NASA TM-83519

NASA Technical Memorandum 83519

NASA-TM-83519 19840006052

Design and Performance of a Fixed, Nonaccelerating, Guide Vane Cascade That Operates Over an Inlet Flow Angle Range of 60°

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Prepared for the
Twenty-ninth Annual International Gas Turbine Conference
sponsored by the American Society of Mechanical Engineers
Amsterdam, The Netherlands, June 3–7, 1984

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E-1868

ABSTRACT

A unique set of wind tunnel guide vanes are designed with an inverse design code and analyzed with a panel method and an integral boundary layer code developed at the NASA Lewis Research Center. The fixed guide vanes, 80 feet long with 6-foot chord length, were designed for the NASA Ames 40 x 80/80 x 120 ft Wind Tunnel. Low subsonic flow is accepted over a 60° range of inlet angle from either the 40 x 80 leg or the 80 x 120 leg of the wind tunnel, and directed axially into the main leg of the tunnel where drive fans are located. Experimental tests of 1/10-scale models were conducted to verify design calculations.

1. INTRODUCTION

To expand testing capabilities an 80 x 120 ft. leg has recently been added to the National Aeronautics and Space Administration's Ames Research Center 40 x 80 ft. wind tunnel (AWST, 1). This new leg joins the original tunnel circuit at a 45° angle just upstream of the drive fans. At this juncture a cascade of guide vanes is required to turn the flow 45° from the new leg into the original circuit for axial inflow to the drive fans. However no air turning is desired for operation in the 40 x 80 mode when the new leg is closed off. It is obviously impractical to alternately install and remove guide vanes depending on the operating mode.

The guide vane design goals were mechanical simplicity and ruggedness to lessen cost and expedite operations, and low aerodynamics losses to minimize energy consumption. Low losses were desired not only at the two operating points with inlet air angles of 0° and 45° but also over an extended range from about -5° to +55°. This was to accommodate expected spatial nonuniformities inherent in the flow approaching the guide vanes particularly for the 80 x 120 operation. In this mode large models with vortex wakes are located just upstream of the vanes. The NASA Lewis Research Center has experience in designing airfoils in cascade not only with conventional techniques but also with computational fluid dynamics codes it has recently developed. Therefore it proposed some guide vane designs for Ames' consideration.

The Lewis aerodynamic design studies considered three different concepts. One was the use of classical 65-series airfoil cascade data. A review of these data (NACA TR1368, 2) indicated that less than one-half of the desired 60° inlet air angle range was possible with low loss with a single row of airfoils. A two-row design was developed that would do the job but the first row required a setting or stagger angle in the 40 x 80 mode different from that in the 80 x 120 mode. Another design utilized a single row of 20 percent thick airfoils, the forward segment of which could be pivoted to match the oncoming flow direction. The third and clearly most desirable design concept was a single row of fixed airfoils capable of accepting the flow with low losses at both modes of operation.

The purposes of this report are (1) to describe the aerodynamic design process for a single row of fixed turning vanes that could operate over an inlet flow angle range of 60° with low loss, and (2) to present experimental data of the performance of these vanes and compare these results with design predictions.

There were three computational fluid dynamics codes utilized in the design and off-design analysis of the guide vanes, all developed at the Lewis Research Center. The airfoil in cascade was designed with a recently developed inviscid, inverse design code with a boundary layer correction (Sanz, 3,4). The off-design analysis of the resulting cascade was made by another recently developed inviscid, blade-to-blade, panel method code (McFarland, 5). An integral boundary layer code (McNally, 6) was used in conjunction with the panel analysis.

The final designs were built and tested by the Ames Research Center in 0.10 scale size. Tests were conducted over an inlet air angle range from -5° to +60° at the design inlet Mach number of about 0.2. Because of Reynolds number effects, all analysis cal-

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culations were repeated for comparison with the scale model test results.

2. THE DESIGN PROBLEM, PROCEDURE AND PREDICTIONS

A schematic of the junction of the 40 x 80 and 80 x 120 wind tunnel legs at the Ames Research Center is shown in Fig. 1. The axis of the cascade of guide vanes shown in this figure is staggered at an angle of 25° to the plane of the drive fans. The problem at hand was to find a set of vanes capable of receiving the flow from either leg without the need of moving parts. In other words, find a cascade with acceptable losses in a range of inlet air angle operation of about 60° and with the 25° stagger angle shown.

For the low subsonic speed involved (Mach No. of 0.2) an airfoil with a very blunt leading edge appeared most appropriate to maximize the low loss range of inlet flow angle. NACA 65-series airfoils are relatively blunt and have been used successfully for many years to provide low losses over wide ranges of inlet flow angles and for inlet Mach numbers of 0.6 or less. In reviewing the 65-series cascade data (2), which is for a maximum thickness/chord of 0.10 and a maximum blade solidity of 1.50, it can be seen that less than half the required inlet air angle range of 60° was obtainable with low loss in a single row of airfoils.

The inverse design code has the capability of producing more arbitrary shapes with thick leading edges, if so desired. The technique has the additional advantage of being able to input the surface velocity distribution directly. This allows control of the velocity diffusion to eliminate the boundary layer separation, thus maximizing turning and minimizing loss. Also, the new design code in the subsonic regime provides solutions very quickly; thus a large number of differing shapes can be explored in a short time. The capability of designing with blade solidities of as high as 2.0 was another encouraging feature in its use. To obtain 60° of inlet air angle range of operation in nonaccelerating flow with little or no separation would obviously require relatively high solidities. Thus, it was clear that the best chance of achieving the design objectives with a single row of airfoils was with the inverse design method.

Lewis No. 1 Design

Use of the inverse design code requires the input of a surface velocity distribution and three design parameters that control the solidity, the inlet Mach number and the inlet air angle at the design point. In the search for a single cascade of fixed geometry capable of receiving the flow at both modes of operation, various design inlet air angles could be tried. Nevertheless, because low losses were desired in the 40 x 80 leg, the first design, designated Lewis No. 1, was optimized for the 40 x 80 leg with incidence angle (oncoming air angle relative to mean camber line at leading edge) near zero and with essentially zero turning. However, the vane shape was highly asymmetric because of the 25° of stagger angle of the cascade, which results in an inlet air angle of approximately -25°, relative to a normal to the cascade axis. All inlet and exit air angle values in this report are relative to the drive fan axes for a more convenient reference (Fig. 1).

The design surface Mach number distribution and blade geometry are shown in Fig. 2. The input surface Mach number distribution reflects the necessary asymmetry. This fixed geometry guide vane has a solidity of 1.64, and maximum thickness/chord of 0.209. It was designed assuming a Reynolds number of 6 million (full scale operation).

In the design code the input surface velocity distribution is chosen to tailor the blade geometry according to the design philosophy. The maximum thickness to chord ratio, the leading edge curvature, and the trailing edge thickness can be controlled with this input speed distribution. In the case at hand, the main concern is range of operation and a thick blade with a blunt leading edge was designed. Continuous flow acceleration to a peak velocity near 55 percent chord on the upper surface was specified to provide more incidence angle range for off-design operation. This peak velocity was higher than that required for a fixed inlet flow angle and diffusion from this peak results in relatively higher losses. If only the 40 x 80 mode with 0° inlet air angle was to be considered, a much thinner body with less diffusion could be designed. The criteria in this case was not to optimize loss at the design point, but to have a wide range of operation with acceptable losses.

The panel method code was used to analyze the designed airfoil. This code was particularly well suited for the task because it can readily accept arbitrary blade shapes and quickly solve the flow field at different off-design conditions. Compressible flow effects are approximated in this analysis code, but the method is accurate for the low speed flow being considered in this design. Figure 3 shows the analysis results of the Lewis No. 1 design in the 80 x 120 mode with an inlet air angle of 45°.

Due to Reynolds number differences, boundary layer calculations were performed for both the full scale and 0.10 scale cases. The boundary layer was treated differently for these cases to reflect actual testing conditions.

In the full scale wind tunnel, tests are often run on full scale airplanes with engines running. Buildup of soot and other contaminants on guide vane surfaces is likely, and it is anticipated that these contaminants will cause tripping of the boundary layer to turbulent flow near the leading edge. Boundary layer calculations for the full scale guide vanes were therefore forced turbulent near the leading edge. In some cases this tripping is probably unrealistic in regions of acceleration, but in such cases the calculations should provide conservative estimates of performance.

The boundary layer calculations for the 0.10 scale model vanes were made by letting a laminar boundary layer develop in a natural fashion from the leading edge. This approach was taken because it proved difficult to properly simulate the tripping in the scale model tests. Consequently, no tripping was attempted and the calculations were then conducted to compare with the test. In all cases, the steepness of the surface velocity gradients caused the laminar boundary layer to separate rather than pass through natural transition (zero skin friction predicted in Cohen-Reshotko laminar boundary layer method, see ref. 6). The boundary layer was reattached as turbulent using an initial momentum thickness equal to the value at separation (conservation of momentum), and an incompressible form factor of 1.4 (equilibrium layer). If the Reynolds number based on momentum thickness for the initial turbulent boundary layer using this method was less than 320, then the momentum thickness was increased to a value corresponding to 320 (Preston, 7).

The results of the calculations for inlet air angles of 0° and 45° are presented in Fig. 4 and are plotted as incompressible form factor against percent of chord. Only the turbulent portion of the calculation is plotted. Turbulent boundary layer separation is assumed to occur when the incompressible form factor exceeds 2.0.

Loss coefficients (total pressure loss over inlet dynamic head) were estimated based on the boundary layer calculations using the method of Speidel (8,9), which incorporates a simple model for calculating loss with separated boundary layers. No loss calculations were attemped for inlet angles greater that 45° because at those very high angles the surface velocity gradients are high and turbulent boundary layer separation is likely over 20 to 25 percent of the blade surface. The loss model was not constructed for such extreme conditions, and should not be relied on for exact predictions. It was calculated for the 45° inlet angle only to provide an approximate guide for comparing trends. Based on these calculations, losses estimated for the full scale blading were 0.036 in the 40 x 80 mode, and 0.043 for the 45° inlet angle in the 80 x 120 mode. For the 0.10 scale model, the corresponding estimated losses were 0.038 and 0.097, respectively. Regardless of Reynolds number effects, the loss levels appeared acceptable for a fixed geometry design operating over the required wide operating incidence angle range. Thus a hinged version of this guide vane known as the Lewis No. 2 design was not pursued.

Lewis No. 3 Design

Although the concept of a fixed geometry was proved viable in initial experimental tests of the Lewis No. 1 design at inlet air angles of 0° and 45°, its losses at 55° were unacceptable (see Section 4). To increase the overall operating range a compromised design point was pursued. This would result in some negative incidence angle of the flow in the 40 x 80 mode, but still acceptable losses, and correspondingly a lower incidence angle in the 80 x 120 mode. A few degrees of blade camber and a modest increase in solidity were deemed necessary for such a design.

To increase the range in this mode of operation, two options were available. The first one would be to increase the solidity of the cascade, with the subsequent reduction of the peak velocity near the leading edge on the upper surface. This option should keep nearly the same performance in the 40 x 80 mode. The second option available was to design the blade with both a higher solidity and a few degrees of turning in such a way that the flow would have some negative incidence in the 40 x 80 mode, but still acceptable losses, and correspondingly a lower incidence angle in the 80 x 120 mode.

The inverse design code was again used in this compromised design point approach to design a new blade following this second alternative. The new blade, the Lewis No. 3 design, is shown in Fig. 5. It has a solidity of 1.94 and a maximum thickness/chord of 0.19. At nominal conditions, the vane will receive the flow from the 40 x 80 leg with a negative incidence of 6.7° and in the 80 x 120 mode it will receive it with 38° of positive incidence. This combined with the higher solidity was expected to increase the low loss range of operation in the 80 x 120 mode at the expense of some increase in losses in the 40 x 80 mode. The panel code analysis in the 80 x 120 mode with an inlet air angle of 45° is shown in Fig. 6. The reduction in the peak suction surface Mach number on the upper surface from the Lewis No. 1 design is indicated.

The reduction in peak upper surface Mach number is considered essential for stable, high inlet angle operation. The very steep velocity gradients at the leading edge of the Lewis No. 1 design (80 x 120 mode) present a distinct danger of completely stalled flow at high inlet angle. The calculated loss of 0.043 for the full scale version of that blade assumes an unstalled condition, i.e., laminar separation and turbulent reattachment near the leading edge. Since, even with this assumption, the last 15 percent of the upper surface boundary layer is separated, the calculated loss of 0.043 should be viewed more as an estimate that an exact prediction. Clearly, using an integral boundary layer method to calculate loss for blades having incidence angles of 45° and substantial boundary layer separation, should only be done to compare similar designs and indicate trends.

Boundary layer calculations for the Lewis No. 3 blade are shown for inlet air angles of 0° and 45° in Fig. 7. Differences between Lewis No. 1 (Fig. 4) and Lewis No. 3 boundary layer behaviour occurred principally on the upper surface. For the 40 x 80 mode the Lewis No. 3 upper surface boundary layer shows earlier separation. This is due to the more adverse pressure gradient that results from compromising the 40 x 80 design as previously discussed. A loss of 0.042 was calculated for the 0.10 scale (laminar boundary layer at leading edge); a loss of 0.042 was also calculated for the full scale blade assuming a turbulent boundary layer beginning near the leading edge.

In the 80 x 120 mode at a 45° inlet air angle (Fig. 7(c) and (d)) losses of 0.093 and 0.052 were calculated for the 0.10 scale and full scale blades, respectively. At inlet angles greater than 45°, the loss was increasingly controlled by the extent of a laminar separation bubble at the leading edge. It is not feasible to incorporate this quantitatively into the simple loss model used, so no loss estimates are made for inlet angles greater than 45°. A summary of these losses and the implications for the full scale performance are presented, and will be discussed, in Section 4.

In Fig. 8 the shapes for the Lewis No. 1 and Lewis No. 3 designs are shown along with their mean camber lines and their edge angles relative to the drive fan axes. The camber line for Lewis No. 1 is S-shaped with both ends aligned with the drive fan axes. Thus the incidence angle would be near 0° in the 40 x 80 operating mode, and near 45° in the 80 x 120 mode. The camber line for Lewis No. 3 differs from No. 1 mainly near the leading edge. Most of the upsweep to become aligned with the drive fan axes has been removed resulting in a non-optimum incidence angle of -6.7° in the 40 x 80 mode and one near 38° in the 80 x 120 mode. The camber line at the trailing edge is aligned with the drive fan axes for both modes. In Fig. 9 the surface pressure distributions for both designs are compared in both modes of operation.

3. TEST FACILITY, MODELS AND INSTRUMENTATION

The experimental verification of the vane set design was conducted at and by the Ames Research Center (Norman, et al., 10). The test facility is an open loop wind tunnel with rectangular cross-section in which a cascade of 0.10 scale vanes with aspect ratio of 5 was tested. There was no tunnel wall boundary layer control and no tailboards to improve vane wake periodicity. A schematic of the test facility is shown in Fig. 10.

The inlet flow angle into the cascade was set at seven different values (-5°, 0°, 5°, 45°, 50°, 55°, and 60°) by alternate inlet pieces. The cascade test section consisted of eight vanes plus the flat tunnel walls. Three vanes (nos. 4 to 6, Fig. 10) were instrumented with static pressure taps, 15 on the upper and 14 on the lower surface. Static pressure taps were also located on all four tunnel walls to verify that the vane set and not the tunnel walls were turning the flow.

Total pressure surveys were made upstream and downstream of the cascade (see Fig. 10). These surveys at midspan were used to calculate the pressure loss coefficient for the vane set. About 25 locations across each blade gap and all blade gaps were utilized to determine the average exit total pressure. More than half of these locations were within the blade wake. In addition, the flow angle at the cascade exit was measured to determine if the vane set over-or-under turned the flow.

It should be noted that all design calculations necessarily assumed two-dimensional and periodical flow. The purpose of the experimental tests, on the other hand, was to simulate conditions in the actual wind tunnel. Since no wall boundary layer control was desired, the degree to which flow approached two-dimensionality depended upon the aspect ratio. The aspect ratio is reasonably large both for the scale model and for the full size tunnel.

The Lewis No. 1 and No. 3 vanes were designed at a Reynolds number of 6 million, which is the full scale wind tunnel operational Reynolds number. The 0.10 scale test cascade operates with a Reynolds number of 600 000. In this context, the test results should be looked upon as a conservative estimate of the full scale performance. Boundary layer calculations were performed at both Reynolds numbers and the calculations, in all cases, predict an improvement in the full scale model with respect to the results for the 0.10 scale model.

Boundary layer transition and separation locations were determined by oil flow visualization techniques. Their interpretation can be very subjective, but they do provide a qualitative check on boundary layer calculation methods, and therefore on the design philosophy. Of more direct interest and relevance is the measure of blade section loss.

4. EXPERIMENTAL RESULTS AND COMPARISONS WITH PREDICTIONS

The 0.10 scale model experimental and theoretical results for the Lewis No. 1 design will be discussed first, followed by that for the Lewis No. 3 design. A summary of the losses in total pressure from the boundary layer analysis of Lewis No. 3 for both scale model and full size blades concludes this section. There, comparisons with the scale model data are discussed along with implications for the full scale performance.

Lewis No.1 Design

40 x 80 mode. There were no surface pressure taps available for the initial tests of the Lewis No. 1 design. However at an inlet air angle of 0°, the measured loss coefficient was only 0.029, an excellent level. This compares favorably with a predicted value of 0.038 from Speidel's loss calculation method (8). The measured exit air angle was -2° (overturned) which matches the design value.

80 x 120 mode. The chordwise distribution of static pressure coefficient, C_p, is shown in Fig. 11 for an inlet air angle of 45°. Measurements on the upper and lower surfaces of three adjacent vanes are indicated as is the panel code prediction. There is a good match between the data and the prediction over the forward half-chord. The flattening out of the data on the upper surface over the aft one-third chord is indicative of boundary layer separation. This was in fact confirmed by oil flow visualization tests which indicated that the turbulent boundary layer separated near 64 percent chord on the upper surface. There was no oil flow evidence of turbulent layer separation on the lower surface.

As tabulated in Fig. 11 for an inlet air angle of 45°, the measured total pressure loss coefficient was 0.13. This loss is higher than desired but acceptable in these scale model (low Reynolds number) tests. The exit air angle was very good, within 1° of that desired.

Also tabulated in Fig. 11 are the loss coefficients and exit air angles for inlet air angles of 50° and 55°. The loss coefficient increased sharply, to 0.27 and 0.74 at these inlet angles. Such loss levels are not acceptable and the Lewis No. 3 design was developed specifically to improve on this off design performance at 50° and 55°. Oil flow visualization indicated the entire upper surface was in reverse flow at 55°.

Lewis No.3 Design

40 x 80 mode. The C_p distribution is shown in Fig. 12 for an inlet air angle of 0°. There is a good match between the data and the prediction except over the forward 20 percent chord on the lower surface. Here there are increasingly more negative C_p values peaking near the leading edge. This is similar to what a more negative inlet air angle than 0 might show on the lower surface but there is not a corresponding change in the C_p data on the upper surface. The effect of this unexplainable early peak was a higher than expected loss coefficient of 0.076. The oil flow visualization results and the C_p data do not indicate any turbulent boundary layer separation on either surface, and the oil flow on the lower surface did not show a change in character of the boundary layer.

The boundary layer calculation for the lower surface, however, predicted transition at 47 percent chord and possible turbulent separation at 81 percent chord, depending upon how conservatively the critical incompressible form factor is selected. But oil flow visualization showed no change in character of the boundary layer over the entire lower surface. This would be consistent with an early transition to a turbulent boundary layer due to the high negative pressure coefficient, and a turbulent layer extending over the entire lower surface. An early transition to a turbulent boundary layer would produce a larger boundary layer thickness at the trailing edge, and consequently a higher loss than anticipated and close to the level measured. Also tabulated in Fig. 12 is the performance at inlet air angles of -5° and 5° where the loss coefficients are 0.10 and 0.065, respectively.

80 x 120 mode. The chordwise distribution of C_p is shown in Fig. 13 for an inlet air angle of 45°. Again there is a good match between the data and the prediction over the forward half-chord. There also is a modest delay in the flattening out of the data on the upper surface compared to the Lewis No. 1 results (Fig. 11). This too was also confirmed by oil flow visualization tests which indicated that the turbulent boundary layer separated near 69 percent chord on the upper surface, further aft than the Lewis No. 1 results. There was no oil flow evidence of turbulent layer separation on the lower surface.

As tabulated in Fig. 13 (45° inlet air angle), the measured loss coefficient was 0.12, about 10 percent less than for the Lewis No. 1 design. The change in C_p minimum from -5.4 for Lewis No. 1 (Fig. 11) to -3.2 for Lewis No. 3 (Fig. 13) has reduced the adverse pressure gradient on the upper surface. This in turn has decreased the boundary layer growth and reduced the losses. The exit air angle of 1.5° for Lewis No. 3 was also good.

The improved off-design response of the Lewis No. 3 design in the 80 x 120 mode is shown in Fig. 14. Here experimental C_p distributions (faired lines from discrete data points for clarity) for inlet air angles of 50° and 55° are shown along with the 45° value just discussed. Loss coefficients and exit air angles for these varied inlet angles are also tabulated.

The pressure gradient on the upper surface becomes more adverse as inlet air angle is increased. However, this effect is primarily in a region near the leading edge. There is little difference in C_p over the remaining chord between the 50° and 45° inlet angle cases and loss levels are the same, 0.12. This is less than half the loss for Lewis No. 1 at 50° (Fig. 11).

At 55° the loss for Lewis No. 3 was 0.17 compared to 0.74 for Lewis No. 1. Although not shown on Fig. 14, the loss at 60° for Lewis No. 3 was 0.29. In all cases the exit air angle was close to axial as desired. Oil flow visualization indicated turbulent layer separation near 70 percent chord for 45°, 50°, and 55° for Lewis No. 3. This is a reasonable agreement with the start of flattening of the upper surface C_p curves.

<u>Summary of Losses in Total Pressure for Lewis No. 3 Design</u>

There are three parts to Table 1 summarizing the Lewis No. 3 performance results as follows: (a) 0.10 scale experimental, (b) 0.10-scale theoretical, and (c) full-scale theoretical. The chordwise locations of transition from laminar to turbulent boundary layer and turbulent layer separation are given along with loss coefficients. This information is organized by operating mode, inlet flow angle, and blade surface.

<u>40 x 80 mode</u>. There is a significant difference in the measured loss coefficient of 0.076 from that predicted of 0.042 for the 0.10-scale model. This is believed due to the premature transition on the lower surface near the leading edge instead of at the 47 percent chord location predicted. The premature transition is due directly to the difference in lower surface pressure distribution between measurement and calculation. The reason for the difference in pressure distribution is not evident. The predicted loss for the full scale operation is the same as for the 0.10 scale model. This is coincidental since the assumed transition locations are considerably different, and the Reynolds numbers differ.

<u>80 x 120 mode</u>. A comparison of the experimental with the theoretical results for the 0.10 scale model in the 80 x 120 mode indicates the experimental separation location on the upper surface for the turbulent layer is about 70 percent compared to theoretical values of 76 percent, irrespective of inlet air angle from 45° to 55°. Also, there is agreement that no turbulent layer separation occurs before the trailing edge on the lower surface. Corresponding comparisons of transition locations generally agree within about 10 percent. Prediction of loss coefficient with the Speidel method is not accurate when significant regions of boundary layer separation exist (incompressible form factor > 2.0). Such separation does occur for all operation in the 80 x 120 mode. However, to indicate the trend, a loss coefficient of 0.093 is presented for the minimum air angle of 45° in the 80 x 120 mode and is about 20 percent less than the measured value of 0.12.

Of greater significance is the predicted loss at this condition for the full scale application. Here, because of the beneficial effects of a much thinner boundary layer with a 10-fold increase in chord Reynolds number and the resulting movement of separation further aft (see Table 1(c)), the loss is reduced by almost a factor of two. Note also this full scale calculation is made with an early trip to turbulent flow which may not occur on both surfaces. Although the actual magnitude of the loss reduction due to scale may not be as high as 100 percent, the trend to a lower loss value in full scale is believed reliable. Thus the full scale loss coefficient in the 80 x 120 mode for inlet angles of 45° and 50° is expected to be less than 0.10 and that at 55° less than 0.15.

5. SUMMARY

A fixed geometry low speed guide vane set has been designed with an inverse design code, analyzed with panel method and integral boundary layer codes and tested at 0.10 scale. The set is to operate in two different modes in the 40 x 80/80 x 120 foot NASA Ames wind tunnel which requires inlet air angles from about -5 to +55°. In both modes the exit flow is aligned with the axes of the drive fans.

The experimental tests were conducted on a cascade of 0.10 scale models with a chord-based Reynolds number of 600 000. Measured losses in total pressure (expressed as percent of inlet dynamic head) were 17 percent or less over a range of inlet air angles from -5 to +55° while the exit air angles remained within 1.5° of axial. Losses are expected to be even less in the full scale application because of the positive influence of higher Reynolds number.

6. ACKNOWLEDGMENT

Many people, too numerous to mention, from two NASA laboratories contributed to the design and experimental evaluation of the new guide vanes. The authors wish to acknowledge the efforts of members of the Low Speed Aircraft Research Branch and Low Speed Wind Tunnel Investigations Branch from NASA Ames Research Center; and members of the Fan and Compressor Branch and Computational Fluid Mechanics Branch from NASA Lewis Research Center.

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TABLE I. - SUMMARY OF LOSSES IN TOTAL PRESSURE FOR LEWIS NO. 3 DESIGN

			(a) 0.1 - scale experimental			(b) 0.1 - scale theoretical			(c) full-scale theoretical		
			Percent chord		Loss	Percent chord		Loss	Percent chord		Loss
Operating mode	Inlet flow angle, degs.	Blade surface	Trans.	Turb. sep.	coeff.	Trans.	Turb. sep.	coeff.	Trans.	Turb. sep.	coeff.
40 x 80	0	Upper	66	100	0.076	62	80	0.042	Forced	85	0.042
		Lower	Near L.E.	100		47	81		Near L.E.	100	
80 x 120	45	Upper	No visual	69	0.12	5	76	0.093	↓	82	0.052
		Lower	59	100		53	100			100	
/	50	Upper	1, small bubble	72	0.12	1	76	-----		82	-----
		Lower	59	100		53	100			100	
	55	Upper	1, 1/4-inch bubble	69	0.17	1	76	-----		82	-----
		Lower	37-59	100		53	100			100	

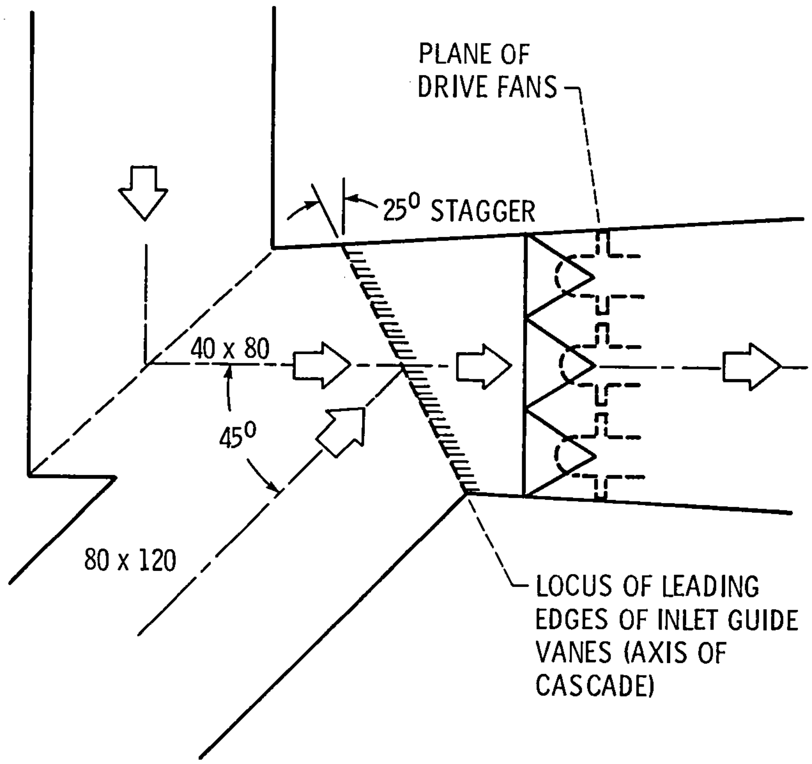


Figure 1. - Schematic of the junction of the NASA Ames 40 x 80/ 80 x 120 foot wind tunnel.

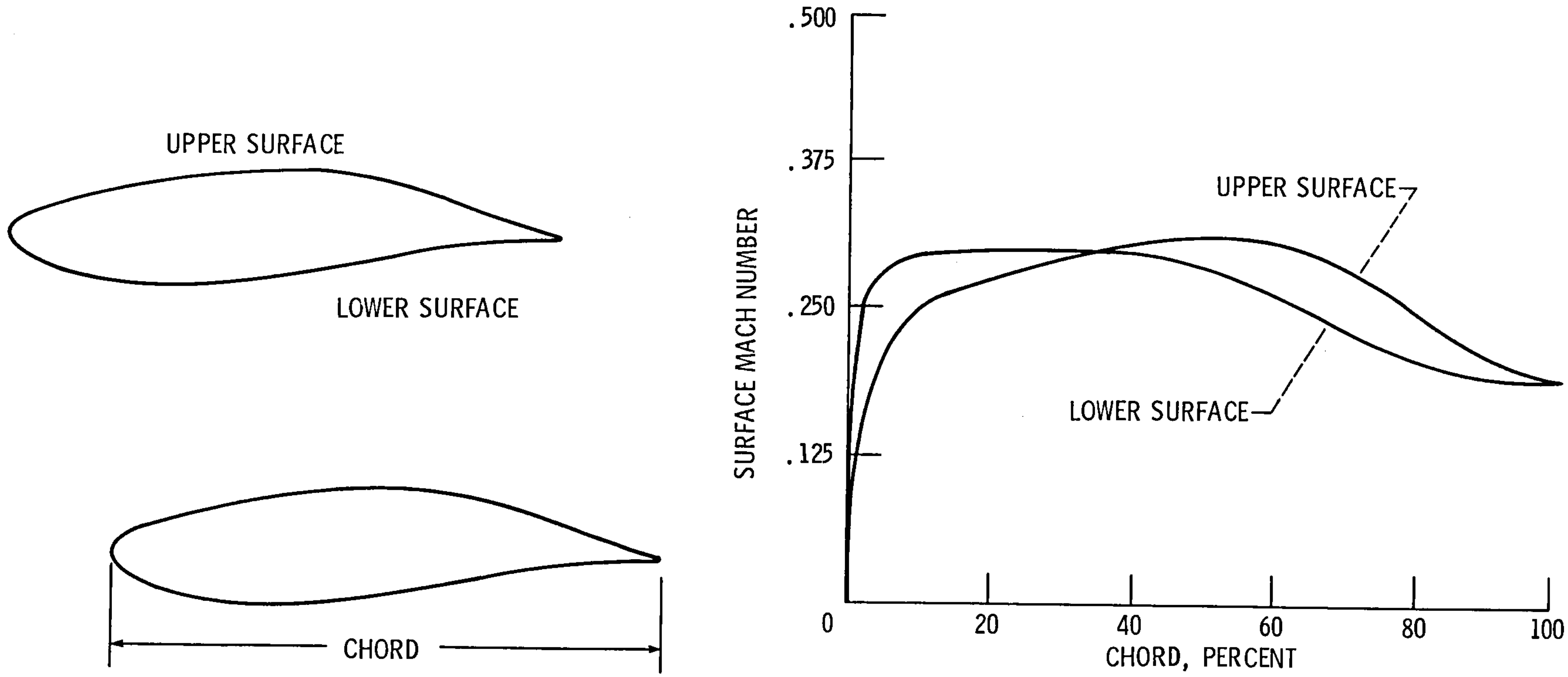


Figure 2. - Lewis number 1 design.

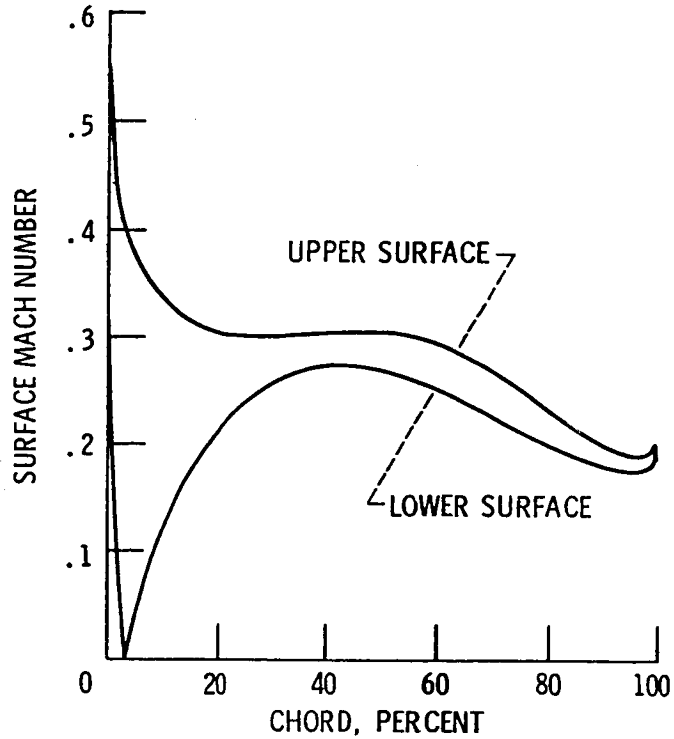


Figure 3. - Analysis of Lewis number 1 design operating in 80 x 120 mode, inlet air angle of 45^0.

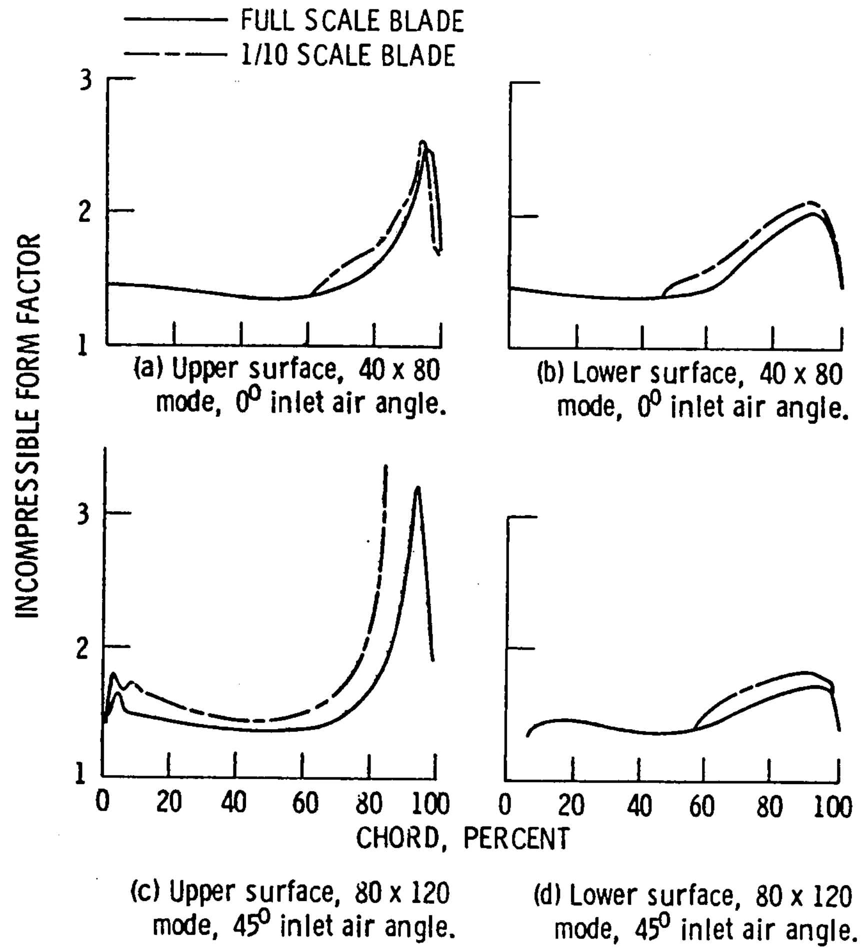


Figure 4. - Theoretical boundary layer performance for Lewis number 1 in both operating modes.

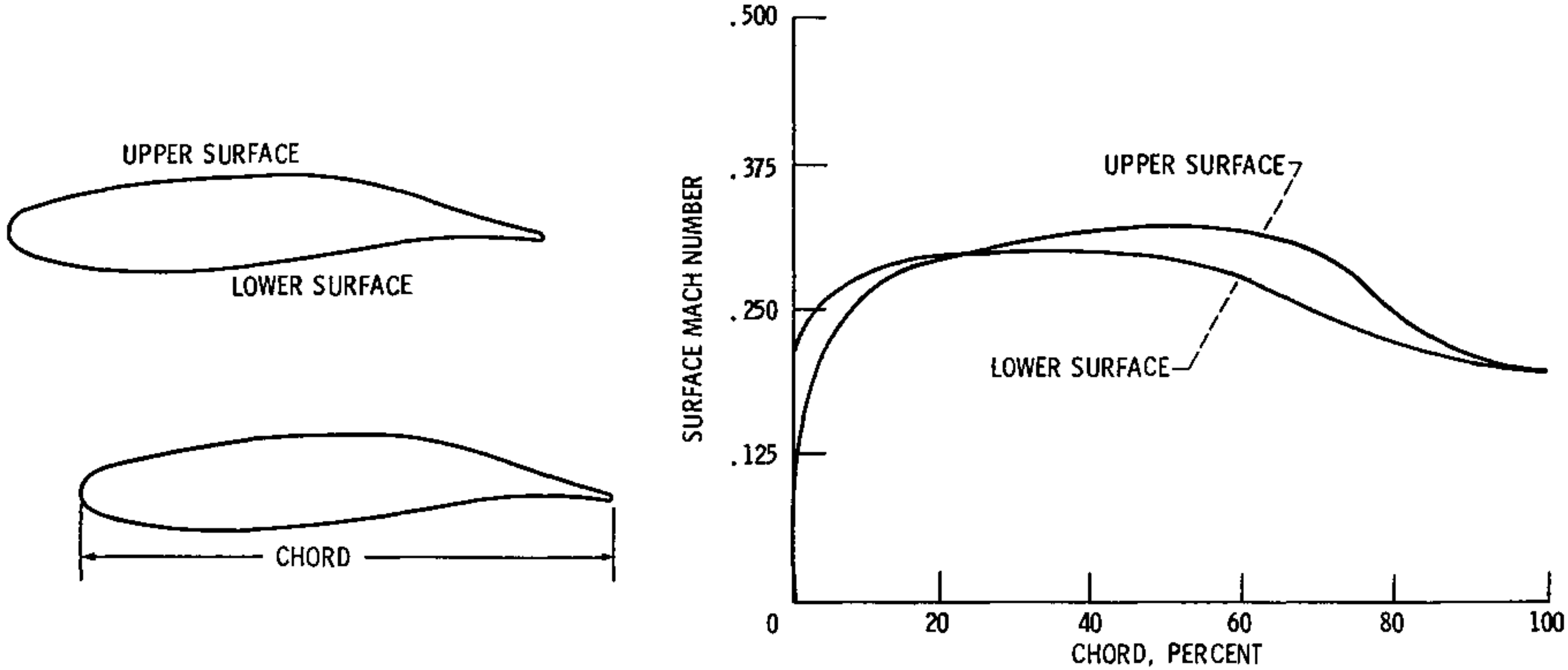


Figure 5. - Lewis number 3 design.

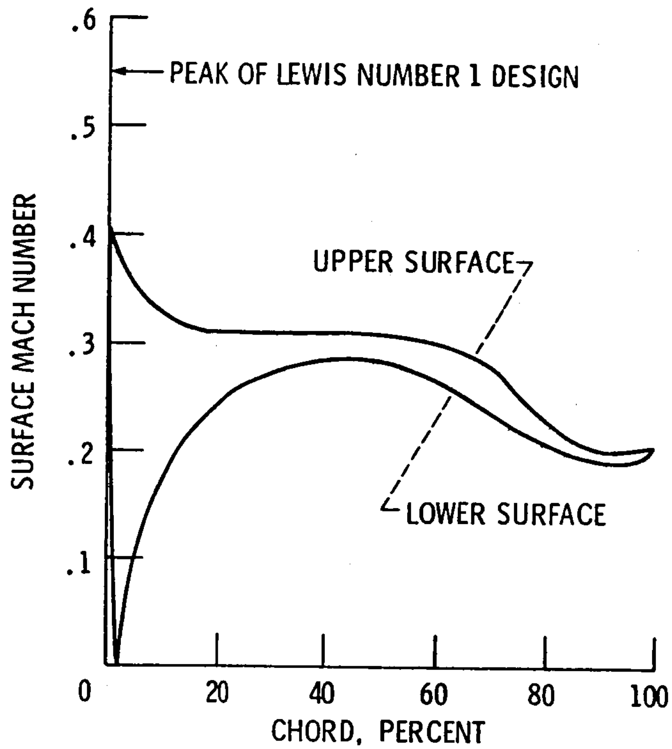


Figure 6. - Analysis of Lewis number 3 design operating in 80 x 120 mode, inlet air angle of 45°.

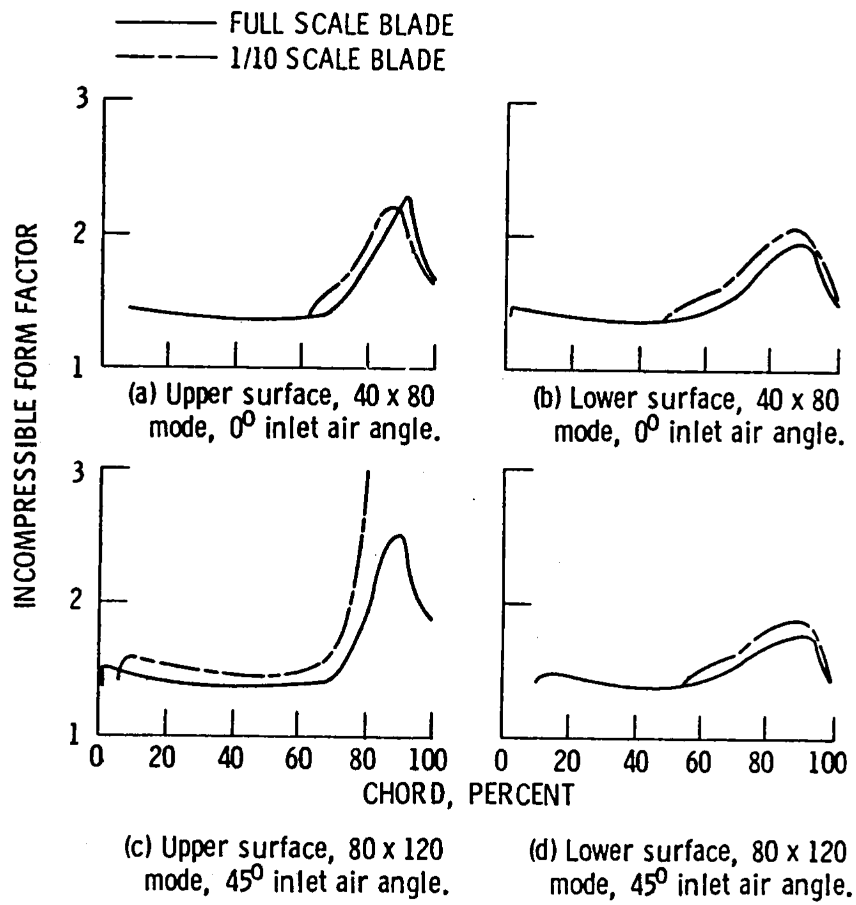
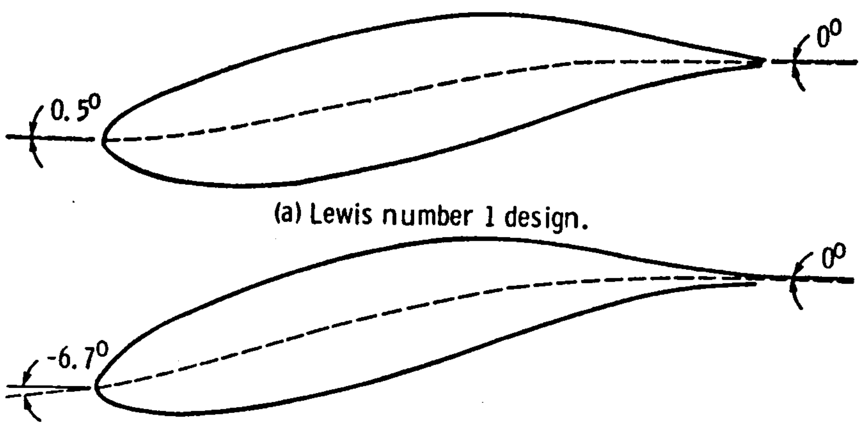


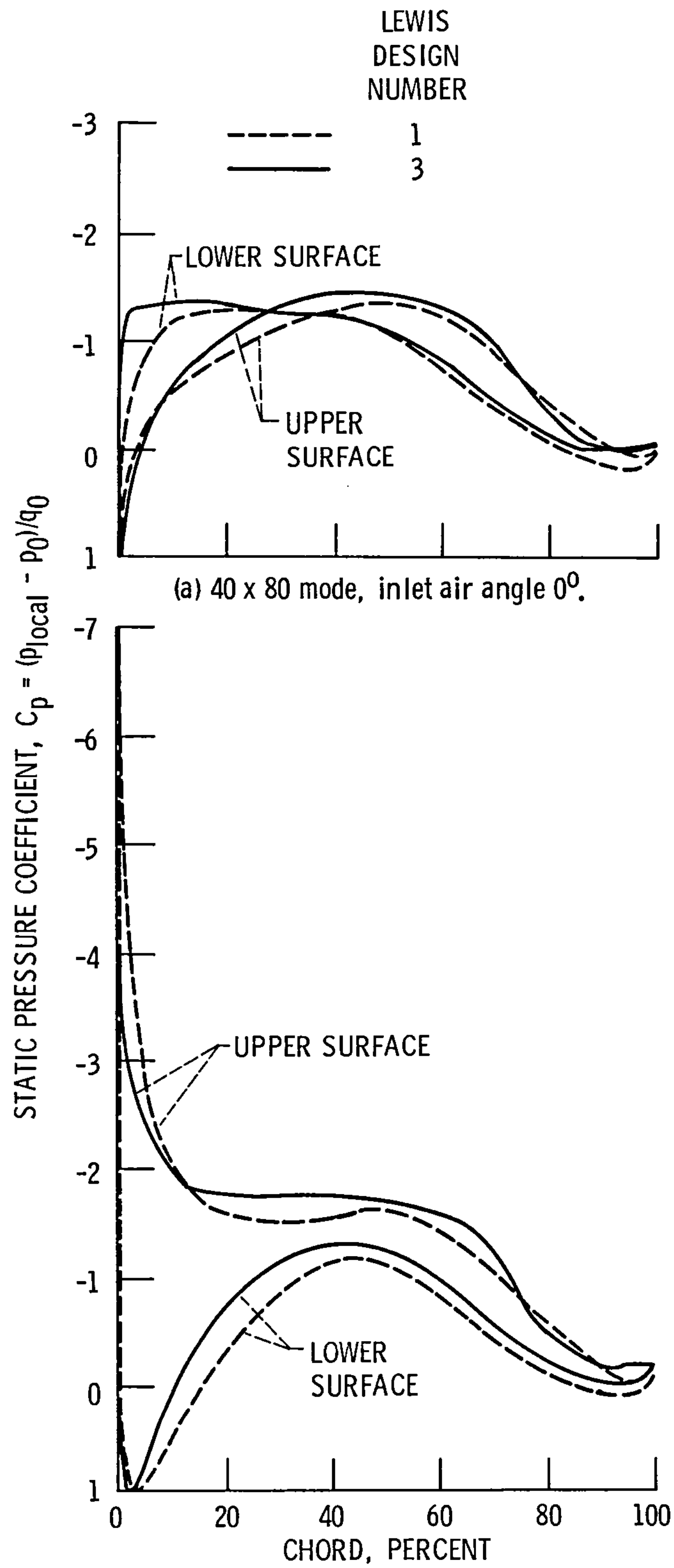
Figure 7. - Theoretical boundary layer performance for Lewis number 3 in both operating modes.



(a) Lewis number 1 design.

(b) Lewis number 3 design.

Figure 8. - Blade shapes and mean camber lines for two designs with edge angles relative to drive fan axes.



(a) 40 x 80 mode, inlet air angle 0°.

(b) 80 x 120 mode, inlet air angle 45°.

Figure 9. - Comparison of theoretical surface pressure distributions for Lewis number 1 and number 3 designs in both operating modes.

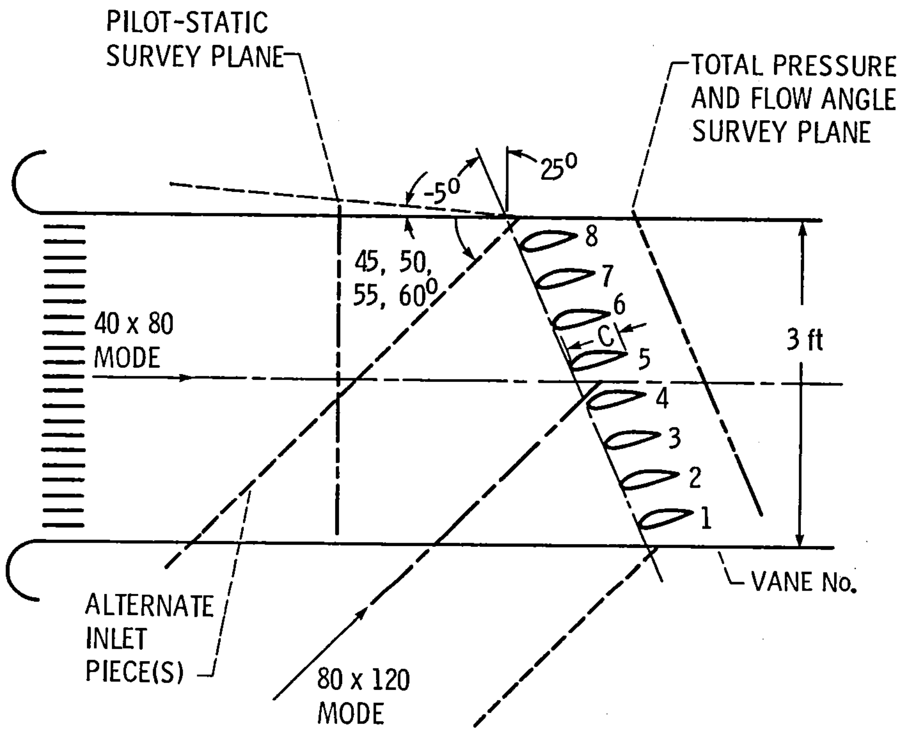


Figure 10. - Schematic of 0.1-scale test facility.

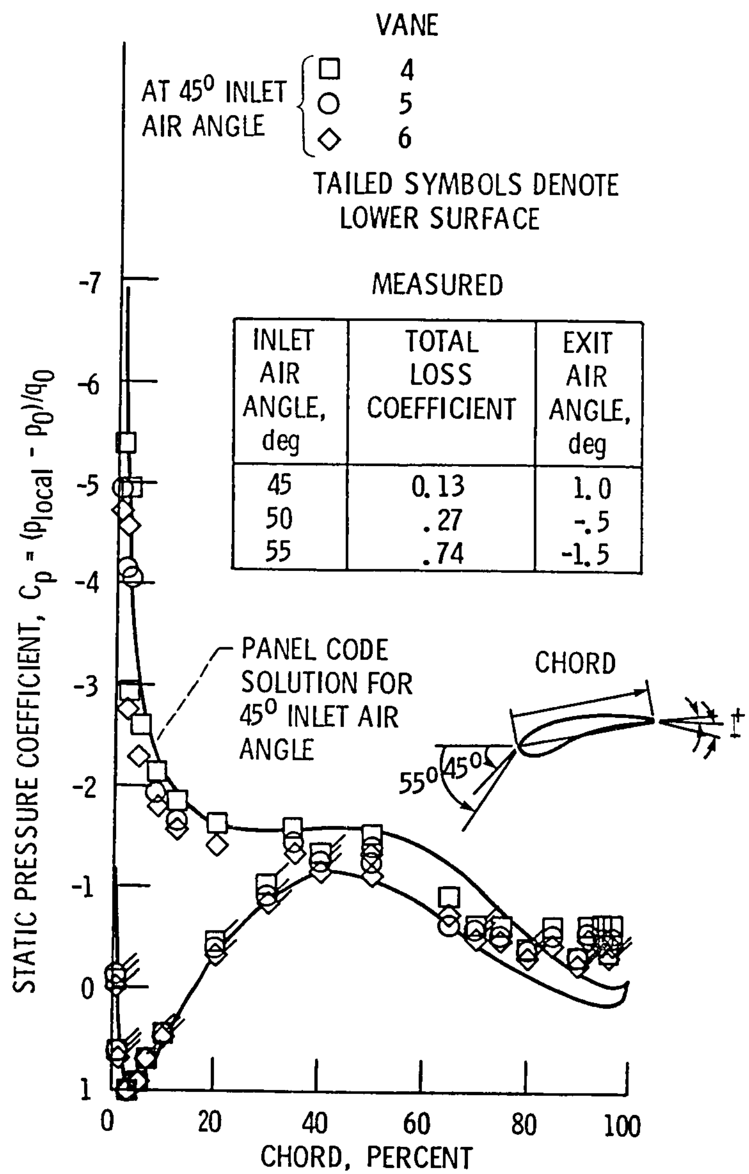


Figure 11. - Experimental surface pressure distributions compared to theoretical for Lewis number 1 design in the 80 x 120 operating mode.

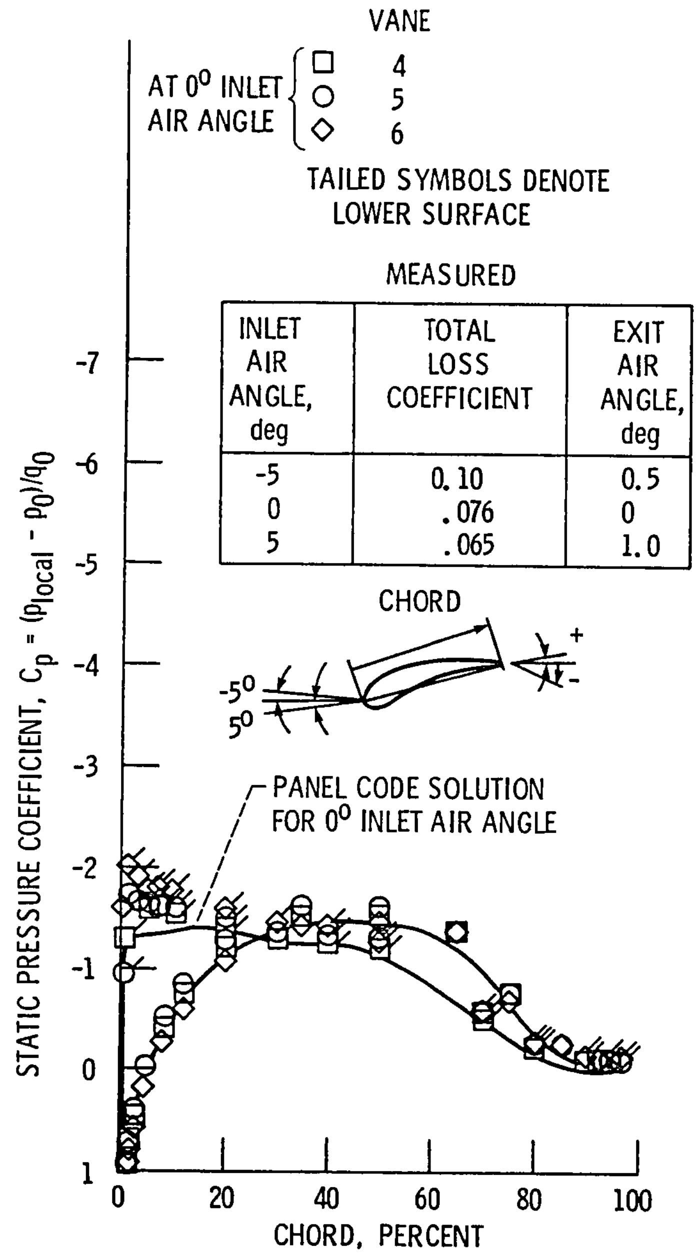


Figure 12. - Experimental surface pressure distributions compared to theoretical for Lewis number 3 design in 40 x 80 operating mode.

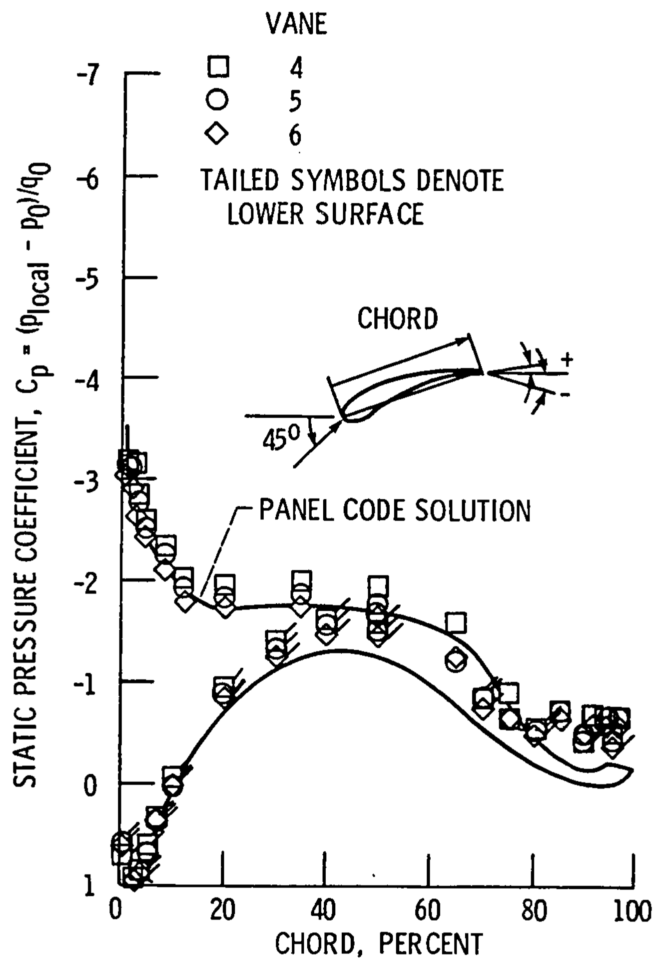


Figure 13. - Experimental surface pressure distributions compared to theoretical for Lewis number 3 design in the 80 x 120 operating mode with inlet air angle of 45°. Total loss coefficient, 0.12; exit air angle, 1.5°.

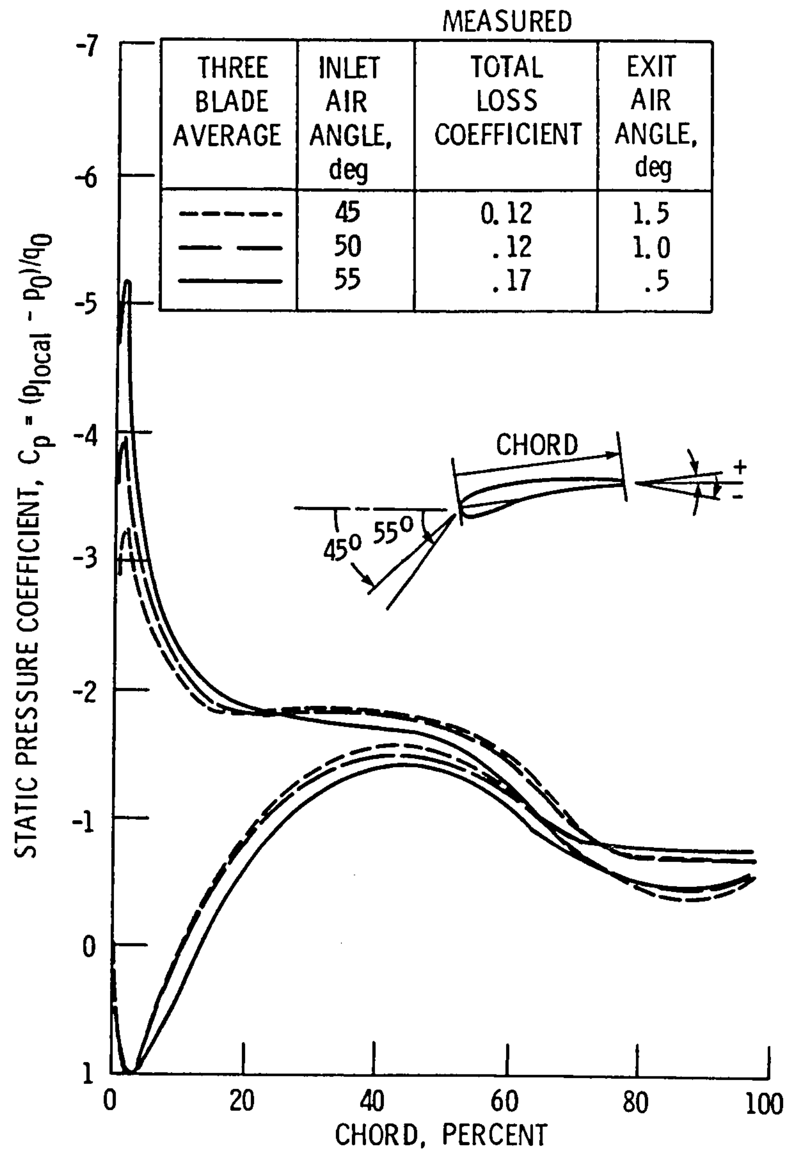


Figure 14. - Comparison of experimental surface pressure distributions for Lewis number 3 design at three inlet air angles in the 80 x 120 operating mode.

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA TM-83519		

4. Title and Subtitle

Design and Performance of a Fixed, Nonaccelerating, Guide Vane Cascade That Operates Over an Inlet Flow Angle Range of 60°

5. Report Date

6. Performing Organization Code

505-31-0A

7. Author(s)

José M. Sanz, Eric R. McFarland, Nelson L. Sanger, Thomas F. Gelder, and Richard H. Cavicchi

8. Performing Organization Report No.

E-1868

10. Work Unit No.

9. Performing Organization Name and Address

National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

11. Contract or Grant No.

13. Type of Report and Period Covered

Technical Memorandum

12. Sponsoring Agency Name and Address

National Aeronautics and Space Administration
Washington, D.C. 20546

14. Sponsoring Agency Code

15. Supplementary Notes José M. Sanz, Research Scientist presently employed by Universities Space Research Association; Eric R. McFarland, Nelson L. Sanger, Thomas F. Gelder, and Richard H. Cavicchi, NASA Lewis Research Center. Prepared for the Twenty-ninth Annual International Gas Turbine Conference sponsored by the American Society of Mechanical Engineers, Amsterdam, The Netherlands, June 3-7, 1984.

16. Abstract

A unique set of wind tunnel guide vanes are designed with an inverse design code and analyzed with a panel method and an integral boundary layer code developed at the NASA Lewis Research Center. The fixed guide vanes, 80 feet long with 6-foot chord length, were designed for the NASA Ames 40 x 80/80 x 120 ft Wind Tunnel. Low subsonic flow is accepted over a 60° range of inlet angle from either the 40 x 80 leg or the 80 x 120 leg of the wind tunnel, and directed axially into the main leg of the tunnel where drive fans are located. Experimental tests of 1/10-scale models were conducted to verify design calculations.

17. Key Words (Suggested by Author(s))		18. Distribution Statement
Design	Performance	Unclassified - unlimited
Guide vanes	Wind tunnel	STAR Category 02
Cascade	Boundary layer	

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of pages	22. Price*
Unclassified	Unclassified		

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