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SEVENTH SEMIANNUAL STATUS REPORT
VOLUME I

1 April 1981 - 30 September 1981

ENERGY EFFICIENT ENGINE
COMPONENT DEVELOPMENT AND INTEGRATION PROGRAM

30 October 1981

Contract NAS3-20646

(NASA-CR-173084) ENERGY EFFICIENT ENGINE. VOLUME 1: COMPONENT DEVELOPMENT AND INTEGRATION PROGRAM Semiannual Status Report, 1 Apr. - 30 Sep. 1981 (Pratt and Whitney Aircraft) 228 p HC A11/MF A01 G3/07

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Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Lewis Research Center
Cleveland, Ohio

PWA-5594-179

UNITED TECHNOLOGIES
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In reply please refer to:
WBG:DH(00330) - E1M3
LC-81-121

30 October 1981

To: National Aeronautics and Space Administration
Lewis Research Center
21000 Brookpark Road
Cleveland, Ohio 44135

Attention: Mr. C.C. Ciepluch, Mail Stop 301-4

Subject: Submittal of the Seventh Semiannual Status Report (Volumes I and II) - PWA-5594-179, Contract NAS3-20646, Energy Efficient Engine Component Development and Integration Program

References:
(a) Contract No. NAS3-20646
(b) Letter, N.T. Saunders to W.B. Gardner, 'Guidelines for Semiannual Status Reports,' dated April 11, 1980, ref. 2320
(c) Revised distribution list for the Seventh Semiannual Status Report, dated October 7, 1981

Enclosures: 18 Copies of the Subject Report

Gentlemen:

In accordance with the requirements of references (a) and (b), we are pleased to submit 18 copies of the subject report. Distribution is being made to all authorized recipients in accordance with reference (c).

Sincerely yours,

UNITED TECHNOLOGIES CORPORATION
Pratt & Whitney Aircraft Group
Commercial Products Division

D. C. Hamm/for

W. B. Gardner
Program Manager

cc: Administrative Contracting Officer
Air Force Plant Representative Office
Pratt & Whitney Aircraft Group
East Hartford, Connecticut 06108

UNITED TECHNOLOGIES

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COMMERCIAL PRODUCTS DIVISION

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 April 1981 through 30 September 1981. 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.

Selected portions of the data (pertaining to specific component design details) are considered to have significant early commercial potential. As such, these data are designated as Category-2 Data under NASA FEDD (For Early Domestic Dissemination) Policy and are restricted from foreign dissemination for at least two years from the date of this report. Category-2 data may be duplicated and used by the recipient with the expressed limitation that the data will not be published or released to foreign parties during this period without the expressed permission of Pratt & Whitney Aircraft and appropriate export licenses. Release of this Category-2 data to other domestic parties shall only be made subject to the limitations that all recipients must agree, prior to the receipt of these data, to abide by the limitations of the FEDD legend which appears on the cover of this report.

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TABLE OF CONTENTS

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LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (continued)

LIST OF ILLUSTRATIONS (continued)

LIST OF ILLUSTRATIONS (continued)

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LIST OF ILLUSTRATIONS (continued)

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LIST OF ILLUSTRATIONS (continued)

LIST OF ILLUSTRATIONS (continued)

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 Aircraft. The Pratt & Whitney Aircraft 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 Aircraft 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 on emissions (meet the proposed EPA 1981 regulation) and noise (meet FAR 36-1978 standards) have been established.

The Pratt & Whitney Aircraft 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 Aircraft 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 Aircraft Project Team and defined in the original Program Work Plan.

Task 1, Propulsion System Analysis, Design and Integration - 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.

Task 2, Component Analysis, Design and Development - 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 current 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:

- o Deletion of the Task 3 effort and the addition of a second test to the Task 4 integrated core/low spool effort.

- o Redefinition of the fan effort to replace the primary shroudless fan with a shrouded design and reduction of the Hollow Blade Supporting Technology Program to a fabrication feasibility effort.

- o Addition of a third test to the High-Pressure Compressor Rig Program.

- o Addition of a tangential on-board injection (TOBI) rig 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.

- o Deletion of the machining of one set of advanced combustor liner segments and liner supports.

- o Addition of a benefit/cost study of potential fuel saving technologies in order to identify those suitable for follow-on technology development programs.

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 (30 September 1981) 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 sub-tasks 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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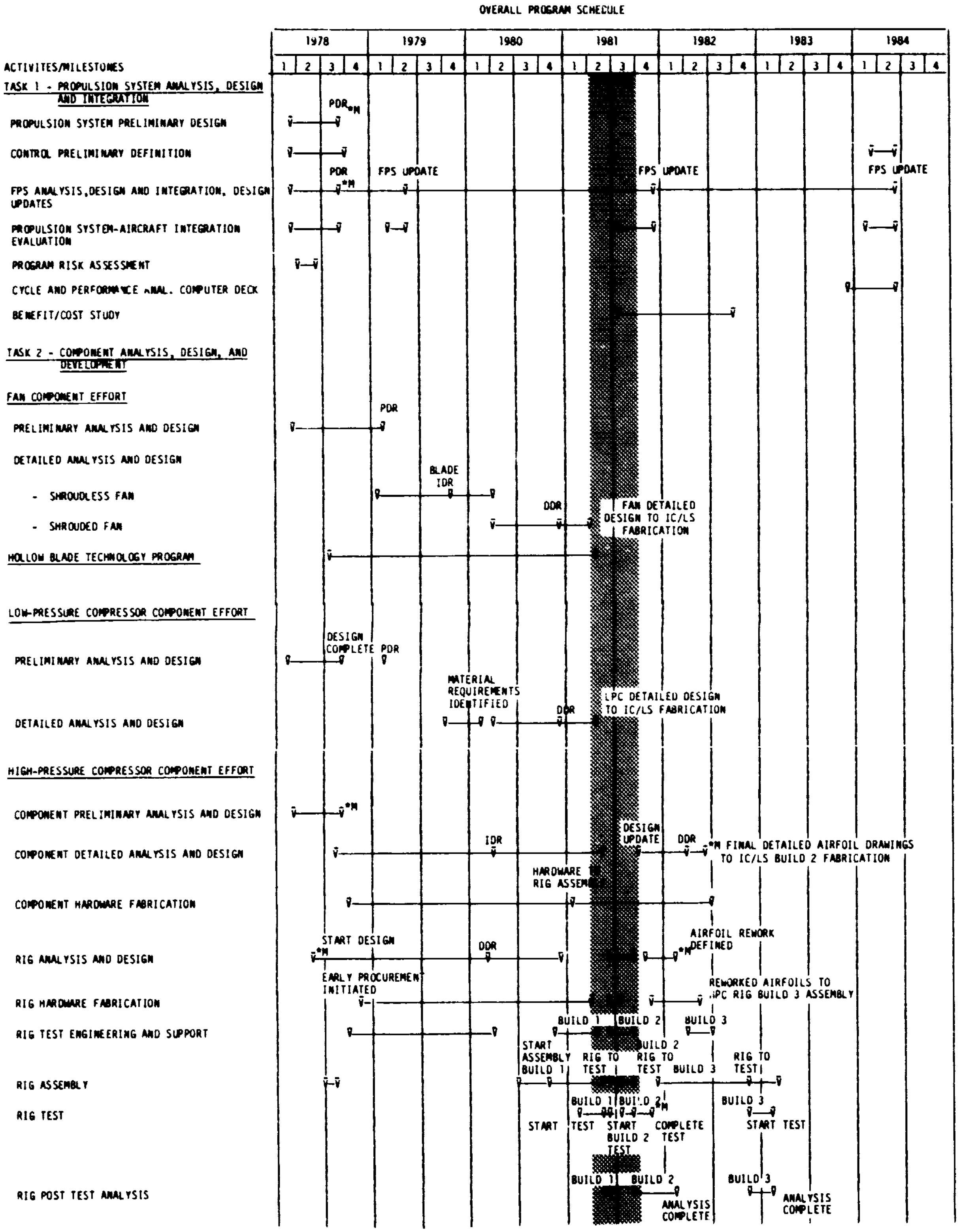

Figure 1   Overall Program Logic Diagram

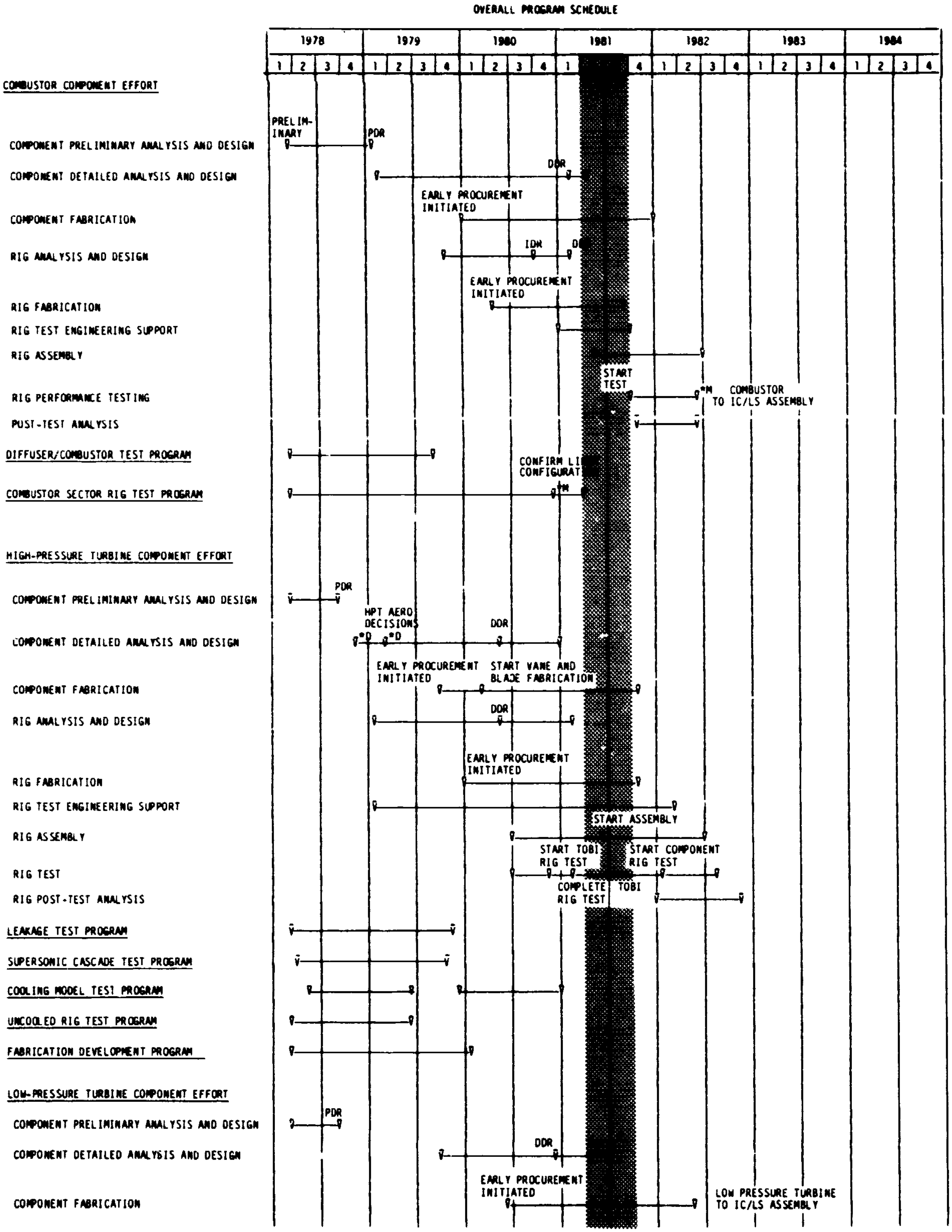

Figure 1 Overall Program Logic Diagram (continued)

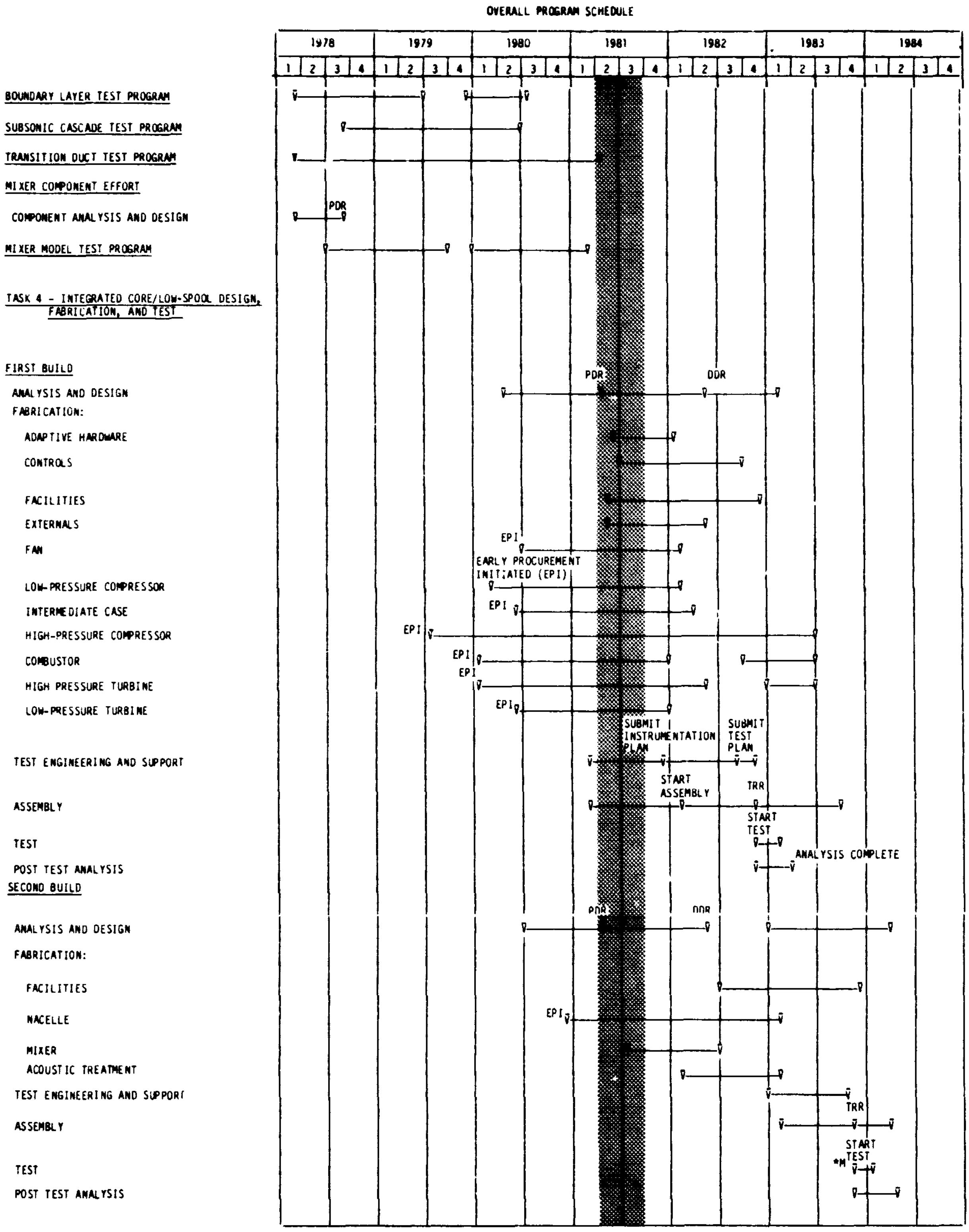

Figure 1 Overall Program Logic Diagram (continued)

2.0 HIGHLIGHTS OF WORK ACCOMPLISHED

- Updated analysis of the flight propulsion system incorporating secondary airflow changes and rematching impacts on turbine operating conditions have resulted in high- and low-pressure turbine efficiency improvements to 89.1 percent and 91.6 percent, respectively. These turbine efficiency improvements contribute to an overall 15 percent reduction in thrust specific fuel consumption for the flight propulsion system relative to the JT9D-7A reference engine.

- Groundrules for conducting flight and economic performance evaluations for the flight propulsion system update were prepared and submitted to NASA for review and approval. NASA subsequently approved these updated groundrules pending inclusion of additional fuel pricing data.

- NASA requested that Pratt & Whitney Aircraft reschedule the second preliminary analysis and design update for the flight propulsion system to the spring of 1982. This schedule allows updated flight propulsion system flight and economic performance evaluations to coincide with completion of the integrated core/low spool Detail Design Review.

- Final manufacturing operations conducted at TRW on prototype hollow, shroudless fan blades were completed early in the report period. However, isothermal forging of these laminated titanium blades produced mixed results due to thermal distortion of the forging dies. An aerodynamically-contoured, finished-form blade was sent by TRW to Pratt & Whitney Aircraft for evaluation.

- Assembly effort for the high-pressure compressor rig was completed and initial rig testing begun early in the reporting period. Unacceptably high stress indications were noted in the rig rear thrust balance piston. Testing was subsequently interrupted to allow for the design, fabrication, and reinstallation of a revised thrust balance piston. Testing then resumed and the thrust balance piston showed no further activity of concern. Some performance data were acquired before failure of the slip ring drive mechanism forced a second interruption. Revised drive hardware is currently being fabricated and test data are being analyzed.

- The instrumentation and test plan for the full annular combustor rig was completed during the report period and submitted to NASA for review and approval. The fabrication effort directed toward the inner combustor case, advanced segmented liners and liner supports for the combustor component were completed.

- Fabrication of the high-pressure turbine MERL 76 disk compaction, for use in the component rig, progressed into finish machining (lathe turning) during the report period. Approximately 90 percent of all component rig hardware items have been successfully fabricated with completed parts being fit-checked preparatory to the rig assembly effort initiated late in the report period. Component rig testing is scheduled to commence in early-1982.

- By mutual agreeement between Pratt & Whitney Aircraft and NASA, high-pressure turbine vane castings for integrated core/low spool (build 1) will be poured using PWA 1422 directionally-solidified material instead of PWA 1480 single crystal alloy. Material substitution was necessitated by a slippage in the casting vendor's schedule to finalize the PWA 1480 vane casting process. This approach prevents any schedule slippage in testing the integrated core/low spool. However, effort by the casting vendor to provide a minimum of three acceptable PWA 1480 vanes for the first integrated core/low spool build is continuing on an expedited basis.

- NASA approval of the Integrated Core/Low Spool Preliminary Design Review was received during the report period. By mutual agreement between Pratt & Whitney Aircraft and NASA, the Detail Design Review for the integrated core/low spool has been rescheduled to May 1982.

- An update to the integrated core/low spool status performance level using expected turbine efficiencies (previously mentioned for the flight propulsion system), secondary system flow levels, and refined exhaust system pressure losses resulted in a forecasted 10 percent benefit in thrust specific fuel consumption relative to the JT9D-7A reference engine.

- NASA Contract NAS3-20646 was modified during the reporting period with the major effects of this modification being (1) deletion of the Task 3 effort and the addition of a second test to the Task 4 integrated core/low spool effort, (2) redefinition of the fan effort to replace the shroudless fan with a shrouded design, (3) addition of a third test to the high-pressure compressor rig program, and (4) addition of a benefit/cost study of potential fuel saving technologies in order to identify those suitable for follow-on technology development programs.

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 sub-task 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 sub-tasks: (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. The logic diagram for Task 1 is shown in Figure 2, and the work plan schedule, in Figure 3.

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.

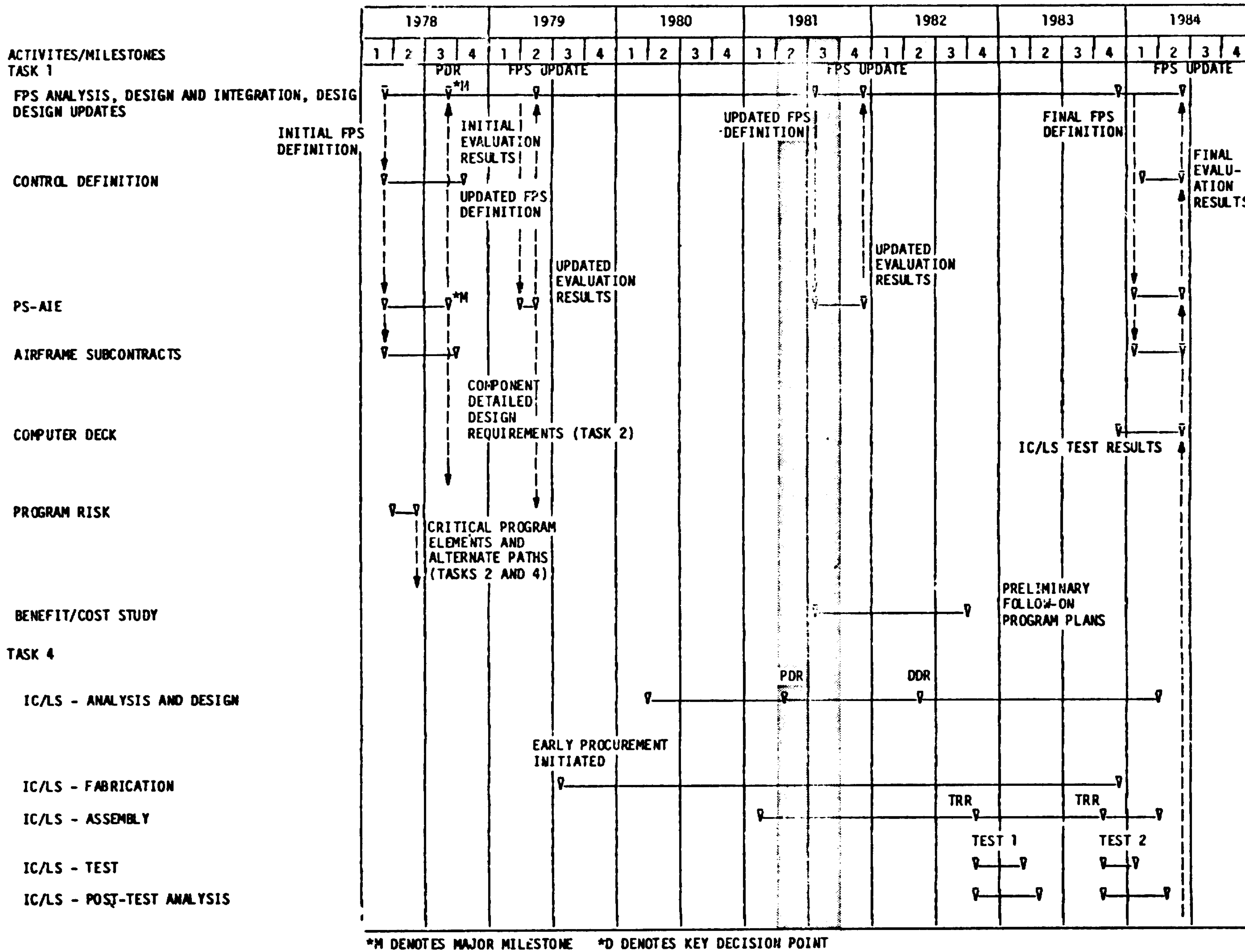

Figure 2 Task 1 Logic Diagram

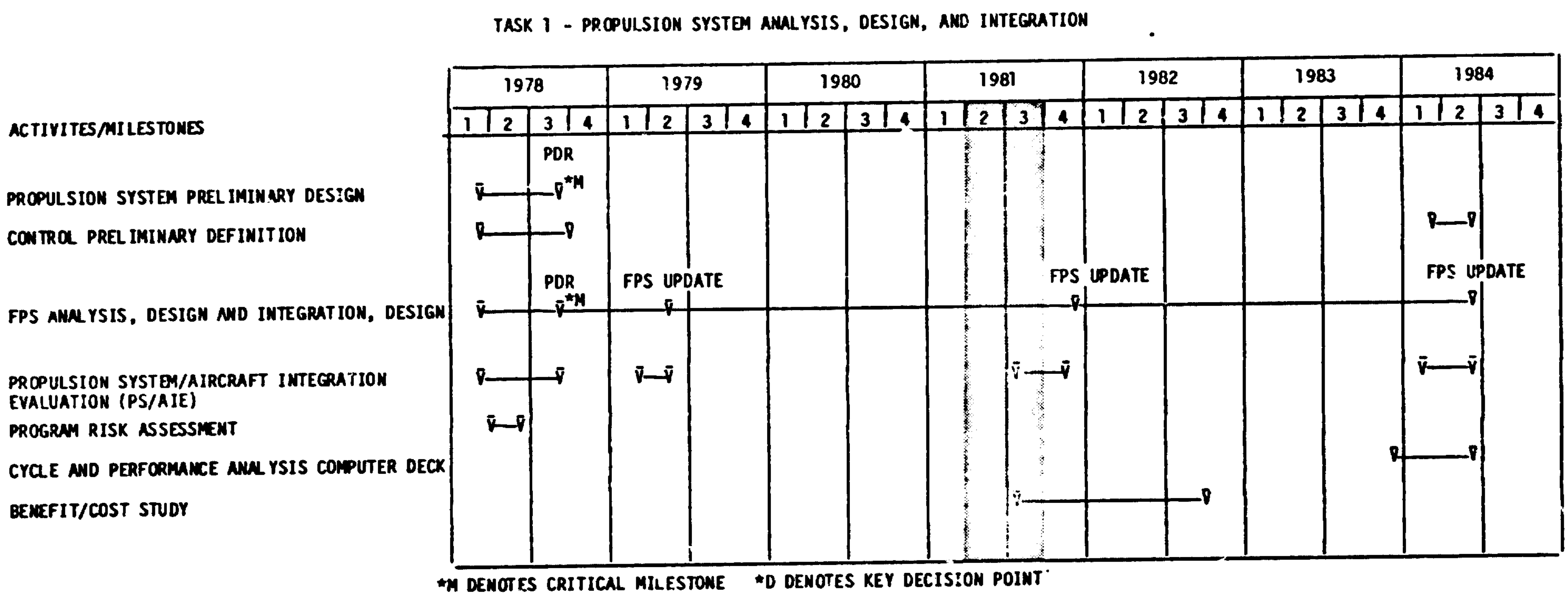

Figure 3 Task 1 Work Plan Schedule

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All of the work planned and approved from contract award through the end of the current reporting period (30 September 1981) 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 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.

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 aft shroud to provide efficiency improvement. The low-pressure compressor utilizes controlled endwall loss and reduced airfoil loss concepts to raise compressor efficiency 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 counter-rotates 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.

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

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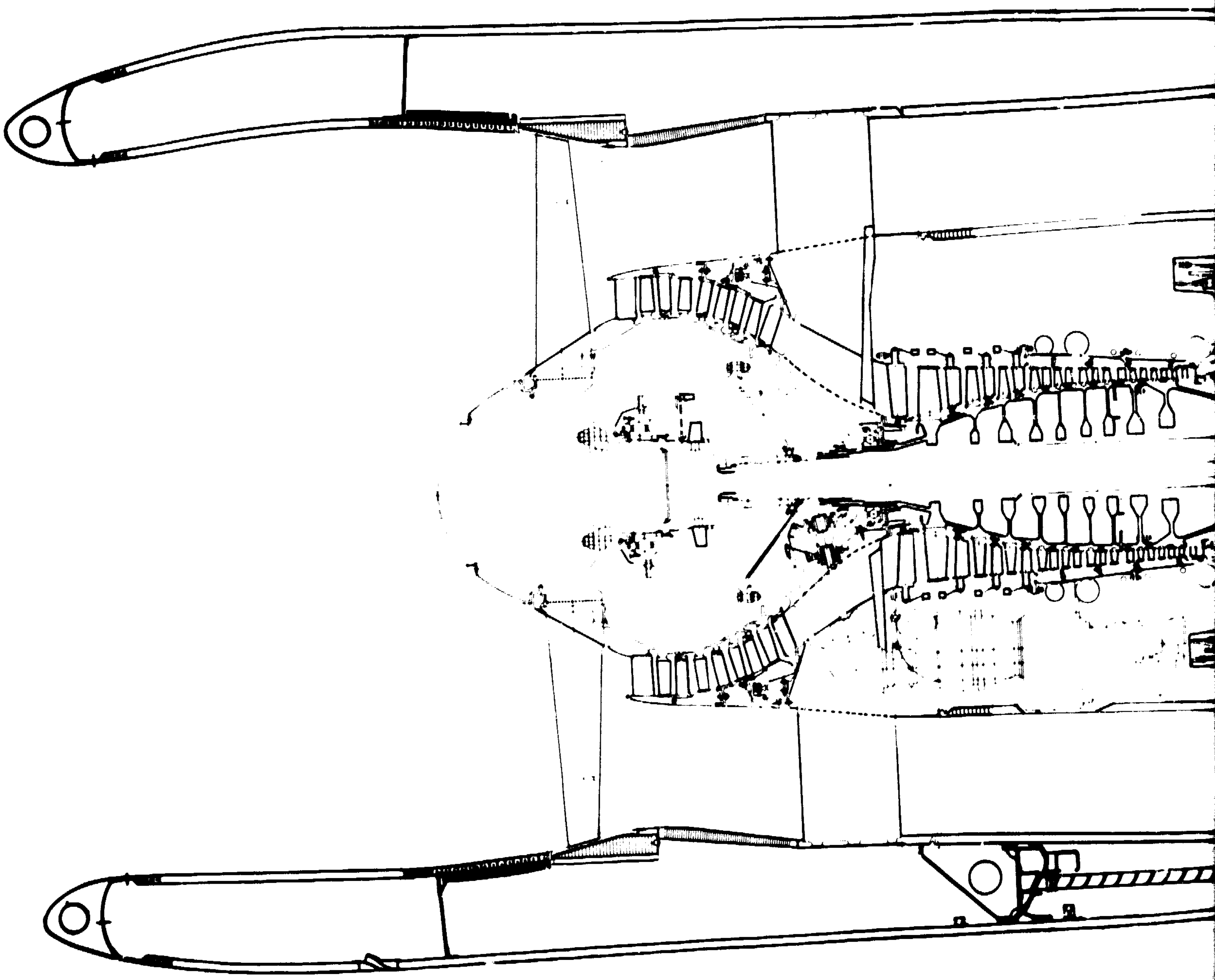

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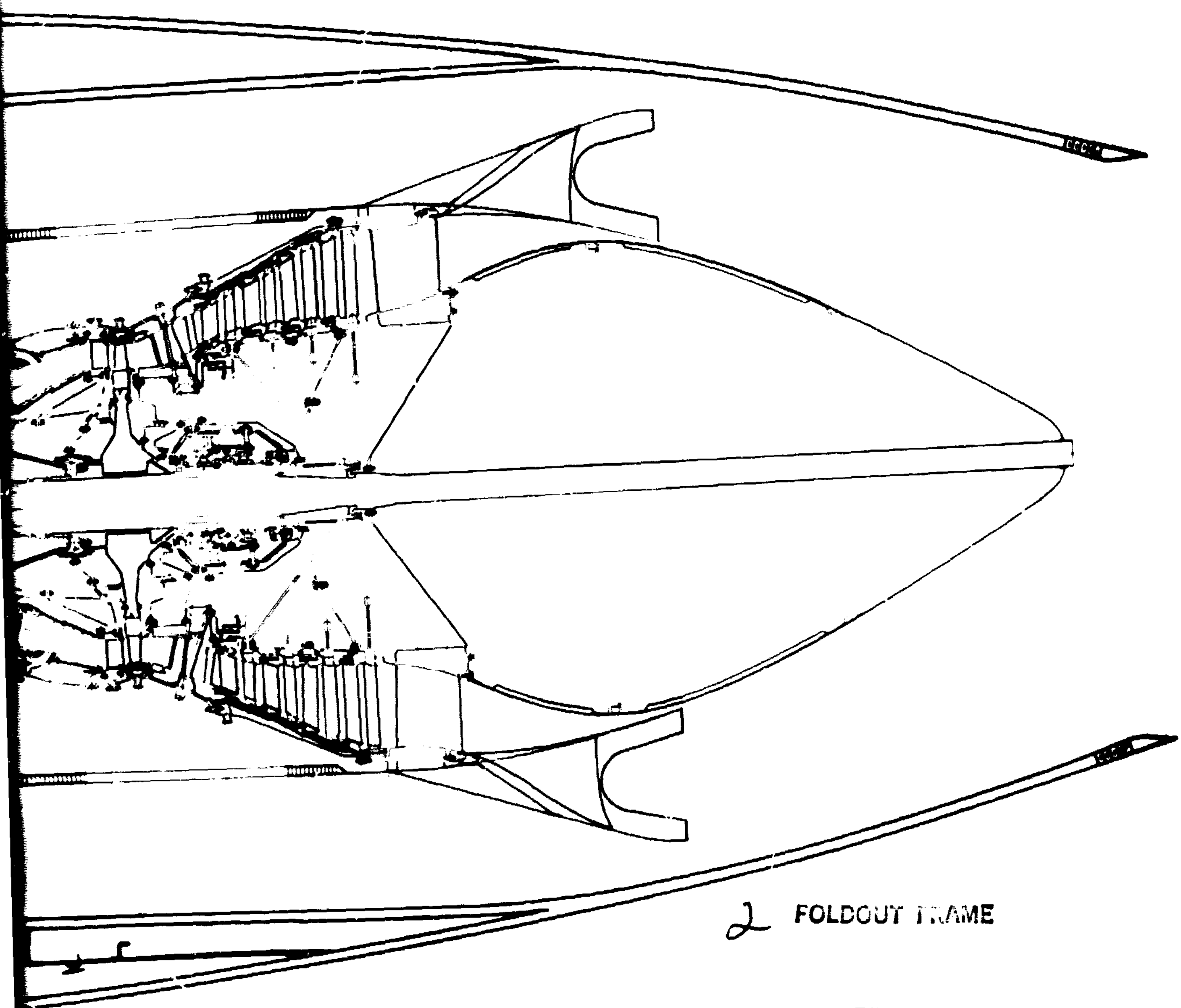

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Figure 4 Energy Efficient Engine Cross Section

Figure 3 identifies those tasks completed during the previous reporting periods and indicates that work on sub-Task 3 was continued during the current reporting period. It also indicates that work on sub-Tasks 4 and 7 was scheduled to begin late in the current reporting period. Work for sub-Tasks 3, 4 and 7 is described in detail in the following sections.

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 reoptimization 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 definitions of the propulsion system and its components have been periodically updated as program technical objectives have been met. The evolutionary status of the system performance for the flight propulsion system design, compared to the NASA goals and the JT9D-7A reference engine, is shown in Table 1. Results from the propulsion system/aircraft integration evaluations, where these were conducted, are also included in this table. Table 2 lists the current flight propulsion system performance parameters at significant engine operating conditions.

TABLE 1

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 2

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.

TABLE 3

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 (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

The material selection was updated for both the flight propulsion system and the integrated core/low spool. These materials are listed in Table 4 along with an '*' symbol indicating 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 for reference is presented in Table 5.

Performance Parameters and Detailed Drawings

The secondary airflow system was refined to incorporate provisions necessary to achieve the required high and low pressure rotor thrust balance and component case cooling as defined at the completion of the low pressure spool component detailed designs. Figure 5 is the new secondary airflow and pressure map. System revisions include refinement of controlling areas in the high- and low-pressure turbine and rear bearing compartment regions; refinement of clearances of the number 3 bearing compartment intershaft seal, number 4 1/2 bearing compartment buffer seal, and the low-pressure turbine front thrust balance seal; extension of the center vent pipe to the exit of the exhaust nozzle; and changing low-pressure turbine active clearance control bleed air source from the 10th stage of the high-pressure compressor to the 8th stage.

The flight propulsion system cross section was also updated during this reporting period. Major modifications to the previous cross section include (1) incorporation of a shrouded, solid fan blade replacing the shroudless, hollow fan blade, and (2) details resulting from the design of the low pressure spool components. The new cross section is presented in Figure 6.

Performance parameters (including sea level static takeoff, maximum climb, and maximum cruise conditions) were updated to account for changes occurring because of completion of the low pressure spool detail design and refinement of the secondary airflow system.

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

TABLE 4

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

TABLE 4 (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 4 (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	FPS 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	

TABLE 4 (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 4 (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 5

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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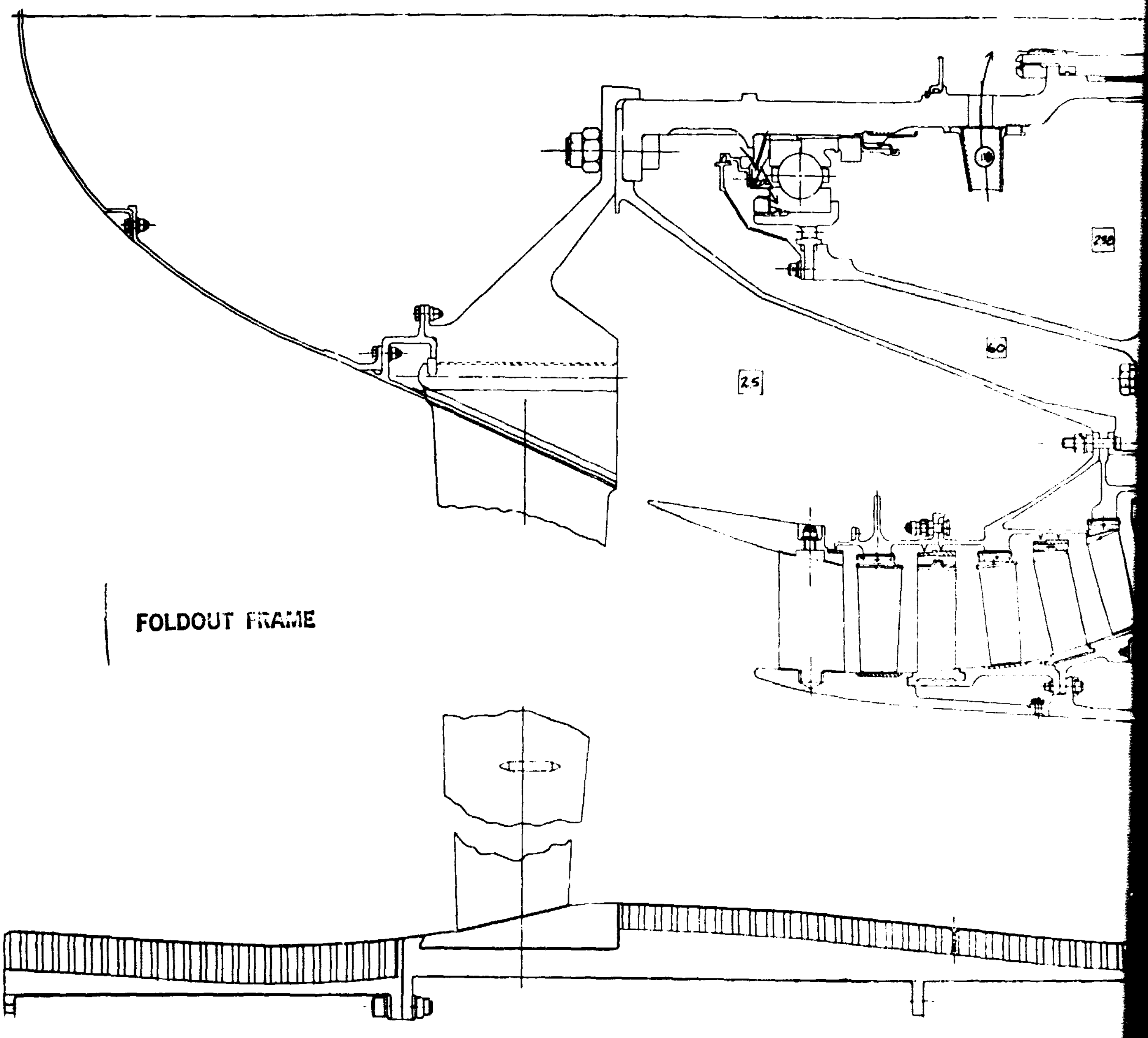

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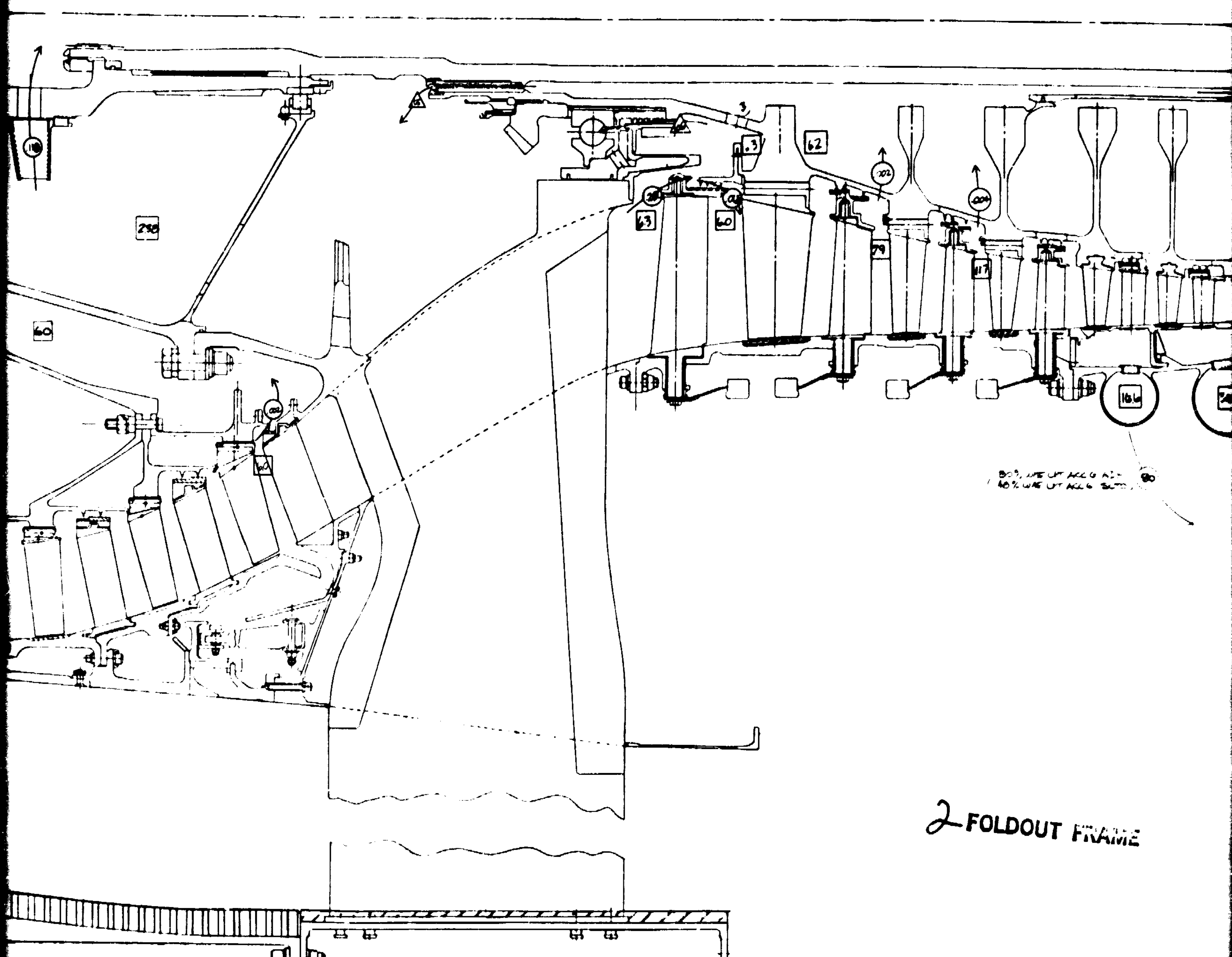

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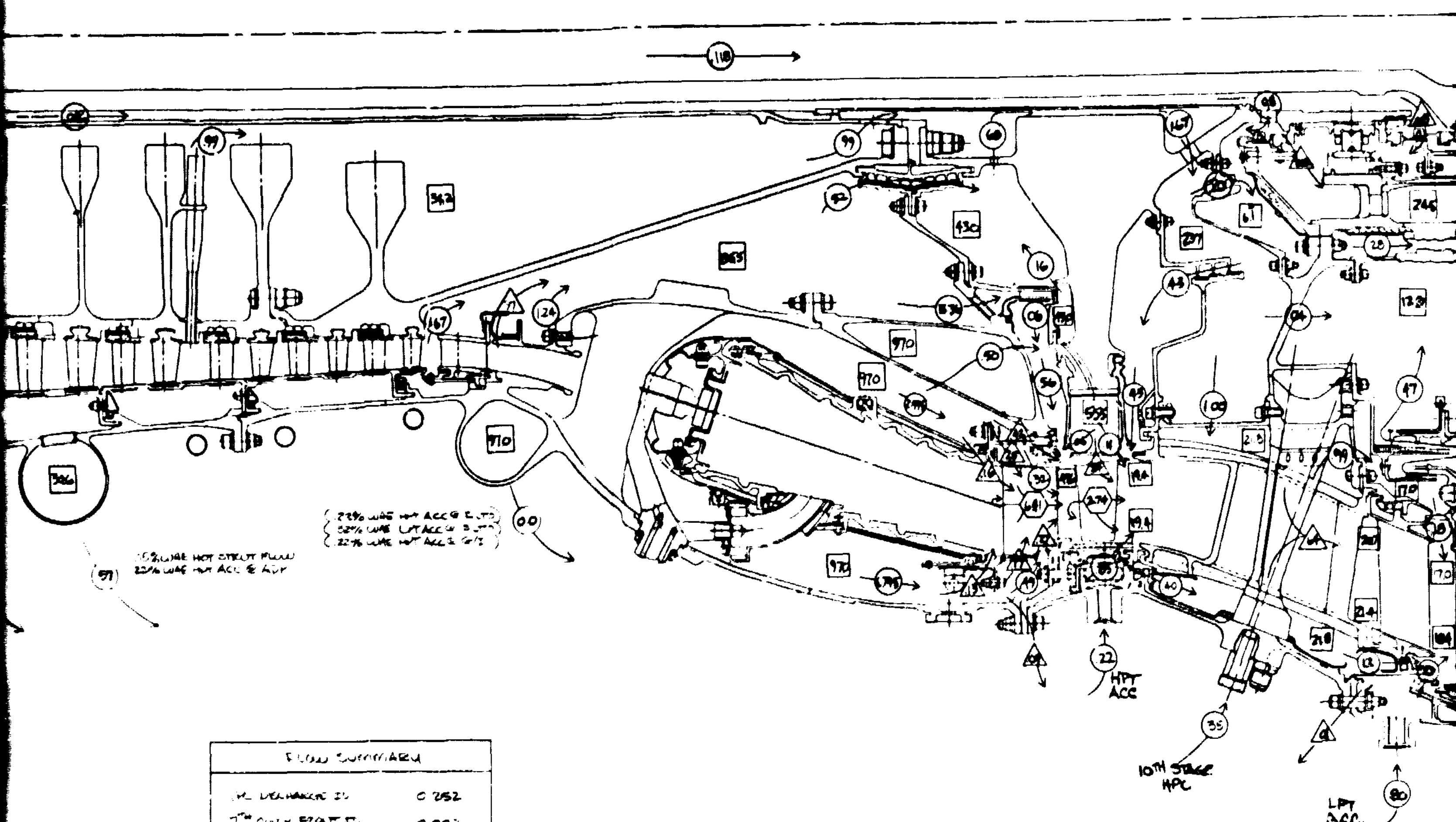

FLOW SUMMARY	
HPC DISCHARGE ID	0.252
7TH COMP FRONT ID	0.002
5TH COMP FRONT ID	0.004
8TH COMP REAR OD	0.800
10TH COMP REAR OD	0.570
7TH COMP REAR ID	0.910
5TH COMP REAR ID	1.880
BURNER SECONDARY ID	3.245
BURNER [illegible] OD	6.745
Σ	6.738 %WAE
FAN DISCHARGE OD	0.500
TOTAL COOLING AIR	7.238 %WAE

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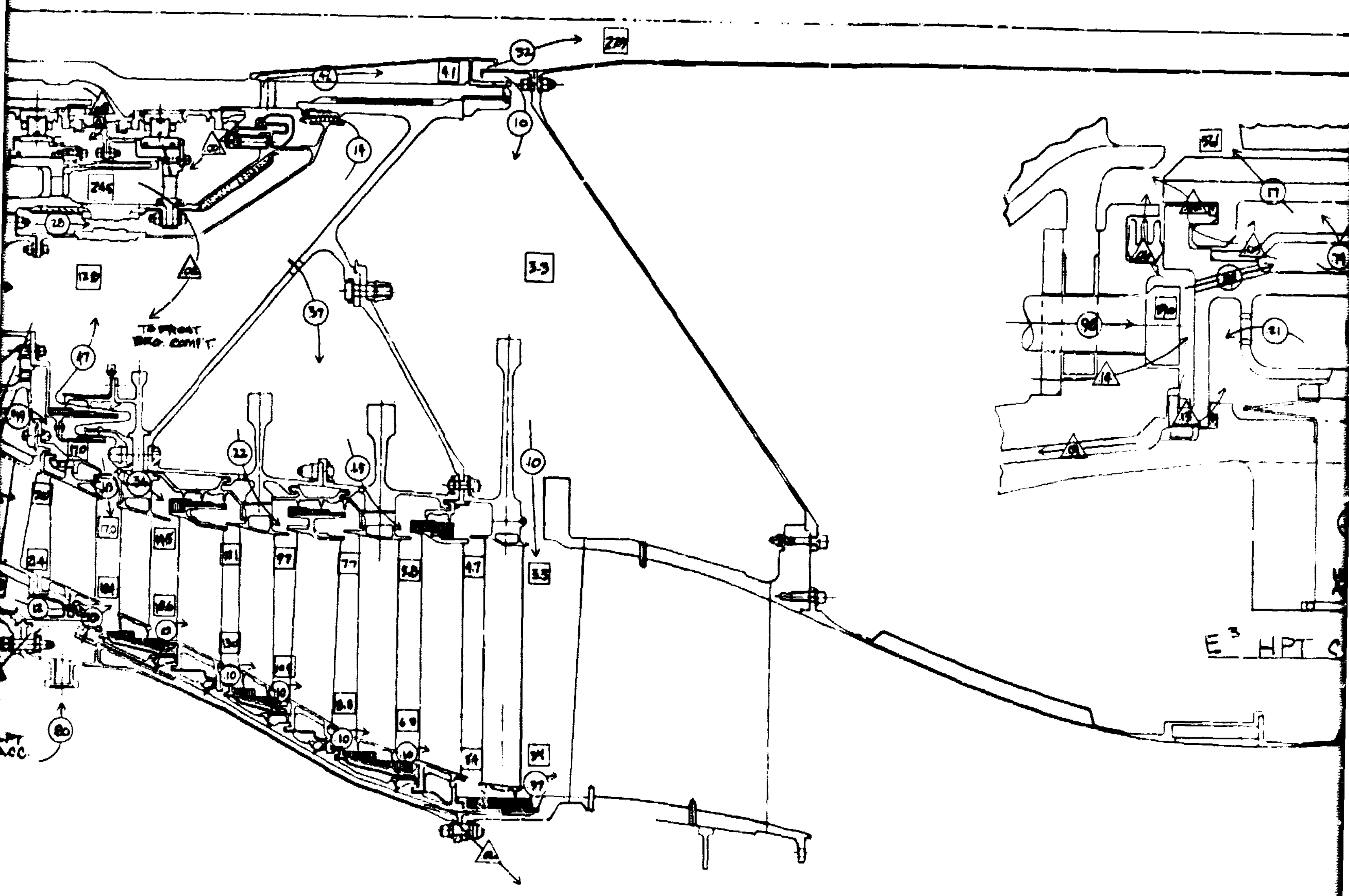

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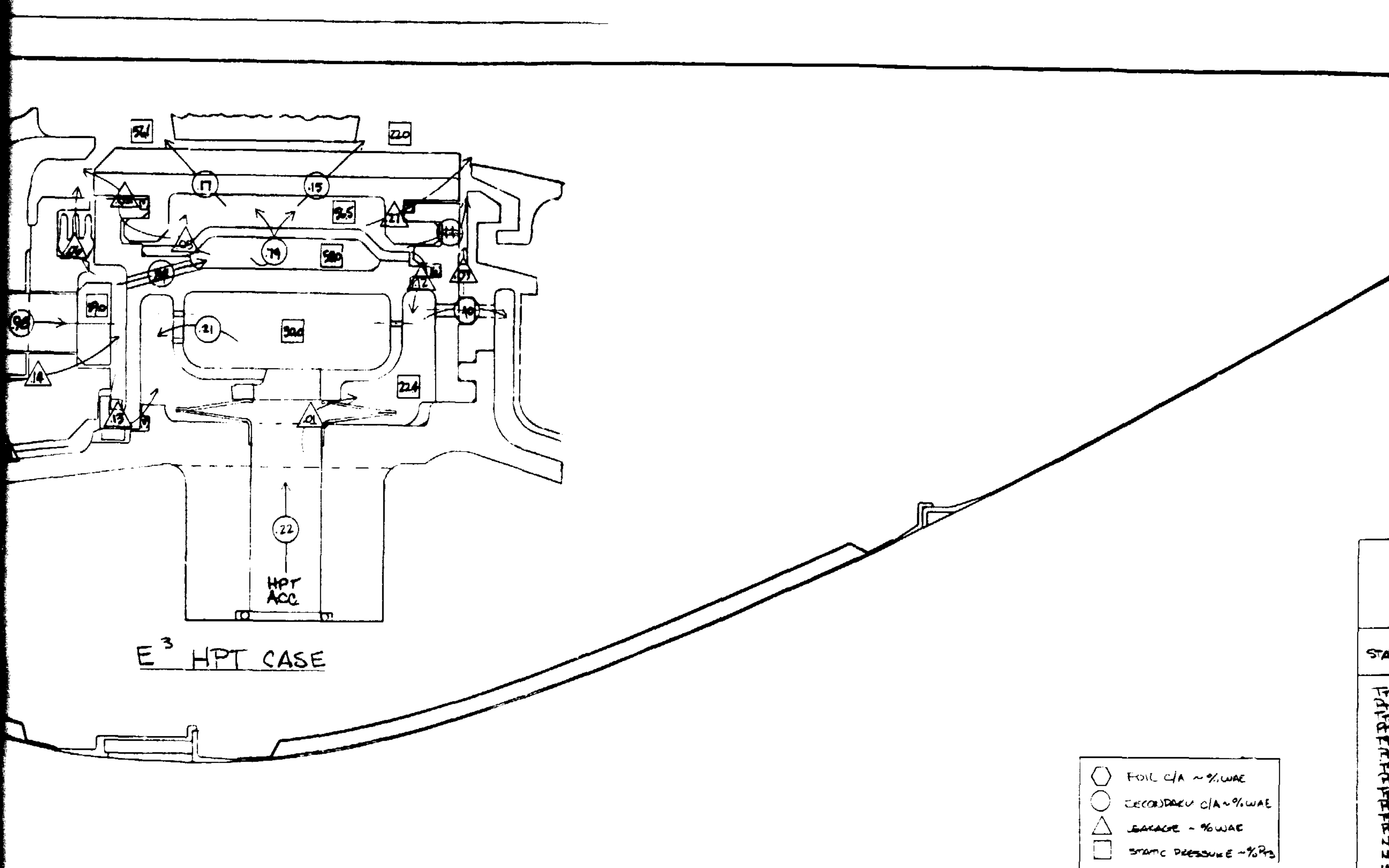

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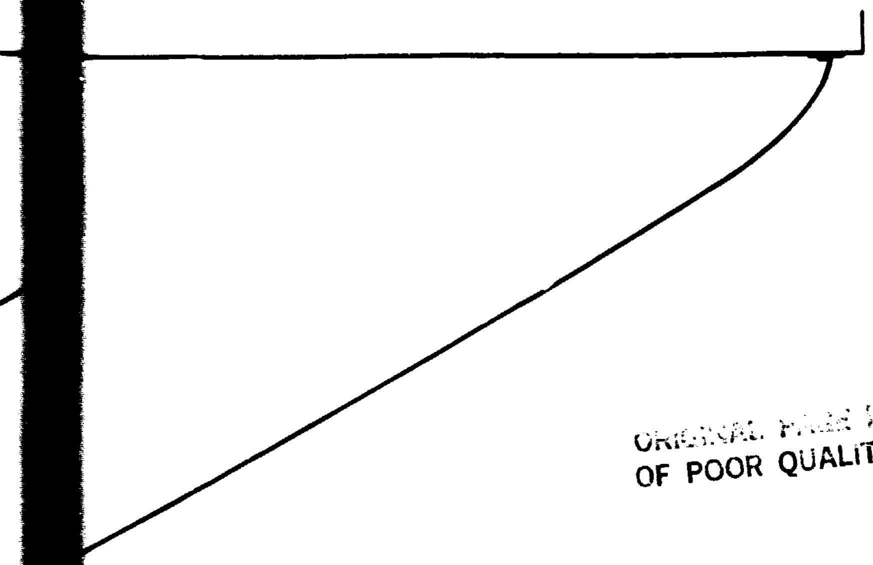
Figure 5 New Flight Propul
and Pressure Map



E^3 PERFORMANCE DESIGN TABLE # 2108		
STATION	+25°F SLTO P # 4 (½ DET)	A D P P # 1 (NO DET)
TT2 (°F)	84	-16
PT2 (PSIA)	14.7	5.27
TT2.5	250	152
PT2.5	356	16.5
TT3	1077	899
PT3	461	203
TT4	2659	2352
PT4	435	192
TT45	1822	1540
PT45	107	47.6
TT49	1162	895
PT49	21.5	8.38
N1 (RPM)	3923	3902
N2	14045	13178
WAE (#/S)	146	70.5

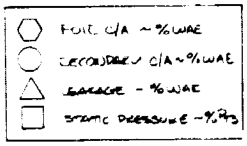


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Figure 5 New Flight Propulsion System Secondary Airflow and Pressure Map

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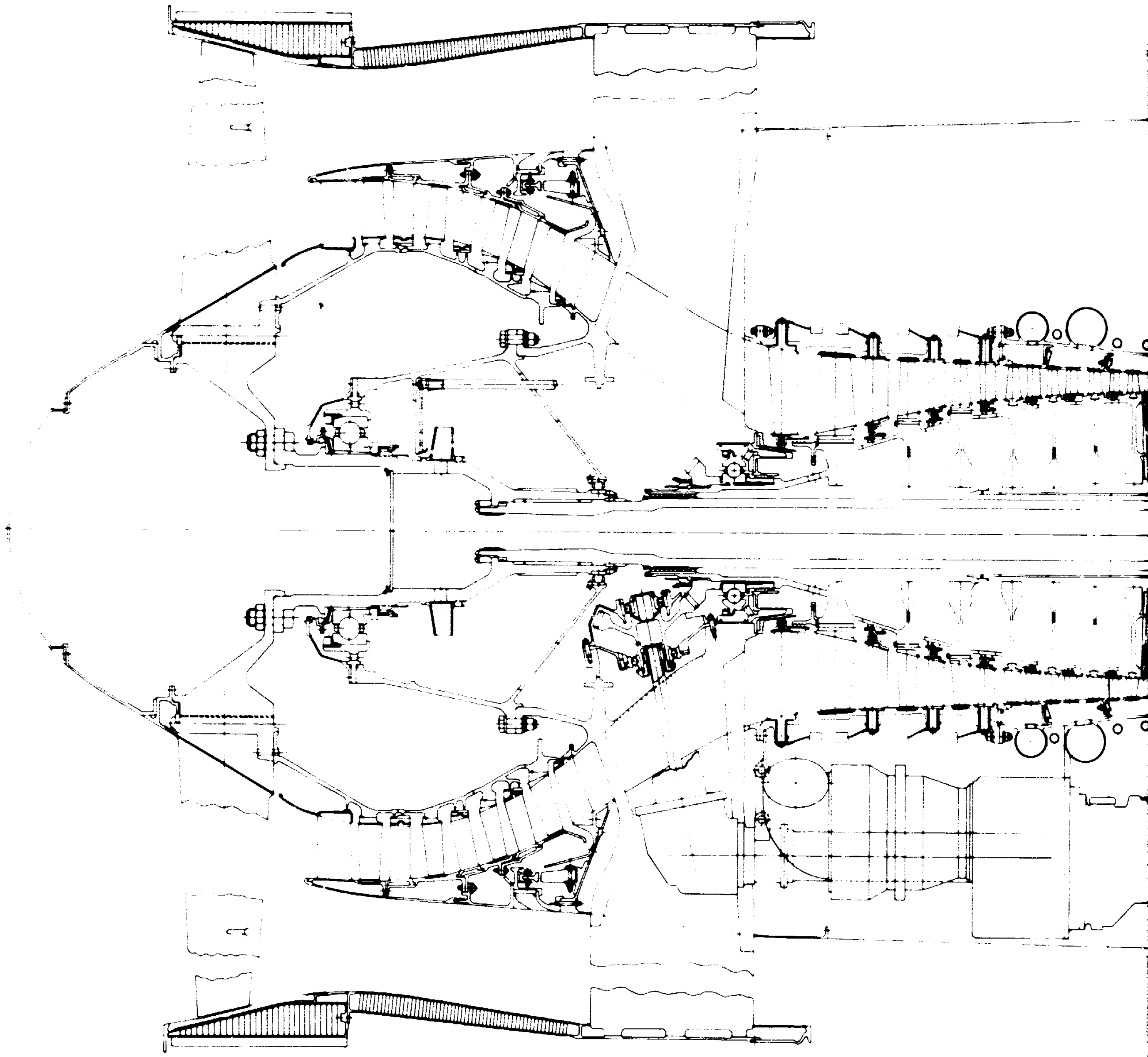
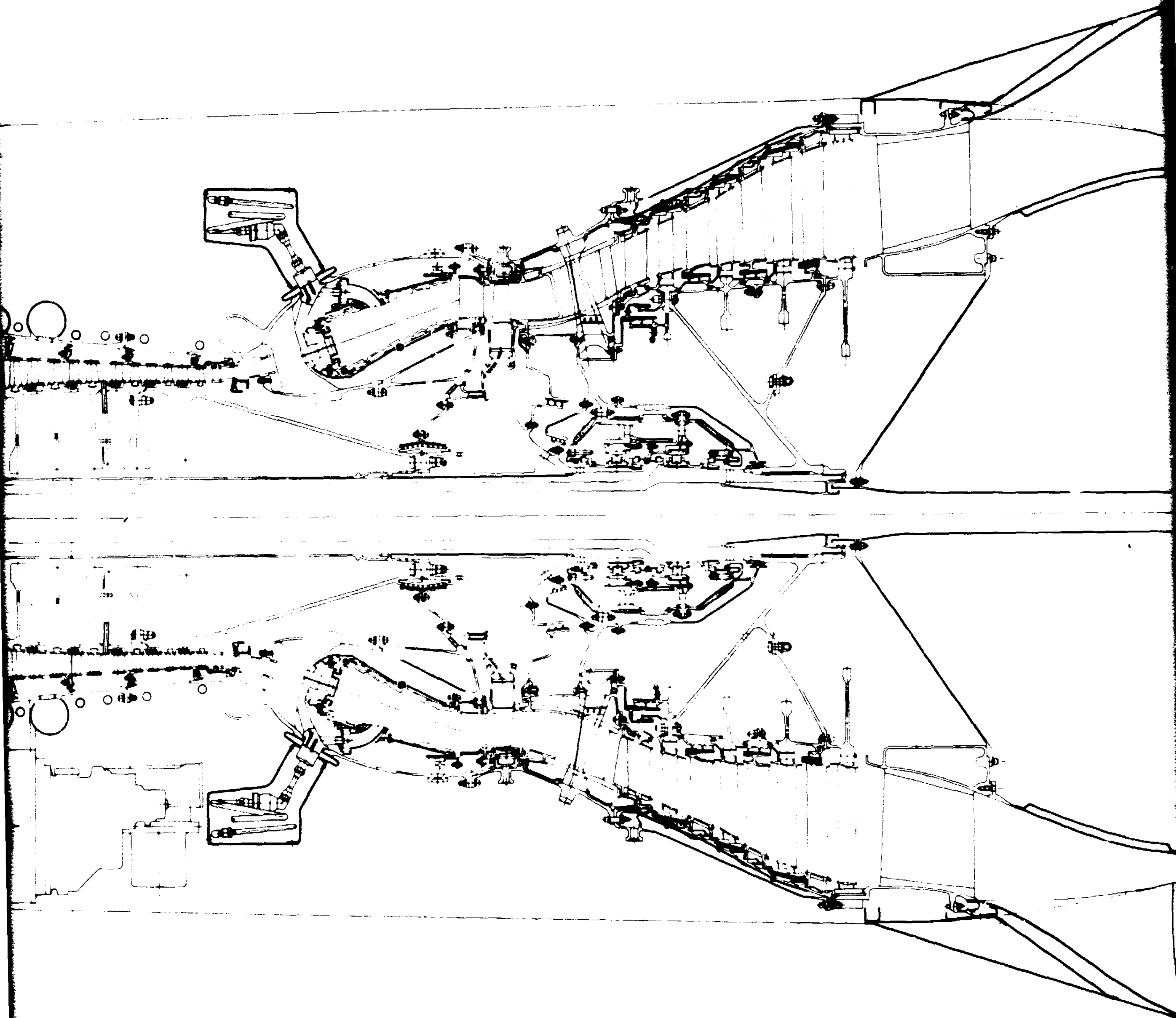


Figure 6 Updated Flight Propulsion System Cross Section

PRATT &
COMMERC



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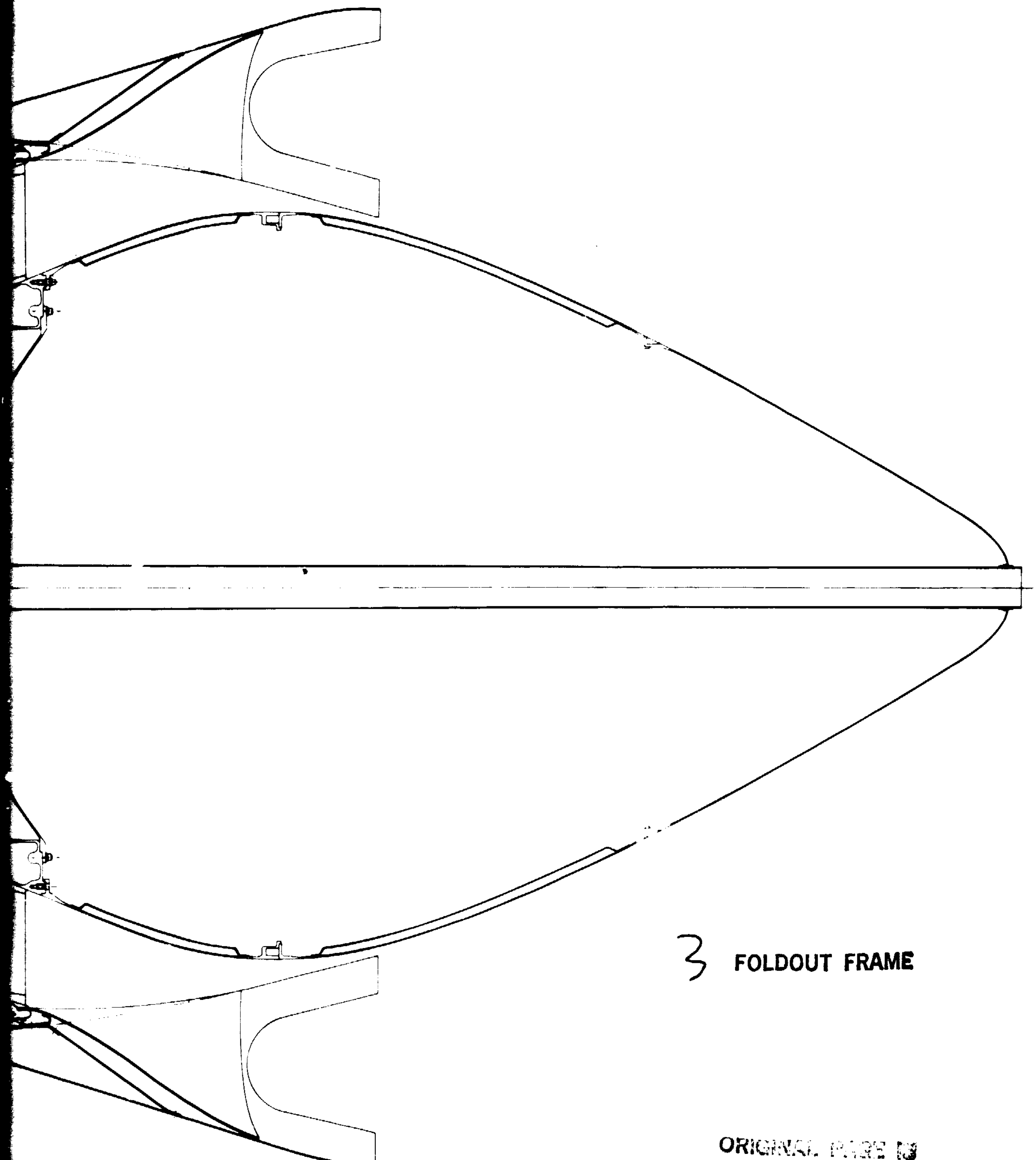

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System-Related Activities

All of the work planned for system-related activities for this reporting period was completed. Table 6 presents a summary of the work accomplished, including updates to the design of the flight propulsion system, integrated core/low spool design support, economic and fuel burn analyses support, and NASA requests. Results and conclusions are summarized in the following paragraphs.

Flight Propulsion System Performance Update: An update to the flight propulsion system performance was conducted to reflect design changes since March 1980. Thrust specific fuel consumption status was determined using the computerized flight propulsion system performance simulation. Component and system changes accounted for since the March 1980 status analysis resulted from (1) completed detailed designs of the low pressure spool components, (2) the completion of testing on the turbine transition duct, and (3) a refinement of the secondary airflow system. No change was required as a result of Phase II exhaust mixer model testing. Included in these component and system changes were:

- a change to a shrouded fan configuration and fan map refinement;
- improvements in the tip clearances of the fan and low-pressure compressor;
- improvement in the turbine transition duct pressure loss;
- high-pressure turbine airfoil cooling air and secondary airflow system improvements;
- a change in high-pressure compressor bleed source for low-pressure turbine active clearance control;
- an increase in low-pressure turbine active clearance control air quantity;
- a reduction in low-pressure turbine rotor cooling/leakage air;
- a reduction of low-pressure turbine inlet annulus area;
- an increase in low-pressure turbine tip clearance;
- reassessment of technology benefits for the low-pressure turbine.

TABLE 6

SUMMARY OF ANALYSIS AND DESIGN UPDATING EFFORT FOR THE CURRENT REPORTING PERIOD

EFFORT	REQUIREMENT: FPS DESIGN REFINEMENT	COMPONENT/IC/LS DESIGN SUPPORT	BENEFIT EVALUATION	NASA REQUEST
FPS Performance Update	X	X	X	
FPS Direct Operating Cost Update			X	
FPS Fan/LPC Flowpath Revision	X	X	X	
Second Analysis/Design Update	X	X	X	X
FPS Cross Section Update	X	X	X	
Materials Update	X	X	X	
IC/LS Bifurcated Duct Configuration Design Requirements		X		
IC/LS Secondary Airflow System Map		X	X	
IC/LS Preliminary Design Action Items		X	X	X
IC/LS Performance Update		X	X	
IC/LS Transient Analysis		X		

Factors contributing toward the predicted status flight propulsion system maximum cruise thrust specific fuel consumption are shown in Table 7. The itemized component and system changes occurring since March 1980 resulted in a net thrust specific fuel consumption penalty of 0.1 percent. The new predicted level represents a 15.0 percent advantage for the flight propulsion system compared to the JT9D-7A reference engine.

TABLE 7

FLIGHT PROPULSION SYSTEM
THRUST SPECIFIC FUEL CONSUMPTION STATUS SUMMARY (June 1981)
Maximum Cruise (35000 ft, 0.8Mn, Standard Day), JT9D-7A Reference Engine

Component/System Change	TSFC Change (%)
-0.8% Δ Fan O.D. Efficiency (Incorporation of Shroud, 0.081 to 0.055 in. Tip Clearance, New Map)	+0.40
+0.1% Δ LPC Efficiency (0.021 to 0.019 in. Average Tip Clearance)	-0.01
+0.4% Δ HPT Efficiency (Vane Cooling Air Reduction, Blade Cooling Air Revision, Secondary Airflow Reduction, Rematching	-0.24
-0.8% Δ Turbine Transition Pressure Loss (Testing)	-0.26
+0.1% Δ LPT Efficiency (Aero. Reassessment, 0.019 to 0.035 in. Average Tip Clearance, Secondary/Active Clearance Control Airflow Changes, -5% Inlet Area	+0.05
Secondary Airflow System/Turbine Cooling Air Refinement (+0.3% Core Airflow)	+0.20
Total	+0.14
3/80 Status	-15.1
6/81 Status	-15.0

A breakdown of the major elements contributing to the June 1981 thrust specific fuel consumption advantage of 15.0 percent was defined. These elements and their contributions are presented in Table 8.

TABLE 8

FLIGHT PROPULSION SYSTEM THRUST SPECIFIC FUEL CONSUMPTION IMPROVEMENT STATUS - JUNE 1981
Maximum Cruise (35000 ft., 0.8 Mn, Standard Day), JT9D-7A Reference Engine

Contributor		TSFC Change (%)
Low Pressure Spool		-5.8
High Pressure Spool		-3.8
Cycle		-3.2
Mixing/Installation		-2.2
	Total	-15.0

Flight Propulsion System Direct Operating Cost Update: The effect of the June 1981 maximum cruise thrust specific fuel consumption prediction on direct operating cost for the study airplanes and missions was estimated. (Weight and costs were not updated at the time, so their effects on direct operating cost are unchanged from those of October 1979 and March 1980.) The results of this assessment showed a negligible change in direct operating cost advantage for the flight propulsion system relative to the JT9D-7A reference engine, as compared to the March 1980 status. These results are summarized in Table 9 for the average airplane/mission.

TABLE 9

FLIGHT PROPULSION SYSTEM AVERAGE DIRECT OPERATING COST STATUS UPDATE (JT9D-7A REFERENCE ENGINE)

Flight Propulsiom System Change		Direct Operating Cost Change - %
+0.1% Δ TSFC (Max. Cruise)		+0.05
Δ Weight (Not Updated)		-
Δ Cost (Not Updated)		-
Δ Maintenance Cost (Not Updated)		-
	Total	+0.05
	3/80 Status	-7.6
	6/81 Status	-7.6

Flight Propulsion System Fan/Low-Pressure Compressor Flowpath Revision: The updated (June 1981) performance simulation was incorporated into the flowpath analytical system so that a refined flight propulsion system flowpath could be defined with the shrouded fan. The blade root slope was increased to blend the 4.0 aspect ratio fan into the same low-pressure compressor. Optimization of the blending required a minor recontouring of the compressor inlet stator. An additional optimization reduced the inlet annulus area of the low-pressure turbine by 5 percent relative to the integrated core/low spool turbine design to properly match the flight propulsion system component performance levels. The resulting updated flight propulsion system flowpath is shown in Figure 7.

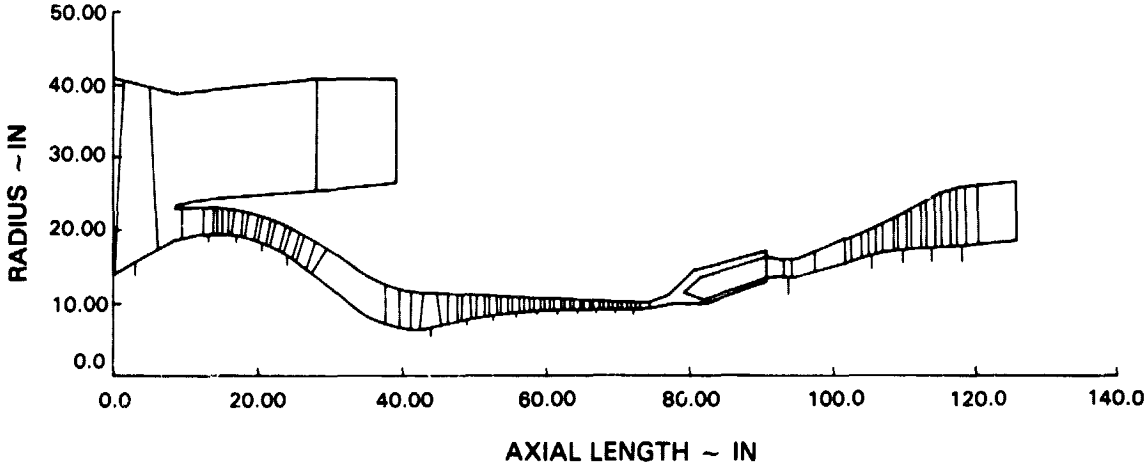


Figure 7 Updated Flight Propulsion System Flowpath

Flight Propulsion System Second Analysis/Design Update: Initial planning for the second preliminary analysis and design update effort, targeted for a December 1981 Preliminary Design Review, was completed in the reporting period. Results and recommendations have been reviewed and discussed with NASA. Their approval of work to be conducted and approaches to be taken was received orally at the end of the reporting period, along with a directive to reschedule the Preliminary Design Review until the Spring of 1982 to be consistent with the Detailed Design Review for the Integrated Core/Low Spool.

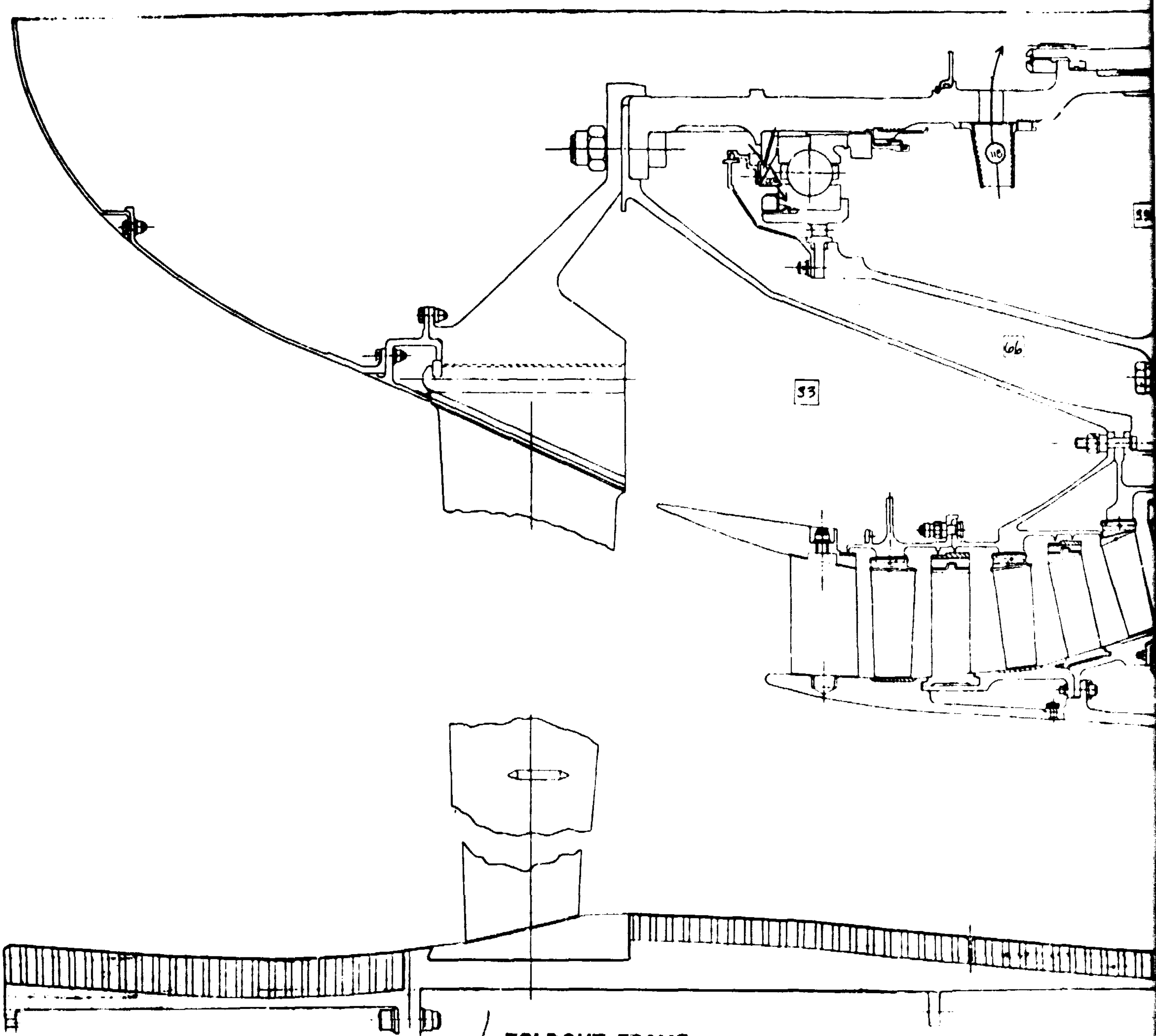
Integrated Core/Low Spool Bifurcated Duct Configuration Design Requirements: Definition of the nacelle performance design parameters for the integrated core/low spool with the bifurcated fan duct was completed during the reporting period. Compression system stability margin sensitivity to possible variations in component performance was investigated. Results showed the selected base exhaust nozzle areas, 1160 sq. in. individual fan duct exhaust nozzle area (2320 sq. in. total fan duct exhaust nozzle area) and 648 sq. in. core exhaust nozzle area, to provide adequate surge margins for all components assuming 'worst' component performance (90% probability of achievment) in combination with 1 percent flow capacity 'misses.'

Stability and performance sensitivity analyses were conducted to evaluate effects of component efficiency, pressure loss, and flow capacity 'misses' on operating optimization. A recommendation was made to have the capability to vary both duct and core nozzle areas by ± 5 percent relative to the nominals for the integrated core/low spool testing.

Analysis also indicated that maximum corrected airflow (corrected to station 13.0) entering the fan duct during testing of the integrated core/low spool will be a total of 1172.3 lb/sec. This is 586.15 lb/sec per individual bifurcation. An evaluation of aerodynamic performance in the design duct configuration will be made with this number during the design effort.

Integrated Core/Low Spool Secondary Airflow System Map: A secondary airflow system was developed for the integrated core/low spool. This analysis was done in conjunction with that for the flight propulsion system discussed in an earlier section on Performance Parameters and Detailed Drawings. The map developed is presented in Figure 8. Controlling areas and seal clearances for the integrated core/low spool and the flight propulsion system are generally the same. However, the low-pressure turbine front thrust balance seal for the integrated core/low spool is a stepped 3-1 configuration as compared to the stepped 3-2 configuration of the flight propulsion system. A tighter seal clearance (0.019 in. vs. 0.022 in.) is therefore used for the integrated core/low spool at the sea level, zero Mach number, hot day takeoff condition to achieve the same pressure drop and seal airflow.

The low- and high-pressure spool thrust balance of the integrated core/low spool was evaluated. Results showed the low rotor load to be 25 percent higher than that calculated previously. The increase to 32130 lb rearward is caused by (1) replacement of the hollow, shroudless fan blade with the solid, shrouded configuration, (2) routing oil lines through the number 4/5 bearing support which allows air to leak from the high-pressure turbine thrust balance cavity into the low-pressure turbine thrust balance cavity, (3) finalizing low-pressure turbine aerodynamics and blade running clearances, and (4) increasing low-pressure turbine front thrust balance seal radius as a result of changing the seal configuration from a 4-knife edge labyrinth to the stepped 3-1 configuration. The high rotor thrust load was calculated to be unchanged at the sea level, zero Mach number, hot day takeoff condition.



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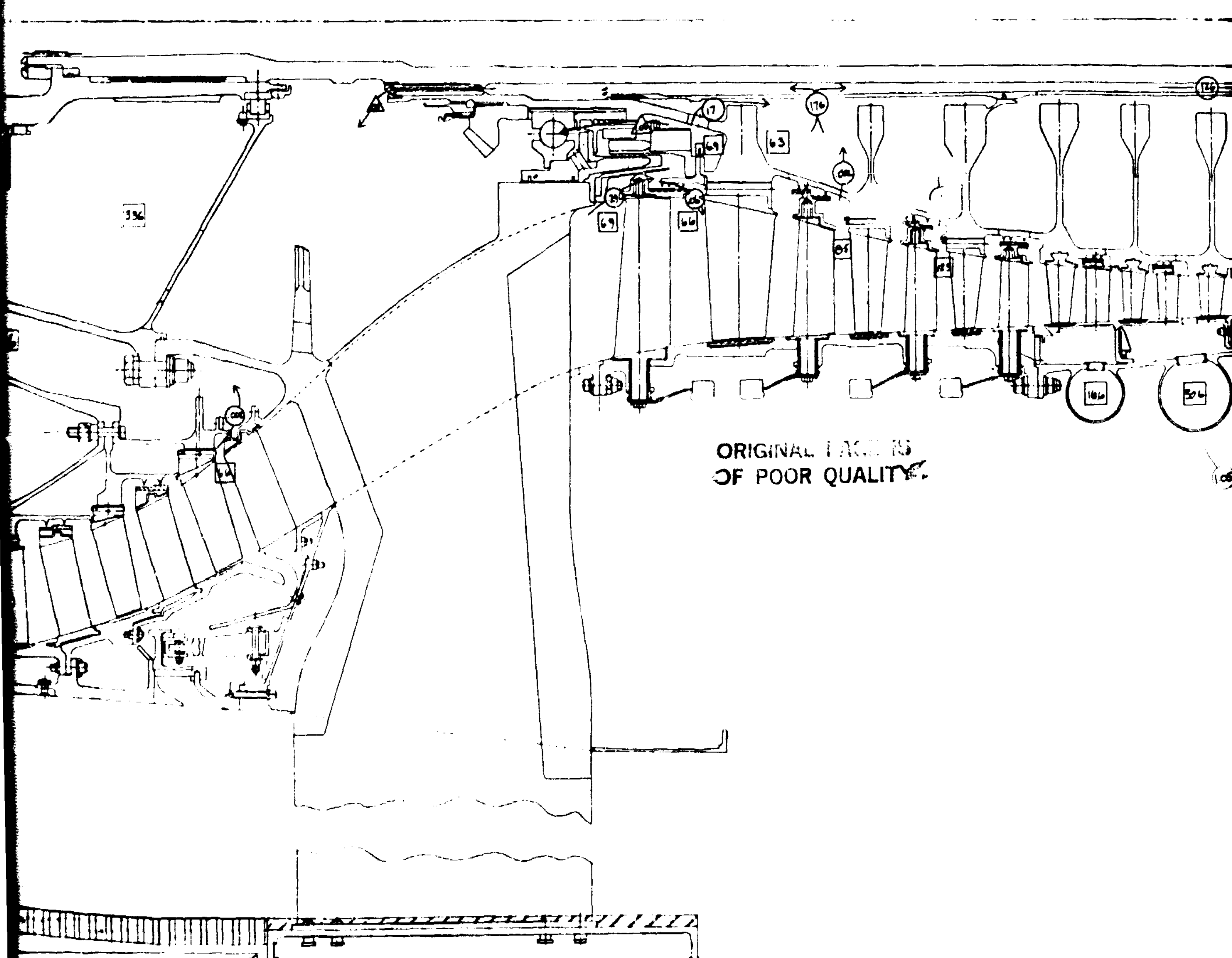

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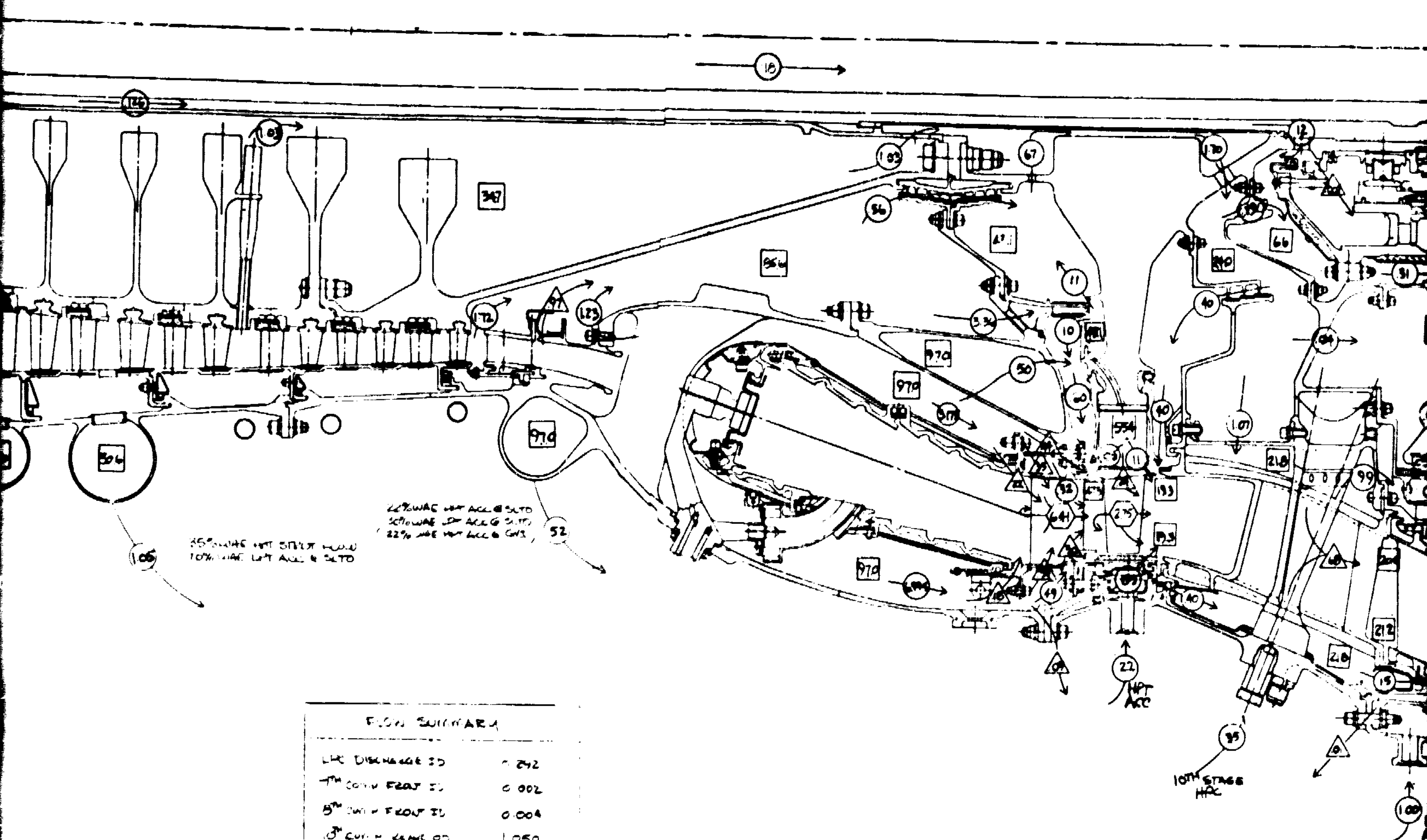

FLOW SUMMARY	
LPC DISCHARGE ID	[illegible].242
7TH COMP FRONT ID	0.002
8TH COMP FRONT ID	0.004
10TH COMP REAR OD	1.050
12TH COMP REAR ID	[illegible]
[illegible]TH COMP REAR ID	[illegible]
[illegible]TH COMP REAR OD	[illegible]
BURNER SECONDARY ID	3.675
BURNER SECONDARY OD	6.445
Σ	[illegible] %WAE
FAN DISCHARGE OD	0.500
TOTAL COOLING AIR	[illegible] %WAE

3

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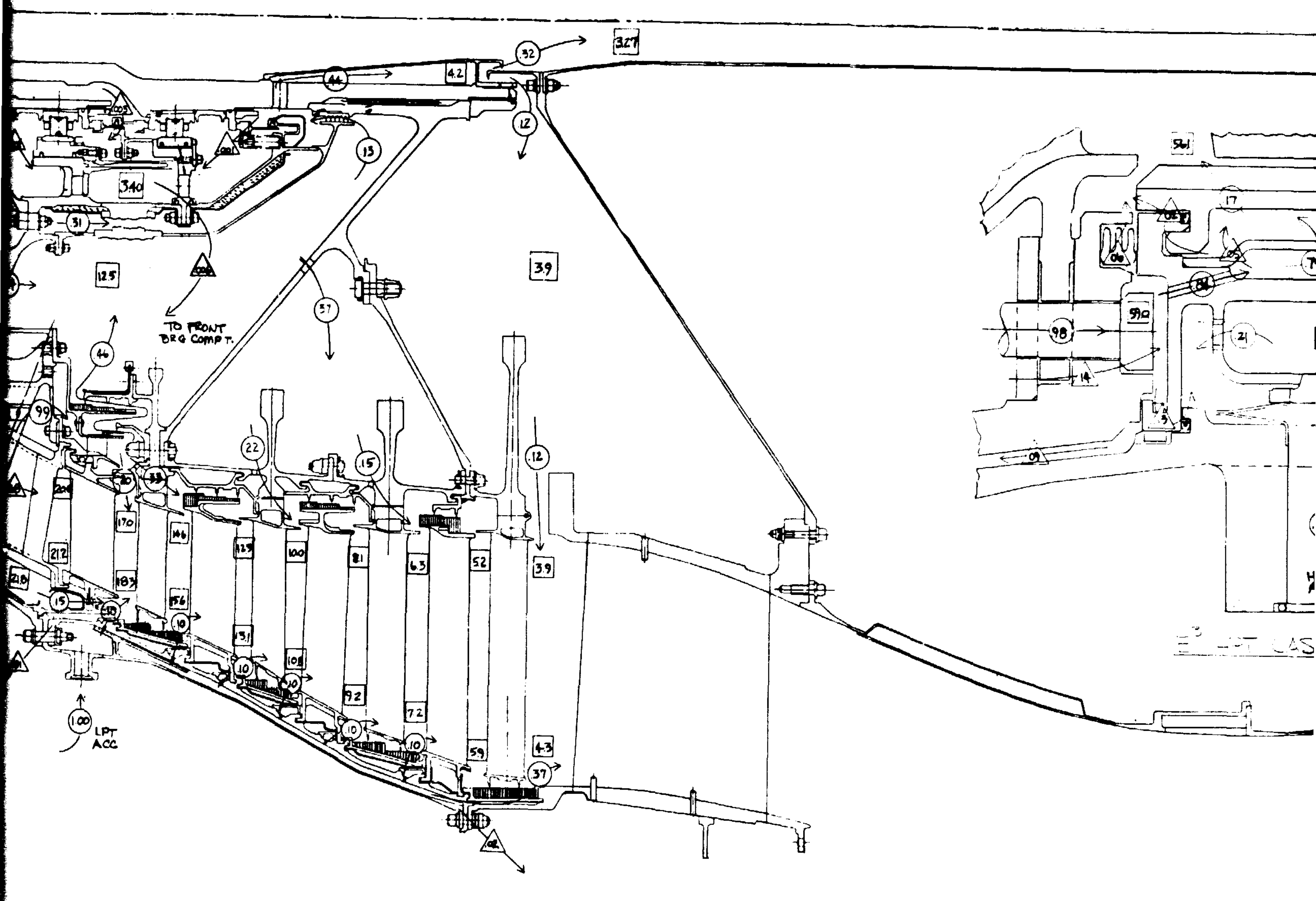

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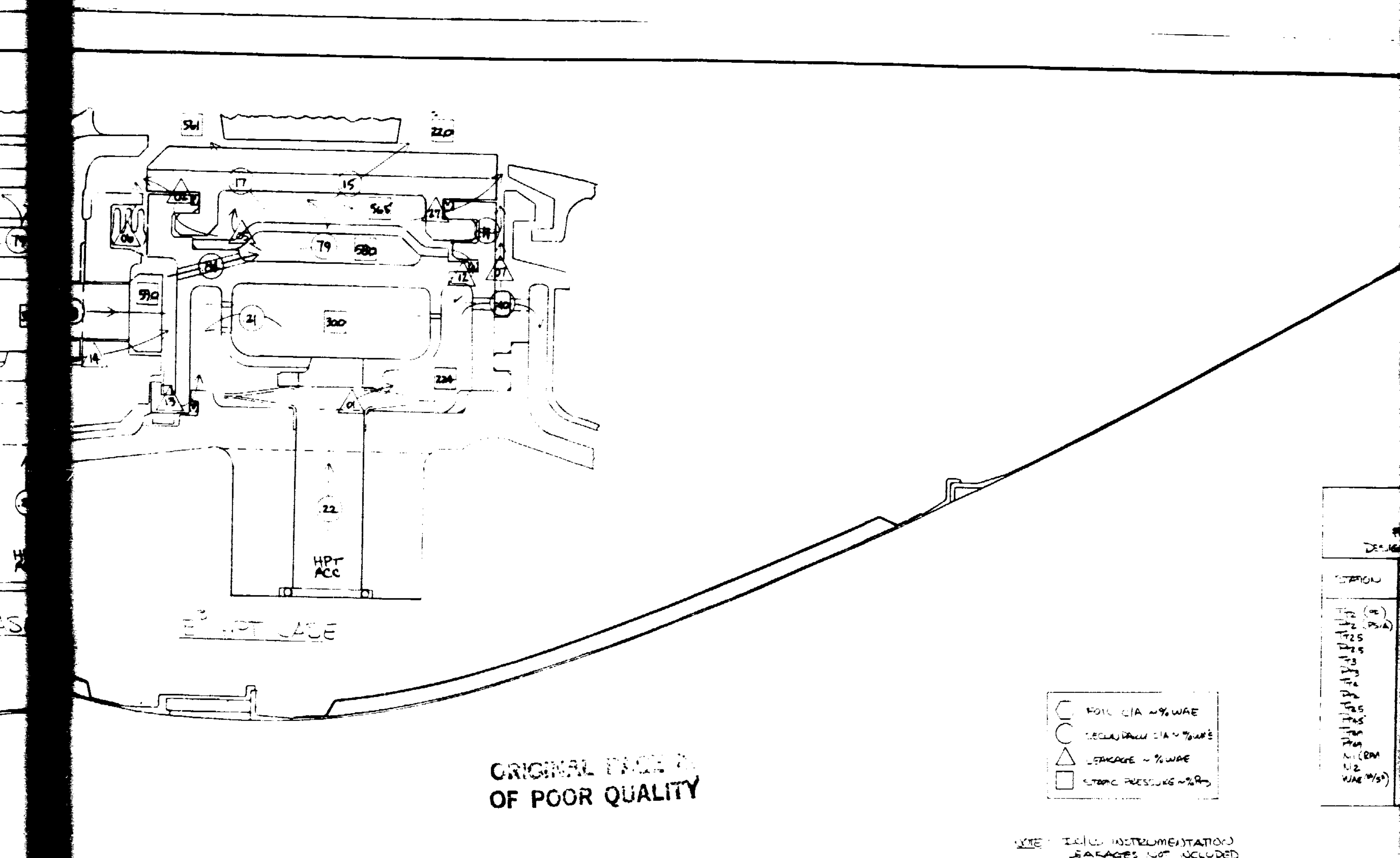

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Figure 8 Integrated Core/Low Spool S

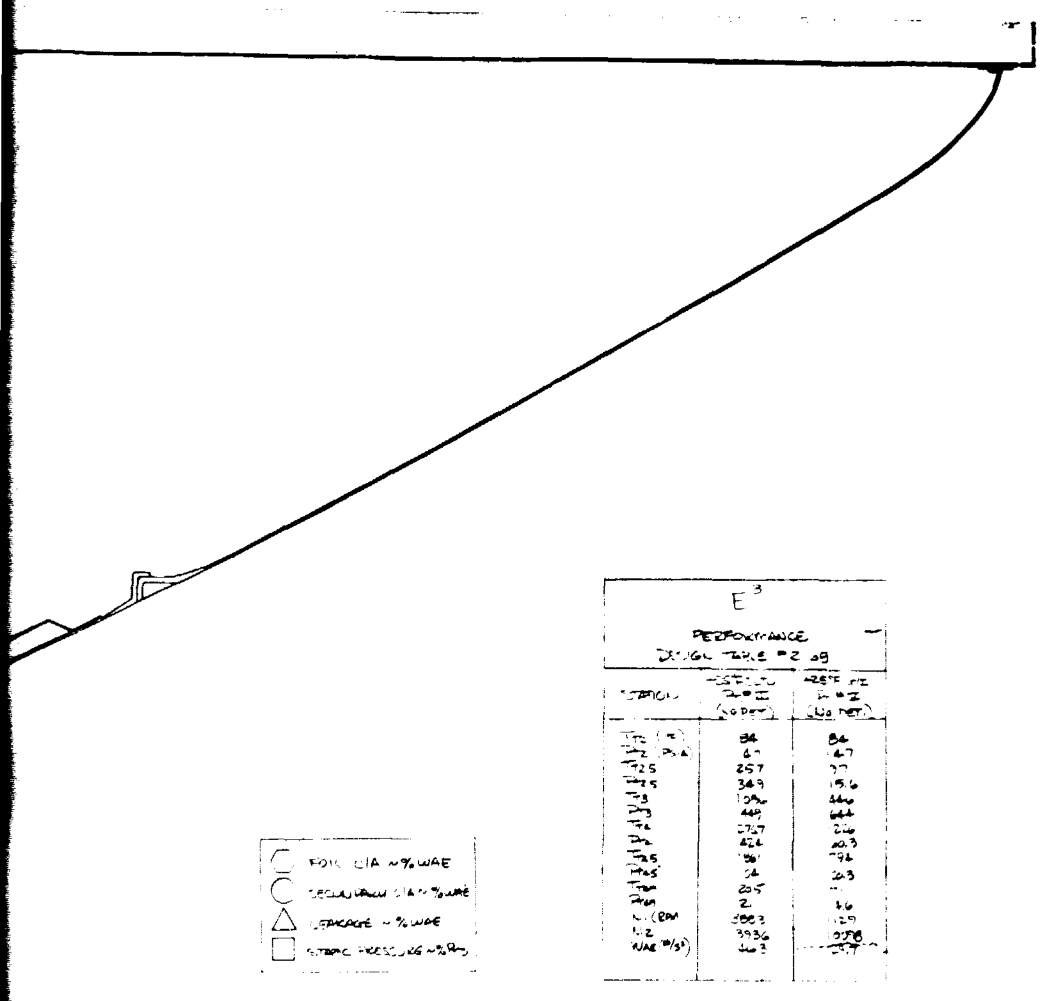

Figure 8 Integrated Core/Low Spool Secondary Airflow System Map

Integrated Core/Low Spool Preliminary Design Action Items: Several action items from the Integrated Core/Low Spool Preliminary Design Review were addressed. These items included (1) exhaust mixer benefit projections and performance details, (2) integrated core/low spool starting torque requirements and light-off speed estimate from combustor sector rig testing, and (3) ways to map the fan during integrated core/low spool testing.

A decision was made to incorporate the best performing exhaust mixer from the Phase II mixer model testing into the integrated core/low spool. The impact of this decision on expected test thrust specific fuel consumption was assessed. This assessment indicated a penalty of 0.5 percent would result. A breakdown of the pressure loss increases causing this penalty is shown in Table 10.

TABLE 10

PHASE II EXHAUST MIXER MODEL TEST IMPACT ON
INTEGRATED CORE/LOW SPOOL THRUST SPECIFIC FUEL CONSUMPTION
(Maximum cruise, 35000 ft., 0.8 Mn, standard day)

	Expected Performance (50% Probability of Achievement)		
	5/80 Status	Best Model - 3/81	TSFC %
Fan Duct Pressure Loss	.0074	.0084	+.08
Core Stream Pressure Loss	.0057	.0077	+.06
Duct Stream Pressure Loss	.0029	.0064	+.30
Exhaust Nozzle Pressure Loss	.0040	.0044	+.05
Mixing Efficiency	.85	.85	0
Thrust Coefficient	.9944	.9944	0
	Thrust Specific Fuel Consumption Increase		+.49

An estimate of motoring torque was made for the integrated core/low spool to determine starter sizing requirements. Results are shown in Figure 9. Superimposed on this figure is an estimate of the spool speed at which combustor lighting is accomplished for the integrated core/low spool. This lighting speed estimate of 2050 rpm was predicted from an extrapolation of performance parameters into the region below idle in combination with relight test results from the combustor sector rig. The prediction shows adequate torque margin for starting with the use of an ATS 200-11 starter.

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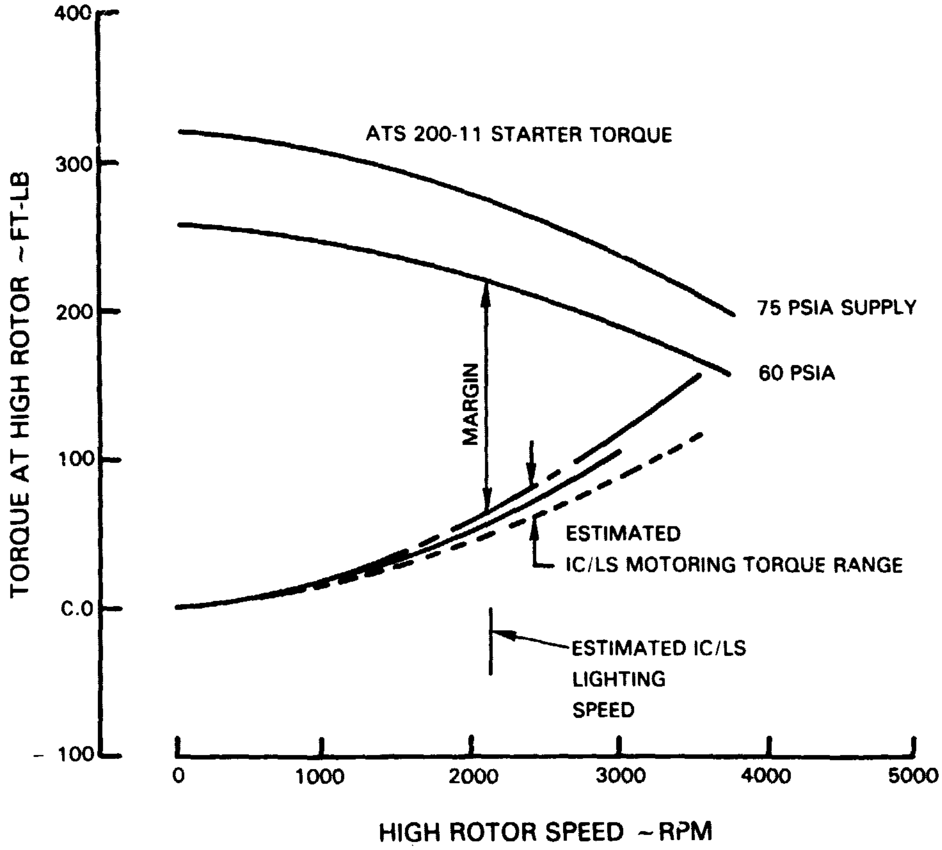

Figure 9 Integrated Core/Low Spool Torque Estimate and Starter Torque Capability (Standard Day)

An earlier investigation into ways to map the fan during sea level testing of the integrated core/low spool (refer to the Fourth Semiannual Status Report, dated April 1980) was expanded to include separate exhaust possibilities as well as to update mixed exhaust considerations. Mapping methods considered included:

- o combustor exit temperature variation;
- o exhaust nozzle area variations;
- o mixer area variations;
- o fan rotor exit bleed;
- o inlet pressure loss;
- o ambient temperature variation;
- o fan duct pressure loss;
- o high-pressure compressor bleeds;
- o and low-pressure compressor flow capacity variation.

Mapping capabilities with the ± 5% exhaust nozzle area variation planned for the integrated core/low spool are shown for the mixed and separate exhaust configurations in Figures 10 and 11, respectively. Results showed that the portions of the map available with only 5 percent nozzle area variation are not extensive. Shaft torque and speed limits restrict mapping beyond the sea level static, zero Mach number, hot day takeoff operating point. In addition, closing the duct nozzle of the separate exhaust configuration causes a fan stability problem in the idle region. To correct this condition would require fan duct bleed. Other general conclusions were:

- o opening the exhaust nozzle area and bleeding from the fan duct defines the region below the operating line;
- o closing the exhaust nozzle area defines the region above the operating line;
- o changing combustor exit temperature defines the region along the operating line from idle to takeoff;
- o reducing fan inlet temperature raises the shaft speed limit;
- o reducing fan inlet pressure raises the shaft torque limit;
- o and varying exhaust nozzle area ± 20 percent is a reasonable maximum, but far duct bleed is required to run in the idle region with the separate exhaust configuration.

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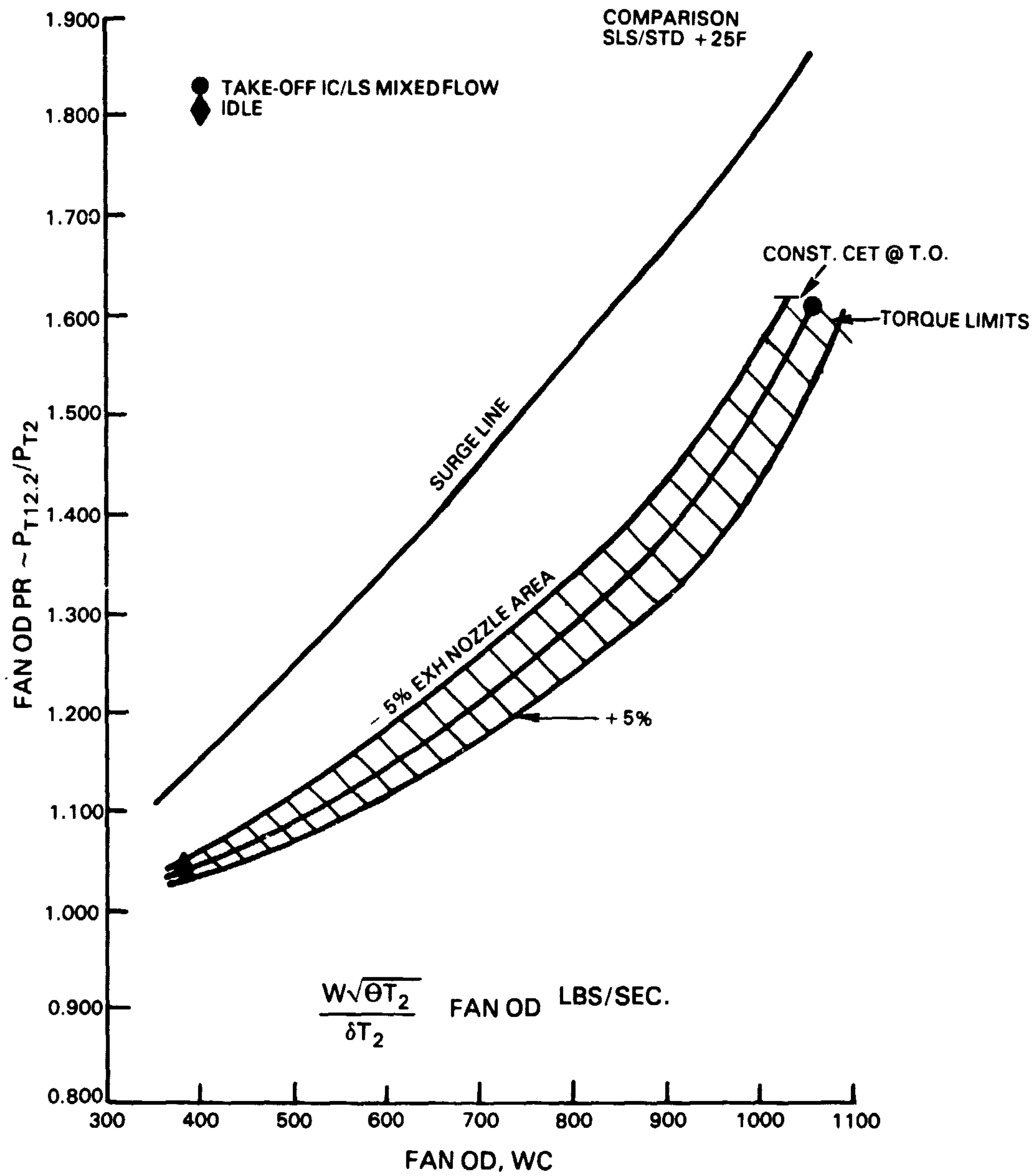

Figure 10 Integrated Core/Low Spool With + 5 Percent Exhaust Nozzle Area Variation For the Mixed Exhaust Configuration

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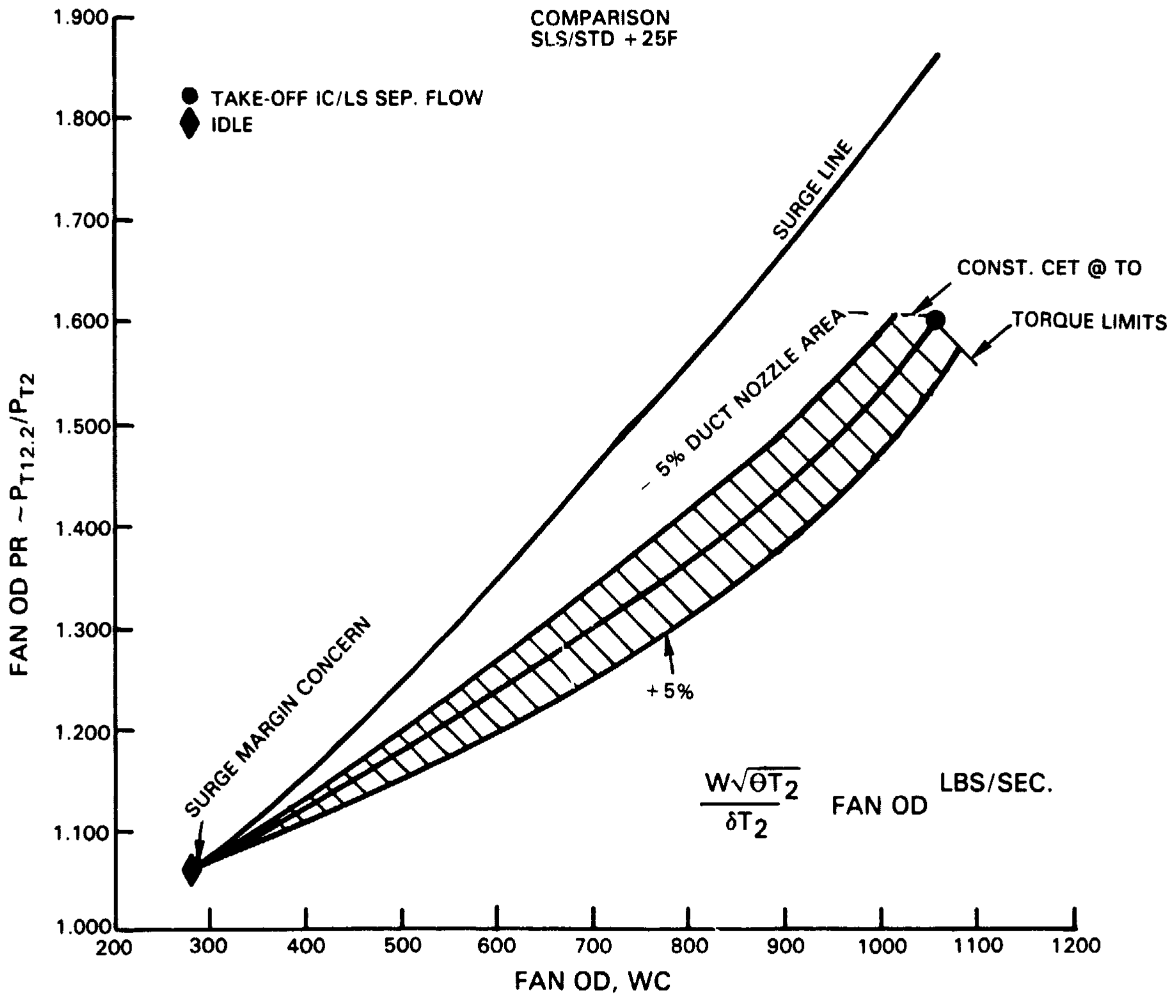

Figure 11 Integrated Core/Low Spool With ± 5 Percent Exhaust Nozzle Area Variation For the Separate Exhaust Configuration

Recommendations, if fan mapping is to be done during integrated core/low spool testing, are:

- o map the fan from idle to takeoff by varying combustor exit temperature and mixed (or, if necessary, duct) exhaust nozzle area;
- o provide ± 20 percent exhaust nozzle area variation capability for maximum operating line excursion;
- o provide at least 20 percent fan duct bleed capability for fan idle stability (if the mapping is done with the separate exhaust configuration).

Integrated Core/Low Spool Performance Update: An update to the integrated core/low spool performance was conducted to reflect design changes since May 1980. Expected thrust specific fuel consumption was determined using the computerized integrated core/low spool performance simulation. Four configurations of the integrated core/low spool were evaluated:

- o mixed exhaust without instrumentation;
- o mixed exhaust with instrumentation;
- o separate exhaust without instrumentation;
- o separate exhaust with instrumentation.

Component and system changes accounted for since the May 1980 status analysis resulted from (1) the completion of the detailed design of the low pressure spool, (2) updating of the cooling and secondary airflow systems, (3) completion of testing of the Phase II exhaust mixer models, and (4) designing the aerodynamics of the exhaust system. Included in these estimates were:

- o improvements in the tip clearances of the fan and low-pressure compressor;
- o revisions in the secondary airflow system and airfoil cooling air in the high-pressure turbine;
- o an increase in tip clearance, revision in the secondary airflow system, and reestimates of the aerodynamic design of the low-pressure turbine;
- o revisions to the fan map;
- o revisions in the pressure losses of the exhaust mixer and exhaust nozzle.

Factors contributing toward the predicted status integrated core/low spool maximum cruise thrust specific fuel consumption for the mixed exhaust configuration without instrumentation are shown in Table 11. The itemized component and system changes occurring since May 1980 resulted in a net thrust specific fuel consumption penalty of 0.3 percent. The newly predicted expected level represents a 10.0 percent advantage for the mixed exhaust integrated core/low spool without instrumentation relative to the JT9D-7A reference engine.

TABLE 11

INTEGRATED CORE/LOW SPOOL EXPECTED THRUST SPECIFIC FUEL CONSUMPTION STATUS - JULY 1981
MIXED EXHAUST, NO INSTRUMENTATION
(Maximum Cruise: 35000 ft, 0.8Mn, Standard Day); JT9D-7A Reference Engine

Component/System Change	TSFC Change (%)
+0.2% Δ Fan O.D. Efficiency (0.081-0.055 in. Tip Clearance)	-0.11
+0.1% Δ LPC Efficiency (0.021-0.0195 in. Average Tip Clearance)	-0.01
+0.3% Δ HPT Efficiency (Airfoil Cooling/Secondary Airflow Reductions/Revisions	-0.18
+0.2% Δ LPT Efficiency (Aero. Reassessment, 0.019-0.041 in. Average Tip Clearance, Net Secondary Airflow Revisions Offsetting)	-0.14
+0.29% Δ Duct Mixer Pressure Loss	+0.25
+0.29% Δ Core Mixer Pressure Loss	+0.08
-0.08% Δ Exhaust Nozzle Pressure Loss	-0.09
Secondary Airflow System and Turbine Cooling Airflow Changes (+.1% Core Airflow)	+0.20
Rematching Effects	+0.30
Total	+0.30
5/80 Status	-10.3
7/81 Status	-10.0

Based on this updated prediction at maximum cruise, a breakdown of the major elements contributing to the thrust specific fuel consumption improvement was defined. These elements and their contributions are presented in Table 12.

TABLE 12

INTEGRATED CORE/LOW SPOOL THRUST SPECIFIC FUEL CONSUMPTION
IMPROVEMENT STATUS - JULY 1981
MIXED EXHAUST, NO INSTRUMENTATION
(Aerodynamic Design Point: 35000 ft., 0.8Mn, Standard Day)
JT9D-7A Reference Engine
(50% Probability of Achievment)

Contributor		TSFC Change (%)
Low Pressure Spool		-3.1
High Pressure Spool		-1.7
Cycle		-3.4
Mixing/Installation		-1.8
	Total	-10.0

A comparison of the four configurations of the integrated core/low spool and the flight propulsion system in terms of sea level static takeoff and cruise performance are presented in Tables 13 and 14, respectively.

TABLE 13

COMPARISON OF INTEGRATED CORE/LOW SPOOL AND
FLIGHT PROPULSION SYSTEM PERFORMANCE
Takeoff: Sea Level, Zero Mn, 84°F day, Uninstalled

	JULY 1981 IC/LS*				JUNE 1981 FPS
	Mixed - No Instr.	Mixed - Instr.	Separate- No Instr.	Separate - Instr.	
Fan Pressure Ratio (Duct)	1.61	1.60	1.60	1.58	1.58
Bypass Ratio	6.71	6.77	6.72	6.81	6.83
Overall Pressure Ratio	32.0	31.8	31.8	31.3	31.10
Turbine Rotor Inlet Temp (°F)	2620	2620	2620	2620	2485
Thrust (lb)	38060	37200	37455	35925	37025
Thrust Specific Fuel Consumption (lb/hr/lb)	.352	.355	.355	.362	.327
Corrected Total Airflow (lb/sec)	1235	1235	1226	1216	1214

*Test inlet and exhaust nozzle incorporated

TABLE 14

COMPARISON OF INTEGRATED CORE/LOW SPOOL AND
FLIGHT PROPULSION SYSTEM PERFORMANCE
Maximum Cruise: 35000 ft., 0.8Mn, Standard Day, Uninstalled

	JULY 1981 IC/LS*				JUNE 1981 FPS
	Mixed - No Instr.	Mixed - Instr.	Separate- No Instr.	Separate - Instr.	
Fan Pressure Ratio (Duct)	1.71	1.70	1.71	1.69	1.71
Bypass Ratio	6.57	6.60	6.55	6.64	6.60
Overall Pressure Ratio	38.5	38.6	37.6	37.3	37.3
Turbine Rotor Inlet Temp (°F)	2310	2310	2310	2310	2195
Thrust (lb)	9085	8870	8625	8265	8935
Thrust Specific Fuel Consumption (lb/hr/lb)	.590	.600	.610	.625	.548
Corrected Total Airflow (lb/sec)	1395	1404	1360	1360	1356

*Test inlet and exhaust nozzle incorporated

Integrated Core/Low Spool Transient Analysis: The performance simulations were prepared so that the transient operation could be explored for the several configurations of the integrated core/low spool. Initial transient operation trials were conducted for the simulated configurations. Included in these trials were definitions of the operating lines and surge margins. Both rate limited and snap decelerations and accelerations were evaluated.

Transient operating results were analyzed. Several anomalies were observed which will require more detailed investigation and perhaps some refinement to the simulation.

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 sub-task 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 Aircraft 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 Aircraft 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 sub-task 3, Propulsion System Analysis and Design Update. Pratt & Whitney Aircraft 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

Planning was initiated for the second update of the propulsion system/aircraft integration evaluation. Flight and economic performance groundrules for conducting these evaluations were prepared and submitted to NASA for approval. Approval was received orally at the end of the reporting period, with the only change being the addition of a second fuel price level for evaluation to determine economic sensitivity. In addition, this effort was rescheduled by NASA for consistency with revised timing (Spring of 1982) for the second preliminary analysis and design update.

3.1.5 Benefit/Cost Study

3.1.5.1 Objective

Identify advanced fuel-saving technologies, whose timing is beyond the 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

The total benefit/cost study effort is divided into four sub-tasks, described as follows:

1) Benefit/Cost Study Ground Rules and Screening: a two-month effort undertaken to 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. These concepts are ranked and the 20 deemed most feasible are submitted to NASA for approval to proceed with a refined assessment.

2) Benefit/Cost Study Refined Assessment: a two and one-half month effort which further evaluates the 20 concepts selected in sub-task 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 further work.

3) Integration of Benefit/Cost Study Concept into Engine System: a four-month effort designed to integrate the best concepts selected in sub-task 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: provides for the preparation and submittal of oral and written status reports required by the contract.

3.1.5.3 Technical Progress

Efforts on the first sub-task, Benefit/Cost Study Ground Rules and Screening, were initiated late in the reporting period with submittal of proposed study ground rules to NASA for review and approval. Verbal approval was received at the end of the reporting period.

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 Aircraft 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 12, and the work plan, in Figure 13. 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.

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) Comfirm 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 (30 September 1981) has been completed. Exceptions are indicated in the appropriate technical progress sections of this report. Figure 13 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.

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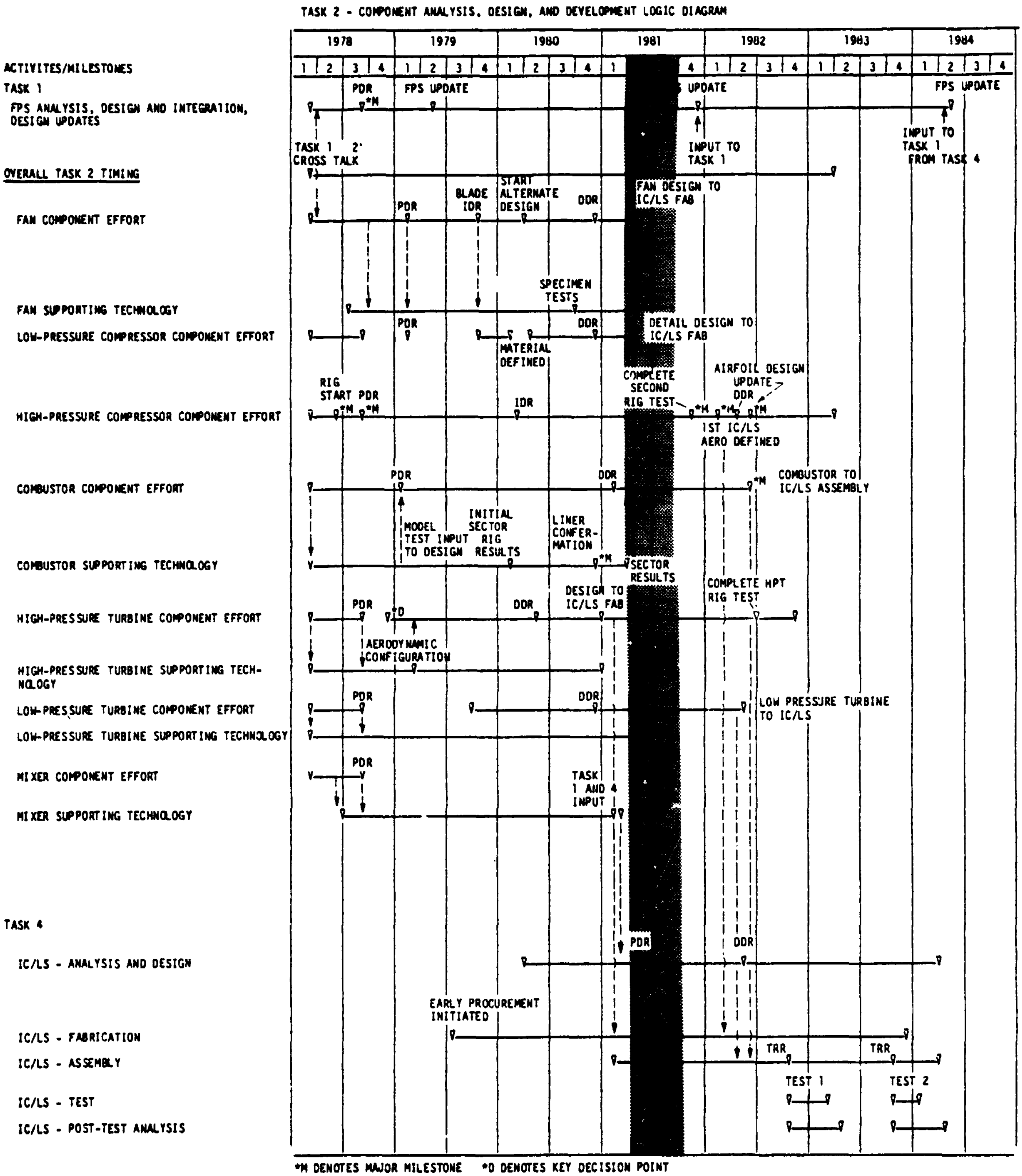

Figure 12 Task 2 Logic Diagram

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TASK 2 EFFORT

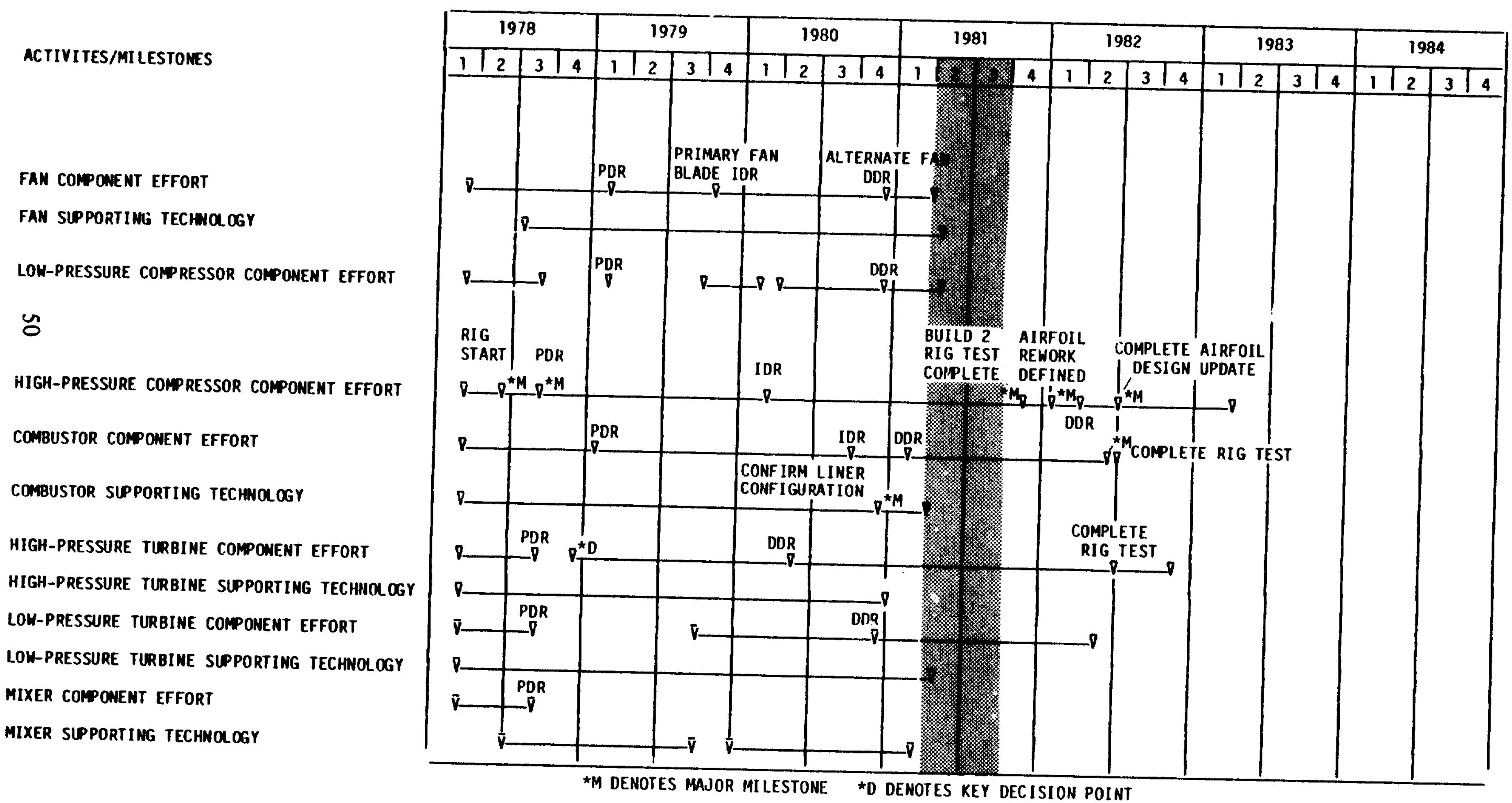


Figure 13 Task 2 Work PLan Schedule

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 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 re-directed to place more emphasis on design and fabrication of the shrouded fan design. This re-direction 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.

3.2.3.2 Component Program Overview

The overall task effort consists of (1) a shrouded fan component detailed design effort which provides a conventional design for use in the Task 4 IC/LS program, 2) the preliminary analysis and design of a shroudless hollow fan blade to determine the feasibility of this design concept, followed by a detailed design effort which completes the blade design and produces drawings suitable for fabricating hardware, and 3) a Shroudless Blade Supporting Technology Program which evaluates the feasibility of specific hollow blade conceptual fabrication techniques.

The shrouded blade component design was accomplished in 1980 to provide final hardware drawings for fabrication of integrated core/low spool fan component hardware in early 1981.

Figure 14 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.

3.2.3.3 Component Effort

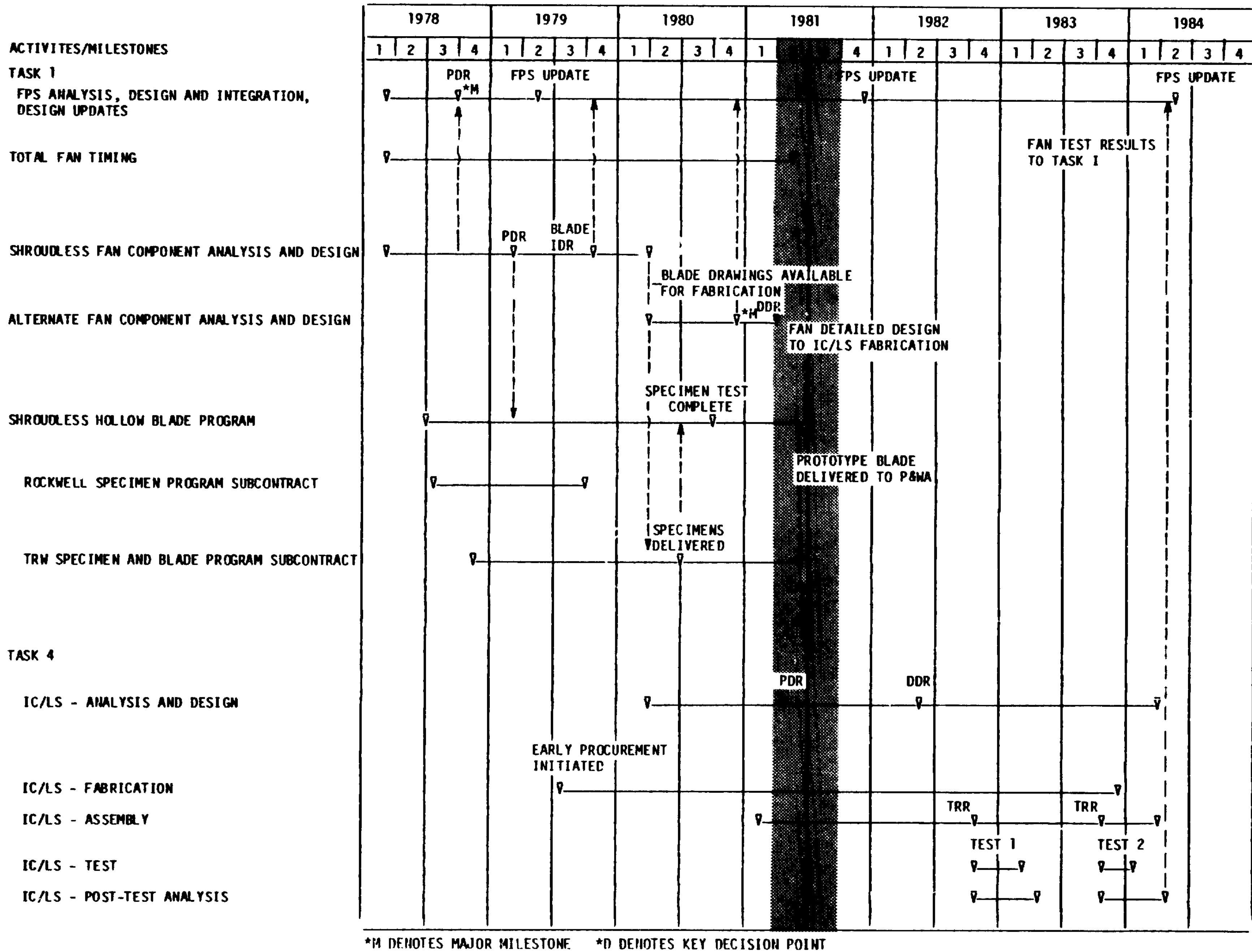
3.2.3.3.1 Objective

The primary objective of this effort is to design an aft part span shroud single stage fan component for the integrated core/low spool in Task 4. The aspect ratio of the blade is 4.0 and the experimental expected efficiency for the component is 84.7%.

A secondary objective of this effort is to conduct the preliminary and detailed design of a shroudless, hollow fan blade with an aspect ratio of 2.5 for use in the Hollow Blade Supporting Technology Program.

3.2.3.3.2 Scope of Total Work Planned

The fan component effort is initiated with the shroudless fan 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.



*M DENOTES MAJOR MILESTONE   *D DENOTES KEY DECISION POINT

Figure 14   Fan Program Logic Diagram

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.

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 14 indicates that all technical effort associated with the fan component design program is complete. Program results are reported in NASA CR-165466.

3.2.3.4. Supporting Technology

3.2.3.4.1 Scaled Fan Rig Test Program

Pratt & Whitney Aircraft 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 reduced.

3.2.3.4.2 Hollow Blade Technology Program

3.2.3.4.2.1 Objective

Evaluate and select a construction technique suitable for use with hollow fan blades.

3.2.3.4.2.2 Scope of Total Work Planned

This supporting technology program effort consists of the phases shown in Figure 15. Rockwell International is retained as a subcontractor to provide a test specimen fabricated utilizing the superplastic forming/difusion bonding (SPF/DB) technique. The completed specimen is used to evaluate the possible application of this technique to the fabrication of hollow fan blades and fan exit guide vanes.

A subcontract is also issued to TRW to evaluate a construction method that uses multiple layers of titanium laminate that are hot isostatically pressed (HIP) diffusion bonded, and isothermally resized to final aerodynamic shape.

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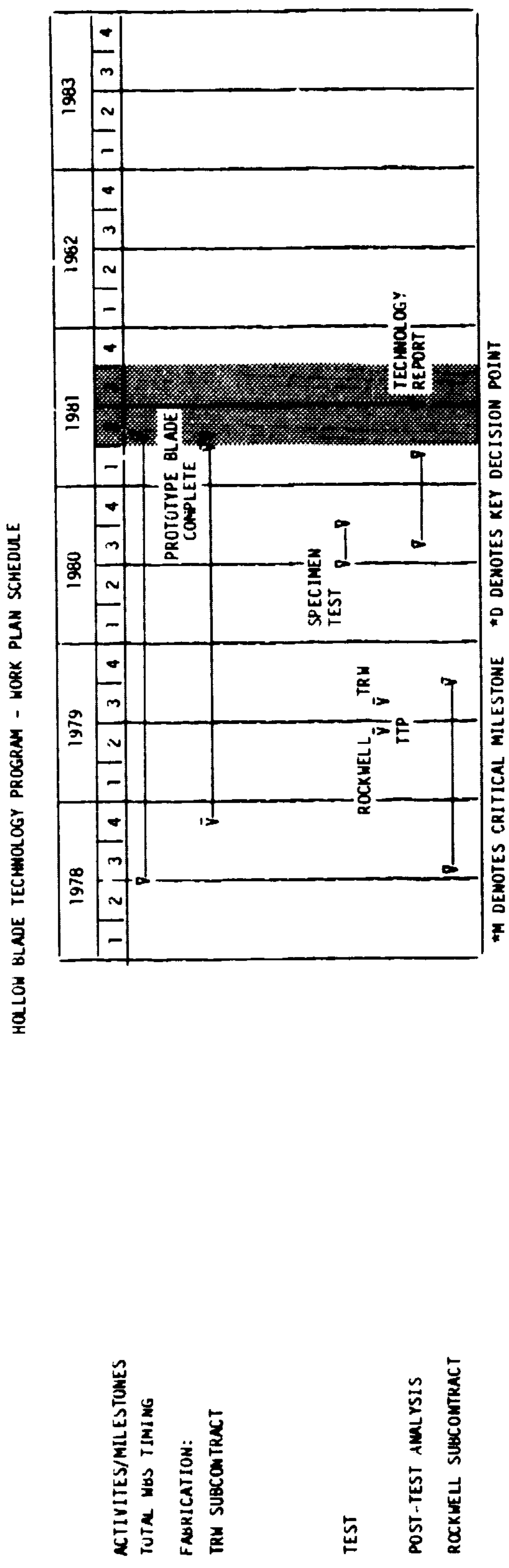


Figure 15 Hollow Blade Technology Program Work Plan Schedule

Initially, test specimens are fabricated by Rockwell and TRW in order to develop their respective fabrication process and provide material for metallographic, tensile, and fatigue evaluations. In addition, full-size prototype blades are fabricated by TRW utilizing (1) the design established during the Shroudless Blade Detailed Design component effort, and (2) the fabrication technique evolved from the initial test specimen effort. This effort establishes the feasibility of TRW's fabrication approach.

At the completion of subcontractor efforts and following suitable structural tests, a construction technique will be selected.

Figure 15 indicates that all tasks under this supporting technology program have been completed. Information detailing all efforts under this program will be published in a technology report currently being prepared by Pratt & Whitney Aircraft for submittal to NASA.

3.2.3.4.2.3 Technical Progress

3.2.3.4.2.3.1 Summary of Work Previously Completed

Rockwell Efforts

Rockwell, originally scheduled to produce twenty-four specimens to be used in evaluating the superplastic forming/diffusion bonding process, encountered technical difficulties during the fabrication of the original prototype test specimens. Rockwell was unable to resolve these problems with the available subcontractor funds; therefore, it was mutually decided to terminate all technical efforts. These efforts were summarized by Rockwell in a final technical report that was submitted to Pratt & Whitney Aircraft. The Rockwell final report was reproduced in its entirety in the Appendix to the Fourth Semiannual Status Report.

TRW Efforts

TRW subcontract efforts up to the current reporting period have concentrated on (1) selecting a fabrication method for full-size hollow blades and (2) the fabrication of diamond shaped test specimens to be used in verifying the selected fabrication process.

Diamond-Shaped Specimen Fabrication And Testing: A fully-laminated approach was selected as being more cost-effective in production quantities than conventional forging or isothermal forging. The diamond shaped test specimens were fabricated using laminated titanium sheets and steel alloy cores to create the hollow sections. By varying fabrication parameters, TRW was able to make a preliminary assessment of the process margins required for fabrication of full-size blades. However, TRW encountered significant difficulties in fabricating diamond shaped specimens suitable for testing. These difficulties included (1) hot isostatic press container leaks, (2) iron core shift, (3) microporosity at the bond interface, and (4) irregular hollow cavity surface finish. Table 15 indicates the corrective action taken for each problem.

TABLE 15

SUMMARY OF DIAMOND SPECIMEN FABRICATION DIFFICULTIES

PROBLEM	CORRECTIVE ACTION
Hot Isostatic Press Container Leaks	Increased the width of the container flange which permitted use of a more effective seam weld rather than the fusion butt weld previously used.
Iron Core Shift During Assembly and Hot Isostatic Press Cycle	X-ray inspection was added to the assembly procedure to ensure proper core registry during assembly. Core shift during the Hot Isostatic Press Cycle will require adjustments in the isothermal forging process.
Microporosity at the Bond Interface	Some improvement has been achieved by removing argon from the weld process and by heating the titanium to 900 F while hot vacuum outgassing to eliminate entrapped gases during the container sealing process.
Irregular Hollow Cavity Surface Finish	Substantial improvement in finish was achieved through the use of higher carbon alloy steel cores. However, this alloy caused regions of the titanium to crack while cooling. This is unacceptable and the use of higher crabon steel alloy has been rejected. The impact of these irregularities will therefore be evaluated during the specimen tests.

Testing included the evaluation of 'coupon' specimens sectioned from two of the diamond shaped panels. These coupons were bonded on one side to a shake table and excited. This excitation process produced a high cycle fatigue stress in the critical internal core leading edge radius. As designed, this radius was to be .062 inches nominal. As fabricated, however, the radii were irregular and as low as .002 inches. These irregularities were the result of the titanium ply endings not forming properly around the steel core radius during the hot isostatic pressing cycle.

Tests were run at successively higher stress levels. Runout at 10^7 cycles was considered a successful test. Crack generation prior to runout at 10^7 cycles was considered a failure. Results of this testing indicated runout ranging from 30 KSI to 45 KSI depending on the specimen. Minimum material manual high cycle fatigue strength for this titanium alloy is 45 KSI. The average runout level of 38 KSI is 15 percent below minimum. The poor radii on the specimens is the cause of the reduced properties.

Metallographic review of the specimens after failure revealed that none of the cracks generated were associated with the titanium laminate bonds. This fact gives some credibility to the basic blade fabrication approach of laminated hot isostatic pressing diffusion bonding.

Blade Process Development And Fabrication: Full-scale blade fabrication activities included (1) definition of the geometrical characteristics of the plies and completion of a program to generate the plies with core cutouts, (2) selection of the most cost-effective machining approach for producing cores in experimental quantities, (3) completion of the design of the hot isostatic press cans, (4) completion of all tooling required to camber, twist, and forge the blades, (5) completion of all ply and core details required to fabricate twelve prototype blades, and (6) the assembly and hot isostatic pressing (HIP) of blades. The modified flat assembly approach used to produce the full-scale fan blade was discussed in detail in the Fifth Semiannual Status Report. NASA, TRW, and Pratt & Whitney Aircraft mutually agreed that no further fan blades would be fabricated following completion of the ninth blade. Table 16 describes the configuration and status of each of these nine blades. Processing of blades numbered six through nine continued through isothermal forging.

TABLE 16

Blade Number	Plys	Cores	Root	Status
1	Hand Cut	None	Blocks	HIPed, leaked
2	Hand Cut	Partial	Blocks	HIPed, leaked; reHIPed; leaked
3	NC Machined	Full	None	HIPed, leaked
4	NC Machined	Full	Blocks	HIPed, leaked
5	NC Machined	Full	Blocks	HIPed, leaked, used for cambering trial
6	NC Machined	Full	None	HIPed, well bonded, cambered, twisted
7	NC Machined	Full	None	HIPed, well bonded, cambered, twisted
8	NC Machined	Full	None	HIPed, well bonded, cambered, twisted
9	NC Machined	Full	None	HIPed, well bonded, cambered, twisted

3.2.3.4.2.3.2 Current Technical Progress

TRW Efforts

The isothermal forging process of five pre-twisted and cambered blades was being prepared late in the preceding report period. This forging was completed during the current report period without any technical problems with respect to the blade processing procedure. However, there were some problems associated with forge die closure near the tip of the blade.

The tip end of the dies did not fully contact the blade preform during the initial forge trial. The forge dies were adjusted several times in an attempt to remedy the problem but it became apparent that a manufacturing iteration was necessary to dimensionally correct the die set. This iteration was not part of the program. Therefore, the best setup possible was attained. Blades 4, 6, and 9 were used to set the final die position.

Blades 7 and 8 were isothermally forged with what was considered the best setup. The blades were forged without incident. Dimensional analysis of the forged blades is shown in Table 17. The pitch thicknesses measured for both blades either met or are very close in meeting blueprint requirements. Only the last section inspected indicates a large oversize condition. This was expected due to the lack of die contact at the tip which was mentioned earlier. The lack of contact in this area during forging (as indicated by ply endings present on surface) is shown in Figure 16.

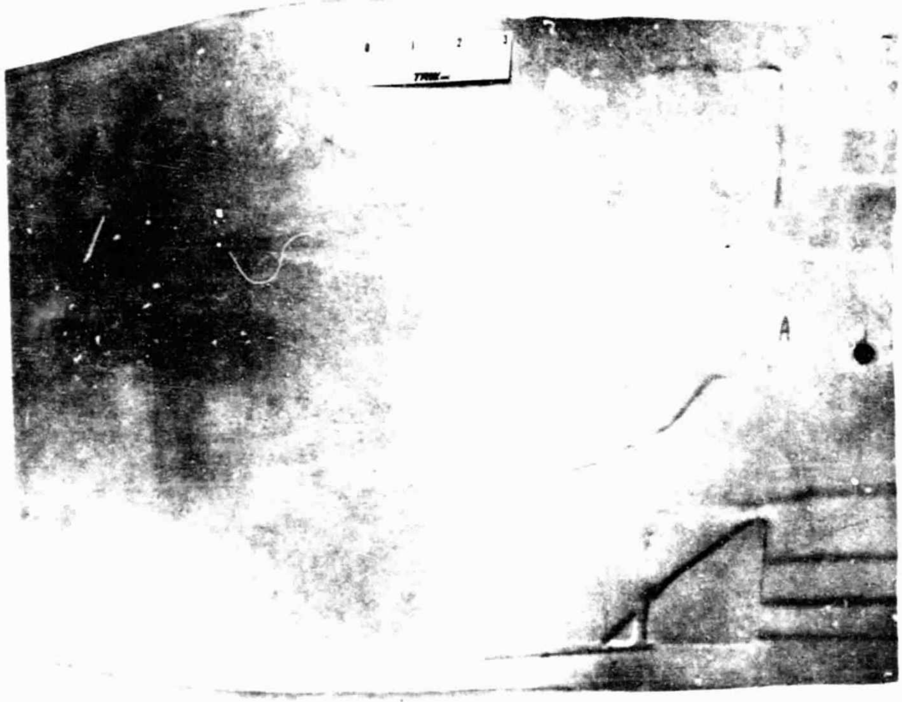


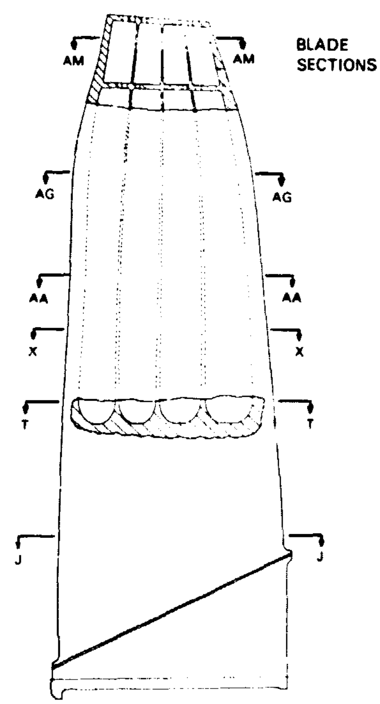
Figure 16 Tip Section of Blade Number 7 After Isothermal Forging Showing Lack of Contact During Forging in This Area as Indicated by Ply Endings Present on Surface

TABLE 17

DIMENSIONAL ANALYSES OF BLADES 7 AND 8

	PITCH			LEADING EDGE		
Section	Blueprint Requirements*	Actual Blade 7	Actual Blade 8	Blueprint Requirements*	Actual Blade 7	Actual Blade 8
J-J	0.852	0.866	0.865	0.081	0.184	0.161
T-T	0.828	0.833	0.832	0.079	0.201	0.195
X-X	0.755	0.746	0.742	0.073	0.146	0.147
AA-AA	0.652	0.642	0.639	0.063	0.121	0.122
AG-AG	0.461	0.467	0.469	0.047	0.071	0.070
AM-AM	0.345	0.392	0.390	0.046	0.115	0.112

	TRAILING EDGE		
Section	Blueprint Requirements*	Actual Blade 7	Actual Blade 8
J-J	0.089	0.037	0.018
T-T	0.089	0.105	0.094
X-X	0.079	0.118	0.118
AA-AA	0.069	0.100	0.094
AG-AG	0.049	0.104	0.096
AM-AM	0.038	0.113	0.109



* - Tolerance of ± 0.008 inch allowed

In general, the leading and trailing edges were found to be heavy in thickness except for one section. The heavy condition is attributed to the thermal expansion of the dies during heating. The dies want to bow outward along the width. The wider the chord, the more significant the problem. It is normal to expect this condition in conventional fan blades. Die iterations are generally necessary in most new applications. However, the wider chord of the Energy Efficient Engine blade has added to the problem. If the program were to continue through the fabrication of additional blades, the results of this first iteration would be used to make a reoperation to the dies for the next set of blades.

All five isothermally forged blades were X-rayed. There was no evidence of disbond within the blades. It was found, however, that some spanwise and chordwise flowing of the core material occurred during the forging process. Excess material had been added to the blades to insure that 'steps' on the blade surface, formed by the edges of the laminated sheets, would be smoothed out during the forming process. This added material is thought to have caused an uneven pressure distribution on the core which resulted in the observed core deformation. Tayloring the distribution of this excess material could be a solution to the problem.

Blades 4 and 6 were sectioned after forging to inspect the interior cavities. More severe cavity surface irregularities were noted in these blades than were noted during the specimen fabrication phase of the program. It is apparent from these sections that a more resilient core material is required, one that will not soften at the 1700°F hot isostatic press and isothermal forge temperature. Figure 17 shows a typical condition of these interior cavities.

Sections of blade number 4 were examined metallographically for disbonds and/or contaminates at the bond lines. No evidence of either condition was found. Figure 18 shows a typical microstructure observed from blade 4.

Blades 7, 8, and 9 were machined after forging. This machining effort consisted of cutting the blade to the correct length and width dimensions and blending the edges. Polishing of the airfoils was accomplished in the tip region to remove the ply indications not removed during forging. Racetrack holes, shown in Figure 19, were machined in the end of each blade to allow for acid leach of the core material. The cores were then removed using a water/nitric acid (2:1 ratio) at 180°F. The blades were subsequently chemically milled to remove .001 inch of titanium from the blade and then vapor blasted. The finished blades are shown in Figure 20. Blade 7 was shipped to NASA, blade 8 to Pratt & Whitney Aircraft, and blade 9 was retained by TRW.

TRW prepared their final technology report summarizing the program effort and published it late in the report period. This report has been reproduced in its entirety in Volume II, Seventh Semiannual Status Report. The hollow fan blade supporting technology program is now complete.

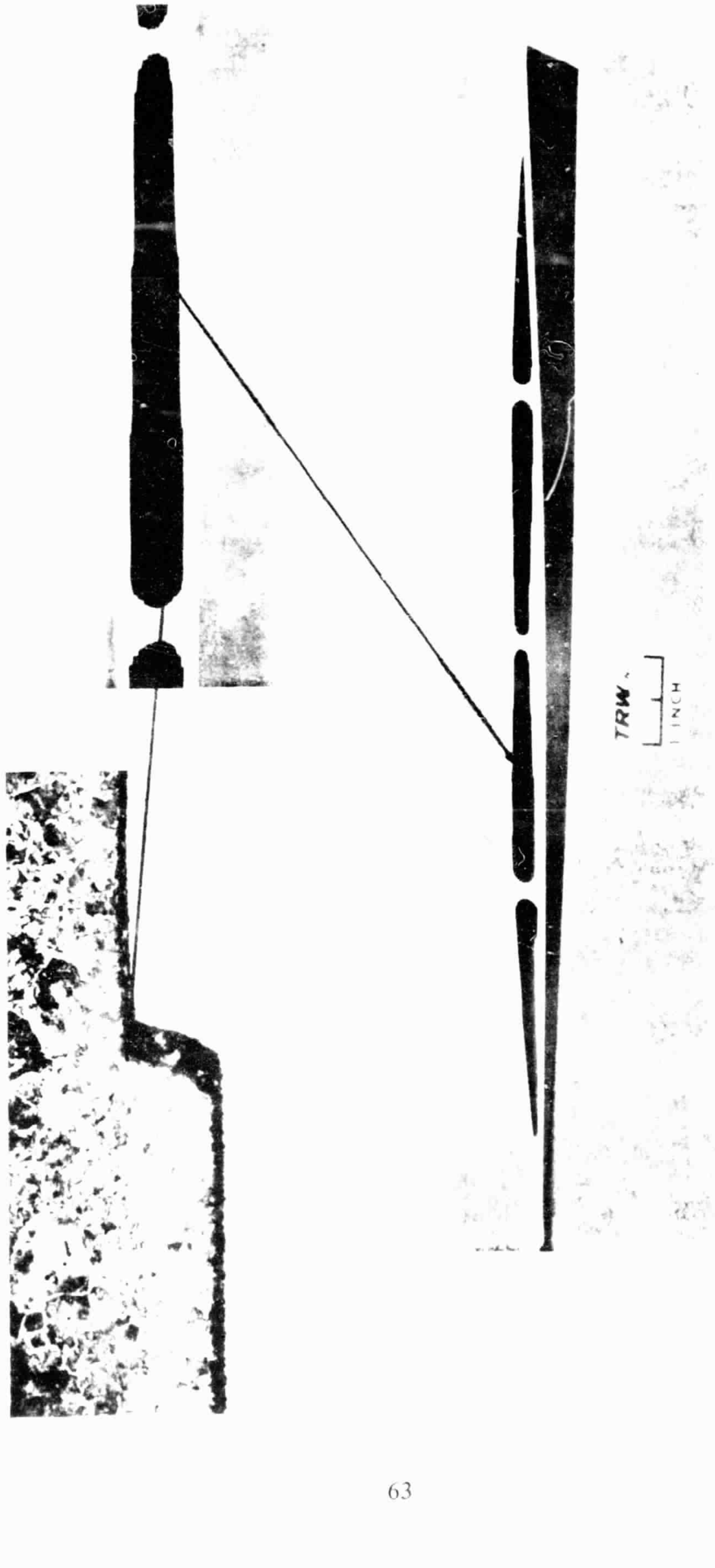


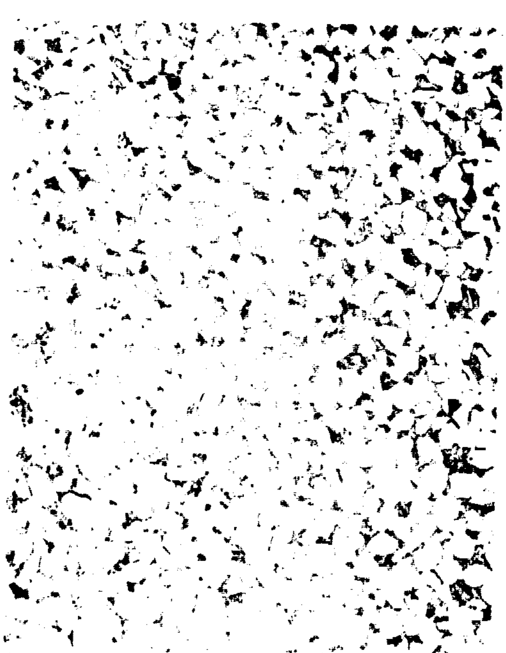
Figure 17 Section of Blade Number 4 With Core Removed Showing Typical Internal Wall Surface Condition

Bond Line

500X

Etch

98 H_2O
2 HNO_3
2 HF



200X

Etch

60 H_2O
35 HNO_3
5 HF

Bond Line

500X

Figure 18 Microstructure Observed for Blade 4

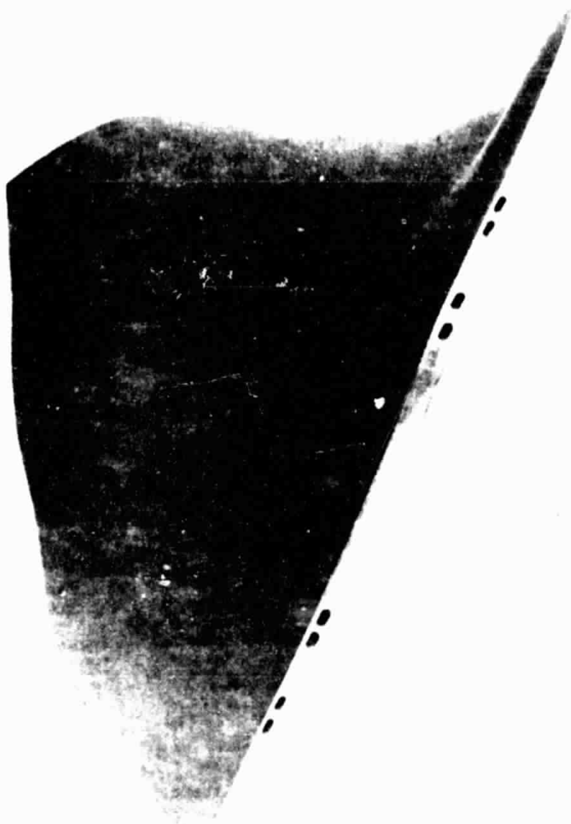


Figure 19 Racetrack Holes in Blade Tip

Figure 20 Blades 7, 8, and 9 After Finish Machining and Core Leaching

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 Aircraft programs such as an in-house supercritical cascade program, the NAVAIR Supercritical Cascade Test (Contract), 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 21, 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.

All technical efforts directed toward this component design program are complete. Program results are reported in NASA CR-165354.

3.2.5 High-Pressure Compressor

3.2.5.1 Overall Objective

Design a ten-stage, high-pressure compressor that produces a pressure ratio of 14, and has an adiabatic efficiency of 88.2 percent and an average blade aspect ratio of 1.5. The corresponding expected efficiency for the experimental integrated core/low spool component 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.

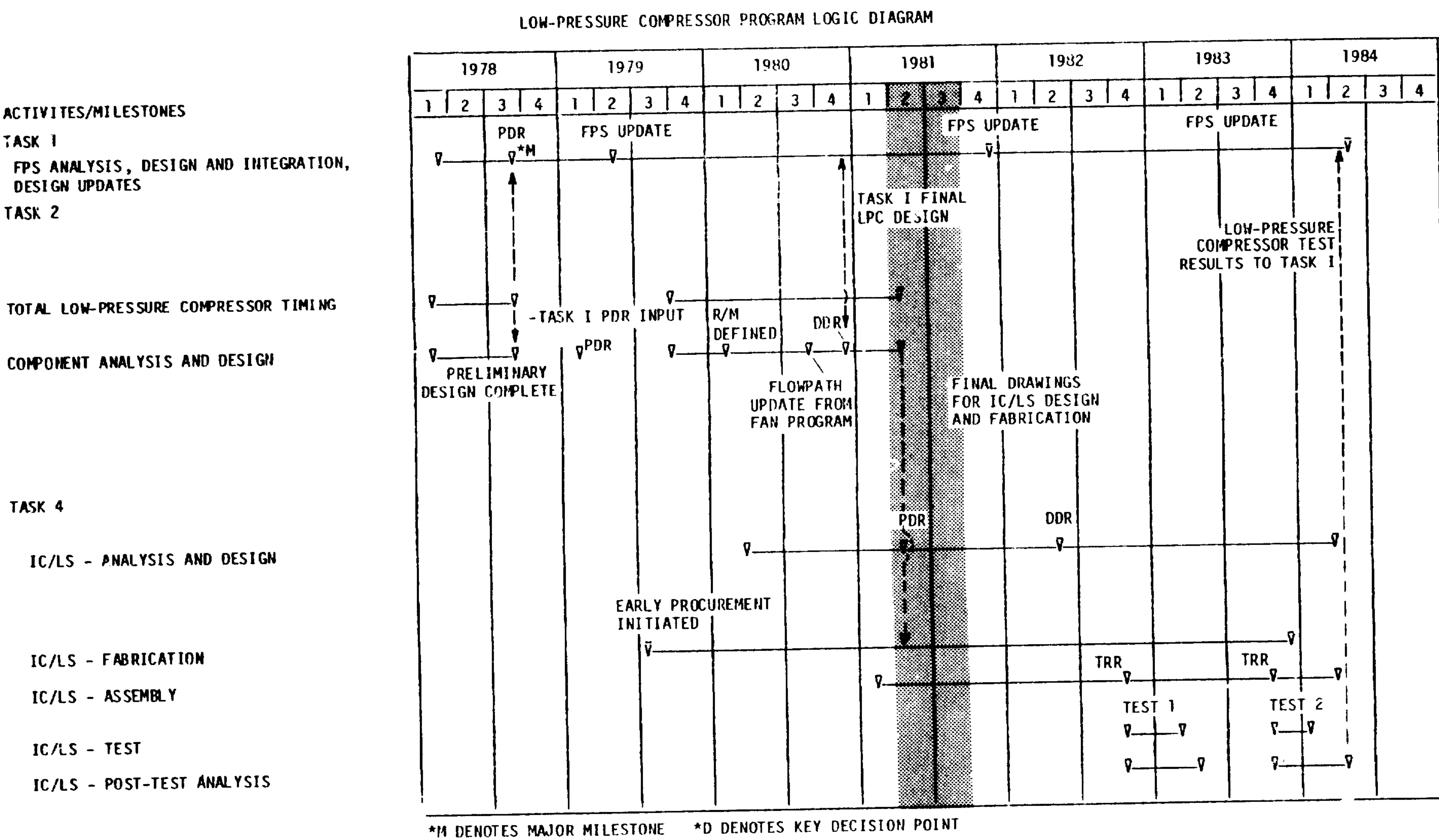


Figure 21 Low-Pressure Compressor Program Logic Diagram

3.2.5.2 Scope of Total Work Planned

The program consists of (1) a high-pressure compressor preliminary analysis and design phase, which determines the feasiblility of the compressor design; (2) a detailed analysis and design phase, which provides the hardware design for the high-pressure compressor rig program and integrated core/low spool program; (3) a high-pressure compressor hardware fabrication program, which supplies non-rotating and rotating component hardware to the high-pressure compressor rig program; and (4) a high-pressure compressor component rig program to verify and optimize the compressor design. The design effort does not require a separate supporting technology phase to provide design and verification of advanced concepts. This information is obtained principally from other Pratt & Whitney Aircraft programs such as an in-house supercritical cascade program and a NAVAIR supercritical cascade test.

The preliminary design activity provides layout drawings and substantiating data, which were presented to NASA for approval at a preliminary design review in September 1978. The results of the detailed design activity were presented for NASA approval at a detailed design review in February 1980.

Figure 22 shows the relationship between the elements of this task and contract Tasks 1 and 4. The program shown in the figure 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. Component and rig hardware is fabricated simultaneously. All component hardware is transferred to the rig program in October 1980 for assembly in the test rig. Upon analysis of the second build test data, the high-pressure compressor airfoil designs are updated, as required, to optimize the compressor design. The resultant updated airfoil requirements are utilized for hardware fabrication and transferred to Task 1 for the final flight propulsion system update. The third rig test incorporates the reworked airfoils and evaluates the performance of this optimized compressor design. The critical milestones for the high-pressure compressor effort are shown in the work plan schedule presented in Figure 23. 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, and (3) rig build 1 assembly, test, and post-test analysis activities. Component fabrication efforts are continuing, and rig build 2 efforts have been initiated.

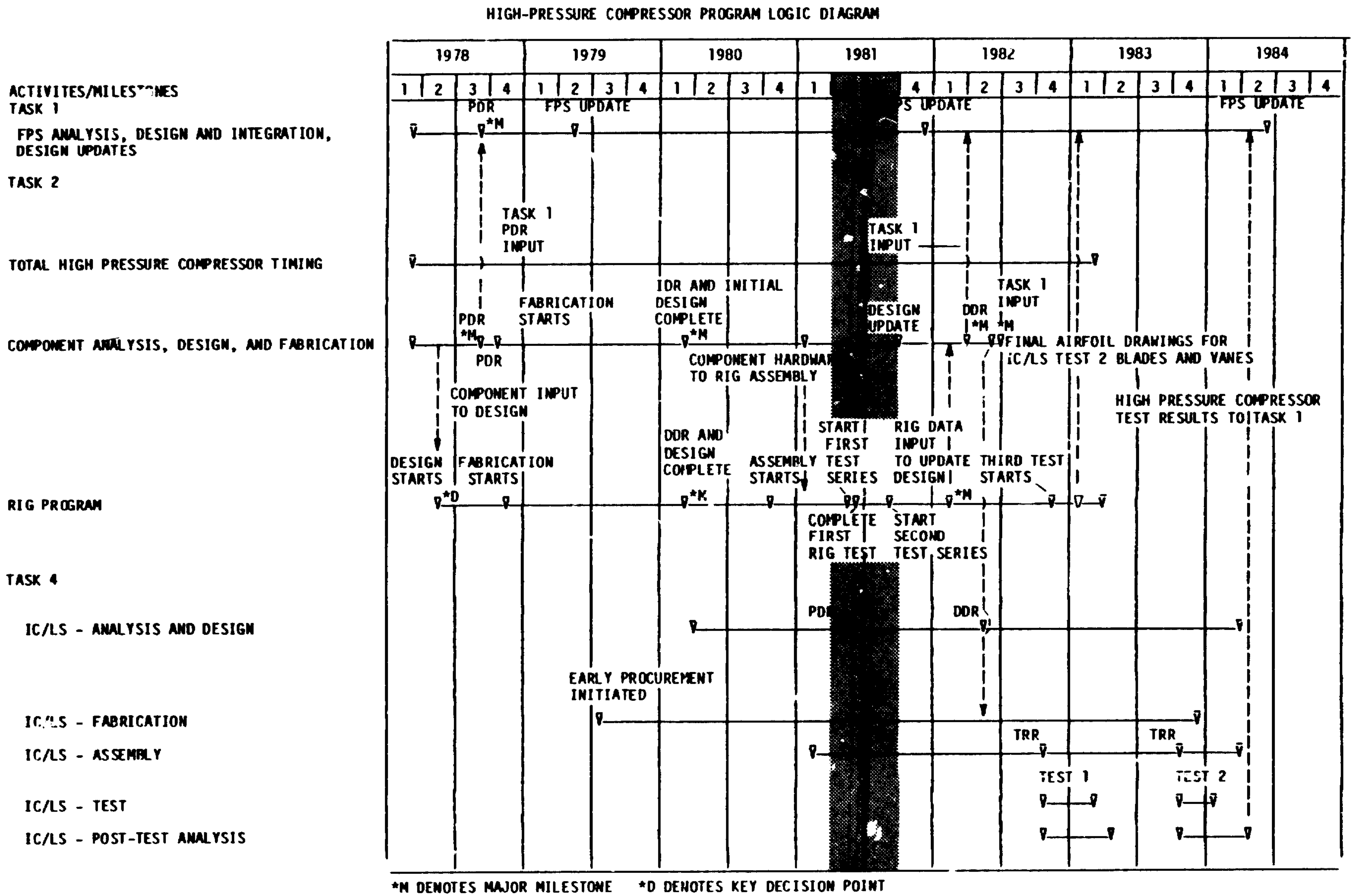


Figure 22 High-Pressure Compressor Program Logic Diagram

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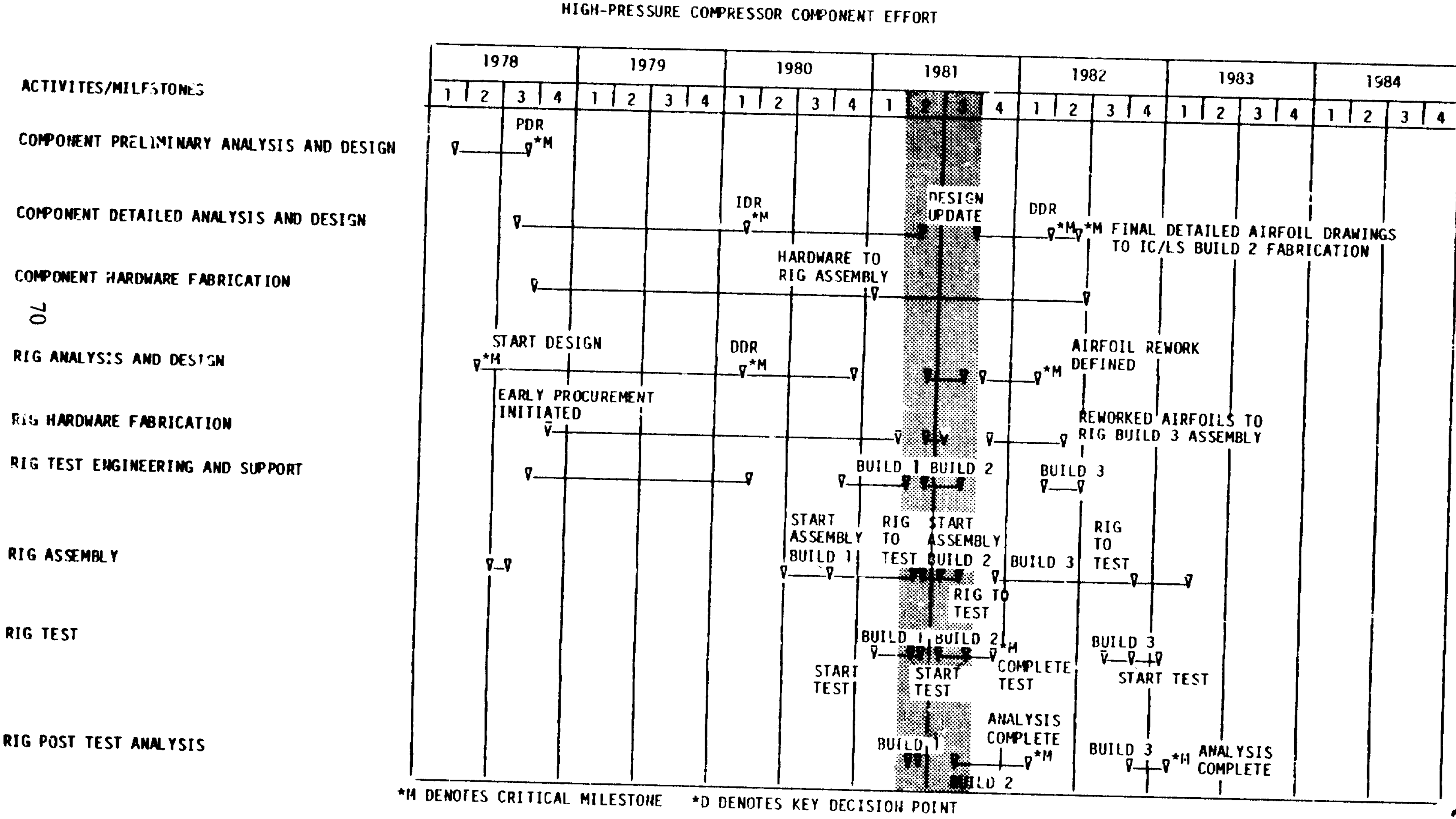


Figure 23 High-Pressure Compressor Component Effort Work Plan Schedule

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 its companion rig design are illustrated in Figures 24 and 25, respectively. All detailed analysis and design work for the high-pressure compressor component and rig was completed during a previous reporting period, 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 compressor has ten stages. The first four stages have variable geometry stators. The front case is a split case configuration to accomodate the variable geometry stators. The rear case is a single piece. Active clearance control is incorporated in the rear stages.

This high-pressure compressor design also features a drum rotor construction, extensive use of titanium in the static structure, and significantly fewer airfoils. These technology concepts combine to make the compressor assembly lighter, less costly, and easier to maintain. Current performance parameters for the compressor component at significant engine operating conditions are presented in Table 18.

The design of the ten-strut compressor intermediate case was included in the high-pressure compressor preliminary analysis and design effort and was continued into the detailed analysis and design phase. The nine most important functions of the case are listed below.

1. Supports the fan case.
2. Provides a portion of the fan flowpath and provisions for clamping of nacelle D-ducts.
3. Carries the nacelle loads (load sharing assumed).
4. Contains the fan exit vanes.
5. Supports the low-pressure compressor static structure and bleed actuating mechanism.
6. Forms the low- to high-pressure compressor flowpath.
7. Supports the fan and high-pressure compressor rotors.
8. Provides front mount locations.
9. Supports accessory drive shafts and gears.

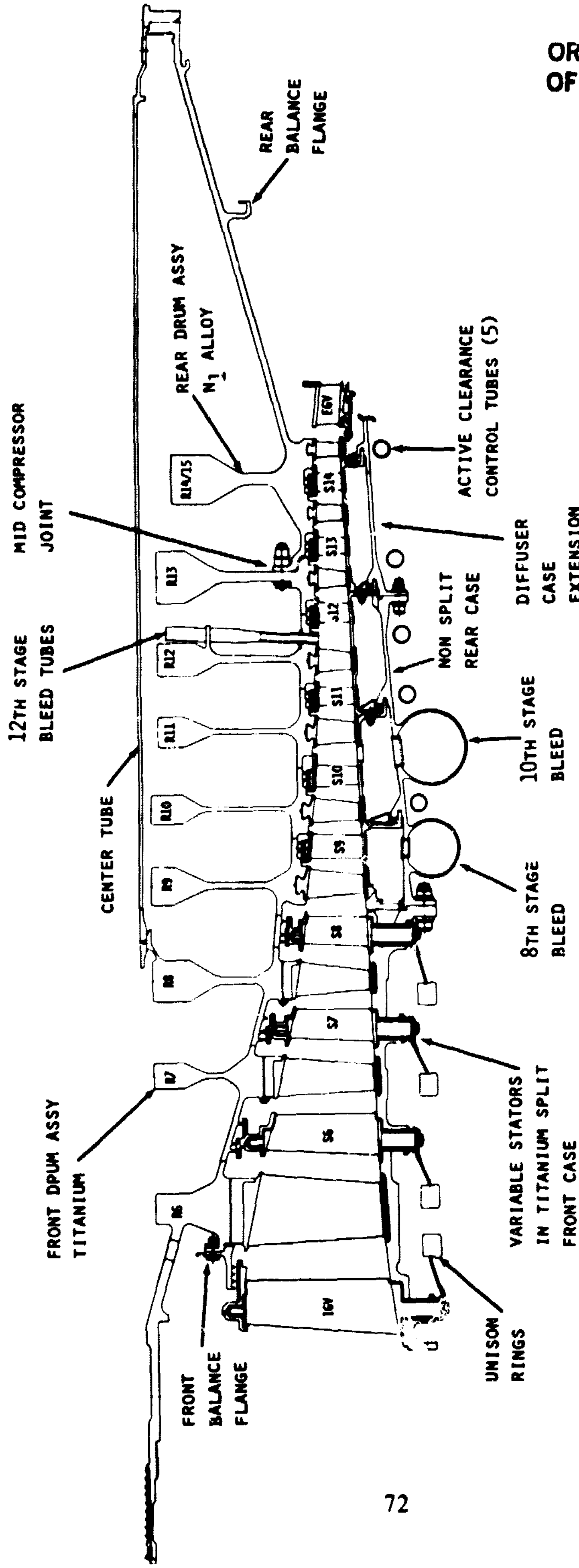
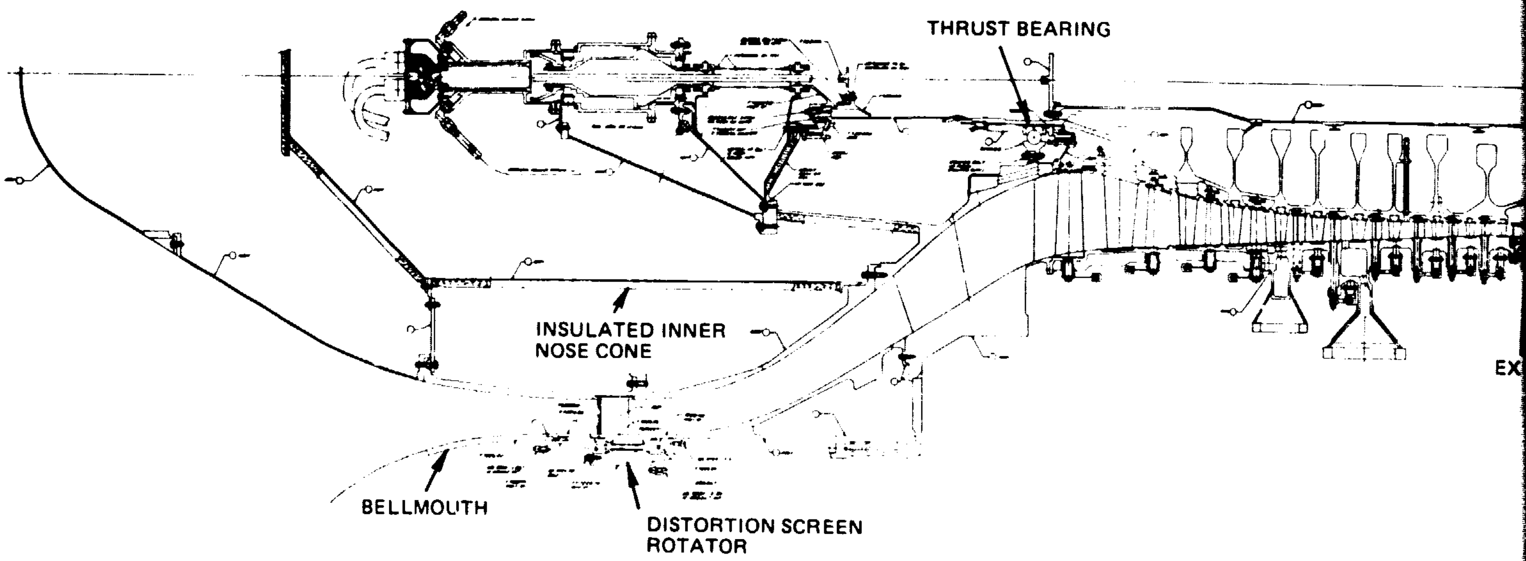


Figure 24 High-Pressure Compressor Component

PRATT & WHITNEY AIRCRAFT GROUP
COMMERCIAL PRODUCTS DIVISION



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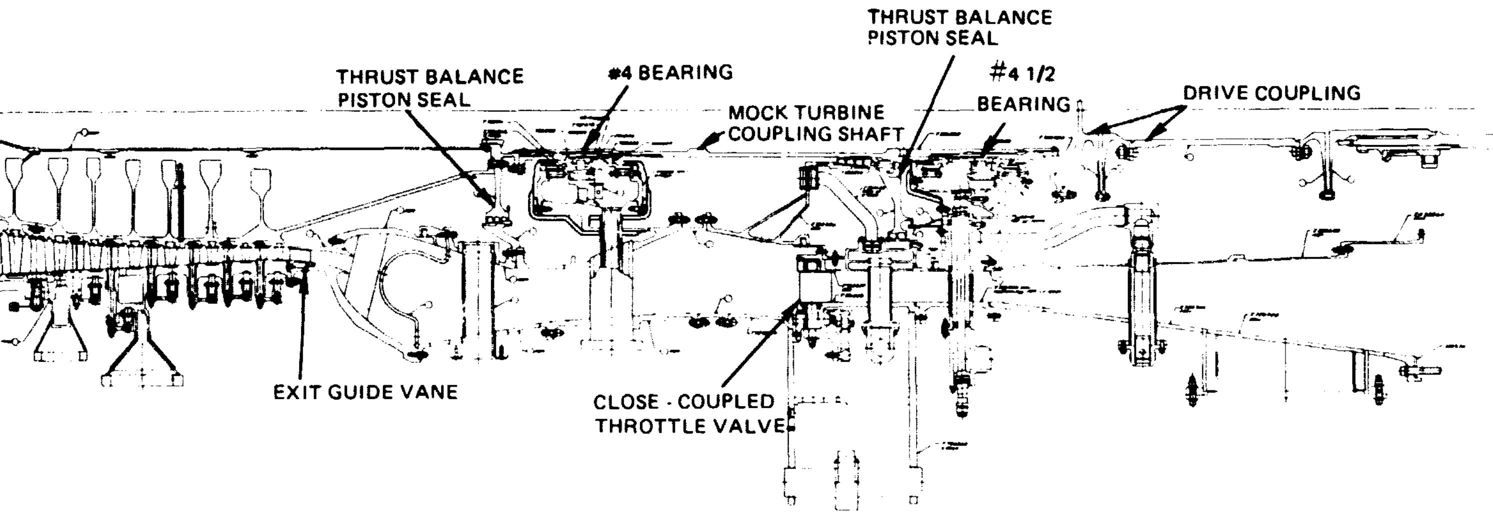

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Figure 25 High-Pressure Compressor Rig

TABLE 18

CURRENT HIGH-PRESSURE COMPRESSOR
PERFORMANCE PARAMETERS

	Engine Operating Condition			
	Aero. Des. Point	Maximum Cruise	Maximum Cruise	Takeoff
Pressure Ratio	14.00	13.85	14.25	13.05
Efficiency (percent)				
(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 (F)	898	883	976	1060

Fabrication of the component and rig-unique hardware, in support of the build 1 rig test program, was completed prior to the current reporting period. All hardware was delivered to assembly for incorporation into the component rig. Assembly efforts included trial fit-ups of the major rig subassemblies, installation of blades on the rotor assembly followed by rotor balance and blade tip grinding, installation of vanes, shrouds, vane arms and unison rings in the compressor cases, and instrumentation of all major subassemblies.

3.2.5.3.2 Current Technical Progress

Final assembly of the high-pressure compressor test rig (Build 1) was completed during the report period and the rig was delivered to the test facility (X-211) on 17 May 1981. Figure 26 shows the rig assembly in the final stage of preparation immediately preceding delivery to the test stand.

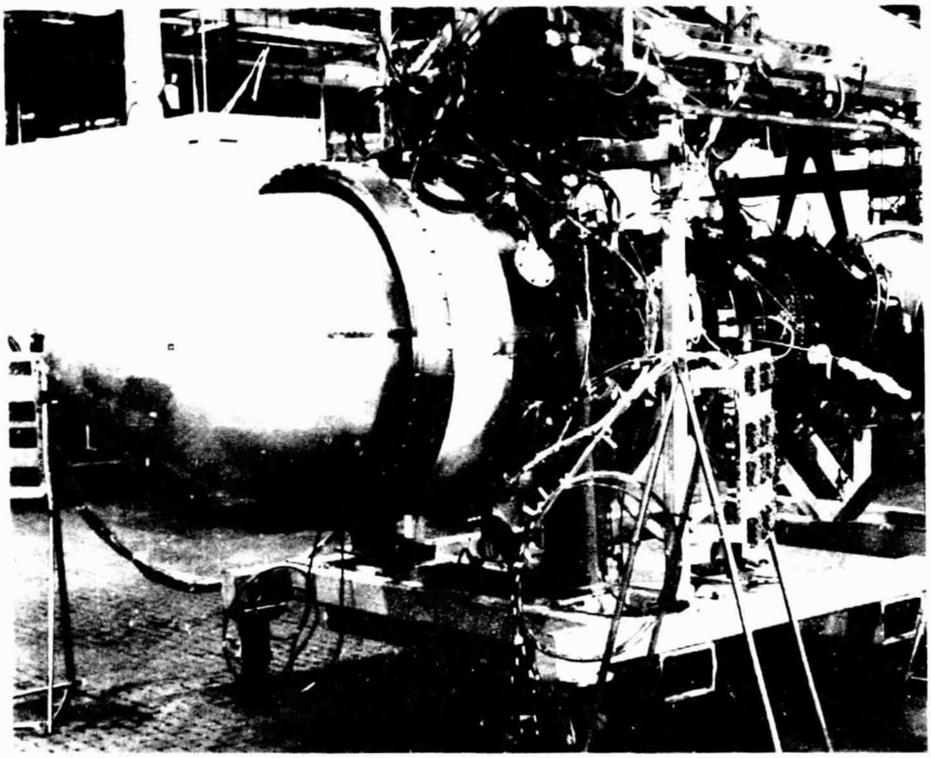

Figure 26 Assembled High-Pressure Compressor Rig Build 1

High-Pressure Compressor (Build 1) Test: The mounting of the compressor rig in the X-211 test stand was completed on 28 May 1981. The initial test series scheduled for the following day was successfully completed. This shakedown program consisted of evaluating all test stand and rig mechanical systems at rig speeds up to 70 percent of the design goal.

After completing the stress and vibration survey portion of the rig test in which acceleration to 100 percent speed on a wide open throttle valve discharge line was accomplished, high stresses were found on the rig rear thrust balance piston seal. This piston seal is identified in a cross section of the high-compressor rig shown in Figure 25. The high stress was attributed to seal flutter occurring when the pressure differential across the piston was increased. Following a review of stress data results, the rig was dismounted from the test stand and returned to the assembly area for removal and inspection of the rear thrust balance piston seal.

Analysis of data acquired from the build 1 rig test provided no insight into compressor performance since no data at or near the aerodynamic design point were obtained. These data will be acquired from the build 2 rig test.

High-Pressure Compressor Rig (Build 2) Design: A review of stress data on the rear thrust balance piston seal revealed that a high stress condition existed. In order to eliminate this flutter induced stress condition, a replacement seal was designed. The new seal incorporates a more substantial cross section, a single knife edge seal, and two rim ring dampers to provide improved stability. Figure 27 shows the original rig build 1 thrust balance piston seal design and the redesigned part used in rig build 2.

High-Pressure Compressor Rig Build 2 Fabrication: Fabrication of the redesigned thrust balance piston seal was completed during the report period. No other new parts were required for build 2.

High-Pressure Compressor Rig (Build 2) Assembly: Following removal of the original thrust balance seal, the redesigned seal was instrumented with strain gages and installed in the second build. The build 2 assembly effort was completed on 27 July 1981 and the rig was delivered to the X-211 test facility.

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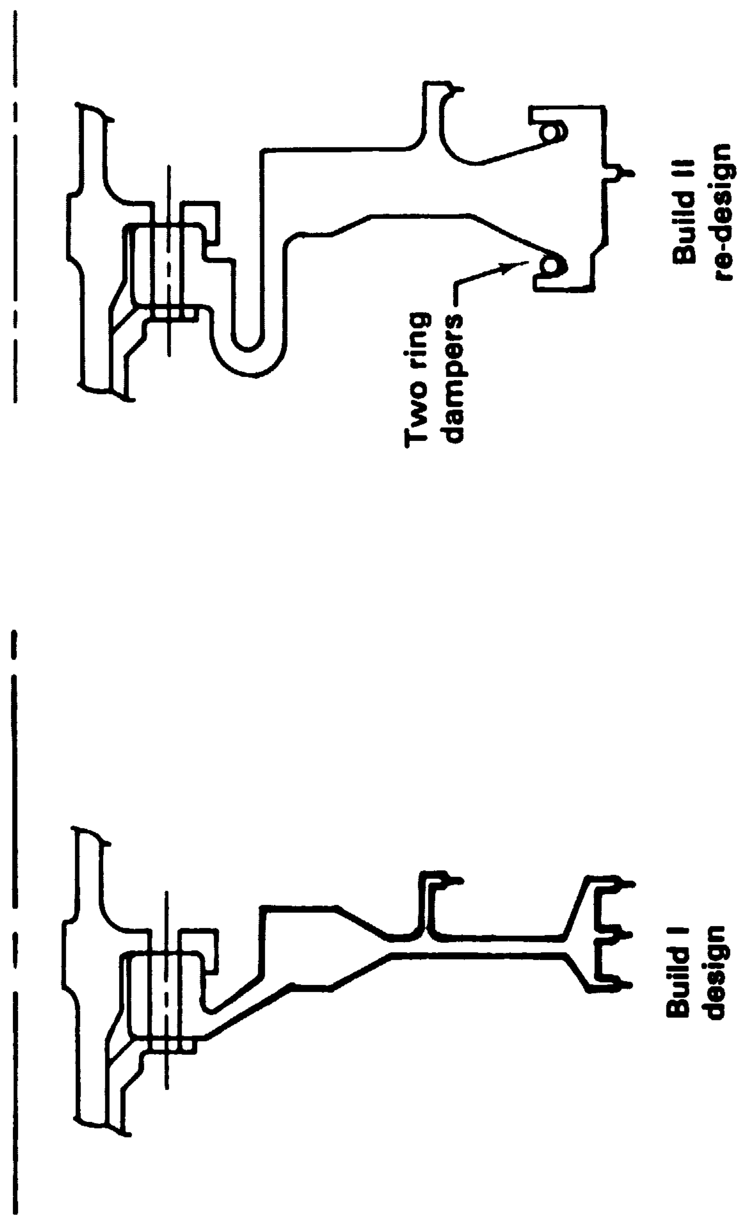

Figure 27 High-Pressure Compressor Rig Thrust Balance Seals Showing the Original Build I Design and the Redesigned Build II Configuration

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High-Pressure Compressor Rig (Build 2) Test: The test program commenced on 17 August 1981. The initial test series, consisting of evaluating all test stand and rig mechanical systems, was successfully completed. A preliminary analysis of data acquired from the stress and vibration survey portion of the rig test, in which acceleration to 105 percent rotor speed was accomplished, indicated acceptable stresses on the thrust balance piston throughout the rig running range. However, high vibration occurred on the front of the rig during this stress survey. Investigation after shutdown revealed that the rotating strain gage slip ring drive shaft separated from the front of the titanium rotor with subsequent damage to several parts in the front bearing compartment. The effected bearing compartment and drive shaft area is shown in Figure 28. Efforts to repair the damaged parts were initiated late in the report period.

HPC RIG SLIP RING DRIVE

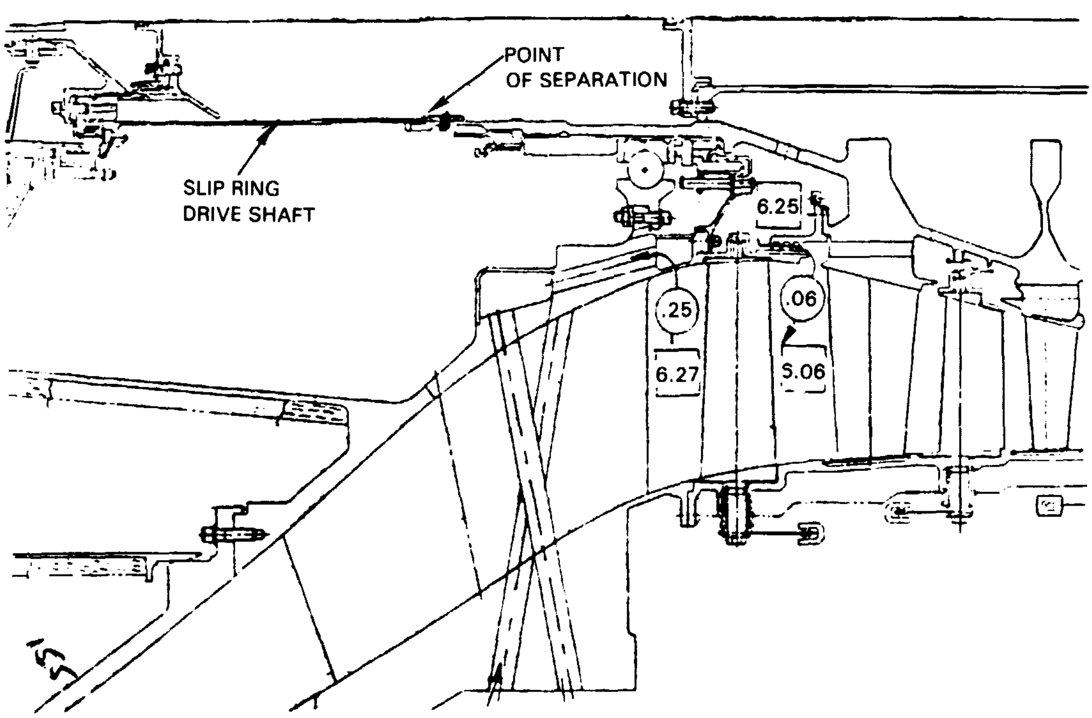


Figure 28 High-Pressure Compressor Rig Slip Ring Drive

3.2.6 Combustor

3.2.6.1 Overall Objective

Design a full-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 combustor liner featuring advanced wall cooling. The goals established for the combustor rig (see Table 19) are the same as those set for the flight propulsion system component.

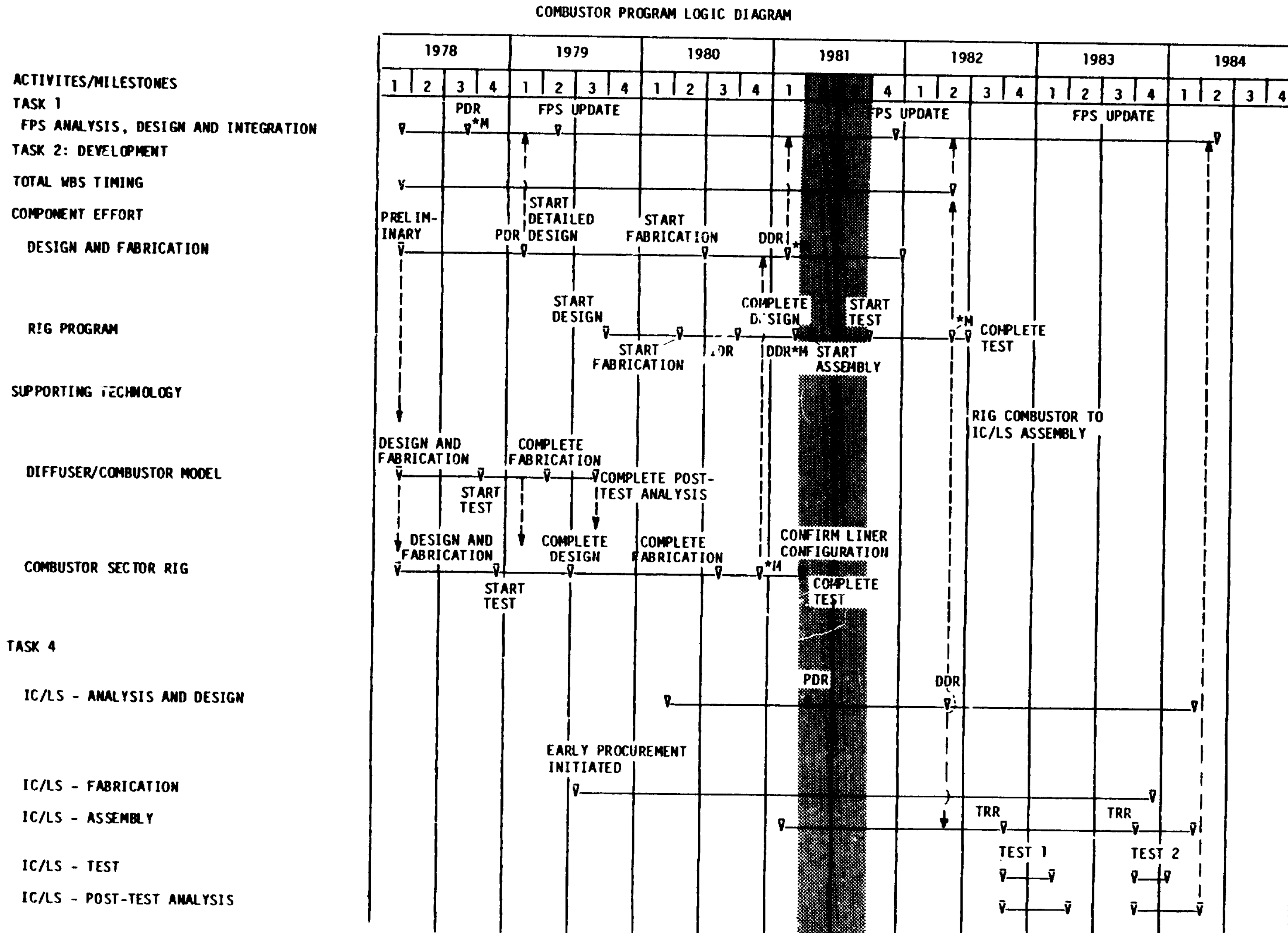
TABLE 19

GOALS ESTABLISHED FOR THE COMBUSTOR COMPONENT RIG

Pattern Factor, Maximum	0.37
Section Pressure Loss	5.5 percent P_{T3}
Hydrocarbon EPAP	0.4
Carbon Monoxide EPAP	3.0
NOx EPAP	3.0
SAE Smoke Number	20
Radial Profile	250 degrees average-peak
Liner Life	8000 hours, 4900 missions

3.2.6.2 Scope of Total Work Planned

The overall task effort consists of a component effort and two supporting technology sub-tasks. The component effort comprises the analysis and design of the combustor component and a combustor rig test program. The two supporting technology programs are (1) the diffuser/combustor model test program and (2) the combustor sector rig program. Figure 29 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 30 and critical milestones are noted.



*M DENOTES MAJOR MILESTONE *D DENOTES KEY DECISION POINT

Figure 29 Combustor Program Logic Diagram

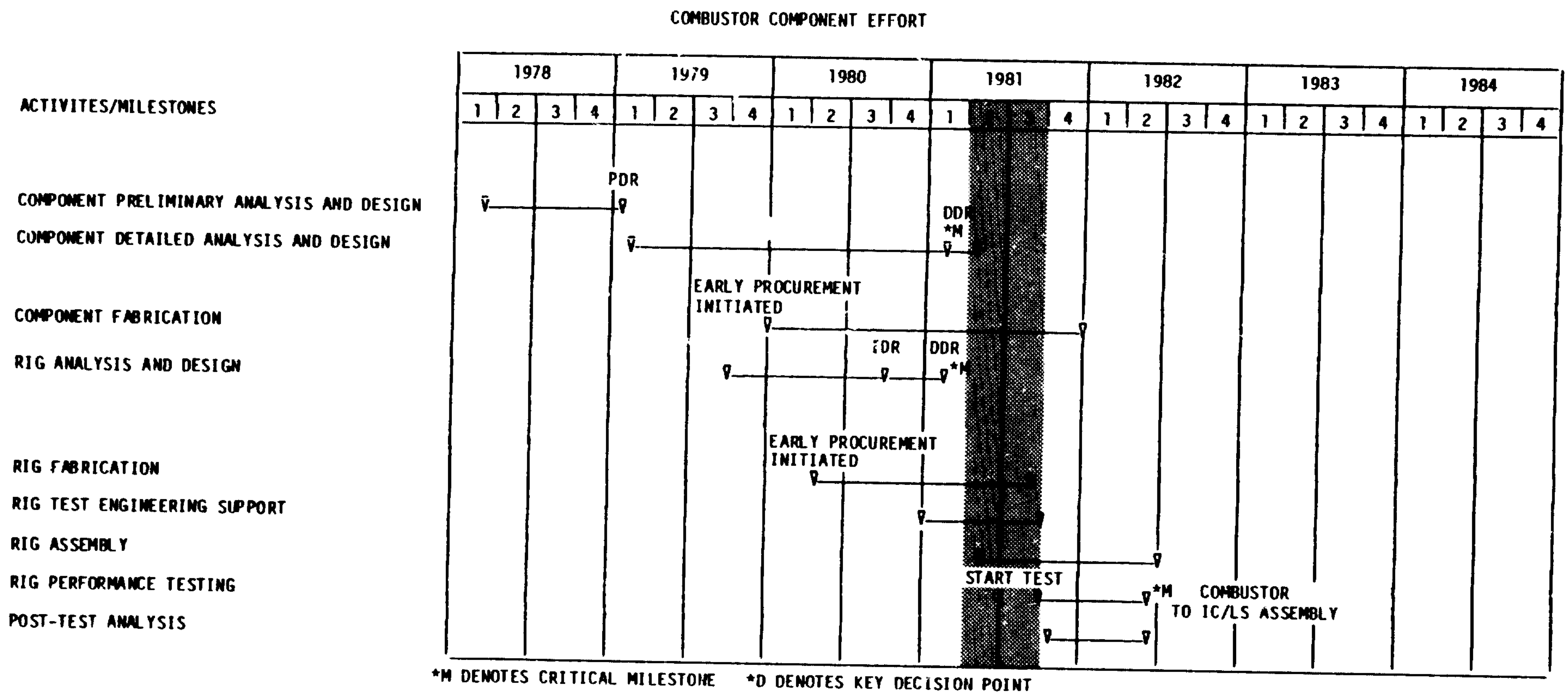


Figure 30 Combustor Component Effort Work Plan Schedule

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 testing activities necessary to develop a full-annular combustor that meets the program goals.

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 sub-tasks shown in Figure 30. A preliminary design activity is conducted to establish the feasibility of the combustor as proposed for the flight propulsion system. The studied designs provide configuration definitions to the supporting technology programs. This preliminary activity results in layout drawings and substantiation of design data which are presented to NASA at a preliminary design review in January 1979.

Detailed design activity starts in March 1979. Results available 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. The results of this effort are presented to NASA in a Detailed Design Review (DDR) in February 1981. Detailed drawings are scheduled for completion approximately two months later.

Design and fabrication of combustor rig parts progresses 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 configuration are tested to develop the final configuration that satisfies program goals. Testing of the various configurations consists mainly of air schedule variations to demonstrate emissions, exit radial temperature profile, performance goals, and durability. In May 1982, the final diffuser/combustor configuration is transferred to the first build integrated core/low spool assembly effort.

All of the work planned and approved from contract award through the end of the current reporting period (30 September 1981) has been completed. Figure 30 indicates that all component and rig design activities have been completed and that rig fabrication efforts were near completion at the end of 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 during prior reporting periods. A rig design review was held at NASA-Lewis Research Center in September, 1980. This was followed by the component detaled design review in February, 1981.

The combustor component is illustrated in Figure 31, and its companion full annular rig is shown in Figure 32. The basis for the component design was the two-stage combustor investigated in the NASA Experimental Clean Combustor Program (ECCP). The design therefore incorporates two distinct burning 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.

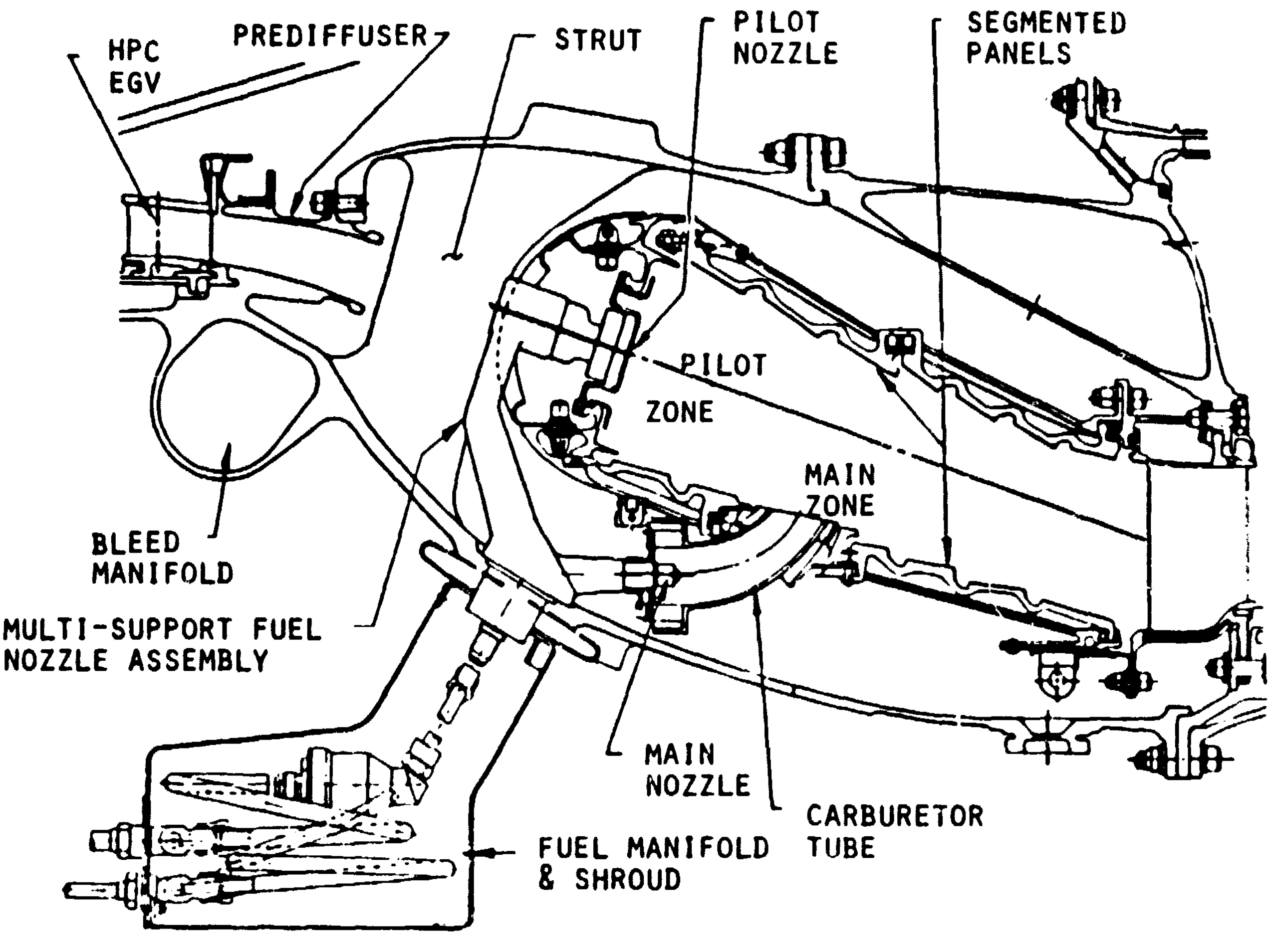


Figure 31 Combustor Component

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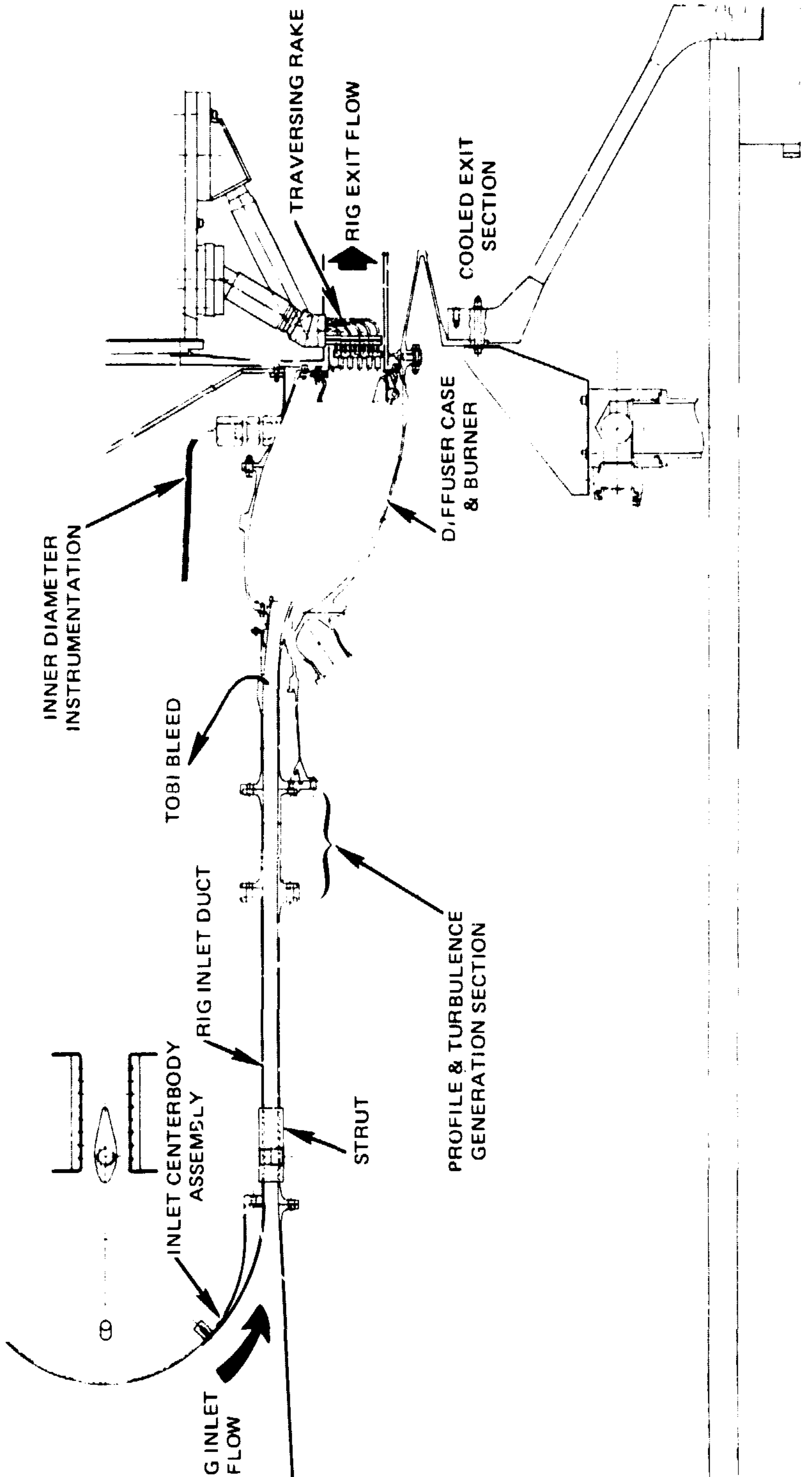

Figure 32 Combustor Component Full Annular Rig

The design of the compressor exit guide vane assembly was included in the design of the combustor component because of its interaction with the prediffuser duct. The assembly is shown in Figure 33. This configuration features: (1) a vane having 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 exit guide vane and the inner prediffuser duct wall, and (4) feather seals to control air leakage through the gaps between segments.

The exit guide vane design shown in Figure 33 incorporates a single row airfoil. 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 Build 2 high-pressure compressor rig.

Current performance parameters for the combustor component at significant engine operating conditions are shown in Table 20.

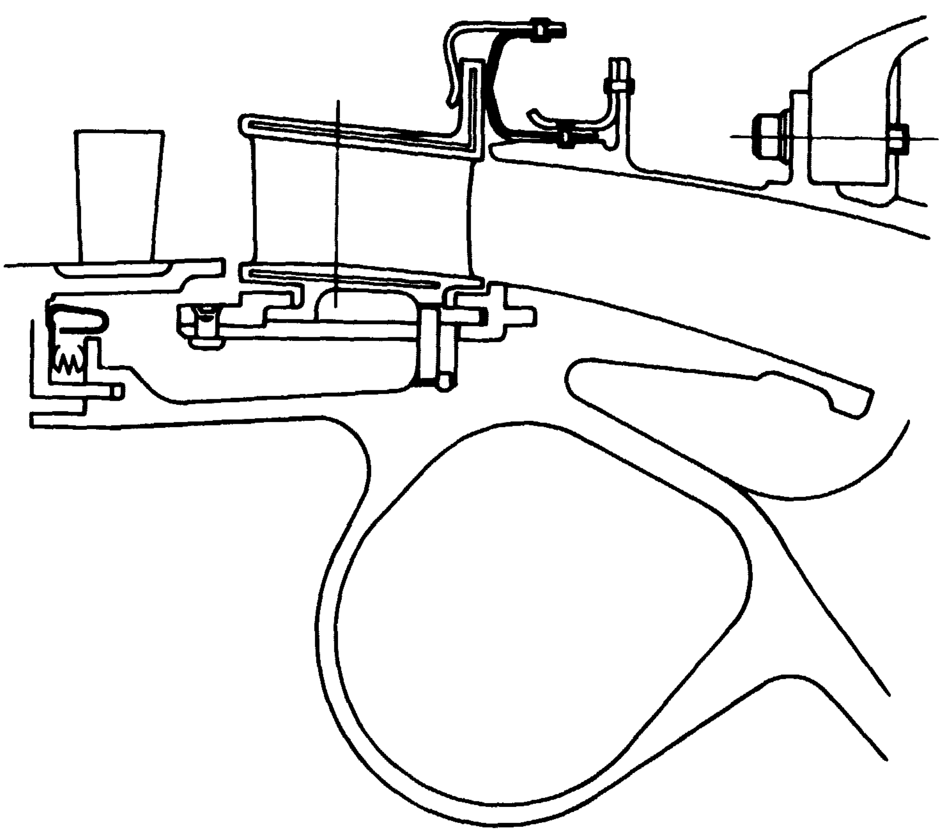

Figure 33 High-Pressure Compressor Exit Guide Vane Assembly

TABLE 20

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 (lb/in^2 Abs)	203	197	212	456
Inlet Temperature (^{o}F)	898	883	976	1060
Section Pressure Loss (Percent)	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 (Percent)	99.95	99.95	99.95	99.95

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

Component Hardware: Initiated fabrication of the diffuser case, fuel nozzle and support assembly, inner combustor case, tangential on-board injector (TOBI) duct, high-pressure compressor seal and seal support, advanced segmented liners, inner and outer combustor support frames, combustor bulkhead, and main zone carburetor tubes.

Rig-unique Hardware: Completed fabrication of the transtional ducts to the diffuser case and initiated fabrication of the air inlet ducts, tangential on-board injection (TOBI) bleed system, inner rear combustor case, inner and outer support rings, exit section heatshields, station 4.0 rakes, and station 3.0 pressure and temperature probes.

3.2.6.3.3.2 Current Technical Progress

Combustor Component Fabrication

The fabrication of combustor component hardware, required as part of the combustor component full annular test rig assembly, was almost completed during the current reporting period; the exception being the fuel manifold sealing shroud and the exit guide vane (EGV) seal assembly. The fuel manifold sealing shroud is scheduled for completion early in the next reporting period during the initial phase of component fuel system assembly for the full annular rig test program. The fabrication effort directed toward detail pieces of the exit guide vane seal assembly, not required to support the rig testing effort, is progressing on a low priority basis. This assembly is scheduled to be completed and available for integrated core/low spool testing planned for 1982. A discussion on the major items of the combustor component appears below.

Diffuser Case Assembly: The diffuser case assembly requires the welding of forged Inconel 718 front and rear sections to the Inconel 718 detail case casting prior to final machining. The welding and final machining operations were completed by the machining vendor during the reporting period. The finished assembly (Figure 34) was delivered to the assembly floor area for incorporation into the combustor component rig assembly.

Combustor Bulkhead and Hood Assemblies: In-house fabrication of these two assemblies is complete. The parts have been delivered to the assembly floor area for incorporation into the combustor component rig assembly. Figure 35 shows the completed combustor bulkhead assembly.

Inner Combustor Case: In-house fabrication of the inner combustor case assembly was completed during the current reporting period. This case assembly will be utilized in the development testing of the high-pressure turbine component rig, but is not required for the full annular combustor rig test program.

Tangential On-Board Injection (TOBI) Duct. Vendor fabrication of the tangential on-board injection air duct assembly was completed during the current reporting period. This assembly (Figure 36) will be utilized in the development testing of the high-pressure turbine component rig, but is not required for the full annular combustor rig test program.

Exit Guide Vane Seal Assembly: As mentioned previously, the fabrication effort directed toward detail pieces of the exit guide vane seal assembly, not required to support the rig testing effort, is progressing on a low priority basis. This assembly is scheduled to be completed and available for integrated core/low spool testing planned for 1982.

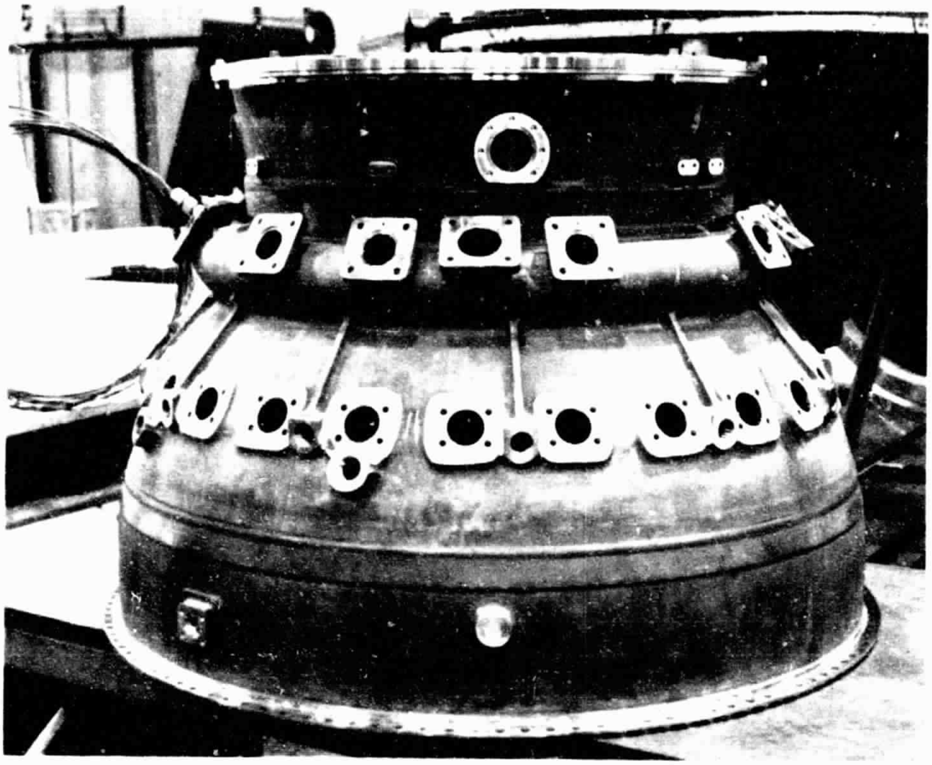

Figure 34 Combustor Component Diffuser Case Finished Assembly

Figure 35 Completed Combustor Component Bulkhead Assembly

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Figure 36 Tangential On-Board Injection Air Duct Assembly for use in Development Testing of the High-Pressure Turbine 'Warm' Rig

Fuel Nozzle Support Assembly: The fuel nozzle support casting, modified to eliminate a potential interference problem with the outer combustor support frame, was completed and finish machined during the current reporting period. The fuel nozzle support assemblies, shown in Figure 37, were delivered to the assembly floor area for incorporation into the combustor component rig assembly.

Carburetor Tube Assembly: Carburetor tube casting tooling was modified to correct previously encountered core shift problems. Detail castings were successfully poured and provided to a vendor for final machining and braze assembly of the secondary discharge tube to the main body of the carburetor tube. The finished carburetor tube assemblies shown in Figure 38 were delivered to the assembly floor for incorporation into the combustor component rig assembly.

Ignitor Plugs: Vendor fabrication of ignitor plugs for the combustor was completed during the reporting period. The parts were delivered to Pratt & Whitney Aircraft for incorporation into the combustor component rig assembly.

Fuel Jumper Tubes and Manifolds: Vendor fabrication of the combustor fuel system's fuel supply jumper tubes and pilot manifolds was completed during the current reporting period. However, trial fit revealed a design error in locating the fuel nozzle support fuel inlet fitting during design layout of the fuel tubes. Some minor modification to the pilot zone fuel jumper tubes is required. New jumper tubes will be fabricated to fit between the main zone pressure equalizing valves and fuel nozzle support. This work will take place early in the next reporting period.

Fuel Manifold Sealing Shroud: In-house fabrication of the fuel manifold sealing shroud is scheduled for completion in the next reporting period. This hardware item will be trim-to-fit during the initial phase of component fuel system assembly for incorporation into the combustor component assembly prior to rig testing.

Advanced Combustor Liner Segments. The in-house fabrication of the first set of advanced segmented combustor liners was completed during this reporting period. Typical liner segments are shown in Figures 39 and 40. The first set of advanced liner segments has been delivered to the assembly floor area for incorporation into the combustor component rig assembly.

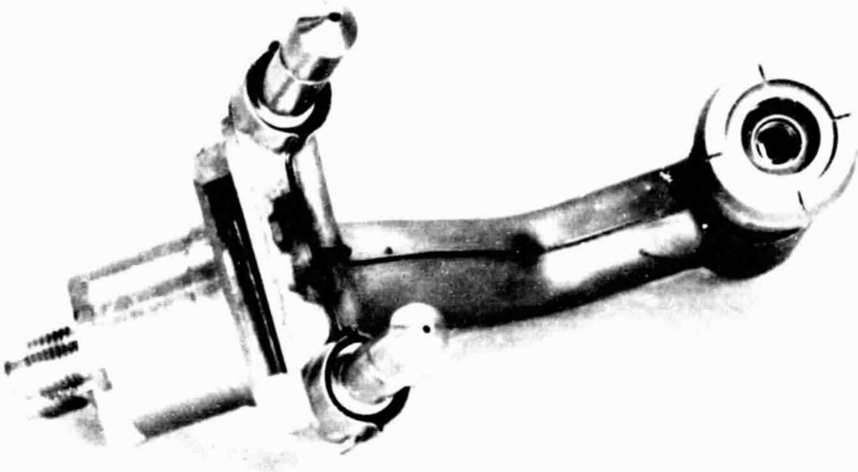


Figure 37 Fuel Nozzle Support Assembly

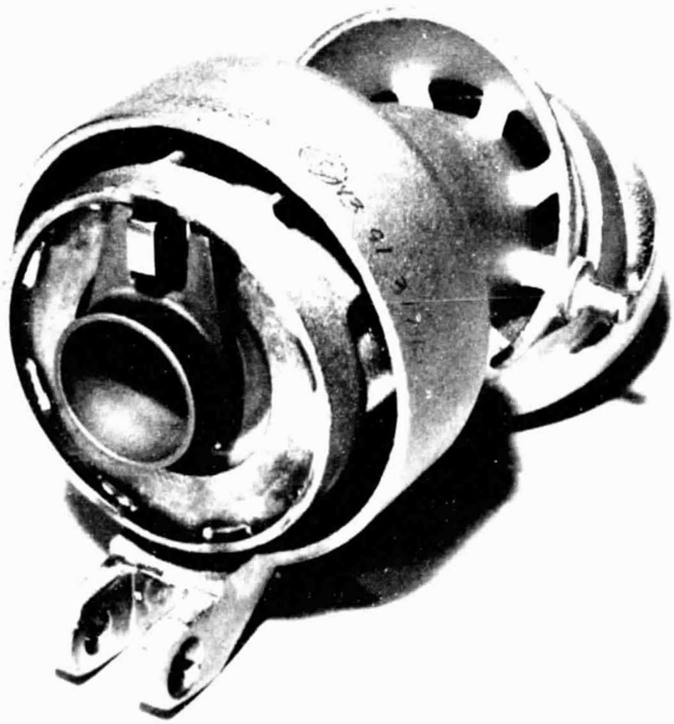
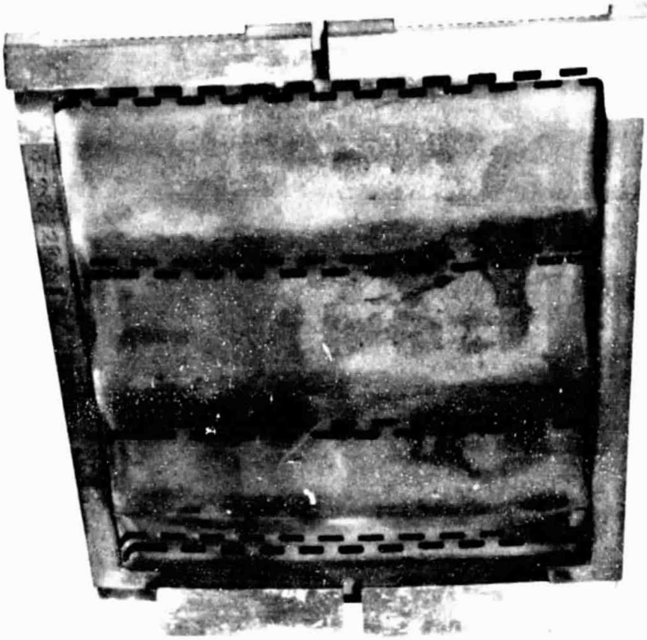


Figure 38 Finished Carburetor Tube Assembly



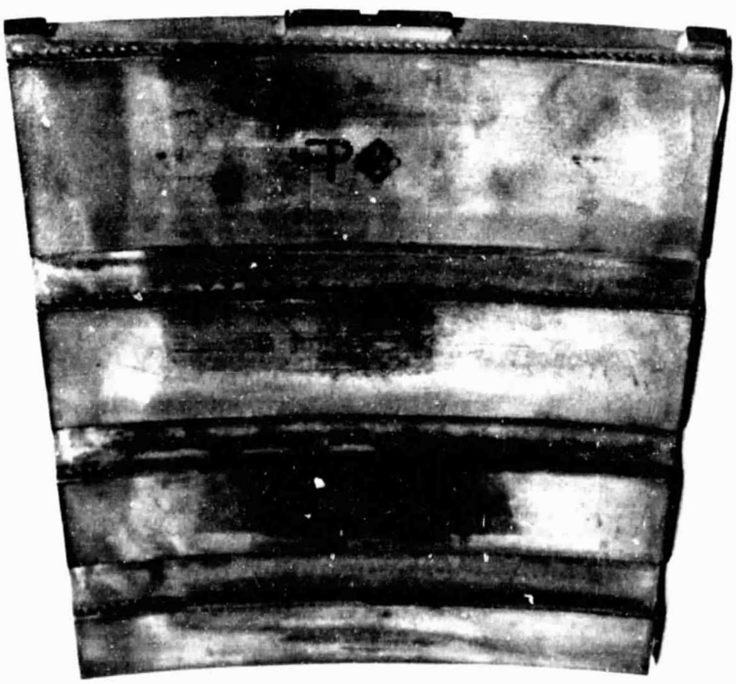


Figure 39 Combustor Component Outer Rear Advanced Liner Segment

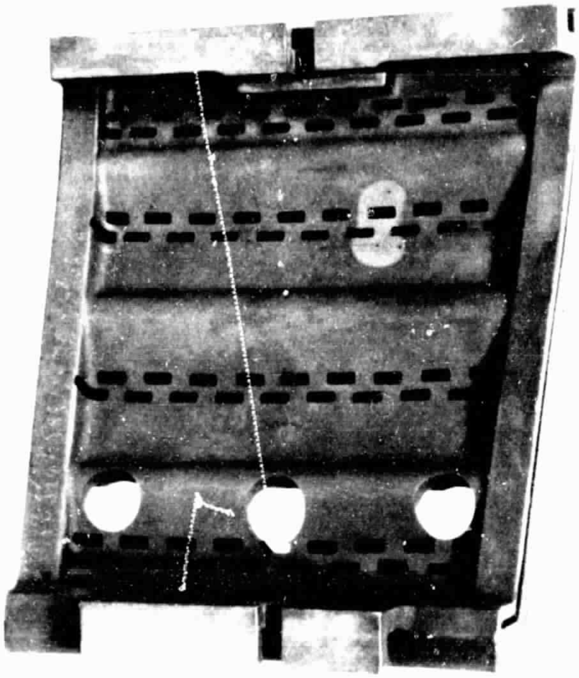


Figure 40 Combustor Component Inner Rear Advanced Liner Segment

Combustor Support Frames: In-house machining of the inner and outer AMS 5754 combustor liner segment support frame forgings was completed during the reporting period. These hardware items have been delivered to the assembly floor area for incorporation into the combustor component rig assembly. Figure 41 shows the inner combustor frame and Figure 42 shows the outer combustor support frame.

Combustor Rig Fabrication Effort

Fabrication of rig-unique hardware for the full annular combustor rig assembly was completed during this reporting period. All hardware items have been delivered to the assembly floor area. These include the outer flow path ducting, including the rig inlet case, and the inner flow path ducting, including the rig inlet case. Figure 43 shows a total pressure and total temperature rake for station 3.0. These probes will be installed in the diffuser case assembly. Figure 44 shows a total pressure and emissions sampling rake along with a total temperature rake for station 4.0.

Combustor Component Rig Assembly

Assembly of the combustor component rig was initiated during the current reporting period. All mating rig hardware was trial-fitted to insure proper bolt hole alignment. Surface irregularities on the inner flowpath duct (aft of the inlet case) were removed by machining. This condition resulted from weld distortion during fabrication. Static pressure sensing instrumentation was installed in the rig inlet ducting. Trial assembly of the diffuser/combustor identified several minor interferences that were subsequently corrected.

Figure 45 shows the diffuser case assembly with several fuel nozzle supports and station 3.0 pressure and temperature probes installed. Figure 46 shows an inner combustor support frame with several advanced liner segments installed. An outer combustor support frame with several advanced liner segments installed and several carburetor tube assemblies mounted is shown in Figure 47 while Figure 48 shows the combustor bulkhead mated to the inner and outer combustor support frames.

A typical instrumented advanced liner segment for the combustor is shown in Figure 49. Installation of thermocouples used to measure advanced liner segment metal temperatures was completed at the end of the reporting period along with instrumentation of fuel nozzle supports, diffuser case, and the combustor bulkhead.

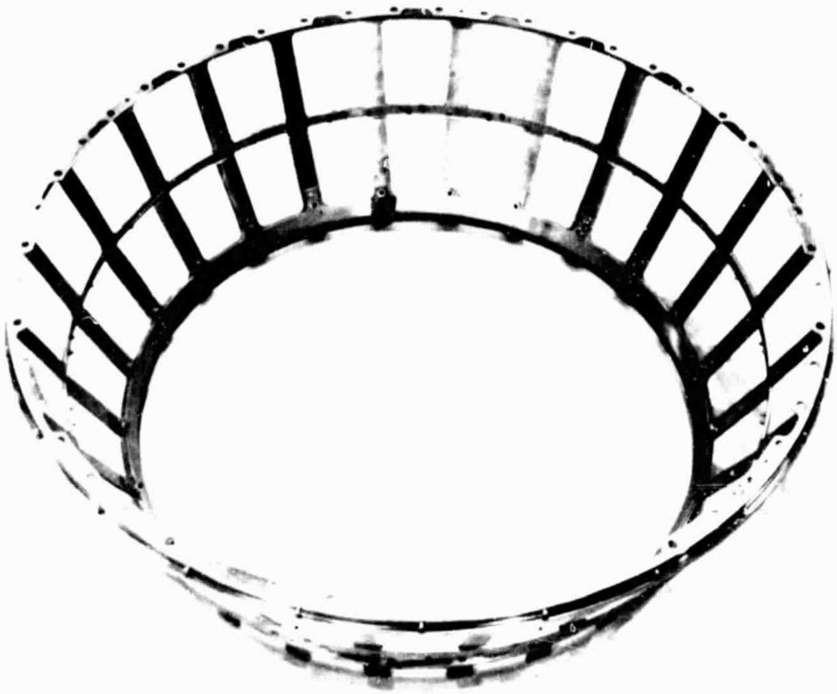


Figure 41 Combustor Component Inner Combustor Advanced Liner Support Frame

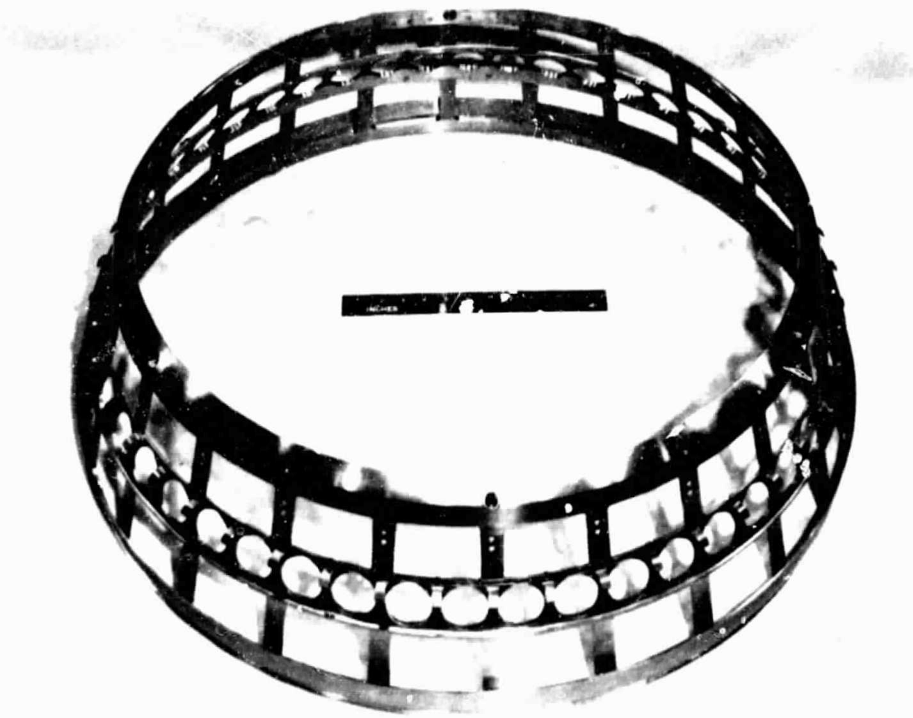


Figure 42 Combustor Component Outer Combustor Advanced Liner Support Frame

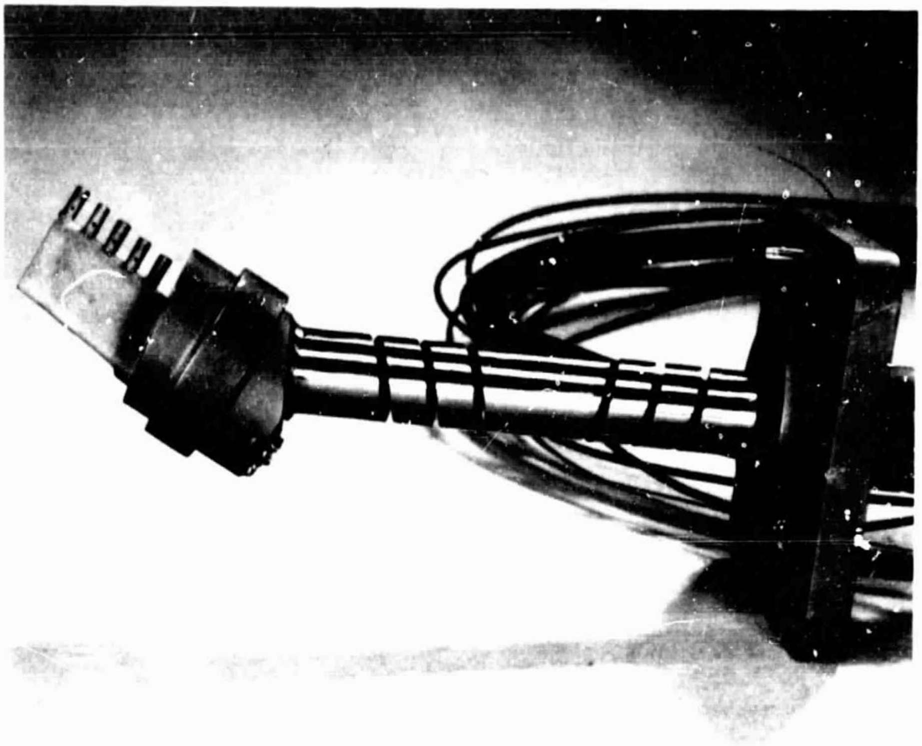


Figure 43 Rig Instrumentation - Station 3.0 Total Pressure and Total Temperature Rakes

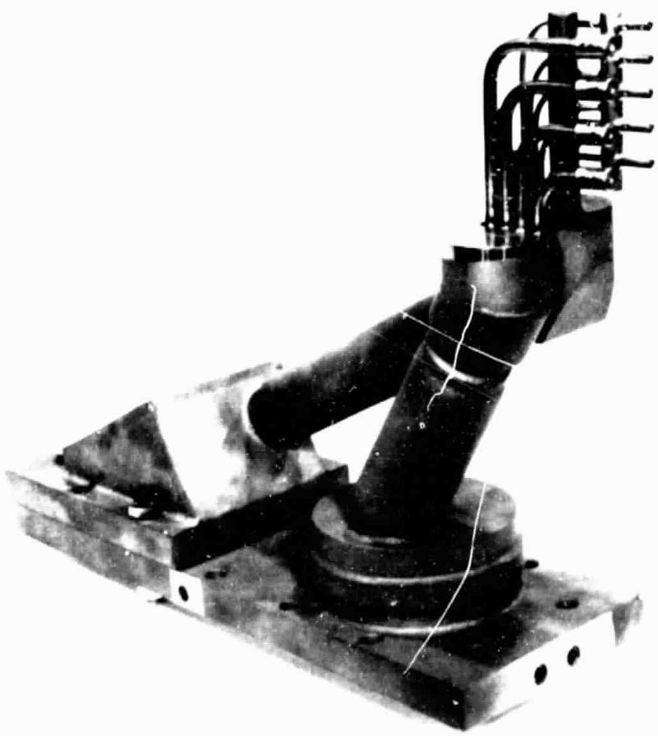


Figure 44 Rig Instrumentation - Station 4.0 Total Temperature Rake and Total Pressure and Emissions Sampling Rake

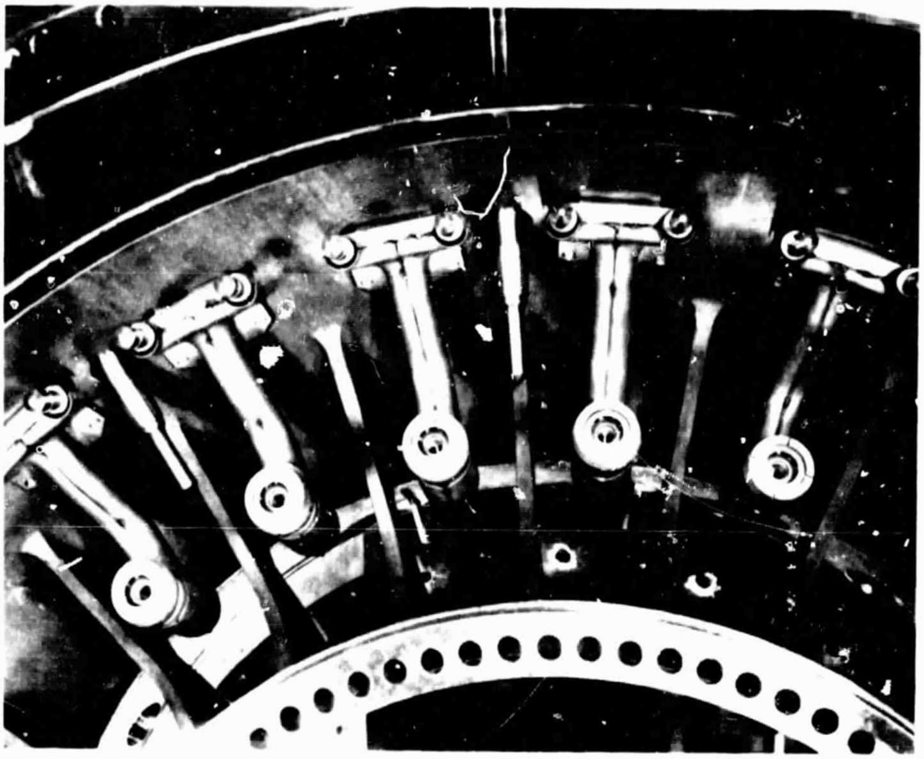


Figure 45 Combustor Component Diffuser Case Assembly With Several Fuel Nozzle Supports and Station 3.0 Pressure and Temperature Probes Installed

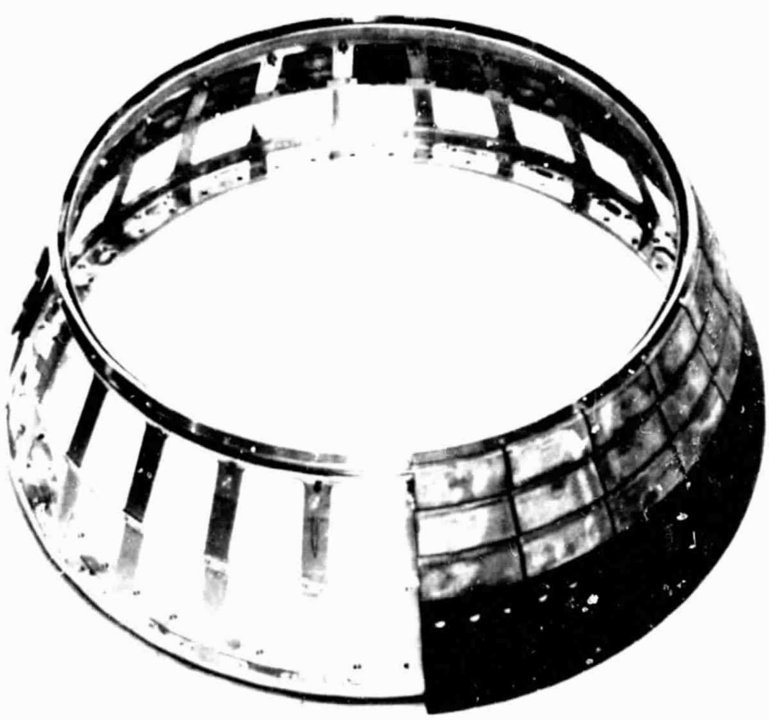


Figure 46 Inner Combustor Support Frame with several Advanced Liner Segments Installed

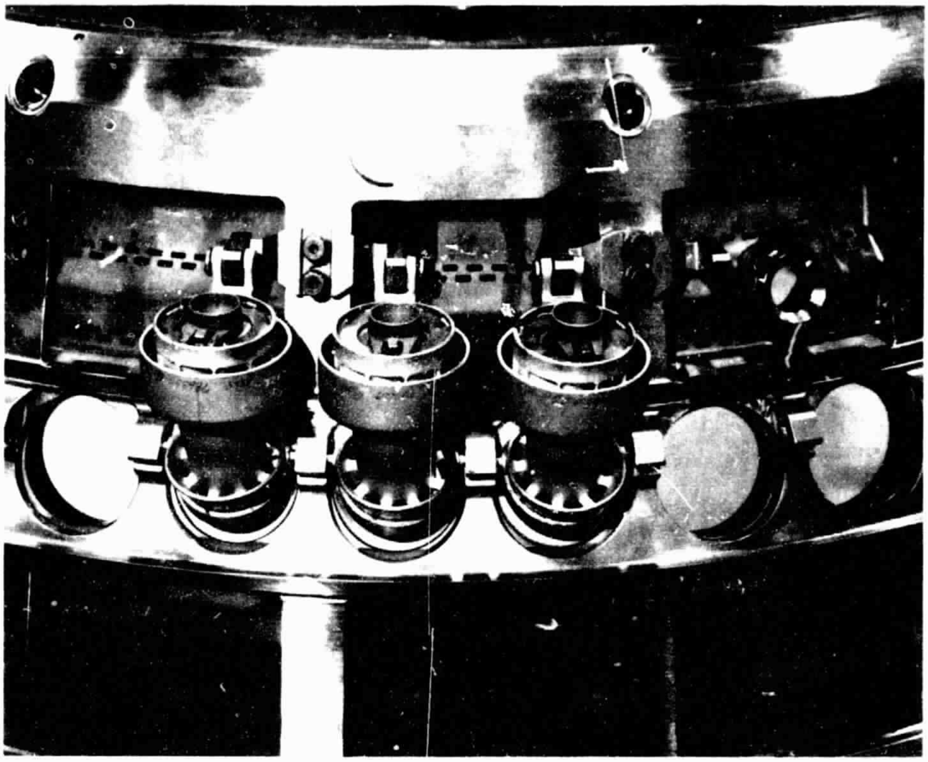


Figure 47 Outer Combustor Support Frame with several Advanced Liner Segments Installed and several Carburetor Tube Assemblies Mounted

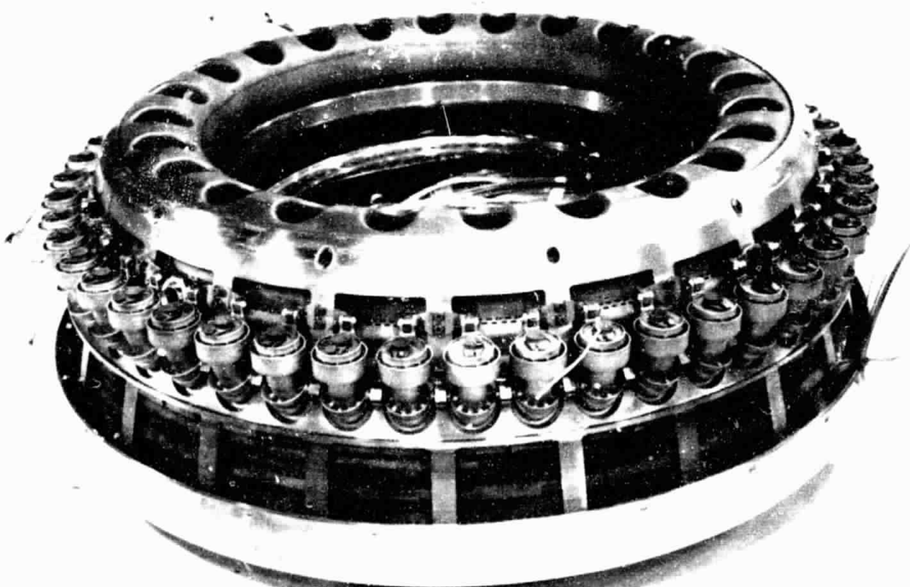


Figure 48 Combustor Bulkhead Mated to the Inner and Outer Combustor Support Frames

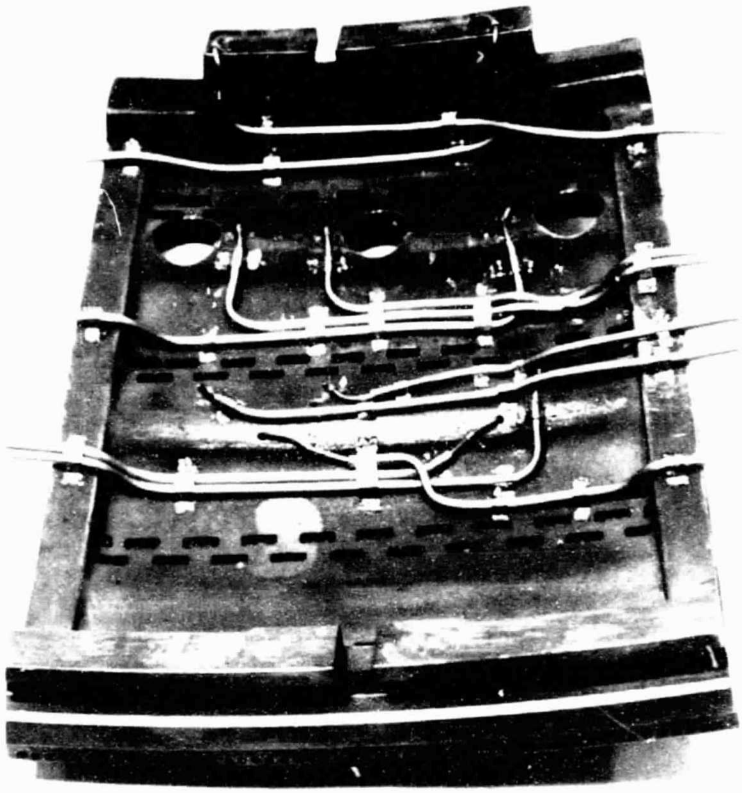


Figure 49 Combustor Component Instrumented Advanced Liner Segment

Rig hardware from the combustor sector rig supporting technology program has been transferred to the combustor component rig assembly effort. Minor modifications to this hardware are currently underway to incorporate the cast carburetor tubes. Both the conventional louver liner and advanced liner sector rigs will utilize the cast carburetor tube.

Combustor Component Rigs Performance Test

The combustor component test and instrumentation plan, in support of the integrated full annular and sector rig test programs, was prepared and submitted to NASA for review and was approved during the current reporting period. Program support effort for data acquisition computer system programming and rig operational procedures is continuing.

3.2.6.4 Supporting Technology

3.2.6.4.1 Diffuser/Combustor Model Test Program

The objective of this supporting technology program was to experimentally optimize and document the aerodynamic performance of the diffuser/combustor system employing an outboard-canted combustor located downstream of a strutless, curved-wall diffuser. All technical work for this supporting technology program was completed in a prior reporting period and results appear in NASA Report CR-165157.

3.2.6.4.2 Combustor Sector Rig Test Program

3.2.6.4.2.1 Objective

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 are aimed at reducing the emissions, pattern factor, and cost and weight as well as improving the durability and maintainability of the combustor section. Specific emissions and performance goals are the same as those of the combustor component.

3.2.6.4.2.2 Scope of Total Work Planned

The combustor sector rig test program consists of the five phases shown in Figure 50, which indicates that analysis and design, fabrication, and assembly were completed prior to the current report period. Subsequent testing and post-test analysis were completed during the current reporting period. A modular high-pressure test rig representing a 90-degree sector of the full annular engine diffuser/combustor section is designed, fabricated, and assembled. This rig includes the transition duct (circular to sector cross section) inlet section, diffuser case, combustor, and instrumentation used to sample pressure, temperature, and exhaust gas. Modular design features are employed to facilitate variation of prediffuser contour, diffuser case strut geometry, prediffuser dump/combustor front-end geometry, and combustor pilot and main zone geometry. The ability to vary the number and type of main zone fuel injectors is also incorporated into the design. A second diffuser case/combustor assembly is also fabricated. The fabrication and assembly efforts for the rig and the second diffuser case/combustor assembly are phased to permit modification of one combustor while the other is being tested. A third combustor liner assembly featuring segmented liners and advanced cooling techniques is also fabricated.

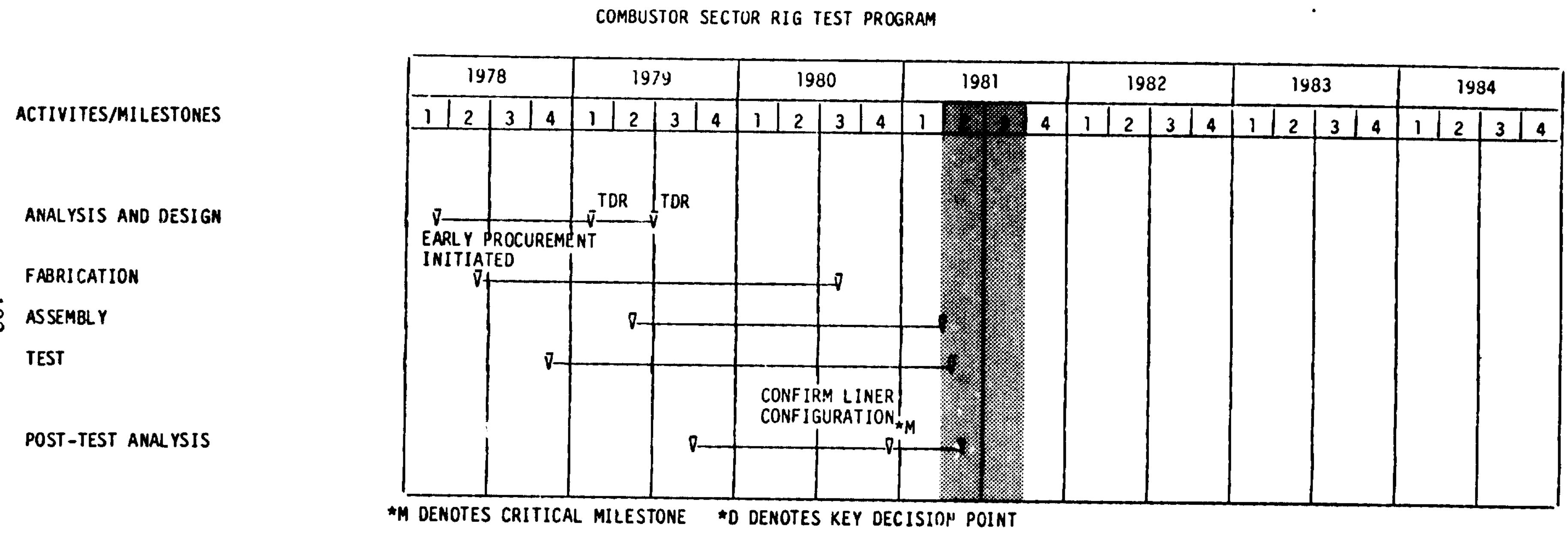


Figure 50 Combustor Sector Rig Test Program Work Plan Schedule

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Testing includes pilot zone optimizations, changes in temperature history, variation in number of fuel injectors and fuel spray characteristics, and combustion air variations in the pilot and main zone to regulate exit temperature profiles and emissions. Tests consist of cold flow pressure loss measurements, parametric idle testing with only the pilot zone fuel injectors flowing, definition of lean blow-out characteristics, and pilot/main zone fuel split variations to minimize emissions.

Performance and emissions data are recorded at every test point by utilizing an automatic data recording system. Following definition of an optimum combustor, utilizing the high-pressure facility, the sector rig is transferred to the altitude relight facility and tests conducted to determine the altitude relight and stability characteristics of the combustor.

Processed data are analyzed in depth following a test sequence with a particular configuration to evaluate the status of the emissions, performance, and durability characteristics relative to the program goals. Modifications of the rig hardware are then formulated to improve deficient areas.

3.2.6.4.2.3 Technical Progress

3.2.6.4.2.3.1 Summary of Work Previously Completed

The baseline sector rig assembly with an instrumented vane pack installed in the exit plane is shown in Figure 51. The liners used in early rig testing are of conventional louver construction, employing Hastelloy X, to expedite configuration changes during the primary test program. The vane pack assembly consists of eight instrumented vanes, four pressure vanes and two 'dummy' vanes.

Following the change to segmented liners in the component design, the sector rig combustor design was revised to be compatible with the component design. This revised combustor assembly, featuring the inner and outer liner support frames and the four types of segments, is shown in Figure 52.

Sixteen sector rig tests were conducted using the conventional louvered design. The tests consisted of (1) a shakedown test, (2) pilot zone injector comparison tests, (3) baseline performance tests, and (4) development tests exhibiting evolutionary changes to the baseline configuration. The performance characteristics of the most promising configuration are summarized in Table 21. The emissions parameters include margins for variability and development. The NOx parameter is based on a pilot zone fuel/air ratio of 0.003 at climb and sea level takeoff conditions. All emissions data for climb and takeoff conditions were scaled for inlet pressure effects.

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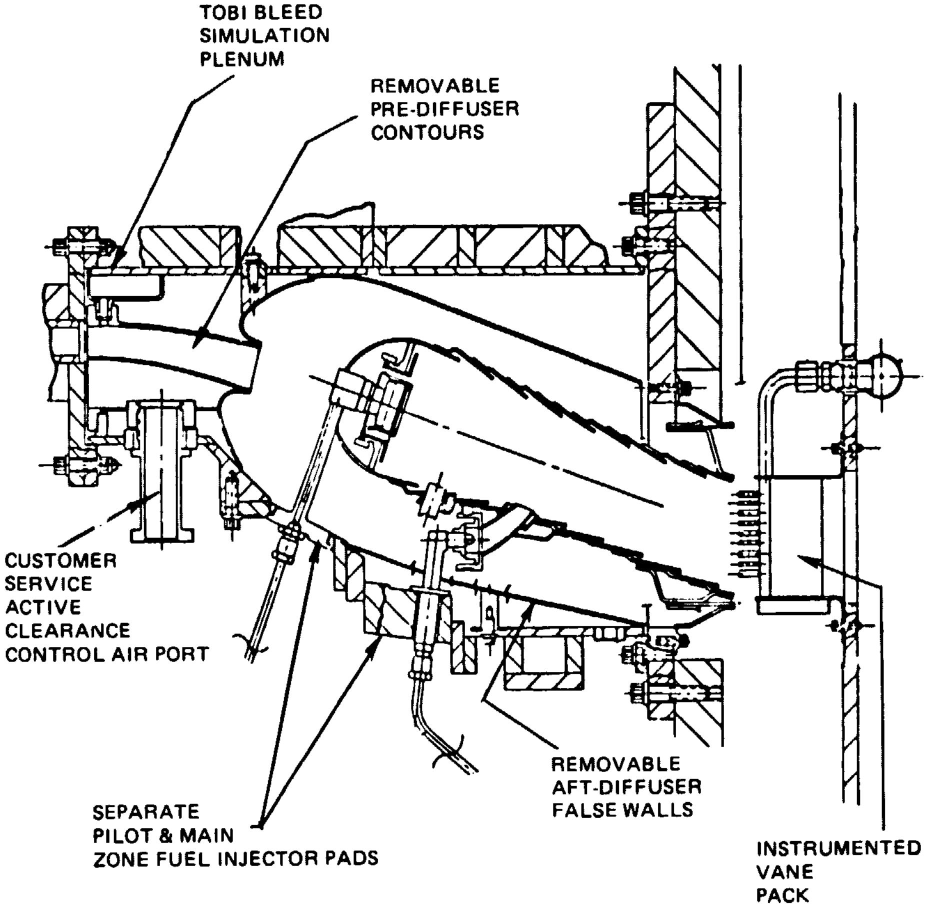

Figure 51 Combustor Sector Rig Cross Section - The combustor liners for rig application are of conventional louver construction with instrumented vane pack installed in the exit plane

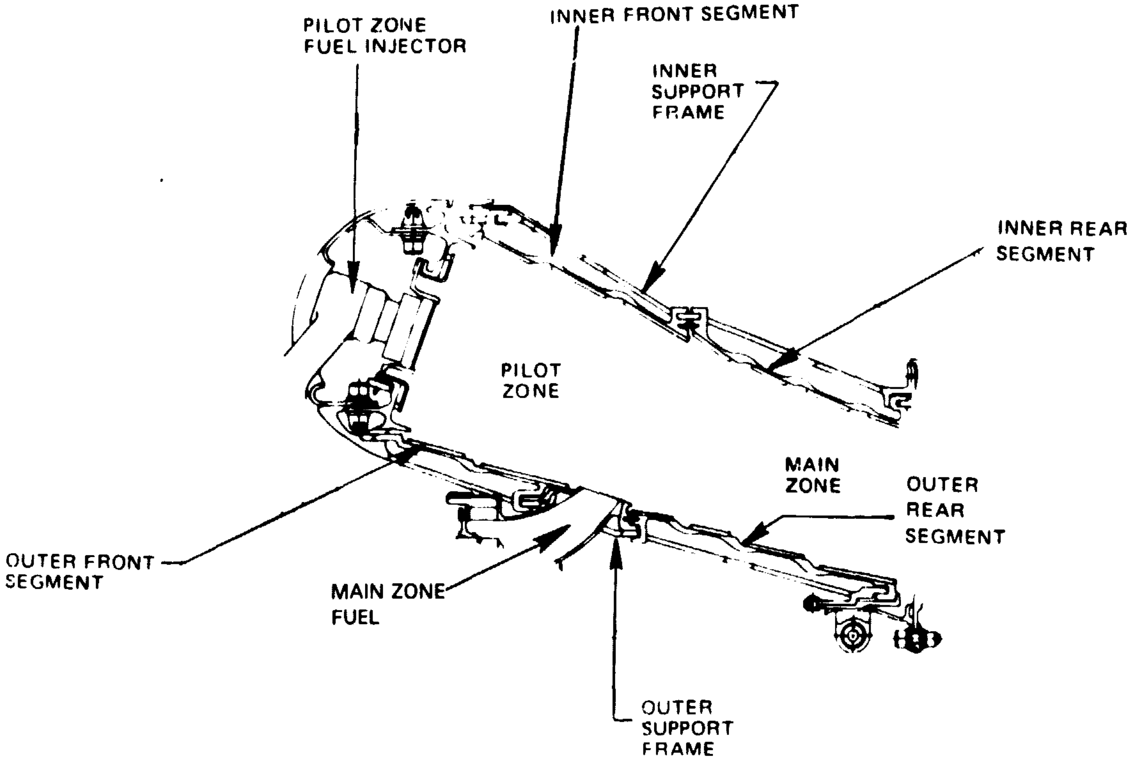

Figure 52 Revised Combustor Assembly with Segmented Liners

TABLE 21

SECTOR RIG CANDIDATE COMBUSTOR PERFORMANCE SUMMARY

Pressure Drop (% Pt3)		Environmental Protection Agency parameters	
Section	: 5.37	Carbon Monoxide (CO):	2.07
Outer Liner	: 2.34	Total Unburned Hydrocarbons (THC):	0.26
Inner Liner	: 3.05	Oxides of Nitrogen (NOx):	4.65
		Smoke No.·	5

Pattern Factor: 0.18
Radial Temperature Profile: 70°F (peak-to-average at 50% span)

All program performance and emission goals (except NOx emissions) and the altitude relight requirements were achieved with this configuration. The pilot and main zone fuel injector characteristics were incorporated into the component detailed design. Revised combustor air flow distribution, characteristic reference velocities, and liner areas are shown in Figure 53.

The altitude relight and sea level start characteristics of the Energy Efficient Engine combustor were evaluated using the most promising configuration and two candidate pilot nozzle designs. Testing was conducted at combustor inlet conditions representative of the compressor windmilling over the Energy Efficient Engine flight envelope. Fuel flow was varied at each condition to determine the minimum level of fuel flow required for ignition. The altitude relight results for the best nozzle design are presented in Figure 54. As shown over the range of conditions representative of the flight envelope, ignition was achieved up to an altitude of 35,000 feet with fuel flows as low as 48 lb/hr, both of which exceeded the required envelope.

At sea level conditions (see Figure 55), ignition was achieved at fuel flows of 192 lb/hr, exceeding Energy Efficient Engine requirements. Although both nozzle designs exhibited similar altitude relight capabilities, sea level start requirements could only be satisfied with one design.

Three sector rig tests were conducted using the advanced segmented liners. These tests consisted of an initial 'shakedown' test and two development tests directed at demonstrating the advanced liner concept at full engine operating conditions (maximum operating pressure extended up to the 400 psia level).

COMBUSTOR AIRFLOW DISTRIBUTION (BASED ON SECTOR RIG TEST RESULTS)

o ALL FLOWS ARE IN PERCENTAGE OF COMBUSTOR AIRFLOW (W_{A4})

o REFERENCE VELOCITIES BASED ON SECTOR RIG SIMULATED SLTO CONDITIONS

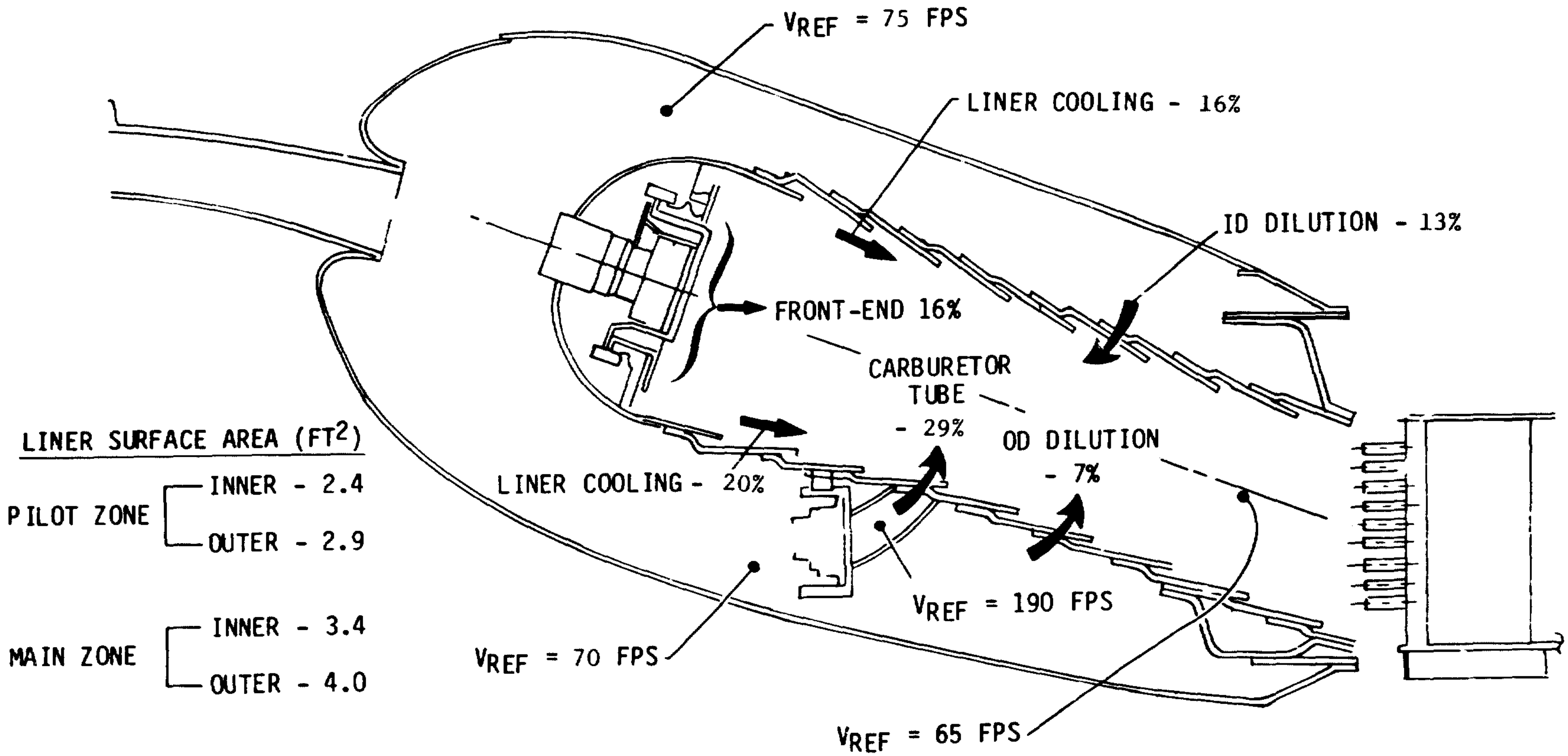

Figure 53 Revised Combustor Airflow Distribution, Characteristic Reference Velocities, and Liner Areas

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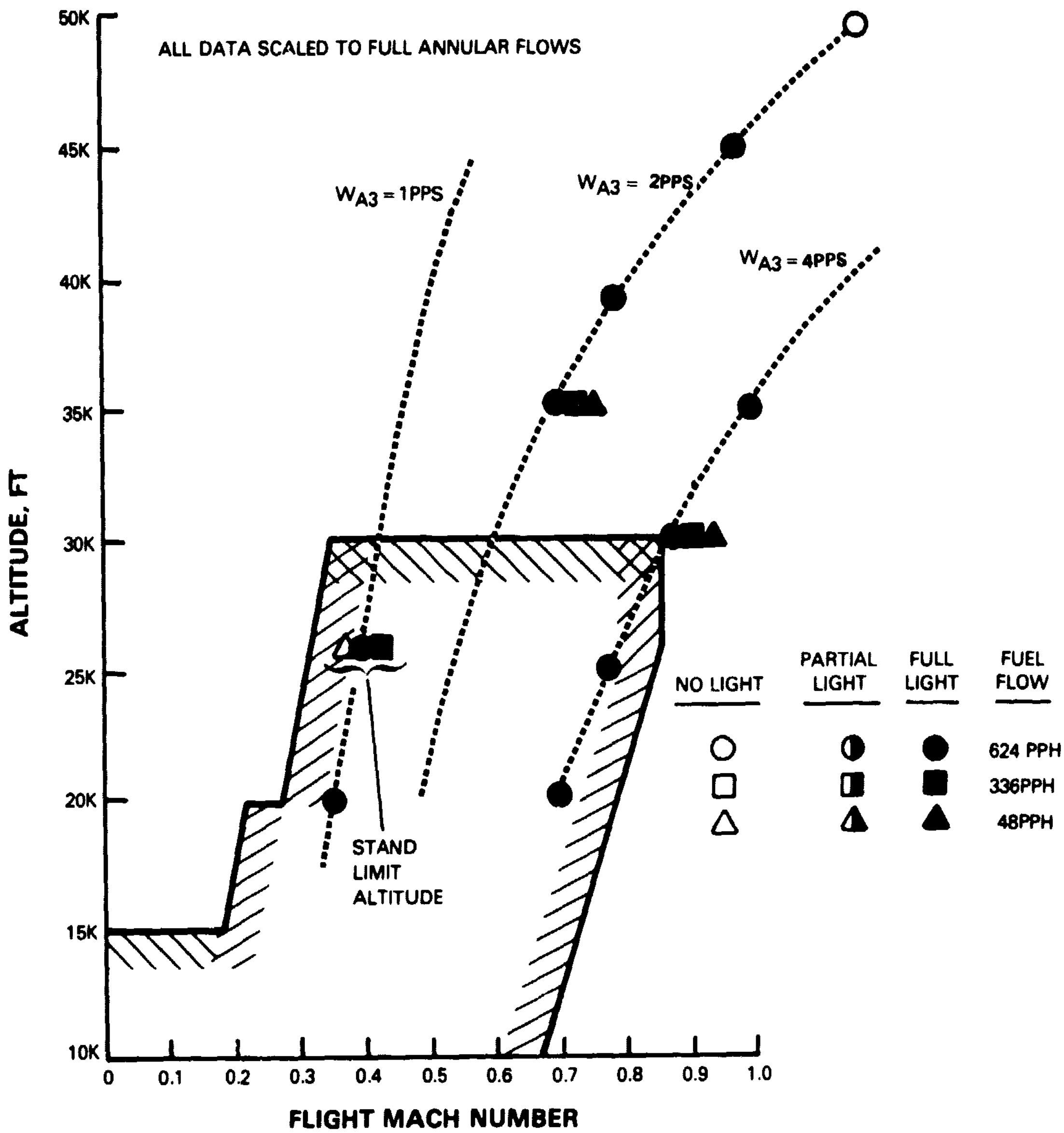

Figure 54 Altitude Relight Results

PRATT & WHITNEY AIRCRAFT GROUP
COMMERCIAL PRODUCTS DIVISION

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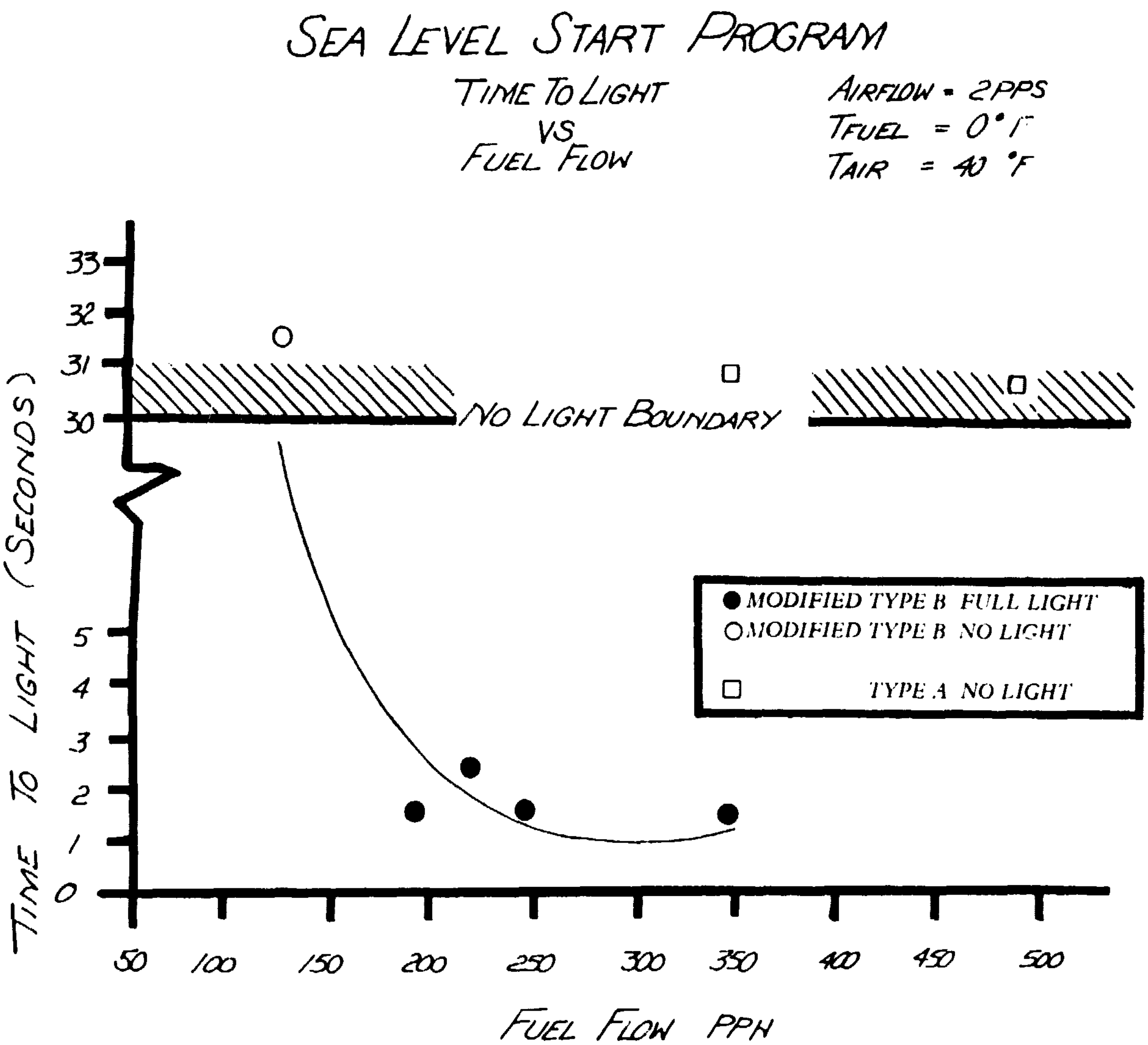

Figure 55 Altitude Relight Results (Sea Level Takeoff)

As shown in Figure 56, emissions levels for the advanced segmented liner configuration were essentially equal to those emissions levels attained by the best louvered liner configuration. All emissions and smoke goals were achieved except for the NOx EPA parameter which remained approximately 50 percent above design level goal. The exit radial temperature profile, as shown in Figure 57, was similar to the louvered liner configuration except for the overcooled region near the OD wall.

3.2.6.4.2.3.2 Current Technical Progress

The combustor sector rig effort during this reporting period consisted of evaluating results from the final advanced liner test (Run 21 configuration) which was conducted as part of the NASA-sponsored Broad Specification Fuel Program. The tests were conducted to evaluate the affect of broad specification fuels on the operational characteristics of an advanced two-stage combustor. The combustor was evaluated over the full range of operating conditions with Jet-A fuel and two additional test fuels (ERBS 1 and ERBS 2) of progressively higher aromatic content.

Emissions, performance and operational characteristics were measured and compared. Results generally indicated that low power emissions exhibited very little dependency on aromatic content while high power oxides of nitrogen emissions increased with increasing aromatic content. These results were consistent with trends observed during previous tests with a single-stage conventional combustor. Liner temperature trends differed somewhat from the results obtained with the single-stage combustor with radiative affects less severe and convective affects more pronounced. Detailed results of this NASA-sponsored Broad Specification Fuels program will be presented in a future contractor report (NAS3-22392).

Additional testing was included with Jet A fuel to provide information relative to the hot streaking along the inner wall as evidenced in previous advanced liner segment testing described in the Sixth Semiannual Status Report. The tests were directed at evaluating the affects of pilot injector radial position, pressure level, and pilot to main zone fuel split. Imbedded liner thermocouples were used to obtain trends at the hot streak locations.

Prior to the start of testing, the center three pilot fuel injectors (upstream of the inner liner streaks) were relocated to assess the possibility of nonuniform thermal growth of the sector rig burner liner contributing to the hot streaks. The nozzles were moved approximately 3/16 of an inch radially outboard by installing shims. Except for this modification, the run 21 configuration was identical to the run 20 configuration.

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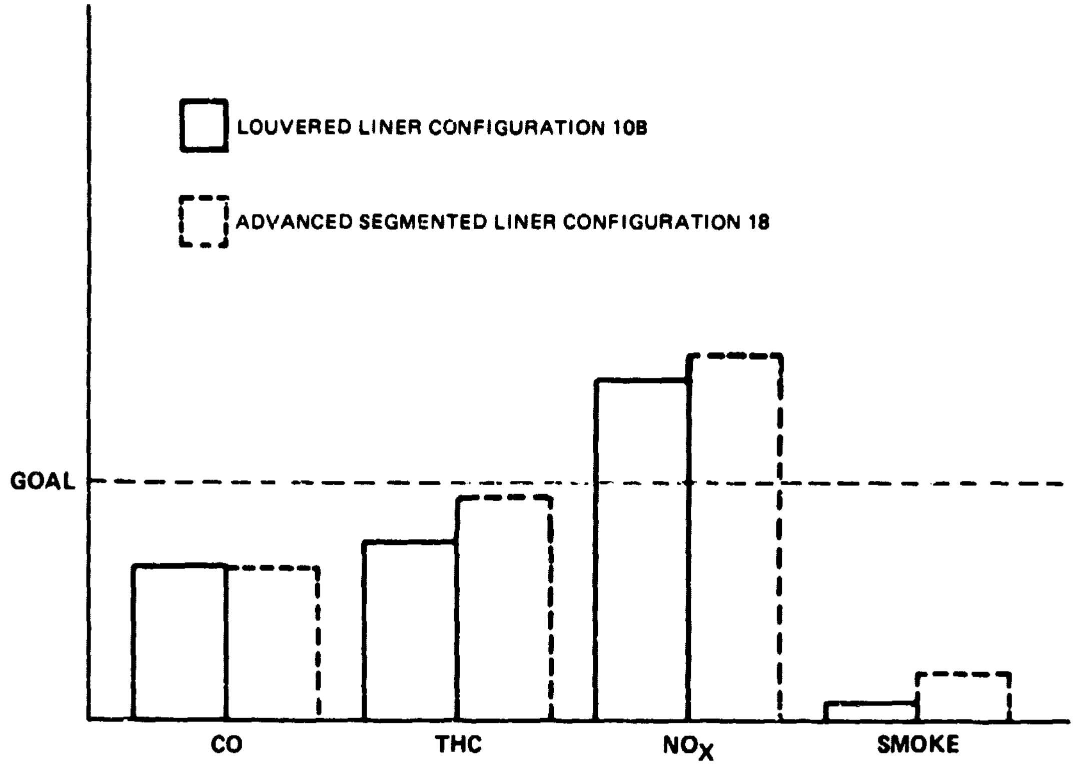

Figure 56 Segmented Liner Emissions Characteristics

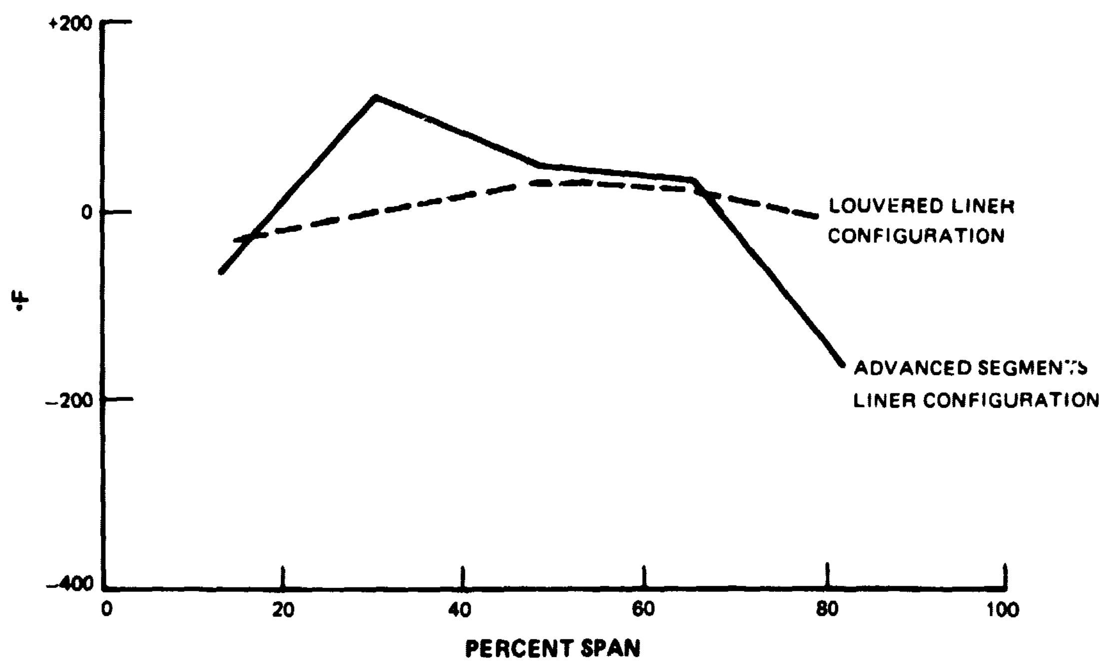

Figure 57 Comparison of Exit Radial Temperature Profiles

The test results showed that relocating the center three pilot fuel injectors reduced the highest (center) streak temperature approximately 150°F. The other instrumented streak increased approximately 130°F which resulted in a near equalization of streak temperatures. The potential geometry impact on streaking will be resolved during the upcoming full annular rig test program.

As shown in Figure 58, inner liner streak temperatures were affected by pressure level. Maximum streak temperatures were approximately 150°F higher at an operating pressure of 400 psia versus 300 psia. The affect of pressure was not as severe at the other instrumented locations.

Inner liner streak temperature was not significantly affected by pilot-to-main zone fuel split. Combustion processes in both zones appear to be interacting to cause the streaks.

The performance and emission characteristics of the advanced liner (Run 21) configuration are presented in Table 22. As indicated, all program design goals were met with the exception of the oxide of nitrogen (NOx) Environmental Protection Agency emission parameter. This parameter exceeded the goal but remained near the Pratt & Whitney Aircraft target level demonstrated in the Engine Clean Combustor Program two-stage combustor.

TABLE 22

ADVANCED SEGMENTED LINER PERFORMANCE SUMMARY

Pressure Drop (Percent Pt3)		Environmental Protection Agency Parameter (EPAP)	Emission Levels	
			Run 21	Goal
Section:	5.22	Carbon Monoxide	2.10	3.0
Outer Liner:	2.41	Unburned Hydrocarbons	0.38	0.4
Inner Liner:	2.70	Oxides of Nitrogen	5.1	3.0
		Smoke Number	4	20

Pattern Factor: 0.26
Radial Profile: 170°F peak-to-average at 65 percent span

The combustor airflow distributions and exit radial temperatures for the Run 21 advanced liner configuration are presented in Figures 59 and 60, respectively.

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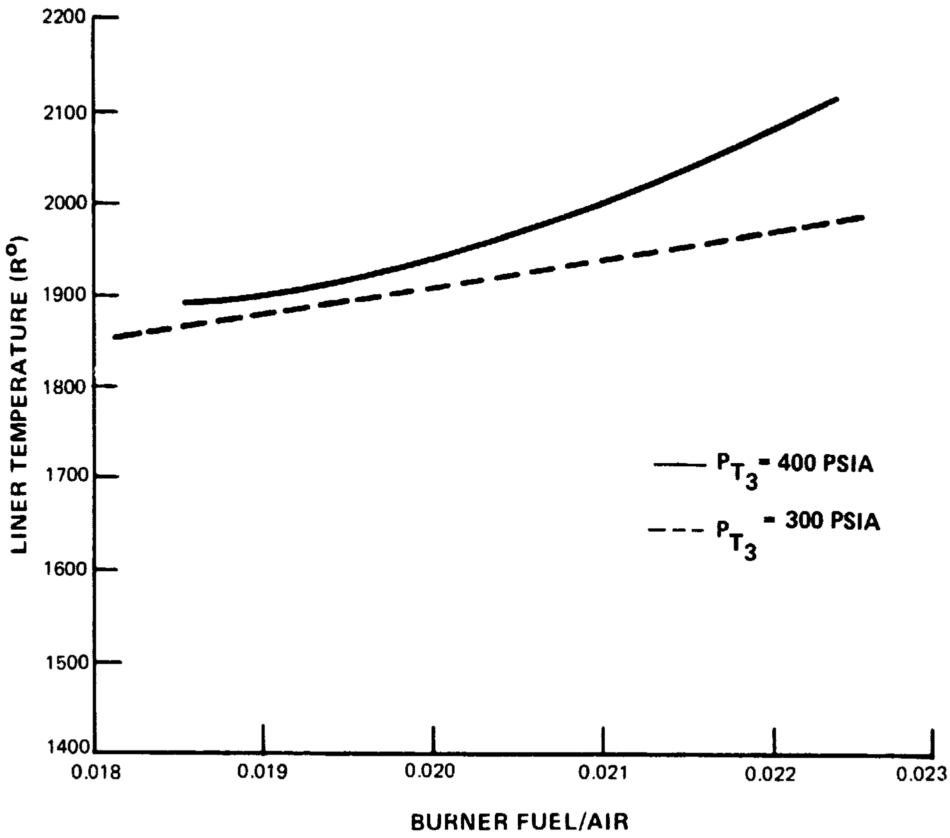

Figure 58 Comparison of Maximum Inner Diameter Segment Temperatures at Two Pressure Levels

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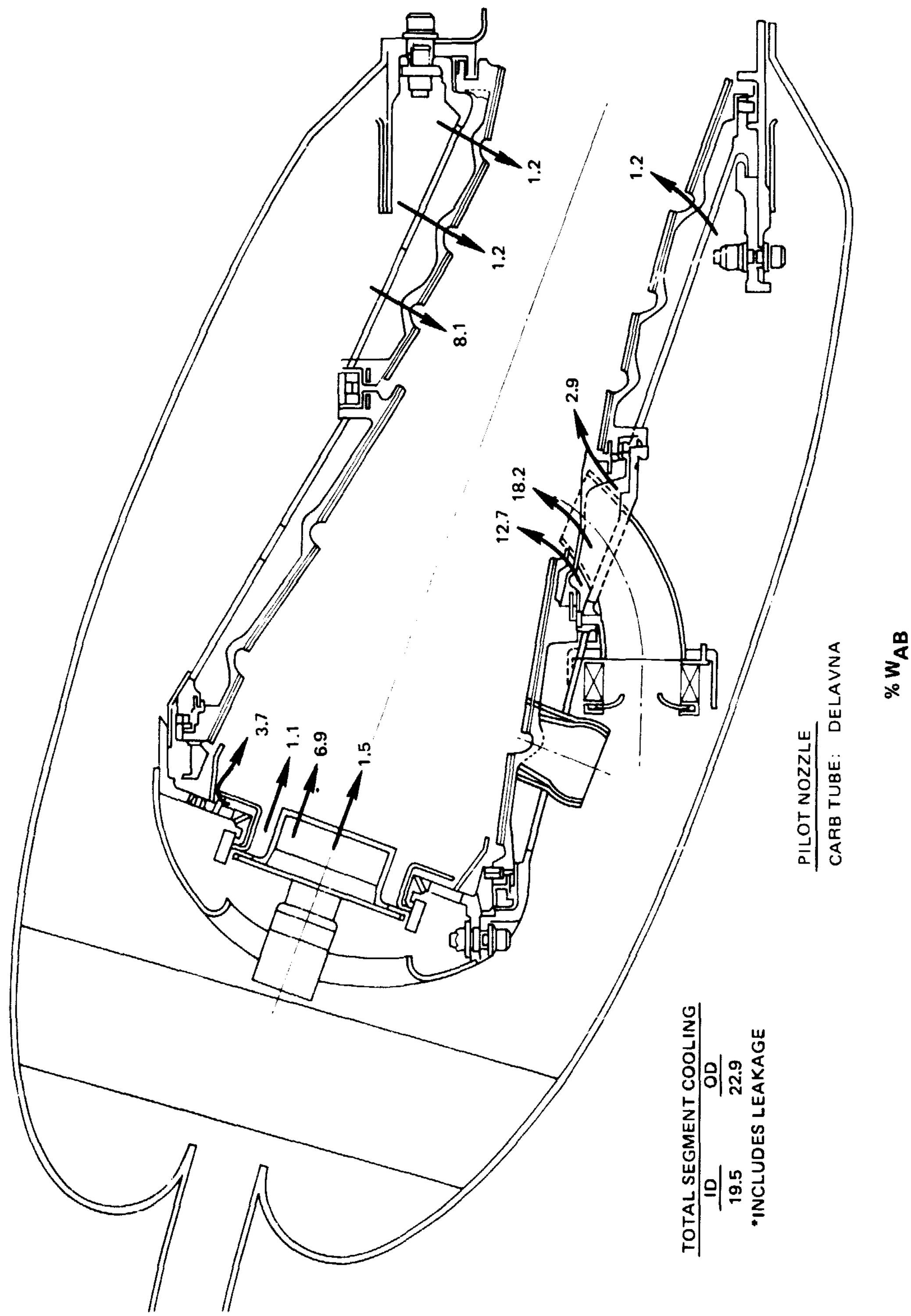

Figure 59 Advanced Segmented Liner Combustor Airflow Distribution

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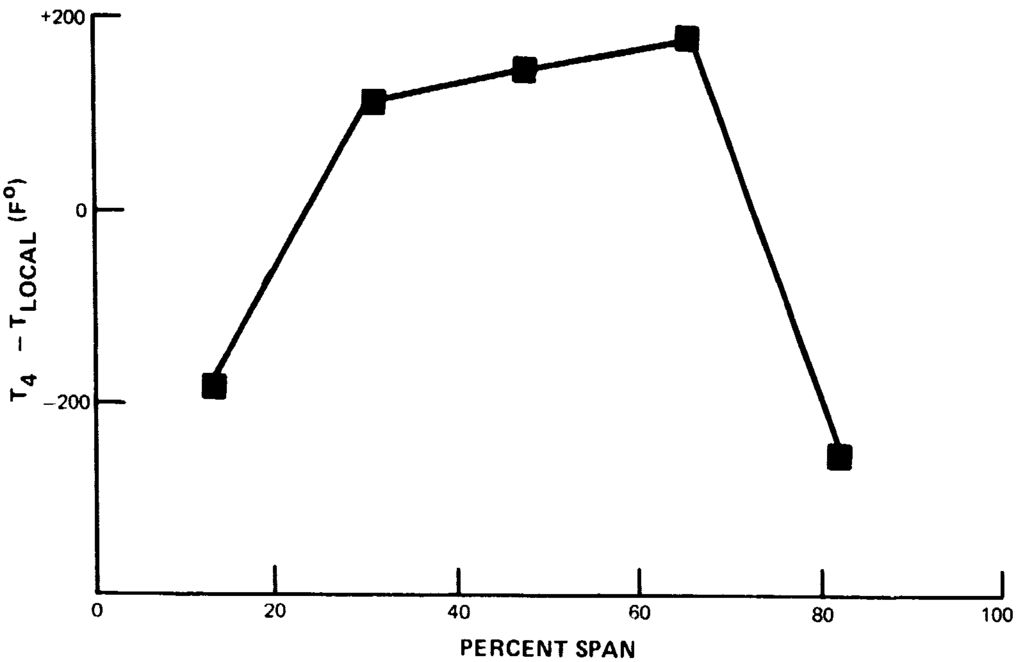


Figure 60 Advanced Segmented Liner Exit Radial Temperatures Profile

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 Wae and life of 10,000 hours on the blade and vanes and 20,000 hours on 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 61 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 62 and critical milestones are noted.

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 sub-tasks shown in Figure 62.

A six-month preliminary design activity is conducted to establish the feasibility of the high-pressure turbine as proposed for the flight propulsion system, Task 1. 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.

HIGH-PRESSURE TURBINE PROGRAM LOGIC DIAGRAM

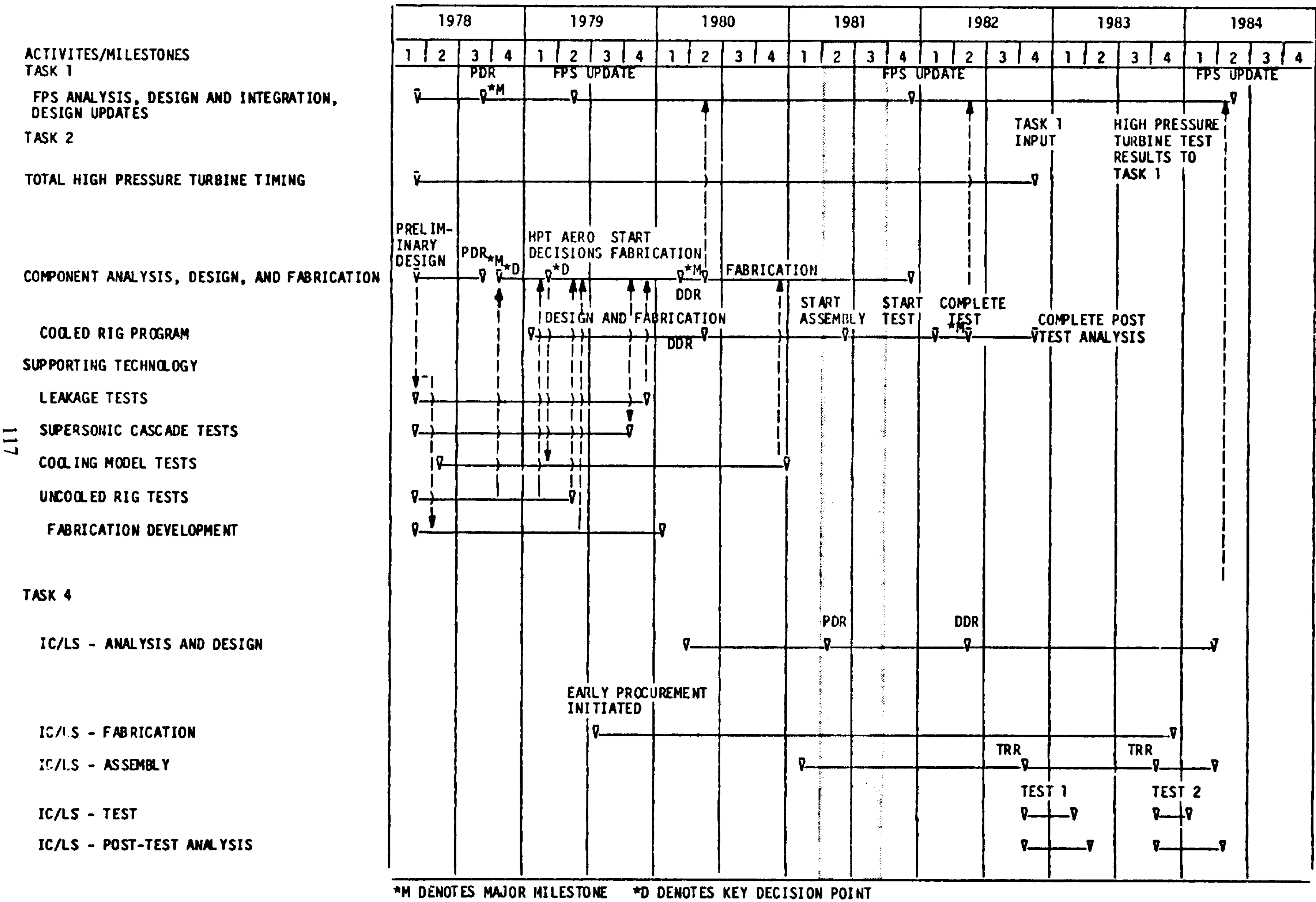
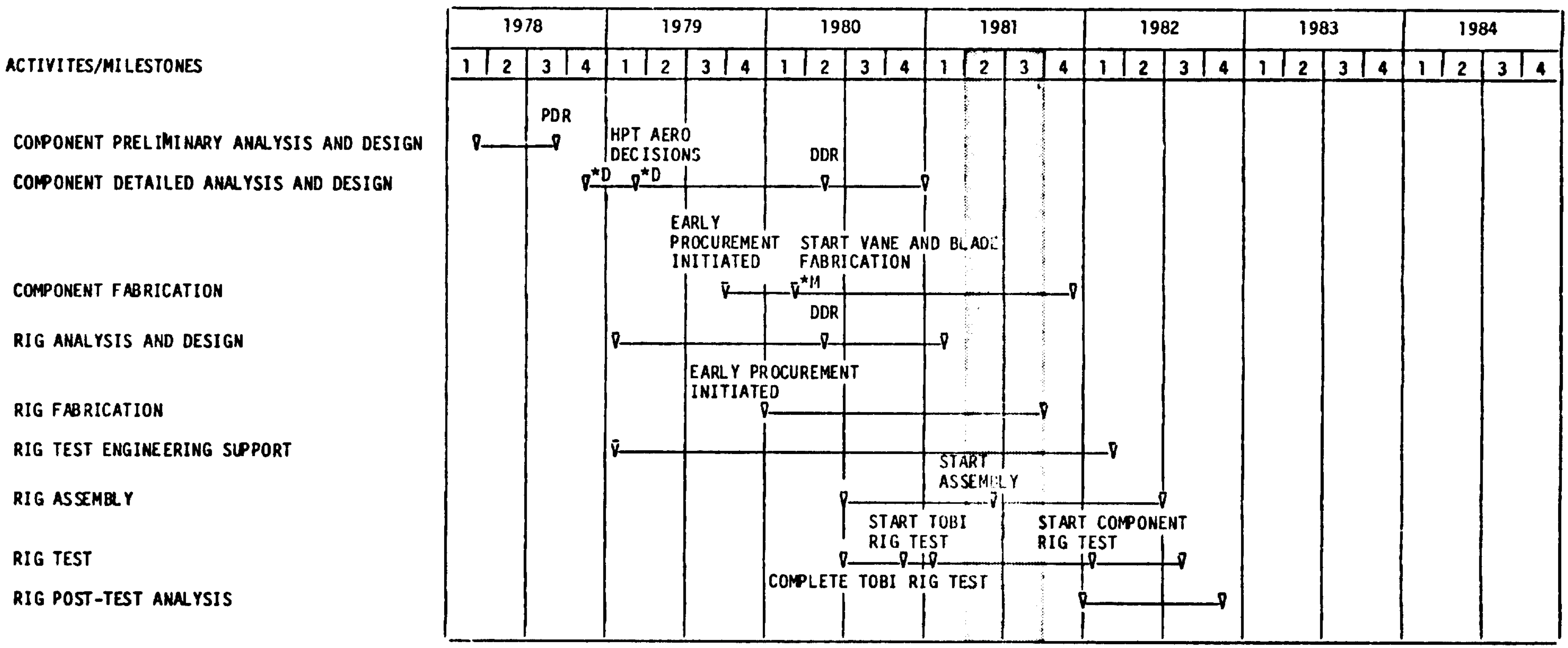


Figure 61 High-Pressure Turbine Program Logic Diagram

HIGH-PRESSURE TURBINE COMPONENT EFFORT



*M DENOTES CRITICAL MILESTONE *D DENOTES KEY DECISION POINT

Figure 62 High-Pressure Turbine Component Effort Work Plan Schedule

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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.

3.2.7.3.3 <u>Technical Progress</u>

3.2.7.3.3.1 <u>Summary of Work Previously Completed</u>

The high-pressure turbine component design that evolved from preliminary design efforts and detailed design efforts is illustrated in Figure 63. Major features are noted. Its companion 'warm' test rig is illustrated in Figure 64. The major aerodynamic parameters of the high-pressure turbine remained unchanged and are shown in Table 23. Current performance parameters at significant engine operating conditions are listed in Table 24.

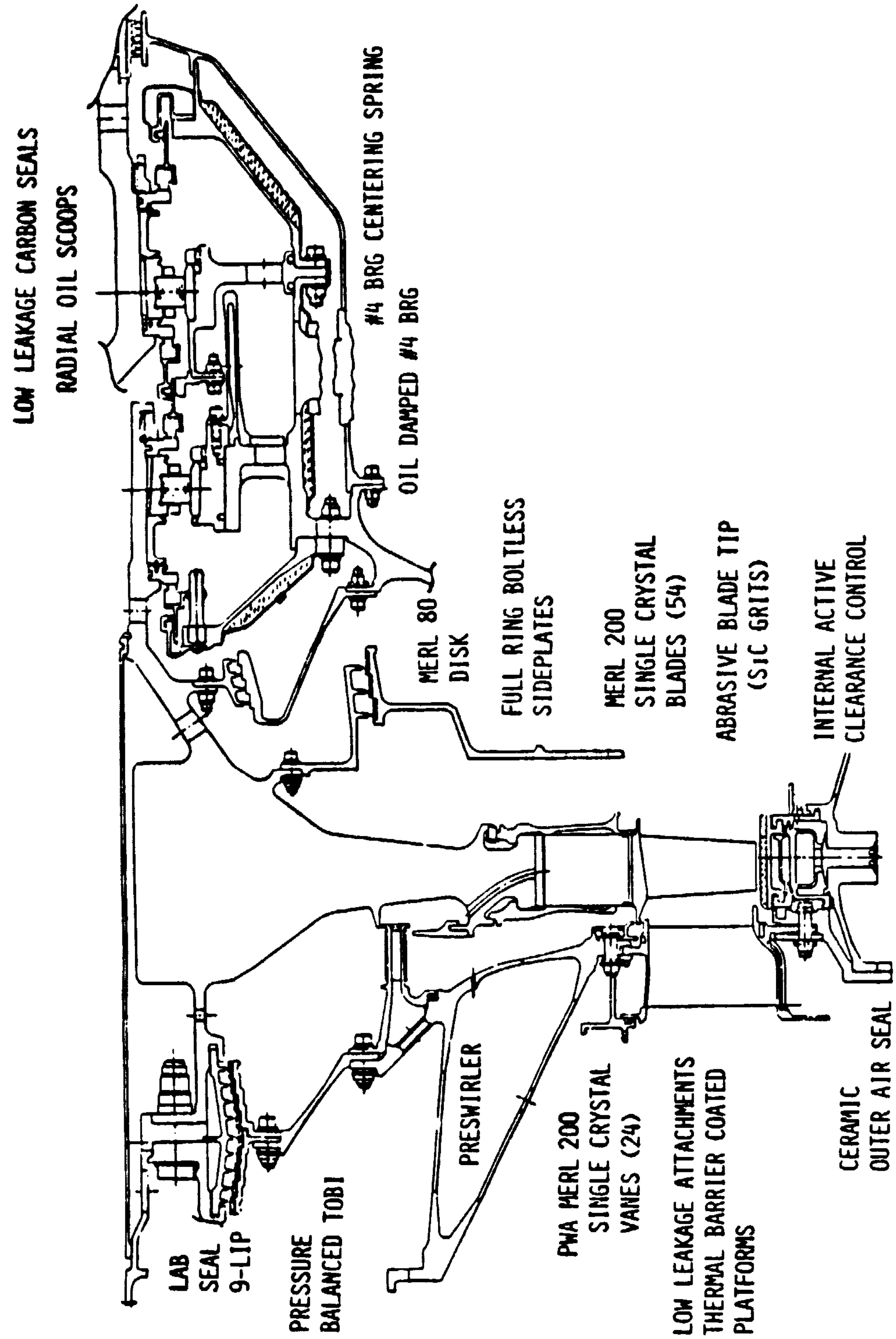


Figure 63 High-Pressure Turbine Component

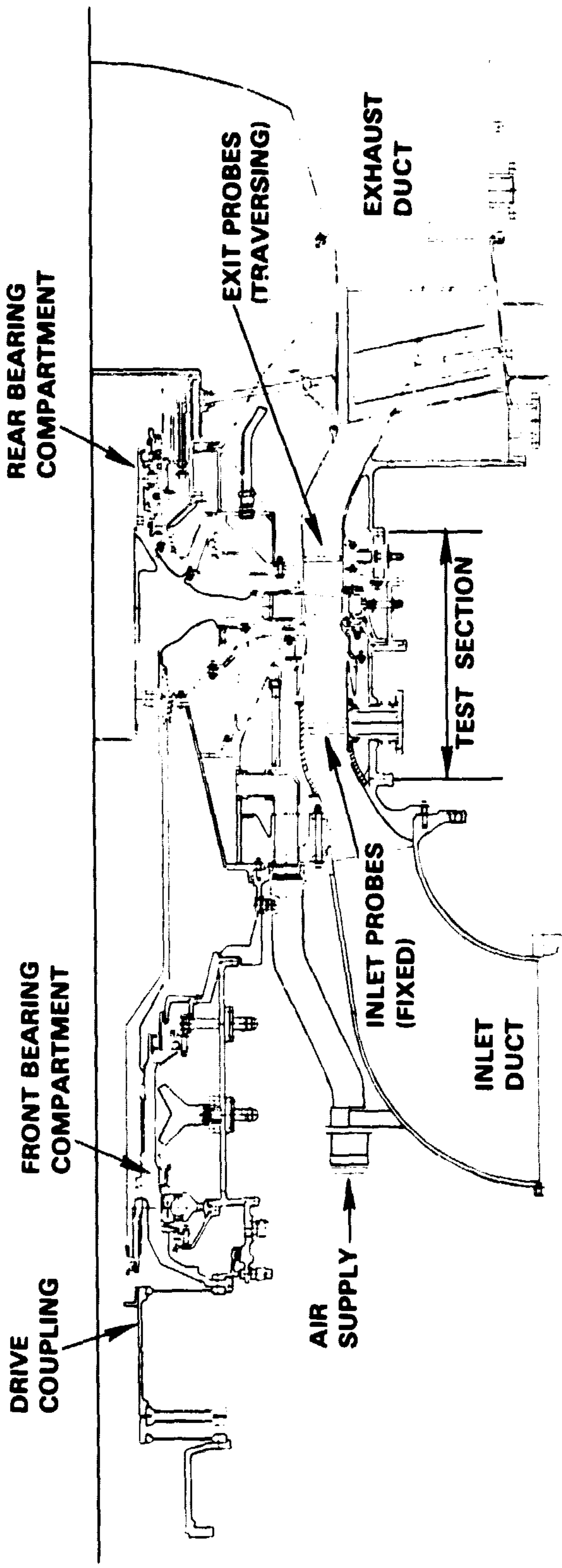


Figure 64 High-Pressure Turbine 'Warm' Rig

TABLE 23

HIGH-PRESSURE TURBINE AERODYNAMIC DESIGN SUMMARY

No. of Stages	1
Expansion Ratio	4.0
Mean Velocity Ratio/NASA Load Coefficient	0.56/1.59
An^2 - Maximum	49×10^9 in.^{2}-rpm^2
Rim Speed - Maximum	1730 ft/sec
Specific Enthalpy (h), btu/lb - SLTO	208
Mean Blade Loading (Y)	0.92
Mean Blade Turning	118 deg.
Mean Reaction Level	0.43
Number of Blades	54
Number of Vanes	24

TABLE 24

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

	Aero. Des. Point	Maximum Cruise	Maximum Climb	Takeoff
Inlet Flow Parameter (lb_m ^oR)(in.^{2}/sec)(lb_f)	16.70	16.70	16.65	16.65
Rotor Inlet Temperature (^oF)	2235	2315	2410	2485
Pressure Ratio	3.99	3.99	3.97	3.98
Adiabatic Efficiency (percent)	89.1	89.1	89.1	89.2
Enthalpy Change (btu/lb)	190.3	187.5	202.5	208.7
Transition Section Pressure Loss (percent)	.70	.70	.70	.70
Total Cooling Airflow (% Core Airflow)	16.7	16.7	16.7	16.7

The high-pressure turbine component rig (shown in Figure 64) uses as many existing rig parts as possible to adapt to the Energy Efficient Engine design, thereby minimizing cost.

The secondary flow system provides rig flows that duplicate all integrated core/low spool design cooling air and leakage flows and pressures associated with the high-pressure turbine flowpath. Separate flow controls will be provided for cooling air to the primary and secondary tangential on-board injection.

All of the analysis and design efforts associated with the high-pressure turbine component and rig were completed during a previous reporting period, and a detailed design review was held at NASA-Lewis Research Center on May 21 and 22, 1980. NASA approved this design on 11 June 1980. Details of this design are summarized in the Fifth Semiannual Status Report.

The performance of the high-pressure turbine is compared to the goal performance estimates in Table 25. The goal efficiency levels are exceeded for both the flight propulsion system and the integrated core/low spool primarily because (1) the actual design tip clearance is less than the previously established value, (2) vane and blade cooling air requirements have been revised, and (3) chargeable secondary airflow has been reduced.

TABLE 25

HIGH-PRESSURE TURBINE PERFORMANCE VS. GOALS

	Goal		Current Status	
	FPS	IC/LS TEST	FPS	IC/LS
Efficiency (percent)	88.2	86.7	89.1	87.6

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: Initiated fabrication of vanes, blades, disks (including front and rear sideplates, rear thrust balance seal and vortex plate), rotating high-pressure compressor discharge seal, active clearance control hardware, and number 4 bearing compartment parts. 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 the rig parts.

Rig-Unique Hardware: Completed fabrication of front compartment bearings and initiated fabrication of rear compartment bearings and instrumentation probes and rakes.

3.2.7.3.3.2 Current Technical Progress

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

High-Pressure Turbine Component Fabrication

Turbine Vane Fabrication: The vendor continues to experience diffuculty in producing acceptable single-piece, PWA 1480 single crystal vane castings. Results from casting trials indicate various degrees of oversize (thick) airfoils, extraneous grain formation, and lack of trailing edge fill. Corrective actions taken to alleviate these problems included revised mold dewaxing techniques, mold bracing schemes, casting parameter adjustments, revised shell mold compositions, core positioning pins, and reductions in the number of vanes cast per mold.

These actions were effective in reducing the amount of oversize thickness and the degree of extraneous grain formation. However, additional effort will be required to produce acceptable PWA 1480 single crystal vane castings for integrated core/low spool engine running.

Machining of the first set of PWA 1422 vanes for incorporation into the component rig has progressed through completion of grinding inner and outer platform rails and electro-discharge machining airfoil cooling holes. The effort directed toward machining platform cooling holes and feather seal slots is progressing satisfactorily. An example of a semi-finished vane is shown in Figure 65.

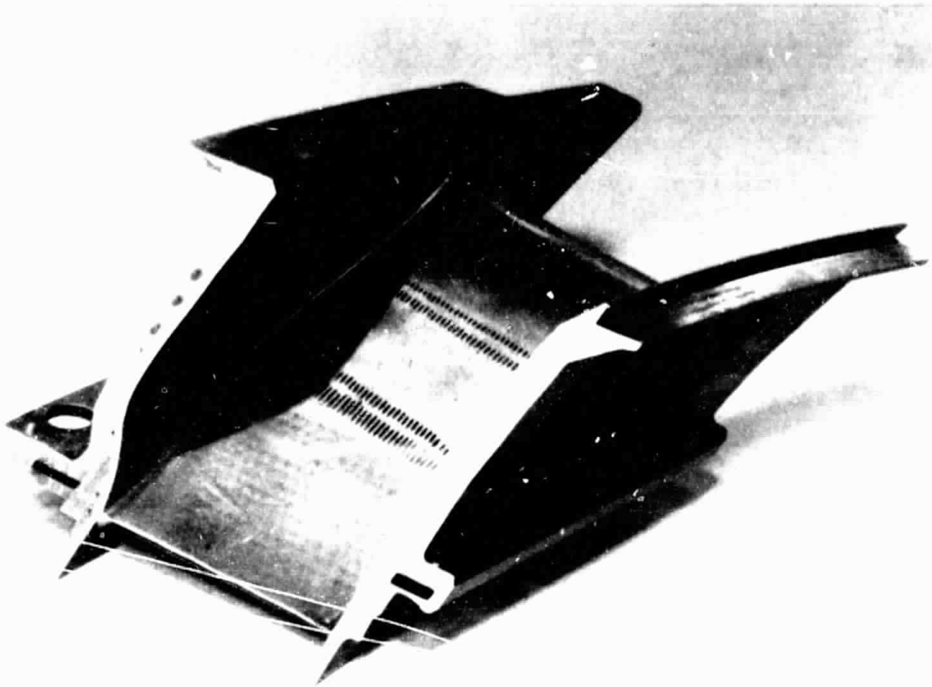


Figure 65 High-Pressure Turbine Component Rig Vanes With Machined Inner and Outer Platforms and Airfoil Cooling Holes Installed

Sample sheet metal impingement tube inserts were submitted by the vendor, reviewed and approved by Pratt & Whitney Aircraft. Completion of the part order is currently in process.

Turbine Blade Fabrication: Vendor effort on PWA 1480 material single crystal blade castings progressed through completion of one-half of the first set of castings required for the component rig. The main casting difficulty encountered by the vendor was core breakage at the trailing edge tip. This problem has been alleviated by adding five additional rear cavity pedestals in the tip section and opening up the tip trailing edge discharge hole.

Machining of three initial blade castings has progressed through root attachment, platform and airfoil tip grinding. The results of these machining efforts are shown in Figure 66.

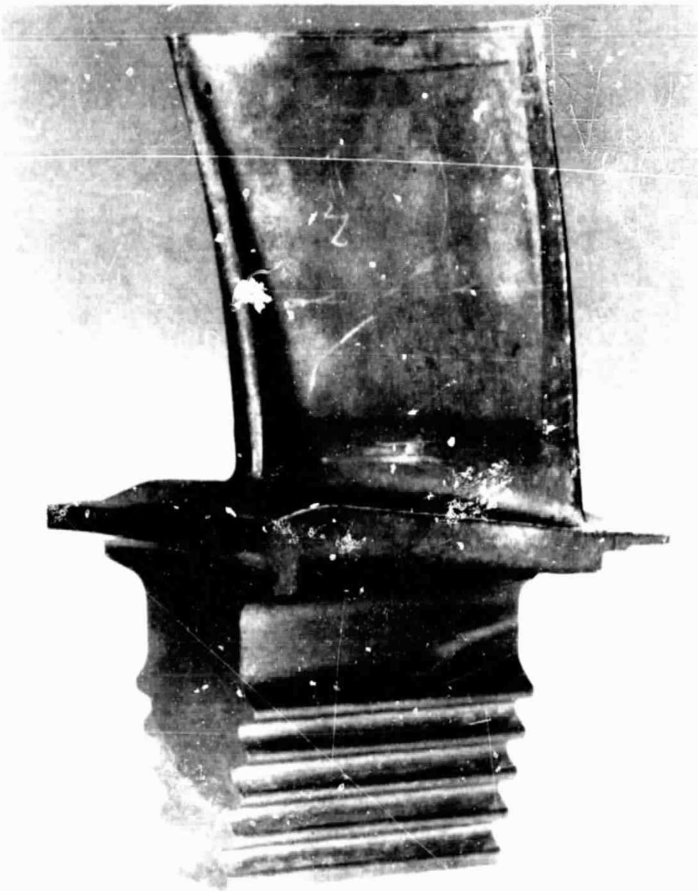


Figure 66 High-Pressure Turbine PWA 1480 Single Crystal Blade Casting With Machined Root Attachment, Platform and Airfoil Tip

Turbine Disk and Attachments Fabrication: The turbine disk fabrication effort during the reporting period included hot isostatic pressing a MERL 76 compaction, initial lathe turning of the rough shaped compaction, heat treating, surface cleaning and sonic inspection, and finish machining (lathe turning). Major operations remaining to be completed are blade attachment broaching, hole drilling, slot milling, electro-chemical machining of cooling air holes, and polishing. Figure 67 shows the disk as it appeared during the lathe turning operation.

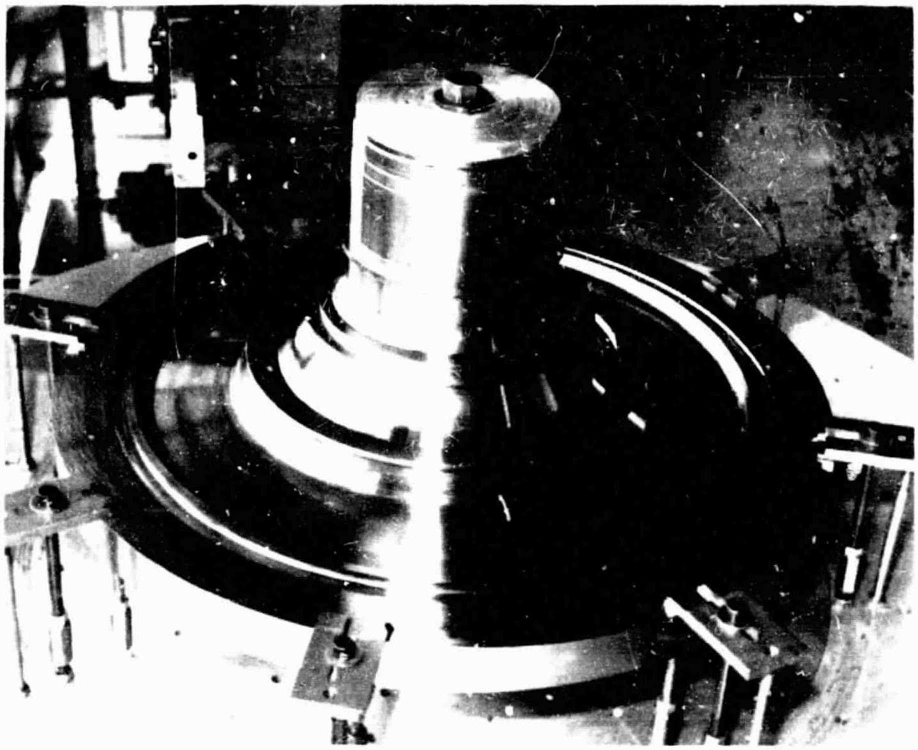


Figure 67 Semifinished (Lathe Turned) High-Pressure Turbine Disk

Development of the process to electro-chemically machine curved, elliptical cooling air holes in the disk rim has progressed to the point where acceptable test pieces have been machined. Based on this success, the process will be utilized in machining elliptical cooling air holes in the second disk compaction. These air holes are designed to conduct cooling air from the disk front side to the root attachment area of each blade.

Work was completed on the front disk side plate, shown in Figure 68. This plate was machined from hot isostatically pressed MERL 76. Work on the disk vortex plate located near the TOBI nozzle exit was also completed while work on the rear disk side plate progressed to 80 percent completion.

High-Pressure Compressor Discharge Seal: Work was completed on the 9-lip discharge seal and the front and rear high-pressure compressor discharge seal land supports.

Air Seals: Also completed during this reporting period was all work on the disk rear thrust balance seal, the number 4 and 5 bearing compartment buffer air seal, and the disk rim front air seal.

Active Clearance Control System: Items completed for the blade outer air seal active clearance control system include the rear rail support, the outer air seal segment castings, shown in Figure 69, and the outer air seal cooling air impingement plate. The front rail support and manifold assembly progressed to 75 percent completion while machining of seal segments was initiated during the report period.

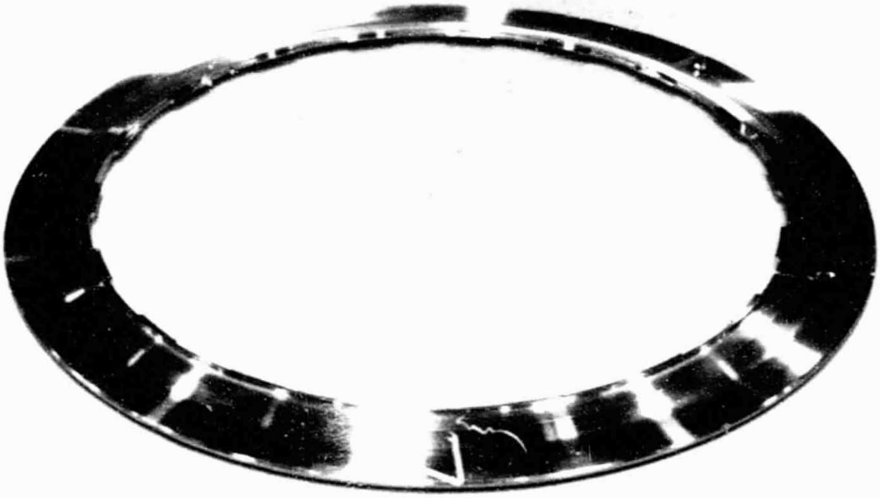


Figure 68 High-Pressure Turbine MERL 76 Front Disk Side Plate

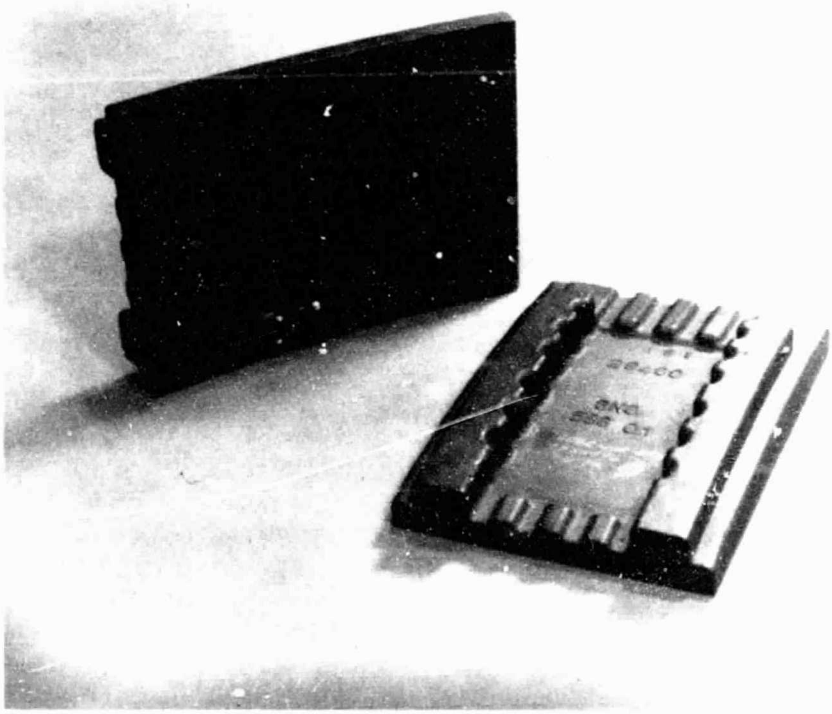


Figure 69 Active Clearance Control System Blade Outer Air Seal Rear Rail Support

Vane Supports: The inner vane support and the outer vane support were both completed.

Number 4 Bearing: Fabrication of the number 4 bearing was completed by the bearing vendor.

Miscellaneous Parts: Fabrication of approximately 90 percent of high-pressure turbine miscellaneous hardware has been completed.

High-Pressure Turbine Component Rig

Rig Fabrication: Approximately 90 percent of all hardware required for the component rig (exclusive of component parts) has been fabricated and delivered to Pratt & Whitney Aircraft to-date. Major rig hardware items completed include the exit instrumentation traverse ring, the inner diameter exit flow path, disk gauge spacer, outer diameter inlet case (shown in Figure 70), front bearing buffer air seal, disk rear thrust balance seal support, and front bearing stub shaft (shown in Figure 71).

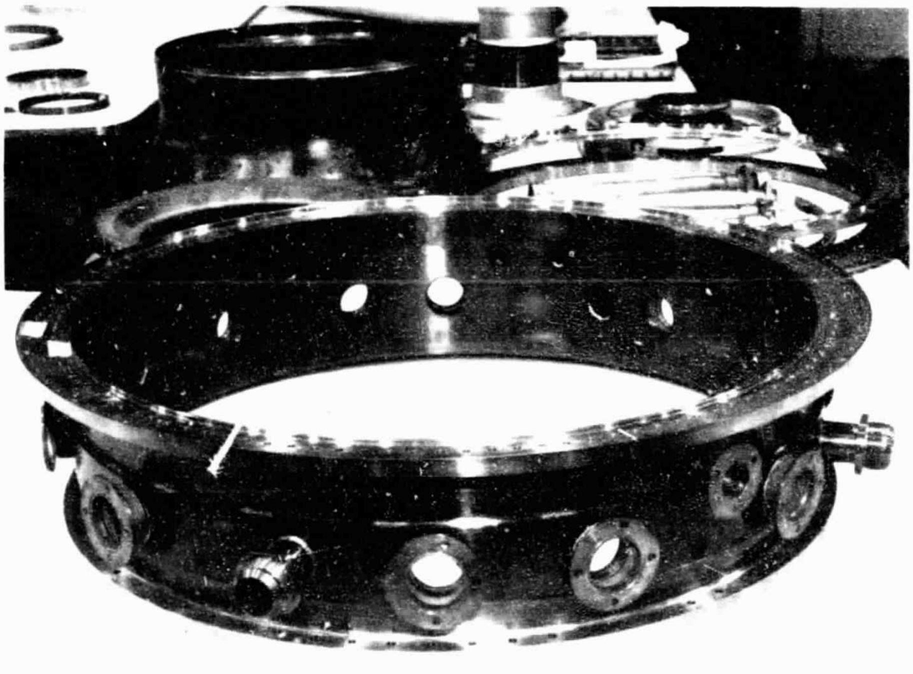


Figure 70 High-Pressure Turbine Outer Diameter Inlet Case

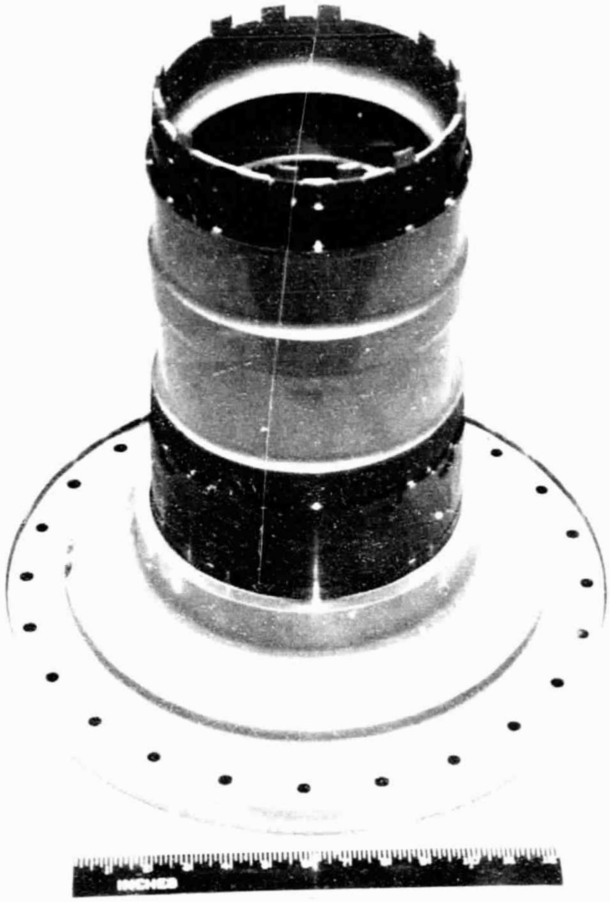


Figure 71 High-Pressure Turbine Front Bearing Stub Shaft

Engineering and Support: 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.

Rig Assembly: Fabrication effort was completed on the fixtures required to cold airflow various rig assemblies. Fit-checking of parts was accomplished as parts were delivered to assembly. Two instances of fit discrepencies were discovered and part reoperations accomplished to correct the problems. 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.

Rig Performance Test: Procurement of special facility test equipment required for the secondary airflow cooling system is 90 percent complete.

3.2.7.4 Supporting Technology

3.2.7.4.1 Leakage Test Program

All efforts under this supporting technology program are complete. Program results are reported in NASA CR-165202.

3.2.7.4.2 Supersonic Cascade Test Program

All technical effort for this supporting technology program is complete. Program results were presented in the Third Semiannual Status Report. A draft of the technology report has been submitted to NASA for review and approval.

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 sub-tasks. 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 72 shows the relationships between these activities and their relationship to Tasks 1 and 4. The work plan is shown in Figure 73.

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 73. 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.

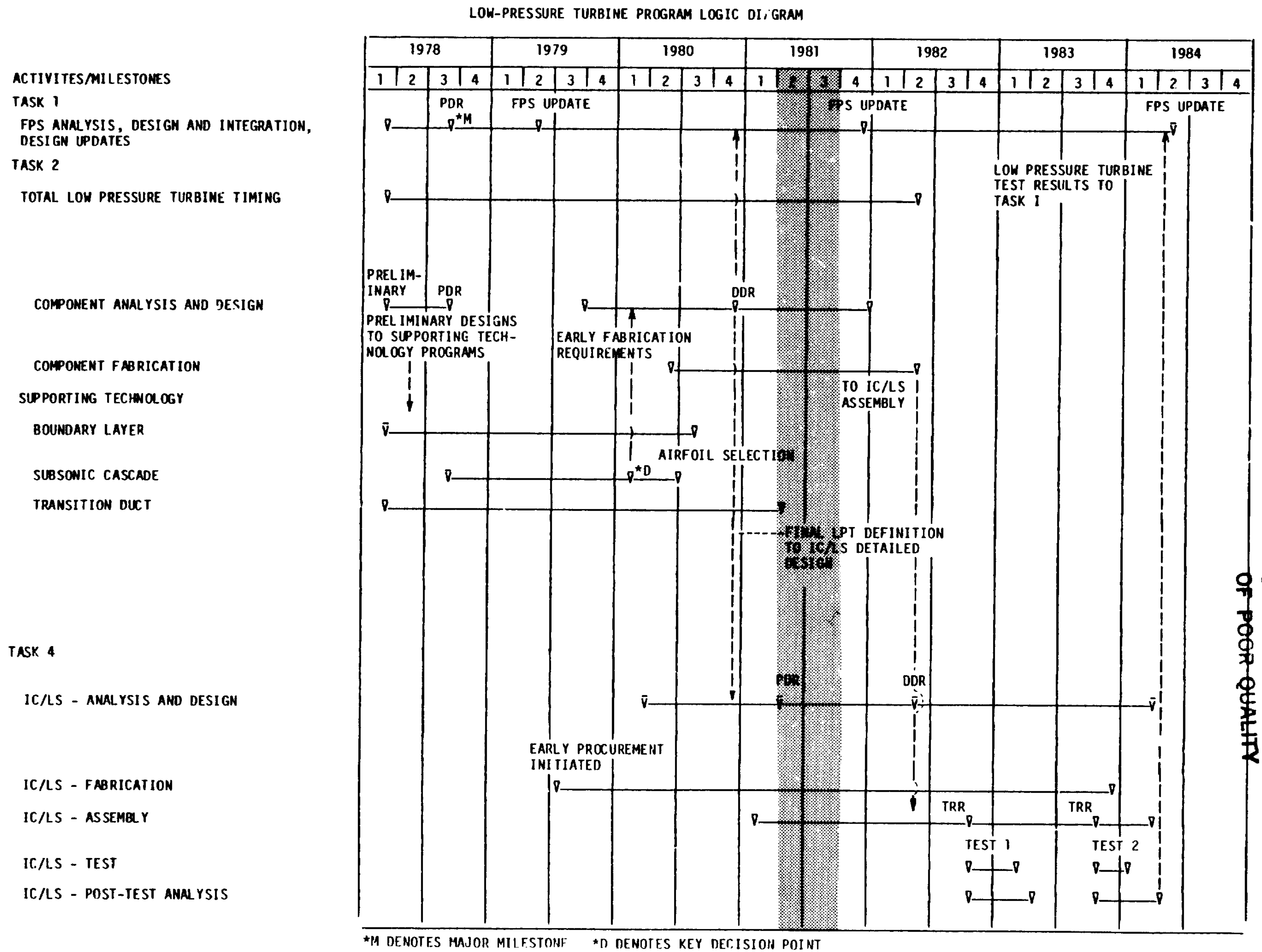


Figure 72 Low-Pressure Turbine Program Logic Diagram

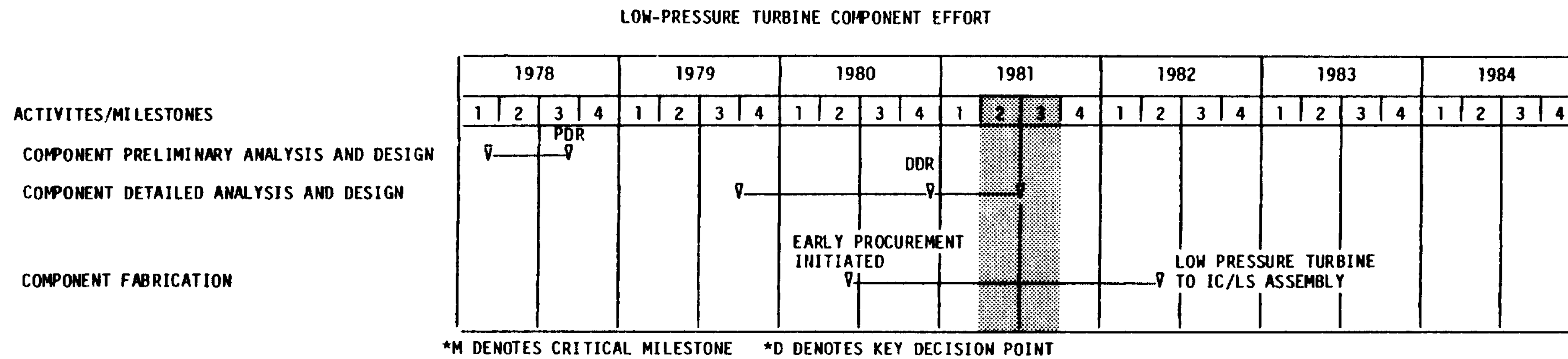


Figure 73 Low-Pressure Turbine Component Effort Work Plan Schedule

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 73 indicates that all of the work associated with the preliminary analysis and design of the low-pressure turbine component was completed during a previous reporting period. The figure also shows that component detailed analysis and design work was completed while design follow-up and fabrication work continued during the current reporting period.

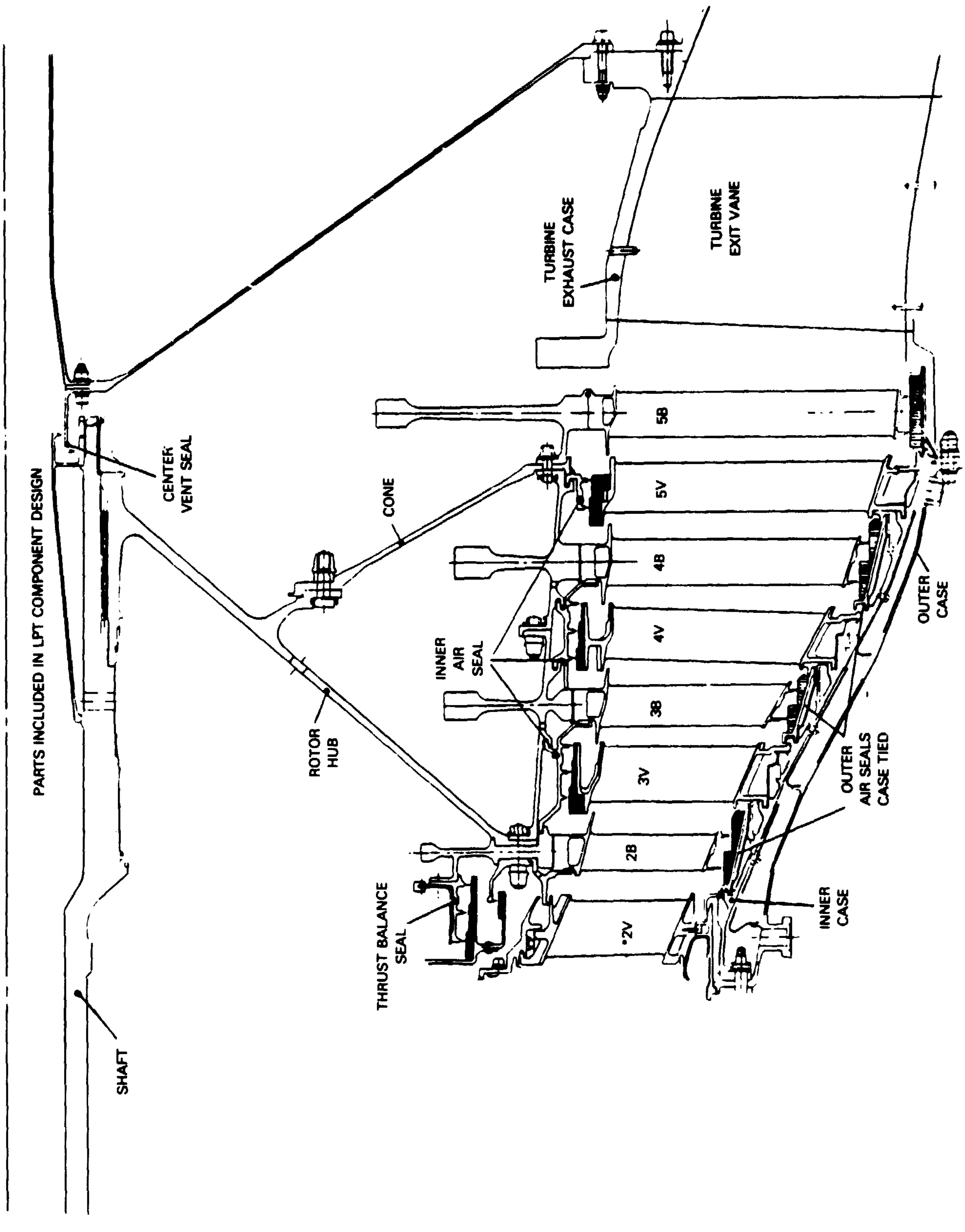
3.2.8.3.3 Technical Progress

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 74.

The major mechanical features of this design are as follows:

- 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.



*LPT INLET GUIDE VANE (IGV) IS 2ND STAGE VANE OF COMPLETE ENGINE TURBINE SECTION

Figure 74 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 thru 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 thru an internal active clearance control system. These general aerodynamic characteristics are listed in Table 26 while the current performance parameters at significant engine operating conditions are shown in Table 27.

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 75 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.

A summarization of the fabrication efforts prior to the current reporting period is listed below.

- Long lead time raw material processing continued.
- Forgings for the low-pressure turbine case and hot strut outer case were delivered.
- Preparation of wax pattern tooling for the vanes and blades proceeded.
- The second vane tool was completed and delivered to the casting vendor.
- Wax pattern tooling for the hot strut fairing was also delivered to the casting vendor and trial waxes were poured and inspected.
- Turbine exhaust case vane castings were ordered along with number 5 bearing area parts.

TABLE 26

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 (^{o}R)	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 (Δ h/U^2)	.464 (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 27

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 ^{o}R)(in.^{2}/sec)(lb_f)	65.40	65.50	65.25	65.40
Rotor Inlet Temperature (^{o}F)	1540	1505	1675	1735
Pressure Ratio	5.72	5.66	5.81	5.09
Adiabatic Efficiency (percent)	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

3.2.8.3.3.2 Current Technical Progress

Low-Pressure Turbine Component Fabrication

Disks and Inner Air Seals: Hot isostatically pressed MERL 76 powder material is being used for all low-pressure turbine disks and inner air seals. The hot isostatic press cans required for one set of disk and inner air seal compactions plus spares were completed this report period. Filling of the cans with MERL 76 powder is progressing satisfactorily.

Shaft: Work on the first low-pressure turbine shaft forging was terminated after cracks developed on the front end of the shaft following a heat treat operation. The heat treat operation was determined to be improper. A second shaft was forged and rough machining was initiated during the report period. An approved heat treat procedure will be used for this shaft.

Blades: Casting work by the vendors progressed through initial casting trials to completion of acceptable sample parts. Inspection reports on the sample castings for all low-pressure turbine blades were completed by the vendors, reviewed and accepted by Pratt & Whitney Aircraft. Vendors are currently casting and processing all low-pressure turbine blades according to schedule.

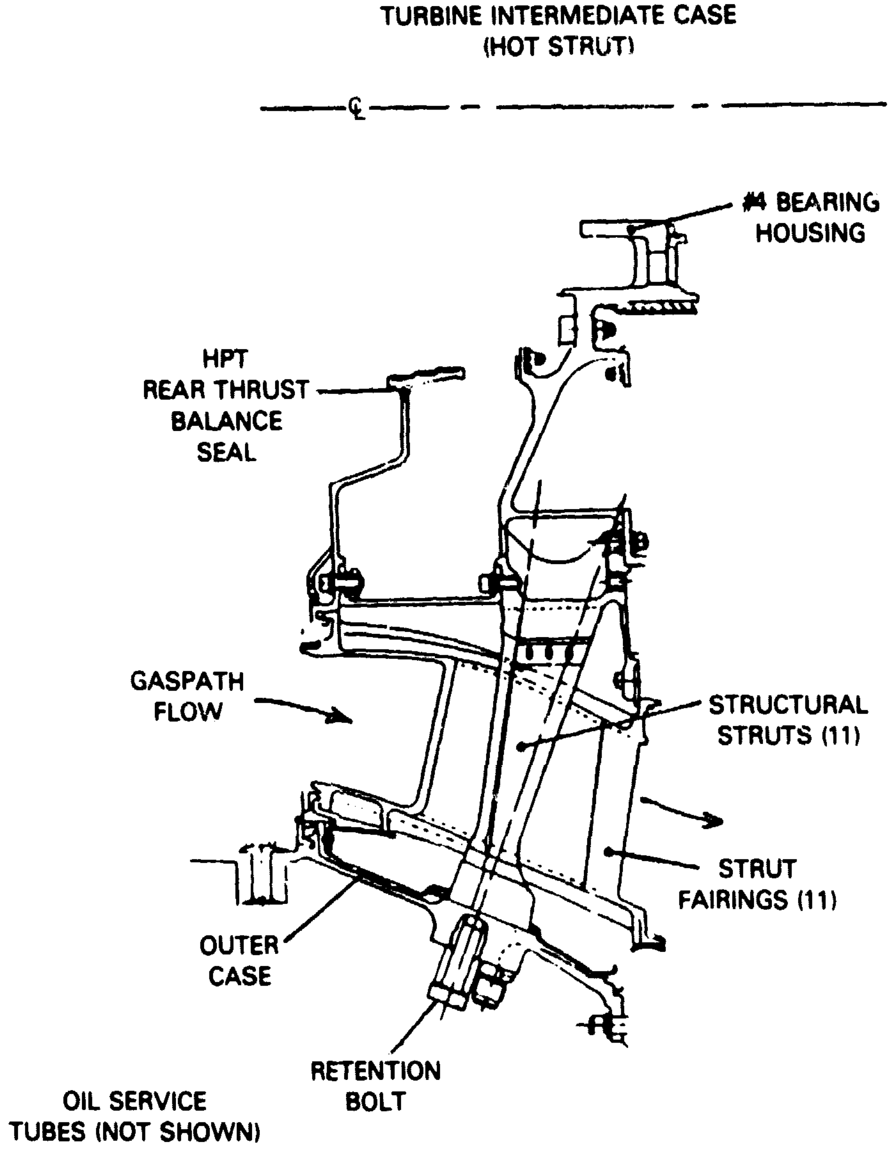


Figure 75 Turbine Intermediate Case Major Design Features

Vanes: Vane casting work by the vendors progressed through completion of initial trials to fabrication of acceptable sample parts. All vane sample casting inspection reports were completed by these vendors, reviewed and accepted by Pratt & Whitney Aircraft. Casting of all vanes is progressing according to schedule. All low-pressure turbine airfoils from the second vane stage through the fifth blade stage are shown in Figure 76.

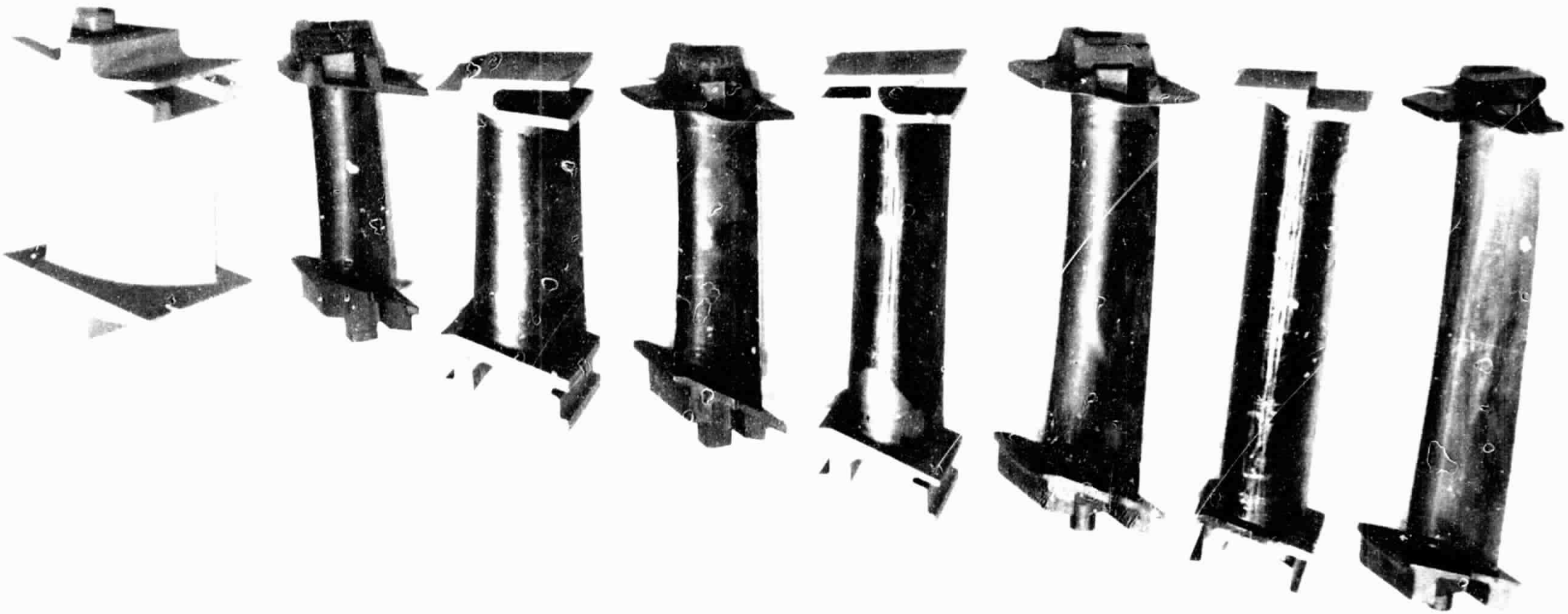


Figure 76 Low-Pressure Turbine Airfoils From the Second Vane Stage Through the Fifth Blade Stage

Hot Strut Case

All forging material for the case parts was received during the report period. Initial rough machining of the case details, prior to welding, progressed to 60 percent completion. Following a successful pour of sample hot strut fairing castings, shown in Figure 77, the vendor is directing effort toward developing an acceptable casting straightening procedure.

Turbine Exhaust Case

All forging material for the case details was received and rough machining of the case details was started. Turbine exhaust case vanes have been cast, as shown in Figure 78, and sample parts are being submitted to Pratt & Whitney Aircraft for inspection and review.

Turbine Case Assembly

Three ring-forgings for the turbine case were received and electron beam welding of the rings to form the rough shape case, as shown in Figure 79, was completed. Inspection of the welds is in process prior to heat treat and final machining.

3.2.8.4 Supporting Technology

3.2.8.4.1 Boundary Layer Test Program

All efforts under this supporting technology program are complete. Program results are reported in NASA CR-165338.

3.2.8.4.2 Subsonic Cascade Test Program

3.2.8.4.2.1 Objective

The objective of this program is to develop (through analysis and testing) the design of low-loss, highly loaded airfoils for the flight propulsion system low-pressure turbine component design.

3.2.8.4.2.2 Scope of Total Work Planned

All technical effort for this supporting technology program is complete. Program results were presented in the Fifth Semiannual Status Report. A draft of the technology report is currently being revised following NASA review.

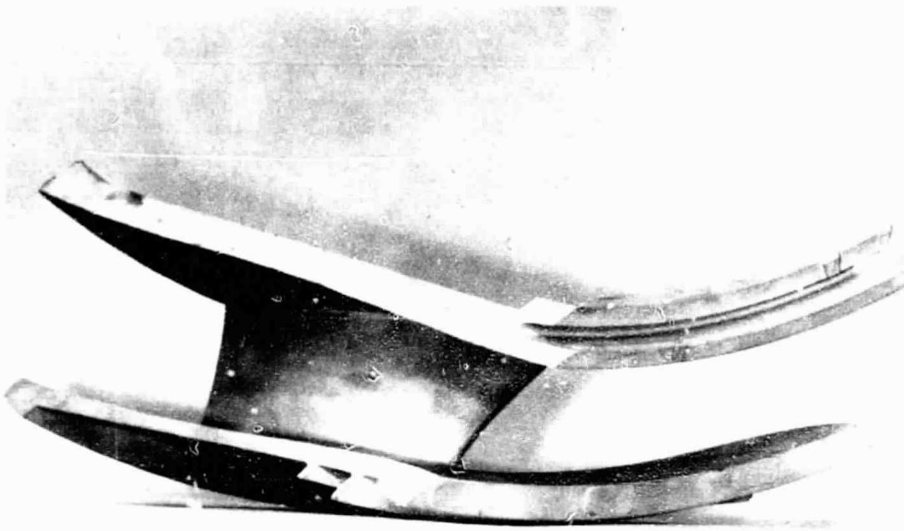


Figure 77 Sample Hot Strut Fairing Casting

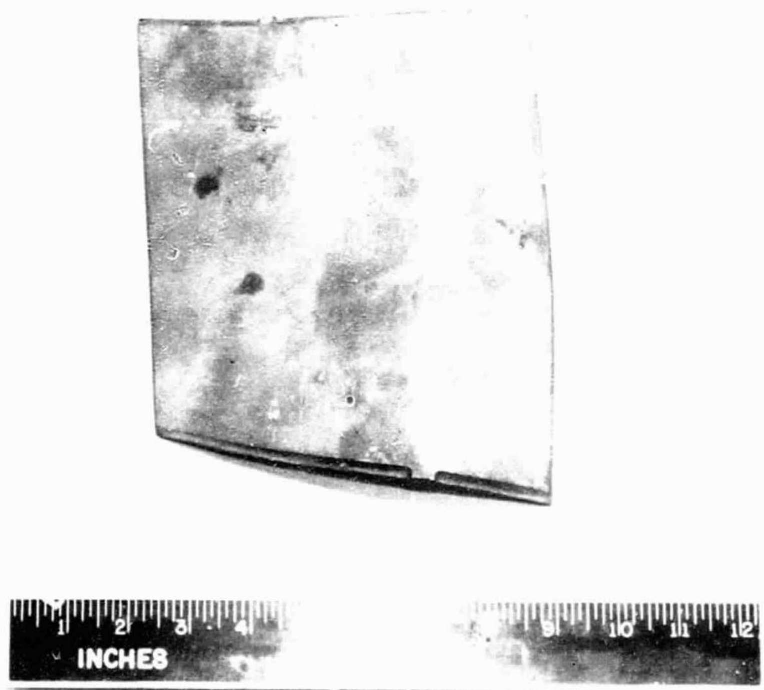


Figure 78 Turbine Exhaust Case Sample Vane Castings

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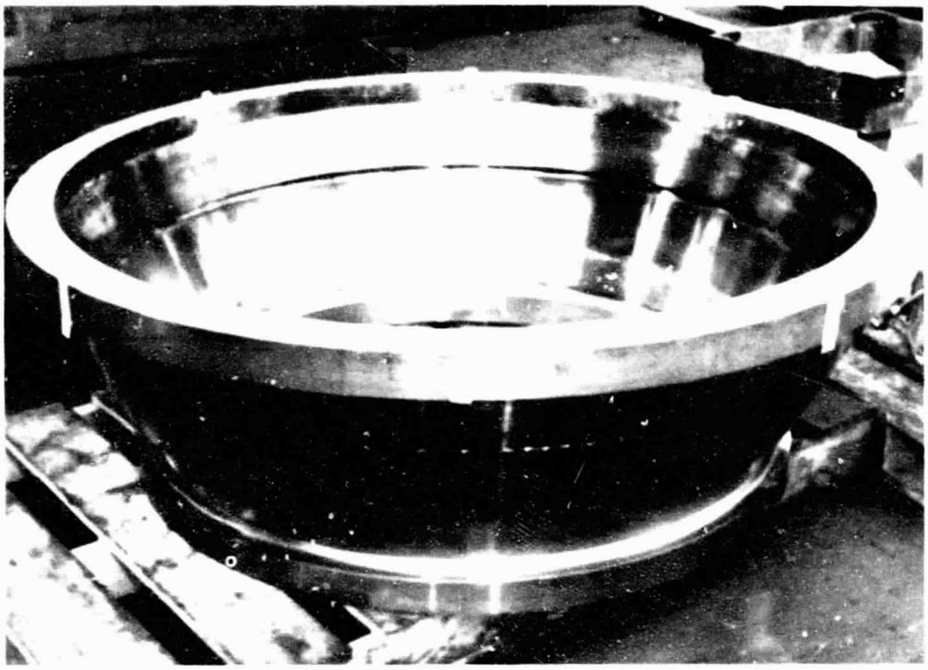

Figure 79 Rough Turbine Case Assembled from Three Ring-Forgings Electron Beam Welded Together

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

The transition duct test program consists of four phases. Figure 80 shows these phases and identifies those tasks and activities that were completed during the previous reporting periods and indicates that post-test analysis was completed during the current reporting period.

An inviscid/viscid analytical coupling procedure is used until incipient stability is obtained. This analysis is applied to an initial transition duct design obtained from minimum structural length considerations. The inner and outer diameter contours, strut fairing shape, and locations are varied to obtain the most desirable design. This first design is incorporated as the base low-pressure turbine transition duct. A second duct is used to model the integrated core/low spool transition duct. The design includes the intermediate case strut and low-pressure turbine inlet vane. The models, hardware, and instrumentation are designed, fabricated, and then assembled into an air flow tunnel facility. Pressure and air angle measurements are conducted at Energy Efficient Engine conditions. Aerodynamic losses, local separation, and nonuniform flow patterns are analyzed.

3.2.8.4.3.3 Technical Progress

3.2.8.4.3.3.1 Summary of Work Previously Completed

Prior to the current reporting period, test and post-test analysis of the first transition duct model were completed at design and off-design flow conditions. Test and post-test analysis of the second transition duct model were completed at design point flow conditions. Results of these activities are summarized below.

First Duct Model:

- o At design flow conditions, the total pressure loss including the low-pressure turbine first vane is 1.5 percent versus a 2.6 percent design value. At off-design swirl conditions up to 5 degrees, pressure loss increased to 2.1 percent.

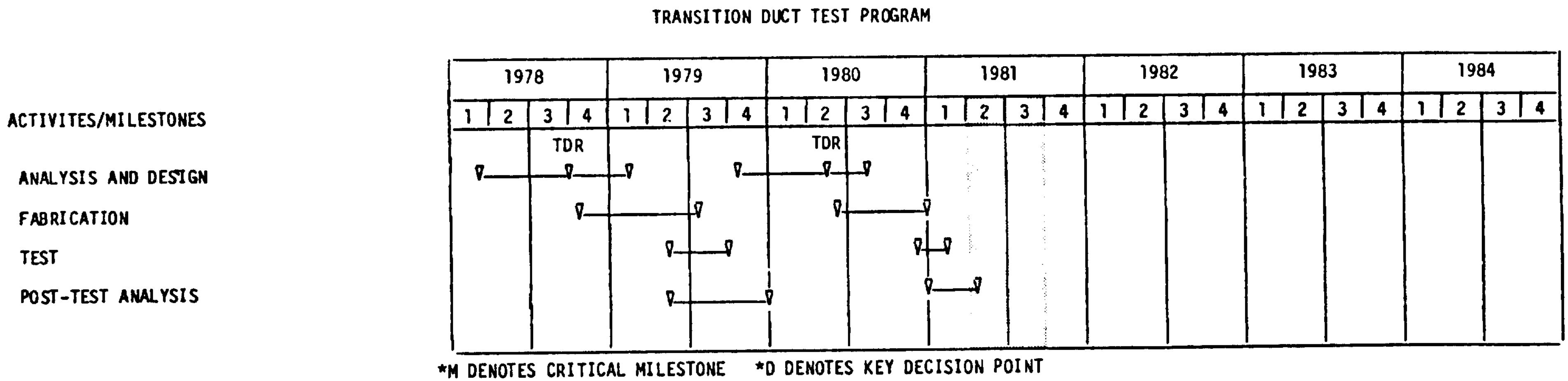

Figure 80 Transition Duct Test Program Work Plan Schedule

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- The pressure coefficient along the duct outer diameter wall indicated that the desired diffusion was attained across the strut. Strut airfoil pressure distribution data indicated an unloaded condition with no separation.

- Inlet, strut exit, and first vane exit air angle data bracket design values with some overturning in the root areas. With a five-degree off-design inlet swirl angle, the struts returned the unturned flow to within one degree of the design point swirl.

- The measured inlet turbulence was 2 percent versus a 4 percent rig design prediction.

Second Duct Model:

- The transition duct inner and outer wall loadings for the design point showed that the air flow diffuses from the hot strut leading edge to the hot strut trailing edge. The flow then accelerates through the low-pressure turbine inlet guide vane. These wall loadings indicate the flow to be separation free.

- Preliminary mass-averaged air angle spanwise data profiles indicated the average inlet angle had 1.6° more swirl than design, the average strut exit angle had 2.8° less swirl than design, and the second vane average exit angle had 1.2° less swirl than design.

- With these aerodynamics, the transition duct and hot strut loss was 1.59 percent $\Delta P/P$ compared to the design prediction of 1.5 percent $\Delta P/P$. The second vane measured loss was 0.49 percent $\Delta P/P$.

3.2.8.4.3.3.2 Current Technical Progress

Post-test analysis of the second model test data was completed during the current reporting period and is discussed in the following paragraphs. A draft of the technology report covering all program efforts is currently being prepared for submittal to NASA.

The post-test analysis effort focused on performance of the second transition duct model tested at an off-design inlet swirl angle to simulate high-pressure turbine exit flow characteristics at off-design operating conditions.

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The mass-averaged air angle spanwise data profiles are compared to (1) design predictions and (2) results from design point inlet swirl angle tests in Figures 81, 82, and 83. Data were recorded at the rig inlet, strut exit, and low-pressure turbine inlet guide vane exit (i.e., stations 1, 2, and 3 shown in Figure 84). Data in Figure 81 indicate that the measured off-design inlet swirl was approximately 6.7 degrees less than measured design point inlet swirl. The impact of this on strut exit air angle is shown in Figure 82, where measured strut exit swirl is approximately 2.8 degrees less than that represented by design point data. Figure 83 indicates that measured inlet guide vane exit swirl at the tested off-design rig inlet conditions was within 0.75 degrees of design point data. This indicates that off-design high-pressure turbine operation will have little effect on the angle of the flow entering the first stage rotor of the low-pressure turbine.

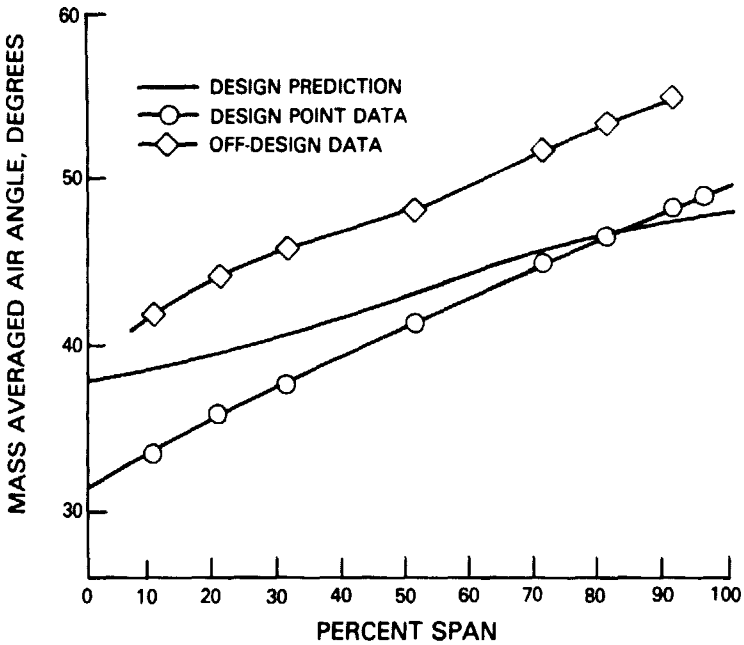

Figure 81 Inlet Air Angle - Station 1

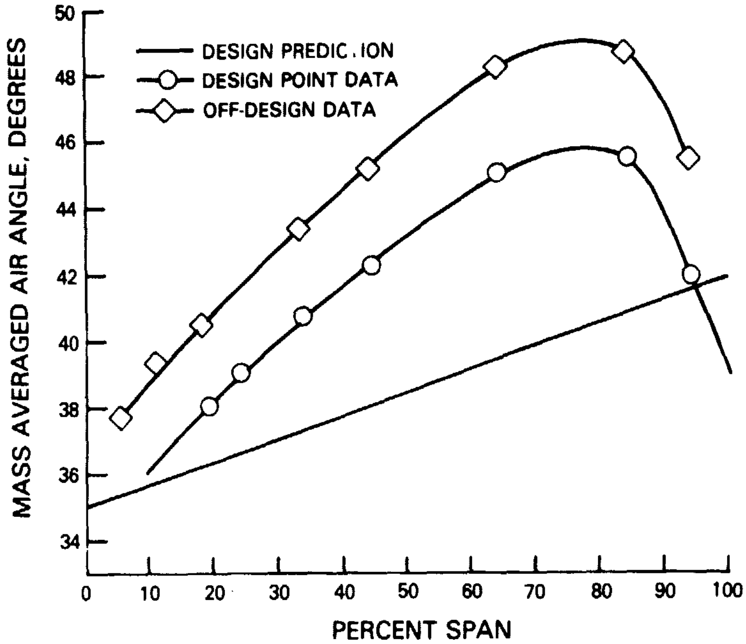

Figure 82 Hot Strut Exit - Station 2

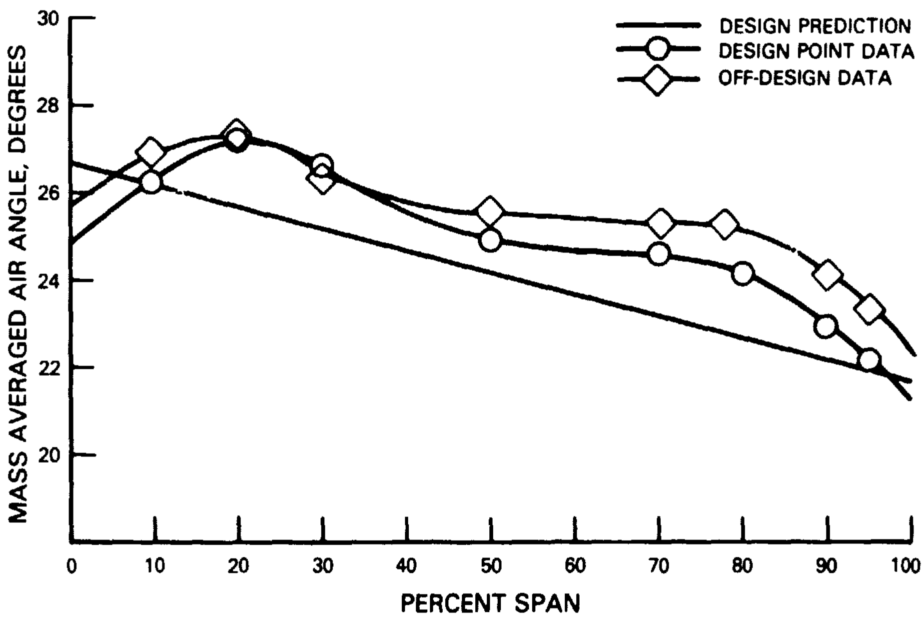

Figure 83 Inlet Guide Vane - Station 3

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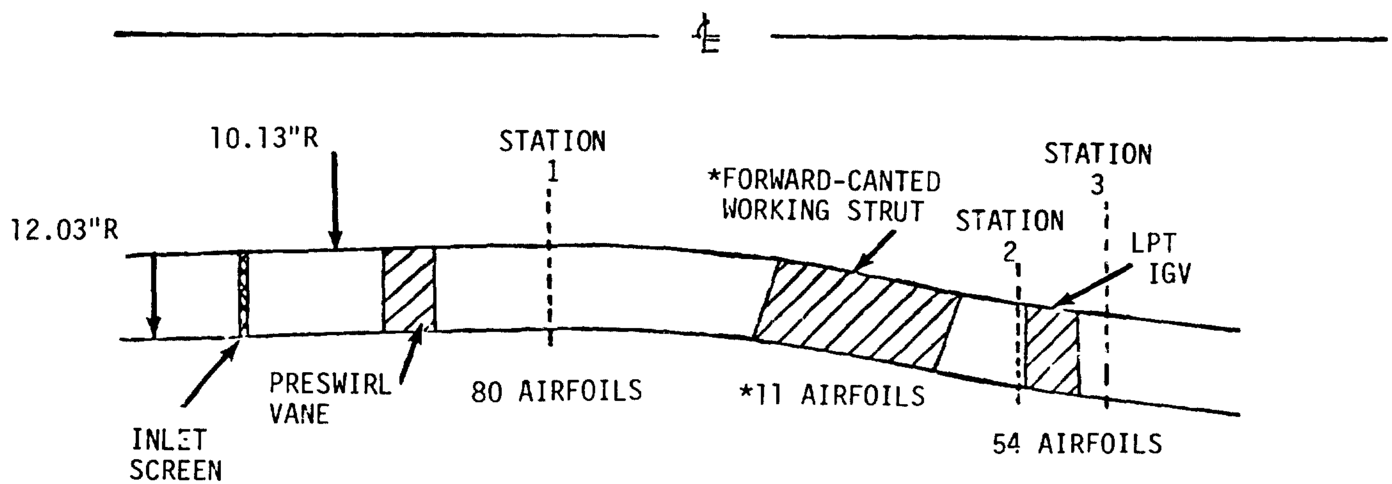
LOW PRESSURE TURBINE TRANSITION DUCT RIG - SECOND BUILD

SCALED FROM ENGINE

SCALE FACTOR = 0.7434



*ANNULUS AREA RATIO (A1/A2) = 1.57
*FLOW AREA RATIO = 1.42

(*) - CHANGE FROM FIRST BUILD

Figure 84 Low-Pressure Turbine Build 2 Transition Duct

The transition duct outer and inner wall loadings are shown in Figures 85 and 86, respectively. These data, compared to the design point data, show the reduction in diffusion through the strut due to the change in inlet angle. The flow then accelerates through the low-pressure turbine inlet guide vane. These wall loadings indicate the flow to be separation free.

With these aerodynamics, the transition duct and hot strut loss was 1.37 percent Δ Pt/Pt compared to the design point loss of 1.59 percent. The second vane loss was 0.68 percent Δ Pt/Pt compared to the design point loss of 0.49 percent.

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 sub-task. 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 Aircraft and airframe subcontractors. The design resulting from this refinement process is fed into the model test program. The final refinements to the mixer design are completed in the Task 4 analysis and design work package. A test facsimile is fabricated and tested in Task 4.

The Mixer Model supporting technology program consists of an extended period of scale model testing, starting with the exhaust system basically defined in Task 1. The first model program (Phase I) evaluates pertinent major variables, with the results used for updating the full scale flight propulsion system design in late-1979. A second model program (Phase II) is conducted to refine the design, prior to defining the configuration for the integrated core/low spool tests. Each phase includes model design, fabrication, testing, and post-test analysis. During model analysis and design, configurations are selected for testing. Following NASA Project Manager approval of the configurations, detailed design and fabrication is initiated.

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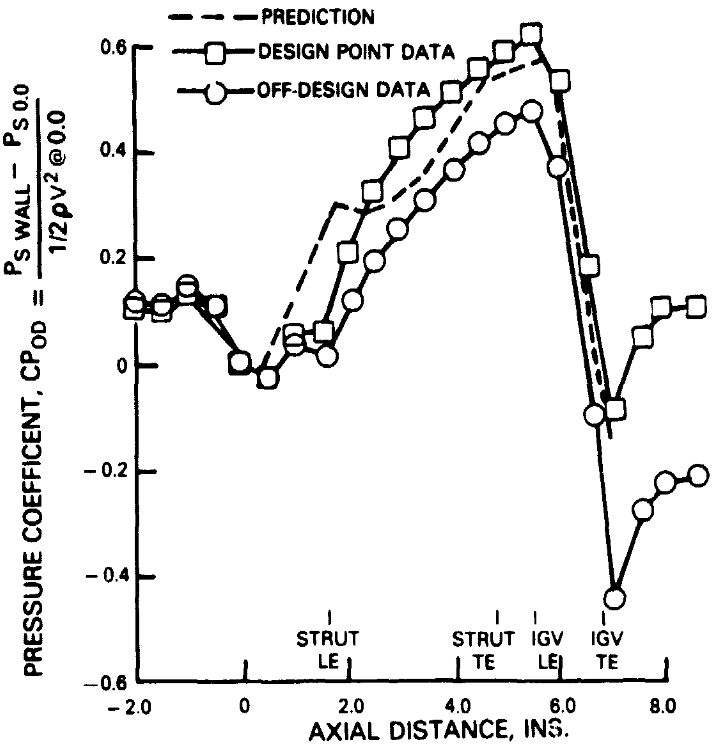

Figure 85 Transition Duct Rig (Build 2) Outer Wall Loading

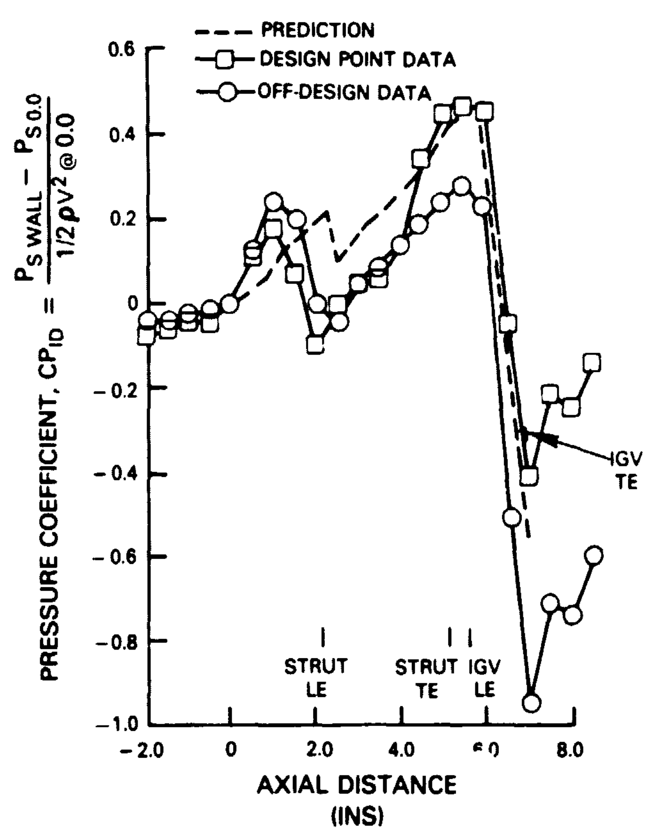

Figure 86 Transition Duct Rig (Build 2) Inner Wall Loading

The proposed test hardware is fabricated to Pratt & Whitney Aircraft specifications by FluiDyne Engineering Corporation. The tests are conducted in the FluiDyne Channel 11 static thrust facility. Nozzle thrust and flow coefficients are measured for each test point. Analysis of test data occurs during and after the test period and identifies the most promising configurations. The conclusions from these analyses are then used as final inputs to the flight engine exhaust mixer design definition.

All of the 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. A draft of the technology report summarizing the mixer model program effort has been prepared and submitted to NASA for review and approval.

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

3.3.1 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, and to obtain initial indications of structural integrity. Testing of the second build determines overall system performance, emissions, and noise. The following goals have been established for these tests:

TSFC	0.342 lbm/hr-lbf (corrected to standard day)	
Emissions	1981 EPA 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 between task activities is shown in Figure 87.

First Build Integrated Core/Low Spool: The work plan structured to achieve the first build objectives is shown in Figure 88.

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, 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 any modifications to the number 3 bearing area required 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 and 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.

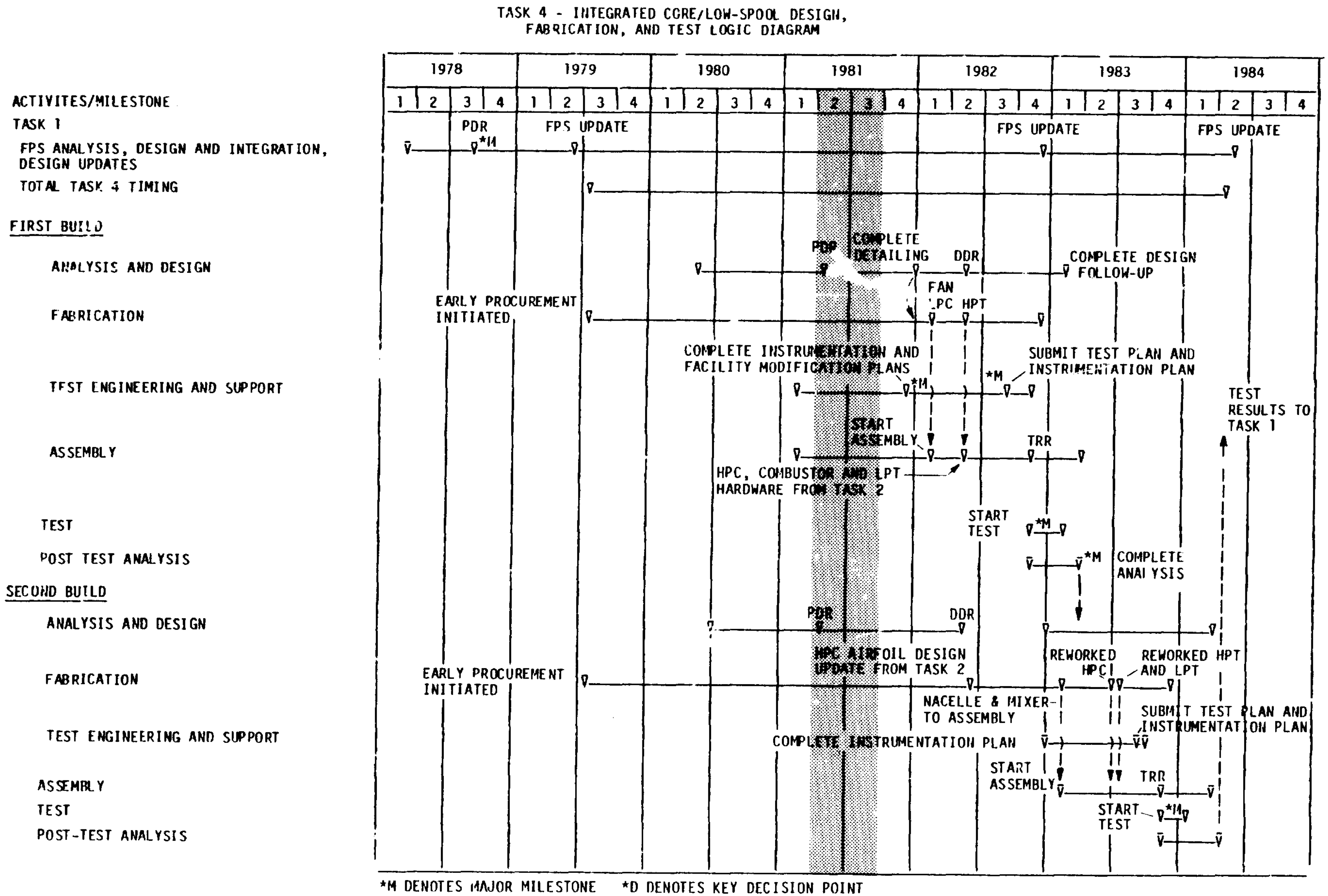

Figure 87 Integrated Core/Low Spool Design, Fabrication, and Test Logic Diagram

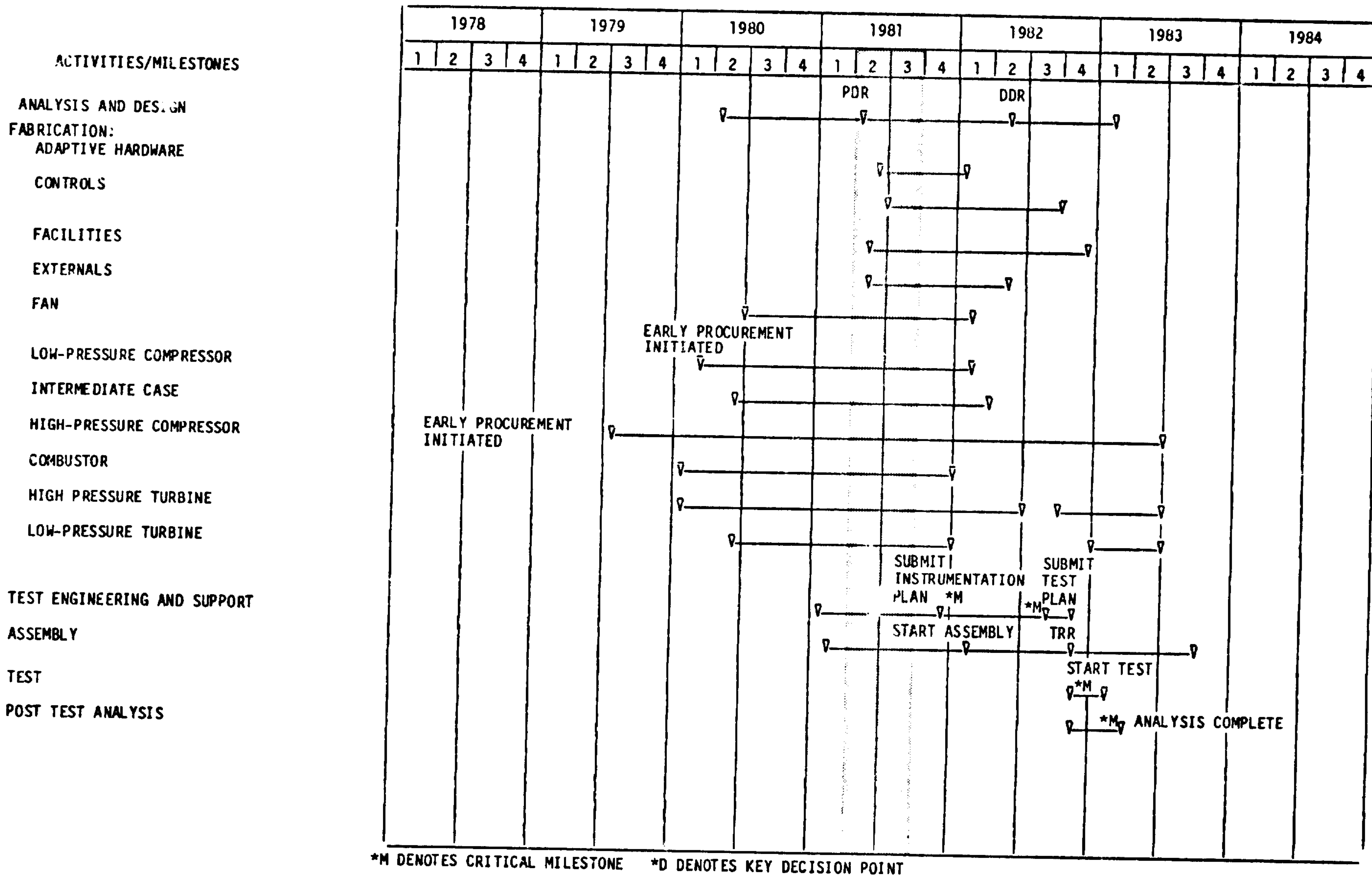

Figure 88 Integrated Core/Low Spool (First Build) Work Plan Schedule

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 rig test results. A final combustor hole pattern evolves through combustor component rig testing. Fabrication of hardware is to be performed by Pratt & Whitney Aircraft and approved vendors.

The integrated core/low spool is assembled for the first build with the shrouded fan rotor and associated hardware in an unmixed 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 the integrated core/low spool 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 maesurements and active clearance control system temperature, pressure, and flow measurements are used to compare system operation against pre-test 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.

Second Build Integrated Core/Low Spool: The work plan structured to achieve the second build objectives is shown in Figure 89.

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 the restagger of high- and low-pressure turbine blades and vanes if build 1 testing results define a requirement for such a change. 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 following 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, high- and low-turbine blade and vane attachment machining, and fabrication of an acoustically treated 'boilerplate' nacelle and test mixer. Any 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 test, the integrated core/low spool is disassembled for 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, currently available hardware from various Pratt & Whitney Aircraft engine models has been adopted to the integrated core/low spool as the design evolved.

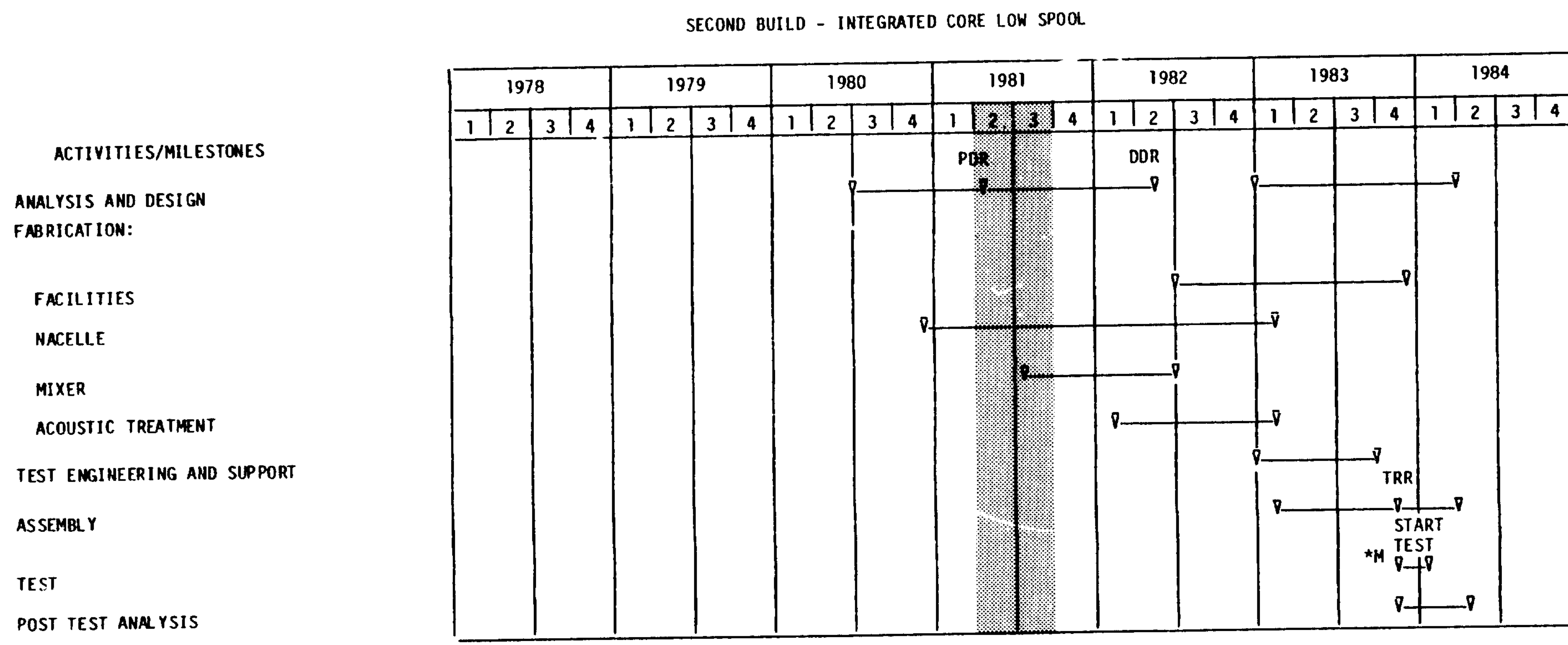

Figure 89 Integrated Core/Low Spool (Second Build) Work Plan Schedule

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

A full-size wooden mock-up of the integrated core/low spool was fabricated to facilitate the design of exterior plumbing. A schematic of the plumbing required for the active clearance control system is shown in Figure 90.

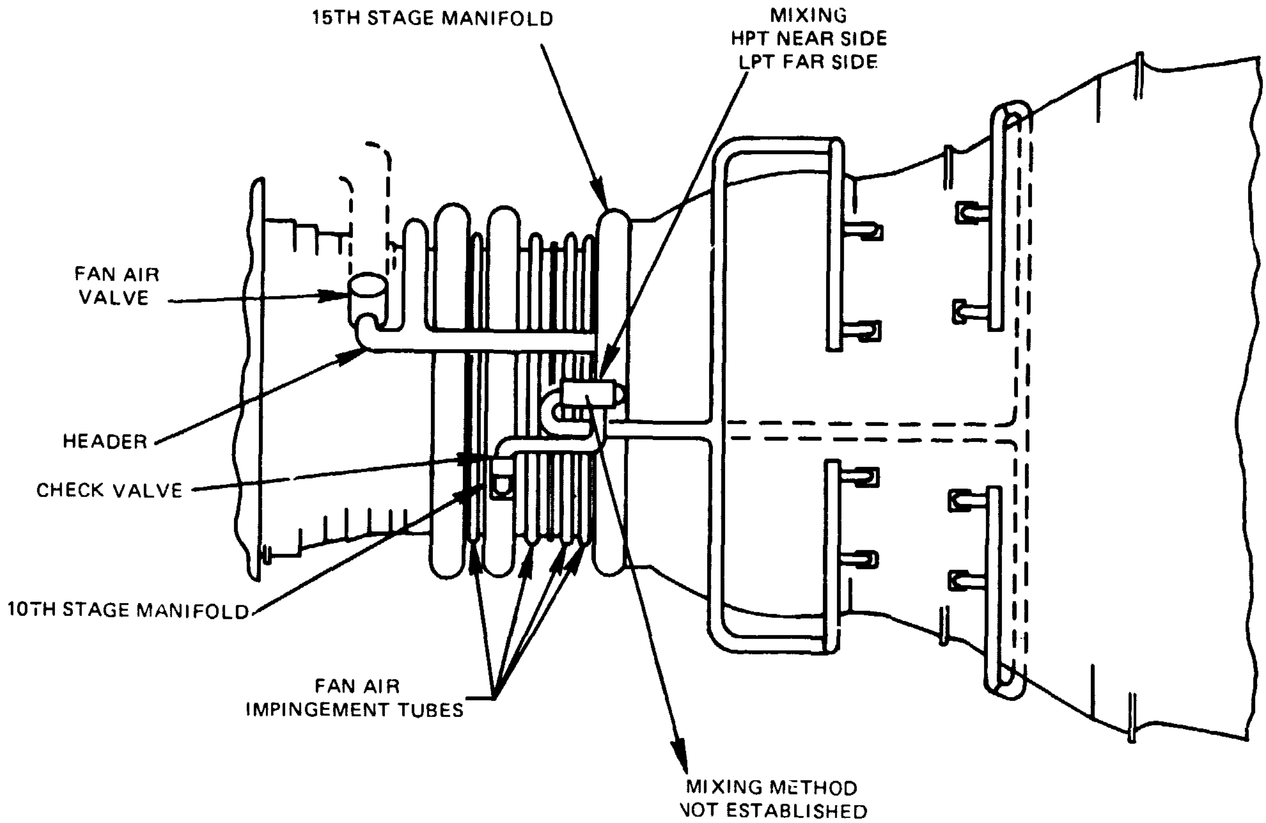
Each high- and low-pressure turbine mixing system incorporates four shutoff valves with metering orifices to regulate the flow of 15th stage air. Air from the 15th stage is mixed with 10th stage air before being ducted to the turbine case manifold. A temperature test was conducted on a currently available 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, 10th 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 for the integrated core/low spool was selected 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 in 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 are incorporated in the control for engine structural protection.

To accommodate the large amount of instrumentation required for testing, a bifurcated duct configuration was evolved. This air flow split facilitates access to the core and decreases losses that normally result from crossing the fan stream with instrumentation. The decision was made to modify the design of an existing JT9D bifurcated duct configuration for adaptation to the integrated core/low spool. Layout and reoperation of these ducts was initiated.



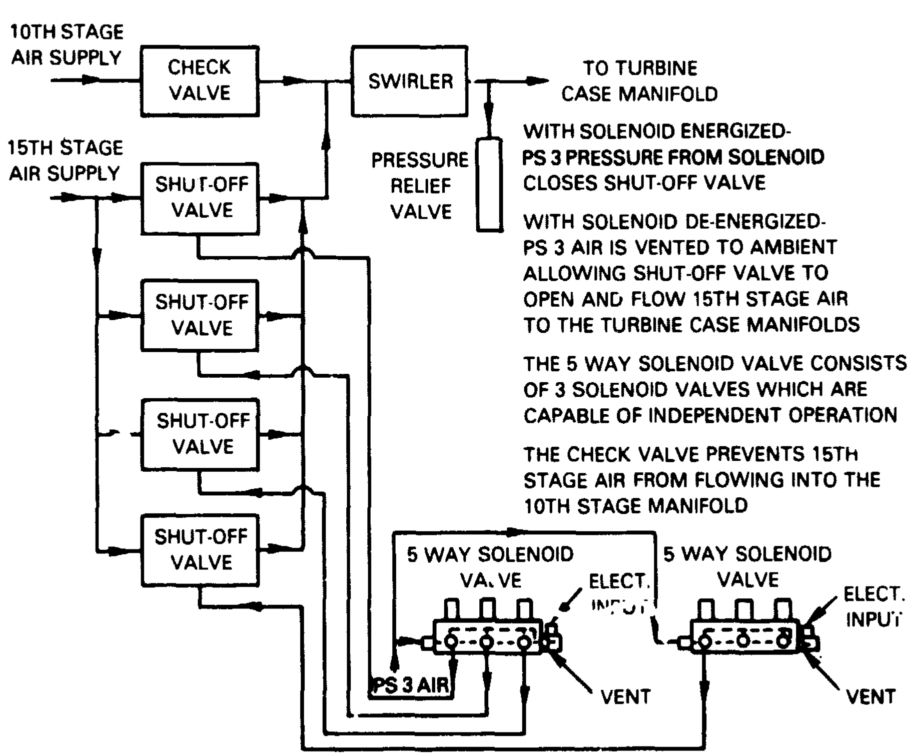


Figure 90 Integrated Core/Low Spool Active Clearance Control System

A preliminary design of the full-scale nacelle and its acoustically treated sections was completed. Rohr Industries submitted information detailing their fabrication cost and lead time estimates for Pratt & Whitney Aircraft review. Full-scale mixer design effort was initiated based on mixer model (Phase II) test results.

Instrumentation planning was initiated. An instrumentation roadmap and instrumentation requirements list were submitted to NASA for review and approval.

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.

3.3.3.1.2 Current Technical Progress

The Integrated Core/Low Spool Preliminary Design Review (PDR) was conducted at NASA-Lewis Research Center and approved in May 1981. Instrumentation plans for the integrated core/low spool were presented to NASA for review during the current reporting period. Instrumentation requirements for the integrated core/low spool are described in the Sixth Semiannual Status Report. These requirements remain unchanged. An engine roadmap identifying the instrumentation type and location, as shown in Figure 91, was also sent to NASA for review. This figure reflects updated information evolving from the ongoing design effort being directed toward the integrated core/low spool. Additional information submitted to NASA for review during the reporting period included an updated plan reflecting the current assembly schedule for both builds of integrated core/low spool.

Established test plans for the two builds, previously presented in the Preliminary Design Review, are shown in Figures 92 and 93. Test stands X-18 and C-11 were selected during the reporting period for the testing of integrated core/low spool builds 1 and 2. The capabilities and configuration for each facility are shown in Figures 94, 95, 96, and 97, respectively. A design effort was initiated during this reporting period to modify the X-18 test stand as required for the first test of the integrated core/low spool.

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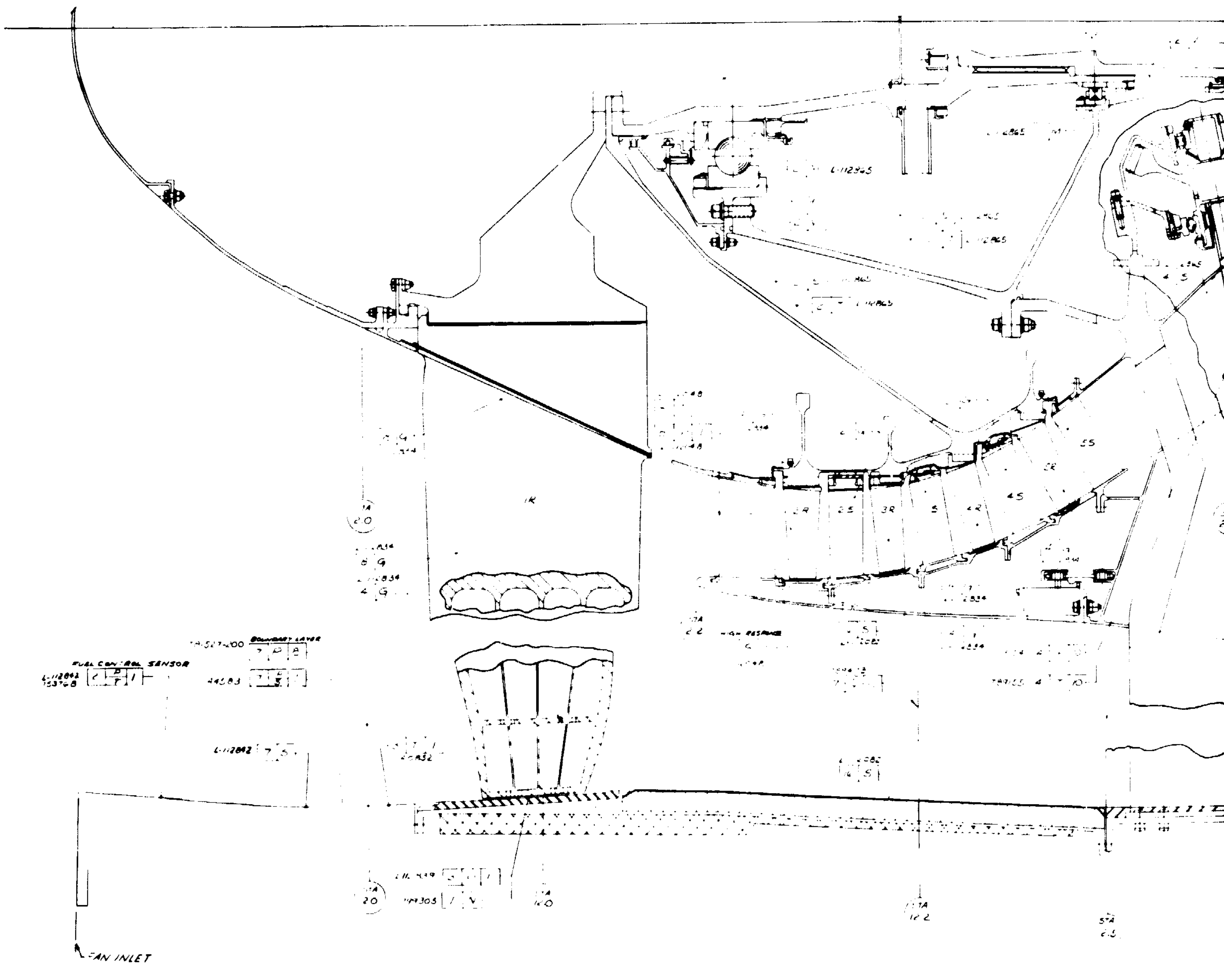

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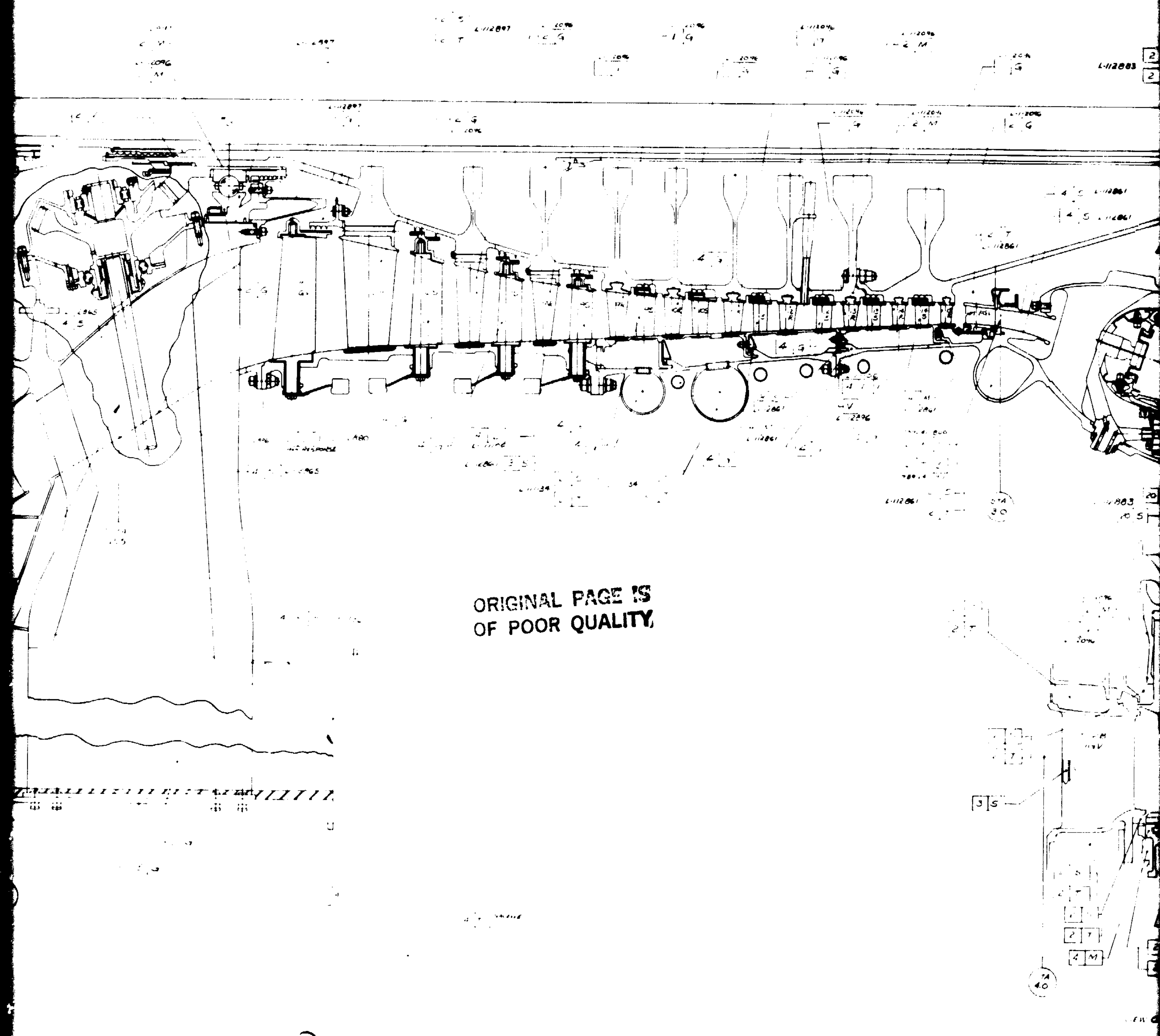

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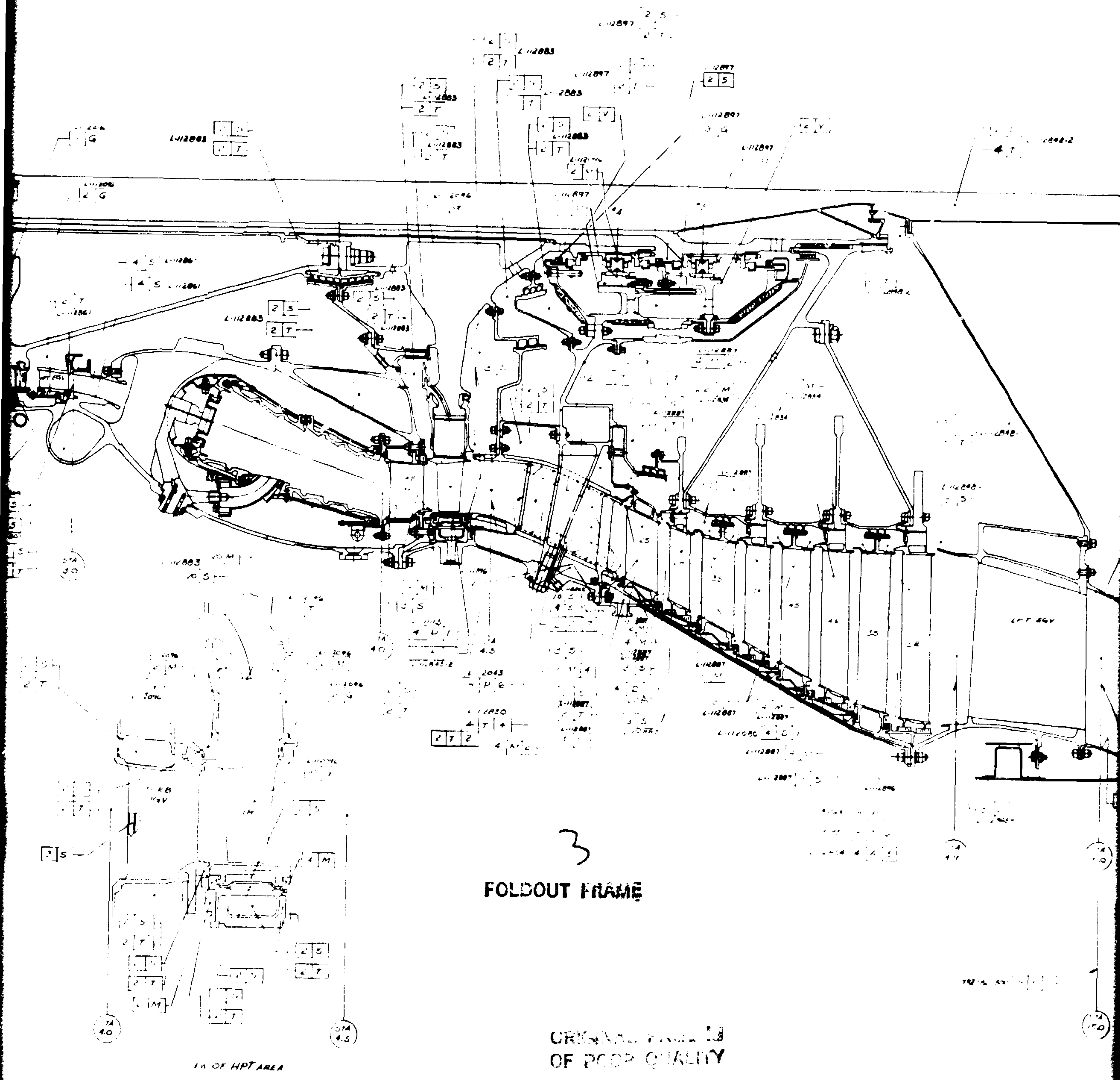

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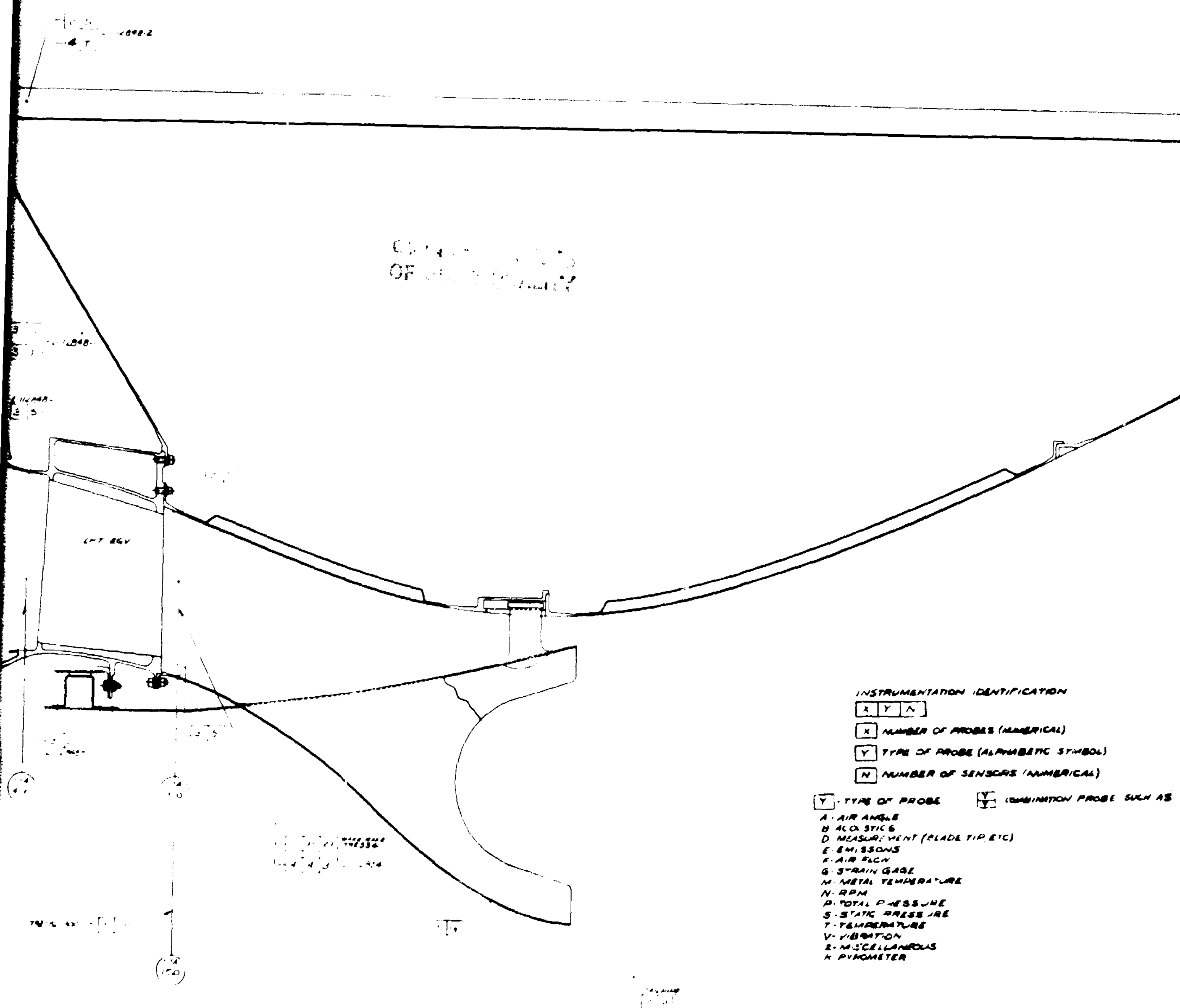

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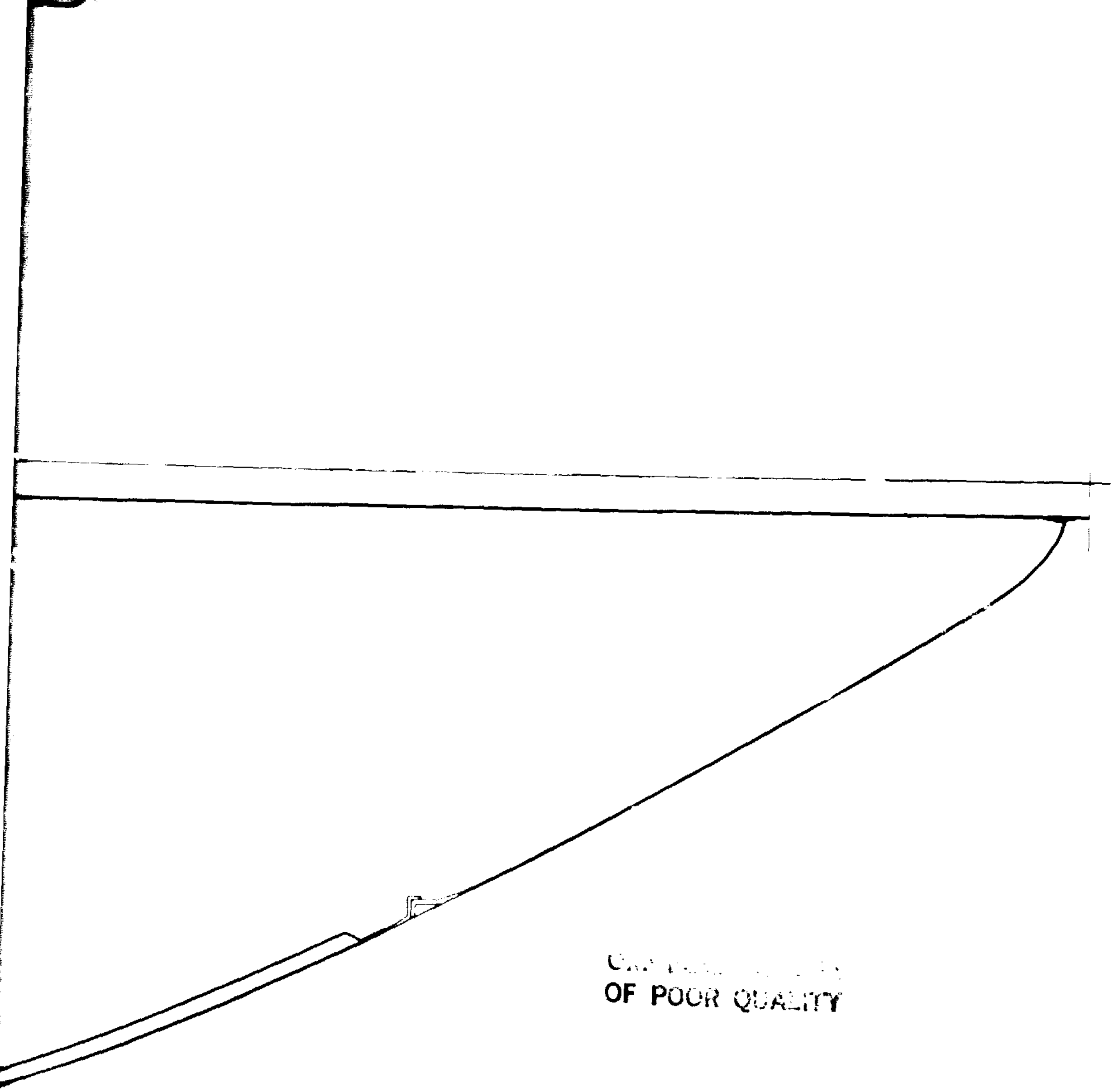
Figure 91 Integrated Core/Low Spool (Build 1) Eng
Identifying Instrumentation Type and Lo



Integrated Core/Low Spool (Build 1) Engine Roadmap
Identifying Instrumentation Type and Location

TEST SCHEDULE - BUILD 1

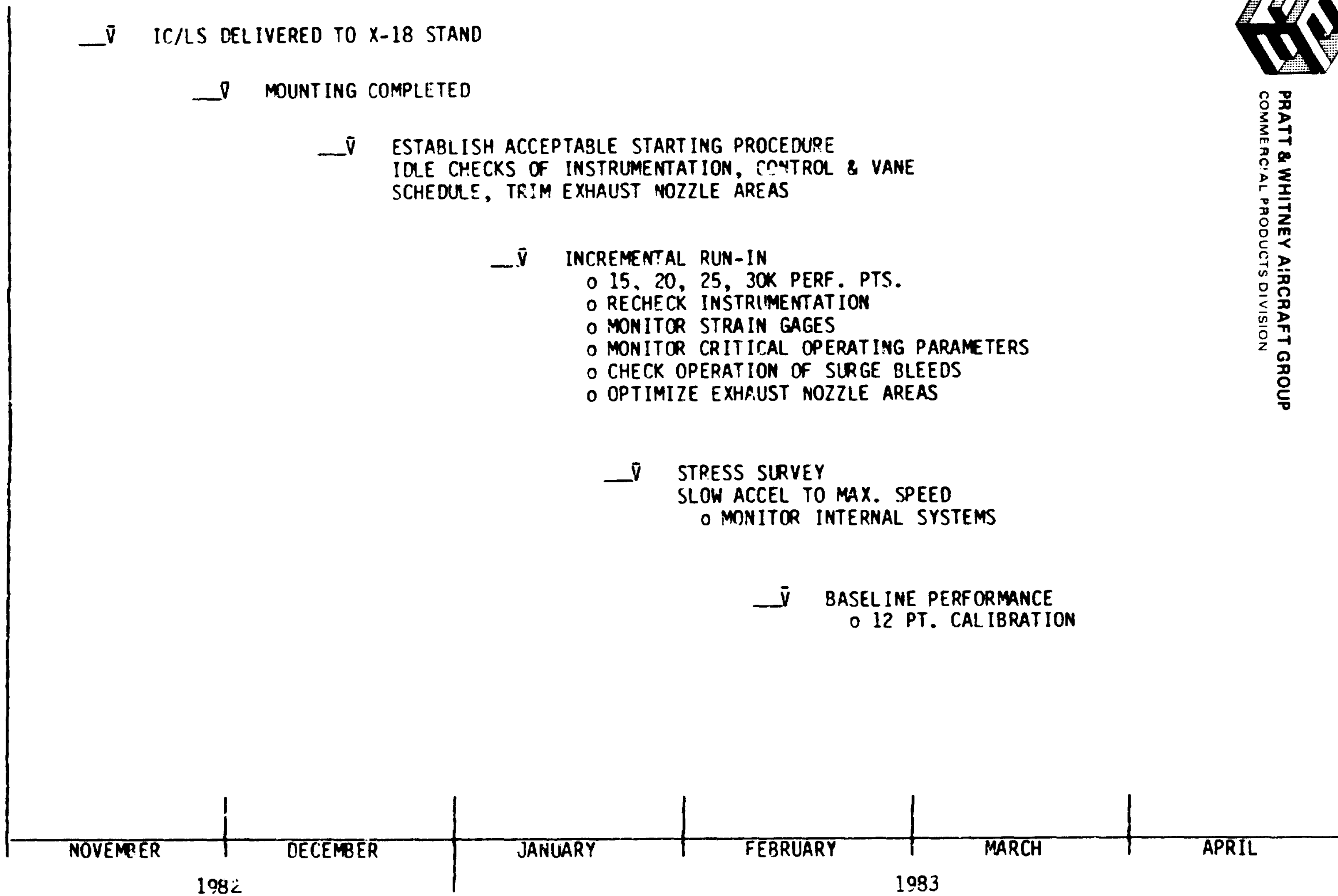


Figure 92   Integrated Core/Low Spool (Build 1) Test Schedule

TEST SCHEDULE - BUILD 2

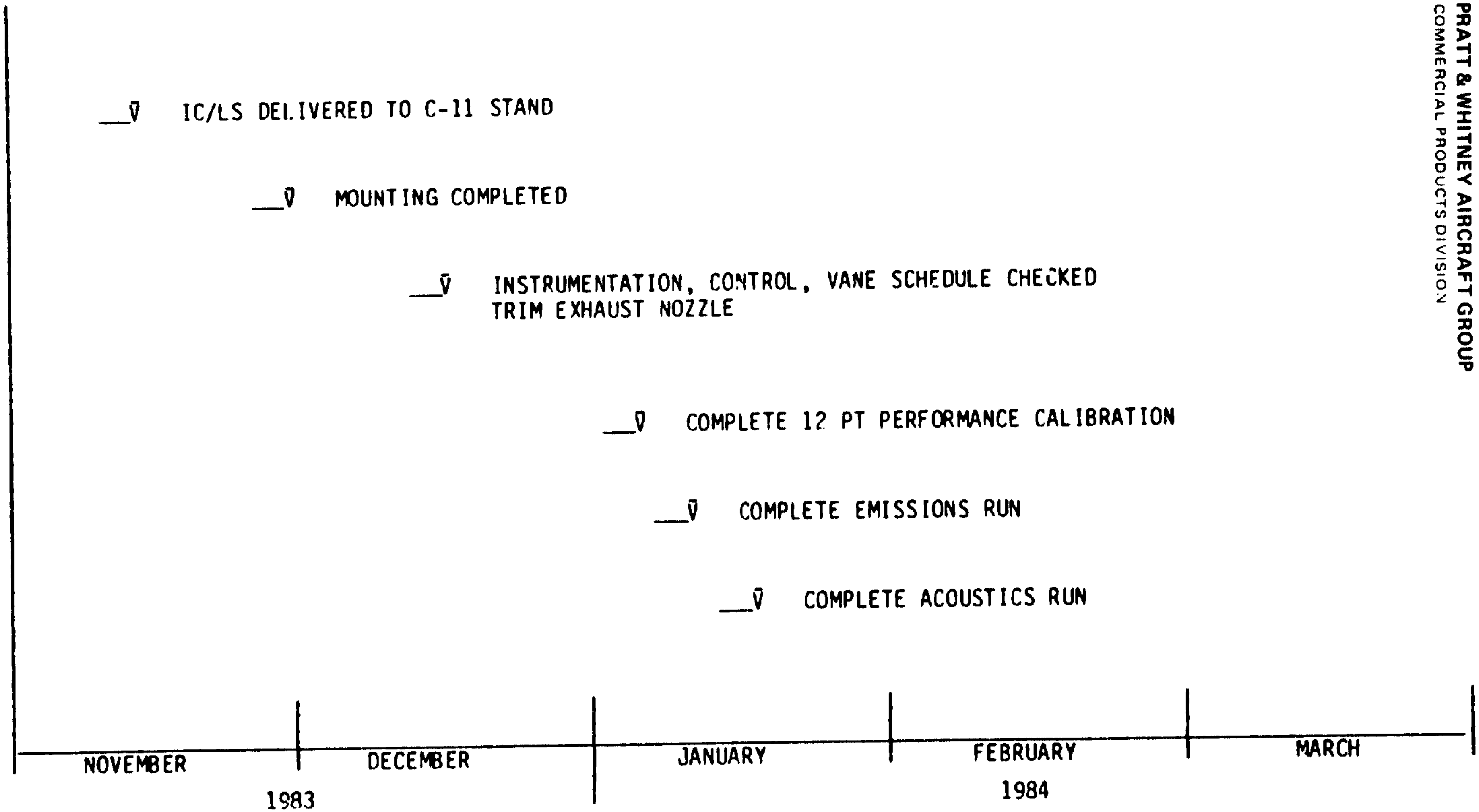


Figure 93 Integrated Core/Low Spool (Build 2) Test Schedule

C-3

TEST STAND - TEST 1

STAND DESIGNATION: X-18 E. HARTFORD, CT

STAND TYPE: STATIC SEA LEVEL

CONSTRUCTION: REINFORCED CONCRETE

INLET FLOW CAPACITY: 1550 PPS

THRUST MEASUREMENT: STRAIN GAGED LOAD CELL (CAPACITY 50,000 LBS.)

AUTOMATIC DATA SYSTEM:

600 TEMPERATURES   600 PRESSURES
+ MOBILE VAN AVAILABILITY

SERVICES AVAILABLE:

JET A FUEL	FIRE SAFETY
INDUSTRIAL WATER @ 125 PSIG	WATER FOG NOZZLE SYSTEM
DEMINERALIZED WATER UP TO 800 PSIG	FIRE HOSE SYSTEM
LOW & HIGH PRESSURE STEAM (25 & 140 PSIG)	ANSUL WAGON
SHOP AIR	

Figure 94 Integrated Core/Low Spool X-18 Test Stand Capabilities - Test 1

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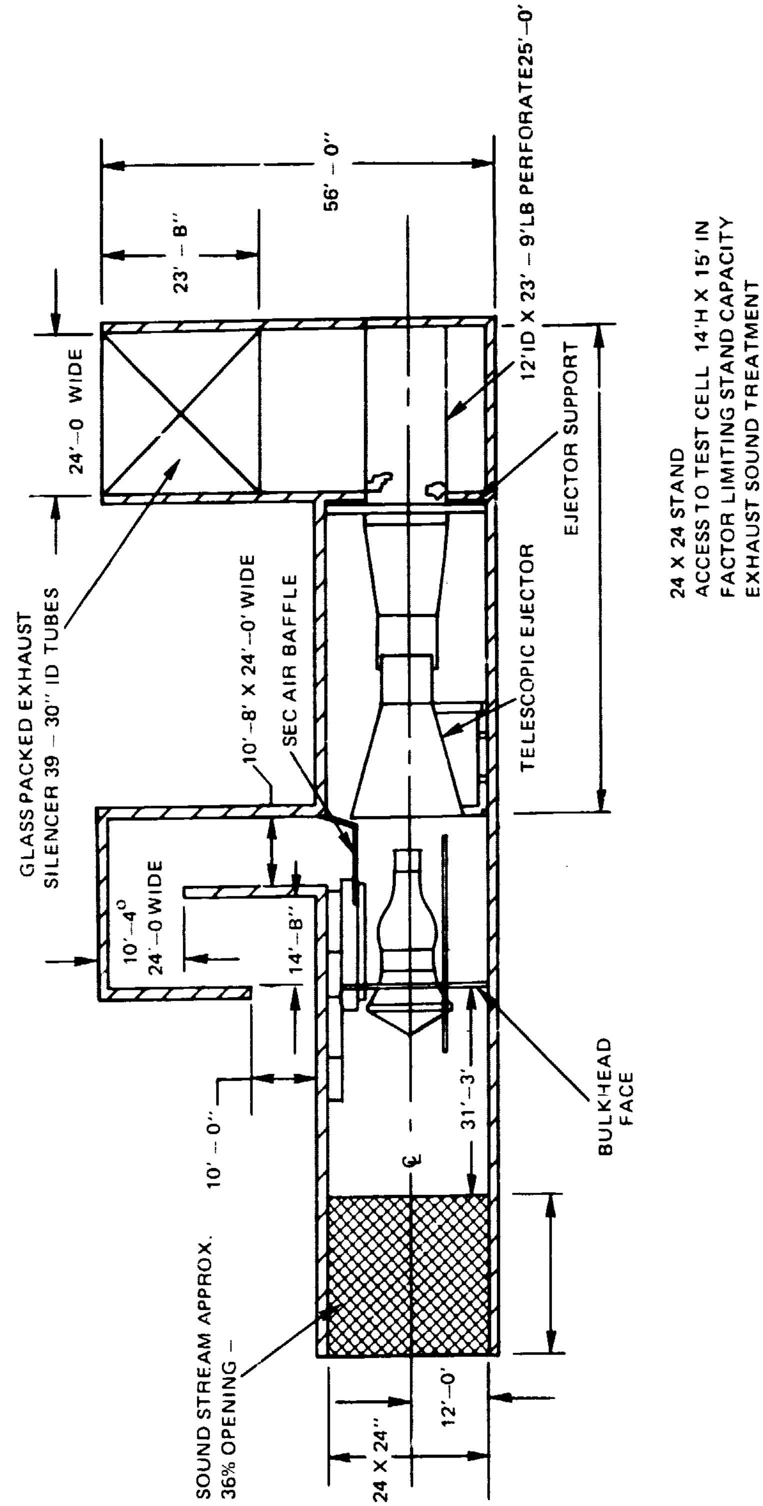

Figure 95 X-18 Test Stand

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TEST STAND - TEST 2

STAND DESIGNATION: C-11 W. PALM BEACH, FLA.

STAND TYPE: OUTDOOR STATIC SEA LEVEL WITH SPECIAL CAPABILITY FOR ACOUSTIC EVALUATION

CONSTRUCTION: SINGLE CIRCULAR TOWER WITH CANTILEVER BEAM WITH ENGINE ROTATIONAL CAPABILITY OF 270°

THRUST MEASUREMENT: STRAIN GAGED LOAD CELL (CAPACITY 100,000 LBS.)

AUTOMATIC DATA SYSTEM:

150 TEMPERATURES — 40 PRESSURES

+ MOBILE VAN CAPABILITY

SERVICES AVAILABLE:

JET A FUEL	FIRE SAFETY
SHOP AIR	WATER FOG NOZZLE SYSTEM
	FIRE HOSE SYSTEM

Figure 96 Integrated Core/Low Spool C-11 Test Stand Capabilities - Test 2

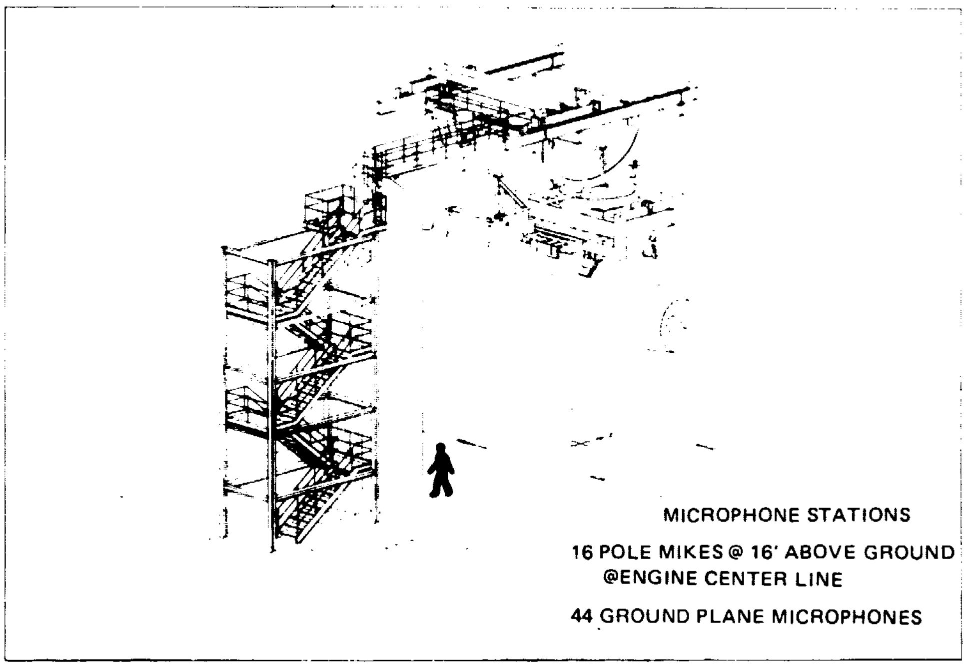

Figure 97 C-11 Test Stand

The primary design effort during the current reporting period continued to concentrate on the external adaptation of engine accessory hardware and the plumbing associated with these items. The full-size wooden mock-up of the integrated core/low spool was utilized to ensure no overlapping of hardware envelopes. Significant progress was also made in the area of instrumentation planning and coordination, including major station probe designs and instrumentation routing, and layout of the actual reoperation of JT9D bifucated ducts for adaptation to the build 1 integrated core/low spool.

Simulated mountings of external engine accessory hardware and associated plumbing completed to-date on the mock-up are shown in Figures 98, 99, 100, and 101. Mounting of all accessory hardware was completed during the current reporting period. 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. In addition, pneumatic and hydraulic external plumbing routes are currently being defined. Figure 102 represents the latest plumbing schematic.

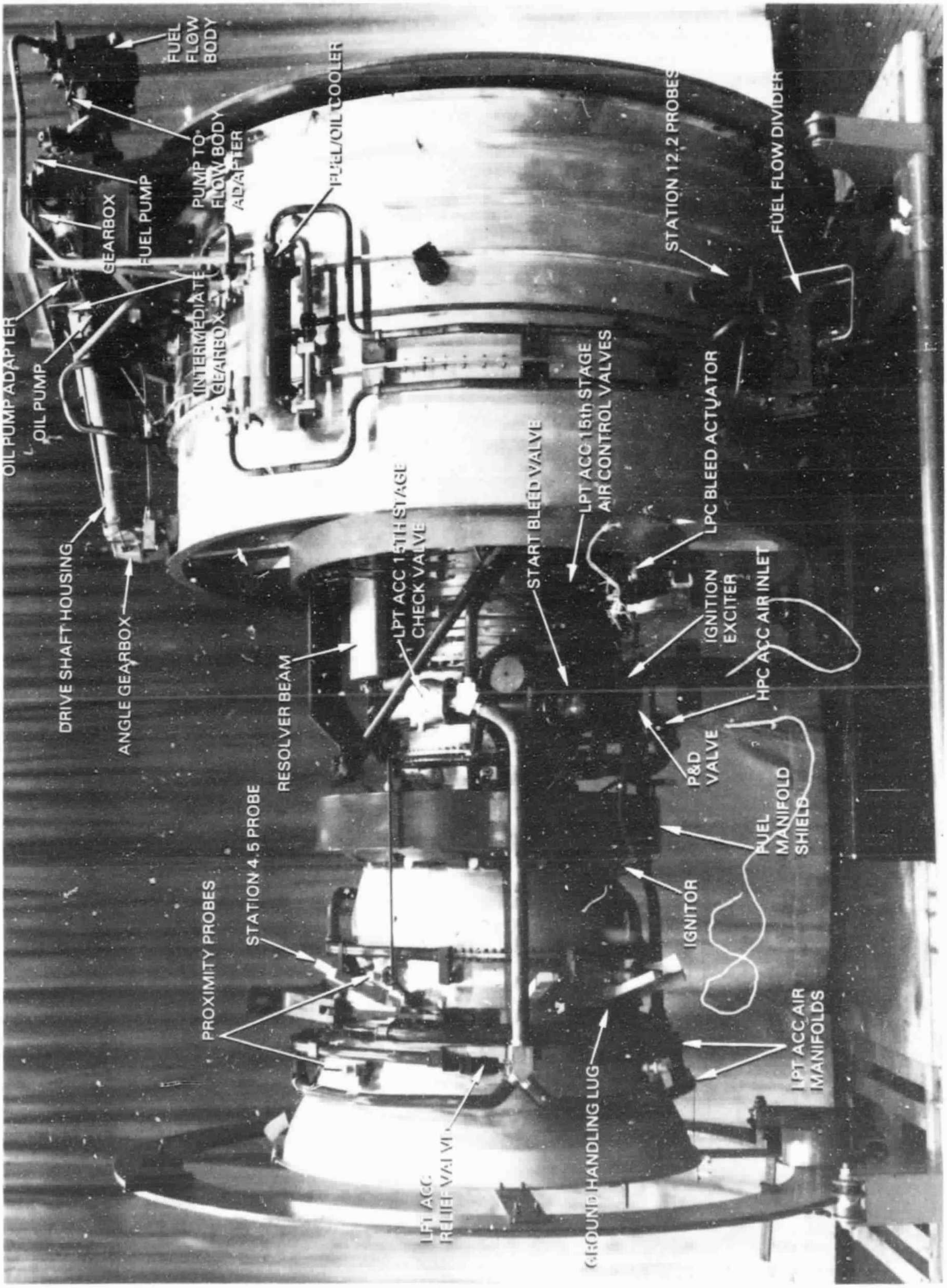

Figure 98 Integrated Core/Low Spool Mock-up Refelecting Simulated Mountings of External Engine Accessory Hardware and Associated Plumbing

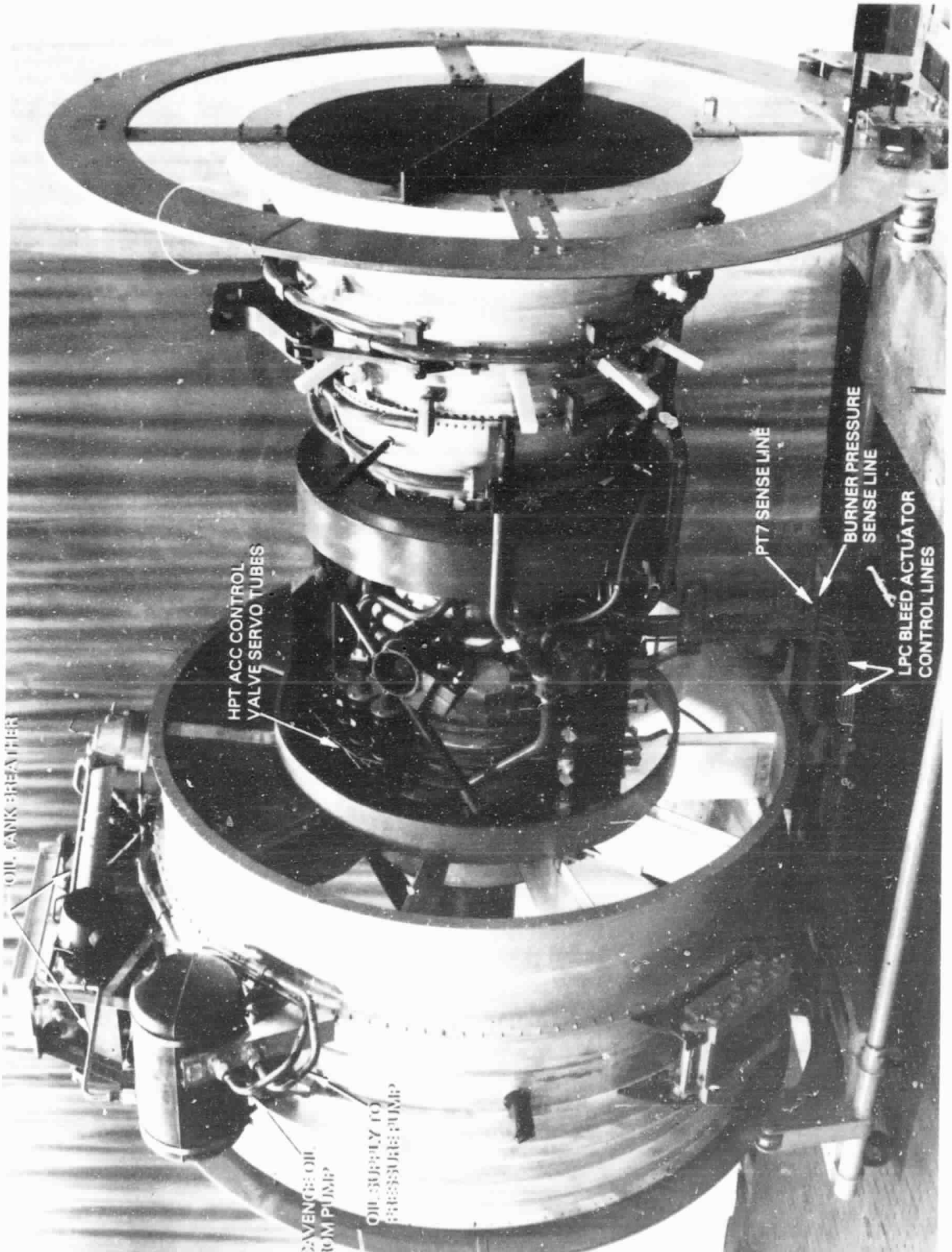

Figure 99 Integrated Core/Low Spool Mock-up Refelecting Simulated Mountings of External Engine Accessory Hardware and Associated Plumbing

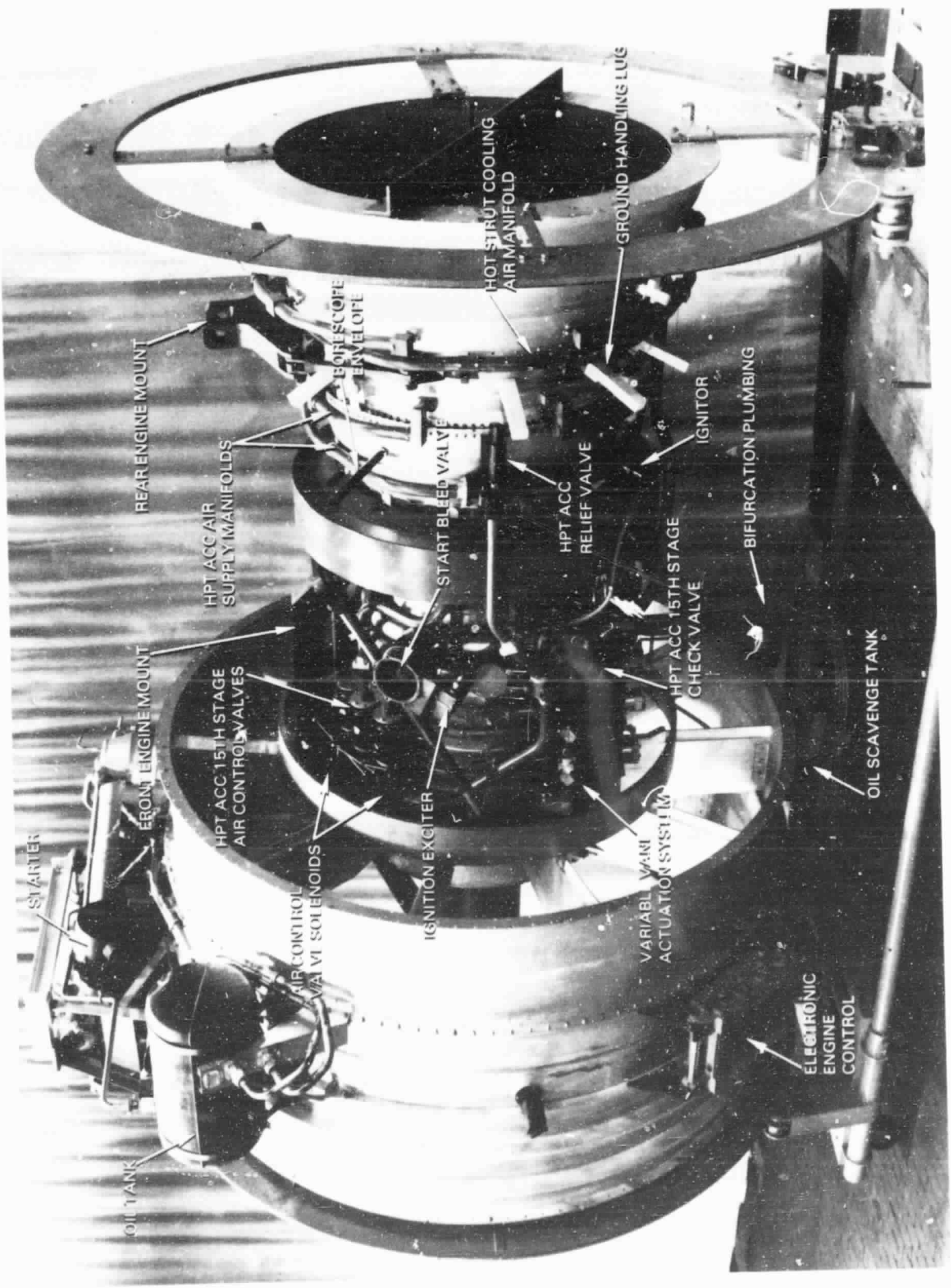

Figure 100 Integrated Core/Low Spool Mock-up Refelecting Simulated Mountings of External Engine Accessory Hardware and Associated Plumbing

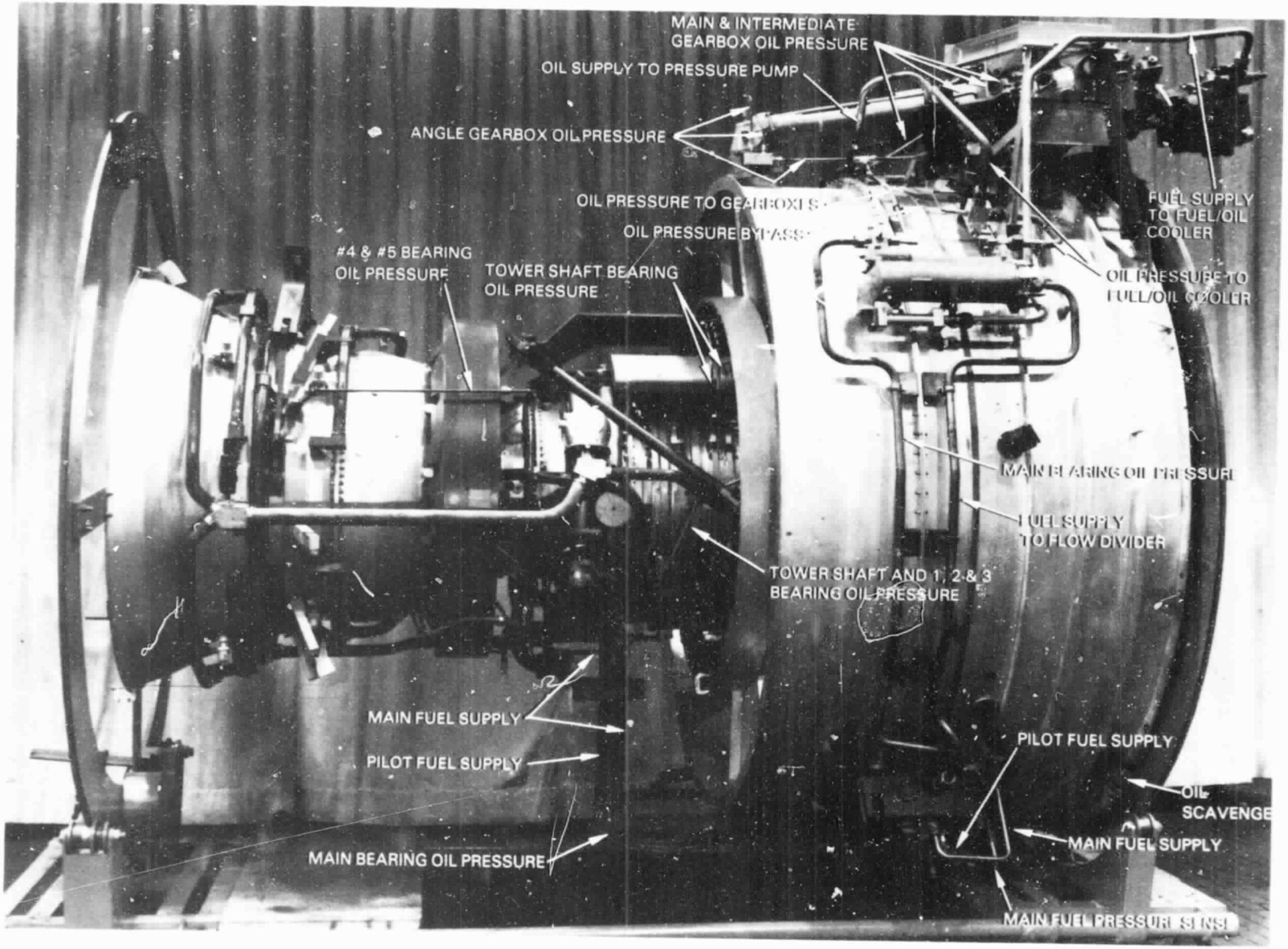

Figure 101 Integrated Core/Low Spool Mock-up Refelecting Simulated Mountings of External Engine Accessory Hardware and Associated Plumbing

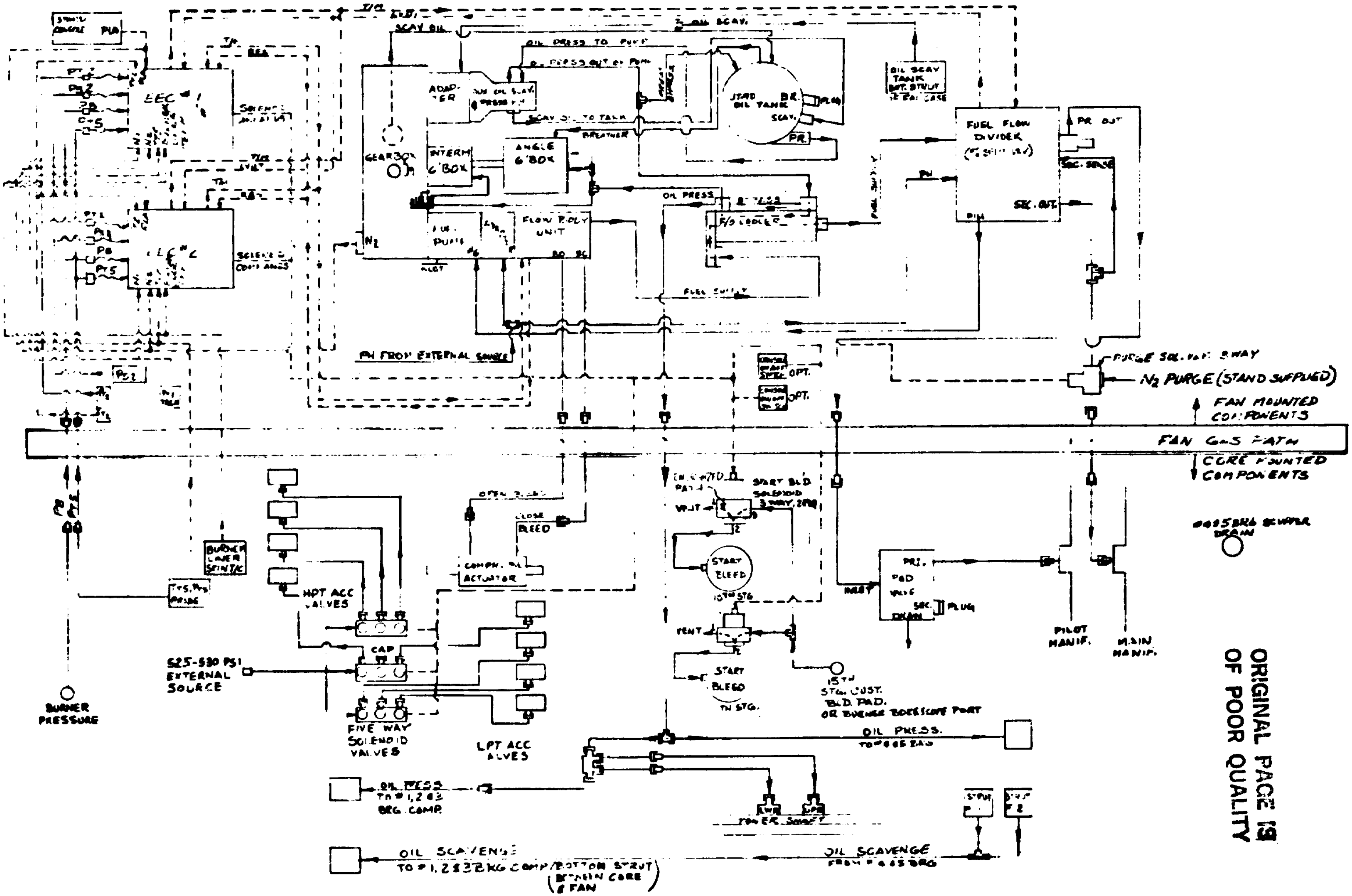

Figure 102 Integrated Core/Low Spool Plumbing Schematic

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All major instrumentation probe designs have been completed along with the high and low rotor routing to the telemetry and slip ring units. The packaging of the telemetry unit is shown in Figure 103 while Figure 104 illustrates the general arrangement of the low rotor slip ring unit. Static instrumentation routing is nearing completion with only a small number of bearing compartment items remaining to be routed.

The reoperation layout of the revised JT9D bifurcated duct was completed. This duct, designed to accommodate instrumentation by splitting the air flow into two equal segments, and other salient features of the integrated core/low spool build 1 configuration are shown in Figure 105. In conjunction with this arrangement, a one-piece fiberglass bellmouth/inlet case design is nearing completion. This same bellmouth/inlet unit, shown in Figure 106, will also be used for build 2 which features the full nacelle configuration. Negotiations with a vendor are currently underway for budgetary and planning cost and leadtime estimates for non-acoustically treated fiberglass D-ducts in the full nacelle configuration. In addition to these negotiations, a final decision for the incorporation of nacelle acoustical treatment shown in Figure 106 will be made in the first quarter of 1982.

The design effort directed toward the mixer, tailplug arrangement was continued during the reporting period. This design effort continues to be a scale-up of the technically successful Phase II mixer model configuration discussed under Task 2 - Mixer Component section in the Sixth Semiannual Status Report.

Specifications are currently being written to identify the requirements of the dual channel, full-authority electronic control fuel system. Computer programming of the electronic control fuel system to meet these specifications is progressing. The hardware procurement effort for associated valving and actuators continued.

In December 1981 a decision will be made on whether to incorporate into the fuel control system a centrifugal pump to supply fuel to a dual metering flow body designed to receive electrical commands from the control. This fuel control system is shown schematically in Figure 107.

3.3.3.2 Integrated Core/Low Spool - Fabrication

3.3.3.2.1 Summary of Work Previously Completed

Prior to the current reporting period, approval was requested from NASA for early procurement of long lead time raw material for adapting hardware, externals, and engine mounts. Fabrication efforts directed toward the major component hardware progressed satisfactorily during the reporting period and are summarized in the following paragraphs.

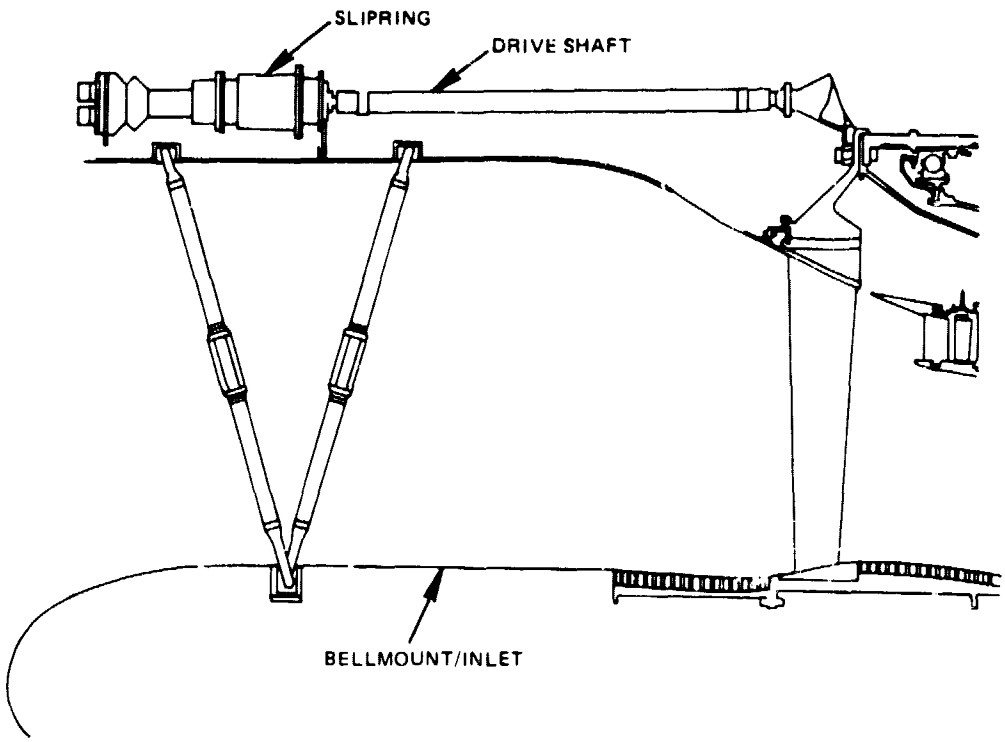

Figure 104 Low Rotor Slip Ring Unit Arrangement

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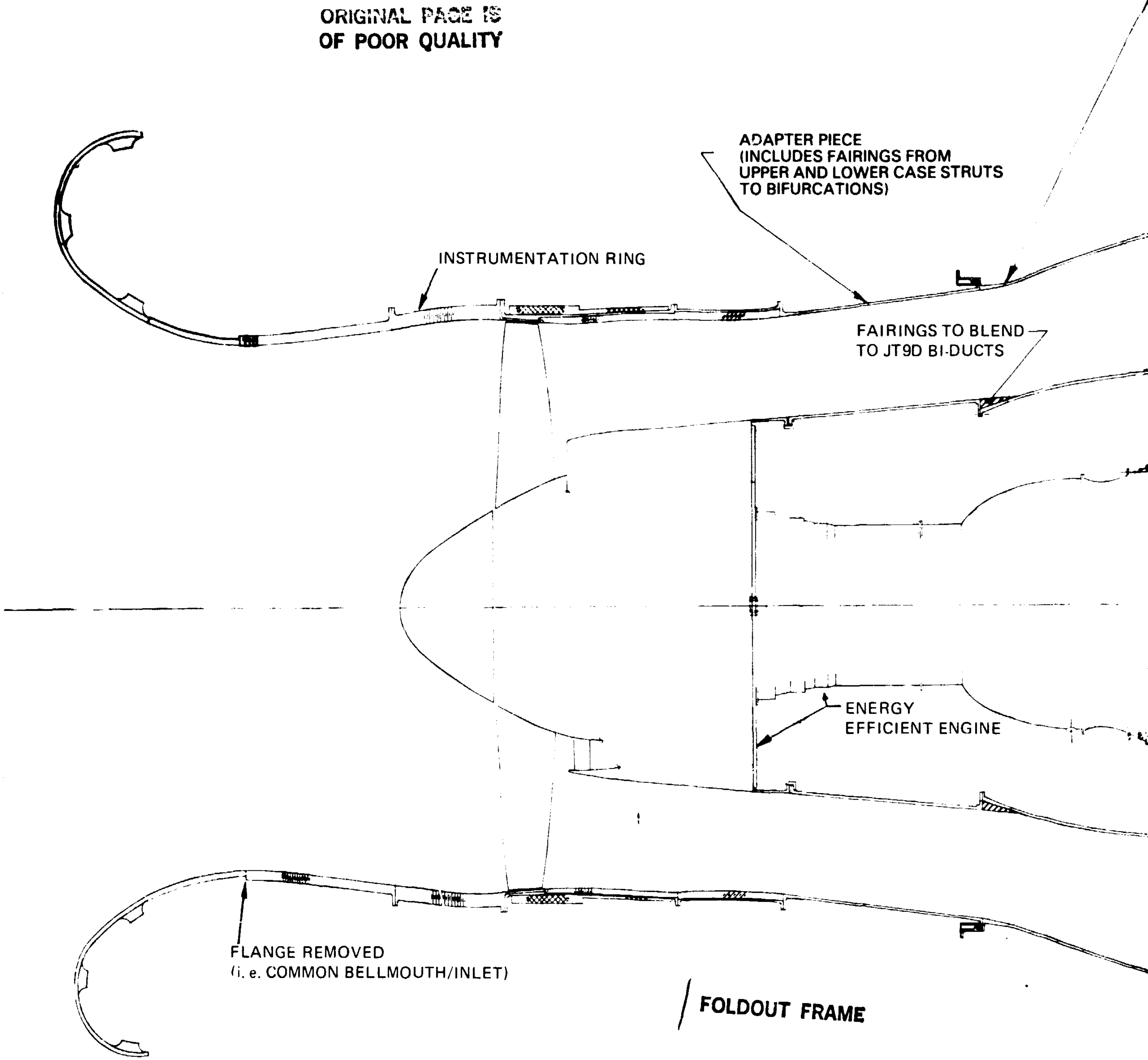

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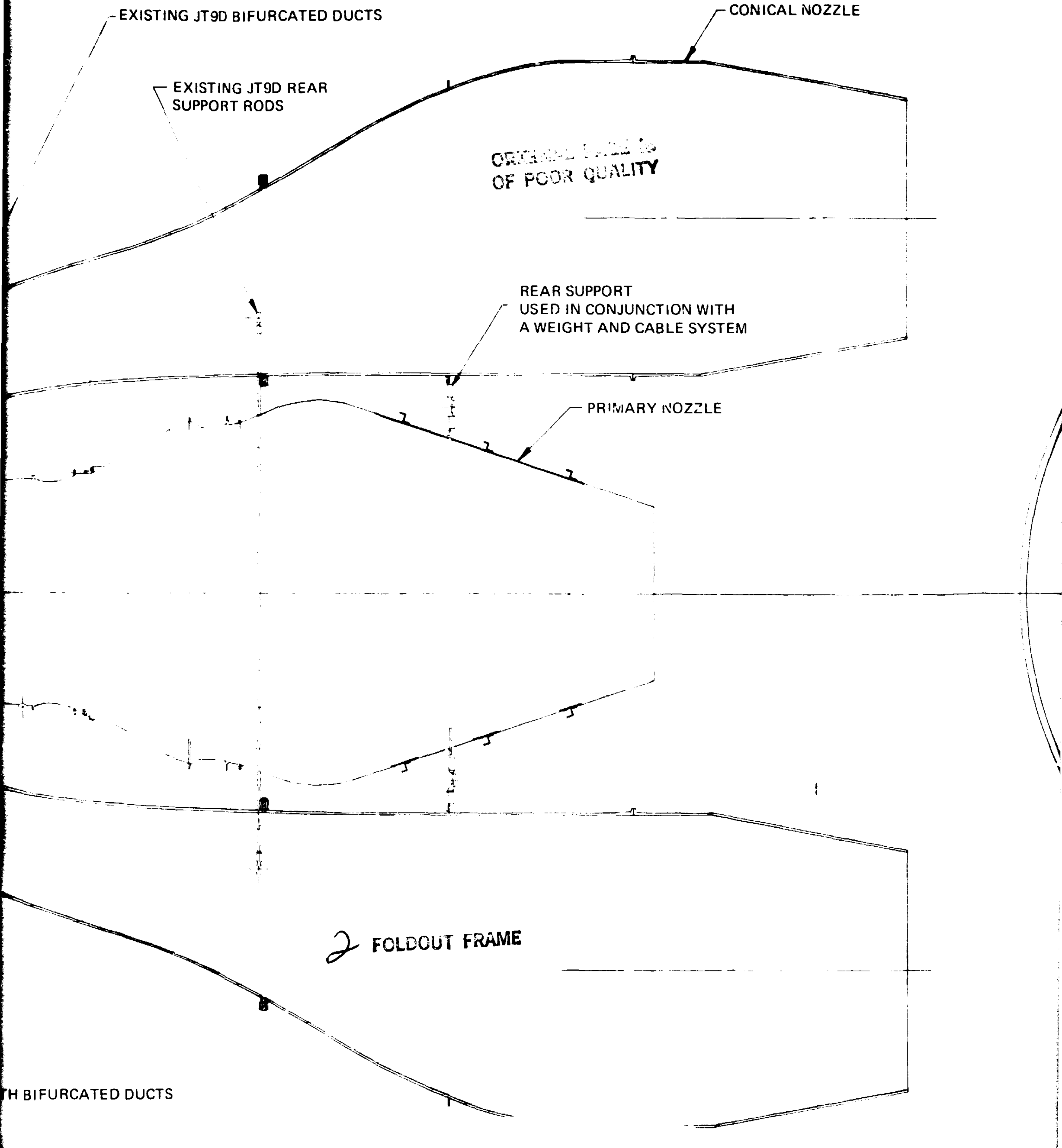
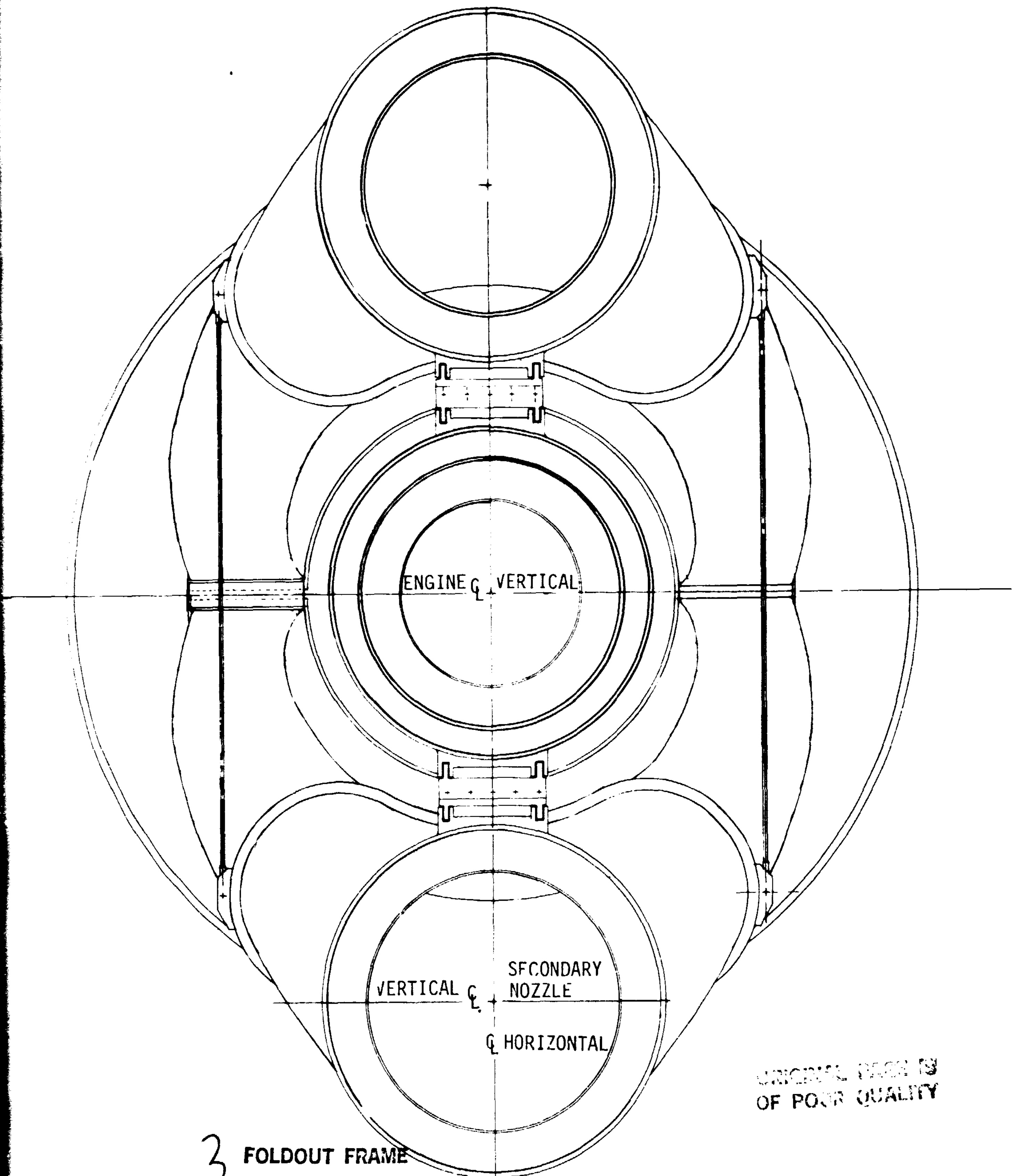
Figure 105 Integrated Core/Low Spool Build 1 Configuration Featuring Reoperated JT9D Bifurcated Duct

TOP VIEW WITH BIFURCAT


EXISTING JT9D BIFURCATED DUCTS
CONICAL NOZZLE
EXISTING JT9D REAR SUPPORT RODS
REAR SUPPORT USED IN CONJUNCTION WITH A WEIGHT AND CABLE SYSTEM
PRIMARY NOZZLE
H BIFURCATED DUCTS



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REAR VIEW

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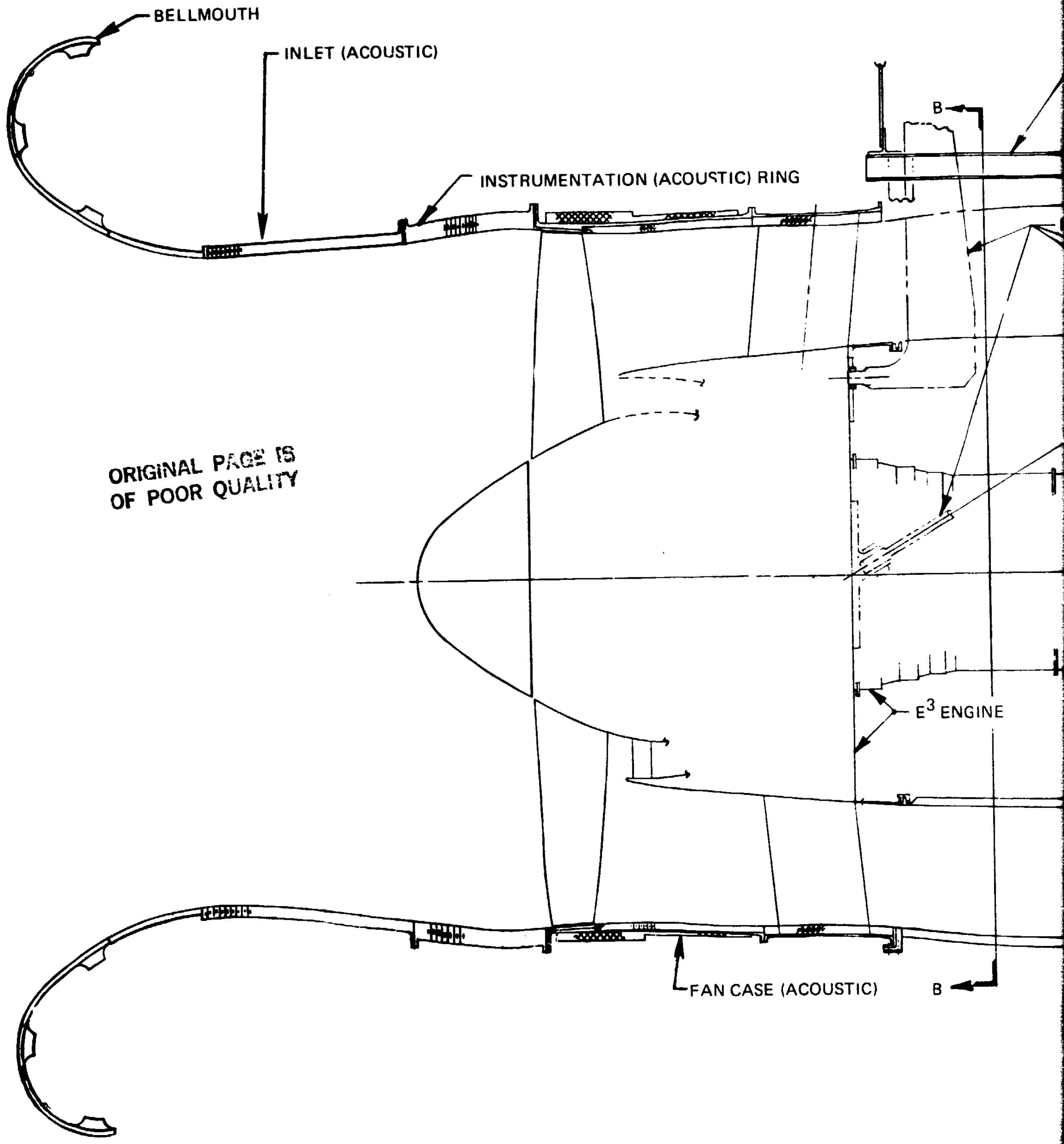

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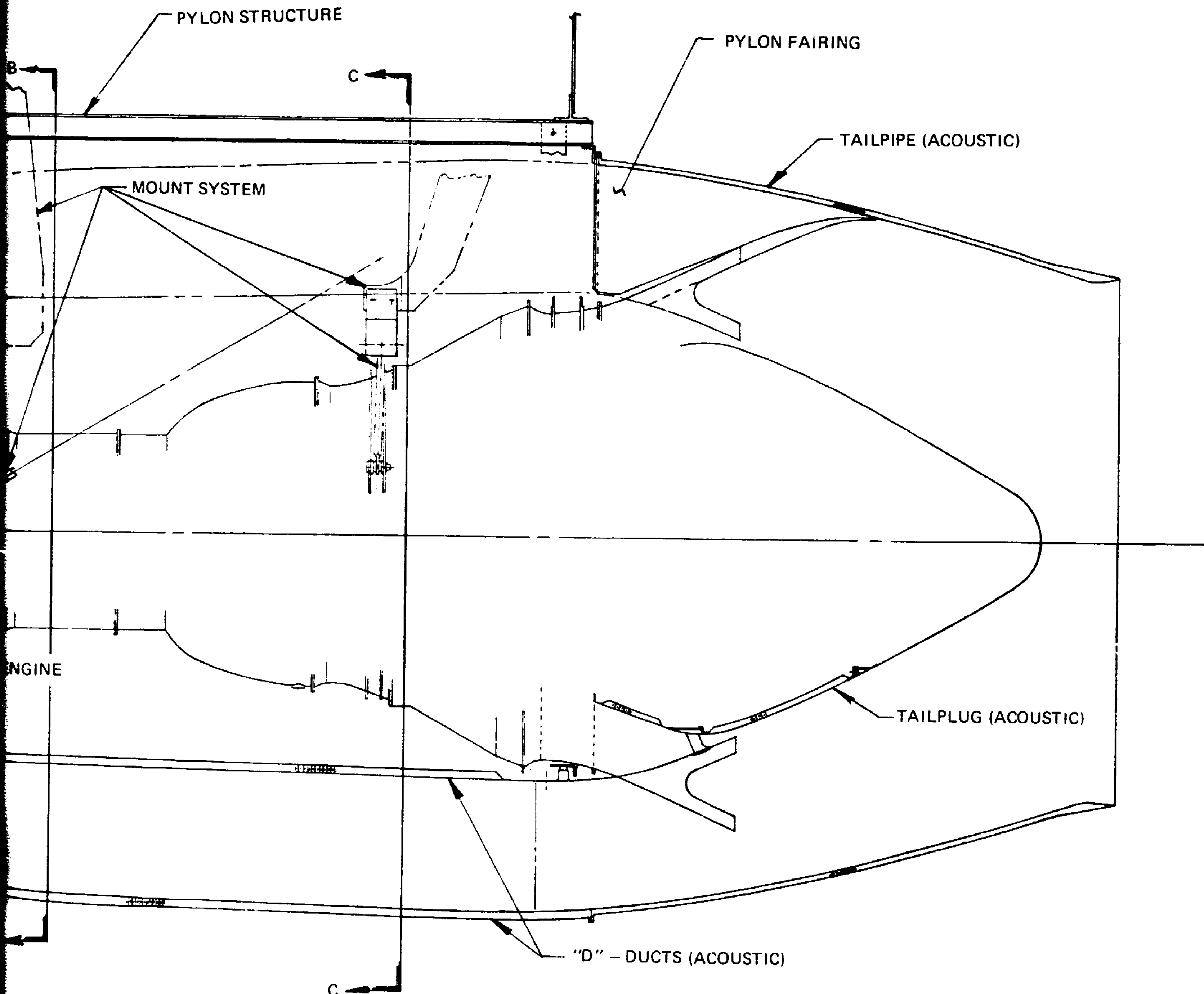

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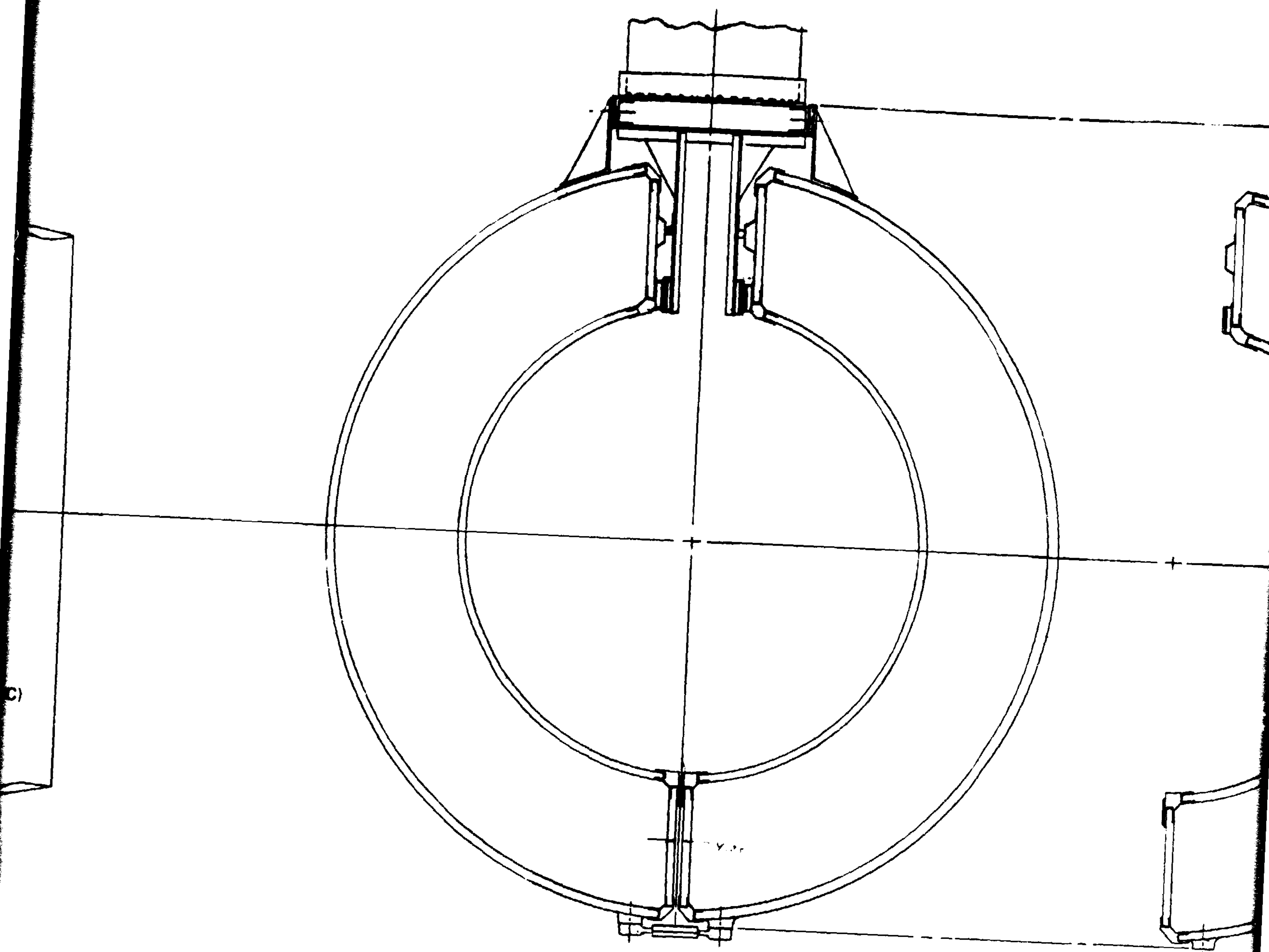

SECTION B-B

'D' DUCTS REMOVE HORIZON

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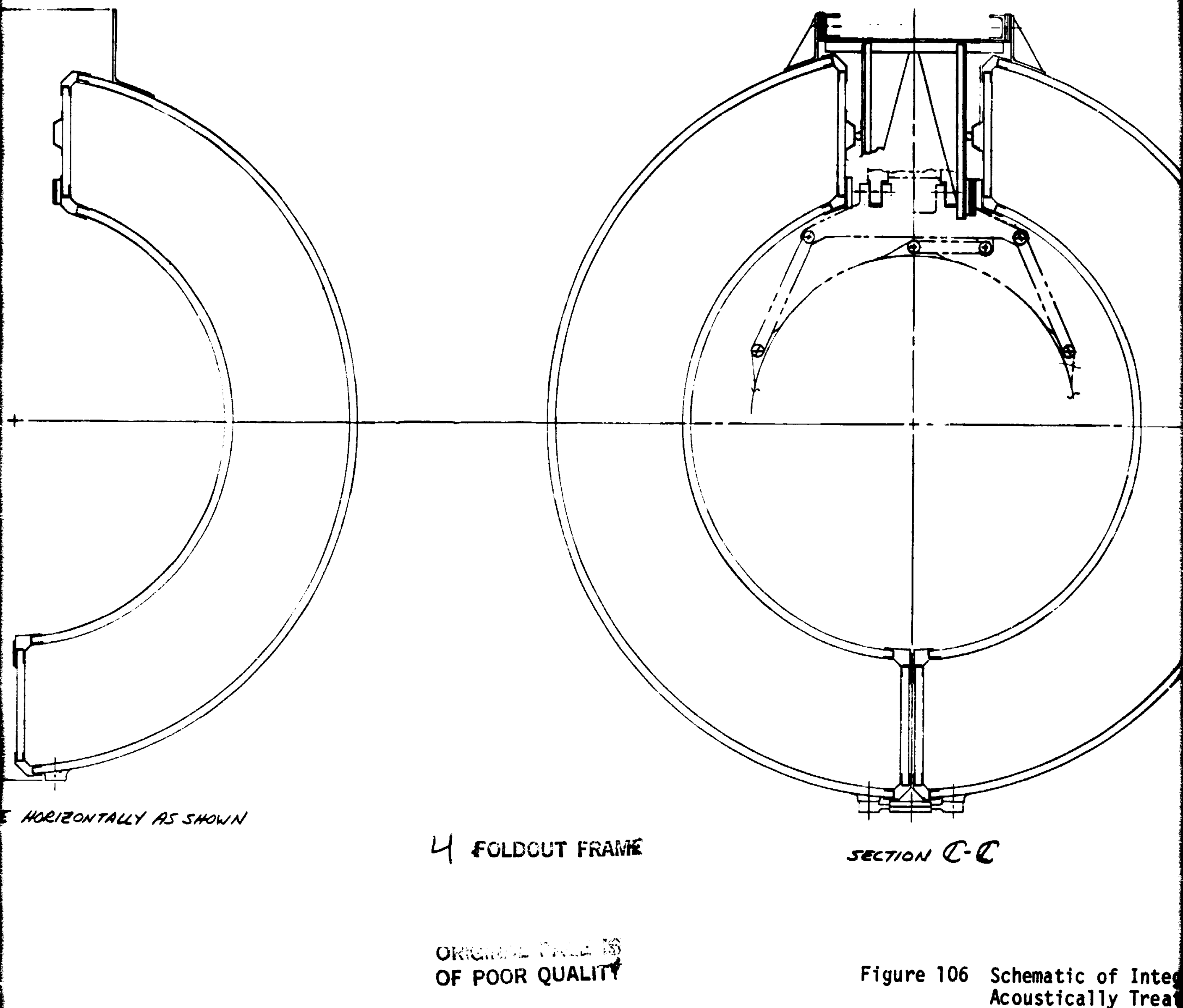

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SECTION C-C

Figure 106 Schematic of Integ
Acoustically Treat
Bellmouth/Inlet Ca

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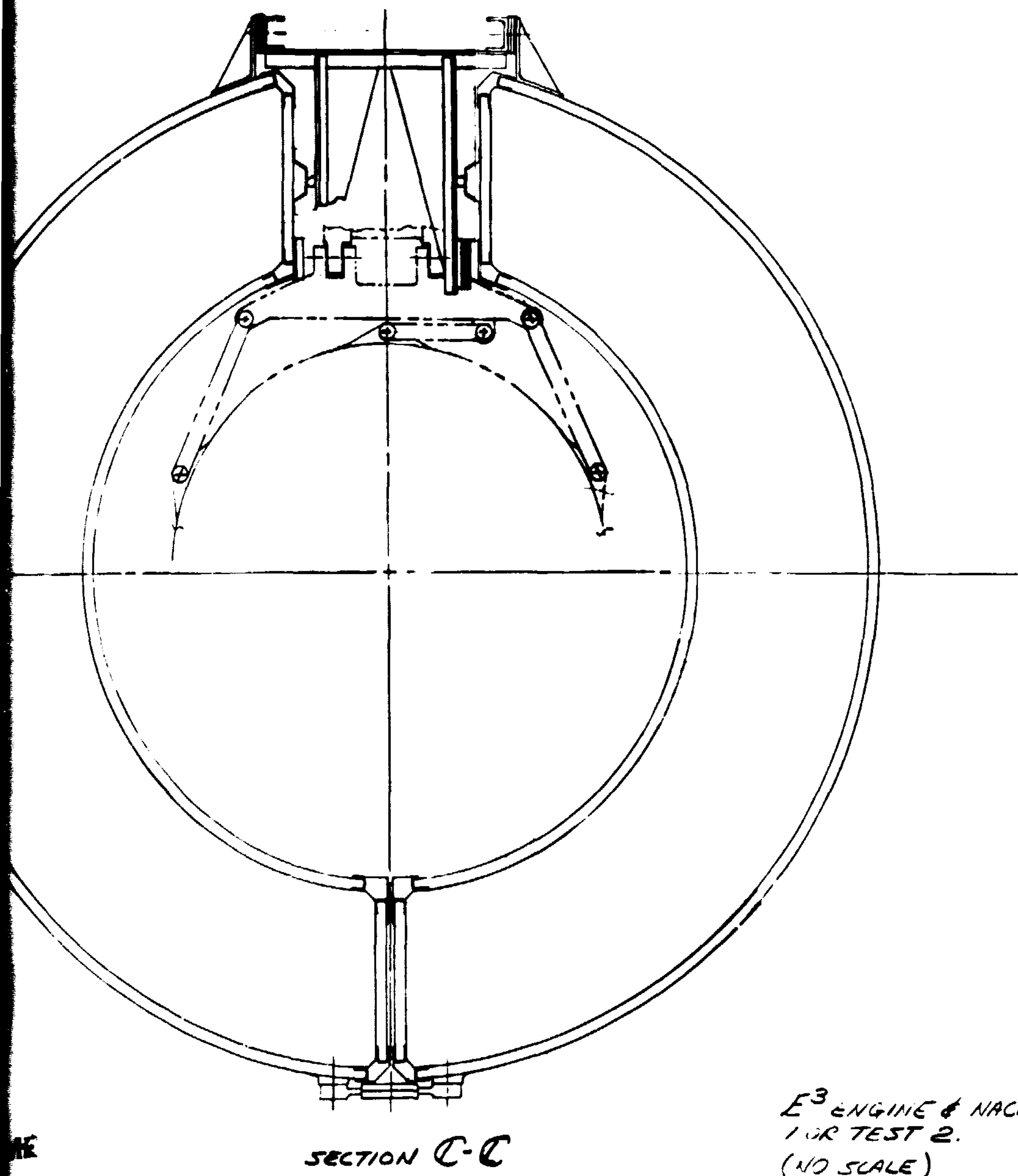

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E^3 ENGINE & NACELLE TEST CONFIGURATION
FOR TEST 2.
(NO SCALE)

Figure 106 Schematic of Integrated Core/Low Spool Full Nacelle Showing Acoustically Treated Sections and One-piece Fiberglass Bellmouth/Inlet Case

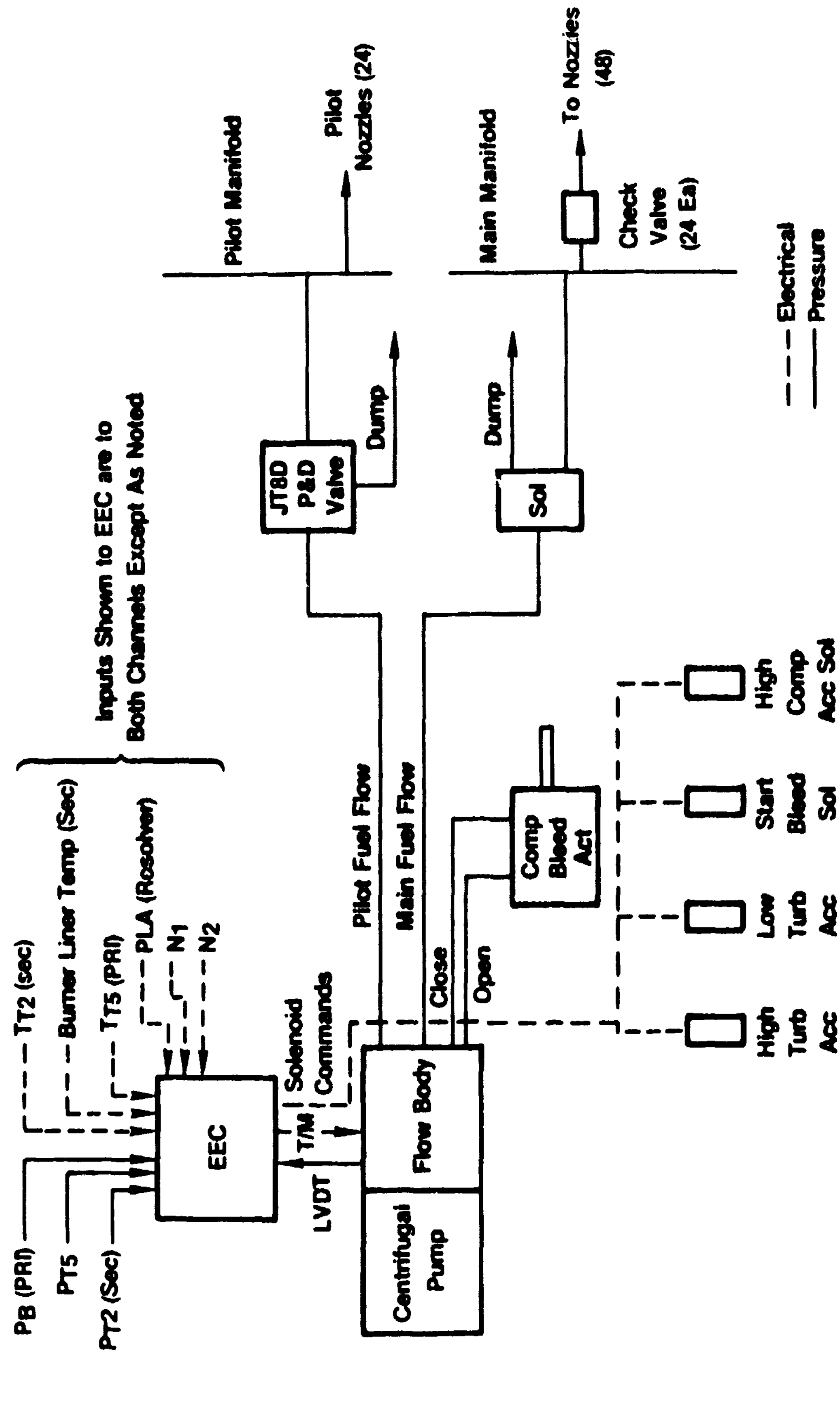

Figure 107 Integrated Core/Low Spool Fuel Control System Schematic

Fan: The titanuim fan blade forgings were prepared and delivered. A purchase order for final machining of these forgings was placed with a vendor. Fan containment case steel forgings were received and operation sheets detailing the procedures required to machine these steel forgings were prepared. Vendor contacts for installation of acoustic treatment were initiated. Raw material procurement continued for the remaining fan component hardware.

Low-Pressure Compressor: The aluminum vane and vane case forgings were delivered. Rotor steel forging and blade titanuim procurement continued. Bearing compartment hardware purchase orders were placed.

Intermediate Case: Based on a cost and fabricability comparison between steel struts machined from solid barstock or built up from sheets and titanium struts fabricated by a diffusion bonding/superplastic forming process, NASA approved the latter construction method for the intermediate case. Fabrication process planning was subsequently initiated as was design and preparation of tooling required for (1) fabrication of the intermediate case and towershaft drive gears, (2) refurbishment of forming dies used in the demonstration effort, and (3) fabrication of inspection gauges.

Quotations were received and evaluated for parts to be vendor fabricated. Included in these parts are: non-structural struts; solid leading and trailing edge inserts for the structural struts; pylon and center bottom bifurcation strut fairings pieces; towershaft components; and various lubrication system items.

Receipt and processing of raw materials continued. All case materials were received. Nominal camber structural strut fabrication was initiated toward the end of the reporting period.

High-Pressure Compressor: The titanium forging for the variable vane case was received from the vendor and fabrication of this case was initiated. The titanuim forging for the compressor bleed case was also received. Vendors were contacted for fabrication cost and lead time estimates.

Blade and vane purchase orders were placed for all airfoils except the exit guide vane (EGV). Available material was transferred to the vendors and the fabrication effort initiated.

Machining of the titanium rotors was initiated on three of the rotor details (R-6, R-7, and R-12) and procurement of raw material for the remaining stages was initiated.

A spare MERL 76 disk compaction from the High-Pressure Compressor Rig Program was transferred to the integrated core/low spool and rough machining of this compaction was initiated.

Inner shrouds for the inlet guide vane and 6th, 7th, and 8th stage vanes were completed and unison rings for the variable vanes received. The raw material procurement effort continued for the remaining high-pressure compressor component hardware.

3.3.3.2.2 Current Technical Progress

Early raw material procurement approval has been received from NASA and fabrication efforts have been initiated for much of the external, fuel system control, facilities and adaptive hardware required by the workplan. Included in these hardware items are rear and front engine mounts, reoperated gearbox and main oil tank, tubes and manifolds, and active clearance control bleed valves and external brackets. In addition, NASA approval was granted to proceed with fabrication of station 2.0 pressure rakes, station 2.5 pressure and temperature rakes, station 4.9 pressure and temperature rakes, and a station 5.0 air angularity probe located aft of the low-pressure turbine exit guide vane.

Integrated Core/Low Spool - Fan

Final machining of the shrouded fan blade forgings was initiated at the vendor during the reporting period and has progressed satisfactorily. Figure 108 shows the titanium blade forging after initial machining of the root. A December 1981 delivery of finished parts is expected.

Vendor machining of the fan containment case steel forging detail was completed. This finished case, shown in Figure 109, was delivered to the Pratt & Whitney Aircraft finish stores area for later use in the assembly effort. Fan blade tip rub strip material segments for use in the fan containment case are currently being manufactured.

Aluminum raw material for the fan component nose cone and cap has been released to the vendor for initial fabrication effort. The machining effort on these two items was initiated during the reporting period.

Fabrication of the fan blade retaining ring, shown in Figure 110, was completed and the ring was delivered to Pratt & Whitney Aircraft. Additional machining of this ring is necessary in order to satisfy existing instrumentation requirements. This machining effort is scheduled to be performed immediately preceding the integrated core/low spool assembly effort.

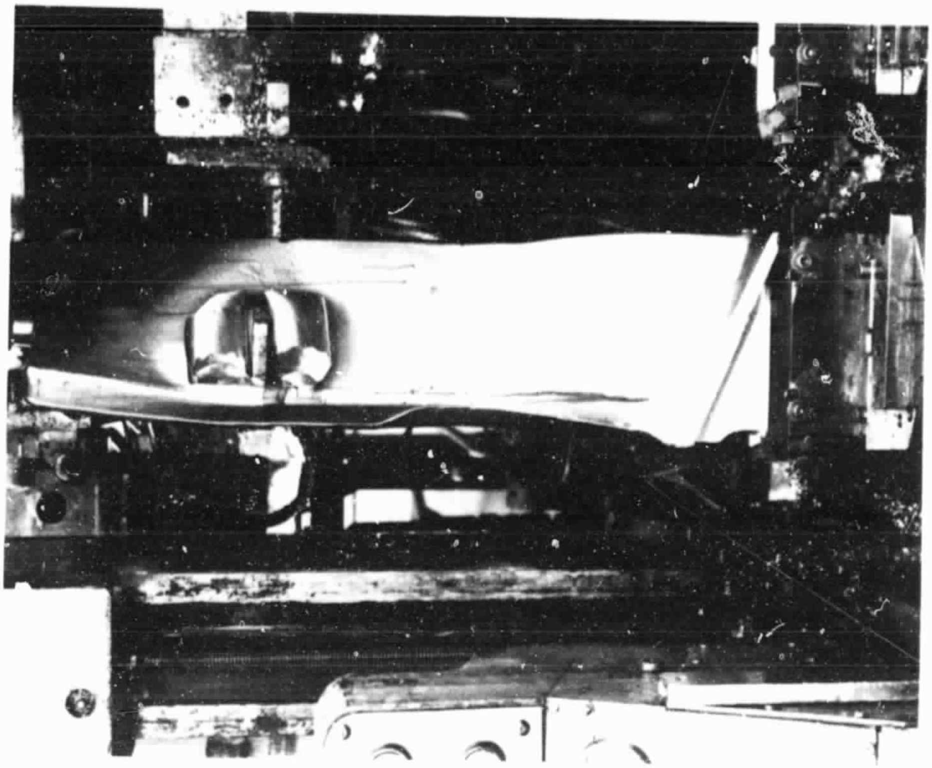

Figure 108 Fan Blade Forging During Initial Shroud Machining

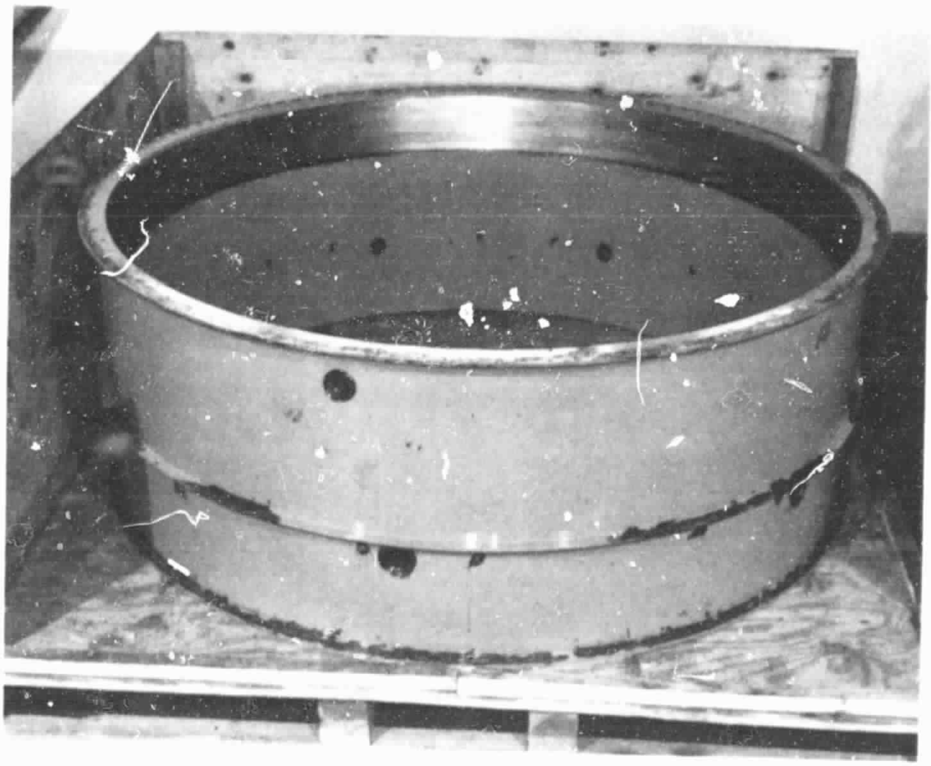

Figure 109 Finished Fan Containment Case Steel Forging Detail

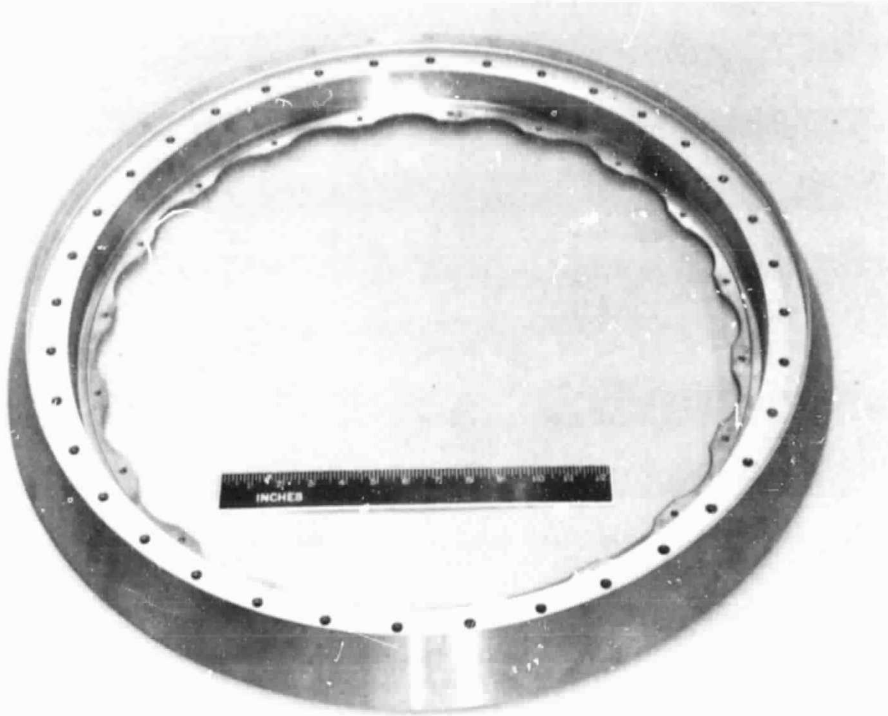

Figure 110 Fan Blade Retaining Ring

Procurement of raw material for the fan hub and fan stubshaft was completed during the current reporting period. The fan hub forging, shown in Figure 111, was forwarded to the vendor and initial machining effort commenced late in the reporting period. The stubshaft forging was also received and forwarded to the machining vendor during the current report period. Machining is scheduled to begin early in the next reporting period.

Integrated Core/Low Spool - Low-Pressure Compressor

Purchase orders for all low-pressure compressor blades and vanes were placed during this reporting period. Raw material has been received and initial fabrication effort directed toward preparation of blade and vane masters is currently in process. Finish parts for all stages of the low-pressure compressor are targeted for a December 1981 delivery. Steel forging material for second, third, and fifth stage low-pressure compressor disks has been received at Pratt & Whitney Aircraft. The fourth stage disk material has also been received and is currently in the machining phase.

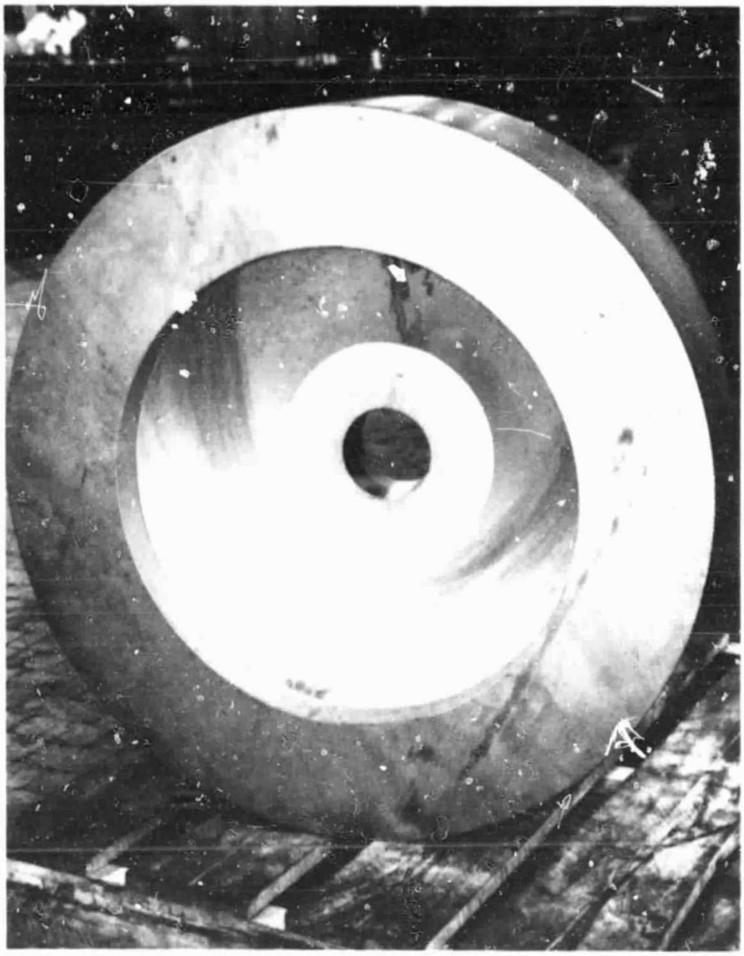

Figure 111 Fan Hub Raw Material Forging

Fabrication of the stubshaft bearing compartment de-oiler has been completed. Figure 112 shows the finished part. The number 1-2 bearing support, the number 1 bearing housing, and number 2 bearing assembly are currently being fabricated.

Fabrication of the low-pressure compressor inlet guide vane inner shroud details has been completed. Fabrication of the compressor vane case assembly was initiated during the report period. Cascade castings of low-pressure compressor bleed discharge turning vanes for the full annular bleed system are in process along with the procurement effort being directed toward bleed system actuation hardware such as bushings, pins and valve brackets.

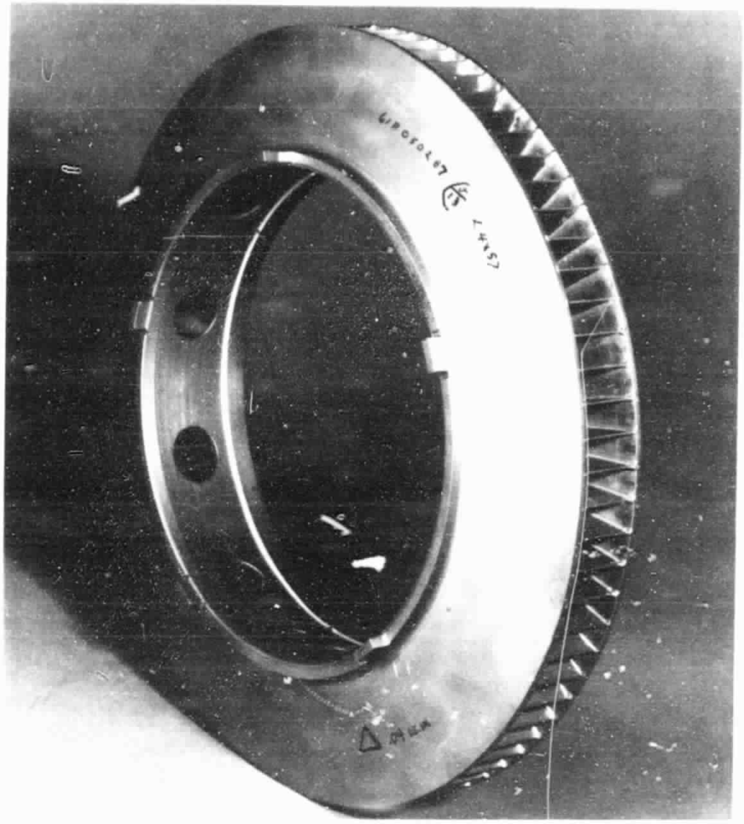

Figure 112 Stubshaft Bearing Compartment De-oiler

Integrated Core/Low Spool - Intermediate Case

Planning effort for the intermediate case was essentially completed by the end of the reporting period despite some earlier delays experienced in shop process planning. The planning effort now indicates that completion of the case assembly will be in July 1982 rather than in March 1982. This date will still allow the integrated core/low spool to meet its first test date, but the structural test for the compressor intermediate case will have to be rescheduled until after the first test.

The tooling design effort continued. Since a new precamber die was required for the nominal structural strut, several die and bonding stop-off pattern modifications were made. Design of the assembly fixture tooling was completed. Preparation and verification of computer tapes for the 15 degree uncamber structural strut were completed toward the end of the reporting period. Preparation of computer tapes for cutback of the structural strut leading and trailing edge was started.

Fabrication of tooling for the intermediate case continued. A new precamber die, with some improvements, was fabricated for the nominal structural strut because of the deteriorated condition of the die made during the technology demonstration program. Fabrication of inspection tooling for the structural struts was completed. Fixturing for welding solid leading and trailing edge stiffeners to the struts was finished. Die fabrication for the 15 degree uncamber strut was initiated. Checkout and correction of tapes to be used in the strut tooling fabrication was completed, and the die patterns and master were made. Dimensional verification of the master's accuracy has been completed. Preparation of forms from which to make the dies has been initiated. Machining of the stop-off material pattern mask for the 15 degree uncamber strut was finished.

Nominal camber structural strut fabrication was continued. Stop-off pattern and precamber die changes to improve forming quality were extensive enough that one strut was taken through the complete fabrication process before committing the remaining struts. This strut was successfully bonded and superplastically formed as shown in Figure 113.

Dimensional checking uncovered some errors in the inspection gauge, and repairs were made. External strut dimensions met the design intent, but internal checking indicated an inadequate bond in the vicinity of the leading edge which will be corrected by relocation of future strut packs in the die. Preparation of titanium sheet was completed for all struts, and bonding of the strut packs was accomplished. One strut pack was formed in the precamber die to verify elimination of previous dimensional deviations by relocation of the pack in the die. Acceptable dimensions resulted, so all bonded packs were successfully formed in the precamber die. A strut was successfully expanded in the finishing die. Dimensional checking indicated strut thickness to be at a minimum design tolerance. Spacing between upper and lower dies was adjusted. Final forming of the rest of the struts was started. At the close of the reporting period, seven of the struts had been final formed.

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 (see Figures 114 and 115), the 'eagle beak' shaped section of the pylon strut, and the pylon strut inner body shown in Figure 116.

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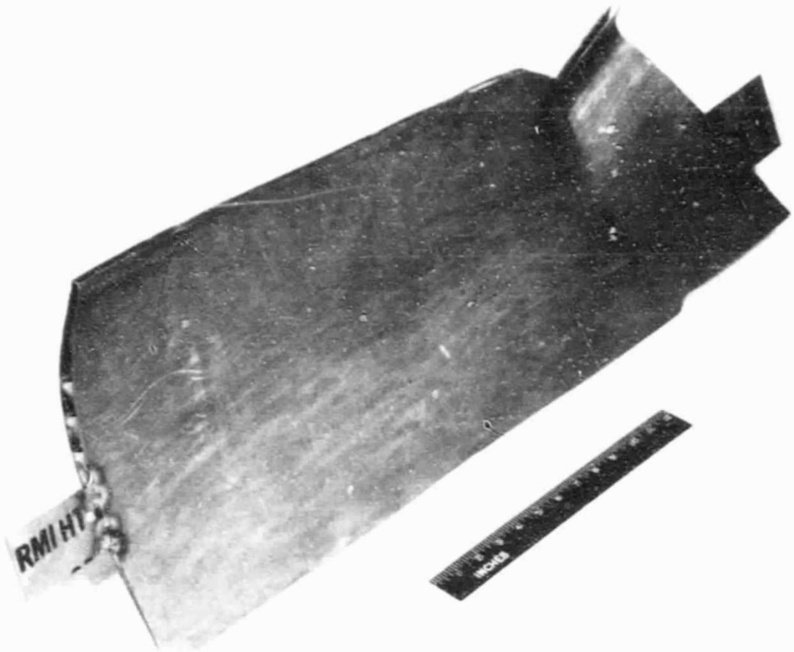

Figure 113 Compressor Intermediate Case Nominal Strut

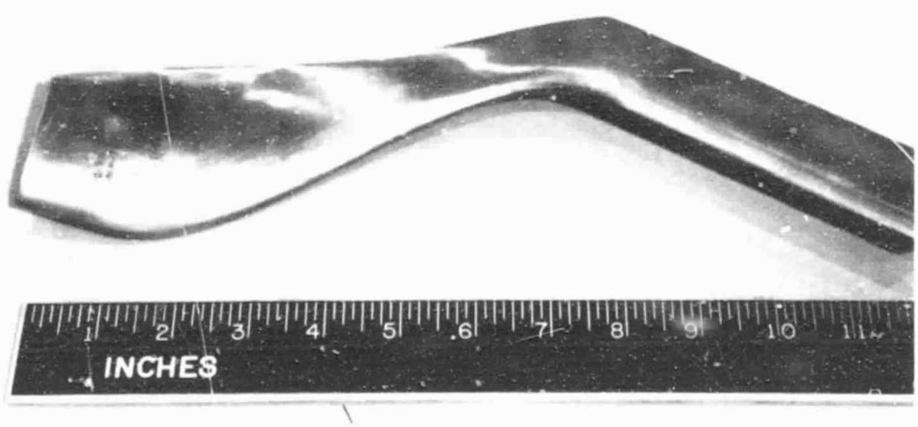

Figure 114 Compressor Intermediate Case Solid Strut Leading Edge

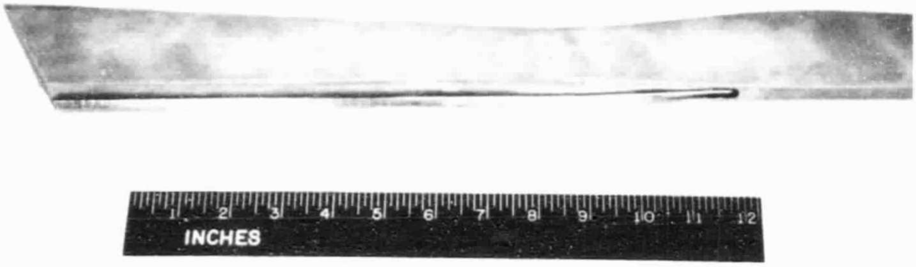

Figure 115 Compressor Intermediate Case Solid Strut Trailing Edge

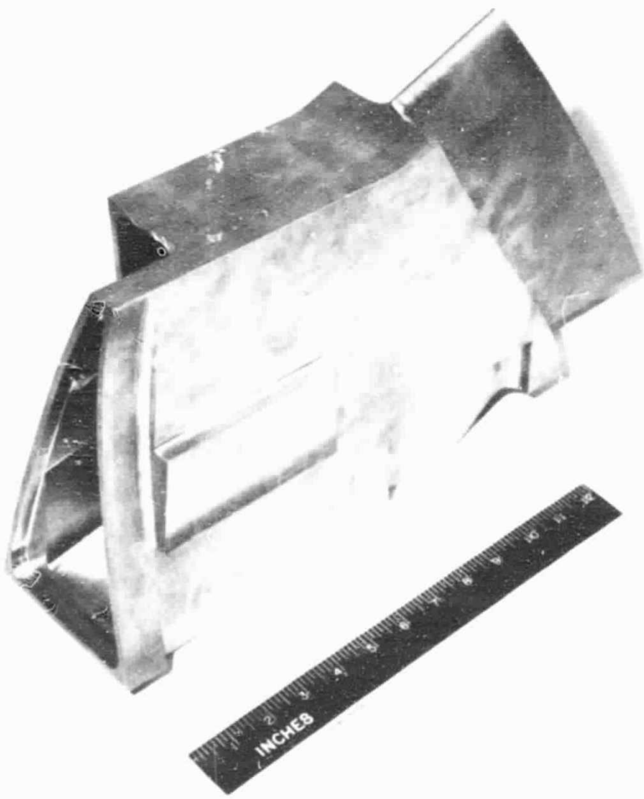

Figure 116 Compressor Intermediate Case Pylon Strut Inner Body

Fabrication of other case parts was initiated. Machining of non-structural struts is underway at a vendor. Machining of the aluminum outer case ring and the inner core flowpath ring, shown in Figure 117, was started at Pratt & Whitney Aircraft. All flanges were completed except for the rear inner fan duct. Rear segments for the outer core flowpath ring were finished.

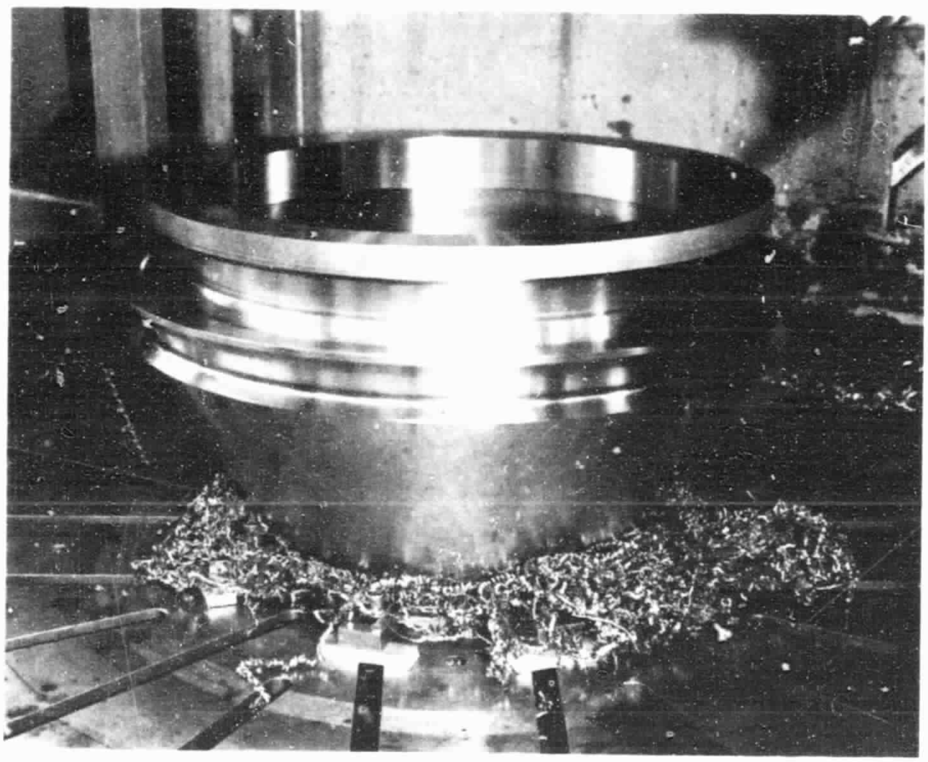

Figure 117 Compressor Intermediate Case Inner Core Ring

Accessory Drive Fabrication: Shop process planning and tooling design 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. Gear tooth cutters were received, and the first tooth form cut was made for the first gear set shown in Figure 118. Dimensional checking has indicated some inaccuracies, and adjustments are currently being made to the machining program and the teeth are being recut.

Inner and outer towershaft fabrication progressed at the vendor, and the center towershaft coupling was received from the vendor. Inspection showed the center shaft to be out-of-print and unrepairable. A new part is scheduled for receipt in early-December 1981.

Completion of the center towershaft bearing is expected in late-October 1981.

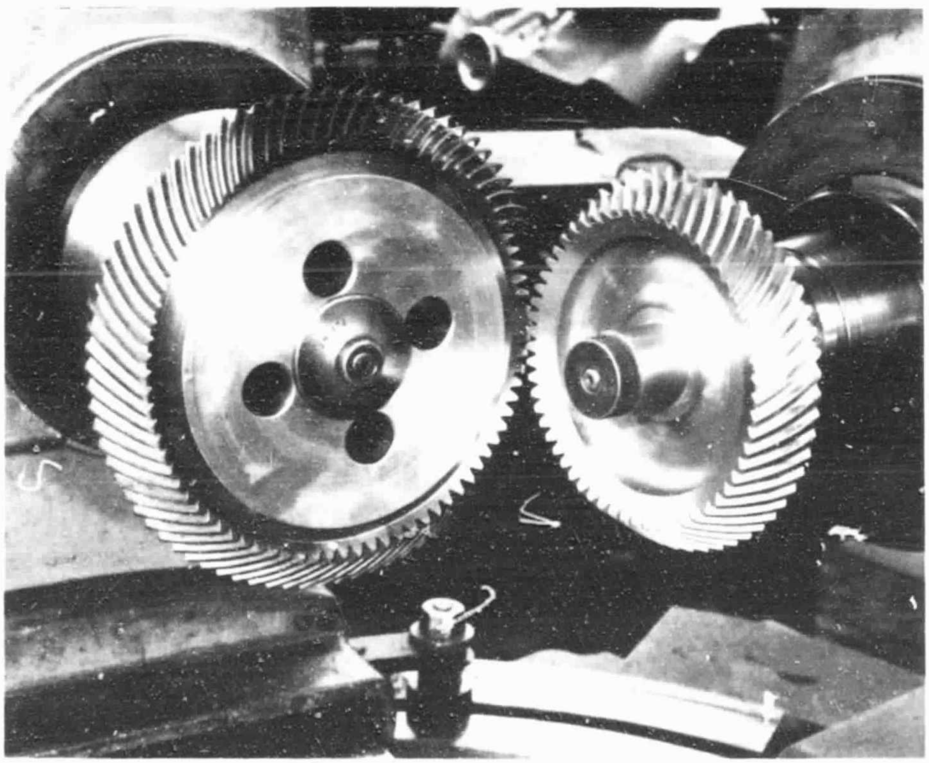

Figure 118 Compressor Intermediate Case Towershaft Bevel Gears

Integrated Core/Low Spool - High-Pressure Compressor

The fabrication effort for the high-pressure compressor inlet guide vane, and the 6th, 11th, and 12th stage vanes was completed. Fabrication of the remaining stage vanes is progressing toward their respective completion dates in the fourth quarter of 1981. After receipt of all finish parts, each piece will be fully inspected via RADOC, a computer aided inspection technique. Results of this inspection will be beneficial in optimizing the aerodynamics of each stage in the high-pressure compressor component.

The fabrication effort for all high-pressure compressor blades continued during the reporting period with delivery of finish parts scheduled for late-December 1981. As in the case of the vanes, these blades will be fully inspected via RADOC. Results of this inspection will also be beneficial in optimizing the aerodynamics of each stage in the high-pressure compressor component.

The high-pressure compressor titanium front split case, shown in Figure 119, is semifinished and ready for flange weldment and finish machining. The bleed case, including details, is currently being fabricated. The fabrication completion date for this case is scheduled for March 1982.

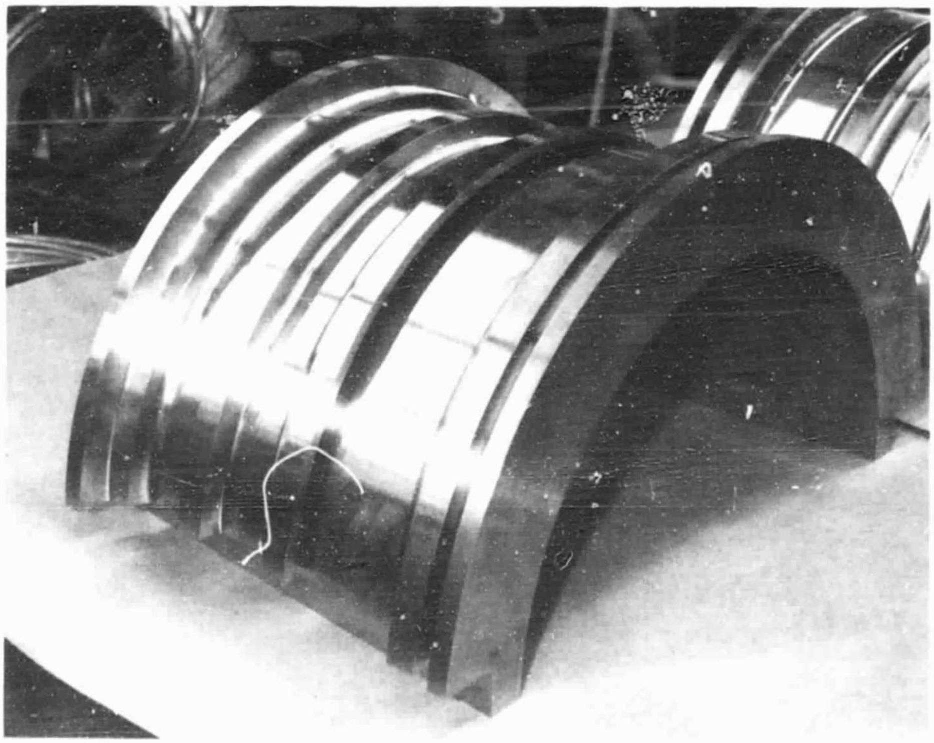

Figure 119 High-pressure Compressor Titanium Front Split Case

The individual disks which comprise the high-pressure compressor drum rotor have been fully machined to the pre-weld configuration. Instrumentation provisions remain to be installed prior to the welding activity. Some of the rotor details are shown in Figures 120 through 123. 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 treat. Final machining will be initiated in the fourth quarter of 1981.

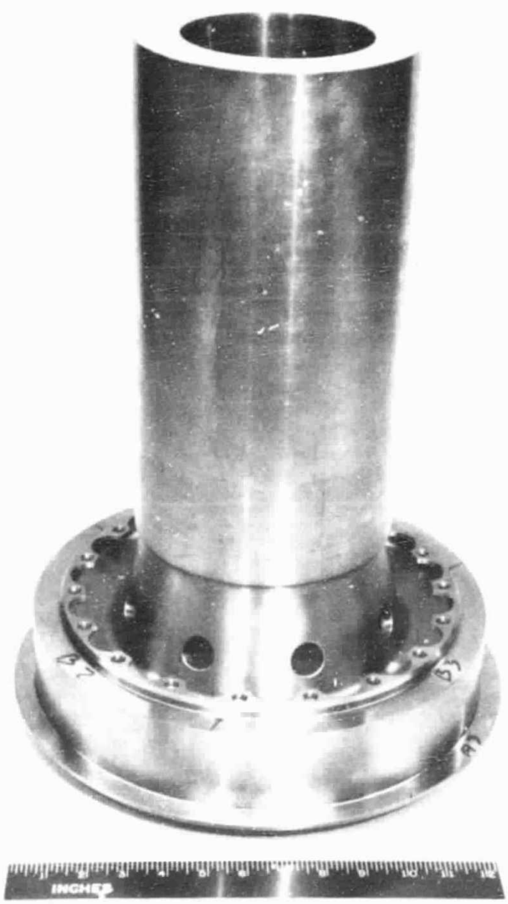

Figure 120 High-Pressure Compressor Sixth Stage Rotor Detail

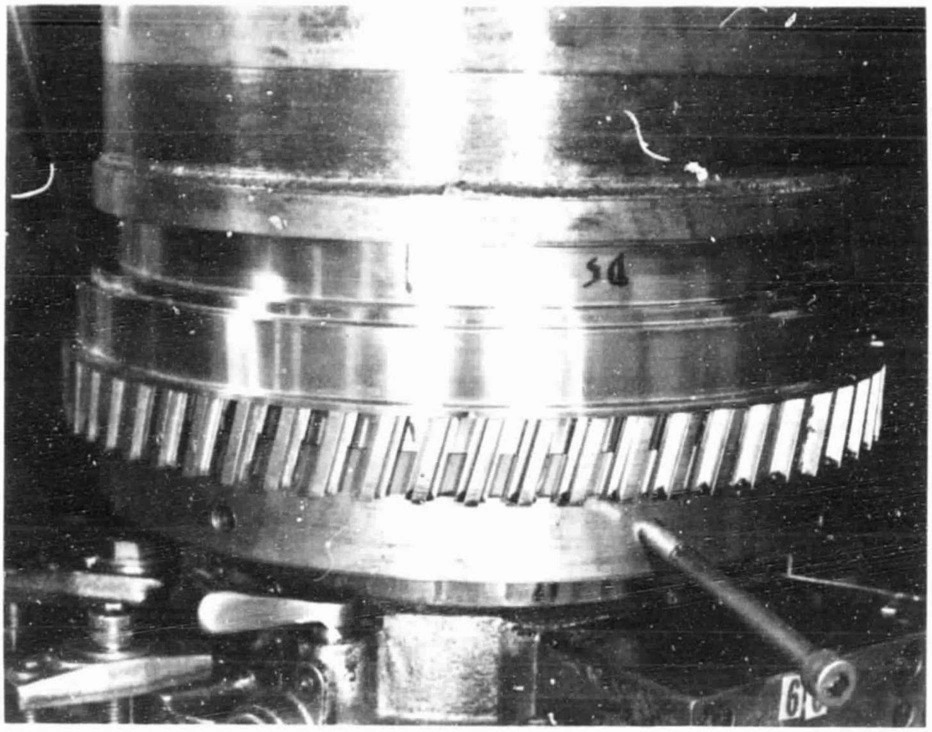

Figure 121 High-Pressure Compressor Eighth Stage Rotor Detail

Figure 122 High-Pressure Compressor Ninth Stage Rotor Detail

Figure 123 High-Pressure Compressor Twelveth Stage Rotor Detail

Integrated Core/Low Spool - Combustor

The spare diffuser case was successfully cast and delivered to Pratt & Whitney Aircraft during the reporting period. Raw material was received for the diffuser case front and rear skirt, and the prediffuser duct inner wall detail. This raw material is planned as 'back-up' for the component diffuser case assembly which will be available for the integrated core/low spool upon completion of the combustor component development program. Raw material was also received for the inner combustor case and the fabrication effort initiated.

Integrated Core/Low Spool - High-Pressure Turbine

Vanes: In order to maintain the assembly and test schedule established for the first build integrated core/low spool, a decision was made to cast a set of high-pressure turbine vanes using PWA 1422 directionally-solidified alloy instead of PWA 1480 single crystal alloy for integrated core/low spool use. A set of PWA 1422 vanes was previously successfully cast for use in the high-pressure turbine component rig. Effort directed toward successfully casting PWA 1480 vanes will continue with the intention of providing a minimum of three vanes for the initial integrated core/low spool build and a full set of vanes for the second integrated core/low spool build.

Predicted integrated core/low spool life for the PWA 1422 vanes is 60 hours hot time (i.e., sea level takeoff - hot day) compared to 100 hours hot time for the PWA 1480 vanes. This is adequate life to meet the planned integrated core/low spool test program requirements.

Blades: Casting work on the integrated core/low spool blade set is progressing satisfactorily toward a December 1981 completion date. These blades will incorporate five additional cooling air pedestals in the trailing edge core cavity. Addition of these pedestals was based on results from blade water flow cooling model tests.

Disk: Work on the integrated core/low spool disk progressed through hot isostatic pressing, rough lathe turning, heat treat, surface cleaning and is currently undergoing ultrasonic inspection.

Integrated Core/Low Spool - Low-Pressure Turbine

The only part presently included in this task effort is the fabrication of the number 5 bearing which is progressing on schedule for a December 1981 delivery.

3.3.3.3 Integrated Core/Low Spool - Assembly and Inspection

3.3.3.3.1 Current Technical Progress

The assembly floor planning, and tool design and fabrication efforts associated with the first build integrated core/low spool assembly were initiated during the reporting period. Assembly of the first build of the integrated core/low spool is scheduled to begin in July 1982.