Table 1 Physical properties of the wing–aileron system

b, ft (m)	m, slugs (kg)	I_β, slug-ft^2 (kg-m^2)	S_β, slug-ft (kg-m)	k_h, lb/ft (kN-m^{-1})	k_β, lb-ft-rad^{-1} (N-m-rad^{-1})
3.125 (0.95)	0.65 (9.5)	0.1 (0.136)	0.045 (0.2)	1642 (24)	308.4 (418)

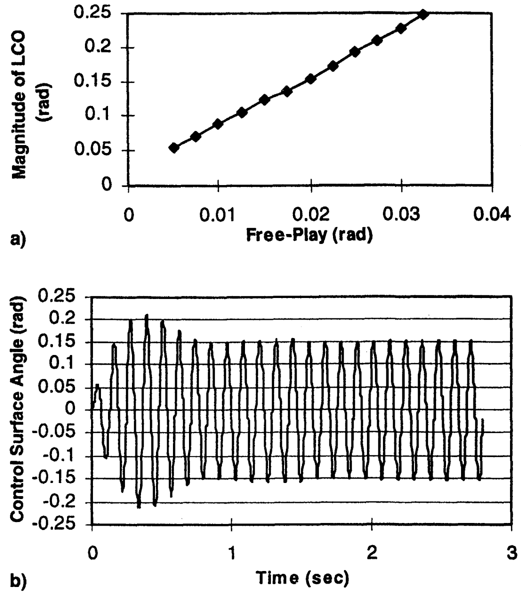

Fig. 2 Effect of freeplay on the aeroelastic response of the control surface: a) effect of freeplay band, β_0, on the LCO amplitude; and b) typical time–history of the nonlinear system.

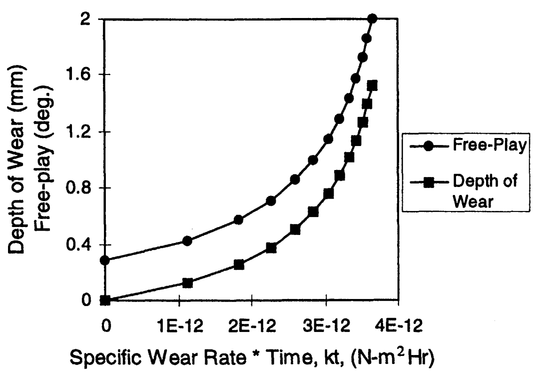

Fig. 3 Development of wear and freeplay in the two-degree-of-freedom nonlinear system.

amount of freeplay corresponding to the new aeroelastic response of the system is calculated. Figure 3 shows the development of backlash in the joint.

Conclusions

In this paper, the wear of a control linkage was considered. The existence of LCOs in a wing–aileron system with freeplay nonlinearity was shown. The wear in the wing–aileron system with freeplay has also been shown to be a coupled problem; i.e., increased wear results in increased amplitude of the LCO and applied load, which in turn, gives rise to increasing wear rate.

Similar feedback phenomena exist in the development of cracks, where crack growth rates increase with crack length. The rate at which the aging process accelerates is the critical issue for maintenance, and accurate prediction depends on the sophistication of the aeroelastic model and the wear model. Here, a two-degree-of-freedom model of an unbalanced aileron, and a wear model that ignores the effect of impact that will occur as the system traverses the freeplay gap, have been implemented.

Acknowledgment

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Particle Image Velocimetry Study of Wing-Tip Vortices

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Introduction

OCCURRING in a wide variety of flows such as aircraft wakes and helicopter blade-vortex interactions, wing-tip vortices are of great practical significance and have been the subject of extensive research for many decades. Although it is well known that the behavior of wing-tip vortices, particularly in the far field, is strongly influenced by complex atmospheric conditions, the wind tunnel does provide a controlled environment for the detailed study of the vortices in the near field at a relatively low cost (when compared to, say, field studies). Valuable to the understanding of the physics of vortex formation, the data thus obtained can also be used for the validation of near-field simulations, and as the near-field input for far-field predictions.

The wing-tip vortex of an unswept wing at low speeds has been studied[1] in the Institute for Aerospace Research (IAR) 9 m × 9 m Wind Tunnel using the particle image velocimetry (PIV) technique.[2] In particular, measurements of the crossflow

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components of wing-tip vortices were obtained with different PIV setups and under various flow conditions. A number of features of the wing-tip vortex can be clearly identified from the measurements. The PIV technique has thus been demonstrated as a viable velocity measurement tool in a large-scale facility, although further work will be required to refine its operation and extend its range of applicability.

Experimental Setup

A 14-in. chord, 5-ft long NACA 0015 wing with adjustable angle of attack (AOA) was mounted vertically on the floor of the test section. No boundary-layer transition strips were applied, and the wing tip of the airfoil was blunt. The Dantec PIV system was used in the present test. The double-cavity Nd:YAG laser, PIV processor, and personal computer were all located behind the wall of the test section. A specially designed piece of optics was used to transform laser beams into diverging light sheets with an included angle of 5 deg. The light sheet and, therefore, the measurement plane, was approximately 10 chord lengths downstream of the airfoil. The PIV camera, protected by a Plexiglas® casing and mounted on a tripod inside the test section, was placed approximately 6 chord lengths downstream of the measurement plane. Seeding (vaporized Bayoil) was released in the settling chamber through a small nozzle approximately 150 ft upstream of the airfoil. To align the seeding with the tip of the wing, the smoke nozzle had to be placed approximately 17 ft above the floor of the 82-ft-diam settling chamber.

Accuracy of any PIV measurements depends on a large number of factors (including, for example, the local velocity gradient of the flow), and is in practice difficult to quantify without other experimental or numerical means of verifying the PIV data. Further work is therefore necessary in this direction, although as a first estimate, an accuracy of 5–10% can be reasonably expected.

Results and Discussion

As an example, Fig. 1 shows a PIV image obtained 10 chord lengths downstream of the airfoil, at a wind speed of 15 mph (6.7 m/s) and a Reynolds number of 1.3×10^5. The airfoil was set at 14-deg AOA. At such a high AOA, the airfoil is stalled and flow separation over the airfoil may well be a source of unsteadiness to the wake vortex downstream. The measurement area of this image is 320 × 240 mm, the largest measurement area with acceptable velocity measurements. The core appears to be laminar, and is surrounded by a shear layer rolling up around it. The small structures along the shear layer are believed to be secondary vortices, which were not captured in the corresponding velocity vector map (Fig. 2) because of the lack of resolution. The several erroneous vectors on the left of the vector map are a result of insufficient seeding in the region. A plot of the streamwise vorticity (not shown) shows a core with a rather constant level of vorticity, as would be expected from a forced vortex core. The outer flowfield has a minimal level of vorticity and is essentially irrotational.

Because of centrifugal effects, seeding particles tend to drift away from the center of the core. In this region, therefore, seeding particles are usually sparse (cf. Fig. 1) and no meaningful velocity measurements can be obtained. This poses a drawback in such cases where detailed crossflow measurements are required, e.g., in the study of vortex stability. However, in the study of tip vortex as a direct aircraft hazard, the center of the core represents only a small part of the flowfield responsible for the (undesired) rolling moment on the following aircraft.

A plot of tangential velocity (U_θ) against the radial distance from the core (not shown) shows that the core is not fully axisymmetric. By averaging the U_θ data along six axes 30 deg apart, we can obtain an averaged U_θ distribution plot. Figure 3 shows 12 such sets of averaged data. The vortex is quite strong, with the maximum U_θ reaching 0.4 of the freestream velocity, which is not too surprising because the airfoil was set at a rather high AOA of 14 deg. Of more interest is the degree of scattering of the data, not only in terms of the $(U_\theta)_{max}$, but also in terms of the core radius r_c, i.e., the position of $(U_\theta)_{max}$. Note that the U_θ distributions with the larger $(U_\theta)_{max}$ usually have the smaller r_c, as may be expected in the vortex stretching process, where angular momentum is conserved.

Referring to Table 1, the core sizes measured here have been compared with those obtained in other similar (but not identical) studies,[3–7] and were found to be within the same order

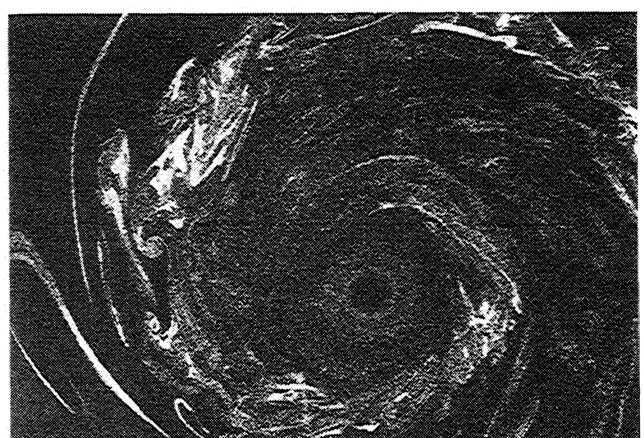

Fig. 1 PIV image taken 10 chord lengths downstream of the airfoil. Fifteen mph (6.7 m/s), $Re_c = 1.3 \times 10^5$, and 14-deg AOA. Image area is 320 mm × 240 mm.

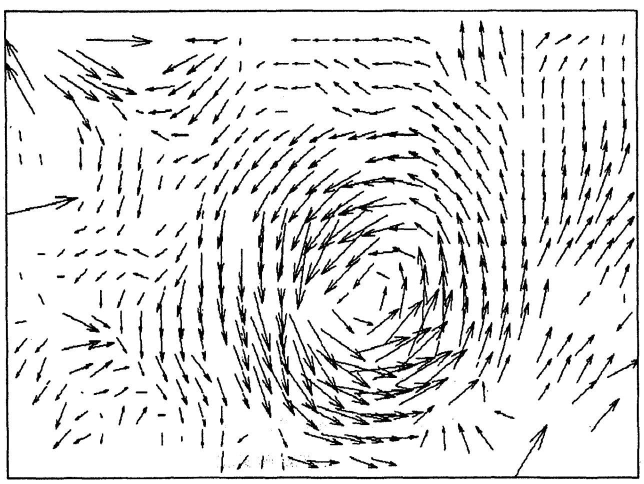

Fig. 2 Corresponding velocity vector map of Fig. 1.

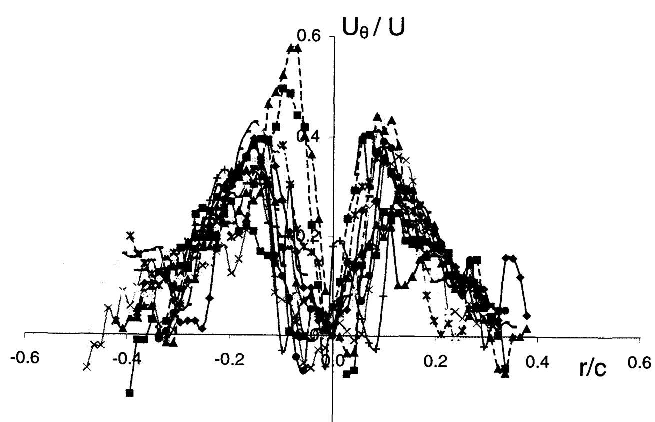

Fig. 3 U_θ distributions of 12 averaged data sets.

Table 1 Comparison of core radius (r_c), with r_c defined in various ways

Flow conditions and model configuration	Measurement plane	r_c/c	$r_c/(b/2)$	Definition[a]
NACA 0015, $Re_c = 1.3 \times 10^5$, $\alpha = 14$ deg (present test)	$x/c = 10$ ($x/b \approx 1$)	0.09–0.13	0.02–0.03	[1]
Moore and Saffman[4] equation	$x/c = 10$	0.03	——	[3]
ONERA Peaky airfoil, 35-deg sweptback, $Re_c \approx 5 \times 10^5$ (Ref. 5)				
$\alpha = 11$ deg	$x/b = 0.35$	——	0.02	[2]
$\alpha = 12.8$ deg	$x/b = 0.35$	——	0.04	[2]
NACA 0015, $Re_c \approx 3 \times 10^5$ (Ref. 6)				
$\alpha = 8$ deg	$x/b = 10$	——	0.02	[1]?
$\alpha = 12$ deg	$x/b = 10$	——	0.02	[1]?
NACA 66-209, $Re_c = 6.83 \times 10^5$, $\alpha = 7.1$ deg (Ref. 7)	$x/c = 10$	——	0.03	[1]
NACA 0012, $Re_c = 5.3 \times 10^5$, $\alpha = 5$ deg (Ref. 8)	$x/c = 10$	——	0.05	[2]

[a][1] Positions of the two tangential velocity maxima; [2] edge of viscous wake, e.g., 1% total pressure loss contour; and [3] size of Rankine vortex.

of magnitude, though many of these earlier studies used point-measurement techniques without taking into account the effects of wandering, and made limited mention of the fluctuations (if any) in core sizes. In the present test, the extent of core movement, or wandering, has been found to be about 0.1c, which is comparable with the radius of the core itself. Knowing the time interval between data points, the velocity of wandering can also be estimated and found to be about 0.025–0.030 m/s, or roughly 4% of the freestream velocity.[1]

The cause of the unsteadiness is not certain, although it has often been attributed to the unsteadiness in the wind-tunnel freestream.[8] It may also be a result of flow separation over the airfoil, and may even be a genuine feature of any wing-tip vortex. In any case, the wandering motion is slow and could, in principle, be separated from other (turbulence) measurements using methods such as spectral decomposition[8] whenever a point-measurement technique is used. According to one particular study,[3] the measurement error in the peak tangential velocity can be as much as 35% if wandering effects are not included. As a global measurement technique, PIV therefore presents a definite advantage over point-measurement techniques such as hot wires and pressure probes.

Conclusions

Velocity measurements of wing-tip vortices have been obtained successfully in the IAR 9 m × 9 m Wind Tunnel using the PIV technique. At 14-deg AOA, the wing-tip vortices have been found to be highly unsteady, and the PIV technique has been shown to be an effective tool in obtaining instantaneous measurements of such unsteady flows. Further work is underway to refine its operation and extend its applicability under a wider range of flow conditions.

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Gurney Flap Experiments on Airfoil and Wings

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Introduction

IN Ref. 1, Myose et al. present recent wind-tunnel results on the effects of Gurney flaps on the aerodynamic characteristics of several two- and three-dimensional lifting surfaces. The Gurney flap is a small vertical tab mounted at the trailing edge of a lifting surface normal to its pressure side. The idea behind this trailing-edge modification is to shift the location of the Kutta condition to enhance the lift generated by the lifting surface at given angle of attack. In Ref. 1, Myose et al. indicate that few results are available in the open literature on the effects of Gurney flaps for three-dimensional configurations. Their study includes lift and drag results for a wing with a natural laminar flow section shape and Gurney flaps of varying span. The purpose of this Note is to bring attention to a wind-tunnel study on the effects of Gurney-flap-like modifications for a three-dimensional wing with a similar section shape conducted ~10 years ago,[2] to compare the lift and aerodynamic efficiency (L/D) results obtained in this earlier test with those obtained in Ref. 1, and to point out possible aerodynamic advantages of three-dimensional modifications in the geometry of the Gurney flap.[3]

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