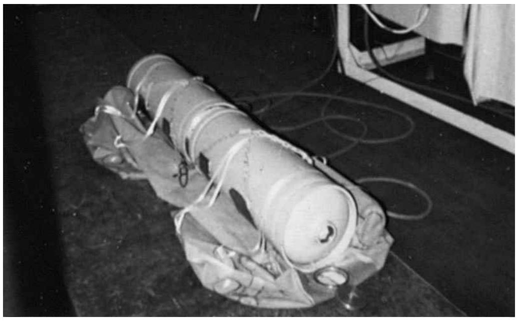

Fig. 2 Typical model–airbag system at the end of the drop test.

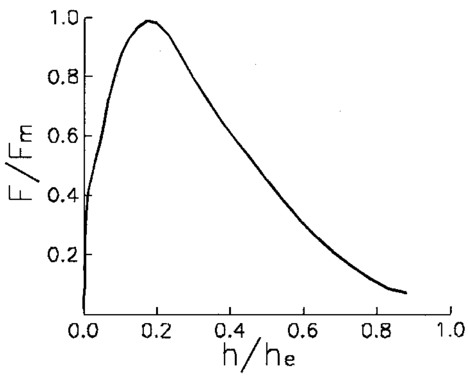

Fig. 3 Relationship between the retarding force and stroke.

cylindrical bags were not used for dissipating the impact energy, but instead were used for combining the two cube bags. Shear pin-type vents were applied to release gas to avoid rebounding at ground impact. The vent area was calculated with the method presented by Idomir.[2] The parameters of the circular-shaped bag were determined by the same procedure.

A photoelectron-quick-release and cable-suspension system suitable for an indoor test was used. With this equipment, the vertical- and horizontal-velocity components could be obtained. Accelerometers were glued near the model's c.g. to measure the vertical and horizontal acceleration. To monitor the dynamic pressure during impact, a pressure sensor was installed in the airbag. Test data were recorded on the oscillometer and processed by a computer.

In the case of vertical drops in various initial yaw or/and roll conditions, it is shown that the multicompartment airbag can provide better attenuation. A slight tipover occurs only under the poorest test conditions, 19-deg roll and 16-deg yaw. When the horizontal velocity component is in a given range. (<2 m/s, limited by the room size), the payloads drop steadily. The range is related to the width of bag and the height of the payload. Figure 2 shows a photograph of the typical model–airbag system at the end of the drop test.

The dynamic properties of the bags are also studied systematically. It is found that several factors affect the impact deceleration. Among these factors, the drop velocity and initial pressure have significant effects on the peak deceleration. Experiments indicate that the retarding force-stroke curve (Fig. 3), is different from the curve presented by Knache.[1]

Conclusions

A multicompartment structural airbag for preventing the payload–airbag system from tipover has been designed. Drop tests show that when the horizontal velocity component is in a given range, the new technique is very effective.

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Application of the Subsonic Doublet Lattice Method to Delta Wings

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Nomenclature

C_l = lift coefficient of wing
C_m = pitching moment coefficient about wing root midchord
C_p = pressure coefficient difference across wing
k_r = reduced frequency based on mean wing semichord
M = Mach number
n = number of boxes on wing

Introduction

THE subsonic doublet lattice method (DLM)[1–3] is commonly used for the calculation of unsteady airloads on aircraft. This method can be regarded as an extension of the vortex lattice method (VLM) that is widely used for steady load calculation. In both of these methods a lifting surface is divided into a number of smaller panels or boxes. A lifting line and a collocation point is associated with each box. The lift force acting on each box is assumed to act at the lifting line, whereas the boundary condition of tangential velocity is enforced at the collocation point. The lifting line is located along the box quarter-chord and the collocation point at the spanwise center of the box at three-quarter-chord. Downwash factors are calculated for each lifting line/collocation point combination. The set of lift forces that satisfy the boundary condition at all of the collocation points is solved from a set of linear equations.

The downwash factors in the DLM are calculated as the sum of a steady component, identical to that of the VLM, and an unsteady component. Whereas the steady component is exact, the unsteady component is approximated. The error introduced by the approximation increases as the box aspect ratio, i.e., the ratio of the box span to the box chord, is increased. Upper limits to the box aspect ratio have consequently been suggested to ensure acceptably accurate results.[3]

It is not difficult to panel wings with moderate taper ratios so that the box aspect ratio is fairly constant over the wing. Delta wings are different in this respect because the aspect ratio of the boxes in the outboard spanwise strip depends only on the number of chordwise divisions and the wing leading-edge sweep angle. If sufficient chordwise boxes are used to satisfy the convergence criteria with respect to the maximum box chord at the wing root, excessive box aspect ratios usually occur at the wing tip. It has been suggested[4] that the DLM totally breaks down for root-to-tip paneling schemes of delta wings. Delta wings are often divided into a number of spanwise sections with different numbers of chordwise divisions, as illustrated in Fig. 1. In the case of large models, this practice has the advantage of reducing the number of boxes in the model, and consequently, the computational effort required for a solution. In

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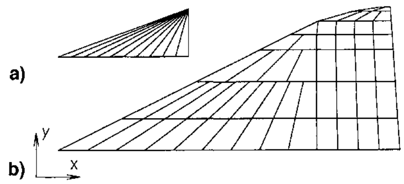

Fig. 1 Paneling of delta wings: a) outboard strip of a 70-deg delta wing with 10 chordwise boxes from root to tip, and b) tip region of a typical delta wing with an elevon, divided into spanwise sections with different numbers of chordwise boxes.

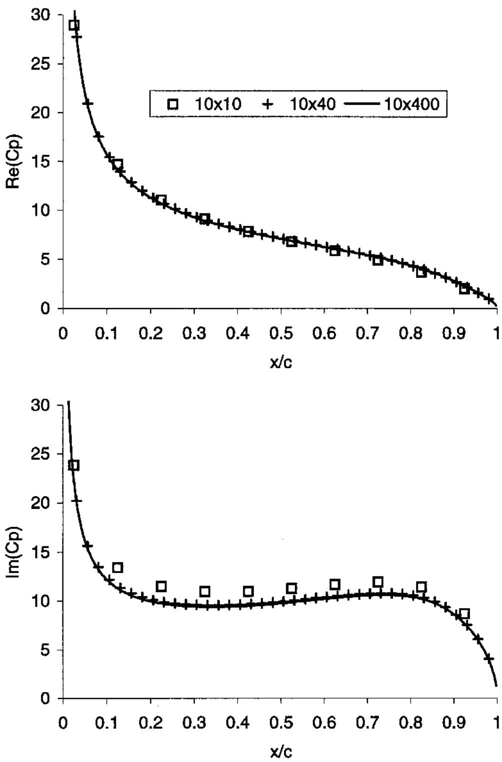

Fig. 2 Pressure distribution over the outboard strip of a 70-deg delta wing.

some cases, this saving does not justify the effort required to perform the paneling. This study shows that the excessive box aspect ratios resulting from using a root-to-tip paneling scheme for delta wings has no detrimental effect on the accuracy of the solution.

It has been shown[5] for the AGARD wing and horizontal tail, which have moderate taper ratios, that simultaneously refining the spanwise and chordwise paneling in such a way that the box aspect ratio is kept reasonably constant, results in almost linear convergence with respect to $1/\sqrt{n}$. This linear convergence behavior provides a means of estimating the discretization error by employing a fine and a coarse paneling scheme, and extrapolating to obtain an estimate of the fully converged solution. For delta wings, the condition of maintaining a constant box aspect ratio cannot be complied with. This study shows that simultaneously refining equally spaced spanwise and chordwise divisions of a delta wing also results in approximately linear convergence with respect to $1/\sqrt{n}$.

The computer code used to calculate the results presented here is an implementation of the DLM described in Ref. 3. Single-precision arithmetic was used throughout.

Effect of Excessive Box Aspect Ratios

Figure 2 shows the pressure distribution over the outboard spanwise strip of a 70-deg delta wing pitching about the root midchord at $k_r = 2$, $M = 0.8$. Pressure coefficient values are plotted at the box quarter-chord positions. The results were calculated using 10×10, 10×40, and 10×400 (span $\times$ chord) paneling schemes. The respective box aspect ratios at the tip are 7.28, 29.1, and 291. The results appear smooth for all three paneling schemes. In the real part, there is very little difference between the results for the different paneling schemes. In the imaginary part, there is a significant difference between the results for the 10×10 and 10×40 paneling schemes, but the results for the 10×40 and 10×400 paneling schemes are close to each other.

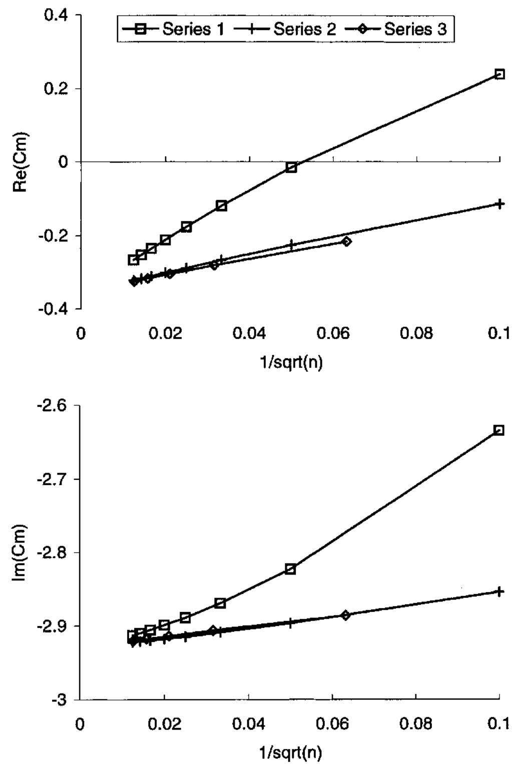

Fig. 3 Convergence histories for simultaneous refinement of spanwise and chordwise paneling.

Convergence Behavior

The convergence behavior of the DLM for a 70-deg delta wing is illustrated by plotting the unsteady pitching moment coefficient against $1/\sqrt{n}$ for simultaneous refinement of the spanwise and chordwise divisions (see Fig. 3). The first series of paneling schemes is (span $\times$ chord) 10×10, 20×20, 30×30, 40×40, 50×50, 60×60, 70×70, and 80×80; the second series is 5×20, 10×40, 15×60, 20×80, 25×100, 30×120, 35×140, and 40×160; and the third series is 5×50, 10×100, 15×150, 20×200, and 25×250. Equally spaced divisions were used for both the spanwise and the chordwise paneling.

The converged values were estimated as the y offsets of linear fits through the last four points of each series. The estimated converged values are -0.358–$i2.937$, -0.358–$i2.929$, and -0.354–$i2.930$ for series 1, 2, and 3, respectively. This represents a maximum relative difference (the norm of the maximum variation divided by the norm of the mean value) of only 0.3%.

Comparison with Slender Wing Theory

The lift and moment slope coefficients of a delta wing, calculated using the DLM at Mach 0.8, are compared with the slender wing approximation of Miles,[6] for leading-edge sweep angles from 70 to 89 deg in Fig. 4. It is seen that the DLM results approach the slender wing results as the leading-edge sweep angle is increased. There is very little difference between the DLM results for the 10×10 and 40×40 paneling schemes.

The comparison was extended to leading-edge sweep angles of 89.5, 89.75, 89.9, and 89.95 deg. The converged DLM results were estimated as twice the results for the 40×40 paneling scheme minus the results for a 20×20 paneling scheme. The differences between

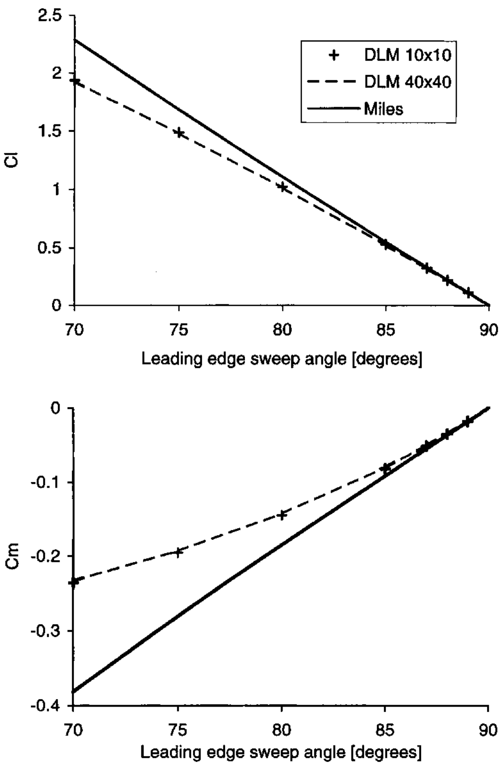

Fig. 4 Comparison with Miles' slender wing theory.[6]

the estimated converged DLM results and the slender wing theory decrease monotonically from 0.4% in C_l and 1% in C_m at 89.5 deg to 0.02% in both C_l and C_m at 89.95 deg. It would therefore seem that the DLM converges to the correct results for slender delta wings in steady flow.

Conclusions

The subsonic DLM is sufficiently robust to give credible solutions for delta wings with root-to-tip paneling schemes, despite the excessive box aspect ratios that usually occur at the tip. The present study suggests that there is no practical upper limit to box aspect ratio, with a box aspect ratio of 291 having been used without encountering any difficulties.

The convergence with respect to simultaneous refinement of equally spaced chordwise and spanwise divisions is approximately linear with respect to $1/\sqrt{n}$. This property is useful for estimating the discretization error of a particular paneling scheme, or for estimating the fully converged solution using extrapolation.

A comparison with slender wing theory indicates that the DLM converges to the correct results for slender delta wings in steady flow.

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Improvement to Numerical Predictions of Aerodynamic Flows Using Experimental Data Assimilation

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I. Introduction

THE idea to use measurements and/or observational data for correcting and updating numerical solutions has already been established in the context of weather forecasting in meteorology.[1–3] In this case, the capability of a numerical forecasting model depends not only on the resolution of the model and the accuracy with which dynamic and physical processes are represented, but it is also dependent critically on the initial conditions employed for integrating the model.

It is known from meteorology that observational data and/or measurements cannot be directly used to initialize a numerical forecast.[1] The data must be modified in a dynamically consistent manner to obtain a suitable data set for model initialization. This process is usually referred to as "data assimilation." Although the data assimilation technique is already established in the context of weather forecasting, an extensive literature survey showed that there is no systematic research regarding such an approach in the area of aerodynamics.

The aim of the present study is to investigate the idea of experimental data assimilation in the context of aerodynamic computations of turbulent compressible flows over airfoils. Therefore, the objectives of the paper are as follows:

1) To present the implementation of the data assimilation approach in conjunction with a compressible and turbulent Navier–Stokes solver.

2) To investigate the effects of pressure and velocity forcing on the numerical prediction of subsonic and transonic flows over an airfoil.

II. Experimental Data Assimilation

The compressible two-dimensional Navier–Stokes equations, in conjunction with a two-equation turbulence model, formed the computational basis for the present study. An implicit-unfactored method,[4] which solves the Navier–Stokes and turbulence transport equations in a strongly coupled fashion, has been employed. The solver uses a first order in time-implicit discretization scheme with Newton-type subiterations and Gauss–Seidel relaxation. A third-order characteristic-based scheme is used for discretizing the inviscid fluxes. The Launder–Sharma (LS)[5] and Nagano-Kim (NK)[6] k-ε models have also been employed. To improve the predictions of the LS model in separated flows, the Yap-correction term[7] has been included in the transport equation for $\tilde{\varepsilon}$.

Experimental data are usually available for surface pressure distributions, and, possibly, for the velocity profiles at certain positions around the geometry. We consider an airfoil for which experimental data for the pressure coefficient distribution C_p on the suction

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