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RESEARCH MEMORANDUM

METHOD FOR SHORTENING RAM-JET ENGINES BY BURNING HYDROGEN FUEL IN THE SUBSONIC DIFFUSER

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

METHOD FOR SHORTENING RAM-JET ENGINES BY BURNING HYDROGEN FUEL IN THE SUBSONIC DIFFUSER

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SUMMARY

Merging of the subsonic diffuser and the combustor appears feasible with a highly reactive fuel such as hydrogen. At typical ram-jet operating conditions the flame speed of this fuel is high enough for burning to be stabilized at velocities of 600 feet per second by means of a fuel injector alone. Thus it was possible to seat the flame at a station where the Mach number was 0.4 to 0.5 rather than 0.2 as is done conventionally. Besides the decrease in engine length and weight, a further gain was found due to a uniform exhaust-gas temperature distribution which varied as little as ±7 percent from the mean of 2100° R. Also, combustion efficiency was improved at lean fuel-air mixtures; values of 90 percent or higher were obtained at equivalence ratios as low as 0.1. However, the total-pressure loss coefficient increased to 9.8 from 2.6 for the conventional system at a temperature ratio across the combustor of 2.8. An estimate of the net effect of these factors was made on the basis of range potential of a ram-jet-powered missile. At a lean fuel-air ratio the gain in range was estimated as 16 percent above that of hydrogen fuel used in a conventional configuration.

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INTRODUCTION

This program is part of an NACA investigation to determine the possibility of improving the range of jet-propelled aircraft through the use of high-energy fuels. Aerodynamic analyses have shown (ref. 1) that a ram-jet-powered missile of a given weight has a range potential as much as 70 percent greater with hydrogen than with hydrocarbon fuels. It may be possible to further increase the potential of these special fuels such as hydrogen by making use of their high reactivity to reduce engine length and weight.

One of the obstacles to shorter engines is the subsonic diffuser. Its function is to reduce air velocities from Mach numbers near 1 to values of around 0.2 at the diffuser exit. The method most frequently used to shorten diffuser length is to make the divergence angle greater than the conventional 6°. However, this often results in nonuniform air flows and fuel-air mixtures and consequent reduction in thrust due to temperature stratification as well as engine failure due to overheating of walls.

The objective of this program was to investigate the feasibility of merging the subsonic diffuser and the combustor. This would have the obvious advantage of shortening the over-all engine length. Another possible advantage would be a uniform flow profile at the point of fuel introduction. In addition, with a controlled rate of heat release, sharp 4119
divergence angles could be tolerated without separation, since a favorable pressure gradient could be maintained. Obviously, these advantages would be weighed against the increase in total-pressure loss due to addition of heat at a high Mach number.

The results of experiments with a $9\frac{3}{4}$-inch ram jet installed in a free-jet facility are reported. Tests were conducted at an inlet-air pressure of about 1 atmosphere and over a range of temperatures from 100° to 400° F. Burner-inlet Mach numbers based on maximum combustor cross section were around 0.2. The variables investigated and their effect on range were diffuser and fuel-injector geometry. In addition, an estimate of the rate of temperature rise along the combustor axis was made for two dissimilar fuel injectors and for a range of inlet-air temperatures.

SYMBOLS

The following symbols are used in this report:

A	cross-sectional area, sq ft
L	diffuser length, ft
M	Mach number
P	total or stagnation pressure, lb/sq ft abs
$\mathscr{P}$	total-pressure-loss factor
q	dynamic pressure, lb/sq ft abs
T	total or stagnation temperature, °R
$\mathscr{T}$	temperature-stratification factor
t	static temperature, °R

V linear velocity, ft/sec

$\mathcal{W}$ engine-weight factor

γ specific-heat ratio

η combustion efficiency

τ ratio of total temperatures

Subscripts:

A,B,C,D, E,F,G,H stations in analytical engine

m mean

x station in analytical engine

1,2,3,4, 5 ... engine stations (see fig. 2)

APPARATUS

Free-Jet Facility

A free-jet facility is a very useful tool for evaluating supersonic ram-jet engines since it provides a good simulation of supersonic flight. However, for many combustion studies a much simpler and cheaper connected-pipe facility is adequate. An attempt to combine the simplicity of connected-pipe operation with supersonic flight simulation was made in the installation diagramed in figure 1. As can be seen in this drawing, the system is a free jet which can also be operated as a connected-pipe facility through the simple manipulation of valves. The accessibility of a connected-pipe rig has been retained since most of the test engine is exposed.

Combustion air was supplied by the laboratory air supply system. The same supply furnished primary air for the ejector altitude exhaust. Combustion air was metered and throttled before entering the inlet plenum. From the plenum the flow path led through the free-jet nozzle, the engine and spill-air line, and to the altitude exhaust system. Modulation of the motivating air flow to the ejectors was the only means for controlling the pressure level in the engine.

Ram-Jet Engine

A sketch of the engine is shown in figure 2. The engine inlet for all configurations tested was a spike diffuser designed for a flight Mach number of 1.8. This diffuser was identical to that used for the free-flight configuration of reference 2.

Pilot fuel was introduced into the centerbody between stations 1 and 2, and ignited by a spark plug. The air for the pilot recirculated from the downstream end of the centerbody, and the hot gases from the pilot ignited the main fuel. The main fuel was injected in the plane of station 2, and the combustion chamber extended to station 9. This station was the inlet to a convergent-divergent exhaust nozzle designed for the same flight conditions as the inlet.

Combustor Configurations

The various configurations investigated are shown in figure 3. All of the configuration changes consisted of variations in the centerbody and fuel injectors. The axial distances between stations were held constant. The pertinent features of each configuration are summarized as follows:

4119

Configuration	Centerbody design		Fuel injector
	Station 1 to station 2	Station 2 to station 7	
A	Conical section, $6\frac{1}{8}$-inch O.D. at station 1 tapered to 4-inch O.D. at station 2	No centerbody	Eight radial spokes injecting normal to airstream
B	Conical section of configuration A plus vortex generators attached at station 1	↓	↓
C	Cylindrical section, $6\frac{1}{8}$-inch O.D.	↓	Annular ring injecting normal to airstream
D	↓	Conical section tapered from $6\frac{1}{8}$-inch O.D. to 4-inch O.D. with length of 11 inches	↓
E	↓	Conical section tapered from $6\frac{1}{8}$-inch O.D. to 4-inch O.D. with length of $5\frac{1}{2}$ inches	↓
F	Cylindrical section, $6\frac{1}{8}$-inch O.D., $11\frac{5}{8}$ inches long. $2\frac{3}{8}$-inch-long conical section tapering to 5.14-inch O.D.	Conical section tapered from 5.14-inch O.D. to 4-inch O.D. with length of $3\frac{1}{8}$ inches	↓
G	Cylindrical section, $6\frac{1}{8}$-inch O.D., $11\frac{5}{8}$ inches long. $2\frac{3}{8}$-inch-long conical section tapering to 5.82-inch O.D.	Conical section tapered from 5.82-inch O.D. to 4-inch O.D. with length of $8\frac{5}{8}$ inches	↓
H	↓	↓	Annular ring injecting downstream

Fuel System

The hydrogen fuel was supplied in cylinders with total capacities of 420 pounds of hydrogen and a gas pressure of 2400 pounds per square inch gage. The fuel was taken from the cylinders through pressure-reducing valves, a metering orifice, and a throttle valve to the engine manifold. A gas analysis showed the fuel to be more than 99 percent pure.

4119

Instrumentation

The combustion air and spill air were measured by orifices in their respective ducts. The fuel-flow rate was measured by an orifice conforming to A.S.M.E. standards. The differential pressures were indicated on manometers, and the line pressures were indicated on Bourdon-tube gages.

Inlet and exhaust plenum pressures and static pressures at various stations along the engine and free-jet nozzle were indicated on manometers.

A self-balancing potentiometer recorded exhaust temperatures measured by 20 chromel-alumel thermocouples at station 9. Temperatures in the fuel and combustion air lines were read on nonrecording potentiometers.

Probes designed to measure total gas temperature and total pressure at a point were used to traverse the diameter of the duct at stations 7, 8, and 9. Static taps were located in the engine wall opposite these probes. Each probe consisted of a 1/8-inch-diameter total-pressure tube and a chromel-alumel thermocouple, both located in a 3/8-inch-diameter radiation shield. This shield was parallel with stream flow and open at either end, allowing flow over the thermocouple and total-pressure tube.

PROCEDURE

Free-jet operation of the facility required simply opening of the spill-air valve and maintaining a minimum pressure ratio across the system. Supersonic flow was thus established in the free nozzle, and a maximum of one-half of the air flow was captured by the engine. The remainder passed through and was metered in the spill-air line. Engine air flow was the difference in total air metered ahead of the plenum and spill air. Since the altitude exhaust capacity was quite limited, the required pressure ratio could be maintained only with the Mach 1.4 free-jet nozzle.

During connected-pipe operation the spill-air-line valve was closed, and all of the air flow metered by the combustion air orifice passed through the engine. The flow velocity through the free-jet nozzle was subsonic.

A ram-jet combustor and diffuser of conventional design in which air is diffused to a Mach number of approximately 0.2 before fuel is injected was investigated initially and the results were used for reference pur-
4119 poses. In subsequent configurations, fuel was introduced into a higher velocity airstream. This variation in air velocity at the fuel station was achieved by suitable manipulation of the cross-sectional area of the centerbody, rather than by actual movement of the fuel injectors. Flow blockage at station 2 is shown in the following table:

Configuration	Flow blockage at station 2, percent of total cross section	
	Fuel injector	Centerbody
A	10.1	18.8
B	10.1	18.8
C	10.5	41.0
D	10.5	41.0
E	10.5	41.0
F	11.2	27.9
G	11.2	35.5
H	11.2	35.5

With each configuration the data required to determine combustion efficiency, temperature and velocity profiles, and burner total-pressure loss were recorded.

Combustor Operating Conditions

The various configurations were compared at the following combustor operating conditions:

Inlet-air static pressure, in. Hg abs . . . . 30±2
Inlet-air total temperature, °F . . . . 220±10
Inlet-air velocity, ft sec . . . . 250 to 270

These values correspond approximately to the combustor-inlet conditions in a ram jet flying at a Mach number of 1.9 at a 40,000-foot altitude. Two of the combustor configurations were also operated at inlet-air temperatures of 100° and 400° F. Since a section of the burner immediately downstream of the fuel injectors was uncooled, maximum fuel-air ratio was limited by burner wall temperature. This limit varied somewhat with configuration, but the usual maximum was an equivalence ratio of 0.3.

Temperature and Velocity Surveys

Temperature and velocity surveys at stations 7, 8, and 9 were made with movable probes. The combustor-inlet conditions listed in the preceding section and the maximum fuel-air ratio that could be measured without damage to the survey instruments were held constant during the survey. Data were recorded at each position of the survey probe, at approximately 1-inch intervals across the combustor diameter. Wall static pressure was measured at each survey station and was assumed to be constant for that plane. From this static pressure and the total pressure measured at a given point, the local Mach number was determined. The local velocity was then readily calculated from this Mach number and the measured total temperature. Corrections to the total-temperature readings were assumed to be negligible, since a radiation shield was used.

Combustion Efficiency

Combustion efficiency was determined over a range of equivalence ratios with each configuration. Equivalence ratio is the metered fuel-air ratio divided by the stoichiometric fuel-air ratio of 0.0294 for hydrogen and air. The temperature of the exhaust gases at station 9 was measured with 20 bare-wire thermocouples. The method for correcting the thermocouple readings and for calculating combustion efficiency was similar to that of reference 3. Since the thermocouples were located upstream of the nozzle rather than at the nozzle throat as in reference 3, the radiation and recovery factors were slightly different for the two cases. An arithmetic average of the corrected temperatures at station 9 was used to determine the enthalpy rise of the exhaust products. Combustion efficiency was defined as the ratio of the enthalpy rise of the exhaust products to the heating value of the fuel used. Gaseous hydrogen at 75° F has a lower heating value of 51,571 Btu per pound.

Burner Pressure Loss

Burner-pressure-loss coefficient was defined as $(P_1 - P_9)/q_7$. The total pressure at the burner inlet P_1 was calculated from measured values of air flow and static and total temperatures at station 1; P_9 was obtained either by direct measurement or by the same method used to determine P_1; q_7 is a fictitious velocity pressure which would be realized at the maximum burner area if an isentropic expansion occurred between station 1 and the maximum burner cross section. Pressure-loss data were obtained both with cold air flow and burning for most of the configurations.

RESULTS

Eight combustor configurations were evaluated on the basis of uniformity of temperature and velocity of the combustion gases, total-pressure loss, and combustion efficiency as a function of burner length. These results were obtained with connected-pipe operation of the facility except where noted.

Temperature and Velocity Profiles

The temperature and velocities measured at stations 7, 8, and 9 with configuration A are shown in figure 4(a). Configuration A was considered representative of conventional designs for hydrogen fuel where the subsonic diffuser area ratio is sufficient to reduce the flow Mach number to a value near 0.2 ahead of the combustor. In all of the remaining configurations fuel was injected into a higher velocity airstream where the Mach number was 0.4 to 0.5. These profiles were quite typical parabolic distributions with the peaks at the center of the duct, and the average temperature increasing with combustor length. The regions of high temperature were also high-velocity regions, which resulted in a fairly uniform Mach number at a given station. Near the combustor wall gas temperatures were essentially the same as the inlet-air temperature, indicating a very low mixing rate. The ratio of maximum to minimum gas temperature was greater than 2. The use of vortex generators in the diffuser, configuration B, produced slightly flatter profiles (fig. 4(b)) than those measured with configuration A.

Configuration C represented the opposite extreme from the two preceding configurations in that no length was allowed for diffusion from a high to a low subsonic velocity. Fuel was injected into the uniform flow field before this sudden expansion. With configurations in which fuel is injected into a high-velocity airstream, the annular injector was believed to give better fuel distribution than the radial tube system. An initial test with configuration C, an annular fuel injector having 128 orifices, 3/64 inch in diameter, exhibited a poor temperature profile. Additional tests made with an injector having 96 of the same size orifices showed an improved temperature profile. Results for the 96-hole injector are shown in figure 4(c). Some flattening of the temperature profile over that found with configurations A and B was noted, and the mean temperatures at stations 7, 8, and 9 indicated that combustion was completed at station 7. However, burning stability at lean fuel-air ratios was poor with this injector, and succeeding tests were conducted with the 128-hole injector.

The addition of a conical section to the centerbody, configuration D, provided a more gradual transition in flow area downstream of the fuel-injection station. Since with hydrogen fuel flame is readily seated on the fuel injectors, a simultaneous heat addition and diffusion process occurred. The temperatures and velocities measured with configuration D are presented in figure 4(d). Excellent temperature profiles were obtained with a variation of $\pm 150^{\circ}$ in 2100° R at station 7 where burning was complete. Similar results, shown in figure 4(e), were obtained with configuration D in free-jet tests. Engine operation was smooth with supercritical diffuser flow.

4119

From the standpoint of temperature profiles, little improvement could be expected from further modifications. However, as is discussed later, the total-pressure loss with configuration D was higher than with configurations A and B. The following discussion reviews the results of attempts to obtain a combination of a uniform profile and a low total-pressure loss.

Two methods for reducing pressure loss were tried; one consisted of reducing the Mach number at the point of fuel injection and the other reducing the heat release in the diverging part of the combustor. The Mach number at the fuel injectors was reduced by increasing the flow area at the point of fuel injection. The heat release in the diverging part of the combustor was reduced by two methods; one was to increase the diffusion angle and the other was to inject the fuel in a downstream direction. The diffuser angle was increased in configuration E; the diffuser angle was increased and the fuel injectors were located in a larger annular area in configuration F; and the fuel injectors were relocated in configuration G. Fuel was injected downstream in configuration H. The temperature and velocity profiles with these configurations are given in figures 4(f) to (i). The temperature surveys with these configurations at station 9 gave the following results. With configuration E the variation in temperature was $\pm 110^{\circ}$ F; with F, $\pm 200^{\circ}$ F; with G, $\pm 60^{\circ}$ F; and with H, $\pm 460^{\circ}$ F at a mean temperature of about 1800° R.

Pressure Loss

The total-pressure-loss coefficient $(P_1 - P_9)/q_7$ across the various combustor configurations is shown as a function of combustor temperature ratio in figure 5. The total pressures were determined at stations 1 and 9, and the dynamic pressure q was based on the maximum burner area.

The pressure losses for configurations A and B were lower than for the others. Configuration D, which gave uniform exhaust temperature profiles, had a pressure-loss coefficient of 9.8. However, with the exception of configuration E all of the cold-flow coefficients were moderate. With this configuration the cold-flow coefficient increased from 4.5 to 7.4 with the addition of a typical value of fuel rate. The volume of fuel-air mixture before combustion increased by 14 percent. It is particularly interesting to note the trend in pressure loss with temperature ratio across the combustor (configuration G, e.g.). The two limit curves represent the maximum and minimum losses possible assuming all of the heat was released in either the minimum or maximum diffuser area. At low fuel-air ratios the pressure-drop data follow the minimum curve but shift towards the maximum with richer operation. This shift indicates that the quantity of fuel burned in the minimum diffuser area increased with increased fuel-air ratio. The effect of fuel-injection method is seen by comparing the data of configurations G and H. Normal fuel injection gave higher losses than downstream injection.

Combustion Efficiency

The burner-inlet conditions at which these tests were conducted were quite favorable for combustion. Pressure was 1 atmosphere or greater, temperature was 100° to 400° F, and velocity was 240 to 300 feet per second. It was desired, however, to operate at lean fuel-air mixtures, which according to reference 4 have essentially zero flame speed. Obviously, if high combustion efficiencies are to be obtained at over-all lean fuel-air ratios, burning must proceed rapidly enough so that the locally rich mixture at the injector is burned before it is diluted. The combustion efficiencies shown in figure 6 can be considered in this light. Fuel injected normal to the airstream (configuration G) mixed more rapidly than that injected downstream (configuration H). Consequently, combustion efficiencies dropped off more rapidly as the fuel-air mixture became leaner with normal than with downstream injection.

An even more rapid loss in efficiency was observed with configuration A, which also had normal fuel injection. A possible explanation for this was that the fuel injectors were located in a region where the air flow was nonuniform and locally aggravated the mixture dilution.

At equivalence ratios greater than 0.25 combustion efficiencies of 90 to 100 percent were obtained with all the configurations tested.

DISCUSSION

Evaluation of a combustor configuration must consider the combined effects of combustion efficiency, pressure loss, temperature stratification, and engine length on over-all engine performance. Since a ram-jet engine burning hydrogen would probably find application in a long-range missile, the influence of some of these parameters is considered in reference 5. The conclusions of that study are the basis for analysis of the experimental results. Moreover the authors of reference 5 supplied some unpublished relations that made possible an estimate of the effect of temperature stratification on range.

Very briefly, the various factors influencing range were estimated in the following manner. Range is directly proportional to combustion efficiency. An isothermal pressure-loss coefficient of 1 is equivalent 4119
to a 1-point loss in combustion efficiency. For the case of burning in the diffuser, the pressure drop above the minimum was considered an increase in the isothermal pressure-loss coefficient. The exhaust-gas temperature-stratification effect was estimated as a 2-percent loss for a parabolic profile where the ratio of maximum to minimum temperature was about 2, and there was no loss for the flat-profile case. The engine-weight parameter was calculated as follows. An increase in engine weight from 0 to 150 pounds per square foot reduced range 9 percent. However, since the supersonic diffuser and the exhaust nozzle were estimated to represent one-half of the engine weight, eliminating the subsonic diffuser and the combustor would increase range less than 5 percent. For a constant-angle subsonic diffuser, the length required to diffuse from sonic velocity to a Mach number of 0.2 is $4\frac{1}{2}$ times the length required for diffusion from sonic to 0.5 sonic speed. Therefore, the simplification was made that the first part of the subsonic diffuser was weightless. It should be noted that the range comparison is specifically for the conditions covered in reference 5. The rating of the various combustors varies with assigned operating conditions. However, the comparison presented, in addition to being illustrative, provides a comparison at fairly typical ram-jet conditions.

The net effect on range is the product of the combustion efficiency η, pressure-loss factor $\mathscr{P}$, engine-weight factor $\mathscr{W}$, and temperature-stratification factor $\mathscr{T}$. For example, with configuration A at an equivalence ratio of 0.15, combustion efficiency was 82 percent; the cold-flow pressure-loss coefficient, $\Delta P/q$ was 0.5 and therefore $\mathscr{P}$ was 0.995; the temperature-stratification loss was 2 percent, which gave a $\mathscr{T}$ of 0.98; and the engine-weight parameter $\mathscr{W}$ was 0.91. The range parameter was the product of these values and equalled 0.74 for this case and 0.86 for configuration H. Similar estimates for other configurations and equivalence ratios were made, and the range parameters are plotted in figure 7. For lean operation, configuration H was best mainly because combustion efficiency was the highest (97 percent at an equivalence ratio of 0.15). At an equivalence ratio of 0.3, the range of all three configurations was the same, but at slightly leaner operation configuration G was the best.

It is important to note that configuration G is best suited for operation over a limited range of fuel-air ratio, but within these limits its range potential is the greatest. The reduction in over-all engine length with this configuration was estimated as 50 percent of the conventional engine design. Since a long-range ram-jet engine cruises at a fixed temperature ratio across the combustor, a method for designing a configuration such as G for operation at a given fuel-air ratio is discussed in the following sections.

4119

ANALYSIS AND DESIGN APPLICATION

Analysis

It has been shown from a range standpoint that merging of the subsonic diffuser and combustor is desirable, particularly for engines designed to operate over a narrow range of flight Mach number and fuel-air ratio. Since decelerating flow is conducive to flow separation, this design problem may be solved by maintaining a constant gas velocity in the diffuser by the addition of the right amount of heat. To achieve this flow condition the relation between temperature rise and area change must be determined. Also, experimental data for heat-release rate as a function of inlet conditions, fuel-air ratio, and fuel injector geometry are required. The relation between temperature and area ratio for constant gas velocity and the methods used to arrive at total temperature from static-pressure measurements are discussed in appendixes A and B, respectively.

A typical plot of total temperature along the axis of the combustor as calculated from static-pressure measurements is shown in figure 8. With fuel injection normal to the airstream, the temperature rose rapidly in the first half of the burner, and little rise was noted in the downstream half. The average temperature measured at the combustor exit is 1720° R compared with a calculated temperature of 2070° R. In temperature surveys made with configuration G and shown in figure 4(h), no measurable rise in temperature was noted between stations 7 and 9. A comparison of measured temperature at the combustor exit with calculated temperature at station 7 shows good agreement, and it appears that the further calculated temperature rise is due to friction pressure drop past station 7.

The temperature rise between stations 1 and 6 as calculated from wall static-pressure measurements is shown in figure 9. The variables investigated were fuel-air ratio, inlet-air temperature, and fuel-injection method. At this test condition inlet-air temperature did not influence the rate of heat release.

Temperature rise increased with increasing fuel-air ratio for both normal and downstream injection, but the rate of increase was much greater with normal than with downstream injection. From these results it was concluded that at these operating conditions mixing of the fuel and air rather than chemical-reaction rate controls the rate of heat release.

Design Application

The design application is based upon a method of keeping the velocity constant in the terminal part of the subsonic diffuser where the 4119
Mach number is 0.6 or less. As discussed in the previous section, the criterion of constant velocity in the diffuser gives a reasonable compromise between pressure loss and temperature profile.

Constant velocity in the diffuser is achieved by letting

$$\frac{T_x}{T_2} \approx \frac{A_x}{A_2}$$

This can be seen in the continuity expression if incompressible flow is assumed or in the numerical examples in appendix A in which compressible flow relations are used.

In order to select a rate of area change with length, it is necessary to know the rate of change of temperature with length. The average rate of change of temperature ratio can be found from the experimental data. The average rate of change of temperature ratio is approximately equal to $\frac{T_6}{T_2} \frac{V_m}{(L_6 - L_2)}$, where V_m is the arithmetic mean velocity, station 2 is the point of fuel injection, and station 6 is the point of measurement of outlet temperature.

For constant velocity to exist in the diffuser, A_6/A_2 must be approximately equal to T_6/T_2, or τ, and so it may be written that

$$\frac{A_6}{A_2} \frac{V}{L} \approx \tau \frac{V_m}{L_6 - L_2}$$

A plot of this parameter is shown in figure 10 for two methods of fuel injection. The effects on rate of heat release discussed in connection with figure 9 are apparent in figure 10 also.

The parameter $V_m A_6/LA_2$ can then be used in designing a constant-velocity flow region. For operation at a fixed fuel-air ratio, as is

usually assumed for a long-range ram-jet cruise, the normal fuel-injection design has a greater range advantage than downstream injection. From considerations such as these, a value of $V_m A_6/LA_2$ is determined. The gas velocity V_m and diffuser area ratio A_6/A_2 are calculated at the fuel-injection station, and the diffuser length L required can then be obtained. If increased thrust is required, additional fuel can be added downstream of the diffuser. It may be possible to apply the design
4119 parameter $V_m A_6/LA_2$ to lower and higher values of velocity V_m than those (540 to 820 ft/sec) covered in this investigation.

CONCLUSIONS

Shortening ram-jet engines by merging of the subsonic diffuser and the combustor appears feasible with a high-energy fuel such as hydrogen. When this technique was used, the temperature profile varied 7 percent at an average combustion temperature of 2100° R. This profile was obtained with a total-pressure-loss coefficient of 9.8.

Heat-release rate, which determines the total-pressure loss, is influenced by the method of fuel injection. Injection normal to the airstream gives a higher heat-release rate, more uniform temperature profile, and higher total-pressure loss than downstream injection. Both systems gave efficiencies of 95 to 100 percent at equivalence ratios above 0.25; however, at an equivalence ratio of 0.1 downstream injection gave combustion efficiencies 20 percent higher than normal injection. Reasonable total-pressure loss and temperature profiles may be obtained by keeping velocity constant in the diffuser.

It was estimated that engine length could be reduced by 50 percent by merging the subsonic diffuser and combustor, which would result in a 3-percent gain in the range of a ram-jet missile. Further gains with the system are due to improved temperature profile and higher combustion efficiency. However, increased pressure drop reduced range slightly, and the net gain in range was estimated as 16 percent.

Lewis Flight Propulsion Laboratory
National Advisory Committee for Aeronautics
Cleveland, Ohio, July 30, 1956

APPENDIX A

FLOW ANALYSIS WITH SIMULTANEOUS HEAT ADDITION AND DIFFUSION

A calculation of the required temperature rise to maintain constant velocity throughout a diverging combustor for various inlet Mach numbers can be made once the flow area is known as a function of length.

The following flow assumptions are made: 4119

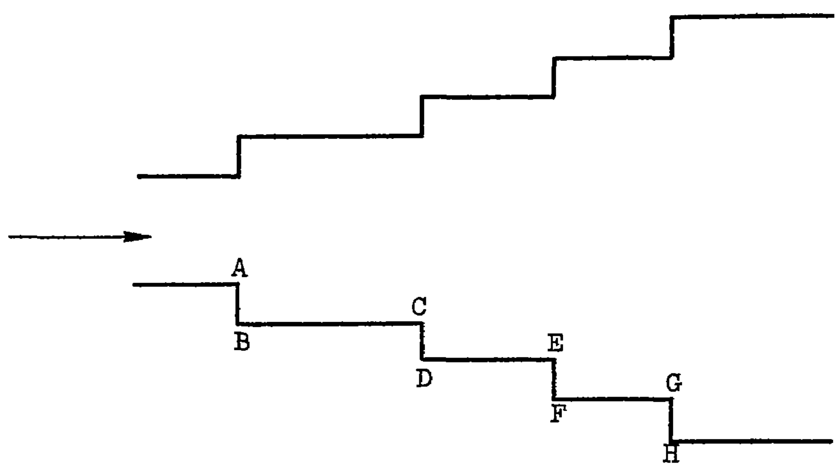
(1) All flow is one-dimensional.

(2) No viscous losses or frictional effects exist.

(3) The gas constant, specific-heat ratio, and mass flow are constant.

(4) No momentum effects of fuel addition exist.

The diffuser flow process is assumed to occur in a series of steps, each involving an isentropic expansion and a constant-area temperature rise. Sufficient heat addition at constant area is used to regain the velocity lost in the expansion. The following is a sketch of the assumed step diffuser, with various stations noted:



The isentropic relations applied to the expansion step are:

$$\frac{A_A}{A_B} = \frac{M_B}{M_A}\left(\frac{1 + \frac{\gamma - 1}{2} M_A^2}{1 + \frac{\gamma - 1}{2} M_B^2}\right)^{\frac{\gamma+1}{2(\gamma-1)}}$$

4119

$$P_A = P_B$$

$$\frac{t_A}{t_B} = \left(\frac{1 + \frac{\gamma - 1}{2} M_B^2}{1 + \frac{\gamma - 1}{2} M_A^2}\right)$$

The relations applied for heat addition in the constant-area step (following table) are:

$$\frac{T_C}{T_B} = \frac{M_C^2}{M_B^2}\left(\frac{1 + \gamma M_B^2}{1 + \gamma M_C^2}\right)^2\left(\frac{1 + \frac{\gamma - 1}{2} M_C^2}{1 + \frac{\gamma - 1}{2} M_B^2}\right)$$

$$\frac{P_B}{P_C} = \left(\frac{1 + \gamma M_C^2}{1 + \gamma M_B^2}\right)\left(\frac{1 + \frac{\gamma - 1}{2} M_B^2}{1 + \frac{\gamma - 1}{2} M_C^2}\right)^{\frac{\gamma}{\gamma-1}}$$

$$\frac{t_C}{t_B} = \frac{M_C^2}{M_B^2}\left(\frac{1 + \gamma M_B^2}{1 + \gamma M_C^2}\right)^2$$

The ratio of Mach numbers at points A and C becomes

$$\frac{M_A}{M_C} = \sqrt{\frac{t_C}{t_A}}$$

These calculations of the total-temperature rise and expected total-pressure loss were made for a range of combustor-inlet to exit-area ratios and are presented in the following table for a burner-inlet Mach number of 0.4 and equal velocities at stations A, C, E, G, I, K, M, and P. It may be noted that configuration D had a subsonic diffuser area ratio of 1.37 and from the table it may be seen that a temperature ratio of 1.32 is necessary to maintain constant velocity in the diffuser.

Station	T_x/T_A	A_x/A_A	M_x	P_x/P_A	Process used to get from previous station
A	1.0	1.0	0.40	1.0	
B	1.0	1.2	.32	1.0	Isentropic area change
C	1.2	1.2	.36	.984	Constant-area heat addition
D	1.2	1.37	.32	.984	Isentropic area change
E	1.37	1.37	.34	.975	Constant-area heat addition
F	1.37	1.61	.29	.975	Isentropic area change
G	1.60	1.61	.31	.972	Constant-area heat addition
H	1.60	1.83	.28	.972	Isentropic area change
I	1.80	1.83	.30	.965	Constant-area heat addition
J	1.80	2.01	.27	.965	Isentropic area change
K	2.00	2.01	.28	.960	Constant-area heat addition
L	2.00	3.00	.19	.960	Isentropic area change
M	3.00	3.00	.23	.945	Constant-area heat addition
N	3.00	4.05	.17	.945	Isentropic area change
P	4.00	4.05	.20	.932	Constant-area heat addition

4119

APPENDIX B

CALCULATION OF TOTAL TEMPERATURE FROM STATIC PRESSURE

The total temperatures at stations 3, 4, 5, 6, 7, and 9 were calculated from static pressures at each of these stations. These pressures were measured under burning conditions and calculated for isothermal flow at the same inlet Mach number. A particular station calculation of temperature consisted of two steps:

(1) Calculation of a fictitious static pressure and Mach number based on isothermal flow: Configurations G and H were run using typical values of fuel and air flow, but with no burning. Isothermal losses were determined across the fuel injectors and over the abrupt area change at the downstream end of the centerbody. These losses, expressed as $\Delta P/q$, were applied to combustor-inlet conditions that existed during normal burning operation. Static pressures and Mach numbers were calculated at various axial stations.

(2) Calculation of total temperature under burning conditions: During the burning runs with configurations G and H, static-pressure measurements were made at stations 1, 3, 4, 5, 6, 7, and 9. The relation of the hypothetical flow conditions, as calculated in step (1), to the actual flow conditions at any axial station is given by the constant-area heat-addition flow relation. Therefore, the total temperature at each axial station may be determined.

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1. Olson, Walter T., and Gibbons, Louis C.: Status of Combustion Research on High-Energy Fuels for Ram Jets. NACA RM E51D23, 1951.

2. Disher, John H., and Jones, Merle L.: Flight Investigation of Pentaborane Fuel in 9.75-Inch-Diameter Ram-Jet Engine with Downstream Fuel Injection. NACA RM E55G01, 1955.

3. Dangle, E. E., and Kerslake, William R.: Experimental Evaluation of Gaseous Hydrogen Fuel in a 16-Inch-Diameter Ram-Jet Engine. NACA RM E55J18, 1955.

4. Jost, Wilhelm: Explosion and Combustion Processes in Gases. McGraw-Hill Book Co., Inc., 1946.

5. Weber, Richard J., and Luidens, Roger W.: Analysis of Ram-Jet Engine Performance Including Effects of Component Changes. NACA RM E56D20, 1956.

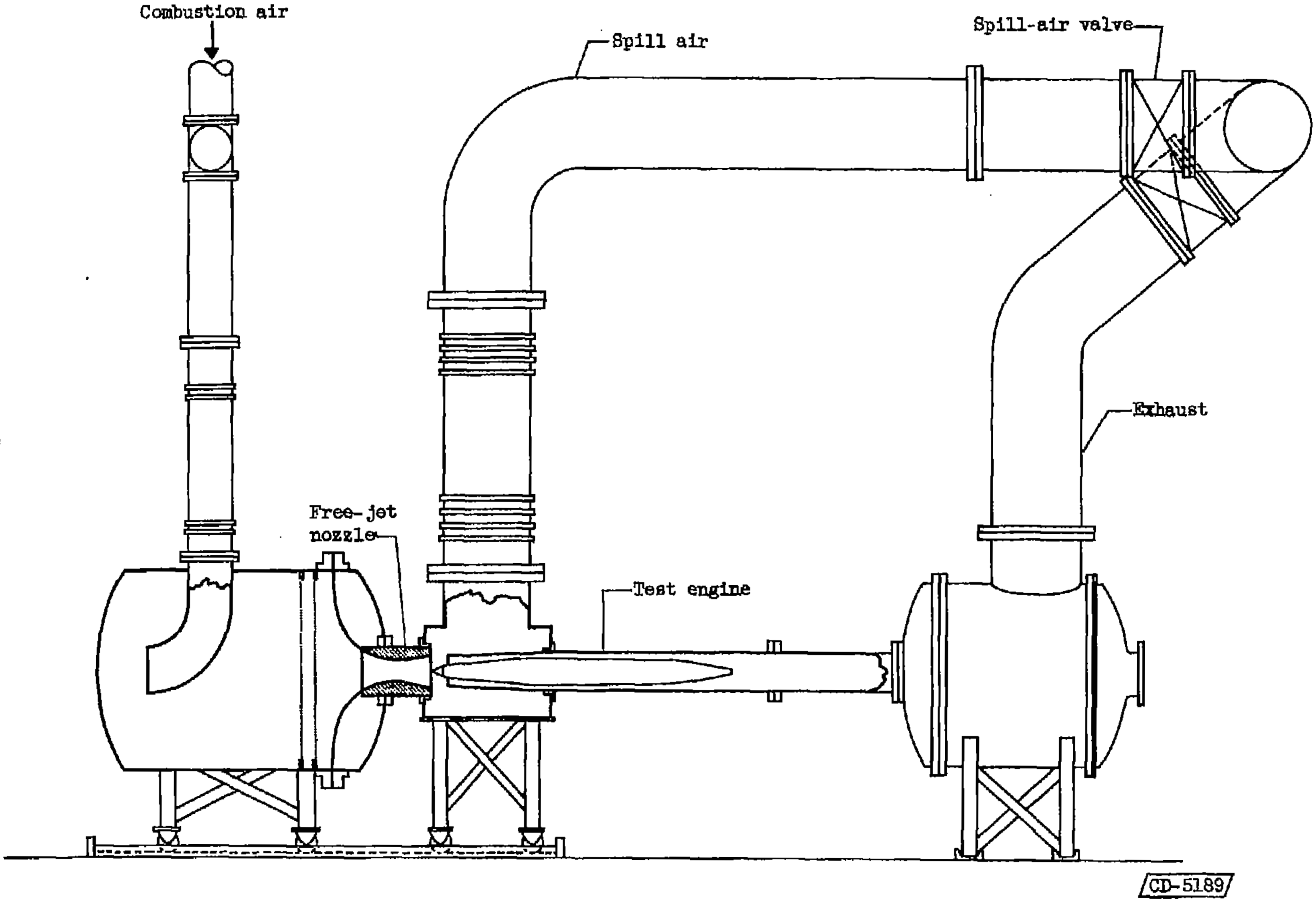


Figure 1. - Installation of ram-jet engine in free-jet test facility.

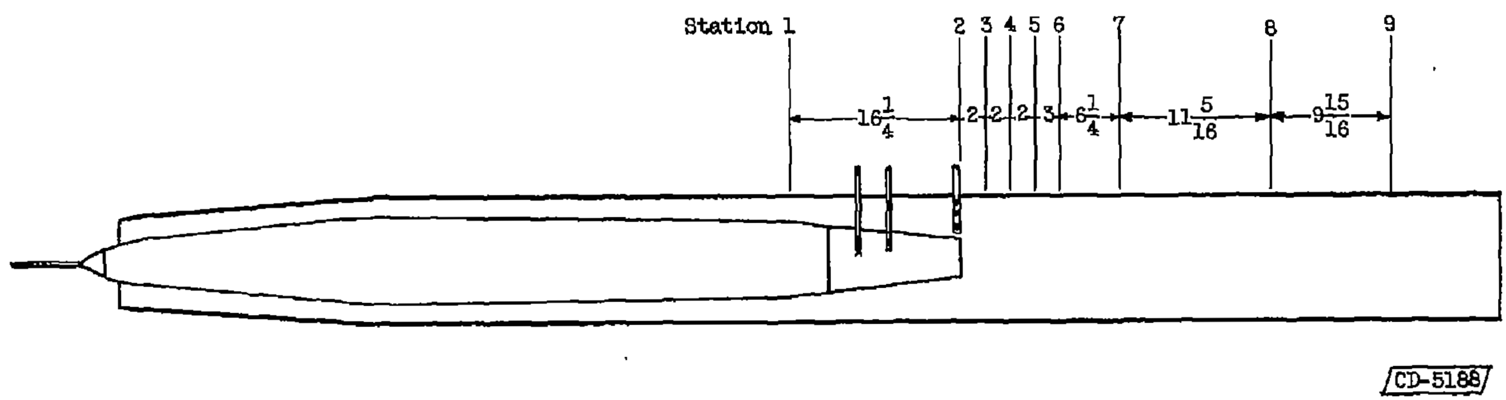
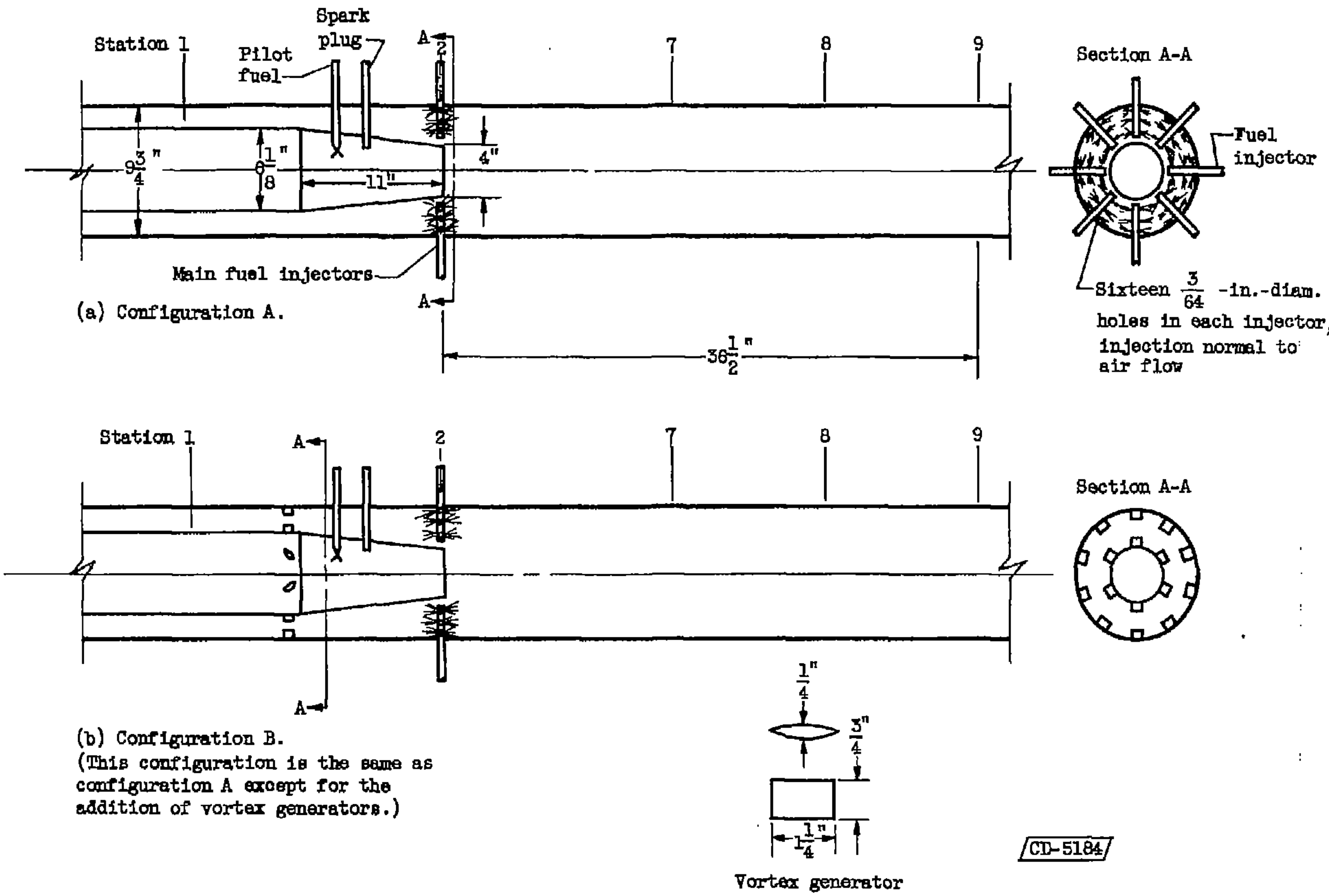


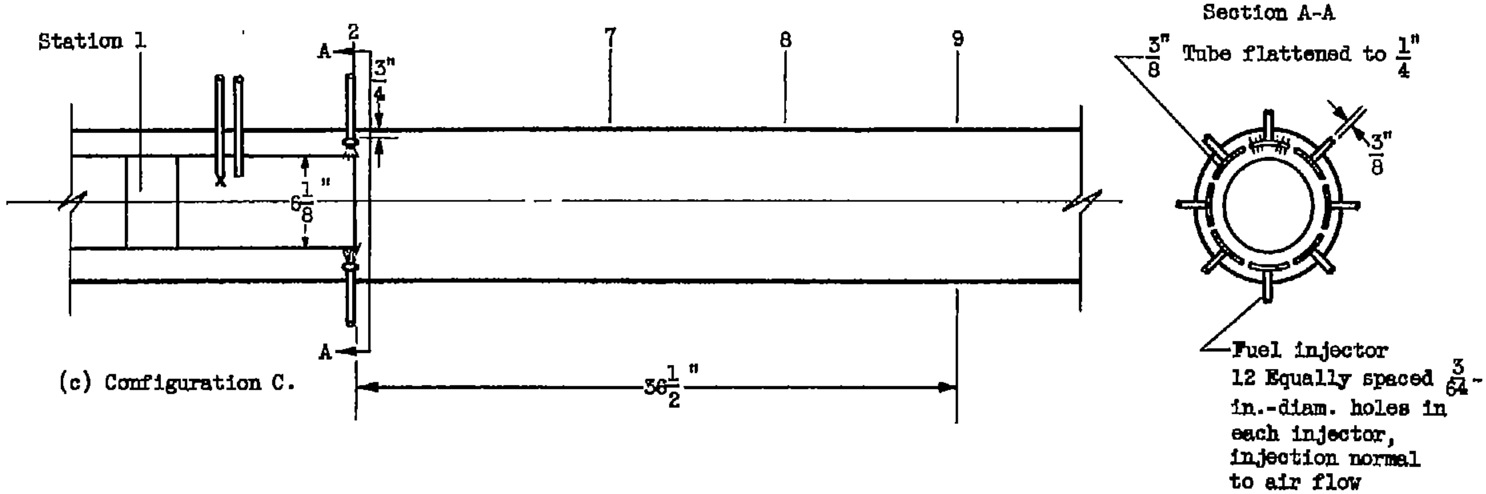
Figure 2. - $9\frac{3}{4}$ -Inch-diameter ram-jet engine. (Dimensions in inches.)



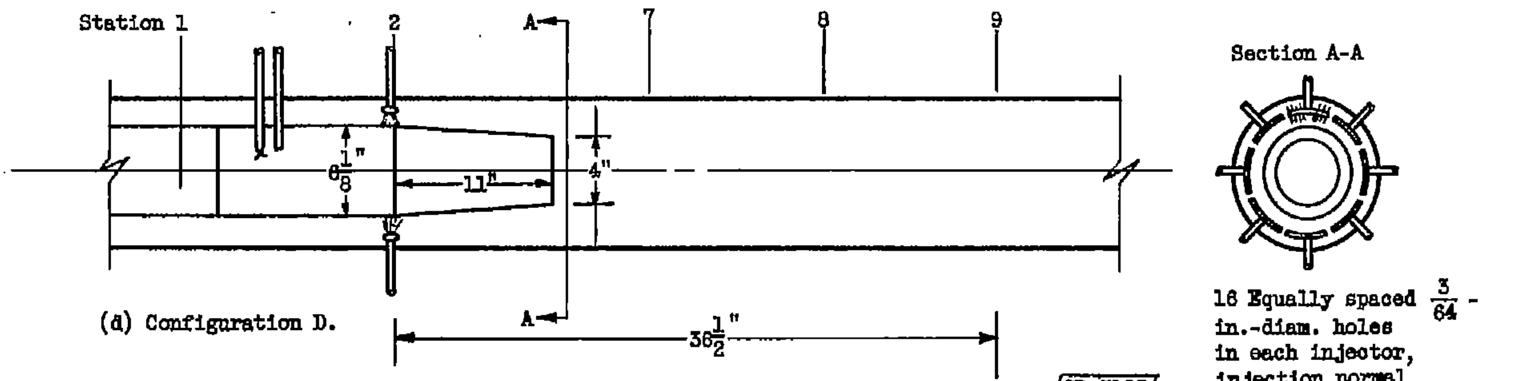
(a) Configuration A.

(b) Configuration B.
(This configuration is the same as configuration A except for the addition of vortex generators.)

Figure 3. - Combustor configurations.

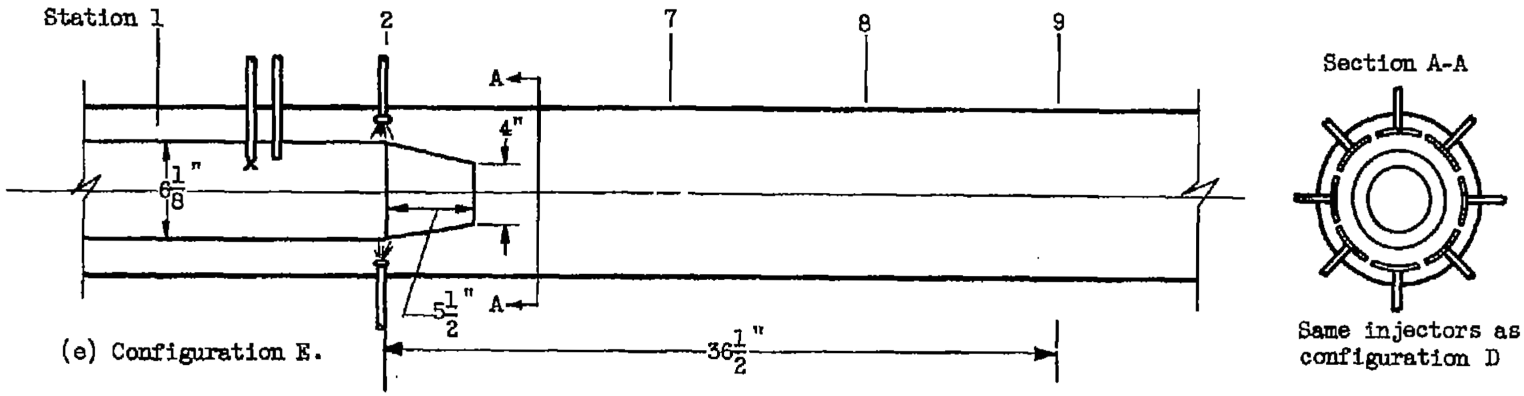


(c) Configuration C.

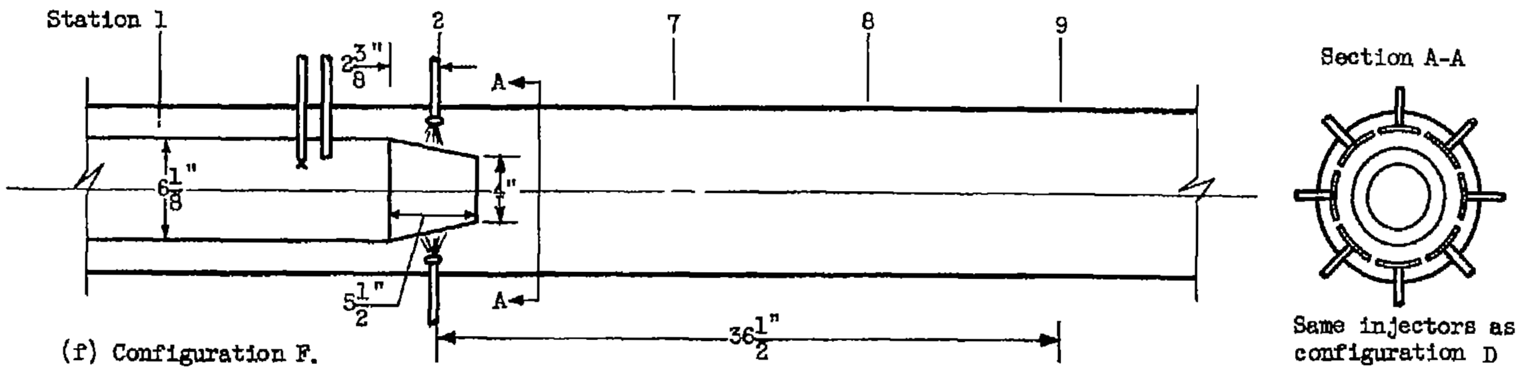


(d) Configuration D.

Figure 3. - Continued. Combustor configurations.



(e) Configuration E.



(f) Configuration F.

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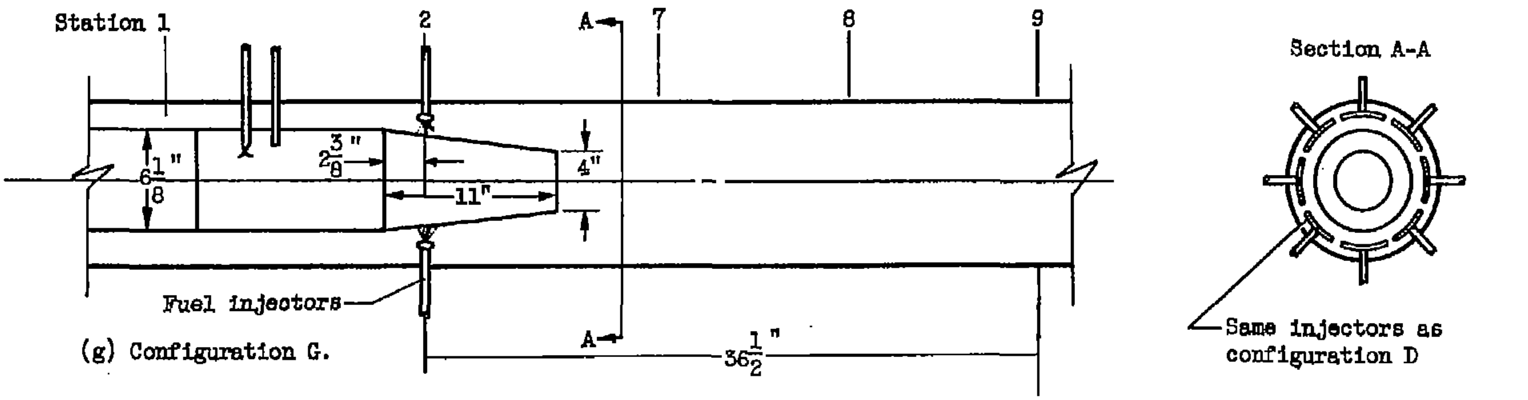
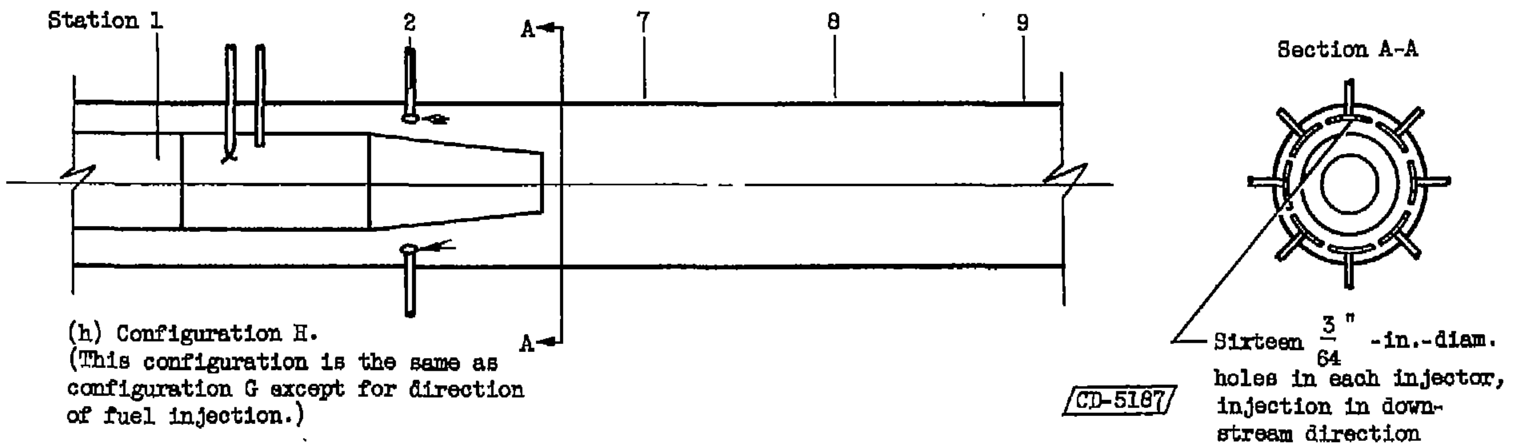
Figure 3. - Continued. Combustor configurations.

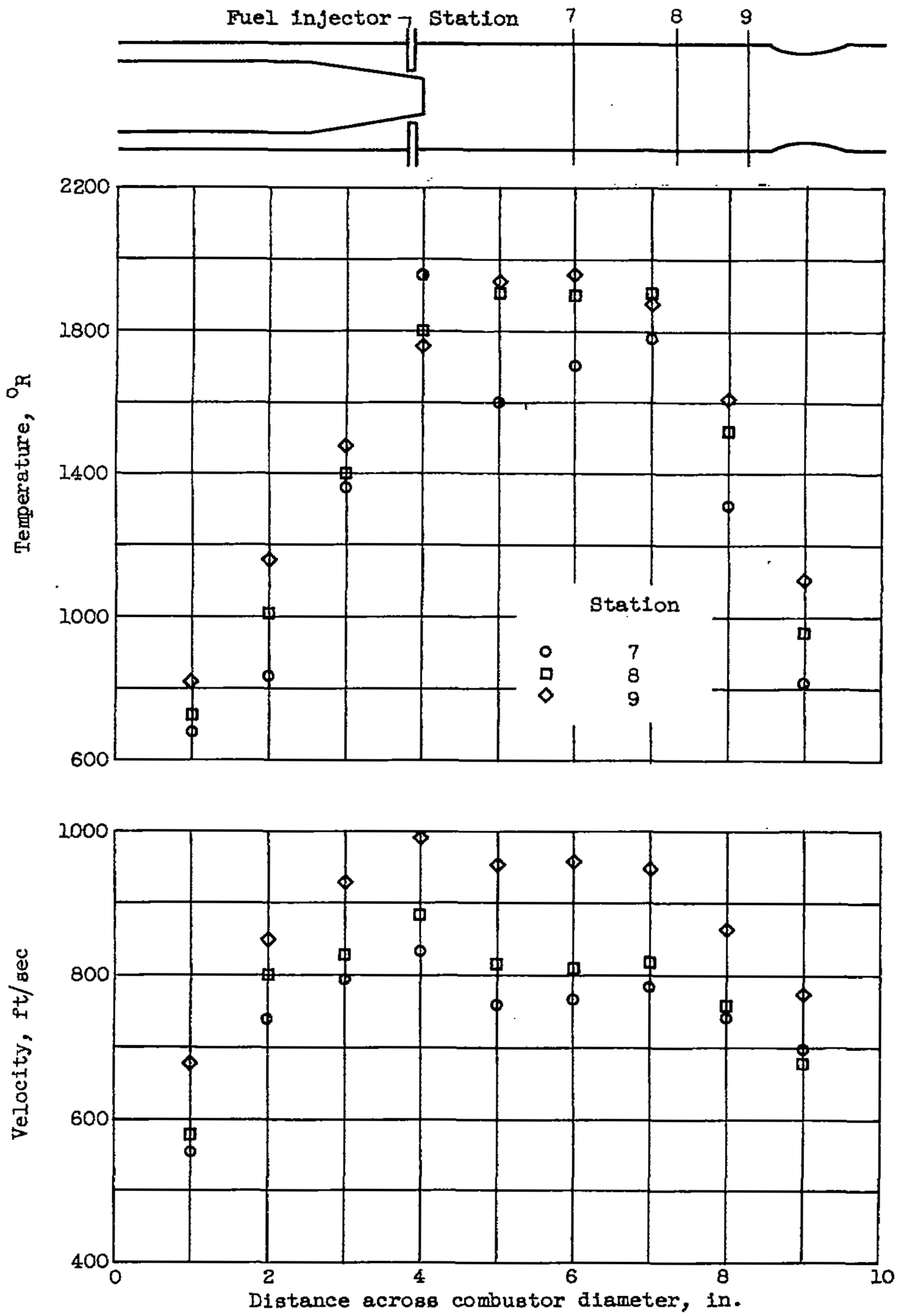


(g) Configuration G.



(h) Configuration H.
(This configuration is the same as configuration G except for direction of fuel injection.)

Figure 3. - Concluded. Combustor configurations.



(a) Configuration A.

Figure 4. - Temperature and velocity profiles in combustion zone.

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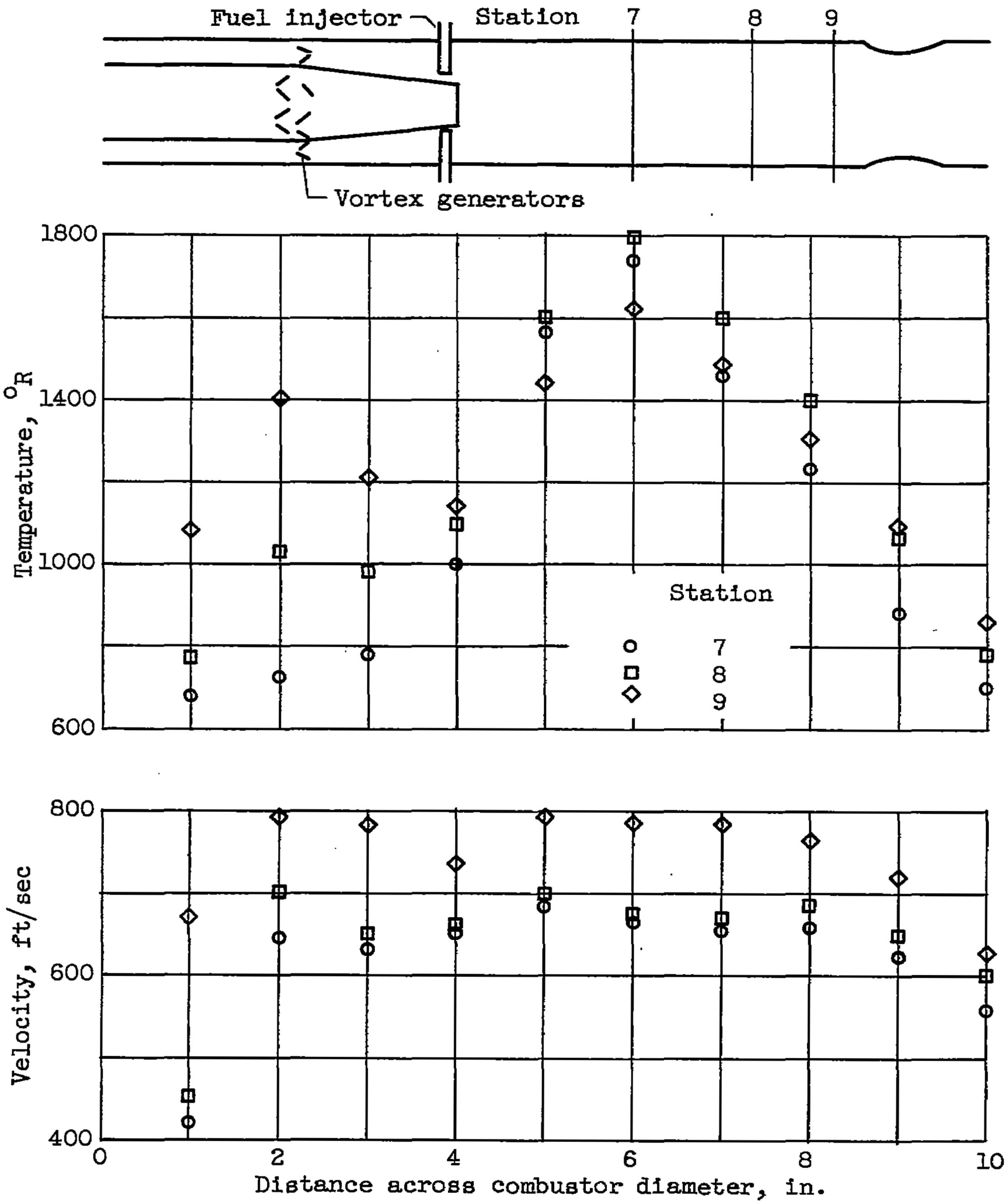
CD-4 back



(b) Configuration B.

Figure 4. - Continued. Temperature and velocity profiles in combustion zone.

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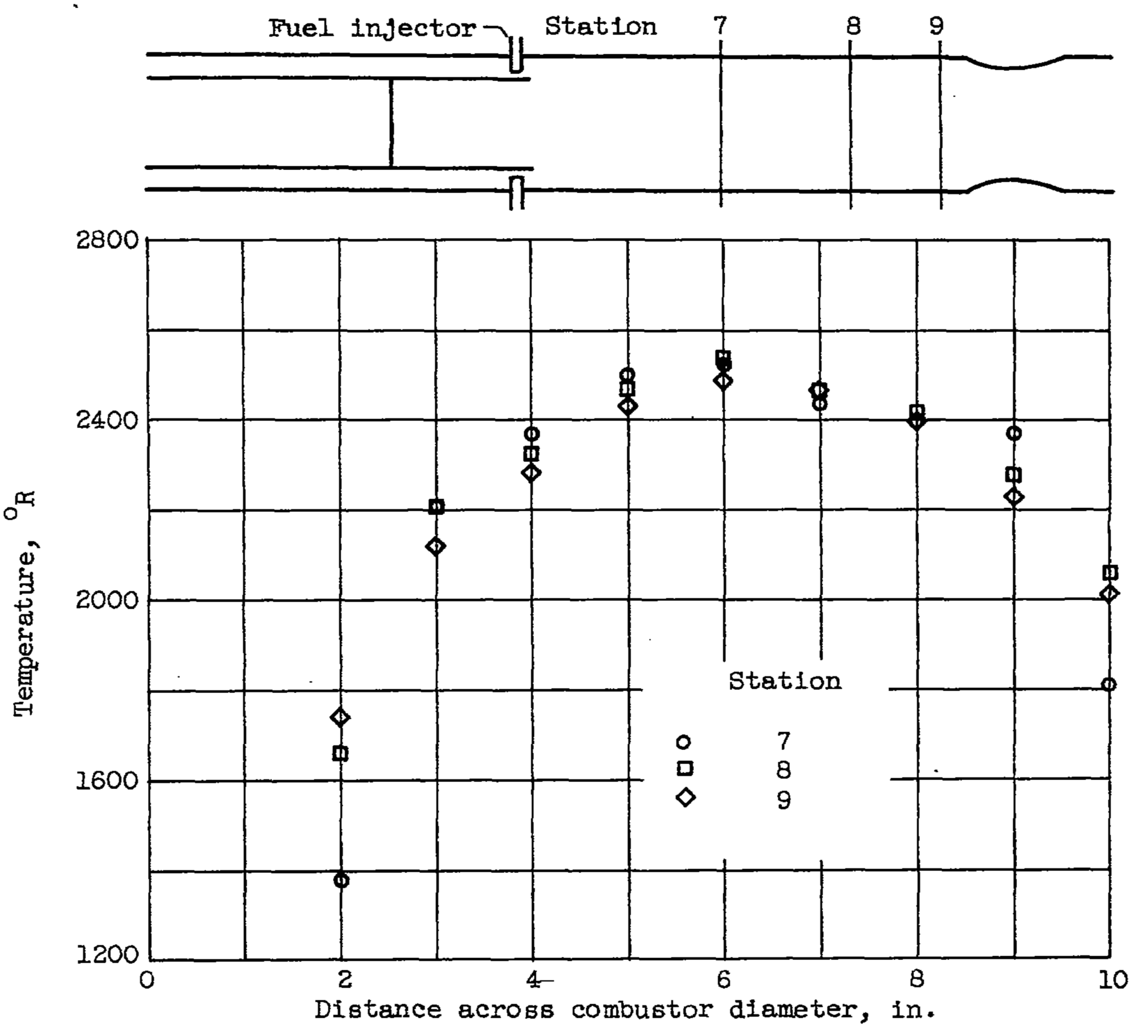

(c) Configuration C.

Figure 4. - Continued. Temperature and velocity profiles in combustion zone.

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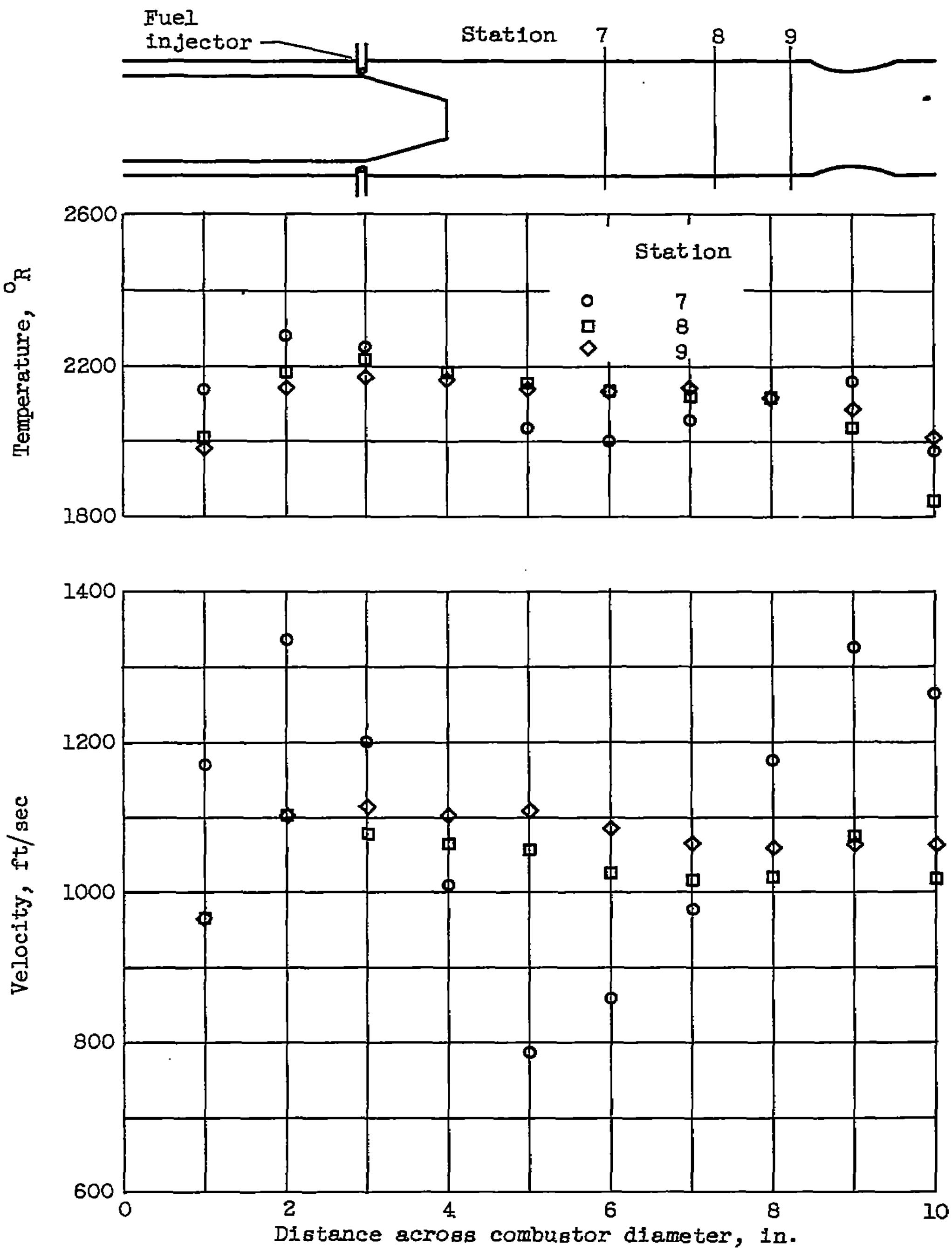

(d) Configuration D.

Figure 4. - Continued. Temperature and velocity profiles in combustion zone.

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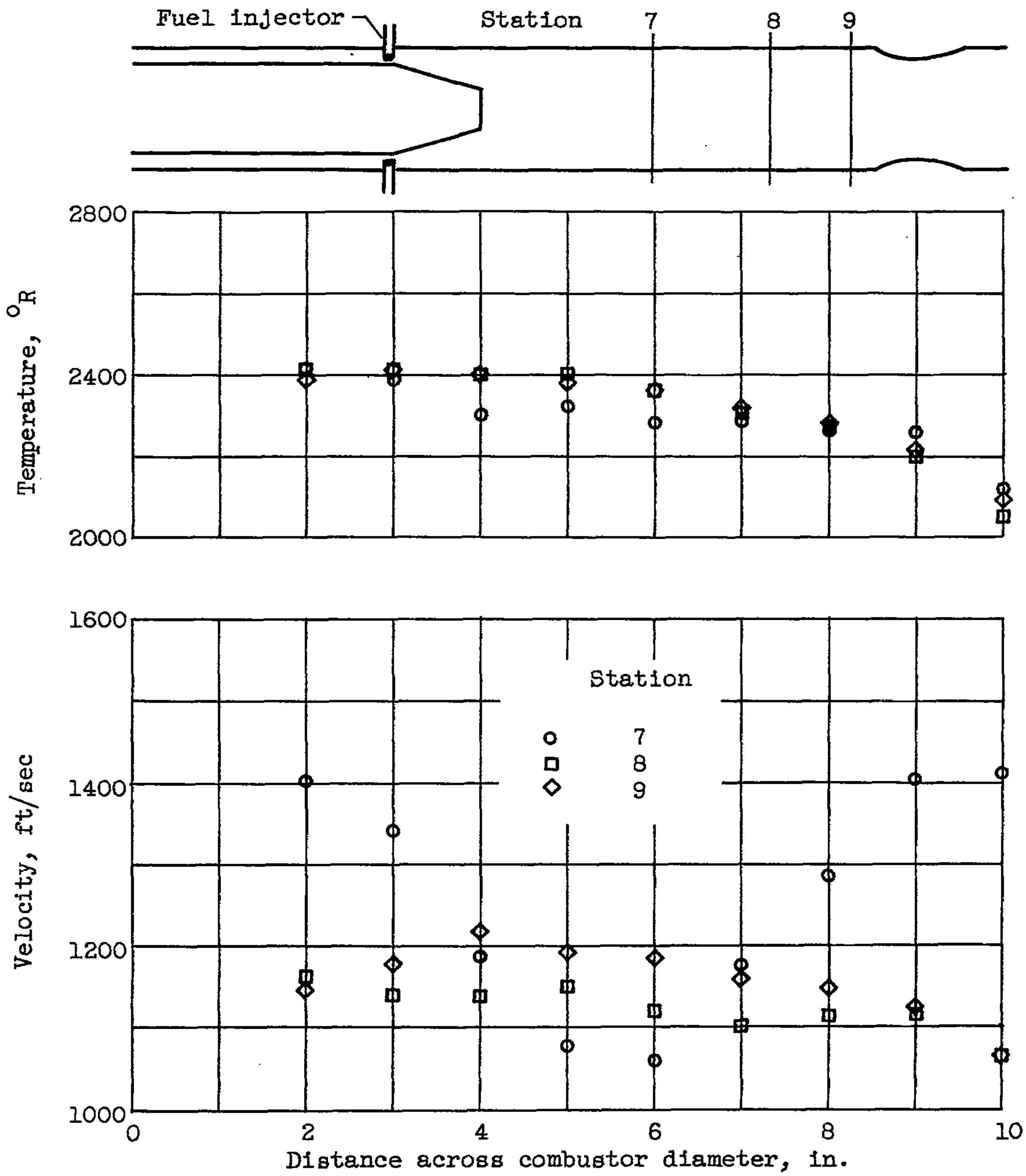

(e) Configuration D with free-jet operation.

Figure 4. - Continued. Temperature and velocity profiles in combustion zone.

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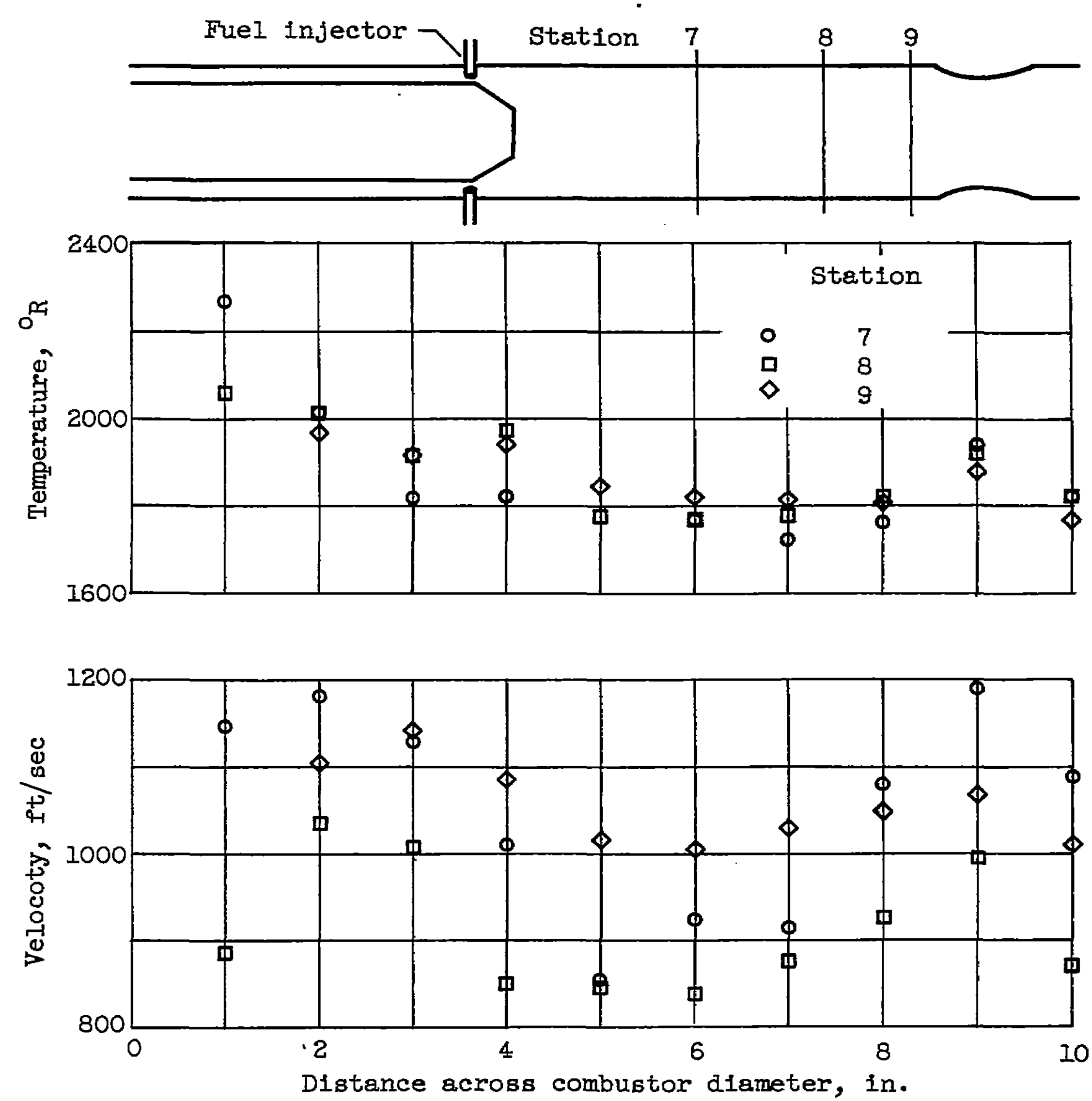

(f) Configuration E.

Figure 4. - Continued. Temperature and velocity profiles in combustion zone.

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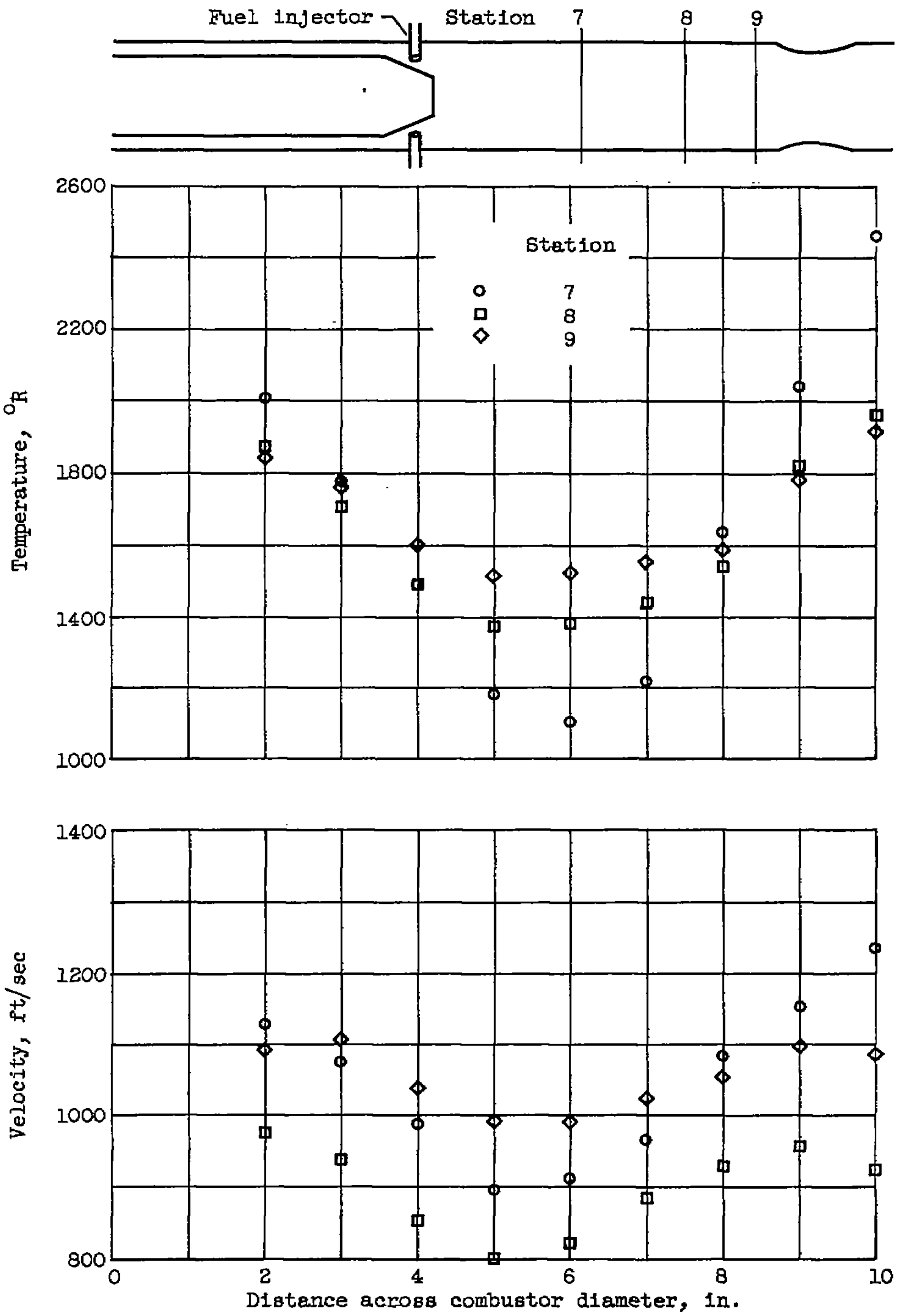

(g) Configuration F.

Figure 4. - Continued. Temperature and velocity profiles in combustion zone.

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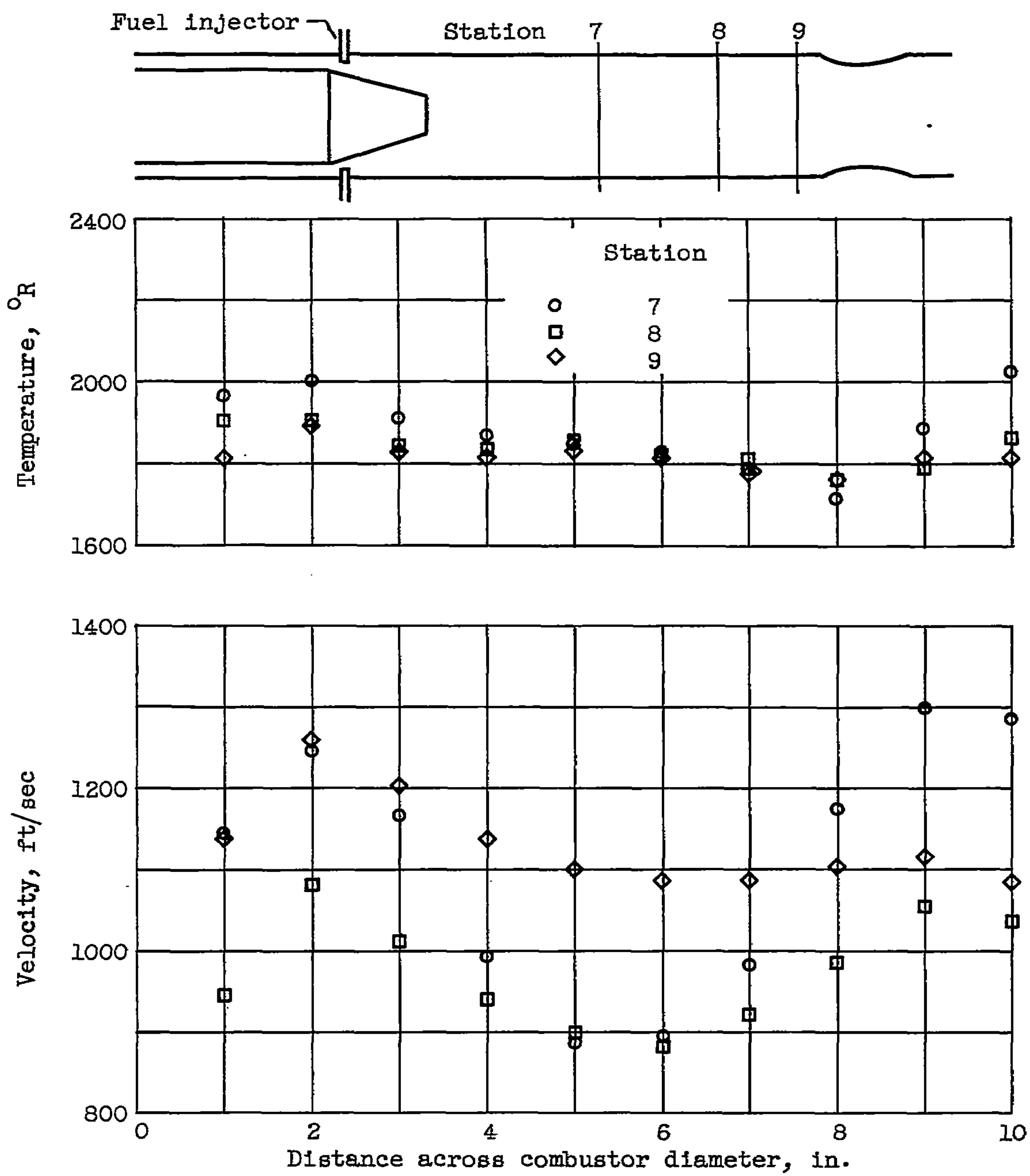

(h) Configuration G.

Figure 4. - Continued. Temperature and velocity profiles in combustion zone.

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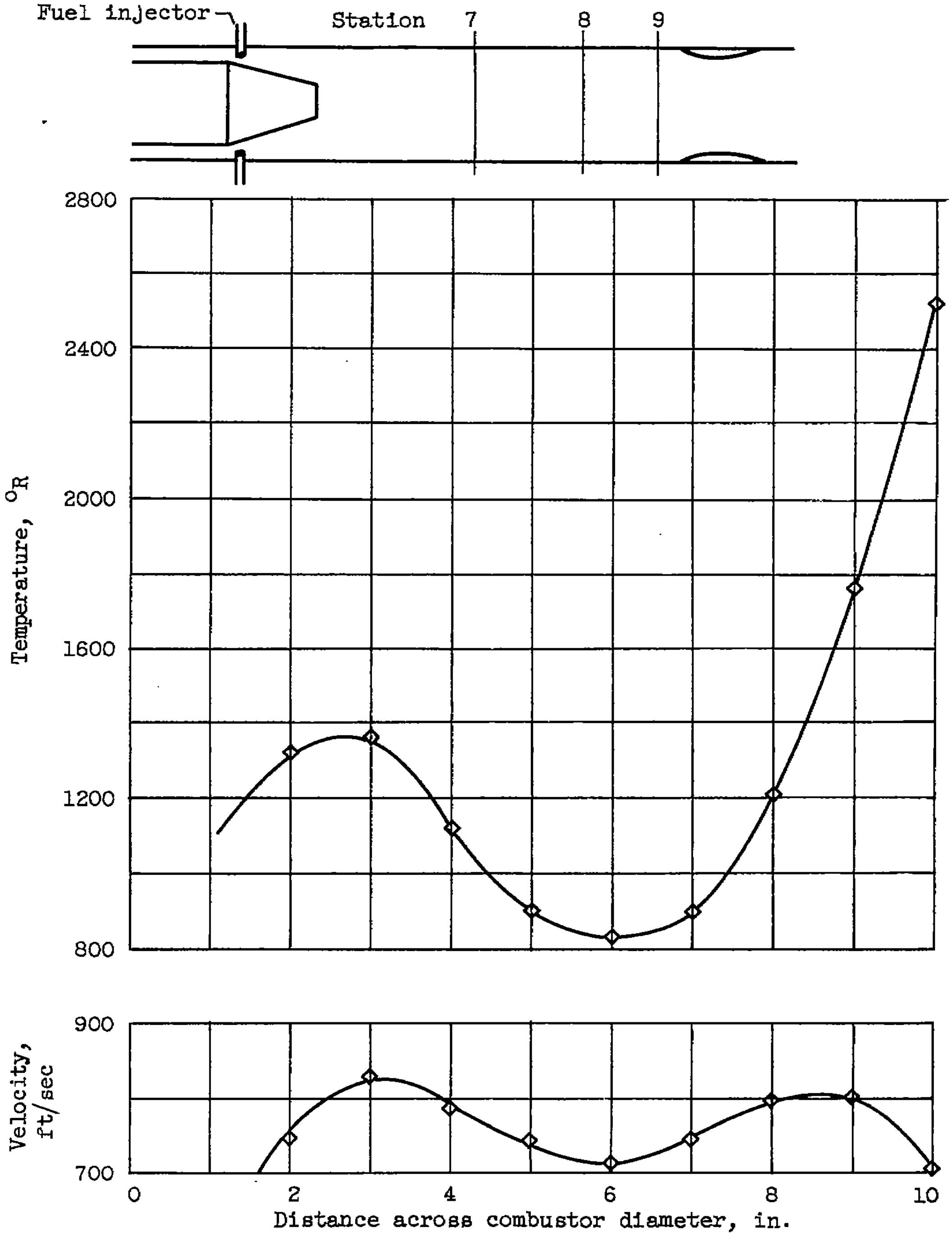

(i) Configuration H. Data taken at station 9.

Figure 4. - Concluded. Temperature and velocity profiles in combustion zone.

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CD-5 back

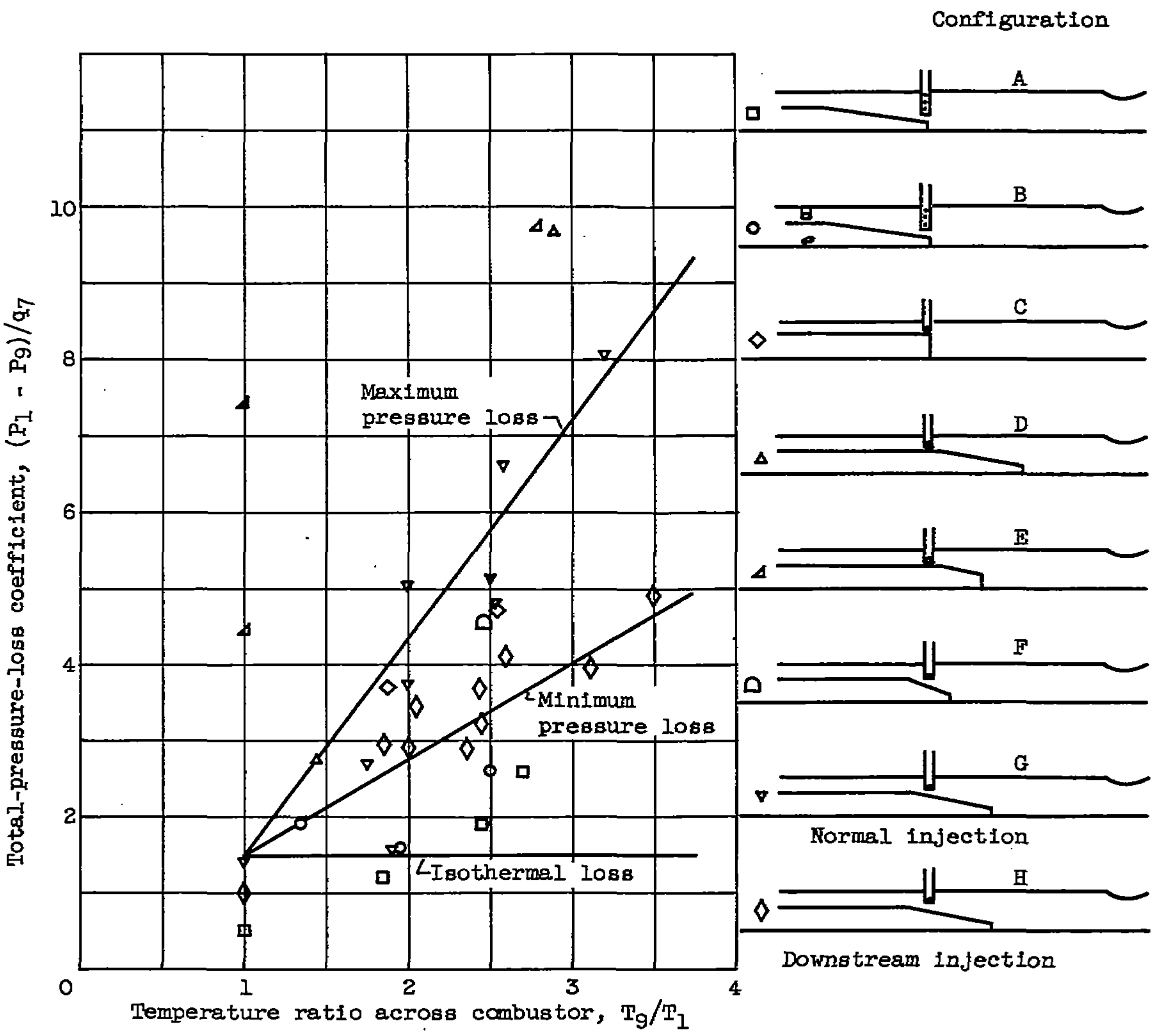

Figure 5. - Pressure-loss coefficient against temperature ratio for various combustor configurations.

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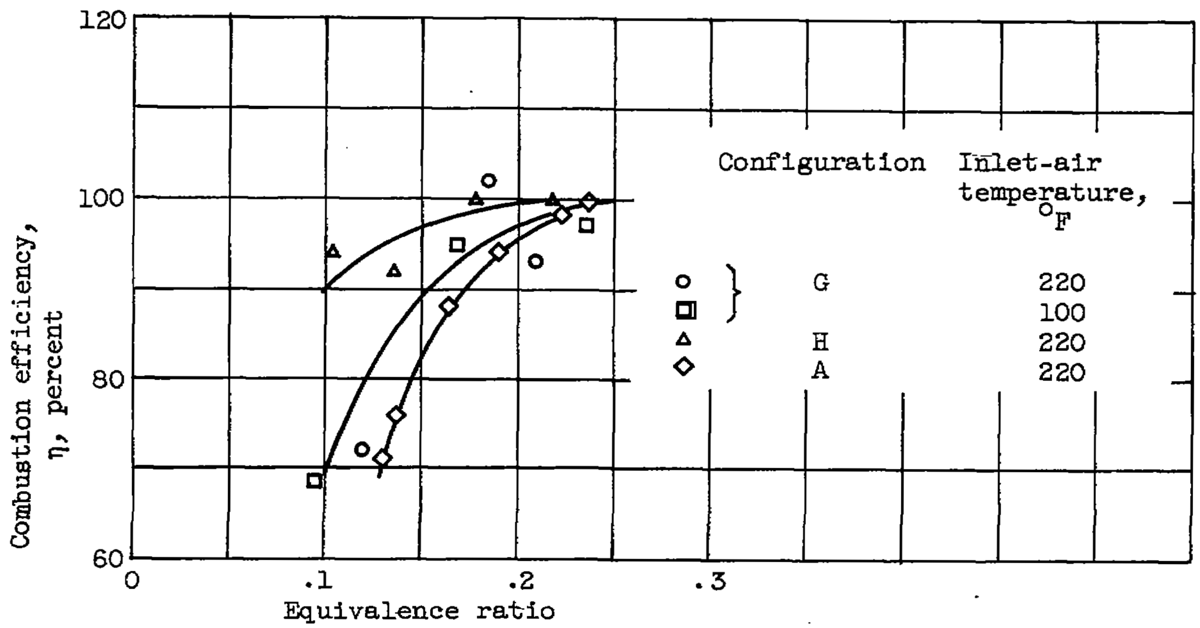

Figure 6. - Combustion efficiency of three configurations for varying equivalence ratio.

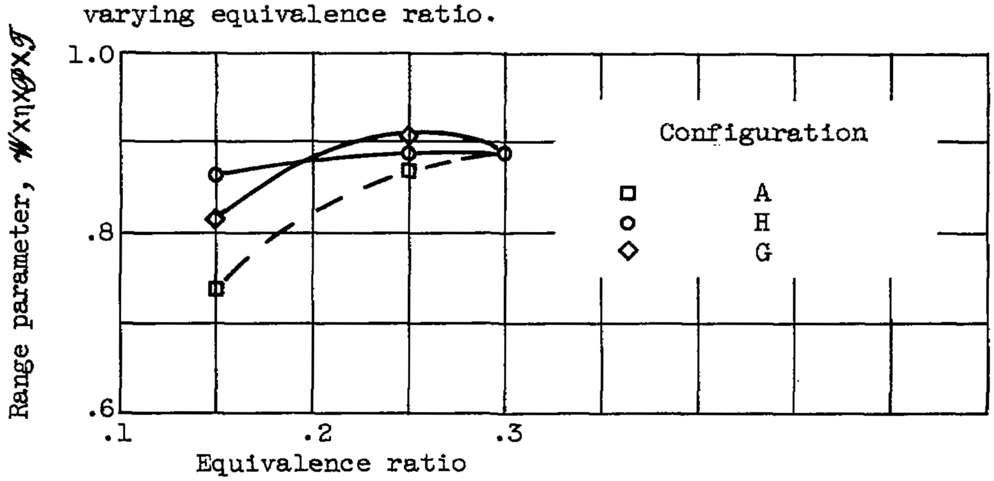

Figure 7. - Relative range of three engine configurations for varying equivalence ratio.

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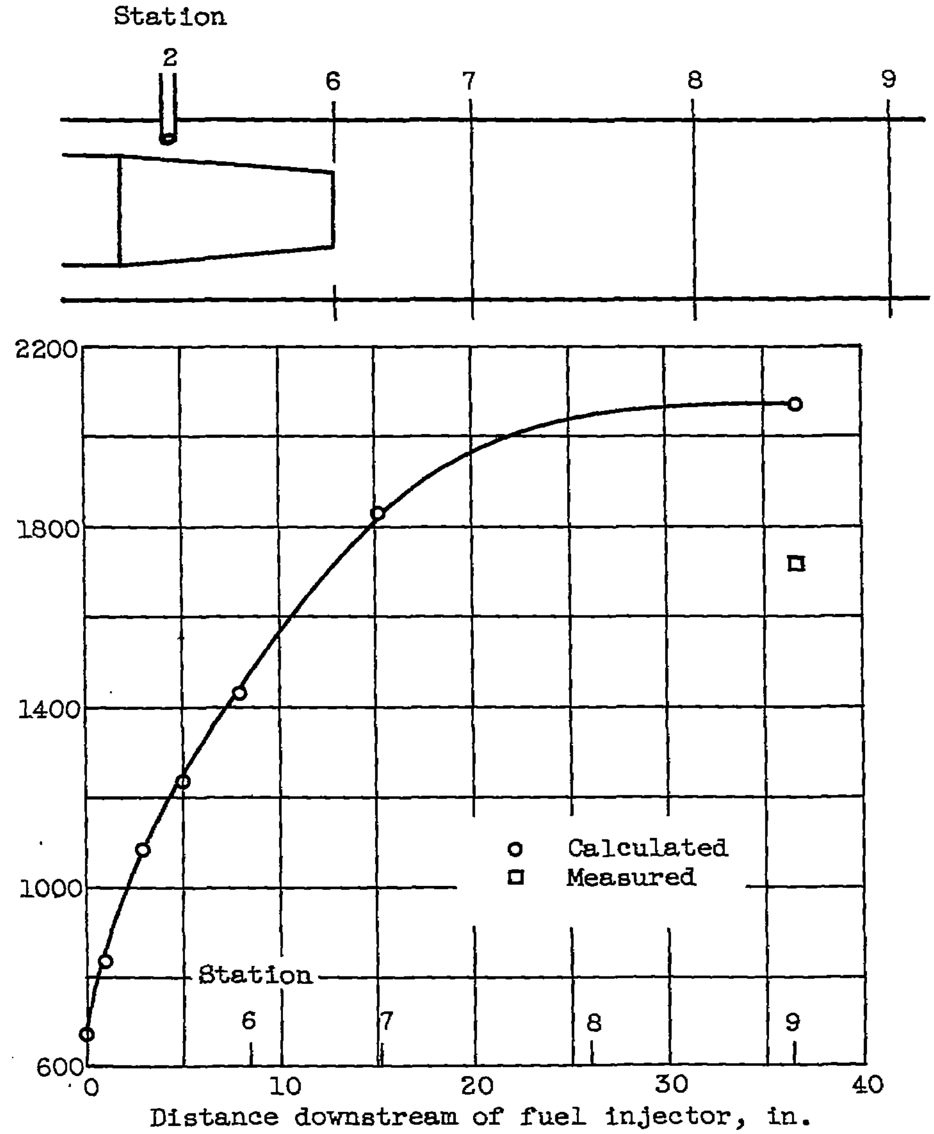

Figure 8. - Gas temperature along burner axis.

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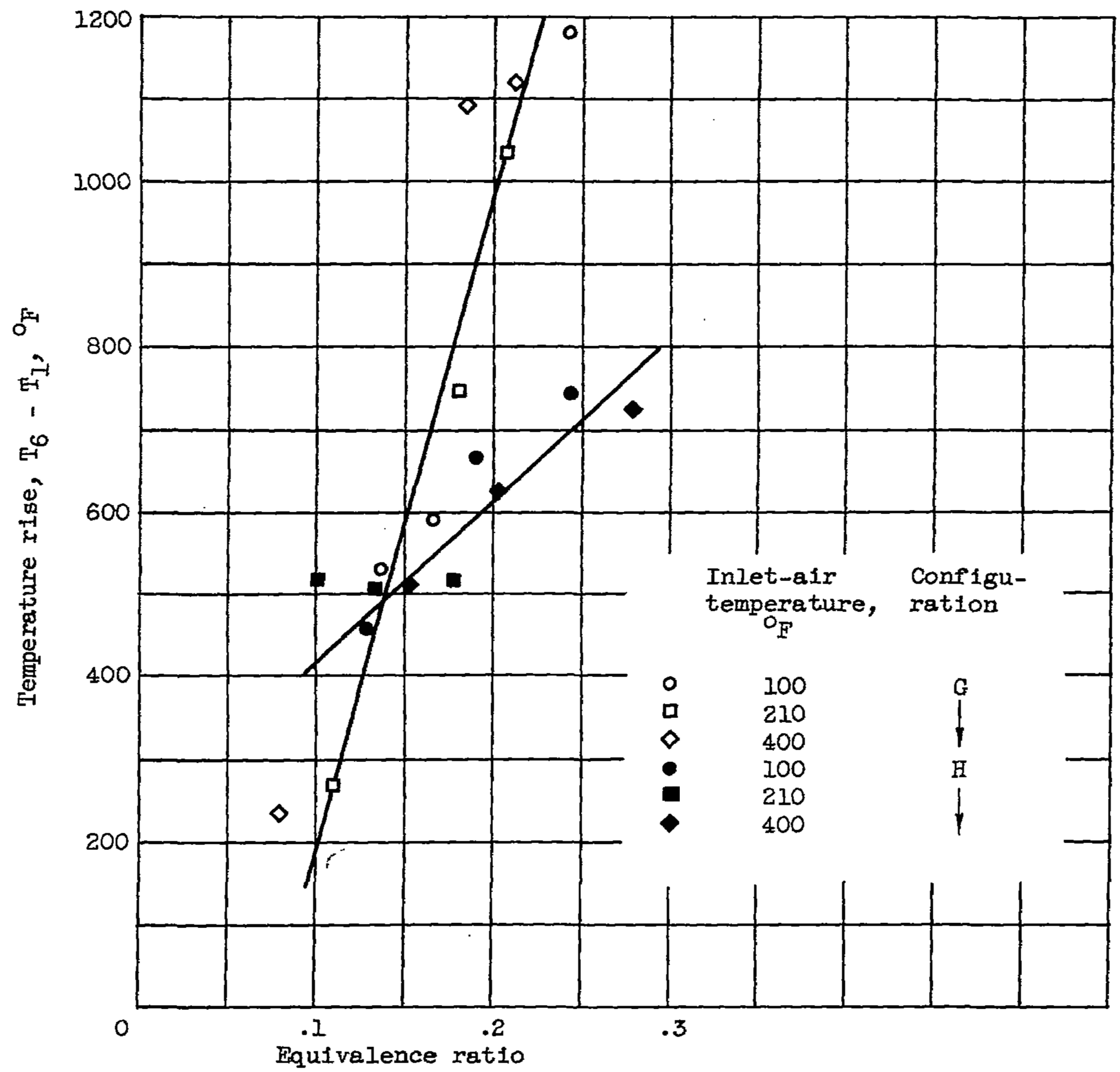

Figure 9. - Calculated temperature rise across diffuser for two configurations and three inlet-air temperatures.

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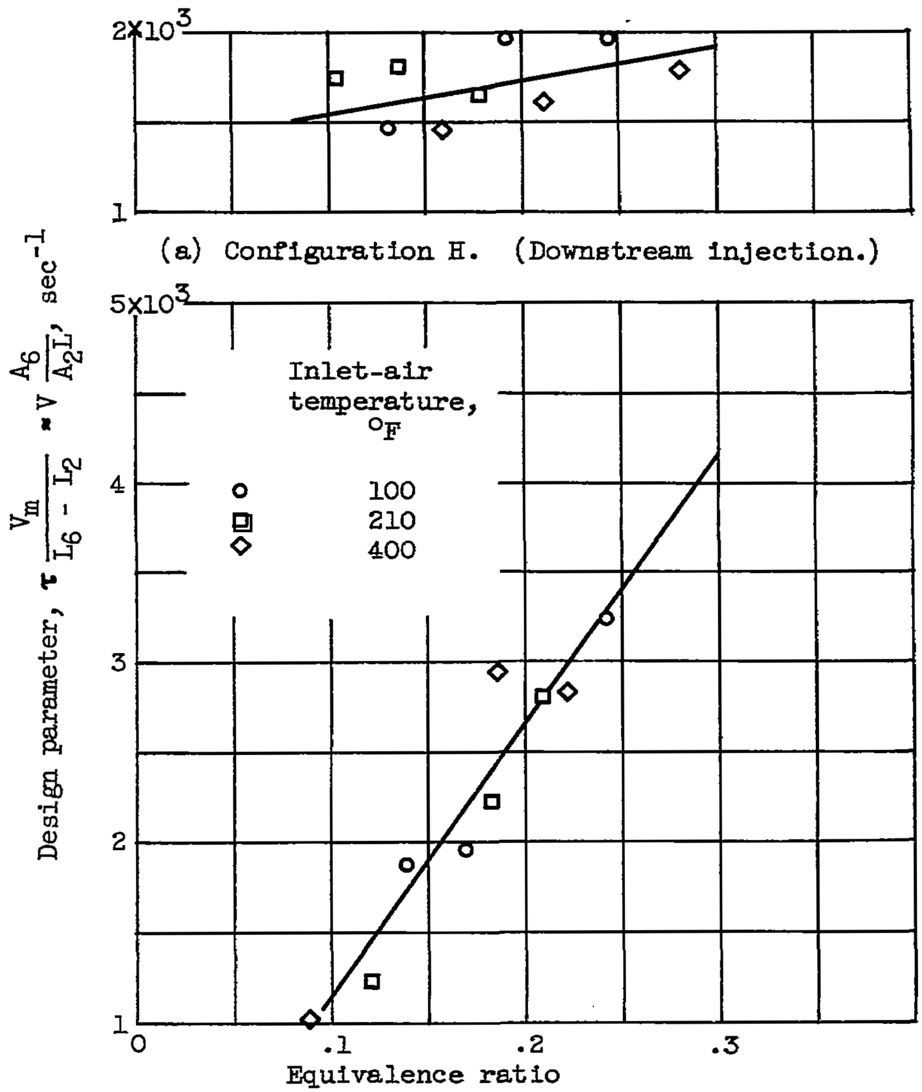

(a) Configuration H. (Downstream injection.)

(b) Configuration G. (Normal injection.)

Figure 10. - Design parameter for determining diffuser length of integrated diffuser-combustor configurations.

Diffusers, Subsonic	1.4.2.1.1
Engines, Ram-Jet	3.1.7
Combustion and Combustors	3.5
Combustion - Ram-Jet Engines	3.5.2.3
Fuel Systems - Engines, Ram-Jet	3.12.1.7

4119 Cervenka, A. J., and Sheldon, J. W.

METHOD FOR SHORTENING RAM-JET ENGINES BY BURNING HYDROGEN FUEL IN THE SUBSONIC DIFFUSER

Abstract

Merging of the subsonic diffuser and the combustor appears feasible with a highly reactive fuel such as hydrogen. At typical ram-jet operating conditions the flame speed of this fuel is high enough for burning to be stabilized at velocities of 600 feet per second by means of a fuel injector alone. Thus it was possible to seat the flame at a station where the Mach number was 0.4 to 0.5 rather than 0.2 as is done conventionally. With this configuration engine length was reduced, exhaust temperature profiles were more uniform, and combustion efficiency was improved at lean fuel-air ratios over the conventional design. However, combustion pressure loss increased, and the net effect of these factors on range was estimated.

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