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REDESIGN AND CASCADE TESTS OF A SUPERCRITICAL CONTROLLED DIFFUSION STATOR BLADE-SECTION

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Abstract

A supercritical stator blade section, previously tested in cascade, and characterized by a "flat-roof-top" suction surface Mach number distribution, has been redesigned and retested. At near design conditions, the losses and air turning were improved over the original blade by 50 percent and 7 percent respectively. The key element in the improved performance was a small blade reshaping. This produced a continuous flow acceleration over the first one-third chord of the suction surface which successfully prevented a premature laminar separation bubble. Several recently available inviscid analysis codes and one fully viscous (Navier-Stokes) analysis code were used in the redesign process. The validity of these codes was enhanced by the test results.

Introduction

At the AIAA 21st Aerospace Sciences Meeting (1983) Boldman et al.(1) reported on cascade tests of shockless supercritical airfoils. The basic airfoil design had a "flat-roof-top" Mach number distribution on the suction surface that was supersonic. This surface Mach number distribution is typical of the early designs utilizing the Bauer, Garabedian and Korn(2) (BGK) inverse design code. As reported in Ref. 1, the design point performance of the supercritical blade section was disappointing due to a premature laminar separation bubble and the subsequent early separation of the reattached turbulent boundary layer. As discussed in the Concluding Remarks of Ref. 1, a laminar boundary layer may not persist to the downstream end of the "flat-roof-top" Mach number distribution as predicted for the design. The location of transition and or laminar separation on an airfoil surface with a "flat-roof-top" Mach number distribution is especially sensitive to such factors as Reynolds number, inlet turbulence intensity and surface roughness. Thus, its location is almost impossible to predict. This, in turn can have a significant effect on the initial thickness of a reattaching turbulent boundary layer on that surface.

Therefore, as concluded by Boldman as well as by Stephens and Hobbs,(3) and Rechter et al.,(4) a "flat-roof-top" surface Mach number distribution should be avoided and replaced by one with a continuous flow acceleration to a peak near one-third of the chord. In this way the location of transition or laminar separation is forced to occur near this peak for most inlet flow conditions. Such a distribution should also increase the low-loss range of off-design, positive incidence angles which is desired in compressor designs.

Shortly after Boldman's cascade tests were completed, a Navier-Stokes (viscous) analysis code became available. This code was originally developed by Steger(5) for an isolated airfoil, but later modified and adapted to cascade flow by Donovan.(6) This Navier-Stokes code was utilized to predict the design point performance of one of the supercritical blade designs which had been tested. It correctly predicted the measured performance in terms of the near leading edge loss of laminar flow on the suction surface and the subsequent early separation of the turbulent boundary layer near 70 percent chord.

Based on this experience a blade redesign and retest effort was initiated. The redesigned blade section was required to have a continuous flow acceleration on the suction surface to a peak Mach number near the 35 percent chord location. An additional requirement was that the redesigned blade geometry had to fall within the cross section of the original design so that it could be fabricated by removing material from the original blade. Finally, a significant improvement in performance as predicted by the Navier-Stokes code was necessary before giving approval to machine the redesigned blade.

Three inviscid, two-dimensional/quasi-three-dimensional flow codes have recently become available for turbomachinery blade designs that were crucial to obtaining a satisfactory redesign within a few weeks time. One, a panel method code by McFarland(7) was used to quickly determine allowable blade cross sections that would provide the desired flow acceleration on the suction surface. This code, while fast, is limited to shock-free flows and is inaccurate at supersonic Mach numbers. A second code by Farrell(8) accurately calculates transonic flows (to local Mach numbers of 1.4) and captures shock-waves. This code was used to analyze the promising blade shapes indicated by the panel method code. The third code, developed by Beauchamp,(9,10) is an addition to Farrell's code that can reshape a blade with a strong shock-wave into one that is shock-free. It utilizes the fictitious gas concept originated by Sobieczky.(11) An integral boundary layer code by McNally(12) was also used in conjunction with the three inviscid codes to account for the boundary layer displacement thickness along the blade.

The objectives of the present paper are: (1) to describe the use of these recently developed flow analysis codes in redesigning a supercritical, controlled diffusion stator blade section for improved performance over that previously reported, and (2) to present and discuss cascade test results for the redesigned section with comparisons to the original.

This paper is organized as follows: First, a brief review of the original blade design and an assessment of its test results are presented. This is followed by a description of the blade redesign process. The test results for the redesigned blade are then discussed and summarized.

Acknowledgements

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Original Blade Design and Assessment of Cascade Test Results

The original design for application to a compressor stator midspan section was done by the general method of BGK, Ref. 2. This is a two-dimensional, inverse design method in that the blade surface pressure distribution is prescribed while the blade geometry is an output of the solution. The solutions of present interest are for supercritical but shock-free blades in cascade flow. The BGK method has recently been extended to higher blade solidities in cascade flow by Sanz.[13, 14] Further details of the original design can be found in Ref. 1 which describes the design and testing of two cascade airfoil sections which differed in trailing edge thickness. This difference in thickness occurred because of different Reynolds number (based on chord) application. The design with the thicker trailing edge is the subject of the present redesign effort because of its earlier availability for remachining.

Test results from the original blade near its design point (inlet Mach number, 0.75; inlet air angle, 35.3°, and AVDR, 1.09) are presented in Fig. 1 along with predicted surface Mach number distributions from two different analyses. AVDR, an axial velocity density ratio across a cascade airfoil, is a measure of the flow blockage across the cascade airfoil (see ref. 1). There is a definite leveling off of the suction surface Mach number data near 70 percent X/C_x (see sketch on fig. 1). Flow turning and total pressure loss data are summarized in Table 1. There is a higher than expected loss in total pressure of 7 percent (throughout the paper total pressure loss is normalized by the cascade inlet total-to-static pressure difference). In addition the measured air turning was 28.4° as compared to the design value of 35°. This combination of evidence is indicative of turbulent boundary layer separation long before the trailing edge. A further test of injecting alcohol into the pressure taps on the suction surface near 8 and 17 percent, X/C_x, indicated upstream, recirculating flow and thus a large laminar separation bubble. This contradicts the prediction of laminar flow to about 35 percent, X/C_x, by McNally's integral boundary layer code. Based on this prediction, the turbulent boundary layer was started at this location in the inverse design code, and no turbulent boundary layer separation prior to the trailing edge was indicated by that code.

The Navier-Stokes analysis, which is a viscous flow solution of the complete flow field, predicts an early turbulent boundary layer separation near 70 percent, X/C_x, similar to the experimental data (see fig. 1). It also indicates an unstable transition at about 6 percent, X/C_x, which is consistent with the observed laminar separation bubble.

The measured variation in loss of total pressure with inlet air angle at design inlet Mach number (0.75) is shown in Fig. 2. At inlet air angles less than design, the loss is reduced. Minimum loss occurs at an inlet angle 2.7° less than design. This loss reduction at negative incidence angles is a result of suppressing the premature laminar separation bubble on the suction surface. The surface Mach number distribution for the minimum loss condition is shown in Fig. 3 along with a repeat of the distribution for the near design condition from Fig. 1. The minimum loss of 5 percent instead of 7 percent near design is accompanied by increased air turning while the suction surface Mach number distribution tended to decrease all the way from the peak at 35 percent, X/C_x, to the trailing edge. In addition, there is only a modest adverse pressure gradient on the suction surface between 8 and 17 percent, X/C_x, and observing the alcohol injection, as before, revealed no large recirculating flows at the minimum loss condition.

Thus, as previsouly discussed, a redesign and retest effort was initiated, aimed at avoiding premature laminar separation and maintaining the reattached turbulent boundary layer all the way to the trailing edge.

Blade Redesign

Three inviscid, two-dimensional/quasi-three-dimensional flow analysis codes were utilized in conjunction with an integral boundary layer code (to account for displacement thicknesses) in the redesign process. A two-dimensional, fully viscous, Navier-Stokes analysis code was also used to substantiate the final redesign selected from the inviscid plus boundary layer approach. These are discussed next.

Inviscid Plus Boundary Layer Analysis. A panel method inviscid flow code[7] was utilized for its speed in determining allowable blade shapes that would provide the desired flow acceleration on the suction surface. This method however is limited to shock-free flows and is inaccurate at supersonic Mach numbers. Thus, a full potential solution flow code[8] (called Qsonic), which can handle shock waves with Mach numbers up to 1.4, was used on the most promising shapes from the panel method. This Qsonic flow code has a body fitted mesh with an artificial density imposed in the transonic region to insure stability and capture of shock waves. Finally, an addition[9,10] to the basic Qsonic flow code was utilized to reshape blades with predicted shocks into ones that are shock-free. An integral boundary layer code[11] was used to determine boundary layer displacement thicknesses. These displacement thicknesses were subtracted from the effective blade shapes (used by the inviscid codes) to obtain the actual blade coordinates.

Surface Mach number predictions from the inviscid plus boundary layer analyses for the preliminary and final redesigned blade shapes are shown in Fig. 4(a). The dashed line distribution labelled "preliminary" was obtained from the Qsonic flow code. This Mach number distribution exhibited the desired acceleration to a peak near 35 percent, X/C_x, but the peak level (1.44) was too high and it was followed by a steep decline to subsonic values, indicative of a strong shock wave. Several trial-and-error blade shape changes were analyzed to eliminate or soften the shock wave but none were successful. Fortunately, a modification[9,10] was recently made to the Qsonic flow code which utilizes the ficticious gas concept[11] and changes the effective blade coordinates wetted by the supersonic flow region such that the flow remains supersonic, but shock-free. With the ficticious gas concept, whenever the governing flow equations become hyperbolic (Mach number exceeds unity) the density-speed relationship is switched from the real relationship to one for a denser, but ficticious gas that returns the equations to elliptic behavior which is shock-free. This flow code changes the sonic line shape, prescribes a new one that results in supersonic but shock-free flow within the supersonic pocket, and calculates the required blade surface modification. Most important, the code avoids lengthy trail and error geometry changes which usually fall far short of shock-free flow. This code is a valuable numerical tool for designing supercritical shock-free stator blades. The results of this process are shown in Fig. 4(a) by the solid line distribution labelled "final." The peak Mach number is now a reasonable value (1.27) and no shock-wave is indicated in the velocity diffusion to the trailing edge. The desired continuous flow acceleration to about the 35 percent, X/C_x, has been achieved.

Viscous Navier-Stokes Analysis. Because the Navier-Stokes Code[6] was successful in predicting the significant performance features of the original design, it was applied to the proposed final design. The results of its prediction are shown in Fig. 4(b). The Mach number distribution is essentially the same as that shown in Fig. 4(a). Details of this analysis showed no laminar flow instabilities until near the peak Mach number (at 35 percent, X/C_x) on the suction surface. From the detailed output of the Navier-Stokes code, a very small localized separation region was detected immediately adjacent to the surface on the rear portion of the blade. However, this separation region was so small it did not appreciably affect the flow over the rear portion of the blade. The loss prediction from the Navier-Stokes code, although not proven reliable in absolute value, did indicate the redesigned blade should have less than half the loss of the original.

Redesign Blade Section Geometry. The redesigned blade is compared to the original in Fig. 5. These blade cross sections are those with the boundary layer displacement thicknesses subtracted form the effective blade shapes analyzed, with one exception. The laminar boundary layer was so thin over the accelerating flow region of the redesigned blade, its displacement thickness was ignored (same as for the original blade). As shown, the change in blade geometry occurs only over the first 35 percent, X/C_x, on the suction surface. Although the indicated changes are small, local thickness equivalent of 1-percent chord or less, they do result in large predicted changes in the supersonic flow regime.

Cascade Test Results and Discussion of Redesigned Blade

Tests with the redesigned blade were performed in exactly the same way as for the original[1] and resulted in a reduced measured value of AVDR(1.05). Test results from the redesigned blade near the design point are presented in Fig. 6, along with the flow predictions previously discussed. The surface Mach number data is in very good agreement with both analysis predictions. This confirms the usefulness of these codes. Also, there is no leveling off of the suction surface Mach number before the trailing edge, indicative of early turbulent boundary layer separation. The measured total pressure loss is 3.5 percent and air turning is 30.7°. Comparable values for the original blade were 7 percent loss and 28.4° turning. This comparison of total pressure loss and flow turning data between the redesigned and original blades is presented in Table 1. As before, alcohol was injected into the pressure taps on the suction surface from 1-1/2 to 17 percent, X/C_x, (additional taps were installed for the redesigned blade) and no flow recirculation was observed.

Schlieren images of the suction surface for the original and redesigned blade sections at near design conditions are shown in Fig. 7. Near 45 percent chord there is much less shock activity indicated for the redesigned compared to the original blade. Also, little boundary layer separation is indicated over the latter half chord of the redesigned blade compared to the significant separation starting near 70 percent chord for the original. These images are compatible with the pressure data (figs. 1 and 6) previously discussed.

The significant performance improvements of the redesigned over the original blade near design conditions resulted from small (1 percent chord and less) blade shape changes to the suction surface over the first one-third chord. Further improvements with similar design requirements are believed likely. Smaller trailing edge thicknesses should further reduce losses somewhat (see ref. 1). Also, air turning predictions should get better as flow deviation rules are developed from tests to account for the present deficiencies in modeling trailing edge flows with this new family of controlled diffusion blades.

Also of interest is the off-design performance of the blade section. Some off-design performance is shown in figure 8 for both the redesigned and original blade. At design inlet Mach number, the loss in total pressure as a function of inlet air angle (fig. 8(a)) for the redesigned blade is not only down 50 percent from the original near the design inlet angle, but is less over the entire range tested. There is a wider low loss range of inlet air angle for the redesigned blade as well. Near design inlet air angle, the loss in total pressure as a function of inlet Mach number (fig. 8(b)) decreases to 2 percent at the lower Mach numbers tested. However, as the inlet Mach number is increased above the design value the loss increases dramatically.

This performance is probably caused by the appearance of strong shocks and their influence on thickening and/or separating the suction surface boundary layer. Comparable data for the original blade are not available.

Summary of Results

A supercritical stator blade section, previously tested in cascade, and characterized by a "flat-roof-top" suction surface Mach number distribution, has been redesigned and retested. The disappointing performance of the original blade resulted from a large premature laminar separation bubble which in turn caused early separation of the turbulent boundary layer. The main results from this redesign and retest effort were as follows:

(1) The key element in the redesign was a small blade reshaping. This produced a continuous flow acceleration over the first one-third chord of the suction surface which successfully prevented a premature laminar separation bubble. The redesigned shape resulted in a change of local thickness equivalent to no more than 1 percent chord different from the original.

(2) At near design conditions, the losses and air turing were improved over the original blade by 50 percent and 7 percent respectively. Schlieren images confirmed the improved flow conditions over the redesigned sections. Off-design performance was also better for the redesigned blade.

(3) Several recently available inviscid analysis codes and one fully viscous (Navier-Stokes) code were utilized in the redesign progress. The inviscid analyses quickly selected the proper blade shape with the desired Mach number distribution, without a shock wave and no turbulent boundary layer separation. The viscous analysis successfully postdicted the performance of the original blade and predicted that of the redesign. The validity of these analysis codes was enhanced by the test results.

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TABLE 1. - CASCADE PERFORMANCE OF ORIGINAL AND REDESIGNED BLADE SECTIONS; INLET MACH NUMBER, 0.75

	Measured Data		
	Original design		Redesign
	Design incidence	Minimum loss incidence	Design incidence
Total Pressure loss, percent	7	5	3.5
Air turning, degrees	28.4	29.4	30.7

Design air turning, 35 degrees.

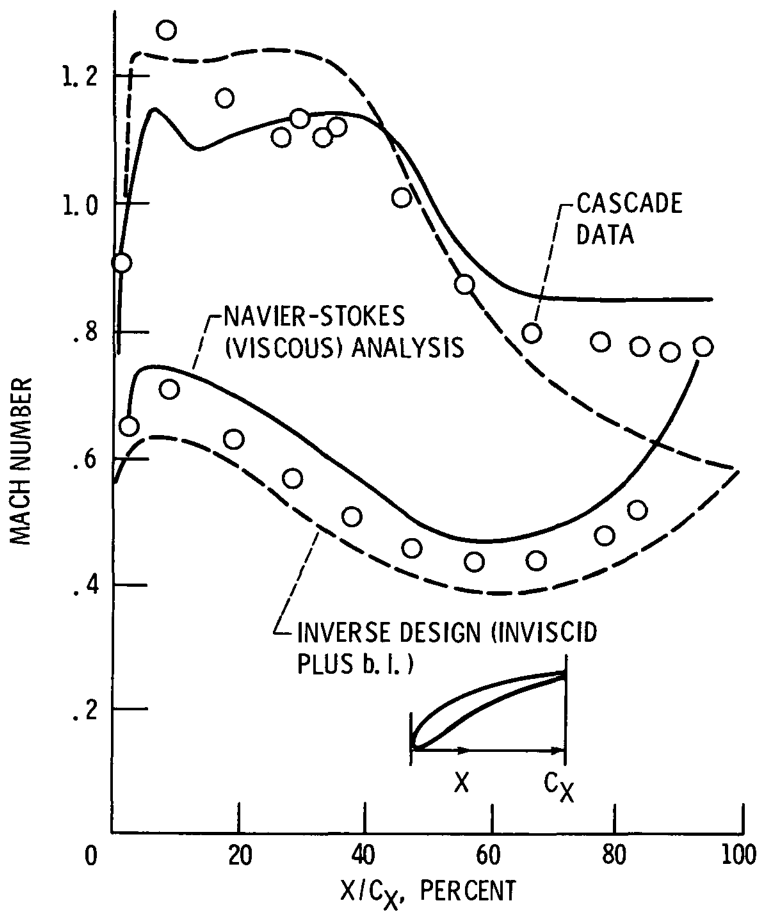


Fig. 1. - Surface Mach number for original blade near design point: inlet Mach number, 0.75; inlet air angle, 35.3°; AVDR, 1.09.

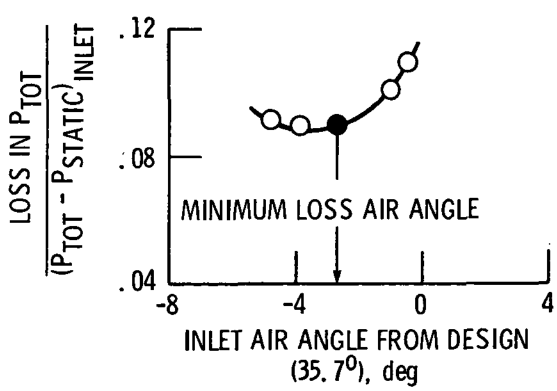


Fig. 2. - Variation of loss in total pressure with inlet air angle for original blade at design inlet Mach number, 0.75.

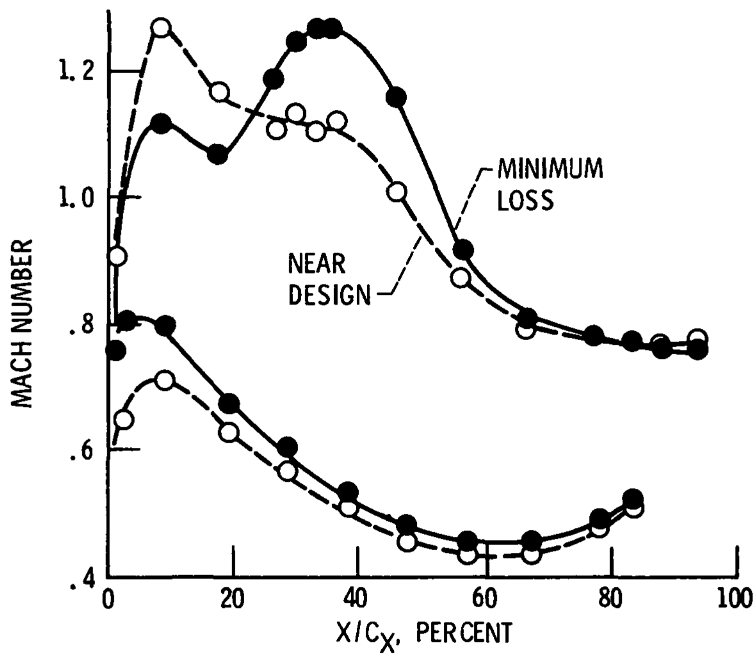
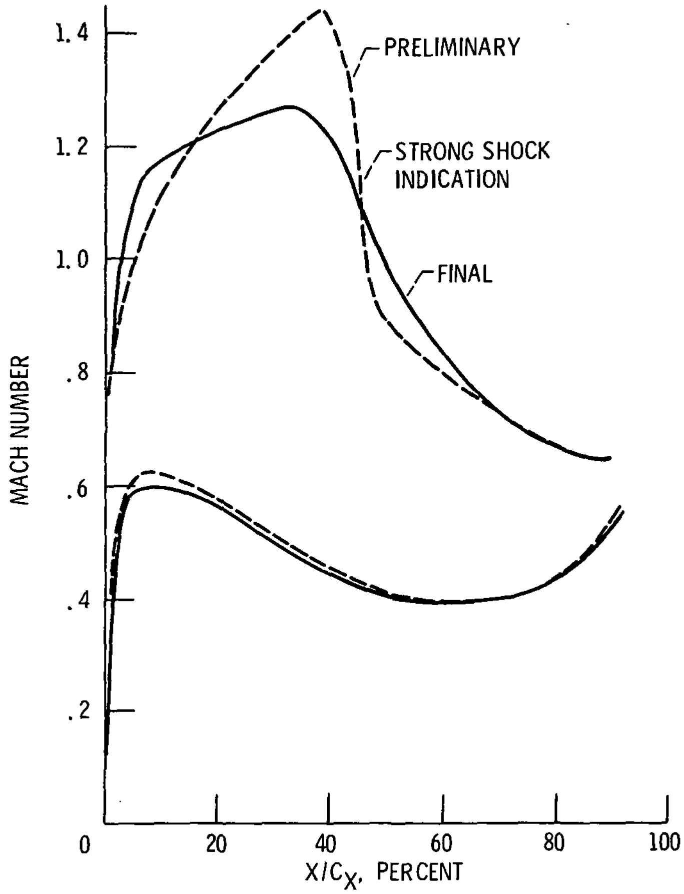
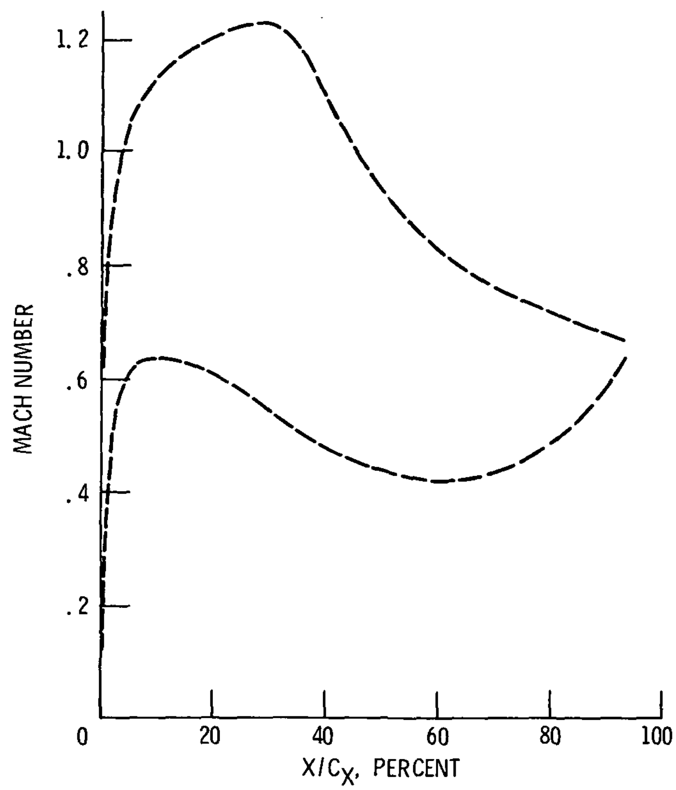


Fig. 3. - Surface Mach numbers near design-, and at minimum loss-inlet air angles for original blade at design inlet Mach number, 0.75.



(a) Inviscid plus b. l. analysis.



(b) Navier-Stokes (viscous) analysis.

Fig. 4. - Surface Mach number predictions for redesigned blade at design point: inlet Mach number, 0.75; inlet air angle, 35.7°.

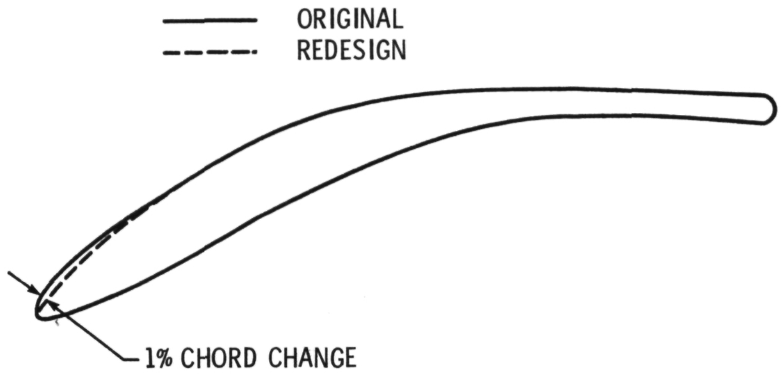


Fig. 5. - Blade section geometry of original and redesign.

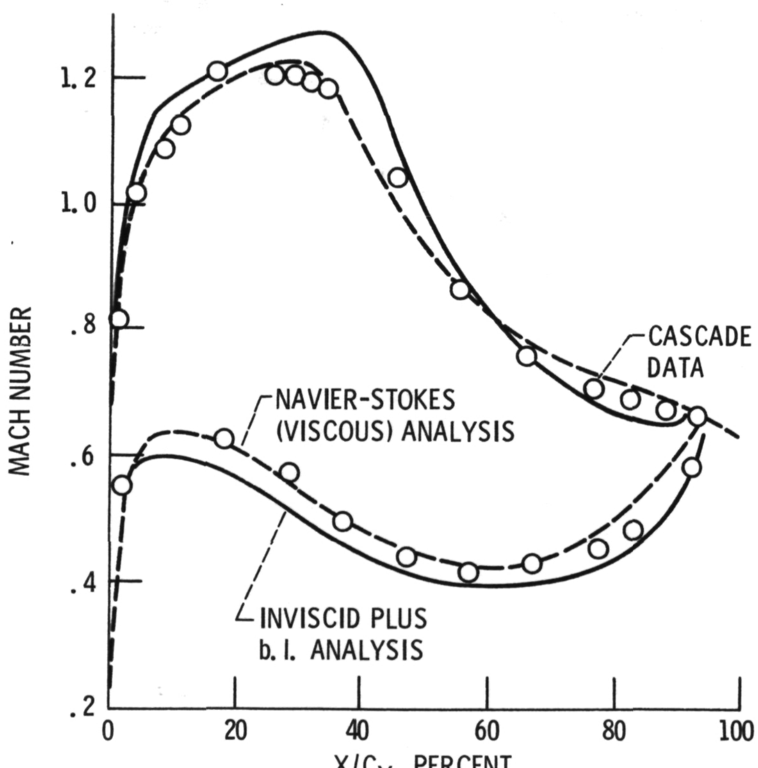
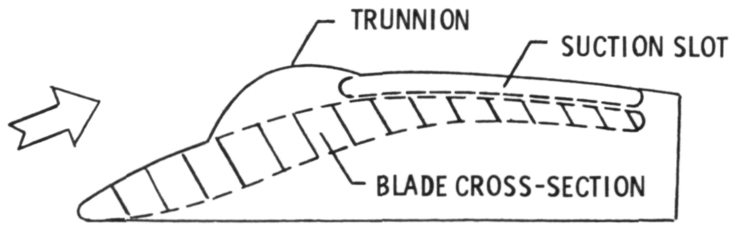
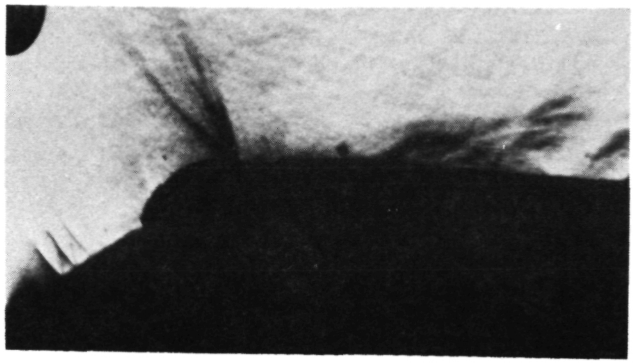


Fig. 6. - Surface Mach number for redesigned blade near design point: inlet Mach number, 0.75; inlet air angle, 35.3^{0}; AVDR, 1.05.



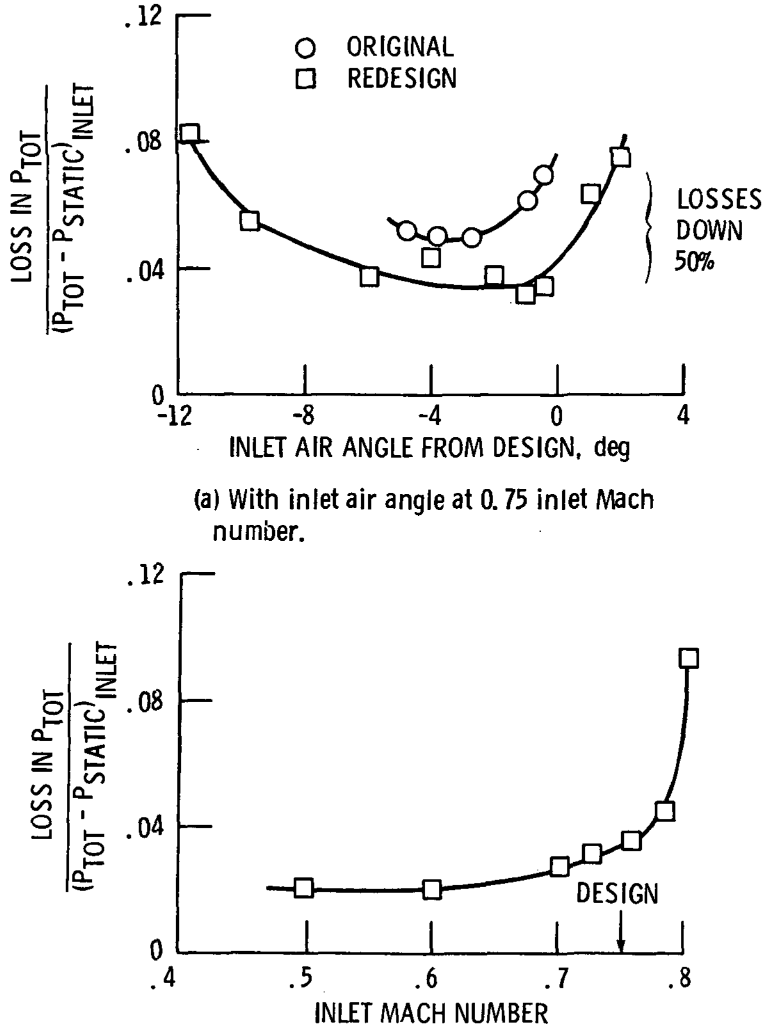
(a) Set-up.



(b) Original, 7% loss.

(c) Redesign, 3.5% loss.

Figure 7. - Schlieren images for original and redesigned blade. Near design point: inlet mach no., 0.75; inlet air angle, 35.3°.



(a) With inlet air angle at 0.75 inlet Mach number.

(b) With inlet Mach number at -0.4° from design inlet air angle.

Fig. 8. - Off-design variation of loss in total pressure for original and redesigned blade.

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