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The Energy Efficient Engine (E^3) — Advancing the State of the Art

The NASA sponsored E^3 Program, being conducted by Pratt & Witney Aircraft, is described, including program objectives and goals. The evolution of the Flight Propulsion System design and the related work of the aircraft manufacturers is discussed. The status of the component technology substantiation program is summarized.

INTRODUCTION

Background

In 1974, NASA initiated the conceptual definition of future aircraft engines with significantly lower fuel consumption than today's turbofans. An exploratory study (1) was contracted with Pratt & Whitney Aircraft Group to identify the design approach and technology requirements of such an engine.

In 1977, with the engine conceptual definition complete, NASA embarked on the Energy Efficient Engine Program (E^3). As a first step, preliminary design and integration studies (2) provided the baseline engine definition to be used in the current program, contracted in 1978.

E^3 Component Development and Integration Contract

The objectives of the current E^3 program(3) are to develop and demonstrate the technology for reducing fuel consumption in future environmentally acceptable turbofan engines. The specific goals, listed on Table I, address performance, economics and environmental factors.

The program consists of four major tasks as shown in Fig. 1. Task 1, providing the base upon which the balance of the program builds, consists of the preliminary design of a Flight Propulsion System (FPS) configured to meet the goals, and the update of that design at the end of the program. The final update serves as the measurement of how well the initial goals were met.

TABLE I

FLIGHT PROPULSION SYSTEM GOALS

PERFORMANCE, ECONOMICS AND ENVIRONMENT ARE CONSIDERED

Fuel Consumption*	
o SFC Reduction	≥ 12%
o Deterioration Reduction	≥ 50%
Operating Costs*	
o DOC Reduction	≥ 5%
Environmental	
o Noise	FAR-36 (1978)
o Emissions	EPA-1981

*REL JT9D-7A

Task 2 is divided into two parts; (1) supporting technology and (2) design, fabrication and tests of the components. The supporting technology efforts are performed to verify advanced concepts prior to their incorporation into the component detail designs. Full scale components to be individually tested are the high-pressure compressor, combustor, and high-pressure turbine. While the fan is not tested in an aerodynamic rig, the blades will be subjected to extensive mechanical testing to verify structural concepts.

Task 3 provides for the design, fabrication and test of the core engine while Task 4 combines the Task 3 core with the fan, low pressure compressor and low pressure turbine in a complete systems test.

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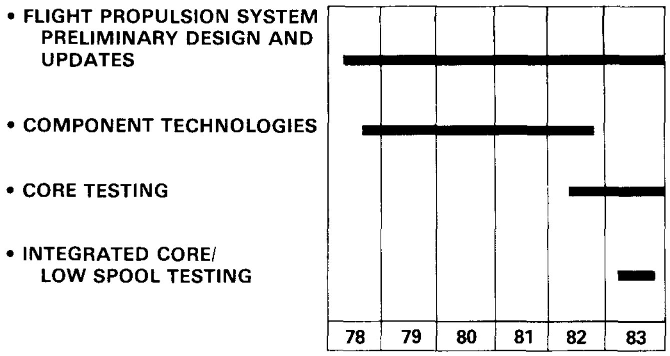

Fig. 1 Schedule. Program structured to meet goals.

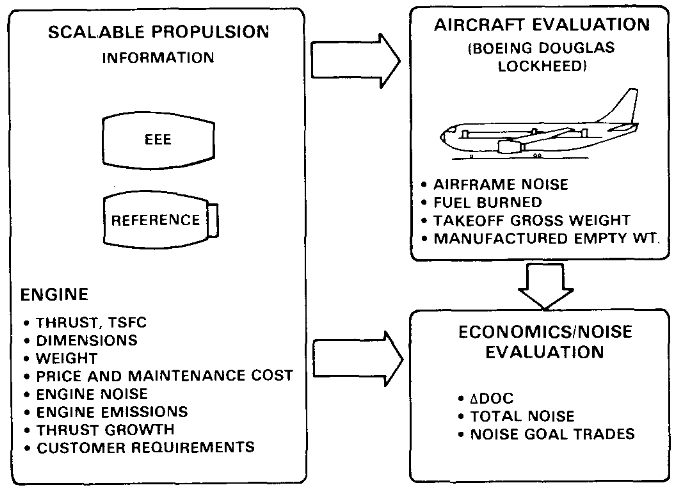

Fig. 2 Evaluation Process. Economics and noise were evaluated jointly.

FLIGHT PROPULSION SYSTEM (FPS) PRELIMINARY DESIGN

The preliminary design of the FPS cycle and configuration, defined during the 1977 E^3 contract (2) and described at the NASA CTOL Transport Technology Conference (4) in 1978, was further refined under Task 1 of the current contract. Design feasibility was confirmed.

Design Process

Starting with the engine defined in the 1977 contract, the low spool was further refined. Perturbations in each component were made and, using trade factors, the results were assessed and an optimum configuration defined. Other subsystems, including the active clearance control system, bearing and lube systems, and the secondary airflow system were defined and analyzed in this effort.

As the design was being refined, representative engine data were sent to Boeing, Douglas, and Lockheed, who provided independent aircraft evaluations of the FPS capabilities to aid the refinement as shown in Fig. 2. Scalable propulsion information was provided to Boeing, Douglas and Lockheed. The airplane manufacturers sized and evaluated the performance of their advanced airplanes with both the JT9D-7A, as the base, and the E^3; and furnished airplane weights and dimensions, aerodynamics, fuel burned on design and typical missions, engine size, and airframe noise at FAR 36 noise measuring points. They also provided design and typical mission sensitivities to TSFC, propulsion system weight, and nacelle drag. This information was used by Pratt & Whitney Aircraft in assessing the engine capabilities relative to the goals as the preliminary design evolved.

Flight Propulsion System Description

The resulting flight propulsion system preliminary design from this process is shown in Fig. 3. The cross section shows the large shroudless fan blades and waspwaisted engine core gaspath, the two-frame rotor support system, and the acoustically treated, single-nozzle nacelle, with fan-only thrust reversing.

The engine has the following cycle characteristics: a 6.5 fan bypass ratio, a 1.74 fan pressure ratio, a 38.6 overall pressure ratio, a 1225°C (2240°F) turbine rotor inlet temperature, a design

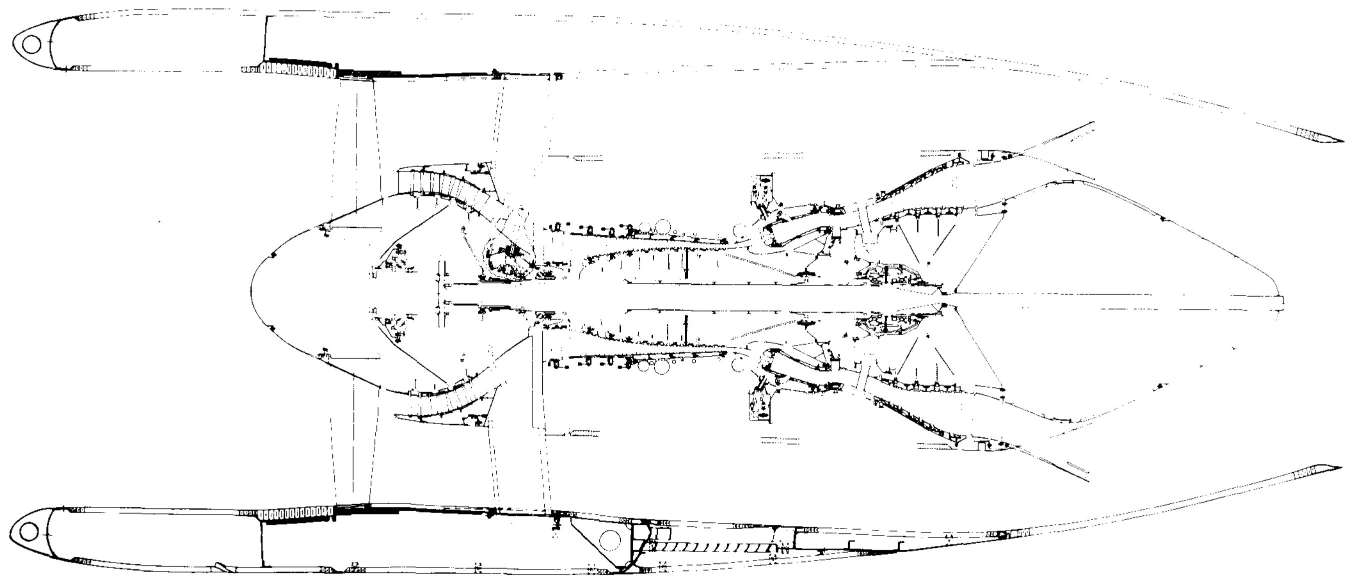

Fig. 3 Cross Section. FPS design for energy efficiency.

fan corrected airflow size of 625 kg/sec (1375 lb/sec) at 10,668 m (35,000 ft) Mach 0.8 design point, and a mixed flow exhaust. These cycle characteristics were selected to emphasize fuel economy and low operating costs. Overall pressure ratio was selected at as high a level as possible considering TSFC, thrust growth, nitrogen oxide (NO_x) generation, and design practicality. Turbine inlet temperature selection was based on minimizing TSFC while allowing uncooled low-pressure turbine airfoils for cost savings. The fan cycle parameters were selected to minimize direct operating cost. The mixer was chosen to reduce both fuel consumption and direct operating cost.

Core air is pressurized by the fan inner section, a four-stage low-pressure compressor and a ten stage high-pressure compressor. The combustor has two in-line combustion zones to control emissions. A one-stage, aircooled high-pressure turbine drives the high-pressure compressor. An uncooled four-stage low-pressure turbine drives the fan and low pressure compressor. The low and high spools counter-rotate, permitting lightly cambered low-pressure turbine inlet vanes for increased turbine efficiency. The core exhaust and the fan duct air are mixed by means of chutes positioned around a large diameter central tailplug and then discharged through a common-flow exhaust nozzle.

Active clearance control on the rear section of the compressor (four external air impingement tubes) and on the turbines (internal case systems) is designed to closely match rotor and case diameters during cruise operation without inducing rubs during takeoff and climb maneuvers.

Two major frames and main shaft bearing compartments are situated between the compressors and between the turbines to independently support both rotors. Compartment ventilation is provided by a center vent breather system utilizing cool low-pressure compressor exit air which is discharged into the low spool shaft and exhausted through the tailplug.

The nacelle is designed to share flight loads through the load-carrying fan ducts which transmit much of the inlet gust moment and other cowl loads around the engine cases and to the mounts. The nacelle system consists of an inlet section, fan case cowling, a D-shaped duct mid-section, and the exhaust system, as shown in Fig. 4.

The inlet section is designed to bolt directly to the fan case front flange to eliminate flowpath steps or gaps near the fan front face. The fan cowling section consists of two doors which are hinged from the pylon and latched along the bottom of the nacelle to provide access to the fan case mounted engine control, the thrust reverser air motors, the air lines, the jack-shaft drive units, and the oil tank. The mid-section of the nacelle is made up of two large D-shaped semi-circular structural doors which provide full access to the rear of the fan and lower service lines, the core mounted full duty gearbox and accessories, the compressor, the combustor, and the turbines. The exhaust system is designed as a one-piece unit suspended on rollers which run on a track in the pylon.

The engine control system is a full authority digital control, vibration isolation mounted on the fan case and air cooled. Electronics implementation will be with large and very large scale integrated circuits.

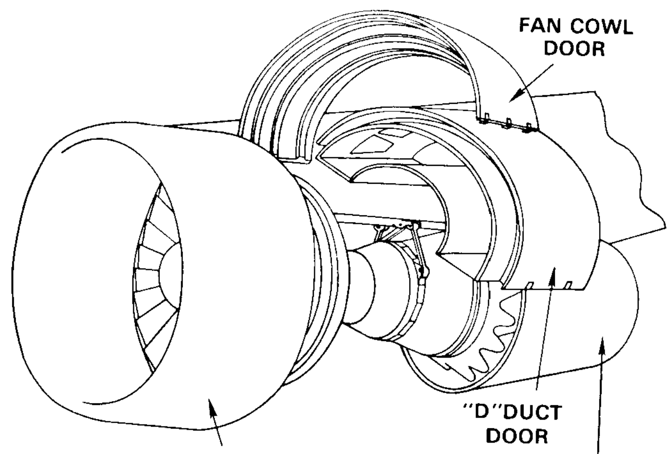

Fig. 4 Pylon Mounted Nacelle Schematic. Nacelle designed for load sharing with accessibility.

Clearance Control - Straddle mounting of the high spool rotor, optimal three bearing support of the low spool rotor, structural integration of the nacelle and engine, and active clearance control are used to provide the tight running clearances needed for high performance. Straddle-mounting the short, stiff, high-spool rotor, by locating the two mainshaft bearings just fore and aft of the front and rear stages, centralizes the masses between and near the support structures causing the cases and the rotors to have similar deflections under normally encountered flight loads. The fan is overhung from two widely-spaced bearings, providing a firm wheel base. The low-pressure compressor is centered between the bearings to control rotor deflections. The rear bearing is positioned beneath the two front low-pressure turbine stages.

Inlet cowl moments and other aerodynamic loads are transmitted through the fan ducts to provide load sharing. The larger diameter, stiffer fan ducts are designed to carry up to 80 percent of the total load to substantially reduce engine deflections in the fan region.

The active clearance control system, shown schematically in Fig. 5, uses external fan air impingement tubes on the compressor and a dual manifold compressor air feed system to supply the turbines with different cooling air between takeoff and cruise. Electrically actuated valves are used to control the air flow.

Status Towards Program Goals

The analysis and design efforts show that the Flight Propulsion System design is feasible and capable of meeting all design goals except for the EPA-proposed 1981 NO_x regulation. Results of tne most recent evaluation of the engine capabilities, made during October 1979, are compared to the goals on Table II and are discussed below.

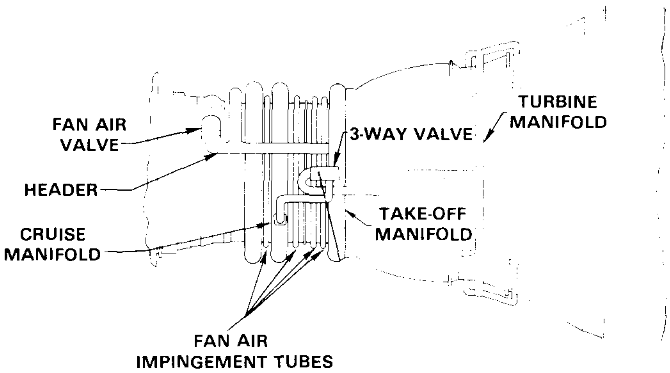

Fig. 5 Active Clearance Control System Schematic. Compressor system is external, while turbine system is internal.

TABLE II
STATUS FLIGHT PROPULSION SYSTEM CAPABILITIES
MAJORITY OF GOALS ARE EXCEEDED

	Goals	Status
Fuel Consumption*		
o SFC Reduction	12%	14.7%
o Deterioration Reduction	50%	50%
Operating Costs*		
o DOC Reduction	5%	8.1%
Environmental		
o Noise	FAR-36 (1978)	Meet With 2 DB Margin
o Emissions	EPA-1981	Meet All But NO_x

*REL JT9D-7A

Performance predictions for the flight propulsion system indicate a 14.7 percent installed cruise TSFC reduction relative to the JT9D-7A, as shown in Fig. 6. Projected improvements in the components account for two-thirds of the reduction. The other one-third is provided by the 50 percent higher pressure ratio Energy Efficient Engine cycle, and by exhaust mixing.

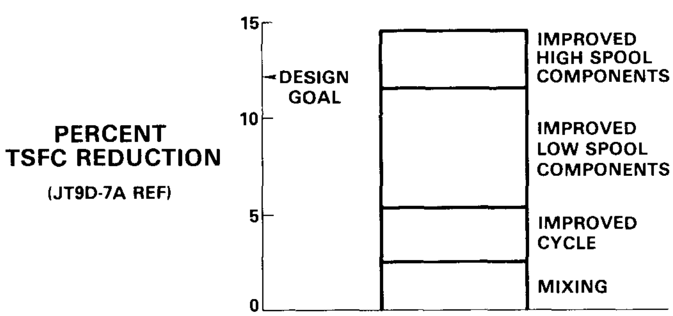

Fig. 6 TSFC Goal Exceeded. Both component and cycle improvements are needed to achieve TSFC reduction.

Many of the design features to control clearances are expected to reduce performance deterioration caused by in service clearance increases. The stiff rotor and cases and load sharing nacelle will reduce deflections and rubs under severe maneuver loads. The use of abradable rub strips over abrasive blade tips results in a local region of clearance increase between the tip and seals; blade tip diameter and clearance over a large percentage of the circumference remains unchanged. Therefore, the preferential wear characteristics of the abradable seal system minimize clearance deterioration during severe airplane maneuvers, case ovalization, and other operational conditions, which may cause blade-to-seal rubs.

Rotor-frame analysis indicates that early rapid deterioration can be essentially eliminated with advanced design concepts, and that the deterioration rate from 100 to 1000 cycles for current high bypass ratio engines can be matched, as shown in Fig. 7 despite the higher pressure ratio cycle and smaller compressor blades that increase sensitivity to tip clearances and airfoil surface roughness, and increase the concentration of erosive particles in the gaspath. Therefore, the initial 14.7 percent E^3 TSFC advantage, which is based on a comparison of newly installed engines, should rapidly increase to over 16 percent for the major period of in-service operation.

DOC predictions, based on common economic ground rules equally applied to the seven study aircraft described in Table III, are shown in Fig. 8. The DOC reduction with increasing range is principally a result of the higher fuel fraction associated with long range aircraft. The average DOC reduction is 8.1 percent which exceeds the goal of 5 percent. Fuel prices of 10.5¢/l (40¢/gal) for the domestic aircraft and 11.9¢/l (45¢/gal) for the intercontinental aircraft were assumed in converting the fuel burned data to fuel cost.

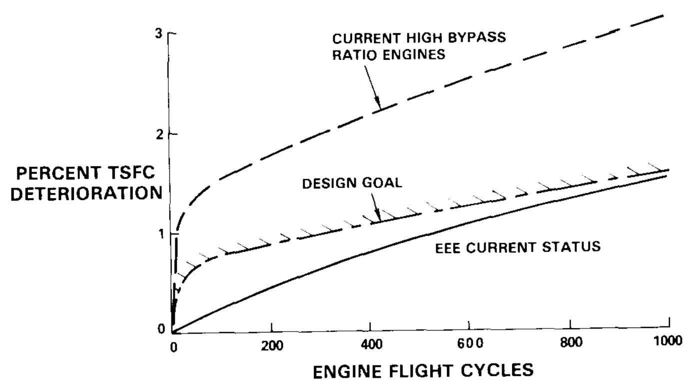

Fig. 7 Deterioration Rate Meets Goal. Early high deterioration rate is eliminated by advanced concepts.

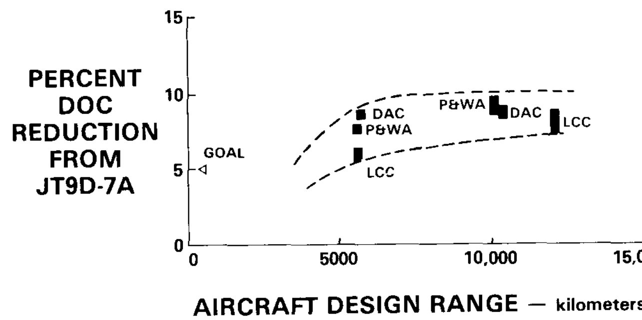

Fig. 8 DOC Goal Is Exceeded. DOC reduction improves with range.

TABLE III

ADVANCED AIRCRAFT USED IN DOC ANALYSIS

AIRCRAFT PROJECTIONS WERE PROVIDED BY MANUFACTURERS

	DOMESTIC				INTERNATIONAL		
	Boeing	Douglas	Lockheed	P&WA	Douglas	Lockheed	P&WA
Type	Twinfan	Trifan	Trifan	Trifan	Trifan	Quadfan	Quadfan
In-Service Date	1990's	1990's	1990's	1990's	1990's	1990's	1990's
Passengers	196	458	500	440	438	500	510
Design Range (n.mi.)	2000	3000	3000	3000	5500	6500	5500
Typical Range (n.mi.)	1000	1000	1400	700	1500	3000	2000

Noise predictions for study aircraft systems of Boeing, Douglas, Lockheed, and Pratt & Whitney Aircraft are summarized together with the limits of the three FAR noise measurement points on Fig. 9. The E3 is predicted to meet FAR-Part 36 (1978) noise limits in six of the seven study aircraft. In the Boeing twinjet, the takeoff limit is exceeded by approximately 0.5 dB, while the approach and sideline limits are met. The noise levels of the trijet and quadjet airplanes are 2 dB or more below the limits. Refined matching of the acoustic configuration of the flight propulsion system to each airplane, and/or further optimization of the airplanes for minimum noise, could further reduce noise levels.

Emissions indices (EI's) (pounds of pollutant per pound of fuel) were obtained from test results of the Experimental Clean Combustor Program (5) (ECCP) two-stage combustor. These tests correlated the EI with operating pressure, temperature, and fuel-to-air ratio for the three pollutants at each of the five operating modes. Margins for development and engine-to-engine variations were superimposed on these estimates. The EI's and individual mode engine performance levels were then combined with the time-in-mode information to arrive at the final EPA parameter estimates. The CO and HC emissions are below their limits while NOx exceeds its limit. Advanced NOx-reducing combustor technology from other programs is being assessed for possible incorporation into the E3 design to further reduce NOx toward the goal levels. Smoke estimates are based on correlations of SAE smoke number versus pressure and fuel-to-air ratio, assuming the presence of carburetor tubes in the main zone to reduce both NOx and smoke. Smoke levels are estimated to meet the goal of an SAE smoke number of 20.

TECHNOLOGY SUBSTANTIATION STATUS

As indicated in the introduction, the preliminary design definition of the FPS was accomplished under Task I. The component designs are being carried out in detail as part of Task II. Also under Task II, sub-component supporting technology programs are being conducted to verify the advanced concepts to be included in the component detailed designs.

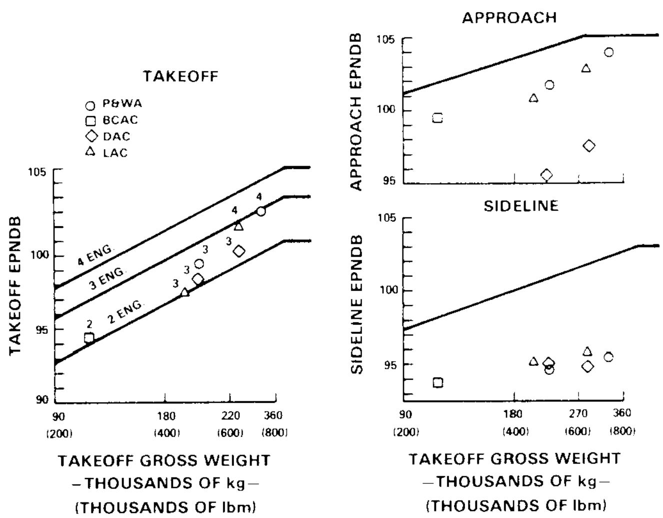

Fig. 9 Predicted Noise Levels Vs. FAR Part 36-1978 Design Goal. All aircraft meet goals with noise trades.

Supporting Technology

To make large advances in the state-of-the-art of energy efficiency, it became necessary to go beyond the currently available design tools. As shown in Table IV, thirteen supporting technology programs covering the fan, combustor, turbine and mixer were needed to verify design concepts and/or calibrate design tools. While all of the programs represent advances in technology, the programs of most significance are indicated by asterisks. Separate papers on the Diffuser/Combustor (6) and Uncooled High Pressure Turbine (7) have been published and are presently available. The fabrication development, transition duct, mixer, and fan programs, for which results have not been previously published, are discussed below.

Fabrication Development - The single stage high-pressure turbine incorporates long chord vanes and highly twisted tapered blades, both of which are to be cast in two pieces from single crystal nickel alloys and bonded together. The objective of the fabrication development program is to adapt the two-piece casting process to the complex E3 geometries. Over 30 vanes and blades have already been successfully cast and bonded in this program. Fig. 10 shows a successfully bonded blade. No compromise to the airfoil design was necessary for fabrication purposes; confidence is now high that the high-pressure turbine component hardware can be readily fabricated using the two-piece approach.

TABLE IV
SUPPORTING TECHNOLOGY

PROGRAMS VERIFY ADVANCED DESIGN CONCEPTS

Fan

- o Scaled Rig*
- o Hollow Blade*

Combustor

- o Diffuser/Combustor Rig*
- o Sector Rig*

High Pressure Turbine

- o Uncooled Rig*
- o Supersonic Cascade
- o Leakage
- o Cooling
- o Fabrication Development*

Low Pressure Turbine

- o Subsonic Cascade
- o Boundary Layer
- o Transition Duct*

Mixer

- o Model tests*

*Indicate programs of most significance

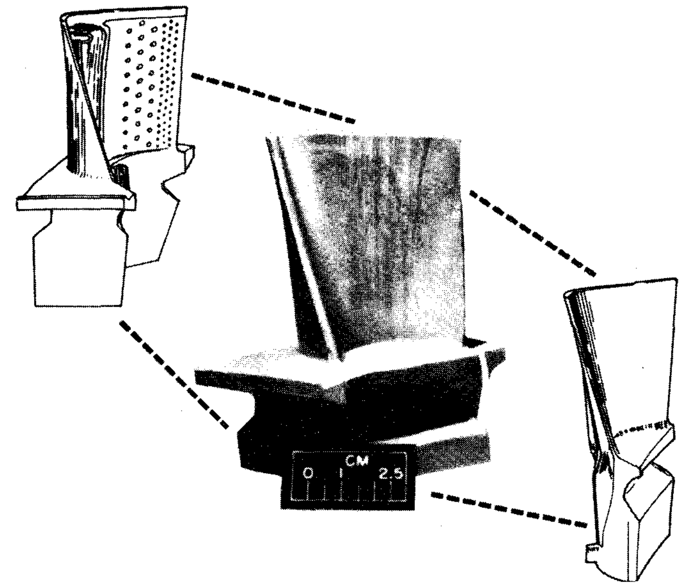

Fig. 10 Successfully Bonded two-Piece Turbine Blade. No compromise in design was required.

Transition Duct - The E3 incorporates a transition duct between the exit of the HPT and the inlet of the larger diameter LPT. The duct incorporates aerodynamic fairings around the structural rotor support members. The objective of the transition duct program is to establish the level of pressure loss which could be expected in a design of this type. It was originally estimated that the pressure loss would be 1.5%; however, three-quarter scale model tests to date have demonstrated a loss of only 0.7%. This result translates into an additional 0.25% reduction in TSFC.

Mixer - The 1977 E3 studies showed that an engine with a forced mixer could provide a 3% reduction in TSFC relative to a comparable separate exhaust engine. The mixer design resulting from Task 1 efforts is about 30 percent shorter than current technology mixed-flow exhaust systems. The short mixing length, in combination with a high mixing efficiency goal of 85 percent, represents a major aerodynamic design challenge. The mixer shown in Fig. 3 consists of twelve lobes equally spaced around the circumference, with cut-outs or scallops, on the sides of the lobes to enhance mixing.

A parametric mixer model program was run with the objective of empirically evaluating the effects of length, lobe number, and lobe geometry on performance potential. A total of 30 configurations was tested at the Fluidyne Engineering Corporation facilties. Fig. 11 shows a representative model. Several of the models demonstrated high levels of mixing efficiency accompanied by high pressure loss. The best performing model demonstrated an equivalent TSFC reduction of 2.2%. The more promising geometries will be refined to reduce losses and retested in a second model test series where it is expected that the performance goal will be achieved.

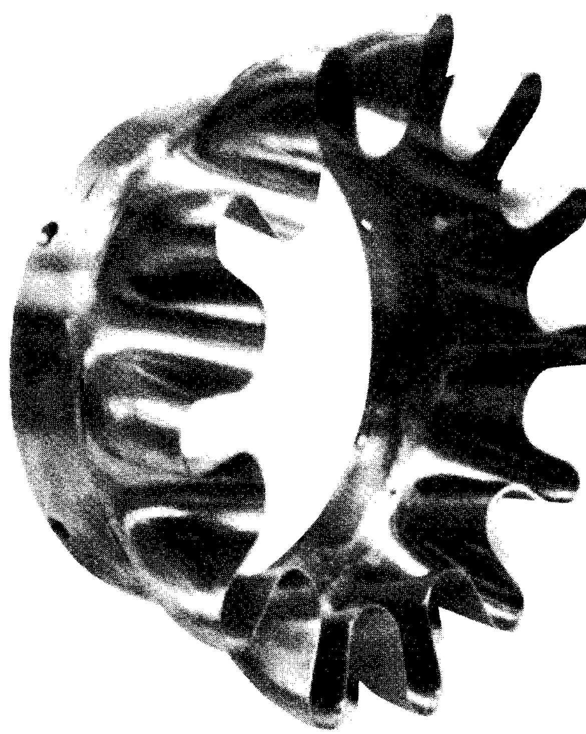

Fig. 11 Representative Model Mixer. 30 Models used to evaluate performance.

Fan - The fan blades are shroudless designs with low aspect ratio. Elimination of a shroud can increase fan efficiency by 0.7 to 1.0%. However, the low aspect ratio design needed structurally, causes an increase in stage weight. To reduce the weight penalty, the blades have been designed to be partially hollow. Two supporting technology programs, a scaled fan rig and hollow blade development, are included in the program to address the aerodynamic and structural questions. The objectives of the scaled fan program are to determine the efficiency benefit of shroud removal and to measure blade stress levels when subjected to a 2 per revolution disturbance. Information on how to aerodynamically match the exit guide vane and pylon to the fan blade discharge flow field for best performance will also be obtained. This program is scheduled to start in late 1980.

The hollow blade development effort is divided into two major areas, fabrication and structure. The blades are to be made from laminated titanium sheets which are hot isostatically pressed to form consolidated airfoils and then isothermally forged to the finished contours. The fabrication technology program is directed towards establishing the details of the manufacturing process, while the structural program will examine foreign object damage (FOD) tolerance, structural integrity, and vibratory response characteristics. The fabrication program has already been initiated. Fig. 12 shows a test specimen which has been hot isostatically pressed. Start of structural testing is planned in 1981.

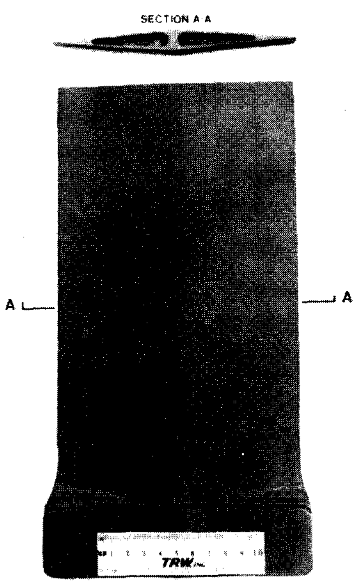

Fig. 12 Hot Isostatically Pressed Fan Blade Specimen. Specimens with hollow sections have been successfully fabricated.

Component Detailed Design Status

As design concepts have been verified in the supporting technology programs, the detailed designs of experimental components have started. The intent is to make the experimental hardware identical to the FPS. It is recognized that precise duplication is not possible in all areas since procurement lead times for certain materials may be excessive or that lower cost materials may serve just as well for ground test evaluation; however, the component and engine hardware will have the same flow path, cooling and leakage flows, and mechanical and structural design as the FPS.

Fan - Detailed Design of the fan blade was completed early in the program so that the final design was available for use in the fabrication technology program. The design of the remainder of the fan module (disks, cases, shafts, etc.) will be completed in 1981.

The blade geometry remained essentially unchanged from the preliminary design and is shown in Fig. 13. Table V lists the key design parameters. The airfoil hub-to-tip radius ratio is set low to improve efficiency and reduce weight. The blade planform, with a 2.5 aspect ratio for increased stiffness, increases in chord from root to tip. The thickness-to-chord ratio decreases from root to tip and the airfoils are twisted 70 degrees from root to tip. Because of shroud elimination, the airfoil detailed design geometry includes compensation for uncamber and untwist at running speeds. The blades have solid titanium dovetail attachments to the disk. The transition region from the airfoil to dovetail is wide and shallow to maximize the stiffness required by the shroudless airfoils. The broach angle is low to control attachment region stresses.

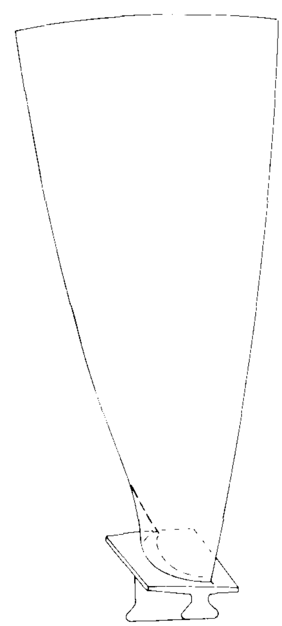

Fig. 13 Shroudless Fan Blade. Chord increases from root to tip.

TABLE V

FAN DESIGN PARAMETERS
VALUES WERE SELECTED TO PROVIDE HIGH EFFICIENCY

Duct Pressure Ratio	1.74
Duct Efficiency (%)	87.3
Tip Speed (Ft/Sec)	1500
Bypass Ratio	6.5
Corrected Flow (ADP) (Lb/Sec)	1373
Inlet Specific Flow (Lb/Sec Ft^2)	43
Inlet Hub/Tip Ratio	0.34
Number of Blades	24
Aspect Ratio	2.5
Number of Duct Exit Guide Vanes/Struts	30

Aerodynamically, the blades are designed for minimal shock losses by utilizing contoured airfoils based on a "through-the-row", quasi three dimensional design system. Blade relative Mach number at the row inlet varies from 0.77 at the root to 1.48 at the tip. Respective exit relative Mach numbers are 0.55 at the root and 0.93 at the tip. A tip speed of 1500 ft./sec. results in moderate aerodynamic loading levels comparable to current fans.

The increased efficiency of this fan configuration, from shroud elimination and other aerodynamic improvements, can reduce specific fuel consumption by about 3 1/2 percent and reduce airline DOC by about 1 percent relative to a JT9D-7A type fan.

In deciding on fan blade internal geometry, various degrees of hollowness and several internal stiffening rib patterns were examined to produce acceptable flutter and vibration operational margins. The internal geometry which evolved during this design effort, Fig. 14, is two-thirds hollow for lightness with three radial stiffening ribs. Wall thickness in the hollow sections is varied with thicker walls at the bottom of the cavity and along the middle rib in accordance with the local stress field.

Because the blade is shroudless and hollow, careful attention was given to the structural design, especially vibration and FOD durability. A finite element analytical model of the blade was made in which plate elements were used to describe the skin, internal ribs, and solid inner portion of the blade to assess vibration characteristics. As shown in Fig. 15, two of the critical vibration modes (2nd bending and 1st torsional) are calculated to occur safely above redline speed. The 1st bending vibratory mode for two excitations per revolution cannot be placed fully outside the operating range. Therefore, this mode is positioned as low as possible in the operating range, just above flight idle, to minimize the vibrational energy. Stall flutter and unstalled supersonic flutter margins have been analyzed and are acceptable. Foreign object damage sensitivity was examined by analyzing bird ingestion leading edge and bending stress parameters. By controlling solid leading edge depth and tip section wall thickness, stress parameters were held at or below current levels.

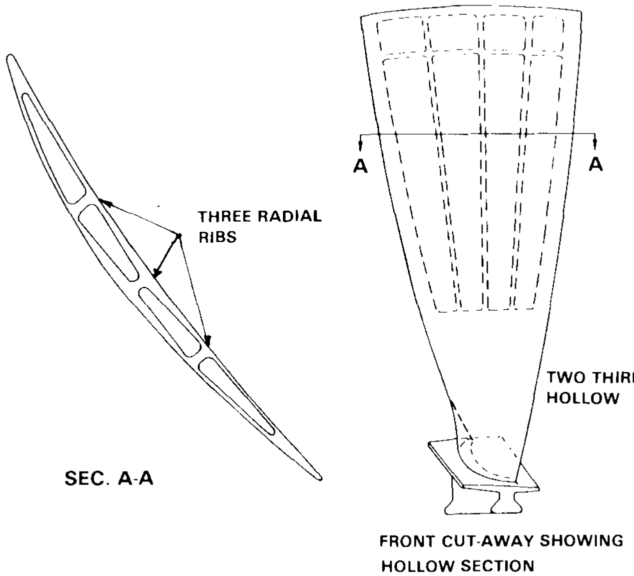

Fig. 14 Fan Internal Configuration. Blade is two-thirds hollow for lightness.

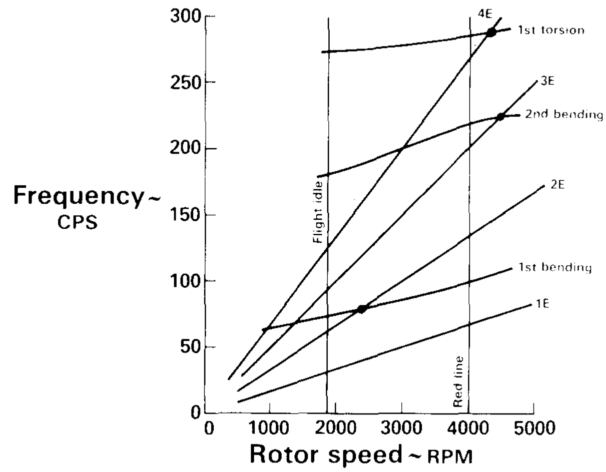

Fig. 15 Predicted Fan Vibration Characteristics. First-bending occurs low in operating speed range; second bending and first torsion intersect above redline speed.

Pending experimental verification, the fan blade is projected to meet all of the originally established mechanical, aerodynamic and structural requirements.

High Pressure Compressor - The high-pressure compressor detailed design is the product of continuing evolution as technical information became available from other compressor research activities. All of the advanced concepts and design tool verification efforts have been conducted outside the contract. The compressor is designed for 14:1 pressure ratio in ten stages. Table VI is a listing of the key design parameters, and Fig. 16 is a cross-section showing major design features.

Major subassemblies include a drum rotor system with only one bolt circle, an axially split front case with four variable stagger stator vane rows, a nonsplit mid-compressor case with two bleed manifolds, and an active clearance control system consisting of four air impingement tubes over the rear of the compressor. A forward extension of the diffuser case supports the rear shroud assemblies.

TABLE VI

HIGH-PRESSURE COMDPRESSOR DESIGN PARAMETERS

SELECTED VALUES RESULT IN LOW AIRFOIL COUNT AND HIGH EFFICIENCY

Number of Stages	10
Pressure Ratio	14
Adiabatic Efficiency	88.2%
Corrected Airflow, Lb/Sec	77.5
Inlet Corrected Tip Speed, Ft/Sec	1245
Inlet Flow Capacity, Lb/Sec/Ft^2	38.0
Inlet Hub/Tip Ratio	0.56
Exit Hub/Tip Ratio	0.924
Average Aspect Ratio	1.52
Average Gap-Chord Ratio	0.89
Number of Airfoils	1297
Number of Variable Stator Rows	4

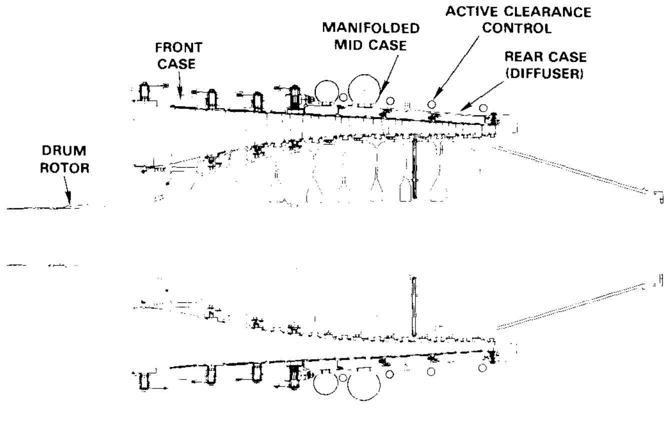

Fig. 16 High-Pressure Compressor Cross-Section. Compressor has drum rotor for stiffness and small rim cavities for efficiency.

The rotor is comprised of a titanium front drum and a higher temperature nickel alloy rear drum. Eight blade rows are included in the front drum with two rows in the rear drum. Rim cavities in the drum are small in volume and carefully sealed to improve efficiency.

The front case/variable stator assembly is of a proven design. Both the case and stator inner shroud rings are axially split 180 degrees apart for assembly over the drum rotor. The mid-compressor case is a full-hoop configuration to reduce leakage and avoid thermally induced ovalization. To assemble the compressor, the case is slid over the stator assemblies to engage the stator vane retaining hooks. In the rear section of the compressor, both the inner and outer stator shrouds are segmented to permit the active clearance control system to establish the tip clearance over the blades. External tubes are used to impinge cool fan air directly on the rear casing to reduce tip clearance at cruise.

Bleed air is extracted at two mid-compressor O.D. locations and from behind the seventh blade row at the I.D. Manifolds are used to collect the O.D. air while the I.D. air is extracted through tubes which carry it inward to the disk bore region for use in internal engine cooling.

Airfoil selection is based on the Mach number and incidence range requirements of each row. The front two blade rows operate at transonic relative Mach numbers leading to the selection of multiple circular arc (MCA) airfoils with straight leading edges on the suction surface to reduce both the supersonic acceleration in front of the normal shock and the shock viscous losses. For the remainder of the blade rows and for all of the stator vane rows, controlled diffusion airfoils (CDA), with increased freedom in airfoil geometry, are used to address the high subsonic Mach numbers encountered in the mid rows and the large incidence angle swings required in the rear rows.

Analytical predictions of flutter and resonance, thermal gradients and stress levels have been made to verify the structural adequacy of the design. Testing of the complete compressor as a component is scheduled for early in 1981 with subsequent testing of a second compressor build in 1982.

Combustor - Fig. 17 shows a cross-section of the diffuser-combustor. The combustor design evolved from the Experimental Clean Combustor Program (ECCP) where both high and low-power emissions were reduced by incorporating two combustion zones, each individually optimized to have specific low emission characteristics. The strutless curved wall pre-diffuser turns the air outward, eight degrees, to align the compressor discharge air with the center line of the combustor. The 24 thin and aerodynamically shaped diffuser case struts, located in the dump section, tie together the inner and outer cases. The combustor incorporates two inline burning zones. The pilot zone is designed for stability and relight characteristics with low emissions at idle. There are 24 single pipe aerated fuel nozzles for good atomization characteristics. The main zone is used for all power settings above ground idle and is optimized for low NO_x and smoke. The main zone incorporates 48 low pressure fuel nozzles which discharge into carburetor tubes where the fuel-air mixture is prepared and then delivered to the combustor.

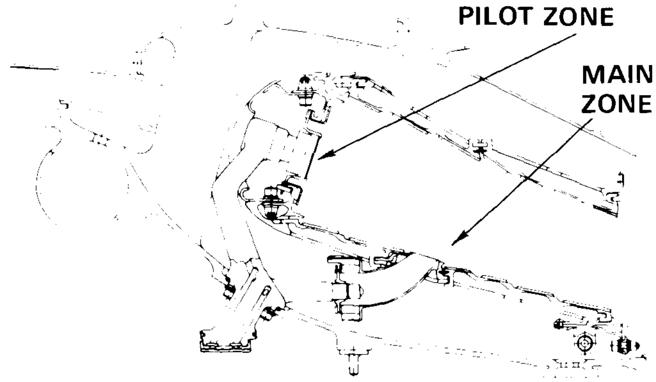

Fig. 17 Combustor Cross-Section. Two combustion zones produce low emissions.

An advanced cooling concept, Counter-Parallel FINWALL® (CPFW) is used for the liner walls, shown in Fig. 18. Coolant air enters through slots in the cold wall, flows up and downstream in discrete cooling passages, and is injected through continuous slots on the hot side as a protective film. The liner, segmented to improve liner life relative to a full hoop design, is made of cast B-1900 turbine alloy. The segments are supported by a Hastalloy-X framework, with feather seals between adjacent segments to reduce leakage. Fig. 19 is an isometric rendering of the liner and support frame.

The detail design is now essentially complete and the status of combustor emission parameters is shown on Table VII. All values meet or exceed requirements except NO_x which misses the requirements by 1.4 units. Sector rig fabrication is in process with testing scheduled to begin in July, 1980. Results from this rig test will be used to update the final design in late 1980 so that fabrication and testing of the full annular rig can commence by late 1981.

High Pressure Turbine - The detailed design of the high pressure turbine has been completed. Fig. 20 is a cross-sectional drawing showing the important design features, and Table VIII is a listing of key design parameters. The technical success of this single stage turbine hinges on achieving low leakage, effectively utilizing new materials and fabrication methods, and reducing the number of airfoils from previous designs, in addition to advancing the state-of-the-art in transonic aerodynamics.

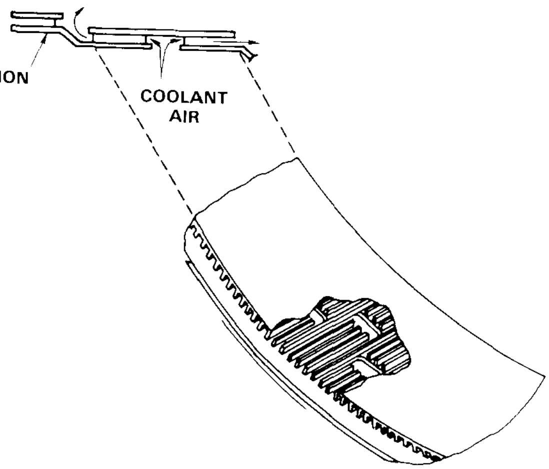

Fig. 18 Advanced Counter-Parallel FINWALL® Cooling System. Combustor liner has double wall construction for effective cooling.

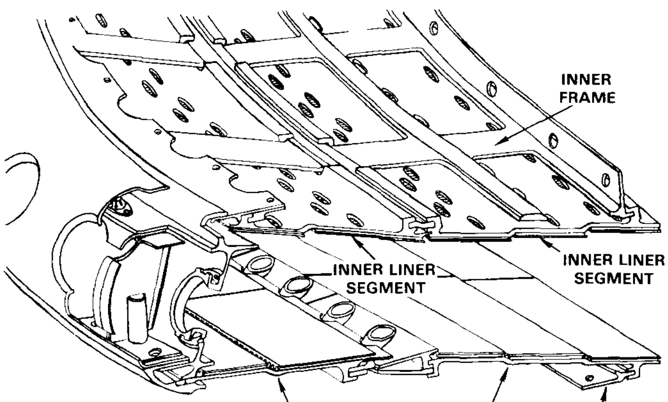

Fig. 19 Combustor Segments Supported By Framework. Liner is segmented for increased life.

TABLE VII

EPA PARAMETERS

. REQUIREMENTS ARE MET EXCEPT FOR NO_X

Pollutants	Goal	Status (W/Margins)
HC	0.4	0.2
CO	3.0	1.7
NOx	3.0	4.6
Smoke	20	20

Close attention was paid to reducing leakage. The overall design concept is one of minimizing the number of gaps which can produce air leaks. The vanes are individually restrained at both ends to allow the use of low-leak, full-ring, flexible metal seals at the attachments. Chordal cuts on the vane attachments eliminate a leakage path around the buttresses as the vanes tilt from differential inner and outer diameter axial thermal growth. Improved feather seals are used between vane platforms. Vane support rails are machined at assembly to avoid steps and attendant leakage paths. A fishmouth seal is placed between the vane inner platform and the disk, and feather seals are located between vane platforms to further reduce leak paths. Full ring sideplates on the disk cover the blade root attachment region and are sealed with "W" shaped seals.

TABLE VIII

HIGH-PRESSURE TURBINE DESIGN PARAMETERS

ONE STAGE DESIGN PROVIDES LOW DOC AND HIGH EFFICIENCY

Number of Stages	1
Pressure Ratio	4.0
Adiabatic Efficiency	0.882
Mean Velocity Ratio	0.56
NASA Work Factor	1.59
Mean Reaction Level	0.43
Maximum Rim Speed (Ft/Sec)	1730
Maximum AN2 (Ft^2 RPM^2)	46×10^9
Number of Airfoils	78

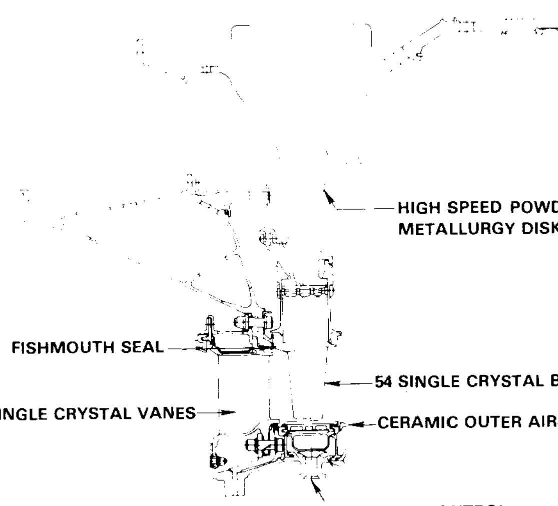

Fig. 20 High-Pressure Turbine Cross-Section. Turbine design requires new materials, advanced aerodynamics and effective sealing.

The high speed turbine disk (1730 ft./sec. rim speed) and sideplates will be made from MERL 80, a second generation powder metal nickel alloy. This alloy has high strength and long LCF life due to improved fracture mechanics characteristics. The vanes and blades are made from MERL 200, which is a second generation single crystal nickel alloy with a temperature capability of +100°F over B-1900 alloy. The turbine outer airseal is made from a metal backed ceramic material.

The vanes and blades are fabricated in two halves which permits casting-in sophisticated, high effectiveness internal cooling passages prior to bonding the halves together. The vane cooling system, shown on Fig. 21, includes three internal cavities for crossflow impingement internal cooling augmented by leading edge region coolant film holes from the forward cavity. The remainder of the coolant is exhausted through the pedestal trailing edge section. The selected blade configuration, shown in Fig. 22, is cast in two halves and bonded at the leading and trailing edge sections and along the ribs of the multipass internal cooling system. Front cavity air cools the leading edge section internally and is ejected through showerhead holes along the leading edge of the airfoil. Additional cooling air flows into the second cavity and traverses the length of the blade twice before exiting through a pedestal array in the trailing edge region. Internal trip strips are used to promote turbulence for increased heat transfer effectiveness.

The low number of cooled turbine airfoils is the major contributor to projected maintenance cost reduction of the E^3. Only 78 airfoils are cooled as opposed to 410 for the JT9D-7 two stage turbine. The E^3 high-pressure turbine design, with its high wheel speed and blade root stress together with increased blade loading, represents the practical minimum in the number of airfoils.

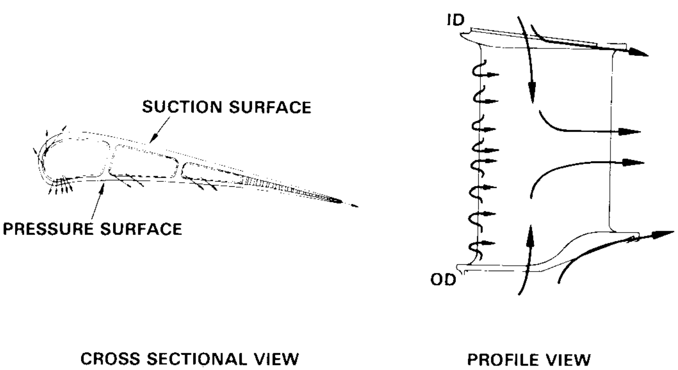

Fig. 21 High-Pressure Turbine Vane Cooling Configuration. Combination of convective, film and showerhead cooling techniques used to increase cooling effectiveness.

The aerodynamic improvements envisioned for the $E3$ turbine derive from the observation that efficiency could be increased by raising the reaction level, which lowers the vane exit Mach number and increases the expansion across the blade. In this design the reaction was raised until the rotor rearward pressure load reached a limiting value at 43 percent reaction. The combination of a large gaspath annulus, high rim speed, and high reaction level is calculated to increase turbine efficiency by 2 percent relative to current design practice.

Detailed vibration and stress analyses have been completed on the rotor system to assure structural adequacy of the detailed design. Component verification testing is scheduled for 1981.

CONCLUSIONS

The E^3 program has been formulated to address the concerns of increased fuel cost and decreased availability. A flight propulsion system has been designed to meet the specific goals set by NASA. Supporting technology programs are being conducted to verify many advanced design concepts. The detailed design of the core components is essentially complete and fabrication of the hardware has started. The program is on schedule towards testing the core in 1982 and the IC/LS in 1983.

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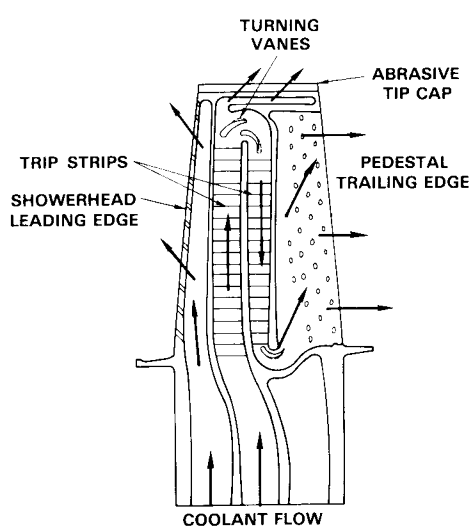

Fig. 22 High-Pressure Turbine Blade. Advanced multi-pass cooling system used to increase cooling effectiveness.