NASA
7N-07-CR
73741

Contract NAS 3-20629

p-335

Jet Engine Performance Improvement — Program Review — June 28, 1979

(NASA-CR-190146) JET ENGINE PERFORMANCE IMPROVEMENT. PROGRAM REVIEW: 28 JUNE 1979 (GE) 335 p

N92-70461

Unclas
Z9/07 0073741

Prepared For:
National Aeronautics & Space Administration
NASA - Lewis Research Center
Cleveland, Ohio 44135

GENERAL ELECTRIC

Aircraft Engine Group
Cincinnati, Ohio 45215

Introduction

Component Improvement Concepts on Contract

- ✓ Fan
- ✓ Short Core Exhaust
- HP Turbine Aerodynamic Improvement
- ✓ Front Mount
- HP Turbine Roundness Control
- HP Turbine Active Clearance Control
- LP Turbine Active Clearance Control

✓ Completed

Discuss

- Description
- Payoff
- NASA Program
- Summary

CF6 Jet Engine Performance
Improvement Program

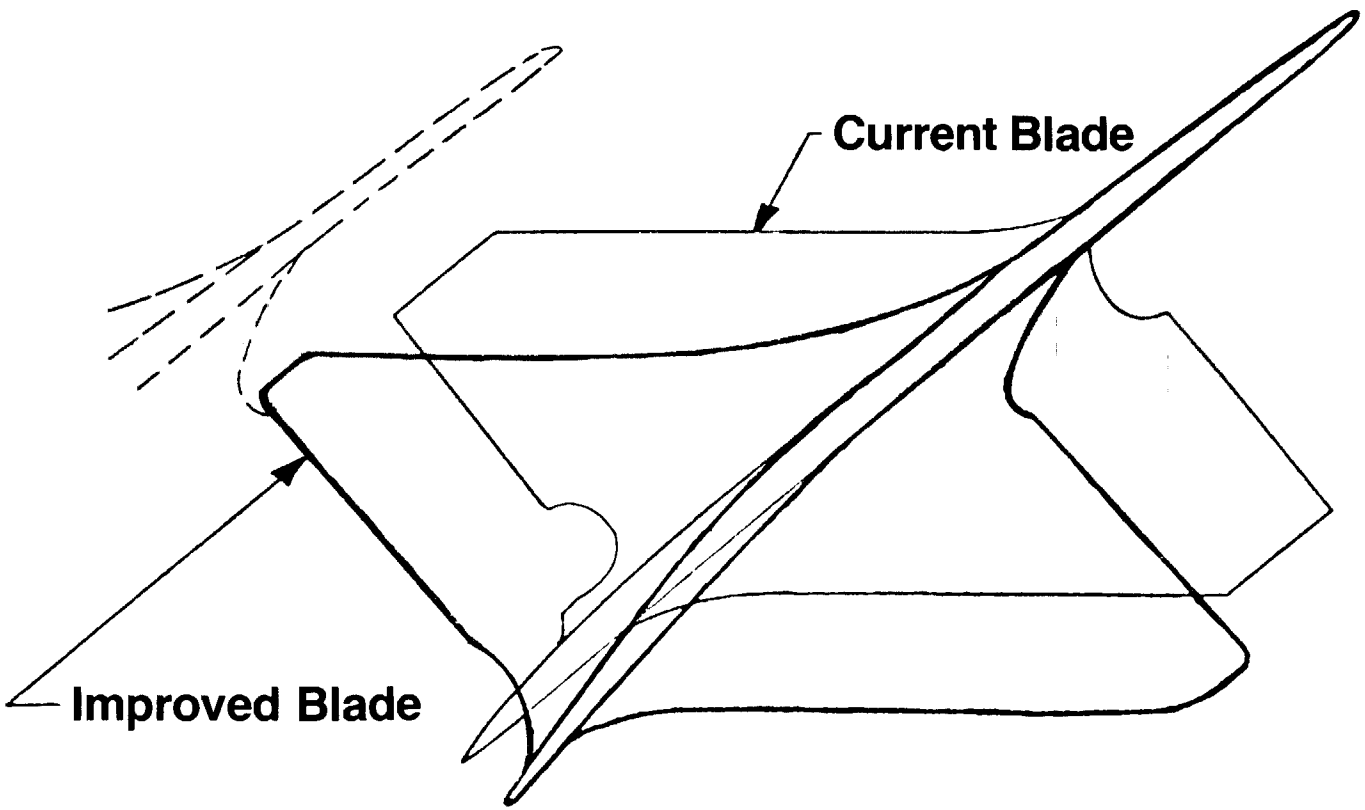
Task 2.1
Fan Performance Improvement Program

Fan Performance Improvement Concept

- Aerodynamic Redesign of Fan Blade
  - Camber & Stagger Changes
  - Reduce Part-Span Shroud Blockage
- Decrease Running Clearance
  - Stiffener Ring to Detune Case
- Optimize Operating Line for Cruise
  - Open Fan Nozzle Area (A_{18})
    (Deleted Because of No Payoff)
- Closed VSV Schedule
  (Added to Increase Performance)
- Smooth Tip Shroud
  (Added to Increase Performance)

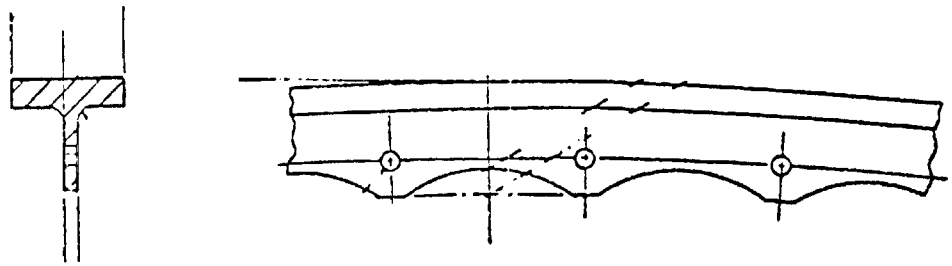
This top view of the part span shroud shows the current CF6 fan blade and its shroud position superposed on the improved fan blade. It can be seen that the shroud is further aft and the airfoil has more camber than the current blade.

Fan Blade Comparison

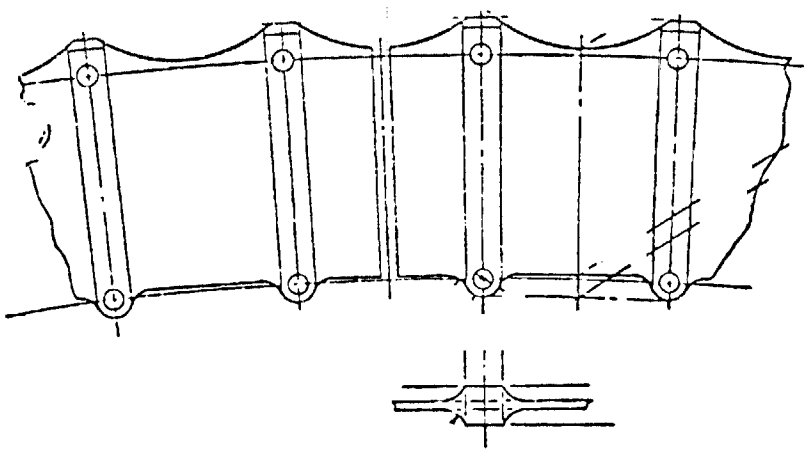


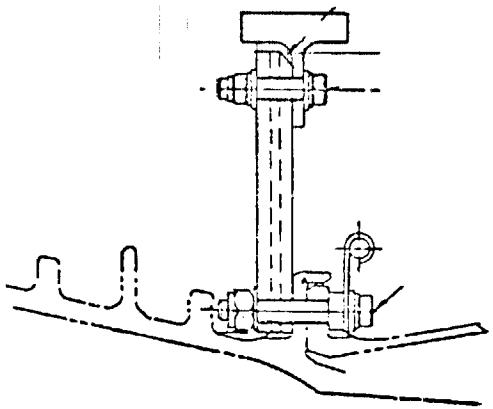
The stiffener consists of 5 web assemblies shown in the lower left figure and full 360° "T"-shaped flange bolted to the top of the web. This flange is shown in the top figure.

CF6-50 Fan Case Ring Stiffener



360° Ring





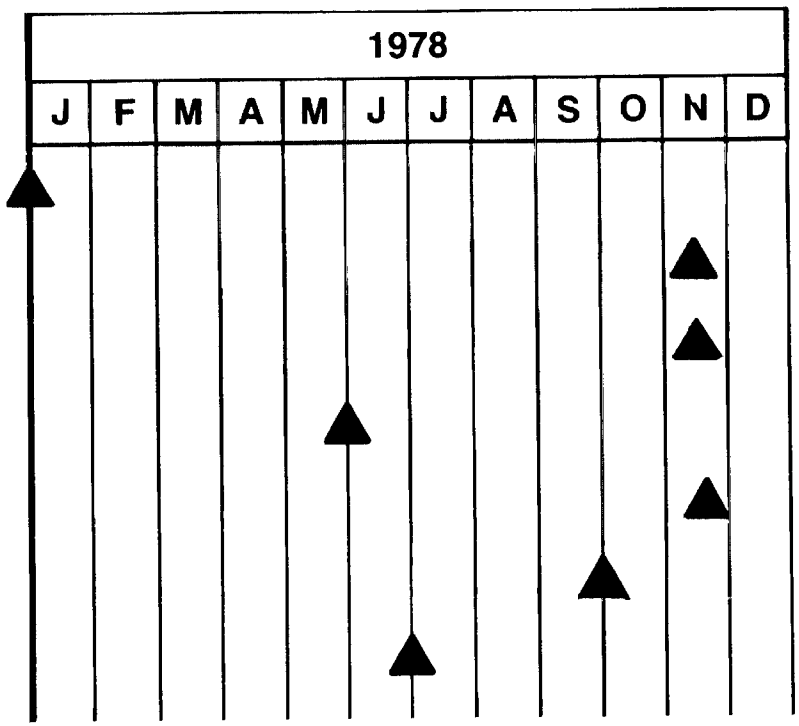
Segmented Web

This chart shows objective and demonstrated cruise SFC improvements. The increase in fan nozzle exit area (A18) did not pay off and was deleted from the fan package. With the revised variable stator vane (VSV) schedule the cruise SFC improvement for the fan package exceeds the objective.

Item	Improvement % △ Cruise SFC		Equivalent % △ SFC At SLS, $F_n = 40{,}000$	
	Objective	Demonstrated	Objective	Demonstrated
Improved Fan Blade	– 1.0		– 1.9	
Reduced Fan Clearance	– 0.6		– 0.8	
Increased A18	– 0.2	Deleted	0	Deleted
Total	– 1.8	– 2.0[1)]	– 2.7	– 2.9[1)]

[1)] Including Closed VSV Schedule and Smooth Casing Tip Shroud

Program Plan
Fan Performance Improvement



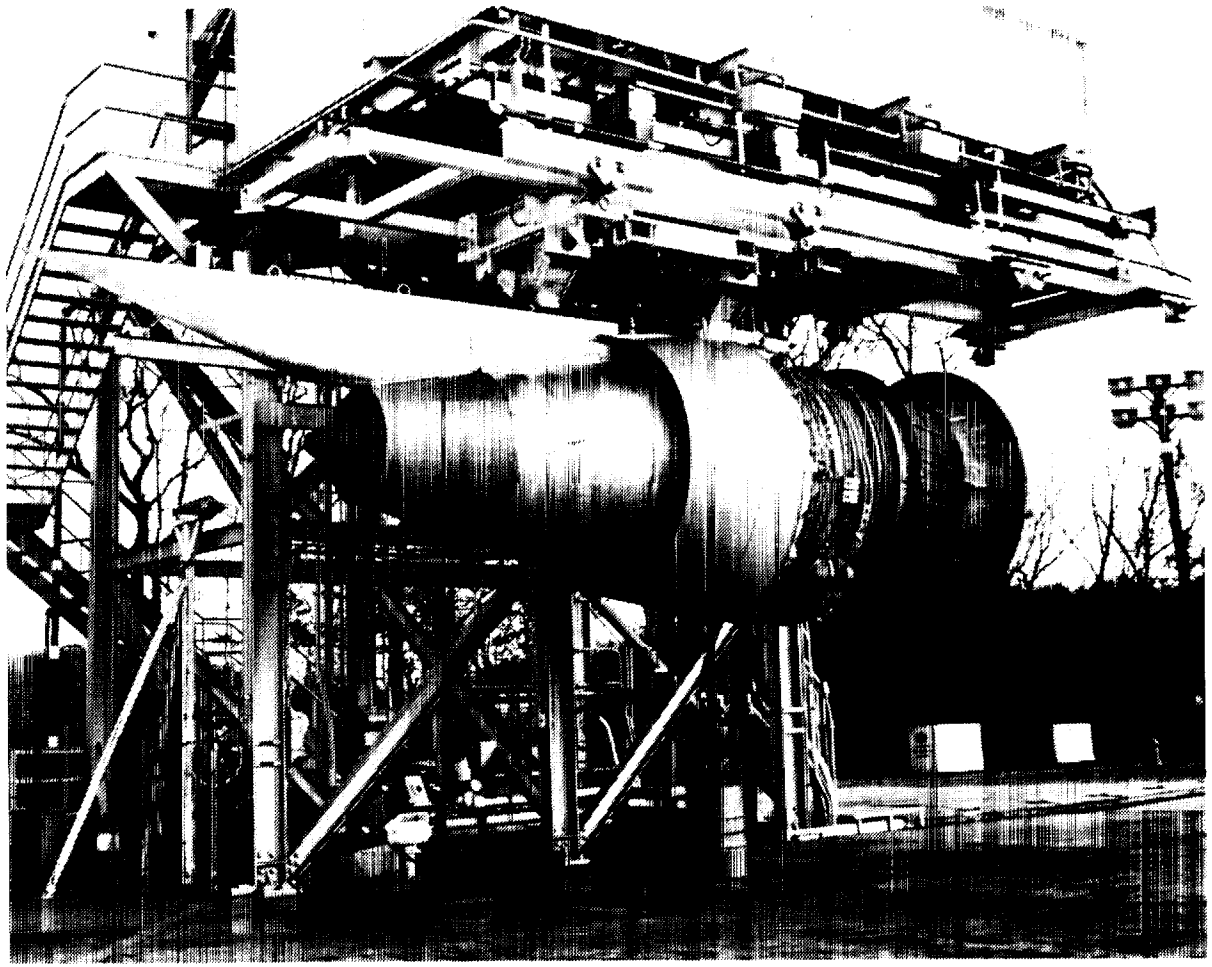
Performance Test

- **Objective**
  - Determine Performance Improvement in Back-to-Back Engine Ground Test
- **Status:**
  - Test Completed, Reported in Last Oral
- **Results:**
  - Fan Provides Expected Efficiency Improvement
  - SFC Improvement Exceeds Expectations
  - Insignificant Performance Improvement for Open A18
  - Incroporated Closed VSV Schedule and Smooth Casing Tip Shroud to Improve Performance
  - Successful Altitude Cruise Simulation with Diffusers

Power Management Test

- **Objective**
  - To Define Full Scale Flight Configuration Nozzle Coefficients for In-Flight Thrust Determination.
  - Determine Flight Configuration Thrust VS. Fan Speed Characteristics.
- **Status**
  - Test Completed

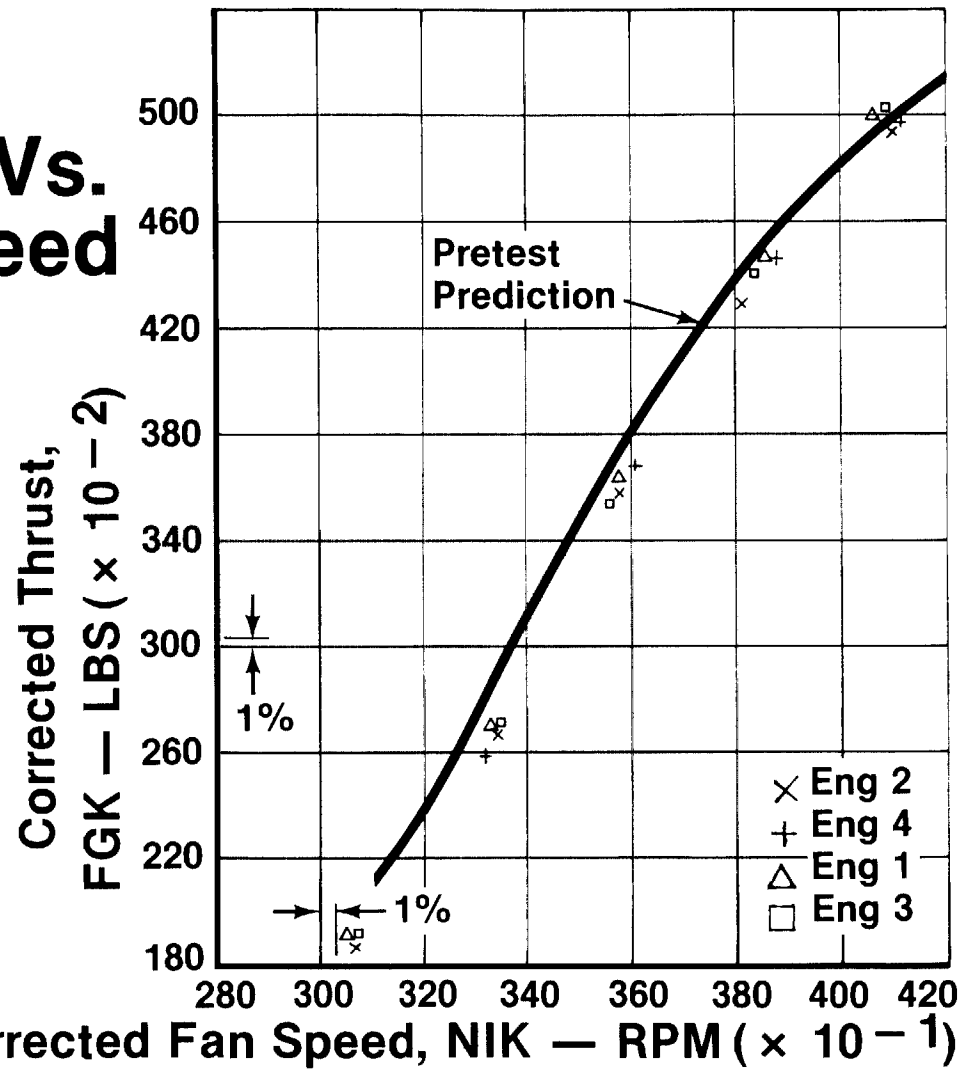
Test Site IVD



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Corrected Thrust Vs. Corrected Fan Speed

M = 0.85

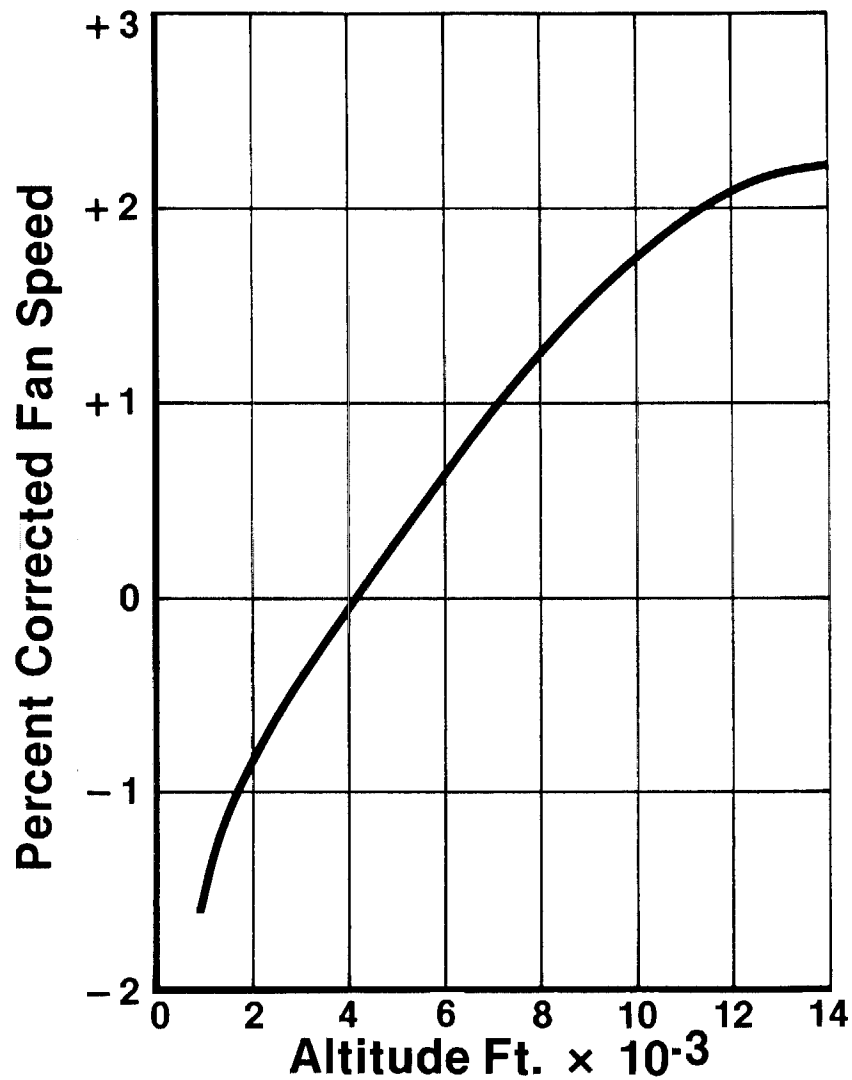


PRECEDING PAGE BLANK NOT FILMED

The difference in thrust vs. N1 characteristics with the improved fan blade necessitates a significant change to power management fan speeds.

Takeoff Power Management Difference

Advanced Fan Vs. Base Engine



Conclusions

- Power Management Defined for Advanced Fan on CF6-50 Engine for DC-10-30, 747-200 and A300B

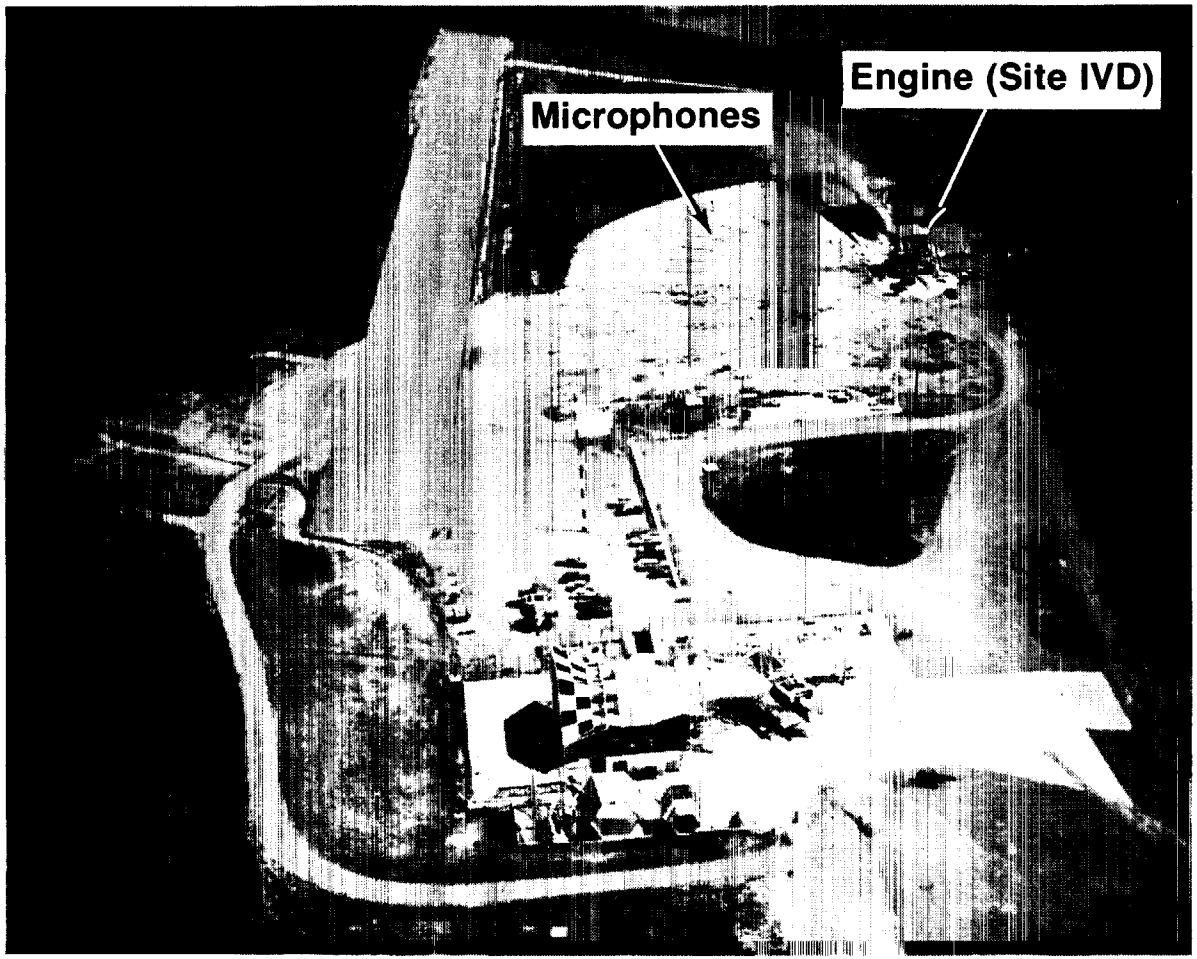
Acoustic Testing

- **Objectives**
  - To Establish the Acoustical Effects of the Fan Performance Improvement Package on CF6-50 Engine Noise
  - To Assess the Impact of the Subject Engine Modification on Community Noise Levels for Typical Aircraft Approach and Takeoff Flight Conditions
- **Status**
  - Test Completed

Test Program

- Back-to-Back Testing of Improved Fan Configuration and the Current Production Fan Configuration
- Testing of Reduced Tip Clearance and Smooth Shroud
- All Testing Performed at the Peebles Outdoor Test Facility

Noise Test



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Test Procedure

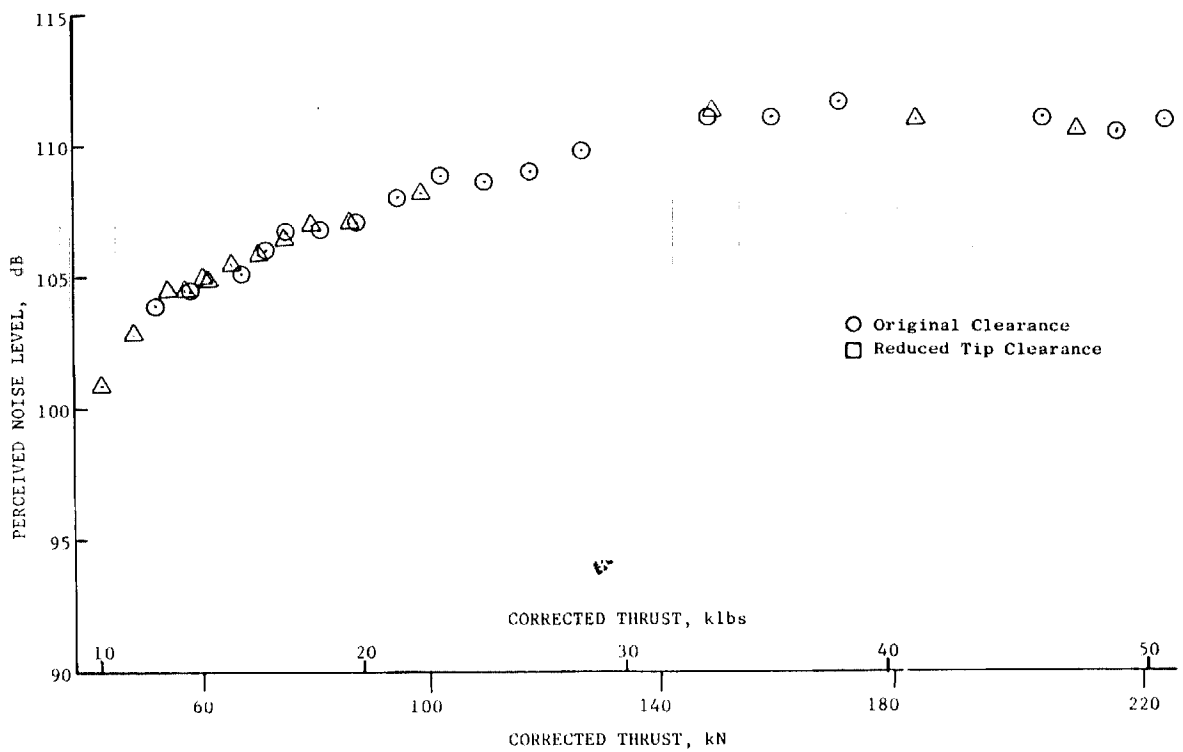
- 21 Far Field Microphones Located on a 150 Ft Arc, 13.0 Ft. Above Pad (Engine ℄ Height)
- Microphones Positioned at 10°-160°, 10° Spacing Additional Microphones Positioned at 85°, 95°, 105°, 115°, and 125°
- 19 Fan Speed Points, Minimum of Two Repeats
- Data Obtained at Equivalent Thrust for Production and Advanced Fan Engine Configurations

Results

- **Reduced Fan Blade Tip Clearance**
  - No Significant Changes in Engine Noise Spectra
  - Perceived Noise Level Comparisons as a Function of Thrust Exhibit Acoustic Equivalency

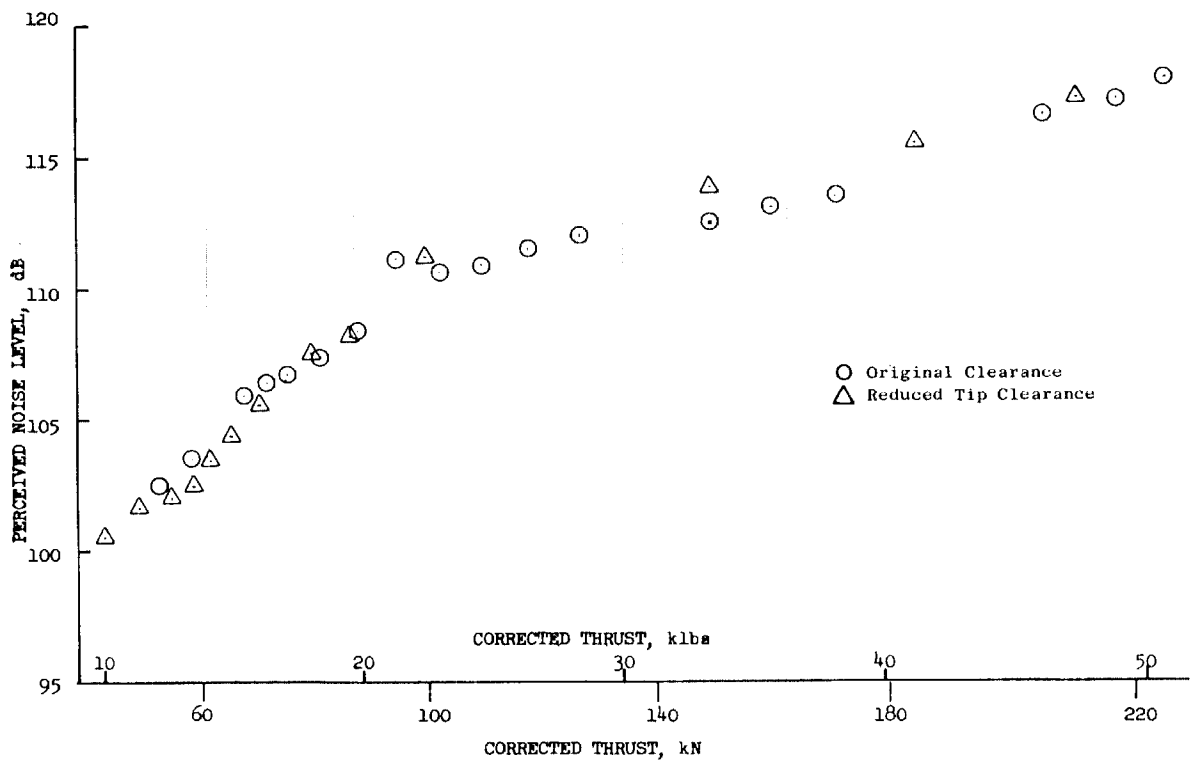
Data were averaged and extrapolated to a 400 ft. sideline.

Influence of Reduced Fan Blade Tip Clearance on PNL for 60° at 122 m Sideline



Data were averaged and extrapolated to a 400 ft. sideline.

Influence of Reduced Fan Blade Tip Clearance on PNL at 120° at 122 m Sideline



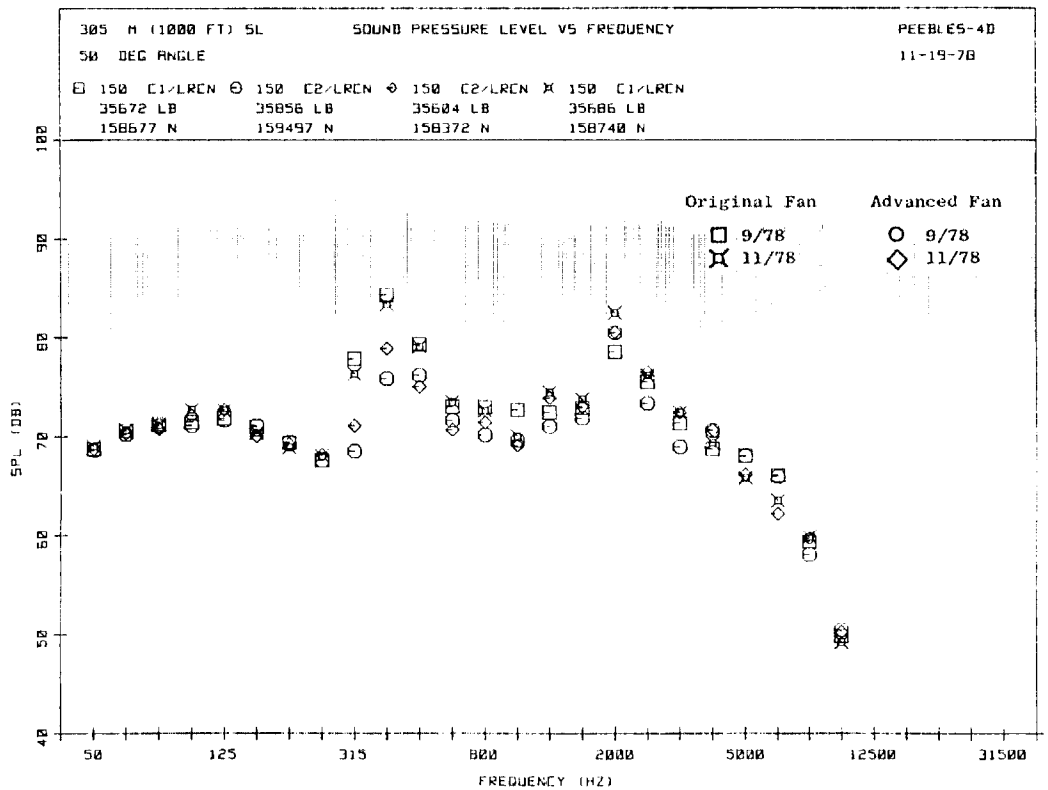
Results

- **Advanced Fan Blades**
  - Significant Reduction of MPT's (Buzz Saw Noise)
  - No Significant Changes in Engine Noise Spectra Other Than MPT's
  - No Significant Changes in Perceived Noise Level Directivity and Behavior as a Function of Thrust

Reduction of buzz-saw noise is most pronounced in the 315-500 Hz 1/3 octave bands.

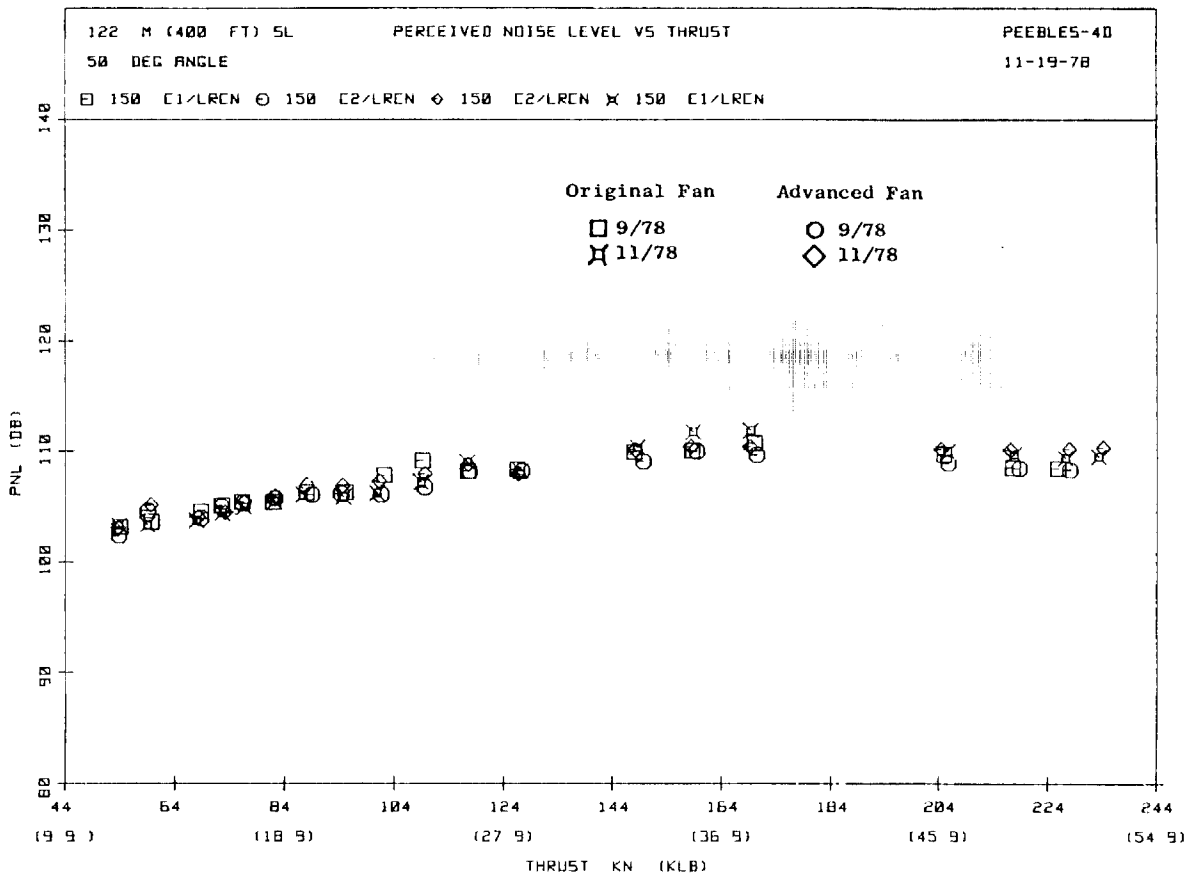
One-Third Octave Spectrum Comparison at 50°

(305 m Sideline and 158 kN Corrected Thrust)



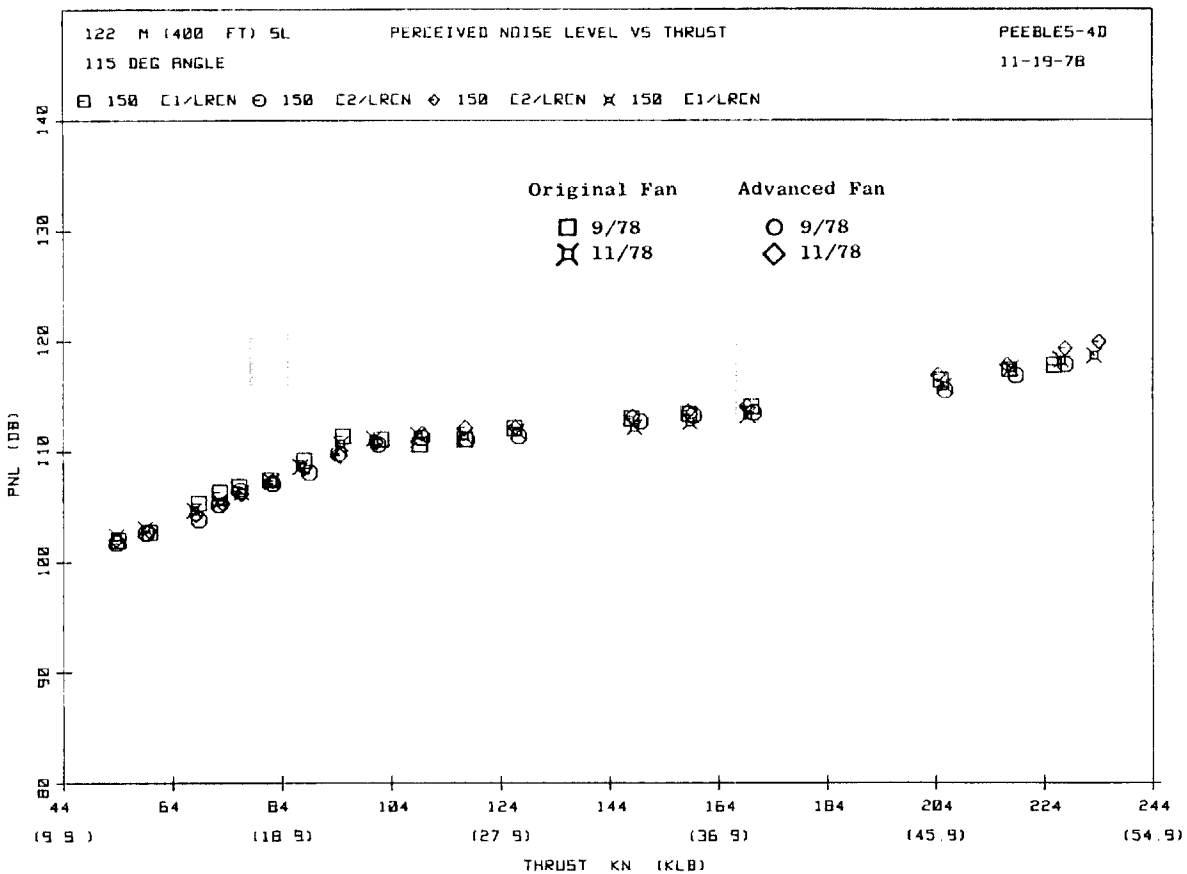
Data were extrapolated to a 400 ft. sideline and averaged.

Perceived Noise Level Directivity Comparison for 50° at 122 m Sideline



Data were extrapolated to a 400 ft. sideline and averaged.

Perceived Noise Level Comparison For 115° at 122 m Sideline



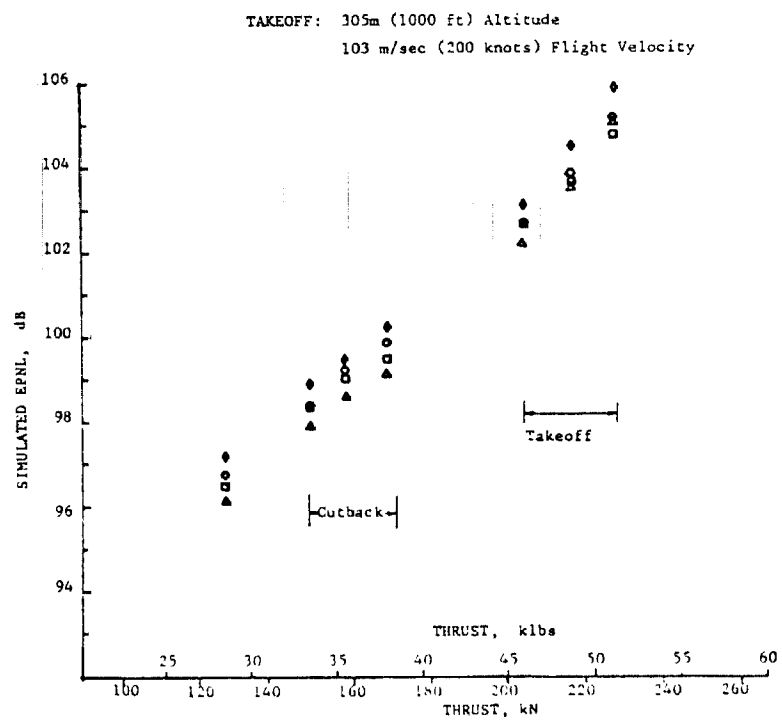
Community Noise Impact

- Simulated EPNL Comparisons as a Function of Thrust
- Results Show No Significant Change

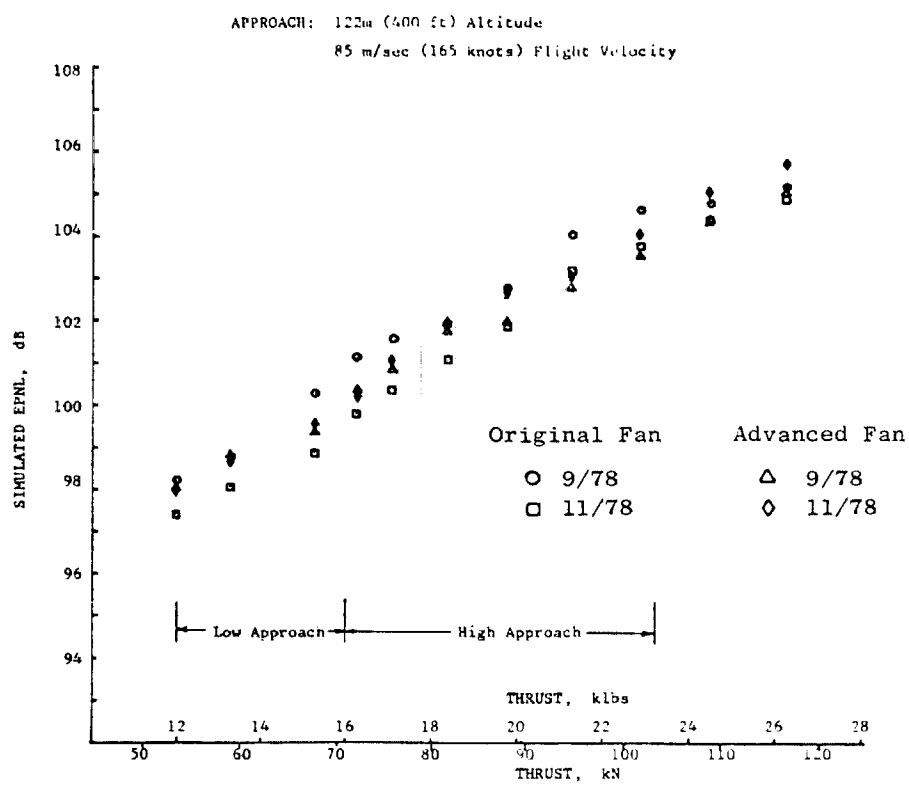
Simulated EPNL were computed by assuming that static PNL data were measured flight levels recorded at typical aircraft velocities. No flight effect corrections were applied to the data. The differences between the fan configuration data are not regarded as significant effects.

Noise Comparison at Takeoff

Original Fan	Advanced Fan
○ 9/78	△ 9/78
□ 11/78	◊ 11/78



Noise Comparison at Approach



Conclusions

- Advanced Fan in CF6-50 Engines Results in Equivalent Community Noise Levels
- Significant Reduction in Buzz-Saw Noise
- FAA Has Accepted the Acoustic Equivalency of the Advanced Fan with the Current Production Fan

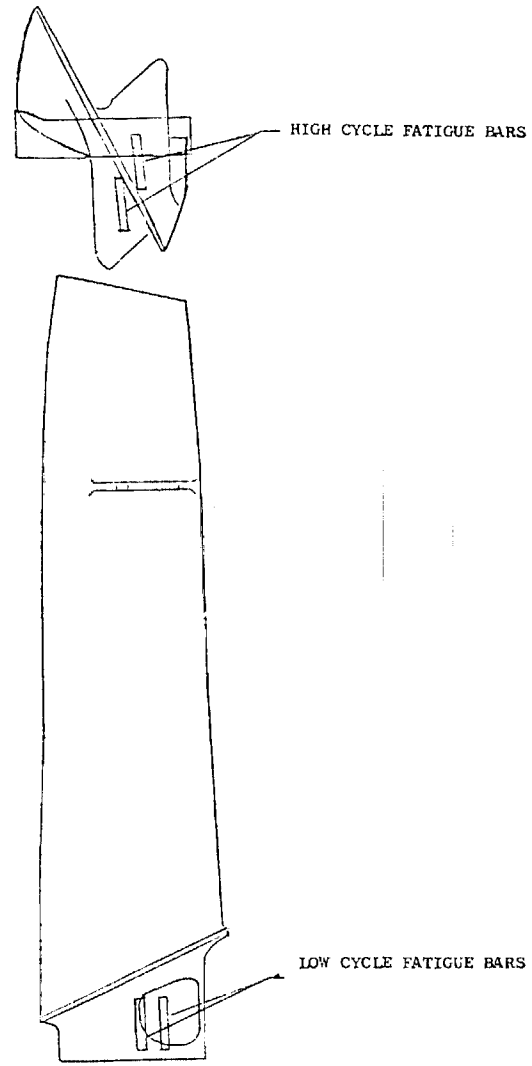
Fatigue tests are run on any new fan blade design. The purpose of fatigue tests are listed as the objectives. When these objectives are met, the design has been proven adequate for airlines service.

Bench Fatigue Test

- **Objectives**
  - Demonstrate the Design to be Free of Stress Risers
  - Demonstrate an Acceptable Manufacturing Process
  - Demonstrate Adequate Design Margin to Preclude Field Problems
- **Status**
  - Test Completed

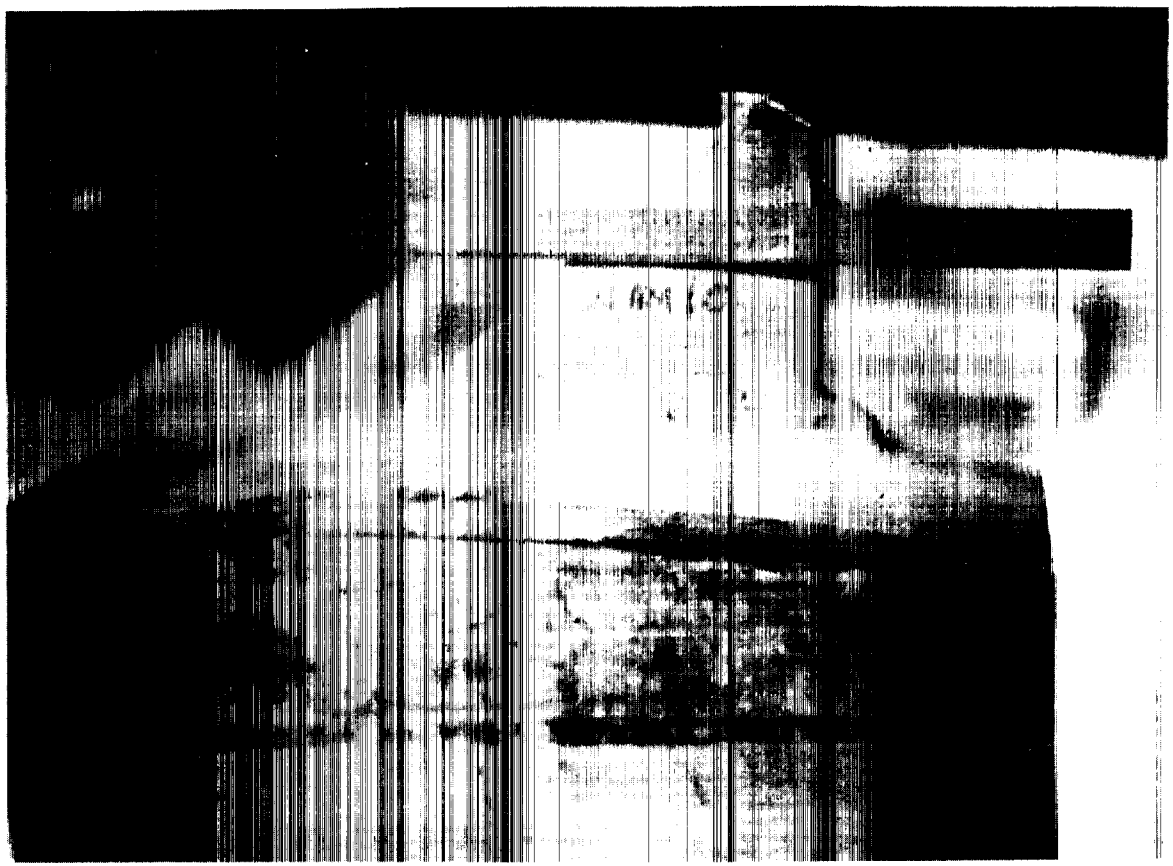
This sketch shows where the test bars were taken from the fan blade forging. The bars in the shank and dovetail area are tested in a low cycle fatigue mode and they are notched to simulate the stress concentration of the dovetail. The bars taken from the shroud area are tested in a high cycle fatigue mode and are smooth bars. The shroud and outer panel area are typically stresses more in high cycle loading than low cycle loading.

Location Of Test Samples



This photograph shows the section of the fan blade cast in kirksite with one of the midspan shrouds ready to be tested in high cycle fatigue. The entire kirksite block is clamped to a heavy table during the fatigue cycling.

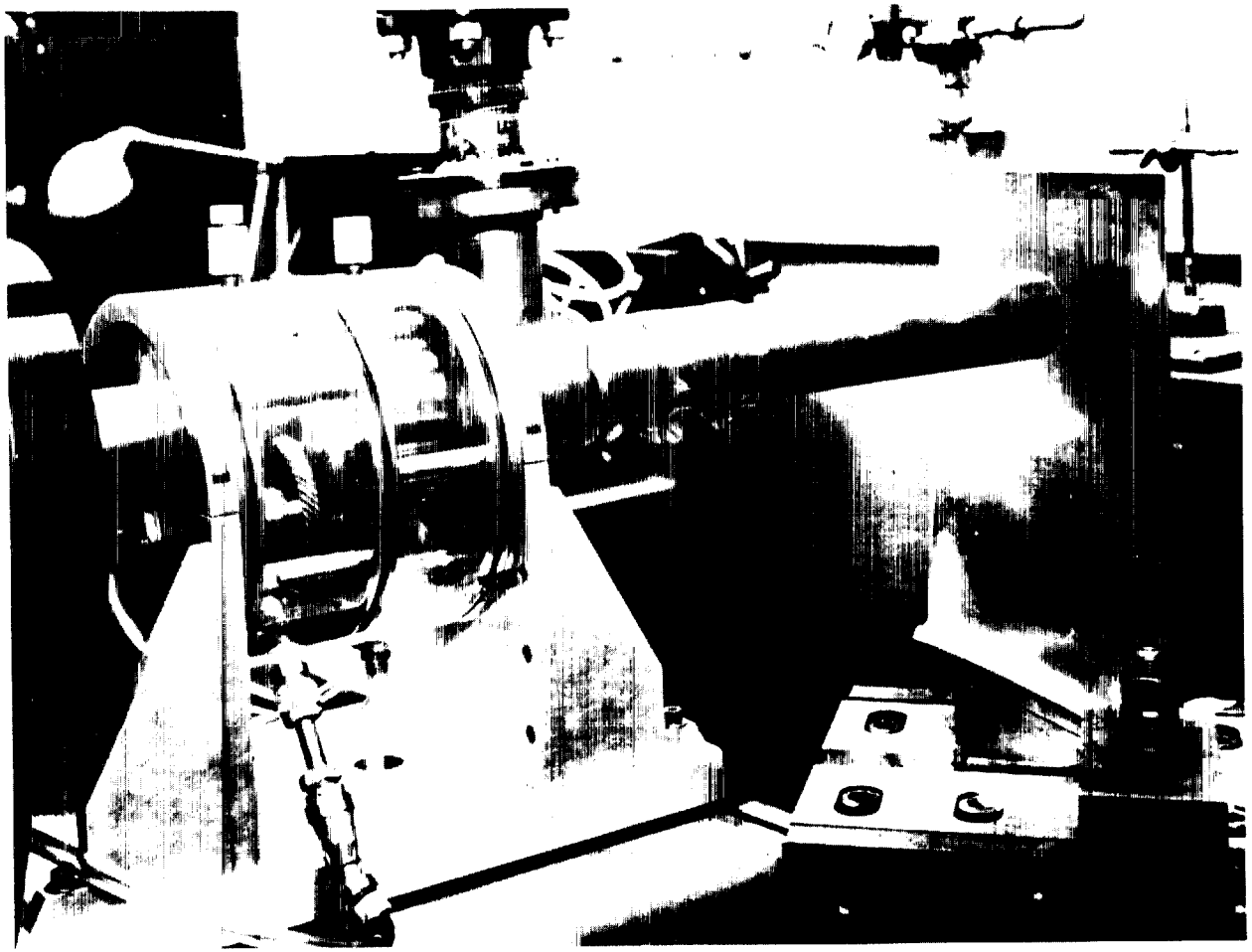
Fan Blade Shroud Segment Test Specimen



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

This photograph shows the inner panel section of the fan blade being tested with the air siren. The blade is clamped at the dovetail and supported as a cantilever beam.

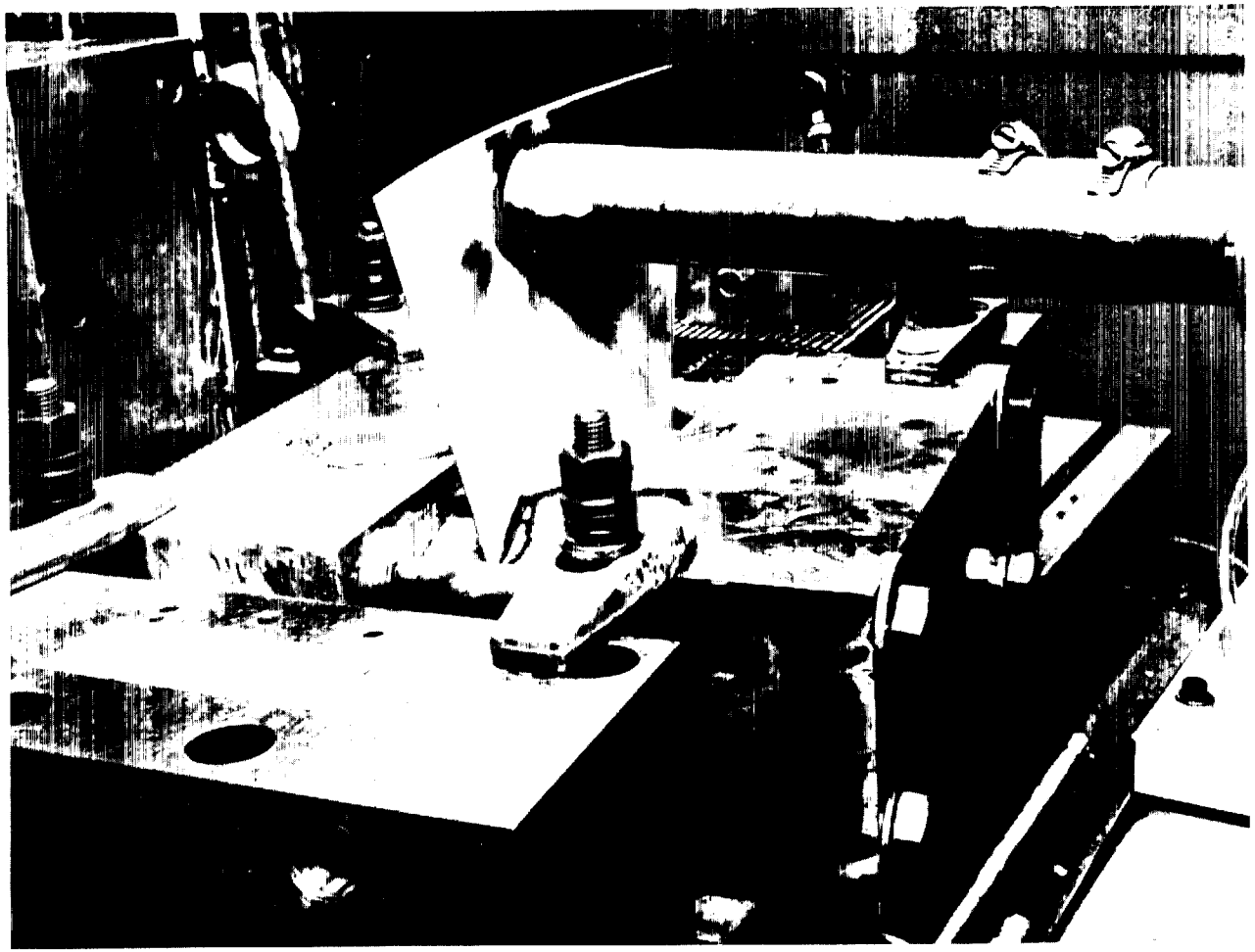
Inner Panel Test Set-Up



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

This photograph shows the outer panel cast in the kirksite block and clamped to the table. The air siren is shown adjacent to the blade during the testing.

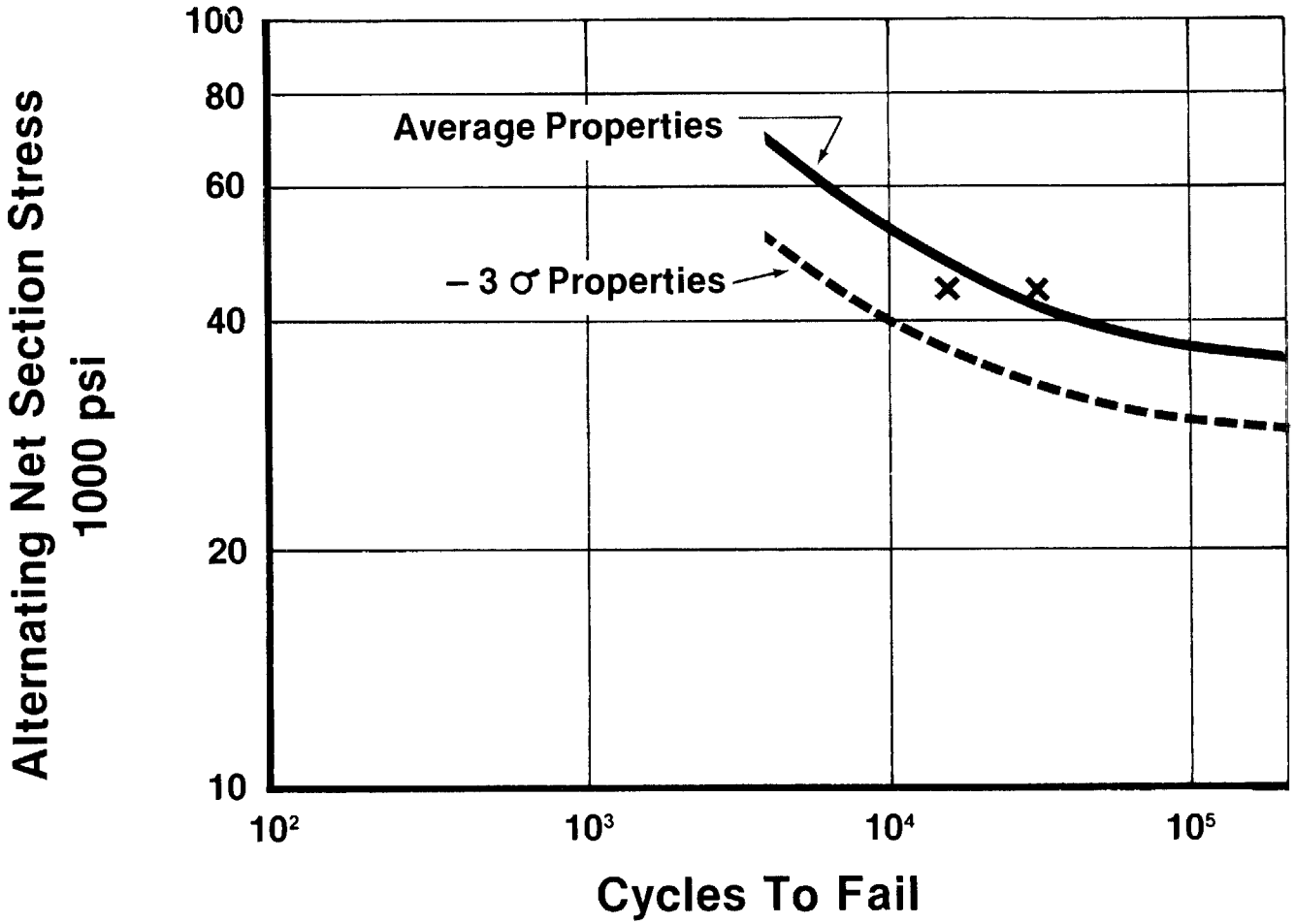
Outer Panel Test Set-Up



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

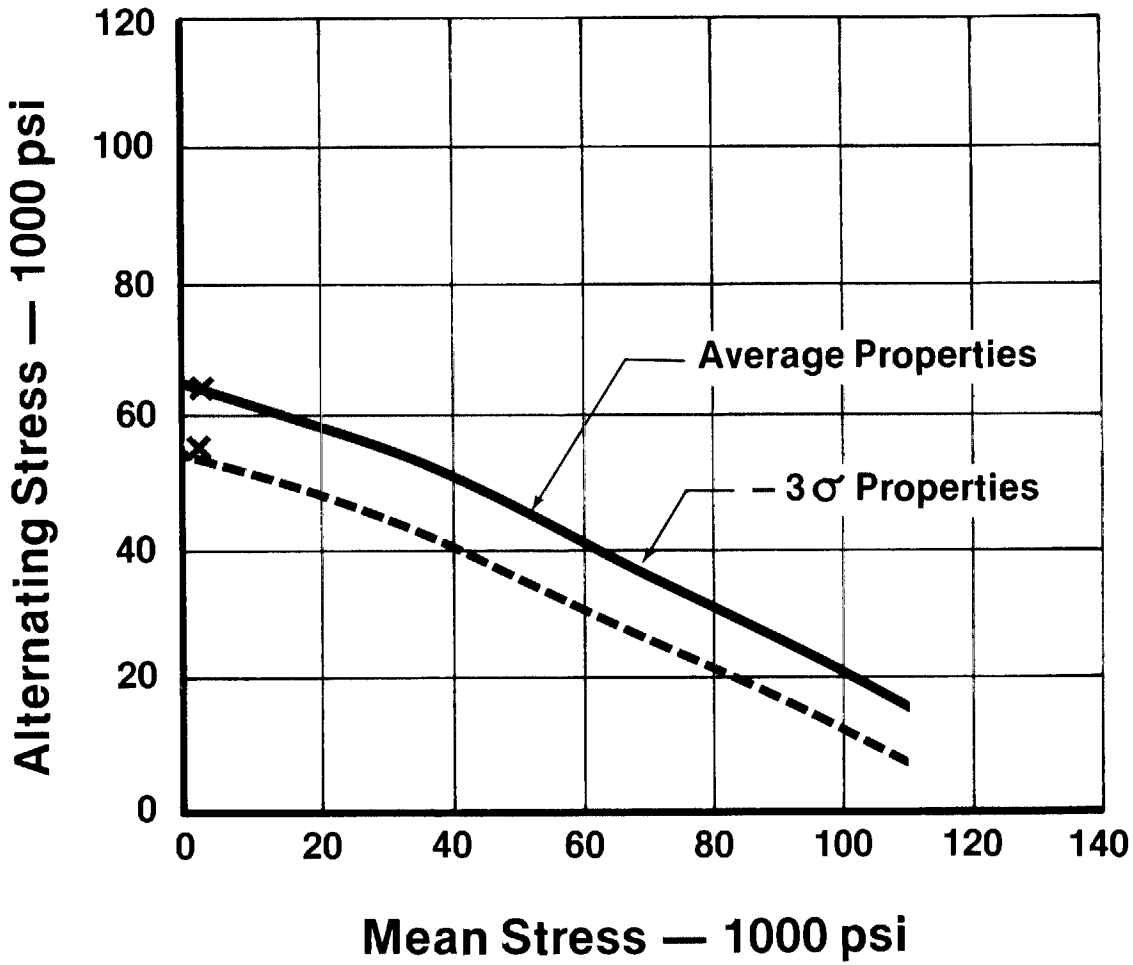
The solid curve shows average titanium low cycle fatigue capability of notched bars ($K_t = 1.5$) versus cycles to failure. The dotted line represents three standard deviations from average fatigue strength of titanium. The two test points represent the two test bar failure points. As can be seen, these two points straddle the average curve.

Titanium Low Cycle Fatigue



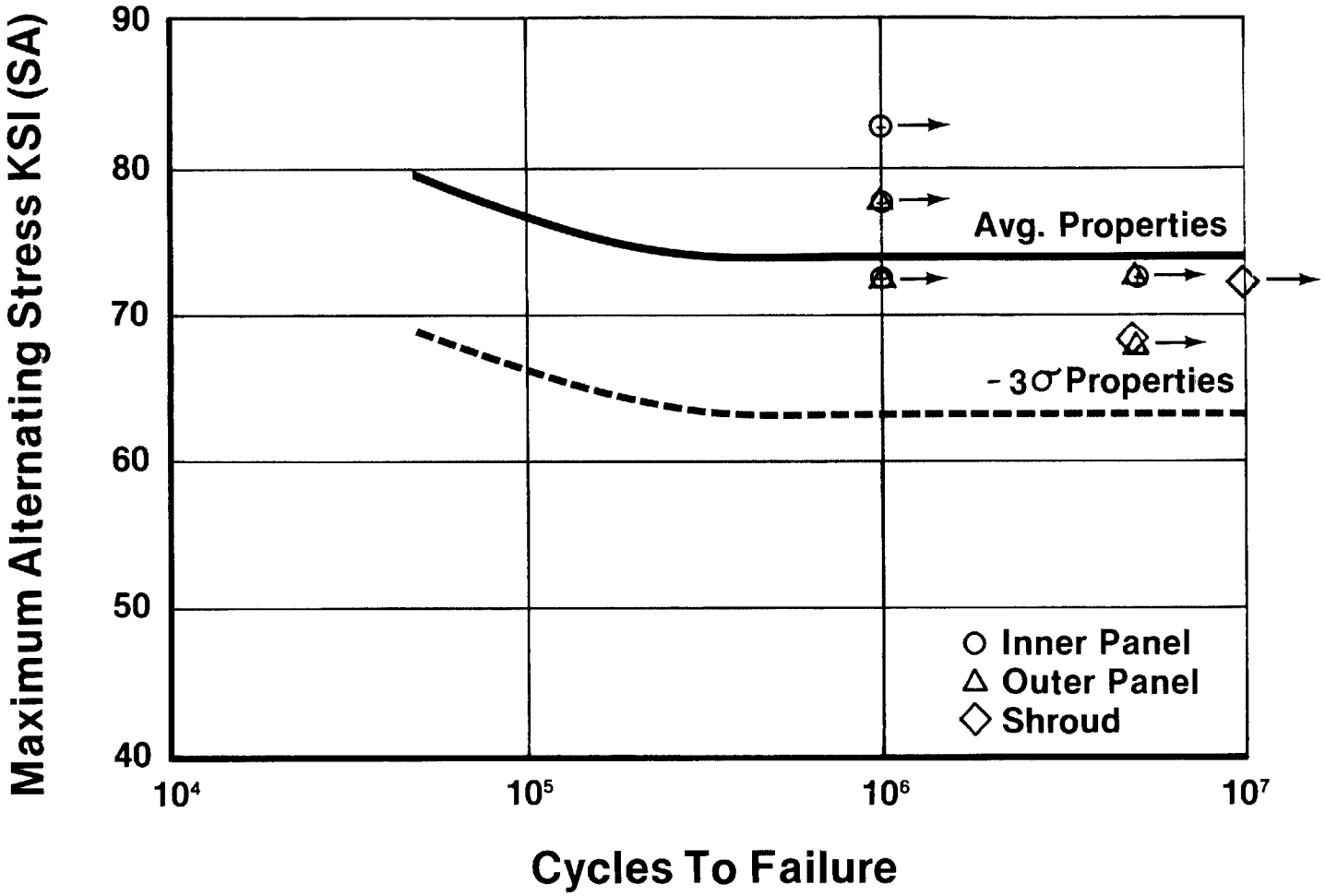
The solid curve shows average high cycle fatigue capability of smooth titanium material. The dotted line represents three standard deviations from average. The two test points show runouts of the two stress levels tested. This shows that the material was at least as good as average.

Titanium High Cycle Fatigue



These two curves show the average and minimum stress versus number of cycles to failure for titanium. The runout test points are shown with the various symbols. These points all represent testing of finished fan blades.

Fatigue Test



The detailed test results for the round bar testing are shown here. In both low cycle and high cycle testing the material was as good as average.

Bench Fatigue Test

Results

- **Round Bar — Low Cycle Fatigue**

  From Forging Dovetail — Spanwise Direction
  90 KSI — Axial-Axial — KT = 1.5 — Unpeened
  - Life = 15,528 and 23,192 Cycles
  - Materials Curve = 20,000 Cycles Avg.; 6000 Cycles −3 Std Dev.

- **Round Bar — High Cycle Fatique**

  From Forging Mid Span — Circumferential Direction
  Bending-Bending-KT = 0.1-Unpeened
  - Life = 10^7 Cycles @ 55 KSI Alt, 1.5 × 10^7 Cycles @ 65 KSI Alt.
  - Materials Curve = 64 KSI Avg.; 54 KSI −3 Std Dev.

The results of the fatigue testing on the finished fan blade inner and outer panel are shown here. When the first test went 5,000,000 cycles, the stress levels were increased by 5 KSI until failure was achieved. Failure occurred between 1000 and 1,000,000 cycles.

Bench Fatigue Test

- **Finished Airfoil — Inner Panel**
  - First Flex Inner Panel
  - Test #1 5 × 10^6 Cycles @ 67.5 KSI + 10^6 @ 72.5 KSI + 10^6 @ 77.5 KSI + Failure @ 82.5 KSI
  - Test #2 5 × 10^6 Cycles @ 72.5 KSI + 10^6 @ 77.5 KSI + 10^6 @ 82.5 KSI + Failure @ 87.5 KSI
  - Materials Curve = Avg. Runout @ 74 KSI; −3 Std Dev Runout @ 63 KSI
- **Finished Airfoil — Outer Panel**
  - First Flex Outer Panel
  - Test #1 5 × 10^6 Cycles @ 67.5 KSI + 10^6 @ 72.5 KSI + 10^6 @ 77.5 KSI + Failure @ 82.5 KSI
  - Test #2 5 × 10^6 Cycles @ 72.5 KSI + Failure @ 77.5 KSI
  - Materials Curve = Avg Runout @ 74 KSI; −3 Std Dev. Runout @ 63 KSI

The fatigue results of the shroud testing are shown. The shroud could not be failed as the maximum output of the siren was insufficient.

Bench Fatigue Test

- **Finished Airfoil — Shroud**
  - First Flex Shroud
  - 5×10^6 Cycles @ 67.5 KSI + 10^7 Cycles @ 72.5 KSI No Failure
  - Materials Curve = Avg Runout @ 74 KSI; −3 Std Dev Runout @ 63 KSI

The fatigue results from the test bars and the finished airfoil demonstrated that the new design is equal to the current design for fatigue margin and in over 12,000,000 flight hours the current design has never experienced a fatigue failure.

Conclusions/Summary

- Demonstrated Design to be Free of Stress Risers and Manufacturing Process is Acceptable
- New Design Margin is Equal to Current Design and Current Design has Never Experienced a Fatigue Failure

Crosswind Test

- **Objective**
  - Subject Improved Fan to Inlet Distortion Induced by 90° Crosswinds and Measure Vibratory Stresses
- **Status**
  - Test Completed, Reported in Last Oral
- **Results**
  - The Improved Fan has Similar Crosswind/Distortion Characteristics as the Original Fan

Endurance Testing

- **Objective**
  - Conduct 1000 "C" Cycles With Improved Performance Fan Blade, Case Stiffener, and .115" Clearances to Simulate Typical Flight Cycle. Assess Wear Characteristics/Structural Integrity of Improved Fan Package.
- **Status**
  - Test Completed, Reported in Last Oral.
- **Results**
  - Hardware in Excellent Condition.

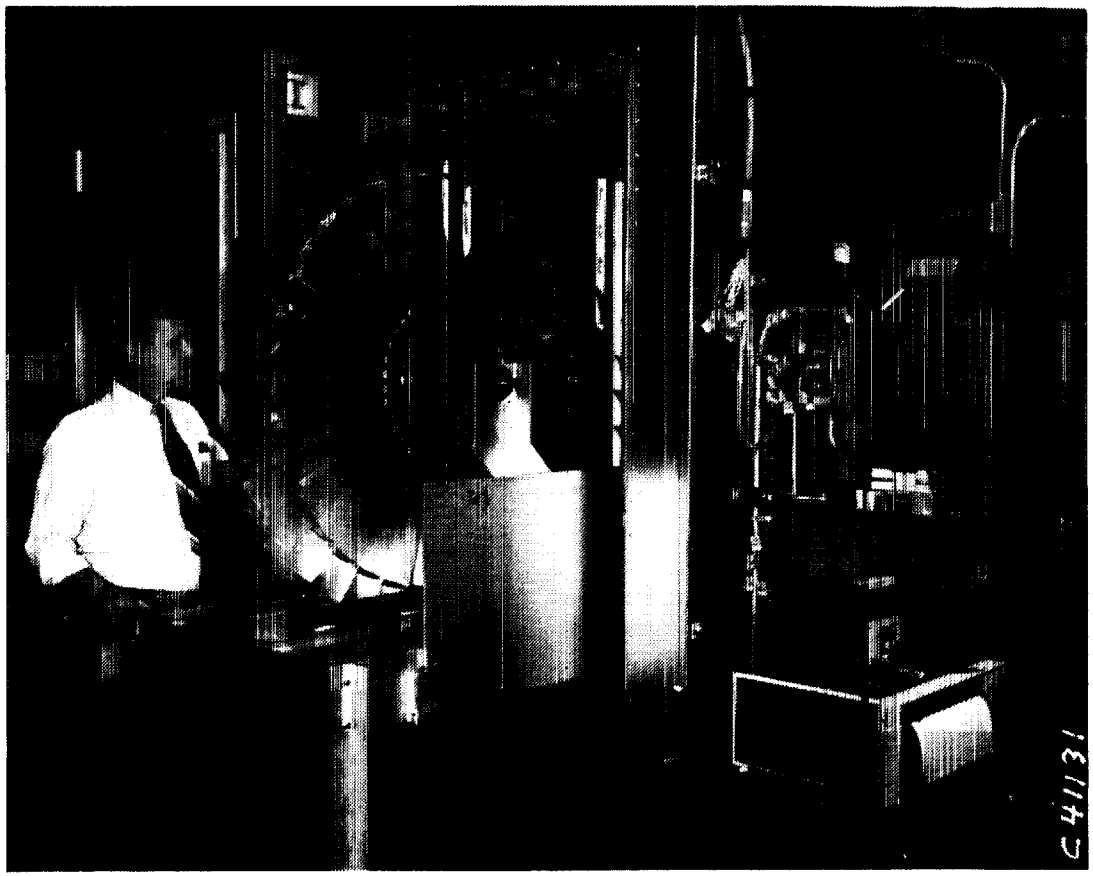
The photoelastic test was conducted on the new fan blade to investigate local areas of high stress and complex stress distribution. Three different shank designs were tested to determine the best design for the new fan blade.

Fan Photoelastic Test

- **Objective**
  - Obtain Detailed Stresses in Blades with 3 Shank Versions, Especially in High-Stress Areas and in Locations of Complex Concentration.
- **Status**
  - Test Completed

This photograph shows the photoelastic model of the fan rotor in the rotating stress freezing test facility at General Electric.

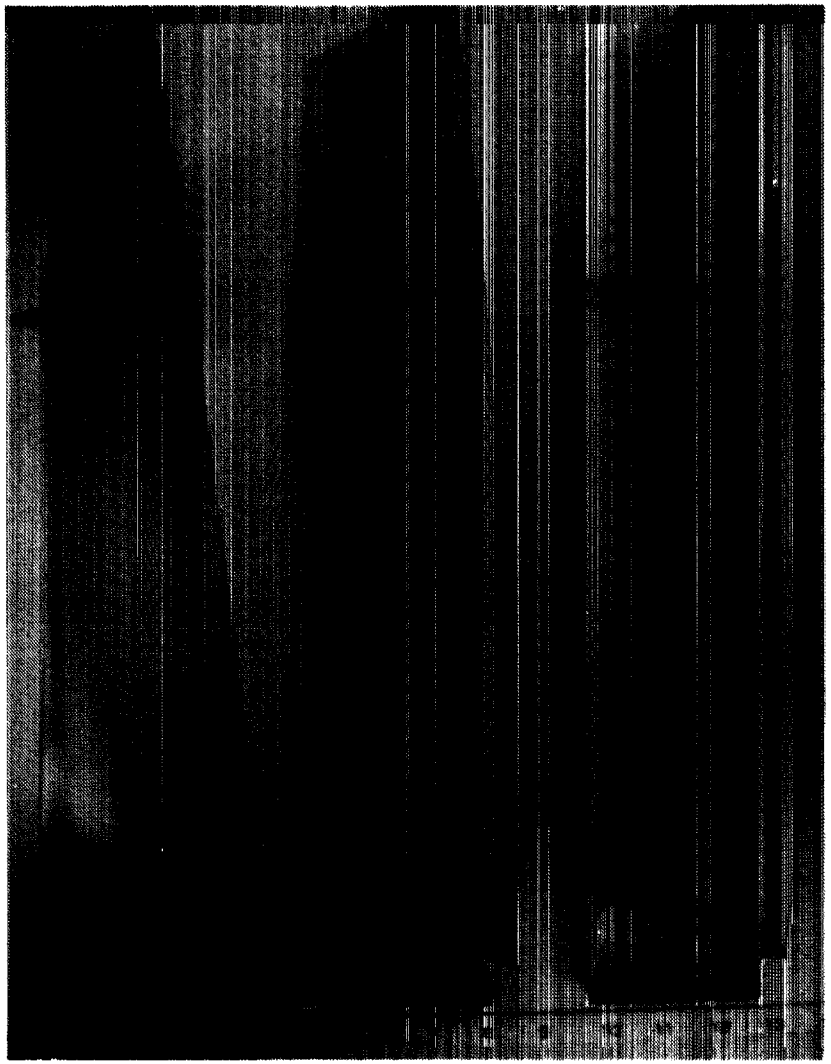
Photoelastic Model Of Fan In Stress-Freezing Test Facility



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

A photograph of the three different shank configurations is shown. The rest of the fan blade was identically the same.

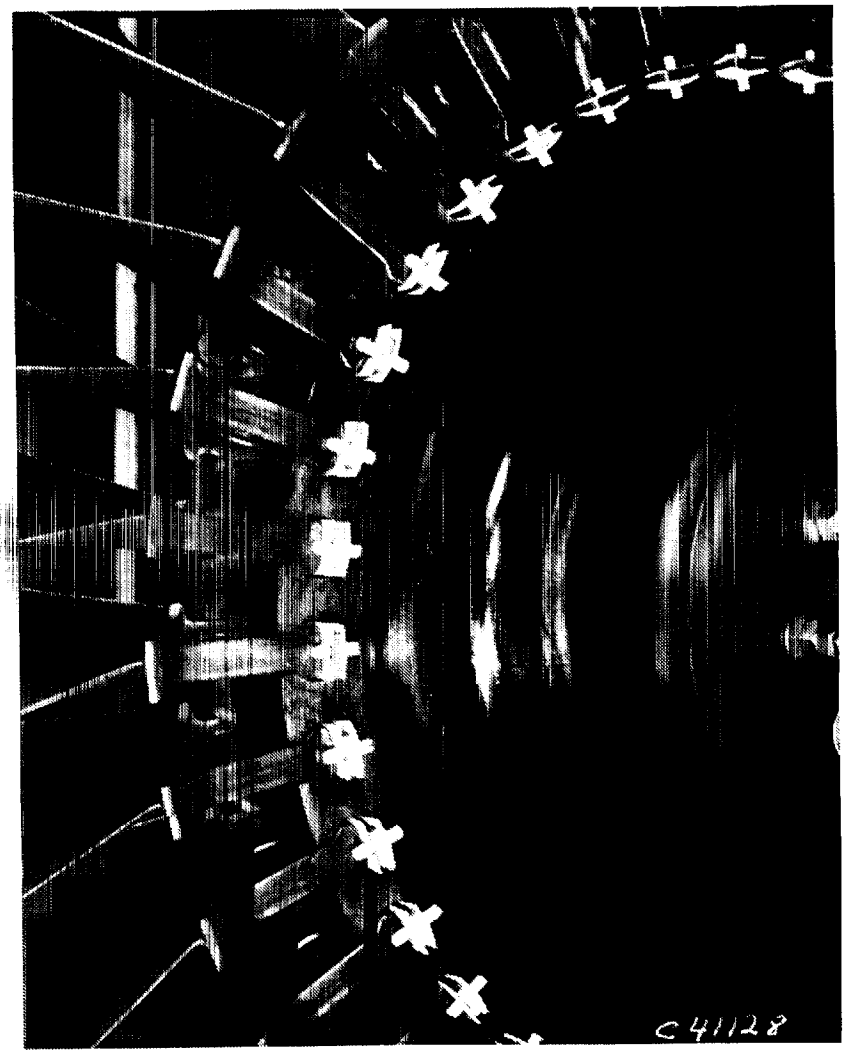
Blade Models — Full Shank, Half-Pocket Shank, and Full-Pocket Shank



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

This photograph shows a close-up view of the shank region of the fan blade. The simulated fan blade retainers are shown under the blade dovetail.

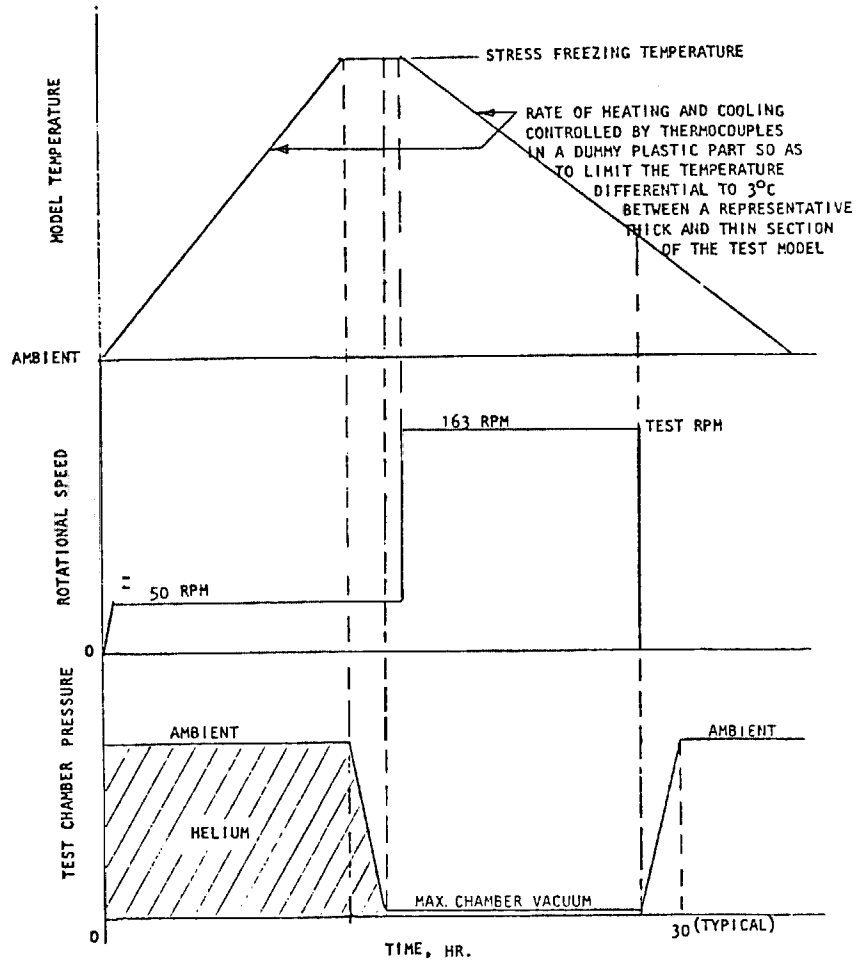
Photoelastic Model Of Fan Rotor



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

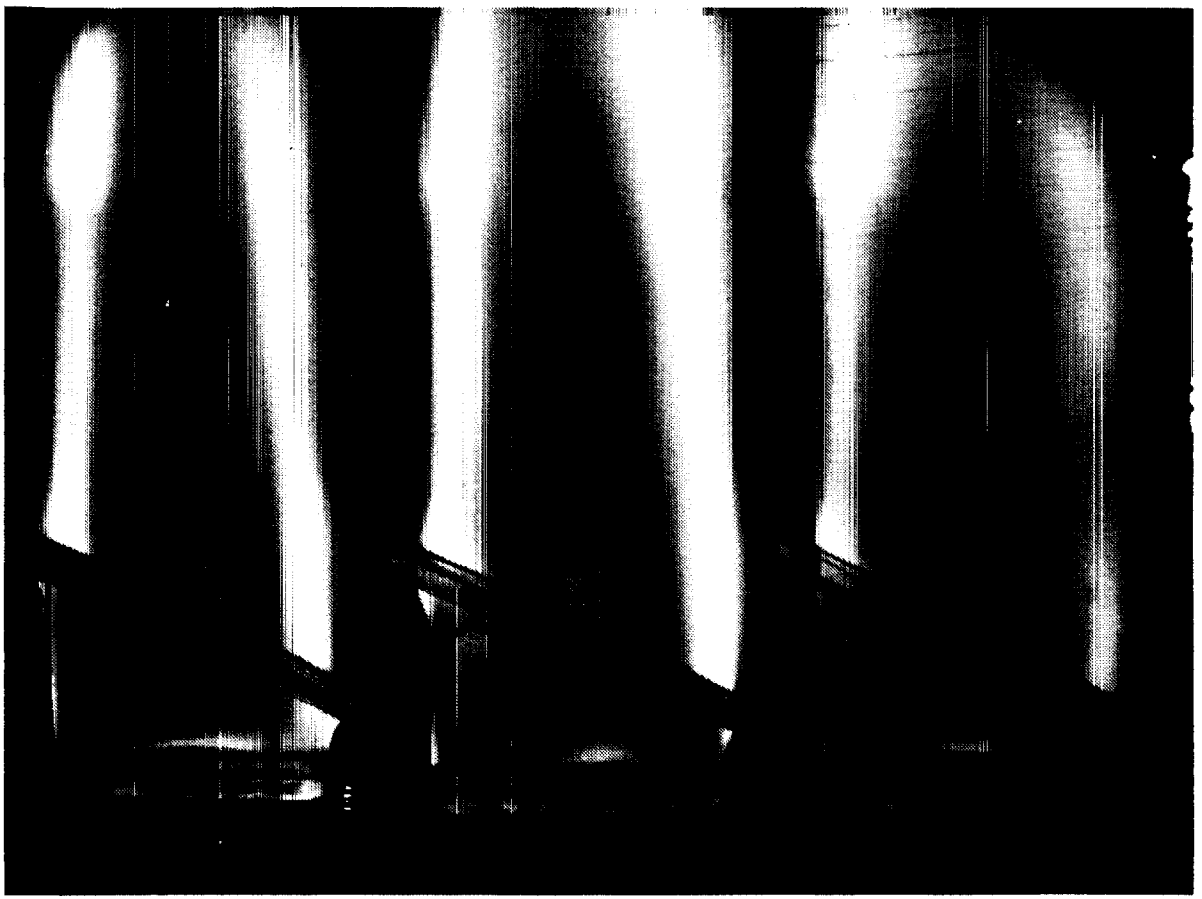
These three curves show how the temperature, rotational speed and chamber pressure were varied with time for the stress freezing cycle. The total time required for this cycle was approzimately 40 hours.

Rotating Stress Freezing Cycle



This photograph shows the frozen stress fringe pattern for the three different shank models.

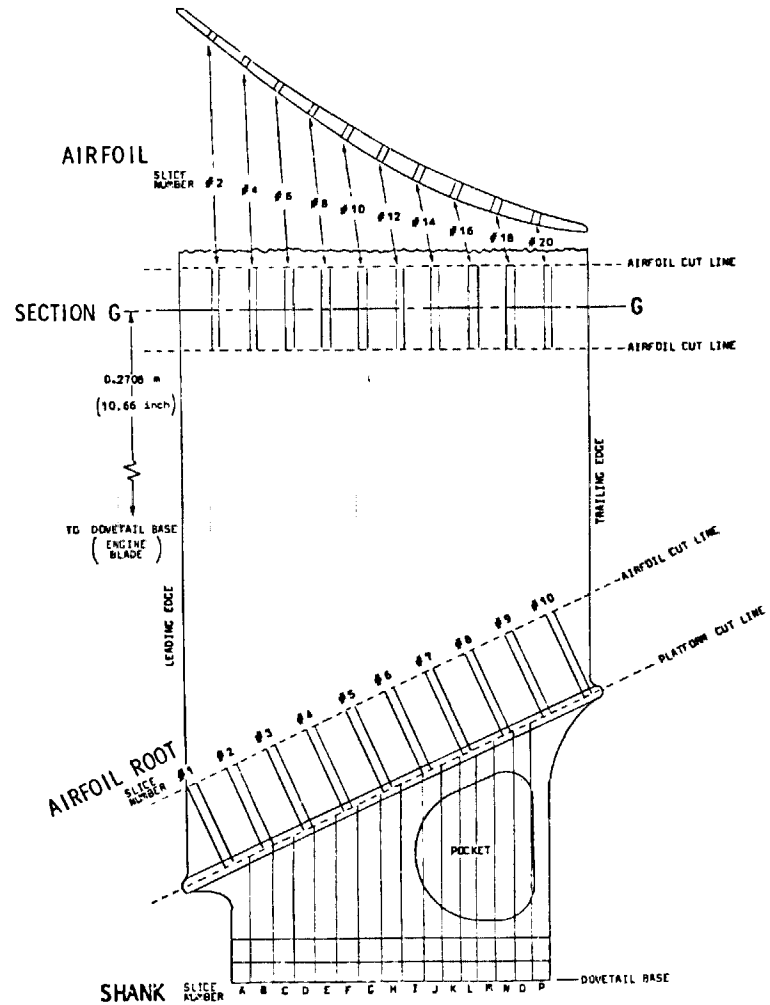
Frozen-Stress Fringe Patterns In The Three Types Of Blades



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

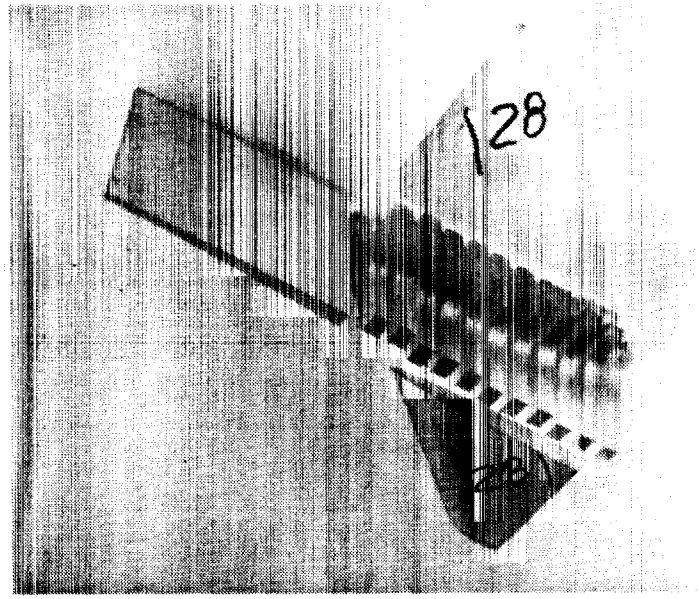
This sketch shows the slicing pattern used to investigate the stress in the shank/dovetail, airfoil root and airfoil pitch section.

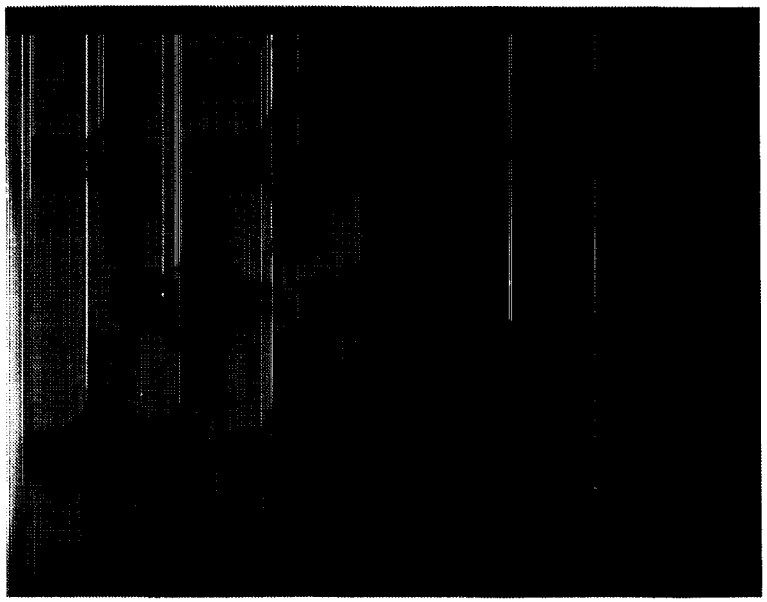
Blade Slicing Diagram



The upper photograph shows the slices that were made around the airfoil/shroud intersection. The slices have been reassembled to show their relative position. The lower photograph shows the slices laid-out from the forward section in the upper left hand corner to the aft section at the lower right corner.

Shroud Slices

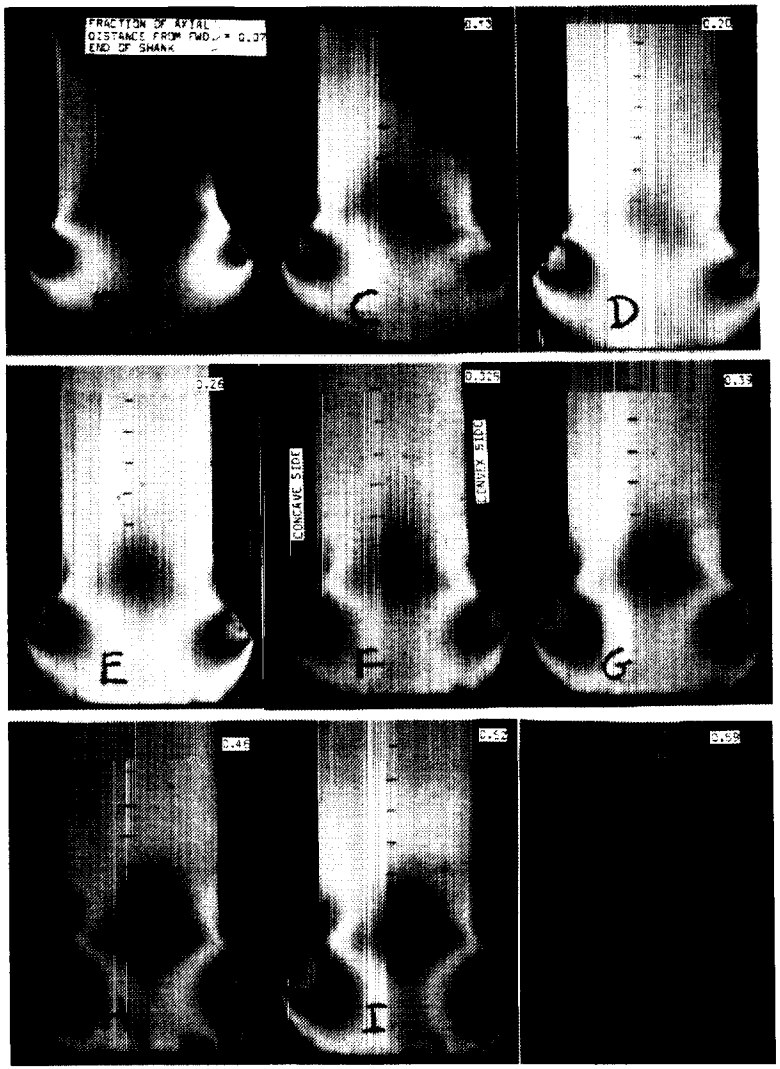




ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

This is a photograph of the shank/dovetail sections with the frozen stress fringe pattern. The section "B" is near the leading edge and section "J" is just aft of the center of the dovetail.

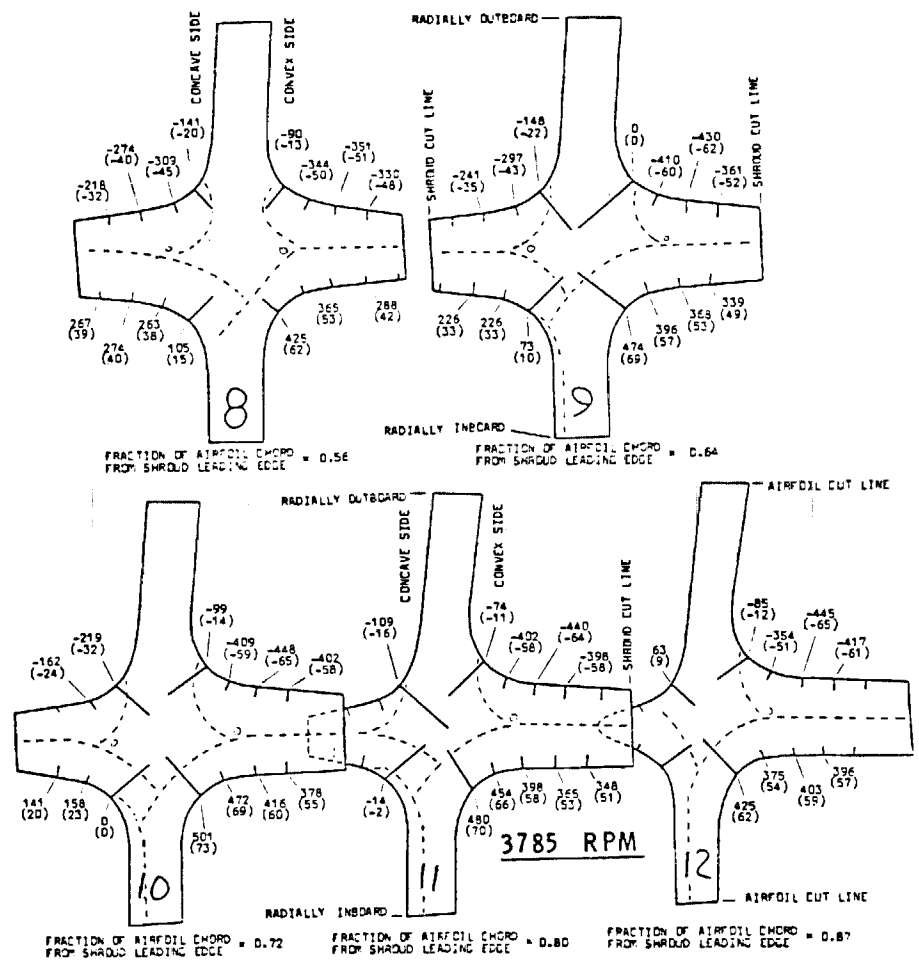
Fringe Patterns In Typical Series Of Dovetail Slices



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

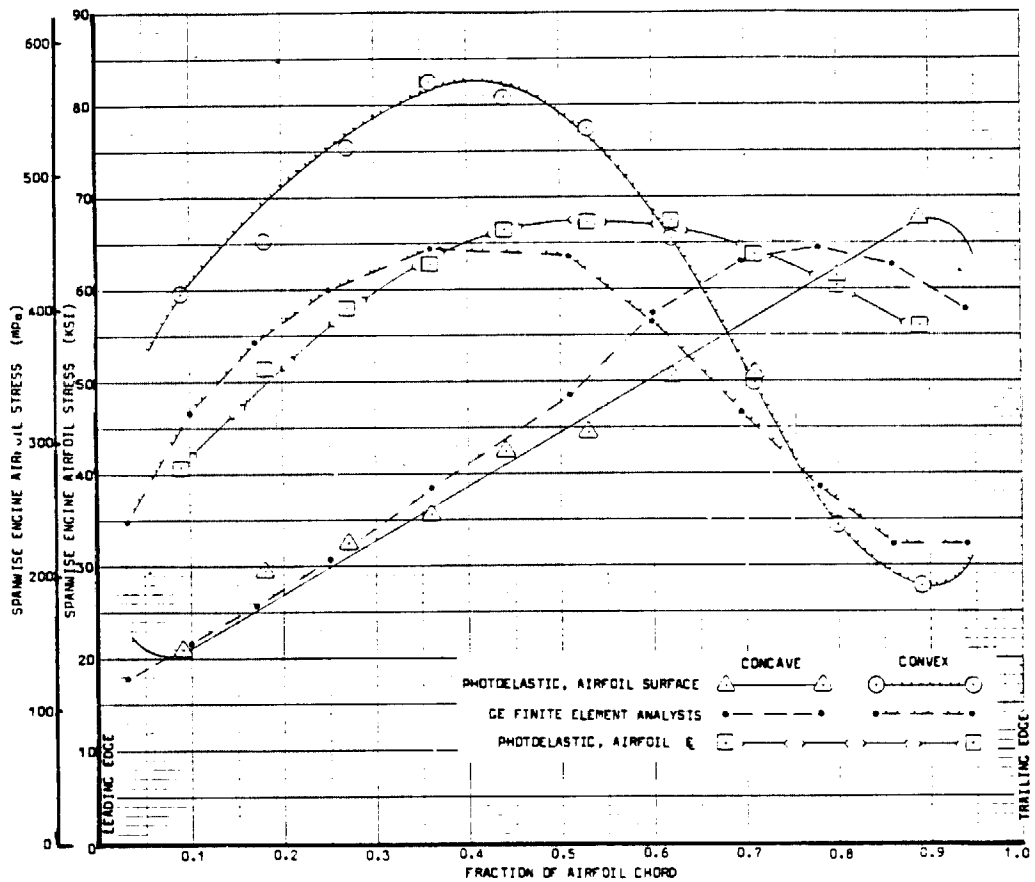
This sketch shows some typical slices throught the shroud/airfoil intersection with the stresses at the root of the shroud and around the fillet. Stresses are highest where the shroud overhang is largest.

Typical Shroud Slices Showing Local Stresses In MPa and (KSI)



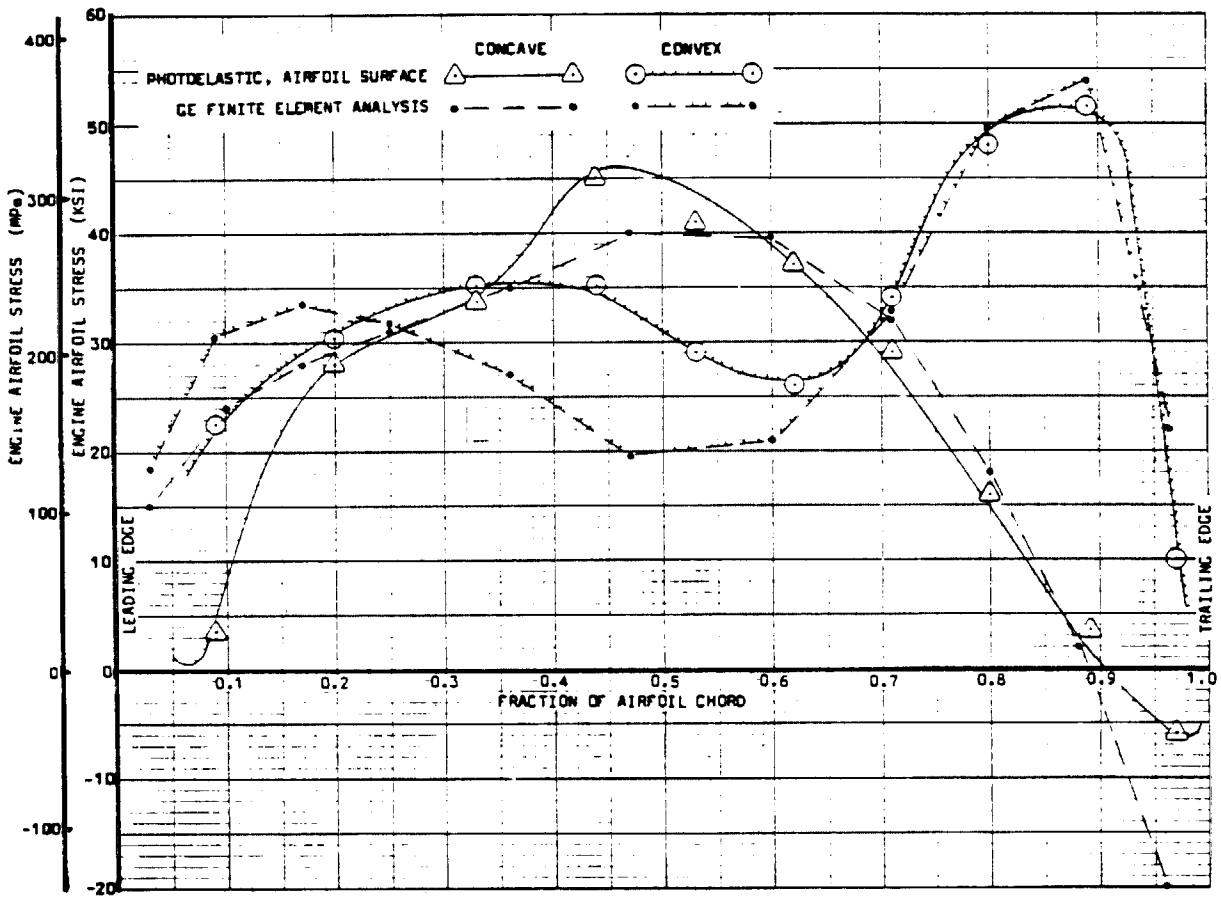
This curve shows the stresses in the airfoil at Section G-G, a previously analyzed section. The photoelastic results are compared to a finite element analysis and shows relatively good agreement.

Spanwise Airfoil Stress Comparison



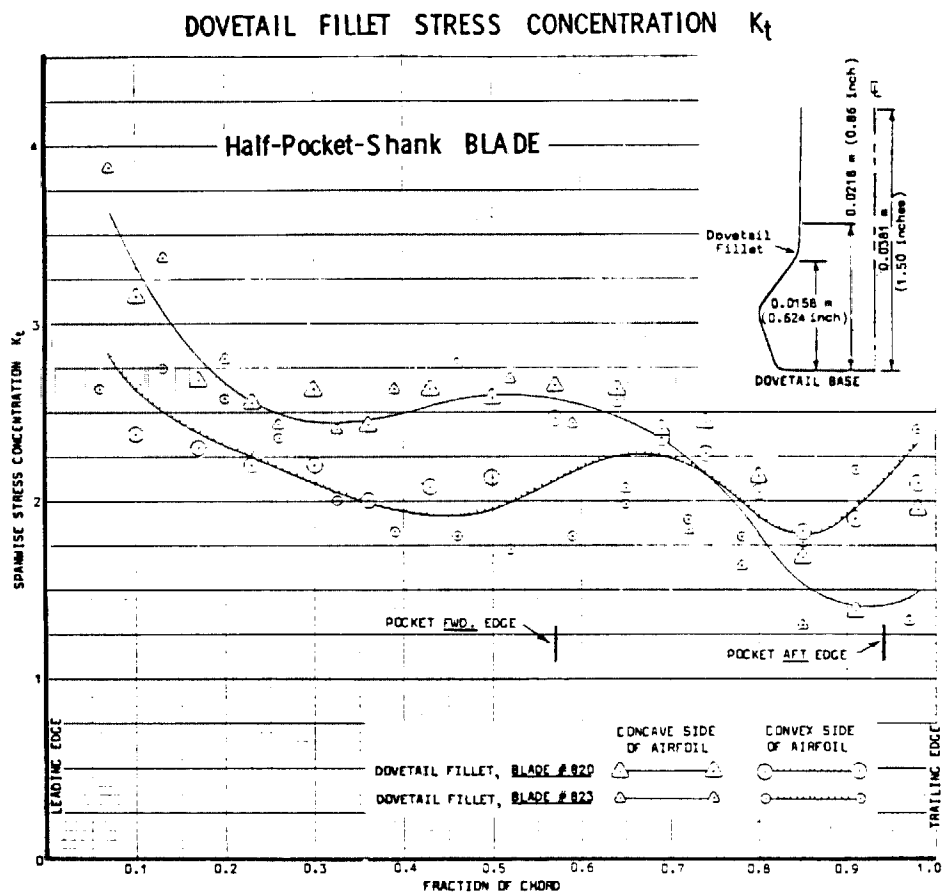
This curve shows the stresses in the airfoil root section. The photoelastic results are compared to the finite element analysis and show relatively good agreement.

Airfoil Root Stress Comparison



This curve shows the stress concentration effect in the shank region of the half pocket blade. These results confirmed the half pocket design as being best.

Dovetail Fillet Stress Concentration



The photoelastic results show no life limiting stresses in the new blade with good comparative results to previous analysis. In some cases, the photoelastic results were somewhat higher, but these higher results could generally be explained.

Conclusions

- No Life-Limiting Stresses in the Advanced Fan Blades.
- Half-Pocket Shank Design Confirmed.
- Photoelastic Results Similar to Finite Element Results.
- Local Concentrations Found Photoelastically are Higher Than Those From Other Methods.

Summary

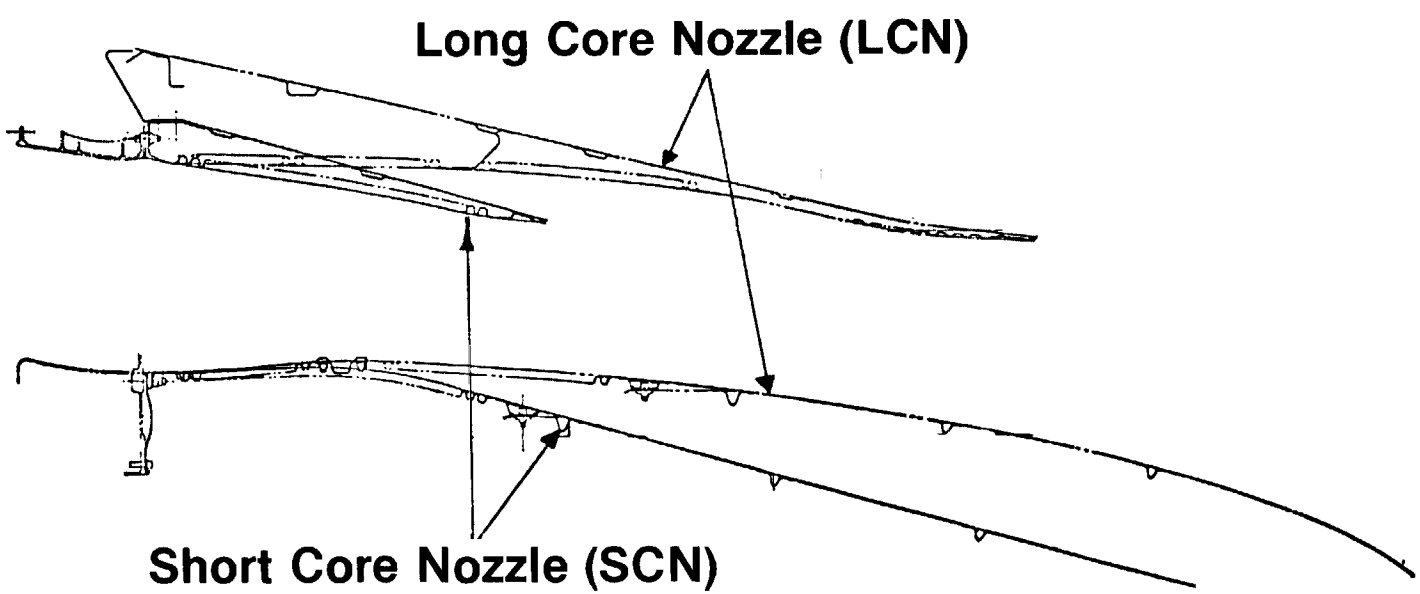
Fan

- Program Complete
- Performance Objectives Exceeded
  – 2% Δ SFC at Cruise
- DC-10, 747, A300B Flight Tests Completed
- Fan Certified and Introduced in CF6-50 C2/E2
- Also Offered for CF6-6 Retrofit
- Sold to (8) Airlines on CF6-50C2, CF6-50E2 and CF6-6

CF6 Jet Performance
Improvement Program

Task 2.2
Short Core Exhaust
Performance Improvement Concept

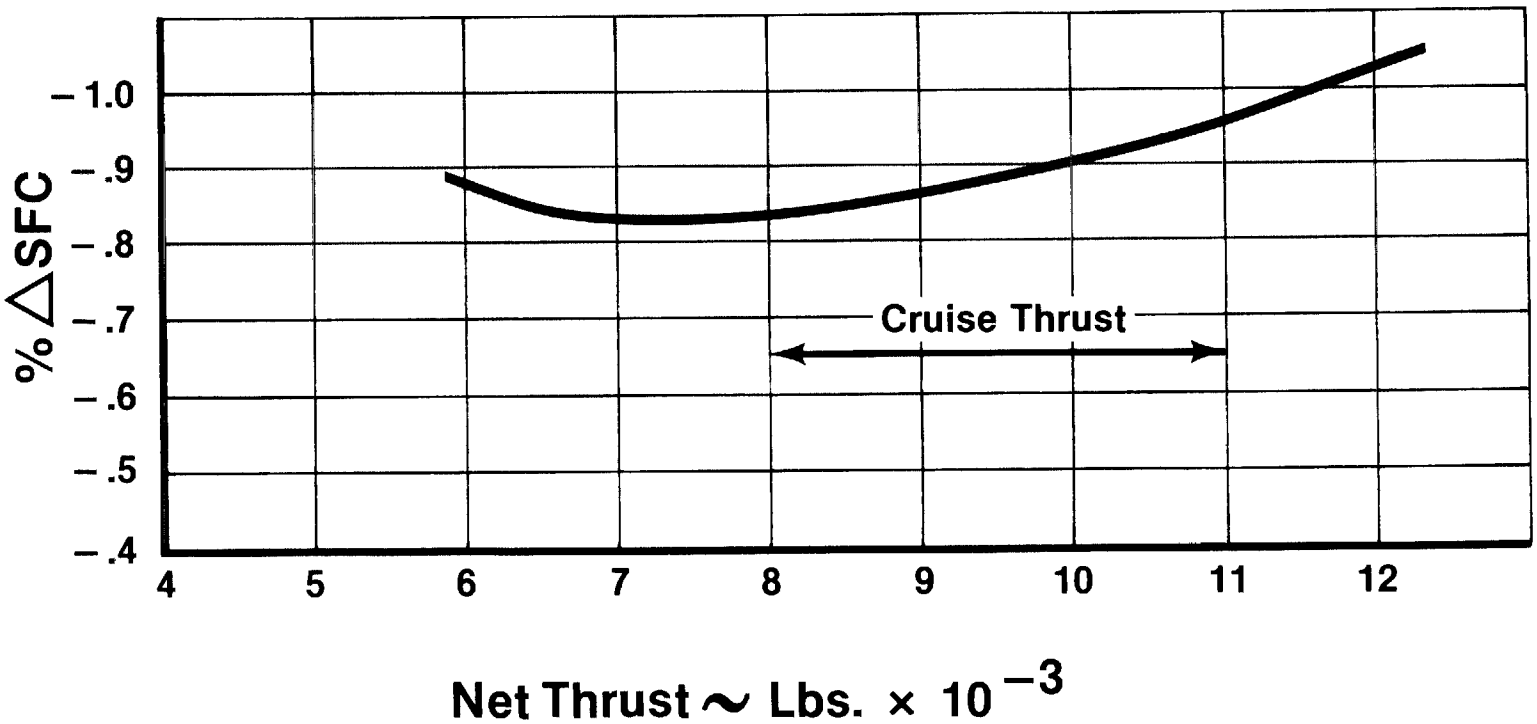
Comparison — Short Core Nozzle to Long Core Nozzle



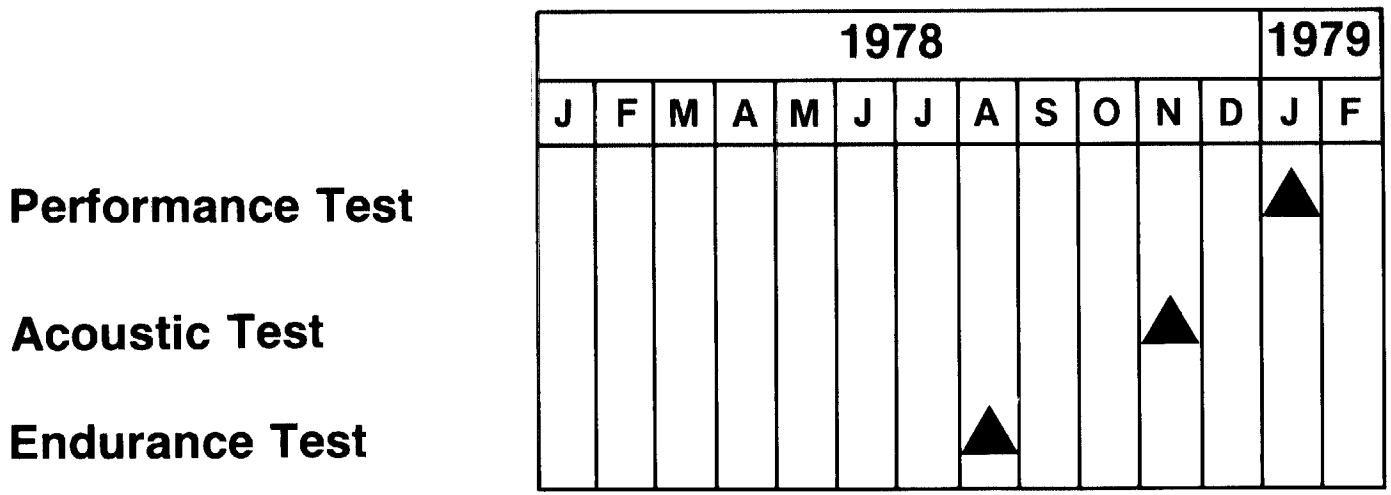
Uninstalled Performance Improvement

35,000 Ft. Alt., 0.85 Mp

Cycle Deck Matched to Engine 517-330
Sea Level Test Cell Data



Program Plan
Short Core Exhaust



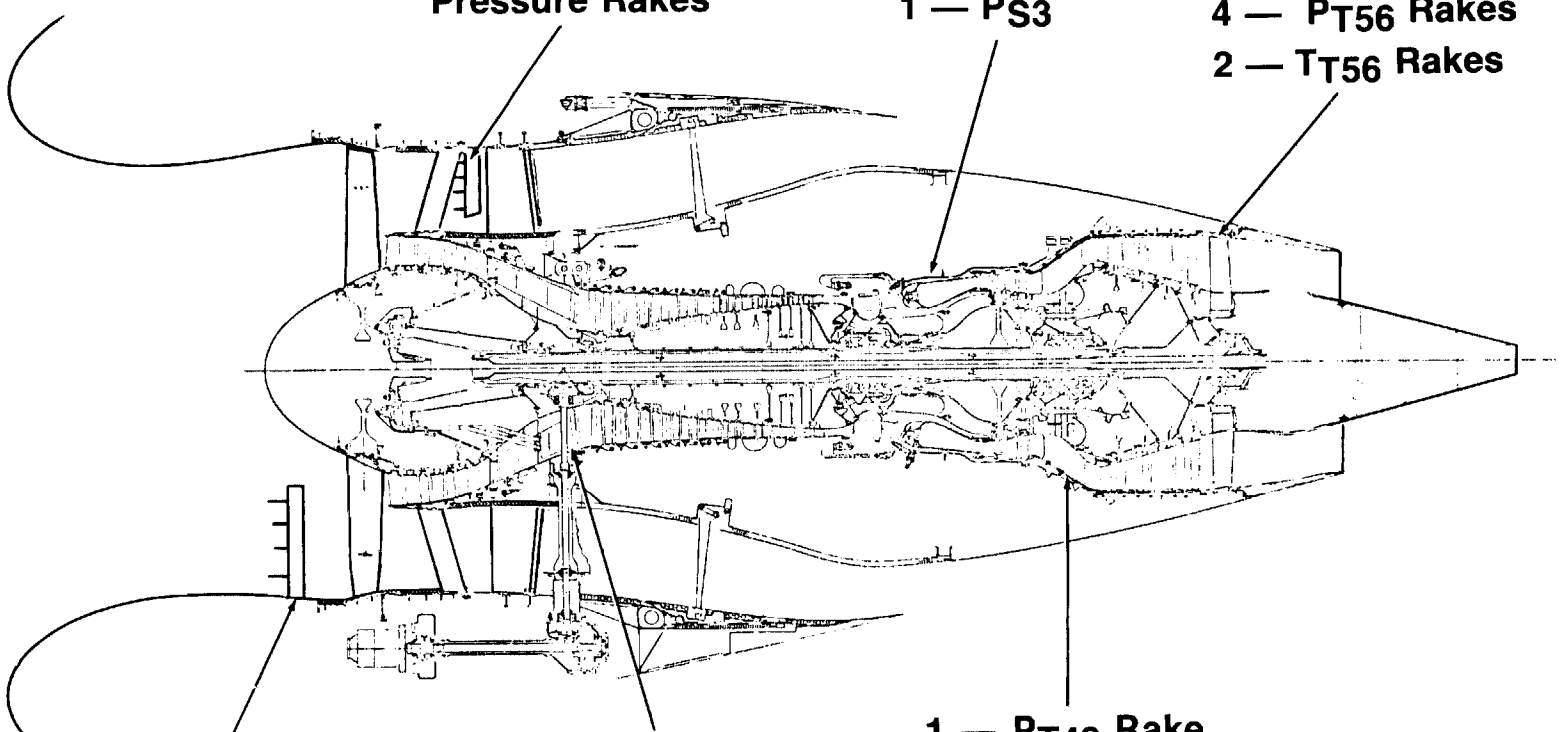
Endurance Test

- **Objective**
  - Conduct 1000 "C" Cycles Which Simulate Typical Flight to Assure Structural Integrity
- **Status**
  - Test Completed, Reported in Last Oral
- **Results**
  - No Indication of any Distress

Engine Performance Test

- **Objective**
  - Measure Performance Improvement at SLS by Back-to-Back Testing — LCN and SCN
- **Status**
  - Test Completed

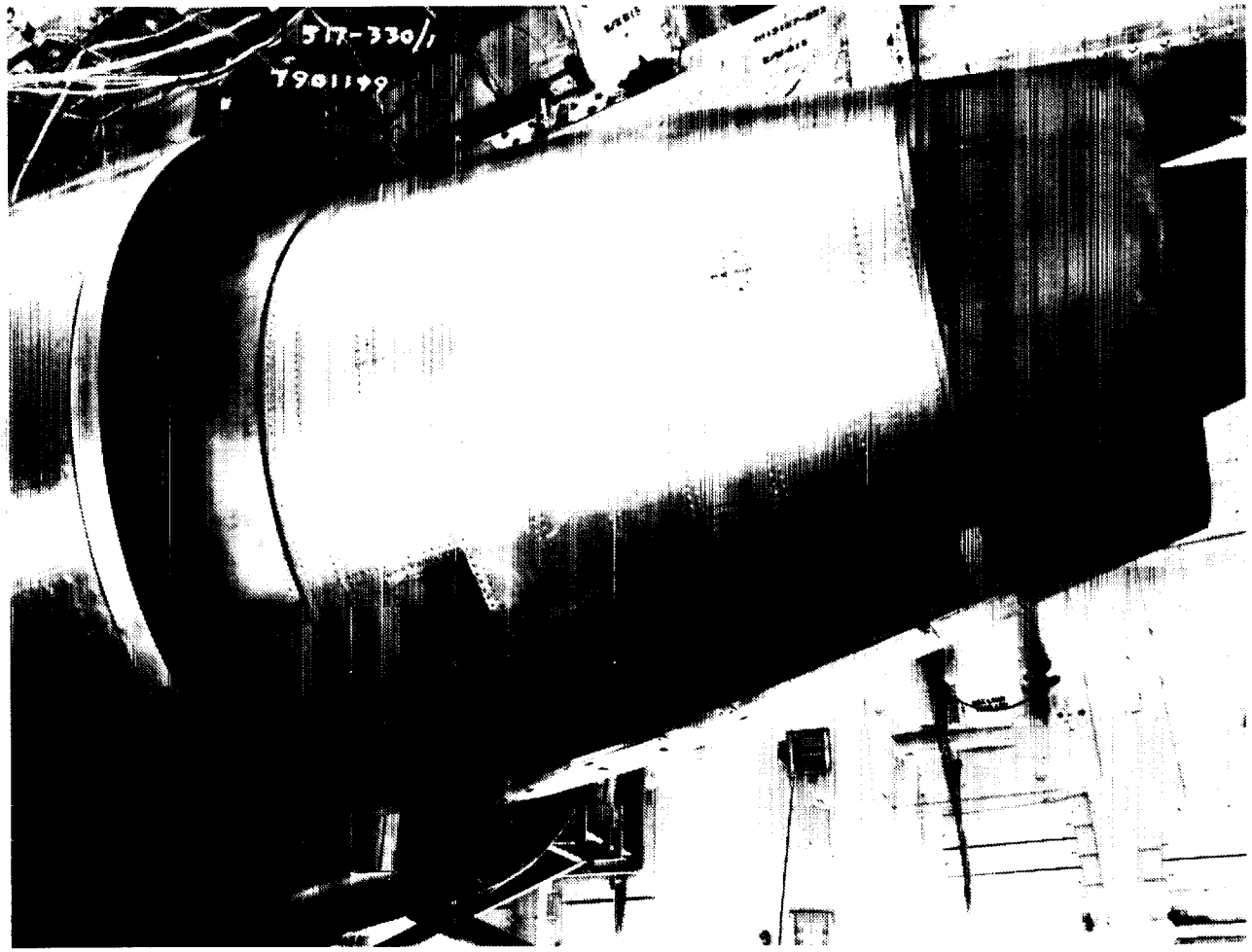
Short Core Nozzle Performance Test



Test Cell Installation of Long Core Nozzle

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Test Cell Installation of Short Core Nozzle



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Overall thrust coefficient is defined as measured thrust over ideal thrust. Ideal thrust is calculated based on a total inlet airflow (from PT2 and PS2 inlet instrumentation), a calculated core airflow, and nozzle discharge pressures for the fan and core. Method A & B are similar, just a different effective area used in calculating core airflow. Method C is based on the measured thrust difference assuming a point on the power calibration can be selected where the ideal thrust calculations are identical for both short and long nozzle. Based on the scale model tests at FluiDyne, the thrust coefficient difference is a 0.35% CT improvement for the short core nozzle when compared to the long core nozzle. From the following three methods of analysis it is seen that the full scale tests verifies the results of the scale model tests.

Method Of Analysis

- Overall Thrust Coefficient Defined as:

$$CT = \frac{F_{GM}}{(F_{I,\,FAN} + F_{I,\,CORE})}$$

Method A: Core Flow Calculated from C_{F8} Core

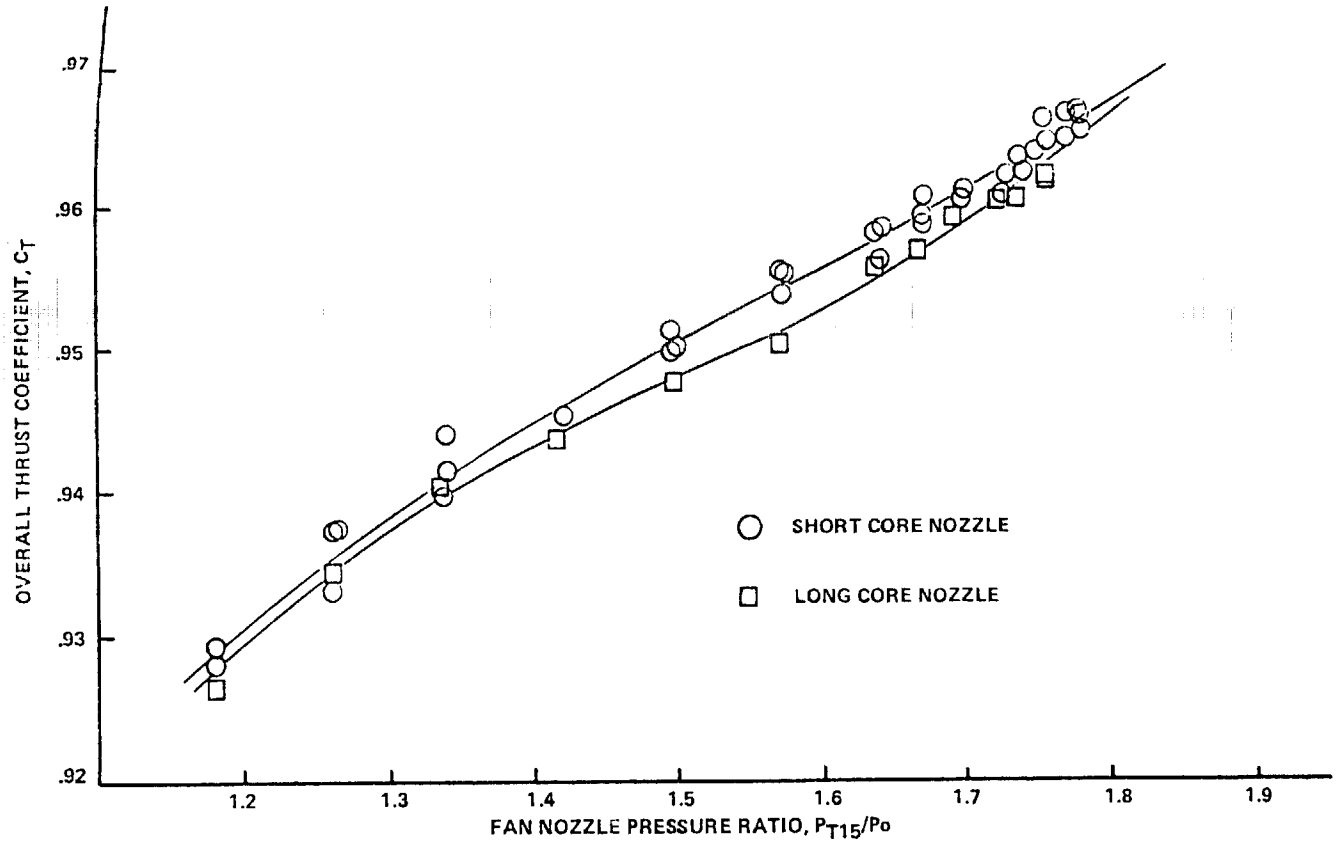
Method B: Core Flow Calculated from LPT Flow Function

Method C: ΔThrust at Equal Effective Throat Areas

This curve indicates a thrust coefficient difference for the case where core flow is calculated based on physical A8 measurements and scale model core nozzle flow coefficients, CF8. The magnitude of the thrust coefficient difference is $\approx 0.25\%$.

Overall Thrust Coefficient Based on A_8, C_{f8} & Measured Inlet Flow

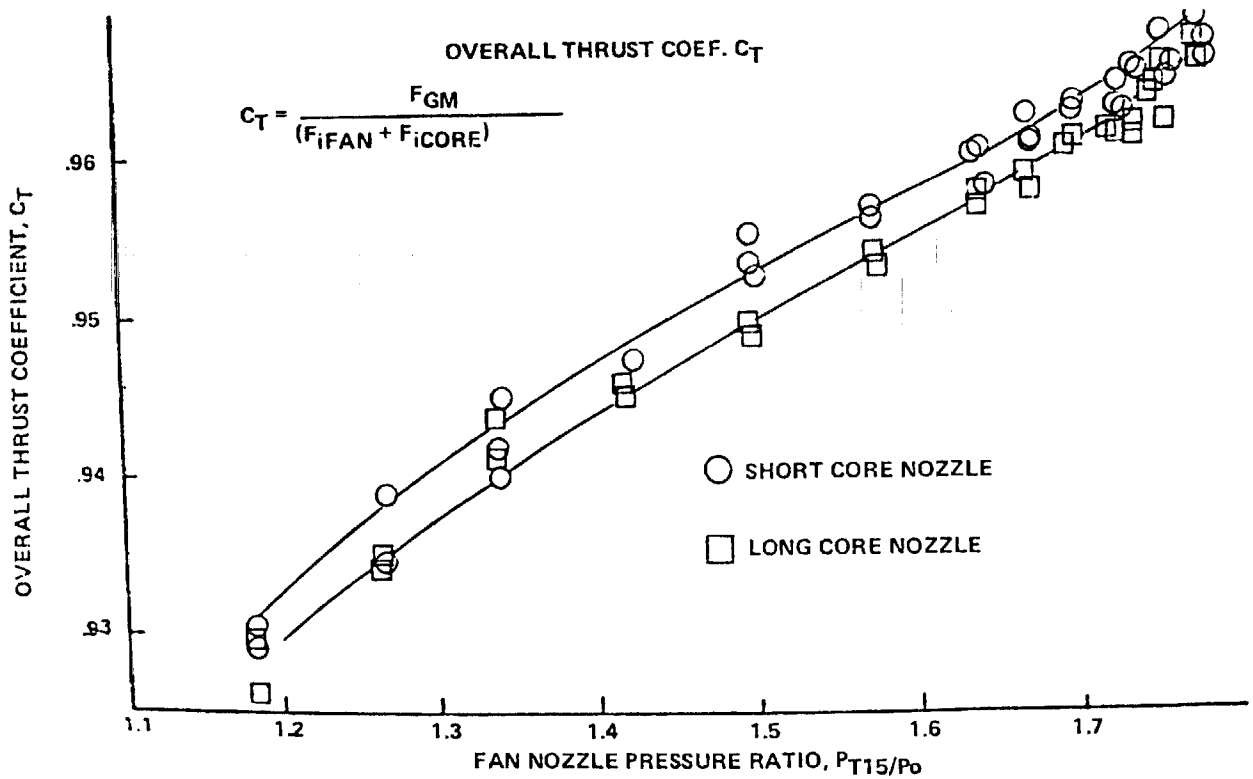
(Method A)



This curve indicates a thrust coefficient difference for the case where core flow is calculated based on low pressure turbine effective area. The magnitude of the thrust coefficient difference is again 0.25%.

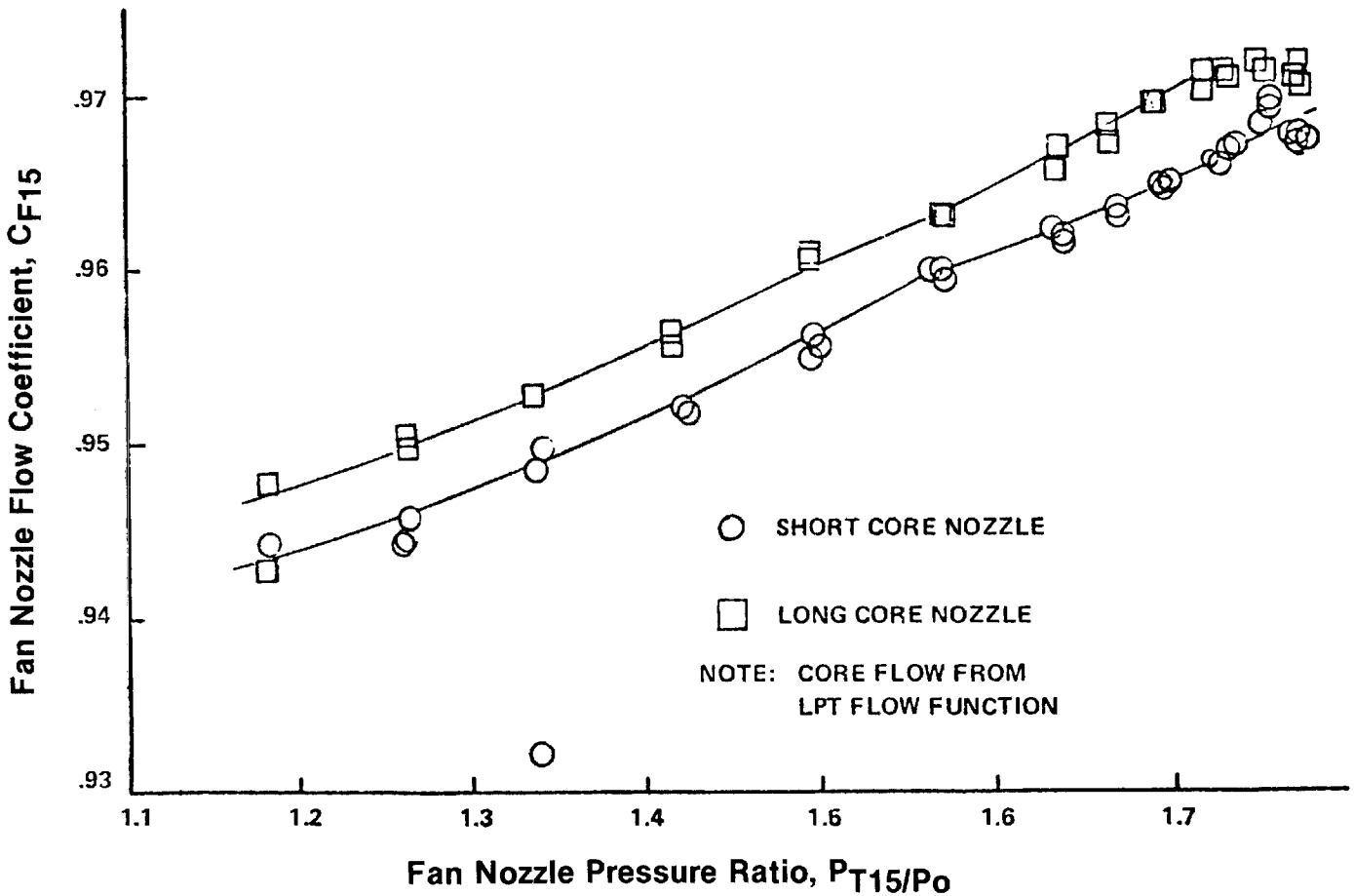
Overall Thrust Coefficient Based on LPT Flow Function & Measured Inlet Flow

(Method B)



The flow coefficient of the fan should not change for the different core nozzle configurations. This curve indicates data inconsistencies in either the total airflow measurement or the fan discharge rakes. These inconsistencies require another method of analysis (Method C) that is independent of the airflow or average nozzle pressures.

Fan Nozzle Flow Coefficient

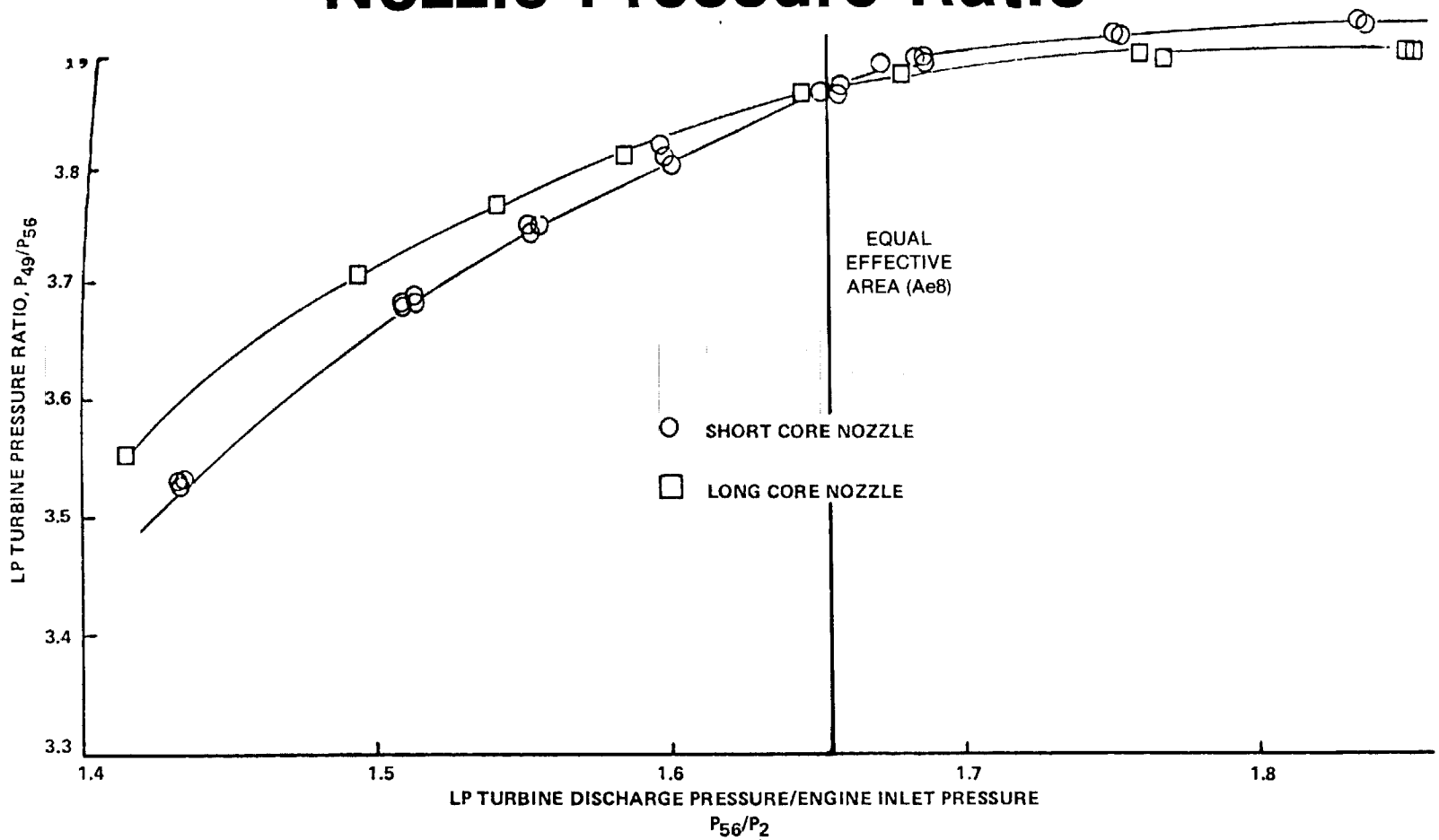


Method C

- Method C Based on Equal Effective Core Areas (Ae8)
- Equal Effective Core Areas is a Condition of Equal Ideal Thrust for the Long and Short Nozzles
- At Equal Ideal Thrust Overall Improvement is Reflected in Increase in Actual Measured Thrust Delta

The low pressure turbine pressure ratio is the same at the point in the power curve where the short and long nozzle effective areas Ae8 are identical. This point occurs at $\approx$ 1.65 PT5/PT2 and is consistent with 51,000 LBS thrust.

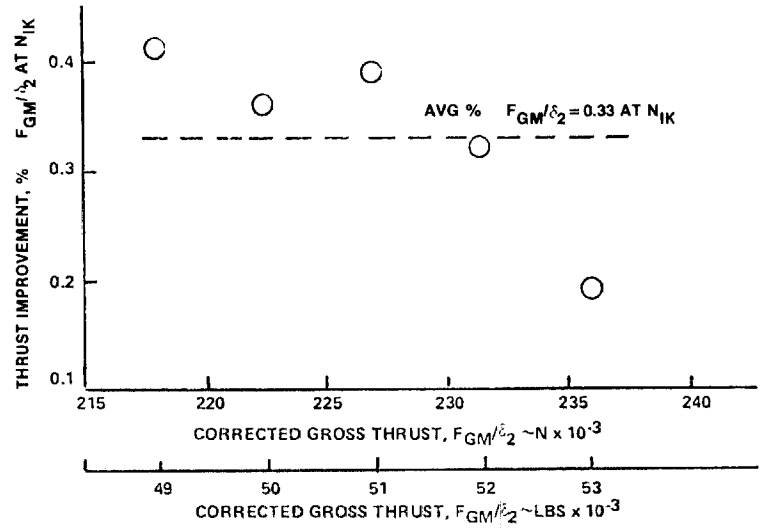
LPT Pressure Ratio Versus Nozzle Pressure Ratio



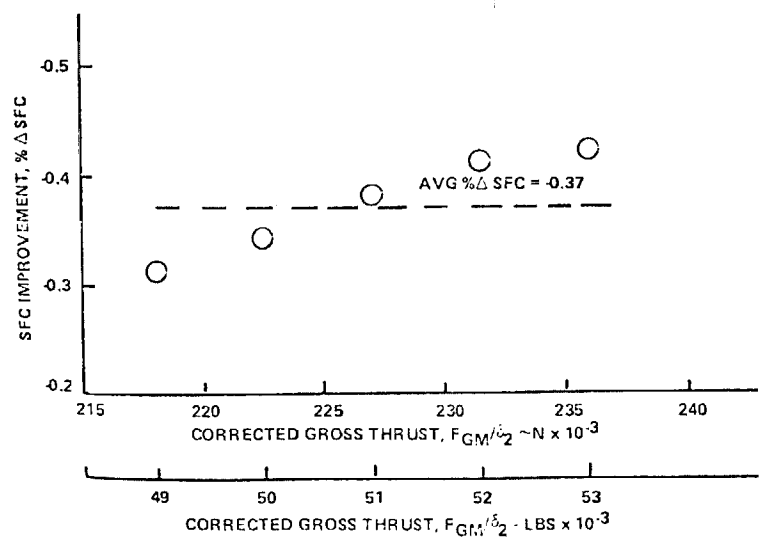
GENERAL ELECTRIC COMPANY
AIRCRAFT ENGINE GROUP

The measured thrust difference between nozzles at the 51,000 lb. point is indicated in thrust at constant corrected fan speed and again in SFC at thrust. A short portion of these curves (thrust vs. corrected fan speed and SFC vs. thrust) is presented in the region of 51,000 lbs. thrust and averaged to established a measured thrust difference between short and long core nozzles.

Thrust Improvement With Short Nozzle



SFC Improvement With Short Nozzle



Summary Of Results

- Methods Based on Calculated Core Flow and Measured Inlet Flow Result in $\Delta CT = 0.25\%$
- At Equal Ideal Thrust the Measured Thrust Indicates $\Delta CT = 0.33\%$ and $\Delta SFC = -0.37\%$

Power management has been established based on the long core nozzle corrected thrust vs. corrected fan speed characteristics. The short core nozzle thrust vs. fan speed characteristic results in the same or slightly higher thrust at fan speed and therefore no power management changes are necessary.

Conclusions

- Close Agreement Between Full Scale and Model Test at Sea Level Verifies $\Delta CT = 0.35\%$ Determined from Scale Model Tests.
- Model Test Results Simulating Altitude Operation can be Used for Estimating "On-Wing" Improvements.
- Short Core Nozzle Does Not Require a Power Management Change

Acoustic Testing

- Objectives
  - To Establish the Acoustical Effects of the Short Core Exhaust Nozzle on CF6-50 Engine Noise
  - To Assess the Impact of the Short Core Exhaust Nozzle on Community Noise Levels for Typical Aircraft Approach and Takeoff Flight Conditions
- Status
  - Test Completed

Test Program

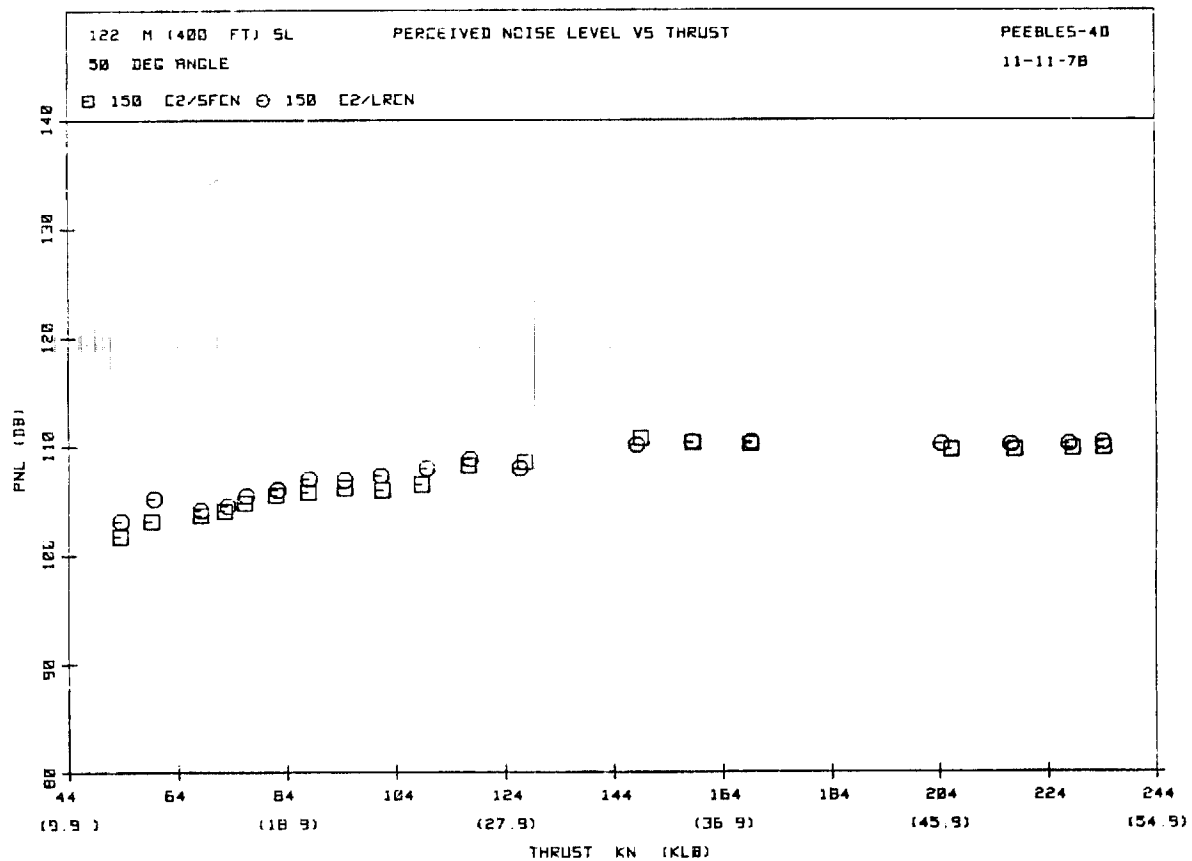
- Back-to-Back Testing of Long Reversing Core Nozzle (LRCN) and Short Core Nozzle (SCN)
- All Testing Performed at the Peebles Outdoor Test Facility
- Same Test Procedure as Fan Noise Test

Results

- No Significant Changes in Engine Noise Spectra
- Dominant Noise Components (Fan, LPT, and Core Jet) Were Not Significantly Impacted
- Perceived Noise Level Comparisons as a Function of Thrust Show No Significant Changes

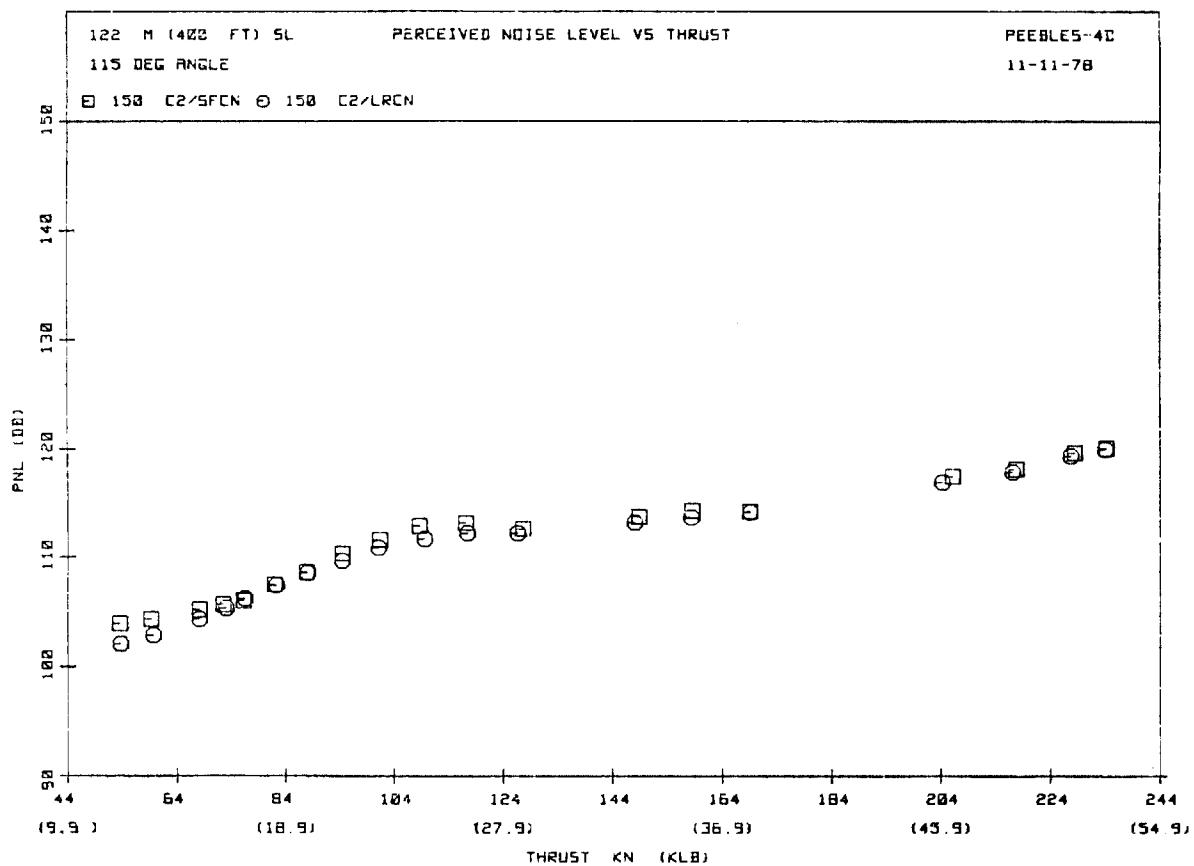
Data averaged and extrapolated to a 400 ft. sideline. Data exhibited for peak forward and aft angles.

Perceived Noise Level Comparison For 50 Degrees At 122 M Sideline



Data averaged and extrapolated to a 400 ft. sideline. Data exhibited for peak forward and aft angles.

Perceived Noise Level Comparison For 115 Degrees At 122 M Sideline

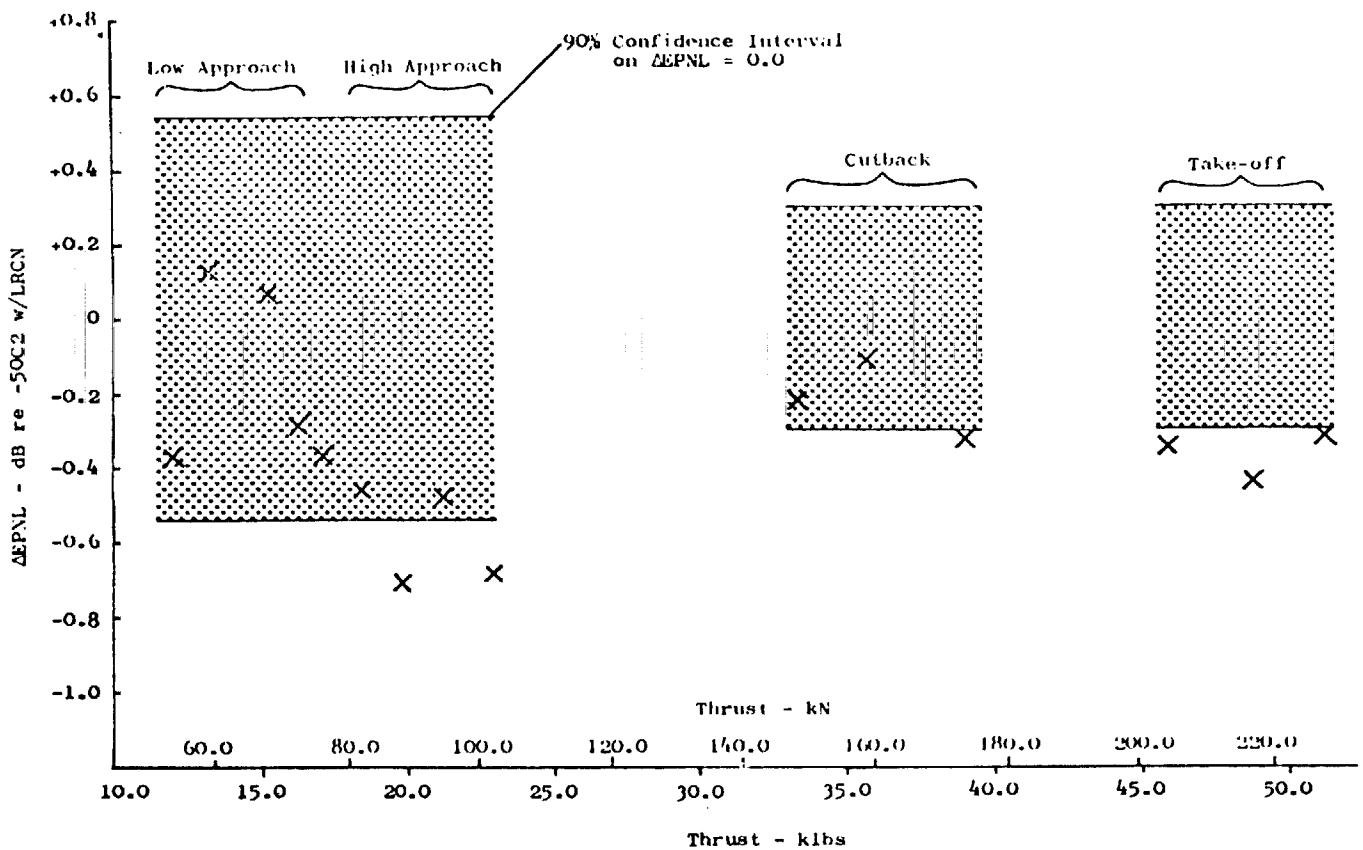


Community Noise Impact

- Simulated EPNL Comparisons as a Function of Thrust
- Results Show No Significant Change

Simulated EPNL computed by assuming that static PNL data were measured flight levels recorded at typical aircraft velocities. No flight effect corrections were applied to the data. The difference between configuration levels are compared to 90% confidence bands for the statistical hypothesis that $\triangle$ EPNL = 0.

CF6-50, Short Core Nozzle Community Noise Impact



Conclusions

- Short Core Nozzle Does Not Produce Any Significant Change in the CF6-50C2 Engine Configuration Noise Signature
- Short Core Nozzle Produces Equivalent or Lower Community Noise Levels Than the Long Reversing Core Nozzle for the CF6-50C2 Engine Configuration
- FAA has Accepted the Acoustic Equivalency of the Short Core Nozzle with the Long Reversing Core Nozzle

Summary

Short Core Nozzle

- Program Completed
- Internal Performance Improvement at Cruise — 0.9% ΔSFC
- Improvement Demonstrated in Flight Tests
- Short Core Certified and in Production
- Sold to (6) Airlines on CF6-50C2

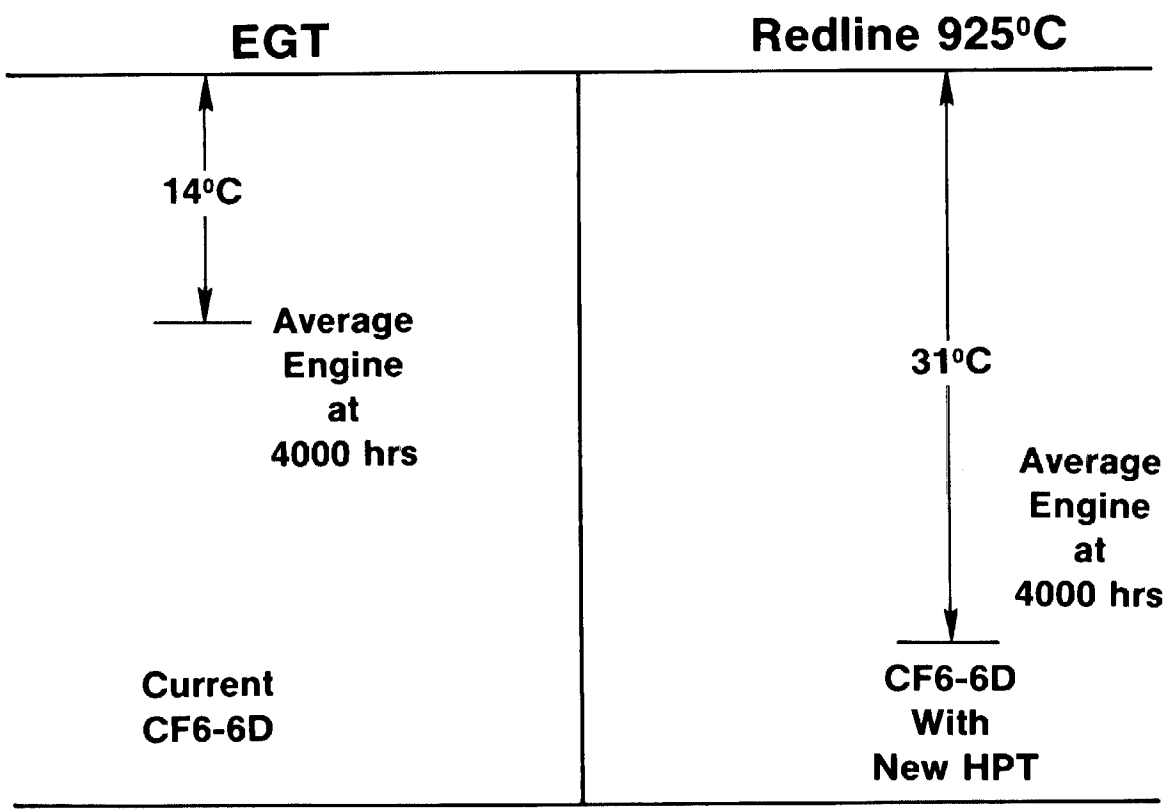
CF6-6
Jet Engine Performance
Improvement Program

Task 2.3
HP Turbine Aerodynamic Performance Improvement Concept

Performance Improvement Features

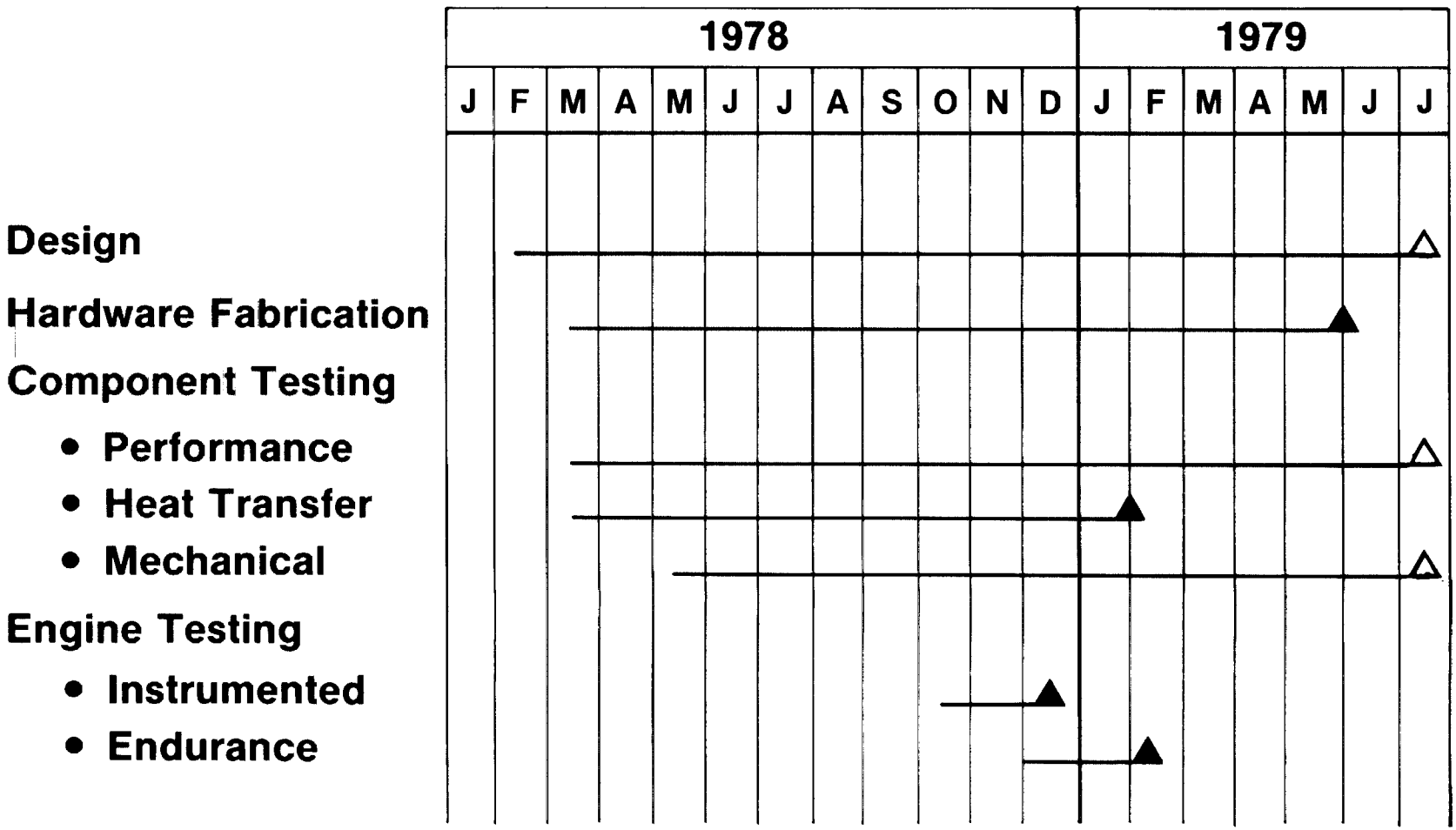
- Improved Stage 2 Vane Aerodynamics
- Reduced Exit Swirl
- Improved Cooling Effectiveness
- Improved Wheel Space Seals
- Tighter Tip Clearance
- No Mating Face Shank Leakage

CF6-6D



- –1.0% SFC Improvement (Relative to Current HPT)
- 17°C Improvement at 4000 Hours

NASA Program



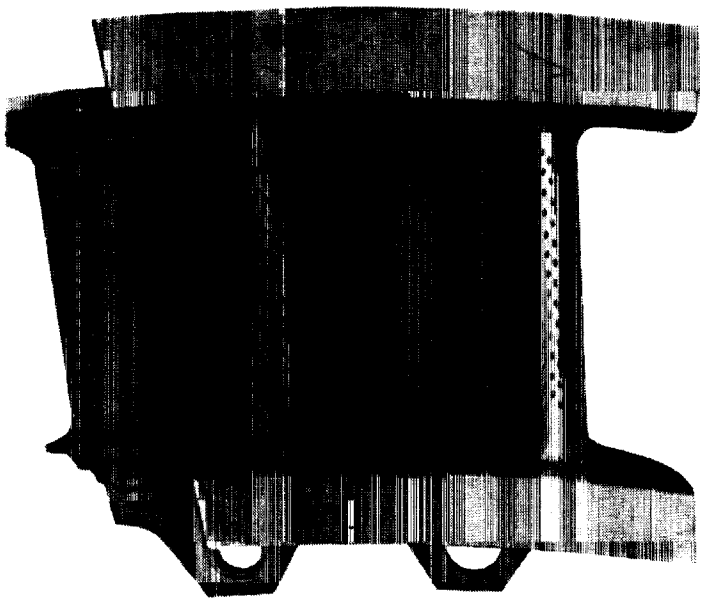
Component Test Program

	Completion Dates
Performance Tests	
• Stage 1 Vane Cascade Test	8/79
• Stage 2 Vane Leading Edge Flow Evaluation	Complete
Heat Transfer Tests	
• Model Test of Stage 1 Vane Trailing Edge Cooling Passages	Complete
• Stage 2 Leading Edge Heat Transfer Evaluation	Complete
• Stage 2 Vane Trailing Edge and Pin Fin Test	Complete
Mechanical Tests	
• Turbine Blade Dynamic & Steady State Strain Distribution	Complete
• Frequency and Amplitude Responses As a Function of Damper Force	Complete
• Rim Area Model Stress Test	8/79

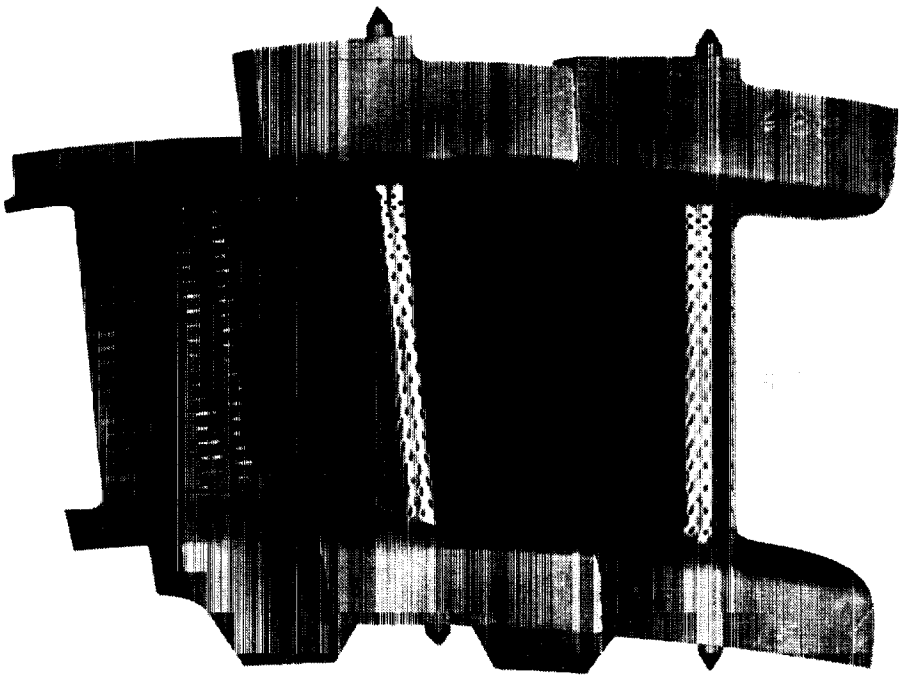
Stage 1 Vane Cascade Test

- **Objective:**
  - Evaluate the Aerodynamic Efficiency of the Current Production and the Improved Stage 1 HPT Vanes.
- **Status:**
  - Testing and Data Reduction are in Progress.

Test Hardware



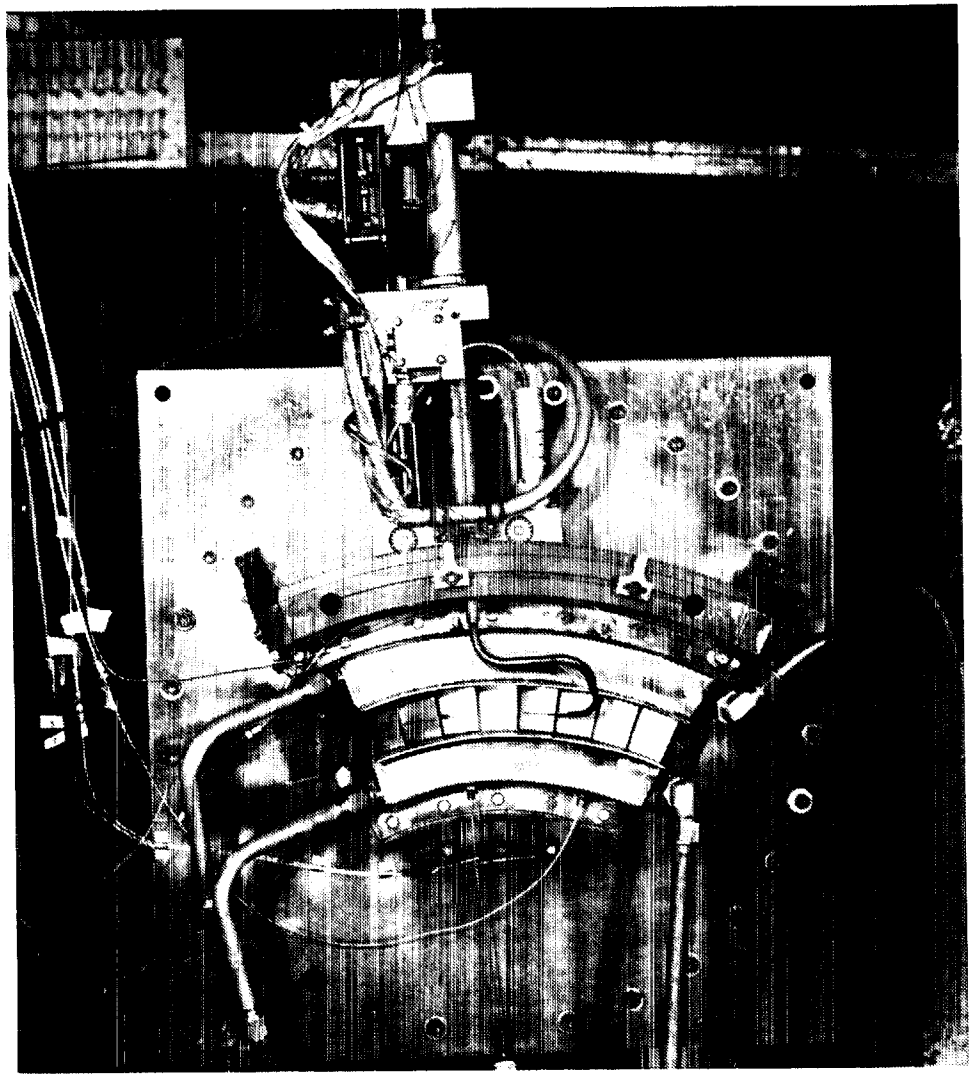
Current Production



Improved Design

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Test Set-Up



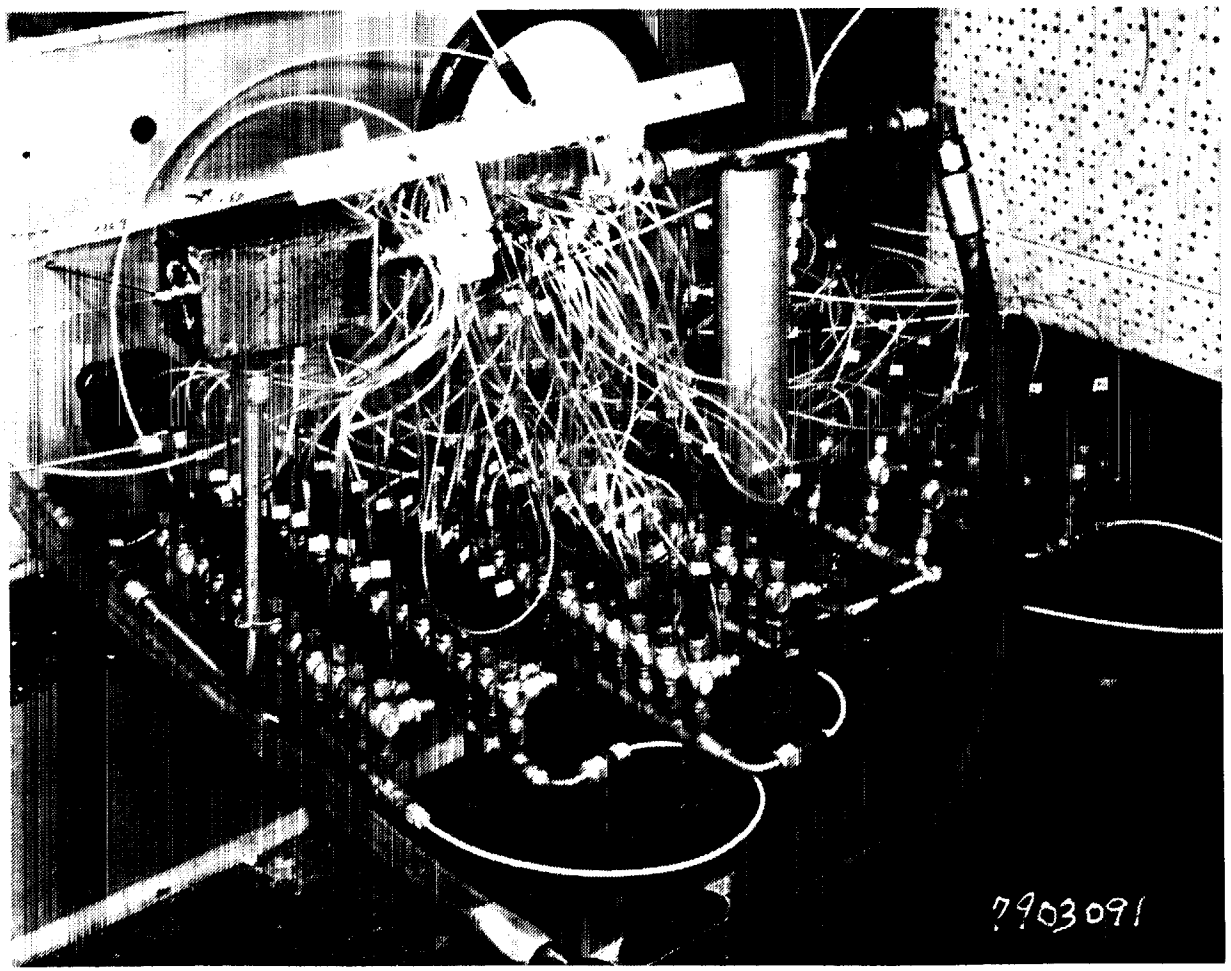
ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Stage 2 Vane Leading Edge Flow Evaluation

- **Objective:**
  - Detemine the Vane Flow Distribution Between the Vane Shell and the Impingement Insert, and Radially within the Insert for the Current Stage Two Vanes.
- **Status:**
  - Test Completed

To determine the pressure distribution between the vane shell and insert, pressure taps were installed through the vane shell on the pressure and suction surfaces. A pressure probe was traversed radially within the insert. Air flow was measured at the outer band supply to the vane and inner band discharge with the trailing edge discharging to atmospheric pressure. Engine design pressure ratio conditions were set between inlet and discharge. Pressures and flows were recorded after stabilizing.

Test Set-Up



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Conclusions

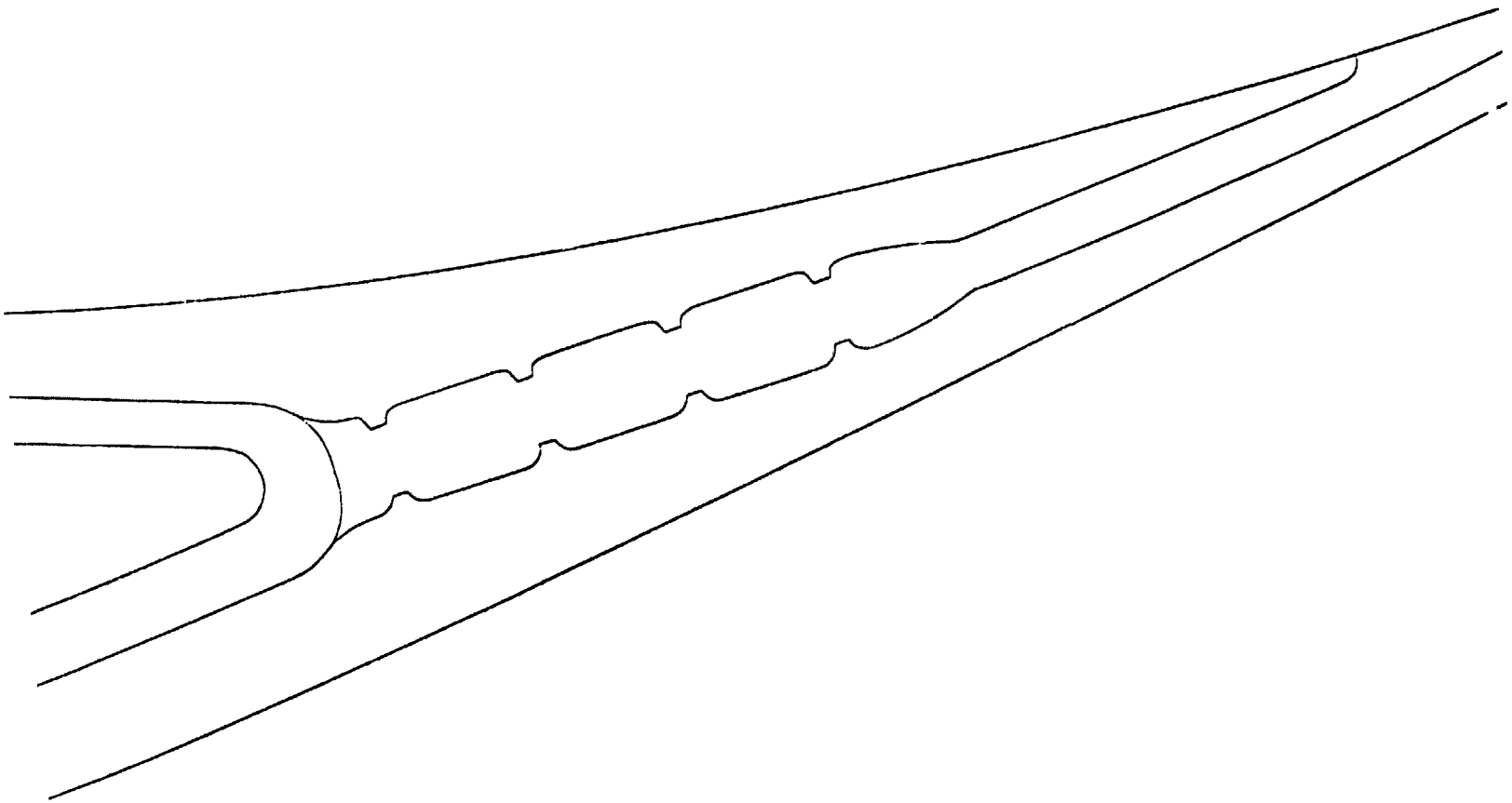
- 0.2% W25 Reduction in Cooling Flow Has Been Achieved
- Total Flow Meets Design Intent
- Adjustment of Trailing Edge Area is Required to Achieve Design Flow Split
- Radial Pressure Gradient Within Insert Has Been Reduced Eliminating the Low Impingement ΔP Condition at the Outer Band
- Pressure Distribution Between Vane and Shell is as Expected

Stage 1 Vane Trailing Edge Heat Transfer

- **Objective:**
  - Determine the Heat Transfer Characteristics of the High Pressure Turbine Stage One Vane Trailing Edge Flow Passages.
- **Status:**
  - Test Completed

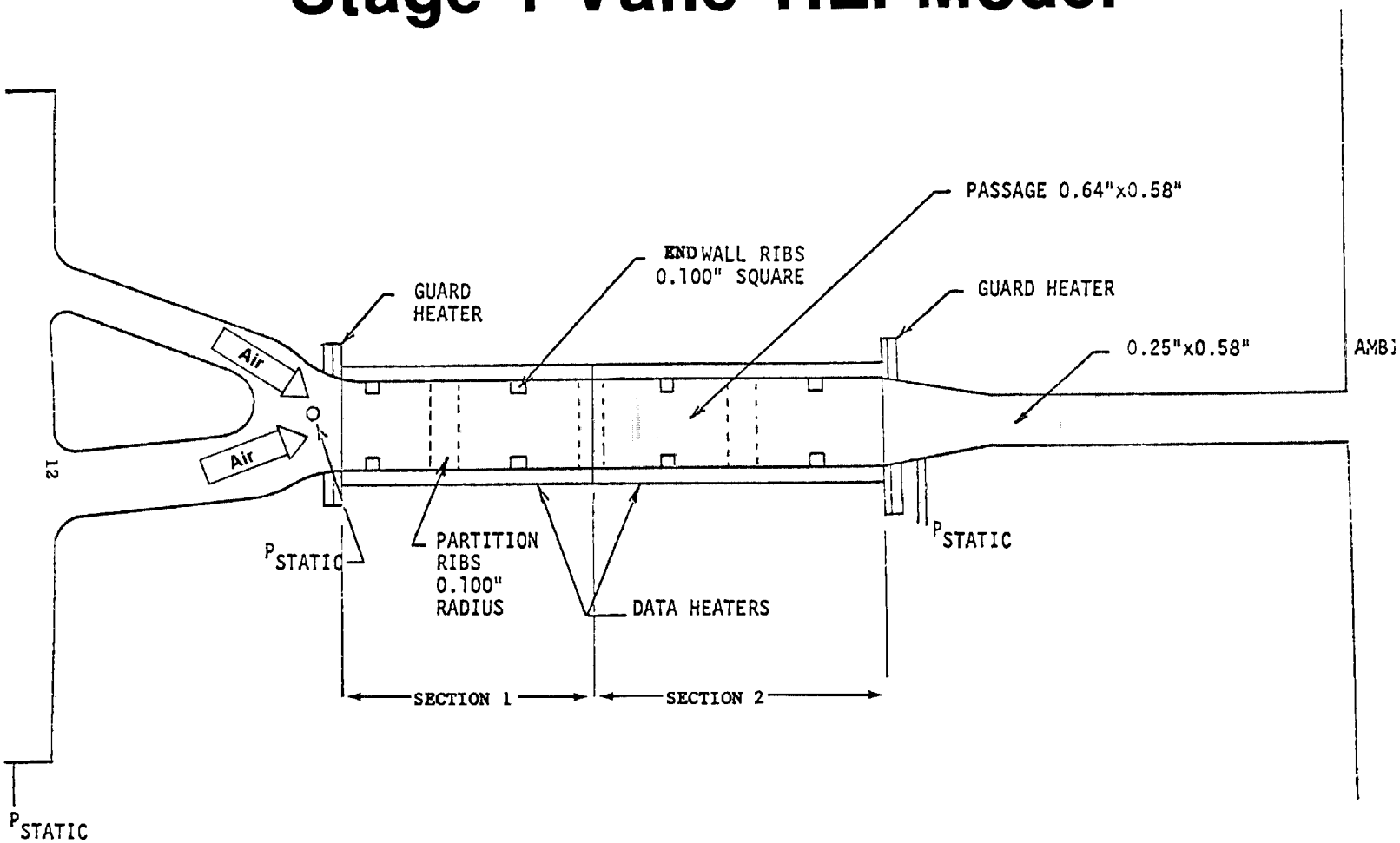
Cross section of vane showing ribbed portion of trailing edge flow circuit which was investigated.

Pitchline — Stage 1 Vane T.E. Cross Section



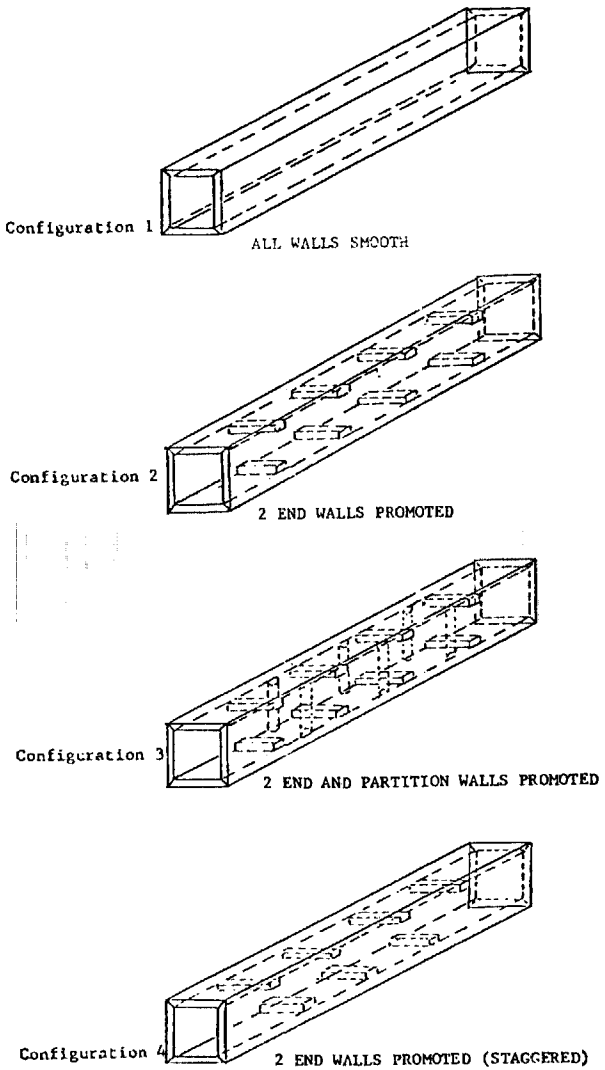
Cross section of 10X scale model of vane shown previously. Four data heaters were installed in both sections 1 and 2 to define heat transfer distribution in the passage.

Stage 1 Vane T.E. Model



Four test geometries which were installed in sections 1 and 2 of model shown previously for comparison of heat transfer performance.

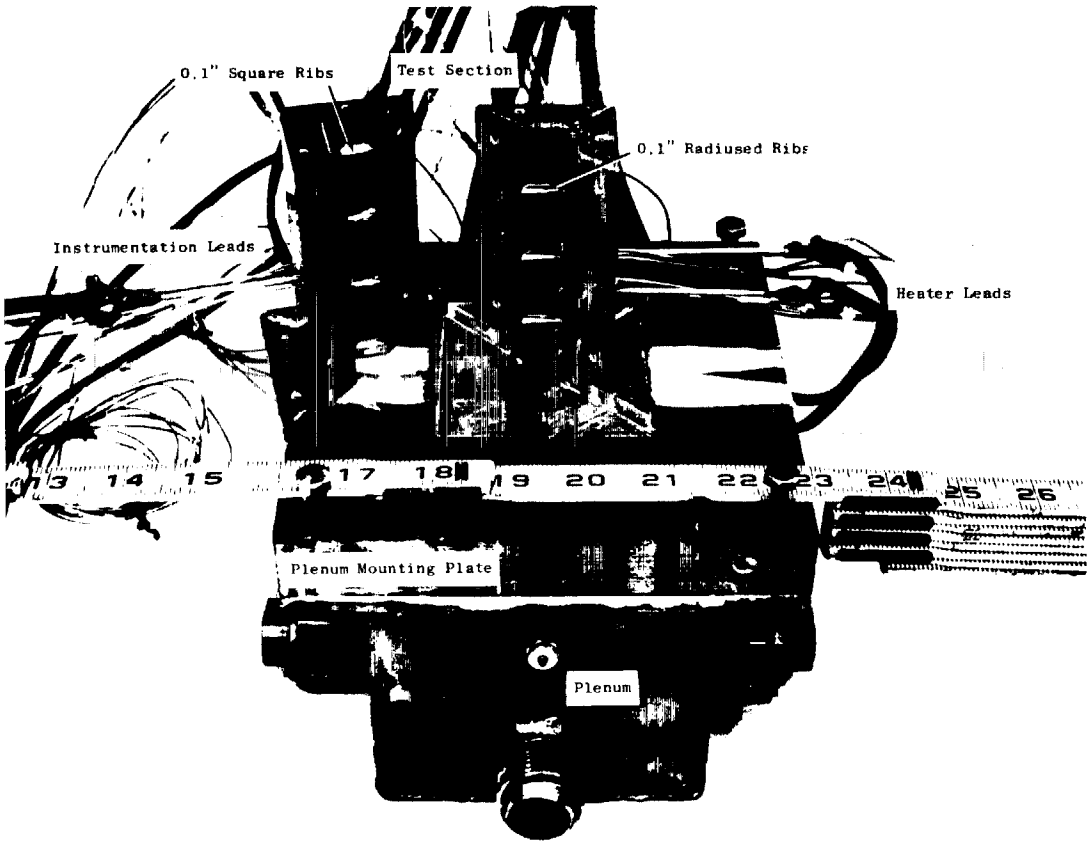
Test Geometries



Photograph of test model with test section opened up to show test geometry. Test model is mounted on flow supply plenum. Electrical heater leads provide known heat inputs to copper plates which are used for calculation of heat transfer performance.

Configuration No. 3

(Two Endwalls Promoted/Two Promoted Partition Walls)



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Average heat transfer in ribbed section compared to smooth walls. the configuration with two wall promoted and the ribs staggered was chosen based on its net effect on vane temperatures.

Ribbed Wall Results Compared To Smooth Wall

Configuration	Heat Transfer Promotion
All Walls Smooth	1.0
4 Walls Promoted	
• Ribbed Partiton Walls	2.47
• Ribbed End Walls	3.21
2 Walls Promoted	
• Smooth Partition Wall	1.67
• Ribbed End Wall	2.79
2 Walls Promoted Staggered	
• Smooth Partition Walls	1.70
• Ribbed End Walls	2.44

Conclusions

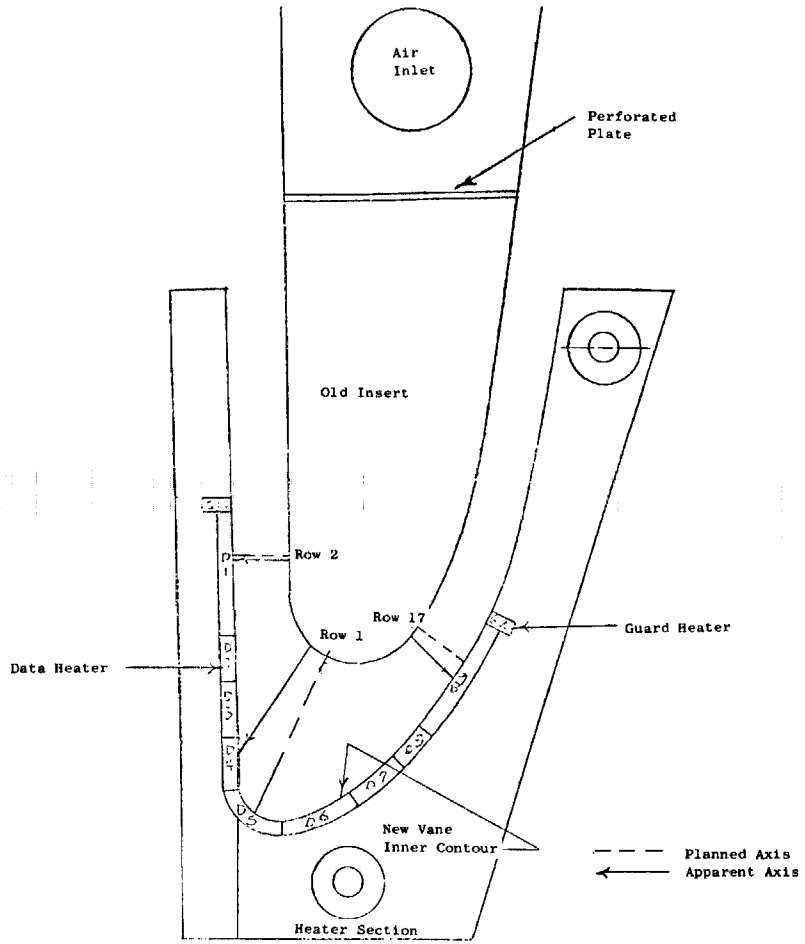
- Removing Turbulence Promoters From Partitions Walls and Staggering Results in Temperature Increases of Only 10°F in Promoted Region. Reduces Temperature at Trailing Edge by 30°F.

Stage 2 Vane L.E. Impingement

- **Objective:**
  - Investigate the Leading Edge Region Impingement Heat Transfer Characteristics of Both The Current Production and Improved Stage Two Vane Impingement Inserts
- **Status:**
  - Test Completed

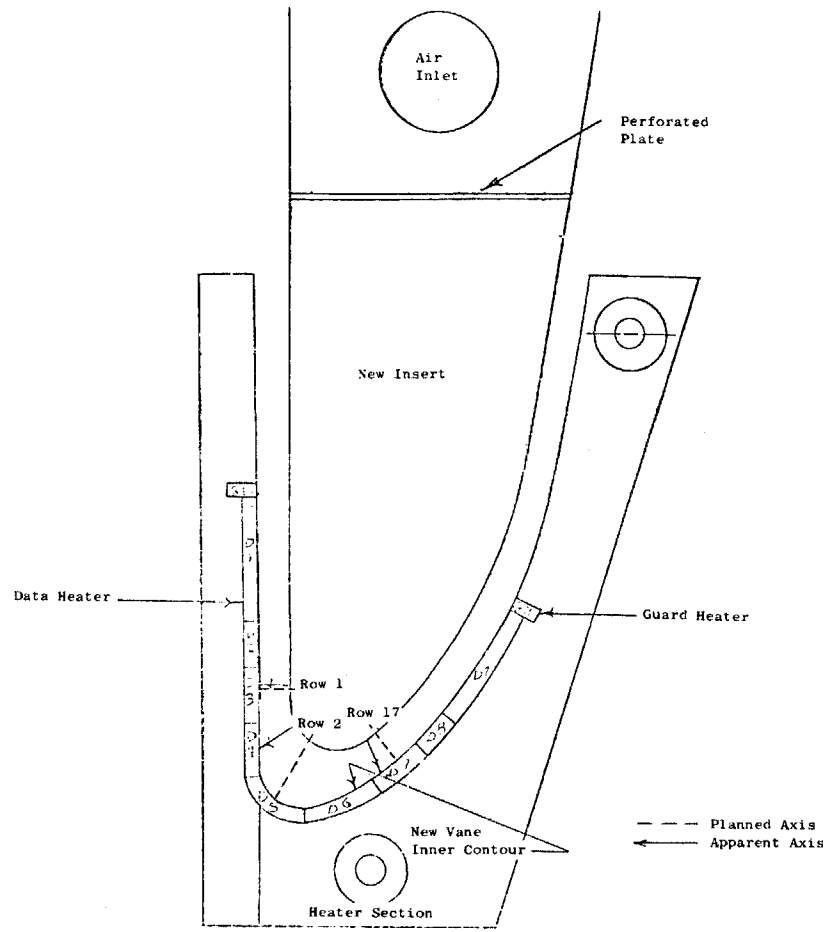
Cross-section of current production vane leading edge model. Model was 11.5X engine scale. Note wide spacing between leading edge impingement point and insert. Heat transfer data were taken at data heater positions shown.

Current Stage-2 Vane Insert Model



Cross-section of improved vane leading edge model. This model was scaled to 7.0X engine size to better match contour of heater plate model for current production vane. Note increased concentration of cooling at leading edge.

Improved Stage-2 Vane Insert Model



Conclusions

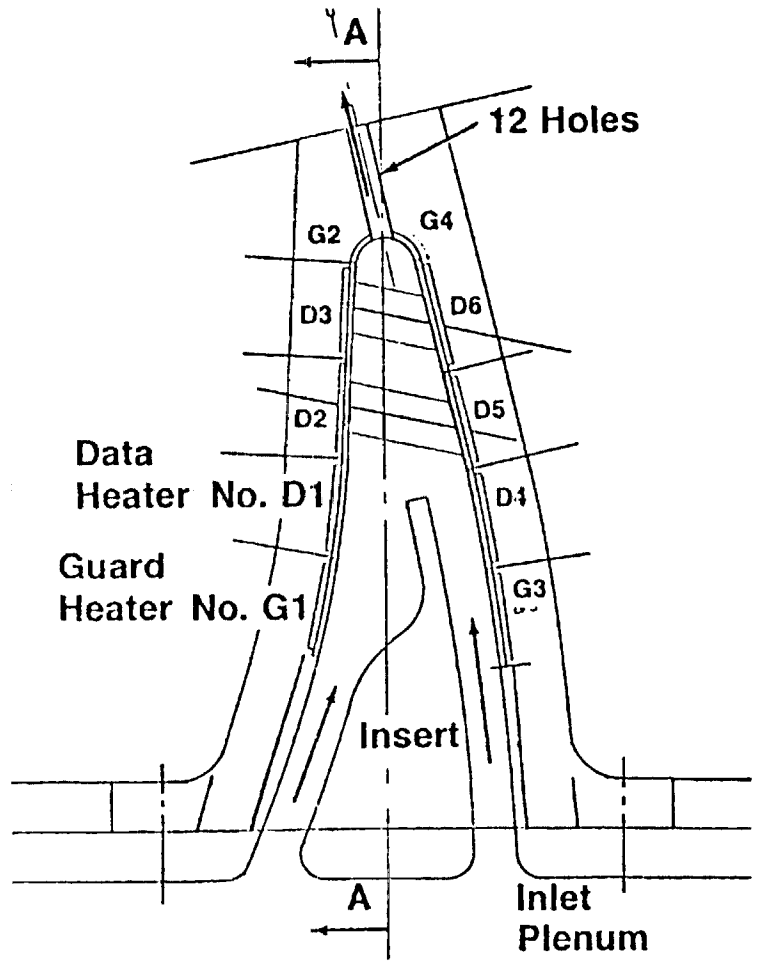
- Improved Leading Edge Impingement Geometry has Reduced Vane Leading Edge Temperature by 25°F.

Stage 2 Vane Trailing Edge Heat Transfer

- **Objective:**
  - Determine the Heat Transfer Characteristics of the Pin Fin Geometry Selected for the Stage Two Vane Trailing Edge.
- **Status:**
  - Test Completed

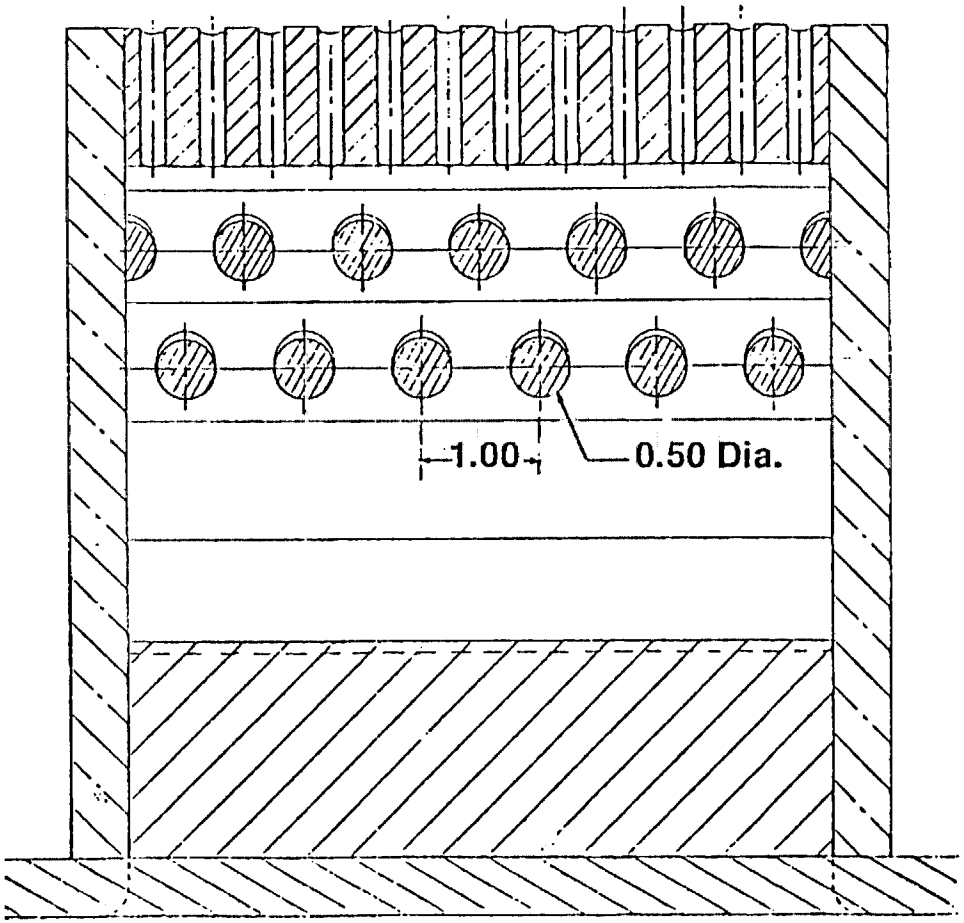
Cross-section of model of stage two vane trailing edge pin fin region. Model was 10X engine scale. Heat transfer data were taken at each data heater location using measured electrical power input. Tests were run with and without pin fins.

Stage 2 Vane Trailing Edge Model



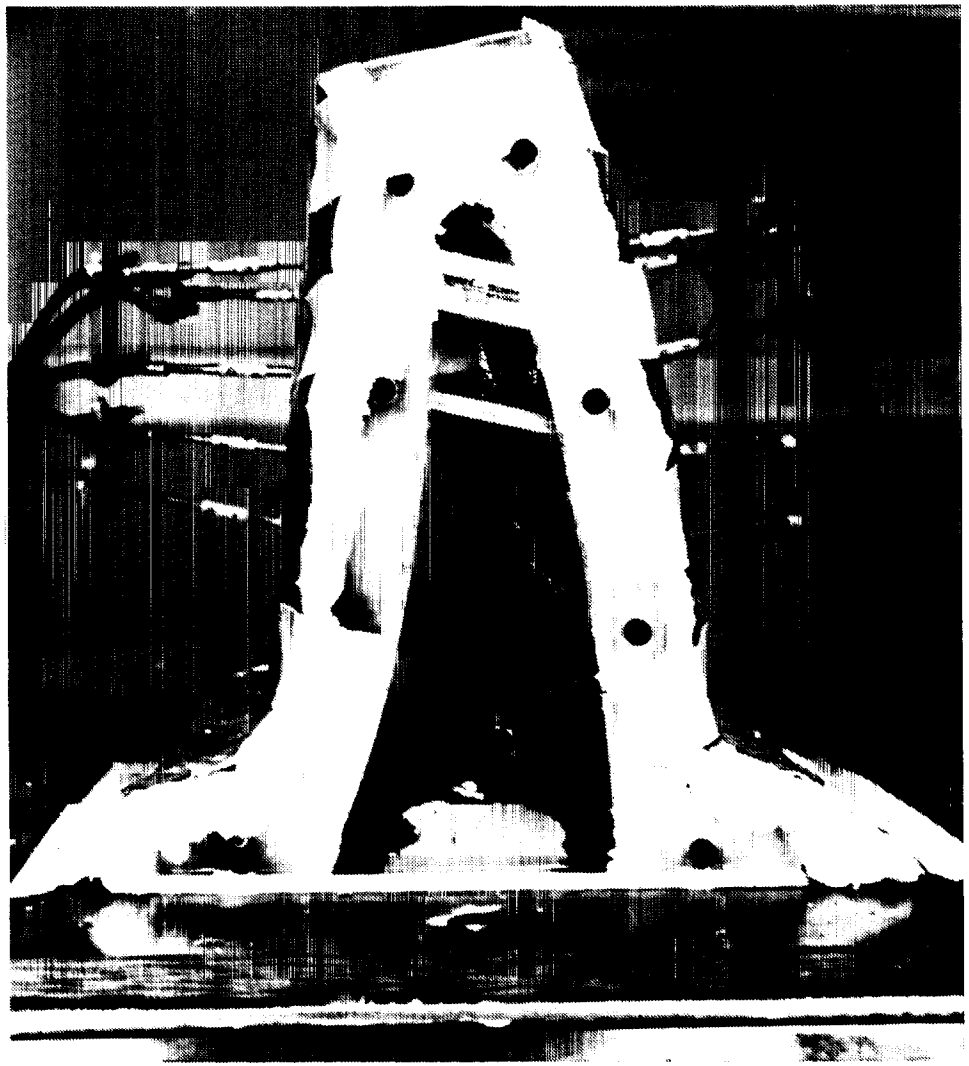
Planar cross-section of model illustrated pin spacing.

Stage 2 Vane Trailing Edge Model



Photograph of test model. Model was mounted on a plenum from which flow was supplied. Flow discharged to atmospheric pressure. Leads shown are for electrical power input and thermocouples for heater plate temperatures.

Stage 2 Vane Trailing Edge Model



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Conclusion

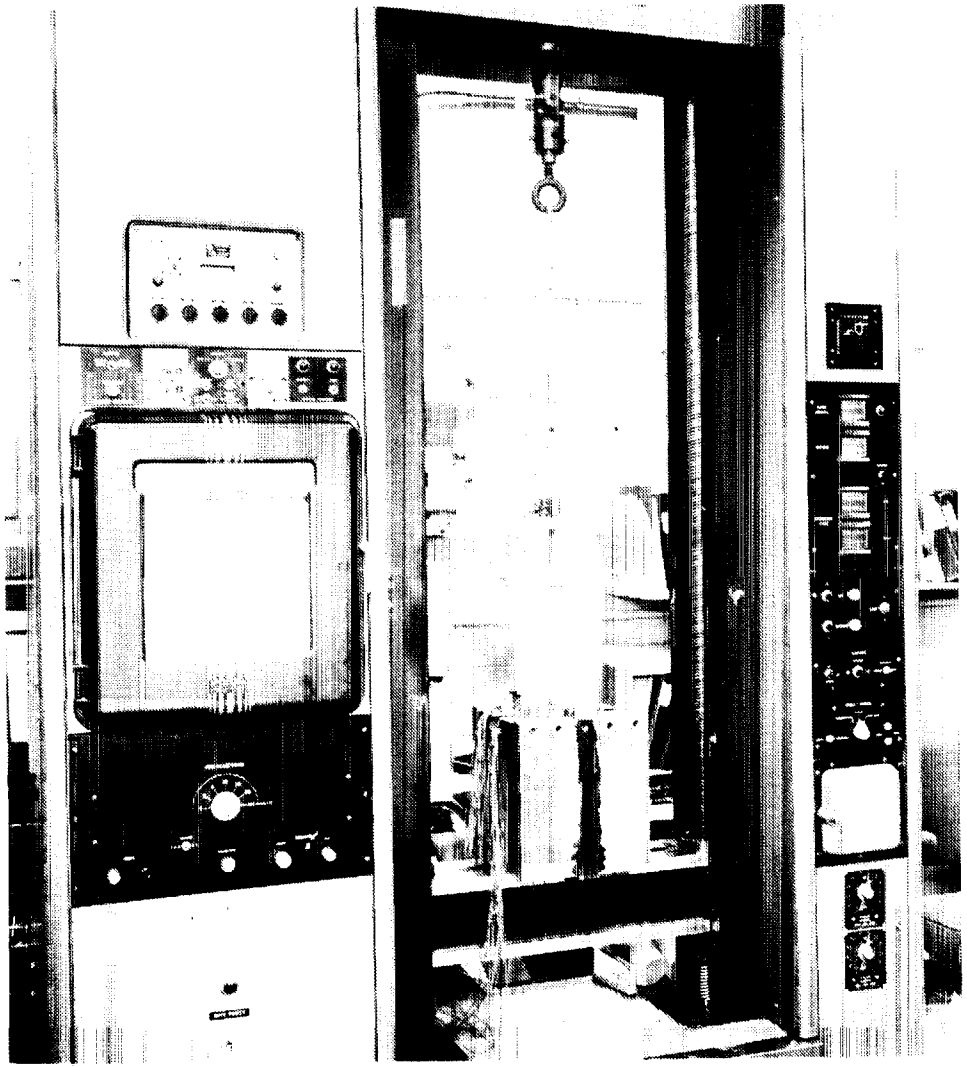
- The Selected Pin Fin Arrangement Increased the Wall Heat Transfer to 3× That of a Smooth Wall.
- Test Results are Currently Being Incorporated into Analysis

Blade Dynamic and Steady-State Strain Distribution

- **Objective:**
  - To Determine the Dynamic and Steady-State Strain Distribution in the Stage 1 and 2 Single Shroud Turbine Blades
- **Status:**
  - Test Completed

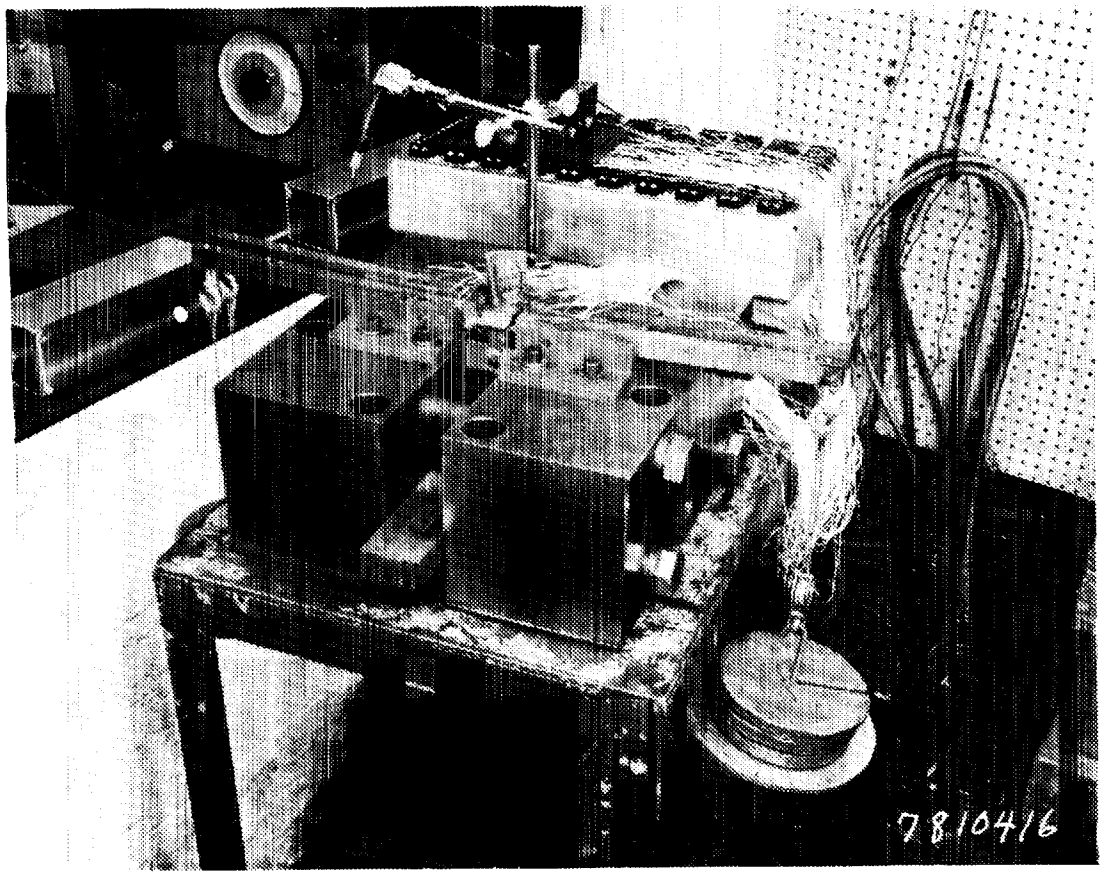
Photo showing a typical test setup to determine blade steady state strain distribution.

Steady-State Test Set-Up



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

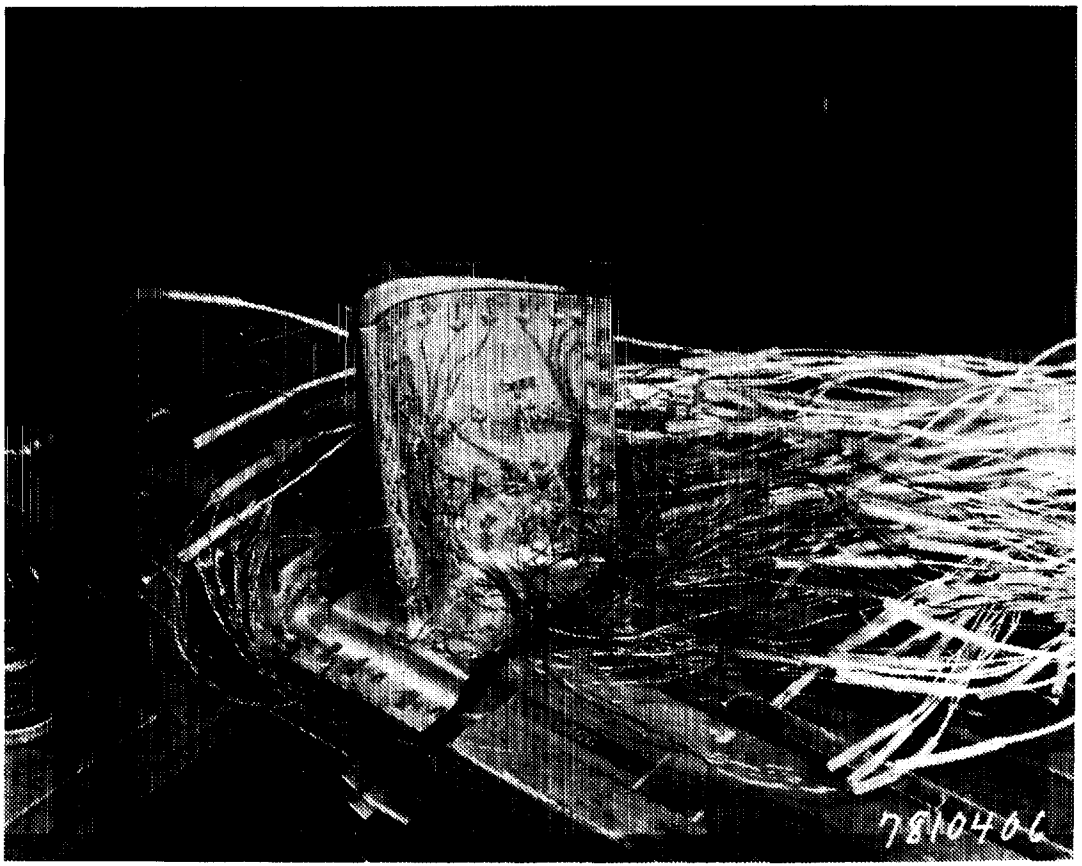
Component Test Setup for Stress Distribution.



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Instrumented turbine blade showing strain gage application to determine vibratory stress distribution.

Closeup View of Foil Type Strain Gages



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Results

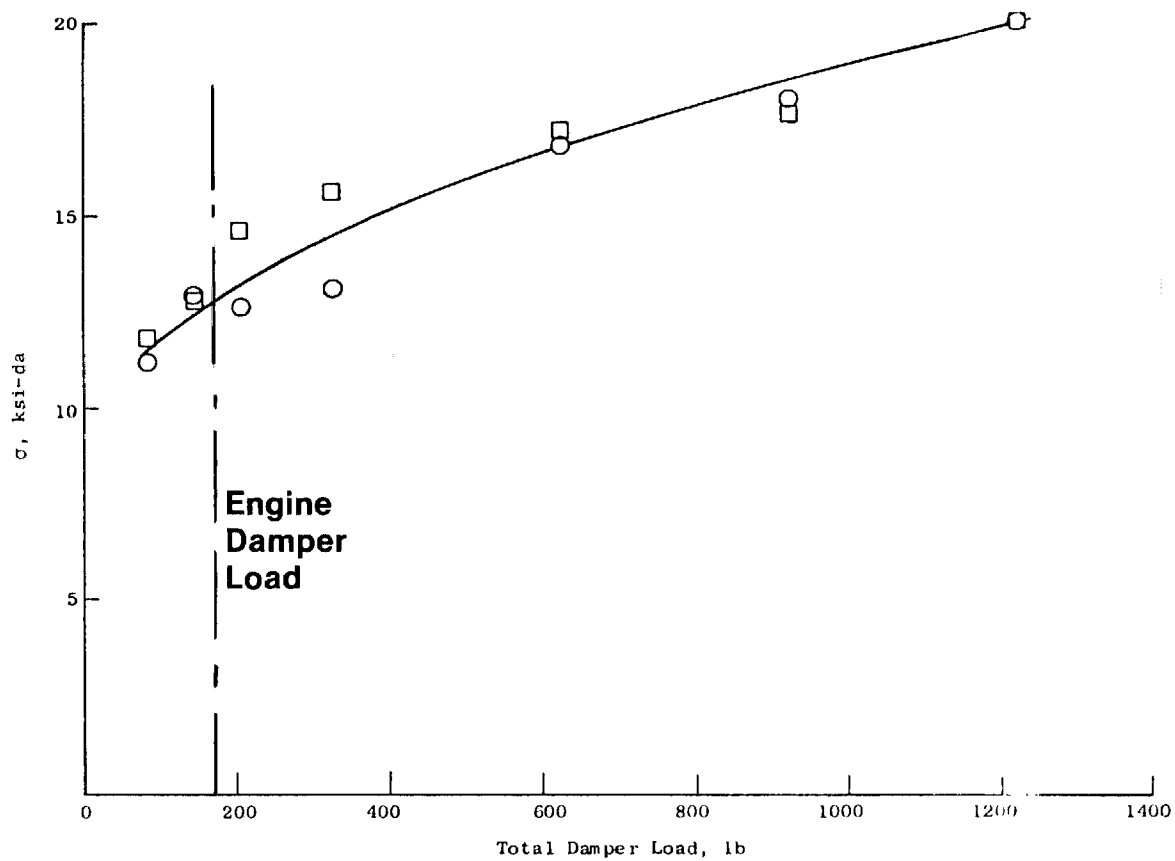
- Steady State and Vibratory Strain Distributions Successfully Determined for Stage 1 and Stage 2 Blades.
- Results of the Steady State Distribution Testing Revealed No Deleterious End Effects
- Results of the Vibratory Stress Distribution Were Used to Determine Engine Strain Gage Locations and to Determine Blade Vibratory Stress Capabilities and Safe Operating Limits

Frequency And Amplitude Response As Function Of Damper Force

- **Objective:**
  - To Determine the Blade Frequency and Response Amplitude as a Function of the Damper Load.
- **Status:**
  - Test Completed

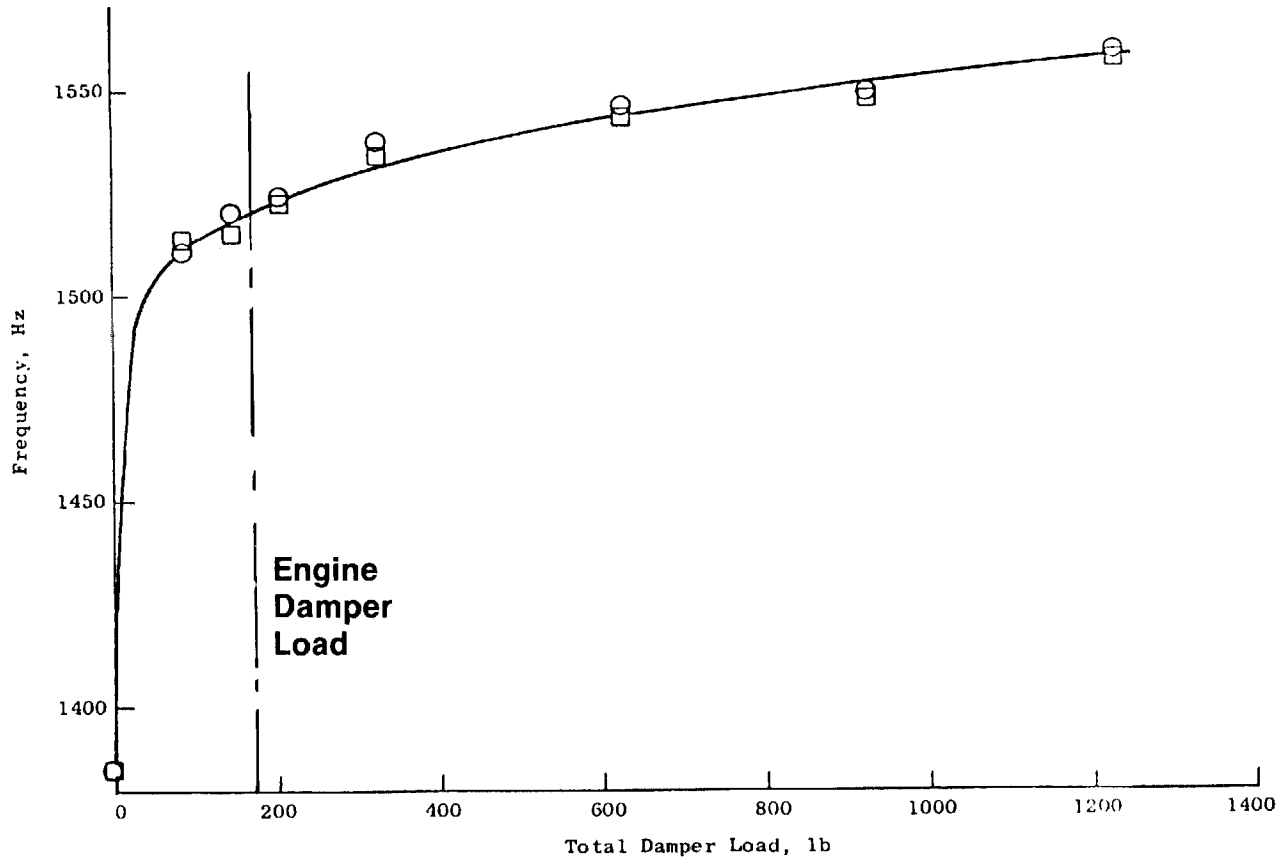
Blade stress vs. damper load chart shows decrease in stress as damper load decreases. Note that engine damper load takes advantage of the majority of the stress reduction available.

Root Leading Edge Stress Vs. Damper Load, Stage 2 Blade



Blade flex frequency vs. damper load. Note that engine damper load is high enough to raise the blade frequency significantly but low enough to provide good damping. (Reference previous chart).

Frequency Vs. Damper Load, Stage 2 Blade



Results

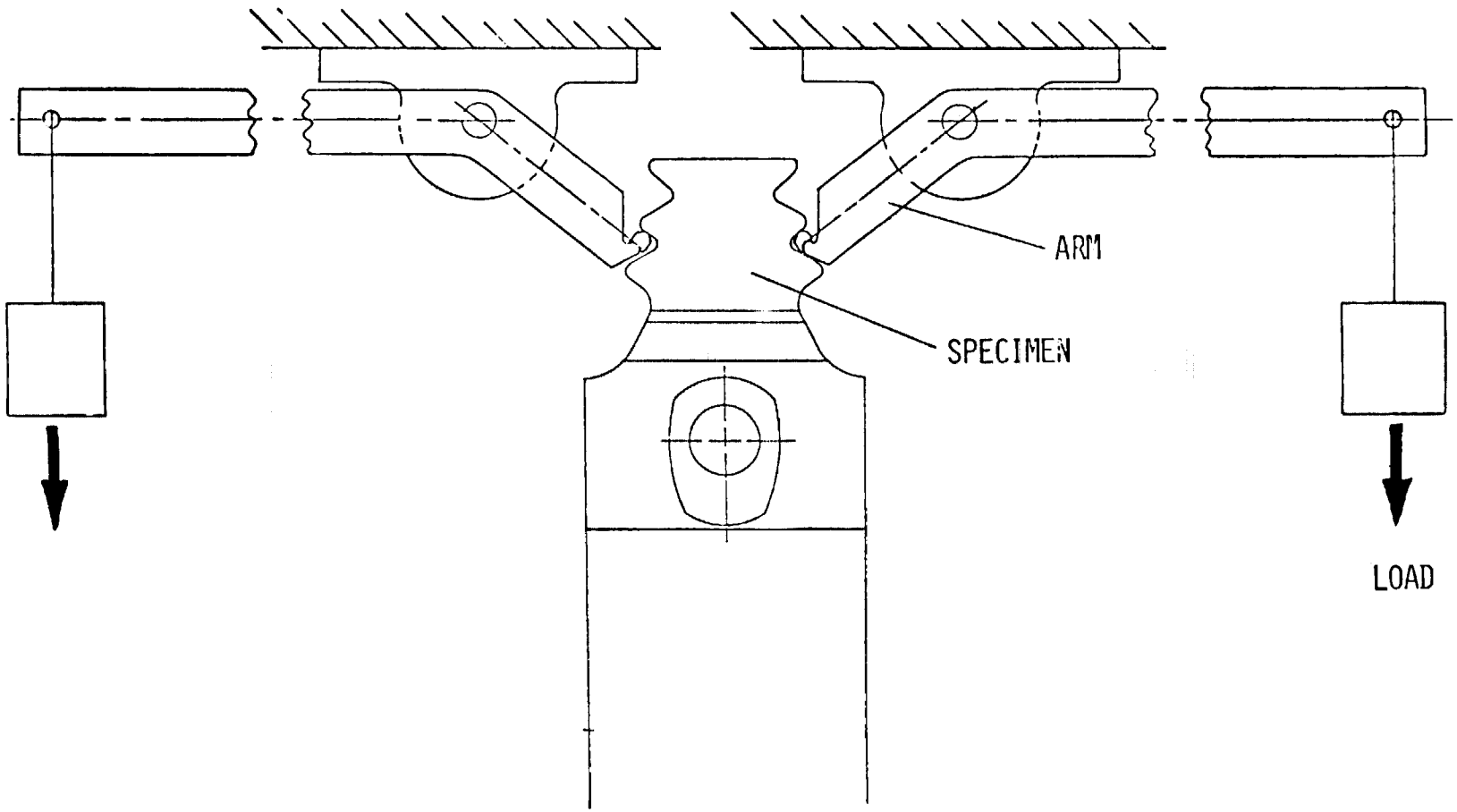
- Blade Frequency, Tip Deflection, and Root Leading Edge Stress Were Determined Versus Damper Load for the Stage 1 and 2 Blades
- Testing Confirmed Damper Effectiveness for Both Stages

Disk Rim Area Stress Distribution

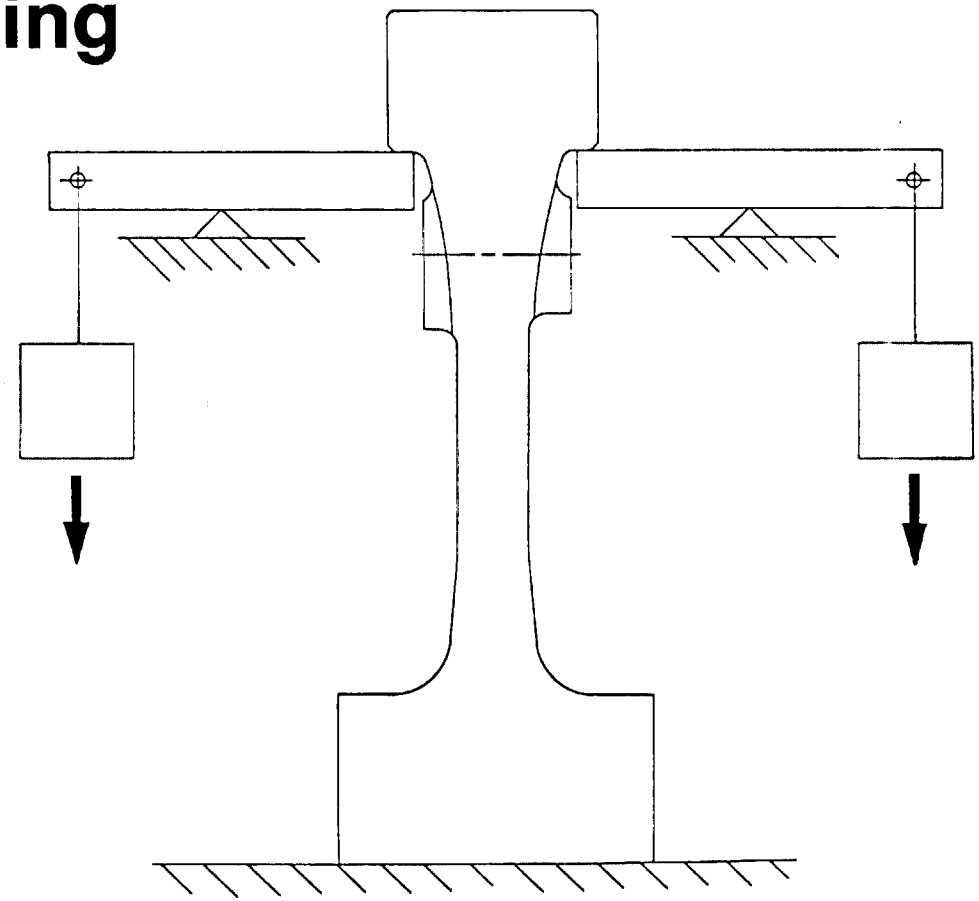
- **Objective:**
  - To Determine the Stress in the Rim Area of the Stage 1 and 2 Single Shank HP Turbine Disk to:
    1) Determine Local Stress Concentration
    2) Modify Model to Reduce High Local Stress
    3) Provide Input for Final Life Analysis
- **Status:**
  - Rim Area Models Have Been Manufactured and Delivered. Instrumentation Has Been Initiated.

Sketch showing typical model loading technique. Note use of seperate pullers to load each side of each tang. This permits the elimination of bending and also allows the effect of each seperate load on stress to be determined. Final results are analysed using principle of superposition.

Specimen Test Setup, Dovetail Loading



Specimen Test Setup, Rabbet Loading



Engine Testing Milestones

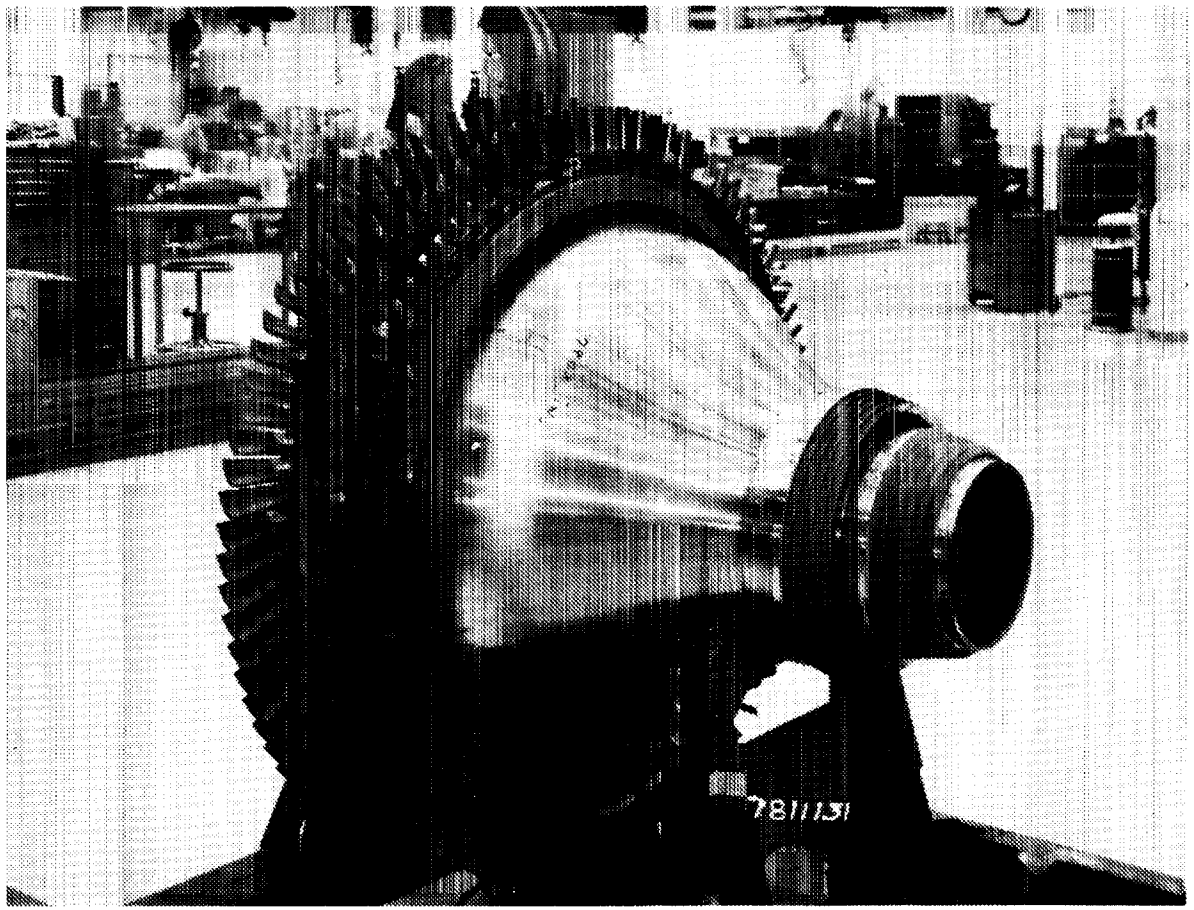
	Target to Fire
• Core Engine Baseline Test	Complete
— Current Production Hardware	
— GE Funded	
• Core Engine Instrumented Test	Complete
— New HPT Hardware	
• Core Engine Cyclic Test	Complete
— New HPT Hardware	
— Instrumented Hardware Replaced	
• Fan Engine Performance Test	7/79
— Back-to-back Testing of Current Production and New HPT Hardware	
— GE Funded	

Instrumented Engine Test

- **Objective**
  - Verify That Stresses, Temperatures and Flows Meet Design Requirements
- **Status**
  - Testing Completed

Instrumented Single Shank High Pressure Turbine Rotor Assembly for core test.

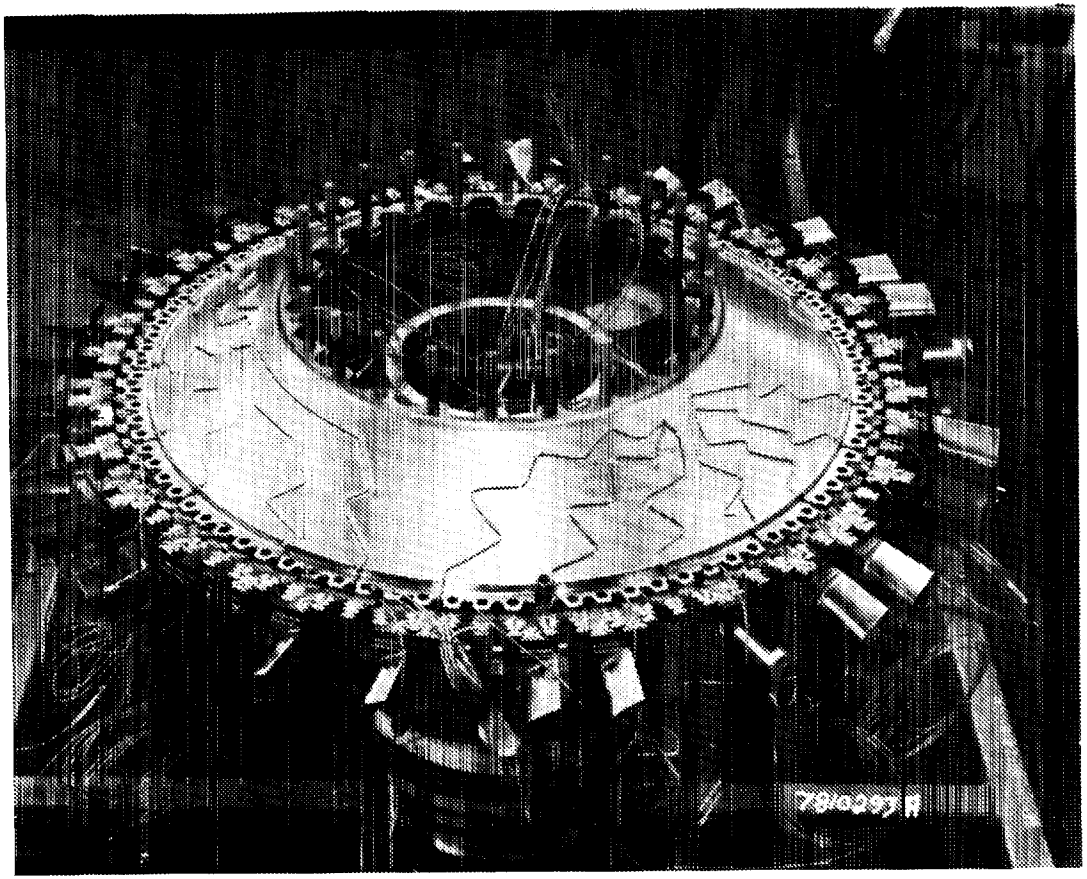
Single Shank Turbine Rotor Front End View.



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Instrumented Stage 1 blades, disk, and spacer.

Installation of Instrumented Stage 1 Turbine Blades

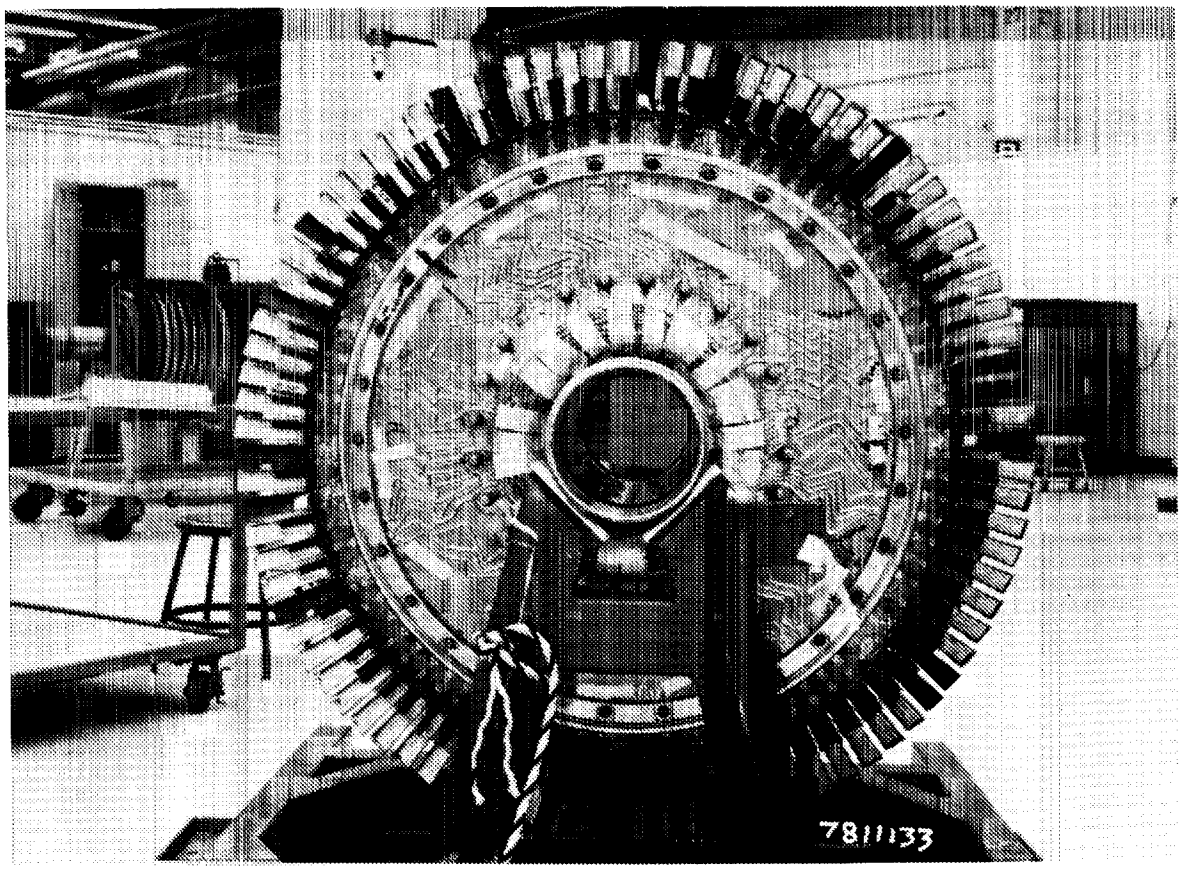


ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

C-3

Aft looking forward view of instrumented rotor showing amount of instrumentation.

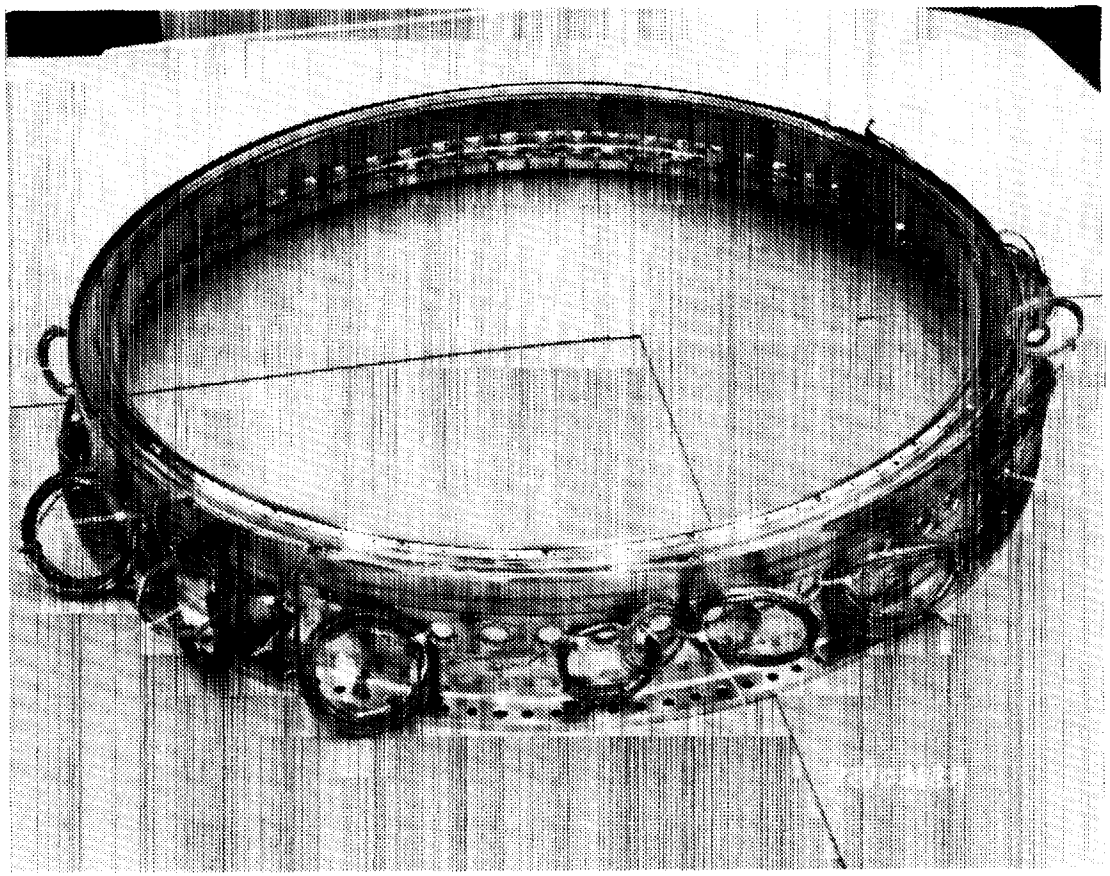
Instrumentation Lead-out on Aft Side of Stage 2 Disk



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

This is a photo of the instrumented stage 2 nozzle support prior to buildup into the instrumented engine. This assembly is only partially built up and does not include the stage 2 vanes and interstage seals.

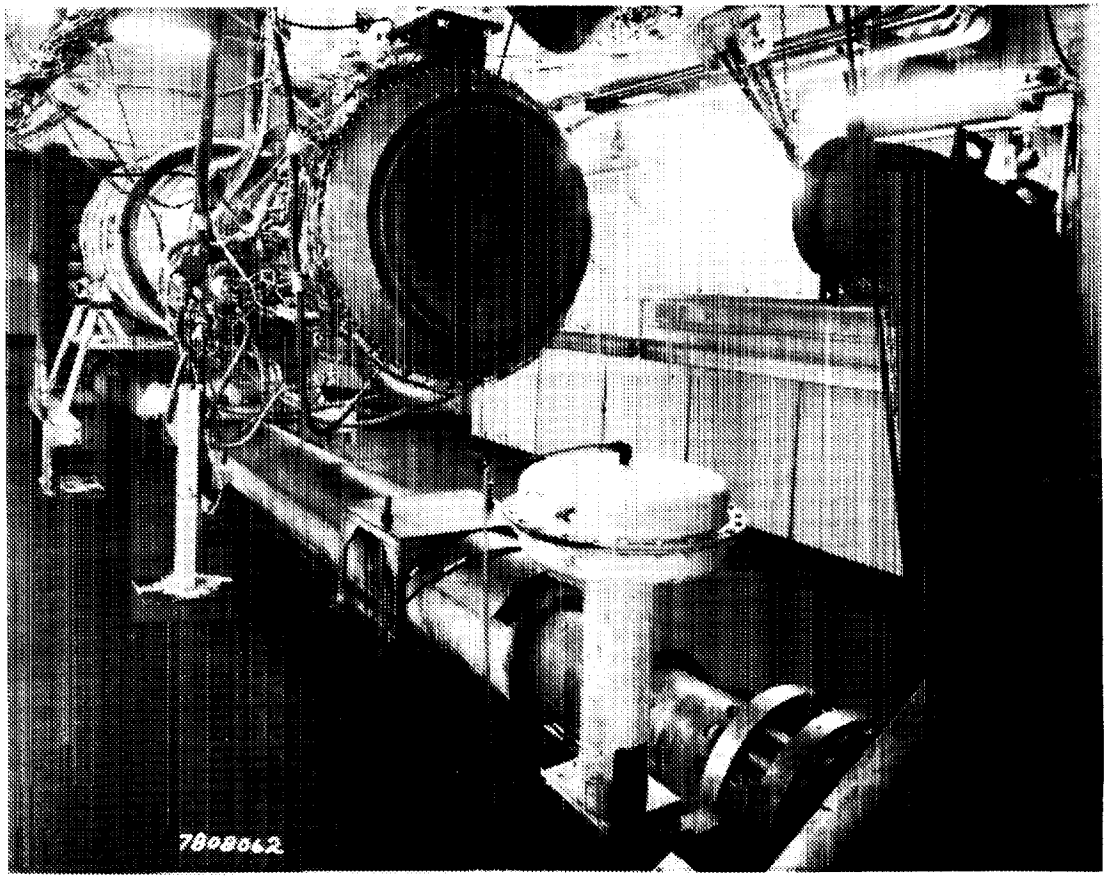
Instrumented Stage 2 Nozzle Support



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

This photo shows engine 441-019/8 installed in test cell 37. The photo shows the engine exhaust nozzle, the hot gas re-circulation scoop and distribution piping as well as the butterfly valve and actuator for the hot gas recirculation piping.

TF39 Core Engine 441-019/8 in Test Cell #37



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

This chart indicates a number of data points obtained for both steady state and transient conditions during instrumented engine testing. A total of ten million bits of data were collected.

Single Shank Turbine Program

019 Instrumented Test

Data Obtained

- 205	Steady State Data Points
- 300	Transient Data Runs
- 10 Million	Total Bits Of Data

Both steady state and transient conditions were explored during instrumented engine testing as shown on this chart. The engine was run to 107% speed and to max temperature conditions during the steady state operation. The transient operations were conducted to simulate a wide variety of engine operating conditions.

Single Shank Turbine Program

019 Instrumented Test

- **Conditions Explored - Steady State**
  - Blade Vibration to 107% N
  - Rotor Vibratory Stress to 103% N
  - Temp Data To LM2500 & CF6-6 Max Power
  - Thrust Balance
  - Cooling Flow
  - Cavity Pressures

  } Throughout The Operating Range
- **Transient Conditions - Temperature Data Obtained**
  - Cold Rotor Starts
  - Bursts, Chops To CF6 & LM2500 Max Power
  - Cycles Planned For 019 Endurance
  - Shutdown From Ground And Flight Idle
  - Shutdown From LM2500 Max Power
  - Restart After LM2500 Shutdown
  - Hot Rotor Reburst From CF6-6 Max. Power
  - Step Thermal Cycles For Heat Transfer Correlation

No comment required.

Instrumented Test Preliminary Results

Blade Vibratory Stress Results

- Stage 1 Blade
  - No Observable Resonance Response in Any Mode
  - Measured Levels Very Low
- Stage 2 Blade
  - Low Response
  - First Flex Only
  - 10% of Limits — 10/Rev Peak
  - 20% of Limits at Top Speed (Near 8/Rev)
- Confirmed Low Vibratory Stress Adds Confidence for Achieving Long Life

Blade Metal Temperatures

- **Stage 1 Blade**
  - Bulk Temperature Slightly Lower Than Prediction
  - Local Temperatures Agree with Revised Heat Transfer Model
  - Warm Region Found on Pressure Side Being Analysed
- **Stage 2 Blade**
  - Measured Temperatures are Very Close to Prediction
- **Conclusion**

  Measured Temperatures Confirm That Long Life Can Be Achieved With Reduced Cooling.

Turbine Rotor Temperatures

- **Measured Metal Temperatues**
  - Agree With Prediction
  - Except Front Shaft Measured 60° Warmer
  - May Decide to Add Windage Cover
- **Measured Cavity Temperatures**
  - Confirms That Reduced Purge Flow is Satisfactory
  - May Need to Bypass Some Interstage Flow

Clearance improvements which were demonstrated during the instrumented engine testing indicated that the redesign turbine had no rubs on accel or decel, that the warm rotor reburst capability had been improved and that the shroud structure was current and more uniform in temperature. These observations from tests combined with other design features not directly measureable, lead one to conclude that the clearance improvement for the redesign turbine appear to equal or exceed those quoted at the beginning of the program.

Summary Of Clearance Improvements

From Engine Test

- No Rubs Accel/Decel - Same As Current -6
- Warm Rotor Reburst Capability Improved
- Shroud Hooks Cooler And More Uniform
  - Less Thermal Distortion

Other Improvements - Not Directly Measureable

- Reduced Elastic Deflection At Cruise
- Increased Resistance To TMF Distortions

Conclusion - Clearance Improvements Appear To Equal Or Exceed Quoted Increments

Stage 2 nozzle flows. Stage 2 nozzle flows were measured to be approximately .4% higher than design intent. This high flow will be reduced by means of improved sealing on stage 2 vane inserts. This feature has been incorporated on parts for both performance and endurance testing.

Cooling Flows

- **Rotor Flow**
  - Instrumented Build Flowed High by 1% W_{2C} Due to Leaks Thru Instrumentation Leadout Holes
  - Endurance Test Rotor With No Instrumentation Shows Flow to be 0.2% W_{2C} Higher Than Intent
  - Aft Shaft Rabbit Being Incorporated for Block II
- **Stage Two Nozzle Flow**
  - Flow is Approximately 0.4% W_{2C} High
  - Improved Sealing at Inserts Incorporated for Performance and Endurance Testing

Overall Results Of Instrumented Testing

- Obtained All Required Data
- Most Design Goals Were Achieved or Exceeded
- Problem Areas Not Yet Resolved
  - Rotor Flow Leakage
  - Stage 1 Blade Metal Temperature
  - Stage 2 Nozzle Leakage
  - Front Shaft Metal Temperature
  - Rotor Thrust Balance

The objective of the endurance engine test was to run 750 severe endurance cycles. This testing has been completed and 1000 core engine endurance cycles were successfully achieved.

Endurance Engine Test

- **Objective**
  - Run 750 Severe Endurance Cycles
- **Status**
  - Testing Complete — 1000 Core Engine Endurance Cycles Achieved

During 1000 cycles of testing, the engine obtained 5000 minutes at the redline conditions shown on this chart. This is roughly equivalent to 3500 hot day takeoffs and more than 350 hour certification tests.

Cyclic Endurance Test

1000 Cycles Completed
5000 Minutes at Redline*

$T_{4.1}$	2550° F
T_3	1010° F
EGT	1675° F
N_2	9900 RPM

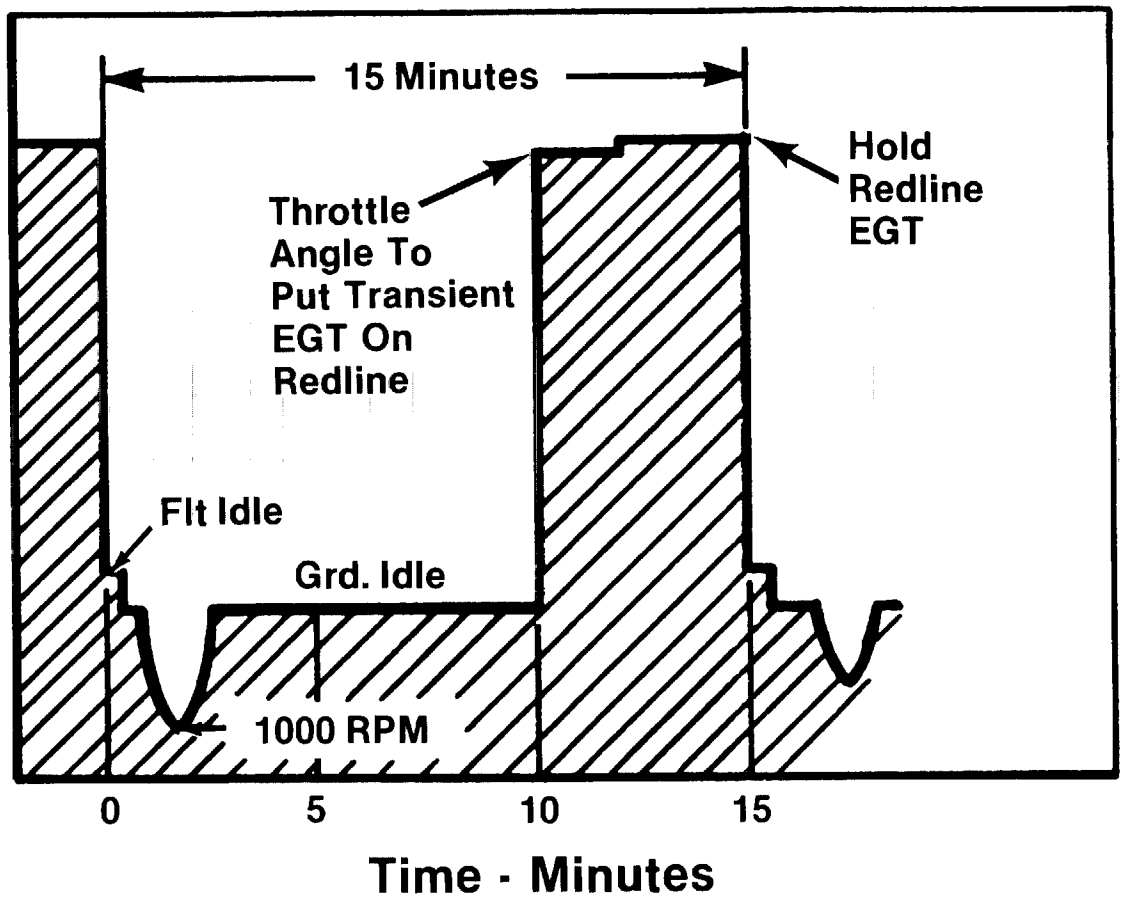
*Hot Time Equivalent To:

- 3500 Hot Day Takeoff's
- 3 + (150 Hour) Certification Tests

This chart depicts a typical cycle of 15 minutes duration to which the turbine was subjected during 1000 cycles of endurance test. Note that the throttle angle is varied to hold the engine on the redline EGT during the 5 minutes at redline conditions. This compensates for the overshoot and soak-back normally seen on an accel.

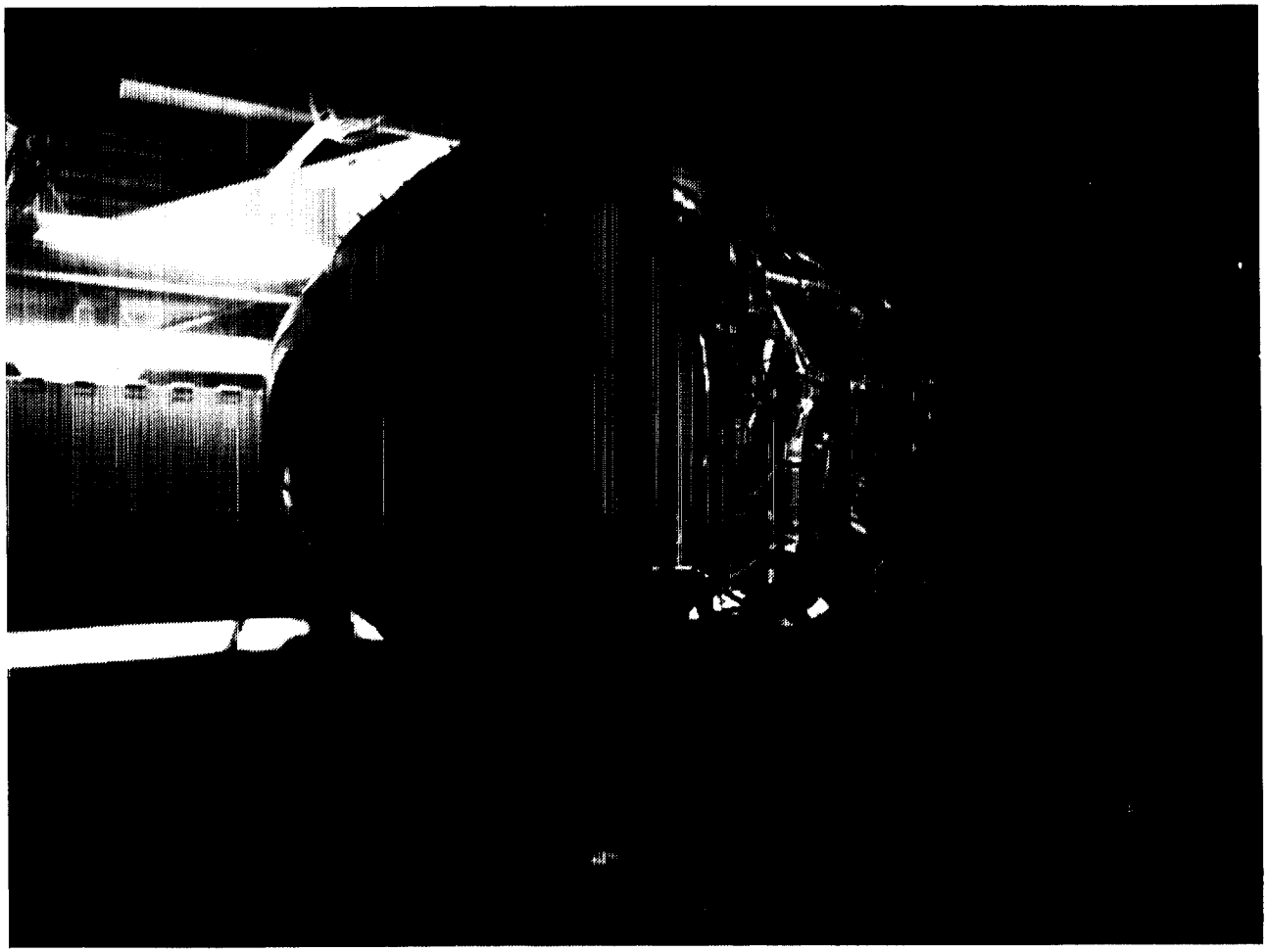
Core engine endurance cycle. All transients are maximum rate. Takeoff power is held for 5 minutes with periodic adjustments made to insure redline conditions as engine "soaks in". Flight idle, shut-down, ground idle legs are used to cool rotor structure and to increase stress range by reducing minimum centrifugal stress.

Core Engine Test Cycle



This photo shows the engine at redline conditions during cyclic endurance testing.

Hot Running Engine



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

1000 cycles which were achieved during the endurance testing are equivalent to between 3000 to 9000 hours of Airline operation. This is based on the amount of distress seen on the stage 1 shroud during the 1000 cycles which looks like greater than 3000 flight hours on a typical airline engine and on the calculated stage 2 rupture life of 9600 hours on a mission mix basis.

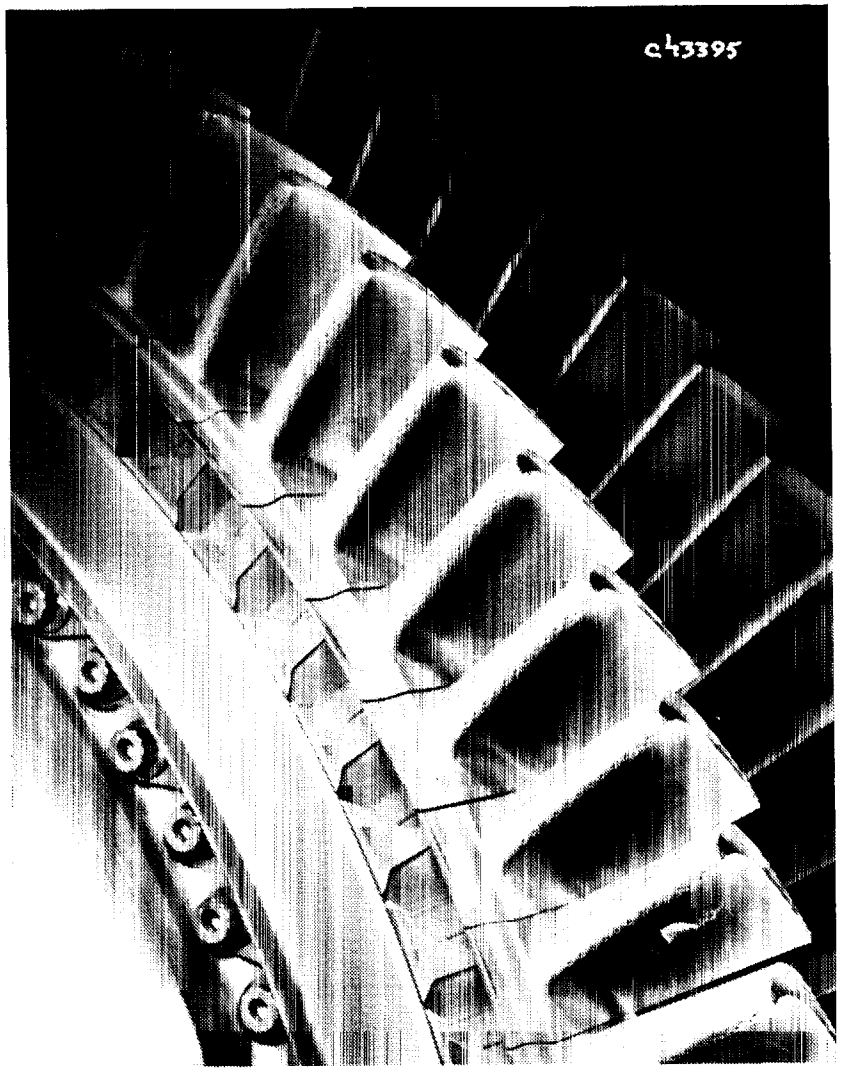
Cycle Severity

- **Severity of 1000 Cycles**
  - Equivalent to 3000 - 9000 Hour Airline Operation
- **Stage 1 Shroud Distress —**
  - Looks Like 3000 + Flight Hours
- **Stage 2 Blade Rupture Life**
  - 9600 Hours Mission Mix With Full Rated 10% of Takeoff's
    9% Derate 90% of Takeoff's

Turbine Blades

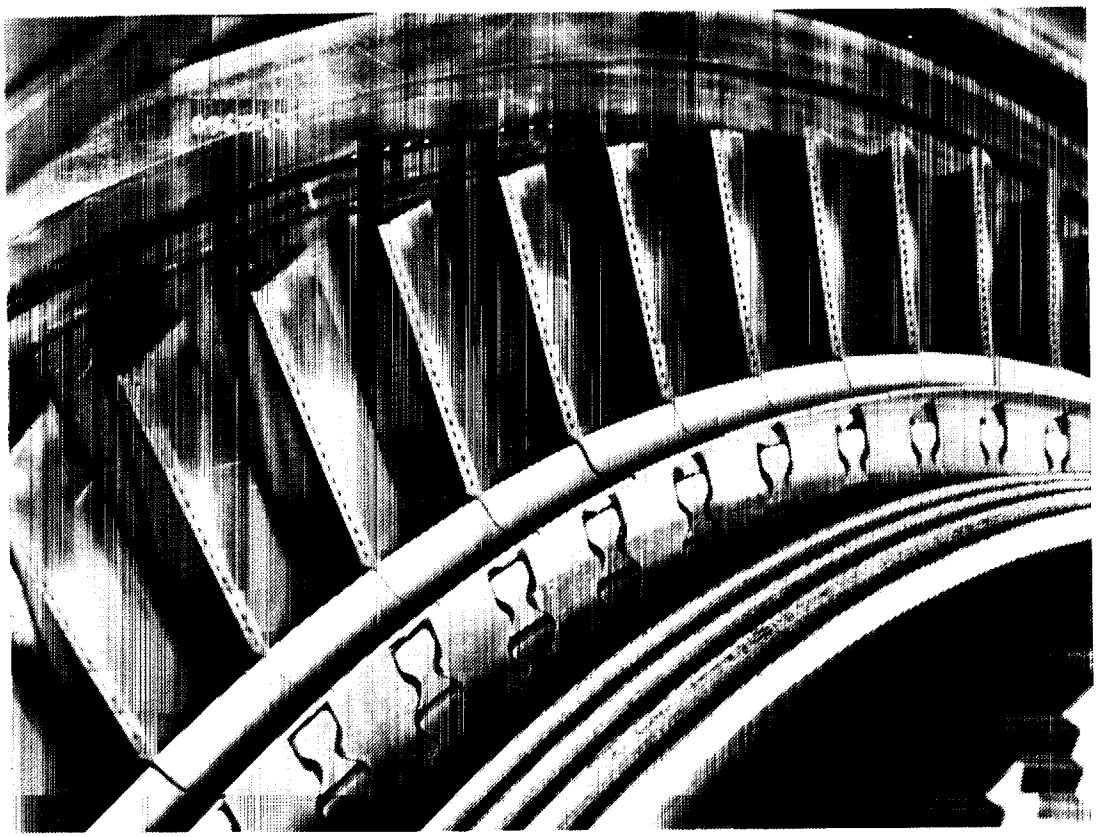
- Stage 1
  - No Trailing Edge Cracks
  - No Squealer Tip Cracks
  - Only Distress is Crack Initiation in Suction Gill Holes
  - All Parts are Serviceable — Will be Retested
- Stage 2
  - No Distress!

Stage 1 Blade Condition After 1000 Cycles



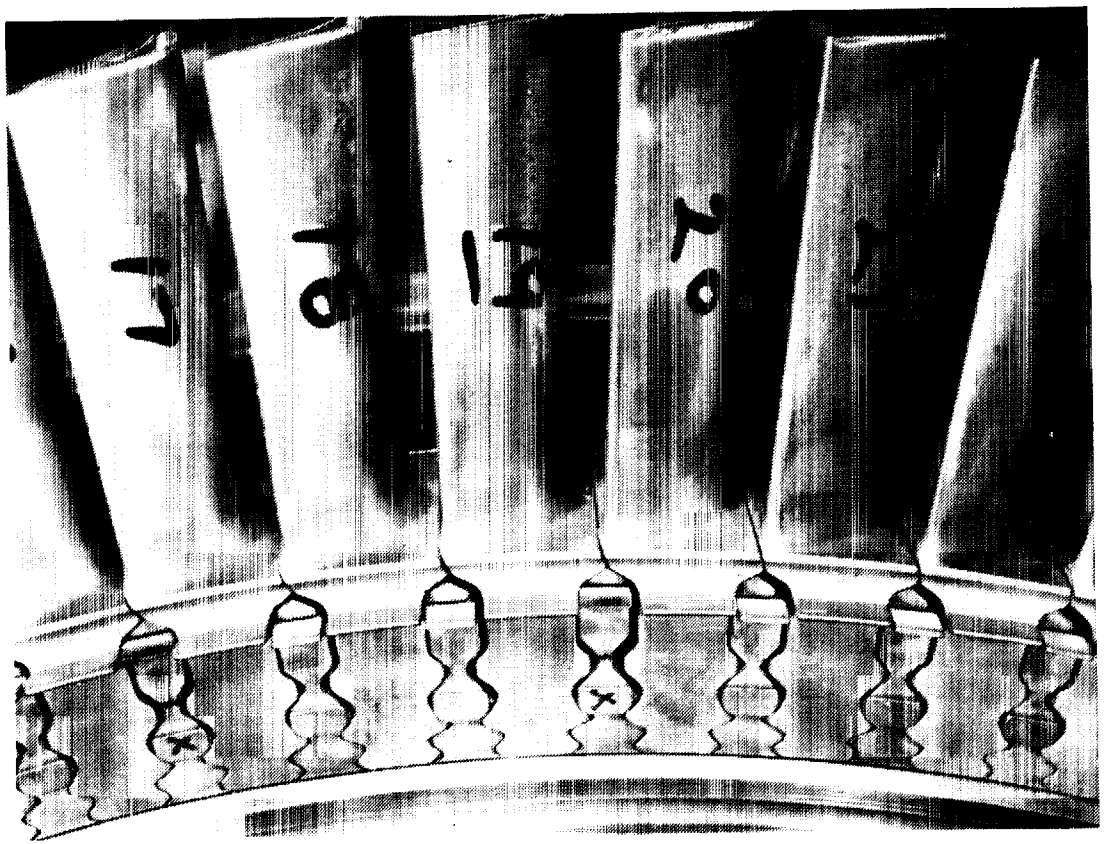
ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Stage 1 Blade Condition After 1000 Cycles



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

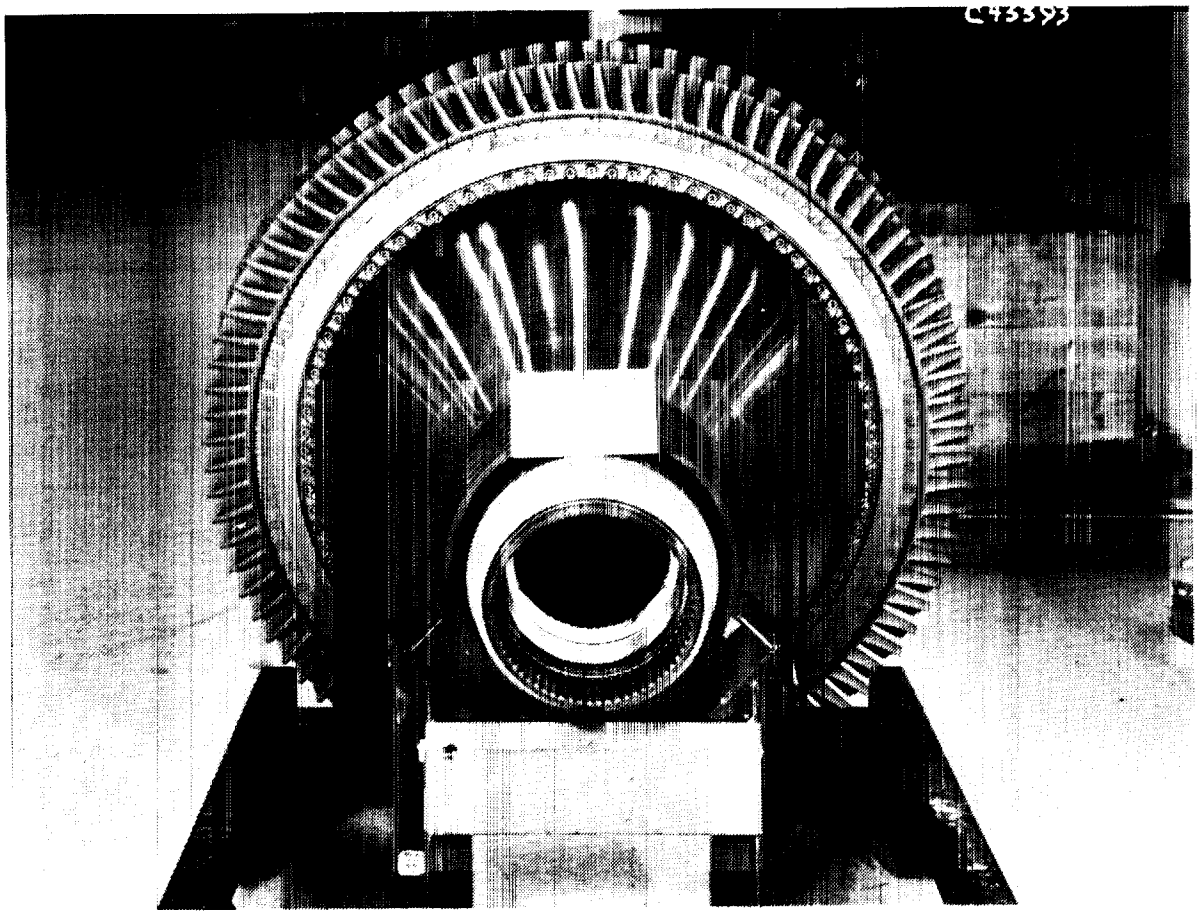
Stage 2 Blade Condition After 1000 Cycles



Rotor

- All Rotor Parts Except Stage 2 Seal Dampers are Serviceable and Will be Retested.

Turbine Rotor After 1000 Cycles



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

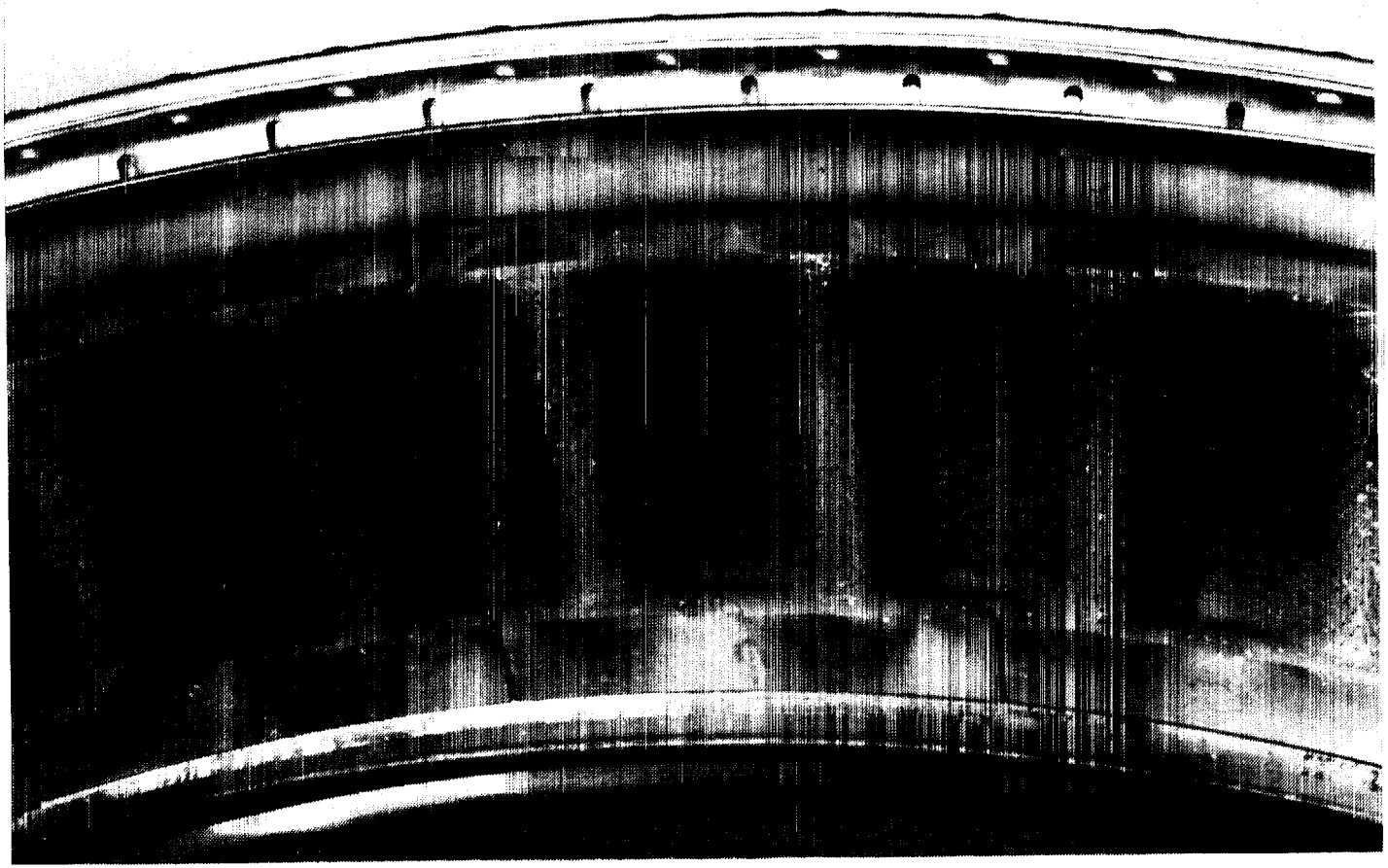
The condition of the stage 1 vanes follows 1000 hours of endurance testing and in indicated on this chart. The most significant distress seen is the pressure side cracking, the probable cause for which is base pressure side of the vane is running hotter than design intent. Rework inserts to avoid impingement flow additional pressure side film holes have been incorporated on additional test parts. Testing of the cracked parts will continue.

Stage 1 Vanes

- No LE Cracks
- No Ballooning or Distortion
- Trailing Edge Minor Distress Only
- Some Pressure Side Cracking
  - Probable Cause - Hot Pressure Side
- Plan
  - Rework Insert to Adjust Impingement
  - Add Pressure Side Film Row
  - Run Samples on −019 & −422
  - Continue Testing Cracked Parts

This photo shows a typical pressure side crack on a stage 1 vane. These vanes show substantially more accumulation of dirt and buildup on the vane surface than is normally seen in airline parts with comparable hot time exposure. This is thought to be caused by sea level operation in a test cell environment.

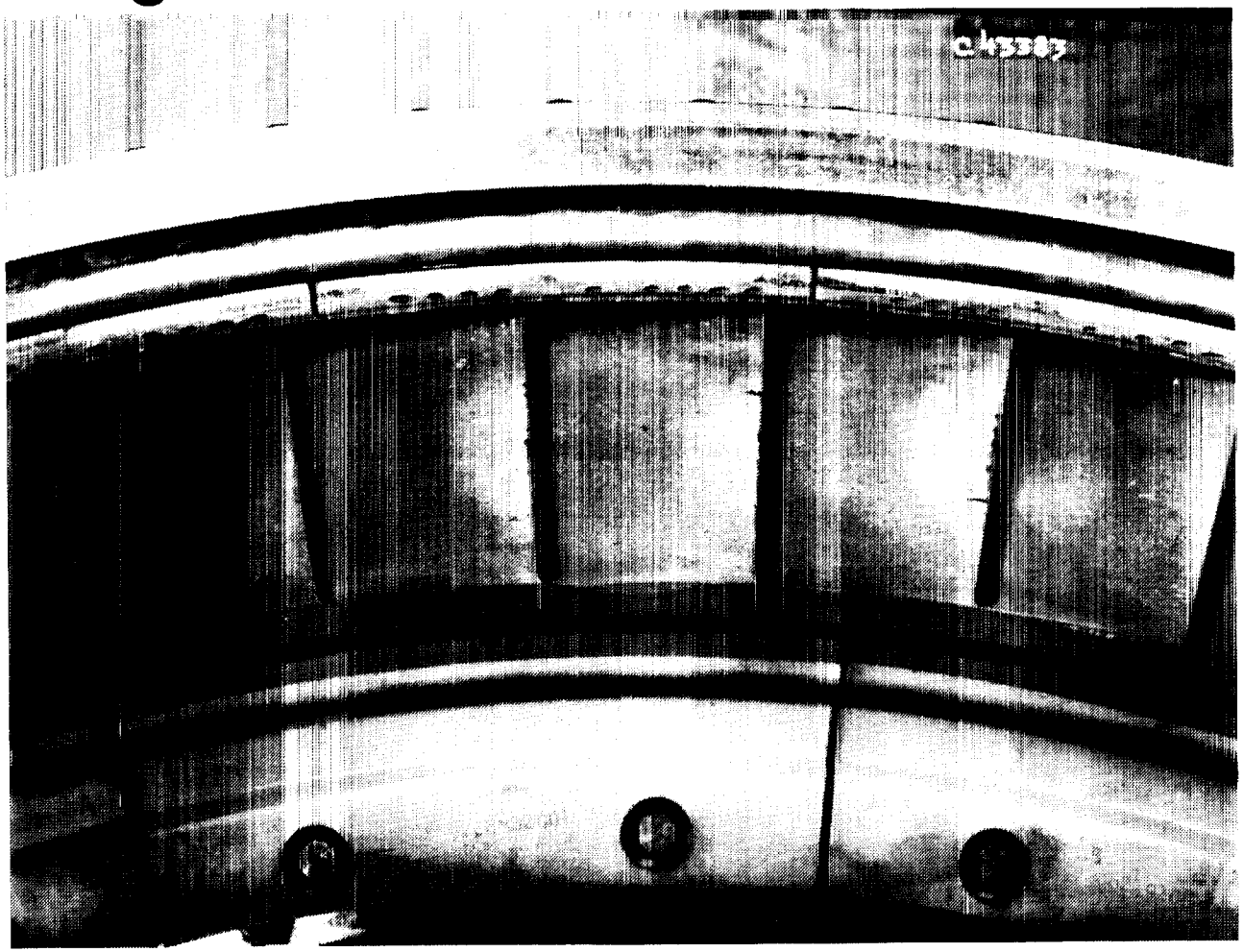
Stage 1 Vanes After 1000 Cycles



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

This photo indicates the degree of distress on the vane trailing edges. Note that only minor cracking is evident.

Stage 1 Vanes After 1000 Cycles



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

This chart summarizes distress seen on the stage 2 vane. No leading edge cracking could be detected indicating that the redesign vane successfully avoids the current field problem.

Stage 2 Vane

- No LE Cracking - Avoids Current Field Problem
- Some Suction Side Cracking
- Will Make Cooling or Mechanical Adjustments

This photo shows the typical suction side crack on the stage 2 vane. Note the absence of leading edge cracks.

Stage 2 Vanes After 1000 Cycles

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

Stage 1 shrouds showed evidence of distress during the 1000 cycle endurance test which was primarily a loss of Bradelloy filler material over a small portion of the total shroud surface area. These shrouds had not been redesigned as part of the single shank turbine design effort. However, improvements in the shroud are now undergoing service evaluation including increased cooling and improved sealing against cooling air contamination. Additional improvements are planned which include a film-cooled shroud design. Shrouds of this design are currently being procurred for engine test.

Stage 1 Shrouds

- Now Service Evaluating
  - Increased Cooling
  - Improved Screen Seal
- Single Shank Needs Additional Improvements
  - Film Cooled Design
  - Same Air Flow as Above
  - Start Test July 27

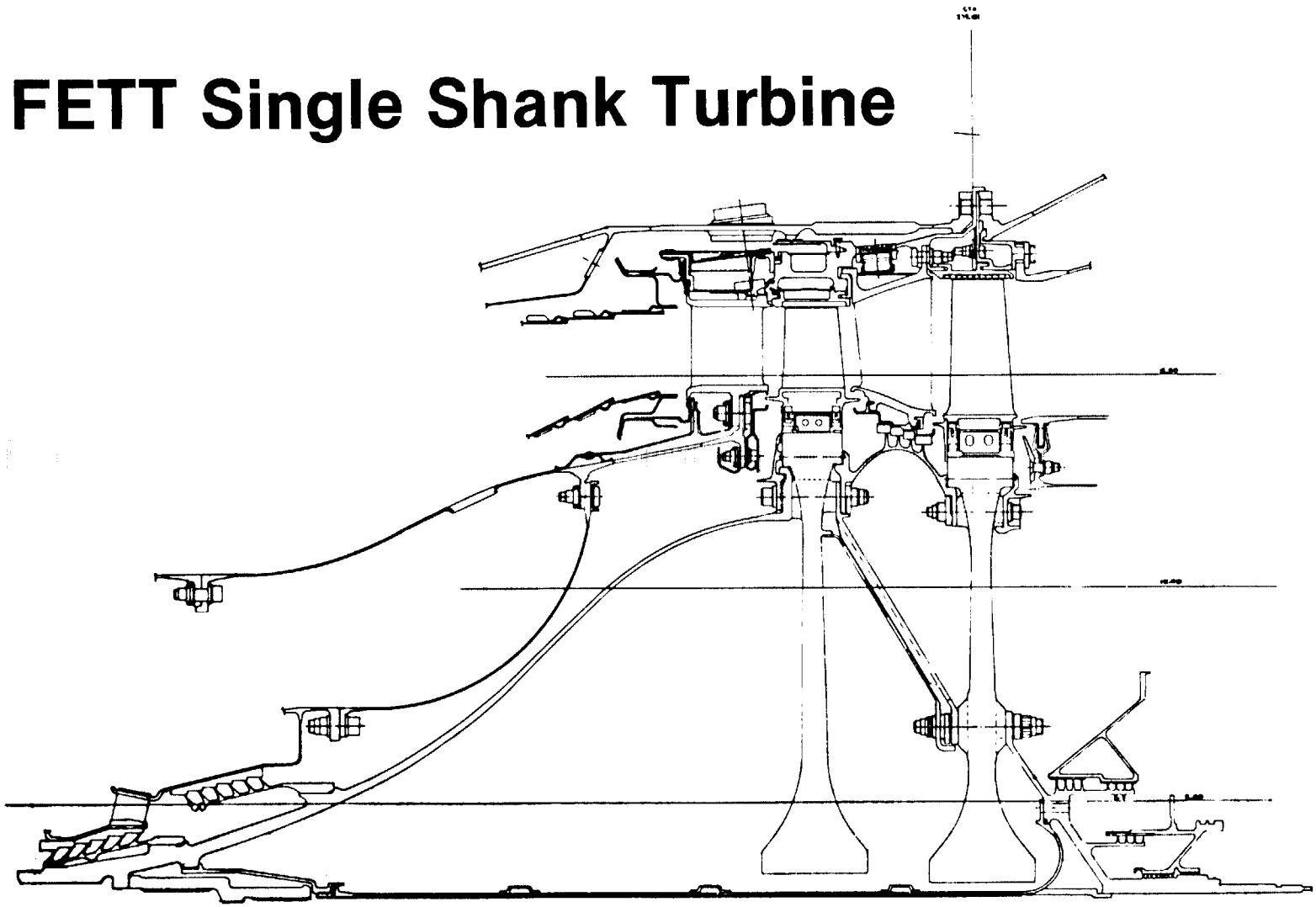
Fan Engine Performance Test

- **Objective**
  - Back-to-Back Testing of Current Production and New High Pressure Turbine Hardware for Performance Evaluation
- **Status**
  - New Production Engine 451-539 on Test
  - Test Completion - 7/79

Single Shank Turbine Development Plan

CF6-6 Fan Engine Performance Test	July '79
CF6-6 Fan Engine Endurance Test, 1000 Cycles	Aug. '79
CF6-32 First Engine Test with Single Shank Turbine	Apr. '80
Airline Service Evaluation	1st Qtr. '80
Production Introduction CF6-6	2nd Qtr. '81

FETT Single Shank Turbine



Summary

HPT Aero

- Program Close to Completion
- Design Successfully Demonstrated
- Estimated Performance Improvement Potential at Cruise -
  - 1.3% Δ SFC (– 1.6% at 3000 Hr.)
- Strong Airline Interest
  - Planned for CF6-6 Upgrade

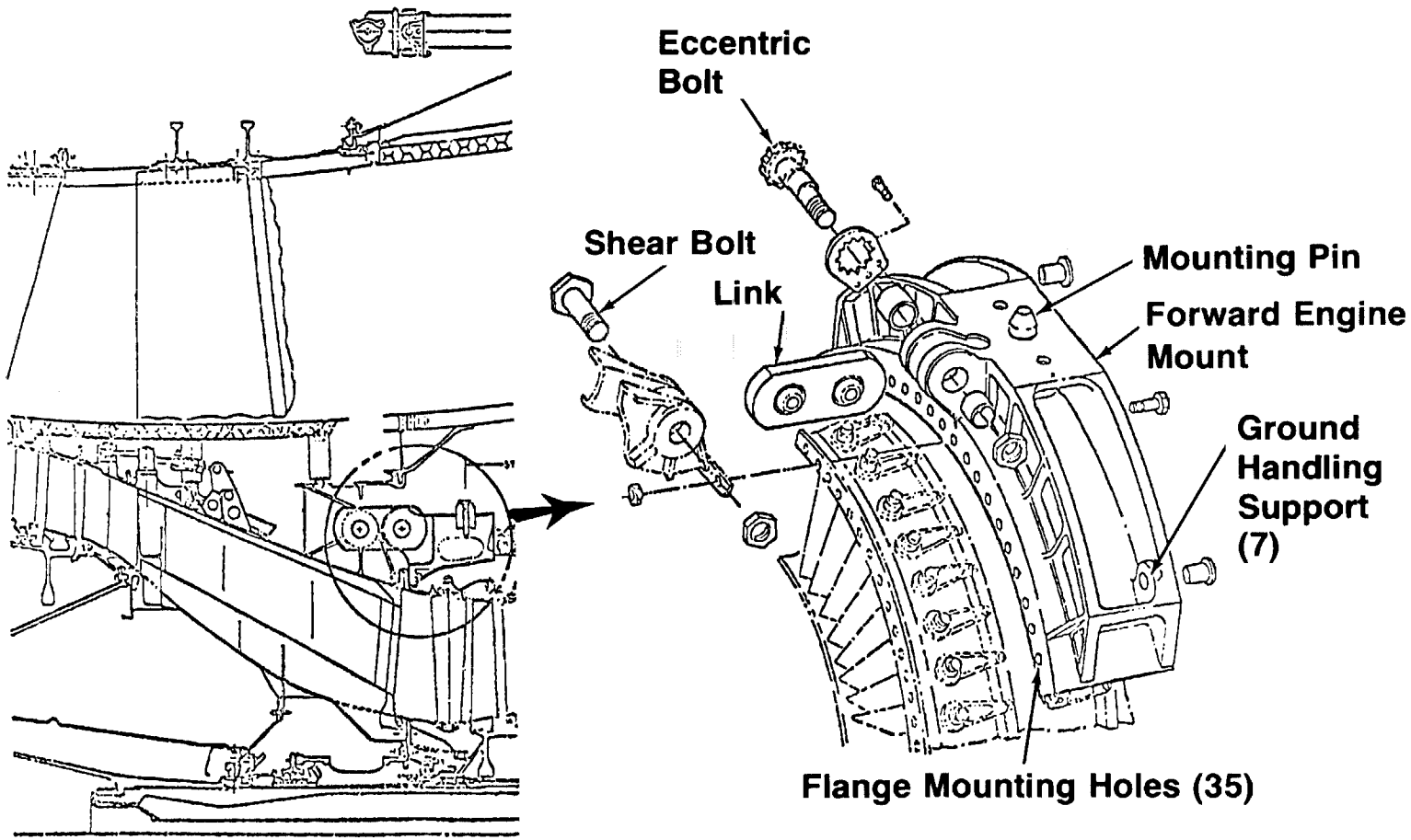
CF6 Jet Engine Performance
Improvement Program

Task 2.4
Front Mount Performance Improvement Concept

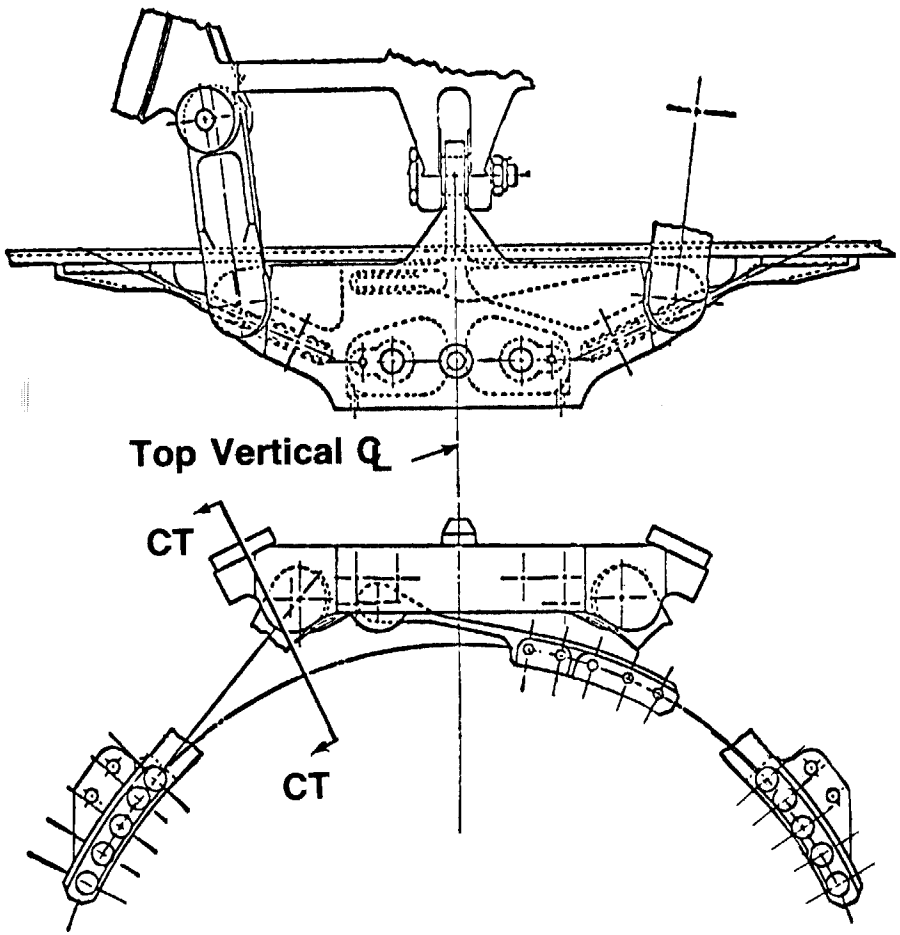
Description of Concept

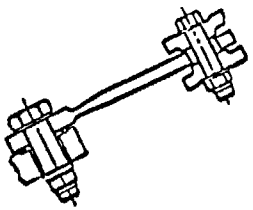
- Two-Point Thrust System Was Developed
  - — Engine Thrust Reacted at $\pm 30^0$ from Top Vertical
  - — Vertical and Side Forces
  - — Series of Tangential Links
- Must Be Failsafe
- Minimizes the Impact on the Aircraft Pylon
- Reworkable with Present Hardware
- 'Double Safety' Locking for All Attachments

Present CF6 Forward Mount System



Revised Forward Mount System



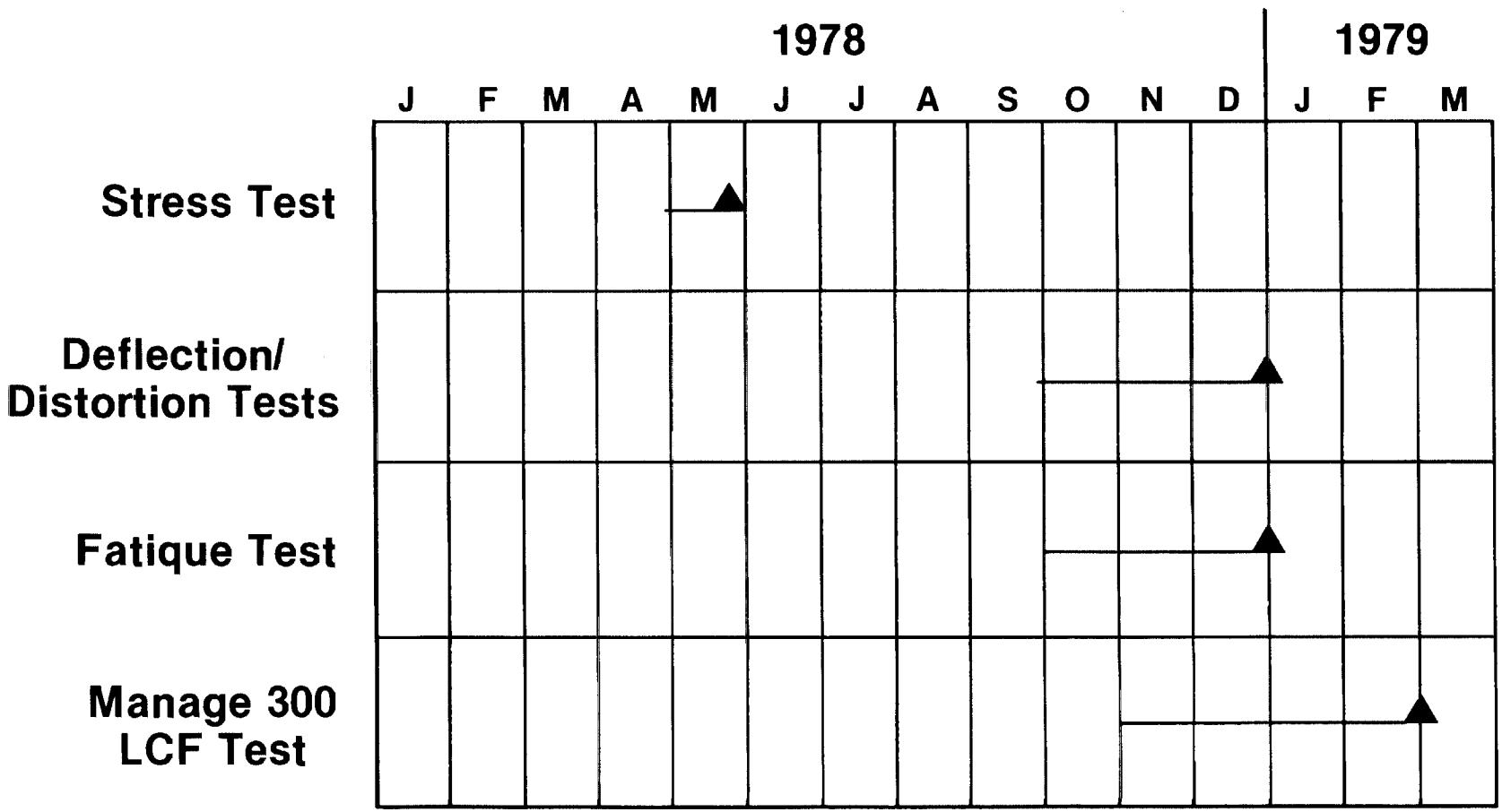


Section CT-CT
2 Places

Estimated Performance Improvement

- Reduction in Compressor Casing Deflections Translated into Performance Improvement
  - ΔSFC (Cruise) — 0.1%
  - ΔEGT (Take-off) — 1.5°C
  - ΔStall Margin (Take-off Flow) + 16%
  - ΔStator Angle Margin (Take-off) + 10%
- Improved Performance Retention

Program Plan–Improved Front Mount



GENERAL ELECTRIC COMPANY
AIRCRAFT ENGINE GROUP

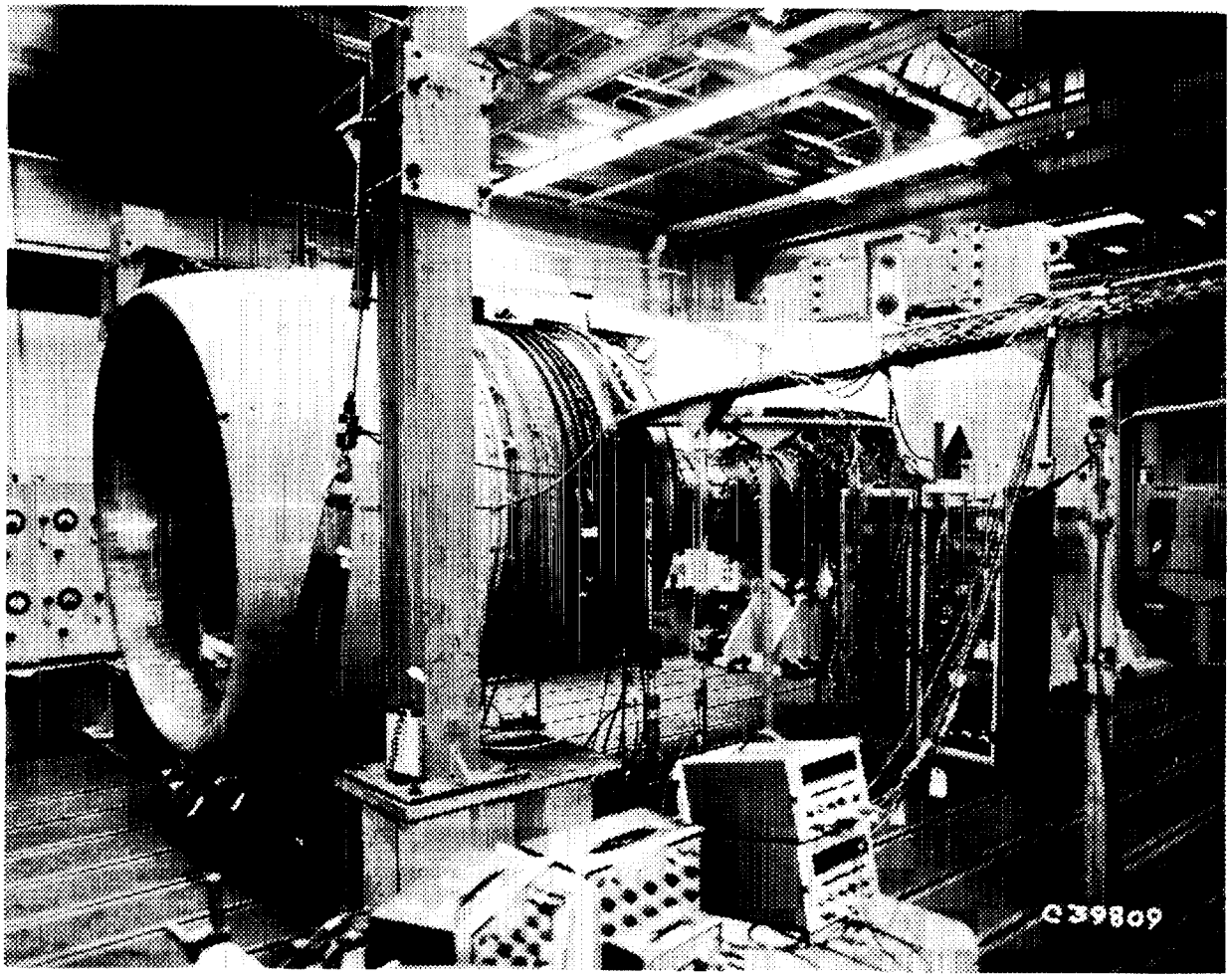
Stress Test (Stress Cote)

- **Objective**
  - Locate High Stress Locations
  - Confirm Calculated Stress Levels
- **Status**
  - Test Completed. Reported in Last Oral
- **Results**
  - Calculated Stress Levels Confirmed
  - Local High Stress Regions Located
  - Additional Strain Gage Locations Identified

Deflection/Distortion Tests

- Three Test Programs to Measure Stress & Deflection on Complete Engine Shell Structure on Wing Pylon.
  1 — Casing Radial Deflection & Mount Stress — Take-off Power
  2 — Failsafe Test — Simulated Partial Failure of Mount
  3 — Installation Stresses & Thermal Growth Stress Correlation

Component static load tests were performed on the engine structural shell assembly supported by an aircraft pylon to determine the effects of mount loadings on the compressor, turbine casings.



ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

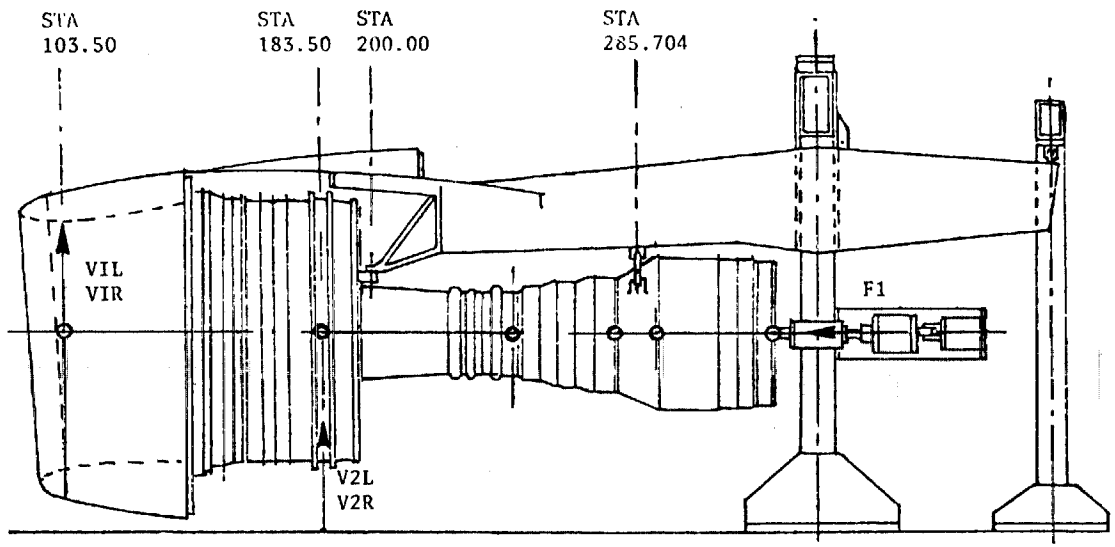
Deflection/Distortion Test.

Deflection/Distortion Test #1

- **Objective**
  - Back to Back Tests of Original & New Front Mount
  - Demonstrate Improvement in Casing Radial Deflection
  - Measure Stress & Deflection in New Mount
  - Maximum Static Thrust & Take-off at Rotation Conditions
- **Status**
  - Test Completed
- **Results**
  - New Mount Appreciably Reduces Casing Radial Deflection (– 29%)
  - Calculated & Measured Stresses Agree Well

The test loading for the deflection/distortion tests were defined such that the tare weight of the test set-up gave the correct 1'G' down reaction at the front mount and very close agreemont with the compressor casing bending moment.

Test Loads for Deflection/Distortion Tests

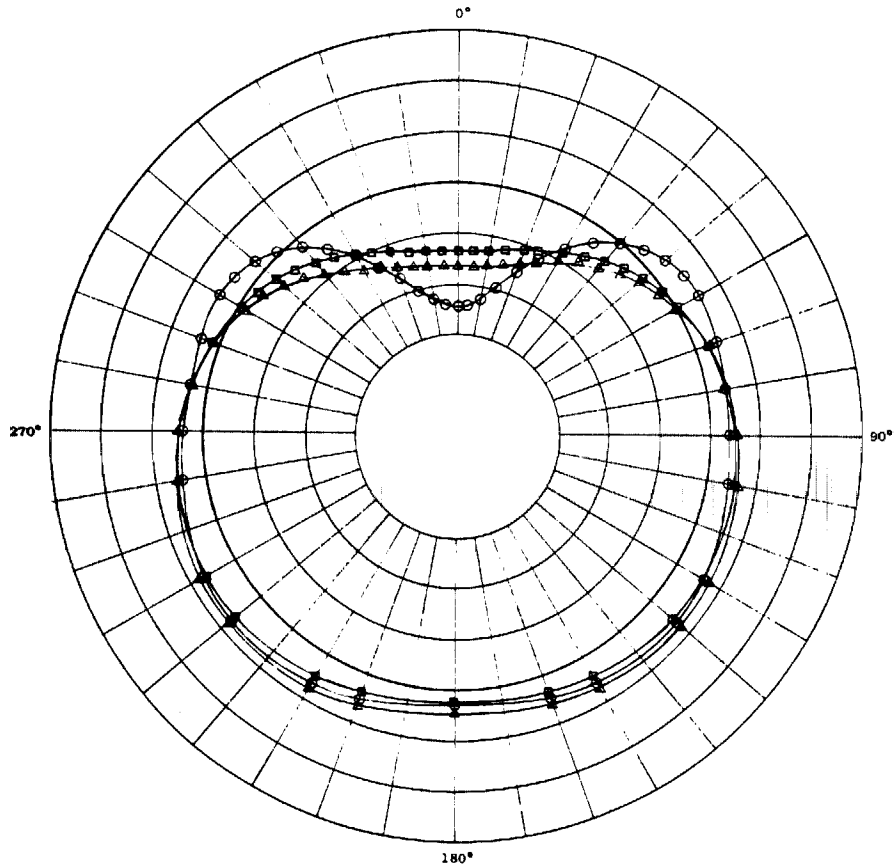


TEST CONDITION	TEST LOADS N - (LBS)				
	F_1	VIL	VIR	V2L	V2R
Maximum static thrust	244650 (55000)	0	0	-9150 (-2058)	-9150 (-2058)
Take-off at rotation	184601 (41500)	19570 (4400)	19570 (4400)	9150 (-2058)	9150 (-2058)

A typical plot at a high pressure compressor station will show a cardioid shape due to the forward engine mount reaction.

Compressor Casing Radial Deflection

Stage 3 — Take-off at Rotation Condition



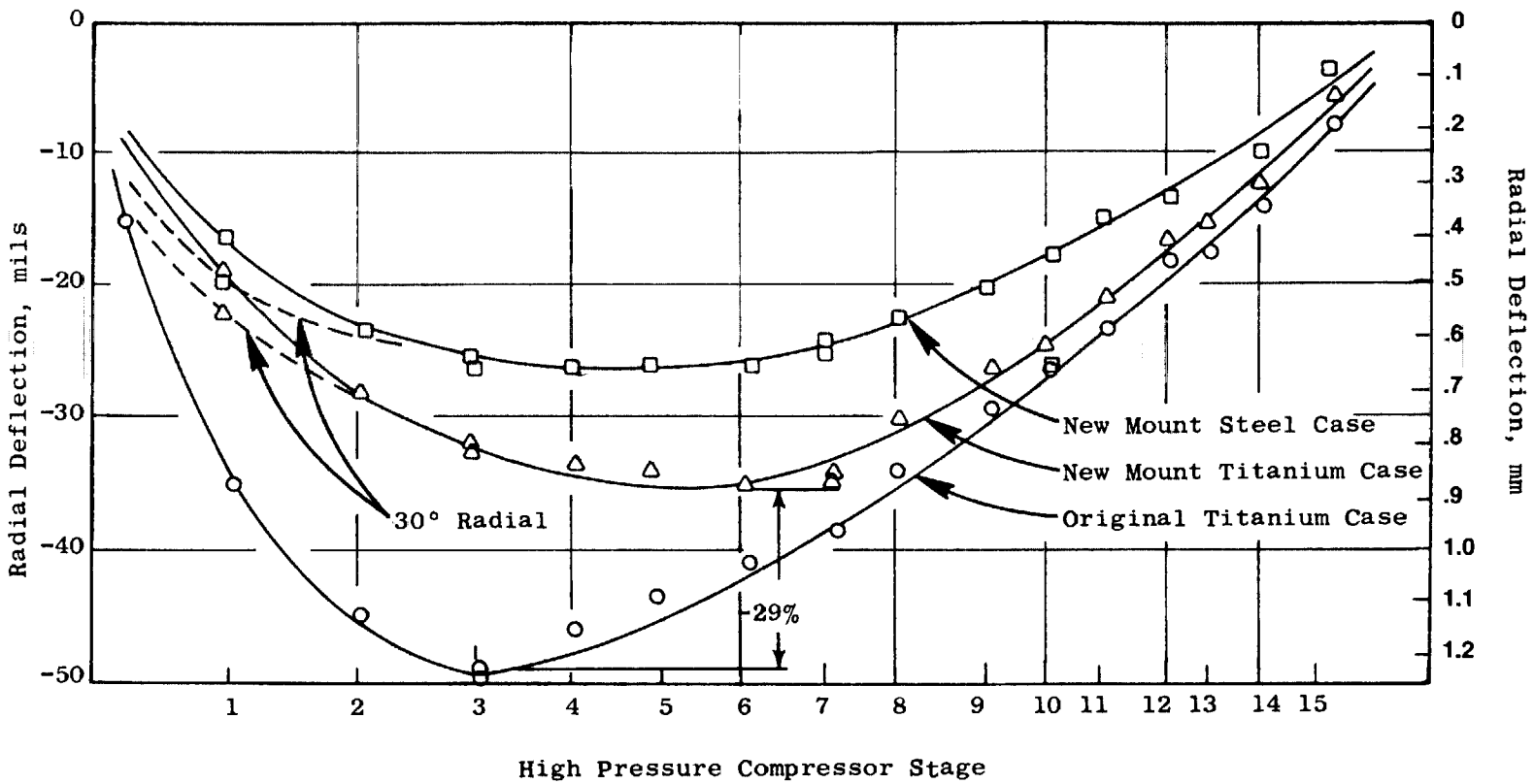
○ Original Mount, Titanium Casing
△ New Mount, Titanium Casing
□ New Mount, Steel Casing

Scale: 0.508 mm (0.020 in.) per Division

The figure shows the compressor case radial deflection at 12 o'clock along the length of the casing.

Compressor Casing Deflections

12 O'clock Radial Position — Take-off at Rotation



Deflection/Distortion Test #2 — Failsafe Loads

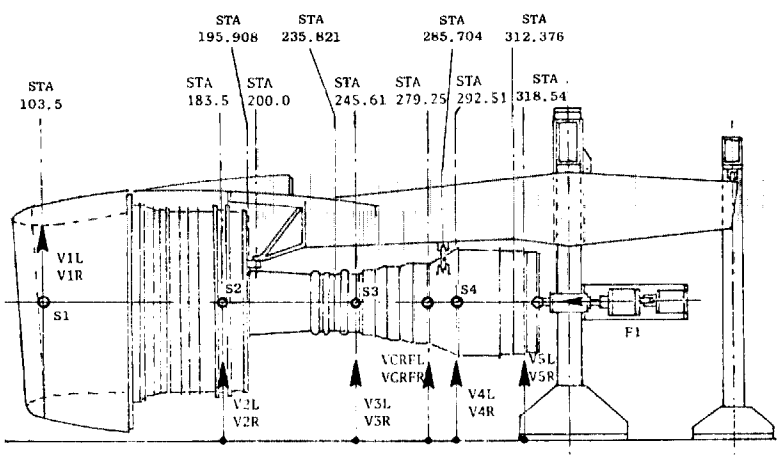
- **Objective:**
  - Test to Simulate Partial Failure of Front Mount
  - Measure Stress/Deflection of Mount & Links
  - Measure Compressor Casing Deflection
  - Loads on Failsafe Tongue
- **Status:**
  - Test Completed

Test Conditions

- **Simulated Failure**
  - Left Side Link Failed
  - Left Side Link & Left Thrust Link Failed
  - Right Thrust Link Failed
  - Center Link Failed (Swaylink)
- **Test Loads**
  - Take Off at Rotation (Operating Normal)
  - 4G Vertical (Ultimate Condition)
  - Max Side Load on Mount (Ultimate Condition)

The three failsafe loading conditions provide the maximum axial, ride, and vertical design loads.

Failsafe Load Conditions

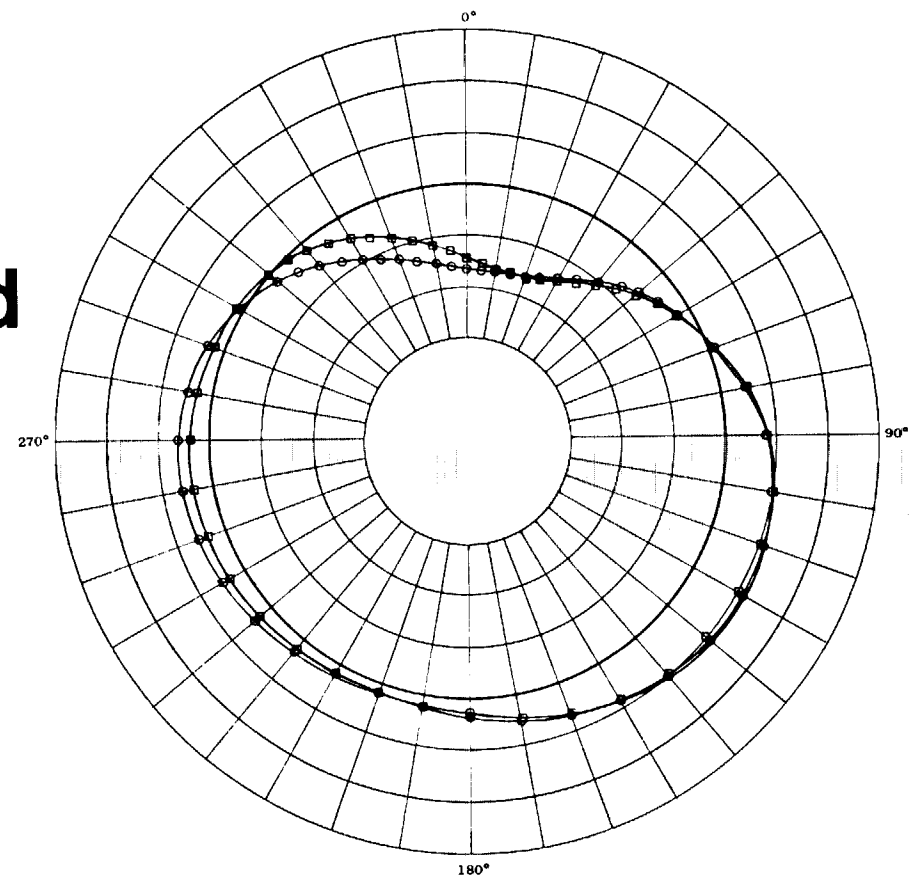


FAILSAFE LOADS, (LB)

TEST LOADS	CASE 1	CASE 2	CASE 3
	T O AT ROTATION	SIDE LOAD COND.	VERTICAL LOAD COND
V1L	4300	-1850	-2400
V1R	4300	-1850	-2400
S1	0	0	0
V2L	6150	-5600	-13400
V2R	-13850	-5600	-13400
S2	1300	8600	0
V3L	13100	-1700	-3600
V3R	-14900	-1700	-3600
S3	0	600	0
VCRFL	14000	0	0
VCRFR	-14000	0	0
V4L	- 950	-1800	-3800
V4R	- 950	-1800	-3800
S4	0	650	0
V5L	-18950	-1800	-3800
V5R	+18950	-1800	-3800
S5	0	650	0
F1	45000	-6500	0

A typical plot at a high compressor station shows a small change in diametrical clearance for a simulated mount failure condition.

Effect of Simulated Side Link Failure on Compressor Casing Distortion



○ Test 3 - New Mount Complete

△ Test 6 - New Mount Left Side Link Failed

Scale: 0.508 mm (20 mils) per Division

Test Results

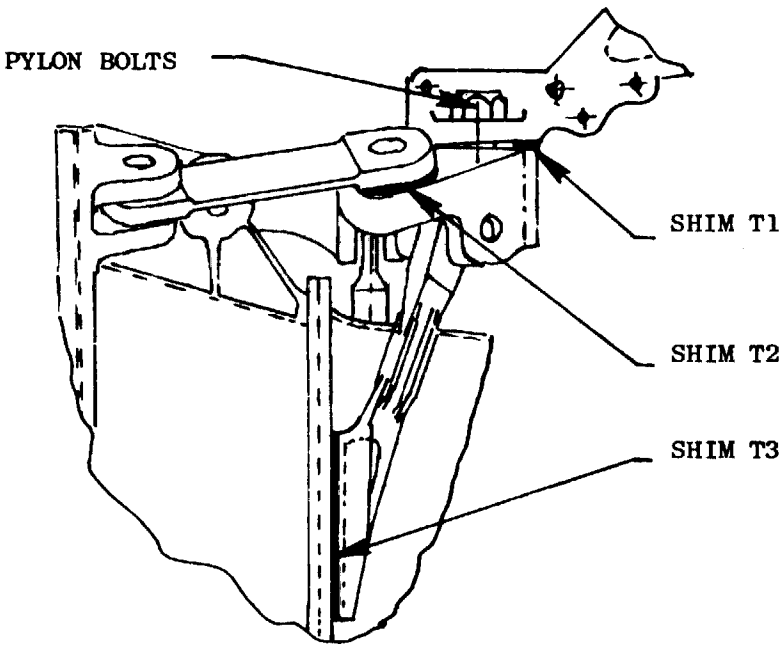
- Partially 'Failed' Mount Performance Adequate
- Additional Radial Deflection is Acceptable (.002"—.004")
- Stress in Mount Components Agree with Calculation
- Low Stresses Permit Repeated Flight Cycles Under the Stipulated 'Failed' Condition

Deflection/Distortion Tests #3 — Thermal & Assembly Stress Correlation

- **Objective:**
  - Confirm Calculated Stress Analysis
  - Thermal Growth of Core Casing
  - Installation Stress with Adverse Dimensional Stack-up
- **Status:**
  - Test Completed

The table shows the shimming sequence in order to measure the stresses induced in the mount system due to the simulated thermal growth and assembly misalignment.

Simulated Thermal and Assembly Stress Test Conditions



	TEST DESCRIPTION	SHIM T1	SHIM T2	SHIM T3	PYLON BOLT TORQUE
19	Thermal correlation Axial displacement	-	-	x	Tight *
20	Thermal correlation Vertical displacement	-	x	-	Tight *
21	Assembly correlation Mount pitch rotation	x	-	-	Tight *
22	Assembly correlation Axial displacement	-	-	x	Loose **
23	Assembly correlation Vertical displacement	-	x	-	Loose **

* Loosen Mating Faces Sufficient to Insert Shim and then Retorque. Shim Sizes 0.127 mm (0.005 in.), 0.254 mm (0.010 in.) and 0.381 mm (0.015 in.)

** Lossen all Bolts, Insert Shims and Retorque All Except Pylon Bolts. Shim Sizes 0.127 mm (0.005 in.), 0.254 mm (0.010 in.) and 0.381 mm (0.015 in.)

Test Results

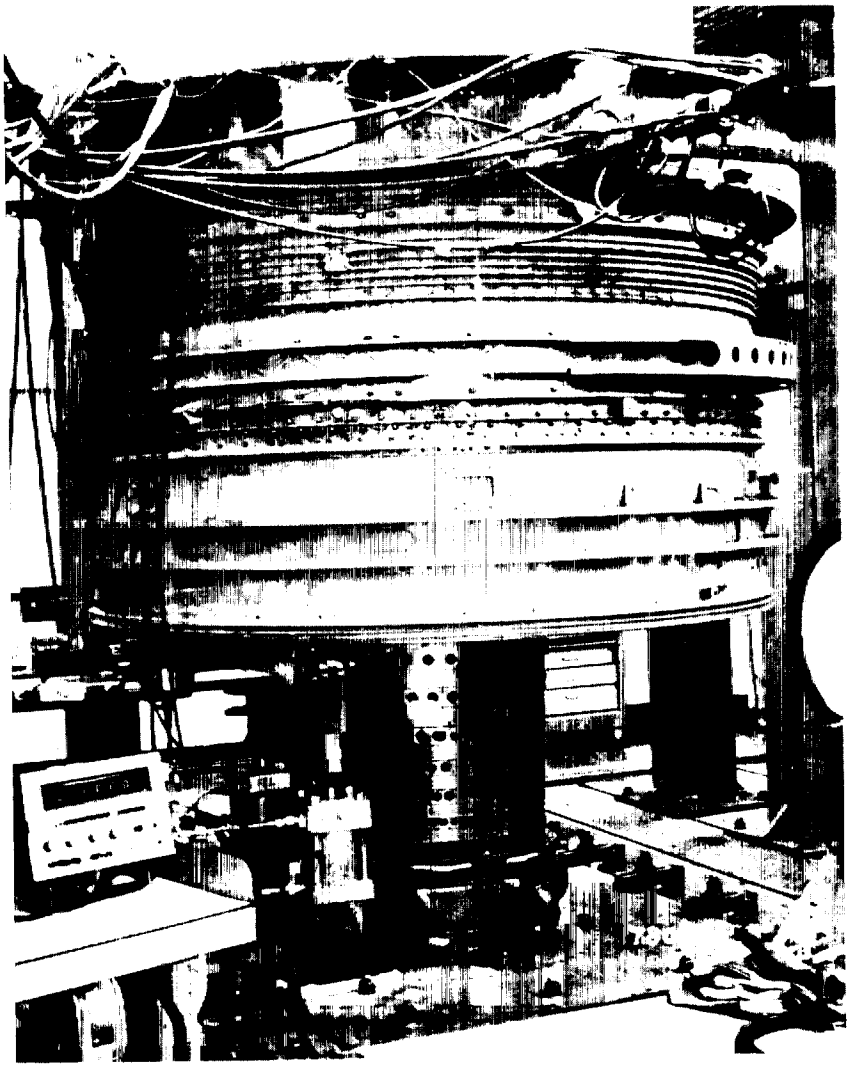
- Good Agreement Between Measured & Calculated Stress Levels
- Confirmed That Max Stresses Are Very Low

Fatigue Test

- **Objective:**
  - Demonstrate Two Lifetimes Cyclic Load Capability
  - One Life = 35,000 Cycles, Including 230 Limit Load Cycles
- **Status:**
  - Test Completed

A vertical set-up of a fan module including the high pressure forward compressor case used for the engine mount fatigue test.

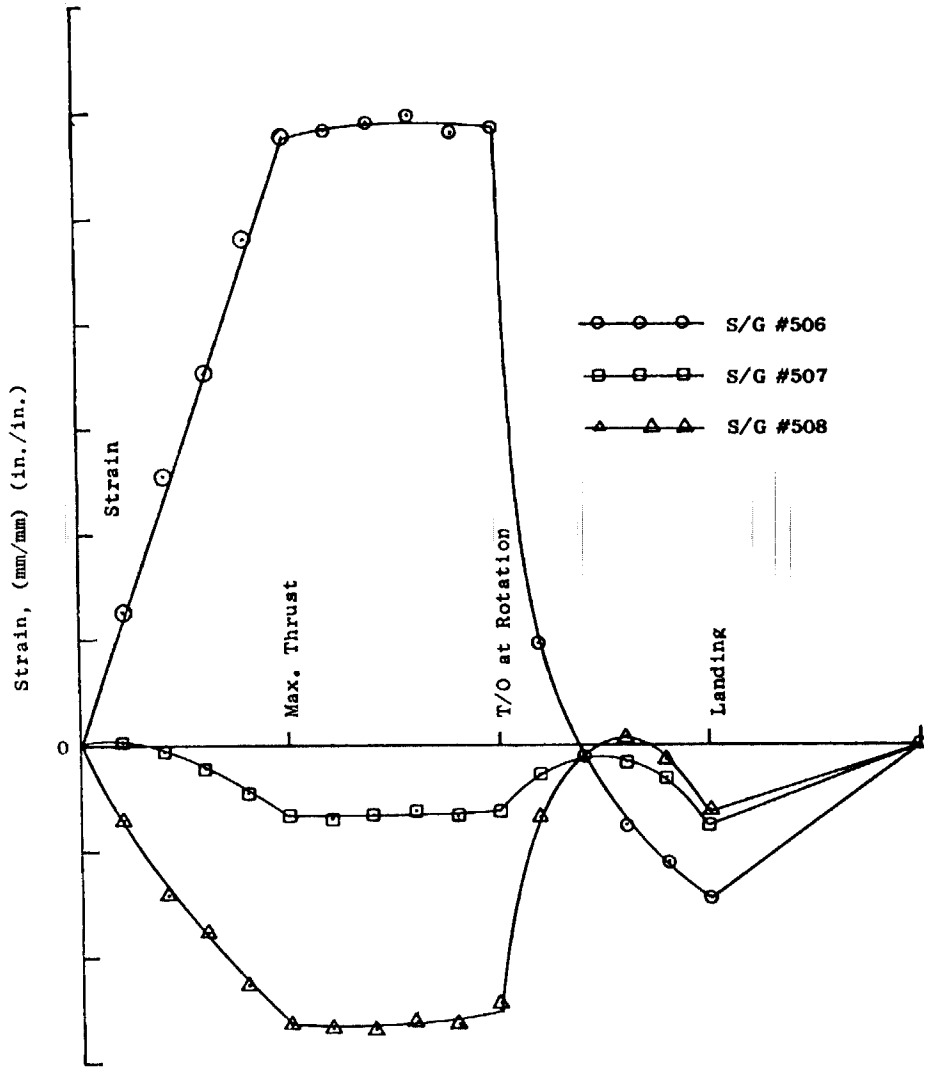
Fatigue Test



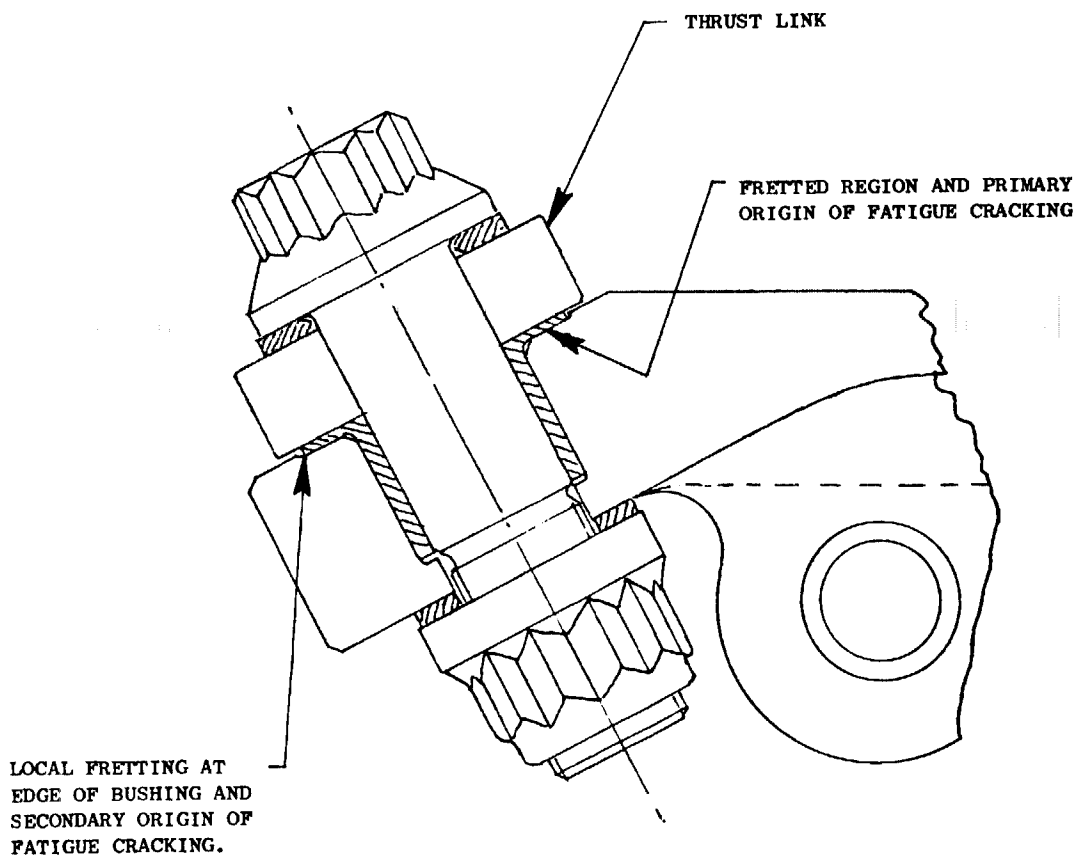
ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

This plot shows the analysis of the results from three strain gage rossettes on the mount platform vs. the load duty cycle.

Strain Variation During Load Cycle



Cross Section of Left Lug & Thrust Link Showing Origin of Fatigue Cracks.



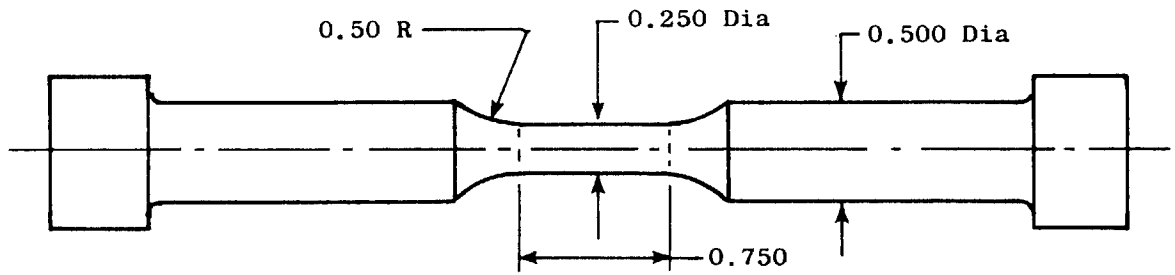
Test Results

- One Lifetime (35,000 Cycles) at Average Load Level Demonstrated
- Second Lifetime at 20% Increased Load to 35% of Second Life Time
- Failure Initiated by Fretting (Stress Level Low at Failure Initiation)

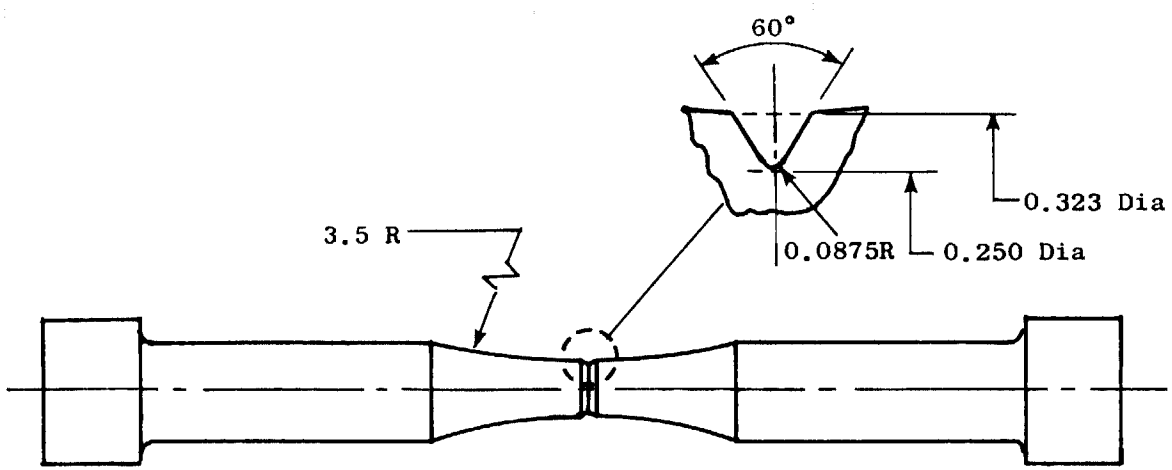
Low Cycle Fatigue Tests — Marage 300 Steel

- **Objective:**
  - Provide High Confidence Low Cycle Fatigue Data
  - 400°F, K_T = 1.0 & 1.5, 'A' Ratio: .6, 1.0, & 2.5
  - Two Heats of Material
- **Status:**
  - Test Completed

Fatigue Test Specimens



Smooth Specimen (K_t = 1.0)



Notched Specimen (K_t = 1.5)

Test Results

- 56 Test Coupons Tested at 20 CPM Cyclic Frequency
- Original Data Used for Design Were 5% Conservative

Front Mount

- Program Completed
- Performance Improvement at Cruise $-0.1\% \Delta$SFC with Titanium Casing
- New Mount Certified and in Production
- Benefits Reduced for Steel Casing Compressors

CF6 Jet Engine Performance
Improvement Program

Task 2.5
HP Turbine Roundness Control Performance Improvement Concept

Factors Affecting Turbine Clearances

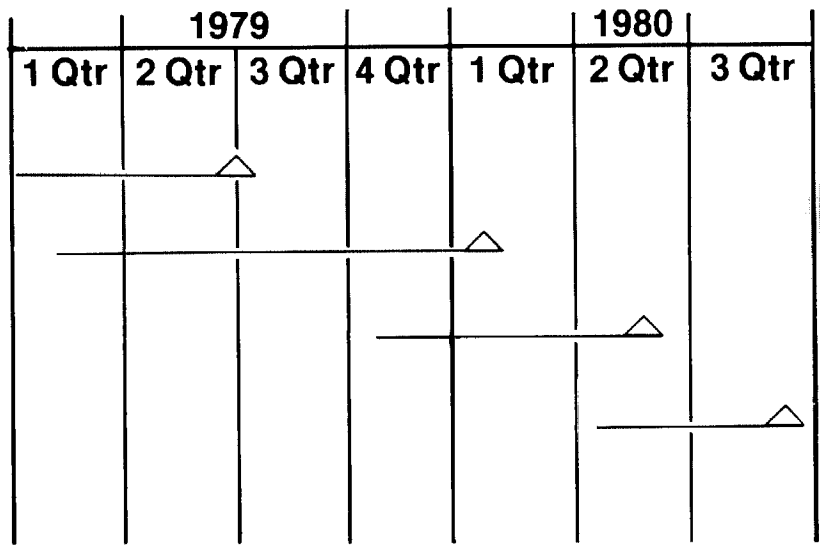
- Transient and Steady State Radial Growth Match of Rotor and Stator
  - Keep Steady State Clearances Small
  - Avoid Rubs During Engine Transient Operation
  - Requires a Modified, Definable, Controlled Shroud Support Structure Thermal Environment and Inertia
- Out-of-Round Distortion of Engine Casings and Structures
  - Causes Local Rubbing at Points of Radially Inward Deformation
  - Local Rubs Wear Blade Tips Uniformly, Thus Increasing Average Clearance
  - Requires Improved Circumferential Uniformity of Mount Reaction Loads, Structural Stiffness, and Thermal Gradients

This chart shows the calculated new engine cruise clearance improvement levels for the improved clearance and roundness control design.

Estimated Payoff

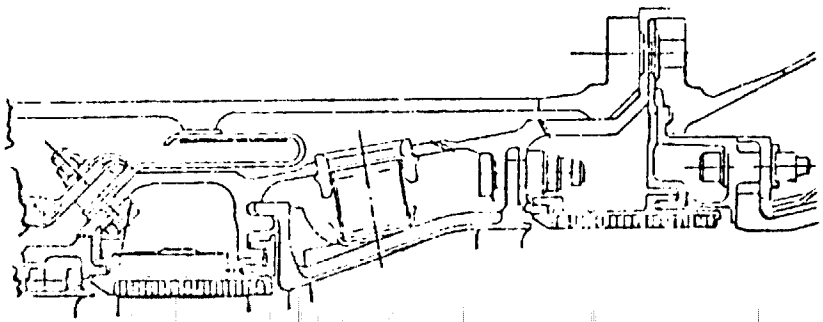
	$\Delta\eta_T$(Pts)	ΔSFC Cruise
HPT Stator Changes	+ .44	– .35
TMF Roundness Changes	+ .54	– .44
Total (New Engine)	+ .98	– .79%

HPT Roundness Control Program Plan

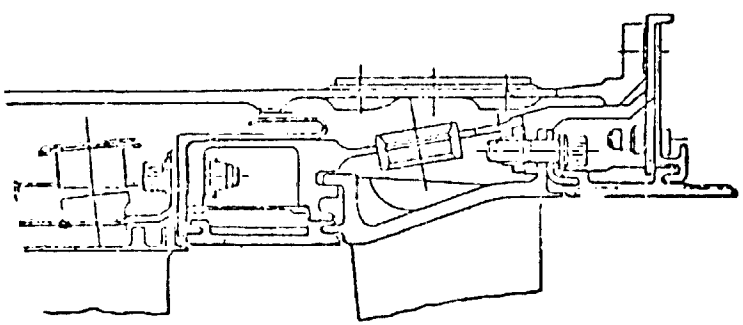


RAA description of the design of a shroud structural supporting system which addresses the factors summarized on the previous page. The major features discussed include those which isolate the structure from the gas stream and create for it an analytically determinable engine environment, reduced deformations due to applied loading, and altenate materials and means of attachment.

Better Thermal Matching
Improved Stage 2 Nozzle Support



Current Design



New Design

- Reduced Cavity Size and Shielding from Gas Recirculation
- Added Mass and Heat Shield for Rotor Response Match
- Reduced Sensitivity to Inward Pressure Deflections
- Segmented Hangers to Isolate Support from Thermal Non-Uniformities

$$\Delta\eta_T = +0.44 \text{ Pts.}$$

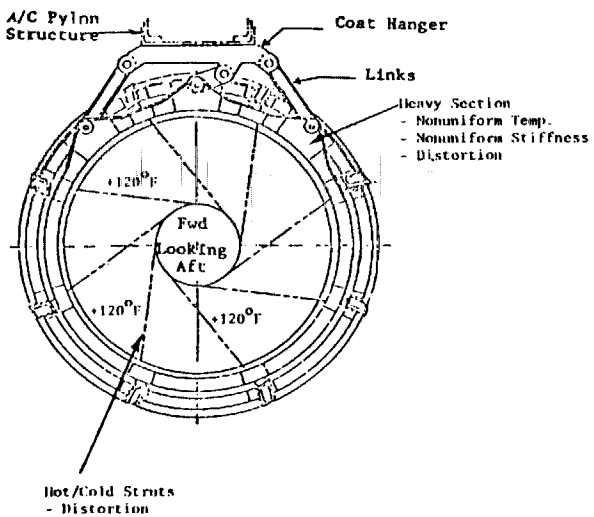
This chart describes modifications to be made to the turbine mid-frame structure to improve its overall structural roundness during engine operation.

Improving Out-Of-Roundness

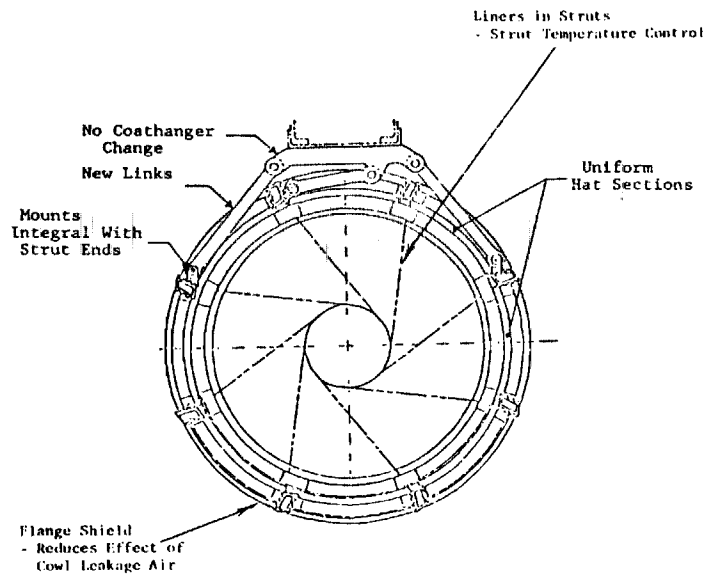
- New Aft Mount Construction
- Uniform Casing Stiffness/Temperature
- Alter Strut Temperature Distribution
- Shielded Flange
  - Eliminate Flange Distortion Caused By Nacelle Leakage

This chart shows the areas of the turbine mid-frame changed to obtain roundness improvements.

Turbine Mid Frame Roundness Improvement



Current Configuration

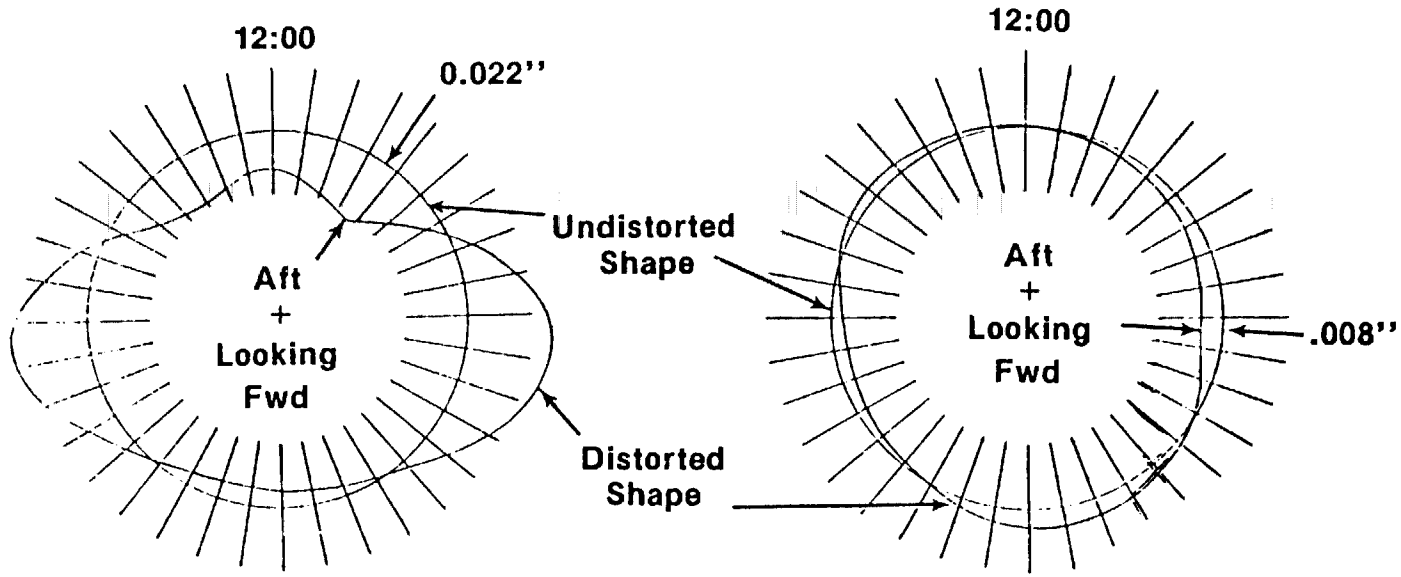


New Configuration

This chart shows the calculated improvement in steady state takeoff out-of-roundness of the stage 1 shroud resulting in the changes identified on the previous page. Also identified is the effect of these changes on turbine efficiency.

Turbine Mid Frame Roundness Improvement

H.P. Turbine Stg 1 Shroud Distortion Due to TMF Steady State Takeoff



Current Configuration **New Configuration**

$\Delta\eta_T = +0.54$ Pts.

Static Load Test

- **Objective:**
  - Determine Effect of TMF Skin Circumferential Gradients, TMF Strut Temperature Differentials, Engine Torque Reaction, Engine Thrust Reaction on HPT Roundness
- **Status:**

  Planned for Mid 1980, Hardware on Order

Instrumented Engine Test

- **Objective:**
  - Conduct Instrumented Engine Test of Improved Turbine Stator and Mid Frame. Conduct Analytical Teardown
- **Status:**
  - Planned for Mid 1980, Hardware Ordered

Endurance Test

- **Objective:**
  - Conduct Engine Endurance Test of Improved Turbine Stator and Mid Frame. Conduct Analytical Teardown
- **Status:**
  - Planned for Mid 1980, Hardware on Order

Summary

HPT Roundness

- Program on Schedule
- Detail Drawings Issued
- Test Hardware Ordered
- Estimated Cruise SFC Improvement for New Engine
  — 0.79% ΔSFC

CF6 Jet Engine Performance
Improvement Program

Task 2.6
HP Turbine Active Clearance Control Performance Improvement Concept

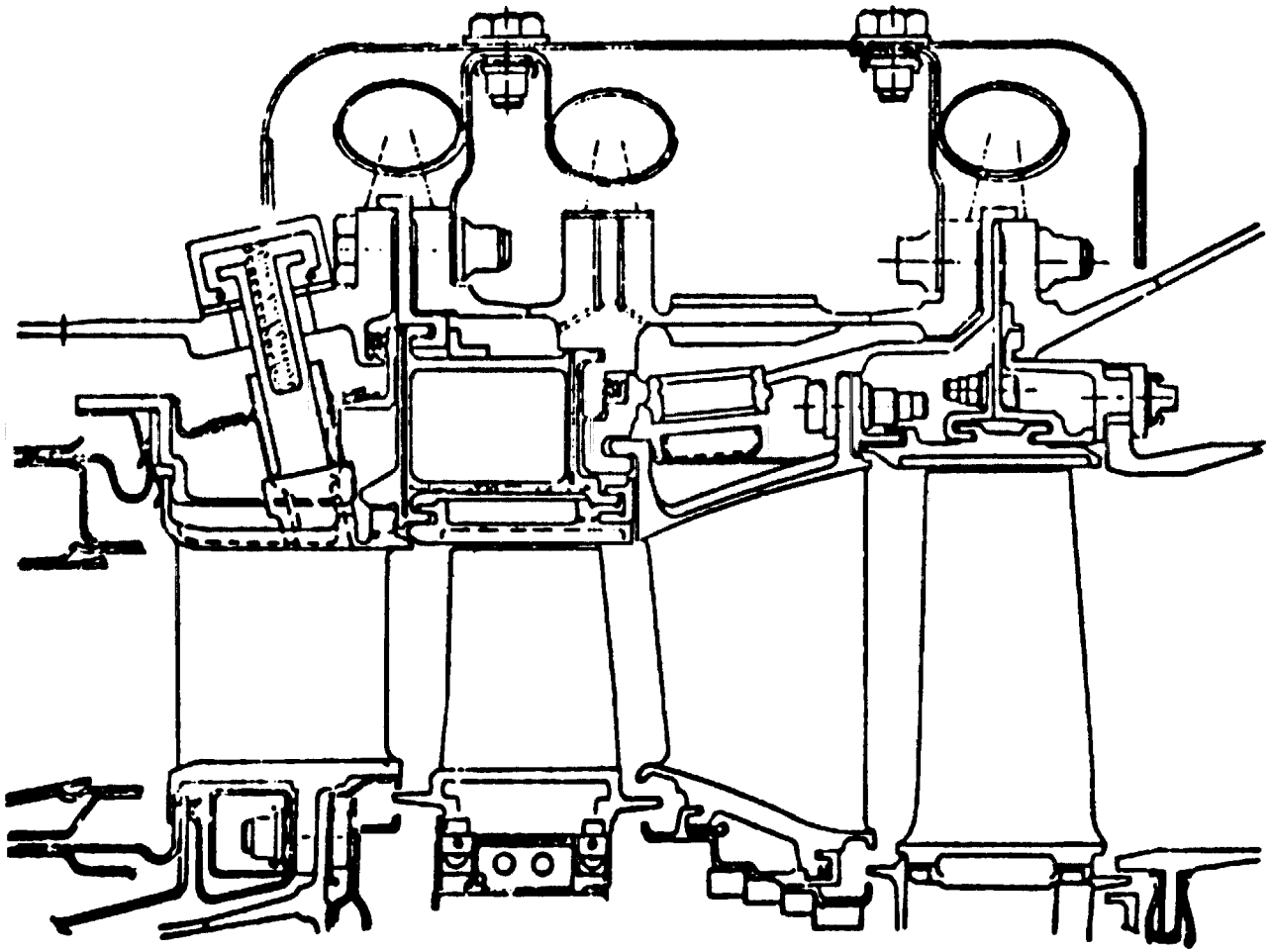
Reduced casing distortion minimizes rub, thereby maintaining tighter clearances. When going from max climb to cruise, rotor shrinks more than stator; external impingement at that time brings stator in to match rotor diameter. A combination of the above optimizes SFC.

Concept Rationale

- Clearance at Cruise Can Be Reduced Without Rub
  - Reduced Casing Distortion
  - Rotor Shrinks Relative to Casing
- Control External Impingement to Shrink the Stator
- Reduce Tip Clearance to Lower SFC at Cruise

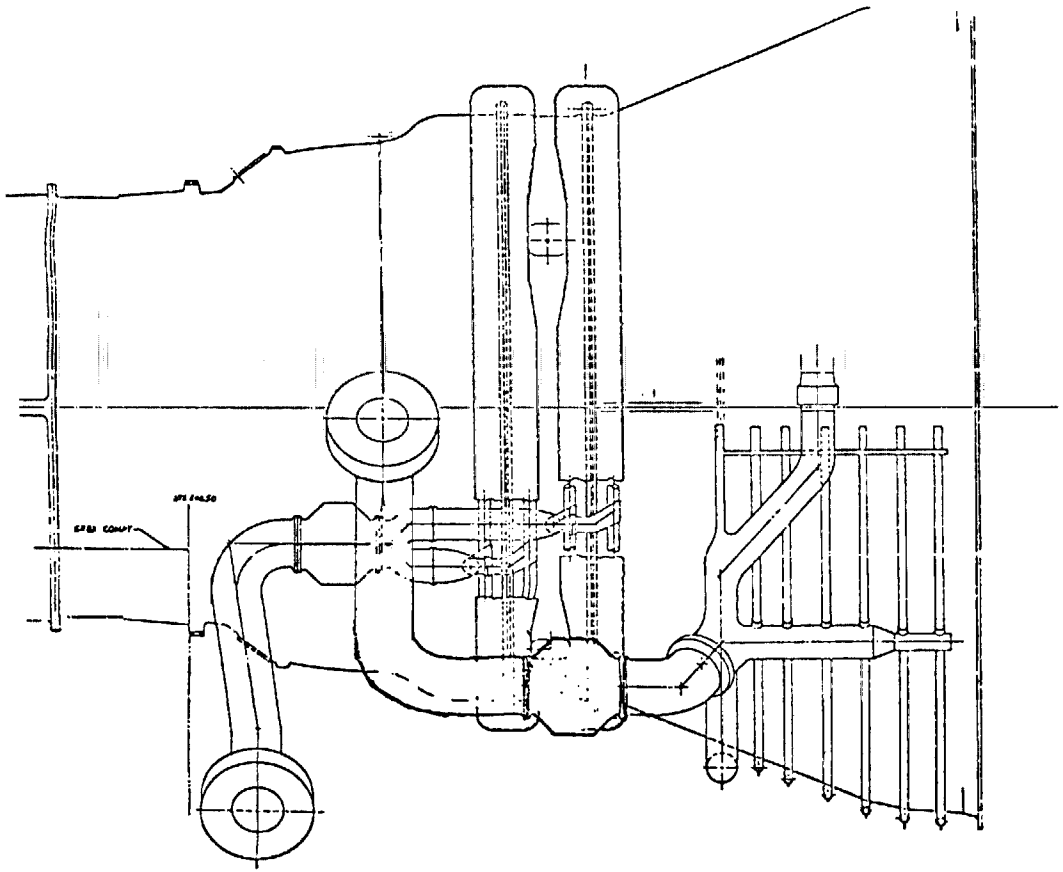
This sketch shows external impingement on the three main stator structural rings. Cooling of these rings effectively reduces the total stator diameter. A cover is used to enhance circumferential temperature uniformity and better utilizes the cooling air.

HPT Turbine Active Clearance Control



This sketch shows how a single extraction and piping system can be used for supplying air to both the HPT and LPT external impingement systems. This minimizes complexity and weight.

Piping and Valves for HPT & LPT External Impingement



By keeping modification to the casing only simplifies new and retrofit introduction. Major teardowns are not necessary and major hardware is reusable with modification. Pre-modified parts changeout can reduce downtime. Same approach can be used for all CF6 family.

Active Clearance Control External System

Summary

- Case Mounted for Simplified Active System
- Major Hardware Reworkable
- Maintain Modular Interchangeability
- Basic Approach Applicable to −6, −32, −50, −80

The SFC improvement on both new and used engines is significant even with the air extraction penalty.

Performance Payoff

HPT Active Clearance Control

	New Engine	3000 Hr.
Cruise △ SFC		
Turbine Efficiency	– .44	– .72
Active Air Penalty to the Cycle	.15	.15
NET	– .29	– .57
Takeoff EGT	– 4°C	– 9°C

Payoff Based On

- ½ of Potential Clearance Improvement
- Delta from Single Shank Turbine

The program has been fully planned and is on schedule.

Active Clearance Control-HPT

NASA Program Schedule

No.	Milestones/Events	YRS / MOS	1979						1980						1981					
			JF	MA	MJ	JA	SO	ND	JF	MA	MJ	JA	SO	ND	JF	MA	MJ	JA	SO	ND
1.	Design Freeze						△ 8-15													
2.	Material Release — HPT				△ 5-1															
3.	Material Release — Configurations					△ 7-15														
4.	Detail Drawings Issued							△ 11-1												
5.	Hardware Delivery										△ 6-1			△ 12-15		△ 4-15				
6.	Component Testing																			
	— Control & System Bench Tests											△ 7-1								
	— Structures Static Load Test										△ 6-1									
	— Piping & Bracket Vib Testing										△ 6-1									
7.	Engine Test												△ 10-1			△ 3-1		△ 7-1		
8.	Analysis					△ 7-15						△ 8-1				△ 5-1				
9.	Final Report																			△ 9-81

GENERAL ELECTRIC COMPANY
AIRCRAFT ENGINE GROUP

The design activity is well underway and on schedule. Critical hardware has been released. The required configuration studies have been completed, and the configuration has been defined.

Design Status

- Turbine Layout Complete
  - Maintains Single Shank Turbine Features
  - Reworkable from Single Shank or Current Production Hardware
- Material Release Complete
- Heat Transfer & Mechanical Analysis
  - Models Complete
  - Studies Underway
- System Arrangement Selected
  - Air Scoop Location Selected
  - Air Flow Requirements Defined
  - Piping Studies Started
  - Slave Valves Released for Procurement
- Control System Studies Started

The program continues on schedule with all goals looking feasible at this time

Summary

HPT Active Clearance Control

- Program Defined
- Design Work on Schedule
- Estimated Cruise SFC Improvement
  .3% Δ SFC — New
  .6% Δ SFC — (3000 Hrs)

CF6 Jet Engine Performance
Improvement Program

Task 2.7
LP Turbine Active Clearance Control Performance Improvement Concept

This chart summarizes the concept being pursued to develop active clearance control for the LPT. A two-position valve activated as a function of altitude will allow increased airflow at cruise in order to cool the casing thereby reducing radial clearances and improving efficiency.

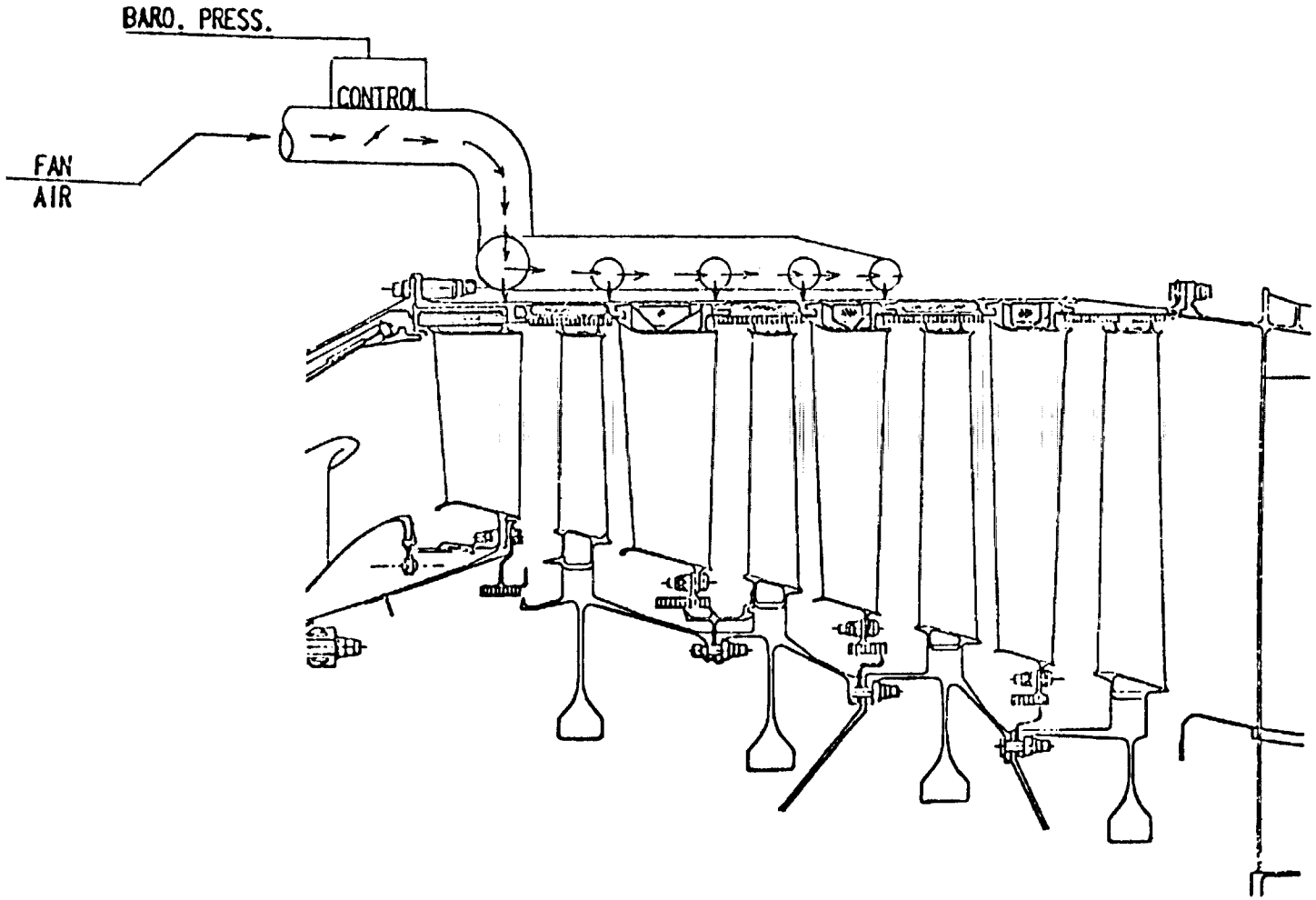
Active Clearance Control Concept

- External Case Cooling Using Fan Discharge Air
  - Same As Present System

- Active Clearance Control During Cruise
  - Continuous Flow for all Power Settings
  - Increased Cooling Flow During Cruise
  - 2 Position Airflow Valve
  - Barometric Pressure (Altitude) Actuated

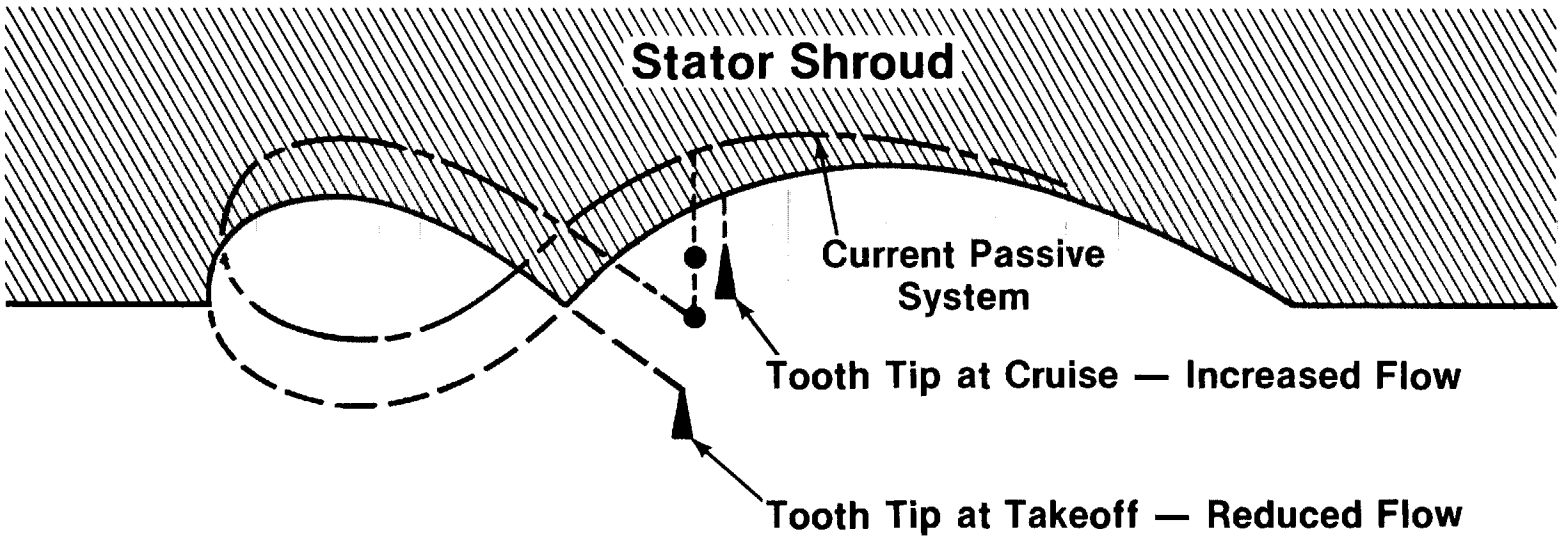
A pictorial representation of the LPT active clearance control system is shown. When the higher level of airflow is activated during cruise, the honeycomb clearances are reduced. This reduces aero losses and fuel consumption. Note that system concentrates on the stages with higher performance payoff.

LPT Active Clearance Control



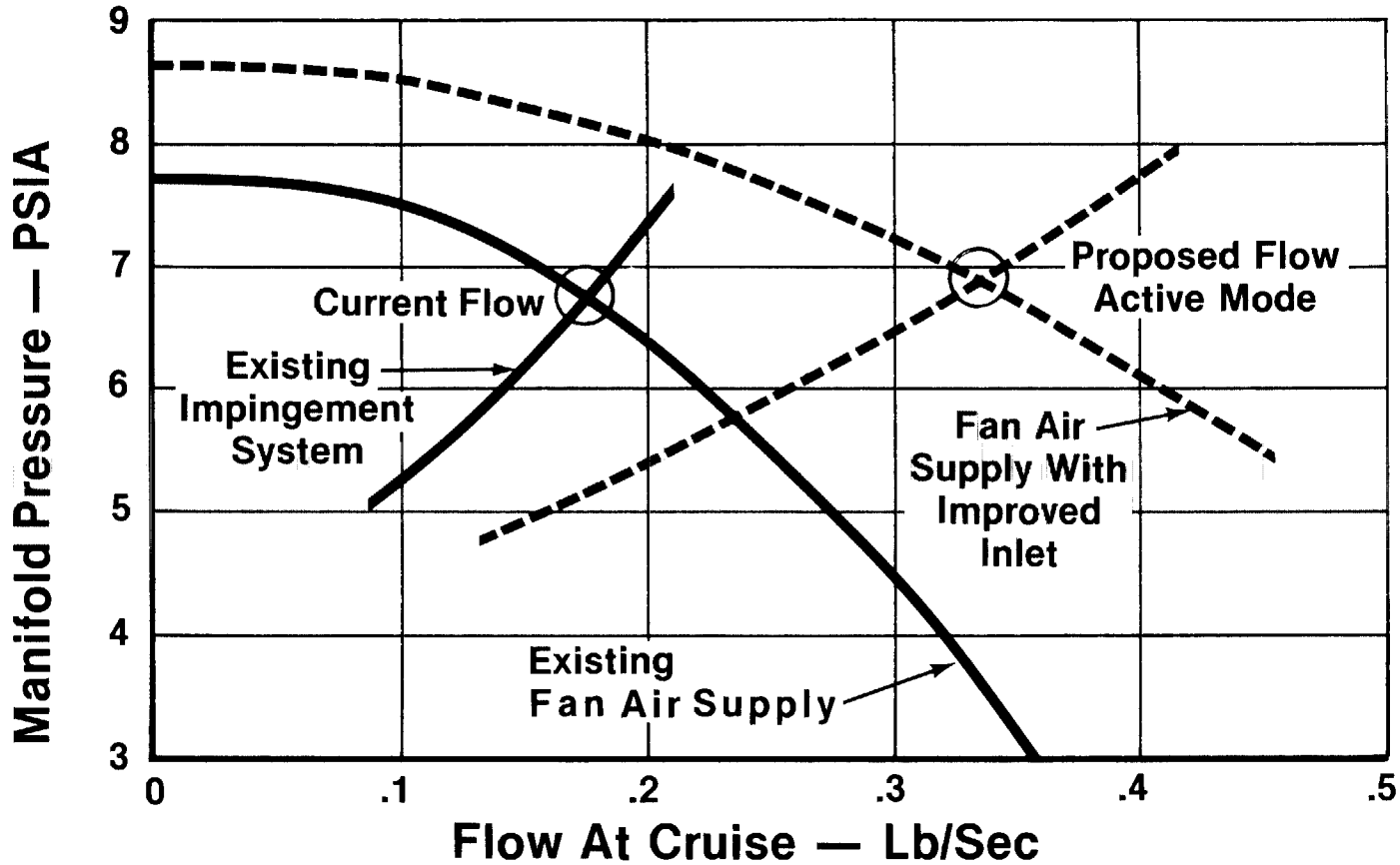
The solid and dashed lines describe the path of a typical blade tip relative to the honeycomb surface during the takeoff transient and on into takeoff and cruise operation. By reducing the level of cooling flow used at takeoff, a reduction in the transient rub depth occurs which is a reduction in cruise steady state clearance.

How It Works


Stator Shroud
Current Passive System
Tooth Tip at Cruise — Increased Flow
Tooth Tip at Takeoff — Reduced Flow

This chart shows flow characteristics of the current and proposed cooling manifold systems. The combination of improved fan supply inlet characteristics and a manifold with increased flow capability allows the use of nearly double the present level of flow in order to accomplish a reduction in casing operating temperature.

Cruise Flow Characteristic

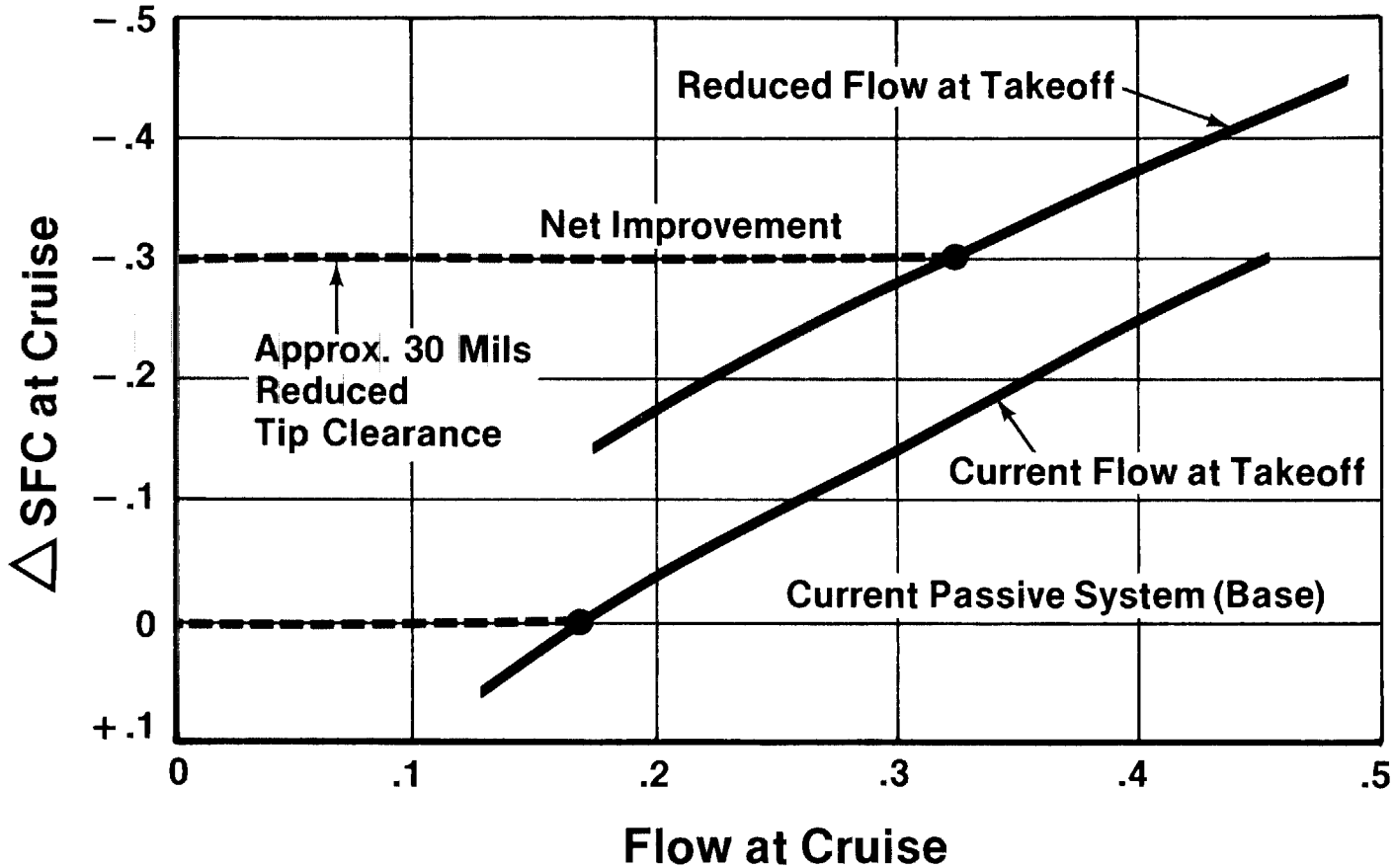


- Conclusion: Airflow Required for Active Mode Can Be Achieved With Existing System

This chart shows the combined improvements of added cruise manifold flow and reduced takeoff transient flow. The net potential in terms of SFC @ cruise is estimated to be .3% based on reduction in running clearance of 30 mils at cruise.

Combined Benefit

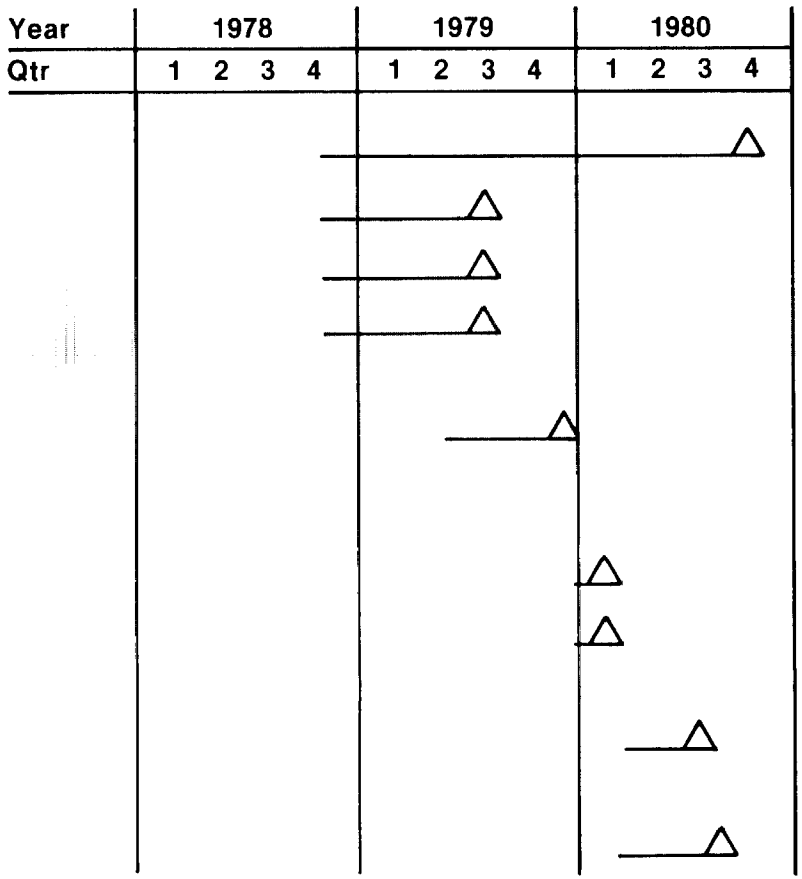
- Reduced T/O Flow
- Increased Cruise Flow



NASA LPT Active Clearance Control Program

Program Plan

- Design and Analysis
  - Heat Transfer
  - Cooling Flow
  - Mechanical
- Hardware Procurement
- Component Test
  - Vibration
  - Flow
- Engine Test
- Report



Summary

- Program on Schedule
- Estimated Cruise SFC Improvement
  - 0.3% Δ SFC

Summary

- Program Status
- Payoff

Status

- Fan, Short Core and Front Mount Programs Complete
- HPT Aero Program Close to Completion
- Remaining Four Programs on Schedule
- Fan, Short Core and Front Mount in Production
- High Production Probability for Remaining Four Concepts

Payoff

- **Demonstrated Cruise SFC Improvement of Completed Program**
  - Fan — 2%
  - Short Core — 0.9%
  - Front Mount (Ti Casing) — 0.1%
- **Estimated Cruise SFC Improvements of Remaining Programs**
  - HPT Aero — 1.3% (−1.6% at 3000 Hrs.)
  - HPT Roundness — 0.8%
  - HPT Active Clearance Control — 0.6%
  - LPT Active Clearance Control — 0.3%
- **Total Projected Fuel Savings for All Concepts**
  - 2 — 2.8 Billion Gallons