Figure 3 shows the results obtained on the fine grid with the two dissipation models. The effect of reducing artificial dissipation levels by means of MDM is seen to be minimal on both the wing and the fuselage solutions, indicating that the solutions are nearly grid-converged, on the fine grid employed.

Figure 4 shows the solutions with MDM for $M_\infty = 0.80$, $\alpha = 2$ deg, on only the fine grid. Overall comparison with the experimental data is again found to be very good on the wing surface. The pressure distributions compare quite well with experimental data, except near the nose and the wing-fuselage juncture. There are perhaps two reasons: 1) grid skewness near the nose of the fuselage, and 2) inadequacy of the simple algebraic turbulence model (Baldwin-Lomax) for predicting complex flows that are encountered in the wing-fuselage juncture region. Further details can be found in Ref. 9.

Summary and Conclusions

Transonic Navier-Stokes solutions were obtained for a transport (RAE) wing-fuselage configuration. A *C-O* grid topology was used since it has the advantage of requiring strong clustering one coordinate direction for resolving the boundary layers developing on both wing and fuselage surfaces. Extremely fine grid density (1.52 million nodes) was employed to obtain accurate numerical solutions, and the resulting pressure distributions compared well with experimental data. Solution convergence for each case was found to be very fast, the CPU time being approximately 8 h.

A grid refinement study was also conducted to assess the effect of artificial dissipation models (SDM and MDM), and truncation errors on the numerical solutions. Based on this study, it was concluded that the fine grid solutions were grid-converged. Computed results using MDM on the fine grid compared best with the experimental data.

Acknowledgments

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Effect of Leading-Edge Geometry on Delta Wing Unsteady Aerodynamics

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Introduction

THE complexity of the flowfield on aircraft and aircraft-like configurations at high angles of attack prohibits the use of numerical computational methods for preliminary design. Also because of the continual changes in the early design, a purely experimental method cannot be used. One needs rapid computational methods to guide the early stages of preliminary design until a firmer design has evolved on which experimental and numerical methods can be applied.

The simple flow concept developed by Polhamus,[1] i.e., the leading-edge suction analogy, was used in Ref. 2 as a starting point in the development of a fast prediction method for the unsteady aerodynamics of sharp-edged delta wings. This note extends the prediction to include the effect of leading-edge cross-sectional shape.

Figure 1 shows how the delay of crossflow separation to $\alpha_{LE} > 0$, caused by the leading-edge geometry, results in a delay to $\alpha > \alpha_v$ before leading-edge vortices are generated, where α_v is

$$\alpha_v = \tan^{-1}(\tan \alpha_{LE} \sin \theta_{LE}) \tag{1}$$

When the leading-edge geometry is of the type sketched in Fig. 1a, α_{LE} is determined directly by the geometry as $\alpha_{LE} = \delta_{LE}$. However, in the case of a rounded leading edge (see sketch in Fig. 1b), $\alpha_{Le} = \alpha_{sn}$, where α_{sn} is the crossflow sep-

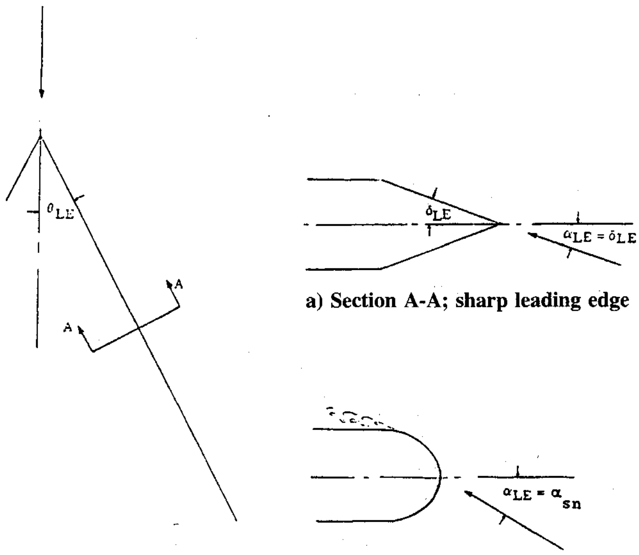

Fig. 1 Delta wing leading-edge geometry.

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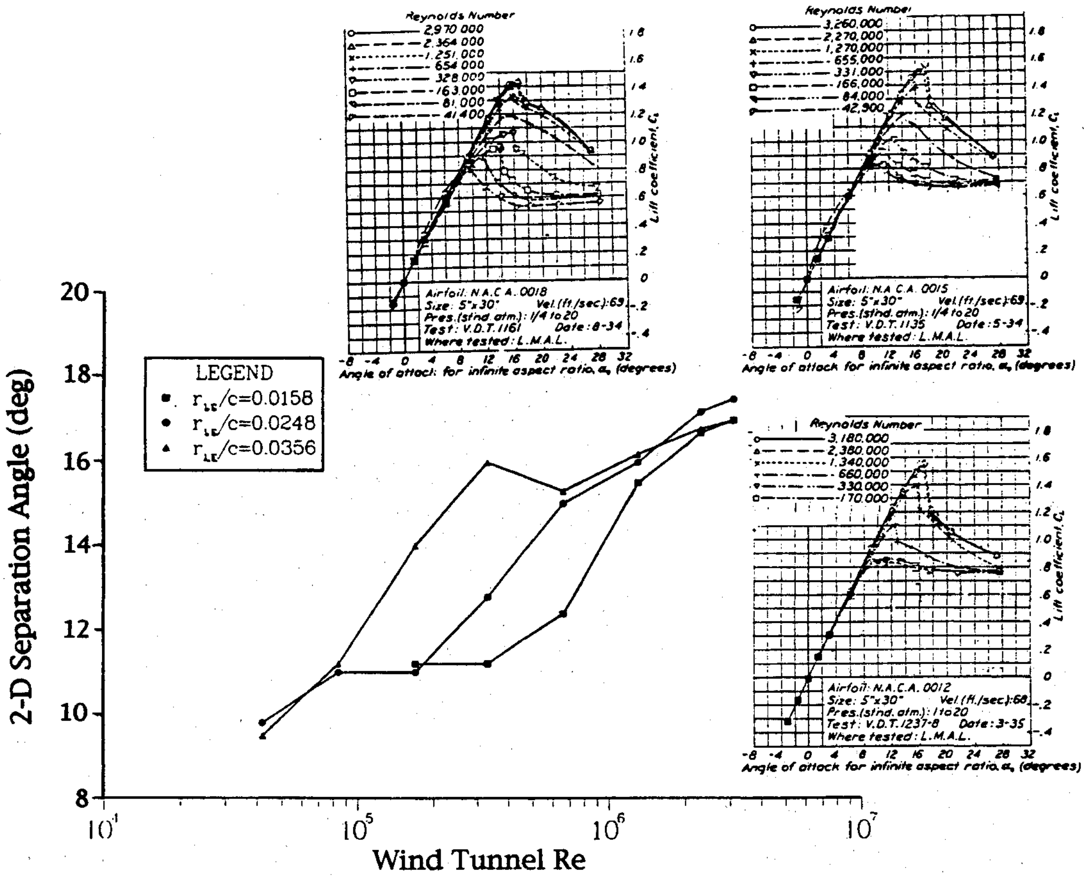


Fig. 2 Effect of Reynolds number on airfoil section characteristics.

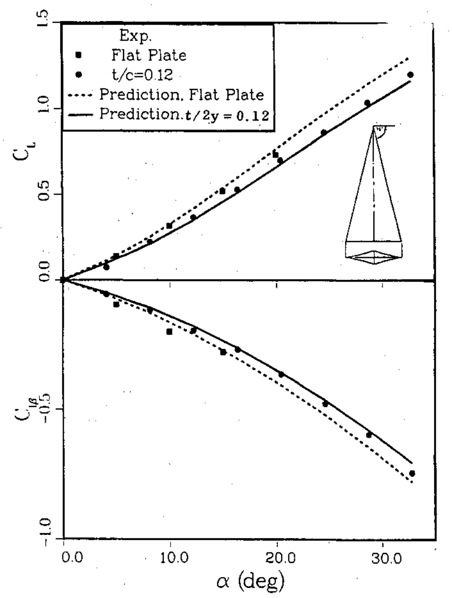


Fig. 3 Effect of wing thickness on the longitudinal aerodynamics of a 76-deg delta wing.

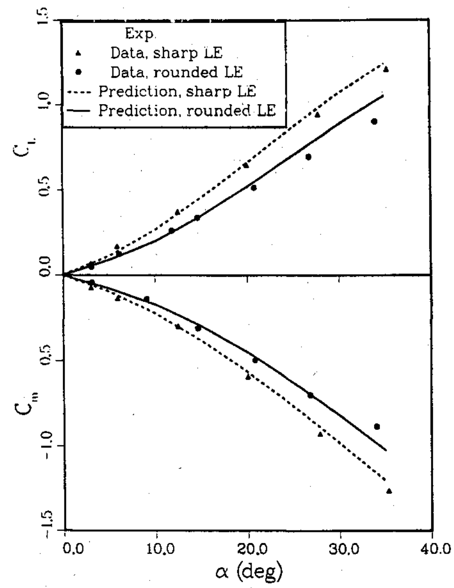


Fig. 4 Effect of leading-edge roundness on the static longitudinal aerodynamics of a 76.5-deg delta wing.

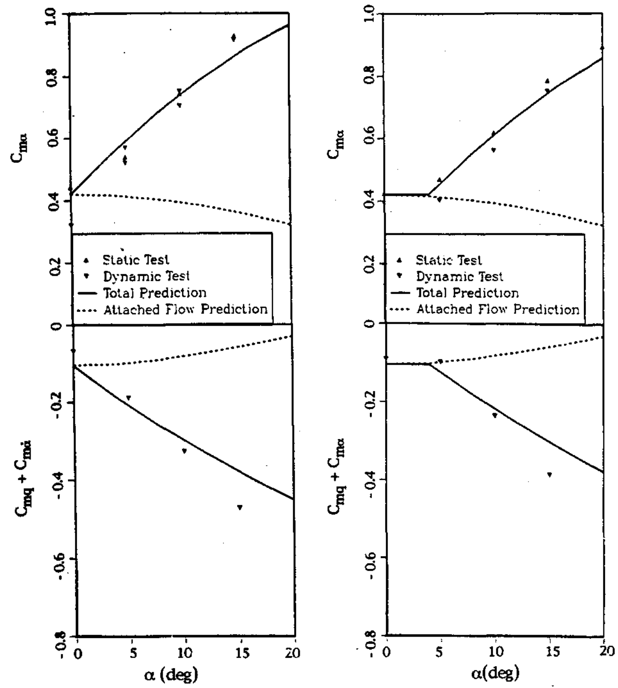
aration angle determined by the nose radius in combination with the crossflow Reynolds number.

Figure 2 shows experimental results obtained for NACA-OOXX airfoils.[3] The limiting value for the separation angle appears to be $\alpha_s \approx 11$ deg for low and $\alpha_s \approx 18$ deg for high Reynolds numbers. It is shown in Ref. 4 that the growth of the vortex-induced lift is limited to the first 40% of the wing extent downstream of the apex. The effective (mean) Reynolds number for this region of the wing is roughly 20% of the Reynolds number based on the center chord. Thus, in a first approximation, a value $\alpha_{sn} \approx 11$ deg can be used for wind-tunnel test data and $\alpha_{sn} \approx 18$ deg for full-scale flight.

As the leading-edge flow separation with associated vortices does not occur until $\alpha > \alpha_v$, the angle of attack α in the equations in Ref. 2 for a sharp leading edge need to be substituted by the effective angle of attack $(\alpha - \alpha_v)$ in order to give the aerodynamic characteristics for wings of finite thickness. It should be pointed out that α_{sn} could be expanded to include the effect of leading-edge blowing[5,6] by expressing α_{sn} as follows:

$$\alpha_{sn} = (\alpha_s)_{2D} + \Delta\alpha_{sn}(C_\mu) \tag{2}$$

$\Delta\alpha_{sn}(C_\mu)$ is the additional delay of crossflow separation due to tangential leading-edge blowing, $C_\mu > 0$.



a) Flat plate b) Round leading edge

Fig. 5 Effect of leading-edge roundness on the steady and unsteady longitudinal aerodynamics of a 69.6-deg delta wing (c.g. at 75% chord).

Figures 3–5 demonstrate that the experimentally observed,[7–9] effects of wing thickness and leading-edge roundness on delta wing aerodynamics are well predicted. Inclusion of the effect of the cross-sectional geometry in the rapid prediction method developed earlier significantly improves the agreement with experiment. The high-alpha steady and unsteady aerodynamics of slender delta wing and wing-body configurations[10] are predicted with sufficient accuracy for preliminary design, as long as vortex breakdown does not occur.

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Alternative Solution to Optimum Gliding Velocity in a Steady Head Wind or Tail Wind

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Introduction

THE problem of graphically determining the velocity for the maximum gliding distance of an aircraft in a given atmosphere has been addressed by several authors.[1–4] A comprehensive numerical analysis was performed by Jenkins and Wasyl[5] in which they presented solutions for the optimum glide velocity and crab angle in a given wind and air mass sink rate. For the particular case of a direct head wind or tail wind and zero air mass sink rate, they found that the solution for the optimal velocity could be expressed as

$$V = (1/2a)[4a^2v^2 + 4a(c - bv)]^{1/2} - v \tag{1}$$

where V is the optimal glide velocity, v is the wind speed (tail wind is positive), and the coefficients a, b, and c are functions of wing loading, aspect ratio, the slope of the profile drag vs Reynolds number, and the best glide speed in zero-wind. This equation gives a mathematical solution to what has traditionally been shown by graphical analysis, that optimum glide velocity increases in a head wind and decreases in a tail wind.

An alternative solution to this particular problem can be found by assuming a parabolic drag polar for the glider and making small angle approximations to the glide angle. The optimum glide velocity is solved as an infinite power series involving only the wind speed and zero-wind optimum glide velocity, thus avoiding the requirement to find the coefficients in the previous equation.

Discussion

The analysis is begun by looking at an aircraft in a constant glide with velocity V and wind v (Fig. 1). The flight path angle of the glider with respect to the ground γ_g can be written as

$$\tan \gamma_g = (\dot{h}/V_g) \tag{2}$$

where $\dot{h}$ is the sink rate and V_g is the velocity of the glider over the ground. By making the approximation that γ_g is small, the flight path angle can be expressed as

$$\gamma_g = [\dot{h}/(V + v)] \tag{3}$$

The sink rate can be eliminated by the relationship

$$\dot{h} = (C_L/C_D)V \tag{4}$$

with C_D and C_L the aircraft coefficients of drag and lift, respectively. The coefficient of drag can be expressed in the form of a parabolic drag equation, where the first term is the

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