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EIGHTH SEMIANNUAL STATUS REPORT

1 October 1981 - 31 March 1982

ENERGY EFFICIENT ENGINE
COMPONENT DEVELOPMENT AND INTEGRATION PROGRAM

30 April 1982

Contract NAS3-20646

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Lewis Research Center
Cleveland, Ohio

PWA-5594-202

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FOREWORD

This contract effort is being conducted as part of NASA's Energy Efficient Engine Project. It is managed by the NASA-Lewis Research Center, with C.C. Ciepluch serving as the NASA Project Manager and J.W. Schaefer serving as NASA's Assistant Project Manager responsible for this contract.

This semiannual report covers the work performed under contract NAS3-20646 for the period of 1 October 1981 through 31 March 1982. It is published for technical information only and does not necessarily represent recommendations, conclusions, or the approval of NASA. The data generated under this contract are being disseminated within the United States in advance of general publication to accelerate domestic technology transfer. Since all data reported herein are preliminary information, they should not be published by the recepients prior to general publication by either the contractor or NASA.

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1.0 INTRODUCTION

The Energy Efficient Engine Component Development and Integration Program is currently being conducted under parallel contracts with General Electric and Pratt & Whitney. The Pratt & Whitney effort is funded under NASA Contract NAS3-20646. The program is under the overall direction of Mr. C.C. Ciepluch, who is assisted by Mr. J.W. Schaefer, NASA Project Manager for the Pratt & Whitney effort.

The objective of the program is to develop, evaluate, and demonstrate the technology for achieving lower installed fuel consumption and lower operating costs in future commercial turbofan engines. NASA has set minimum goals of a 12-percent reduction in thrust specific fuel consumption (TSFC), 5-percent reduction in direct operating cost (DOC), and 50-percent reduction in performance degradation for the Energy Efficient Engine (flight propulsion system) relative to the JT9D-7A reference engine. In addition, environmental goals for emissions (meet the proposed EPA 1981 regulation) and noise (meet FAR 36-1978 standards) have been established.

The Pratt & Whitney program effort is based on an engine concept defined under the NASA-sponsored Energy Efficient Engine Preliminary Design and Integration Studies Program, Contract NAS3-20628. This program was completed under an earlier low-energy consumption contract effort, and is discussed in detail in NASA Report CR-135396. The Pratt & Whitney engine is a twin-spool, direct drive, mixed-flow exhaust configuration, utilizing an integrated engine-nacelle structure. A short, stiff, high rotor and a single-stage high-pressure turbine are among the major features in providing for both performance retention and major reductions in maintenance and direct operating costs. Improved clearance control in the high-pressure compressor and turbines, and advanced single crystal materials in turbine blades and vanes are among the major features providing performance improvement.

To meet the program objectives, four technical tasks were established by the Pratt & Whitney Project Team and defined in the original Program Work Plan.

<u>Task 1, Propulsion System Analysis, Design and Integration</u> - provides for the preliminary design of the Energy Efficient Engine flight propulsion system and for evaluation of the propulsion system/aircraft integration with the assistance of Boeing, Douglas, and Lockheed.

<u>Task 2, Component Analysis, Design and Development</u> - consists of designing, fabricating, and testing the high risk components as well as supporting technology tests in critical areas. The task includes the designing of all components, plus a technology program to obtain design data on hollow fan blade test specimens; two builds of the high-pressure compressor; a full annular combustor and supporting programs to define diffuser parameters and combustor geometry for low emissions; a cooled high-pressure turbine rig and supporting technology programs in aerodynamics, leakage control, and blade fabrication; aerodynamic rigs supporting the design of a low-pressure turbine; and scale model mixer testing.

Task 3, Core Design, Fabrication and Test - provides the design, fabrication, and test of two builds of the core engine. The core consists of the high-pressure compressor, combustor, and high-pressure turbine. The test programs are structured to obtain aerodynamic and thermodynamic performance of the components and core. These test programs also evaluate the mechanical behavior of the structural design.

Task 4, Integrated Core/Low Spool Design, Fabrication and Test - consists of design, fabrication, and test of the fan, low-pressure compressor, low-pressure turbine, and mixer, all of which will be installed in a boiler plate nacelle and integrated with the core engine. The boiler plate nacelle will be acoustically treated and its lines will duplicate the internal flow lines of a representative flight nacelle. The integrated core/low spool will be tested to obtain aerodynamic and thermodynamic performance, component matching characteristics, and data on acoustic and emission characteristics. These tests will also evaluate mechanical behavior of the integrated core/low spool.

Several program changes were effected during the previous reporting period as a result of contract modification. The program work plan was revised in June 1981 to incorporate these changes and was subsequently approved by NASA. Relative to the previous work plan (February 1980), the most significant changes are:

- Deletion of the Task 3 effort and the addition of a second test to the Task 4 integrated core/low spool effort.
- Redefinition of the fan effort to replace the shroudless fan blade with a shrouded design and a redefinition of the Hollow Blade Supporting Technology Program to a fabrication feasibility effort.
- Addition of a third test to the High-Pressure Compressor Rig Program.
- Addition of a tangential on-board injection (TOBI) test to the high-pressure turbine component rig test effort and re-ordering of the component rig program to conduct the full stage testing first and then, if necessary, the annular cascade test.
- Deletion of the machining of one set of advanced combustor liner segments and liner supports.
- Addition of a benefit/cost study of potential fuel saving technologies in order to identify those suitable for follow-on technology programs.

Near the end of the current reporting period, federal government funding changes resulted in the deletion from the NASA contract task efforts associated with the integrated core/low spool. The relative impact of this to the on-going program effort is currently being assessed.

The program logic diagram in Figure 1* indicates the task schedules and the relationships between these tasks and their elements over the duration of the program.

Most of the work planned and approved from contract award through the end of the current reporting period (31 March 1982) has been completed. Exceptions are indicated in the appropriate technical progress sections of this report.

The remainder of this report presents background information and technical progress for each of the subtasks of Tasks 1, 2, and 4. The technical progress sections are appropriately divided to reflect (1) previously completed work that has an impact on the technical progress for the current reporting period, and (2) work accomplished during the current reporting period.

* For all program logic diagrams and work plan schedules presented in this report, the shaded region represents the current reporting period; '*M' denotes a major milestone; and '*D' denotes a key decision point.

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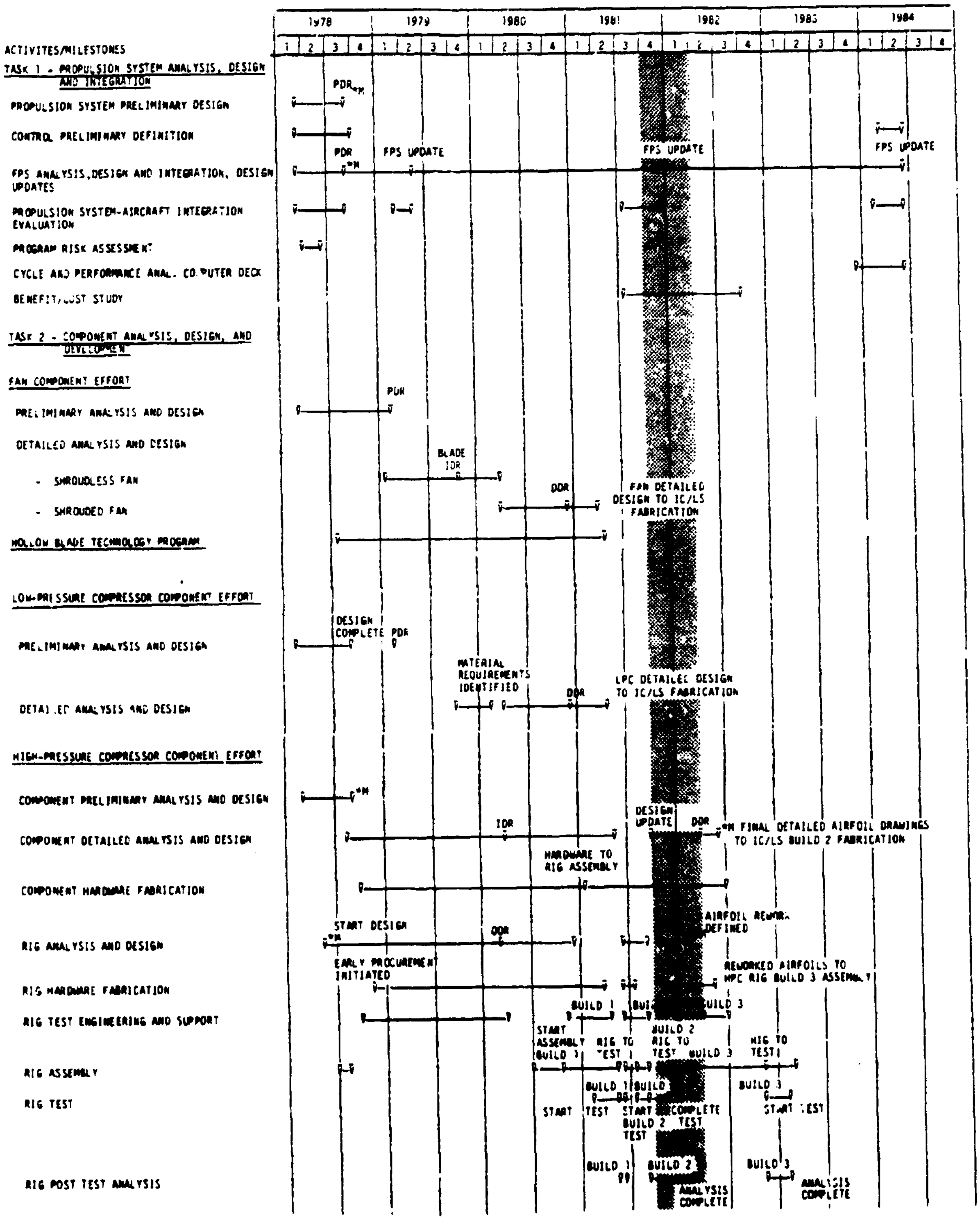

Figure 1 Overall Program Logic Diagram

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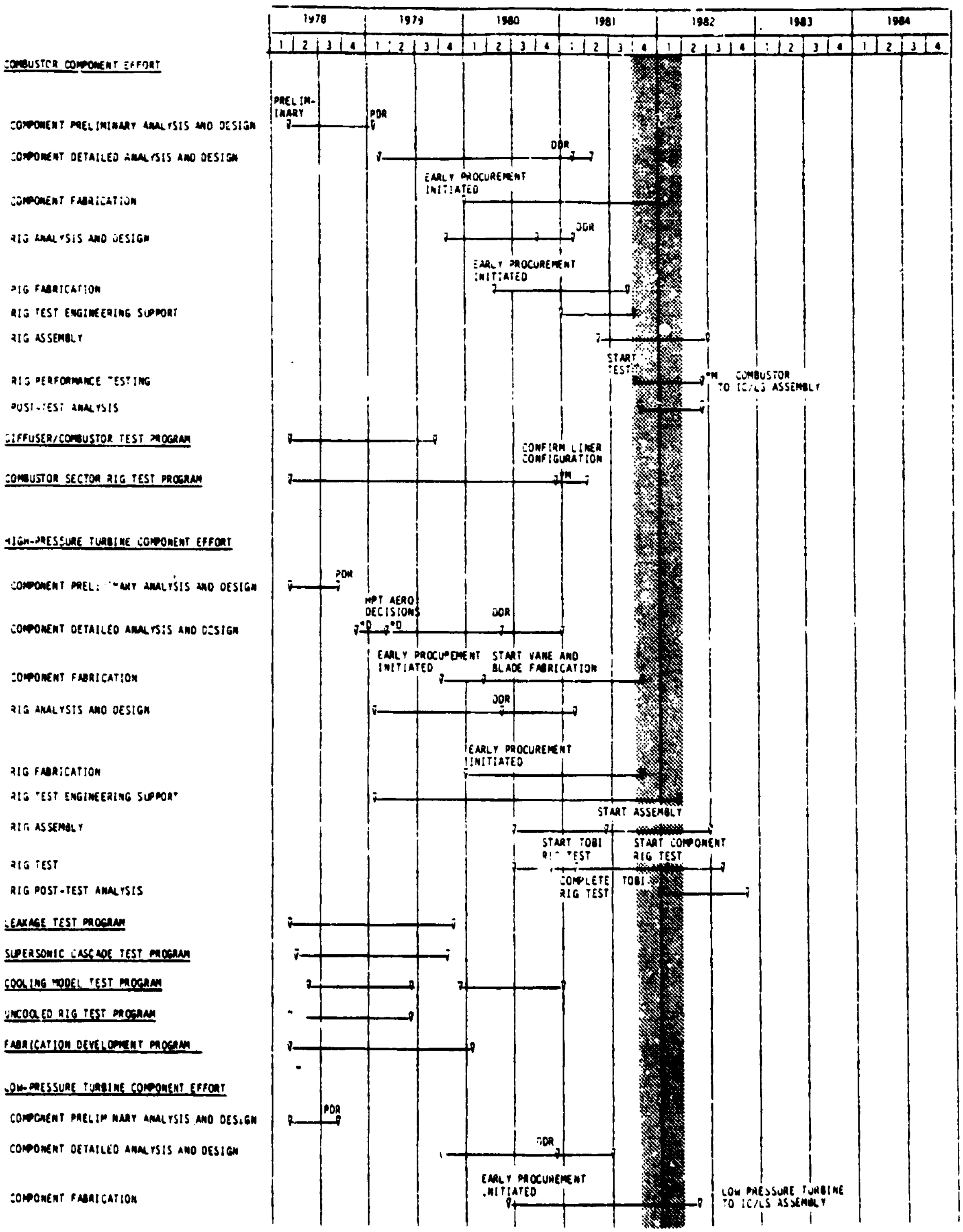

Figure 1 Overall Program Logic Diagram (continued)

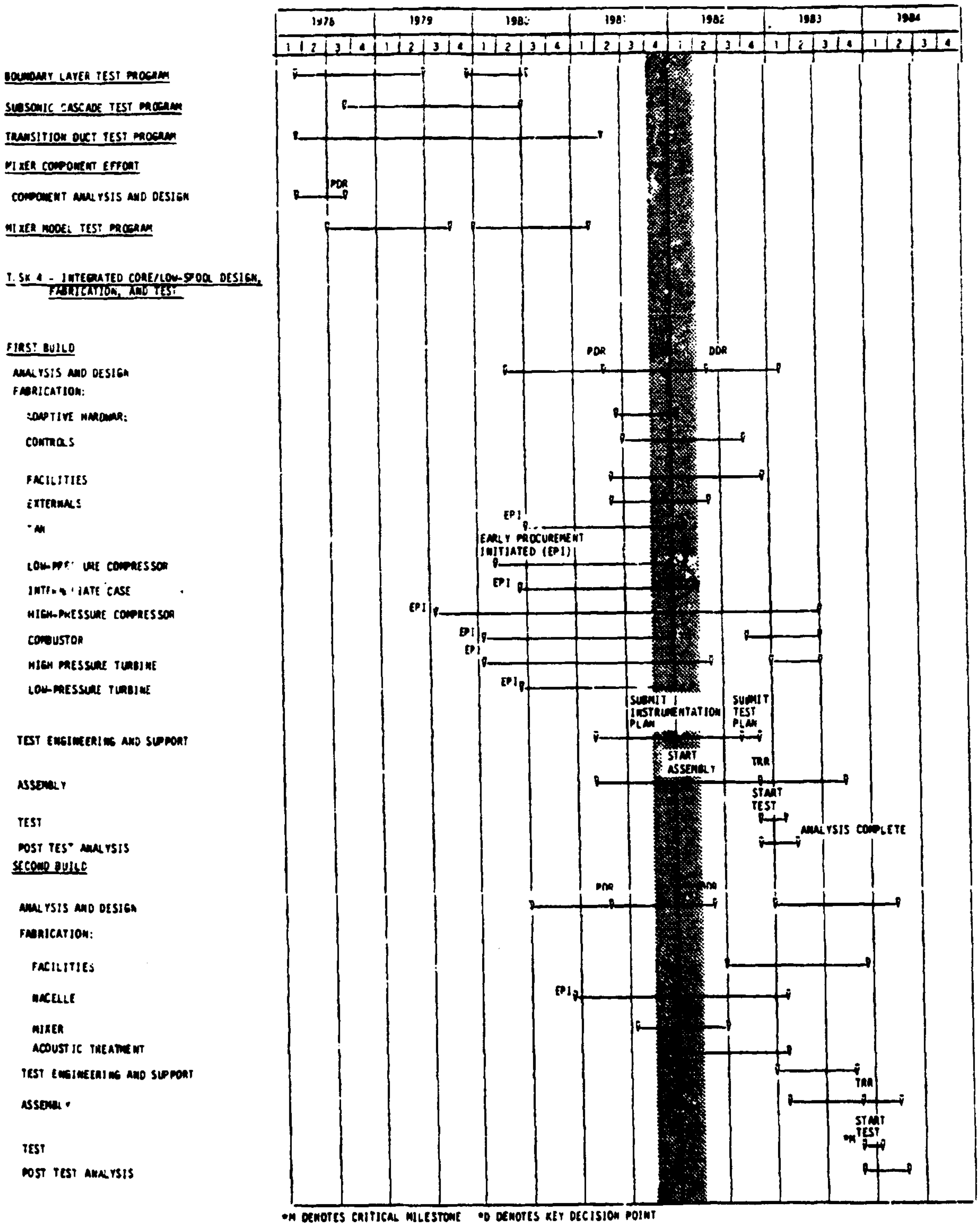

Figure 1 Overall Program Logic Diagram (continued)

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2.0 HIGHLIGHTS OF WORK ACCOMPLISHED

- A reference engine concept was defined and used as the baseline for assessing fuel-saving technology concepts as part of the benefit/cost study program. Evaluation of forty-three fuel-saving technologies was completed. As a result of this evaluation, twenty concepts were recommended to and approved by NASA for refined assessment.

- Testing of the high-pressure compressor rig (build 2A) was successfully completed, with a total accumulation of 256 hours of running time and an acquisition of 744 data points. Adiabatic efficiency was within five-tenths of a percentage point of the goal efficiency of 86.0 percent, while goals for other key parameters such as pressure ratio and airflow were met. A post-test inspection of the compressor rig showed that the component parts were in excellent condition. The high-pressure compressor rig (build 2A) test memo was completed and submitted to NASA.

- Testing of the full annular combustor rig (build 1) was successfully completed. Test results indicate emissions and aerothermal performance trends were similar to those demonstrated during the preceding Sector Combustor Rig Technology Program. These results were used to establish a performance and emissions baseline with which to compare data obtained from succeeding full annular and sector rig testing. Assembly and testing of combustor sector rig (build 22) was subsequently completed. Results from post test analyses indicate an unsatisfactory increase in fuel system carburetor tube airflows. Tube modifications are currently being made to reduce secondary (outer) airflow and increase velocity to provide the desired performance characteristics.

- High-pressure turbine rig test and instrumentation plans were finalized, printed and submitted to NASA. Following assembly and delivery of the rig to the stand, testing was initiated late in the report period. Design and off design performance data have been obtained and data reduction is currently in process.

3.0 TECHNICAL DISCUSSION

The following sections describe the scope of the total technical effort at the major task level. Work planned for the current reporting period is identified at the subtask level, and progress and results relative to this planned work are discussed in detail.

3.1 TASK 1 FLIGHT PROPULSION SYSTEM DESIGN

3.1.1 Overall Objective

Produce and maintain the flight propulsion system definition over the period of performance for the contracted work.

3.1.2 Task Overview

The definition of the flight propulsion system (1) forms the basis for assessing the capabilities of the flight propulsion system and integrated core/low spool (measured against program goals) and (2) establishes the design of the experimental hardware for Tasks 2 and 4.

The overall Task 1 effort is accomplished in seven subtasks: (1) propulsion system preliminary design, (2) control preliminary definition, (3) propulsion system analysis and design update, (4) propulsion system/aircraft integration evaluation, (5) program risk assessment, (6) cycle and performance analysis computer deck, and (7) a technology benefit/cost study. Figures 2 and 3 present the Task 1 logic diagram and work plan schedule, respectively.

The two major milestones of the Task 1 work plan schedule are (1) the flight propulsion system preliminary design review and (2) the propulsion system/aircraft integration evaluation. The first milestone is important because detailed design of the components cannot start until the preliminary design of the flight propulsion system is approved. Results of the propulsion system/aircraft integration initial evaluations provide the first major indications of the flight propulsion system capabilities measured against design goals.

Most of the work planned and approved from contract award through the end of the current reporting period (31 March 1982) has been completed. This included (1) completion of the flight propulsion system preliminary design and first design update (plus the companion effort associated with the propulsion system/aircraft integration evaluation*), (2) completion of the control preliminary definition, and (3) completion of the initial risk assessment. The

* Documented in NASA reports CR-159487 and CR-159488, respectively.

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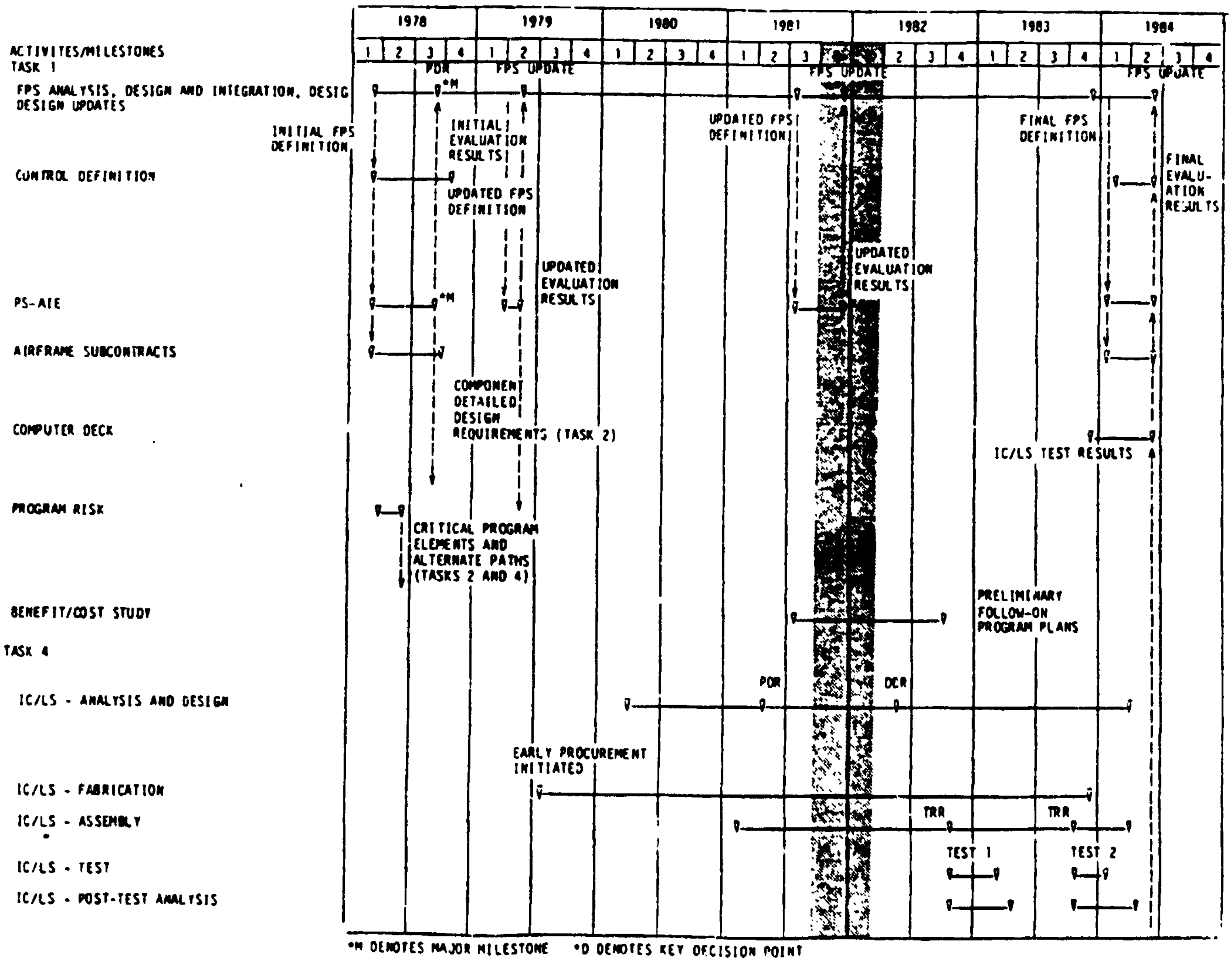


Figure 2 Task 1 Logic Diagram

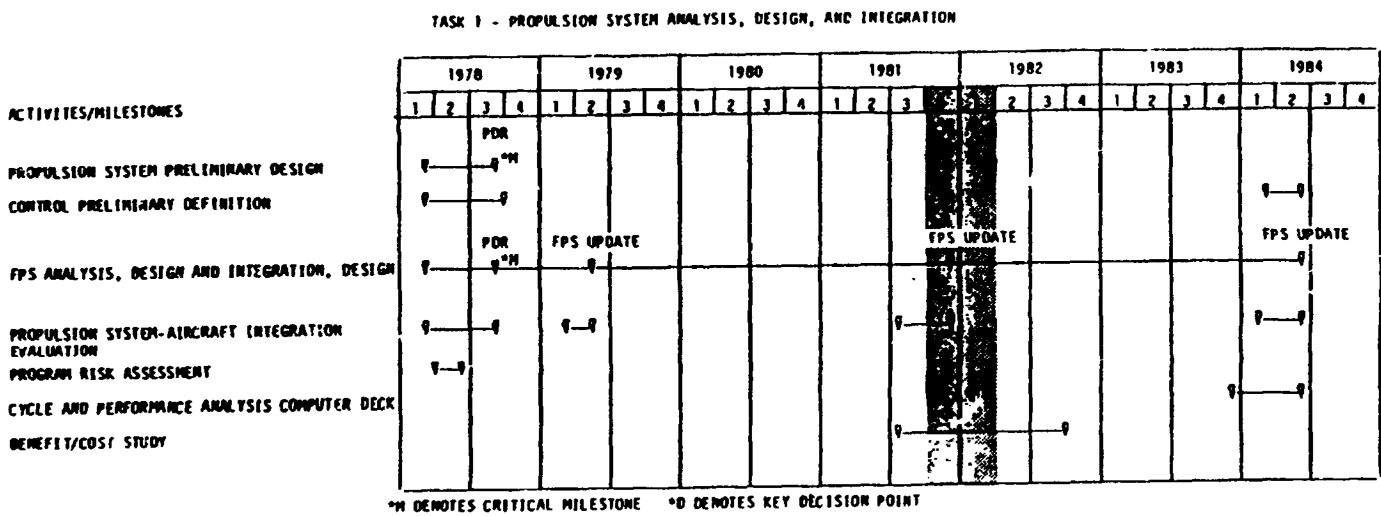
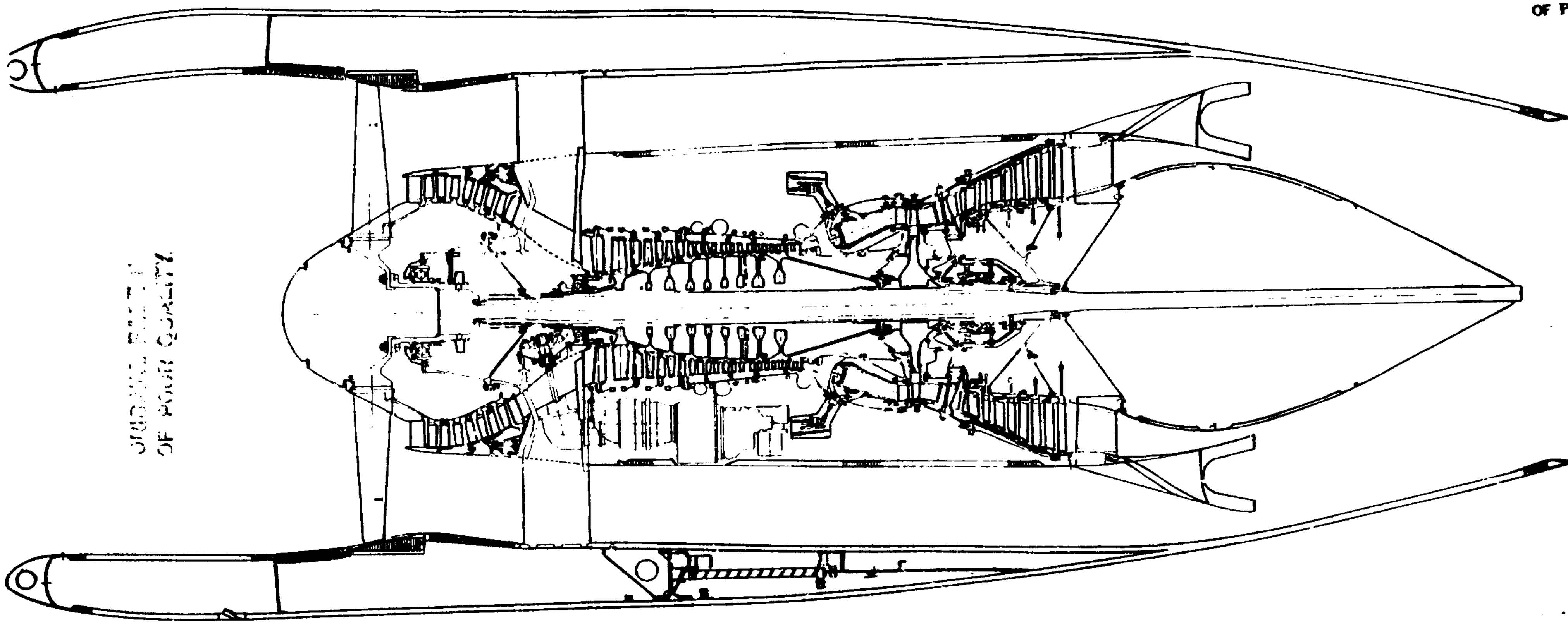


Figure 3 Task 1 Work Plan Schedule

flight propulsion system preliminary design and first design update demonstrated that the flight propulsion system can potentially meet NASA program objectives. The control preliminary definition established a full authority, digital electronic system as the primary concept for the flight propulsion system. The initial risk assessment identified the fan, high-pressure compressor, turbine, core, and integrated core/low spool as having the critical program paths that pace the program as scheduled.

Figure 3 identifies those tasks completed during the previous reporting periods and indicates that work on subtasks 3, 4 and 7 was scheduled to be performed during the current reporting period. However, the second flight propulsion system analysis and design update in subtask 3, scheduled to have been initiated during the current report period, has been postponed until a later date (Spring of 1982) at NASA request. Work on subtask 4 (PS/AIE) was also postponed at NASA request in order to be consistent with the revised timing for the second preliminary analysis and design update. Information detailing the current status for these subtasks is presented in the following sections.

The definition of the flight propulsion system and its components is periodically updated as program technical objectives are completed. The present propulsion system (see Figure 4) is a five-bearing design with two main support frames and two main bearing compartments. The fan features a single-stage, mid-part span shrouded blade to provide efficiency improvement. The low-pressure compressor utilizes concepts to control endwall loss and reduce airfoil loss levels. The high-pressure compressor similarly employs these low loss concepts. The high-pressure compressor operates at higher rotor speeds relative to the JT9D-7A high rotor for reduced weight and cost. It also incorporates an active clearance control system for improved efficiency. A two-stage combustor is utilized for low emissions. The high-pressure turbine features a single-stage design to provide a significant reduction in initial cost and engine maintenance cost. Single crystal alloy airfoils are used to reduce cooling and leakage flows. The high-pressure turbine also incorporates active clearance control to improve component efficiency. The low-pressure turbine counterrotates relative to the high-pressure turbine and incorporates active clearance control to increase component efficiency. The exhaust mixer is a scalloped design for reduced pressure loss, increased efficiency, and light weight. A full authority digital electronic control is used to promote efficient engine operation and reduce the effects of deterioration. The key nacelle features are an integrated engine-nacelle structure which improves engine performance retention by reducing engine deflections caused by thrust and cowl loads. The nacelle is constructed of composite and honeycomb materials for reduced weight and incorporates improved internal and external contouring and advanced sealing techniques for reduced losses.



FOLDOUT FRAME

Figure 4 · Energy Efficient Engine Cross Section

FOLDOUT FRAME

3.1.3 Propulsion System Analysis and Design Update

3.1.3.1 Objective

Continually review the predicted performance levels for the flight propulsion system and integrated core/low spool designs as test data are obtained from Tasks 2 and 4.

3.1.3.2 Scope of Total Work Planned

The propulsion system is updated at the completion of (1) the fan and combustor preliminary designs, (2) the detailed design reviews for the components and integrated core/low spool, and (3) the program. The final update includes cycle optimization for the flight propulsion system based on the overall program results. At the completion of the component and integrated core/low spool design efforts, and at the end of the program, a propulsion system preliminary design review is conducted at NASA-Lewis Research Center to cover the updated and revised analysis and design efforts.

3.1.3.3 Technical Progress

3.1.3.3.1 Summary of Work Previously Completed

The definition of the propulsion system and its components has been periodically updated as program technical objectives have been met. Table 3-I, not updated during the current reporting period, presents a comparison of the evolutionary status of the system performance for the flight propulsion system design with NASA program goals and the JT9D-7A reference engine. Also included in this table are results from the propulsion system/aircraft integration evaluations. Table 3-II lists the current flight propulsion system performance parameters at significant engine operating conditions.

TABLE 3-I

SUMMARY OF FLIGHT PROPULSION SYSTEM DESIGN EVALUATION

	NASA GOAL	PRELIMINARY DESIGN	PSAIE REP'T	FIRST FPS DESIGN UPDATE	STATUS - MAY, 1979	STATUS - OCT., 1979	STATUS - MARCH 1980	STATUS - JUNE 1981
TSFC Reduction*	12.0	14.9	14.9	14.9	14.9	14.7	15.1	15.0
DOC Reduction*								
Domestic Mission (Avg.)	5.0	7.7	7.6	7.2	7.1	6.5	6.7	6.6
International Mission (Avg.)	5.0	9.9	9.8	9.4	9.3	8.7	8.9	8.8
Noise	FAR 36 (1978)	FAR 36-2 to -4	**	**	**	**	-3 to -5	**
Emission								
Carbon Monoxide	3.0	2.0	2.0	1.7	1.7	1.7	**	**
Unburned Hydrocarbons	0.4	0.3	0.2	0.2	0.2	0.2	**	**
Oxides of Nitrogen	3.0	4.3	4.3	4.6	4 6	4.6	**	**
Reduction in Engine Weight*	-	8.6	7.6	2.5	1.3	-3.9	**	**
Reduction in Engine Cost*	-	5.9	4.7	1.0	1.4	-1.6	**	**
Reduction in Maintenance Cost*	-	6.2	4.6	4.2	4.7	2.4	**	**

*Relative to scaled JT9D-7A base engine
**Not updated

TABLE 3-II

CURRENT FLIGHT PROPULSION SYSTEM PERFORMANCE PARAMETERS

	Engine Operating Condition			
	Aero. Des. Point	Maximum Cruise	Maximum Climb	Takeoff
Altitude (ft)	35000	35000	35000	0
Mach Number	0.8	0.8	0.8	0
Ambient Temperature (°F)	-66	-66	-48	84
Net Thrust (Uninstalled) (lb)	9355	8935	9960	37025
Thrust Specific Fuel Consumption (lb/hr/lb)				
(Uninstalled)	0.550	0.548	0.570	0.327
(Installed)	0.576	0.575	0.596	0.330
Overall Pressure Ratio	38.55	37.35	40.25	31.05
Bypass Ratio	6.51	6.60	6.39	6.83
Fan Pressure Ratio (Duct Section)	1.74	1.71	1.78	1.58
High-Pressure Turbine Rotor Inlet Temperature (°F)	2235	2195	2410	2485

As part of the evolutionary design process, the propulsion system was resized to obtain the maximum technology benefit for smaller thrust engines expected to be required in the late 1980's. The inlet hub/tip ratio of the high-pressure compressor was also changed to improve aerodynamic performance. These changes are summarized in Table 3-III.

TABLE 3-III

1979 PROPULSION SYSTEM DESIGN CHANGES

	Original	Revised
Sea Level Static Takeoff Thrust (Uninstalled, lb)	41,100	36,200
Overall Pressure Ratio	38.6	No Change
Bypass Ratio	6.51	No Change
Fan Pressure Ratio	1.74	No Change
Turbine Rotor Inlet Temperature (°F) (at 84°F Day Takeoff Condition)	2,500	No Change
Exhaust System Configuration	Mixed Flow	No Change
High-Pressure Compressor Inlet Hub/Tip Ratio	0.63	0.56

3.1.3.3.2 Current Technical Progress

Current Flight Propulsion System Design

Current Materials

There were no updates during the reporting period to previously established listings of materials selected for both the flight propulsion system and integrated core/low spool. These materials, selected during previous reporting periods, are listed in Table 3-IV along with an '*' symbol indicating the materials changed since January 1981. An explanation of any differences in materials between the flight propulsion system and the integrated core/low spool is also provided. A material equivalency listing is presented in Table 3-V.

Performance Parameters and Detailed Drawings

The flight propulsion system cross section was revised during the report period. Major modifications to the cross section include (1) corrections to the number and arrangement of active clearance control pipes for the high-pressure compressor and (2) incorporation of the current configuration for the tailplug. The updated cross section is presented in Figure 5.

There were no updates during the reporting period to performance parameters or to previously established drawings showing active clearance control system, piping, and mount configurations.

System-Related Activities

Written confirmation to reschedule the second preliminary analysis and design update to the Spring of 1982 was received from NASA at the beginning of the report period. A planning update of subtasks to be conducted for the second flight propulsion system update was completed with the intention of conducting a Preliminary Design Review at the NASA Lewis Research Center in mid-May 1982. However, work planned for late in the report period was not initiated pending the possibility of a program redirection.

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TABLE 3-IV

ENERGY EFFICIENT ENGINE MATERIAL COMPARISON

	FPS	IC/LS	Rationale for IC/LS Difference
Fan			
Blade	AMS4928	AMS4928	
Disk	PWA1215	AMS4928	Schedule
Stubshaft	PWA733	PWA733	
Containment Case	AMS4150/Kevlar	AMS5062	Cost Saving
Sound Treatment	Al Honeycomb	None*	Cost Saving
LPC			
Blades	AMS4928	AMS4928	
Disks	AMS4928	AMS6414	Cost Saving
Hub	AMS4928	AMS6414	Cost Saving
Vanes			
S1	AMS4312	AMS5613	Aluminum is difficult to instrument
S2-S5	AMS4312	AMS4312	
Cases	AMS4150	AMS4312	Schedule
Intermediate Case			
Structural Struts	AMS4911	AMS4911	
Inner Case	PWA1262	AMS4928	Cost Saving
Non-Structural Struts	AMS4911	AMS4135*	Cost Saving
Other Case	AMS4150	AMS4135	Schedule

*Revised since January 1981

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TABLE 3-IV (continued)

	FPS	IC/LS	Rationale for IC/LS Difference
HPC			
Blades			
R6-R7	PWA1202	AMS4928	Schedule
R8-R15	PWA1010	PWA1010	
Disks			
R6-R7	AMS4928	AMS4928	
R8-R11	PWA1224	PWA1224	
R12-R13	PWA1225	PWA1226	Availability and Cost
R14-R15	MERL80	PWA1099	FPS Material Not Available
Non-Vortex Tubes	AMS4911	AMS4911	
Center Tube	AMS5613	AMS5613	
Vanes			
IGV	AMS4132	AMS5613	Aluminum is difficult to instrument
S6-S8	AMS5613	AMS5613	
S9-S12	AMS5508	AMS5616	Cost Saving and Schedule
S13-S14	AMS5596	AMS5662	Cost Saving and Schedule
EGV	PWA649	AMS5663	Cost Saving and Schedule
Front Case	AMS4928	AMS4928	
Rear Case	PWA1214	PWA1214	
IGV ID Shroud	AMS4132	AMS5613	For Compatibility with Vane

TABLE 3-IV (continued)

	FPS	IC/LS	Rationale for IC/LS Difference
Diffuser/Burner			
Diffuser			
Inner Prediffuser Wall	AMS5662	AMS5662	
Strut Assembly	PWA649 (HIP)	PWA649	Schedule
Burner			
Bulkhead	AMS5754	AMS5754	
OD Liner Segments	PWA1455	PWA1455	
OD Bird Cage	AMS5754	AMS5754	
ID Liner Segments	PWA1455	PWA1455	
ID Bird Cage	AMS5754	AMS5754	
HPT			
Rotor			
Blade	MERL200	PWA1480	[illegible]S Material Not Available
Disk/Hub	MERL80	PWA1099	FPS Material Not Available
Sideplates - FRT/RR	MERL80	PWA1099	FPS Material Not Available
Vortex Plate	MERL80	PWA1099	FPS Material Not Available
HPC Discharge Seal	AMS5895	AMS5895	

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TABLE 3-IV (continued)

	FPS	IC/LS	Rationale for IC/LS Difference
Static			
Vane S1	MERL200	PWA1480	FPS Material Not Available
OAS	PWA655/Ceramic	PWA655/Ceramic	
OAS Supports-FRT/RR	PWA1007	PWA1007	
TOBI System	PWA649/AMS5596	AMS5754	Cost Saving and Schedule
Outer Case	AMS5662	AMS5662	
Turbine Intermediate Case			
Hot Strut			
Aero Fairings	MERL200	PWA647	FPS Material Not Available
#4-5 Bearing Support	AMS5662	AMS5662	
Structural Struts	AMS5662	AMS5662	
LPT			
Rotor			
Blades R2	PWA1447	PWA1447	
Blades R3-R4	PWA655	PWA655	
Blades R5	MERL101	PWA655	FPS Material Not Available
Disks	PWA1099	PWA1099	
Spacers/Seals	PWA1099	PWA1099	
Hub	PWA1003	PWA1003	

TABLE 3-IV (continued)

	FPS	IC/LS	Rationale for IC/LS Difference
Static			
Vanes S2	MERL200	PWA1480	FPS Material Not Available
Vanes S3	PWA1447	PWA1455	Cost Saving
Vanes S4-S5	PWA655	PWA655	
Shrouds & Seals	AMS5536/AMS5754	AMS5536/AMS5754	
Inner Case	AMS5662	AMS5662	
Outer Case	AMS5858/AMS5895	AMS5858/AMS5895	
Exhaust Case			
ID/OD Case	MERL101	AMS5616	FPS Material Not Available
Struts	MERL101	AMS5354	FPS Material Not Available
LPT Shaft	PWA733	*AMS6304	Cost Saving
Mixer & Exhaust			
Mixer			
Mixer	PWA1231	AMS5599	Cost Saving
Mixer Support	MERL101	AMS5666	FPS Material Not Available
Exhaust			
Tailplug	AMS5599/AMS4910	AMS5599	Cost Saving
Center Vent Static	AMS5504	AMS5504	

*Revised since January 1981

TABLE 3-V

MATERIAL EQUIVALENCY

PWA 647	MAR-M509
PWA 649	Inconel 718
PWA 655	Inconel 713C
PWA 733	17-22-A; Templex (Low Alloy Steel)
PWA 1003	Incoloy 901
PWA 1007	Waspaloy
PWA 1010	Inconel 718
PWA 1099	Modified IN-100 Alloy (Formerly MERL 76)
PWA 1202	Titanium (8AL-1MO-1V)
PWA 1214	Titanium (6AL-2SN-4ZR-2MO) High Creep Strength
PWA 1215	Titanium (6AL-4V) Forged Below Beta Transus
PWA 1224	Titanium (6AL-2SN-4ZR-2MO) Forged Below Beta Transus
PWA 1225	Titanium (6AL-2SN-4ZR-2MO) Forged Above Beta Transus
PWA 1226	Titanium (6AL-2SN-4ZR-2MO) Forged, Beta Annealed, Precipitation Heat Treated
PWA 1231	Titanium (6AL-2SN-4ZR-2MO) Cross Rolled, Beta Annealed, Precipitation Heat Treated
PWA 1262	Titanium (6AL-4V) Cast
PWA 1447	MAR-M-247
PWA 1455	Modified B-1900
PWA 1480	Single Crystal NI Alloy
MERL 80	Modified IN-100 Alloy
MERL 101	Titanium Aluminide Alloy
MERL 200	Single Crystal NI Alloy

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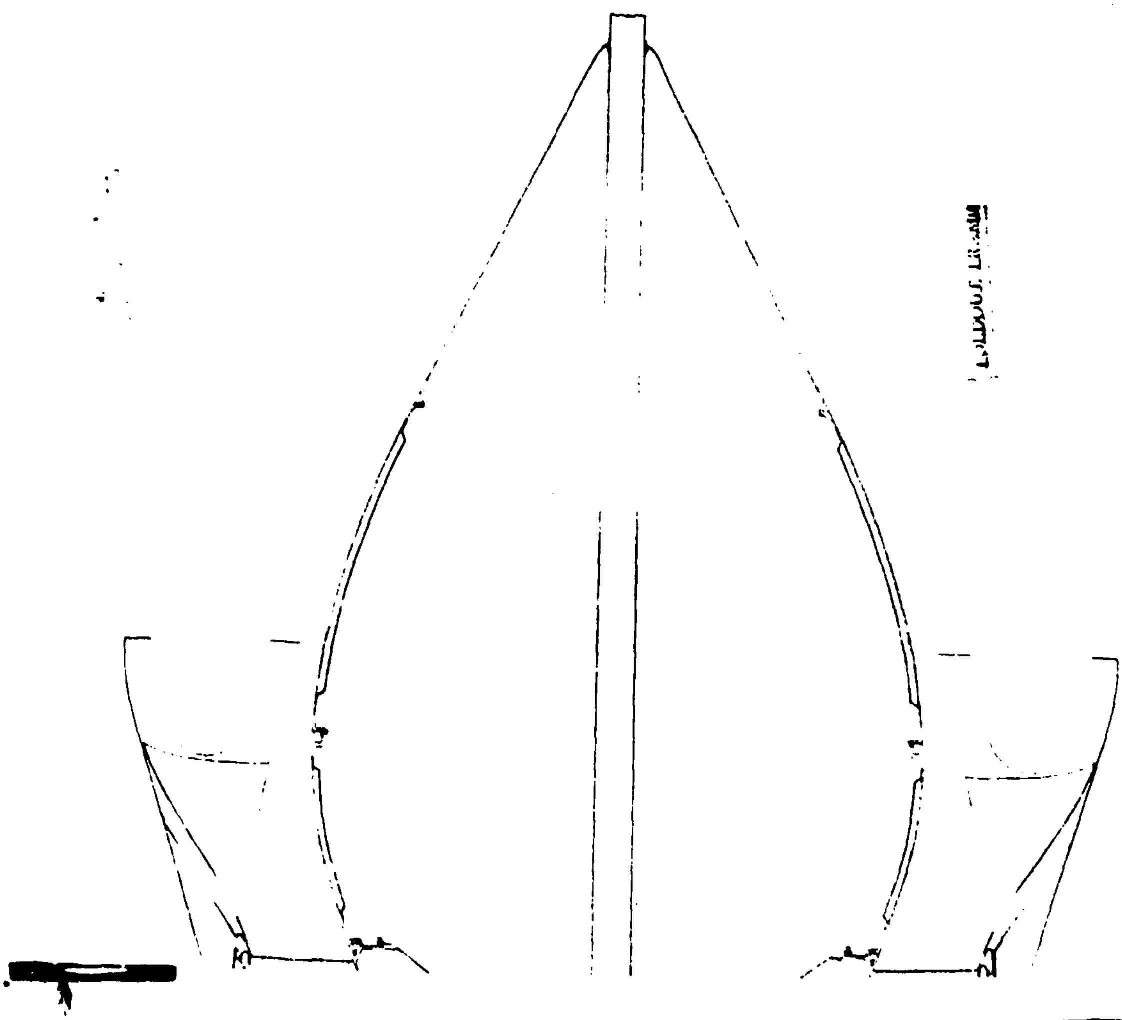
FOLDOUT FRAME

Figure 5   Updated Flight Propulsion System Cross Section



3.1.4 Propulsion System/Aircraft Integration Evaluation

3.1.4.1 Objective

Maintain a current measure of Energy Efficient Engine flight propulsion system performance against the program goals that reflect both the latest engine program status and the latest airframe technology.

3.1.4.2 Scope of Total Work Planned

This subtask assesses the ability of the flight propulsion system to meet the program design goals of thrust specific fuel consumption, direct operating cost, exhaust emissions, and noise. Engine/airplane operating economics, fuel burned, and integration requirements are also evaluated. Boeing, Douglas, and Lockheed assist in evaluating airplane performance and installation requirements for domestic and international aircraft. Three updates occur during the contract period concurrent with the three specific propulsion system designs scheduled in 1979, 1981, and 1983.

The propulsion system/aircraft integration evaluation procedure generally follows the plan of the proposed program. Boeing is studying only a domestic aircraft and may participate in the third update scheduled for 1983. Lockheed and Douglas are scheduled to participate in the initial evaluation and all subsequent updates. The division of work between Pratt & Whitney and the airplane companies remains unchanged from the proposed program.

3.1.4.3 Technical Progress

3.1.4.3.1 Summary of Work Previously Completed

The 1979 update of the propulsion system/aircraft integration evaluation (PS/AIE) was conducted at Pratt & Whitney without participation from Lockheed and Douglas. Values for fuel burned, direct operating cost, and return on investment were updated to reflect the engine design changes evolving from subtask 3, Propulsion System Analysis and Design Update. Pratt & Whitney estimated the effect of these engine changes on the Boeing, Douglas, and Lockheed airplanes, using Energy Efficient Engine airplane simulations and information from the PS/AIE reports of the airframe manufacturers (NASA CR-159488).

3.1.4.3.2 Current Technical Progress

Written approval of revised flight and economic performance ground rules for conducting the second update of the propulsion system/aircraft integration evaluation was received from NASA at the beginning of the report period. However, planning for this update was not initiated as intended pending the possibility of a program redirection.

3.1.5 Benefit/Cost Study

3.1.5.1 Objective

Identify advanced fuel-saving technologies, whose timing is beyond the scope of fuel-saving technology being developed in the current Energy Efficient Engine program, and incorporate these technologies into a preliminary design of the flight propulsion system.

3.1.5.2 Scope of Total Work Planned

To accomplish these objectives, the Benefit/Cost Study program was structured into four subtasks:

Subtask	Title and Purpose
1)	Benefit/Cost Study Ground Rules and Screening: Establish the basic ground rules to be used in the benefit/cost study as well as providing for the selection and initial screening of at least 30 candidate technology concepts. Following completion of the screening effort, twenty fuel-saving technology concepts will be recommended to NASA for approval to proceed with a refined assessment, as defined under subtask 2.
2)	Benefit/Cost Study Refined Assessment: Further evaluate the 20 concepts selected in subtask 1. Design and analysis efforts are conducted to obtain refined fuel savings, cost, and environmental characteristics. Technology development risk and probability of success assessments aid in ranking the concepts. Technology programs are defined, including the elements, schedule, cost, and testing requirements. The 20 technology concepts are ranked and at least 10 of the more promising concepts are recommended to NASA for approval of further evaluation under subtask 3.
3)	Integration of Benefit/Cost Study Concept into Engine System: Integrate the best concepts selected in subtask 2 into the Energy Efficient Engine propulsion system. A system cross section is prepared and analyzed for use in performance and weight estimates. System benefits are determined and a preliminary technology development program plan prepared.
4)	Benefit/Cost Study Reports: Prepare and submit oral and written status reports required by the contract.

3.1.5.3 Technical Progress

3.1.5.3.1 Summary Of Work Previously Completed

Effort directed toward the first subtask, Benefit/Cost Study Ground Rules and Screening, was initiated late in the previous report period with submittal of proposed study ground rules to NASA for review and approval.

3.1.5.3.2 Current Technical Progress

Reference Engine Definition -- A reference engine concept, representing the full performance potential with Energy Efficient Engine technology, was defined as a baseline for assessing fuel-saving technology concepts that extend beyond the scope of Energy Efficient Engine program technology. Engine overall pressure ratio and turbine inlet temperature levels of the flight propulsion system were held fixed to be consistent with Energy Efficient Engine materials and cooling technology. The bypass ratio was re-examined using $1.50/gallon in fuel cost to reflect the increase in cost from when the flight propulsion system cycle was selected in 1977 using a fuel price of 35-45 cents a gallon.

A comparison of the overall cycles selected for both the 1977 study and 1981 study is presented in Table 3-VI. The only difference is a 10 percent higher bypass ratio and corresponding lower fan pressure ratio in the 1981 study to improve performance.

TABLE 3-VI
SYSTEM CYCLE SELECTION
(35,000 ft, 0.8 Mn, Maximum Cruise)

	Flight Propulsion System (1977)	Benefit/Cost Study Ref. Engine (1981)
Bypass Ratio	6.5	7.2
Fan Pressure Ratio	1.74	1.65
Overall Pressure Ratio	38.6	38.6
Combustor Exit Temperature (deg-F)	2315	2315

In addition to a redefinition of the system cycle, configurational changes to the benefit/cost study reference engine were also made relative to the flight propulsion system to further improve performance. These modifications, identified in Figure 6, are listed below.

(1) A three inch increase in fan diameter which results in a higher bypass ratio and corresponding lower fan pressure ratio.

(2) A three inch reduction to the axial length of the high-pressure compressor component.

(3) Replacement of the two-zone (pilot and main) combustor with a single-zone combustor to reduce cost, weight and pressure loss.

(4) A two-stage high-pressure turbine component that provides an improvement in efficiency of two and one-half percentage points compared to the flight propulsion system single-stage high-pressure turbine.

(5) A five-stage low-pressure turbine that provides an efficiency improvement of one percentage point compared to the flight propulsion system four-stage low-pressure turbine.

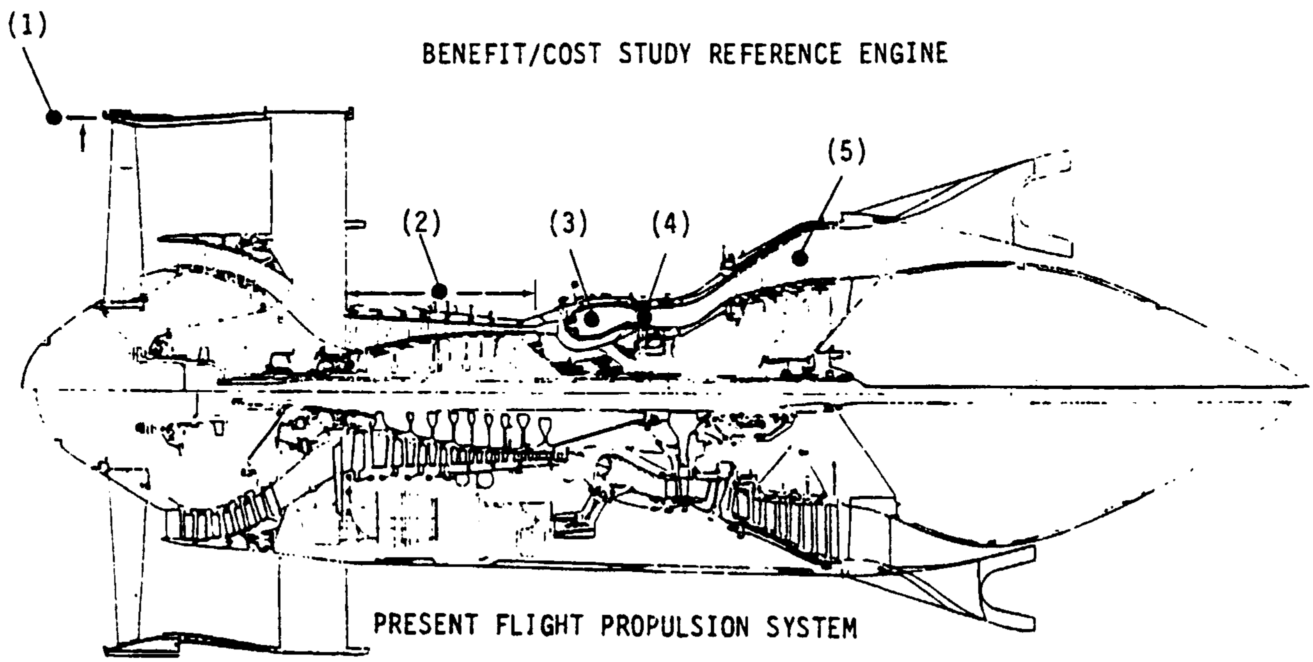


Figure 6 Benefit/Cost Study Reference Engine Indicating Configurational Differences Compared to the Current Flight Propulsion System

Comparative performance, weight, and cost estimates were made for the two engine configurations. These estimates were used to assess fuel burned and direct operating cost for a short range twinjet, a medium range trijet, and a long range quadjet. Study results, summarized in Table 3-VII, indicate a 5 percent reduction in fuel burned and a 0.9-2.5 percent reduction in direct operating cost relative to the Energy Efficient Engine flight propulsion system.

TABLE 3-VII

BENEFIT/COST STUDY REFERENCE ENGINE
STATUS RELATIVE TO FLIGHT PROPULSION SYSTEM

Performance, Weight and Cost Estimates:	
Thrust Specific Fuel Consumption (Installed) (%)	-4.7
Weight (Installed)* (lbm)	+230
Acquisition Cost (Installed)*	+$70K
Maintenance Cost (Installed)*	+$13/EOH**
Fuel Burned	
1500 Nautical Mile Twinjet	-5.1
3000 Nautical Mile Trijet (%)	-5.4
5500 Nautical Mile Quadjet (%)	-6.1
Direct Operating Cost ($1.50/gallon fuel cost)	
1500 Nautical Mile Twinjet	-0.9
3000 Nautical Mile Trijet (%)	-1.7
5500 Nautical Mile Quadjet (%)	-2.5

*Constant Installed Cruise Thrust
**Engine Operating Hour

Initial Screening of Technology Concepts -- Written approval of benefit/cost study ground rules, proposed in the last report period, was received from NASA early in the current report period. Forty-three concepts were selected and initially screened by evaluating potential fuel savings and direct operating cost reductions based on incorporation of these concepts into the benefit/cost reference engine configuration. Results from these evaluations, presented in Table 3-VIII, indicate the majority of concepts evaluated offer reductions to fuel burned and direct operating cost of between 0.1 and 0.5 percent. These relatively inconsequential differences make a justifiable definition of rankings impossible. Therefore, criteria used for selecting candidates for refined evaluation under subtask 2 were revised to include those concepts that: (1) offer fuel savings and direct operating cost reduction, (2) represent an evolutionary extension of current technology rather than an innovative design approach, and (3) are amenable to meaningful analysis. Using these criteria, concepts were selected as candidates for refined assessment, recommended to and approved by the NASA Project Office.

Initial effort directed toward refined assessment of the technology concepts, scheduled to begin during the report period, has been delayed by mutual agreement between NASA and Pratt & Whitney pending a decision on possible redirection of the benefit/cost study program.

TABLE 3-VIII

TECHNOLOGY BENEFIT/COST STUDY SCREENING SUMMARY

Concept Code	Concept Title	TSFC Reduction (%)	Fuel Savings % Block Fuel	Direct Operating Cost Reduction (%)
Fan				
F-1	Shroudless, Hollow Fan Blade	0.6	0.6	0.0
F-2	Tuned Fan Blade	0.6	0.3	0.2
F-3	Reduced Hub/Tip Fan	0.4	0.5	0.3
F-4	Swept Fan Blade	0.5	0.5	0.1
F-5	Fan Blade Clearance Adjustment	0.1	0.1	0.1
F-6	Fan Exit Guide Vane Endwall Suction	0.1	0.2	0.1
Compressor				
C-1	Radial Work Endwall Improvement	0.2	0.2	0.1
C-2	2nd Generation Controlled Diffusion Airfoils	0.2	0.2	0.2
C-3	Pressurized Inner Seal Cavities (Dropped From Study)			
C-4	Variable Compression (Alt. Cycle Changer Simpler)			
C-5	Centrifugal Compressor	0.0	---	---
C-6	Integrated Exit Guide Vane/Strut (Deferred to Subtask 2)			
Combustor				
CB-1	Mark 4 Combustor	1.2	1.4	0.7
CB-2	Advanced Segmented Liner	1.0	1.2	1.0
High-Pressure Turbine				
HT-1	Leaned/Bowed Vanes	0.5	0.6	0.4
HT-2	Increased AN^2 (annulus area times wheel speed (disk rpm))	0.4	0.4	0.2
HT-3	Increased Efficient Blade Cooling	0.1	0.1	0.1
HT-4	Airfoil Thermal Barrier Coating	0.3	0.4	0.3
HT-5	Single Crystal-1000 Vane with PS200 Coating	0.3	0.4	0.3
HT-6	Single Crystal-2000 Vane with PS200 Coating	0.1	0.1	0.1
HT-7	Single Crystal-3000 Vane with PS200 Coating	0.1	0.1	0.1
HT-8	Fabricated Vane	0.0	0.0	0.2

TABLE 3-VIII (continued)

TECHNOLOGY BENEFIT/COST STUDY SCREENING SUMMARY

Concept Code	Concept Title	TSFC Reduction (%)	Fuel Savings % Block Fuel	Direct Operating Cost Reduction (%)
Low-Pressure Turbine/Exhaust Mixer				
LT/M-1	Improved Mixer	0.2	0.2	0.1
Air Management				
A-1	Modulated TOBI System	0.1 (Cruise)	0.0	0.0
A-2	Modulated Combustor Air	0.1 (Cruise)	0.0	0.0
A-3	Modulated Vane Cooling Flow	0.4 (Cruise)	0.2	0.0
A-4	Radial Flow TOBI	0.0	0.0	0.0
A-5	Optimized Customer Bleed	0.7	0.9	0.5
A-6	Closed Loop Active Clearance Control	0.5 (Above 20,000 ft)	0.4	0.1
A-7	Precooled Turbine Cooling Air with Fuel Coolant	0.4	Deleted because of coking potential	
A-8	Improved Low-Pressure Turbine Active Clearance Control	0.2	0.3	0.3
Installation				
I-1	Low Pressure Loss Duct	0.2	0.2	0.1
I-2	Low Isolated Drag Nacelle	0.7	0.9	0.6
I-3	Low Interference Drag Instal.	3.0	3.6	2.2
I-4	Nacelle Vent Thrust Recovery	0.2	0.2	0.1
I-5	Engine Torque @ Front Mount (Deferred to Subtask 3)			
I-6	Variable Jet Area	0.1	0.2	Unknown
I-7	All Electric Power Extraction	1.5	1.8	1.1
Structures/Mechanics				
S/M-1	Composite Fan Cases	0.0	0.1	0.1
S/M-2	Composite Intermediate/Fan Exit Case	0.0	0.2	0.1
S/M-3	Integrated Fan Containment/ Nacelle Inlet	0.0	0.1	0.05
S/M-4	Composite Core Cowl	0.0	0.1	0.0
S/M-5	High Efficiency Reduction Gear	3.0	3.1	1.6

3.2 TASK 2 COMPONENT TECHNOLOGY

3.2.1 Overall Objective

The overall objectives for Task 2 are to: (1) establish preliminary component configurations, (2) conduct supporting technology programs to evaluate Energy Efficient Engine concepts, (3) produce component detailed designs, and (4) evaluate the Energy Efficient Engine high-pressure compressor, combustor, and high-pressure turbine in full-scale component rigs.

3.2.2 Task Overview

The Task 2 effort focuses on the design, fabrication, and testing of the major components to be used in the Task 4 integrated core/low spool experimental verification program. In addition, the results of Task 2 testing are fed into the flight propulsion system analysis and design updates of Task 1. Specific performance goals for these components are shown in the subsequent component effort sections of this report.

The preliminary component designs are based largely on results from the Energy Efficient Engine Preliminary Design and Integration study (NAS3-20628) combined with results of other government and Pratt & Whitney related programs. There are areas where additional evaluation of Energy Efficient Engine concepts is necessary before committing to the Energy Efficient Engine detailed design. In these areas, supporting technology programs provide that evaluation in a timely manner. The detailed component designs are accomplished as an extension of the preliminary component designs, reflecting supporting technology program results, as applicable, and Task 1 input.

Preliminary component designs are 'flight' designs and support the propulsion system preliminary design effort of Task 1. Systems (lubrication, breather, thrust balance, and active clearance control) are worked jointly between Tasks 1 and 2 during the preliminary design phase. A detailed design of the exhaust mixer is not accomplished under Task 2. Instead, a test mixer detailed design is provided as part of Task 4.

Program fabrication schedules are stringent, and certain constructions require early starts. In general, raw material is ordered as early as rough shapes can be defined, thus ensuring material availability at the time detailed drawings are completed. As hardware definition becomes known during the detailed design phase, those parts requiring early fabrication are identified and permission to proceed is requested from NASA.

The program logic diagram is shown in Figure 7, and the work plan, in Figure 8. The logic diagram shows the relationship of the supporting technology programs to the component effort. Specific logic diagrams for each component are shown in subsequent sections.

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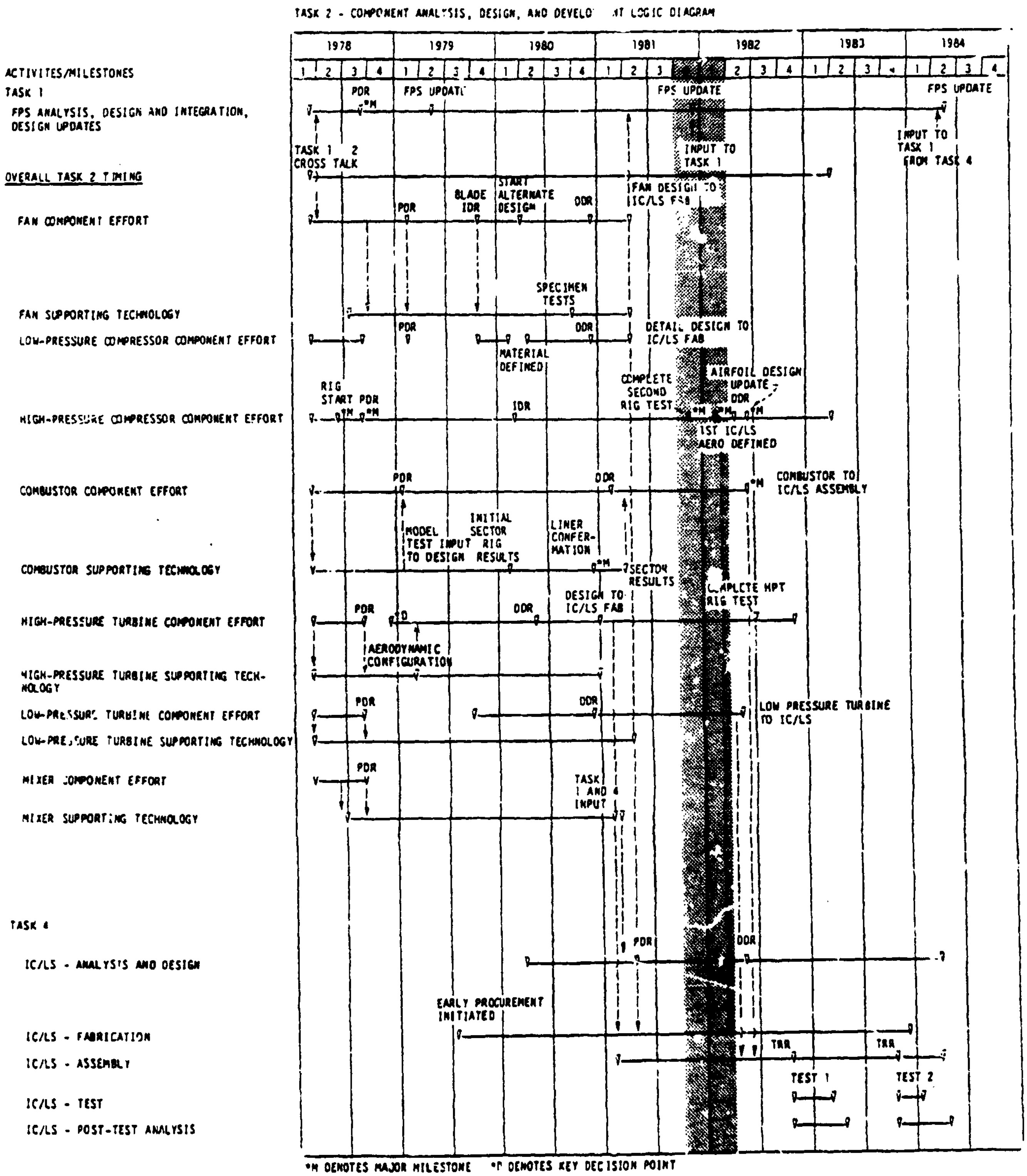

Figure 7 Task 2 Logic Diagram

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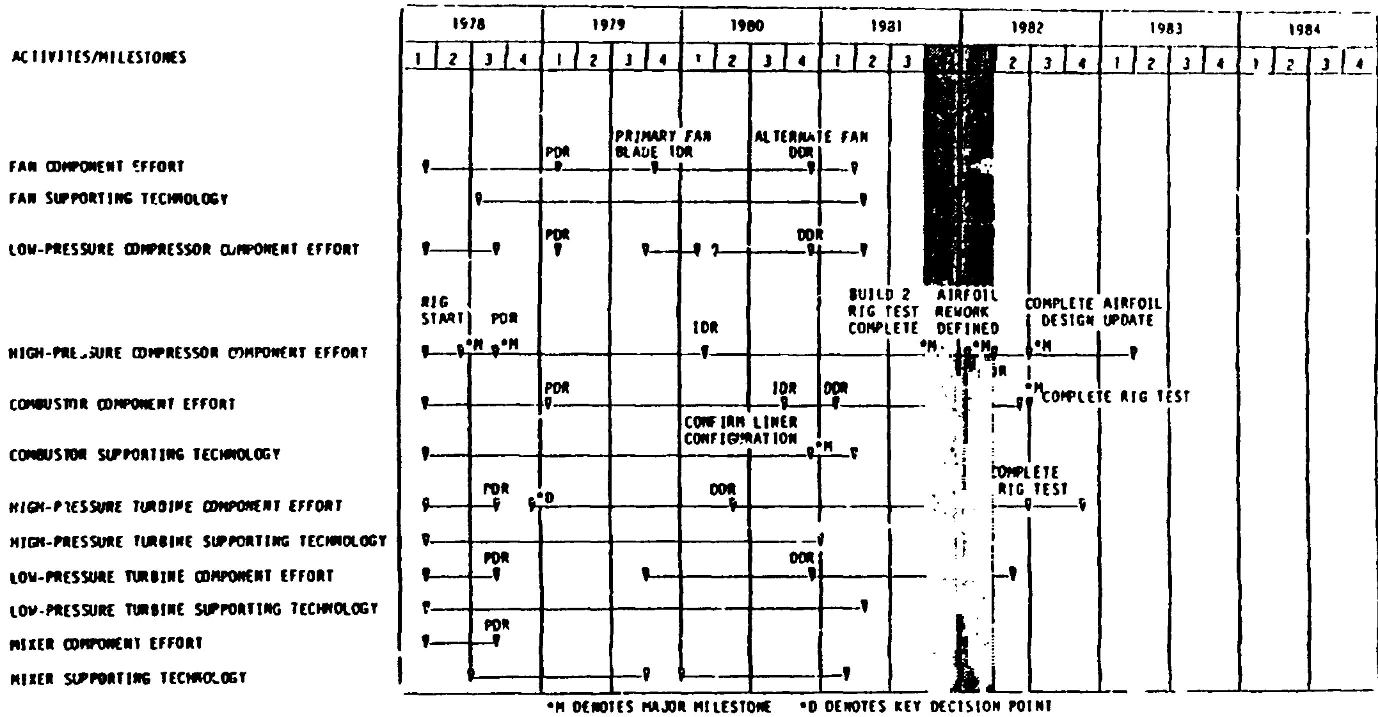

Figure 8 Task 2 Work Plan Schedule

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Critical Milestones

(1) High-Pressure Compressor:

(a) Complete first high-pressure compressor rig test.

(b) Complete high-pressure compressor airfoil design update.

(c) Define the high-pressure compressor airfoil rework for the core.

(2) Diffuser/Combustor:

(a) Confirm the combustor liner configuration.

(b) Complete the annular combustor rig test.

(3) High-Pressure Turbine:

(a) Complete high-pressure turbine rig testing.

Most of the work planned and approved from contract award through the end of the current reporting period (31 March 1982) has been completed. Exceptions are indicated in the appropriate technical progress sections of this report. Figure 8 identifies tasks that were completed during the previous reporting periods. It also identifies tasks which were initiated, continued, or completed during the current reporting period. The component discussions that follow describe this work in more detail.

Major program changes affecting Task 2 include (1) elimination of the scaled fan supporting technology program, (2) transfer of the shrouded fan analysis and design effort from Task 4 to Task 2, (3) transfer of the shroudless blade fabrication effort from Task 4 to the TRW subcontract effort in Task 2, (4) addition of a tangential on-board injection rig test to the High-Pressure Turbine Rig Test Program, (5) reduction of the Hollow Blade supporting technology program to a fabrication feasibility effort, (6) addition of a third test to the High-Pressure Compressor Rig Test Program, and (7) deletion of the machining of one set of advanced combustor liner segments and liner support frames.

3.2.3 Fan

Fan program effort has been redirected to place more emphasis on design and fabrication of the shrouded fan design. This redirection was necessary when it became apparent that delays in fabrication of the shroudless, hollow blades precluded their availability in time for integrated core/low spool testing in Task 4. The objectives and scope of effort reflect this change in emphasis.

3.2.3.1 Overall Objective

The primary objective of this effort is to design a single stage, aft part span shroud fan blade component for use in the integrated core/low spool in Task 4. The fan is designed to produce a pressure ratio of 1.74 outer diameter/1.56 inner diameter with a goal flight propulsion system (FPS) adiabatic efficiency of 86.3 percent. The aspect ratio of the blade is 4.0. Experimental fan component expected efficiency for the integrated core/low spool is 84.7 percent.

A secondary objective of the fan component effort is to design a single stage fan which utilizes a shroudless hollow titanium 2.5 aspect ratio fan blade and to explore the feasibility of fabricating such an airfoil by 1) lamination of multiple titanium sheets, and 2) the superplastic forming/diffusion bonding technique.

Figure 9 shows the relationship between program activities and contract Tasks 1 and 4. The preliminary and detailed design phases provide design input to the blade fabrication effort in the Hollow Blade Technology Program.

3.2.3.2 Scope of Total Work Planned

The fan component effort is initiated with the shroudless fan blade preliminary design which consists of a twelve month design effort to establish the feasibility of this design concept and provide configuration definition to the supporting technology program. This design phase provides a layout drawing of the fan component, a fan blade fabrication approach, and a substantiating design data package presented to NASA at a Preliminary Design Review (PDR) in February 1979.

Immediately following NASA approval of the preliminary design, an eight month detailed design of the shroudless fan blade is undertaken to provide the Hollow Blade Technology program with a blade design to be fabricated under a TRW subcontract. Moreover, a fabrication feasibility study is conducted under subcontract with Rockwell International to explore the suitability of employing the superplastic forming/diffusion bonded technique for fabrication of hollow, shroudless fan blades.

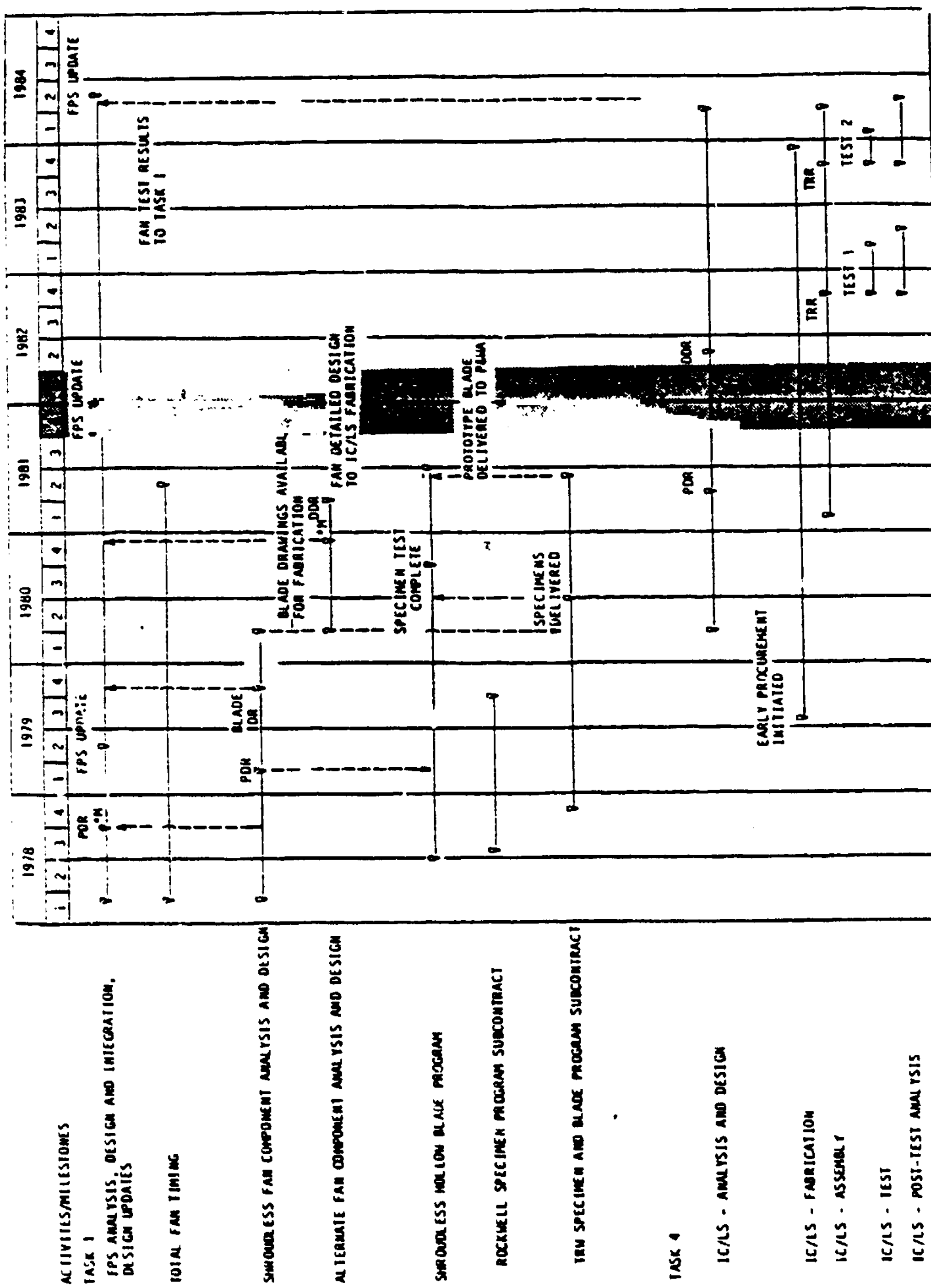

Figure 9 Fan Program Logic Diagram

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The shrouded fan detailed design effort is accomplished in 1980 to allow sufficient fabrication time for incorporation in Task 4. The results of this detailed design effort are presented to NASA at a Detailed Design Review in December 1980. Shrouded fan hardware fabrication is accomplished as part of the Task 4 effort. Figure 9 indicates that all technical effort associated with the fan component design program is complete. Program results are reported in NASA CR-165466.

3.2.3.3. Supporting Technology

3.2.3.3.1 Scaled Fan Rig Test Program

Pratt & Whitney and NASA mutually agreed to delete the scaled fan supporting technology effort from the overall program because (1) results of rig testing would not be available to affect the final shroudless blade design and (2) re-assessment of the balance between available contract funds and cost of technical work planned for the remainder of the program indicated that the technical effort would have to be reduced.

3.2.3.3.2 Hollow Blade Technology Program

All technical work for this supporting technology program has been completed. Program results appear in NASA Report CR-165586.

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3.2.4 Low-Pressure Compressor

3.2.4.1 Overall Objective

Design a four-stage low-pressure compressor with a design pressure ratio of 1.77 and an adiabatic efficiency of 89.9 percent. The corresponding expected efficiency for the low spool component of the experimental integrated core is 87.5 percent. Additional design goals are an inlet flow of 142.1 lb/sec, a surge margin of 20 percent, and a life of 20,000 missions and 30,000 hours.

3.2.4.2 Scope of Total Work Planned

The program consists of (1) a preliminary analysis and design phase that determines the feasibility of the low-pressure compressor design, and (2) a detailed analysis and design phase that completes the compressor design for use in the integrated core/low spool (Task 4). There is no component rig program or supporting technology program. The design data and the verification of advanced concepts are obtained principally from related Pratt & Whitney programs such as an in-house supercritical cascade program, the NAVAIR Supercritical Cascade Test, and the NASA Front Stage Program (Contract No. NAS3-20899). Low-pressure compressor hardware for the low spool portion of the integrated core/low spool phase is fabricated in Task 4. As shown in Figure 10, the preliminary design effort starts at the beginning of the contract in support of Task 1. The results are presented in a preliminary design review in February 1979. The low-pressure compressor detailed analysis and design begins in October 1979.

Following the acceptance by NASA of the low-pressure compressor detailed design, the low-pressure compressor component is fabricated and tested in the Task 4 integrated core/low spool program.

3.2.4.3 Technical Progress

All technical work for Task 2 component analysis and design, as shown in Figure 10, has been completed. Results from this program effort appear in NASA Report CR-165354.

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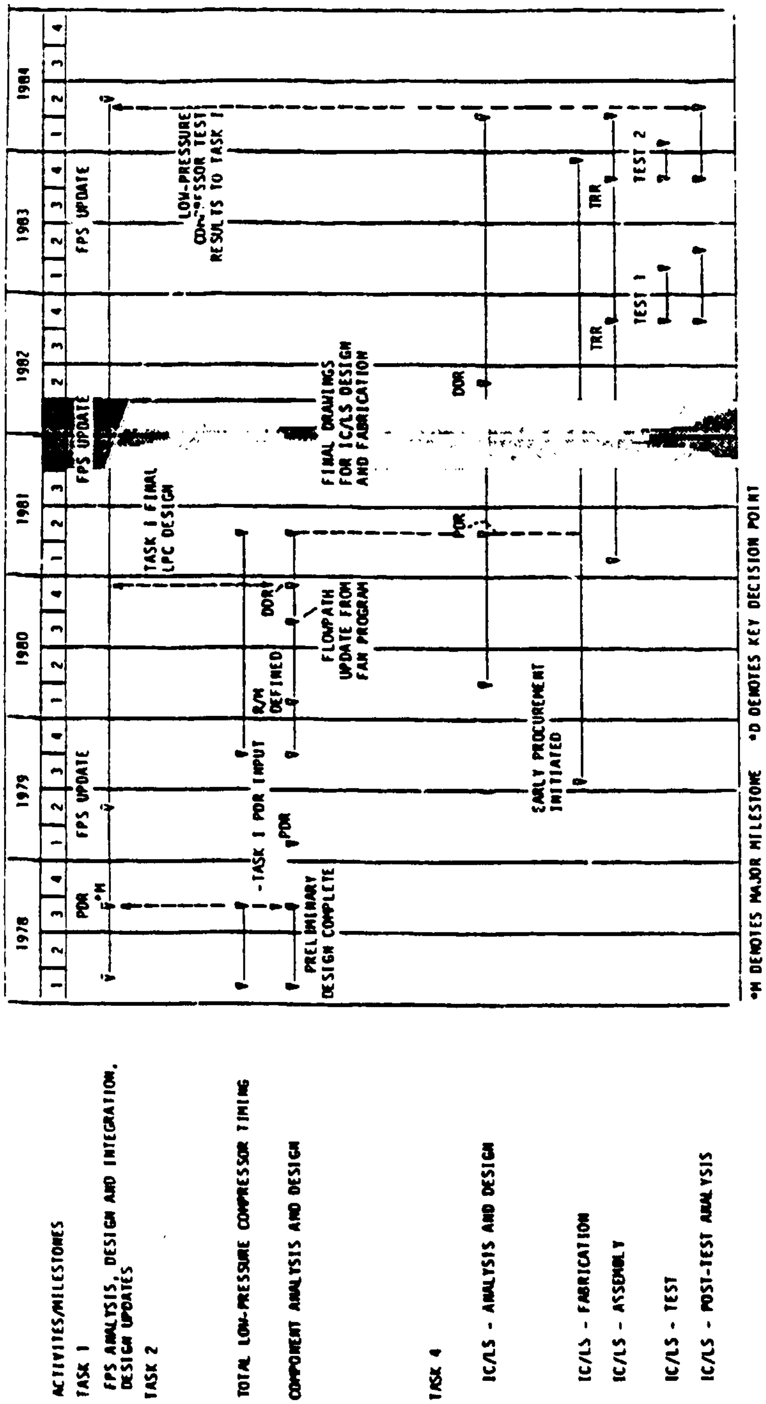

Figure 10 Low-Pressure Compressor Program Logic Diagram

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3.2.5 High-Pressure Compressor

3.2.5.1 Overall Objective

Design a ten-stage, high-pressure compressor with a pressure ratio of 14:1, an adiabatic efficiency of 88.2 percent and an average blade aspect ratio of 1.5. The efficiency expected for the high-pressure compressor component in the integrated core/low spool is 86.5 percent. Additional design goals are an inlet corrected flow of 77.5 lb/sec, a surge margin of 20 percent, and life of 20,000 missions and 30,000 hours.

3.2.5.2 Scope of Total Work Planned

The program consists of four major efforts. The first is the preliminary analysis and design phase, in which the feasibilility of the compressor design is determined. Next is the detailed analysis and design phase to provide the hardware design for both the high-pressure compressor rig program and integrated core/low spool program. Fabrication of nonrotating and rotating hardware for the high-pressure compressor rig program is the third major effort. The fourth is the high-pressure compressor component rig program, which is aimed at verifying and optimizing the compressor design.

Figure 11 shows the relationship between the elements of this task and contract Tasks 1 and 4. As shown, the program begins with the preliminary design activity, which provides design input to the high-pressure compressor component rig program and to Task 1 as well as to the detailed design activity that immediately follows. Layout drawings and substantiating data evolving from this preliminary design effort were presented to NASA for approval at a preliminary design review in September 1978. Results from the detailed design activity were presented for NASA approval at a detailed design review in February 1980. Component and rig hardware is fabricated simultaneously. All hardware is transferred to the rig program in October 1980 for assembly in the test rig. Upon completion of analysis of the second build test data, the airfoil designs are updated, as required, to optimize the compressor design. The resultant airfoil requirements are utilized for hardware fabrication and transferred to Task 1 for the final flight propulsion system update. The third rig test uses the reworked airfoils and evaluates the performance of the optimized compressor design.

The critical milestones for the high-pressure compressor effort are shown in the work plan schedule presented in Figure 12. As shown in this figure, the following task efforts have been completed to date: (1) preliminary design of the component; (2) detailed design of both the component and rig; (3) rig build 1 assembly, test, and post-test analysis activities; and (4) build 2 test and assembly activities. Component fabrication efforts are continuing, as are build 2 post-test analysis efforts. Component rig (build 3) efforts have been delayed pending results of build 2 post-test analysis.

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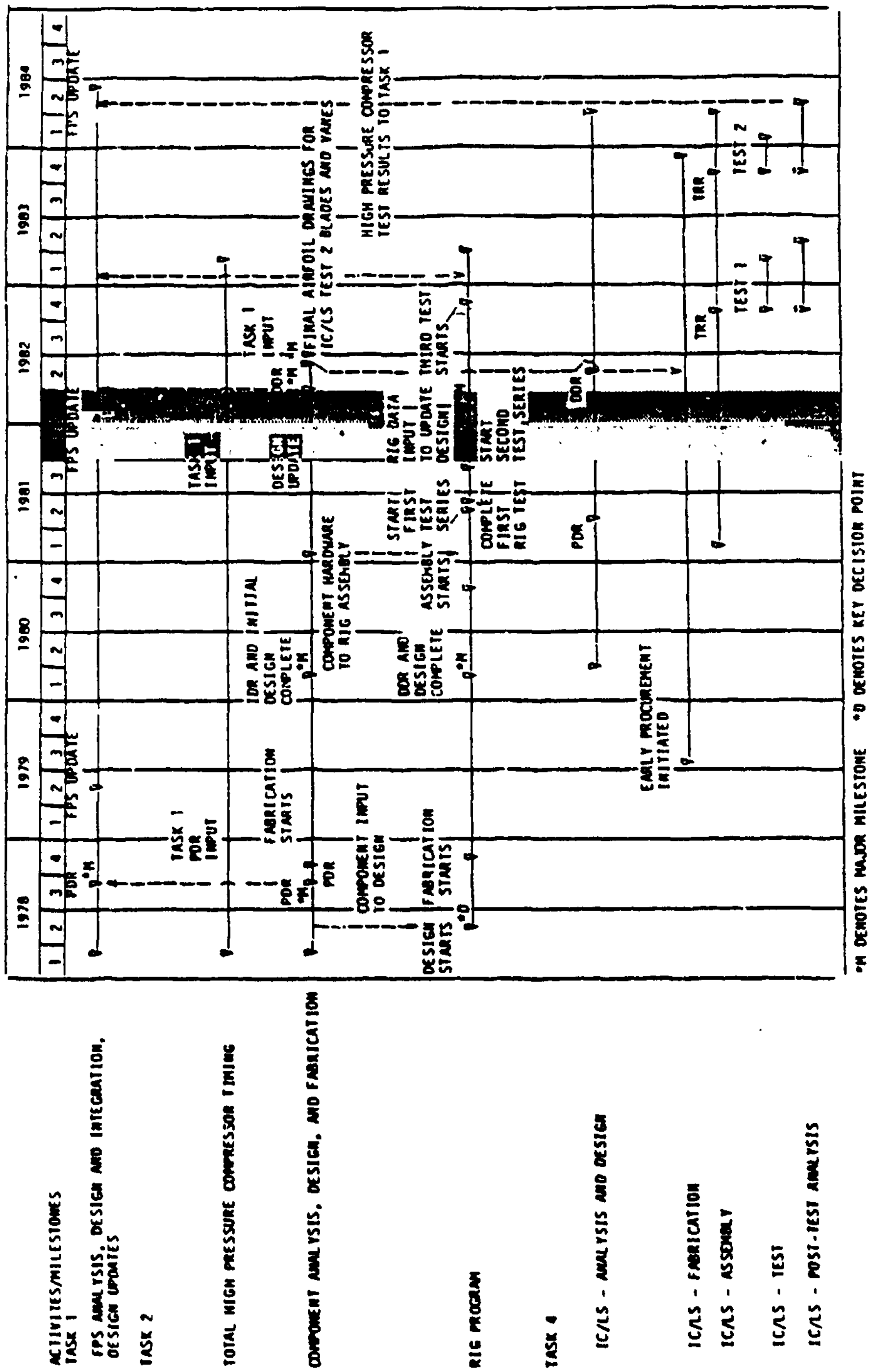

Figure 11 High-Pressure Compressor Program Logic Diagram

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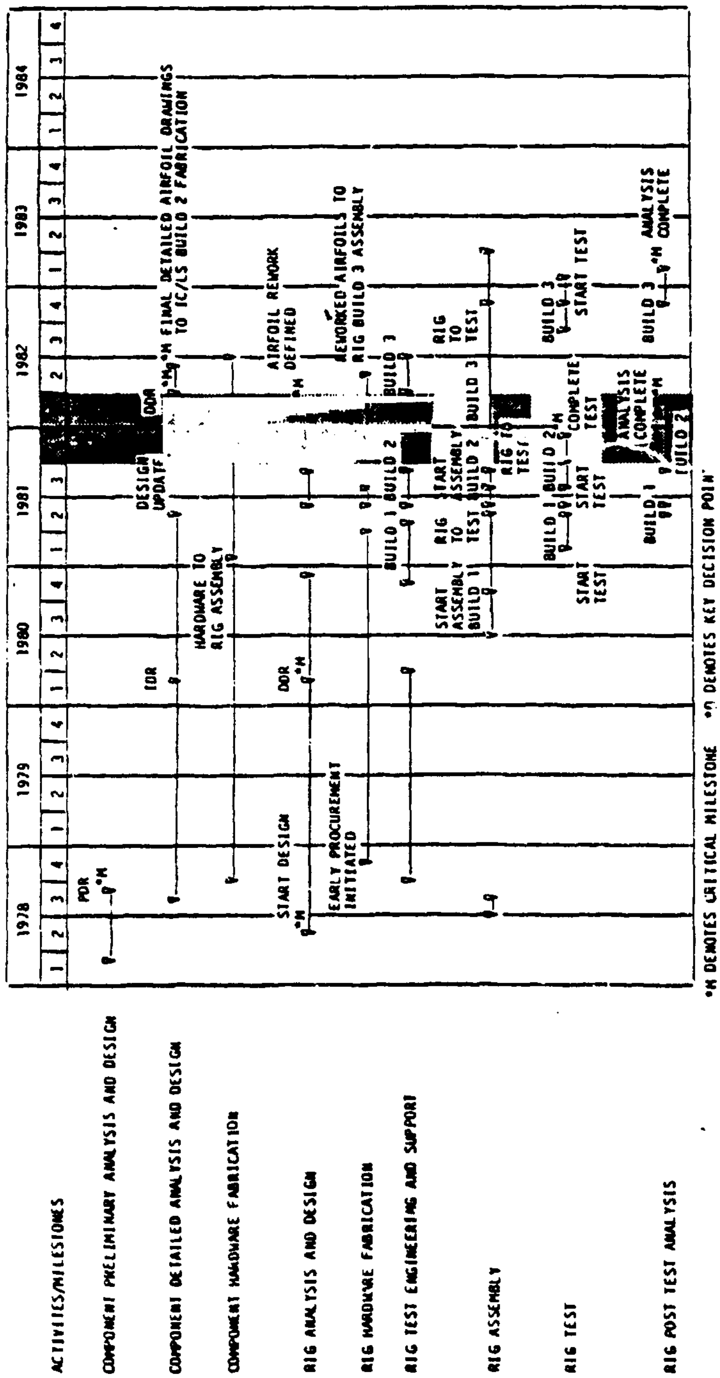

Figure 12 High-Pressure Compressor Component Effort Work Plan Schedule

3.2.5.3 Technical Progress

3.2.5.3.1 Summary of Work Previously Completed

The Energy Efficient Engine high-pressure compressor component and companion rig designs are illustrated in Figures 13 and 14, respectively. All detailed design and analysis of the compressor component and rig has been completed, and a design review was held at NASA-Lewis Research Center in February 1980. A detailed discussion of the results of this effort is presented in the Fourth Semiannual Status Report.

The high-pressure compressor for the Energy Efficient Engine has ten stages. The first four stages contain variable geometry stators. The front case is a split configuration to accommodate the variable stators, while the rear case is of single piece construction. Active clearance control is incorporated in the rear stages. The compressor design also features a drum rotor construction, extensive use of titanium in the static structure, and significantly fewer airfoils. Incorporation of these technology concepts into the assembly results in a component that is lighter, less costly, and easier to maintain. Current performance parameters for the compressor component at significant engine operating conditions are presented in Table 3-IX.

The design of the ten-strut compressor intermediate case was also included in the high-pressure compressor preliminary analysis and design effort, and continued into the detailed analysis and design phase. The intermediate case is designed to:

- support the fan case;
- provide a portion of the fan flowpath and provisions for clamping the nacelle D-ducts;
- carry nacelle loads (load sharing assumed);
- contain the fan exit vanes;
- support the low-pressure compressor static structure and bleed actuating mechanism;
- form the low to high-pressure compressor flowpath;
- support the fan and high-pressure compressor rotors;
- provide front mount locations;
- support accessory drive shafts and gears.

Fabrication of the component and rig-unique hardware in support of the build 1 rig test program was completed. The rig was assembled and delivered to the Pratt & Whitney test facility, stand X-211, on 17 May 1981. Subsequent testing of the build 1 configuration was terminated early in the program because of high stresses on the rig rear thrust balance piston seal. The high stress was attributed to seal flutter occurring when the pressure differential across the piston was increased. Since no data were acquired either at or near the aerodynamic design, the build 1 rig test provided no insight into compressor performance.

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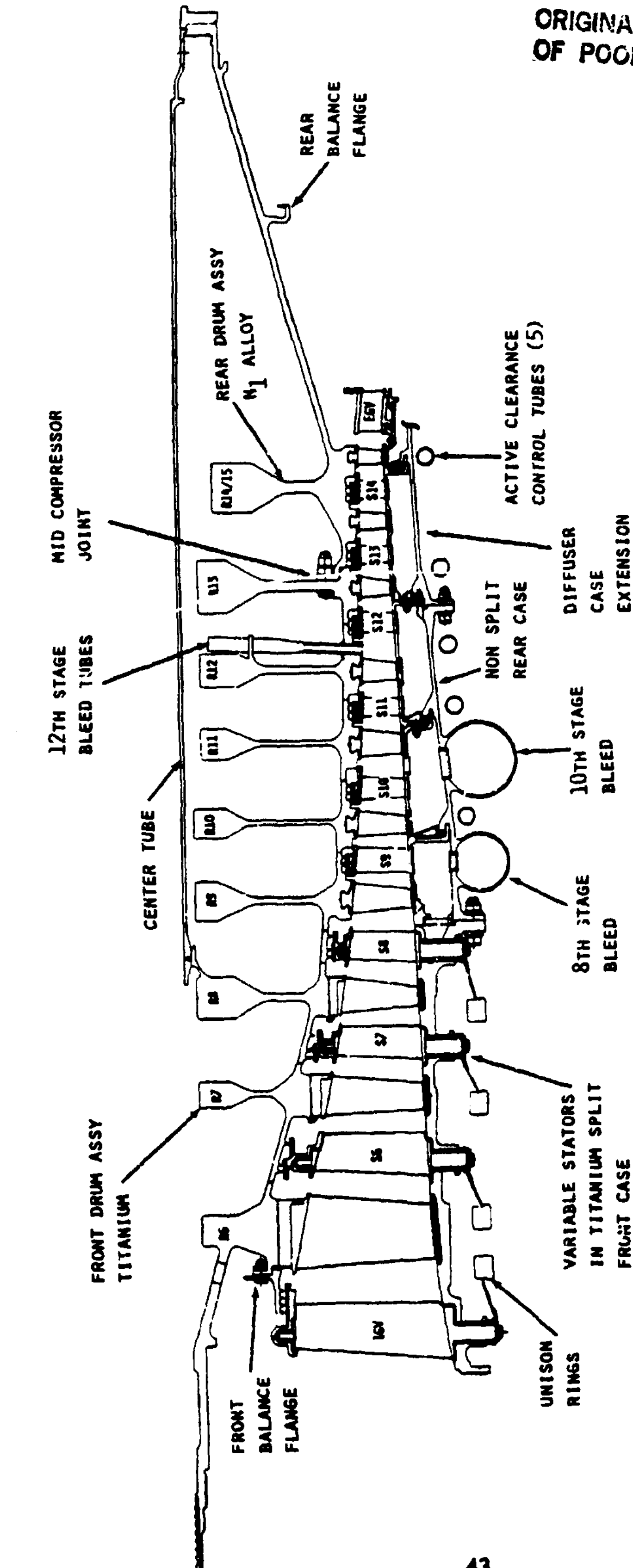

Figure 13 High-Pressure Compressor Component

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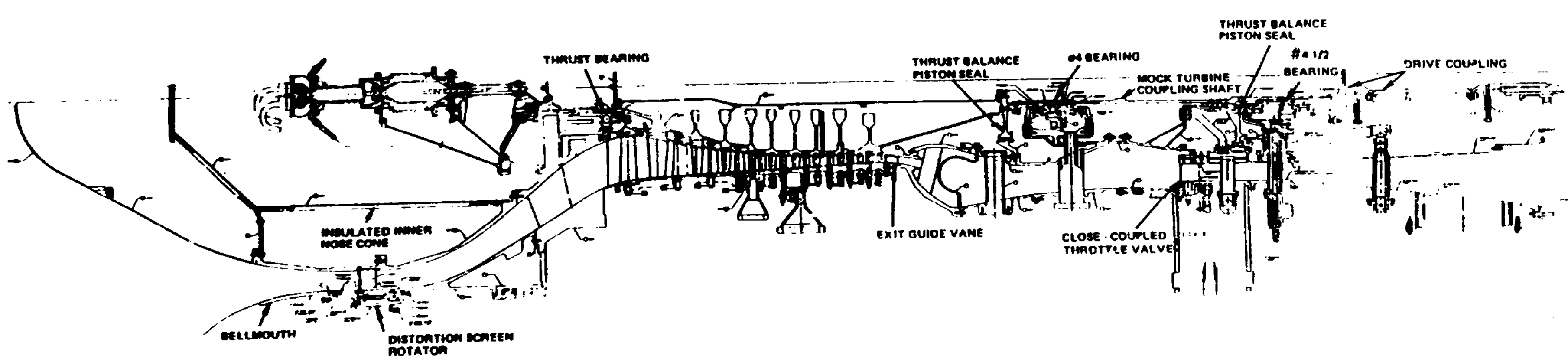

Figure 14 High-Pressure Compressor Rig

FOLDOUT FRAME

TABLE 5-IX

CURRENT HIGH-PRESSURE COMPRESSOR PERFORMANCE PARAMETERS

	Engine Operating Conditions			
	Aero. Des. Point	Maximum Cruise	Maximum Cruise	Takeoff
Pressure Ratio	14.00	13.85	14.25	13.05
Efficiency (%)				
(Adiabatic)	88.3	88.4	88.1	89.4
(Polytropic)	91.7	91.7	91.6	92.4
Inlet Corrected Airflow (lb/sec)	77.65	77.05	78.40	74.20
Inlet Specific Airflow lb/sec/ft^2)	38.0	37.7	38.4	36.3
Inlet Corrected Tip Speed (ft/sec)	1245	1240	1250	1225
Rotor Speed (rpm)	13180	13090	13585	13970
Exit Temperature (^{o}F)	898	883	976	1060

For testing in build 2, the rear thrust balance piston seal was modified to eliminate the flutter-induced stress condition. The revised seal featured a larger cross section, a single knife edge, and two rim ring dampers for better stability. Assembly of the build 2 rig was completed on 27 July 1981, and the rig was delivered to the X-211 test facility.

The test program commenced on 17 August 1981. A preliminary analysis of stress and vibration survey data, in which an acceleration to 105 percent rotor speed was made, indicated acceptable stresses on the thrust balance piston. However, high vibration occurred on the front of the rig during this stress survey. Following rig shutdown, an investigation disclosed that the rotating strain gage slip ring drive shaft separated from the front of the titanium rotor and damaged several parts in the front bearing compartment. Testing was again terminated prematurely. The rig was removed from the test stand and the effort required to repair the damaged parts was initiated. The rig, thus modified, was designated Build 2A.

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3.2.5.3.2 Current Technical Progress

High Pressure Compressor Rig (Build 2A) Design, Fabrication and Assembly -- The slip ring drive system was modified to eliminate the problem experienced with the original configuration, as reported in the Seventh Semiannual Status Report. A comparison of the original system design with the new system design is shown in Figure 15. The rig slip ring drive modifications, identified in Figure 15, consist of (1) a shorter shaft axial length to reduce the overhung moment, (2) a reduction in the axial length of the inner intermediate case front flange, and (3) replacement of the riveted joint on the compressor front hub with a threaded joint. Fabrication of the revised slip ring drive shaft was completed along with other minor parts to replace those parts damaged during the test (build 2). A new front carbon seal was purchased from existing inventory, and the rig intermediate case inner flange was modified. Changes to the compressor front hub were installed without removal of the rotor from the rig. All parts were available for rig assembly by early October 1981. The revised slip ring drive configuration was installed in the rig and the slip ring was mounted. Instrumentation removed previously to make the repairs was reconnected. The assembled rig was returned to the test facility on 20 October 1981 for installation.

High-Pressure Compressor Rig (Build 2A) Test -- Testing resumed on 28 October 1981, and the program was completed on 10 December 1981 without interruption. The series of tests outlined in the Test and Instrumentation Plan was completed as defined, except for testing at maximum speed with a heated inlet. This portion of the program was aborted after several unsuccessful attempts were made to achieve an acceptable inlet temperature profile from the facility heater.

Total rig running time for builds 1, 2 and 2A was 256 hours. A total of 744 data points was recorded, including 33 surges. No blading modifications were made during the test period other than restaggering the variable stator rows.

The compressor initially underflowed the altitude cruise design point by approximately 6 percent, with efficiency slightly more than 2.5 percentage points below the goal. This performance was the result of work in the middle to rear stages being less than design. An initial attempt at restaggering the vane schedule achieved the design flow level and increased the efficiency by 2 percentage points. Further optimization of stage matching increased efficiency an additional two-tenths of a point.

Following optimization of compressor efficiency, a change in the demonstrated performance level was produced by the occurance of two events. The first was the initial high speed surge and the second was a rapid acceleration to a rotor speed in excess of that required for the integrated core/low spool engine. Performance measurements showed a net deterioration of 0.8 percent in flow and 1.2 points in efficiency. A subsequent teardown inspection at the conclusion of testing indicated that an increase in rotor tip clearances and knife edge seal clearances was responsible for the degradation in performance.

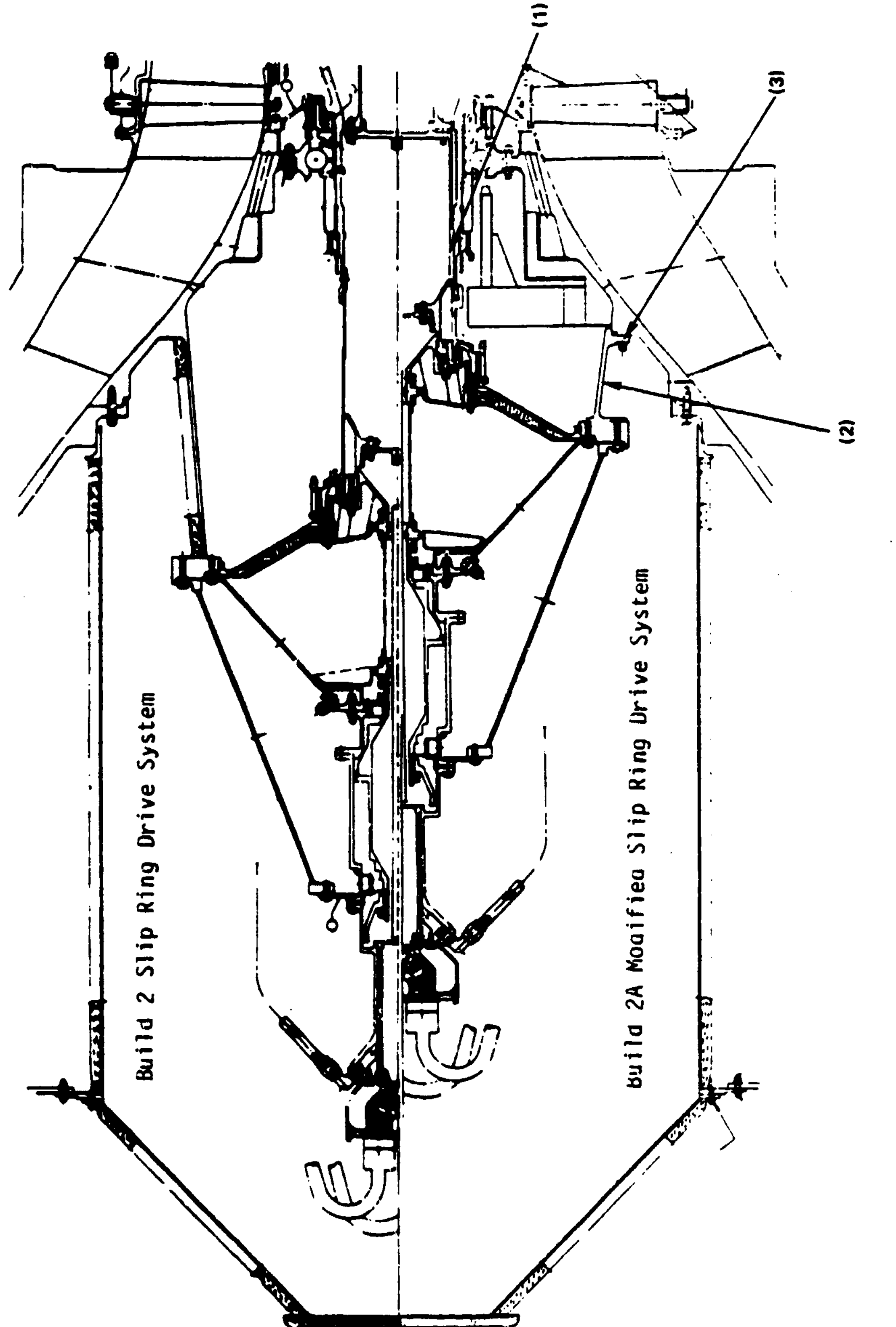

Figure 15 Original (Build 2) and Revised (Build 2A) High-Pressure Compressor Rig Slip Ring Drive Systems

Before the deterioration in performance occurred, however, the compressor was optimized for the best combination of surge margin and flow. No loss in operating line efficiency was noted, while surge margin was increased by about 4 percent. This version of compressor rematch has been proposed for the first build of the integrated core/low spool engine, for which performance has been documented over the full range of engine operating conditions. Additional tests were conducted to determine the effects of Reynolds number, inlet distortion and bleed quantities on performance.

The compressor maps in Figures 16 and 17 show the demonstrated performance for the high speed and starting regions, respectively. These results are based on data acquired from the six total pressure and six total temperature radial instrumentation rakes at the compressor exit. All data points were recorded with the bleeds operational and in an inlet Reynolds number range appropriate for the engine. Corrections to flow, pressure ratio and efficiency are referenced to the inlet station instrumentation located at the intermediate case strut leading edge.

Adjustments to the performance maps account for discharge wake rake and pole rake sampling, inlet and discharge probe Mach number recovery, estimated inlet and leading edge instrumentation losses, and the deterioration measured over the period of the test. A listing of these adjustments is presented in Table 3-X.

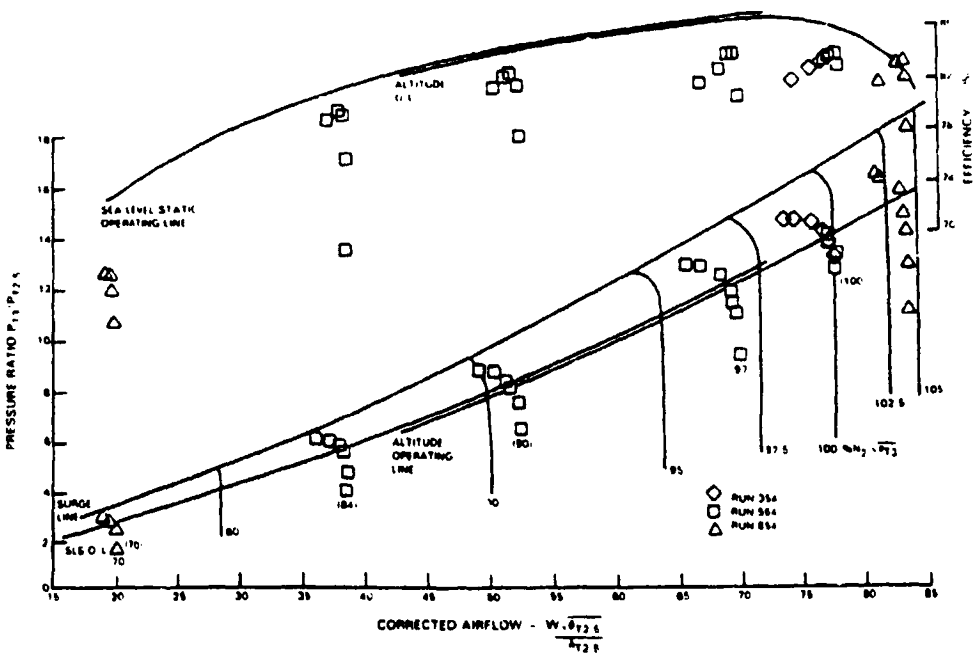

Figure 16 High-Pressure Compressor Performance at High Speed Operation

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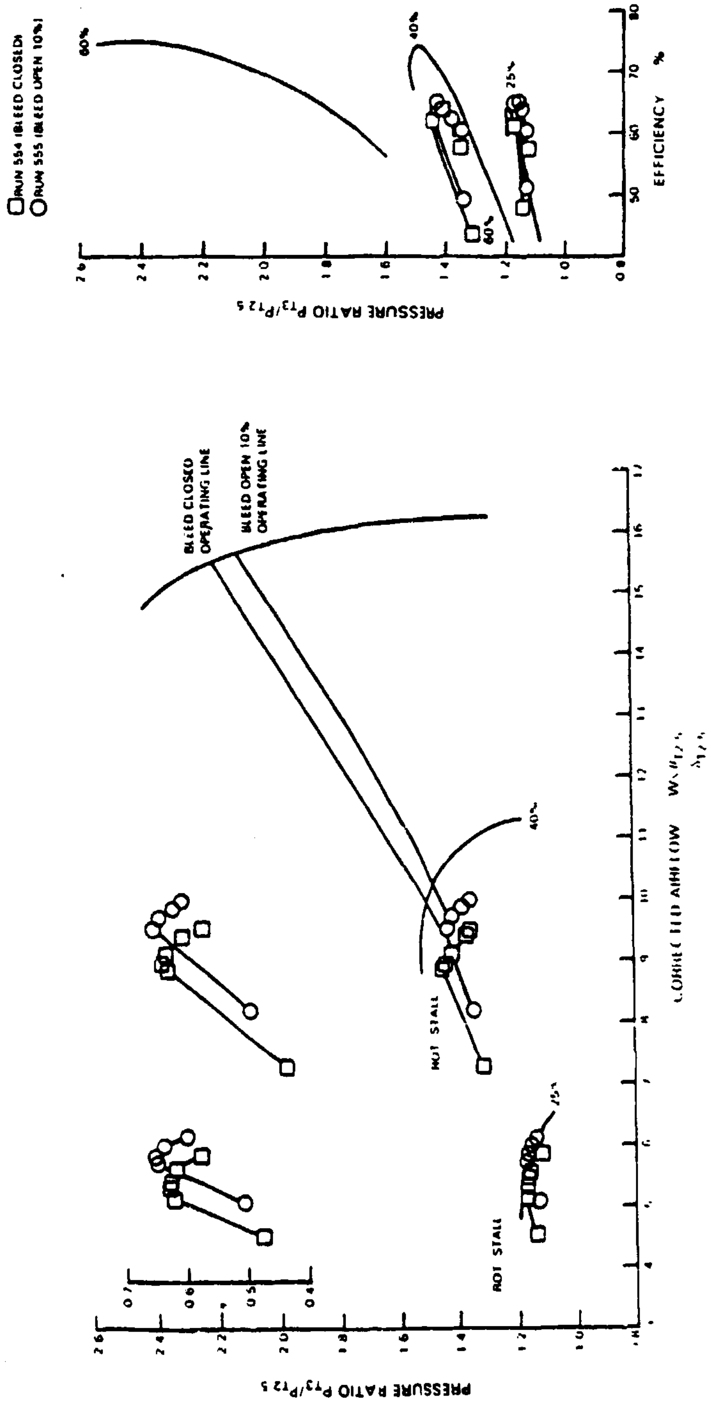

Figure 17 High-Pressure Compressor Performance in the Starting Region

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TABLE 3-X

SUMMARY OF COMPRESSOR PERFORMANCE ADJUSTMENTS
(All values in percent)

	100% Corrected Speed Cruise Op Line	97% Corrected Speed SLS Op line
Test Efficiency	83.6	83.8
6 Total Pressure and 6 Total Temp Rakes Interchanged	0.1	0.1
Wake Rake vs. Pole Rake Sampling (Gap Plus Radial Averaging)	-0.3	-0.3
Total Temp Probe Mach Number Recovery	0.4	0.4
Inlet and Lead Edge Instrumentation	0.5	0.5
Deterioration	1.2	1.2
Adjusted Rig Efficiency	85.5	85.7
First Aero Build Goal	86.0	---
Reynolds Index Adjustment	----	0.1
Rig Penalty - Leakage in Variable Rear Stages	0.5	0.5
First integrated core/low spool Compressor Efficiency	86.0	86.3
First integrated core/low spool Efficiency Goal	86.5	87.1

With these adjustments, compressor performance can be summarized as follows:

- The design airflow and pressure ratio were achieved at rig build clearances.
- The demonstrated adiabatic efficiency was within 0.5 point of the rig goal.
- Surge margin, although less than the flight propulsion system goal, is adequate for the first test of the integrated core/low spool engine.
- Start region stall margin and efficiency are comparable to those experienced with core compressors of other engines having satisfactory starting characteristics.
- The compressor is insensitive to inlet pressure distortion throughout the operating range.

A comprehensive analysis of rig aerodynamic performance data has been started. Potential areas for performance improvement, as indicated by analysis to date, include velocity profile weakness at various blade hubs or tips, stator-rotor mismatch in incidence, less than optimum incidence at the engine operating line, and greater than design exit guide vane loss.

Modifications will be made to the blading for the next rig build, as required, based on final analysis of several different stator schedules tested during the program.

High-Pressure Compressor Rig (Build 2A) Disassembly -- After completion of testing, the rig was disassembled for an examination of the hardware. In general, the post-test condition of the compressor components was excellent. However, the following anomalies were noted.

(1) Coking in the number 4 bearing compartment
(2) Six fractured tab locks on the mid-rotor bolt joint
(3) Several cracks on the bleed case in the bleed boss welds
(4) Delamination of the abradable material on the tangential on-board injection inner seal land
(5) Piston ring wear on the tangential on-board injection duct piston
(6) Blade tip wear in stages 9 through 15
(7) Blade rub strip wear in all stages
(8) Rotor knife edge seal wear in stages 9 through 14
(9) Vane inner shroud seal land wear
(10) Crack indications on six 13th-stage blade tips

Blade and case measurements were obtained to determine the change in clearances as a result of blade tip rubbing. Also, the calibrations of the laser tip probes were checked. Stator vane angles at the rigging pin location were verified and compared to test data readings. The inspection results were summarized and made available for use in the analysis of test data.

Following inspection, all hardware was packaged and prepared for storage.

3.2.6 Combustor

3.2.6.1 Overall Objective

Design an annular two-stage combustor and demonstrate three advanced technology concepts: (1) a curved-wall, strutless diffuser; (2) a two-stage combustor having a pilot zone and carburetor tube main zone; and (3) a segmented liner with an advanced cooling scheme. The goals established for the combustor rig (Table 3-XI) are the same as those established for the flight propulsion system component.

TABLE 3-XI

COMBUSTOR COMPONENT RIG GOALS

PERFORMANCE	
Pattern Factor, Maximum	0.37
Section Pressure Loss	5.5 percent P_{T3}
Radial Profile	250 degrees average-peak
EMISSIONS	
Hydrocarbons*	$\leq$ 0.4
Carbon Monoxide*	$\leq$ 3.0
Oxides of Nitrogen*	$\leq$ 3.0
SAE Smoke Number	$\leq$ 20
LIFE	
Liner Life	8000 hours, 4900 missions

* Environmental Protection Agency Parameter

3.2.6.2 Scope of Total Work Planned

The overall task effort consists of a component effort and two supporting technology subtasks. The component effort comprises the analysis and design of the combustor component and a combustor rig test program. The two supporting technology programs are the Diffuser/Combustor Model Test Program and the Combustor Sector Rig Program. Figure 18 shows the relationships between these activities and their relationship to Tasks 1 and 4. The work plan schedule for the component effort is shown in Figure 19 and critical milestones are noted.

3.2.6.3 Component Effort

3.2.6.3.1 Objective

Conduct the design, analysis, hardware procurement, and both full annular and sector rig test activities necessary to develop a full annular combustor that meets the program goals.

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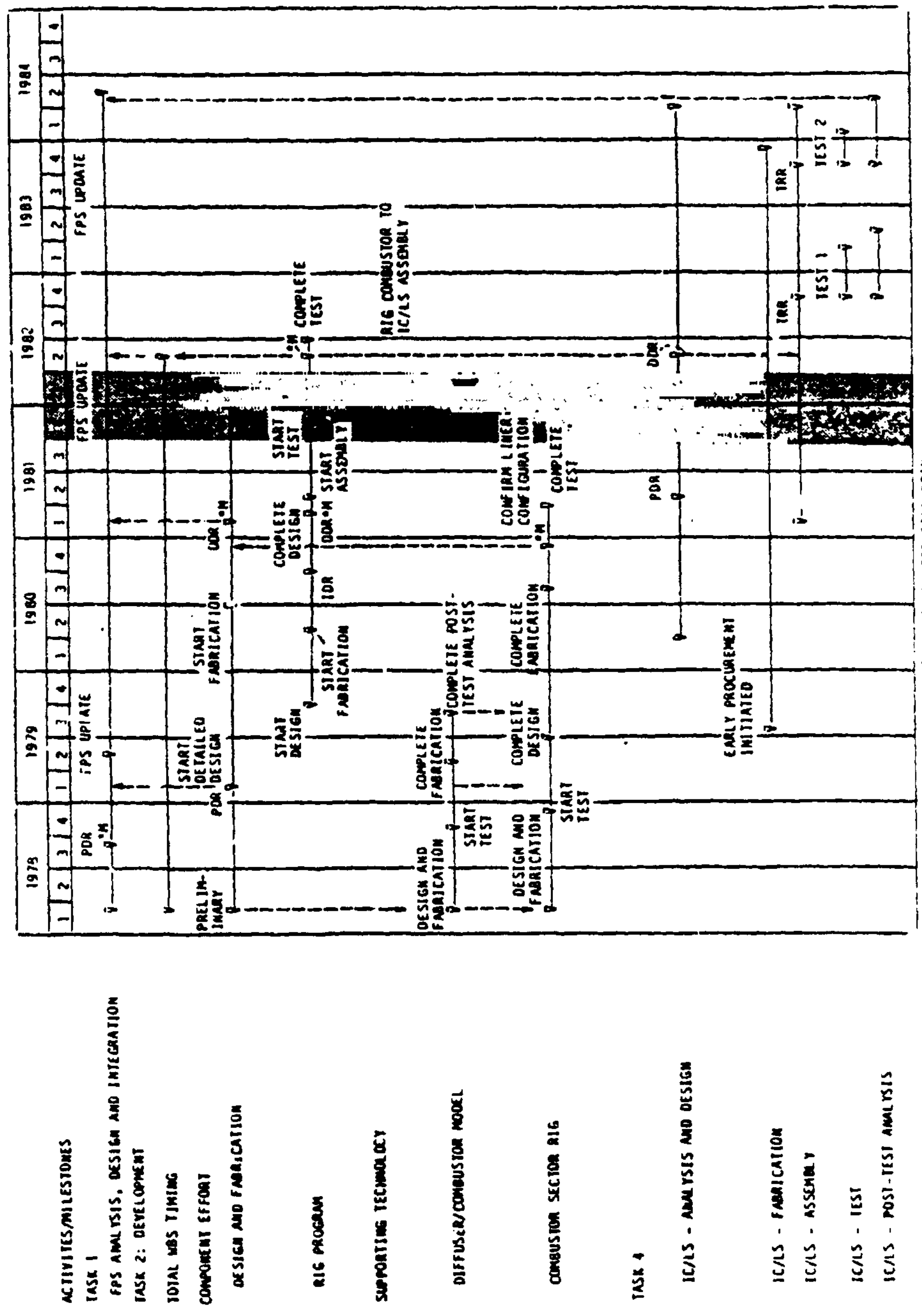

Figure 18 Combustor Program Logic Diagram

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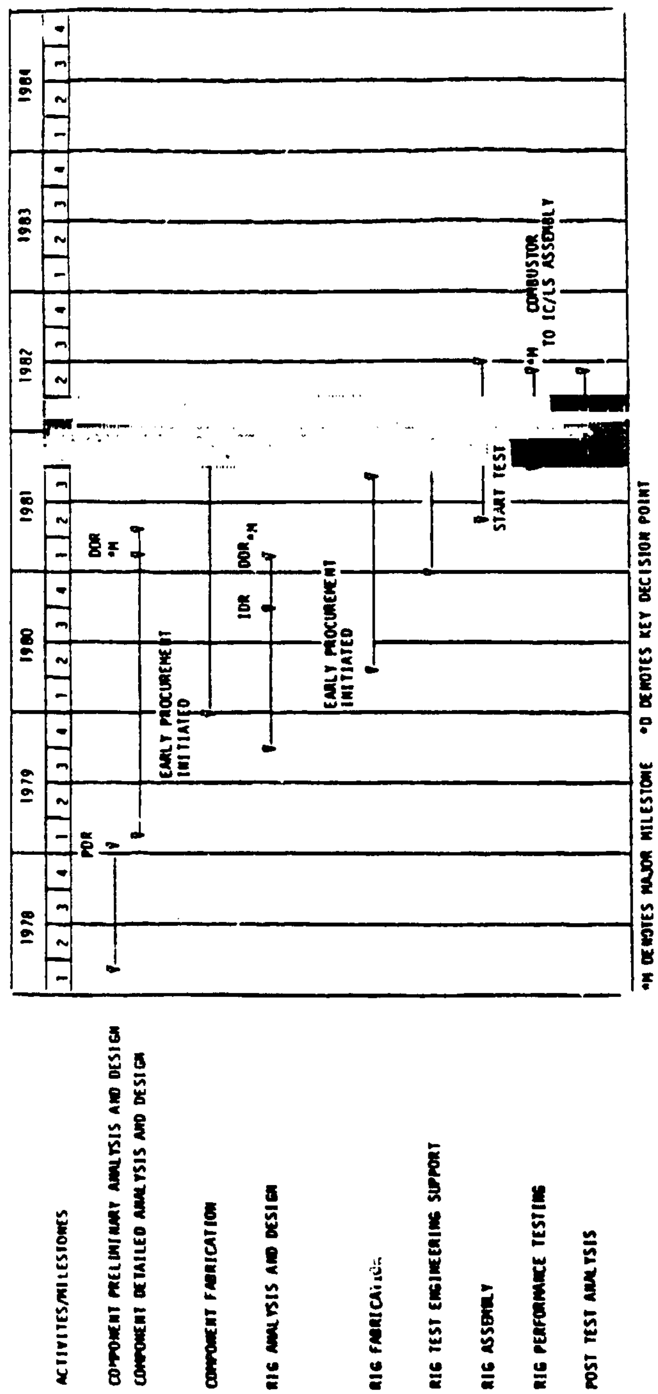

Figure 19 Combustor Program Effort Work Plan Schedule

3.2.6.3.2 Scope of the Total Work Planned

The analysis and design effort consists of both a preliminary and a detailed analysis and design phase. The rig program entails the six subtasks shown in Figure 19. A preliminary design activity is conducted to establish the feasibility of the combustor proposed for the flight propulsion system. The designs studied provide configuration definitions to the supporting technology programs. This activity results in layout drawings and substantiation of design data that are presented to NASA at a Preliminary Design Review in January 1979.

Detailed design activity starts in March 1979. Results from the supporting technology programs are used to substantiate or improve the configurations established in the preliminary design. Also, more sophisticated design and analytical procedures than those employed in the preliminary effort are used. Results are presented to NASA in a Detailed Design Review in February 1981. Detailed drawings are scheduled for completion approximately two months later.

Design and fabrication of combustor rig parts progress concurrently with those of the component parts, permitting the start of full-scale rig assembly in the second quarter of 1981. Various modifications to the combustor are tested to develop a final configuration that satisfies the program goals. Testing consists mainly of air schedule variations to demonstrate emissions, exit radial temperature profile, performance, and durability. In May 1982, the final diffuser/combustor configuration is transferred to the first build of the integrated core/low spool assembly effort.

All of the work planned and approved from contract award through the end of the current reporting period (31 March 1982) has been completed. Figure 19 indicates that all component and rig design and fabrication activities have been completed and that rig testing has been initiated during the current reporting period.

3.2.6.3.3 Technical Progress

3.2.6.3.3.1 Summary of Work Previously Completed

All detailed analysis and design work for the combustor component and rig was completed in prior reporting periods. A rig design review was held at the NASA-Lewis Research Center in September 1980. This was followed by a component detailed design review in February 1981.

The combustor component is illustrated in Figure 20, and the companion full annular rig is shown in Figure 21. The basis for the component design was the two-stage combustor investigated in the NASA-sponsored Experimental Clean Combustor Program. As shown in Figure 20, the combustor has two distinct combustion zones: a pilot zone designed to minimize idle emissions, provide adequate stability and relight characteristics and a main zone that provides fuel-lean combustion to minimize emissions of smoke and oxides of nitrogen.

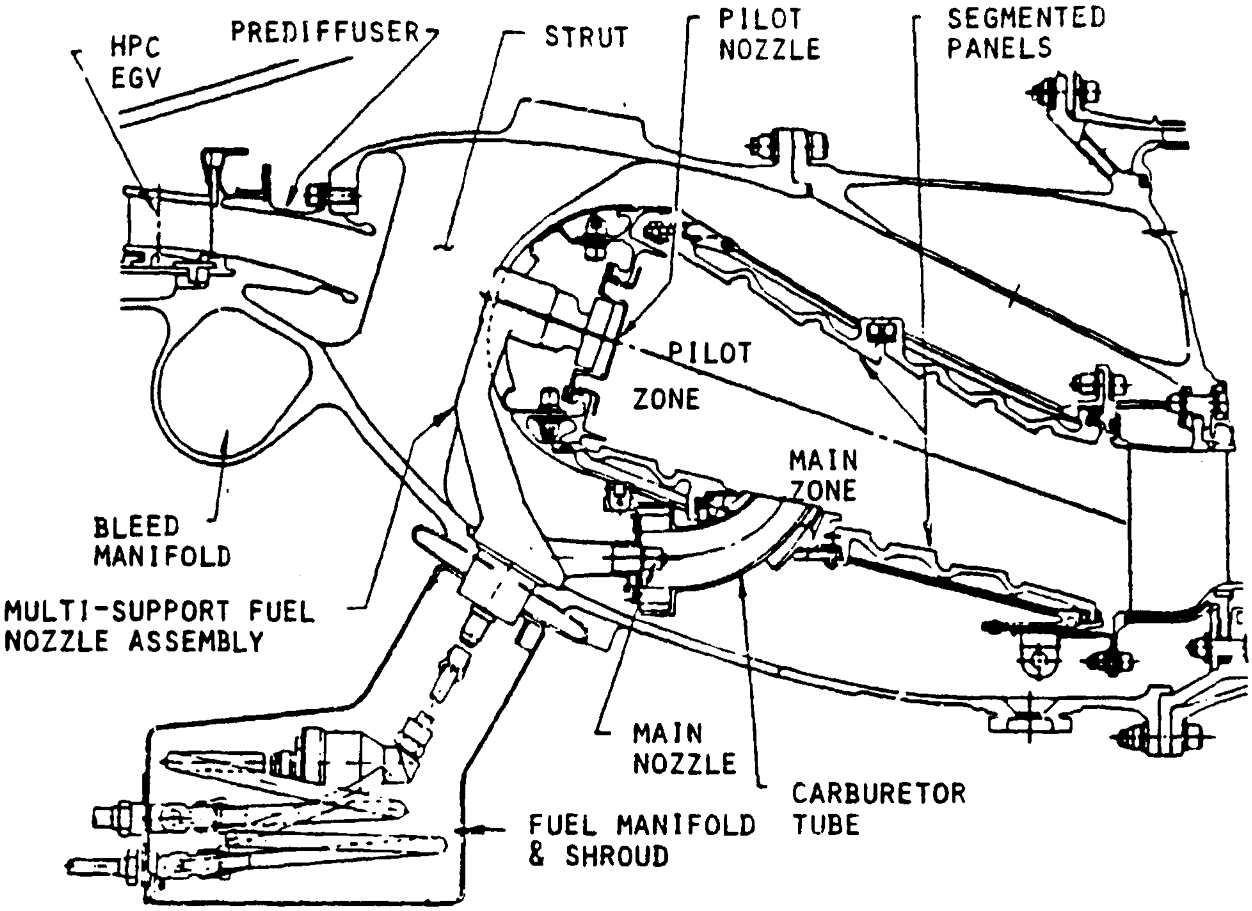

Figure 20 Combustor Component

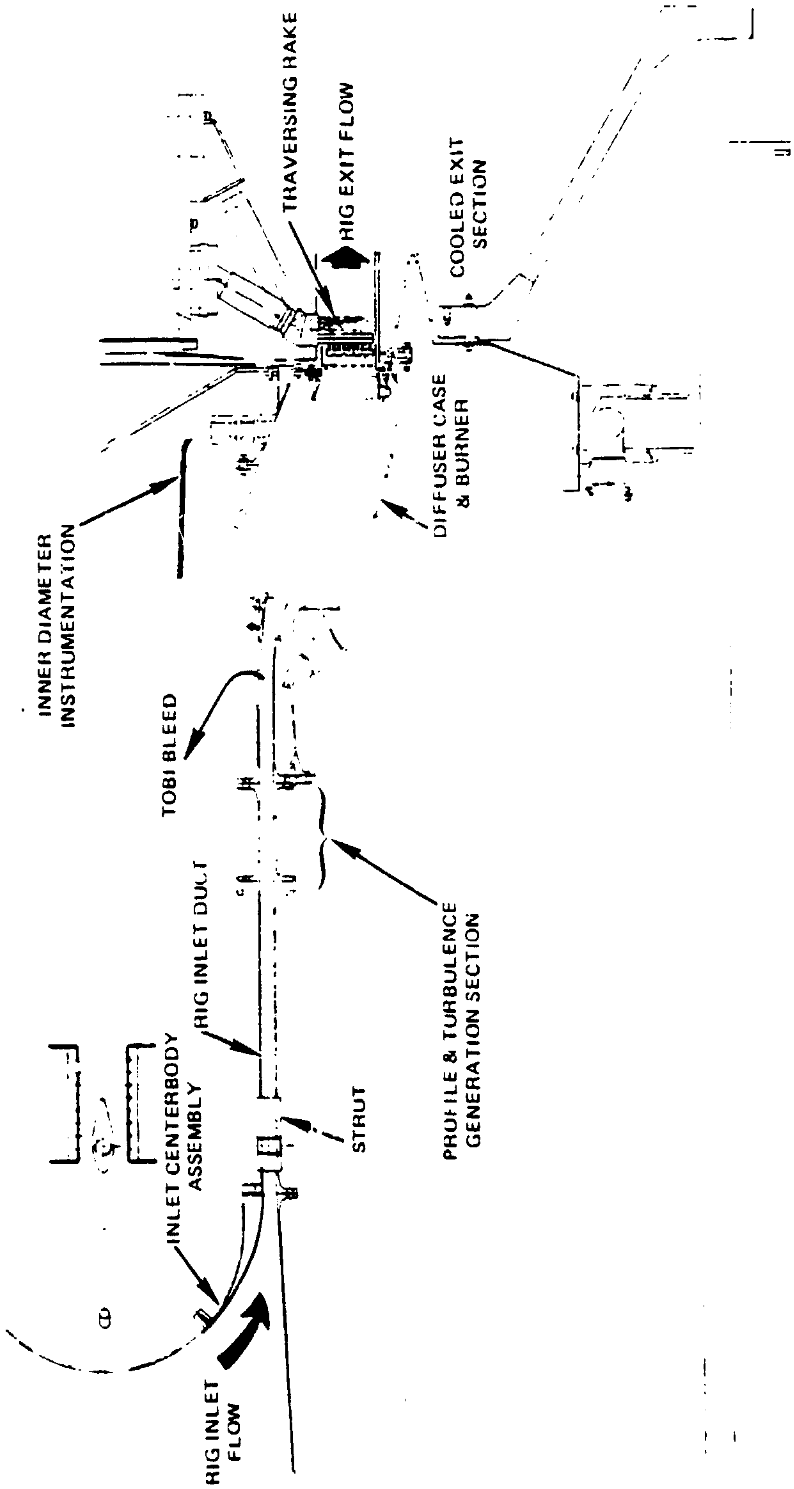

Figure 21 Combustor Component Full Annular Rig

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The design of the compressor exit guide vane assembly was also included in the design of the combustor component because of its interaction with the prediffuser duct. This assembly features: (1) a vane with integrally attached inner and outer shrouds but circumferentially separated into groups of five vanes to relieve thermal gradient stresses, (2) decoupled inner and outer prediffuser duct walls, (3) a sheet metal seal for the gap between the vane and inner prediffuser duct wall, and (4) feather seals to control air leakage between segments.

The exit guide vane design consists of a single row of airfoils. A back-up design, based on a dual row airfoil approach for air entrance and turning angle, will be fabricated if the expected efficiency of the single row exit guide vane is not demonstrated during testing of the second build of the high-pressure compressor rig.

Current combustor component performance parameters at significant engine operating conditions are shown in Table 3-XII.

TABLE 3-XII

CURRENT COMBUSTOR COMPONENT PERFORMANCE PARAMETERS

	Engine Operating Condition			
	Aero. Des. Point	Maximum Cruise	Maximum Climb	Takeoff
Inlet Corrected Airflow (lb/sec)	6.90	6.92	6.86	6.95
Inlet Pressure (psia)	203	197	212	456
Inlet Temperature (^{o}F)	898	883	976	1060
Section Pressure Loss (%)	5.50	5.53	5.43	5.58
Fuel/Air Ratio	0.02420	0.02365	0.02651	0.02667
Exit Temperature (^{o}F)	2360	2315	2540	2615
Combustor Efficiency (%)	99.95	99.95	99.95	99.95

Fabrication of hardware required for the full annular combustor test rig assembly was nearing completion. The only outstanding components were the fuel manifold sealing shroud and the exit guide vane seal assembly. Major subassemblies that have been fabricated and installed in the combustor component rig include: the diffuser case, bulkhead and hood assemblies, inner case, tangential on-board injection air duct, fuel nozzle support, advanced liner segments, and liner support frame. In addition, all unique rig hardware for the full annular rig has been fabricated. This hardware consists of the inner and outer flowpath ducting, including the rig inlet case, as well as the station 3.0 and 4.0 instrumentation rakes.

With the fabrication/procurement effort nearly complete, rig assembly was initiated. All mating rig hardware was trial fitted to ensure proper bolt hole alignment. Trial assembly of the diffuser/combustor identified several minor interferences that were subsequently corrected. Figure 22 shows an inner combustor support frame with several liner segments installed. Figure 23 shows the combustor bulkhead mated to the inner and outer combustor support frames. Hardware from the combustor sector rig supporting technology program was also transferred to the combustor component rig assembly effort.

The combustor component test and instrumentation plan for the integrated full annular and sector rig test program was submitted and approved by NASA.

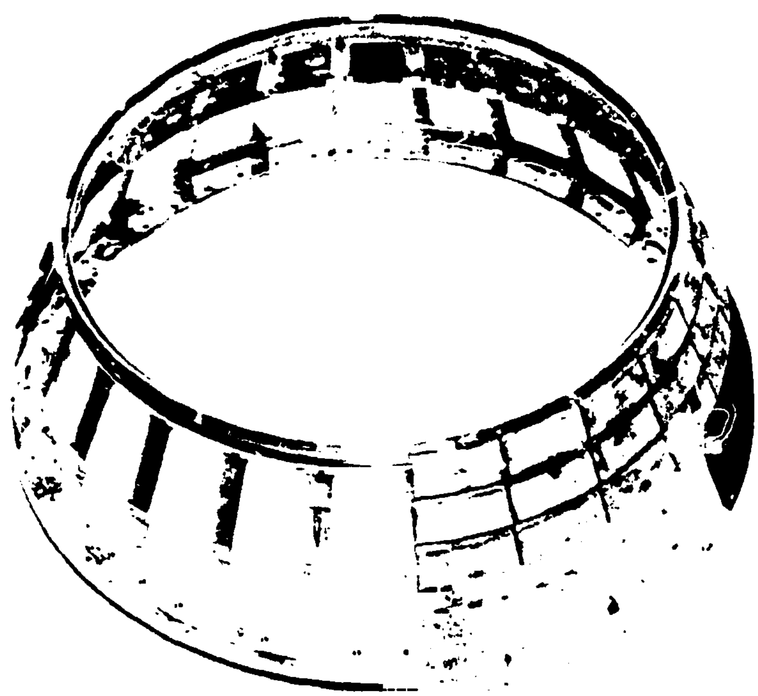

Figure 22 Inner Combustor Support Frame with Liner Segments Installed

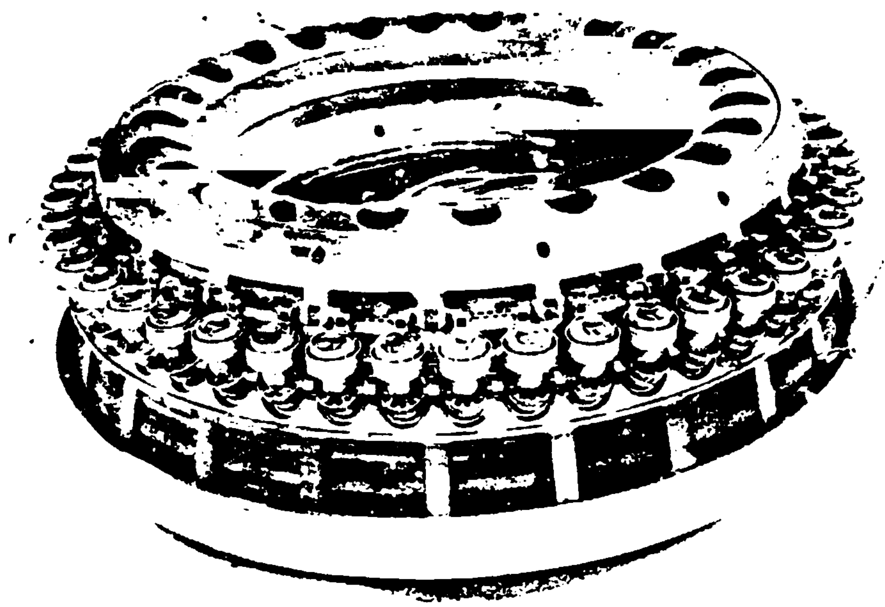

Figure 23 Combustor Bulkhead Mated to Inner and Outer Support Frames

3.2.6.3.3.2 Current Technical Progress

Combustor Component Fabrication

In this reporting period, all remaining planned fabrication was completed. A summary of the effort completed since the last report period is presented in the following paragraphs.

Fuel Jumper Tubes and Fuel Manifold Sealing Shroud -- A minor revision was made to the jumper tube between the main zone pressure equalizing valve and fuel nozzle support to eliminate an interference. A comparison of the original and revised tube configurations is shown in Figure 24.

The fuel manifold sealing shroud was installed and fit-checked during assembly of the combustor component. The installation of the sealing shroud without the outer cover in place is shown in Figure 25.

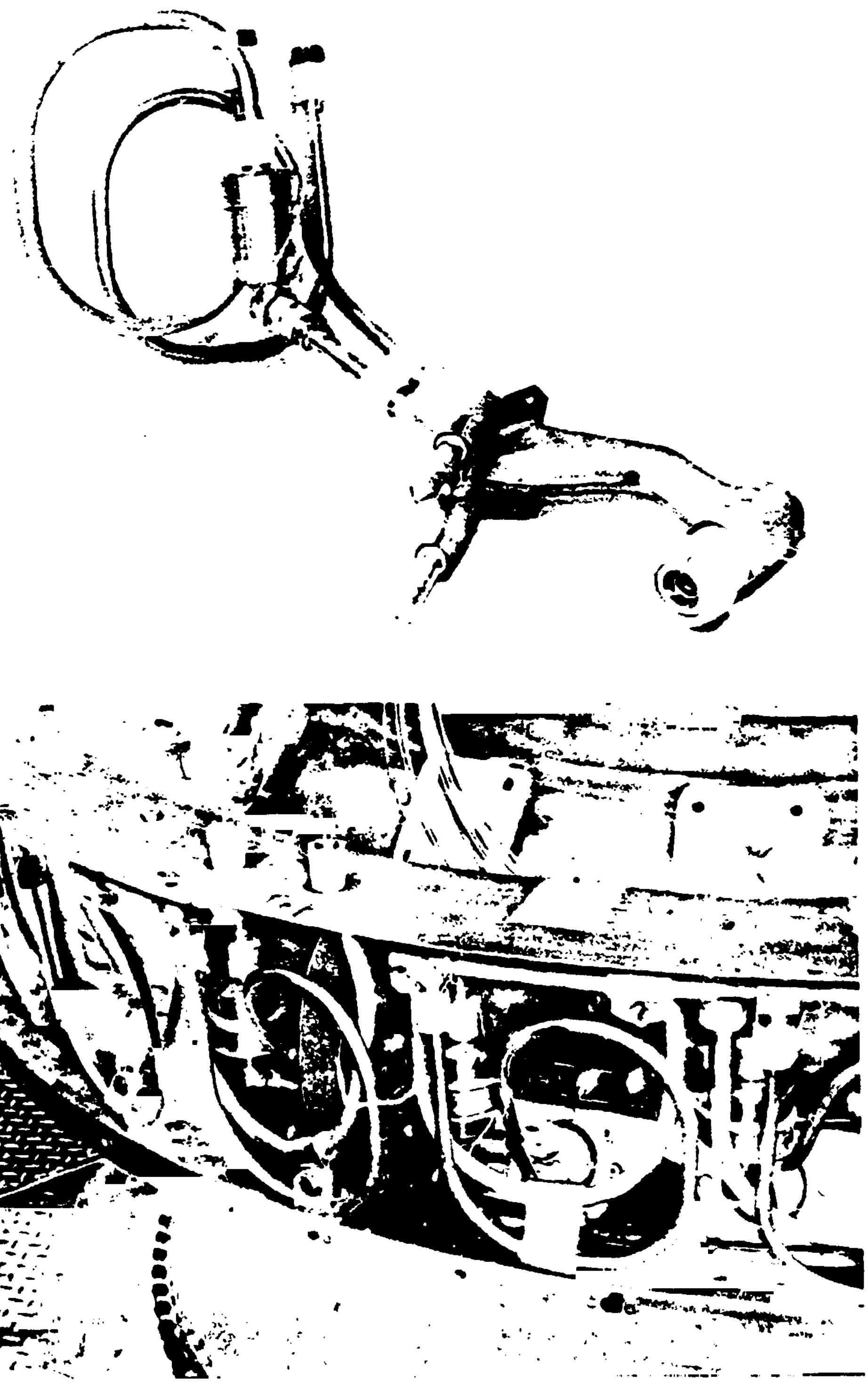

Figure 24 Original (Top) and Revised Fuel Jumper Tube Configurations

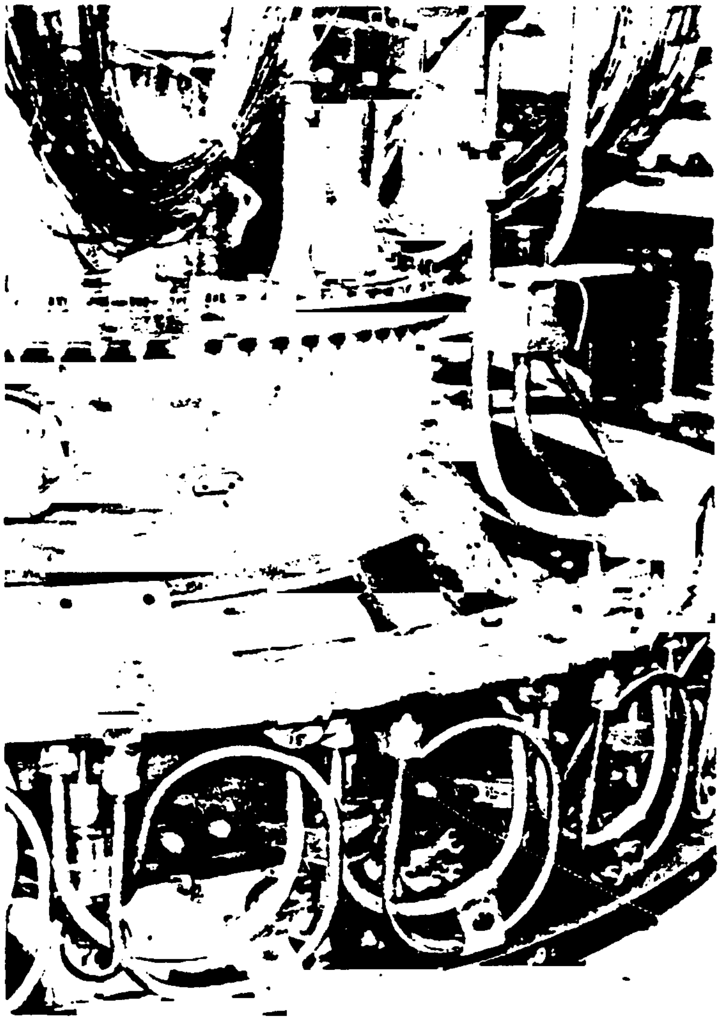

Figure 25 Combustor Component with Fuel Manifold Sealing Shroud

Combustor Component Rig Assembly

Assembly of the full annular combustor and the combustor rig was completed. In Figure 26, the diffuser case with rig inlet case and rig profile generation section is shown. Also shown is the tangential on-board injection (TOBI) bleed air manifold on the inlet case. Bleed air from the rig inner annulus flows through three of eight struts in the inlet case to the manifold. A test facility system meters and controls the flow. Figure 27 presents a view looking downstream into the inlet case, in which the flowpath struts can be seen.

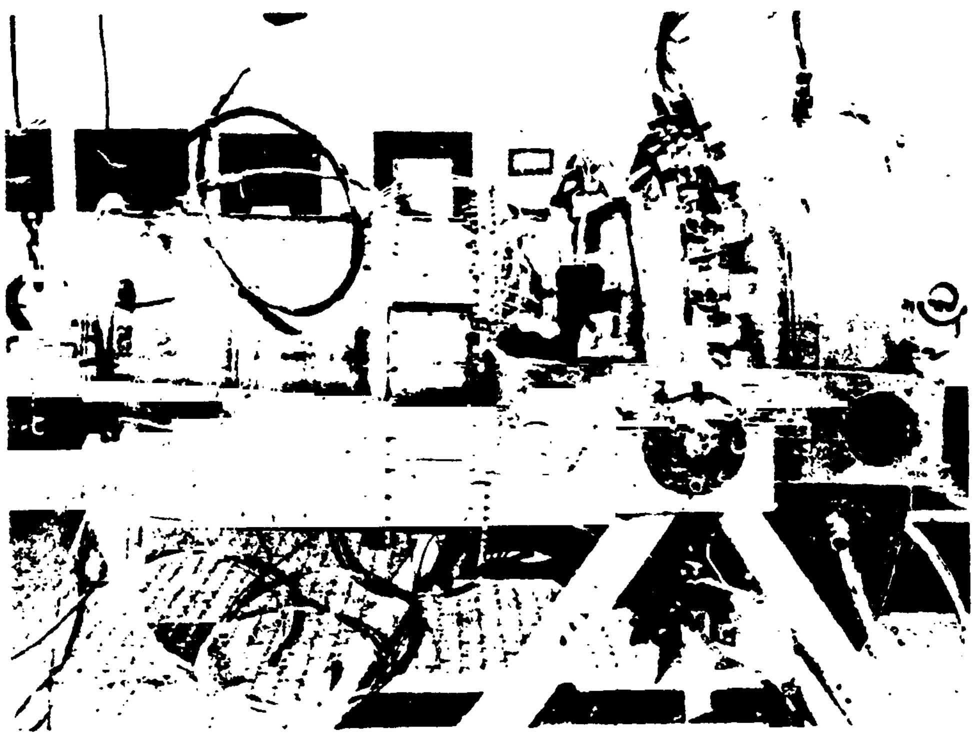

Figure 26 Diffuser Case with Rig Inlet Case

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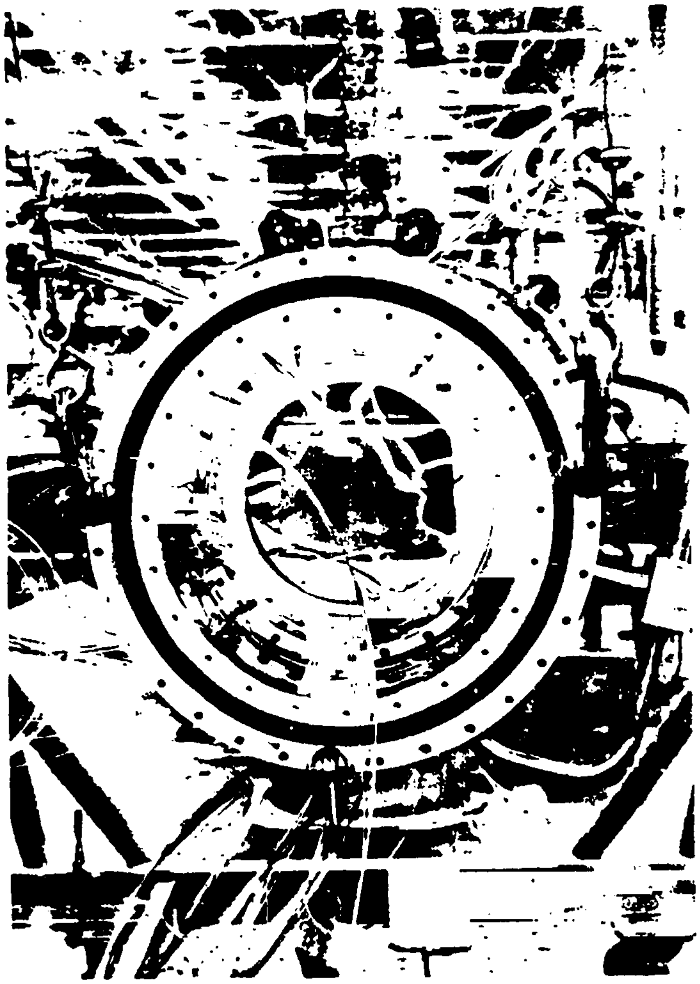

Figure 27 Front View Looking Into the Inlet Case -- Note flowpath struts

The assembled full annular combustor rig is shown in Figure 28 prior to installation on the support diaphragm that connects the rig to the pressure capsule. In this figure, the insulation blankets on the rig inlet ducts are shown. These blankets are intended to reduce radiation effects upon the cooling air flowing in the annulus formed by the case outer walls and the rig capsule.

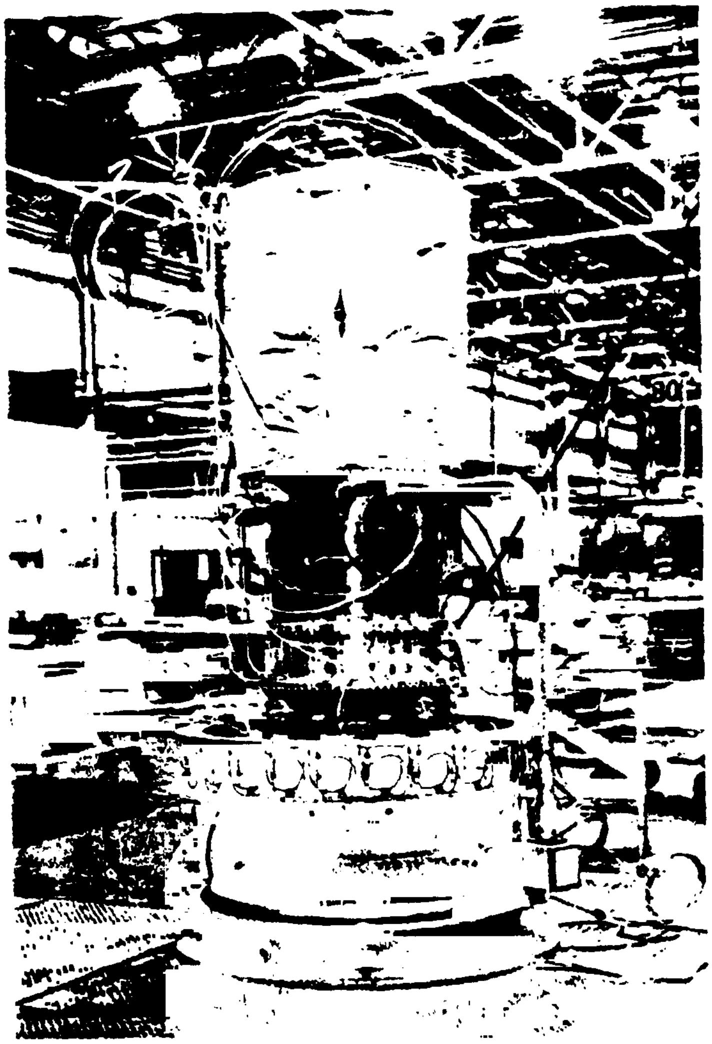

Figure 28 Full Annular Combustor Test Rig

Figure 29 shows the pressure capsule. Quick disconnect panels for pressure sensors and multi-pin connectors for temperature sensors are mounted on the instrumentation ring. This facilitates mounting and removal of the test rig in the capsule. Figure 30 shows the rig before installation in the pressure capsule. Initially, the quick disconnect panels are installed after the rig is in place. However, after the panels have been installed, only plug-in connections are required for subsequent installations.

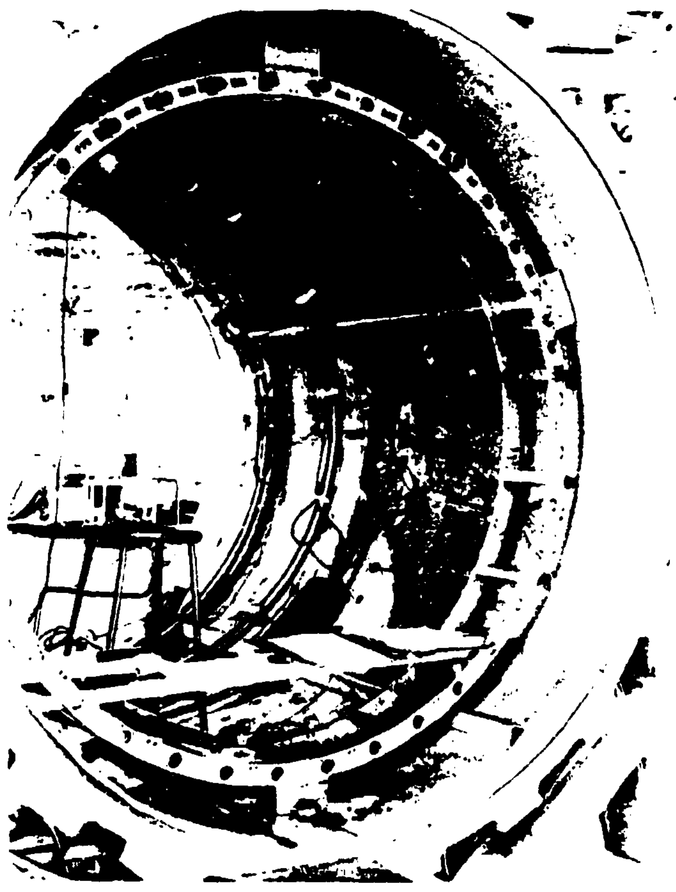

Figure 29 Pressure Capsule

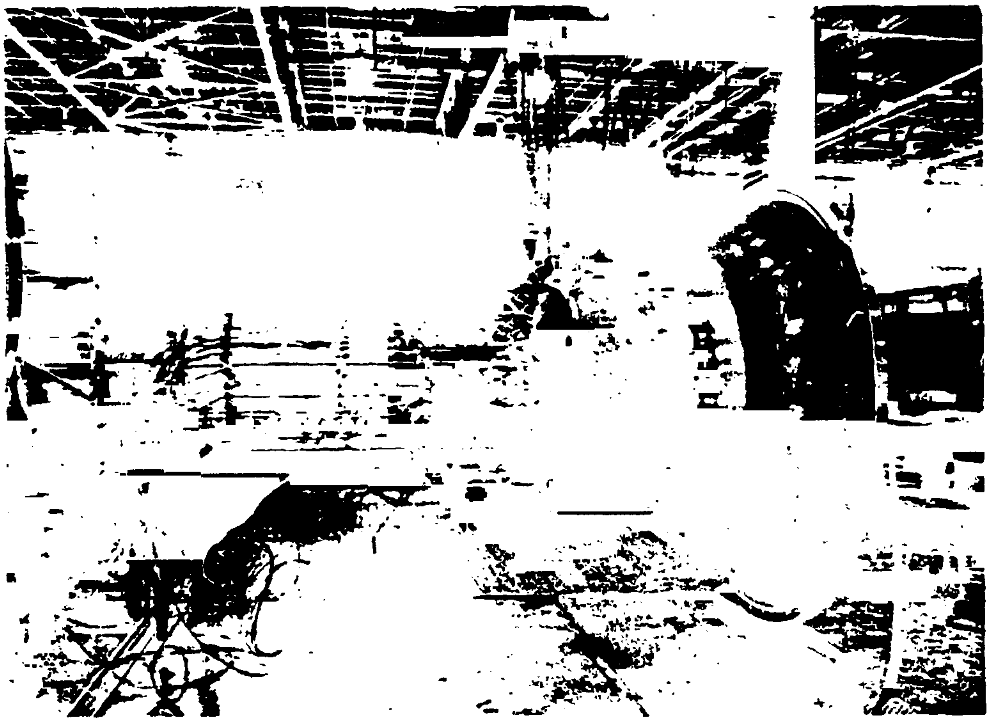

Figure 30 Test Rig Prior to Installation in Pressure Capsule

Combustor Component Rig Performance Test Program

During this report period, shakedown and performance evaluation tests were conducted with the full annular and updated sector test rigs. Both rigs incorporate the design improvements demonstrated during the recent Combustor Sector Rig Supporting Technology Program. The objective of the test programs was to obtain baseline emissions and performance levels for comparison with values demonstrated during the sector rig program. Attaining exit temperature measurements with the full annular combustor rig was of particular importance.

Full Annular Combustor Test Configuration and Test Conditions -- A schematic of the full annular combustor rig is presented in Figure 31. The airflow schedule for the initial full annular configuration was established on the basis of the results from the preceding Combustor Sector Rig Supporting Technology Program, while the overall combustor size was scaled down slightly to reflect the latest engine design. In addition, the carburetor tube design was modified to improve castability and tube installation. The pilot nozzle design is essentially the same as the optimized sector rig design.

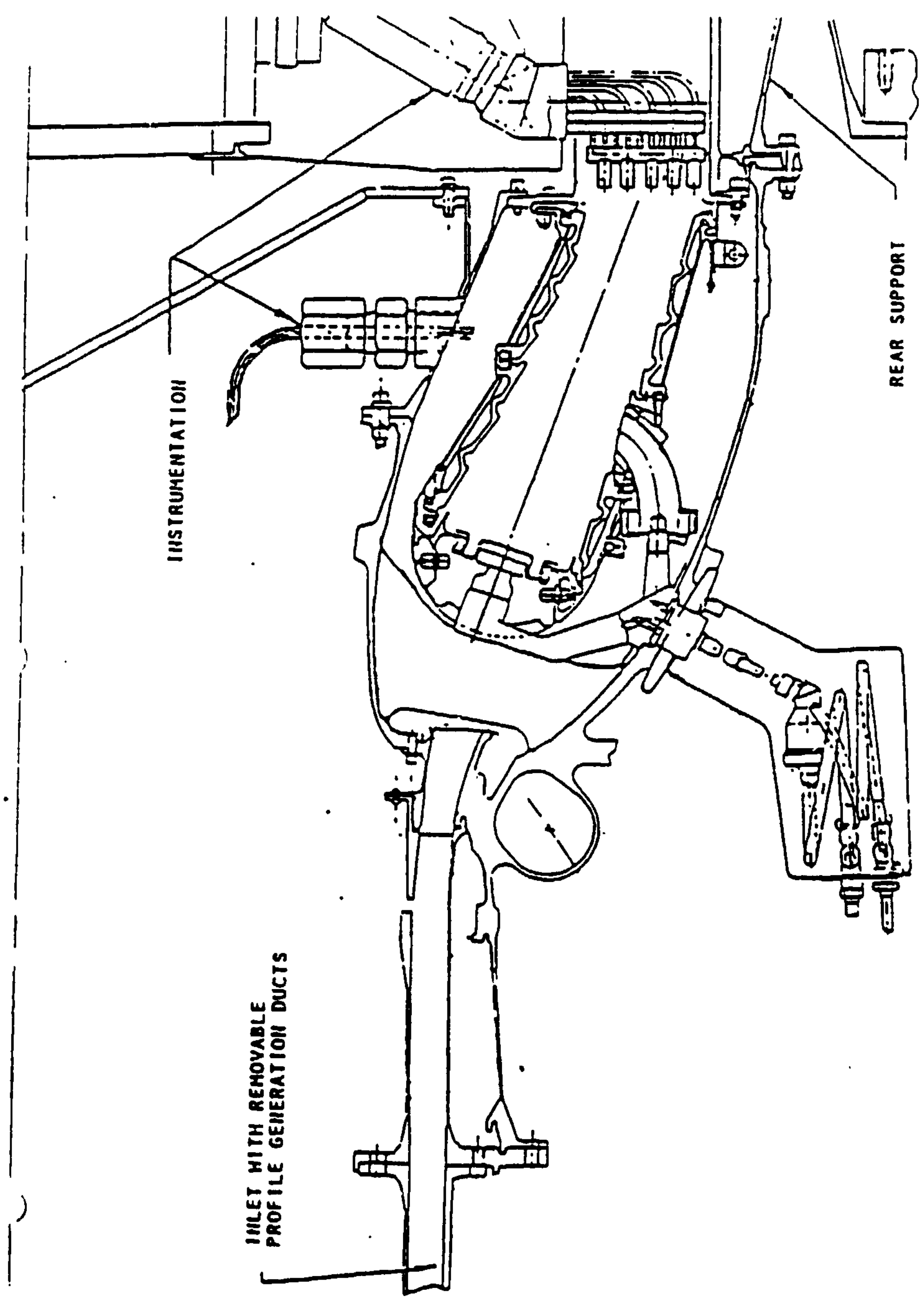

Figure 31 Full Annular Combustor Rig Cross Section

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The test conditions selected for the full annular combustor match the actual engine operating conditions specified by the Environmental Protection Agency for calculating the emissions EPA Parameters. The operating conditions listed in Table 3-XIII correspond to the idle, approach, climb, and sea level takeoff conditions anticipated for the Energy Efficient Engine. Parametric variations of combustor fuel/air ratio were investigated at all operating conditions. At high power conditions, the pilot to main zone fuel split was varied while maintaining a constant total fuel flow. The resulting data permitted identification of the optimum fuel distribution between the pilot and main combustion zones.

TABLE 3-XIII

Condition	Combustor Inlet Pres (psia)	Combustor Inlet Temp (Deg-F)	Combustor Inlet Flow (lb/sec)	Fuel/Air Ratio
Idle	63	391	30.0	0.0098
Approach	167	659	68.0	0.0150
Climb	270*	935	100	0.023
Takeoff	300*	991	100	0.025

* Flow parameter simulation at facility-imposed maximum operating conditions.

Full Annular Rig Test Results -- Figure 32 presents a comparison between the full annular rig (Build 1-Run 1) combustor airflow distribution and the distribution tested in sector rig (build 21) of the preceding Combustor Sector Rig Supporting Technology Program. As indicated, the full annular combustor exhibited higher flow percentages in both the pilot fuel injectors and main zone carburetor tubes.

Top Numbers Shown Represent Sector Rig (Build 21) Values
Bottom Numbers Shown Represent Full Annular Rig (Build 1) Values

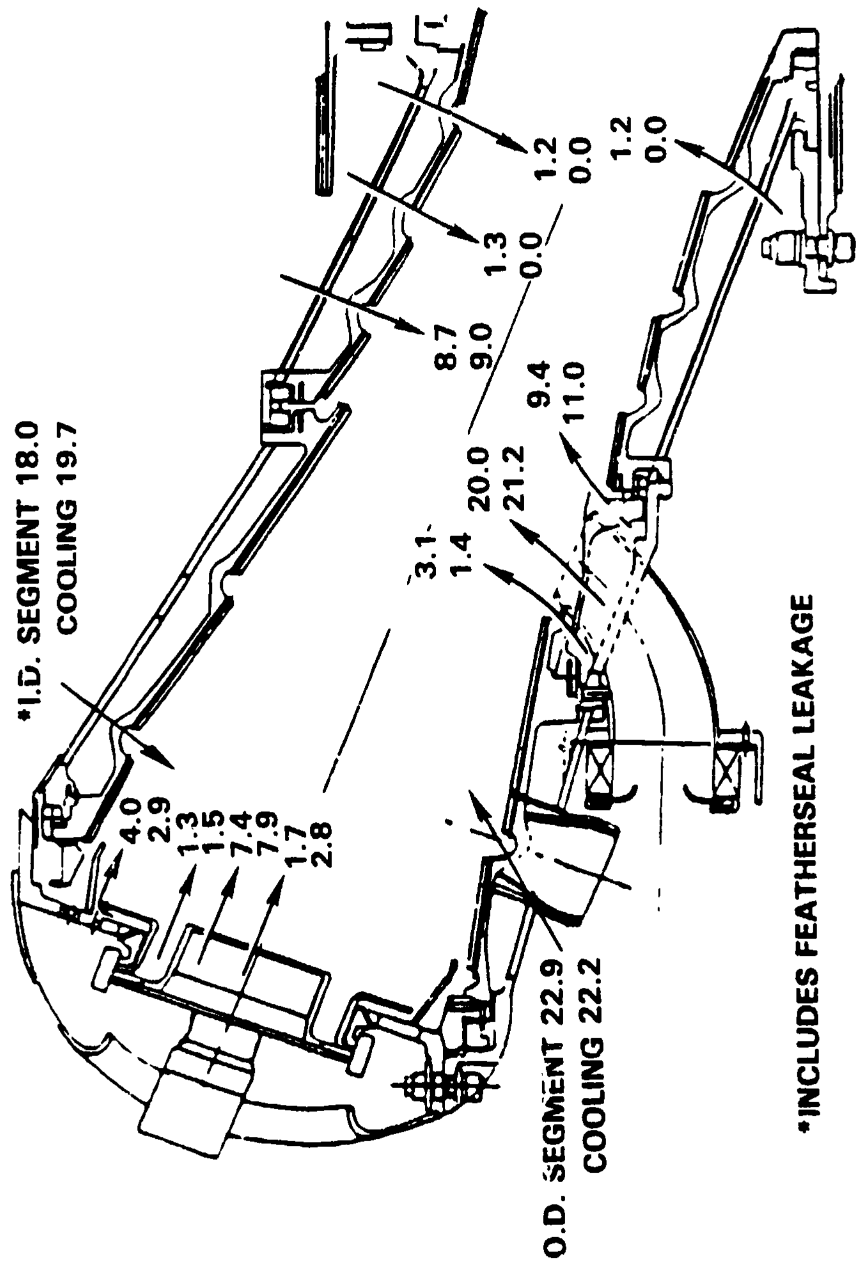

Figure 32 Comparison of Combustor Airflow Distribution

Performance and emissions results are summarized in Table 3-XIV. The overall pressure loss goal of 5.5 percent was exceeded as a result of a slight increase in inner and outer liner pressure losses caused by the elimination of three rows of dilution air holes. The radial exit temperature profile (shown in Figure 33) is comparable to that achieved in the sector configuration used in build 21, with temperatures slightly higher along the outer wall and slightly lower near the inner wall. The goal profile was achieved. Also, pattern factors were below the goal for the two pilot to main zone fuel splits evaluated. The higher pilot/main fuel split (pilot fuel/air of 0.0062) produced a slightly lower pattern factor.

TABLE 3-XIV

FULL ANNULAR COMBUSTOR (BUILD 1) PERFORMANCE SUMMARY

Pressure Drop (Percent Pt3)		Environmental Protection Agency Parameter (EPAP)	Emission Levels Run 1	Goal
Overall:	5.8	Carbon Monoxide	5.6	3.0
Outer Liner:	2.9	Unburned Hydrocarbons	.88	.40
Inner Liner:	2.9	Oxides of Nitrogen	4.88	3.0
		Smoke Number	1.00	<20

Pilot Fuel/Air Ratio	Resulting Pattern Factor
0.0062	.26
0.0034	.30

Figure 34 compares carbon monoxide (CO) and unburned hydrocarbon (THC) emissions trends for sector rig (build 21) and full annular rig (build 1) at the idle operating condition. As shown, carbon monoxide and hydrocarbon levels increased approximately 50 percent, compared to the levels demonstrated in the previous sector rig run. This result is indicative of a lean primary zone. In addition, carbon monoxide and hydrocarbon levels also increased significantly at the approach condition. These increases were probably caused by higher flowing carburetor tubes.

	Condition	Pilot F/A	Performance F/A
● Sector Rig (build 21)	Climb	.0037	.018
□ Full Annular Rig (build 1)	SLTO	.0062	.022
△ Full Annular Rig (build 1)	SLTO	.0032	.022

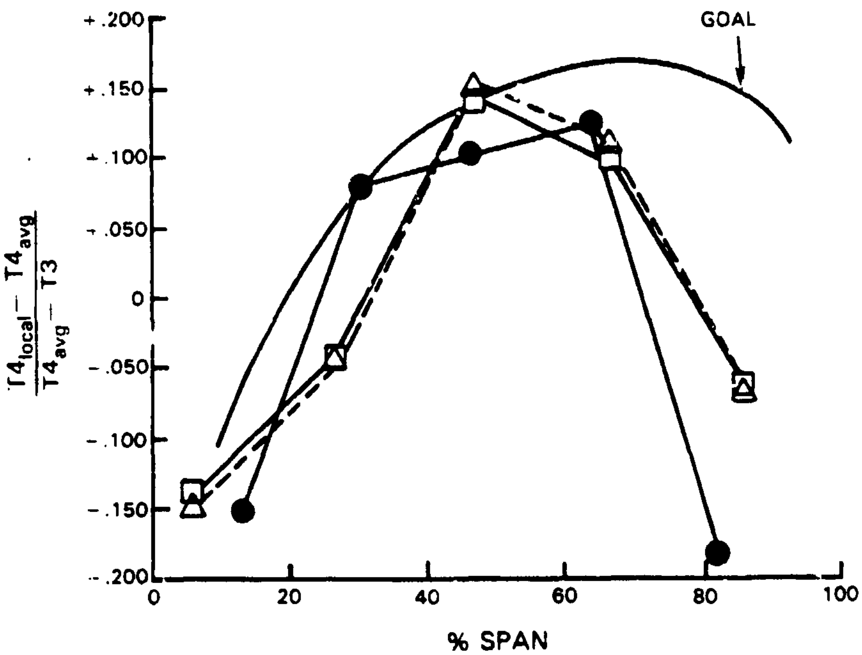

Figure 33 Combustor Radial Exit Temperature Profiles

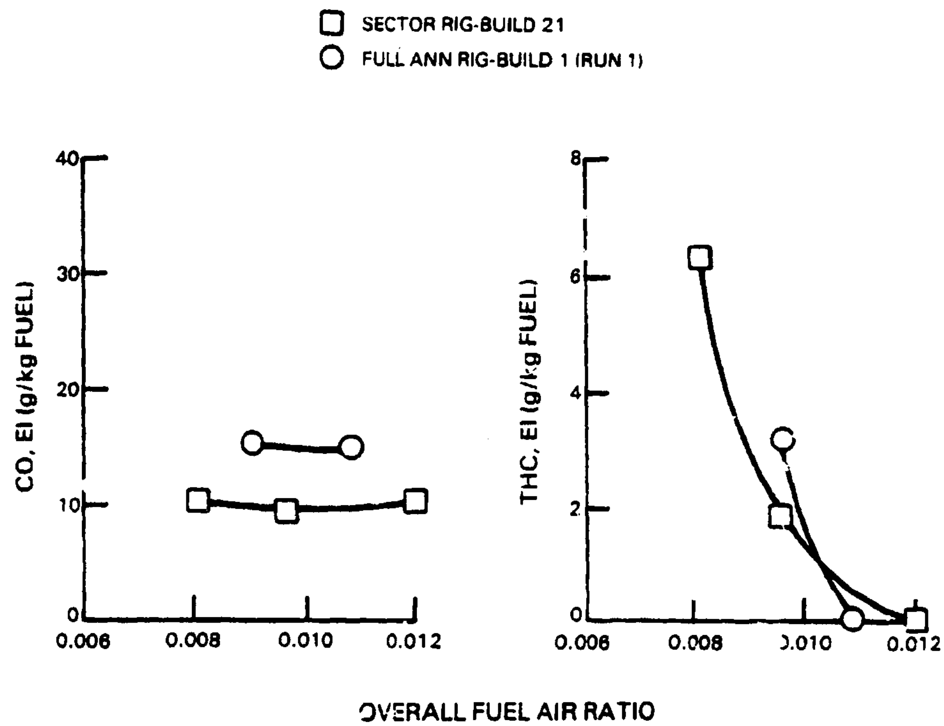

Figure 34 Carbon Monoxide and Unburned Hydrocarbon Emissions Trends at Idle Conditions

An inspection of the combustor at the end of this sequence of testing showed no evidence of thermal distress. Figure 35 shows the excellent condition of the combustor bulkhead. A post-test thermal paint analysis of the combustor liners revealed no significant high temperature streaks during operation at the 0.022 fuel/air ratio. The hottest areas were along the feather seals on both the outer and inner liners. Streak locations and patterns were similar to those exhibited with the sector rig, although temperature levels were significantly lower. The reduction in liner temperature levels was confirmed at full operating conditions during subsequent sector rig testing, as discussed in the following section. Figures 36 and 37 show the typical post-test condition of the inner and outer liners, respectively.

Combustor Sector Rig Configuration and Test Conditions -- Combustor sector rig (build 22), tested in this report period, was similar to the build 21 configuration evaluated in the recently completed Combustor Sector Rig Supporting Technology Program. The only difference was the use of cast carburetor tubes with effective airflow areas duplicating the initial full annular rig configuration.

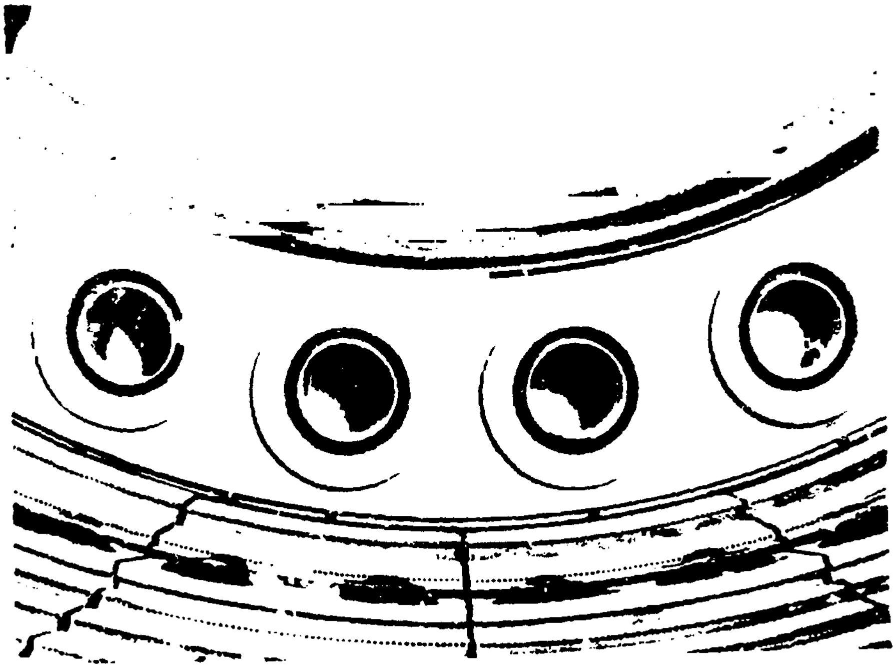

Figure 35 Post-Test Condition of Combustor Bulkhead

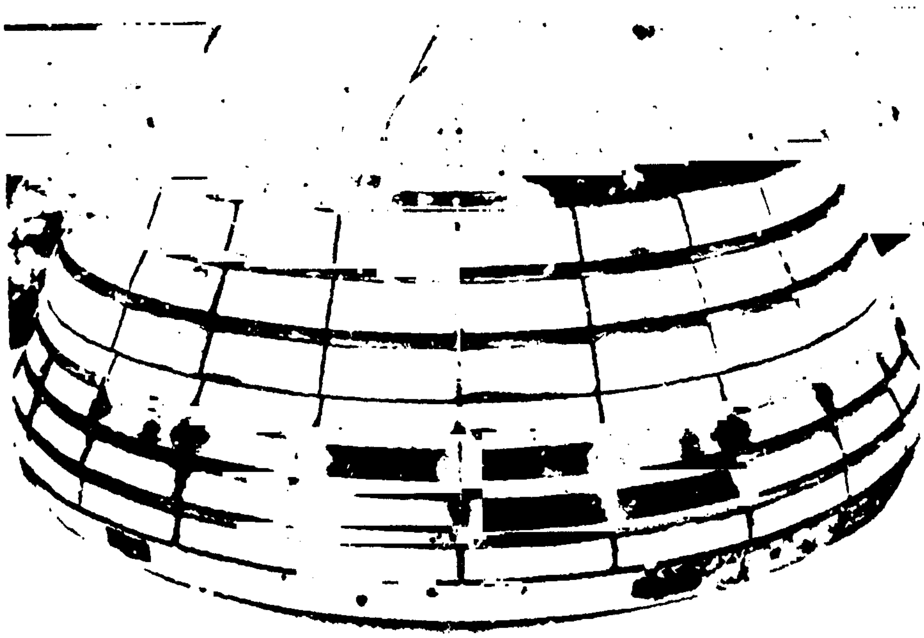

Figure 36 Typical Post-Test Condition of Inner Liner

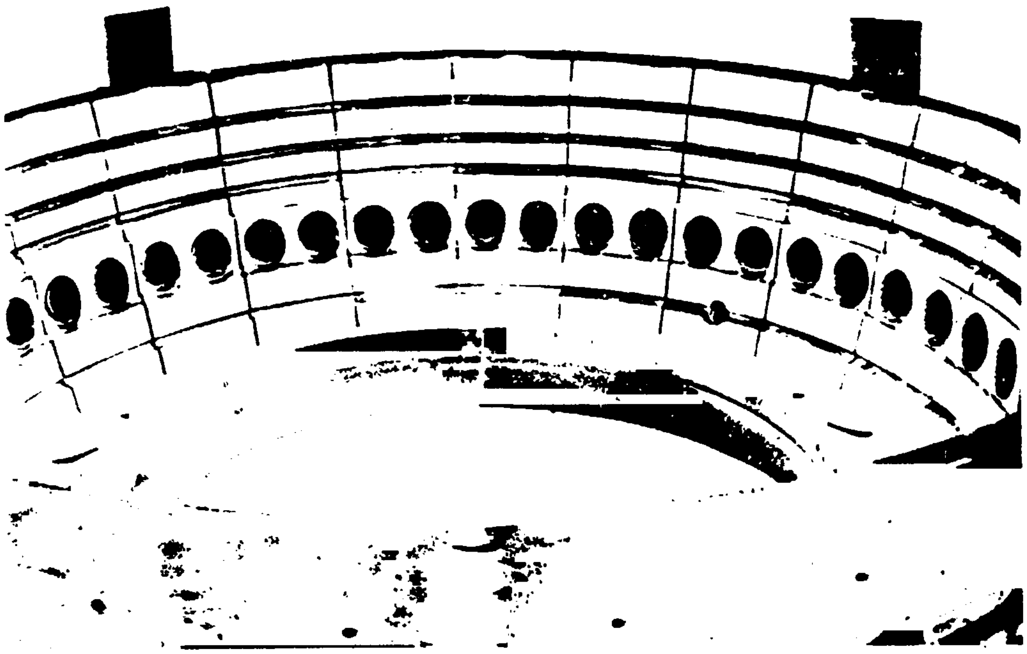

Figure 37 Typical Post-Test Condition of Outer Liner

Testing was conducted at the operating conditions listed in Table 3-XV. Parametric variations in fuel/air ratio were investigated at idle and the two high power pressure level takeoff conditions.

TABLE 3-XV

Condition	Combustor Inlet Pres (psia)	Combustor Inlet Temp (Deg-F)	Combustor Inlet Flow (lb/sec)	Fuel/Air Ratio
Idle	63	391	7.5	0.0098
Approach	168	659	16.1	0.0150
Takeoff*	300	991	25.4	0.0250
Takeoff	444	991	37.3	0.0250

* Reduced pressure takeoff conditions

Sector Rig Test Results -- Testing with the initial sector rig configuration (build 22) was directed at obtaining baseline emissions, performance and liner temperature data with the cast carburetor tubes. The primary objective was to obtain additional data to better understand the increase in carbon monoxide and unburned hydrocarbons emissions at the approach condition and the decrease in maximum liner temperatures observed during the initial full annular combustor test effort. Since the cast carburetor tube was the only modification, a direct comparison with previous sector rig results was possible.

The combustor airflow distributions for sector rig builds 21 and 22 are shown in Figure 38. As expected, core and secondary airflows with the cast carburetor tubes were higher than for build 21. Carbon monoxide and unburned hydrocarbon emissions at the approach condition did increase significantly, which confirmed that high carbon monoxide and unburned hydrocarbon emissions observed during the full annular combustor test were a result of the higher flowing carburetor tube.

Pattern factor increased as a result of the more peaked exit radial temperature profile. This condition, as shown in Figure 39, is attributed to excess dilution air in the rearmost inner and outer liners. The profile would be very similar to the full annular rig profile if the last three rows of dilution air holes were utilized to reduce temperatures at the peak.

A post-test thermal paint analysis showed that significant reductions in liner temperatures were achieved with the cast carburetor tube design. Figures 40 and 41 show the typical post test condition of the inner and outer liners, respectively.

Main zone inner segment streak temperatures are compared in Figure 42. As indicated, temperatures at the segment axial edges were reduced as much as 200°F and local hot spots were reduced by approximately 100°F. The maximum estimated liner temperature is 1900°F at the sea level takeoff (hot day) design point.

Main zone outer liner temperatures were unchanged when compared to sector rig build 21. Also, they were equal to or below design values and essentially streak free. The improved liner temperature patterns with the cast carburetor tubes were similar to those patterns observed in the initial full annular rig test.

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Top Numbers Shown Represent Sector Rig (Build 22) Values
Bottom Numbers Shown Represent Sector Rig (Build 21) Values - (Build 21 configuration was basically identical to Build 20 except the pilot nozzles were shimmed .1875 inch)

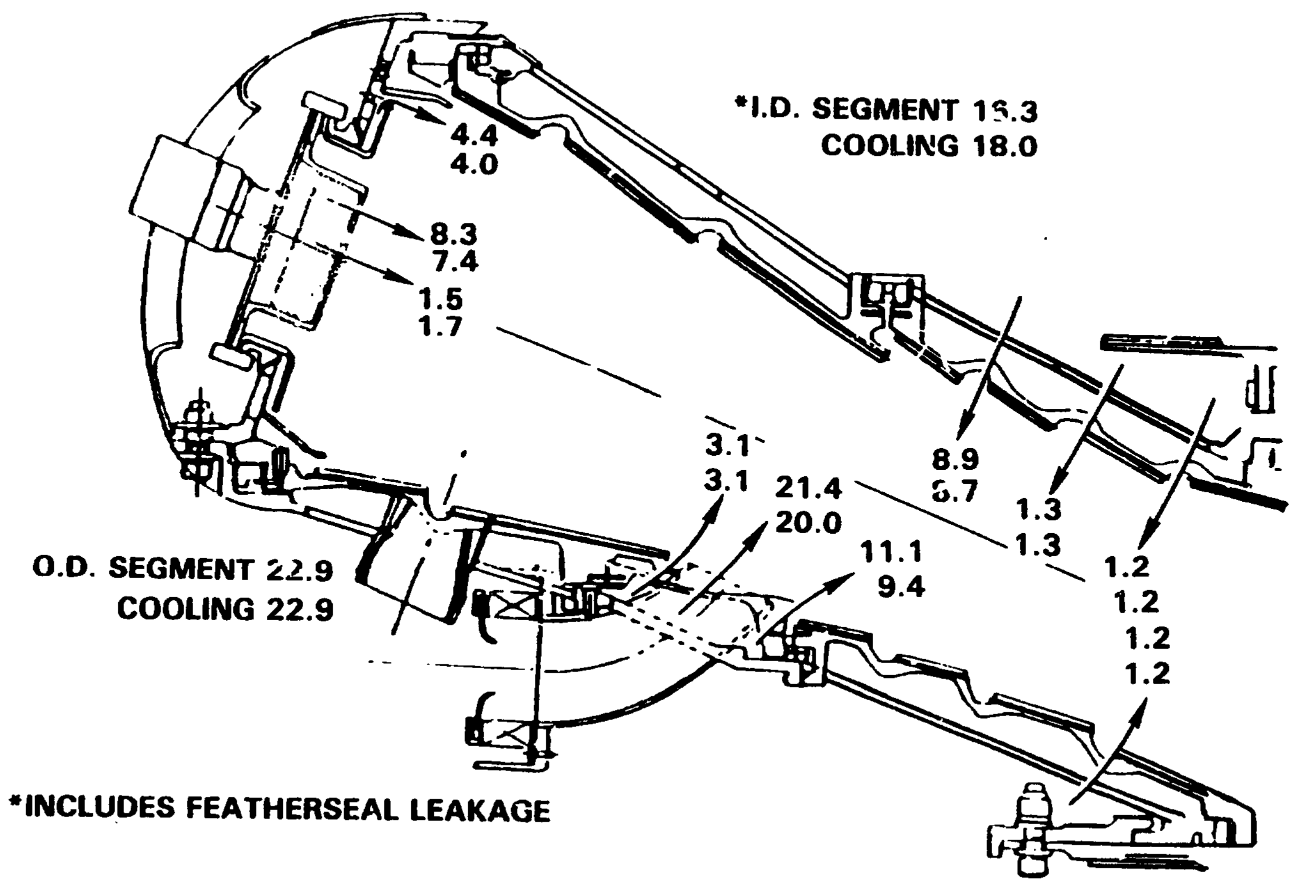

Figure 38 Combustor Airflow Distribution

AVERAGE EXIT RADIAL TEMPERATURE PROFILE

	Condition	Pilot F/A	Performance F/A
○ Sector Rig (build 22)	SLTO	.0036	.024
△ Sector Rig (build 21)	CLIMB	.0037	.018
□ Full Annular Rig (build 1)	SLTO	.0032	.022

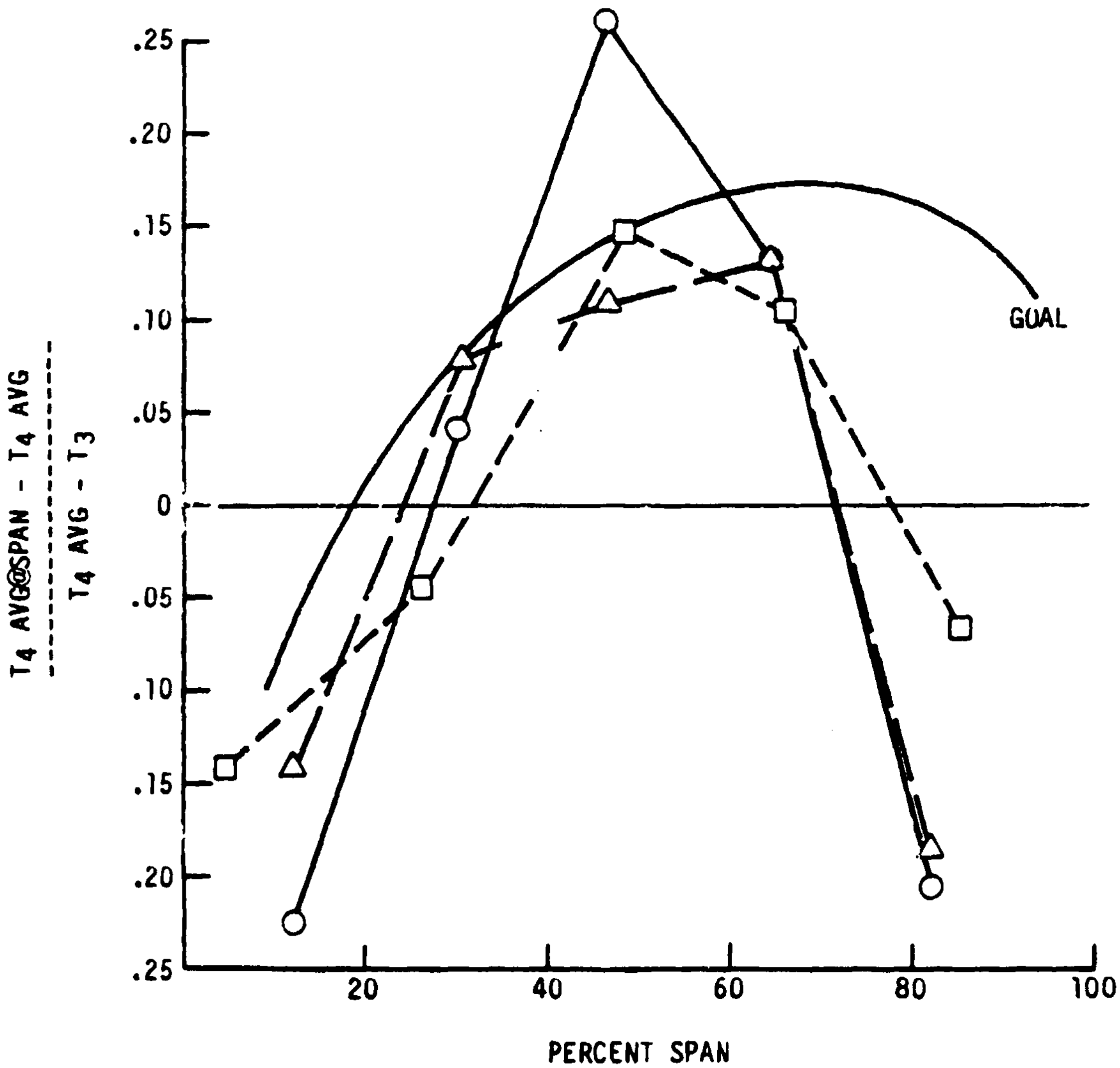

Figure 39 Sector Combustor Rig Exit Radial Temperature Characteristics

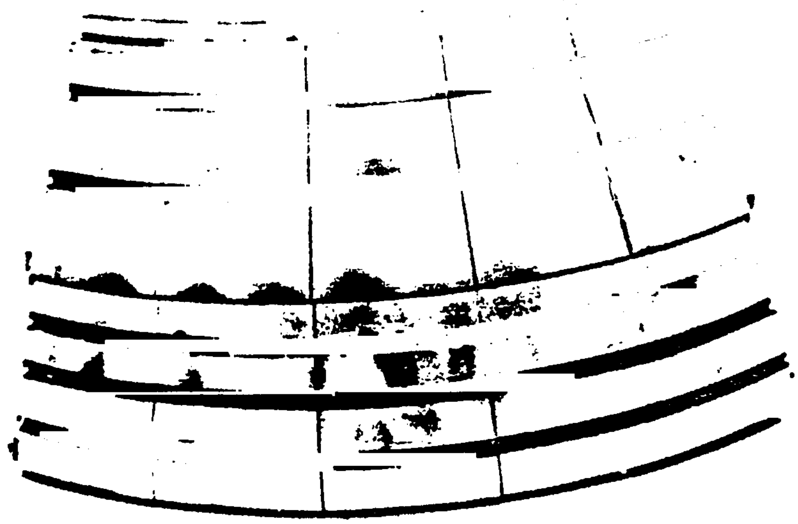

Figure 40 Typical Post-Test Condition of Inner Liners

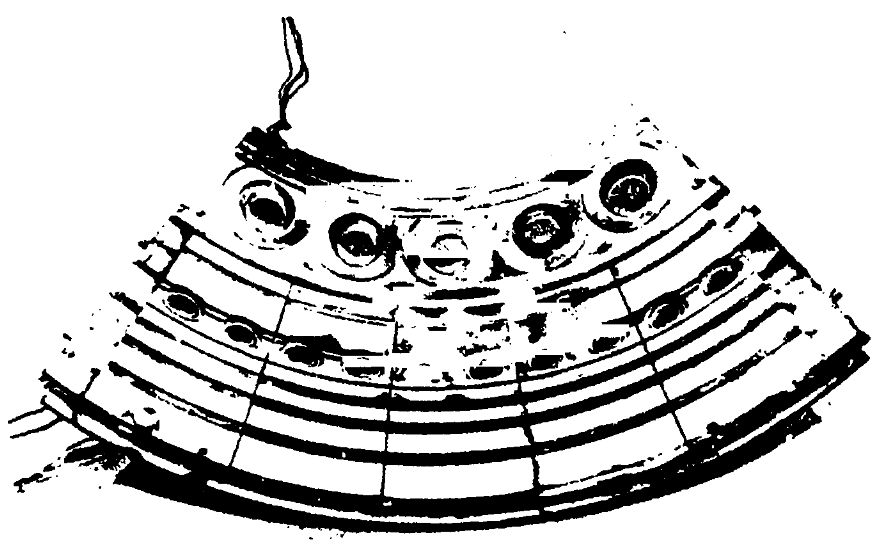

Figure 41 Typical Post-Test Condition of Outer Liners

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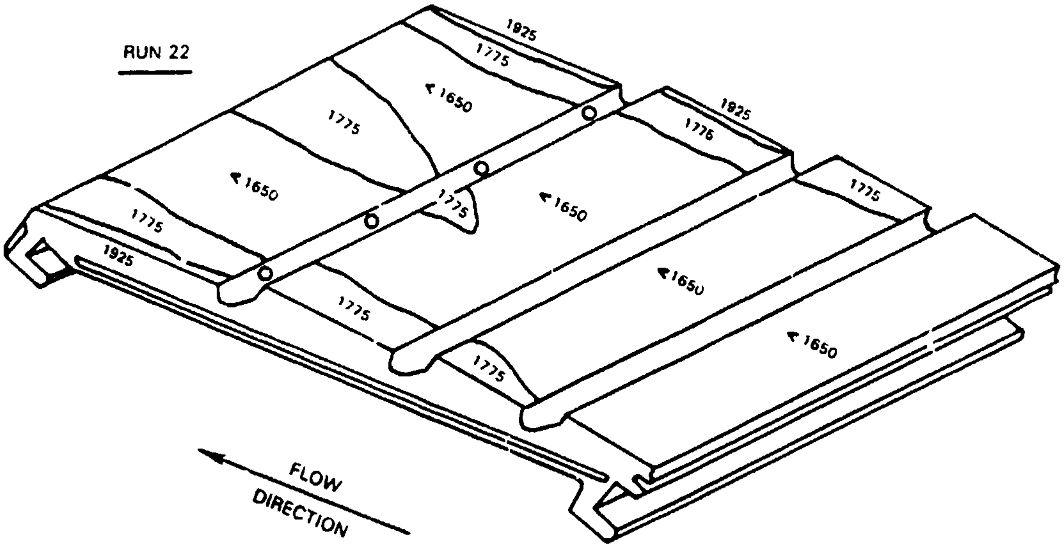

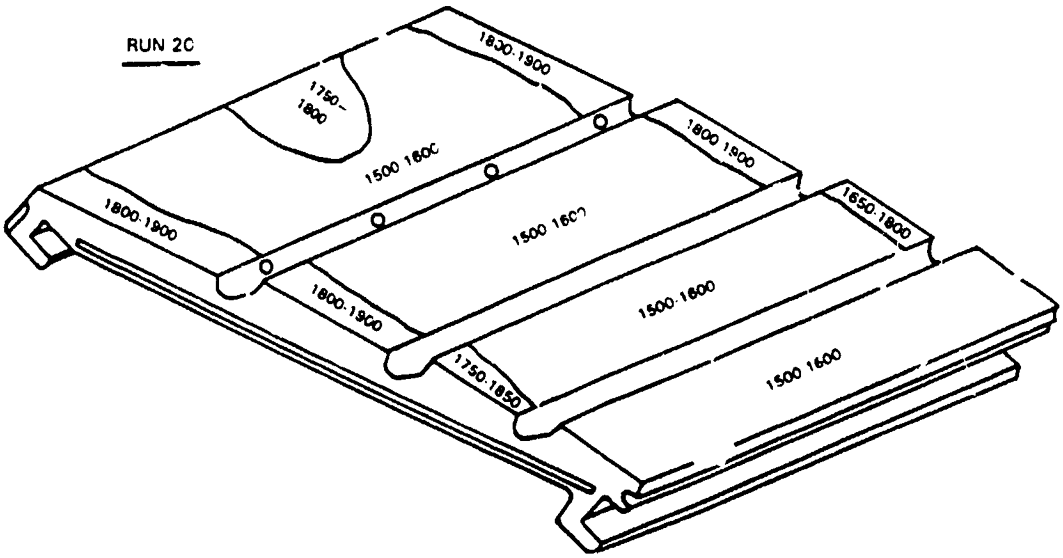

Figure 42 Temperature Map of Main Zone Inner Liner Segment

3.2.6.4 Supporting Technology

3.2.6.4.1 Diffuser/Combustor Model Test Program

All technical work for this supporting technology program has been completed. Program results are published in NASA Report CR-165157.

3.2.6.4.2 Combustor Sector Rig Test Program

3.2.6.4.2.1 Objective

The objective of this supporting technology program was to evolve and experimentally substantiate the design features of the two stage (aerated nozzle pilot and carburetor tube main zone) combustor. The modifications formulated during the program were aimed at reducing emissions, pattern factor, cost, and weight as well as improving combustor durability and maintainability. Specific emissions and performance goals were the same as those for the combustor component. All technical work for this supporting technology program was completed in a previous reporting period. A Contractor Report, presenting the salient results of this program, has been submitted to NASA for review and approval.

3.2.7 High-Pressure Turbine

3.2.7.1 Overall Objective

Develop the technology to design a highly efficient single-stage high-pressure turbine. Fabricate and test a full-scale high-pressure turbine rig to substantiate the technology advancements selected for this component. The performance goal for this turbine is 88.2 percent cooled efficiency. Design goals are a combined cooling and leakage flow of 11.2 percent of the total component airflow and life of 10,000 hours for the blades and vanes and 20,000 hours for the disk. In addition, blade and vane coating goal life is 6,000 hours.

3.2.7.2 Component Program Overview

The overall task effort consists of a component effort and five supporting technology subtasks. The component effort is composed of the analysis and design of the high-pressure turbine component and a high-pressure turbine rig test program. The five supporting technology programs are (1) the Leakage Test Program, (2) the Supersonic Cascade Test Program, (3) the Cooling Model Test Program, (4) the Uncooled Rig Test Program, and (5) the Material Fabrication Program. Figure 43 shows the relationships between these activities and their relationships to contract Tasks 1 and 4. The work plan schedule for the component effort is shown in Figure 44 and critical milestones are noted.

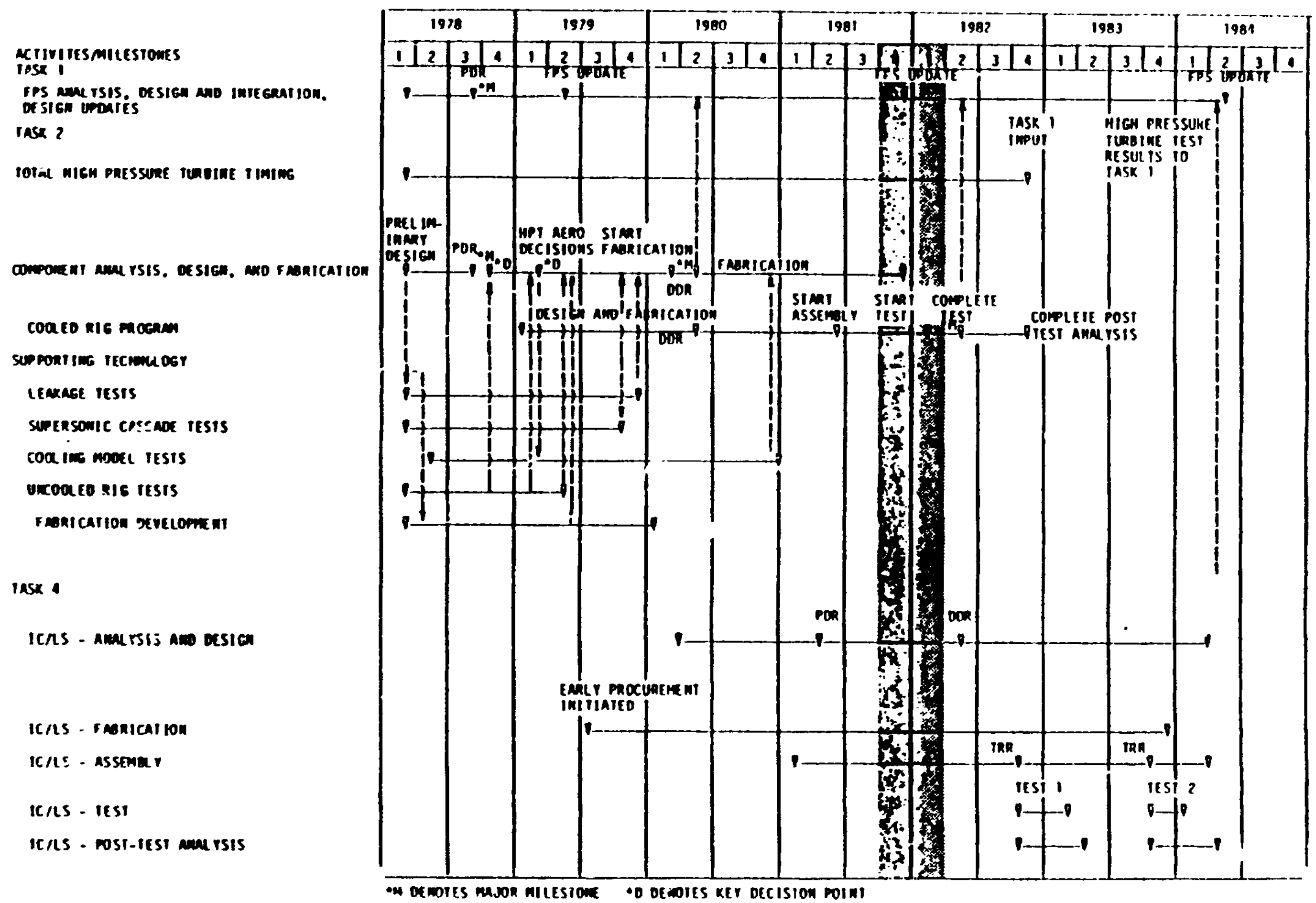

Figure 43 High-Pressure Turbine Program Logic Diagram

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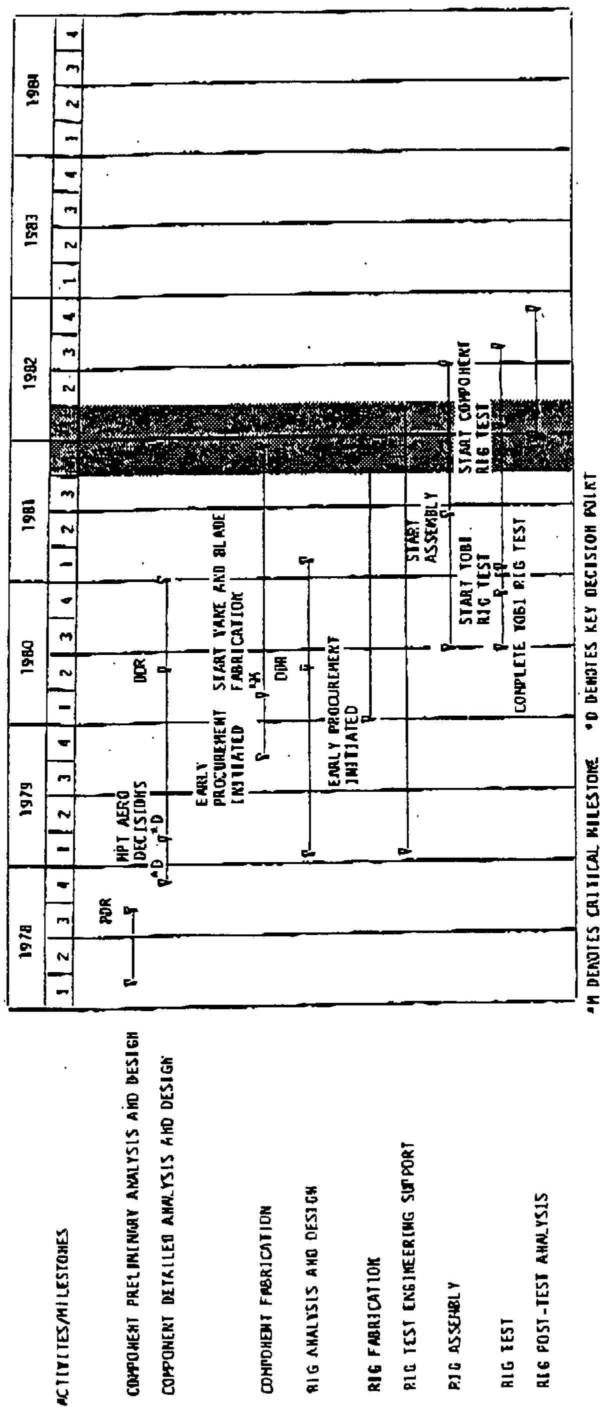

Figure 44 High-Pressure Turbine Component Effort Work Plan Schedule

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3.2.7.3 Component Effort

3.2.7.3.1 Objective

Conduct the design, analysis, hardware procurement and rig testing necessary to develop a full-scale high-pressure turbine that meets the established goals.

3.2.7.3.2 Scope of Total Work Planned

The analysis and design effort consists of a preliminary analysis and design phase and a detailed analysis and design phase. The rig program comprises the six subtasks shown in Figure 44.

A six-month preliminary design activity is conducted to establish the feasibility of the high-pressure turbine as proposed for the Task 1 flight propulsion system. The studied designs provide configuration definitions for the supporting technology programs. This preliminary activity results in layout drawings and substantiating design data, which are presented to NASA at a preliminary design review in September 1978.

Approximately two months after the preliminary design review, the detailed design work on the high-pressure turbine starts. Results available from the supporting technology programs are used to substantiate or improve the configurations established in the preliminary design. Significant supporting technology input is provided by results of the uncooled high-pressure turbine rig testing. The performance results from this rig allow selection of optimized single-stage aerodynamics. The results of the detailed design effort are completed layout drawings and substantiated design data that form the basis for a detailed design review to be conducted for NASA in May 1980. Detailed drawings are scheduled for completion approximately two months later. The design and analysis of parts peculiar to the test rig are conducted concurrently with the detailed design of the component. Fabrication of rig test parts begins in late 1979 as the designs of the rig parts are completed. Fabrication of the component hardware is not initiated until late-1979, after the feasibility of the vane/blade casting process has been established.

A component rig test program consisting of three phases is conducted. The first phase of this test program assesses tangential on-board injection in order to improve injection nozzle performance. This phase is initiated in November 1980 and lasts approximately three months. The second phase comprises full-stage (vane and rotor) testing to determine the overall design and off-design performance of the high-pressure turbine component. The third phase comprises a first vane annular cascade test to determine vane aerodynamic performance. The second and third phases of this component rig test program commence in January 1982 and last approximately six months.

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3.2.7.3.3 Technical Progress

3.2.7.3.3.1 Summary of Work Previously Completed

All analysis and design effort associated with the high-pressure turbine and its companion 'warm' rig has been completed. Results of these efforts are presented in NASA CR-165608.

Fabrication efforts prior to the current reporting period focused on component and rig-unique hardware required for the component test rig program. These efforts are summarized below.

Component Hardware: Problems experienced by the vendor in casting PWA 1480 single crystal vanes led to a mutual decision with NASA to replace the single crystal material with PWA 1422 directionally-solidified material for use in component rig testing. Subsequently, the first set of vanes was cast in PWA 1422 and machined for use in the component rig.

Finish machining of the high-pressure turbine disk (first compaction) was completed along with work on the front and rear sideplates, rotating high-pressure compressor discharge seal, active clearance control hardware, and number 4 bearing compartment parts.

Rig-Unique Hardware: Approximately 90 percent of all hardware required for the component rig (exclusive of component parts) was fabricated and delivered to Pratt & Whitney. Major rig hardware items completed include the exit instrumentation traverse ring, the inner diameter exit flow path, disk gauge spacer, outer diameter inlet case, front bearing buffer air seal, disk rear thrust balance seal support, and front bearing stub shaft.

Work was initiated on preparation of general instructions (GI's) which are necessary for specifying data acquisition processes and data reduction procedures for the computerized data systems. Work was also started on preparing the detailed component rig test plan for eventual submittal to NASA. Analytical work is also in process to define secondary flow system automatic valve operation that is compatible with the rig supervisory control.

Fabrication effort was completed for the fixtures required to cold airflow various rig assemblies. Cold airflowing of the tangential on-board injection air duct was completed with the results showing a 4 percent overflow condition at the test point. Review of the data is in process and scheduled for completion in the next reporting period.

3.2.7.3.3.2 Current Technical Progress

All work conducted during this reporting period was directed toward completing fabrication of all high-pressure turbine component and rig parts and assembly of the component rig. These efforts are summarized in the following subsections.

High-Pressure Turbine Component Fabrication

Turbine Vane Fabrication: Machining and assembly work, including welding of platform plenum covers and impingement plates, on PWA 1422 directionally solidified material vanes was completed. An example of a finished vane is shown in Figure 45. Bench airflow testing of the vane indicated that the cooling passage average airflow was 14 percent less than design intent. This condition was considered acceptable for the rig test since the inlet temperature will only be 800°F and the low cooling flow can be bookkept in the efficiency calculation. The reason for the low flow condition was determined to be slightly smaller than intended holes in both the impingement tube insert and the airfoil walls. Measurements of the vane nozzle throat area showed the area to be 2 percent higher than desired.

Turbine Blade Fabrication: Machining of the PWA 1480 single crystal material blade castings was completed. An example of a finish machined blade is shown in Figure 46. A measurement of the trailing edge flow area indicated an area 4 percent less than intended.

Bench airflow testing on the complete blade indicated that average cooling passage flow was within 1 percent of design intent. The blades were then made available for rig assembly.

Turbine Disk and Attachments: Turbine disk fabrication efforts, including successful electro-chemical machining of the 54 curved, elliptical cooling air feed holes, was completed. These air holes are designed to conduct cooling air for the blades from the disk front side to the root attachment area of the blade. The process involved development of an electro-chemical machining procedure requiring accurate feed control of the electrode. Following spin balancing, the completed disk was made available for rig assembly. Figure 47 illustrates the disk after completion of the cooling air holes and prior to blade attachment final broaching. In addition, rear disk side plate machining was also completed.

Active Clearance Control System: Machining of the blade outer air seal segment castings was completed along with the front rail support and manifold assembly.

Fabrication of all remaining component parts for the high-pressure turbine section was completed prior to this report period.

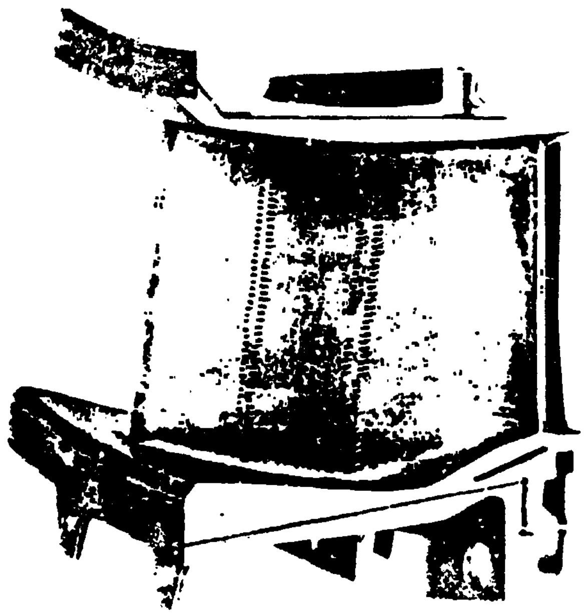

Figure 45 Finish Machined PWA 1422 High-Pressure Turbine Vane

Figure 46 Finish Machined PWA 1480 High-Pressure Turbine Blade

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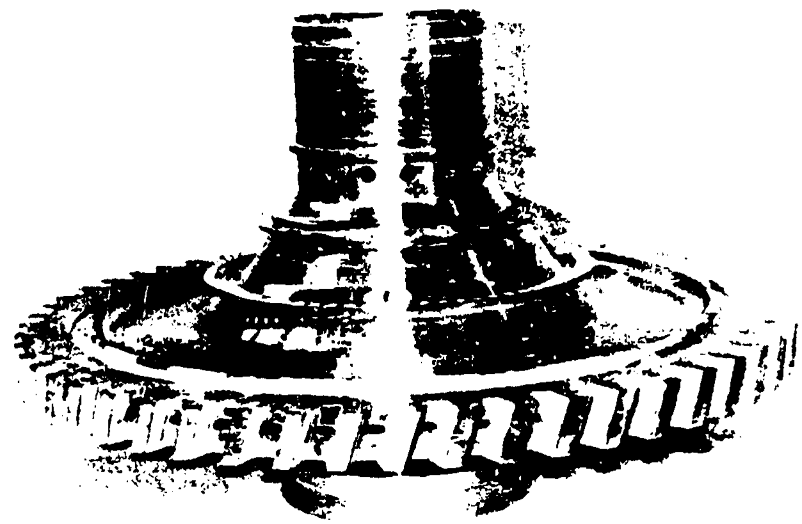

Figure 47 High-Pressure Turbine Disk Prior to Blade Broaching

High-Pressure Turbine Component Rig

Rig Fabrication: Fabrication of all hardware items for rig assembly, including unplanned airseal vibration dampers, was completed.

Engineering and Support: Preparation of general instructions for data acquisition systems was completed along with a subsequent computer checkout of these instructions. A detailed high-pressure turbine rig test and instrumentation plan was finalized, printed and submitted to NASA.

Rig Assembly: Assembly of the instrumented component and rig hardware was completed during the report period. As shown in Figure 48, the rig consists of an inlet section, test section and exhaust section. The main aerodynamic and low leakage technology features being tested in the rig are shown in Figure 49.

Secondary Flow Systems: The rig secondary cooling system is designed to simulate the Energy Efficient Engine. Air supply lines, as shown in Figure 50, supply metered coolant flows to the vane, tangential on-board injection (TOBI) system, bore cavity, mini-tangential on-board injection system, disk rear side thrust balance flow, and active clearance control system.

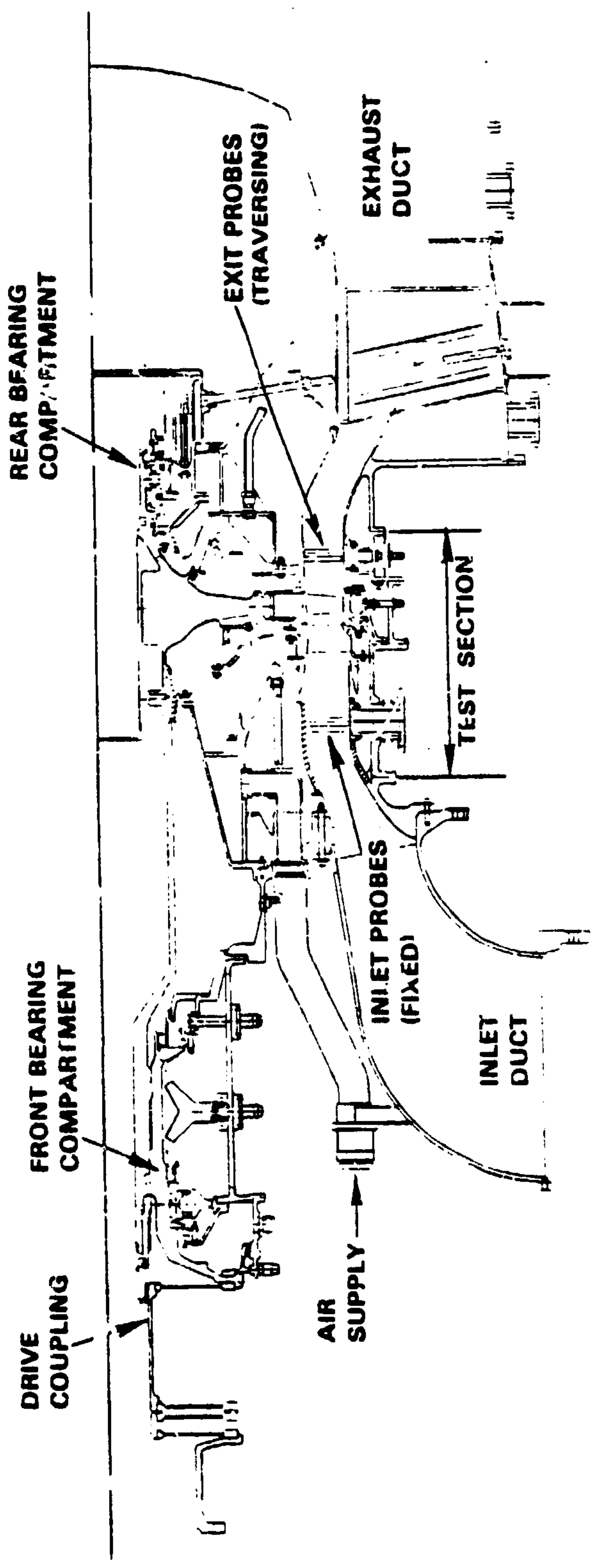

Figure 48 High-Pressure Turbine Component Test Rig

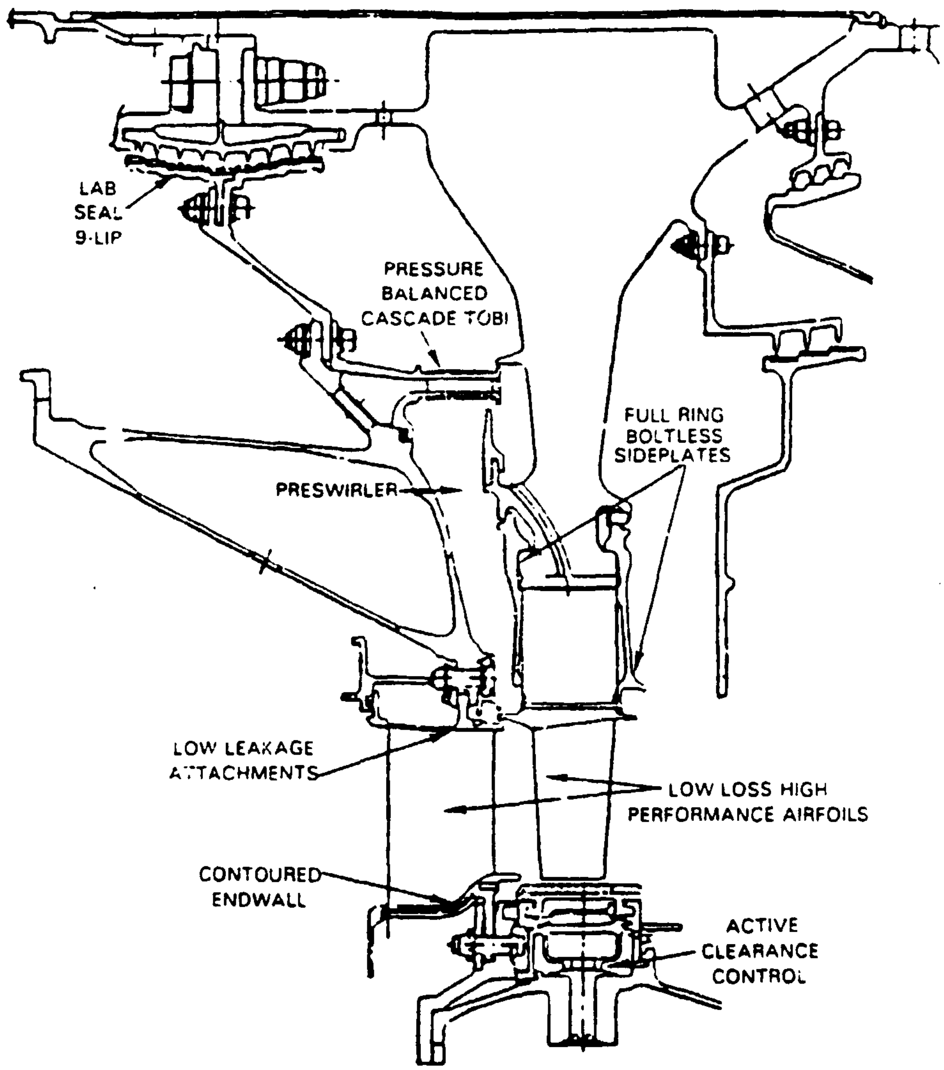

Figure 49 High-Pressure Turbine Component Aerodynamic and Low Leakage Technology Features

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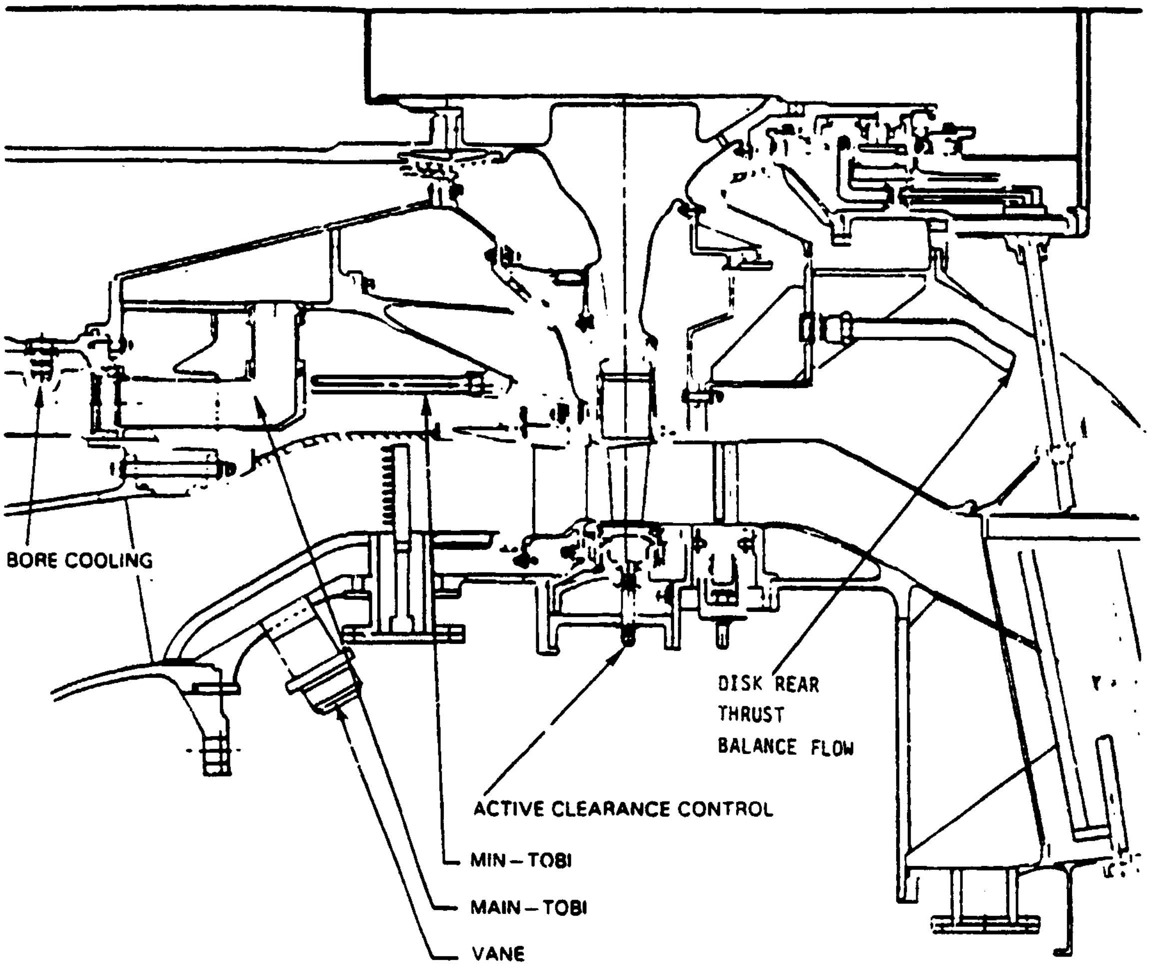

Figure 50 Turbine secondary Flow System Air Supply Lines

Separate controls for each of these secondary flows as well as the main flow are provided. The active clearance control system is capable of varying the air temperature over a range of approximately 300°F to facilitate blade tip clearance change. The main flow temperature will be 800°F with the secondary air temperature being 145°F to simulate engine operating conditions (coolant to mainstream temperature ratio).

Bench airflow tests were conducted on the following secondary airflow systems:

1. Vane assembly
2. Active clearance control system
3. Tangential on-board cooling air injection duct
4. Disk and blade assembly

Specific coolant and leakage flow splits were determined from these flow calibrations to aid in the analysis of rig test data. Results of this flow testing for each of the secondary airflow systems follow.

1. Vane Assembly

The vane inner diameter platform area leakage was approximately 2 times the predicted flow. The outer diameter platform area overflowed by approximately 1.5 times the predicted level. The vane cooling airflow was 15 percent under prediction which confirmed results of bench airflow tests on individual vanes.

2. Active clearance control system

The active clearance control system overflowed its prediction by 20 percent at a 2.0 pressure ratio. The holes metering flow for the vane outer diameter cooling air overflowed by 15 percent at 1.5 pressure ratio. Four of these holes were plugged for subsequent rig running to reduce flow to the design intent.

3. Tangential On-Board Injection (TOBI) Cooling Air Duct

The main tangential on-board injection duct flowed 4 percent higher than predicted but was judged to be acceptable for the rig test. The mini-TOBI, which swirls air in front of the turbine disk, overflowed by 18 percent. A total of eight holes was plugged in this duct to reduce flow to design intent.

4. Disk and Blade Assembly

Cooling airflow to the blades was only slightly underflowing the prediction and was in good agreement with the bench cooling flow results. Leakage past the attachment area of the blade was two times the prediction. This increase in leakage resulted because all rotor cold flowing was done with the rotor static. The sideplates and damper seals would be expected to seal more effectively under the centrifugal loading applied during rotating conditions.

Rig Instrumentation

Figure 51 presents an instrumentation map of the complete high-pressure turbine rig. The instrumentation employed, including type of sensor, quantity and location, was determined on the basis of analyses and previous test experience.

As shown in Figure 51, the rig contains extensive instrumentation to monitor and record turbine aerodynamic behavior. Pressure and temperature sensors were installed at the inlet, throughout the flowpath and secondary system, and at the exit to obtain accurate cooled turbine performance. Also, static pressure taps were installed on selected vanes to acquire pressure distribution data. In certain areas, instrumentation was provided for redundant measurements to enable supervisory control of test conditions, facilitate establishment of data validity, enhance measurement precision, and minimize downtime resulting from sensor failures.

Turbine blade tip clearance will be measured with laser proximity probes. Structural characteristics will be monitored by strain gages on some probes and rig hardware and vibration pickups on rig and test facility hardware.

Performance Instrumentation

The inlet section was instrumented with stationary pressure and temperature instrumentation rakes to properly establish rig inlet conditions. Three vanes were instrumented with extensive airfoil instrumentation. Two vanes will determine airfoil pressure distributions and one vane will determine suction wall film effectiveness through the use of thermocouples.

Laser proximity probes were installed at four circumferential locations to measure blade tip clearance at different operating conditions. Figure 52 shows the laser probe installation through the outer air seal shoe.

Circumferential traversing instrumentation was installed to record turbine exit temperature, pressure and air angle. The traversing instrumentation ring contains four kiel-headed total pressure pole rakes, four kiel-headed total temperature rakes and four radially traversing air angle wedge probes. The total range of the circumferential travel for the ring is 30 degrees, corresponding to two vane pitches. The circumferential positions of the rakes and wedge probes were selected to provide performance data in four quadrants of the rig to permit potential circumferential variations in blade tip clearance to be properly bookkept.

Structural Integrity Instrumentation

Certain condition monitoring sensors were installed in the rig to record vibration and bearing pressure and temperature as a safeguard to ensure the structural integrity of the turbine rig. In addition, the high pressure air seals were instrumented with strain gages as a precaution against a potential fatigue failure. Only static components were instrumented with strain gages. These components are identified in Figure 53.

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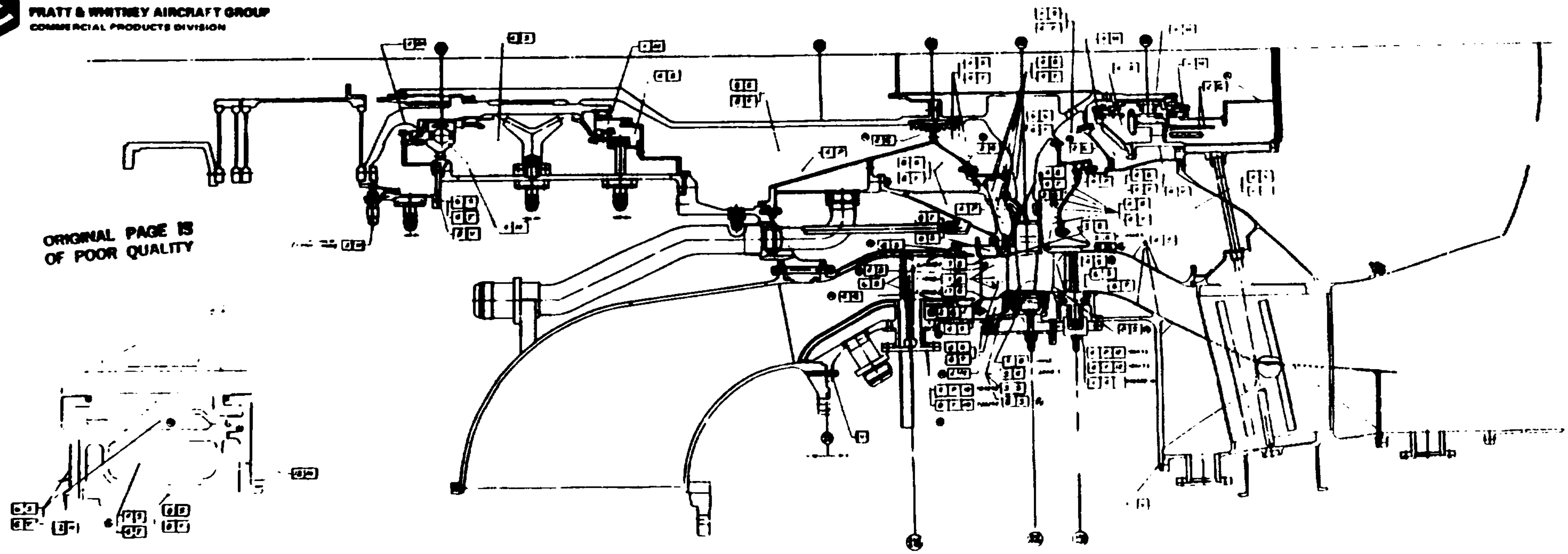

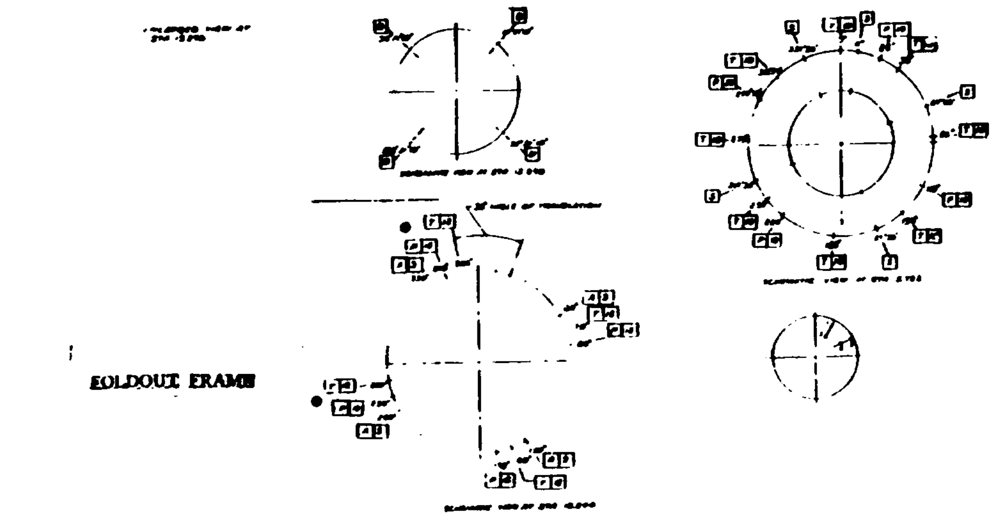

Notes:
1 - All schematic views are looking forward
2 - Radial sensor locations numbered from outer diameter to inner diameter
3 - Angular locations are clockwise from top dead centerline (TDC)
4 - Scale: None

Instrumentation Identification

[1][T][4]

[1] Number of probes (numerical)
[T] Type of probe (alphabetic symbol)
[4] Number of sensors (numerical)

[T] Type of Probe

A - air angle
B - acoustics
D - measurement (blade tip, etc.)
E - emissions
F - airflow
G - strain gage
H - metal temperature

N - rpm
P - total pressure
S - static pressure
T - temperature
V - vibration
Z - miscellaneous

FOLDOUT FRAME

FOLDOUT FRAME

Figure 51 Instrumentation Map of High-Pressure Turbine Component Rig

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Figure 52 Laser Probe Installation Through Outer Air Seal In the High-Pressure Turbine Rig

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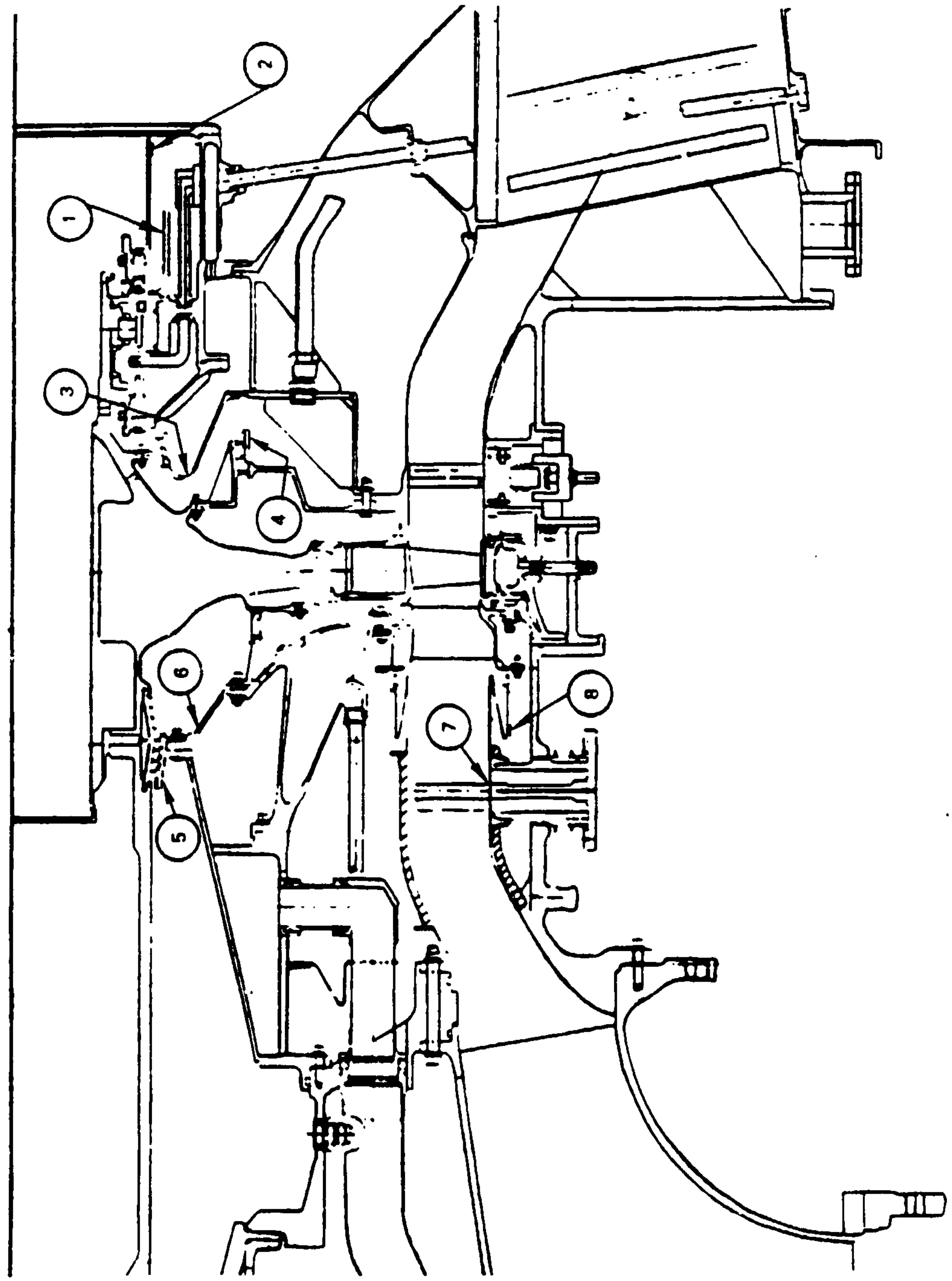

Figure 53 Location of Turbine Structural Integrity Strain Gage Instrumentation

The completed rig, as it appeared prior to shipment to the test cell, is shown in Figures 54 and 55.

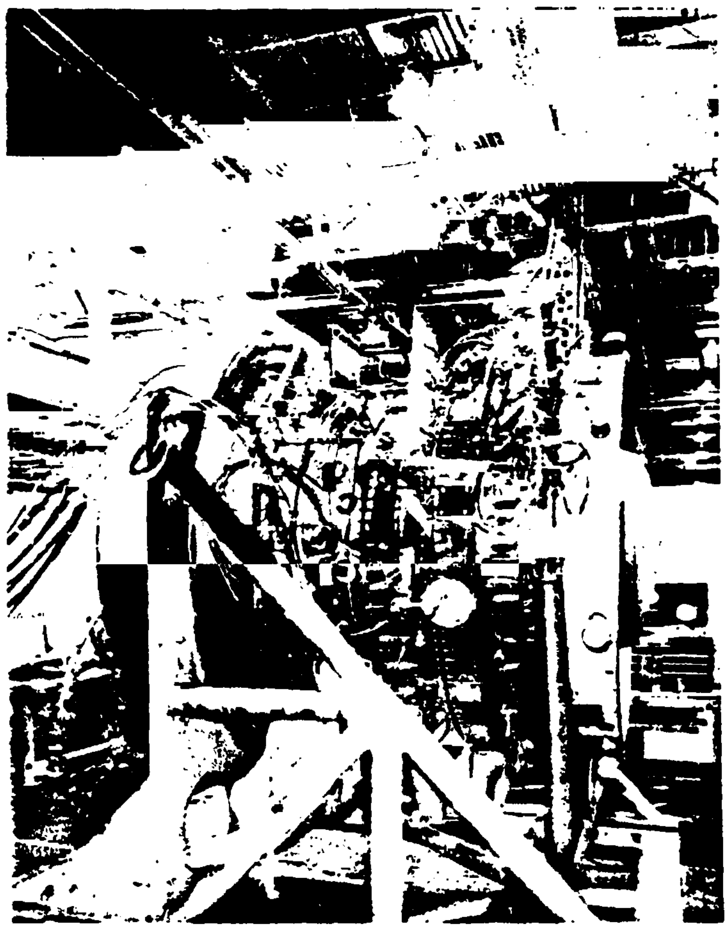

Figure 54 Assembled High-Pressure Turbine Rig

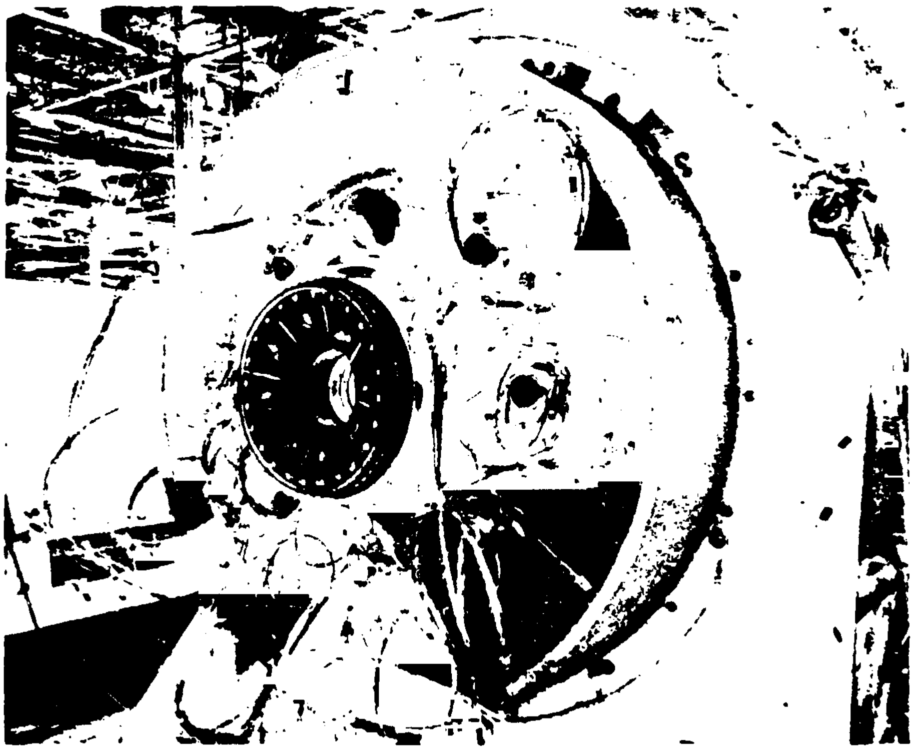

Figure 55 Assembled High-Pressure Turbine Rig

Rig Performance Test

The assembled rig was delivered to the Pratt & Whitney Andrew Willgoos Laboratory's X-203 dynamometer stand in mid-February 1982. A schematic of this test stand is shown in Figure 56. Each of the two dynamometers are capable of absorbing up to 10,000 horsepower. Only one dynamometer will be used for the Energy Efficient Engine high-pressure turbine rig. Installation of the rig and hookup of the instrumentation was completed in early-March 1982.

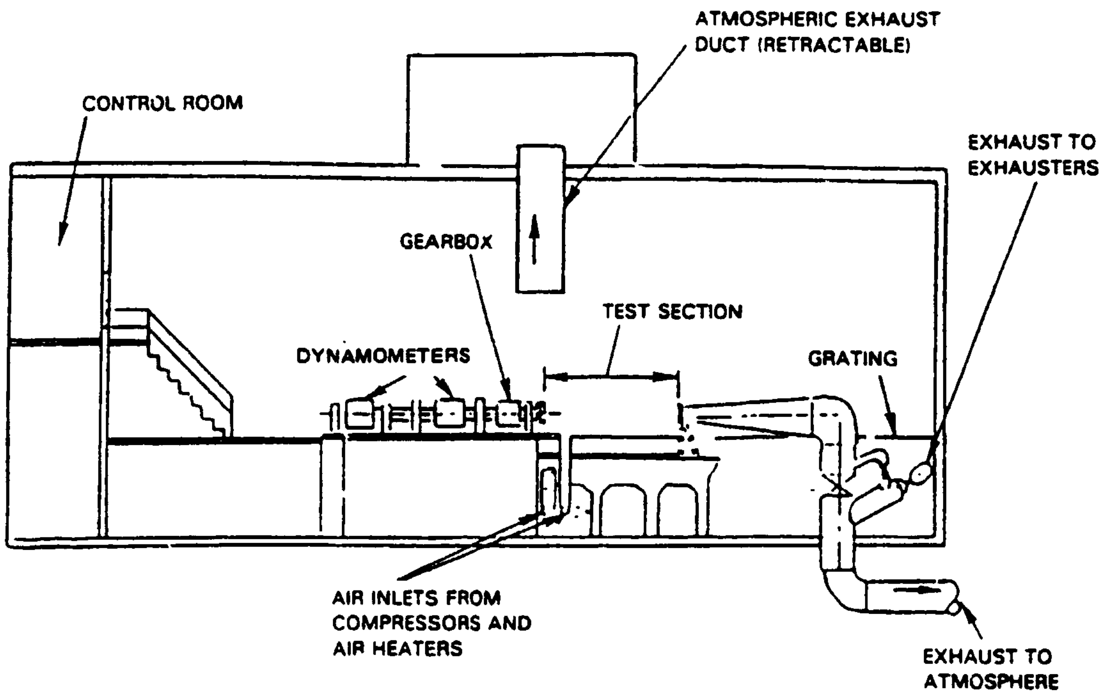

Figure 56 Schematic of Test Stand X-203

A rig supervisory control system is being used to aid in setting performance points, maintaining test parameters within preprogrammed limits, and monitoring rig and facility safety parameters for early detection of problems. The supervisory control is a digital computer control with appropriate input and output signal circuiting. Input signals are compared to preprogrammed levels stored within the supervisory control computer. The system output controls the feedback mechanism to trim the facility equipment, as required.

In addition to the rig supervisory control system, permanently installed stand monitoring systems are used to ensure both rig and facility safety.

The test matrix planned for the program is shown in Figure 57. Performance testing of the rotating configuration was completed late in the reporting period. Analysis of the resultant data is scheduled to begin in the next reporting period.

A second part of the test will involve removal of the rotor from the rig and air flowing of the nozzle vane assembly as a cascade. The cascade test could be eliminated depending on test results from the full-stage rig.

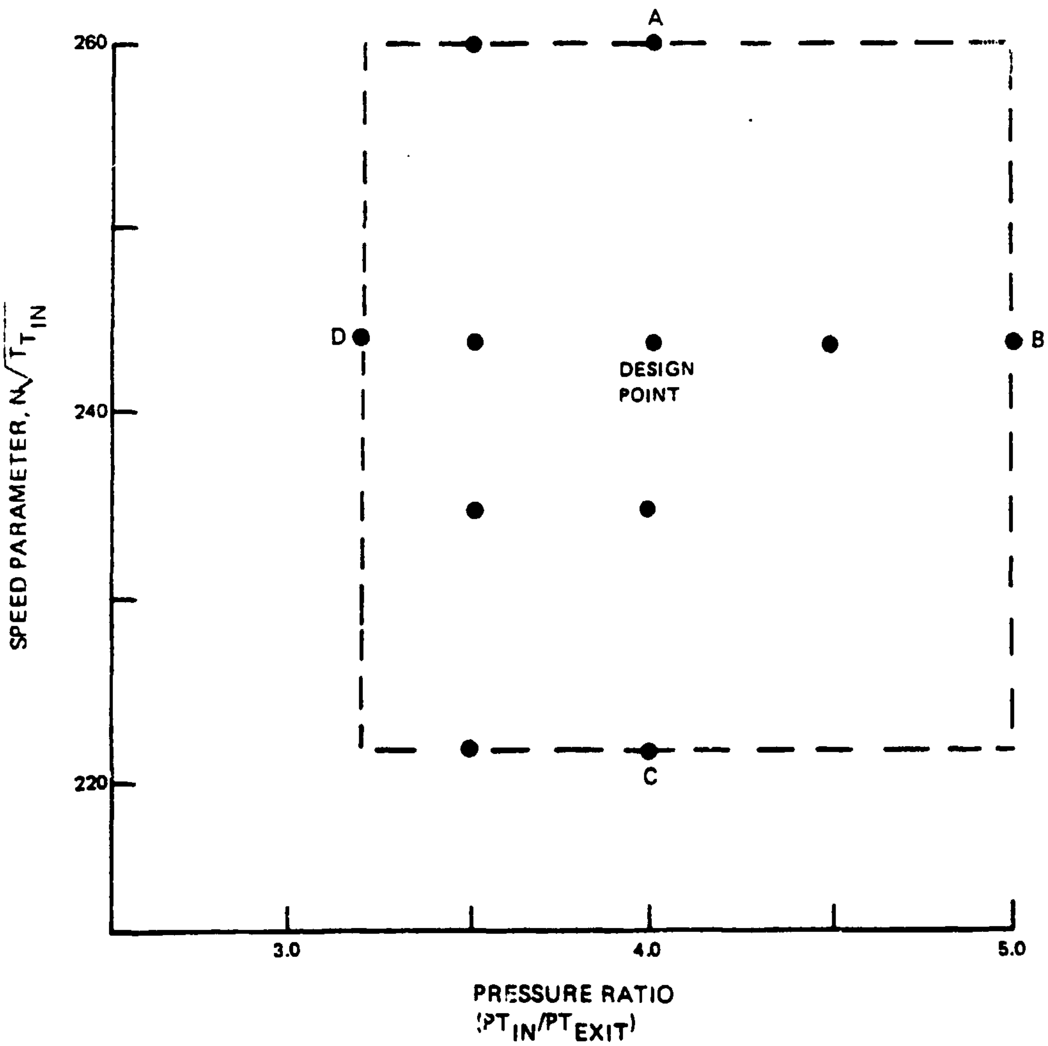

Figure 57 Test Envelope for Full Stage Turbine Test Program

3.2.7.4 Supporting Technology

3.2.7.4.1 Leakage Test Program

All technical work for this supporting technology program has been completed. Program results appear in NASA Report CR-165202.

3.2.7.4.2 Supersonic Cascade Test Program

All technical work for this supporting technology program has been completed. Program results appear in NASA Report CR-165567.

3.2.7.4.3 Cooling Model Test Program

All technical work for this supporting technology program has been completed. Program results appear in NASA report CR-165374.

3.2.7.4.4 Uncooled Rig Test Program

All technical work for this supporting technology program has been completed. Program results appear in NASA report CR-165149.

3.2.7.4.5 High-Pressure Turbine Fabrication Development Program

All technical work for this supporting technology program has been completed. Program results appear in NASA report CR-165400.

3.2.8 Low-Pressure Turbine

3.2.8.1 Overall Objective

Develop the technology required to design a highly efficient low-pressure turbine, and to incorporate this technology into design and fabrication to demonstrate the potential for achieving the Energy Efficient Engine flight propulsion system low-pressure turbine performance goals of 91.5 percent efficiency, 0.7 percent pressure loss in the transition duct, and 0.9 percent pressure loss in the exit guide vane. Design goals are disk life of 20,000 missions/30,000 hours, blade and vane life of 15,000 hours, hot strut life of 9,000 hours/15,000 missions and vane, blade, and transition duct coating life of 9,000 hours.

3.2.8.2 Component Program Overview

The overall task effort consists of a component effort and three supporting technology subtasks. The component effort comprises the analysis and design and fabrication of the low-pressure turbine component. The three supporting technology programs are (1) the Boundary Layer Test Program, (2) the Subsonic Cascade Test Program, and (3) the Transition Duct Test Program. The original program effort included a turbine exit guide vane supporting technology test program. This program was cancelled at the first work plan update in March 1979 because it was judged to be of minimal technical risk. Figure 58 shows the relationships between these activities and their relationship to Tasks 1 and 4. The work plan is shown in Figure 59.

3.2.8.3 Component Effort

3.2.8.3.1 Objective

Conduct the design, analysis, and hardware procurement activities necessary to develop a low-pressure turbine that meets the established goals.

3.2.8.3.2 Scope of Total Work Planned

The analysis and design effort consists of a preliminary analysis and design phase and a detailed analysis and design phase as shown in Figure 59. A six-month preliminary design activity is conducted to establish the aerodynamics of the low-pressure turbine flowpath and to determine the mechanical and structural feasibility of that configuration. This preliminary activity results in layout drawings and substantiating design data to be presented to NASA at a preliminary design review in September 1978.

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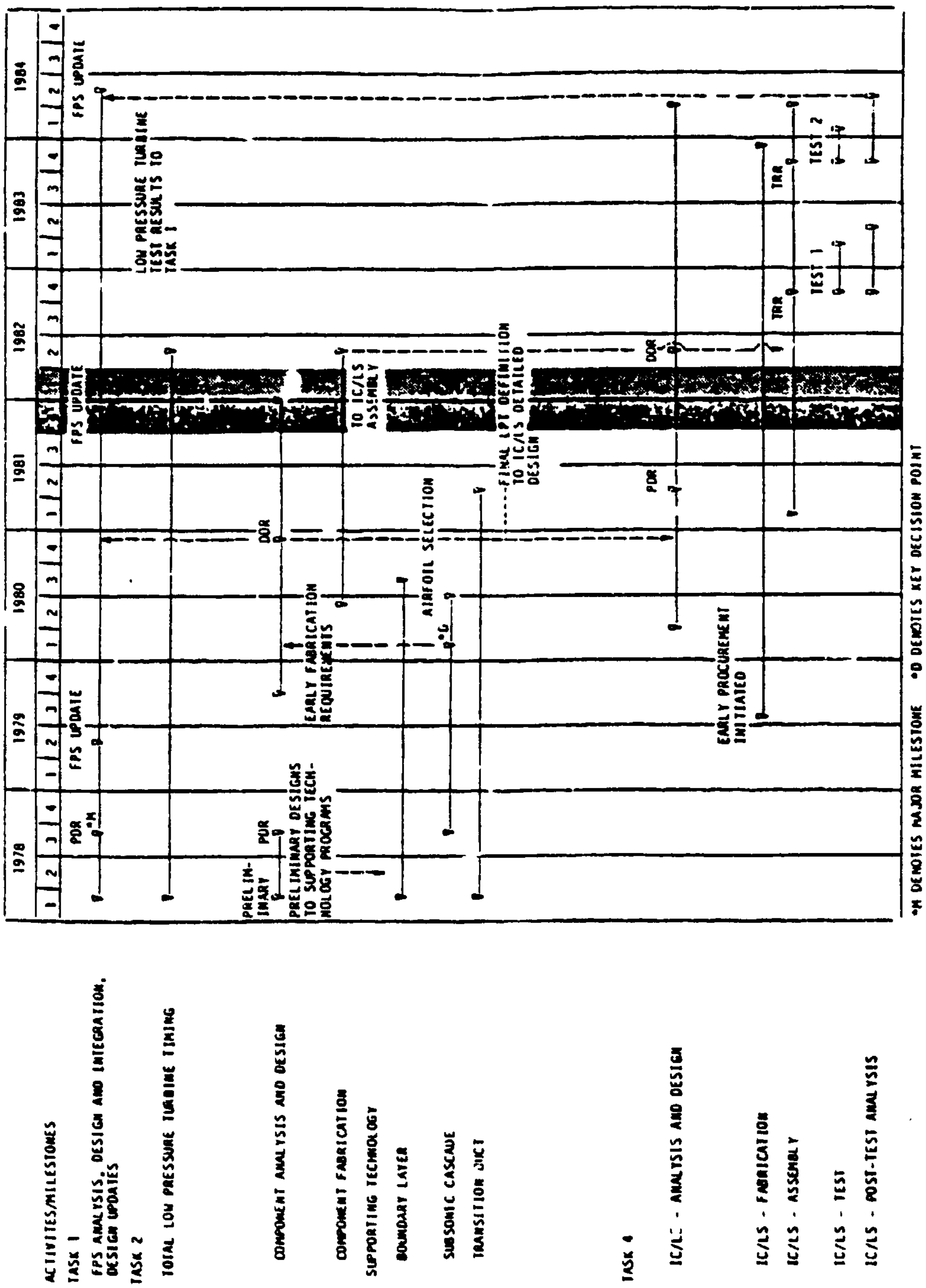

Figure 58 Low-Pressure Turbine Program Logic Diagram

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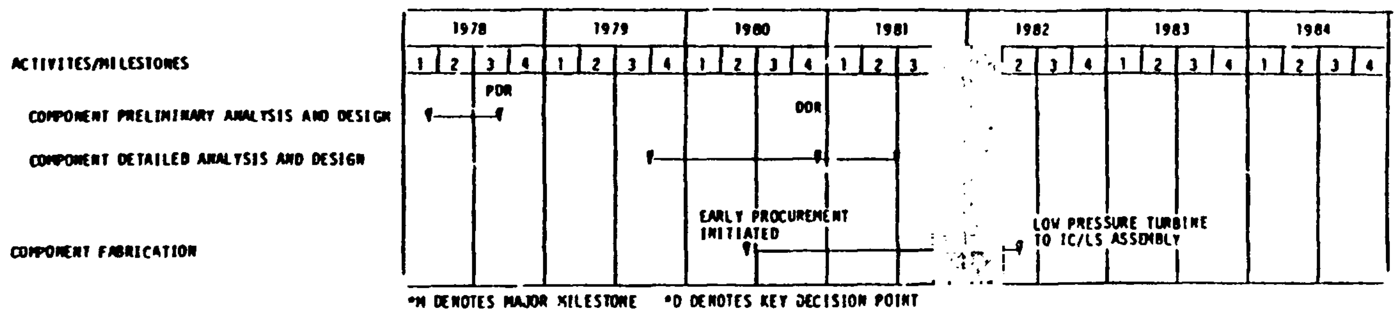

Figure 59 Low-Pressure Turbine Component Effort Work Plan Schedule

Approximately 12 months after the preliminary design review, a detailed design activity starts. Results available from the supporting technology programs are used to substantiate or improve the configurations established in the preliminary design. More sophisticated design and analytical procedures than those of the preliminary effort are used. The results of this effort are presented to NASA at a detailed design review in December 1980. Fabrication of the component parts is scheduled to start in the second quarter of 1980 and be completed in the second quarter of 1982.

Figure 59 indicates that all of the work associated with component preliminary and detailed analysis and design of the low-pressure turbine component was completed during previous reporting periods. The figure also shows that the component fabrication effort was continued during the current reporting period. A draft copy of the component test hardware detailed design report has been submitted to NASA for review and approval.

3.2.8.3.3 Technical Progress

3.2.8.3.3.1 Summary of Work Previously Completed

The low-pressure turbine design that evolved from the preliminary and detailed design activities is shown in Figure 60. The major mechanical features of this design are listed below.

- An 'A-frame' rotor hub to control deflections caused by maneuver loads.
- Disk rim spacers/knife edge seals separate from the rotor structure to shield the rotor from the hot gaspath air.
- Two-tooth blade attachments for all disks.
- Gaspath flow guides on the flowpath inner wall to reduce cavity ingestion and improve efficiency.
- Cooled disk rims.
- A double-wall outer case to accommodate the internal active clearance control system.
- Inner air-seal shrouds that are integral parts of the vanes.
- Internal active clearance control for the outer air seals to control tip clearance and maximize efficiency.

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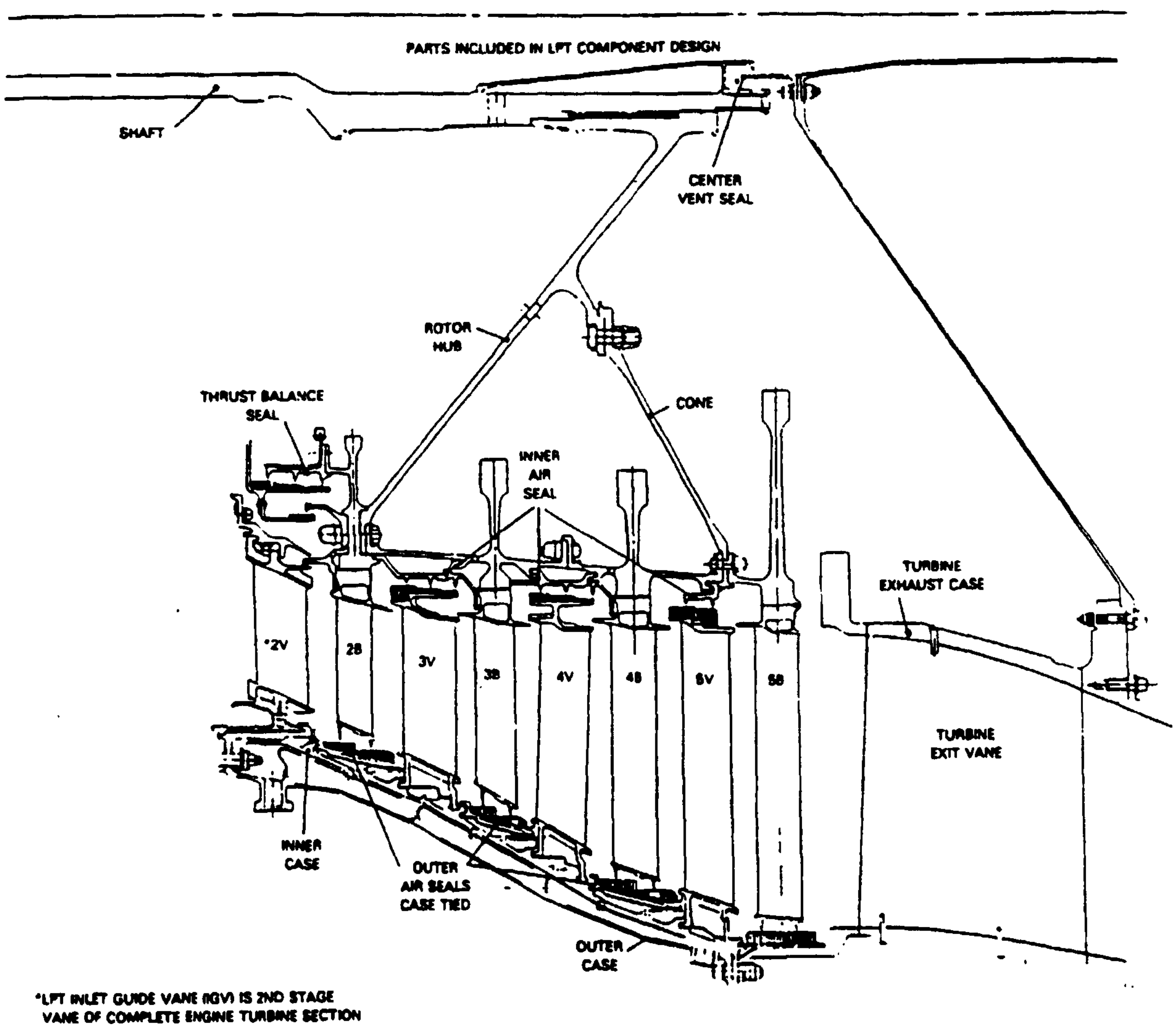

Figure 60 Low-Pressure Turbine Component

The general aerodynamic features of the design remained unchanged from those previously reported. These include (1) counter-rotation relative to the high-pressure turbine, (2) a low velocity ratio with a low ratio of through flow to wheel speed (Cx/U), (3) a relatively high pressure ratio per stage, (4) controlled vortexing, (5) low loss airfoil designs, and (6) minimized tip clearance through an internal active clearance control system. These general aerodynamic characteristics are listed in Table 3-XVI while the current performance parameters at significant engine operating conditions are shown in Table 3-XVII.

TABLE 3-XVI

INTEGRATED CORE/LOW SPOOL - LOW-PRESSURE TURBINE
GENERAL AERODYNAMICS
(AERODYNAMIC DESIGN POINT)

Stages	4
Total Number of Vanes	318
Total Number of Blades	438
Rotation	Counter
Speed (rpm)	3902
Inlet Total Pressure (psia)	46.3
Inlet Total Temperature (^oR)	2090
Inlet Corrected Flow (lbs/sec)	69.342
Exit Corrected Flow (lbs/sec)	323.17
Pressure Ratio	5.51
Specific Enthalpy (h) (Btu/sec)	12760
Mean Velocity Ratio	.464
Work Factor ($\Delta h/U^2$)	2.32
Average Flow Coefficient (Cx/u)	.79
Work Split	.23/.24/.26/.27
Mean Reaction	.45/.45/.45/.46
Goal Clearances (in)	.020
Goal Efficiency Split (%)	90/88.9/90.4/90.9
Goal Overall Efficiency (%)	91.5

TABLE 3-XVII

LOW-PRESSURE TURBINE CURRENT PERFORMANCE
PARAMETERS AT SIGNIFICANT ENGINE OPERATING CONDITIONS

	Engine Operating Conditions			
	Aero. Des. Point	Maximum Cruise	Maximum Climb	Takeoff
Inlet Flow Parameter (lb_m ^oR)(in^2/sec)(lb_f)	65.40	65.50	65.25	65.40
Rotor Inlet Temperature (^oF)	1540	1505	1675	1735
Pressure Ratio	5.72	5.66	5.81	5.09
Adiabatic Efficiency (%)	91.6	91.5	91.7	90.5
Enthalpy Change (Btu/lb)	175.6	171.5	189.6	181.2
Exhaust Case Pressure Loss (%)	0.90	0.87	0.95	0.69

The turbine exit guide vane is designed to provide the mixer with a low Mach number, zero swirl gas stream. A controlled diffusion airfoil design was used to (1) produce an attached boundary layer and (2) attain the desired gas exit angle. The exit guide vane airfoil contours and their predicted loading diagrams were reported in the Fifth Semiannual Status Report.

Figure 61 identifies the major design features of the turbine intermediate case assembly. The assembly comprises the high-pressure turbine outer case, high-pressure turbine blade tip seal, eleven structural struts that traverse the gaspath and are shielded by aerodynamic fairings, an inner ring torque box that forms an interface between the structural struts and the rear bearing support structure, and second stage turbine vane inner support, and front and rear secondary air seal lands. Engine mount and ground handling attachment lugs are located on the outer case between the pads where tiebolts and dowels secure the bearing support structure to the case. The strut and its associated support structure serve to maintain structural integrity of the bearing support frame in the event of turbine failure, minimize case ovalization caused by engine mount loads, and provide a route for oil service lines to the number 4-5 bearing compartment.

TURBINE INTERMEDIATE CASE
(HOT STRUT)

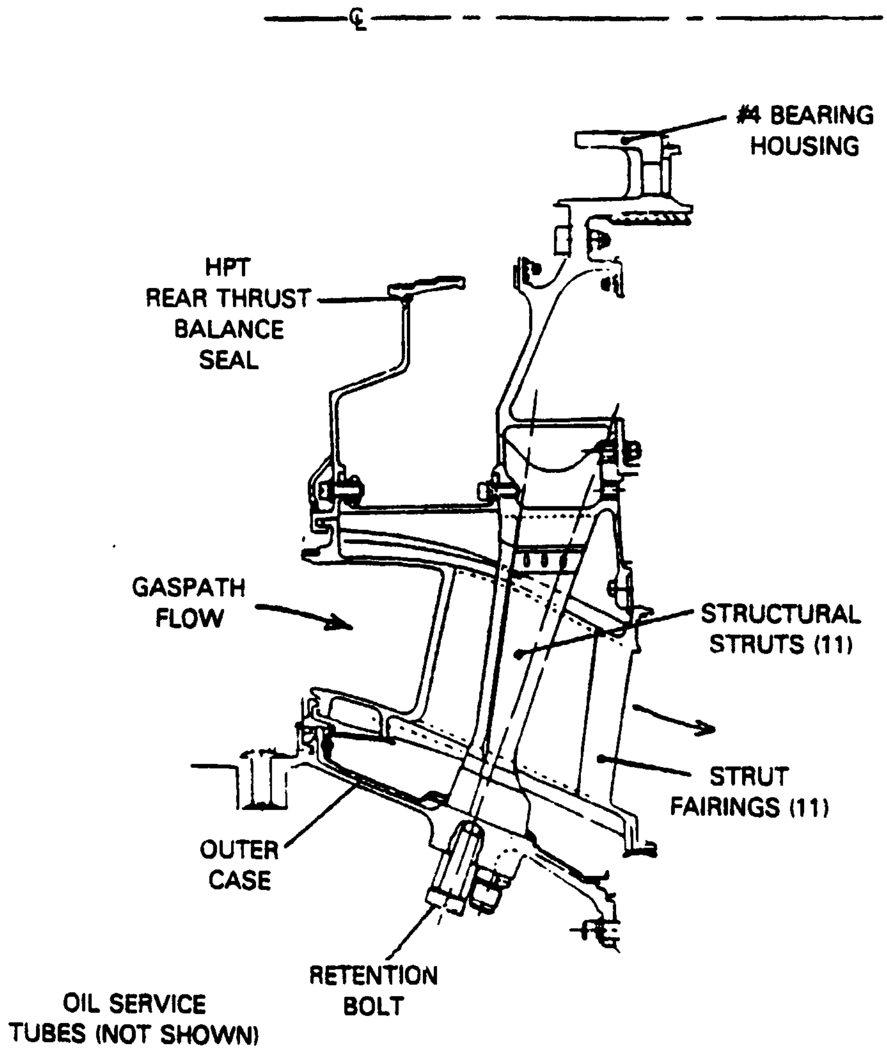

Figure 61 Major Design Features of the Turbine Intermediate Case

Fabrication efforts prior to the current reporting period focused on component hardware required for integrated core/low spool assembly. A summarization of these fabrication efforts is detailed below.

- o Three ring-forgings for the low-pressure turbine case were received and assembled by electron beam welding the details to form the rough shape case.

- o Initial rough machining of the turbine intermediate case forgings progressed to 60 percent completion and sample hot strut fairing castings were poured.

- o Low-pressure turbine blades and vanes were cast and accepted by Pratt & Whitney after completing a review of vendor inspection reports. Vendor casting of all blades and vanes is progressing according to schedule.

- o Fabrication of tooling required to hot isostatically press MERL 76 powder material for low-pressure turbine disks and inner air seals was completed.

- o Improper heat treatment caused cracks to develop on the front end of the low-pressure turbine shaft. A second shaft was forged and is currently undergoing initial rough machining prior to heat treat.

- o Forged materials for turbine exhaust case details were received and rough machining initiated. Turbine exhaust case vane castings were completed and sample parts inspected by Pratt & Whitney.

3.2.8.3.3.2 Current Technical Progress

Low-Pressure Turbine Component Fabrication

Disks and Inner Air Seals: Hot isostatic pressing of MERL 76 powder material for the first set of low-pressure turbine disk compactions and inner air seal compactions was completed. These parts are currently being heat treated and machined to a sonic inspection shape.

Shafts: The initially fabricated shaft was shortened to remove existing cracks that developed during the first heat treat operation. A revised operational procedure was defined and successfully used, as determined by hardness checks, to heat treat the shortened shaft. A second shaft was then forged, rough machined and is currently undergoing the same heat treat operation successfully applied to the shortened shaft. This second shaft is scheduled for use in the integrated core/low spool.

Blades: The initial sets of PWA 655 (Inconel 713C) and PWA 1447 (MAR-M-247) material castings for all stages of blades were delivered and accepted by Pratt & Whitney. With planning work and fabrication of the necessary machine tooling near completion, finish machining of the parts is scheduled to start in the next reporting period.

Vanes: PWA 655 (Inconel 713C) and PWA 1455 (Modified B-1900) material vane castings for stages 3 through 5 and PWA 1480 single crystal material vane castings for stage 2 were delivered and accepted by Pratt & Whitney. A sufficient number of these castings was received to yield a full set of finish machined low-pressure turbine vanes. With planning work and fabrication of the necessary machine tooling near completion, finish machining of the parts is scheduled to start in the next reporting period.

Turbine Intermediate Case

Machining of the turbine intermediate outer case assembly was completed. Fabrication of detail parts for the case struts and inner shroud ring was completed. These parts are now ready for welding.

Turbine intermediate case fairing castings were returned to the vendor for straightening to an acceptable shape to provide improved matching of the platforms at the trailing edge outer diameter. The vendor is currently inspecting the dimensions of these straightened castings as well as casting additional sample parts.

Turbine Exhaust Case

Machining of all turbine exhaust case detail parts was completed and these parts were made available to the welding vendor for assembly.

Number 4 and 5 Bearing Compartment

Fabrication of all minor parts for the number 4 and 5 bearing compartment was completed while fabrication of the number 5 bearing progressed to 85 percent completion.

Rotor Hub and Cone

Machining of the low-pressure turbine rotor hub and rear cone support progressed to 80 percent completion.

3.2.8.4 Supporting Technology

3.2.8.4.1 Boundary Layer Test Program

All technical work for this supporting technology program has been completed. Program results appear in NASA report CR-165338.

3.2.8.4.2 Subsonic Cascade Test Program

All technical work for this supporting technology program has been completed. Program results appear in NASA report CR-165592.

3.2.8.4.3 Transition Duct Test Program

3.2.8.4.3.1 Objective

Develop and experimentally verify the design of a short, low-loss, advanced technology transition duct for the flight propulsion system low-pressure turbine component design.

3.2.8.4.3.2 Scope of Total Work Planned

All technical work for this supporting technology program has been completed. Program results are summarized in the Seventh Semiannual Status Report. A draft copy of the transition duct technology report has been submitted to NASA for review and approval.

3.2.9 Exhaust Mixer System

3.2.9.1 Overall Objective

Design and develop exhaust mixer aerodynamics that will achieve the goal mixing efficiency of 85 percent, both for the flight propulsion system component and for the experimental integrated core/low spool.

3.2.9.2 Program Overview

The overall task effort consists of a component effort and a mixer model supporting technology subtask. The initial mixer component analysis and design effort is aimed at defining the mixer/tailpipe flowpath. Mixer deflections, stresses, and thermal loading are then estimated and a preliminary layout is defined. This preliminary layout is incorporated into the overall nacelle design, and the total system is evaluated through interface meetings between Pratt & Whitney and airframe subcontractors. The design resulting from this refinement process is fed into the mixer model test support technology program. Final refinements to the mixer design are completed in the Task 4 analysis and design work package. In addition, a test facsimile is fabricated and tested in Task 4.

All work associated with the preliminary analysis and design of the mixer component is complete. Results of this program effort are summarized in the Sixth Semiannual Status Report.

3.2.9.3 Supporting Technology

All technical work for the Mixer Model Supporting Technology program has been completed. Program results appear in NASA report CR-165592.

3.3 TASK 4 - INTEGRATED CORE/LOW SPOOL DESIGN, FABRICATION AND TEST

3.3.1 Task Objective

Design, fabricate and test two builds of the Energy Efficient Engine integrated core/low spool. The purpose of the first build is to evaluate component and subsystem performance as well as obtain initial indications of structural integrity. Testing of the second build will determine overall system performance, emissions and noise. The following goals have been established for these tests:

Thrust Specific Fuel Consumption
(lb/hr/lb corrected to standard day) -- 0.342

Emissions -- 1981 Environmental Protection Agency Rule

Noise (EPNdB)

Takeoff	98.9
Approach	98.2

3.3.2 Scope of Total Work Planned

The following paragraphs describe the work planned for each build of the integrated core/low spool. The interrelationship among task activities is shown in Figure 62.

First Build Integrated Core/Low Spool -- The work plan structured to achieve the first build objectives is shown in Figure 63. The analysis and design effort associated with the first build involves those items required to test the components of Task 2 as an integrated system, as well as the necessary instrumentation and test hardware. This effort includes provisions for engine accessories, plumbing, active clearance control, bleed, and fuel and lubrication systems. Any necessary instrumentation not designed in Task 2 is designed in Task 4. This includes a high-pressure rotor telemetry package and modifications to the number 3 bearing area to accommodate the package. The impact of these modifications on engine critical speed is assessed.

A bellmouth inlet, bifurcated fan exhaust ducts, tailpipes, mount hardware, and related equipment designs are provided. At an appropriate point in the design cycle, a preliminary design review is conducted. It addresses the design status as well as integrated core/low spool test and instrumentation plans and schedules. Following approval, the design is completed, at which time a detailed design review is conducted to address the final build 1 design as well as a refinement of plans and schedules.

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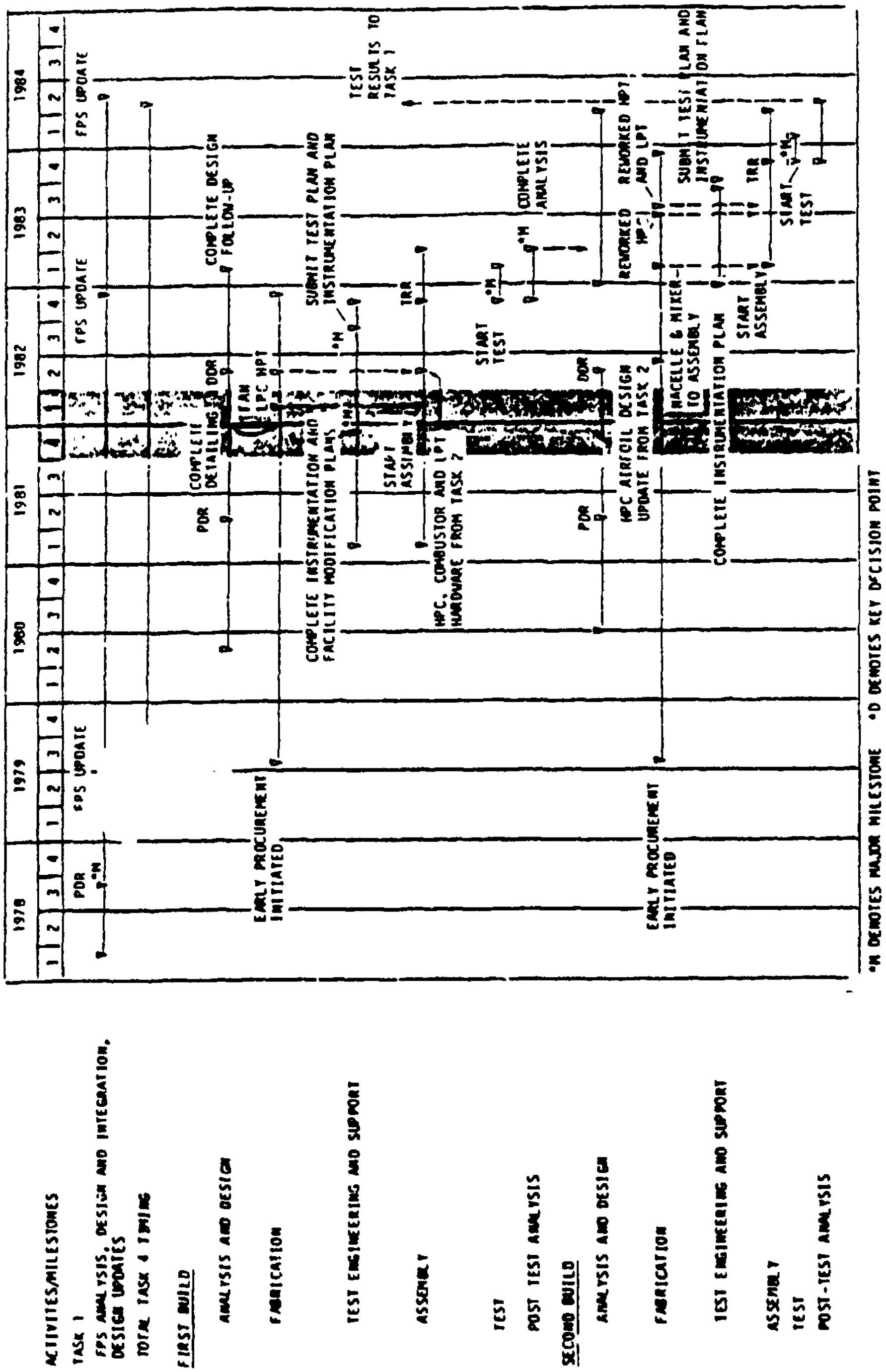

Figure 62 Integrated Core/Low Spool Design, Fabrication and Test Logic Diagram

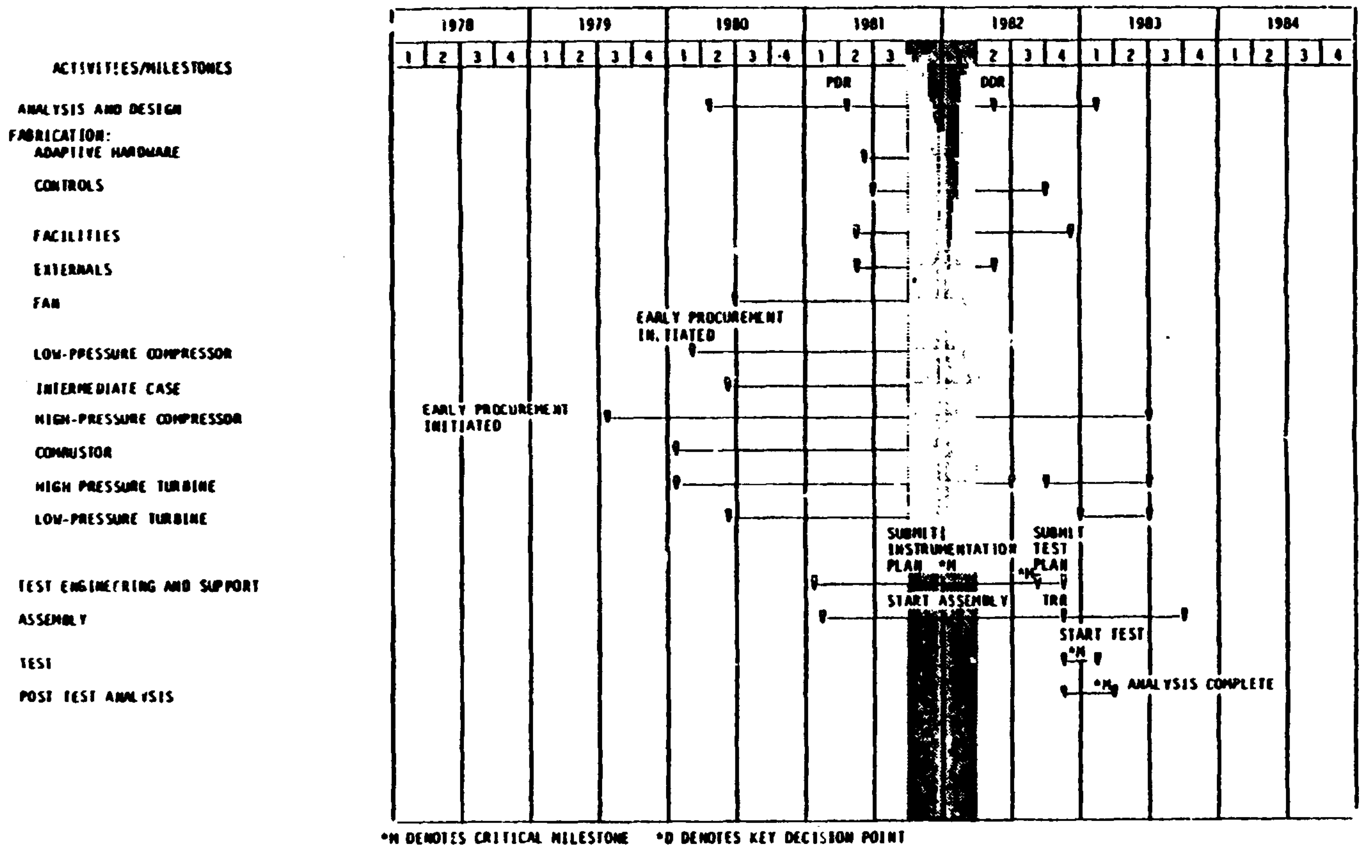

Figure 63 Integrated Core/Low Spool (First Build) Work Plan Schedule

Hardware fabrication is initiated upon receipt of NASA approval of the detail design. Components are those initially designed in Task 2, with the exception of the high-pressure compressor aerodynamics and final combustor hole patterns. High-pressure compressor blades are modified to reflect changes defined by results of compressor build 2 testing. Similarly, high-pressure compressor stator details are assembled with stagger angles defined by build 2 test results. A final combustor hole pattern evolves through combustor component rig testing. Fabrication of hardware is performed by Pratt & Whitney and approved vendors.

The integrated core/low spool is assembled for the first build with the shrouded fan rotor and associated hardware in an nonmixed exhaust configuration using bifurcated fan ducts. Major structural cases are tested to verify axial and radial spring rates. Leakage and flow checks are conducted on turbine subassemblies, main shaft seals, bearing compartments, and oil jets. In addition to major engine station gaspath instrumentation, extensive strain gage, static pressure and temperature instrumentation is installed at assembly. A test readiness review is conducted at the conclusion of assembly.

The first build of the integrated core/low spool is tested in an indoor sea level test stand. Initial testing is directed toward obtaining both high and low spool strain gage data to provide an early indication of mechanical behavior. This is followed by performance testing. During performance testing, the active clearance control system is calibrated.

Performance data are taken over a range of rotor tip clearances, controlled by manual operation of the active clearance control system. Tip clearances are determined by use of optical proximity probes. Internal instrumentation is monitored throughout the test to ensure that internal pressures and temperatures are in agreement with predicted values. In addition, boroscope inspection is periodically conducted to visually check the condition of the airfoils and combustor liner.

Test data are reduced and plotted on gas generator curves to assess the overall gas generator performance. Plots compare test data to predicted characteristics to determine component and spool match. More detailed evaluation of the component data provides assessment of individual component performance. Tip clearance measurements and active clearance control system temperature, pressure, and flow measurements are used to compare system operation against pretest predictions.

Following testing, the integrated core/low spool is disassembled to the extent necessary to incorporate desired changes and to inspect certain critical parts prior to assembly for build 2.

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Second Build Integrated Core/Low Spool -- The work plan structured to achieve the second build objectives is shown in Figure 64. The analysis and design of the second build of the integrated core/low spool consists of the design of a boilerplate nacelle and an exhaust mixer. In addition, it may include a re-stagger of high and low-pressure turbine blades and vanes if build 1 testing results define such a requirement. Because the analysis and design activity of both builds is concurrent, combined build 1 and 2 preliminary and detailed design reviews are anticipated.

Fabrication of hardware peculiar to build 2 commences after approval of the detailed design. This includes hardware for high-pressure compressor blade rework, assembly of stators to incorporate changes resulting from high-pressure compressor build 3 rig testing, highand low-pressure turbine blade and vane attachment machining, and fabrication of an acoustically treated boilerplate nacelle and mixer. Appropriate modifications to build 1 hardware are also performed.

The second build of the integrated core/low spool is assembled with the shrouded fan, acoustically-treated boilerplate nacelle, and exhaust mixer. Instrumentation is less extensive than that used in build 1. A test readiness review is conducted at the conclusion of assembly.

The second build of the integrated core/low spool is mounted in an outdoor sea level test stand. Following baseline performance testing, all instrumentation not required for monitoring engine safety and basic gas generator performance is removed. Subsequent performance testing includes thrust specific fuel consumption demonstration, noise and emissions. Following testing, the integrated core/low spool is disassembled for an inspection of parts.

Test data are reduced and analyzed. Performance results are compared to design assumptions and pre-test performance predictions. Overall performance (thrust specific fuel consumption), emissions and noise levels are compared to integrated core/low spool predicted levels.

3.3.3 Technical Progress

3.3.3.1 Integrated Core/Low Spool Analysis and Design

3.3.3.1.1 Summary of Work Previously Completed

For reasons of expediency and economics, available hardware from various Pratt & Whitney engine models has been adopted for the integrated core/low spool as the design evolved.

Initial analysis and design efforts were directed toward adaption of a gearbox and angled drive configuration previously used in another experimental program. This configuration, which is mounted on the top of the fan case, has been successfully run and was selected because of its availability and low cost.

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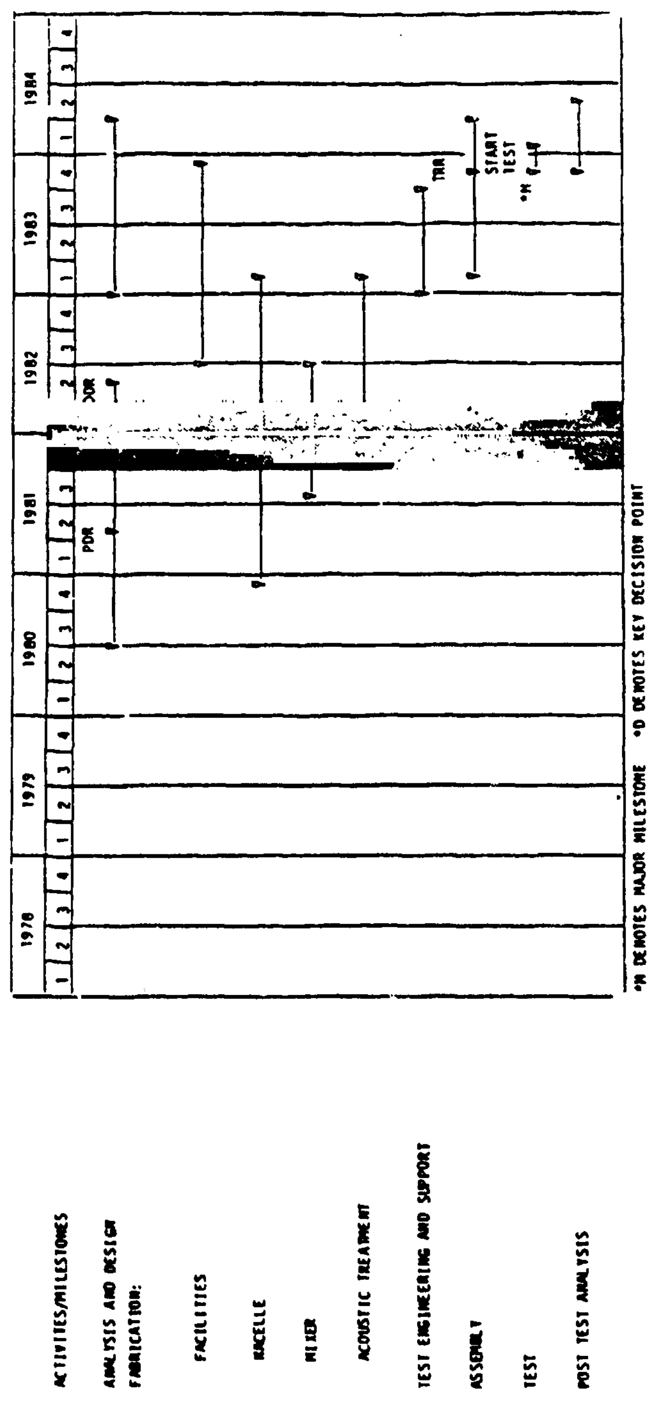

Figure 64 Integrated Core/Low Spool (Second Build) Work Plan Schedule

A full-size wooden mock-up of the integrated core/low spool was constructed to facilitate the design of exterior plumbing. A schematic of the plumbing required for the active clearance control system is shown in Figure 65.

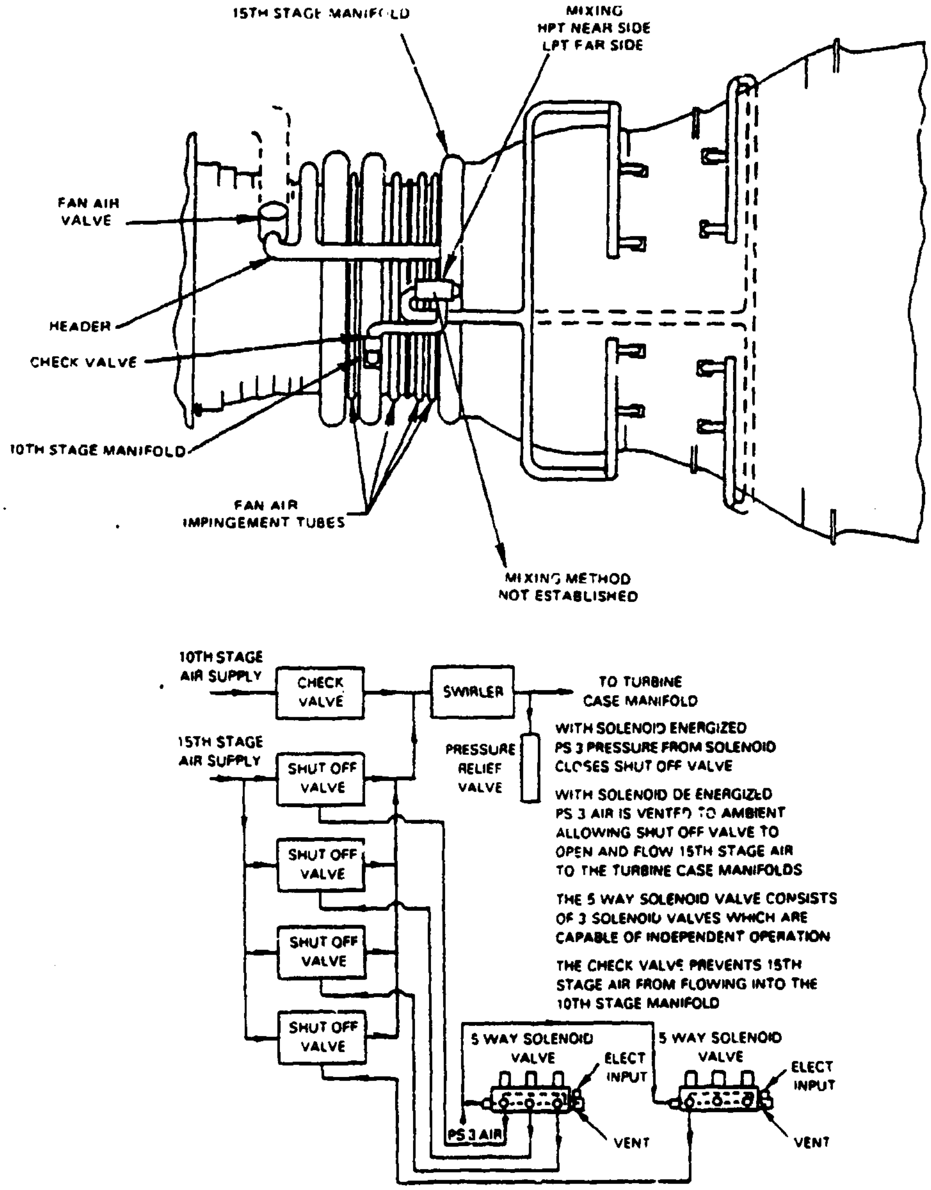

Figure 65 Integrated Core/Low Spool Active Clearance Control System

Each high- and low-pressure turbine mixing system incorporates four shutoff valves with metering orifices to regulate the flow of fifteenth stage air. Air from the fifteenth stage is mixed with tenth stage air before being ducted to the turbine case manifold. A temperature test was conducted on a currently available shutoff valve to evaluate its operational capability at 1100°F. The valve was cycled so that it was open for three minutes and closed for thirty seconds for a total of 780 on/off cycles. Results indicated that the valve is satisfactory for use in the integrated core/low spool active clearance control system at temperatures of 1100°F for up to 40 hours.

Layouts of the gearbox, oil tank, fuel oil cooler, fuel flow body, fuel flow divider, starter, ignition, tenth-stage start bleeds, and a portion of the high-pressure turbine active clearance control plumbing were completed. There were no significant changes to the lubrication system.

The fuel system selected for the integrated core/low spool and features a dual channel, full-authority electronic control mounted on the fan case. The electronic control is a modified version of a design used on existing engines. Commands from the control establish the primary and secondary fuel flow splits in the flow divider or split valve. During shutdown, the pilot zone fuel is dumped by a pressurizing and dump valve. A solenoid performs the same function for the main zone. The main zone fuel downstream of the manifold passes through 24 check valves (1 for each pair of nozzles). These valves are designed to permit the whole main manifold to fill before the fuel starts to flow out of the nozzles.

Preliminary control logic for the integrated core/low spool test has been designed and verified. Total fuel flow is scheduled as a function of rate limited power lever angle or as a function of engine pressure ratio versus power lever angle. A number of topping loops is incorporated in the control for engine structural protection.

A critical speed/forced response analysis was completed for the integrated core/low spool. It was found that a damper was required at the number 5 bearing to reduce the shaft strain energy and desensitize the mode to unbalance. With the introduction of this oil damper, no high strain energy low rotor modes are predicted in the running range. The fan and low-pressure turbine modes have low strain energy and occur below minimum cruise speed. Acceptable speed margin is expected for the high energy shaft mode which occurs well above the maximum low-pressure compressor rotor speed.

To accommodate the large amount of instrumentation required for testing, a bifurcated duct configuration was evolved. This facilitates access to the core and decreases losses that normally result from crossing the fan stream with instrumentation. A decision was made to modify the design of an existing JT9D bifurcated duct configuration for adaptation to the integrated core/low spool. A re-operation layout of the revised JT9D bifurcated duct was completed. In conjunction with this arrangement, a one-piece fiberglass bellmouth/ inlet case design is nearing completion. This unit will be also used for build 2, which features the full nacelle configuration.

Prior to this current reporting period, the Integrated Core/Low Spool Preliminary Design Review was conducted at the NASA-Lewis Research Center and subsequently approved in May 1981. Also, instrumentation plans as well as an engine instrumentation map (as shown in the Seventh Semiannual Status Report) for the integrated core/low spool were presented to NASA. For the testing builds 1 and 2 of the integrated core/low spool, test stands X-18 and C-11 have been selected. A design effort was initiated to modify stand X-18, as required, for the first test.

A full-size wooden mock-up of the integrated core/low spool has been used in the design of external of engine accessory hardware and associated plumbing to ensure no overlapping of hardware envelopes. Using this engine mock-up, mounting of all accessory hardware was completed. After completing simulated mock-up routing of the fuel and lubrication system to ensure proper clearances for all externals, installation of final configured tubing was initiated.

3.3.3.1.2 Current Technical Progress

In this period, design efforts associated with most of the instrumentation required for builds 1 and 2 of the integrated core/low spool engine (as shown in Figure 91, Seventh Semiannual Status Report) were completed. Only the design of static pressure instrumentation in the area of the high-pressure turbine and total temperature instrumentation at the leading edge of the turbine intermediate case fairings remains to be completed.

The design of ducting required for builds 1 and 2 also continued. The bifurcated duct configuration, which was completed at the end of the last reporting period, required an adapter duct. The adapter consists of two coannular duct halves used to adapt the flow lines of the compressor intermediate case to the flow lines of an existing set of bifurcated ducts. Upper and lower fairings are required between the intermediate case struts and the bifurcation walls of the duct halves. The adapter walls employ three equally-spaced groups of wall stiffening flanges. A minor modification to the forward contour of the existing bifurcated duct is required. The number of bolts fastening the bifurcated duct adapters to the intermediate case is sufficient to withstand shear and blowoff loads. However, the bending moment of the duct system must be prevented from acting on the intermediate case flanges by use of either a constant support hangar or counterweight system. The particular method of support has not been defined to date.

Early definition of the integrated core/low spool configuration included a fiberglass bellmouth and an acoustically-treated inlet duct and forward fan case, with the latter containing the required inlet test instrumentation. However, because of a potential reduction in fan blade tip clearance due to excessive overhung weight, this configuration was replaced by a lighter weight one-piece fiberglass inlet/bellmouth duct and an aluminum instrumentation ring. At present, this configuration does not incorporate noise treatment since a decision on the amount of treatment required is still pending.

Preliminary design effort for the nacelle D-ducts was initiated during the report period. Design coordination efforts with the vendor indicate fiberglass is a feasible and practical material for use in fabrication of nacelle D-ducts.

Except for the alternate fuel system, the mounting of plumbing and external accessories on the wooden mock-up of the integrated core/low spool engine has been completed.

The planning of test stand modifications as well as the design of the transport stand and engine overhead mounts were initiated. It appears that reoperation of an available transport stand will be possible, and the current design effort is moving in that direction.

The mixer and tailplug designs for integrated core/low spool build 1 and build 2 were completed. The mixer design, shown in Figure 66, is based on the optimum mixer definition that evolved from Phase II of the Mixer Model Supporting Technology program conducted under Task 2 of the contract. This design features 18 lobes with hoods (sheet metal fairings) attached to the upstream portion of the lobes. The function of these hoods is to improve the characteristics of the flow entering the lobe area and provide the added structural support required to make the lobes self-supporting. This latter enhancement eliminates the requirement for tie-rods between the mixer lobes and the exhaust plug, which could adversely affect the flow characteristics.

Inner and outer shells, ribs, and structural rings provide the mixer component with the necessary structural integrity. The outer shell consists of 36 stampings of Inconel 625 sheet metal, 0.063 inch thick and welded together to form the 18-lobed mixer fan stream flowpath. The 18 lobe weldment is then joined by rivets, and screwed to structural rings and ribs (one rib per lobe) to form the primary structure for the mixer. The triangular region bounded by the structural rings, rib, and hood is effectively a hollow box beam. The beams, from which the downstream mixer lobes are cantilevered, are supported by two structural rings. The rear ring provides a single plane attachment for the entire mixer assembly to the rear outer flange of the turbine exhaust case.

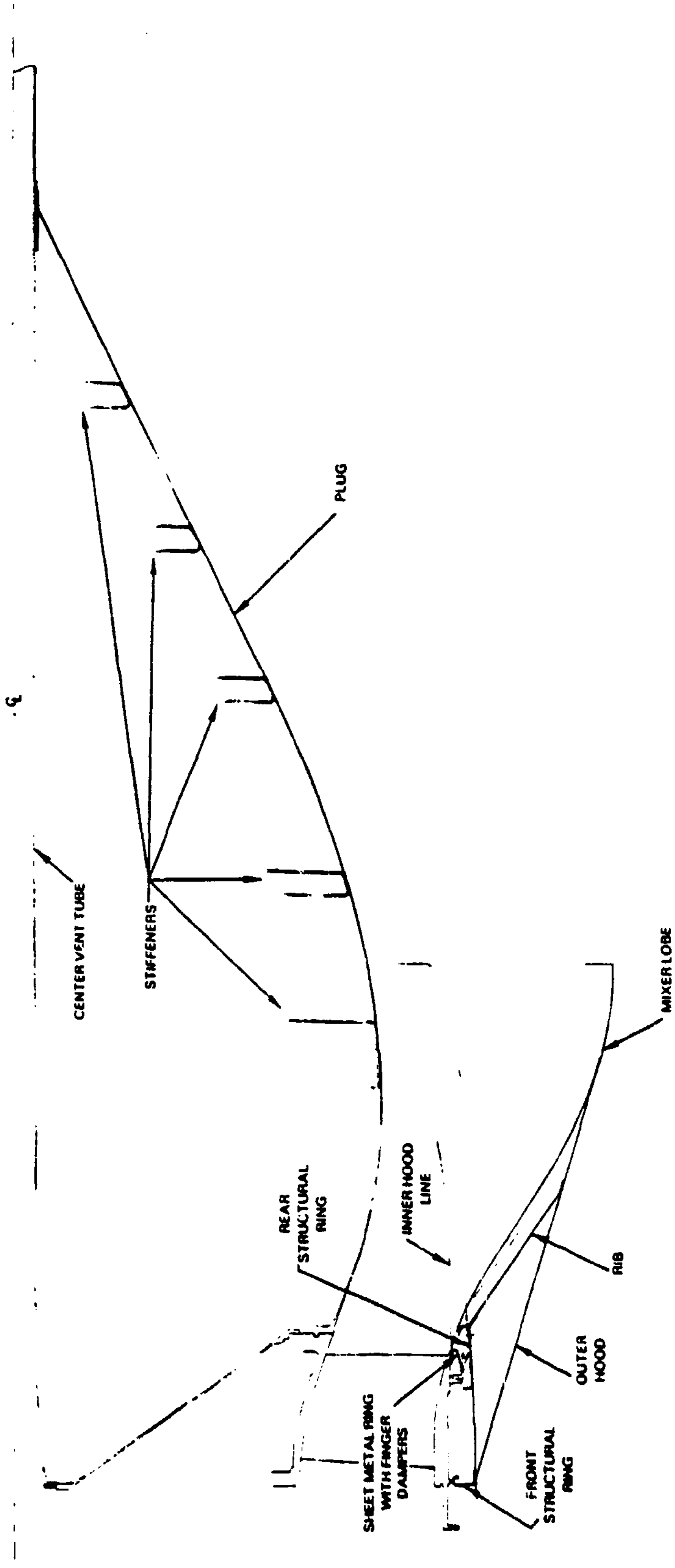

Figure 66 Integrated Core/Low Spool Exhaust Mixer and Plug Design

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The 18-lobed inner shell is fabricated from two stampings of Inconel 625 sheet metal, 0.043 inch thick. The two inner shell configurations, nested within the outer shell, are then joined together by blind rivets along the axial overlap joint shown in Figure 67. These inner shells are joined to the outer shell by rivets only in a plane at Section C-C. This allows thermal freedom between the inner shell which is exposed to hot engine gases and the outer shell, which is exposed to the relatively cold fan stream. The forward edge of the inner shell is riveted to a sheet metal ring which provides stability and damping action for the inner shell through use of finger contacts with the rear support ring.

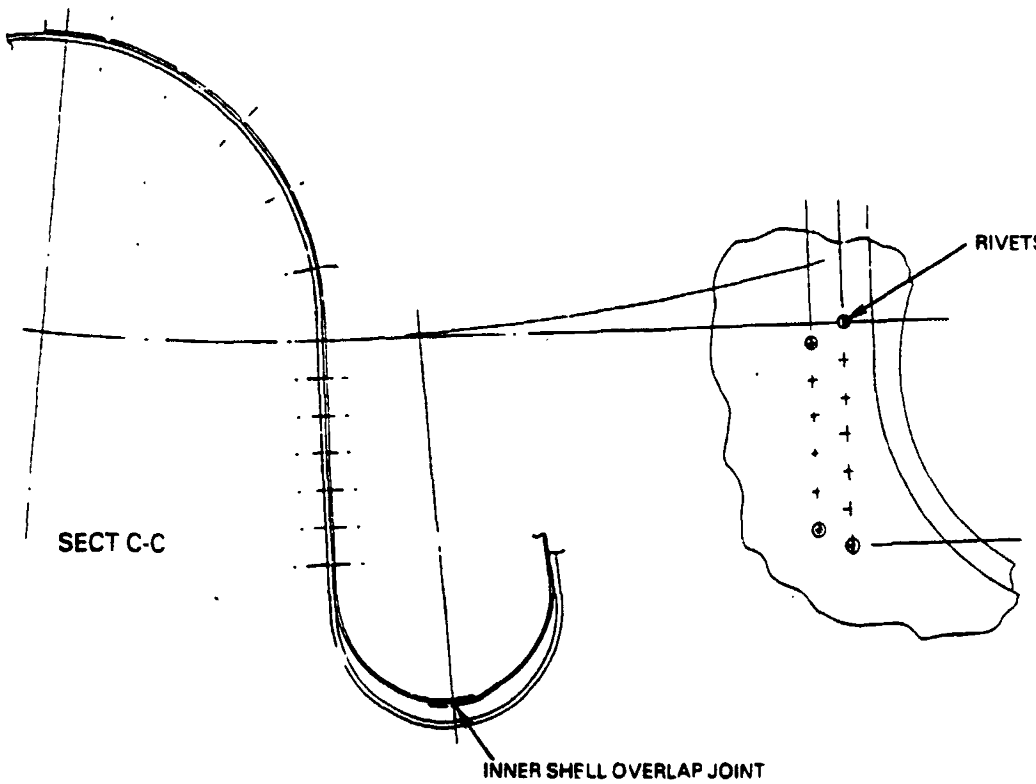

Figure 67 Mixer Inner and Outer Shell Joining Technique

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The exhaust plug, shown in Figure 66, is a structurally simplified version of the flight propulsion system design. It is fabricated from AMS 5510 material sheet, 0.050 inch thick. Buckling resistance is provided by simple ring stiffeners. The stiffener in the plane of the mixer lobe is wider than the others in order to provide an attachment point for a lobe damper, should that become necessary. Assembly is accomplished by sliding the plug over the engine center vent tube and bolting it to the turbine exhaust case inner rear flange.

A specification document, which defines the software requirements for the electronic control, has been completed. A decision was made to continue with the alternate fuel system, which incorporates a centrifugal pump and dual metering valve. A decision to use the original system, in the event of any unforeseen difficulties, can be made in the second quarter with no adverse effect on the integrated core/low spool (build 1) test date.

3.3.3.2 Integrated Core/Low Spool - Fabrication

3.3.3.2.1 Summary of Work Previously Completed

Approval for early procurement of raw material was received from NASA, and fabrication efforts were initiated for much of the external hardware, fuel system control, facilities, and adaptive hardware required by the work plan. In addition, NASA approval was granted to proceed with fabrication of various pressure and temperature instrumentation rakes along with a station 5.0 air angularity probe located aft of the low-pressure turbine exit guide vane.

A summary of the fabrication and procurement efforts of major external and internal hardware items for the integrated core/low spool engine is presented in the following paragraphs.

Integrated Core/Low Spool Fan -- Final machining of the shrouded fan blade forgings was initiated. Also, machining of the fan containment case steel forging detail was completed and the case was delivered to finish stores for later use in the assembly effort.

Machining operations were started on the aluminum nose cone and cap as well as on the fan hub forging. Fabrication of the fan blade retaining ring was completed.

Integrated Core/Low Spool Low-Pressure Compressor -- Purchase orders for all low-pressure compressor blades and vanes were placed. Raw material was received, and fabrication of blade and vane masters was initiated. In addition, steel forging material for all of the disks was received at Pratt & Whitney along with the aluminum vane and vane case forgings.

Fabrication of the stubshaft bearing compartment de-oiler was completed, while fabrication of the number 1-2 bearing support, the number 1 bearing housing, and number 2 bearing assembly continues. The inlet guide vane inner shroud details were completed, and machining of the vane case assembly was initiated.

Integrated Core/Low Spool Compressor Intermediate Case -- NASA approved using titanium struts fabricated by a diffusion bonding/superplastic forming process. Fabrication process planning was subsequently initiated as well as the design and preparation of tooling for (1) fabricating the intermediate case and towershaft drive gears, (2) refurbishing forming dies, and (3) fabricating inspection gauges.

The planning effort was completed. Indications are that the completion of the case assembly will slip. Although this slippage will not prevent meeting the first test date scheduled for the integrated core/low spool, the compressor intermediate case structural test will have to be rescheduled to follow this test.

Tooling design and the fabrication of tooling for the intermediate case continued. Since a new precamber die was required for the nominal structural strut, several die and bonding stop-off pattern modifications were made, and the new precamber die was fabricated. In addition, fabrication of inspection tooling for the structural struts and fixturing for welding the solid leading and trailing edge stiffeners to the struts were completed. Die fabrication for the 15-degree uncamber strut was initiated and verification of tapes, used in the manufacture of strut tooling, was completed. Die patterns and the master were made and dimensional verification of the master's accuracy was completed. In addition, design of the assembly fixture tooling was completed.

Nominal camber structural strut fabrication continued. As a trial, one strut was fabricated, successfully bonded and superplastically formed. A dimensional check in the inspection gauge, however, uncovered some gauge errors which were subsequently corrected. Strut external dimensions were in conformance to the design specifications, but internal checking disclosed an inadequate bond in the vicinity of the leading edge. The problem will be resolved by relocating strut packs in the die.

Preparation of titanium sheet material was completed for all nominal camber struts, and bonding of the strut packs was accomplished. One strut pack was successfully formed in the precamber die to verify elimination of previous dimensional deviations by relocation of the pack in the die.

All raw materials and many vendor-supplied hardware items were received. Included were such major parts as solid leading and trailing edge stiffeners for the structural struts, the 'eagle beak' shaped section of the pylon strut, and the pylon strut inner body. Fabrication of other case parts was initiated. All flanges were completed, except for the rear inner fan duct. Also, the rear segments for the outer core flowpath ring were finished.

Shop process planning and tooling designs were completed for the towershaft drive gears. Forgings were received for both the driven and driving gears, and rough machining of the six gear sets was completed. Fabrication of the inner and outer towershaft progressed, and the inner center towershaft coupling was received from the vendor. Inspection showed that this coupling did not conform to specifications. Since the part is not repairable, a new coupling will be manufactured.

Integrated Core/Low Spool High-Pressure Compressor -- Fabrication of the inlet guide vanes along with the sixth, eleventh, and twelfth stage vanes was completed. Fabrication of the remaining stage vanes and all compressor blades was continued.

The individual disks that comprise the high-pressure compressor drum rotor have been fully machined to the pre-weld configuration. The tandum disk raw material was forged to correct a material envelope problem. The part has been machined to a suitable shape for inspection and heat treatment.

The titanium forging for the variable vane case was received, and fabrication has been initiated. The titanuim forging for the compressor bleed case was also received. Semifinishing operations for the titanium front split case were completed and the part is ready for flange weldment and finish machining.

Integrated Core/Low Spool Combustor -- A spare diffuser case was successfully cast and delivered to Pratt & Whitney. Fabrication of the inner combustor case was also initiated.

Integrated Core/Low Spool High-Pressure Turbine:

Vanes -- To maintain the established assembly and test schedule for the first build integrated core/low spool, a decision was made to cast a set of vanes using PWA 1422 directionally-solidified alloy instead of PWA 1480 single crystal alloy. This decision was based on favorable experience in casting PWA 1422 vanes for use in the component rig. Meanwhile, efforts continue to successfully cast PWA 1480 vanes with the intention of providing a minimum of three vanes for the first integrated core/low spool build and a full set of vanes for the second build.

Blades -- Casting of a set of blades for the integrated core/low spool has progressed satisfactorily. These blades incorporate five additional cooling air pedestals in the trailing edge core cavity based on results from blade water flow cooling model tests.

Disk -- The fabrication effort directed toward the turbine disk has progressed through hot isostatic pressing, rough lathe turning, heat treating, and surface cleaning.

Integrated Core/Low Spool Low-Pressure Turbine -- The only effort presently included in this task is fabrication of the number 5 bearing, which is progressing on schedule.

3.3.3.2.2 Current Technical Progress

Integrated Core/Low Spool External and Adaptive Hardware -- Fabrication of the engine external hardware is well underway. This includes gearbox modifications, active clearance control plumbing and valving, oil tank engine mounts, bifurcated ducts and adapter, and bracketing.

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Adaptive hardware not previously ordered but required to support the first integrated core/low spool engine test was placed on order. Fabrication of the high-pressure spool telemetry carrier and support was completed along with the instrumentation probes listed below. Meanwhile, initial assembly of the electronics was started.

Station 2.0 boundary layer probe
Station 2.0 pitot static probe
Station 3.0 pressure sensing head
Station 3.0 temperature sensing head
Station 4.9 pressure probe
Station 4.9 temperature probe
Station 4.9 and 5.0 air angle probe

Integrated Core/Low Spool Fan -- Fabrication of the solid, shrouded fan blades continued this report period. At present, airfoil machining has been completed (see Figure 68). The remaining operations to be performed include final machining of the dovetail and polishing the airfoil.

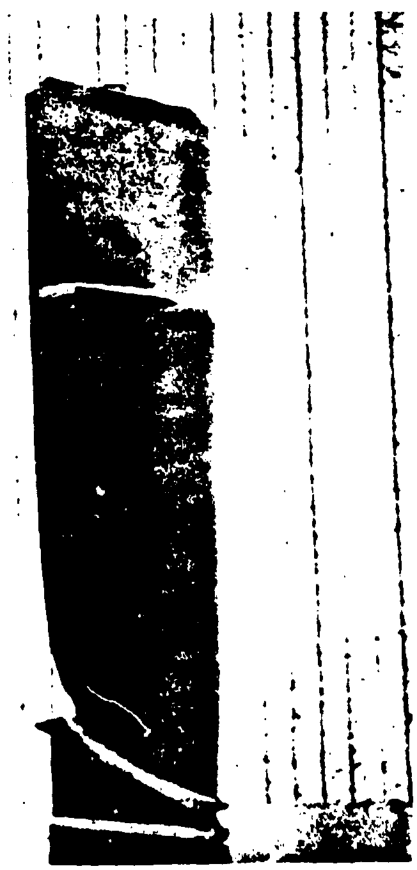

Figure 68 Fan Blade Nearing Final Stage of Fabrication

The rubstrip material for the fan containment case was received. Vendor installation of this rubstrip material is scheduled for completion in the next report period.

Machining of the stubshaft was completed during this period. The completed shaft is shown in Figure 69. In addition, fabrication of the fan rotor nose cone and cap was also completed, as shown in Figure 70.

Integrated Core/Low Spool Low Pressure Compressor -- All low-pressure compressor airfoils, except fifth stage blades, have been manufactured. Vendor fabrication of the fifth stage blades has been delayed by higher priority work. Figure 71 shows the airfoils in their representative position in the flowpath.

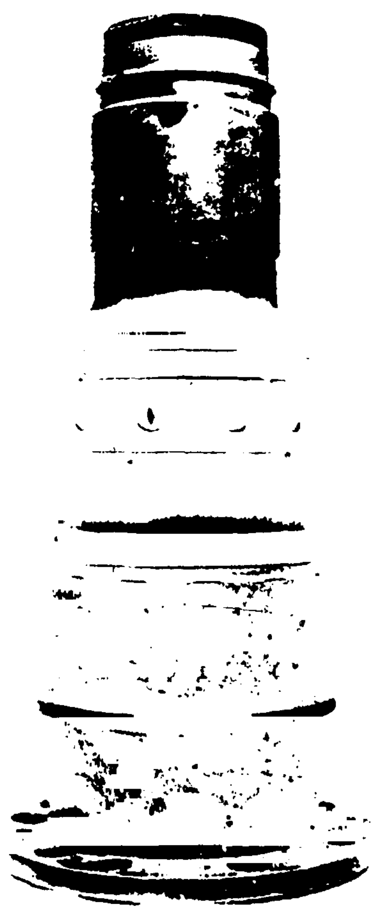

Figure 69 Completed Fan Stubshaft Subassembly

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Figure 70 Completed Fan Rotor Nose Cone and Cap

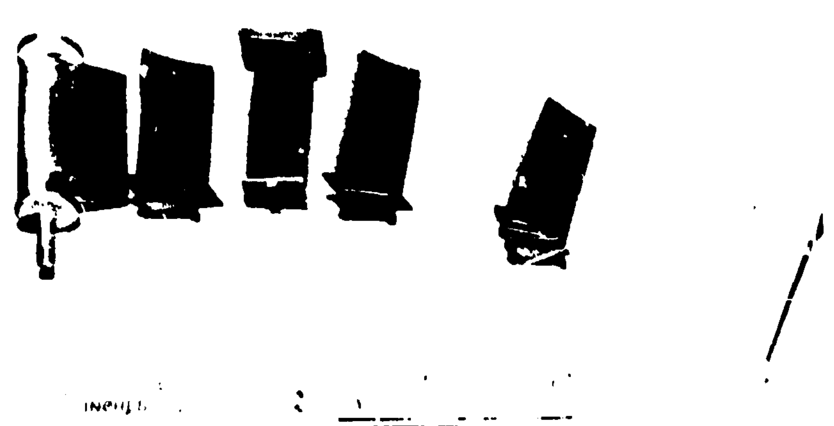

Figure 71 Completed Low-Pressure Compressor Airfoils

Machining of the rotor disks is nearing completion. The remaining operation is installation of the blade slots, which is currently scheduled for completion in April 1982.

Fabrication of the bleed case was completed. Bleed case turning vane cascades were received, along with the bleed valve ring and associated hardware. The bleed case is shown in Figure 72 and the turning vane cascade wax pattern is shown in Figure 73.

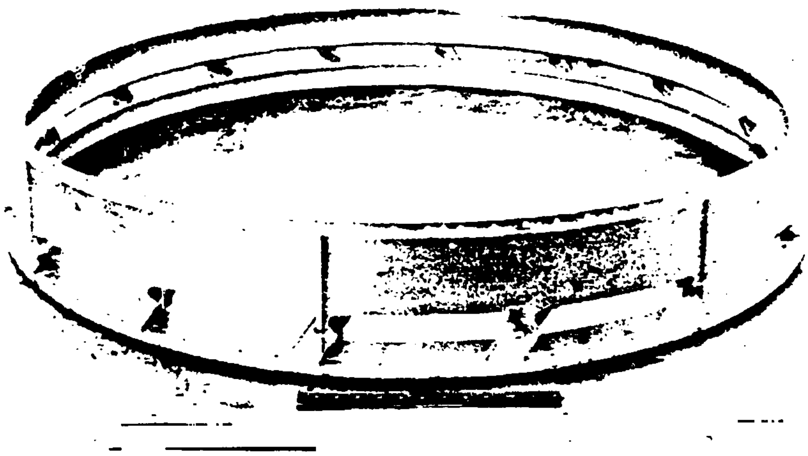

Figure 72 Completed Low-Pressure Compressor Bleed Case

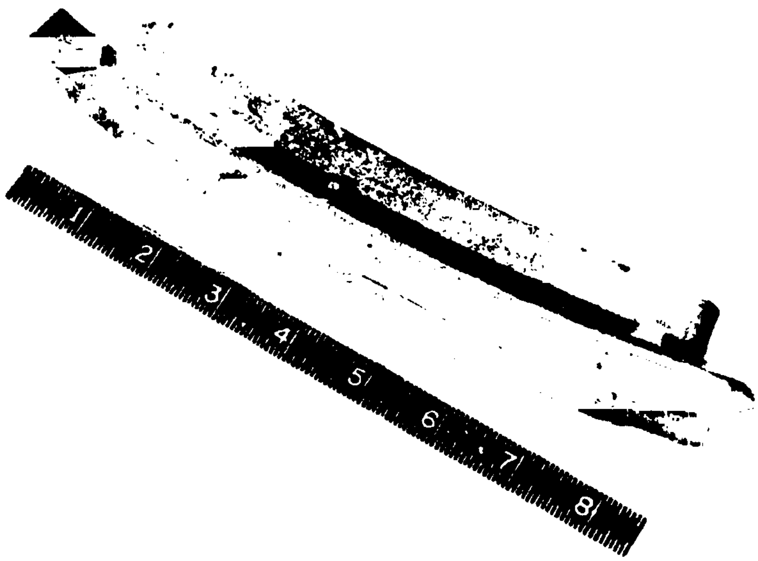

Figure 73 Completed Low-Pressure Compressor Vane Cascade Wax Pattern

Integrated Core/Low Spool Compressor Intermediate Case -- The design and fabrication of tooling has been completed. Attention concentrated on expediting the fabrication of the schedule-pacing 15-degree uncamber structural struts. Preparation of both the precamber and finishing dies was completed. The strut packs were successfully bonded and precambered. However, final superplastic expansion of the initial strut showed that the bonding pattern was misindexed because of an improperly placed locating hole in the mask used to apply the stop-off material that prevents unwanted bonding. This error precluded the use of these first two struts in the case assembly, and two new packs were prepared with the correct stop-off pattern.

The new 15-degree uncamber strut packs were then successfully bonded and formed in the precamber and finishing dies. Prior to machining the leading and trailing edges to receive the solid stiffeners, the struts were trimmed. Figures 74 and 75 show two views of a 15-degree uncamber strut at this stage of fabrication.

Figure 74 Bonded and Formed 15-Degree Uncamber Strut

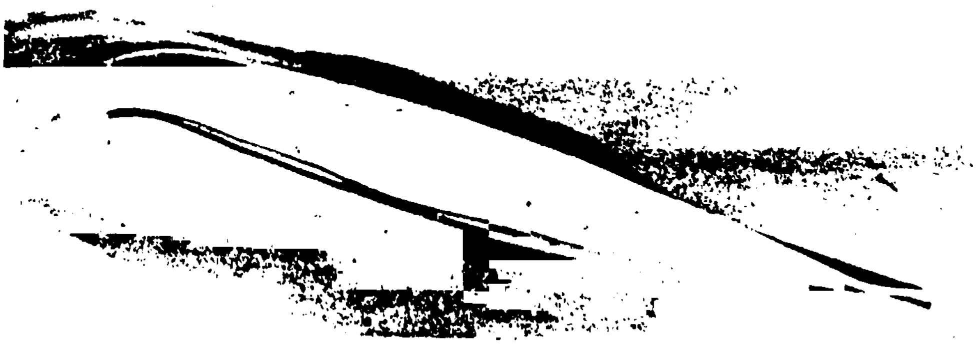

Figure 75 Bonded and Formed 15-Degree Uncamber Strut (Inner End View)

Machining of the 15-degree uncamber strut edges was initially delayed by warping of the nominal camber structural struts which occurred during welding of the solid stiffeners. Once the problem was understood, however, machining and welding operations were successfully completed. Figures 76 and 76a show the 15-degree uncamber strut immediately after stiffener welding. At the end of the reporting period, the struts were heat treated and essentially ready for the start of case assembly.

Figure 76 15-Degree Uncamber Strut Immediately After Stiffener Welding

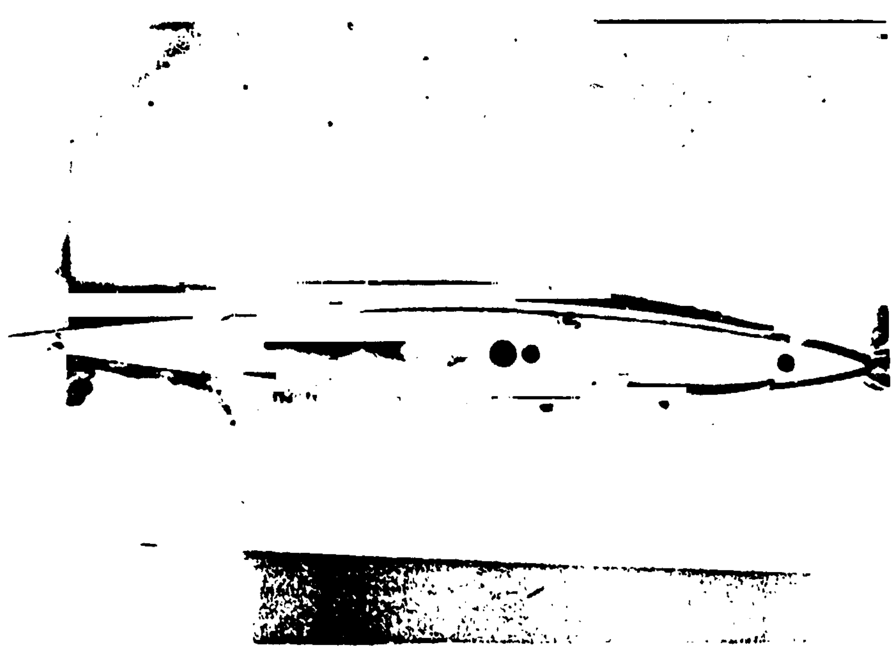

Figure 76a 15-Degree Uncamber Strut Immediately After Stiffener Welding

Fabrication of the nominal camber structural struts was completed with superplastic expansion of the struts in the finishing die accomplished. After completing this operation, the seven best of the ten struts were selected and their leading and trailing edges were trimmed and machined to accept the solid stiffeners. Welding of the stiffeners to the struts was then completed.

After heat treatment, all of the nominal camber struts exhibited some degree of warpage. A subsequent investigation attributed this problem to fixturing that did not provide the proper constraint, with some degree of weld shrinkage being a contributing factor. A method to resolve this problem was developed, and additional fixtures were fabricated.

At present, reprocessing of the seven nominal camber struts has been successfully completed. The struts are now ready for the start of case assembly.

All nonstructural vanes were procured, but an inspection uncovered inconsistencies between attachments from one vane design to the next. As a result, the vanes were returned to the vendor where the proper attachments were incorporated.

Pylon strut fabrication became the pacing item for the initiation of case assembly during the second half of the period. The side panels of the strut were successfully formed, but machining was impeded by the lack of computer tapes. Fabrication of assembly tooling was completed on schedule, but an inspection of the tooling uncovered several discrepancies that were subsequently corrected. Strut components were installed in their fixtures, and welding of the pylon subassemblies was completed. The subassembly is shown in Figure 77. The fixture assembly is shown in Figure 77a prior to final welding. Welding was initiated at the end of the report period.

Also completed was the fabrication of the other case parts, including the inner duct rear flange and ring, outer duct ring, outer core front ring segments, and inner core ring. The completed inner core ring is illustrated in Figure 78.

Figure 77 Integrated Core/Low Spool Intermediate Case Pylon Strut Subassembly

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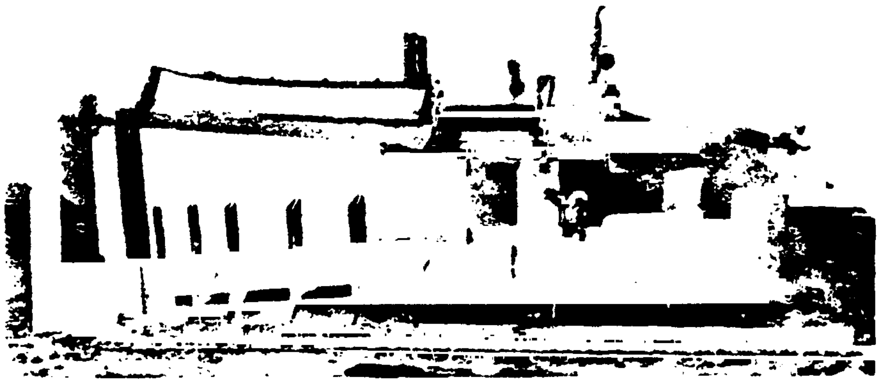

Figure 77a Integrated Core/Low Spool Intermediate Case Fixture Assembly Prior To Final Welding

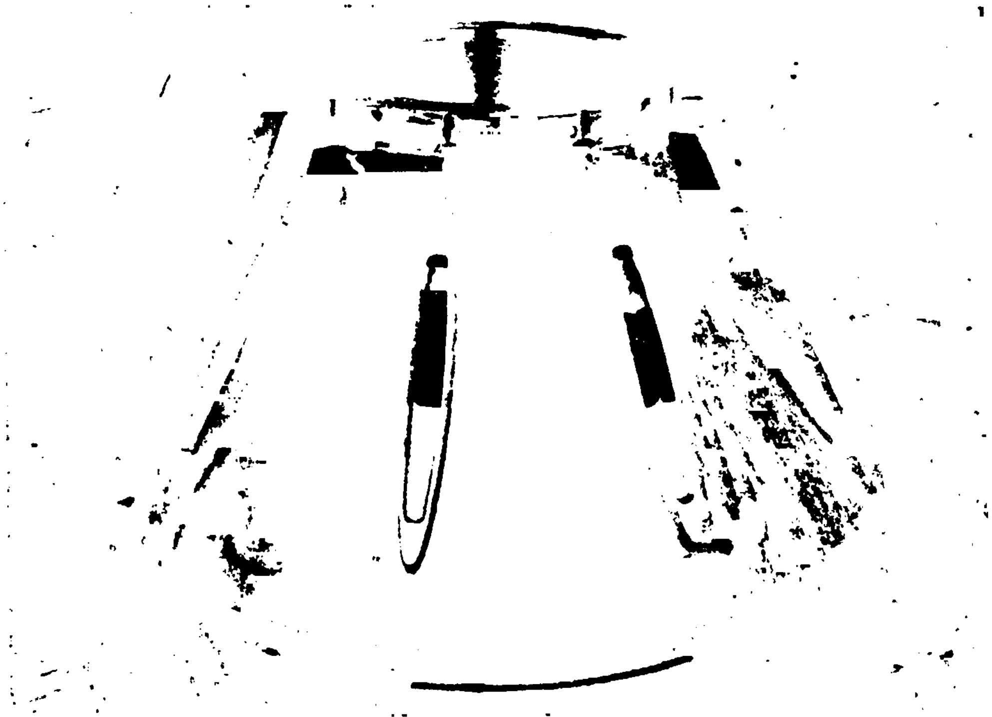

Figure 78 Completed Integrated Core/Low Spool Intermediate Case Inner Core Ring (Centerbody)

As part of the intermediate case accessory drive fabrication effort, teeth on the first gear set were recut to compensate for initial contact deviations. A general inspection indicated that the desired profiles were achieved. Splines were cut on the gears, and a detailed inspection was conducted prior to heat treatment. After heat treatment, both gears of the set were inspected. Results indicated that both the driving gear and the driven gear will be acceptable following minor additional tooth machining. Some minor machining on the front and rear faces of the first gear set has been completed.

Fabrication of all remaining accessory drive system parts was successfully completed during the reporting period. These vendor-fabricated items include the outer and inner towershafts, the outer and inner center towershaft couplings, and the center towershaft bearing.

Integrated Core/Low Spool High-Pressure Compressor -- With the exception of the single row exit guide vanes, all blades and vanes in the high-pressure compressor have been manufactured, and an in-house inspection of the airfoils is in process. Meanwhile, fabrication of the compressor exit guide vanes is in progress. Figure 79 shows the high-pressure compressor airfoil array.

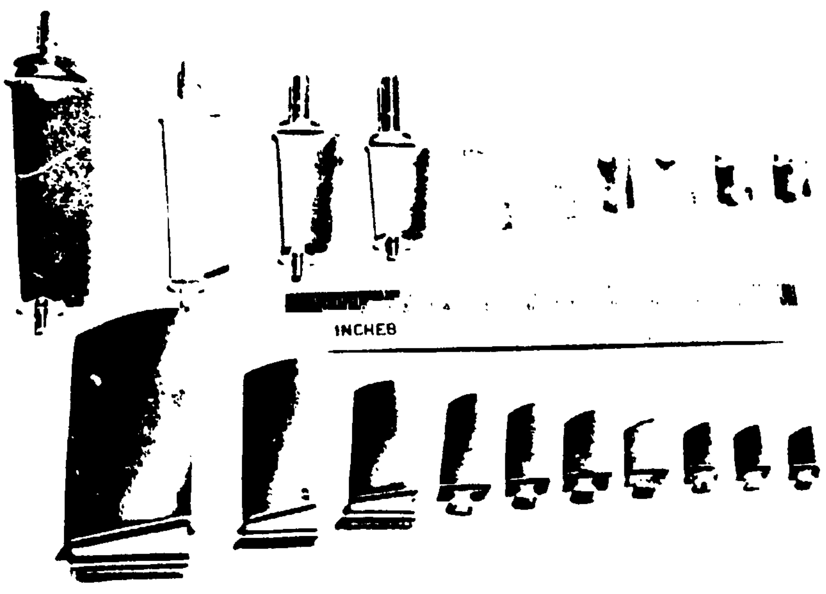

Figure 79 High-Pressure Compressor Airfoil Array

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The flange weldment on the front split compressor case was completed, and finished machining has been initiated. Also, the compressor bleed case was machined to the preweld configuration. The bleed manifolds and flanges were then welded to the case, and final machining of the case was started.

Provisions for instrumentation have been installed in the titanium rotor disks. The disks were electron beam welded together, as shown in Figure 80. X-ray inspection verified the integrity and quality of the welds. Final machining operations on the MERL 76 rear rotor were started . To date, all lathe work has been completed. Installation of flange details has been initiated. Figure 80a shows the rotor prior to installation of flange holes. Fabrication of the detail parts for the fixed stator shroud assembly was initiated. All tooling was prepared, and machining of the outer shroud cases was started.

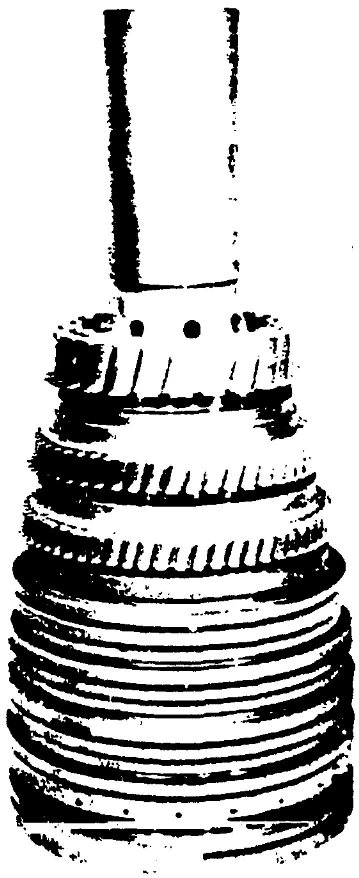

Figure 80 Integrated Core/Low Spool High-Pressure Compressor Assembled Rotor

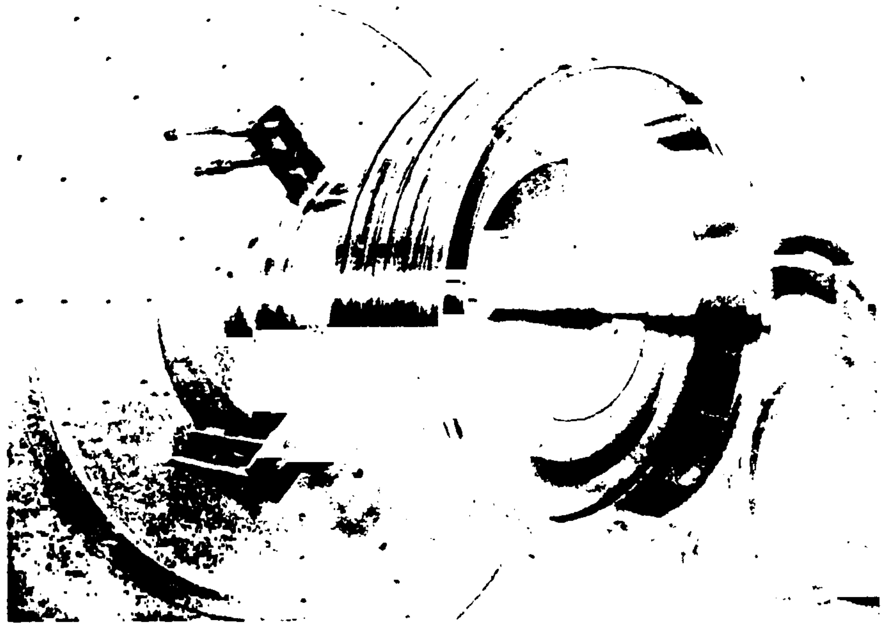

Figure 80a Integrated Core/Low Spool High-Pressure Compressor MERL 76 Rotor

Integrated Core/Low Spool Combustor -- No significant fabrication effort directed toward combustor hardware was conducted during the current reporting period.

Integrated Core/Low Spool High Pressure Turbine:

Vanes -- The casting vendor continued efforts to successfully cast PWA 1480 single crystal vanes. Casting problems reported previously have for the most part been successfully resolved. However, a new difficulty has been experienced involving cracking of the airfoil at the leading edge to outer shroud fillet. Because of casting difficulties with the PWA 1480 material, PWA 1422, a directionally solidified material, has been used to satisfy vane requirements for the integrated core/low spool engine. To date, a total of 32 castings has been delivered, including three castings of PWA 1480 material determined to be acceptable for use in the integrated core/low spool engine.

Blades -- Core shift problems have been experienced with casting of the blades during this report period. Consequently, only 32 castings out of an order of 70 were delivered.

Disk -- Flourescent penetrant inspection of the heat treated high-pressure turbine disk disclosed a fine crack extending 360 degrees around the circumference adjacent to the integral test material area on the rear side of the web. An additional metallurgical inspection showed that the material had been overtemperatured during the heat treat process. Rather than repeating the hot isostatic pressing operation and risk distortion of the sonic shape, it was decided to use the spare disk compaction. At present, the spare compaction has been machined to a sonic shape and completed heat treatment. The part is now ready for sonic inspection.

Integrated Core/Low Spool Low-Pressure Turbine -- Fabrication of the number 5 bearing is expected to be completed in April 1982.

3.3.3.3 Integrated Core/low Spool - Assembly and Inspection

3.3.3.3.1 Summary of Work Previously Completed

The assembly floor planning as well as tool design and fabrication efforts associated with the assembly of the first build of the integrated core/low spool engine were initiated.

3.3.3.3.2 Current Technical Progress

A detailed assembly schedule has been formulated, firming up the previously projected assembly start date of July 1982. An estimated 200 new tooling items are required for assembly of the integrated core/low spool engine. Of these, 184 are at least in the design phase with fabrication of some items nearly complete.

3.3.3.4 Integrated Core/low Spool - Test Engineering and Support

3.3.3.4.1 Current Technical Progress

System-Related Activities -- All of the work planned for integrated core/low spool system-related activities for this reporting period was completed. The following table presents a summary of the work accomplished with a discussion of these efforts contained in subsequent paragraphs.

TABLE 3-XVIII

SUMMARY OF SYSTEMS ANALYSES FOR THE INTEGRATED CORE/LOW SPOOL

Predicted Performance Update Comparison
Operating Sensitivity to Core Efficiency Levels
Estimated Transient Operation
Fuel Control Software Review
Starting Requirements Review
Detailed Design Review Planning

Predicted Performance Update Comparison -- Analysis of the July 1981 performance update of the integrated core/low spool was completed. Separate and mixed flow exhaust configurations were evaluated with and without instrumentation. Results for the more pertinent performance and operating parameters are presented in Table 3-XIX for the sea level static, hot day takeoff condition and in Table 3-XX for the maximum cruise flight condition. Design data for the integrated core/low spool are included for comparison.

A comparison of results shows status component and overall performance differences are generally small relative to design levels. However, rotor speeds may be limited during testing because rematching has caused high- and low-pressure spool speeds to increase 0.9 and 0.6 percent, respectively, at takeoff and 0.6 and 2.6 percent, respectively, at maximum cruise compared to design values. In addition, rated airflows have increased accordingly. Temperatures at the ratings appear close enough to design levels so that no problems are anticipated for testing the integrated core/low spool.

TABLE 3-XIX

INTEGRATED CORE/LOW SPOOL COMPONENT PERFORMANCE COMPARISON
(July 1981 Status)
(TAKEOFF: 0 ft, 0 Mn, 84°F Day -- 50% Probability of Achievement)

	Separate Exhaust W/O Inst	Separate Exhaust With Inst	Mixed Exhaust W/O Inst	Mixed Exhaust With Inst	Design
Fan					
Pressure Ratio					
(Duct)	1.60	1.58	1.61	1.60	1.57
(Core)	1.46	1.45	1.47	1.47	1.45
Corrected Flow (lb/sec)					
(Duct)	1065	1060	1075	1075	1055
(Core)	158.9	155 7	160.2	158.9	150.4
Rotor Speed (rpm)	3940	3910	3965	3955	3885
Exit Temperature (°F)					
(Duct)	175	173	176	176	171
(Core)	154	153	155	155	152
Low-Pressure Compressor					
Pressure Ratio	1.64	1.64	1.65	1.65	1.64
Inlet Corr Flow (lb/sec)	115.4	113.8	115.9	115.3	110.2
Exit Temperature (°F)	258	256	261	260	257

TABLE 3-XIX (continued)

INTEGRATED CORE/LOW SPOOL COMPONENT PERFORMANCE COMPARISON
(July 1981 Status)
(TAKEOFF: 0 ft, 0 Mn, 84°F Day — 50% Probability of Achievement)

	Separate Exhaust W/O Inst	Separate Exhaust With Inst	Mixed Exhaust W/O Inst	Mixed Exhaust With Inst	Design
High-Pressure Compressor					
Pressure Ratio	13.25	13.20	13.20	13.15	12.85
Inlet Corr Flow (lb/sec)	75.95	75.35	75.70	75.40	72.35
Rotor Speed (rpm)	14120	14065	14130	14110	13935
Exit Temperature (°F)	1100	1095	1100	1100	1085
Combustor					
Inlet Corr Flow (lb/sec)	6.99	6.95	7.00	6.98	6.94
Exit Temperature (°F)	2770	2770	2770	2770	2770
High-Pressure Turbine					
Rotor Inlet Temp. (°F)	2620	2620	2620	2620	2630
Pressure Ratio	4.07	4.03	4.07	4.06	4.08
Low-Pressure Turbine					
Inlet Temperature (°F)	1840	1845	1840	1845	1860
Pressure Ratio	4.88	4.80	5.01	5.00	4.86
Exit Temperature (°F)	1170	1180	1160	1165	1205

TABLE 3-XX

INTEGRATED CORE/LOW SPOOL COMPONENT PERFORMANCE COMPARISON
(July 1981 Status)
(MAXIMUM CRUISE: 35000 ft., 0.8 Mn, Standard Day
-- 50% Probability of Achievement)

	Separate Exhaust W/O Inst	Separate Exhaust With Inst	Mixed Exhaust W/O Inst	Mixed Exhaust With Inst	Design
Fan					
Pressure Ratio					
(Duct)	1.71	1.69	1.71	1.70	1.71
(Core)	1.55	1.54	1.57	1.58	1.55
Corrected Flow (lb/sec)					
(Duct)	1180	1180	1210	1220	1180
(Core)	180.2	178.0	184.4	184.8	178.2
Rotor Speed (rpm)	3855	3845	3930	3955	3855
Exit Temperature (°F)					
(Duct)	72	70	72	72	71
(Core)	52	52	55	56	52
Low-Pressure Compressor					
Pressure Ratio	1.73	1.74	1.78	1.80	1.75
Inlet Corr Flow (lb/sec)	125.0	123.9	126.4	126.2	123.6
Exit Temperature (°F)	152	152	160	163	153
High-Pressure Compressor					
Pressure Ratio	14.00	13.95	13.75	13.60	13.85
Inlet Corr Flow (lb/sec)	78.65	77.90	77.65	76.95	77.05
Rotor Speed (rpm)	13220	13175	13245	13240	13160
Exit Temperature (°F)	916	912	924	924	911
Combustor					
Inlet Corr Flow (lb/sec)	6.97	6.93	6.98	6.97	6.94
Exit Temperature (°F)	2445	2445	2445	2445	2445
High-Pressure Turbine					
Rotor Inlet Temp (°F)	2310	2310	2310	2310	2315
Pressure Ratio	4.09	4.05	4.08	4.06	4.09
Low-Pressure Turbine					
Inlet Temperature (°F)	1595	1600	1595	1600	1605
Pressure Ratio	5.50	5.46	5.69	5.75	5.46
Exit Temperature (°F)	948	953	937	935	962

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Operational Sensitivity to Core Efficiency Levels -- Potential operational problems resulting from possibly lower than expected high-pressure compressor and turbine efficiencies were evaluated. The analysis was conducted for a separate exhaust configuration with instrumentation. Results are summarized for pertinent component parameters in Tables 3-XXI and 3-XXII for the sea level static, hot day takeoff and the maximum cruise conditions, respectively. Data from the July 1981 performance update of the integrated core/low spool are included as the basis for comparison.

TABLE 3-XXI

EFFECT OF CORE COMPONENT EFFICIENCIES ON INTEGRATED CORE/LOW SPOOL PERFORMANCE
(Separate Exhaust With Instrumentation - Takeoff: 0 ft, 0 Mn, 84°F Day)

	July 1981 Status	-1% Change High Compressor Efficiency	-1% Change High Turbine Efficiency
Fan			
Pressure Ratio			
(Duct)	1.58	1.57	1.57
(Core)	1.45	1.45	1.45
Corrected Flow (lb/sec)			
(Duct)	1060	1050	1050
(Core)	155.7	152.9	153.0
Rotor Speed (rpm)	3910	3880	3885
Exit Temperature (°F)			
(Duct)	173	172	172
(Core)	153	152	152
Low-Pressure Compressor			
Pressure Ratio	1.64	1.64	1.64
Inlet Corrected Flow (lb/sec)	113.8	112.2	112.2
Exit Temperature (°F)	256	255	256
High-Pressure Compressor			
Pressure Ratio	13.20	13.05	13.00
Inlet Corrected Flow (lb/sec)	75.35	74.25	74.10
Rotor Speed (rpm)	14065	13990	13990
Exit Temperature (°F)	1095	1090	1085
Combustor			
Inlet Corrected Flow (lb/sec)	6.95	6.95	6.93
Exit Temperature (°F)	2770	2770	2770
High-Pressure Turbine			
Rotor Inlet Temperature (°F)	2620	2620	2620
Pressure Ratio	4.03	4.03	4.03
Low-Pressure Turbine			
Inlet Temperature (°F)	1845	1845	1355
Pressure Ratio	4.80	4.76	4.77
Exit Temperature (°F)	1180	1185	1190

TABLE 3-XXII

EFFECT OF CORE COMPONENT EFFICIENCIES OF INTEGRATED CORE/LOW SPOOL PERFORMANCE
(Separate Exhaust, With Instrumentation --
Maximum Cruise: 35000 ft, 0 Mn, Standard Day)

	July 1981 Status	-1% Change High Compressor Efficiency	-1% Change High Turbine Efficiency
Fan			
Pressure Ratio			
(Duct)	1.69	1.68	1.69
(Core)	1.54	1.54	1.54
Corrected Airflow (lb/sec)			
(Duct)	1180	1175	1180
(Core)	178.0	175.7	176.0
Rotor Speed (rpm)	3845	3825	3830
Exit Temperature (°F)			
(Duct)	70	69	70
(Core)	52	51	51
Low-Pressure Compressor			
Pressure Ratio	1.74	1.75	1.75
Inlet Corr Flow (lb/sec)	123.9	122.6	122.7
Exit Temperature (°F)	152	151	152
High-Pressure Compressor			
Pressure Ratio	13.95	13.75	13.75
Inlet Corr Flow (lb/sec)	77.90	76.85	76.75
Rotor Speed (rpm)	13175	13110	13110
Exit Temperature (°F)	912	911	903
Combustor			
Inlet Corr Flow (lb/sec)	6.93	6.93	6.91
Exit Temperature (°F)	2445	2445	2445
High-Pressure Turbine			
Rotor Inlet Temperature (°F)	2310	2310	2310
Pressure Ratio	4.05	4.05	4.04
Low-Pressure Turbine			
Inlet Temperature (°F)	1600	1600	1605
Pressure Ratio	5.46	5.45	5.46
Exit Temperature (°F)	953	955	960

Results with lower core efficiencies showed no operational problems. This is primarily because the integrated core/low spool will operate to a rated temperature rather than a rated thrust. Consequently, rotor speeds, airflows and pressure levels are generally reduced by additional inefficiencies. More specific results are summarized in the following table.

TABLE 3-XXIII

EFFECT OF UNEXPECTEDLY LOW HIGH-PRESSURE COMPRESSOR AND HIGH-PRESSURE TURBINE EFFICIENCIES ON INTEGRATED CORE/LOW SPOOL OPERATION

High-Pressure Compressor

- o Temperature Limit at Rating Precludes Operating Problems - Thrust Reduced
- o Pressure, Pressure Ratio, Airflow, and Rotor Speed Levels Reduced
- o Bypass Ratio and Thrust Specific Fuel Consumption Increased
- o Cooling Air Temperature Lowered
- o High-Pressure Turbine Parameters Not Affected
- o Low-Pressure Turbine Work Reduced and Exit Temperature Increased

High-Pressure Turbine

- o Temperature Limit at Rating Precludes Operating Problems - Thrust Reduced
- o Pressure, Pressure Ratio, Airflow, and Rotor Speed Levels Reduced
- o Bypass Ratio and Thrust Specific Fuel Consumption Increased
- o Cooling Air Temperature Lowered
- o High-Pressure Turbine Work Lowered and Exit Temperature Increased
- o Low-Pressure Turbine Work Not Affected and Exit Temperature Increased

Estimated Transient Operation -- An analysis of engine transient operation was conducted to assess compression system stability. In this study, the separate exhaust configuration was evaluated at sea level and the mixed exhaust configuration was evaluated both at sea level and altitude. The simulation used in these analyses also included the effects of instrumentation. Preliminary surge margin requirements for the flight propulsion system were used to measure stability adequacy.

Evaluations were made for the three transient modes planned for the integrated core/low spool control system. These include a rate-limited deceleration, snap (emergency) deceleration, and rate-limited acceleration. Rate limited power lever movements were on the order of 20 seconds at sea level and 16.7 seconds at altitude. Sea level transients were between takeoff and ground idle, while altitude transients were between maximum climb and flight idle (minimum fuel flow).

Results of this analysis showed that fan stability is adequate. Transient and steady state operating lines were coincident for the different configurations and conditions. However, some potential instabilities were evident with the low- and high-pressure compression systems. Figures 81 through 83 present the typical operating characteristics of the low-pressure compressor. Similar information is shown in Figures 84 through 86 for the high-pressure compressor. The results in these figures are for a mixed configuration at sea level conditions, but they are indicative of the operating trends shown at sea level with a nonmixed configuration and at altitude with the mixed configuration.

On the basis of this analysis, the following conclusions have been made.

- A potential deficiency in low-pressure compressor surge margin exists in the region above idle during a rate-limited deceleration, as shown in Figure 81.

- Immediate opening of the exit surge bleed provides protection for the low-pressure compressor during a snap deceleration, as shown in Figure 82.

- A potential deficiency in high-pressure compressor surge margin exists in the region above idle during a rate-limited acceleration, as shown in Figure 86.

- No other potential stability problems are anticipated for the compression system.

Several control system revisions have been recommended to obviate any potential compression system instabilities. These consist of the following.

- Actuation of the low-pressure compressor exit surge bleed should be moved up 350 rpm in high-pressure rotor speed to preclude any low-pressure compressor surge margin inadequacy during rate-limited deceleration, as shown in Figure 81.

- Actuation of the low-pressure compressor exit surge bleed should be based on mechanical rather than corrected high-pressure rotor speed to provide adequate surge margin for the low-pressure compressor during rate-limited deceleration at altitude.

- A slower power lever angle movement should be used during rate-limited acceleration to provide surge margin adequacy for the high-pressure compressor, if high-pressure compressor rig testing indicates that stability is worse than the analytical results in Figure 86 show.

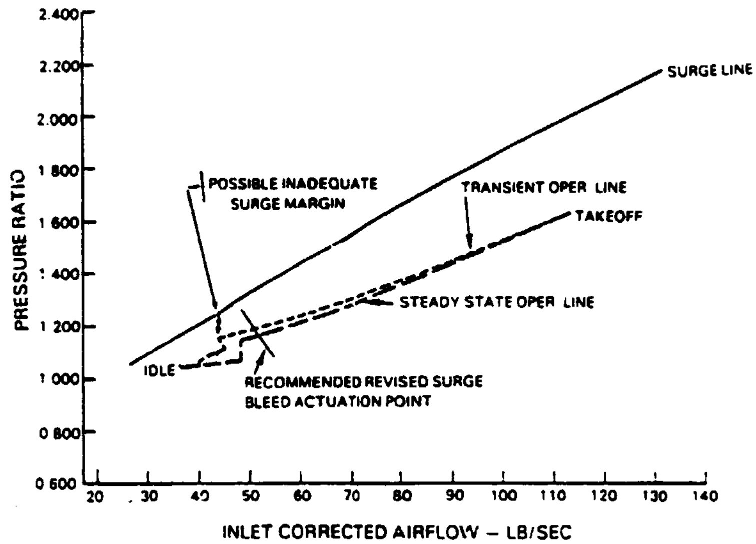

Figure 81 Low-Pressure Compressor Rate-Limited Deceleration Operating Characteristics for a Mixed Exhaust Configuration at Sea Level Static Conditions

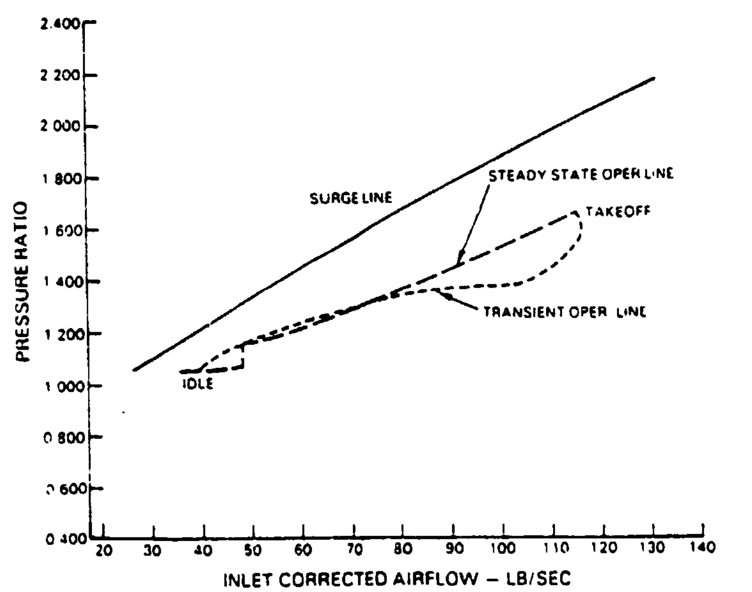

Figure 82 Low-Pressure Compressor Snap Deceleration Operating Characteristics for a Mixed Exhaust Configuration at Sea Level Static Conditions

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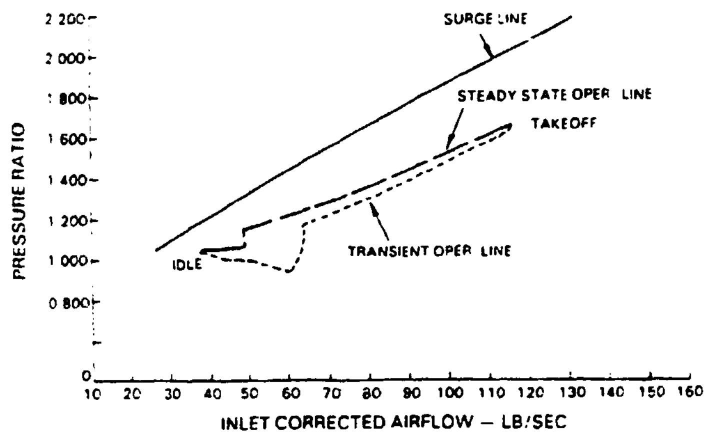

Figure 83 Low-Pressure Compressor Rate-Limited Acceleration Operating Characteristics for a Mixed Exhaust Configuration at Sea Level Static Conditions

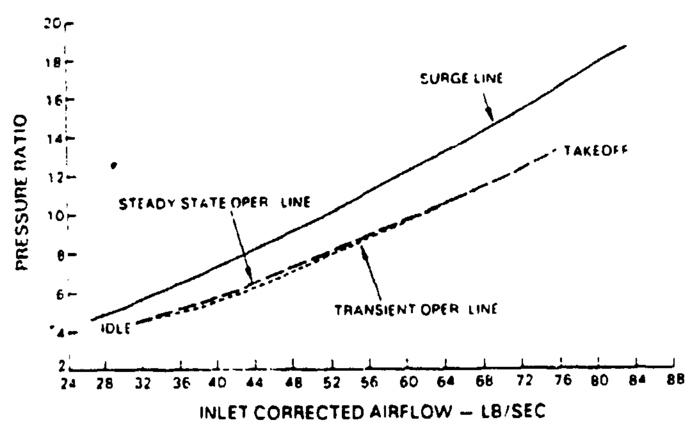

Figure 84 High-Pressure Compressor Rate-Limited Deceleration Operating Characteristics for a Mixed Exhaust Configuration at Sea Level Static Conditions

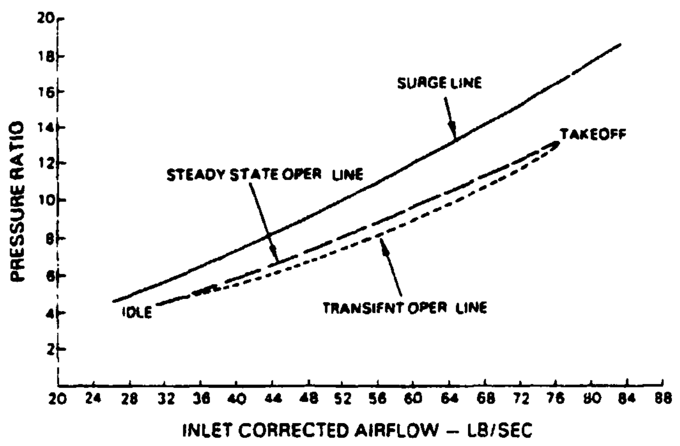

Figure 85 High-Pressure Compressor Snap Deceleration Operating Characteristics for a Mixed Exhaust Configuration at Sea Level Static Conditions

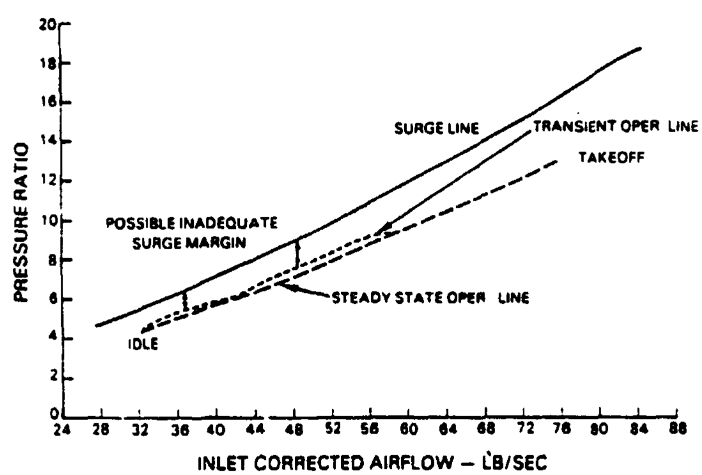

Figure 86 High-Pressure Compressor Rate-Limited Acceleration Operating Characteristics for a Mixed Exhaust Configuration at Sea Level Static Conditions

Fuel Control Software Review -- A review of the control system software specified for the integrated core/low spool was conducted. Component and overall system data from the July 1981 performance update and the transient analysis were the basis for this review.

Several changes were recommended, in addition to those made as a result of the transient operation analysis. A Mach number bias was suggested in the event that the integrated core/low spool should be tested at altitude. Increases in maximum high- and low-pressure rotor speed limits were recommended to provide more realistic operating margins relative to speed levels currently expected during integrated core/low spool testing.

Starting Requirements Review -- A review of starting requirements for the integrated core/low spool was also made in this reporting period. The different items investigated with respect to their impact on starting included high-pressure compressor rig surge testing, high-pressure turbine operating (motoring) line predictions, combustor rig lighting experience, stability airbleed capabilities, and other engine experience.

High-pressure compressor testing in the starting region indicated that regions of rotating stall exist in the vicinity of the operating (motoring) line defined by the July 1981 performance simulation. However, since the operating line is very difficult to locate accurately in the starting region, other factors were examined to determine the direction and magnitude of their impact.

The influence of the fifteenth stage compressor bleed was assessed since compressor rig testing showed that the tenth stage starting bleed was not effective in providing more surge margin in the starting region. Fifteenth stage bleed capability was estimated to be 7.4 percent with the current active clearance control plumbing design and up to a maximum of 15.0 percent with plumbing modifications. Surge margin improvements of 0.8 and 1.6 percent were projected for the respective bleed capabilities.

A reassessment of high-pressure turbine operation at low power conditions indicated that the effective controlling vane area is larger than assumed in the performance simulation, as shown in Figure 87. This should provide an additional high-pressure compressor surge margin benefit of at least 1.2 percent.

Combustor rig testing indicated that lighting will be 'soft' and accomplished at low fuel flow rates. Therefore, combustor ignition should not significantly influence high-pressure compressor operation during starting.

Finally, a comparison of integrated core/low spool starting analysis results was made with other Pratt & Whitney engine starting experience. The conclusions reached are that analytical predictions of starting capability have proven to be rather pessimistic in the past and that the integrated core/low spool predictions appear to be comparable to those engines that have never experienced starting difficulties.

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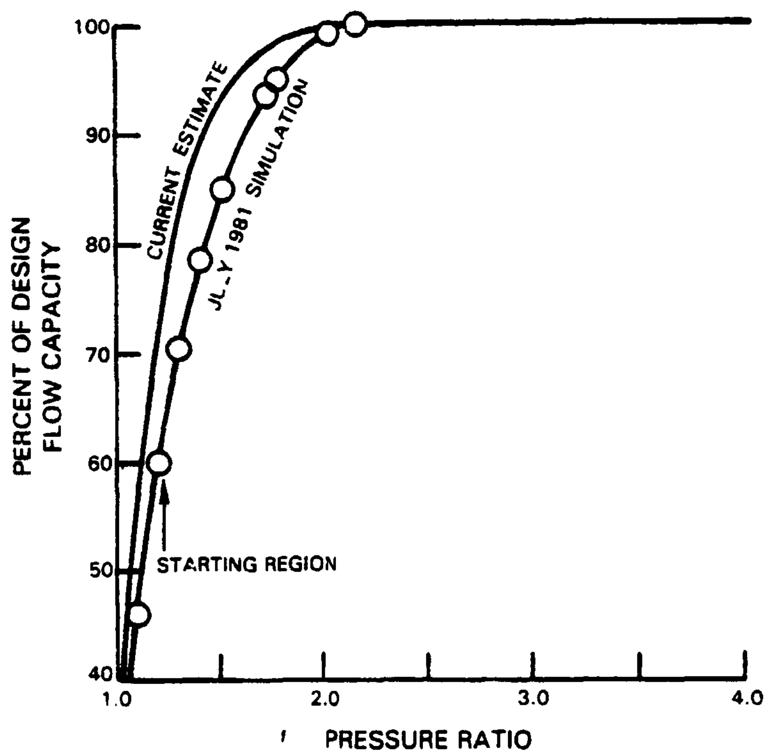

Figure 87 Integrated Core/Low Spool High-Pressure Turbine Flow Capacity Comparison

From this review, the following recommendations were made.

- Provision should be made to have the capability to extract the maximum possible bleed from the fifteenth stage active clearance control plumbing system.
- High-pressure turbine rig testing should include starting region operation to confirm analytically-predicted flow capacity levels.
- The starter should be kept engaged to the highest possible high-pressure rotor speed to help accelerate the spool to idle.

Detailed Design Review Planning -- Planning of performance updating activities for the detailed design review of the integrated core/low spool was completed. However, the performance effort was not started.