Inverse Aerodynamic Design Method for Aircraft Components

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Abstract

An existing, semi-inverse, aerodynamic design algorithm is modified to permit the unrestricted geometric design of aircraft components with prescribed aerodynamic surface pressures. A brief description of the present design procedure is given and computed results are presented for both a two-dimensional airfoil and a three-dimensional nacelle configuration.

Contents

The motivation for using automated, inverse aerodynamic design methods is to reduce the overall effort required to develop aircraft geometries possessing favorable aerodynamic performance or aerodynamic interference characteristics. In Ref. 1, Garabedian and McFadden described an iterative aerodynamic design procedure suitable for automated wing design (referred to here as the GM method). They demonstrated their method by incorporating it into a three-dimensional, full-potential, transonic wing, aerodynamic analysis code.

In the GM design method, an auxiliary partial differential equation governing the spatial location of wing surface ordinates was solved iteratively in the computational plane, together with the fluid flow equation, to achieve given target surface-pressure distributions. The technique, as noted in Ref. 1, was recommended for use over only a limited portion of the wing geometry. From the present authors' experience, the original GM method of updating wing ordinates in a normal, or nearly normal, direction to the surface can lead to irregularities in the final design pressures near the leading edge of the design geometry. This is due primarily to the nonuniform stretching that can occur near the leading edge where the surface normal directions are parallel to the longitudinal axis of the component geometry.

The present inverse design procedure is formulated in a manner similar to that of the original Garabedian-McFadden scheme, but the auxiliary equation is solved directly in the physical domain, rather than in the computational domain. The new design method, which will be referred to in the following paragraphs as the modified Garabedian-McFadden (MGM) method,[2] was developed to improve airfoil or wing designs where the control of leading-edge pressures is desirable and to extend the method to handle different types of configurations, including axisymmetric or asymmetric body geometries. The surface perturbations generated with the MGM procedure are interpreted as changes in the coordinate direction perpendicular to the longitudinal axis of the geometry. This interpretation for the movement of the surface coordinates leads to smoother leading-edge geometry designs.

A two-dimensional airfoil design problem is used here to illustrate the salient features of the MGM procedure. For this two-dimensional application, the original GM auxiliary equation is rewritten as

$$\beta_0 \frac{\partial z}{\partial t} + \beta_1 \frac{\partial^2 z}{\partial x \partial t} + \beta_2 \frac{\partial^3 z}{\partial x^2 \partial t} = \Delta Q^2 = Q^2 - q^2 \tag{1}$$

In Eq. (1), the Q's are user specified target pressures expressed as flow velocities, the q computed flow velocities predicted by the given fluid flow solution procedure, and the β user-defined constants that improve the convergence of the algorithm. The time coordinate in Eq. (1) is actually a pseudotime variable representing different iterations in the solution process.

As Q approaches q, the right-hand side of Eq. (1) vanishes and the surface coordinates $z(x,t)$ stop varying with pseudotime. Partial derivatives with respect to the time coordinate are interpreted as a change in the surface coordinate Δz between any two design iterations. In order to apply Eq. (1) correctly to both upper and lower surfaces, the value of Δz must have opposite signs on each surface for equal values of the quantity, $Q^2 - q^2$.

Equation (1) is normally evaluated along the complete surface of the design geometry. Target pressures are supplied at discrete points around the geometric contour and then interpolated at locations corresponding to computational mesh points on the aerodynamic surface. These target pressures are then converted to target values of surface velocity. The computed surface velocities required to evaluate the right-hand side of Eq. (1) can then be obtained from the fluid flow solution procedure without the need for further interpolation.

Next, finite difference expressions are written for each term of Eq. (1). Assuming that there are a total of N computational points on the airfoil surface, Eq. (1) is written for each of these points i, where $1 < i < N$. A typical equation evaluated at the ith point of the surface is

$$A_i \Delta z_{i-1} + B_i \Delta z_i + C_i \Delta z_{i+1} = \Delta Q_i^2 \tag{2}$$

where, for example, at the ith point of the upper surface

$$A_i = \frac{-\beta_1}{(x_{i+1} - x_i)} + \frac{-2\beta_2}{(x_{i+1} - x_i)(x_{i+1} - x_{i-1})} \tag{3}$$

A more complete description of the finite-difference expressions used in Eq. (2) can be found in Ref. 2.

Equation (2) is evaluated at each point i, leading to a system of N equations in N unknowns (the Δz_i values). Note that at

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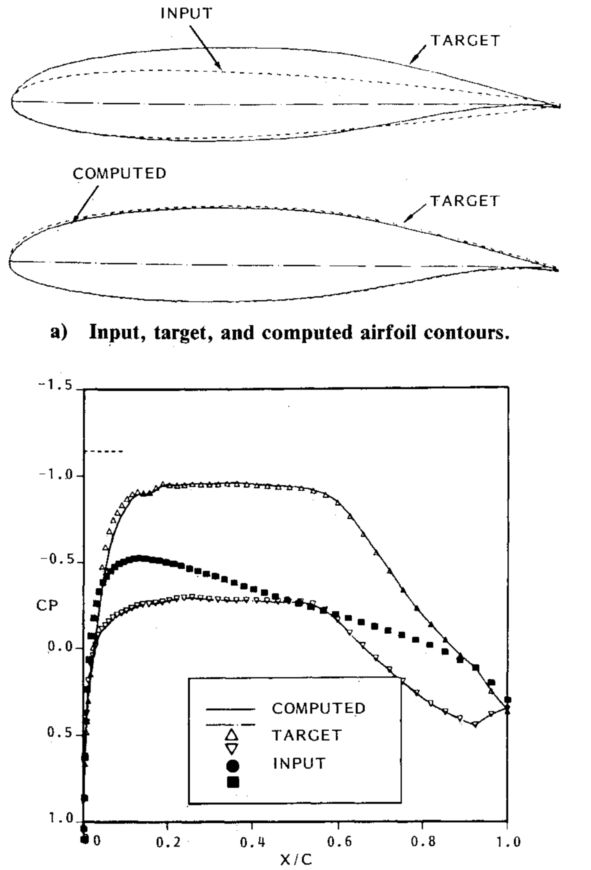


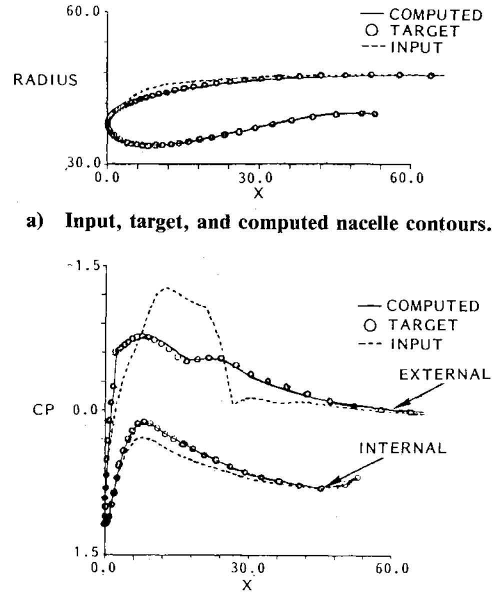
b) Input, target, and computed airfoil pressure distributions.

Fig. 1 Input, target, and computed airfoil contours and surface pressures for supercritical airfoil design problem.

each point on the aerodynamic surface, Δz_i is coupled to values at neighboring points. The resulting equations form a tridiagonal system that is solved for the values of Δz_i using the well-known Thomas algorithm.

Special treatment is required at two locations on the design geometry contour. At the leading edge, an ambiguity arises as to the direction to apply the upwinding used to evaluate certain derivatives in Eq. (2). To eliminate this problem, the leading-edge point is constrained to move as the average of both the upper- and lower-surface downstream points. This also permits the local angle of attack to change during the design process. Also, at the trailing edge, values of Δz are needed to evaluate the second derivative terms. Currently, these values are set to zero. Thus, during the design process, the trailing-edge thickness remains constant. Actually, the baseline, or starting geometry used to initiate the design process, serves primarily to fix the trailing-edge thickness of the final design geometry.

The MGM design procedure as applied to airfoil shapes is illustrated in Fig. 1. Here, the design algorithm is coupled to a full-potential airfoil code.[3] The figure shows the results of an airfoil design problem in which a symmetric baseline airfoil with a NACA 0012 contour is modified into a highly cambered, aft-loaded supercritical airfoil shape. Surface pressures were first obtained by analyzing a known supercritical airfoil, the GA(W)-1, and then applied as targets during the design process. To compute these results, boundary-layer effects were included. Shown in Fig. 1a is a comparison of the initial vs target and target vs final airfoil contours. A comparison of the baseline pressures, target pressures, and final design pressures is given in Fig. 1b. The design pressures plotted were obtained from a separate analysis of the final geometry in order to verify that an adequate design was obtained.



b) Input, target, and computed nacelle pressure distributions.

Fig. 2 Input, target, and computed nacelle contours and surface pressures for an asymmetric nacelle design problem.

An application of the MGM design procedure for three-dimensional nacelle configurations is illustrated in Fig. 2. For this sample case, the design procedure is coupled to a three-dimensional, full-potential nacelle code for arbitrary nacelle inlet configurations.[4] This case is for an asymmetric nacelle operating at a freestream Mach number of 0.8, an angle of attack of 2.0 deg, and an inlet mass flow ratio of 0.7. The object was to determine if the original nacelle contour could be recovered if its pressure distribution was used as the target distribution and if a perturbed geometry was used as the initial geometry estimate. Figure 2a illustrates the upper symmetry meridian nacelle contours for the original, designed, and initial estimate geometries. Figure 2b illustrates the corresponding surface pressure distributions. As is observed, the analysis successfully recovers the original contour.

In conclusion, a well-known design procedure, the Garabedian-McFadden method, has been modified to permit design of airfoil and wing leading-edge regions and to extend the range of configurations that can be handled by this technique. The modified design procedure has been incorporated by the authors into several existing aerodynamics programs and sample design problems have been presented for airfoil and nacelle geometries. Although simple in nature, the changes to the existing design algorithm have, in all cases examined by the authors, improved the overall versatility of the method. Test problems that were successful using the original method were not adversely effected by the modified scheme presented here. However, several classes of problems, an example of which is the case shown here for the two-dimensional airfoil, were successful only when the modified algorithm was used during the design process.

References

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