To solve this we introduce a new coordinate system $o-\xi\eta\zeta$ which is local to the control surface in the wake We set the ξ axis parallel to the trailing vortex there and η axis on the xy plane To find the direction of the local trailing vortex an LDV can be utilized By tracing the velocity jump in the wake the trajectory of the trailing vortex is determined Since the coordinate transformation is by translational and rotational displacements only the absolute value of the velocity is unchanged Then together with requirement (11) Eq (14) reduces to

$$v_\xi^{*2}=v_\xi^2-2\int_{y_w}(v_z\omega_{Bx}-v_x\omega_{Bz}+\nu\nabla^2 v_y)\,dy \qquad 15)$$

For most of the working conditions of a hydrofoil or airfoil it can be assumed that viscous wake vorticity is almost normal to the xy plane Also the curvature of v_y in any direction is considered to be small With these assumptions the equation further reduces to

$$v_\xi^{*2}=v_\xi^2+2\int_{y_w}v_x\omega_{Bz}\,dy$$

The z component of the vorticity for viscous wake ω_{Bz} can be approximated by $\partial v_y/\partial x-\partial v_x/\partial y$ as the z component of the trailing vorticity is considered to be small Thus

$$v_\xi^{*2}=v_\xi^2+2\int_{y_w}v_x\left(\frac{\partial v_y}{\partial x}-\frac{\partial v_x}{\partial y}\right)dy=v_\xi^2$$

$$+2\int_{y_w}v_x\frac{\partial v_y}{\partial x}\,dy-v_x^2 \qquad (16)$$

v_x^* is now given through the inverse transformation

Results

The values of profile drag computed from the measured velocity distribution by an LDV[10] at various streamwise locations are presented in Fig 1 for two types of foil sections (two dimensional foil) The results for Foil A and Foil B at $Re=9\times10^5$ have a scatter Since rather old data was used for these two cases in which the use of the formula for the profile drag was not particularly intended there may be some coarseness in the data Foil B at $Re=2\times10^6$ shows fairly constant value On the same figure the range of values for each case taken from Ref 11 is shown for comparison For Foil B no available data could be found so a similar foil section was chosen instead For Foil A and Foil B at $Re=2\times10^6$ the computed values compare well with the other source Foil B at $Re=9\times10^5$ shows values a little higher However since the design lift coefficient for this foil is higher than the one shown for comparison slightly higher values of drag may be expected Figure 2 shows spanwise variation of the profile drag for a finite span swept back foil It is noted that near the tip the value goes down and then climbs up drastically This is due to the highly three dimensional in teractions of the boundary layer and tip vortex [10] There profile drag" does not have much meaning The trend shown in the figure is pointed out in Ref 12

Conclusions

A formula has been developed to give the profile drag based on the measured velocity data by laser Doppler velocimetry The formula was applied to some examples and reasonable agreements with the published data were obtained The derivation was based on low turbulence assumption However this assumption may not be valid in some cases It should therefore be extended in order to incorporate the turbulence components In such a case cross terms like $\rho v_x v_y$ will appear in the formula This will suggest a need for simultaneous multicomponent measurements which were not made in the present work

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Engineering Analysis of Drooped Leading-Edge Wings Near Stall

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Introduction

SINCE the late 1970 s low speed wind tunnel experiments[1] and flight tests[2 5] have conclusively demonstrated that wings with a discontinuous leading edge extension and in crease in camber (leading edge droop) exhibit a smoothing of

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the normally abrupt drop in lift coefficient C_L at stall and generation of a relatively large value of C_L at very high poststall angles of attack As a result, an airplane with a properly designed drooped leading edge has increased resistance towards stall/spins—behavior of great interest to the general aviation community In response to this interest, an extensive experimental investigation of the fundamental aerodynamic characteristics of drooped leading edge wings is being conducted, a sampling of which is reflected in Refs 6 9

Some preliminary theoretical support for these ex perimental results is given in Ref 10 which is an application of numerical lifting line theory to drooped leading edge wings below and above the stall However, lifting line theory has several deficiencies when applied to this problem not the least of which is summarized by the following statement quoted from Ref 10: "It is wise not to stretch the applicability of lifting line theory too far For the high angle of attack cases presented here, the flow is highly three dimensional and only an appropriate three dimensional flowfield calculation can hope to predict the detailed aerodynamic properties of such flows " The purpose of the present Note is to describe an extension of the work of Ref 10 namely to present the results of an "appropriate three dimensional flowfield calculation" for drooped leading edge wings

The only other published theoretical work directly bearing on this problem is that due to Katz,[11] who utilized an un steady vortex lattice program in a time asymptotic mode The present Note complements Katz's work in that a simpler engineering approach is taken, without any apparent loss of accuracy

Method

This Note presents numerical results obtained with a three dimensional vortex panel computer program for the calculation of inviscid, incompressible (potential) flow over infinitely thin finite wings with camber This program is

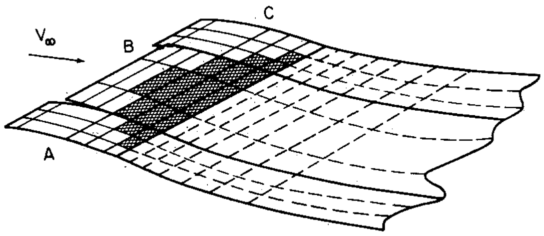

Fig 1 Panel distribution to simulate the effects of flow separation Sketch also shows the three separate sections used to model the effect of the leading edge discontinuities

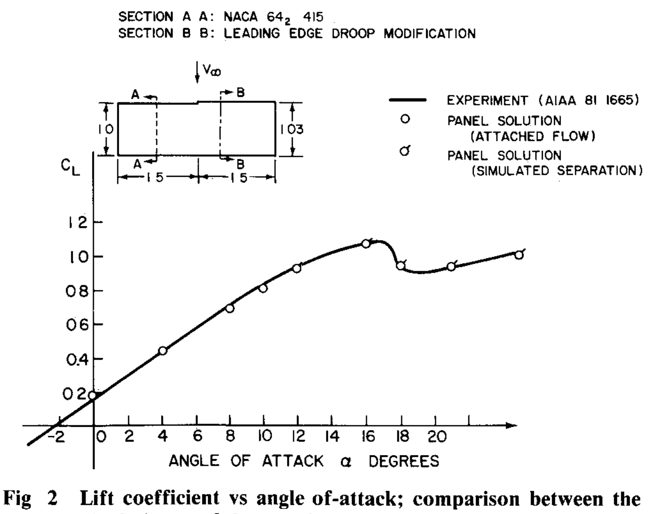

Fig 2 Lift coefficient vs angle of-attack; comparison between the present calculations and the experimental data of Ref 7

specially constructed for application to wings with drooped leading edge discontinuities [12] The program is essentially a numerical representation of lifting surface theory involving both spanwise and chordwise distributions of vorticity Across each panel, the vorticity is assumed to vary linearly in both the spanwise and chordwise directions; hence this is a second order panel method

The present results also include two approximations of an "engineering" nature First, the effect of the leading edge discontinuities is modeled by assuming that the vortices eminating from these discontinuities aerodynamically divide the wing into three distinct sections of lower aspect ratio as sketched in Fig 1 Some direct experimental evidence of this effect is discussed in Ref 1 Hence, the present calculations were made with three separate sections with different camber butted together (Sections A, B, and C in Fig 1) This allows an appropriate discontinuity of vorticity in the chordwise piecewise continuous vorticity distribution In this fashion the vortex panel analysis is made to "see ' the leading edge discontinuities without explicitly inserting separate vortex filaments originating at the discontinuities Although the actual flowfield will have an external vortex generated at each drooped leading edge discontinuity, the theoretical model simply has a stronger vortex filament at the discontinuity at the start of the droop Secondly, the effect of the separated flow at high angle of attack is modeled by applying rec tangular vortex panels with a *varying* vortex strength over only those portions of the wing with attached flow The separated region of the wing is covered with constant strength vortex panels associated with a value of the pressure coef ficient $C_p = -0\ 6$ This is a reasonable value of C_p in separated regions on wings, in low-speed flow as shown by numerous experiments [1 6 9 13] These constant strength panels in the separated region are represented by the shaded region in Fig 1 Obviously this modeling requires a knowledge of the separation lines on the finite wing For the present results these separation lines are obtained from surface oil flow visualization experiments such as described in Ref 14 (It is interesting to note that the experiments described in Refs 6 9 and 14 clearly demonstrate that the separated region over the top surface of a wing just beyond the stall forms a "mushroom shaped" pattern)

Comparison with Experiment

This modeling of the drooped leading edge discontinuities and of the separated flow is a simple engineering approach and is not meant to be the final theoretical answer to the analysis of such flows However this modeling taken in conjunction with the second order vortex panel program described above yields amazingly good results as shown in Fig 2 Here C_L vs angle of attack is given for the drooped leading edge wing sketched in the figure The solid line represents a curve through the experimental data of Ref 7; the open circles give numerical results obtained with the present analysis The agreement is excellent at all angles of attack both below and above the stall The purpose of this Note is to demonstrate that such results can be obtained with the simplified engineering approximations discussed above Moreover by taking into account the three dimensional flow effects the present results represent a substantial im provement over the C_L vs α results obtained from lifting line theory in Ref 10

Concluding Remarks

The present calculations are not totally predictive They re quire a knowledge of the experimentally observed separation lines However they reflect a reasonable capability to calcu late and understand the complicated three dimensional sepa rated flow over drooped leading edge wings at and beyond the stall

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Errata

"A Computer System for Aircraft Flyover Acoustic Data Acquisition and Analysis"

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