in this investigation as compared with the J35 engine (reference 2). Because of the increased air flow, it was necessary to fabricate a larger burner to reduce the burner-inlet velocity and burner total-pressure loss. The modified burner had the largest diameter that could be used without making major changes in the airplane fuselage. This final configuration incorporated the best flame holder and fuel system previously determined from the investigation reported in reference 2.

Tail-pipe burner performance at several flight conditions is given in both tabular and graphical forms and compared with performance of the standard engine and of the tail-pipe burner reported in reference 2.

APPARATUS AND INSTRUMENTATION

Engine

Tail-pipe burner performance was investigated using a J47 turbojet engine having a static sea-level thrust rating of 5000 pounds at an engine speed of 7900 rpm and a turbine-outlet temperature of 1275°F (1735°C). The air flow is approximately 94 pounds per second at static sea-level rated conditions. The principal components of this engine are a 12-stage axial-flow compressor, eight cylindrical through-flow combustion chambers, a single-stage turbine, and a tail pipe. The over-all length of the engine with standard tail pipe is 143 inches and the maximum diameter is 37 inches. The length of the standard-engine tail pipe is 35 inches. A fixed-area exhaust nozzle with which rated conditions were obtained had an outlet area of 280 square inches.

Installation

The engine and tail-pipe burner were mounted on a wing section that spanned the 20-foot-diameter test section of the Lewis altitude...
wind tunnel (fig. 1). For this investigation the standard-engine tail-pipe was replaced by a tail-pipe burner assembly which was attached to the downstream flange of the turbine casing. In order to provide accessibility for instrumentation, no cowling was installed around the engine.

Air entered the engine inlet through a duct from the tunnel make-up air system. This duct was connected to the engine inlet by means of a slip joint with a frictionless seal so that thrust and drag could be measured by the balance scales.

**Tail-Pipe Burner**

The tail-pipe burner was designed by the manufacturer in conformance with the available space in the airplane. A cross-sectional view showing the dimensions of the tail-pipe burner is shown in figure 2. The three main sections of the tail-pipe burner assembly are the diffuser section, the combustion chamber, and the nozzle section, as shown in figure 3. A fixed conical exhaust nozzle was used for the entire investigation inasmuch as a variable-area nozzle was not available for this burner. Cooling liners, providing a 1/2-inch radial space between the liner and the burner shell and extending the full length of the tail-pipe combustion chamber, were installed in a manner similar to that discussed in reference 1.

The minimum diffuser diameter, which occurred at the downstream end of the diffuser inner body, was 25 inches and the maximum diameter in the combustion chamber was $31\frac{1}{2}$ inches. The diameter of the conical exhaust nozzle was $25\frac{3}{32}$ inches, which gave a fixed area of 494 square inches, and the tail-pipe burner length including the exhaust nozzle was $138\frac{3}{4}$ inches. A typical two-annular V-type flame holder used throughout the investigation is shown in figure 4. The flame holder, which was mounted $2\frac{3}{4}$ inches downstream of the combustion-chamber inlet, had outer and inner ring diameters of 23 inches and 14 inches, respectively, and blocked 31.5 percent of the burner cross-sectional area.

Fuel was injected at two stations into the burner normal to the direction of gas flow (fig. 5). At the upstream station, fuel was injected from 12 tubes $1\frac{5}{16}$ inches long extending from the diffuser inner cone and located about 44 inches upstream of the flame holder.
Each tube had two 0.031-inch-diameter holes. At the downstream station, fuel was injected from two sets of different length radial spray tubes 18\(\frac{1}{2}\) inches upstream of the flame holder. One set consisted of six fuel tubes 8\(\frac{1}{2}\) inches long with six 0.041-inch-diameter holes per tube and the other of six tubes 7\(\frac{5}{16}\) inches long with four 0.031-inch-diameter holes per tube. Rate of fuel injection from the three sets of spray tubes was determined by a flow divider installed in the tail-pipe fuel system as shown in figure 5. The fuel-flow schedule provided by the flow divider is shown in figure 6. At low tail-pipe fuel flows, approximately 60 percent of the tail-pipe fuel was injected through only the primary system which consisted of the six short tubes and the 12 spray tubes on the inner cone ring. The remaining fuel was injected through the six long tubes.

Ignition of the tail-pipe fuel was accomplished by either of two systems. For one ignition system, fuel was injected through a 30-gallon-per-hour spray nozzle installed in the center of the diffuser inner cone. Ignition of this fuel was obtained by two spark plugs inserted in the pilot cone (fig. 3). The other system provided ignition by a momentary increase in fuel flow to the fuel nozzle in one of the engine combustors. This excess fuel in one engine combustor resulted in a burst of flame through the turbine, thereby igniting the tail-pipe fuel. With this system of ignition, the cone fuel also had to be on to carry the flame back to the flame holder.

Instrumentation

Instrumentation was installed at the engine inlet to measure engine air flow and in the tail-pipe to measure the burner performance. Location of the instrumentation in the tail-pipe burner is shown in figure 3, and cross sections of instrumentation at stations 6 and 8 are shown in figure 7. The instrumentation at station 7 consisted of four static wall orifices approximately 9 inches upstream of the flame holder. The exhaust-nozzle rake installed at station 8 was water cooled.

PROCEDURE

Data were obtained over a range of altitudes from 15,000 to 45,000 feet for a range of flight Mach numbers from 0.22 to 0.76. At each flight condition, performance data were obtained for
several tail-pipe fuel flows between the lean combustion blow-out limit and the fuel flow required for limiting turbine-outlet temperature. All tail-pipe burning data were obtained at rated engine speed (7900 rpm).

Dry refrigerated air was supplied to the engine from the tunnel make-up air system at the standard temperature for each flight condition, except that the minimum temperature obtained was about -20°F. The air was throttled from approximately sea-level pressure to the desired total pressure at the engine inlet. Complete freestream ram-pressure recovery was assumed at each flight condition in the calculation of flight Mach number.

Thrust measurements were obtained from the tunnel balance scales and from the tail-rake pressure survey at the exhaust nozzle. The thrust values presented were obtained from the balance-scale measurements. The thrust obtained from tail-rake measurements was used in determining exhaust-gas temperature and combustion efficiency. Methods of obtaining the important tail-pipe burner parameters and the symbols used are given in the appendix.

The fuel used in the engine was AN-F-32 kerosene having a lower heating value of 18,550 Btu per pound and a hydrogen-carbon ratio of 0.155. Fuel used in the tail-pipe burner was AN-F-48b, grade 80, gasoline having a lower heating value of 19,000 Btu per pound and a hydrogen-carbon ratio of 0.186.

RESULTS AND DISCUSSION

As a preliminary phase of this investigation, a smaller diameter tail-pipe burner having a converging burner section similar in shape to the one discussed in this report was investigated on a J47 turbojet engine. Unpublished results showed that tail-pipe burner operation was unsatisfactory at rated engine speed because of the high combustion-chamber-inlet velocities. The combustion-chamber-inlet velocities under these conditions were in the order of 700 feet per second. In order to reduce this velocity, the tail-pipe burner diameters were increased approximately 3 inches. This increase in diameter was the largest that could be used without making major changes to the airplane fuselage. As a result of this enlargement of the burner, the combustion-chamber-inlet velocity was reduced approximately 20 percent. Results of the investigation of this larger burner are presented graphically to show the effect of altitude and flight Mach number on the significant burner-performance characteristics. All data for this investigation are also presented in table I.
Burner Performance Characteristics

Effect of altitude. - The effect of altitude on burner performance is shown in figures 8 to 10 for a flight Mach number of 0.23 and at rated engine speed. The variation of combustion-chamber-inlet velocity and tail-pipe burner-inlet conditions with tail-pipe fuel-air ratio is shown in figure 8. The tail-pipe fuel-air ratio is defined as the ratio of tail-pipe fuel flow to the unburned air flow entering the tail-pipe. For the range of tail-pipe fuel-air ratios, increasing the altitude from 15,000 to 45,000 feet had little effect on the combustion-chamber-inlet velocity and the tail-pipe burner-inlet temperature. The turbine-outlet total pressure was reduced approximately in proportion to the reduction in compressor-inlet pressure. For the entire range of altitudes investigated, the combustion-chamber-inlet velocity varied between 525 and 575 feet per second.

The effect of altitude on the variation of exhaust-gas total temperature and tail-pipe combustion efficiency with tail-pipe fuel-air ratio is shown in figure 9. Maximum combustion efficiency was reduced from 0.86 to 0.80 as the altitude was increased from 15,000 to 35,000 feet. Further increase in altitude greatly reduced the combustion efficiency. The maximum combustion efficiency occurred at tail-pipe fuel-air ratios between approximately 0.040 and 0.050.

The exhaust-nozzle-outlet temperature, over the range of tail-pipe fuel-air ratios investigated, was reduced about 600° R with increase in altitude from 15,000 to 45,000 feet, with the greater portion of this decrease occurring at an altitude above 35,000 feet. This decrease in exhaust-gas temperature is a result of a change in engine characteristics with a change in Reynolds number. With an increase in altitude, the Reynolds number effect on the compressor (reference 5) caused a decrease in compressor efficiency which necessitated increasing the exhaust-nozzle area in order to obtain rated engine conditions at limiting turbine-outlet temperature. Consequently, because the data presented herein are for a fixed-area exhaust nozzle, the available exhaust-gas temperature with afterburning is considerably lower at 45,000 feet than at 15,000 feet. For optimum performance the nozzle area should be adjusted by use of a variable-area nozzle so that the pressures and temperatures at the turbine outlet are the same with the afterburner operative or inoperative.

Variation of the over-all burner performance characteristics with altitude at a tail-pipe burner-inlet temperature of 1700° R
and rated engine speed is shown in figure 10. These results were obtained by cross-plotting and in some cases by slight extrapolation of jet thrust and total fuel flow at a constant tail-pipe burner-inlet temperature of 1700° R.

Thrust-augmentation characteristics are presented in terms of the augmented thrust ratio, which is defined as the ratio of augmented thrust to normal thrust as defined in the appendix. For altitudes up to 25,000 feet, the augmented thrust ratio was 1.30 and then decreased to 1.13 as the altitude increased to 45,000 feet. The total-pressure loss across the standard-engine tail pipe was 0.032 of the tail-pipe burner-inlet total pressure for all flight conditions at rated engine speed. This value was obtained during previous operation of the engine with a fixed-conical exhaust nozzle. With the tail-pipe burner installed but inoperative, this value increased from 0.052 to 0.071 as the altitude was increased from 15,000 to 45,000 feet at a flight Mach number of 0.23. As a result of this greater pressure loss across the tail-pipe burner, the augmented thrust ratio with tail-pipe burner inoperative was 0.98 at 15,000 feet and decreased to 0.96 at 45,000 feet.

The specific fuel consumption of the standard engine based on net thrust increased from 1.15 to 1.25 as the altitude was raised from 15,000 to 45,000 feet. With tail-pipe burning, the specific fuel consumption was 2.95 at 15,000 feet and 3.40 at 45,000 feet. This expected increase in specific fuel consumption with increase in altitude resulted from the lower tail-pipe combustion efficiency at higher altitudes.

Effect of flight Mach number. - The effect of flight Mach number on burner performance is shown in figures 11 to 13 for an altitude of 25,000 feet and rated engine speed. The variation of combustion-chamber-inlet velocity and tail-pipe burner-inlet conditions with tail-pipe fuel-air ratio is shown in figure 11 for several flight Mach numbers. At a given tail-pipe fuel-air ratio, increasing the flight Mach number from 0.23 to 0.76 had no effect on the combustion-chamber-inlet velocity. At tail-pipe burner-inlet temperatures above 1500° R, the combustion-chamber-inlet velocity was about 525 feet per second for all flight Mach numbers investigated. Variations of flight Mach number, over the range investigated, had little effect on tail-pipe burner-inlet temperature. The tail-pipe burner-inlet temperature was approximately 50° R higher at a given tail-pipe fuel-air ratio for a flight Mach number of 0.23 than for the two higher flight Mach numbers investigated. The tail-pipe burner-inlet pressure increased approximately in proportion to the compressor-inlet pressure.
The effect of flight Mach number on variation of exhaust-gas total temperature and tail-pipe combustion efficiency with tail-pipe fuel-air ratio is shown in figure 12. Peak tail-pipe combustion efficiencies of 0.82 at a flight Mach number of 0.23, and 0.84 at flight Mach numbers of 0.54 and 0.76 were obtained at a tail-pipe fuel-air ratio of approximately 0.043. Exhaust-gas total temperature was unaffected by changes in flight Mach number and increased to a maximum value of 3480°C as the tail-pipe fuel-air ratio was raised to 0.0625.

Variation of the over-all burner performance characteristics with flight Mach number at a tail-pipe burner-inlet temperature of 1700°C and at rated engine speed is shown in figure 13. These results were obtained by cross-plotting and extrapolating in a manner similar to that previously mentioned. The augmented thrust ratio increased from 1.29 at a flight Mach number of 0.23 to 1.4 at a flight Mach number of 0.76. This curve indicates that further increases in flight Mach number would have a very small effect on augmented thrust ratio. The tail-pipe total-pressure loss with the tail-pipe burner installed, but inoperative, was 0.058 of the tail-pipe burner-inlet total pressure and the augmented thrust ratio was about 0.98 for all flight Mach numbers investigated.

The specific fuel consumption based on net thrust of the standard engine was approximately 1.2 for all flight Mach numbers investigated. With tail-pipe burning, the specific fuel consumption was 2.95 at a flight Mach number of 0.23. Increasing the Mach number to 0.54 raised this value to 3.30 and a further increase in Mach number to 0.76 lowered the specific fuel consumption to 2.98.

Performance characteristic curves for a converging-conical tail-pipe burner on a J35 turbojet engine are also shown on figure 13. These data are not directly comparative inasmuch as they are for a lower tail-pipe burner-inlet total temperature (1650°C) and a smaller exhaust-nozzle outlet area (296 square inches) than for the data obtained with the J47 engine. The curves are presented, however, to indicate the relative performance of two converging-conical burners on two different turbojet engines. The augmented thrust ratio for the burner on the J35 engine is 10 to 15 percent higher and the specific fuel consumption is about 25 percent lower than for the burner on the J47 engine. The augmented thrust ratio was lower for the burner used on the J47 engine than for the burner used on the J35 engine, probably because of the greater tail-pipe total-pressure loss ratio. At the operating conditions shown in figure 13, the pressure loss was about 0.15 of the tail-pipe burner-inlet total pressure for the burner used on the J47 engine as compared with 0.065 in the J35 tail-pipe burner.
Burner Operational Characteristics

Two methods were used to ignite the tail-pipe fuel. The first method utilized the pilot cone at the downstream end of the diffuser inner body to provide a sheltered region in which to install a fuel nozzle and spark plugs. Fuel was supplied to the ignition region by the nozzle in the pilot cone and by the fuel-spray tubes on the diffuser inner cone. This system gave successful starts only at altitudes below 20,000 feet and reduced engine speed. When fuel was supplied to the nozzle in the pilot cone at rated engine speed, the flame holder appeared very hot.

In the second method employed, fuel was momentarily injected into one of the engine combustors with a resultant burst of flame through the turbine and into the tail-pipe. This method provided satisfactory ignition of the tail-pipe fuel up to an altitude of 45,000 feet at rated engine speed and a flight Mach number of 0.23. Successful burner operation was also possible at these conditions.

The use of an internal cooling liner extending the full length of the combustion chamber having a 1/2-inch gap between the liner and the burner shell provided adequate shell cooling at all flight conditions investigated.

SUMMARY OF RESULTS

As a preliminary phase of this investigation, a smaller diameter tail-pipe burner having a converging burner section similar in shape to the one discussed in this report was investigated, and the results indicated that tail-pipe burner performance was unsatisfactory with combustion-chamber-inlet velocities in the order of 700 feet per second. Satisfactory performance was obtained with an enlarged burner in which the combustion-chamber-inlet velocities were approximately 550 feet per second.

The following results were obtained from an investigation of the enlarged tail-pipe burner having a converging burner section and a fixed-area exhaust nozzle on a J47 turbojet engine in the NACA Lewis altitude wind tunnel:

1. A maximum tail-pipe combustion efficiency of 0.86 was obtained at an altitude of 15,000 feet, a flight Mach number of 0.23, and a tail-pipe fuel-air ratio of 0.040.
2. At an altitude of 25,000 feet, a tail-pipe burner-inlet temperature of 1700° R, rated engine speed, and flight Mach numbers from 0.23 to 0.76, the ratio of augmented thrust to normal thrust increased from 1.3 to 1.4 and the specific fuel consumption based on net thrust of the standard engine varied from 3.0 to 3.3.

3. Tail-pipe burner operation was possible up to an altitude of 45,000 feet at a flight Mach number of 0.23.

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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_J**: jet velocity coefficient, ratio of scale jet thrust to rake jet thrust
- **C_T**: thermal-expansion ratio, ratio of hot exhaust-nozzle area to cold exhaust-nozzle area
- **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
- **f/a**: fuel-air ratio
- **g**: acceleration due to gravity, 32.2 ft/sec^2
- **H**: total enthalpy, Btu/lb
- **h_c**: lower heating value of fuel, Btu/lb
- **M**: Mach number
- **P**: total pressure, lb/sq ft absolute
- **P_{8'}**: total pressure at exhaust-nozzle survey station in standard-engine tail pipe, lb/sq ft absolute
- **p**: static pressure, lb/sq ft absolute
- **R**: gas constant, 53.4 ft-lb/(lb)(^oR)
- **T**: total temperature, ^oR
t  static temperature, °R
V  velocity, ft/sec
W_a  air flow, lb/sec
W_c  compressor leakage air flow, lb/sec
W_f  fuel flow, lb/hr
W_g  gas flow, lb/sec
W_f/F_n  specific fuel consumption based on total fuel flow and net thrust, lb/(hr)(lb thrust)
γ  ratio of specific heats for gases
η_b  combustion efficiency

Subscripts:
a  air
e  engine
f  fuel
i  indicated
j  jet
m  fuel manifold
s  scale
t  tail-pipe burner
x  inlet duct at frictionless slip joint
0  free-stream conditions
1  engine inlet
6  tail-pipe burner inlet (turbine outlet)
tail-pipe combustion-chamber inlet

exhaust nozzle, 1 inch upstream of fixed portion of exhaust-nozzle outlet

exhaust-nozzle outlet

Methods of Calculation

Flight Mach number and airspeed. - Flight Mach number and equivalent airspeed were calculated from ram-pressure ratio by the following equations assuming complete pressure recovery at the engine inlet:

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

(1)

\[ V_0 = M_0 \sqrt{\gamma_1 s R T_1 \left( \frac{P_0}{P_1} \right)^{\gamma_1}} \]  

(2)

Air flow. - Air flow at the engine inlet was determined from pressure and temperature measurements obtained with four survey rakes in the inlet annulus. The following equation was used to calculate air flow:

\[ W_{a,1} = P_1 A_1 \sqrt{\frac{2 \gamma_1 s}{(\gamma_1 - 1) R T_1} \left[ \left( \frac{P_1}{P_1} \right)^{\gamma_1} - 1 \right]} \]  

(3)

Air flow at station 6 was obtained by taking into account the measured air leakage at the last stage of the compressor in the following manner:

\[ W_{a,6} = W_{a,1} - W_c \]  

(4)
Temperatures. - Static temperature was calculated from indicated temperature, using an impact recovery factor of 0.85.

\[
T_t = \frac{T_i}{1 + 0.85 \left[ \frac{P}{P_i} \right]^{\gamma - 1} - 1}
\]  

(5)

Total temperature was calculated by the adiabatic relation between temperatures and pressures.

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

(Engine) \[ W_{g,6} = W_{a,6} + \frac{W_{f,e}}{3600} \]  

(Tail pipe) \[ W_{g,7} = W_{g,8} = W_{a,6} + \frac{W_{f,e} + W_{f,t}}{3600} \]  

(6)

Tail-pipe combustion-chamber-inlet velocity. - Velocity at the combustion-chamber inlet during burning was calculated from the continuity equation using static pressure measured at station 7, approximately 9 inches upstream of the flame holder, and assuming constant total pressure and total temperature from the tail-pipe burner inlet to the combustion-chamber inlet.

\[
v_7 = \frac{W_{g,7}RT_6}{P_7^\gamma} \frac{1}{P_7^\gamma A_7 \left( \frac{W_{g,7}}{P_7} \right)^\gamma} \]  

(7)

Tail-pipe fuel-air ratio. - The tail-pipe fuel-air ratio is defined as the ratio of the weight flow of fuel injected in the tail-pipe burner to the weight flow of unburned air entering the tail-pipe burner from the engine. Weight flow of unburned air was determined by assuming the fuel injected in the engine combustor was completely burned therein.

\[
(f/a)_t = \frac{W_{f,t}}{3600(W_{a,6})} - \frac{W_{f,e}}{0.068}
\]  

(8)

where 0.068 is the stoichiometric fuel-air ratio for the engine fuel.
The total fuel-air ratio for the engine and tail-pipe burner is:

\[
(f/a) = \frac{W_{f,e} + W_{f,t}}{3600 W_{a,e}}
\]  

(9)

Combustion efficiency. - Tail-pipe combustion efficiency was obtained by dividing the enthalpy rise through the tail-pipe burner by the product of the tail-pipe fuel flow and the lower heating value of the tail-pipe fuel, disregarding dissociation of the exhaust gas.

\[
\eta_{b,t} = \frac{3600 W_{g} \Delta H_{t}}{W_{f,t} h_{c,t}}
\]  

(10)

The engine fuel was assumed to be completely burned in the engine; inasmuch as the engine combustion efficiency has been found to be about 98 percent at rated engine speed, this assumption involves an error in tail-pipe combustion efficiency of less than 0.5 percent. The enthalpy of the products of combustion was obtained from a chart using the hydrogen-carbon ratio and the temperature. This method is explained in reference 5.

Augmented thrust. - Scale jet thrust was determined from the balance-scale measurements by use of the following equation:

\[
F_{j,s} = B + D + D_{T} + \frac{W_{a,e} V_{x}}{g} + A_{x} (p_{x} - p_{0})
\]  

(11)

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 and with a blind flange installed in the engine inlet to prevent air flow through the engine. Drag of the exhaust-nozzle survey rake was measured over a range of jet Mach numbers by a hydraulic balance piston mechanism.
Scale net thrust was obtained by subtracting the equivalent free-stream momentum of the inlet air from the scale jet thrust

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

(Rake thrust is given by the following equation based on pressure obtained at station 8):

\[ F_{j,8} = \frac{2C_T A_0 P_8 \gamma_9}{\gamma_9 - 1} \left[ \frac{P_8}{P_0} \right]^{\frac{\gamma_9 - 1}{\gamma_9}} - 1 + A_8 C_T (P_8 - P_0) \]  

The value of \( \gamma_9 \) was obtained from an approximate exhaust-nozzle-outlet temperature calculated from scale thrust.

Normal thrust. - Normal thrust is the theoretical thrust that would be obtained with the engine and the standard-engine tail pipe equipped with a nozzle of a size that gives the same turbine-outlet pressures and temperatures as were encountered with tail-pipe burning under the same flight conditions. Such a definition of normal thrust takes into account engine deterioration. The normal thrust of the engine was calculated from measurements during the tail-pipe burning program of total pressure and temperature at the turbine outlet, gas flow at the turbine outlet, and from previously determined total-pressure loss ratio across the standard-engine tail pipe.

\[ F_{n,e} = \frac{W_{g,6}}{g} C_J \sqrt{\frac{2 \gamma_6}{\gamma_6 - 1} \frac{g R T_6}{P_0} \left[ \left( \frac{P_8'}{P_0} \right)^{\frac{\gamma_6 - 1}{\gamma_6}} - 1 \right]} - \frac{W_{a,1} V_0}{g} \]  

Experimental data indicated that the total-pressure loss through the standard-engine tail pipe from station 6 to station 8 at rated engine speed was nearly 0.032 \( P_8 \); \( P_8' \) is therefore equal to approximately 0.968 \( P_8 \). The jet velocity coefficient \( C_J \) used in equation (14) is shown as a function of tail-pipe pressure ratio:
Augmented thrust ratio is defined as:

\[
\frac{F_{n,s}}{F_{n,e}}
\]

As previously stated, for the normal thrust, \( P_8 \) was approximately equal to 0.968 \( P_6 \). Experimental data also showed that with the tail-pipe burner inoperative, \( P_8 \) was approximately equal to 0.942 \( P_6 \). This thrust ratio was calculated as follows:

\[
\frac{F_{n}^*}{F_{n,e}} = \frac{C_J \frac{W_{g,s}}{g} \sqrt{\frac{276}{\gamma - 1} gRT_6 \left[ 1 - \left( \frac{P_0}{P_6} \right)^{\frac{\gamma - 1}{\gamma}} \right]} - \frac{W_{a,1}}{g} V_0}{C_J \frac{W_{g,s}}{g} \sqrt{\frac{276}{\gamma - 1} gRT_6 \left[ 1 - \left( \frac{P_0}{P_6} \right)^{\frac{\gamma - 1}{\gamma}} \right]} - \frac{W_{a,1}}{g} V_0}
\]

where \( F_{n}^* \) is the thrust with tail-pipe burner inoperative. At all flight Mach numbers, \( T_6 \) was assumed a value of 1700°C. For various flight Mach numbers, \( P_6/P_0 \) and \( V_0 \) were obtained from tail-pipe burning data. The jet velocity coefficients were assumed to be equal for both configurations.
Exhaust-gas total temperature. - The total temperature of the exhaust gas was calculated from the thrust equation and is based on rake thrust.

\[
T_g = \frac{(\gamma - 1) s(F_g, 8)^2}{\left[ \frac{\gamma - 1}{\gamma} \right] 2 \gamma R (W_g, 8)^2 \left[ 1 - \left( \frac{P_0}{P_g} \right)^\frac{\gamma}{\gamma - 1} \right]}
\]

(16)

REFERENCES


<table>
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<th>Flight Mach number $M_0$</th>
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<th>Engine inlet Total pressure $P_{in}$ (lb/ft$^2$)</th>
<th>Engine fuel flow $W_f$ (lb/hr)</th>
<th>Tail-pipe fuel flow $W_{sp}$ (lb/hr)</th>
<th>Tail-pipe fuel distribution</th>
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<th>Secondary system</th>
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<th>Net (lb)</th>
<th>Engine inlet fuel consumption $W_{in}/P_{in}$ (lb/thr)</th>
<th>Specific fuel consumption $W_{in}/P_{ch}$ (lb/hr)</th>
<th>Total tail-pipe fuel ratio $f/f_s$</th>
<th>Total tail-pipe combustion efficiency $\eta$</th>
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<th>Combustion chamber inlet velocity $V_0$ (ft/sec)</th>
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(continued)
Figure 1. - Installation of engine and tail-pipe burner in altitude wind tunnel.
Figure 2. - Cross-sectional view of tail-pipe burner.
Figure 3. - Schematic diagram of tail-pipe burner.
Figure 4. - Typical two-annular V-type flame holder.
Detail A: four holes per tube, 0.031-inch diameter; fuel spray perpendicular to gas flow.

Detail B: two holes per tube, 0.031-inch diameter; fuel spray perpendicular to gas flow.

Detail C: six holes per tube, 0.041-inch diameter; fuel spray perpendicular to gas flow.

Figure 5. - Schematic diagram of tail-pipe fuel system for configuration C.
Figure 6. - Flow divider schedule for tail-pipe burner.
(a) Tail-pipe-burner inlet, station 6, 9\(\frac{3}{4}\) inches downstream of tail-pipe-burner inlet.

Figure 7. - Cross sections of measuring stations showing location of instrumentation. Viewed from upstream.
(b) Exhaust nozzle, station 8, 1 inch upstream of exhaust-nozzle outlet.

Figure 7. Concluded. Cross sections of measuring stations showing location of instrumentation. Viewed from upstream.
Figure 8. — Effect of altitude on variation of combustion-chamber-inlet velocity and tail-pipe-burner-inlet conditions with tail-pipe fuel-air ratio. Engine speed, 7900 rpm; flight Mach number, 0.23.
Figure 9. - Effect of altitude on variation of exhaust-gas temperature and tail-pipe combustion efficiency with tail-pipe fuel-air ratio. Flight Mach number, 0.23; engine speed, 7900 rpm.
Figure 10. - Variation of over-all burner performance characteristics with altitude. Tail-pipe burner-inlet temperature, 1700 °R; Mach number, 0.23; exhaust-nozzle-outlet area, 494 square inches; engine speed, 7900 rpm.
Figure 11. - Effect of flight Mach number on variation of combustion-chamber-inlet velocity and tail-pipe-burner-inlet conditions with tail-pipe fuel-air ratio. Altitude, 25,000 feet; engine speed, 7900 rpm.
Figure 12. - Effect of flight Mach number on variation of exhaust-gas temperature and tail-pipe combustion efficiency with tail-pipe fuel-air ratio. Altitude, 25,000 feet; engine speed, 7900 rpm.
Figure 13. - Variation of over-all burner performance characteristics with flight Mach number. Altitude, 25,000 feet; engine speed, 7900 rpm.