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An Inverse Method For The Aerodynamic Design of Three-Dimensional Aircraft Engine Nacelles

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R. A. Bell R. D. Cedar

GE Aircraft Engines
One Neumann Way
Cincinnati, OH 45215

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Abstract

A fast, efficient and user-friendly inverse design system for three-dimensional nacelles has been developed. The system is a product of a two-dimensional inverse design method originally developed at NASA Langley Research Center (LaRC) and the CFL3D analysis code which was also developed at NASA LaRC and modified at GEAE for nacelle analysis. The design system uses a predictor/corrector design approach in which an analysis code is used to calculate the flow field for an initial geometry, the geometry is then modified based on the difference between the calculated and target pressures. A detailed discussion of the design method, the process of linking it to the modified CFL3D solver and its extension to three-dimensions is presented in this paper. This is followed by a number of examples of the use of the design system for the design of both axisymmetric and three-dimensional nacelles.

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Introduction

The purpose of a nacelle, on a high bypass ratio turbofan engine is to supply the airflow required by the engine efficiently with low distortion levels, provide a low drag aerodynamic enclosure for the engine hardware, and expand the exhaust gasses from the engine through an exhaust system with maximum efficiency. The nacelle has three major components, the inlet, the fan cowl and the exhaust system (Figure 1). The nacelle's crown, side and keel cuts are also shown in Figure 1. In this paper a three-dimensional inverse design technique is presented for the aerodynamic design of the fan cowl of the nacelle. In this inverse design method, the designer analyzes an initial geometry and then interactively modifies the resulting pressure distribution to remove any undesirable features. The method then determines the nacelle geometry that will give the desired pressure distribution.

A diverse variety of inverse methods have been developed for airfoils and wing design. An overview of these methods can be found in review papers by Slooff [1] and Dulikravich [2]. Many of these methods, however, are often only suited to specific applications and are not easily extendible to meet the requirements of nacelle design. Examples of this are the Hodograph method of Bauer, Garabedian and Korn [3], which is limited to two-dimensional airfoil and turbomachinery flows and the fictitious gas method [4] which is only suitable for the design of transonic shock free flows.

Unfortunately, modern high bypass ratio turbofan engine nacelles are far from axisymmetric and to obtain a meaningful solution a fully three-dimensional analysis has to be performed. There are also geometric constraints imposed on the design process that the inverse method must be able to handle. In an aero engine, the fan nozzle acts as the throttle controlling the engine operating characteristics; therefore the fan nozzle area, and thus the radius of the trailing edge of the fan cowl, must remain fixed in the design process. Similar constraints apply to the nacelle inlet area, so that the radius of the leading edge of the fan cowl must also remain fixed during the design process. The aerodynamic designer, therefore has to design a surface between two fixed endpoints. In reality, the situation can be further constrained. The trend in the aero engine industry has been to produce derivative families of engines. When designing the nacelle of a derivative engine there can be large economic incentives to keep as much of the hardware common between the members of an engine family. This is especially true for complex components that have high initial tooling costs such as the inlet anti-icing system and the thrust reverser. In these cases, the aerodynamic designer may be limited to changes in the geometry between about 5% and 60% of the fan cowl length. The new surface having to blend smoothly with the existing hardware.

In this paper an inverse design technique is described that meets the needs of the fan cowl designer in that it is three-dimensional and allows either all or a portion of the fan cowl to be modified. The method incorporates the CFL3D [5] analysis code and the inverse design technique of Campbell and Smith [6,7].

Aerodynamic Code

CFL3D was developed by the Computational Fluid Dynamics Laboratory at NASA Langley Research Center [5] and modified at General Electric Aircraft Engines (GEAE) for nacelle analysis [8]. The modified code solves the Euler equations by using a finite volume discretization method. Solutions are advanced in time with a spacially-split three-factor approximate factorization method in diagonalized form. The flux quantities are represented using the flux-vector-splitting approach of Van-Leer with third-order spacial accuracy. Special features include multigrid convergence acceleration and the

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ability to handle multiple grid blocks with a variety of block interfaces and boundary conditions.

The Euler equations express the conservation of mass, momentum, and energy for an inviscid, non-conducting gas in the absence of external forces. The conservation form of the equations in generalized coordinates is:

$$\frac{\delta \hat{Q}}{\delta t} + \frac{\delta \hat{F}}{\delta \xi} + \frac{\delta \hat{G}}{\delta \eta} + \frac{\delta \hat{H}}{\delta \zeta} = 0$$

where

$$\hat{Q} = \frac{Q}{J} = \frac{1}{J}\begin{bmatrix} \rho \\ \rho u \\ \rho v \\ \rho w \\ e \end{bmatrix}$$

$$\hat{F} = \frac{1}{J}\begin{bmatrix} \rho U \\ \rho U u + \xi_x P \\ \rho U v + \xi_y P \\ \rho U w + \xi_z P \\ (e+P)U \end{bmatrix}$$

$$\hat{G} = \frac{1}{J}\begin{bmatrix} \rho V \\ \rho V u + \eta_x P \\ \rho V v + \eta_y P \\ \rho V w + \eta_z P \\ (e+P)V \end{bmatrix}$$

$$\hat{H} = \frac{1}{J}\begin{bmatrix} \rho W \\ \rho W u + \zeta_x P \\ \rho W v + \zeta_y P \\ \rho W w + \zeta_z P \\ (e+P)W \end{bmatrix}$$

The equations are non-dimensionalized in terms of the reference density ρ_∞ and the speed of sound a_∞. The cartesian velocity components are u,v,w in the x,y and z cartesian directions. The pressure, P, is related to the conserved variables, Q, through the ideal gas law:

$$P = \frac{(\gamma - 1)}{2}\left[e - \rho\left(u^2 + v^2 + w^2\right)\right]$$

where γ is the ratio of specific heats ($\gamma = 1.4$). U, V, and W are the contavariant velocity components in the ξ, η, and ζ directions respectively. J is the Jacobian of the transformation and e represents the internal energy.

The boundary conditions used consist of far-field, solid surface and fan face boundary conditions. The farfield boundary condition is based on Riemann invariants for a one-dimensional flow. On the solid surface of the nacelle and spinner, the velocity normal to the wall is set to zero and a slip condition is imposed. In order to simulate the exhaust plume, the fan cowl is extended downstream of the fan nozzle as a solid body. A full description of how this is achieved is given in reference 8. The intake flow rate is controlled by setting the fan face boundary condition (static pressure). The fan face static pressure required for a given flow rate is calculated using a one-dimensional flow equation. Since the pressure does not account for the three-dimensional effects and loss in the flowfield it is readjusted based on results from the one dimensional analysis. A symmetry boundary condition is also placed at the nacelles's vertical plane of symmetry when cross-wind or yaw requirements are not

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imposed. For these cases a 180 degree grid is used for the analysis. For flow conditions involving cross-wind or yaw, a full 360 degree grid is used with a continuous boundary condition specified at the vertical plane of geometric symmetry (Figure 2). An option is also available to analyze axisymmetric configurations that only requires one computational cell in the circumferential direction. This code has been validated against test data for a large range of nacelle designs and operating conditions. For further details on the modified version of CFL3D for three-dimensional nacelle analysis see reference 8.

Design Method

The predictor/corrector approach used by this design method is illustrated in Figure 3. A target pressure distribution is specified by the designer that has desirable characteristics such as no shocks, no steep diffusions or favorable pressure gradients for natural laminar flow. The aerodynamic analysis code (CFL3D) is used to determine the pressure distribution on an initial geometry. The nacelle surface pressure coefficients are compared with the target pressure distributions in the design module. The initial geometry is then modified based on the pressure differences. The grid is then perturbed and the new geometry is analyzed in the analysis code to determine its pressure distribution. This process continues until the convergence criteria specified by the designer is reached.

Design Algorithm

The design method, described in reference 6, uses two design algorithms, one for subsonic flow and the other for supersonic flow. The supersonic algorithm is blended with the subsonic algorithm to design regions of transonic flow. Both algorithms assume that ΔC_p is proportional to the change in geometry.

The subsonic algorithm is based on the assumption that changes in curvature are directly proportional to changes in pressure coefficient. The relationship used to express the change in curvature as a function of change in pressure coefficient is:

$$\Delta C = \Delta C_p A(1 + C^2)^B$$

where

C is the curvature
C_p is the pressure coefficient
A = +1 for the upper surface, -1 for the lower surface
B = input constant ranging from 0.0 to 0.5

The derivation of this equation is given in reference [6]. The change in curvature is converted to a change in r'' by using the formula

$$\Delta r'' = \Delta C \left[1 + (r')^2\right]^{1.5}$$

where

r is the surface radius
r' is the surface slope
r'' is the second derivative of surface radius

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This equation assumes that the changes in the surface slope are small.

The supersonic algorithm is based on supersonic thin airfoil theory. Based on relations between the pressure coefficients and surface slope the expression

$$\Delta r' = K \Delta C_p$$

can be derived [6]. Differentiating this expression gives the following relationship between r'' and ΔC_p.

$$\Delta r'' = K \frac{d(\Delta C_p)}{dx}$$

The value for the constant K is 0.05 and is used to under relax the changes in the geometry during each design iteration.

Using these equations the required change in r'' is calculated at each point along the fan cowl. To change the magnitude of r'' at point I without changing r'' at other locations, points I+1 through N are rotated through a given angle. Figure 4 shows the result of this process.

Closure of the Design Range

It is clear from Figure 4 that in general the last point in the design range will not remain fixed and therefore a method of closing the geometry is required. The method suggested in reference 6 was to rotate the newly designed section about the most forward point of the design range so that the end of the design range closes. This process, however, leaves a surface slope discontinuity at the beginning of the design range. If the beginning of the design range was the nacelle's leading edge, then an option would be to blend a new leading edge geometry into the modified nacelle. This was felt to be undesirable as a nacelle's off-design (takeoff, climb etc.) performance is critically dependent on the leading edge shape. An alternative would have been to smooth the geometry in the region of the slope discontinuity but there is no guarantee that the resulting pressure distribution in this region will be smooth.

A solution of this problem was found in a paper by Lin etal. [9] where they advocate modifying the target pressure distribution to ensure that the end point of the design range remains fixed. In this method a sine function is added to the target pressure with the maximum modification at the center of the design region, and zero at the ends (Figure 5). The amplitude of the sine function is iteratively determined by using the secant method. Figure 6 shows the logic used to close the geometry by modifying the target pressure distribution. This process is performed at each design update and it has been found that this scheme normally converges in three iterations.

Coupling the Design to the Analysis Code

For ease of use, the design algorithm has been incorporated as a module in a modified version of the CFL3D analysis code. The information that is passed from the design module to CFL3D is an updated computational grid that reflects the changes in surface geometry calculated by the design algorithm. Rather than regridding the complete configuration every design calculation, a grid perturbation scheme has been developed. In this scheme the grid lines along the fan cowl surface are moved radially to account for the change in surface geometry. This is repeated for the grid line away from the nacelle surface but the change in radius is reduced linearly with the local radius, so that the outer boundary does not move.

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The normal procedure for designing a nacelle is that the designer analyzes a first guess at the nacelle geometry using the CFL3D code. An interactive graphic program has been written that presents the designer with the nacelle surface Mach number distribution and allows the designer to specify the portion of the surface to be modified. The designer can interactively alter the Mach number distribution in order to obtain the desired characteristics for the target pressure distribution. CFL3D is then run with the design option active. The converged solution from the initial nacelle geometry can be used as the starting solution. The difference between this solution and the required target pressure distribution is used by the design module to calculate a new geometry. The grid is perturbed and the geometry is re-analyzed. Numerical studies have shown that after each pass through the design module the analysis does not have to be fully converged. It has been found that only 40 iterations of CFL3D are needed, where as, 250 iterations would be required for full convergence. About 20 passes through the design calculation are needed to obtain a pressure distribution that matches the modified target pressure distribution to engineering accuracy for a typical design. Convergence is slowed if the original geometry has large supersonic patches with strong shocks or if the designer is making large changes to the pressure distribution.

Axisymmetric Results

Because the design method does not account for three-dimensional (circumferential) effects, the first test cases that were run were purely axisymmetric. The results of two of these runs are presented in Figures 7 and 8. The design range for both of these cases is the complete length of the fan cowl. In the first test case the the inverse design method was used to eliminate a shock on the fan cowl as shown in Figure 7a. A comparison of the final pressure distribution and the initial and modified target pressure distributions are shown in Figure 7b. The final pressure distribution matches the modified target distribution almost perfectly. As shown, the difference between the initial and modified target pressure distributions, is quite small. Figure 7c shows a comparison of the initial and final geometry.

The same geometry is used in the second test case (Figure 8) but larger changes are being made to the pressure distribution. In this case a significant change in the target pressure distribution is required to close the geometry but the characteristics of the final pressure distribution are still similar to the designers intent. Figure 8b shows that the final pressure distribution matches the modified target distribution quite well. At the trailing edge, however, there is a small difference because the geometry downstream of trailing edge is fixed during the design, resulting in a discontinuity in surface slope and curvature. This highlights the problem of how one specifies a pressure distribution that ensures that the geometry at the end points of the design range match and the pressure distribution remains smooth.

Three-Dimensional Extension

Having shown that the axisymmetric inverse design code works well, the next stage was to extend it to the design of three-dimensional nacelles. In the axisymmetric version only one radial cut is considered. For three-dimensional nacelles, the radius varies from crown to keel and so a number of circumferential cuts must be taken into account during the design process.

Three options were considered for the three-dimensional version. The first option requires the designer to specify the target pressure distribution at each circumferential cut of the grid (typically 13 cuts are used on a 180-degree sector). The problem with this approach is that there is no guarantee of a

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smooth geometry in the circumferential direction (Figure 9). Thus, the designer would have to know how to distribute the pressure distribution circumferentially to ensure a smooth geometry. It was felt that this would be difficult to achieve, and therefore this option was rejected.

The second option requires the designer to specify the target pressure distribution on only three cuts (crown, side and keel). The design procedure would be used on only these cuts and the remaining cuts would be designed by parabolically interpolating the new radii circumferentially. The problem with this method is that the original cross-sectional geometric shape of the nacelle is not preserved during the design. As shown in Figure 10, the elliptic shape of the original nacelle cross-section is altered to a parabola by the interpolation scheme.

The third option considered was very similar to the second one. With this option the designer specifies the target pressure distribution on the crown, side and keel cuts but the remaining cuts are designed by parabolically interpolating the changes in the radii from these three cuts (Figure 11). With this approach the essence of the original cross-sectional shape of the nacelle is preserved and some degree of smoothness in the circumferential direction is ensured.

The third option was chosen to be used in the three-dimensional version of the inverse design method. As stated before, the designer specifies the target pressure distribution on the crown, side and keel cuts, of the nacelle. At each design iteration all three cuts are redesigned using the same design method as had been used in the axisymmetric version. No attempt has been made to extend the relationship between change in C_p and change in geometry to account for three-dimensional (circumferential) effects. The changes in geometry from the three cuts are then interpolated for the other radial cuts and the grid is perturbed in a similar manner to the axisymmetric version. Experimentation has shown that although changes made in the crown cut, for instance, do effect the flow at the side cut and to a lesser extent the keel cut, these effects do not cause instabilities in the design scheme.

Three-Dimensional Results

The results for a three-dimensional test case are shown in Figures 12 through 14. In this case the design range started at the nacelle leading edge and ended 10 inches upstream of the nacelle trailing edge. The Mach number distributions along the crown, side and keel cuts of the original nacelle as well as the desired target Mach number distribution are shown in Figure 12. Figure 13 shows the Mach number distribution achieved after 40 design iterations and the initial and final target Mach number distributions. The resulting modifications to the geometry is shown in Figure 14. It should be noted that the vertical scale has been expanded so that the change in geometry can be clearly seen.

Summary

A predictor corrector design method originally developed by Campbell and Smith has been coupled to a modified version of the CFL3D analysis code and extended to allow the design of three-dimensional nacelles. A designer can interactively modify the Mach number distribution of the crown, side and keel cuts of a fan cowl and the required geometry is automatically calculated. The method is capable of designing any local region of the fan cowl, the remainder being fixed, although further work is required in determining how to specify the pressure distribution so that both the geometry and pressure distribution are smooth at the end points of the design range.

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Further work is being pursued to try and reduce the computational time required by the method. The aim is to reduce the cost of an inverse design calculation from the present value of about four times that of an analysis calculation to about twice. Future work on the choice of an optimum pressure distribution that meets both geometric constraints and off-design performance criteria is also being considered.

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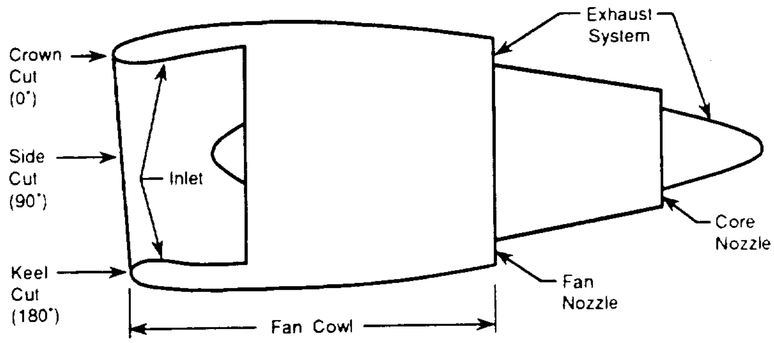

Figure 1. The Aerodynamic Components of a Nacelle.

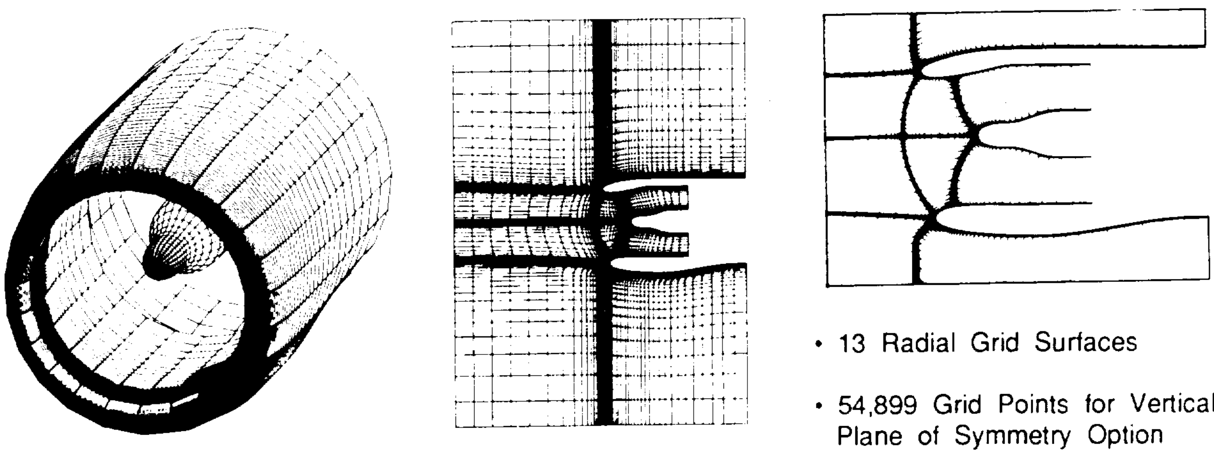

Figure 2. Typical Computational Grid for Nacelle Design.

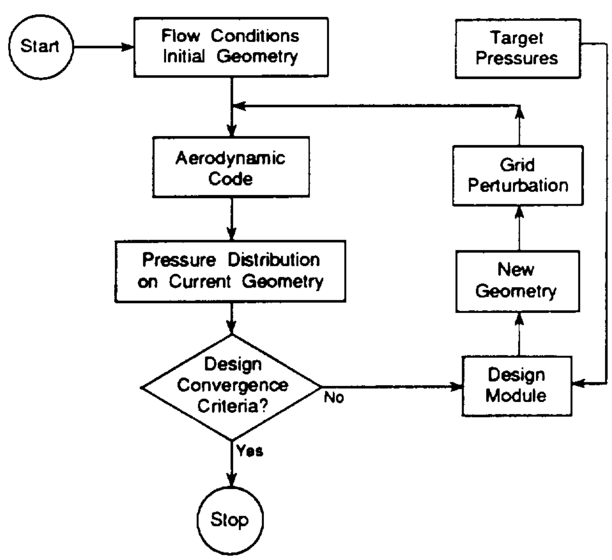

Figure 3. Flow Diagram of Automated Predictor/Corrector Design Method.

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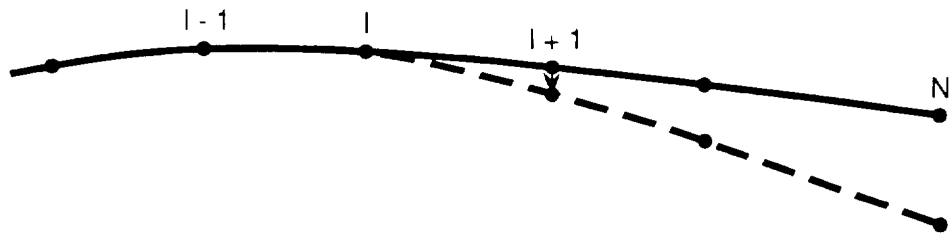

Figure 4. Method of Changing Curvature at One Point Only.

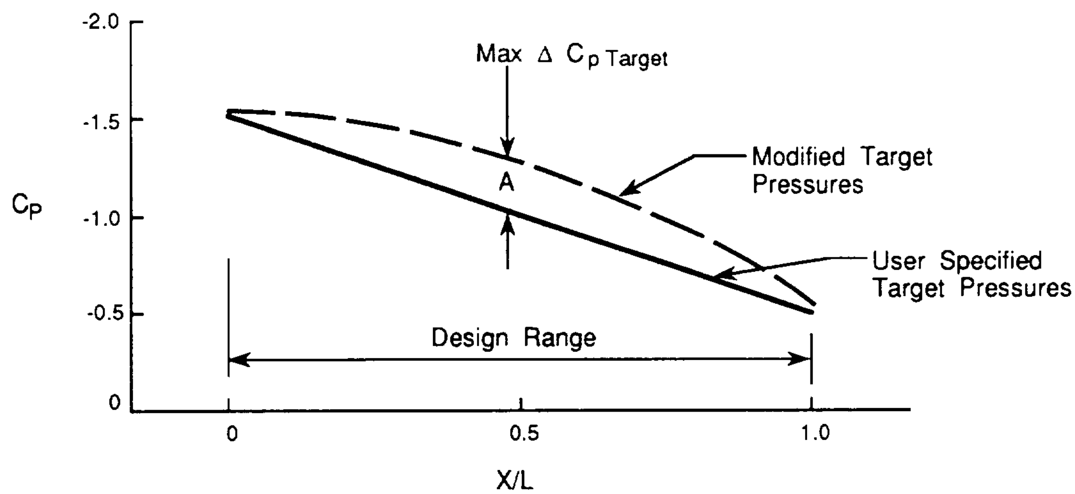

Figure 5. Modification of Target Pressure Distribution by a Sine Function.

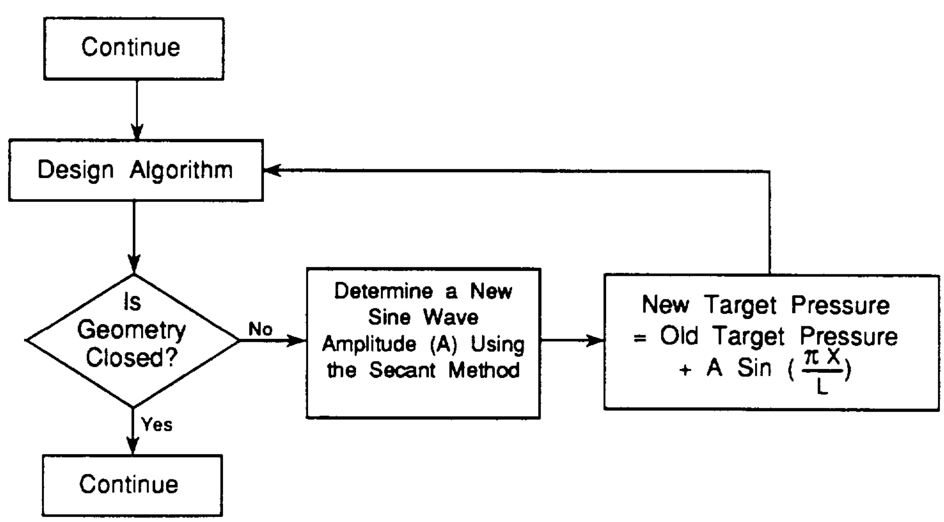

Figure 6. Logic Used to Close the Geometry by Modifying the Target Pressure Distribution.

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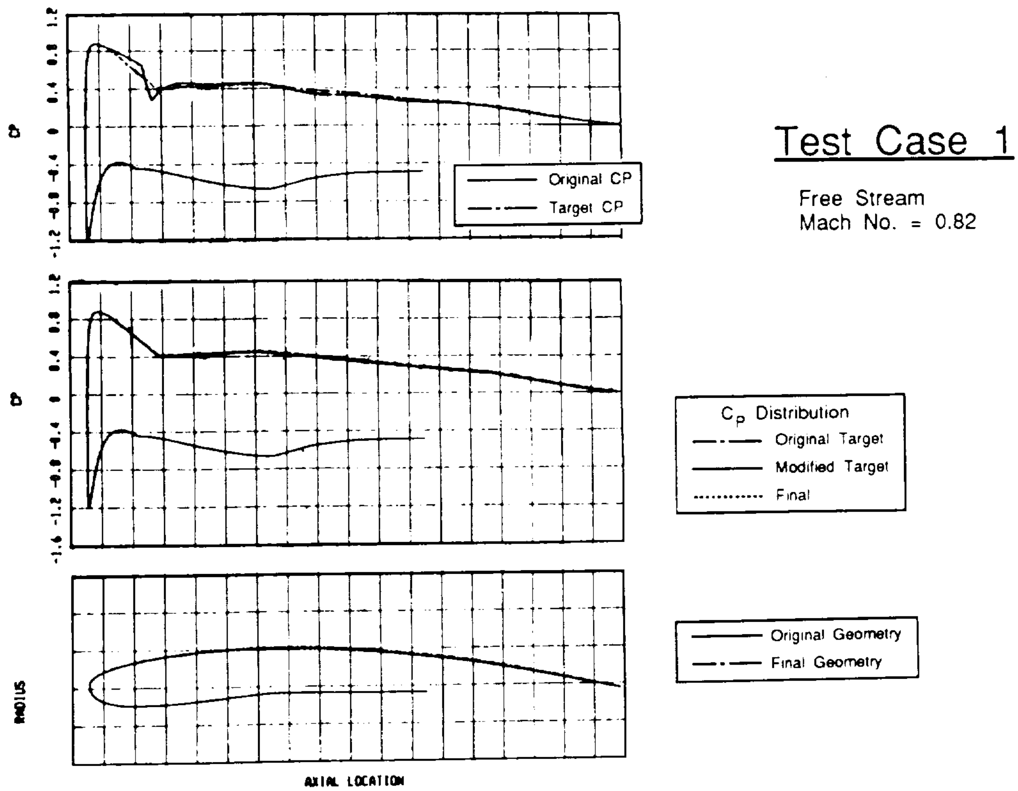

Figure 7. Results for Axisymmetric Test Case 1.

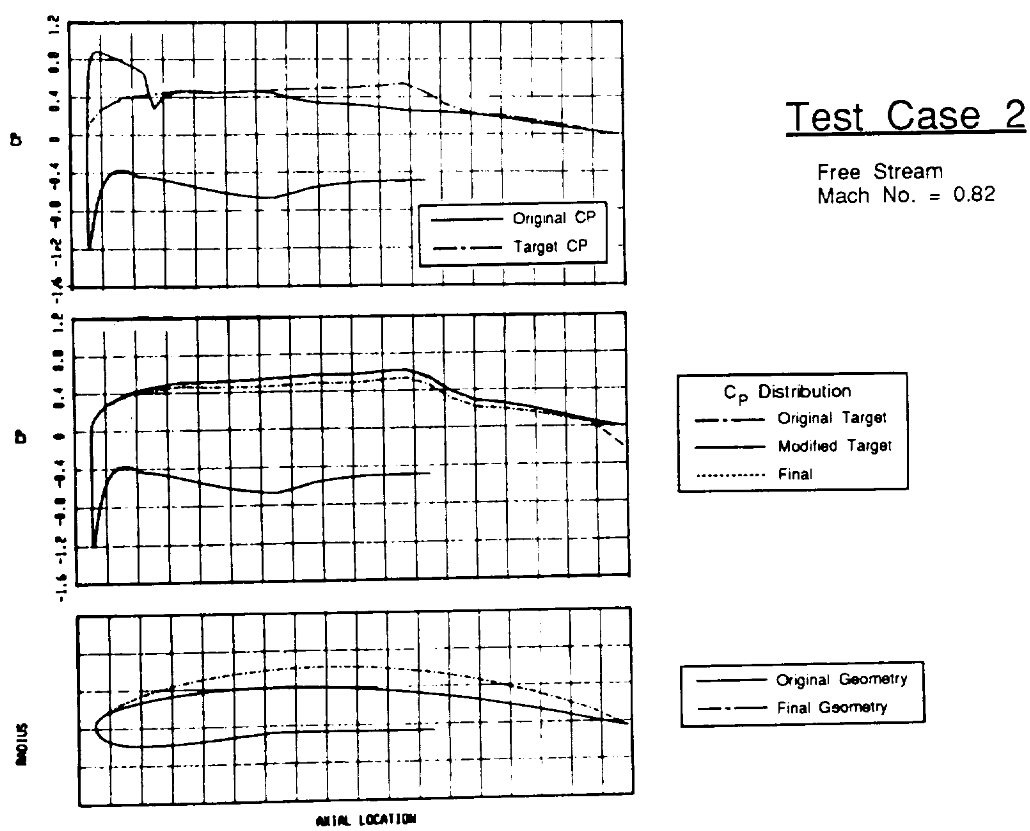

Figure 8. Results for Axisymmetric Test Case 2.

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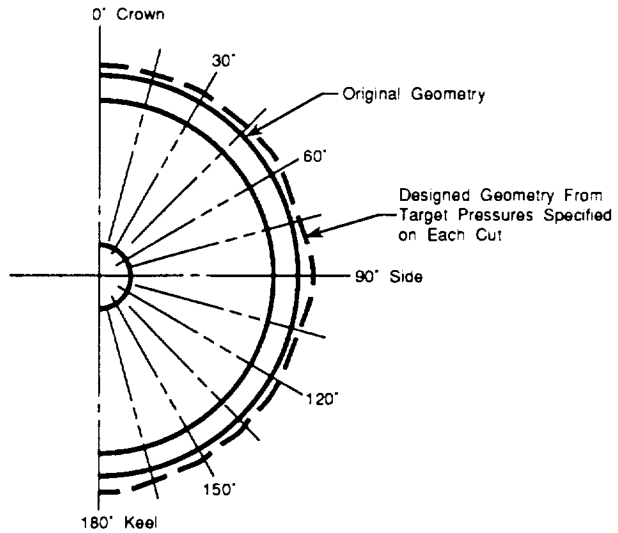

Figure 9. Three-Dimensional Inverse Design Option 1.

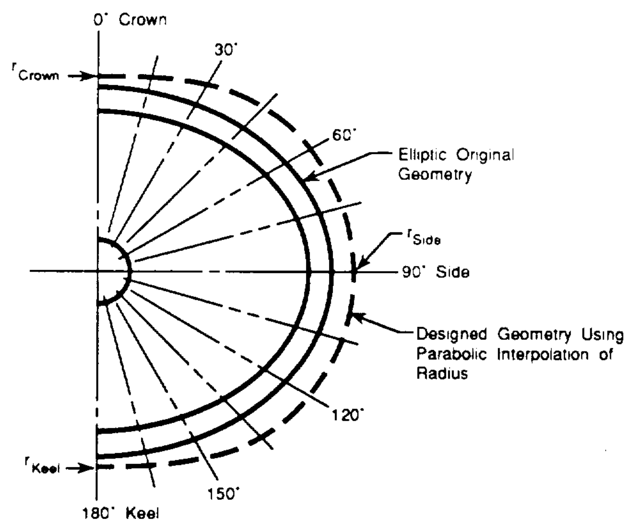

Figure 10. Three-Dimensional Inverse Design Option 2.

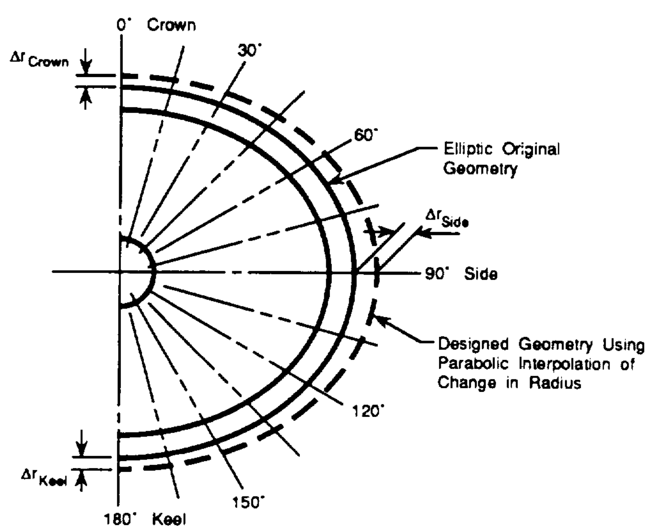

Figure 11. Three-Dimensional Inverse Design Option 3.

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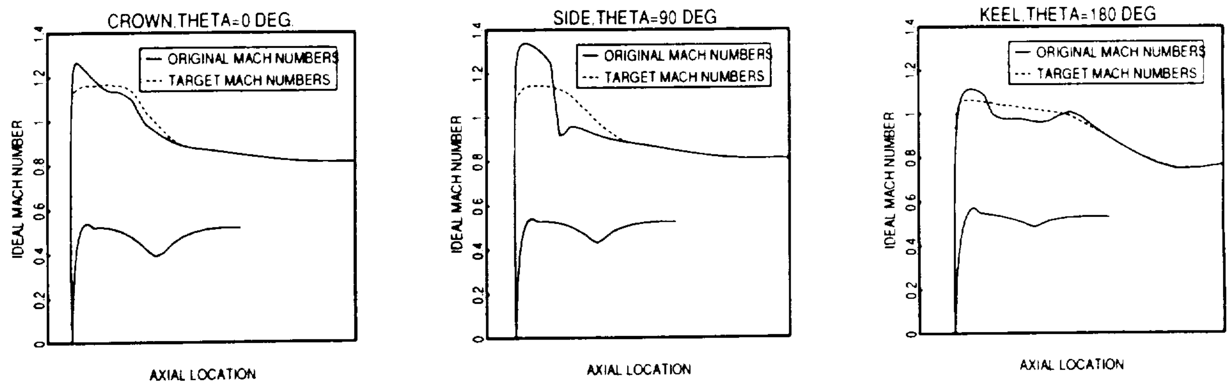

Figure 12. Original and Target Mach Number Distributions for the Three-Dimensional Test Case.

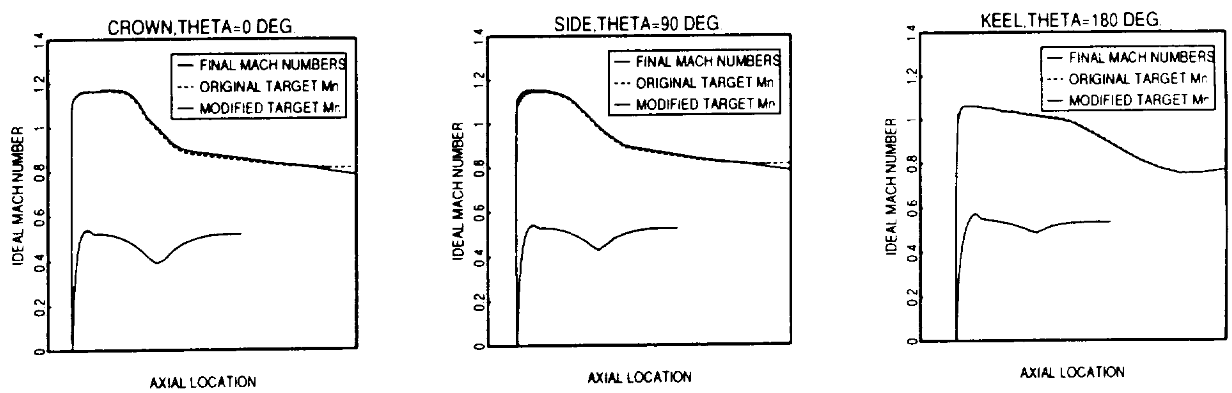

Figure 13. Final Mach Number Distributions and Original and Final Target Mach Number Distributions for the Three-Dimensional Test Case.

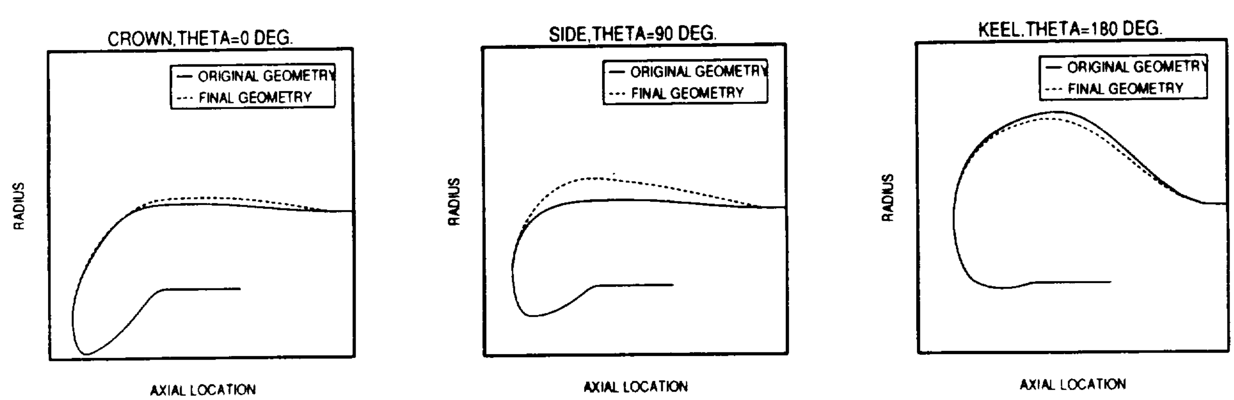

Figure 14. Original and Final Geometries for the Three-Dimensional Test Case.