(NASA-CR-167945) ENERGY EFFICIENT ENGINE COMBUSTOR TEST HARDWARE DETAILED DESIGN REPORT (United Technologies Corp.) 105 p HC A06/MF A01 CSCL 21E
N85-10950
Unclas
G3/07 24781

NASA CR-167945
PWA-5594-197

ENERGY EFFICIENT ENGINE COMBUSTOR TEST HARDWARE DETAILED DESIGN REPORT

by

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UNITED TECHNOLOGIES CORPORATION
Pratt & Whitney Aircraft Group
Commercial Products Division

Prepared for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Lewis Research Center
Contract NAS3-20646

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In reply please refer to:
WBG:WS (0317K) 118-35
LC-83-30

17 June 1983

To: National Aeronautics and Space Administration
Lewis Research Center
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Attention: Mr. Carl C. Ciepluch, Mail Stop 301-4

Subject: Submittal of Energy Efficient Engine Combustor Test Hardware Detailed Design Report, CR-167945 (PWA-5594-197)

Reference: (a) Contract NAS3-20646
(b) Ltr EEE-P250 dated 29 June 1982

Enclosure: Twenty (20) copies of the subject report

Gentlemen:

In accordance with the references (a) and (b), we are pleased to submit twenty (20) copies of the subject report.

Sincerely yours,

UNITED TECHNOLOGIES CORPORATION
Pratt & Whitney Aircraft Group
Commercial Engineering

W B Gardner

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Program Manager

1. REPORT NO.	2. GOVERNMENT AGENCY	3. RECIPIENT'S CATALOG NO.
NASA CR-167945		

4. TITLE AND SUBTITLE	5. REPORT DATE
Energy Efficient Engine Combustor Test Hardware Detailed Design Report	MARCH 1982
	6. PERFORMING ORG. CODE
7. AUTHOR(S)	8. PERFORMING ORG. REPT. NO.
M. H. Zeisser, W. Greene and D.J. Dubiel	PWA-5594-197
9. PERFORMING ORG. NAME AND ADDRESS	10. WORK UNIT NO.
UNITED TECHNOLOGIES CORPORATION Pratt & Whitney Aircraft Group Commercial Products Division	11. CONTRACT OR GRANT NO. NAS3-20646
12. SPONSORING AGENCY NAME AND ADDRESS	13. TYPE REPT./PERIOD COVERED
National Aeronautics and Space Administration Lewis Research Center 21000 Brookpark Road Cleveland, Ohio 44135	Contractor Report 14. SPONSORING AGENCY CODE

15. SUPPLEMENTARY NOTES

Combustor Program Manager: Daniel E. Sokolowski, NASA Lewis Research Center, Cleveland, Ohio 44135

16. ABSTRACT

The combustor for the Energy Efficient Engine is an annular, two-zone component. As designed, it either meets or exceeds all program goals for performance, safety, durability, and emissions, with the exception of oxides of nitrogen.

When compared to the configuration investigated under the NASA-sponsored Experimental Clean Combustor Program, which was used as a basis for design, the Energy Efficient Engine combustor component has several technology advancements. The prediffuser section is designed with short, strutless, curved-walls to provide a uniform inlet airflow profile. Emissions control is achieved by a two-zone combustor that utilizes two types of fuel injectors to improve fuel atomization for more complete combustion. The combustor liners are a segmented configuration to meet the durability requirements at the high combustor operating pressures and temperatures. Liner cooling is accomplished with a counter-parallel FINWALL® technique, which provides more effective heat transfer with less coolant.

17. KEY WORDS (SUGGESTED BY AUTHOR(S))

Advanced Diffuser Aerodynamics
Two-Zone Combustor
Segmented Liner with Advanced Cooling
Energy Efficient Engine
Emissions Reduction Technology

19. SECURITY CLASS THIS (REPT)	20. SECURITY CLASS THIS (PAGE)	21. NO. PGS	22. PRICE *
UNCLASSIFIED	UNCLASSIFIED		

FOREWORD

The Energy Efficient Engine Component Development and Integration Program is being conducted under parallel National Aeronautics and Space Administration (NASA) contracts with Pratt & Whitney Aircraft Group and General Electric Company. The overall project is under the direction of Mr. Carl C. Ciepluch with Mr. John W. Schaefer serving as NASA's Assistant Project Manager for the Pratt & Whitney Aircraft effort under NASA contract NAS3-20646. Mr. Daniel E. Sokolowski is the NASA Project Engineer responsible for the portion of the project described in this report. Mr. William B. Gardner is Manager of the Energy Efficient Engine Program at Pratt & Whitney Aircraft Group with Messrs. M. H. Zeisser, W. Greene and D. J. Dubiel the engineers responsible for work described in this report.

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SECTION 1.0
SUMMARY

The combustor component for the Energy Efficient Engine is an annular, two-zone configuration. As designed, this component either meets or exceeds all program goals for performance, safety, durability, and emissions, with the exception of oxides of nitrogen.

The Energy Efficient Engine combustor component has several advanced features to enhance aerothermal performance and durability without compromising emissions reduction. The prediffuser has short, strutless, curved walls to provide a uniform combustor inlet profile. The combustor is a two-zone design with a refined fuel supply system. It employs two types of fuel injectors to improve fuel atomization for more complete burning. The liners are segmented for better durability at the high operating pressure/temperature conditions envisioned for the Energy Efficient Engine. The segments are constructed from a high-temperature capability turbine alloy (B-1900 + Hf), and cooling is accomplished with an advanced counter-parallel FINWALL® technique. This technique permits more effective heat transfer with less coolant.

Of particular importance are the results acquired from the related Sector Combustor Rig Test Program, in which several combustor configurations were tested at actual high pressure and high temperature conditions anticipated for the Energy Efficient Engine. Table 1-I presents a comparison of program goals with demonstrated performance levels from this test program.

The combustor component, as designed for the Energy Efficient Engine, represents a considerable extension of current combustor technology that has application in future gas-turbine engines.

TABLE 1-I

DEMONSTRATED COMBUSTOR PERFORMANCE

Aerothermal Performance	Goal(**)	Demonstrated
Pattern Factor	0.37	0.26
Section Pressure Loss (% P_{T3})	5.50	5.22
Exit Radial Temperature Profile (maximum peak-to-average) Δ°C (Δ°F)	139 (250)	83 (150)
Emissions Parameters (EPAP)*		
Unburned Hydrocarbons	0.4	0.38
Carbon Monoxide	3.0	2.30
Oxides of Nitrogen	3.0	4.70
Smoke Number	20	4
Durability		
Life Capability	8000 hours/4900 missions	11,700 (estimated)

* Environmental Protection Agency Parameter - Includes Margins for Development and Variability

** Maximum for Aerothermal Performance and Emissions

SECTION 2.0
INTRODUCTION

The Energy Efficient Engine Component Development and Integration Program, sponsored by the National Aeronautics and Space Administration (NASA), is directed toward developing the technology to achieve greater fuel efficiency for future commercial aircraft gas-turbine engines. The overall goals for the program include a reduction in fuel consumption of at least 12 percent and a reduction in direct operating cost of at least 5 percent relative to the Pratt & Whitney Aircraft JT9D-7A base engine. To verify the technology to attain these goals, the Energy Efficient Engine Program is currently organized into three technical tasks:

Task 1	Propulsion System Analysis, Design, and Integration
Task 2	Component Analysis, Design and Development
Task 4	Integrated Core/Low Spool Design, Fabrication and Test

Under Task 2, an advanced combustion system has been designed for the Energy Efficient Engine. This combustor has several advanced features to provide a compact system that can reduce emissions and exit temperature pattern factor, plus meet commercial durability requirements. With only minor exceptions, the combustor design is the same for both the flight propulsion system, which is the analytical study engine in the Energy Efficient Engine Program, and the integrated core/low spool, the test vehicle for evaluating fuel-efficient technology concepts. Figure 2-1 presents the program schedule for the combustor component effort, including two supporting technology programs that provided both technical guidance and design substantiation.

This report presents the detailed design of the combustor component. The following section, Section 3, provides an overview of the combustor design. Section 4 describes the aerodynamic design of the component, while Section 5 presents the thermal-mechanical design. A synopsis of the work completed in the related supporting technology programs is presented in Section 6. In Section 7, the test program for evaluating the full annular combustor component is summarized. Concluding remarks are presented in Section 8.

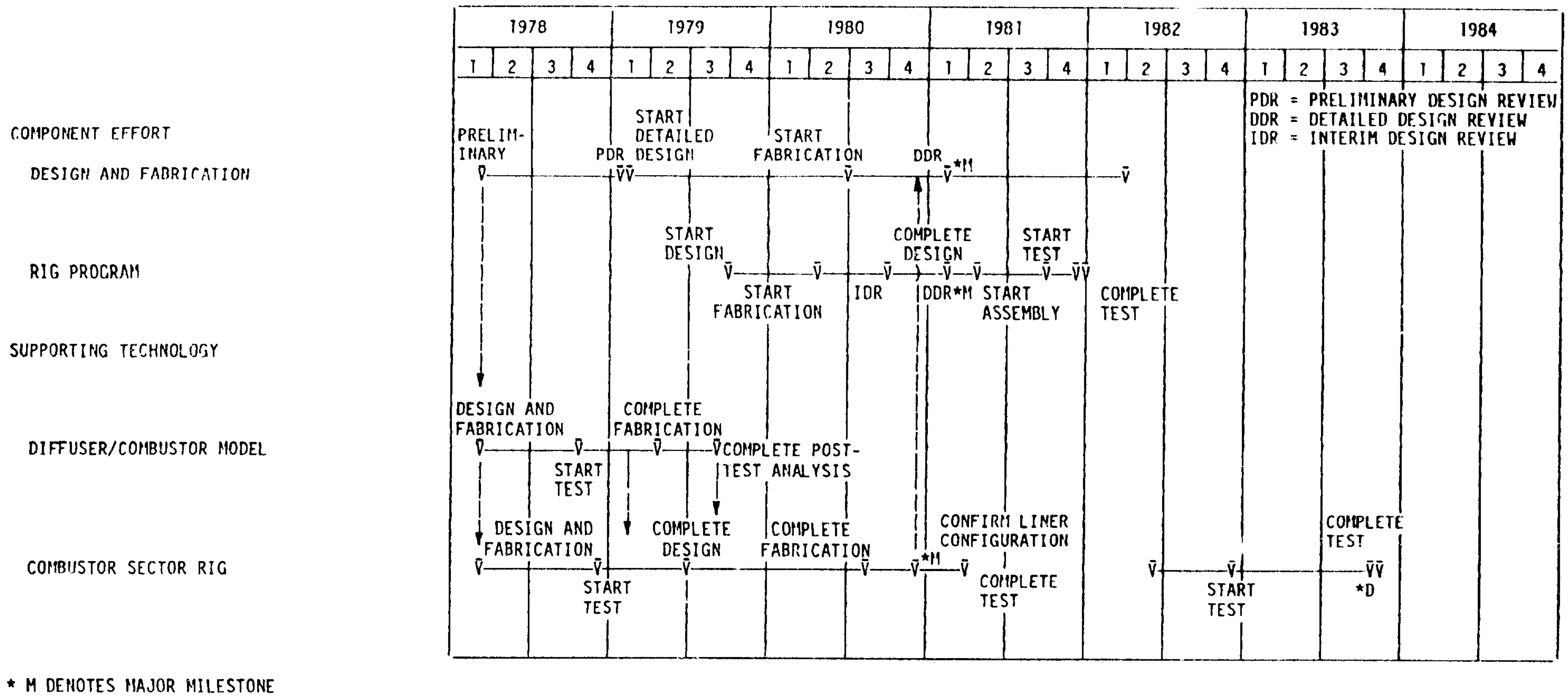


Figure 2-1 Combustor Component and Supporting Technology Program Schedule

SECTION 3.0
DESIGN OVERVIEW

3.1 COMBUSTOR SYSTEM DESIGN

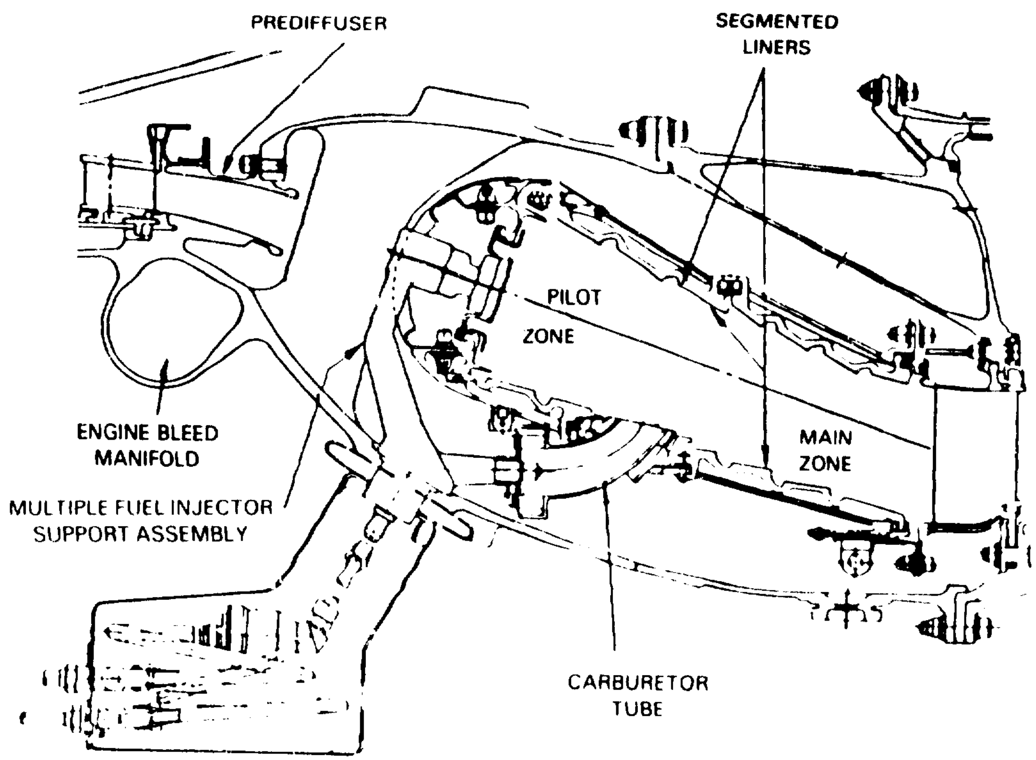
The Energy Efficient Engine combustor design is based on the technology investigated in the Experimental Clear Combustor Program sponsored by the National Aeronautics and Space Administration. It combines this technology with advances in the areas of aerodynamics and structure-mechanics to provide a compact system capable of low emissions and high performance. For commercial acceptance, the design also addresses durability, mechanical simplicity and the capability to operate on broad specification fuels. Figure 3.1-1 presents a cross-sectional view of the combustor and identifies the salient features. These include a high performance diffuser, two-zone combustion system and segmented liners.

The diffuser section consists of a short prediffuser and dump region and the inner and outer annuli around the combustor liners. The prediffuser flowpath is a strutless, curved-wall design that turns the high-pressure compressor exit flow towards the combustor center line to reduce pressure loss from flow turning in the combustor hood section. The dump region contains the structural struts which are designed as aerodynamic members of the diffuser to minimize combustor flow maldistribution and to enhance liner durability and pattern factor.

The combustor has two distinct combustion zones -- a pilot and a main zone. The pilot zone operates at all flight conditions and is designed to minimize emissions at idle, plus ensure adequate stability and relight characteristics. In the main zone lean combustion occurs to minimize oxides of nitrogen and smoke. This zone is operative at conditions above idle. In comparison to current single-zone combustors, this two-zone system provides more effective control of exhaust emissions throughout the flight spectrum.

Emissions reduction is further enhanced by the adaptation of the Vorbix combustion method (vortex mixing and burning) in the main zone. This approach, as demonstrated under the Experimental Clean Combustor Program, exploits the benefit of swirling airflow to promote a rapid and thorough mixing of the fuel and air for a more uniform combustion process. The fuel injector in the pilot zone is a single pipe aerated design that relies on the shearing action of low velocity fuel surrounded by high velocity air streams for improved atomization. In the main zone, a compact carburetor tube injection system mixes the fuel and air prior to introduction into the combustion zone for better atomization and lower smoke emissions.

Both combustion zones are enclosed by a unique segmented liner construction. Although it represents a considerable departure from conventional louvered designs, this concept offers the potential for superior durability, better maintainability and the capability to operate at higher combustion pressure levels. The segments, as shown in Figure 3.1-2, are cast from an advanced nickel base alloy. Effective heat transfer is provided with minimum cooling air by the use of a counter-parallel FINWALL® cooling technique.



ADVANCED TECHNOLOGY FEATURES

- **Segmented liner with advanced cooling technique**
- Short, curved wall, strutless prediffuser
- Two zone fuel injection
  - Single pipe aerated pilot injectors
  - Main zone carburetor tubes

Figure 3.1 1 Energy Efficient Engine Combustor Design

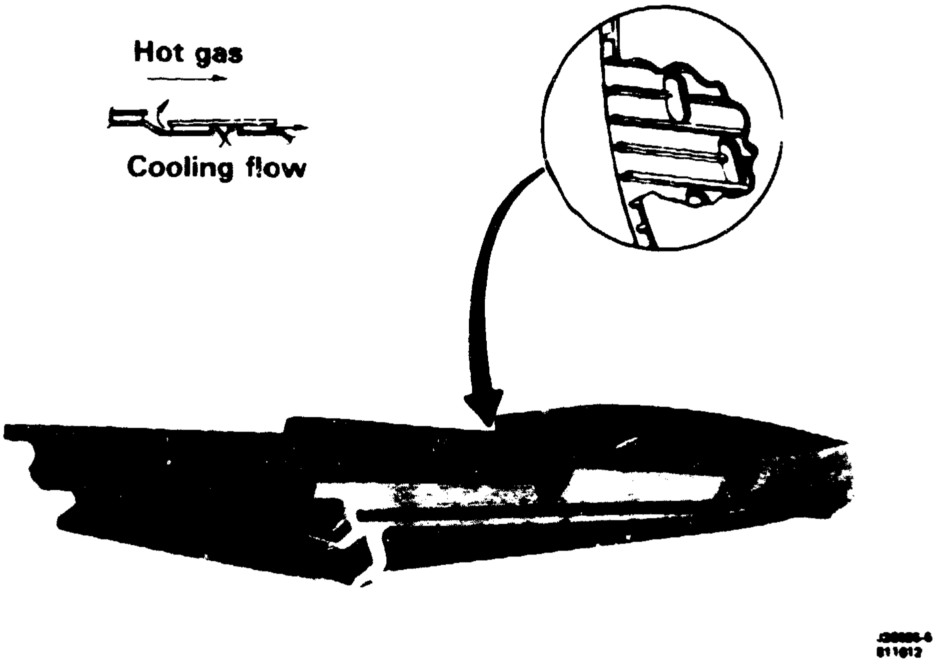


Figure 3.1-2 Advanced Segmented Liner With Counter-Parallel FINWALL® Cooling Technique

3.2 PREDICTED PERFORMANCE

The combustor performance goals are outlined in Table 3.2-I. This table also shows the levels demonstrated during a related supporting technology test program, using a 90-degree sector of the Energy Efficient Engine combustor component. As indicated, all performance goals were successfully met or surpassed. Emissions goals for carbon monoxide and unburned hydrocarbons were surpassed, while the goal for oxides of nitrogen, considered very difficult for high pressure ratio engines, was closely approached.

Testing also served to demonstrate the structural integrity of the advanced segmented liner design. This liner system offers a predicted life of 11,700 hours for the engine combustor component, which well exceeds the design goal of 8000 hours.

In summary, these results substantiate the technology features in the combustor design as well as the effectiveness of a two-zone system in reducing emissions.

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TABLE 3.2-I
ENERGY EFFICIENT ENGINE COMBUSTOR COMPONENT GOALS
AND DEMONSTRATED PERFORMANCE

	Goal (max)	Louvered Liner (2)		Segmented Liner (3)	
Exhaust Emissions					
Hydrocarbons (1)	0.4	0.18*	0.26**	0.26*	0.38**
Carbon Monoxide (1)	3.0	1.54*	2.07**	1.71*	2.30**
Oxides of Nitrogen (1)	3.0	3.80*	4.65**	3.85*	4.70**
Smoke, SAE Number	20		1		4
Pattern Factor	0.37		0.15		0.26
Section Pressure Loss, (%P_{T3})	5.5		5.37		5.22
Exit Radial Profile, Δ°C (Δ°F)	139 (250) (peak to avg)		39 (70)		83 (150)

(*) As measured
(**) Includes margins for development and variability
(1) Environmental Protection Agency Parameter (pound pollutant/1000 pounds-thrust hour/cycle).
(2) Tested up to 1.6 MPa (230 psia).
(3) Tested up to 3.1 MPa (445 psia).

SECTION 4.0
COMBUSTOR AERODYNAMIC DESIGN

4.1 AERODYNAMIC DESIGN GOALS

The design goals for the Energy Efficient Engine combustor component apply to both the future flight propulsion system and the integrated core/low spool. Figure 4.1-1 graphically presents the combustor exit temperature goals. As indicated, the maximum pattern factor at sea level takeoff conditions is 0.37. The average exit radial temperature profile factor is 0.17 with a 139°C (250°F) peak to average differential limit. Altitude relight and lean blowout characteristics must be adequate to allow safe aircraft operation in commercial service.

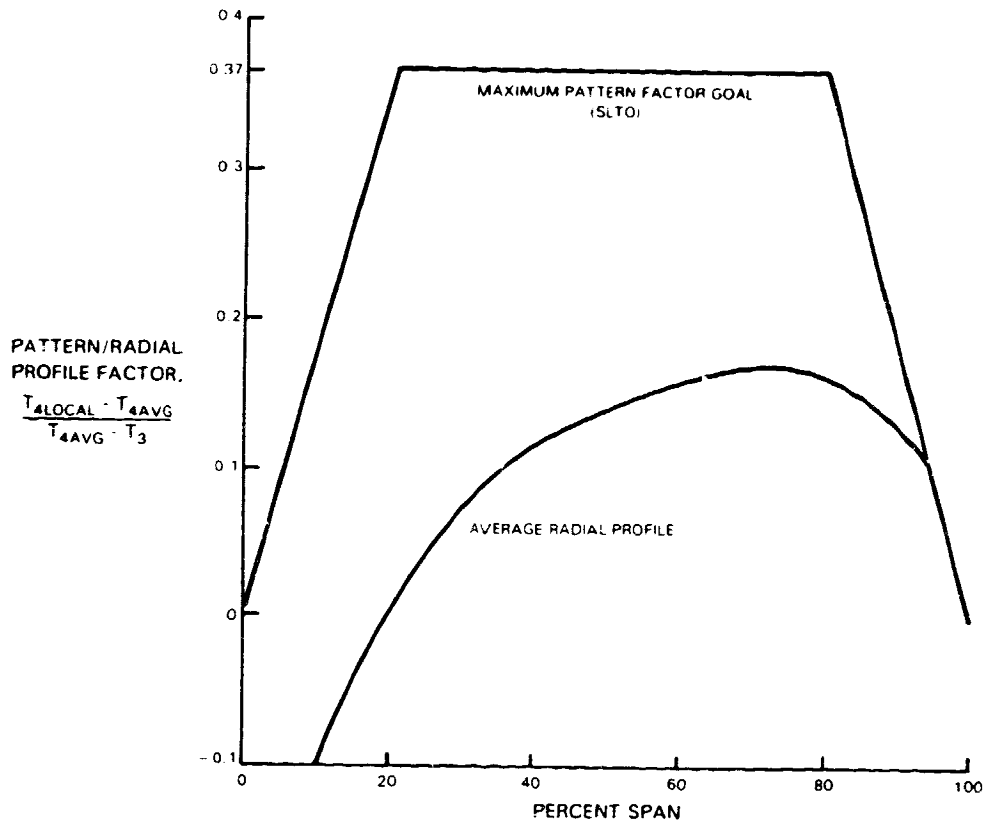

Figure 4.1-1 Combustor Exit Temperature Goals

The total pressure loss goal of 5.5 percent of the total compressor exit pressure is based on a diffuser pressure loss of 3.0 percent and a liner pressure loss of 2.5 percent. The diffuser loss includes the attendant losses in the prediffuser, dump and inner and outer annuli. The 2.5 percent liner loss goal is necessary to ensure a positive turbine cooling supply pressure.

Emissions goals are consistent with the original 1981 Environmental Protection Agency emissions regulations to control exhaust pollutants for newly certified engines below an altitude of 914 m (3000 ft). Values for carbon monoxide (3.0), unburned hydrocarbons (0.4) and oxides of nitrogen (3.0) are based on the Environmental Protection Agency Parameter of pound pollutant per 453 kg (1000 lb) thrust hour cycle. The smoke goal is a maximum Society of Automotive Engineers (SAE) smoke number equal to or below 20 over the range of operation.

4.2 AERODYNAMIC DESIGN APPROACH

The basic aerodynamic definition of the combustor was established during the preliminary design phase of the Energy Efficient Engine Program. This preliminary definition was predicated on design criteria that reflected both the program goals and overall engine cycle requirements. It was also influenced significantly by the experience acquired from the earlier Experimental Clean Combustor Program. In essence, the Energy Efficient Engine combustor design is an evolutionary extension of the Vorbix combustor technology evaluated under the Experimental Clean Combustor Program.

The final component design was influenced largely by the results of the Diffuser/Combustor Model Test Program and the Sector Combustor Rig Test Program, which were in process concurrent with the design effort. These programs provided diagnostic and technical guidance as well as design substantiation.

4.2.1 Vorbix Type Combustor

The two-zone Vorbix combustor of the Experimental Clean Combustor Program was selected as the type of design for the Energy Efficient Engine combustor component. A cross-sectional view of the Vorbix configuration is shown in Figure 4.2.1-1. The combustor has two axially-separated burning zones, a pilot and a main zone, each independently fueled and each optimized for one extreme of the engine operating range. Separating both fuel and air introduction into two distinct zones allows the pilot zone to be designed for the high efficiency required for low idle emissions, while the main zone can be optimized for low oxides of nitrogen emissions.

The pilot zone is a conventional swirl-stabilized bulkhead-type combustor with fuel injected directly into the combustion zone. It is sized to provide a heat release rate consistent with high combustion efficiency, i.e., low carbon monoxide and total unburned hydrocarbon emissions, at the low pressures and temperatures associated with idle operation with all fuel injected into the pilot zone.

Vorbix (vortex burning and mixing) is a concept applied in the main zone to reduce the formation of oxides of nitrogen, while retaining high combustion efficiency. This concept employs basic combustion principles and takes advantage of the staging concept to enhance its application. To achieve low levels of oxides of nitrogen, relatively short residence times in a low temperature environment are required. However, the antithesis of these conditions is required for low levels of carbon monoxide and unburned hydrocarbons. Consequently, an extremely delicate balance of time and temperature in the burning and mixing region is necessary. The process is further complicated by fuel preparation. If liquid droplets persist after the burning and mixing region, excessive levels of carbon monoxide and hydrocarbons can be produced.

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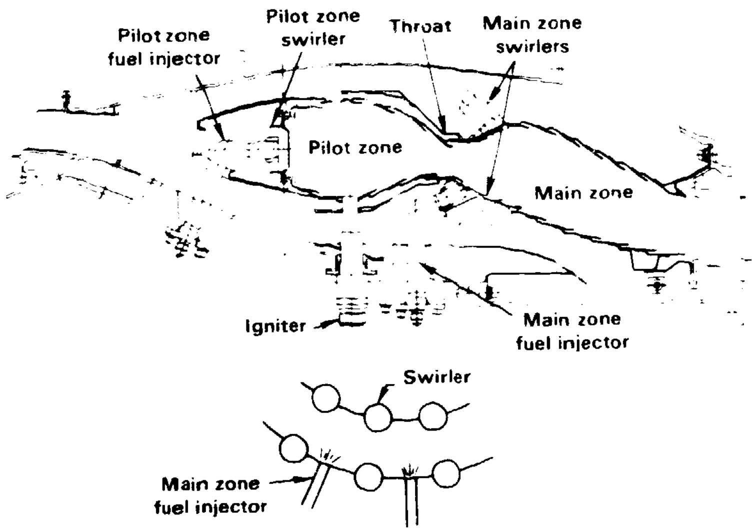

Figure 4.2.1-1 Experimental Clean Combustor Program (ECCP) Vorbix Combustor

The Vorbix configuration evaluated under the Experimental Clean Combustor Program was successful in satisfying the rigid constraints on the burning and mixing process. In this design, as shown in Figure 4.2.1-1, main zone fuel is introduced into the hot gases at the exit of the pilot zone to provide partial mixing and prevaporization before entering the main combustion zone. The rich fuel mixture enters the main zone with a degree of homogeneity exceeding that of a conventional combustor where the liquid fuel is injected directly into the combustion region. This method of fuel preparation is essential for complete combustion with rapid mixing induced by introducing main zone combustion air through swirlers on both the inner and outer liners. Rapid mixing is essential to reduce the time at the high temperatures produced by the reaction of the rich fuel/air mixture at the interface between the swirler air and the pilot exhaust gases. This accelerated mixing process is the key to lower levels of oxides of nitrogen.

In comparison to the Energy Efficient Engine Program's reference JT9D-7A engine combustor, the Vorbix configuration reduced oxides of nitrogen by 58 percent, carbon monoxide by 69 percent and unburned hydrocarbons by 96 percent (Ref. 1). Maximum smoke emissions, however, exceeded the goal of the Experimental Clean Combustor Program by approximately 50 percent. In terms of performance, combustion efficiency and exit temperature profile were comparable to the production engine combustor. However, certain other deficiencies were noted in the areas of durability, coking and altitude relight.

The aerodynamic definition of the Energy Efficient Engine combustor builds on the emissions reduction technology demonstrated with the Vorbix concept and addresses the areas of deficiency.

4.2.2 Design Criteria

Besides the Energy Efficient Engine Program goals for performance and exhaust emissions, the design criteria addressed aspects that would provide commercial attractiveness. These include durability, low weight, mechanical simplicity, maintainability, and low cost.

Of particular importance is mechanical simplicity. A two-zone combustor is inherently more complex than a conventional single zone-design because of the fuel staging requirement, method of fuel-air preparation and greater amount of liner surface area requiring cooling. This complexity is an impediment toward achieving low weight, low cost and better maintainability. In the aerodynamic definition, a major emphasis was placed on establishing a compact combustion system design with a fuel-air management approach more amenable to a commercial application.

Engine and cycle requirements dictated the design point conditions as well as the thermal and mechanical constraints imposed on the aerodynamic definition. Unlike the other engine components which are designed for a cruise condition, the combustor design point is the takeoff condition. This is the most stringent condition in terms of inlet pressure and exit temperature in the Energy Efficient Engine flight envelope. Table 4.2.2-I presents a comparison of the combustor operating parameters at takeoff and the cruise condition.

TABLE 4.2.2-I
COMBUSTOR PERFORMANCE PARAMETERS AT THE
TAKEOFF DESIGN POINT COMPARED TO THE CRUISE POINT

	Takeoff*	Cruise
Inlet Corrected Flow, kg/sec (lb/sec)	3.15 (6.96)	3.13 (6.91)
Inlet Pressure, MPa (psia)	3.1 (448)	1.4 (203)
Total Pressure Loss, %	5.58	5.50
Inlet Temperature, °C (°F)	567 (1053)	481 (898)
Exit Temperature, °C (°F)	1435 (2615)	1298 (2370)
Temperature Rise, °C (°F)	850 (1562)	800 (1472)
Combustion Efficiency, %	99.95	99.95

* Hot day, 29°C (84°F) ambient condition

4.2.3 Supporting Technology Programs

In the design process, the results from the related supporting technology programs were used to verify the aerodynamic concepts and definition of the diffuser/combustor system. The Diffuser/Combustor Model Test Program (Ref. 2) was conducted to document and optimize the aerodynamic performance of the prediffuser/combustor section. A full scale, full annular model rig was designed and tested to investigate the effects of various configurational changes on pressure loss and flow separation characteristics. The significant results from this effort are summarized in Section 6.0.

Refinement as well as substantiation of the combustor aerodynamic design was accomplished during the Sector Combustor Rig Test Program (Ref. 3). The significant results from this effort are also summarized in Section 6.0.

4.3 DIFFUSER/COMBUSTOR AERODYNAMIC DESIGN

The design of the Energy Efficient Engine, particularly the single stage high-pressure turbine, influenced the aerodynamic definition of the diffuser/combustor system. The large diameter of the turbine required that the combustor accommodate a relatively large offset between the high-pressure compressor exit and the turbine inlet. In addition, the straddle mounted, or simply-supported, high-pressure spool support system eliminated the requirement for a bearing support within the annular combustor and the thick diffuser case struts used to contain the bearing service lines in conventional designs. These factors, combined with the desire for a short burner and overall section length for liner durability, engine weight and performance considerations, led to the combustion system flowpath shown in Figure 4.3-1. This design is predicated on the use of a strutless, curved-wall diffuser, an integrated dump/burner front end region with structural struts, and a short axially-staged combustor using main zone carburetor tube air-fuel inlets located in the outer airflow annulus.

The aerodynamic design of the diffuser and combustor is described in the following sections.

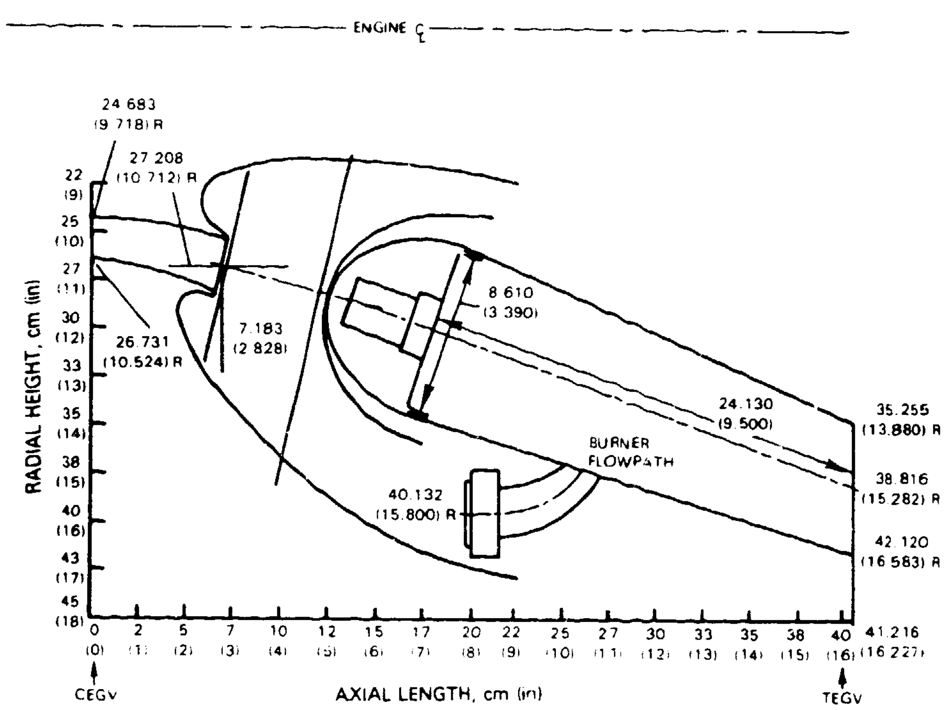

Figure 4.3-1 Energy Efficient Engine Combustor Flowpath Hot Dimensions

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4.3.1 Diffuser Section

The diffuser section consists of the prediffuser, dump section and combustor case inner and outer annuli. These subassemblies, along with other features of the diffuser section, are shown in Figure 4.3.1-1.

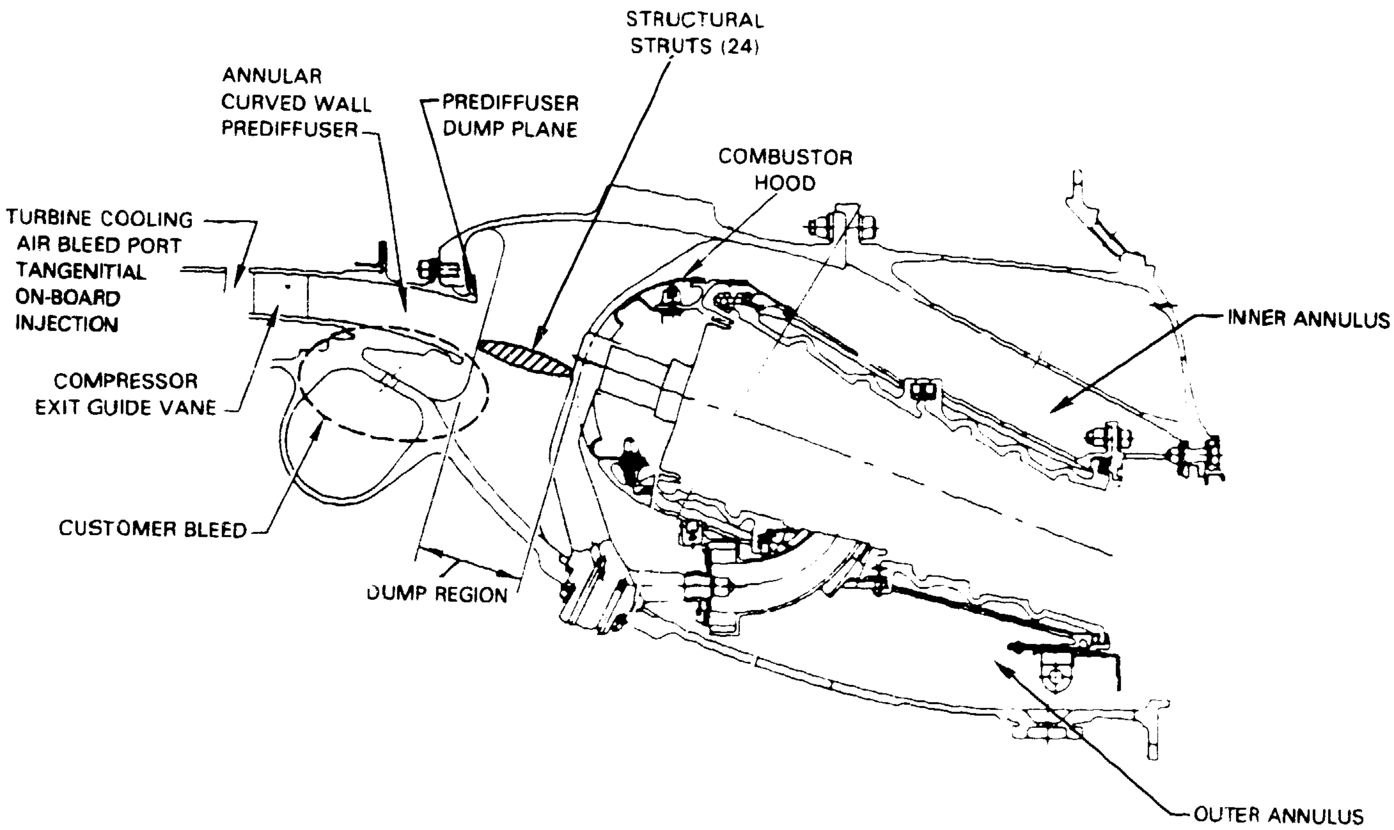

Figure 4.3.1-1 Energy Efficient Engine Combustor Diffuser Section

The annular prediffuser is a strutless, curved-wall design that slows and turns the high-pressure compressor exit air towards the outward canted combustor. Air exiting the prediffuser at the dump plane and not entering the combustor front end through openings in the hood around each pilot fuel injector is guided smoothly into the inner and outer annuli by the combustor hood. Twenty-four thin aerodynamically-shaped struts are located in the low velocity region between the prediffuser exit and the hood to provide the connecting structure between the inner and outer walls of the diffuser case. The outer annulus is designed to accommodate the blockage resulting from the main zone fuel injectors.

The diffuser design inlet Mach number is 0.28. The overall diffuser pressure loss is approximately 2.7 percent of the total high-pressure compressor exit pressure, which is less than the goal of 3.0. This low loss level has been confirmed by both model and sector rig tests.

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4.3.1.1 Prediffuser

The prediffuser is designed to maintain stable, separation-free airflow at all operating conditions, provide diffusion and turning in a relatively short length, and meet the overall pressure loss goal. The large radial offset between the high-pressure compressor exit and the high-pressure turbine inlet required the use of a curved-wall prediffuser. This type of flowpath is necessary to turn the airflow outward to better align it with the combustor centerline and reduce the pressure losses associated with airflow turning around the front end of the canted combustor. In contrast to most straight wall designs, the flowpath is free of structural struts to preclude the possibility of cross flows on the strut surfaces as a result of spanwise static pressure variations in the curved-wall prediffuser.

Prediffuser Flowpath

In the design evolution, the preliminary or baseline prediffuser configuration was ambitious from an aerodynamic standpoint. This design had a 1.5 area ratio, a short length to inlet height ratio (3.0), and a 19-degree turning angle relative to the engine centerline. After results of preliminary analyses indicated that this amount of turning might be too aggressive in the short length, outward canting of the compressor exit guide vane was evaluated as an alternative approach.

The benefit to the prediffuser for varying the amount of outward turning of the high-pressure compressor exit guide vanes from zero to eight degrees is indicated in Figure 4.3.1-2. As shown, the impact is most significant on the outer wall where separation is the primary concern. Results of this study indicated that five degrees of outward cant in the exit guide vanes would improve the wall static pressure coefficient and, thereby, be aerodynamically beneficial to both the prediffuser and the high-pressure compressor.

Another parametric study was undertaken to determine the effects of length, amount of turning and rate of axial diffusion on the prediffuser design. This study assumed a prediffuser area ratio of 1.5 and a compressor exit guide vane cant angle of five degrees. The results are presented in Figure 4.3.1-3 in terms of diffuser pressure loss versus prediffuser nondimensional length. Also shown are the design pressure loss goal and the regions of prediffuser stable and unstable operation. In addition, lines indicating the total amount of turning relative to the engine centerline are shown. These results show that the baseline design, with a nondimensional length of 3.0 and 19 degrees of airflow turning, operated in the unstable region. Stability was achieved by increasing the nondimensional length to 3.5 and reducing the amount of turning to 14 degrees. As indicated in Figure 4.3.1-3, this recommended design was estimated to have a pressure loss below the goal.

Prior to selecting the final configuration, the recommended design as well as two more aggressive variations (one having an increased area ratio and less turning and the other having a shorter length) were evaluated in the Diffuser/Combustor Model Test Program. These tests verified the superiority of the recommended prediffuser design, by demonstrating low pressure loss, good pressure recovery and separation-free operation. In addition, it had insensitivity to inlet profile, air extraction, and dump spacing variations and annuli flow split variations.

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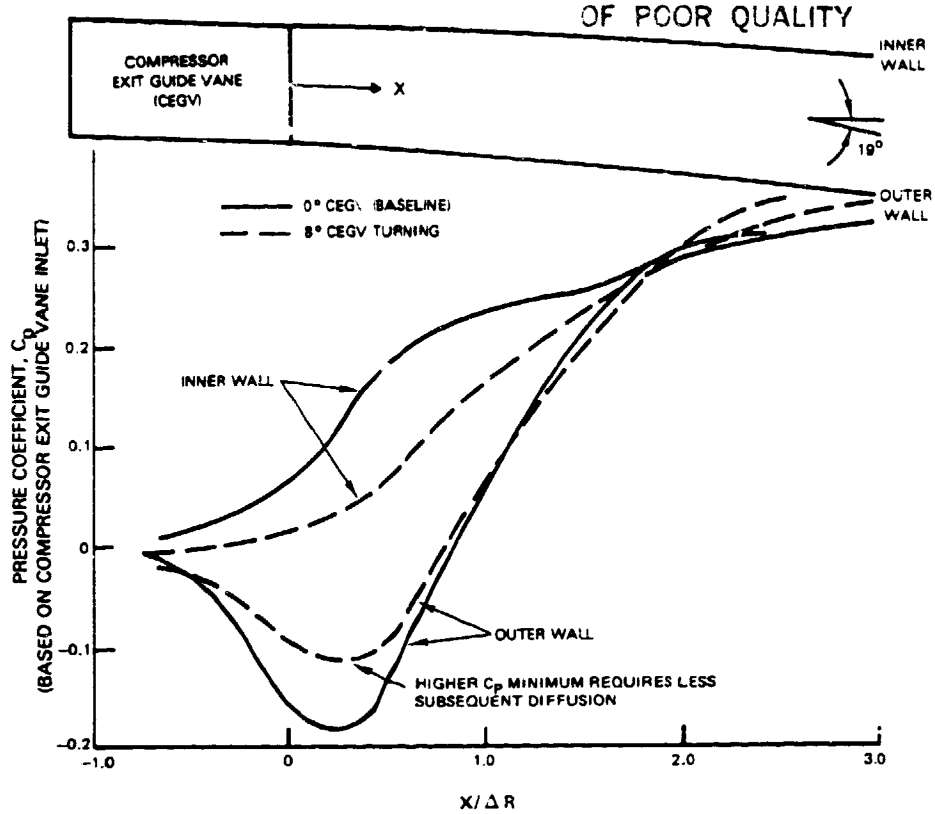

Figure 4.3.1-2 Effect of High-Pressure Compressor Exit Guide Vane Turning On Combustor Prediffuser Performance

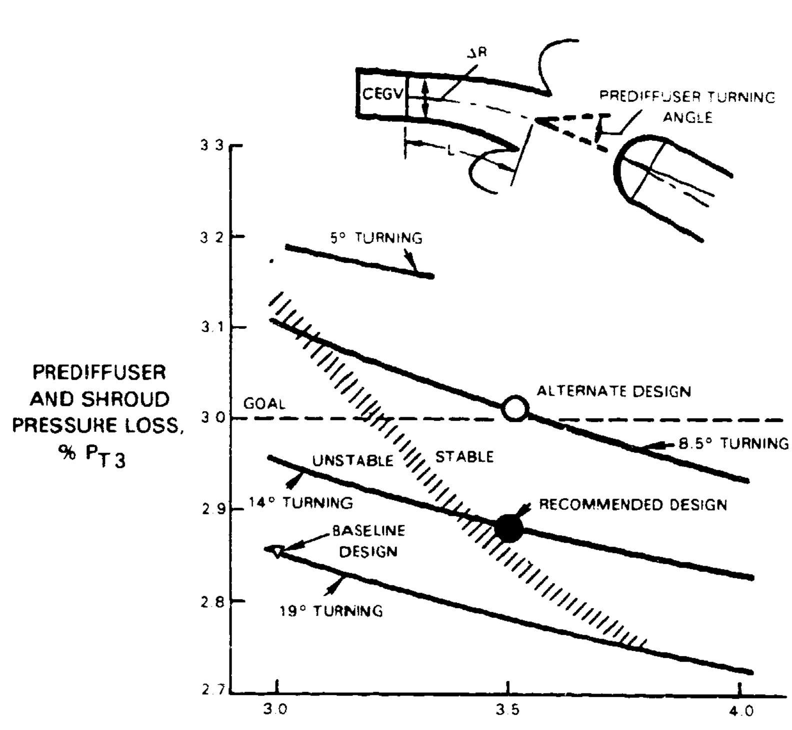

Figure 4.3.1-3 Parametric Study Results in Which Prediffuser Length, Turning, and Axial Turning Were Varied

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A summary of the prediffuser final aerodynamic design is presented in Table 4.3.1-I.

TABLE 4.3.1-I
SUMMARY OF FINAL PREDIFFUSER AERODYNAMIC GEOMETRY

Area Ratio	1.5
Length, cm (in)	7.1 (2.8)
Length/Inlet Height	3.5
Airflow Turning Angle (with CEGV), deg	14
Inlet Mach Number	0.28
Exit Mach Number	0.18
Total Pressure Loss, %	0.8

Prediffuser Bleed Requirements

The prediffuser aerodynamic definition considered air extraction locations for customer bleed air and high-pressure turbine tangential on-board injection (TOBI) cooling air. As shown in Figure 4.3.1-1, air for customer usage is extracted downstream of the high-pressure compressor exit, which is typical of modern aircraft gas-turbine engines. In contrast, the extraction of tangential on-board injection air forward of the high-pressure compressor exit guide vane differs from current engine design practices. A low-pressure customer bleed flow of 9.3 percent of the total high-pressure compressor exit flow was defined as representative of service requirements. The tangential on-board injection bleed extracts 3.5 percent of the compressor exit flow for turbine cooling.

Tangential on-board injection air extracted from the diffuser case is normally of sufficient pressure to ensure proper operation. Analytical studies, however, predicted that the Energy Efficient Engine would experience significant windage losses at the high-pressure compressor rear hub with this approach because of the relatively large hub surface area and the high operating speed. This would expose the hub to higher temperature levels, thus requiring extra cooling to maintain the desired temperature.

Additional analysis indicated that extracting the air ahead of the exit guide vane would be a more efficient approach. At this location, the air would contain sufficient swirl to greatly reduce the windage losses and result in lower operating temperatures for the rear hub. As a secondary benefit, the efficiency of the tangential on-board injection system itself would be improved because of the lower temperature of the supply air. In turn, the amount of air required for cooling could be reduced. This method provided a sufficient pressure level to ensure choked operation of the tangential on-board injection system.

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High-Pressure Compressor Exit Guide Vane

The aerodynamic design of the high-compressor exit guide vane was accomplished under the compressor design effort. As stated previously, the exit guide vane is designed with five degrees of outward cant. The exit guide vane, as integrated in the prediffuser flowpath, is shown in Figure 4.3.1-4.

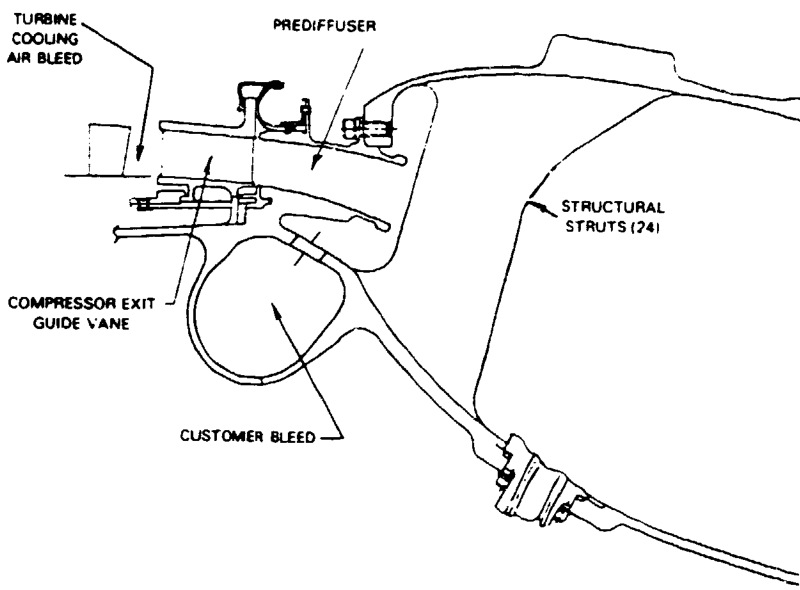


Figure 4.3.1-4 Combustor Prediffuser Section with Compressor Exit Guide Vane

A single row controlled diffusion design was selected with the intent to capitalize on the potential for short length, fewer airfoils and the lower weight inherent with a single row as opposed to a double row design. However, enough axial space was provided in the event that a double row approach would be necessary.

The vane aspect ratio is 0.52 and the pitch to chord ratio is 0.4 to limit the diffusion factor to an acceptable level. The resultant airfoil profile and spacing is illustrated in Figure 4.3.1-5. A summary of airfoil design aerodynamics is contained in Table 4.3.1-II.

4.3.1.2 Diffuser Dump Region

The initial dump region, which is at the prediffuser exit, is defined by the diffuser case height and contour. This region is sized to provide positive separation, or dump, of the prediffuser exit flow without immediate flow attachment to the case walls. The combustor hood is axially and radially located downstream of the prediffuser exit to divide the flow and guide it smoothly into the inner and outer annuli and the combustor front end with minimum pressure loss. The design flow splits, shown in Figure 4.3.1-6, were determined from the Diffuser/Combustor Model Test Program.

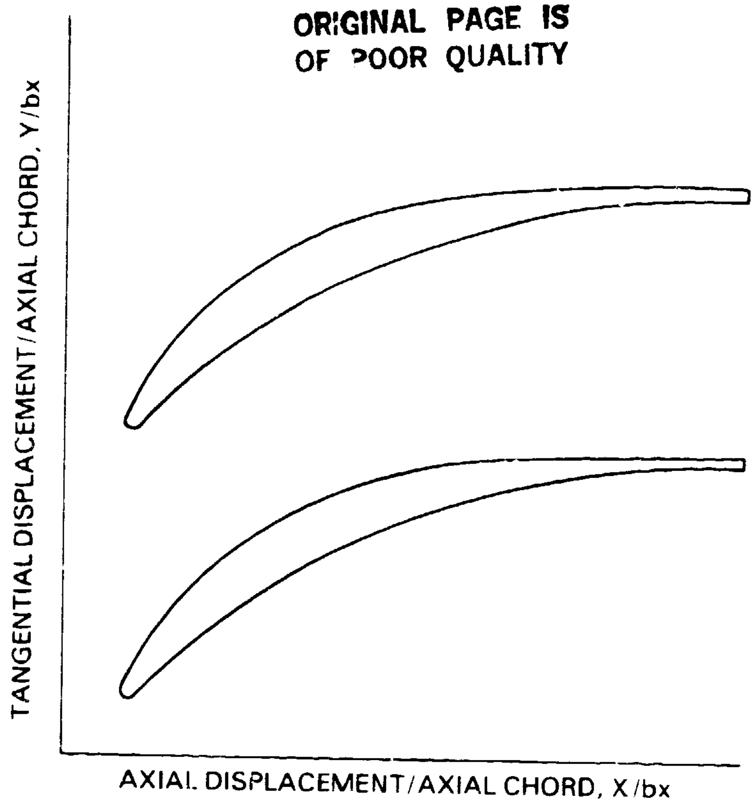


Figure 4.3.1-5 High Pressure Compressor Exit Guide Vane Airfoil Geometry

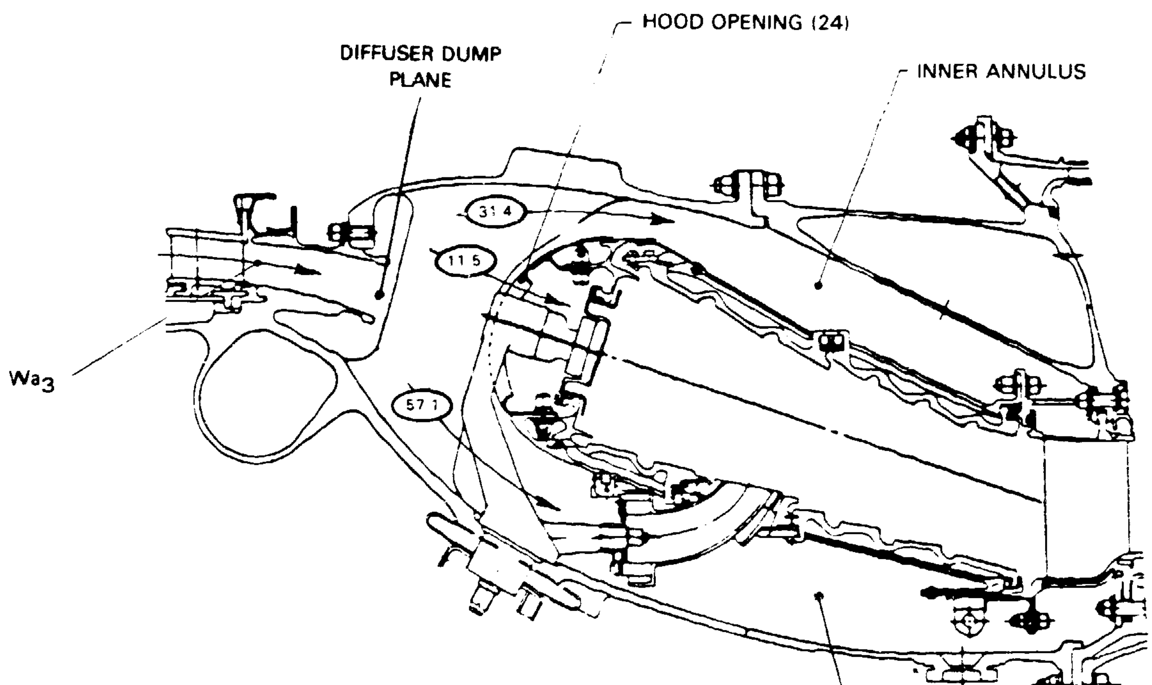


Figure 4.3.1-6 Diffuser Design Flow Splits

TABLE 4.3.1-II
CEGV AIRFOIL AERODYNAMIC SUMMARY
(Single Row Airfoil)

SL	V-1 M/SEC	V-2 M/SEC	VM-1 M/SEC	VM-2 M/SEC	VO-1 M/SEC	VO-2 M/SEC	RHOVM-1 KG/M2 SEC	RHOVM-2 KG/M2 SEC	PO/PO INLET	TO/TO INLET	%EFF-A TOT-INLET	%EFF-P TOT-INLET	EPSI-1 RADIAN	EPSI-2 RADIAN
1	260.9	145.7	146.8	145.5	215.7	8.7	837.13	885.70	14.0056	2.2796	84.05	88.58	0.0065	0.0982
2	263.5	154.8	158.4	154.8	210.5	1.2	922.59	960.69	14.1048	2.2400	87.19	90.85	0.0111	0.0972
3	260.5	154.8	157.3	154.8	207.7	0.5	929.25	974.50	14.1156	2.2097	89.50	92.51	0.0146	0.0961
4	257.6	154.9	155.3	154.9	205.5	0.4	926.30	983.94	14.1200	2.1895	91.10	93.66	0.0191	0.0929
5	258.5	156.5	154.1	156.5	207.6	0.4	906.91	981.28	14.1198	2.2177	88.90	92.08	0.0209	0.0876
6	259.3	157.4	152.0	157.4	210.1	0.6	881.42	972.05	14.1122	2.2500	86.47	90.34	0.0209	0.0840
7	253.3	148.1	133.2	147.9	215.5	8.3	756.71	894.97	14.0026	2.2908	83.27	88.02	0.0203	0.0792

SL	V-1 FT/SEC	V-2 FT/SEC	VM-1 FT/SEC	VM-2 FT/SEC	VO-1 FT/SEC	VO-2 FT/SEC	RHOVM-1 LBM/FT2SEC	RHOVM-2 LBM/FT2SEC	PCT TE SPAN	TO/TO INLET	%EFF-A TOT-INLET	%EFF-P TOT-INLET	EPSI-1 DEGREE	EPSI-2 DEGREE
1	856.1	478.2	481.8	477.3	707.7	28.4	171.45	181.40	0.1000	2.2796	84.05	88.58	0.370	5.627
2	864.4	507.8	519.9	507.7	690.6	4.1	188.95	196.76	0.2000	2.2400	87.19	90.85	0.637	5.568
3	854.8	508.0	516.0	508.0	681.5	1.6	190.32	199.59	0.3000	2.2097	89.50	92.51	0.837	5.505
4	845.1	508.2	509.5	508.2	674.3	1.2	189.72	201.52	0.5000	2.1895	91.10	93.66	1.093	5.320
5	848.2	513.6	505.5	513.6	681.2	1.2	185.74	200.97	0.7000	2.2177	88.90	92.08	1.196	5.022
6	850.7	516.4	498.6	516.4	689.2	1.9	180.52	199.08	0.8000	2.2500	86.47	90.34	1.200	4.815
7	831.1	486.0	436.9	485.3	707.0	27.2	154.98	183.30	0.9000	2.2908	83.27	88.02	1.162	4.537

SL	B-1 DEGREE	B-2 DEGREE	M-1	M-2	INCS DEGREE	INCM DEGREE	DEV DEGREE	TURN DEGREE	D-FAC	OMEGA-B TOTAL	LOSS-P TOTAL	PO2/PO1	PO/PO STAGE	TO/TO STAGE	%EFF-A TOT-STG	%EFF-P TOT-STG
1	55.7	3.4	0.4846	0.2670	0.0	0.0	0.0	52.25	0.5930	0.0943	0.0182	0.9863	1.1737	1.0503	85.74	86.03
2	52.9	0.5	0.4938	0.2861	0.0	0.0	0.0	52.48	0.5659	0.0935	0.0182	0.9860	1.1703	1.0498	85.42	85.72
3	52.8	0.2	0.4914	0.2881	0.0	0.0	0.0	52.60	0.5607	0.0881	0.0173	0.9869	1.1714	1.0497	86.21	86.49
4	52.9	0.1	0.4876	0.2896	0.0	0.0	0.0	52.72	0.5560	0.0846	0.0169	0.9876	1.1738	1.0500	87.01	87.28
5	53.4	0.1	0.4859	0.2909	0.0	0.0	0.0	53.23	0.5550	0.0883	0.0179	0.9871	1.1767	1.0507	86.87	87.16
6	54.1	0.2	0.4833	0.2905	0.0	0.0	0.0	53.86	0.5571	0.0920	0.0190	0.9865	1.1783	1.0513	86.51	86.78
7	58.2	3.2	0.4680	0.2708	0.0	0.0	0.0	55.04	0.5837	0.0946	0.0195	0.9870	1.1817	1.0518	86.88	87.15

NCORR INLET RPM	WCORR INLET LBM/SEC	WCORR INLET KG/SEC	TO/TO INLET	PO/PO INLET	EFF-AD INLET %	EFF-P INLET %	TO/TO STAGE	PO2/PO1	PO/PO STAGE	EFF-AD STAGE %	EFF-P STAGE %
12136.50	77.52	35.16	2.2387	14.0678	87.16	90.82	1.0507	0.9870	1.1761	91.22	90.60

The design of the dump region is based on data obtained from Pratt & Whitney Aircraft Independent Research and Development programs and combustor design experience. The dump gap, defined as the length between the prediffuser exit and combustor hood, is a compromise considering aerodynamic pressure loss and structural strut requirements. The gap length is 7.24 cm (2.85 in). Although this is somewhat greater than the aerodynamic optimum, it does not incur a significant pressure loss penalty.

Elimination of the bearing service lines through the diffuser allowed the use of aerodynamically-shaped struts in the dump region to provide the structural connection between the diffuser inner and outer case walls. The dump region is designed with a total of 24 struts. This number of struts enables the use of a thinner and shorter geometry to achieve the necessary structural integrity. In comparison to the thick struts used in current engine designs, a thin design reduces pressure loss and low pressure wake regions downstream in the inner and outer annuli. In turn, this reduces associated liner hot streaks, improves liner durability and enhances the attainment of a low pattern factor.

On the basis of structural analyses, modifications were required to the initial strut geometry. The modifications consisted of thickening the trailing edge from 0.317 to 0.762 cm (0.125 to 0.300 in), along with extending the inner chord length, as shown in Figure 4.3.1-7.

Aerodynamically, the revised configuration only slightly increases the total pressure loss. Figure 4.3.1-8 shows the results of performance testing during the Diffuser/Combustor Model Test Program. Although the diffuser pressure loss increased 0.2 percent with the revised strut, the overall loss level is below the goal. Also shown in Figure 4.3.1-8 are the inner shroud wake rake traverse results. These data confirm the absence of wakes in the inner shroud flow with both the original and modified strut designs.

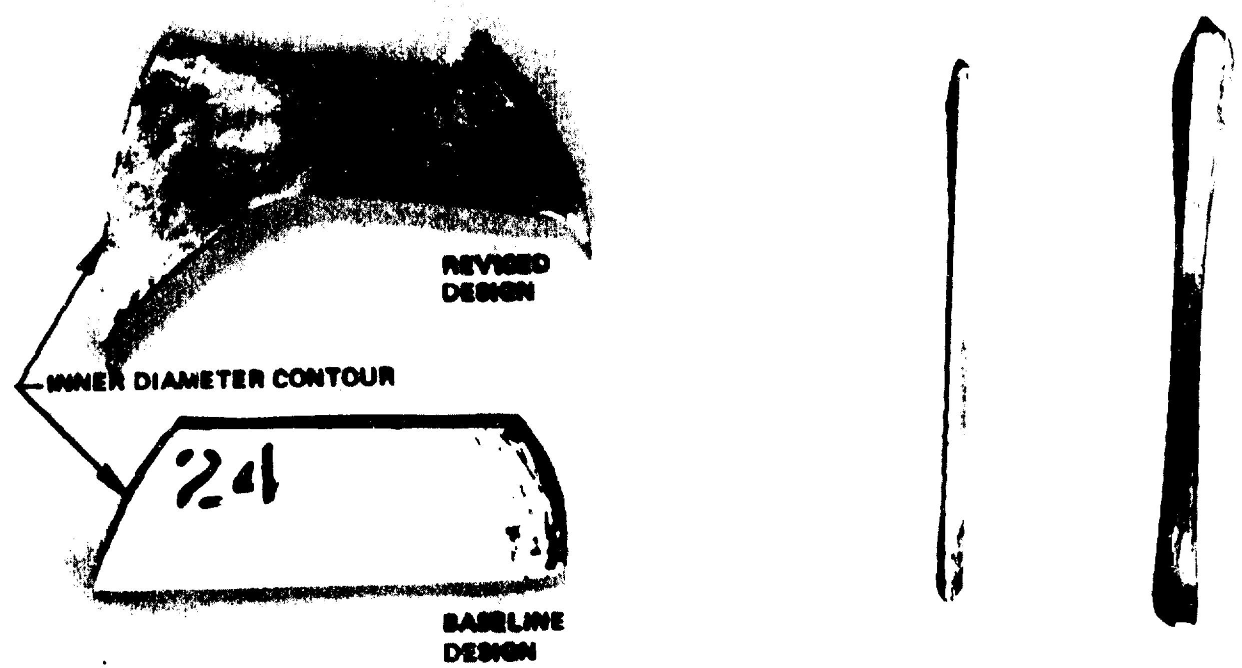
4.3.1.3 Inner and Outer Annuli

The inner and outer annuli are designed to minimize diffuser pressure loss and provide a positive static pressure difference across the liner at all locations to prevent aspiration (local reverse flow). Both annuli are sized to ensure a normal air velocity of less than or equal to 45 m/sec (150 ft/sec) over the full length of the annuli at the design airflow. The outer annulus is sized based on a net airflow area that accounted for blockage by the fuel injector support assemblies and the carburetor tubes.

4.3.2 Combustor Section

The combustor is a two-zone design for effective emissions control throughout the entire operating range. The pilot zone is designed to minimize idle emissions and provide adequate stability and relight characteristics. In the main zone, lean combustion occurs to minimize the formation of oxides of nitrogen and smoke.

As shown previously by the flowpath definition in Figure 4.3-1, the axial length of the combustor flowpath is 24.1 cm (9.5 in). The internal flowpath height at the dome of the pilot zone is 8.6 cm (3.4 in) and tapers to a height of 6.8 cm (2.7 in) at the exit plane. Relative to the engine centerline, the combustor cant angle is 19 degrees, and the maximum radial height, as measured at the exit, is 42.1 cm (16.6 in).



EXTENDED CHORD ID

BASELINE DESIGN

REVISED DESIGN

THICKER TRAILING EDGE
0.762 cm (0.300 in) vs. 0.317 cm (0.125 in)

Figure 4.3.1-7 Comparison of Baseline and Revised Strut Configurations

COMBUSTOR SECTION
TOTAL PRESSURE LOSS, % P_{T_3}

	DEMONSTRATED BASELINE STRUT	REVISED STRUT	PROGRAM GOALS
INNER ANNULUS	2.2	2.4	≤ 3.0
OUTER ANNULUS	2.7	2.95	≤ 3.0
SECTION	5.2	5.5*	5.2(cold)

*REVISED COMBUSTOR HOLE PATTERN ENABLES GOAL SECTION LOSS TO BE MET

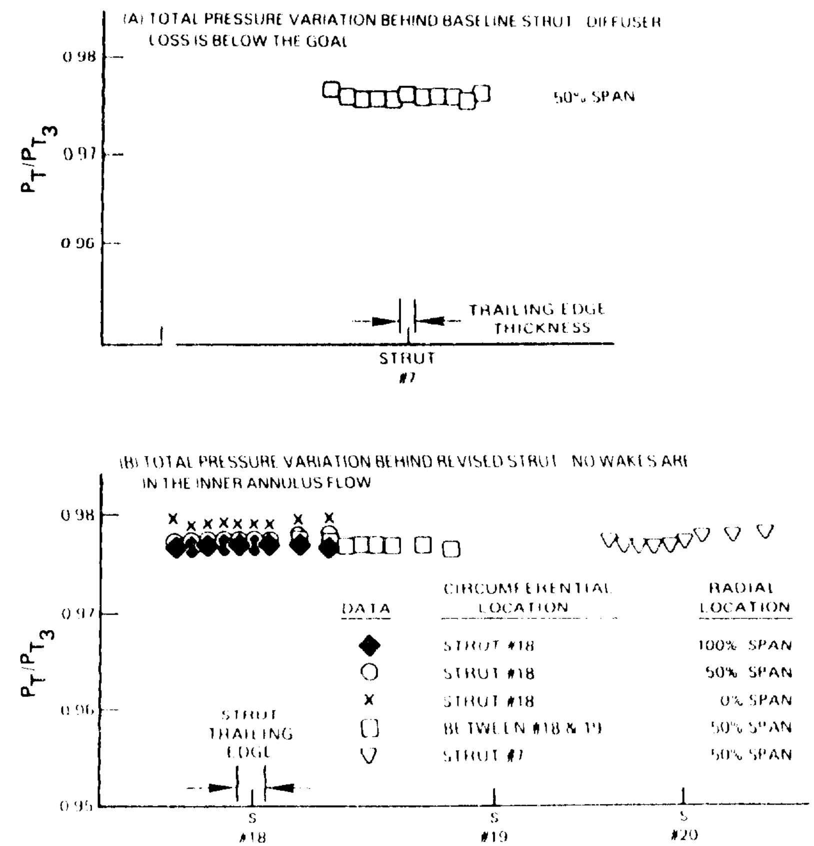


Figure 4.3.1-8 Inner Annulus Test Results Indicate Modified Strut Design Acceptable

4.3.2.1 Pilot Zone

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The aerodynamic definition of the pilot zone is based on the design and test experience acquired during the Experimental Clean Combustor Program (ECCP). In the Energy Efficient Engine combustor, the pilot zone is a direct fuel injection swirl-stabilized design with fuel introduced through single pipe aerated nozzles. Compared to a scaled Vorbix combustor, the pilot zone is designed with an increased volume and dome height. This results in a lower heat release rate, lower velocity and higher residence time, which are conducive to reduced low-power emissions and good relight and stability characteristics. Table 4.3.2-I presents pilot zone key parameters for the Energy Efficient Engine combustor and the configuration for the Experimental Clean Combustor Program.

TABLE 4.3.2-I
COMPARISON OF PILOT ZONE PARAMETERS
(Energy Efficient Engine and ECCP Vorbix Design)

	ECCP Vorbix	Energy Efficient Engine Combustor
Dome Height, cm (in)	8.8 (3.5)	8.6 (3.4)
Length, cm (in)	15.2 (6.0)	10.9 (4.3)
Pilot Zone Heat Release Rate,		
Joules/m^3 hr atm	124.8×10^7	67.0×10^7
(Btu/ft^3 hr atm)	(12.3×10^6)	(6.6×10^6)
Characteristic Velocity, m/sec (ft/sec)	12 (40)	7 (24)
Residence Time, msec	12	16
Fuel Injectors		
Number	30	24
Type	Pressure Atomized	Aerated

Figure 4.3.2-1 shows a cross sectional view of the single pipe aerated injector design. This type of injectors relies on shearing of low velocity fuel by the surrounding high velocity airstreams to achieve good fuel atomization. The internal and external airstreams are corotationally swirled to distribute the fuel and provide the recirculatory flow required for combustion stability.

The combustor hood is designed to provide sufficient volume to achieve a uniform, high static pressure air feed of at least 99 percent of the diffuser inlet total pressure to the fuel injectors and combustor bulkhead. This is necessary to meet emissions goals as well as desirable for exit temperature pattern development and liner durability. Hood openings, which are located around each pilot injector, capture the air for the combustor front end. These openings were initially sized to pass 11 percent of the inlet flow with minimum spillage and impact on inner and outer annuli airflow over the hood. However, to prevent an interference between the pilot fuel injector and combustor hood, the capture area was increased to pass approximately 15 percent of the inlet flow.

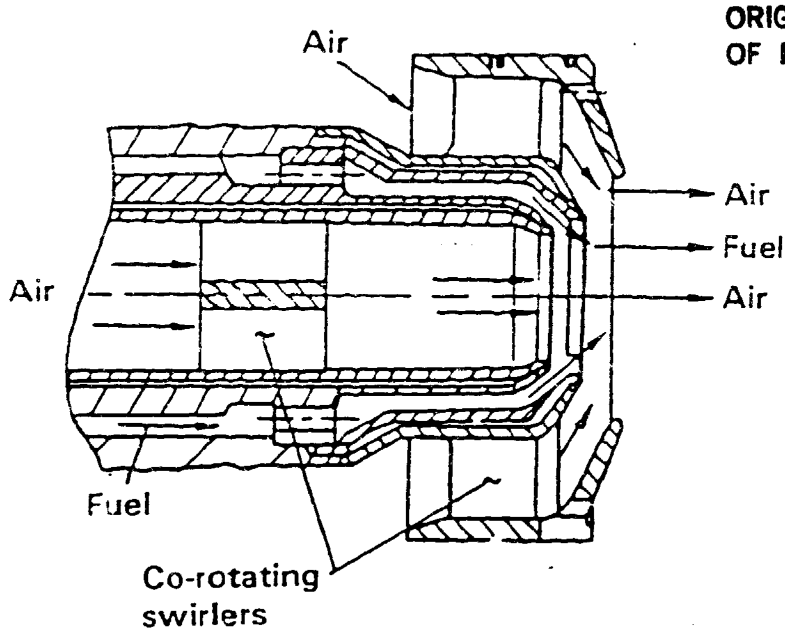


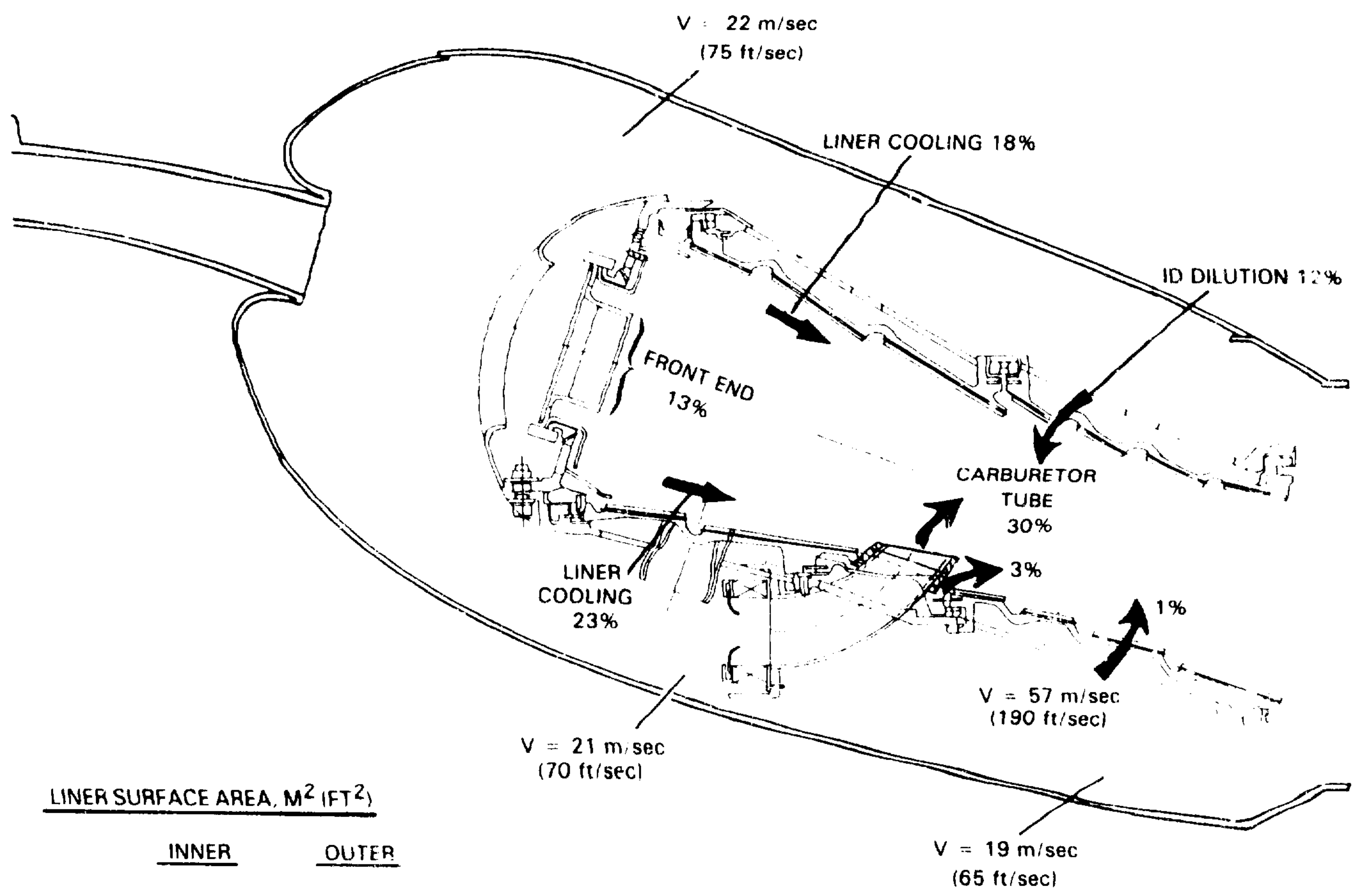
Figure 4.3.2-1 Single Pipe Aerated Pilot Fuel Injector

Optimization of the pilot zone airflow schedule was successfully accomplished during the related Sector Combustor Rig Test Program. The relative percentages of the final airflows are indicated in Figure 4.3.2-2. Testing demonstrated that carbon monoxide and unburned hydrocarbons are minimized at idle by maintaining an equivalence ratio of approximately 1.2. At high power conditions, the equivalence ratio is reduced to approximately 0.35 to minimize the formation of oxides of nitrogen. The minimum equivalence ratio for the pilot zone is defined by overall lean blowout considerations, combustion efficiency and the requirement to maintain a sufficient pilot zone temperature to vaporize and ignite the main zone fuel.

4.3.2.2 Main Zone

The main zone design reflects a substantial refinement over its precursor — the Vorbix. This zone is designed with a short length of 13.2 cm (5.2 in) to reduce the liner surface area and resulting cooling requirements. Residence time, however, was maintained near the Vorbix design level to ensure complete combustion. Table 4.3.2-II presents main zone key parameters for the Energy Efficient Engine combustor and configuration for the Experimental Clean Combustor Program.

Besides the apparent geometry differences, a major difference between the two designs is the method by which fuel is injected into the main zone. In the Vorbix combustor, fuel is injected directly into the combustion zone through pressure atomizing nozzles where it mixes with swirling air jets introduced through the inner and outer liners (see Figure 4.2.1-1). This method of fuel-air mixing has been significantly refined by the compact carburetor tube arrangement used in the Energy Efficient Engine combustor component.



LINER SURFACE AREA, M^2 (FT^2)

	INNER	OUTER
PILOT ZONE	0.22 (2.4)	0.27 (2.9)
MAIN ZONE	0.31 (3.4)	0.37 (4.0)

- ALL FLOWS ARE IN PERCENTAGE OF COMBUSTOR AIRFLOW (W_{A3})
- VELOCITIES BASED ON SECTOR RIG SIMULATED SEA LEVEL TAKEOFF CONDITIONS

Figure 4.3.2-2 Combustor Airflow Distribution as Determined from Sector Rig Testing

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TABLE 4.3.2-II
COMPARISON OF MAIN ZONE PARAMETERS
(Energy Efficient Engine and ECCP Vorbix Design)

	ECCP Vorbix	Energy Efficient Engine Combustor
Height, cm (in)	8.1 (3.2)	6.6 (2.6)
Length, cm (in)	23.6 (9.3)	13.2 (5.2)
Velocity, m/sec (ft/sec)	51 (170)	26 (88)
Residence Time, m sec	5	4.4
Fuel Injectors		
Number	60	48
Type	Pressure Atomized	Carburetor Tube

A cross-sectional view of the carburetor tube injection system is shown in Figure 4.3.2-3. This design, which is based on the results of the Sector Combustor Rig Test Program, premixes the fuel and air prior to injection into the combustor for better atomization. Fuel is supplied to each of the 48 carburetor tubes by a simplex pressure atomizer. A fraction of the fuel is vaporized in the tube and the remaining fuel is centrifuged to the walls of the tube by air introduced through a radial inflow swirler. The fuel film formed at the exit plane of the tube is sheared into droplets by the swirling core and secondary airflow jets.

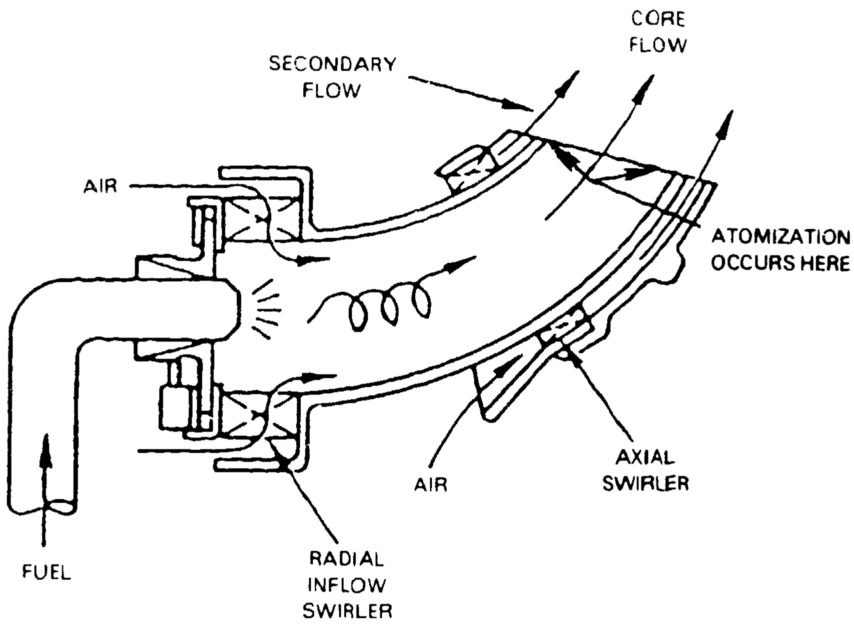

Figure 4.3.2-3 Carburetor Tube Main Zone Injector

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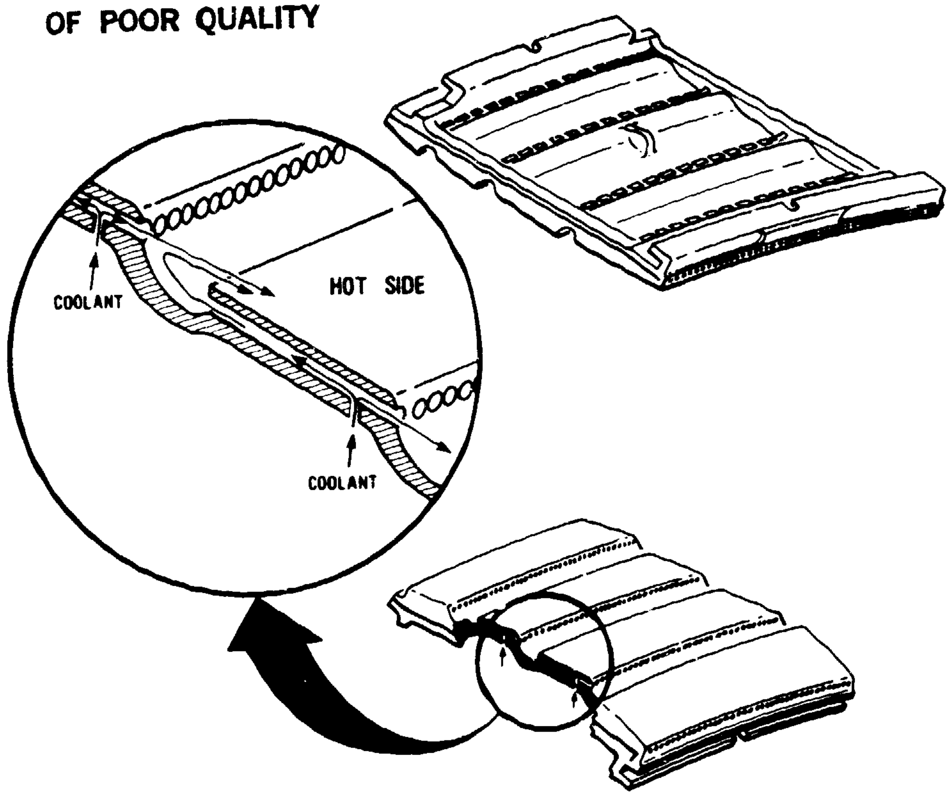

Figure 4.3.2-4 Counter-Parallel FINWALL® Cooling Technique

The core radial inflow swirler consists of ten curved vanes with an overall width of approximately 1.75 cm (0.7 in). The secondary airstream swirler contains nine 20-degree axial vanes.

The relative percentages of main zone airflow are shown in Figure 4.3.2-2. Most of the combustion air is supplied through the carburetor tube core and secondary passages. The remaining airflow enters the combustor through dilution holes in the liners and cooling air slots on the cold side of each panel. A main zone equivalence ratio of approximately 0.7 results at sea level takeoff conditions if 50 percent of the inner liner wall dilution air, along with the total carburetor tube flow, is included in the combustion process.

4.3.2.3 Segmented Liner

The requirement for a segmented liner design was dictated by the life goal of 8000 hours, the high combustor operating pressure at sea level static takeoff conditions and the liner cooling airflow limit of 35 percent of the total combustor airflow. This liner cooling limit was established to ensure adequate air supply for dilution and emissions control. The liner is designed for a pressure loss of 2.5 percent of the total high-pressure compressor exit pressure.

The inner and outer liners, which define the combustor gas path, are formed by arranging the segments circumferentially in both the pilot and main zones. A total of 120 segments is contained in the full annular combustor. The inner combustor liner is comprised of 48 segments while the outer liner is comprised of 72 segments. A typical segment is illustrated in Figure 4.3.2-4. Segments range in size from the smallest with a length of 6.3 cm (2.5 in) and width of 9.9 cm (3.9 in) to the largest with a length of 12.1 cm (4.8 in) and width of 10.9 cm (4.3 in). Axial feather seals on the sides of the segments are used to control leakage between adjacent segments.

The total liner surface area requiring cooling is 1.2 m^2 (12.7 ft^2), 0.5 m^2 (5.3 ft^2) in the pilot zone and 0.7 m^2 (7.4 ft^2) the main zone. The method selected for cooling, based on thermal analyses of several concepts as described in Section 5.3.3.2, is the counter-parallel FINWALL® technique. This is an advanced convective/film cooling approach that permits more effective heat transfer with less cooling air.

The flowpath for this technique is shown in Figure 4.3.2-4 and consists of a series of axial cooling holes 0.088 cm (0.035 in) in diameter located 0.165 cm (0.065 in) on center across the panel. Cooling air enters the liner through 0.444 by 0.152-cm (0.175 by 0.060-in) slots on the liner cold wall. At sea level takeoff, the coolant temperature is 565°C (1050°F) and the pressure is 3.1 MPa (448 psia). The air flows upstream and downstream in the discrete cooling passages, and the exiting air flows along the hot liner wall to provide a cooling film. The use of high temperature capability material (B-1900 + Hf) allows operation at a maximum metal temperature of 1037°C (1900°F).

SECTION 5.0
COMBUSTOR THERMAL-MECHANICAL DESIGN

5.1 THERMAL-MECHANICAL DESIGN GOALS AND PHILOSOPHY

The objective of the mechanical design effort was to provide a commercially acceptable combustor configuration that would meet the performance and emissions goals described in Section 4.1 within the constraints of the aerodynamic definition. The specific goal for component life/durability is 8000 hours or 4900 cycles. This level is consistent with commercial service requirements.

The technical approach employed to achieve this objective was based on three advanced technology concepts: a curved-wall, strutless prediffuser; a two-zone combustion system; and segmented liners. Incorporation of these concepts into the component design resulted in a compact diffuser/combustor arrangement compared to the Vorbix and reference JT9D-7A combustors.

5.2 DESIGN APPROACH

The design effort was an evolutionary process, in which the preliminary design incorporated reference data available from the NASA Experimental Clean Combustor Program and Pratt & Whitney Aircraft Independent Research and Development programs. The mechanical design was also based on results from the aerodynamic design effort.

During the preliminary design, primary attention was directed towards: (1) defining the front end structural characteristics, particularly the diffuser case and structural struts; (2) assessing liner durability; and (3) determining a fuel injector clustering arrangement that would permit casting the diffuser case while minimizing wall penetrations to improve case structural integrity. As the design progressed, results from the supporting technology programs, particularly the Sector Combustor Rig Test Program, were used for optimization of the base configuration.

The most notable change in the design approach involved the combustor liners. Initially, it had been hoped that life goals could be achieved with a conventional film-cooled louvered construction using Hastelloy X material for the outer shell and an oxide dispersion strengthened (ODS) alloy for the gas side section. However, preliminary analysis indicated that the goals were unattainable with this configuration because of high pressure/temperature operating levels and the limited cooling airflow available for the liner. Consequently, thermal and structural design studies were conducted to screen various convective cooling techniques offering a higher thermal effectiveness (see Section 5.3.3.2) than film cooling.

5.3 COMBUSTOR THERMAL-MECHANICAL DESIGN

The combustor component assembly is comprised of four major subassemblies. These include: (1) the high-pressure compressor exit guide vanes; (2) the diffuser case, including the prediffuser, bleed manifold, and struts; (3) the combustor, including hood, bulkhead, segmented liners, liner support structures, and carburetor tubes; and (4) the fuel management system, including fuel ignitors, fuel injectors, fuel manifold, and a circumferential fuel system shroud. Information pertaining to the mechanical and structural designs of these subassemblies is presented in the following sections.

5.3.1 High-Pressure Compressor Exit Guide Vanes

5.3.1.1 Mechanical Design Features

The design of the high-pressure compressor exit guide vane assembly was included in the design of the combustor component because of its interaction with the prediffuser duct, as shown in Figure 5.3.1-1. The exit guide vanes are mounted on an inner and outer platform with a turbine-style vane foot attachment. The vane assembly is comprised of 100 detail vanes machined from Inconel 718 bar stock and welded together to produce 20 five-vane clusters supported off a shelf on the diffuser case. The five-vane cluster approach was chosen as a judicious compromise to satisfy the requirements for damping, high-pressure compressor stage interaction, assembly, leakage, and fabrication considerations. For the integrated core/low spool, vanes are machine fabricated because it is a low cost approach that meets the lead time requirements. The exit guide vane assemblies for the flight propulsion system will be cast.

The vane inner platform has a simple sheet metal seal that attaches to the prediffuser wall using a vane segment retainer ring to prevent leakage and recirculation of air at the prediffuser location. This seal is required at this location because of the large axial excursion of the diffuser case and the pressure differential (344,740 Pa maximum (50 psi)) between the axial flowpath and the high-pressure compressor rotor rear bearing compartment which could result in recirculation of airflow around the root platform. The vane segment retainer ring is fabricated from Inconel 718 material and fastened to the primary ring structure with a hollow rivet. A countersunk screw and nut combination could be also used at this location if the rivet loosens up. Additional sealing is provided by feather seals at both the inner and outer vane segment walls to prevent leakage at the segment gaps.

5.3.1.2 Structural Analysis

Structural analysis of the exit guide vane assembly package showed that all stress levels were within limits, with a maximum stress occurring at the outer attachment nearest the prediffuser duct as shown in Figure 5.3.1-2. Large airfoil chords combined with simple outer hook attachments serve to minimize gas bending stresses in the airfoils; the maximum being 38.6 MPa (5.6 ksi) in the outer leading edge. By attaching the segments with hook retainers, appreciable support damping is provided to minimize the risk of fatigue failure.

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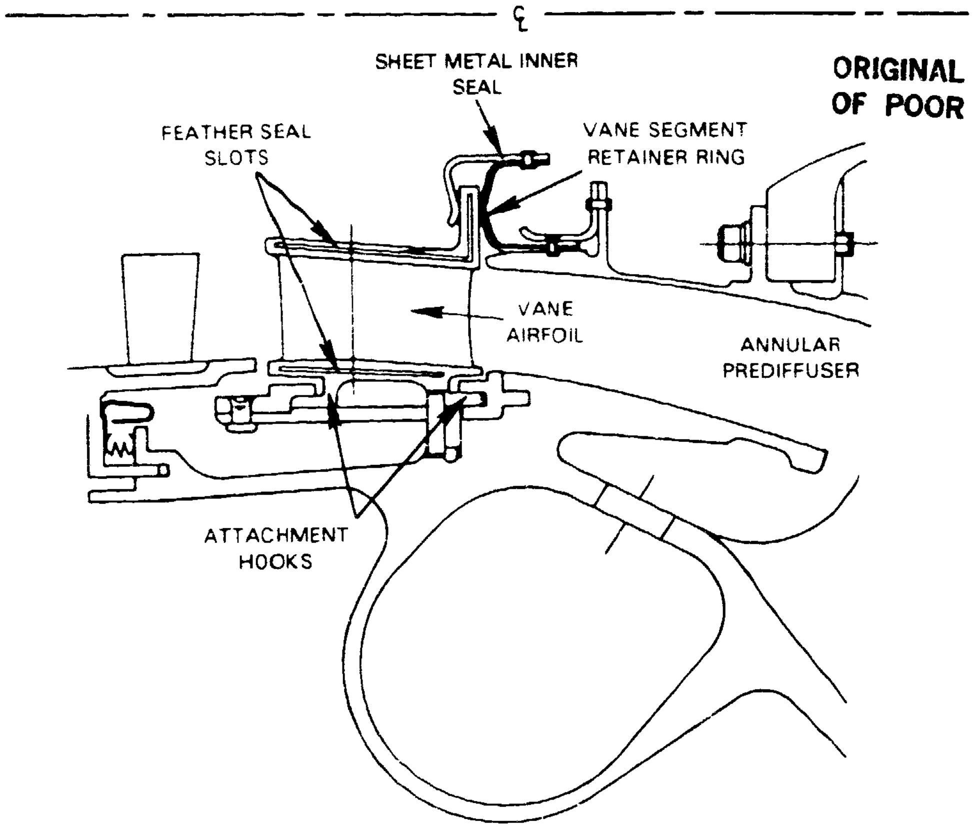

Figure 5.3.1-1 High-Pressure Compressor Exit Guide Vane Assembly

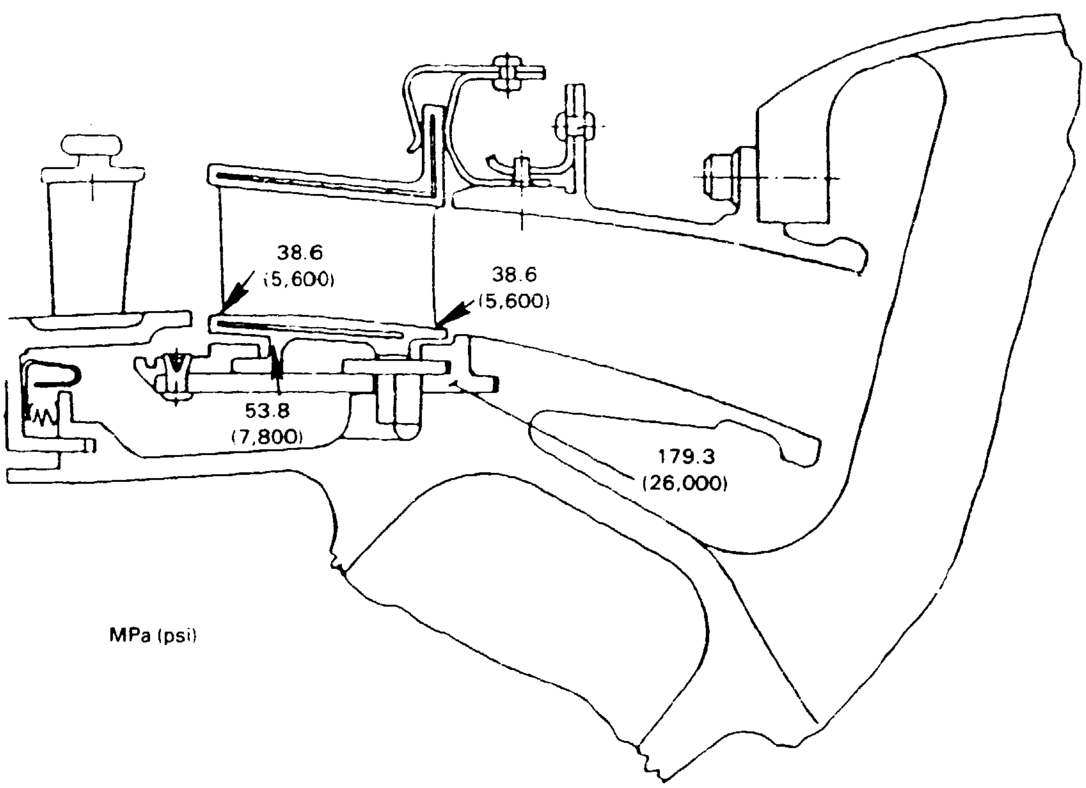

Figure 5.3.1-2 High-Pressure Compressor Exit Guide Vane Assembly Stress Summary

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5.3.2 Diffuser System

5.3.2.1 Mechanical Design Features

The diffuser case assembly, shown in Figure 5.3.2-1, includes 24 internal structural struts, a bleed manifold, and inner and outer cases, all of which are fabricated from cast and wrought Inconel 718 material. Connected to the inner flange of the diffuser case is the tangential on-board injection support case which is designed to sustain a blowoff load of 572,038 N (128,600 lb) onto the diffuser case. Located on the outer flange of the diffuser case are the bleed manifold and ports which take high-pressure compressor exit air that dumps at the prediffuser exit plane and directs this air to the active clearance control system and customer bleed. Welded to this outer flange is an extension of the diffuser case that is bolted to the high-pressure compressor rear case to provide the necessary frontal support for the subassembly. The outer prediffuser wall is integral with the diffuser case casting, while the inner wall is fabricated separately and bolted to the case. Fuel injector support assembly bosses, combustor support pin bosses, and ignitor bosses are located in the cast portion of the outer diffuser case. A forged rear skirt is welded to the outer case casting and bolted at the rear to the high-pressure turbine case. This skirt includes a port at the bottom to drain fuel that may accumulate during engine starting.

The diffuser case for the flight propulsion system will be cast and hot isostatically pressed for adequate fatigue strength to meet life requirements. However, to minimize cost and meet the schedule for fabricating the diffuser case in the integrated core/low spool, the case will not be hot isostatically pressed. Although the fatigue strength will be lower, it is adequate for the limited cyclic testing planned for the integrated core/low spool.

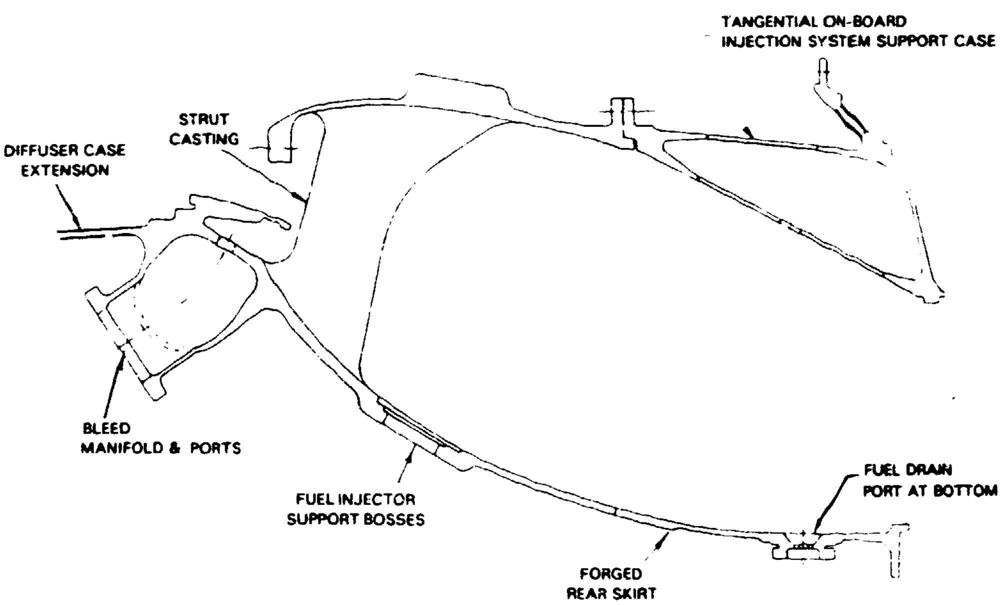

Figure 5.3.2-1 Diffuser Case Assembly

5.3.2.2 Structural Analysis

The diffuser case design evolved through several iterative configurations. A preliminary configuration, shown in Figure 5.3.2-2(a), started with simple radial struts connecting the inner flowpath structure to the outer engine casing. These struts were configured to carry the large 572,038 N (128,600 lb) axial blowoff load imposed on the inner structure by the tangential on-board injection support case. The design was subsequently modified to minimize deflection at the high-pressure compressor discharge seal that could result from radial forces acting on the high-pressure turbine inlet guide vane support. This modification, shown in Figure 5.3.2-2(b), involved moving the inner support ring further downstream and canting the strut further downstream to provide increased stiffness. However, detailed finite element analysis of this configuration indicated high stresses in the strut trailing edge when exposed to thermal gradients and blowoff loads, as shown in Figure 5.3.2-3. To reduce these stresses, another modification was made to thicken the strut at the trailing edge, fill in the leading edge inner diameter region of the strut and center the inner support ring closer to the middle of the strut. These revisions reduced stresses to the acceptable levels shown in Figure 5.3.2-4.

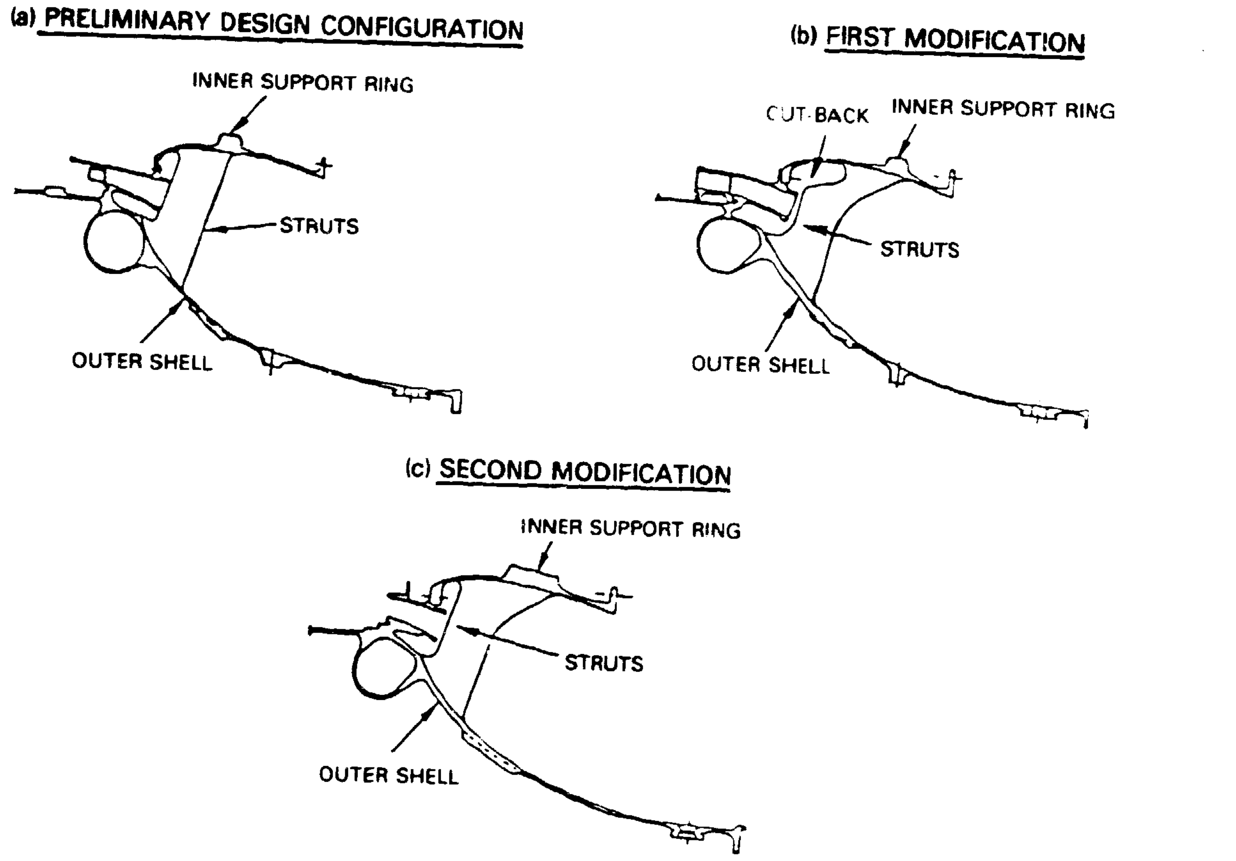

Figure 5.3.2-2 Evolution of the Diffuser Case Design To Satisfy Structural Requirements

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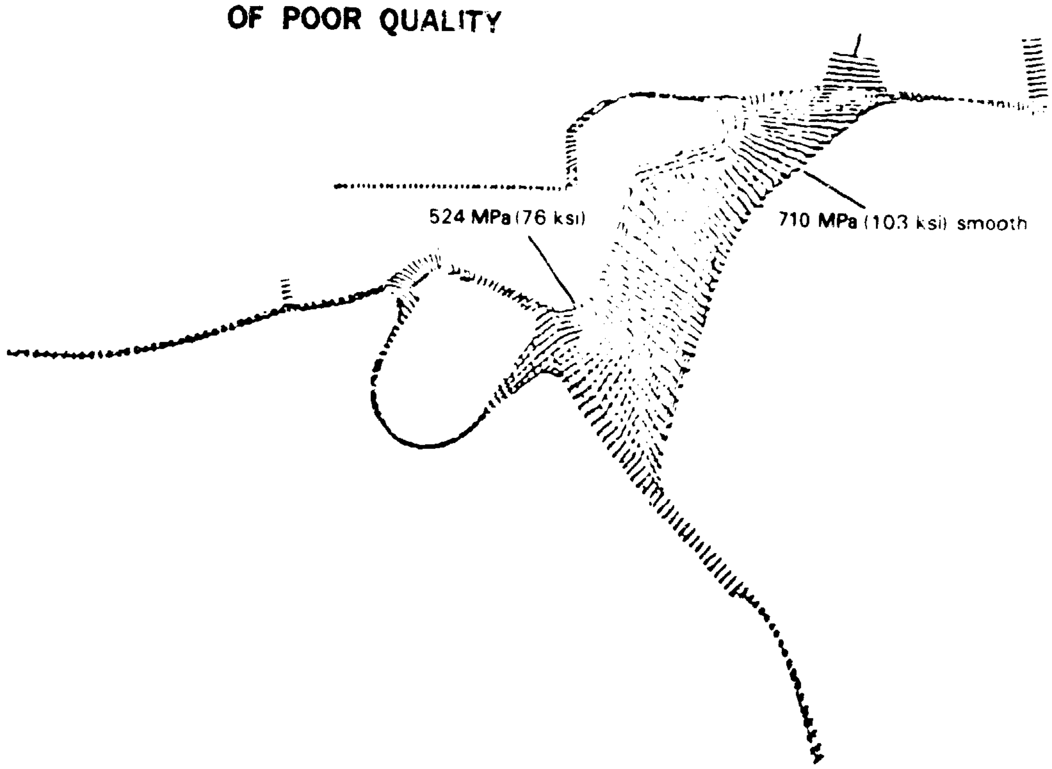

Figure 5.3.2-3 Finite Element Model Showing Stresses Resulting From Initial Modifications To The Diffuser Case Struts

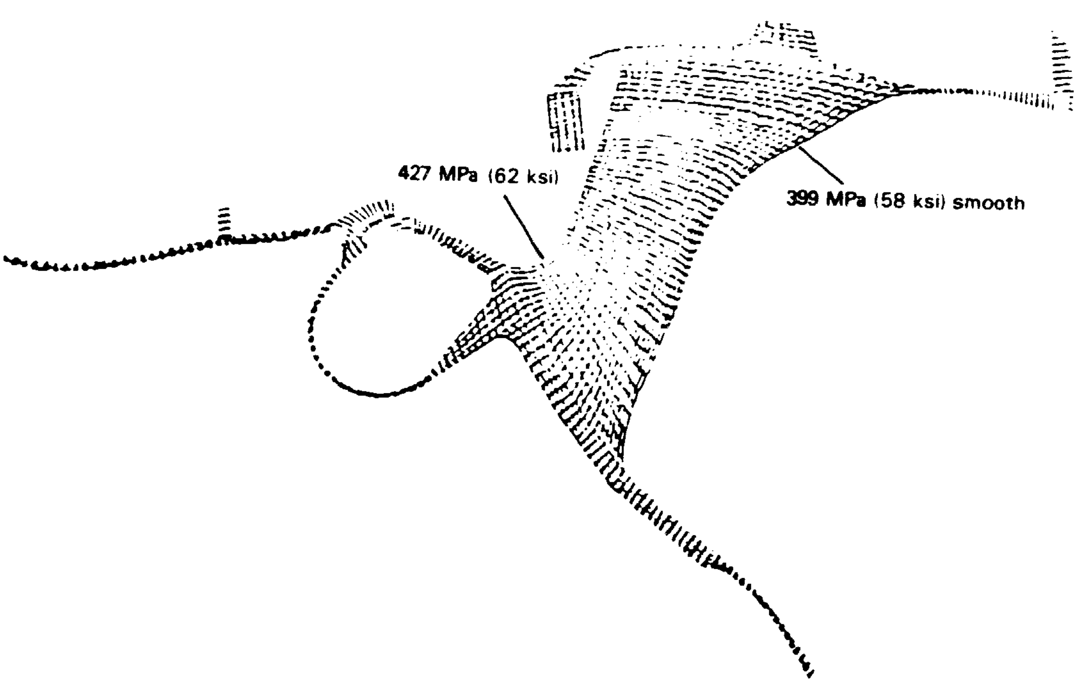

Figure 5.3.2-4 Finite Element Model Showing Stresses Resulting From The Final Modifications To The Diffuser Case Struts

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The diffuser case final configuration, Figure 5.3.2-2(c), will meet the flight propulsion system life goal of 15,000 cycles when hot isostatically pressed. Figure 5.3.2-5 summarizes the principle stresses in the diffuser case assembly as calculated for engine operation at hot day sea level takeoff conditions.

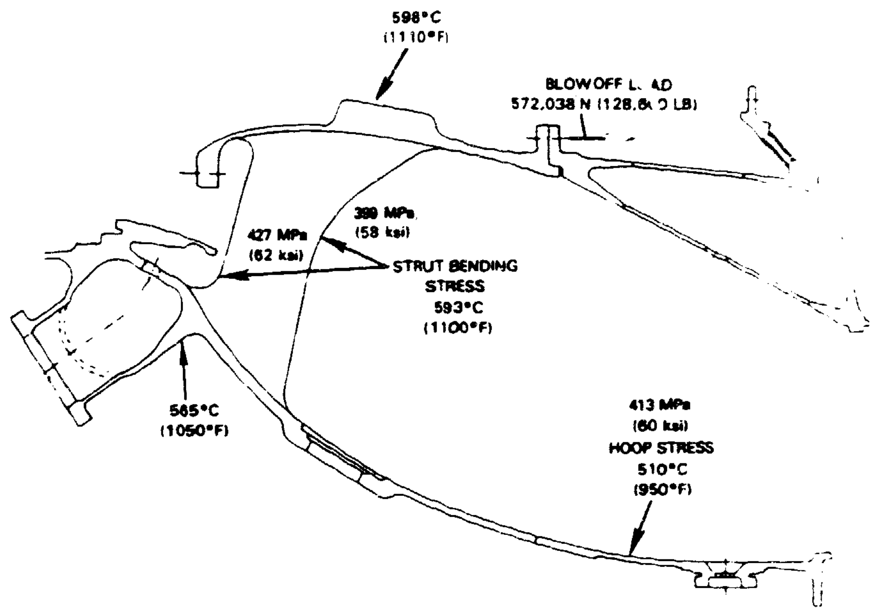

Figure 5.3.2-5 Diffuser Case Stress and Temperature Summary at Sea Level Takeoff Condition on an 28°C (84°F) Day

5.3.3 Combustor Section

The combustor section, shown in Figure 5.3.3-1, includes the front-end subassembly, the segmented liners and liner support structure, and the carburetor tubes. A detailed discussion of these subassemblies is presented in the following sections.

5.3.3.1 Front-End Subassembly

The front-end subassembly is illustrated in Figure 5.3.3-2. The primary elements are the hood, bulkhead, pilot zone fuel injector guides and heatshields. The hood directs the airflow exiting from the prediffuser into the inner and outer flow annuli as well as passes a portion of the air to feed the pilot zone combustion process. It is supported by the bulkhead, which acts as the major supporting member connecting the inner and outer liner support frames, hood and nozzle guides to the engine outer case (via the combustor mount pins). The pilot zone fuel injector guide accommodates the relative motion between the combustor and fuel injector support assembly as well as facilitates assembly. Heatshields protect the bulkhead support structure from pilot zone radiation heating. These heatshields are cooled by impingement air from upstream of the bulkhead. All major hardware is fabricated from Hastelloy X material.

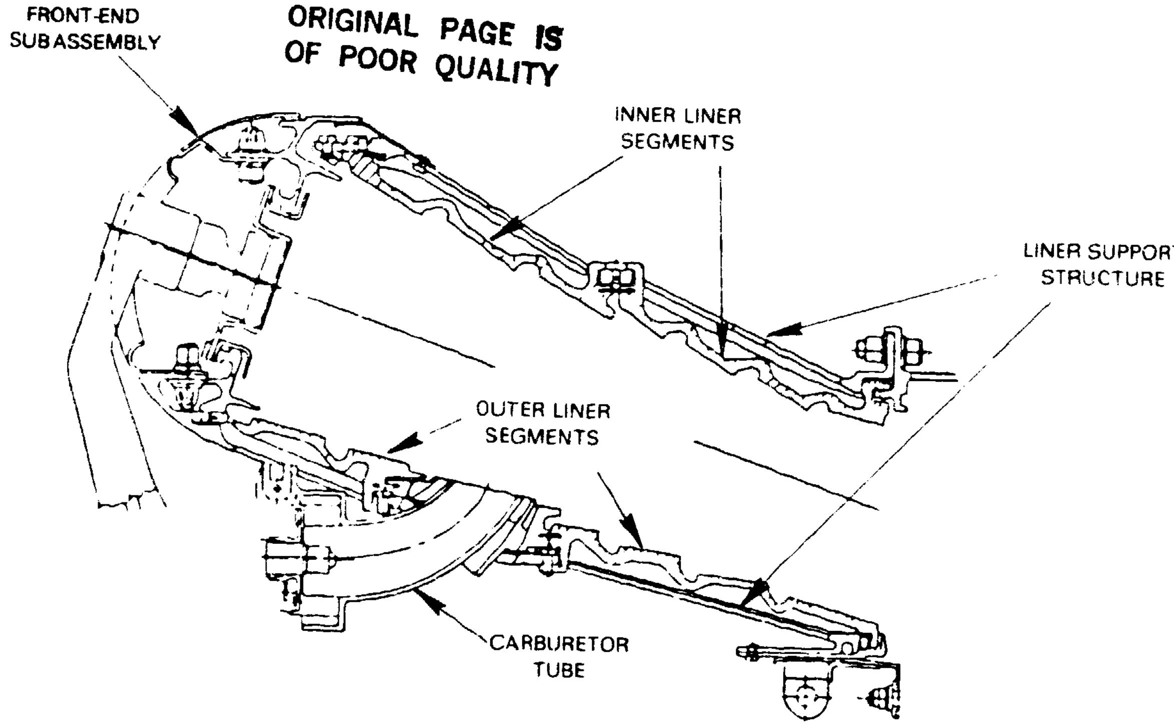

Figure 5.3.3-1 Major Subassemblies For The Energy Efficient Engine Combustor

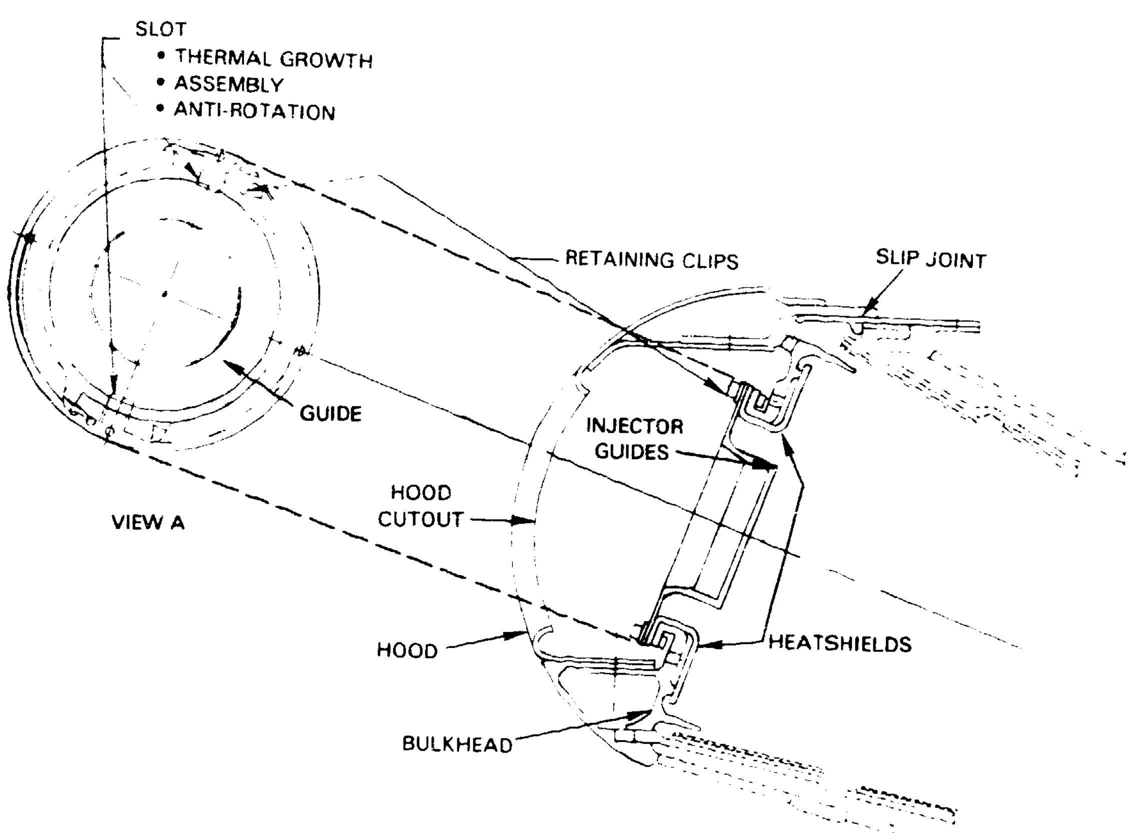

Figure 5.3.3-2 Combustor Front-End Subassembly

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5.3.3.2 Liners and Liner Support Structure

5.3.3.2.1 Mechanical Design Features

Life requirement for the flight propulsion system combustor liners and other structural hardware is 8000 hours or 4900 missions to satisfy the short mission application for the Energy Efficient Engine. Thermal and structural design studies were conducted to evaluate cooling techniques that offered high thermal effectiveness and to identify the most promising liner construction and material. A film cooled, full hoop, louvered liner was chosen as the baseline configuration from which to compare the other cooling techniques. Advanced cooling concepts such as counterflow film cooling (CFFC), counter-parallel FINWALL® (CPFW), impingement/transpiration and impingement/film were evaluated. Results indicated a cast segmented liner with the counter-parallel FINWALL® technique improves life by reducing hoop stress and providing higher thermal effectiveness.

As shown in Figure 5.3.3-3, both inner and outer liners are divided into circumferential segments in the pilot and main zones. Since the liner is constructed of segments, the use of a cast high temperature turbine-type alloy is possible to further improve liner life. The material selected for life analyses was B-1900 + Hf, an investment cast nickel base alloy with good strength in the 760 to 1037°C (1400 to 1900°F) temperature range.

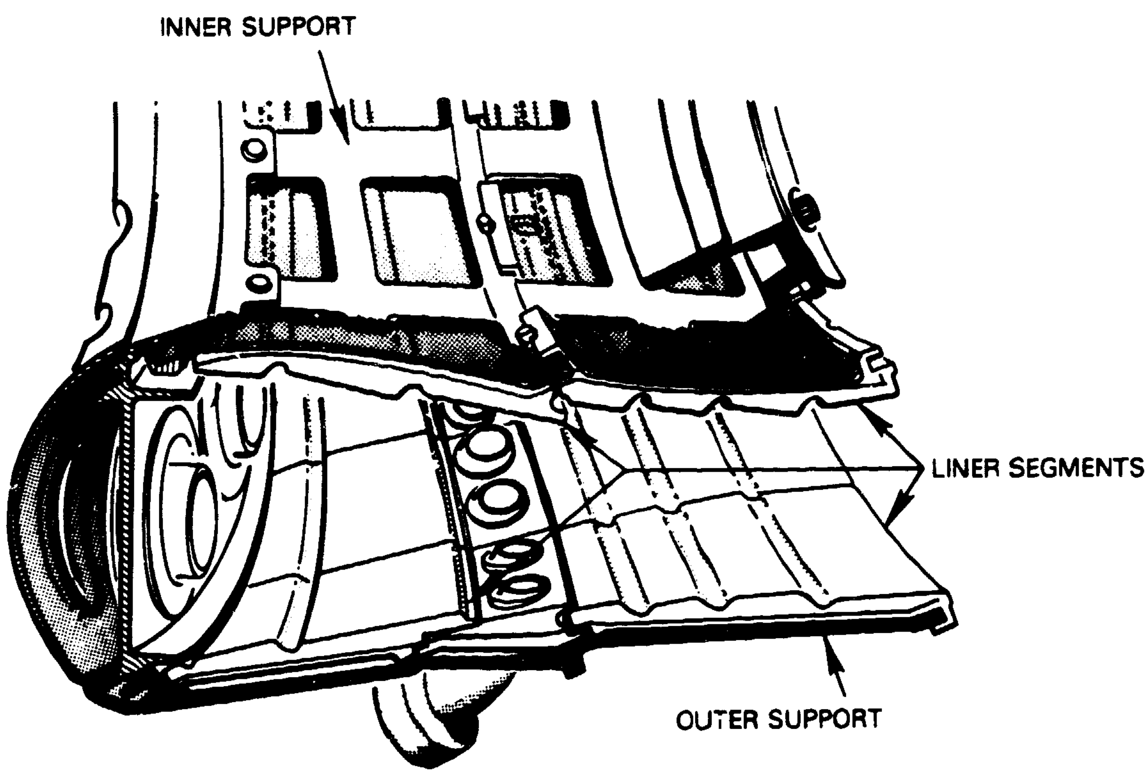

Figure 5.3.3-3 Combustor Section Illustrating Segmented Liners and Liner Support Structure

The segments are supported by inner and outer structural frames machined from Hastelloy X material. Conventionally machined hooks on the back of each segment engage with the circumferential rails on the structural frame to position the segments. Axial and circumferential feather seals control air leakage through the gaps between the adjacent segments. Experience gained in feather seal technology from the Energy Efficient Engine High-Pressure Turbine Leakage Technology Program (Ref. 4) was used in designing these seals. Slots for these feather seals are installed by electro-discharge machining. In addition to improving cooling air management and durability, the replaceable segments enhance component maintainability. Liner front-end support is provided by the bulkhead in the front-e d subassembly which, in turn, connects the liner support structures to the engine case though the use of mount pins.

Counter-parallel FINWALL®, illustrated in Figure 5.3.3-4, is primarily a convective cooling technique that offers higher thermal effectiveness compared to conventional film cooling methods. Air enters the cool side of the liner through slots and flows both upstream (counter to hot gas flow) and downstream through channels to convectively cool the liner. When exiting from the upstream end of the channel, the flow turns 180 degrees and joins the downstream effluent to 'film cool' the liner hot side. Cooling air feed slots, which intersect the counter-parallel FINWALL® channels and supply cooling air from the cold side of the liners, are electro-discharge machined to a dimension of 0.444 by 0.152 cm (0.175 by 0.060 in). The counter-parallel FINWALL® channels used to convectively cool the liner hot side are electrochemically machined to a 0.088-cm (0.035-in) diameter. This method of fabrication was selected as being the most feasible and cost effective compared to two other fabrication methods investigated: (1) one-piece casting with the cooling geometry cast in place and (2) a multipiece casting and subsequent bonding. The one-piece casting method required that the cooling channels be formed using 0.088-cm (0.035-in) diameter quartz rods (approximately 250 rods per segment), which would be difficult to hold in position during the casting process. The multipiece casting and bonding method required a bonding development program to establish new tooling and bonding techniques to successfully fabricate a liner that meets the high temperature/high pressure requirements.

5.3.3.2.2 Thermal and Structural Analyses Leading to Liner Selection

In conjunction with initial design studies, thermal screening of five different candidate liner cooling configurations was conducted at the conditions identified in Table 5.3.3-I. These conditions are representative of hot day sea level takeoff operation. Results from this analysis are shown in Figure 5.3.3-5. This figure indicates that the film cooled and impingement/film cooled louver designs would operate at a nominal maximum wall temperature in excess of 1026°C (1880°F) at 100 percent of available combustor airflow. With possible 50 - 100°C (90 - 180°F) increases in wall temperature resulting from local hot spots caused by maldistribution of fuel and/or air and material temperature limits, the wall temperature levels preclude the utilization of these two configurations. The three convective cooling configurations (counterflow film cooling, counter-parallel FINWALL®, and impingement/transpiration) operate at maximum nominal wall temperatures below 876°C (1610°F) at 100 percent of available combustor airflow and, thus were chosen for further durability analysis as described in Section 5.3.3.2.3.

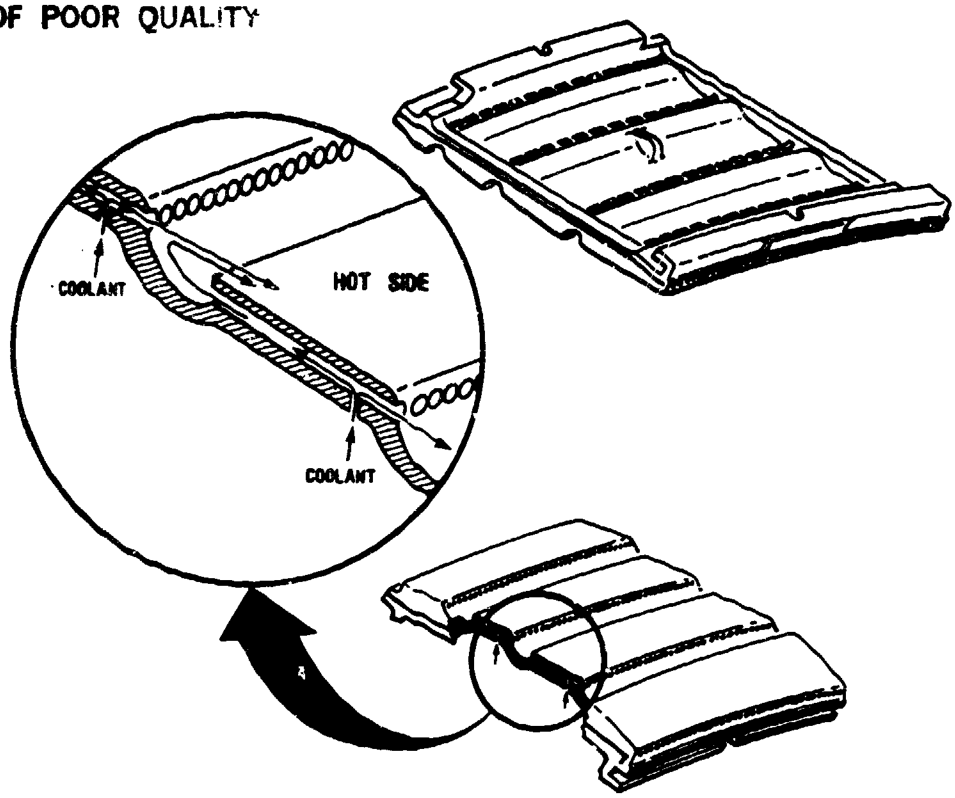

Figure 5.3.3-4 Typical Combustor Liner Segment Illustrating the Counter Parallel FINWALL® Convective Cooling Technique

TABLE 5.3.3-I

COMBUSTOR OPERATING CONDITIONS DURING SCREENING STUDIES
(Hot Day, 29°C (84°F) Sea Level Takeoff Conditions)

Combustor Inlet Temperature °C (°F)	580 (1077)
Combustor Exit Temperature °C (°F)	1482 (2700)
Combustor Inlet Atmospheric Pressure MPa (psi)	3.1 (455.7)
Overall Fuel/Air Ratio	0.028
Liner Pressure Drop, % of P_{T3}	2.5

Durability analysis results indicated life goals could only be achieved with a segmented liner construction employing counter-parallel FINWALL® cooling and higher temperature capability materials. A detailed thermal analysis was subsequently performed to optimize the candidate convective cooling scheme and establish the temperature distribution on each of the liner segments. This analysis was made using deteriorated engine sea level takeoff, hot day (29°C (84°F) ambient temperature) conditions. The results, presented in Figure 5.3.3-6, indicate the cooling passage size and spacing required to achieve the liner temperature distribution shown.

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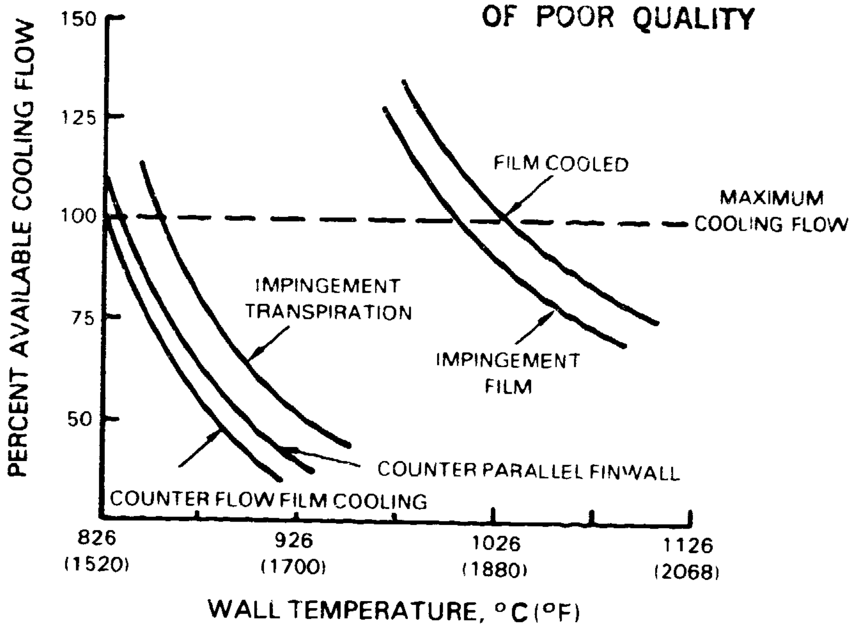

Figure 5.3.3-5 Cooling Technique Analysis Results

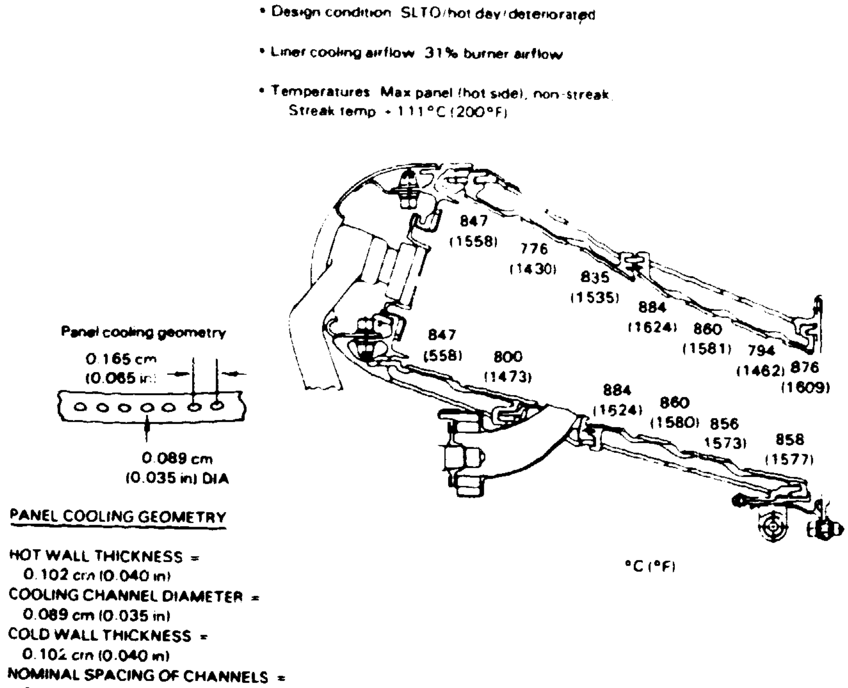

Figure 5.3.3-6 Optimized Segment Panel Cooling Geometry for Liners

Conventional sheet metal or machined liners are characterized by very short thermal time constants. This allows the whole system to adjust rapidly to changing boundary conditions as experienced in engine snap accelerations and decelerations. Since there is no thermal lag in such liners, calculation of thermal gradients at steady state maximum power operation is sufficient to determine the maximum induced stresses in the liner. Considering the increase in mass for a cast segmented liner and the relative isolation of the segment hook from the hot panel, thermal lag might be expected during a transient. To investigate this possibility, a three-dimensional transient thermal analysis was conducted on the thickest portion of a longitudinal section of a segment.

The thermal model used in the analysis is shown in Figure 5.3.3-7 and consists of a narrow slice of a main combustion zone segment, which includes the last counter-parallel FINWAL® cooling panel in the segment, one half cooling channel and web thickness, and the rear hook. Typical engine transient snap acceleration data were used to establish the time variation of the boundary conditions for the Energy Efficient Engine. The results of the analysis, presented in Figure 5.3.3-8, indicate that the maximum temperature gradient between hot and cold side surfaces occurs at about nine seconds after starting an engine acceleration.

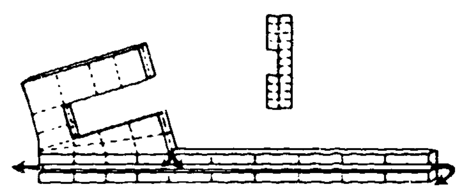

Figure 5.3.3-7 Thermal Model of Longitudinal Section of the Advanced Liner Segment

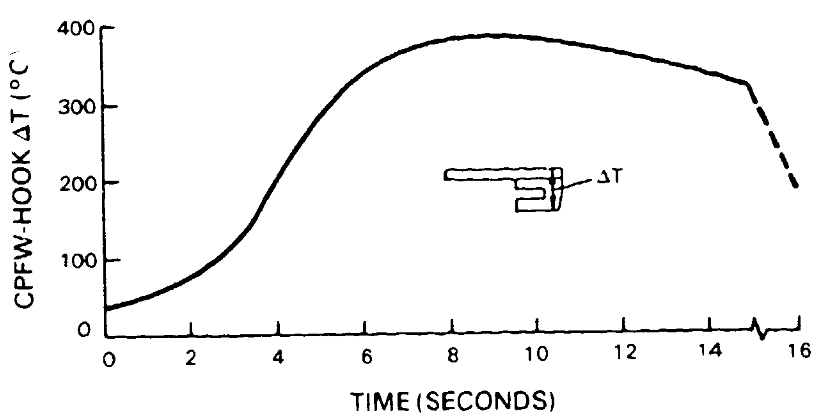

Figure 5.3.3-8 Analysis of Thermal Model of Longitudinal Section of the Advanced Liner Segment

A detailed structural analysis was based on an analytical model that represented the segment geometry, as shown in Figure 5.3.3-9. A finite element computer program with nonlinear analysis capability was utilized. The nonlinear analysis was used to study any dimensional instability that might result from accumulating creep and plasticity during cycling. An elastic analysis was used to set attachment clearances and calculate cyclic life.

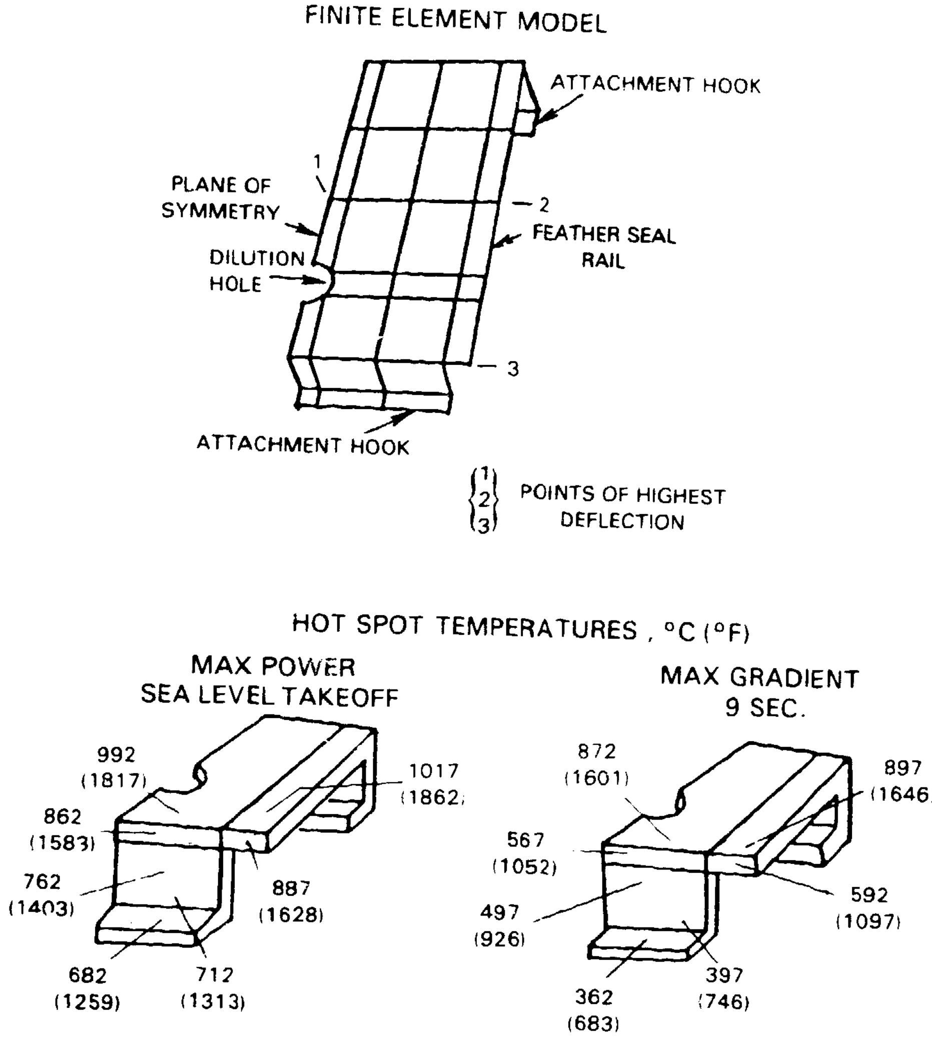

Figure 5.3.3-9 Structural Model of Longitudinal Section of the Advanced Liner Segment Geometry and Predicted Temperatures

Two conditions were analyzed by the finite element program to determine segment maximum stresses. These were sea level takeoff and the maximum temperature gradient during acceleration. The results, shown in Figure 5.3.3-10, indicate that a maximum stress of 462 MPa (67.7 ksi) was observed when the maximum temperature difference occurred at the hook section during acceleration (condition 2). Maximum stress was consistently located at the ends of the segment over the attachment rails; axial cracks would eventually be initiated in streaked segments in those locations. The maximum elastic stress was converted to an equivalent strain that was used to refine the prediction for crack initiation.

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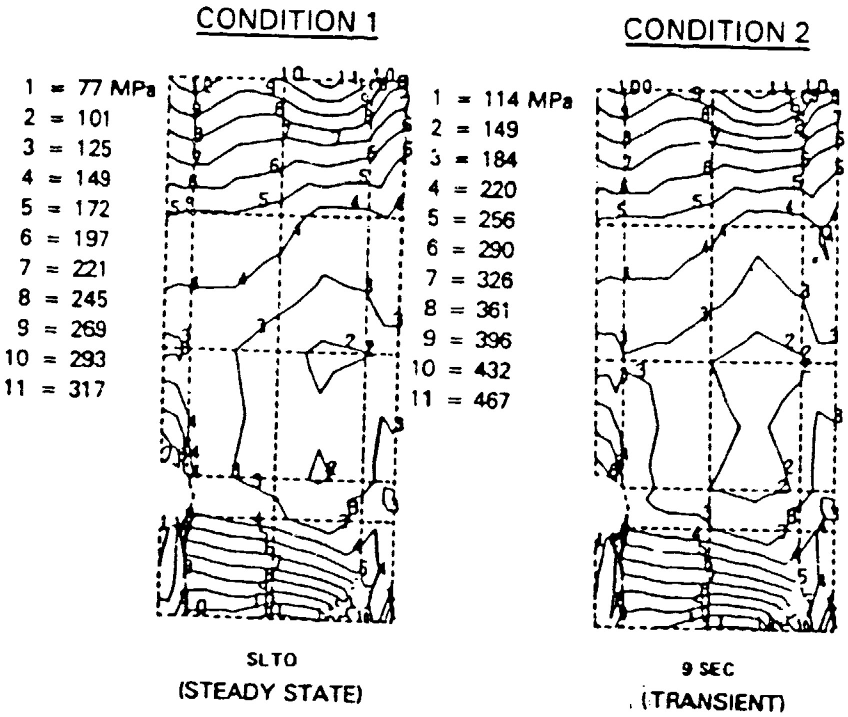

Figure 5.3.3-10 Advanced Liner Segment Maximum Stress Results

5.3.3.2.3 Durability Assessment

Full Hoop Liner Construction

Results from thermal and structural analyses were utilized to assess life characteristics of liners utilizing counterflow film cooling, counter-parallel FINWALL® and impingement/transient convective cooling techniques. Liner lives are generally limited by thermally-induced strains. The axial and radial liner temperature distributions obtained in the thermal analysis were utilized to estimate the cyclic life for the three candidate cooling configurations. Plastic and creep strains were calculated and a strain range partitioning method was used to obtain cyclic lives to crack initiation for the counter-parallel FINWALL® and counterflow film cooling techniques. Fracture mechanics analysis was employed for the impingement/transpiration configuration where the assumption of an initial crack was made and the subsequent cyclic life for transpiration hole-to-hole crack link-up was calculated. The results of the initial life study are presented in Table 5.3.3-II. None of the three cooling configurations met the Energy Efficient Engine Program life goals in a full hoop conventional louvered liner. Although counter-parallel FINWALL® and counterflow film cooling techniques are very similar, the presence of holes in the hot sheet of the counterflow film cooling severely limits the low cycle fatigue life.

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TABLE 5.3.3-II

PREDICTED FULL HOOP LINER LIFE COMPARISON
WITH CANDIDATE COOLING CONFIGURATIONS
(Material: Hastelloy X)

Cooling Scheme	Life Relative to Program Goals (percent)
Counter-Parallel FINWALL®	50
Counterflow Film Cooling	20
Impingement/Transpiration	20

Segmented Liner Construction

The segmented liner construction was subsequently investigated as a way to improve life through reducing hoop stress. A life prediction model based on creep relaxation and ductility exhaustion was used for the counterflow film cooling and counter-parallel FINWALL® configurations, while a fracture mechanics approach was retained for the impingement/transpiration configuration.

The effect of segmenting and material type on combustor liner life with counter-parallel FINWALL® cooling is shown in Table 5.3.3-III. The potential for a fourfold improvement in life, relative to a Hastelloy X full hoop liner, is evident when segments of high temperature alloys such as B-1900 + Hf (PWA 1455) are used. The ranking of candidate segmented configurations with the different cooling techniques is shown in Table 5.3.3-IV. As the tables indicate, the only segmented configuration that meets or exceeds program goals is the counter-parallel FINWALL® cooling technique using B-1900 + Hf material.

TABLE 5.3.3-III

EFFECT OF SEGMENTING AND MATERIAL ON LINER LIFE
(Counter-Parallel FINWALL® Cooling)

Construction	Material	Relative Life (Percent) Program Goal	Relative Life (Percent) Full Hoop
Full Hoop	Hastelloy X	50	100
Segmented	Hastelloy X	80	160
Segmented	B-1900 + Hf (PWA 1455)	200	400

TABLE 5-IV

PREDICTED LINER LIFE RANKING FOR SEGMENTED CONSTRUCTION
EMPLOYING CANDIDATE COOLING TECHNIQUES
(Material: B-1900 + Hf)

Cooling Scheme	Life Relative to Program Goals (percent)
Counter-Parallel FINWALL®	200
Impingement/Transpiration	50
Counterflow Film Cooling	20

The structural integrity of the liner design was demonstrated successfully during the Sector Combustor Rig Test Program at the full pressure and temperature conditions of the Energy Efficient Engine combustor.

5.3.3.3 Carburetor Tubes

Fuel is introduced into the main zone through the carburetor tubes shown in Figure 5.3.3-11. The carburetor tube design evolved from fuel injector characterization tests described in Section 6.3.1. Fuel is supplied to each of the carburetor tubes by a simplex pressure atomizer. A fraction of the fuel is vaporized in the tube, while the remaining fuel is centrifuged to the walls of the tube by air introduced through a radial inflow swirler. The fuel film formed at the exit plane of the tube is sheared into droplets by the swirling core and secondary airflow jets. The core radial inflow swirler consists of ten curved vanes with an overall width of approximately 1 cm (0.7 in). The secondary airstream swirler contains nine 20-degree axial vanes. The main nozzle enters the tube through a floating guide that accommodates the relative displacement between the carburetor tube and pressure-atomizing fuel nozzle and facilitates assembly.

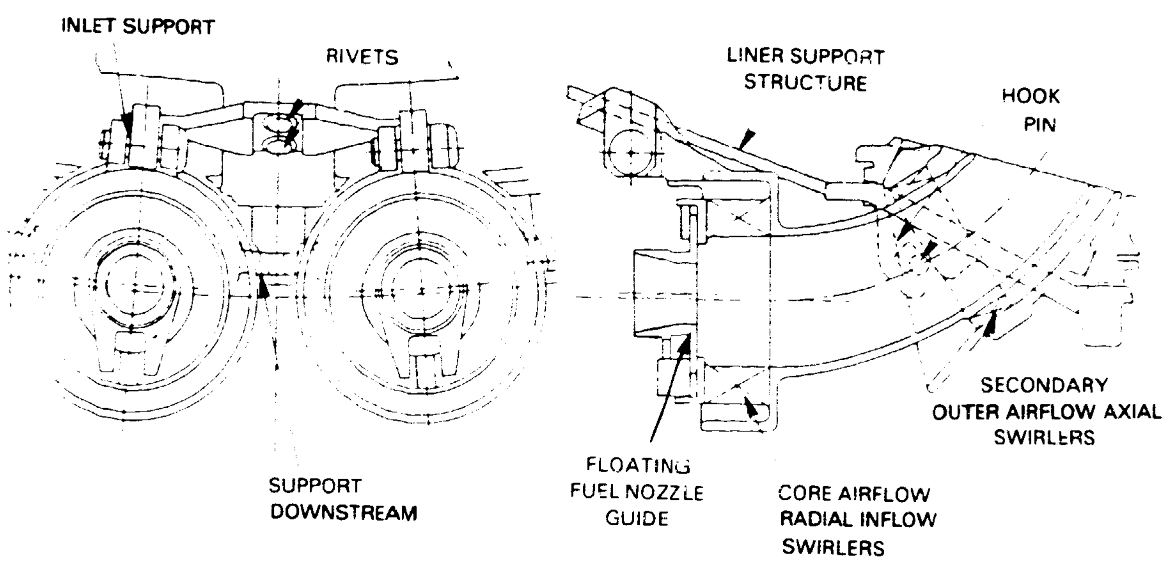

Figure 5.3.3-11 Combustor Main Zone Carburetor Tube Assembly

Carburetor tube assemblies are cast from Inconel 625 material, a non-cobalt alloy that provides adequate strength. Because of the complexity of the design, a two-piece casting approach is utilized with the bond joint (which separates the castings) being located near the plane of the secondary swirlers.

The carburetor tube assembly is supported with a simple riveted joint on the outer liner support structure. The downstream end of the tube is supported by pins that protrude from the tube casting and engage simple hooks on the liner support structure.

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5.3.4 Fuel Management System

The combustor fuel management system is comprised of the fuel ignitors, fuel injectors, and injector structural support assemblies, plus the fuel manifold safety and manifold shroud. The mechanical design features of these subassemblies and a structural analysis of the fuel injector support assemblies are discussed in the following paragraphs.

5.3.4.1 Mechanical Design Features

Fuel Ignitors

The ignitor plug penetrates the pilot zone outer liner wall. The two ignitor plugs required for the Energy Efficient Engine combustor are identical and of conventional design. A dimensional review of ignitors that are currently available indicated an existing configuration used in the Pratt & Whitney Aircraft J52 engine could be adapted easily in the combustor component. The only required modification is an extension of existing mounting threads where the ignitor threads into the diffuser case near the fuel injector support assembly and protrudes through the pilot zone outer segmented liner. This thread extension is shown in Figure 5.3.4-1 and provides the necessary penetration into the pilot zone area to allow proper ignition over the range of conditions representative of the flight envelope. This approach provides the combustor with an ignitor that is proven reliable and at a substantially lower cost compared to producing a unique prototype part.

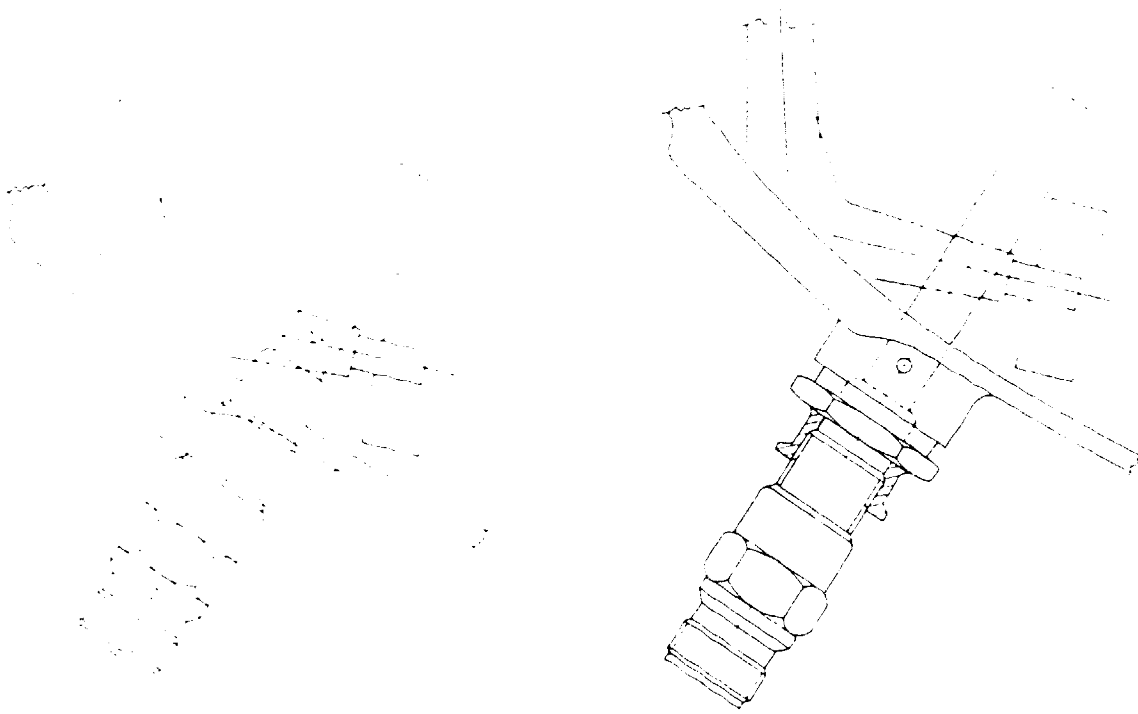

Figure 5.3.4-1 Modified Ignitor for Energy Efficient Engine Combustor

Injector Structural Support Assemblies and Fuel Injectors

The fuel injector structural support assembly, illustrated in Figure 5.3.4-2, is made from cast AISI 347 stainless steel material which possesses high temperature capability, good castability and good ductility. Each support assembly provides for two main zone pressure atomizers injectors and a pilot zone single pipe aerated injector. Each of the 24 fuel injector assemblies is inserted through the case from the inside of the engine and secured to the case with four bolts. To minimize the possibility of fuel coking as the fuel passes through the fuel injector assembly, all exposed surfaces are protected with welded-on sheetmetal heatshields. Separate fuel connections for the pilot zone and main zone are provided at each fuel injector support assembly location.

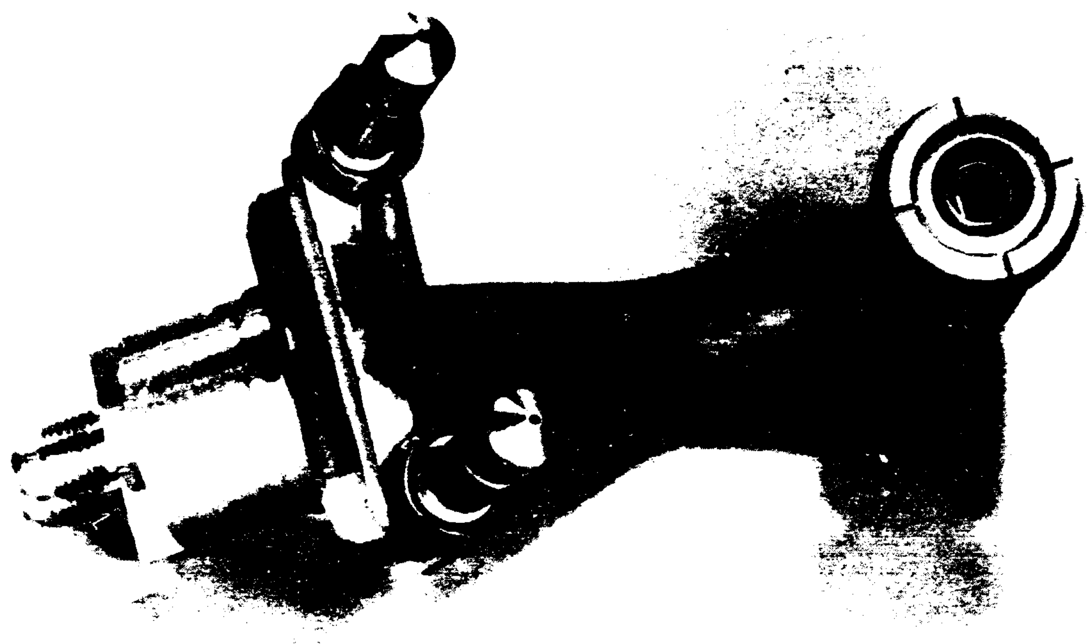

Figure 5.3.4-2 Pilot Zone/Main Zone Fuel Injector Support Assembly

The fuel inlet connections, illustrated in Figure 5.3.4-3, are provided with standard cone-type fittings brazed to the casting and include internal 'last chance' fuel filters and an orifice metering plate on the pilot zone fuel supply. This orifice metering plate permits fine-tuning of the fuel flow.

Fuel Manifold and Manifold Safety Shroud

The three major features of the fuel manifold system, illustrated in Figure 5.3.4-4, are: (1) the pigtail connectors from the fuel manifold to the fuel injector, assemblies (2) the fuel manifolds and their connections to the fuel supply lines, and (3) the sealing safety shroud surrounding the manifolds.

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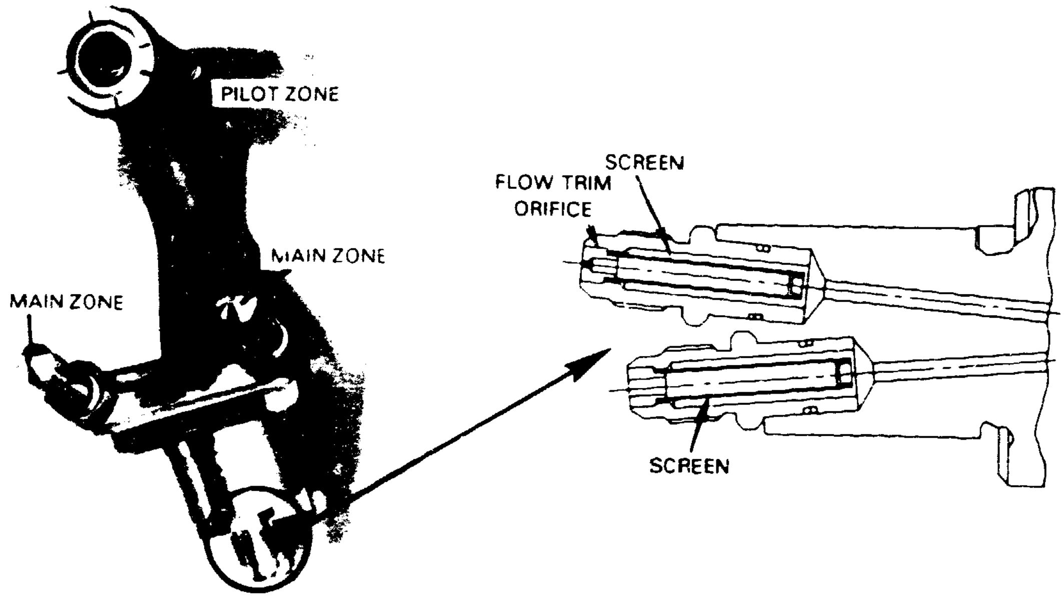

Figure 5.3.4-3 Fuel Inlet Section of Fuel Injector Support Assembly

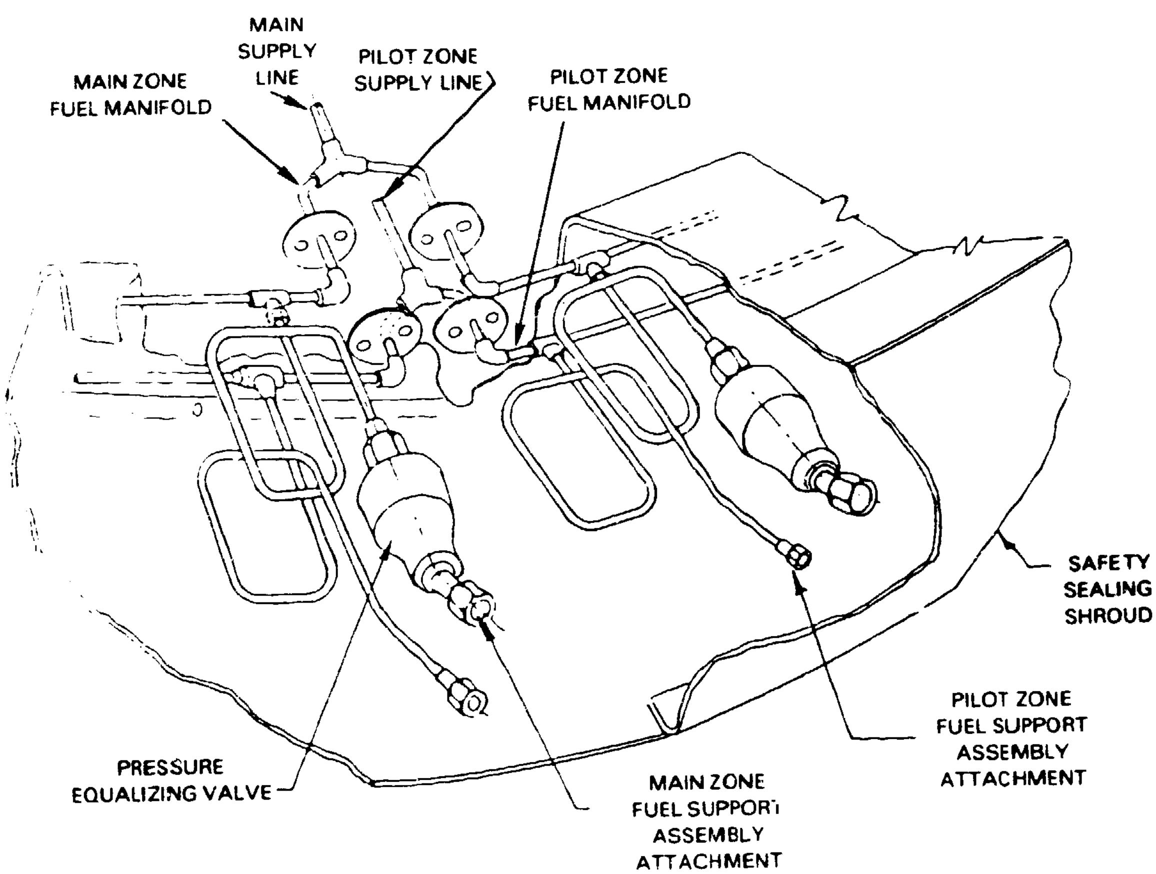

Figure 5.3.4-4 Fuel Manifold and Safety Shroud

The connector tubes from the fuel manifold to the fuel injector support assemblies are referred to as pigtail connector tubes. They are designed with a helix type flexibility to accommodate the thermal growth differences between the relatively cool fuel manifold and the hot fuel injector/diffuser case. The pilot zone pigtail connector is a simple pipe connection, while the main zone fuel system includes both the pigtail connector and a pressure equalizing valve. The pressure equalizing valve is used in the main zone fuel system to ensure that uniform fuel pressure is supplied to injectors located at both the top and bottom of the engine. To prevent vibration of these connector tubes, support clips are provided between the tubes in the helix region. In addition, support clips tie the pressure equalizing valve to the sealing shroud. Identical pilot zone and main zone connector tubes are used for each of the 24 fuel injector support assemblies, thereby minimizing the cost of tube fabrication.

Separate manifolds are used to supply fuel to the pilot and main zones. The fuel supply is manifolded circumferentially around the engine by using a succession of bent tube sections attached to tee fittings. Each tee fitting provides the connection point for the pigtail tube at each of the 24 fuel injectors. Building the fuel manifold by this succession of identical bent tubes and tee fittings minimizes the cost of the fuel manifold. To minimize vibration of the fuel manifolds, clips are provided to connect the fuel manifold to the sealing shroud at several circumferential locations. Fuel supply to the fuel manifolds is brought in at the bottom location through sealed holes in the sealing shroud.

A sheetmetal sealing shroud is provided to prevent fuel leaks from impinging on hot engine cases. This shroud is also designed to provide an enclosure in which cool air will be used to ventilate the fuel manifold system during the full annular rig testing. This cool air ventilation minimizes the chance of any fuel coking caused by the high temperature air environment of the combustor rig. The radial growth incompatibility between the sealing shroud and the diffuser case is accommodated by spring connectors between the cases. These spring connectors have a cross section in the form of the letter 'C.' One side of the 'C' is bolted to the diffuser case with the same bolts used to attach the fuel injector support assemblies. The other side of the 'C' is bolted to the sealing shroud. To facilitate assembly of the sealing shroud, it is segmented into two 180-degree circumferential sectors so that all fuel plumbing can be assembled before the shroud is slid over it. Potential vibration of the sealing shroud case is minimized by several brackets connecting the front end of the shroud to bleed ports on the diffuser case manifold.

5.3.4.2 Structural Analysis

Detailed structural analysis of the fuel management system was confined to analysis of the fuel injector support assemblies to ensure adequate strength under any potential vibratory loading. The resultant support structure incorporates a thickness distribution suitable for maintaining stresses due to fore, aft and lateral (sideways) loadings below maximum allowable limits established through engine experience. As expected, maximum stresses occur at the junction of the pilot zone injector support structure and the diffuser case attachment point. These injector support stresses are summarized in Figure 5.3.4-5.

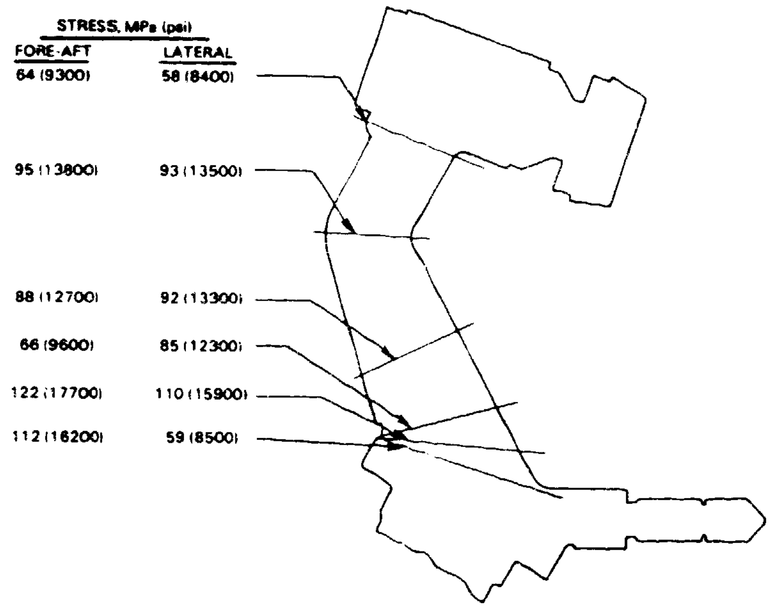

Figure 5.3.4-5 Fuel Injector Support Assembly Stress Summary

5.3.5 Component Weight Summary

Preliminary weight analyses were conducted for the combustor component as configured for the integrated core/low spool. Results of these analyses are presented in Table 5.3.5-I. A detailed weight assessment will not be performed until the final flight propulsion system preliminary design update.

TABLE 5.3.5-I

PRELIMINARY WEIGHT SUMMARY FOR INTEGRATED CORE/LOW SPOOL
COMBUSTOR COMPONENT

	Weight, kg (lb)
Diffuser Case	151 (335)
Combustor Front End	16 (35)
Inner Combustor Liners and Support Frame	21 (47)
Outer Combustor Liners and Support Frame	45 (99)
Fuel and Ignition Systems	87 (194)
Total	320 (710)

SECTION 6.0
SUPPORTING TECHNOLOGY PROGRAMS AND DESIGN SUBSTANTIATION

6.1 INTRODUCTION

To provide technical insight and substantiation of key design features in the Energy Efficient Engine combustor component, two supporting technology programs were conducted. The Diffuser/Combustor Model Test Program and Sector Combustor Rig Test Program were performed concurrent with the component design effort. Aerodynamic performance was evaluated in a full annular Plexiglas® model of the prediffuser/combustor section. Critical technology concepts such as the fuel injector designs and segmented liner were evaluated in a modular, high-pressure test rig using a 90-degree sector of the full annular combustor component.

Pertinent results of these programs are summarized in the following sections. Complete documentation of the results is contained in the Energy Efficient Engine Diffuser/Combustor Model Test Report (CR-165157) and Energy Efficient Engine Sector Combustor Rig Test Program Technology Report (CR-167913).

6.2 DIFFUSER/COMBUSTOR MODEL TEST PROGRAM

6.2.1 Overview

The objective of this supporting technology program was to optimize and document the aerodynamic performance of the Energy Efficient Engine diffuser/combustor design, which employs an outwardly canted combustor located downstream of a strutless, curved-wall prediffuser. Performance goals included demonstration of a separation-free prediffuser flow field, a diffuser pressure loss of less than 3 percent of the total high-pressure compressor exit pressure, and an overall component pressure loss of less than 5.5 percent of the high-pressure compressor exit pressure at the design airflow distribution.

Annular Plexiglas® models of three prediffuser/combustor configurations were tested in the rig shown schematically in Figure 6.2.1-1. The key parameters of the three models are defined in Table 6.2.1-I. Configuration I was the design recommended from analytical studies, and configurations II and III represent two more aggressive aerodynamic designs.

TABLE 6.2.1-I
GEOMETRIC CHARACTERISTICS OF PREDIFFUSER CONFIGURATIONS

Config	$L/\Delta R$	Area Ratio	Turning (degrees)*
I	3.5	1.50	14
II	3.5	1.57	10
III	3.0	1.50	14

* Includes 5 degrees wall cant before the prediffuser inlet.

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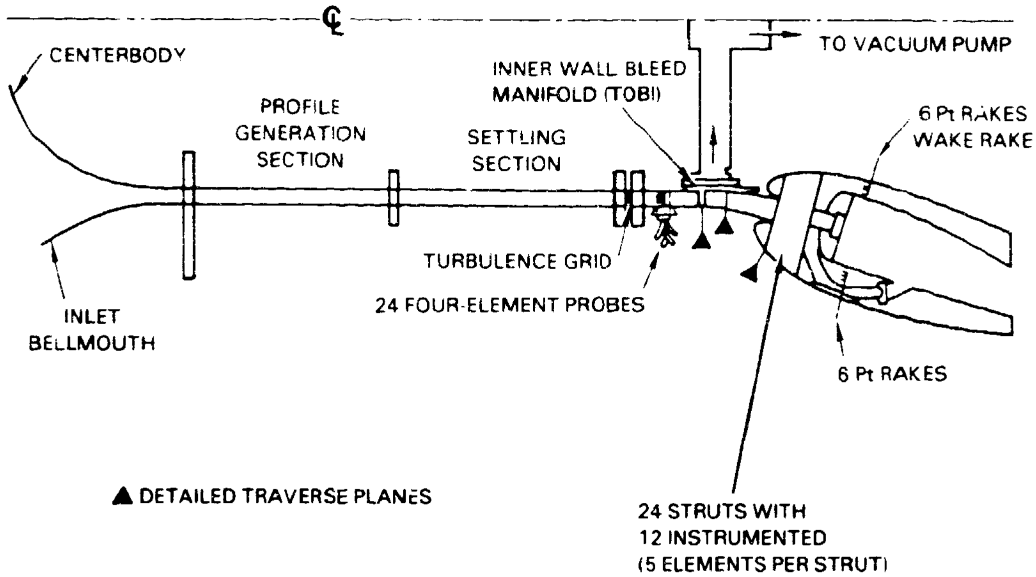

Figure 6.2.1-1 Diffuser/Combustor Model Test Rig

The major program elements consisted of: (1) prediffuser performance evaluations, (2) prediffuser/combustor performance integration tests, and (3) aerodynamic sensitivity assessment. Testing was conducted at near atmospheric pressure and temperature, with the rate of engine inlet airflow adjusted.

6.2.2 Test Results

Prediffuser Performance

Initial flow visualization tests, using tufts attached to the wall and a tufted wand, indicated that all configurations were separation free with inner, center and outer peaked inlet profiles. However, configuration III was judged to be closer to separation than the others.

Prediffuser/Combustor Performance

The performance of each prediffuser was measured with a simulated combustor module located downstream to provide the hood design back pressuring effect and the required inner to outer annuli flow split. The results, presented in Table 6.2.2-I, indicate that the diffuser total pressure losses are nearly identical and below the goal for each design. The flow turning angle was 14 degrees for configuration I and 10 degrees for configuration II. Less turning in the prediffuser resulted in a larger flow incidence angle on the combustor hood. Thus, the lower exit Mach number of configuration II was apparently negated by increased losses associated with flow around the combustor front end.

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TABLE 6.2.2-I
DIFFUSER/COMBUSTOR SYSTEM PERFORMANCE CHARACTERISTICS

Config	Prediffuser Press. Recov. (Cp) Inner	Outer	Efficiency	Exit Mn	Diffuser Pressure Loss (%) Inner	Outer
I	0.45	0.34	0.71	0.176	2.2	2.6
II	0.52	0.36	0.74	0.168	2.2	2.6
III	0.47	0.33	0.72	0.176	2.2	2.7

Based on the results of the stability and performance testing, configuration I, recommended from preceding analytical studies, was selected as the candidate for the Energy Efficient Engine.

Aerodynamic Sensitivity Study

Tests demonstrated that prediffuser performance was insensitive to inner and outer diffuser case annuli airflow shifts of up to 6 percent of the prediffuser exit airflow from the baseline value. The prediffuser pressure loss variation was less than or equal to 0.2 percent. The stability parameter, defined as the rate of change in pressure coefficient with branch flow splits in Figure 6.2.2-1, indicates that the prediffuser/combustor aerodynamic behavior was stable or flowing full throughout the branch flow split range of interest.

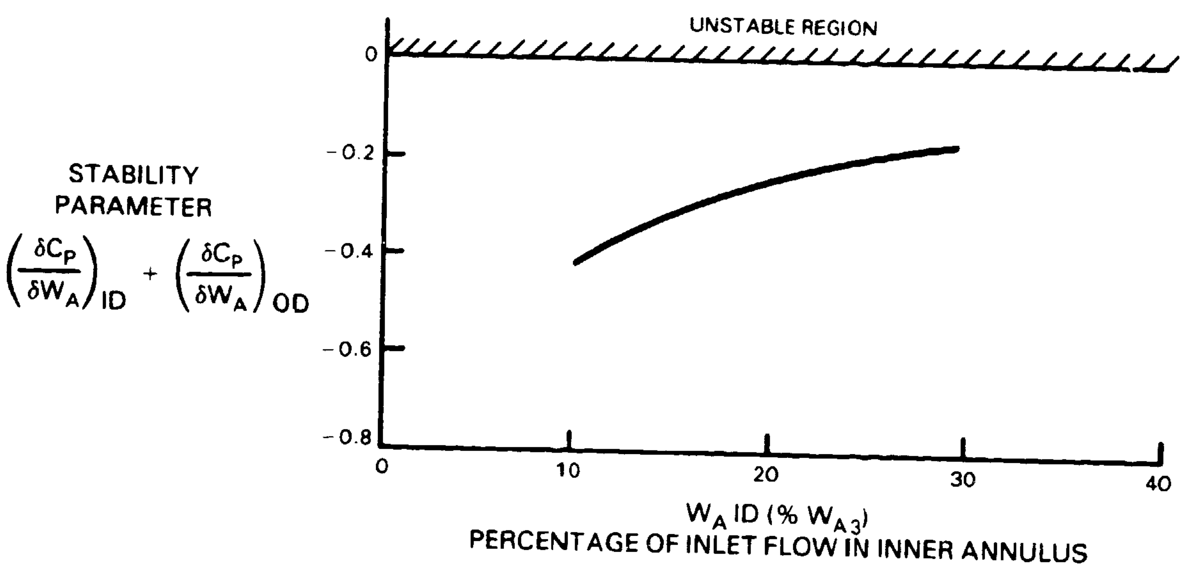

Figure 6.2.2-1 Prediffuser Stability Characteristics as a Function of Downstream Airflow Splits

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Tests to assess the impact of increasing the dump gap indicated that the effect on prediffuser pressure recovery was small for the relatively large baseline nondimensional dump gap of 2.85 (gap/prediffuser inlet radial height). Increases in the measured total pressure loss, as a function of change in nondimensional dump gap, are shown in Figure 6.2.2-2. The results indicate that the increase in pressure loss should be less than 0.2 percent based on design gap tolerances. Radial movement of the hood by 6 percent ΔR inward and 24 percent ΔR outward relative to the baseline position had a negligible effect on performance parameters. Table 6.2.2-II shows the low performance sensitivity characteristics of the prediffuser/combustor system to variations in inlet air profile.

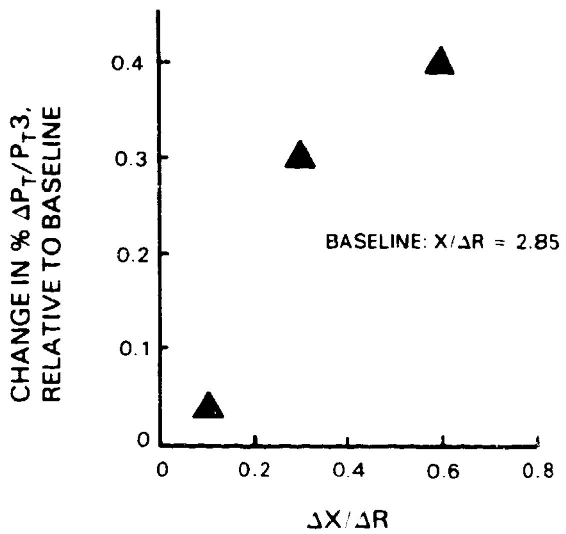

Figure 6.2.2-2 Change in Diffuser Total Pressure Loss as a Function of Increased Dump Gap

TABLE 6.2.2-II
EFFECT OF PREDIFFUSER INLET PROFILE ON PERFORMANCE

Profile	Press. Recov. (Cp) Inner	Press. Recov. (Cp) Outer	Press. Loss (%) Inner	Press. Loss (%) Outer	Flow Split (% Wa) Inner	Flow Split (% Wa) Outer
Outer Peaked	0.45	0.34	2.2	2.6	20.7	60.6
Center Peaked	0.46	0.35	2.3	2.8	21.2	59.6
Inner Peaked	0.47	0.35	2.3	2.7	22.8	58.5

The turbine cooling bleed (tangential on-board injection) at the inlet to the compressor exit guide vanes and extraction through the outer wall in the dump region (customer bleed) did not significantly affect performance, as indicated in Table 6.2.2-III. Customer bleed flow of 9.3 percent of the total high-pressure compressor flow was regarded as an average installation requirement. For the customer bleed case, the major change in flow occurred in the outer annulus. The relative flow distribution in the three branches downstream of the dump plane (customer\ bleed is similar, both with and without bleed extraction, indicating a minimum impact on the pilot zone combustion process.

TABLE 6.2.2-III
EFFECT OF BLEED AIR ON PERFORMANCE

Turbine Cooling Air Bleed - Configuration I

	Cp 1-2 ID	Cp 1-2 OD
No Bleed	0.37	0.35
3.5% Wa	0.37	0.36

Customer Bleed - Configuration II

	Cp 1-2		$\Delta P_t/P_{t1}$ Baseline Struts		Flow Splits (%Wa)		
	ID	OD	ID	OD	ID	Hood	OD
No Bleed	0.52	0.36	2.2	2.6	20.7	18.7	60.6
9.3% Wa	0.52	0.36	2.2	2.6	18.7	17.1	54.9

These tests demonstrated that the Energy Efficient Engine diffuser/combustor design is insensitive to inner and outer diffuser case annuli flow splits, geometric tolerances, inlet pressure profile shape, and air bleed variations.

Testing of the final design configuration with the modified diffuser case struts demonstrated shroud losses below the 3 percent goal and no significant wakes downstream in the inner shroud.

6.3 SECTOR COMBUSTOR RIG TEST PROGRAM

The overall objective of the Sector Combustor Rig Test Program was to evaluate the emissions, performance and structural integrity, of the two-zone combustor configuration over the Energy Efficient Engine flight spectrum. A portion of the test also included an evaluation of relight characteristics at simulated altitude conditions. The program was organized into three phases of testing. The first consisted of fuel injector characterization tests, which were performed both prior to and during rig testing. The second phase involved a series of performance and emissions optimization tests using a sector rig with a conventional louvered liner. The third phase was directed towards evaluating the advanced segmented liner at full engine sea level takeoff pressure and temperature levels. Approximately 500 hours of rig operation were accumulated and approximately 1000 performance/emissions data points were acquired.

6.3.1 Fuel Injector Characterization Tests

6.3.1.1 Overview

Airflow characterization and fuel spray droplet size tests were conducted with both pilot and main zone fuel injection systems. This effort was aimed at assessing initial configurations as well as evaluating and substantiating design refinements.

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6.3.1.2 Program Results

The fuel injectors in the pilot zone are designed for airblast fuel atomization in which the fuel is filmed between two swirling air streams before being atomized. Two types of designs were evaluated for airflow capacity and spray quality. The results are summarized in Table 6.3.1-I.

Both injectors exhibited acceptable droplet size and airflow capacity. The swirl strength of injector B, however, was judged to be too low for good combustion stability. This result was corroborated during subsequent sector rig testing.

TABLE 6.3.1-I
PILOT ZONE FUEL INJECTOR TEST RESULTS

Injectors	AC_D, cm^2 (in^2) Inner	Outer	Inlet Swirl Strength (Torque/Thrust, cm (in)	Droplet Size (Microns) Idle	Starting
A	0.20 (0.08)	0.50 (0.20)	1.5 (0.6)	22	110
B	0.07 (0.03)	0.58 (0.23)	0.7 (0.3)	32	90

Similar tests were conducted with the main zone carburetor tube injector. The effect of main zone carburetor tube core flow on droplet size is shown in Figure 6.3.1-1. As indicated, the vane height of the radial inflow swirler does not affect droplet size if the core airflow velocity and swirl strength are the same. Also, droplet size decreases with increasing core flow velocity and decreasing fuel flow rate.

The impact of secondary to core airflow split on droplet size is shown in Figure 6.3.1-2 for three different core exit velocities. Trends show that droplet size is substantially reduced by increasing the ratio of secondary to total airflow up to approximately 40 percent. At levels greater than 40 percent, there is a negligible decrease in the droplet size. On the basis of these results, the optimum droplet size was established as 35 to 40 microns at the design core airflow exit velocity (53-60 m/sec (175-200 ft/sec)) at sea level takeoff conditions.

6.3.2 Sector Combustor Rig Testing

6.3.2.1 Overview

Emissions and performance optimization tests were conducted with a 90-degree sector rig, which duplicated the aerodynamic features of the Energy Efficient Engine combustor component. During initial testing, a combustor with a conventional louvered liner configuration was used in order to facilitate the installation of design refinements. The rig contained sufficient instrumentation to measure key aerodynamic parameters and liner temperatures. In addition, instrumented vanes were installed at the combustor exit plane to acquire exit temperature, pressure and emissions data.

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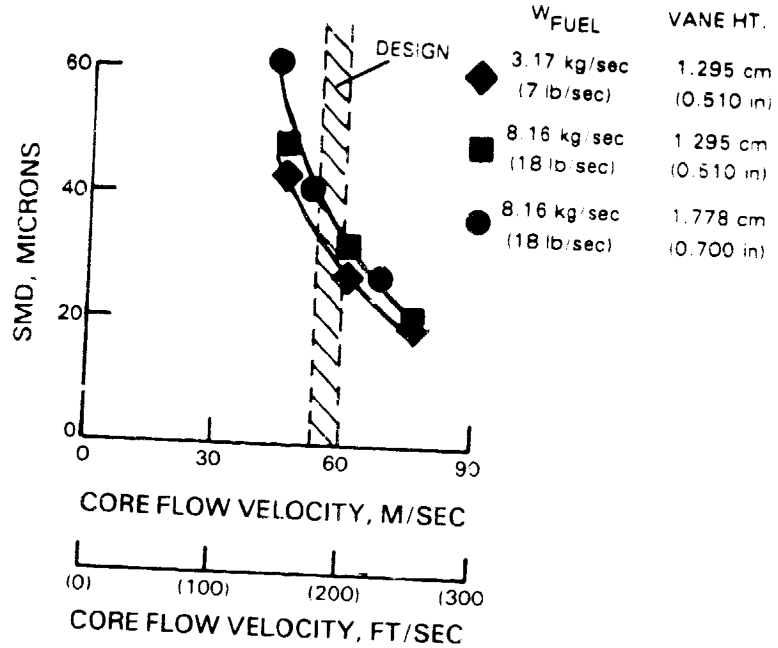

Figure 6.3.1-1 Effect of Tube Core Flow on Droplet Size

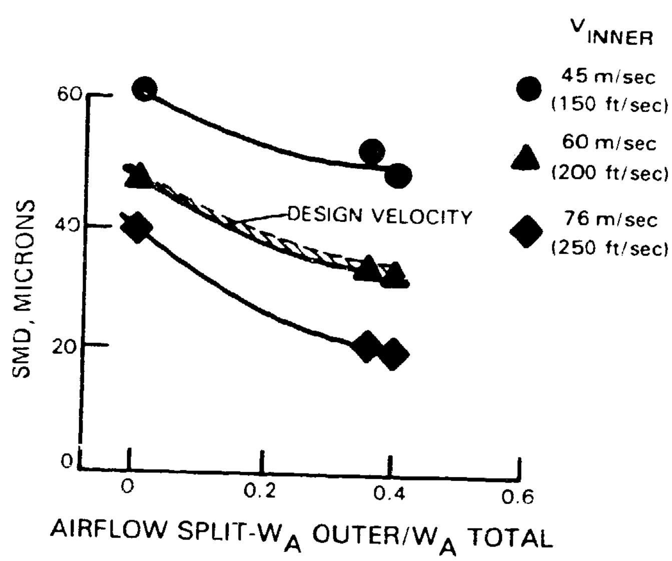

Figure 6.3.1-2 Effect of Secondary (Outer) Airflow on Droplet Size

6.3.2.2 Combustor Emissions and Performance Characterization Tests

Initial tests with the louvered liner sector rig concentrated on reducing low power emissions. Certain pilot zone modifications were evaluated to determine the effect on reducing carbon monoxide and unburned hydrocarbons at idle conditions. These modifications included changes to the airflow schedule between the pilot fuel injector inner/outer airblast passages and fuel injector design. Target idle carbon monoxide levels were established for meeting the Environmental Protection Agency Parameter goals.

Subsequent tests were directed toward optimizing the main zone for efficient fuel staging at the approach condition and reducing oxides of nitrogen at high power operation. These modifications consisted of changes to the carburetor tube design and airflow schedules.

Pilot Zone Optimization

The pilot zone airflow schedule for high combustion efficiency at idle conditions was determined in the early phase of testing. Three configurations, modified for successive increases in zone equivalence ratio, were tested to define emissions and aerothermal performance. As indicated in Figure 6.3.2-1, significant reductions in carbon monoxide and unburned hydrocarbon emissions were achieved. Zone equivalence ratios were increased by eliminating all pilot dilution air, reducing fuel injector airflow with blockage rings and redesigning the fuel injector floatation arrangement to eliminate extraneous air from entering through the expansion slots. Results showed that a pilot zone equivalence ratio near or at the 1.1 value provides effective emissions control at idle.

A series of tests was completed to optimize the pilot injector design, specifically the airflow split between the inner and outer airblast passages. Blockage rings were employed to vary the airflow rates. The different area variations, along with the corresponding results, are presented in Figure 6.3.2-2. These results show that an appreciable reduction in idle emissions was attained by decreasing the inner flow passage area.

Information from these tests provided the guidance to reconfigure pilot injector B for optimum swirl strength and passage flow area. The defined swirl strength was 1.2 cm (0.5 in). Passage area was 0.19 and 1.07 cm^2 (0.030 and 0.167 in^2), respectively, for the inner and outer passages. Results acquired with the redesigned injector B, which produced the lowest overall emissions at idle, are also presented in Figure 6.3.2-2.

Main Zone Optimization

Several carburetor tube design variations were tested to determine the effect on oxides of nitrogen and smoke reduction as well as combustion performance. Based on the results obtained from the preceding fuel injector characterization tests, two approaches were evaluated for improving tube performance and combustor emissions. The first approach focused on increasing core airflow to enhance fuel/air preparation and lower the overall zone equivalence ratio. Core flow was increased by installing longer length radial inflow swirlers. This modification produced reductions in oxides of nitrogen of 20 percent at climb, as shown in Figure 6.3.2-3.

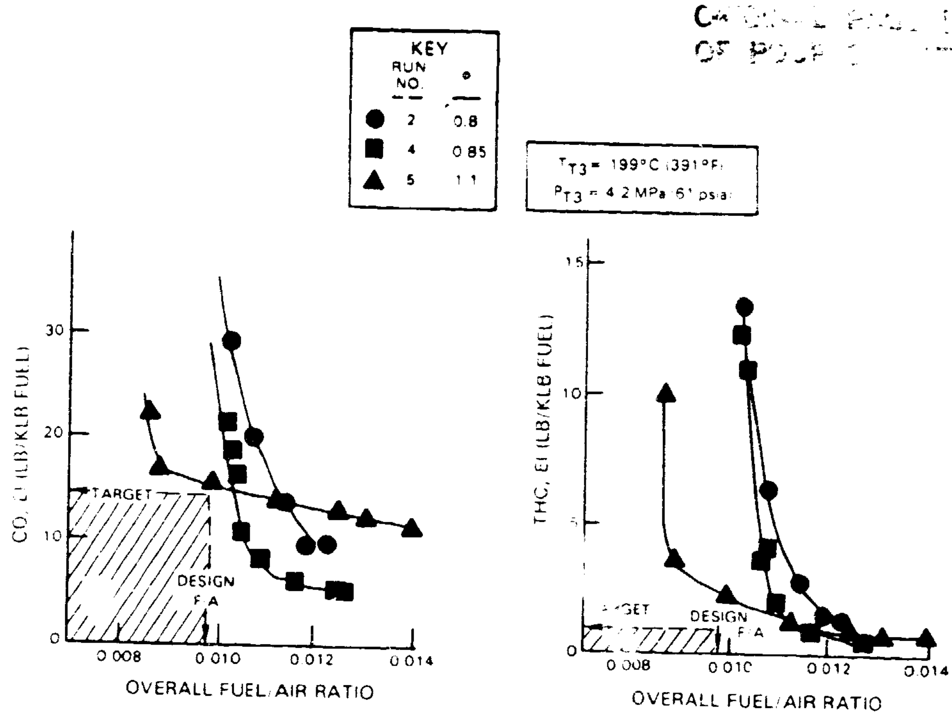

Figure 6.3.2-1 Comparison of Idle Emissions as a Function of Fuel/Air Ratio for Various Peak Equivalence Ratios (Ø)

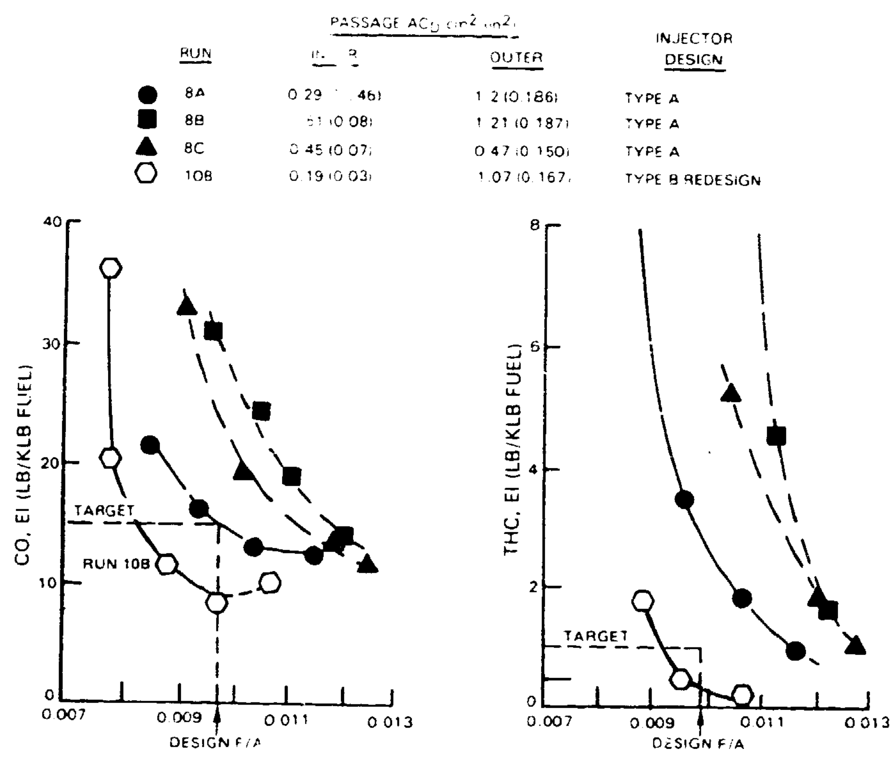

Figure 6.3.2-2 Comparison of Idle Emissions for Several Pilot Fuel Injector Configurations

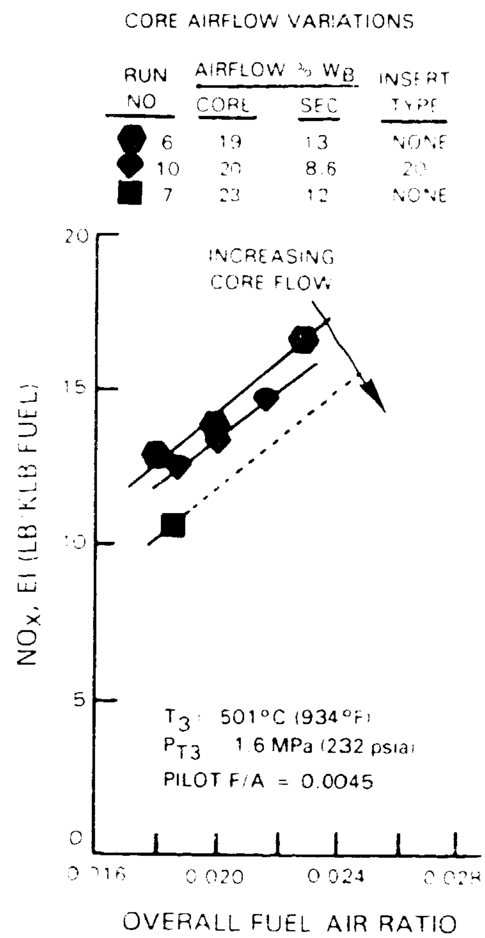

Figure 6.3.2-3 Impact of Increasing Carburetor Tube Core Airflow on High Power Oxides of Nitrogen Emissions

With the second approach, swirl was introduced in the secondary air passage to provide more rapid mixing between secondary and core airflows. The finger seals in the base design (Figure 6.3.2-4) were replaced with swirl generating inserts, angled in the direction of flow (co-rotating with the core flow). Insert angles of 20 and 35 degrees were evaluated. The inserts also produced a reduction in the secondary passage effective area because of the high degree of blockage associated with turning the flow. The 20-degree swirler, which necessitated a large number of individual inserts relative to the 35-degree configuration, produced the greatest degree of blockage.

The effects of swirl on high power oxides of nitrogen are shown in Figure 6.3.2-5. Installation of the 20-degree inserts in the secondary air passage contributed to a significant reduction in oxides of nitrogen with no adverse impact on either carbon monoxide or hydrocarbon emissions. However, increasing the angle to 35 degrees failed to produce any additional reduction to the oxide of nitrogen level. Results showed that the oxides of nitrogen level increased because of the reduced core flow. The modification also had a negative effect on carbon monoxide and hydrocarbon levels at the approach condition. On the basis of these results, the 20-degree insert was selected for the final design.

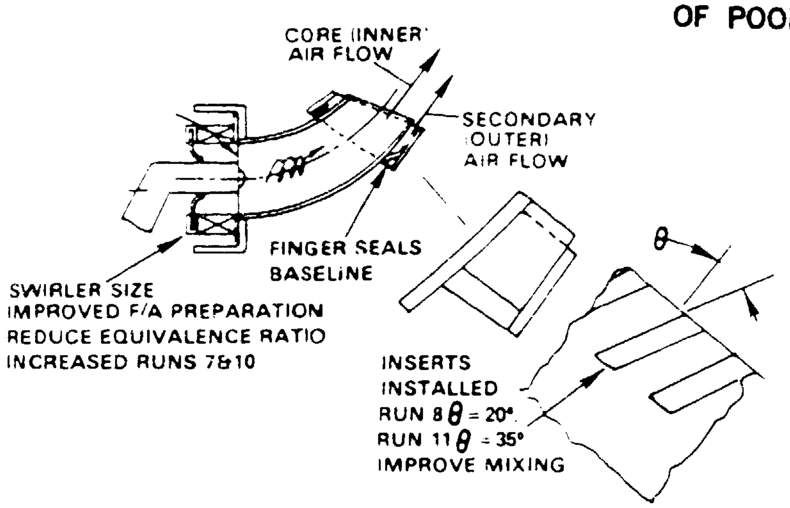

Figure 6.3.2-4 Main Zone Fuel Injector Assembly Showing Modified Axial and Secondary Airflow Sleeve (Swirlers)

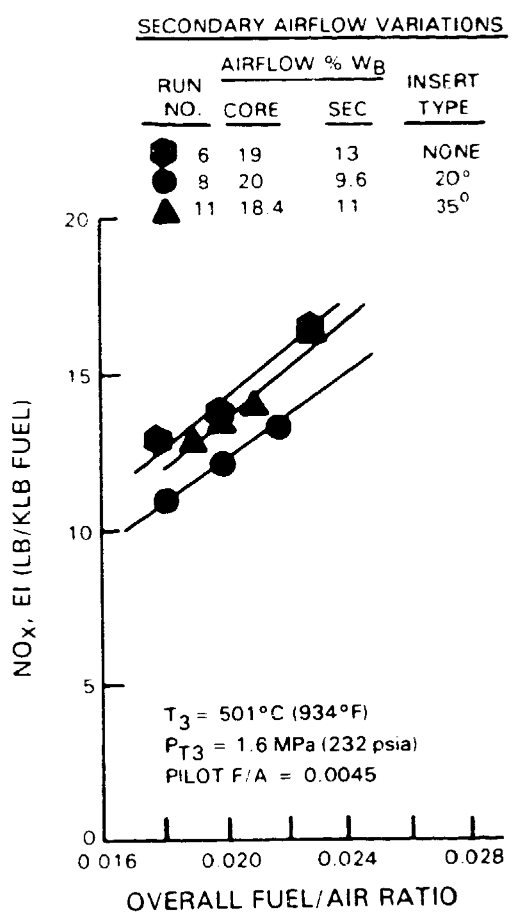

Figure 6.3.2-5 Effect of Carburetor Tube Secondary Passage Swirl on High Power Oxides of Nitrogen Emissions

Altitude Relight and Sea Level Starting Characteristics

Altitude relight and sea level start goals were surpassed with the optimized combustor configuration. Testing was conducted with combustor inlet conditions representative of compressor windmilling over the Energy Efficient Engine flight envelope. Fuel flow was varied at each condition to ascertain the minimum required flow for ignition. The results of testing are summarized in Figure 6.3.2-6. As indicated, ignition was achieved over the range of conditions in the flight envelope. Moreover, ignition was demonstrated at an altitude of 10,668 m (35,000 ft) with fuel flows as low as 21 kg/hr (48 lb/hr), which surpassed the goal.

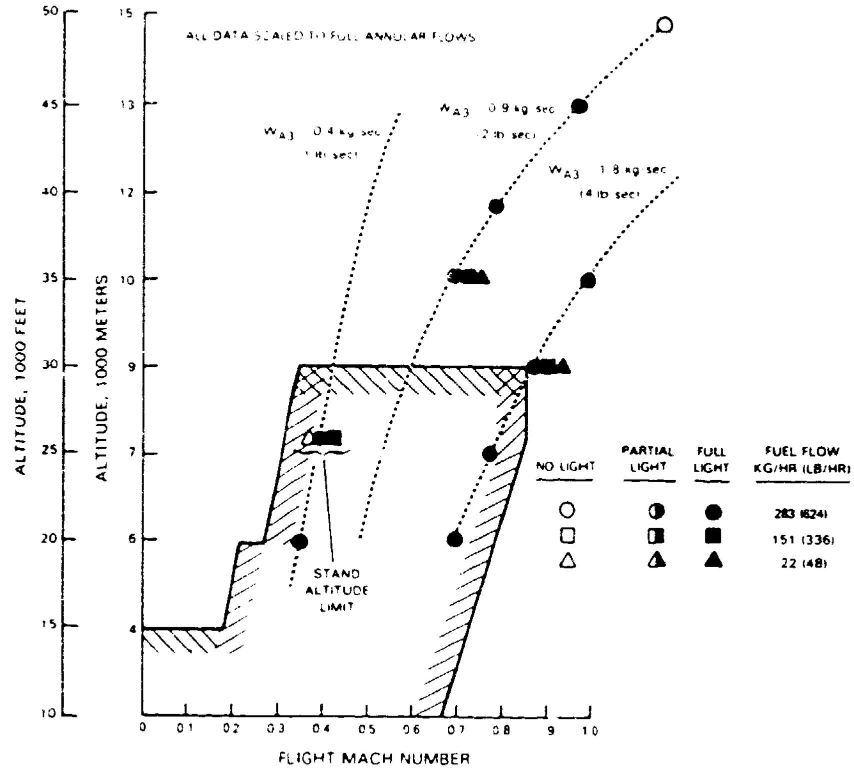

Figure 6.3.2-6 Altitude Relight Test Results

Exit Temperature Distribution

Pattern factor was measured at high power conditions, usually with the pilot zone fuel/air ratio of 0.003. This level corresponds to the optimum fuel split for low emissions at high power conditions. Pattern factors generally ranged between 0.15 and 0.25. The exit radial temperature profile for the final sector rig configuration is presented in Figure 6.3.2-7. These results show that the goal profile was achieved.

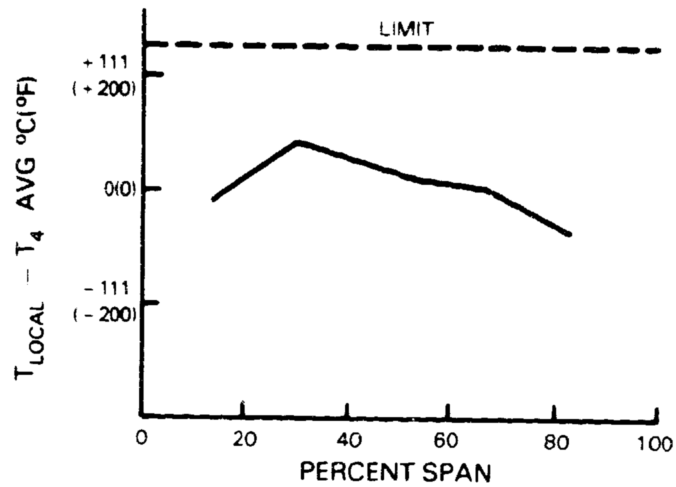

Figure 6.3.2-7 Combustor Exit Radial Temperature Profile

6.3.2.3 Advanced Segmented Liner Tests

Three combustor configurations having the advanced segmented liners were evaluated during this phase of the Sector Combustor Rig Test Program. The airflow schedule and design refinements evolved from testing with the conventional louvered liners were incorporated in the advanced liner configurations.

Testing was performed over the full range of operating conditions for the Energy Efficient Engine. As shown in Figure 6.3.2-8, emissions characteristics remained essentially the same as those demonstrated during the preceding tests with the louvered liner. Except for oxides of nitrogen, all emissions goals as well as smoke were achieved. It should be emphasized that although differences existed in the cooling characteristics of the advanced segmented liner and conventional louvered liner designs, specifically in the cooling air boundary layers, comparable emissions and performance were achieved with both designs. This demonstrates the feasibility of using louvered liners to reduce program costs for developing advanced combustion systems. A summary of combustor aerothermal and emissions performance is presented in Table 6.3.2-I.

The structural integrity of the segmented liner configuration was demonstrated during these series of tests. The typical post-test condition of the liners is shown in Figure 6.3.2-9. The liners are essentially in excellent condition with no indications of thermal distress such as cracking or buckling. An analysis of thermal sensitive paint results showed temperature levels and patterns generally within the established limits. Some evidence of streaking, however, was observed on the inner rear segment, in which localized temperatures exceeded the 1037°C (1900°F) limit.

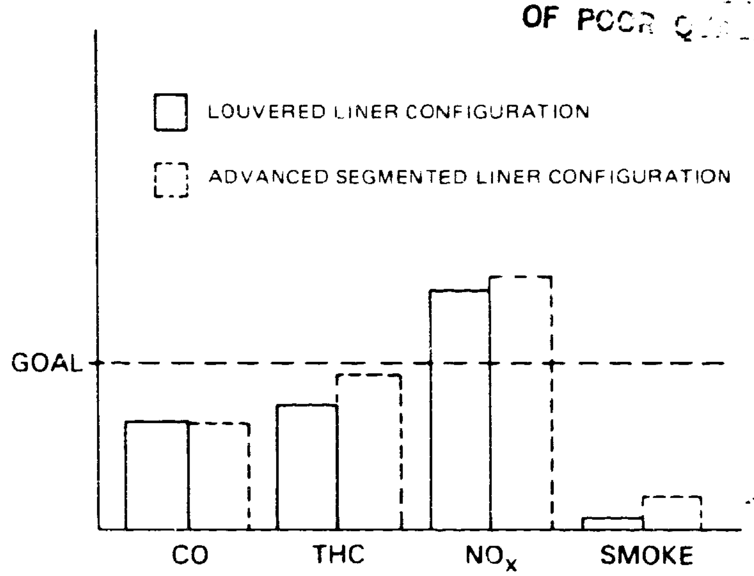

Figure 6.3.2-8 Comparison of Environmental Protection Agency Parameters for Emission and Smoke Number for Conventional Louvered and Advanced Liner Sector Combustor Configurations

TABLE 6.3.2-I
SUMMARY OF ADVANCED TWO-ZONE SECTOR COMBUSTOR PERFORMANCE AND EMISSIONS CHARACTERISTICS

Aerothermal Performance	
o Total Pressure Loss	5.22-percent P_{t3}
o Pattern Factor	0.26
o Radial Temp. Profile	65°C (150°F) Peak-to-Average
o Relight Capability	10,668 m (35,000 ft) Altitude
o Sea Level Start	Acceptable
o Lean Blow Out	Acceptable
Emissions	
o Carbon Monoxide	2.30*
o Unburned Hydrocarbons	0.38*
o Oxides of Nitrogen	4.7*
o Smoke Number	4

* Environmental Protection Agency Parameter; with Margin added for Development and Varability.

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Figure 6.3.2-9 Post-Test Condition of Segmented Liner After 15.5 Hot Hours of Testing

SECTION 7.0
FULL ANNULAR COMBUSTOR COMPONENT TEST RIG AND PROGRAM

7.1 OVERVIEW

The objectives of the Full Annular Combustor Component Test Program are to: (1) demonstrate that the two-zone combustor designed for the Energy Efficient Engine meets the aerothermal and environmental goals, and (2) identify areas where refinements could be made to meet the combustor requirements for a future flight propulsion system. To accomplish these objectives, the program is organized into an integrated sequence of full annular combustor rig tests and sector combustor rig tests. Full annular combustor rig testing will concentrate mainly on aerothermal performance optimization, such as radial and circumferential temperature profiles. Sector rig testing will serve as a development tool to provide supporting diagnostic data relating to liner durability and aerothermal performance to guide the full annular rig effort.

The program schedule is presented in Figure 7.1-1. The entire program consists of a planned 15-month effort. An estimated 235 total hours of testing (133 hours of hot time) will be accomplished during the eight full annular and seven sector rig tests. Also, approximately 250 performance/emissions data points will be acquired during the program.

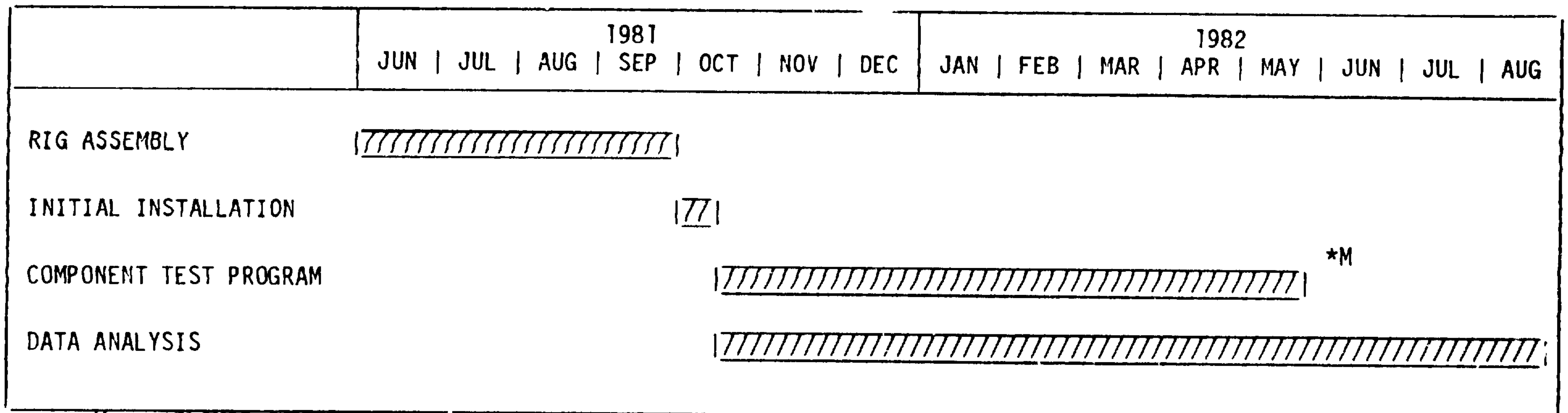
7.2 RIG DESIGN

7.2.1 Full Annular Combustor Test Rig

A cross sectional view of the full annular combustor test rig is presented in Figure 7.2.1-1. Basically, the rig incorporates the Energy Efficient Engine diffuser and combustor sections, along with the appropriate mounting and adapting hardware for installation into the Pratt & Whitney Aircraft High-Pressure Combustion Laboratory (Stand X-960) located in Middletown, Connecticut. Details of the component test section are shown in Figure 7.2.1-2.

For test flexibility, the rig inlet section contains a removable profile generating duct. This permits testing with intentional flow profile variations to simulate various compressor discharge profiles, while the rig is installed in the facility. In addition, simulated tangential on-board injection (TOBI) bleed at the prediffuser inlet, customer bleed, and inner/outer annuli turbine cooling air features have been incorporated for a realistic representation of bleed effects. Combustor exit gas temperatures, pressures and emissions samples will be acquired with a sophisticated multiprobe traversing system.

The full annular combustor rig utilizes component hardware in the diffuser and combustor sections. Following completion of testing, the diffuser and combustor hardware shown in Figure 7.2.1-3 will be removed from the test rig and incorporated into the integrated core/low spool.



*M - DELIVER COMBUSTOR TO IC/LS

Figure 7.1-1 Full Annular Combustor Component Rig Test Program Schedule

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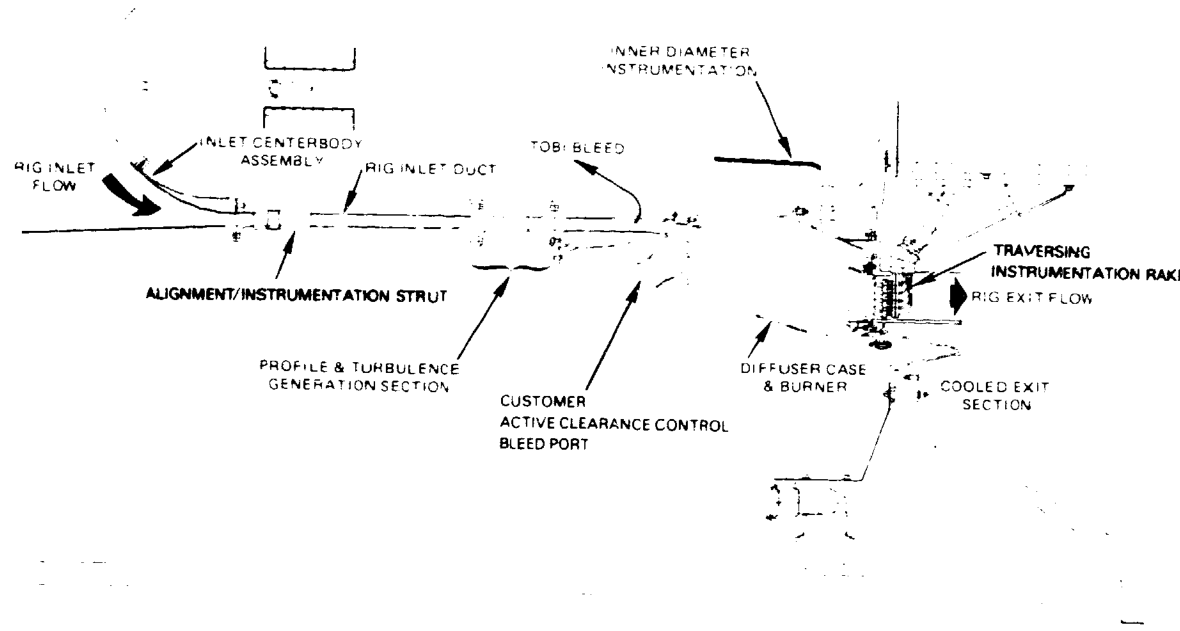

Figure 7.2.1-1 Full Annular Combustor Component Rig

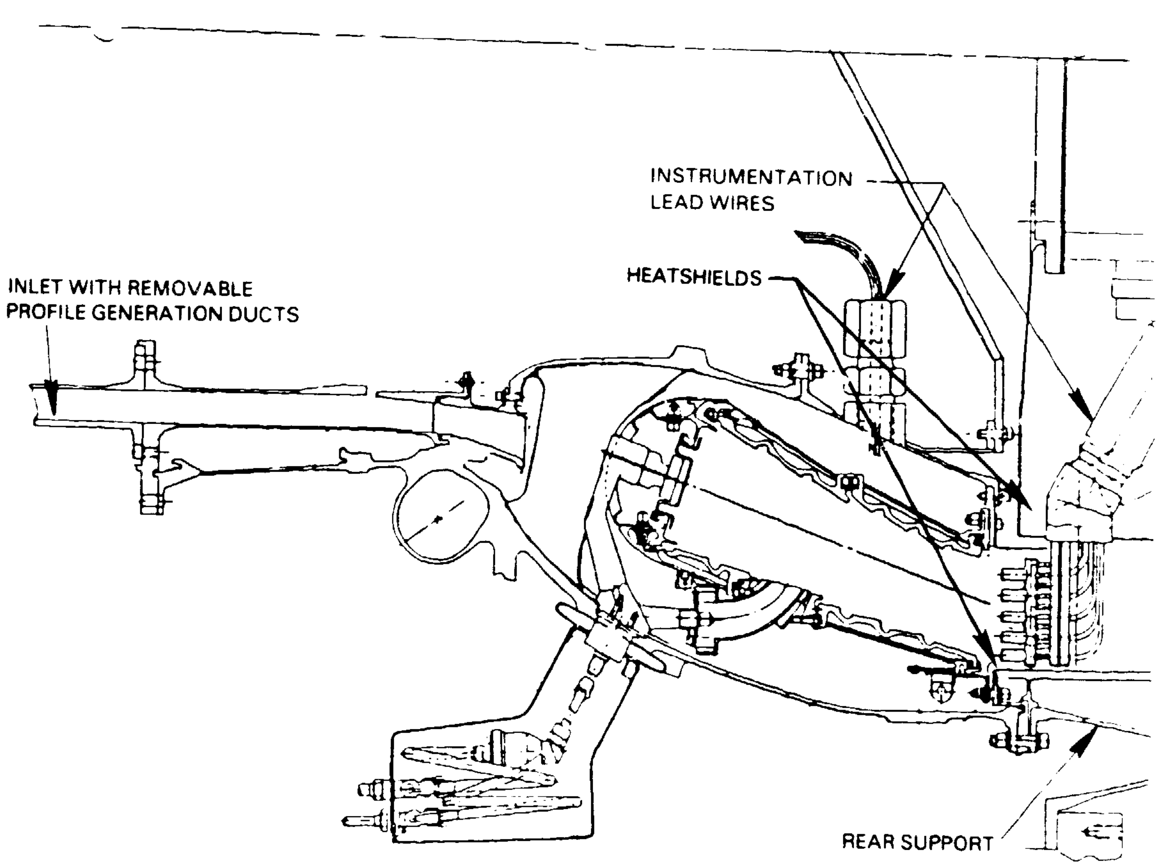

Figure 7.2.1-2 Test Section of Full Annular Combustor Component Rig

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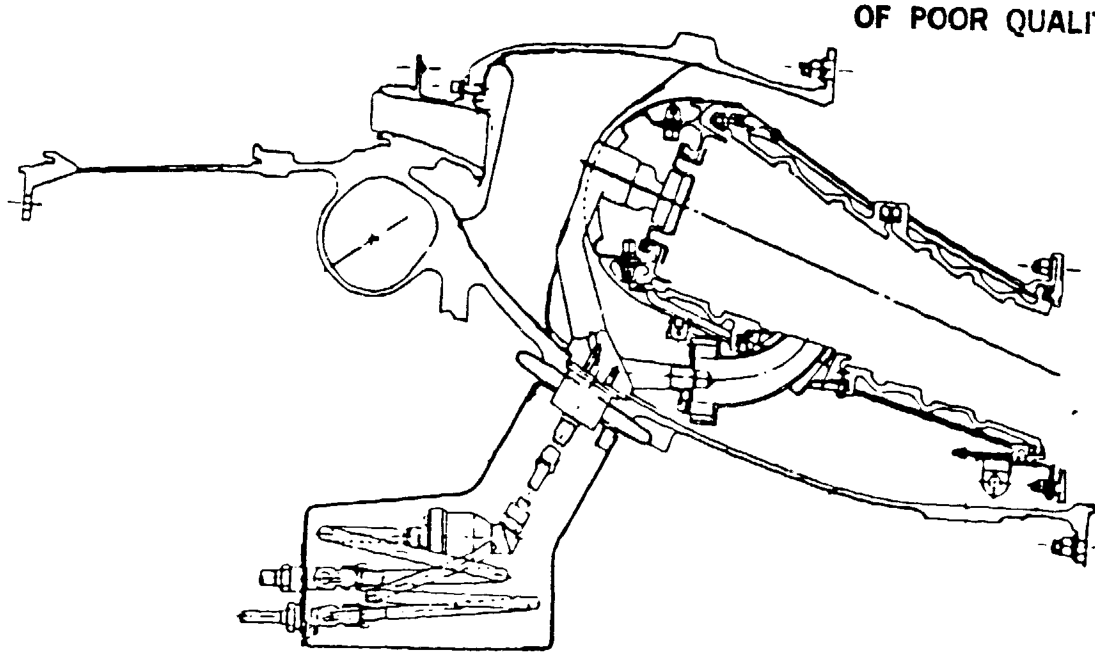

Figure 7.2.1-3 Rig Diffuser and Combustor Section Hardware Utilized in Integrated Core/Low Spool Experimental Engine

7.2.2 Sector Combustor Rig

The sector combustor test rig, as shown in Figure 7.2.2-1, is the same as that utilized in the Sector Combustor Rig Test Program. The rig is a 90-degree sector of the Energy Efficient Engine combustor. The modular design enables expeditious installation of configurational changes to reduce cost and down time.

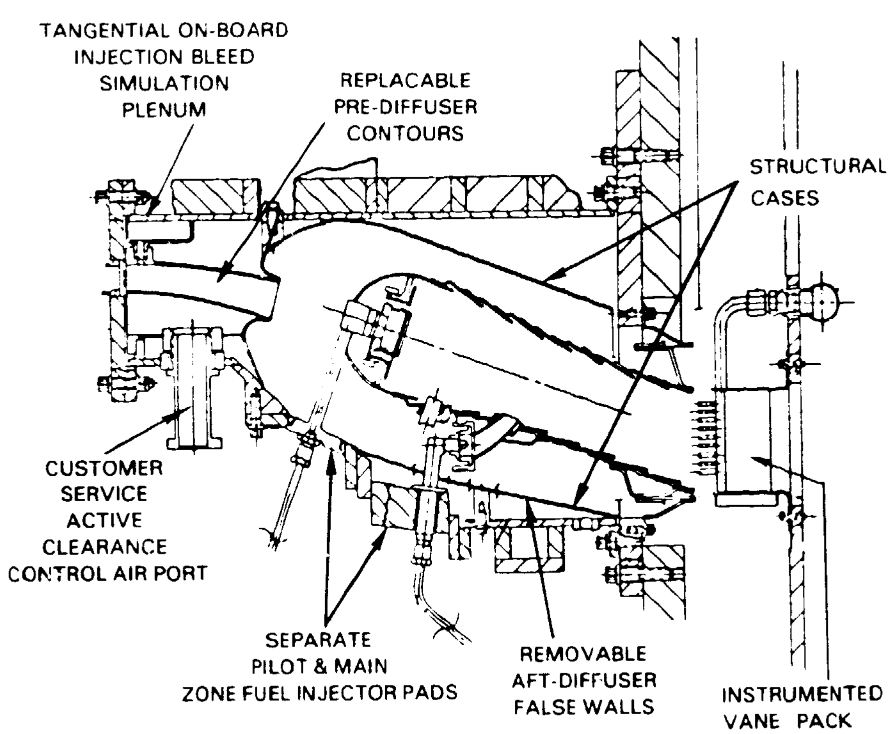

Figure 7.2.2-1 Cross Section of Sector Combustor Rig with Conventional Louvered Liner Configuration

The combustor section is encased by a removable diffuser wall. In addition, the pilot and main fuel injectors are installed separately for added test flexibility. At the exit plane, instrumented vane packs are used to acquire pressure, temperature and emissions data.

For this test program, two sector rig combustors will be available. One utilizes a conventional sheetmetal louvered liner to permit easy modification in order to evaluate a variety of cooling/dilution flow schedules. The second uses the advanced segmented liner for testing at the sea level takeoff pressure and temperature conditions expected for the Energy Efficient Engine combustor.

7.3 TEST INSTRUMENTATION

Both the full annular and the sector combustor test rigs will incorporate a variety of instrumentation to monitor operating conditions such as inlet flows, temperature and pressures as well as to record combustor performance, emissions and structural characteristics. Imbedded thermocouples will be installed in the liners to record surface metal temperatures. Instrumentation requirements, including the quantity, location and type of sensor, have been determined on the basis of analyses, previous sector rig testing and Pratt & Whitney Aircraft's experience in the development of advanced combustion systems.

7.3.1 Full Annular Combustor Rig Instrumentation

The full annular test rig is equipped with a complete complement of pressure, temperature and gas sampling sensors to provide a thorough documentation of performance and emissions. Table 7.3.1-I lists the location, type and quantity of instrumentation used to monitor performance and Figure 7.3.1-1 presents the instrumentation map. As indicated, the combustor inlet and exit locations are extensively instrumented.

Combustor exit radial and circumferential temperature distributions and exhaust gas samples will be obtained with a traversing multiprobe system. It consists of seven multi-element probes. Five of the probes are used for measuring total pressure and acquiring gas samples for a determination of total unburned hydrocarbons, carbon monoxide, oxides of nitrogen, and smoke levels. Each of the five probes has four radial sampling elements. The remaining two probes are used for measuring total temperature. These probes have five radial sensing elements and are positioned at the combustor exit plane in the same axial location as the first stage turbine vane. They are cooled with supplementary facility air and the gas sample lines are water cooled. The gas sampling lines are manifolded so that emissions are averaged.

The data acquisition system includes a combination of scanivalves and individual pressure pickups. Steady state pressures are recorded in the normal mode by transducers mounted in the scanivalves. Airflow is measured by two force balance transducers. A precision pressure gauge is used to measure the reference barometric pressure. Based on previous applications, a pressure measurement uncertainty of less than ± 0.10 percent of transducer full scale pressure can be expected for steady state data. To maintain pressure accuracy, a primary calibration of the high accuracy steady state transducers is performed just prior to starting the test. This calibration is performed using dead weight testers that are calibrated against National Bureau of Standards' traceable

TABLE 7.3.1-I
FULL ANNULAR COMBUSTOR RIG INSTRUMENTATION LIST

LOCATION	MEASUREMENT/TYPE	QUANTITY	PURPOSE
INLET	5 ELEMENT TOTAL PRESSURE PROBE 5 ELEMENT TOTAL TEMPERATURE PROBE INNER/OUTER WALL STATIC PRESSURE TAPS	6 6 4/4	o RIG INLET TOTAL PRESSURE AND TEMPERATURE
OUTER ANNULUS	2 ROWS WALL STATIC PRESSURED TAPS HYDROCARBON SNIFFER	11 & 14 each row 5	o LINER FEED PRESSURE MAP o SAFETY
INNER ANNULUS	2 ROWS WALL STATIC PRESSURED TAPS	14 each row	o LINER FEED PRESSURE MAP
COMBUSTOR HOOD PILOT INJECTOR INNER PASSAGE	STATIC PRESSURE TAPS TOTAL PRESSURE KIELHEAD	9 7	o BULKHEAD FEED PRESSURE o INJECTOR FEED PRESSURE
COMBUSTOR OUTER AND INNER DIAMETER LINERS	IMBEDDED THERMOCOUPLE TEMPERATURE SENSITIVE PAINT	102	o LINER TEMPERATURE
EXIT	TRAVERSE RAKE 5 TOTAL PRESSURE/SAMPLE HEADS TRAVERSE RAKE 2 TOTAL TEMPERATURE HEADS	4 TOTAL PRESSURE /GAS SAMPLE PORTS 5 TOTAL PRESSURE /PORTS	EXIT MEASUREMENTS

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CODE

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POS 4°	10 285	11 292	12 296	13 300	14 307	15 310	16 315	17 322	

℄ LOOK UPSTREAM CLOCKWISE
TDC 0°

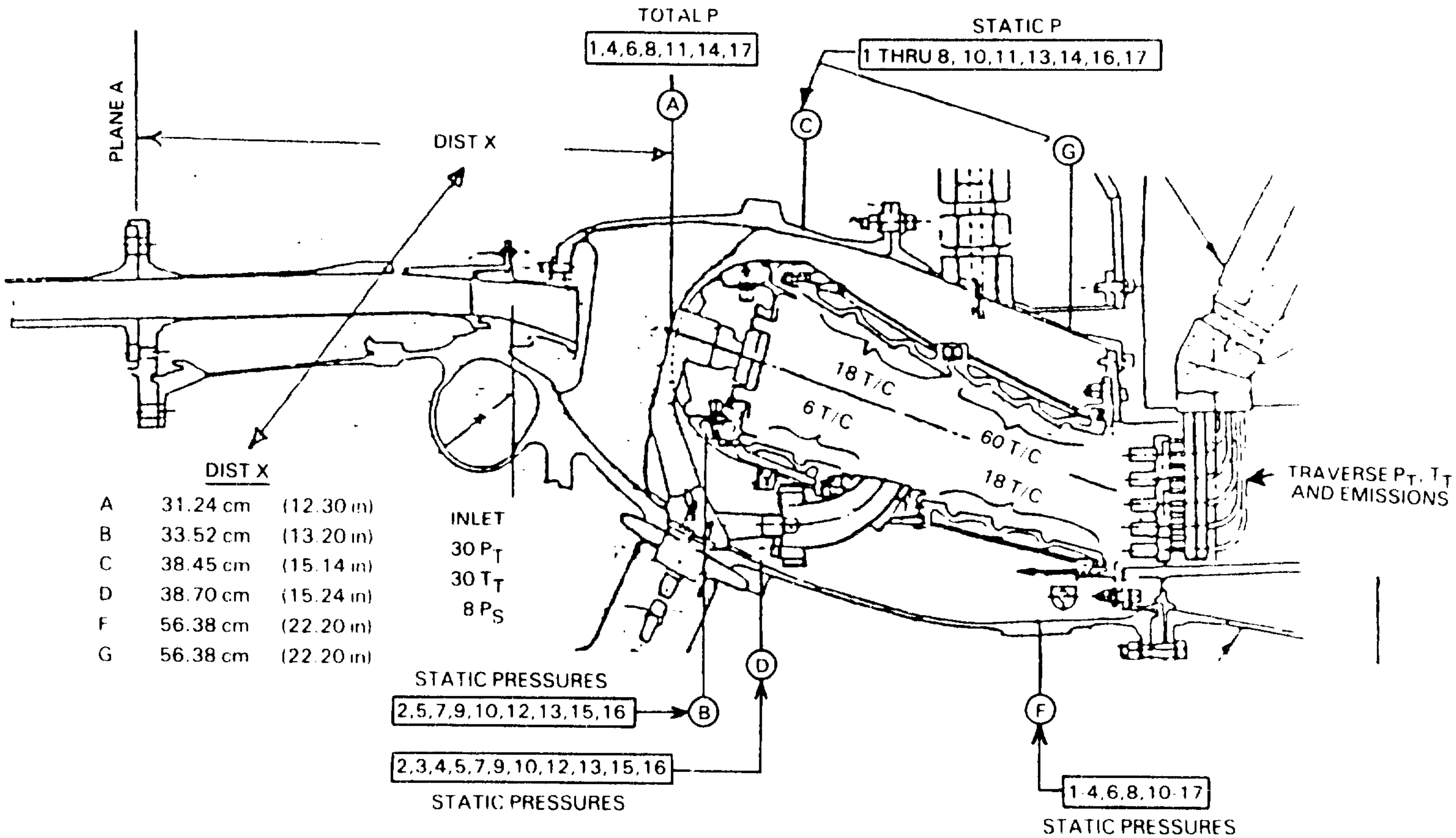

Figure 7.3.1-1 Full Annular Combustor Component Rig Instrumentation Map

standards. A secondary calibration, consisting of the application of at least two known pressures to each high accuracy transducer during each data scan, is available for updating the primary calibration curves.

A total of 102 thermocouples will be used for recording liner metal surface temperatures. In addition, thermal sensitive paint will be used on selected panels to show liner thermal gradients. Seventy-eight thermocouples are located on the inner liner and 24 are on the outer liner. The thermocouples are imbedded in the liner to record hot wall surface metal temperatures as well as temperature levels on the cold wall at the liner segment hook attachment area.

Thermocouple instrumentation will provide complete coverage of a 90-degree quadrant of the full annular rig. Individually instrumented 15-degree panels of four liner segments (inner rear, inner front, outer rear, and outer front segments) will be installed for each test. Thermocouple locations for these panels are shown in Figures 7.3.1-2 and 7.3.1-3. The rear inner liner segment has received the most extensive thermocouple coverage. This coverage is required on the basis of results acquired from sector rig testing, which showed the presence of high metal surface temperatures.

Temperatures ranging to 815°C (1500°F) will be measured using calibrated thermocouples connected to reference junctions attached to uniform temperature reference blocks located in the test cells. The temperatures of these reference blocks are monitored relative to highly accurate electronic ice point cells, which are calibrated to National Bureau of Standards' traceable standards. Based on recent experience, a temperature measurement uncertainty of $\pm$0.14°C ($\pm$0.25°F) up to $\pm$0.56°C ($\pm$1°F) can be expected for steady state data. Exit gas temperature measurement uncertainty is $\pm$0.5 percent.

Instrumentation will be installed to ensure the safety of the test vehicle. Thermocouples will be installed at different locations throughout the combustor to measure metal surface temperatures and detect hot spots. Also, a light-off detector provides an indication of ignition as well as detects the occurrence of a lean blow out. Hydrocarbon detectors will be used to monitor the hydrocarbon content in the outer combustor annulus and fuel manifold safety shroud areas.

7.3.2 Sector Combustor Rig Instrumentation

The sector combustor rig will incorporate sufficient instrumentation to evaluate performance, emissions and liner durability characteristics. A list of the sector rig performance instrumentation is presented in Table 7.3.2-I.

Sufficient pressure and temperature probes will be installed in the diffuser and combustion sections for a complete documentation of aerothermal performance. Exit total pressures and temperatures will be recorded by stationary vane packs. Also, exhaust emissions and smoke samples will be obtained at the combustor exit utilizing eight stationary vane packs. The exit probes are air cooled and the gas sample lines are steam cooled. In addition, the gas sampling lines are manifolded so that emissions can be averaged radially or circumferentially. However, it is anticipated that a bulk average sample will be used. Since this system is essentially identical to the exit instrumentation in the full annular rig, the same pressure measurement uncertainty level of $\pm$0.10 percent can be expected and the same calibration procedures will be employed.

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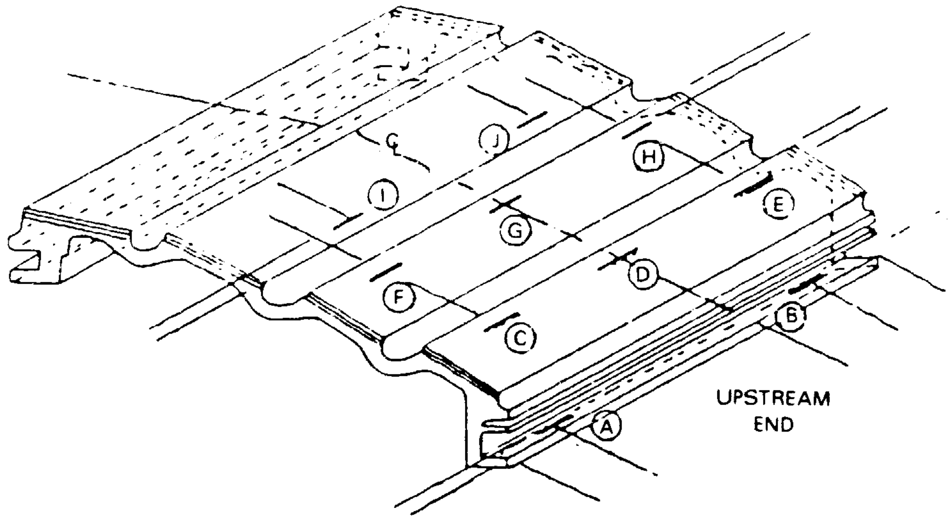
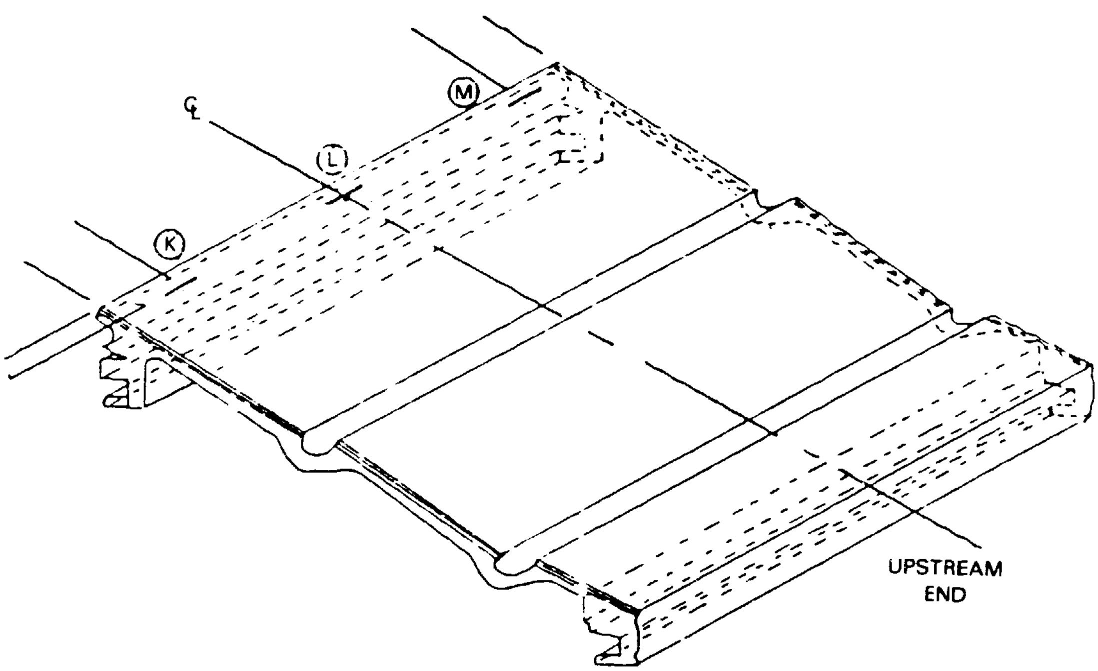


(a) TYPICAL INNER REAR LINER SEGMENT THERMOCOUPLE LOCATIONS



(b) TYPICAL INNER FRONT LINER SEGMENT THERMOCOUPLE LOCATIONS

Figure 7.3.1-2 Full Annular Combustor Component Rig Thermocouple Locations For Inner Liner Segments

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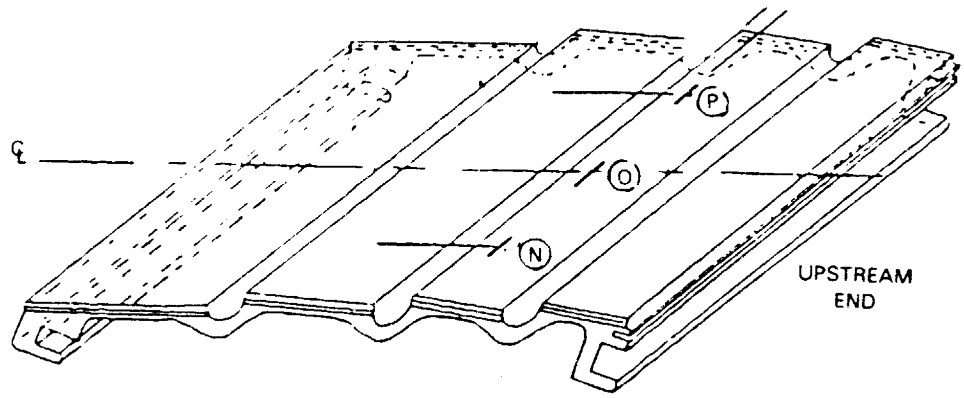
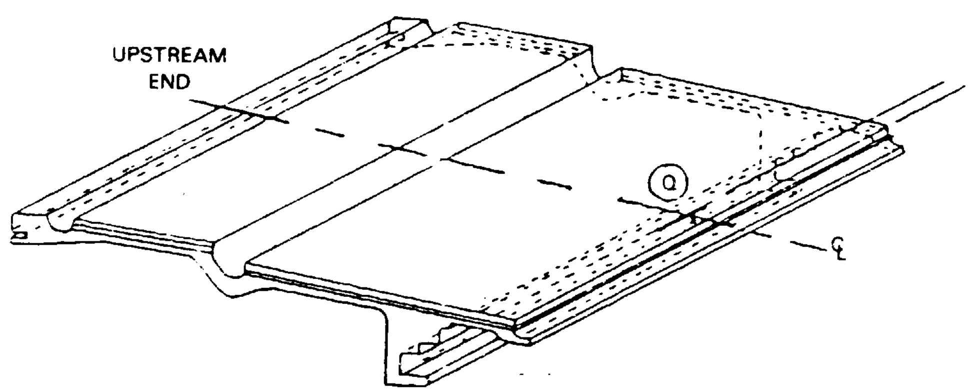


(a) TYPICAL OUTER REAR LINER SEGMENT THERMOCOUPLE LOCATIONS



(b) TYPICAL OUTER FRONT LINER SEGMENT THERMOCOUPLE LOCATIONS

Figure 7.3.1-3 Full Annular Combustor Component Rig Thermocouple Locations For Outer Liner Segments

TABLE 7.3.2-I
SECTOR COMBUSTOR RIG INSTRUMENTATION LIST

LOCATION	MEASUREMENT/TYPE	QUANTITY	PURPOSE
INLET	o 4-ELEMENT TOTAL PRESSURE PROBES o 4-ELEMENT TOTAL TEMPERATURE PROBES o INNER/OUTER WALL STATIC PRESSURE TAPS	6 5 5/5	o RIG INLET TOTAL PRESSURE AND TEMPERATURE PROFILES
PREDIFFUSER	o INNER WALL STATIC PRESSURE TAPS (2 ROWS) o OUTER WALL STATIC PRESSURE TAPS (2 ROWS)	6 6	o PREDIFFUSER PERFORMANCE
DIFFUSER CASE STRUTS	o 5-LEADING EDGE TOTAL PRESSURE	4 STRUTS	o PREDIFFUSER EXIT PROFILE
OUTER ANNULUS	o 2 ROWS-WALL STATIC PRESSURE TAPS o HYDROCARBON DETECTOR	4 EACH ROW 1	o LINER FEED PRESSURE MAP o SAFETY
INNER ANNULUS	o 2 ROWS-WALL STATIC PRESSURE TAPS o HYDROCARBON DETECTOR	6 EACH ROW 1	o LINER FEED PRESSURE MAP o SAFETY
COMBUSTOR HOOD	o STATIC PRESSURE TAPS	2	o BULKHEAD FEED PRESSURE
COMBUSTOR LINERS	IMBEDDED THERMOCOUPLE AND TEMPERATURE PAINT	20	o LINER TEMPERATURE
EXIT	o VANE PACK 5 LEADING EDGE THERMOCOUPLE 4 LEADING EDGE GAS SAMPLING PORTS 5 TOTAL PRESSURE PORTS	 8 VANES 8 VANES 4 VANES	

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To monitor liner overall thermal-mechanical performance, ten imbedded thermocouples will be installed on the inner and outer liners. Also, thermal sensitive paint will be applied to selected panels to show thermal gradients.

The sector rig has the same type of safety instrumentation as the full annular rig. This safety instrumentation includes thermocouples for temperature level monitoring, light-off detectors and hydrocarbon sensors.

7.4 TEST PROGRAM

7.4.1 Full Annular Combustor Rig Test Program

The program planned for the full annular combustor component rig is a four phase effort, consisting of a total of eight separate tests. These tests are outlined in Table 7.4.1-I.

The baseline full annular combustor configuration used in the first phase of testing will be a simulation of the last configuration evaluated during advanced liner evaluation in the previous Sector Combustor Rig Test Program. This will provide a direct measure of combustor behavior and commonality between the full annular and preceding sector rig results, as well as a baseline characterization of performance for subsequent testing. However, the test configuration will not be identical to sector combustor rig hardware. The full annular combustor design is scaled down for a 12 percent smaller core airflow capacity, incorporates cast carburetor tubes, and has smaller diameter pilot fuel injectors.

The second phase is directed towards establishing baseline durability characteristics at conditions simulating sea level takeoff. In the third phase, modifications to improve performance and durability will be evaluated. Potential modifications include the following:

(1) Main zone cast carburetor tube geometry to alter the airflow and/or swirl strength of both the core and secondary (outer) airflow. Changes will be limited to those achievable within current castings.

(2) Variations in the dilution air injection location and amount of air.

(3) Varying pilot fuel injector airflow and/or swirl strength.

The specific test rig configurations for each test will evolve from the results of preceding tests, including the results from the companion sector rig program.

The matrix presented in Table 7.4.1-II outlines the sequence and test conditions for the full annular combustor rig test. All of the performance and emissions testing planned in Phases 1, 3 and 4 will include combustor operation at points 1 through 18. The liner durability evaluation in Phase 2 will consist of test points 1-6 and point 19, which simulate a hot day 29°C (84°F) sea level takeoff condition.

TABLE 7.4.1-I
COMBUSTOR COMPONENT TEST PROGRAM
FULL ANNULAR RIG TESTING

Test Number	Type of Test	Configuration/Variables	Comment
Phase 1			
FA-1	Shakedown and Performance/Emissions Baseline	Simulate Sector Rig Configuration	Commonality with previous advanced liner sector rig test also substantiated.
Phase 2			
FA-2	Liner Temperature (Thermal Paint) and Performance Emissions	FA-1 for 180 Deg Modified Carburetor Tube or Dilution Air 180 Deg	Baseline durability test at simulated sea level takeoff conditions.
Phase 3			
FA-3 through 7	Performance/Emissions	Potential Modifications o Carburetor Tube (Airflow, Swirl) o Dilution Air o Pilot Fuel Injector Including those evolved in sector rig tests	Evolve combustor to meet performance and emissions goals. Monitor liner temperatures.
Phase 4			
FA-8	Performance/Emissions Verification	Integrated Core/Low Spool Combustor	Full emissions/performance documentation at actual and simulated high power conditions.

TABLE 7.4.1-II
FULL ANNULAR COMBUSTOR TEST PROGRAM
TEST MATRIX

Point	Condition	T_{T3} °C (°F)	P_{T3} MPa (psia)	(F/A)	W_{A3} kg/sec (lb/sec)	Pilot/Main Fuel Split	Comments
1	Cold Flow Pressure Loss	148 (300)	0.7 (100)	-	18.7 (41.4)	-	Inlet Mn Variation
2				-	20.8 (46.0)	-	
3				-	22.9 (50.6)	-	
4	Idle	199 (391)	0.43 (63)	0.0090	13.6 (30.0)	100/0	Fuel/Air Variation
5				0.0098			
6				0.0110			
7				Min.			Lean Blowout
8	Approach	348 (659)	1.2 (167.7)	0.015	30.8 (68.0)	100/0	Fuel Split Variation
9						85/15	
10						70/30	
11				0.014			Fuel/Air Variation with Design Split
12				0.016			
13	Climb	501 (935)	1.8 (270*)	0.020	45 (100*)	15/85	Fuel Split Variation
14						30/70	
15				0.023			Fuel/Air Variation with Design Split
16	SLTO	532 (991)	2.1 (300*)	0.021	45 (100*)		Fuel/Air Variation with Design Split
17				0.023			
18				0.025			
19	SLTO,Hot	563 (1055)	2.1 (305*)	0.027	45 (100*)	15/85	Durability test only.

*Corresponds to flow parameters simulation at facility-imposed maximum operating conditions.

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At the end of testing, the combustor will be inspected and thermal paint patterns will be analyzed for a determination of temperature gradients and maximum temperature levels.

7.4.2 Sector Combustor Rig Test Plan

The planned sector combustor rig program will be conducted concurrent with the full annular test effort. This program consists of the three phases shown in Table 7.4.2-I, and the test matrix is presented in Table 7.4.2-II. The matrix is essentially the same as that for the full annular program, except that exit pressures, temperatures and emissions will be measured with a stationary vane pack. Also, the procedures are consistent with those used in the Sector Combustor Rig Test Program.

Visual inspection of the test rig will be made on a regular basis to detect temperature-related problems. These inspections will also include analyses of thermal paint patterns.

TABLE 7.4.2-I
COMBUSTOR COMPONENT TEST PROGRAM
SECTOR RIG TESTING

Test Number	Type of Test	Configuration/Variables	Comment
Phase I			
S-1	Liner Temperature/ Performance/Emissions	Advanced Liner, Simulate Sector and F.A. #1 Cast Carburetor Tubes	Baseline, thermal paint plus commonality assessment.
Phase II			
S-2 thru 6	Performance/Emissions Development	Louvered Liner, Potential Modifications: o Dilution Air o Carburetor Tube	Performance optimization, results to full annular.
Phase III			
S-7	Liner Temperature/ Thermal Paint	Advanced Liner, Integrated Core/ Low Spool Combustor Configuration	Liner temperatures at full sea level takeoff conditions for durability assessment.

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TABLE 7.4.2-II
SECTOR COMBUSTOR RIG TEST PROGRAM
TEST MATRIX

Point	Condition	T_{T3} °C (°F)	P_{T3} MPa (psia)	(F/A)	W_{A3} kg/sec (lb/sec)	Pilot/Main Fuel Split	Comments
1	CFPL	148 (300)	0.7 (100)	-	5.5 (12.3)	-	
2				-	5.9 (13.1)	-	Cold Flow
3				-	6.3 (14.0)	-	Pressure Loss (CFPL)
4	Idle	199 (391)	0.43 (63)	0.0090	3.87 (8.54)	100/0	Fuel/Air Variation
5				0.0098			
6				0.0110			
7				Min.			Lean Blowout
8	Approach	348 (659)	1.2 (167.7) (167.7)	0.015	8.2 (18.2)	100/0	Fuel Split Variation
9						85/15	
10						70/30	
11				0.014			
12				0.016			Fuel/Air Variation at Design Split
13	Climb	501 (935)	1.4 (205*)	0.020	9 (20*)	15/85	
14						30/70	
15				0.023			Fuel/Air Variation at Design Split
16	SLTO, STD	532 (991)	1.4 (210*)	0.021	9 (20*)		Fuel/Air Variation at Design Split
17				0.023			
18				0.025			
19	SLTO, Hot	568 (1055)	3.0 (444)	0.027	18.7 (41.4)	15/85	
X-960							Advanced Liner,
Temperature							Air supply
							Evaluation

*Corresponds to flow parameter simulation at X-903 air supply maximum operating conditions, louvered liner configuration.

SECTION 8.0
CONCLUDING REMARKS

The combustor design for the Energy Efficient Engine represents a substantial improvement in mechanical simplicity, durability and overall performance relative to the first generation two-zone Vorbix system evaluated during the Experimental Clean Combustor Program. The results from supporting technology programs have provided the necessary design substantiation, demonstrating all aerothermal performance and emissions goals, except for oxides of nitrogen.

The segmented liner concept – the most prominent feature of the combustor design – offers the structural integrity necessary to sustain operation at the high pressure and temperature levels envisioned for future fuel efficient gas-turbine engines. This design introduces new avenues where further technology advances can be made. A substantial weight reduction could be attained by modifying the liner support structure. Also, liner fabrication costs could be sharply reduced by simplifying the internal cooling technique of the liner. Recent advances under independent research and development programs have identified other cooling techniques that would enable the use of larger, as-cast liner segments that require little machining, as opposed to the current FINWALL® approach. These advanced concepts also offer improved thermal effectiveness.

The combustor fuel management system is a second area where significant improvements have been made. The unique main zone carburetor tube design is a major contributor, providing improved fuel atomization for emissions control, low smoke and pattern factor, while permitting a shorter main zone combustion section. The carburetor tube fuel injector, manifolded integrally with the pilot zone aerated fuel injector, provides an attractive fuel system approach in the event that the current oxides of nitrogen emissions regulatior dictate the use of a two-zone combustor in future commercial aircraft.

Overall, the technology evolved through the combustor design and experimental verification processes has wide application.

LIST OF ABBREVIATIONS AND SYMBOLS

A	area
AC_D	effective flow area
atm	atmospheric unit of pressure (14.7 psi)
bx	axial chord
CO	carbon monoxide
Cp	pressure recovery coefficient
EI	emissions index (gm pollution/kgm fuel)
EPAP	Environmental Protection Agency Parameter
F/A	fuel/air ratio
HC	hydrocarbons
i or ID	inner annulus or inner diameter
L	prediffuser axial length
LBO	lean blowout
micron	unit of length equal to one thousandth of a millimeter (0.00004 in)
NOx	oxides of nitrogen
o or OD	outer annulus or outer diameter
ODS	oxide dispersion strengthened
ΔP	pressure differential
P_S	static pressure
P_T	total pressure
R	radius from engine centerline
r	prediffuser inlet annulus height
SLTO	sea level takeoff
SMD	Sauter Mean Diameter indicating fuel spray droplet size
THC	total unburned hydrocarbons
TOBI	tangential on-board injection
T_T	total temperature
T/C	thermocouple
V	velocity
W_a	total mass airflow rate (measured)
X	axial displacement
$X/\Delta R$	baseline nondimensional prediffuser dump gap
Y	tangential displacement
$\emptyset$	Equivalence ratio

Terms from Table 4.3.1-II

Term	Definition
B-1	Absolute air angle at row inlet
B-2	Absolute air angle at row exit
B'-1	Relative air angle at row inlet
B'-2	Relative air angle at row exit
ESPI	Angle between tangent to streamline projected on meridional plane and axial direction

Terms from Table 4.3.1-II (Cont'd)

Term	Definition
DEV	Deviation angle (Exit air angle minus the metal angle at trailing edge
D-FAC	Diffusion factor (see above)
EFF-A or -AD	Adiabatic efficiency
EFF-P	Polytropic efficiency
EPSI-1	Slope of meridional streamline at row inlet
EPSI-2	Slope of meridional streamline at row exit
INCS	Incidence angle between inlet air direction and line tangent to blade suction surface at leading edge, degrees
INCM	Incidence angle between inlet air direction and line tangent to blade mean camber line at leading edge, degrees
LOSS-P	Loss parameter
M-1	Mach number at row inlet
M-2	Mach number at row exit
M'-1	Relative Mach number at row inlet
M'-2	Relative Mach number at row exit
NCORR	Low-pressure rotor speed, corrected
OMEGA-B	Total pressure loss coefficient (Mass average defect in relative total pressure divided by difference between inlet stagnation and static pressures)
PCT TE SPAN	Percent trailing edge span
PO/PO	Pressure ratio

Terms from Table 4.3.1-II (Cont'd)

Term	Definition
P02/P01	Static pressure ratio
RHOVM-1	Density times meridional velocity at row inlet
RHOVM-2	Density times meridional velocity at row exit
TURN	Relative air turning from inlet to exit, degrees
TO/TO	Temperature ratio
U-1	Rotor tangential speed at row inlet
U-2	Rotor tangential speed at row exit
V-1	Air velocity at row inlet
V-2	Air velocity at row exit
VM-1	Meridional velocity at row inlet
VM-2	Meridional velocity at row exit
V'-1	Relative air velocity at row inlet
V'-2	Relative air velocity at row exit
Vθ-1	Tangential velocity at row inlet
Vθ-2	Tangential velocity at row exit
Vθ'-1	Relative tangential velocity at row inlet
Vθ'-2	Relative tangential velocity at row exit
WCORR	Airflow, corrected
WC1/A1	Stage inlet corrected airflow divided by stage inlet area

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