improved accuracy of the channel efficiency, the individual test efficiencies for a number of flight test conditions are averaged. Thus there will be an average efficiency for each instrumented position on the wing leading edge.

Local surface pressure coefficients are required in order to calculate the local surface velocity and local Reynolds number. The local Reynolds number is needed to determine the local heat transfer coefficient. Local surface pressure coefficients are derived from potential flow computer codes for a Mach number of 0.35. The Karman-Tsien pressure correction formula is used to adjust these data for any variaton in aircraft Mach number. Local surface temperatures, required for calculating the channel efficiency, were measured during the test flights.

Utilizing the calculated pressure data and measured surface temperatures, the average channel efficiencies are computed. Icing analysis is performed by using the computed channel efficiency to predict spanwise and chordwise surface temperature distributions for any aircraft flight and icing condition. Surface pressure data and the local average channel efficiencies are used, along with the required aircraft con-

ditions and bleed air conditions, to compute local wing surface temperature.

Discussion of Results

The test of the validity of such a procedure as that proposed is to calculate wing leading edge surface temperatures based on known test conditions and compare them with the corresponding measured values. This was done for all of the test data available. A typical example of the results for natural icing conditions is shown in Fig. 3. It is seen that predicted surface temperatures are generally within ±10 °F of the measured values. It is apparent that the method is, in general, very conservative in that it tends to predict surface temperatures that are colder than those measured.

A comparison of calculated and measured values of T_S/T_A for natural icing conditions for all six instrumented wing stations and for each thermocouple location indicated that there appears to be a consistent ±5% scatter and that the method tends to be quite conservative by 5-10% of T_S/T_A.

Conclusions

The formulation of an accurate method for calculating chordwise and spanwise surface temperature distribution for the heated portion of a wing leading edge has been accomplished. It provides for the effects of droplet size and liquid water content and requires only information about aircraft, atmospheric, and hot bleed air conditions.

For icing conditions, the accuracy of the method is ±10% or less when compared to measured data and tends to be conservative. Thus the leading edge anti-icing system should maintain more of the leading edge clear of ice than indicated by the calculated surface temperature profiles.

Acknowledgment

The author expresses appreciation to the Gates Learjet Corporation for its cooperation in releasing the data for publication.

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[2]Heinrich, A., Ross, R., and Ganesan, N., "Engine Inlet Anti-Icing System Evaluation Procedure," FAA-RD-80-50, Jan. 1980.
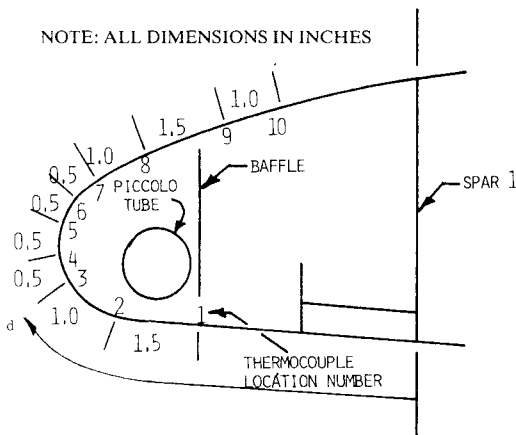

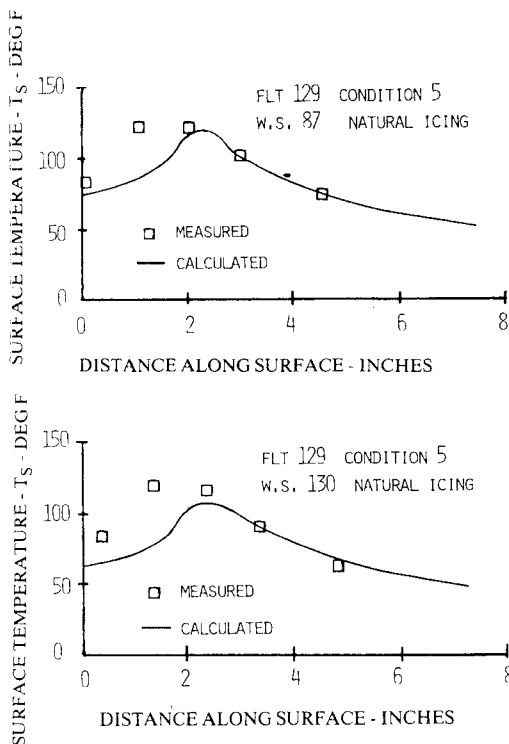Fig. 2   Typical instrumentation points on wing leading edge.

Fig. 3   Comparison of calculated and measured surface temperature, Flight 129, Condition 5.

Application of Panel Methods in External Store Load Calculations

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Nomenclature

$K1$	= wing leading-edge definition parameter
M_∞	= freestream Mach number
u, v, w	= perturbation velocity components, Fig. 1
V_∞	= freestream velocity

x, y, z = coordinate system, Fig. 1
α = angle of attack

Introduction

IN Ref. 1 the PAN AIR panel code was applied to external store load calculations. For reasons of cost effectiveness the linearized "thin wing" approximation was employed for the carrier aircraft wing. Although this would appear adequate, the author is not aware of any reported quantitative substantiation in this regard. This is one aspect to be addressed in the present Note. Also, wing thickness was not simulated in Ref. 1. Although this appears to have been acceptable in the cases considered in Ref. 1, the present Note will demonstrate that in general neglecting wing thickness is not permitted.

Another aspect in connection with cost effectiveness concerns the paneling of the aircraft. In calculating the aerodynamic forces and moments on flight configurations with the panel method, it is common engineering practice to apply a rather coarse paneling and thus reduce computer costs. This is usually adequate to obtain sufficiently accurate results for overall effects, such as forces and moments. In the calculation of the aerodynamic loads on external stores in close proximity to the carrier aircraft, it is of interest to know whether the influence of the aircraft on the external store loads is similarly well represented with coarse paneling of the aircraft. In this regard, the aircraft wing is of particular interest.

The effect of the modeling of the aircraft wing on external store loads is investigated by computing its influence on the external perturbation flow field generated by the wing. First, the effect of the paneling/boundary condition method applied (nonplanar, planar) on the perturbation flow field below the wing is considered. Second, the effect of the wing paneling in the leading edge region on the flow field is investigated, as the singularity distributions in this region show large gradients; both lift effects and thickness effects are considered.

Calculation Method

The results for the nonplanar boundary condition method are obtained with version B of Woodward's USSAERO computer code.[2] The panels lie on the upper and lower surfaces of the w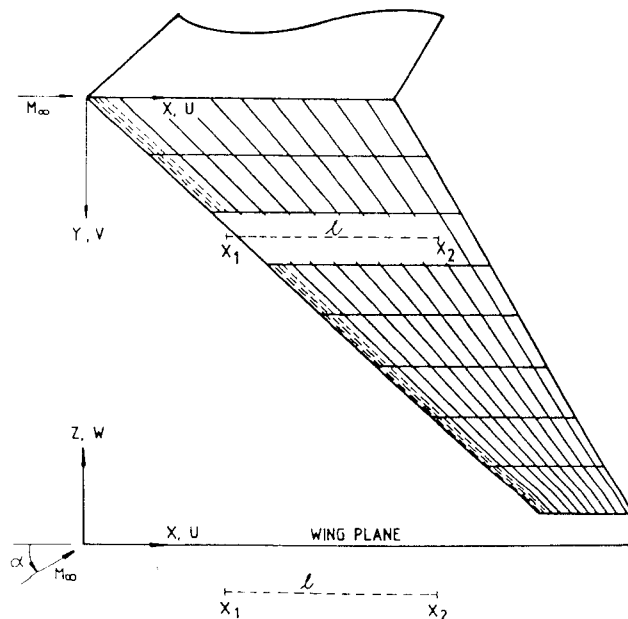ing; the boundary conditions of tangential flow are applied on the wing surface. The results for the more cost effective, but less accurate, planar boundary condition method are obtained with the USTORE code,[3] developed by the author using the basic concepts of the USSAERO code.[4] Now the panels lie in the mean wing plane and the boundary conditions are applied in this plane. In the planar b.c. method, the wing thickness effects are separated from the lift effects by equating the strengths of the source distributions (which represent the wing thickness) to the local wing thickness slope. This is the equivalent of the "thin wing" approximation. For a round leading edge, the local thickness slope is infinite and a special treatment is required to obtain a finite value for the leading edge source strength (and thus ensure finite perturbation velocities). This finite value must be such that it represents the round nose effects well. To this end, USSAERO and USTORE use a parameter $K1$ in the input data. A round leading edge is specified by $K1 = 3$, ensuring special treatment for the leading-edge source strength. A sharp leading edge is specified by $K1 = 1$.

Results

The wing to be considered is swept and tapered; it has NACA 65A006 airfoil sections with a leading-edge radius of 0.229% chord. The wing geometry and paneling are shown in Fig. 1. The spanwise paneling is fixed. Two chordwise paneling cases are considered. In Case A, the chords are evenly divided into ten parts. In Case B, the first 10% of the chords (the leading edge region) is paneled more densely by adding panel edges at 1.25, 2.5, and 5% chord (the dashed lines in Fig. 1). The axial line, along which the perturbation flow field components are calculated, is indicated as line ℓ in Fig. 1. This would be about the position of the longitudinal axis of an external store in carriage position.

Figure 2 shows the flow field velocity components calculated with the nonplanar and planar boundary condition methods, respectively, for Case B (denser paneling near the leading edge). The differences between the two methods are seen to be small along line ℓ.

The faster planar boundary condition method will be considered further. The computational results are presented in Fig. 3, showing the influence of wing thickness and leading-edge paneling on the perturbation flow field. For the zero-
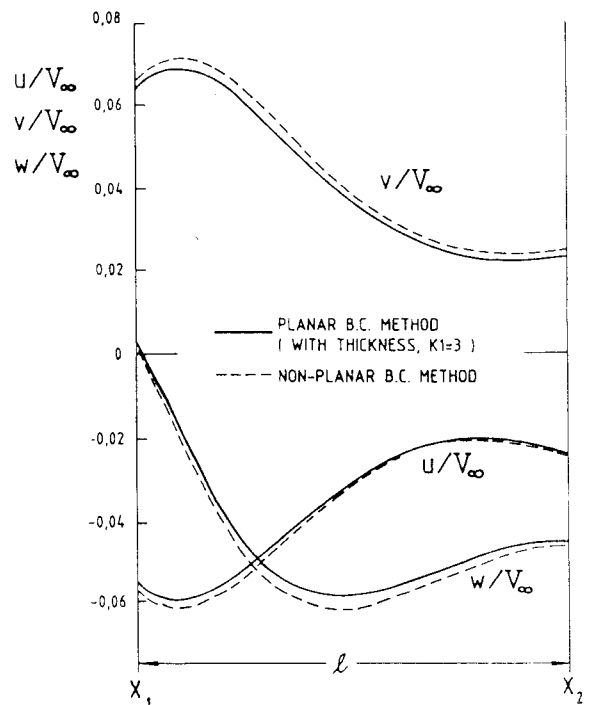

Fig. 1 Wing planform and paneling.

Fig. 2 Perturbation flow field along line ℓ (Case B: densely paneled leading edge); $M_\infty = 0.4$, $\alpha = 6$ deg.
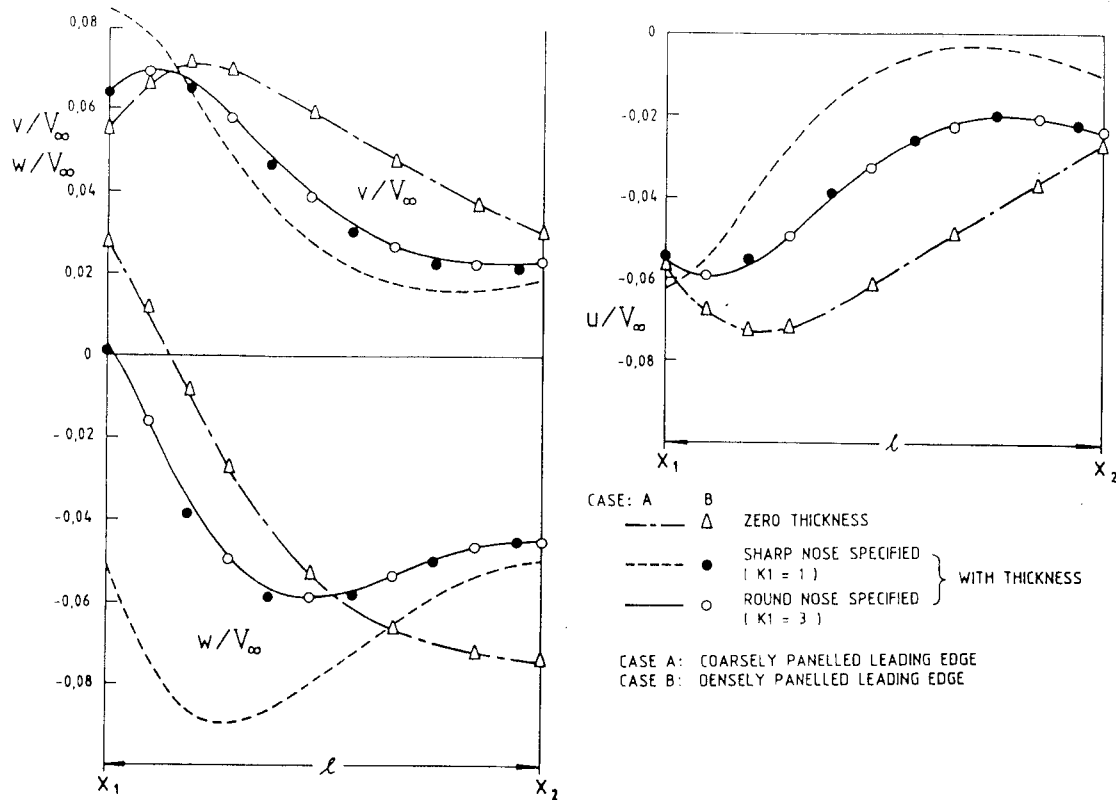
Fig. 3 Perturbation flow field along line ℓ (planar boundary condition approximation); $M_\infty = 0.4$, $\alpha = 6$ deg.

thickness case, the results for Cases A and B match closely. Thus, no dense wing leading-edge paneling is required as far as the lift-induced flow field is concerned. The effect of the wing leading-edge paneling on the thickness-induced flow field is determined by comparing Cases A and B for the data obtained with thickness taken into account. As Fig. 3 shows that the influence of the wing thickness on the flow field is severe, particular attention is required. Figure 3 demonstrates the trivial result that the differences in the perturbation velocities obtained with $K1 = 3$ (round nose) and $K1 = 1$ (sharp nose) are small if the leading-edge region is paneled more densely (Case B). However, for a coarse leading-edge paneling (Case A), these differences become very significant, the correct leading edge definition ($K1 = 3$) yielding correct results, whereas the $K1 = 1$ results are totally unacceptable. Thus, as far as the thickness-induced flow field is concerned, no dense wing leading-edge paneling is required, provided that the roundness of the leading edge is taken into account properly.

Conclusions

As to the modeling of carrier aircraft wings in external store load calculations using the panel method, it was demonstrated for a particular wing with round leading edge that:

1) The planar boundary condition approximation may be applied to the aircraft wing.

2) Wing thickness may not be ignored.

3) Using the planar b.c. approximation, a coarse paneling of the aircraft wing leading-edge region is allowed, provided the leading-edge roundness is incorporated properly in the source strengths representing the aircraft wing thickness.

Although these conclusions have been demonstrated using particular codes (USSAERO, USTORE), they should have a much more general validity.

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A Nonlinear Analysis of the Cushion Stability of Slowly Oscillating ACV's

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Introduction

THE main dynamic effects which would appear in the fan-duct-plenum system of ACV's might be enumerated as: 1) unsteady operating characteristics of the fan, 2) unsteady flow in the ducting, 3) wave propagation phenomena in the

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