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Subsonic Panel Method for Designing Wing Surfaces From Pressure Distribution

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Subsonic Panel Method for Designing Wing Surfaces From Pressure Distribution

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SUMMARY

An iterative method has been developed for designing wing section contours corresponding to a prescribed subcritical distribution of pressure. The calculations are initialized by using a surface panel method to analyze a baseline wing or wing-fuselage configuration. A first-order expansion to the baseline panel method equations is then used to calculate a matrix containing the partial derivative of potential at each control point with respect to each unknown geometry parameter. In every iteration cycle, the matrix is used both to calculate the geometry perturbation and to analyze the perturbed geometry. The distribution of potential on the perturbed geometry is established by simple linear extrapolation from the baseline solution. The extrapolated potential is converted to velocity by numerical differentiation, and velocity is converted to pressure by Bernoulli's equation. Not only is the accuracy of the approach good for very large perturbations, but the computing cost of each complete iteration cycle is substantially less than one analysis solution by a conventional panel method. Example design solutions are presented to demonstrate that the method is accurate, numerically stable, and converges in only three to five iterations.

1. INTRODUCTION

The surface panel approach has proved to be very successful for the analysis of subcritical, potential flow around geometrically complex aircraft configurations (references 1-10). It is well established that the better formulated panel methods consistently predict accurate wing pressure distributions, including the effect of strong fuselage-nacelle interference. A typical example is illustrated in Figure 1. Several investigators who have been aware of the power of surface panel methods have established iterative inverse techniques for designing wing section geometry corresponding to a prescribed distribution of pressure (references 11-14). However, each of the existing three-dimensional wing design methods suffers from two or more of the following shortcomings:

i) the calculations will not converge,

ii) the designed geometry is unrealistically wavy chordwise and/or spanwise,

iii) the designed geometry does not generate the prescribed pressure distribution near the wing leading edge,

iv) the computing cost of each iteration cycle is substantial.

The new design method presented in this report was developed by McDonnell Aircraft Company (MCAIR) under contract to NASA, Langley Research Center. The method is believed to be the first that can reliably and efficiently solve the wing design problem of Figure 2. The success of the method is attributed to the development of a formulation that overcomes the design dilemmas depicted in Figure 3.

The basic mathematical approach is similar to that of the two-dimensional design method developed for multi-element airfoils (references 8 and 15). However, the present design method has a significant new cost-savings feature that is especially appropriate for wing-fuselage configurations. The "perturbation analysis method" of references 9 and 10 is employed in each design iteration cycle. As will be demonstrated in Section 2, the perturbation analysis method is an accurate and extremely efficient method for analyzing large changes to wing section geometry. The present design method is designated the "perturbation design method" because it is an inverse formulation for the perturbation analysis method. The mathematical formulation and example design solutions are presented in Section 3.

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2. PERTURBATION ANALYSIS METHOD

The standard solution approach to prescribed pressure design problems is to divide each iteration cycle into an analysis, pressure calculation step and an inverse, geometry correction step. In the two-dimensional method of references 8 and 15, an entire panel method solution is calculated during each analysis step. Furthermore, a new geometry-velocity perturbation matrix is calculated for each inverse step. In spite of the fact that the number of computations in each iteration cycle is a cubic function of the number of panels, the total computing cost is relatively small. The reason is that typical two-dimensional problems require fewer than one hundred panels.

However, the number of panels required for wing-fuselage configurations is an order of magnitude greater, and the preceding design procedure would be extremely expensive. During the initial phase of the present contract, it became clear that a practical wing-on-fuselage design procedure could not be established on the basis of existing panel method technology. Consequently, the perturbation analysis method was developed. The method is an extremely efficient tool for analyzing the pressure distribution corresponding to a series of arbitrary, small perturbations to a baseline wing-fuselage geometry (Figure 4). The following features of the perturbation analysis method make it especially practical for application to an iterative wing section design method.

(1) The computational expense for analyzing each successive geometry perturbation is more than an order of magnitude less than that of a conventional panel method,

(2) The pressure distribution prediction accuracy is competitive with conventional surface panel methods for very large perturbations to wing section geometry,

(3) A pre-calculated matrix of partial derivatives for the paneled baseline configuration is available. Each element of the matrix is the rate of change of potential at a boundary condition control point with respect to a geometry parameter perturbation. For design applications, the geometry-potential perturbation matrix can be efficiently converted to a geometry-pressure perturbation matrix.

The purpose of this section is to familiarize the reader with the fundamentals and power of the perturbation analysis method (a more complete description is available in references 9 and 10). An inverse formulation for the perturbation analysis method is reserved for the next section. The inverse formulation is the foundation for the iterative, prescribed pressure design method for wings and wings on fuselages.

2.1 MATHEMATICAL APPROACH - The perturbation analysis approach requires an initial baseline calculation of a conventional panel method solution for an arbitrary baseline configuration. Subsequently a matrix consisting of the partial derivatives of velocity potential with respect to geometry coordinates is calculated. The baseline solution and derivative matrix are calculated one time only and then stored for repetitive use. For each geometry perturbation, the solution surface distribution of velocity potential is constructed by multiplying the derivative matrix by a new right-hand-side. This procedure bypasses the two computationally expensive steps of a conventional panel method: calculating the influence coefficients and solving a large system of linear algebraic equations.

Although the perturbation analysis method is appropriate for predicting the effect of arbitrary small changes to wing planform and fuselage geometry, the real power of the method is the accuracy with which large perturbations to wing thickness, camber, and twist can be analyzed. A simple two-dimensional example is the perturbation from a NACA 0012 airfoil to a circular cylinder illustrated in Figure 5. Unlike conventional "small disturbance" or "linearized" flow methods, predictions by the perturbation analysis method are accurate even at the leading edge of a wing. The reason is that only the surface potential - not velocity and pressure - is linearized with respect to geometry coordinate perturbations. It can be shown that the nonlinear terms are much smaller for potential than for either velocity or pressure. For example, consider the flow about an elliptic cylinder. As illustrated in Figures 6 and 7, the variation of potential with thickness is exactly linear.

Three computer programs are required in order to apply the perturbation analysis method (see Figure 8). The first program is the conventional MCAIR panel method for analyzing the baseline configuration. The second program employs an output solution file from the first program plus a differential mathematical formulation to calculate the matrix of partial derivatives of perturbation potential. For each baseline configuration, the first two programs generate an input file for the third program - the Perturbation Analysis Program.

After the input file is generated, the first two programs are no longer required. The Perturbation Analysis Program can be executed repeatedly at low computing cost for the analyses of a series of perturbations to the panel corner coordinates $(x,y,z)_j$ of the complete aircraft configuration. The method used to analyze each perturbation is the same as the conventional panel method calculation with two significant exceptions. First, no influence coefficients are calculated; second, no large system of linear algebraic boundary condition equations is solved. Instead, the perturbation potential at each control point is calculated by linear extrapolation. The conversion of potential to surface velocity is based on numerical differentiation.

Usually the second program is executed twice for each baseline configuration, once at 0° incidence and once at 90° incidence. By employing the principle of linear superposition, the perturbation analysis is automatically performed at any intermediate angle of attack (Figure 9).

2.2 EXAMPLE PERTURBATION ANALYSIS SOLUTIONS - Demonstration of prediction accuracy for the perturbation analysis method is presented in Figures 10-12. The baseline geometry is an isolated low aspect ratio wing with no camber and no twist. Coordinate displacements normal to the chord plane were applied in order to generate the fighter wing geometry illustrated in Figure 10b. The incompressible, potential flow solution for the fighter wing was calculated using both the conventional surface panel methods and the perturbation analysis method. The distributions of pressure, forces, and moment (Figures 12a and 12b) demonstrate that the perturbation analysis method is extremely accurate for very large perturbations.

In order to evaluate prediction accuracy for a configuration with strong wing-fuselage interference, the paneled YAV-8B Harrier of Figure 13 was selected as the baseline configuration. The good accuracy of the conventional MCAIR panel method is demonstrated by comparing the calculated and experimental pressure distributions on the baseline wing (Figure 14). Compressibility effects were simulated by a Gothert correction, which is available in a later version of the panel method than the incompressible version of reference 9. Approximately 420 seconds computing time on the CDC CYBER 176 were required for the analysis, where 537 panels on each side of the symmetry plane were used to model the YAV-8B.

For application to the perturbation analysis method, the partial derivative of perturbation potential (ϕ_i) at each control point on the YAV-8B configuration was calculated with respect to the waterline displacement (z_j) of each of the 150 panel corner points on the wing. The total computing time expended by the second program for 0° and 90° angles of attack was 1500 seconds on the CDC CYBER 176.

The perturbation analysis method was then used to calculate the wing-fuselage pressure distribution corresponding to a large change in the wing geometry. The actual cambered supercritical wing with 8° twist (Figure 13) was perturbed to an uncambered, untwisted wing with constant NACA 0012 section geometry. It is obvious that this change is not recommended for the purpose of improving aerodynamic performance. It is simply intended to permit assessment of the accuracy of the perturbation analysis method for a relatively large change. As shown in Figure 15, the predicted pressure distribution agrees closely with the nearly exact potential flow solution calculated by the conventional surface panel method. However, the perturbation analysis required only eight seconds computing time compared to 420 seconds for the conventional analysis. On McDonnell Douglas computers, the total computing cost (including input/output) for the perturbation analysis was less than 1/20 the cost of the conventional analysis.

3. PERTURBATION DESIGN METHOD

Consider a paneled representation of a wing or wing-on-fuselage at a fixed, arbitrary angle of attack. Suppose that prescribed pressure coefficients are assigned to the panel centers. The objective of the design method is to determine the change in wing section geometry - twist, camber, and thickness - that most nearly corresponds to the prescribed pressure distribution (Figure 2).

In the preceding section, it was shown that the pressure distribution corresponding to large changes in wing section geometry can be accurately calculated by the perturbation analysis method. Conversely, it is reasoned that large changes in wing section geometry could be accurately designed by an inverse formulation for the perturbation analysis method. That approach is the basis for the perturbation design method of this report. The inverse formulation requires iteration because pressure is a non-linear function of the wing section geometry coordinates. However, the fact that perturbation potential is a nearly linear function eliminates the need to perform extensive computations in each iteration cycle.

A schematic of the perturbation design method is presented in Figure 16. The method can be used to solve very general aero-dynamic design problems. For example, the prescribed aerodynamic quantity at a panel center can be either a velocity component or pressure coefficient. Arbitrary geometry parameters such as wing chord or fuselage shape can be selected for design. Furthermore, design constraints such as fixed camber or thickness can be imposed. Most design problems, however, are of the type illustrated in Figure 17. This type, designated the "standard wing design problem", is defined in detail below. The remainder of this section presents the mathematical formulation for the perturbation design method and example design solutions.

3.1 STANDARD WING DESIGN PROBLEM - As illustrated in Figure 17, the region of panels subject to design is identified by corner points in the range $(i_A, j_A) \leq (i,j) \leq (i_B, j_B)$, where the limits (i_A, i_B, j_A, j_B) are selected by the user. If j_A and j_B are points on the upper and lower surface trailing edge respectively, then the geometry of the complete wing section at each span station i will be designed. At the center of each panel in the design region, the desired pressure coefficient is prescribed by the user.

The unknowns are Δz at the panel corner points $\{(i_A, j_A+1) \leq (i,j) \leq (i_B, j_B-1)\}$. However, not all of the unknowns are permitted to be independent. As illustrated in Figure 18, less than one-half of the unknowns are independent. The remaining unknowns are generated by interpolation through the independent unknowns. On the span stations $i = i_A, i_A+2, i_A+4, \ldots, i_B$, each dependent unknown $\Delta z(i,j)$ is established by least squares quadratic interpolation through the path of points (j-3, j-1, j, j+1, j+3) on the baseline configuration. For the remaining span stations ($i = i_A+1, i_A+3, \ldots, i_B-1$), each unknown $\Delta z(i,j)$ is established by straight line interpolation through $\Delta z(i-1,j)$ and $\Delta z(i+1,j)$. For this type of interpolation to be accurate, it is implicitly assumed that the three points (i-1,j), (i,j), and (i+1,j) lie on nearly the same per cent chord line. Typical wing panelling is consistent with this assumption. A short computer program has been written to perform the interpolation on the baseline panelled configuration for all of the dependent variables.

The reason for limiting the number of independent unknowns a priori is to prevent numerically unstable design calculations. Figures 19a and 19b illustrate two types of design instabilities that could occur if every panel corner in the design region were allowed to be an independent unknown.

Consistent with the nomenclature of the perturbation analysis method (reference 9), each independent unknown perturbation is assigned an index k_S. The value of Δz for perturbation number k_S is designated S_{k_S}. A schematic of the independent unknowns is presented in Figure 20. The objective of the perturbation design method is to calculate the values S_{k_S} ($1 \leq k_S \leq NKS$) that most nearly correspond to the prescribed pressure distribution.

3.2 MATHEMATICAL FORMULATION FOR PERTURBATION DESIGN METHOD - For any panelled baseline configuration, application of the perturbation design method requires that the arrays ϕ_i and $\partial\phi_i/\partial S_{k_S}$ have been calculated a priori. ϕ_i is the perturbation potential at the i^{th} control point and $\partial\phi_i/\partial S_{k_S}$ is the rate of change of ϕ_i with respect to independent geometry perturbation number k_S. The MCAIR 3-D Subsonic Potential Flow Program and MCAIR 3-D Geometry Influence Coefficient Program will automatically calculate the required arrays and store them on a computer disk file. The perturbation design method can then be used to calculate the geometry perturbation that most nearly matches prescribed aerodynamic properties within the limitations of a minimal least square error.

At any panel center selected by the user, one or more properties can be prescribed. The property can be either pressure coefficient (c_p) or velocity component in an arbitrary, specified direction. The prescribed value of an aerodynamic quantity at a panel center is designated Q_{P_i}, where there is one index i for each prescribed value ($1 \le i \le n_{DES}$).

Arbitrary geometric constraints can be imposed upon the independent unknowns (S_{k_S}). Each geometric constraint is expressed as a linear equation

$$\sum_{i_D=1}^{n_D} [c_{i_D} \cdot S_{k_S(i_D)}] = CHRS \qquad (1)$$

Where n_D, c_{i_D}, and CRHS are arbitrary values specified by the user. Depending upon the values specified, the constraint equation can be used to fix the cross-sectional area of a wing section, fix the thickness at one point, and so forth.

The aerodynamic design problem can now be expressed in mathematical form. The objective is to calculate the array of independent geometric unknowns S_{k_S} $(1 \leq k_S \leq NKS)$ that will minimize the following function E.

$$E = \sum_{i=1}^{n_{DES}} \left\{ WDES_i^2 \cdot AREA_i \cdot (Q_i - Q_{P_i})^2 \right\}$$
$$+ \sum_{j=1}^{n_{CON}} \left\{ WCON_j^2 \cdot \left[\sum_{i_D=1}^{n_D} c_{i_D} \cdot S_{k_S(i_D)} - CRHS \right]_j^2 \right\} \qquad (2)$$

where (1) each i $(1 \leq i \leq n_{DES})$ corresponds to one prescribed aerodynamic quantity at one panel center,

(2) each j $(1 \leq j \leq n_{CON})$ corresponds to a geometric constraint equation,

(3) the weights $WDES_i$ and $WCON_j$ are specified by the user (typically, $WDES_i = 1.00$ and $WCON_j >> 1.00$),

(4) $AREA_i$ is the area on the baseline configuration of the panel corresponding to prescribed quantity Q_{P_i}, and

(5) $(Q_i - Q_{P_i})$ is the difference between the calculated and prescribed values of an aerodynamic property.

Iteration is used to solve for the array of unknowns S_{k_S} corresponding to the minimum value of E. As depicted in Figure 16, each iteration cycle is divided into an inverse step in which the geometry perturbation is calculated and a direct step in which the perturbed or updated geometry is analyzed.

In the inverse step of each iteration cycle, the matrix of derivatives is calculated. Then the change in Q_i induced by a small perturbation to the array S_{k_S} can be expressed as

$$dQ_i = \sum_{k_S=1}^{NKS} \left[\frac{\partial Q_i}{\partial S_{k_S}}\right] \cdot dS_{k_S} \tag{3}$$

By incorporating equation (3) in equation (2) and minimizing E with respect to dS_{k_S}, a system of linear, algebraic equations is established. The solution by standard matrix algebra provides the values dS_{k_S}. The updated geometry is then analyzed by the perturbation analysis method.

One might expect that recalculation of the matrix $[\frac{\partial Q_i}{\partial S_{k_S}}]$ during each iteration cycle would requre substantial computing expense. However, the following approach has proved to be both very efficient and accurate.

Consider an aerodynamic quantity Q_i, which can be either pressure coefficient or a velocity component at the center of some panel. Figure 21 illustrates the panel of interest and neighboring control points. If Q_i is incompressible pressure coefficient, then

$$Q_i = C_p = 1 - (V_x^2 + V_y^2 + V_z^2) \tag{4}$$

$$\frac{\partial Q_i}{\partial S_{k_S}} = -2(V_x, V_y, V_z) \cdot \frac{\partial}{\partial S_{k_S}} (V_x, V_y, V_z) \tag{5}$$

If Q_i is velocity component in an arbitrary, fixed direction (with direction cosines cosx, cosy, cosz), then

$$Q_i = \vec{V} \cdot (\cos x, \cos y, \cos z) \tag{6}$$

$$\frac{\partial Q_i}{\partial S_{k_S}} = (\cos x, \cos y, \cos z) \cdot \frac{\partial}{\partial S_{k_S}} (V_x, V_y, V_z) \tag{7}$$

Equation (5) and (7) indicate that whether Q_i is pressure coefficient or velocity component, the problem of calculating $\frac{\partial Q_i}{\partial S_{k_S}}$ is essentially reduced to calculating the velocity derivatives $(\frac{\partial V_x}{\partial S_{k_S}}, \frac{\partial V_y}{\partial S_{k_S}}, \frac{\partial V_z}{\partial S_{k_S}})$. This is discussed below.

At the center of the panel of interest (Figure 21), the velocity vector $\vec{V}$ is

$$\vec{V} = V_x \vec{e}_x + V_y \vec{e}_y + V_z \vec{e}_z = \vec{V}_\infty + \vec{\nabla}\phi \tag{8}$$

On the basis of the perturbation potential ϕ at approximately 20 neighboring control points j, $\vec{\nabla}\phi$ can be calculated by numerical differentiation. This is expressed mathematically as

$$\vec{\nabla}\phi = \sum_{j=1}^{\sim 20} \phi_j \cdot (a_j \vec{e}_x + b_j \vec{e}_y + c_j \vec{e}_z) \tag{9}$$

where the scalars a_j, b_j, and c_j are functions of the corner point coordinates of the neighboring panels. A simplified analogue of equation (9) for one-dimensional numerical differentiation is presented in Figure 22. (The three-dimensional numerical differentiation that is actually employed is based upon the assumption that neighboring panels stretch or shrink approximately the same percentage. A more accurate algorthim could be introduced at the expense of greater compexity, but the present approach has been demonstrated to be adequate.)

By substituting equation (9) into equation (8) and differentiating, the desired velocity derivatives can be expressed as

$$\frac{\partial}{\partial s_{k_S}} (V_x, V_y, V_z) = \sum_{j=1}^{\sim 20} \left\{ \frac{\partial \phi_j}{\partial s_{k_S}} \cdot (a_j \vec{e}_x + b_j \vec{e}_y + c_j \vec{e}_z) + \phi_j \cdot \frac{\partial}{\partial s_{k_S}} (a_j \vec{e}_x + b_j \vec{e}_y + c_j \vec{e}_z) \right\} \tag{10}$$

The only term in equation (10) that requires substantial expense to compute is $\frac{\partial \phi_j}{\partial s_{k_S}}$. Fortunately, it is also the only term that is nearly independent of perturbations to wing thickness, camber, and twist. Therefore, the precalculated baseline matrix $[\frac{\partial \phi_i}{\partial S_{k_S}}]$ that is available on computer disk file can be used in equation (10) during every iteration cycle. Substitution of equation (10) into equation (5) or (7) yields the desired value, $\frac{\partial Q_i}{\partial S_{k_S}}$.

A significant feature of the preceding approach is that the accuracy of the calculated matrix $[\frac{\partial Q_i}{\partial S_{k_S}}]$ is competitive with an exact first order expansion during each iteration cycle. However, much less computing effort is required.

In fact, the number of computations required for a complete iteration cycle is relatively small. The reason is apparent upon consideration of each calculation step in Figure 16. For example, consider the system of linear, algebraic equation to be solved for the perturbations to the independent unknowns. Typically, fewer than one hundred unknowns are sufficient for wing design, compared to several hundred for a conventional panel method analysis of a wing-fuselage. Also, consider the last step of each iteration cycle - analyzing the updated geometry. The extremely efficient perturbation analysis method is used for that calculation.

The perturbation design method has been automated and is operational on the McDonnell Douglas CYBER 176. The computer program is designated "MCAIR Perturbation Design Program (Version 1)." A demonstration of the accuracy, efficiency, and numerical stability of the method is presented in the next section.

3.3 EXAMPLE DESIGN SOLUTIONS - Two example solutions by the perturbation design method are presented below. The calculations were performed on the McDonnell Douglas CDC CYBER 176.

In the first design problem, the analytical pressure distribution for an infinite circular cylinder was prescribed. The baseline geometry is a NACA 0012 airfoil represented by 26 panels. The calculations converged to the minimum root-mean-square error in four iterations. As shown in Figure 23, the solution geometry is nearly circular. The slight discrepancy that does exist is attributable to the sparse panelling aft of 50% chord. The streamwise geometry interpolation techniques of Figure 18 would have been substantially more accurate if 30 or more panels were used to represent the geometry, instead of only 26.

In the second example, the objective is to design the fighter wing section geometry of Figure 10b by starting from the baseline wing panelling of Figure 10a. The fighter wing pressure distribution at 0° angle of attack was prescribed at the center of the 208 panels. The converged solution after 3 iterations is presented in Figures 24a and 24b. Although the designed geometry is close to the target, it is again apparent that slightly greater chordwise panel density would have permitted a better match on the lower surface at the tip.

Additional design examples have been successfully calculated using the panelling of Figure 10a as a baseline. A typical solution requires 3 to 5 iteration cycles. The computing time for each complete iteration cycle is approximately 5 seconds, compared to 53 seconds for one conventional panel method analysis.

4. CONCLUSIONS

The perturbation analysis and perturbation design methods are similar to classical thin wing theory in the sense that small disturbance "linearized" assumptions are employed. This mathematical simplification generates extensive computational savings for aerodynamic problems involving successive iteration, such as design.

On the other hand, the restrictions of classical thin wing theory have been elminated. Compared to an exact potential flow solution, the present approach is quite accurate for thick wings, large leading edge radius or camber, and high angle of attack, The success of the perturbation design method is attributed to the inclusion of all significant first-order geometry-pressure peturbation terms in each iteration cycle. This leads to rapid solution convergence, in spite of the fact that the entire distribution of surface potential is constructed by simple linear extrapolation.

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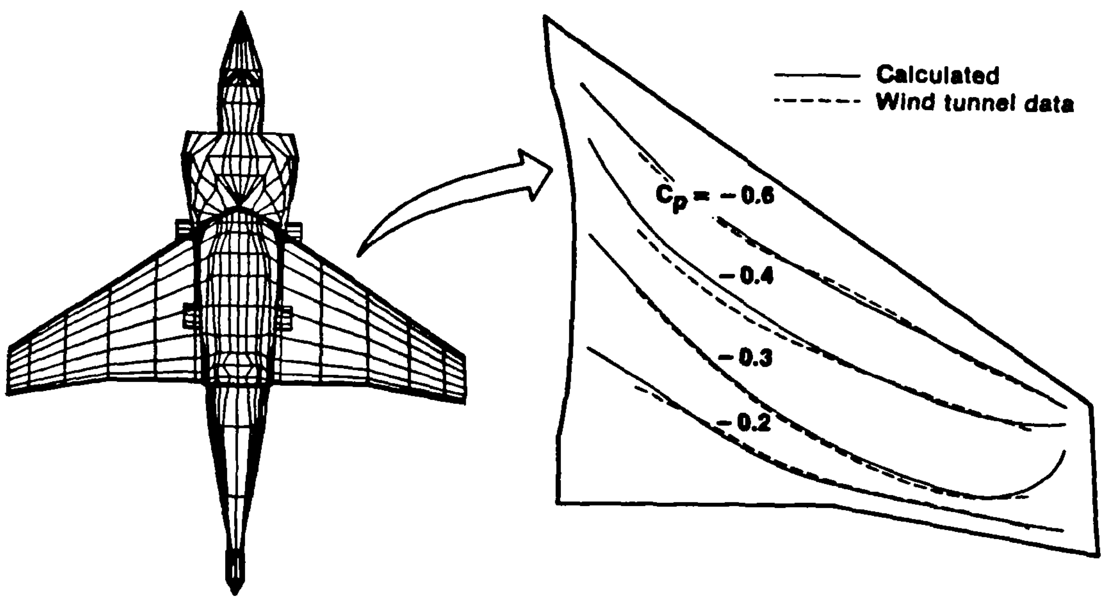


YAV-8B Surface Panel Modeling

Upper Surface Isobar Pattern
Mach = 0.5 6.4° Angle-of-Attack

Figure 1. Prediction Accuracy of MCAIR Panel Method

Given:
- Fuselage Geometry
- Baseline Wing
- Angle-of-Attack
- Prescribed Pressure Distribution (C_{p_i})

Calculate: Wing Section Geometry

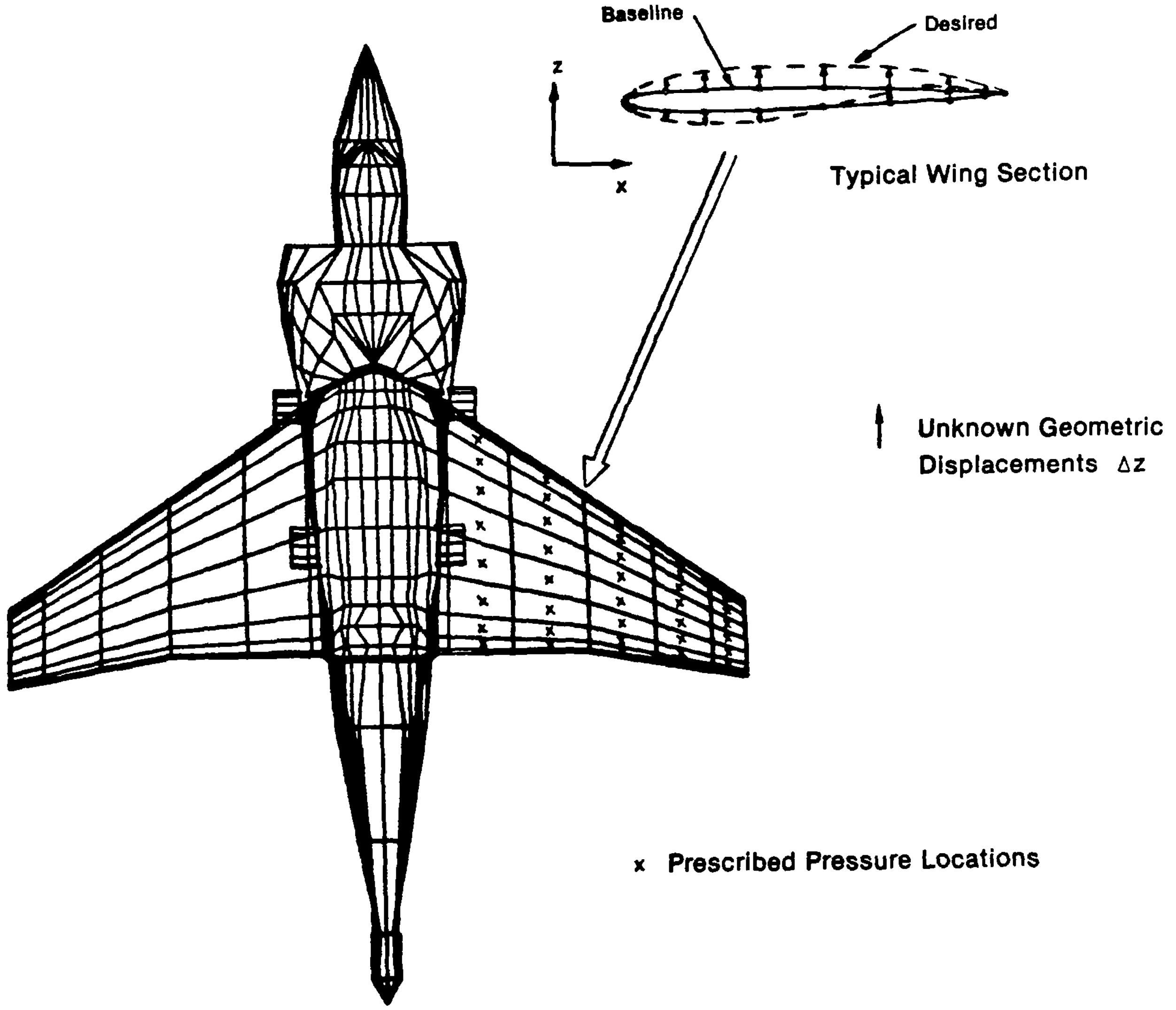
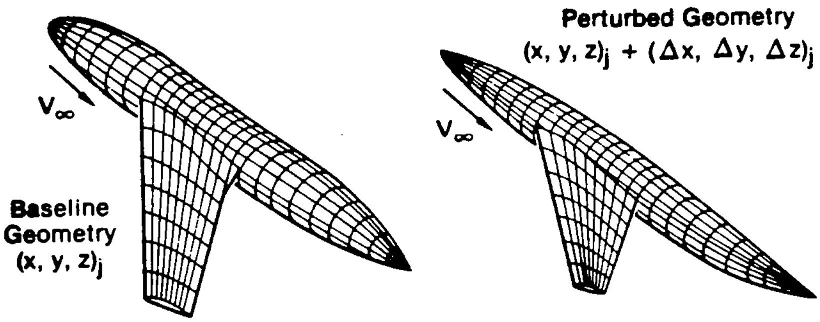


Figure 2. Objective of the Design Method

SYMPTOMS	CAUSE	REMEDY
• CALCULATED PRESSURES OVERLY SENSITIVE TO CONTOUR SMOOTHNESS	• SOURCE SINGULARITIES	• COMBINED SOURCE - DOUBLETS (GREEN'S IDENTITY)
• CALCULATIONS DO NOT CONVERGE • INACCURATE LEADING EDGE DESIGN CONTOURS	• DIRICHLET BOUNDARY CONDITIONS • TRANSPIRATION PRINCIPAL (EQUIVALENT BLOWING)	• FIRST-ORDER MATHEMATICAL EXPANSION ($\partial c_p / \partial z$)
• UNREALISTICALLY WAVY DESIGN CONTOURS	• ONE UNKNOWN COORDINATE PER PRESCRIBED PRESSURE	• MORE PRESSURES THAN UNKNOWNS • LEAST SQUARES
• HIGH COST PER ITERATION	• COMPLETE PANEL METHOD ANALYSIS DURING ITERATION CYCLE	• PERTURBATION ANALYSIS METHOD

Figure 3. Common Barriers to a Successful Wing Design Method



Objective

- Subsonic Inviscid Analysis of Multiple Geometry Perturbations at Small Additional Cost

Approach

- Precalculated Baseline Matrix of Potential Derivatives $\left\{\frac{\partial\phi i}{\partial x_j}, \frac{\partial\phi i}{\partial y_j}, \frac{\partial\phi i}{\partial z_j}\right\}$
- Linear Extrapolation
  $$(\phi_i + \Delta\phi_i) = \phi_i + \sum_j \left\{\frac{\partial\phi i}{\partial x_j}\Delta x_j + \frac{\partial\phi i}{\partial y_j}\Delta y_j + \frac{\partial\phi i}{\partial z_j}\Delta z_j\right\}$$

Figure 4. Perturbation Analysis Method

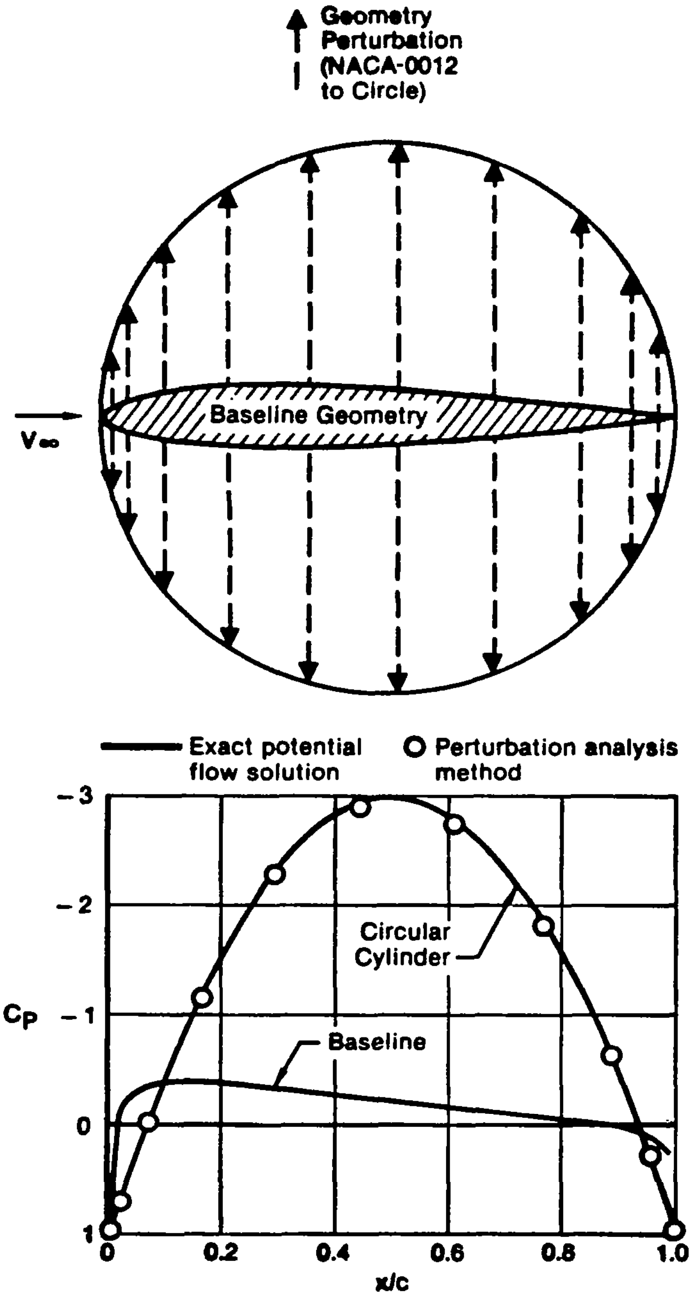


Figure 5. Simple Demonstration of Perturbation Analysis Method
2-D Incompressible Flow

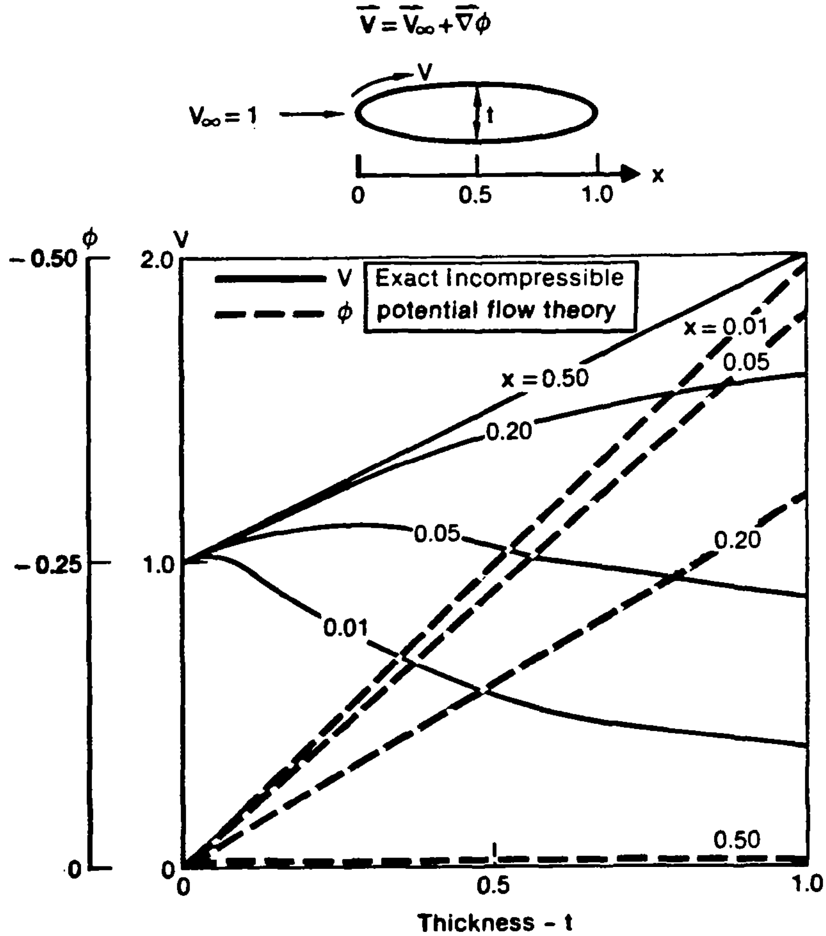


Figure 6. **Variation of Flow Properties with Thickness**
Elliptical Cylinder at 0° Incidence

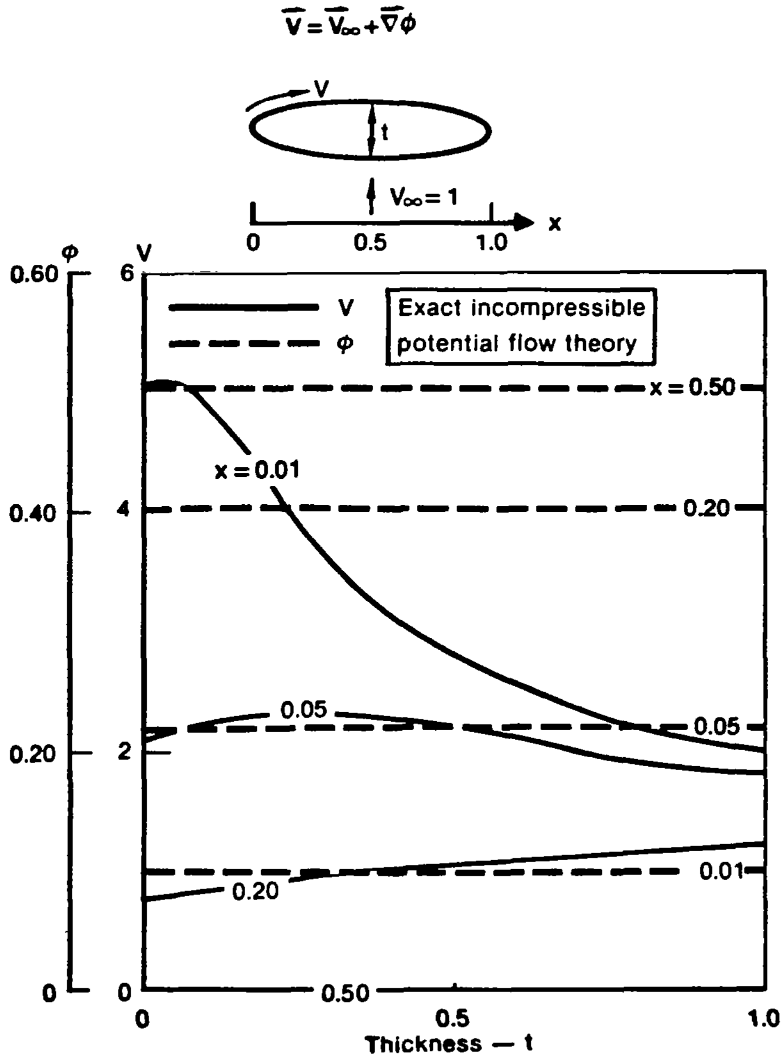
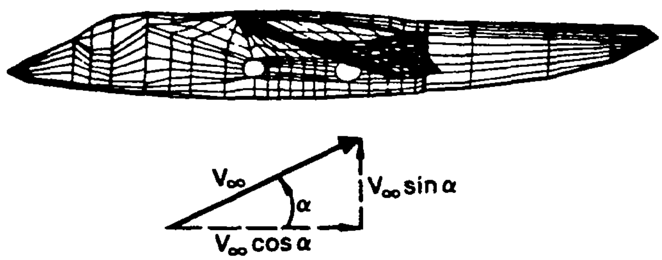


Figure 7. Variation of Flow Properties with Thickness Elliptical Cylinder at 90° Incidence

	MCAIR 3-D Subsonic Potential Flow Program	MCAIR 3-D Geometry Influence Coefficient Program	MCAIR 3-D Perturbation Analysis Program
Input	• Baseline Geometry $(x, y, z)_j$	• Baseline Geometry $(x, y, z)_j$ • Baseline Potential Distribution ϕ_i	• Baseline Properties $(x, y, z)_j, \phi_i$ • Derivative Matrix $(\partial\phi_i/\partial x_j, \partial\phi_i/\partial y_j, \partial\phi_i/\partial z_j)$ • Geometry Perturbation $(\Delta x, \Delta y, \Delta z)_j$
Approach	• Conventional Panel Method	• First-Order Expansion to Panel Method Formulation	• Linear Extrapolation $\Delta\phi_i = \sum_j [(\partial\phi_i/\sigma x_j)\,\Delta x_j + (\partial\phi_i/\partial y_j)\,\Delta y_j + (\partial\phi_i/\partial z_j)\,\Delta z_j]$
Output	• Baseline Potential Distribution ϕ_i • Baseline Aerodynamic Properties	• Derivative Matrix $(\partial\phi_i/\partial x_j, \partial\phi_i/\partial y_j, \partial\phi_i/\partial z_j)$	• Aerodynamic Properties of Perturbed Geometry

Figure 8. The Three Computer Programs for the Perturbation Analysis Method



$$\phi_i = \left\{ \begin{array}{l} + \cos\alpha \cdot \left[\phi_{i_{\text{baseline}}} + \sum_j \left(\frac{\partial \phi_i}{\partial x_j} \Delta x_j + \frac{\partial \phi_i}{\partial y_j} \Delta y_j + \frac{\partial \phi_i}{\partial z_j} \Delta z_j \right) \right]_{0^\circ \alpha} \\ + \sin\alpha \cdot \left[\phi_{i_{\text{baseline}}} + \sum_j \left(\frac{\partial \phi_i}{\partial x_j} \Delta x_j + \frac{\partial \phi_i}{\partial y_j} \Delta y_j + \frac{\partial \phi_i}{\partial z_j} \Delta z_j \right) \right]_{90^\circ \alpha} \end{array} \right\}$$

Figure 9. Linear Superposition of 0° and 90° Solutions Perturbation Analysis Method

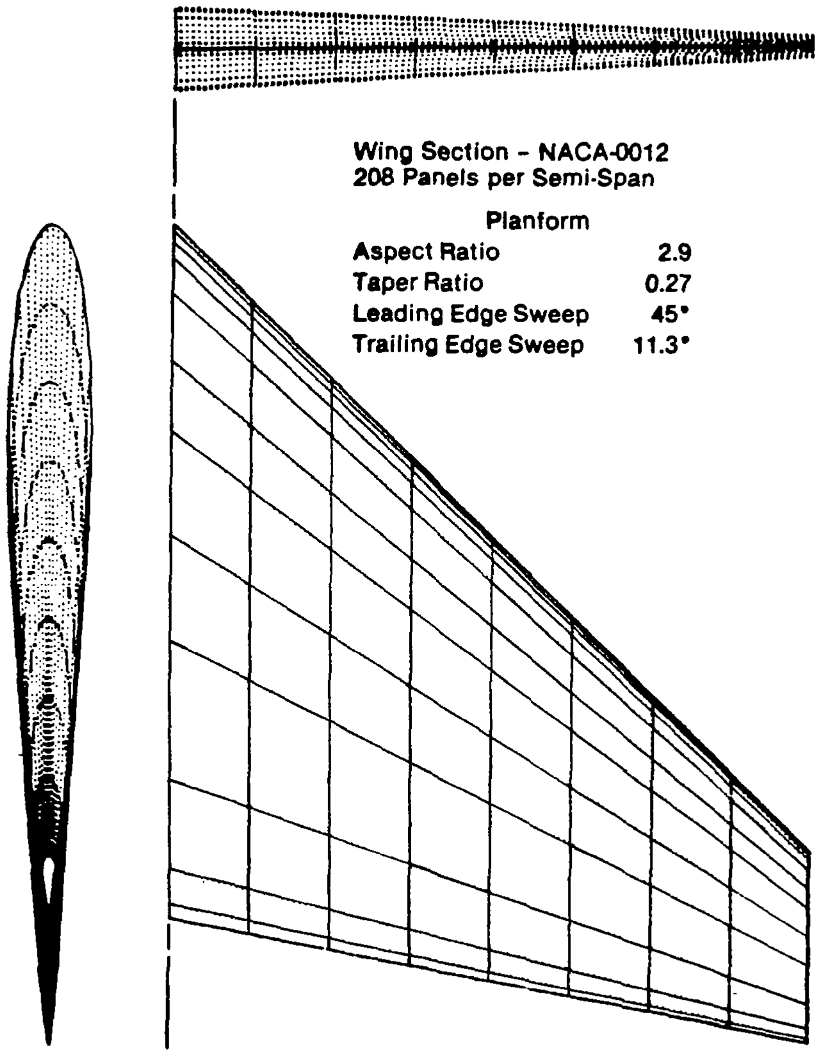


Figure 10a. Baseline Wing Paneling for Perturbation Analysis Test Cases

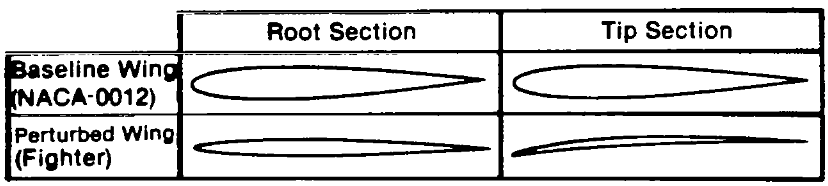


Figure 10 b. Geometry Perturbation Test Case
Fixed Planform Geometry

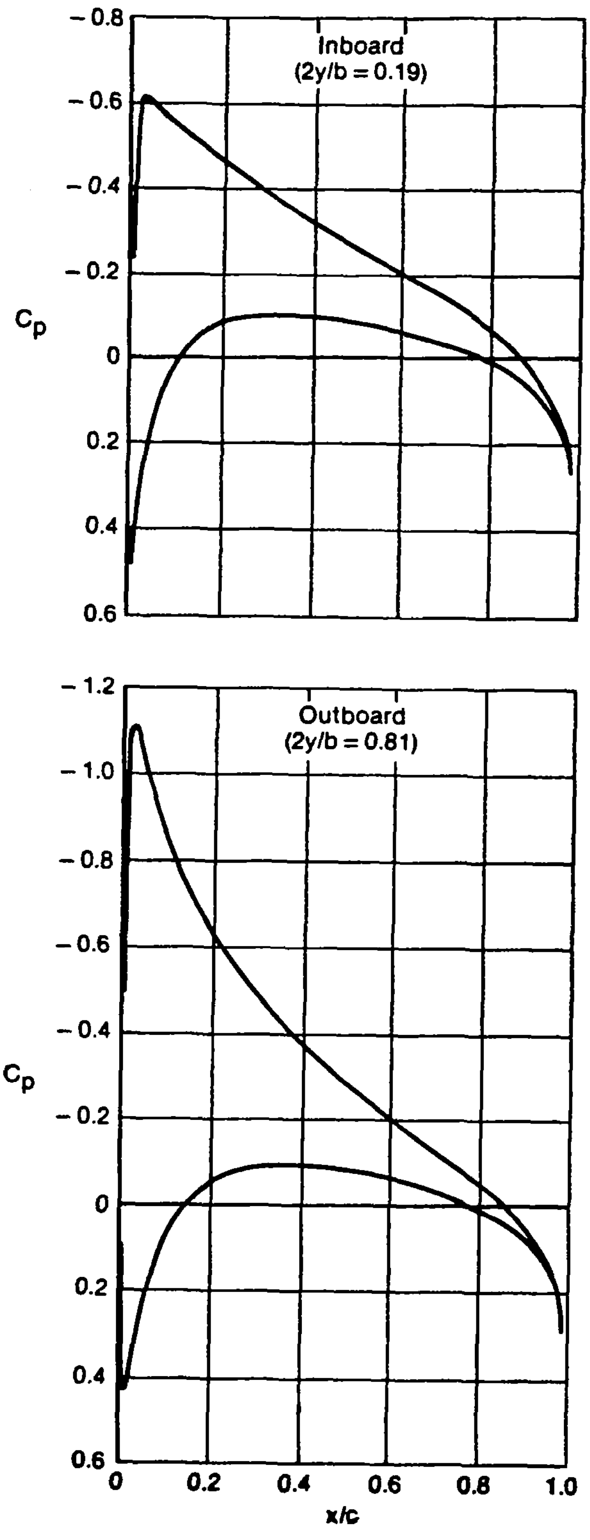


Figure 11a. Baseline Pressure Distribution at 5° Angle-of-Attack MCAIR Panel Method

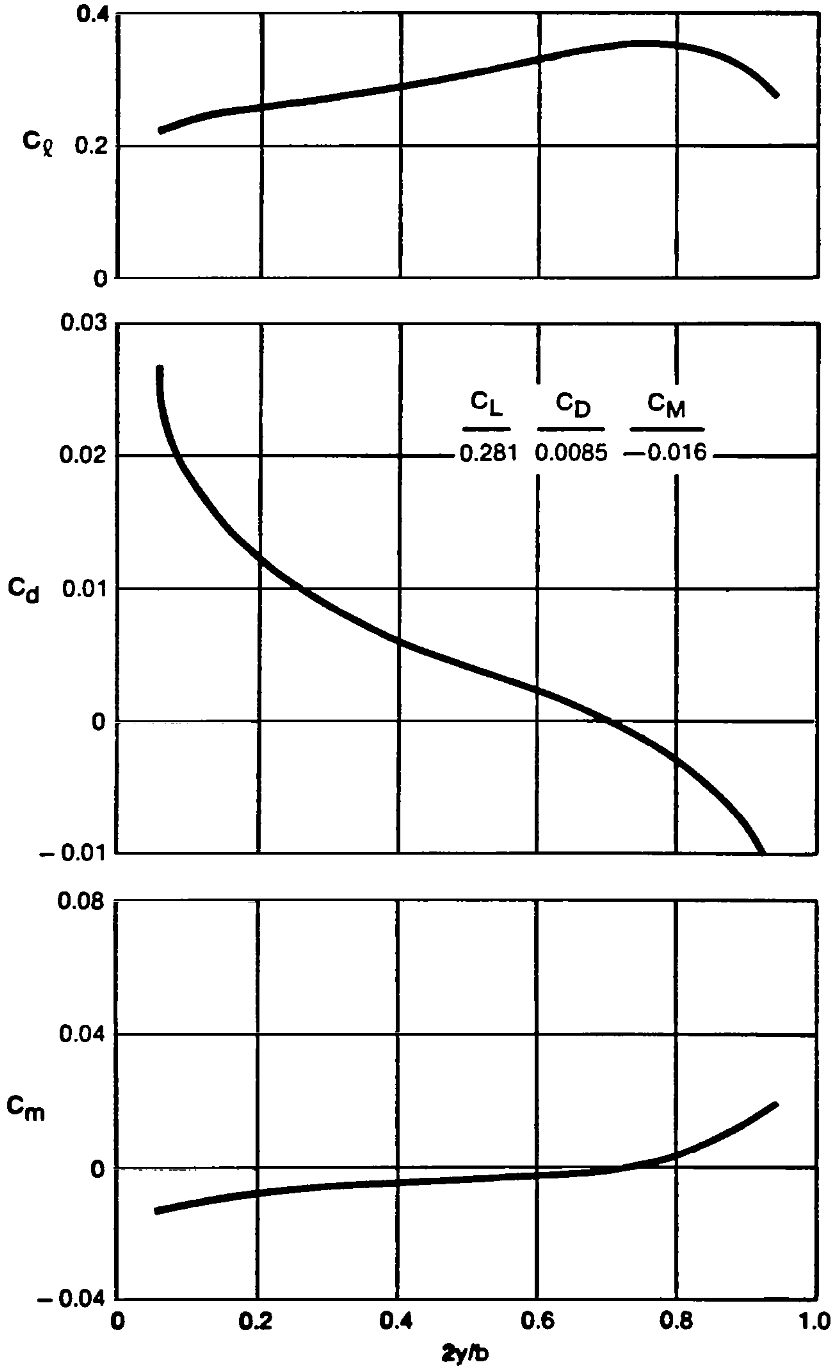


Figure 11b. Baseline Force and Moment Distribution at 5° Angle-of-Attack MCAIR Panel Method

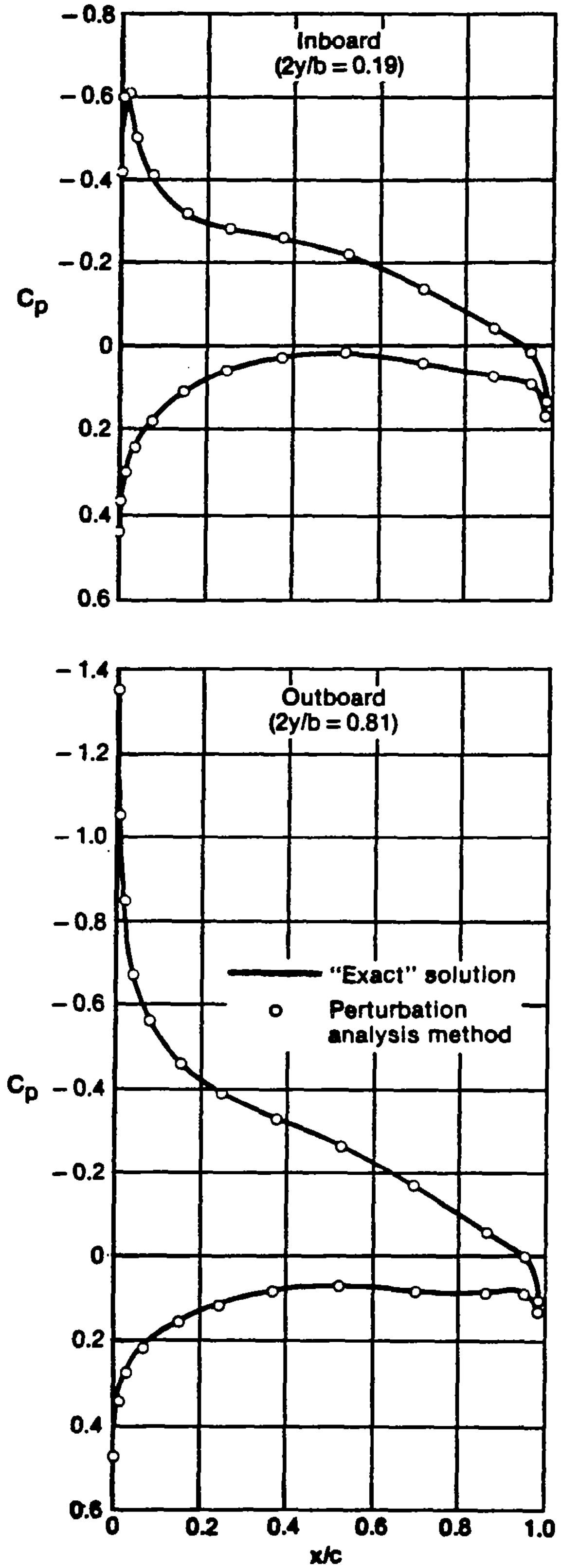


Figure 12a. Pressure Distribution for Perturbed Wing at 5° Angle-of-Attack

Fighter

		C_L	C_D	C_M
———	"Exact" solution	0.318	0.0112	−0.038
o	Perturbation analysis method	0.320	0.0106	−0.038

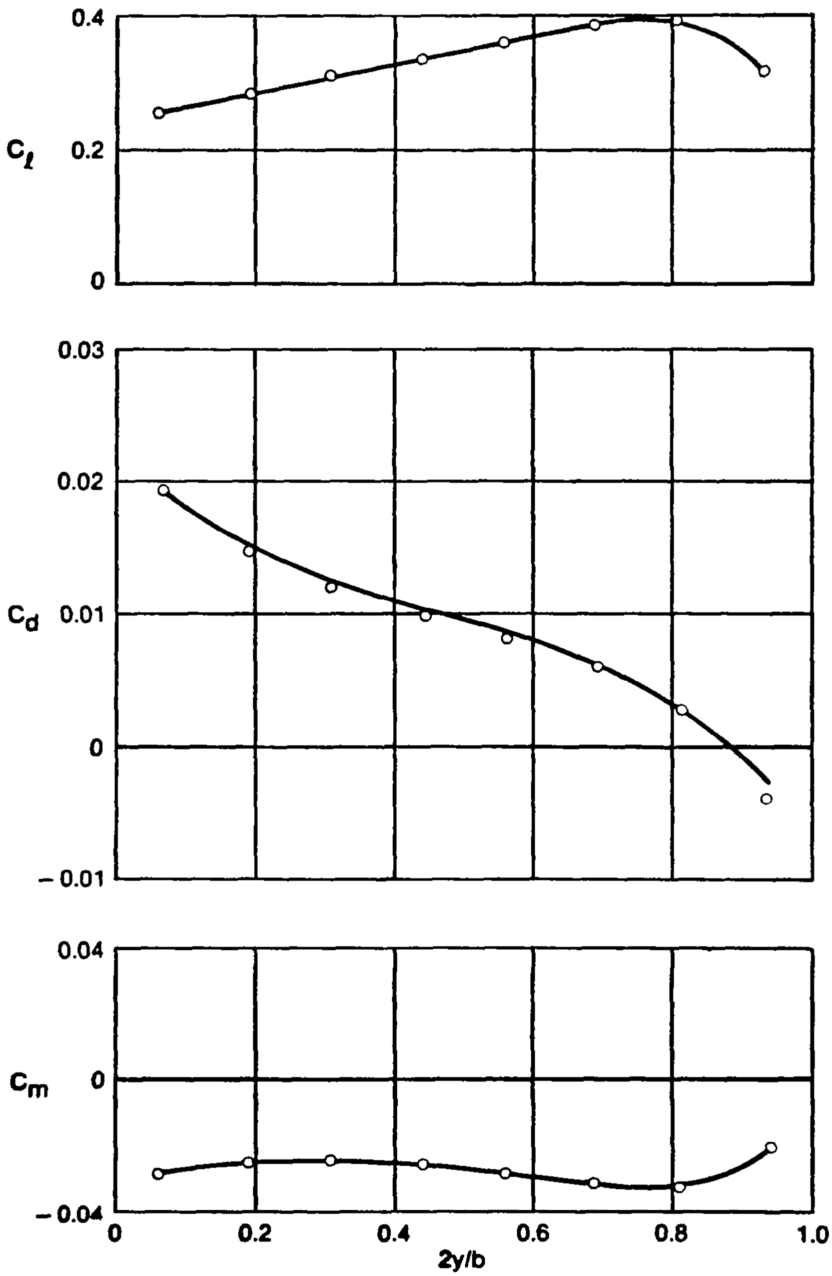


Figure 12b. Force and Moment Distribution for Perturbed Wing at 5° Angle-of-Attack

Fighter

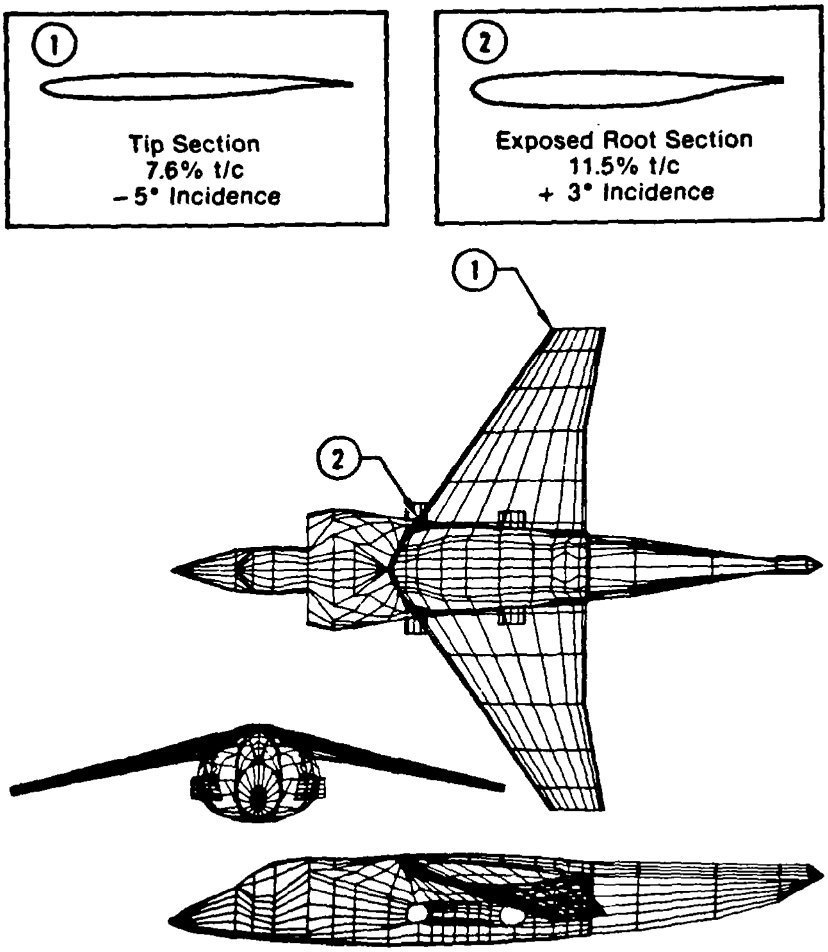
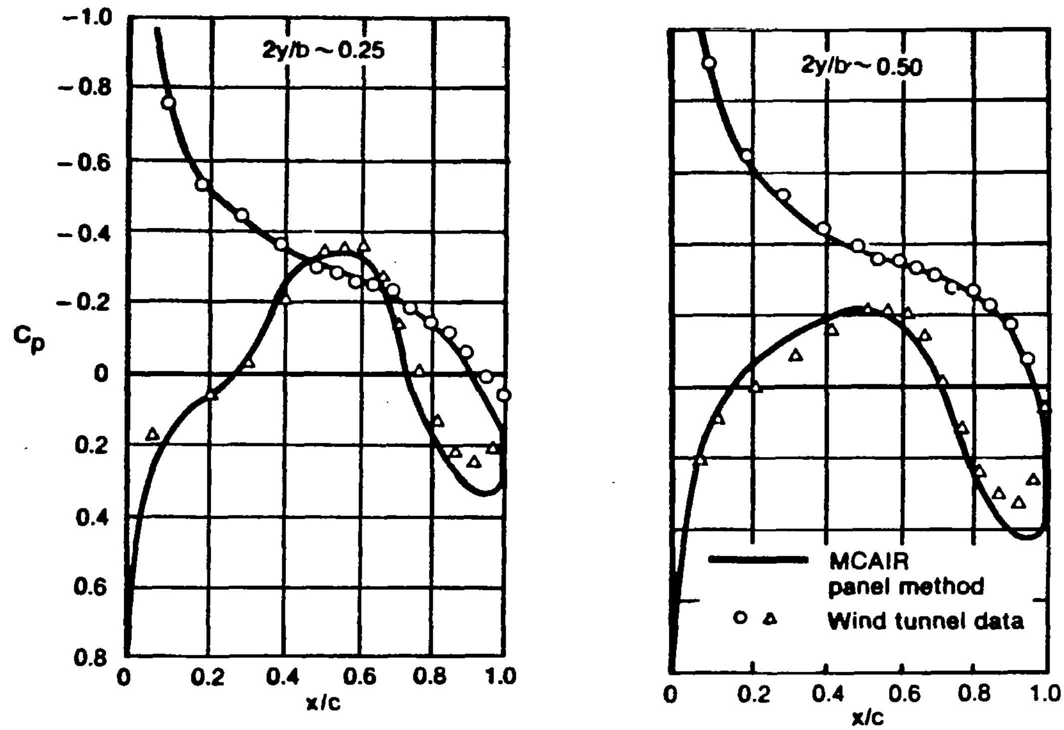
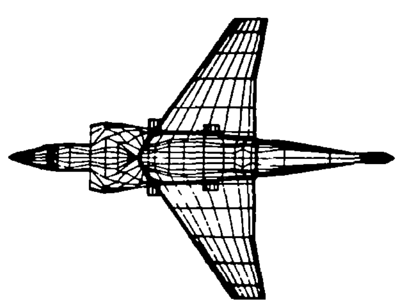


Figure 13. Paneled Representation of the YAV-8B 537 Panels on Each Side of Symmetry Plane





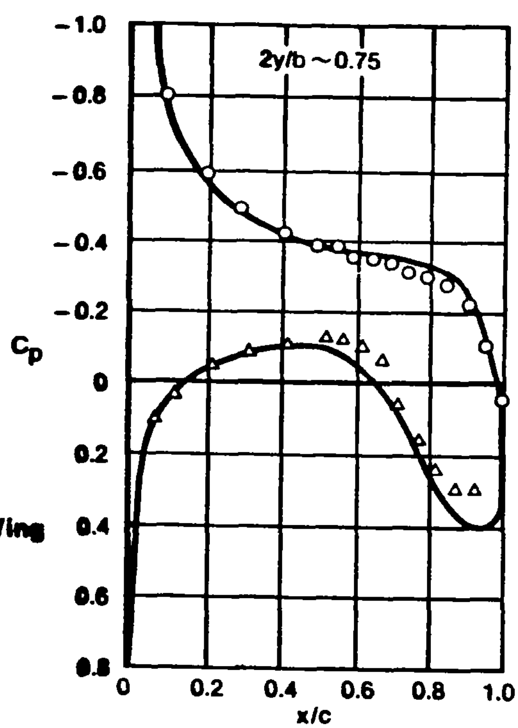
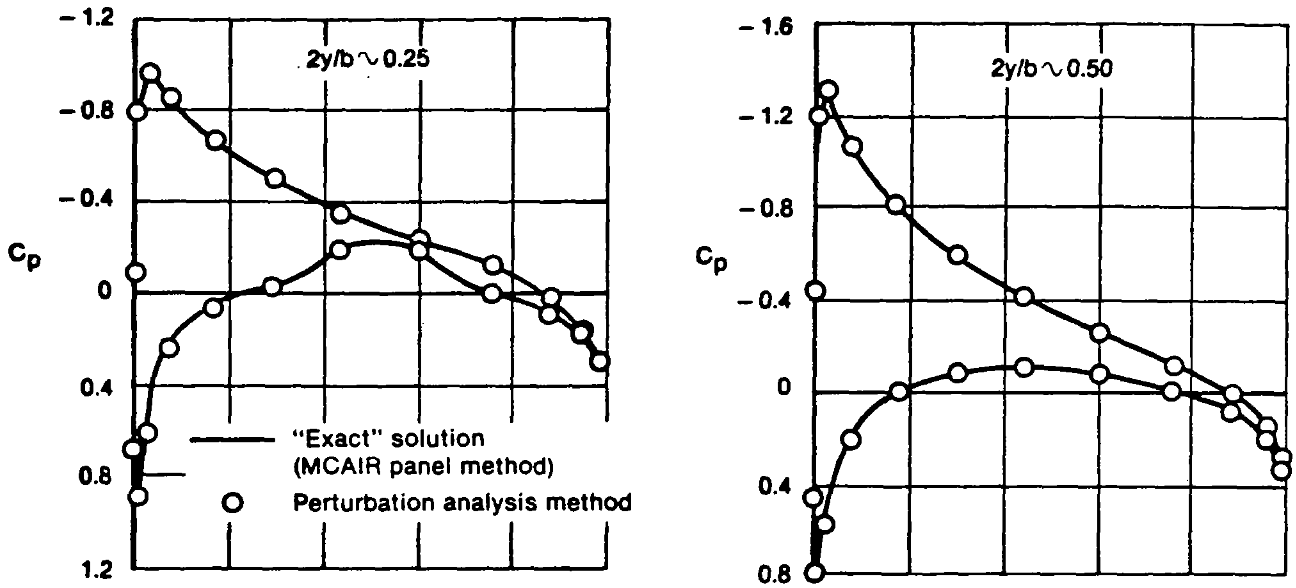
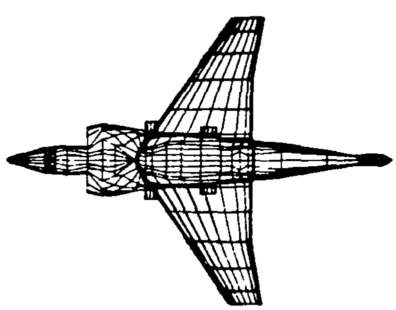


Figure 14. Pressure Distribution on YAV-8B Wing
6.4° Angle-of-Attack Mach 0.50





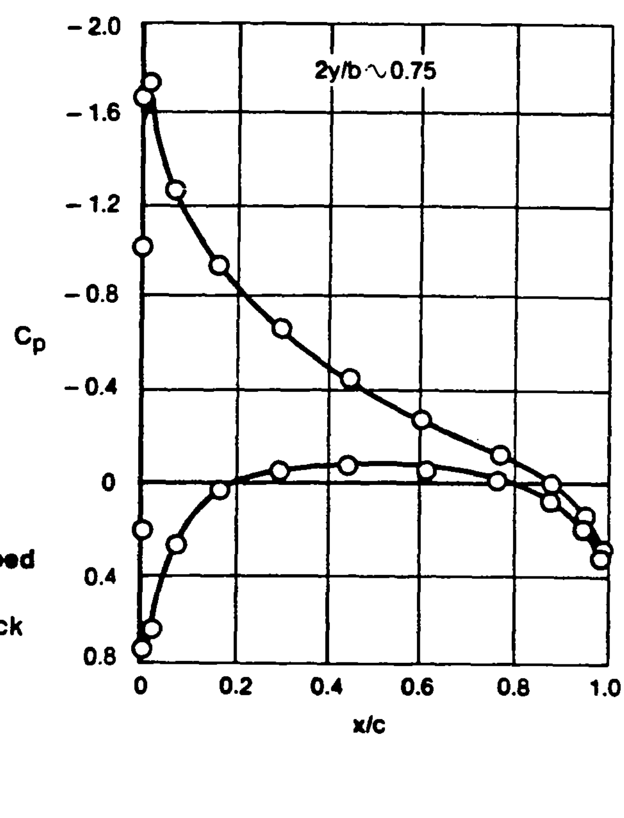


Figure 15. Calculated Pressure Distribution on Perturbed YAV-8B Wing

Uncambered, Untwisted Wing 6.4° Angle-of-Attack

Mach 0.50

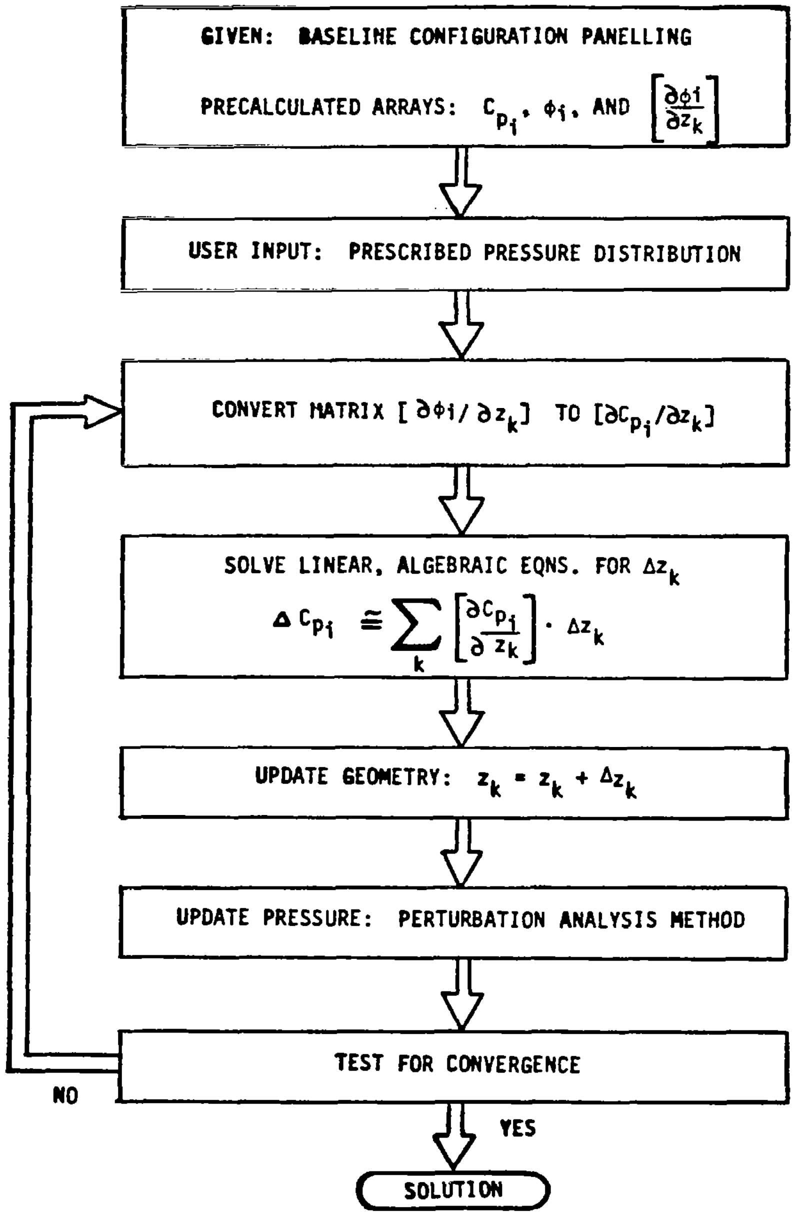


Figure 16. Schematic of Perturbation Design Method

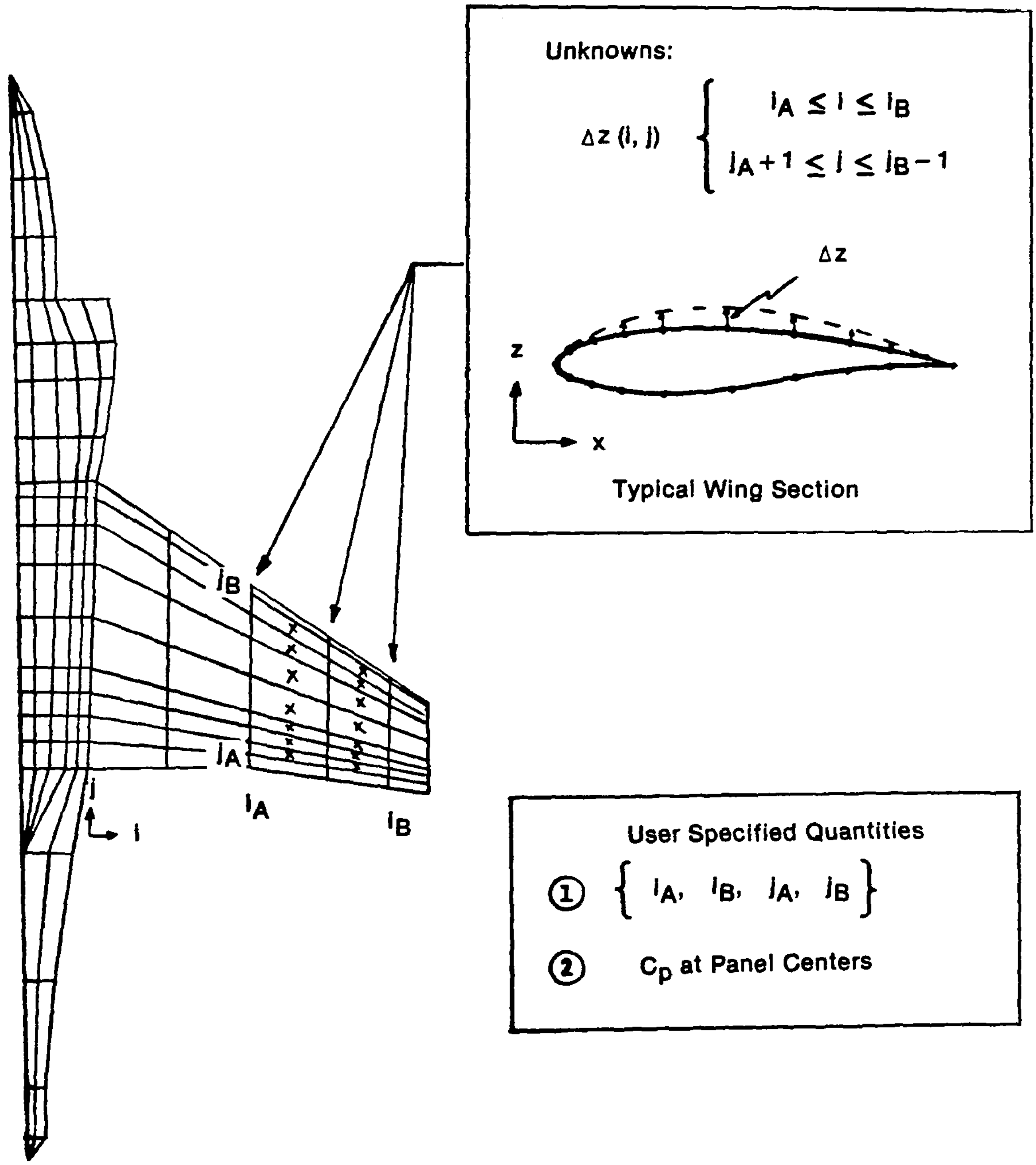


Figure 17. Standard Wing Design Problem

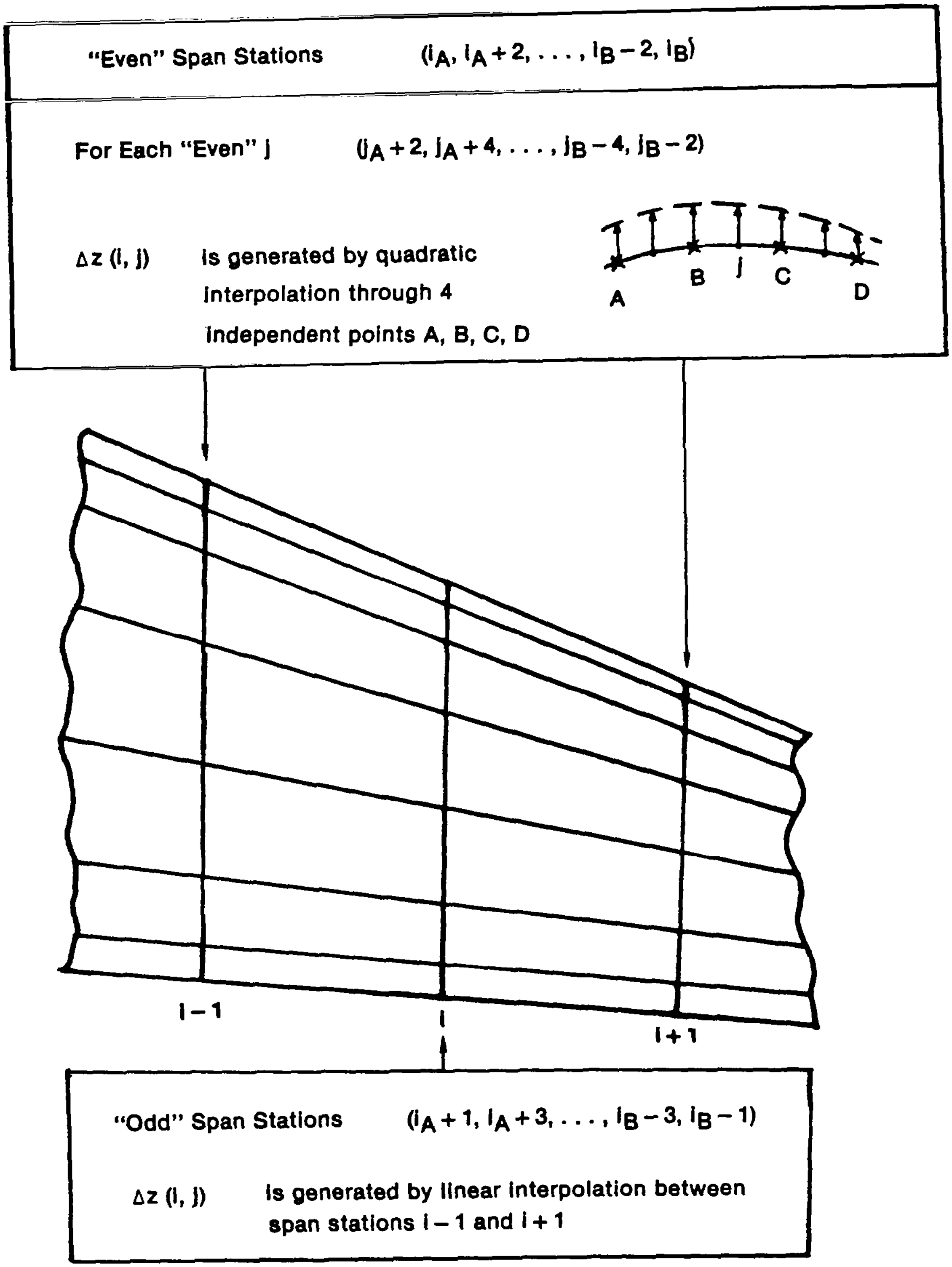
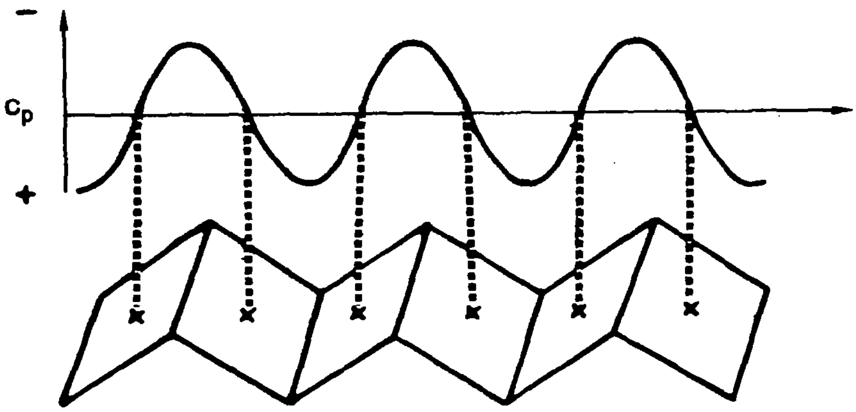
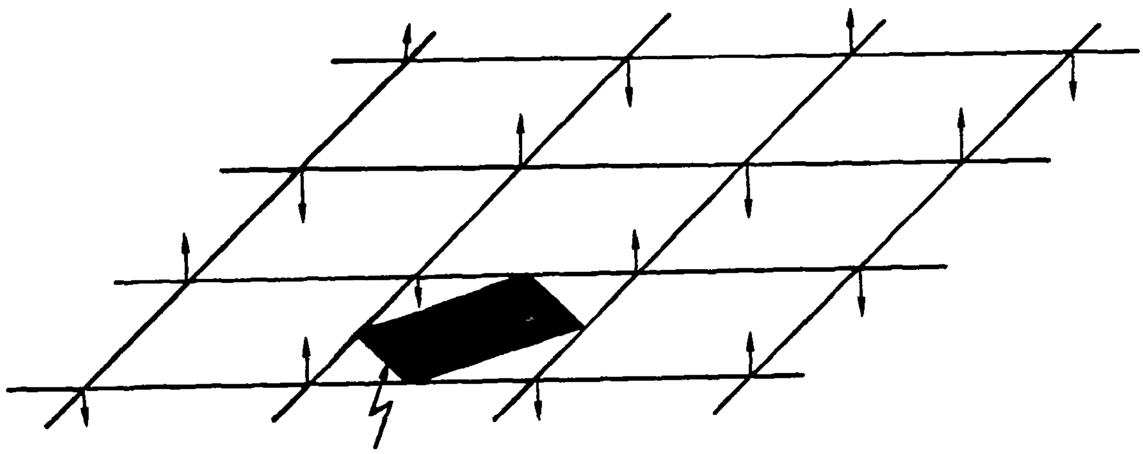


Figure 18. Interpolation Scheme for Dependent Unknowns



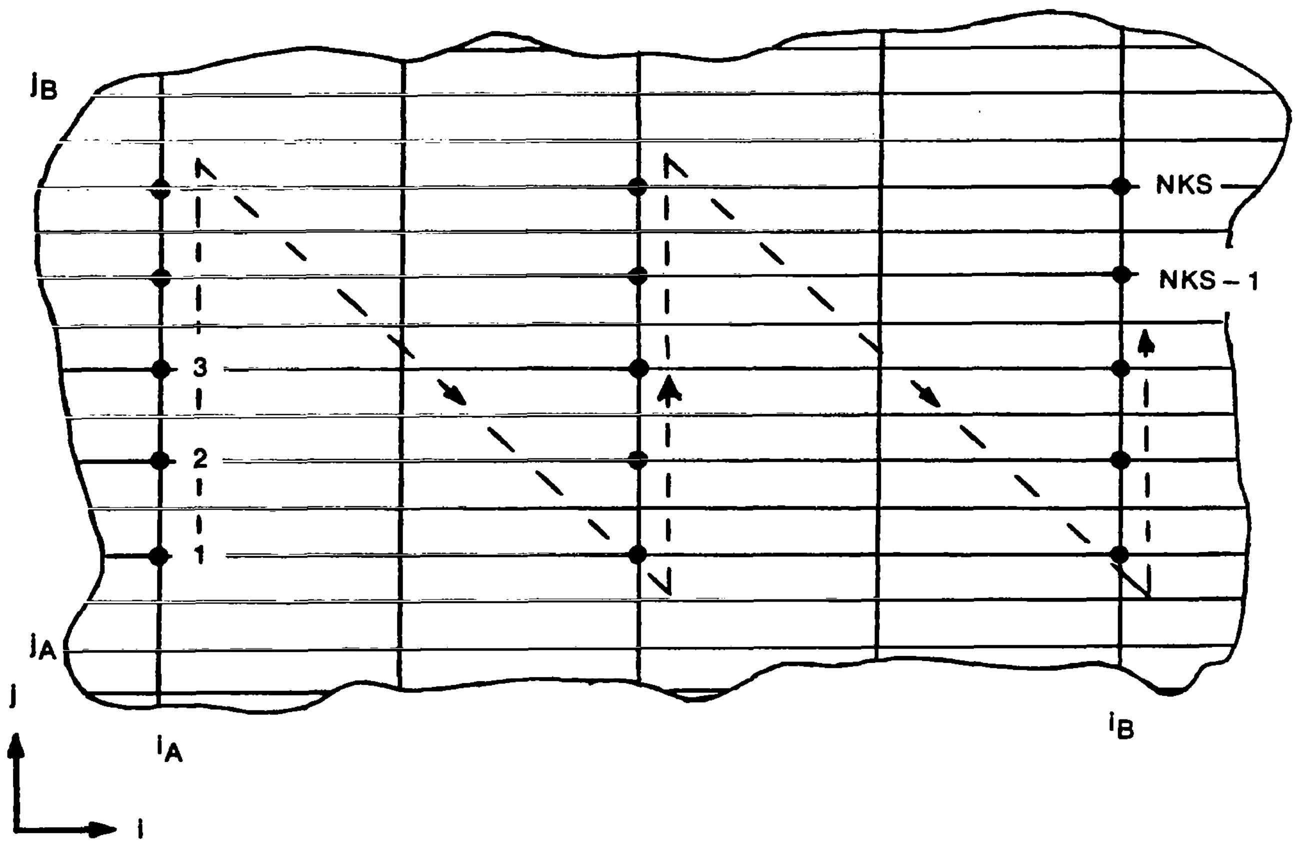
Pressure at Panel Center Does Not Control
Streamwise Slope Oscillations

Figure 19a. Wavy Wall Design Instability



Average Plane of Panel Does Not Control
Corner Point Oscillations

Figure 19b. Four-Corner Design Instability



$$S_{k_S} = \Delta z \qquad (1 \le k_S \le NKS)$$

Figure 20. Ordering of Independent Unknowns S_{k_S}

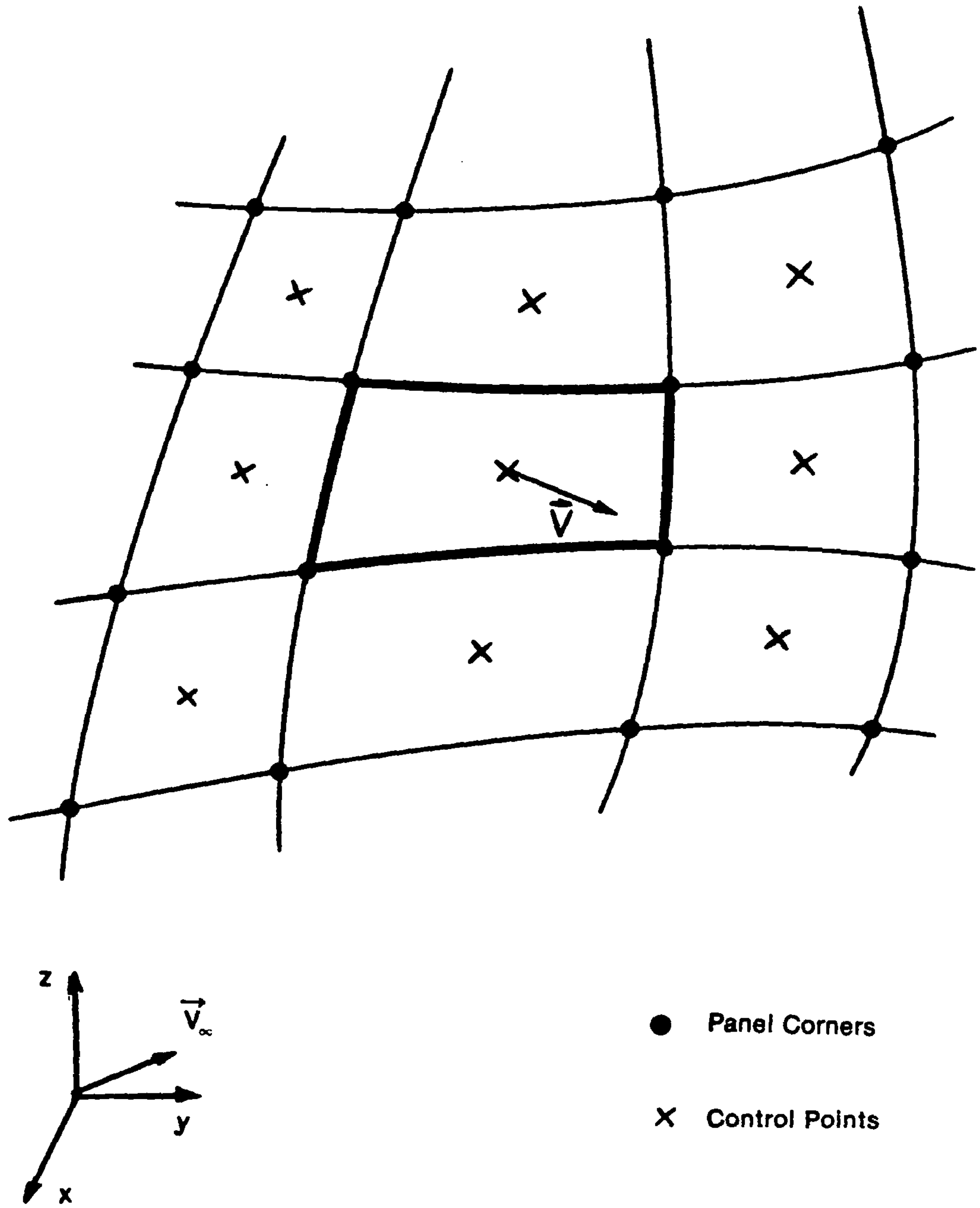
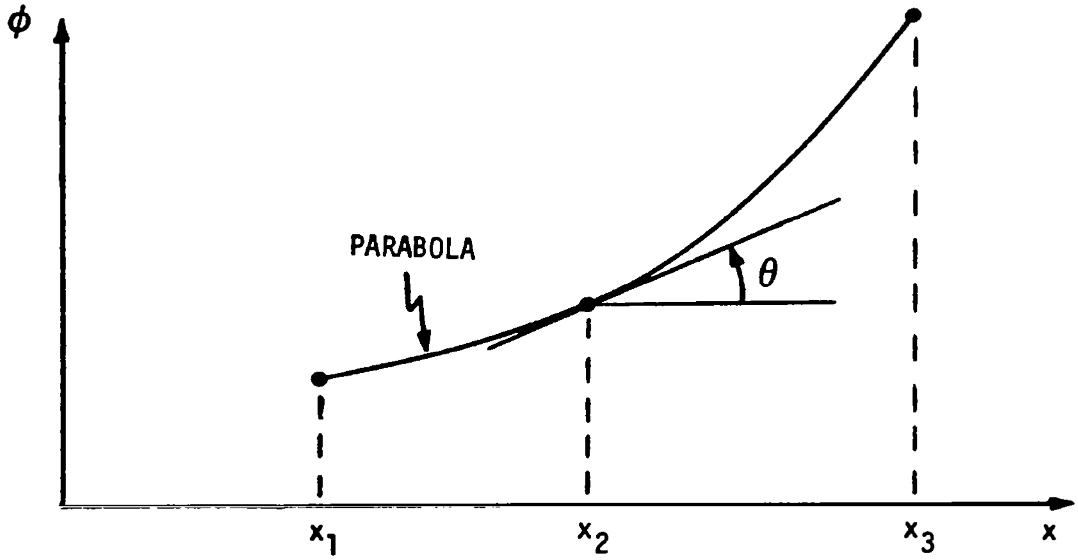


Figure 21. Schematic of a Grid of Panels

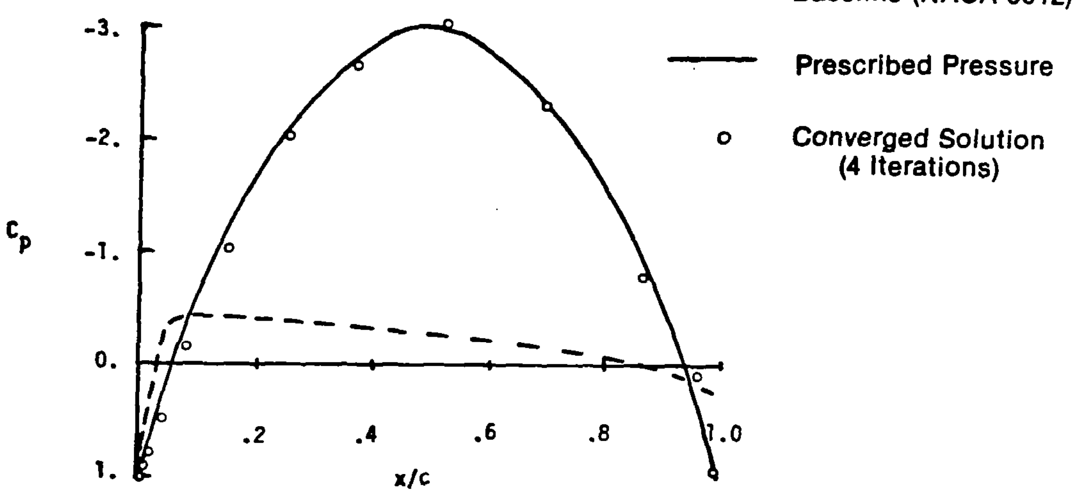


$$\tan\theta = \left(\frac{d\phi}{dx}\right)_{x_2} = a_1\phi_1 + a_2\phi_2 + a_3\phi_3$$

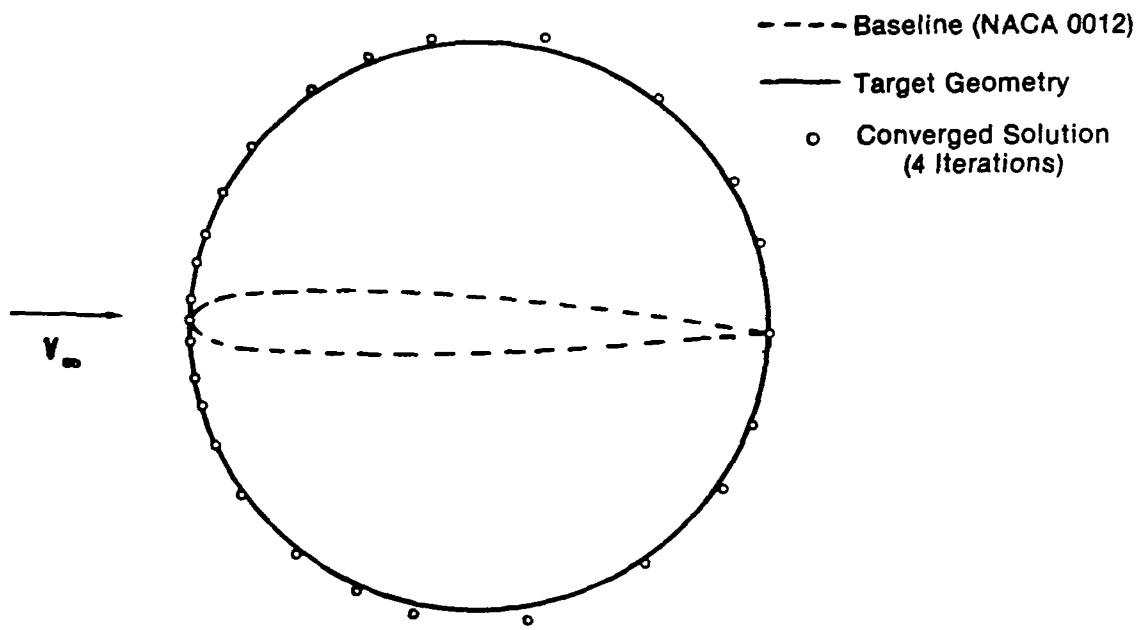
WHERE

$$\begin{cases} a_1 = \dfrac{-(x_3-x_2)}{(x_2-x_1)(x_3-x_1)} \\[2ex] a_2 = \dfrac{\left(\dfrac{x_3-x_2}{x_2-x_1}\right)-\left(\dfrac{x_2-x_1}{x_3-x_2}\right)}{(x_3-x_1)} \\[2ex] a_3 = \dfrac{(x_2-x_1)}{(x_3-x_2)(x_3-x_1)} \end{cases}$$

Figure 22. Simple One Dimensional Numerical Differentiation

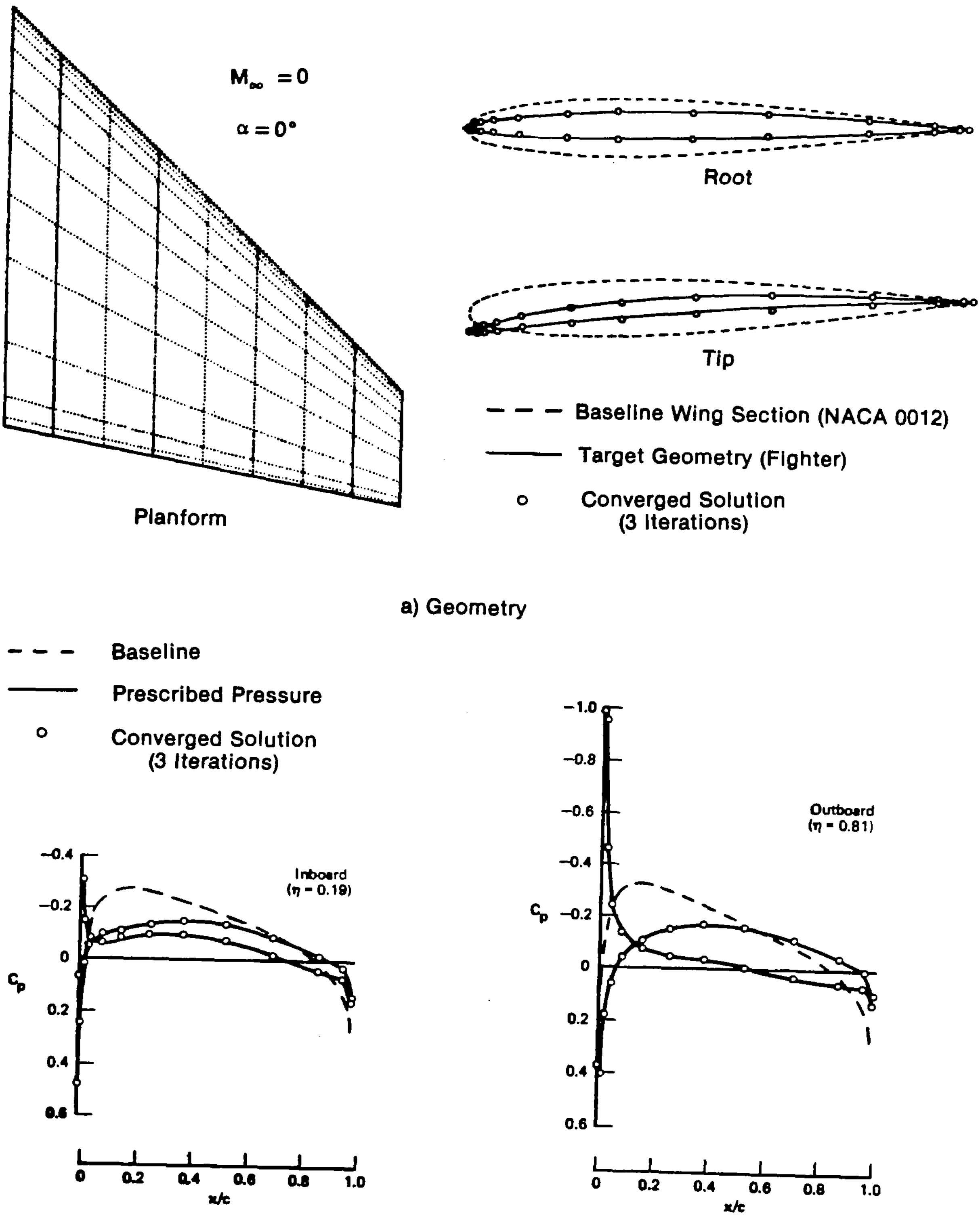


a) Pressure Distribution



b) Geometry

Figure 23. Circular Cylinder Design Solution



a) Geometry

b) Pressure Distribution

Figure 24a. Example Wing Design Solution

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15. Supplementary Notes

Langley Technical Monitor: Robert Weston
Topical Report

16. Abstract

An iterative method has been developed for designing wing section contours corresponding to a prescribed subcritical distribution of pressure. The calculations are initialized by using a surface panel method to analyze a baseline wing or wing-fuselage configuration. A first-order expansion to the baseline panel method equations is then used to calculate a matrix containing the partial derivative of potential at each control point with respect to each unknown geometry parameter. In every iteration cycle, the matrix is used both to calculate the geometry perturbation and to analyze the perturbed geometry. The distribution of potential on the perturbed geometry is established by simple linear extrapolation from the baseline solution. The extrapolated potential is converted to pressure by Bernoulli's equation. Not only is the accuracy of the approach good for very large perturbations, but the computing cost of each complete iteration cycle is substantially less than one analysis solution by a conventional panel method. Example design solutions are presented to demonstrate that the method is accurate, numerically stable, and converges in only three to five iterations.

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Panel Method	Wing Design	
Potential Flow		
Pressure Distribution		
Body-Wing Configuration		Subject Category 05

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