Design and Test of a Low-Cost Combustor for Expendable Turbine Engines

S. Moskowitz* and F. Spindler†
Curtiss-Wright Corporation, Wood-Ridge, N. J.

Application of the gas turbine for use in unmanned military aircraft and missiles, industrial or vehicular power applications, and inexpensive general aviation aircraft is possible only through development of low-cost engine components. This paper presents the design and development of an annular fuel vaporizing combustor for a low-cost turbojet for use in a supersonic missile and other expendable or limited-life applications. The design of the low-cost gas turbine and the factors that influence the selection of component characteristics, design configurations, materials, and methods of manufacture are discussed. In order to provide high specific power, the highest burner temperature possible, without introducing design complexity and performance and cost penalties associated with turbine cooling, was selected. Consequently, the combustor's influence on turbine performance and durability had a significant effect on the engine design and cost. The combustor performance goals, including efficiency and temperature pattern factor as provided within envelope, reliability, and cost constraints, are described. An experimental combustor test program was conducted to evaluate the design pertaining to liner cooling, pyrotechnic ignition, fuel introduction, primary and dilution air distribution, and exit gas temperature profiles. The combustor test results and the influence of combustor performance on turbine life are discussed.

Introduction

INCREASED consideration has been given to the application of low-cost gas turbine engines for use in tactical missiles and drones. Furthermore, use of this type of gas turbine for application in general aviation and for industrial power drives or vehicular equipment has brought attention to the need for development of low-cost components. For the missile applications, engine design philosophy must be redirected from the conventional criteria of long life in combination with performance of high-technology components to low cost. The new philosophy must provide design configurations with appropriate compromises to 1) optimum cycle and internal gas conditions, 2) high technology and performance levels, and 3) margins of strength rarely, if ever, used in man-rated applications.

Engine Requirements

In order to provide a design specifically for application in Navy supersonic tactical missiles, an engine configuration was established which satisfied a unit cost target of $5 per pound of sea-level static thrust for quantity production. Design constraints included the requirement to be within an envelope established by the dimensions of 14 in. diam and 44 in. length. The engine weight of under 375 lb was specified. Reliability of 99% and storage capability of 5 years without maintenance also were specified. The engine thrust rating at the primary design point was 2300 lb at Mach 1.5, sea level. A thrust requirement of at least 1600 lb also was specified at Mach 0.9, sea level.

In order to satisfy these requirements at the lowest engine cost, the following engine design criteria were implemented: 1) high airflow per unit frontal area by selecting an axial flow transonic compressor; 2) high thrust per unit airflow by selecting optimum pressure ratio and high burner temperature; 3) minimal mechanical design complexity; 4) intensive utilization of the integral investment casting process.

Presented as Paper 76-768 at the AIAA/SAE 12th Propulsion Conference, Palo Alto, Calif., July 26-29, 1976; submitted Oct. 14, 1976; revision received April 8, 1977.
Index category: Airbreathing Propulsion.
*Director, Energy Systems Research and Development.
†Program Manager, Combustion.

The low-cost gas turbine engine resulting from the design effort was a single-spool turbojet weighing 264 lb and having a four-stage axial flow compressor, an annular combustor using a fuel vaporizing system, a single-stage axial flow turbine, and an extended plug-type fixed area exhaust nozzle. All design specifications were satisfied. The maximum diameter of 14 in. occurred at the combustor and turbine section.

The engine operating conditions at Mach 1.5, sea level are as follows: 1) airflow, lb/sec, 51; 2) pressure ratio, 3.63; and 3) turbine inlet temperature, °F, 1900. The effect of pressure ratio and turbine inlet temperature on specific thrust was investigated to optimize and select the cycle conditions. The results of this study are shown in Fig. 1, which presents the turbojet cycle analysis at sea level and 1000 knots flight speed.

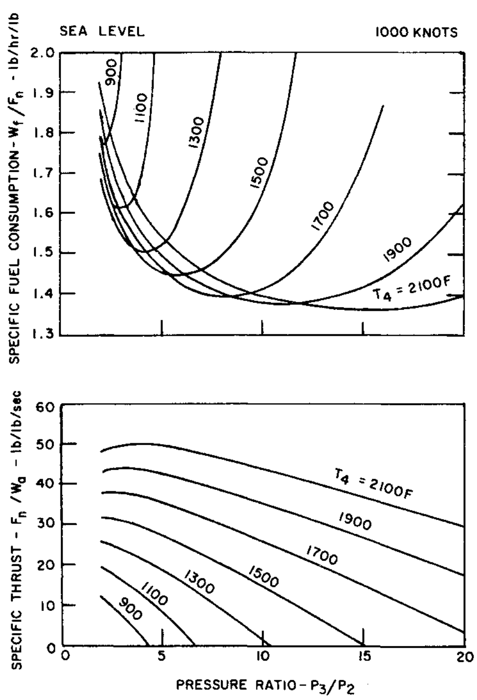

Fig. 1 Turbojet cycle analysis – effect of turbine inlet temperature.

It is clearly shown that, in order to maximize specific thrust, a low compressor pressure ratio of below 4:1 is required. Cycle analysis at high subsonic flight speeds also indicates that specific power is maximized at a compressor pressure ratio range of 4 to 5.

This figure shows that increases in thrust per pound of air also would result from higher burner temperatures, and therefore the possibility of using a cooled turbine is suggested by this cycle analysis. Further design study indicated that the addition of turbine cooling for a small gas temperature increase of 100-200°F would be impractical because of 1) the decrease in turbine efficiency resulting from thicker trailing edges, 2) the compromised airfoil shape associated with incorporating the blade internal geometry for cooling passages, and 3) the loss of cooling air to the cycle, which would offset the apparent cycle performance gain. It also was concluded that a substantial temperature increase such as 400°F would be required to offset the performance losses caused by cooling. Although this temperature increase would result in a 20% reduction in airflow and a 10% decrease in compressor diameter, the combustor and turbine size would not be reduced unless the compressor pressure ratio were increased by an additional stage. This study confirmed that cost would be minimized by elimination of turbine cooling.

Figure 1 also presents the effect of pressure ratio and burner temperature on specific fuel consumption. This illustrates that at the selected pressure ratio specific fuel consumption is only moderately increased as burner temperatures above 1500°F are selected. This also is characteristic for the subsonic condition.

Since increased burner temperatures produce increased specific power, the design objective for the expendable turbine engine was to use the highest turbine inlet temperature possible with existing turbine alloys without introducing the mechanical design complexity and the performance and cost penalties associated with turbine cooling. Consequently, the performance of the combustor in terms of gas temperature uniformity and radial profile became a most significant factor in the overall engine design and cost.

Expendable turbine engine cost analyses have shown that the compressor is almost 50% of the total cost of the engine where an uncooled turbine design is incorporated; the turbine represents 14%, the combustor is 11%, the control, accessories, and bearings are 20%, and the inlet and exhaust housings are the remaining 5%. However, a turbojet design using an air-cooled turbine produces both a marked change in cost distribution and a substantially increased total cost for the engine. The air-cooled turbine would represent about 40% of the overall engine cost. It was concluded that a low-cost engine requires a high temperature rise combustor, but one that presents very low gas temperature distortion entering an uncooled turbine.

Finally, in order to achieve the low-cost objectives for the limited life gas turbine, special design features must be incorporated to simplify manufacture and minimize assembly time. Key elements, which were most effective in minimizing the cost of this engine, are as follows:

1) Unitized compressor rotor – each compressor stage is an integral investment-cast blade and disk assembly requiring minimal machining. The stages and shaft are electron-beam-welded together to form a unitized rotor.

2) Uncooled turbine.

3) No accessory drives – the alternator is driven directly off the main shaft at the front. A turbine impingement starter uses gas from a solid propellant to provide cranking energy.

4) All circumferential joints between housings are welded to avoid the cost of machining and bolting flanges.

5) Minimum number of parts – dual functions for parts are arranged, such as the alternator rotor also serving as the front bearing retaining nut. The number of parts in the assembly involves slightly more than one hundred details.

6) Low alloy materials – selection of 17-4 PH steel for the compressor, type 321 stainless alloy for the combustor and engine casings was made.

Many of these design features are shown in the engine design arrangement presented in Fig. 2.

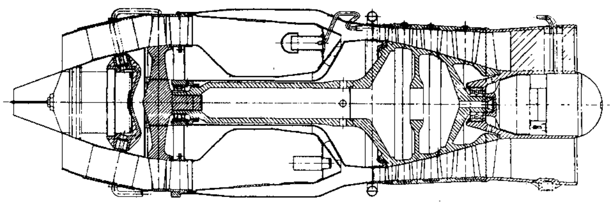

Fig. 2 Low-cost turbojet design features.

Combustor Performance Goals

The combustor performance design goals were established by engine performance requirements. Some of the desired combustor characteristics have a strong influence on successful engine operation and, therefore, must be satisfied through development; other characteristics may be accommodated by performance adjustments in other components. Table 1 presents the combustor design goals, the areas of engine performance affected by these goals, and the significance on engine performance if the component goal is not achieved completely.

Figure 3 presents the temperature profile goals for the combustor design. The maximum profile represents the envelope of the highest temperature that may occur at any circumferential location at the combustor exit. This is sensed by one or more turbine stator vanes. Since this profile is a composite of the peak temperature at each radius, and all peaks do not occur at one circumferential location, then any one vane is exposed to only a portion of this profile. Notwithstanding, the vanes are designed for exposure to the full temperature profile.

The mean profile is the envelope of the circumferential average temperature at the combustor exit and is used in determining the temperature envelope that the turbine rotor feels. The relative gas temperature represents the temperature the turbine rotor blades sense when operating at design rpm.

Combustor Design Requirements

The combustor concept that was selected for the expendable engine is a full annular combustion chamber, incorporating a

Table 1 Effect of combustor design goals

Combustor parameter	Design goal	Effect on engine	Significance on engine performance
Efficiency	98%	SFC, stability range, off-design efficiency, emissions	Moderate
Temperature profile factor	0.19	Turbine life	High
Pressure drop	7.3%	SFC	Low
Smoke	Invisible	Vulnerability	N.A.
Durability	1 hr	Engine or turbine life	High

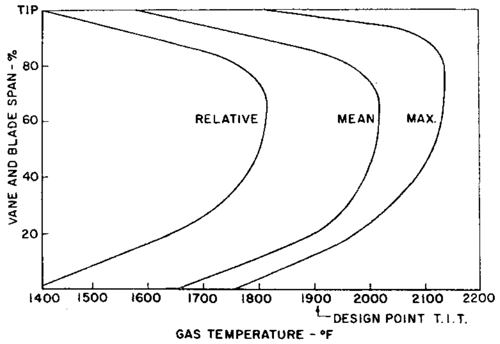

Fig. 3 Combustor exit gas temperature profiles.

vaporizing fuel injection system, consisting of ten "mushroom" shaped vaporizer tubes. The design requirements that led to the selection of this concept include: 1) envelope, 2) performance, and 3) cost.

Envelope

The engine envelope limitation of 14 in. maximum diameter and 44 in. maximum length dictated an uprated level of combustor technology. This requirement resulted in a design with a high reference velocity (135 fps) and a high space rate (9.7 million Btu/hr ft^3 atm). Furthermore, diluent air entry was only feasible through the combustor outer liner, since the available annular space between the inner liner and the engine shaft heat shield precluded reasonable flow rates or velocities in the passage to the inner liner diluent station. This situation increases the difficulty in providing a low exit temperature distortion for either fuel vaporizing or atomizing combustor systems.

The envelope constraint also required that a short length diffuser system be provided to accept a compressor exit Mach number of 0.35 at design point, and to reduce the air velocity and to split and meter the flow to multichannel annular passages at the combustor entry station.

Performance

The performance requirements of the combustor increased the difficulties imposed upon it by engine envelope constraints. A high temperature rise of 1228°F in this short length combustor and an efficiency of 98% were required. Furthermore, a temperature distortion factor T_{max}/T_{avg} of 1.10, a mean to average temperature ratio of 1.05, and a temperature pattern factor $(T_{max}-T_{avg})/(T_{avg}-T_{inlet})$ of 0.19 were required to assure satisfactory turbine life at a combustor average exit temperature of 1900°F. Performance requirements for this compact combustor also include supersonic flight over a wide operating envelope from sea level to 35,000 ft and light-off capability at altitude.

Cost

Conventional combustor designs utilize a series of axially positioned machined or sheet metal liner rings with air metering holes circumferentially arranged to provide splash-plate film cooling of the gas-side surface of the superalloy liner. In order to satisfy the requirement for low cost, the vaporizing combustor design for the expendable engine was able to take advantage of the nonluminuous low radiative flame characteristic (10^5 Btu/hr ft^2), so that a simpler liner cooling scheme could be used. Here a single perforated sheet of stainless steel is wrapped to form the cylindrical outer or inner liner. The annular arrangement provides a symmetry that enhances the manufacture and producibility of the component.

Finally, the fuel vaporizing concept is a low-pressure fuel distributor system, so that relatively large diameter fuel tubes with finished-cast fuel vaporizer turbes are used. A fuel delivery pressure of 50 psi above combustor pressure is required.

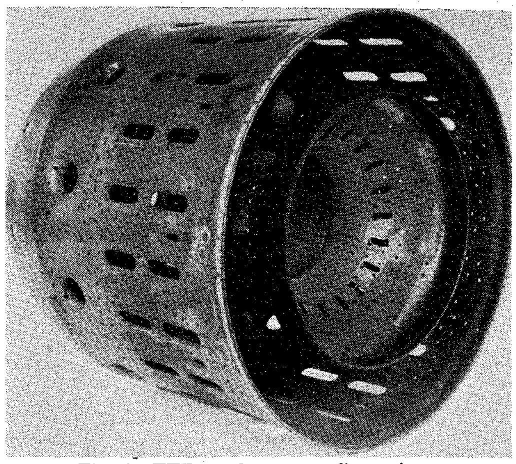

Fig. 4 ETE combustor configuration.

Combustor Design Description

The combustor and housing assembly is a welded composite annular structure consisting of a combustor basket, diffuser, outer housing, inner housing or shaft heat shield, fuel vaporizing tubes, pyrotechnic igniter, fuel manifold, and fuel metering tubes. AISI-type 321 stainless steel is used throughout, except for the vaporizer tubes located in the primary zone which are cast of Hastelloy X.

The combustor basket shown in Fig. 4 consists of an inner and outer liner, a headplate, and a diffuser flow splitter section. The liners are made of perforated sheet metal. The perforated sheet concept provides a boundary layer of cooling air on the hot-side wall surfaces. The inner and outer liners have large radial injection holes for primary and diluent air mixing. The liners are connected together at their forward end by the headplate to form an integral basket. The flow splitter rings, which form the annular air inlets, project from the front of the basket and serve to split the compressor exit airflow among inner liner, outer liner, and headplate flow paths. The basket is supported at the front by the inner housing through ten flat struts and at the rear by a slip joint at the forward edge of the turbine stator inner and outer shrouds.

The vaporizer tubes are tack-welded to the headplate. The metering tubes, which supply fuel to the vaporizer tubes, are connected to a torodial manifold circumscribing the combustor housing. The combustor inlet diffuser system was designed to diffuse the compressor exit air initially in a conical annular diffuser section, through an area ratio of 1.42, to reduce the air velocity before dividing into three components. Inner and outer channel flows were further diffused through a small area ratio of 1.12, while headplate flow was simply dumped into the headplate chamber.

The 3-way airflow splitting requirement at the combustor entrance region is delayed to a region of low velocity. This minimizes the turning losses, improves diffuser performance and stability by providing downstream blockage, and by avoiding splitters near the compressor exit, and minimizes sensivity of the diffuser and combustor to compressor radial profile changes. The inlet flow splitter established 73.5% of the compressor airflow to the outer liner for primary and diluent air mixing and liner cooling functions, 17.7% to the inner liner for the primary air and cooling functions, and 8.8% through the headplate for the primary air mixing function. Diffuser loss coefficients ($\Delta P/q$) of 15% for the outer liner passage, 17% for the inner passage, and 36% for the headplate were established for this design.

The combuster was designed to have approximately stoichiometric fuel-air mixture ratio in the primary zone. Airflow through the vaporizer tube was set for fuel carburetion and tube cooling. With this air allocation, fuel coking is precluded and carbon formation in the combustor

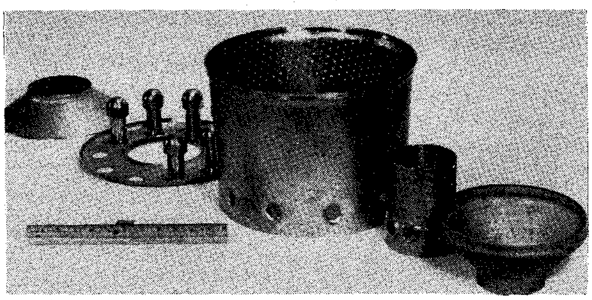

Fig. 5 Combustor low-cost manufacturing details.

primary zone is minimized. The fuel is directed from the fuel tube into the vaporizer tube inlet with low injection velocity. The fuel mixes with the air in the vaporizer tube and is directed into the combustor in a reverse flow direction towards the combustor headplate. Fuel vaporization and combustion with the remaining primary zone airflow introduced through the outer and inner liner air injection ports occurs in the primary zone. Strong recirculatory patterns are set up in the primary zone by the interaction of the axial flow from the vaporizer tubes and the radially opposed flow issuing from the outer and inner liner primary air injection ports.

The remaining airflow is used for diluent air mixing and for cooling the liners through high velocity convective heat transfer and film cooling. The combustor's operating temperatures were calculated to be 1695°F for the inner liner and 1638°F for the outer liner. The diluent air distribution, which is initially established during the design from test data on similar combustor arrangements, normally requires empirical refinement more than any other aspect of the combustor design. The diluent air entry serves to set the combustor exit temperature profiles so that the life goals of the highly stressed turbine rotor and stator component may be achieved.

Combustor Manufacture

The low manufacturing cost of the combustor design was demonstrated during this program. The inner and outer liners were fabricated from 0.080 in. perforated flat sheet stock by 1) wrapping the sheet into a cylinder, 2) TIG (tungsten inert gas) welding the butted edges, 3) planishing the weld joint section, and 4) expansion die forming the fore and aft section diameters, to provide the liner contour and to match the adjacent details.

The primary zone air mixing holes were produced by an undersize piercing die when the liner was in flat sheet form. After the liner was formed, the holes were enlarged and plunged to form a radially inward lip. The outer diffuser case was also produced from 0.080-in. flat sheet stock, formed into a cylinder and contoured by expansion die forming. The headplate was fabricated from 0.080-in. flat sheet stock, using the drop-hammer forming process. The holes in the headplate, which are provided for installation of the cast vaporizer tubes, were produced by die punching the flat sheet stock prior to forming.

Figure 5 shows the semifinished detail elements of the combustor prior to preparation for welding into a combustor basket. The cost of the combustor is estimated at under $600 based on production lots of 1000/year. The manufacturing processes for each detail were maintained at minimum cost by 1) avoiding machining, 2) using perforated sheet for the liners, and 3) prepunching primary and diluent air mixing holes in the flat sheet. The loosely controlled hole sizing and location resulting from these techniques did not appear to affect combustor performance. This was evident when combustor serial No. 2 provided a repetition of the gas temperature profile observed in the test evaluation of the serial No. 1 unit, despite the dimensional variations, which were larger than are expected for the production tooling.

Test Approach

The combustor performance and durability test evaluation was conducted in a test rig connected to a test facility air supply. Testing was conducted in the Development Laboratory located at Curtiss-Wright's Wood-Ridge, N. J. facility. The facility has the capability of providing the test rig with about 35 lb/sec airflow at pressures up to 6 atm. A facility air heater capable of up to 1200°F provides non-vitiated air to the combustor rig at the temperature, simulating any compressor exit condition over the flight envelope.

The combustor test rig housings were designed to simulate the geometric and aerodynamic environment surrounding the engine combustor basket, so that the design predictions of critical performance characteristics could be confirmed.

The combustor performance data were obtained by means of a rotating probe assembly, which is located at the exit plane of the combustor. Exit temperature was measured by means of a five-element rake and exit pressures by a three-element rake. These rakes are mounted on a rotating centerbody, which provides a 360 deg sweep of the combustor exit annulus. The centerbody is supported by an outer housing through four air-cooled radial supports. The rakes were constructed of high-melting-point platinum alloy in order to operate wihtout cooling. Platinum/platinum-rhodium alloy was used for the thermocouple sensing elements. The rakes were positioned in the same plane as the leading edge of the turbine vanes, and the radial position of each sensing element was located at the center of equal annular areas.

Gas temperatures and pressures were recorded continuously during the complete annular survey of the combustor exit plane on *X-Y* recorders for continuous visual observation of combustor performance during the setting of each test point. The temperature data to be used for analysis also were recorded every 3 deg of probe circumferential position by an automatic data acquisition system, which records millivolt output of the sensing element onto a punched tape. The tape is processed through a computer terminal in the laboratory and computer program analysis of combustor performance is obtained during the test.

Chromel/alumel thermocouples were used to measure metal temperatures on the combustor inner and outer liners and the vaporizer tubes. Multiple-point strip-chart recorders recorded this data. Temperature-sensitive paints were applied to the combustor for additional data in determining location of peak metal temperatures and range of metal temperature gradients.

Two combustor baskets were evaluated during a 110-hr test program. The initial combustor was tested for over 63 hr, primarily to develop the diluent port design. This experimental evaluation was performed on a total of nine design modifications. The second combustor was tested for 14 hr to verify the performance of the final diluent design configuration and to confirm the repeatability of exit gas temperature uniformity and profile for a combustor design using low-cost punch-perforated sheet metal manufacturing techniques.

Initial testing was conducted to confirm predicted airflow distribution, diffuser performance, liner metal temperatures, and primary zone efficiency. Subsequent tests representing the major portion of the development effort were directed at obtaining the design combustor outlet temperature profile characteristics. The initial test showed the as-designed diluent configuration had an excessive TPF (temperature pattern factor) in 14% of the combustor exit annulus area.

The combustor airflow distribution was determined initially because all subsequent performance tests are strongly influenced by airflow distribution. Furthermore, airflow distribution in a combustor is difficult to predict because of the accuracy involved in establishing hole flow discharge coefficients in high velocity and rapidly turning passages surrounding the combustor, and the sensitivity of airflow distribution on pressure recovery in the multichannel diffuser design.

A simplified technique for measuring airflow distribution involves a series of installations of the combustor in the test rig. For the first installation, all airflow ports, except for one region (i.e., vaporizer tubes in the headplate), are covered with metal adhesive-backed tape. An airflow test is conducted over a range of pressure drops across the combustor. During a series of subsequent combustor reinstallations, tape is systematically removed from other sections of the combustor, and airflow checks are repeated. This procedure is repeated until all airflow ports are uncovered. These tests confirmed the design predictions for airflow distribution.

Results of Test Evaluation

The results of the experimental evaluation indicated that the combustor design goals had been accomplished satisfactorily, and are discussed in the following paragraphs.

Efficiency

The combustor efficiency was calculated from the ratio of the measured to the ideal temperature rise. The measurements consisted of over 500 gas temperature readings at the combustor exit annulus, plus inlet air temperature, calibrated airflow and fuel flow, and the calorimetric analysis of JP5 fuel for actual heating value. The ideal temperature rise was based on 18,550 Btu/lb. The shielded thermocouple and the uncooled exit duct of the combustor rig reduced thermocouple probe radiation and conduction errors to insignificant levels.

The combustor demonstrated stable and efficient combustion at all conditions, which include a range of inlet air pressures from 1-6 atm and inlet air temperatures from ambient to 700°F. Design point efficiencies of 99% or better were measured consistently. Also, no sign of smoke was visible. The stability range was not explored fully, since a broad stability range is not required for the planned engine application. Nevertheless, testing at 20% of design fuel-air ratio was fully stable.

Durability

The combustor exhibited excellent durability over a test period that exceeded the required design life by orders of magnitude. Combustor serial No. 1, which was used for the major portion of diluent air mixing port development, was in excellent condition upon post-test inspection, after 63 hr of testing. The liner metal temperatures measured during this test program at design point conditions are presented in Fig. 6. These data appear to confirm the hardy durability demonstration of the design. Figure 7 shows the excellent condition of the primary zone after 34 hr of testing.

Exit Temperature Profile

Each of the five elements in the temperature rake that was used to establish the exit gas temperature in the combustor annulus provided over 100 readings in a 360-deg circumferential survey. From these data, the final combustor outlet temperature profile at the design point was obtained, and is presented in Fig. 8. This figure presents a plot of isotherms seen by the turbine stator leading edge. Contour islands of constant temperature are plotted in terms of the ratio of local gas temperature to overall average gas temperature of the entire exit annulus. The figure shows three very small regions, totalling an annulus area of less than 1%, which have a peak temperature of 50°F higher than the goal. The highest temperature region in the exit gas stream is located at approximately 230-deg circumferential position. However, all locations at which the temperature ratio exceeds 1.10 fall between turbine vanes, and span only about 20% of the annulus height. Relocating the turbine vanes by rotating the assembly several degrees can accommodate a shift in the peak temperatures due to compressor swirl. Furthermore, these localized temperatures on the vanes are acceptable in the area of the vane midsection where the stresses are low. Therefore, the effect of these small temperature peaks on turbine durability is not considered to be significant.

Figure 9 presents the stress rupture margin for the turbine stator based on the actual radial temperature profile of the gas at 210-deg location. The minimum rupture strength of the cast WI-52 alloy for the turbine vanes is shown to be substantially above the calculated stresses in the vane. Although the actual temperatures near the turbine i.d. shroud are slightly higher

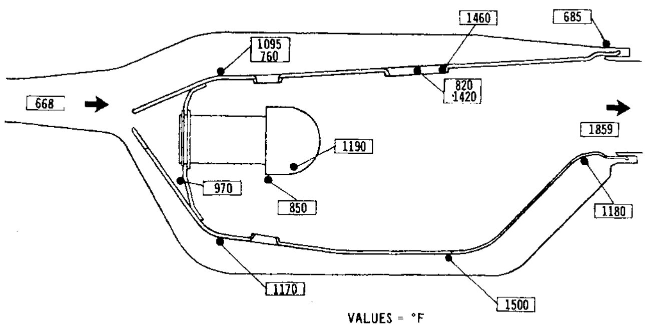

Fig. 6 Combustor liner metal temperature survey (test point 65, 6 atm).

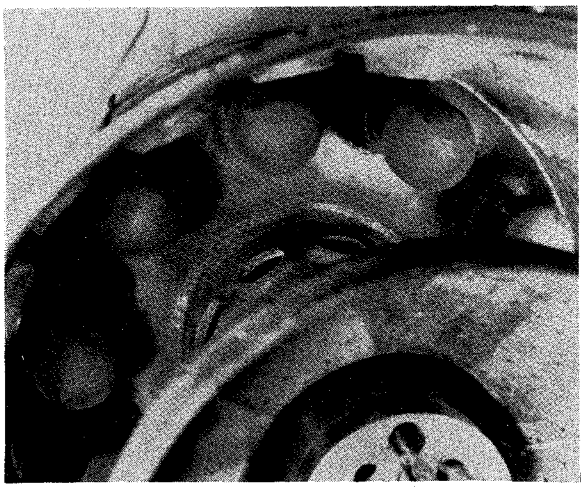

Fig. 7 ETE combustor headplate after 34 hr.

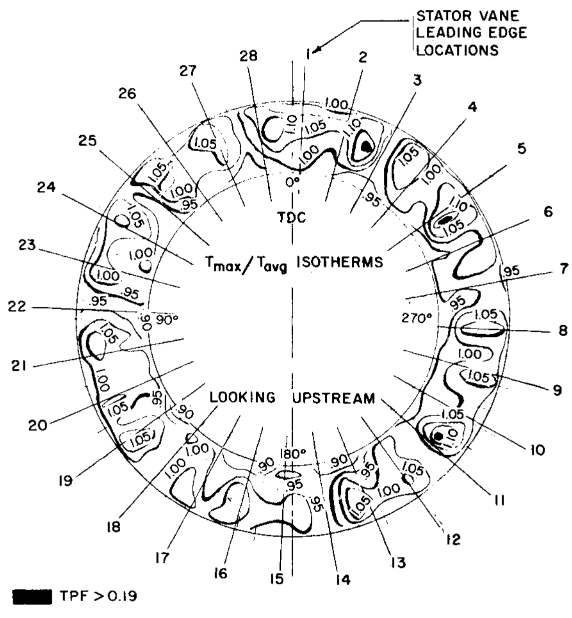

Fig. 8 Exit temperature contour map.

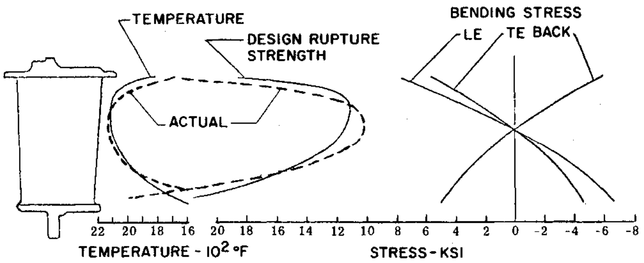

Fig. 9 Turbine stator vane stress and temperature summary.

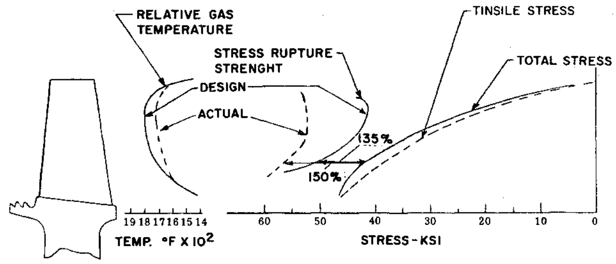

Fig. 11 Turbine rotor blade stress and temperature summary.

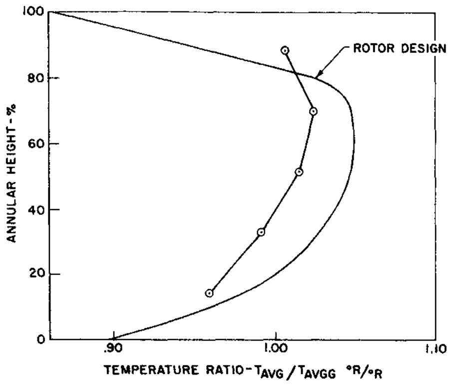

Fig. 10 Exit temperature profile (rotor).

than the predicted profile at other circumferential locations, this cobalt-base alloy still provides adequate strength margin at the shroud to vane attachments. The vane end attachment has been designed to support the full gas bending load at either end. The end mounted vane design provides the desirable feature of very low stresses in the midspan region, where peak gas temperatures normally occur.

Figure 10 shows the average radial gas temperature profile for the developed combustor at design point. This influences the operating temperature and life of the turbine rotor blades. The radial profile is presented as a ratio of the average gas temperature at a specified radius for the 360-deg circumferential survey to the overall average gas temperature of the entire exit annulus. The figure shows that the actual radial temperature profile is considerably less peaked than the design goal.

This measured total temperature profile is used to calculate the relative gas temperature sensed by the rotor blades which is shown in Fig. 11. It is noted that the rotor blade relative temperature is 184°F lower at the hub than the measured combustor exit gas temperature, and 243°F lower at the tip.

The relative gas temperature profile is used to establish the stress rupture strength of the blades. The comparison of the actual and design predicted stress rupture margin also is presented in Fig. 11. The data indicate that the measured combustor exit profile has resulted in an improved stress rupture margin for the blades. The effect of this condition on engine performance is significant in two possible ways. This increased margin at the design temperature provides a substantially longer turbine life. Alternatively, the combustor exit average temperature may be raised by 50°F or more, which would either produce increased engine power or accommodate unforeseen deficiencies in other engine components.

The design decision for diluent air mixing holes size and locations was influenced by historical data, which indicate that the exit gas temperature peaks occur in-line with primary air mixing ports. Therefore, ten diluent ports were located in the outer liner, in-line with the primary air ports, and ten smaller diluent ports were located in-line with the vaporizer tubes.

An investigation was conducted to evaluate the use of a single axial plane for the location of the diluent ports. Since the diluent entry ports were limited to the outer liner, diluent holes with plunged lips to promote penetration also were investigated. The study indicated that the radially inward lip of the plunged hole would operate at excessive metal temperatures and tend to shed. The results also indicated that a single diluent plane would violate the requirement that the minimum distance between diluent hole circumferential spacing should be 2 to 3 hole diameters. Closer spacing between diluent ports tends to inhibit mixing in the diluent zone.

The selected diluent port configuration consisted of 20 square-edge rectangular ports in each of two axial planes. The ports all are approximately the same size. The ports are in-line, since a staggered arrangement would have resulted in the same effect on the gas temperature profile as a single-plane configuration with undersize spacing between ports. The slots were designed with a length-to-width ratio of 3:1 in order to enhance mixing and penetration for the final development tests. Ten small slots, representing less than 4% of the area of the main diluent ports, were provided to reduce the localized peak gas temperature near the outer wall. A similar set of 20 slots was provided in the inner liner rear cone to offest a similar pattern of peak temperatures near the inner wall.

Total Pressure Loss

The total pressure loss characteristic for the combustor system including diffuser amounted to only 93.6% of the compressor discharge dynamic pressure. This relatively low design goal resulted from the attempt to provide rated engine thrust at a minimum turbine inlet temperature. Although low pressure drop inhibits aerodynamic mixing and increases the difficulty in attaining design temperature profiles, the desired temperature profiles and pressure drop were achieved with the final dilution arrangement. It is interesting to note that increased combustor pressure drop does not have a severely penalizing effect at the supersonic sea level engine design condition, since an additional 1% combustor pressure drop is equivalent to less than 8°F in turbine temperature rating.

Ignition

A high-energy electrical spark ignitor and a fuel primer were used for repetitive light-offs during the gas temperature profile development tests. However, in order to demonstrate the proposed low-cost ignition concept for this engine, ignition tests also were conducted using a cartridge of solid propellant of zirconium nitrate. Ignition is initiated by a 24-V dc bridgewire, which is used to light an initiator charge. The initiator lights the column of main grain propellant sized to burn for 8 sec and to emit a plume of flame 4 to 6 in. in length.

Combustor light-off was achieved successfully with the ignitor installed radially and flush with a port in the outer liner. The ignitor was located axially in the plane of the vaporizer tube dome. The plume of flame from the ignitor was directed forward. Simulation of compressor flow during engine starting was set at a combustor pressure drop of 0.2%. A fuel-air ratio of 125% of design was established for successful light-offs. Light-off did not occur when directing the plume radially inward. These tests demonstrated that the pyrotechnic concept was viable for the expendable engine combustor light-off and that the envelope of air fuel flow conditions was reasonable.

Fuel Distribution

A calibration of the low-cost fuel system was conducted to determine flow characteristic and distribution. The results show excellent fuel distribution at the high-power or high-combustor outlet temperature operating conditions. Fuel flow deviation in excess of the average for any metering tube was less than 5%. The flow distribution at start and very low power conditions exhibited a larger deviation, in excess of 30%. The larger flow deviation is attributed to 1) the pressure head effect in the fuel manifold circumscribing the combustor, and 2) the very low rates through a simple orifice provided by swaging the fuel tube tip to a selected diameter. However, the flow deviations at these conditions are not serious, since the combustor outlet average temperatures are not high. Analysis of the system fuel flow requirements for the range from the sea level static maximum power condition to the maximum power condition over the flight envelope indicates that the total fuel deviation in excess of the average flow is less than 3%. This provides assurance that the use of a low-cost fuel system is not expected to affect engine performance or durability.

Conclusions

Based upon the results of the experimental test evaluation, the low-cost sheet metal combustor and diffuser design satisfied all performance goals including efficiency, temperature pattern factor, and pressure drop, and exceeded the durability requirements. The use of radial entry primary air injection for the fuel vaporizing combustor system, instead of the conventional design using axial entry primary aircups mounted contiguous to the vaporizer tubes, was completely successful, since no change to the original design of the primary zone was necessary during the development program.

It was noted that the combustor exit gas temperature profile achieved through the 110 hr of development testing is more favorable than design predictions. This condition has resulted in an increased performance capability for the engine. Turbine life is substantially greater than that required for any expendable application, and engine power ratings may be increased through increased turbine inlet temperature.