Table 2 Data for two maneuvers

Dummy variable y	Maneuver X_1	X_2
0	0.58579	2
0	2	2
0	3.41421	2
1	5	4.58579
1	5	6
1	5	7.41421

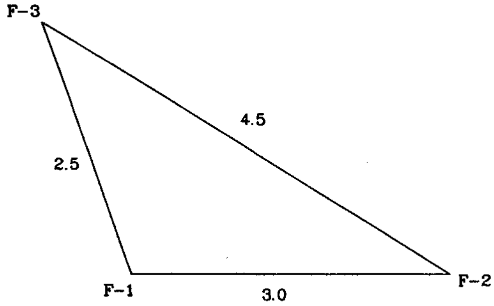

Fig. 1 Agility distance between three aircraft.

correlation coefficient R at each step, which can be used in Eq. (2). Importantly, this program also calculates a statistical test for the significance of the set of remaining variables not yet entered into the regression. It should be noted that the vector of regression coefficients calculated at each step is equal to the vector $\boldsymbol{d}'S^{-1}$ which appears in Eq. (1) (see Ref. 2), so that Eq. (1) could be used as well to compute the Mahalanobis distance at each step.

Furthermore, let D_p^2 and D_q^2 denote the squared Mahalanobis distances between two aircraft based on p maneuvers and any subset of q maneuvers, respectively. It can be shown[4] that, in general, $D_p^2 \geq D_q^2$. Thus, if we have calculated a Mahalanobis distance between two aircraft for each maneuver, we would select from these distances the largest one as the best indicator of distance between these aircraft based on a single maneuver. (It would be a mistake to average these distances.) Thus, the different distances between aircraft based on one maneuver tell us which single maneuver best, or least, reveals an agility difference. Again, however, the least revealing maneuver should not be discarded, not only for the reasons noted above, but also because that maneuver may be more revealing for other aircraft in future tests.

For the case $p = 2$, Kramer[2] gives an example completely worked out for real data from an industrial experiment. Pedantically, in order to facilitate computation, we have constructed hypothetical data as shown in Table 2.

Using Eq. (1), we easily get $D^2(X_1 \ \& \ X_2) = (3 \quad 4)\ (I)\ (3 \quad 4)' = 25$, where I is the identity matrix, and $D^2(X_1 \ \& \ X_2)$ is the squared Mahalanobis distance based on both X_1 and X_2. Similarly, $D^2(X_1) = 9$ and $D^2(X_2) = 16$, where both of these last results also can be obtained easily using Eq. (2).

Suppose there are three aircraft, the F-1, F-2, and F-3, with a calculated distance of 3.0 between the F-1 and the F-2, 2.5 between the F-1 and the F-3, and 4.5 between the F-2 and the F-3. These results can be shown on a triangle as in Fig. 1.

If there are four aircraft and a calculated distance between each pair, the results can be similarly shown on a tetrahedron, which makes an interesting and revealing visual display.

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Effect of a Fuselage on Delta Wing Vortex Breakdown

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Nomenclature

b	=	wing span
c_r	=	root chord of exposed wing panel
D	=	fuselage diameter
K_b	=	upwash factor for vortex breakdown
K_W	=	slender-body-theory Beskin upwash factor
Re_{c_r}	=	Reynolds number, based on root chord
x	=	axial distance measured aft of apex of exposed wing panel
x_b	=	axial distance to mean breakdown position measured aft of apex of exposed wing panel
α_∞	=	angle of attack of freestream flow relative to wing root chord
$\alpha_{b,w}$	=	angle of attack of wing-alone configuration for breakdown at x_b/c_r
$\alpha_{b,wb}$	=	angle of attack of wing-body configuration for breakdown at x_b/c_r
α_{eq}	=	equivalent angle of attack
α_0	=	angle-of-attack offset
Λ_w	=	leading-edge sweep angle

Introduction

THE importance of leading-edge vortex flows to slender aircraft has been well-documented in recent years in many reports.[1-3] As a result, the aerodynamic characteristics of leading-edge vortex flows have been extensively studied both experimentally and computationally. Much of this research has been in the area of vortex breakdown, the sudden and dramatic change that results in the turbulent dissipation of the vortex. Various investigators[4-8] have reported the many factors influencing the vortex breakdown on delta wings. Some of those factors include aspect ratio, camber, cropping, flaps, surface roughness, and trailing-edge geometry. Most of the previous work has reported on wing-alone (without fuselage)

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planforms. The authors have been unable to locate any systematic studies regarding the effect of adding a fuselage to a wing planform on the leading-edge vortex breakdown characteristics. In this note, the effect of adding a generic fuselage to a delta planform is experimentally investigated with regard to the onset and forward progression of leading-edge breakdown with increasing angle of attack.

Description of the Experiment

The experiment was conducted in the NASA Langley Research Center (LaRC) water tunnel.[9] The wing planforms used in this study were installed in the vertically arranged 16- × 24-in. test section so that the wing plane was parallel to the 16-in. side at an angle of attack of 0 deg. The test section was equipped with a pitch mechanism that allows for an angle-of-attack range of −30 to +30 deg. For the present investigation, a 30-deg offset model support sting was used to allow for higher angles of attack ranging from $\alpha \approx 0$–45 deg. For the present investigation, the test section velocity was held constant at 0.25 ft/s to minimize freestream flow turbulence and allow for good dye flow visualization. The Reynolds number Re_{c_r} was approximately 2×10^4. No blockage corrections were made.

The models used in this investigation are sketched in Fig. 1. Each planform was made out of 0.05-in.-thick, flat, aluminum sheet to minimize both camber and thickness effects. The wing-body configuration was a delta planform with a leading-edge sweep of 69.3 deg, mounted on a 14-in.-long axisymmetric body with a 3.5 tangent-ogive forebody and a 1.5-in.-diam cylindrical afterbody. The wing-body model was bolted to a $\frac{3}{8}$-in.-diam support sting at the centerline of the axisymmetric body and then mounted in the tunnel on the angle-of-attack mechanism. The second configuration was a wing-alone delta planform derived by simply removing the fuselage of the original wing-body. The wing-alone configuration was mounted directly to the support sting.

A pressurized dye injection system was used for flow visualization. To entrain dye into the core in order to visualize the leading-edge vortex and its breakdown, dye ports were placed on the windward side of the model planforms at the apex of each planform leading edge. Location measurements of vortex breakdown were facilitated by a grid laid out on the wing planforms. Videotape and still film photography were used during the test program to document the leading-edge vortex trajectory and breakdown locations. Early in the investigation, it was determined from dye issuing from ports near the apex of the forebody that the influence of the weak fuselage-shed vorticity on the wing leading-edge vortex burst characteristics was minimal. Previous work by Hall[10,11] confirms this result.

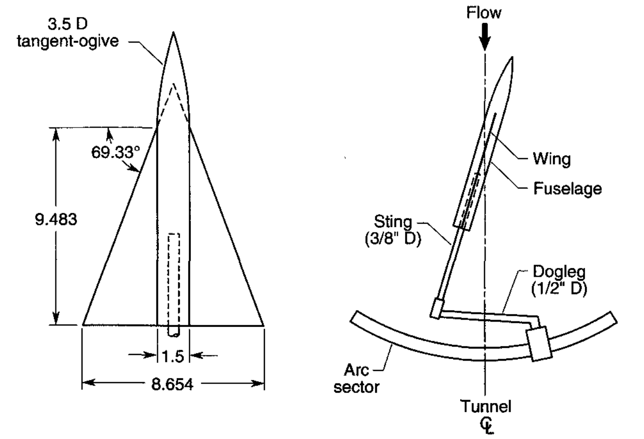

Fig. 1 Model configuration dimensions (in.) and mounting system.

Results and Discussion

Figure 2 shows a comparison of the leading-edge vortex breakdown locations for the two configurations at an angle of attack of 27.5 deg. Time mean values of the chordwise progression of vortex burst with increasing angle of attack are illustrated in Fig. 3 for both planforms. A value of $x_b/c_r = 1.0$ corresponds to the trailing edge. Also shown are several sets of related data from previous investigations by Erickson,[5] Wentz and Kohlman,[7] Roos and Kegelman,[12] Payne,[13] and Brennenstuhl and Hummel.[14] Each of these investigations, with the exception of Brennenstuhl and Hummel, tested wing-alone 70-deg sweep delta planforms at Reynolds numbers ranging from 3×10^4 to 1×10^6. Brennenstuhl and Hummel tested a wing-alone 69.3-deg swept delta, similar to the wing-alone planform of the present investigation, at a Reynolds number of 1.3×10^6.

Removing the fuselage from the wing-body delta planform configuration dramatically delayed the upstream progression

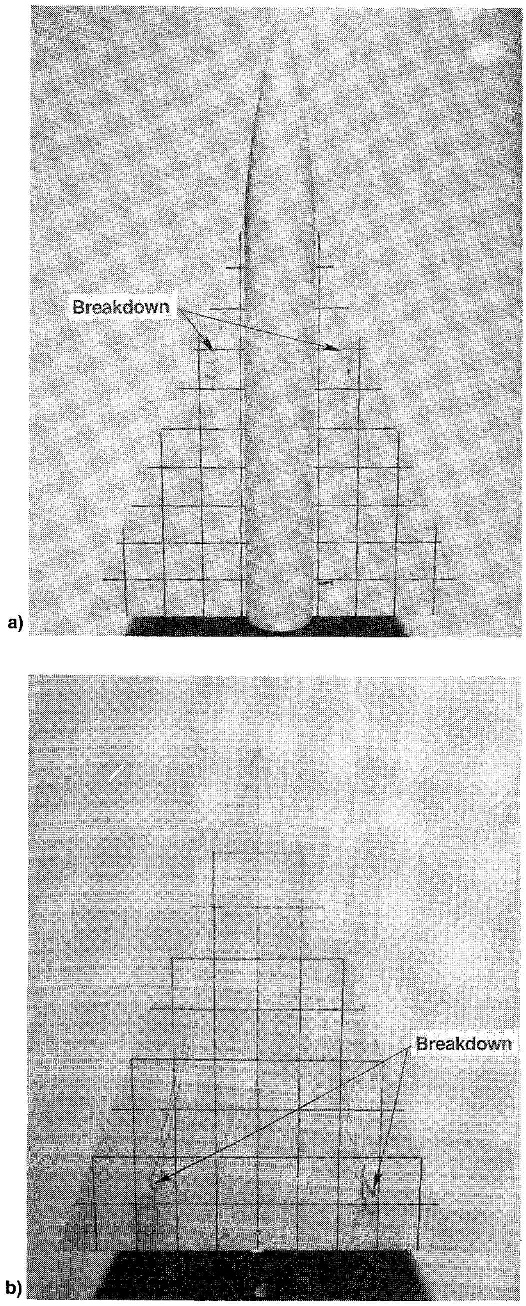

Fig. 2 Comparison of leading-edge vortex breakdown locations for α = 27.5 deg: a) wing-body and b) wing-alone.

	Re_{cr}	Λ_w	Wing	Reference
—●	1.8 x 10^4	69.3°	Wing-body	Present work
—■	2.2 x 10^4	69.3°	Wing-alone	Present work
○	3.0 x 10^4	70.0°		Erickson[5]
□	4.0 x 10^5	70.0°		Roos & Kegelman[12]
◇	4.3 x 10^5	70.0°		Payne[13]
△	1.0 x 10^6	70.0°		Wentz & Kohlman[7]
▽	1.3 x 10^6	69.3°		Brennenstuhl & Hummel[14]

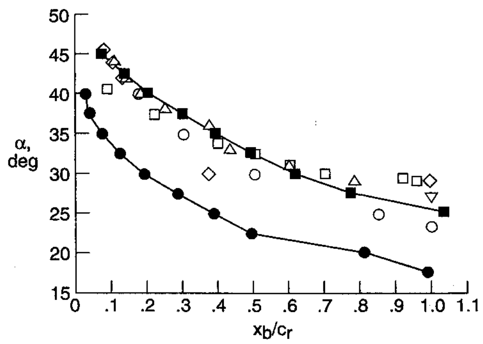

Fig. 3 Effect of fuselage on vortex breakdown characteristics of a delta wing.

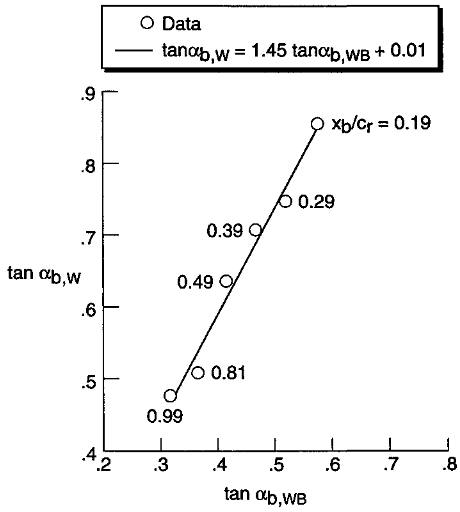

Fig. 4 Comparison of angles of attack at which vortex breakdown occurs at a specific axial location for wing-body and wing-alone configurations.

of vortex burst. The effect of adding the body to the delta planform can be seen to adversely affect both the onset and normalized chordwise progression of vortex breakdown on the wing. Trailing-edge breakdown occurred at an approximately 8 deg higher angle of attack without the fuselage than when it was present. The vortex breakdown results for the wing-alone planform correlate well with previous investigation data sets for the similar 70-deg-sweep wing-alone planform.

The effect of the cylindrical fuselage on the flow prior to vortex burst over the wing panels as a function of angle of attack can be correlated using the equivalent angle-of-attack concept of Hemsch and Nielsen.[15] They suggested that classical slender-body theory (SBT) for wing-body interference[16] could be extended to high angles of attack, including vortex lift, by applying an effective upwash factor to the angle of attack rather than the normal force, i.e.

$$\tan \alpha_{eq} = K_W \tan \alpha_\infty \tag{1}$$

where K_W accounts for the effective upwash imposed on the exposed wing panels by the fuselage flowfield. The quantity α_{eq} is the angle of attack at which an isolated wing (composed of the two exposed wing panels) would generate the normal force acting on the fuselage-mounted panels. For a fuselage with constant diameter cross section in the vicinity of the wing panels, the classical SBT upwash factor K_W is solely a function of the ratio of the wing span to the fuselage diameter.[16] For the wing-body configuration tested in this investigation, the SBT value for K_W is 1.17.

The equivalent angle-of-attack concept is not likely to be applicable in the presence of vortex breakdown. However, the form of Eq. (1) appears to be useful for correlating the fuselage effects. In Fig. 4, the data of Fig. 3 for the wing-alone and wing-body planforms are replotted with tan $\alpha_{b,w}$ and tan $\alpha_{b,wb}$ corresponding to their individual burst angles of attack, respectively. The data points correspond to identical normalized axial positions of vortex burst for the two configurations. The data are accurately fit within the experimental accuracy by

$$\tan \alpha_{b,w} = K_b \tan \alpha_{b,wb} + \tan \alpha_0 \tag{2}$$

where K_b is 1.45 and α_0 is approximately 0.6 deg. Unfortunately, wing panel forces were not measured, so it is not known if the experimental value of K_b determined in this way applies to wing panel loads as well. Equation (2) may prove to be a useful means for extending wing-alone results when additional data for different planforms and D/b ratios become available. However, it should be pointed out that the physics underlying Eq. (2) are not understood.

Final Remarks

The present investigation illustrates the dramatic effect of fuselage/forebody upwash on leading-edge vortex breakdown. Although it is premature to generalize on the presented data, the results presented in this Note may provide insight in the quest to achieve higher angle-of-attack flight by delaying the onset of leading-edge vortex breakdown, one possibility being the use of a blended fuselage-wing aircraft designed to reduce the effect of body upwash on the wing. However, more data on wing-body and wing-alone configurations will be needed in order to derive the expressions needed to extend SBT to vortex breakdown.

Acknowledgments

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