ICAS Paper No. 68-22

STATE OF-THE-ART IN SHORT COMBUSTORS

by

William H. Roudebush
Airbreathing Engines Division
NASA-Lewis Research Center
Cleveland, Ohio, U. S. A.

The Sixth Congress of the International Council of the Aeronautical Sciences

DEUTSCHES MUSEUM, MÜNCHEN, GERMANY/SEPTEMBER 9-13, 1968

Preis: DM 2.00

STATE OF THE ART IN SHORT COMBUSTORS

by William H. Roudebush
Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio

Abstract

Increased attention is being given to high temperature combustors for advanced aircraft engine applications. Severe durability problems are likely to be encountered. Using substantially shorter combustors can help in this respect since there is less surface to cool and better structural strength. It has been difficult in the past to make improvements in this direction without hurting efficiency and exit temperature profile. The high temperature combustors, however, may provide an environment in which the penalties in performance associated with the shorter designs are not so severe. To examine this possibility NASA has undertaken a program on very short annular combustors for high temperature aircraft engines. Some of the results of that program are described in this paper.

Introduction

Consideration of supersonic aircraft with cruise Mach numbers of 2.7 to 3.2 presents the combustor designer with operating conditions well outside the range of substantial experience. Combustor inlet air temperatures of 1150° F and pressures of 90 psia are typical cruise values. Turbine inlet temperatures of 2200° F and higher are desirable[1]. The inlet air temperature is, of course, the liner coolant temperature as well. The extremely high temperature of the coolant combines with the high temperature level throughout the combustor to create a formidable liner cooling and durability problem. In engines where the pressure is higher the problem is made even more severe; both by higher mechanical loads imposed on the liner and by increased radiant heat flux to the liner[2].

Large amounts of cooling flow tend to be required under these severe conditions. As discussed in Ref. 3, this leads to a deterioration of the exit temperature profile. The temperature profile, however, is more critical than ever in high temperature engines and the problem is further compounded.

One approach to alleviating the problem is to shorten the liner. This reduces the area of surface to be cooled as well as the unsupported length. Other advantages also accrue, such as decreased engine weight and greater flexibility in engine design. Of course, shorter combustors have always been desirable; and while length has seldom been a determining factor in combustor research, the knowledge acquired over the years has led to decreased length. At a given state of the art, however, the minimum length that must be provided is determined chiefly by the performance requirements and the operating conditions to be met. The operating conditions in high temperature engines promote high combustion efficiency. It is hoped that a potential is thereby created in these combustors for the use of such things as higher turbulence level, very lean front ends, smaller liner depths, internal flow deflectors and other devices to decrease combustor length.

In order to evaluate this potential a number of short annular combustor projects have been started by NASA. Figure 1 shows combustor length from compressor exit to turbine inlet plotted against airflow at the sea level takeoff condition. Although other factors than airflow rate are important in determining combustor length, it can be used to show a general trend. The large shaded area coincides roughly with the present combustor design art. The smaller shaded area covers the NASA short combustor projects. Some of these projects are being carried out under contracts with industrial organizations. Others are being conducted partially or wholly in-house. The present paper describes the progress to date on several of these projects.

Twin Ram Induction Combustor

The twin ram induction annular combustor (fig. 2) was designed for Mach 3 cruise conditions at 65 000 feet. The design sea level takeoff airflow is 260 lb/sec. The external diameter of the combustor is 40 inches and the overall length, including the diffuser, is 20 inches. The distance from the fuel nozzles to the turbine inlet station is 12 inches. The 20 inch length compares with an overall length of more than 50 inches for the combustor in the J58 engine currently used in the SR-71/YF-12.

Tests of the ram induction concept were made in a 90 degree sector of a full-scale combustor. The program was conducted under NASA contract and is reported in Ref. 4.

Design

The original combustor design is shown in cross section in Fig. 2. To minimize parasitic pressure losses and thereby increase the pressure drop available for mixing, the diffuser inlet Mach number of 0.28 is diffused only moderately to 0.20. This relatively high velocity air is then turned by means of scoops with internal vanes into the combustor liner where it is discharged at the same Mach number. The high velocity and steep angle of the entering jet promote rapid mixing of air and fuel in the primary zone and of diluent air and burned gases in the mixing zone. This leads to rapid burning of the fuel and increased uniformity of the exit temperature profile[5]. The center shroud scoops divide the annulus into two concentric annuli having higher values of length/height ratio than a single annulus would have had. This was also expected to help produce a uniform temperature profile in the short length allowed.

The combustor was designed to have an isothermal pressure loss of 6% of inlet total pressure at the cruise condition. At this condition inlet total pressure is 90 psia, inlet total temperature is 1150° F and reference velocity is 150 ft/sec. The mass flow at cruise is 96 lb/sec for the full annulus.

The design airflow in the two concentric effective annuli was the same. Within each annulus the air was divided equally between inner and outer scoops, which were opposed. The primary scoops (first row) admitted 24% of the total airflow. Each scoop had a square exit 0.432 inches on a side. Forty-three percent of the airflow entered through the secondary scoops (second and third rows). Each of these scoops had a square exit 0.579 inches on a side. For a full annulus combustor of this design a total of 512 scoops is needed. The remaining airflow was divided between swirlers (12%) and film cooling (21%).

The diffuser inlet annulus height was 1.6 inches and the combustor casing height was 5.7 inches. In order to maintain acceptable diffusion rates in the 8 inch diffuser length, four splitter plates were used to divide the diffuser into five sections. The diffuser entry Mach number of 0.28 was decreased to 0.2 at the diffuser exit in the passages leading to the inner and outer shrouds. In the center three passages the Mach number was reduced to about 0.1 to decrease the loss involved in dumping air for the swirlers. The remaining air was reaccelerated into the inner, outer and center shrouds.

There were 32 fuel nozzles required for each annulus in the full annular design. Simplex nozzles were used for the test program. A radial inflow swirler surrounded each nozzle. The fuel used was ASTM A-1.

Testing

Tests were conducted in a 90 degree sector of a full-scale combustor at a simulated take-off condition (inlet temperature 600° F) and a simulated Mach 3.0 cruise condition (inlet temperature 1150° F). In each case the outlet temperature was 2200° F. The pressure was 60 psia instead of the intended 90 psia because the sector housing turned out to be overstressed at the higher pressure. Exit temperatures and pressures were measured with a five-point total pressure-total temperature traversing rake. The thermocouples were shielded and aspirated, with junctions of platinum and 90/10 platinum rhodium alloy. Readings were made at 3 degree intervals across the sector. Efficiencies were calculated from mass-weighted exit temperatures.

Development

A number of problems were encountered and major and minor changes made in the course of development tests.

The diffuser housing and the extreme inner and outer diffuser splitter plates warped and it was finally necessary to eliminate these two splitters entirely. The performance of the diffuser was not measured but it is quite possible that the flow was separated. A coarse screen added near the diffuser exit late in the test program resulted in an improvement in exit temperature profile.

Liner and firewall overheating occurred. This was corrected to some extent by adding a cooling gap behind the second row of scoops, holes and deflectors around the nozzles, and thumbnail scoops, as required, on the liner. However, overheating and warping of the liner were continuing problems in the program.

The third row of scoops was deleted from the center shroud early in the program to decrease the airflow to this region. Later it was found that deleting all third row scoops was required for a good exit temperature profile, resulting in the configuration shown in Fig. 3.

Spark ignition failed to ignite the burner reliably. Rather than spend time on this problem in the preliminary stages of the program it was decided to use pyrophoric ignition. Throughout the remainder of the program, with only a few exception, ignition was obtained by injection of triethylborane.

Results

Performance data were obtained over a wide range of operating conditions for combustor model number 14. Subsequent changes to the combustor to improve exit temperature profile resulted in model number 18 (Fig. 3) which had a superior exit temperature profile. Model 14 differed in a major way from the initial configuration (Fig. 2) by having only two rows of center shroud scoops. Model 18 differed further (Fig. 3) by having all third row scoops removed and the remaining scoops resized. The program ended without taking a wide range of data on model 18. The points at which data were taken are presented along with the data on model 14 and show no difference in performance at these isolated points.

Efficiency was high over a range of fuel/air ratios and for reference velocities up to 189 ft/sec as shown in Fig. 4. Efficiency as a function of reference velocity is shown in Fig. 5. At low pressure the hydraulic diameter is known to have an important effect on combustion efficiency[6]. Therefore these results are encouraging for a highly turbulent burner with such a small hydraulic diameter.

Pressure loss characteristics are shown in Fig. 6. In spite of the high annulus velocities and poorly designed diffuser the pressure losses are not excessive. The diffuser pressure loss at an inlet Mach number of 0.28 was estimated to be about 3%. It seems likely that more care in future diffuser design can reduce overall pressure loss or alternately permit the use of higher liner and jet velocities.

The average radial temperature profile obtained with model 18 is shown in Fig. 7 for the cruise condition. The measured profile exceeds the desired profile by less than 100 degrees. The circumferential profile (Fig. 8) is also flat. The parameter ΔTVR (the ratio of maximum temperature increase at any point to average temperature increase) has the value 1.28 which compares reasonably well with existing combustors.

Some limited ignition testing was done at pressures of 20 psia and slightly below. The burner would not ignite at fuel/air ratios below 0.0185 at 20 psia. It would not ignite at all below 16.6 psia. This poor ignition had some contributing factors, however. The ignitors were badly located on a side wall of the combustor sector. Furthermore, the simplex nozzles had pressure drops as low as 10 psid at the low flows required for the ignition tests.

Since these factors could have contributed substantially to the ignition difficulty, the severity of the problem cannot be determined from the limited testing done.

Further Work

A full scale 360 degree version of this combustor is now being built for further testing. In these tests development changes will be made as required to provide acceptable performance and temperature profile characteristics over a wide range of operating conditions. Low pressure blowout and relight problems will be studied. Some cyclic endurance tests will also be conducted if the performance of the unit developed is sufficiently good.

Distributed Burner

Instead of using a single large flame zone, as provided by the dome and liner in a traditional annular combustor, a large number of small flame holders can be distributed across the combustor annulus. Such a configuration offers some advantages for short combustors as described below.

A distributed burner was designed with a 40 inch outer diameter and 20 inches in overall length. The combustor was designed for Mach 3.0 cruise operation at 65,000 feet and has a design sea level takeoff airflow of 275 lb/sec. The diffuser inlet annulus height is 2 inches and the duct height is 10 inches. Tests of a full-scale full-annulus version of the combustor will be conducted at Lewis Research Center as a part of the inhouse research program.

Design

The distributed burner departs from the concept of diverting diluent air around a large sheltered zone and then mixing the diluent air with burned gases in the form of high velocity jets. Instead, all of the airflow is brought through the burning zone of the combustor which is composed of a large number of simple flameholders as in the rectangular sector shown in Fig. 9. The mixing process is similar to that occurring downstream of any bluff body except that there are interaction effects with neighboring wakes.

The pressure loss associated with this type of mixing is small compared to that required for penetration jets. The possibility also exists for tailoring radial profile by adjusting fuel flow at various radii. The liner, serving no diluent air function, can be made simple and rather far removed from the hottest gases. This leads to lower liner cooling flows and better endurance. Finally, this combustor lends itself well to very large temperature rises without fuel staging.

The combustor was designed for a cruise isothermal pressure drop of about 5% at a diffuser inlet Mach number of 0.3. The inlet temperature at cruise is 1150° F, the exit temperature 2200° F and the pressure 90 psia. Reference velocity is 150 ft/sec. One hundred and twenty cans 2 inches in diameter at the exit are used in the full annulus. The projected area blockage of the cans is approximately 34%. The diffuser has an included angle of more than 35 degrees and two splitter plates will be used in the first model to prevent separation.

The fuel is ASTM A-1 and the flow to each can is externally metered by a high pressure drop orifice. The fuel enters each can and is mixed with a small amount of air entering the upstream face of the can through an orifice. The flame seats inside the can and part of the air flowing around the can recirculates and enters the hot wake to complete the burning.

Testing

Testing to date has been done at Lewis Research Center in a 12- by 30-inch rectangular sector. The flameholder array shown in Fig. 9 was mounted at the downstream end of a 30 degree diffuser (Fig. 10). A section of straight housing and a simple exhaust ramp completed the combustor. The total combustor length was 39 inches. A film-cooled liner was used on all four sides of the duct.

The rectangular sector simulates a section of a much larger combustor than the design described above. Therefore, the information obtained in these tests does not assure the same results in the smaller model. Nevertheless, the modular nature of the distributed burner lends itself to a change in size. Furthermore, the ratio of combustor length to diffuser inlet height is comparable in the two designs.

Temperatures and pressures at the exit to the combustor were measured with a 7 point total pressure-total temperature traversing rake. The thermocouples were shielded and aspirated, with junctions of platinum and 90/10 platinum rhodium alloy.

Tests were conducted at inlet temperatures of 540° and 1140° F, exit temperatures up to 2200° F, pressures of 17, 45, and 90 psia, and reference velocities up to 190 ft/sec. Only very limited testing at 90 psia was done since bowing of the rectangular housing occurred at that condition.

Results

Many configurations were tested with a variety of can sizes and airflow splits. Simple scoops were sometimes used just downstream of the can exit to deflect air into the burning zone. The data presented here were taken with the configuration shown in Fig. 9. The cans have an exit diameter of 2.9 inches.

The efficiency is shown in Fig. 11 for reference velocities up to 190 ft/sec. The pressure was 45 psia and the inlet temperature 540° F. At an inlet temperature of 1140° F the efficiency is nearly 100% over the same range. The pressure loss characteristics are given in Fig. 12. It can be seen by comparing with Fig. 6 that this is a lower pressure loss combustor than the twin ram. About half of the pressure loss was attributed to the diffuser.

The average radial temperature profile is shown in Fig. 13. The measured profile agrees very well with the desired profile. The value of ΔTVR of 1.28 compares favorably with other combustors.

Further Work

Testing is continuing in the rectangular duct to further develop the concept. Tests will be conducted to determine low pressure performance and relight characteristics. It is clear that this will be a difficult area of operation for these burners. Meanwhile the full annulus combustor is being built for tests at severe conditions of pressure and temperature and further development in the more realistic configuration.

Diluent Stator Combustor

One of the unusual devices being studied both under contract and inhouse at NASA is the combined stator-combustor concept shown schematically in Fig. 14. This is a somewhat smaller combustor than the two described above, having an outer diameter of 30 inches. The overall length is 14 inches from compressor exit to first stage turbine stator exit. The distance from the dome to the first stage turbine stator exit is 9.5 inches. The burner was designed for operation at a sea level static condition of 574° F inlet temperature and 2200° F outlet temperature with a reference velocity of 136 ft/sec at a pressure of 9 atmospheres. A 60 degree test sector was designed and built under contract and the testing will be done by NASA inhouse.

Design

The purpose of this combustor is to mitigate the exit temperature profile problems aggrevated by short combustor length. The technique used is to bring all of the diluent air, approximately half of the total airflow, through the first stage turbine stator. A part of this air, from 12 to 19% of the total airflow, is used to cool the stator. The remainder is discharged through a row of holes distributed radially near the stator leading edge. No diluent air is admitted through the liner as in traditional combustors. About 9% of the total airflow is used for film cooling. The remainder, approximately 41%, enters through the dome.

It is hoped that the more positive control of the diluent air distribution will provide better control of the radial temperature profile. The problem of circumferential profile distortion at the combustor exit is also alleviated with this concept because the first turbine member to encounter the combustion exit gases is a rotor. The rotor, since it averages out circumferential differences, is affected chiefly by radial profile. Although circumferential temperature distortions may be carried through to later stator rows, the temperature level is significantly lower after the first stage rotor. Of course, the problem of cooling the diluent stator itself is a very severe one.

The diluent stators, as shown schematically in Fig. 14, are much larger than traditional stators in order to accommodate the high mass flow. Two blade cooling designs were made, one utilizing film cooling of the suction and pressure surfaces (Fig. 15), the other (Fig. 16) using a transpiration skin, Nichrome V. The transpiration cooled blades were designed for a maximum skin temperature of 1200° F using 12% of the airflow for cooling. The film cooled blades were allowed to go to 1500° F and 19% of the airflow was required. Detailed heat transfer analyses of these blades were made in which luminous flame radiation was assumed and conduction within the blade skin was included. Preliminary tests at NASA of a single uncambered blade section of the film-cooled design, placed in the wake of a simple flame-holder combustor, indicate that the cooling flow is adequate.

A pair of adjacent blades is shown in Fig. 17. The blades are located at 12 degree intervals circumferentially with the blade leading edges midway between fuel tubes. This was done to keep locally hot regions, associated with individual fuel tubes, from impinging directly on the blade leading edges. The high solidity passages provide short penetration distances for the diluent jets.

The primary zone selected for this project was a vaporizer design. In this design all of the primary air enters through the vaporizer head plate. Part of this air enters through fuel tubes in which it is mixed with the externally metered fuel before entering the combustor. Deflector caps on the fuel tubes direct the fuel-air mixture upstream. The remainder of the air enters through slotted air cups located circumferentially midway between the fuel tubes. There are 30 fuel tubes and 60 air cups in the full annulus design. Although this type of primary zone has been referred to traditionally as "vaporizing", it is likely that preheating the fuel and premixing it with air are the true functions performed.

The combustor was designed for ASTM A-1 fuel. Ignition will be accomplished by a spark igniter and primer fuel injector.

Testing

Initial tests will be started soon at NASA. The early testing will be done in an atmospheric rig for good visual access. When it has been established that the blade cooling is adequate rough measurements of efficiency and pressure loss will be made at atmospheric pressure. If the combustor can be developed to a promising level of performance, connected duct tests will be conducted over a wider range of conditions and detailed performance measurements will be made.

Concluding Remarks

From the tests described above and from other recent work by NASA with combustors operating at high temperatures some preliminary observations can be made. For operation at temperature levels of 600° to 1150° in and 2200° F out and pressures of 15 to 90 psia, there is hope for achieving high performance combustors with substantially decreased length. The most difficult performance problem to overcome will probably be the low pressure relight problem.

Problems associated with diffuser separation and flow distortion are also severe. A much better understanding of practical annular diffuser operation is needed. The effects of radial struts and snout blockage on diffuser flow are largely matters of conjecture at the present time. Total pressure loss problems, present for all combustors, are aggrevated in short combustors because decreasing length generally means that higher pressure drops are required for mixing. The cursory understanding of the relation of pressure drop to jet penetration and mixing that has served in the past may no longer be adequate.

Finally, the problem of liner cooling and durability has proven to be severe in all of the testing done by NASA at 1150° F inlet temperature. Tests at pressures from 90 to 400 psia which have recently started at NASA, will no doubt reveal even more serious problems. Improvements over present cooling schemes may be required to provide long combustor life at these formidable temperature levels.

References

1. Dugan, James F., Jr.; Koenig, Robert W.; Whitlow, John B., Jr.; and McAuliffe, Timothy B.: Turbojet and Turbofan Engines for a Mach 3 Supersonic Transport. Paper No. 64-244, AIAA, June 1964.

2. Reeves, D.: Heat Transfer in Chambers. Experimental Methods in Combustion Research. J. Surugue, ed., AGARD, Pergamon Press, 1961, sect. 3.2, pp. 5-29.

3. Dooley, Philip G.: Design and Development of Combustion Chambers for Turbine Engines. Paper No. 64 - WA/GTP-8, ASME, Nov. 1964.

4. Kitts, D. L.: Development of a Short-Length Turbojet Combustor. Rep. PWA-FR-2433, Pratt & Whitney Aircraft (NASA CR-54560), Mar. 18, 1968.

5. Chamberlain, John: The Ram-Induction Combustor Concept. Presented at the AIAA Third Propulsion Joint Specialist Conference, Washington, D. C., July 18, 1967.

6. Olson, Walter T.: Combustion Chamber Developments. Design and Performance of Gas Turbine Power Plants. Vol. XI of High Speed Aerodynamics and Jet Propulsion. Princeton Univ. Press, 1960, pp. 289-350.

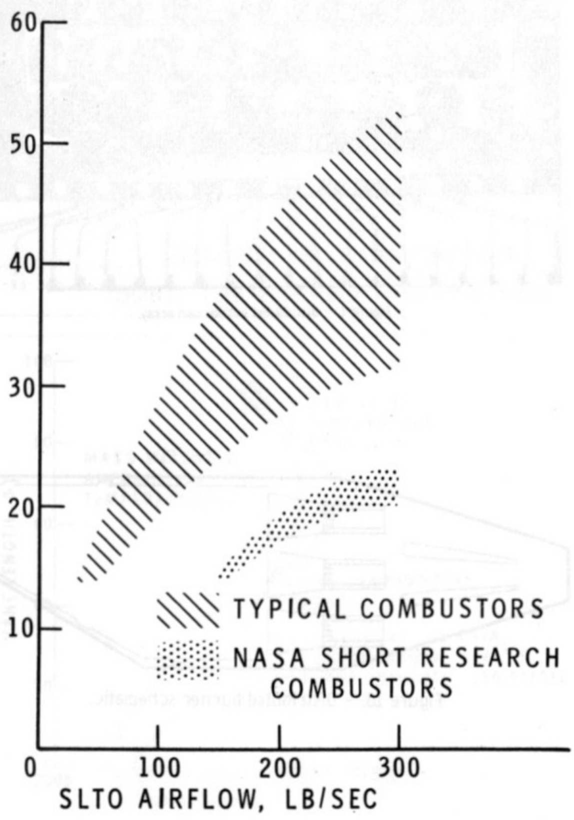

Figure 1. - Combustor length versus SLTO airflow.

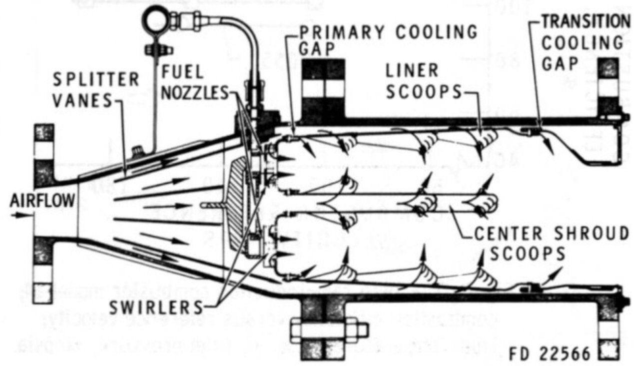

Figure 2. - Twin ram-induction combustor model 1.

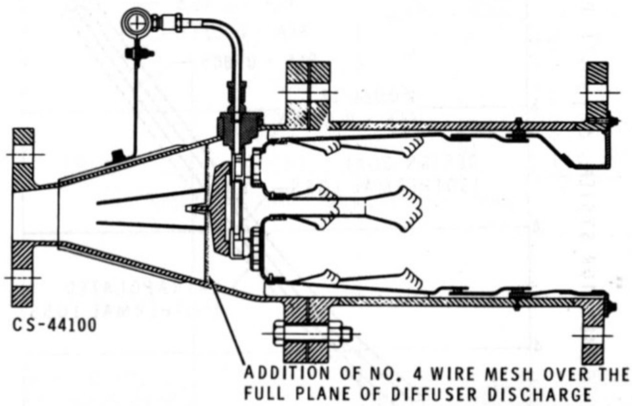

Figure 3. - Twin ram-induction combustor modification model 18.

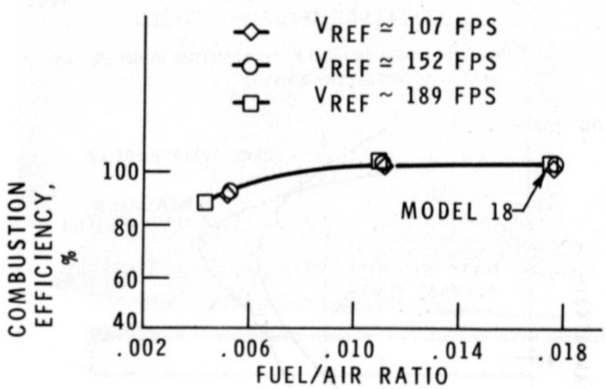

Figure 4. - Twin ram-induction combustor model 14; combustion efficiency versus fuel/air ratio; inlet temperature, 1150° F; inlet pressure, 60 psia.

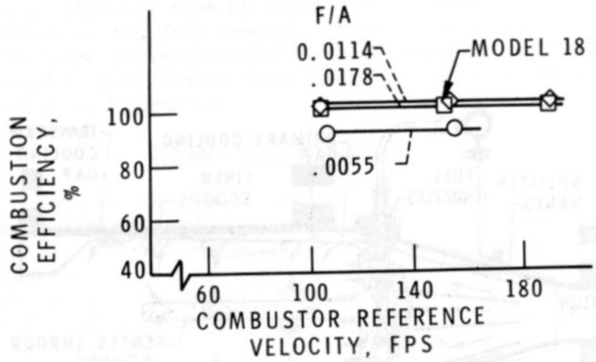

Figure 5. - Twin ram-induction combustor model 14; combustion efficiency versus reference velocity; inlet temperature, 1150° F; inlet pressure, 60 psia.

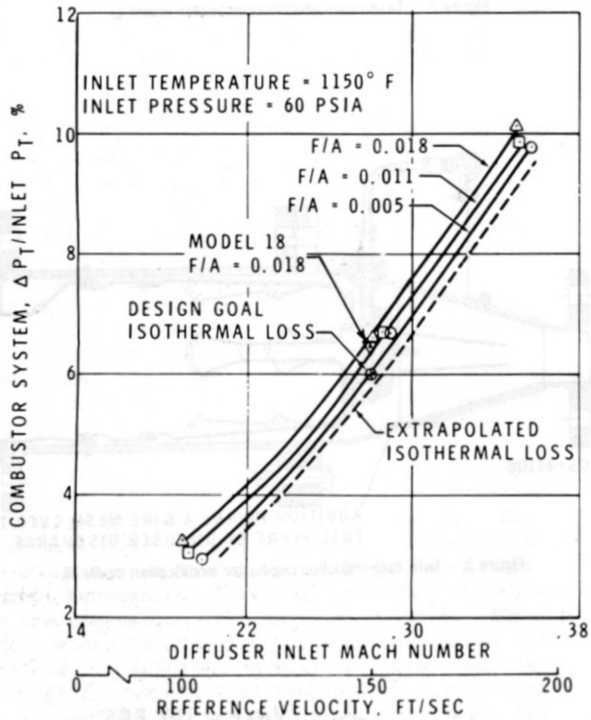

Figure 6. - Twin ram-induction combustor model 14; combustion system pressure loss.

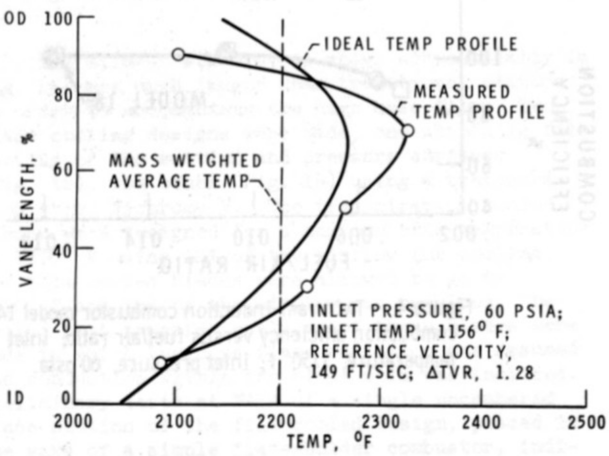

Figure 7. - Twin ram-induction combustor model 18; radial mass-weighted temperature profile.

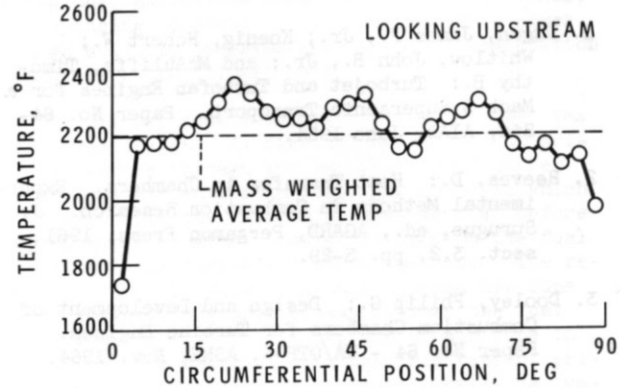

Figure 8. - Twin ram-induction combustor model 18; circumferential mass-weighted temperature profile; inlet temperature, 1156° F; inlet pressure, 60 psia; reference velocity, 149 ft/sec; ΔTVR, 1.28.

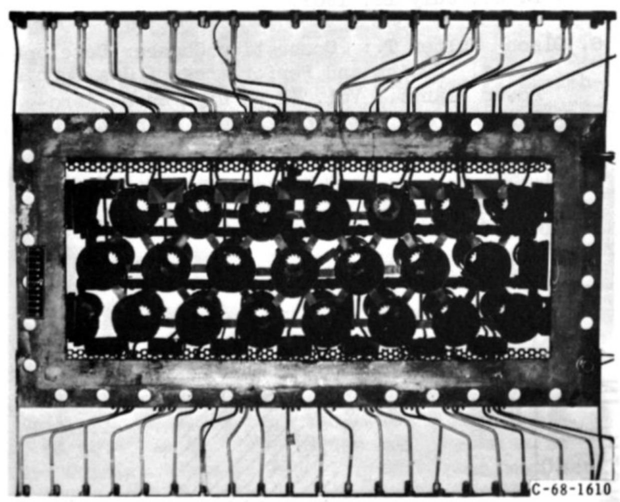

Figure 9. - Distributed burner can array.

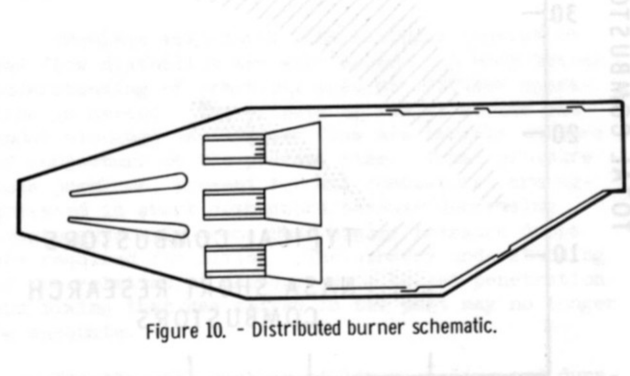

Figure 10. - Distributed burner schematic.

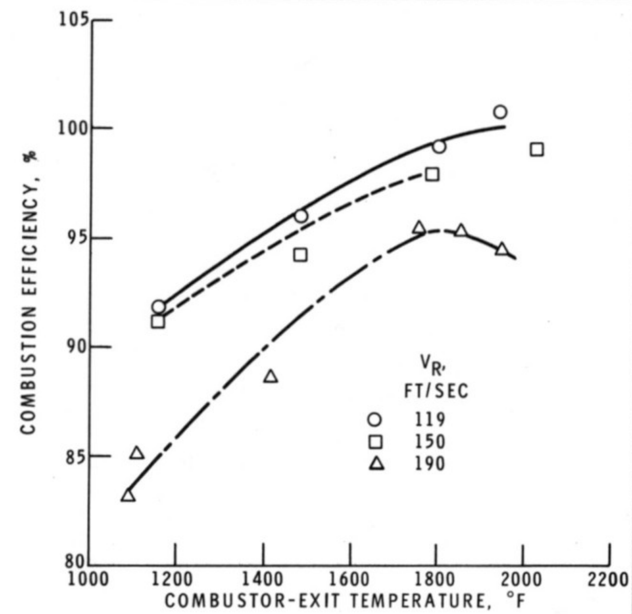

Figure 11. - Distributed burner combustion efficiency; inlet temperature, 540° F; inlet pressure, 45 psia.

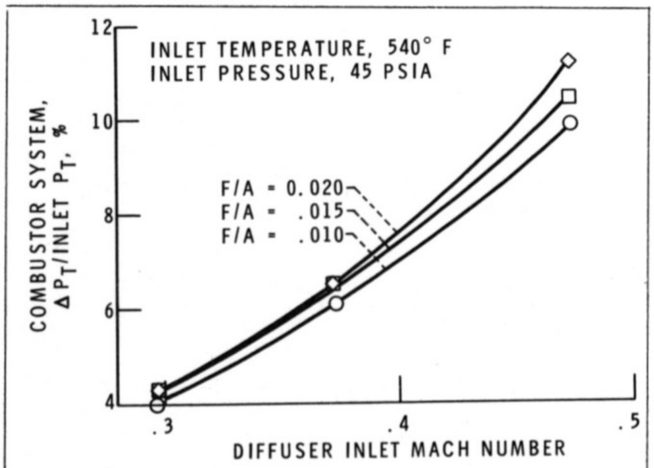

Figure 12. - Distributed burner system pressure loss.

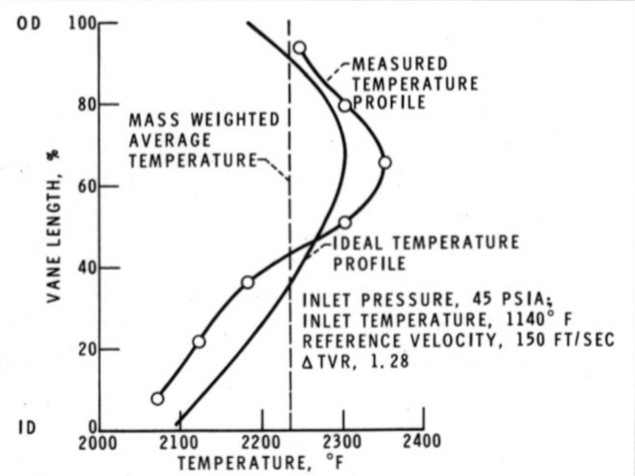

Figure 13. - Distributed burner radial mass-weighted temperature profile.

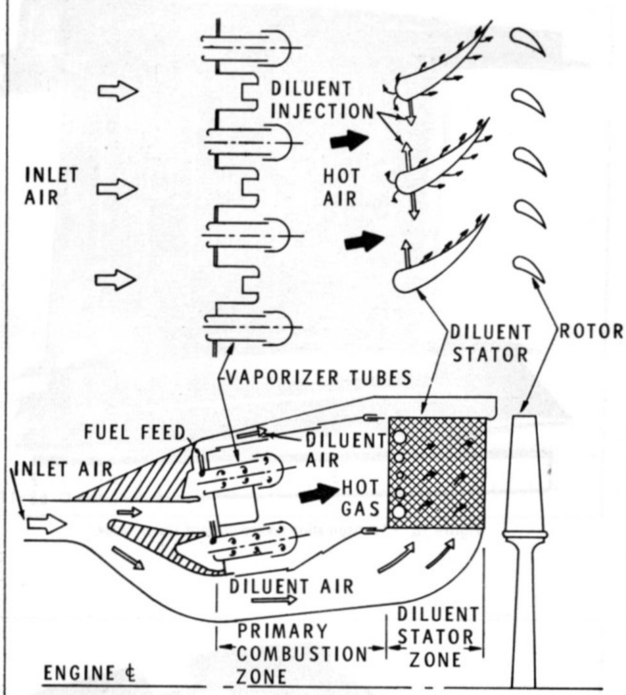

Figure 14. - Diluent - stator combustor concept.

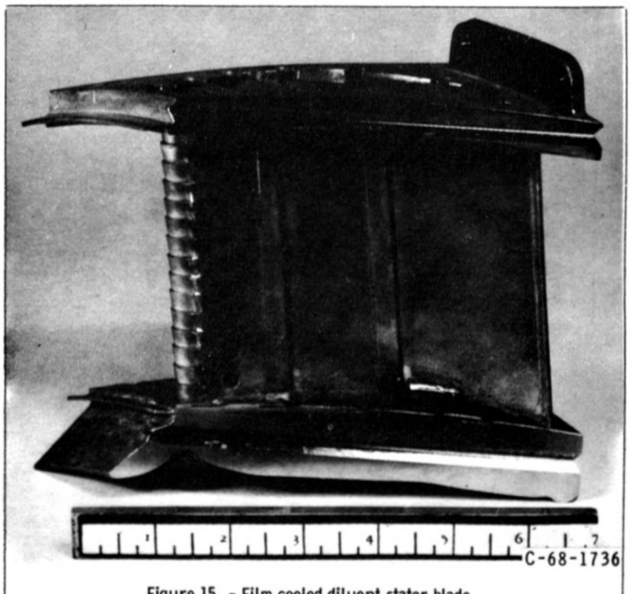

Figure 15. - Film cooled diluent stator blade.

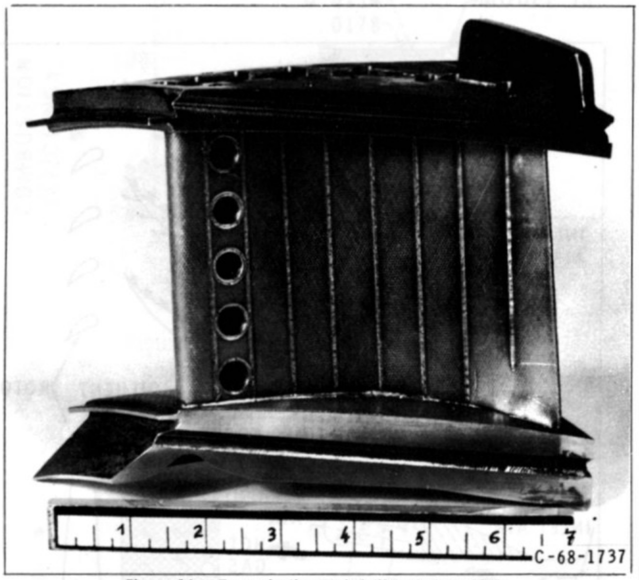

Figure 16. - Transpiration cooled diluent stator blade.

Figure 17. - Adjacent diluent stator blades.