April 30, 1981

R81AEG316

ENERGY EFFICIENT ENGINE

COMPONENT DEVELOPMENT & INTEGRATION

NAS3-20643

SEMIANNUAL REPORT No. 6

OCTOBER 1, 1980 - MARCH 31, 1981

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LEWIS RESEARCH CENTER
21000 BROOKPARK ROAD
CLEVELAND, OHIO 44135

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GENERAL ELECTRIC

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CINCINNATI, OHIO 45215

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General Electric Project Manager
T.L. Hampton

Prepared for

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FOREWORD

This contract effort is being conducted by NASA as part of the Energy Efficient Engine project. It is managed by the NASA-Lewis Research Center, with Carl C. Ciepluch serving as the NASA Project Manager.

This semiannual report covers the work performed under Contract NAS3-20643 for the period of October 1, 1980 through March 31, 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 U.S. in advance of general publication in order to accelerate domestic technology transfer. Since all data reported herein are preliminary information, they should not be published by the recipients prior to general publication of the data by either the Contractor or NASA.

Selected portions of the data (that is, those 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 2 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 Category 2 data may be duplicated and used by the recipient with the expressed limitation that the data will not be published or released to foreign parties during this period without the expressed permission of the General Electric Company and appropriate export licenses. Release of these Category 2 data to other domestic parties shall only be made subject to the limitations that all recipients must agree, prior to receiving these data, to abide by the limitations of the FEDD legend on the cover of this report.

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TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Concluded)

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LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Concluded)

LIST OF TABLES

LIST OF TABLES (Concluded)

INTRODUCTION

This is the sixth semiannual report under Contract NAS3-20643 - NASA Energy Efficient Engine (E^3) Component Development and Integration Program. The report covers the period October 1, 1980 through March 31, 1981.

The program objective is the development of technology that will improve the energy efficiency of propulsion systems for subsonic commercial aircraft of the late 1980's or early 1990's. The following goals have been established.

- Fuel Consumption

  A reduction in Flight Propulsion System (FPS) cruise installed sfc of at least 12% compared to the reference CF6-50C engine. (Cruise is defined as Mach 0.8 and 35,000 feet on a standard day at maximum-cruise power without bleed or power extraction.)

- Direct Operation Cost (DOC)

  A DOC reduction of at least 5% based on advanced aircraft with E^3 compared to scaled CF6-50C.

- Noise

  FAR-Part 36 (as amended July 1978) with provision for engine growth corresponding to future engine application.

- Emissions

  EPA new engine standards January 1981.

CO	(lb per 1000 lb thrust-hr per cycle)	3.0
HC	(lb per 1000 lb thrust-hr per cycle)	0.5
NO_x	(lb per 1000 lb thrust-hr per cycle)	3.0
SMOKE	(SAE-SN)	20.0

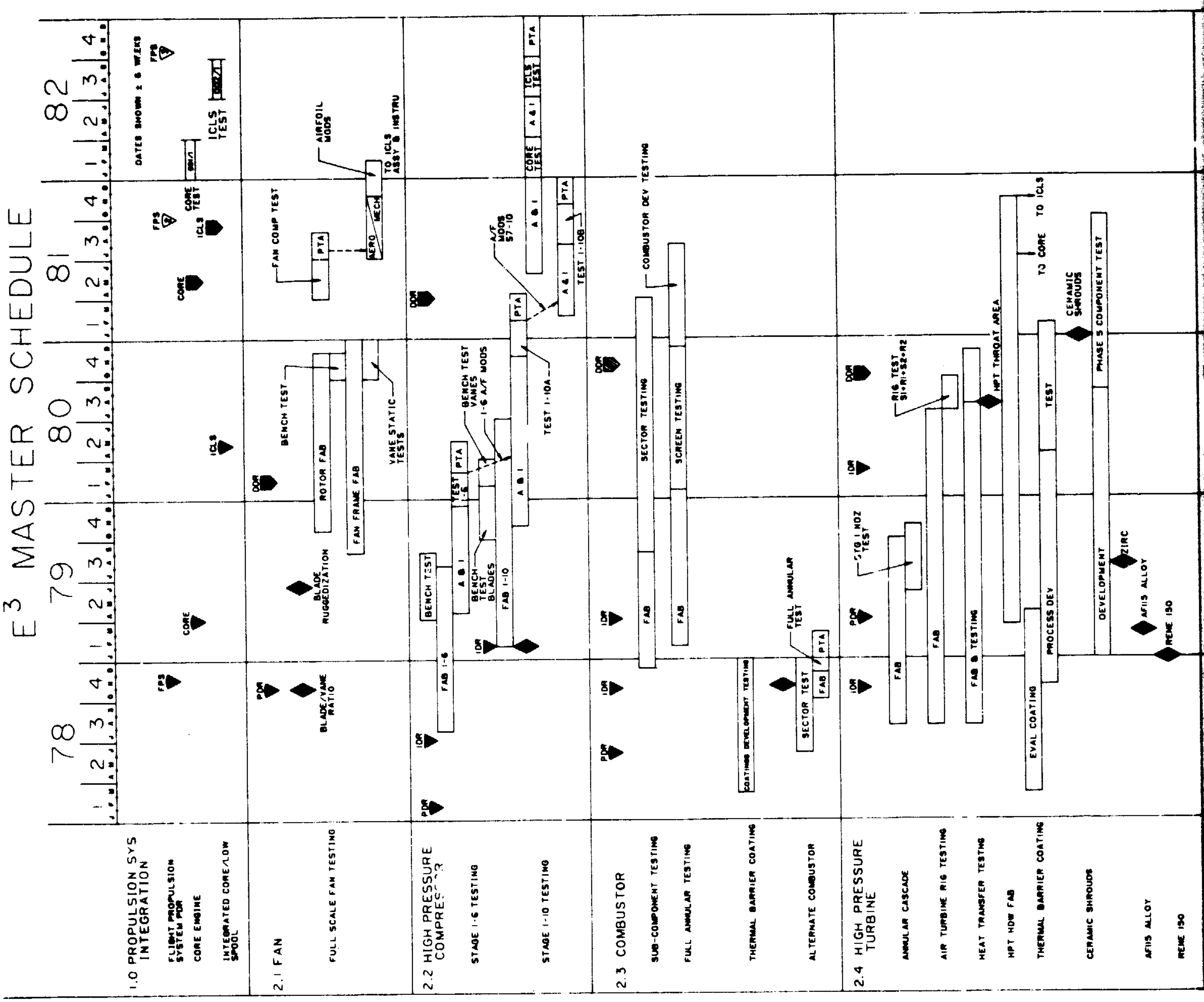
General Electric is projecting a reduction of 14.4% in cruise installed sfc (0.572 versus 0.668 for the CF6-50C) and a DOC reduction in excess of the 5% goal. Noise and emissions projections are consistent with the established goals.

In addition to the foregoing, growth of the FPS is being considered from the outset. A minimum installed-thrust level of 36,500 lbf has been established for the FPS at takeoff. A planned growth of 20% maximum-climb thrust over the baseline rating is desired without compromise of the foregoing goals. Components of the engine are to be designed with consideration for growth and the competitiveness of the initial engine.

Commercial transport engine requirements will be factored into the development effort, as appropriate, with the objectives of (1) making the resulting technology useful for subsequent commercial application and (2) normalizing the risk of any future commercial developments. Components of the engine are to be designed with consideration for growth and the competitiveness of the initial engine.

Four major technical tasks have been established for the E^3 program. Task 1 addresses the design and evaluation of the E^3 Flight Propulsion System; this propulsion system is designed to meet the requirements for commercial service and includes a flight nacelle. The Task 1 results will establish the requirements for the experimental test hardware including the components, core, and integrated core/low spool. Task 2 consists of the design, fabrication, and testing of the components and includes supporting technology efforts. These supporting technology efforts are to be performed where required to provide verification of advanced concepts included in the propulsion system design. In addition, more advanced technologies that are not specifically included in the propulsion system design but which provide the potential for further performance improvements are also to be explored. Task 3 involves the design, fabrication, and test evaluation of a core engine consisting of the compressor, combustor, and high pressure turbine. Integration of the core with the low-spool components and test evaluation of the integrated core/low spool comprise Task 4. At the conclusion of the program, the latest performance of the experimental hardware (integrated core/low spool and concurrent core and component efforts) will be factored into a final propulsion system/aircraft evaluation (as part of continual, ongoing evaluations in Task 1) to determine achievable performance as compared to program goals. Task 5 is a nontechnical task that encompasses the Project/Program Management and Control System established for the contract. The Master Program/Project Schedule is shown in Figure 1.

This report provides a review of the work accomplished during the first 39 months of the E^3 contract, with emphasis on accomplishments during the



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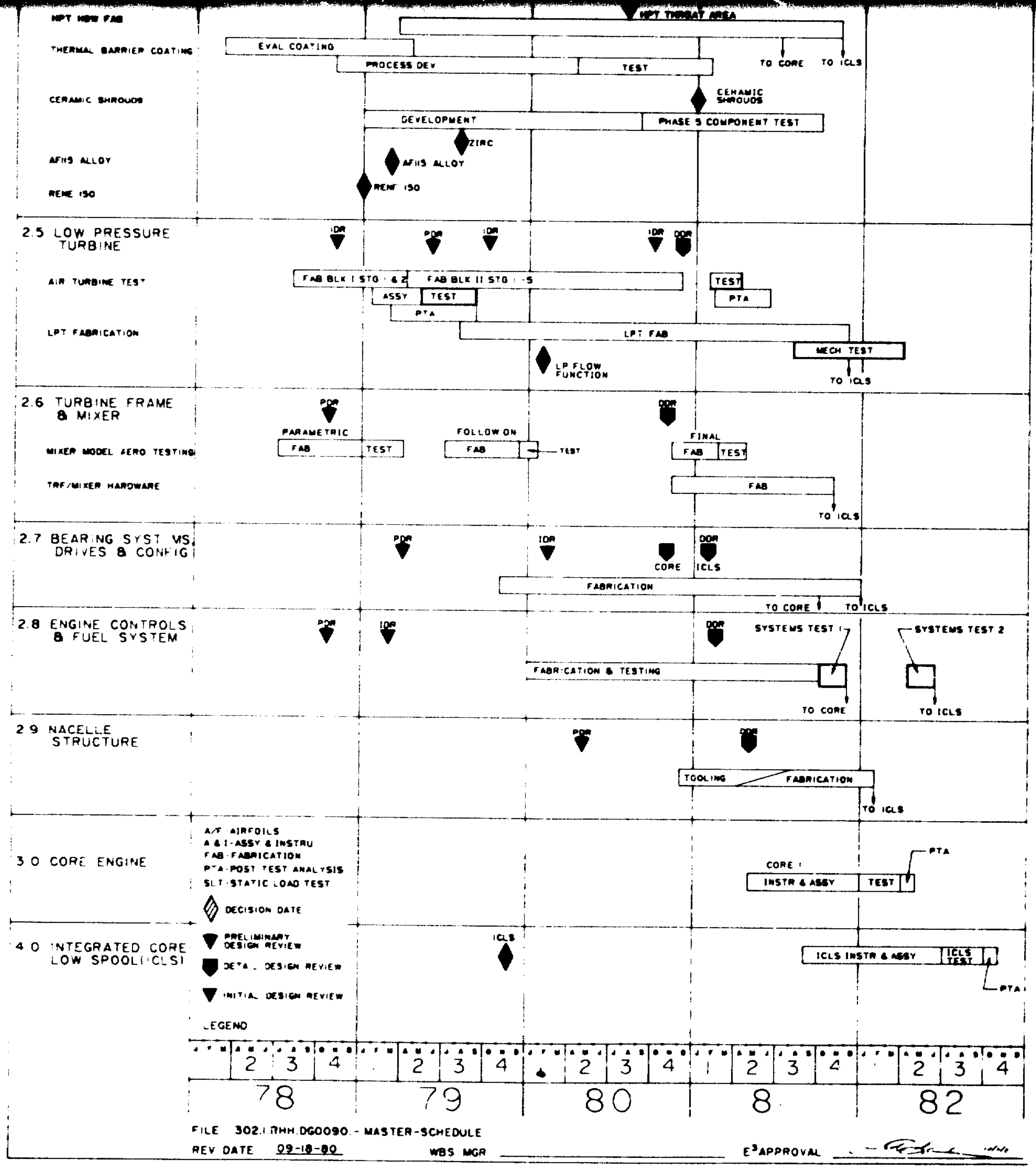

Figure 1. Master Program Schedule.

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past 6 months. It is a progress report, and the data contained herein are subject to change during the remaining period of performance. The report is responsive to and consistent with the requirements of the Statement of Work.

Summary of Progress

The first 60 days under contract were devoted to planning the entire 5-year program. This period culminated in the Work Plan, the basis of which is the Work Breakdown Structures (WBS) established in accordance with the elements contained in the contract Statement of Work. The WBS is presented in Table I in an abbreviated form. Program and design highlights, progress, and overall perspective are reported herein for the systems and components. All WBS items in which work was performed during this report period are documented. Some of the highlights during this report period are

- Sector combustor configurations have been tested which demonstrated successful ground start ignition below the 45% core speed requirement

- Full annular combustor testing of the third development combustor configuration for ignition performance was completed; further development testing is planned

- All LP turbine ICLS hardware has been released for manufacture

- Substantial progress has been made in the manufacture of Core engine HPT hardware

- The 10-stage compressor vehicle was installed in the test facility and testing is in progress

- All fan test vehicle hardware has been received; instrumentation and assembly is approximately 50% complete

- Detail design and drawing releases for hardware fabrication for the Core and ICLS engines is complete; procurement of limiting hardware is being expedited

- Mechanical design and analysis of the turbine frame and mixer has been completed

- Configuration design has positioned all major components and piping requirements have been established

Table I. Work Breakdown Structure.

1.0	Task 1 - Propulsion System Analysis, Design, and Integration
1.1	Propulsion System
1.2	Cycle and Performance
1.3	Materials and Processes
1.4	Acoustic Development
1.5	Propulsion System Aircraft Integration and Development
2.0	Task 2 - Component Analysis, Design, and Development
2.1	Fan
2.2	Compressor
2.3	Combustor
2.4	High Pressure Turbine
2.5	Low Pressure Turbine
2.6	Turbine Frame and Mixer
2.7	Bearings, Systems, Drives, and Configuration
2.8	Engine Controls and Fuel System
2.9	Nacelle Structure
3.0	Task 3 - Core Testing
3.1	Initial Core Build
3.2	Second Core Build
3.3	Core Posttest Analysis
4.0	Task 4 - Integrated Core/Low Spool (ICLS) Testing
4.1	ICLS Engineering and Analysis
4.2	ICLS Fabrication
4.3	ICLS Assembly and Instrumentation
4.4	ICLS Test Facilities Engineering
4.5	ICLS Testing
4.6	ICLS Posttest Analysis
5.0	Task 5 - Management
5.1	Project Management
5.2	Configuration Management
5.3	Reporting

- Defined a new simplified engine model based on engine cycle data curve fits for use in the digital control test stand.

A continuing problem has been experienced with regard to long lead times for procurement of hardware, with subsequent impact on program schedules. Schedule adjustments were made during this reporting period to accommodate hardware slips and are discussed individually in the applicable WBS sections.

1.0 TASK 1 - PROPULSION SYSTEM ANALYSIS, DESIGN AND INTEGRATION

Overall Objectives

The primary objectives of this task are to provide the preliminary design of the flight propulsion system (FPS) and to evaluate the progress of the FPS to ensure that NASA program goals are being met. In addition, periodic updates and reassessments of the FPS will be conducted to ensure that FPS projections contain the most current information available from the component technology development program. Periodic preliminary design reviews (PDR) based on material and data developed under this task will be conducted to provide program status information.

As part of this task, cycle decks and performance projections for the FPS will be developed and kept current. All system integration efforts are conducted through this task, such as FPS layouts, overall assembly drawings, parts lists, and design change monitoring. All aspects of the system, including acoustic evaluation, aircraft integration, engine dynamics, reliability, life management, and fan and compressor compatibility, are coordinated and results incorporated under Task 1. Other results are the evaluation of the system benefits of the FPS and a determination as to whether direct operating cost and fuel savings goals are being met. Another objective of this task is to conduct and carry on a supporting material technology program and to provide the reviews of the material selected for use in the program.

Task 1 also has the objective of providing preliminary Core Engine and Integrated Core/Low Spool (ICLS) designs and appropriate efforts to integrate the designs and to ensure that hardware support is provided, as appropriate, for Core/ICLS testing. In addition, Core and ICLS PDR and Detail Design Reviews (DDR) will be conducted based on the data generated during the task.

Development Approach

Evaluation and updates of the FPS will be accomplished by blending information that becomes available from the preliminary design phases of the FPS, Core, and ICLS with information that becomes available from Tasks 2, 3, and 4 as the program proceeds.

Initially, layouts and system characteristics were developed from the proposal FPS and any subsequent modifications that occurred before and after the first FPS PDR. Information from the supporting efforts under the Propulsion System Design effort will be combined with that coming out of the other Task 1 efforts, such as aircraft integration and evaluation, and acoustic studies to update the FPS status.

As changes and new estimates of component and system performance become available, new layouts and system evaluation updates will be generated for the appropriate FPS PDR.

Continual monitoring of the Core and ICLS design and hardware, along with implementation of the system integration responsibility, will ensure that accurate knowledge of the FPS configurations and characteristics is always available. It also ensures that changes to the Core and ICLS that could adversely affect the performance of the FPS will be known and that action will be taken, when necessary, to prevent FPS degradation.

As the program proceeds, all important aspects of the Core and ICLS performance and characteristics will be translated into meaningful information for FPS PDR's. Final program results and a final FPS projection will be obtained and communicated in this manner.

Flight Propulsion System Description

General Electric's proposed E^3 Flight Propulsion System in a high bypass, dual-rotor, axial-flow turbofan with a fan pressure ratio of 1.65 and an overall pressure ratio of 38 at the maximum climb power matching point. At maximum cruise, the bypass ratio is approximately 7. For the sea level takeoff (SLTO) maximum thrust rating of 36,500 pounds, the combustion temperature for the FPS is projected to be 2450° F. A symmetrical nacelle with a long-duct, mixed-flow nozzle completes the installation. A cross section of the proposed installed engine is shown in Figure 1.1-1.

The major engine components are the fan rotor and stator module; the core engine, consisting of the compressor, combustor, and two-stage turbine; the low pressure turbine module, mounting provisions, and exhaust mixer; and the core-mounted accessories.

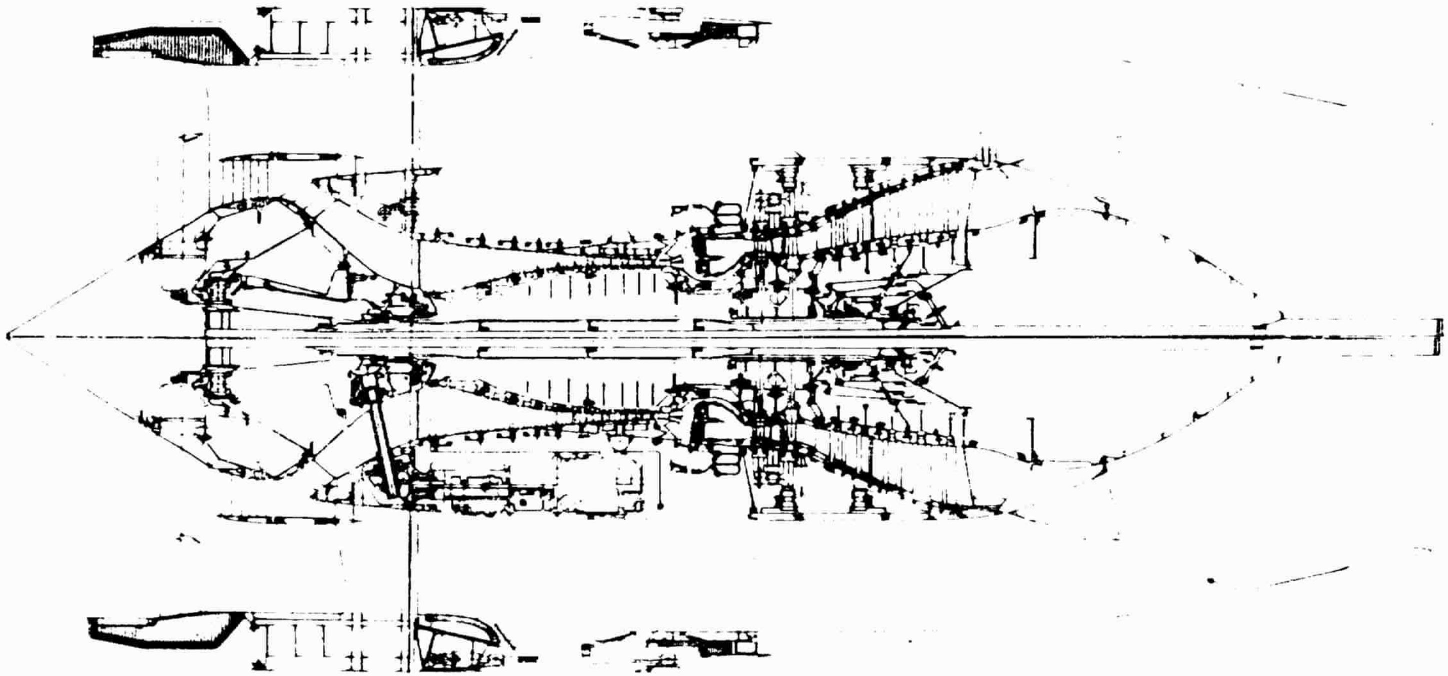

Figure 1.1-1. E^3 Propulsion System.

The fan rotor, consisting of a 32-blade fan stage and a 56-blade quarter stage for core compressor supercharging, is driven by a five-stage turbine. Titanium was utilized for blading to provide foreign object damage (FOD) ruggedness. The quarter stage provides approximately 1.7 times as much air as the core requires. The excess air is bypassed into the fan exhaust duct, providing automatic flow matching and FOD separation. The fan turbine uses tip shrouds and blade/vane root overlap to reduce leakage and aerodynamic losses. A circumferentially continuous fan turbine casing (coupled with modulated casing cooling) provides active clearance control. Increased blade numbers are used in the next-to-last fan trubine stage to raise puretone frequencies and reduce perceived noise.

The fan frame is integrated with the fan outer duct and nacelle to provide a stiff, lightweight structure. The outer frame and duct are constructed of graphite-epoxy materials; the inner frame is aluminum. Containment for the fan blades is provided by a hybrid system of steel backed up by wrapped Kevlar.

The core compressor has 10 stages of rugged, low-aspect-ratio blading to reduce stress levels and erosion, and it provides a compression ratio of 23 at the maximum-climb matching point. The last five stages of compressor have active clearance control and a separate aft casing support to improve running clearances. There are five stages of variable-geometry vanes to improve matching and efficiency at off-design conditions, and seventh-stage bleed is available for starting.

The combustor is a double-annular design for low emissions and is patterned after the combustor developed under a NASA low-emissions combustor development program based on the CF6 engine. A shingle liner configuration is employed; the hot shingle liner is nonstructural (except for cooling and pressure loads) while the cool outer liner is used for all combustor support and positioning functions. The outer combustor nozzles are used for low-thrust conditions; as thrust is increased, the inner set of main nozzles begins to function.

A two-stage turbine completes the core engine configuration. Advanced directionally solidified René 150 is used for the Stage 1 and 2 blading and the Stage 2 vanes. The Stage 1 vanes are constructed of an oxide-dispersion-solidified (ODS) material that has provided good service in other high-temperature nozzle applications. Both turbine disks are boltless designs with smooth side plates for low windage losses. The cooling-air circuits have been configured to provide the coolest air possible with no excess pressure losses or leakage for increased thermodynamic efficiency. The cooling circuit for the Stage 1 blades also provides a cooling layer over the compressor and turbine interconnecting shaft. Active clearance control over the turbine is achieved with an engine control function that selectively cools the support rings of the shrouds, permitting blade running clearance to be changed thermally under transient and steady-state conditions to permit minimum blade-to-shroud average clearances.

A double-wall design is used in the aft hot sump to ensure that the inner sump is adequately cooled. Two separate sources of cooling air purge the outer and inner walls of the aft sump to prevent any inadvertent overheating. A center vent system exhausts the sump pressurization air back through the exhaust cone to the engine nozzle region.

The engine accessory gearbox is located in the engine core compartment. This location allows the nacelle diameter to be smaller for lower nacelle drag. The basic engine design will accommodate an alternate accessory location on the fan case if that arrangement is ultimately preferred by some users.

The engine nacelle is a symmetrical, long-duct, mixed-flow design that makes extensive use of lightweight composites to reduce cost and weight. Sound suppression is integrated into the inner walls of the inlet, fan frame, fan duct, core cowl, and nozzle. The thrust reverser is a directed-cascade type with no links crossing through the fan duct. When deployed, cooling air slots are opened to allow cool ambient air to enter the region aft of the blocker doors to prevent the composite material from overheating due to hot recirculating core exhaust gas. No thrust reverser is used for the core stream since the core thrust is effectively spoiled by the overexpansion of the mixer exhaust into the fan nozzle when the fan air is blocked off.

Mounting provisions permit the point mount loads to be transferred into the engine in a smooth, attenuated manner. Thrust side and vertical loads are transferred into the forward engine frame; roll side and vertical loads are taken out by the aft mount attached to the LP turbine frame.

1.1 PROPULSION SYSTEM DESIGN

1.1.1 System Integration

Technical Progress

Analysis of transient tip clearances determined a potential problem in the high pressure turbine. Rubs could occur if the engine were accelerated to power immediately after start-up. To avoid this, a heating circuit was incorporated into the active clearance control system for the HP turbine. This circuit ducts CDP bleed air to the impingement manifolds around the turbine case and will be used at low power settings.

Two design changes discussed in the previous semiannual report were incorporated into an updated FPS cross section. These were the addition of a flaired mixer/low pressure turbine and removal of the fluid film damper from the aft support bearing. These changes are incorporated in Figure 1.1-1.

The FPS materials are shown in Figure 1.1-2 and operating parameters and cooling flows are shown in Figure 1.1-3.

The FPS parts list is comprised of the drawing status report which has been regularly updated.

The key dimension drawing was updated three times. The changes concerned refinements of the position of the forward bearing relative to the fan blade and the position of the interface between the forward shaft and the LPT shaft.

The E^3 Flight Propulsion System was utilized by two separately funded NASA programs. These were the Materials for Advanced Turbine Engines (MATE) Cost/Benefit Study and the Structures Performance, Benefit, Cost Study. Also, existing E^3 aircraft integration material was provided to Lockheed-California Company for their Integrated Technology Wing Design program with NASA-Langley.

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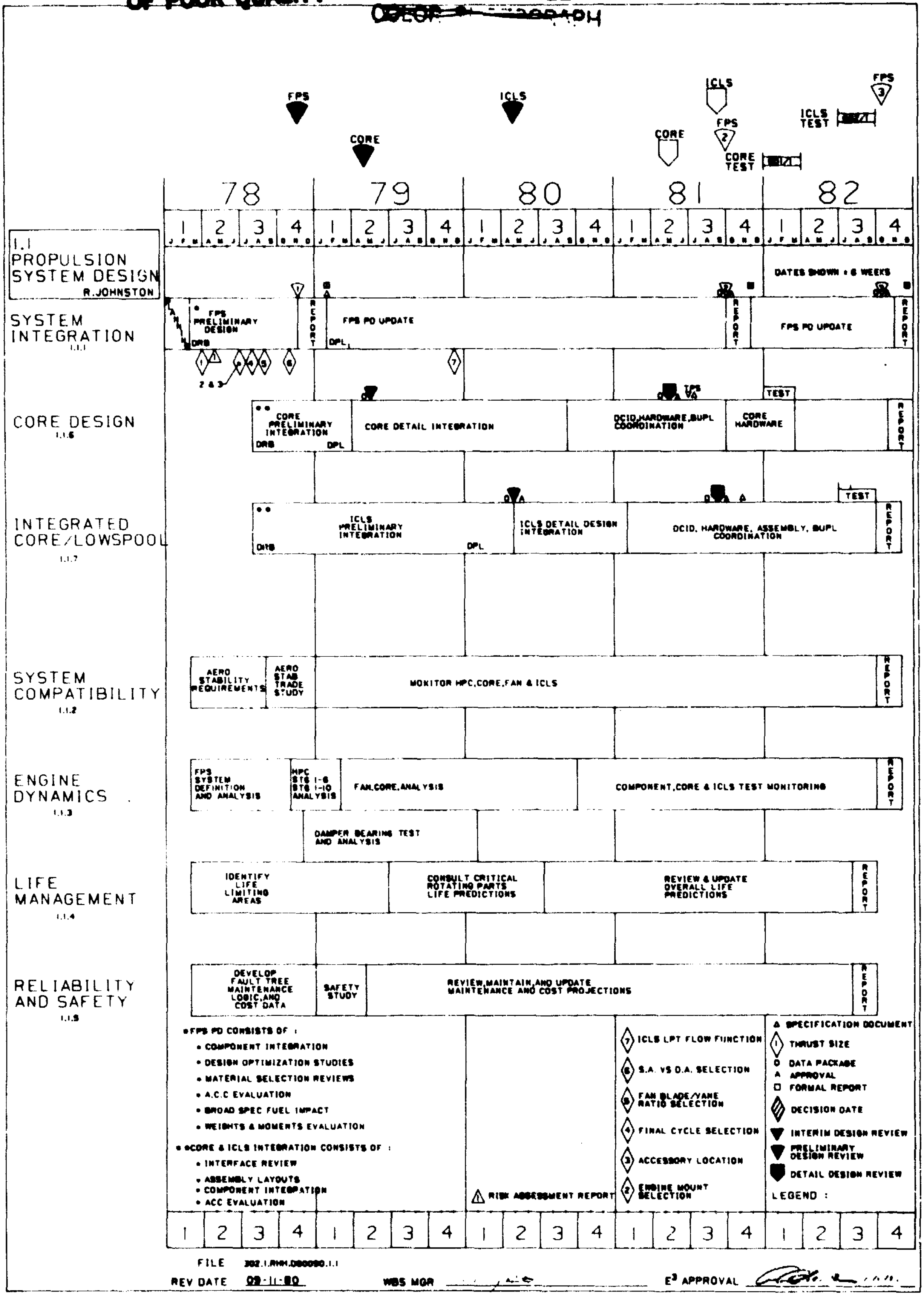


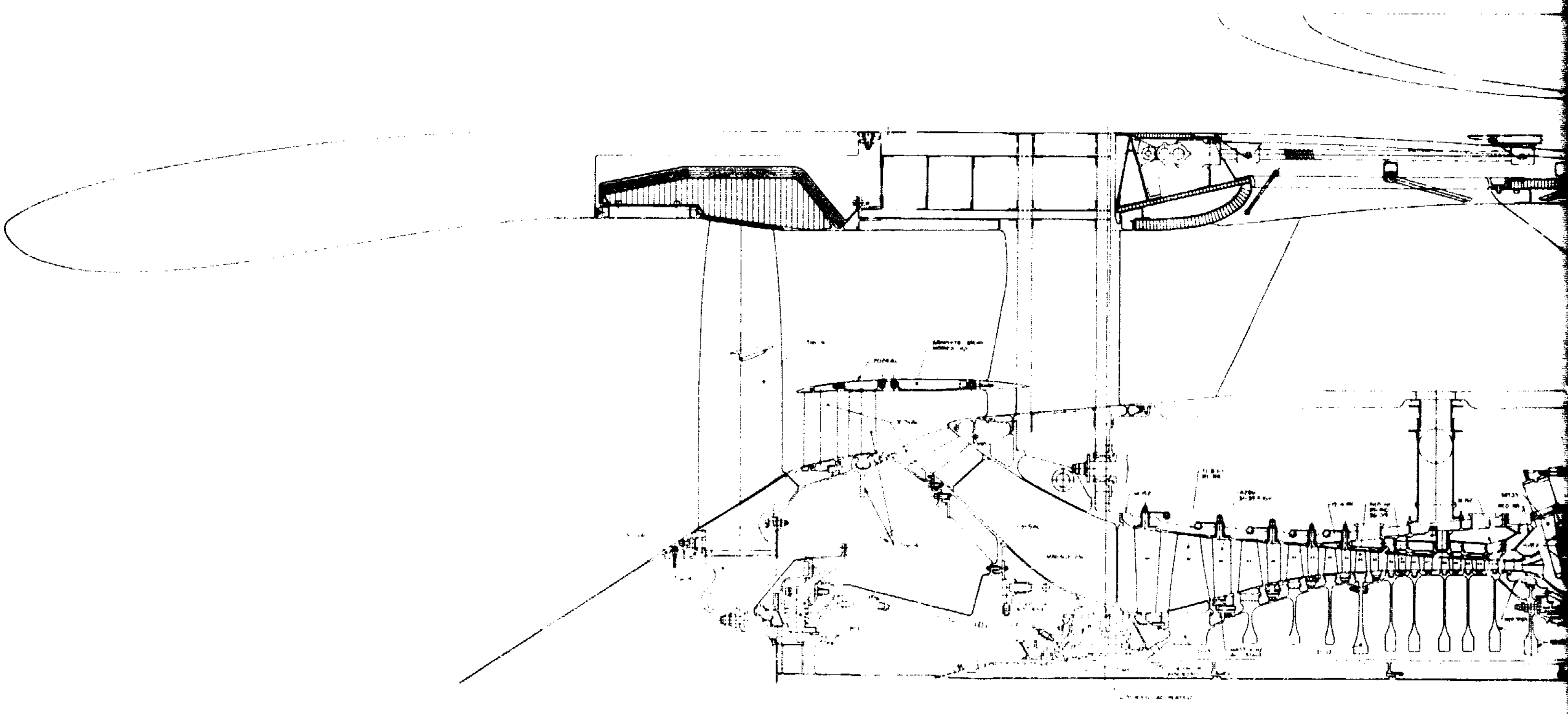
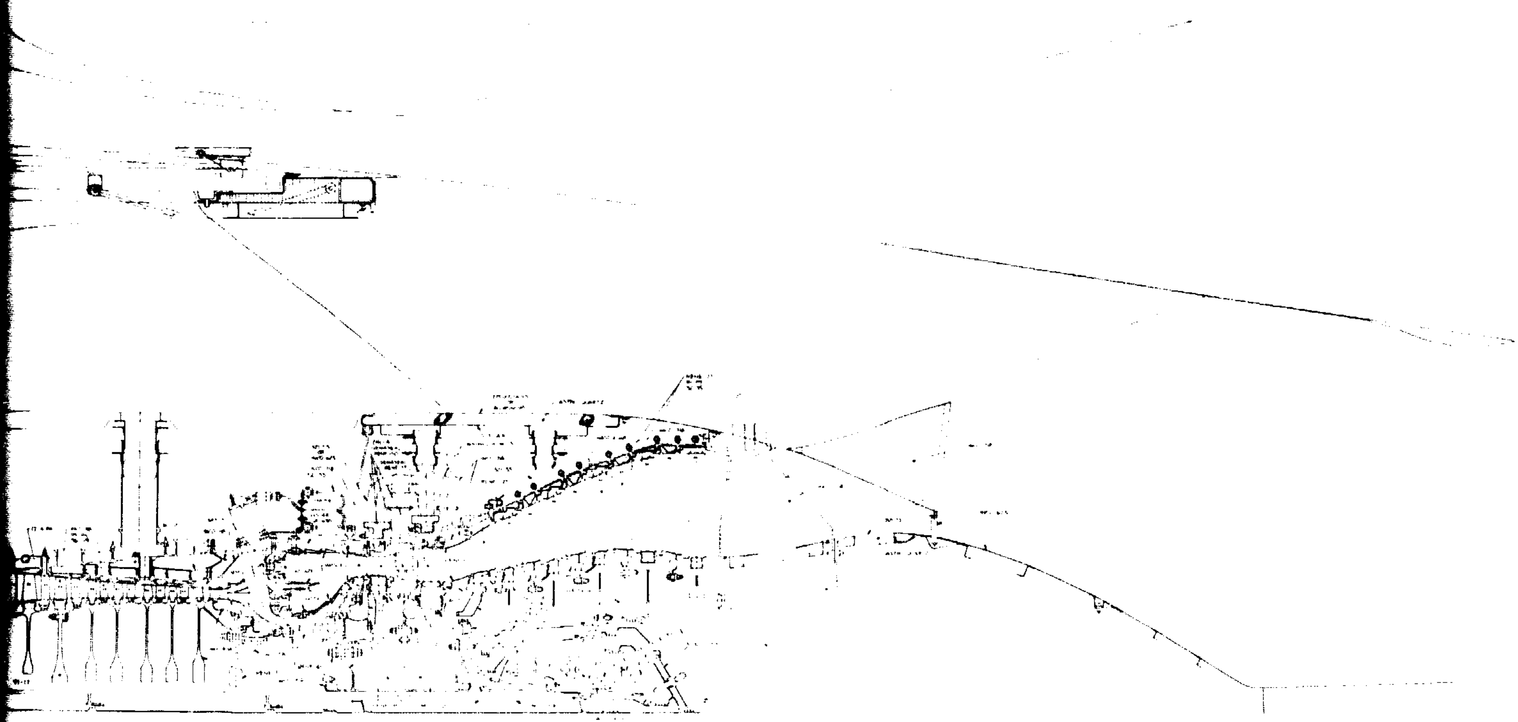


Figure 1.1-2. FPS Material Ident



PS Material Identification.

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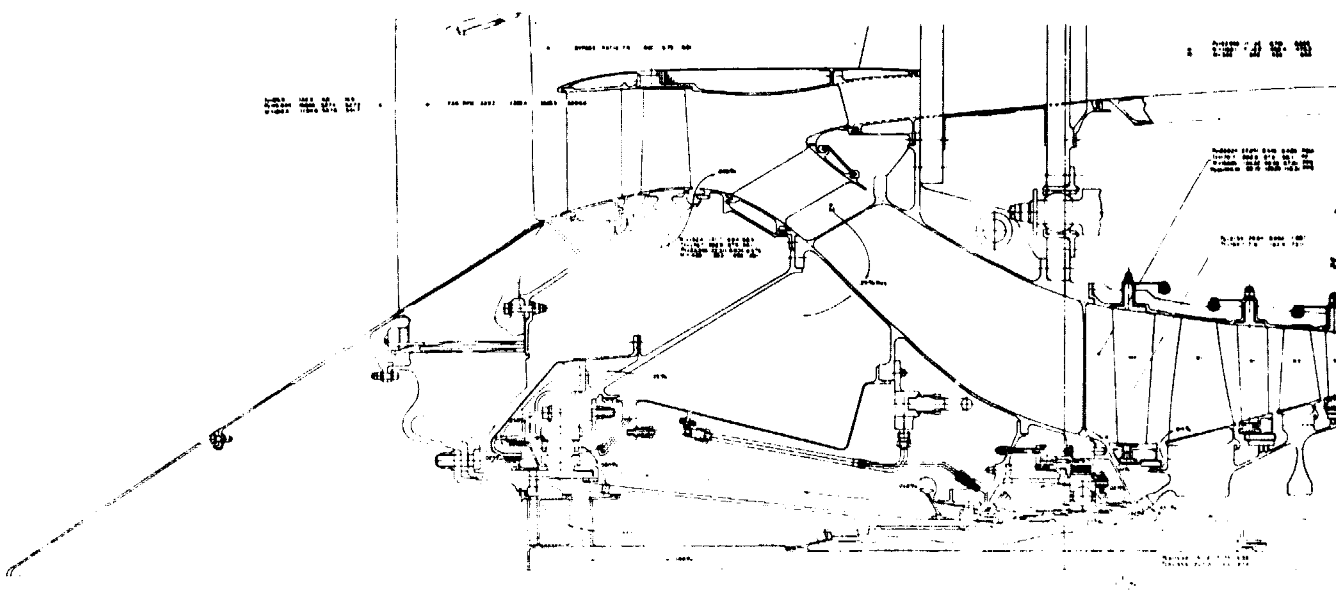

Figure 1.1-3. FPS Operating Parameters and Cool

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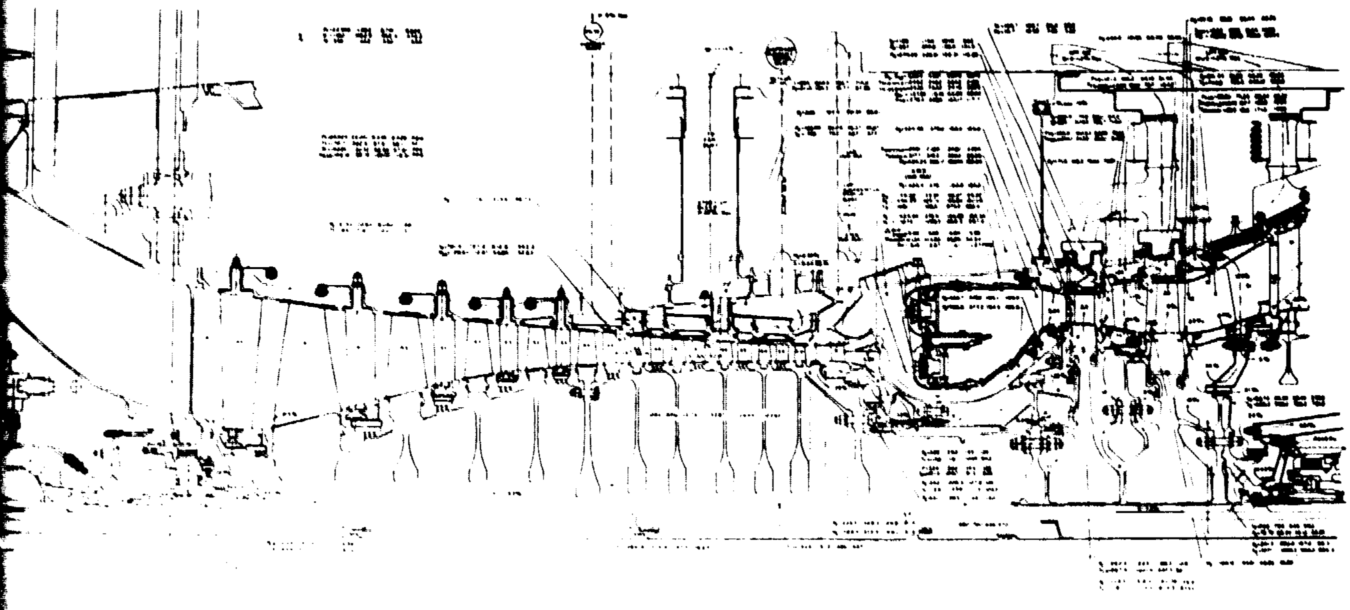

perating Parameters and Cooling Supply System.

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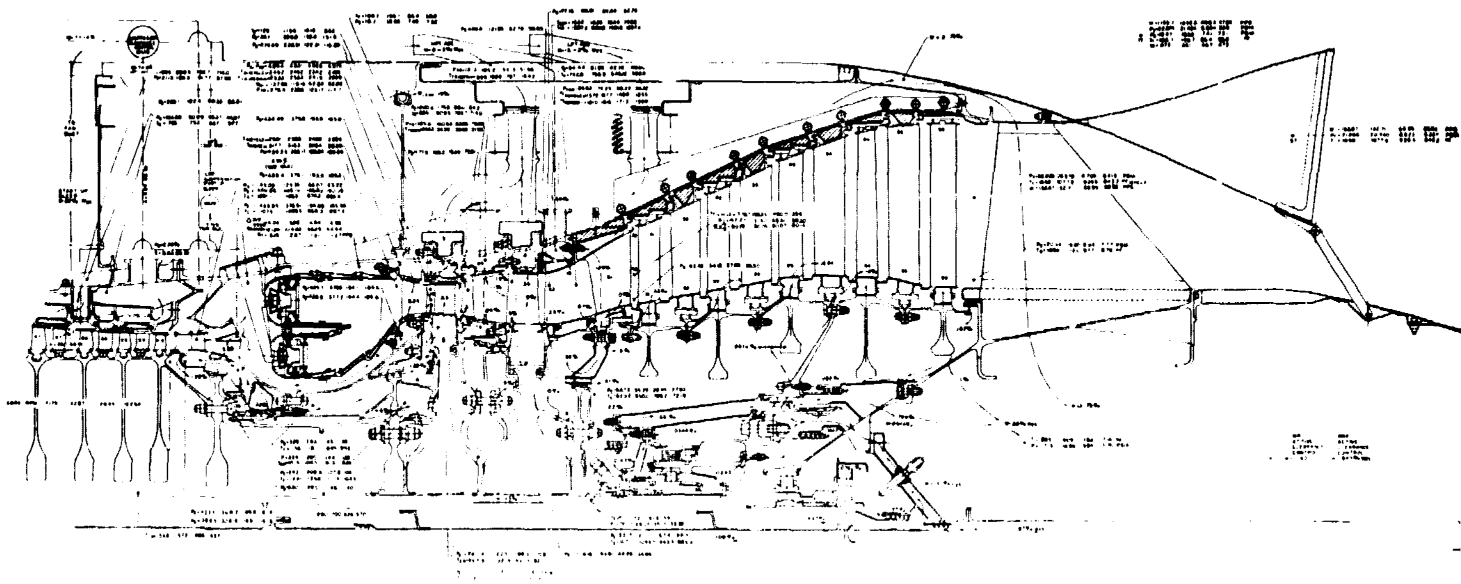

Figure 1.1-3. FPS Operating Parameters and Cooling Supply System

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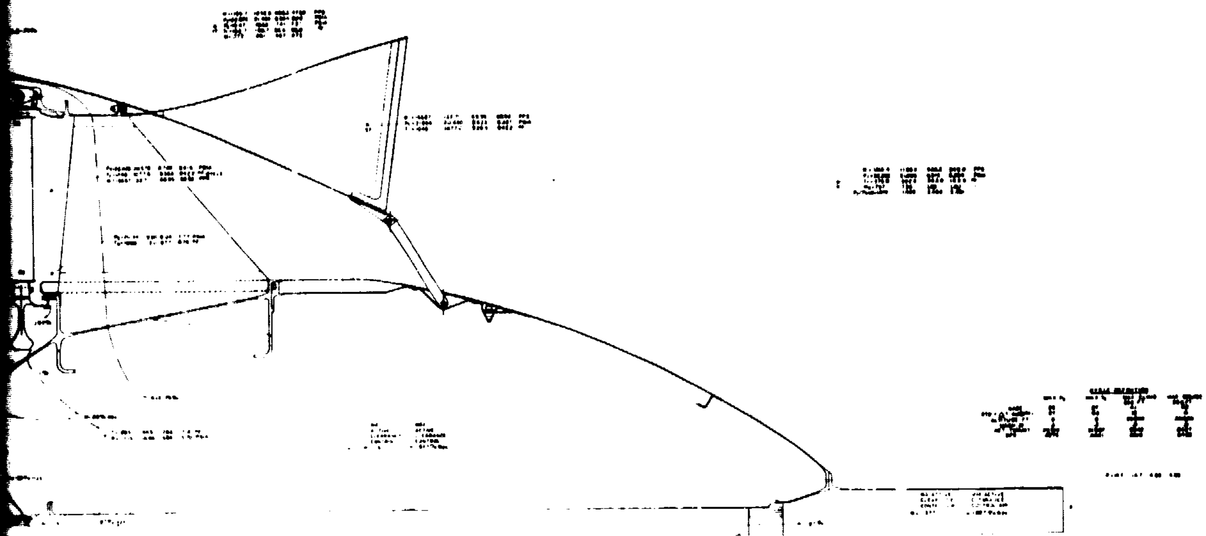

rameters and Cooling Supply System (Concluded).

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Work Planned

- Update weight status
- Update secondary flow and operating parameters drawing
- Continue FPS refinement and optimization
- Prepare the second FPS Preliminary Design Review

1.1.2 System Compatibility

Technical Progress

Distortion screens were designed for the compressor and fan component test rigs. Test planning was conducted, coordinated with areo engineering, for the 1-10 stage compressor test and the fan test.

Work Planned

Monitor the 1-10 stage compressor rig test and analyze the data for determining stability characteristics of the compressor.

1.1.3 Engine Dynamics

Technical Progress

FPS

The FPS planar finite-element model was revised to more accurately represent the combustor, HP turbine, and LP turbine casings. Results from the updated static analysis indicated that beam bending clearance loss was reduced for this configuration. Table 1.1-I summarizes the projected beam bending clearance required for maneuver load combinations, which represent actual flight conditions. The dynamic analyses did not indicate any changes in the core or LP system synchronous vibration characteristics for the revised configuration.

The FPS/ICLS torsional natural frequency was calculated to be 16.4 Hz. An extensive investigation into the potential sources of excitation concluded that there will not be any engine problems related to this torsional frequency.

Table 1.1-I. FPS Clearance Required Due to Combined Loading for Specific Flight Conditions.

Vertical Plane - Positive Deflection Extreme

Stator-Rotor Relative Deflection, Mils (-) Closure at 12:00

Flight Condition	Fan	Booster	HPC 1	HPC 5	HPC 9	CDP Seal	HPT 1	HPT 2	LPT 1	LPT 3	LPT 5
Takeoff Rotation	+20.48	-1.91	- 1.02	- 0.70	- 1.67	- 1.77	- 0.77	- 0.06	- 0.31	-0.28	- 5.30
Second Segment Climb	+23.62	-7.39	+10.49	+10.63	+10.68	+11.17	+11.67	+11.97	+16.40	-0.80	-27.22
Low Mach Cruise	+12.63	-1.22	+ 4.92	+ 4.83	+ 5.06	+ 5.54	+ 6.63	+ 7.21	+ 4.72	+2.70	- 4.09
High Mach Cruise	+18.54	-5.30	+ 8.79	+ 9.39	+ 9.68	+10.22	+10.83	+11.14	+13.00	+0.48	-19.65

Maneuver Loads Which Simulate Flight Conditions

Vertical Plane - Positive Extreme

	Nacelle Airloads		Thrust	Gyro Loads	G-Load
	Mv in.-lb	Fv lb	Fn lb	rad/sec	G's
Takeoff Rotation	317,000	3,950	28,000	+0.10	1 + 0.2
Second Segment Climb	64,000	2,200	18,000	+0.40	1 + 1.8
Low Mach Cruise	108,000	2,800	7,000	+0.10	1 + 1.0
High Mach Cruise	63,000	1,800	8,400	+0.30	1 + 1.5

"RSS" values provided unless magnitude of "SUM" is less than magnitude of "RSS"

"RSS" = Δ airloads + Δ thrust + $\sqrt{(\Delta \text{ gyro loads})^2 + (\Delta \text{ G-loads})^2}$ "SUM" = Δ airloads + Δ thrust + Δ gyro loads + Δ G-loads

ICLS

A comprehensive ICLS LP system vibration analysis was completed, and the clearance loss characteristics were defined for both normal and blade-out unbalance. Table 1.1-II indicates the clearance required to prevent blade rubs and rotor-rotor rubs due to normal unbalance. Table 1.1-III defines the relative deflection expected between blades and case, and between rotors in the event of a blade loss. Figure 1.1-4 illustrates the relative deflection between the LP shaft and the HP rotor due to the loss of 1-1/3 complete fan blades. The blade-out results are conservative as no secondary load paths have been accounted for due to stator-rotor rubs. This verified that although a blade loss would cause blade rubs, no rotor-rotor or case-rotor relative deflections would exist, which could cause fan shaft failure, engine mount failure, or any other catastrophic failure.

Core Test Vehicle

The Core Test Vehicle NOVAS analysis, a modal subsystem method, was completed and the vibration response characteristics were shown to be consistent with previous results generated using VASTH. The analysis confirmed that the damper design will provide effective vibration control for normal and high unbalance.

A system vibration analysis of the Core Test Vehicle was conducted which addressed to the response characteristics of the instrumentation tube. The first n = 1 frequency was computed to be at 14,300 rpm based on the front of the tube being pinned, which was conservative. Based on the front of the tube being clamped, which was the design objective, the first n = 1 frequency was computed to be at 18,100 rpm. Therefore, the mode is not expected to present a problem, since it is undamped, resulting in a sharp buildup which exists completely above the speed range for either of the boundary conditions. A shake test will be conducted to verify the frequency.

Two Shaft Damper Rig

The two shaft damper rig simulates the engine dynamics associated with an LPT rotor mode that occurs at approximately 4000 rpm in the engine. This

Table 1.1-II. ICLS Clearance Required for Normal Unbalance.

Unbalance, and Structural Damping	MILS-SA at LP RPM								
	Stator-Rotor Relative Deflection					Between LP Shaft And Core Rotor		Across Centering Springs	
	Fan	HPC 1	HPT 1	LPT 1	LPT 5	At Mid LP Shaft	At HPT Rotor 1	Fwd	Aft
500 gm-in. at Fan Q = 15	0.5 @ 3500	1.8 @ 3170	3.6 @ 3170	1.6 @ 3170	0.1 @ 3170	1.3 @ 3170	1.6 @ 3170	1.0 @ 3170	2.0 @ 3170
200 gm-in. at LPT 1 Q = 15	0.2 @ 3170	1.9 @ 3170	3.9 @ 3170	1.7 @ 3170	0.3 @ 3500	1.3 @ 3170	1.7 @ 3170	1.0 @ 3170	2.1 @ 3170
200 gm-in. at LPT 5 Q = 15	0.2 @ 3170	2.4 @ 3170	4.9 @ 3170	2.2 @ 3170	0.5 @ 3500	1.7 @ 3170	2.1 @ 3170	1.3 @ 3170	2.7 @ 3170

Table 1.1-III. ICLS Clearance Required Due to Blade Out Conditions.

Unbalance, and Structural Damping	MILS-SA at LP RPM								
	Stator-Rotor Relative Deflection					Between LP Shaft And Core Rotor		Across Centering Springs	
	Fan	HPC 1	HPT 1	LPT 1	LPT 5	At Mid LP Shaft	At HPT Rotor 1	Fwd	Aft
250,000 gm-in. at Fan, Q = 7.5	188 @ 3500	52 @ 3500	99 @ 3500	57 @ 3500	20 @ 3500	95 @ 3500	46 @ 3500	20 @ 3500	20 @ 3500
1350 gm-in. at LPT 1, Q = 7.5	0.7 @ 3170	6.4 @ 3170	13.2 @ 3170	5.7 @ 3170	1.0 @ 3500	4.4 @ 3170	5.7 @ 3170	3.4 @ 3170	7.1 @ 3170
3245 gm-in. at LPT 1, Q = 7.5	1.6 @ 3170	19.5 @ 3170	39.8 @ 3170	17.9 @ 3170	4.1 @ 3500	13.8 @ 3170	17.0 @ 3170	10.6 @ 3170	21.9 @ 3170

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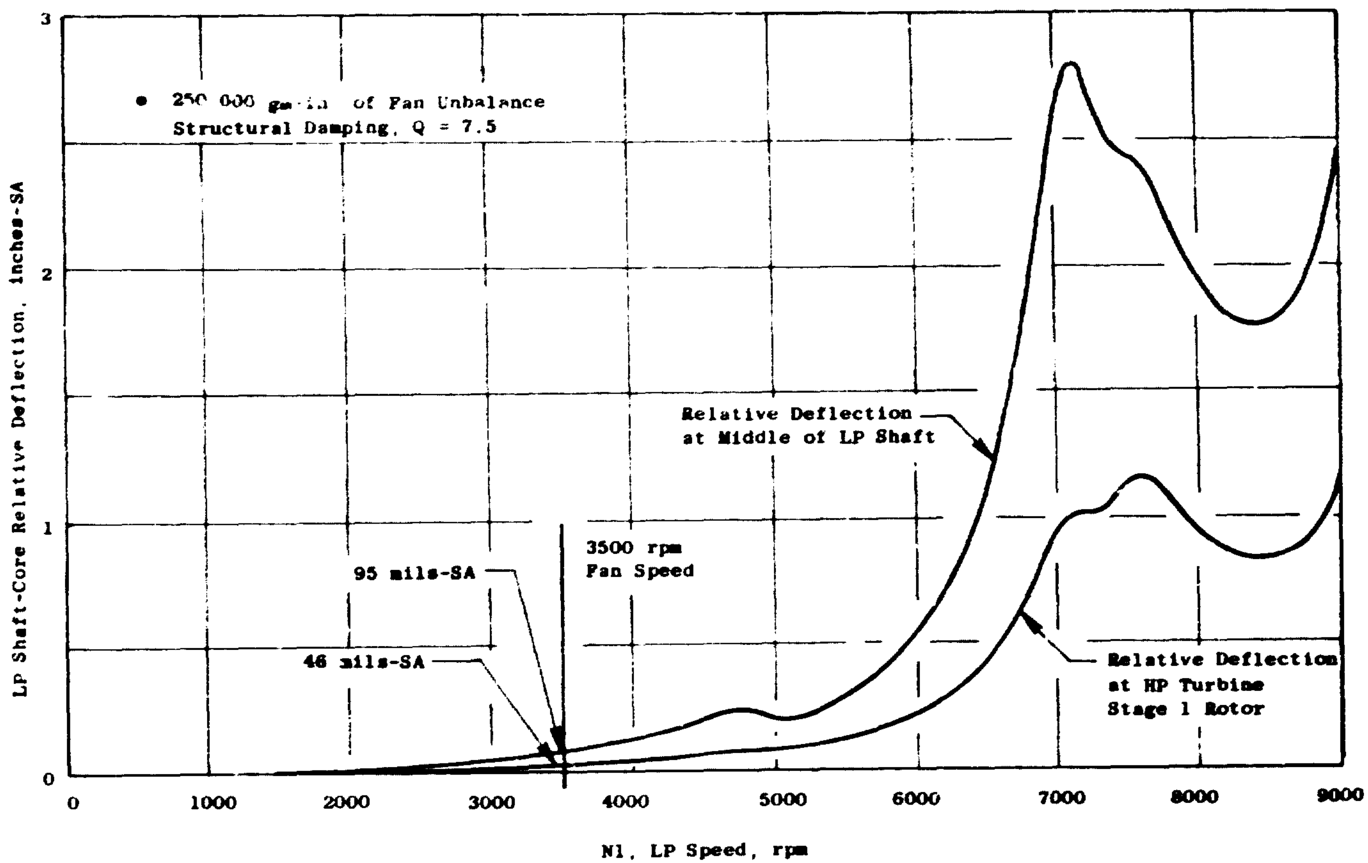


Figure 1.1-4. Relative Deflection LP Shaft/Core Rotor Due to Fan Blade Out Condition.

mode occurred above the rig 4600 rpm N1 red line speed, but the significant response buildup relating to this mode was more than adequate for the purpose of evaluating damper-system performance that could be directly related back to the engine.

Figures 1.1-5 and 1.1-6 show the No. 6 bearing free floating-end flow damper and the multishim, end sealed, spring-mounted superdampers that were tested for evaluation of damper-system performance. Figures 1.1-7 and 1.1-8 show examples of the measured and predicted frequency response characteristics. Figure 1.1-9 shows a comparison of the predictions and test data where both are presented on a normalized basis.

Inspection of this figure shows that the analysis predicts effective performance for the simple end flow damper up to approximately 550 g-in. unbalance while the test results indicate that the limit of effectiveness for the superdamper extends up to approximately 1900 g-in.

The analysis accounts for the increase in oil temperature due to the damper work and indicates that the observed reduction in the superdamper effectiveness is due to oil temperature rise in the damper annulus.

Work Planned

Complete full-scale 10 stage HP compressor test and correlate results with analysis.

Complete full-scale fan test and correlate results with analysis.

Complete ICLS final dynamic analysis and issue report.

Continue FPS dynamic analysis.

Continue FPS clearance studies based on maneuver loads which represent flight conditions.

Write final report for the two-shaft damper rig.

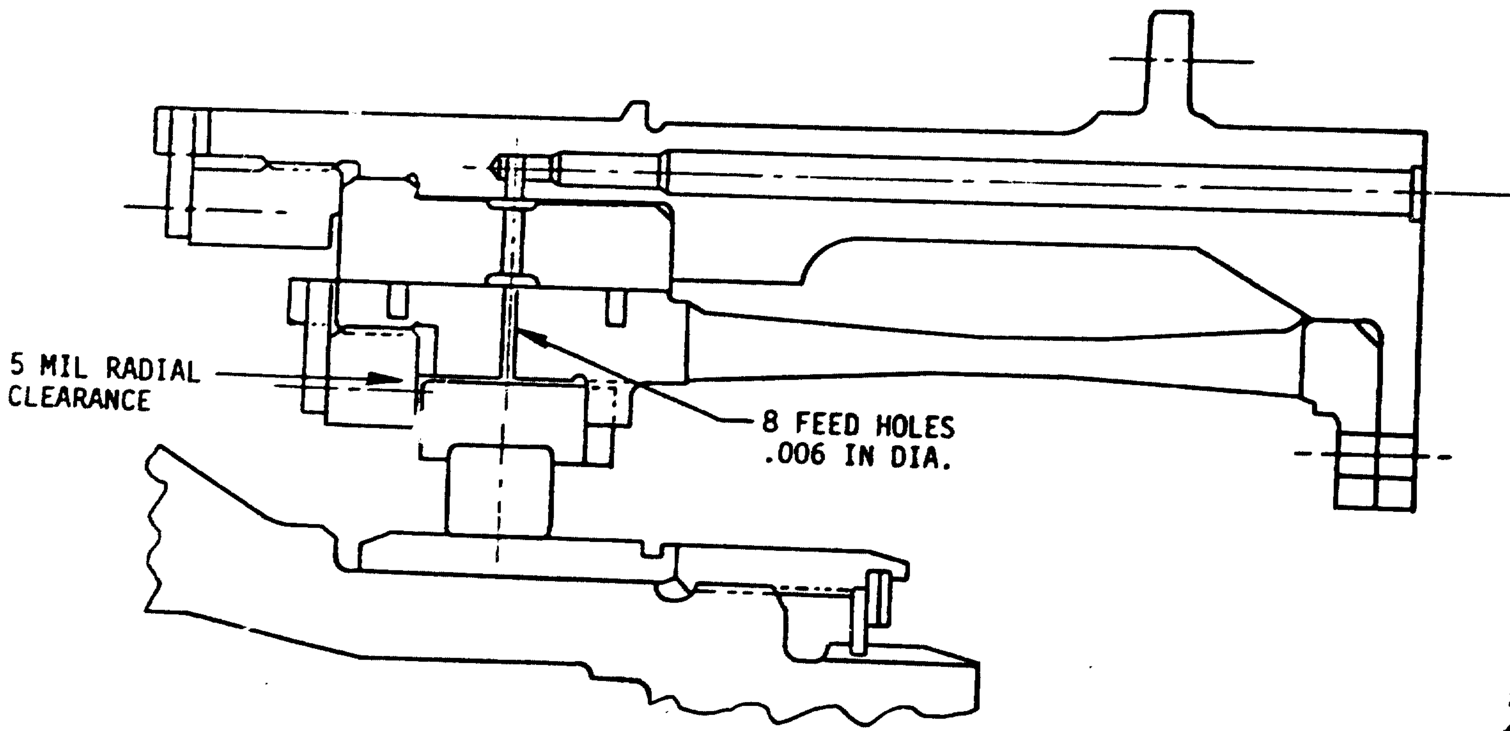
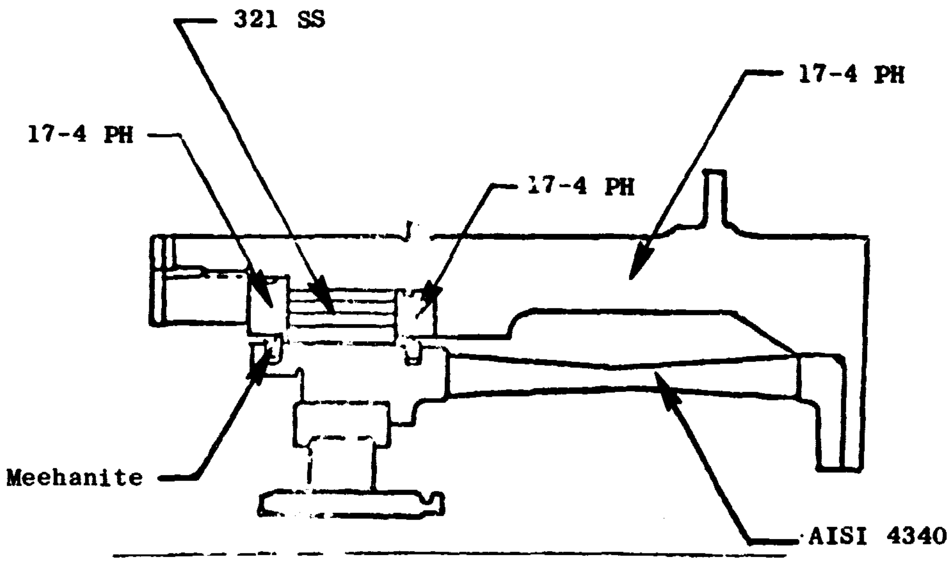


Figure 1.1-5. Two Shaft Damper Rig Free Floating End Flow Damper.



Tested

- Chrome Plated End Plates and Piston Ring Bore
- Four Oil Supply Holes with Check Valves
- Fifty-Two mil Radial Clearance
- Meehanite Arc Bound Piston Rings (10 mil Installed End Cap)
- Four Shims with Drilled Oil Circulation Holes
- 15 Hours 35 Minutes Total Testing Time

Figure 1.1-6. Two Shaft Damper Rig Super Damper Configuration.

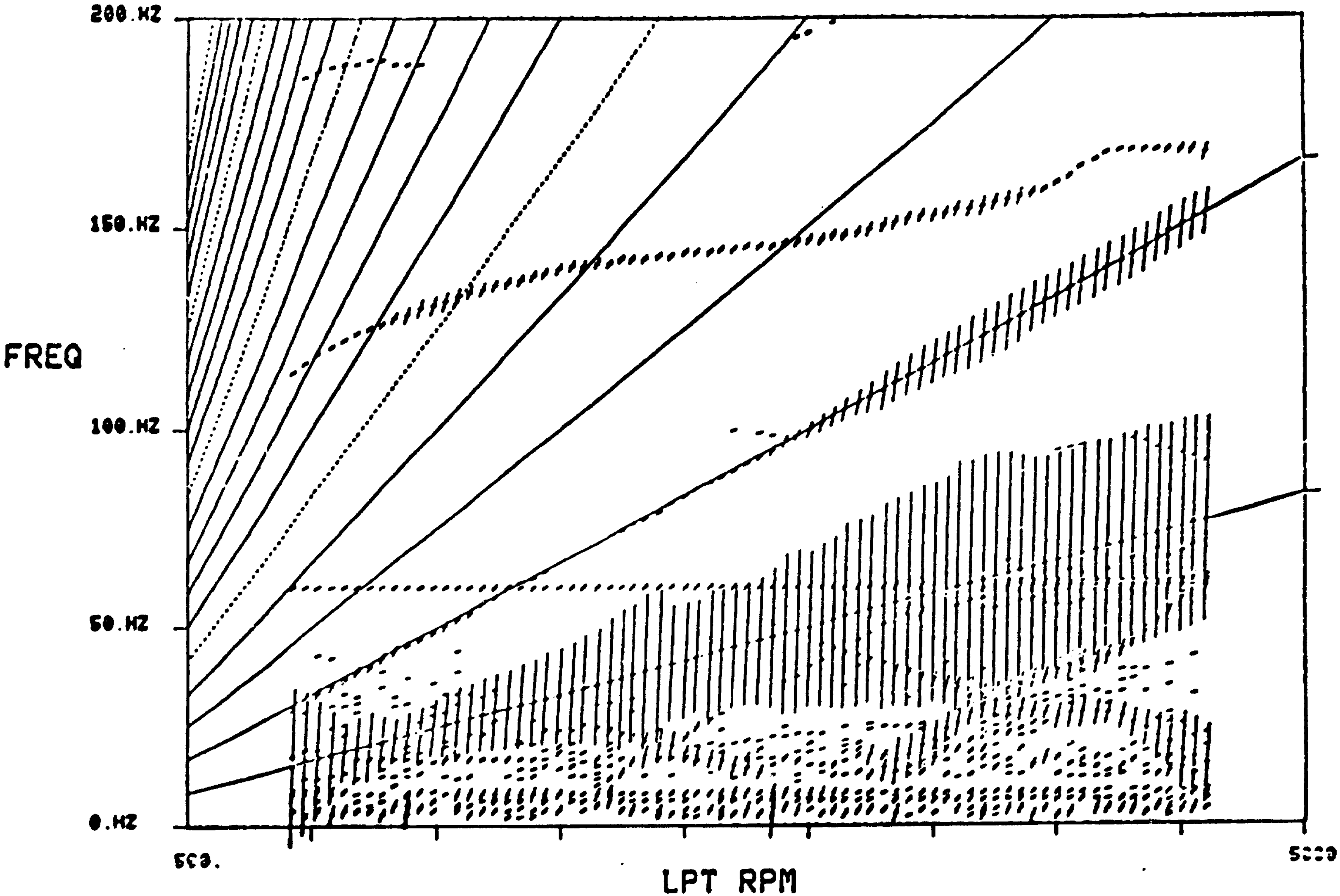


Figure 1.1-7. Super Damper Frequency Response Measurement.

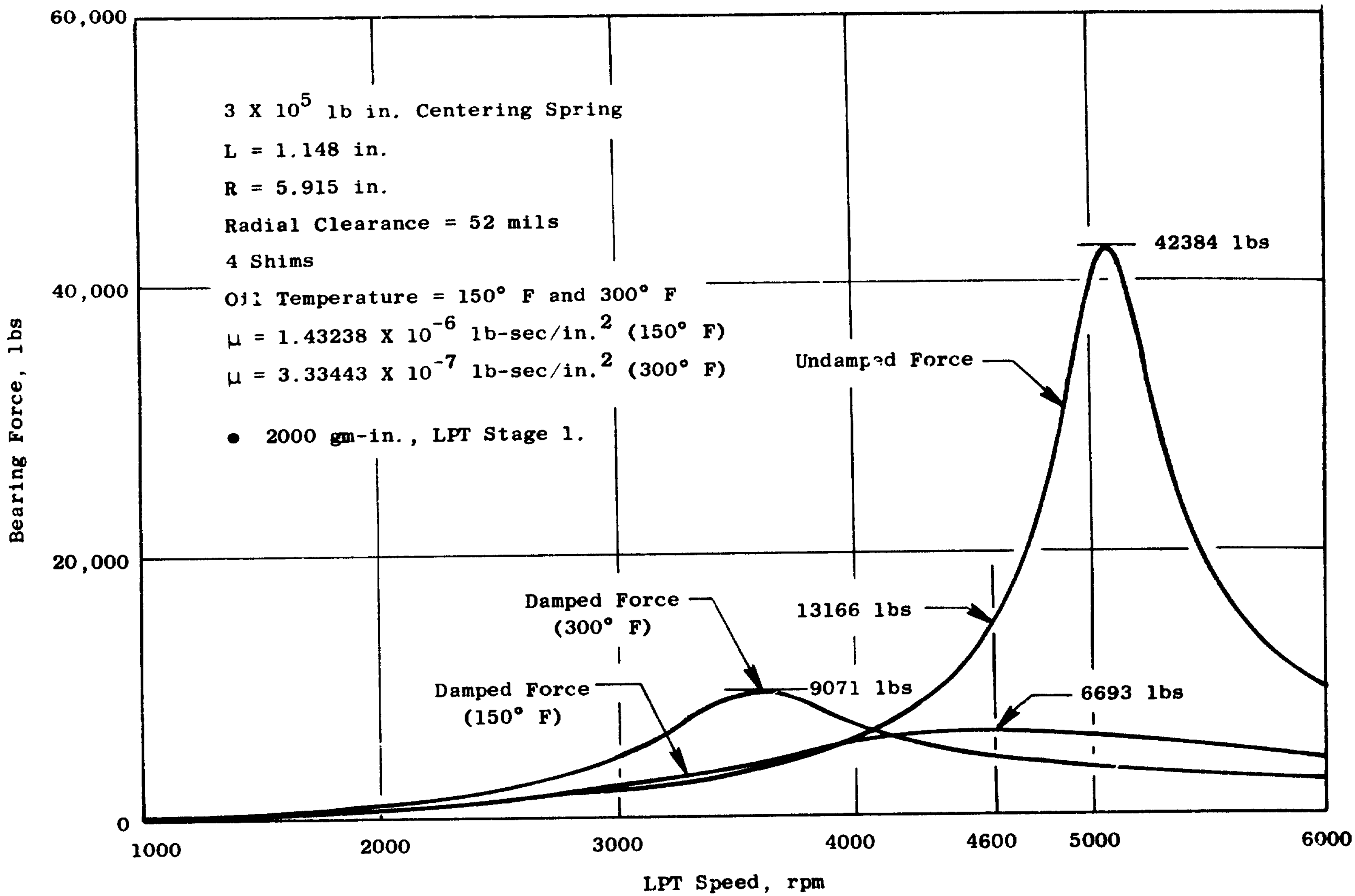


Figure 1.1-8. Two Shaft Damper Rig - Predicted Bearing Force Versus LPT Speed.

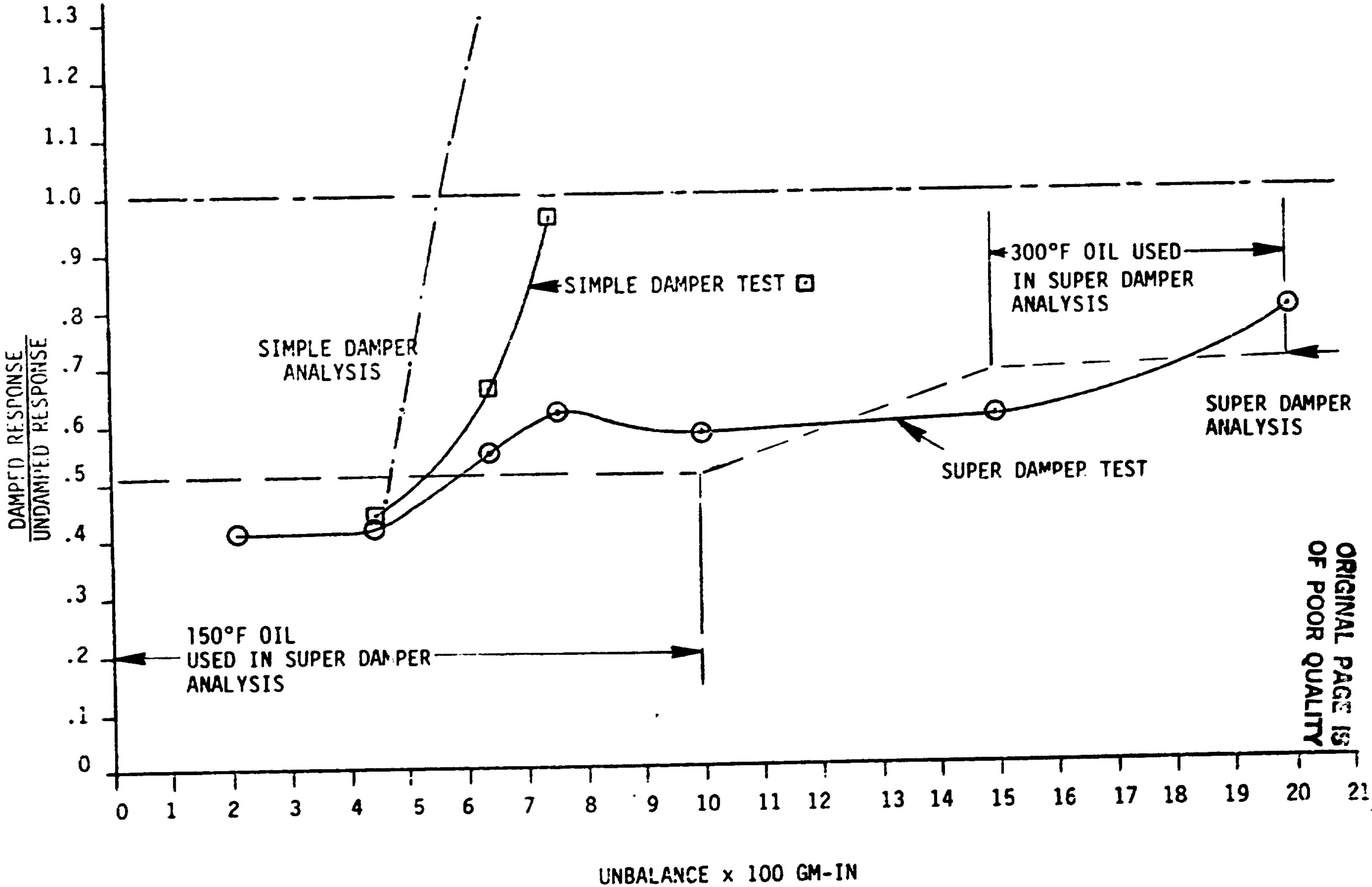


Figure 1.1-9. Two Shaft Damper Rig - Effectivness of Simple and Super Damper Configuration.

1.1.4 Life Management

Technical Progress

An investigation of the effect of minor low cycle fatigue (LCF) cycles during a flight on the life prediction of typical engine rotor components was carried out. The stress and temperature at any given location on the rotors varies considerably during a flight and there is a certain combination of stress and temperature which causes a maximum LCF damage (major LCF cycle). All other variations of stress and temperature which cause damage are considered minor LCF cycles. For example, thrust reverse contributes significantly to minor LCF cycles. An effort was made to quantify the LCF damage due to minor LCF cycles using experience from CF6-6 and CF6-50 rotors. Total damage caused by a minor LCF cycles/flight was calculated as a percentage of the damage/flight caused by the major LCF cycle. The investigation indicated damage due to minor LCF cycles varies from a minimum of zero to a maximum of 55%. The LCF damage assessment was carried out, so that the experience gained in the life extension programs for the CF6-6 and CF6-50 rotor components could be applied to the E^3 components.

A review of the effects of material factors which might affect LCF crack initiation was carried out. Electrical chemical machining (ECM) and electrical discharge machining (EDM) are planned for manufacturing of rotor components of the E^3 HPT. Studies have shown that significant component life reduction may result from surface damage inflicted by the above processes during manufacturing. It was recommended that manufacturing processes be specified to avoid surface damage and, if possible, to specify processing which will enhance part LCF performance.

Work Planned

Consultation regarding Life Management Techniques to E^3 design areas will continue.

1.1.5 Reliability and Safety

Technical Progress

The Preliminary Hazard Analysis was issued. The objective of the hazard analysis was to determine whether probable malfunctions, failures or improper operations will result in hazardous "undesired events." A secondary objective of the analysis was to identify the potential for an in-flight shutdown, which is termed "undesirable" but not necessarily "hazardous." Based on the configuration analyzed and the data available, no probable malfunction, nor any probable single or multiple failure, nor any probable improper operation of the engine will cause the probability of occurrence of a hazardous "undesired event" to exceed an acceptable rate.

Work Planned

No specific tasks are planned.

1.1.6 Core Analysis and Design

Technical Progress

The engine cross-section drawing was updated. Changes were made in the sumps, turbine, and rear frame. The drawing is shown in Figure 1.1-10.

The operating and cooling parameters are shown in Figure 1.1-11, the materials in 1.1-12, and the piping in 1.1-13.

The weight and center of gravity were determined for the core engine demonstrator.

Wishbone structure within the CDP bleed manifolding became overly restrictive to the bleed flow. Flow restrictions were reanalyzed and opened up.

Selection of blade tip and seal clearances is underway. The criteria is to model the clearances of a 3000-hour production engine operating at cruise.

A stack-up analysis and evaluation of critical radial and axial clearances is underway. The purposes of this study are

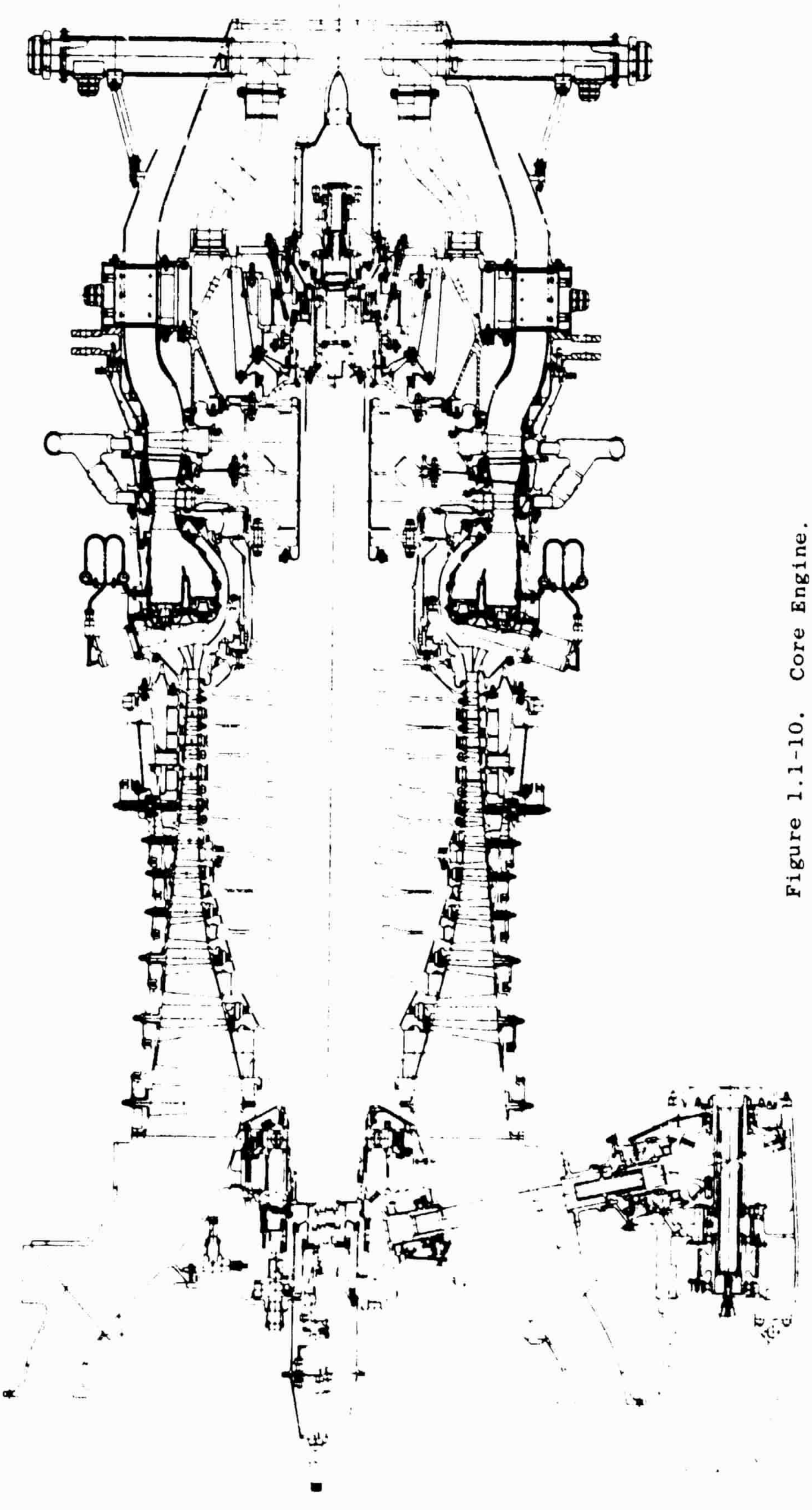


Figure 1.1-10. Core Engine.

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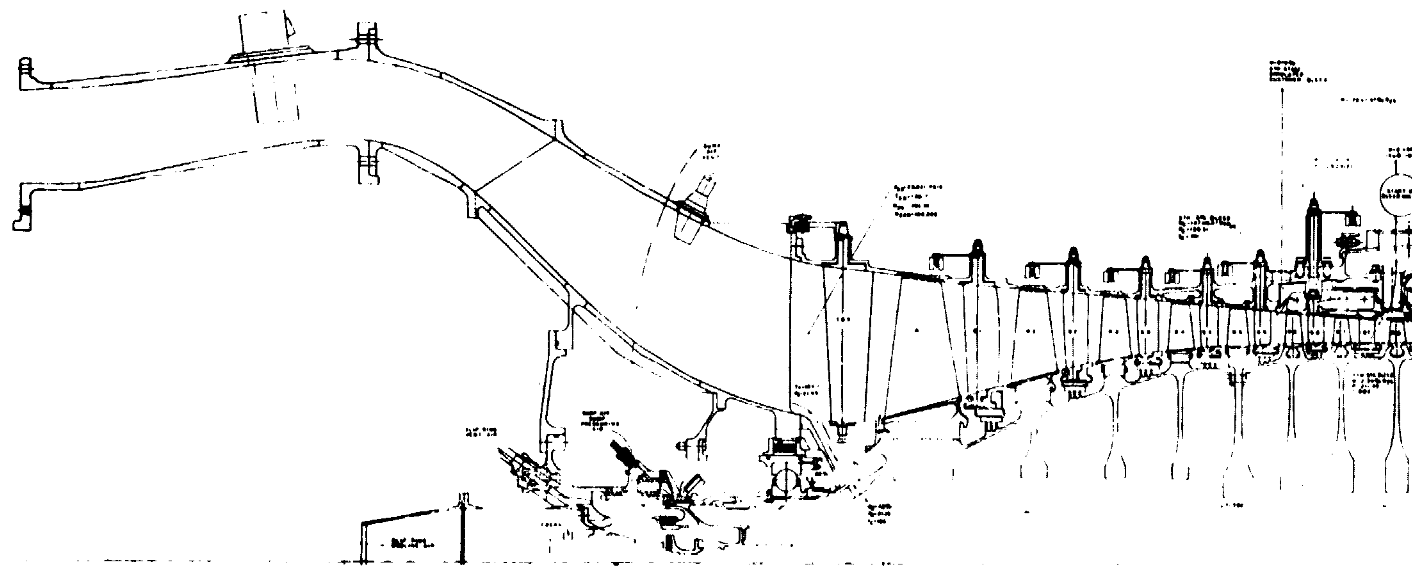

Figure 1.1-11. Core Engine Operating Paramete

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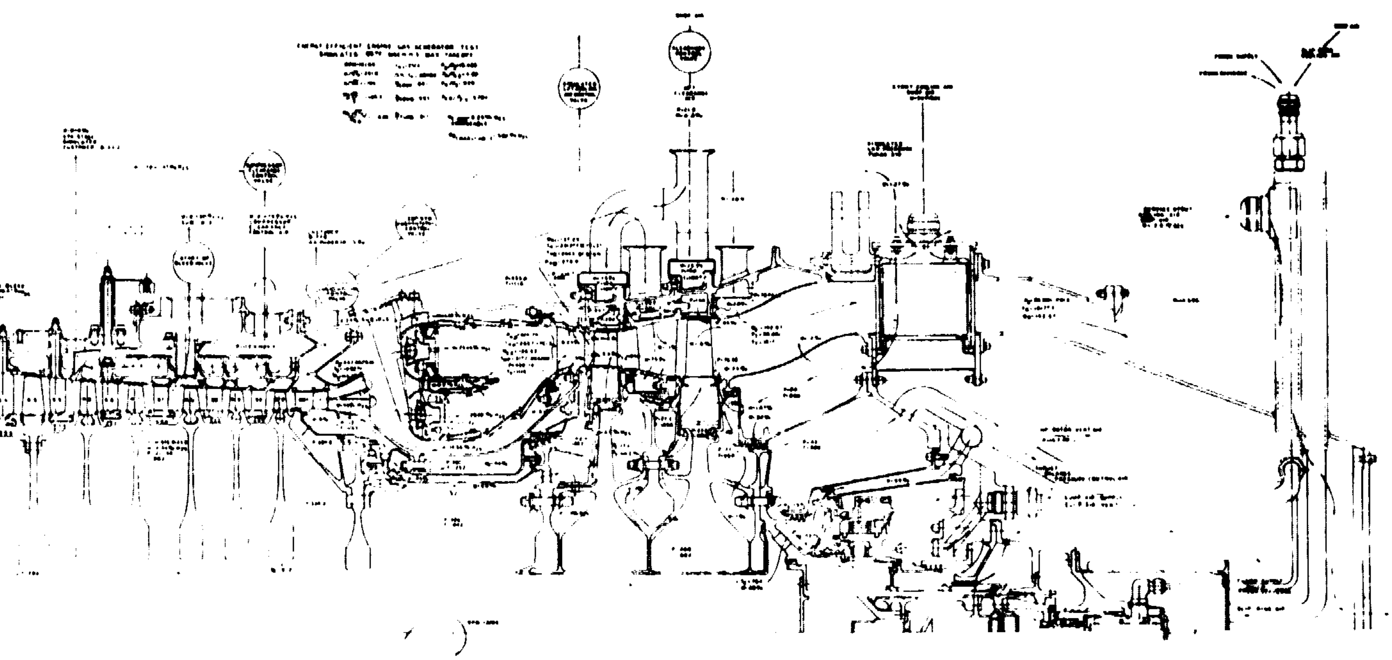

ating Parameter and Cooling Supply System.

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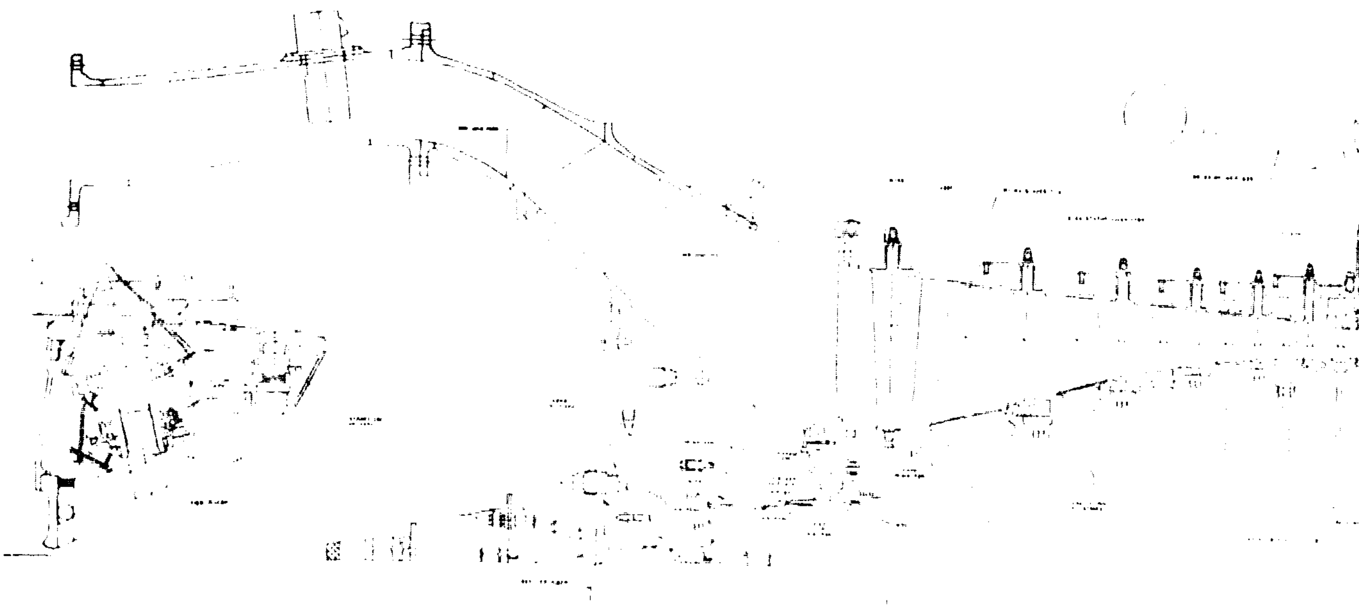

Figure 1.1-12. Core Engine Mat

ore Engine Materials Identification.

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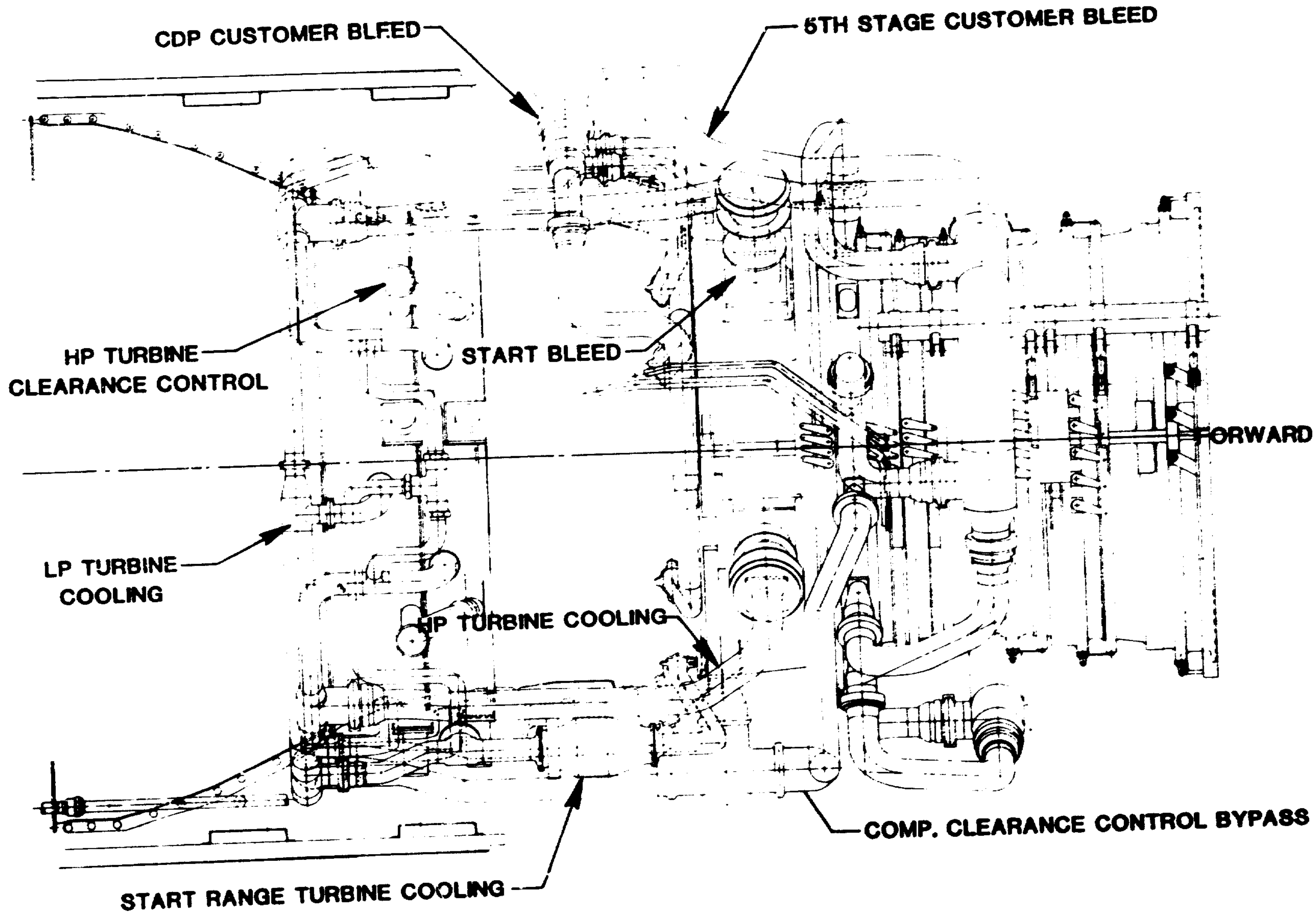

Figure 1.1-13. Core Engine Piping.

- Ensure that the engine can be assembled
- Ensure that the critical clearances are within limits for successful engine operation
- Provide data for selecting shim dimensions
- Document actual build clearances.

The results of this work will be a specification of required clearances and monitoring of hardware clearances as hardware is assembled.

A drawing was established to define interfaces between the test engine and the facility.

An engine stack-up roadmap drawing was essentially completed. This defines responsibilities for stack-ups and records component stack-ups.

Work Planned

Conduct the core detail design review.

1.1.7 ICLS Analysis and Design

Technical Progress

The engine cross section is being updated. The inlet, nacelle, fan frame, low pressure turbine, mixer, and sumps were revised. The updated drawing is shown in Figure 1.1-14.

Operating and cooling parameters are shown in Figure 1.1-15 and materials in 1.1-16.

Responses to additional questions resulting from the Preliminary Design Review were supplied to NASA.

Weight and center of gravity of the test vehicle were established.

Mount link loads were reevaluated for a fan blade-out condition and were refined for final mount geometry. Design refinement of the facility mount links and mount lugs continues.

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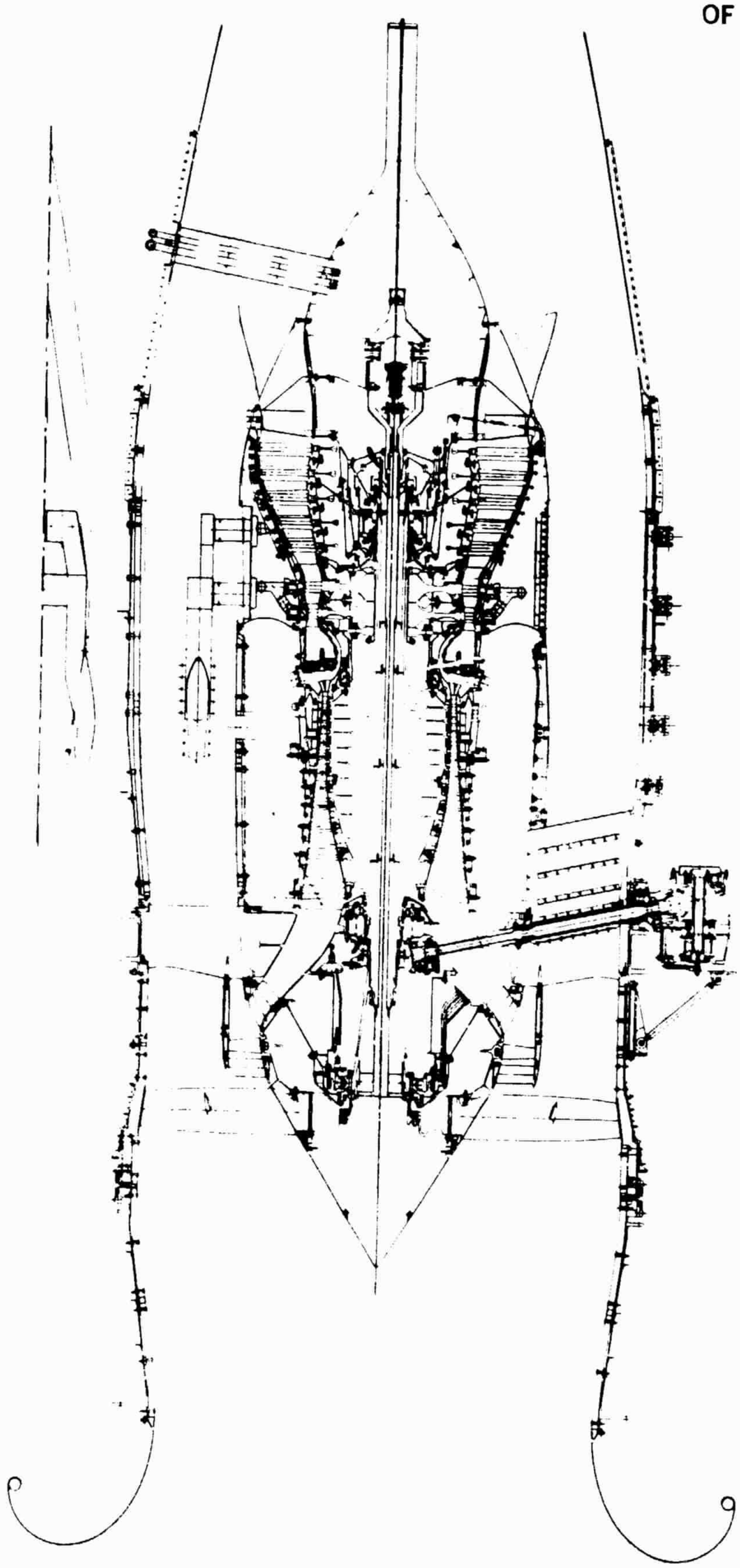

Figure 1.1-14. ICLS Cross Section.

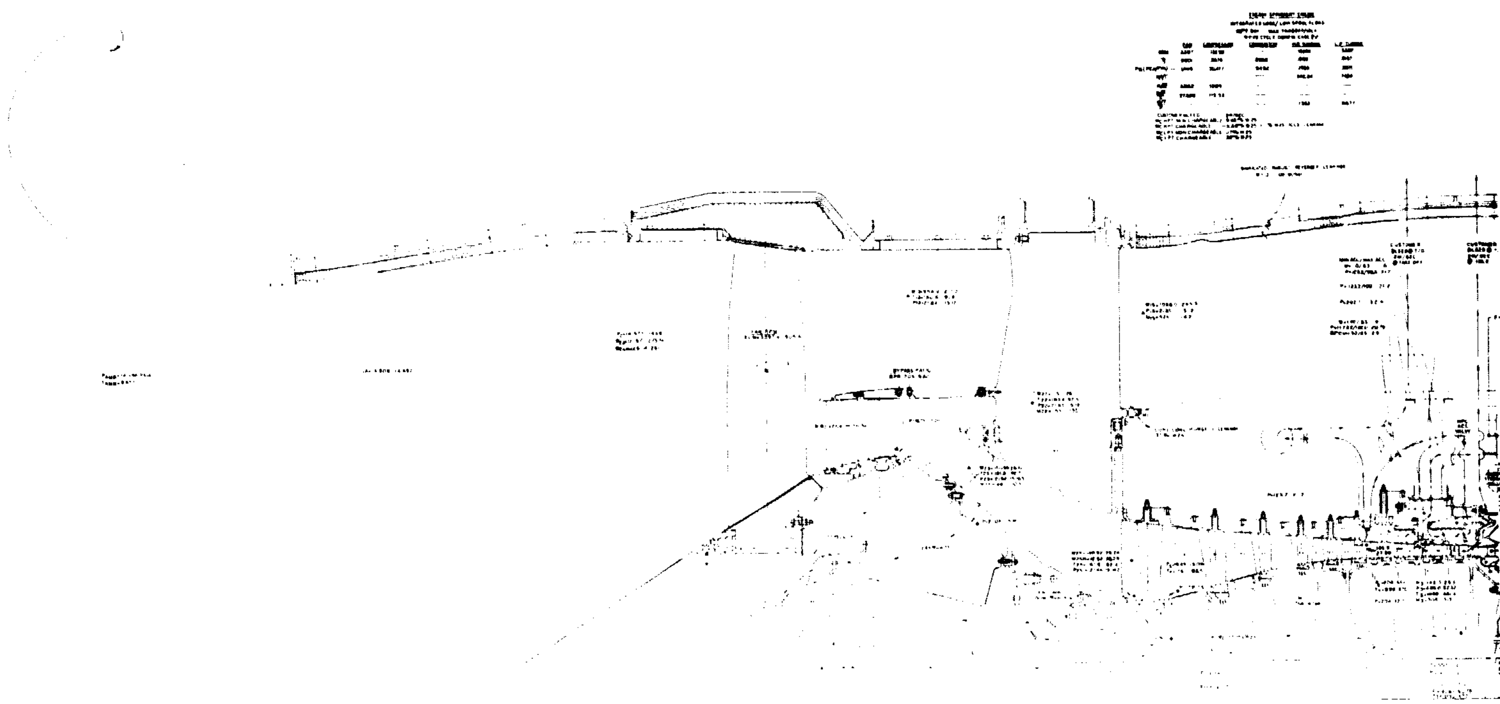

Figure 1.1-15. ICLS Operating Parameters and (

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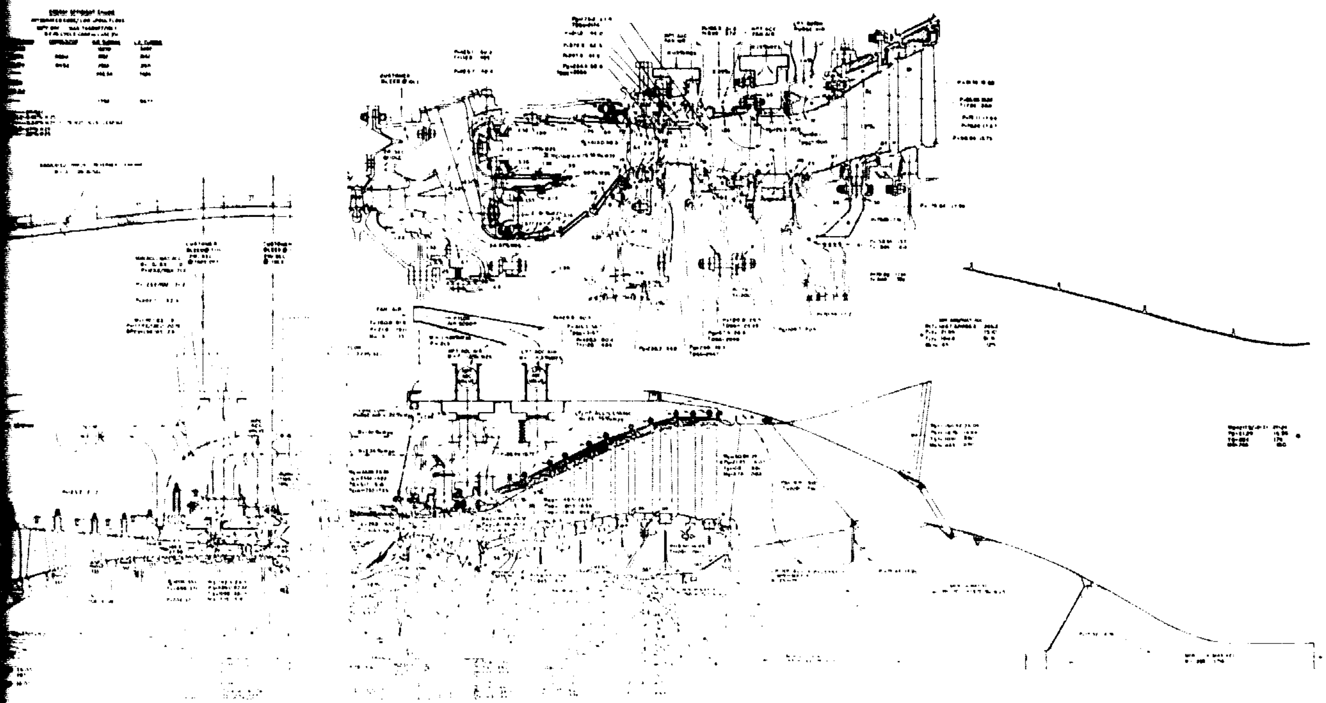

ng Parameters and Cooling Supply System.

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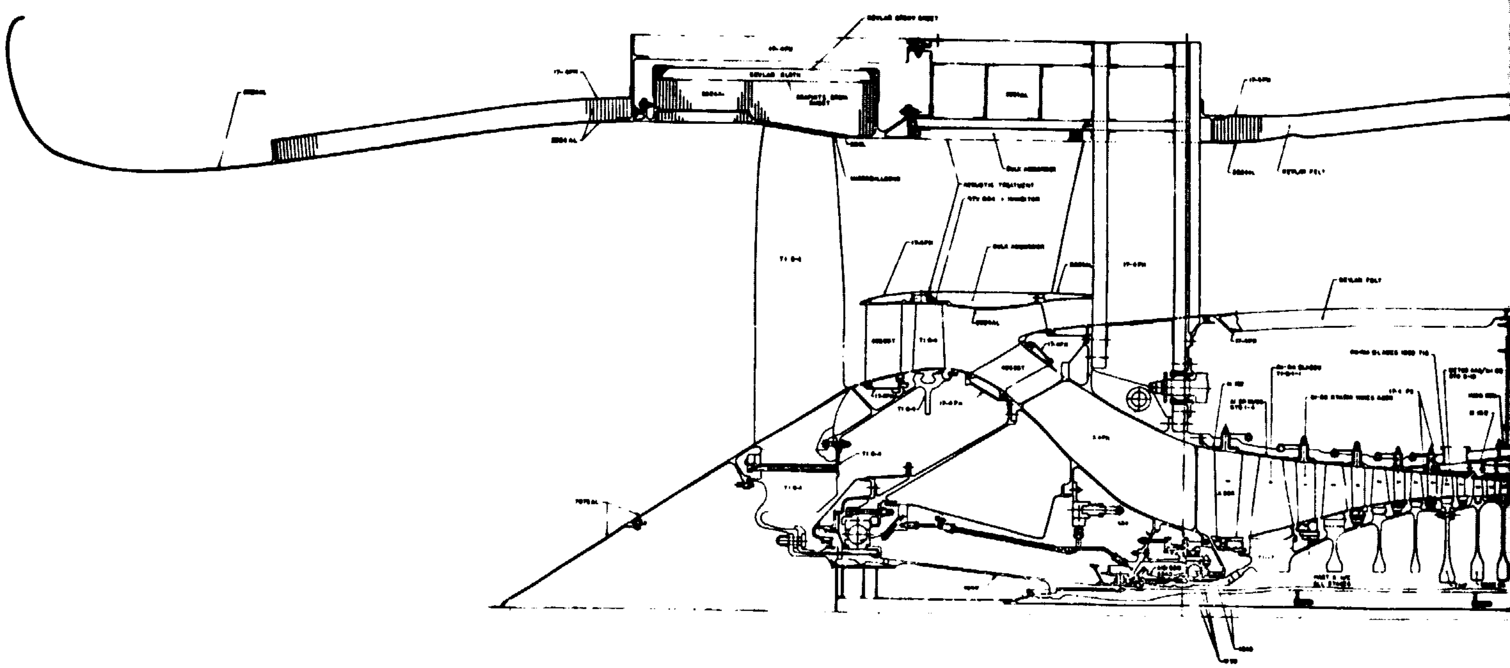

Figure 1.1-16. ICLS Material

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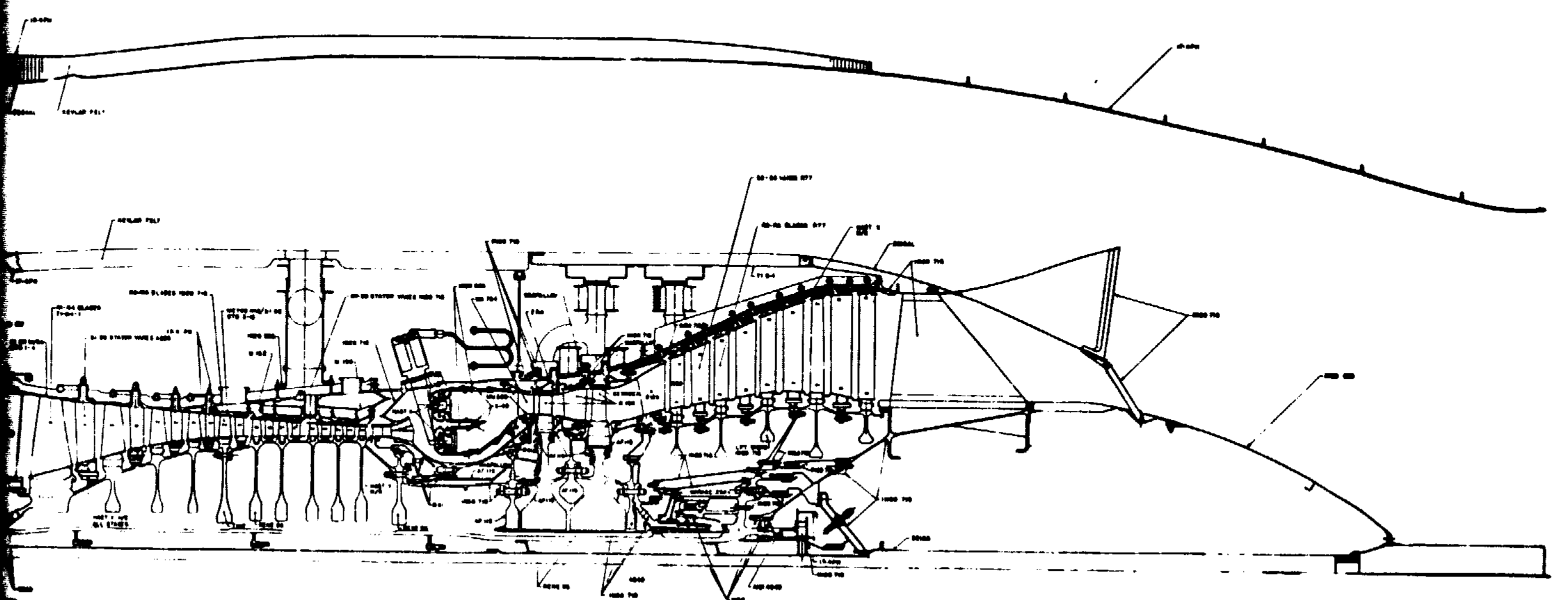

-16. ICLS Material Identification.

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The aft instrumentation strut was configured. This strut carries instrumentation and service lines to the aft slipring. It will be a 6 x 1-1/2 inch streamlined fairing located behind a mixer lobe and extending from the tailcone into the pylon.

Combustor start sequence and means for light-off detection for the pilot and main zones were established.

The key dimension drawing was updated with the same changes discussed in Section 1.1.1.

The hot flowpath was updated. Inner and outer nacelle flow lines and the mixer were changed.

The Multiple Engine Build Up List (MEBUL) was created. Previously, the parts list was defined by the drawing status report.

Work Planned

Continue general support. Update cooling air and secondary flow drawings.

1.2 CYCLE AND PERFORMANCE

Technical Progress

The effort during this reporting period has been concerned with four tasks:

- Engine Deterioration Model
- Update of the ICLS Test Vehicle Cycle
- Cycle Deck logic modifications
- Core Test planning.

The first task has been completed whereas the other three tasks are still in progress. Good progress has been made on the planned cycle deck logic changes (Semiannual No. 5) which are nearly complete. The remaining tasks are continuous through the engine test programs.

Engine Deterioration Model

The effort to define a preliminary E^3 deterioration model based on the CF6-50 Engine Performance Deterioration Report (NASA CR-15987) has been completed. Table 1.2-I shows a revision to the in-service performance losses presented in Semiannual Report No. 5. The E^3 deterioration model was derived by developing a CF6-50C deterioration model based on the hardware evaluation as presented in Section 4.0 of the CF6-50 final report. This model defined efficiency and leakage losses for a 6000-hour engine to produce a 3.26% ΔSFC loss on the E^3 CF6-50C reference cycle deck. It should be noted that the CF6-50 report shows a 6000-hour fan but that the other modules are shown for 4000 hours. It is assumed the same losses would exist for an extrapolation to a 6000-hour time period.

The loss mechanism of each CF6-50C module was evaluated relative to the E^3 design to forecast an improvement in the various loss items. The CF6-50C model was then adjusted for reduced performance losses and evaluated on the E^3 FPS-4 cycle and found to show that a 51% reduction of the 3.2% ΔSFC loss was achieved. Table 1.2-II shows a comparison of the two models and the percent ΔSFC results from the respective cycle decks.

The results of this investigation indicated that a more detailed evaluation of the CF6-50C deterioration mechanisms should be made to ascertain the validity of the E^3 forecast deterioration model.

Update of The ICLS Test Vehicle Cycle

The ICLS test vehicle cycle has been updated to incorporate a 4% increase in the HP turbine nozzle area (A4). The HP efficiency was also increased from the ICLS Forecast level of 91.9% to the 92.4% level demonstrated in the component test. The FPS-4 cycle definition is shown in Table 1.2-III and is the same as reported in Semiannual No. 5.

Table 1.2-IV shows a comparison of the FPS-4, the ICLS-4 forecast, and the new ICLS-4F test vehicle cycles for a 35,000 feet/0.8 Mach/+18° F day flight condition. A copmparison with the table in Semiannual No. 5, shows

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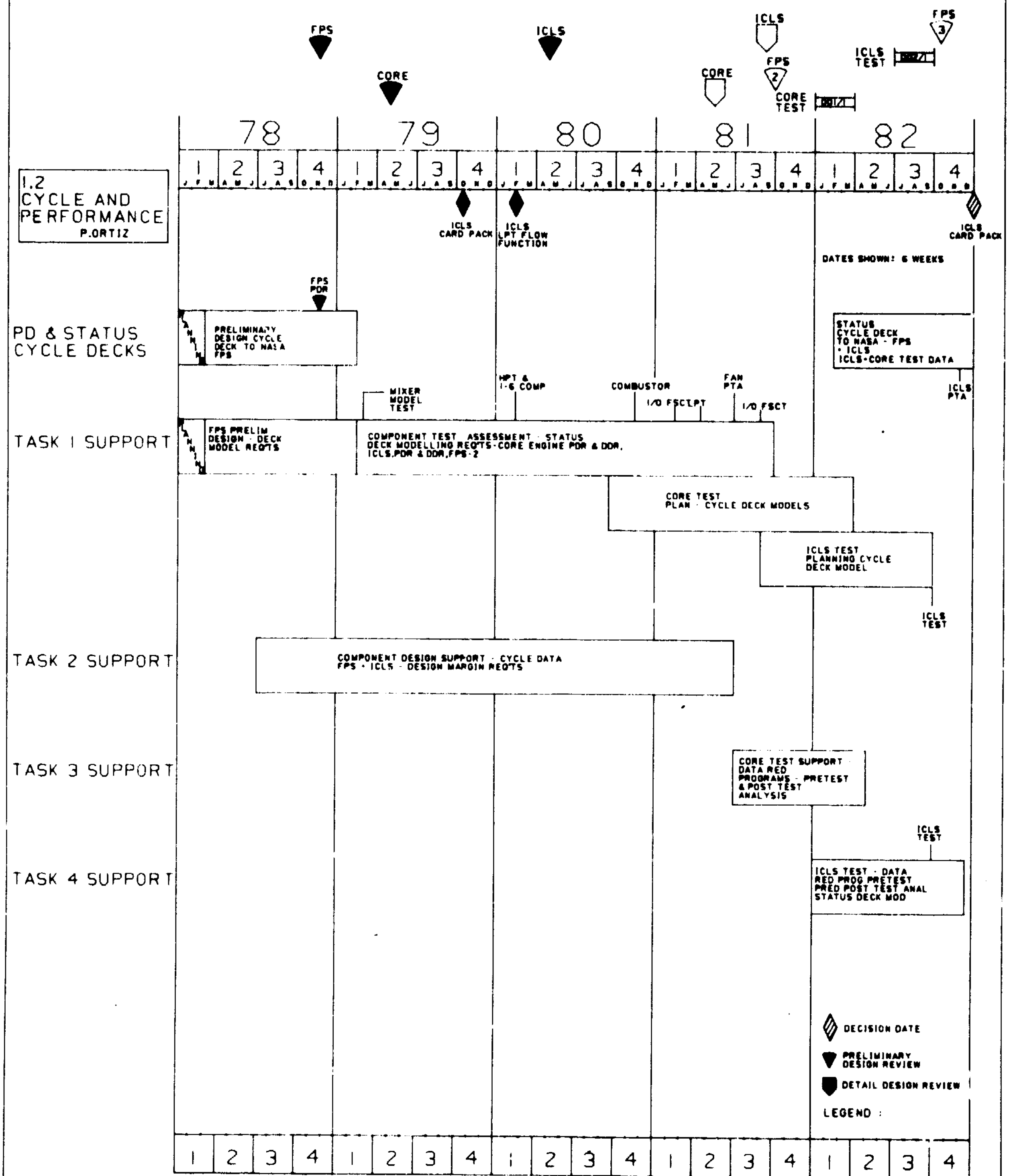

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Table 1.2-I. CF6-50C In-Service Performance Losses.

Cause	Estimated Losses, % Preliminary	Final Report
Clearances		
Fan + Booster	3	12.6
HPC	13	10.1
HPT	29	16.9
LPT	4	7.4
	49	47.0
Leakages		
HPC	10	5.8
HPT	6	3.1
LPT	3	5.2
	19	14.1
Erosion		
Fan + Booster	7	24.2
HPC	8	7.0
HPT	2	3.1
LPT	5	0.9
	22	35.2
Miscellaneous	10	3.7
Total, %	100	100

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Table 1.2-II. Table Comparison of CF6-50C and E^3 Deterioration Models.

Component	CF6-50C	E3 FPS	% Reduction
Fan - η	-2.45	-1.60	-35
Booster - η	-0.16	-0.10	-38
Compressor - η	-0.92	-0.75	-19
HPT - η	-1.01	-0.35	-65
LPT - η	-0.60	-0.40	-33
Leakage - % of W25	-0.135	-0.05	-63
% ΔSFC Loss	-3.26	-1.59	-51

Table 1.2-III. FPS-4 Performance Parameters.

Parameter	Maximum Climb 35,000 feet/ 0.8 M/+18° F	Maximum Cruise 35,000 feet/ 0.8 M/+18° F	Takeoff SLS/+27° F
Net Thrust, lb	9040	8425	36500
SFC (Standard Day)	0.546	0.542	0.294
Bypass Ratio	6.77	6.95	7.34
Overall Pressure Ratio	37.7	35.8	29.7
Fan Bypass Pressure Ratio	1.65	1.61	1.50
Compressor Corrected Airflow, lb/sec	120.0	118.0	108.9
Compressor Pressure Ratio	23.0	22.4	20.0
HPT Rotor Inlet Temperature, ° F	2345	2277	2450

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Table 1.2-IV. FPS and ICLS Maximum Cruise Cycle Comparison.

35,000 Feet/0.8 Mach/Standard Day + 18° F

Parameter	FPS-4	ICLS-4 Forecast	ICLS-4 F* Test Vehicle
Uninstalled Performance			
Net Thrust, lb	8425	8425	8425
SFC (Standard Day)	0.542	0.554	0.555
Bypass Ratio	6.95	6.93	6.90
Fan			
Corrected Flow, lb/sec	1396	1391	1389
Bypass Pressure Ratio	1.61	1.61	1.60
Hub Pressure Ratio	1.63	1.62	1.62
Compressor			
Corrected Flow, lb/sec	118.0	118.0	118.8
Pressure Ratio	22.4	22.4	21.9
HP Turbine			
Flow Function, $W\sqrt{T}/P$	17.6	17.6	18.2
Rotor Inlet Temperature, ° F	2777	2329	2319
LP Turbine			
Flow Function, $W\sqrt{T}/P$	80.3 (Base)	82.3(+2.5%)	83.6 (+4%)
Nozzle Inlet Temperature, ° F	1438	1470	1466
*No Instrumentation Losses			

the SFC increased from 0.554 to 0.555 and that compressor operating point changed from a 22.6 pressure ratio and 119.2 lb/sec flow to 21.9 and 118.8, respectively. Opening A4 decreases the HPT rotor inlet temperature, increases the LPT inlet temperature slightly, and results in a rematch of the fan operating point to a slightly lower pressure ratio and flow.

Cycle Deck Logic Modifications

Progress has been made in the planned modifications identified in Semiannual Report No. 5. These were

- Active Clearance Control System (ACC)
- Centervent Flow Model
- Reverse Thrust Mode of Operation
- Cooling Flow Source and Sink Logic
- Revised Mixer Logic.

The ACC logic effort is on hold but will be resumed during the next reporting period. The remaining four items have been completed.

The centervent logic is employed to account for a substantial amount of air which eventually exhausts at the core nozzle and can be worth about 0.35% in maximum cruise SFC with full fan flow modulation for the AAC system.

Figure 1.2-1 shows a schematic of the centervent thrust model. Each flow segment has an associated pressure loss to match the conditions at the vent exit plane. It is planned to substantiate these losses during the ICLS test and use them in the flow prediction at the cruise condition.

Core Test Planning

Support is continuing for the core engine test plan. The core engine test will be conducted in a cell which has a venturi flow measurement system.

A new logic feature which was added to the cycle deck permits running the deck as a core engine only model. This capability was developed as a result of converting the original cycle deck system to the Executive Cycle Deck system.

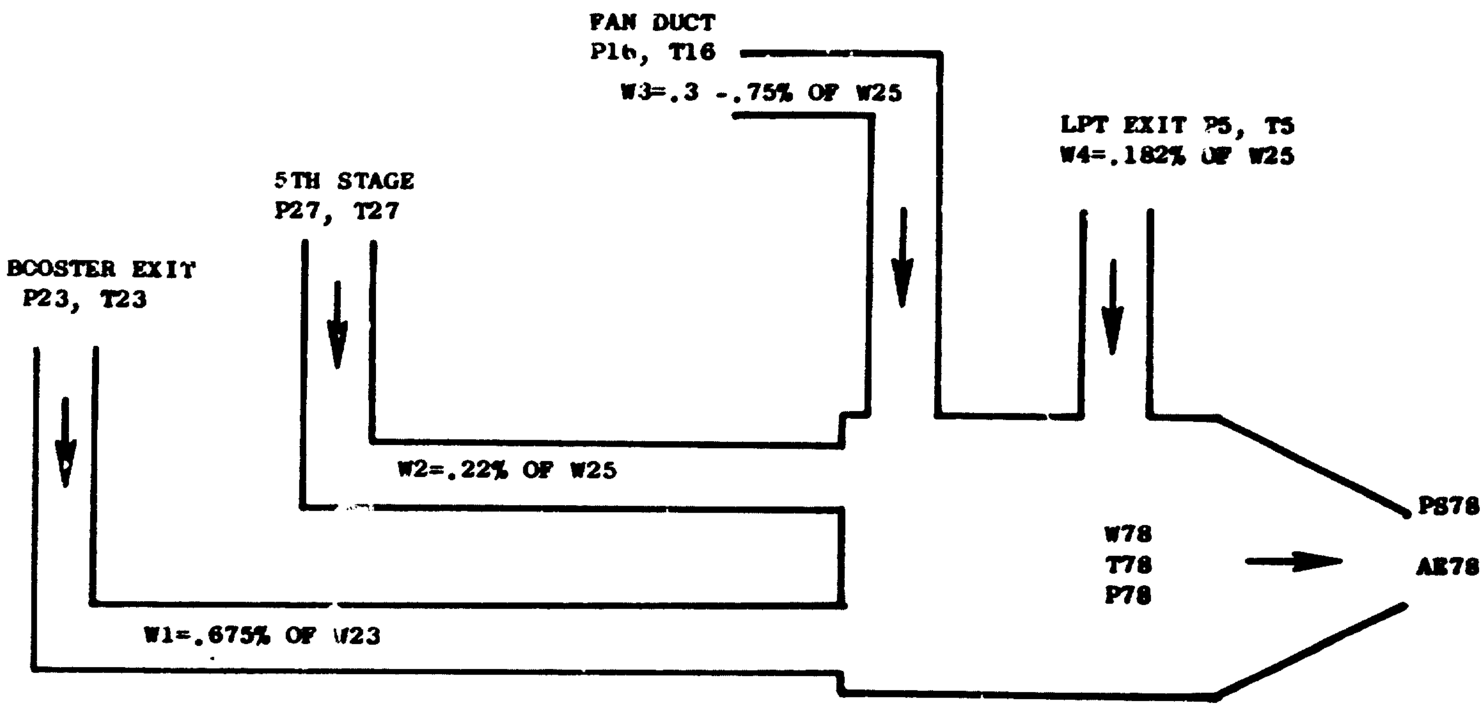


Figure 1.2-1. Centervent Thrust Model.

The core deck model was checked out and used to generate a package of core engine test prediction data which includes instrumentation losses and the HPT efficiency and A4 changes. As a result of the updated data run, the nominal nozzle area increased from 157 to 172 in^2.

Work Planned

- Continue investigation of engine deterioration mechanism
- Continue support of core and ICLS test planning
- Resume work on the cycle deck ACC system model.

1.3 MATERIALS AND PROCESSES

1.3.1 Materials and Processes Component Support

Technical Progress

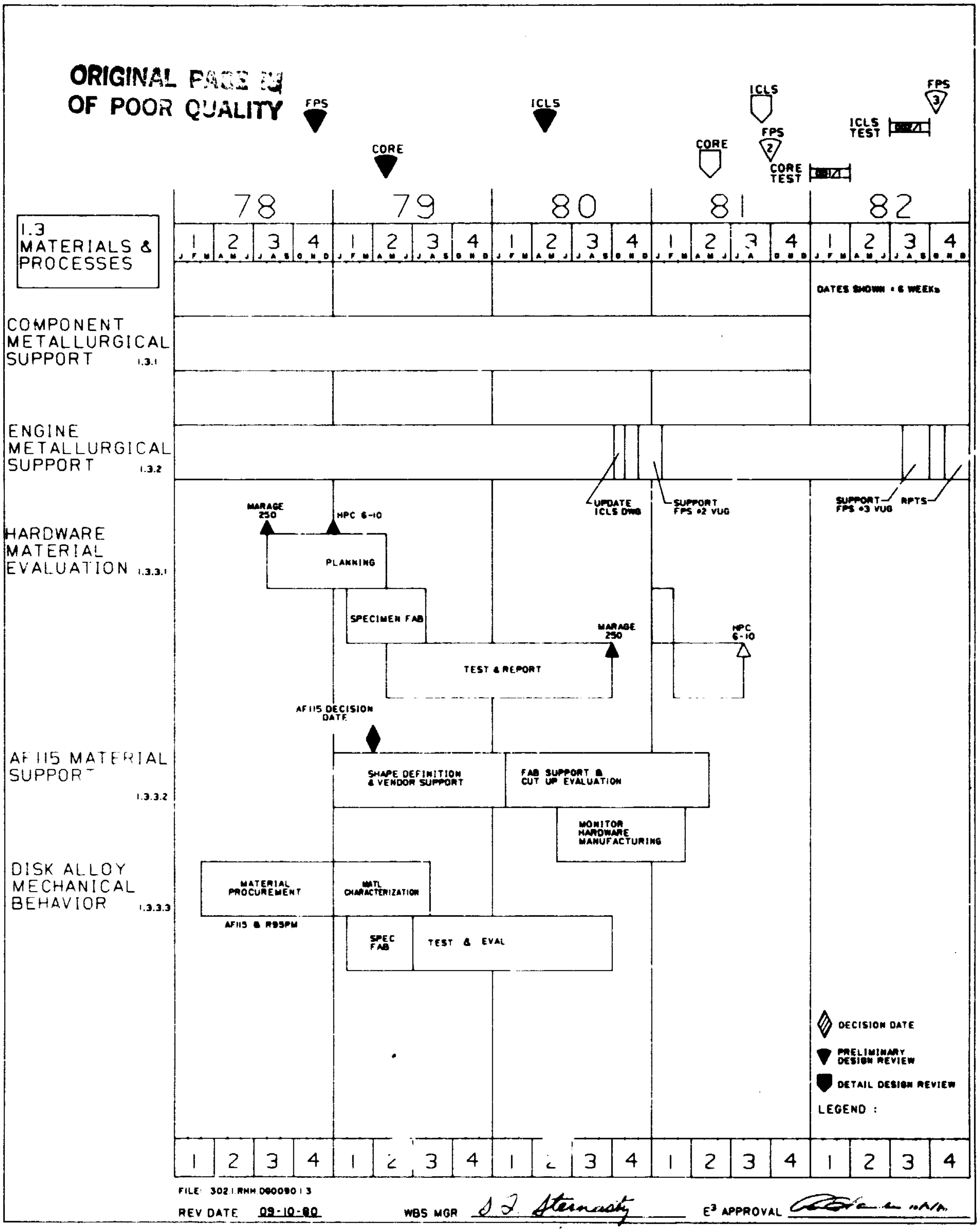
The metallurgical support activities for each component will be described individually. The activities consisted primaril of definition of materials and processing details for drawings, monitoring fabrication processes, and material property evaluation.

HP Compressor 1-10 Rig Test

Technical support and coordination was provided in the fabrication and instrumentation of hardware for this rig test. The vehicle went to test in March.

HP Compressor

Four as-HIP PM René 95 logs were ordered for this program for the 6-10 spools. Mechanical property tests have been completed on test rings from S/N 1, 2, and 3; test results are satisfactory. A test ring is being cut from log S/N 4. Log S/N 1 is in the HP Compressor 1-10 rig test; S/N 2 will go in the core, and later, the ICLS. S/N 3 is a backup.


1.3
MATERIALS & PROCESSES
78
79
80
81
82
COMPONENT METALLURGICAL SUPPORT 1.3.1
ENGINE METALLURGICAL SUPPORT 1.3.2
HARDWARE MATERIAL EVALUATION 1.3.3.1
AF 115 MATERIAL SUPPORT 1.3.3.2
DISK ALLOY MECHANICAL BEHAVIOR 1.3.3.3
FPS
CORE
ICLS
ICLS TEST
CORE TEST
DATES SHOWN + 6 WEEKS
UPDATE ICLS DWG
SUPPORT FPS #2 VUG
SUPPORT FPS #3 VUG
RPTS
MARAGE 250
HPC 6-10
PLANNING
SPECIMEN FAB
TEST & REPORT
AF 115 DECISION DATE
SHAPE DEFINITION & VENDOR SUPPORT
FAB SUPPORT & CUT UP EVALUATION
MONITOR HARDWARE MANUFACTURING
MATERIAL PROCUREMENT
MATL CHARACTERIZATION
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Vendor test results on Ti-17 hardware for the 1-4 spool were completed; they were satisfactory. GE will conduct fracture toughness (K_{IC}) tests in-house on test ring material from these Ti-17 forgings.

Full Scale Fan

Manufacture of the fan frame has been completed. Considerable support was required in the evaluation, repair, and rework of welds. Other specific items of support included evaluation of the anodized coating on the OGV's, and flame spray repair of a mismachined flange on the slave midcase.

Combustor Test Rig

Metallurgical support was required in the evaluation of casting porosity in some austenitic stainless steel castings; no other support was required.

LP Air Turbine

The first, second, fourth, and fifth stage nozzles have been fabricated; Stage 3 is still in process. The manufacture of this hardware has required considerable metallurgical support in establishing and monitoring the precise braze cycles and techniques, and processing sequences.

Work Planned

Support on component activities will continue in the form of material and process definitions for drawings and hardware modifications, monitoring fabrication processes, and evaluation of hardware from component tests.

1.3.2 Materials and Processes Engine Support

Technical Progress

This section describes the technical progress accomplished in support of the Core, ICLS, FPS, and FPS growth engines. The major items of technical support are discussed:

Material Selection

There were no changes in materials for the FPS during this reporting period.

Support Activities

Materials and process drawing definitions continued to be provided for the Core, ICLS, FPS, and FPS growth engines.

Numerous detail drawings were issued including ducts, the diffuser, Stage 2 HPT nozzle, the turbine casing, Stage 2 HPT shrouds, exhaust system components, configurations, and the LPT shrouds, nozzles, blades, disks, and rotor seals.

Coordination was provided on the supporting technology programs. Progress reviews were held in specific areas as follows:

WBS 1.3.3.1 Hardware Material Evaluation

- Torsional LCF data
- Plan for evaluation of HPC rotor

WBS 1.3.3.2 AF115 Support

- Test ring results
- Quench cracking problem, including corrective action

WBS 1.3.3.3 Disk Alloy Mechanical Behavior

- All data generated was reviewed prior to impending publication

WBS 2.2.7.1 VSV Bushing

- Status of endurance testing of bushings

WBS 2.4.7.1.1 Ceramic Shroud Process

- Basis for change from wire mesh attachment to super peg attachment

WBS 2.4.7.2.1 Thermal Barrier Process

- Results of cascade rig and engine tests of TBC hardware

WBS 2.4.7.3.1 Alloy Mechanical Behavior

- Specimen status and test plan for evaluation of coated DS René 150.

Details of the above programs are in the appropriate WBS sections.

Work Planned

Engine support activities for the next 6 months will include material and process drawing definitions, hardware problems, and monitoring fabrication processes.

1.3.3.1 Hardware Material Evaluation

Technical Progress

The purpose of this program is to perform an evaluation of material to be used in the E^3. Components to be included within this program are the Flight Propulsion System (FPS) midfan shaft and the high pressure compressor (HPC). Details of the work conducted in each of these subprograms are given within the following paragraphs.

Midfan Shaft

The FPS midfan shaft is to be manufactured from Marage 250. Additional torsional-low-cycle-fatigue (TLCF) and high-cycle-fatigue data was generated to verify estimated property curves provided to design. The results of these experiments have been presented in previous semiannual reports. Four additional hollow-bar TLCF experiments were to be conducted during the present reporting period to evaluate the effects of the ratio of maximum to minimum diameter. Due to problems encountered in the testing of these four specimens, which were attributed to the electronics and load cell of the test system, it was decided not to use the results of the tests. However, a TLCF design data program for Marage 250 currently being conducted for another engine will supply the required information for E^3.

High Pressure Compressor

The HPC for the E^3 is to be manufactured from -60 mesh René 95. Because of the significant size of this as-HIP'd component, the tensile, stress rupture, low-cycle-fatigue, and cyclic crack growth experiments will be conducted on specimens machined from various locations of the material. Table 1.3-I summarizes the tests to be conducted along with specimen location.

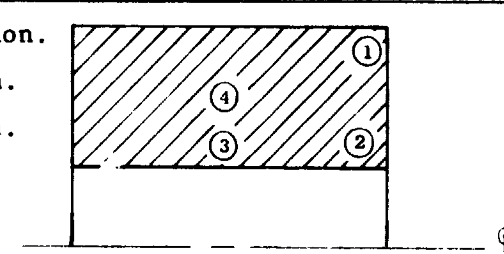
Work Planned

Conduct HPC experimental program.

Table 1.3-I. HPC Experimental Material Test Program.

Test Type	Temperature, ° F	Direction, Degrees	Number of Tests At Each Location*				Total Number Of Tests
			1	2	3	4	
Tensile	RT	0 and 180	2	2	2	2	16
	1200	0 and 180	1	1	1	1	8
Stress Rupture	1200	0 and 180	2	-	-	2	8
LCF	750	0			6		6
	1000	0	6		3	6	15
CCGR	750	0			6		6
	1000	0	6		3	6	15
							74

*Location 1 represents outer corner position.
Location 2 represents outer bore position.
Location 3 represents inner bore position.
Location 4 represents middle position.



1.3.3.2 AF115 Support

Technical Progress

The AF115 compacts from Special Metals have all been delivered with the exception of the blade retainer compact which did not meet dimensional requirements. This part is being remade but will be replaced for the initial engine tests by an AF115 ring cut from the rim of a disk obtained from another program. The Special Metals compacts all exceeded the specification strength requirements, except for low yield strength and stress rupture lives in the thick bore section of the forward seal disk. It appears that our preliminary specification values were too ambitious for this 4.5-inch thick heat treat section. This hardware will be accepted since the yield strength and stress rupture capability in this location does not limit the part.

The Crucible compacts have all been heat treated and are ready to ship pending test results. Several quench cracks were discovered in the Crucible compacts, however, all the cracks cleaned up in machining, leaving sufficient stock to machine the hardware. The quench cracking, however, did point out the need to lessen the severity of the quench at Crucible on future hardware.

LCF testing was performed on QC compacts of the powder blend used to produce the AF115 hardware at Crucible. The data is tabulated in Table 1.3-II. The data exceeds typical release requirements for René 95 and is a significant improvement over the previous material evaluated in this program. All the LCF bars, however, did initiate at Hafnium containing powder related defects, once again indicating the need to further improve the cleanliness of melting/atomization procedures for Hafnium bearing alloys.

Work Planned

Review and analyze the property data from the Crucible AF115 test rings. Begin LCF testing of Special Metals AF115 material.

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Table 1.3-II. AF115 Low Cycle Fatigue Data.

1000° F, Strain Control, A=1, 20 CPM				
Specification	$\Delta\epsilon_T$, %	E, 10^6 psi	Pseudo σ/Alt., ksi	N_F
1-1	0.66	27.8	91.7	24,984
1-2	0.78	27.8	108.4	7,214
1-3	0.90	28.0	126.0	6,292
1-4	0.66	27.9	92.1	37,090
1-5	0.60	27.8	83.4	51,976
2-1	0.66	28.0	92.4	24,572
2-2	0.78	27.9	108.8	11,340
2-3	0.60	27.9	83.7	28,602
2-4	0.66	27.9	91.7	23,688
2-5	0.60	28.2	84.6	36,930

1.3.3.3 Disk Alloy Mechanical Behavior

Technical Progress

Two advanced disk alloy materials, namely as-HIP René 95 and as-HIP AF115, have been selected for use in the E^3. René 95 in the as-HIP condition has been utilized in other General Electric engines; however, limited fatigue data has been developed at the temperatures of interest to this program. Furthermore, -150 mesh powder will be used in application of the E^3 for its better fatigue characteristics, while in the past the earlier engine components were manufactured from -60 mesh material. The other alloy, AF115, is a nickel-base superalloy developed by General Electric under an Air Force Material Laboratory Contract ending June 1974. This material, which has higher creep and stress rupture strengths than René 95 at temperatures of 1000° to 1500° F, has not been designated for use in any other engine other than the E^3. Thus limited data is available for either of these two alloys. The objective of this program was to produce design information and understanding of the materials behavior for a successful design of the E^3. All work within this program has been described in earlier semiannual reports.

1.4 ACOUSTIC DEVELOPMENT

1.4.1 System Acoustic Prediction

Technical Progress

The main effort during this reporting period focused on a system noise prediction update. Takeoff noise levels for the E^3 aircraft were updated through mixer scaled model test results. The results are shown in Table 1.4-I.

Work Planned

- None

Table 1.4-I. Takeoff Noise Estimate for E^3 Aircraft.

- With Mixer Benefit
- EPNdB

	Boeing Twinjet	Lockheed Trijet	Lockheed Quadjet	Douglas Trijet
FAR36	93.8	100.4	104.2	100.9
Status Level	88.8	93.6	98.3	94.4
Margin	-5.0	-6.8	-5.9	-6.5
Revised Level	88.1	93.3	98.0	94.0
Margin	-5.7	-7.1	-6.2	-6.9

1.4.3 Mixer Acoustic Testing

Technical Progress

Aeroacoustic correlation of the tested mixer configuration was completed. It predicted well in terms of sound power level, i.e., the source strength. However, the accuracy of prediction for the SPL spectra at all angles was not completely adeuqate. Figure 1.4-1 illustrates the power level comparison.

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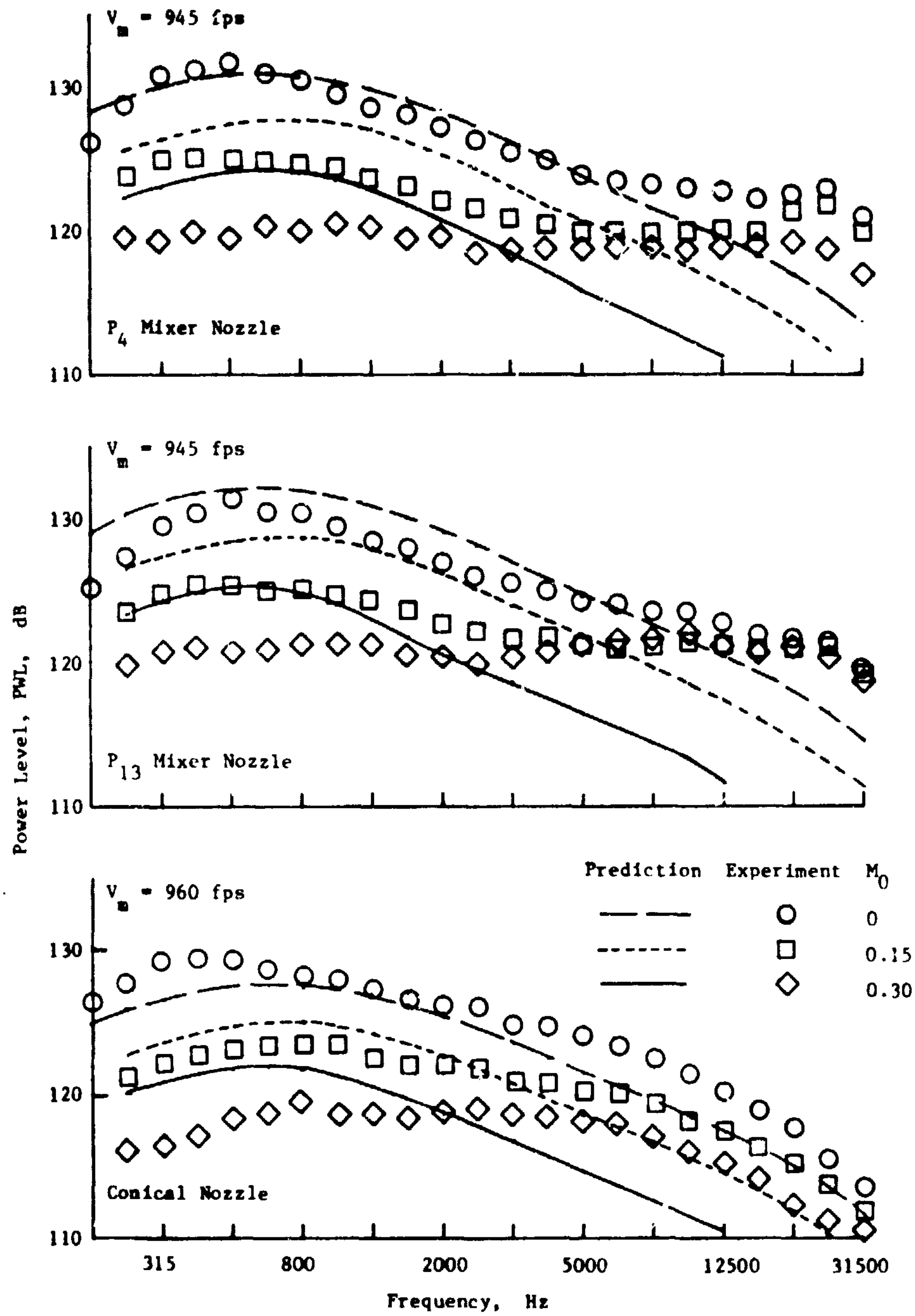

Figure 1.4-1. Comparison of Predicted and Measured PWL Spectra.

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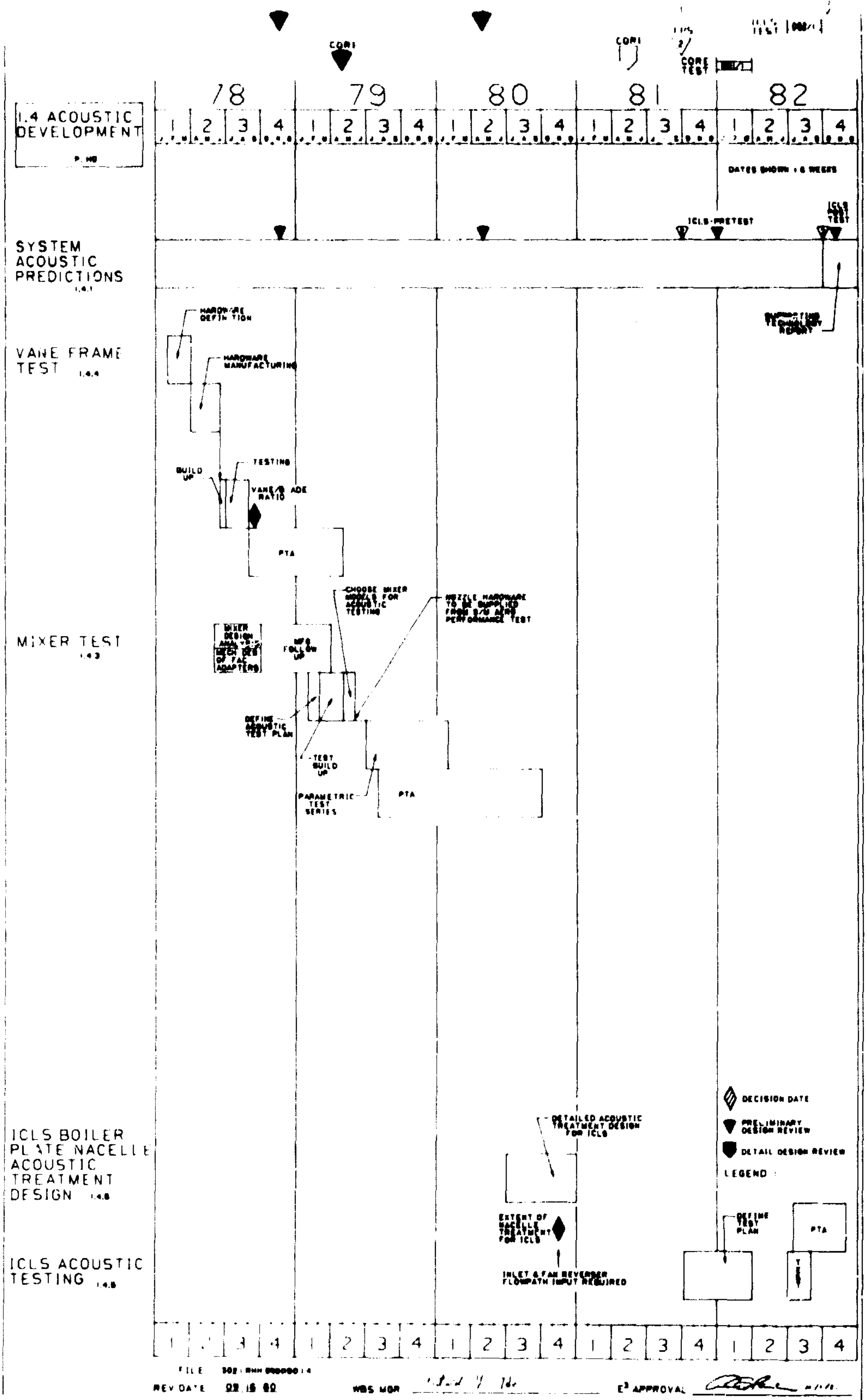

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Work Planned

- None.

1.5 PROPULSION SYSTEM/AIRCRAFT INTEGRATION

1.5.1.3 Nacelle Performance Evaluation - Langley

Technical Progress

A recalibration of the E^3 baseline and extended nacelles was conducted at Boeing at the request of NASA-Langley. It was discovered that the calibration data used for the E^3 nacelle in the Phase II test data reduction was with powered simulator (TPS) Unit No. 5 but the wind tunnel test was run with Unit No. 3. Thus a calibration with TPS No. 3 was necessary. Langley also requested a check calibration of the extended nacelle on TPS No. 3. Final calibration results were received from Boeing, and the nozzle coefficients were derived and forwarded to Langley.

Recalibration results are showing another shift in the nozzle coefficients by 1% to 2%. This was seen previously but it was believed to be due to differences in TPS units. Since the E^3 extended nacelle has been calibrated twice with the same TPS unit, and detailed comparisons of different TPS units indicate very similar operating characteristics, the shifts are apparently caused by some unknown factor from test to test. If the shifts are an inherent part of the calibration procedure, the accuracy of wind tunnel test results will be affected. Analysis of the calibration data will continue to determine a reason for the coefficient shifts.

The above work is being performed under an extended contract to NASA-Langley.

Work Planned

Analyze the Phase II wind tunnel test results if they are made available from NASA-Langley and prepare a data memo.

2.0 TASK 2 - COMPONENT ANALYSIS DESIGN AND DEVELOPMENT

2.1 FAN

Overall Objectives

The primary objective of the fan development effort is to evolve a high technology design able to meet or exceed all commercial certification requirements for noise, performance, life, and bird or ice ingestion. The general configuration of the fan (Figure 2.1-1) will incorporate a high-bypass fan stage with a part-span shroud followed by a quarter stage to provide additional supercharging to the core compressor. The design of the quarter stage, and passages aft of it, will minimize foreign object ingestion into the high-pressure compressor. The quarter stage will also provide good distortion attenuation and tolerance to bypass variation without variable geometry.

Blades will be solid titanium construction to provide the lightest weight fan capable of meeting the operational environment imposed by bird ingestion requirements. The fan will have no life-limiting conditions for the expected operation in commercial service including crosswinds, thrust reversals, and tip rubs. The FPS fan efficiency goals are 0.882 bypass and 0.892 hub at Mach 0.8, 35,000-feet altitude, standard day, maximum cruise power setting.

The fan design will be verified in a fully instrumented, full-scale fan test beginning in the second quarter of 1981. Modifications required to attain goals for performance or mechanical integrity will be identified from the test results and incorporated into the fan design prior to the ICLS test.

Preliminary mechanical design studies were conducted in order to establish the general fan configuration with the best potential for meeting the overall program objectives. In June 1979, the fan-blade parameters related to bird-ingestion tolerance were evaluated based on commercial service experience with improved blade designs now in production. The objective of this selection was to ensure that the E^3 fan blading will be compatible with commercial engine certification requirements.

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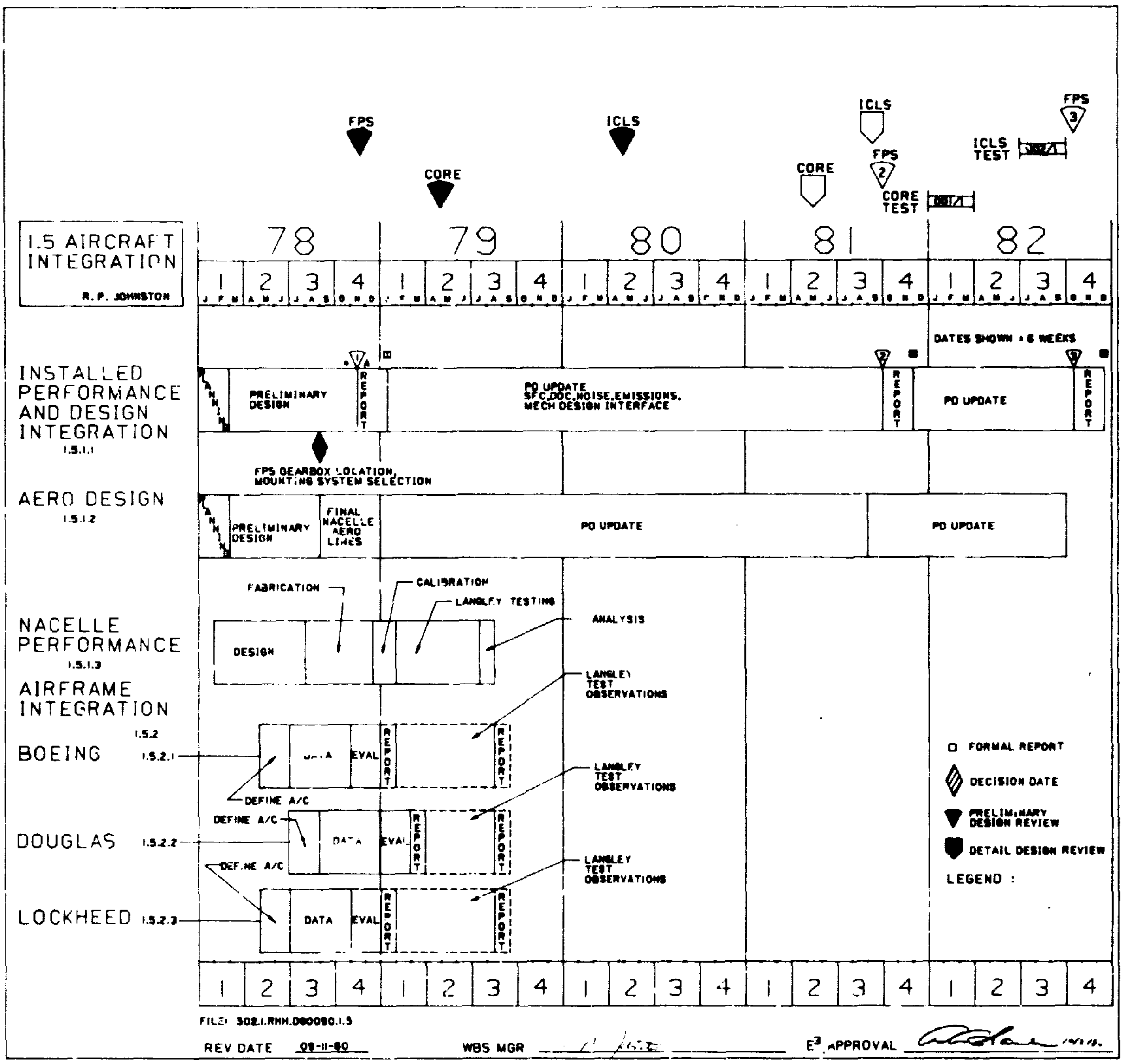

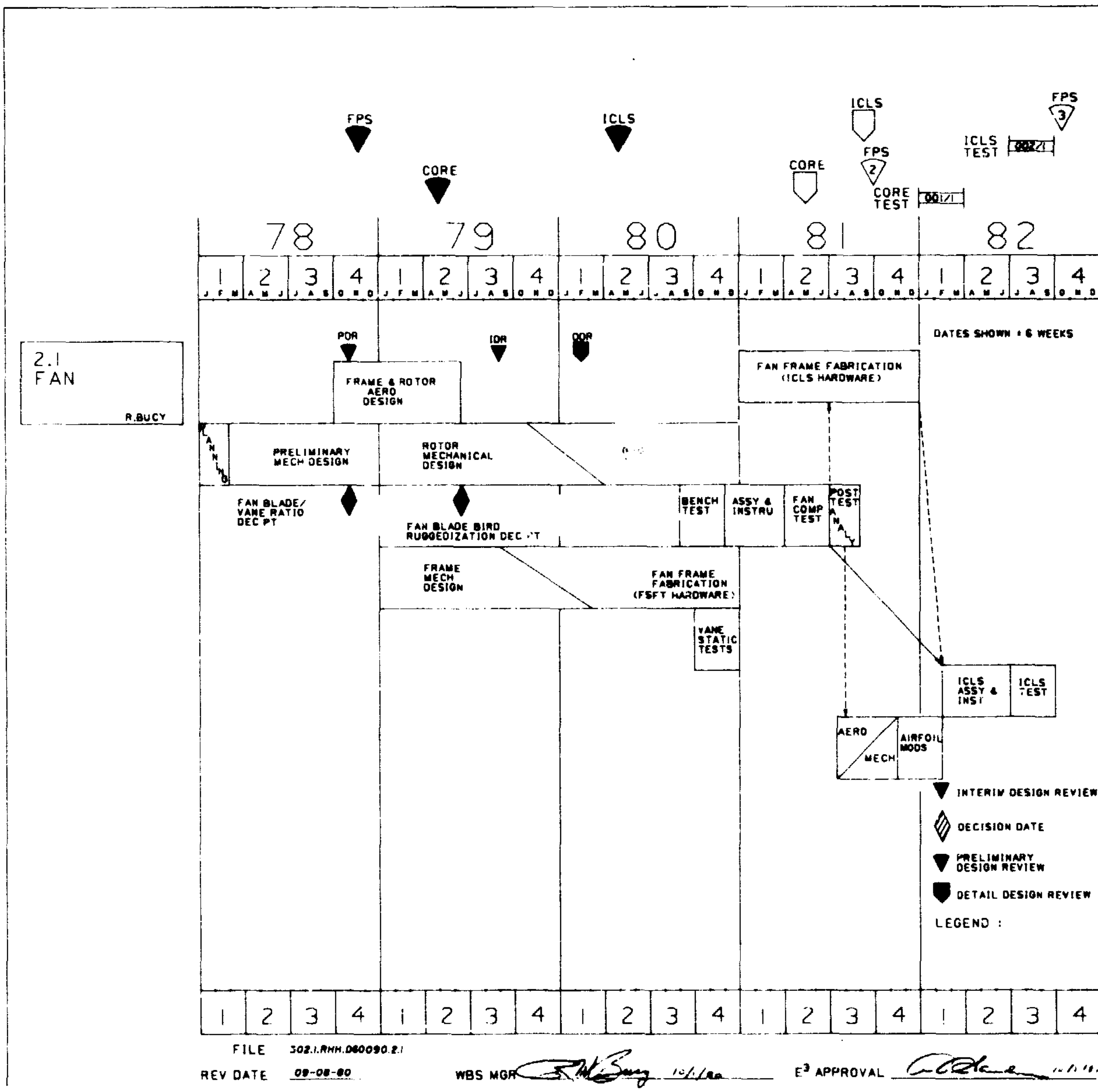

2.1
FAN
R.BUCY
FPS
CORE
ICLS
FPS
ICLS TEST
CORE TEST
78
79
80
81
82
DATES SHOWN ± 6 WEEKS
PDR
IDR
DDR
FRAME & ROTOR AERO DESIGN
FAN FRAME FABRICATION (ICLS HARDWARE)
PLANNING
PRELIMINARY MECH DESIGN
ROTOR MECHANICAL DESIGN
FAN BLADE/ VANE RATIO DEC PT
FAN BLADE BIRD RUGGEDIZATION DEC PT
BENCH TEST
ASSY & INSTRU
FAN COMP TEST
POST TEST ANAL
FRAME MECH DESIGN
FAN FRAME FABRICATION (FSFT HARDWARE)
VANE STATIC TESTS
ICLS ASSY & INST
ICLS TEST
AERO
MECH
AIRFOIL MODS
INTERIM DESIGN REVIEW
DECISION DATE
PRELIMINARY DESIGN REVIEW
DETAIL DESIGN REVIEW
LEGEND :
FILE 302.1.RHH.060090.2.1
REV DATE 09-08-80
WBS MGR
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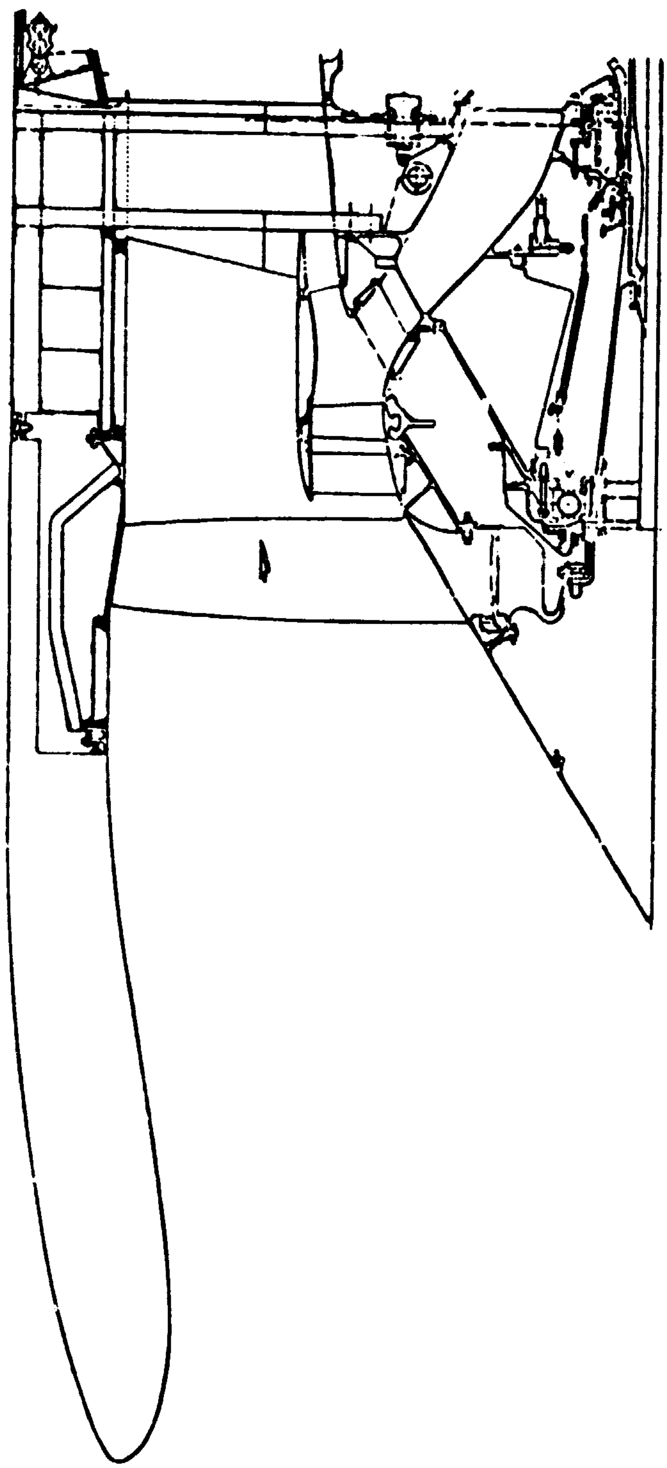

Figure 2.1-1. Fan Module Cross Section.

The final aerodynamic design of the fan rotor was completed during the third quarter of 1979. The final mechanical design was completed in the fourth quarter of 1979 with an Interim Design Review (IDR) in 1979 leading to the fan Detail Design Review (DDR) in February 1980; design parameters are shown in Table 2.1-I. Concurrent with the above effort, the final aerodynamic design of the frame structures was also completed during the last quarter of 1979. The final frame mechanical design was initiated at the beginning of 1979 and conducted during the latter part of 1979. The slave frame design was reviewed in an informal IDR on September 12, 1979 to allow an early start on the relatively long procurement cycle of the slave frame. Fabrication of the slave frame was completed during the first quarter of 1981.

Hardware of the fan rotor and the slave frame was available to initiate the component test buildup by the end of the first quarter of 1981. Following the instrumentation and buildup cycle, the fan component test will begin in the second quarter of 1981. Test data from the component test will be analyzed to incorporate expected minor adjustments into the aerodynamic or mechanical design effort during the third quarter of 1981. This refinement in the design may then be committed to a rework cycle of the Full Scale Fan Test (FSFT) hardware during the remainder of 1981, leading to its availability for the ICLS buildup beginning early in 1982.

2.1.1 Fan Aerodynamic Design

Technical Progress

A very limited amount of activity has taken place in the Semiannual reporting period in the area of fan aerodynamic design. The principal activity has been centered around planning and preparing for the fan test, now scheduled for July 1981.

During this preparation period, the instrumentation plan has been formulated and work on the data reduction procedure has been initiated. The instrumentation plan has been written and the final draft recently approved by NASA. This plan includes a description of the fan test vehicle as well as the Large Fan Test Facility (LFTF) located in Lynn, Massachusetts where the fan will be tested. The aerodynamic instrumentation is described in detail, specifying

Table 2.1-1. Fan Aerodynamic Design Parameters.

Parameters	Maximum Climb 35,000 feet/Mach	Maximum Cruise 0.8/+18° F	Takeoff SLS/+27° F
Corrected Tip Speed, ft/sec	1350	1311	1198
Corrected Airflow, lbm/sec	1419	1396	1274
Flow/Annulus Area, lbm/sec-ft^2	42.8	42.1	38.4
Bypass Stream Pressure Ratio	1.65	1.61	1.50
Bypass Stream Adiabatic Efficiency, percent	87.9	88.7	90.0
Core Stream Pressure Ratio	1.67	1.63	1.51
Core Stream Adiabatic Efficiency, percent	88.5	89.2	89.7
Bypass Ratio	6.8	6.9	7.3

the location and type of sensor, and the associated performance calculations. The vehicle and facility instrumentation are also described in the instrumentation plan. A complete list of each instrument element as well as the instrumentation drawing is provided.

The data reduction plan is currently being formulated to determine what performance calculations are required to best analyze the fan test results and how these calculations are handled in the centralized data reduction computer program. In addition to overall bypass and core stream performance, vane-mounted and traverse instrumentation will be used to measure the individual blade-row performance. Numerous wall static pressures, and boundary-layer rakes will be used to analyze the flow properties near the wall boundaries and to aid in the calculation of the flow at the bellmouth, bypass, core, and booster measuring planes. Dynamic and steady-state pressures over the fan rotor blade tip will be recorded for several data points and will be used to determine the shock structure of the flow in the rotor tip blade passage.

The fan vehicle hardware quality is being monitored by aero personnel as it is delivered for assembly. Instrumentation application to the blades and vanes is also being monitored to assure good quality. The aerodynamic requirements for the calibration of the rakes and vane-mounted sensors have been specified to include the predicted range of test operation.

Work has begun recently to prepare the fan vehicle test plan including the pretest predictions of the fan and quarter-stage.

The detailed design report of the fan test hardware was approved in its final version and issued as required by NASA Contract NAS3-20643. The report number is NASA CR-165148.

Work Planned

- The fan aerodynamic performance map, pretest predictions and test plan will be finalized 3 months prior to the scheduled fan component test.
- The data reduction plan will be formulated and coordination with the engineering and test personnel in Lynn, Massachusetts will continue.
- Test coverage, data analysis, and reporting will be provided by fan aero personnel on the fan vehicle which is scheduled to begin test in July 1981.

2.1.2 Fan Rotor Mechanical Design

2.1.2.1 Fan Rig Rotor Mechanical Design

Technical Progress

All hardware required for the Full Scale Fan Test (FSFT) vehicle buildup has been delivered.

Work Planned

- Follow vehicle buildup and testing.

2.1.2.2 ICLS Fan Rotor

The forward fan shaft detail drawing has been issued and an order for manufacture has been placed.

Work Planned

- Follow hardware procurement
- Issue the ICLS fan rotor assembly drawing.

2.1.3 Fan Stator Mechanical Design

2.1.3.1 Frame Interface Design

Technical Progress

The final interface requirements for the fan stator hardware required for the FSFT were established and all the detail drawings issued by November. The majority of the work effort involved the incorporation of the hardware required to accommodate the gearbox radial driveshaft and the accessory supply lines for the ICLS engine into the hardware configuration for the FSFT. This task required the early establishment of frame interface locations for engine hardware; the gearbox, radial driveshaft and mounting brackets, the nacelle inner and outer cowl doors, service lines, controls, etc. This was necessary to permit the design of the cut-back vanes, fairing, and extension at the 6:00 o'clock location that would interface with the test facility hardware for the FSFT and still adapt to the engine hardware for the ICLS test. Conversion from the FSFT configuration to the ICLS configuration at the gearbox location requires a different service line extension and support hardware, an oil-tube and retainer to prevent oil leakage into the bypass flow stream, and a gearbox mounting adaptor to position the radial driveshaft through the frame. The detail drawings for this hardware will be issued in April 1981.

The remaining fan stator hardware required for the ICLS engine test includes the engine mount brackets, the gearbox attachment brackets, and the nacelle inner and outer cowl door seal rings. In February 1981, a review of the gearbox mounting arrangement to the fan frame resulted in an acceptable configuration and released the hardware designs for fabrication. While the overall engine mount arrangement and the preliminary link pick-up points at the fan frame have been established, several design iterations of the slave engine pylon system have delayed the final approval of the interface between the fan frame mounting brackets and the Test Facility pylon. A final configuration for the engine mount system should be available in April 1981. The interface requirements of the inner and outer cowl door seal rings with the test facility pylon structure and the fan frame have been established and the detail should also be completed in April 1981.

During the past 6 months, engineering support has been required for the fabrication of the fan frame hub in the Development Manufacturing Operation at Evendale and for the final assembly, machining and instrumentation of the fan frame assembly.

Work Planned

- Expedite the determination of the engine mount system configuration to allow the release of ICLS hardware.

2.1.3.2 Fan Casing Design

Technical Progress

The fan casing for the full-scale fan test (FSFT) consists of a containment case and a midcase that are mechanically attached to the steel outer bypass case of the slave fan frame as shown in Figure 2.1-2. The effort during this reporting period consisted of design support for the fabrication and assembly of this hardware. The fabrication is now complete as is the initial trial assembly.

Work Planned

- The final assembly of the FSFT vehicle will be supported.

2.1.3.3 Stage 1 Vane Design

Technical Progress

The mechanical design and the fabrication of the Stage 1 vane and the vane assembly hardware has been completed.

The vanes and stator assembly hardware requiring aerodynamic and mechanical instrumentation have been c pleted and the remaining hardware inspected prior to assembly for the fan test vehicle.

2.1.3.4 Quarter-Stage Vane Design

Technical Progress

The mechanical design and fabrication of the quarter-stage vane and vane assembly hardware have been completed and the required aerodynamic and mechanical instrumentation have been installed. The stator assembly is currently being inspected prior to delivery to the fan test vehicle assembly.

2.1.3.5 Frame Mechanical Design

Technical Progress

Design work on the frame has primarily involved the hardware for ICLS. Following a review of the gearbox mounting arrangement, the design was changed to provide a more positive means of aligning the radial driveshaft and to reduce the loads carried by the stationary flexible oil tube that encapsulates the radial driveshaft. Figures 2.1-3 and 2.1-4 illustrate the current arrangement at the gearbox interface with the fan frame. An adaptor bracket

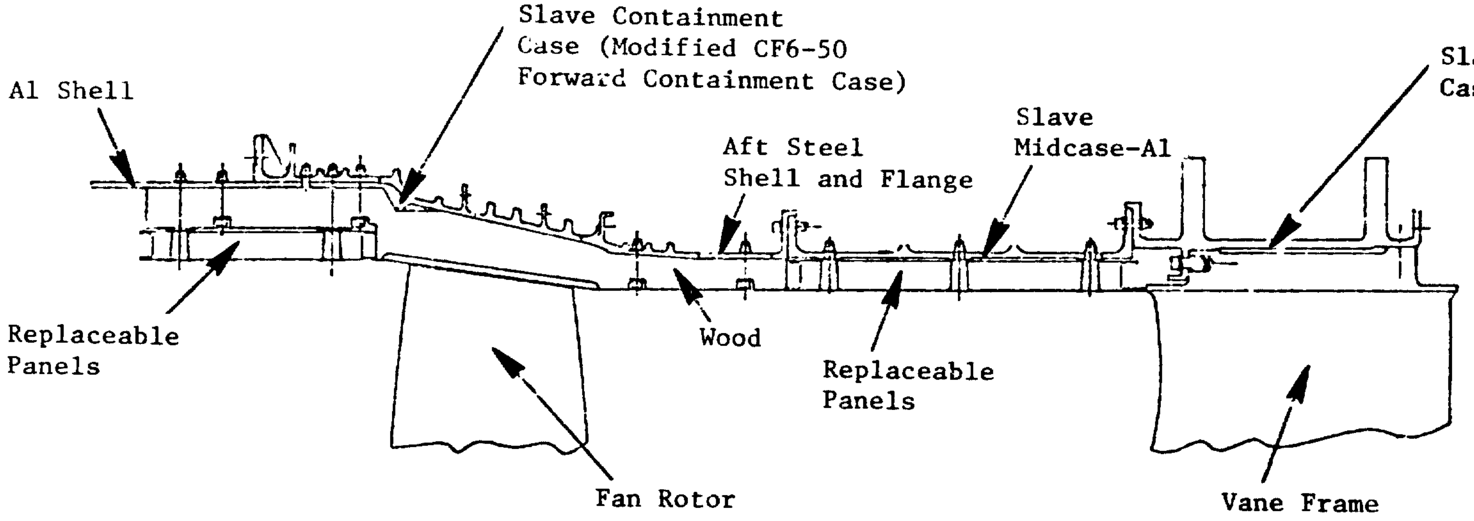

Figure 2.1-2. Fan Casings FSFT and ICLS.

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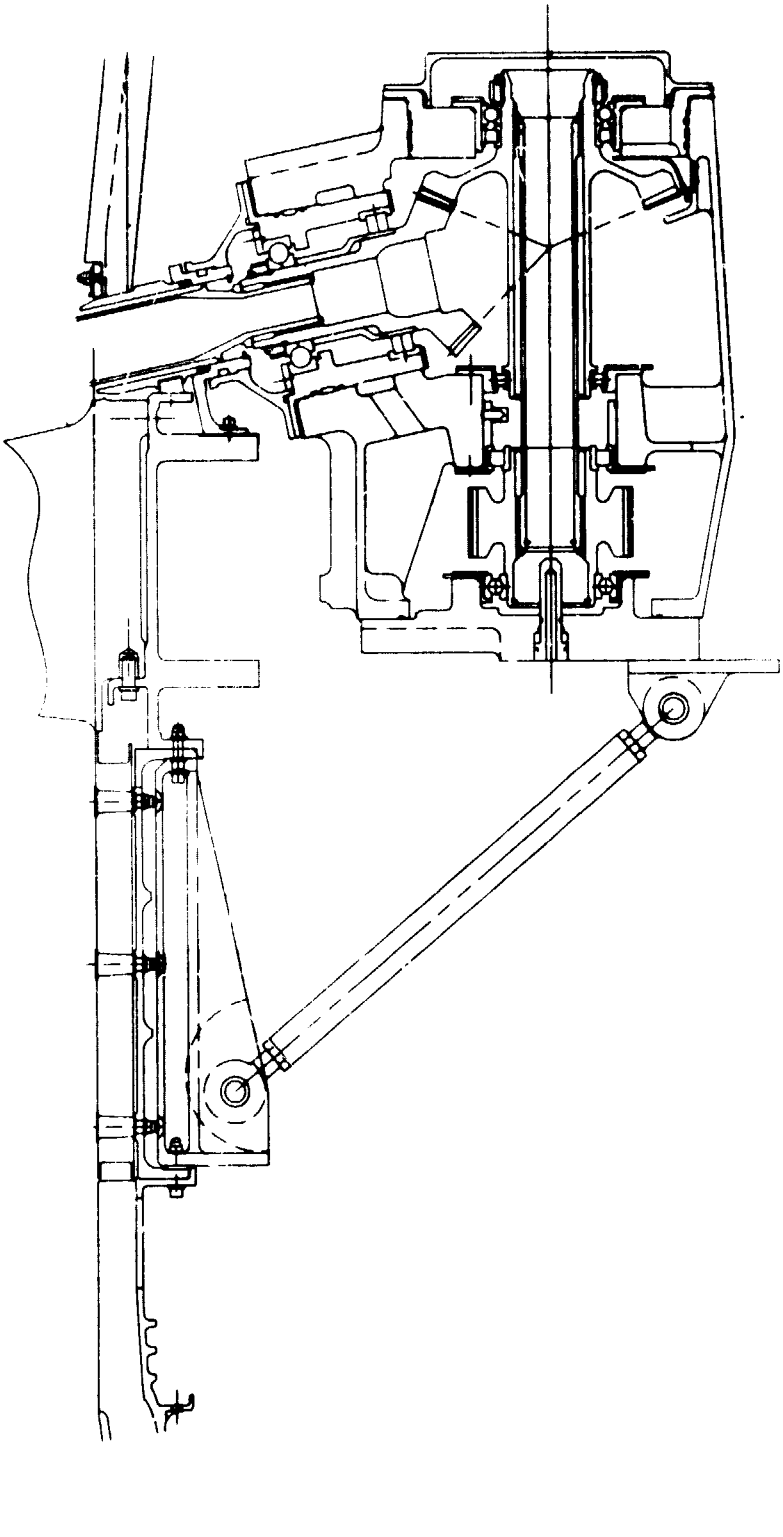

Figure 2.1-3. Gearbox Mounting Arrangement.

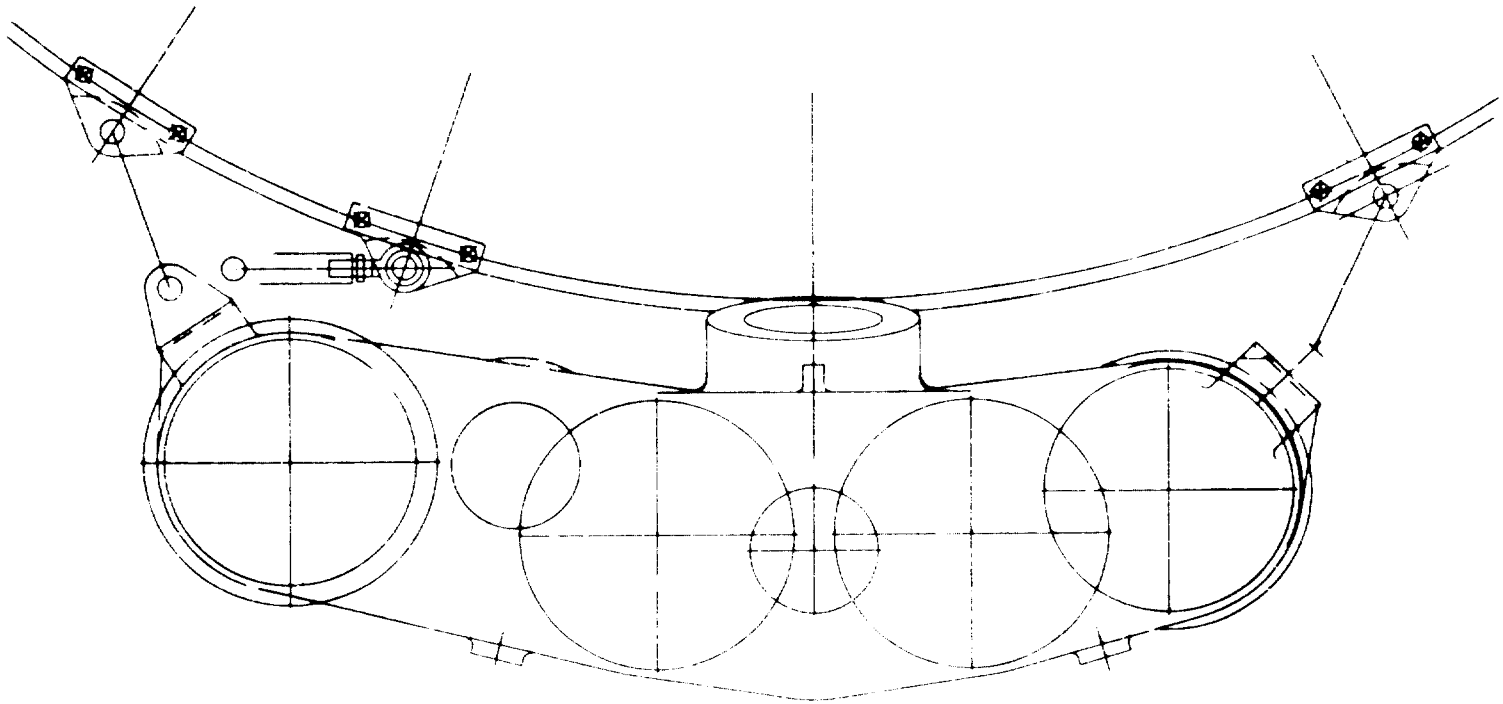

Figure 2.1-4. Gearbox Mounting Arrangement.

which bolts to the fan bypass case will provide a pilot attachment to the gearbox which establishes the positive alignment of the radial driveshaft to the frame. The adaptor bracket will be aligned to the frame by laser optics through three points; the driveshaft centerline at the hub, the midspan bearing bore at the bypass vane ID, and the pilot bore at the adaptor bracket. The gearbox is supported off the frame by two vertical link brackets at approximately $\pm 30°$ from bottom vertical, one tangential link bracket, and a bracket and link to the midcase which provides fore and aft stability to the gearbox.

Figure 2.1-5 illustrates the service line extension for the ICLS configuration. This extension is similar in size to the extension required for the FSFT but must satisfy different operating requirements. The extension for the FSFT is fixed to the facility hardware at the ID and OD and is slip jointed into the vane fairing. The thermal and pressure gradients that will be encountered during the FSFT are minimal. Consequently, one stiffener will be employed to prevent any stress, deflection, or panel frequency problems. For the ICLS engine test, where thermal and pressure gradients across the extension can become significant, stiffeners will be employed at three axial locations.

The forward engine mount attachment to the fan frame is illustrated in Figure 2.1-6. The mount brackets are located symmetrically about the top vertical centerline at $\pm 45°$ on the aft face of the fan frame hub. Each bracket connects to the Test Facility pylon system through a vertical link and an axial thrust link. Vertical, side, and thrust loads are reacted at the forward mount. The design configuration for the brackets is complete. The detail drawing can be completed once the link connecting locations currently under study by the Test Facility Engineering group are finalized.

The cowl door seal ring designs are shown in Figure 2.1-7. The outer seal ring is segmented into two approximately 180° sectors and bolts to the fan frame bypass case. The ring provides a radial seal and a bumper support for the forward end of the outer nacelle door. Additionally, the bracket that supports the service line extension at the bottom bypass vane location bolts to the seal ring. The inner seal ring bolts to the bypass vane aft

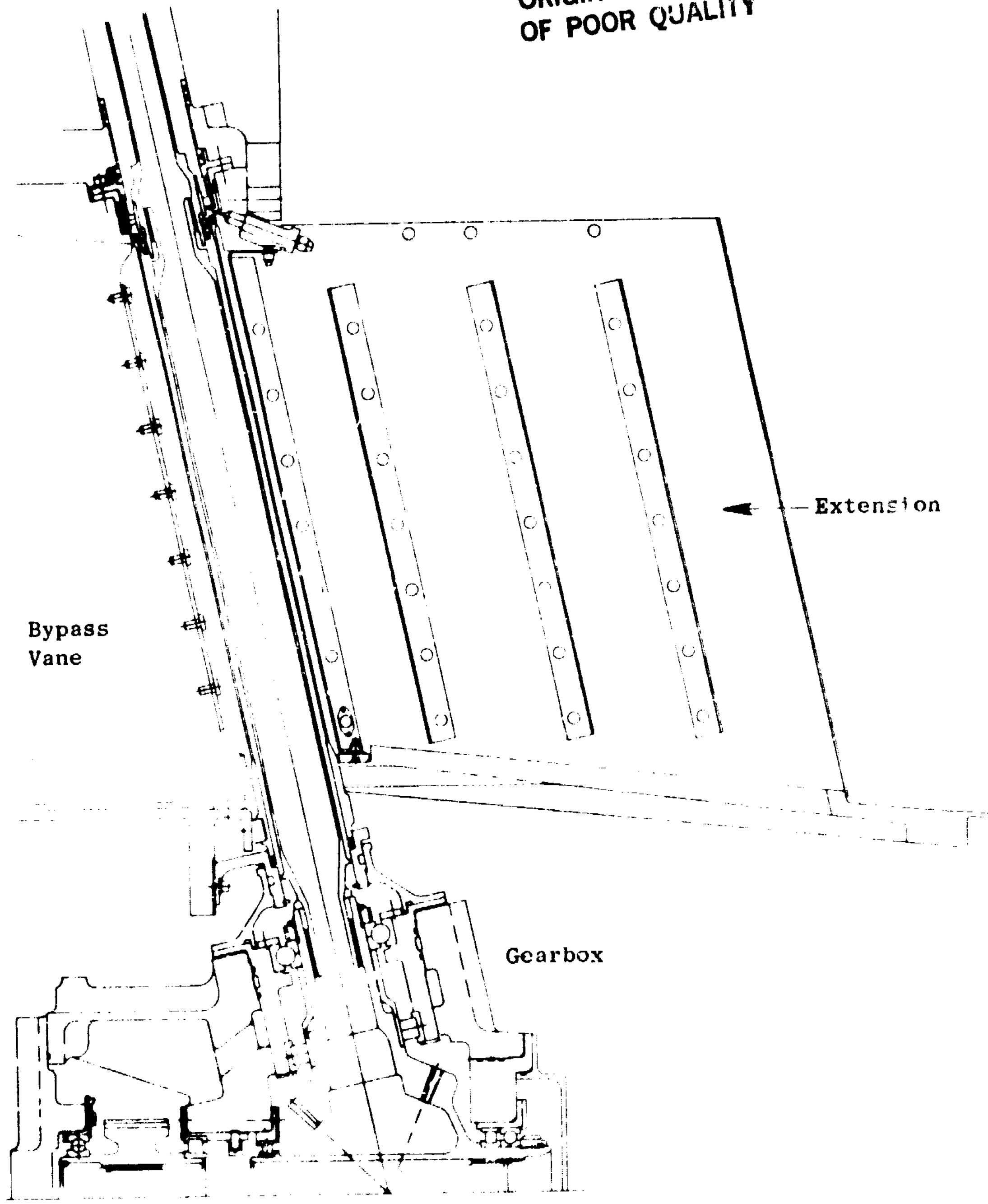

Figure 2.1-5. ICLS Service Line Extension.

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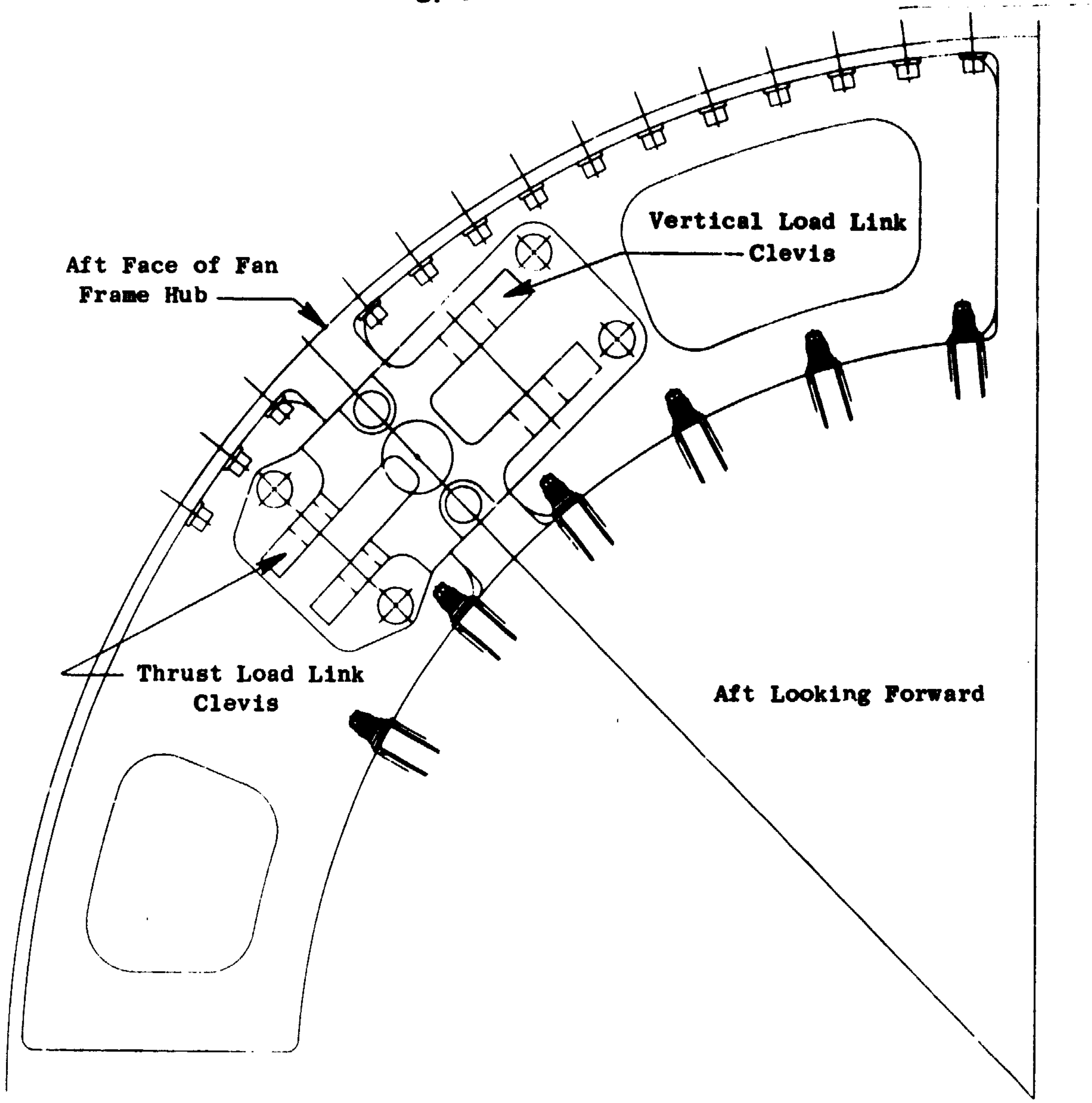

Figure 2.1-6. Forward Engine Mount Bracket.

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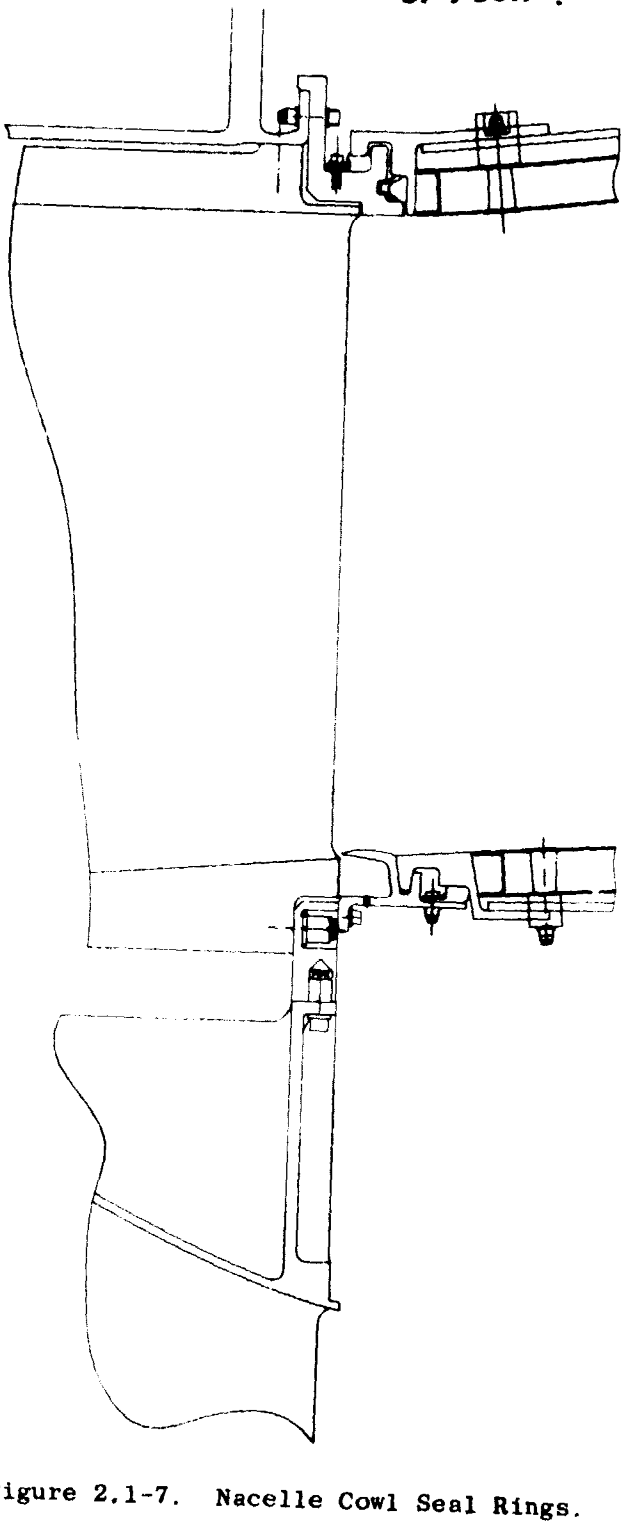

Figure 2.1-7. Nacelle Cowl Seal Rings.

inner support ring and supports the forward end of the nacelle inner cowl door. The seal is designed for a tongue and groove attachment of the cowl door with a bulb seal provided to minimize leakage. The seal is also designed to collect the cowl purge air required for fire safety. The flowpath portion of the seal is set radially outward above the bypass vane inner flowpath. This annulus collects the air which is then ported through holes in the seal to the core cowl cavity. The adjustment of the seal flowpath for the purge air can be seen in Figure 2.1-7.

Work Planned

- Complete design of forward engine mount hardware
- Support fan module assembly and test.

2.1.4 Fan Rotor Design Testing

Bench test of the Stage 1 and Stage 2 blades has been completed. The tests included determining resonant frequencies, nodal patterns, stress distribution, and end effects.

2.1.5 Fan Stator Design Testing

Technical Progress

The major portion of the fan stator bench testing has been completed. Data for vane vibratory frequencies, corresponding nodal line diagrams, and relative strain distributions has been determined for the bypass vane stage including the cutback vanes, the Stage 1 vanes, and the core OGV's. In addition, unit static load tests were conducted on the Stage 1 vane and the core OGV. Based on these data, strain gage locations were established for safety monitoring the vanes during the fan component test and the engine test. A check of the engine gages is planned during the vehicle assembly to verify vane frequencies and gage response. Figures 2.1-8 and 2.1-9 illustrate a bypass vane and a Stage 1 vane setup during bench testing.

Work Planned

- Conduct final strain gage check on vanes during assembly.

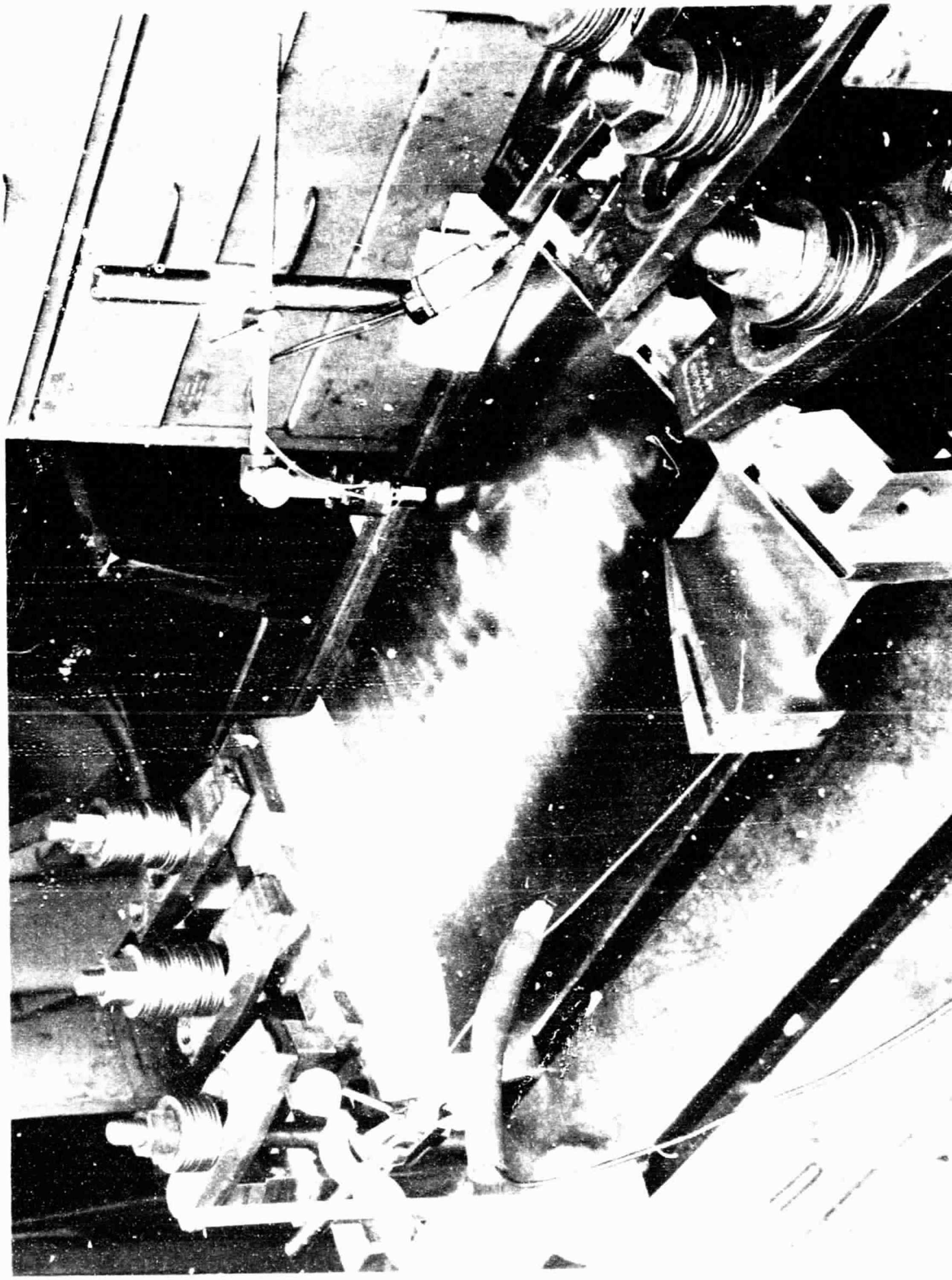

Figure 2.1-8. Bypass Vane: Bench Test Set-Up.

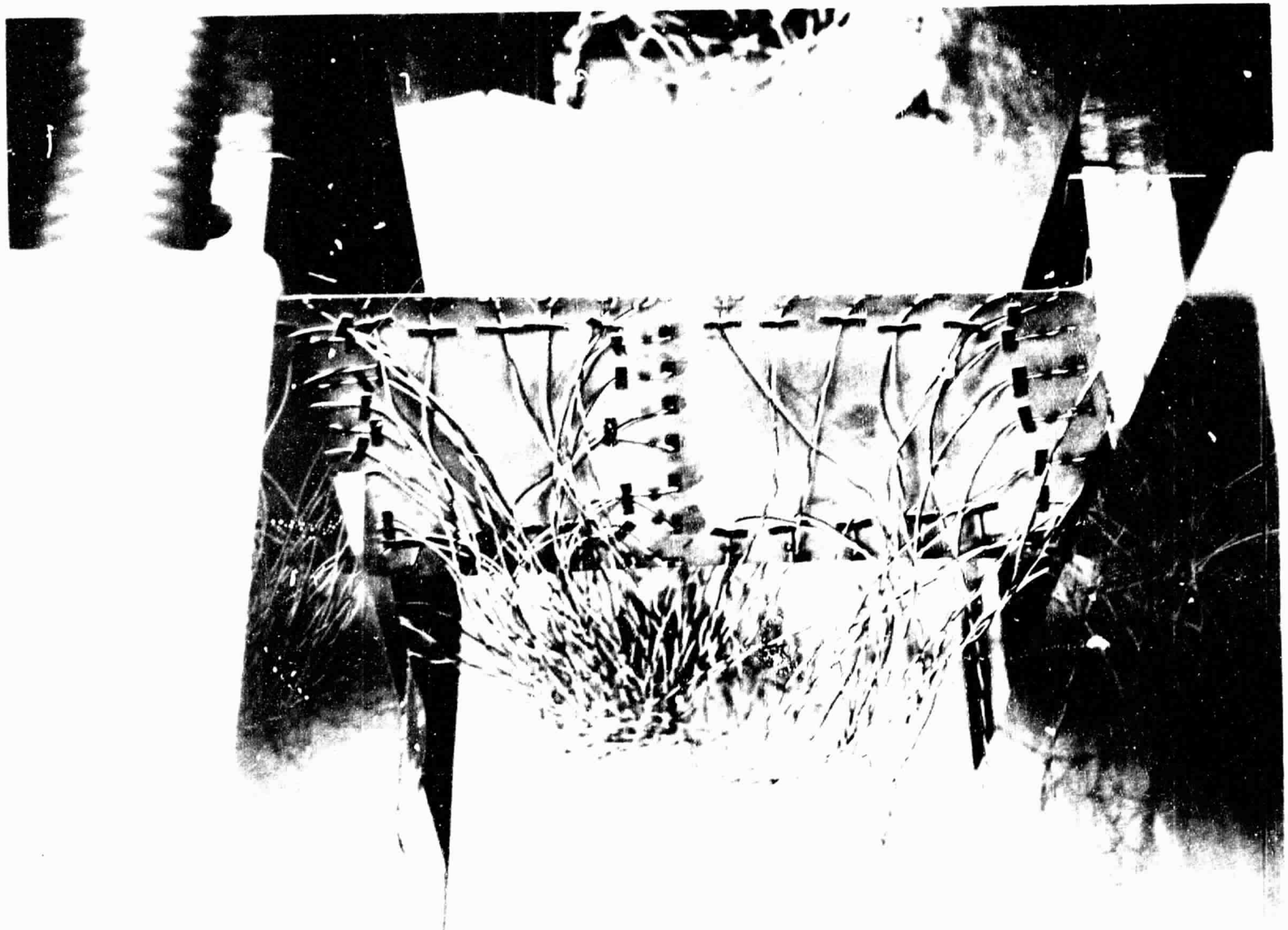

Figure 2.1-9. Stage 1 Vane: Bench Test Set-Up.

2.1.6 Full-Scale Fan Testing

2.1.6.1 Component Development and Evaluation

Technical Progress

During the past reporting period, the Component Development group has continued to coordinate the activities of various organizations required to assemble and test the fan rig. They completed the detailed instrumentation print, along with the computerized data bank code list; started fabrication of the assembly and balance tooling; initiated sensor application onto the fan frame, bypass vanes, and rotor blades; and continued the detailed review of the assembly and balance procedures.

Work Planned

- Monitor fabrication of assembly tooling
- Coordinate and follow instrumentation and assembly of the fan vehicle
- Assist in preparation of the Test Request
- Coordinate with Lynn Test Operation for vehicle testing.

2.1.6.2 Test Facilities Engineering

Technical Progress

Test Facilities Engineering has completed the detail design of inlet and discharge hardware required for fan testing; completed the fabrication of all inlet and discharge hardware; finished the fan vehicle assembly drawing; and placed an order for fabrication of the distortion screen hardware.

Work Planned

- Complete fabrication of distortion screen hardware
- Provide engineering support during buildup of the fan vehicle.

2.1.6.3 Instrumentation Design

Technical Progress

Designs have been finalized for the inlet airflow and distortion rakes using existing designs from the CF6-8C program. New designs were completed for the following rakes and manufacture was initiated of

- Compressor inlet P_T/T_T arc rakes and radial rakes for Planes 23 and 25
- Fan inlet boundary layer P_T
- Planes 13/93 P_T boundary layer
- Plane 14 P_T/T_T arc/radial rakes.

Preliminary designs have been completed for all the component applied aero and mechanical measurement sensors on the fan frame subassembly and stress and temperature sensors for the fan and booster rotors.

Cold probe calibration was completed for aero P_T and T_T sensors on the Plane 23 and 25 rakes and the frame bypass vanes.

Work Planned

- Complete final designs for all applied and separable instrumentation sensors.
- Provide aero calibration of selected P_T and T_T sensors on core OGV and Stage 1 stator vanes.
- Provide engineering consultation and direction to the shops during manufacturing, application and assembly of the test vehicle instrumented hardware.
- Provide engineering coverage for stress monitoring of instrumented rakes and consultation and initial test support for specialized data gathering parameters.

2.1.6.4 Lynn Facilities Engineering

Technical Progress

Lynn Test Facilities Engineering completed the design of the required facility hardware. Work was initiated on modifications to the test cell inlet stack and is continuing. Completed design of discharge valve actuation system and placed orders for required hardware.

Work Planned

- Complete modifications to inlet stack
- Remove and rework the bypass discharge valve system
- Install and check out discharge valve actuator system.

2.1.6.5 Development Assembly

Technical Progress

The Development Assembly group continued the preparation and publication of the assembly and inspection procedures required to assemble the fan vehicle. They also completed the design of assembly and balance tooling.

Work Planned

- Complete the procurement of fan assembly and balance tooling
- Complete the preparation and publication of the detailed assembly and balance procedures.

2.1.6.6 Instrumentation Application

Technical Progress

The Instrumentation Application group initiated the necessary planning for instrumentation of the fan vehicle hardware. They completed the fabrication of the Plane 93, 23, and 25 rakes; started this fabrication of the Plane 41 radial arc rakes and also the bellmouth inlet rakes; and started the application of mechanical and aerodynamic sensors to vanes and rotor blades.

Work Planned

- Complete manufacture of all fan aerodynamic rakes and probes
- Complete the application of the required mechanical and aerodynamic sensor to the fan hardware

- Provide coverage during the assembly of the fan vehicle for leadout and termination.

2.1.6.7 Lynn Testing

Technical Progress

Lynn test has continued to provide consultation to Evendale Evaluation pertaining to the interface between vehicle and test facilities; they have issued schedule and work requirements for rework of the bypass discharge valve system and have provided technical assistance to Lynn Facilities on stack modifications and the discharge valve actuation system.

Work Planned

- Remove and rework bypass discharge valve system
- Prepare cell for fan vehicle
- Install fan vehicle in test cell
- Conduct testing of fan vehicle.

2.1.7 Fan Fabrication

2.1.7.1 Fan Rotor

Technical Progress

All components required for FSFT buildup have been received.

Work Planned

- Follow vehicle buildup.

2.1.7.2 Fan Stator

Technical Progress

The fabrication of the hardware required for the full scale fan test (FSFT) has been completed. The core frame has been welded, final machined, instrumented, and shipped to an outside vendor for final machining of the core frame/bypass vane assembly. Figure 2.1-10 shows the core frame following completion of the EB-welding cycle.

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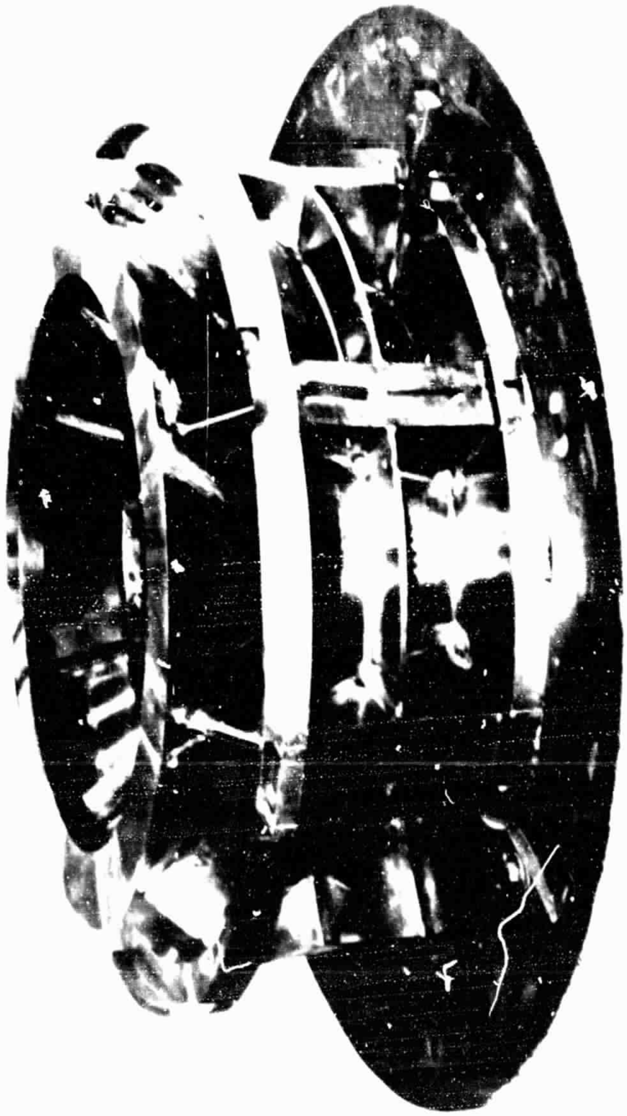

Figure 2.1-10. Fan Core Frame Welded Assembly.

The fabrication of the fan stator vanes has been completed as has been the stator assembly hardware from Mojave. Figures 2.1-11 through 2.1-13 show the completed fan stator vanes. Included in this hardware are the cut-back bypass vanes and the fairing and extension hardware for the bottom cut-back vane.

The fan midcase, containment case, and hardwall panels have also been delivered.

With the completion of the hardware fabrication for the FSFT, orders will be placed following issuance of the detail drawings for the hardware required for the ICLS engine test. Hardware to be procured includes the gearbox brackets, engine mount brackets, nacelle cowl door seal rings, and bottom bypass vane hardware; service line extension and stiffeners, support bracket, and radial driveshaft oil tube and pilot bracket.

Work Planned

- Initiate procurement of ICLS hardware.

2.2 HIGH PRESSURE COMPRESSOR

Overall Objectives

The primary objective of the compressor development effort is to evolve a 10-stage, high-performance, high-stage-loading design (Figure 2.2-1) capable of achieving a pressure ratio of 23:1 at the maximum climb design point. The primary aerodynamic design challenge will be to provide adequate levels of stall margin at part-speed operation while maintaining the high efficiency levels required for this compressor. The FPS compressor efficiency goal is 0.361 at Mach 0.8, 35,000 feet, standard day, maximum cruise power setting.

The mechanical design requirements include the development of an active clearance-control system for the rear block of compressor stages to achieve tip clearances at cruise compatible with efficiency and stall margin goals and to enhance performance retention. The compressor has fewer, longer chord

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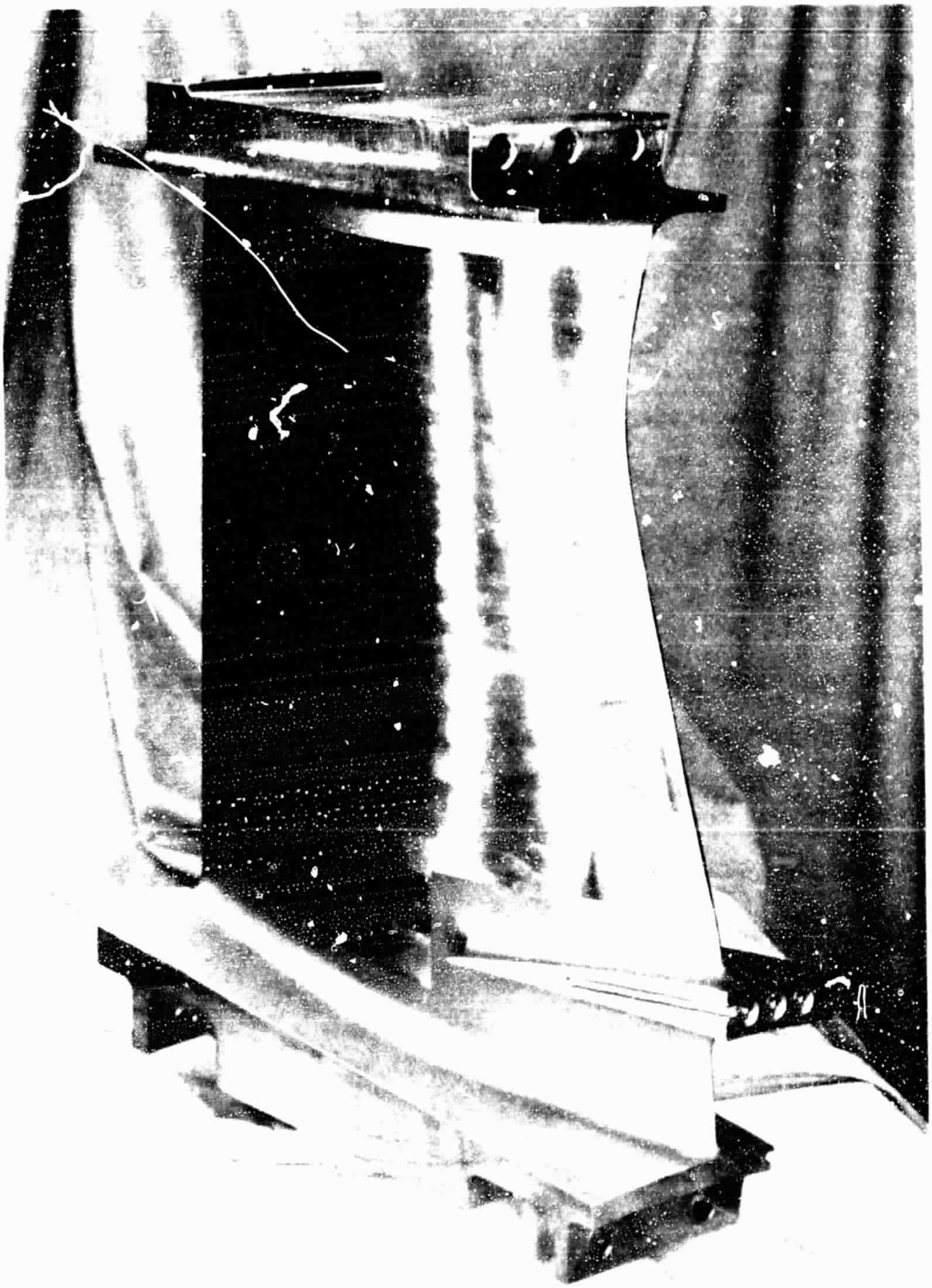

Figure 2.1-11. Fan Bypass Vane.

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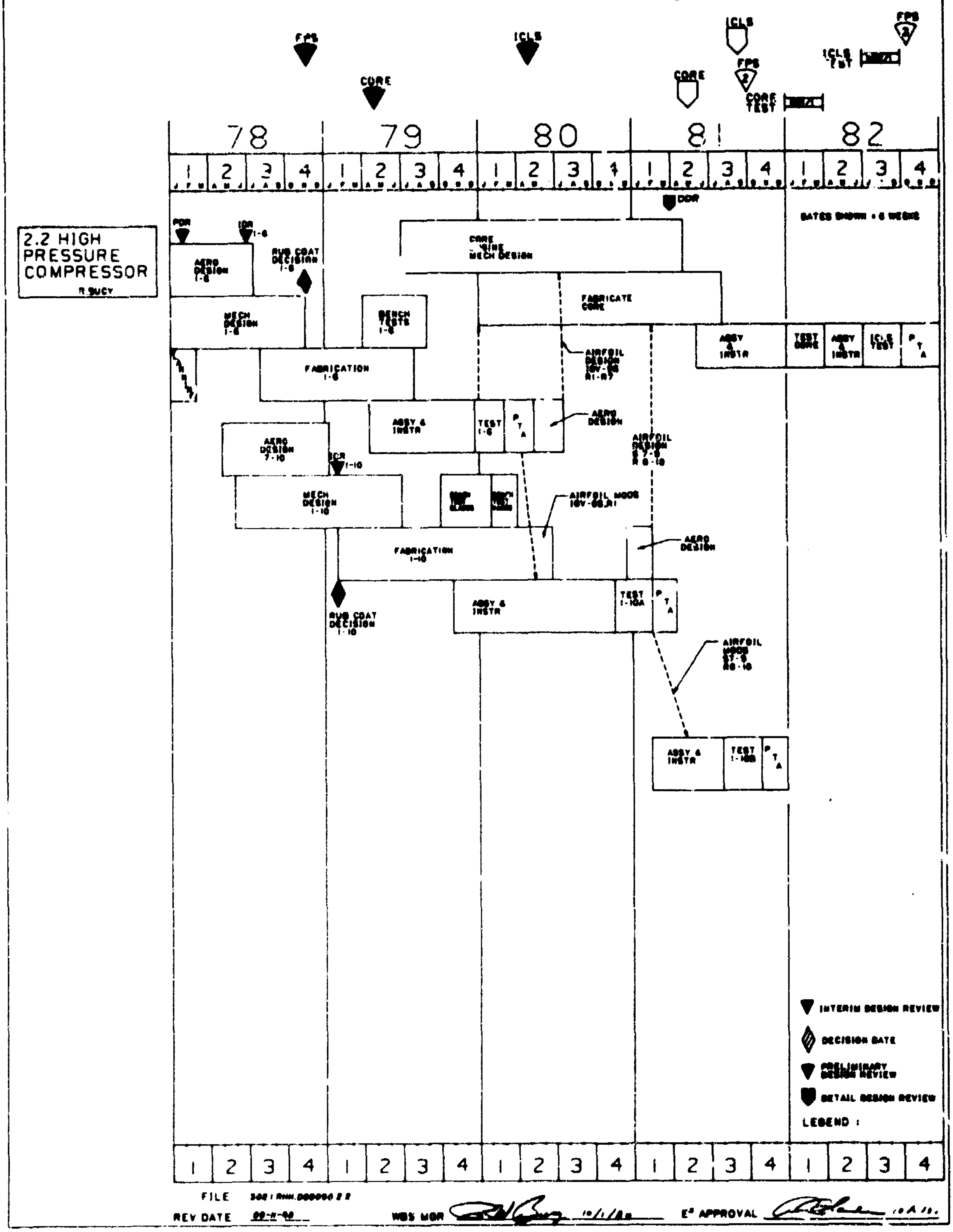

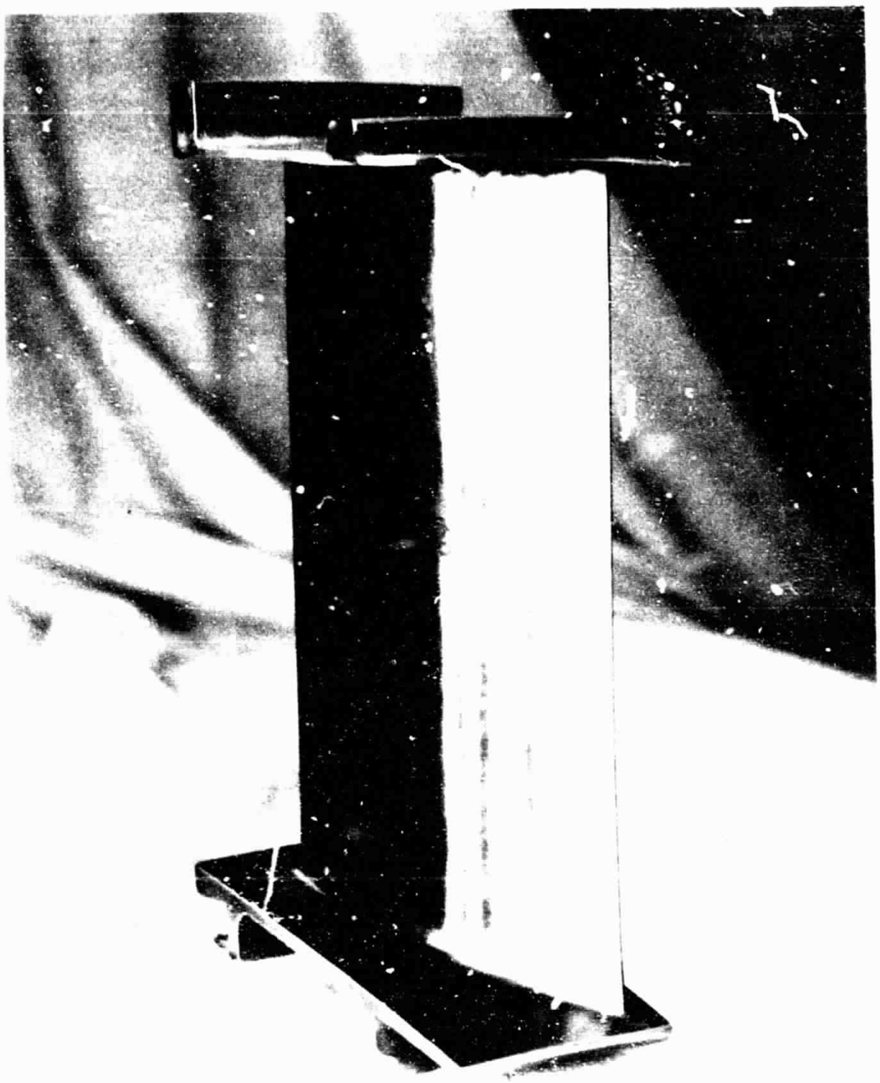

Figure 2.1-12. Fan Stage 1 Vane.

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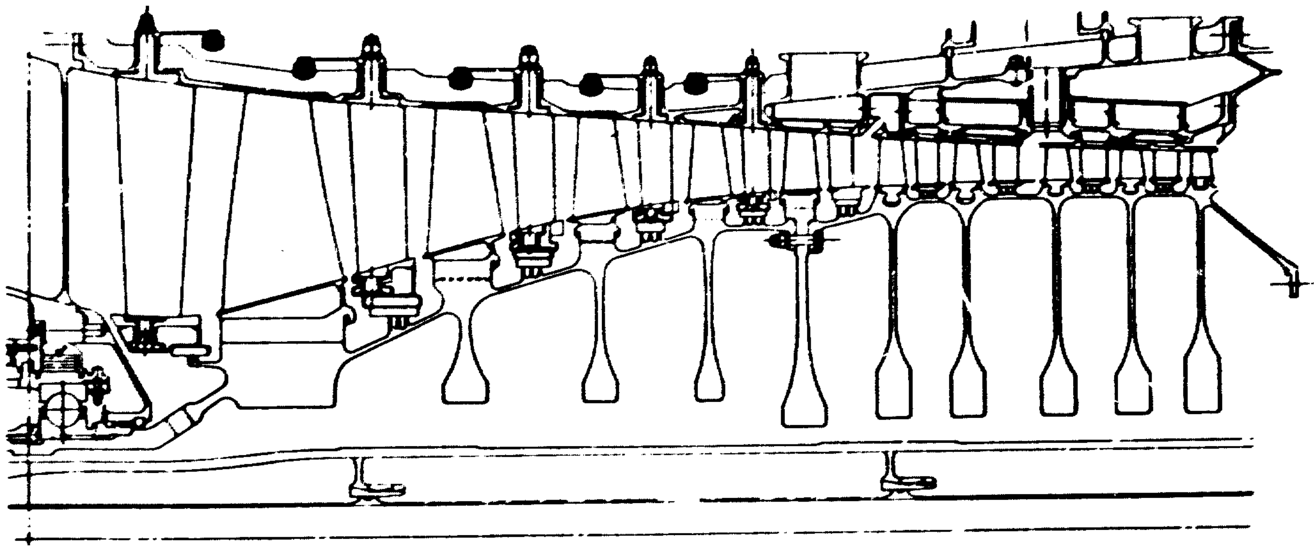
Figure [illegible] Quarter Stage Vane



- Efficiency - MXCR - 86.1%
- 10 Stage, 23:1 PR-MXCL
- Low Aspect Ratio, Rugged Blades
- Steel Casing
- Inertial Welded Spools

- Ti 17 Forward, R95 Aft Spool
- Digital Control of Variable Stators
- Active Clearance Control Stages 6-10
- Adequate Stall Margin

Figure 2.2-1. HP Compressor Cross Section.

airfoils (low aspect ratio) to increase blade life and general ruggedness and to reduce performance deterioration and operational costs. The compressor is short and stiff and, in conjunction with the short combustor and high pressure turbine, permits the use of only two bearings to support the core rotor.

A sequential arrangement of the tests will allow refinements of the design to be introduced throughout the compressor development test program. The test program will culminate with the ICLS test in 1982.

Development Approach

Precontract aerodynamic design studies were completed in sufficient detail to allow the detailed aerodynamic design to commence at contract initiation. The initial contract efforts are devoted to the detail design and testing of the first six stages of the compressor in component tests. The PDR for the compressor was, therefore, completed very early in the program (February 1978). An IDR was held in July 1978 before committing to the hardware fabrication cycle for a 1-6 stage component test shown in Figure 2.2-2. Following instrumentation and assembly, the 1-6 stage component test was completed during the month of February 1980, and posttest analysis of the data has been completed.

The detailed aerodynamic and mechanical design of the rear stages of the compressor was completed in the first quarter of 1979 during the procurement cycle for the first six stages. This permitted an additional IDR for the entire 10-stage compressor in late January 1979; compressor aerodynamic design parameters are shown in Table 2.2-1. Upon successful completion of this IDR, the hardware for the last four stages of the compressor was released for procurement.

Appropriate hardware from the 1-6 stage vehicle was merged with the hardware procured for the aft stages to complete a 1-10 stage component buildup shown in Figure 2.2-3. The first 1-10 stage component test was initiated during the first quarter of 1981.

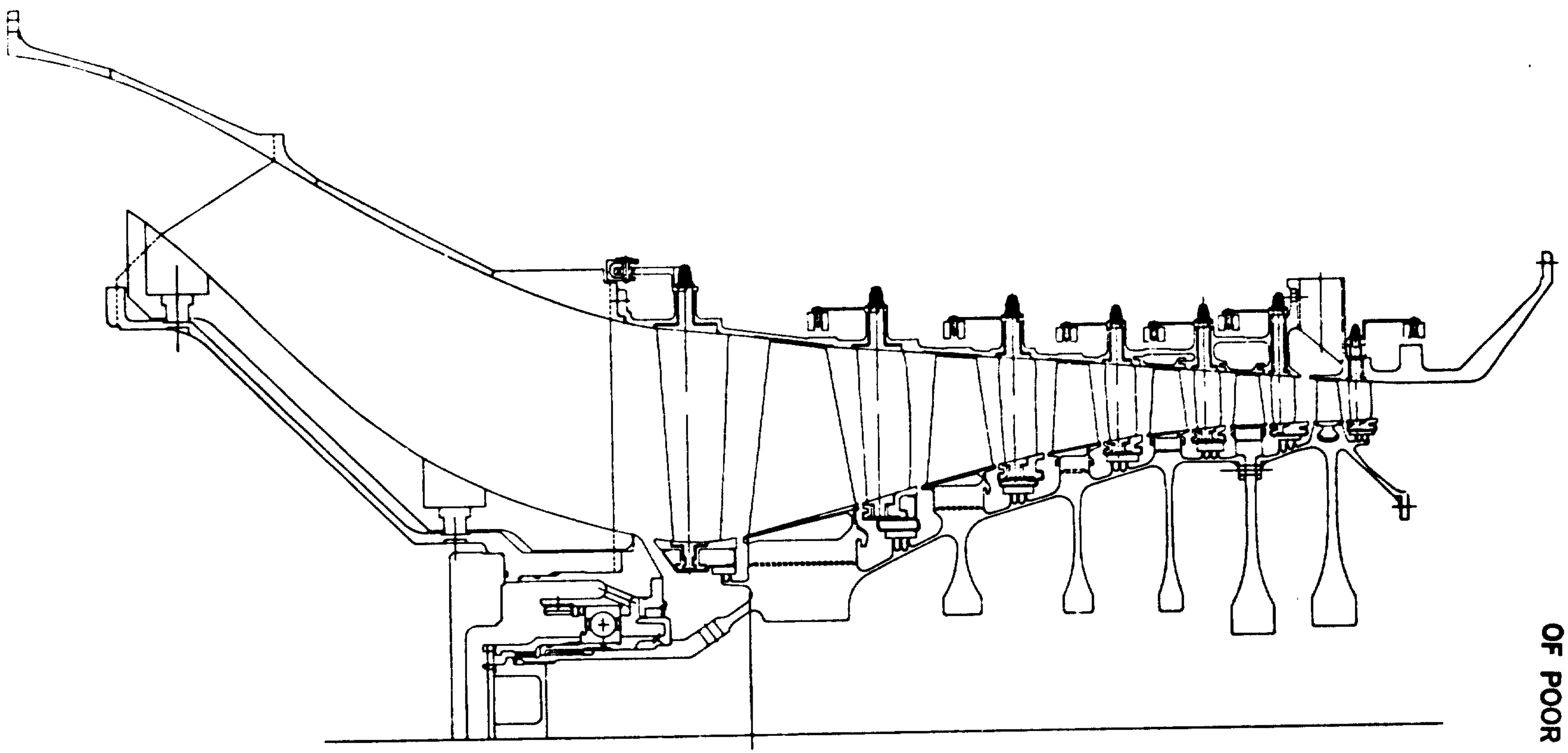


Figure 2.2-2. Stage 1 through 6 Compressor Rig.

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Table 2.2-I. Compressor Aerodynamic Design Parameters.

	Maximum Climb 35,000 ft/0.8 Mach/+18° F	Maximum Cruise 35,000 ft/0.8 Mach/+18° F	Takeoff SLS/+27° F
Corrected Speed, % Design	100.0	99.5	97.7
Corrected Airflow, lbm/sec	120.0	118.0	108.8
Total Pressure Ratio	23.0	22.4	20.0
Adiabatic Efficiency	0.857	0.861	0.865
Polytropic Efficiency	0.903	0.905	0.908
Inlet Temperature, ° R	547.9	542.5	621.6
Inlet Pressure, psia	8.65	8.42	21.84

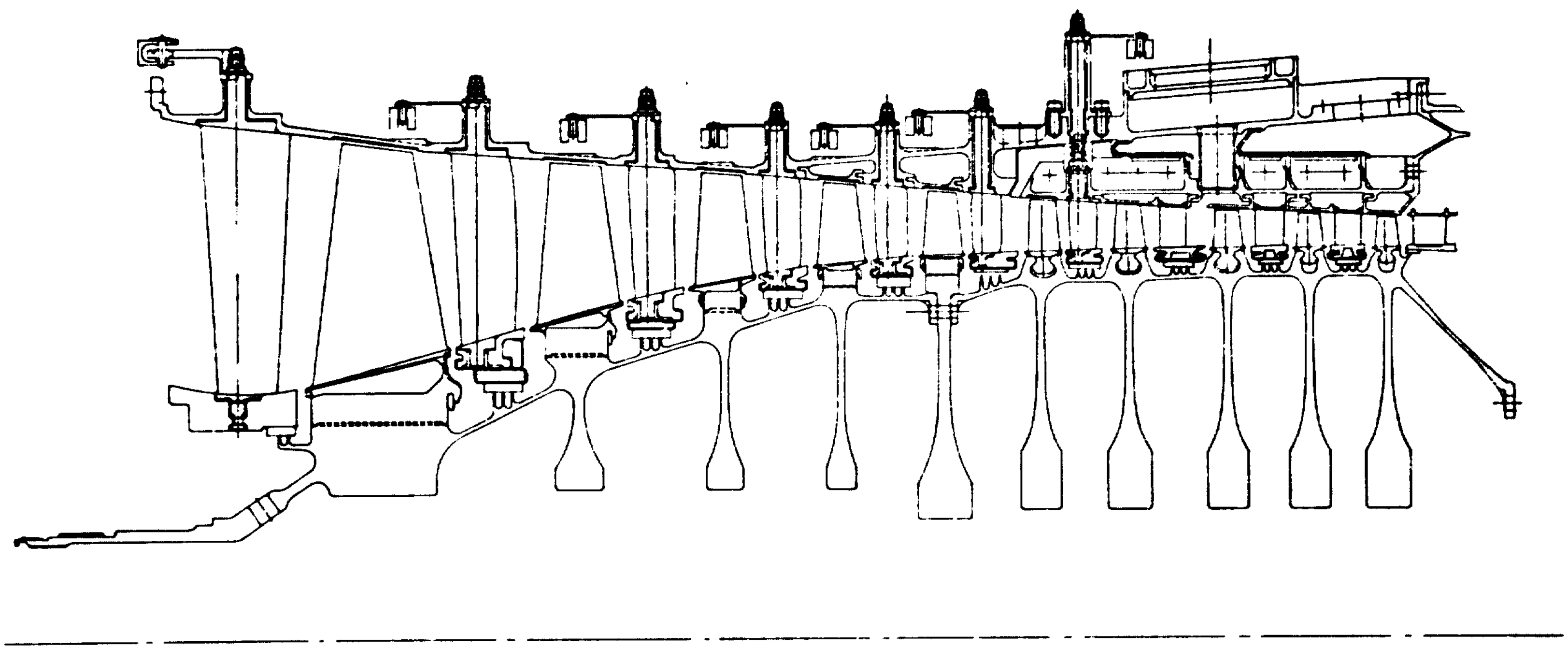

Figure 2.2-3. Stage 1 through 10 Compressor Rig.

Following analysis of the 10-stage test data, refinements in the mechanical and aerodynamic design will be executed in modified hardware in a second 10-stage component test in late 1981.

A complete new set of hardware, refinements for which will be defined based on the first 1-10 test, will be introduced into the first core engine test planned for the first quarter of 1982. Upon successful completion of this test, the low pressure system will be added to the core and built up into the ICLS to be tested in late 1982.

2.2.1 Aerodynamic Design

Technical Progress

2.2.1.1 HPC Aero Component Design and Test

During the past 6 months, the aerodynamic design activity has been limited to preparation for the 10-stage core compressor test. The principal activity has centered about the formulation of the instrumentation and test plans for the 10-stage test. These plans were documented and given final approval by NASA and were issued during the past semiannual period.

Preparations for the first 10-stage compressor test were carried out during the past reporting period. The data acquisition and data reduction plans were finalized and coordination with the Lynn plant engineering and test personnel continued to assure accurate and timely measurements and calculation of all test data. Aero coverage of the test vehicle assembly was provided to assure that hardware quality was consistent with the aerodynamic design intent.

Work Plannned

- The full-scale component test of the 10-stage core compressor will be run in March and April 1981. Testing will include numerous data points taken to define the performance of the compressor with both clean inlet and distorted inlet conditions.
- Stator schedule optimization will be performed and interstage radial traverses will be taken. The resulting test data will be analyzed in detail and a test memo will be written.

2.2.2 Rotor Mechanical Design

2.2.2.1 Design of 1-6 Rig

Technical Progress

This effort has been completed and results have been integrated into the 1-10 rig test.

Work Planned

- No further effort will be expended.

2.2.2.2 Design of First 1-10 Rig

Technical Progress

During this reporting period, the E^3 compressor first 1-10 rig design effort involved:

- Tip grinding of Stage 6 through 10 blades
- Completion of buildup of the instrumented 1-10 rig assembly
- Installation of the 1-10 rig vehicle in the General Electric FSCT facility in Lynn, Massachusetts
- Initiation of 1-10 rig testing.

Work Planned

- Complete the 1-10 rig testing
- Complete vehicle teardown and inspection
- Data reduction and reporting

2.2.2.3 Design of Second 1-10 Rig

Technical Progress

This task was initiated at the beginning of the first quarter of 1980. At that time, the total effort involved procuring all new hardware, except a 6-10 spool; and instrumentation, assembly, testing, and reporting on this effort. This effort is progressing on schedule. The second 1-10 rig build will utilize the first 1-10 rig build hardware and have the option of replacing the Stage 8, 9, and 10 Mod B blades with the Mod A blades or with reworked Stage 7-10 rig blades.

Work Planned

- Upon completion of the first 1-10 rig test and data reduction, possible aerodynamic modifications to the Stage 7 through 10 blades will be defined and drawings issued.

2.2.2.4 Design of the Core HPC

Technical Progress

During this period, the E^3 core HPC design effort invovled

- Release of alternate material for the manufacture of the CDP seal due to unacceptably late forging delivery
- First article inspection of Stage 1, 2, 5, and 6 blades has been completed, with the airfoils mechanically and aerodynamically acceptable
- Design and release of an instrumentation lead-out duct which can be disassembled and removed without rotor teardown (Figure 2.2-4)
- Blade instrumentation rework defined, quotes received, and orders placed
- Based on the axial dovetail flow check conducted in WBS Item 2.2.4, some desirable design changes to minimize leakage have been identified and incorporated into released hardware (including alternate deep blade retainer rings for use on some axial dovetail stages to reduce leakage).

2.2.3 Stator Mechanical Design

2.2.3.2 1-10 Design of 1-10 Rig, Build I

Technical Progress

Engineering coverage was provided during the interstage seal grind and vehicle assembly. Calculated scope limits for strain gages mounted on Stage 7

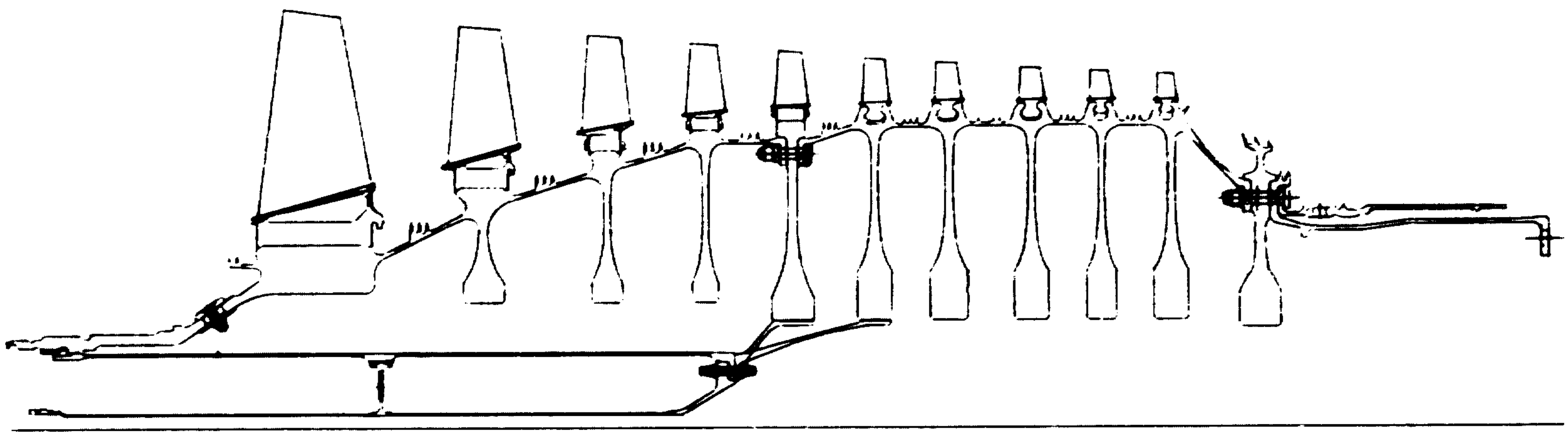

Figure 2.2-4. Core Rotor Assembly with Instrumentation Lead Out Duct.

through 9 vanes to be used for safety monitoring during vehicle test. Scope limits calculated for Stages IGV through 6 for the 1-6 rig test are applicable for the 1-10 test. Safety monitoring is being provided during vehicle testing.

Work Planned

- Continue to provide safety monitoring during vehicle test
- Analyze test data and prepare summary of test results.

2.2.3.3 Design of 1-10 Rig, Build II

Technical Progress

Work on this WBS was initiated in March with design of instrumentation rework and intervane circumferential seals on the cast stator vanes. Engineering coverage of procurement of the cast vanes was also provided.

Work Planned

- Continue coverage of cast vanes procurement.

2.2.3.4 Design of Core Engine

Technical Progress

Thermal, stress, and life analyses of the diffuser frame have yet to be completed. Modeling problems and a preliminary internal design review have resulted in more detailed analyses. An internal design review is scheduled near the end of this reporting period. Analyses will be completed next period.

Detail design of the VSV acuation system is continuing.

Design of all remaining hardware items is complete. Drawings of the VSV bushings, OGV/diffuser frame machining, and Stage 7, 8, and 9 vane machining were issued during this period. The inner support case (wishbone) was redesigned to accommodate the axial engine shim.

Engineering support of the hardware procurement is continuing.

Work Planned

- Complete diffuser frame analyses
- Complete detail design of VSV actuation system
- Continue to support hardware procurement.

2.2.4 Rotor Design Testing

Technical Progress

All bench tests associated with the 1-10 compressor rig blades have been completed. The tests include determination of airfoil natural frequencies, nodal patterns, stress distribution, root end effects, and fatigue strength. A dovetail leakage component test was conducted during the last reporting period but data reduction and analysis was not completed until early in this reporting period.

Work Planned

- Bench testing of the core compressor blades will be initiated in the second quarter of 1981.

2.2.6 Full-Scale Compressor Testing

2.2.6.1 Component Development and Evaluation

Technical Progress

Planning and coordination of compressor stator, front frame, diffuser frame, rear frame instrumentation application, and leadout design methods was completed; finalized preparation of a detailed instrumentation list and instrumentation drawing for the 10-stage vehicle; completed plans for the termination of vehicle aerodynamic and mechanical instrumentation; completed review of requirements for control room display and recording of vehicle operational and mechanical instrumentation; completed preparation and issuance of 10-stage compressor test request; provided engineering direction for the work performed on the 10-stage vehicle in the assembly, instrumentation, inspection, and machine shop areas; and provided engineering direction at Lynn during installation of the 10-stage vehicle into the test facility and during the initial testing phase.

Work Planned

- Provide engineering direction at Lynn during the testing of the 10-stage compressor vehicle
- Provide engineering direction during disassembly of the 10-stage vehicle and during assembly of the second 10-stage vehicle.

2.2.6.2 Test Facilities Engineering

Technical Progress

Fabrication of the 10-stage compressor vehicle distortion hardware was completed; provided engineering support to resolve problems encountered on hardware provided for the 10-stage vehicle during the vehicle assembly and test facility installation activities; they also provided engineering support during initial vehicle testing at Lynn.

Work Planned

- Provide engineering support during the remaining 10-stage vehicle testing
- Provide engineering support during the 10-stage vehicle disassembly and during assembly of the second 10-stage vehicle.

2.2.6.3 Instrumentation

2.2.6.3.1 Instrumentation Application

Technical Progress

Application of aerodynamic sensors to the variable stator vanes was completed; finalized application of aerodynamic and mechanical sensors to the forward and aft stator cases and to the Stage 7, 8, and 9 fixed stator vane sectors; completed instrumentation leadout and routing on the compressor rotor; completed fabrication of the compressor discharge Pt rakes and the diffuser exit rake; completed aerodynamic and mechanical sensor application to the front frame, rear frame, and diffuser frame hardware; completed the fabrication of mounting hardware required for the radial traverse probes and touch probes; completed fabrication of clearanceometer and dynamic pressure system hardware required for installation on the 10-stage vehicle; provided instrumentation support during all assembly activity on the 10-stage vehicle to leadout, route, and tie down instrumentation leads; and completed final routing and termination of 10-stage vehicle instrumentation sensors prior to shipment of the vehicle to the Lynn test facility.

Work Planned

- Provide support during disassembly of the 10-stage vehicle after completion of Lynn testing
- Provide support during assembly of the second 10-stage vehicle.

2.2.6.3.2 Instrumentation Design

Technical Progress

Instrumentation application and leadout drawings for the compressor stator, diffuser frame, and rear frame subassemblies were finalized; completed the aerodynamic calibration of the compressor discharge rakes, variable stator vanes, and fixed stator vanes Pt and Tt sensors; provided calibrated vibration pickups for the internal and external vibration measurements; completed design and configuration change to the clearance probes to update to latest improved design; completed design changes and fabrication of traverse probe mounting hardware for Stages 6, 7, 8, and 9; defined the inlet rake stress measurement requirements for test monitoring; completed the design of the stall detection dynamic pressure system hardware; provided engineering support during the final assembly and instrumentation application activity prior to shipment of the vehicle to the Lynn test facility; and also provided engineering support during test setup of the vehicle in the Lynn test facility and during the initial testing phase.

Work Planned

- Provide engineering support as required during the remaining 10-stage vehicle testing

- Provide engineering support during 10-stage vehicle disassembly

- Initiate instrumentation engineering design work for the second 10-stage compressor vehicle.

2.2.6.4 Development Assembly

Technical Progress

Buildup and tip grind of the instrumented 10-stage compressor rotor were accomplished; completed buildup of the aft stator case with uninstrumented Stage 6, 7, 8, and 9 stator vanes and completed grinding of the honeycomb shroud seals on these stages; completed buildup of the forward stator case with the IGV through Stage 5 variable stator vanes; completed buildup and final balance of the complete rotor system; obtained IGV and Stage 1, 2, 3, 4, 5, and 6 tangent angle checks on the variable stator vane; completed buildup of tne forward stator case with instrumented stator vanes; completed buildup of the aft stator case with instrumented fixed stator vane sectors; completed buildup of the front frame and diffuser frame subassemblies; completed buildup of the rear frame subassembly and initiated main vehicle assembly effort; and completed main vehicle vertical assembly work and turned vehicle horizontal and installed the vehicle transport dolly. Final vehicle assembly work was completed and the vehicle was shipped to Lynn on February 6, 1981.

Work Planned

- Perform partial disassembly of 10-stage vehicle after completion of Lynn testing

- Initiate assembly of second 10-stage compressor vehicle.

2.2.6.5 Lynn Testing

2.2.6.5.1 Test Facilities Engineering - Lynn

Technical Progress

Pressure check of the external bleed systems was completed; cleaned and pickled all internal tank bleed system piping and manifolds; completed installation of water spray system and measuring section; completed the installation of the flow measuring orifice plates into the bleed system piping; completed fabrication of base section required to support inlet distortion screen hardware; completed leak check of main flow discharge valve system. Assembled the remachined coupling shaft, secured the forward water jacket, and rechecked the axial dimensions in preparation for installation of the 10-stage compressor vehicle; provided engineering support during installation of the vehicle into the test facility and during installation of the internal test facility bleed manifolds and piping to the vehicle; and also provided engineering support during the initial testing of the 10-stage compressor vehicle.

Work Planned

- Provide engineering direction and support during the remainder of the 10-stage vehicle test phase
- After removal of the 10-stage vehicle from the test facility, inspect the test facility hardware and prepare hardware for second 10-stage vehicle.

2.2.6.5.3 Stage 1-10 Compressor Testing

Technical Progress

The necessary planning work required to prepare the test facility for delivery of the 10 stage compressor vehicle was completed; received the 10-stage vehicle at the Lynn test facility on February 10, 1981 and installed to the test tank mounting hardware on February 12, 1981; completed connection of the vehicle aerodynamic and mechanical instrumentation; connected the internal tank bleed system piping to the vehicle; completed connection of vehicle lube, hydraulic, and air services; completed installation of vehicle mounted vane angle position readouts, clearanceometers, and touch probes; completed installation of vehicle inlet hardware and mated with test facility inlet ducting; and performed final vehicle and test facility checkouts and initiated testing on March 20, 1981.

Work Planned

- Complete aerodynamic performance testing of the 10-stage compressor vehicle
- Complete removal of the 10-stage vehicle from the test facility and ship to Evendale.

2.2.7 Variable Stator Vanes (VSV)

2.2.7.1 VSV Bushing Application

Technical Progress

Further reduction of materials selections for variable stator bearings for the core engine test has been made, eliminating some earlier candidates on the basis of unfavorable cost, performance, longer development time, or materials availability problems. Selected for the core engine test are the following:

Vane Stages	Description	Material
IGV-4	Spacers*	Composite ZX (Skybond 703 resin)
IGV-4	Bushings	Fabroid XV
4-6	Bushings and Spacers	Mechanical Carbon

* thrust washers

This is not to say that other materials are totally excluded. Experimental bushings will be included as available for comparison, but the above selection will be the primary choices.

Endurance wear tests are in progress to support the core engine and to substantiate 18,000 hours service life; 2,500,000 rub cycles in the component wear test corresponds to 18,000 hours service life. Table 2.2-II shows endurance wear tests which are in progress.

Table 2.2-II. Endurance Wear Tests (200,000 cycles HDTO plus 2,300,000 cycles at cruise).

Material	Variable Stage Geometry	Variable Stage/ Load (lb)	Test Pressure (psig) HDTO/Cruise	Test Temperature (° F) HDTO/Cruise
ZX (703 resin)	1	1/62.8	30.2/12.4	316/190
ZX (NR-150 resin)	1	4/44.8	75.0/39.3	623/485
Fabroid XV	1	1/62.8	30.2/12.4	316/190 (Completed)
Fabroid XV	4	4/37.6	75.0/39.3	623/485
Carbon PBH-20	4	5/30.2	75.0/53.9	723/581

The NR-150 version of Composite ZX has been shown to be superior in heat resistance and wear to the Skybond 703 version. However, long term availability of NR-150 is not guaranteed by DuPont. ZX (NR-150) is not planned for engine test since the difference does not justify switching from 703 which is adequate for this test. The endurance wear test of ZX (NR-150) at Stage 4 temperatures is intended to demonstrate the heat resistance, wear, and potential of such a system, should the availablility situation change in the future.

The following is a status and assessment of materials evaluated to date:

Fabroid X and Fabroid XV - Fabroid X is the TFE-glass fabric/polyimide lined metal bushing with a 10-mil thickness liner, and with capability to about 500° to 600° F. While Fabroid X is a long life, low wear rate material, it was judged to be inadequate to attain 18,000 hours of E^3 service life. In earlier IR&D effort, it was found that modifying the TFE-glass fabric liner to incorporate double the TFE content, more than doubled the endurance wear life. This modified liner is about 15-mils thick (hence the designation "Fabroid XV").

The first 2,500,000 cycle endurance wear test was completed successfully using Fabroid XV at Stage 1 geometry and loading. Only 10 mils of wear were measured on the bushing (only 5 mils of wear were measured on the ZX (703) thrust washer), Table 2.2-III, Test 59. Thus under these Stage 1 conditions, Fabroid XV should have substantial margin to attain 18,000 hours.

Another endurance wear test using Fabroid XV at Stage 4 conditions is planned, but is awaiting delivery of new bushings, which were recently redesigned to add 0.20-inch length. Note that should Fabroid XV fail to meet 2,500,000 cycles at Stage 4 conditions, the alternate material is mechanical carbon PBH-20.

Composite ZX - Using the 15-mil TFE-glass fabric, an all nonmetallic bushing construction has been made which has the desgination Composite ZX. There are two versions of it, either made with Skybond 703 polyimide resin adhesive, or with DuPont NR-150 resin (Figure 2.2-5). ZX bushings could be expected to give longer life than Fabroid XV bushings at lighter weight, but probably at substantially higher cost (especially the NR-150 version). Thrust washers of Composite ZX (Skybond 703) similar to those shown in the upper photo of Figure 2.2-5, are being procured for the core engine test.

GENR-150 - While GENR-150 is a production material, it suffers from two drawbacks which have resulted in its elimination from the program.

- Dependence on NR-150 resin
- Higher than acceptable wear rate.

GENR-150 is the designation for bushings made from DuPont NR-150 polyimide resin reinforced with graphite fabric. The processing required to make such parts was developed by GE (hence the designation GENR-150). Since that time, the supplier has raised the price of the resin almost fivefold, and has announced it will accept no new applications, nor supply the resin after 1985.

For E^3, the wear rate at Stage 4 conditions has been shown to be unacceptably high. An endurance wear test was conducted on the older geometry (0.805-inch length) Stage 4 bushing, made of Xylan coated GENR-150, bushing

Table 2.2-III. Endurance Wear Test Data

Test No.	Material	Cycles (1000's)	Temp. (°F)	Nominal Stage Conditions	Max. Wear Mils	Max. Wear Mils/10^5 Cycles	Actuation Force (Lbs)	Remarks
56	Filimide 7X	501	700	4	B 7 W 5	3.7 2.7	4.5-19	Old style stage 4 bushing.
58	GENR-150/ NR-150 Xylan Coated	1,212	623/485	4	B 32 W 31	2.6 2.6	2.0-17	Endurance t t, old style stage 4 bushing (see text).
59	Fabroid XV bushing, ZX(703) washer	2,500	316/190	1	B 10 W 5	0.4 0.2	7-14	Endurance test (see text).

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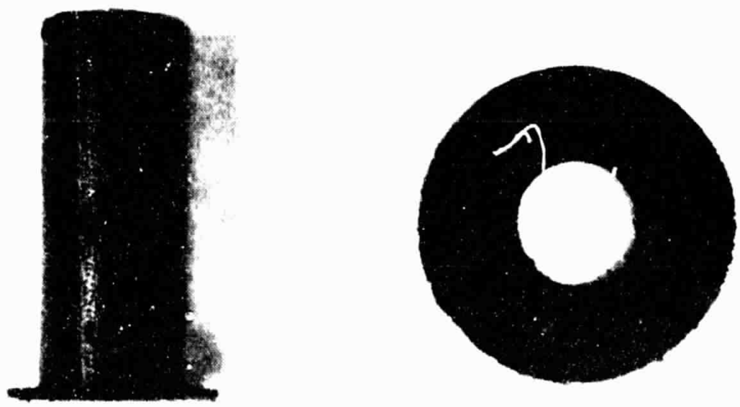

- High Boss, Suitable for E^3

- High Boss, Suitable for CFM56 or F101

- Low Boss, Suitable for CF6 Family

Figure 2.2.5. Composite ZX (NR 150) Produced in Various Configurations.

and washer. The endurance wear test was terminated at 1.2 million cycles, exhibiting 32 mils of wear (Table 2.2-III, Test 58). Even at the new length (1.00 inch) and thicker wall, it is doubtful that GENR-150 could attain 2.5 million cycles. No further effort with this composite is planned at this time for E^3.

PBH-20 Mechanical Carbon - Mechanical carbons remain as the only attractive bearing material for Stages 5 and 6 (700° to 900° F). Mechanical carbon is the term used to describe carbon-graphite pressed into a metal sleeve (or backed by metal as in a thrust washer).

Since the last report, a parametric study has been completed on Pure Carbon Co.* PBH-20 carbon bushings and washers (Table 2.2-IV). Test 57 represented an endurance wear test at Stage 4 conditions with 200,000 cycles at 700° F and 75 psi pressure, plus the remainder of test at cruise conditions of 580° F and 40 psi. This bushing was not of the latest configuration, having the older 0.805-inch length. The bushing wore out at a respectable 2.2 million cycles. This test will be repeated using the newer design at Stage 5 conditions, and it can be expected to pass the 2.5 million cycles target with ease.

Miscellaneous and Advanced Materials - TRW Bearing Division's "Filmide 7X" material was dropped from further consideration because of dependence on DuPont NR-150 and cost. Based on Test 56 (and other related experience), it is doubtful that it has any advantage over GENR-150 in these applications.

DuPont has introduced two new grades of Vespel, which are intended to be improved in heat resistance and dimensional stability over present Vespel grades. DuPont data indicates excellent dimensional stability, with low shrink and low weight loss as compared to present Vesepl after thermal oxidative exposure to air at 675° F, 70 psia for 100 hours. If this behavior is

*An excellent text on this class of materials has been published, written by Dr. R. Robert Paxton, Chief Engineer of Pure Carbon Co., entitled, "Manufactured Carbon: A Self-Lubricating Material for Mechanical Devices," CRC Press, 1979, ISBN 0-8493-5655-5, and LCC 78-10107.

Table 2.2-IV. PBH-20 Parametric Wear Rate Study Results.

Test No.	Cycles (1000's)	Temp. (°F)	Press (psig)	Load (Lbs)	Angle ±(Deg)		Max. Wear Mils	Max. Wear Mils/10^5 Cycles	Actuation Force (Lbs)	Remarks
45	500	700	75	32	19	B*	11	2.2	3-11	Central Conditions 16AA on vane stem, 100 cpm.
						W	8	1.6		
46	500	700	75	$\boxed{16}$	19	B	8	1.6	1-2	----
						W	5	1.0		
47	500	$\boxed{500}$	75	32	19	B	8	1.6	2-4	----
						W	4.5	0.9		
48	369	700	75	$\boxed{48}$	19	B	16	4.3	3-7	Washer deflected and cracked
						W	6	2.2		
49	500	700	75	32	$\boxed{25}$	B	13	2.6	4-8	----
						W	7	1.4		
50	500	700	$\boxed{0}$	32	19	B	9	1.8	3-6	3 cracks in washer, edge chipped in bushing
						W	4	0.8		
51	500	700	75	32	$\boxed{2}$	B	2	0.4	6-13	----
						W	2	0.4		
52	500	$\boxed{750}$	75	32	19	B	9	1.8	3-5	----
						W	7	1.4		
53	500	700	75	32	19	B	10	2.0	3-4	Surface finish on vane stem $\boxed{8\text{-}10AA}$. Other tests 32AA.
						W	8	1.6		
54	51.8	700	75	32	19	B	1	1.9	3-4.5	Cyclic Rate $\boxed{10\ cpm}$, all other tests 100 cpm.
						W	1	1.9		
55	500	700	75	32	19	B	8	1.6	2.5-4	Surface finish on vane stem $\boxed{45AA}$ other tests 32AA.
						W	5	1.0		
57	2,207	700/580	75/40	32	19	B	28	1.3	3.5-15	(See text.)
						W	13	0.6		

$\boxed{\quad}$ Numbers in boxes represent variations from central conditions, Test 45.

*B = Bushing W = Washer

substantiated by our evaluation, then a number of applications are possible; where present Vespel, with its excessive shrinkage above 500° F, renders it unsatisfactory for long service.

In IR&D effort, one objective is to identify less brittle, alternative, dry bearing materials to replace mechanical carbons for service above 675° F. One idea which might have application in E^3 is a metal bearing with multiple carbon-grpahite button inserts. Thrust washers of a size to fit E^3 Stage 4 geometry are on order for evaluation.

Work Planned

- Complete remaining endurance wear tests as outlined earlier
- Coordinate procurement and transitioning of bushings and washers for core engine tests
- Issue Final Report.

2.2.8 High Pressure Compressor Fabrication

2.2.8.1 Compressor Rotor

2.2.8.1.1 1-6 Rig

All hardware was received in the first and second quarters of 1979.

2.2.8.1.2 1-10 Rig Build A

Technical Progress

All hardware required for the 1-10 rig compressor rotor was received during the last quarter of 1979 and the first and second quarters of 1980.

Work Planned

- None.

2.2.8.1.3 1-10 Rig Build B

Technical Progress

No effort was expended on this item during this reporting period since the 1-10 rig second build will use the same hardware as the first build with the possible exception of modified Stage 7-10 blades.

Work Planned

- Following the first compressor rig test, Aerodynamic Design will identify the desired Stage 7-10 blade design changes to be incorporated into the Build B blades. These modifications will be initiated.

2.2.8.1.4 Core

Technical Progress

A complete set of compressor rotor hardware is currently being procured for the core engine buildup. Work on the forward spool is currently in process. The Stage 5 disk has been received, as has the 6-10 spool originally ordered under WBS Item 2.2.8.1.3. The CDP seal material has been released and is now in the manufacturing cycle. The Stage 1, 2, 5, and 6 blades are in the final stages of manufacture while the Stage 3, 4, and 7 blades are proceeding on schedule. The Stage 8-10 blades are in dovetail forming; and airfoil machining will proceed after the design based on initial 1-10 rig test data is determined. The instrumentation leadout duct has been released for manufacture and material procurement has been started. Deep blade retainer rings, designed to reduce axial dovetail leakage, have been released and placed on order.

All core rotor hardware is on schedule for the core buildup.

Work Planned

- Incorporate instrumentation rework into rotor spool components
- Receive the balance of rotor hardware.

2.2.8.2 Compressor Stator

2.2.8.2.2 1-10 Rig Stator Hardware, Build I

Technical Progress

Grinding of Stage 7, 8, and 9 interstage seals completed this WBS.

Work Planned

- None.

2.2.8.2.3 1-10 Rig Stator Hardware, Build II

Technical Progress

Manufacture of Stage 7, 8, and 9 interstage seals is complete.

Dimensional inspections of Stage 7 and 8 cast vane patterns denote that patterns are within blueprint tolerances. Inspection of Stage 9 revealed a deviation to the airfoil contour. This has been corrected by the pattern vendor. One sample each of Stages 7 and 8 has been poured to test casting parameters. Word from the vendor is that these trails were successful. They will be inspected by Engineering personnel.

Work Planned

- Cast Stage 9 trial sector
- Inspect Stage 7, 8, and 9 trial sectors
- Complete casting of all sectors
- Initiate machining of cast sectors.

2.2.8.2.4 Core Stator Hardware

Technical Progress

Manufacture of Stage 7, 8, and 9 cast vane sectors is paralleling the 1-10, Build II sectors; therefore the status of the core sectors is as reported above.

A sample pattern of the OGV has been poured as a casting trial. Tooling required to produce patterns and cores for the diffuser frame is progressing. Tooling and material required for the front frame is being procured. Manufacture of the aft compressor case is complete. Manufacture of the forward compressor case is about half complete. Manufacture of all other stator hardware items is in various stages of completion.

Work Planned

- Work planned for cast sectors is the same as for 1-10 rig, Build II
- Complete casting of OGV ring
- Complete casting of diffuser
- Complete manufacture of all stator hardware items except machining of the cast stator vanes.

2.3 COMBUSTOR

Overall Objectives

The key objective of this program is to design and develop an advanced combustion system capable of meeting both the stringent emissions and long-life goals of the E^3, as well as meeting all of the usual performance requirements of combustion systems for modern turbofan engines. The specific E^3 emissions goals are the emissions standards for carbon monoxide (CO), unburned hydrocarbons (HC), oxides of nitrogen (NO_x), and smoke that have been specified by the EPA for newly certified subsonic aircraft engines. These very stringent CO and HC emissions goals require very high combustion efficiencies at all engine operating conditions, including idle. The FPS combustion efficiency goal is 0.995 minimum at high power settings with a total pressure drop of 5.0% maximum.

To meet these emissions goals and other performance requirements, an advanced, short-length, double-annular combustor design concept has been selected (Figure 2.3-1). This design approach was chosen based on the low-emissions combustor design technology developed in the NASA Experimental Clean Combustor Program (ECCP) and the NASA Quiet Clean Short Haul Experimental Engine (QCSEE) Program. A comparison of the key combustor design parameters for the NASA E^3, NASA ECCP, and NASA QCSEE combustors is shown in Table 2.3-1. In these development programs, it was demonstrated that with the double-annular combustor design concept, low emissions levels could be obtained in addition to obtaining the other combustor performance capabilities required for satisfactory operation of a turbofan such as the E^3. To meet the long-life goals, an advanced, double-walled, axially segmented, cooling liner design concept using both impingement and film cooling has been selected. This advanced cooling liner design concept, in conjunction with the very short length of this design, is projected to result in a combustor configuration with the requisite long life objectives.

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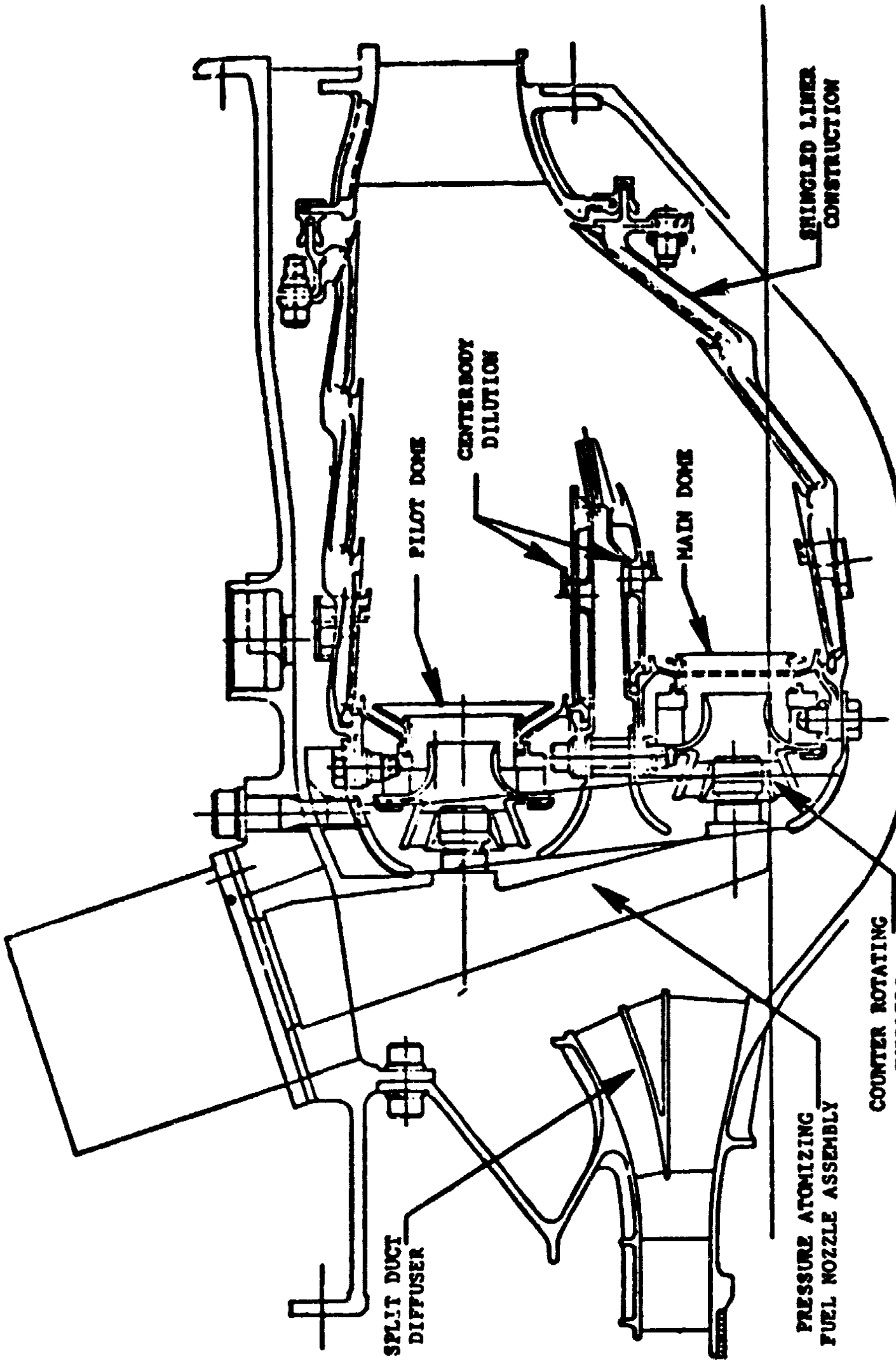
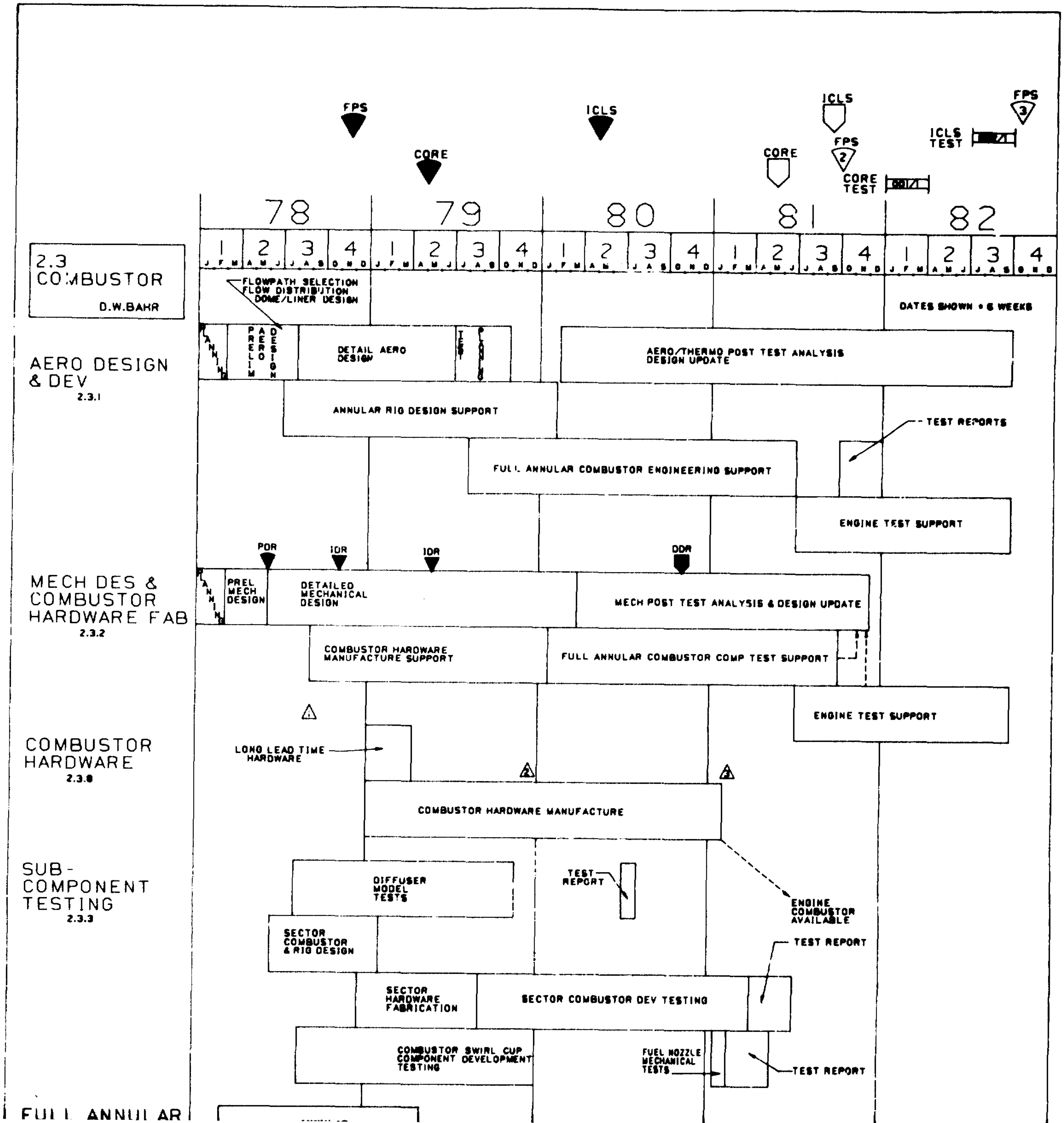


Figure 2.3-1. Combustor Design Concept.



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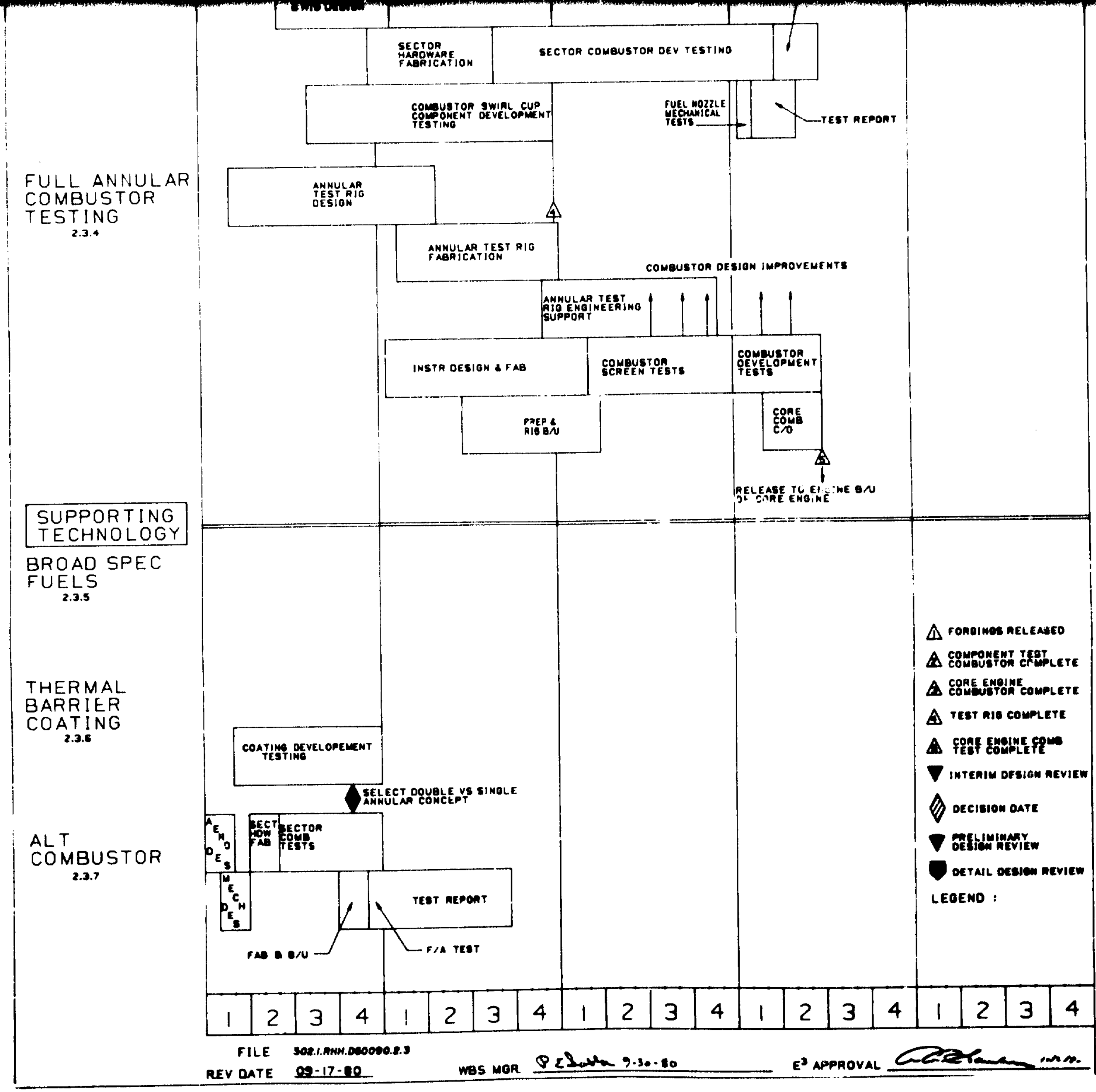

FOLDOUT FRAME 2

Table 2.3-I. Combustor Aerodynamic Design Parameter Comparison.

Combustion System	E^3 Double Annular	ECCP Double Annular	QCSEE Double Annular
Burning Length (in.)	6.7	12.9	7.0
Pilot Dome Height (in.)	2.4	2.7	2.3
Main Dome Height (in.)	2.0	2.4	2.0
Length/Dome Height - Pilot	2.8	4.8	3.0
Length/Dome Height - Main	3.4	5.4	3.5
Number of Fuel Injectors - Pilot	30	30	20
Number of Fuel Injectors - Main	30	30	20
Reference Velocity (ft/sec)	58	75	59
Space Rate ($Btu/hr-ft^3-atm \times 10^{-6}$)	7.0	5.8	8.1

Development Approach

Basic Program (Double-Annular Combustor Design)

The overall design definition and development approach have been selected for the E^3 combustor. The aerodynamic and mechanical design features identified for the diffuser, cowling, dome, and liner of the combustor were based on proven design and analytical techniques evolved in other General Electric combustor designs.

Initially, an extensive series of combustion system aeromechanical design and design tradeoff studies was conducted using existing combustor design tools and correlations to define the optimum combination of the basic combustor design parameters. These design studies included the definition of the combustor inlet diffuser, combustor cooling features, the number of fuel injectors, the swirl cup design features, and the dilution airflow patterns. Based on the results of these design studies, combustion system performance predictions and emission levels were estimated. The final flowpath design was then incorporated into a mechanical design layout.

Following these initial design studies, detailed stress analysis, heat transfer analysis, and life prediction studies were made for both steady-state and transient operation at the most adverse operating conditions.

Based on these detailed aerodynamic and mechanical design studies, the detail features of the combustor evolved into engineering drawings for procurement of development test hardware.

To evaluate and confirm these design analyses efforts, subcomponent tests are being conducted. The purpose of these tests is to permit preliminary development and refinement of the emissions and performance characteristics of the combustor design prior to and during the full-annular combustor development testing. These subcomponent development tests include diffuser tests to develop the diffuser aerodynamic characteristics in order to obtain the required combustor inlet conditions at the various simulated engine conditions from idle to takeoff, individual swirl cup tests to develop the required fuel preparation and introduction characteristics to obtain the desired combustion zone stoichiometry, and sector combustor tests to develop the pilot-stage dome features to meet the idle emissions goals and develop ground-start and altitude-relight capability.

The diffuser test program was conducted in a full size, full annular test rig and is complete. These tests were run with ambient temperature airflow, and the diffuser was tested over a broad range of passage flow-split values with three different inlet-velocity profiles. The results of these tests, based on measured static pressure recovery and total pressure loss coefficients for the inner and outer combustor domes, the centerbody passage, and the inner and outer combustor liner passages, indicated that the diffuser would perform satisfactorily in the development combustor test rig and in the E^3. The test results have been confirmed based on measurements recorded in the development test rig.

The swirl cup spray-visualization tests were conducted to determine the effects of combustor dome pressure drop, swirl cup fuel-to-air ratio, and fuel-to-air momentum ratios on fuel spray characteristics such as spray angle, stability, and atomization. The results of these tests were utilized to select the swirl cup design features for the sector combustor and for the full-annular combustor.

The sector combustor tests are being conducted with a 60° sector of the E3 combustor design. The emissions-related aspects of these tests are primarily directed toward obtaining low CO and HC emissions levels in the pilot stage at idle operating conditions and evaluating the crossfire and fuel-staging characteristics between the pilot and main stages. Tests of the baseline configuration sector combustor and several modifications to the baseline are complete. The objective of the present testing will be to select a preferred pilot-stage configuration to obtain even lower idle emissions. The sector combustor tests will also continue to develop ground-start ignition capability. Several combustor configurations have been evaluated in the sector combustor test rig that have demonstrated significant improvement in low power emissions levels and ground start ignition capability. Later in the test program, the altitude-relight capability will be investigated. Initially, these altitude-relight tests will be conducted with ambient-temperature air. Configurations that light within the envelope will then be tested both with cold air and with cold fuel (about -30° F) at the air temperature and pressure corresponding to the combustor conditions for the altitude and Mach number being evaluated.

A significant portion of the planned E^3 combustor component development effort is being performed with a full-scale, annular test vehicle that duplicates the flowpath of the engine and can accommodate full-scale engine combustor hardware. Testing in the combustor test vehicle is being peformed at both atmospheric and elevated-pressure conditions.

The exit annulus of the test rig is at the same radial location as the turbine nozzle of the engine. The test vehicle will simulate the cooling flows of the turbine nozzle diaphragm and first-stage turbine blades.

The OGV/diffuser section of the full-annular test vehicle simulates the aerodynamic characteristics of the airflow delivered to the combustor from the engine compressor. Provisions to add inlet airflow profiling features to simulate compressor-circumferential and radial distortion are also included. The test rig prediffuser incorporates structural features identical to those of the engine design, including flowpath simulation, strut supports, and bleed capability at the prediffuser trailing edge. Pressure-measurement instrumentation is provided along the diffuser flowpath surfaces to monitor stability and performance.

The combustor section provides the structural pressure vessel to house the combustor and duplicates the flowpath of the engine combustor housing. Ignitor port locations can be incorporated at several circumferential positions to permit selection of location flexibility based on sector combustor-component test data.

Two different exit instrumentation sections are being used with this test vehicle. Atmospheric pressure tests are performed with a mechanically actuated ring mounted from the outer flange. Temperature and pressure rakes are mounted to the ring which is traversed around the combustor circumference. With this test rig setup, the combustor airflow is discharged directly to the atmosphere, and the reaction zone can be viewed directly from the combustor exit. For testing at elevated pressure levels, a high pressure casing containing five internal rotating gas sample rakes is used. This exit section is capable of operation up to pressures of 300 psia.

Two different types of full-annular combustor tests are being conducted: (1) high pressure tests to develop the emissions, performance, and durability characteristics of the combustor at various simulated engine-operating conditions from idle to takeoff and (2) atmospheric-pressure combustor tests to develop the required combustor exit temperature distributions and ground-start ignition capability. The altitude-relight capability of promising combustor configurations will be evaluated in accompanying sector combustor tests.

The evaluation of the baseline combustor configuration as the first phase of development testing has been completed. The design information acquired in the baseline test in conjunction with preferred combustor design features evolved in the sector combustor development program were incorporated into a modified version of the baseline design.

Cold-flow calibrations of this modified E^3 combustor test hardware were performed to verify that the various dome and liner cooling airflows were distributed as intended. Then full-annular tests were conducted at atmospheric pressure to provide additional data on pattern factor, profile factor, and ground-start characteristics. Following completion of the atmospheric tests, evaluation of the improved design for reduced emissions was conducted at ground idle operating conditions. A final high pressure test was conducted to measure combustor metal temperatures at simulated high power conditions. Following analysis of the data from this modified configuration, additional development tests will be conducted to further improve the design.

Upon selection of the final design for engine installation, all of the design features evolved in the development program will be incorporated in the core engine combustor. This hardware will undergo a complete evaluation of all facets of combustor operation including ignition, emissions, and performance. The combustor will then be released to the core engine upon satisfactory completion of the tests.

2.3.1 Aerodynamic Design

The E^3 double-annular combustor concept is an advanced design approach which must meet the engine performance requirements as well as the emissions goals over a wide range of operating conditions. Some of the key operating conditions for the E^3 combustor are shown in Table 2.3-II. Operation of the engine combustor at these varied conditions requires that the combustor fuel

Table 2.3-II. Combustor Aerodynamic Parameters.

EPA Landing/Takeoff Cycle

Sea Level Static/Standard Day

	SLTO	Climb	Approach	Idle	Cruise 35,000 ft/0.8 Mach/+18° F
W_3 (pps)	146.1	131.0	69.3	27.1	63.6
T_3 (° R)	1467	1409	1145	895	1408
P_3 (Psia)	438.9	382.5	175.6	63.4	189
ΔP (%)	5.02	5.08	5.34	5.0	4.97
Fuel Flow (pph)	10730	8776	2930	993	4695
Reference Velocity (fps)	58.8	58.1	54.1	46.1	57.1
Residence Time (ms)	3.3	3.4	2.8	4.3	3.3
Compressor Mach No.	0.305	0.307	0.318	0.302	0.301
Space Rate $\left(\frac{\text{BTU}}{[\text{hr-ft}^3\text{-atm}] \times 10^{-6}}\right)$	7.0	6.6	4.8	4.5	7.9

flow be staged to the two domes as shown in Figure 2.3-2. Therefore, considerable aerodynamic development effort is anticipated to obtain satisfactory operation over this wide range of operating conditions while meeting the very challenging performance and emissions goals for this engine.

At the conclusion of the last reporting period, the Mod. I annular development combustor had completed testing and the results have been published in a Component Test Memo. In general, the results of the test were very encouraging with improvements obtained in low power emissions, ground-start ignition, and exhaust gas temperature distribution. The emissions levels are approaching the program goal level and the exhaust gas temperature distribution satisfies the requirements at a pilot-to-total fuel flow level of 0.40 which is very close to the design level of 0.35. However, the ignition performance requires considerable improvement to provide satisfactory operation in the severe combustor inlet environment encountered during ground start. As shown in Figure 2.3-3, the pilot stage ignition performance is more than adequate to meet the ignition fuel/air ratio schedule, but the main stage ignition performance will not permit staging for operation on both pilot and main below 55% core speed. Therefore, development activities were directed at design changes to improve the main stage ignition while minimizing the impact on other key combustor performance parameters. One of the key differences between the pilot stage dome and main stage dome is air velocity as shown in Figure 2.3-4. The purpose of the higher main stage dome velocity is to reduce residence time in this stage to lower emissions of NO_x at high power operating conditions. However, these high dome velocities which are accomplished through higher airflow and a smaller dome volume adversely impact ignition performance.

Utiliing the design data acquired in testing the Mod. I annular configuration and integrating the results obtained in the ongoing sector combustor development program, a modified annular development combustor was evolved directly primarily at improving main stage ground start ignition capability.

The overall impact of these design changes was expected to imrpove main stage ignition performance to meet ground start requirements, further reduce low power emissions levels, maintain combustor exit temperature distribution to within limits, and reduce combustor metal temperatures in selected regions.

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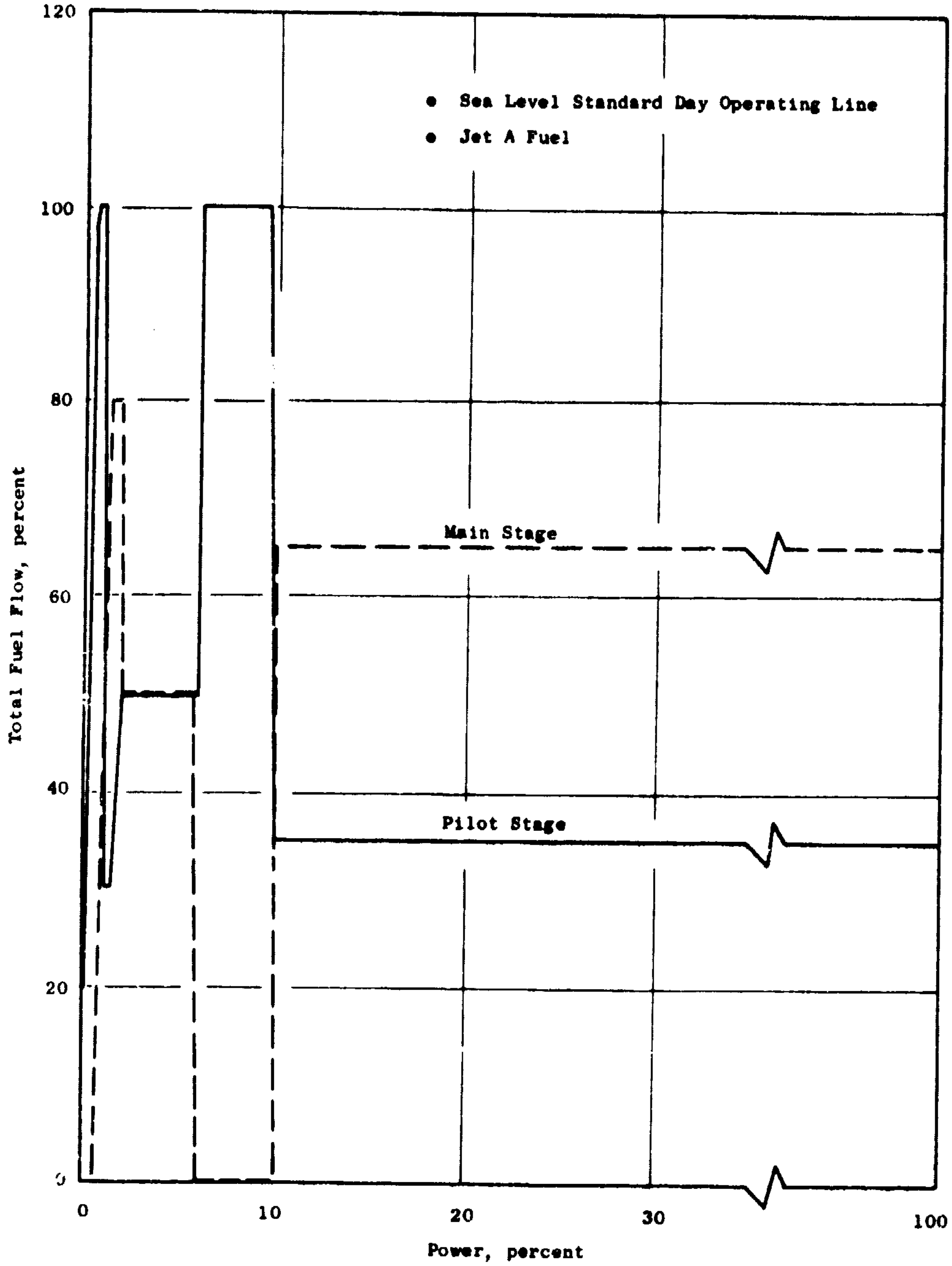

Figure 2.3-2. Fuel Flow Staging.

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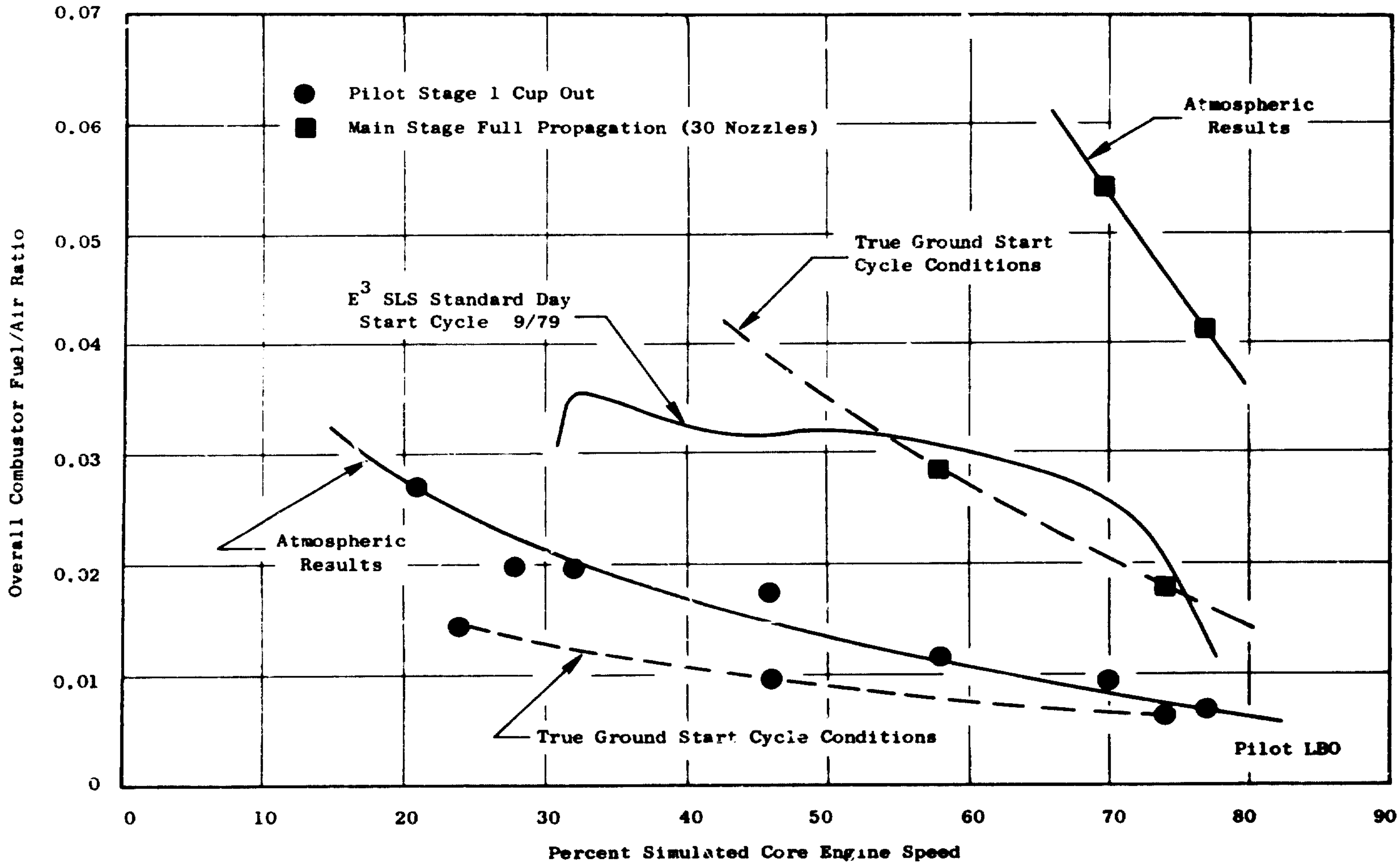

Figure 2.3-3. Development Combustor Mod I Main Stage Ground Start Ignition.

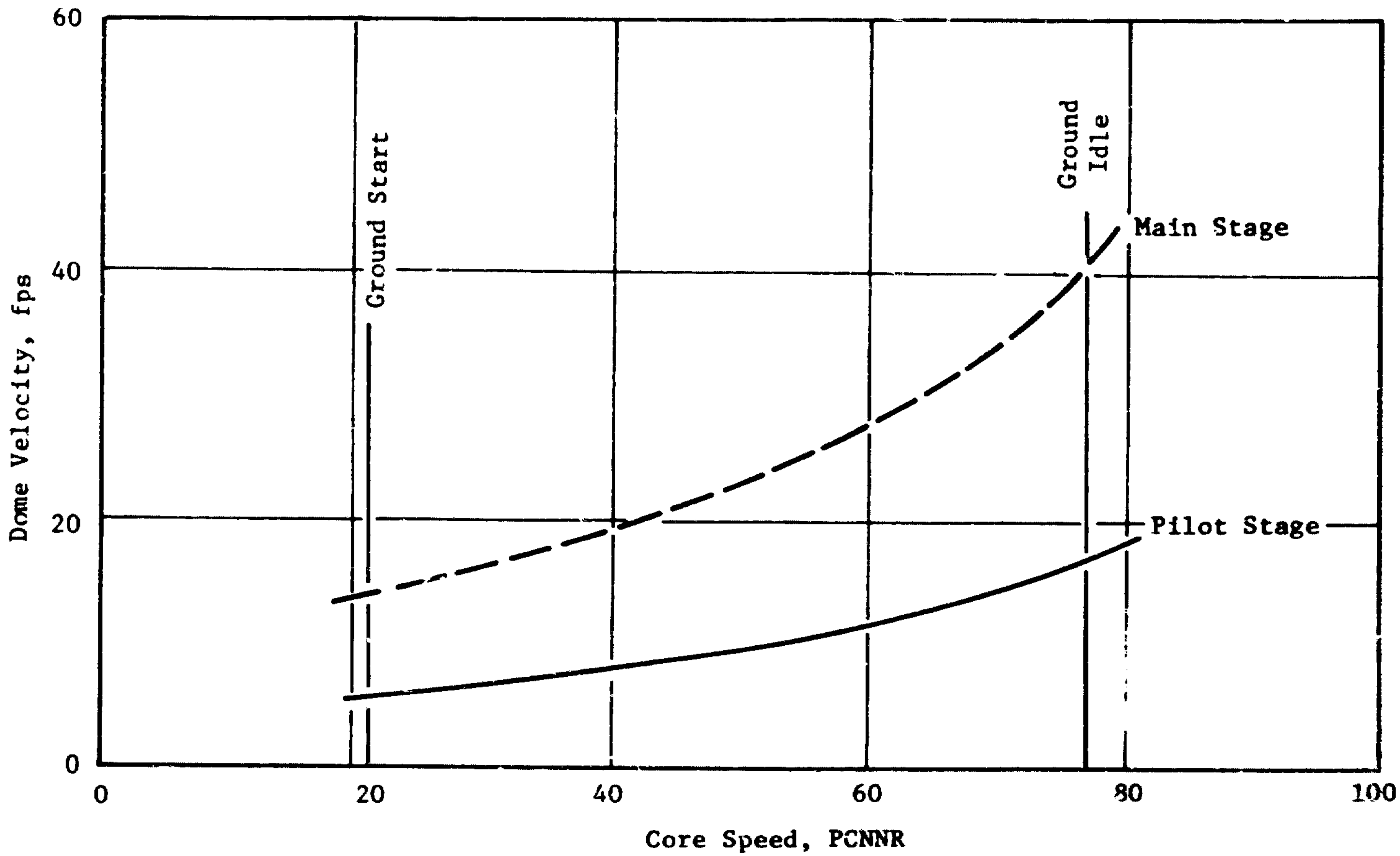

Figure 2.3-4. Double Annular Combustor Dome Velocity Comparison, SLS Standard Day, Mod I Configuration.

However, due to the extreme modifications to the main stage dome airflows, NO_x emissions were expected to increase. The key design changes identified for the Mod. II combustor configuration were as follows:

- The pilot stage dome flows were reduced to obtain lower CO and HC emissions levels, as well as reduce the pilot stage incipient lean extinction limit, thus providing increased fuel/air ratio margin for main stage ignition. This airflow reduction was accomplished by reducing the pilot stage swirl cup airflow. In order to maintain overall combustor pressure drop, the pilot stage dilution and centerbody cooling flows were increased proportionate amounts.

- To improve the main stage ignition, by reducing dome velocity, the main stage swirl cup airflow was reduced substantially. The reduced airflow in the main stage dome again was offset by increasing main stage centerbody cooling, main stage primary dilution, and introducing an additional row of dilution in Panel 2 of the inner liner.

- To improve the mechanical integrity of the centerbody and increase cyclic life of this component, the centerbody multijet cooling ring was shortened 0.70 inch. Based on subcomponent test results, shortening of the centerbody by this small amount had no adverse impact on emissions of ignition performance.

Following completion of the hardware modifications, detailed measurements were made to determine the various combustor airflow orifice sizes in the regions modified. As in past design modifications, these flow area data are then used to determine the expected airflow distribution as well as pressure drop characteristics at the prescribed combustor inlet operating conditions. A comparison of the flow areas obtained for the Mod. I and Mod. II designs are shown in Table 2.3-III.

To obtain a more accurate definition of the combustor aerodynamics expected during component test which account for passage pressure variations, frictional losses, and hole flow coefficient variations, an airflow analysis was conducted using a machine program called Combustor Analysis (COBRA). This program accurately calculates all of the combustor flow parameters based on a given set of combustor inlet flow conditions. A comparison of the analytical results obtained for the airflow distribution for the Mod. I and Mod. II development combustor configurations is shown in Figure 2.3-5.

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Table 2.3-III. Flow Check Results Mod. I and Mod. II Combustor Hardware.

	Effective Area, in.^{2}			
	Measured Mod. I	Measured Mod. II	Design Mod. II	% Difference from Design
Outer Liner	6.53	7.21	7.02	+ 2.7
Inner Liner	11.42	16.59	16.71	- 1.0
Centerbody	6.00	7.18	7.55	- 4.9
Outer Dome	11.48	12.61	10.76	+17.2
Inner Dome	18.12	15.01	14.40	+ 4.2
Total Combustor	53.71	58.55	56.45	+ 3.7

In addition to defining the design definition for the development combustor, an aerodynamic design definition was evolved for the core engine combustor. Pending satisfactory performance of the Mod. II development combustor, the core engine hardware designs could then be released with minimum delay thereby enhancing timely delivery of the core engine hardware. The hardware definition for the core engine combustor was evolved utilizing an aerodynamic analysis program called SODAC (Single Or Double Annular Combustor) which utilizes a known airflow distribution as might be obtained from analytical output such as COBRA and generates the required flow areas needed to obtain the desired airflow distribution. The selected flow distribution and resulting flow area definition for the core engine combustor are shown in Table 2.3-IV.

Table 2.3-IV. Core Engine Combustor Design.

	Effective Area, in.^{2}	Flow, % Wc
Outer Liner	8.3	12.9
Inner Liner	13.2	24.0
Centerbody	8.7	18.1
Outer Dome	10.9	20.9
Inner Dome	14.4	24.1
Total Combustor	55.5	100.0

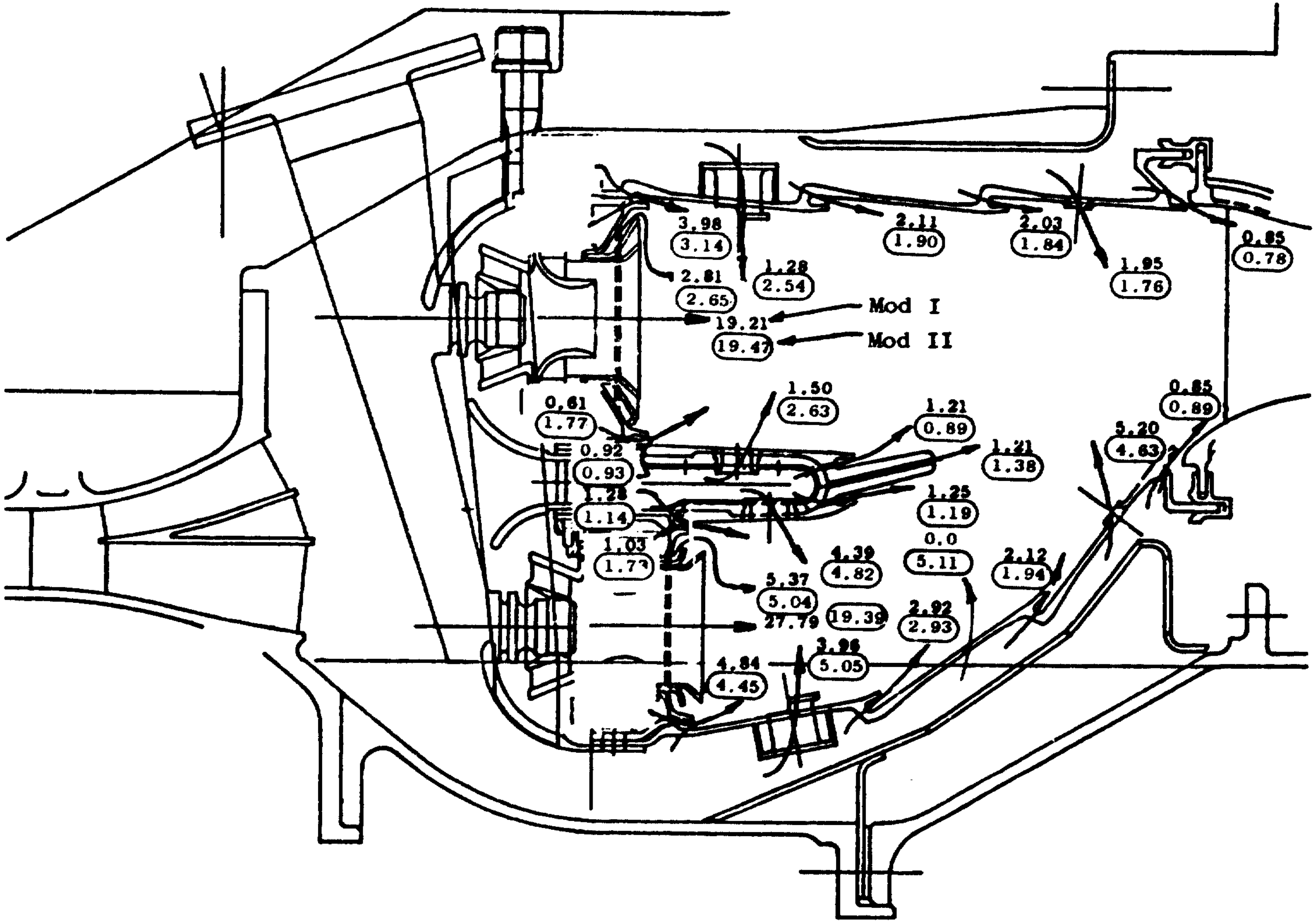

Figure 2.3-5. Airflow Distribution for Mod I and Mod II Development Combustor Designs.

As in the earlier combustor test program, the procedure is to first conduct atmospheric ignition tests followed by atmospheric performance evaluation, then proceed to pressure and emissions testing. The results of the Mod. II development combustor ignition test are provided in detail in Section 2.3.4.3. In general, the main stage ignition performance improvements expected were not realized and, furthermore, pilot stage ignition deteriorated from its previous excellent levels. Visual observations during the test indicated that the dilution flow in the second inner panel penetrates across the end of the centerbody and tends to quench the pilot zone. Therefore, the inner Panel 2 dilution was blocked. As expected, this modified configuration (Mod. IIA) had a slightly different airflow distribution and higher pressure drop compared to Mod. II, as shown in Figure 2.3-6, due to the change in combustor total flow area. The test results for this modification showed significant improvements in pilot stage ignition but no major improvement in main stage ignition, due in part to the undesirable increase in main dome airflow resulting from the total combustor area change. Two phenomena were observed during the Mod. IIA ignition test which might explain the poor main stage ignition performance obtained. First, penetration of the pilot zone flame through the crossfire tubes appeared to be reduced compared to earlier tests and the flame appeared to be swept rapidly downstream. Second, once ignition was obtained in the swirl cups opposite the crossfire tubes, propagat` to adjacent swirl cups was difficult indicating a narrow fuel spray angle or poor stabilization in the combustion zone. As shown in Table 2.3-V, the result of the Mod. II design changes was to make the main stage dome velocity similar to the pilot stage dome. However, the introduction of added cooling in the centerbody increased the ring annulus exit velocity to values higher than in any other configuration tested. This higher cooling ring exit velocity may have also suppressed the swirl cup fuel spray angle. Therefore, additional modifications to the development combustor were identified to eliminate or offset these local effects.

The key changes introduced into the Mod. III were as follows:

- Reduce main stage swirl cup primary swirler to induce a wider fuel spray angle, enrichen the main stage swirl cup, and further reduce main stage dome velocity, as shown in Table 2.3-V

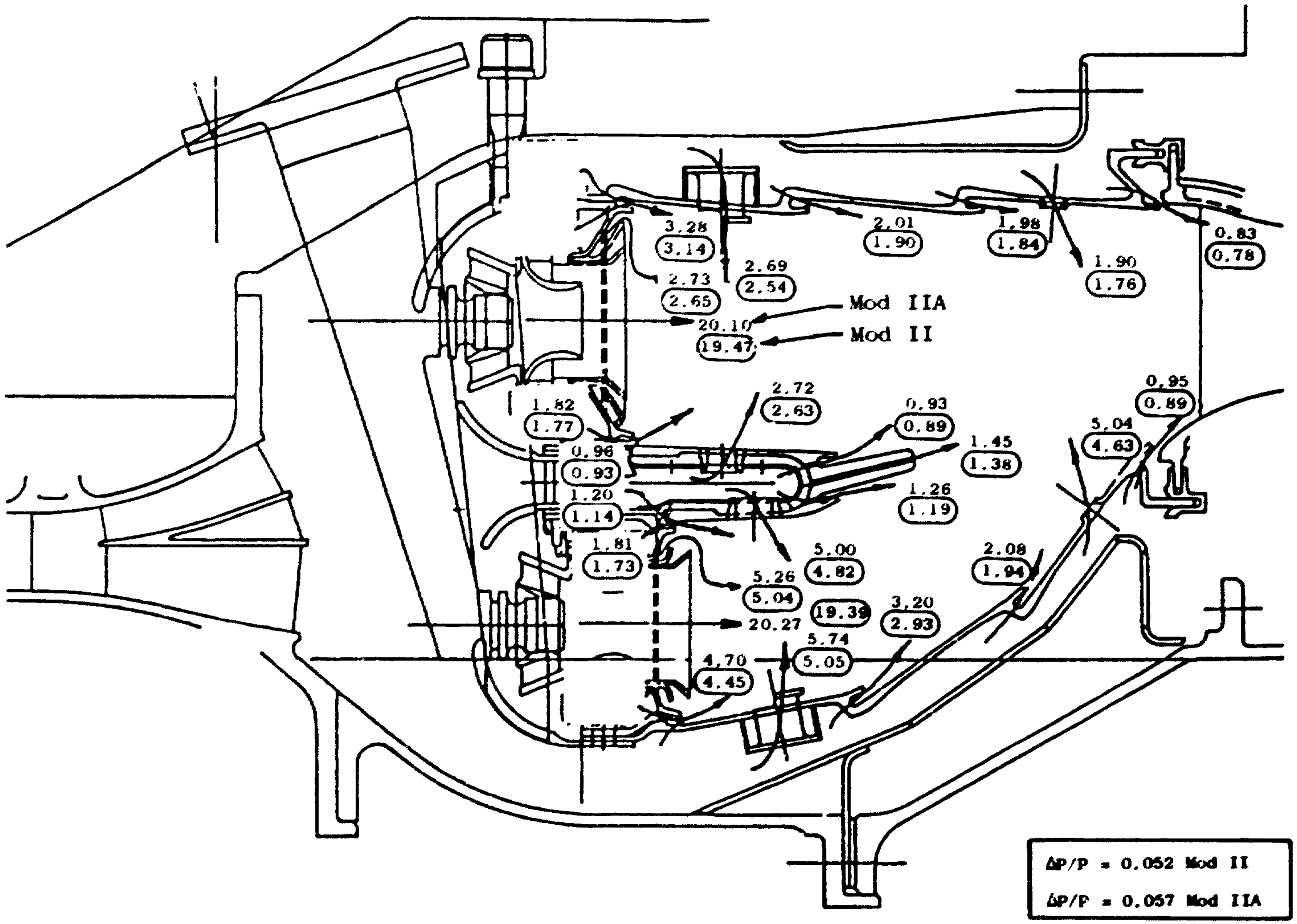

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Figure 2.3-6. Airflow Distribution for Mod II and Mod IIA Development Combustor Designs.

- Reduce main stage splashplate cooling to further reduce spray angle suppression and dome velocity

- Reduce main stage dome outer cooling ring airflow uniformly to reduce spray angle suppression and incorporate additional flow reductions locally in the region upstream of the crossfire tubes

- Modify crossfire tubes to provide a sheltered region at the forward position to enhance pilot zone flame penetration into the main zone

- To compensate for the combustor flow area reductions, the aft outer and inner liner dilution holes will be increased a proportionate amount.

Table 2.3-V. Velocity Comparisons.
- 46% PCNHR
- Velocity in FPS

	Pilot Dome	Main Dome	Main Dome Cooling Ring
Baseline	13.3	19.7	31.5
Mod. I	9.3	21.3	35.3
Mod. II	9.7	17.8	47.9
Mod. III	9.3	14.3	32.0

The resulting airflow distribution expected from this modification is shown in Figure 2.3-7 and compared to the Mod. II design at ground-start ignition combustor inlet conditions. These design changes are expected to provide the desired dome environment to obtain satisfactory ignition of the main stage during ground start.

Due to the major revisions in the development combustor airflow distribution and modifications to the hardware definition, the core engine hardware design definition has been placed on hold until the desired performance results for the development combustor can be confirmed in the component tests. The preferred core engine aerodynamic design definition and associated hardwear configurations will then be released for incorporation into the core engine combustor hardware.

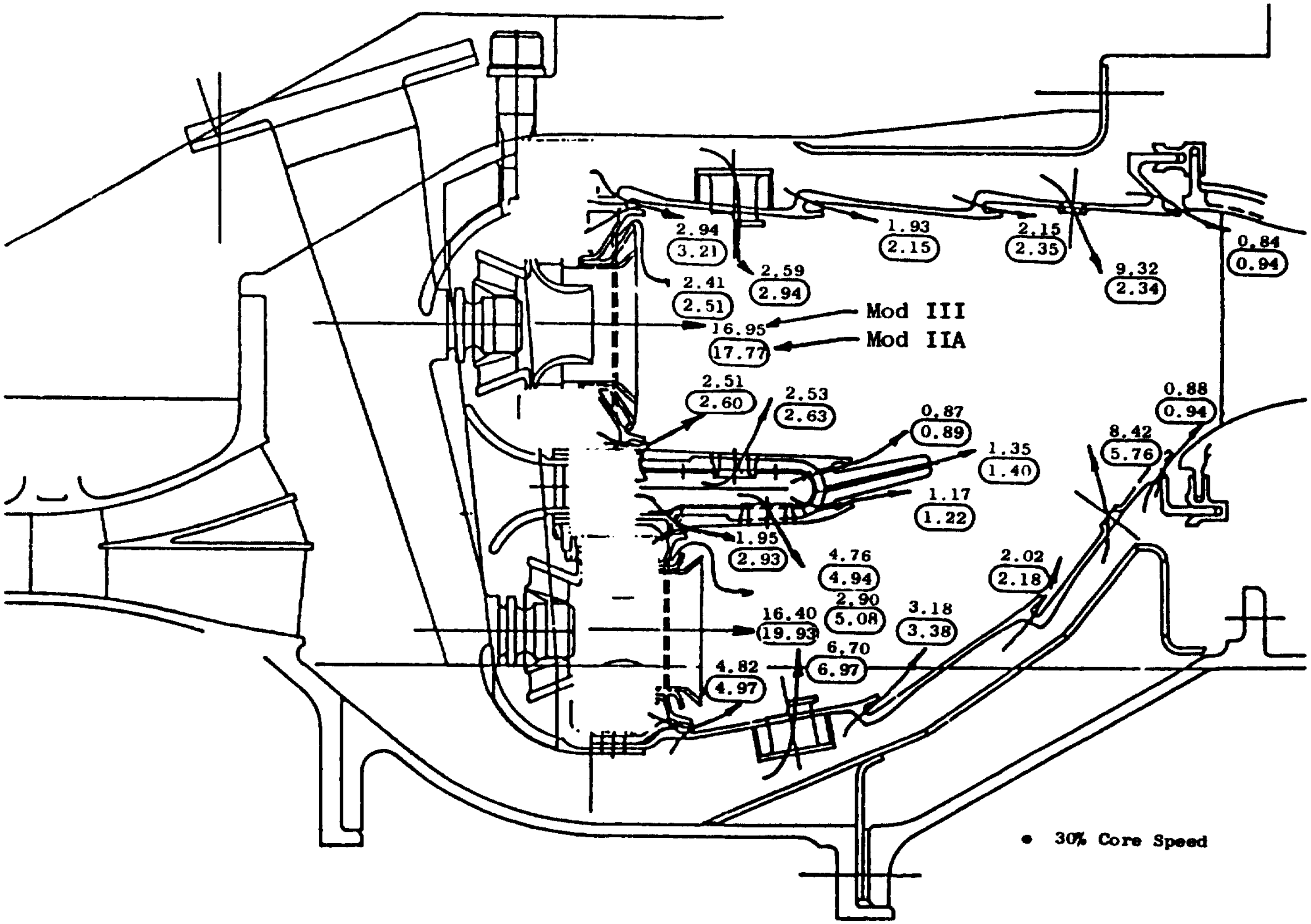

Figure 2.3-7. Airflow Distribution for Mod III and Mod IIA Development Combustor Designs.

Work Planned

- The results of the component tests of the Mod. III development combustor configuration will be analyzed in detail. Based on the findings from the combustor test data analysis and aerodynamic design studies, either additional modifications to the combustor will be investigated or the preferred design configuration will be evolved for the core engine combustor.

- Upon receipt of the core engine hardware, the necessary pretest inspections and aerodynamic analysis will be conducted to ensure that the core engine combustor hardware meets the design intent.

- An extensive detailed test plan will be evolved to provide for a thorough evaluation of the core engine combustor design and component performance.

- Conduct a detailed review of the development and core engine designs.

2.3.2.2 Component Fabrication and Test Support

Primary activity under this task has been involved with procurement of combustor hardware required for the core engine combustor. A listing of the combustor hardware items which have been manufactured to date is provided in Table 2.3-VI. The engine hardware still being manufactured is shown in Table 2.3-VII. As shown, several of these components have been placed on hold pending final aerodynamic design definition of the initial engine combustor build. The parts that are on hold are anticipated to be available in early July of this year based on aero design release in April.

Design studies were conducted to identify an improved centerbody configuration for the engine combustor. A design was identified which features thermal barrier coating to reduce centerbody panel operating temperatures and slitting of the centerbody tip to reduce the induced hoop loads in the centerbody structure. In conjunction with the centerbody design refinement studies, a vibration analysis of the centerbody tip was made. In order to increase the tip frequency above the low harmonic range, approximately 0.75 inch was removed from the centerbody tip. This results in a significantly stiffer design. The first flex frequency is 3000 Hz for the shortened design. These features will provide adequate centerbody cyclic life for the E3 combustor application. The improved centerbody configuration is shown in Figure 2.3-8.

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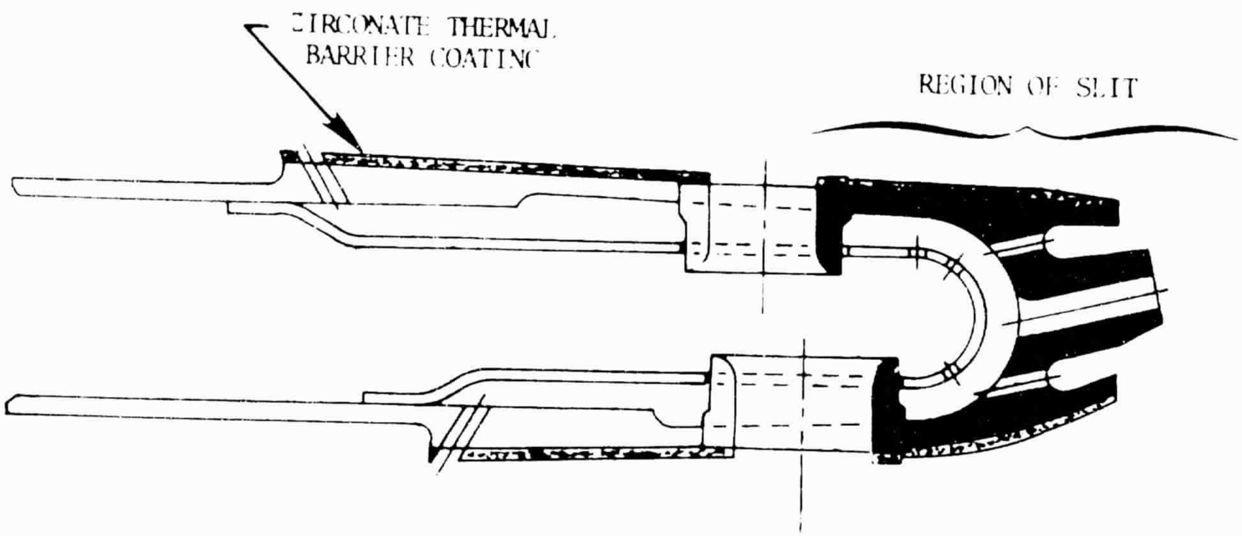
Table 2.3-VI. Available Engine Combustor Hardware.

Item	Part Number	Quantity
Outer Impingement Liner Forging	4013267-459P02	3
Inner Impingement Liner Forging	4013267-460P02	3
Combustor Support Pins	4013249-922P01	65
Outer Aft Seal	4013249-931P01	2
Inner Aft Seal	4013249-932P01	2
Centerbody Forging	4013267-586P01	2
Shingle Castings	4013249-913,914,915,911	325
Outer Liner Retainer	4013249-446P01	2
Inner Liner Retainer	4013249-447P01	2
Combustor Casing Forging	4013267-466P01	2
Inner Fuel Manifold	4013279-056G01,G02	2 ea
Outer Fuel Manifold	4013279-055G01,G02	2 ea
Primary Fuel Manifold	4013279-055G01,G02	2 ea
Primary Swirler Casting	4013267-587P01	205
Secondary Swirler Casting	4013249-442P01	205
Primary Swirler Housing Casting	4013267-585P01	205
Dilution Eyelet Casting	4013249-933P01,P02	200 ea
Fuel Nozzle Seals	4013279-001P01	125
Fuel Manifold Bracket	4013279-095G01	20
Ignitor Dilution Eyelet	4013249 [illegible]2P01	4
Shingle Retainer Rings	4013249-450P01,P02,P03	12
Instrumentation Seal	4013279-002P01	65
CDP Bleed Cover Plate	4013279-012P01	8
CDP Bleed Seal Plates	4013279-011P01	20
Ignitor Seal	4013204-109P01	10
Outer Support Liner	4013249-929P01	2
Inner Support Liner	4013249-930P01	2
Fuel Nozzle Forging	4013279-042P01	95
Ignitors	4013204-112P02	12
Ignition Exciter Box	510M52-P110	2
Miscellaneous Fasteners		4.5 Combustors
Outer Case Machining	4013297-650P01	1
Liner Shingle Machining	4013249-919,920,921,917	110

Table 2.3-VII. Engine Combustor Hardware Currently Being Manufactured.

Item	Part Number	Qty	Delivery Date
Outer Dome Assembly*	4013267-595G01	1	July 15, 1981
Inner Dome Assembly*	4013267-596G01	1	July 15, 1981
Outer Support Liner Hole Drilling*	4013249-568G01	1	July 15, 1981
Inner Support Liner Hole Drilling	4013249-567G01	1	July 15, 1981
Centerbody	4013267-589G02	1	July 15, 1981
Liner Dilution Eyelet*	4013249-933P03,P04	60	July 15, 1981
Ignitor Leads	4013284-325	6	July 1, 1981
Cowl Assembly	4013297-594G03	1	May 1, 1981

*On hold.



- Brazed Construction
- Thermal Barrier Coated
- Slit for Stress Relief
- Adequate Stiffness (3000 Hz)
- Difficult Tip Hole Drill Eliminated

Figure 2.3-8. E^3 Engine Centerbody Configuration.

Work Planned

Manufacture of the core engine combustion-system components will continue to be monitored. Revisions in combustor cooling and dilution flow levels will be incorporated into component manufacturing drawings and the remaining engine combustor components will be released.

2.3.3 Subcomponent Tests

The sector combustor activities during this reporting period were directed at three areas of investigation. First, three sector modifications (Mods. III, IV, and V) were evaluated for ignition and emissions performance improvements. Second, the design definition of the desired fuel nozzle shroud air characteristics relative to emissions performance was identified for incorporation into the core engine fuel nozzles. Finally, test efforts in the sector combustor were turned toward evaluating design modifications to the sector which would improve the ignition performance of the annular combustor main stage.

The Mod. III sector combustor configuration showed significant improvements in main stage ignition compared to the Mod. II design. At true cycle operating conditions, the main stage cross fire and the propagation limit was estimated to occur at a core speed of 40% PCNHR. Based on these encouraging results, the emissions performance of the sector combustor was evaluated next. The key design changes for Mod. III included incorporation of a centerbody with crossfire hole, reduced main stage swirl cup flow, reduced outer panel dilution, increased centerbody cooling, increased pilot trim air to offset flow reductions incorporated, and installation of development-type fuel nozzles identical to those used in full-annular testing. The resulting airflow distributions from these modifications are shown in Figure 2.3-9 and are compared to the Mod. II design.

The emissions test point schedule for low power emissions performance is shown in Table 2.3-VIII. The results obtained for the Mod. III design were disappointing in comparison to the low levels of CO and HC emissions obtained for the Mod. II design. The CO emissions for Mod. III at the 6% ground idle operating condition were about double the levels obtained for Mod. II

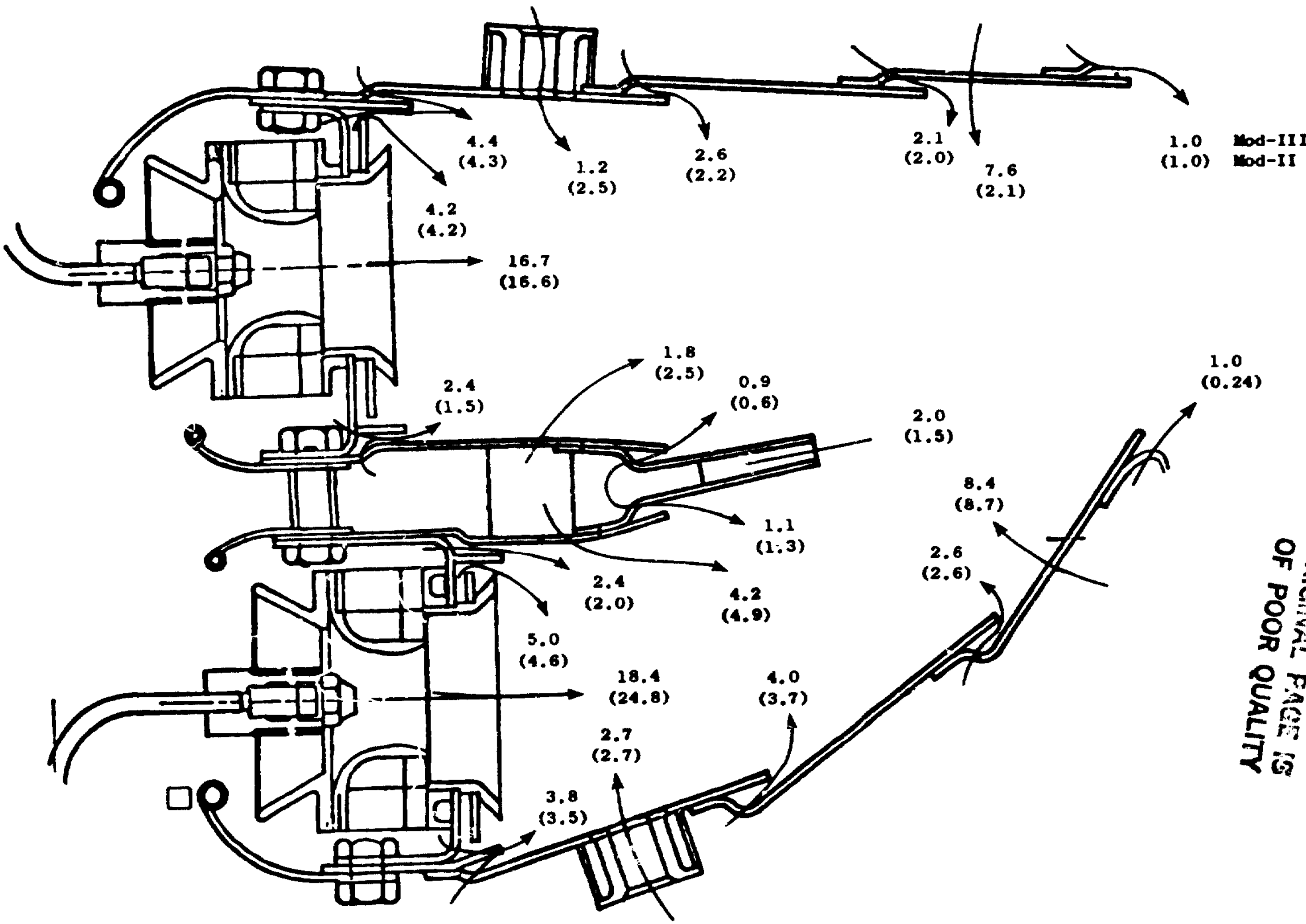


Figure 2.3-9. Sector Combustor Mod II Versus Mod III Airflow Distribution.

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Table 2.3-VIII. Sector Combustor Emissions Test Point Schedule.

Cycle Condition	P_3 psia	T3 °F	W3 pps	Wbleed pps	Wfp pph	Wfm pph	f/a Overall	W[illegible] / W_{rt}
4% Idle	50	379	3.19	0.24	106	0	0.010	1.0
	50	379	3.19	0.24	143	0	0.0135	1.0
	50	379	3.19	0.24	212	0	0.020	1.0
	50	379	3.19	0.24	265	0	0.025	1.0
6% Idle	60	435	3.85	0.29	128	0	0.010	1.0
	60	435	3.85	0.29	156	0	0.0122	1.0
	60	435	3.85	0.29	205	0	0.0160	1.0
	60	435	3.85	0.29	256	0	0.020	1.0
	60	435	3.85	0.29	320	0	0.025	1.0
30% Approach	50	685	2.96	0.22	139	0	0.0141	1.0
	50	685	2.96	0.22	42	97	0.0141	0.30
	50	685	2.96	0.22	56	83	0.0141	0.40
	50	685	2.96	0.22	70	70	0.0141	0.50
	50	685	2.96	0.22	39	59	0.010	0.40
	50	685	2.96	0.22	79	118	0.020	0.40
	50	685	2.96	0.22	99	147	0.025	0.40

as shown in Figure 2.3-10. The only major change between the Mod. II and Mod. III configurations that might produce such a drastic effect on the CO and HC emissions with the pilot stage only fueled is the fuel nozzles. The Mod. II configuration featured simple/simplex prototype fuel nozzles while the Mod. III featured the more complex development-type trumpet/airshroud/simplex fuel nozzles that have a narrower spray angle. This narrow spray angle in combination with the shroud air in the development-type fuel nozzles is suspected to be the cause of the increased emissions levels. A subsequent emissions test with the fuel nozzle shroud air blocked off was conducted at the 6% ground idle conditions only; the results showed an improvement of approximately 13% in CO and HC emissions levels, also shown in Figure 2.3-10. Consequently, the balance of the deterioration in the emissions levels is thought to be the effect of narrower fuel nozzle spray angle. However, this same narrow spray angle may also be a strong contributor to improved ignition performance obtained during ignition testing of the Mod. III configuration.

The CO and HC as well as NO_x emissions levels were also measured at simulated approach conditions (30% F_N SLTO) with the pilot stage only fueled and with both pilot and main stages fueled. With a pilot fuel flow-to-total fuel flow ratio of 0.40, the fuel/air ratio was varied between 0.0140 and 0.020 and emissions data were recorded at selected fuel/air ratios. The results are shown in Figure 2.3-11 for CO and HC emissions and in Figure 2.3-12 for NO_x emissions. The improvement in CO and HC emissions at approach conditions with both stages fueled is attributed to the significantly richer main stage dome. Conversely, the NO_x emissions have also increased slightly.

The effect of varying the fuel flow split between the pilot and main stages on the emissions levels was investigated during the test at a constant design fuel/air ratio of 0.0143. The results are presented in Figures 2.3-13 and 2.3-14 and are compared to the Mod. II configuration results. The level of emissions is less dependent on the fuel flow split in the Mod. III configuration than in the Mod. II configuration. This was expected since the dome stoichiometry in the pilot and main stage domes of the sector combustor are more nearly equal in the Mod. III configuration.

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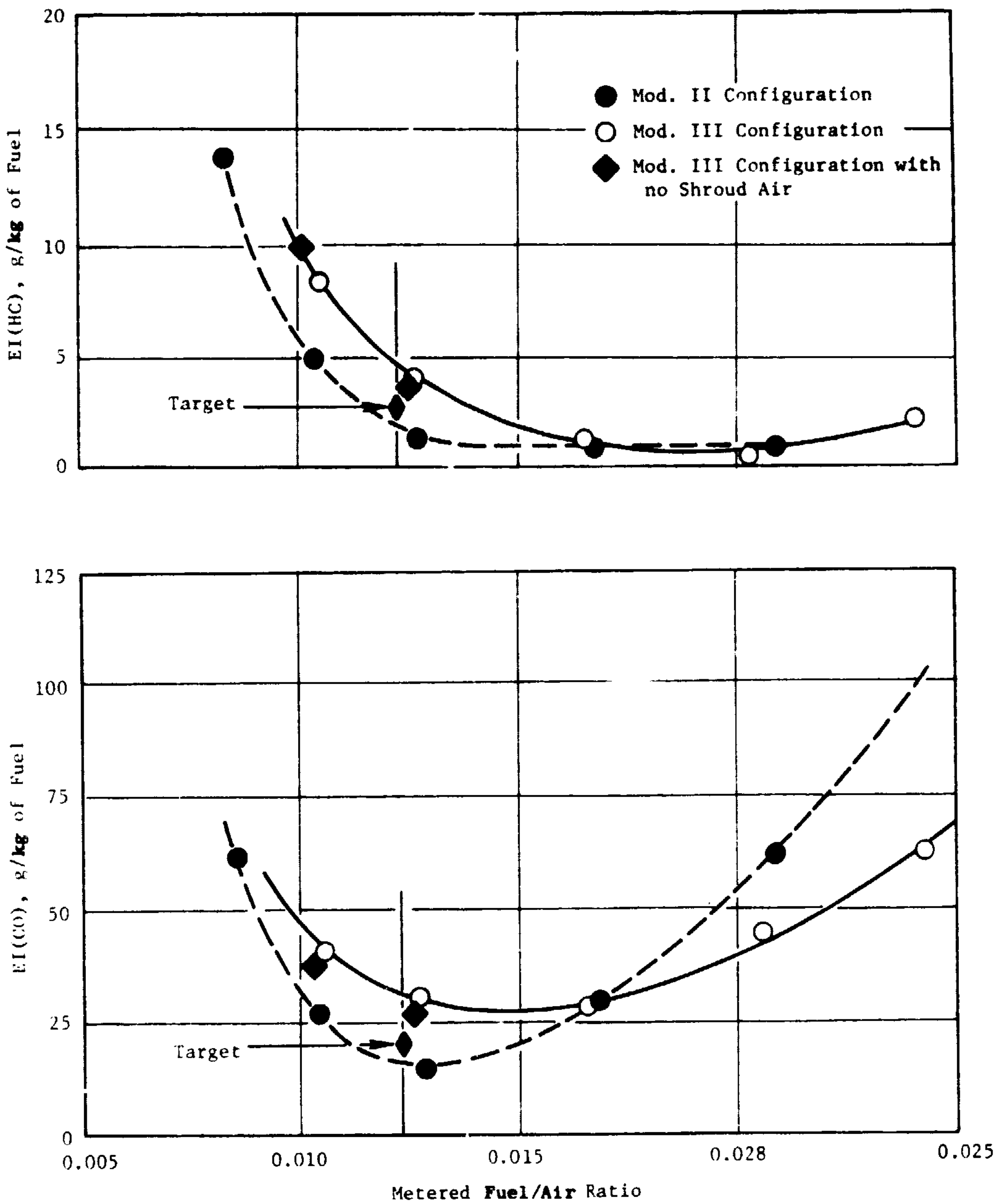

Figure 2.3-10. Sector Combustor Mod III Emissions Test CO and HC Versus Fuel/Air Ratio at 6% Idle.

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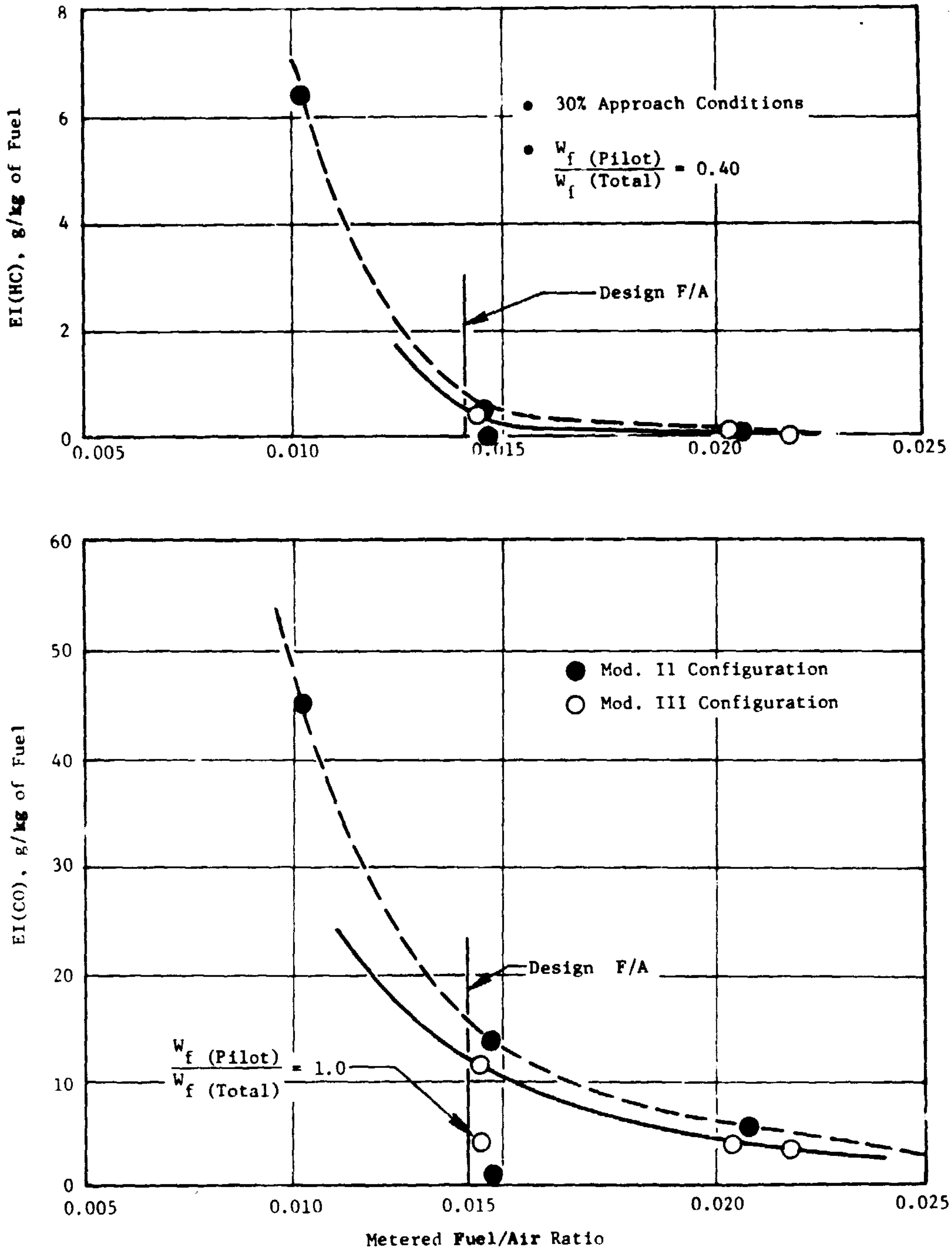

Figure 2.3-11. Sector Combustor Mod III Emissions Test CO and HC Versus Fuel/Air Ratio at 30% Approach.

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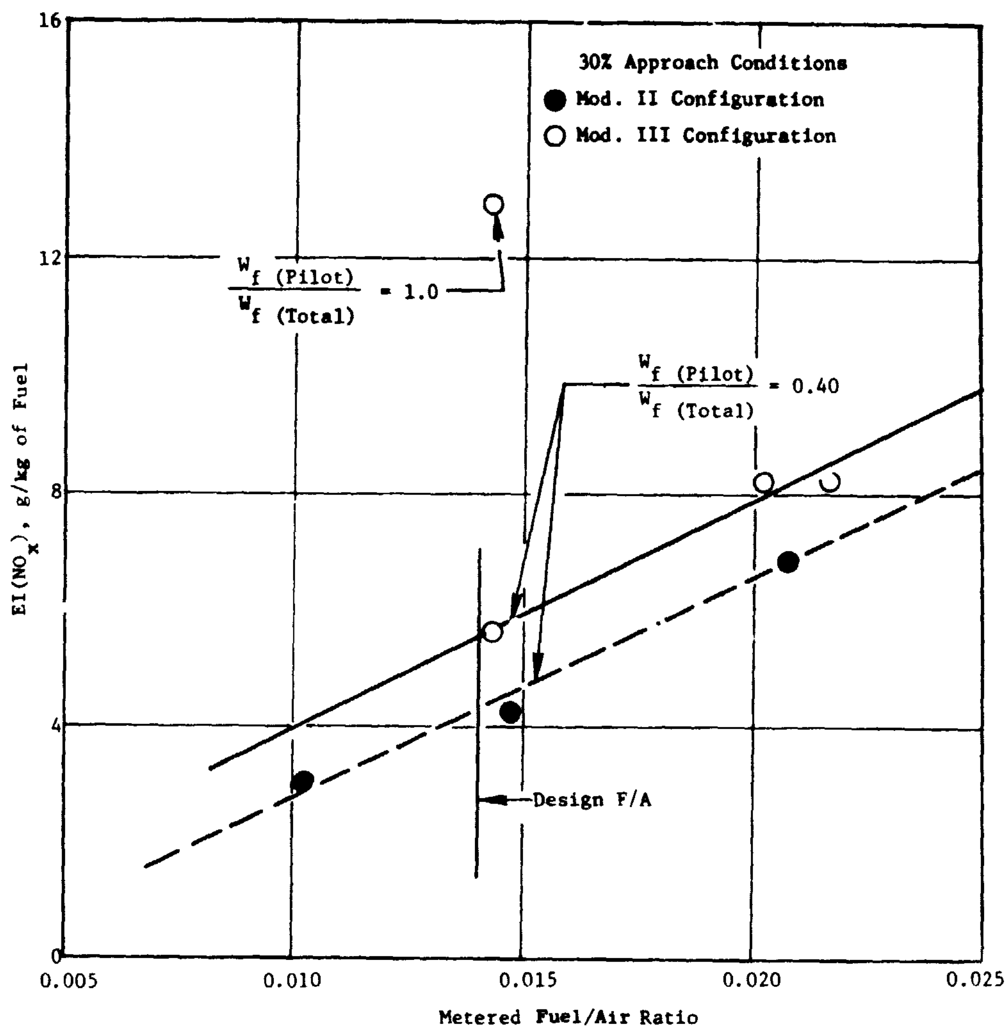

Figure 2.3-12. Sector Combustor Emissions Test NO_x Versus Fuel/Air Ratio.

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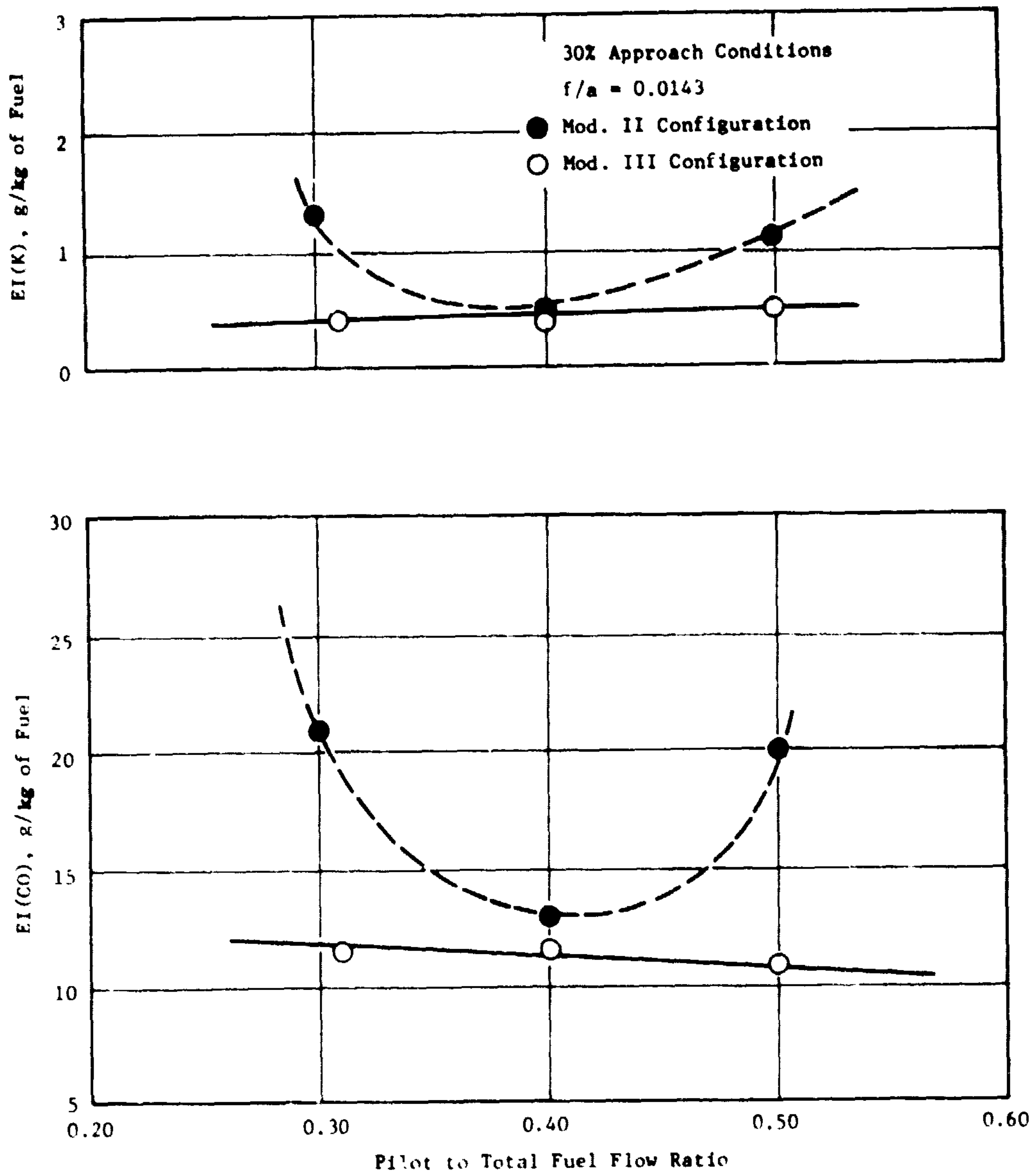

Figure 2.3-13. Sector Combustor Mod III Emissions Test CO and HC Versus Fuel Flow Split.

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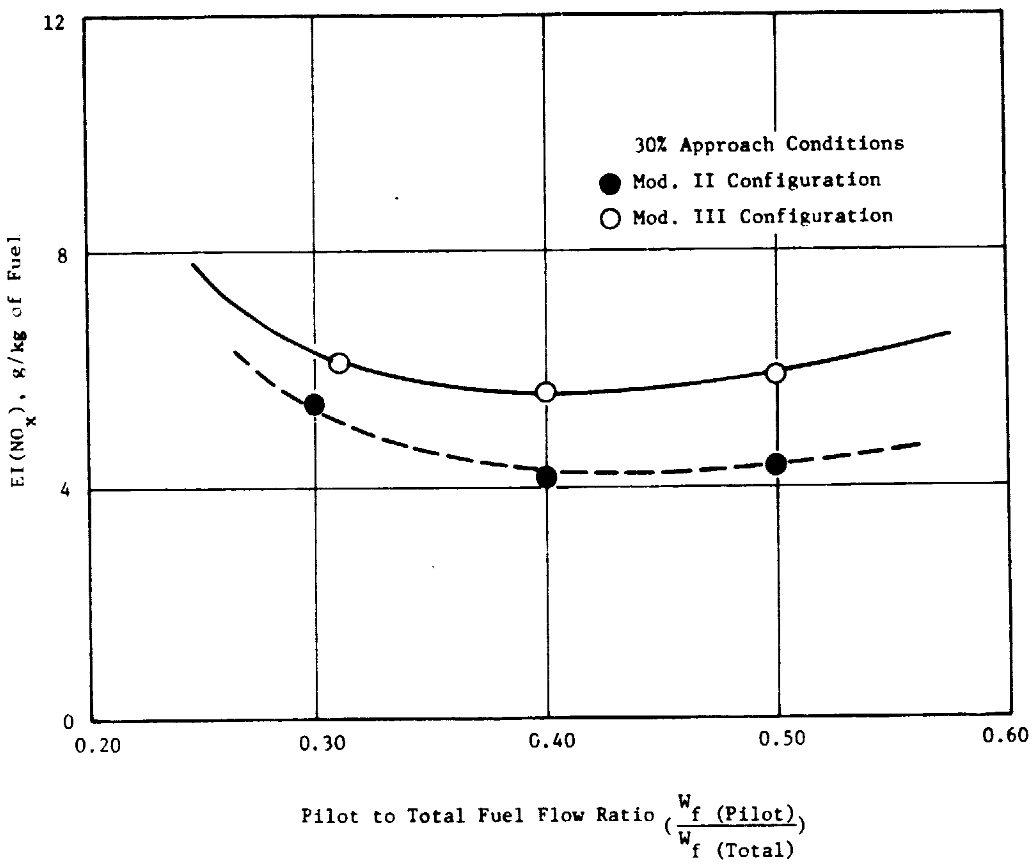

Figure 2.3-14. Sector Combustor Mod III Emissions Test NO_x Versus Fuel Flow Split.

The design changes incorporated into the Mod. III configuration of the sector combustor, specifically, the use of the narrower spray angle development-type fuel nozzles, resulted in a significant increase of the idle emissions levels relative to the Mod. II configuration. However, the use of the development nozzles resulted in improved ignition performance.

To determine the impact of the fuel nozzle design on emissions, a series of investigative tests was conducted with the Mod. III sector combustor at the 6% ground idle operating condition. The primary purpose of these tests was to determine the effects of the fuel nozzle spray angle and the fuel nozzle shroud air on CO and HC emissions levels.

The fuel nozzle features investigated included blocking off the fuel nozzle shroud air, the use of prototype simplex peanut fuel nozzles in place of development nozzles, and air shrouded, higher flow, development-type fuel nozzles. The key design features and corresponding emissions levels obtained are shown in Table 2.3-IX.

The lowest CO and HC emissions levels were obtained with the Mod. IIIB configuration that incorporated the prototype simplex peanut fuel nozzles with an 85° spray angle. However, these levels are still significantly higher than the CO and HC emissions levels obtained for the Mod. II configuration which featured the same fuel nozzles. The key difference between the Mod. II and Mod. IIIB configurations is in the pilot stage dilution where the Mod. IIIB dilution airflow level is approximately 50% of that of the Mod. II configuration. It is suspected that this low dilution airflow did not adequately penetrate the combustion zone and, hence, did not enhance mixing of the fuel and air.

The presence of an air shroud around the fuel nozzle increased the CO and HC emissions by approximately 13% as can be deduced from comparing the Mod. III and IIIA results. The shroud airflow in the pilot stage constitutes approximately 2.0% of the total sector combustor airflow raising the pilot stage swirl cup airflow level from 16.6% to 18.6% W_C when shroud air is present. This increase in the swirl cup airflow could result in the higher CO and HC emissions. However, inspection of the fuel nozzles after each test indicated that the shroud air is necessary to prevent fuel nozzle plugging and carbon buildup on the venturi. Since shroud air is required to prevent fuel nozzles and venturi carboning, it will be included in the swirl cup airflow in future tests and the desired flow level optimized.

Table 2.3-IX. Sector Combustor 6% Idle Emissions Test Results Comparison.

Configuration	Pilot Stage Swirl Cup* Flow, % W_c	Fuel Nozzle Shroud Air (2% W_c)	Pilot Stage Fuel Nozzle Flow Number	Pilot Stage Fuel Nozzle Spray Angle	EI (@ design f/a) CO	EI (@ design f/a) HC
Mod. II	16.6	No	2.5	85°	15.0	2.0
Mod. III	18.6	Yes	2.6	55°	32.0	4.8
Mod. IIIA	16.6	No	2.6	55°	29.0	4.0
Mod. IIIB	16.6	No	2.5	85°	22.0	1.6
Mod. IIIC	18.5	Yes	5.0	70°	30	2.7

- EI's are in g/kg of fuel.
- Flow number = $\frac{W_f}{\sqrt{\Delta P_f}}$ where W_f = fuel flow ~ pph and ΔP_f = fuel nozzle pressure drop ~ psi

*Includes fuel nozzle shroud air.

One of the key concerns raised for the mechanical design of the core engine combustor was the life expectancy of the centerbody, which is not segmented like the outer and inner cooling liners. The centerbody is cantilevered and heated on both sides at high power operating conditions. To improve the life characteristics of the centerbody, the trailing edge was axially segmented to reduce the thermal stresses. However, the stiffness of the multijet cooling ring was also reduced resulting in a low natural frequency that might be excited in the engine operating range and produce a high cycle fatigue due to vibration. To offset the reduced stiffness of the slotted and cantilevered multijet, a reduction in length of the trailing edge of the centerbody was investigated.

The key emphasis of the next sector combustor modification (Mod. IV) was to determine if shortening the centerbody would adversely affect ignition performance and low power emissions, specifically CO and HC emissions levels. In addition, an increase in pilot stage dilution flow was included as indicated by the Mod. III test results. A comparison of the airflow distribution between the Mod. IV and Mod. III configurations of the sector combustor along with the noted key airflow modifications is shown in Figure 2.3-15.

As in previous test investigations, prior to conducting the emissions test, a standard atmospheric ignition test as well as a pressure ignition test were conducted at selected combustor inlet conditions simulating the 9/27/79 E^3 ground start operating line. The ignition test points selected for the pressure test are shown in Table 2.3-X along with the atmospheric test point schedule. The pressure ignition test results were excellent in terms of the lightoff and lean blowout fuel-air ratios, especially at high core speeds associated with operation above atmospheric inlet pressures. The results of the pressure ignition test are compared with the results of the pressure ignition test of the Mod. III sector combustor configuration in Figure 2.3-16 for the pilot and main stages. In general, the Mod. III and IV configurations provide very similar ignition performance and would be expected to provide satisfactory ignition at core speeds above 40% PCNHR based on the measured test results.

Next, the emissions performance was investigated for the ground idle and approach operating conditions utilizing the same test conditions as for

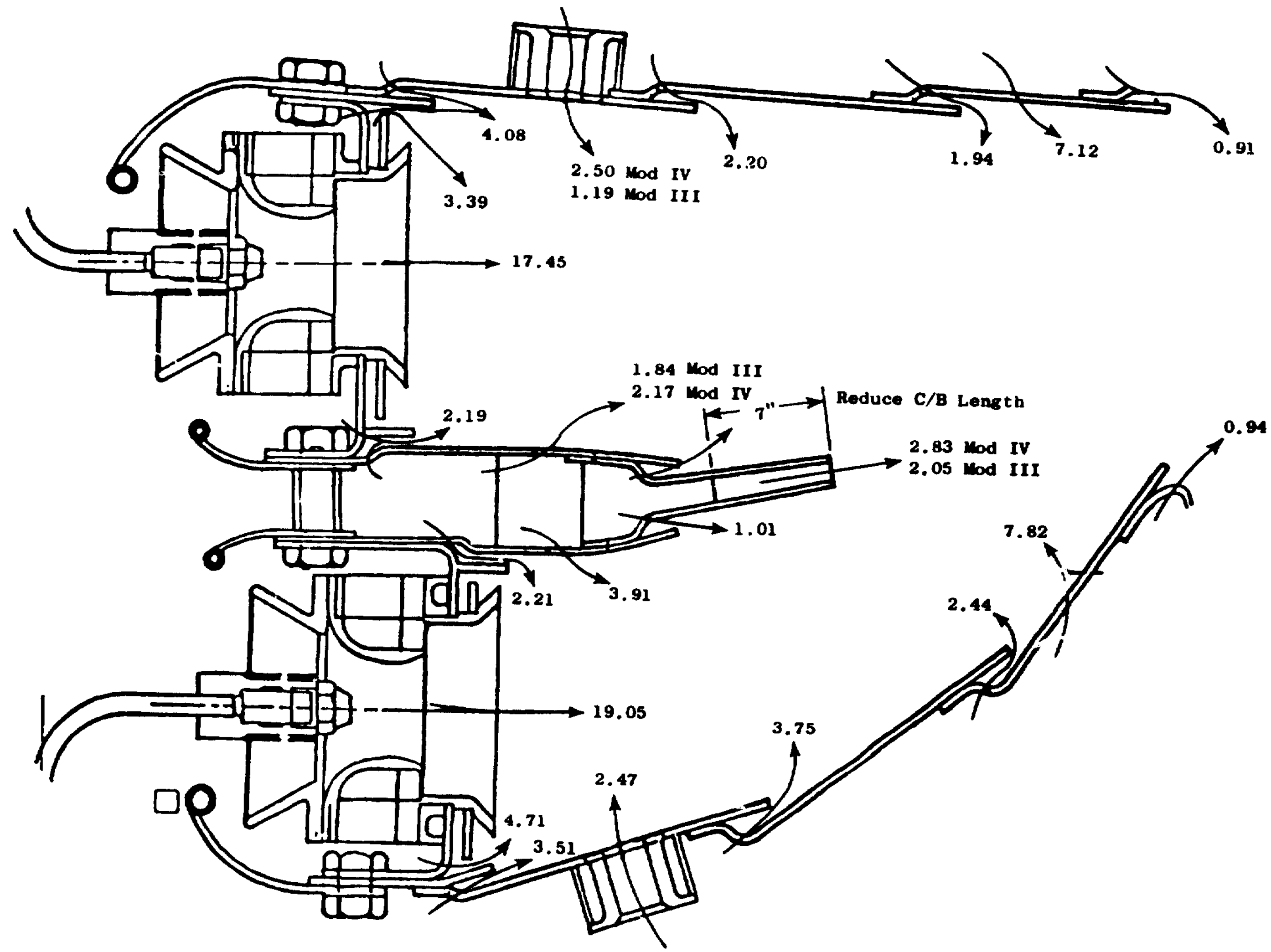

Figure 2.3-15. Sector Combustor Mod IV Airflow Distribution.

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Table 2.3-X. Sector Combustor Ignition Test Point Schedule.

- Based on E^3 9/27/79 start model
- Sector flow conditions (annular flow /6)

XNRH %	P_3 psia	T3 °F	Wc pps	$\frac{W_c \sqrt{T_3}}{P_3}$
(Pressure Test)				
21	15.0	72	0.46	4.29
58	27.0	230	1.25	7.30
70	36.2	310	1.75	8.05
(Atmospheric Test)				
21	14.7	72	0.45	4.29
32	14.7	105	0.56	5.50
46	14.7	160	0.61	6.17
58	14.7	230	0.68	7.30
70	14.7	310	0.71	8.05

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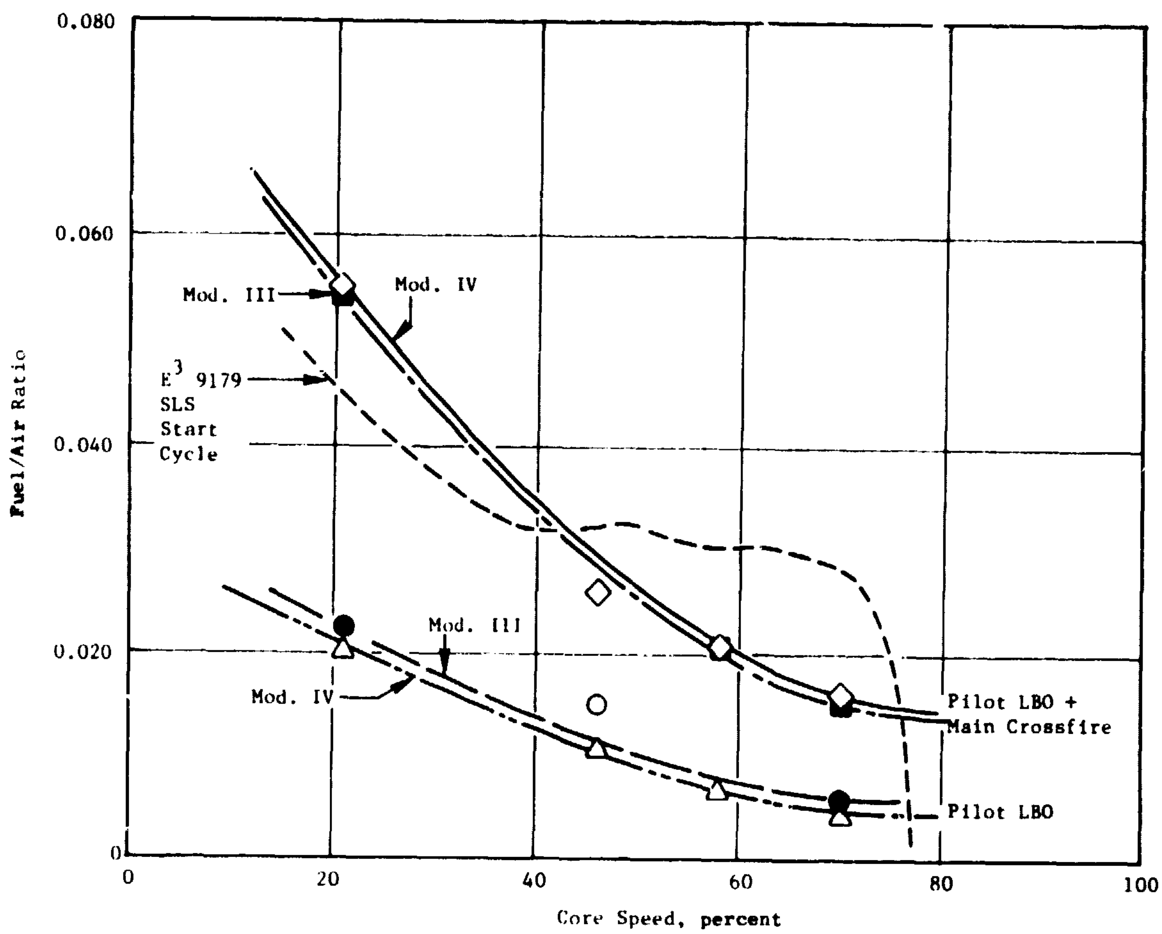

Figure 2.3-16. Sector Combustor Mod IV Ignition Test Results.

previous investigations. The test results were very encouraging in that CO and HC emission levels similar to those of the Mod. III configuration were obtained, as can be seen from Figure 2.3-17 where these two emission categories are shown versus the metered fuel/air ratio. The CO and HC as well as NO_x emissions levels were also measured at simulated approach (30% F_N SLTO) conditions with the pilot stage only fueled and with both pilot and main stages fueled. The results are shown in Figure 2.3-18 for CO and HC emissions and in Figure 2.3-19 for NO_x emissions. The results indicate a small increase in CO and HC emissions and a slight decrease in NO_x emissions compared to the Mod. III design when both domes are fueled.

One of the effects of trimming back the multijet ring was an increase in multijet airflow. This increase is due to a step size hole in the ring. Therefore, cutting back the multijet ring exposed the larger diameter hole and increased the airflow in the region. The increased CO and HC emissions and decreased NO_x emissions at the approach conditions is probably the result of quenching and mixing action from this higher airflow. This increased quenching appears to affect the main stage system more than the pilot stage since the CO emissions were not affected with the pilot stage only fueled. However, the NO_x emissions were reduced due to downstream quenching. Also, the CO emissions are higher when the fuel flow is biased to the main stage when both domes are fueled. This higher-than-desired centerbody cooling flow will not exist in the full-annular development combustor.

The design changes incorporated into the Mod. IV configuration of the sector combustor, specifically the reduction in centerbody length and increased pilot primary dilution, resulted in similar ignition performance and idle emissions levels relative to the Mod. III configuration. However, the modifications did result in higher CO and HC emissions levels and lower NO_x emissions levels at the approach power conditions.

One of the major changes to the sector combustor was the substantial reduction in main dome airflow accompanying the Mod. III configuration. However, no additional major modifications in flow area were made to offset the dome airflow reduction. Therefore, the Mod. V configuration was directed at compensating for the Mod. III main dome airflow reduction by increasing the

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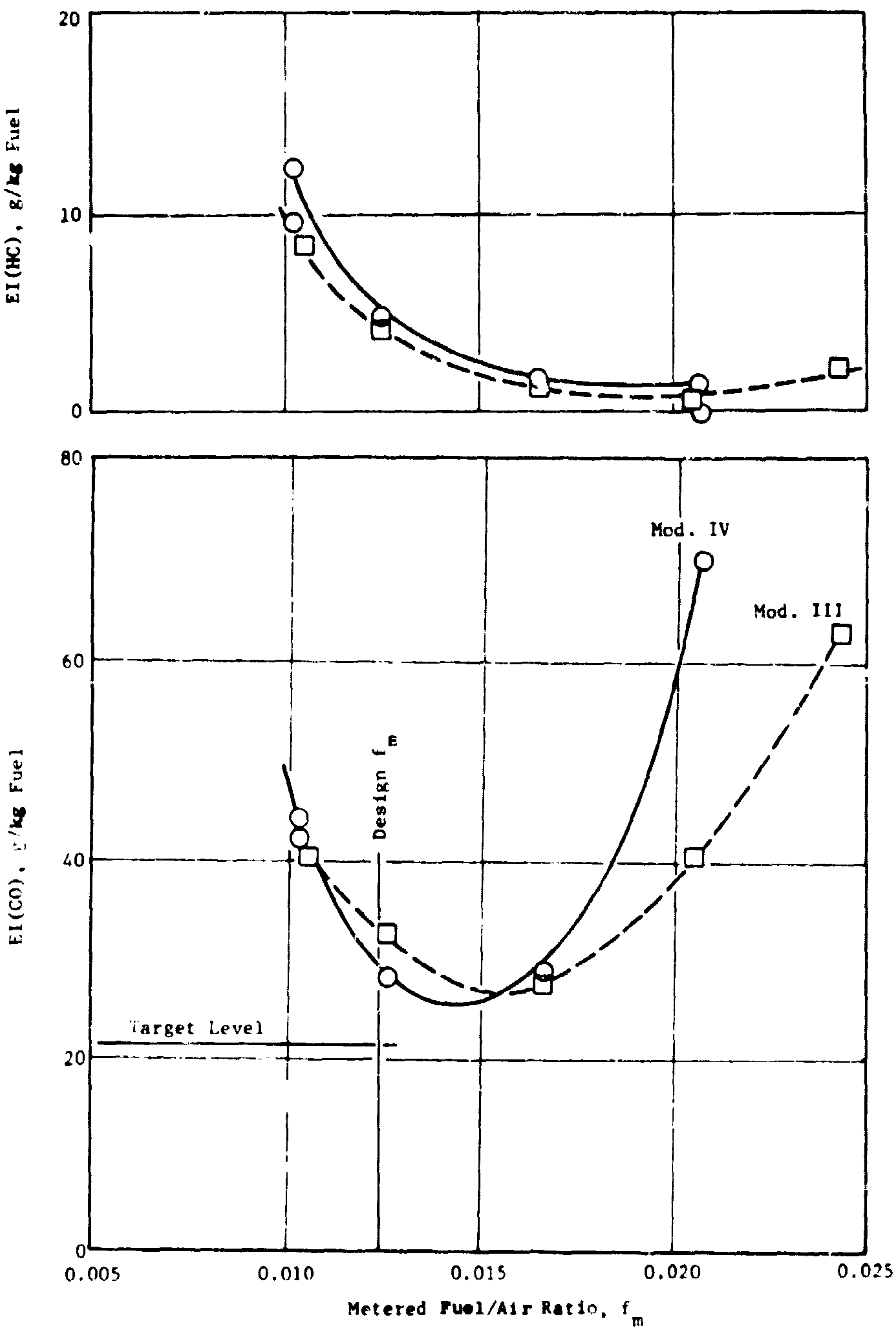

Figure 2.3-17. Sector Combustor Mod IV Emissions Test CO and HC Versus Fuel/Air at 6% Idle.

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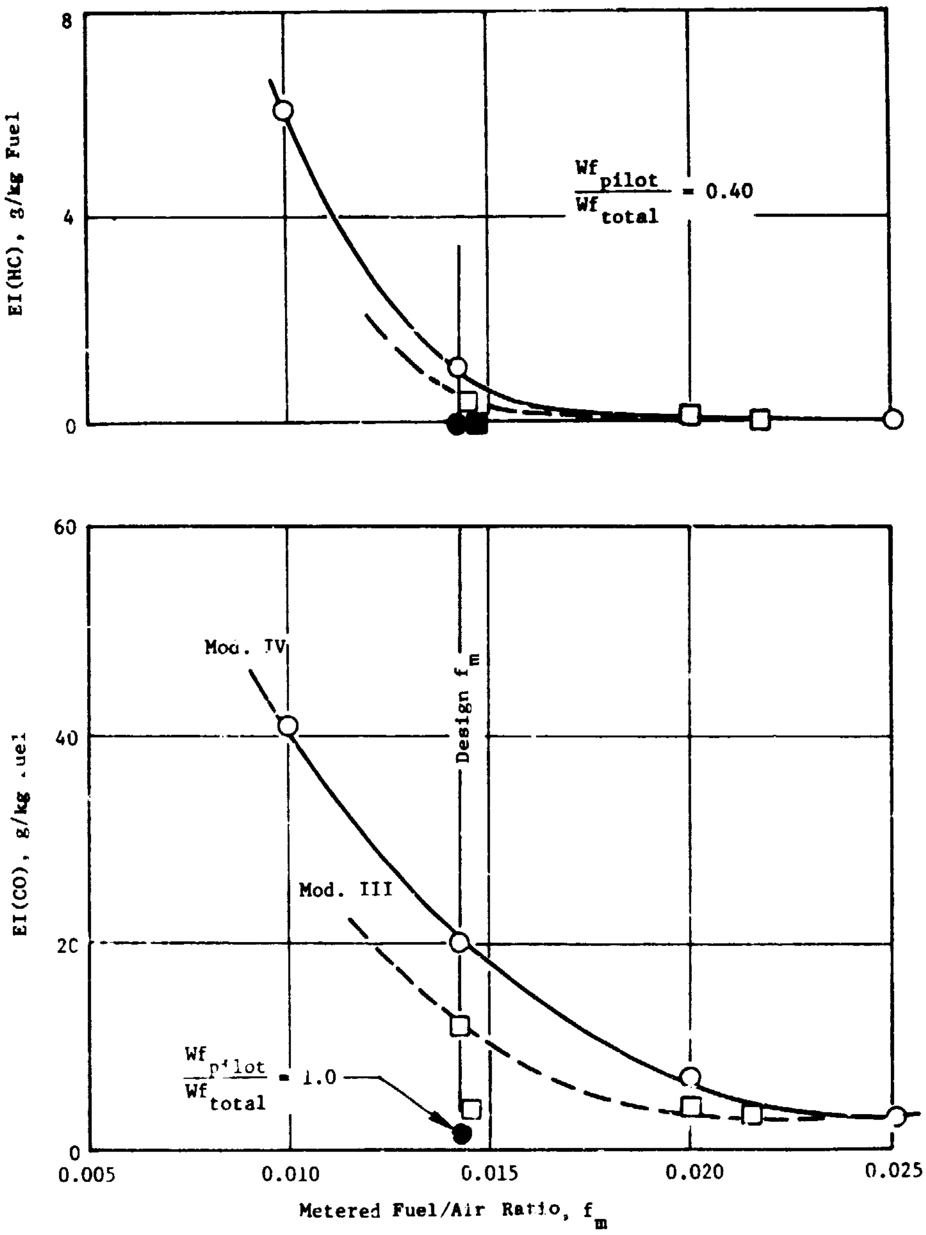

Figure 2.3-18. Sector Combustor Mod IV Emissions Test CO and HC Versus Fuel/Air Ratio at 30% Approach.

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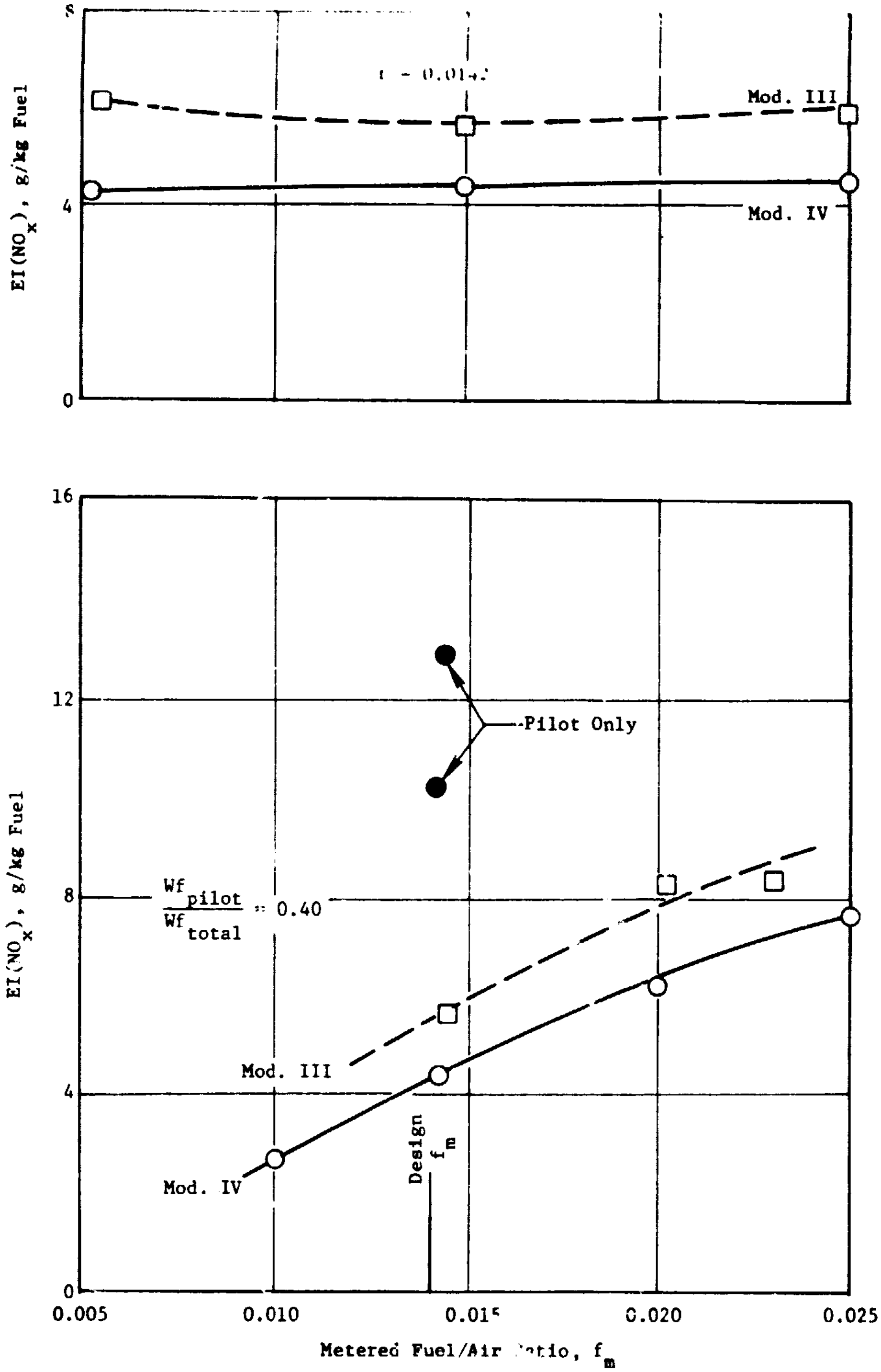

Figure 2.3-19. Sector Combustor Mod IV Emissions Test - NO_x.

main stage primary dilution airflow. The increase in main stage primary dilution hole size is expected to provide several improvements in the design. The increased flow area will result in a further reduction in swirl cup flow as a percentage of total flow. This reduction is expected to further reduce idle emissions and improve ignition performance. The increased dilution flow in the main stage is also expected to further reduce NO_x emissions. However, this reduction in emissions must not result in any deterioration in ground start ignition performance of the main stage system. In addition, the increased overall flow area should provide combustor pressure drop characteristics near the design value.

A comparison of the airflow distribution between the Mod. V and Mod IV configurations is shown in Figure 2.3-20 with key airflow modifications noted. Some change in all of the airflow quantities occurred due to an increase in the overall flow area from the Mod. IV level.

A standard atmospheric ignition test and a pressure ignition test were conducted at selected combustor inlet conditions. The pressure ignition test results were excellent in terms of lightoff and lean blowout fuel/air ratios, especially those of the main stage crossfire where a significant improvement over the Mod. IV configuration was obtained. The main stage lean blowout fuel flows for the Mod. V were too low for accurate measurement by the available flow meter. The pilot stage pressure ignition test results for the Mod. V configuration are shown in Figure 2.3-21. In general, the pilot stage ignition performance remained unchanged except at the low speed point where the lower combustor inlet temperature for the Mod. V configuration adversely affected lightoff. Figure 2.3-21 also shows the sum of the pilot stage lean blowout and main stage crossfire fuel/air ratio versus core speeds along with the 9/27/79 E^3 start schedule for both Mod. IV and Mod. V. The figure suggests that the E^3 start requirement will be met at core speeds of 24% and higher compared to 40% and higher core speeds for the Mod. IV configuration.

Next, emissions performance was investigated for the ground idle, approach, and SLTO operating conditions. The test results were very encouraging in that CO and HC emissions levels were lower than the Mod. IV levels,

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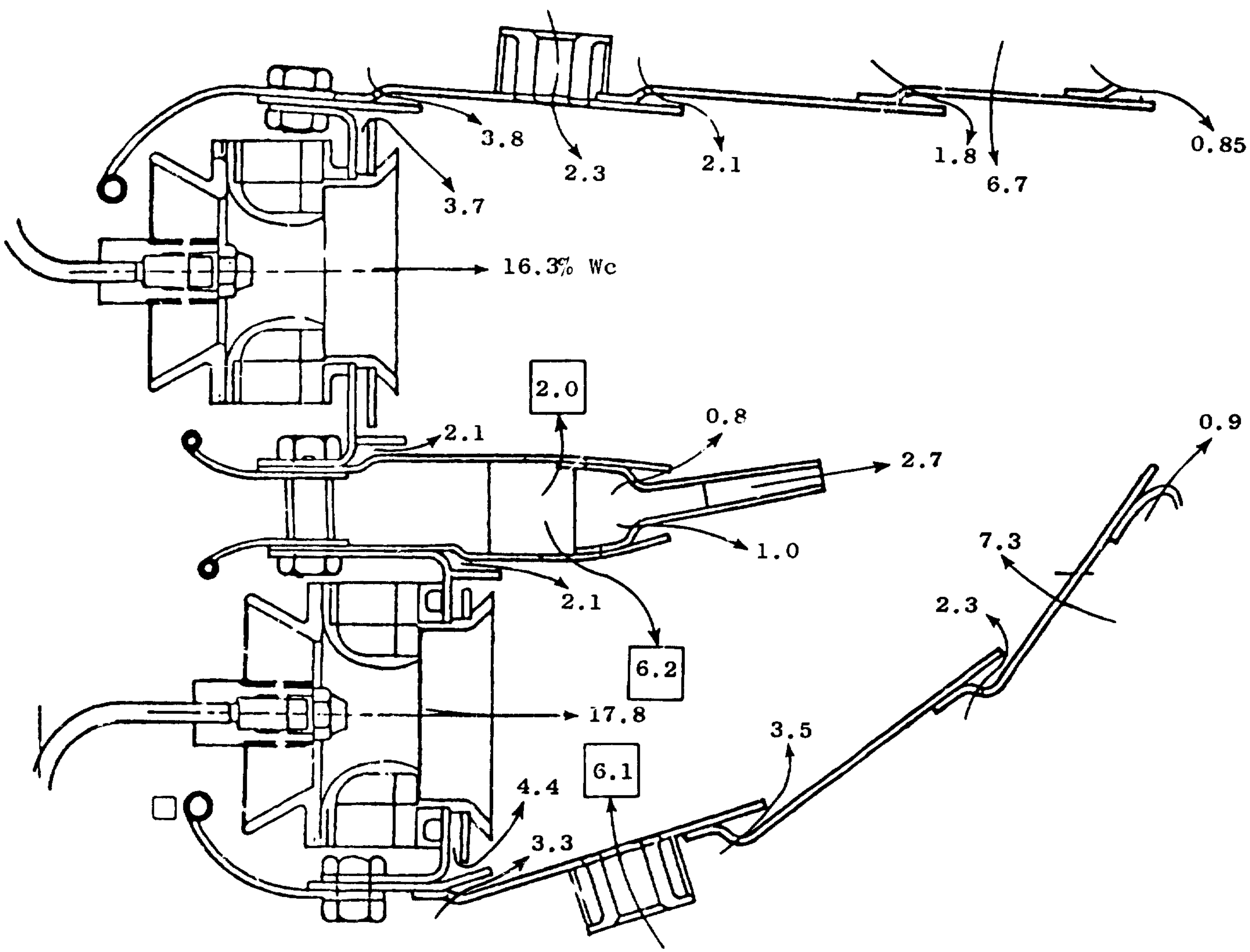

Figure 2.3-20. Sector Combustor Mod V Air Flow Distribution.

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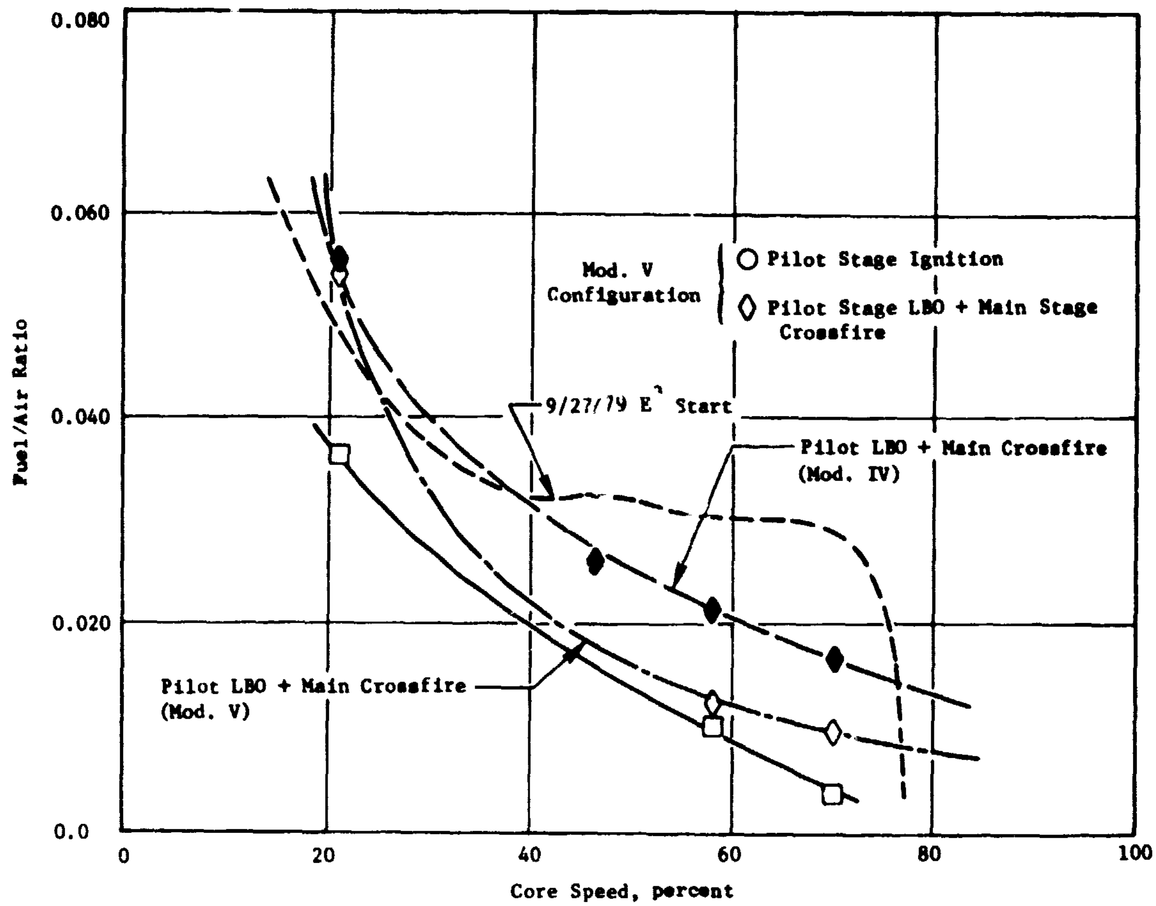

Figure 2.3-21. Sector Combustor Mod V Ignition Test Results.

and closely approached the target levels at the 6% ground idle conditions, as can be seen from Figure 2.3-22 where these two emissions categories are plotted against the metered fuel/air ratio.

The CO and HC emissions levels obtained at 30% F_N SLTO approach conditions are plotted versus the metered fuel/air ratio in Figure 2.3-23 where they are compared to Mod. IV configuration test results. As can be seen, a significant reduction in CO and HC emissions was obtained with the Mod. V configuration when both sector combustor stages are fueled. With the pilot stage only lit, the CO emissions level increased from 1.8 g/kg of fuel for the Mod. IV configuration to 3.0 g/kg of fuel for the Mod. V configuration. This is due to the fact that the change incorporated into the Mod. V configuration resulted in a slightly richer pilot stage bringing the fuel/air ratio in the pilot dome region to slightly above stoichiometry. In the Mod. IV configuration, the dome fuel/air ratio was slightly below stoichiometry. However, the CO and HC emissions levels are still extremely low.

The E^3 target levels for CO and HC emissions at approach conditions are functions of CO and HC emissions at idle conditions. A summary of the sector combustor configurations tested to date is shown in Figure 2.3-24. This figure suggests that the HC emissions will meet the E^3 target level for several of the configurations tested with either pilot only or pilot and main stages fueled. The CO emissions on the other hand, fall short of meeting the E^3 target level for this emissions category in spite of the improvements obtained with the current Mod. V configuration at both idle and approach conditions with a 40/60 fuel flow split. It is apparent from the figure that additional improvement in CO emissions at ground idle conditions is needed to meet the target level for this emission category.

In addition to CO and HC emissions, NO_x emissions were measured at simulated 30% F_N SLTO approach conditions. The results are shown in Figure 2.3-25 versus the metered fuel/air ratio for a 40/60 fuel flow split between the pilot and main stages. As expected, the NO_x emissions were slightly higher for the Mod. V configuration compared to the Mod. IV design due to the richer main stage dome.

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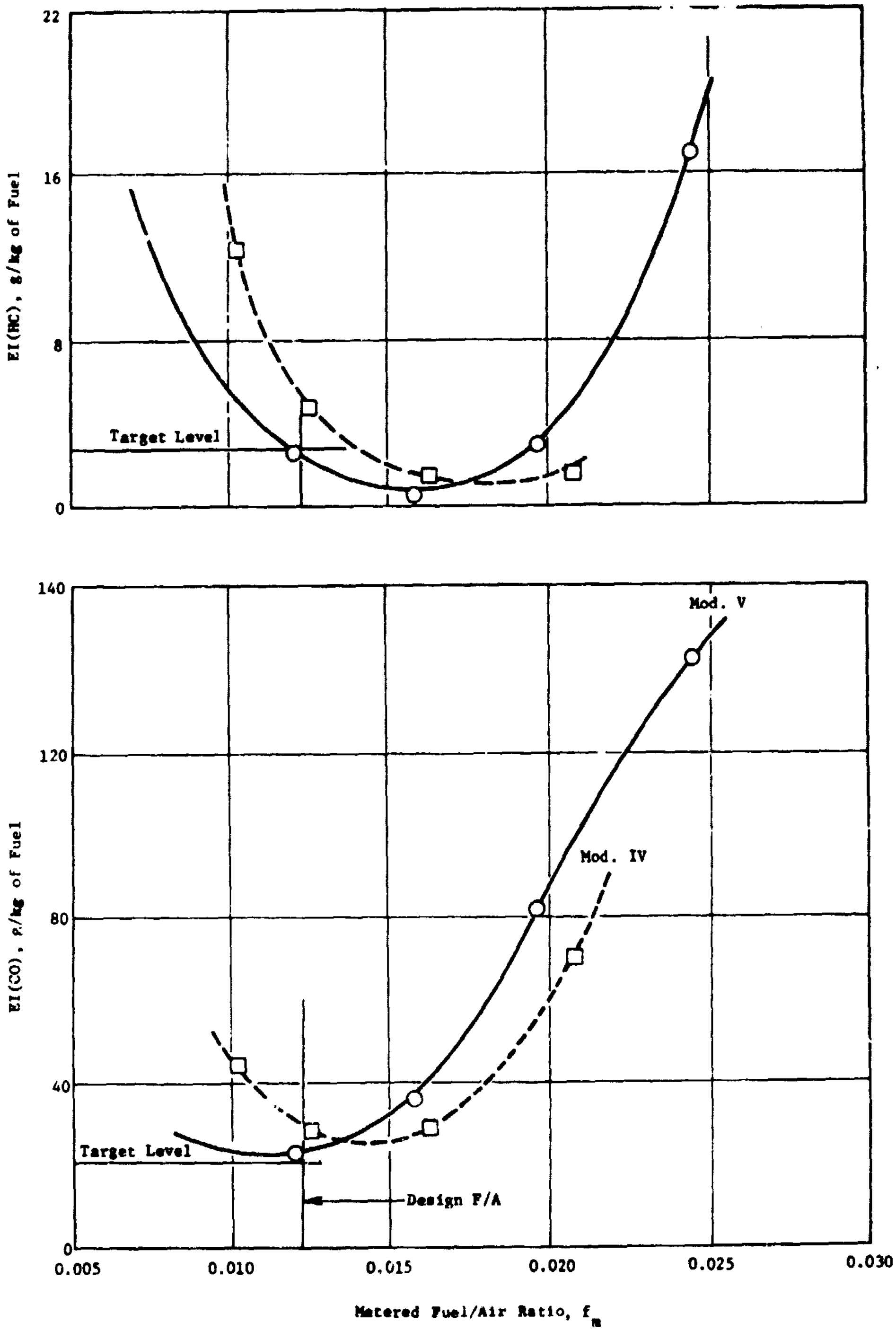

Figure 2.3-22. Sector Combustor Mod V Emissions Test CO and HC Versus Fuel/Air Ratio at 6% Idle.

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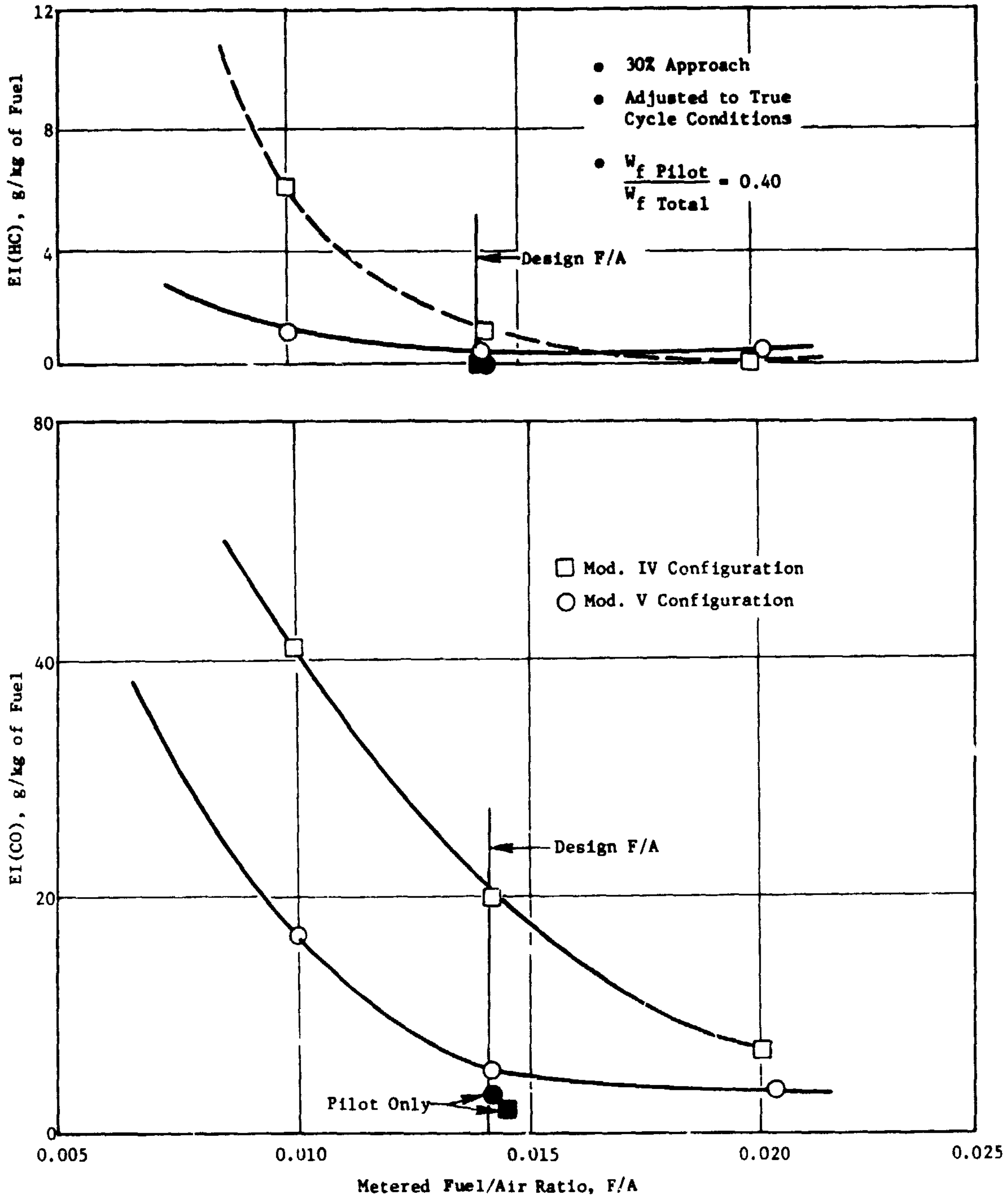

Figure 2.3-23. Sector Combustor Mod V Emissions Test CO and HC Versus Fuel/Air Ratio at 30% Appr^ach.

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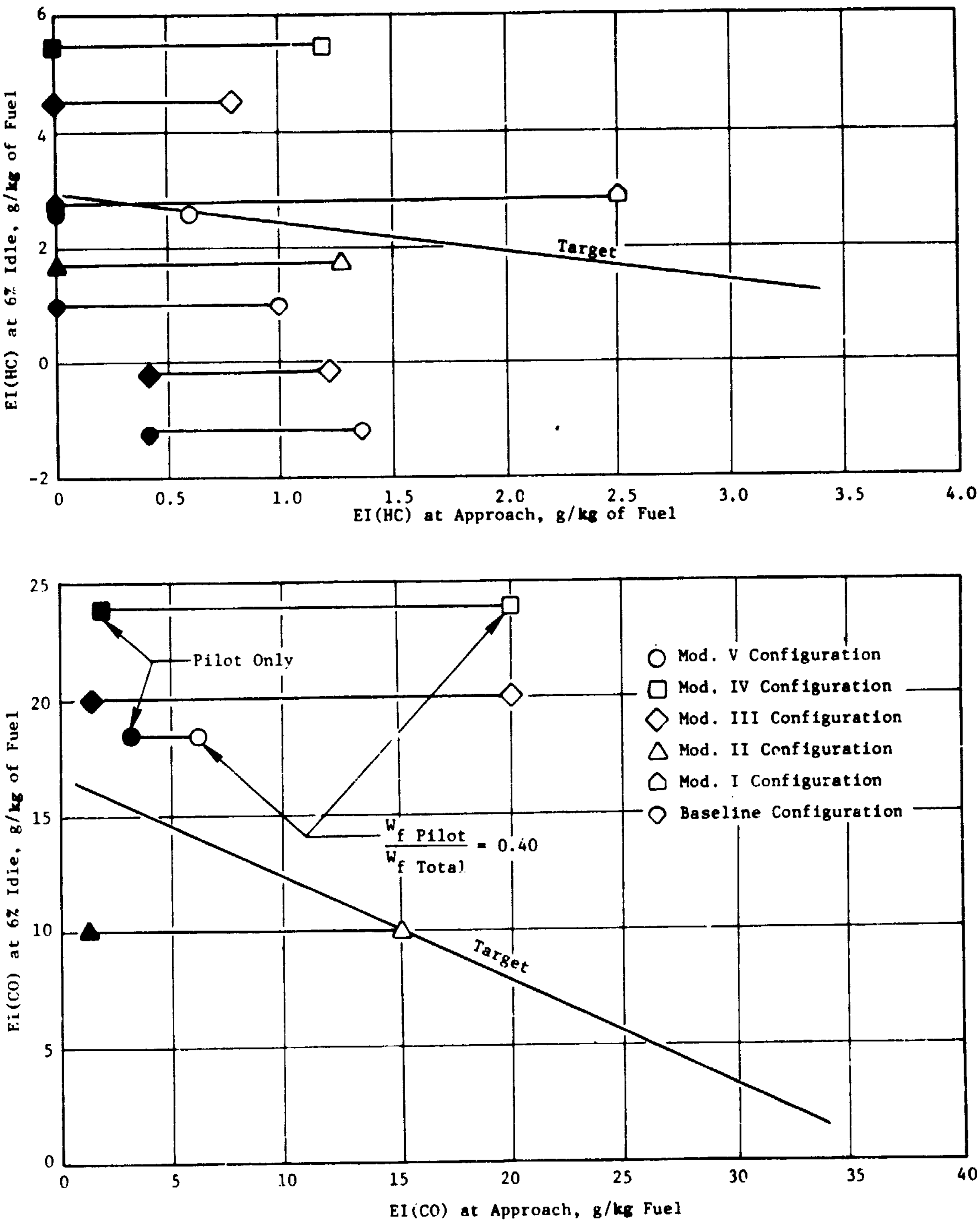

Figure 2.3-24. Sector Combustor Emissions Test Summary for CO and HC.

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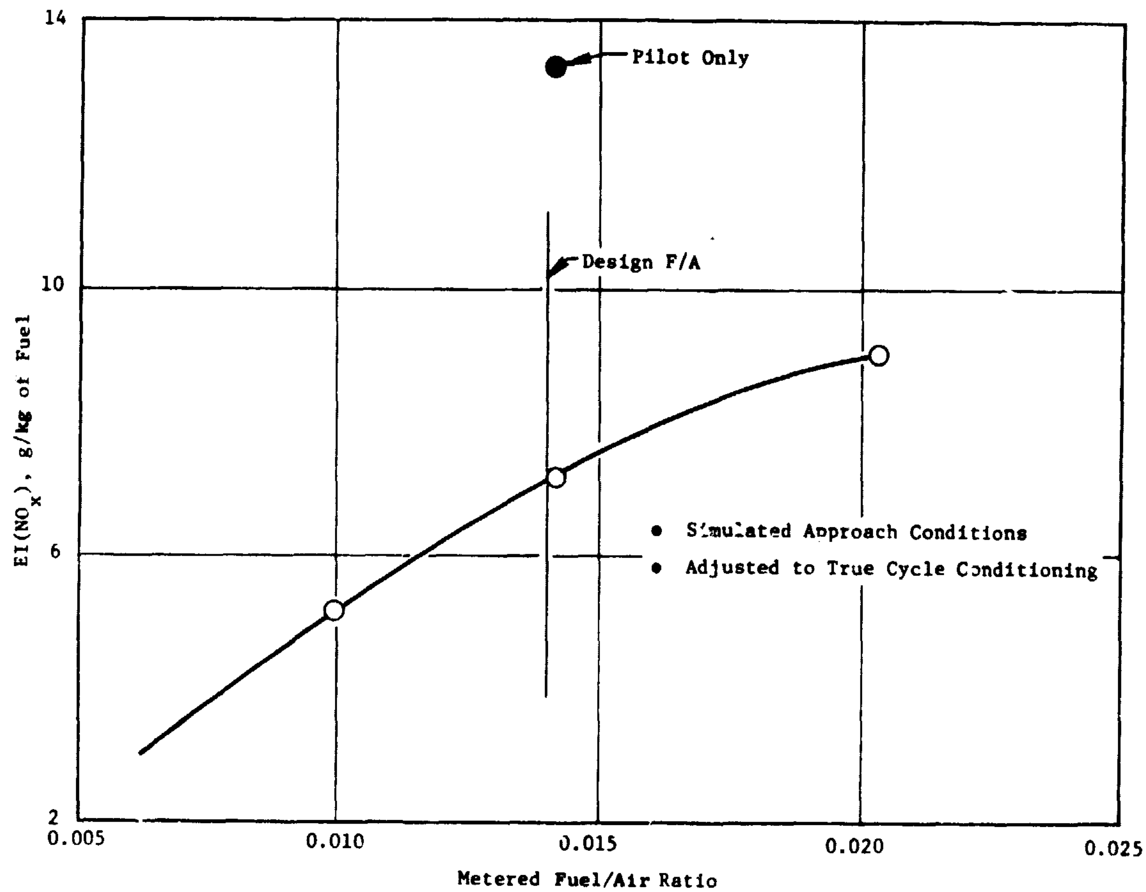

Figure 2.3-25. Sector Combustor Mod V Emissions Test NO_x Versus Fuel/Air at Simulated Approach Conditions.

NO_x emissions were also measured at the simulated sea level takeoff conditions. Figure 2.3-26 presents a plot of NO_x emissions versus selected fuel/air ratios with a pilot-to-total fuel flow ratio of 0.40. The NO_x emissions level corresponding to the design fuel/air ratio of 0.0244 when adjusted to the true cycle conditions is 30.2 g/kg of fuel compared to a target level of 17.5 g/kg of fuel. This data represents the first measurement of NO_x emissions at high power conditions since the main stage dome airflow was reduced to improve ignition performance. As expected, the NO_x emissions level for Mod. V is somewhat higher than levels obtained for the Mod. II configuration (26.4 g/kg of fuel) which was the last main dome high airflow configuration tested. As expected, due to the similarity in airflow levels between the pilot and main stage domes, fuel split has no significant effect on the NO_x emissions levels. Previous full-annular combustor testing has demonstrated generally lower NO_x emissions levels than the sector combustor with similar design features. Therefore, the NO_x levels for an annular configuration similar to Mod. V would be expected to more closely approach the target level. However, the resultant levels would still be expected to be above the goal level. In summary, the increased main stage primary dilution in the Mod. V configuration of the E^3 sector combustor met the main stage ignition requirements and provided a significant reduction of the CO and HC emissions levels at the 30% F_N SLTO approach conditions. However, the NO_x emissions at sea level takeoff conditions increased over the levels obtained from previous testing of earlier configurations with a leaner main stage dome. A summary of the sector combustor development program emissions status in terms of the EPA parameter, is shown in Table 2.3-XI.

Prior to conducting further emissions investigations in the sector combustor, efforts were directed toward identifying the optimum level of fuel nozzle shroud airflow required consistent with low levels of CO and HC emissions and no fuel nozzle carboning. Several shroud airflow levels were tested in the Mod. III and V configurations at ground idle conditions in piggyback tests to determine the effects of shroud air on CO and HC emissions. The fuel nozzles and shroud configurations tested along with the resulting emissions levels obtained are summarized in Table 2.3-XII.

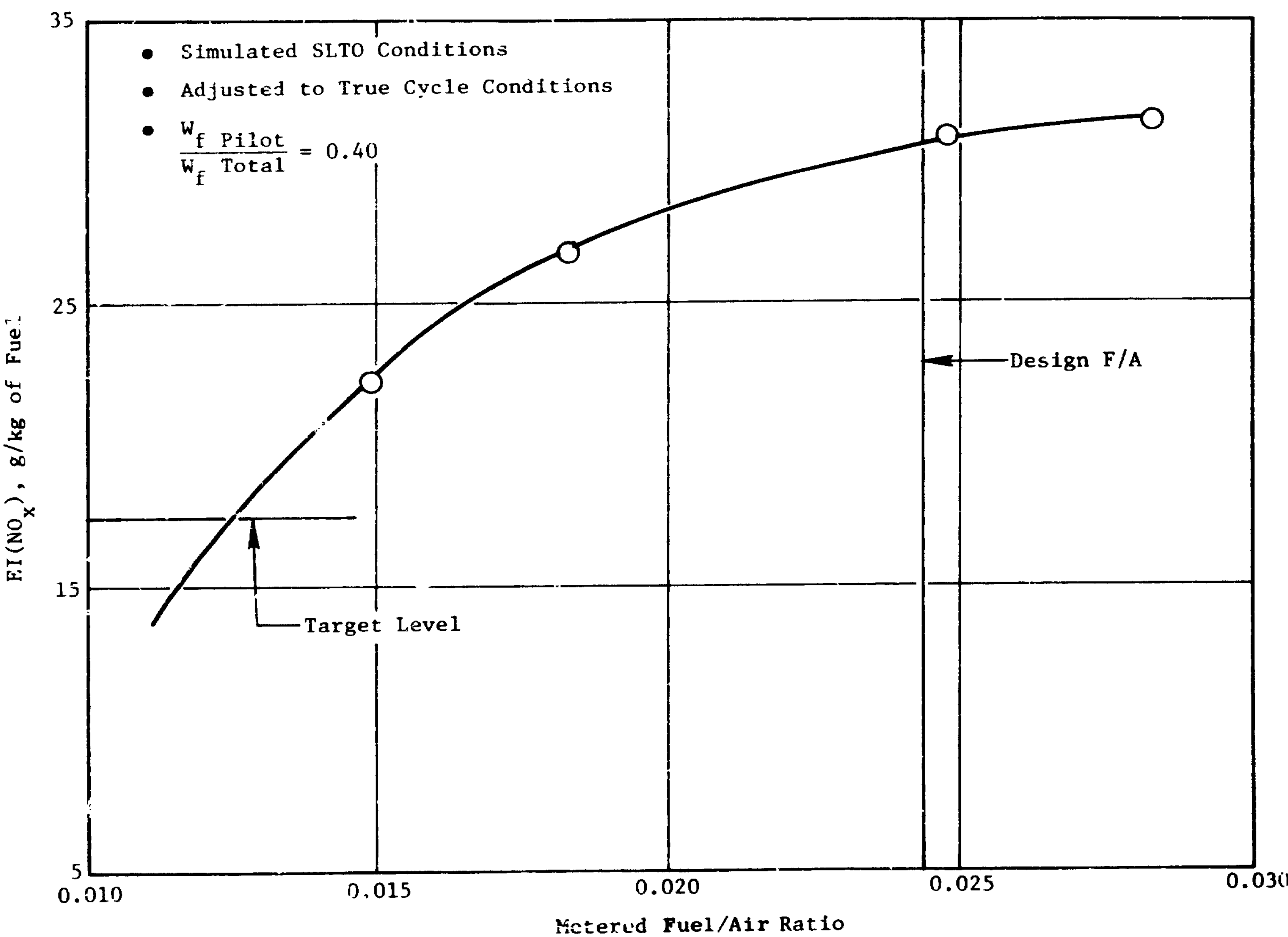

Figure 2.3-26. Sector Combustor Mod V Emissions Tes NO_x Versus Fuel/Air Ratio at Simulated SLTO Conditions.

Table 2.3-XI. Sector Combustor Configuration Emissions Results Summary.

Configuration	Approach Staging Mode	Calculated EPA Parameter lb/1000 lb Thrust Per Hour Cycle		
		CO	HC	NO_x
Baseline	Pilot Only	5.70	0.53	3.30
	Pilot + Main	6.67	0.73	2.39
Mod. I	Pilot Only	3.42	0.37	4.08
	Pilot + Main	4.53	0.52	3.53
Mod. II	Pilot Only	2.11	0.27	4.06
	Pilot + Main	2.98	0.42	3.51
Mod. III	Pilot Only	4.28	0.60	4.87
	Pilot + Main	4.76	0.63	4.43
Mod. IV	Pilot Only	4.01	0.73	3.84
	Pilot + Main	5.10	0.80	3.49
Mod. V	Pilot Only	3.35	0.34	4.68
	Pilot + Main	3.53	0.38	4.32
Target		3.00	0.40	3.00

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Table 2.3-XII. Fuel Nozzle Design Variations Tested for Emissions.

Configuration	Fuel Nozzle Type	Shroud Air (% A_e total)*	Fuel Nozzle Spray Angle, degrees
Mod. III	Development type, 26 pph @ 100 psid	2.0	55
Mod IIIA	Development type, 26 pph @ 100 psid	0 (shroud blocked)	55
Mod. IIIB	Prototype, 25 pph @ 100 psid	0 (no shroud)	85
Mod. IIIC	Development type, 50 pph @ 100 psid	1.8	70
Mod. V	Development type, 26 pph @ 100 psid	1.8	55
Mod. VA	Development type, 26 pph @ 100 psid	0.9 (shroud partially blocked)	55

*The numbers represent the pilot stage fuel nozzle shroud air effective area divided by the total combustor effective area.

G-232

Based on these results, a core engine fuel nozzle shroud design was selected. The shroud air is supplied through 12 holes 0.062 inch in diameter and constitutes approximately 1.2% of the total combustor airflow. This design is expected to provide adequate carboning prevention without adversely impacting the CO and HC idle emissions based on the sector test results.

One additional sector combustor configuration was evolved to attempt to further reduce CO and HC emissions and improve the fuel/air ratio distribution at the sector combustor exit. However, prior to carrying out the investigation, the sector combustor effort was redirected to support the full-annular test program to help identify design features for the main stage dome design to improve ignition performance.

Work Planned

Sector activities will initially be directed at identifying fixes for the annular combustor. Following this effort, the sector combustor configuration will investigate the emissions performance of Mod. VI. Following completion of these tests, altitude relight capability of the selected sector combustor configuration will be evaluated.

2.3.4.2 Annular Test Rig

At the conclusion of the last double-annular development combustor test, several areas on the test rig were identified where improvements are required to improve test rig performance or data quality. The key items of concern are

- Partial plugging of the gas sample rake orifices from carbon deposits
- Inadequate water supply to the gas sample rake system at simulated high power operation
- Difficulty in removing combustor instrumentation routed through the test rig center passage
- Interference between the temperature rake thermocouple elements and the atmospheric test rig exit annulus during data acquisition of combustor exit temperatures with the traverse system
- Poor diffuser performance resulting from extremely flat inlet velocity profiles entering the prediffuser section of the test rig.

C-3

Visual inspection of the gas sample rake inlet orifices, along with airflow calibration, revealed that many of the gas sample rake orifices were either plugged or partially blocked with carbon or water deposits, thus preventing uniform sampling and an adequate supply of exhaust gas to the analysis equipment. Partial plugging of the rakes was suspected during test of the Mod. I combustor and confirmed by gas analysis results, as shown in Figure 2.3-27.

The orifices in the gas sample rakes were set at 0.025-inch diameter to supply adequate sample flow while providing choked flow across the orifice. The pressure drop across the orifice results in a rapid temperature drop in the exhaust gas thereby helping to freeze the exhaust gas reactants until they reach the exhaust gas analyzers. To reduce the potential for plugging gas sample rake orifices, the hole diameter was increased from 0.025- to 0.040-inch diameter. The increased orifice size is expected to eliminate orifice plugging and provide adequate sample flow while maintaining a choked pressure ratio over most of the operating range.

Another problem encountered with the gas sample rakes was an insufficient cooling water supply at simulated high power operating conditions. At test conditions where the test section pressures were high and the cooling water flow rate demands were high, the losses through the supply hoses were excessive resulting in an inadequate water supply to some rakes. As a result, several of the rake elements were damaged and the rakes had to be removed for repair. To prevent this occurrence in future tests, the rake cooling water will be supplied from the instrument spool rotating shaft through slots in the mounting hub, as shown in Figure 2.3-28. This approach will eliminate any flow restrictions and provide sufficient cooling water at all test conditions.

During initial buildup of the test rig, the instrumentation routing hoses, shown in Figure 2.3-29, had to be fabricated from makeshift hardware because the original hoses were not available. The makeshift hoses, though adequate for baseline testing, make it difficult to lead the combustor instrumentation in and out of the rig. The original routing hose hardware was obtained and installed to minimize combustor installation.

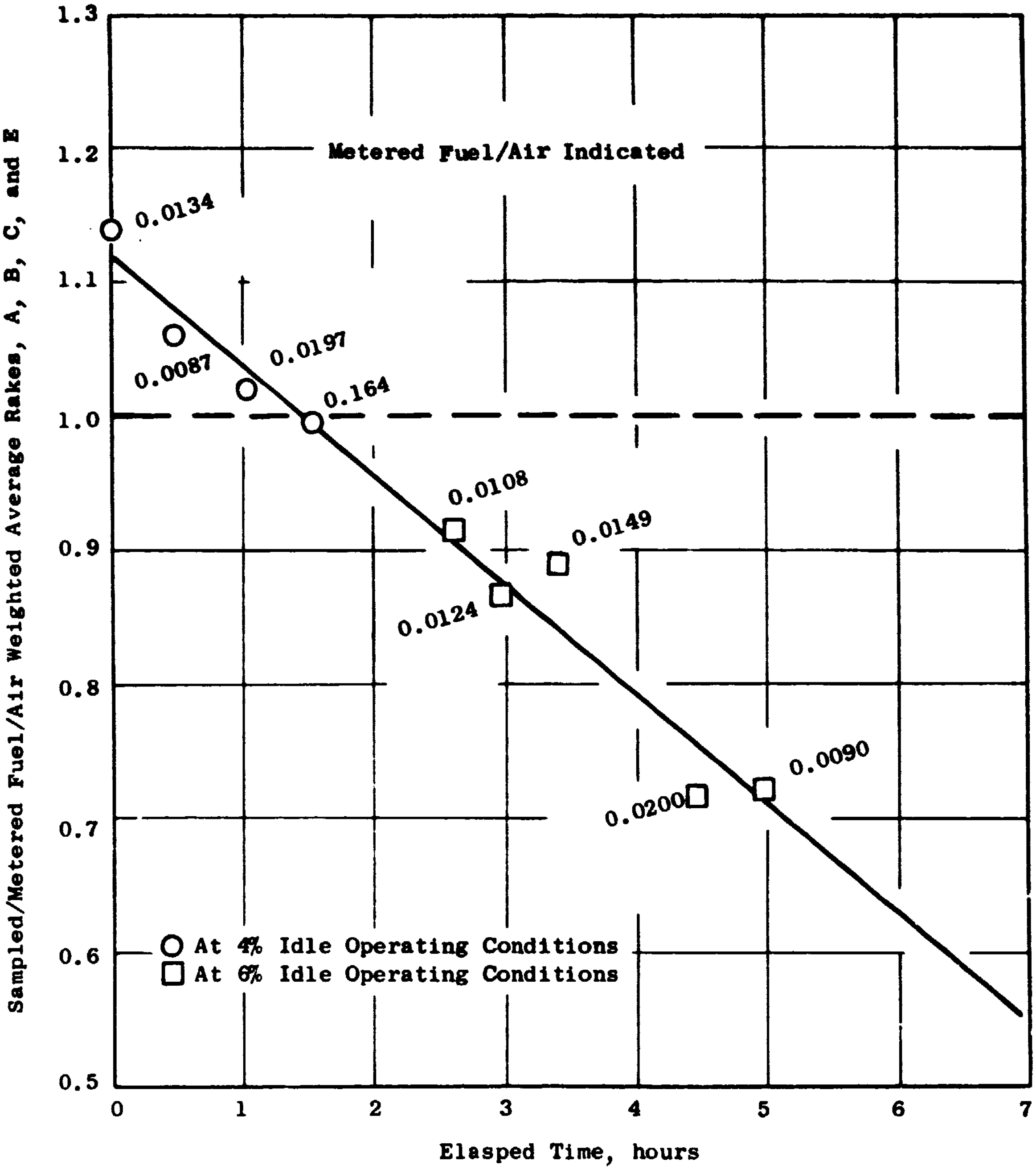

Figure 2.3-27. Gas Sampling Deterioration with Test Time.

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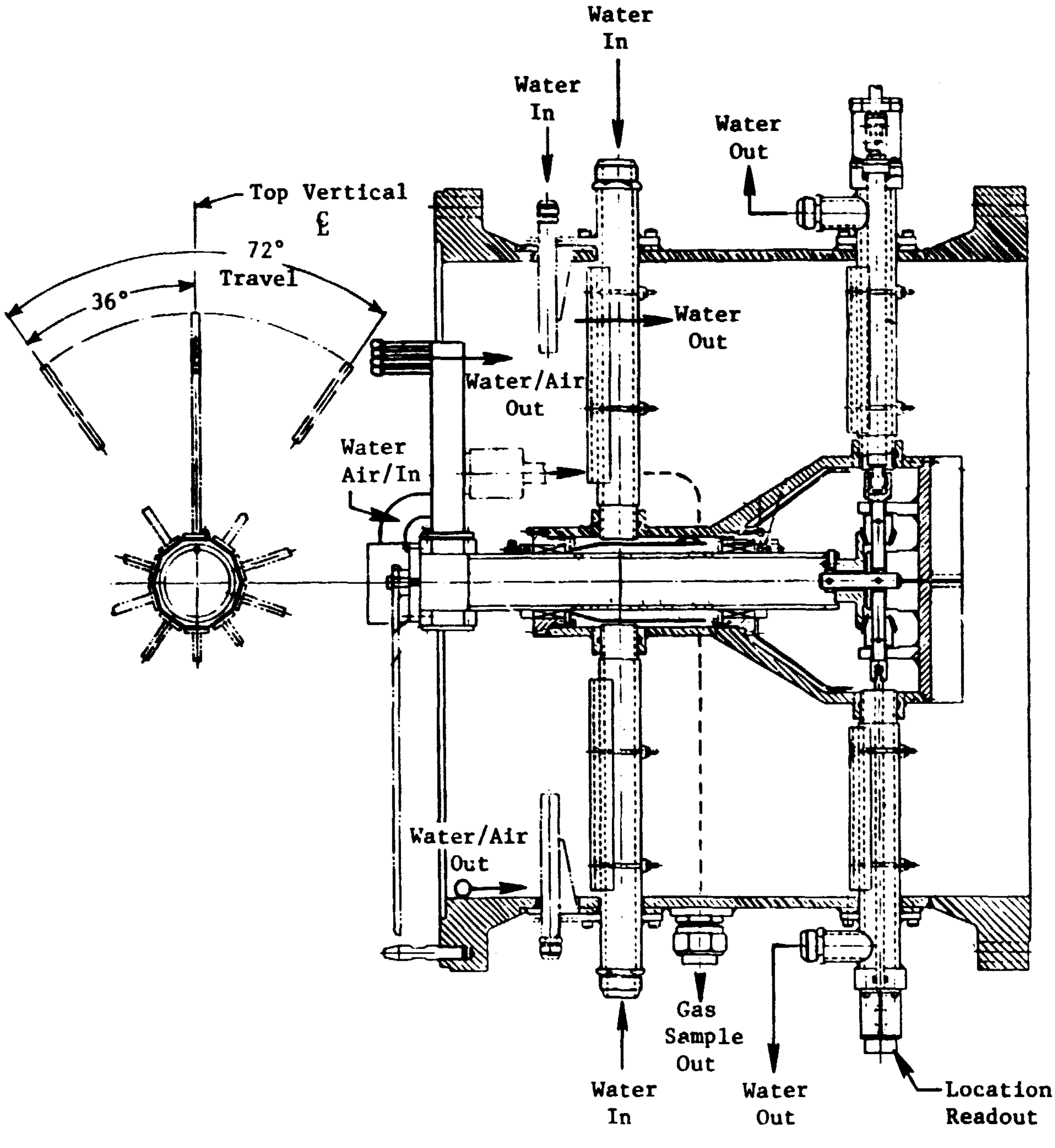

Figure 2.3-28. Rotating Instrument Spool.

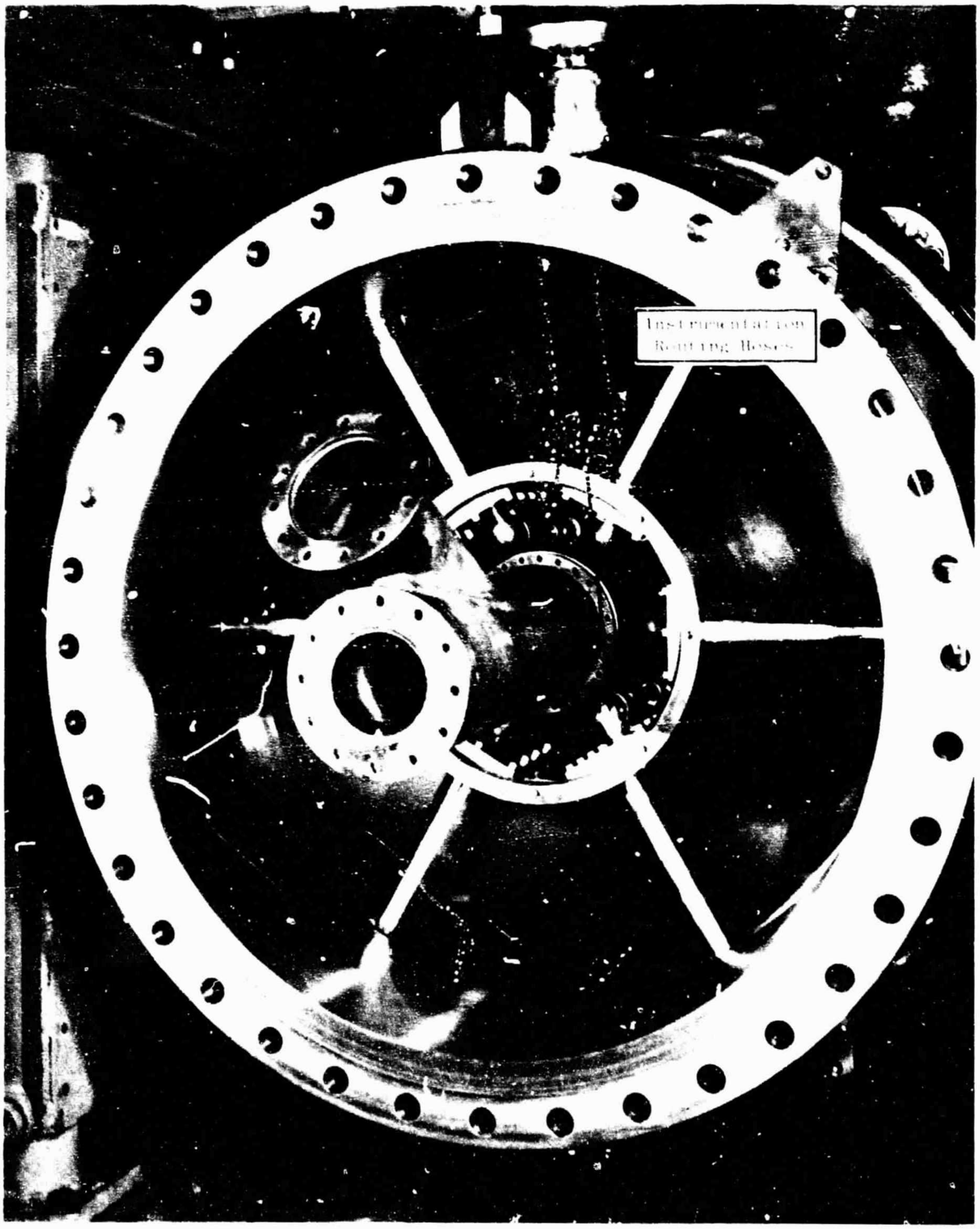

Figure 2.3-29. Combustor Test Rig Showing Instrumentation Routing Hoses.

Data obtained during the baseline and Mod. I development combustor tests, indicated that the inlet velocity profiles entering the combustor inlet diffuser were very flat, as shown in Figure 2.3-30. Additionally, the turbulence levels would be expected to be fairly low since the air enters from a long plenum chamber with few obstructions in the flowpath. Therefore, the combustor inlet diffuser performance would not be expected to be optimized and would result in high pressure losses in the prediffuser annulus. The test rig results were confirmed by comparing the full annular test rig inlet diffuser pressure losses with those obtained in the combustor inlet diffuser model tests where the velocity profiler was eliminated. As shown in Table 2.3-XIII, the baseline and Mod. I combustor configuration pressure losses are very similar to those obtained in the model tests.

To obtain better diffuser performance in the annular test rig, a center peaked velocity profiler, similar to that used in the full annular model tests, has been fabricated and installed in the test rig. The profiler is shown installed in the test rig in Figure 2.3-31. The level of improvement obtained in the model test is shown in Table 2.3-XIV. The same level of improvement is expected in the annular test rig. The reduced combustion system pressure losses should improve combustor dilution and mixing resulting in an overall combustor system performance improvement. The only negative aspect of the installation of the velocity profiler is a slight reduction in test section pressure capability. The added pressure loss of the profiler will reduce the overall test rig pressure level by about 8%. However, this slight decrease in pressure capability is not expected to impact the development program in any way.

The final modification to be incorporated into the test rig is a correction to the thermocouple rake traversing system used during atmospheric performance tests. During the traverse operation at high temperature operating conditions, interference between the test rig annulus flow guides and the thermocouple rake elements occurred in several circumferential locations. This interference appears related to an out-of-round condition in the outer casing or traverse ring when heated. The traverse ring is currently not available to investigate the problem, therefore, corrective action has not yet been identified. However, prior to conducting the next annular atmospheric test, the interference problem will be investigated and the necessary rig modifications identified.

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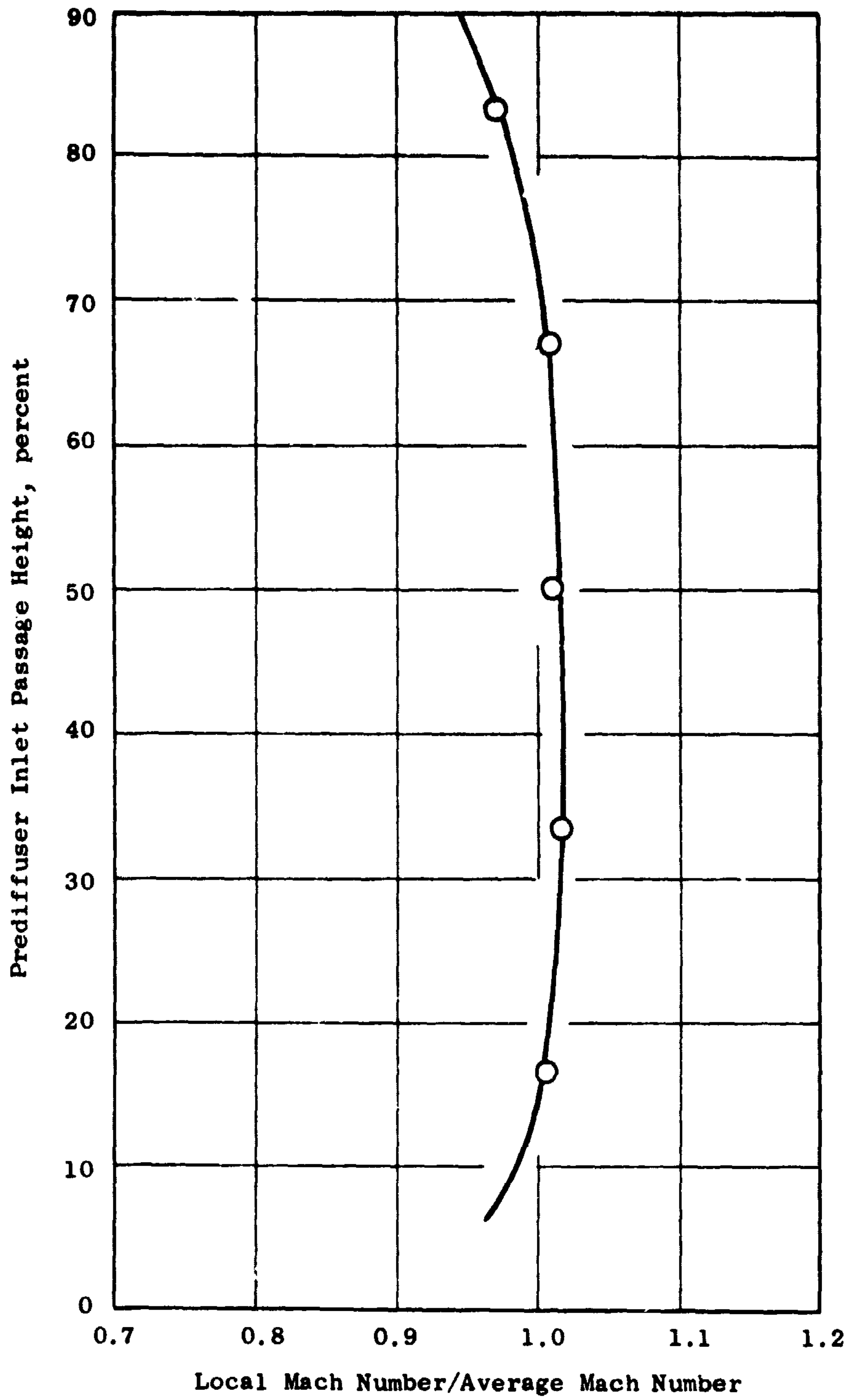

Figure 2.3-30. Test Rig Diffuser Inlet Mach Number Profile.

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Table 2.3-XIII. Test Rig Diffuser Total Pressure Losses.

	Mod-I Configuration	Baseline Configuration	Diffuser Test Flat Profile
Prediffuser Outer Passage	2.52	1.86	2.12
Outer Dump	1.26	1.83	1.92
Total Outer	3.78	3.69	4.04
Prediffuser Inner Passage	1.82	1.79	1.93
Inner Dump	1.32	0.99	1.12
Total Inner	3.14	2.78	3.05
Centerbody	No Data	2.30	2.77
Outer Dome	No Data	2.53	1.16
Inner Dome	2.57	1.72	1.47

Table 2.3-XIV. Diffuser Total Pressure Losses.

- Model Test Results
- SLTO Conditions

Description	Diffuser Test Centerpeaked Profile	Diffuser Test Flat Profile
Outer Passage	3.06	4.04
Inner Passage	2.08	3.05
Centerbody	1.83	2.77
Outer Dome	1.21	1.16
Inner Dome	1.44	1.47

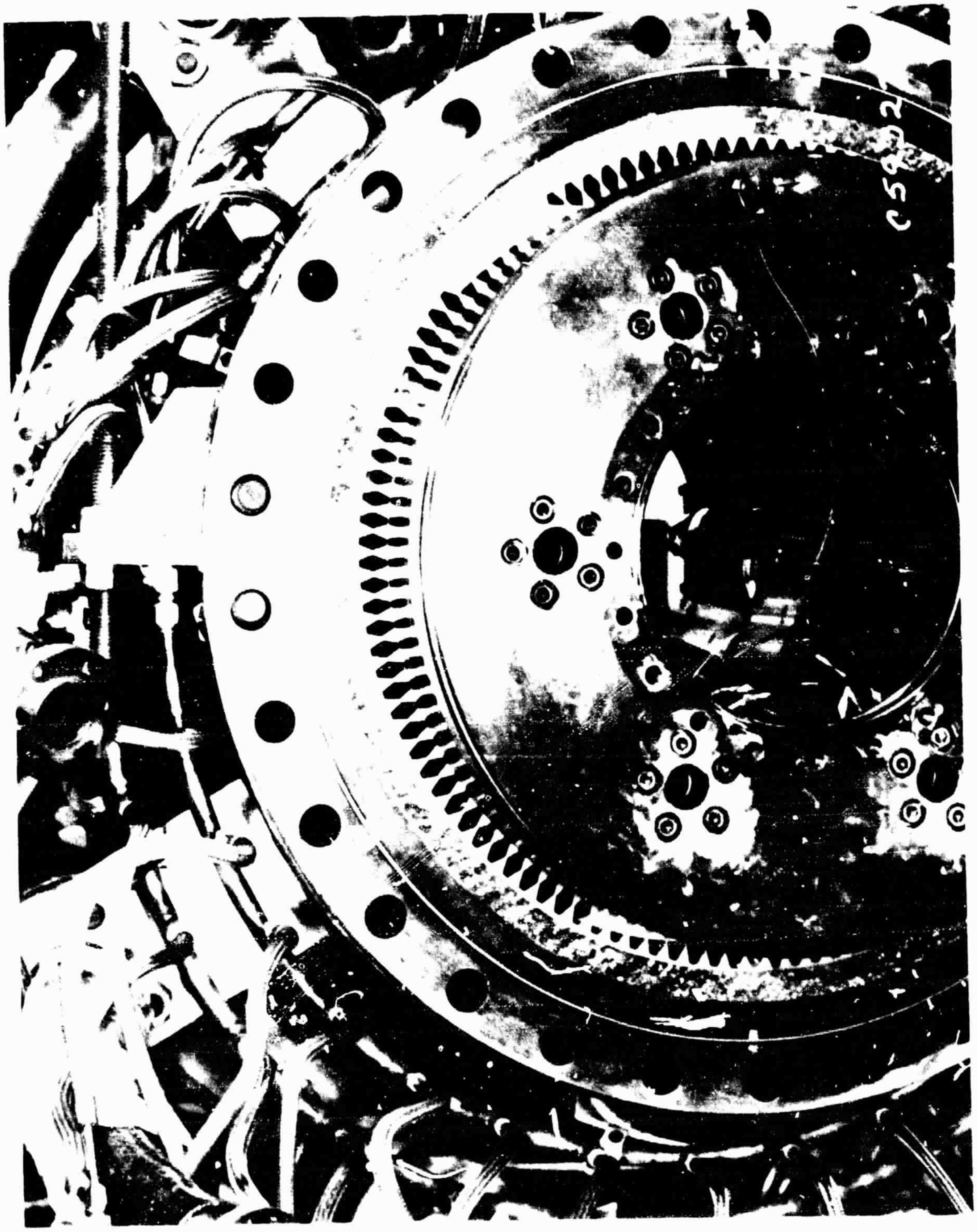

Figure 2.3-31. Test Rig Diffuser Inlet Velocity Profiles.

Work Planned

Investigate atmospheric traverse system interference problem.

2.3.4.3 Full-Annular Test

The objective of the full-annular combustor test program is to develop the emissions and performance features of the E^3 double-annular combustor design utilizing technology from the GE/NASA Experimental Clean Combustor Program (ECCP) and the GE/NASA Quiet Clean Short-Haul Experimental Engine (QCSEE) program as well as evolving new technology. The key objective of this task is to release a combustor, to the core and ICLS engines, which will have verified in component tests that it will meet the emissions and performance goals of the E^3 program.

To accomplish this task, two different types of full-annular combustor tests are being conducted: (1) high pressure tests to develop the emissions, performance, and durability characteristics of the combustor at various simulated engine-operating conditions from idle to takeoff, and (2) atmospheric-pressure combustor tests to develop the required combustor exit temperature distributions and ground-start capabilities. The altitude-relight capabilities of promising combustor configurations will be evaluated in accompanying sector combustor tests. Past experience has shown that altitude relight performance results obtained in sector-combustor tests are quite representative of full-annular combustor altitude-relight performance.

Initially, cold-flow calibrations of the E^3 combustor test hardware are performed prior to the initiation of the combustion tests to ensure that the overall combustor pressure drop is within limits and that the various dome and liner cooling airflows are distributed as intended. Following the verification of the combustor design flow areas, the initial full-annular test is conducted at atmospheric pressure to provide data on ground-start characteristics, pattern factor, and profile factor. No emissions data are obtained during these tests. This type of atmospheric testing has been the main tool for developing the excellent exit-temperature performance of other General Electric combustors. High-pressure tests are then conducted to determine the emissions levels and other performance characteristics of the combustor test configuration over a range of simulated engine-power settings from idle to takeoff.

These high pressure investigations will be conducted primarily at simulated standard day engine operating conditions representing ground idle, takeoff, climb out, and approach conditions commensurate with the specified EPA landing/takeoff cycle. In addition, combustor performance at other simulated engine power conditions is investigated as part of the tests. At certain test conditions, variations of the fuel-flow split between the two combustor annuli are investigated to determine the effects of fuel biasing on emissions and combustor metal temperatures.

Upon selection of the final design for engine installation, all of the design features evolved in the development program will be incorporated in the core engine combustor. This hardware will then undergo complete evaluation of all facets of combustor operation including ignition, emissions, and performance. This combustor will be released to the core engine upon satisfactory completion of the tests.

Preparations for the Mod. II full-annular development combustor were completed in a similar fashion as had been carried out on the baseline and Mod. I designs. Detailed test plans defining the requirements for the ignition, atmospheric performance, and pressure emissions test were prepared and submitted to the test organization for implementation. These test plans provided test instrumentation and test condition requirements as well as testing procedures. The combustor was supplied to the test organization for instrumentation and buildup. The instrumentation requirements were very similar to those described for the baseline configuration. However, the centerbody thermocouples were embedded in the hot side of the metal rather than fixed to the surface as in the Mod. I test. This approach was selected to avoid secondary junctions in the gas path which would give erroneous indications of high metal temperatures.

After completing the instrumentation, the combustor was installed in the test rig in preparation for ground start ignition tests. The fuel nozzles that were installed in the test rig were equipped with 26.5 pph at 100 psid simplex fuel nozzles for both the pilot and main stage fuel system. These fuel nozzles duplicate the primary fuel spray characteristics of the duplex core engine fuel nozzles. The combustor inlet operating conditions for the ground start test are shown in Table 2.3-XV.

Table 2.3-XV. Ignition Test Schedule for Development Combustor Configurations.

Start Time sec	XNRH %	T_3, °F	P_3, psia	W_{36}, pps
10	21	60	Atmospheric	2.76
15	28	60	Atmospheric	3.71
18	32	105	Atmospheric	3.40
30	46	160	Atmospheric	3.64
40	58	230	Atmospheric	4.09
50	70	312	Atmospheric	4.26
55	77	445	Atmospheric	5.13
10	21	75	15.0	2.80
30	46	160	20.9	5.20
40	58	230	27.1	7.50
--	4% G_{Idle}	379	49.9	17.3

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The ground start test results obtained for the Mod. II configuration were disappointing. The pilot stage ignition results are shown in Figure 2.3-32 and are compared to the Mod. I results. The main stage results were unsatisfactory, failing to meet the ground start ignition requirements by a wide margin as shown in Figure 2.3-33.

Visual observations of the flame in the pilot stage indicated that the inner liner dilution flow in Panel 2 was penetrating across the end of the centerbody as shown schematically in Figure 2.3-34. The inner liner Panel 2 dilution holes were blocked off while the combustor was still installed in the test rig to investigate the effects of this dilution flow. An improvement in pilot and main stage ignition was obtained for this configuration (Mod. IIA) compared to the original design (Mod. II) as shown in Figure 2.3-35. However, the main stage ignition performance is still inadequate to meet the ground start ignition requirements. Therefore, the combustor has been removed from the test rig for more extensive modifications.

As part of the ignition development program, systems tests will be conducted using the full-annular test rig hardware in the Cell A3 facility. These systems tests will be directed at simulating the fuel system hydraulic response during the proposed ground-start ignition sequence. The core engine hydraulic characteristics and fuel control inputs will be simulated by a series of electrically actuated solenoids and check valves to measure delay times associated with fuel nozzle fill times and ignition delay times.

The fuel system set up is shown schematically in Figures 2.3-36 and 2.3-37. The response time of the fuel system and ignition delay time will be recorded on real-time Sanborn charts to measure pressures and combustor temperature indications. The proposed test conditions to be evaluated are shown in Table 2.3-XVI. These tests will be conducted after main stage ignition capability is evolved to a satisfactory performance level.

Work Planned

After the modified combustor is received, additional tests will be carried out to develop the ignition performance of the development combustor. Upon

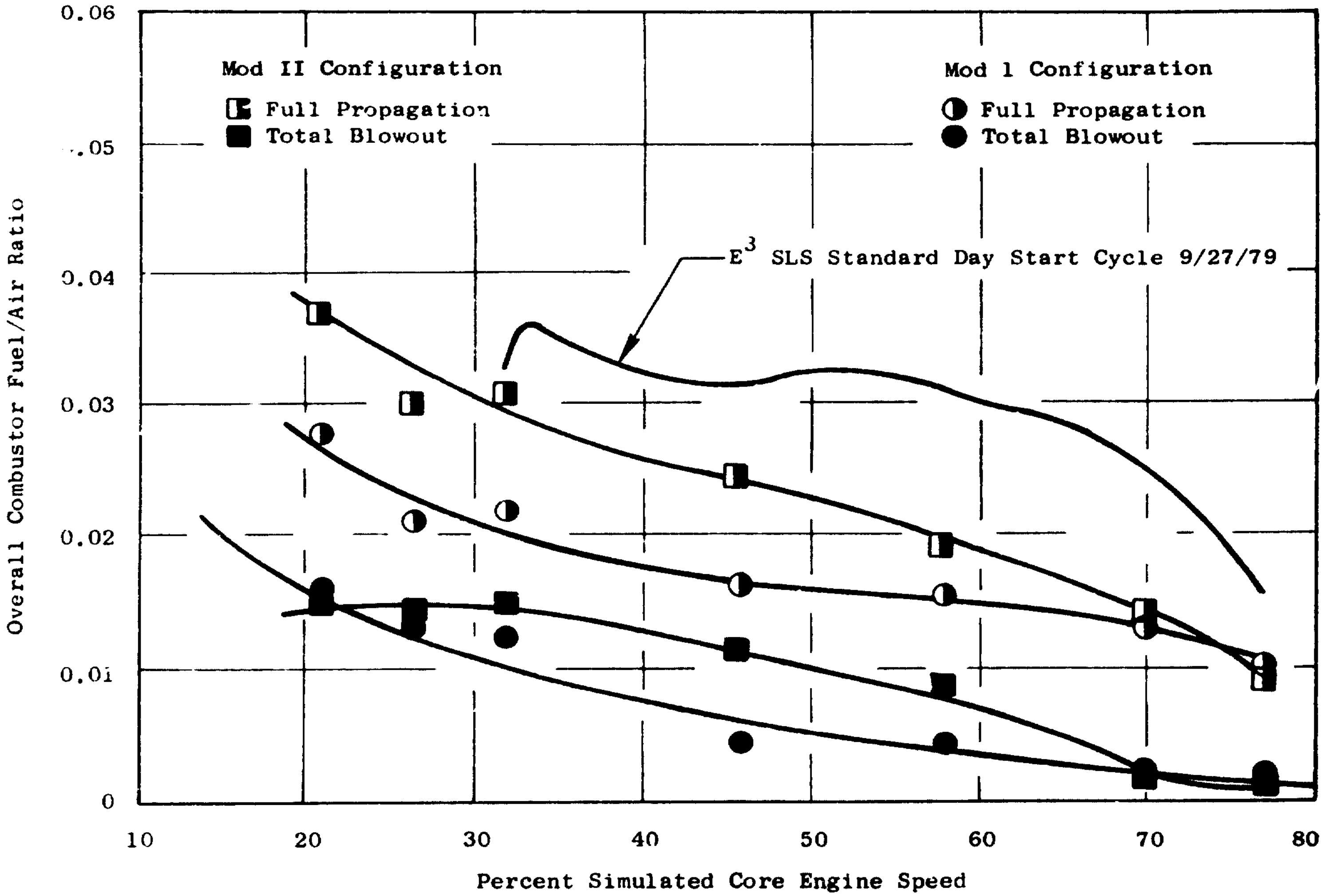

Figure 2.3-32. Development Combustor Ground Start Ignition Pilot Stage.

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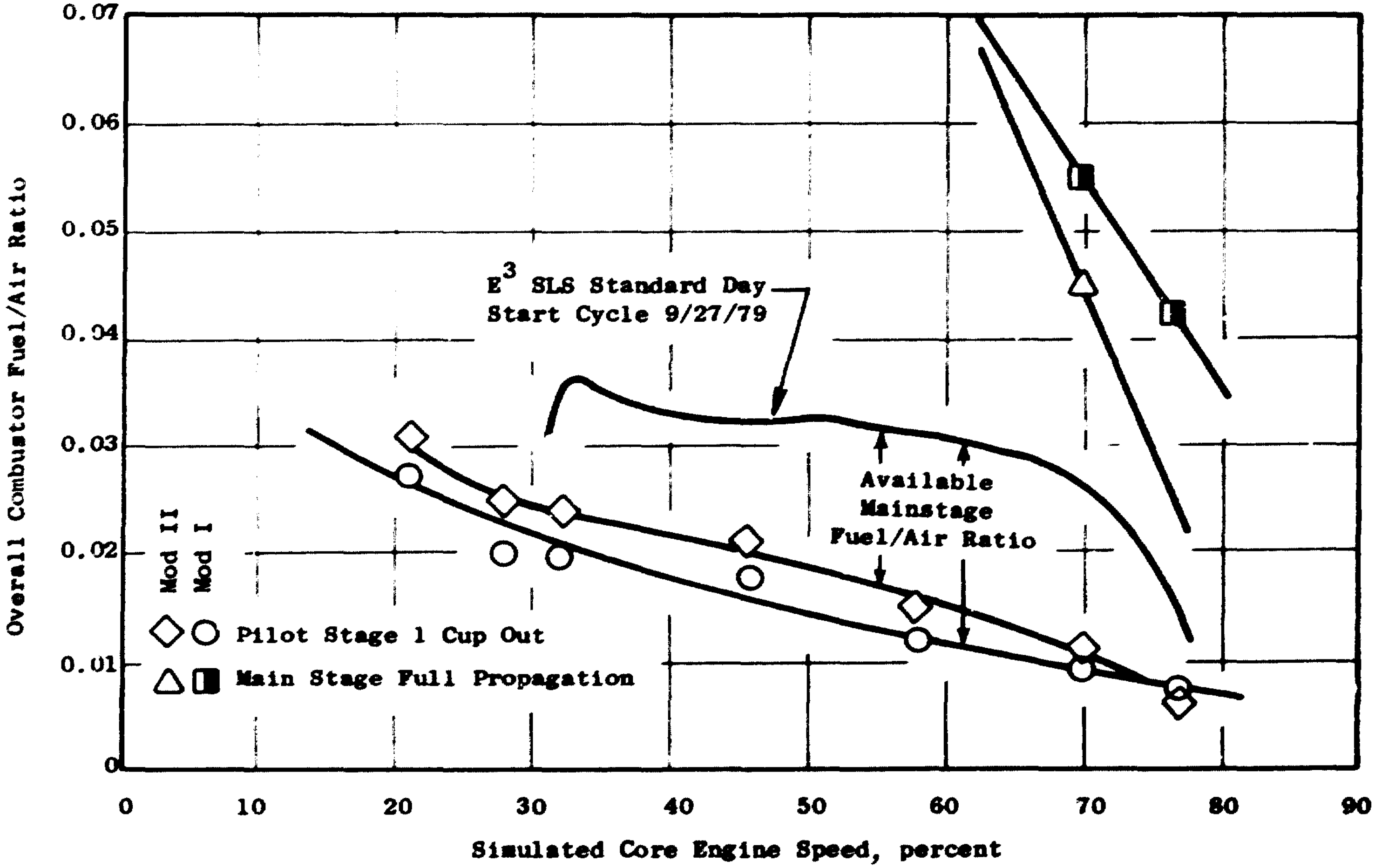

Figure 2.3-33. Development Combustor Main Stage Ground Start Ignition.

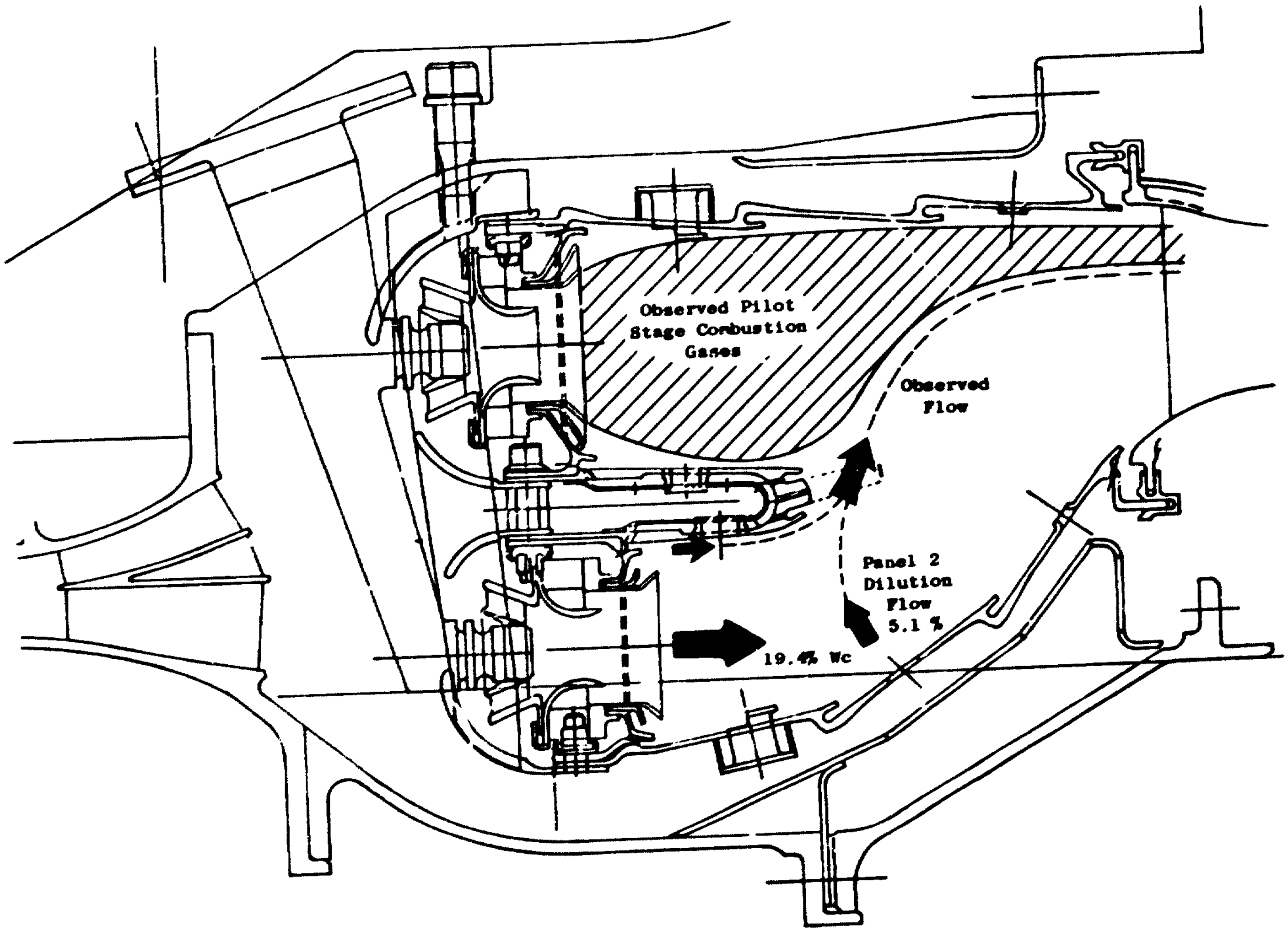

Figure 2.3-34. Mod II Pilot Stage Combustion Gas Flow Pattern.

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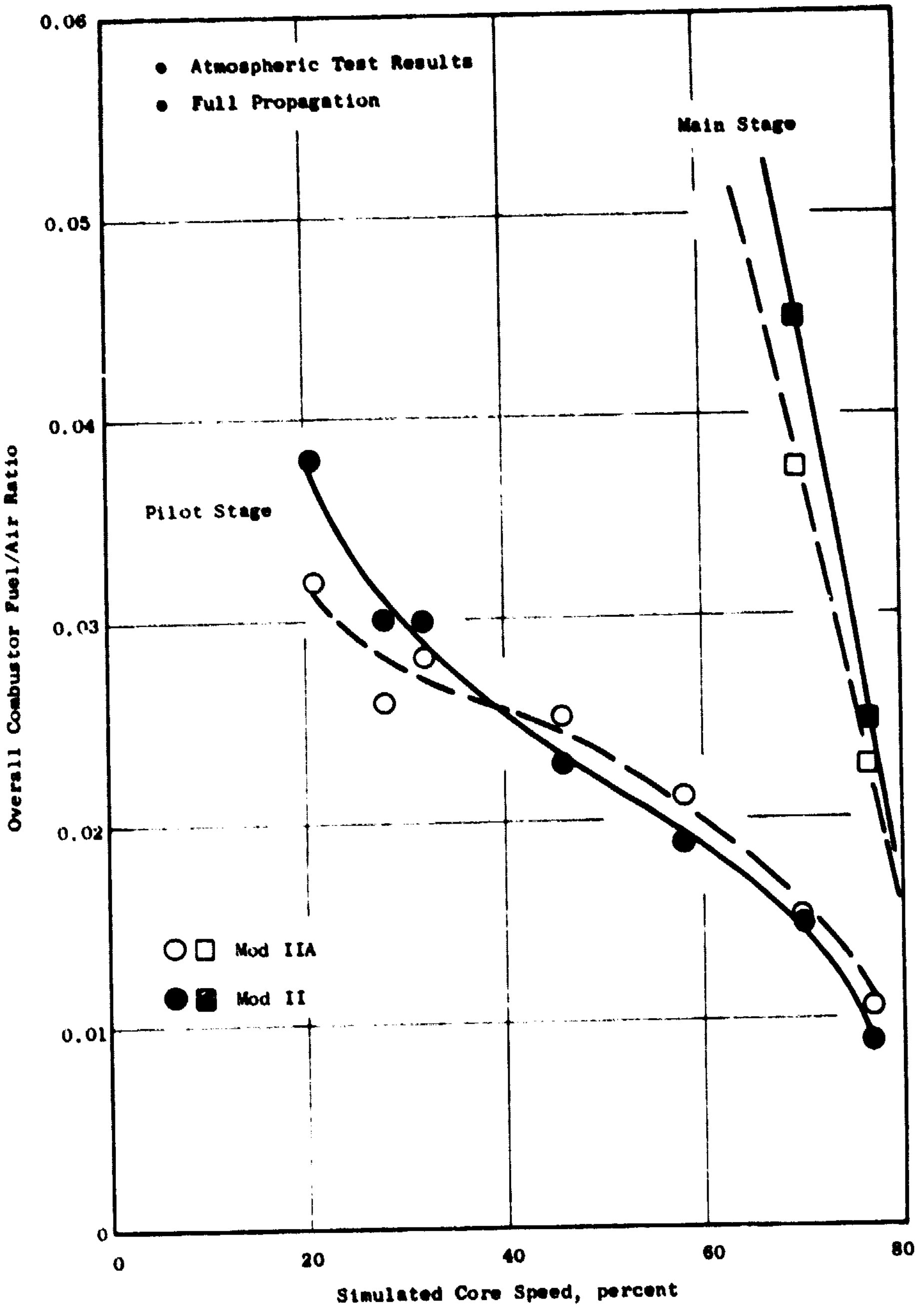


Figure 2.3-35. Development Combustor Ignition Results.

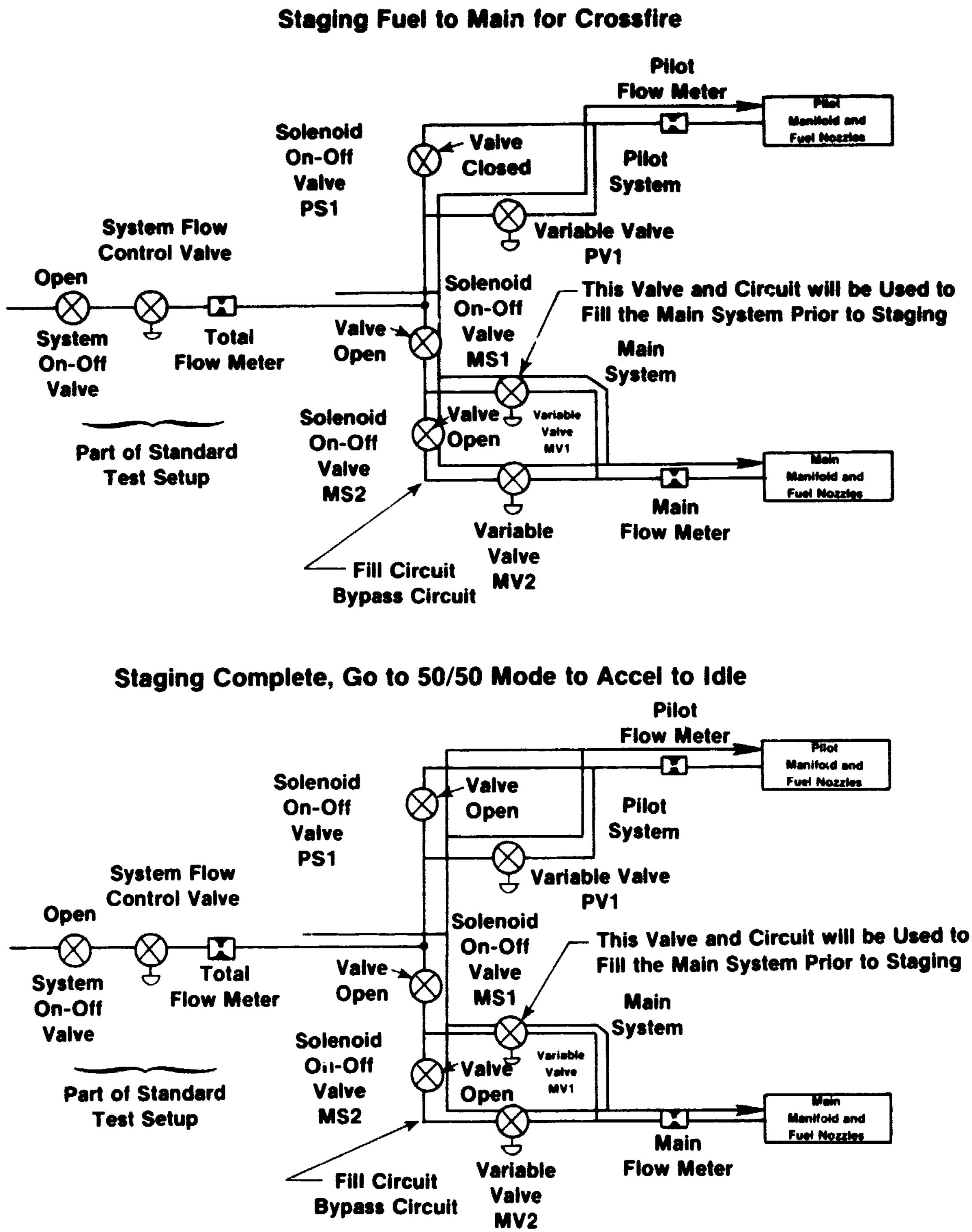
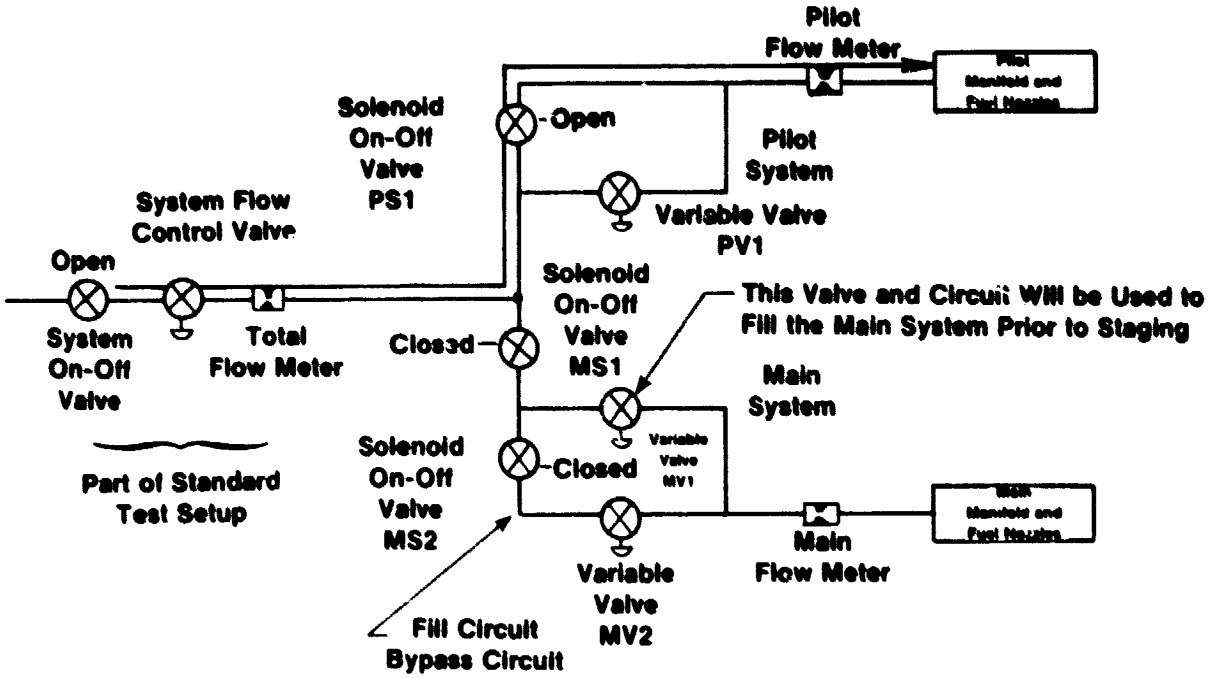


Figure 2.3-36. Test Cell Simulation of Engine Fuel System Response.

Ignition and Propagation of the Pilot Stage



Bleed Pilot System to Fill Main System ~ 5 Seconds

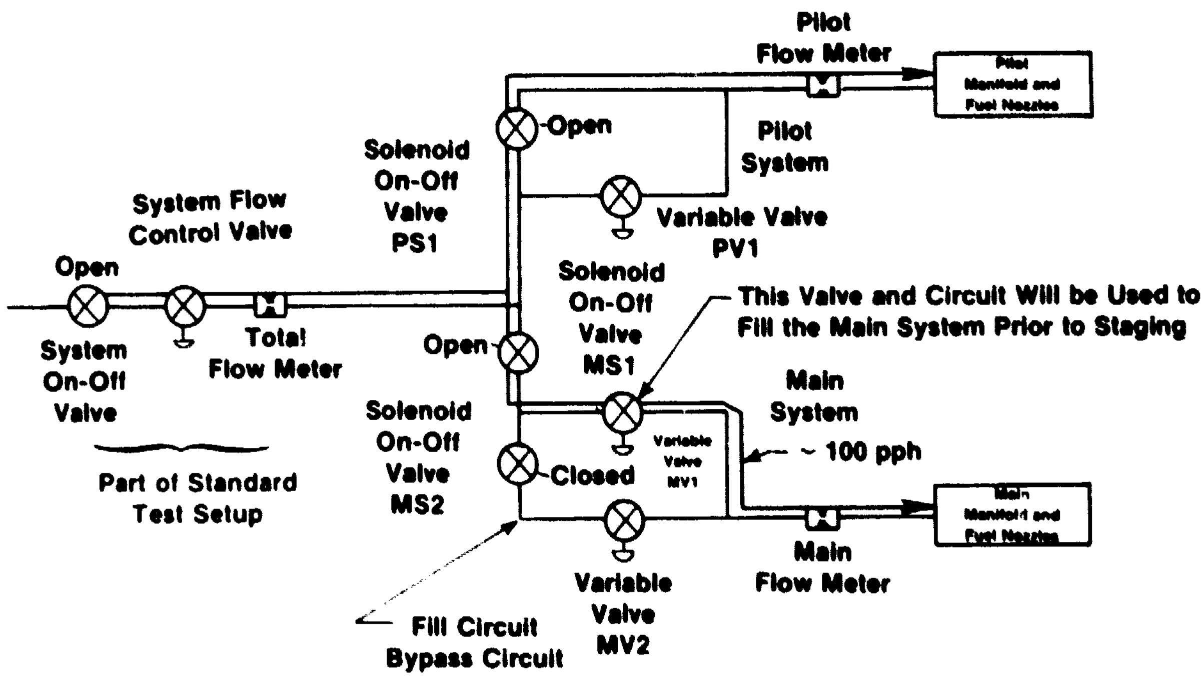


Figure 2.3-37. Test Cell Simulation of Engine Fuel System Response.

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Table 2.3-XVI. Simulated Engine Fuel Control System Ignition Test Schedule.

							Filling		Staging		Accel	
									30%	70%	50%	50%
Test Po[illegible]	Simulated % XNRH	T_3, °F	P_3, psia	W_{comb} pps	W_f, Total pph	f/a_4 overall	Wf pilot pph	Wf Main pph	Wf Pilot pph	Wf Main pph	Wf Pilot pph	Wf Main pph
1a	21	60	AMB	2.76	278	0.028	178	100	83	195	139	139
1b	21	60	AMB	2.76	338	0.034	238	100	101	237	169	169
1c	21	60	AMB	2.76	397	0.040	297	100	119	278	198	199
2a	28	60	AMB	3.71	374	0.028	274	100	112	262	187	187
2b	28	60	AMB	3.71	454	0.034	354	100	136	318	227	227
2c	28	60	AMB	3.71	534	0.040	434	100	160	374	267	267
3a	32	105	AMB	3.40	441	0.036	341	100	132	309	220	221
3b	32	105	AMB	3.40	507	0.041	407	100	152	355	253	254
3c	32	105	AMB	3.40	573	0.047	473	100	172	401	286	287
4a	46	160	AMB	3.64	419	0.032	319	100	126	293	209	210
4b	46	160	AMB	3.64	482	0.037	382	100	145	337	241	241
4c	46	160	AMB	3.64	545	0.042	445	100	164	382	272	273

completion of the ignition development, the atmospheric performance and emissions performance evaluations will be conducted. Additional tests will be conducted as required to assure meeting all of the key combustor performance requirements.

2.3.8 Combustor Fabrication

Required modifications were made to the development combustor hardware in the development testing effort. The Build II modifications were incorporated in November 1980. The combustor was then flow checked and delivered for component test buildup.

Close contact has been maintained with the fuel nozzle manufacturer to assure delivery of this complex engine component by May 1981. Vendors will complete manufacture of all main combustor components in the spring of this year. The combustor components will be available for buildup in July 1981.

2.4 HIGH PRESSURE TURBINE

Overall Objective

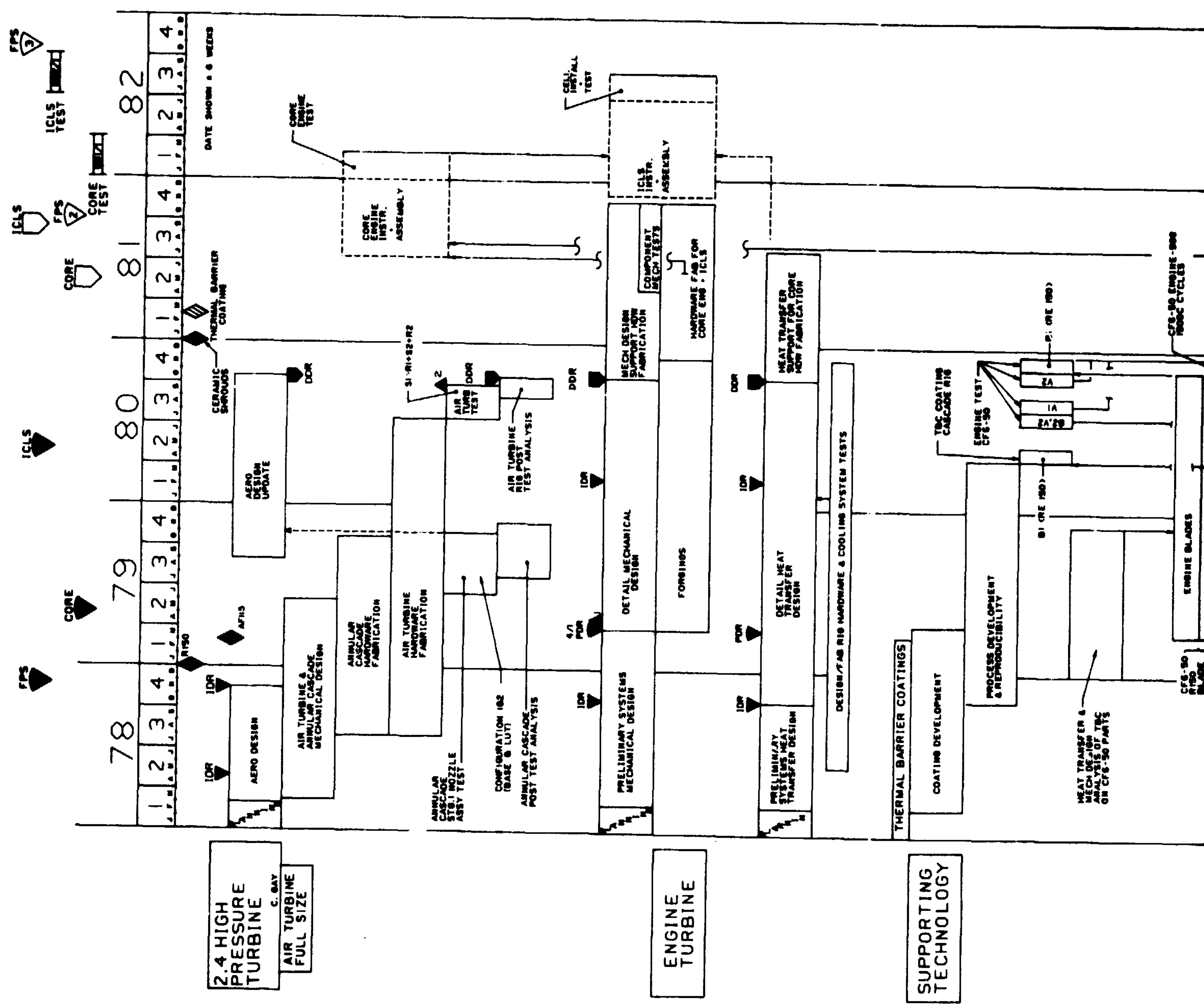
The objective of the HP turbine effort is to develop, evaluate, and demonstrate an efficient, two-stage turbine. The turbine design incorporates features that provide the best balance of efficiency and direct operating costs while achieving required component-life requirements.

Performance achievements shall be aimed at developing high turbine efficiency using moderately loaded airfoils. The HPT efficiency goal for the fully developed FPS is 0.924 at Mach 0.8, 35,000-feet altitude, standard day, maximum cruise power setting. Additionally, the turbine incorporates an active clearance-control system to achieve and maintain improved clearances for enhanced performance, particularly in climb and cruise operation.

Development Approach

The overall program plan for the HP turbine is to establish a turbine mechanical system and configuration that will achieve the projected levels of turbine efficiency and mechanical integrity. The aerodynamnic design studies, initiated in March 1978, are devoted to the design for the air turbine and the aerodynamic airfoils definition for the core and the ICLS engines; current design parameter values are shown in Table 2.4-I. In November 1978, an Intermediate Design Review was presented for the overall air turbine test program, the aerodynamic blade and vane airfoil definition, and the mechanical and heat transfer designs. In March 1979, the Preliminary Design Review was presented and approved for the aerodynamic, mechanical, and heat transfer designs.

The detailed mechanical design began in April 1979 and consisted of an 18-month effort to integrate the experience gained from the materials program, heat transfer cascade tests, air turbine tests, and preliminary mechanical and systems design. The High Pressure Turbine Design Review was presented to NASA on October 10, 1980 and was approved in December 1980. The review consisted of a presentation of all the technological disciplines associated with



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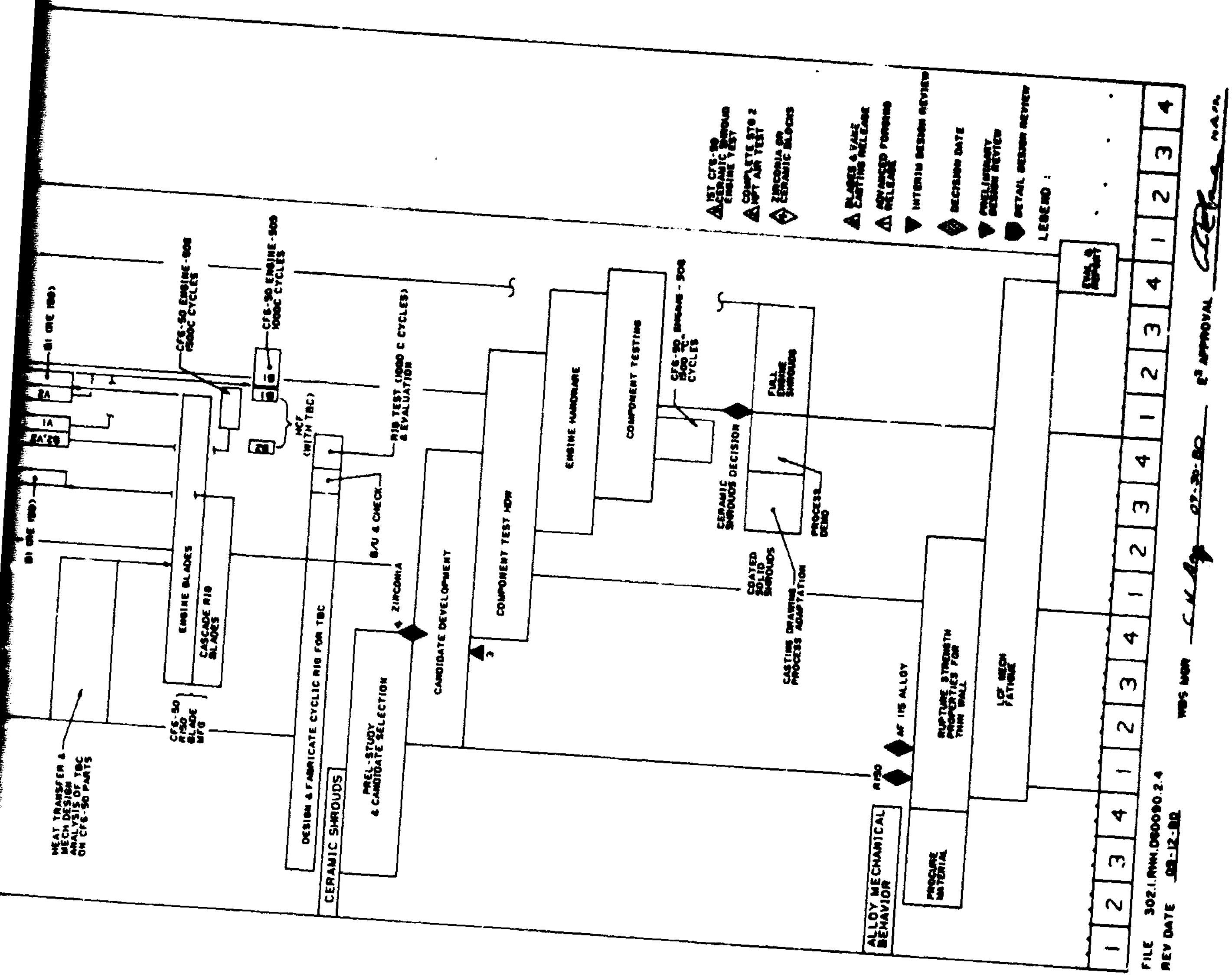

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Table 2.4-I. HPT Aerodynamic Design Parameters.

HPT	RPM	T_{41} °F	P_{41} psia	$\frac{\Delta h}{T}$ BTU/lb-°R	$\frac{N}{\sqrt{T}}$ rpm/°F	ψ	U_T fps	T_3 °F	$\frac{W\sqrt{T}}{P}$ $\frac{\text{lbm-°R}}{\text{sec-psia}}$
SLS/TO +63° F day	13,265	2452	358	0.0848	245.7	0.624	1735	1099	17.6
Maximum Climb +18° F day	12,645	2346	189	0.0851	238.7	0.664	1655	974	17.6
Maximum Cruise +18° F day	12,520	2278	180	0.0851	239.3	0.661	1473	948	17.6

the turbine (that is, heat trasfer, mechanical, aero, and systems review). Upon NASA's approval of the design, the balance of the turbine components not yet released (as an advanced release) were authorized for manufacture. One set of hardware will be purchased which will also be used for both the core and ICLS engine testing. Adequate spares have been ordered for flowpath components. Fabrication and manufacturing interface with vendors has continued within this reporting period.

Components will be instrumented for the core engine tests for engine monitoring and safety. All engine hardware should be delivered for assembly between the third and fourth quarter of 1981 for engine tests. Necessary refinements will be included from the experience gained from the core engine tests and applied to the ICLS engine tests. Posttest analysis will be performed and evaluated for both engine systems.

The major effort of the mechanical design group during this reporting period was in coordinating the manufacturing effort for the procurement of various turbine components. The present manufacturing status can be described as follows:

- All structural rotating and structural static parts are ahead of or on schedule
- Flowpath components are from 4 to 16 weeks behind schedule due to long tooling cycles and some required tooling modifications.

The Stage 1 blade is presently the most limiting part in terms of delivery for the core engine build.

The thermal barrier coating program has progressed through the completion of the coating development and coating selection phase. The selected candidate 20% Y_2O_3 (YZO) has been applied to turbine components from the CF6-50 for a planned 1500 "C" cycle engine test. This test is planned to begin in April 1981 and to be completed by June 1981. Engineering shall monitor the progress of the coated parts during engine testing.

The ceramic process and development program began in March 1978 and extends to the second quarter of 1981. The ceramic shroud program is essentially divided into four major phases. Phases I and II address the material screening evaluation, the process parameters, and candidate selection and development. Phases III, IV, and V comprise fabrication and component tests.

Two basic ceramic shroud design configurations were tested in the CF6-50 engine between June and September 1981. After 625 cycles, the test was discontinued due to turbine FOD. Based on the posttest analysis from this test, the superpeg shroud design was selected as the prime candidate for the E^3 Stage 1 HP shroud.

The two-stage air turbine test was completed and its results reported in the last semiannual report. At present, further refinements of the test data are being completed to establish the turbine efficiency.

2.4.1 High Pressure Turbine Aero Design Analysis

Technical Progress

A detailed design review of the high pressure turbine was held on October 10, 1980 at NASA-Lewis.

A test memo for the two-stage air turbine rig was issued.

The aerodynamic design portion of the high pressure turbine detailed design report was written.

Testing of the two-stage air turbine vehicle was completed on September 11, 1980 as stated in the previous reporting period. A layout of the test rig is shown in Figure 2.4-1. A summary of turbine operating parameters at rig conditions is presented in Table 2.4-II. Analyses of the data, to date, indicate that the quoted level of efficiency is substantiated.

Posttest analysis of the two-stage air turbine data was initiated and will continue through the next reporting period. Circumferential cobra prove traverses were taken at design point operating conditions. Measurements of turbine exit-total pressure, total temperature, and absolute flow angle were made. Figure 2.4-2 shows the radial profiles of pressure, swirl, and temperature obtained by integration of the circumferential traverses. Pressure and temperature have been normalized by their respective inlet values. Positive swirl is in backward running direction.

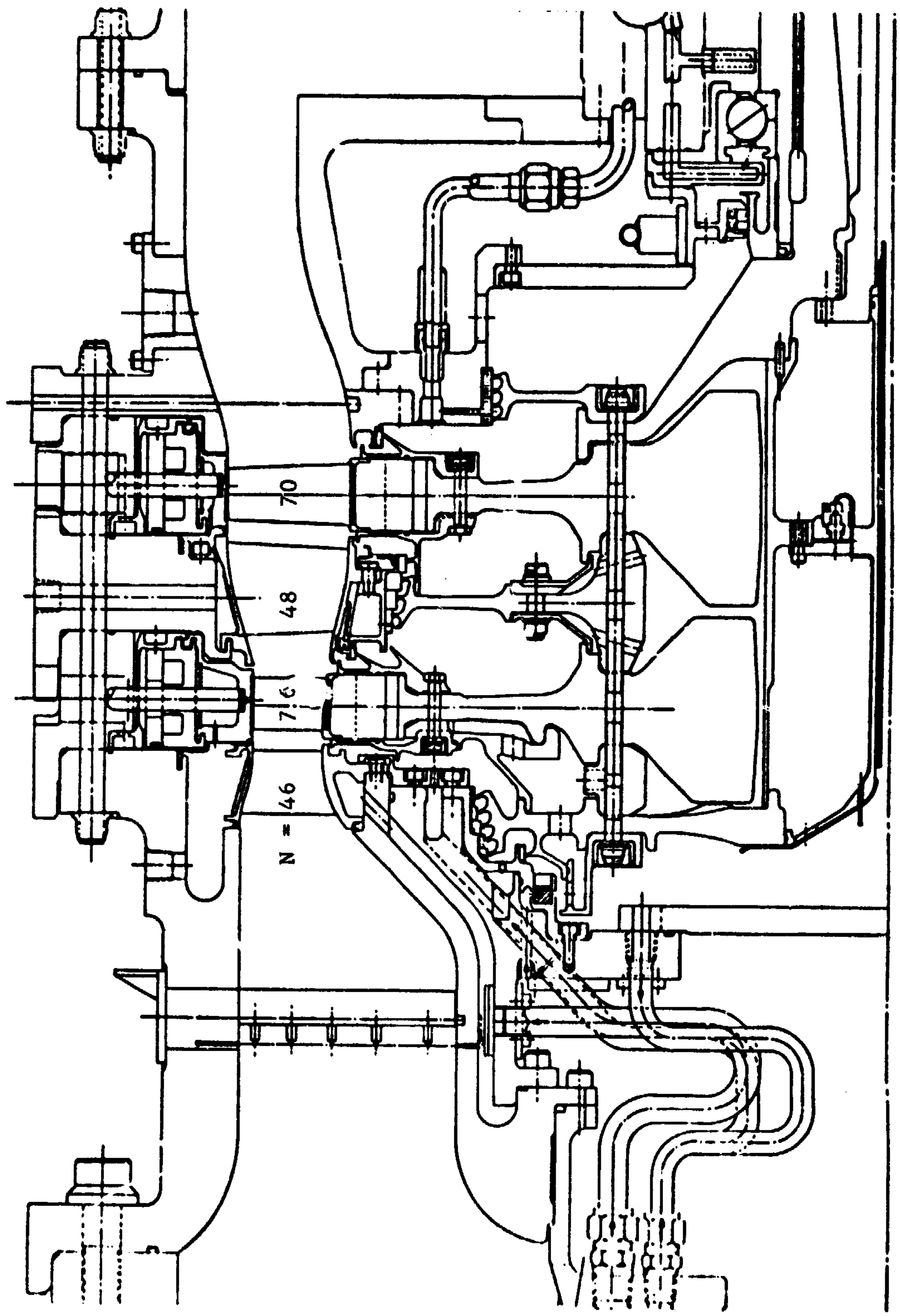

Figure 2.4-1. HP Turbine Rig Layout.

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Table 2.4-II. Summary of Turbine Operating Parameters at Rig Conditions.

	Test	Predicted
Flow Function, $W\sqrt{T}/P$	18.19	18.06
Energy Extraction, $\Delta h/T$	0.0827	0.0814
Corrected Speed, $N/\sqrt{T}$	236.4	236.2
Pressure Ratio, P_T/P_T Total-to-Total	5.02	5.04
Stage Loading, ψ p	0.658	0.648
Root Reaction, R_{x_R}	0.381	0.337
Efficiency, η_T	0.925	0.916

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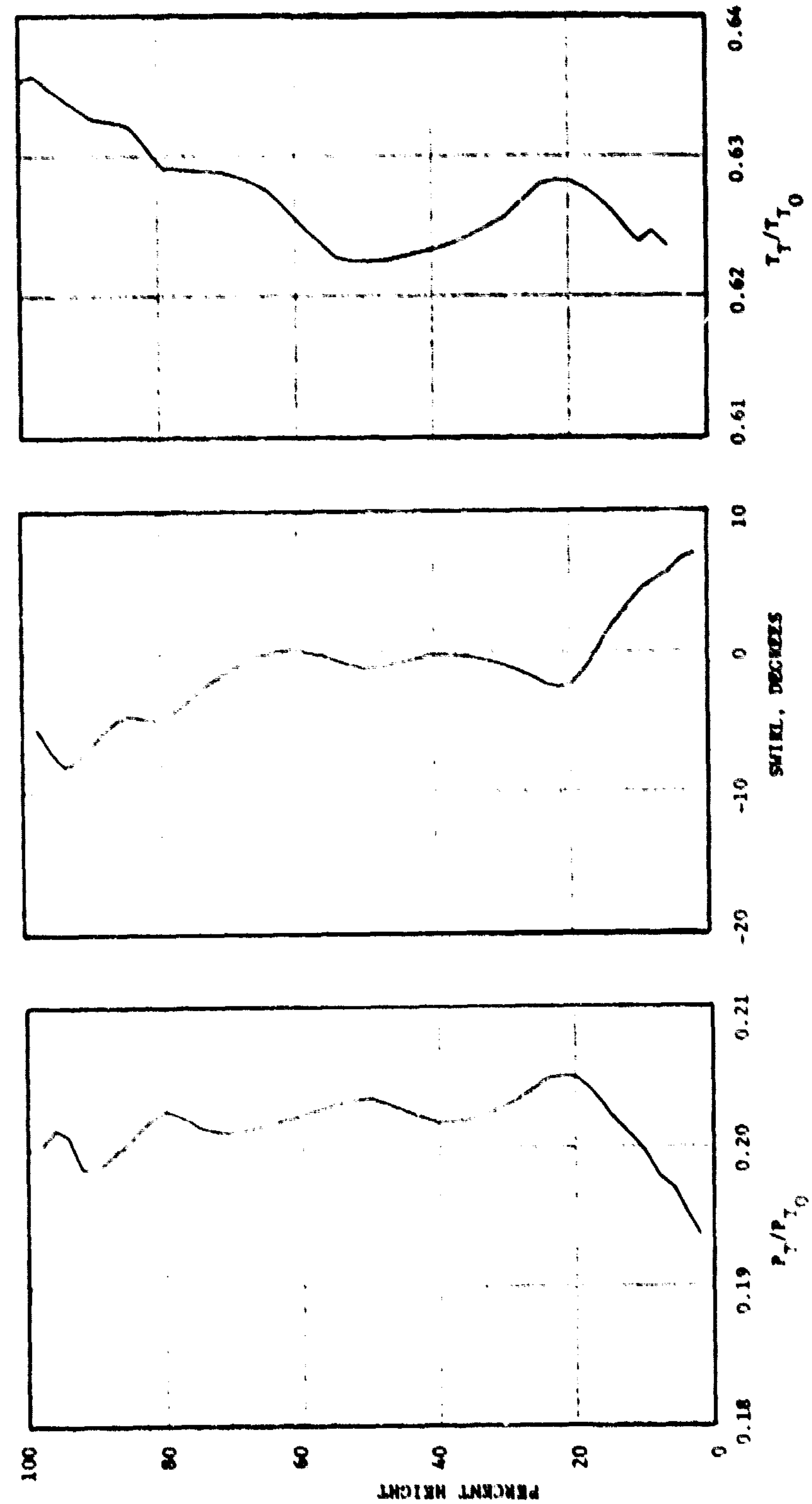

Figure 2.4-2. Turbine Exit Radial Profiles.

Contour plots of the exit flow conditions are depicted in Figures 2.4-3 through 2.4-5 for pressure, swirl, and temperature, respectively; circumferential arc is 0.34 radian. Stage 2 vane wakes are readily observed.

All test readings were scrutinized to identify any errors attributable to malfunctioning instrumentation, incorrectly set point, or point instability. Where possible, these readings were corrected; otherwise, eliminated. All readings with identifiable instrumentation errors have been reprocessed.

Work Planned

- Complete posttest analysis
- Prepare air turbine test report
- Support instrumentation and assembly of core engine.

2.4.2 High Pressure Turbine Heat Transfer Design

Extensive work continued to be performed in the area of heat transfer design of the high pressure turbine. This work has been directed at completing the detailed design heat transfer analysis of the turbine flowpath components and turbine structure. A successful review of the turbine heat transfer was conducted with the Engine Product Review Board. Minor changes were suggested and incorporated into the analysis. A successful detail design review was also presented to NASA on October 10, 1980. The detailed design report which finalizes the 18-month effort has been drafted and almost completed.

The detailed work effort on the high pressure turbine during the last 6 months is herein presented.

Stage 2 Blade

From production engine experience, turbine hardware can experience foreign object damage (FOD) caused by some part of the turbine hardware breaking off and hitting the blades. Because of this and the unique nature of the

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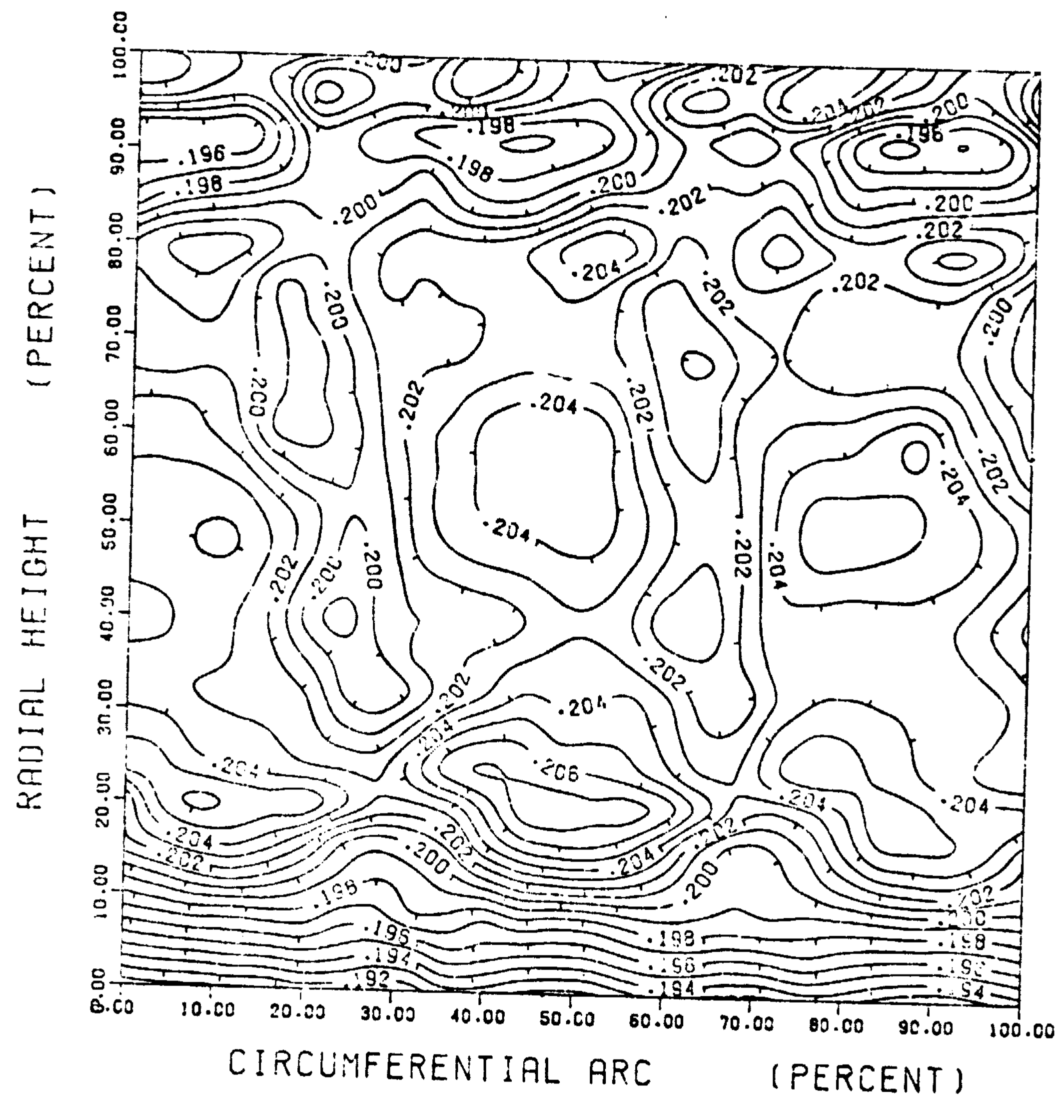

Figure 2.4-3. Turbine Exit Normalized Pressure Contours.

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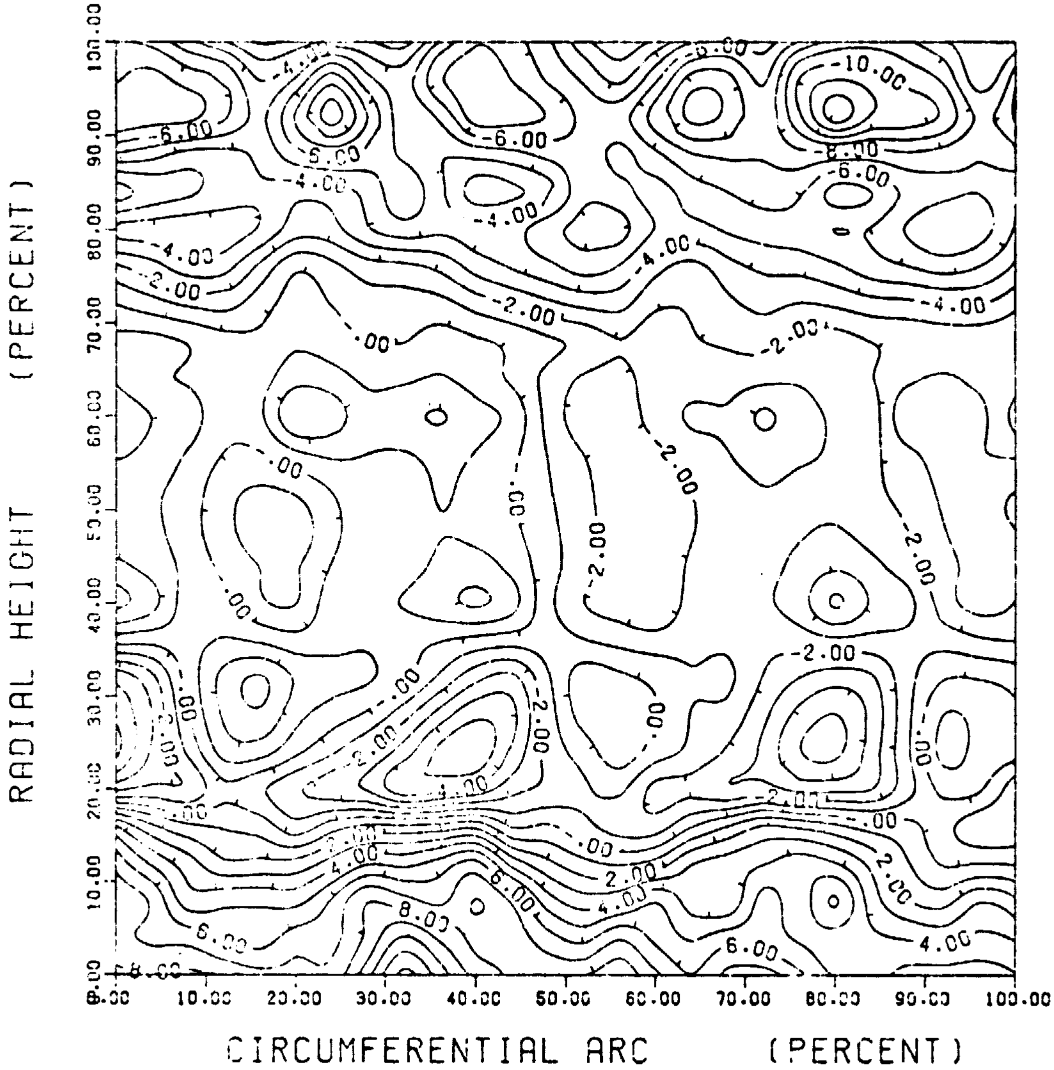

Figure 2.4-4. Turbine Exit Absolute Flow Angle Contours.

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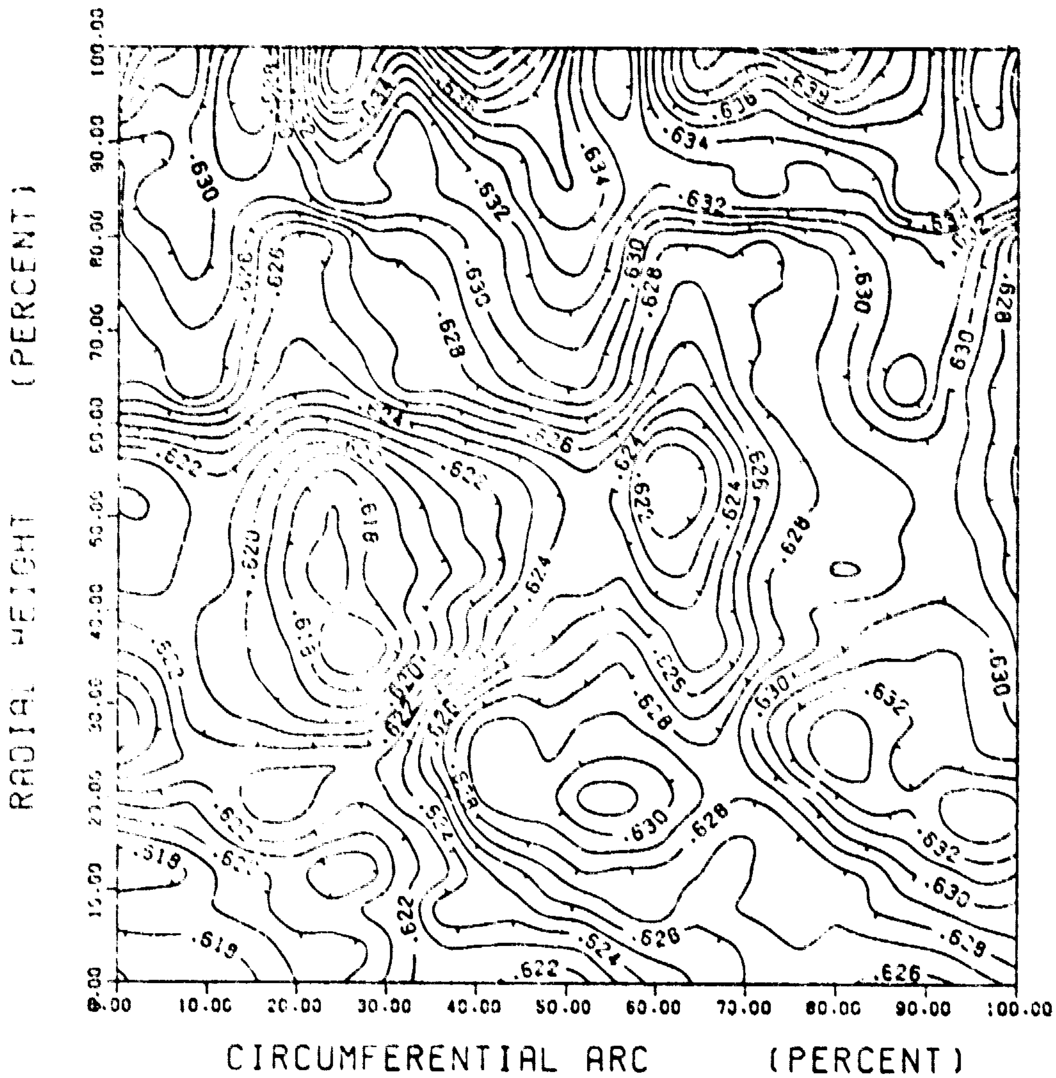


Figure 2.4-5. Turbine Exit Normalized Temperature Contours.

Stage 2 blade cooled circuit, an FOD analysis was conducted. The purpose of this analysis was to define the cooling flow variation to the blade if a hole caused by FOD penetrated the leading edge radial cooling passage. The blade flow variation is presented in Figure 2.4-6 for an FOD hole up to 0.25-inch diameter at the 60% span. The leading edge flow circuit flow is increased quite drastically but, at the same time, the flow to the second and third radial cavity drops from 0.33% to 0.3% when a 0.25-inch diameter hole is opened up. The corresponding pitch-line temperature distribution with and without the 0.25-inch diameter holes is presented in Figure 2.4-7. This analysis shows a 67° F drop in the bulk metal temperature with the biggest variation occurring at the leading edge where the inside part of the airfoil drops as much as 182° F. The end result of this analysis shows no catastrophic loss of blade life due to a loss in cooling.

A similar analysis was conducted for the blade tip cap. The tip cap will be a one-piece construction and brazed in place. There will be no mechanical retention device, as in the first stage. This is feasible since the blade tip gas temperatures are significantly lower on Stage 2. Because the Blade 2 tip-cap retention is relying completely on the braze, a thermal analysis was conducted to determine the impact of the possibility of a braze failure and tip-cap loss. The compressible flow network program, like in the previous analysis, was used for this analysis also. Without the tip cap, the core support holes become the flow circuit restriction. The blade flow went from 0.76% W_{25} to 2.38% W_{25} with no tip cap. The flow in any radial cavity, as shown in Figure 2.4-8, did not drop. The cooling flow in the last two radial passages of the forward circuit went up at least from 0.332% to 0.386%. This analysis, in general, indicated no problem if a Stage 2 blade lost its tip cap.

High Pressure Turbine Active Clearance Control System

The impingement cooling scheme for both the first and second stage rotor clearance control has been finalized. Fan air is used to impinge on the HP casing. The fan air is extracted from the bypass duct through a split scoop that separates the HP and LP active clearnce control (ACC) air. The air, once inside the scoop, is slowed efficiently through a 2:1 area ratio diffuser

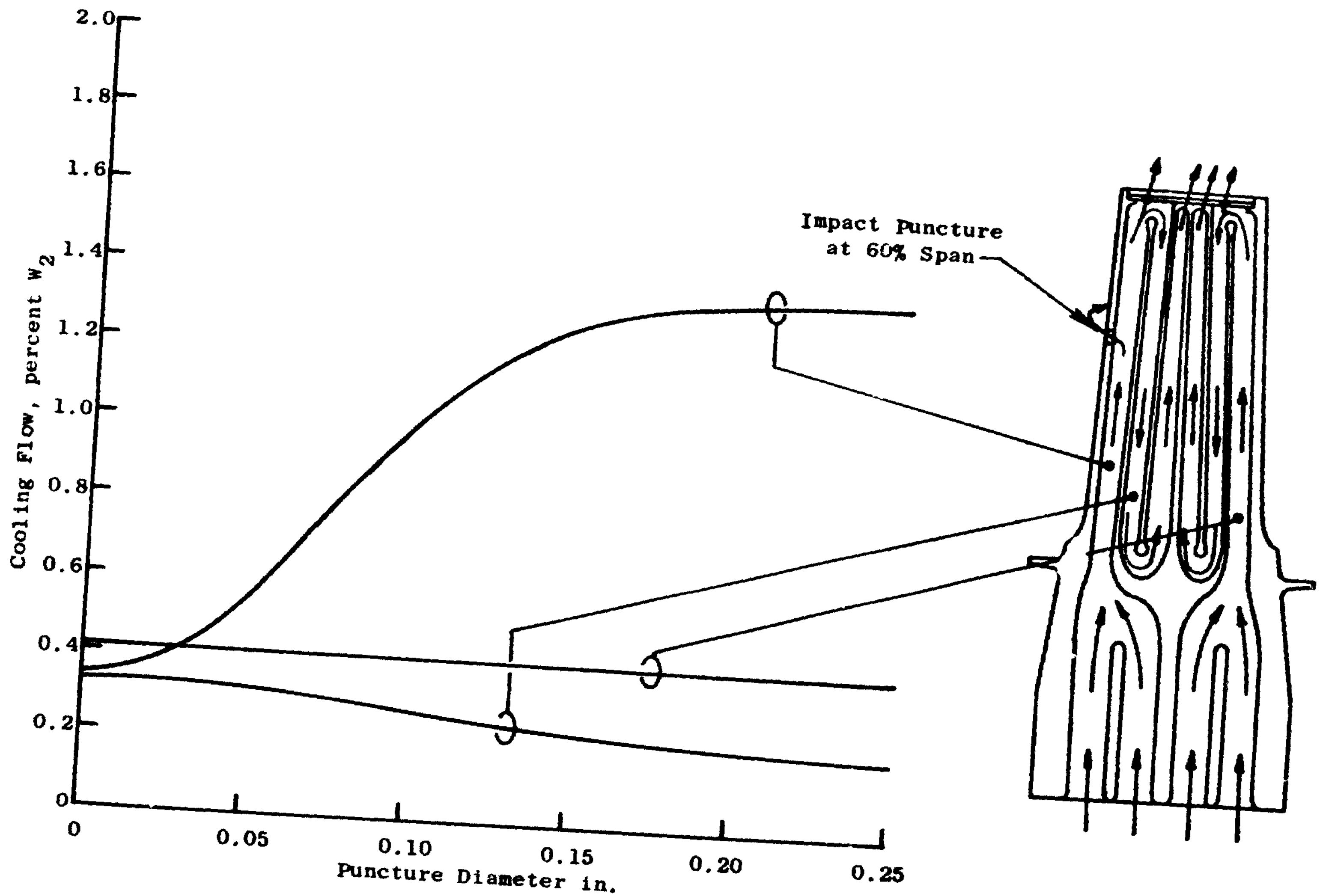


Figure 2.4-6. HPT Stage 2 Blade Cooling Flow with Leading Edge Puncture.

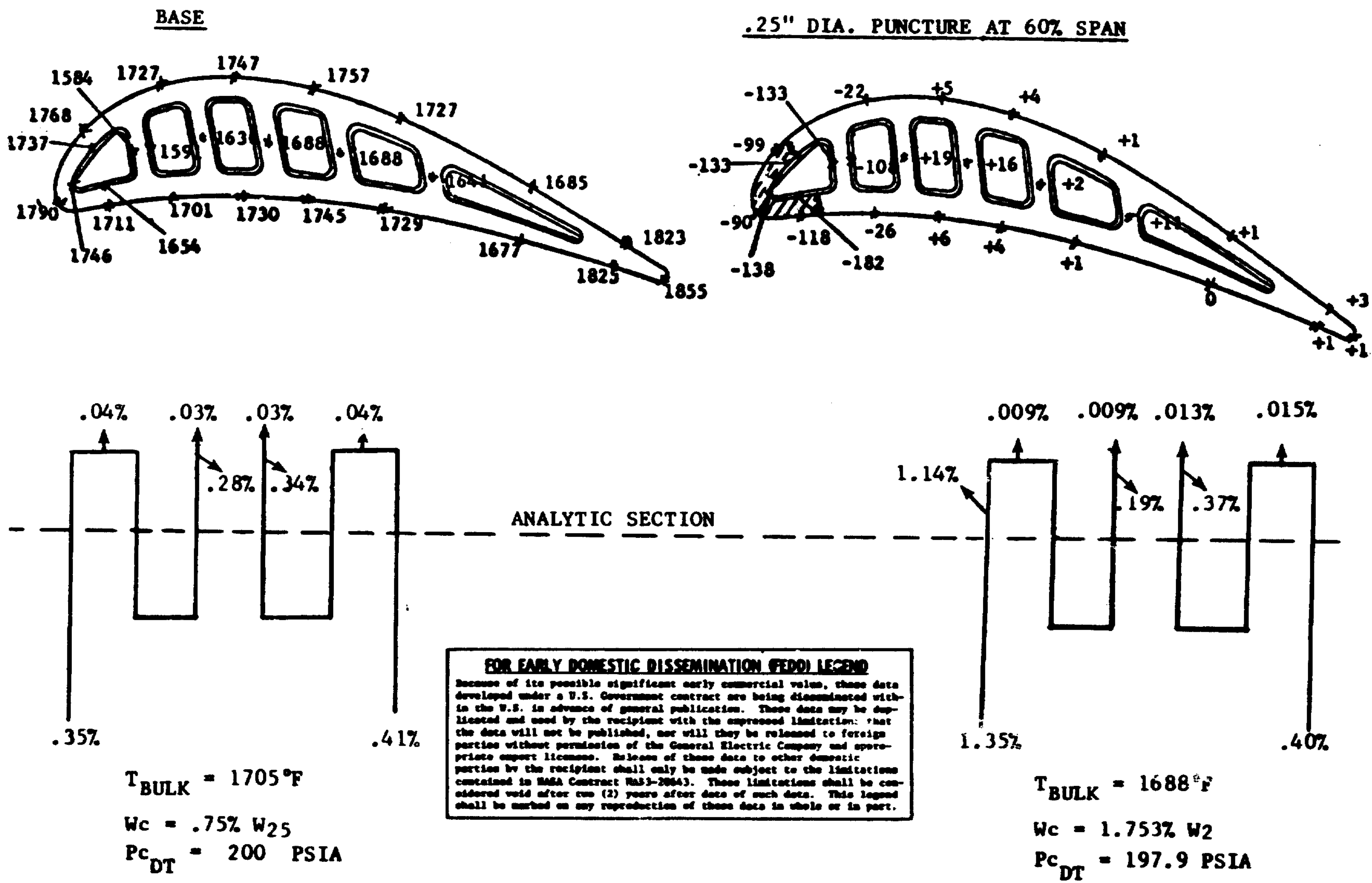
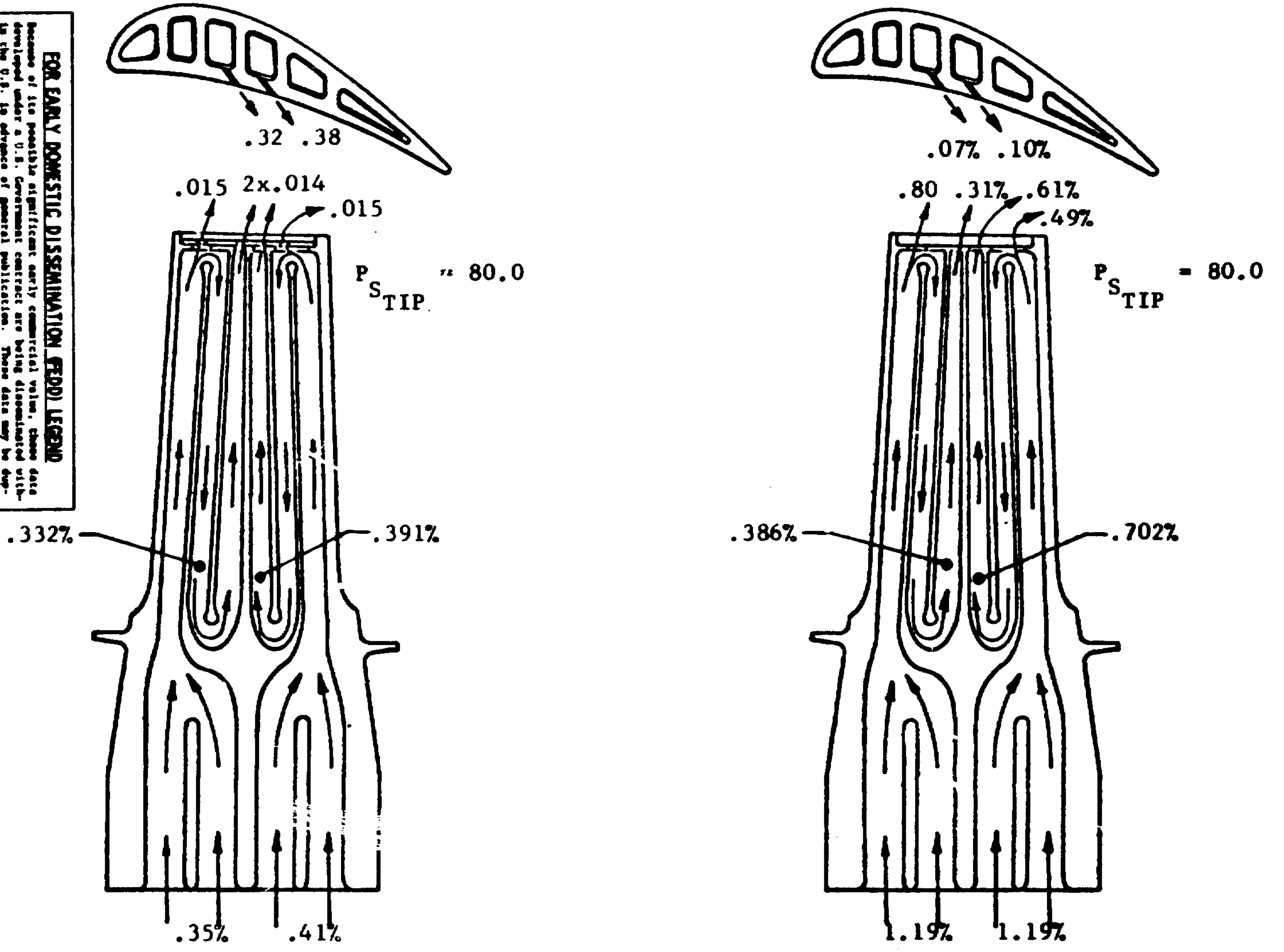


Figure 2.4-7. HPT Stage 2 Blade Leading Edge F.O.D Temperatures Pitch Section at SL Hot Day T/O, Steady State Conditions.



NORMAL BLADE FLOW

P_{TB} = 108.0 P_{DT} = 200.0 W_C = .76%W_2

BLADE FLOW WITHOUT TIP CAP

P_{TB} = 108.0 P_{DT} = 200.0 W_C = 2.38%

Figure 2.4-8. HPT Stage 2 Blade Cooling Flow Distribution Lost Tip Cap Simulation at Steady State Takeoff.

in an effort to recover as much as possible of the Mach 0.5 fan duct dynamic head. After diffusion, the HP ACC air is ducted to the modulation valve in the pylon. After flowing through the valve, the air is delivered to a 270° circumferential rectangular duct, in the area of the HP turbine, that is built into the core cowl. From this circumferential duct, the air is routed through four pipes to the impingement manifold surrounding the HP turbine casing. There are four impingement manifolds surrounding each of the two turbine stages as shown in Figure 2.4-9. The impingement manifolds are of rectangular cross section and allow for the required proximity of the 0.025-inch-diameter impingement holes to the casing clearance control rings and bolt flanges. The clearance of each stage in the HP turbine is accomplished by impinging the fan air on the casing, ACC rings, and bolt flanges. The compartment outside the HP casing has been isolated from the rest of the engine volume between the core engine and the inner fan duct flowpath by means of the fire safety wall. This is necessary since the pressure, after impingement, is lower than the fan duct static pressure at maximum ACC flow rates. With this isolated cavity, the spent ACC air can then discharge through the struts in the rear frame to the aft center body. From there the air discharges out the vent stinger at a velocity such that most of the thrust is recovered from the ACC air.

The HP turbine ACC analysis, to date, has been based on the assumption that the typical engine, in airline service, has been allowed to warm up to a stabilized idle temperature level. Production engine experience indicates that this is not always the case and shorter warm up times are quite possible. Because of this, an extensive study of the cold and warm engine start, short idle, and maximum takeoff transient has been conducted.

The purpose of this analysis was to define the impact on the minimum clearance after accel to maximum takeoff power. Presented in Table 2.4-III is the pinch clearance for the cold engine start and warm engine start. The warm engine start occurs, in airline service, when the engine has been shut down for a period of time, such as a half hour before being restarted. The rotor structure and casing cools off during the engine shutdown, but not at a uniform rate. The casing cools down at a faster rate than the rotor since

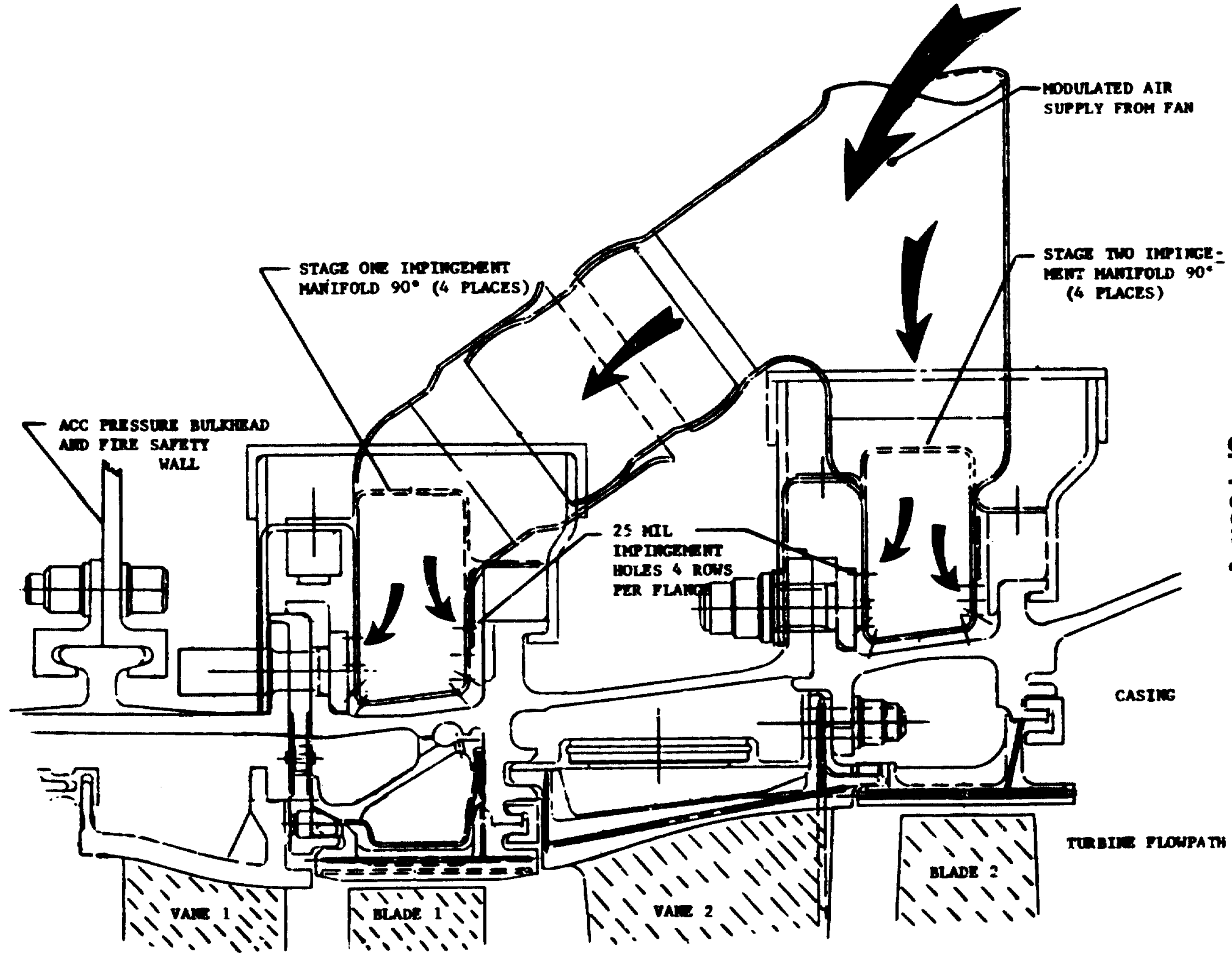


Figure 2.4-9. High Pressure Turbine Active Clearance Control Casing Cooling System.

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it is significantly less massive. The shutdown heat transfer analysis indicated that the Stage 1 casing cools to a temperature that was 125° F below the rotor average temperature after a half hour while the Stage 2 casing cools to a temperature that was 200° F lower than the rotor.

Table 2.4-III. Maximum Takeoff Pinch Clearance After Short Start.

	Time at Idle, Seconds	Cold Start Pinch, Mils	Warm Engine Start Pinch, Mils
Stage 1 Blade	200	16	12
	300	21	16
	400	23	
	500	25	
Interstage Seal	200	-15	-8
	300	-9	-4
	400	-4	
	500	0	
Stage 2 Blade	200	15	6
	300	19	16
	400	24	
	500	25	

The results of the analysis indicate that a substantial reduction in the takeoff turbine blade pinch clearance occurs on both the cold and warm engine when the engine is not given sufficient time to warm up. If the engine accels from idle to maximum takeoff occurs after only a 2 to 3 minute start and warm up, the pinch clearance can easily be reduced from the desired 25 to 10 mils. This then could easily produce a blade tip rub since the pinch will happen when high engine loads and vibration occur. In order to overcome this potential problem, a means of heating the casing, after start, has been devised. The casing heating scheme is accomplished by impinging 0.3% of core compressor discharge air on the outside of the casing during the idle power condition for

200 seconds. After 200 seconds of casing heating, a valve would turn off the warm compressor discharge air and the valve would remain closed until the engine had been shut down for at least 15 minutes. If an acceleration to maximum takeoff occurred while the casing heating valve was open, it would remain open for 30 seconds after reaching full power. There is no advantage in continuing to heat the casing beyond the 30-second point since the clearance pinch had occurred, and continuing to heat the casing would result in extra power loss caused by the compressor discharge air being taken from the cycle and excess clearances in the turbine.

Under normal engine operation, the casing heating valve would open just after the engine achieved idle rpm and would remain open for 200 seconds. The valve would then be shut for the rest of the engine mission. This would have virtually no impact on engine cycle performance and would yield significant improvement in the pinch clearances during takeoff. Presented in Table 2.4-IV is the pinch clearance for the cold and warm engine start with external casing heating for 200 seconds or 30 seconds after accel to full power, whichever occurred first.

The data from Table 2.4-IV shows that the warm engine start/short idle/maximum takeoff produces the most limiting pinch during takeoff. The improvement that can be achieved with the casing heating scheme becomes evident when a comparison is made between the two configurations. For the 200 seconds start/idle, there is a 9-mil increase in the pinch clearance for both the first and second stage blade tip and a 2-mil increase for the interstage seal during takeoff. This casing heating system still does not get the minimum turbine pinch clearance back to the desired 25 mils during maximum takeoff. The minimum pinch clearance is now 19 and 15 mils on the first and second stages, respectively, with the casing heating scheme. This indicates that the buildup clearance might have to be increased if 25 mils is truly required at maximum takeoff. Since this analysis is completely analytical and only has the heat transfer analysis practice from other engines factored in, the engine turbine thermal characteristics need to be defined before the optimum flight propulsion system ACC can be clearly defined. Because of of this, the casing heating system is being incorporated into the ICLS engine. This will then allow the complete evaluation of the ACC with and without the casing heating system. A true system evaluation can then be made to define the need of the heating system in the FPS.

Table 2.4-IV. E^3 HP Turbine Maximum Takeoff Pinch Clearance with External Heating During Engine Warmup.

	Engine Warmup Time, sec	Heating Time, sec	Cold Engine Pinch, mils	Warm Engine Pinch, mils
Stage 1 Blade	200	100-230	25	21
	*300-	100-330	29	22
	*300+	100-300	26	19
	400	100-300	27	
	500	100-300	28	
Interstage Seal	200	100-230	-10	-6
	300-	100-330	-3	-2
	300+	100-300	-6	-4
	400	100-300	-2	
	500	100-300	+2	
Stage 2 Blade	200	100-230	27	15
	300-	100-330	31	19
	300+	100-300	24	17
	400	100-300	26	
	500	100-300	28	

*300- indicates that the takeoff occurred prior to the heating system valve closure signal and this allowed the heating to continue 30 seconds into takeoff mission.

300+ indicates the heating system valve closed at the 300-second point, prior to takeoff accel.

High Pressure Turbine Rotor Structure Heat Transfer Analysis

The heat transfer analysis consisted of complete mission from cold engine start up through the flight mission and engine shutdown. Extensive iterations and feasibility studies were conducted in an effort to yield a transient temperature distribution that would not overstress or overtemperature any component. The areas that presented the biggest challenge were

- Compressor discharge seal disk bolt flange
- Inducer seal disk bolt flange
- Impeller disk bore and rabbet
- Stage 1 disk bore
- Interstage seal disk bolt circle and seal teeth
- Stage 2 disk bore
- Aft seal disk bolt circle.

All the problems were caused by transient temperature gradients except the interstage disk seal teeth which were temperature limited. Because of the massiveness of the disk bores, their thermal response was very slow. This created large temperature gradients during the early part of the takeoff transient.

An additional problem occurred at the Stage 2 disk bore where the axial and radial thermal stresses combined to limit the transient rupture life at that location. This problem was overcome when the heat transfer to the bore was increased. This increased the transient temperature right at the bore surface and caused a higher thermal compressive stress. When combined with the mechanical tensile stress, the effective stress was reduced. The end result was a disk that met the rotor structure life objectives of the growth engine. The improved heat transfer to the bore was accomplished by both increasing the flow under the disk to 0.2% W_{25} and reducing the radial gap from 100 to 50 mils near the center of the bore. The radial gap could not be reduced by a drop in the disk bore, so a 50-mil hump was placed on the outside of cover shaft at the center axial location of the Stage 2 bore. These two approaches not only increased the heat transfer by a factor of three, but the flow increase also improved the thermal response of the air flowing under the disk.

The HPT rotor shutdown temperature transients have also been defined. This data indicates that all the disks undergo a temperature reversal where the disk rim cools off to a temperature that is less than the bore temperature. This temperature reversal gradient puts the bore into compression and will have a definite impact on the cyclic life of the disks. The temperature gradients get up to 300° F at about an hour after shutdown. After this time, the temperature gradients start to diminish.

Stage 1 Blade

Even though the detailed design of the Stage 1 blade has been completed, work is continuing in support of the hardware fabrication. The current work on the Stage 1 blade involves the trailing edge cast cooling slots. Because of the delicate nature of the trailing edge cooling circuit, the casting yield from the first pour was very low. To alleviate the problem, the crossover holes from the last serpentine cavity have been increased in size as shown in Figure 2.4-10.

Work Planned

- Follow engine hardware fabrication for core and ICLS to assure design intent; make necessary adjustments for hardware that is not to print.

2.4.3.2 High Pressure Turbine Detailed Mechanical Nozzles and Structures Design

Technical Progress

The detailed design review of the mechanical design was presented at NASA-Lewis on October 10, 1980. Approval of the design by NASA was officially received in December 1980. The approval then permitted the balance of the hardware not on order (as long lead items) to be released to manufacturing.

- APPROXIMATE 4X INCREASE IN CORE STIFFNESS AT PROBLEM LOCATION

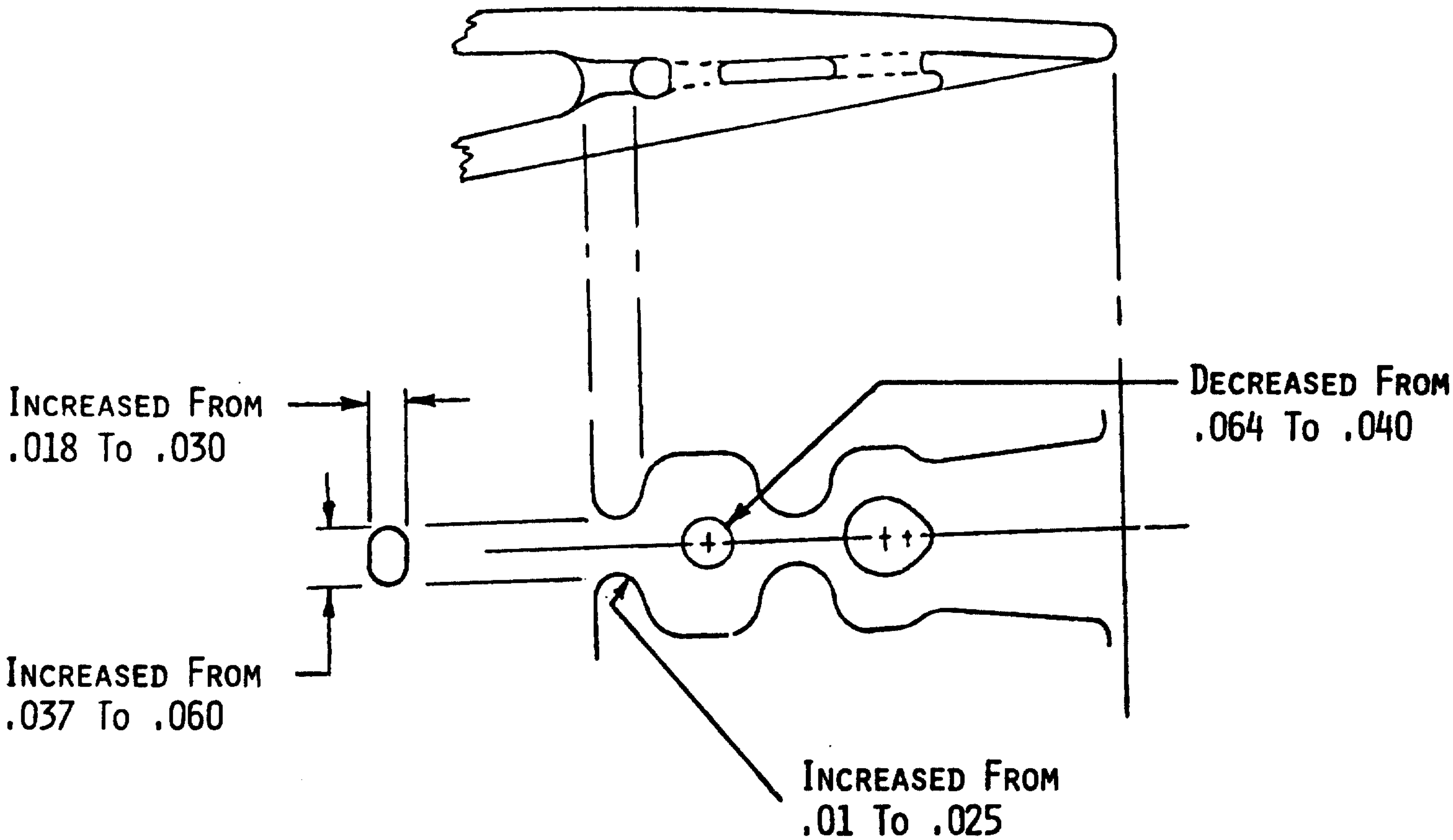

Figure 2.4-10. Stage 1 HP Blade Fix.

The detailed design culminates an 18-month effort by mechanical, heat transfer, aero, and materials engineering to define the final turbine geometry configuration.

Assembly drawing for the Stage 2 nozzle and shroud support is in process of completion. The drawing includes grind dimensions for Stage 1 and 2 shrouds, and Stage 2 nozzle inner honeycomb seal.

The stress analysis for the impingement manifold has been updated to include the use of compressor discharge pressure (CDP) air. The use of CDP air is only used during flight conditions with no idle warmup (only 200 seconds have elapsed after initial engine start up). The heating of the casing will compensate for the reduced clearance reduction due to the casing being at a lower temperature.

Work Planned

- Release Stage 2 nozzle and shroud assembly.

2.4.3.3 Detailed Mechanical Blades and Rotor Design

Technical Progress

The detailed design review for the HP turbine blades and rotor structure was presented at NASA-Lewis on October 10, 1980. The design was approved by NASA in December 1980. The detailed design consisted of an 18-month effort by the mechanical, heat transfer, aero, and materials engineering to define the final turbine geometry configuration.

Extensive stress and life analyses to determine the low cycle fatigue (LCF) life capabilities for all the major components were accomplished during this time period. The analyses included the effect of flight transient conditions on stress and LCF life capabilities. Extensive uses of finite-element analysis was accomplished to determine the localized stresses for the various components.

Work Planned

- All René 95 and AF115 HIP material shall be analyzed using residual life analysis methods
- Release turbine rotor assembly drawing
- Initiate balancing procedure methods.

2.4.3.4 Manufacturing Interface and MRB Discrepancies

Technical Progress

Major effort by engineering to cover manufacturing cycle, schedules, and dimensional discrepancies for various components were in the following areas.

Stage 1 Nozzle

Engineering reviewed the band castings' core and wax die fixtures. Changes were recommended by the casting vendor to improve yields and were incorporated in the design. The band hold drilling features were also reviewed and approved.

Due to band core die late delivery, the manufacturing and fabrication schedule for the nozzle assembly had to be revised. Schedule milestones are being maintained to determine the progress of work, and identify areas where schedule improvements can be made.

Stage 2 Nozzle

Engineering approved the outer band casting dimensional inspection after review. The inner band castings are presently being reviewed.

The vane airfoil cores are showing indications of low core yields. The major problem is cracking of the trailing edge turbulence promoters and the inner flowpath core support. A review is underway to determine the extent of the problem prior to defining a solution.

Stage 1 Blade

Blade core cracking along the trailing edge crossover holes has been the major problem limiting the completion of the blade casting process. Observed breakout of cores during casting, plus the very low core yields, necessitated a redesign of the trailing edge hole sizes.

A redefinition of the trailing edge crossover hole size has been made. The core die rework to include these changes is being completed.

Stage 2 Blade

The only problem discoverd in the process for Stage 2 blade casting was core cracking in the radii of the three small holes located at the bottom of the turning ribs. These holes are used as part of the cores to improve airfoil rib to shank support. The radii around these holes have been increased and core yields have increased.

The first two casting molds processed show casting yields of approximately 50%; further improvements are expected.

Stage 1 Ceramic Shrouds

Review of the wax pattern for the ceamic shroud showed the pegs to be out of print. Core die rework to correct this problem was accomplished and the review of the second wax pattern was approved.

Stage 2 Solid Shroud

The Stage 2 solid shroud received dimensional and visual inspection by Engineering and was approved. Forty shrouds have initially been cast and final inspections are being completed.

Work Planned

- Maintain manufacturing support
- Determine ways to improve Stage 2 vane core yields
- Continue to review any dimensional discrepancies and determine if rework is required.

2.4.3.5 HPT Core - ICLS Support

Technical Progress

Instrumentation definition and locations have been completed. Turbine rotor sensor routing has been defined and the disk rework drawing, which adds holes for instrumentation routing, has been issued.

Blade and nozzle instrumentation requirements have also been defined. Rework drawings for these parts are in process of initiation.

Work Planned

- Release blade and nozzle rework drawings for instrumentation
- Review drawings released through evaluation for instrumentation routing of rotor and stator parts.

2.4.5 High Pressure Turbine Cooling Development Testing

Technical Progress

Due to the critical nature of the HPT and LPT active clearance control, it is important that the system be completely understood before engine testing if the total system payoff is to be realized. The active clearance control system for both the HPT and LPT consists of a scoop to recover fan discharge air, a control device to schedule the correct amount of air, and a distribution system to cool the exterior casings of the respective turbines. Spent air, after cooling the casing, it routed through the LPT aft frame to the stinger where it is injected out the back with sump purge air and compressor purge air.

Between the fan discharge and stinger, there is not a dominating pressure loss so not pressure loss in the system can be considered insignificant. Due to this requirement, it is essential that ACC air extraction from the fan duct be carried out as efficiently as possible. The current extraction system calls for a high performance scoop installed in the side of the pylon skirt. The scoop, shown in Figure 2.4-11, will be split down the middle to segregate the HPT and LPT active clearance control air.

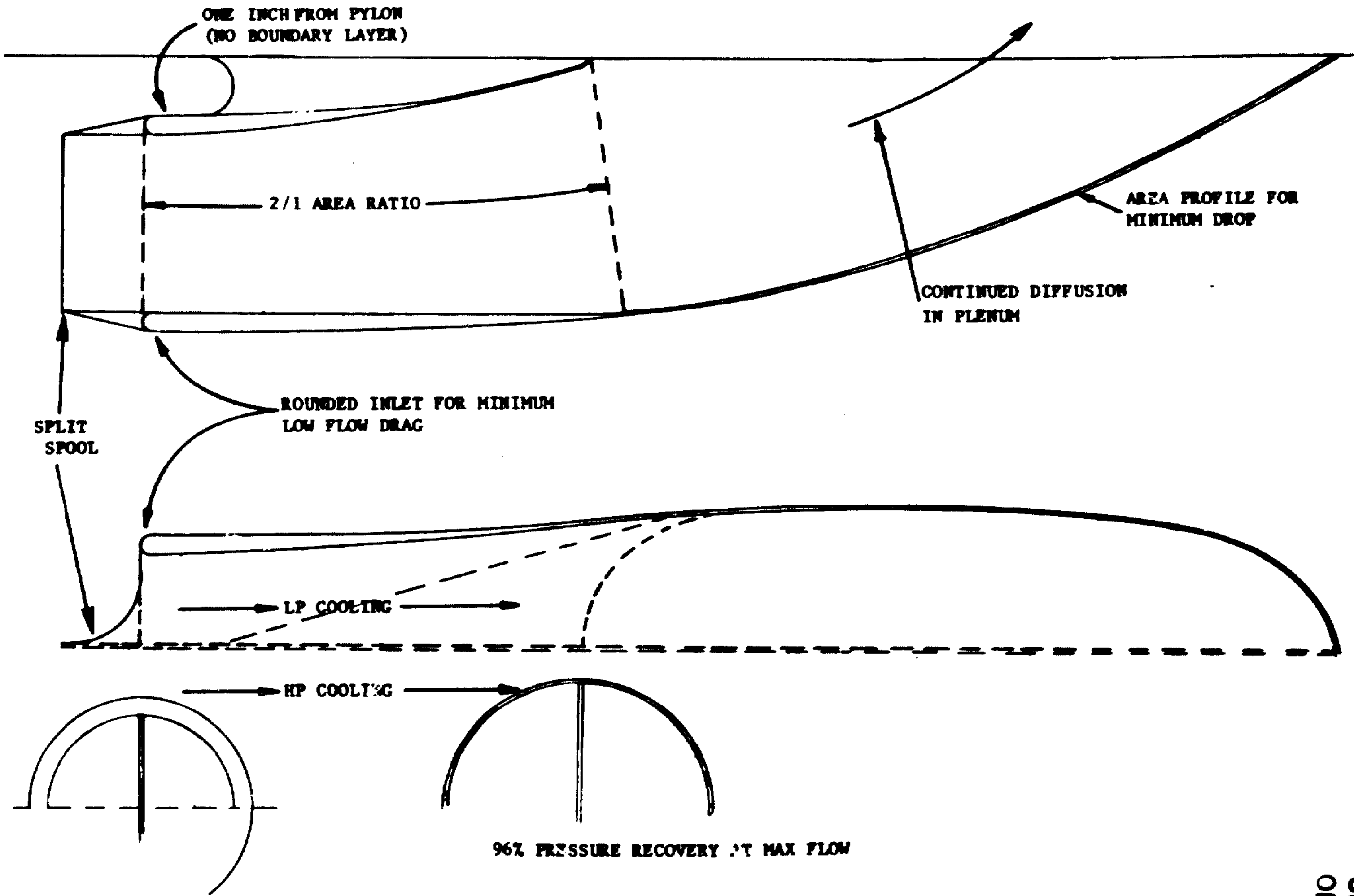

Figure 2.4-11. HP/LP ACC Scoop Features.

The model test, carried out in a wind tunnel at one-half scale, will have three objectives. Of primary importance will be detailed data on recovery pressure for various scoop flows throughout the range of fan discharge air Mach numbers. It is of equal importance to verify the extent of interaction between the LPT and HPT scoop flows. Secondary to the former will be an attempt to experimentally verify the analytical prediction of the aerodynamic penalty associated with putting the scoop in the fan duct.

During this reporting period, the detailed drawings of the ACC scoop model and test duct assembly were completed and released for hardware acquisition in November 1980. The completed scoop assembly was received from the vendor in February 1981 and is shown in Figure 2.4-12. Instrumentation of the scoop was completed shortly after receving the hardware. Installation of the instrumented scoop assembly, shown in Figure 2.4-13, in the test facility was completed on March 18, 1981 with flow and instrumentation checkout taking place at the time of this writing. Testing will begin immediately after checkout is complete.

The testing will cover a range of duct Mach numbers (0.2 to 0.55) in order to simulate the conditions that exist in the fan duct throughout the E^3 flight envelope. Flow extraction will range from zero to a maximum flow of 0.122 pps per side. Thirty-five test points will be set at various combinations of duct Mach numbers and scoop flows. Recovery plenum pressure, and upstream and downstream total pressure traverse measurements will be taken at each point. The results will be analyzed so that accurate scheduling of ACC air will be achieved in the engine test. Wake drag losses will be studied to ensure that they are within acceptable limits.

2.4.7.1.1 Ceramic Shroud Process

Technical Progress

Ten ceramic shrouds were tested in a CF6-50 factory engine. Based on the posttest analysis, the superpeg shroud was selected as the prime ceramic shroud configuration for E^3 core and the ICLS engine testing. The wire mesh configuration, which did not perform as well as the superpeg design, has been deleted from the program.

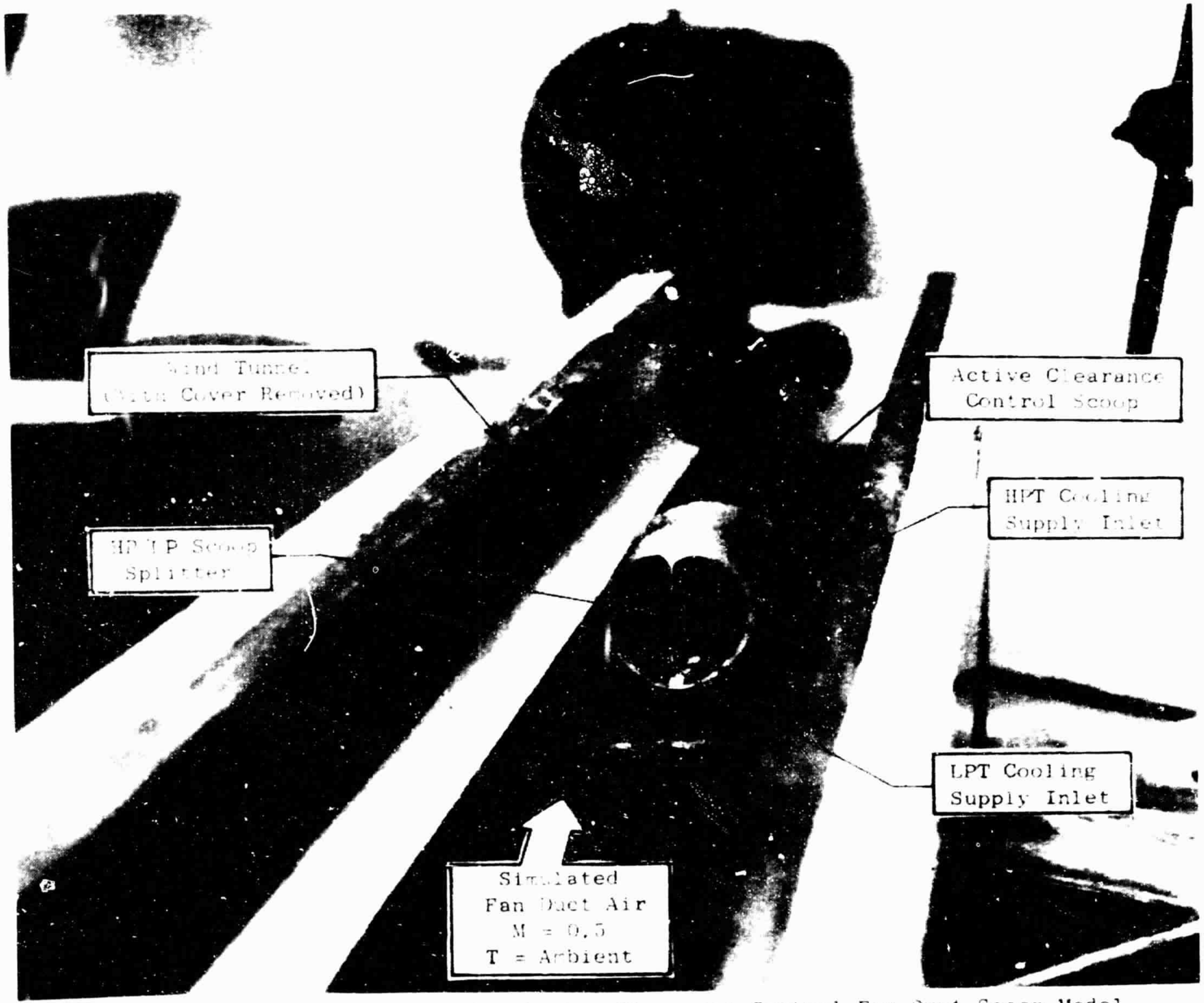

Figure 2.4-12. HP/LP Turbine Active Clearance Control Fan Duct Scoop Model.

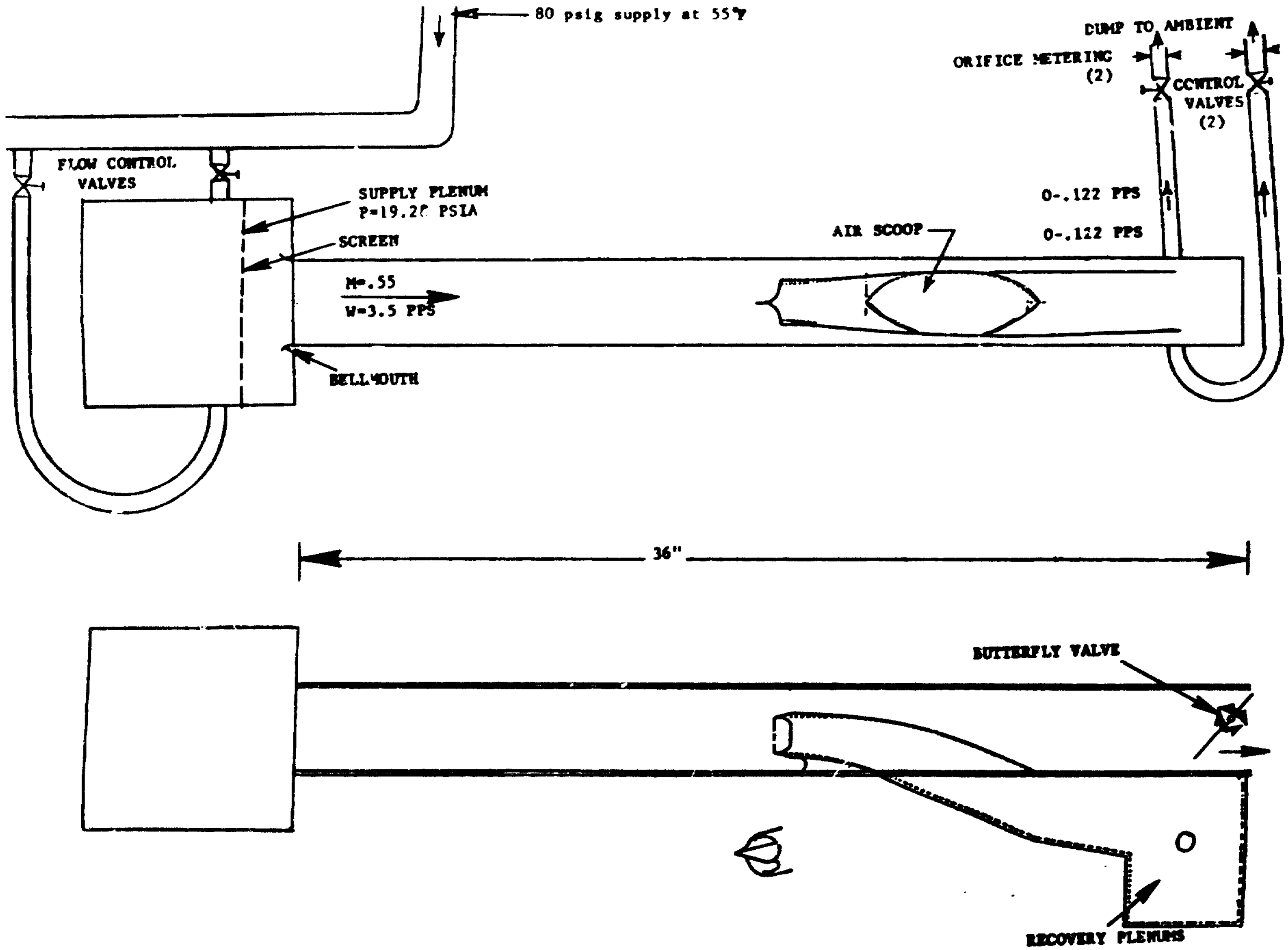

Figure 2.4-13. HP/LP ACC Scoop Model Test Airflow Schematic at Design Point.

An effort was then initiated to improve the plasma spray process for coating the superpeg shroud. Superpeg panels were fabricated to simulate the superpeg shrouds and were coated using several variations in the plasma spray process. Improved spray techniques have been identified but slight refinement is necessary to optimize the coating process.

Two superpeg shrouds were fabricated and plasma spray coated with ZrO_2 - Y_2O_3 powder obtained from two different vendors. These shrouds are awaiting thermal shock testing to identify a ZrO_2 powder source.

Development of a nondestructive evaluation procedure for detecting ZrO_2 coating delaminations is in progress. Holographic and infrared techniques are presently being evaluation.

A program plan was formulated to provide for fabrication of the Stage 1 and 2 superpeg and solid shrouds for E^3 core and ICLS testing.

Work Planned

- Complete superpeg spray process identification
- Identify ZrO_2 powder vendor
- Complete development of coating NDE procedure.

2.4.7.2.1 Thermal Barrier Processes

Technical Progress

The objective of this program is to develop a thermal barrier coating (TBC) system and demonstrate its adequacy for the energy efficient engine. Previous semiannual reports have described the progress of earlier tasks in which the original eight candidate coating systems were reduced to three, and then to a single coating system. The selected coating is a two-layer (duplex) system consisting of a Ni-22Cr-10Al-1Y bond coat and a ZrO_2-$20\%Y_2O_3$ top coat.

A substantial amount of processing effort aimed at obtaining uniform, durable coatings on four high pressure turbine components has been reported.

The coating process selected for the vanes consists of manual arc plasma spraying of both coating layers at atmospheric pressure. For the blades, automatic manipulators were used with the bond coat applied by vacuum plasma deposition (VPD) and the ceramic coating applied by conventional atmosphereic spraying.

The results of mechanical property testing comparing thermal barrier coated specimens to uncoated specimens was reported in the previous semiannual report. The presence of the TBC was found to have no measurable effect on the low cycle fatigue, high cycle fatigue, and stress-rupture properties of X-40, René 80, and DS René 150.

Also reported was the coating of HPT components for rig and engine testing and the damage of TBC Stage 1 HPT blades in the cascade rig test due to problems with the cooling air pressure.

Cascade Rig Test

The thermal barrier coated CF6-50 Stage 1 HPT blades, which were damaged in the cascade rig test, were evaluated using metallography and electron microprobe analysis. A thin deposit which formed on the surface of the ceramic coating during the test was found to contain Fe, Ni, Cr, Co, Cu, and Al. The source of this deposit is believed to be contaminants in the air supply lines to the test cell and possible surface material losses from the transition components which funnel the hot gases from the combustion to the airfoils.

The TBC remained intact in most areas, even adjacent to the leading edge of the airfoils where local melting of the René 150 blades had occurred. Metallographic examination indicated that the TBC remained adherent until the substrate material was lost. Microprobe examination of the blade near the molten zone showed strong aluminum depletion in the bond coat and the EA NiCrAlHf environmental coating. In addition, this region showed a homogenization of the nickel content of the bond coat, EA coating, and the substrate. In a region which had experienced lower temperatures during the test (trailing edge, suction side), these compositional changes were not observed.

While problems with the test apparatus limited the information which was obtained in the cascade rig test, the durability of the TBC under the severe test conditions was encouraging.

Engine Testing

Six thermal barrier coated blades w :e delivered to CF6-50 Engine Systems for engine testing.

The blades were assembled in a rotor, balanced, and tip ground. No damage was observed on the TBC blades after the tip grinding operation. The blades were then assembled in the engine and scheduled to run for 1500 "C" cycles.

Examination of the blades after 16 hours of engine checkout revealed coating damage on all of the TBC blades. The damage was restricted to the upper third of the leading edge on the airfoils and appeared to be small pock marks in the ceramic layer. The damage is believed to have been caused by impact of material lost from development shrouds in the first stage of the high pressure turbine. Subsequent boroscope examination of the blades during the test showed no observable progression of the damage to the leading edge after 316 "C" cycles. However, examination after 467 "C" cycles showed additional ceramic coating loss at the leading edge. No coating loss was observed at any other location on the blades. The test was terminated after 626 "C" cycles due to turbine FOD.

Extensive loss of blade material occurred at the leading edge and blade tip for all Stage 2 blades. The ceramic coating losses from the remaining airfoil surface were about 40% to 70% on the pressure side (concave) and 30% to 40% on the suction side (convex). The bond coat, however, remained considerably intact over all the remaining airfoil surface.

The photomicrograph in Figure 2.4-14 shows the coating damage which was sustained on the suction side of the airfoil near the leading edge. Nearly all of the ceramic top coat was lost due to impact damage. The metallic deposit seen in this micrograph was thickest in this region and covered all of the pressure side of the airfoil and about half of the suction side. The

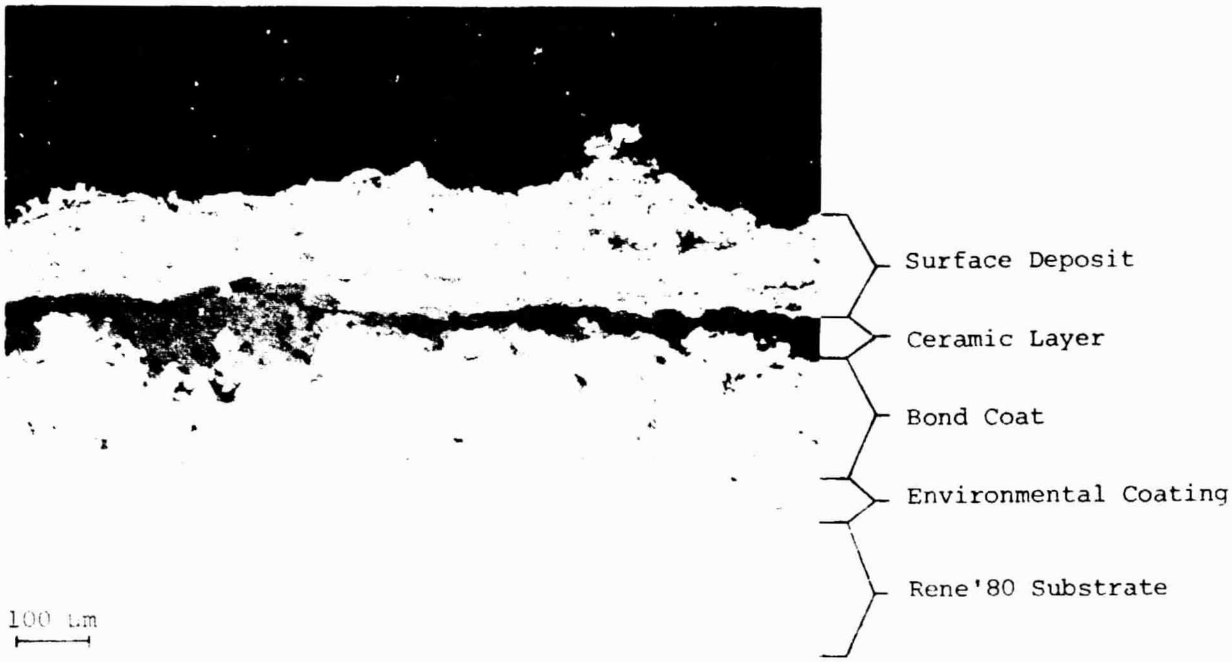

Figure 2.4-14. Microstructure of Engine-Tested Stage 2 Blade on the Suction Side Near the Leading Edge Where Severe Impact Damage Had Occurred.

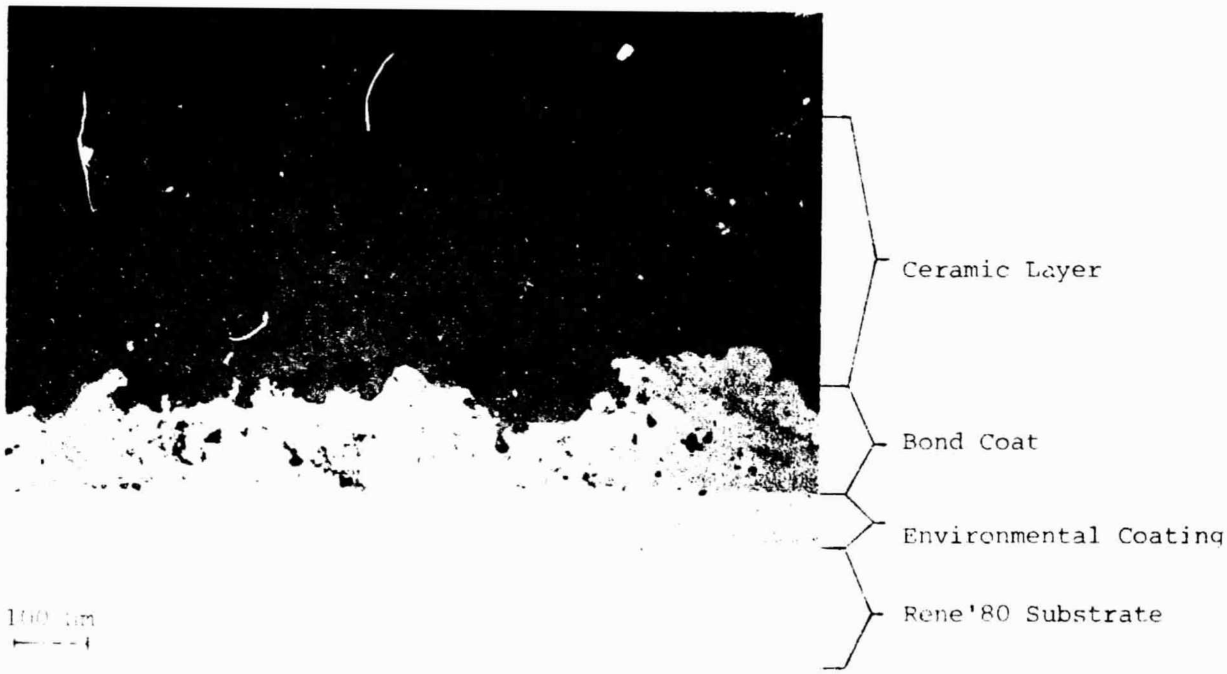

Figure 2.4-15. Microstructure of Engine-Tested Stage 2 Blade on the Suction Side, Mid-Span Where Little Impact Damage Had Occurred.

deposit was analyzed to be René 80 material originating from Stage 1 blades. The coating microstructure in the area which had not sustained impact damage is shown in Figure 2.4-15. The coating is in very good condition and shows no evidence of cracking or separation at the bond lines. X-ray diffraction analysis of the ceramic coating showed the 20% Y_2O_3-ZrO_2 to be mostly cubic, with a small percentage of the monoclinic phase.

Formation of an oxide layer at the top-coat/bond-coat interface, such as had been observed in laboratory specimens, was not generally apparent. This is probably due to the relatively short time (~166 hours) at elevated temperature in the engine test. However, in some areas near the leading and trailing edges, formation of the oxide layer was observed. These areas experienced much higher temperatures than the bulk of the airfoil as was evidenced by the variation of coloration of the oxidation product which formed on the René 80 deposit on the TBC surface.

During this report period, a number of additional CF6-50 high pressure turbine components were thermal barrier coated and submitted to CF6-50 Engineering for engine tests scheduled to start in April 1981. These components are described below.

Stage 1 HPT Vane Segments - The first stage vanes were coated in July 1980. Eight vane pairs were separated and the individual vane halves were coated in designated areas with nominal thicknesses of 0.005-inch NiCrAlY bond coat and 0.012 inch ZrO_2-20%Y_2O_3 top coat. Both coating layers were manually sprayed in air. The ceramic surface was smoothed by hand polishing with grit paper under flowing water. The vane halves were then rewelded and inspected. (This procedure had been evaluated on a vane which subsequently underwent a series of thermal cycle proof tests. The TBC showed no damage in the rewelded zone or in any other area.) Four vane pairs have been assembled for engine testing.

Stage 2 HPT Vane Segments - The areas selected to be coated on Stage 2 vanes did not require the cast pairs to be separated and rewelded; otherwise the TBC and the coating procedure were the same as for Stage 1 vanes. Eight vane segments were coated and submitted for engine testing.

Stage 2 HPT Blades - Ten Stage 2 blades have been submitted for engine testing. All of the blades were coated with nominal thicknesses of 0.005 inch NiCrAlY bond coat and 0.010 inch ZrO_2-Y_2O_3 ceramic top coat. The bond coat was applied in a low pressure, inert gas environment using automated manipulation. The ceramic surface was polished to a smooth finish to improve aerodynamic efficiency.

Three of the blades have a ZrO_2-20%Y_2O_3 ceramic top coat which was applied by a programmed robot manipulator. During this report period, six additional blades were coated in a manner similar to that described above, except that two ceramic powder compositions were used. Three of the blades were coated with ZrO_2-20%Y_2O_3 and three with ZrO_2-8%Y_2O_3. Both powders were obtained from the same source (Metco, Inc.) and were manufactured by the same process. Of these six, two of each are being used for engine tests.

The last three of the ten blades being engine tested are part of a group of blades coated at GE-CRD under NASA Contract NAS3-21727. The ceramic top coat is ZrO_2-8%Y_2O_3 manufactured by Zircoa-Corning, and it was applied by manual plasma spraying. There were several other material and coating process variables which differentiate these blades from those coated at GE-AEBG.

Work Planned

The next planned activity is the evaluation of engine-tested, thermal barrier coated, high pressure turbine components from a CF6-50 engine scheduled for test in the second quarter of 1981.

2.4.7.2.2 Materials Development - Thermal Barrier Cascade Rig Test

Technical Progress

The objective of the thermal barrier coated Stage 1 blade test in the cascade rig is to determine the coating durability in a simulated engine environment. This test, prior to committing the Stage 1 blade for engine test, would provide the level of experience necessary to ensure success during engine tests. The cascade rig test would also be useful in comparing actual temperature measurements versus predicted temperatures. A total of 1000 "C" cycles were planned. The test's excursion would be from idle to maximum transient takeoff, 2-minute hold at takeoff, and deceleration to idle conditions.

During the initial cycling, it was apparent that the measured temperatures were well below the predicted temperatures. The gas stream and coolant temperatures were therefore increased. Additionally, the level of cooling flow to the blade was reduced by a reduction in the air supply pressure to the blade.

After a total of 605 cycles, the blades were inspected and two of the blades were found to have experienced local leading edge burnout from approximately 55% to 85% span. The blades were supplied to the Materials Engineering group for metallurgical examination. The results are reported in Section 2.4.7.2.1.

The cause for the burnout was concluded to be due to the pressure reduction of the air cooling feed to the leading edge circuit. The cooling pressure had been reduced to a level such that no leading edge flow existed. The temperature reading at the leading edge pitch line under this condition was measured to be 2186° F. At this point, the cooling flow was increased for additional cycle tests and the blades, which showed the blade leading edge burnout, were inspected.

It became apparent that further tests would not be desirable since the measured temperatures could not be obtained without a redesign of the blade cooling circuit. This program has, therefore, been terminated until further evaluation of the type of cooling circuit definition that should be made.

Work Planned

No further work is planned at this time.

2.4.7.2.4 CF6-50 Thermal Barrier Coating Heat Transfer Evaluation

Technical Progress

The purpose of this program is to supply the heat transfer support in the development of the zirconia coating for the turbine flowpath components. As the first phase of that development, the zirconia will be evaluated in the CF6-50 turbine.

The problems encountered during cascade testing of the Stage 1 zirconia-coated blade have been reported in the previous semiannual report. No further testing has been done on that blade. However, a considerable amount of analytical work has been done on reassessing and updating temperature distributions on the HP turbine Stage 2 blades and vanes using the latest thermal barrier conductivity data.

Stage 2 Vane

The Stage 2 high pressure turbine vane coating configuration for the CF6-50 engine test is shown in Figure 2.4-16. This configuration was selected in order to minimize the thermal unbalance between the airfoil bulk and leading and trailing edge temperatures. The vane airfoil and bands are integrally cast in pairs thus the entire airfoil surface could not be coated because of the line-of-sight coating technique. The trailing vane (Vane B) was therefore not coated on the trailing edge to prevent a thermal mismatch between the suction and pressure side surfaces.

The vane airfoil and bands were partly coated (Figure 2.4-16) with a 20% yttria-stabilized zirconia (ZrO_2-20%Y_2O_3) bonded to base René 80 by a layer of NiCrAlY. The thickness of the thermal barrier coating and bond layer was 0.012 and 0.005 inch, respectively. A three-dimensional steady state heat transfer analysis has been carried out to investigate the effect of thermal barrier coating on vane temperature. The results are shown in Figures 2.4-17 through 2.4-20. Figures 2.4-17 and 2.4-18 show the outer band temperature distribution, with and without thermal barrier coating for Vanes (A) and (B).

NOTE: TRAILING VANE AT TRAILING EDGE
NOT COATED.

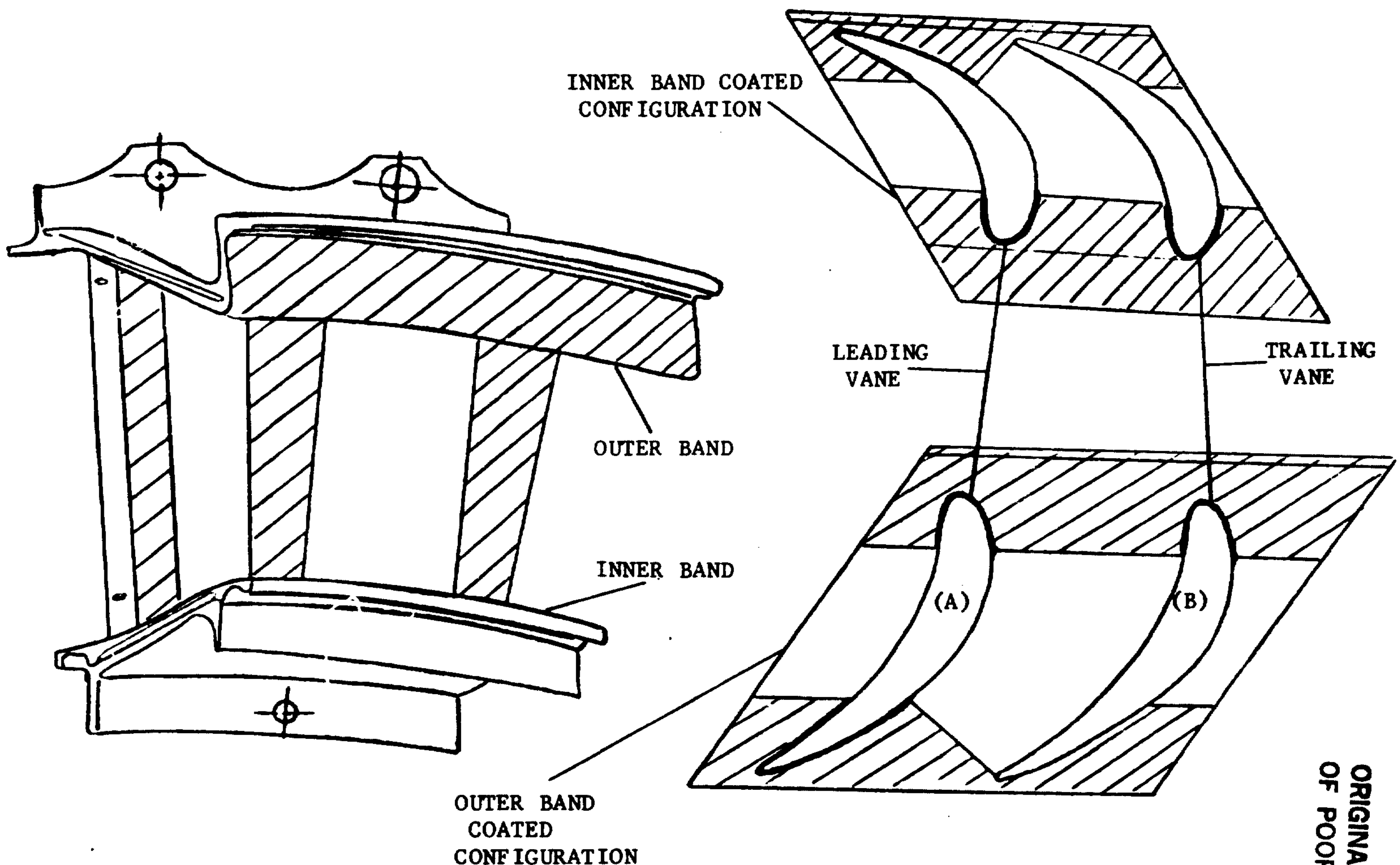

Figure 2.4-16. Stage 2 Vane Thermal Barrier Coating Configuration.

- Outer Band Temperature Distribution Coated Versus (Uncoated)
- Average Temperature 1753° F (1813)
- Shaded Area TB Coated (20% Y_2O_3)

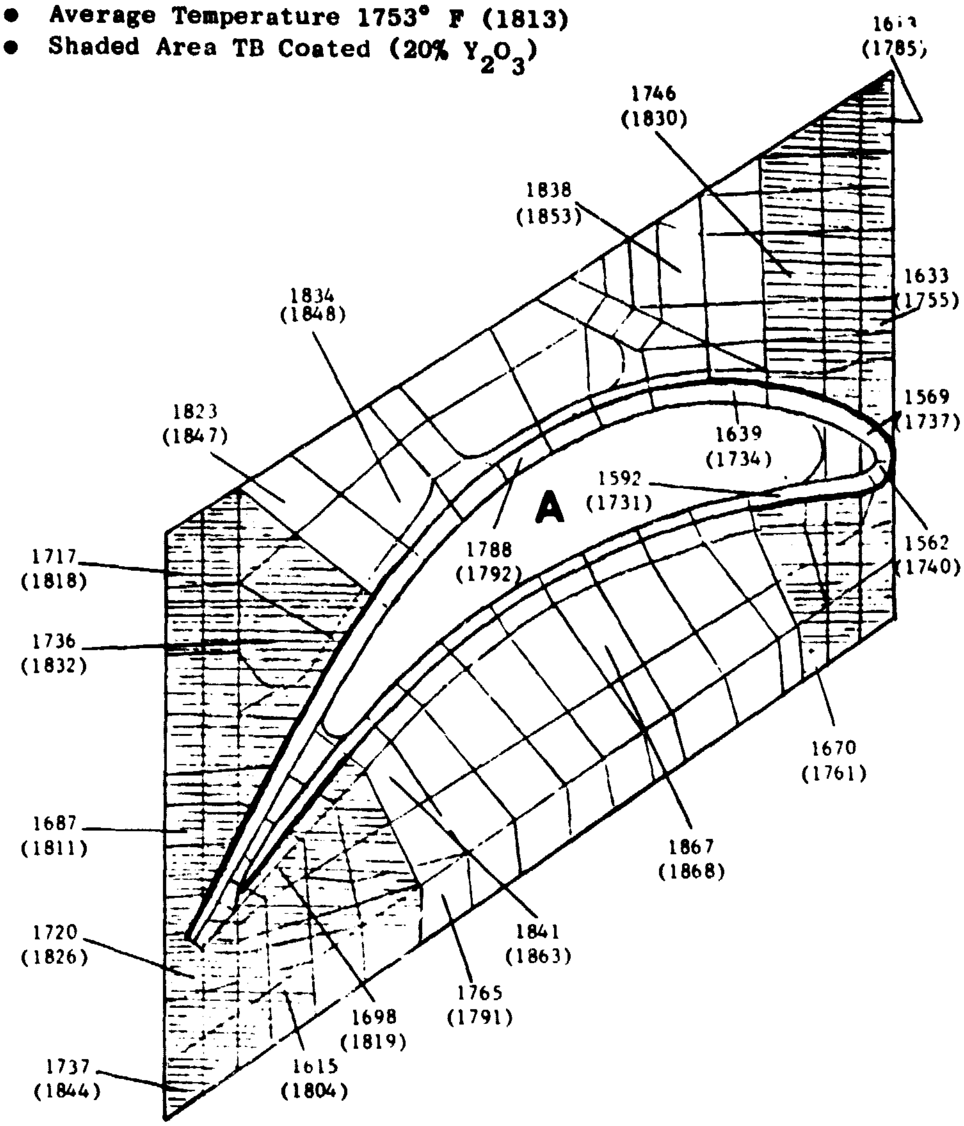

Figure 2.4-17. CF6-50 Stage 2 Vane A and Outer Band.

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- Outer Band Temperature Distribution Coated Versus (Uncoated)
- Average Temperature 1771°F (1813)
- Shaded Area TB Coated (20% Y_2O_3)

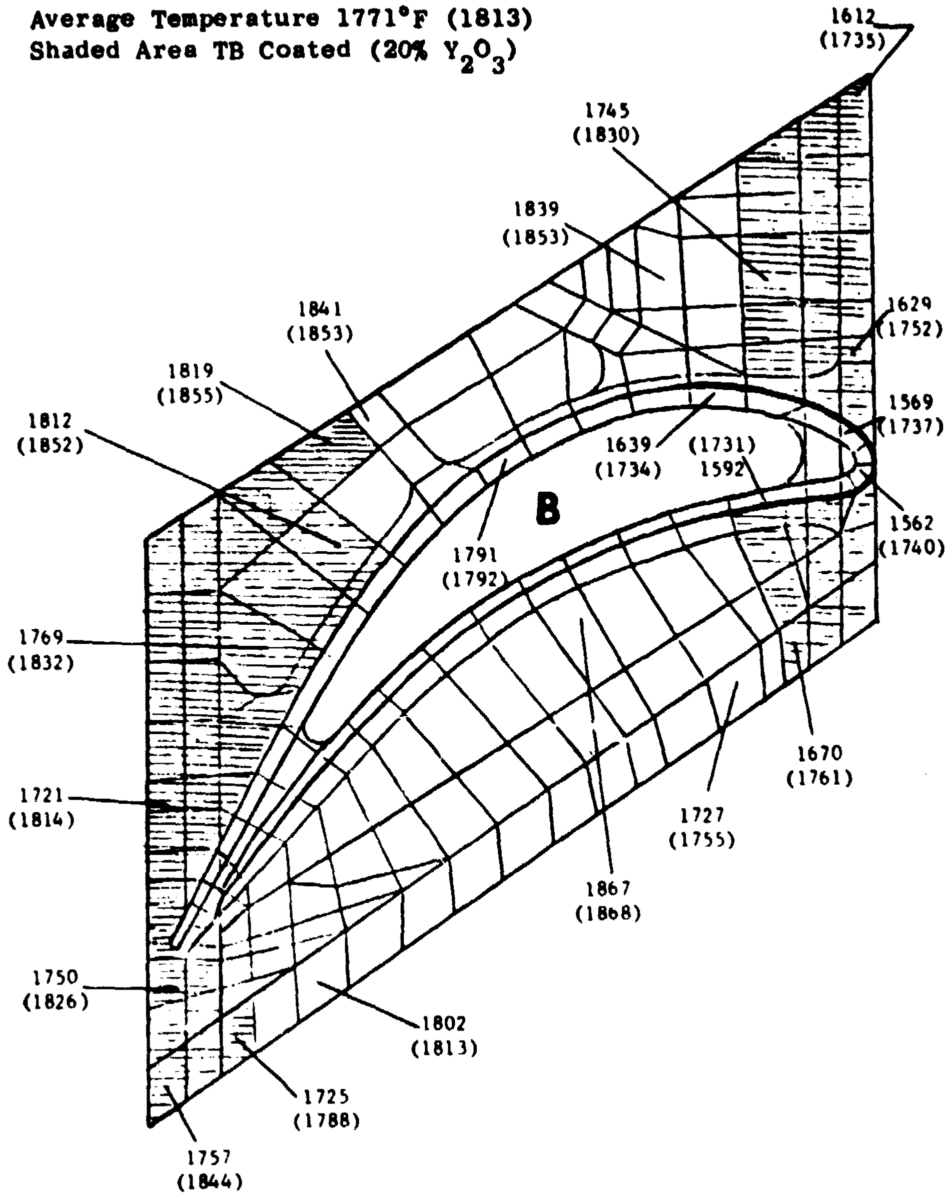


Figure 2.4-18. CF6-50 Stage 2 Vane B and Outer Band.

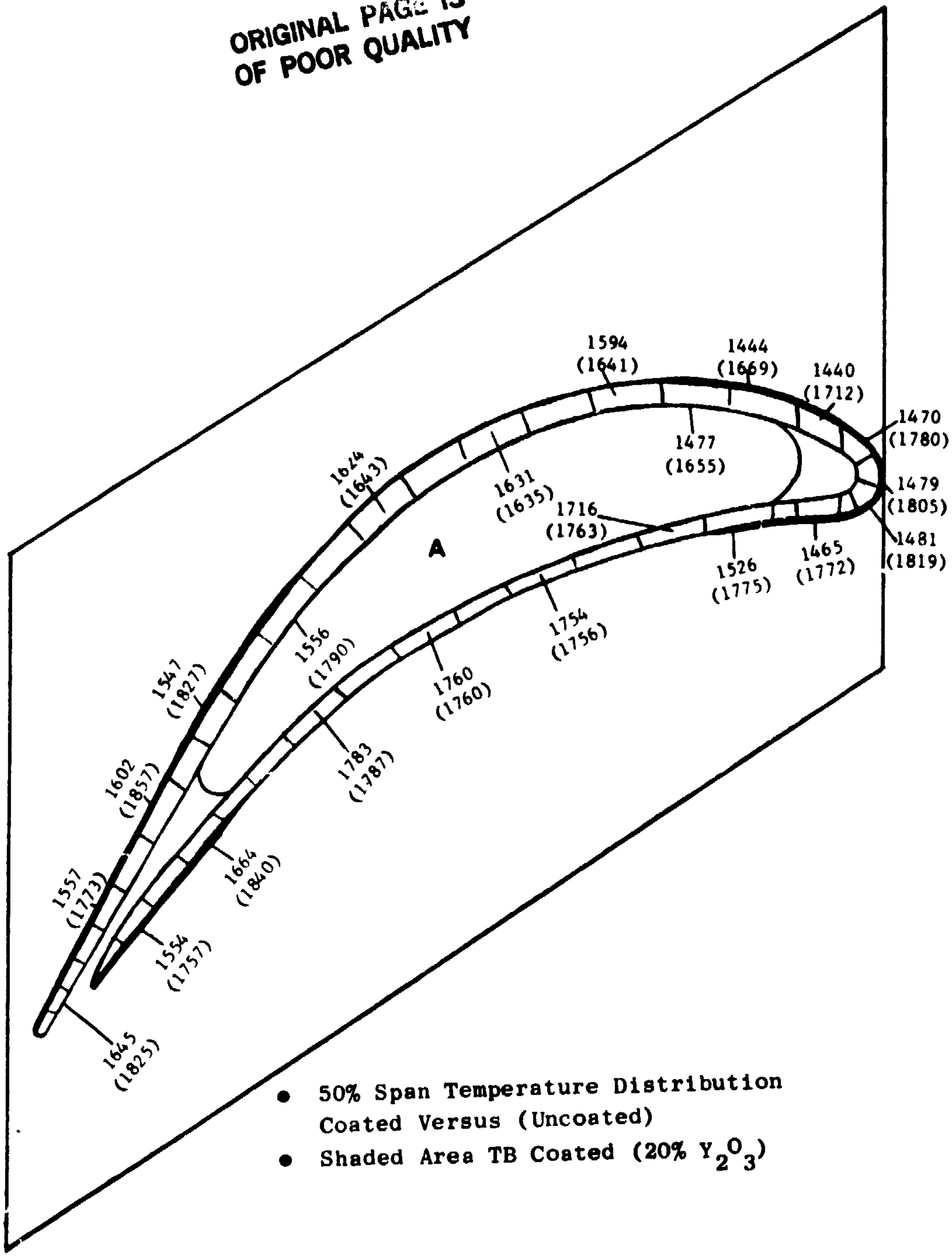


Figure 2.4-19. CF6-50 Stage 2 Vane A.

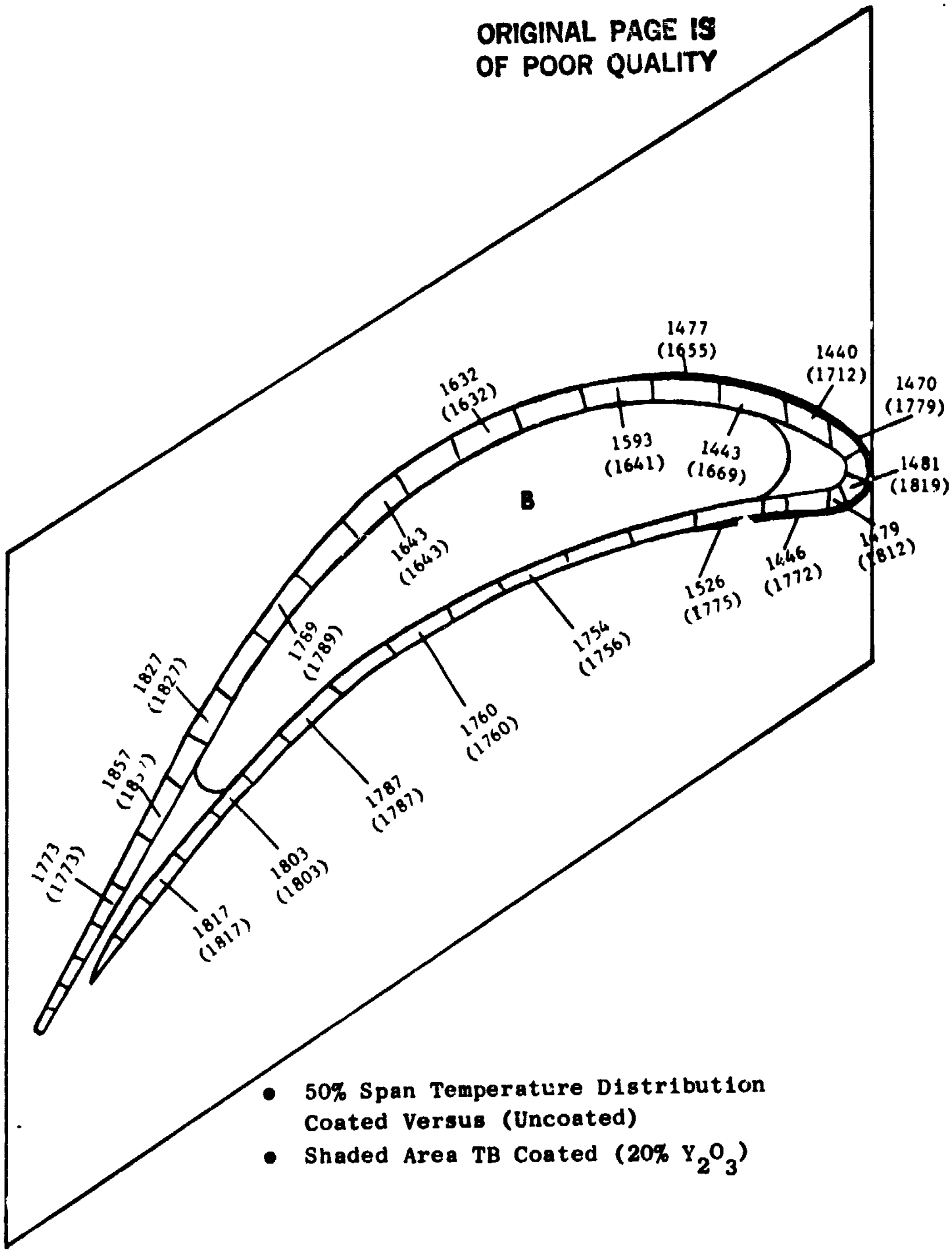


Figure 2.4-20. CF6-50 Stage 2 Vane B.

Figures 2.4-19 and 2.4-20 show the vane aerofoil temperatures (with and without TBC) at 50% span for Vanes (A) and (B). It can be seen that the temperatures in the coated region are substantially lower than those in the uncoated region.

Four pairs of vanes have been thermal barrier coated and are shortly to be engine tested.

Stage 2 Blade

Considerable analytical heat transfer work has been carried out on the thermal barrier coated CF6-50 Stage 2 HP turbine blade. Transient heat transfer analysis has been completed, evaluating the effect of thermal barrier coating spallation on blade temperature. A three-dimensional, steady-state heat transfer analysis has also been completed on this blade.

Two combinations of yttria-stabilized zirconia coatings have been evaluated: 8% yttria-stabilized zirconia and 20% yttria-stabilized zirconia.

The thermal barrier coating thickness was 0.010 inch and was held on to the base René 80 by a 0.004-inch layer of NiCrAlY.

The three-dimensional, steady-state heat transfer analysis was carried out to evaluate the effect of thermal barrier coating on blade temperature. A temperature distribution at 20% blade span is shown in Figures 2.4-21 and 2.4-22 for the 8% and 20% Y_2O_3 coatings. It can be seen that the thermal barrier coating has a considerable effect on blade temperature, reducing the bulk metal temperature from 1669° F (no coating) to 1568° F (20% Y_2O_3 coating).

The transient heat transfer analysis for the various spall conditions was carried out using the cycle shown in Figure 2.4-23. The analysis assumed a boundary layer restart at each spallation. The thermal barrier conductivity data used is shown in Figure 2.4-24. This data is the most recent available and was also used for the steady state and the Stage 2 vane heat transfer analysis.

The effect of coating spallation on transient blade temperature distribution at 50% blade span is shown in Figures 2.4-25 through 2.4-32 for the two different coatings investigated. It can be seen that the 20% Y_2O_3 coating

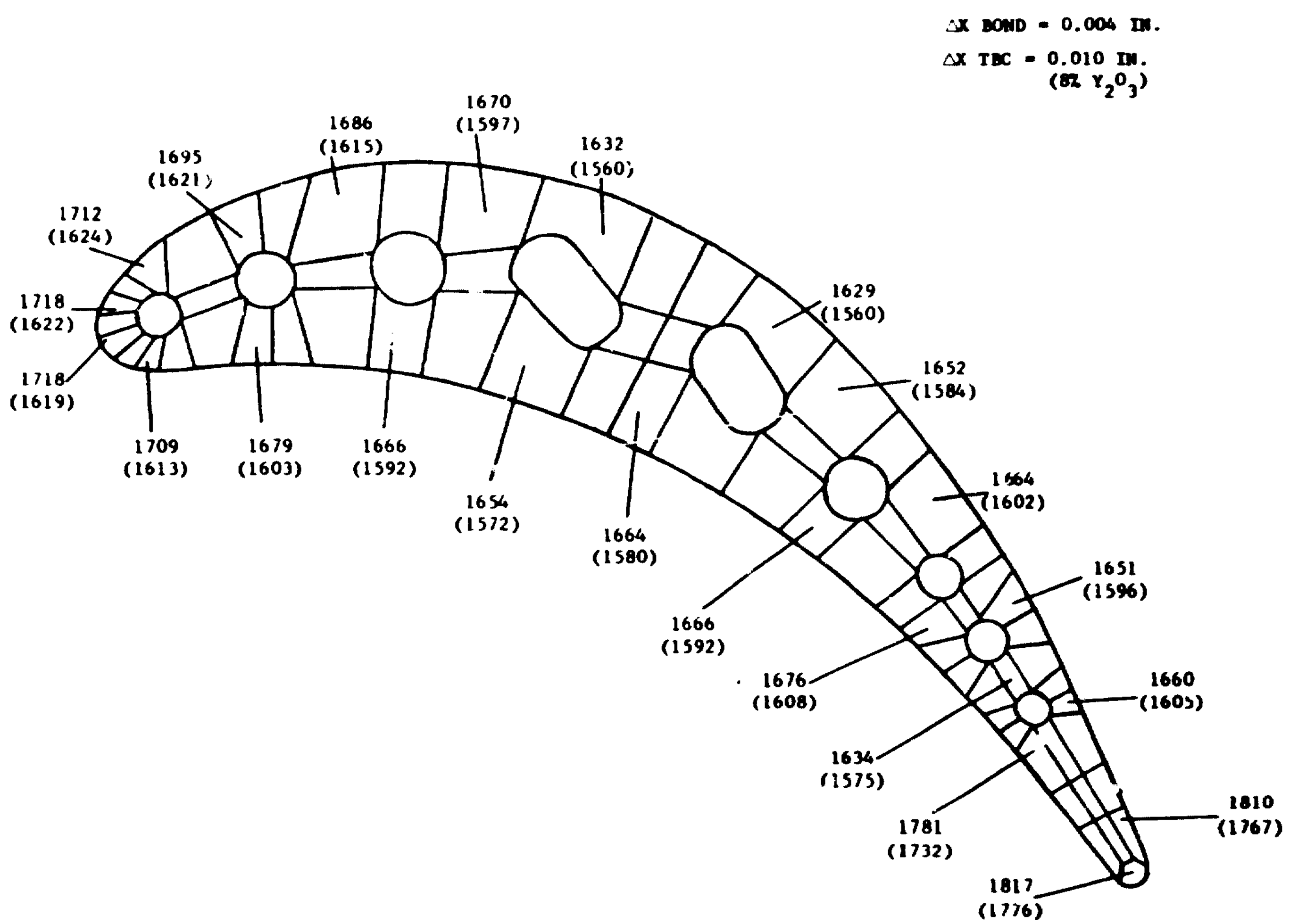


Figure 2.4-21. CF6-50 HPT Stage 2 Blade 20% Span, Steady State Temperature Distribution With and Without TB Coating.

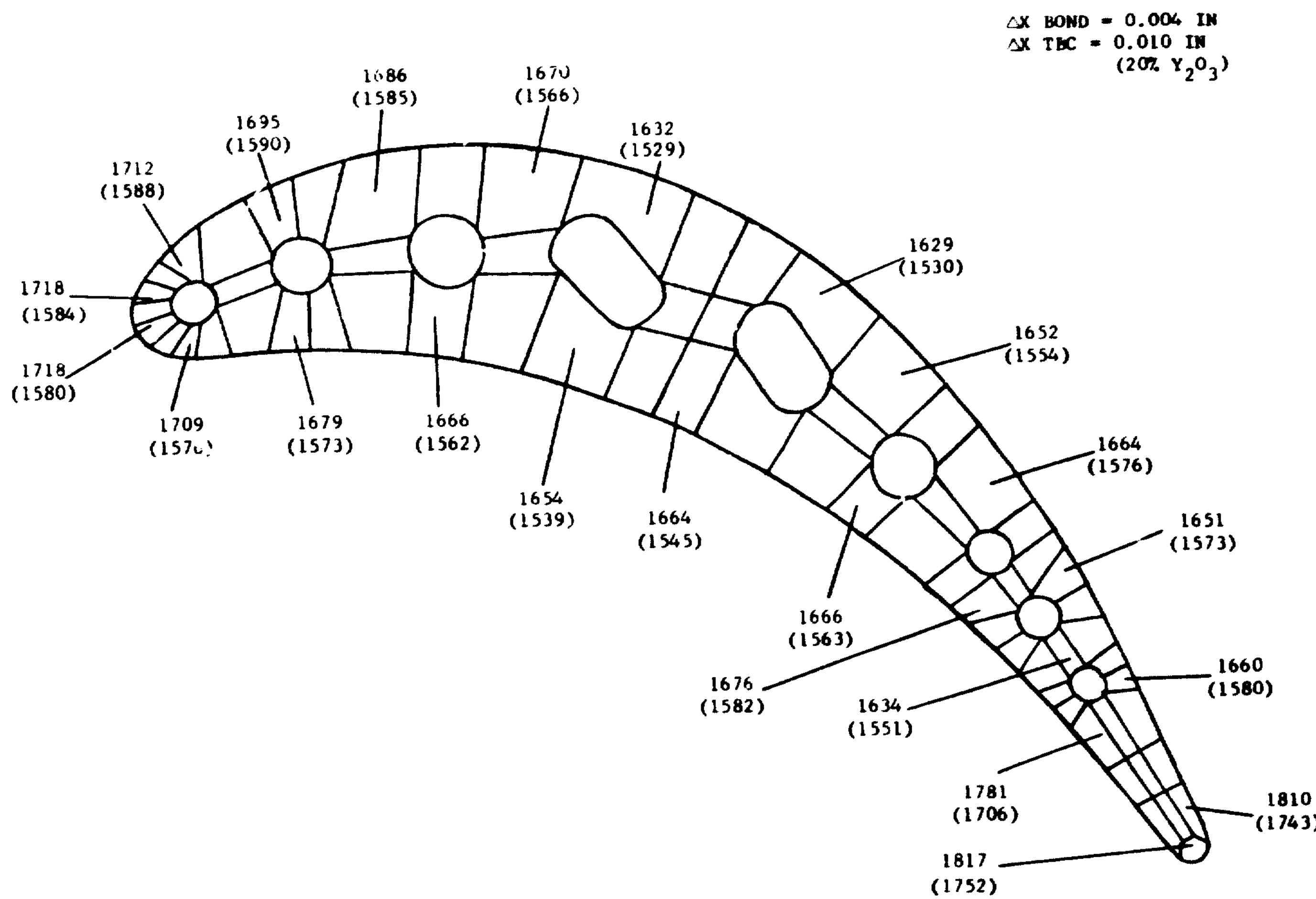


Figure 2.4-22. CF6-50 HPT Stage 2 Blade 20% Span, Steady State Temperature Distribution With and Without TB Coating.

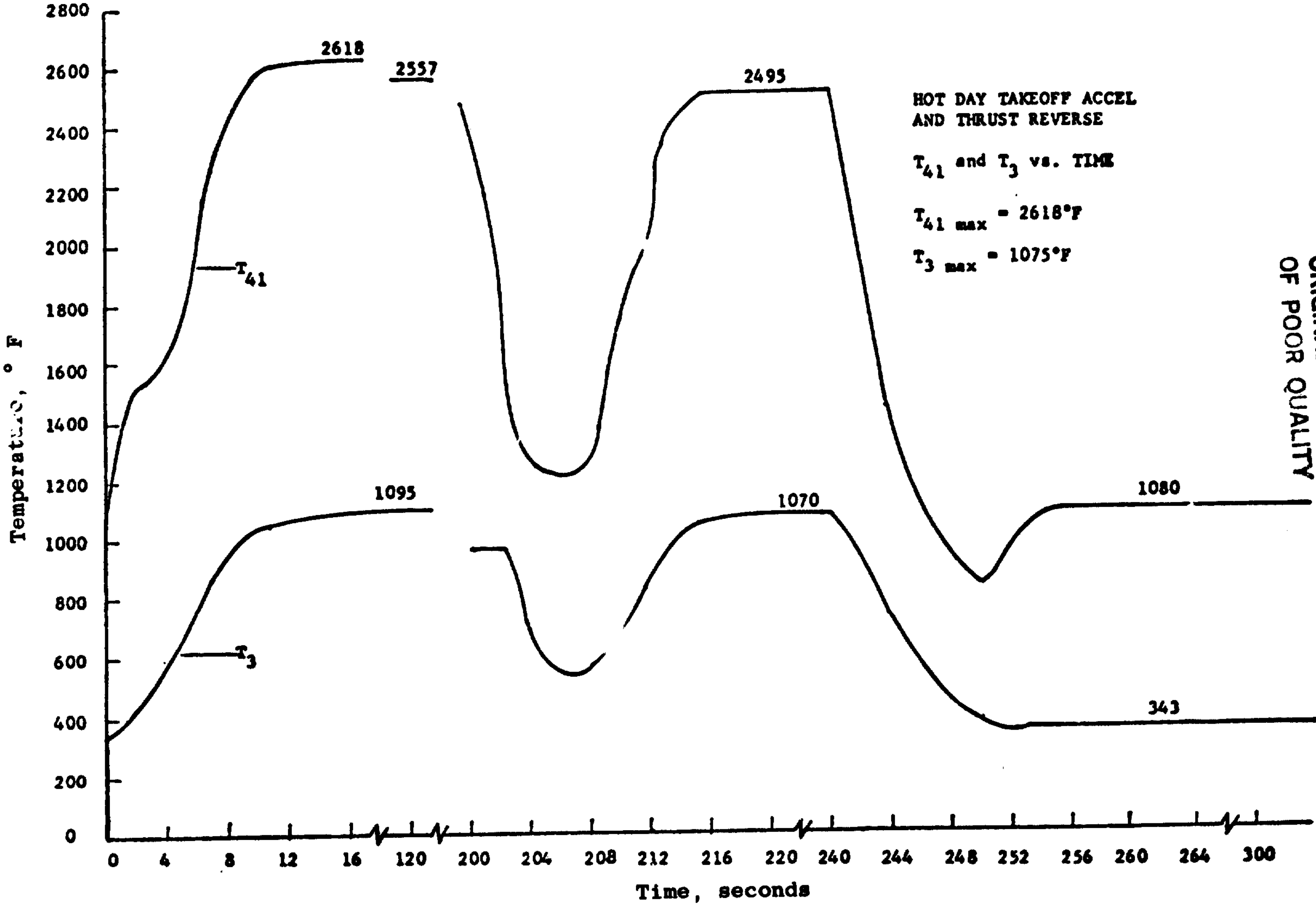


Figure 2.4-23. CF6-50C Transient.

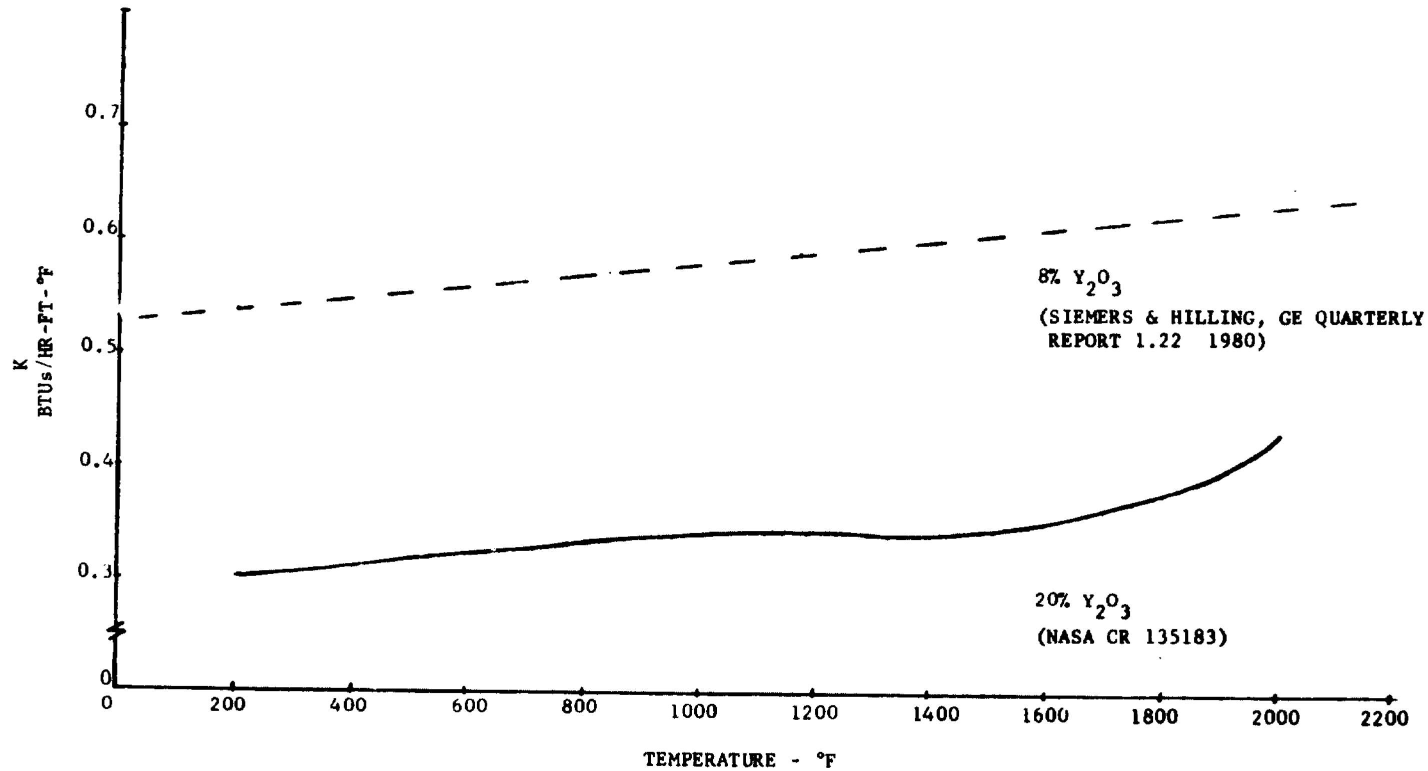
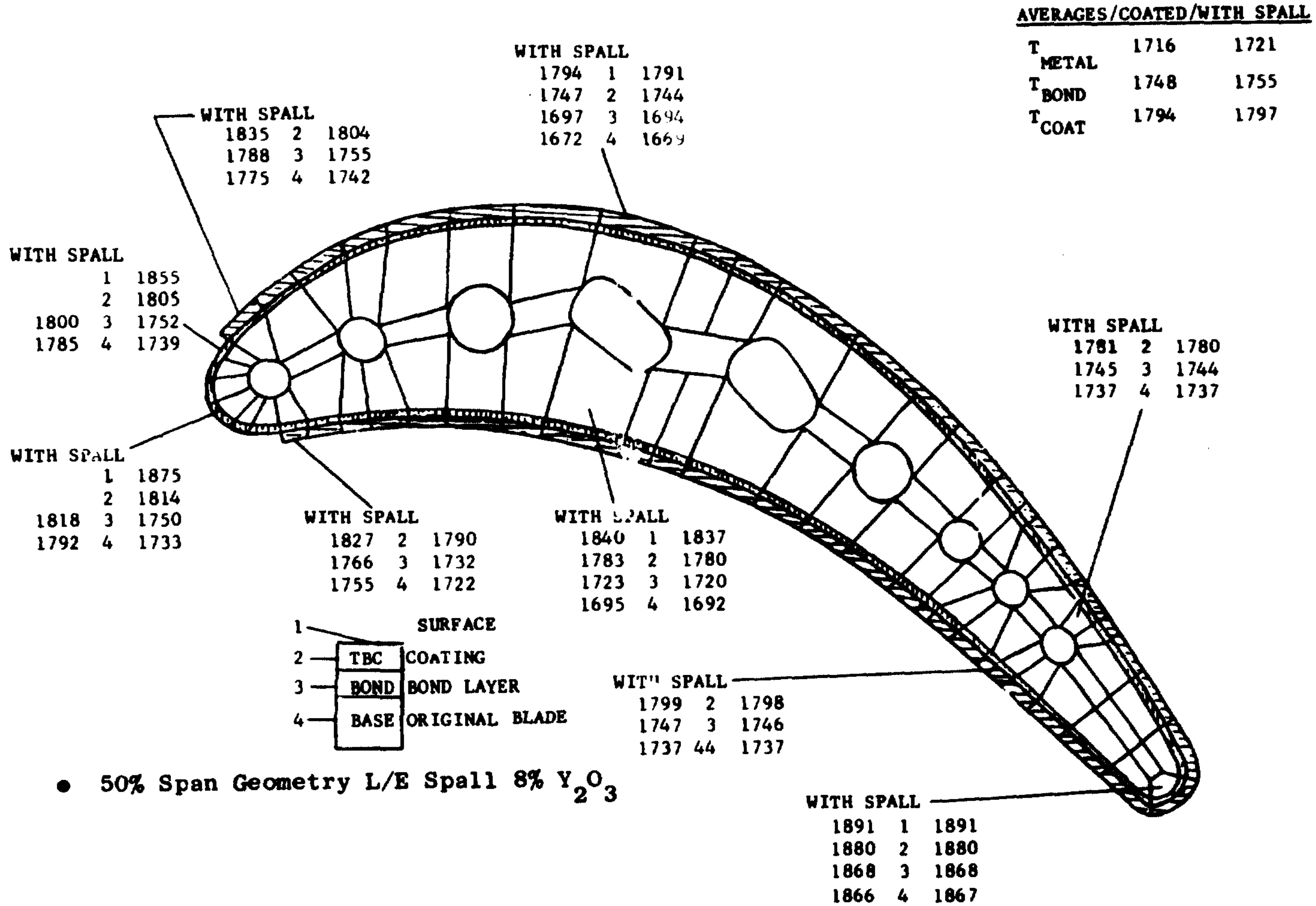


Figure 2.4-24. Thermal Conductivity of Zirconia (Y_2O_3 Stabilized).



- 50% Span Geometry L/E Spall 8% Y_2O_3

Figure 2.4-25. CF6-50 Stage 2 Blade, Showing the Effect of Spalling on Blade Temperature.

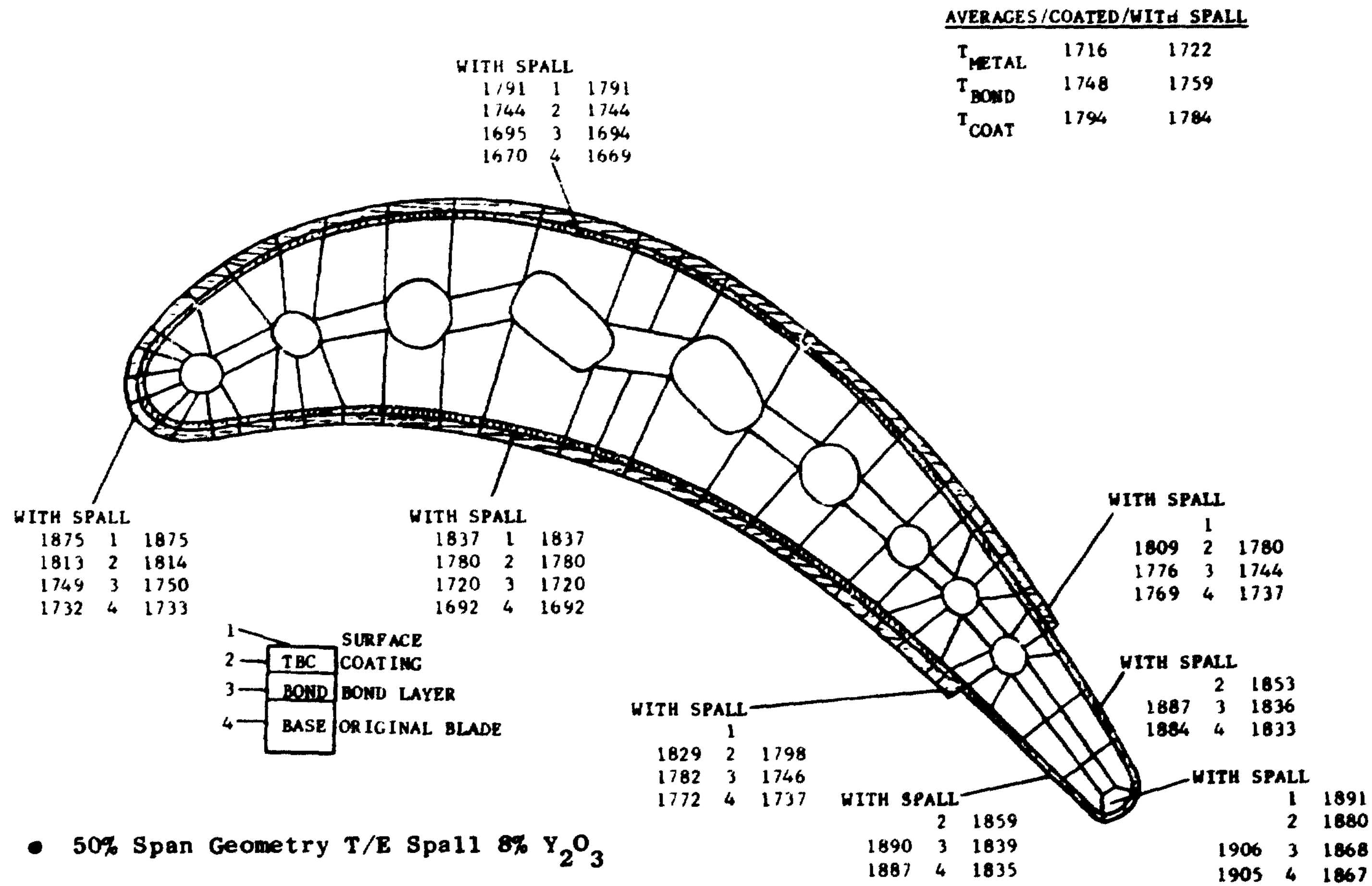


Figure 2.4-26. CF6-50 Stage 2 Blade, Showing the Effect of Spalling on Blade Temperature.

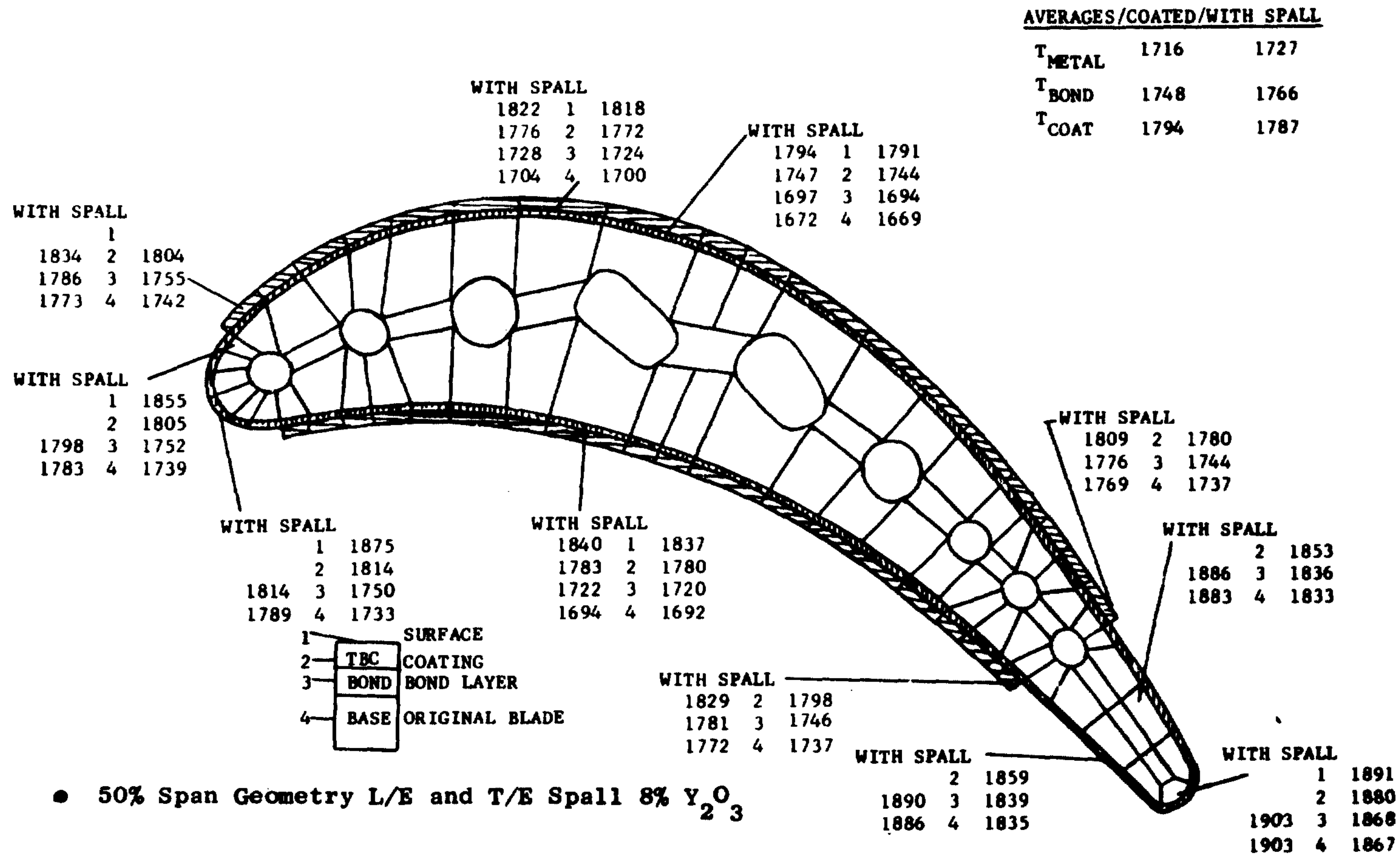
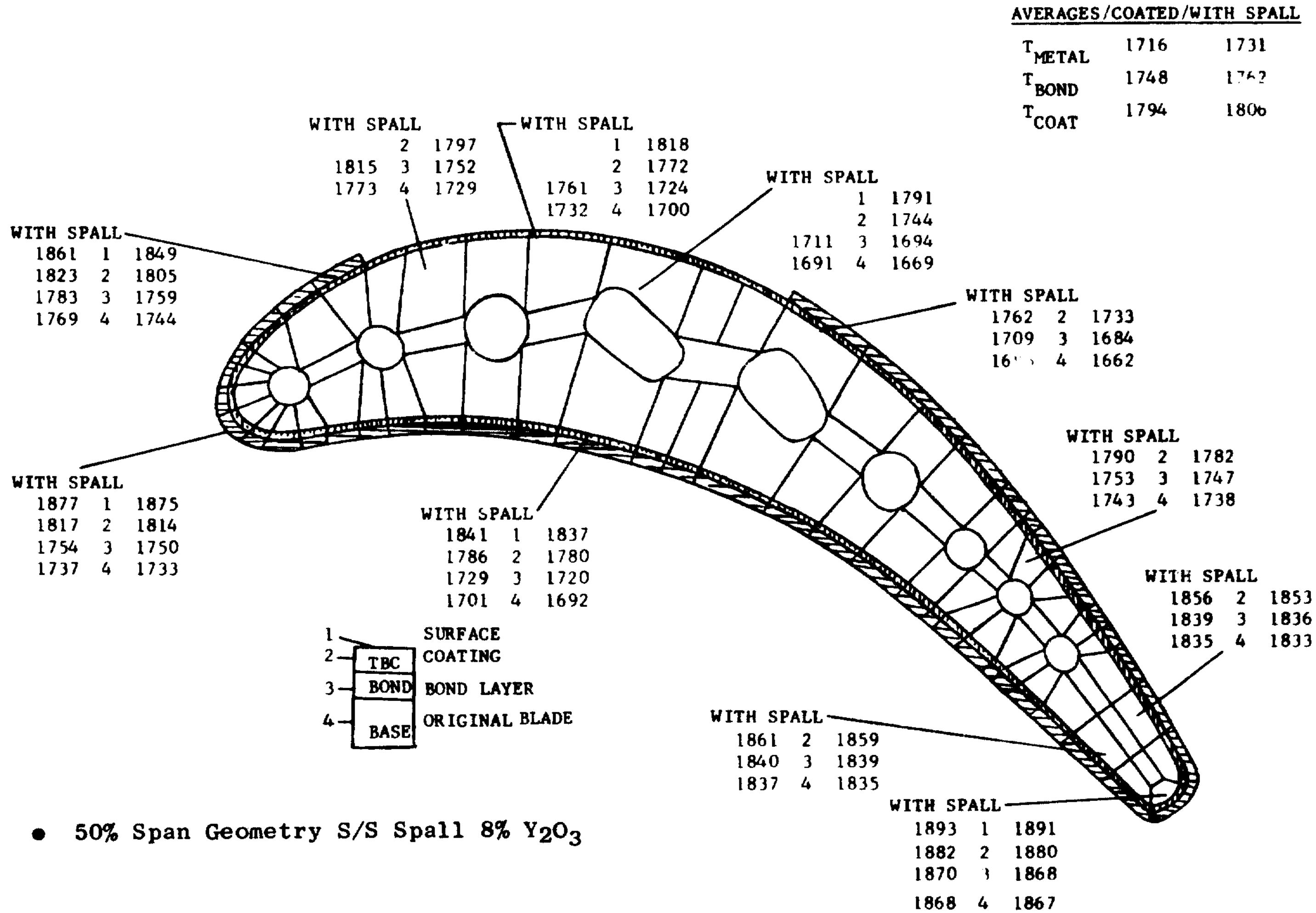


Figure 2.4-27. CF6-50 Stage 2 Blade, Showing the Effect of Spalling on Blade Temperature.



• 50% Span Geometry S/S Spall 8% Y_2O_3

Figure 2.4-28. CF6-50 Stage 2 Blade, Showing the Effect of Spalling on Blade Temperature.

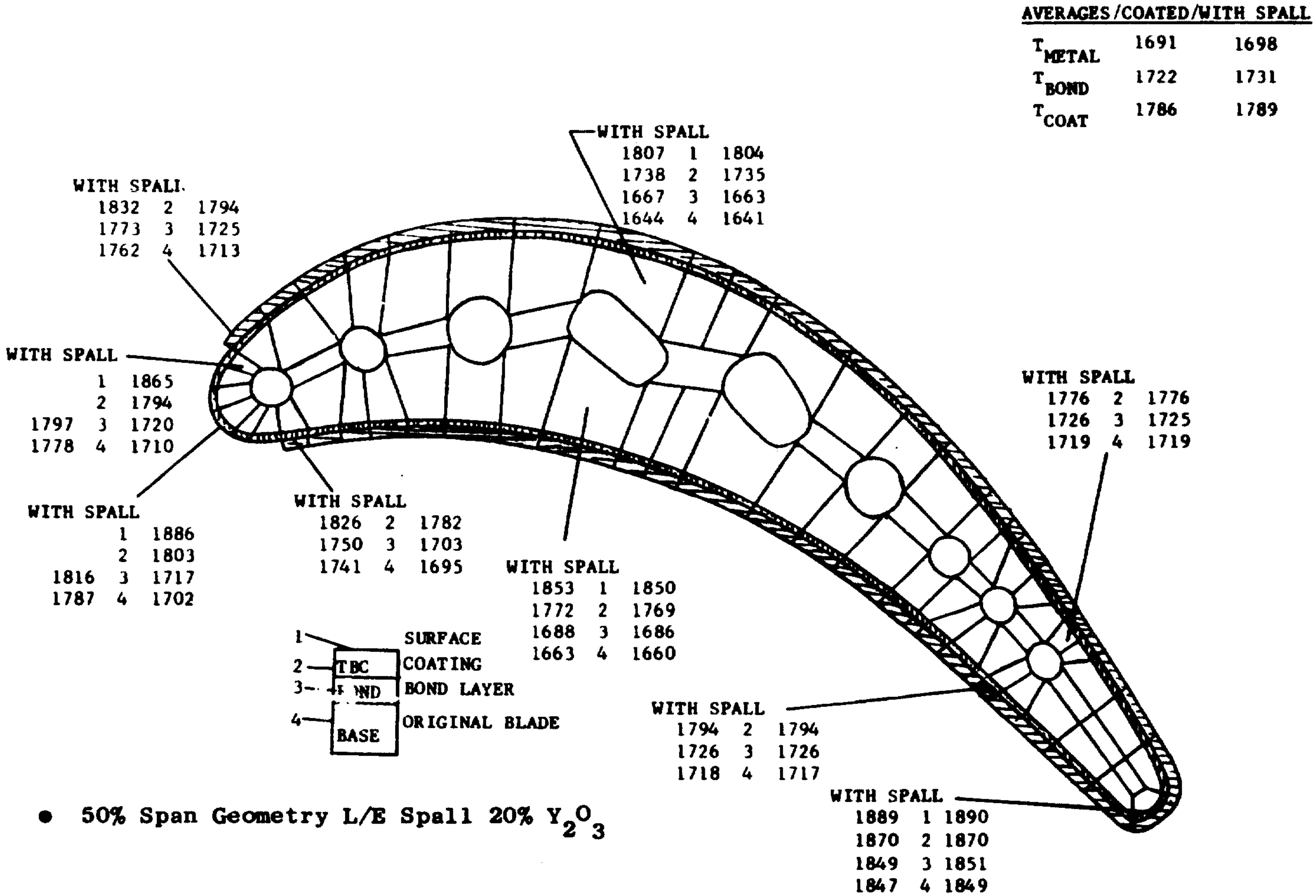


Figure 2.4-29. CF6-50 Stage 2 Blade, Showing the Effect of Spalling on Blade Temperature.

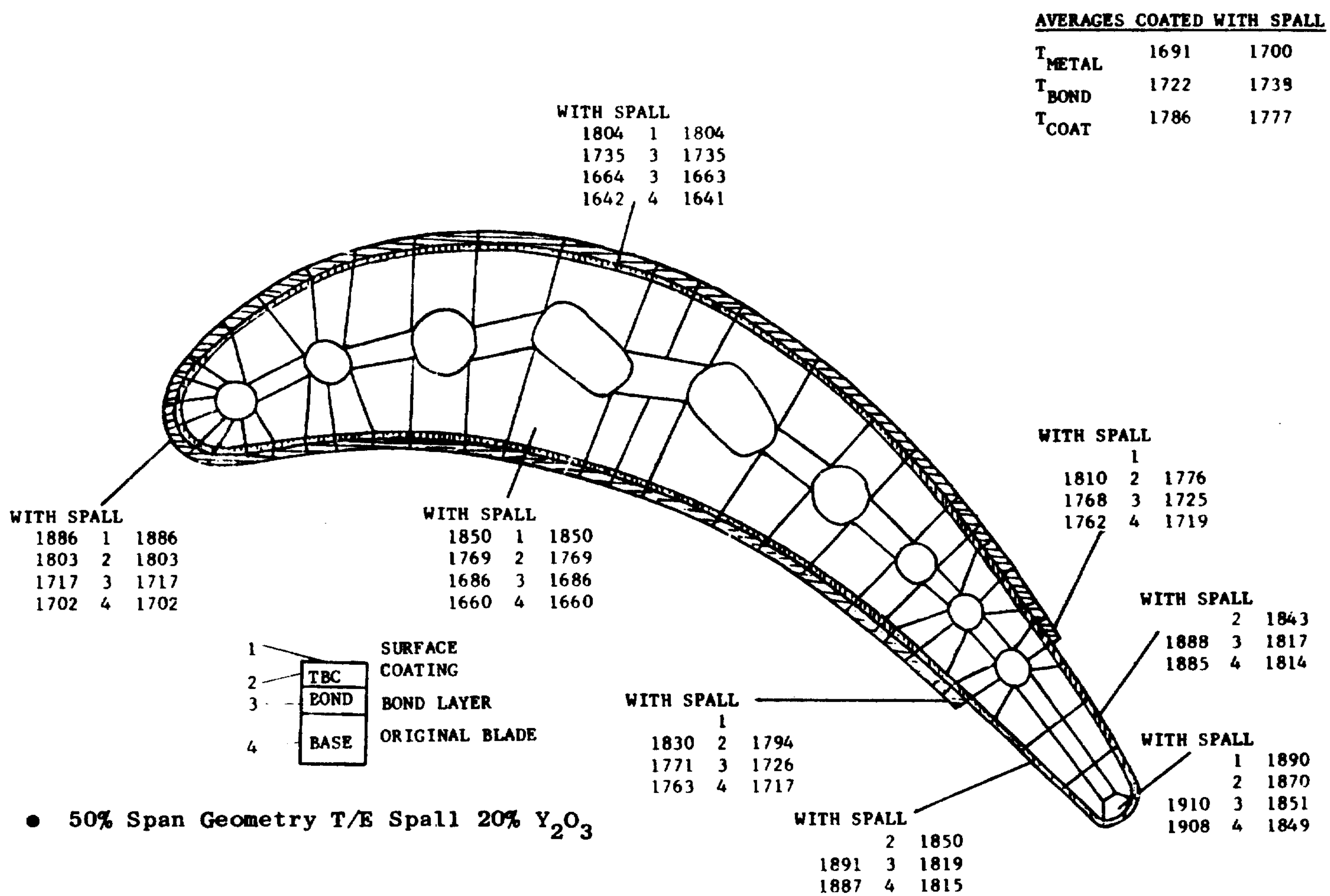
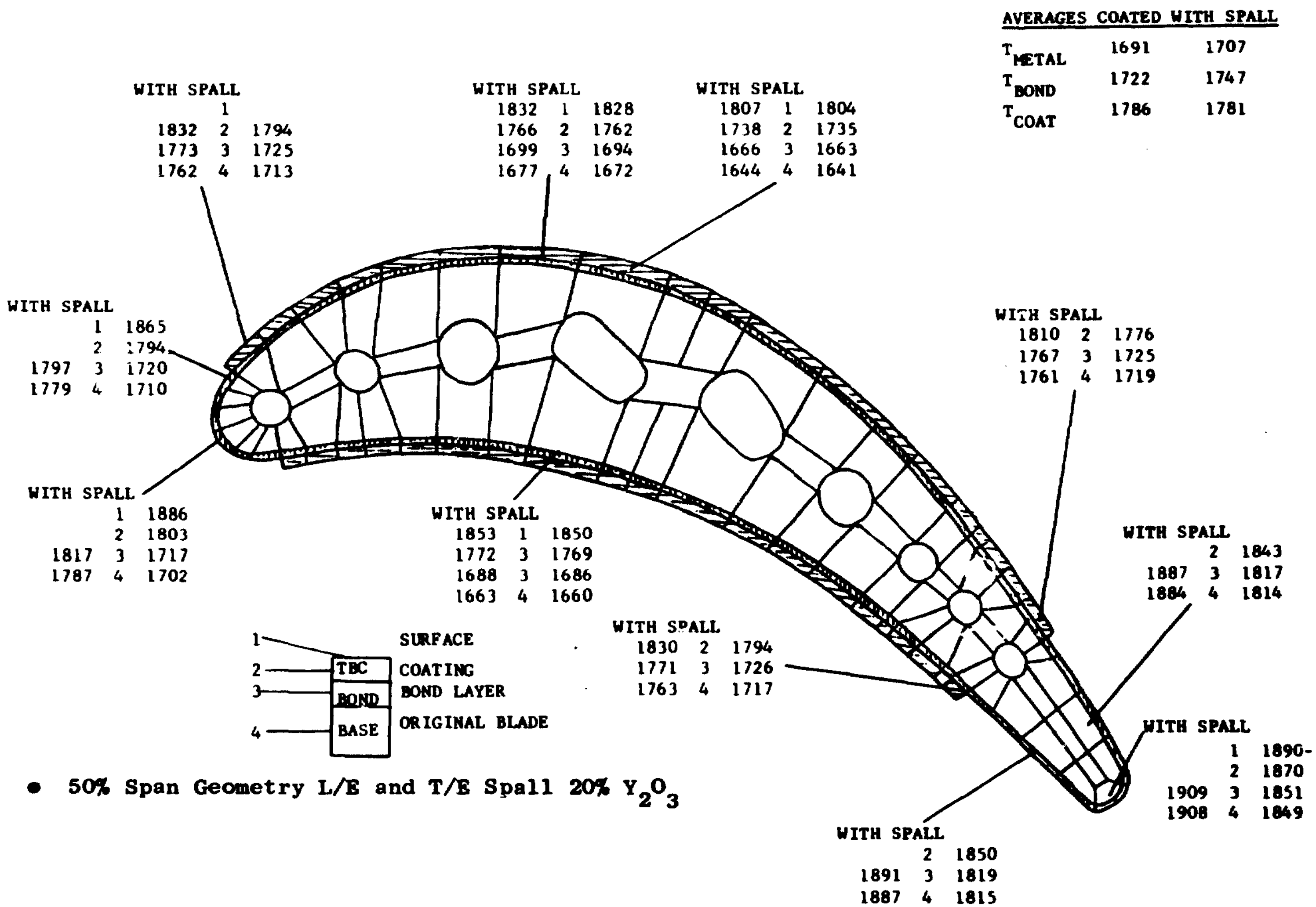
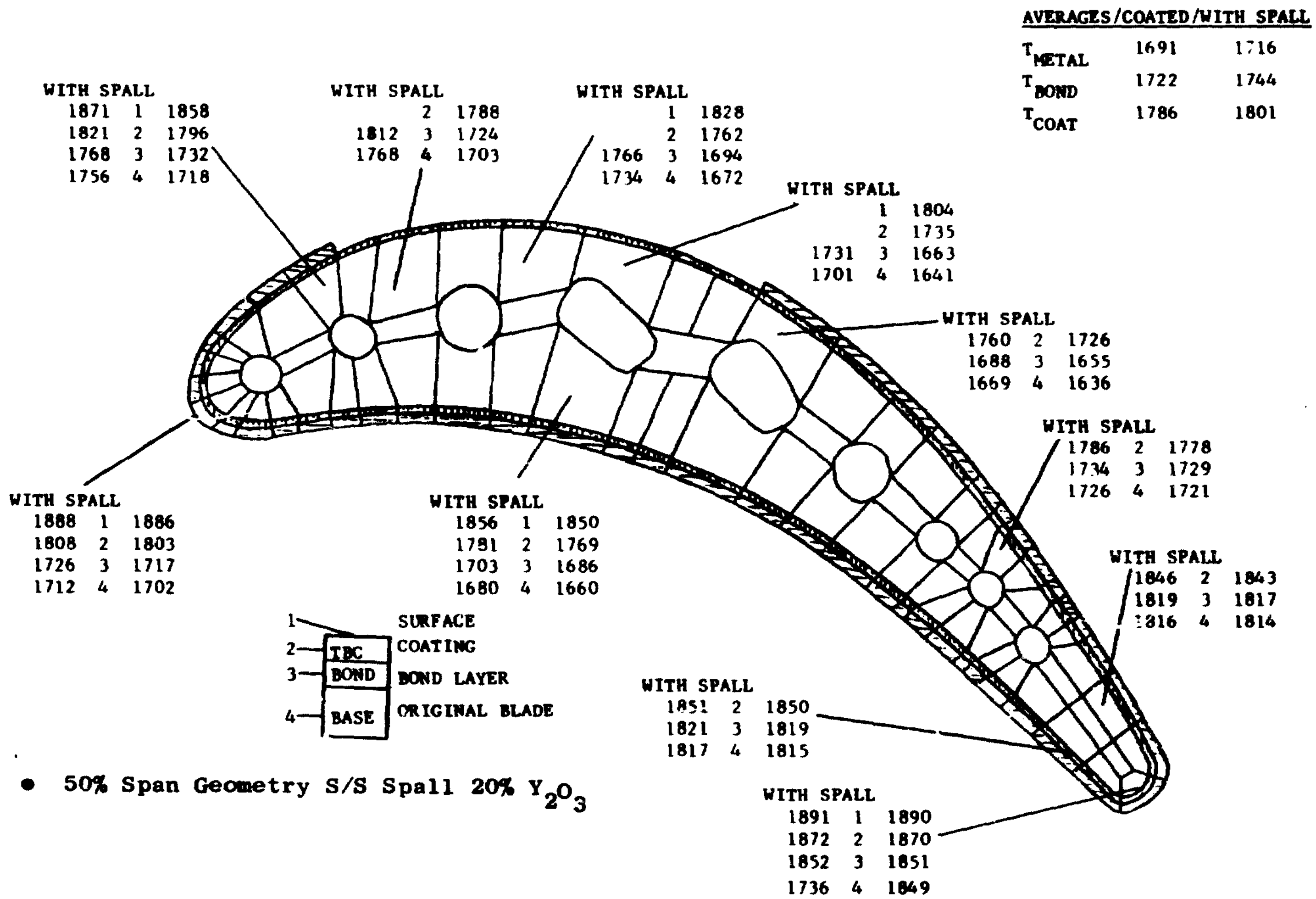


Figure 2.4-30. CF6-50 Stage 2 Blade, Showing the Effect of Spalling on Blade Temperature.



- 50% Span Geometry L/E and T/E Spall 20% Y_2O_3

Figure 2.4-31. CF6-50 Stage 2 Blade, Showing the Effect on Spalling on Blade Temperature.



• 50% Span Geometry S/S Spall 20% Y_2O_3

Figure 2.4-32. CF6-50 Stage 2 Blade, Showing the Effect of Spalling on Blade Temperature.

results in lower blade temperatures than the 8% Y_2O_3. However, this coating is more prone to spallation then the 8% combination and, therefore, both are to be evaluated on an engine test.

2.4.7.2.5 Thermal Barrier Analysis and Test Support

Technical Progress

Thermal barrier coated Stage 2 vanes and Stage 2 blades experienced FOD during the CF6-50 engine test. The test was scheduled for 1500 "C" cycles and was discontinued after 625 "C" cycles.

Engine teardown revealed the vanes and blades to have experienced considerable damage. Metallurgical examination of these parts can be found in WBS 2.4.7.2.1.

A CF6-50 engine is scheduled to initiate a 1500 "C" cycle test in April 1981. As part of the TBC program, the following components will be tested in this engine:

- Four Stage 1 nozzle segments
- Eight Stage 2 nozzle segments
- Ten Stage 2 blades.

Work Planned

Monitor and follow the CF6-50 engine test with TBC parts.

2.4.7.3.1 Alloy Mechanical Behavior

Technical Progress

The successful application of directional alloys to aircraft engine turbine blades and vanes requires careful definition and evaluation of the mechanical behavior of these materials in order to capitalize on their high strength potential and at the same time avoid failures in new modes or weaker off-axis directions. The present effort is aimed at developing and applying understanding of the behavior of the candidate directional materials DS

René 150 and MA754 so that they may be successfully applied in blades and vanes of the E^3.

As described in previous semiannual reports, two major areas of activity are being addressed: (1) thin wall creep rupture of DS René 150 and (2) low cycle and thermal fatigue of DS René 150 and MA754 under conditions relevant to blade and vane fatigue cracking.

Progress in these areas in the past 6 months is described below.

Thinwall Creep and Rupture of DS René 150

Work on this portion of the program has been completed and was reported in the last 6-month progress report.

Complex Fatigue Effects

Work on this portion of the program has progressed to the point of having final machined tubular LCF specimens of René 150 and MA754 (both longitudinal and transverse orientations) on hand. Coating of these specimens has been delayed until coating selection for the HPT blade has been made. This decision will not be finalized until June, although it is known that EA NiCrAlHf will not be used. Some specimens had already been coated with this coating and will have to be stripped.

Work Planned

- The detailed test plan for the tubular LCF specimens will be finalized with E^3 design, including number and configuration of simulated leading edge cooling holes to be introduced into specimens. These holes will be drilled, and then, when the coating choice is finalized (June), the specimens will be coated.
- LCF testing will begin at the end of the 6-month period.

2.4.8.1 Stage 1 Blade Manufacturing

Technical Progress

The casting of the Stage 1 blade, which was to be done by GE Albuquerque has been redirected to another casting source. The new vendor selection was based on a lower competitive cost and schedule.

The blade casting cores have been experiencing very low yields. These are mainly due to the cracking of the crossover holes located in the trailing edge region of the blade airfoil. The appearance of these cracks and cracks in the trailing edge pins consistently appeared during the core curing process. The limited quantity submitted for use in the mold casting also resulted in high casting rejects due to core breakage along the same crossover holes.

In order to improve the core and casting yields, a rework for the core die was defined. Rework involved increasing all trailing edge holes to a larger size opening with increase in the blend radii. Also, the first pin array diameter was slightly reduced. These proposed changes were reviewed with the casting and blade core vendors, and agreement was reached to proceed and rework the core die to the new hole geometry. The core rework was completed March 19, with approximately 5 weeks required for the first rework core deliveries.

A combination of core die rework, core die late delivery, low core yields and low casting yields resulted in the Blade 1 manufacturing being 16 weeks behind schedule. Engineering interface with the casting vendor will continue to determine whether casting delivery dates can be improved.

Blade manufacturing quoted costs are within planned funding. Total Blade 1 manufacturing costs are lower than funding available for this part.

Work Planned

- Continue close interface with casting vendor
- Initiate Phase II blade process procedure, which involves tip cap braze, after receiving blade castings
- Initiate blade machining.

2.4.8.2 Stage 2 Blade Manufacturing

Technical Progress

The casting of the Stage 2 blade is being done by the same casting vendor being used for the Stage 1 blade. Due to late core die delivery, the blade casting is approximately 8 weeks behind schedule.

Two blade casting molds were poured to determine casting soundness and blade dimensions from the first inspection. These dimensions are presently being evaluated. Preliminary airfoil thickness inspections indicate that all inspections are within the tolerance dimensions.

Work Planned

- Initiate tip cap braze
- Release castings to machining vendor
- PVD coat blade.

2.4.8.3 Stage 1 and 2 Disk Manufacturing

Technical Progress

The Stage 1 and 2 disk machining is ahead of schedule. Both disks are presently completing polishing operations. The disks shall be reworked for hole drilling in the aft disk arm (for both stages) for instrumenation routing. Once this operation is completed, the disks shall then be shot-peened.

Both disks in their present state are shown in Figures 2.4-33 and 2.4-34.

Total manufacturing cost for both disks is lower than estimated. This cost reduction is primarily in the tooling costs.

Work Planned

- Complete disk manufacturing
- Review flame spray procedures for weld tacking instrumentation on disk surface
- Assemble for balancing.

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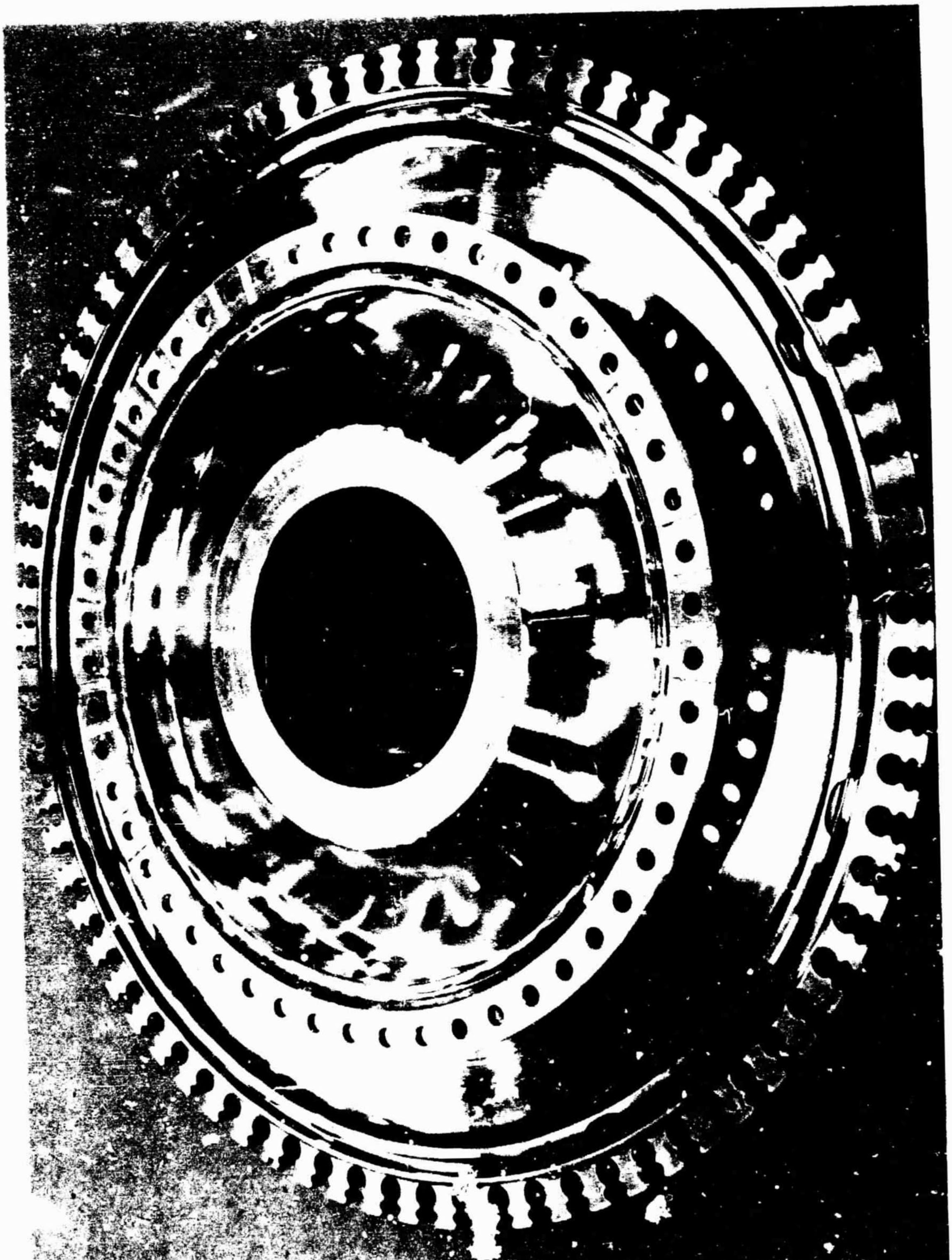

Figure 2.4-33. High Pressure Turbine Disk, Stage 1.

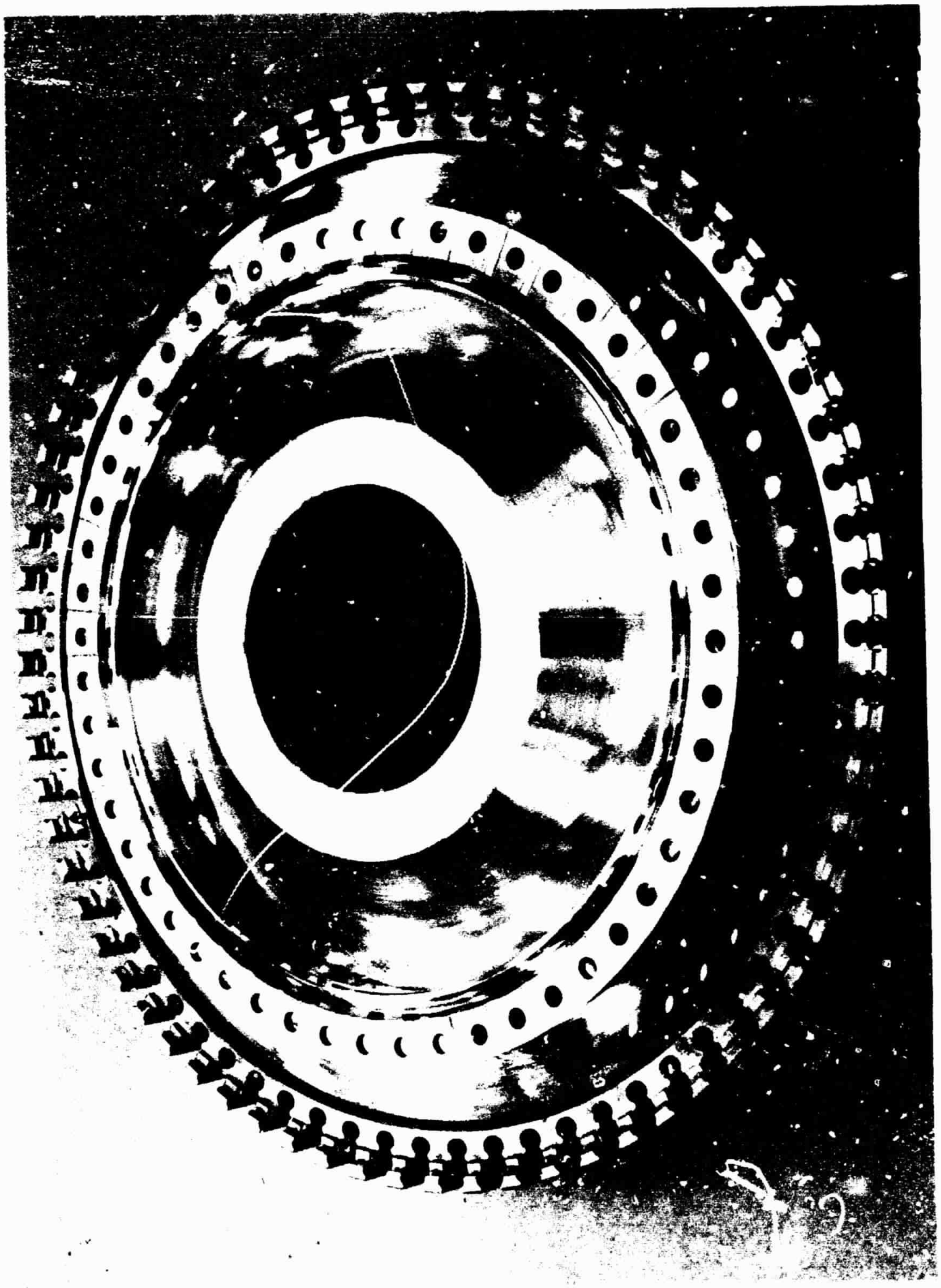

Figure 2.4-34. High Pressure Turbine Disk, Stage 2.

2.4.8.4 Rotating Shafts and Seals

Technical Progress

Stage 1 Aft Blade Retainer

The first attempt to produce the HIP AF115 material log was unsuccessful. Inspection of the log showed the can to contain lower density than acceptable. The cause of this was traced to the cans which had leaked during the autoclave process.

Due to the log being rejected, an alternate plan was defined that would still meet required delivery. This plan consisted of making the first retainer from available René 95 form and the second retainer from AF115 can available through the MPTL. NASA is reviewing the use of René 95 for the blade retainer for the core engine use.

The vendor is in the meantime processing a new log with an expected delivery of May 15, 1981. This log shall be used as a spare.

Delivery of the machined blade retainers is still within the required schedule for core engine buildup.

Stage 2 Aft Blade Retainer

Machining error in one of the two retainers caused a blade retainer to be scrapped. The backup available spare has been released to manufacturing. Both pieces are expected to be delivered by May 1, 1981 (ahead of schedule).

Interstage Disk

The interstage disk HIP AF115 cans were inspected and found to contain surface cracks. A dimensional rework drawing was supplied by engineering that allowed machining of the cracked surface, but still met the minimum envelope required for final machining. This was successful and the two disks were shipped to the machining vendor.

Presently, the original quoted machining time must be reduced to meet the required delivery schedule. Purchasing will now review the machining time and determine methods for improved delivery.

Impeller

One of the three impeller HIP cans showed ultrasonic indications that were above specification levels. This part has been rejected, while the remaining two were found to be acceptable. Because of the length of time required to procure the material and to process the part through the HIP operation, the replacement part is being cancelled. Therefore, the total order is for two cans.

The machining of one impeller is proceeding according to schedule and will be available for core assembly as planned.

Inducer Seal Disk

Three inducer seal disks have been HIP'd from AF115 material and all three were accepted. One disk is presently planned for the program. Machining is continuing and its delivery is within the requirement for core buildup.

HP Shaft

The HP shaft manufacturing has been completed. The shaft is shown in Figure 2.4-35. The material for this component is Inco 718.

Forward Outer Liner

The forward outer liner, made from René 95 material, was successfully HIP'd. This part is being machined and its delivery schedule is within the required assembly date.

Aft Shaft/Seal Disk

Machining of the aft shaft/seal disk is being completed. Material is Inco 718. Part delivery is expected to be the same as that planned for core engine buildup.

Aft Seal Disk Damper

The aft seal disk damper has been completed and is shown in Figure 2.4-36; the material is Inco 718.

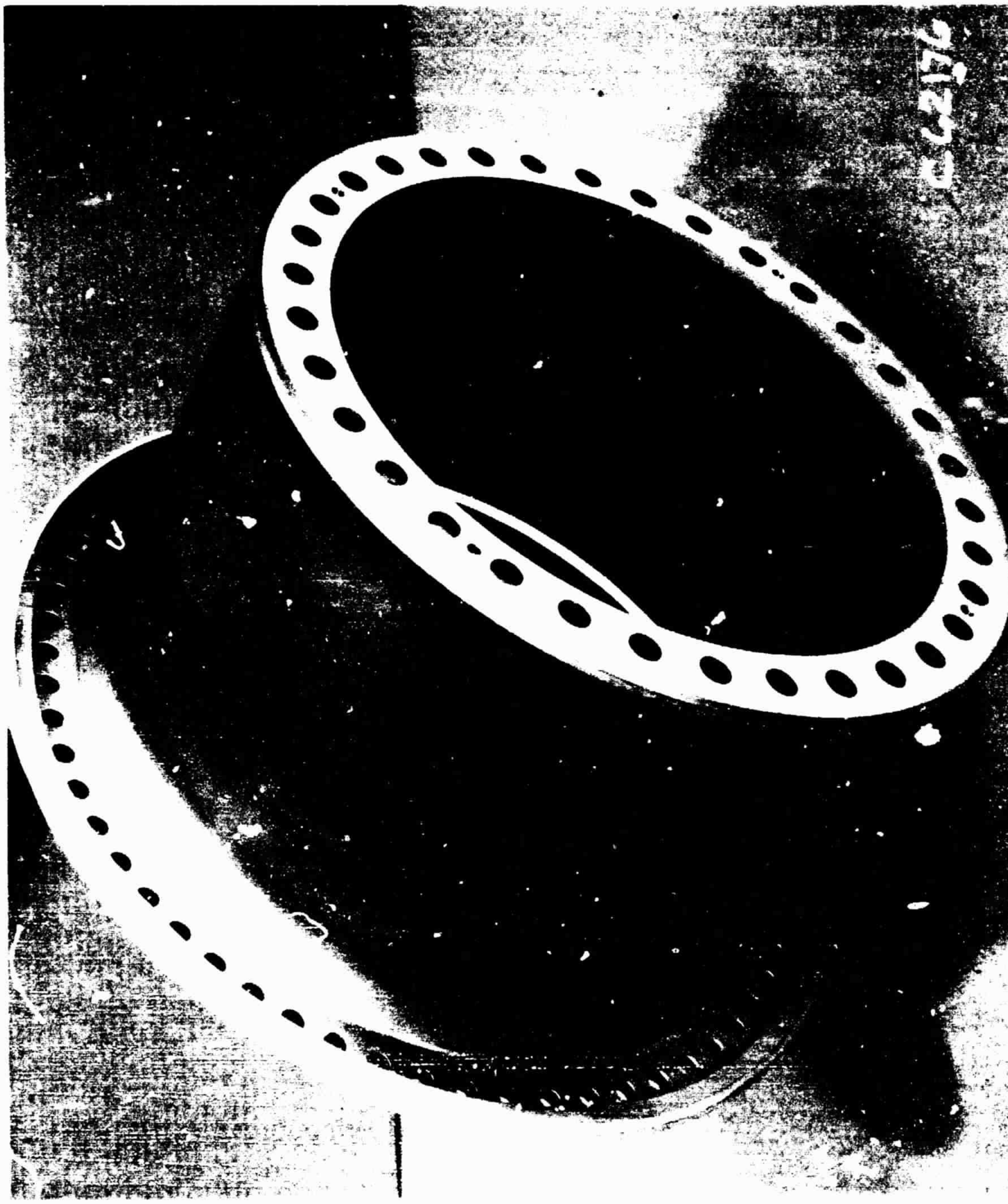

Figure 2.4-35. High Pressure Turbine Forward Shaft.

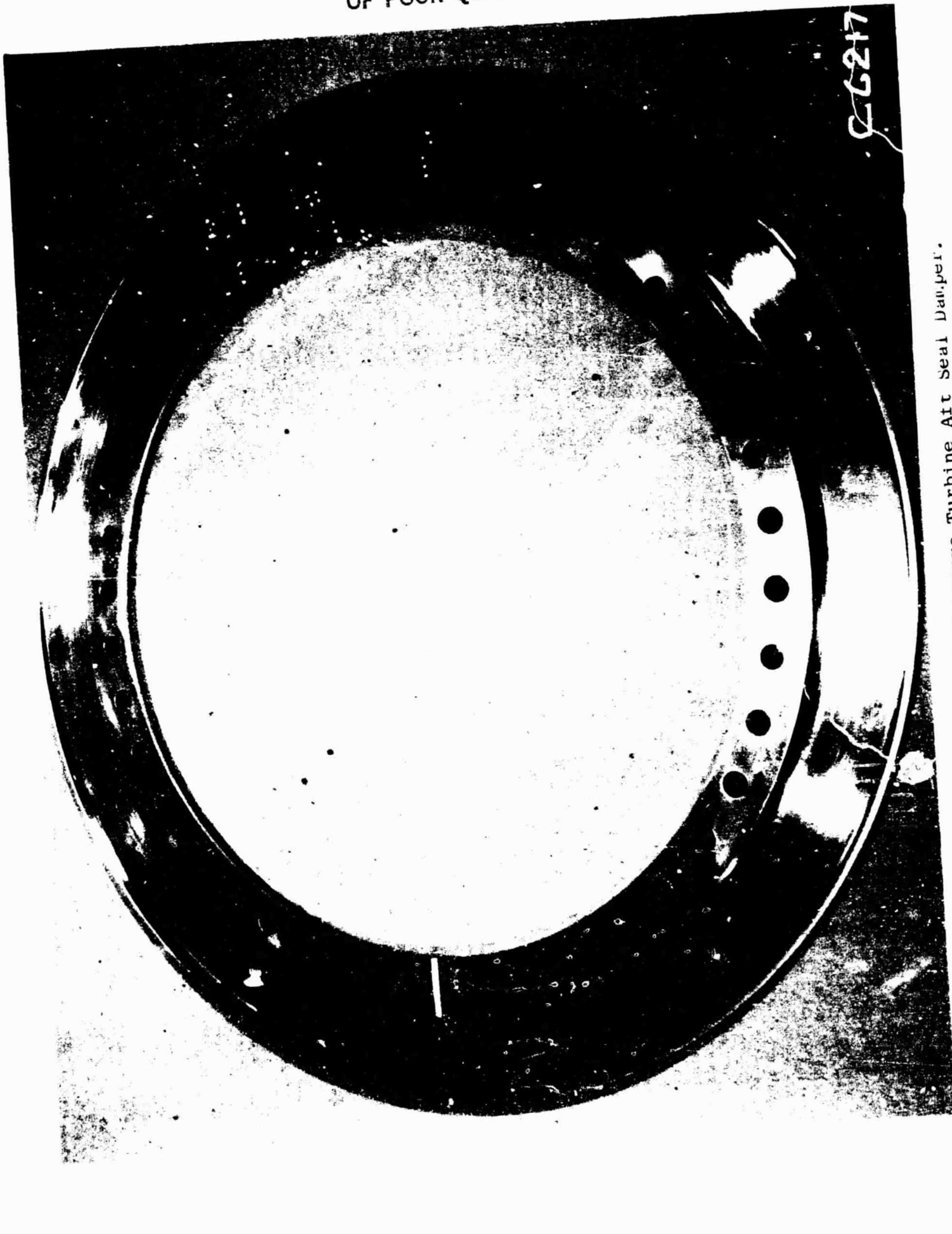

Figure 2.4-36. High Pressure Turbine Aft Seal Damper.

Inner Tube

The inner tube machining is continuing as planned. Two inner tubes are being machined. The first contains holes for instrumentation leadouts to the slipring. After core engine test, this part will be replaced with an identical piece, except that it will not contain any holes. The second piece will be used for the ICLS test.

Work Planned

- Purchasing will determine the methods of improved time delivery for the interstage disk
- Initiate and review flame spray locations for various components; flame spray is used to weld tack instrumentation leads
- Monitor assembly and balancing for turbine rotor.

2.4.8.5.1 Stage 1 Nozzle Diaphragm Fabrication

Technical Progress

Stage 1 Outer Band

The Stage 1 outer band wax die rework was made to improve expected casting yields. These minor reworks were in areas of radii and local widths for improved casting fills.

The first 30 outer band castings were delivered and the first part dimensional inspections were approved. Machining of film holes for cooling air passages has now been completed. Parts are in process of being Codep-coated.

Stage 1 Inner Band

Inspection of the band castings indicated the flowpath contour to be excessively out of print, and therefore rejected. Tool vendor rework of the flowpath inse within the core die was made to correct this problem.

Thirty inner band castings were received for further processing. First casting dimensional inspection has been accepted. The machining vendor is in process of adding film holes to this part.

Airfoil

The vane airfoil is machined from MA754 material. Airfoil machining contour has been completed. The two airfoil cavities are presently being formed by EDM process.

Work Planned

- Review with Manufacturing the possibility of reducing the time required to EDM the inner band hole machining
- Interface and monitor progress of nozzle fabrication.

2.4.8.5.2 Stage 2 Nozzle Diaphragm Fabrication

Technical Progress

Outer Band Casting

The outer band casting, made from René 80 material, has been delivered within schedule. First casting inspection has been reviewed and accepted. The castings will now be processed through the EDM of the airfoil contour in the band flowpath surface.

Inner Band Casting

The inner band casting has been processed and is presently being inspected. Dimensional inspection and casting soundness are being completed and will be reviewed by Engineering.

Vane Airfoil

Core die late delivery and cracking of local areas of the processed casting cores have caused the airfoil casting to be 14 weeks behind schedule for the nozzle fabrication. The airfoil casting, which is then brazed to the inner and outer band casting, is the limiting item for the brazed nozzle fabrication.

Core cracks have primarily been observed in the inner flowpath portion of the airfoil where the core printout intersects the airfoil section and also in the trailing edge turbulence promoters. A larger blend radius for both cracks has now been used for both cracked locations to improve core yield.

Work Planned

- Determine improvements necessary to improve core yields
- Initiate airfoil casting process
- Initiate nozzle fabrication assembly.

2.4.8.6 Support Structure and Inducer Fabrication

Technical Progress

Inducer and Balance Piston Honeycomb Seal

Rough machining for the René 41 static structure and the Inco 903A material for the seal were completed and EB-welded. Visual zyglo and X-ray inspection of the weld are being processed. Delivery of this component for instrumentation and core assembly is as scheduled.

Stage 1 Inner Seal

Rough machining of the René 41 structure and Inco 903A seal, with the subsequent EB-welding of these two materials, have been completed. Visual zyglo and X-ray inspection indicate the weld joint to be acceptable. Final machining is in process.

Stage 1 Nozzle Support

The inner nozzle support was rough machined from forgings and then EB-welded at the end mating surfaces. Final machining has been completed. The remaining manufacturing involves hold flange drilling and mounting the basket captive nuts on the flange which is used for supporting the inner seal.

Outer Casings and Outer Support Structures

All hardware for the casings and outer support structures is being machined. Milling is being completed for all pads in the casing for instrumentation probes and for the active clearance control measurement probe. All hardware delivery dates are within the program schedule requirements.

The active clearance control impingement manifolds are on order with the delivery date scheduled for October 1981.

Ceramic Shrouds

Wax patterns for the shroud have been approved by Engineering. An initial delivery of 40 castings is expected by April 30 for machining and ceramic coating application. Of these initial 40 casings, nine are planned for machining to the soid shroud configuration.

Stage 2 Solid Shrouds

The first initial 40 shroud castings have been poured and inspected.

Work Planned

- Continue manufacturing interface
- Review and approve instrumentation drawings
- Interface with evaluation during component subassemblies.

2.5 LOW PRESSURE TURBINE

Overall Objectives

The objective of the low pressure turbine development effort is to provide a high efficiency LP turbine having material and configuration features that provide maximum opportunity for mechanical success.

Performance improvements will be aimed at achieving the highest turbine efficiency compatible with moderate-to-high turbine aerodynamic loading. The FPS low pressure turbine efficiency goal is 0.917 at Mach 0.8, altitude 35,000 feet, standard day, maximum climb power setting. Also considered are off-design operating points in order to ensure a viable system throughout the operating regime of the engine.

Mechanical integrity is a major goal of the LP turbine mechanical design. The aeromechanical goals are that the blades and vanes should have no instabilities within the operating range of the LP turbine.

Development Approach

The overall development plan for the LP turbine provides a systematic approach to its design. This involves (1) early identification of critical areas and (2) component design that can accommodate the critical requirements. The major LP turbine development areas are (1) the aerodynamic design and air turbine evaluation; (2) the system and mechanical design, which is carried out concurrently with the aero work; (3) hardware fabrication; (4) component bench tests; and (5) instrumentation, assembly, and engine test.

Preliminary LP turbine aero design of the initial flowpath, as well as blade and vane airfoils, comprised the major initial effort which was reviewed at the aero IDR in July 1978. Block I air turbine blading was released at that time; flowpath definition was made the following month. Mechanical and heat transfer design paralleled this aero effort, and an overall LP turbine IDR was held in November 1978 followed by the LP turbine PDR in May 1979. Block I air turbine testing of Stages 1 and 2 was completed in August 1979.

Aero modifications were factored into Block II blading and sequentially released over the period of January through March 1980. An IDR of this work was held in February 1980. Table 2.5-I lists pertinent aerodynamic parameters for the LP turbine as taken from FPS cycle data. It should be noted, however, that the actual Block II aero design was executed to a flow function 4% open relative to the FPS maximum climb condition (Table 1.2-IV). Procurement of Block II hardware is underway, and air turbine tests of all five LP turbine stages are planned beginning with a two stage configuration test in May 1981.

Detailed mechanical design was initiated in June 1979 after the PDR milestone. The next major milestone was the ICLS go-ahead decision given in November 1979, ahead of the scheduled January 1980 date. Continuing detail mechanical design, incorporating the Block II airfoil configurations, was summarized for an August 1980 IDR, and a DDR was held in December 1980. Release of fabrication orders began in September 1980 with cast parts, although advanced forging releases have been made as early as June 1979. The earliest manufactured blades and vanes will be utilized for bench testing. Bench testing will allow identification of responsive locations on the airfoils, where instrumentation will be applied, starting the third quarter of 1981. All components will then be assembled into the ICLS demonstrator; testing is to begin the second quarter of 1982.

2.5.1 LPT Aerodynamic Design

Technical Progress

The LP turbine detailed design review was held at NASA-Lewis on December 11, 1980. Block I rig test results, Block II detailed aero design and performance prediction, and planned Block II rig test schedules were presented as part of the DDR. Formal notification was received from NASA in January 1981 that release of funds for the Block II rig tests had been approved.

The Block II Test and Instrumentation Plan was issued during this reporting period.

All hardware for the two-stage build has been received and instrumentation and assembly are in progress. All hardware for the five-stage build will be received by the end of April 1981.

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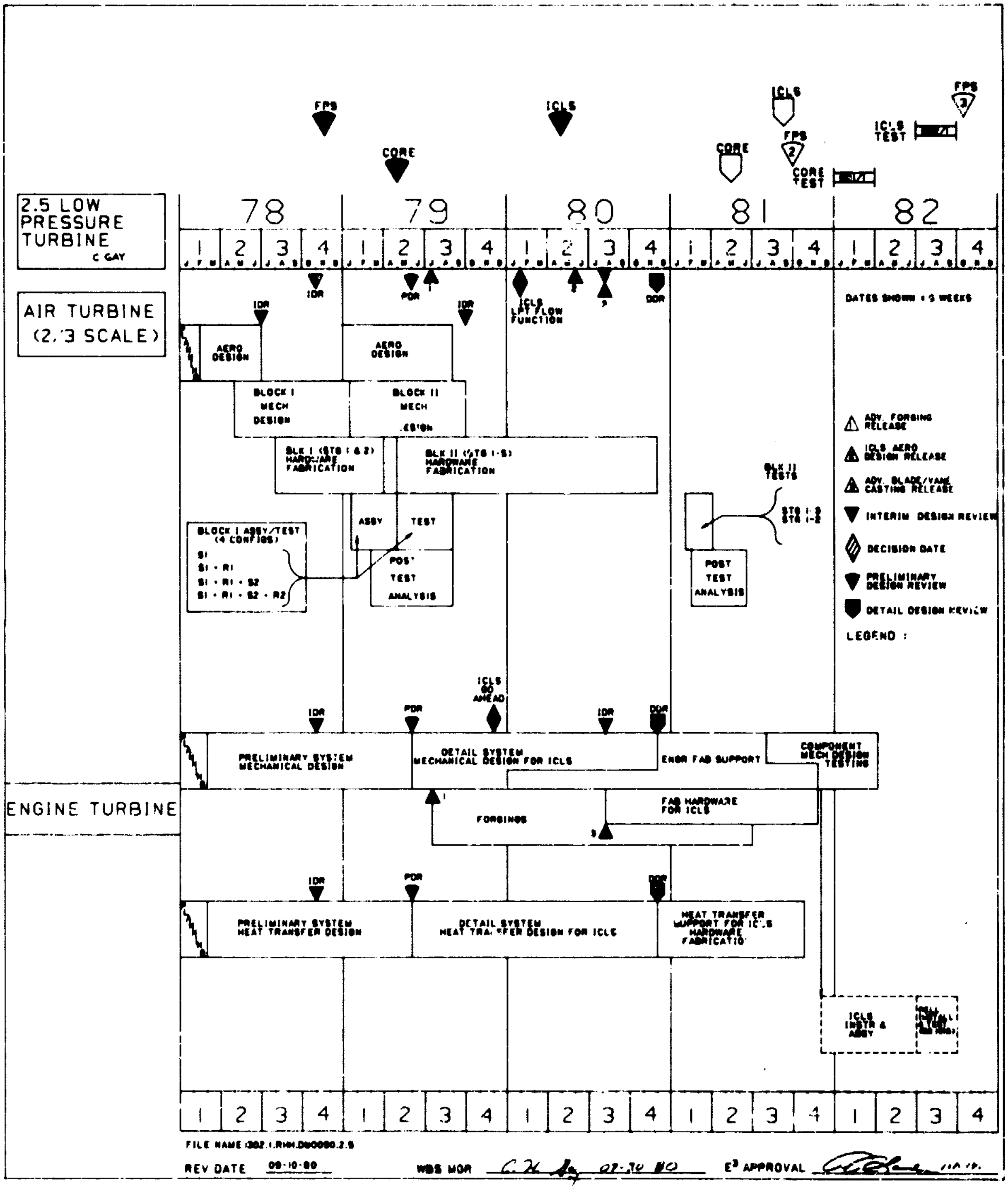

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Table 2.5-I. LP Turbine Aerodynamic Design Parameters (fps Cycle).

LPT	RPM	T_{49} °F	P_{49} psia	$\frac{\Delta h}{T}$ Btu/lb-° R	$\frac{N}{\sqrt{T}}$ rpm/° R	ψ	$\frac{W\sqrt{T}}{P}$ $\frac{\text{lbm-° R}}{\text{sec-psia}}$	T_5 °F
SLS T/O +63° F day	3293	1577	72	0.0695	73.0	1.37	80.7	1071
Max Climb +18° F day	3538	1489	38	0.0780	80.1	1.27	80.4	940
Max Cruise +18° F day	3437	1438	36	0.0770	78.9	1.30	80.5	907

C-4

The following is the latest estimate of the Block II test schedule:

• Instrument and Assemble Two-Stage	In Progress
• Install and Test Two-Stage	4-27-81
• Instrument and Assemble Five-Stage	4-21-81
• Install and Test Five-Stage	6-22-81
• Complete Testing	7-17-81

Work Planned

- All Block II Testing will be accomplished during the next reporting period
- The LPT Aerodynamic portion of the Detailed Design Report will be prepared.

2.5.2 Low Pressure Turbine Heat Transfer Design

During the past 6 months, the detailed heat transfer design was completed. A detailed design review was held with the Engineering Product Review Board. A few minor changes were suggested by the Board and have been factored into the design. A successful review of the detail design was also conducted with NASA. The detail design report is currently being written and will be published during the next reporting period.

During the past 6 months, several studies have been conducted in an effort to complete the detailed design.

HP/LP Wheel Space Purge Design

The LP gas stream and cavity pressures at the 86° F ambient temperature maximum takeoff cycle point are presented in Figure 2.5-1. These cavity pressures were used to define the seal blockage flow rates and various sink pressures for the cooling air. The prime source of LP turbine cooling air is the fifth stage compressor bleed which is delivered to the turbine through six pipes, equally spaced around the Stage 1 LP nozzle cooling supply manifold. This manifold, which is integral with the casing and the outer transition duct hanger, allows the cooling air to distribute itself uniformly around

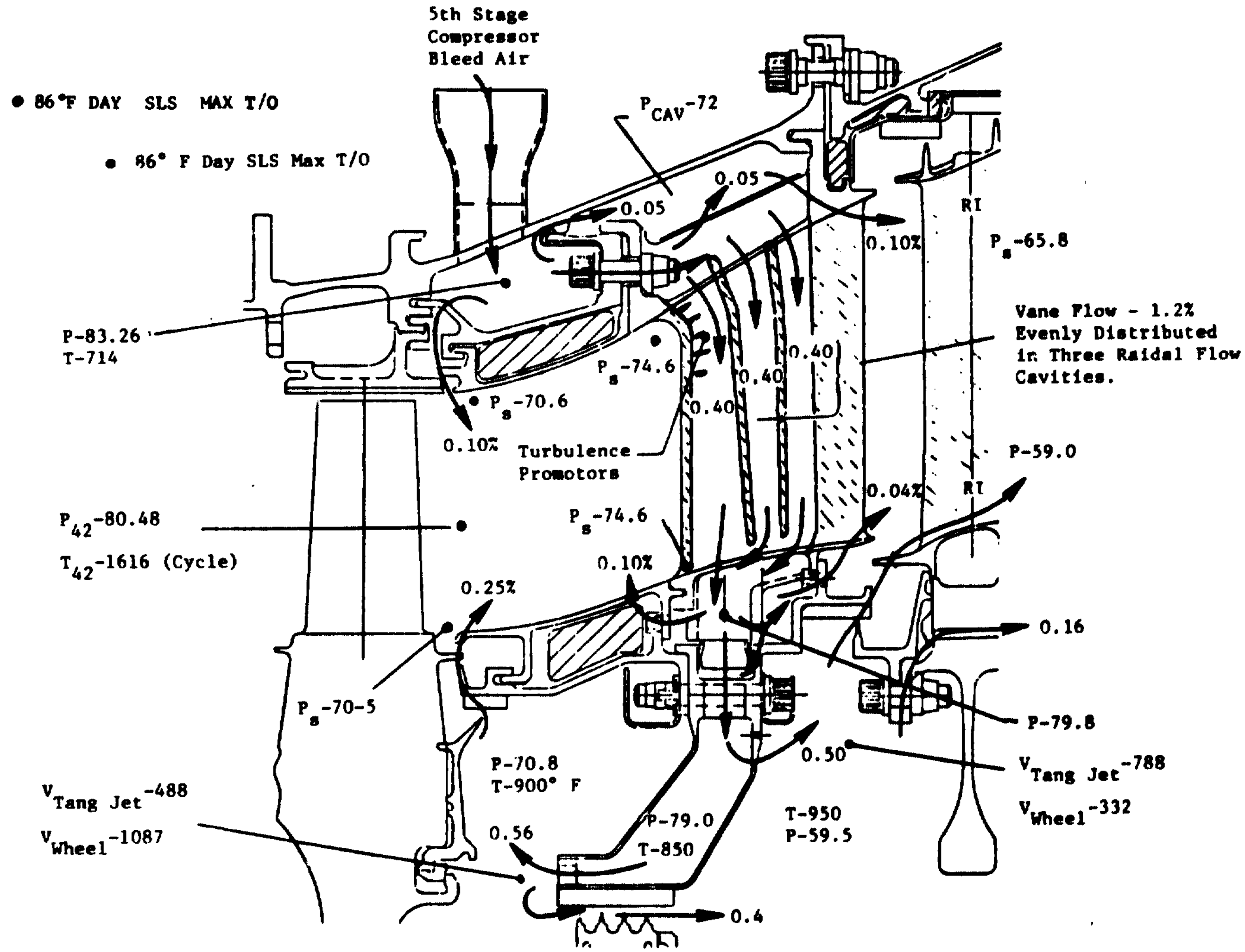

Figure 2.5-1. LPT Cooling Supply System (ICLS).

the inside of the casing. The cooling air then is fed into the 72 nozzle vanes and across the flowpath to the inner nozzle support structure. The cooling air warms up about 130° F while flowing through the vanes. Most of the nozzle cooling is done near the leading edge where the highest stresses occur. Once the cooling air reaches the nozzle hub, it is delivered into the wheel space supply plenum through 72 spoolies. The total cooling air passing through the nozzle is 1.2% W_{25} and of this, 0.14% is used to help purge the nozzle inner flowpath structure. Of the remaining 1.06% that enters the 360° wheel space supply plenum, 0.56% is supplied to the forward wheel space cavity. The plenum supply pressure is 79 psia while the forward wheel space cavity pressure is 70.8 psi. This yields a 1.08 pressure ratio across the forward wheel space injection holes. The holes are angled 60° in the circumferential direction which yields a tangential velocity of 488 ft/sec. This tangential velocity reduces the amount of boundary layer pumping that the high pressure rotor has to do. The wheel space cavity pumping analysis that was conducted indicates that the air will be about 50% of wheel speed. Since the aft cavity, the low pressure rotor cavity, is at a pressure of 59.5 psia, the cooling air injection pressure ratio is higher, thus a higher tangential velocity is achievable. The low pressure rotor is rotating at less than 30% of the high pressure rotor speed, which allows the tangential velocity leaving the injection holes to be better than twice the LP rotor wheel speed. With this system, a substantial amount of work will be obtained from the injected air as it is pumped up on the rotor disks.

Of the 0.56% W_{25} that is injected into the forward wheel space cavity, 0.4% leaks back through the interturbine seal and into the low pressure rotor cavity. Extensive seal clearance studies have been conducted on the interturbine seal to assure the proper quantity of blockage air. Over the engine operating range, it is expected that the seal clearance will vary between 10 and 27 mils, as shown in Figure 2.5-2 - the closest clearance occurring during a cold start takeoff transient of a new seal with the most open clearance occurring at nominal cruise power for a deteriorated seal. The seal will flow 0.67% W_{25} when the clearance has opened up to 36 mils. This could only occur as part of an engine failure and only then will hot flowpath gases start to be injested into the HP turbine aft rotor cavity. In conclusion, the seal blockage air is satisfactory from an engine safety standpoint.

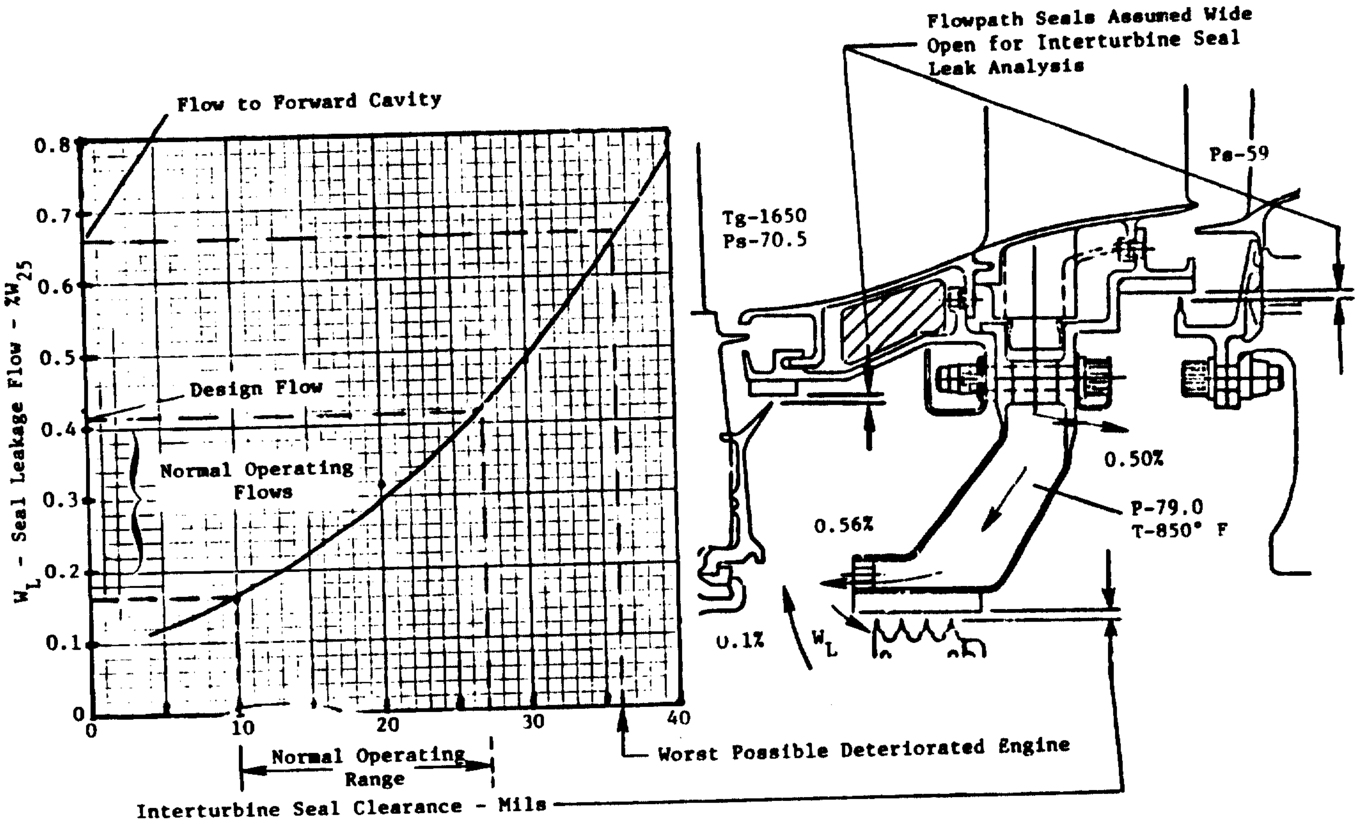

Figure 2.5-2. LPT/HPT Interturbine Seal Blockage.

Further work in the area of wheel space cavity purge is being completed in an effort to better understand the leakage/purge for the ICLS engine. The changes in LPT aerodynamics in the Stage 1 nozzle will change the seal blockage characteristics somewhat.

Stage 1 Vane Heat Transfer Analysis

The rib locations in the Stage 1 vane were located in such a manner that the heat transfer at the leading edge outer band was increased while keeping the pressure losses to a minimum. Turbulence promoters were put in the leading edge cooling cavity of the vane as shown in Figure 2.5-1 in an effort to further enhance the cooling side heat transfer. The wall thickness of the vane leading edge in the region of the vane/band fillet was kept to a level that was no higher than the vane leading edge at the 90% span.

A detailed steady state analysis at both the 95% and 90% span locations was completed. Steady state temperatures at both locations are presented in Figures 2.5-3 and 2.5-4. These temperatures indicate that the desired leading edge metal temperature reduction was achievable and the pressure losses in getting the air through to the rotor cavity were kept to a minimum.

Low Pressure Turbine Casing Heat Transfer Analysis

The CF6-6 engine experience was factored in the heat transfer analysis of the low pressure turbine casing. The heat transfer data indicated that the leakage/gas circulation heat transfer around the shroud hangers was significantly higher than expected. This heat transfer data was factored into the E^3 low pressure turbine casing. The analysis then indicated that the shroud hangers on the first two stages exceeded the 1250° F temperature limit. In order to overcome the excess temperature problem on the Inco 718 casing, the hangers were shortened and the contact area between the hot shrouds support and hangers was minimized. The heat conduction area between the hanger and the casing was also increased. The analysis was redone for the minimum and maximum active clearance control casing cooling air. The results of this analysis are presented in Figure 2.5-5 for minimum cooling and Figure 2.5-7 for maximum

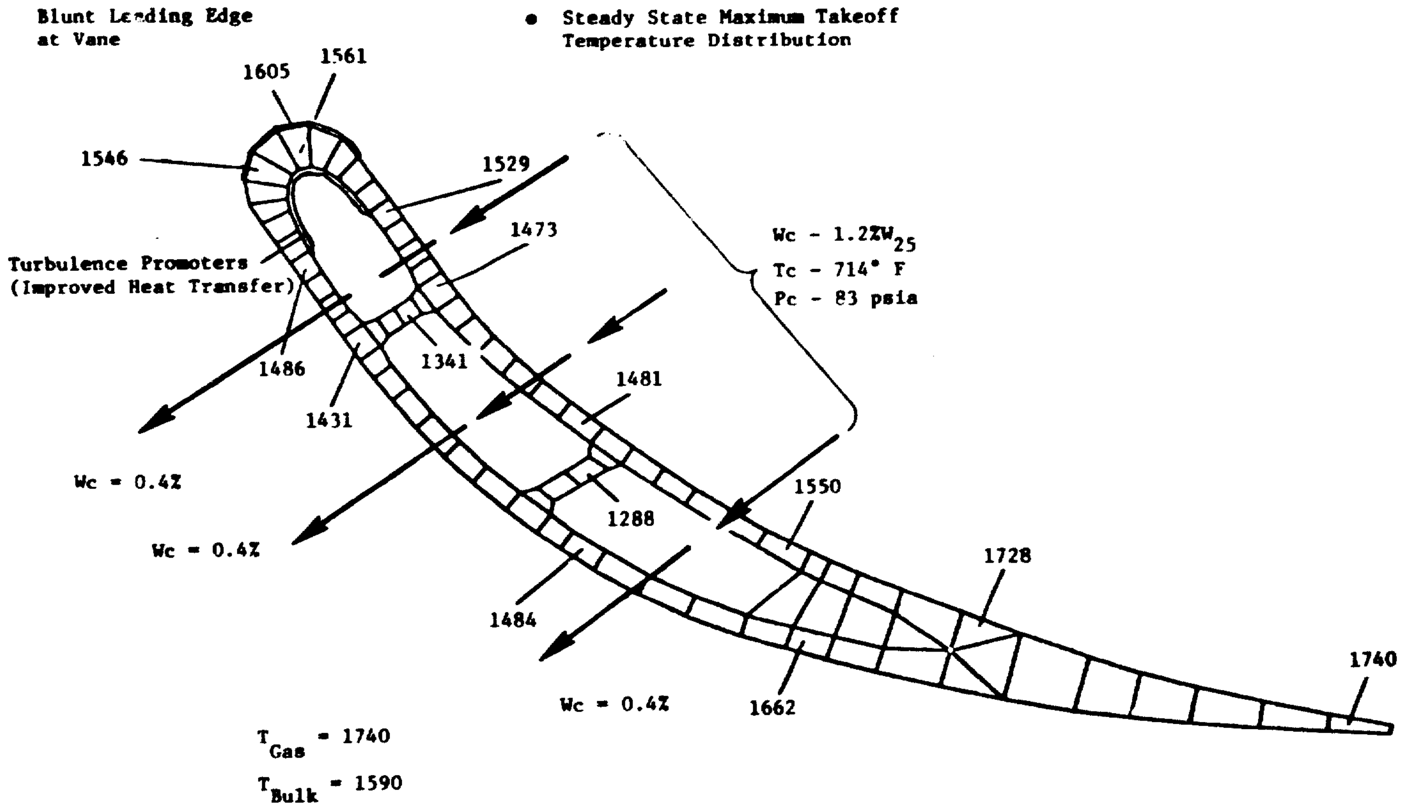

Figure 2.5-3. Low Pressure Turbine Stage One Vane, 95% Span.

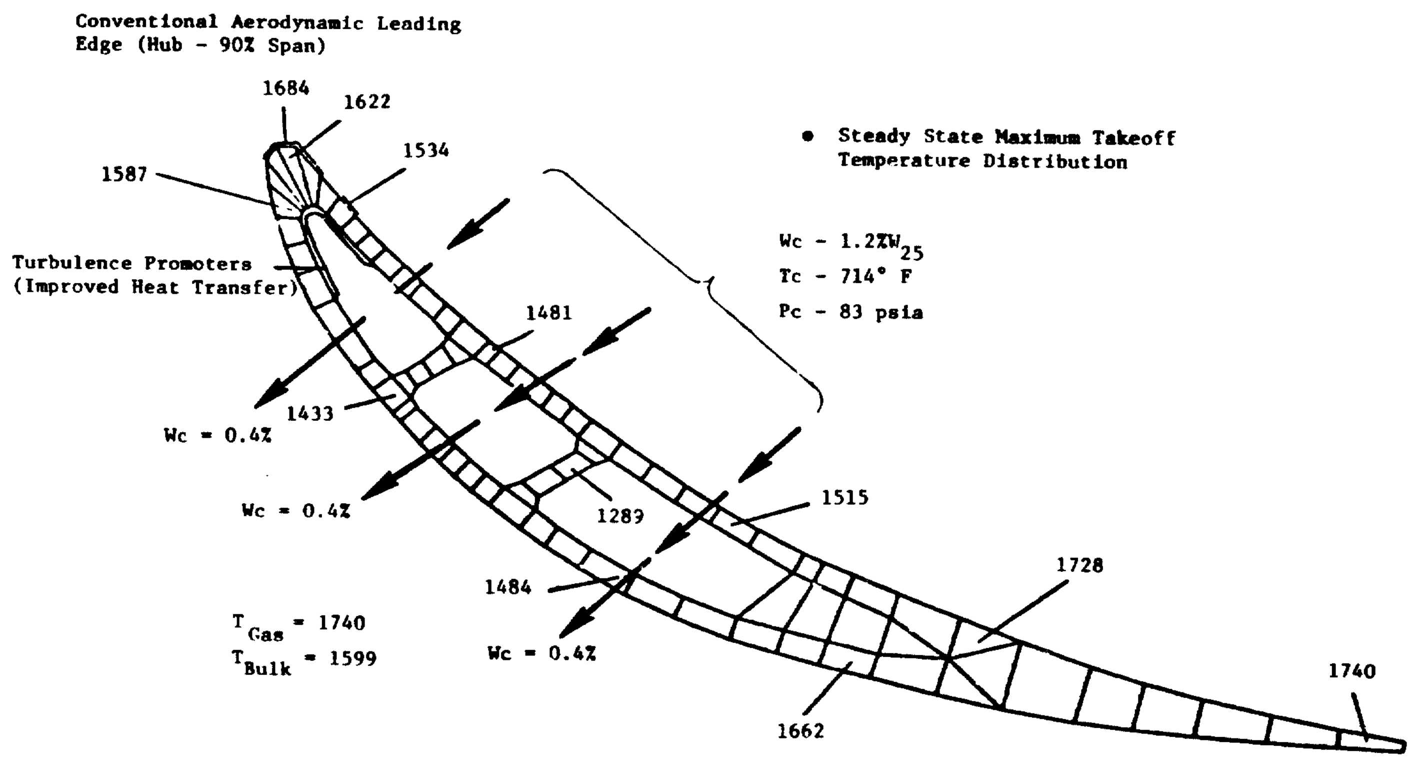

Figure 2.5-4. Low Pressure Turbine Stage One Vane, 90% Span.

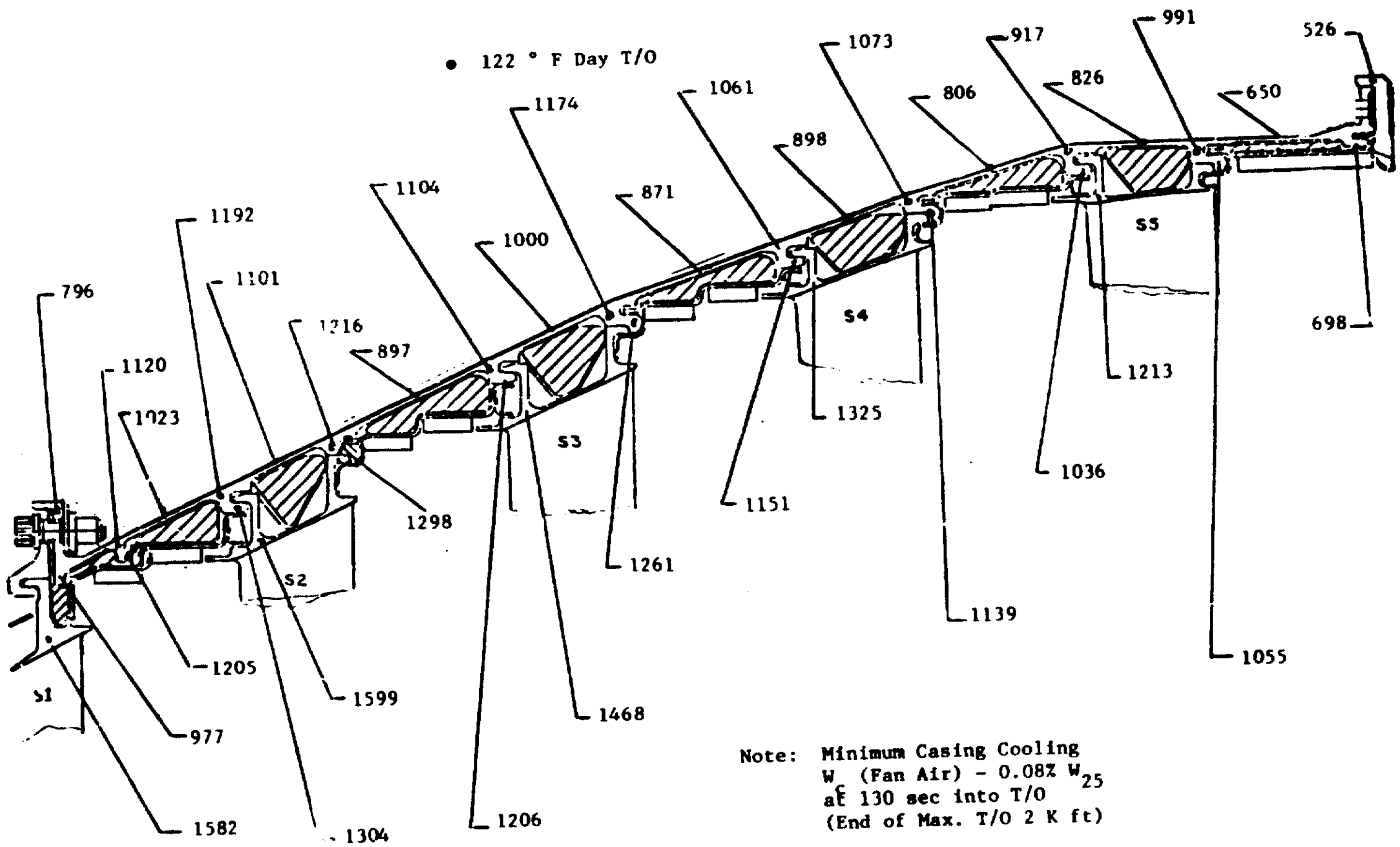

Figure 2.5-5. LPT Casing Transient Temperature Distribution.

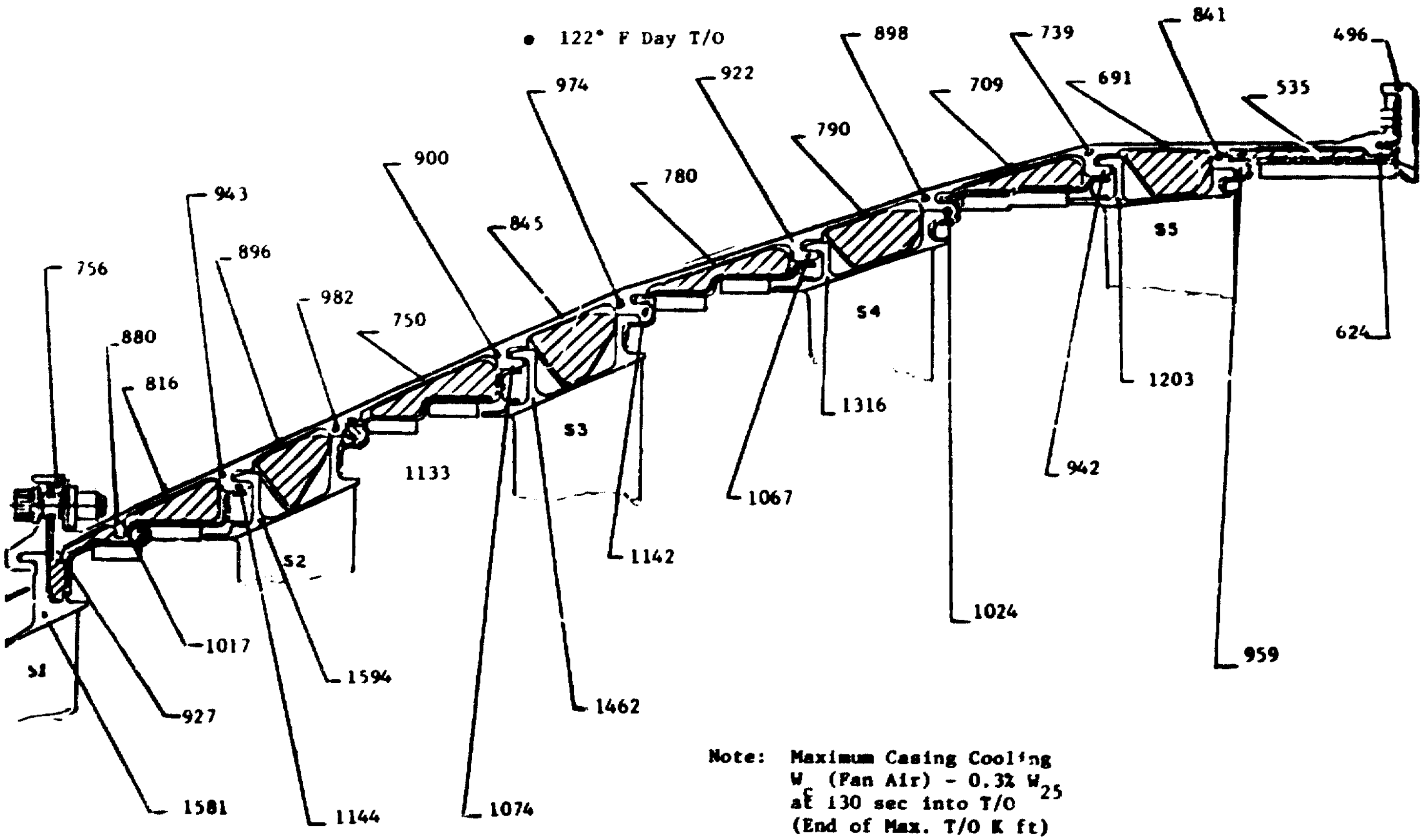

Figure 2.5-6. LPT Casing Transient Temperature Distribution.

cooling at the end of the 2-minute takeoff mission. The analysis indicates that with 0.08% W_{25} casing cooling the second stage nozzle hangers are still over 1250° F. With the maximum cooling of 0.3% W_{25}, the Stage 2 nozzle hangers are well below the 1250° temperature. These areas will be watched very closely in the ICLS to define whether the 0.02% additional flow will be required to keep the casing hanger temperatures at an acceptable level.

The casing cooling is accomplished through means of an impingment manifold which extends around the complete LP casing. The cooling is supplied by way of a scoop which is located in the fan bypass duct. The cooling air is then modulated by means of a valve which is located in the air supply line between the fan duct scoop and the LP casing impingement manifold. The impingement manifold consists of 4° to 90° sectors with one axial distribution plenum for each section, as shown in Figure 2.5-7. Half-inch-diameter tubes, emanating from the axial distribution plenum, supply the cooling air circumferentially around the casing. The cooling air then leaves the tubes through 0.025-inch impingement holes evenly distributed in each circumferential tube. The 0.025-inch-diameter hole is the minimum size that extensive commercial experience has shown does not cause a plugging problem. The hole spacing in each ring has been adjusted to give the desired cooling for each turbine stage. The spacing varies between an Xn/D of 9 up to 16 with 4411 holes, to yield a total impingement flow area of 2.165 in^2.

Stage 1 Nozzle Support Structure

As reported in the last semiannual report, the nozzle support structure has a tendency to separate from the casing during flight descent. This occurs since the nozzle support structure is exposed to the more effective cooling of the turbine purge air as it enters the LP turbine casing and flows circumferentially around and into each vane. A 360° shield had been placed over the seal ring to shield the seal from the turbine purge air. This helped reduce the seal gap to 5 mils. Further shielding is being incorporated in an effort to slow the cooling response even more. This shielding consists of six sheet metal baffles that are located opposite the pipes that feed the cooling air into the casing. These baffles will be 6-inches wide and will be bolted to

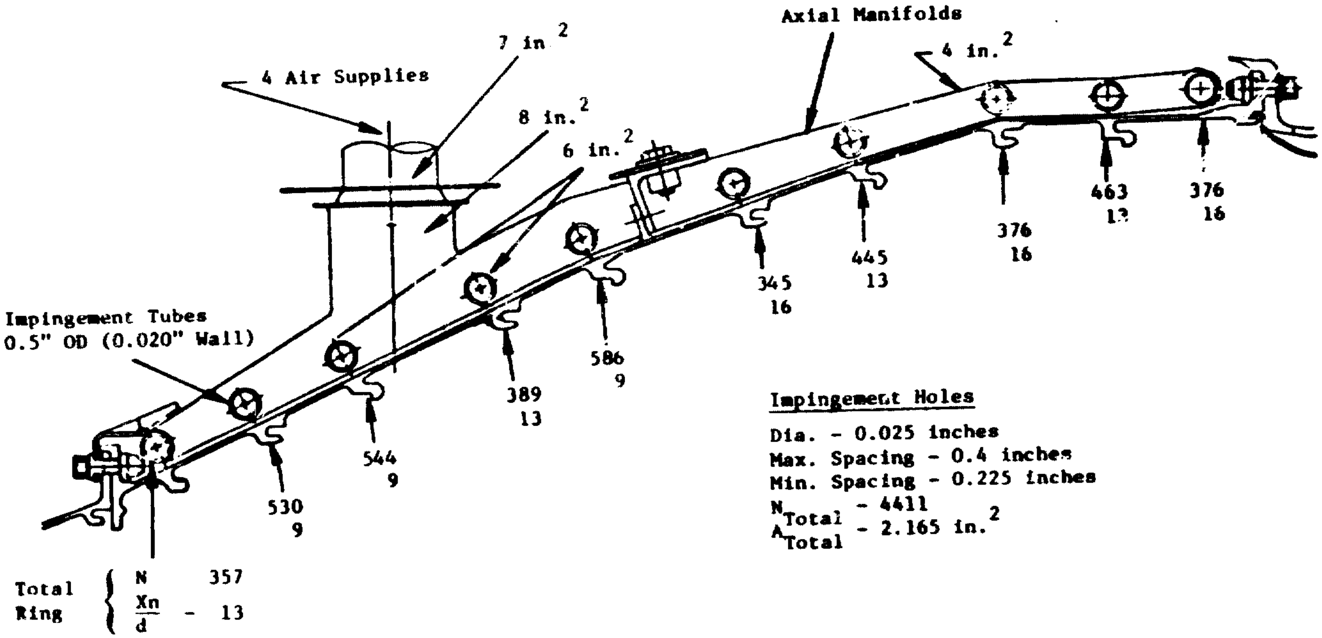

Figure 2.5-7. LPT Cooling/Active Clearance Control Impingement Manifold.

the nozzle support structure. Presented in Figure 2.5-8 is the baffle and the change in average ring temperature. The detailed thermal growth/mechanical growth analysis is currently in progress but the change in temperatures is projected to cut the seal clearance about in half. With an increase in the buildup interference fit of the seal, the gap will be eliminated.

Work Planned

- Follow hardware buildup to assure proper design intent cooling/ purge of the low pressure turbine.

2.5.3.2 LPT Rotor Mechanical Design

Technical Progress

Work during this reporting period has resulted in the completion of the major detail design calculations, their presentation at the NASA LPT Detail Design Review, and the completion and issuing of all rotor detail drawings for the ICLS engine. A detailed description of these accomplishments and the final design configuration follow.

The LPT Detail Design Review was held on December 11, 1980. Formal notification of design approval was received on January 8, 1981.

The present rotor design is essentially the same as in the September 1980 semiannular report with a few additions.

Blades

Analyses were performed to determine combined disk-blade vibration frequencies and to establish that there were no resonances in the engine operating range. The two, four, and six diameter modes were investigated and their Campbell diagrams were plotted as shown in Figures 2.5-9 through 2.5-13. A review of these plots showed that even the potentially limiting mode, the two diameter, was no problem. This had greater-than-desired frequency margin with the 2-per-rev excitation line throughout the engine operating range.

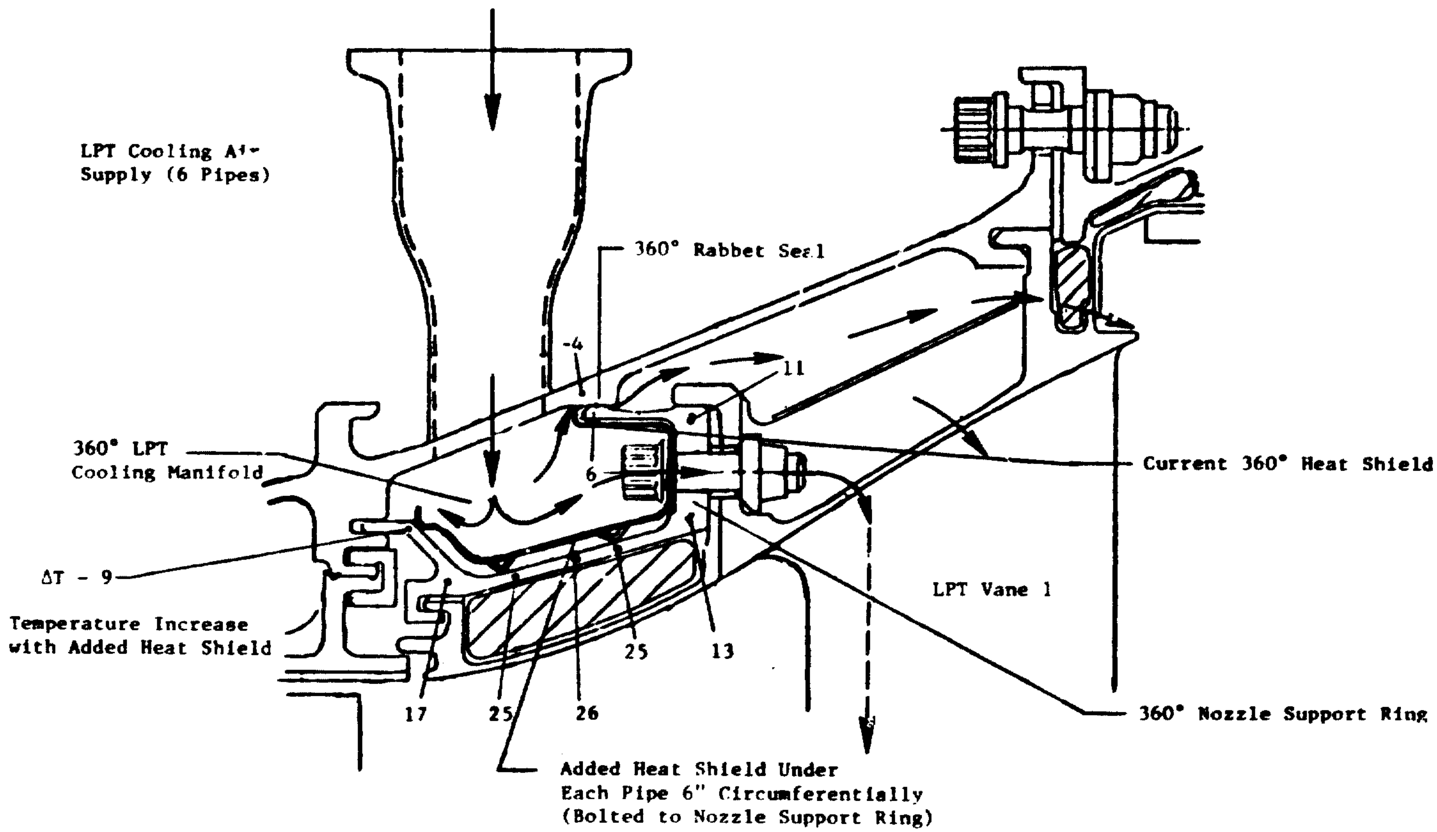

Figure 2.5-8. Low Pressure Turbine Stage Shield Temperature Improvement at 450 Seconds into Descent Transient.

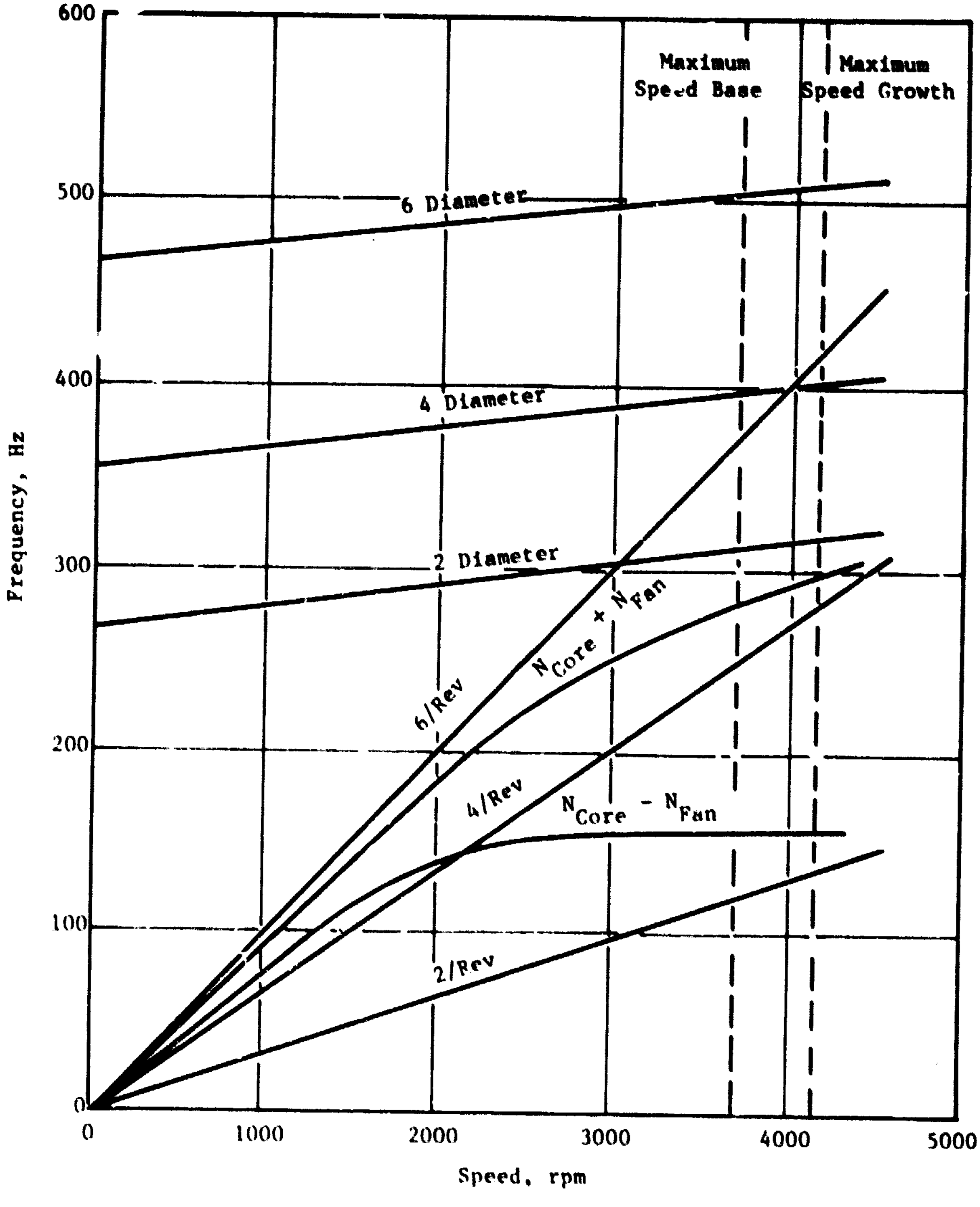

Figure 2.5-9. Stage 1 Coupled Blade Disk Campbell Diagram.

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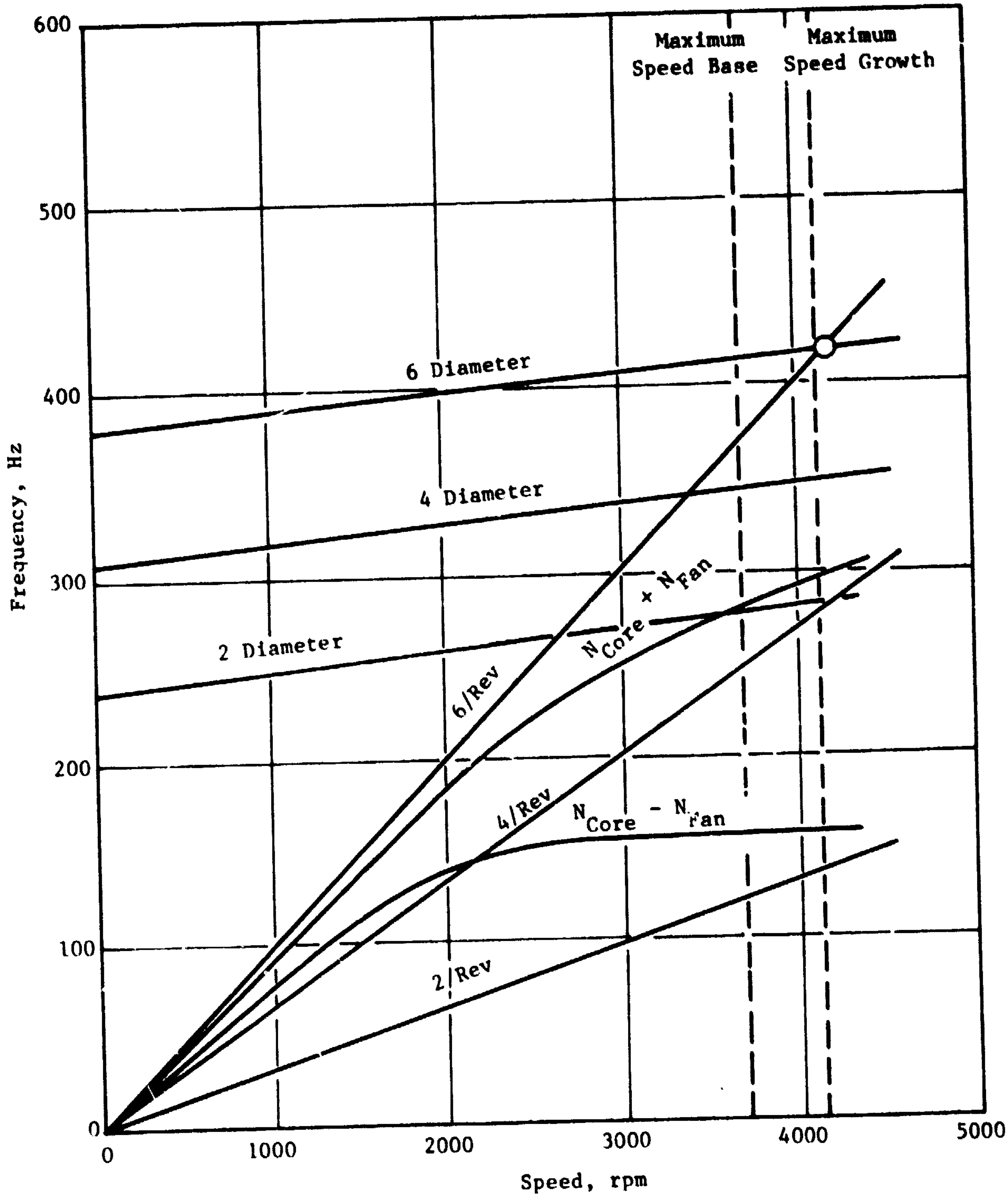

Figure 2.5-10. Stage 2 Coupled Blade Disk Campbell Diagram.

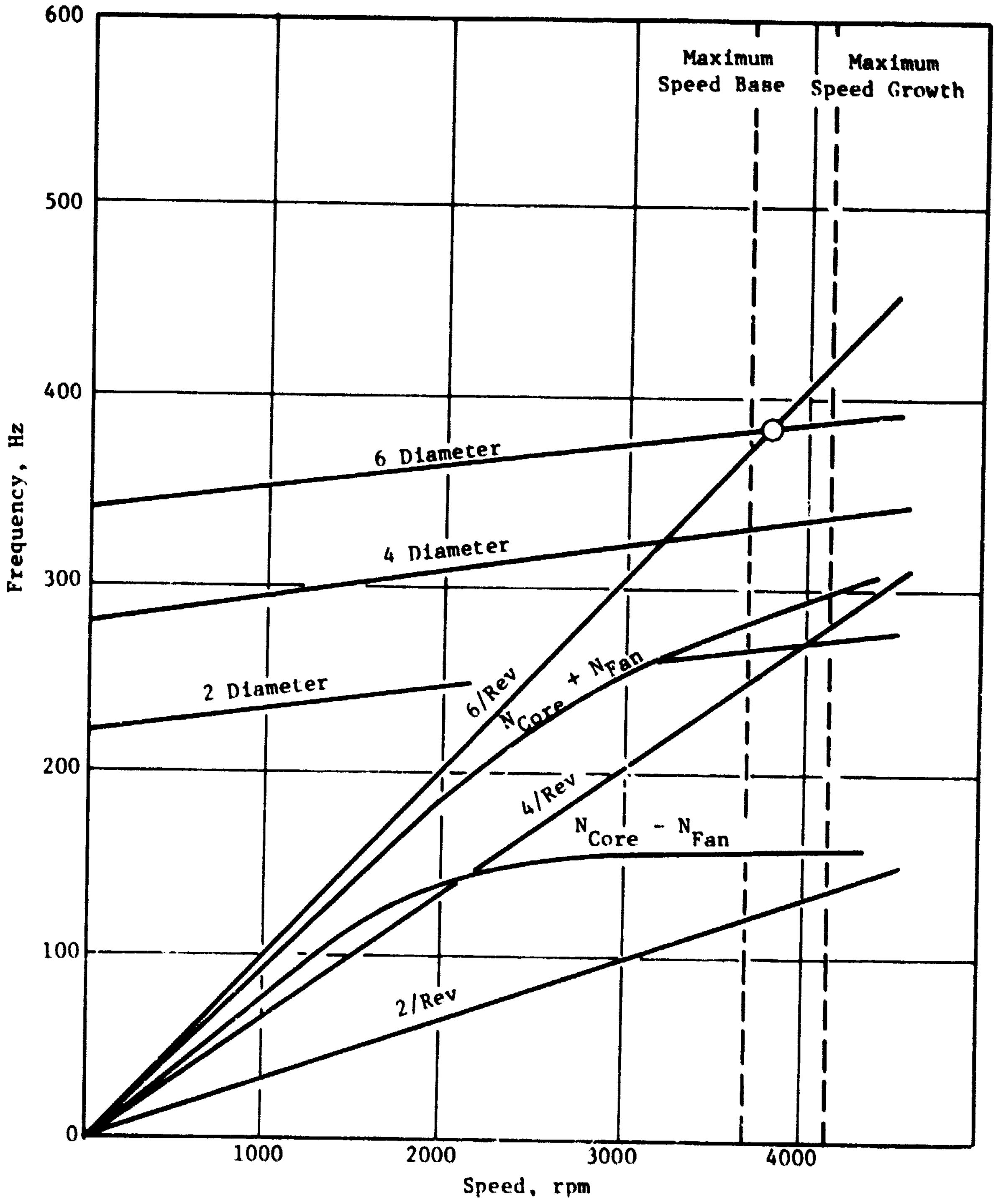

Figure 2.5-11. Stage 3 Coupled Blade Disk Campbell Diagram.

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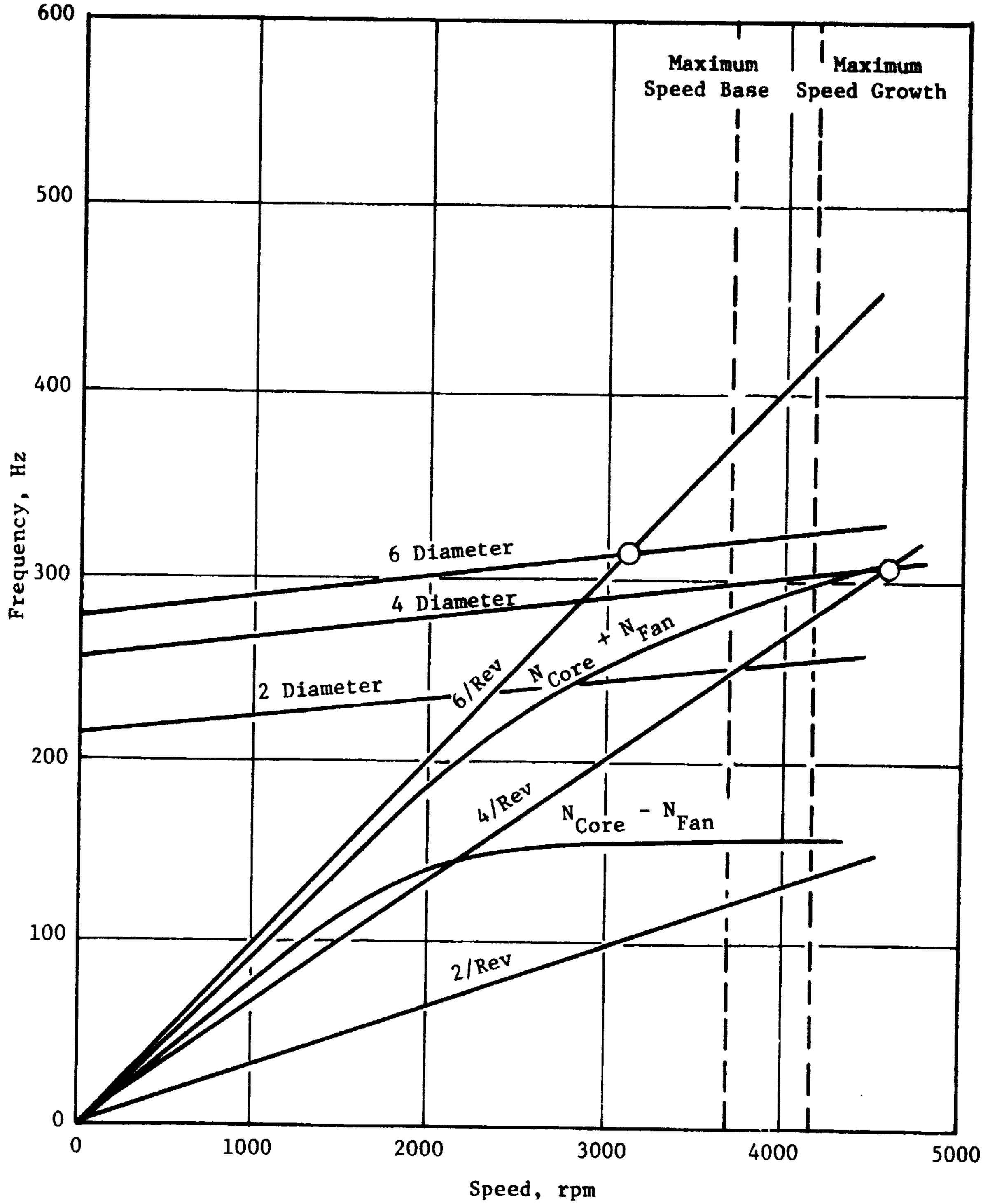

Figure 2.5-12. Stage 4 Coupled Blade Disk Campbell Diagram.

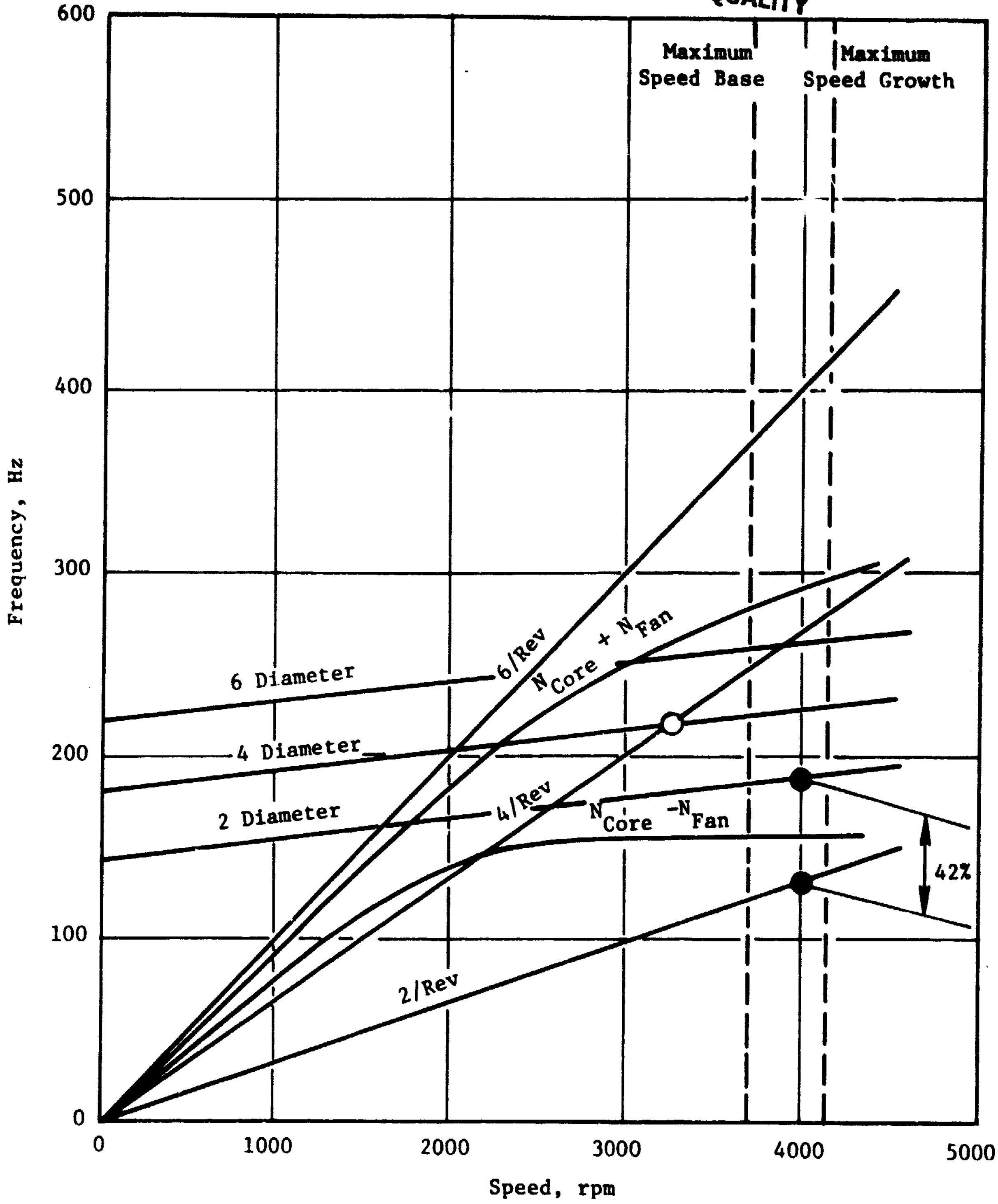

Figure 2.5-13. Stage 5 Coupled Blade Disk Campbell Diagram.

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The LPT blades are retained in their dovetial slots by two different methods. On Stages 1, 2, and 3, they are retained from aft movement by integrally cast retainers on the ends of the dovetails as shown in Figure 2.5-14. They are retained against forward movement by the rotor seals. On Stages 4 and 5, the blades are retained against forward and aft movement by bent-tab retainers as shown in Figure 2.5-15.

The Stage 1, 2, and 3 blade retainers were designed so that they would not exceed material limits when a maximum expected force (larger than the calculated steady-state force) was applied at the end of the retainer. Figure 2.5-14 shows that all three retainers either meet or exceed the required strength.

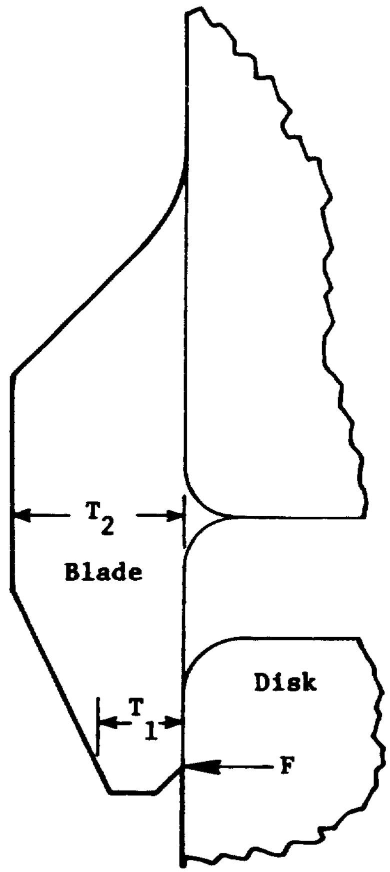
The designs of the Stage 4 and 5 blade retainers were based on correlations with CF65 load tests. Thickening of the retainers was necessary to increase their pushout strengths to meet requirements. The depths of the disk dovetail slots were increased slightly to accommodate the thickened retainers. Based on the data shown in Table 2.5-II, the blade retainers are expected to have pushout strengths greater than their calculated design loads.

Table 2.5-II. Blade Retainer Sizing: Stage 4 and 5.

	Stage	Thickness, in.	I_{MIN}, in.^{4}	Pushout Force, lb	Calculated Load, lb
Test Data (CF6)		0.080	1.17×10^{-5}	480	345
E^3	4	0.090	0.92×10^{-5}	377(est)	238
	5	0.100	1.41×10^{-5}	578(est)	390

Rotor

Detailed analyses were conducted to determine the disk stress concentrations caused by the bolt holes, scalloped flanges, and cooling air grooves (on the flange faces). These analyses show that a slight modification of the foward area of the Stage 1 disk will be needed to reduce the stresses at the



	I_1 in.	I_2 in.	F lb	$\sigma_{Max.}$ ksi*
Stage 1	0.043	0.105	201	90
Stage 2	0.050	0.115	248	91
Stage 3	0.068	0.135	351	92

*0.2 Yield Strength = 92 ksi at 1200° F

Figure 2.5-14. Blade Retainers - Stages 1, 2 and 3.

- Configuration - Sheet Metal
- Material - Inco 718
- Analysis/Design - Utilize CF6-50, -80 Load Tests

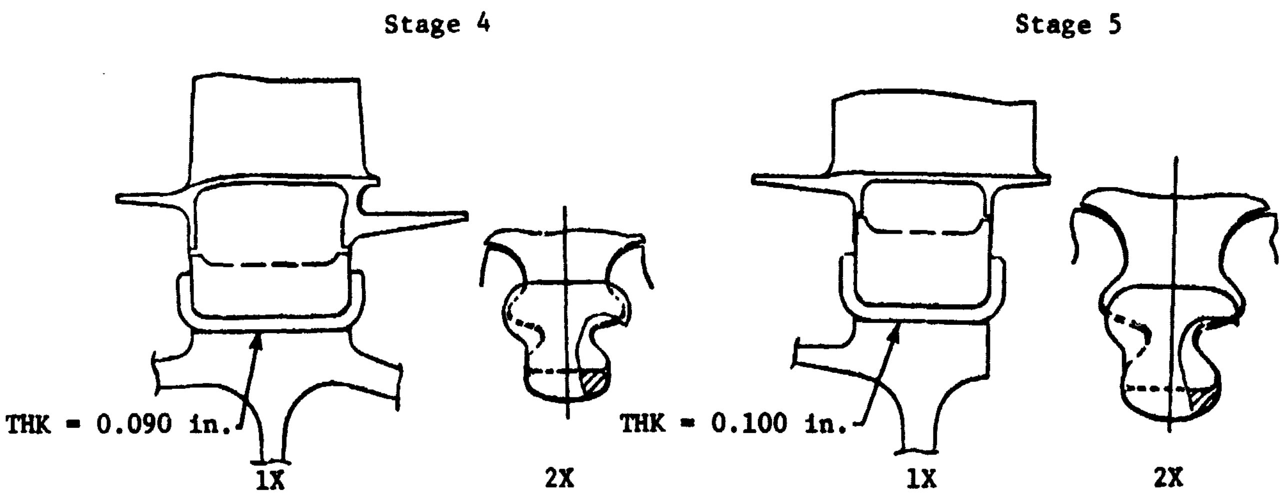


Figure 2.5-15. Blade Retainers - Stages 4 and 5.

bolt holes. This modification can be accompllished within the present rough machined forging envelope with no schedule delay.

The bore of the Stage 4 disk has been modified to allow additional radial space for a sump seal change. The disk bore radius has been moved outward 0.64 inches and thickened radially and in the fillet blend. The new design meets full cyclic life and overspeed requirements with a weight penalty of less than 3 pounds.

Drawing Releases

With the release of the Stage 4 and 5 blade retainer drawings in mid-February, all rotor detail drawings have been issued.

Drawings Issued

The following is a list of the LPT rotor detail drawings issued to date, all of which were issued during this reporting period, with the exception of the forgings.

Drawing Number	Item
4013267-136	Disk, Stage 1-5 Forgings
4013205-784	Disk, Stage 1 Machining
4013205-785	Disk, Stage 2 Machining
4013205-786	Disk, Stage 3 Machining
4013205-787	Disk, Stage 4 Machining
4013205-788	Disk, Stage 5 Machining
4013267-070	Seal, Stage 1-5 Forgings
4013205-968	Seal, Stage 1 Machining
4013205-969	Seal, Stage 2 Machining
4013205-983	Seal, Stage 3 Machining
4013205-970	Seal, Stage 4/5 Machining
4013267-251	Blade, Stage 1 Casting
4013267-252	Blade, Stage 2 Casting
4013267-253	Blade, Stage 3 Casting

Drawing Number	Item
4013267-254	Blade, Stage 4 Casting
4013267-255	Blade, Stage 5 Casting
4013267-271	Blade, Stage 1 Machining
4013267-272	Blade, Stage 2 Machining
4013267-273	Blade, Stage 3 Machining
4013267-274	Blade, Stage 4 Machining
4013267-275	Blade, Stage 5 Machining
4013296-060	Blade Retainer, Stage 4
4013296-061	Blade Retainer, Stage 5
J644P08*	Bolt, Stage 1 Forward
4013205-905	Bolt, Stage 1/2, 2/3, 4/5
4013205-887	Bolt, Stage 3/4
4013205-913	Bolt Retainer, Stage 3/4
4013205-897	Nut, Stage 1 Forward
J980P05*	Nut, Stage 1/2, 2/3, 4/5
J980P06*	Nut, Stage 3/4

* - Standard Drawings

Work Planned

- Complete the analysis of the disk bolt hole stress concentrations and incorporate any changes that are required
- Review ICLS operating conditions and the resulting effects on life and clearances, both axial and radial
- Complete the rotor assembly drawing and ICLS engine parts listing
- Complete detail design resort.

2.5.3.3 LPT Stator Mechanical Design

Technical Progress

Work during this reporting period has resulted in the completion of the major detail design calculations, their presentation at the NASA LPT Detail Design Review, and the completion and issuing of all stator detail drawings for the ICLS engine. A detailed description of these accomplishments and the final design configuration follow.

LPT Shroud Configuration

All five stages of the LPT shrouds are shown in Figure 2.5-16. A detail drawing has been issued for each stage of shrouds. Representative shapes are shown in greater detail in Figures 2.5-17, 2.5-18, and 2.5-19. All of these shrouds have been configured using the same design concept. At the front of each shroud, they engage a hook portion of the LPT casing. A tengential stop brazed into the shroud clip engages a slot in the casing hook. The clip captures the aft leg of the preceding stage nozzle and retains it against the casing hook. This clip serves two functions: (1) it positions the shroud radially at the front and (2) it resists unseating of the aft leg of the nozzle. This is accomplished without the use of a bolted flange and results in a lightweight design.

At the aft end, each shroud, except for Stage 5, is trapped radially between a casing hook and outer flowpath portion of the adjacent nozzle as shown in Figures 2.5-17 and 2.5-18. Stage 5 is positioned radially by a sheet metal part of the shroud at the aft end that clips over the casing hook as shown in Figure 2.5-19.

Axially, each shroud is positioned against the adjacent hardware - that is, the respective stage nozzle for shrouds Stages 1 through 4 and the turbine frame flange for Stage 5. This axial contact with the adjacent hardware provides a spring load that forces the nozzles aft against their axial stops on the casing hooks, and forces the tangential stops in the shroud forward into their slots. The spring load is developed by the windage shields at the forward inner extremity of each shroud stage.

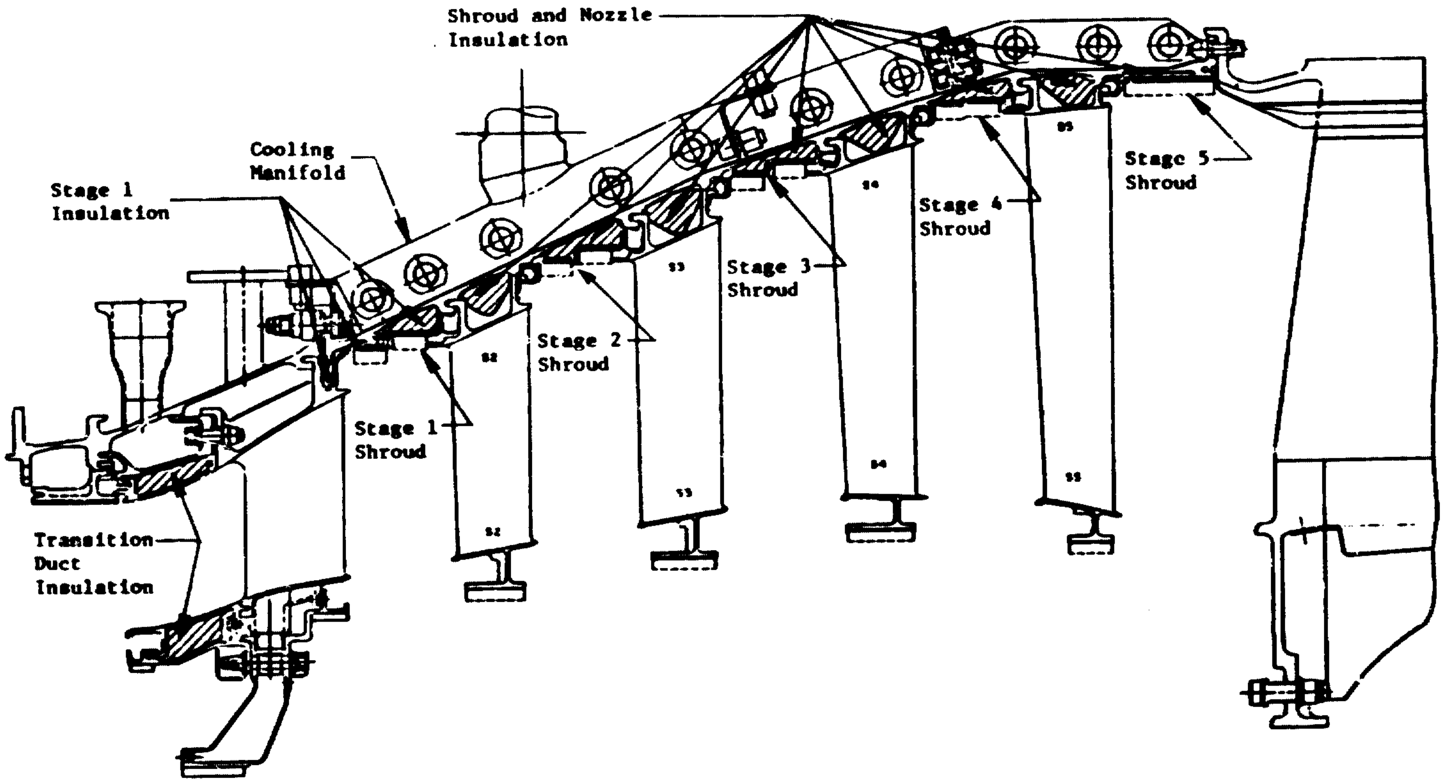


Figure 2.5-16. LPT Stator Insulation, Shrouds and Cooling Manifold.

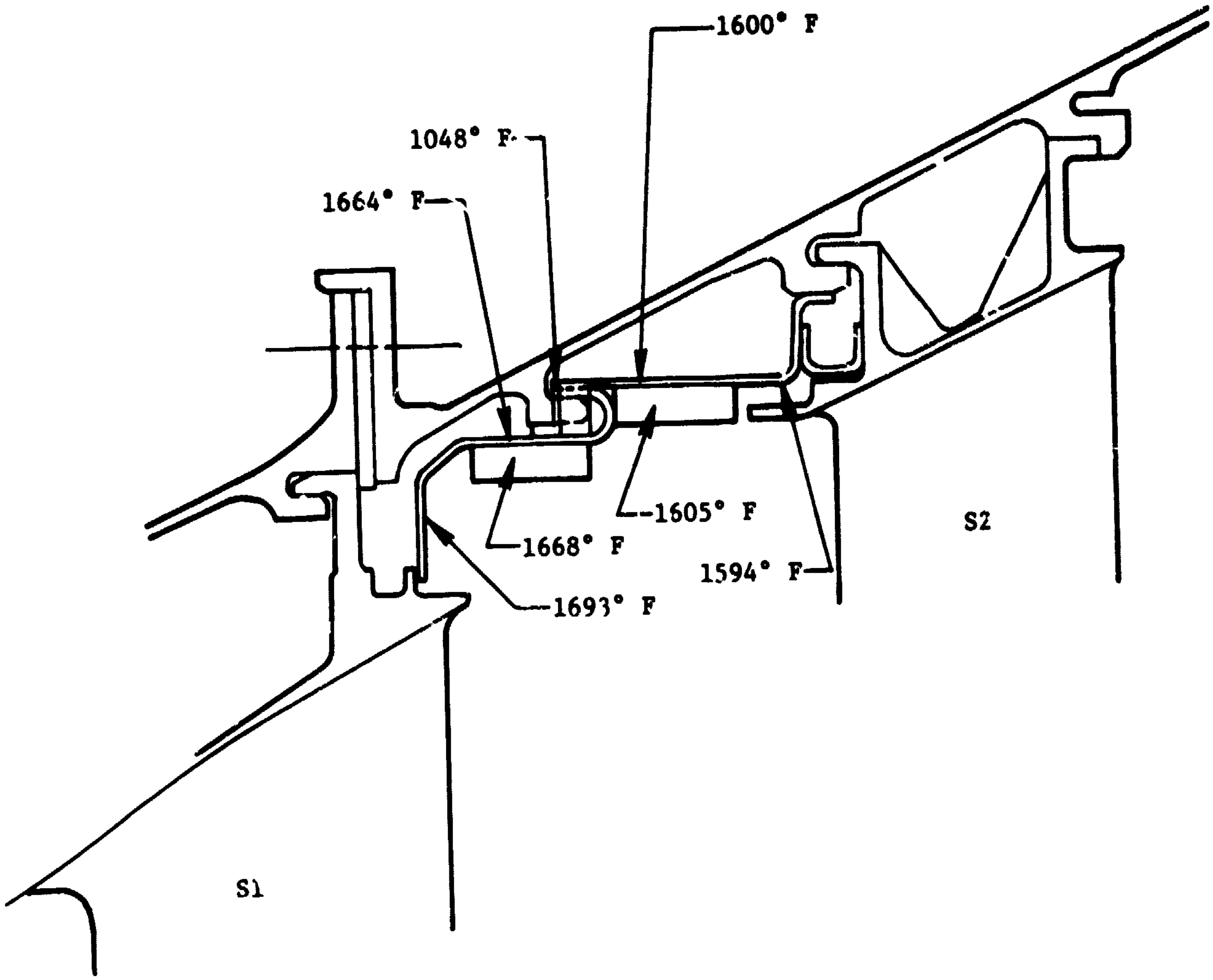


Figure 2.5-17. LPT Stage 1 Shroud Maximum Temperature at End of Takeoff.

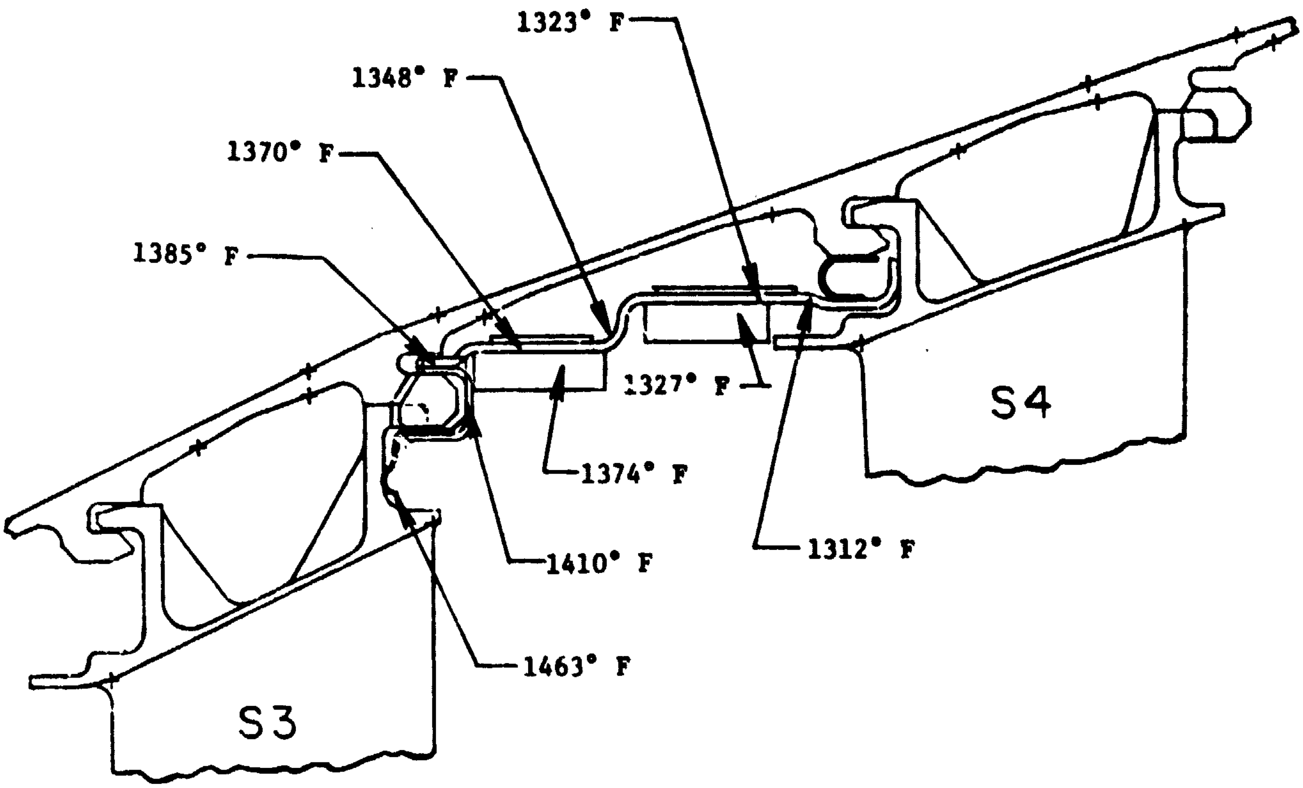


Figure 2.5-18. LPT Stage 3 Shroud Maximum Temperatures at End of Takeoff.

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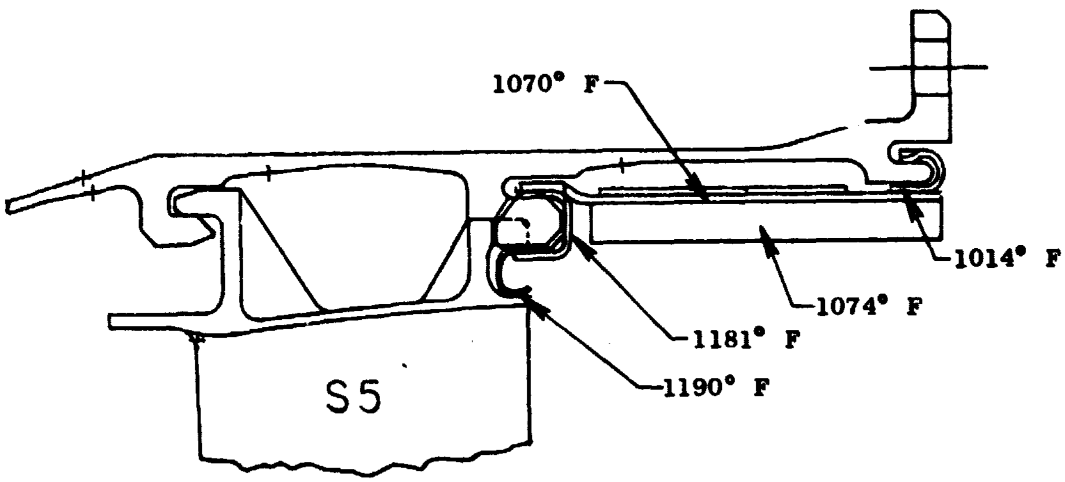

Figure 2.5-19. LPT Stage 5 Shroud Max. Temperature at End of Takeoff.

Clearance for relative thermal expansion between the shroud and casing is provided axially in the forward clip to casing hook interface, and between the tangentially stop to the forward end of its slot interface. Such positioning of the tangential stop provides the key to lock the nozzle, shrouds, and case into their selected tangential positions.

Shroud Temperature and Material

The maximum temperatures for representative stages of the shrouds are shown in Figures 2.5-17 through 2.5-19, and occur at the end of the takeoff condition. To withstand these temperatures, the shrouds are fabricated from Hastelloy X honeycomb (H/C) brazed to René 41 sheet stock. The H/C seals are provided to accommodate blade tip seal teeth rubs with minimal tooth wear.

Insulation

Insulation blankets have been designed and detail drawings have been issued for 13 locations in the LP turbine. These locations are shown in Figure 2.5-16 and are as follows: two in the transition ducts (inner and outer), three over the first stage nozzle and shroud, and eight over the remaining four stages of shrouds and nozzles. These insulation blankets are composed of an insulation material encased in Inco 600 foil. Use of these insulation blankets reduce energy loss from the LPT and lowers the temperature requirements of the LPT casing material.

LPT Cooling Manifold

The LPT cooling manifold design is shown in Figures 2.5-20 and 2.5-21. The design is complete and detail drawings have been issued. The LPT cooling manifold is a part of the Active Clearance Control (ACC) system. The function is to distribute air from the fan air flow to be impinged on selected portions of the LPT case. There are ten such selected locations which are aligned radially with the outer diameter of the ten nozzle and shroud support hooks. By controlling the temperatures of these support hooks, the casing diameter size is controlled which, in turn, controls the shroud seal

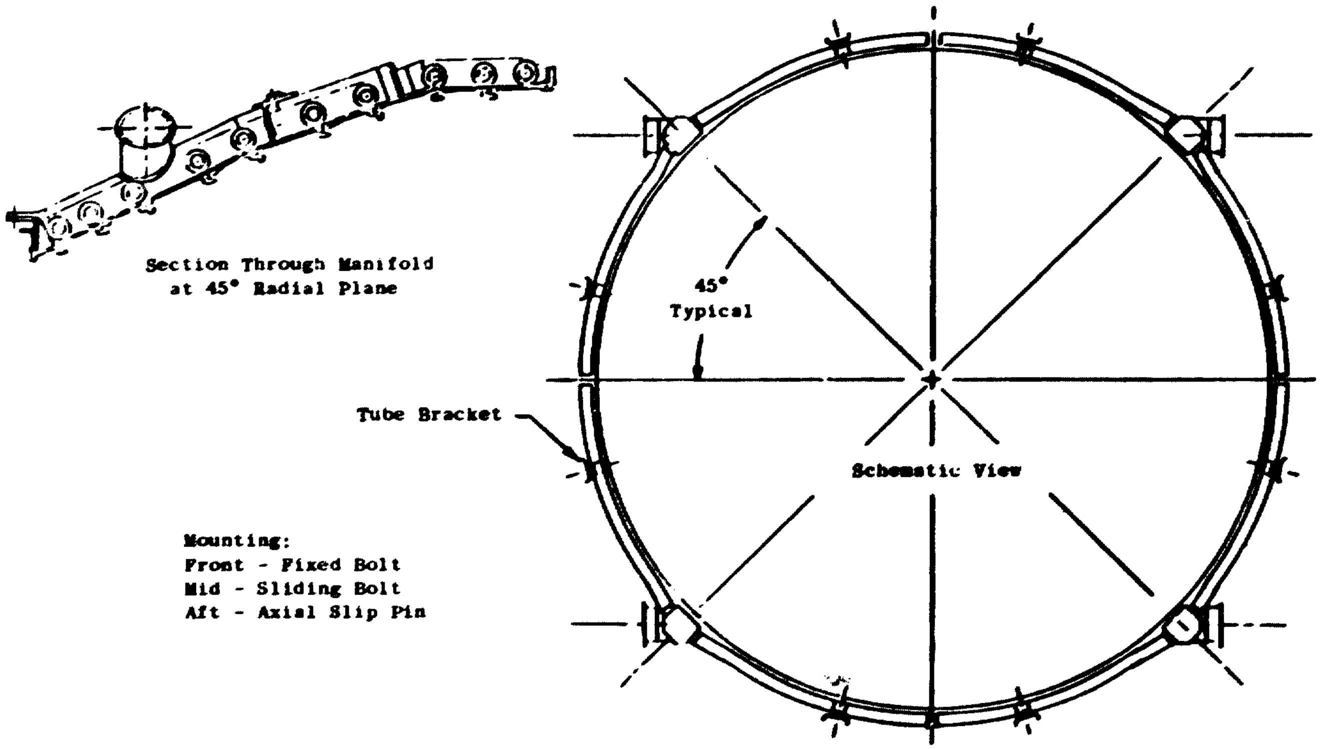

Figure 2.5-20. LPT Cooling Manifold - Axial View.

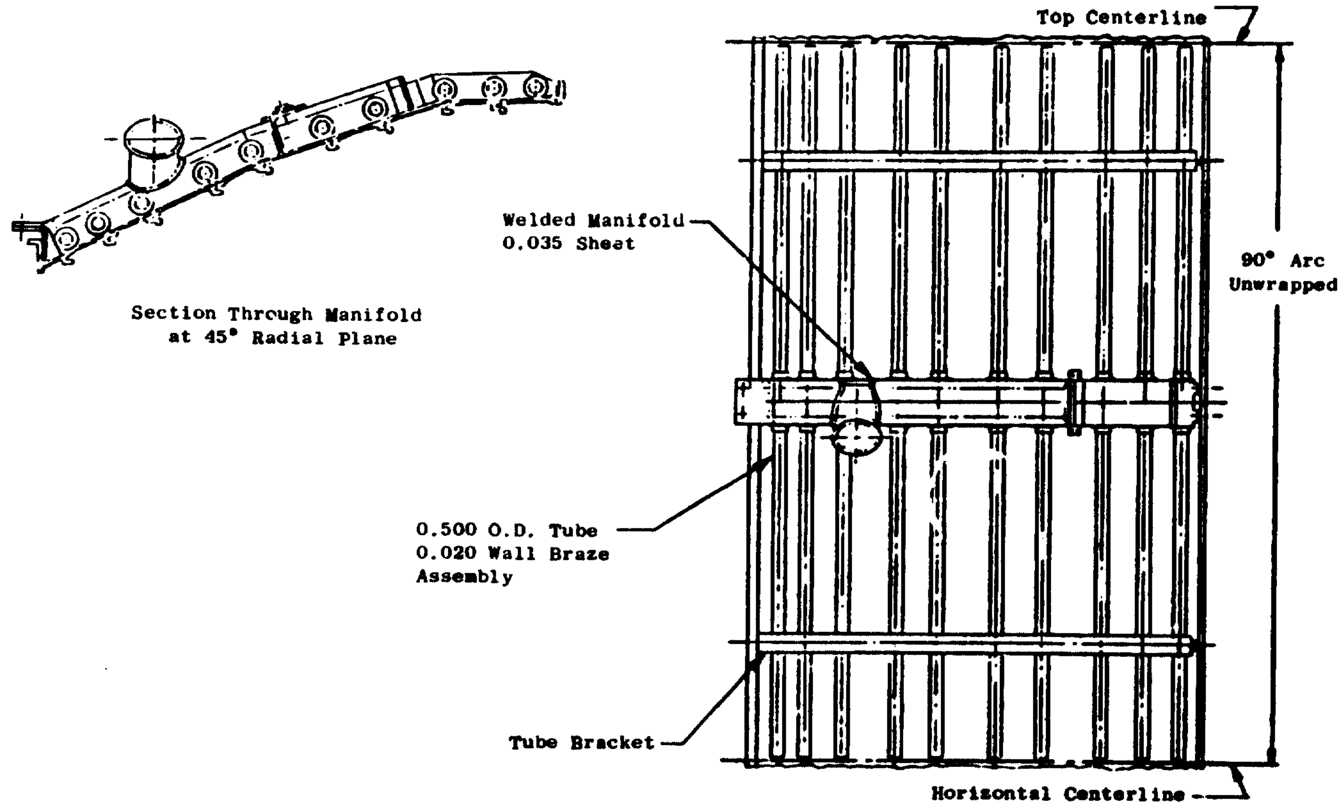

Figure 2.5-21. LPT Cooling Manifold - Unwrapped View.

clearance at the blade tip of all five stages. Thus the efficiency of the LP turbine which is dependent on this clearance is improved.

The LPT cooling manifold design consists of four sectors each covering a 90° arc sector. Each sector is a backbone/rib type configuration - that is, an axial distribution manifold with 10 circumferential tubes extending from each side of the manifold. Each axial manifold is subdivided into an aft and a forward part. The forward part of the manifold contains the seven distribution tubes impinging on the first four stages of the LP turbine, while the aft portion contains the three distribution tubes impinging on Stage 5 of the LP turbine. The two parts of the manifold are connected by a bolted flange. By using a blankoff plate, the impingement cooling can be eliminated or metered to Stage 5. This provides the option to study the effectiveness of the ACC cooling and tip clearance control on Stage 5 of the LP turbine.

The material used to fabricate the cooling manifold (including the tubes and support brackets) is 321 stainless steel. The construction is a welded and brazed assembly of tubing and sheet stock. The manifold assembly has been designed with liberal tolerances and soft tooling considerations to achieve cost effectiveness in fabrication of small lot orders such as this cooling manifold hardware for ICLS.

The positioning of the cooling manifold is accomplished by a forward, mid, and aft mount at both the axial manifold backbone and at the tube support on each side of the axial manifold. The forward mount is a hard mounted, bolted joint. The midmount is a bolted joint, spring mounted to allow axial slip. The aft mount is composed of two pins captured in, and protruding from, the LPT aft flange. These pins engage the aft end of the manifold and the tube supports to provide tangential positioning yet allow axial slip. By allowing axial slip at both the mid and aft support, the differential thermal expansion between the LPT case and the cooling manifold is accommodated.

Another feature of the cooling manifold is the capability of separating the HPT/LPT casing interface flange (that is, removal of the LPT module) without requiring the removal of the cooling manifold. This is accomplished by positioning the forward mount brackets on the aft side of the bolted flange and designing the HPT/LPT flange bolts to be inserted from the aft side of the

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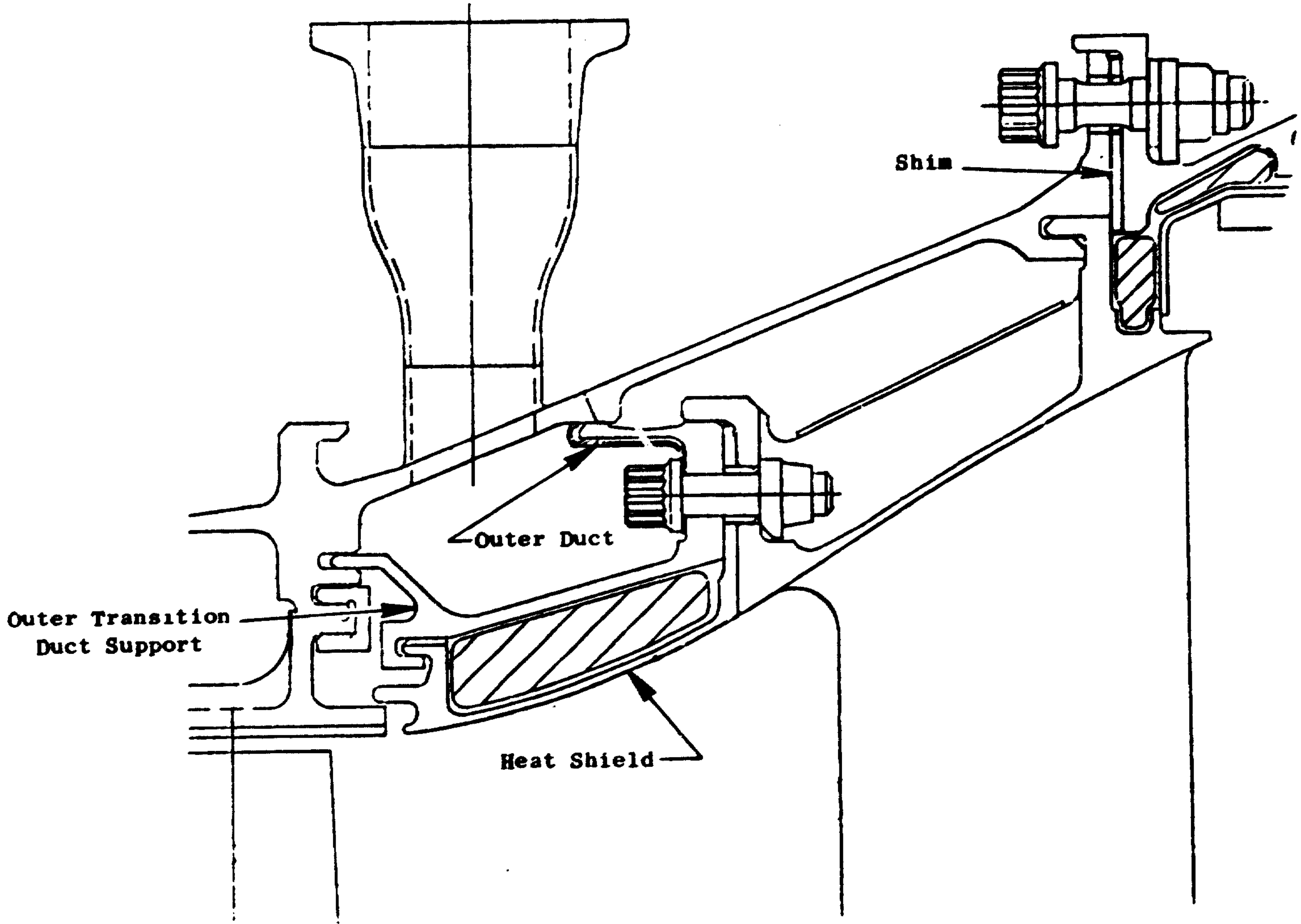


Figure 2.5-22. LPT Outer Duct/Stage 1 Nozzle Configuration.

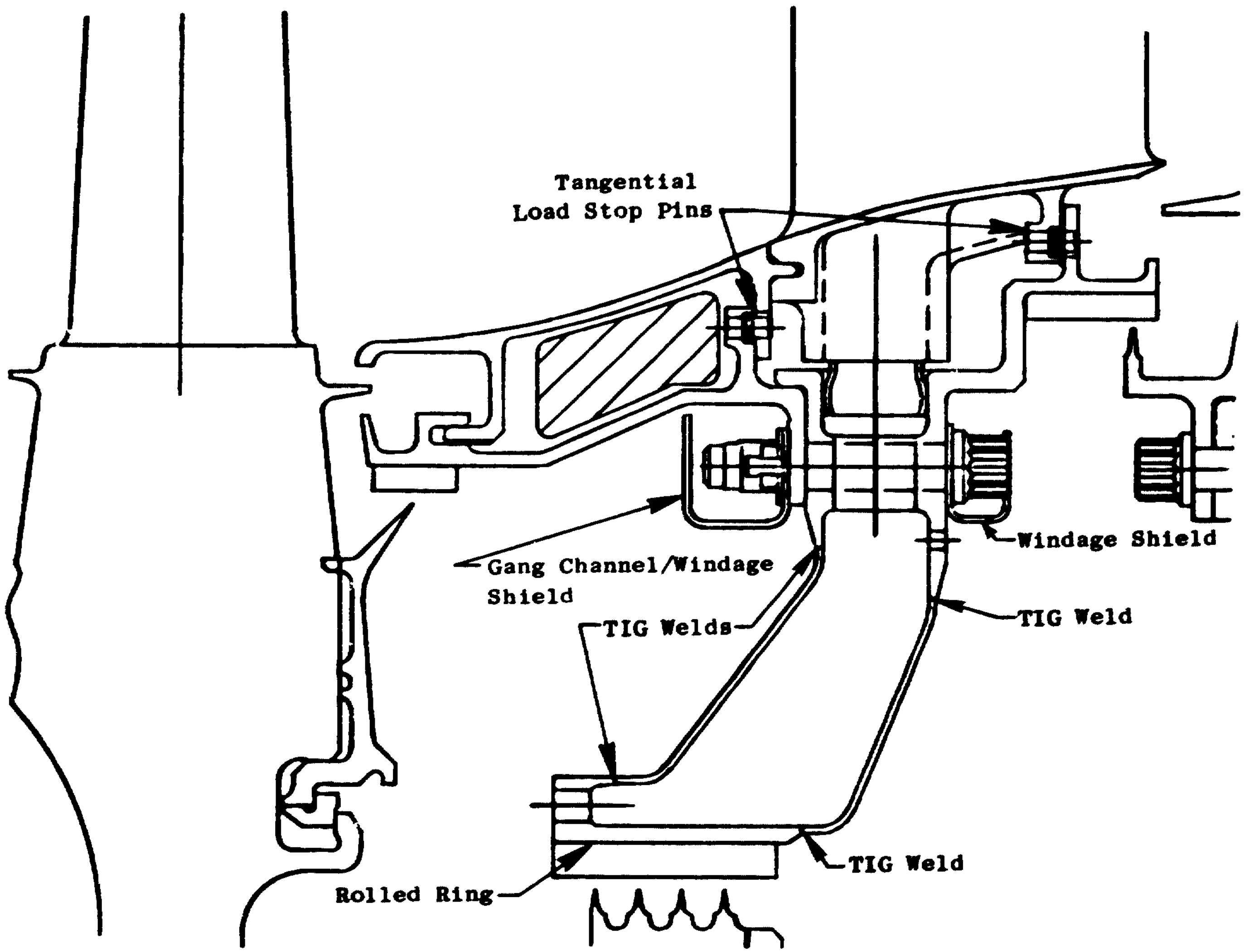
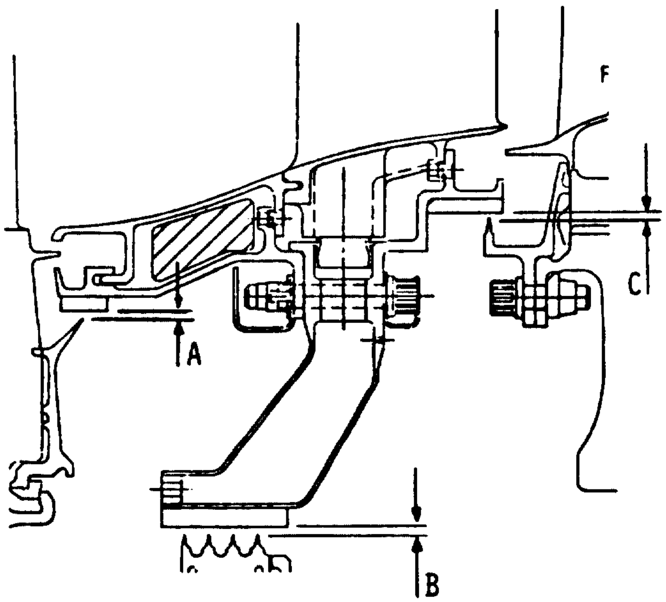


Figure 2.5-23. LPT Forward and Aft Inner Seals.



RADIAL CLEARANCES

(INCHES)

	A	B	C
COLD ASSY.	.030	.030	.030
H.D.T.O.	.037	.027	.024
MAX CLIMB	.016	.015	.023
MAX OPEN (1)	.086	.065	.090
MAX RUB (2)	-.018	-.008	-.024
WORST FLATNESS	.009 @ 1040 SEC	.004	.007 @ 1130 SEC @ 2450 SEC

(+ INDICATES CLEARANCE)
(- INDICATES RUB)

(1) MAX OPEN: ROTOR @ GROUND IDLE
STATOR @ H.D.T.O.

(2) MAX RUB: HOT ROTOR REBURST: ROTOR @ H.D.T.O.
STATOR @ GROUND IDLE

Figure 2.5-24. LPT Stage 1 Nozzle Inner Seals.

flange. Another important benefit of this bolting arrangement is the small envelope required by the bolt head versus the large envelope required by a nut and protruding bolt threads with its large tolerance buildup. The smaller head envelope allows the front tube of the cooling manifold to be positioned over the front shroud support hook without compromising the impingement distance of the cooling air.

Outer Transition Duct Support

The outer duct support design is basically the same as last reported. Provisions were made on the drawing for 14 temperature and pressure probes for ICLS testing which mount on the HPT casing and pierce the outer duct support. A heat shield was added to the aft support arm to keep the arm from being overcooled by fifth stage air during decel and thus minimize separation from the HPT casing (Figure 2.5-22).

Forward Inner Seal Support

The forward inner seal support design has not changed since last reported, except for a slight change in the tangential load stop design where cast lugs on the inner duct were replaced by pressed-in pins on the support (Figure 2.5-23).

Aft Inner Seal Support

The aft inner seal support design was modified slightly to facilitate fabrication, and the new configuration is shown in Figure 2.5-23. Basic changes to the support include changing from EB-welded joints and brazed joints to TIG-welded joints, changing the support material over the forward most honeycomb seal from sheet metal to a rolled ring and changing the tangential load stops from lugs to pressed-in pins. A gang channel nut plate, which also serves as a windage shield, was added.

Radial clearances between the static seals and the rotating seals were calculated at several mission points and are shown in Figure 2.5-24.

Stage 1 LPT Nozzle

The Stage 1 nozzle is basically the same as last reported. Minor changes include a 0.070-inch forward shift in the aft nozzle attachment to accommodate a shim between the HPT and LPT casings, changing from cast lugs to pressed-in pins on the inner shroud for tangential load stops, and the elimination of the vane leading edge type T_{49} probe for ICLS (Figures 2.5-22 and 2.5-23).

The material specified for the FPS design is René 125. Material selection was based on creep and rupture requirements of the FPS design. However, for the ICLS engine, the material was changed to René 77 to reduce cost. The lower creep and rupture strengths of René 77 are still adequate for the relatively short ICLS test program.

Inner and Outer Transition Ducts

The inner transition duct design was changed slightly since last reported to incorporate a slot in the aft rail which accommodates the tangential load stop pin in the forward inner seal support (Figure 2.5-23).

The outer transition duct design has not changed except to accommodate the 14 temperature and pressure probes which pierce the ducts as well as the outer duct support for ICLS testing.

Stage 2 through 5 Nozzles

The forward hook on all of the Stage 2 through 5 nozzles has been beefed up to reduce steady state stresses. This decision was based on experienced gained from a production engine of similar design. Also, the inner band interlock design, which was originally on the E^3 Stage 2 and 3 nozzles and which has been added to the Stage 4 and 5 nozzles, is expected to provide sufficient damping to preclude vibration problems with the nozzle segments.

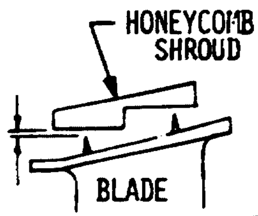
LPT Rotor/Stator Radial Clearances

The LPT radial clearances have been initially set using a 0.015-inch radial rub at the rotating seals and a 0.012-inch radial rub at the blade tip shrouds during maximum climb with nominal cooling as the design point. Using rotor centrifugal and thermal growths as well as stator thermal growths, cold clearances were calculated. Maximum takeoff, worst open, and worst cold clearances were also calculated and are shown in Figures 2.5-25 and 2.5-26. Included in the tables are the summations of asymmetric deflections with symmetric clearances or rubs. These values do not consider the cumulative effect of a severe rub on subsequent flight conditions.

Drawings Issued

The following is a list of all LPT stator detail drawings issued to date:

Drawing Number	Item
4013267-261	Stage 1 Nozzle Casting
4013267-262	Stage 2 Nozzle Casting (Hollow)
4013205-792	Stage 2 Nozzle Casting (Solid)
4013267-263	Stage 3 Nozzle Casting (Hollow)
4013205-793	Stage 3 Nozzle Casting (Solid)
4013267-264	Stage 4 Nozzle Casting
4013267-265	Stage 5 Nozzle Casting
4013205-909	Outer Duct Casting
4013205-929	Inner Duct Casting
4013267-281	Stage 1 Nozzle Machining
4013267-282	Stage 2 Nozzle Machining
4013267-283	Stage 3 Nozzle Machining
4013267-284	Stage 4 Nozzle Machining
4013267-285	Stage 5 Nozzle Machining
4013205-911	Outer Duct Support Machining
4013205-948	Fwd Inner Seal Support Machining
4013205-949	Aft Inner Seal Support Fabrication
4013205-910	Outer Duct Machining
4013205-930	Inner Duct Machining
4013205-972	Gang Channel Nut Assembly
4013205-950	Air Tube
4013205-951	Windage Shield
4013296-018	Heat Shield
4013296-019	Strip Seal
4013296-021	Strip Seal - Stages 2, 3, & 4
4013296-022	Strip Seal - Stage 5
4013296-023	Seal, Nozzle Segment
4013296-169	Borescope Plug
4013296-204	Borescope Cap
4013296-168	Borescope Plug Assembly
4013296-102P02	Spacer, Sleeve (0.625 Long)
4013296-102P03	Spacer, Sleeve (0.100 Thick)
J980P05*	Nut, Self Locking 12 Pt. (Waspaloy Silver Plate)



STAGE	COLD ASSY	MAX T/O (SYMM/ASYMM)	MAX CLIMB (SYMM/ASYMM)	WORST OPEN (SYMMETRIC DEFLECTIONS ONLY)	WORST RUB* (SYMMETRIC DEFLECTIONS ONLY)
1	.017	-.006/-.014	-.012/-.020	.069	-.069
2	.024	.003/-.011	-.012/-.014	.081	-.069
3	.021	-.014/-.008	-.012/-.011	.082	-.080
4	.037	-.007/-.016	-.012/-.021	.056	-.095
5	.053	-.022/-.024	-.012/-.033	.053	-.081

WORST RUB: MAX AFT, WHERE: ROTOR IS AT G. IDLE, STATOR IS COLD

WORST OPEN: MAX FWD (EXCEPT STG 5), WHERE: ROTOR IS AT G. IDLE
STATOR IS AT HDTO
STG 5 ONLY: WORST OPEN IS COLD ASSY

(TARGET CLEARANCE IS -.012" AT MAX CLIMB WITH NOMINAL COOLING)

*HONEYCOMB EFFECTIVE THK. = .150" (.200" ORIGINAL MINUS .050" OVERBRAZE)

(+ Indicates Clearance)
(- Indicates Rub)

Figure 2.5-25. LPT Blade Tip Clearances.

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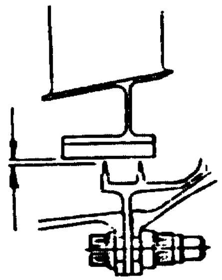

STAGE	COLD ASSY	MAX T/O (SYMM/ASYMM)	MAX CLIMB (SYMM/ASYMM)	WORST OPEN (SYMMETRIC DEFLECTIONS ONLY)	WORST RUB (SYMMETRIC DEFLECTIONS ONLY)
2	.028	-.030/-.013	-.015/-.017	.049	-.047
3	.023	-.032/-.010	-.015/-.012	.048	-.044
4	.027	-.029/-.012	-.015/-.016	.037	-.039
5	.044	-.035/-.020	-.015/-.027	.044	-.044

WORST RUB: STAGE 2 & 3: HOT ROTOR REBURST (ROTOR @ HDTO, STATOR @ IDLE)
STAGE 4 & 5: "MAX AFT" CASE (ROTOR @ GROUND IDLE, STATOR IS COLD)

WORST OPEN: STAGES 2, 3 & 4 "MAX FWD" CASE (ROTOR @ GROUND IDLE, STATOR @ HDTO)
STAGE 5: COLD ASSEMBLY

(+ Indicates Clearance)
(- Indicates Rub)

(TARGET CLEARANCE IS -.015" @ MAX CLIMB WITH NOMINAL COOLING)

HONEYCOMB EFFECTIVE THK. = .150" (.200" ORIGINAL MINUS .050" OVERBRAZE)

Figure 2.5-26. LPT Interstage Rotor Seal Clearances.

Drawing Number	Item
4013205-984G01	Manifold, Air - LPT Stator, Forward
4013205-984G02	Manifold, Air - LPT Stator, Forward
4013205-984G03	Manifold, Air - LPT Stator, Forward
4013205-985G01	Manifold, Air - LPT Stator, Aft
4013205-998P01	Bracket, Tube - Support
4013205-999P01	Bracket, Support - Manifold
4013296-003P01	Support, Tube - Manifold, LPT Stator
4013296-004P01	Plate, Block Off
4013296-004P02	Plate, Block Off
4013296-062P01	Bracket, Manifold
9108M97P01	Washer, Spring - Tension
4013296-063P01	Pin, Headed
4013295-088P01	Bushing
4013296-089P01	Bushing
4013296-102P01	Spacer, Sleeve (0.265)
4013296-102P02	Spacer, Sleeve (0.625)
4013296-102P03	Spacer, Sleeve (Core Use) (0.100)
4013296-103P01	Clip, Spring
R356P08A*	Clamp-Loop, Cushioned, Teflon Impregnated Asbestos (Type Z)
J626P04*	Nut - 10 Point (A286 - Dry Lub)
J979P04*	Nut - 12 Point (Silver Plt - Waspaloy)
J644P07A*	Bolt - 12 Point (In 718) (0.250 Ø X 0.625)
J644P09A*	Bolt - 12 Point (In 718) (0.250 Ø X 0.750)
4013296-153P01	Insulation, Nozzle Stage 2
4013296-153P02	Insulation, Nozzle Stage 3
4013296-153P03	Insulation, Nozzle Stage 4
4013296-153P04	Insulation, Nozzle Stage 5
4013296-099P01	Insulation, Shroud - Forward, Stage 1
4013296-098P01	Insulation, Aft - Stage 1 Shroud
4013296-084P01	Insulation, Shroud, Stage 2
4013296-084P02	Insulation, Shroud, Stage 3
4013296-084P03	Insulation, Shroud, Stage 4
4013296-086P01	Insulation, Shroud, Stage 5
4013296-081G01	Bolt, Slab Head (Inco 718)
4013296-081G02	Bolt, Slab Head (Inco 718; Shoulder)
4013296-102P01	Spacer, Sleeve (0.265 Long)
4013205-966G01	Case, LPT Stator
4013267-065P03	Case, Forward - Low Pressure Turbine
4013267-068P03	Case, Aft - Low Pressure Turbine
4013205-931G01	Shroud, LPT Stator - Stage 1
4013205-932G01	Shroud, LPT Stator Stage 2
4013205-933G01	Shroud, LPT Stator Stage 3
4013205-934G01	Shroud, LPT Stator Stage 4
4013205-935G01	Shroud, LPT Stator Stage 5
4013296-152P01	Insulation, Duct Outer
4013296-152P02	Insulation, Duct Outer
4013296-152P03	Insulation, Duct Outer
4013296-087P01	Insulation, Duct Inner
4013296-164P01	Insulation, Nozzle Stage 1

Work Planned

- Complete Detailed Design Report
- Review ICLS operating conditions and the resulting effects on life and clearances, both axial and radial.

2.5.3.4 LPT Hardware and Test Support

Technical Progress

Cost estimates have been obtained for LPT blade vibration tests and hot fatigue tests. The vibration tests will consist of (1) obtaining nodal patterns and modes of frequency, (2) obtaining stress distributions, and (3) obtaining end effect influences on Stage 1 only. The hot fatigue tests will consist of staircase fatigue tests in the first flexural mode.

Work Planned

- Monitor blade bench test work through design initiation and procurement of test fixtures
- Provide engineering support to vendors
- Complete nozzle bench test planning and issue Test Project Sheets
- Incorporate vendor requested changes into the detail drawings
- Complete assembly drawings
- Initiate instrumentation rework drawings.

2.5.6.1 LPT Rotor (Bench Blades and Tooling)

Technical Progress

The blade casting tooling design is complete on an five blade stages and the tooling vendor has been released to build.

Partial quotes have been obtained for machining the blades (less the required tip shroud interlock hard coat). The possibility of thermal spraying Triballoy T800 on all five stages of blades is being considered. A local vendor capable of performing this operation has been found.

Work Planned

- Place orders for blade machinings.
- Complete blade casting tooling and cast initial parts in each stage.

2.5.6.2 LPT Stage 1 Nozzle

Technical Progress

The Stage 1 nozzle castings are on order. Price as quoted is slightly over plan and the delivery date is later than required. Recent efforts to reduce the cycle time have been fruitful, and close cooperation will continue with the vendor to deliver castings as close to the required date as possible. The nozzle machining drawing has been released to Development Machining Operation (DMO) for planning along with initial funding.

Work Planned

- Complete casting tooling.

2.5.6.3 LPT Stage 2 through 5 Nozzle Fabrication

Technical Progress

The LPT Stage 2 through 5 nozzle castings have been placed on order. As with the Stage 1 nozzle, the Stage 2 through 5 nozzle cost quotes are over plan, and delivery dates are later than required. Visits have been made to tooling vendors, efforts to reduce casting costs and pull in dates have been successful, and we will continue to work with the vendor to make additional cuts in cycle time. Several vendor requested changes have been adopted, including making datum pad integral with gusset walls and moving inspection as well as cutoff points away from wax weld lines. Several more changes are now being reviewed.

The Stage 2 through 5 nozzle machining drawings have been released for quotations.

Work Planned

- Place nozzle machinings on order
- Work with the vendors to reduce delivery time
- Review and incorporate additional casting changes.

2.6 TURBINE FRAME AND MIXER

Overall Objectives

While the E^3 design is based on previous experience with proven engine frames to ensure a long life and a maintainable structure, the E^3 frame will be specifically designed with improvements to meet the FPS requirements.

Performance improvements in engines with confluent exhaust nozzles can be achieved by forced mixing of the core and fan flows. A convoluted mixer is planned for the FPS which effectively mixes the hot, high-velocity, core gas with the relatively lower velocity fan air to produce a more unifrom velocity at the nozzle throat and improved thermodynamic cycle efficiency. The FPS mixer effectiveness goal is 0.75 at Mach = 0.8, altitude = 35,000 feet, standard day, maximum-cruise power setting.

Development Approach

Current technology related to the aerodynamic design of high-bypass mixers is not fully developed. Excellent computer techniques are available which allow good aerodynamic flowpath design in terms of low pressure losses with no separation. However, adequate design criteria that provide guidance for selecting a mixer with high mixing effectiveness and low pressure loss does not exist. Thus the first step in the development of a high performance mixer for the E^3 will be to establish a data base from which an advanced technology, high performance mixer, may be designed.

A scale-model mixer parametric test was conducted early in the E^3 program. The models, 12% scale, were tested in a static thrust stand at both cold flow and simulated hot flow conditions. Selected mixer geometric parameters were systematically varied and tested in order to identify those parameters which significantly impact mixing effectiveness. The mixer models selected were consistent with the E^3 thermodynamic cycle and were within practical mechanical and installation constraints. Results of this test identified initial mixing effectiveness and pressure loss design criteria.

Following the parametric test, five of the scale models were evaluated for noise characteristics. Results of the acoustic tests indicated a two-to-four PNdB noise reduction relative to the separate flow exhaust nozzle. Additionally, no discernible difference in noise level for the various mixers tested was observed. Thus noise generation did not effect the selection of the final mixer design.

Results of the mixer parametric test have been used to design and fabricate scale models for a follow-on mixer performance test. This test was aimed at evaluating overall mixer/exhaust system variables with more emphasis on the total E^3 exhaust system. The follow-on test was added to the original program and was intended to provide the design information necessary to achieve an additional 10% mixing effectiveness (75%) relative to the original program goals (65%). Results of the follow-on tests identified significant exhaust system performance characteristics leading to performance improvements and also pointed out the significance of mixer sidewall shape on mixer performance.

Because of the discovery of the significance of mixer sidewall shape on performance in the follow-on test, the verification test was expanded to investigate mixer shape. Additionally, it has been determined that a change to the flowpath in the last several stages of the turbine and turbine frame can improve the mixer performance by an estimated 0.2% sfc at Mach 0.8 maximum cruise. This change was called the flared turbine design, and it was decided that it would be desirable for an FPS design but not timely for the ICLS flowpath. Thus the verification test included a test of a selected best design for the ICLS flowpath based on the previous tests and analytical studies and a mixer shape investigation on the flared turbine FPS flowpath design. Three mixers will be designed and tested for the FPS flowpath, and the best mixer will be selected for simulated thrust reverser testing.

After the successful completion of the mixer verification tests, the detailed mechanical design of the mixer for the ICLS test vehicle was conducted. This design included an acoustic excitation analysis to establish that no acoustic vibration conditions exist which would result in fatigue life less than the design life of the part. The analysis was carried out

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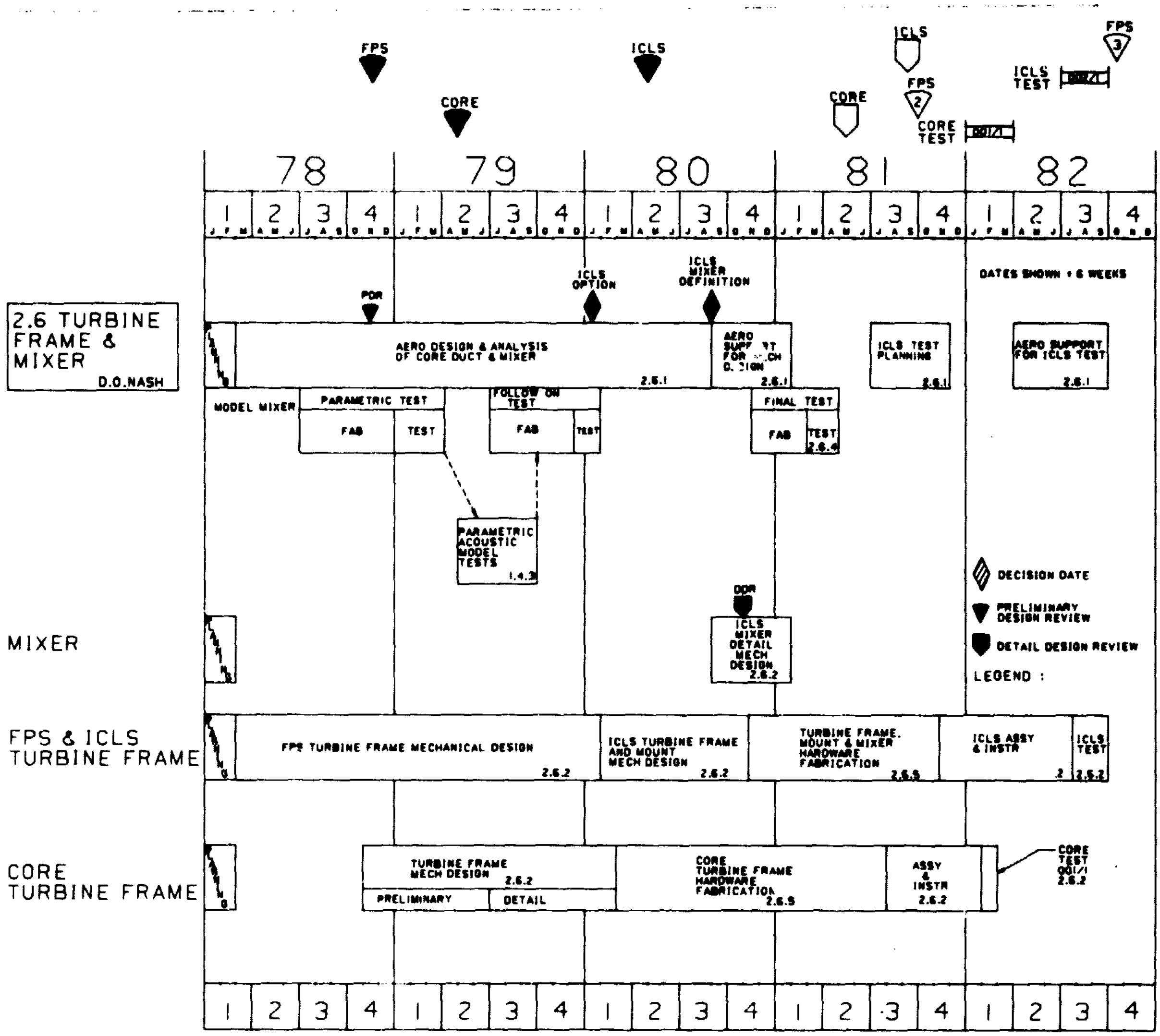

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by determining the elastic and dynamic characteristics of the panel in question by use of the MASS computer program. These results, together with a damping factor based on the type of construction and the predicted or measured acoustic-pressure levels, were entered into RANDEX, a computer program for predicting the response of structures to random excitation. From RANDEX, the expected root mean square (RMS) cyclic panel stress levels were obtained.

The design of the E^3 turbine frame initially involved preliminary layouts and analysis in order to ensure an adequate radial spring constant. The detail design of the turbine frame involved detailed stress analysis under limiting frame load conditions, such as flight maneuver extremes, rotor imbalance, and transient start-up conditions. Analysis of these load and thermal stress conditions was accomplished by use of a three-dimensional, finite-element computer program, Mechanical Analysis of Space Structures (MASS). By using enough nodes in setting up the analytical model, the elastic behavior and stress levels existing under any combination of loading and thermal stress can be determined. Metal temperature distributions, for use in the MASS analysis, are determined by a transient heat transfer analysis computer program.

The NASA Project Manager has approved the detailed design, and the full-scale turbine frame and mixer are being fabricated. After completion of the manufacture, the hardware will be available for engine assembly.

2.6.1 TRF/Mixer Aero Design

Technical Progress

The exhaust nozzle extensions which provide nozzle area variation for the core engine tests were revised due to concern over the ability to estimate the effective area of the previously defined orifice plate nozzles. The new extensions have been changed to conical nozzles and will provide effective areas of nominal, +21%, and -15%.

The nozzle exit survey rakes to determine mixing effectiveness for the ICLS engine tests were defined. The P_T and T_T immersions were scaled directly from the scale model mixer tests with the exception of the innermost element

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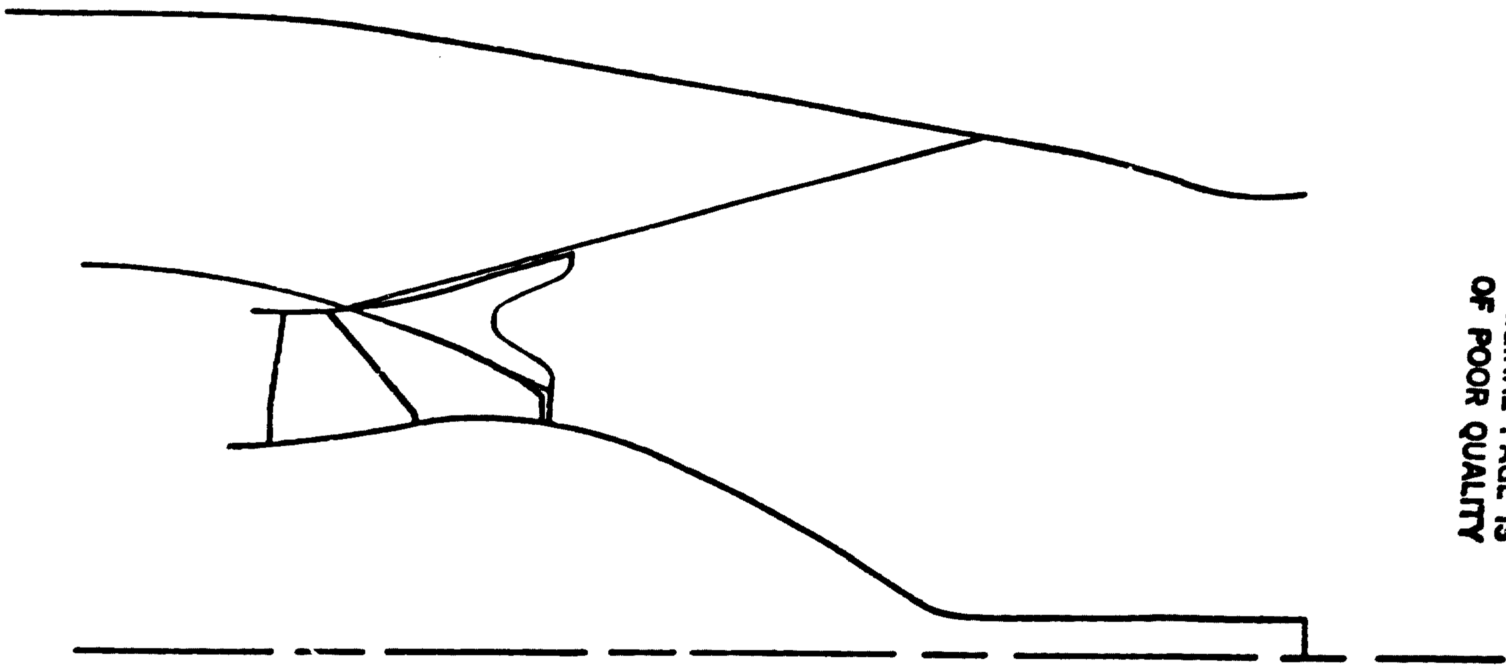
which is set at a 4.0-inch radius to avoid interference with the center vent. A tentative test plan was also defined to provide input for selection of the total number of rakes required (more rakes and less readings versus less rakes and more readings). These rakes will be add-ons to an existing NASA emissions rake mounting system which is capable of remote control variation in an infinite number of circumferential positions.

The ICLS mixer design was selected and finalized based on analysis of the scale model test results. A cross section of the mixer is presented in Figure 2.6-1. The mixer will have 12 lobes with the sidewalls scalloped to promote better mixing. The mixing penetration has been selected to maximize the internal performance tradeoff between mixing effectiveness and mixer pressure loss. Tailpipe length likewise has been selected to maximize total system performance including internal performance, external nacelle drag, and weight. Full scale hot flowpath coordinates were defined for the mixer and exhaust system consistent with the design selection.

The ICLS turbine rear frame/mixer DDR was presented at NASA-Lewis on November 20, 1980 and a follow up for the mixer was presented on January 21, 1981. Background scale model test data and analytical studies leading to the rational for the ICLS design were presented. Design acceptance and authorization to proceed with hardware procurement were given by the NASA E^3 Project Office.

The static pressure which will exist at the base of the ICLS center vent tube in the exhaust nozzle was estimated using the STC program for an SLS maximum power condition. Based on the analysis, in its present location, the static pressure will be 8% above ambient pressure with the nominal nozzle exit area. The tube would have to extend beyond Station 365 to reach the ambient pressure level. Data from scale model tests of similar configurations indicate the base pressure to be 1% to 3% above ambient. The test data is considered to be more accurate than the analytical result, and it is concluded that the base pressure could be 1% to 3% above ambient pressure in its present location. An opportunity exists to measure this pressure on the ICLS exhaust system in the forthcoming mixer test at FluiDyne (WBS 2.6.4) in April. An inquiry will be made to FluiDyne to modify the model to include simulation of the vent and to conduct tests at several SLS points.



- 12 Lobes
- Scalloped, Radial Sidewalls
- 43% Penetration
- 45 Inch Tailpipe Length, $L/D_h = 0.72$
- Integrated with Pylon and Aft Mount

Figure 2.6-1. ICLS Mixer Aero Design.

Work Planned

- The FPS mixer design will be completed following analysis of the Phase III scale model mixer test (WBS 2.6.4).

2.6.2 Turbine Frame and Mixer Mechanical Design and Analysis

Technical Progress

A cross section of the current turbine frame/mixer is shown on Figure 2.6-2. The only significant design changes incorporated into the turbine frame/mixer hardware during the subject reporting period were to accommodate test instrumentation. The portion of the aft inner core cowl situated beneath the pylon was made into a separate piece by means of axial flanges on either side. This allows removal of both sides of the cowl without disturbing the instrumentation leads or the clearance control air valves which are now mounted on this plate. It also permits removal of the cowl without removal of the pylon fairing. The portion of the mixer shroud located under the pylon has also been designed as a separate piece which gives us additional room for instrumentation lead out and simplifies removal of the mixer, since the mount link covers and the pylon fairing can remain in place. In addition, a service strut has been added aft of the mixer to lead out the aft slipring instrumentation and services.

The detailed Transient Heat Transfer (THT) analysis has been completed. A cross section of the three-dimensional computer model used for this analysis is shown in Figure 2.6-3. The worst internal temperature differentials, and thus the worst thermal stresses, were found to occur 60 seconds after the engine's initial acceleration to takeoff power, following a 200-second warm up. A plot of this temperature response is shown in Figure 2.6-4. These transient temperatures were input into the detailed structural computer model of the frame (MASS) and combined with the maximum loads that might be experienced during normal operations. The resultant stresses are illustrated in Figure 2.6-5. The worst stresses were found to occur in the outer casing polygonal panel between the 12 and 1 o'clock struts. Stress concentration factors were added to these stresses to compensate for the welds and steps

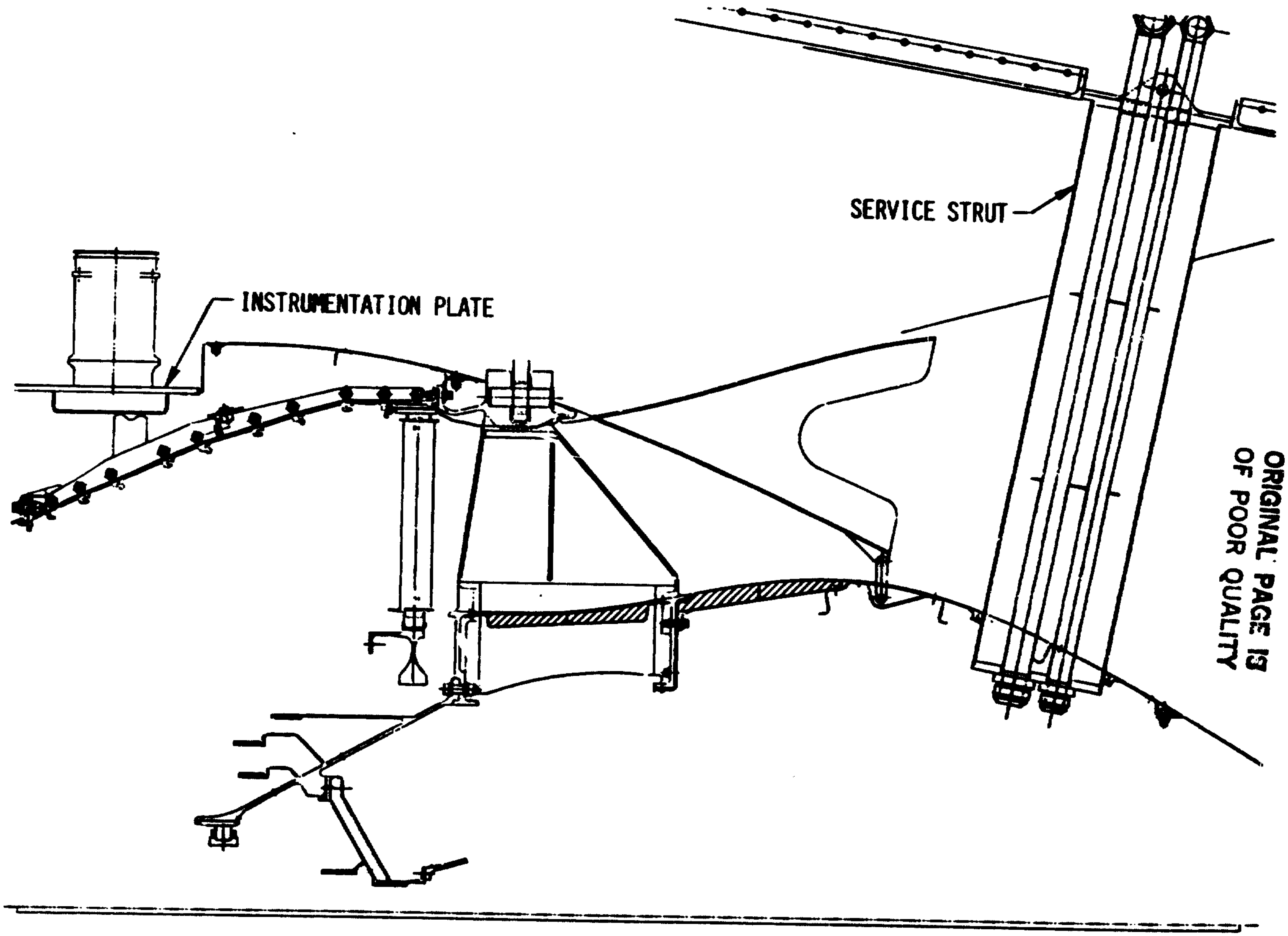


Figure 2.6-2. Turbine Frame and Mixer.

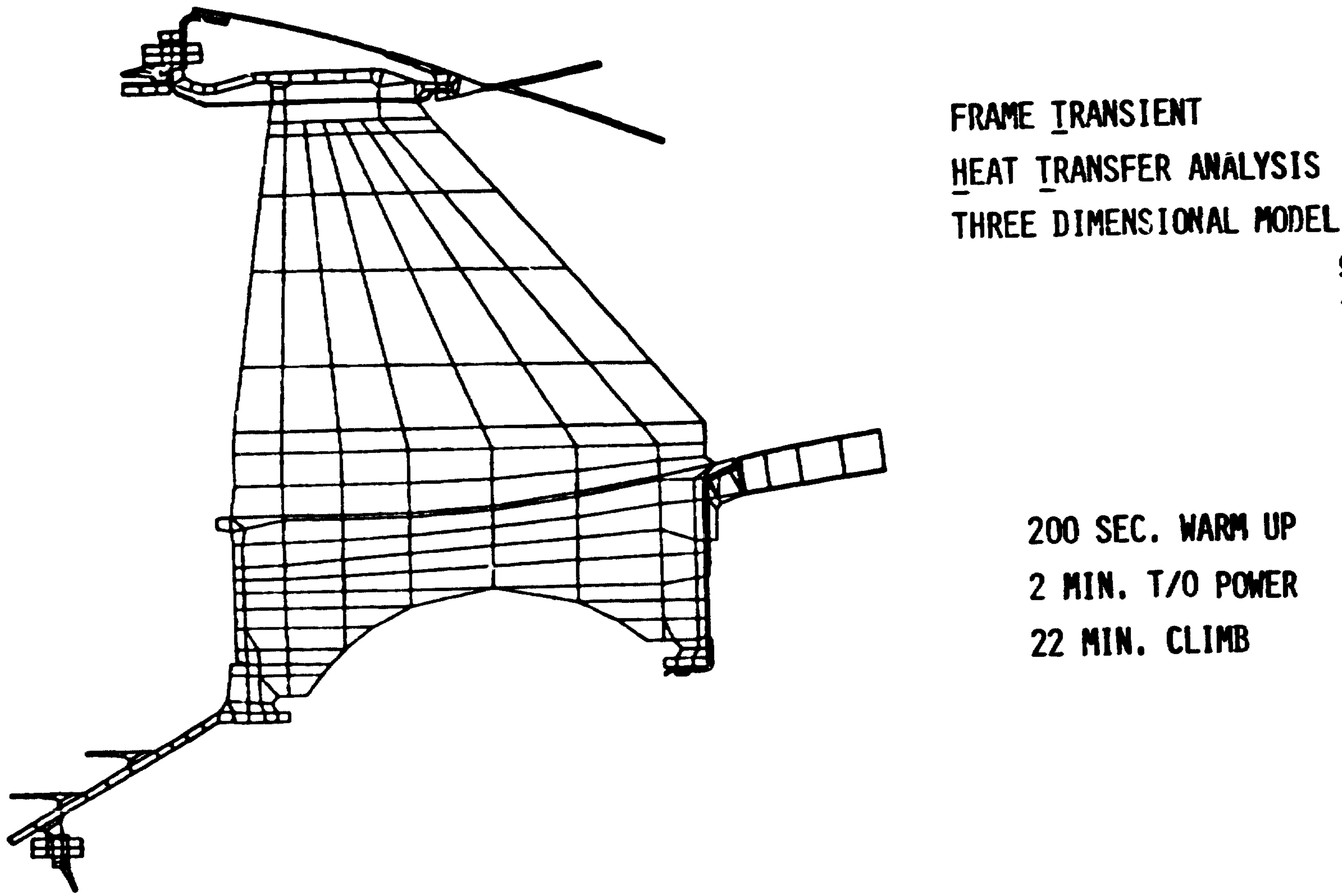


Figure 2.6-3. Turbine Frame THT Model.

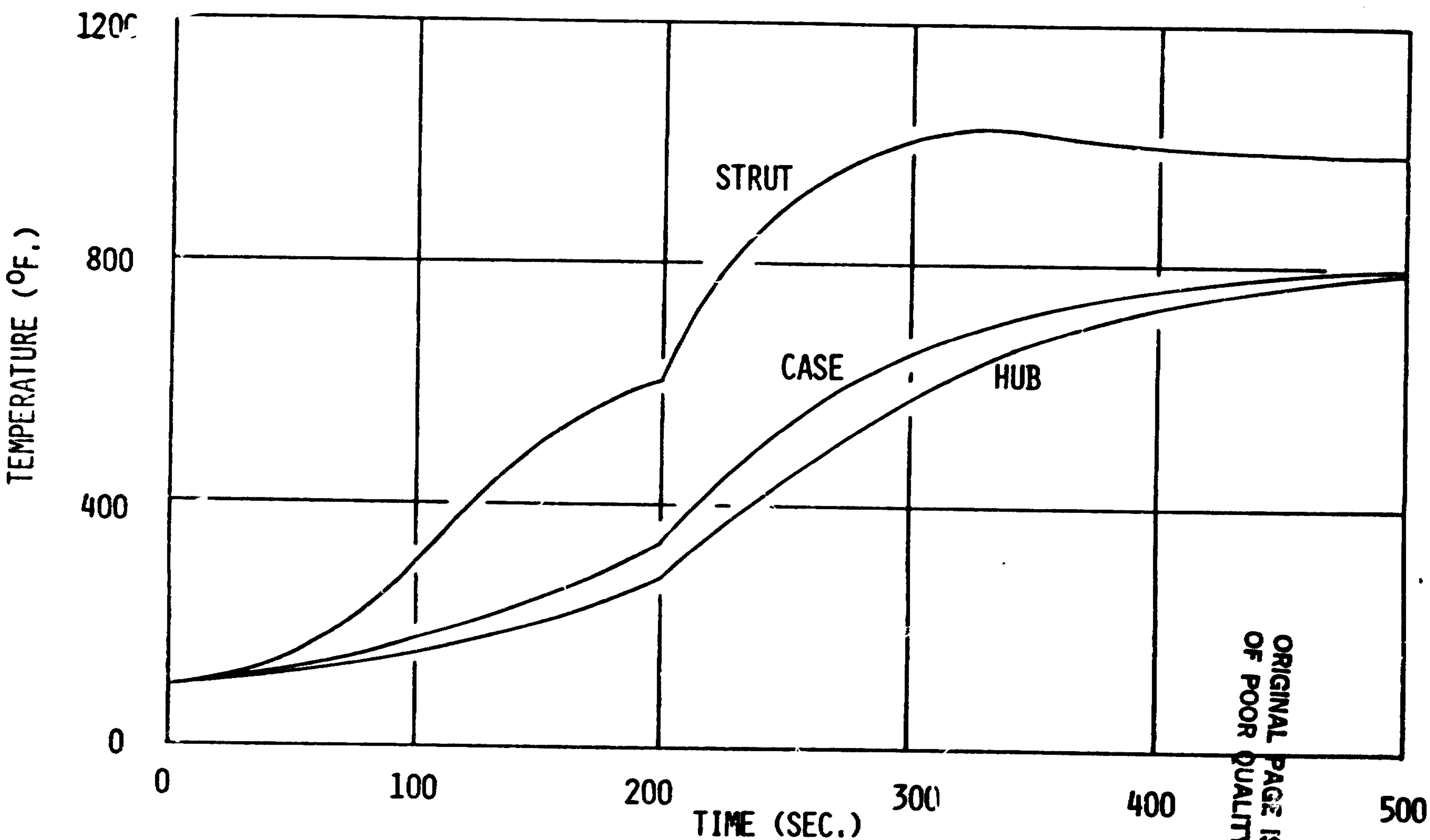


Figure 2.6-4. Turbine Frame Transient Response.

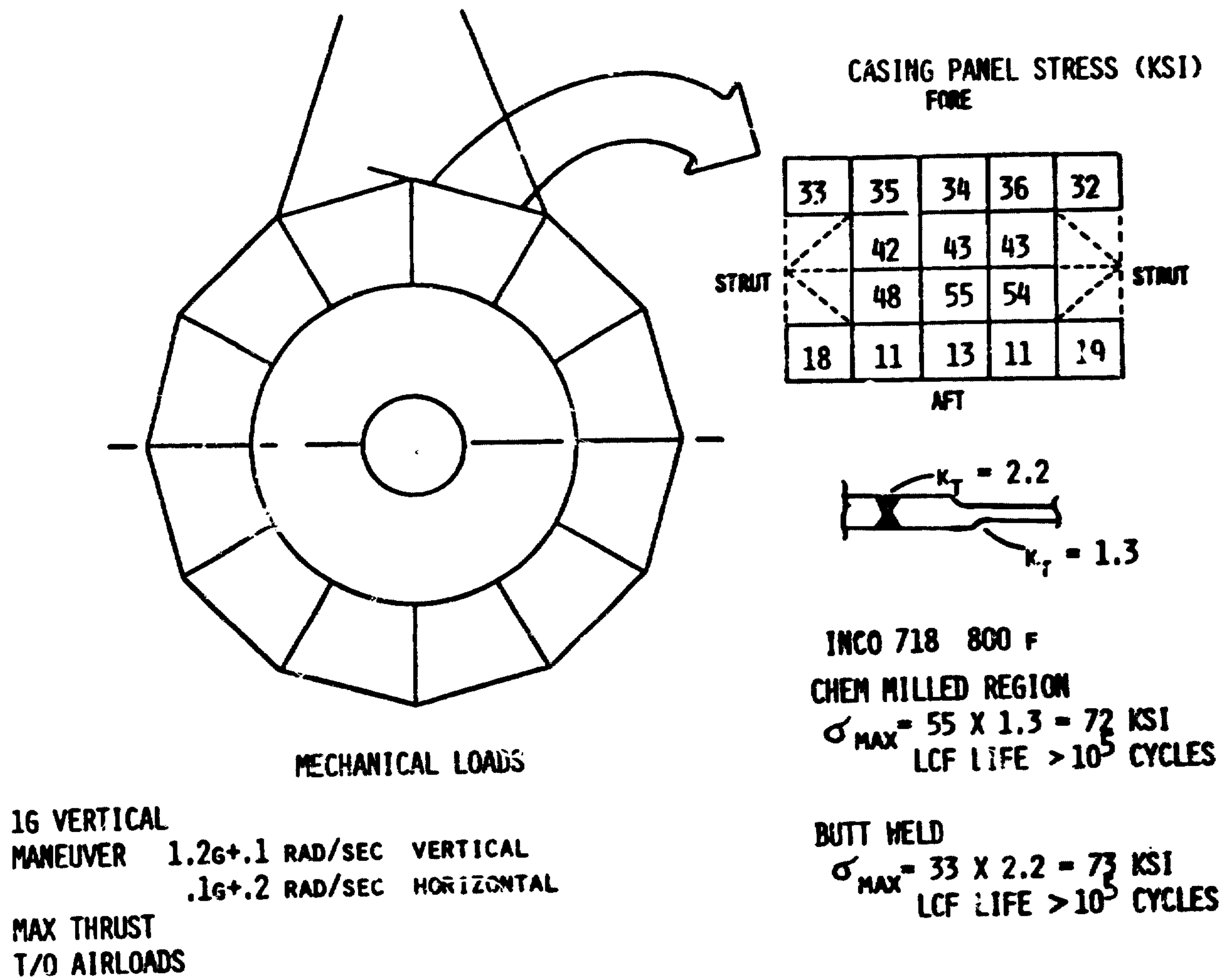


Figure 2.6-5. Casing Combined Thermal and Mechanical Stress.

in the chem-milled outer panel. The resultant worst stresses were 73 ksi at the panel butt weld and 72 ksi in the panel's chem-milled region. The frame's low cycle fatigue life under these stresses is greater than 100,000 cycles.

The two removable ground handling mounts, which, when attached, are bolted to the 2 and 10 o'clock struts, have been stress analyzed. The lugs are designed to withstand a load of 3 g vertical (9600 pounds) and 4 g horizontal. Under these conditions, the lug tearout stress is only 2.4 ksi, while the material's 0.02% yield strength is 116 ksi. The lug is attached to the strut end by two 7/16-inch bolts. The stress in these bolts could be up to 77 ksi, which results in a safety factor of two. When the engine is mounted in the test facility, these mounts are removed, and cover plates are added. Figure 2.6-6 illustrates these ground handling features.

An acoustic fatigue analysis was conducted on the frame heat shield (Figure 2.6-7). The shield is made up of 12 circumferential pieces of corrugated Inco 718. It is captured in front between the frame and low pressure turbine flanges, and in the back at the frame/mixer interface. The acoustic environment was estimated from the sound pressure levels measured for the CF6-50 core nozzle duct outer wall. From this analysis, it was determined the plate natural frequency is 218 hertz, and the acoustic dynamic pressure is Prms = 0.0082 psi $\sqrt{Hz}$. This produces an acoustic stress of 34.8 ksi. For an unlimited high cycle fatigue life, the allowable stress is 40 ksi, yielding a safety factor of 1.2.

The frame struts are used to route rear sump lube tubing outboard of the sump area. There are four tubes for this purpose. A 0.5-inch-diameter oil supply tube is located in the 9 o'clock strut. There are two 0.75-inch-diameter scavenge line tubes, one in the 5 o'clock and one in the 7 o'clock strut, and a 0.375-inch-diameter seal drain tube is located in the 6 o'clock strut. The natural frequency of each of these tubes is listed on Figure 2.6-8. The seal drain tube has the lowest natural frequency of the three, at 328 hertz, and is well above the maximum fan speed of 58 hertz.

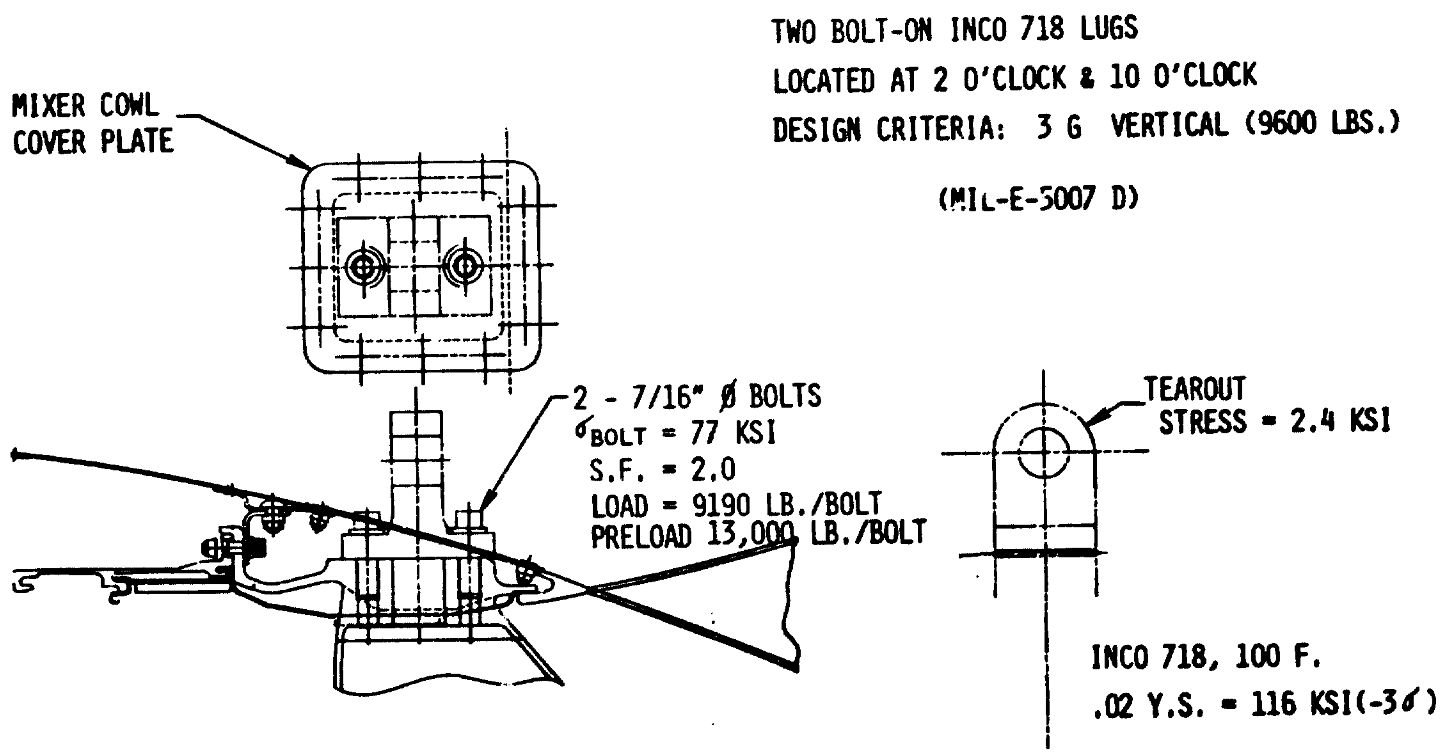
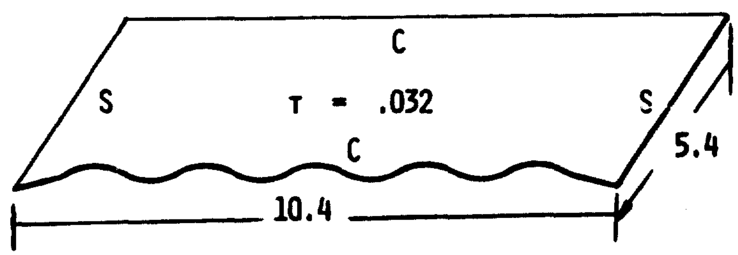


Figure 2.6-6. Ground Handling Features.



$$\sigma^2 = \frac{\pi}{4\zeta} F_N P_R^2 S_o^2 K_T^2$$

ζ = DAMPING RATIO = .02

F_N = NATURAL FREQUENCY = 218 Hz

S_o = PLATE UNIT STRESS = 15.2 KSI

P_R = PRESSURE DENSITY = .0082 PSI/$\sqrt{Hz}$ (149 dB) (CF6-50 DUCT SOUND PRESSURE LEVEL)

K_T = 3.0

$\sigma_{ACOUSTIC}$ = 34.8 KSI

INCO 718, 1000 F.

FATIGUE LIMIT = ~40 KSI

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Figure 2.6-7. Frame Heat Shield Acoustic Stress.

TUBE	O.D.	ω_N
SCAVENGE	.500	448 Hz
SUPPLY	.500	448 Hz
DRAIN	.375	328 Hz

100% FAN SPEED = 58 Hz

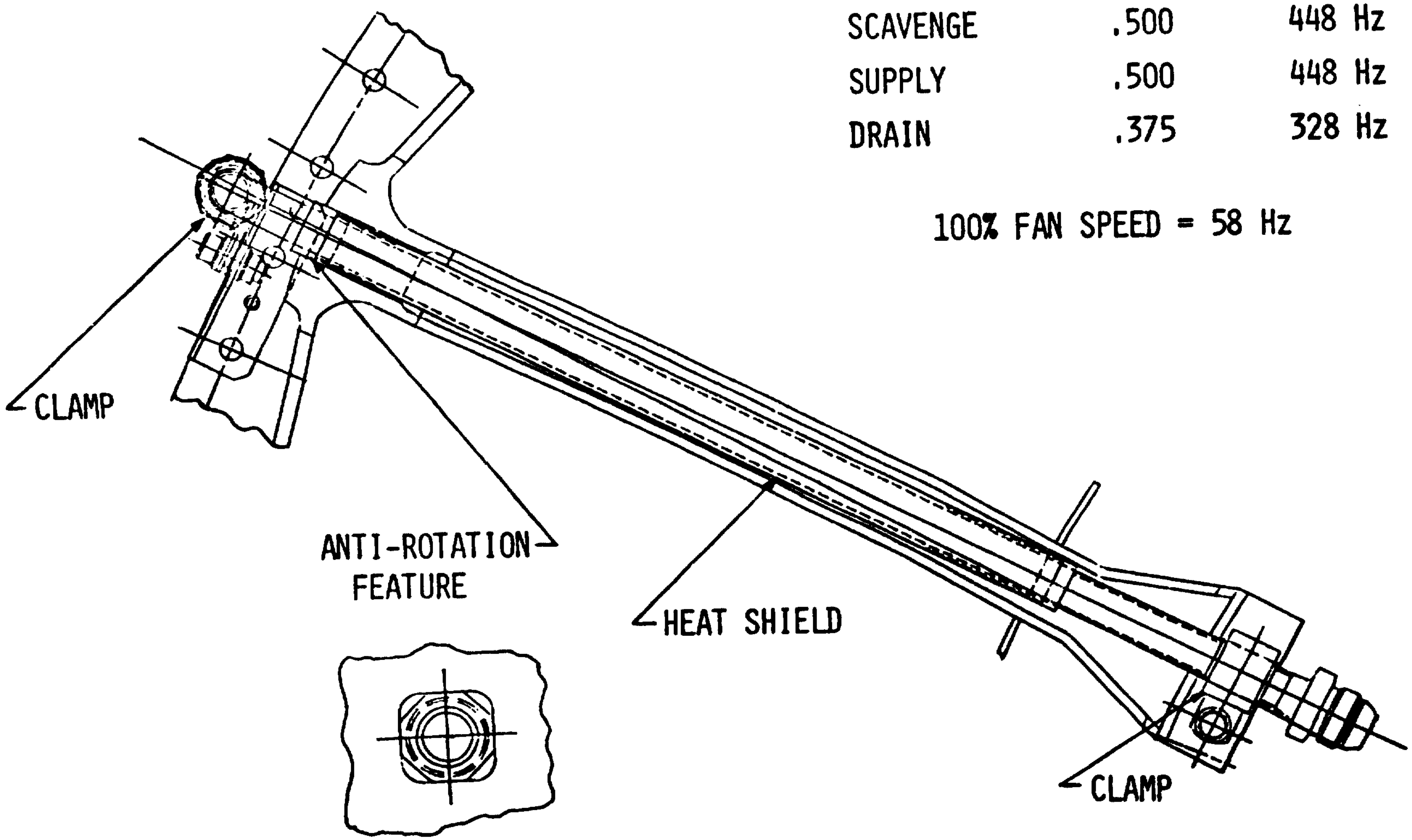

Figure 2.6-8. Rear Sump Lube Tubing.

An analysis was conducted to verify the strength of all the bolt jointed flanges and rabbets in the turbine frame section. The sump cone/frame flange has 48 3/8-inch-diameter bolts. Under ultimate load conditions, the bolts have a safety factor of 1.3 over the 2/3 Johnson's yield strength, and the rabbet safety factor of 1.4 over the 0.2% yield strength. The low pressure turbine/frame flange has 120 0.25-inch-diameter bolts, with a bolt safety factor of 1.4 and a rabbet safety factor of 1.3, also under ultimate load conditions. The centerbody/frame has 48 bolts of 0.25-inch diameter; and under a 10 g load condition, the bolt safety factor is 1.5.

The final ICLS mixer shape has been defined. An acoustic excitation analysis was conducted to establish that no acoustic vibration conditions exist which would result in fatigue life less than the design life. The elastic and dynamic characteristics of the mixer were determined by use of the MASS computer program. A view of the model is shown in Figure 2.6-9. The results show a panel mode natural frequency of 147 hertz, an adjacent chutes in-phase chute flopping mode natural frequency of 73 hertz, and an adjacent chutes out-of-phase chute flopping mode natural frequency of 99 hertz. These results, together with a welded design damping ratio of 0.0076 and the estimated acoustic-pressure levels based on measured levels for the CF6-50 core nozzle, were input into a RANDEX computer program used for predicting the response of structures to random excitation. From RANDEX, the maximum root mean square (RMS) cyclic panel stress was found to be 11.1 ksi. The fatigue limit stress for the 321 stainless steel mixer at these conditions is 30 ksi, which yields a safety factor of 2.6. A Campbell diagram, which shows that all the mixer natural frequencies fall outside the engine operating range, is illustrated in Figure 2.6-10.

The inner portion of the fan flowpath aft of the core cowl is an integral part of the mixer. This piece was analyzed for strength due to an inward loaded pressure differential of 6 psi, and thermal stresses. Results of this analysis led to the addition of a circumferential stiffener six inches aft of the leading edge. The material thickness is 0.04 inch, and the critical buckling pressure is 11.6 psi, giving a safety factor of almost two. The maximum thermal stress occurs where the mixer is attached to the turbine frame by

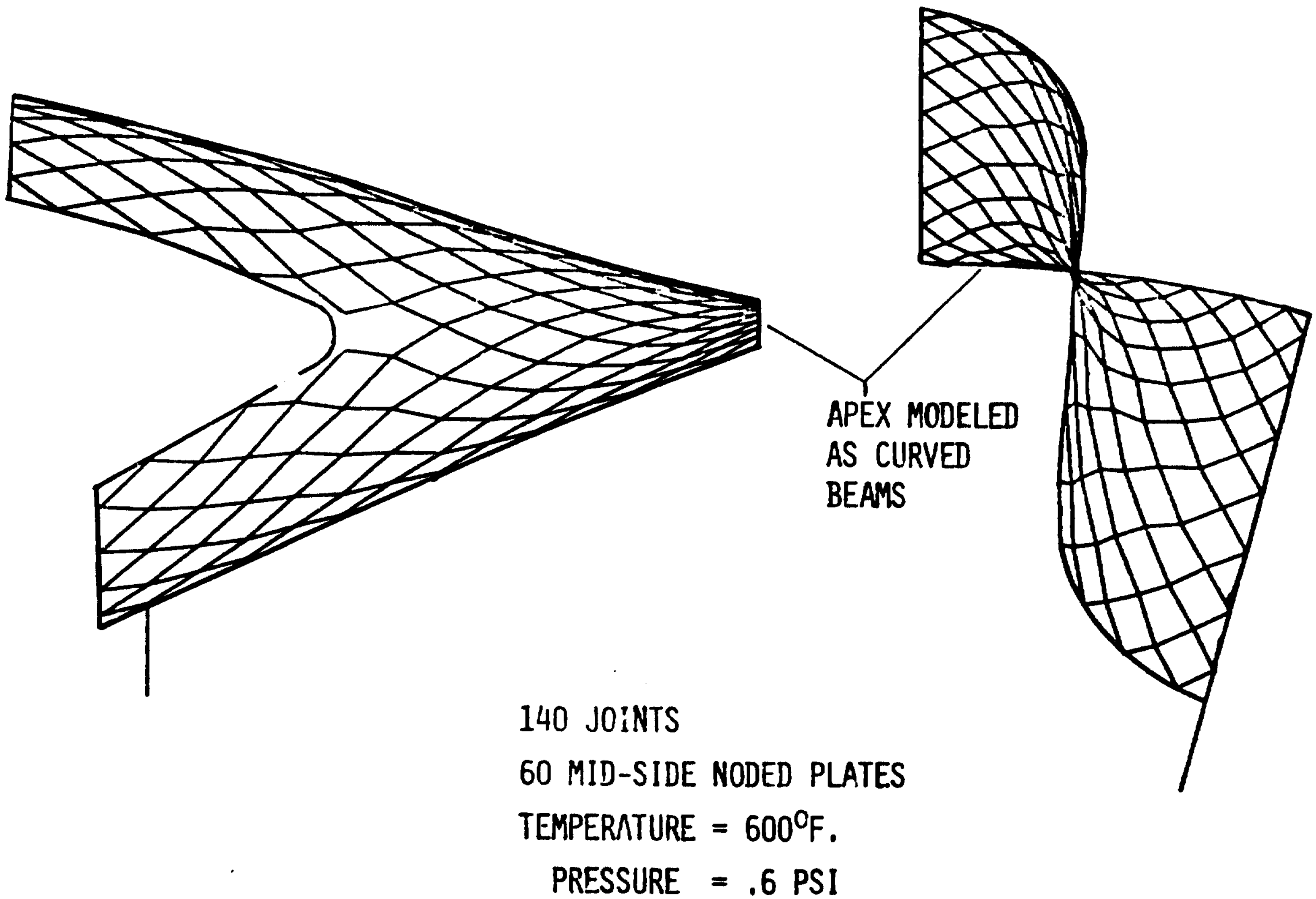

Figure 2.6-9. Mixer Mass Model.

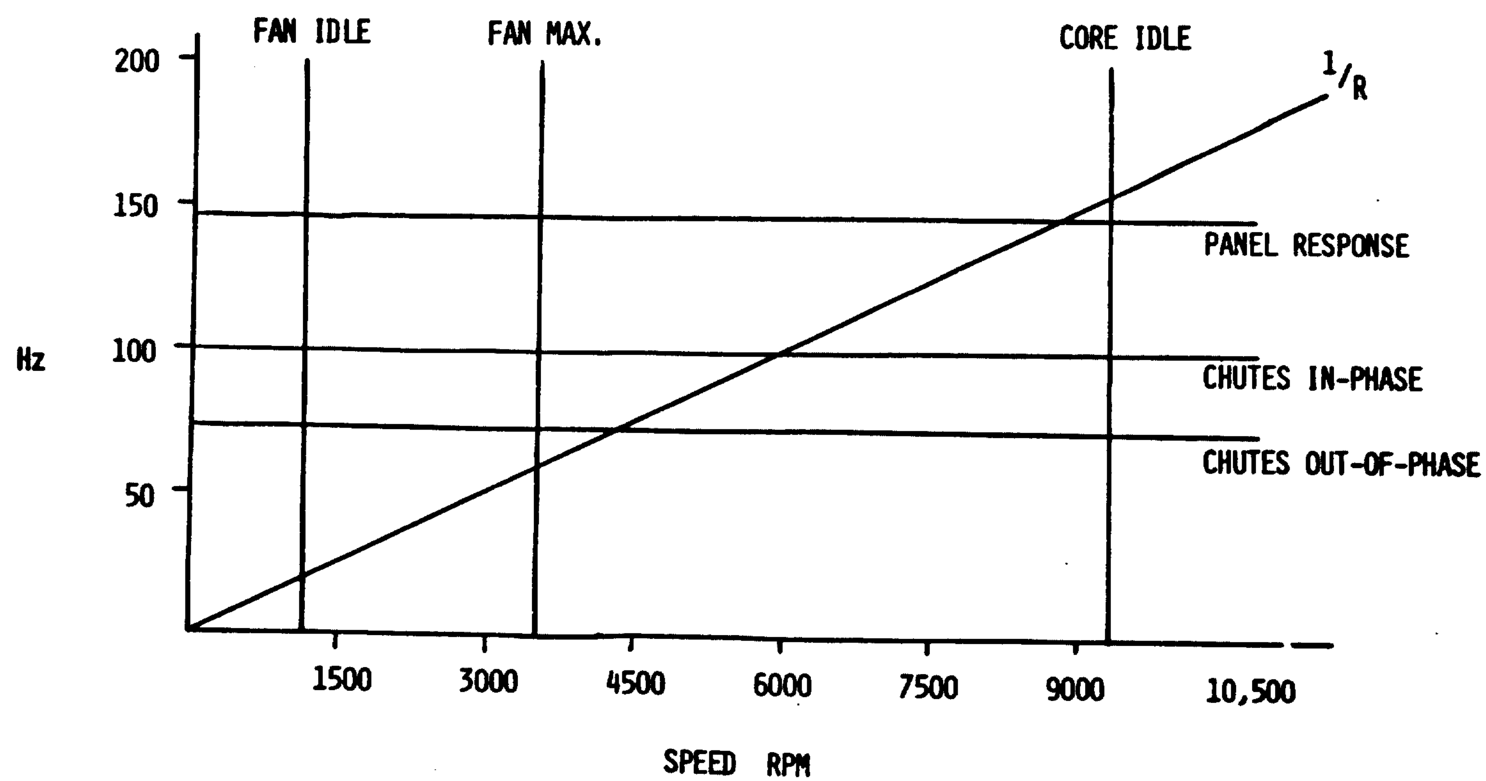

Figure 2.6-10. Mixer Campbell Diagram.

means of 10 local brackets on the low pressure turbine/frame flange. The frame and brackets are hotter than the mixer piece and tend to impose local radial loads on this sheet metal cowl. The resultant stresses were up to 60 ksi, which allows for a low cycle fatigue life on the ICLS design of 8000 cycles.

Mechanical design and analysis of the turbine frame and mixer has been completed. The total weight of the frame, mixer, core cowl, and centerbody, including fasteners, is 653.5 pounds. The weight breakdown is as follows: frame 275.5 pounds, mixer 127.0 pounds, core cowl 164.9 pounds, forward centerbody 53.7 pounds, aft centerbody 10.1 pounds, and fasteners 22.3 pounds.

A detailed design review of the turbine frame was held at NASA-Lewis on November 20, 1980. The mixer detailed design review was held January 21, 1981. The detailed designs of all components were approved by NASA. The detailed drawings for all hardware are complete, and all hardware has been released for manufacture.

Work Planned

- Support manufacturing for the fabrication of the core and ICLS hardware.

2.6.4 Scaled Mixer Performance Testing

Technical Progress

The STC/Spalding analysis to determine the effect of mixer shape on mixing effectiveness was completed. Three mixers with varying mixer shapes were analyzed with the revised STC/Spalding analysis which included circumferential as well as radial streamline analysis. Results indicated some minor improvements in mixing effectiveness but no significant gains. It was decided to extend the analysis to include an evaluation of one of the mixers tested in the Phase II test. Comparison of the predicted exit temperature profile with the measured profile from the model test is shown in Figure 2.6-11. The comparison was not as good as expected and indicated that the STC/Spalding analysis procedure still required improvement before a high degree of confidence in its use is achieved.

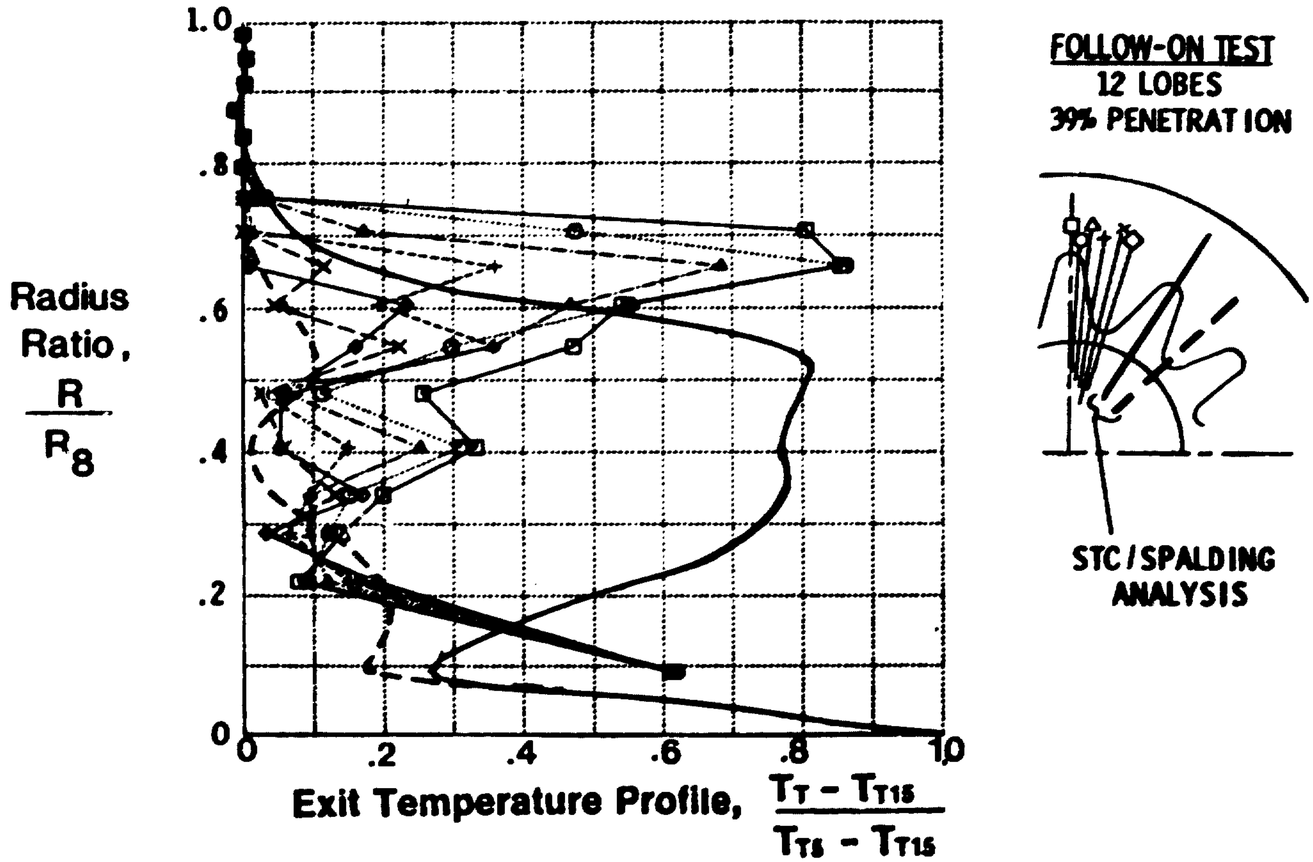

Figure 2.6-11. Predicted Versus Measured Nozzle Exit Temperature Profiles.

As a result of the inability to obtain a clear indication of the types of mixers to be evaluated in the third mixer test from the Spalding analysis, a selection of test configurations was made with NASA concurrence based on the analysis and interpretation of the previous test results. The matrix includes the ICLS mixer/flowpath, three mixers for the FPS flared turbine flowpath, a centerbody variation, the C-D nozzle simulation, and reverse thrust simulation. The three FPS mixers include a baseline mixer similar to the ICLS design, an 18-lobe mixer, and a large mixer with a nonsymmetric cutback which will be defined just prior to test. A revised test specification was sent to FluiDyne for cost quotation. FluiDyne requested that mylars be provided of the mixer cross sections to facilitate model fabrication; these were provided in early January. Preparation of the Test and Instrumentation Plan for NASA has been completed.

FluiDyne has slipped the Phase III mixer test to the end of March, 3 weeks past their quoted start date. This represents a total slip of essentially 1 month from the initial plan; the test will now occur in April-May. The reason is that a program currently in the FluiDyne Channel 11 hot flow stand is running behind schedule. There is no significant effect of the slip on the E^3 program; the ICLS mixer has already been released for the engine test, and the FPS design can be updated at any time.

Work Planned

- Conduct the Phase III scale model mixer test and begin final data analysis and formal report preparation.

2.6.5 Turbine Frame/Mixer Fabrication

Technical Progress

The detailed drawings for all the hardware have been released to the shop. Approximately 15% of the necessary material is in-house. The long lead time machining work has been placed.

Core Turbine Frame

The core engine slave turbine frame and related exhaust system hardware are shown in Figure 2.6-12. Hardware fabrication is 75% complete. Most of the minor parts have completed manufacture, the frame hub and outer case have been formed, and the slots for the struts have been cut in both the hub and outer case. The struts have also completed manufacture.

Work Planned

- Continue planning, material procurement, and fabrication of both ICLS and core engine hardware.

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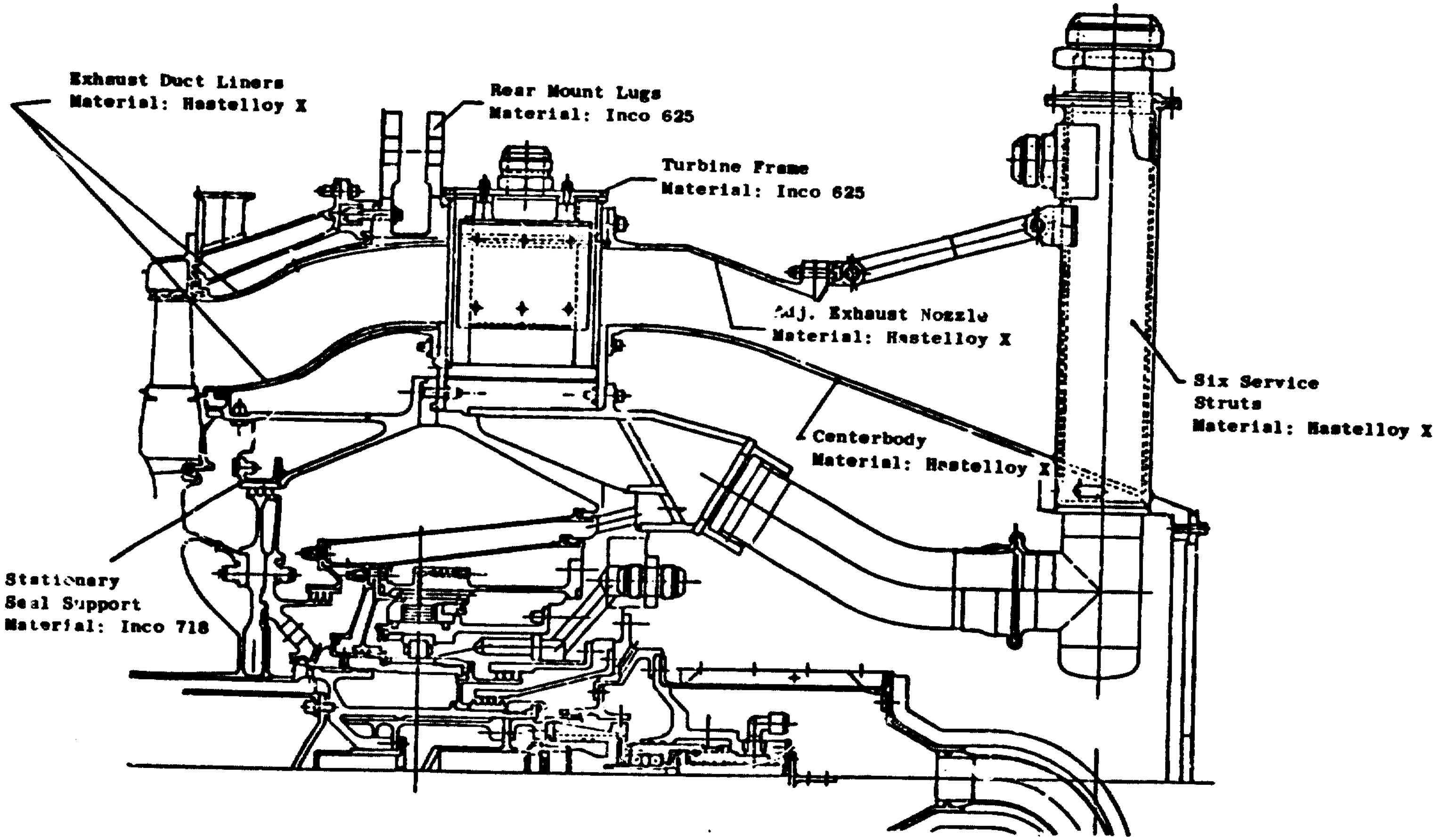

Figure 2.6-12. Core Engine Exhaust System Cross Section.

2.7 BEARINGS, SYSTEMS, DRIVES, AND CONFIGURATIONS

Overall Objectives

Bearings, systems, drives, and configuration components encompass the following:

- Main shaft support bearing and seal components
- Accessory drive system
- Lube system, including rotor thrust balance
- External piping and wiring configuration of the engine.

The main shaft bearings will be designed to properly support the engine rotor systems. These bearings will meet the design requirement for engine life considering fatigue and skidding criteria. They will operate with minimum heat rejection and within specified limitations.

The accessory drive system will provide the means to drive the engine-required accessories. Adequate horsepower capability, the proper pad speeds, and direction of rotation are the primary design objectives. Provision for two starter pads will be provided on the Core/ICLS accessory gearbox. Critical speeds of all gearbox shafting will be kept at least 20% above the engine operating speed. The internal configuration of the accessory gearbox will be designed so that lubricating and cooling oil will be easily scavenged from it. During engine operation, the gearbox will operate within specified temperature limits.

The lube system will be designed to provide a flow network that will deliver and remove specific amounts of oil from various areas of the engine while maintaining predetermined pressure drops in the individual circuits. The rotor thrust balance will be determined as part of the lube system activity, and the thrust load on the fan and core thrust bearings will be established to be compatible with the life requirements of the bearings.

The objective of the configuration design is to provide the required external wiring and piping between various components of the engine. Piping is sized to meet specified flow velocities. The piping and wiring array will be capable of operating in the temperature and vibration environment of the engine.

Development Approach

To achieve the foregoing objectives, a development program has been established that consists of five subprograms defined as follows:

- Forward Sump Mechanical Design (WBS 2.7.1.1)
- Aft Sump Design (WBS 2.7.1.2)
- Lube System Design (WBS 2.7.1.3)
- Accessory Drive System (WAS 2.7.2)
- Configuration (WBS 2.7.3).

The major layout design of the forward and aft sumps was accomplished during 1978 through the third quarter of 1979. Preliminary design work was aimed at obtaining a viable design for the Flight Propulsion System. Studies have been made integrating the sumps with the core (high pressure) and low-spool (low pressure) rotor systems. Bearings and support housing designs are being analyzed in terms of rotor speeds, loads (including blade-out), cost, weight, and maintainability.

The Core PDR was held in the second quarter of 1979, and an Integrated Core/Low Spool (ICLS) IDR was held in the third quarter of 1979. With NASA approval, the detail mechanical design has been initiated with planned completion in the fourth quarter of 1980. The Core bearings and sumps DDR was held in the fourth quarter of 1980. The ICLS DDR is planned for early in the second quarter of 1981. The ICLS bearings and sumps IDR was held in the first quarter of 1980.

Following NASA approval, most of the hardware procurement and fabrication will proceed.

Two bearing tests have been planned to establish lubrication methods, provide heat-rejection data, and establish other design parameters for the core thrust bearing and the intershaft bearing. The thrust-bearing underrace cooling test (WBS 2.7.7.4) will be completed in two phases. The first phase was completed during the second quarter of 1979, and the second phase (which will test an E^3 core thrust bearing) will be completed in the second quarter of 1981. The intershaft bearing test simulating the aft intershaft bearing

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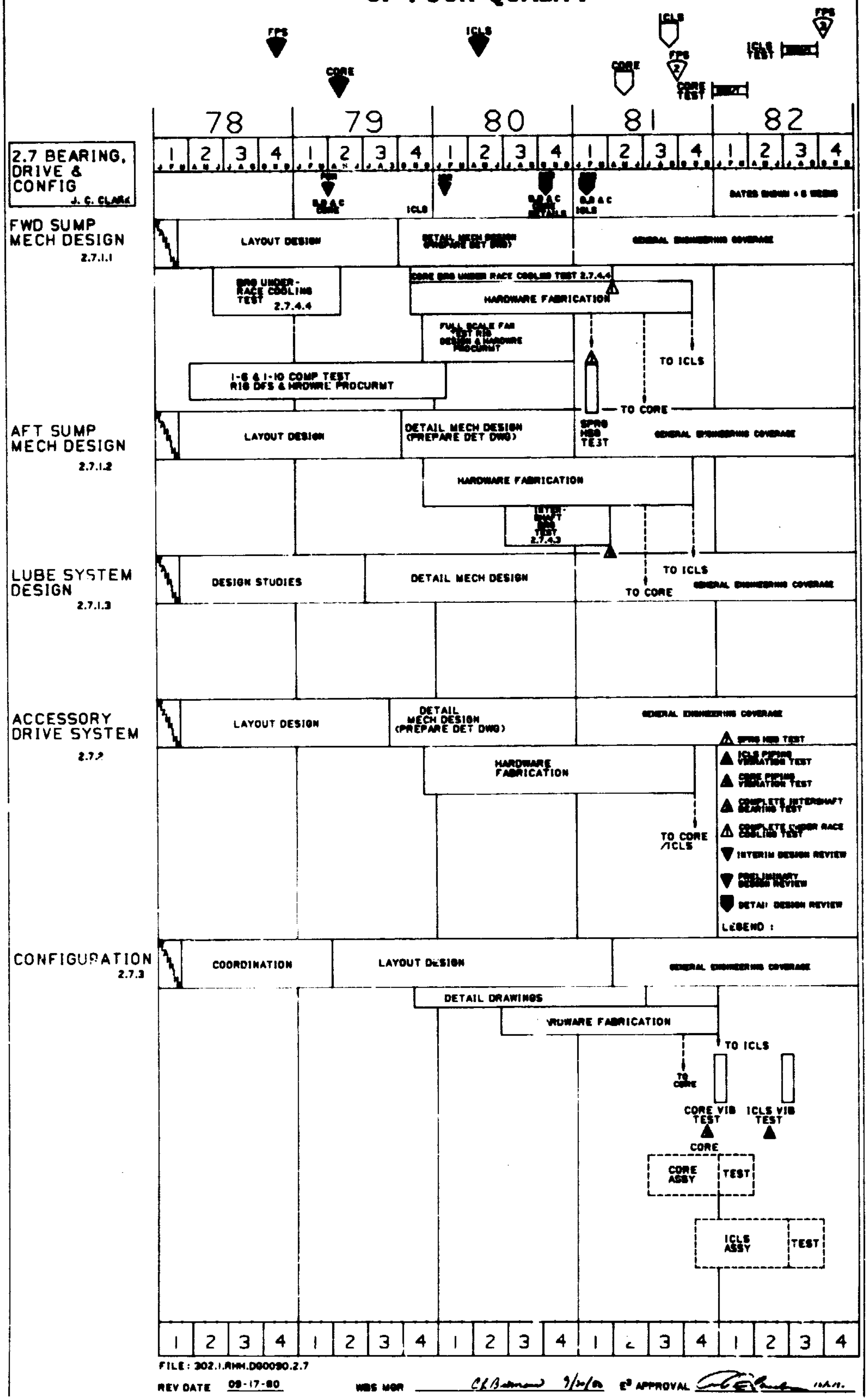

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arrangement of the E^3 design (WBS 2.7.4.3) scheduled to run in the fourth quarter of 1979, was run in the first quarter of 1980. More testing will be needed after rig modifications are made.

The lube system preliminary and detail design efforts will support the mechanical design work in such areas as lube system network, main shaft seal-pressurization networks, and rotor thrust. The PDR for the lube system was scheduled and completed as part of the Core and ICLS PDR's. During the detailed mechanical and design effort, the lube system activity will be concentrating on finalizing such parameters as (1) lube flow and pressures, (2) seal ΔP's at various operating conditions, and (3) the rotor thrust-balance status.

While preliminary mechanical studies are continuing on the sump systems, a parallel work effort will be going on in the accessory drive area. The design effort will be in support of the FPS being designed under the Task I effort. The effort will be centered on the power takeoff (PTO), accessory gearbox (AGB), and the connecting shafting. The PDR for the accessory drive system was scheduled and completed as part of the Core PDR. Detail design work has now been completed for the Core/ICLS engine.

During 1978 and running through the first quarter of 1979, the configuration effort was concerned with coordination with all the interfacing units and design layouts of all external wiring and piping. All detailing will be completed by the second quarter of 1981. Vibration tests were also planned as a supporting effort for the scheduled Core and ICLS tests.

In addition to the aforementioned effort, design and procurement of specific test rig hardware has been provided in support of the 1-6 and 1-10 compressor tests and the full-scale fan test (FSFT).

2.7.1.1 Forward Sump Mechanical Design

Technical Progress

During this reporting period, design effort has been applied in the following areas:

- FSFT forward sump
- Core engine forward sump design
- Forward sump for ICLS engine.

FSFT Forward Sump

The FSFT forward sump configuration is shown in Figure 2.7-1. The design is described in Semiannual Report No. 5 and all hardware is now available for assembly.

Instrumentation leadout methods have been identified and rework drawings have been completed. Test instrumentation requirements have been defined and are summarized as follows:

- Two temperature thermocouples each bearing
- Sump cavity pressure measured one place
- Seal pressurization cavity pressure
- Outer air cavity pressure and flow
- Bearing lube flows and pressures
- Vibration pickups mounted on bearing housings.

Core Forward Sump Design

The present design status of the forward core sump is shown in Figure 2.7-2. The design is basically the same as reported in Semiannual Report No. 4 and detail drawings have been completed and hardware is in the manufacturing cycle.

Minor changes have been made in the rotating labyrinth seal which supports the rotating portion of the slipring. These changes have been made to incorporate an instrumentation duct in the forward end of the compressor rotor. The labyrinth seal locates and traps the instrumentation duct in the compressor rotor and allows instrumentation leadout through the slipring assembly.

Forward Sump for ICLS Engine

The ICLS forward sump is shown in Figure 2.7-3. The configuration incorporates hardware that is comm with the F T and will be tested first during the fan test.

All detail drawings have now been issued for the ICLS forward sump and hardware is in the manufacturing cycle.

The major emphasis during this reporting period has been on the No. 1 bearing area which is shown enlarged in Figure 2.7-4.

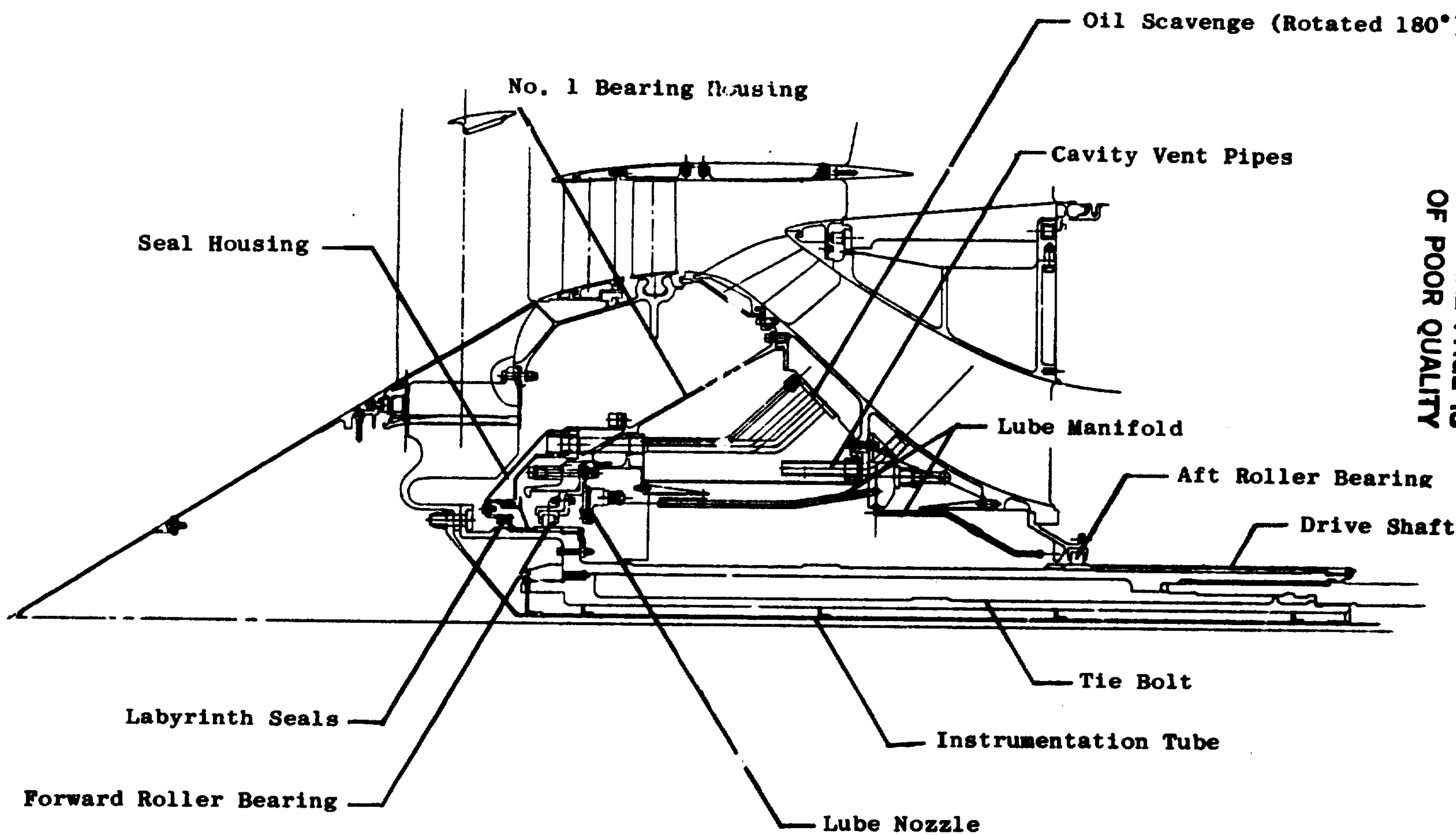

Figure 2.7-1. Forward Sump for Full Scale Fan Test.

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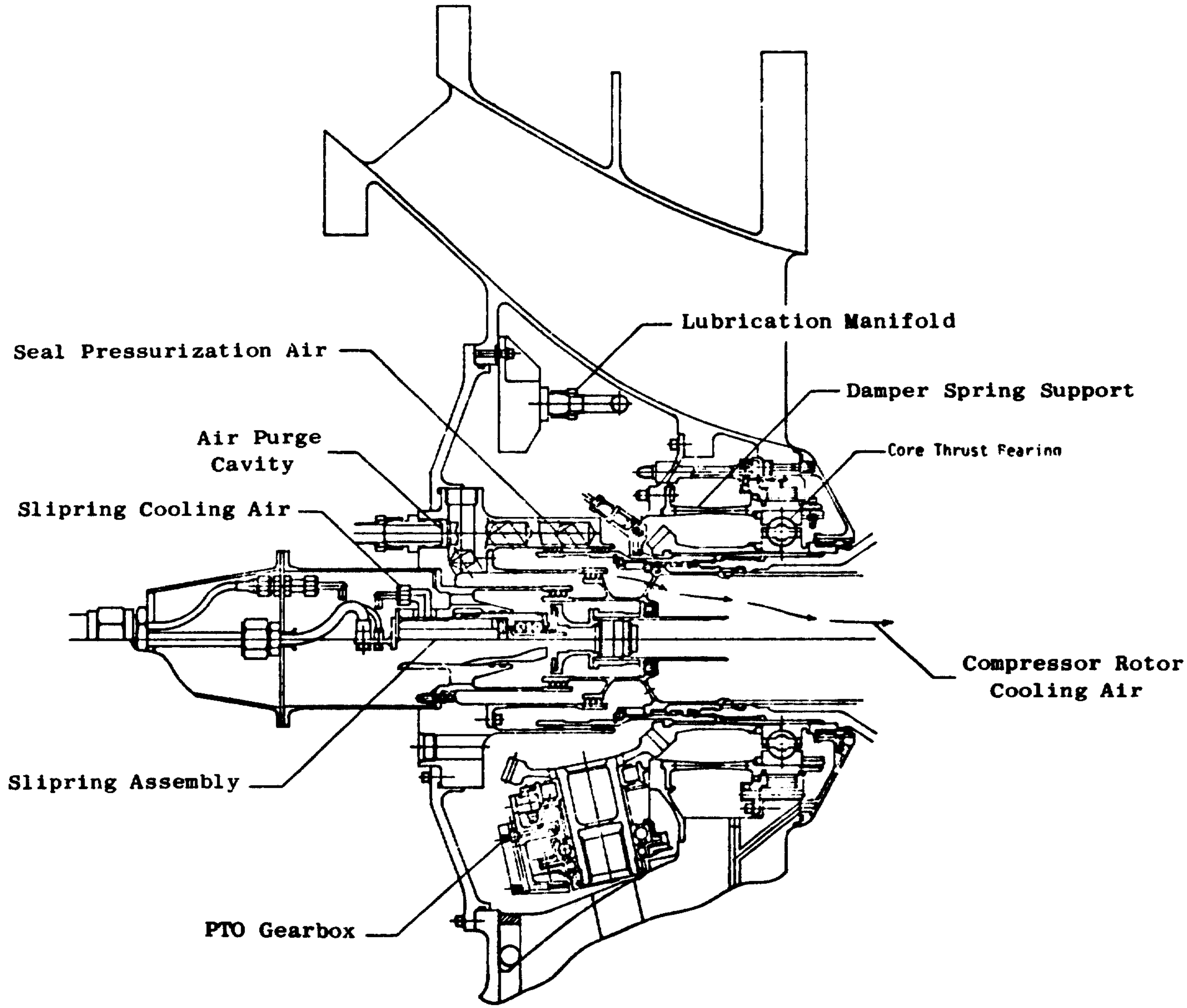

Figure 2.7-2. Core Forward Sump.

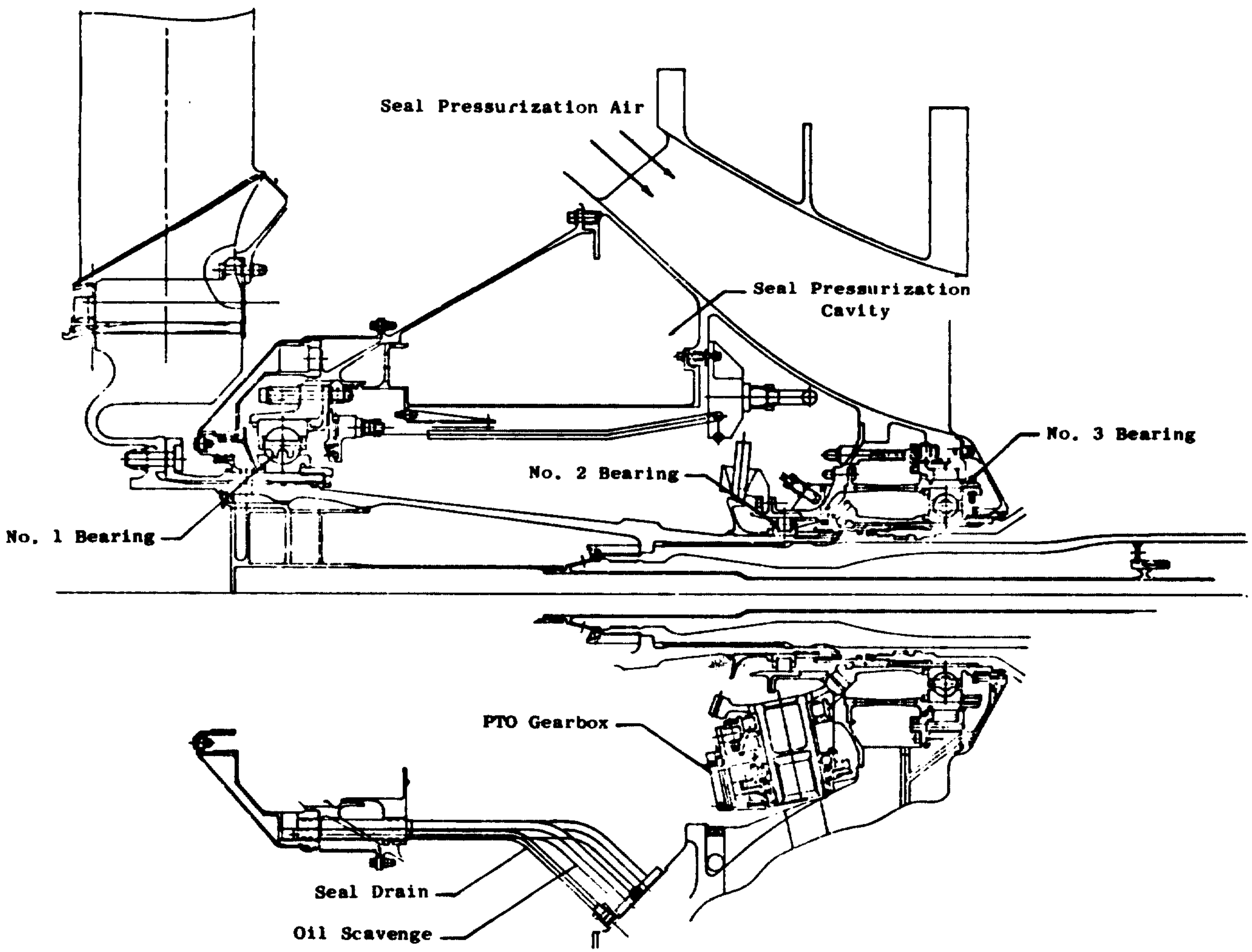

Figure 2.7-3. ICLS Forward Sump.

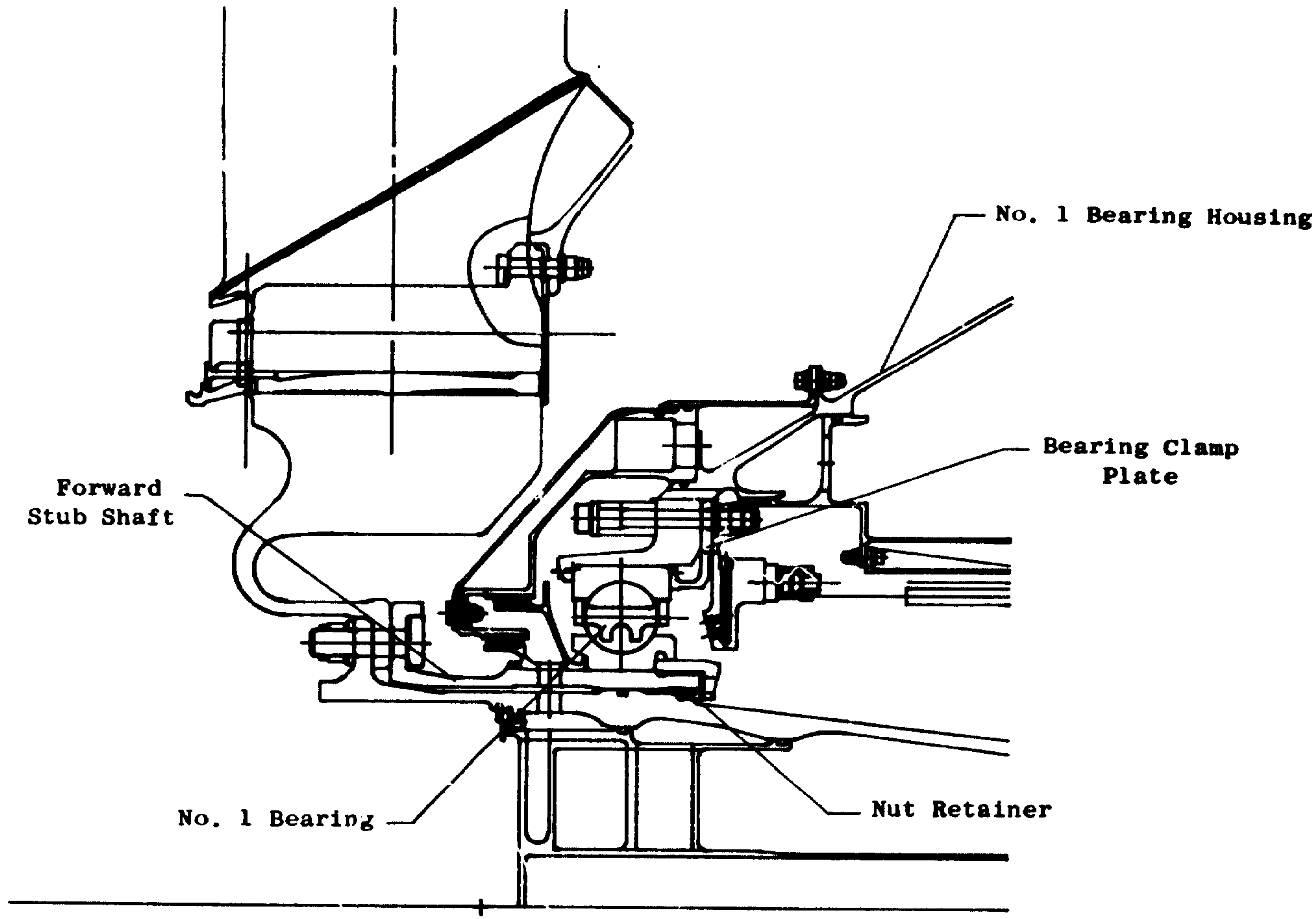

Figure 2.7-4. ICLS No. 1 Bearing Area.

The No. 1 bearing is mounted in a 17-4 PH housing and is located axially by a forward shoulder and a clamp plate. The assembled clamp load is 24,000 to 55,000 pounds which is greater than the bearing expected thrust load of 15,000 pounds. The clamp plate material is AMS 6415 and the assembly stresses and operating stresses are below the strength capability of the material.

The clamp plate stresses have also been checked at expected blade-out conditions and stresses are below the material yield point.

To secure the clamp plate to the bearing housing, 30 0.4375-inch-diameter bolts are used which are more than adequate to react the thrust bearing loads.

The design analysis of the forward stub shaft (Figure 2.7-4) included blade-out condition, and all stress levels are below the yield strength of the AMS 6415 shaft material. Buttress threads are used on the threaded nut to prevent nut from "jumping" threads under extremely heavy thrust loads. This nut is keyed in place by a trapped retainer.

Operating conditions for the thrust bearing have been analyzed and the internal radial clearances and fits established. Both the housing and shaft fit have been designed to remain tight throughout the operating speed range. The contact angle operating range is below 23° and 32° which is consistent with good design practice.

The fits and clearances for the No. 2 bearing have also been established. The effect of the radial growth of the LPT shaft forward spline has been considered in the clearance calculation along with the normal shaft fits. The design intent is to maintain a shaft fit of 0.0002 to 0.0013 inch tight. The operating clearance will be 0.0036 to 0.0048 inch.

Work Planned

- Continue procurement of hardware for Core and ICLS engine
- Provide engineering coverage during mechanical checkout of 1-10 compressor rig
- Support assembly of FSFT rig.

2.7.1.2 Aft Sump Mechanical Design

Technical Progress

During this reporting period design effort has continued in the following areas:

- Aft sump for the Core engine
- Aft sump for the ICLS engine
- ICLS LPT shaft design.

Aft Sump for the Core Engine

The Core aft sump is shown in Figure 2.7-5. All hardware is in the manufacturing cycle. No serious problems have been encountered during the manufacturing process and hardware should be available in adequate time for engine buildup.

Engineering coverage will continue during the manufacturing cycle.

Aft Sump for the ICLS Engine

The aft sump used in the ICLS engine is shown in Figure 2.7-6. All drawings have been completed with the exception of the LPT shaft and the No. 5 bearing housing. Material is available or on order for both of these parts.

Some modifications have been made to the sump since the last reporting period. These include revising the aft air seals as a result of a thermal stability analysis and modifying the centervent tube to prevent possible oil leakage due to an oil buildup.

The fits and clearances for the No. 4 and No. 5 bearing have been established for the ICLS operating conditions.

The No. 4 (Intershaft) bearing is designed to run with a slight interference between the rollers and the inner and outer rings. This interference will be tightest at maximum speed and is estimated at -0.002 inch. This design approach is being used to minimize the tendency of roller to race skidding that is prevalent in high DN, lightly loaded bearings. Running with this tight fit affects the life of the bearing but calculated L_{10} life is greater than 5000 hours which is more than adequate for the overall test program.

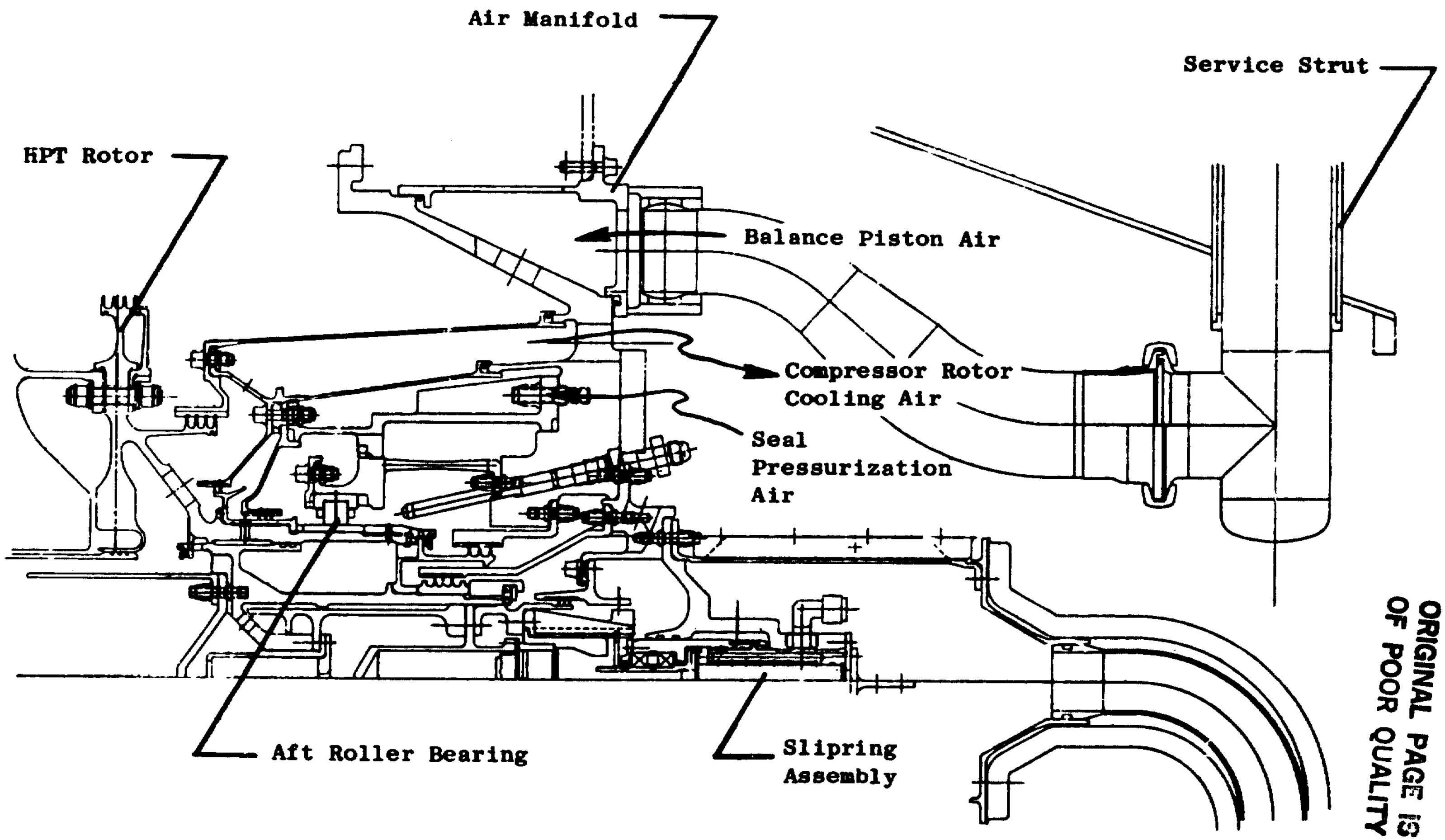

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Figure 2.7-5. Core Engine Aft Sump.

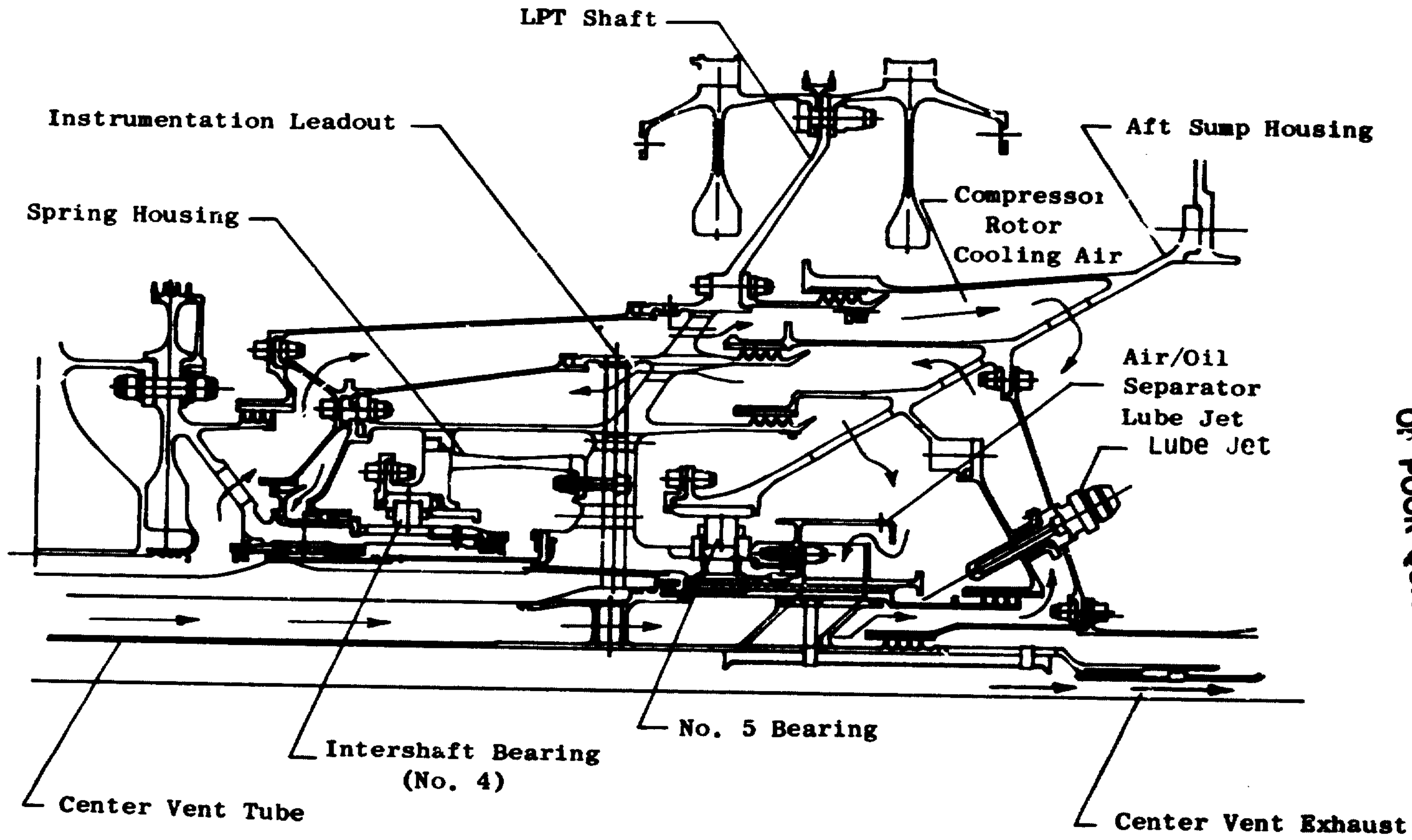

Figure 2.7-6. ICLS Aft Sump.

The No. 5 bearing has been released for manufacture.

This bearing will be underrace cooled by a lube jet mounted in the aft cover. Rotating oil passages are designed so that 20% of the lube jet oil goes to the No. 5 bearing and 80% travels forward and is distributed to the No. 4 bearing.

The detailed analysis of the No. 5 bearing housing is being completed and the detail drawing should be available in the second quarter of 1981.

ICLS LPT Shaft Design

The design analysis of the LPT shaft has been completed and the shaft configuration is shown in Figure 2.7-7. The shaft is machined from a one-piece Inco 718 forging which should be available early in the second quarter of 1981 by which time the finished detail drawing should be available.

Figures 2.7-8 and 2.7-9 show the calculated stresses in the shaft and cone area, respectively. In all areas, the life calculated is greater than the design requirements of 10^4 takeoff cycles.

The forward spline has been configured to use existing tooling and is basically a modified 10/20 pitch spline. The LCF life in the spline area considering stress concentrations, is in excess of 15,000 cycles.

Work Planned

- Complete drawings for LPT shaft and No. 5 bearing housing and place an order
- Continue engineering coverage during procurement of Core and ICLS hardware.

2.7.1.3 Lube System Design

Technical Progress

During this reporting period major emphasis has been on the Core engine system analysis.

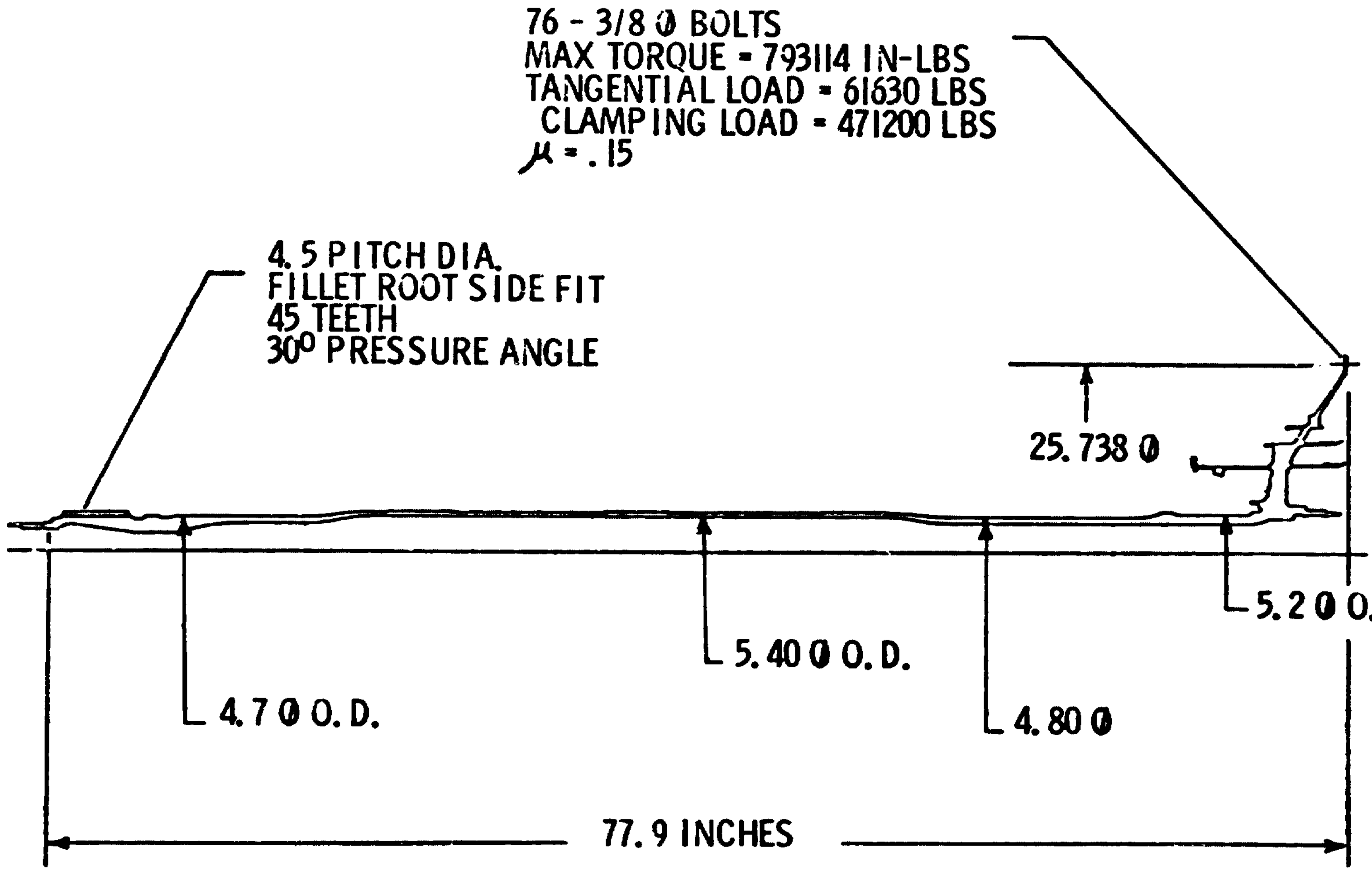

Figure 2.7-7. LPT Shaft Configuration.

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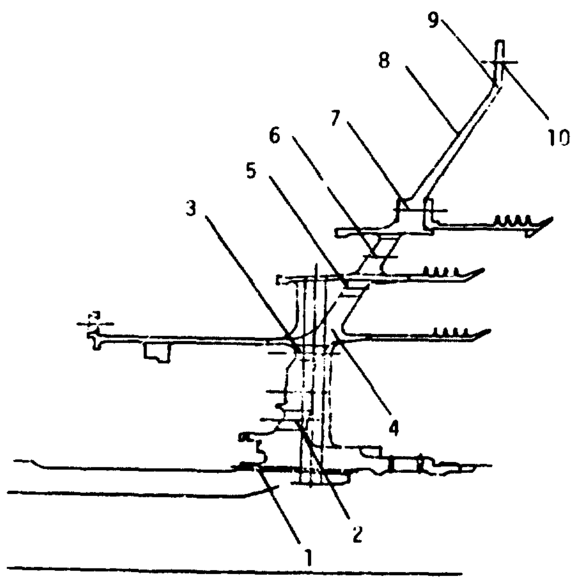
LOCATION	1	2	3
MEAN STRESS (HCF) (KSI) EFFECTIVE TENSILE	80	90	90
ALT. STRESS (KSI) EFFECTIVE TENSILE	27	23	25
ALT. STRESS (KSI) ALLOWABLE -10^7 CYCLES	27	25	25
LOW CYCLE FATIGUE LIFE (CYCLES)	4.5×10^4	2×10^4	2.5×10^4
MEAN STRESS (LCF) EFFECTIVE TENSILE (KSI) $K_t=1$	65	60	63
TEMPERATURE	300°F	650°F	600°F

Figure 2.7-8. LPT Shaft Stresses.



		LCF CALC.		CREEP RUPTURE LIFE		
POINT	TEMP. ^{o}F	$\sigma ss = \sigma alt.$ for A = 1 (KSI)	LCF CYCLES	σeff (max) No. K_K	0.2% CREEP LIFE (Hrs)	RUPTURE LIFE (Hrs)
1	216	64.5	1.5×10^4	43	-	-
2	214	64.5	1.5×10^4	43	-	-
3	247	63	1.5×10^4	72	-	-
4	360	49	$> 10^5$	98	-	-
5	700	67.5	10^4	45	-	-
6	798	69	10^4	46	-	-
7	937	37.5	$> 10^5$	25	-	-
8	997	12.5	$> 10^5$	25	$> 10^5$	$> 10^4$
9	1031	55.5	5×10^4	74	$> 10^5$	$> 10^4$
10	1069	66	10^4	44	$> 10^4$	$> 10^5$

Figure 2.7-9. LPT Shaft Cone Area Stresses.

The lube system for the Core engine is shown in Figure 2.7-10, and lube supply capacity versus scavenge capacity is shown in Table 2.7-I.

Table 2.7-I. Lube Supply Versus Scavenge Pump Capacity.

Component	Lube Supplied, gpm	Scavenge Capacity, gpm
Forward Sump	4.2	17.35
Aft Sump	2.6	7.75
Accessory Gearbox	2.7	11.20
Bypassed Oil	7.3	N/A
	16.8	

From Table 2.7-I, it can be seen that the supply pump (an existing pump is being used) capacity is greater than that required by the engine. The "extra" oil will be bypassed back to the lube tank as is shown in the lube system schematic. Table 2.7-I also shows that the minimum scavenge ratio is 2.98 which should be more than adequate according to GE design practices.

Figure 2.7-11 shows the relationship of the Core thrust bearing load versus Core percent corrected speed for the Core engine. The curve shows rotor thrust calculations for both a 14.7 psia inlet and a 23 psia inlet at nominal A8 with the balance piston cavity at 50 psia The balance piston cavity pressure can be adjusted but for these operating conditions the Core thrust bearing load will vary between 2500 and 6000 pounds.

Figures 2.7-12 and 2.7-13 show an update on the pressures and flows around the Core engine sump.

The sump pressure will be lowered below atmospheric pressure to keep the sump seal flow in the right direction. Sump pressures at ambient inlet conditions will be 10 to 11 psia. With the sumps maintained at these pressures, the external air/oil separators will be flowing 0.05 lb/sec and 0.07 lb/sec for the forward and aft sump, respectively.

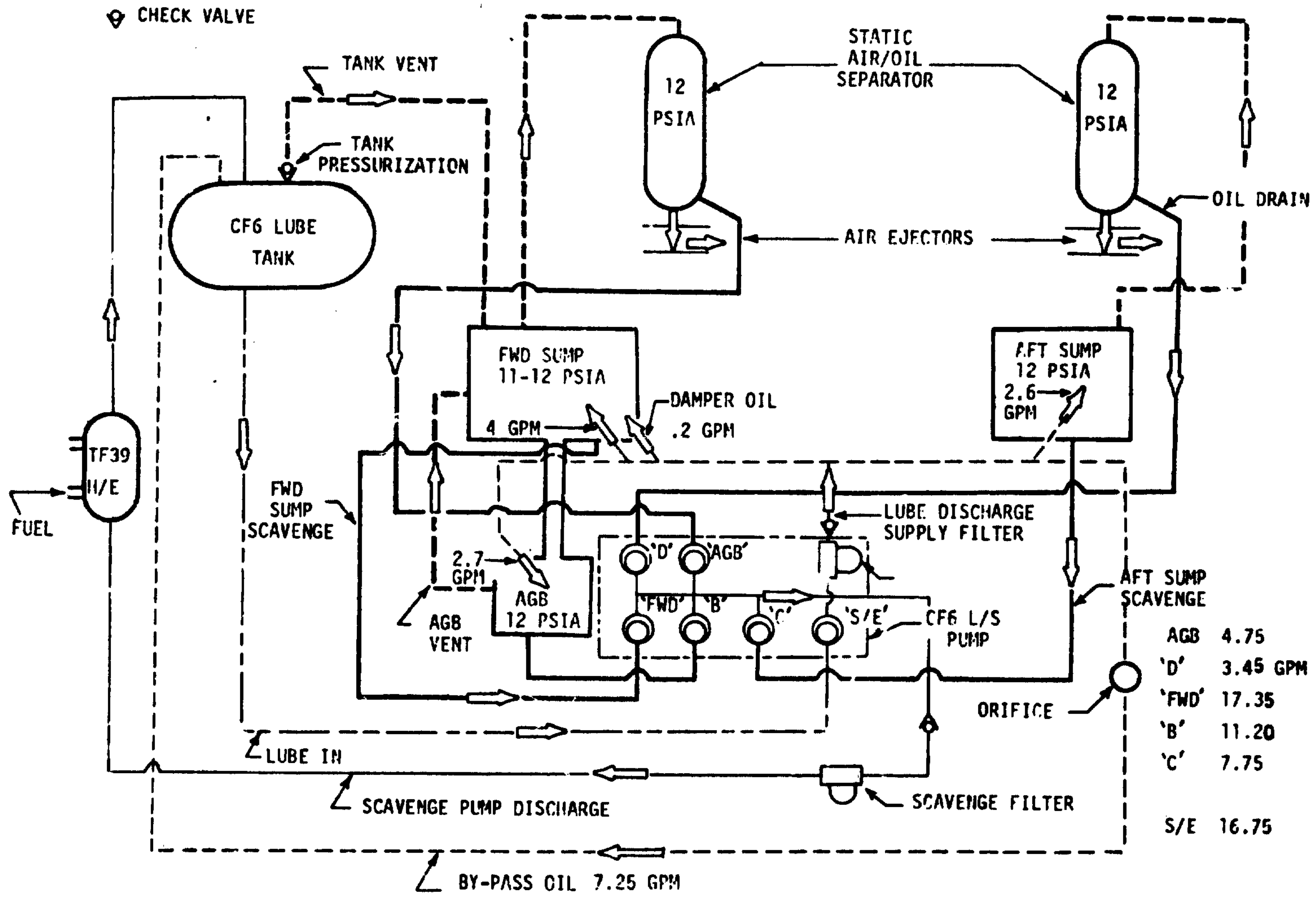


Figure 2.7-10. Core Engine Lube System Schematic.

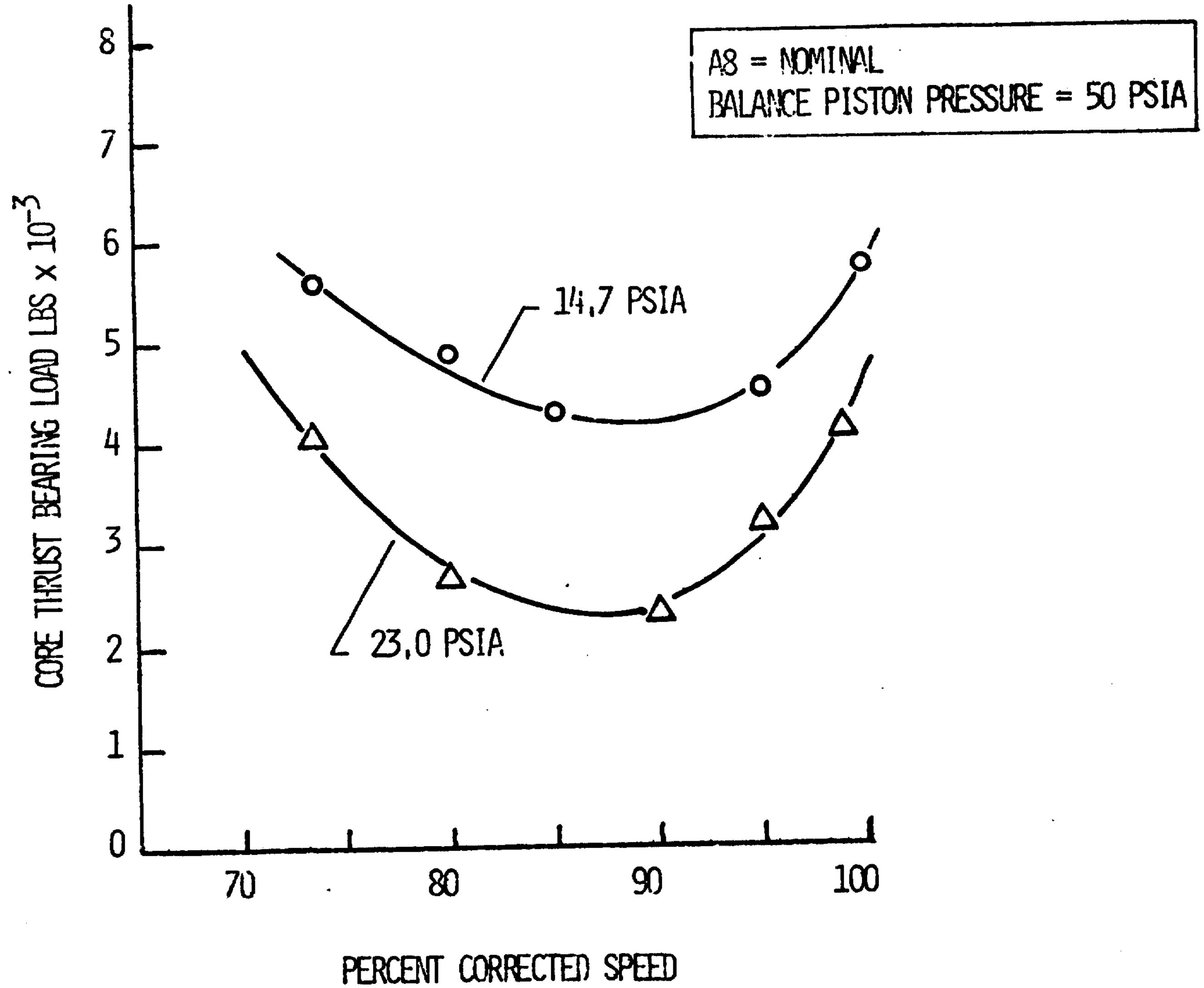


Figure 2.7-11. Core Thrust Bearing Load Versus Percent Corrected Speed.

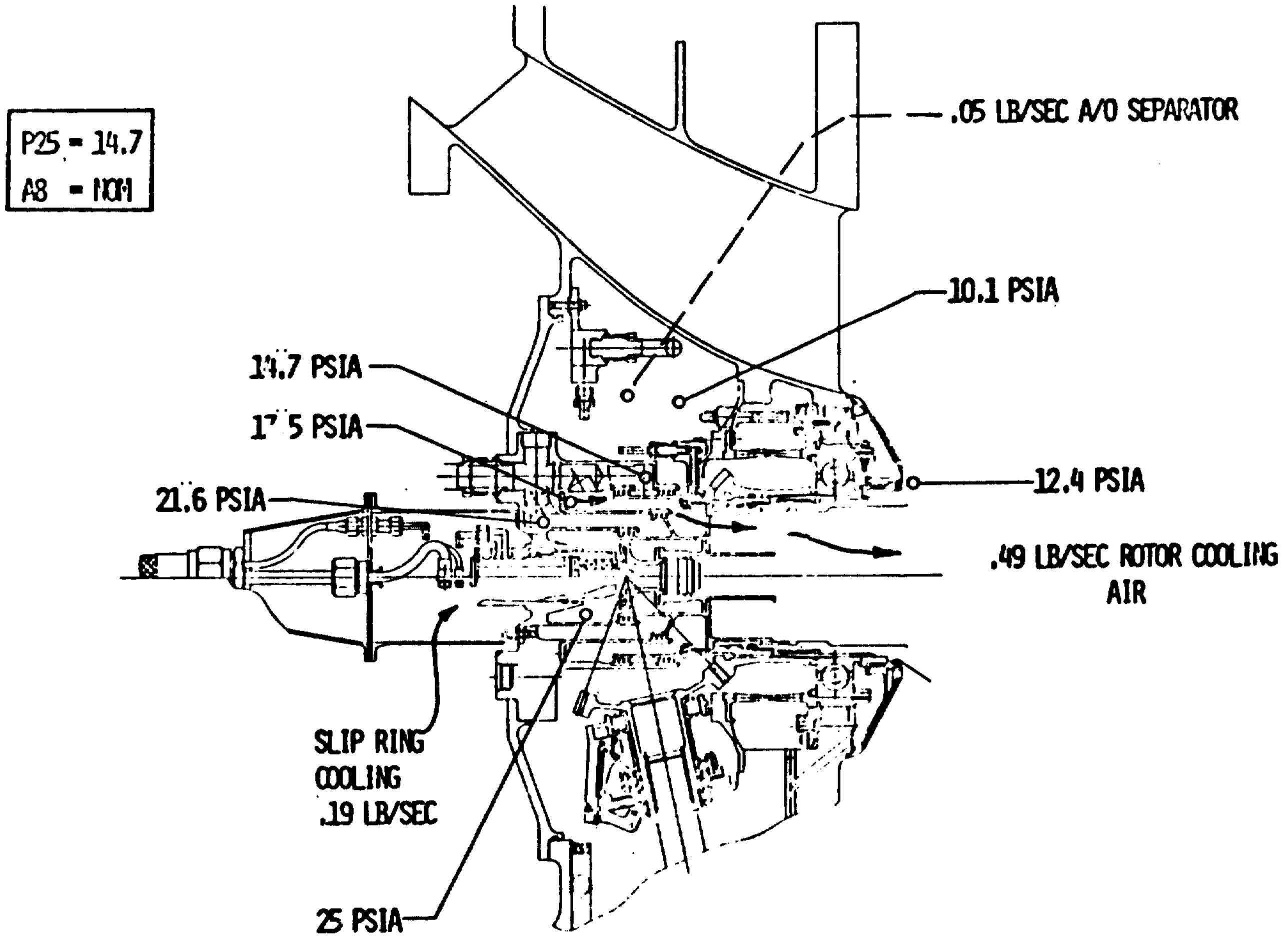


Figure 2.7-12. Core Forward Pressures and Air Flows.

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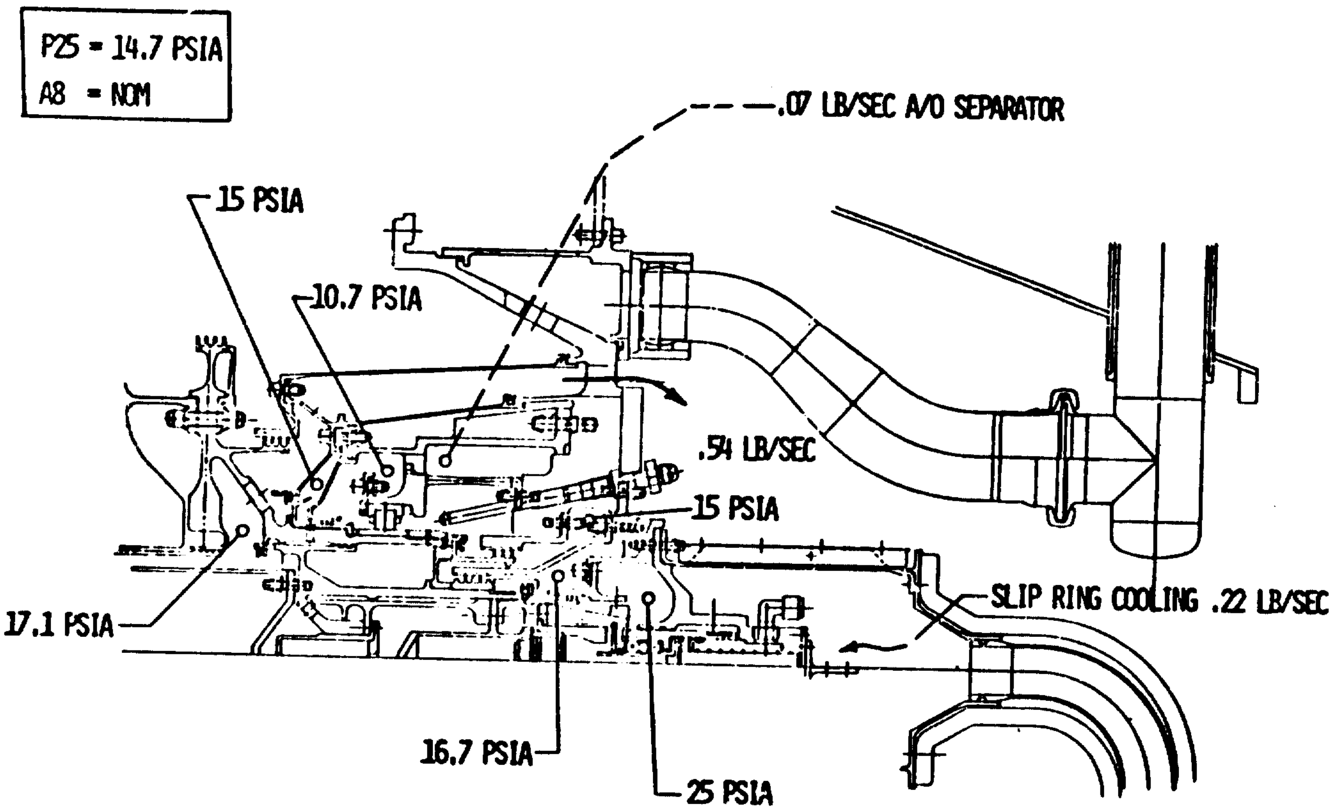

Figure 2.7-13. Core Aft Sump Pressures and Air Flows.

Work Planned

- Continue to coordinate lube system with other engine functions
- Update lube system flow model and reconfirm system lube flows
- Update secondary flow system and rotor thrust for ICLS engine.

2.7.2 Accessory Drive System

Technical Progress

All detail drawings for the accessory drive system are now issued and either out for quote or ordered. Figures 2.7-14 and 2.7-15 show cross sections of the PTO and AGB, respectively.

Figure 2.7-14 shows, in detail, the PTO housing which is fabricated from 17-4 PH. The housing is basically a sector design whose concept has been used successfully for other GE engines.

Work Planned

- Continue procurement of all hardware.

2.7.3 Configuration Design

Technical Progress

All major components have been located and piping requirements have been established. An existing oil tank is being used and simple modifications have been identified for mounting to the engine.

Major emphasis during this reporting period has been on completing drawings for the Core engine. Approximately 30 remain to be issued and these will be completed early in the second quarter of 1981.

Work Planned

- Complete detail drawings for Core and ICLS engine
- Continue procuring hardware.

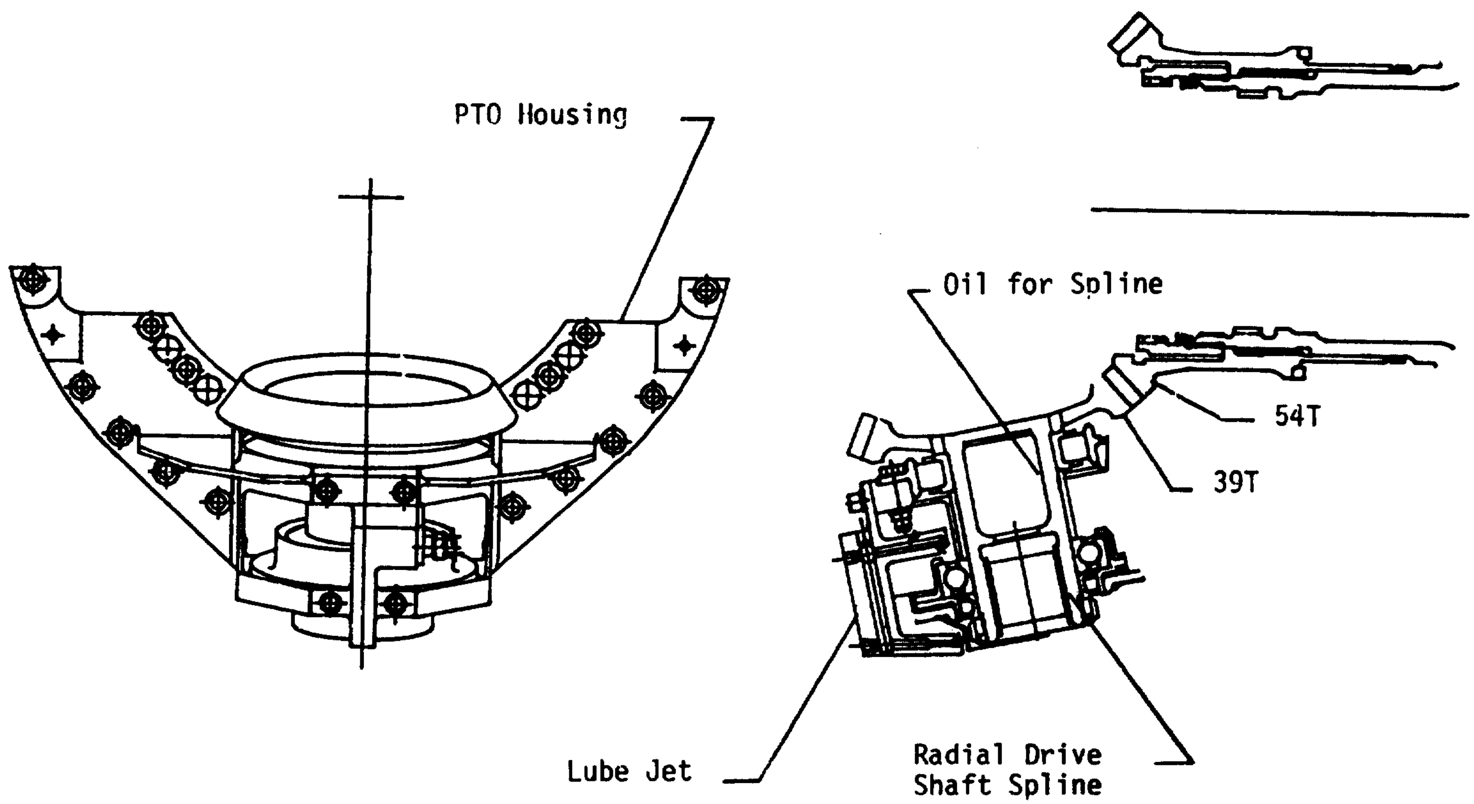

Figure 2.7-14. PTO Gearbox Assembly.

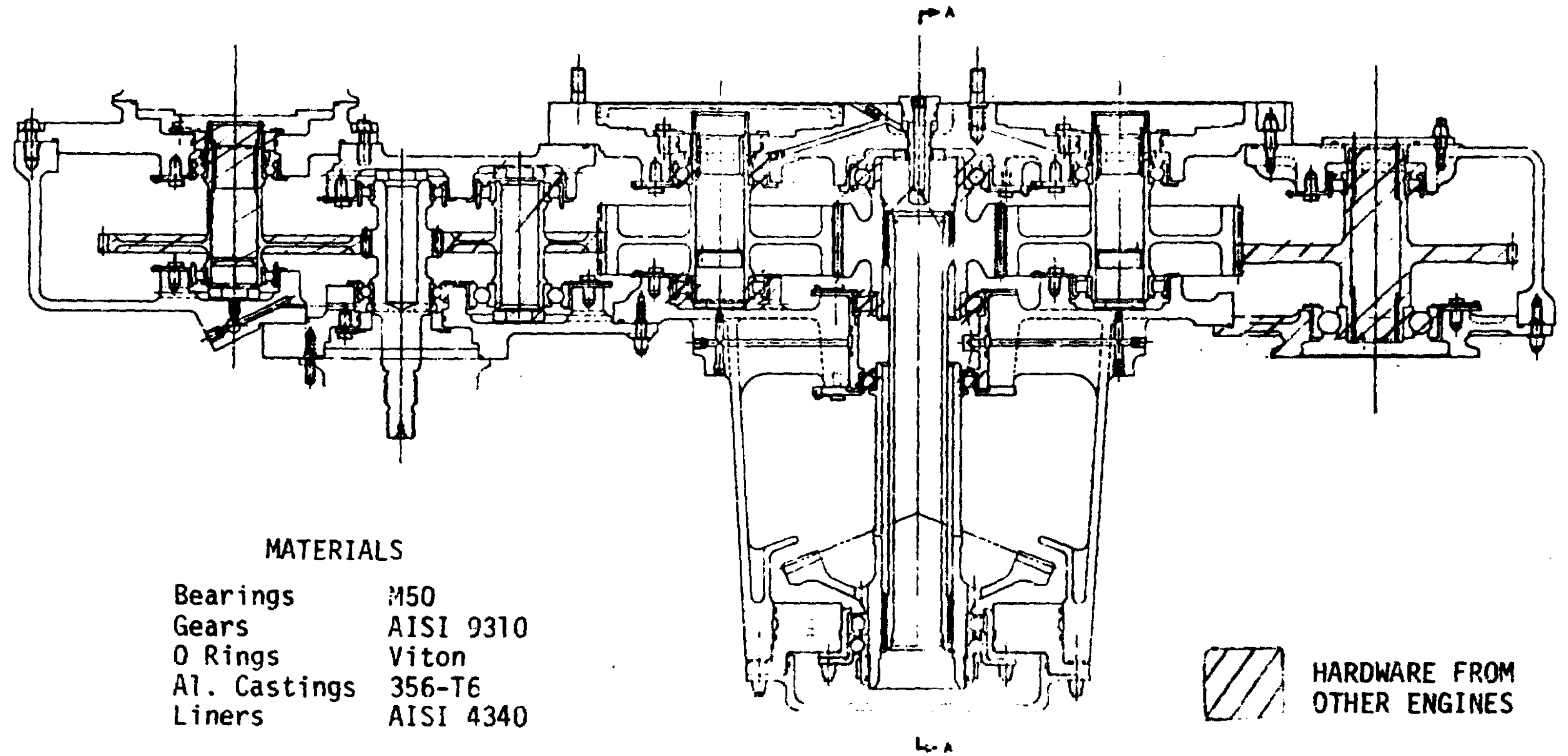

Figure 2.7-15. Core/ICLS Accessory Gearbox.

2.7.4.3 Intershaft Bearing Test

Technical Progress

The design for the required modifications to the intershaft test rig have been completed and hardware is shown in Figure 2.7-16. This modification will allow the application of a radial load to the test bearing which will simulate the expected operating radial load of the engine.

A new test bearing is being procured and is in the final stages of manufacture. Depending on delivery of test bearing, testing should be completed during the next reporting period.

Work Planned

- Procure test bearing
- Complete bearing test.

2.7.4.4 Bearing Underrace Cooling Test

Technical Progress

The second phase of testing will be completed dependent on the delivery of the test bearings and the rig hardware.

Rig modifications have been identified and a cross section of the rig is shown in Figure 2.7-17. Rig hardware drawings have been completed and hardware is either out for quote or on order.

It is expected that the bearings will be available in April 1981 and the rig hardware shortly thereafter.

Work Planned

- Complete procurement of test bearings and rig hardware
- Complete testing.

2.7.5 Bearing, Systems, Drives, and Configuration Fabrication

Technical Progress

Assembly of 1-10 compressor rig has been completed and rig is in test. All major hardware is available for the FSFT assembly and buildup has started.

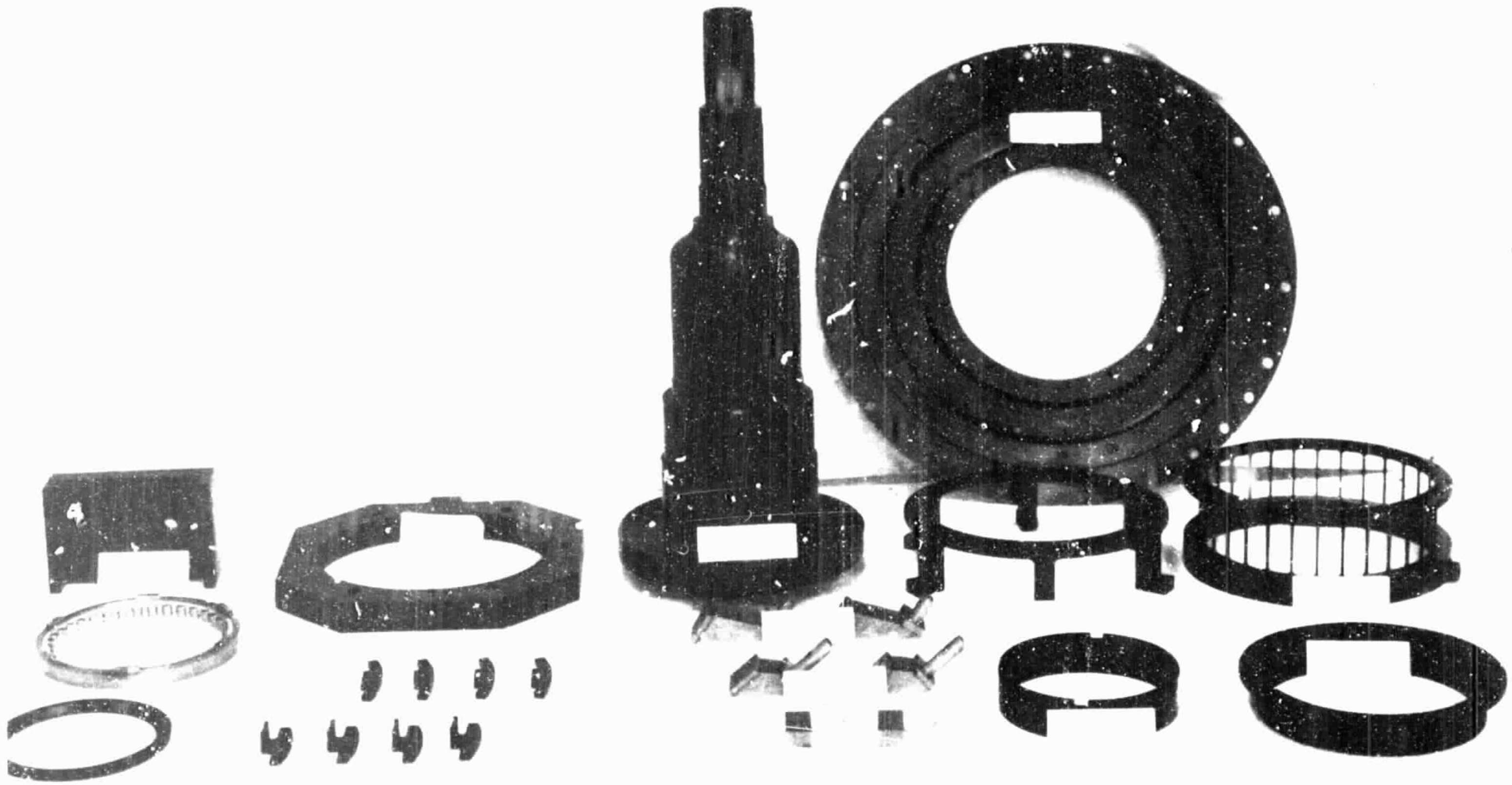

Figure 2.7-16. E^3 Differential Bearing Rig with Side Load.

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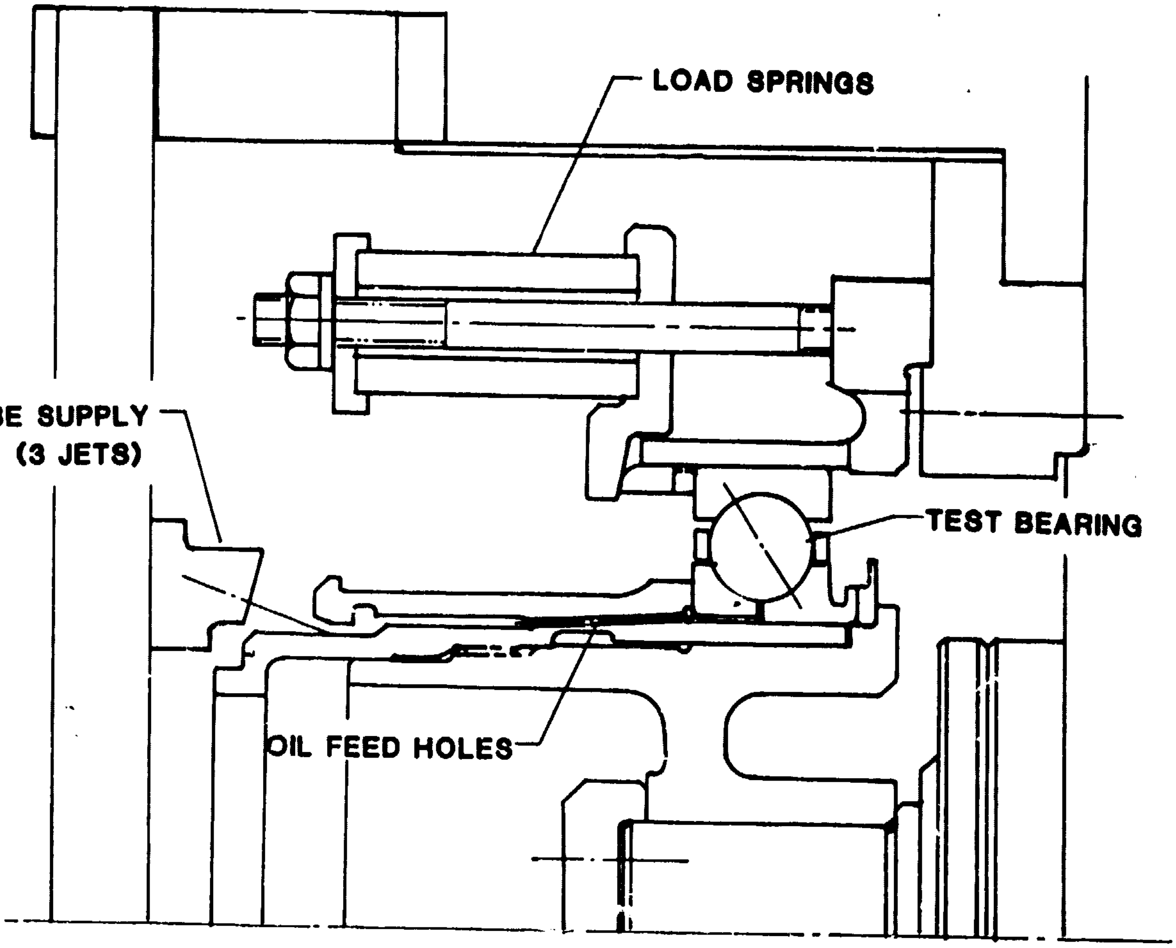

Figure 2.7-17. Under Race Cooling Test Rig.

All hardware is either out for quote or being procured for the forward and aft sump of the Core engine.

Hardware for the accessory drive system, which is common to both the Core and ICLS engine, is out for quote or on order. Quotes have not been received for the machining of the main housing which is the last major part to be ordered. The first casting for the accessory gearbox main housing has been poured and inspections are being made. Results of these inspections should be available in April 1981.

All bearing procurement is basically proceeding on schedule with no known problems.

Work Planned

- Continue procurement of Core related hardware.

2.8 CONTROL AND FUEL SYSTEM

Overall Objectives

The primary objectives of the control and fuel system program is to define a system for E^3 which provides operational capability and reliability equal to or better than that provided by current transport engine control and fuel systems, which thoroughly exploits the fuel conservation features of E^3, and which employs digital electronic computation suitable for interfacing with an aircraft flight control computer. An additional objective is to demonstrate the system functionally on the Core and ICLS engines.

The proposed control and fuel system for the E^3 is based on many of the proven concepts and component designs used on the CF6. The major difference is in the addition of full-authority digital electronic computation, which provides significant improvements in control flexibility, accuracy, and aircraft/engine integration capability.

The digital control is expected to contribute to the low fuel consumption of the E^3 by providing automatic power management and optimum control of variable geometry on the engine over the full range of operating conditions which will be encountered. The control will also help reduce deterioration in engine efficiency by automatically preventing engine overspeed, overpressure, or overtemperature.

The E^3 fuel system employs fuel handling concepts that are basically similar to the CF6. In the flight design, an engine-driven, positive-displacement gear pump with an integral centrifugal boost element is used for pumping. A pump-mounted heat exchanger, downstream of the gear pump element but upstream of the system filter and fuel metering section, provides the dual functions of cooling the engine lube oil and heating the fuel to prevent filter icing. Fuel metering is accomplished by the combined use of the fuel metering valve with a bypass valve that returns excess fuel to the inlet of the gear pump element. Downstream from the metering valve, excess heat from the air being bled from the compressor for use in the aircraft environmental control system (ECS) is transferred into the fuel, thereby improving engine system efficiency by returning waste heat to the engine cycle and reducing the ECS air cooling requirements. After passing through this heat exchanger, the metered fuel is divided as necessary to accommodate the double-annular combustor.

The system also controls several variable geometry elements on the engine including the variable stator vanes and starting bleed valves on the compressor and air valves for controlling clearance in the compressor, HP turbine, and LP turbine.

Engine starting will be accomplished using a gearbox-mounted, air-turbine starter similar to that used on the CF6. Scheduling of fuel flow during the starting sequence will be done by the digital control, which will also provide ignition sequencing logic by energizing the ignition as a function of a starting command input and de-energizing it when the engine has accelerated beyond the point where ignition is needed. The ignition will also be automatically energized if the digital control detects deceleration conditions that indicate a burner blowout has occurred.

Development Approach

The control and fuel system design and development effort began with a preliminary design phase in which various system and component design options were defined and evaluated. Particular emphasis was placed on the method of incorporating advanced technology features such as the full authority digital control, air/fuel heat exchange for waste heat recovery, active clearance control, and fuel flow division for the double-annular combustor. The initial study work resulted in the definition of a preliminary control system design in late 1978. A preliminary design review of this system was held at NASA-Lewis in October 1978 and the design was further reviewed as a part of an overall engine preliminary design review in November 1978.

With the basic system design concepts established, the next task was to define detailed system and component requirements for the demonstrator engine program and proceed with detailed component design. This task, which was essentially completed in 1979, involved several different types of activity. In the system design area, the basic system and component requirements were established and documented. Concurrently a number of supporting analytical

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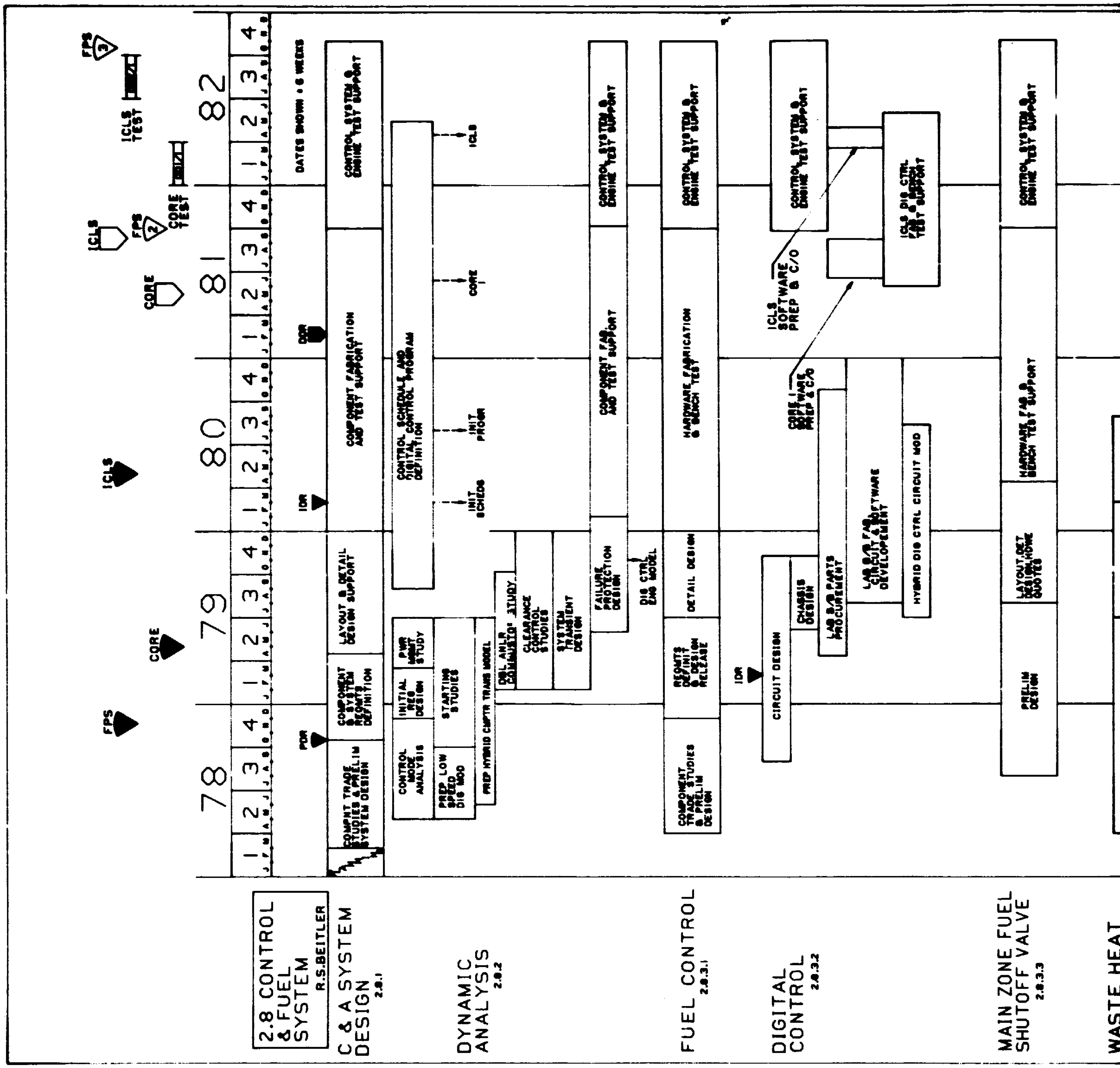

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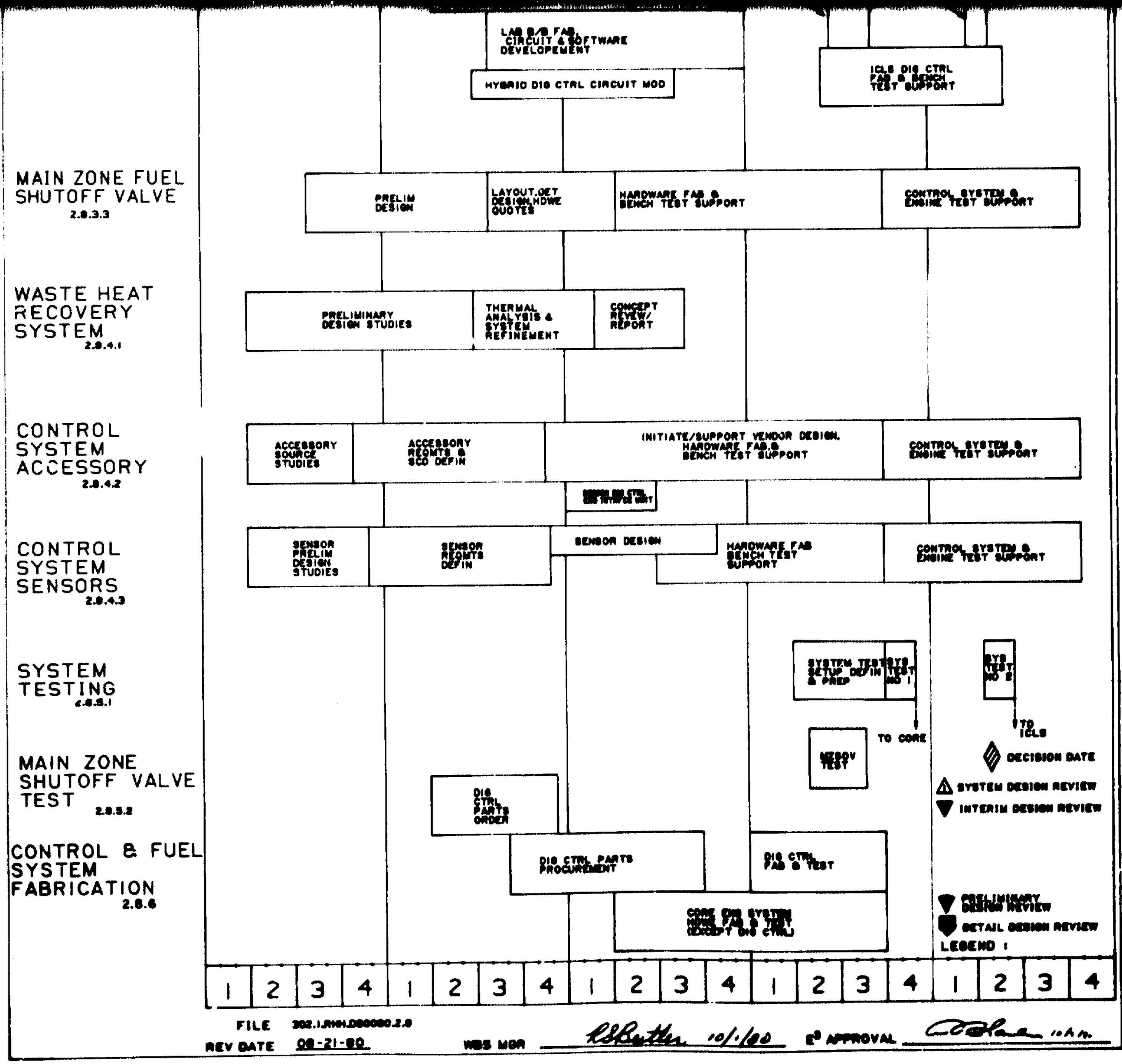

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activities were carried out under the Dynamic Analysis subprogram using computer simulations to investigate engine and control characteristics and to define requirements, particularly those related to control stability and response. Meanwhile, component design and development activity were proceeding as described below.

Fuel control design and development is being done under WBS 2.8.3.1. For the Core and ICLS, a modified F101 fuel control will be used. Detailed requirements for the control have been defined and transmitted to the control vendor. Control modifications were defined in detail, the modifications reviewed with NASA in March 1980, and implementation of the modifications is underway for the core engine under WBS 2.8.6 and for the ICLS under 4.2.4.

Design of the digital control is being carried out under WBS 2.8.3.2. The basic control design process includes initial circuit design, experimental circuit refinement using a laboratory breadboard control, and chassis design. Because construction of the laboratory breadboard requires procurement of electrical parts prior to the control system detailed design review (DDR) scheduled for early 1981, an interim design review (IDR) of the digital control was conducted in March 1979.

Construction of the breadboard was completed in March 1981 and it is being used to check out the initial control software program which has been defined as a part of the Dynamic Analysis activity (WBS 2.8.2). A software refinement and checkout is scheduled in the second quarter of 1981 to accommodate changes which are expected to result from the analysis of data from engine component testing.

The digital control resulting from the above design effort will be an off-engine unit suitable for both the Core and ICLS engines. It will be built and tested under WBS 2.8.6. This control will definitely be used on the Core engine but it is planned that an on-engine unit, constructed under a separate program, will be used for the ICLS engine. The on-engine unit will incorporate an advanced, hybrid, electronic packaging concept.

The main zone fuel shutoff valve covered by WBS 2.8.3.3 is a new valve designed to assist in the fuel flow division required by the double annular combustor. (This valve, previously called the flow divider valve, was renamed because the majority of the flow division task is performed by the fuel nozzles in the system design that has evolved and this valve serves primarily as a shutoff.) Detailed design of this valve was completed and reviewed with NASA in early 1980. Fabrication and test of the valve for the core engine is proceeding under WBS 2.8.6 and 2.8.5.2, respectively.

The air/fuel heat exchanger work under WBS 2.8.4.1 began with thermal model studies to examine, in detail, the potential fuel savings available by transferring heat from the compressor bleed air to the engine fuel. Early in the second quarter of 1980, the results of these studies were assessed by NASA and General Electric and a decision was made to delete the hardware demonstration originally planned for the waste heat recovery concept. This was done primarily because the concept, although unique in function, is implemented with standard components that do not represent new technology.

Design of control system accessories and sensors is being carried out under WBS 2.8.4.2 and 2.8.4.3. These components do not represent advanced technology and consequently, for economic reasons, will be modifications of existing designs. Detailed component requirements were established in 1979 and transmitted to the appropriate design organizations (predominantly outside vendors) to serve as a basis for detailed definition of component modifications. The designs were reviewed in a March 1980 IDR and released for procurement.

As a result of the efforts described above, a full set of control system components will be available in 1981 and will be assembled into a complete system for bench testing in late 1981. Subsequently, the components will be installed for operation of the core engine.

Control system hardware from the core engine will be used on the ICLS except that an on-engine digital control will be used as noted previously and a fan speed sensor and LP turbine clearance control subsystem will be added. The ICLS control system components will also be bench tested as a complete system prior to operation on the engine. Additional control system hardware will be procured to provide a limited, yet adequate, supply of spare parts.

2.8.1 Control System Design and Analysis

Technical Progress

Some additional system design work was required during this reporting period in conjunction with the introduction of two potential new control functions. A temporary reset of the fuel flow split between the two zones of the double-annular combustor has been proposed to promote main zone ignition during engine starts and an HP turbine casing heating feature has been proposed to provide clearance margin for accelerations made before engine temperatures have stabilized after startup.

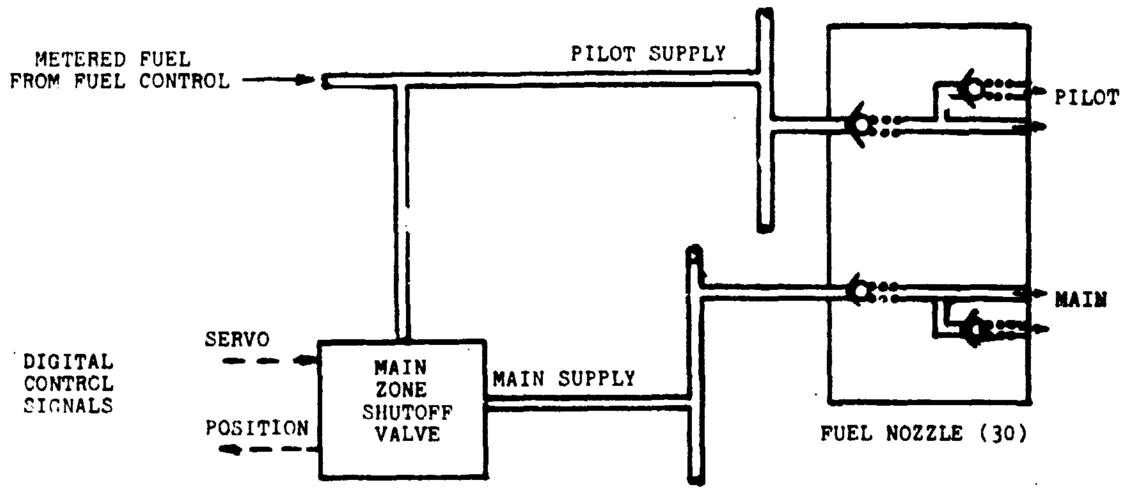
Fuel Flow Split Reset

The potential need for a fuel flow split reset was identified as a result of combustor component testing. For engine starts with the double-annular combustor, ignition can be achieved best at low motoring speed by fueling the pilot zone only but, once ignition has occurred, it is desirable to light the main zone and accelerate to idle with fuel split evenly between main and pilot zones to provide a uniform temperature profile. However, combustor testing has shown that the even fuel split does not provide sufficient fuel for ignition in the main zone. Fortunately, the data also shows that pilot zone fuel can be temporarily cut back after pilot ignition without blowout so that the main zone can be temporarily enriched for ignition purposes.

One method considered for providing temporary main zone enrichment was to retain the same basic fuel distribution arrangement, Figure 2.8-1(a), but to add resistance to the pilot supply line and control the main zone shutoff valve so that it is partly closed for normal main zone operation and temporarily opened fully for main zone ignition enrichment during a start. This approach was rejected because it creates excessive fuel system pressure at high flow.

The fuel distribution system modification shown in Figure 2.8-1(b) has been identified as a suitable means of providing the desired flow split reset. With the pilot zone and main zone valves fully open and offering little flow resistance, fuel flow will be split in accordance with fuel nozzle characteristics as in the current system. With the pilot zone valve closed, the orifice restricts pilot flow and enriches the main zone for ignition.

(a) Existing System



(b) System incorporating Pilot Zone Re-set

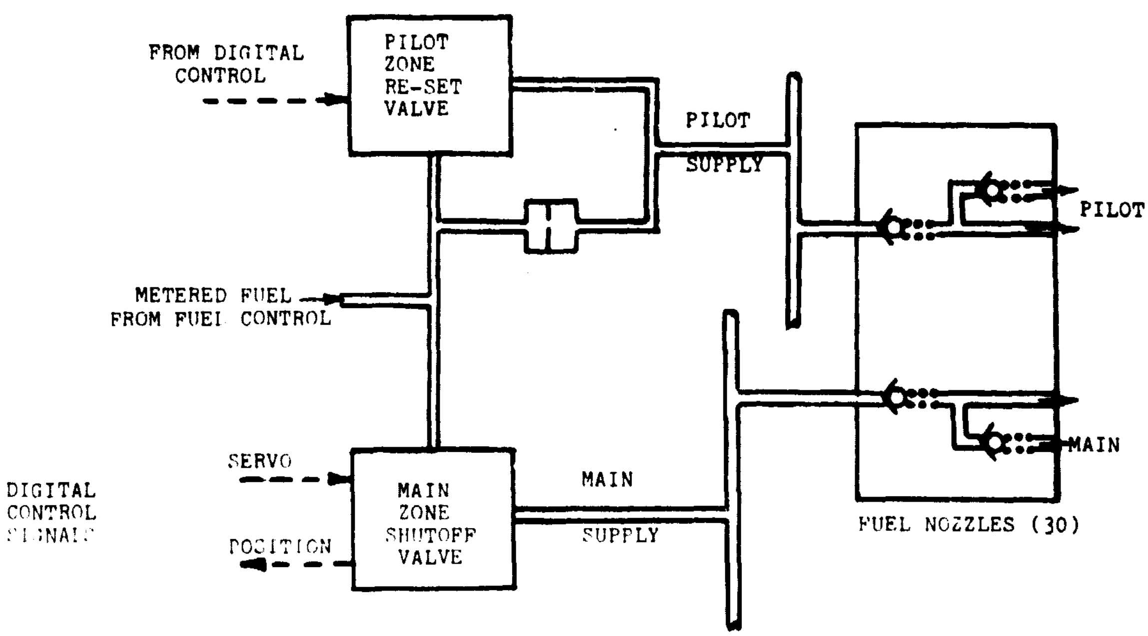


Figure 2.8-1. Fuel Distribution System Options.

The pilot zone valve will be controlled by an on/off signal from the digital control. During a start, both valves would be closed until an increase in T49 gas temperature is sensed by the control at which point the main zone valve is opened. Once the main zone is lit and the engine accelerates a small amount above the speed existing at the initial T49 rise point, the pilot zone valve is opened for the acceleration to idle. At idle, the main zone valve is closed for low emissions as in the existing system.

This new fuel supply arrangement will be implemented on the Core and ICLS with experimental flexibility provided by including a manual control option for the pilot zone valve and providing for pilot supply orifice replacement.

HP Turbine Casing Heating

Studies using the clearance model developed under WBS 2.8.2 indicated that radial clearance in the HP turbine would be quite small if the engine was accelerated to high speed before engine temperatures had stabilized at idle after a start. To prevent insufficient clearance under such conditions, air bled from the compressor can be introduced into the HP turbine clearance control piping to heat the casing during the stabilization period after a start. The system identified for doing this is shown on Figure 2.8-2.

Compressor discharge air for the HP turbine casing heating function is taken from the customer bleed system upstream of the control valving in this system. The air is passed through a solenoid operated on/off valve into a flow control orifice and then to the HP turbine clearance control piping system immediately downstream from the clearance control valve. The solenoid valve is controlled by the digital control through a relay with contacts supplied from 28 volt d.c. power. The initial control strategy, which is still under study, will include both manual and automatic operating modes.

Detail Design Support

There has been considerable control system design effort applied during this reporting period in support of the detailed design of control strategies, system components, and system configuration. Control system requirements were

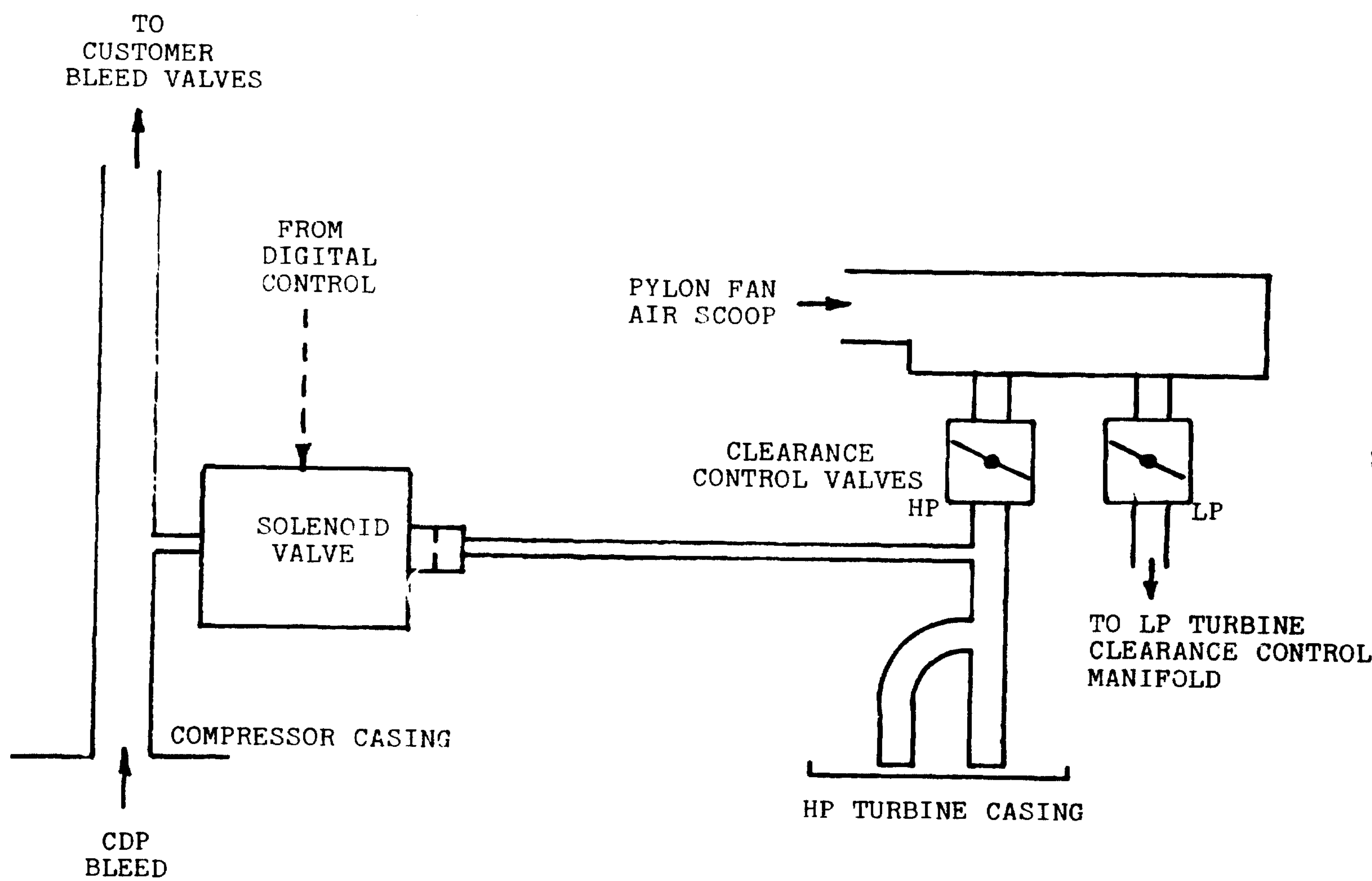


Figure 2.8-2. HP Turbine Casing Heating.

defined in more detail and control strategies were refined. The critical control strategies were translated into digital computer form under WBS 2.8.2 and then subsequently further transformed under WBS 2.8.3.2 into digital control machine language.

System design support was also given to the refinement of control system component details and to the detailed definition of system configuration (that is, component locations, mounting, and interconnections).

Work Planned

- Conduct the control system Detailed Design Review
- Design electrical cables
- Define the changes necessary in the existing initial control strategies and schedules in the digital control to accommodate the core engine
- Monitor and expedite, when necessary, the acquisition of control system hardware for control system bench testing and subsequent core engine testing
- Direct bench system testing of the core engine control system.

2.8.2 Dynamic Analysis

Technical Progress

Two significant tasks were completed during this reporting period: the translation of initial control strategy into computer program form and the definition of a new simplified engine model for use in the digital control test stand. Two other tasks originally planned for this period - Failure Indication and Corrective Action (FICA) strategy completion and core engine control schedule definition - were delayed. The FICA was delayed primarily to promote timely completion of the basic control strategy but the delay was also consistent with a NASA/GE agreement to forego FICA demonstration on the core engine where control system functions are limited (no LP system to control; no compressor stator control, the stator stages being controlled individually by test facility devices). Control schedule definition was delayed because the schedules are to be based on engine component test results and are thus affected by the compressor test delay.

Control Strategy Translation

The control strategy translation task involved starting with the strategy in block diagram or logic diagram form and translating it into a computer program suitable for further transformation into digital control computer language. This effort resulted in a program that is structured as shown on Figure 2.8-3. The relatively short main program sequentially calls upon the various control strategy subprograms. Many of these subprograms call upon additional subprograms that perform input sensing and output servo-driven functions. One fourth level subprogram, a square root function, is called upon at various points in other subprograms.

As part of the control program development process, modules of the program were checked out on a computer with checkout driver programs which applied a wide range of inputs to individual elements to verify that outputs were correct. The complete program was also checked out by running it with the E^3 transient model on a hybrid computer. The finished program was transmitted to the Digital Control Design group where it was transformed into control language under WBS 2.8.3.2.

Simplified Engine Model

The need for a new simplified engine model for use in the digital control test stand arose early in this reporting period. Initially, an engine model based on simplified engine component equations as utilized in the digital control FICA strategy was to be used in the test stand. Recent experience with a digital control for another engine, however, indicated that use of a simplified engine model based on engine cycle data curve fits would expedite preparation of the test stand and thus expedite checkout of the initial digital control breadboard. Therefore, a model of this type was developed. A diagram of this model is shown in Figure 2.8-4.

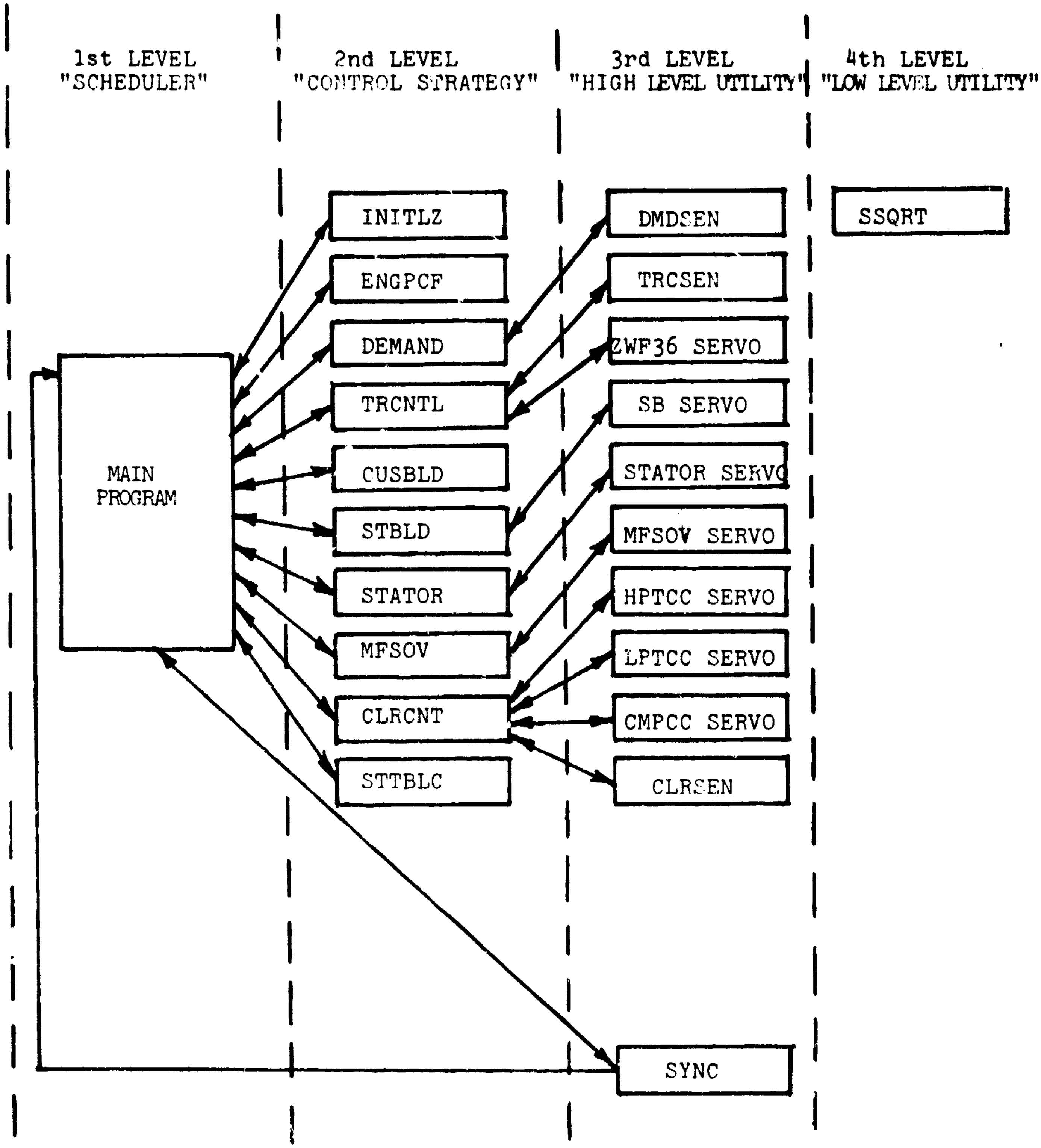


Figure 2.8-3. E^3 Digital Control Code Structure.

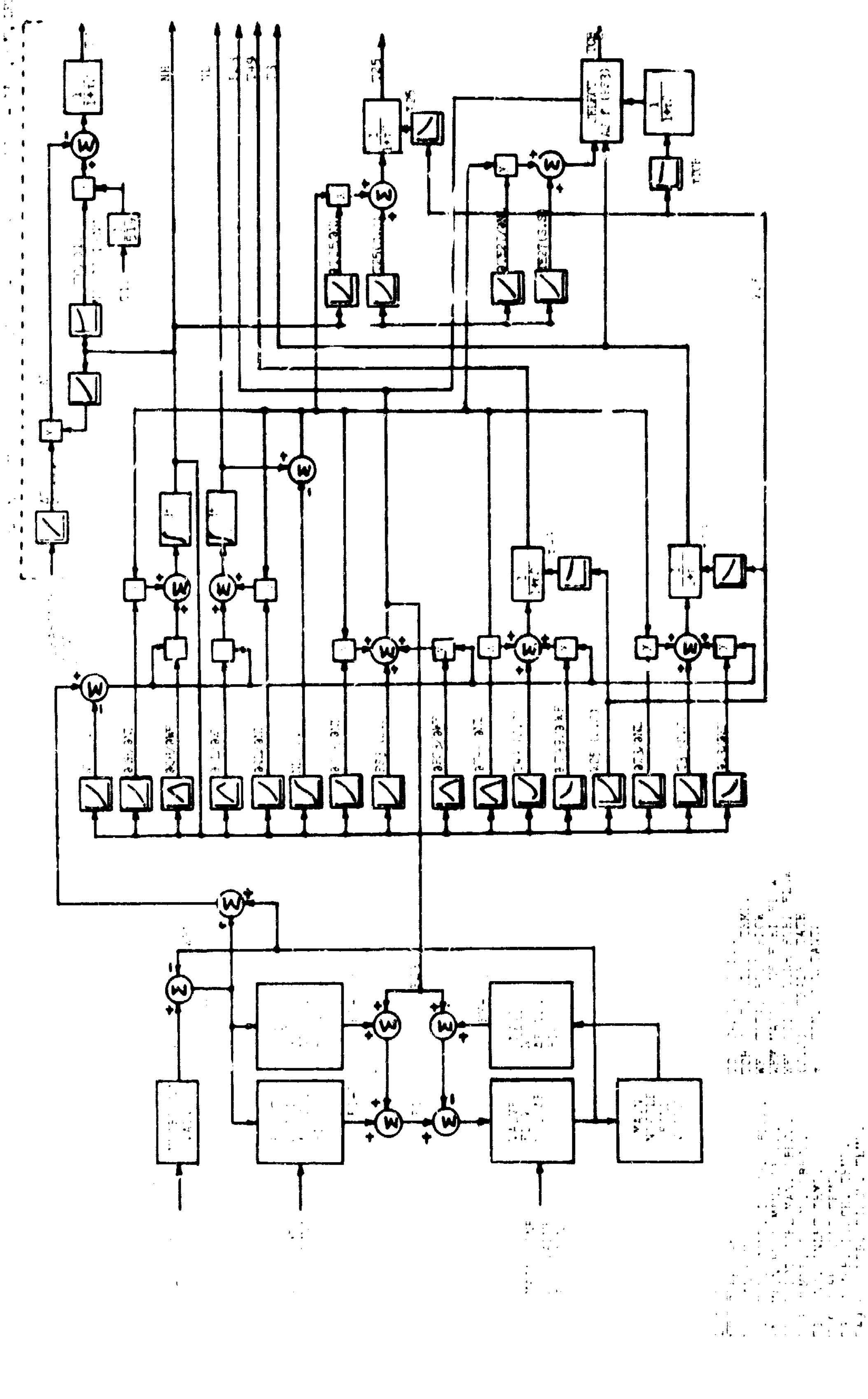


Figure 2.8-4. Simplified Engine Model.

Work Planned

- Revise the subidle engine model to reflect engine component test results and investigate starting characteristics with the revised model
- Prepare core engine fuel schedules for starting, acceleration, and deceleration based on engine component test results
- Provide support if problems are encountered in implementing the control strategy software in the digital control under WBS 2.8.3.2
- Assist in the definition of control system input simultations for the core engine control system bench test.

2.8.3.1 Fuel Control Design

Technical Progress

In conjunction with the new fuel flow split reset function (Section 2.8.1), it was necessary to define a new valve for use in the pilot zone fuel supply network as shown on Figure 2.8-2(b). A two-way, two-position valve is required, capable of operation in response to an existing digital control signal designed for driving a torque motor servovalve. The valve must have negligible leakage in the closed position with an inlet-to-discharge pressure differential up to 30 psi. In the open position, the valve must pass 5000 pounds per hour flow with no more than 5 psi pressure drop.

An inexpensive and readily available valve which appeared to be suitable for this application was a modified J79 pressurizing and drain valve operating in conjunction with a standard torque motor servovalve. While defining the details of the P and D valve modification, however, it was discovered that inadequate valve opening force exists under certain conditions. The valve relies on a fuel manifold pressure higher than servovalve return pressure for opening. The low cracking pressure of the E^3 fuel nozzles results in a fuel manifold pressure lower than fuel inlet pressure (servovalve return) for much of the starting sequence. Therefore, it was concluded that a valve was needed which was servo-powered in both directions.

A review of available valves indicated the best alternative to be an E^3 main zone shutoff valv ith the position feedback transducer deleted for economy. This utilizes an existing valve design and provides a commonality of spare parts with nother element of the E^3 system. Steps have been taken to procure the necessary additional main zone shutoff valve parts to make up a pilot zone reset valve.

The remaining fuel system design activity has been associated primarily with acquisition and modification of hardware. Mounting and connection hardware for fuel system components have been identified and placed on order. A modified fuel pump drive spline has also been identified which will allow the use of an existing CF6 gear in the E^3 gearbox and avoid the cost of a new gear. The relief valves in each of the F101 fuel pumps which will be used for core and ICLS testing were adjusted upward as required by the E^3 fuel system. Relief valve cracking pressures were increased from 1020 to 1230 psid.

Work Planned

- Release control schedules for the fuel control
- Support fabrication and test of fuel system hardware.

2.8.3.2 Digital Control Design

Technical Progress

Digital control progress was made in both hardware and software. The hardware activity was associated with construction and checkout of the digital control breadboard. Some problems were encountered with the power supply in the control which is a new design incorporating a primary section supplied from the engine-driven control alternator and a secondary section supplied from an aircraft (or test facility) 28 volt d.c. source. The power supply elements had been developed independently and when they were mated for the first time some incompatibilities were evident, the most troublesome being noise spikes in the 5-volt regulator output and excessive current drawn from the 28 volt d.c. source. These problems necessitated considerable experimental modification and prevented meeting the December 1980 goal for completion of the breadboard.

By early 1981, the 5-volt regulator noise problem was corrected but the 28 volt d.c. draw problem persisted; therefore, a decision was made to proceed with breadboard fabrication using only the primary section of the power supply in order to expedite initial software checkout. The breadboard was sufficiently complete for the start of software checkout early in March 1981.

A separate power supply breadboard has been constructed and it is being used to develop the dual-power supply while the main control breadboard is involved in software checkout.

Preparation of the initial control software was done in parallel with breadboard fabrication but it was completed earlier because of the breadboard delay. The control strategy computer program, developed under WBS 2.8.2, served as the basis for the control software. A digital control source program was prepared which included the control strategy program along with input/output processing material, memory assignment information, and a variety of material related to further transformation of the program into digital control machine language. This source program was then run with compiler and assembler programs in the time-share engineering computer to produce the machine language program for the digital control memory. Checkout of this program began in March on the control breadboard as noted above.

Another software task completed during this reporting period was the coding of a simplified engine model for the digital control test stand. The computer in this test stand is of the same design as the digital control computer. The model, which was developed under WBS 2.8.2, was incorporated into a source program and run through the same compile/assemble process as the control program to produce a machine language version of the model for the test stand computer. The model was successfully checked out in the test stand computer early in 1981.

Work Planned

- Complete checkout of the initial control software in the breadboard
- Develop a satisfactory design for the dual-power supply
- Prepare software for the core engine digital control
- Support fabrication and test of the core engine digital control.

2.8.3.3 Main Zone Shutoff Valve

Technical Progress

Activity has been limited in this area with almost all design work completed prior to this reporting period. A minor design change was made to incorporate filter fittings in the valve to provide additional protection against contamination in the test fuel system. The only other activity was the identification and ordering of mounting and piping interface items for the valve.

Work Planned

- Support fabrication and testing of the main zone shutoff valve for the core engine.

2.8.4.2 Accessory Design

Technical Progress

The need was established for a new air valve for the HP turbine casing heating function described under WBS 2.8.1. A two-way, two-position valve was needed that can tolerate compressor discharge pressure and temperatures, has negligible leakage in the closed position, and can pass up to 0.3% of core airflow in the open position. A commercially available, solenoid-operated Whittakers Controls poppet valve was identified and ordered for this application.

The other activity in this area was predominantly associated with hardware procurement and preparation for test. Collapsible links for the start bleed valve actuation system were placed on order as were a variety of mounting and connecting hardware items. Some items of test equipment required for air valve testing were also identified.

A detailed review of the vendor's design for the compressor clearance control valve was conducted. This three-way rotary shear valve is one of the few E^3 accessories that is a custom design rather than a modification of an existing design. The design was found to be basically satisfactory with just a few recommendations for changes. Material changes from 440 to 17-4 steel for the input shaft and from 416 to 410 steel for the rotor were recommended and an improved lever retention means was suggested. The design is now released for manufacturing with these changes incorporated.

Work Planned

- Support the acquisition and bench test of accessory hardware.

2.8.4.3 Control System Sensors

Technical Progress

Detailed design of the T3 sensing probe has been completed based on a test instrumentation thermocouple probe which has been applied on several previous engines. An improved lead-out attachment concept has been incorporated to make the probe less susceptible to handling damage which has occasionally been experienced with the basic probe design. The swaged, magnesium oxide-filled metal sheath in which the thermocouple junction is encased is led out, looped, and spliced into a flexible, metal-covered lead before clamping to the probe housing. This design reduces the chance of overstressing and breaking the swaged metal sheath as has happened in the past.

The pressure sensor package has also been designed. The four pressure transducers will be mounted in separate housings along with associated electronics and the housings will be mounted side by side in a module which will mount within the digital control. Pressure ranges for the transducers have been confirmed and procurement activity has been initiated for all the pressure sensing hardware.

Progress has also been made in sensing system design for the HP turbine discharge temperature and the three engine casing temperatures that are used in the clearance control strategy. For the HP turbine discharge temperature input to the digital control, 5 of the 35 thermocouple signals from the T42 test instrumentation rakes will be connected in parallel and spliced into the control system electrical cabling. These 5 signals will each be from different radial and circumferential locations and will be used exclusively for the control system.

For casing temperature sensing, skin thermocouples will be mounted on each of the engine casings that are being used for clearance control (aft compressor, HP turbine, and LP turbine) for exclusive control system use. Three thermocouples will be used at each location, one for initial use and two for spares. Each of the thermocouples will be spliced into a wiring junction to which the control system cabling will be attached.

The toothed disk that will be mounted to the fan shaft for providing a fan speed sensor target has been designed. The disk will have six identical, equally spaced teeth. Initially it had been planned that one tooth would be higher than the others to provide an orientation signal for fan balancing; but other means are available for doing this and it is desirable to have equal radii teeth set to run at the minimum practical clearance from the sensing head in order to provide adequate low speed signal strength.

Activity this reporting period also included the placing of orders for fuel metering valve rotary position transducers and PS3 sensing plugs along with miscellaneous sensor related attaching hardware. Also, a minor electrical connector orientation change was made on a start bleed valve position transducer to avoid interference with recently defined air system piping.

<u>Work Planned</u>

- Complete the design of the T42 and casing temperature sensing systems in conjunction with Instrumentation Design
- Order T3 sensors and remaining miscellaneous sensor interface hardware
- Design the electrical compressor stator position feedback sensing system.

2.9 NACELLE STRUCTURES

Overall Objectives

The flight nacelle configurations for the E^3 will incorporate advanced composite structures and sound-suppression features to meet or exceed commercial certification requirements for noise, performance, life, and thrust reversal of the fan bypass flow. The general configuration of the composite-structure hardware will be a flight inlet, a fixed-cascade thrust reverser, and a fan nozzle and inner cowl hinged to the upper pylon. Kevlar 49, graphite, and fiberglass/epoxy skins will be combined for strength, toughness, light weight, and low production cost. The nacelle structure will have no life-limiting conditions for the expected operation loads of commercial aviation, including crosswinds and thrust reversal. System mode frequencies of the nacelle will avoid strong sources of vibration excitation throughout the engine operating range. Fabrication goals, maintenance goals, and performance goals for the structure's hardware will contribute to the overall goals of the E^3 flight propulsion system to reduce specific fuel consumption and direct operating costs (DOC).

Composite-structure nacelle components for the flight propulsion system will be designed through detail layouts but will not be built. Weight estimates will then be updated based on these layout studies. Components to be studied include the flight inlet, thrust reverser, fan nozzle, and inner cowl.

A boilerplate inlet, an outer fan duct, and an inner fan duct will be built to duplicate the engine flowpath contour for acoustic and performance testing. There will be no thrust reverser in the boilerplate configuration, but space will be left to simulate its effect.

Development Approach

The preliminary aerodynamic design effort was completed in 1978. The acoustic panel design as well as the preliminary mechanical design effort was also completed in 1978.

Having completed the preliminary design, the detailed aerodynamic and mechanical design of the flight-engine composite nacelle was started in January 1979 and was completed in December 1979 with the results presented at the Nacelle PDR in May 1980. The design layouts will provide an accurate basis for calculating weights, costs, and potential performance gains.

Beginning in January 1980, a boilerplate nacelle structure is being designed to duplicate the performance characteristics of the flight-engine design. While no thrust reverser will be incorporated into this boilerplate design, space will be provided to simulate its effect on performance. The boilerplate design was discussed in the Nacelle PDR in May 1980 and will be the subject of a DDR in May 1981.

2.9.1 Nacelle Aero Design

Technical Progress

Definition of the second ICLS conic nozzle to be used for nozzle exit survey tests was completed. The nozzle exit station is the same as the performance nozzle (348.30) and the exit radius is 31.766. The area increase above the performance nozzle is approximately 5% at the exit to account for the blockage of the survey rake. This nozzle design is based on the assumption that the existing NASA emissions rake system will be used for the survey testing. The blockage effect was estimated from an analysis of the rake geometry and previous confluent exhaust, engine test results.

Analysis and design of the core component vent scoop system was completed. The system was designed to provide six air changes per minute at high engine speed conditions. Several approaches were considered in the scoop design selection. The final configuration is a simple annular slot 0.10-inch high contoured to minimize pressure drag. The slot is located just behind the fan frame at Station 200 and is an integral part of the core cowl innerring attached to the fan frame.

Wall static pressures and boundary layer rake requirements were defined for the ICLS aero/acoustic inlet bellmouth. This instrumentation will be used to assess the inlet boundary layer/recovery for performance tests and to verify that the wall Mach number distributions will be as predicted for the noise

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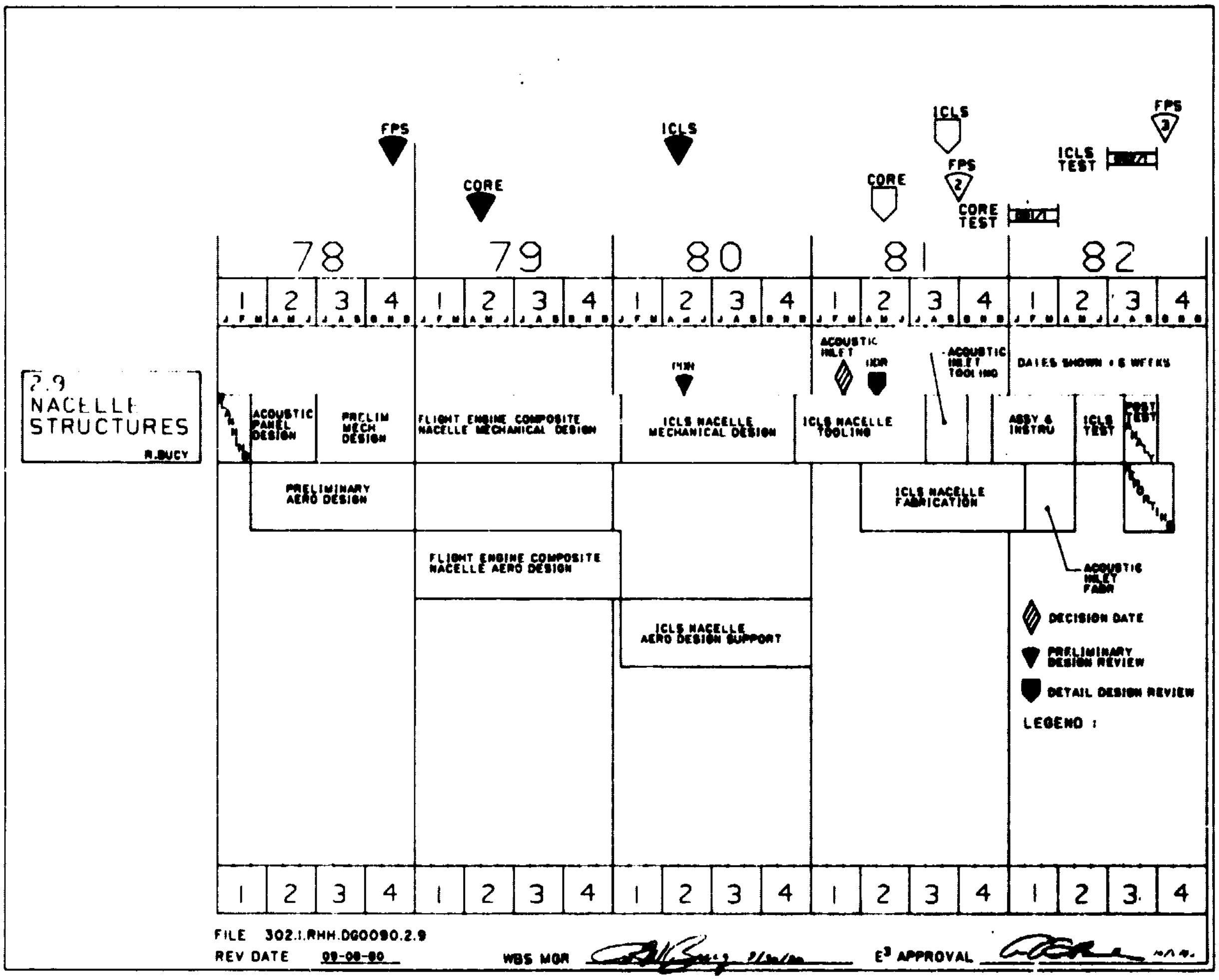

tests. Fifty-eight static pressures were defined from a point just downstream of the inlet/bellmouth lip to the fan face in several axial and circumferential rows. One boundary layer rake will be installed at 70° (clockwise from top centerline, aft looking foward) for all ICLS performance testing. This mount pad location exists as part of the instrumentation requirements for the fan component test. The nine element boundary layer rake which will be used for the component test will also be used on ICLS. Additionally, three linear potentiometers will be installed on the outside of the bellmouth at the slip joint at 0°, 45°, and 90° from top centerline (aft looking forward) to measure relative radial movement between the bellmouth and fan casing for all engine performance testing. These two measurements combined with analytical calculations will provide an appropriate analytical adjustment to the inlet recovery and engine performance data if required.

Work Planned

- The FPS nacelle design will be updated as required to reflect the results of the Phase III scale model mixer test, WBS 2.6.4.

2.9.2 Nacelle Mechanical Design

Technical Progress

The objective of this task is to design and analyze boilerplate nacelle hardware for the ICLS engine test. The basic hardware to be fabricated includes the inlet, outer cowl doors, core cowl doors, fan nozzle, and pylon side walls.

This structure was basically defined in the previous semiannual report and a drawing of the inlet was also shown. The layout drawings of all the ICLS nacelle hardware have now been completed except for incorporation of the instrumentation requirements.

The outer cowl doors are shown in Figure 2.9-1. The doors are hinged to the pylon and latched together at the bottom with five latches and a tie bar (Figure 2.9-2). The tie bar is required at the forward end due to the fact that the doors are cut back in this area to accommodate the lower pylon.

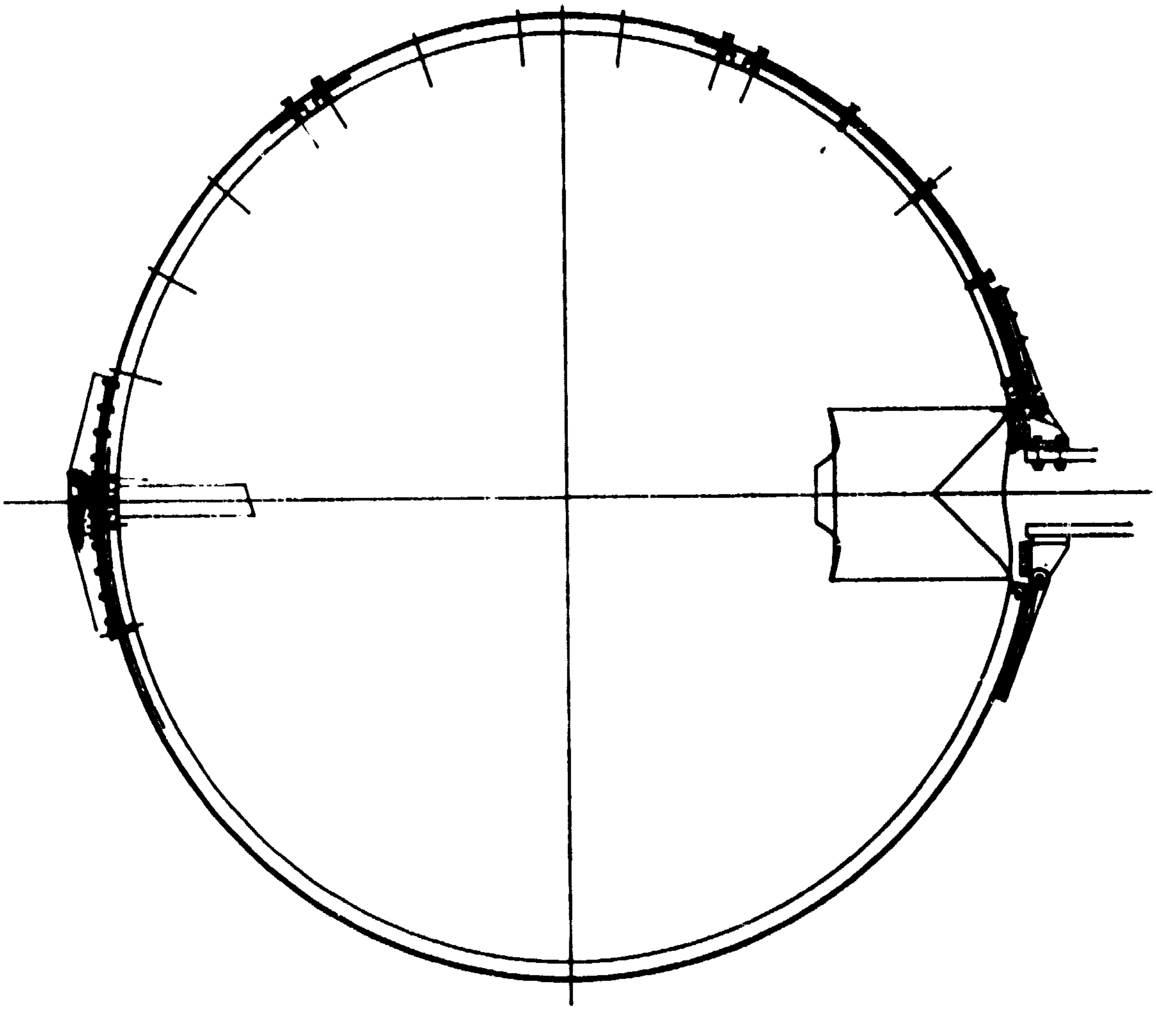

Figure 2.9-1. Outer Cowl Doors - End View.

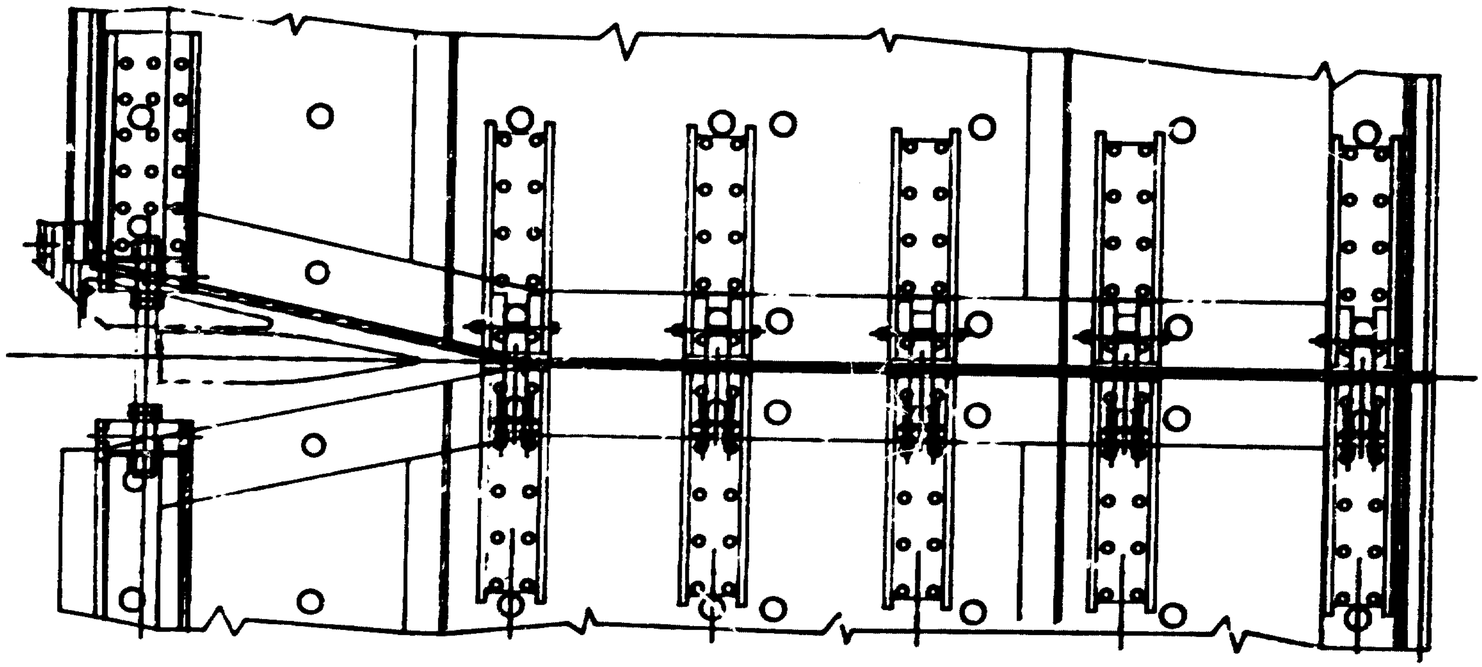

Figure 2.9-2. Outer Cowl Doors - Bottom View.

The core cowl doors, shown in Figure 2.9-3, are hinged to a floating apron structure and latched together at the bottom. Again, provisions had to be made to accommodate the lower pylon. Since there was no way to provide structural continuity to the doors through the lower pylon, they will be latched directly to the fan frame at their lower forward edges. Aft of the pylon, the doors are latched to each other at three locations as shown in Figure 2.9-4. The original concept for the construction of these doors was to make the primary structure a conical steel shell. Once the piping and accessories were located, it was apparent that this concept did not provide sufficient space. In order to provide more room in this area, the lower 90° of the doors was moved outward as shown in Figure 2.9-3. This change did not alter the position of the flowpath surface but made the standoff blocks for the acoustic panels shorter.

The pylon structure will be attached to the facility structure. The pylon incorporates an air scoop and plenums to provide air to the clearance control valves as shown in Figure 2.9-5. The pylon is sealed at the bottom and open at the top.

The fixed fan nozzle is shown in Figure 2.9-6. It is rigidly attached to the facility structure. The forward end contains a grooved rig which accepts a tongue on the aft end of the outer cowl doors. This joint is loose axially but prevents any significant rotational motion between the two structures.

Acoustic treatment is provided for all of the nacelle hardware. One change made during the last reporting period was to make the acoustic treatment in the inlet integral with the structure rather than having replaceable panels. The acoustic treatment in the fan exhaust will be of the replaceable panel type with the possible exception of the forward portion of the fixed fan nozzle.

Work Planned

The ICLS instrumentation requirements will be incorporated in the nacelle layout drawings and design support will be provided for the fabrication effort to be conducted under WBS 2.9.4.

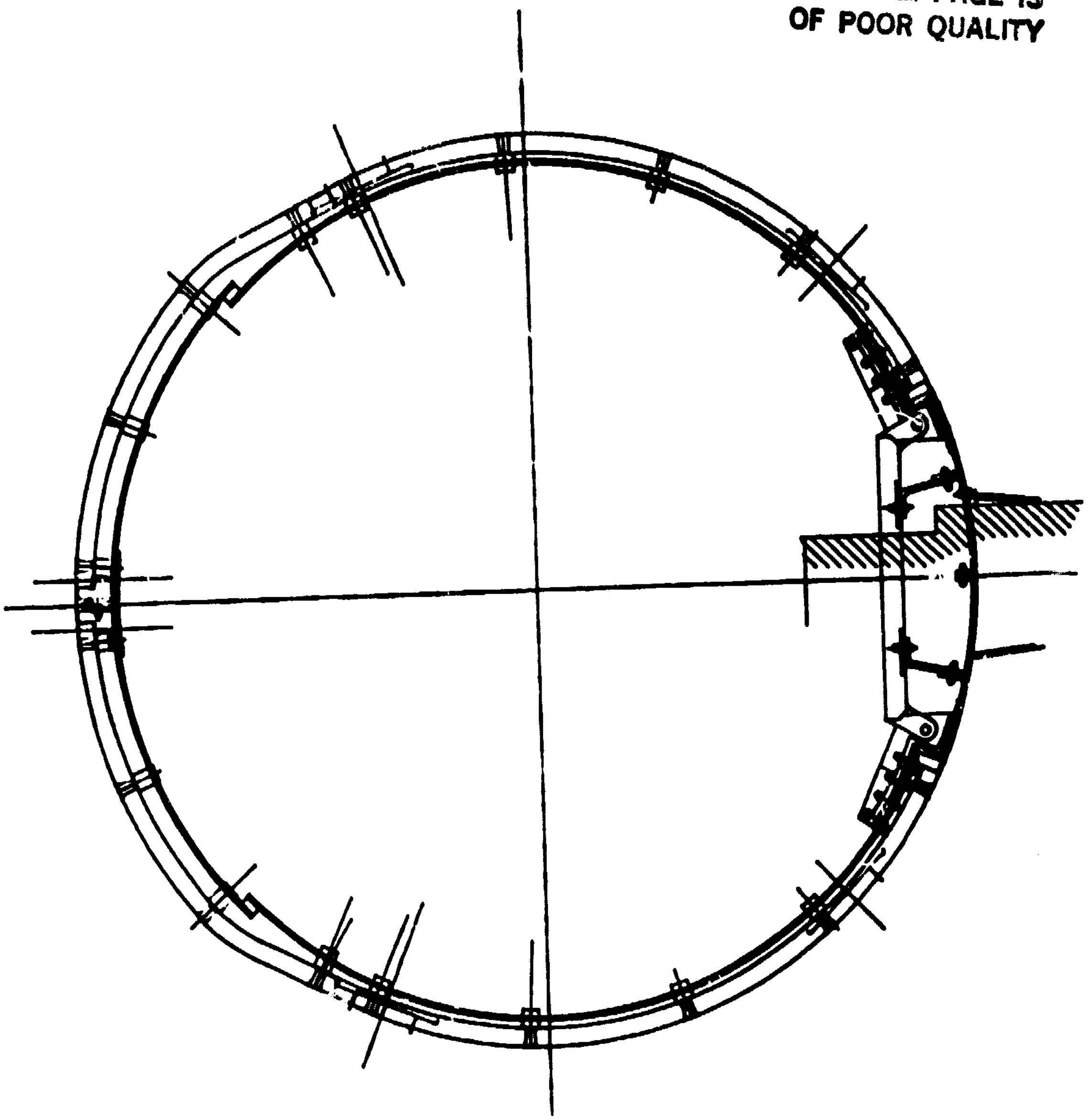

Figure 2.9-3. Core Cowl Doors - End View.

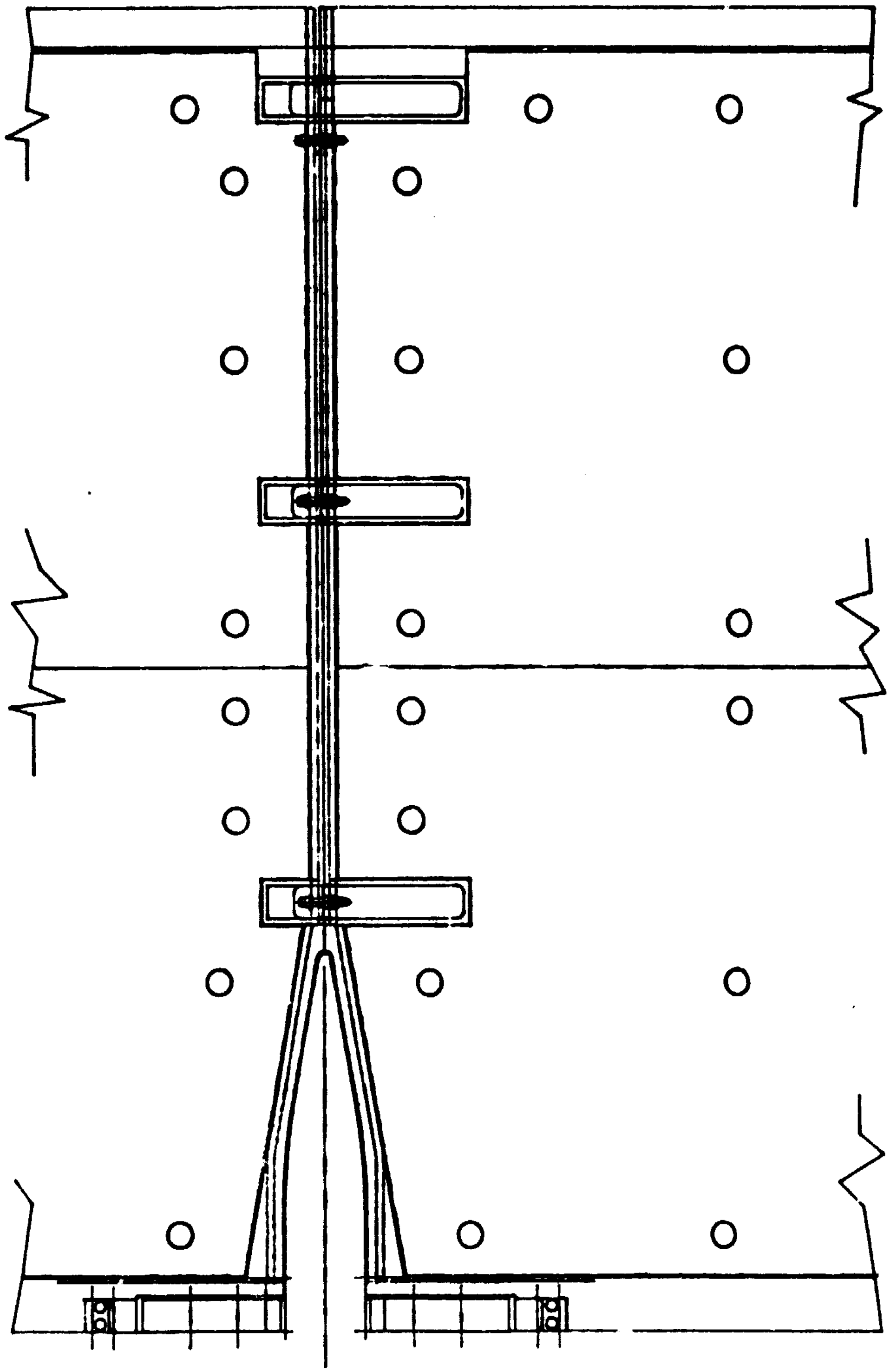

Figure 2.9-4. Core Cowl Doors - Bottom View.

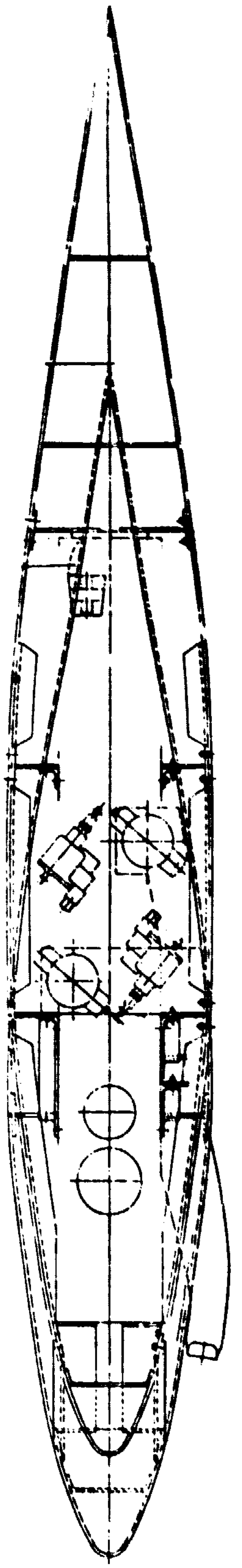

Figure 2.9-5. Pylon - Top View.

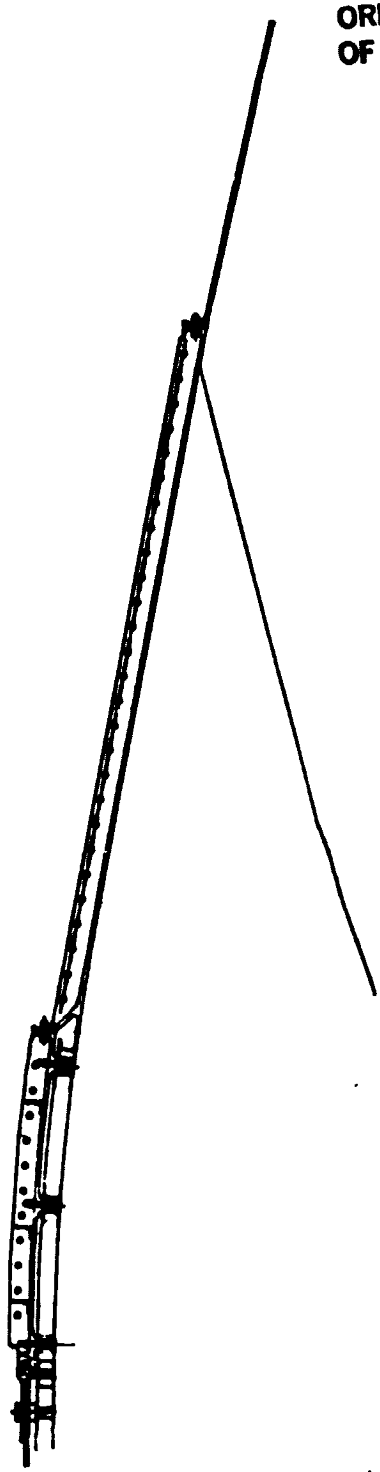

Figure 2.9-6. Fixed Fan Nozzle.

3.0 TASK 3 - CORE TESTING

Overall Objectives

- Design, fabricate, assemble, and test a core engine and obtain experimental evaluation of E^3 components operating as a system.

- Develop methods by which performance of the core can be measured as to its suitability as a core for the projected Flight Propulsion System.

- Evaluate performance and mechanical integrity of the core to identify changes required to meet program goals. Within program timing and cost constraints, incorporate design improvements identified from component and core testing into core and ICLS hardware.

Development Approach

The core engine will incorporate the individual components designed and tested in part or full scale in Task 2 (Component Analysis, Design, and Development). These components will include the high-pressure compressor; the combustor; and the high-pressure turbine, including clearance control devices and a control system adequate to permit starting, steady-state operation, and slow transients. The purpose of the core test will be to evaluate the performance, stability, and mechanical integrity of the components running together as a system, and to identify desirable changes for their incorporation into the ICLS or the Flight Propulsion System.

The core test vehicle will be assembled with extensive performance and mechanical instrumentation, including

- Gas path steady state total temperature and pressure rakes at compressor inlet, compressor exit, and turbine exit

- Rotor mechanical speed measurement

- Fuel flow measurement

- Compressor rotor and stator strain gage insrumentation based on FSCT results sufficient to monitor mechanical integrity

- Turbine rotor and stator strain gage instrumentation sufficient to monitor mechanical integrity

- Inlet airflow instrumentation

- Variable guide vane setting readout
- Measurement of compressor rotor tip clearances for Stages 3, 5, and 10
- Compressor interstage steady-state instrumentation, incorporated, as necessary, to determine compressor interstage conditions
- Measurement of HPT rotor tip clearances
- Parasitic flowpath instrumentation, included to help evaluate the design of sump venting and cooling systems, and to establish the levels of parasitic flows under actual operating conditions; direct measurement of parasitic flows will be done wherever such measurements are practical in the hardware configuration
- Temperature measurements of the compressor and turbine casings for evaluation of active clearance control effectiveness
- Temperature measurements of HPT rotor, stator, and other hot structures.

The core vehicle test program will be designed to exercise the components over a sufficiently wide range to cover operating requirements in the ICLS. This will include operation at SLS ambient inlet conditions and with inlet heat/ram operation. In addition to covering the ICLS operating range, the characteristics of the core engine will be explored up to the levels of corrected airflow (120 pps) required for Flight Propulsion System operation throughout its flight envelope. This will provide an important part of the basis for extrapolating ICLS performance to altitude conditions outside the range of ICLS testing.

Prior to testing, a core engine computer model will be constructed on the basis of component rig tests and core engine configuration assembly measurements. This model will represent the anticipated performance of the core engine and will be used in the generation of pretest predictions and in the analysis of test results.

Using the core engine computer model as a base, a data reduction program will be developed to analyze the measurements during testing. This will ensure consistency of calculation procedures between the core engine computer model and the data reduction program, and substantially facilitate the construction of a Status Performance model.

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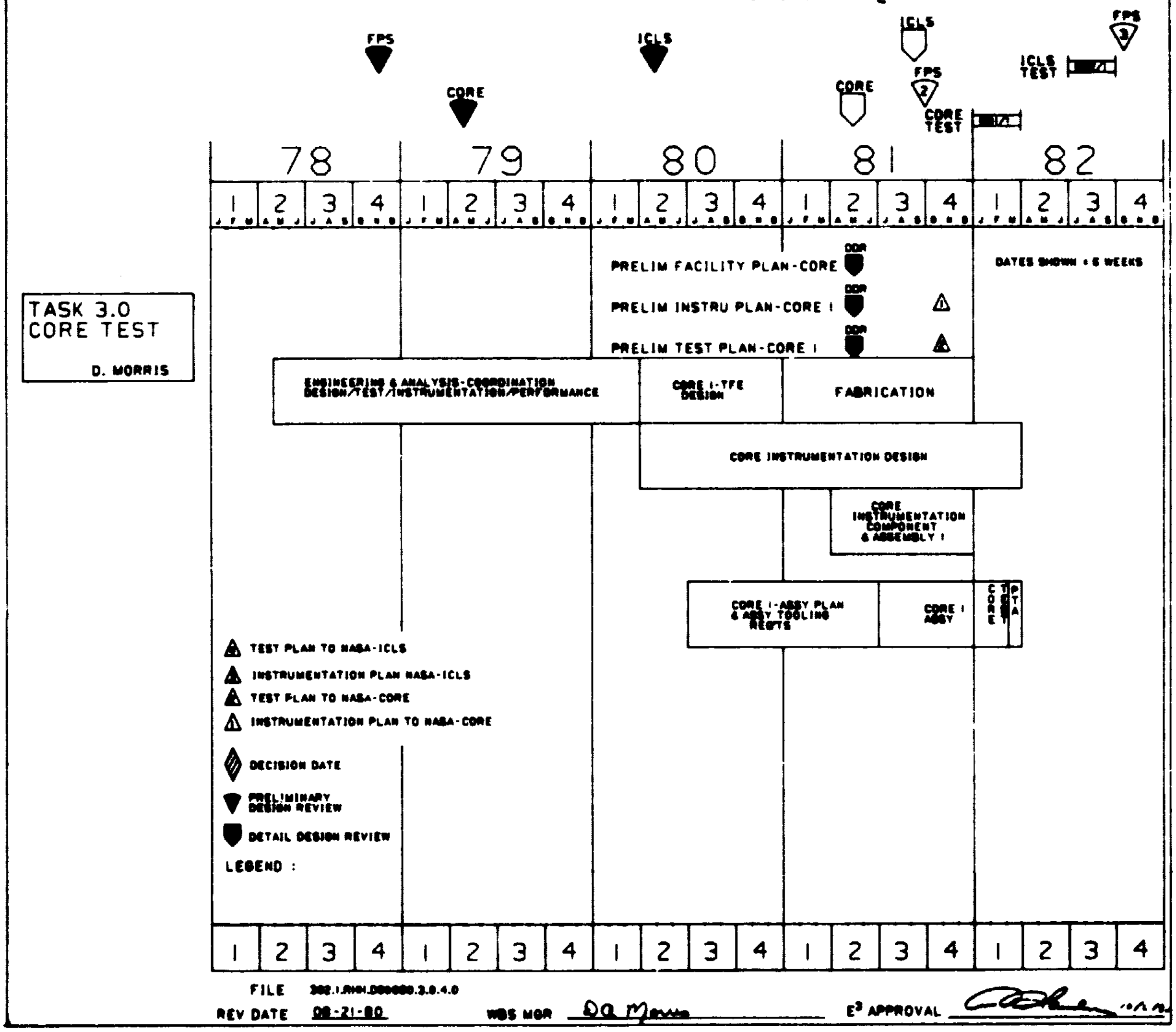

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As the engine test results are obtained, the performance of the individual components and the overall core system will be compared to the performance of the pretest prediction model. Deviations in performance will be identified and reasons for those deviations will be determined. As a result of this analysis, desirable changes will be identified for incorporation in the ICLS or the Flight Propulsion System design.

At the conclusion of the core test, a Core Test Status performance computer model will be established. This will be used to project ICLS performance on the basis of the core engine demonstrated performance. In conjunction with the ICLS test results plus projected improvements anticipated from further component development, the essential basis for Flight Propulsion System performance projections will be provided.

3.1.1 Core Engineering and Analysis

Technical Progress

During this reporting period, the primary task continued to be maintaining the coordination effort between Design Engineering and the contributing support organizations (Instrumentation Design, Development Assembly, and Test Facility Engineering). The majority of the work effort was in the following areas:

- Definition of preliminary test plan
- Definition and refinement to the assembly and instrumentation plans
- Refinement of the assembly schedule
- Monitoring hardware design for assembly and maintainability considerations
- Defining and documentating vehicle and facility interfaces and hardware responsibilities
- Supplying hardware rework definition for the application and lead-out of instrumentation
- Definition of in-assembly machining operations required to maintain engine clearances
- Definition of rotor balance procedures and tooling requirements

- Initiating work on subassembly buildup procedures
- Initiating design activities for instrumentation design, test facility hardware, and assembly tooling
- Initiating core interface drawing.

Work Planned

- Major emphasis will be to complete the major subassembly buildup procedures, design, and begin procurement of assembly tooling and initiate the detailed design for instrumentation application
- The actual application of instrumentation to engine hareware and the assembly of the core will begin
- The detailed design of test facility hardware will be initiated along with the procurement of long lead time items
- Coordination and monitoring effort between Design Engineering and the contributing support organizations will continue.

3.1.2 Core Instrumentation and Assembly

3.1.2.1 Instrumentation Design

Technical Progress

The major work effort during the past 6 months was in defining methods for the application and leadout of instrumentation sensors and associated vehicle hardware rework requirements. These preliminary design activities were in the following areas.

- Defined compressor rotor leadout paths, rework requirements, and instrumentation leadout duct design
- Defined an improved method for routing and sealing compressor aft stator case leads; this will help alleviate the assembly problems encountered on the 1-10 compressor test vehicle
- Defined mounting and sealing provisions for all rakes
- Work continued on defining application and leadout provisions for the combustion case, turbine stator, turbine rotor, and rear frame subassemblies

- Selected accelerometer locations and types
- Began layout of compressor traverse probes and their installation
- Provided definition to Design Engineering as required for hardware design releases.

Work Planned

- Complete definition of application and lead routing techniques for the remaining vehicle components
- Begin detailed design for all rakes and probes plus initiate and complete application drawings as required to meet the vehicle assembly schedule.

3.1.2.3 Core Assembly

Technical Progress

Initiated assembly planning work efforts to provide written assembly procedures. Procedures for the Stage 1 HPT nozzle assembly have been completed and reviewed by Design Engineering. Assembly procedures for the HPT shroud, combustion case, and front frame subassemblies are in progress. Preliminary designs for HP turbine rotor tooling have been submitted by Design Engineering, and are currently being reviewed. The torquing wrench for mating the high pressure rotor main joint has been designed and is in the process of being procured for an integrity test. The detail design for the compressor EROM and vehicle balance tooling has been initiated.

Work Planned

- Provide core vehicle subassembly written procedures as required to meet the assembly schedule
- Procure main rotor joint torquing tool and perform an integrity test to verify its acceptance for use on the core
- Initiate design of assembly and balance tooling and procure tooling as required to meet the assembly schedule.

3.1.2.4 Core Signal Conditioning

Technical Progress

Completed design of the HP turbine rotor optical clearance probes and have released drawings for quotes. Modifications to the engine level assembly drawing to show probe installations are in process. All long lead time electrical components have been placed on order.

Work Planned

- Procure material and begin fabrication of clearance probes
- Begin work on signal conditioning design
- Assist in the redesign of the CDP air cooling cart; this cart is required to cool engine CDP air before being used as purge air for the clearance probes; modifications to the cart may be required to completely satisfy the core vehicle's requirments.

3.1.3 Core Test Facilities Engineering

Technical Progress

Preliminary design of the cell inlet and air supply ducting was initiated. Much of the existing cell ducting can be utilized for the core test including the main air filtration and flow meauring hardware. Design of the slave stator actuation system continued during this reporting period. Interferences between instrumentation traverse probes on the compressor stator case and the actuators have been defined and are currently being resolved. All engine-to-cell interfaces and slave systems have been defined.

Work Planned

- Complete the preliminary design and begin the detailed design of facility hardware
- Place all long lead time material and hardware on order
- Review requirements for the core optical clearance probe cooling cart and define what modifications to the existing cart are required.

4.0 TASK 4 - ICLS TESTING

Overall Objectives

- Design, fabricate, assemble, and test a turbofan demonstrator engine and obtain experimental evaluation of E^3 components operating as a system
- Develop methods by which performance of the turbofan demonstrator engine can be measured as to its suitability toward the projected Flight Propulsion System
- Evaluate performance and mechanical integrity of the turbofan demonstrator engine to identify changes required to meet program goals.

Development Approach

The turbofan engine will incorporate the individual components designed and tested in part or full scale in Task 2 (Component Analysis, Design, and Development). These components will include the high-pressure compressor, the combustor, the high-pressure turbine, the fan, the low-pressure turbine, and the mixer including clearance control devices and a control system adequate to permit starting, steady-state operation, and slow transients. The current plan is to use the applicable core test vehicle and full-scale fan test vehicle component hardware in the assembly of the ICLS. This assumes that there will be no major hardware modifications required as a result of the individual component tests. The purpose of the ICLS test will be to evaluate the performance, stability, and mechanical integrity of the components running together as a system, and to identify desirable changes for the Flight Propulsion System.

The turbofan test engine will be assembled with extensive performance and mechanical instrumentation, including:

- Gas path steady-state total temperature and pressure rakes at the fan inlet, compressor inlet, compressor exit, HP turbine exit, LP turbine exit, and mixer exhaust
- Thrust
- Rotor mechanical speed measurements
- Fuel flow measurement

- Fan rotor and stator strain gage instrumentation based on FSFT results sufficient to monitor mechanical integrity

- Low pressure turbine rotor and stator strain gage instrumentation sufficient to monitor mechanical integrity

- Inlet airflow instrumentation

- Variable guide vane setting readout

- Combustor static pressure (control pressure) measurement

- Compressor interstage static pressure instrumentation; incorporated, as necessary, to determine compressor interstage conditions

- Direct measurement of HPT rotor tip clearances

- Parasitic flowpath instrumentation, included to help evaluate the design of sump venting and cooling system, and to establish the levels of parasitic flows under actual operating conditions; direct measurement of parasitic flows will be done wherever such measurements are practical in the hardware configuration

- Measurements of baseline and fully suppressed acoustical characteristics

- Exhaust emissions measurement

- Temperature measurements of the compressor, HPT, and LPT cases sufficient to evaluate active clearance control effectiveness

- Temperature measurements of LPT rotor, stator, and other hot structures.

The ICLS test program will be designed to test the components over the entire range of ICLS operating conditions. All testing will be at ambient inlet conditions. In addition to covering the ICLS operating range, the characteristics of the core engine will be explored up to the levels of corrected airflow (120 pps) required for Flight Propuslion System operation throughout its flight envelope. This will provide an important part of the basis for extrapolating ICLS performance to altitude conditions outside the range of ICLS testing.

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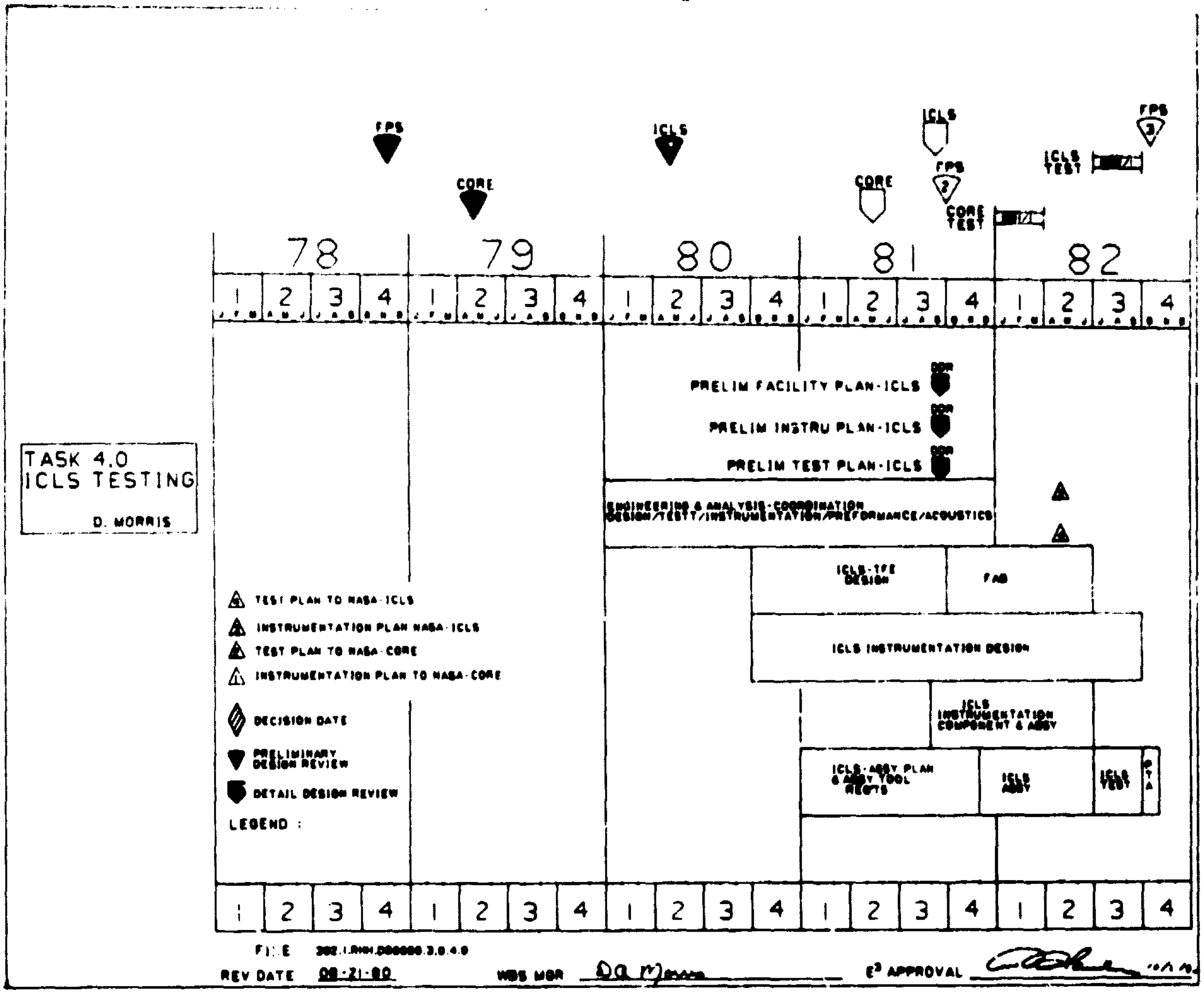

388A

Prior to testing, an ICLS computer model will be constructed on the basis of component rig tests, core engine test, and the ICLS configuration assembly measurements. This model will represent the anticipated performance of the ICLS engine and will be used in the generation of pretest predictions and in the analysis of test results.

Using the ICLS computer model as a base, a data reduction program will be developed to analyze the measurements during testing. This will ensure consistency of calculation procedures between the ICLS computer model and the data reduction program, and substantially facilitate the construction of a Status Performance model.

As the engine test results are obtained, the performance of the individual components and the overall engine system will be compared to the performance of the pretest prediction model. Deviations in performance will be identified and reasons for those deviations will be determined. As a result of this analysis, desirable changes will be identified for the Flight Propulsion System design.

4.1.1 ICLS - Preassembly Engineering and Analysis

Technical Progress

During this reporting period, the primary task continued to be the definition and engineering coordination effort between Design Engineering and the contributing support organizations. The majority of the work effort was in the following areas:

- Definition of preliminary assembly, instrumentation, and test facility plans
- Initiation of facility mount design; coordinated definition of engine mount link loads and cell to engine interfaces
- Monitoring progress of the full scale fan test vehicle hardare and instrumentation designs to assure compatibility with ICLS
- Monitoring hardware design for assembly, maintainability, and installation considerations
- Supplying hardware rework definition for the application and leadout of instrumentation

- Definition and documentation of engine-to-facility interfaces
- Definition of LP turbine rotor balance and assembly procedures
- Definition of in-assembly machining procedures required to maintain engine clearances.

Work Planned

- Continue with the coordination and definition effort between Design Engineering and the support organizations
- Initiate instrumentation design for the fan and LP tubine subassemblies
- Continue to refine the assembly and instrumentation plan
- Define cowling rework for instrunmentation application
- Define main engine level installation of LP turbine module procedure.

4.2.2.1 LPT Rotor Hardware

Technical Progress

Blades

Details on the blades are reported in Section 2.5.6.1.

Rotor

All five major disks have been rough machined, ultrasonically inspected, and macroetched. They will be final machined after receipt of tooling required for machining. A significant cost savings has resulted due to machining the dovetails by wire EDM rather than broaching.

Forgings for the LPT rotor seals have been received and quotes for their machining are significantly below plan.

Work Planned

- Place orders for blade machinings
- Initiate machining LPT rotor seals and continue machining the disks.

4.2.2.2 LPT Nozzle Hardware

Technical Progress

Nozzles

Details on the nozzles are reported in Sections 2.5.6.2 and 2.5.6.3.

Other Hardware

All necessary forgings for the LPT seal supports are available.

Machining drawings for the forward inner seal support and aft inner seal support have been issued and released to Development Machining Operation (DMO) for planning. Several changes were made to the aft inner seal support design to facilitate fabrication, including changing two brazed joints and two EB-welded joints to four TIG-welded joints. Additional minor changes to these drawings are now being reviewed.

Work Planned

- Complete nozzle casting tooling
- Cast initial nozzle pieces.

4.2.2.3 LPT Casings

Technical Progress

Forgings for the outer duct support are available for machining.

The outer duct support machining drawing has been issued and has been released to DMO for planning and machining. Several minor changes to the drawing are now being reviewed. All miscellaneous hardware including nuts, bolts, washers, rivets, etc. are on order.

Casing

Fabrication planning is continuing in-house for the LPT casing. In-process heat treat required prior to machining has been completed. Rough machining of the separate forward and aft shells is scheduled for April, to be followed by welding of the shells and final machining.

LPT Cooling/Active Clearance Control Impingement Manifold

Detail drawings for the LPT cooling manifold have been issued, planned funding has been released, and quotes have been requested. All quotes have not yet been received, but of those returned, the low bid is below the planned funding.

Drawings for all the manifold mounting harware have been issued. W/A's to release funding for procurement of these miscellaneous mounting hardware parts are in process.

Work Planned

- Place all machining orders
- Continue fabrication of casing
- Initiate fabrication of outer duct support.

4.2.2.4 LPT Shrouds and Seals

Technical Progress

The inner and outer transition duct casting drawings have been issued and orders have been placed for the castings. The duct machining drawings have been issued and released to DMO for machining. DMO has reviewed the drawings and minor changes are planned to facilitate manufacture. All spline and hourglass seal drawings (seals which fit between nozzle segments) have been issued and released for quotations.

LPT Shrouds Stages 1 through 5

All five shroud fabrication drawings have been issued and have been sent out for quotes; planned funding has been released for procurement.

Insulation Blankets for LPT

Detail drawings have been issued defining all the insulation blankets in the LPT. Locations are inner and outer transition duct between HPT and LPT, over all five stages of the LPT shrouds, and over the nozzles in Stages 2 through 5. Planned funding has been released for procurement of all the insulation blankets. These parts are now released for quotes.

Work Planned

- Place remaining hardware on order
- Incorporate manufacturing requested changes into the duct machining drawings.

4.2.3 Bearing, Systems, Drives, and Configuration Fabrications

Technical Progress

All drawings for the ICLS forward and aft sumps have been issued with the exception of the LPT shaft and the No. 5 bearing housing. Material for these two major parts is on order and shaft material should be available early in the second quarter of 1981. Housing material should be available by mid-1981.

All sump hardware with the exception of the shaft and housing are either out for quote or on order.

Work Planned

- Place LPT shaft and No. 5 bearing housing on order
- Continue procurement of other hardware.

4.4 ICLS TEST FACILITY ENGINEERING

Technical Progress

Began work on preliminary design of ICLS/facility mount system. Definition of forward mount brackets, pickup coordinates, link loads and links have been completed. Aft mount link design is still in process. Also working on cowling interfaces. The inlet-to-fan frame soft mount has been defined. Working with Design Engineering to define cowling support requirements.

Work Planned

- Complete facility mount design
- Define facility slave service requirments for ICLS
- Define cooling cart requirements for turbine optical clearance probes and outdoor testing.

EXHIBIT B - QUALITY

Technical Progress

During this reporting period, the following has been accomplished:

- Final review of the E^3 Preliminary Hazards Analysis was completed and the Hazards Analysis was issued. The hazards analysis concluded that, "based on the configuration analyzed, no probable malfunction, single or multiple failures, nor improper malfunction will cause the probability of occurrence of a hazardous 'undesired event' to exceed an acceptable rate."

- Components were selected for review of Quality Plan requirements and these Quality Plans were requested from the component manufacturer. To date, approximately 60% of the Quality Plans for the components selected have been reviewed by Engineering and found to be adequate.

- Assembly of the 1-10 Compressor Test Rig was completed, documentation was reviewed and the assembly was shipped to GE-Lynn for test. The installation in the GE-Lynn Compressor Test Facility including the Test Control and Data Center was reviewed. All systems were found to be acceptable for this test.

- Participated in Component design, and program and hardware reviews throughout the reporting period.

Work Planned

- Monitor test of 1-10 Compressor Rig

- Review and monitor the buildup and test of the E^3 fan at the Fan Test Facility at GE-Lynn

- Continue to obtain Quality Plans for selected components for Engineering Review

- Continue to monitor and review all aspects of the programs participating in program and design reviews to assure compliance with the Quality Assurance Program Plan.