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HIGH BYPASS TURBOFAN COMPONENT DEVELOPMENT

Phase II—Fan Detail Design

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December 1979

TECHNICAL REPORT AFAPL-TR-2103

Final Report for December 1978 to October 1979

Air Force Aero Propulsion Laboratory
Air Force Wright Aeronautical Laboratories
Air Force Systems Command
Wright-Patterson Air Force Base, Ohio 45433

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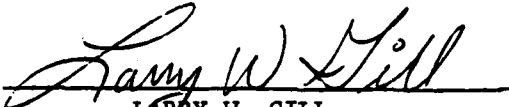
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20. ABSTRACT (Continue on reverse side if necessary and identify by block number) The objective of this program is to develop an advanced, small, high bypass turbofan component applicable to an advanced, fuel-efficient engine for a new Air Force primary trainer aircraft. The program consists of two phases. Phase I consisted of the preliminary design and life cycle cost analysis of candidate engines. This report documents Phase II which was the detail design of the selected fan configuration. Primary design point of the fan is 1.8 pressure ratio at 55.9 lbm/sec flow. At a lower speed, the fan will produce 1.65 (cont)			

→ pressure ratio at a flow compatible with the GMA500 advanced technology core engine. The resultant engine satisfies all requirements of the contract. The fan stage meets all requirements of Mil-E-5007D, including bird ingestion and predicted noise levels are below FAR Part 36 requirements. ←

Preface

This report was prepared by D. C. Chapman of Detroit Diesel Allison, Division of General Motors Corporation, Indianapolis, Indiana.

Design details of the fan stage designed in Phase II of Contract No. F33615-78-C-2014, High Bypass Turbofan Component Development, sponsored by the A.F. Aero Propulsion Laboratory are presented. The Air Force contract monitor was Capt. Larry Gill.

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SUMMARY

An advanced technology fan stage adaptable to a small, high bypass turbofan engine for a future Air Force primary trainer has been designed. Primary design point is at 1.8 pressure ratio and 55.9 lb/sec flow. At a lower speed, this fan stage will produce a 1.65 pressure ratio at a flow compatible with the GMA500 core engine to form a high bypass engine meeting all requirements for the trainer application.

At the 1.8 pressure ratio design condition, the fan rotor operates at 1606 ft/sec tip speed with an inlet annulus specific flow of $42.3 \text{ lbm/sec/ft}^2$. The inlet hub/tip radius ratio is 0.40. The rotor has 20 blades of multiple circular arc airfoil sections with an aspect ratio of 1.64. Maximum thickness-to-chord ratio varies nonlinearly from 8.5% at hub to 2.5% at tip.

Forty-two vanes of multiple circular arc cross section are tilted rearward at the tip to increase blade to vane spacing for noise considerations. The vane aspect ratio is 2.32 and the maximum thickness-to-chord ratio varies from 6% at the hub to 8% at the tip.

The design meets all structural design requirements including steady and vibratory stresses, blade flutter, and bird ingestion. Substantial margins exist for blade and wheel permanent set at 122% speed and for burst at 130% speed. Adequate margins also exist for low cycle fatigue, considering 12,000 cycles to design speed with $K_t = 3$ at blade leading and trailing edges, $K_t = 1.4$ at blade crown, $K_t = 1.4$ in the wheel rim, and $K_t = 2.0$ in the wheel web. Allowable blade vibratory stress exceeds the required ± 15 ksi at resonance points and the required ± 5 ksi at nonresonance points.

The blading was also checked for torsional stall flutter and found to be satisfactory.

Bird ingestion requirements of Mil-E-5007D forced a slight thickening of the blade leading edge region in conjunction with a material change from titanium to stainless steel.

The predicted noise levels are substantially below FAR Part 36 levels at take-off and approach. Furthermore, the ground idle noise levels are greatly reduced from those of the current trainer configuration.

SECTION I

INTRODUCTION

The current Air Force primary trainer (T37) fleet is approaching the end of its useful life, and a replacement aircraft will be needed. A fuel-efficient engine for the replacement aircraft must be developed. Advanced technology engines, such as the GMA500, suitable for the core of such a high bypass engine are being developed. Advanced technology fan stages in this size class are not available. To fill this void in technology, Detroit Diesel Allison (DDA), Division of General Motors Corporation, has conducted the High Bypass Turbofan Component Development Program for the United States Air Force Aero Propulsion Laboratory, Wright-Patterson AFB, Ohio. The program consisted of two phases:

- Phase I--Preliminary Design and Life Cycle Cost Analysis of Candidate Engines
- Phase II--Detailed Design of the Fan stage chosen from Phase I

The Phase I studies were reported in Report No. AFAPL-TR-79-2034, High Bypass Turbofan Component Development, Phase I--Preliminary Design and Life Cycle Cost Analysis of Candidate Engines by D. C. Chapman and W. A. Redmond. This report documents the design of the fan stage completed in Phase II of the program.

Section II

DESIGN REQUIREMENTS

The core engine of choice in the Phase I selection process was the GMA500, an advanced technology turboshaft engine which was a winner in the United States Army Advanced Technology Demonstrator Engine (ATDE) Competition. DDA is currently under contract to complete 500 hours of running on the GMA500 engine starting early in CY 1979. The engine consists of a two-stage centrifugal compressor, foldback annular combustor, two-stage gasifier turbine, and two-stage power turbine.

In Phase I of this program, fans of 1.5, 1.65, 1.8, and 2.0 pressure ratio were matched with the GMA500 core engine to form candidate high bypass ratio engines for system life cycle cost analysis. The performance of all engines met or exceeded the requirements of this contract. Using representative aircraft characteristics, these engines, designated PD418, were applied to the mission requirements established by the Air Force. Both aircraft gross weight and system life cycle cost were minimized with a fan pressure ratio of 1.65, although the advantage of that pressure ratio over 1.5 and 1.8 pressure ratios was not great. DDA, therefore, recommended to the Air Force that the 1.65 pressure ratio fan be selected for detail design in Phase II of the program.

The Air Force identified a higher technology level with the 1.8 fan pressure ratio and because the Life Cycle Cost (LCC) penalty was small, selected that pressure ratio for Phase II. DDA preferred the 1.65 pressure ratio not only because of the LCC analysis but because the engine sea level static thrust level was approximately 7.5% greater than the engine with 1.8 fan pressure ratio. A mutually agreeable set of design conditions were established wherein the fan would be designed to achieve 1.65 pressure ratio at a flow compatible with the GMA500 core engine and also to achieve 1.8 pressure ratio at a higher flow and speed. The primary design point thus established is:

- Pressure ratio 1.8:1
- Corrected flow 55.86 lbm/sec
- Efficiency 85%

The secondary design point, which matches the GMA500 core engine requirement at 25,000 ft, 0.5 Mach number is:

- Pressure ratio 1.65:1
- Corrected flow 52.8 lbm/sec
- Efficiency 87%

Structurally, the fan should meet the requirements of Mil-E-5007D, including bird ingestion capability. Furthermore, the fan noise levels should be within the limits of FAR Part 36 at both takeoff and approach. An unofficial goal was to achieve a substantial reduction in noise at ground idle compared to the existing primary trainer.

SECTION III AERODYNAMIC DESIGN

The aerodynamic design is presented at the 1.8 fan pressure ratio operating condition. Flow-path and vector diagram details are followed by rotor and stator blading information.

FLOW-PATH AND VECTOR DIAGRAMS

The design parameters for the small high bypass fan are:

● Stage pressure ratio	1.8:1
● Corrected flow rate, lbm/sec	55.86
● Adiabatic efficiency, %	85.2
● Rotor inlet hub/tip radius ratio	0.40
● Corrected tip speed, ft/sec	1606
● Corrected speed, rpm	21685
● Mechanical speed, rpm	20223
● Corrected specific flow rate, lbm/sec/ft ²	42.28

The velocity diagrams of the fan were obtained using the DDA Axial Compressor Design System. A description of the design system is given in Appendix A.

The fan was designed at an altitude cruise condition of 25,000 ft at 0.5 Mach number. This point represents the maximum mechanical speed achieved by the fan in a representative trainer mission.

A schematic of the fan flow-path is shown in Figure 1. The fan has a constant tip diameter of 16.974 in. and a rotor inlet hub-to-tip radius ratio of 0.40. The rotor hub ramp angle is 31.25 deg. The number of rotor airfoils is 20 while the stator has 42 vanes. The number of vanes and the vane-blade spacing were consequences of acoustical considerations.

The average value of the blade inlet absolute Mach number is 0.617. The blade inlet relative Mach numbers are supersonic for the outer 75% of the span. The exit relative Mach numbers are all subsonic (Figure 2). The average vane exit Mach number is 0.465. The inlet and exit Mach number profiles for the vane are shown in Figure 3.

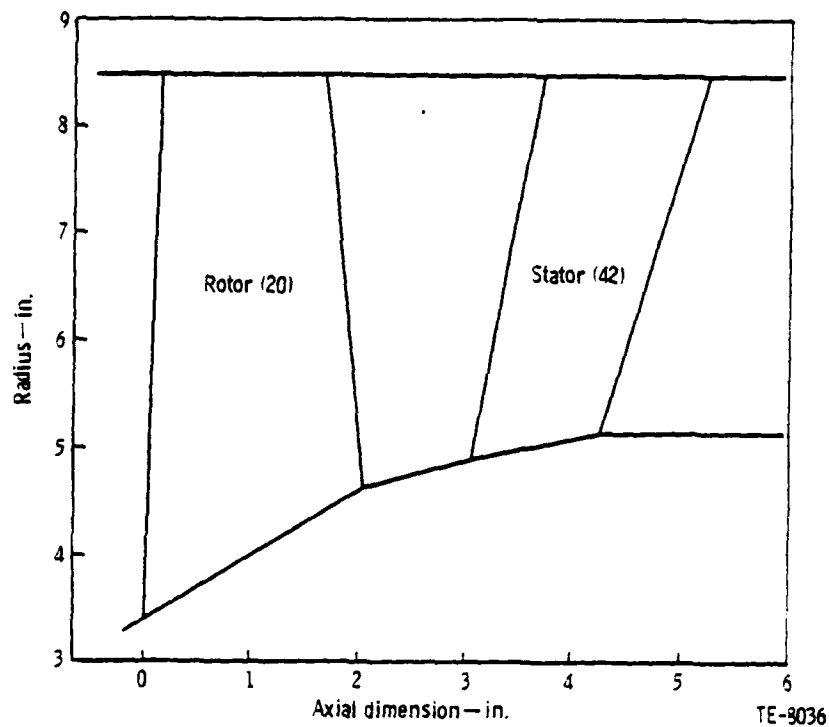


Figure 1. Schematic of fan flow path.

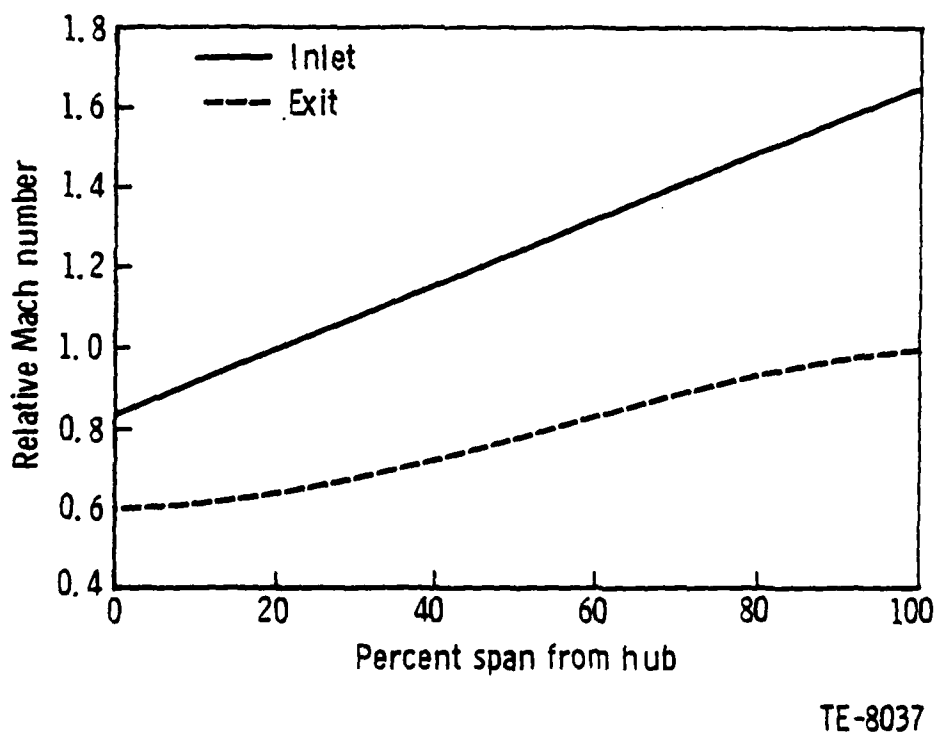


Figure 2. Blade Mach numbers.

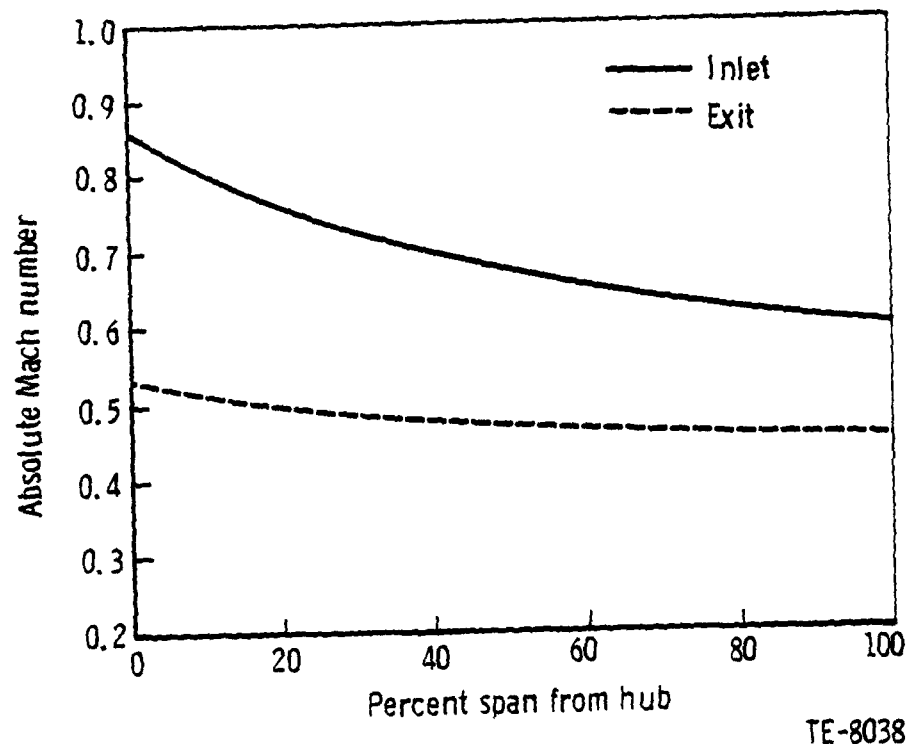
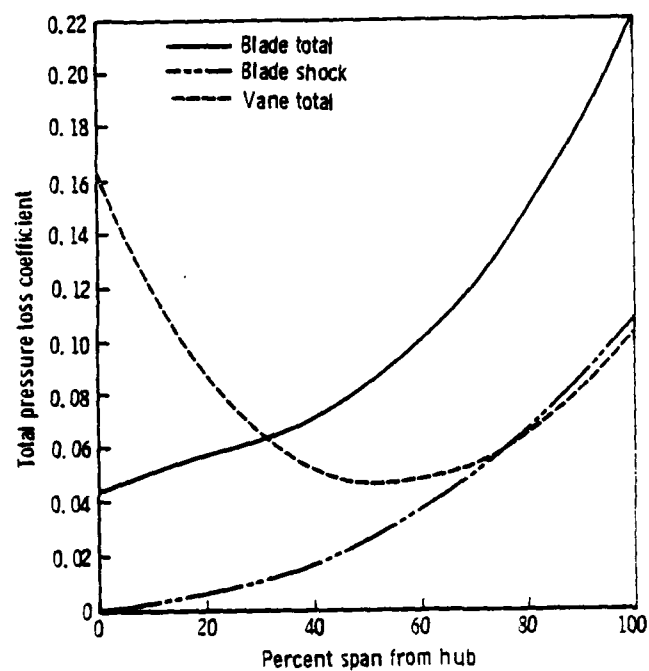


Figure 3. Vane Mach numbers.

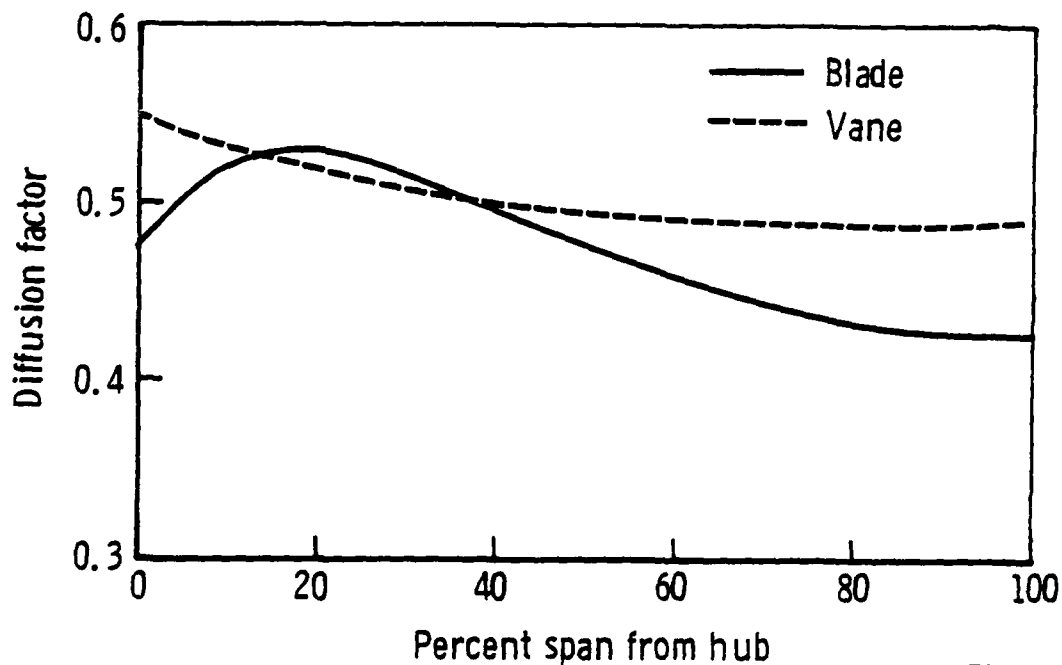
The predicted blade and vane total pressure loss coefficients are illustrated in Figure 4. The resulting average efficiencies are 88.4% for the blade and 85.2% for the stage. The spanwise distribution of the design point loadings (diffusion factors) are shown in Figure 5. They are moderately high but the estimated surge margin for the fan is 18.6%. This surge margin estimate is based on a correlation of blade aspect ratio, relative Mach number, and tip loading at surge for various single stage compressors (Figure 6).

Figure 7 shows the blade inlet and exit relative air angles while Figure 8 is a plot of vane turning angles. The exit air angle from the vane is designed to be 0.0 degrees (axial).



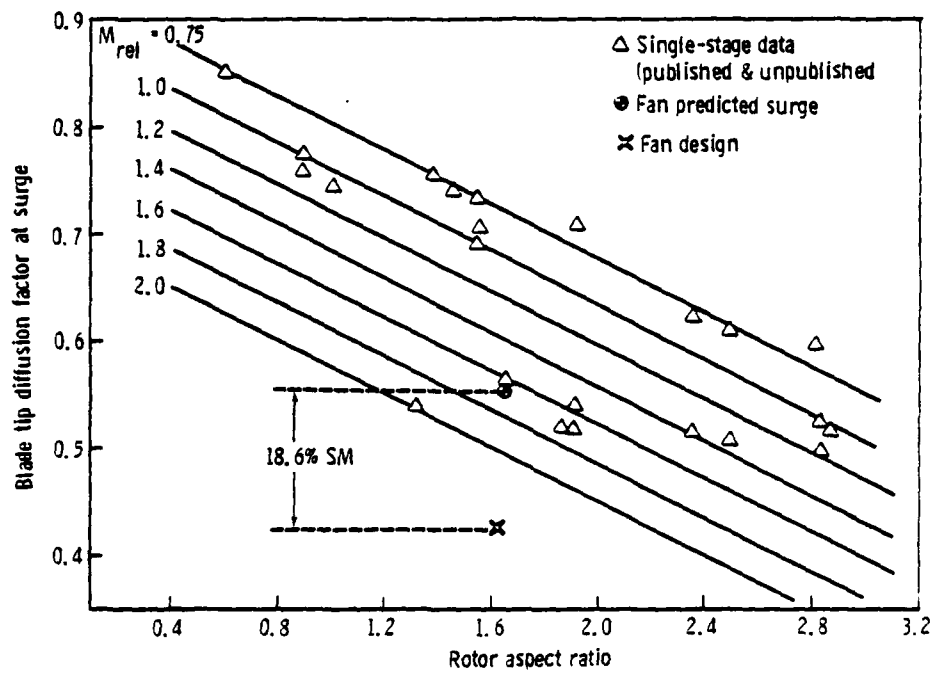
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Figure 4. Radial distribution of total pressure loss coefficient.



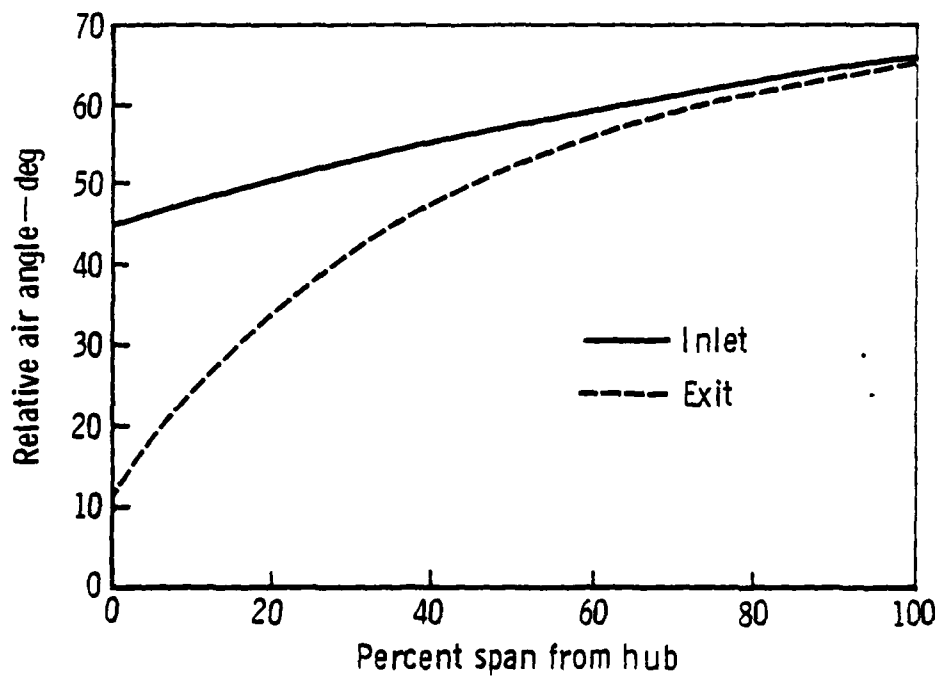
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Figure 5. Blade and vane loading distributions.



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Figure 6. Single stage surge margin correlation.



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Figure 7. Blade air angles.

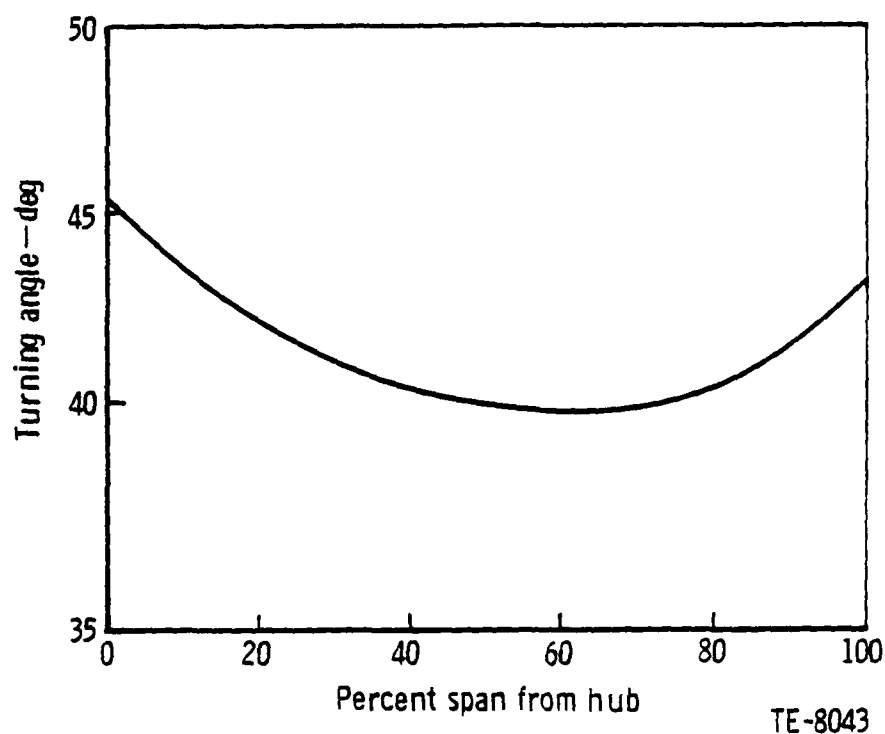


Figure 8. Vane turning angles.

The design point vector diagrams, calculated along streamlines, are tabulated for the blade and vane leading and trailing edge stations in Appendix B.

BLADE DESIGN

There are 20 blades with an aspect ratio of 1.64 (based on average span and true mean chord). The blade consists of multiple circular arc (MCA) airfoil sections designed on conical surfaces approximating streamlines of revolution. An MCA airfoil is shown schematically in Figure 9. It is made up of two circular arcs which define three metal angles: inlet (β_1^*), exit (β_2^*), and inflection (β_i^*). A metal angle is the angle between the axial direction and the mean camber line at a specified location. A blade section is designed by adjusting the metal angles to satisfy incidence, deviation, and starting

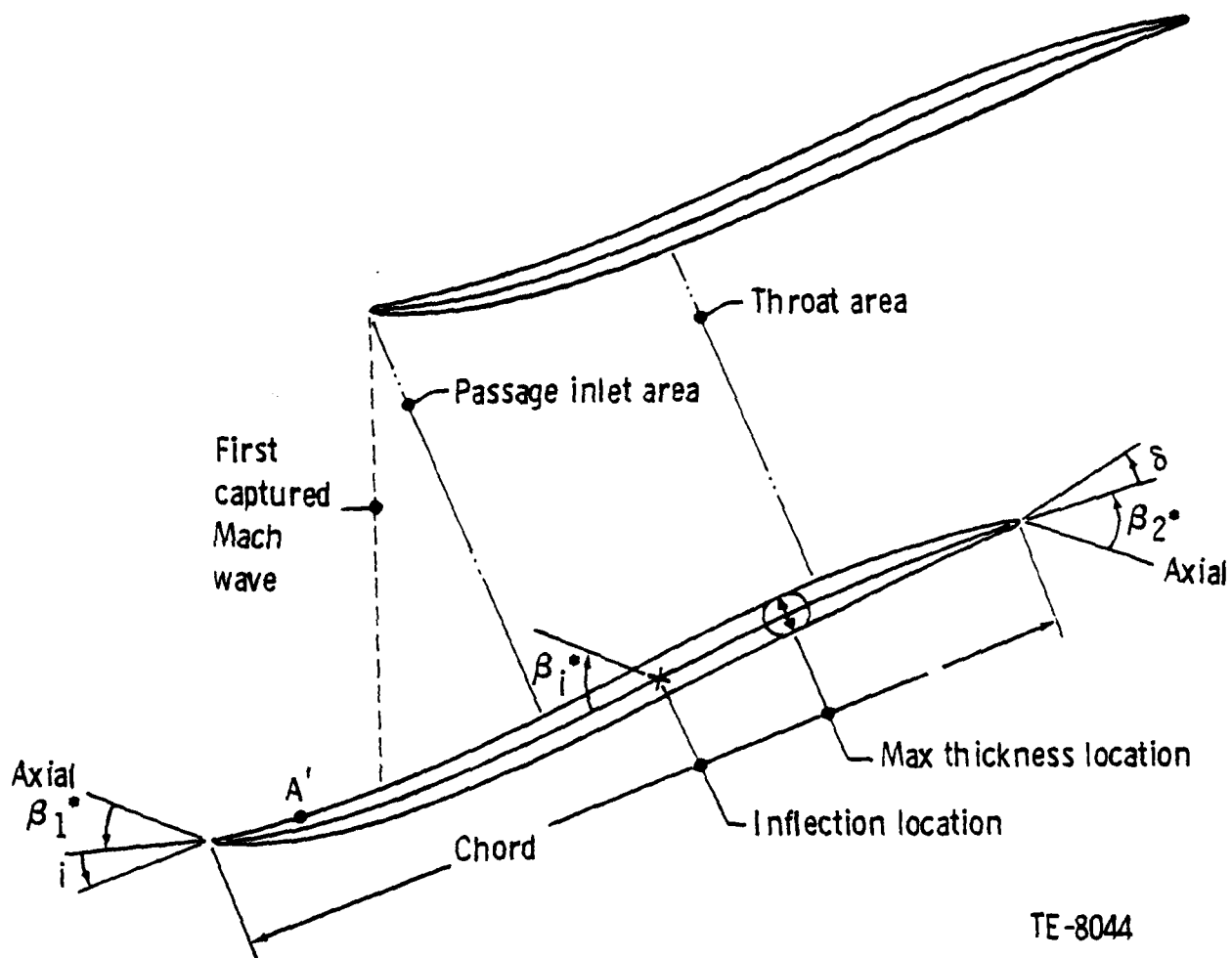


Figure 9. MCA airfoil definitions.

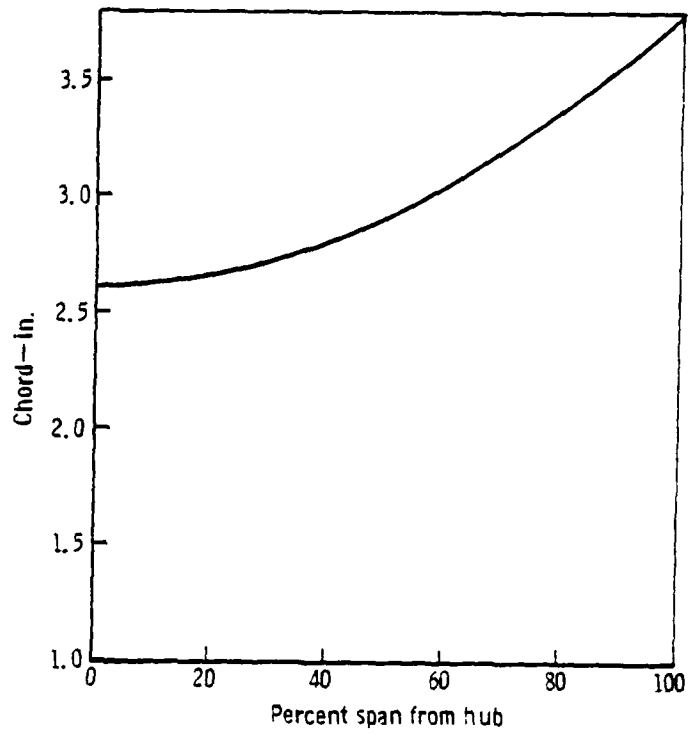
margin criteria. In the outer portion of the fan blade, where the inlet relative Mach number is supersonic, the airfoils were shaped to minimize shock loss. In the subsonic region of the blade, the airfoil shape transitions from the first supersonic section down to a near double-circular arc airfoil section at the hub.

A low aspect ratio, and, therefore, a long average chord, was selected to meet flutter criteria without the use of part-span shrouds. The spanwise chord taper was selected to satisfy the solidity requirements and also be viable from a weight and stress standpoint. The radial distributions of chord and solidity for the blade are shown in Figures 10 and 11, respectively. The maximum thickness to chord ratio (Figure 12) was set to avoid responsive resonant conditions and to maintain radial uniformity of blade mechanical properties. One of the mechanical considerations in the design was blade integrity with bird ingestion. The leading edge radius of the blade was set at 0.0125 in. from the hub to 60% span and then tapered to 0.010 in. at the tip. This maximized blade strength in the primary impact area while at the same time minimizing the efficiency penalty in the high inlet Mach number area at the blade tip from increased shock loss.

For the portion of the blade which has supersonic relative inlet Mach numbers, incidence was set on the suction surface at a point halfway between the leading edge and the emanation point of the first captured Mach wave (point A' of Figure 9). This incidence is the offset of the suction surface from a "free" streamline, which would exist if there were no blade forces, and it establishes the maximum flow the cascade can pass when the throat is not the limiting factor. The incidence value was set at 1.5 deg and is intended to account for leading edge blockage, suction surface boundary layer, and the bow shock wave. In the subsonic portion of the blade, the meanline incidence for each airfoil section was selected to locate the throat near the passage inlet.

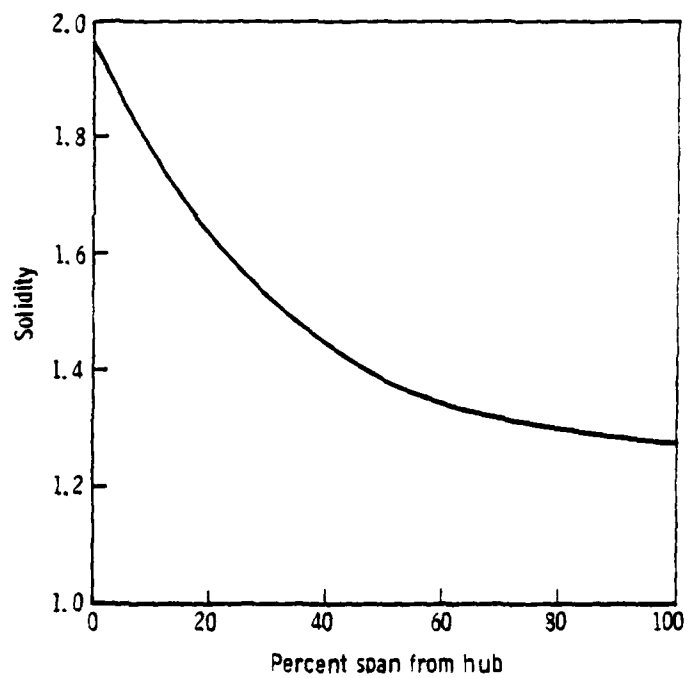
Deviation angles were calculated using a modified form of the NACA 2-D rule for circular arc meanlines and then adding an empirical adjustment. The modification is a circulation correction based on the radius change of the streamline across the blade airfoil section.

The radial distributions of meanline incidence angle and deviation angle are shown in Figure 13. The resulting meanline blade angles are shown in Figure 14.



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Figure 10. Blade chord.



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Figure 11. Blade solidity.

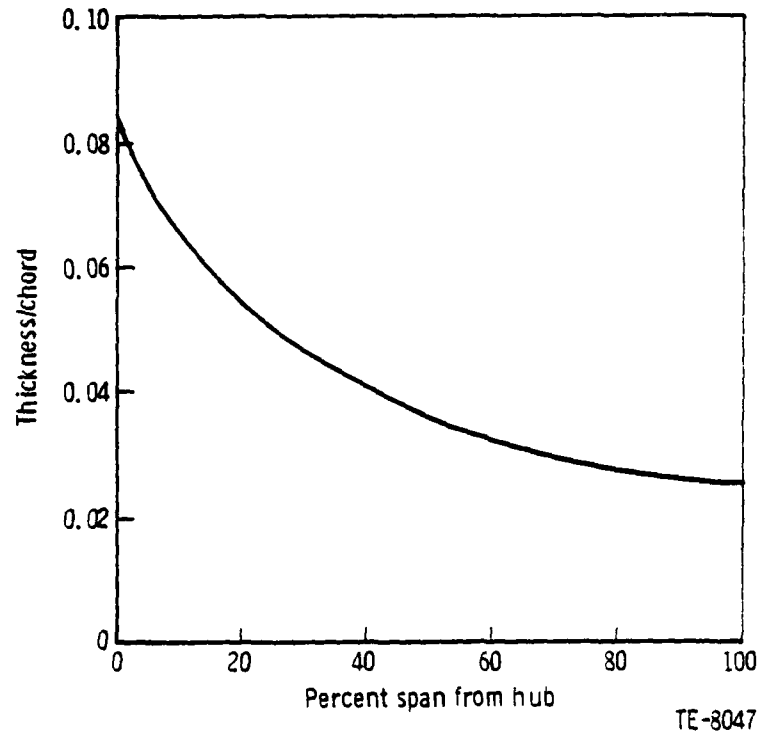


Figure 12. Blade maximum thickness/chord ratio.

The chordwise location of maximum thickness and circular arc inflection are shown in Figure 15. These were selected for two reasons: (1) to avoid accelerating suction surface curvatures ahead of the anticipated passage inlet shock wave location and (2) to set the passage inlet area and contour the airfoil passage between the passage inlet and the throat to minimize the velocity change through the passage. The design throat minimum critical area ratio (A/A^*_{min}) distribution for the supersonic airfoil sections is set to 1.03 for a normal shock total pressure loss applied at the passage entrance with a linear distribution of profile loss from the leading to trailing edge of the airfoil section. Streamtube contraction and the effect of radius change are accounted for.

Figure 16 shows the blade hub, mean, and tip conical airfoil sections in engine orientation. For manufacturing purposes, the airfoil sections were redefined on planes normal to the stacking line. The stack line is a radial line passing through the center of gravity of the hub conical section. The blade manufacturing coordinates are listed in Appendix C with definitions given in Figure C-1.

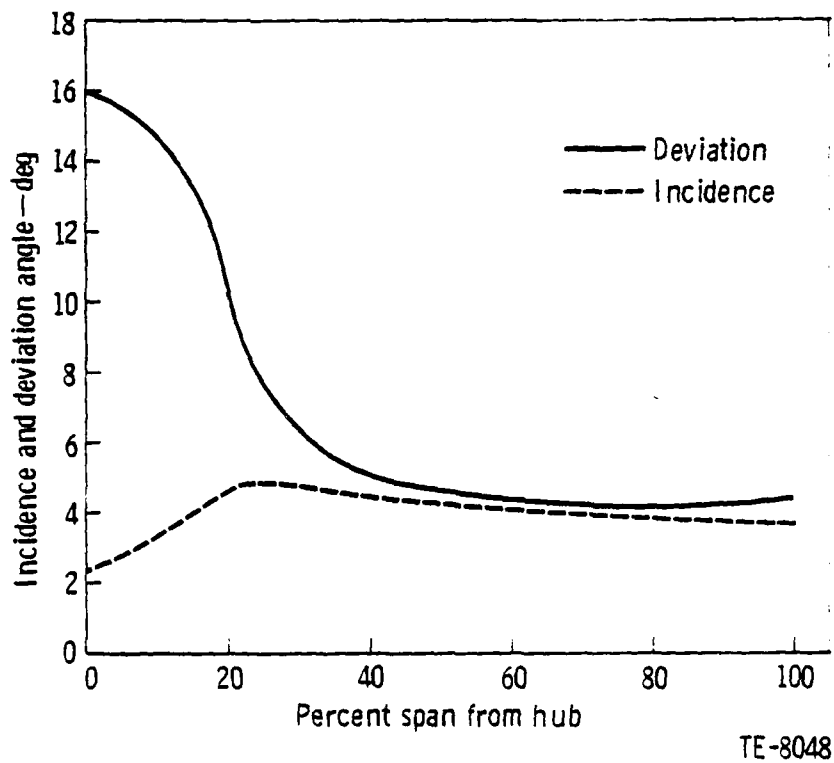


Figure 13. Blade incidence and deviation angles.

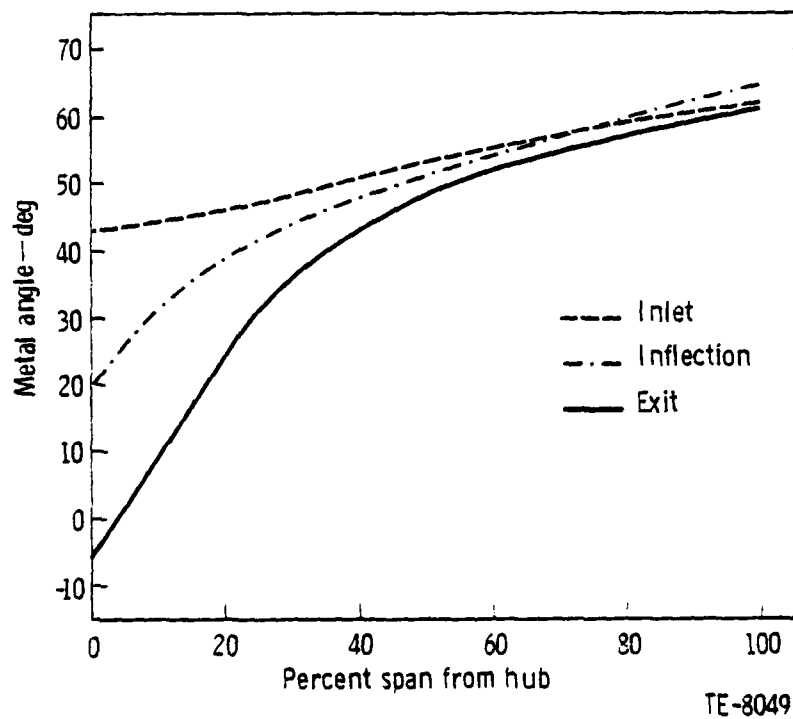


Figure 14. Blade metal angles.

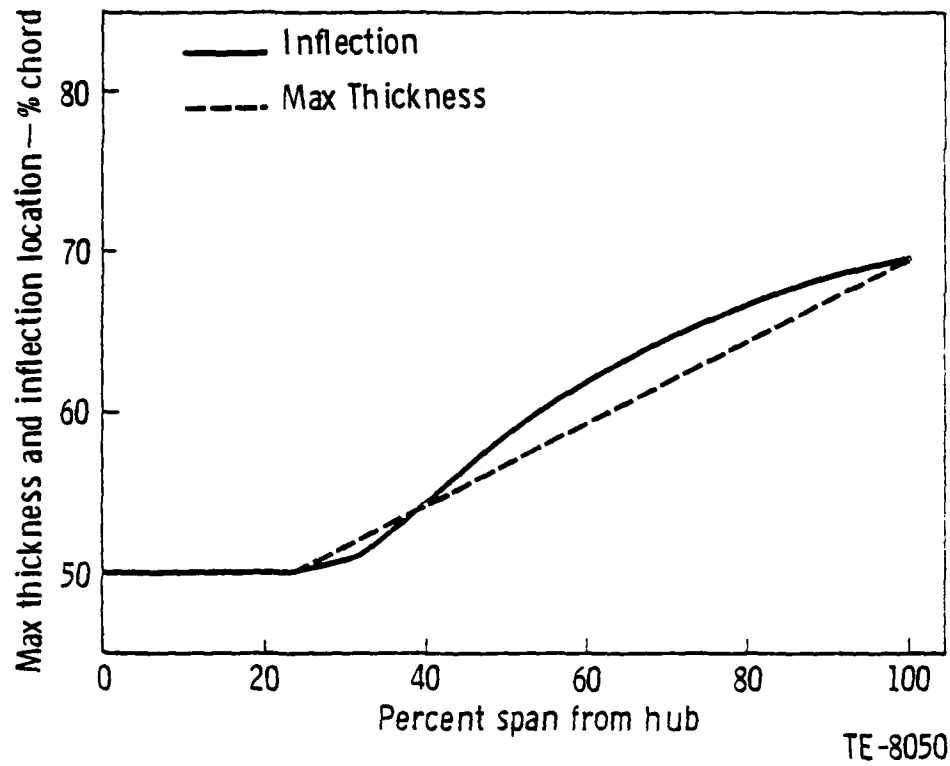


Figure 15. Blade maximum thickness and inflection locations.

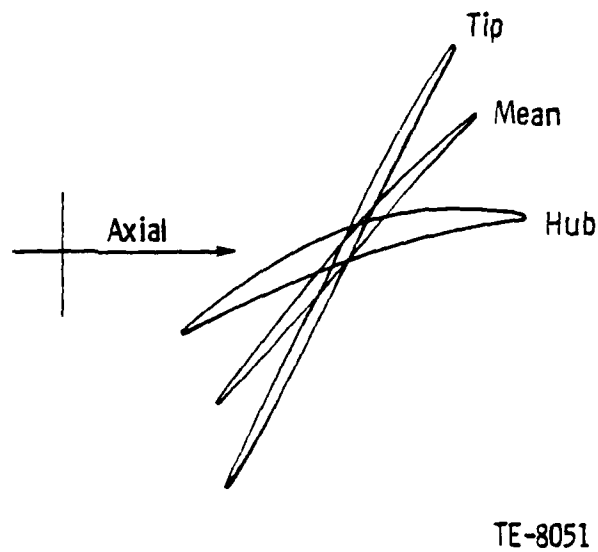


Figure 16. Blade conical airfoil sections.

VANE DESIGN

The vane is also made up of MCA airfoil sections. The vane axis is tilted rearward from radial at an angle of 13.5 deg. This gives a more desirable acoustic spacing between the blade and vane at the tip while minimizing hub length for bypass engine applications. There are 42 vanes with an aspect ratio of 2.32 and a solidity of 1.78 at the I.D. and 1.30 at the O.D. (Figure 17). The radial distribution of chord is shown in Figure 18. The maximum thickness to chord ratio varies linearly from 6% at the hub to 8% at the tip.

The incidence angles were selected to position the throat location at the vane passage inlet. The passage throat margins were based on minimum loss cascade data (Figure 19). Deviation angles were determined from the NACA 2-D rule with an empirical adjustment. The incidence and deviation angles for the vane are presented in Figure 20. Vane metal angles are shown in Figure 21. Vane hub, mean, and tip conical airfoil sections are illustrated in Figure 22.

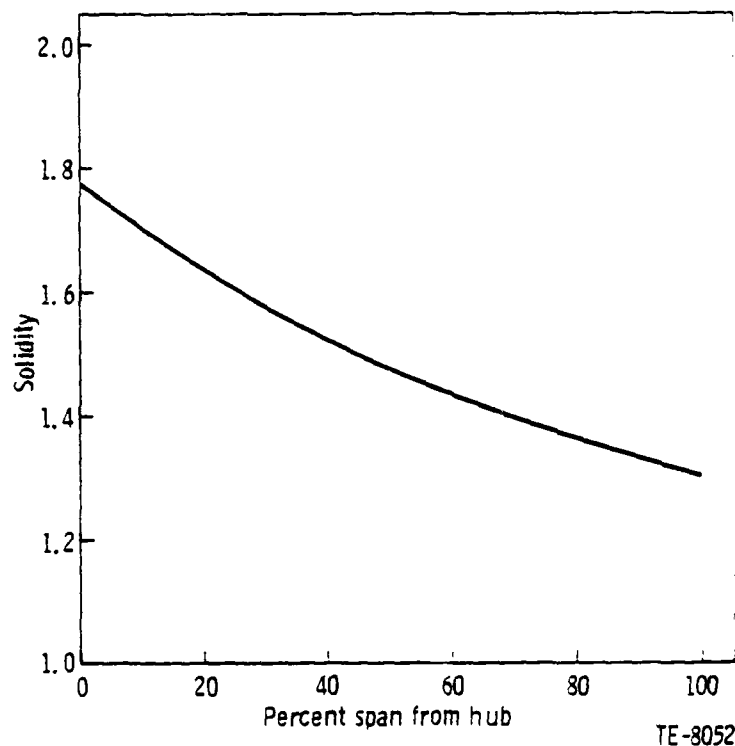
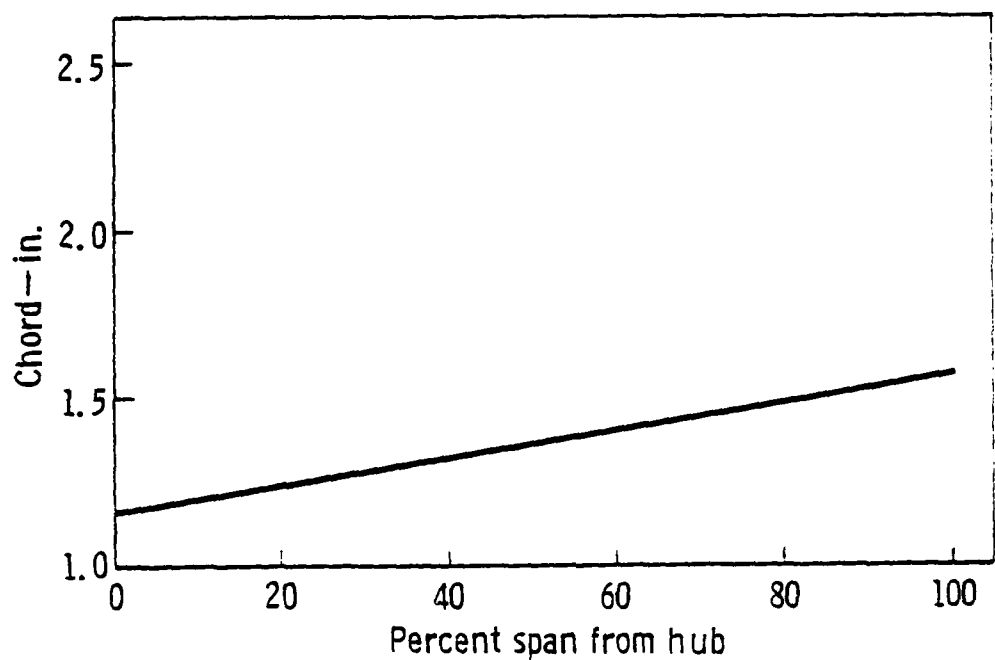
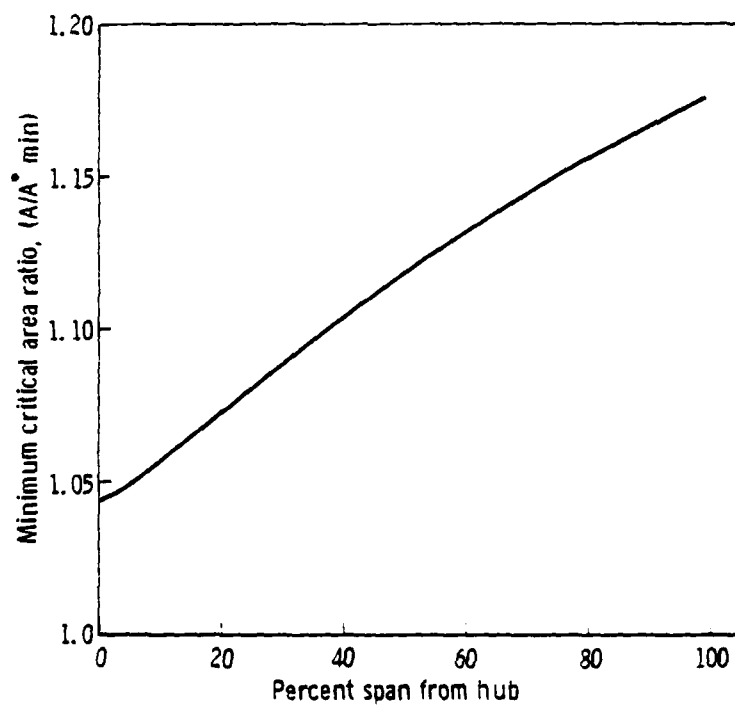


Figure 17. Vane solidity.



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Figure 18. Vane chord.



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Figure 19. Vane passage throat minimum critical area ratio.

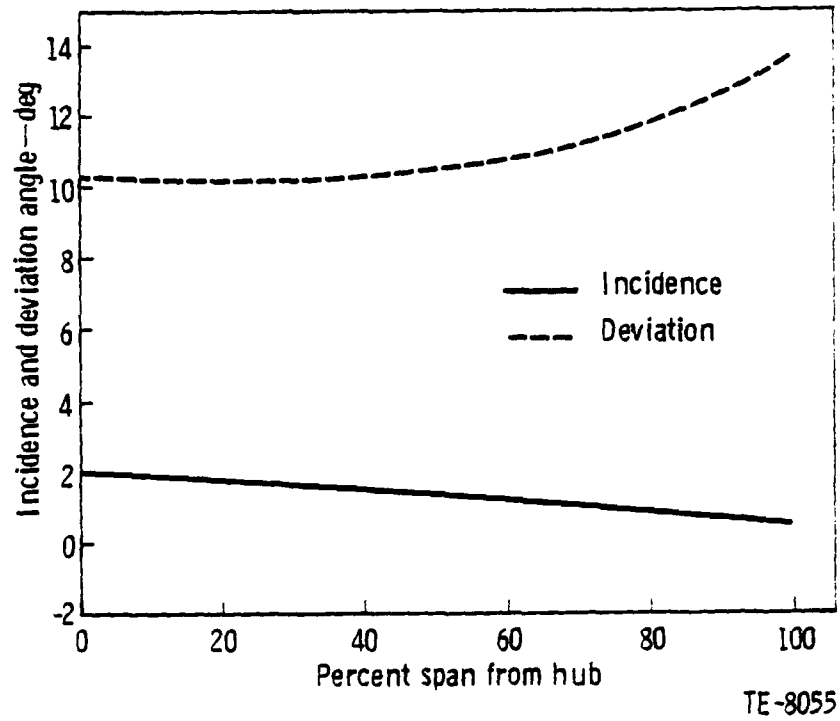


Figure 20. Vane incidence and deviation angles.

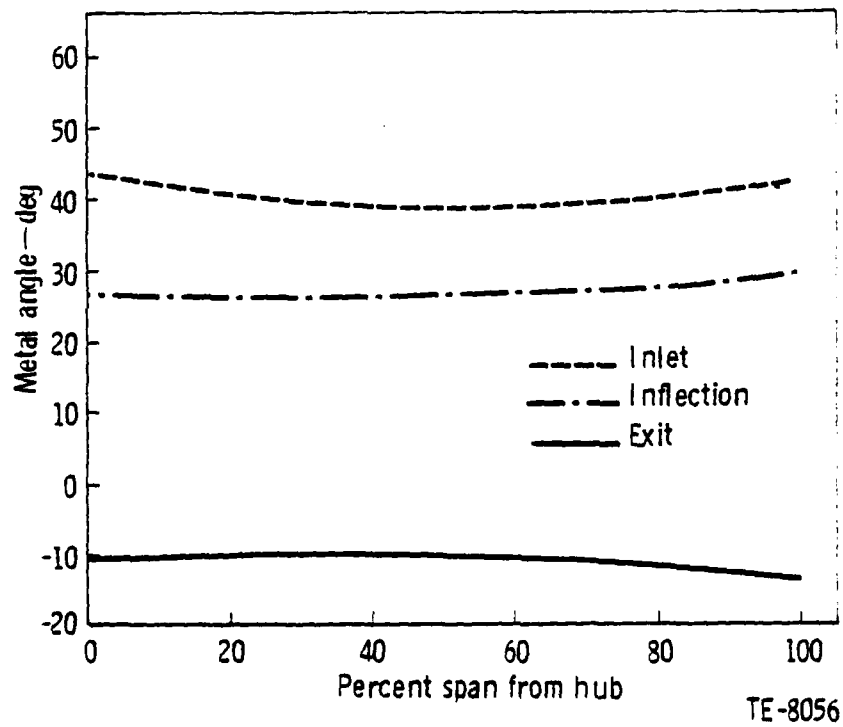
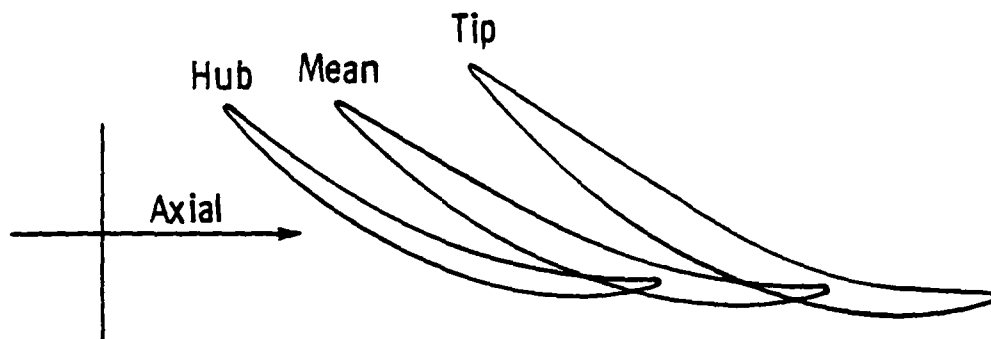


Figure 21. Vane metal angles.



TE-8057

Figure 22. Vane conical airfoil sections.

The manufacturing coordinates for the vane are given in Appendix C with pertinent airfoil section definitions given on Figure C-2. The section coordinates were defined on planes normal to a stacking line. The stack line for the vane is on a radial line passing through the vane hub section c.g.

SECTION IV

STRUCTURAL ANALYSIS

Structural analysis of the fan rotor consisted of calculating airfoil and wheel steady-state stresses, airfoil vibrational characteristics, and bird ingestion capabilities. Satisfying all these requirements while retaining an aerodynamically acceptable configuration required numerous iterations on the design with changes in flow-path shape, spanwise chord and thickness distribution, and eventually a material change. The material change resulted from the inability to analytically satisfy bird ingestion requirements in an aerodynamically viable blade with titanium material. A stainless steel material (17-4 PH) was, therefore, substituted with consequent weight penalties. The final design meets or exceeds the requirements of Mil-E-5007D.

A titanium rotor (Ti-6Al-4V) design was near completion when the material change was deemed necessary. A satisfactory match-up of blade to wheel had not been obtained, and a valid hot-to-cold run was yet to be completed when the steel rotor analysis was started. Airfoil stresses reported here for the titanium rotor were determined with internal program boundary conditions for clamped hub, free tip. These automated constraints are not accurate for an integral blade-wheel rotor; preliminary blade/wheel match-up attempts indicated that airfoil crown stresses would be increased approximately 10 KSI over the reported results while leading edge and trailing edge stresses would be reduced.

The results of the analysis of the steel rotor are reported first. Titanium rotor results follow in a skeletonized format.

STEADY-STATE STRESSES

Design criteria for the rotor are given in Table 1. No steady-state blade stress will exceed 95% of the 0.2% yield strength of the material at 122% of design mechanical speed. High cycle fatigue requirements for the blade are a 15,000-psi allowable vibratory stress at resonance with a $K_t = 3.0$ at leading and trailing edges to allow for foreign object damage. Low cycle fatigue requirements for both wheel and blade are for greater than 12,000 start-stop cycles (zero-to-maximum stress) with a reliability of 0.9999. Finally, the wheel burst speed must exceed 130% of design speed.

TABLE 1
Structural design criteria.

Blade

Permanent set	95% 0.2% yield @ 122% speed
Low cycle fatigue	12,000 start-stop cycles
High cycle fatigue	15 ksi allowable vibratory stress at resonance. FOD ($K_t = 3.0$) at leading and trailing edge

Wheel

Wheel burst	130% speed
Low cycle fatigue	12,000 start-stop cycles

Design point stress analyses were performed for the fan rotor at mechanical speeds of 20,223 and 18,950 rpm corresponding to 1.8 and 1.65 pressure ratios, respectively. The analysis is accomplished with finite element computer models which account for centrifugal loads, air loads, temperature effects, airfoil tilts, airfoil untwist, and wheel deflection. Airfoil bending stresses were minimized by tilting the airfoil in the direction of air loads.

Steel Rotor

Airfoil principal stress levels on both suction and pressure surfaces are shown for 1.8 and 1.65 pressure ratio design points in Figures 23 and 24, respectively. The maximum level of 110 ksi occurs near the airfoil hub on both suction and pressure surfaces at the higher pressure ratio and 100 ksi at the lower pressure ratio. The 110 ksi local principal stress on the blade surface compares with an average section stress of 61.5 ksi at design speed. To check the requirement for no damaging permanent set at 122% of design speed, it is

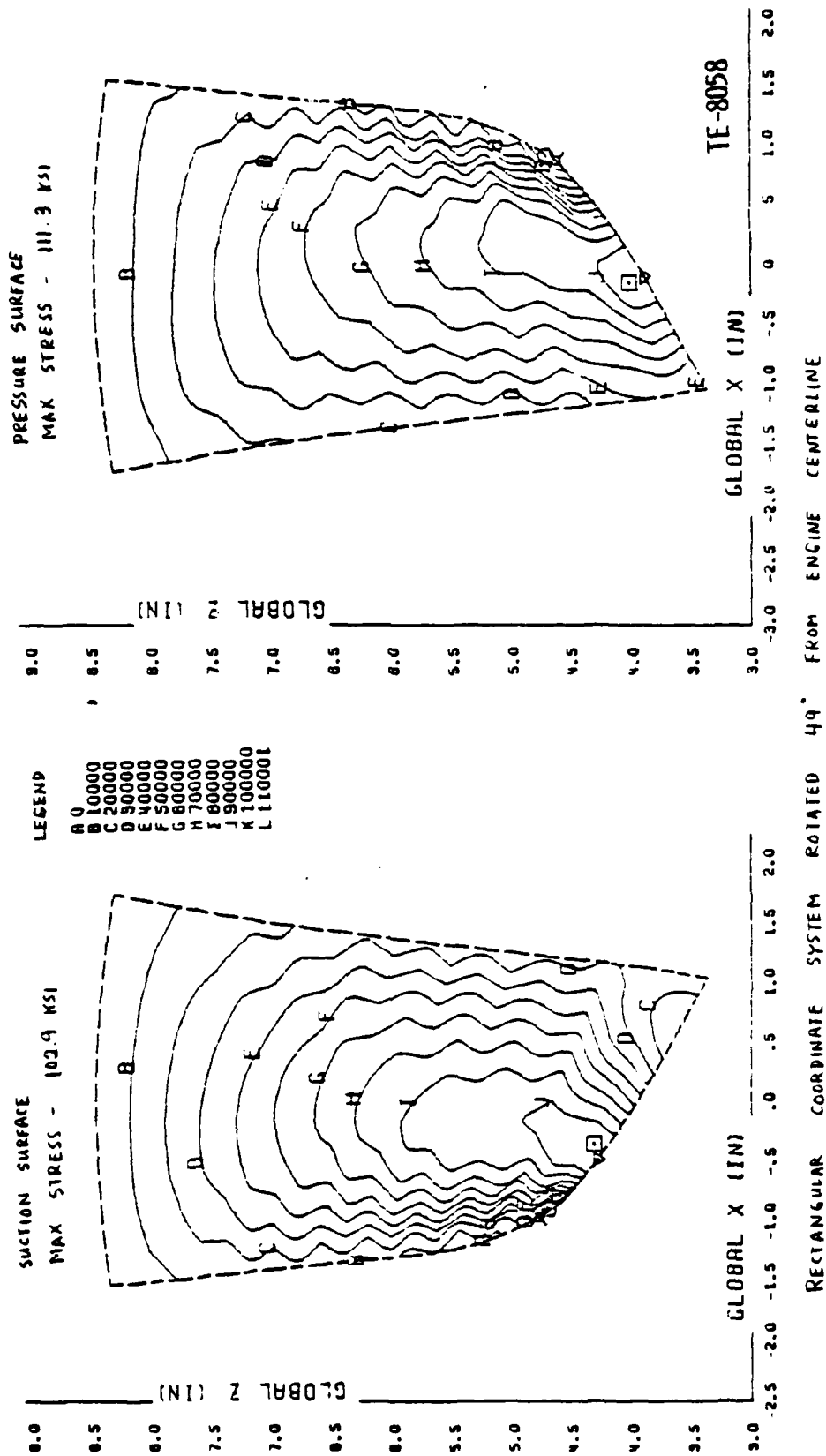
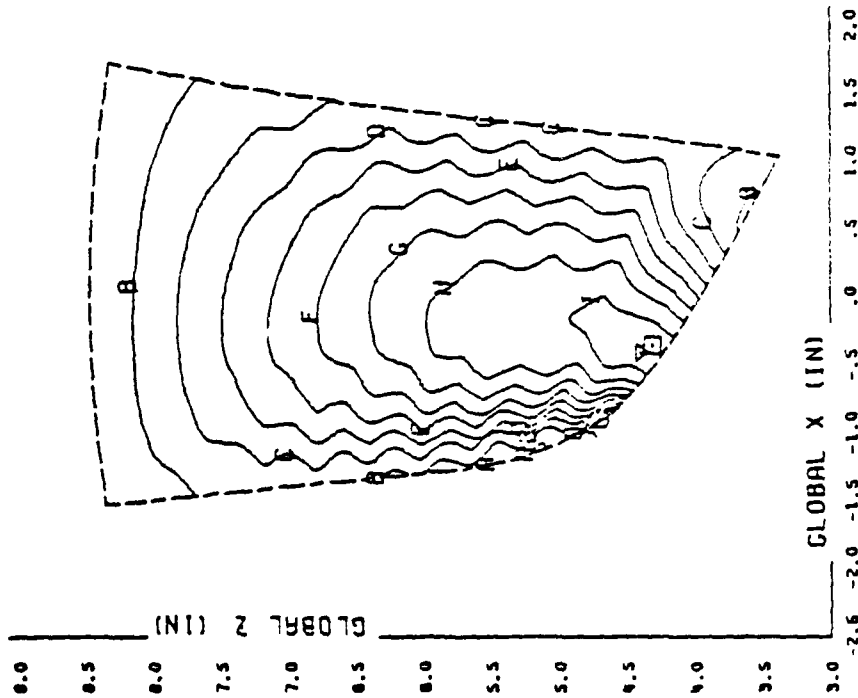
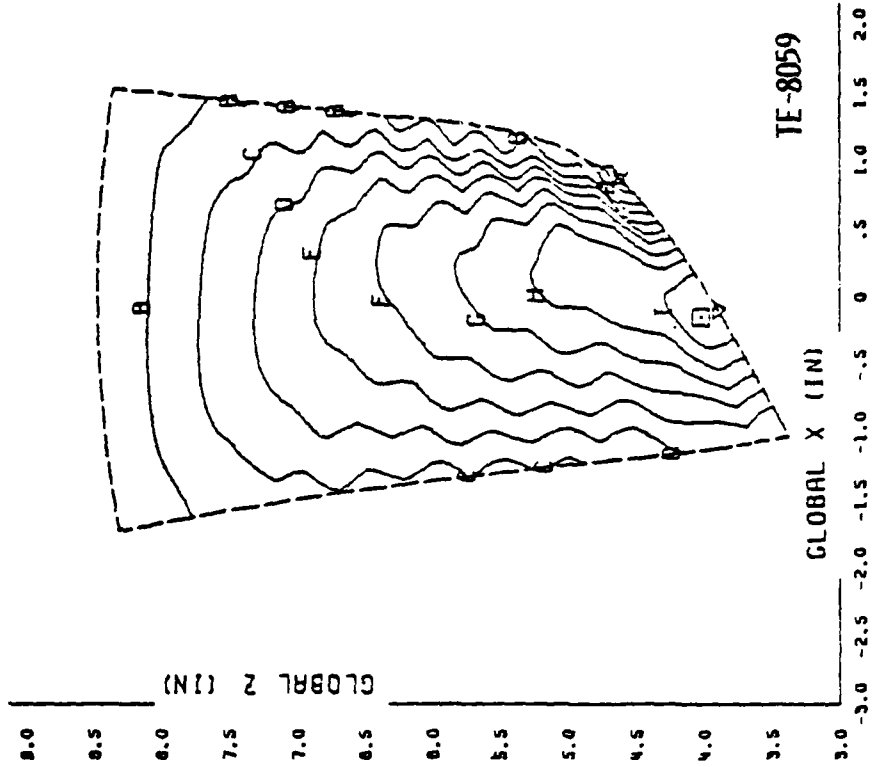


Figure 23. Principal blade stresses at 20,223 rpm.

SUCTION SURFACE
MAX STRESS - 89.7 KSI



PRESSURE SURFACE
MAX STRESS - 101.0 KSI



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RECTANGULAR COORDINATE SYSTEM ROTATED 49° FROM ENGINE CENTERLINE

Figure 24. Principal blade stresses at 18,950 rpm.

TABLE 2
Blade stress summary.

<u>Type of Failure</u>	<u>Criteria</u>	<u>Allowable Stress and Location, ksi</u>	<u>Calculated Section Average Stress</u>	
			<u>1.8 R_c, ksi</u>	<u>1.65 R_c, ksi</u>
Permanent set	95% F _{Ty} @ 122% speed	137 crown	91.5	80.4
Burst	95% F _{Tu} @ 130% speed	151 crown	103.9	91.3
			<u>Calculated Section Max Stress</u>	
			<u>1.8 R_c, ksi</u>	<u>1.65 R_c, ksi</u>
Low cycle fatigue @ 100% speed	12,000 cycles K _t = 3	66 lead edge	46.3	39.2
	K _t = 3	66 trial edge	22.4	19.7
	K _t = 1.4	125 crown	110.0	100.3
High cycle fatigue	+15 ksi vibratory @ resonance	(Refer to Table 3)		
	+5 ksi vibratory @ nonresonance	+5 ksi crown (Required)	+12.0 (allowable)	+14.4 (allowable)

necessary to scale the 61.5 ksi by the square of the speed ratio which gives an average stress level of 91.5 ksi at the overspeed condition. As shown in Table 2, this compares with an allowable stress of 137 ksi. Similarly, for a check of failure at 130% speed, the average stress scales to 104 ksi which compares with an allowable stress of 151 ksi.

Referring to the S-N diagram of Figure 25 at the 12,000-cycle requirement for low cycle fatigue, the airfoil leading and trailing edge allowable stresses are found to be 66 ksi based on a K_t = 3.0. The crown fillet allowable stress is 125 ksi based on a K_t = 1.4. Again, referring to Table 2, the calculated maximum principal stresses are well below these allowables.

The Goodman diagram of Figure 26 indicates that the 110 ksi maximum steady stress at the hub fillet, in conjunction with a fillet radius concentration factor of K_t = 1.4, provides a vibratory allowable stress at that location of ±12.0 ksi. Since this is not a potential resonance condition, a level of ±5 ksi would be considered satisfactory. The vibratory allowable stress at the hub fillet at the lower pressure ratio condition is ±14.4 ksi.

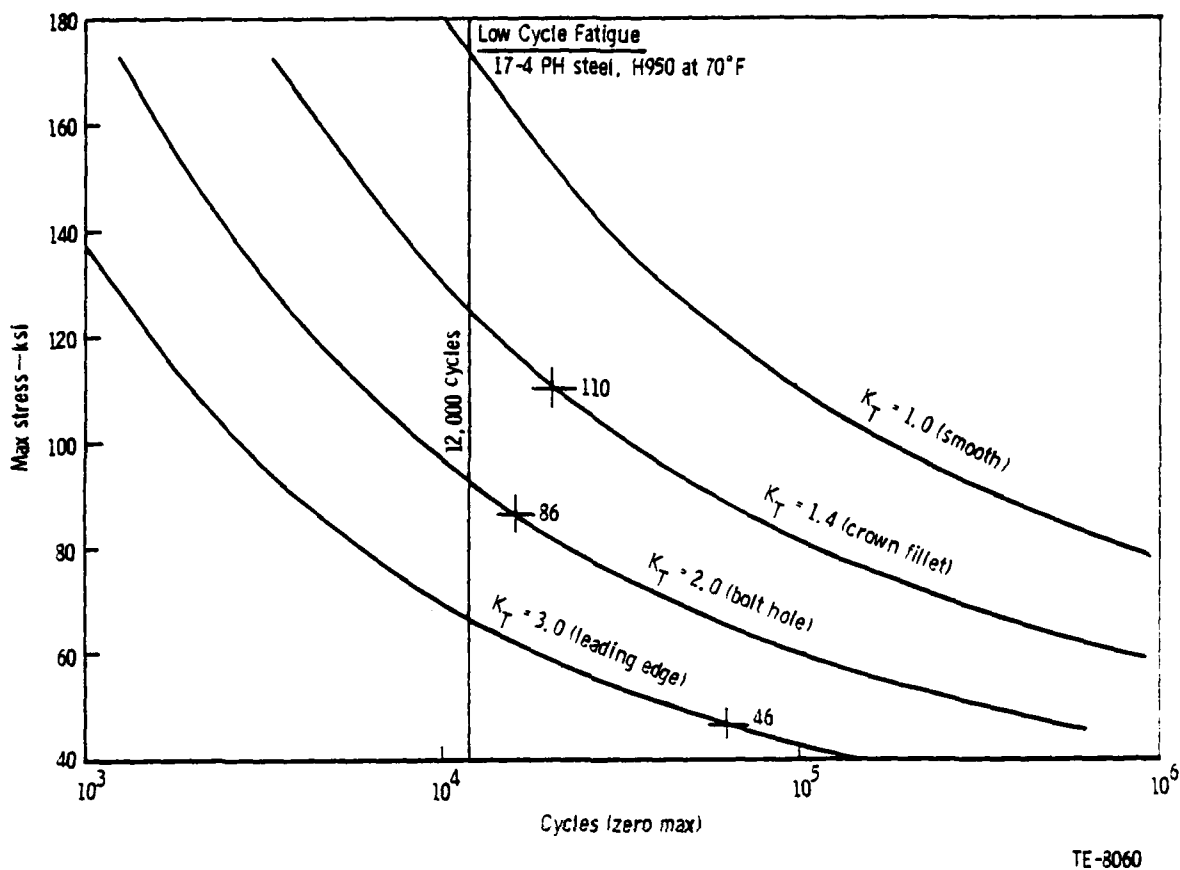
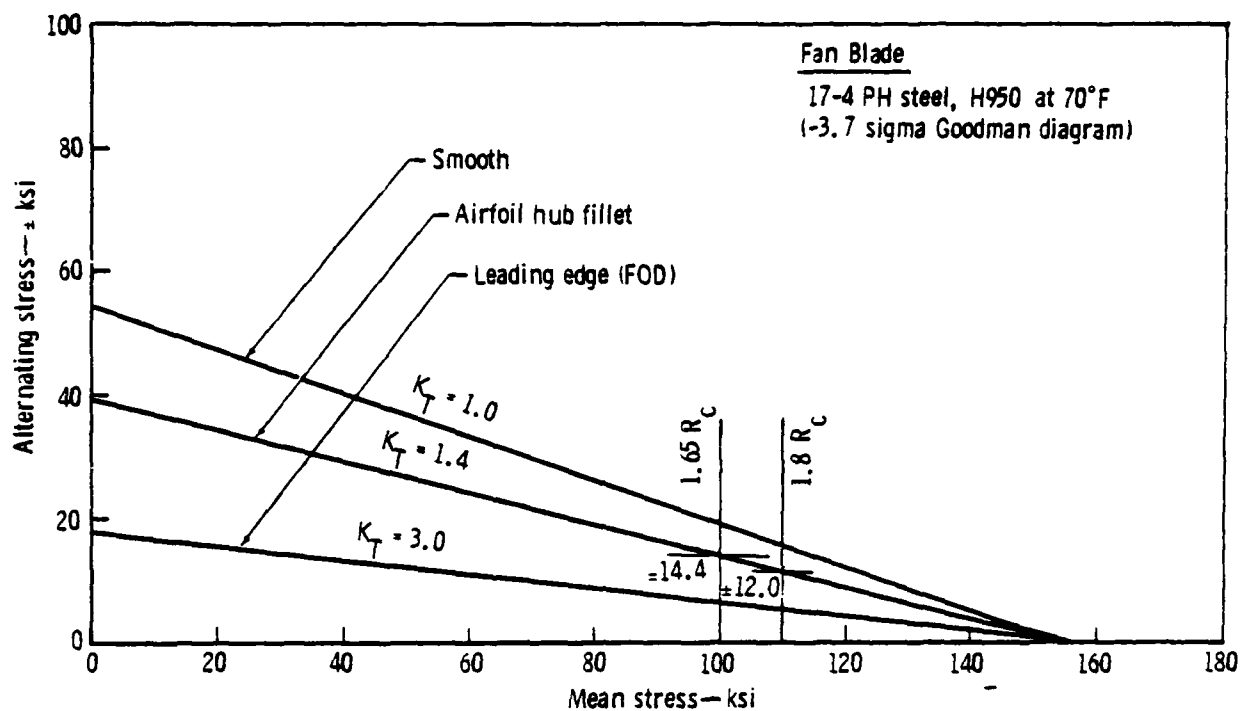


Figure 25. Fan blade and wheel S-N diagram.

Blade vibration analysis, to be discussed more fully later, indicates two potentially troublesome resonances in the operating envelope of the fan. These resonances are a second engine order-first bend mode coincidence at 11,800 rpm and a fourth engine order-first torsional mode coincidence at 18,800 rpm. Allowing a reasonable scatter of individual blade frequencies, the maximum static stresses at the maximum resonance speed and at the critical vibratory stress points are given in Table 3. Entering the Goodman diagram of Figure 26 with those static stresses and the appropriate K_t values, the allowable vibratory stresses shown in Table 3 are defined. All these allowables exceed the goal of ± 15 ksi vibratory allowable. It should also be noted that in the torsional mode, the dynamic stresses at location A and D are 50% and 85%, respectively, of the maximum dynamic stress which occurs at location B.



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Figure 26. Fan blade Goodman diagram.

TABLE 3

Steel blade stress summary at maximum dynamic response.

High cycle fatigue allowables

17-4 PH steel

Cast properties

Pressure

Suction

<u>Mode</u>	<u>Location</u>	<u>Resonance Speed, rpm</u>	<u>Max Static Stress at Resonance, ksi</u>	<u>K_t</u>	<u>Allowable Vibratory Stress, ksi</u>
First bend	A	11,800	35.0	1.4	± 30.4
First torsion	A	18,800	95	1.4	± 15.7
	D		23	3.0	± 15.6
	B		8	3.0	± 17.3

The airfoil, therefore, meets all static and dynamic stress criteria with appropriate K_t factors in cast 17-4 PH material.

Wheel equivalent stresses are shown in Figure 27. Referring to Table 4, a web equivalent stress of 128 ksi at 122% of design speed compared with an allowable yield stress of 137 ksi assures no detrimental permanent set at that condition. Checking wheel burst at 130% of design speed finds a calculated mean hoop stress of 102 ksi and a maximum web radial stress of 127 ksi at that condition which compares with an allowable stress level of 137 ksi. Actual wheel burst is assumed to occur when the mean hoop stress of the wheel reaches 95% of the ultimate strength of the material. The burst speed thus calculated is 163% of design speed or 33,000 rpm. In terms of low cycle fatigue, the calculated values of stress at rim, web, and bore are all well under the allowable stresses taken from the S-N diagram of Figure 25 at 12,000 cycles and the appropriate K_t factors.

TABLE 4
Steel wheel stress summary.

<u>Type of Failure</u>	<u>Criteria</u>	<u>Allowable Stress and Location, ksi</u>	<u>Calculated stress, ksi</u>	
			1.8 R_c	1.65 R_c
Permanent set	95% F_{Ty} @ 122% speed	137 web equiv.	128	112.4
Burst	86% F_{Tu} @ 130% speed	137 mean hoop	102	89.6
		137 web radial	127	111.5
Low cycle fatigue	12,000 cycles @ 100% speed			
	$K_t = 1.4$	125 rim hoop	56	49
	$K_t = 2.0$ (bolt hole)	92.5 web equiv.	86	75.5
	$K_t = 1.0$	174 bore hoop	92.8	81.5
	$K_t = 3.0$	67 balance holes	40	35

Therefore, the wheel also meets all design stress criteria in cast 17-4 PH material. The weight of the wheel and blades is approximately 16.3 lb.

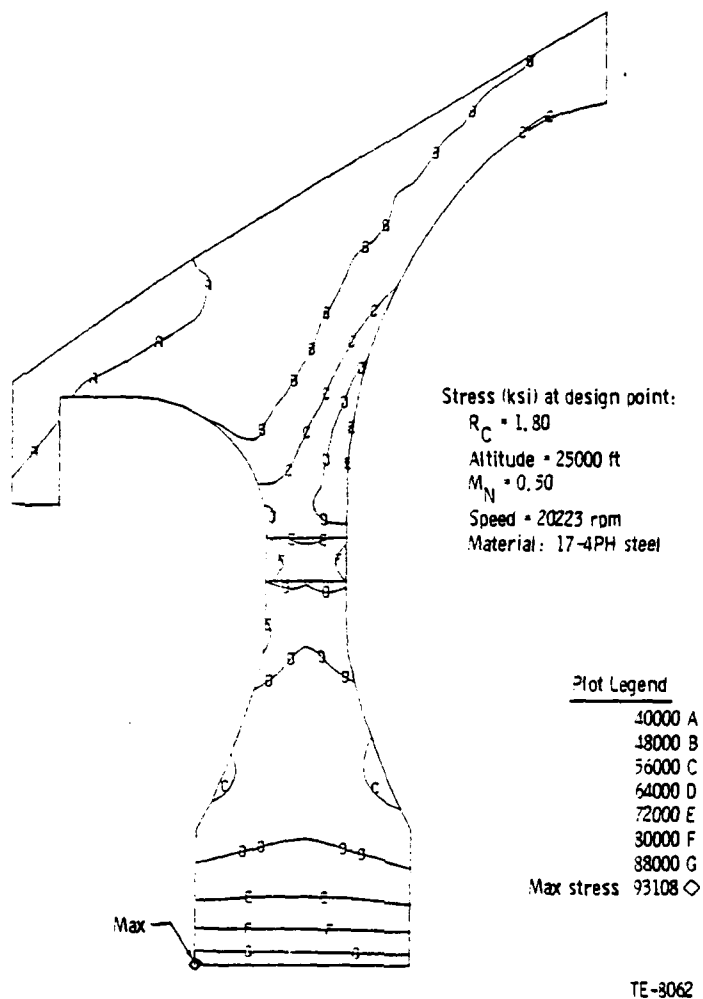


Figure 27. Wheel equivalent stresses.

Titanium Rotor

Blade stresses for the titanium rotor are summarized in Table 5 where allowable stresses for permanent set and burst are comfortably above calculated stresses. In the area of low cycle fatigue however, calculated stresses exceed the allowable at the leading edge and equal the allowable stress at the trailing edge for the 1.8 pressure ratio condition. These stresses are calculated for a rigidly clamped airfoil; including wheel rim flexibility would lower the edge stresses significantly.

TABLE 5
Blade stress summary.

Type of Failure	Criteria	Allowable Stress and Location, ksi	Calculated Section Average Stress	
			1.8 R _c , ksi	1.65 R _c , ksi
Permanent set	95% F _T @ 122% speed	92 crown	57.5	50.5
Burst	95% F _{T_u} @ 130% speed	99 crown	65.3	57.4
			Calculated Section Max Stress	
			1.8 R _c , ksi	1.65 R _c , ksi
Low cycle fatigue @ 100% speed	12,000 cycles K _t = 3	39 lead edge	50	44
	K _t = 3	39 trail edge	39	34.5
	K _t = 1.4	77 crown	53	47
High cycle fatigue	+15 ksi vibratory @ resonance	(Refer to Table 3)		
	+5 ksi vibratory @ nonresonance	+5 ksi crown (required)	10.0 (allowable)	11.1 (allowable)

Referring to Table 6, there are two resonances in the operating envelope of the titanium fan. The second engine order-first bend mode coincidence at 11,500 rpm produces a maximum vibratory response at location A where the static stress is 11.6 KSI. With a fillet radius concentration factor of 1.4 the allowable stress in cast titanium is +17.5 KSI which exceeds the requirement of +15 KSI. Similarly, the fourth engine order-first torsional mode coincidence at 18,800 rpm produces an acceptable allowable dynamic stress of +18.4 KSI.

The titanium wheel stresses are summarized in Table 7 where the allowable stresses are seen to exceed calculated stresses by comfortable margins for all conditions. The calculated burst speed of the titanium wheel is 34,500 rpm or 171% of design speed.

The weight of the titanium wheel and blades is approximately 9.3 lbs, or approximately 7 lbs less than the steel rotor.

TABLE 6

Titanium blade stress summary at maximum dynamic response.

High cycle fatigue allowables

Titanium-6-4

Cast

<u>Mode</u>	<u>Location</u>	<u>Resonance Speed, rpm</u>	<u>Max Static Stress at Resonance, ksi</u>	<u>K_t</u>	<u>Allowable Vibratory Stress, ksi</u>
First bend	A	11,500	11.6	1.4	<u>+17.5</u>
First torsion	B	18,800	7.1	1.4	<u>+18.4</u>

TABLE 7

Titanium wheel stress summary.

<u>Type of Failure</u>	<u>Criteria</u>	<u>Allowable Stress and Location, ksi</u>	<u>Calculated stress, ksi</u>	
			<u>1.8R_c</u>	<u>1.65 R_c</u>
Permanent set	95% F _T @ 122% speed	92 web equiv.	79	69
Burst	86% F _{T_y} ^y F _{T_u} ^u @ 130% speed	89 mean hoop	56.7	49.8
		89 web radial	79	69
Low cycle fatigue	12,000 cycles @ 100% speed			
	K _T = 1.4	77 rim hoop	32	28
	K _T = 2.0	56 web equiv.	53	47
	K _T = 1.0	102 bore hoop	52	46

VIBRATION ANALYSIS

Dynamic analyses of the airfoil, both vibration and flutter, are unaffected by the material change.

Frequencies, mode shapes, and relative dynamic stress distributions for all modes up through the vane passage frequency (42 EO) were calculated, using finite element techniques. The frequency versus speed interference diagram showing the first three modes is shown in Figure 28. The overall interference diagram is presented in Figure 29. Note that the first bending mode (1B) crosses 2 EO at relatively low speed (60%) such that the excitation levels due to inlet distortion will be low.

The relative dynamic stress distributions are determined to locate the maximum dynamic stress location for each mode to assess, in combination with the steady-state stress calculation, the allowable vibratory stress levels. Figures 30 and 31 show the relative radial dynamic stress distributions for the first two modes. These first two modes are given particular emphasis since the excitation force levels produced by the lower four engine orders are us-

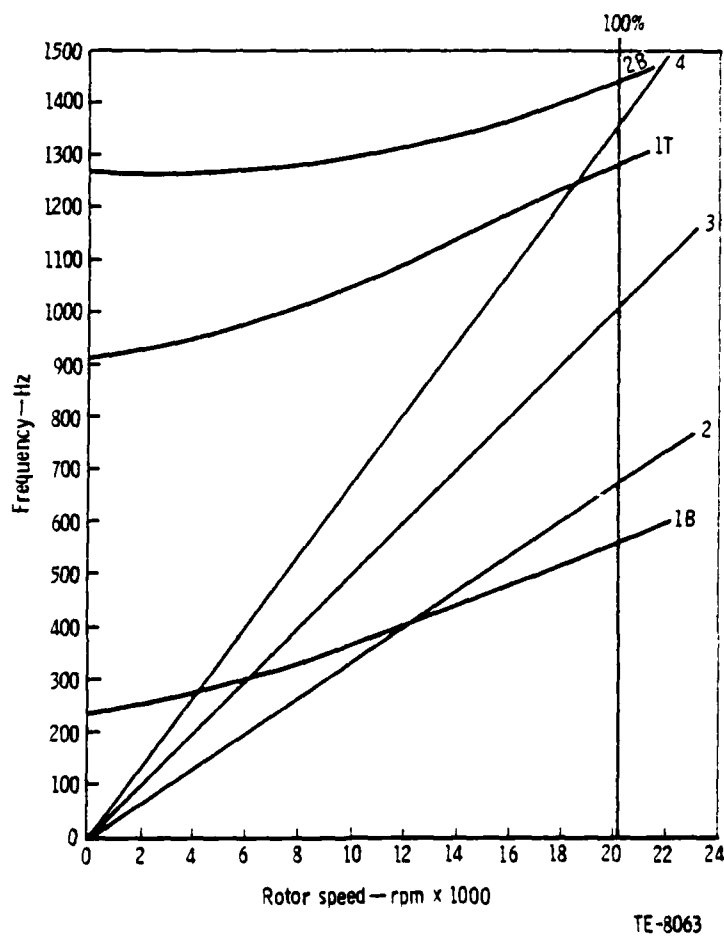


Figure 28. Frequency-speed interference diagram (first 3 modes).

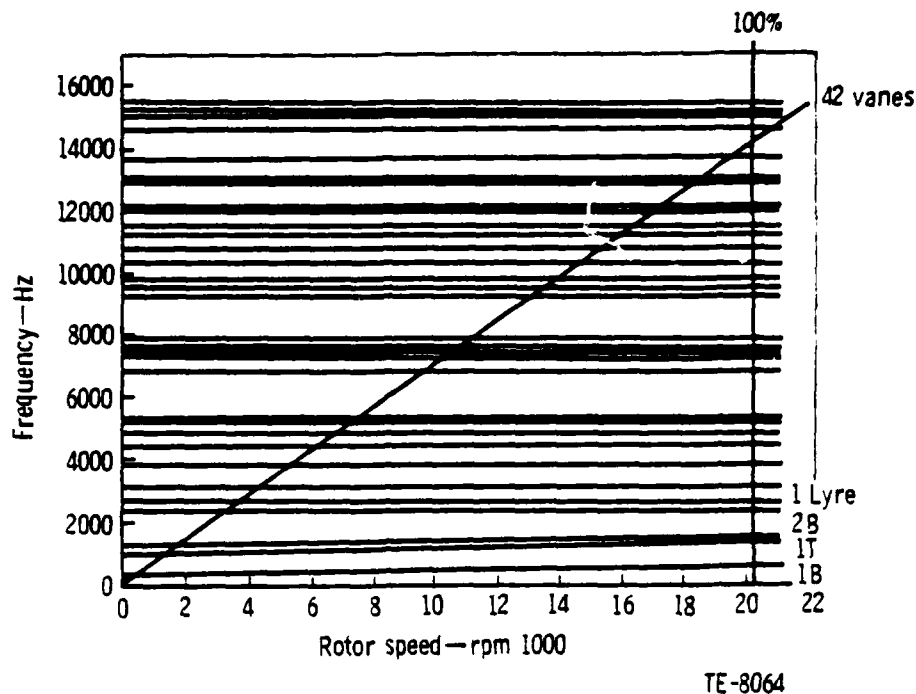


Figure 29. Frequency-speed interference diagram (first 32 modes).

ually higher than those generated by higher harmonics of rotor rotation. As previously discussed, the results of the allowable vibratory stress determination using the Goodman diagram, predicted steady-state stress at coincidence speed, and relative dynamic stress distributions satisfied the ± 15 ksi vibratory stress criteria.

Blade response because of a coincidence of high modes with vane passage (42 EO) are expected to be very low because of the large axial distance (1.5 chord lengths) that the vane row is located aft of the blade row. This large spacing is a result of the noise design criteria.

FLUTTER ANALYSIS

The results of the torsional stall flutter analysis are shown in Figure 32. The predicted margin of safety is an adequate 4 deg of incidence angle above the estimated operating line. The calculated bending stall flutter reduced frequency parameter is well above the 0.25 criterion at 0.302. Similarly, the supersonic unstalled reduced torsional frequency parameter at 0.68 satisfies the 0.60 requirement.

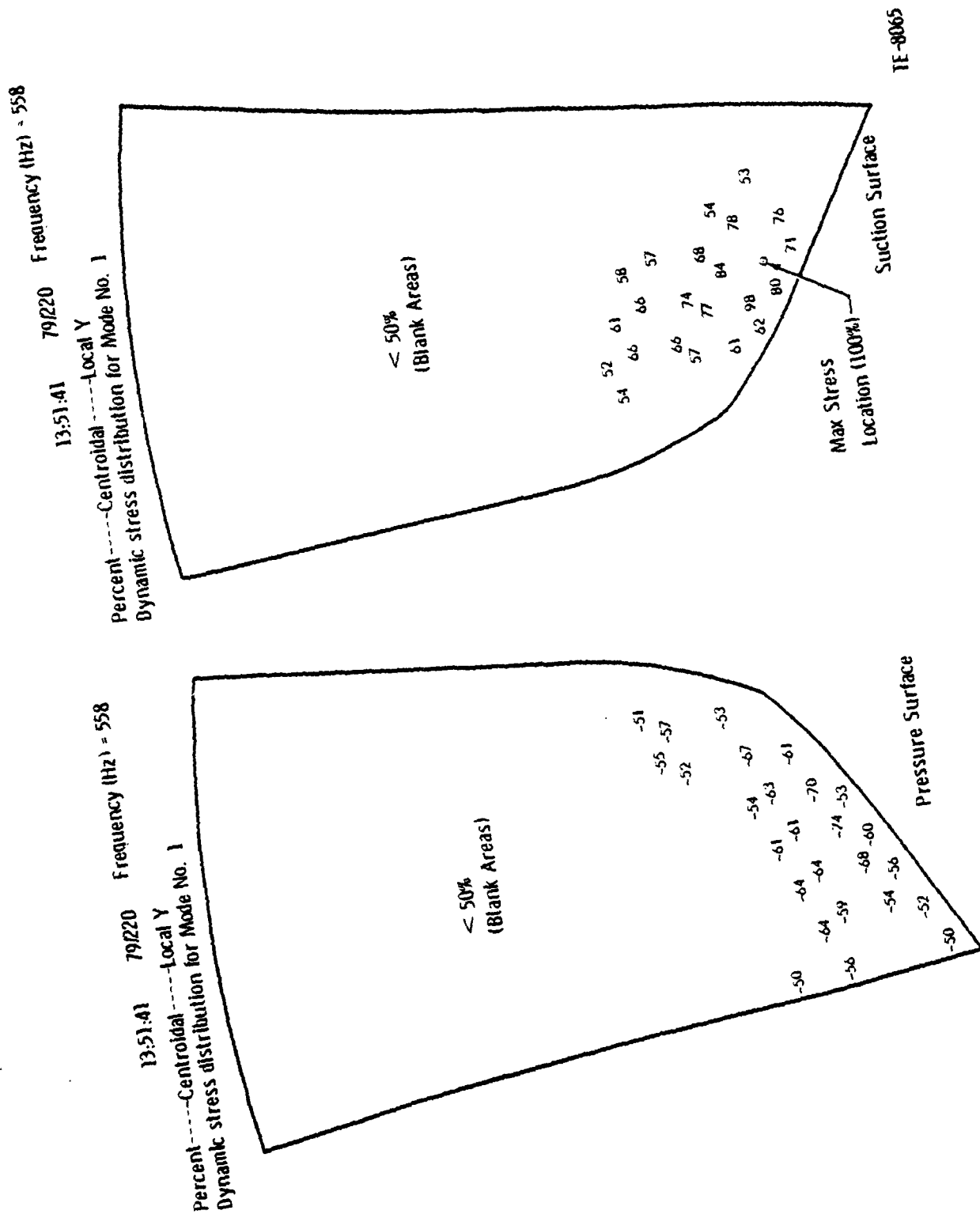


Figure 30. Relative dynamic stress distribution--IB mode at 20,223 rpm.

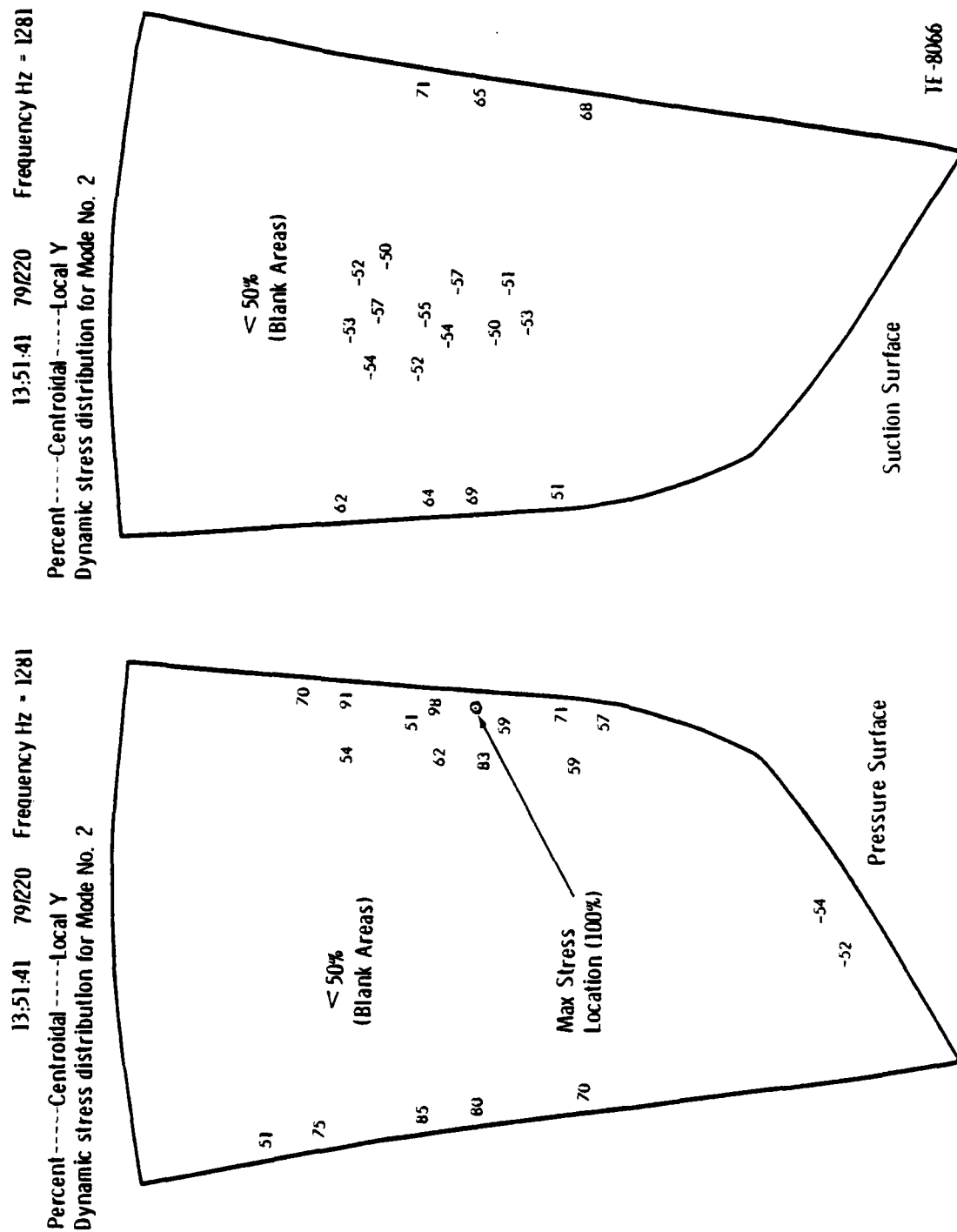
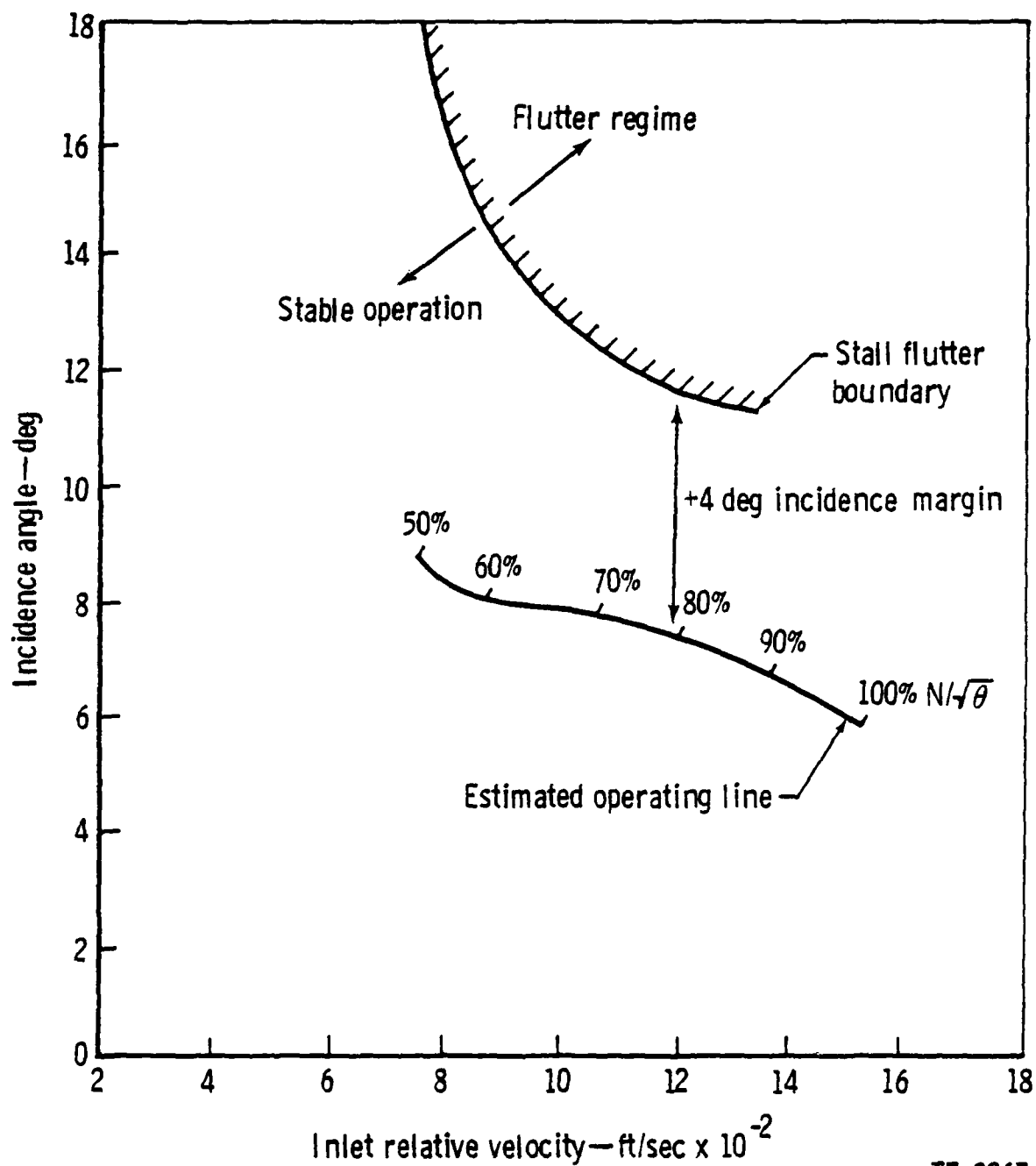


Figure 31. Relative dynamic stress distribution--1F mode at 20,223 rpm.



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Figure 32. Stall torsional flutter analysis.

BIRD INGESTION ANALYSIS

The USAF requirements for bird ingestion are defined in Mil-E-5007D, paragraph 3.2.5.6.1. A summary of the specified bird sizes and engine conditions is given in Table 8. The small high bypass fan and typical trainer aircraft speeds corresponding to the Mil-E-5007D requirements are given in Table 9. In Table 9, the aircraft liftoff, climb, cruise, and descent speeds for the 1.80 pressure ratio fan are assumed to be the same as for the 1.65 pressure ratio fan application.

The annular inlet area of this fan is less than 200 in.² which sets the maximum bird size at two pounds (Ref. Mil-E-5007D par 3.2.5.6.1e). For bird impacts up to 2 lb, no failure shall result which will cause shutdown of the engine although some damage to engine parts may occur.

The failure mode considered here for bird ingestion is local impact damage in the leading edge region of the blade. The calculation of a local damage index, based on a Lycoming criterion approach (Ref. FAA-RD-77-55), has been incorporated into the DDA bird ingestion analysis. This approach relates significant bird slice, impact area, and airfoil parameters to a damage index value that corresponds to critical blade damage. An acceptable damage index level is determined by correlation with actual bird ingestion test data. Engine survivability for new blade designs at the critical ingestion conditions is then predicted with some confidence by use of the damage index calculation.

The shear-penetration damage index is expressed as

$$D_i = 0.273 V_N \sqrt{\pi/4 \left(\frac{e D_B K_B}{h \tau_y} \right)}$$

where:

- V_N = the normal component of impact velocity (ft/sec)
- e = the bird density = 0.045 lb/in.³
- D_B = the bird diameter (in.)
- h = the target mid-thickness (in.)
- τ_y = the target material shear yield (psi)
- K_B = the bird fragmentation parameter

TABLE 8

Mil-E-5007D bird ingestion requirements.

- A. Birds weighing 2 to 4 ounces (a maximum of sixteen at a time) and birds weighing 2 pounds (one at a time) ingested at a bird velocity equal to the take-off flight speed, with the engine at maximum rated speed.
- B. Birds weighing 2 to 4 ounces (a maximum of sixteen at a time) and birds weighing 2 pounds (one at a time) ingested at a bird velocity equal to the cruise flight speed with the engine at maximum continuous speed.
- C. Birds weighing 2 to 4 ounces (a maximum of sixteen at a time) and birds weighing 2 pounds (one at a time) ingested at a bird velocity equal to the descent flight speed with the engine at an associated engine speed.
- D. Birds weighing 4 pounds ingested at a bird velocity based on the most critical flight speed with the engine at maximum rated speed.

Note: Condition D does not apply since the trainer fan inlet is less than 200 in². A maximum of four birds weighing 2 to 4 ounces must be considered for the trainer fan.

TABLE 9
Fan and aircraft speeds for Mil-E-5007D conditions.

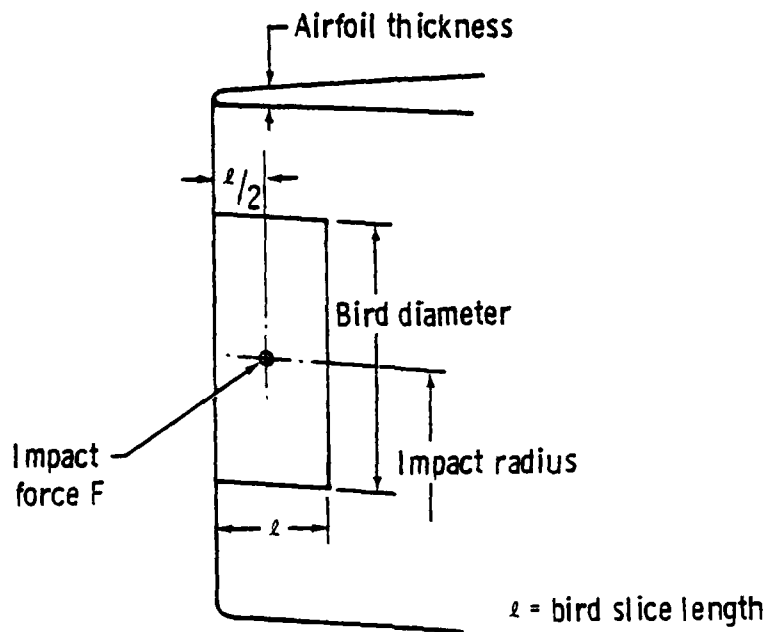
	Pressure ratio <u>1.65</u>	Pressure ratio <u>1.80</u>
<u>Condition A</u>		
Engine take-off fan speed (rpm)	17,303	18,465
Lift-off speed of typical aircraft (kts)	90	90
Climb speed of typical aircraft (kts)	198	198
<u>Condition B</u>		
Engine maximum continuous fan speed (rpm)	17,292	18,454
Cruise speed of typical aircraft (kts)	194	194
<u>Condition C</u>		
Engine descent fan speed (rpm)	10,377	11,074
Descent speed of typical aircraft (kts)	198	198
<u>Condition D</u>		
Engine cruise fan speed (rpm)	10,496	11,201
Cruise speed of typical aircraft (kts)	194	194

The values of D_B and K_B are defined by:

$$D_B = 3.48 (\text{bird weight})^{0.33}$$

$$K_B = \begin{cases} 1.0 \\ 2.40 - 0.0054 V_N \\ 2.5 \end{cases} \quad \text{for} \quad \begin{cases} V_N \leq 260 \text{ ft/sec} \\ 260 < V_N < 400 \\ 400 \leq V_N \end{cases}$$

Figure 33 illustrates the region of leading edge impact.



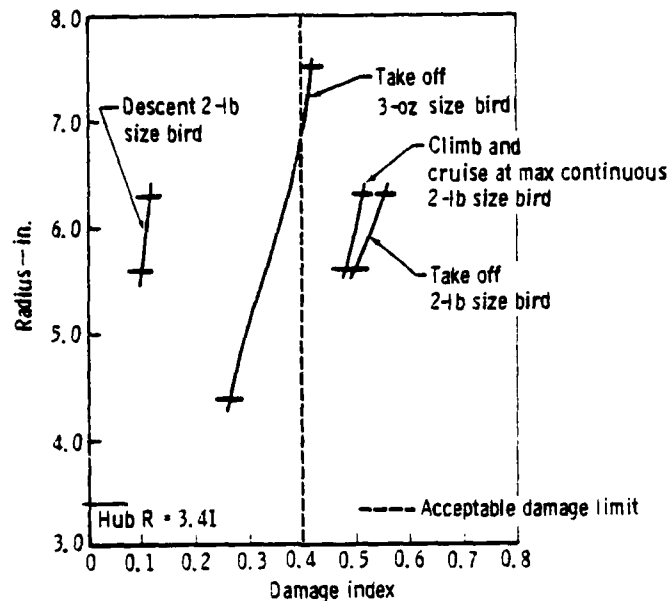
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Figure 33. Bird impact area and blade thickness illustration.

The ballistic-limit velocity for total shear penetration is reached as the damage index approaches 1.0 and theoretically should cause maximum structural damage. In practice, the maximum allowable damage index must be determined by correlation with test.

The calculated damage index along the airfold span for a base-line titanium design at a 1.80 pressure ratio is shown in Figure 34. The maximum damage is predicted to occur at the outermost radial position permitted by the outer case and the 2-lb bird diameter. The maximum allowable damage index for shroudless airfoils is set at 0.40 which limits the expected damage to span-wise tears. A damage index greater than 0.40 could produce loss of airfoil section (for several blades at the 2-lb size) and would cause rotor unbalance that would require engine shut-down. On this basis, the base-line titanium design is judged unacceptable.

Tip R = 8.47



TE-3071

Figure 34. Bird ingestion damage index--titanium blade.

A design study was made to establish the damage index sensitivity to leading edge radius, thickness/chord, number of blades, etc, to identify an acceptable damage index domain for the titanium fan. A titanium airfoil with a 0.025-in. leading edge radius was found to have an acceptable index. However, the performance penalties associated with the required design changes were excessive. Thus, based on the bird ingestion requirements (which are particularly important for trainer engines), a switch to 17-4 PH stainless material was made. The advantage of the material is an increase in shear strength from 66.5 to 106 ksi.

The calculated damage index along the airfoil span for the steel design is shown in Figure 35. Again, the maximum index value of 0.39 is found to occur at the outermost radial position permitted by the outer case for the 2-lb size bird. The steel design (with a 0.0125 leading edge radius) thus satisfies the acceptable damage index limit of 0.40 for the this type of blading.

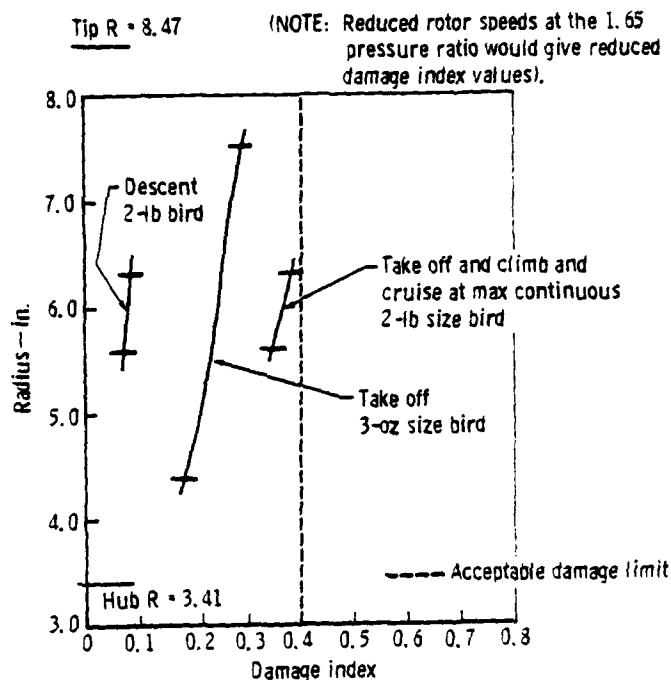


Figure 35. Bird ingestion damage index—steel blade.

Bird ingestion requirements have forced a change in leading edge thickness and material for this fan blade. The penalties are approximately one-half percentage point in stage efficiency and an increase in rotor weight from 9.25 to 16.25 lb. Any degradation in engine response during power transients have not been quantified.

SECTION V
NOISE PREDICTION

Noise goals for the high bypass turbofan powered undergraduate trainer are to be in compliance with Federal Air Regulation Part 36 requirements and maintain ground idle noise substantially below current levels. Engine cycle and fan design data were combined to estimate trainer noise levels for Part 36 and idle conditions. A brief description of the noise prediction methods used and the noise levels estimated for the undergraduate trainer are presented in this section.

Estimated noise levels for a PD 418 powered trainer are presented in Table 10 and show that the above noise goals are met.

TABLE 10
Undergraduate trainer noise levels.

FAR Part 36 Levels (EPNdB)

	<u>Takeoff</u>	<u>Approach</u>
Part 36 Requirements	89	98
PD418 Powered Trainer	78	93

Ground Idle Tone Corrected Perceived Noise Levels

PNdBt at 250-ft Radius (Single Engine)

	<u>Front</u>	<u>Rear</u>
T37 with J69-T-25	118.3	99
PD418 Powered Trainer	79	84

Noise estimates were made through a DDA computer program developed for turbofan noise prediction. A noise generation model for each source, fan, jet, turbine, and combustor, is contained in the program so that the noise output

from each source is dependent on its individual operating conditions and engine total noise reflects the contribution from each noise source.

High bypass ratio turbofan engines are usually fan noise dominated with the jet and combustor being secondary sources. The PD418 engine incorporates the following design features to reduce fan and jet noise generation:

- o Ample space between the fan and the outlet guide vanes (1.5 fan chords)
- o Ratio of outlet guide vanes to fan blades > 2 to cut off blade passing tone
- o Internal mixer to reduce nozzle exit velocity

The noise estimates for the PD418 include the noise reductions provided by these features. The PD418 incorporates a 1.8:1 pressure ratio fan (design point) matched to the GMA500 core and operates at part speed for takeoff, climb, and approach. The engine is fan noise dominated at these conditions so that fan duct acoustic treatment could be used to achieve levels lower than the 78 and 93 EPNdB predicted for takeoff and approach. These levels are 11 and 5 EPNdB below the FAR Part 36 requirements.

At the ground idle condition, the PD418 peak levels at 250-ft radius are expected to be 79 and 84 PNdBt (tone corrected PNdB) in front and rear of the aircraft. In dBA units, front and rear levels are 60 and 67. These levels translate into noise reductions of 34 PNdBt or 35 dBA when the PD418 trainer is compared with the T37B with the J69-T-25 engine.* In predicting the ground idle noise levels, it was assumed that ingested ground level turbulence would prevent cutoff and so the blade passing tone is included in the above levels.

*Speakman, J. D., Power R. G., and Lee, R. A., Community Noise Exposure Resulting from Aircraft Operations, AMRL-TR-73-110, Vol. 4, February 1978.

APPENDIX A

AXIAL COMPRESSOR DESIGN SYSTEM

The vector diagram calculation used for axial compressor design assumes an axisymmetric flow field and obtains a solution of the continuity, energy, and radial equilibrium equations. The design analysis is identified as the Axial Compressor Design System (ACDS) Program BD76. Viscous terms are omitted; however, the equations do account for streamline curvature, radial gradients of total enthalpy and entropy, and blade force terms arising from non-radial blade surfaces. Calculations may be performed at the leading or trailing edges of the airfoils by slanting the calculation stations.

Enthalpy rise across a rotor is given by Euler's turbine equation, and the continuity equation is adjusted for local as well as endwall blockage.

Used as a design tool, the calculation provides detailed examination of the aerothermodynamic solution of the flow process through the compressor. The solution is iterative and must rely on profile loss estimates which are correlated as a function of aerodynamic loading (diffusion factor). This data has been obtained from test data for a wide range of compressor designs and is continually updated.

The equilibrium equation is in the form of:

$$\begin{aligned} \left. \frac{dV_z^2}{dr} \right|_c &= - \left. \frac{d(V_\theta^2)}{dr} \right|_c - \left. \frac{d(V_r^2)}{dr} \right|_c + 2 \left[\left. \frac{dH_o}{dr} \right|_c - T \left. \frac{ds}{dr} \right|_c \right] \\ &+ 2 \left[V_z \left. \frac{dV_r}{dz} \right|_\psi - \frac{V_\theta^2}{r} \right] + 2V_z \left. \frac{dV_z}{dz} \right|_\psi \frac{dz}{dr} \Big|_c + \\ &2V_z \left. \frac{d(rV_\theta)}{dz} \right|_\psi \frac{d\theta}{dr} \Big|_c \end{aligned}$$

where:

r	radial distance
z	axial distance
θ	tangential distance
V_r	radial velocity
V_z	axial velocity
V_θ	tangential velocity
T	total temperature
s	entropy
H_0	total enthalpy
c	projection of the calculating station on relative stress surface
ψ	relative to stream surface

The continuity equation is:

$$W_a = 2\pi \int_{y_h}^{y_t} K_\gamma \rho V_m \sin(\lambda - \epsilon) r dy$$

where:

W_a	airflow
V_m	meridional velocity
K_γ	blockage factor
ρ	density
Y	length along the calculating station
ϵ	angle between tangent to the streamline projected on the meridional plan and axial direction
λ	angle between calculation station and axial

APPENDIX B

DESIGN POINT VECTOR DIAGRAMS

TABLE B-1.

ROTOR INLET

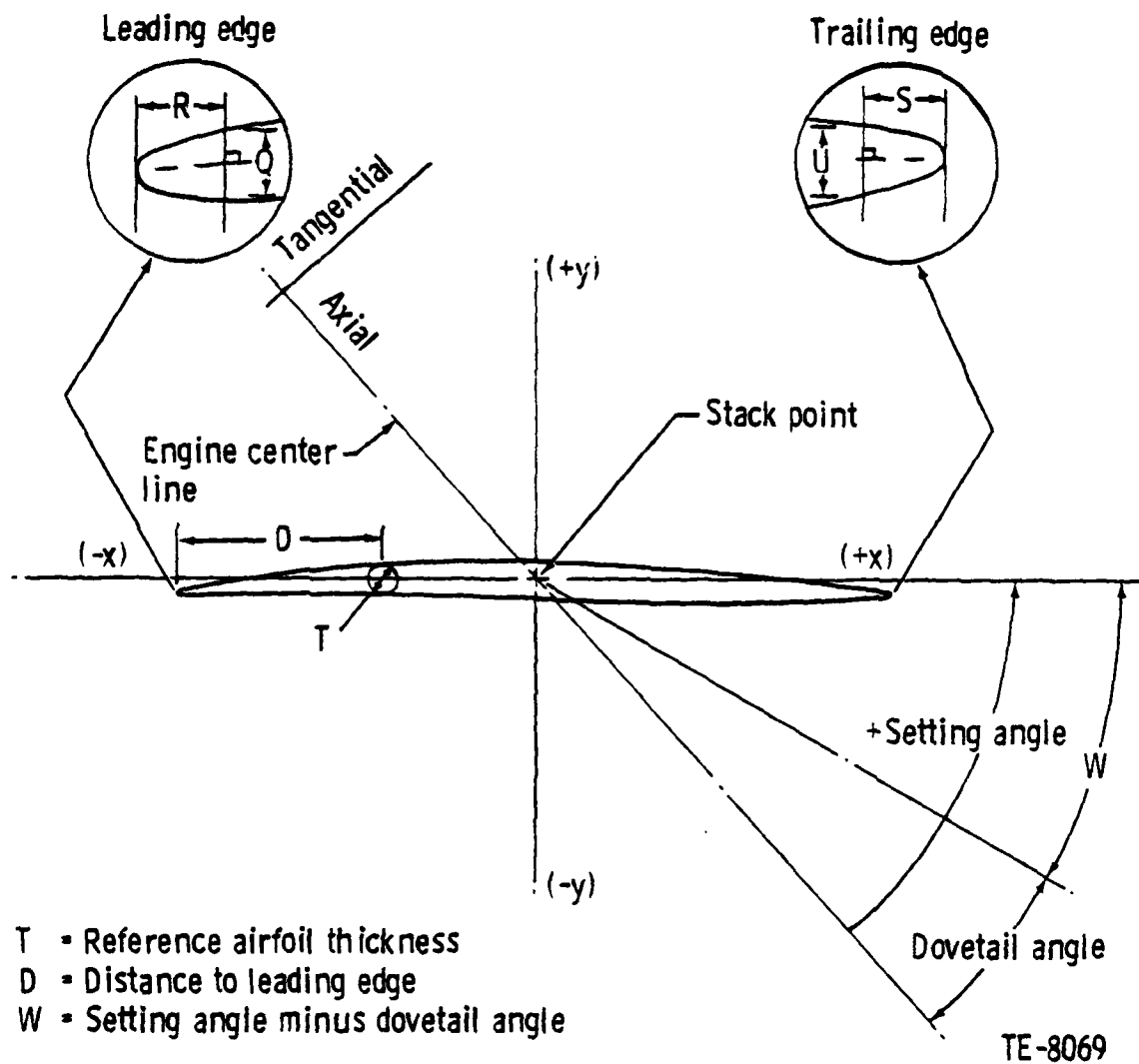
TABLE B-1. (Continued)

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S-L-IND.	DIFFUSION FACTOR	WEGA BAR	DELTA P/100	SURF. DIV.	TOTAL TURNING	ABSOLUTE FLOW RANGE	EQUIVALENT DIFFUSION
1	0.549	1	0.000	1	4	0	1
2	0.539	1	0.000	1	4	0	1
3	0.529	1	0.000	1	4	0	1
4	0.510	1	0.000	1	4	0	1
5	0.504	1	0.000	1	4	0	1
6	0.488	1	0.000	1	4	0	1
7	0.491	1	0.000	1	4	0	1
8	0.491	1	0.000	1	4	0	1
9	0.491	1	0.000	1	4	0	1
10	0.491	1	0.000	1	4	0	1
11	0.491	1	0.000	1	4	0	1
12	0.491	1	0.000	1	4	0	1
13	0.491	1	0.000	1	4	0	1
14	0.491	1	0.000	1	4	0	1
15	0.491	1	0.000	1	4	0	1
16	0.491	1	0.000	1	4	0	1
17	0.491	1	0.000	1	4	0	1
18	0.491	1	0.000	1	4	0	1
19	0.491	1	0.000	1	4	0	1
20	0.491	1	0.000	1	4	0	1
21	0.491	1	0.000	1	4	0	1
22	0.491	1	0.000	1	4	0	1
23	0.491	1	0.000	1	4	0	1
24	0.491	1	0.000	1	4	0	1
25	0.491	1	0.000	1	4	0	1
26	0.491	1	0.000	1	4	0	1
27	0.491	1	0.000	1	4	0	1
28	0.491	1	0.000	1	4	0	1
29	0.491	1	0.000	1	4	0	1
30	0.491	1	0.000	1	4	0	1
31	0.491	1	0.000	1	4	0	1
32	0.491	1	0.000	1	4	0	1
33	0.491	1	0.000	1	4	0	1
34	0.491	1	0.000	1	4	0	1
35	0.491	1	0.000	1	4	0	1
36	0.491	1	0.000	1	4	0	1
37	0.491	1	0.000	1	4	0	1
38	0.491	1	0.000	1	4	0	1
39	0.491	1	0.000	1	4	0	1
40	0.491	1	0.000	1	4	0	1
41	0.491	1	0.000	1	4	0	1
42	0.491	1	0.000	1	4	0	1
43	0.491	1	0.000	1	4	0	1
44	0.491	1	0.000	1	4	0	1
45	0.491	1	0.000	1	4	0	1
46	0.491	1	0.000	1	4	0	1
47	0.491	1	0.000	1	4	0	1
48	0.491	1	0.000	1	4	0	1
49	0.491	1	0.000	1	4	0	1
50	0.491	1	0.000	1	4	0	1
51	0.491	1	0.000	1	4	0	1
52	0.491	1	0.000	1	4	0	1
53	0.491	1	0.000	1	4	0	1
54	0.491	1	0.000	1	4	0	1
55	0.491	1	0.000	1	4	0	1
56	0.491	1	0.000	1	4	0	1
57	0.491	1	0.000	1	4	0	1
58	0.491	1	0.000	1	4	0	1
59	0.491	1	0.000	1	4	0	1
60	0.491	1	0.000	1	4	0	1
61	0.491	1	0.000	1	4	0	1
62	0.491	1	0.000	1	4	0	1
63	0.491	1	0.000	1	4	0	1
64	0.491	1	0.000	1	4	0	1
65	0.491	1	0.000	1	4	0	1

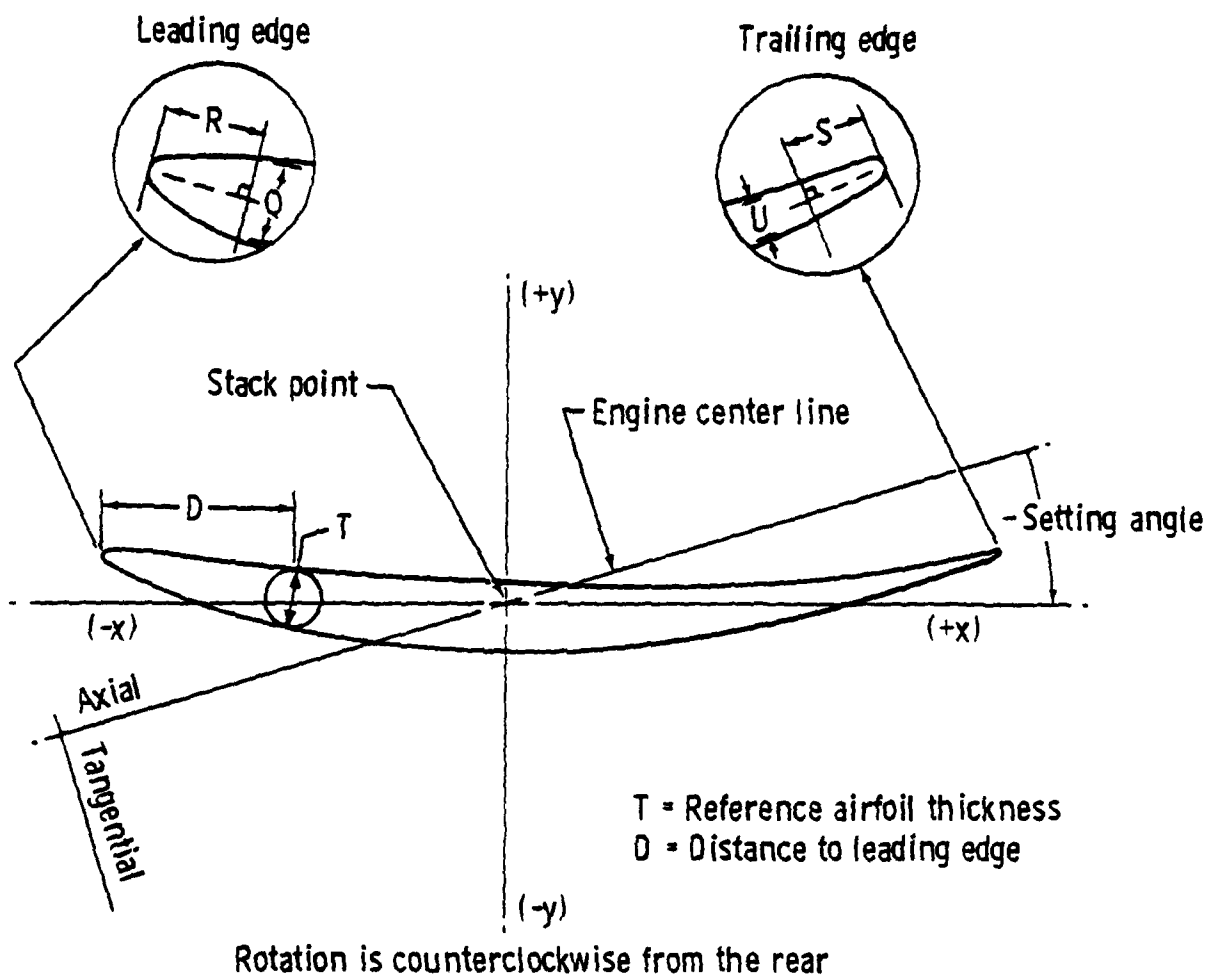
APPENDIX C

ROTOR AND STATOR BLADE COORDINATES



Rotation is counterclockwise from the rear

Figure C-1. Blade manufacturing dimension definitions.



TE-8070

Figure C-2. Vane manufacturing dimension definitions.

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FAN COMPRESSOR BLADE

CBC 12120

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RACII L.E.	T.E.
3.2500	-9.175 -9C 10M 28S	0.2982	1.3000	0.014	0.015

LEADING EDGE AXIAL TANGENT POINT -0.9448

Q DIMENSION	C.0368	U DIMENSION	0.0322
R DIMENSION	C.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES 0.1476 -0.2879
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y	
1	-0.8491*	-0.6670	21	0.4213	0.1936	41	0.4470
2	-0.8472	-0.6594	22	0.5064	0.1547	42	0.3554
3	-0.8133	-0.5831	23	0.5919	0.1073	43	0.2831
4	-0.7769	-0.5081	24	0.6987	0.0266	44	0.1943
5	-0.7276	-0.4167	25	0.8336	-0.0540	45	0.1243
6	-0.6850	-0.3457	26	0.8675	-0.1464	46	0.0556
7	-0.6396	-0.2769	27	0.9703	-0.2901	47	-0.0286
8	-0.5789	-0.1943	28	1.0501	-0.4210	48	-0.0940
9	-0.5271	-0.1314	29	1.1461	-0.6077	49	-0.1579
10	-0.4724	-0.0717	30	1.1643	-0.6482	50	-0.2360
11	-0.3998	-0.0023	31	1.1706*	-0.6684	51	-0.3269
12	-0.3383	0.0486	32	1.1469	-0.6659	52	-0.3713
13	-0.2574	0.1056	33	1.1267	-0.6388	53	-0.4294
14	-0.1893	0.1453	34	1.0250	-0.5158	54	-0.4864
15	-0.1181	0.1789	35	0.9448	-0.4311	55	-0.5560
16	-0.0250	0.2112	36	0.8461	-0.3410	56	-0.6105
17	0.0521	0.2282	37	0.7682	-0.2749	57	-0.6639
18	0.1314	0.2372	38	0.6913	-0.2277	58	-0.7292
19	0.2329	0.2359	39	0.5962	-0.1745	59	-0.7803
20	0.3160	0.2237	40	0.5212	-0.1410	60	-0.8304

* INDICATES EXTREME POINTS

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FAN COMPRESSOR BLADE

CBC 12120

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RACII L.E.	T.E.
3.9500	11.933 11D 55M 58S	0.2241	1.3000	0.013	0.014

LEADING EDGE AXIAL TANGENT POINT -0.9108

Q DIMENSION	C.0337	U DIMENSION	0.0321
R DIMENSION	C.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES 0.0754 -0.0674
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y	
1	-1.0362*	-0.5001	21	0.4402	0.1430	41	0.4084
2	-1.0310	-0.4886	22	0.5133	0.1149	42	0.3243
3	-0.9711	-0.4315	23	0.5840	0.0776	43	0.2567
4	-0.9101	-0.3758	24	0.6683	0.0172	44	0.1712
5	-0.8322	-0.3083	25	0.7317	-0.0429	45	0.1023
6	-0.7685	-0.2561	26	0.7910	-0.1137	46	0.0336
7	-0.7035	-0.2056	27	0.8585	-0.2180	47	-0.0531
8	-0.6419	-0.1641	28	0.9064	-0.3148	48	-0.1228
9	-0.5531	-0.0973	29	0.9577	-0.4526	49	-0.1928
10	-0.4818	-0.0531	30	0.9665	-0.4823	50	-0.2747
11	-0.3929	-0.0018	31	0.9641*	-0.4940	51	-0.3494
12	-0.3205	0.0358	32	0.9520	-0.4559	52	-0.4364
13	-0.2284	0.0780	33	0.9393	-0.4751	53	-0.5059
14	-0.1336	0.1073	34	0.8721	-0.3839	54	-0.5742
15	-0.0780	0.1324	35	0.8144	-0.2482	55	-0.6617
16	0.0178	0.1569	36	0.7423	-0.1248	56	-0.7506
17	0.0954	0.1702	37	0.6816	-0.0206	57	-0.7987
18	0.1731	0.1769	38	0.6194	0.0601	58	-0.8837
19	0.2648	0.1754	39	0.5537	0.1150	59	-0.9515
20	0.3463	0.1661	40	0.4746	0.0931	60	-1.0152

* INDICATES EXTREME POINTS

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FAN COMPRESSOR BLADE

CBC 12120

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
4.6500	29.957 290 57M 255	C.1476	1.300C	C.013 C.014

LEADING EDGE AXIAL TANGENT POINT -0.8742

Q DIMENSION C.0323 U DIMENSION 0.0375
R DIMENSION C.0600 S DIMENSION 0.0600

STACK POINT COORDINATES 0.0
CENTER OF GRAVITY COORDINATES C.0286, -C.0344
COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y	
1	-1.1771*	-C.2956	21	0.4744	0.1136	41	0.4503
2	-1.1685	-C.2827	22	0.5525	0.0970	42	0.3575
3	-1.0928	-C.2472	23	0.6287	0.0747	43	0.2824
4	-1.0167	-C.2127	24	0.7205	0.0382	44	0.1877
5	-0.9210	-C.1713	25	0.7906	0.0017	45	0.1115
6	-0.8441	-C.1395	26	0.8554	-0.0425	46	0.0351
7	-0.7667	-C.1089	27	0.9283	-0.1086	47	-0.0619
8	-0.6695	-C.0726	28	0.9802	-0.1703	48	-0.1403
9	-0.5913	-C.0452	29	1.0359	-0.2386	49	-0.2186
10	-0.5127	-C.0191	30	1.0457	-0.3166	50	-0.3166
11	-0.4140	-C.0112	31	1.0482*	-0.3950	51	-0.3950
12	-0.3347	-C.0337	32	1.0318	-0.4931	52	-0.4931
13	-0.2352	-C.0594	33	1.0191	-0.5716	53	-0.5716
14	-0.1532	-C.0780	34	0.9514	-0.6501	54	-0.6501
15	-0.0731	-C.0947	35	0.8936	-0.7284	55	-0.7284
16	0.0260	-C.1129	36	0.8171	-0.8270	56	-0.8270
17	0.1085	-C.1236	37	0.7529	-0.9057	57	-0.9057
18	0.1910	-C.1299	38	0.6843	-0.9764	58	-0.9764
19	0.2934	-C.1311	39	0.5959	-1.0318	59	-1.0318
20	0.3746	-C.1266	40	0.5236	-0.0370	60	-1.1621

* INDICATES EXTREME POINTS

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FAN COMPRESSOR BLADE

CBC 12120

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
5.4000	43.936 43C 56M 95	0.1180	1.300C	C.013 C.013

LEADING EDGE AXIAL TANGENT POINT -0.8401

Q DIMENSION C.0305 U DIMENSION 0.0356
R DIMENSION C.0600 S DIMENSION 0.0600

STACK POINT COORDINATES 0.0
CENTER OF GRAVITY COORDINATES C.0196, -C.0104
COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y	
1	-1.2862*	-C.1293	21	0.4983	0.0782	41	0.4873
2	-1.2752	-C.1162	22	0.5832	0.0651	42	0.3832
3	-1.1400	-C.0963	23	0.6675	0.0574	43	0.2948
4	-1.1049	-C.0776	24	0.7720	0.0386	44	0.1953
5	-0.9984	-C.0552	25	0.8547	0.0203	45	0.1116
6	-0.9132	-C.0383	26	0.9364	0.0012	46	0.0278
7	-0.8279	-C.0223	27	1.0369	-0.0329	47	-0.0779
8	-0.7213	-C.0036	28	1.1158	-0.0820	48	-0.1627
9	-0.6360	-C.0102	29	1.2121	-0.1031	49	-0.2473
10	-0.5507	-C.0231	30	1.2331	-0.1133	50	-0.3336
11	-0.4440	-C.0378	31	1.2338*	-0.1133	51	-0.4386
12	-0.3587	-C.0463	32	1.2222	-0.1133	52	-0.5444
13	-0.2918	-C.0603	33	1.2024	-0.1133	53	-0.6301
14	-0.1888	-C.0690	34	1.1024	-0.1081	54	-0.7154
15	-0.0808	-C.0763	35	1.0221	-0.0938	55	-0.8227
16	0.0282	-C.0839	36	0.9201	-0.0729	56	-0.9077
17	0.1123	-C.0882	37	0.8334	-0.0604	57	-0.9933
18	0.1984	-C.0902	38	0.7562	-0.0503	58	-1.1003
19	0.3059	-C.0893	39	0.6531	-0.0404	59	-1.1865
20	0.3916	-C.0860	40	0.5703	-0.0345	60	-1.2725

* INDICATES EXTREME POINTS

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FAN COMPRESSOR BLADE

CBC 12120

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RACII L.E.	T.E.
6.4000	54.367 54C 22M 2S	0.0933	1.3000	0.013	0.013

LEADING EDGE AXIAL TANGENT POINT -0.8147

Q DIMENSION	C.3287	U DIMENSION	0.0318
R DIMENSION	C.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0
 CENTER OF GRAVITY COORDINATES C.0127: C.0167
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y
1	-1.4171*	-0.3197	21	0.5151	41	0.5118
2	-1.4048	-0.3071	22	0.6071	42	0.3978
3	-1.3109	-0.3016	23	0.6990	43	0.3066
4	-1.2172	-0.3098	24	0.8138	44	0.1924
5	-1.1004	-0.3194	25	0.9054	45	0.1007
6	-1.0071	-0.3265	26	0.9969	46	0.0090
7	-0.9140	-0.3332	27	1.1112	47	-0.1057
8	-0.7979	-0.3408	28	1.2024	48	-0.1976
9	-0.7052	-0.3463	29	1.3162	49	-0.2896
10	-0.6126	-0.3514	30	1.3389	50	-0.4047
11	-0.4971	-0.3571	31	1.3499*	51	-0.4969
12	-0.4048	-0.3611	32	1.3357	52	-0.6123
13	-0.2896	-0.3655	33	1.3126	53	-0.7049
14	-0.1975	-0.3686	34	1.1977	54	-0.7975
15	-0.1055	-0.3713	35	1.1060	55	-0.9136
16	0.0093	-0.3740	36	0.9414	56	-1.0067
17	0.1012	-0.3758	37	0.8999	57	-1.0999
18	0.1929	-0.3772	38	0.8085	58	-1.2168
19	0.3079	-0.3781	39	0.6943	59	-1.3105
20	0.4000	-0.3773	40	0.6031	60	-1.4045

* INDICATES EXTREME POINTS

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FAN COMPRESSOR BLADE

CBC 12120

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RACII L.E.	T.E.
7.4000	60.445 60D 26M 43S	0.0787	1.3000	0.011	0.011

LEADING EDGE AXIAL TANGENT POINT -0.8169

Q DIMENSION	C.0258	U DIMENSION	0.0286
R DIMENSION	C.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0
 CENTER OF GRAVITY COORDINATES C.0183: C.0467
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y
1	-1.5796*	0.0365	21	0.5475	41	0.5464
2	-1.5677	0.0484	22	0.6490	42	0.4203
3	-1.4635	0.0529	23	0.7306	43	0.3192
4	-1.3596	0.0571	24	0.8777	44	0.1929
5	-1.2303	0.0619	25	0.9795	45	0.0919
6	-1.1273	0.0654	26	1.0814	46	-0.0092
7	-1.0245	0.0686	27	1.2090	47	-0.1355
8	-0.8964	0.0723	28	1.3112	48	-0.2367
9	-0.7942	0.0750	29	1.4393	49	-0.3380
10	-0.6923	0.0774	30	1.5649	50	-0.4647
11	-0.5832	0.0801	31	1.6355*	51	-0.5862
12	-0.4636	0.0820	32	1.6331	52	-0.6934
13	-0.3389	0.0842	33	1.4373	53	-0.7934
14	-0.2137	0.0857	34	1.3086	54	-0.8975
15	-0.0833	0.0872	35	1.0061	55	-1.0256
16	0.0926	0.0888	36	1.0184	56	-1.1283
17	0.1936	0.0900	37	0.9766	57	-1.2313
18	0.2936	0.0912	38	0.8751	58	-1.3605
19	0.3936	0.0926	39	0.7485	59	-1.4642
20	0.4208	0.0936	40	0.6474	60	-1.5683

* INDICATES EXTREME POINTS

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FAN COMPRESSOR BLADE

CBC 12120

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
8.2500	64.569 64C 34M 7S	0.0724	1.3000	0.010 0.010

LEADING EDGE AXIAL TANGENT POINT -0.8253

Q DIMENSION	0.0231	U DIMENSION	0.0264
R DIMENSION	0.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES 0.0310 0.0733
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y
1	-1.7497*	0.0754	21	0.5793	41	0.5790
2	-1.7387	0.0859	22	0.6908	42	0.4401
3	-1.6245	0.0883	23	0.8025	43	0.3290
4	-1.5107	0.0905	24	0.9424	44	0.1903
5	-1.3490	0.0931	25	1.0545	45	0.0794
6	-1.2561	0.0949	26	1.1669	46	-0.0315
7	-1.1435	0.0966	27	1.3076	47	-0.1702
8	-1.0032	0.0985	28	1.4205	48	-0.2811
9	-0.8913	0.0998	29	1.5619	49	-0.3921
10	-0.7797	0.1011	30	1.5903	50	-0.5311
11	-0.6404	0.1026	31	1.5998*	51	-0.6424
12	-0.5292	0.1037	32	1.5889	52	-0.7817
13	-0.3904	0.1051	33	1.5604	53	-0.8934
14	-0.2795	0.1062	34	1.4183	54	-1.0053
15	-0.1686	0.1073	35	1.3051	55	-1.1455
16	-0.0302	0.1089	36	1.1644	56	-1.2580
17	0.0803	0.1102	37	1.0522	57	-1.3708
18	0.1912	0.1117	38	0.9404	58	-1.5122
19	0.3297	0.1138	39	0.8011	59	-1.6258
20	0.4406	0.1158	40	0.6899	60	-1.7398

* INDICATES EXTREME POINTS

23 AUG 79

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FAN COMPRESSOR BLADE

CBC 12120

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
8.7500	66.695 66D 41M 41S	0.0698	1.3000	0.010 0.009

LEADING EDGE AXIAL TANGENT POINT -0.8293

Q DIMENSION	0.0216	U DIMENSION	0.0253
R DIMENSION	0.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES 0.0421 0.1897
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y
1	-1.8537*	0.0981	21	0.5981	41	0.5983
2	-1.8435	0.1079	22	0.7158	42	0.4513
3	-1.7232	0.1091	23	0.8338	43	0.3344
4	-1.6033	0.1103	24	0.9816	44	0.1881
5	-1.4544	0.1116	25	1.1001	45	0.0711
6	-1.3355	0.1126	26	1.2188	46	-0.0458
7	-1.2170	0.1134	27	1.3676	47	-0.1919
8	-1.0893	0.1144	28	1.4870	48	-0.3089
9	-0.9515	0.1152	29	1.6367	49	-0.4258
10	-0.8339	0.1159	30	1.6667	50	-0.5722
11	-0.6873	0.1169	31	1.6757*	51	-0.6844
12	-0.5701	0.1177	32	1.6658	52	-0.8362
13	-0.4239	0.1187	33	1.6355	53	-0.9538
14	-0.3071	0.1197	34	1.4850	54	-1.0716
15	-0.1904	0.1208	35	1.3653	55	-1.2152
16	-0.0445	0.1224	36	1.2165	56	-1.3376
17	0.0722	0.1237	37	1.0980	57	-1.4563
18	0.1896	0.1248	38	0.9799	58	-1.6051
19	0.3349	0.1262	39	0.8328	59	-1.7247
20	0.4518	0.1267	40	0.7154	60	-1.8446

* INDICATES EXTREME POINTS

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FAN COMPRESSOR VANE

CSC 12122

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
4.5009	-24.741 -240 59M 26S	0.0696	0.7450	0.008 0.006

LEADING EDGE AXIAL TANGENT POINT -0.0227

U DIMENSION	0.0242	U DIMENSION	0.0213
R DIMENSION	0.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES -0.1199; -0.0516
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y	
1	-0.7152*	0.0645	21	0.1320	-0.0459	41	0.1474
2	-0.7059	0.0692	22	0.1704	-0.0396	42	0.0963
3	-0.6850	0.0573	23	0.2085	-0.0318	43	0.0547
4	-0.6251	0.0459	24	0.2553	-0.0199	44	0.0020
5	-0.5730	0.0325	25	0.3223	-0.0036	45	-0.0406
6	-0.5322	0.0223	26	0.3286	0.0043	46	-0.0839
7	-0.4913	0.0147	27	0.3731	0.0223	47	-0.1366
8	-0.4402	0.0015	28	0.4079	0.0389	48	-0.1787
9	-0.3904	0.0008	29	0.4503	0.0616	49	-0.2208
10	-0.3525	0.0146	30	0.4586	0.0664	50	-0.2739
11	-0.3076	0.0235	31	0.4681*	0.0616	51	-0.3153
12	-0.2688	0.0300	32	0.4664	0.0580	52	-0.3675
13	-0.2159	0.0376	33	0.4572	0.0514	53	-0.4091
14	-0.1752	0.0427	34	0.4419	0.0401	54	-0.4504
15	-0.1345	0.0476	35	0.4387	-0.0303	55	-0.5016
16	-0.0942	0.0523	36	0.4453	-0.0394	56	-0.5422
17	-0.0545	0.0569	37	0.45101	-0.0434	57	-0.5825
18	-0.0149	0.0556	38	0.45736	-0.0464	58	-0.6323
19	0.0243	0.0542	39	0.46263	-0.0339	59	-0.6717
20	0.0635	0.0515	40	0.4873	-0.0064	60	-0.7105

* INDICATES EXTREME POINTS

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FAN COMPRESSOR VANE

CSC 12122

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
4.8500	-23.613 -230 37M 3S	0.0753	0.7450	0.008 0.005

LEADING EDGE AXIAL TANGENT POINT -0.5719

U DIMENSION	0.0248	U DIMENSION	0.0221
R DIMENSION	0.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES -0.0469; -0.0171
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y	
1	-0.6658	0.0969	21	0.2119	-0.0092	41	0.2268
2	-0.6564	0.1038	22	0.2520	-0.0032	42	0.1733
3	-0.6141	0.0923	23	0.2913	0.0044	43	0.1302
4	-0.5717	0.0813	24	0.3410	0.0161	44	0.0753
5	-0.5189	0.0683	25	0.3799	0.0272	45	0.0311
6	-0.4765	0.0585	26	0.4183	0.0398	46	-0.0134
7	-0.4342	0.0492	27	0.4654	0.0578	47	-0.0683
8	-0.3814	0.0383	28	0.5025	0.0739	48	-0.1119
9	-0.3392	0.0302	29	0.5378	0.0963	49	-0.1555
10	-0.2969	0.0226	30	0.5567	0.1010	50	-0.2098
11	-0.2442	0.0138	31	0.5664	0.0963	51	-0.2531
12	-0.2020	0.0074	32	0.5623	0.0923	52	-0.3070
13	-0.1493	0.0001	33	0.5547	0.0856	53	-0.3499
14	-0.1071	0.0052	34	0.5411	0.0538	54	-0.3925
15	-0.0650	0.0099	35	0.4799	0.0304	55	-0.4454
16	-0.0129	0.0149	36	0.4352	0.0037	56	-0.4873
17	0.0343	0.0173	37	0.3979	-0.0155	57	-0.5289
18	0.0693	0.0182	38	0.3593	-0.0327	58	-0.5803
19	0.1203	0.0172	39	0.3096	-0.0514	59	-0.6209
20	0.1713	0.0146	40	0.2686	-0.0640	60	-0.6611

* INDICATES EXTREME POINTS

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FAN COMPRESSOR VANE

CSC 12122

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
5.1500	-22.567 -220 33M 34S	0.0800	0.7450	0.008 0.005

LEADING EDGE AXIAL TANGENT POINT -0.5277

Q DIMENSION	0.0254	U DIMENSION	0.0225
R DIMENSION	0.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES 0.0169 0.0099
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y
1	-0.6223*	0.1242	41	0.4417	0.0198	0.2963
2	-0.6128	0.1311	42	0.4233	0.0257	-0.0491
3	-0.5993	0.1393	43	0.4047	0.0331	-0.0392
4	-0.5827	0.1490	44	0.3859	0.0416	-0.03647
5	-0.5673	0.1596	45	0.3665	0.0513	-0.03884
6	-0.5527	0.1713	46	0.3465	0.0619	-0.04688
7	-0.5383	0.1838	47	0.3259	0.0737	-0.05617
8	-0.5240	0.1970	48	0.3048	0.0865	-0.06662
9	-0.5097	0.2109	49	0.2832	0.1001	-0.07822
10	-0.4954	0.2254	50	0.2611	0.1146	-0.09097
11	-0.4811	0.2405	51	0.2385	0.1300	-0.10477
12	-0.4668	0.2561	52	0.2154	0.1463	-0.11962
13	-0.4525	0.2722	53	0.1918	0.1635	-0.13552
14	-0.4382	0.2888	54	0.1677	0.1816	-0.15247
15	-0.4239	0.3059	55	0.1431	0.2006	-0.16947
16	-0.4096	0.3234	56	0.1180	0.2204	-0.18652
17	-0.3953	0.3413	57	0.0924	0.2410	-0.20362
18	-0.3810	0.3596	58	0.0663	0.2624	-0.22077
19	-0.3667	0.3782	59	0.0397	0.2846	-0.23797
20	-0.3524	0.3971	60	0.0135	0.3076	0.1169

* INDICATES EXTREME POINTS

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FAN COMPRESSOR VANE

CSC 12122

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
5.5000	-21.834 -210 33M 35S	0.0855	0.7457	0.008 0.006

LEADING EDGE AXIAL TANGENT POINT -0.4099

Q DIMENSION	0.0257	U DIMENSION	0.0230
R DIMENSION	0.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES 0.0335 0.0403
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y
2	-0.5853*	0.1566	41	0.3807	0.03517	0.3792
3	-0.5708	0.1639	42	0.3609	0.04276	-0.0216
4	-0.5563	0.1720	43	0.3404	0.05051	-0.0319
5	-0.5418	0.1807	44	0.3191	0.05831	-0.0413
6	-0.5273	0.1899	45	0.2971	0.06616	-0.0507
7	-0.5128	0.1996	46	0.2744	0.07406	-0.0601
8	-0.4983	0.2098	47	0.2511	0.08201	-0.0695
9	-0.4838	0.2204	48	0.2272	0.09001	-0.0789
10	-0.4693	0.2315	49	0.2027	0.09806	-0.0883
11	-0.4548	0.2430	50	0.1776	0.10616	-0.0977
12	-0.4403	0.2549	51	0.1519	0.11431	-0.1071
13	-0.4258	0.2672	52	0.1256	0.12251	-0.1165
14	-0.4113	0.2799	53	0.0987	0.13076	-0.1259
15	-0.3968	0.2930	54	0.0713	0.13906	-0.1353
16	-0.3823	0.3065	55	0.0434	0.14741	-0.1447
17	-0.3678	0.3204	56	0.0149	0.15581	-0.1541
18	-0.3533	0.3347	57	0.0000	0.16426	-0.1635
19	-0.3388	0.3494	58	0.0000	0.17276	-0.1729
20	-0.3243	0.3645	59	0.0000	0.18131	-0.1823
21	-0.3098	0.3799	60	0.0000	0.18991	-0.1917

* INDICATES EXTREME POINTS

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FAN COMPRESSOR VANE

CSC 12122

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
6.0000	-21.296 -21D 17M 40S	0.0933	0.7450	0.008 0.006

LEADING EDGE AXIAL TANGENT POINT -0.3821

U DIMENSION	0.0264	U DIMENSION	0.0438
S DIMENSION	0.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES 0.2050 0.0819
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y	
1	-0.4874*	0.1998	41	0.4857	0.0945	41	0.5000
2	-0.4779	0.2070	42	0.5310	0.1002	42	0.4397
3	-0.4309	0.1967	43	0.5761	0.1075	43	0.3908
4	-0.3840	0.1867	44	0.6322	0.1189	44	0.3292
5	-0.3254	0.1749	45	0.6768	0.1299	45	0.2797
6	-0.2755	0.1653	46	0.7210	0.1425	46	0.2302
7	-0.2317	0.1571	47	0.7758	0.1605	47	0.1893
8	-0.1732	0.1468	48	0.8192	0.1768	48	0.1211
9	-0.1265	0.1390	49	0.8833	0.1994	49	0.0731
10	-0.0798	0.1310	50	0.9417*	0.2043	50	0.0123
11	-0.0214	0.1223	51	0.9892	0.1993	51	0.0343
12	0.0294	0.1163	52	0.9392	0.1945	52	0.0935
13	0.0834	0.1086	53	0.8799	0.1872	53	0.1406
14	0.1305	0.1029	54	0.8318	0.1528	54	0.1874
15	0.1772	0.0970	55	0.7915	0.1274	55	0.2454
16	0.2230	0.0917	56	0.7393	0.0986	56	0.2914
17	0.2608	0.0886	57	0.6960	0.0779	57	0.3171
18	0.3262	0.0871	58	0.6515	0.0594	58	0.3393
19	0.3833	0.0875	59	0.5944	0.0394	59	0.4382
20	0.4288	0.0890	60	0.5476	0.0261	60	0.4624

* INDICATES EXTREME POINTS

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FAN COMPRESSOR VANE

CSC 12122

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
6.5007	-20.917 -20D 54M 37S	0.1012	0.7450	0.008 0.004

LEADING EDGE AXIAL TANGENT POINT -0.2398

U DIMENSION	0.0270	U DIMENSION	0.0246
S DIMENSION	0.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES 0.3480 0.1230
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y	
1	-0.4023*	0.2438	41	0.4079	0.1372	41	0.6225
2	-0.3932	0.2508	42	0.4581	0.1428	42	0.5595
3	-0.3443	0.2409	43	0.5021	0.1501	43	0.5085
4	-0.2953	0.2313	44	0.5607	0.1615	44	0.4443
5	-0.2345	0.2198	45	0.6072	0.1726	45	0.3927
6	-0.1858	0.2110	46	0.6535	0.1853	46	0.3412
7	-0.1374	0.2025	47	0.6910	0.2036	47	0.2779
8	-0.0764	0.1927	48	0.7264	0.2231	48	0.2290
9	-0.0274	0.1847	49	0.7616	0.2431	49	0.1782
10	0.0207	0.1773	50	0.7938	0.2641	50	0.1162
11	0.0614	0.1685	51	0.8324*	0.2830	51	0.0869
12	0.1049	0.1618	52	0.8698	0.2979	52	0.0055
13	0.1508	0.1539	53	0.9058	0.3104	53	0.0432
14	0.1989	0.1474	54	0.9391	0.3194	54	0.0917
15	0.2476	0.1423	55	0.9706	0.3161	55	0.1519
16	0.2964	0.1354	56	0.9927	0.3116	56	0.2196
17	0.3452	0.1266	57	1.0127*	0.3057	57	0.2869
18	0.3942	0.1150	58	1.0306	0.2974	58	0.3555
19	0.4432	0.1007	59	1.0471	0.2877	59	0.4213
20	0.4923	0.0829	60	1.0622	0.2629	60	0.4977

* INDICATES EXTREME POINTS

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FAN COMPRESSOR VANE

CSC 12122

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
7.0000	-20.611 -200 36M 375	0.1090	0.7450	0.008 0.006

LEADING EDGE AXIAL TANGENT POINT -0.1943

J DIMENSION	0.0276	J DIMENSION	0.0255
R DIMENSION	0.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 0.0
 CENTER OF GRAVITY COORDINATES 0.320 0.1637
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y
1	-0.3100*	0.2990	21	0.7310	41	0.7460
2	-0.3083	0.2968	22	0.7800	42	0.6803
3	-0.2957	0.2897	23	0.8238	43	0.6277
4	-0.2850	0.2770	24	0.8896	44	0.5609
5	-0.2719	0.2593	25	0.9381	45	0.5073
6	-0.2514	0.2363	26	0.9862	46	0.4557
7	-0.2210	0.2076	27	1.0340	47	0.3880
8	-0.1827	0.1733	28	1.0934	48	0.3362
9	-0.1327	0.1244	29	1.1521	49	0.2847
10	-0.0755	0.0713	30	1.2103	50	0.2204
11	0.0233	0.0149	31	1.2678	51	0.1693
12	0.1238	0.0059	32	1.3248	52	0.1198
13	0.2238	0.0000	33	1.3815	53	0.0753
14	0.3238	0.0000	34	1.4381	54	0.0251
15	0.4238	0.0000	35	1.4947	55	-0.0251
16	0.5238	0.0000	36	1.5513	56	-0.1004
17	0.6238	0.0000	37	1.6079	57	-0.1553
18	0.7238	0.0000	38	1.6645	58	-0.2158
19	0.8238	0.0000	39	1.7211	59	-0.2603
20	0.9238	0.0000	40	1.7781	60	-0.3110

* INDICATES EXTREME POINTS

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FAN COMPRESSOR VANE

CSC 12122

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
7.5000	-20.645 -200 41M 415	0.1163	0.7450	0.008 0.006

LEADING EDGE AXIAL TANGENT POINT -0.0944

J DIMENSION	0.0283	J DIMENSION	0.0263
R DIMENSION	0.0600	S DIMENSION	0.0600

STACK POINT COORDINATES 0.0 7.0
 CENTER OF GRAVITY COORDINATES 0.5461 0.2072
 COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y
1	-0.2291*	0.3408	21	0.6854	41	0.8707
2	-0.2193	0.3379	22	0.7300	42	0.8023
3	-0.2068	0.3313	23	0.7896	43	0.7472
4	-0.1918	0.3213	24	0.8584	44	0.6777
5	-0.1740	0.3088	25	0.9381	45	0.6220
6	-0.1533	0.2938	26	1.0188	46	0.5602
7	-0.1297	0.2767	27	1.1000	47	0.4981
8	-0.1023	0.2577	28	1.1818	48	0.4381
9	-0.0713	0.2373	29	1.2645	49	0.3790
10	0.0233	0.2153	30	1.3481	50	0.3213
11	0.1238	0.1913	31	1.4328	51	0.2653
12	0.2238	0.1653	32	1.5185	52	0.2107
13	0.3238	0.1373	33	1.6052	53	0.1583
14	0.4238	0.1073	34	1.6928	54	0.1083
15	0.5238	0.0753	35	1.7815	55	0.0607
16	0.6238	0.0413	36	1.8711	56	-0.0053
17	0.7238	0.0053	37	1.9618	57	-0.0753
18	0.8238	0.0000	38	2.0535	58	-0.1507
19	0.9238	0.0000	39	2.1462	59	-0.2207
20	1.0238	0.0000	40	2.2400	60	-0.2853

* INDICATES EXTREME POINTS

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FAN COMPRESSOR VANE

CSC 12122

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
3.0000	-21.35 -210 174 63	0.1248	0.7450	0.008 0.006

LEADING EDGE AXIAL TANGENT POINT 0.0118

Q DIMENSION 0.0289 U DIMENSION 0.0271
R DIMENSION 0.0600 S DIMENSION 0.0600

STACK POINT COORDINATES 0.3 0.3
CENTER OF GRAVITY COORDINATES 0.0594 0.2576
COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y		
1	-0.1436*	0.4029	21	0.9772	0.2762	41	0.9951	0.1708
2	-0.1337	0.4098	22	1.0296	0.2827	42	0.9240	0.1580
3	-0.1374	0.3984	23	1.0420	0.2912	43	0.8604	0.1516
4	-0.0252	0.3874	24	1.1473	0.3046	44	0.7939	0.1474
5	0.0425	0.3742	25	1.1992	0.3176	45	0.7357	0.1488
6	0.0966	0.3643	26	1.2509	0.3326	46	0.6777	0.1518
7	0.1508	0.3543	27	1.3150	0.3442	47	0.6049	0.1603
8	0.2181	0.3426	28	1.3658	0.3738	48	0.5513	0.1784
9	0.2720	0.3336	29	1.4287	0.4012	49	0.4959	0.1954
10	0.3259	0.3147	30	1.4414	0.4071	50	0.4270	0.2054
11	0.3741	0.3140	31	1.4511	0.4015	51	0.3723	0.2054
12	0.4171	0.3087	32	1.4483	0.3945	52	0.3043	0.2234
13	0.4515	0.2973	33	1.4377	0.3872	53	0.2503	0.2382
14	0.4868	0.2901	34	1.3827	0.3740	54	0.1968	0.2531
15	0.5223	0.2833	35	1.3363	0.3121	55	0.1305	0.2779
16	0.5583	0.2755	36	1.2757	0.31758	56	0.0730	0.2977
17	0.5910	0.2671	37	1.2252	0.2447	57	-0.0281	0.3186
18	0.6233	0.2539	38	1.1731	0.2264	58	-0.0381	0.3468
19	0.6539	0.2437	39	1.1061	0.2013	59	-0.0887	0.3704
20	0.6915	0.2308	40	1.0511	0.1845	60	-0.1387	0.3955

* INDICATES EXTREME POINTS

10 AUG 79

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FAN COMPRESSOR VANE

CSC 12122

RADIAL DISTANCE	SETTING ANGLE	REFERENCE AIRFOIL THICKNESS	DISTANCE TO LEADING EDGE	RADII L.E. T.E.
3.5000	-22.794 -220 47M 375	0.1329	0.7450	0.008 0.007

LEADING EDGE AXIAL TANGENT POINT 0.1274

Q DIMENSION 0.0294 U DIMENSION 0.0279
R DIMENSION 0.0600 S DIMENSION 0.0600

STACK POINT COORDINATES 0.0 0.0
CENTER OF GRAVITY COORDINATES 0.7694 0.3233
COMPRESSOR ROTATION IS COUNTER CLOCKWISE FROM THE REAR

REFERENCE COORDINATE POINTS

STATION NO.	X	Y	X	Y	X	Y		
1	-0.0640*	0.4823	21	1.0981	0.3393	41	1.1170	0.2269
2	-0.0538	0.4897	22	1.1523	0.3462	42	1.0423	0.2136
3	0.0024	0.4774	23	1.2087	0.3554	43	0.9818	0.2070
4	0.0588	0.4659	24	1.2762	0.3702	44	0.9057	0.2040
5	0.1288	0.4513	25	1.3279	0.3847	45	0.8447	0.2058
6	0.1848	0.4403	26	1.3813	0.4017	46	0.7840	0.2112
7	0.2408	0.4296	27	1.4473	0.4262	47	0.7113	0.2209
8	0.3107	0.4169	28	1.4993	0.4486	48	0.6337	0.2302
9	0.3666	0.4071	29	1.5639	0.4799	49	0.5562	0.2408
10	0.4225	0.3977	30	1.5767	0.4867	50	0.5248	0.2561
11	0.4923	0.3864	31	1.5873	0.4811	51	0.4680	0.2694
12	0.5484	0.3777	32	1.5846	0.4759	52	0.3976	0.2891
13	0.6180	0.3673	33	1.5741	0.4658	53	0.3418	0.3060
14	0.6740	0.3594	34	1.5348	0.4178	54	0.2884	0.3242
15	0.7299	0.3518	35	1.4717	0.3823	55	0.2180	0.3489
16	0.8001	0.3429	36	1.4095	0.3420	56	0.1638	0.3702
17	0.8539	0.3370	37	1.3574	0.3131	57	0.1103	0.3928
18	0.9074	0.3335	38	1.3033	0.2874	58	0.0643	0.4229
19	0.9757	0.3324	39	1.2334	0.2598	59	-0.0078	0.4483
20	1.0301	0.3340	40	1.1758	0.2416	60	-0.0391	0.4754

* INDICATES EXTREME POINTS