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RESEARCH MEMORANDUM

COMBUSTION PERFORMANCE OF TWO EXPERIMENTAL TURBOJET ANNULAR COMBUSTORS AT CONDITIONS SIMULATING HIGH-ALTITUDE SUPERSONIC FLIGHT

By Eugene V. Zettle, Carl T. Norgren, and Herman Mark

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NATIONAL ADVISORY COMMITTEE
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SUMMARY

The performance of two experimental annular turbojet combustors was investigated at operating conditions typical of high-altitude supersonic flight. Each combustor consisted of a one-quarter sector of a single annular combustor designed to fit in a housing with an outside diameter of $25\frac{1}{2}$ inches, an inside diameter of $10\frac{5}{8}$ inches, and a combustor length of approximately 23 inches. Liquid fuel was injected into the combustion chamber from the upstream face of the combustor; in addition, a fuel-staging technique was investigated.

Combustion efficiencies near 100 percent were achieved in both experimental combustors operating with combustor reference velocities of 200 feet per second and greater at simulated supersonic flight conditions. These high efficiencies were maintained at the highest combustor-outlet temperatures investigated, namely, 1800° F for one combustor and 2000° F for the other. Reasonably flat outlet-temperature profiles were obtained and were considered satisfactory. The maximum total-pressure losses were 10.2 and 12.6 percent for the two combustors at a velocity of 165 feet per second and a temperature ratio of about 1.7. For combustor pressure losses of this magnitude, calculations indicated that the increase in engine specific-fuel consumption resulting from combustor pressure losses would be no greater in the engine for supersonic propulsion than in current turbojet engines. These pressure losses therefore appear acceptable for the supersonic flight conditions. Combustor-liner durability and carbon-deposition characteristics of the combustors were not evaluated in this investigation.

INTRODUCTION

Research on compressor and turbine aerodynamics has indicated that increases in air flow per unit frontal area of as much as 30 percent are possible for these components of the turbojet engine (refs. 1 to 4).

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In addition, the advancement of turbine-cooling techniques indicates that increases in operating temperatures of as much as 500° F over current practice are possible in future engines (refs. 5 and 6). High supersonic flight speeds with turbojet engines can be more easily realized with the greater power resulting from higher air flows and temperatures. The turbojet combustor designed for use in an engine incorporating these advancements and powering an aircraft at high supersonic speeds (Mach numbers of 2.0 to 3.0) will be required to operate with much higher air flows and at higher temperature levels. This means higher combustor velocities, if the combustor frontal area is not to exceed that of the other engine components. The high air flows also indicate higher fuel flows and higher 3201
heat-release rates.

From the results of previous investigations (ref. 7), increased combustor flow velocities that are desirable in supersonic propulsion would be expected to result in decreased combustion efficiency. The higher pressures and combustor-inlet temperatures encountered at supersonic flight conditions may, however, tend to alleviate the adverse effect of velocity on combustion efficiency. The increased flow velocities will also increase combustor pressure loss, which adversely affects engine fuel consumption. The higher combustor-inlet and -outlet temperature levels (of the order of 400° to 500° F above current combustors) will increase the durability problems involved in the combustor parts.

The preliminary investigations that are reported herein are a part of a general research program at the NACA Lewis laboratory to determine design criteria of combustors for turbojet engines operating at high altitudes and supersonic flight speeds. Performance characteristics of two experimental single-annulus combustors were obtained at combustor-inlet-air conditions approximating those of an engine with advanced design components operating in the range of altitudes from 60,000 to 80,000 feet and of flight Mach numbers from 2.0 to 3.0. One-quarter sectors of the combustors were investigated in a direct-connect system. Pressure-atomized liquid fuel was used in both combustors. Hollow-cone spray nozzles injected fuel axially from the upstream face of the combustors; in addition, one combustor was equipped with flat spray nozzles injecting radially into the combustor for the purpose of fuel staging at high flow rates.

The performance of each combustor was evaluated at a single inlet-air temperature of 870° F, a range of inlet-air pressures from 10 to 30 pounds per square inch absolute, and a range of combustor velocities from 125 to 225 feet per second. Combustion efficiencies, pressure losses, and combustor-outlet-temperature profiles were determined at these conditions. Combustor-liner durability and carbon deposition were not evaluated during this investigation.

APPARATUS

Combustors

Two experimental annular combustors are herein described; neither of these combustors necessarily represents an optimum design. Since the air flow per unit frontal area of the engine is very high in a turbojet engine for supersonic application, an annular configuration was selected for the combustors in order to maintain as low a flow velocity as possible. Each combustor consisted of a one-quarter sector of a single-annular combustor designed to fit into a housing with an outside diameter of $25\frac{1}{2}$ inches, an inside diameter of $10\frac{5}{8}$ inches, and a combustor length of approximately 23 inches. The maximum combustor cross-sectional area of the sector was 105 square inches, which corresponds to 420 square inches for the complete combustor. In each of the combustors the primary air was admitted gradually and the secondary air, rapidly through large rectangular slots. Three-quarter cutaway views of the assembled combustors are shown in figure 1. The combustor longitudinal cross-sectional and air-entry hole geometries are shown in figures 2 to 4.

The geometric shape of combustor A (fig. 2(a)) was similar to that reported in reference 8 in that the combustor occupied the same volume and position within the housing. The primary zone was designed with a series of circular holes (fig. 3(a)) which allowed primary air to enter between the fuel nozzles, thus establishing alternate fuel- and air-rich zones. Large secondary slots (fig. 3(a)) were used to provide adequate penetration of the secondary air and to minimize flow restrictions. Fuel was introduced through five hollow-cone spray nozzles (10.5 gal/hr; 60° spray angle) located at the upstream face of the combustor. The design of combustor A was the result of the research described in reference 8 aimed toward the development of a high-performance combustor for high-altitude, subsonic flight conditions.

Combustor B was specifically designed to meet the requirements of high-altitude, high Mach number flight of an engine with advanced design components. Since a high combustor velocity is encountered at the design flight conditions, maintaining a minimum pressure loss in combustor B was a primary consideration. An attempt was made to reduce the annular losses by designing combustor B with a somewhat smaller combustion space (fig. 2(b)). Making the combustion space small, however, also adversely affects combustion efficiency, particularly at low pressures (ref. 7). Analytical studies of several flight missions of interest for supersonic turbojet aircraft have indicated that the combustor pressure will be above about 1 atmosphere at all flight conditions considered in the analysis (unpublished data). It will therefore not be necessary to meet the requirements of providing high efficiencies at very low pressures; this makes possible a compromise in combustor design to obtain lower pressure losses at the expense of combustion efficiency at low pressures.

Primary air was admitted into combustor B through a number of inverted louvers, as shown in figure 2(b). In addition, a relatively large proportion of the primary air was admitted in the upstream half of combustor B (see fig. 4) which served to further decrease the pressure-loss coefficient. Provision was made for fuel staging in combustor B since this technique was indicated to be particularly advantageous at high heat-release conditions (ref. 9). When the combustor was operated without fuel 3201
staging, all the fuel was injected through nine, hollow-cone, swirl-type nozzles (10.5 gal/hr; 60° spray angle) at the upstream end of the combustor liner. During operation with fuel staging, two-thirds of the fuel was injected through eight fan-spray injectors located 6 inches downstream and spraying radially into the combustor as shown in figure 1(b).

Combustor Installation

A schematic diagram of the combustor installation is shown in figure 5. Air of desired quantity, pressure, and temperature was drawn from the laboratory air-supply system, passed through the combustor, and exhausted into the altitude-exhaust system. Combustor-inlet temperatures were controlled by use of a gasoline-fired preheater which burned a portion of the air upstream of the combustor. The quantity of air flowing through the preheater, the total air flow, and the combustion-chamber static pressure were regulated by three remote-control valves. Two observation windows were installed in the test section in order to permit visual observation of the combustion process.

Instrumentation

Total temperatures and pressures were measured at the three stations indicated in figure 5. The position of the instruments in each of the three planes is shown in figure 6. Combustor-inlet total temperatures were measured with three bare-junction, unshielded, iron-constantan thermocouples at station 1, as shown in figure 6(a). Slightly upstream were located 12 total-pressure tubes, three tubes in each of four rakes as shown in figure 6(a). Combustor-outlet total temperatures were measured with 30 bare-junction, unshielded, chromel-alumel thermocouples; five thermocouples in each of five rakes were located across the duct at station 2, 23 inches from the upstream end of the combustor (fig. 6(b)). At station 3 were located 15 total-pressure tubes in three rakes of five pressure tubes each (fig. 6(c)). All instruments were located at approximate centers of equal areas. Static-pressure orifices were installed at the wall, as shown in figure 6(c). Construction details of the pressure and temperature probes are shown in figure 7.

Rotameters were used to measure the fuel flow; MIL-F-5624A grade JP-4 fuel was used throughout the investigation.

PROCEDURE

The combustor was operated at conditions considered to be representative for an engine with a compressor ratio of 7, flight Mach numbers from 2.0 to 3.0, and flight altitudes from 60,000 to 80,000 feet.

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Flight analyses such as shown in reference 10 indicated that the minimum combustor-inlet pressures encountered would be above 1 atmosphere. The minimum combustor-inlet pressure was 18.4 pounds per square inch absolute for a given interceptor flight plan and was therefore chosen as a standard test point. Combustor-inlet pressures of 10 and 30 pounds per square inch absolute were also included in the test schedule to show the effect on performance of variations in inlet pressure. Combustor reference velocities typical for these conditions of supersonic flight ranged from approximately 150 to 200 feet per second. Data were obtained over a range of velocities from 125 to 225 feet per second to determine the effect of variation in velocity on performance. Minimum combustor-inlet temperatures were determined to be about 870° F, and therefore this value was chosen as a standard test parameter. Turbine-inlet temperatures of 2000° F have been shown (ref. 10) to be desirable for obtaining the high thrust necessary for high supersonic speed. Because of instrumentation limitations, average combustor-outlet temperatures were maintained at 1800° F for most of the runs; a single run was made at a 2000° F outlet temperature with combustor B. The test conditions are shown in tabular form in the following table:

Combustor-inlet total pressure, lb/sq in. abs	Combustor-inlet total temperature, °F	Combustor reference velocity, ft/sec (a)	Combustor-outlet temperature, °F
10.0	870	165	1800
30.0	870	165	1800
18.4	870	125	1800
18.4	870	165	1800
18.4	870	204	1800
18.4	870	225	1800
18.4	876	204	2000

[a]Based on maximum combustor cross-sectional area of 105 sq in. and combustor-inlet air density.

Combustion efficiency, outlet temperature profile, and pressure losses were evaluated for each combustor. Combustion efficiency was computed as the percentage ratio of actual to theoretical increase in enthalpy from the combustor-inlet to the combustor-outlet instrumentation planes by using the method of reference 11. The arithmetic mean of the 30 outlet thermocouple readings was used to obtain the value of the combustor-outlet enthalpy. The accuracy of the combustion efficiency calculated from these readings was considered to be about ± 3 percent. The radial outlet-temperature distribution was determined for an average outlet temperature of approximately 1800° F. The temperature at each of 3201
five radial positions was computed as the average of six circumferential thermocouple readings at each position. The pressure loss was computed as the percentage ratio of pressure loss through the combustor to the inlet total pressure.

RESULTS AND DISCUSSION

The performance of two experimental annular combustors, over a limited range of operating conditions that are representative of supersonic flight, are discussed subsequently. The performance criteria considered include combustion efficiency, combustor pressure loss, and outlet-temperature profile.

Combustion Efficiency

Effect of velocity. - The effect of combustor reference velocity on combustion efficiency is shown in figure 8 for each of the two combustors operating at a constant value of inlet-air temperature of 870° F, inlet-air pressure of 18.4 pounds per square inch absolute, and an average combustor-outlet temperature of approximately 1800° F. Data are shown for a range of reference velocities from 125 to 225 feet per second. Combustor velocity, as discussed herein, is based on the density of the combustor-inlet air and on the maximum cross-sectional area of the combustor. The combustion efficiency of combustor A was essentially 100 percent at all velocities investigated except the lowest velocity (125 ft/sec) where the combustion efficiency was 97 percent. For combustor B without fuel staging the combustion efficiency decreased from 100 percent at a reference velocity of 165 feet per second to 88 percent at 225 feet per second (fig. 8). Fuel staging served to improve the performance of combustor B at the higher velocities. At a reference velocity of 225 feet per second, the combustion efficiency of combustor B with fuel staging was 97 percent. A single data point was obtained for combustor B with fuel staging at a higher combustor-outlet temperature (2000° F). This data point is included in figure 8 and shows that the combustion efficiency remained high at this higher outlet temperature.

The data presented in figure 8 indicate that high combustion efficiency can be obtained at the high combustor reference velocities that are anticipated in future turbojet engines operating at high altitudes and supersonic flight speeds. Moreover, the high combustion efficiencies can be obtained with at least two experimental combustors of significantly different design.

Effect of pressure. - The effect of combustor-inlet pressure on the combustion efficiencies of each of the two combustors is shown in figure 9 for a constant combustor-inlet-air temperature of 870° F, an average outlet temperature of 1800° F, and a reference velocity of 165 feet per second. Above 18.4 pounds per square inch absolute, the combustion efficiency was approximately 100 percent for both combustors; however, as the pressure was reduced to 10 pounds per square inch absolute, the combustion efficiency of combustor A decreased to 82.5 percent and that of combustor B, to 62.5 percent. The marked effect of low pressure on the combustion efficiency of combustor B is the result of the design compromises previously noted (small combustion space and rapid entry of primary air). The effect of low pressures on the efficiency of combustor A is partly due to the fact that this combustor configuration was developed (ref. 8) for use with a fuel prevaporizer, while in this investigation liquid fuel was used. Combustor pressures below about 1 atmosphere would not be encountered in the turbojet-powered aircraft capable of flight at high supersonic Mach numbers which were considered in analytical studies conducted at this laboratory; as shown in figure 9, near 100-percent combustion efficiency was obtained with both combustors at these conditions.

The indicated combustion efficiencies at pressures of 10 pounds per square inch absolute may be low by several percentages because of oxygen depletion in the inlet air due to the gas-fired preheater. Oxygen depletion has been shown to have a more severe effect at low pressures (ref. 12).

Combustor-Outlet Temperature Profile

Typical combustor-outlet isothermal contour patterns for combustors A and B are shown in figure 10, and the radial outlet-temperature profiles in figure 11. A maximum average temperature deviation from inner to outer wall of 220° F was obtained with combustor A and a maximum deviation of 70° F was obtained with combustor B. Preliminary analysis has indicated that uniform temperature distributions such as these are particularly applicable for cooled turbine blades that may be used in engines for high supersonic flight, inasmuch as the preferred gas-temperature profile for cooled turbine blades is radially more uniform than for uncooled turbines. Previous studies (ref. 13) describe methods of controlling outlet-temperature profiles. It is expected that no significant sacrifice in other performance characteristics would be required to provide temperature profiles different from those shown in figure 11.

Combustor Pressure Losses

The percent total-pressure loss of each of the two combustors is shown as a function of reference velocity in figure 12. The pressure loss of combustor A is approximately 20 percent as compared with 15 percent for combustor B at a reference velocity of 204 feet per second and a temperature ratio across the combustor of about 1.7. The lower pressure losses obtained with combustor B are the result of the features (small combustion space and rapid entry of primary air) that were incorporated in the design to obtain lower pressure-loss coefficients. The pressure losses represented by the curves of figure 12 are not to be considered the minimum required for high efficiency at the conditions investigated, since the design variables were investigated to a very limited extent. 3201

The specific-fuel consumption, as a function of the total-pressure losses, was calculated by using the method of reference 14 for a representative subsonic and supersonic flight condition. The specific-fuel consumption is plotted in figure 13 as the ratio of the actual to the ideal specific-fuel consumption with no pressure loss in the combustor assumed. A pressure loss of about $5\frac{1}{2}$ percent, which is obtained in many current combustors for subsonic flight conditions, results in an increase in specific-fuel consumption of about 2.3 percent in the 5:1 pressure-ratio engine, as shown in figure 13. For this same effect on specific-fuel consumption, pressure losses of 9.2 percent are permitted at the supersonic flight condition in a 7:1 compressor pressure engine; therefore, it is evident that higher pressure losses can be tolerated in the engine for supersonic flight than in current engines for subsonic flight while equivalent performance levels are maintained. Pressure loss has a lesser effect on specific-fuel consumption at the supersonic flight conditions mainly because the ram-temperature-rise ratios encountered in supersonic flight are high (ref. 14). In any application, however, it is obviously desirable to design for a minimum value of pressure loss.

Carbon and Durability

During the investigation which included operation at pressures as high as 30 pounds per square inch absolute, no carbon deposits were evident; however, with sustained high-temperature operation over several hours moderate to severe liner deterioration occurred.

CONCLUDING REMARKS

The performance results presented indicate that combustion efficiencies over 95 percent and satisfactory outlet-temperature profiles can be obtained in annular combustors operating at simulated supersonic flight conditions with combustor velocities as high as 225 feet per second. These high efficiencies were maintained to the highest combustor-outlet temperatures investigated, 1800° F for one combustor and 2000° F for the other. The total-pressure losses of the two experimental combustors were acceptable by present standards, since calculations indicated that the increase in engine specific-fuel consumption resulting from combustor pressure losses would not be significantly greater in the engine for supersonic propulsion than in current turbojet engines. These pressure losses, therefore, appear acceptable for the supersonic flight conditions. Further engine performance gain could be realized, however, if the losses could be reduced. Liner durability may well prove to be one of the largest problems facing the combustor designer for combustor applications involving temperature levels of interest for high supersonic flight.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, January 18, 1954

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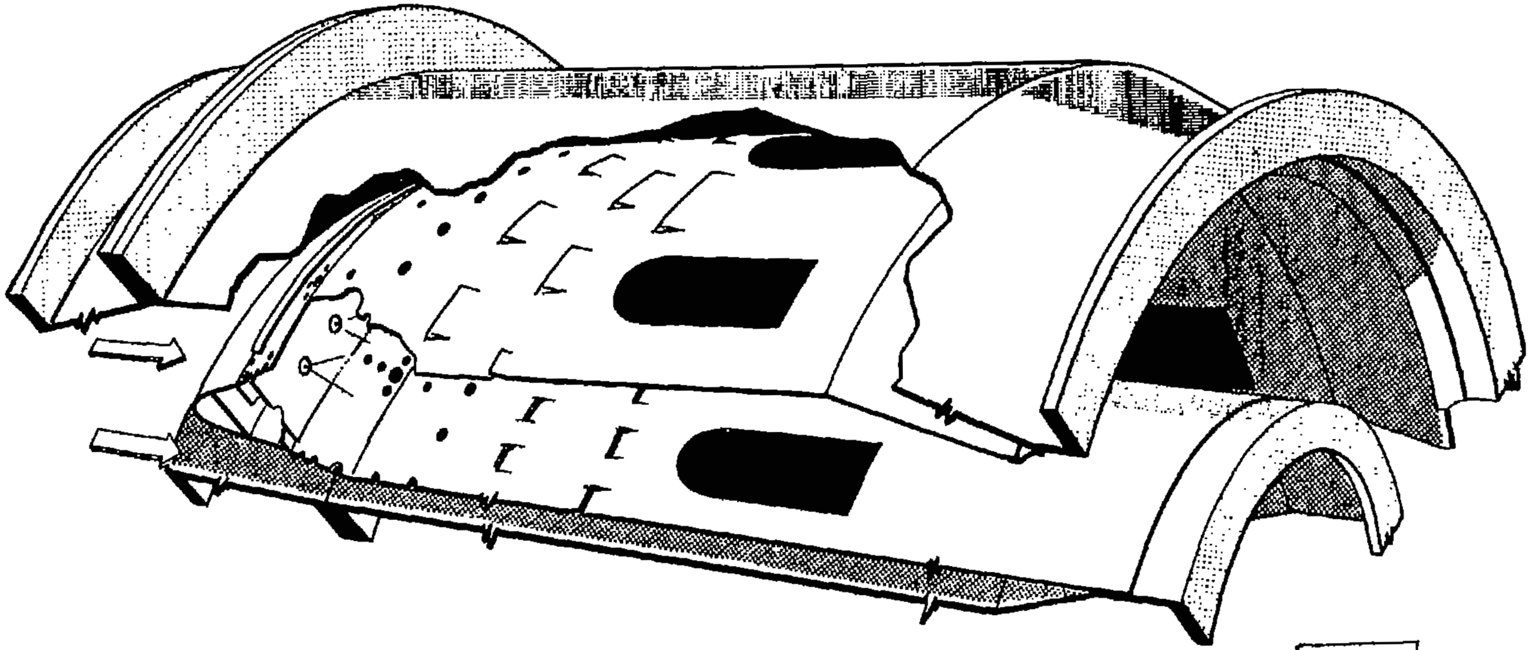
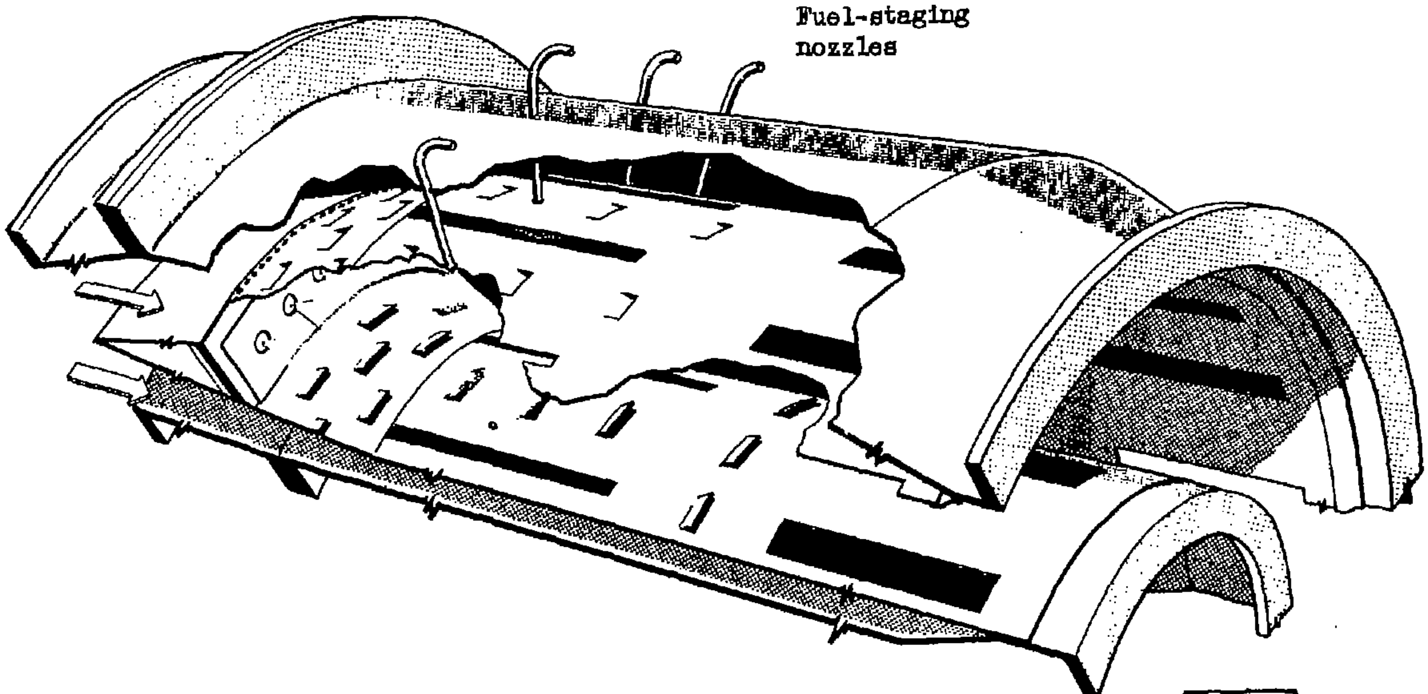


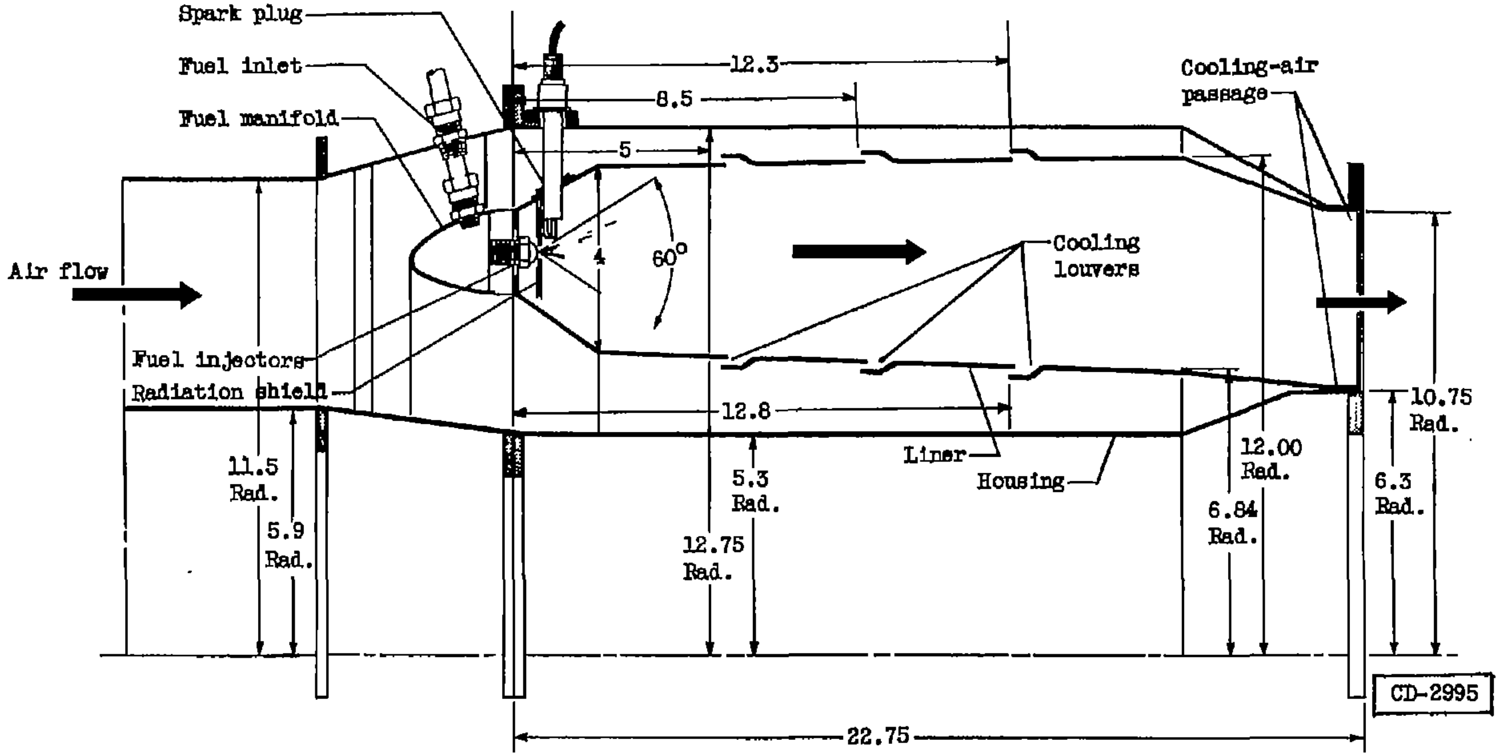
(a) Combustor A.

Figure 1. - Cutaway view of experimental annular turbojet combustors assembled in housing.



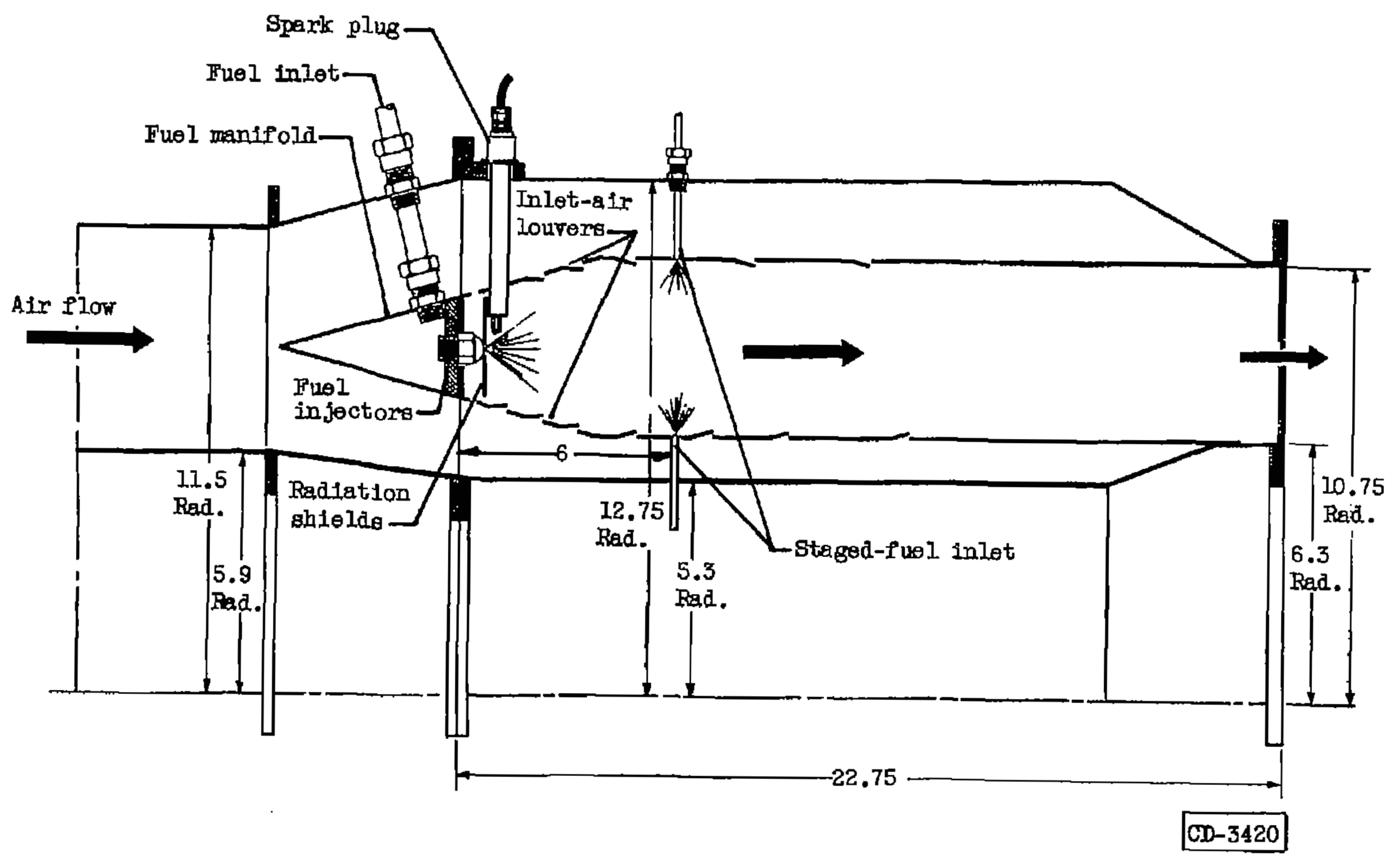
(b) Combustor B.

Figure 1. - Concluded. Cutaway view of experimental annular turbojet combustors assembled in housing.



(a) Combustor A.

Figure 2. - Longitudinal cross-sectional view of experimental annular turbojet combustors. (Dimensions are in inches.)



(b) Combustor B.

Figure 2. - Concluded. Longitudinal cross-sectional view of experimental annular turbojet combustors. (Dimensions are in inches.)

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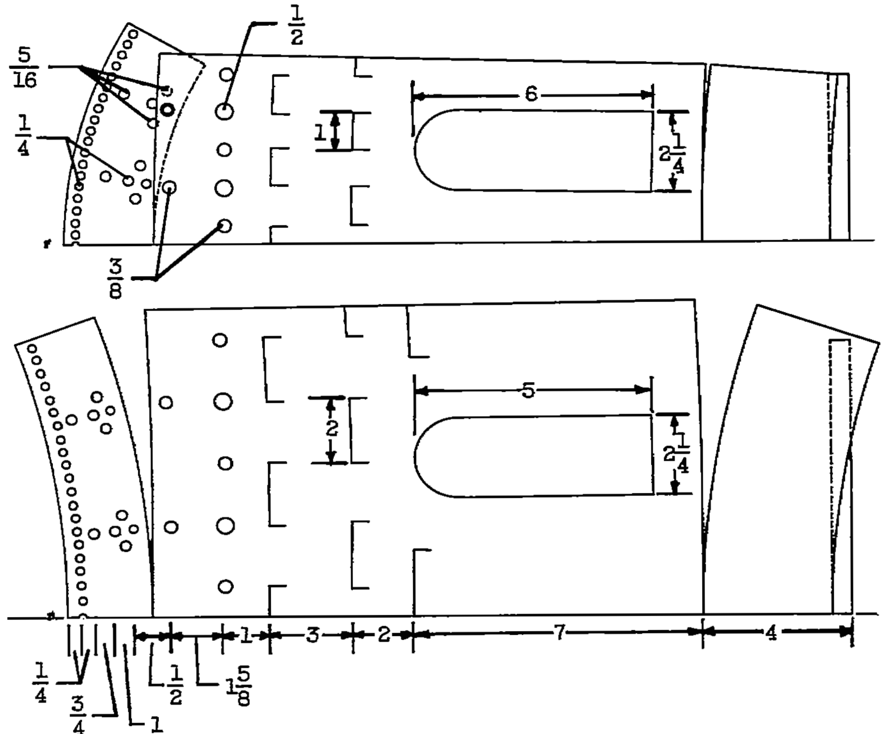
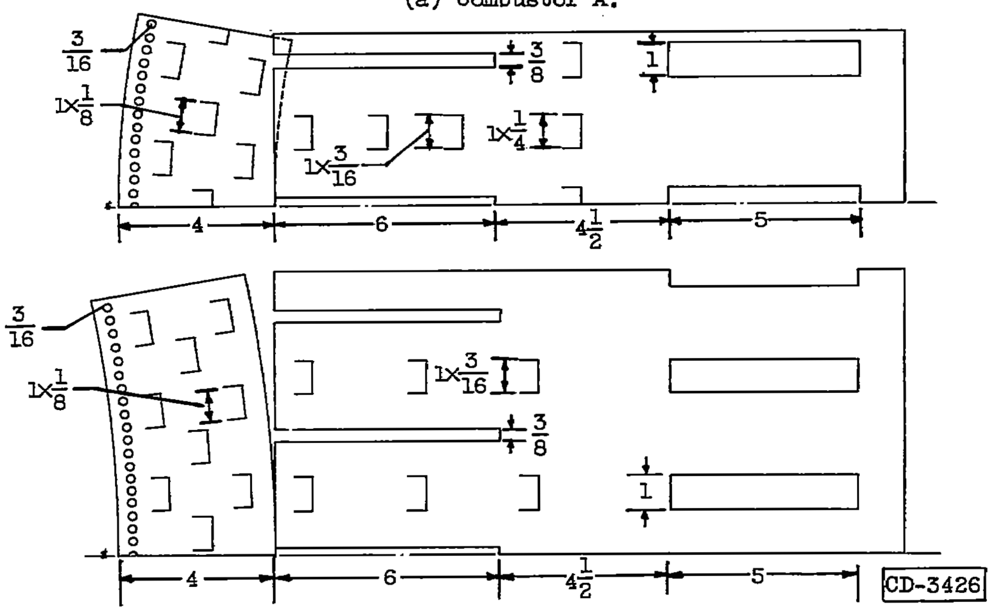


(a) Combustor A.



(b) Combustor B.

Figure 3. - Liner air-entry hole patterns of experimental annular turbojet combustors. (Dimensions are in inches.)

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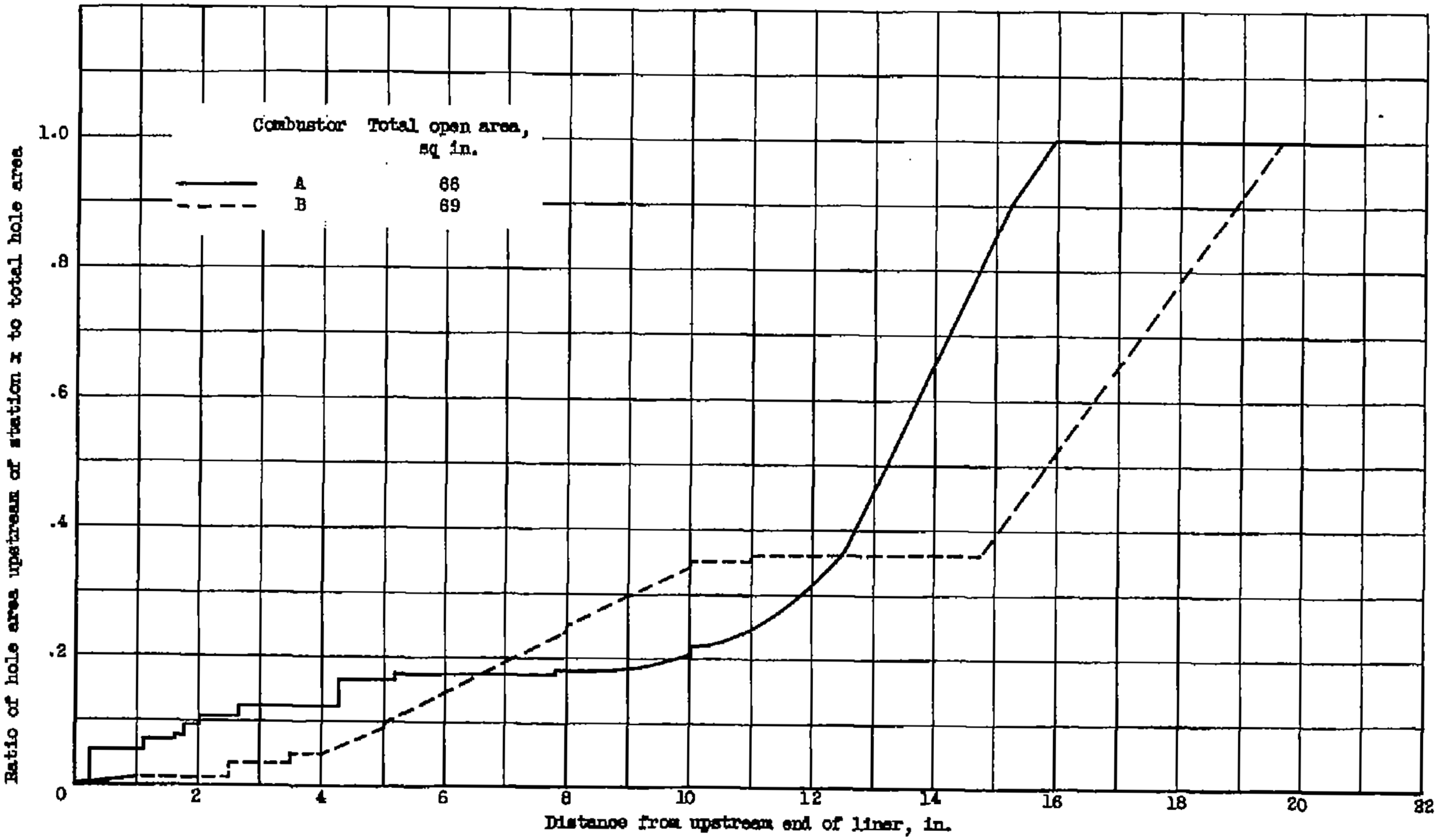

Figure 4. - Air-entry open-hole-area distribution for experimental annular turbojet combustors.

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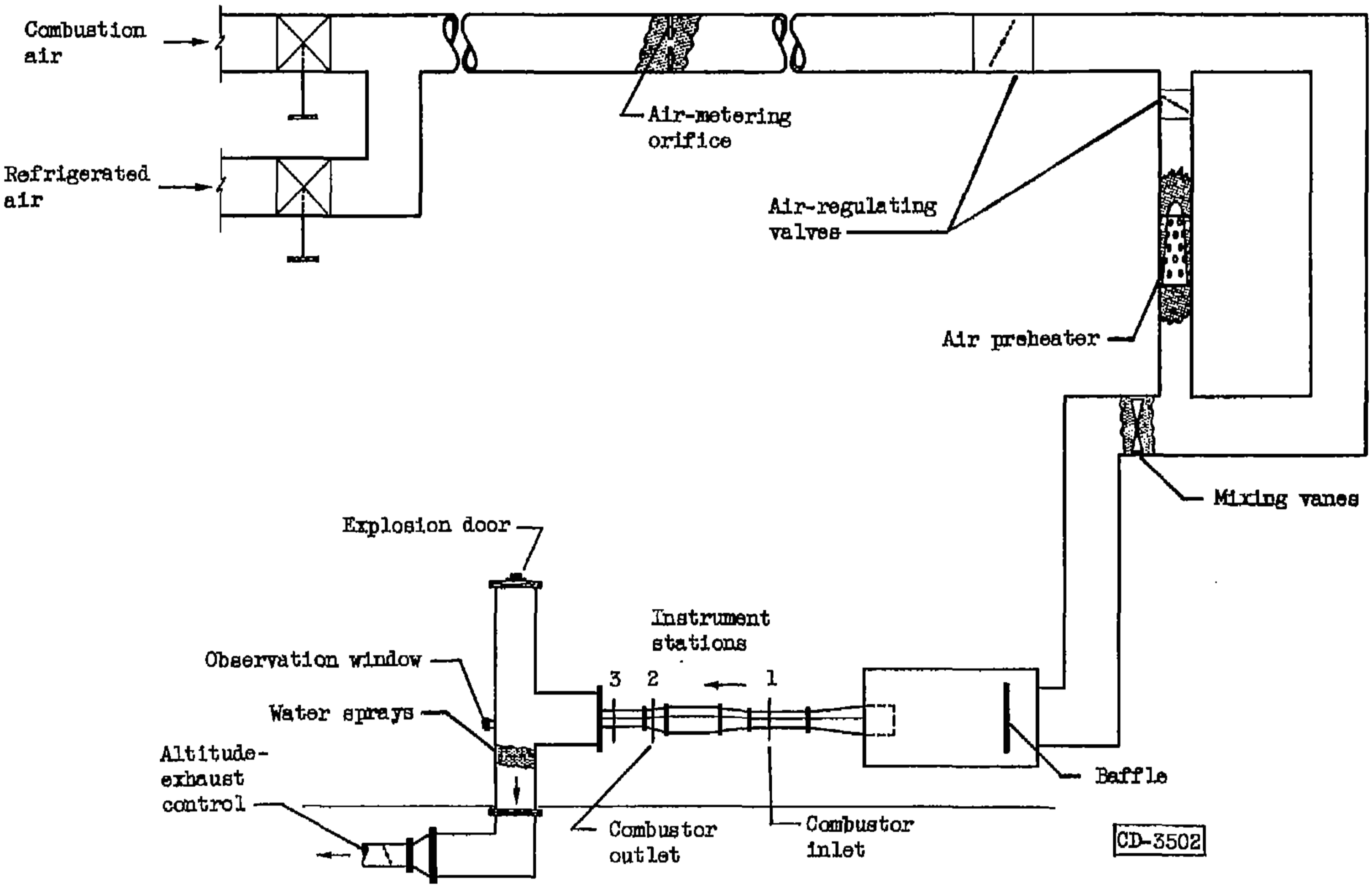
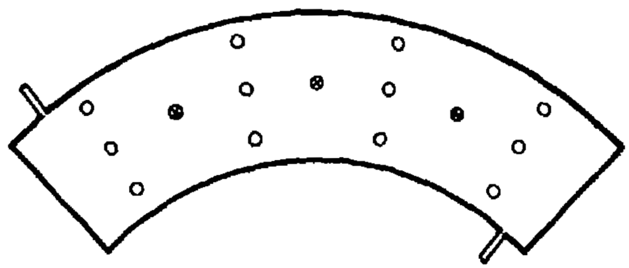
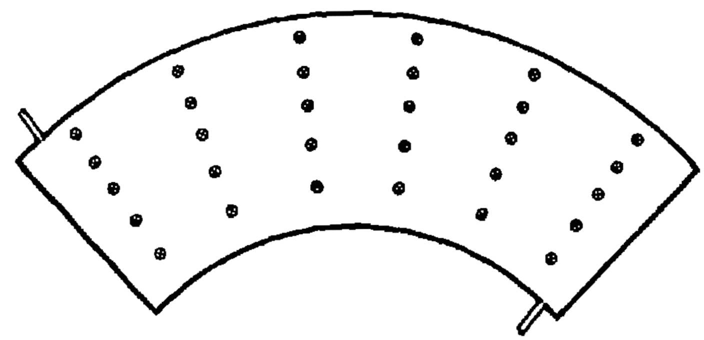


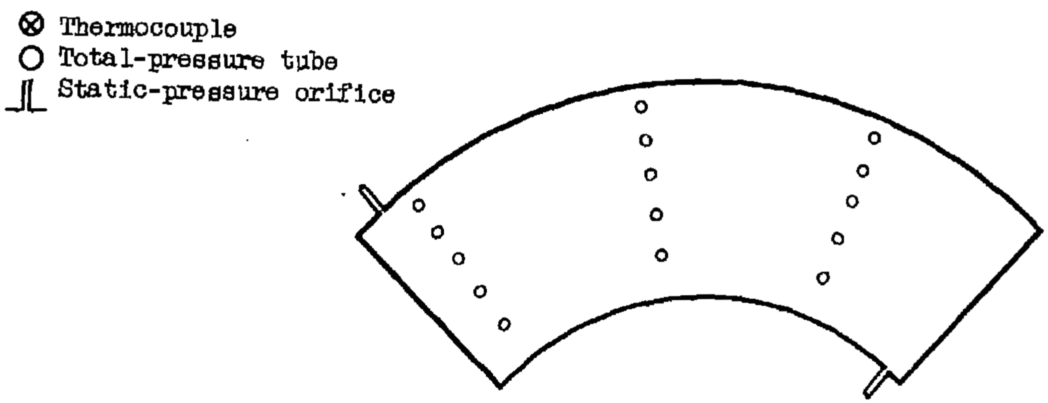
Figure 5. - Installation of experimental annular turbojet combustors.



(a) Inlet thermocouples (iron-constantan) and inlet total-pressure rakes in plane at station 1.



(b) Outlet thermocouples (chromel-alumel) in plane at station 2.



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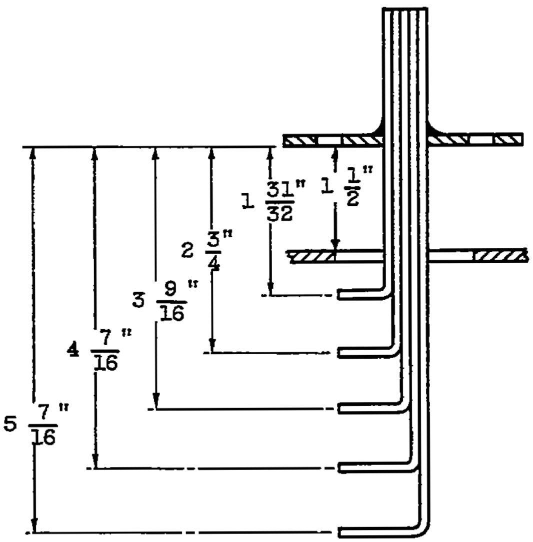
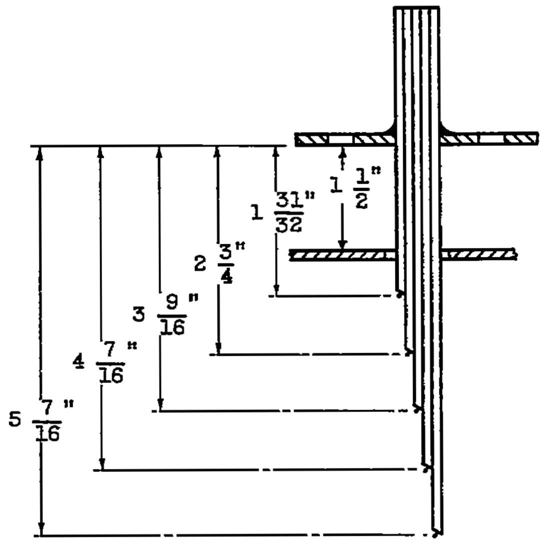
(c) Outlet total-pressure rakes in plane at station 3.

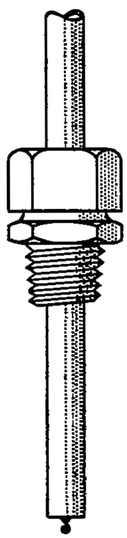
Figure 6. - Instrumentation in experimental annular turbojet combustors.



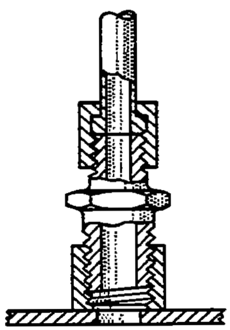
(a) Outlet total-pressure rake.



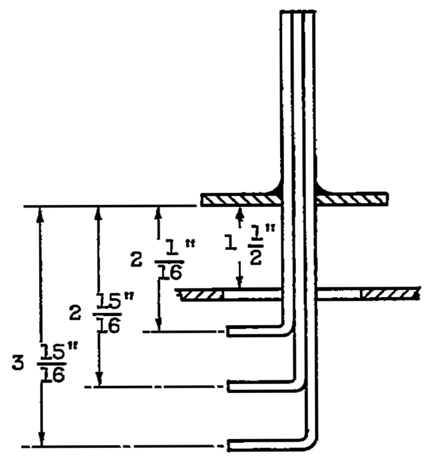
(b) Outlet thermocouple rake.



(c) Inlet thermocouple.



(d) Static-pressure orifice.



(e) Inlet total-pressure rake.

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Figure 7. - Details of instrumentation in annular turbojet combustors.

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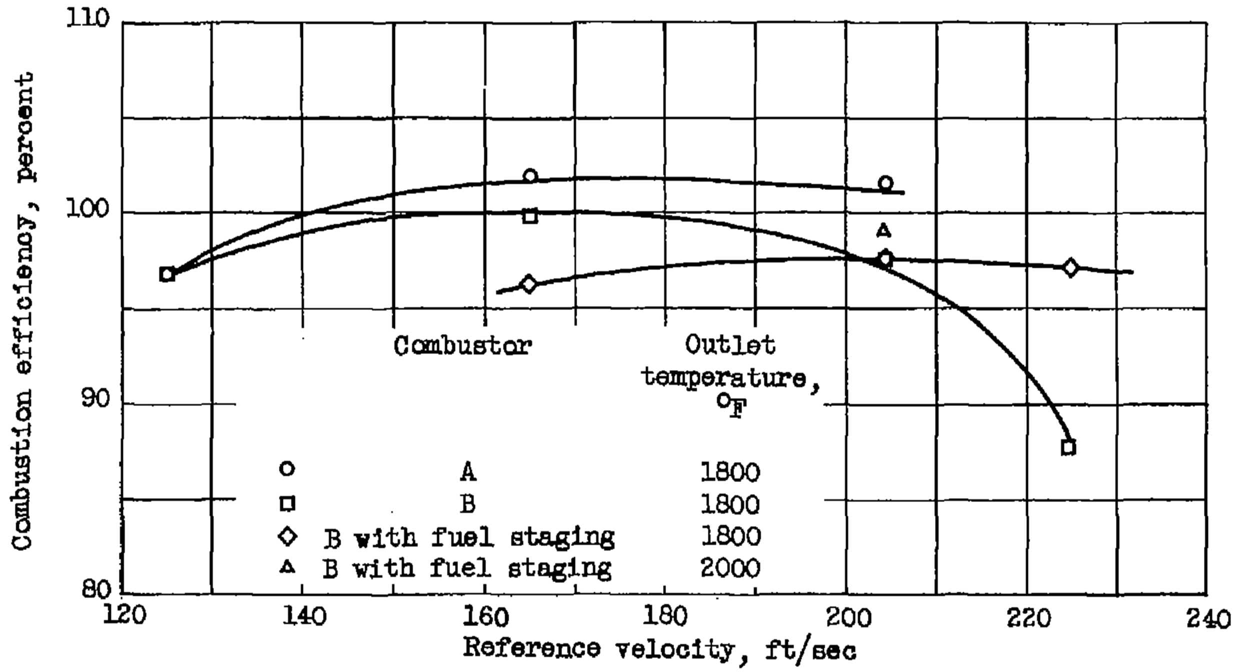

Figure 8. - Effect of velocity on combustion efficiency of experimental annular turbojet combustors. Inlet-air pressure, 18.4 pounds per square inch absolute; inlet-air temperature, 870° F.

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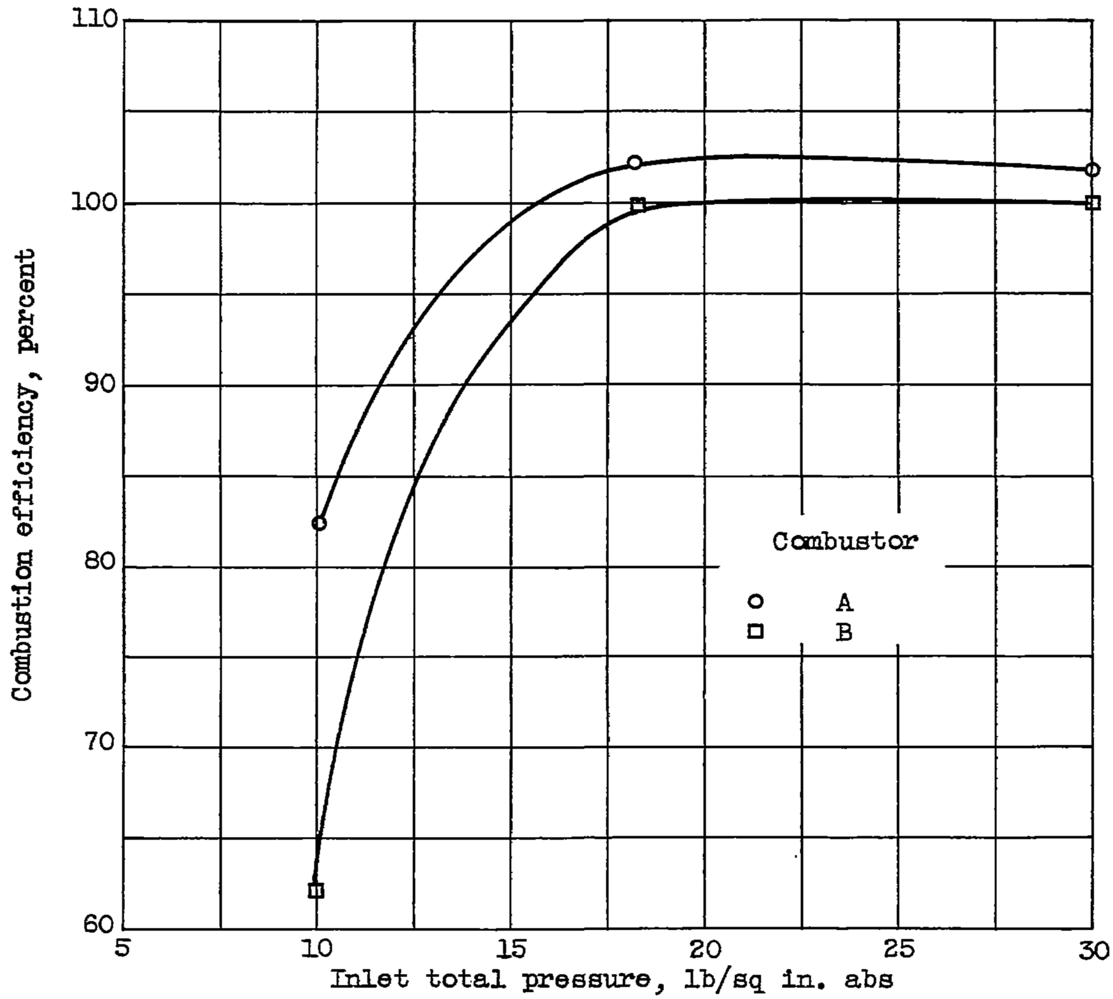
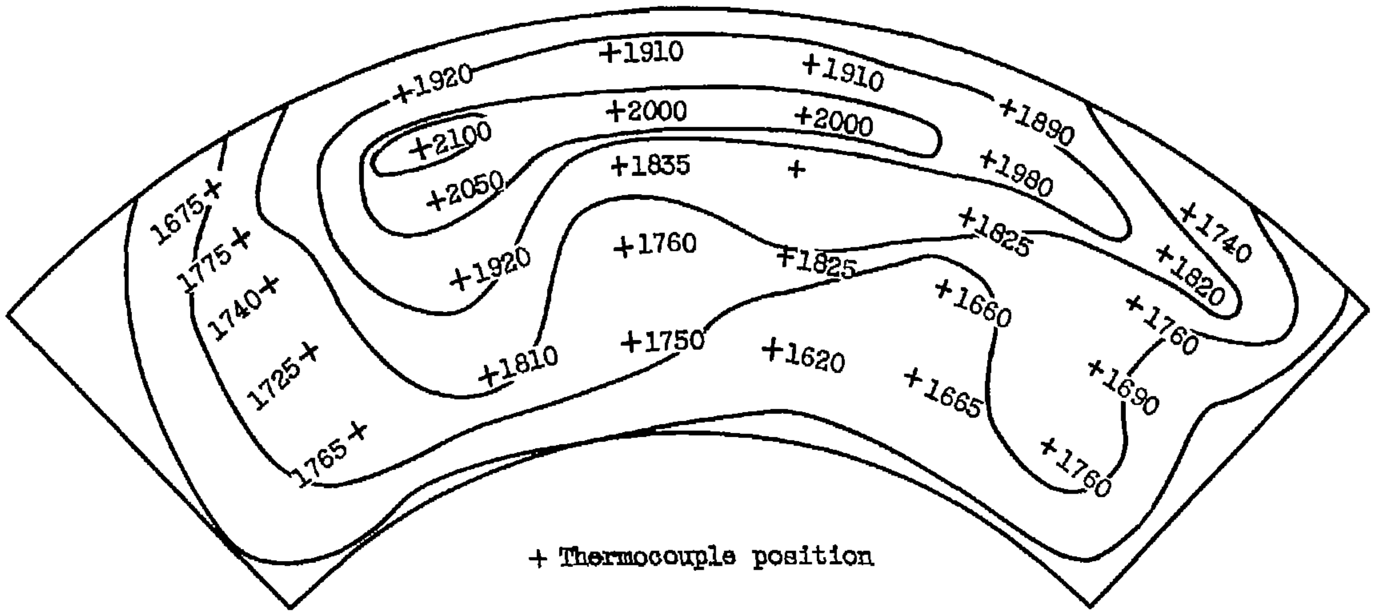
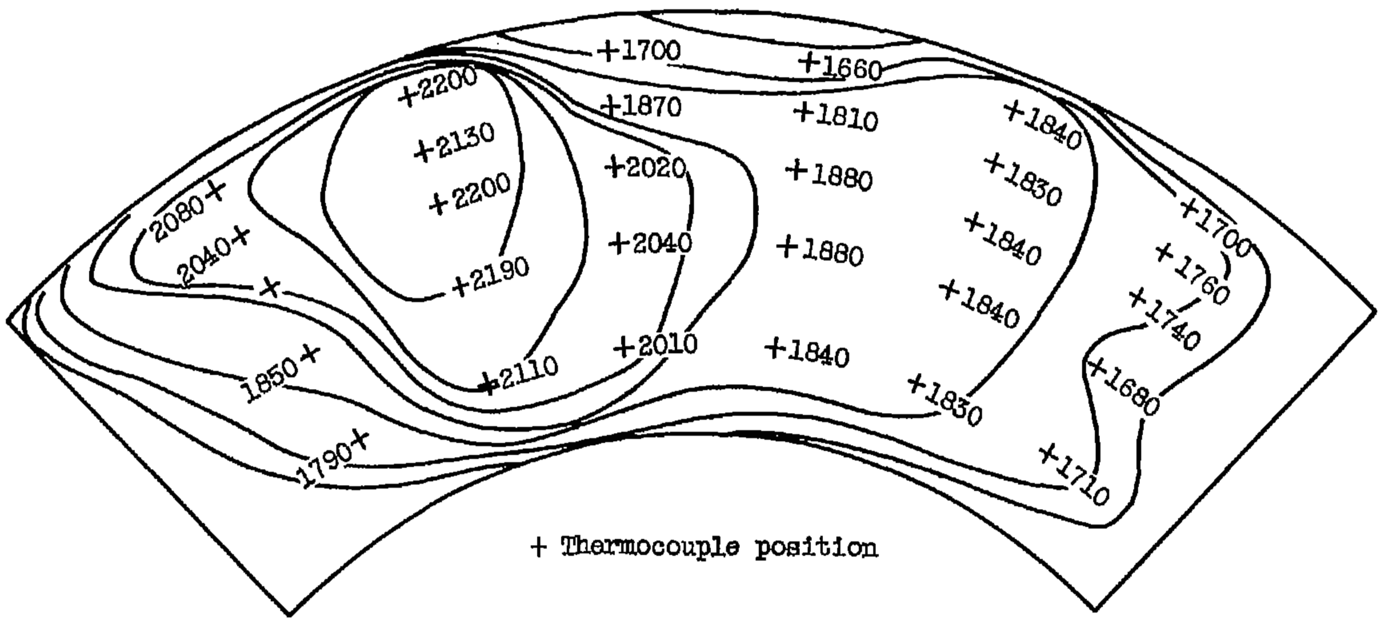


Figure 9. - Effect of pressure on combustion efficiency of experimental annular turbojet combustors. Reference velocity, 165 feet per second; inlet-air temperature, 870° F; average outlet temperature, 1800° F.



(a) Combustor A.

Figure 10. - Isothermal contour patterns at combustor outlet of experimental annular turbojet combustors. Reference velocity, 165 feet per second; inlet-air pressure, 18.4 pounds per square inch absolute; inlet-air temperature, 870° F; average outlet temperature, approximately 1850° F.



(b) Combustor B.

Figure 10. - Concluded. Isothermal contour patterns at combustor outlet of experimental annular turbojet combustors. Reference velocity, 165 feet per second; inlet-air pressure, 18.4 pounds per square inch absolute; inlet-air temperature, 870° F; average outlet temperature, approximately 1850° F.

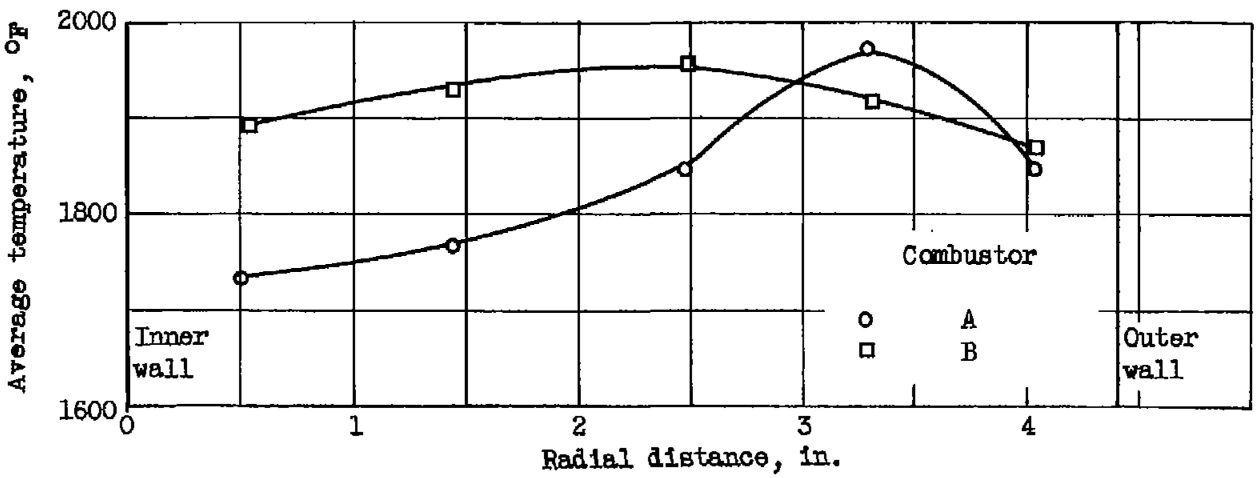


Figure 11. - Radial temperature profile at combustor outlet of experimental annular turbojet combustors. Reference velocity, 165 feet per second; inlet-air pressure, 18.4 pounds per square inch absolute; inlet-air temperature, 870° F; average outlet temperature, approximately 1850° F.

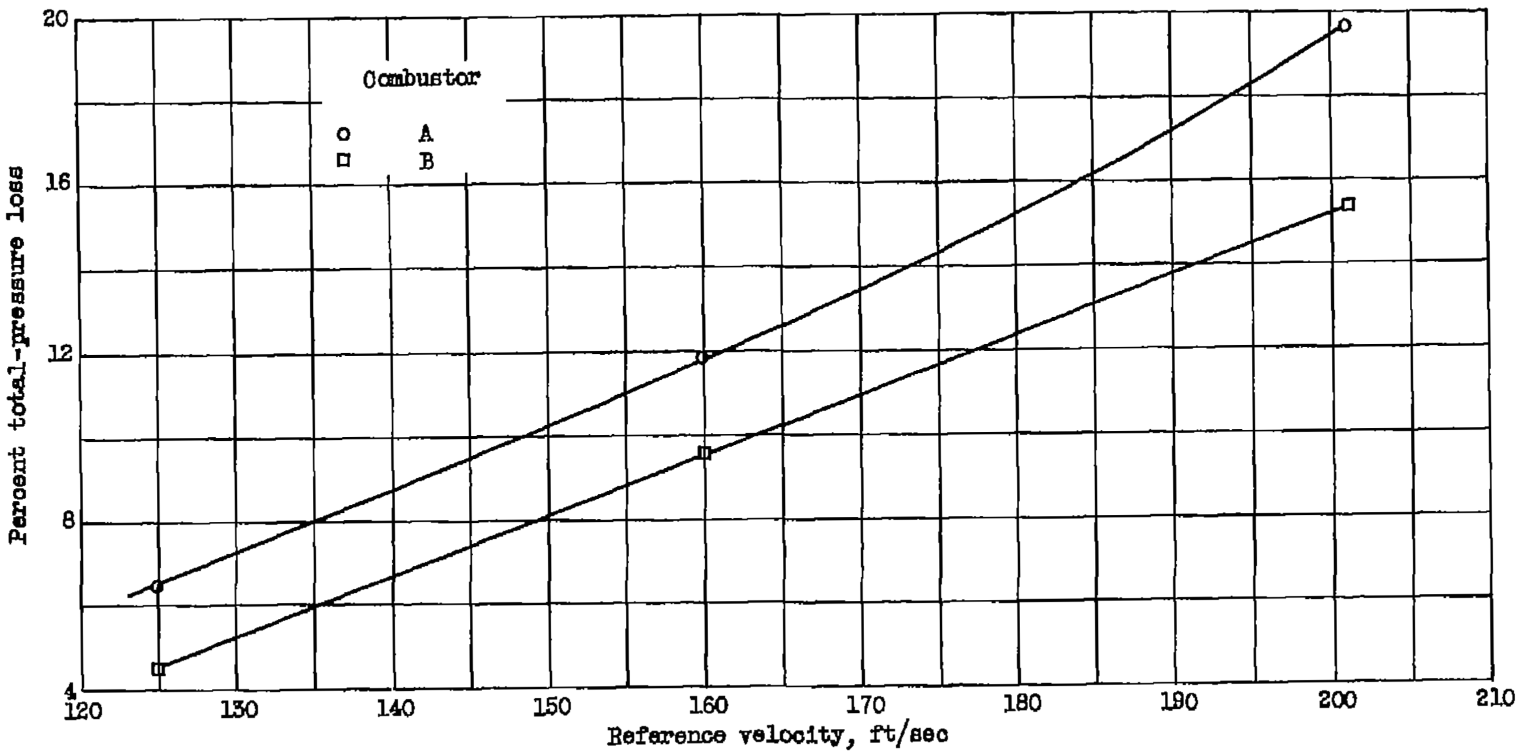


Figure 12. - Effect of velocity on pressure loss of experimental annular turbojet combustors. Inlet-air pressure, 18.4 pounds per square inch absolute; inlet-air temperature, 870° F; outlet temperature, approximately 1800° F.

[redacted]

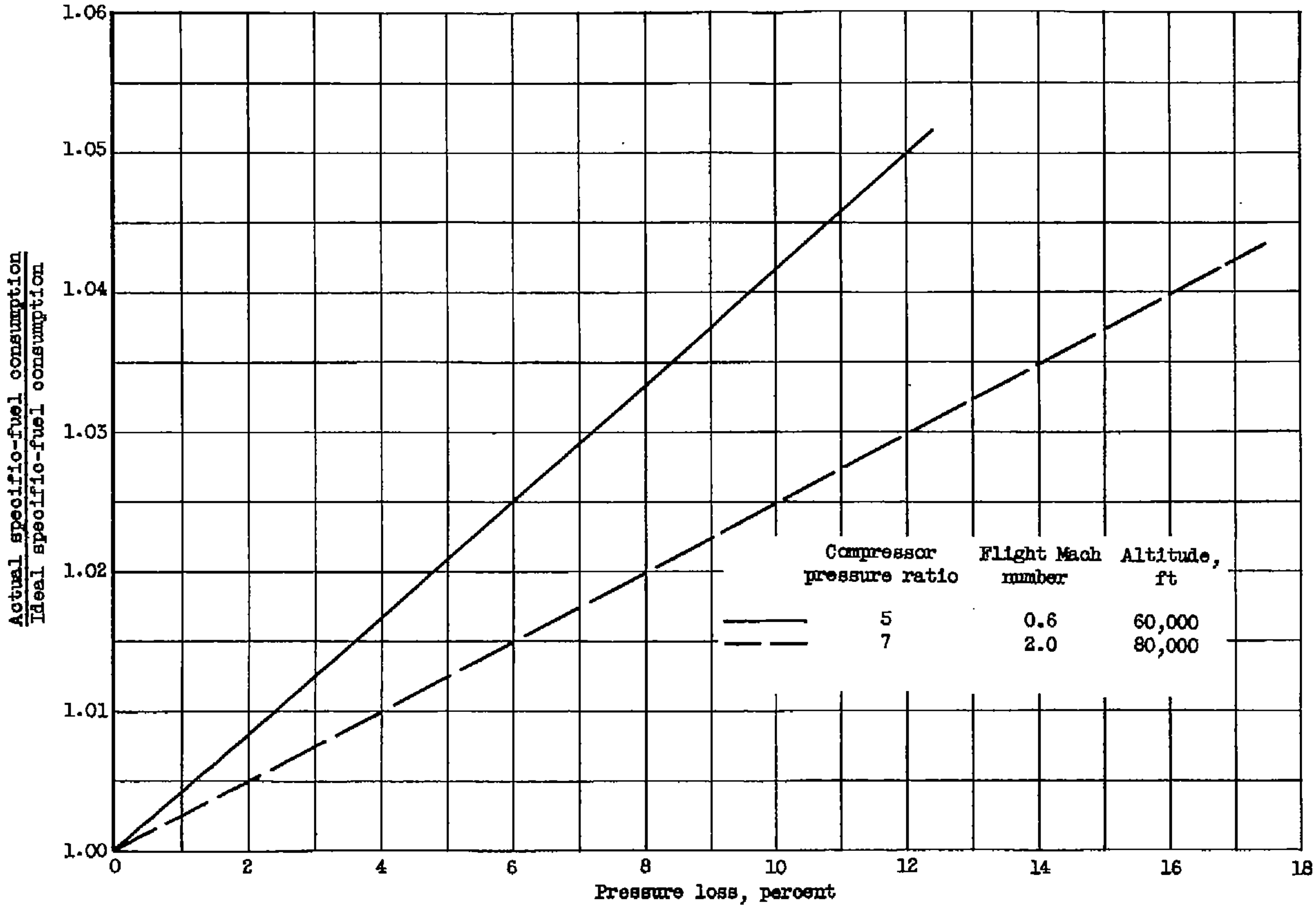


Figure 13. - Effect of pressure loss on specific-fuel consumption for representative subsonic and supersonic flight operation.

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