RESEARCH MEMORANDUM

ALTITUDE PERFORMANCE INVESTIGATION OF TWO FLAME-HOLDER
AND FUEL-SYSTEM CONFIGURATIONS IN SHORT AFTERBURNER

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Page 33, figure 10: The curves are reversed. The solid curve should be dashed and the dashed curve should be solid.

Page 38, figure 15: The phrase "at rated turbine-outlet temperature" should be added to the legend.
SUMMARY

During an investigation in an altitude chamber, the altitude performance characteristics of two dissimilar flame-holder and fuel-system configurations were evaluated in a short afterburner. One configuration had a double annular V-gutter flame holder with fuel injectors located several inches upstream. The other had a double annular H-gutter flame holder and an additional annular V-gutter located a few inches downstream with the fuel injectors located immediately upstream of the H-gutter flame holder.

The altitude operating limits and the altitude performance of the V-gutter configuration were superior to those of the H-gutter configuration. At a Mach number of 0.6, the altitude limit of the V-gutter configuration was 50,000 feet as compared with 42,000 feet for the H-gutter configuration. The combustion efficiency was appreciably better for the V-gutter configuration; for example, at an altitude of 30,000 feet, the combustion efficiency at limiting turbine-outlet temperature was 74 percent for the V-gutter configuration as compared with 61 percent for the H-gutter configuration. As soon as fuel was introduced into the burner, spontaneous ignition was obtained with the V-gutter configuration to the maximum altitude at which ignition was attempted, 45,000 feet. Ignition of the H-gutter configuration required the use of a torch igniter.

INTRODUCTION

An investigation was conducted in an altitude chamber at the NACA Lewis laboratory to obtain a flame-holder and fuel-system configuration which would operate at high altitude in a short afterburner on an axial-flow turbojet engine. The afterburner shell used for this investigation was a production model furnished by the engine manufacturer.
The first part of the investigation was a brief evaluation of several types of flame-holder and fuel-injection systems (reference 1) to obtain an afterburner configuration that would meet the engine manufacturer's requirements for efficient operation at altitudes to at least 40,000 feet.

The configurations investigated could be separated into two distinct types: One type had a double-annular H-gutter type flame holder with a V-type gutter mounted several inches downstream. The fuel-injection system was mounted immediately upstream of the H-gutter flame holder and, therefore, did not provide a practical mixing length for evaporation of fuel and mixing of fuel and air. The other type consisted of a double-annular V-gutter type flame holder with the fuel-injection manifold located several inches upstream to provide a mixing length to evaporate the fuel and obtain additional mixing of fuel and air.

For the second phase of the investigation, one of the most promising configurations of each type was selected for further evaluation over a range of flight Mach numbers as well as altitude. In this phase of the program, the requirements were that satisfactory operation be obtained to an altitude of at least 50,000 feet.

The results of this complete evaluation of the altitude performance and operational characteristics of the two types of flame-holder and fuel-system configurations are reported herein. Operational limits of each configuration were determined for a wide range of simulated flight conditions and the performance characteristics were obtained at various altitudes for the complete operable range of afterburner fuel-air ratios. Comparative data are presented to show the performance variations with altitude at a flight Mach number of 0.6. Data are also presented to show the performance variation with flight Mach number of each configuration at the highest altitude where a reasonably wide range of afterburner fuel-air ratio could be obtained. The starting limits of both configurations at a flight Mach number of 0.6 are also discussed.

APPARATUS

Engine

An axial-flow-type turbojet engine with an afterburner was used in this investigation. With the afterburner inoperative, the engine has a static sea-level dry-thrust rating of 5100 pounds at rated engine speed, 7900 rpm, and at a turbine-outlet temperature of 1300°F. At this operating condition, the air flow is approximately 86 pounds per second and the fuel flow is 5740 pounds per hour. The over-all length
of the engine is approximately 195 inches and the maximum diameter is 43 inches. The main components of the engine are an 11-stage axial-flow compressor, eight cylindrical through-flow combustors, a single-stage turbine, and an afterburner.

Installation

The engine was installed in an altitude chamber as shown in figure 1. The altitude chamber is 10 feet in diameter and 60 feet long. A honeycomb is installed in the chamber upstream of the test section to provide smooth flow of the inlet air. The forward baffle separates the inlet air from the exhaust and provides a means of maintaining a pressure difference across the engine. A 14-inch butterfly valve, installed in the forward baffle, was used to provide cooling air for the engine compartment. The rear baffle acts as a radiation shield and prevents the recirculation of exhaust gases about the engine. The exhaust gas from the jet nozzle was discharged into an exhaust diffuser. The pressure rise in this diffuser assisted in simulating an altitude pressure in the test section. Combustion in the afterburner was observed through a periscope located in the exhaust duct behind the engine.

Afterburner

A drawing of the afterburner assembly including the inlet diffuser is shown in figure 2 with a typical flame-holder and fuel-injection system installed. The afterburner had an inlet diameter of 31 inches and an over-all length including the variable-area exhaust nozzle of 47\frac{1}{2} inches, giving a length-diameter ratio of only 1.53. The variable-area nozzle was a two-position clamshell-type nozzle. During afterburning, the exhaust nozzle was in the open position and the area was approximately 357 square inches. In the closed position, the exhaust-nozzle area was adjusted to give rated turbine-outlet temperature at rated engine speed. Fuel was supplied to the afterburner by an air-turbine fuel pump which was driven by air bled from the compressor outlet.

Cooling of the burner section was accomplished by an ejector cooling shroud. The exhaust jet discharging through this shroud induced a flow of cooling air over the burner shell. The air entered the cooling shroud from the test section of the altitude chamber at approximately the simulated altitude ambient pressure and at a temperature of approximately 100\degree F.
Configurations

The installations of the two configurations used in this investigation are shown in figure 3. The H-gutter flame holder with a trailing V-gutter and the close-coupled fuel system are designated as configuration A. The V-gutter flame holder with the upstream fuel system is designated configuration B.

Configuration A. - The flame holder of configuration A consisted of two annular H-sections connected by eight radial H sections with a trailing V-section. Critical dimensions of the flame holder are shown in figure 3(a). Orifices in the cross member of the H sections metered fuel and air to the sheltered zone of the gutters. The fuel-injection manifold consisted of three concentric rings connected by four radial tubes. Also, the two outer rings were connected by 12 additional tubes. Fuel orifices were located in the rings and connecting tubes to provide injection in an upstream direction but at an angle to the gas flow. The location of the fuel orifices is shown in figure 4(a).

Configuration B. - The flame holder consisted of two annular, staggered V-sections with six radial interconnecting V-sections. Dimensions of the flame holder are shown in figure 3(b). The fuel-injection manifold consisted of 12 radial tubes connected by an outer and an inner ring. Fuel orifices were located in the radial tubes only and placed to provide a uniform radial distribution of fuel (see fig. 4(b)).

Ignition system. - The same afterburner ignition system was provided for both configurations. Ignition was provided by a momentary increase in fuel flow to one of the engine combustors (see reference 2). This excess fuel in one combustor caused a burst of flame through the turbine and ignition in the afterburner.

Instrumentation

Pressures and temperatures were measured at stations throughout the engine and afterburner (fig. 1(b)). Compressor air flow was determined by the use of survey rakes mounted at the engine inlet, station 1. Instrumentation was installed for measuring both the engine midframe air-bleed and the air bled from the compressor outlet that was used to drive the air turbine of the afterburner fuel pump. These air flows were subtracted from the compressor air flow to obtain the afterburner air flow. Afterburner-inlet total pressure and temperature were determined from a survey at the turbine outlet (station 5, fig. 5(a)). Static-pressure measurements were obtained with three wall orifices located at the burner inlet (station 6, fig. 1(b)). Total pressures were measured at the exhaust-nozzle inlet with a water-cooled survey
rake (station 7, fig. 5(b)). Ambient pressure in the region of the exhaust-nozzle outlet was determined by static probes in the plane of the nozzle exit (station 8) and altitude pressure by static probes in the plane of the shroud exit (station 0). Engine and afterburner fuel flows were measured by calibrated rotameters.

Procedure

The altitude operational limits of each configuration were determined for a range of flight Mach numbers from 0.4 to 1.0. The minimum fuel flow at each flight condition was determined by imminent blow-out and the maximum fuel flow was determined by either rated turbine-outlet temperature or rich blow-out. The maximum operable altitude was determined by increasing altitude and holding the fuel flow and the flight Mach number constant until blow-out occurred.

Performance data were obtained for altitudes from 10,000 to 45,000 feet for a flight Mach number of 0.6 with both configurations. Also, performance data were obtained for flight Mach numbers of 0.4 to 1.0 at an altitude of 30,000 feet with configuration A and at an altitude of 40,000 feet with configuration B. Performance data at each flight condition were obtained at several afterburner fuel flows at rated engine speed.

Starting data were obtained at a flight Mach number of 0.6. The starting technique consisted in supplying fuel to the afterburner with the exhaust nozzle in the closed position. If autoignition was obtained, the jet nozzle was opened to the afterburning position. If autoignition was not obtained, the torch fuel flow was turned on for a period not exceeding 2 seconds. When ignition took place, the jet nozzle was quickly opened and the torch fuel flow was shut off. Afterburner ignition was investigated over the operable fuel-air ratio range of the afterburner by presetting fuel flows to those corresponding to the steady-state afterburner operation.

The engine-inlet air for each flight condition was supplied at the total temperature and total pressure corresponding to NACA standard atmospheric conditions; 100-percent ram-pressure recovery at the engine inlet was assumed. The symbols used in this report and the methods used to compute the performance parameters are presented in the appendix.

Two fuel-air ratios are defined and used in computing and presenting the results of the investigation:

(1) The afterburner fuel-air ratio \((f/a)_t\) is defined as the ratio of the afterburner fuel flow to the engine air flow (air flow
entering compressor minus air bled from compressor). This fuel-air ratio was used when only the flight condition, the engine speed, and the afterburner fuel flow were recorded. The air flow values were taken from the engine air-flow calibration curves.

(2) The unburned-air afterburner fuel-air ratio \( (f/a)_{ua} \) is defined as the ratio of the afterburner fuel flow to the unburned air flow entering the afterburner (engine air flow minus the air burned in the engine). This fuel-air ratio was used when complete performance data were recorded.

The fuel used in this investigation was MIL-F-5624A, grade JP-3, having a lower heating value of 18,680 Btu per pound and a hydrogen-carbon ratio of 0.172.

RESULTS AND DISCUSSION

Operational Limits

The afterburner operable range of fuel-air ratios is shown for both configurations in figure 6 as a function of altitude for flight Mach numbers from 0.4 to 1.0. For both configurations at a flight Mach number of 0.6, minimum and maximum fuel-air ratios increased with altitude resulting in no great change in operable range of fuel-air ratio except near the altitude limits where they converged. The data indicate that the trends would be the same for other flight Mach numbers. During the determination of the maximum altitude limits, complete data were not obtained and it was quite possible to exceed the rated turbine-outlet temperature. The rated turbine-outlet-temperature lines were based on subsequent performance data.

As the fuel-air ratio increased near limiting altitudes for configuration A, a rich blow-out occurred before rated turbine-outlet temperature was reached (fig. 6(a)). Conversely, with configuration B, the maximum fuel-air-ratio operable limit was generally established not by blow-out but when the maximum allowable turbine-outlet temperature was reached (fig. 6(b)).

The effect of increasing flight Mach number on the operational limits of the two configurations is shown in figure 6 as a general upward shift in the altitude operational limit. The variation of altitude limit with increasing flight Mach number is shown in figure 7(a). The maximum operable altitude of both configurations increased at about the same rate with flight Mach number but that of configuration B was 6000 to 9000 feet higher than that of configuration A. At a flight Mach number of 0.6, the altitude limits were about 42,000 and 50,000 feet for configurations A and B, respectively. The variation of
Fuel-air-ratio limit with increasing flight Mach number at an altitude of 40,000 feet is shown in figure 7(b). Configuration A would not operate below a flight Mach number of 0.5, and the operable range of fuel-air ratio, which was limited by blow-out, increased with flight Mach number. The operable range of fuel-air ratio for configuration B was about the same over the range of flight Mach numbers investigated. The decrease in fuel-air ratio of configuration B with increasing flight Mach number is a result of increasing combustion efficiency. Also, the lower operating fuel-air ratios for configuration B result from higher combustion efficiency. The leaner fuel-air ratios and higher operating limits of configuration B indicate the combined advantage of uniform circumferential fuel distribution, greater mixing length between fuel injection station and flame holders, and V-gutter instead of H-gutter flame holders.

Performance Characteristics

Effect of altitude. - The performance data are presented in table I and are shown graphically in figures 8 to 12 for a flight Mach number of 0.6, rated engine speed, and several altitudes. The variation in afterburner-inlet conditions with afterburner fuel-air ratio is presented in figure 8. The fuel-air ratio used here is based on the unburned air available at the afterburner inlet as defined in the Procedure section. The turbine-outlet temperature and turbine-outlet pressure increased with increasing fuel-air ratio, as expected for operation with a constant-area jet nozzle. At a given fuel-air ratio, the turbine-outlet temperature and the turbine-outlet pressure decreased with increasing altitude approximately in proportion to the decrease in engine-inlet temperature and pressure. The burner-inlet velocity varied only slightly with fuel-air ratio, decreasing from about 400 feet per second to about 385 feet per second from the minimum to maximum operable fuel-air ratio.

The variation of exhaust-gas total temperature and afterburner combustion efficiency with afterburner fuel-air ratio is shown in figure 9 for the altitudes investigated. At a given fuel-air ratio, the exhaust-gas total temperature decreased with increasing altitude primarily because of a decrease in afterburner combustion efficiency (fig. 9(b)) and to a lesser extent because of decreased turbine-outlet temperatures (fig. 8(b)).

The rich fuel-air-ratio limit of operation, as determined by limiting turbine-outlet temperature (maximum thrust), occurred near the peak of the afterburner-efficiency curves at an altitude of 10,000 feet. With increasing altitude, this rich limit for both configurations occurred at fuel-air ratios progressively greater than that for peak combustion efficiency. The combustion efficiencies at limiting
turbine-outlet temperature for configuration A were 81 percent at 10,000 feet and 61 percent at 30,000 feet. The corresponding efficiencies of configuration B were 88 and 74 percent, respectively. The effect of altitude on combustion efficiency is shown graphically in figure 10 for a flight Mach number of 0.6. Although both configurations show a rapid decrease in combustion efficiency with increasing altitude, configuration B had a 7 to 13 percent higher combustion efficiency than configuration A.

Flame holders such as configuration B with uniform fuel distribution were capable of operation at high altitude without decrease in maximum burner efficiency (reference 1). The unexpected drop in maximum efficiency at altitude for configuration B was probably due to a less uniform fuel distribution than was obtained with better configurations in reference 1. Because the fuel was injected 45° upstream instead of normal to the stream, there was less tendency to distribute fuel completely across the stream, especially at the lower fuel flows corresponding to the higher altitudes.

The variation in net thrust and net thrust specific fuel consumption with altitude and fuel-air ratio is shown in figure 11 for both configurations. A cross plot of the net thrust at rated turbine-outlet temperatures is presented in figure 12(a) and is, of course, essentially equal for the two configurations. Because the rated turbine-outlet temperatures occurred at lower fuel-air ratio for configuration B than for configuration A, the net thrust specific fuel consumption (fig. 12(b)) was lower for configuration B. At an altitude of 30,000 feet, the net thrust specific fuel consumption for configuration A was 2.62 whereas that of configuration B was 2.33. As altitude increased, an even greater advantage of configuration B was evident.

Effect of flight Mach number. - The variation of afterburner performance at rated engine speed, with flight Mach number and fuel-air ratio, is shown in figures 13 to 17. Performance of configuration B was investigated at an altitude of 40,000 feet and flight Mach numbers of 0.4, 0.6, 0.8, and 1.0. Later when the performance of configuration A was investigated, it was found that operation of the afterburner could not be obtained at any fuel-air ratio at a flight Mach number of 0.4 at 40,000-foot altitude and operation at a flight Mach number of 0.6 was possible only over a very limited range of fuel-air ratios (fig. 6). The performance calibration of configuration B was therefore conducted at 30,000 feet altitude. The data for the two configurations then represent the altitudes at which performance at flight Mach numbers from 0.4 to 1.0 could be obtained over a range of fuel-air ratios with stable combustion.
The variation in afterburner-inlet conditions with flight Mach number and fuel-air ratio is presented in figure 13. The afterburner-inlet velocity (fig. 13(a)) is relatively unaffected by flight Mach number but shows the same small decrease with increasing fuel-air ratio previously discussed. The turbine-outlet gas temperature (fig. 13(b)) and the turbine-outlet pressure (fig. 13(c)) increased with increasing flight Mach number. This increase in turbine-outlet gas temperature and pressure is not quite proportional to the increase in engine-inlet temperature and pressure with increasing flight Mach number because of a secondary effect due to compressor Reynolds number.

The afterburner combustion efficiency and exhaust-gas temperature are shown in figure 14. The increase in exhaust-gas temperature (fig. 14(a)) with increasing flight Mach number results primarily from the increased combustion efficiency (fig. 14(b)). The combustion efficiency rises with increasing flight Mach number because of the increased burner-inlet pressure and temperature (fig. 13). For both configurations, the maximum combustion efficiency for each flight Mach number investigated occurred at a fuel-air ratio of about 0.03. The effect of flight Mach number on afterburner efficiency at rated turbine-outlet temperature is shown in figure 15 (a cross plot of fig. 14(b)). At flight Mach numbers above 0.7, the afterburner efficiency of configuration B at 40,000 feet was about the same as the afterburner efficiency of configuration A at 30,000 feet. Below a flight Mach number of 0.7, the sensitivity of configuration A to flight Mach number was far more pronounced than that of configuration B even with the lower altitude advantage.

The variation of over-all engine performance is presented in figure 16 as a function of afterburner fuel-air ratio for several flight Mach numbers. Net thrust increased (fig. 16(a)) and net thrust specific fuel consumption (fig. 16(b)) decreased slightly with increasing flight Mach number. The net thrust increased with increasing flight Mach number primarily because of the accompanying higher air flows. The lowered specific fuel consumption with increasing flight Mach number resulted primarily from the higher cycle and combustion efficiencies. Figure 17 is a cross plot of the net thrust specific fuel consumption at limiting turbine-outlet temperature. Although configuration B was operated at the higher altitude (40,000 ft), it provided the lower net thrust specific fuel consumption throughout the range of flight Mach numbers investigated.

Ignition Characteristics

As previously discussed, two methods of ignition, (1) autoignition and (2) torch ignition, were used to initiate combustion in the afterburner. The range of altitudes and afterburner fuel-air ratios (f/a)ₜ
over which ignition was obtained is shown in figure 18 for a flight Mach number of 0.6. The dashed curves are the operational limits of the two burner configurations from figure 6. Because the fuel flow was measured at the moment ignition occurred, before fuel flow had stabilized, accurate fuel-air ratios cannot be expected for the starting data. Autoignition was obtained with configuration B throughout the operational range of fuel-air ratios to an altitude of 45,000 feet, the highest altitude at which afterburner ignition was attempted. Autoignition could not be obtained with configuration A possibly because of the cooling of the flame holder by the impingement of unvaporized fuel from the closely coupled fuel manifold. With configuration A, the afterburner could be ignited by the torch method throughout the operable fuel-air-ratio range to an altitude of at least 30,000 feet. Two out of three attempts to obtain ignition were successful at a lean fuel-air ratio at 35,000 feet.

CONCLUDING REMARKS

An investigation of the altitude performance of two flame-holder and fuel-system configurations in a short afterburner has been conducted on an axial-flow turbojet engine in an altitude chamber. One configuration consisted of a double annular H-gutter flame holder having an additional V-gutter several inches downstream. The fuel injection manifold was located directly upstream of the H-gutter flame holder. The other configuration had a double annular V-gutter flame holder with the fuel-injection manifold mounted several inches upstream. This arrangement allowed more time for vaporization of the fuel and mixing of the fuel and air. The double annular V-gutter configuration provided the better altitude performance and operating characteristics. At a flight Mach number of 0.6, the altitude limits were about 42,000 and 50,000 feet for the H-gutter and V-gutter afterburner configurations, respectively. The respective combustion efficiencies of the two configurations at limiting turbine-outlet temperature and at an altitude of 30,000 feet and a flight Mach number of 0.6 were 61 and 74 percent. At these operating conditions, the improved combustion efficiency reduced the afterburner net thrust specific fuel consumption from 2.62 to 2.33 pounds of fuel per pound of net thrust.

Ignition of the V-gutter configuration afterburner occurred spontaneously when the fuel was introduced into the afterburner. This autoignition was possible to an altitude of 45,000 feet, the highest altitude at which ignition was attempted. Ignition of the H-gutter configuration, however, was not spontaneous and the use of a torch igniter was required.
The superiority of the V-gutter configuration in the short afterburner is attributed to (1) improved circumferential fuel distribution, (2) increased fuel mixing and vaporization time, and (3) use of the V-gutter flame holder with its superior flame-holding characteristics.

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APPENDIX - METHODS OF CALCULATION

Symbols

The following symbols are used in this report:

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td>area, sq ft</td>
</tr>
<tr>
<td>Cd</td>
<td>flow (discharge) coefficient, ratio of effective flow area to measured area</td>
</tr>
<tr>
<td>CT</td>
<td>thermal expansion ratio, ratio of hot exhaust-nozzle-throat area to cold exhaust-nozzle-throat area</td>
</tr>
<tr>
<td>Cv,e</td>
<td>effective velocity coefficient, ratio of actual jet thrust to calculated jet thrust</td>
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<tr>
<td>F</td>
<td>thrust, lb</td>
</tr>
<tr>
<td>f/a</td>
<td>fuel-air ratio</td>
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<td>p</td>
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<td>gas constant, 53.3 ft-lb/(lb)(°R)</td>
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<tr>
<td>Wa</td>
<td>air flow, lb/sec</td>
</tr>
<tr>
<td>Wf</td>
<td>fuel flow, lb/hr</td>
</tr>
</tbody>
</table>
\( W_g \)  
 gas flow, lb/sec

\( \gamma \)  
 ratio of specific heats

\( \eta \)  
 combustion efficiency

Subscripts:

\( c \)  
 calculated

\( e \)  
 engine

\( i \)  
 indicated

\( j \)  
 jet

\( n \)  
 net

\( t \)  
 afterburner

\( u_a \)  
 unburned air

\( 0 \)  
 free-stream ambient conditions

\( 1 \)  
 engine inlet

\( 3 \)  
 compressor outlet

\( 5 \)  
 turbine outlet (diffuser inlet)

\( 6 \)  
 afterburner inlet

\( 7 \)  
 exhaust-nozzle inlet

\( 8 \)  
 exhaust-nozzle throat

Methods of Calculation

Flight speed and Mach number. - The simulated flight speed and Mach number at which the engine and afterburner were operated were determined from the following equations:

\[
V_0 = \sqrt{\frac{gRT_1}{\gamma_1 - 1} \left[ 1 - \left( \frac{P_0}{P_1} \right)^{\frac{\gamma_1 - 1}{\gamma_1}} \right]} \quad (1)
\]
\[ M_0 = \sqrt{\frac{2}{\gamma - 1} \left[ \frac{\gamma - 1}{\frac{P_1}{P_0}} \right] - 1} \]  

where \( \gamma \) was assumed to be 1.4 and the total temperature was assumed to be equal to the indicated temperature inasmuch as the thermocouple recovery factor was 0.96.

**Gas flow.** - The compressor-inlet air flow was computed as

\[ W_{a,3} = A_1 P_1 \sqrt{\frac{g}{R T_1}} \sqrt{\frac{2\gamma_1}{\gamma - 1} \left[ \frac{\gamma - 1}{\frac{P_1}{P_1}} \right] - 1} \]  

The engine air flow at station 3 was calculated by subtracting the midframe air-bleed and the air flow required to drive the afterburner fuel pump from the compressor inlet air flow. The midframe air-bleed and afterburner fuel-pump air flow were calculated in a similar manner to the compressor-inlet air flow. The total gas flow at the turbine outlet was calculated as

\[ W_{g,5} = W_{a,3} + \frac{W_{f,e}}{3600} \]  

The total gas flow at the exhaust-nozzle throat was calculated as

\[ W_{g,8} = W_{g,5} + \frac{W_{f,t}}{3600} \]  

**Turbine-outlet total temperature.** - The turbine-outlet total temperature was corrected for impact.

\[ T_5 = \frac{T_{5,i} \left( \frac{P_5}{P_5} \right)}{1 + 0.95 \left[ \frac{\gamma - 1}{\gamma} \right] - 1} \]  

The value 0.95 is the thermocouple recovery factor.
Afterburner inlet velocity. - The continuity equation was used to calculate the afterburner inlet velocity. The static pressure and area were measured at station 5. The total-pressure and -temperature measurements from station 5 were used and it was assumed that there was no loss between the two stations.

\[ V_6 = \frac{W_{e,5} \cdot R T_5}{A_6 P_6} \left( \frac{P_6}{P_5} \right)^{γ_6-1} \]  

(7)

Afterburner fuel-air ratio. - Two afterburner fuel-air ratios are used in this report and are defined as follows:

1. The ratio of the afterburner fuel flow to engine air flow,

\[ \left( \frac{f}{a} \right)_t = \frac{W_{f,t}}{3600 W_{a,3}} \]  

(8)

2. The ratio of the afterburner fuel flow to the unburned air entering the afterburner,

\[ \left( \frac{f}{a} \right)_{ua} = \frac{W_{f,t}}{3600 W_{a,3} - \frac{W_{f,e}}{0.0672}} \]  

(9)

The assumption used in obtaining this equation was that the fuel injected in the engine was completely burned. The value of 0.0672 is the stoichiometric fuel-air ratio for the fuel used.

Exhaust-gas total temperature. - The exhaust-gas total temperature was determined by

\[ T_8 = \frac{2g}{R} \left( \frac{A_8 C_d C_T P_8}{W_{e,8}} \right)^2 \left( \frac{γ_8-1}{γ_8} \right) \left( \frac{P_7}{P_8} \right)^{γ_8-1} \left( \frac{P_7}{P_8} \right)^{-1} \]  

(10)

The flow coefficient \( C_d \) was obtained from reference 3. The exhaust-nozzle-throat area \( A_8 \) was measured at room temperature. Values of the thermal expansion ratio \( C_T \) of the exhaust nozzle were determined from the thermal expansion coefficient for the exhaust-nozzle material and the measured skin temperature.
Exhaust-nozzle-throat static pressure \( p_8 \) was determined as follows:

\[
p_8 = \begin{cases} 
p_0 & \text{for subsonic flow} \\
p_7 \left( \frac{2}{\gamma_8+1} \right)^{\gamma_8-1} & \text{for sonic flow} 
\end{cases}
\]

Afterburner combustion efficiency. - The afterburner combustion efficiency was calculated from

\[
\eta_t = \frac{h_a \left[ \left( \frac{r}{a} \right)_e + \left( \frac{r}{a} \right)_t \right]_{T_r}^{\gamma + 1} - \eta_e \left( \frac{r}{a} \right)_e h_c}{h_c \left[ \left( \frac{r}{a} \right)_t + \left( \frac{r}{a} \right)_e (1 - \eta_e) \right]}
\]

(11)
The engine fuel was not assumed to be burned completely in the engine. The unburned engine fuel was charged to the afterburner. The engine combustion efficiency was 0.96; this value was obtained from an altitude calibration of a similar engine. The term $\frac{Am+B}{m+1}$ accounts for the difference between the enthalpy of the carbon dioxide and water vapor in the burned mixture and the enthalpy of the oxygen removed from the air by their formation (reference 4). Dissociation was not considered inasmuch as its effect is negligible for temperatures to $3200^\circ R$.

**Thrust.** - The jet thrust was calculated from

$$F_j = C_v,e \left[ \frac{W_{g,8}}{g} \sqrt{RT_8} \frac{2\gamma_8}{\gamma_8 - 1} \left[ 1 - \left( \frac{P_8}{P_7} \right)^{\frac{\gamma_8 - 1}{\gamma_8}} \right] + A_8C_T \left( P_8 - P_0 \right) \right]$$

The values of $P_8$ and $C_T$ are explained in the discussion of equation (10). The effective velocity coefficient $C_v,e$ was obtained from reference 1.

Net thrust was obtained from the jet thrust by

$$F_n = F_j - \frac{W_{a,1}}{g} V_0$$

**Net thrust specific fuel consumption.** - The net thrust specific fuel consumption was calculated from

$$\frac{W_f}{F_n} = \frac{W_{f,e} + W_{f,t}}{F_n}$$
REFERENCES


TABLE I - PERFORMANCE

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<tr>
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<th>Engine speed (rpm)</th>
<th>Afterburner fuel flow $W_{f,b}$ (lb/hr)</th>
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TABLE III - PERFORMANCE

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### Configuration A

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<th>Engine Inlet</th>
<th>Turbine Inlet</th>
<th>Exhaust Inlet</th>
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#### Data with Afterburning

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<th>Afterburner Combustion Efficiency</th>
<th>Engine Inlet Total Pressure</th>
<th>Turbine Inlet Total Pressure</th>
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<td>Tt (%)</td>
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### Configuration B

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<th>Engine Inlet</th>
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<td>Pt (lb/sq ft)</td>
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(a) Photograph of installation.

Figure 1. - Engine and afterburner installation in altitude chamber.
(b) Schematic diagram of installation.

Figure 1. - Concluded. Engine and afterburner installation in altitude chamber.
Figure 2. - Inlet diffuser and afterburner assembly. (All dimensions in inches.)
(a) Configuration A.

Figure 3. - Installation of fuel-system and flame-holder configurations. (All dimensions in inches.)
(b) Configuration B.

Figure 3 - Concluded. Installation of fuel-system and flame-holder configurations. (All dimensions in inches.)
(a) Configuration A; diameter of orifices, 0.025 inch; number of orifices, 226.

(b) Configuration B; diameter of orifices, 0.030 inch; number of orifices, 108.

Figure 4. - Location of fuel orifices in manifolds.
(a) Turbine outlet (diffuser inlet), station 5, \(4\frac{1}{2}\) inches downstream of turbine flange.

(b) Exhaust-nozzle inlet, station 7, 5 inches upstream of outlet.

Figure 5. - Location of pressure and temperature instrumentation installed in the afterburner; looking downstream.
Figure 6. Operable range of afterburner configurations at several flight Mach numbers.

(a) Configuration A.
(b) Configuration B.
Figure 7. - Effect of flight Mach number on operational limits.

(a) Effect of flight Mach number on altitude limit.
(b) Effect of flight Mach number on afterburner fuel-air ratio at altitude of 40,000 feet.
Figure 8. - Variation of afterburner inlet conditions with fuel-air ratio at several altitudes. Flight Mach number, 0.6.
Figure 9. - Variation of exhaust-gas total temperature and afterburner combustion efficiency with fuel-air ratio at several altitudes. Flight Mach number, 0.6.
Figure 10. - Effect of altitude on combustion efficiency at rated turbine-outlet temperature. Flight Mach number, 0.6.
Figure 11. - Variation of over-all afterburner performance with afterburner fuel-air ratio at several altitudes. Flight Mach number, 0.6.
Figure 12. - Variation of over-all afterburner performance with altitude at rated turbine-outlet total temperature. Flight Mach number, 0.6.
Figure 13. - Variation of afterburner inlet conditions with afterburner fuel-air ratio at several flight Mach numbers.
Figure 14. - Variation of exhaust-gas total temperature and afterburner combustion efficiency with afterburner fuel-air ratio at several flight Mach numbers.
Figure 15. - Effect of flight Mach number on afterburner combustion efficiency. (Cross plot of fig. 14(b).)
Figure 16. - Variation of overall afterburner performance with afterburner fuel-air ratio at several flight Mach numbers.
Figure 17. - Effect of flight Mach number on net thrust specific fuel consumption at rated turbine-outlet total temperature and engine speed.
Figure 18. - Starting range of afterburner configurations. Flight Mach number, 0.6.

(a) Configuration A, H- and V-gutters.
(b) Configuration B, V-gutter.