Wing Sweep and Blunting Effects on Delta Planforms at $M = 20$

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Nomenclature

C_D	=	drag coefficient, D/qS
$C_{p_{max}}$	=	maximum pressure coefficient
C_L	=	lift coefficient, L/qS
C_{L_α}	=	lift curve slope at $\alpha = 0°$, per deg
d	=	wing leading edge diameter, in. (Fig. 1a)
d/l	=	wing thickness ratio
D	=	drag, lb
l	=	wing length, in. (Fig. 1a)
L	=	lift, lb
L/D	=	lift-drag ratio, C_L/C_D
M_∞	=	freestream Mach number
q	=	dynamic pressure, psi
r	=	leading edge radius, in.
S	=	wing planform area, in.^{2}
α	=	angle of attack, deg
Λ	=	sweep angle, deg (Fig. 1a)
γ	=	ratio of specific heats

THIS paper is a brief résumé of the preliminary results obtained from an investigation recently conducted at a Mach number of 20 in helium flow to determine the effects of leading edge blunting and sweep angle on the static longitudinal stability characteristics of basic delta planforms, with particular emphasis on maximum lift-drag ratio. For the purpose of brevity, only summary plots are presented.

The experimental results were obtained in the Langley 22-in. helium tunnel utilizing a contoured nozzle to obtain a uniform freestream Mach number of 20.3.[1] Stagnation temperature for the tests was approximately 80°F. Reynolds number, based on chord length, ranged from 1.5×10^6 to 5.5×10^6, but for wings having thickness ratios less than 0.034, a constant Reynolds number of 3.7×10^6 was maintained in order to minimize any possible Reynolds number effects. A wing having a thickness ratio of 0.01 and sweep angle of 80° was tested at Reynolds numbers of both 3.7×10^6 and 6.6×10^6, and this difference in Reynolds number yielded no appreciable difference in L/D_{max}. Angle of attack was limited to a 0° to 30° range, and wind-tunnel blockage, caused by the relatively large size of some of the models, did not permit all models to reach 30° angle of attack. The models used in the present investigation were hemispherically and hemicylindrically blunted delta planforms, and the ratio of leading edge diameter to body length varied from 0 to 0.3. The sweep angles considered were 45°, 60°, 70°, 80°, 85°, and 90°. A sketch of the basic delta planform models used is shown in Fig. 1a, and the accompanying table indicates the pertinent dimensions, including the presence of a balance housing body, which became necessary on the more slender configurations. Figure 1b contains a sketch of the models used to simulate zero leading edge thickness and the balance housing body associated with the models. In order to shed some light on the effect of the presence of the balance housing bodies, the wing with an 80° sweep angle and thickness ratio of 0.01 was tested using the balance housing body from the wings shown in Fig. 1b. The base of this body was located 0.3 in. from the base of the wing.

The maximum lift-drag ratios obtained in the investigation are shown in Fig. 2 as a function of thickness ratio; the flagged symbols denote wings that had balance housing bodies. As previously mentioned, the wing having an 80° sweep angle and thickness ratio of 0.01 was tested with the smaller balance housing body shown in Fig. 16 (the smaller body had about

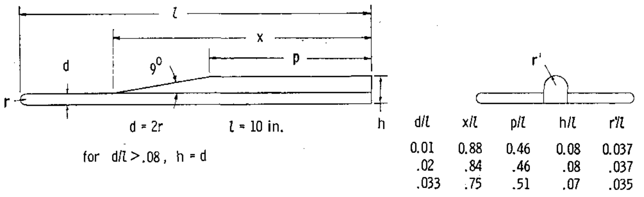

Fig. 1a Sketch of wings having balance housing bodies.

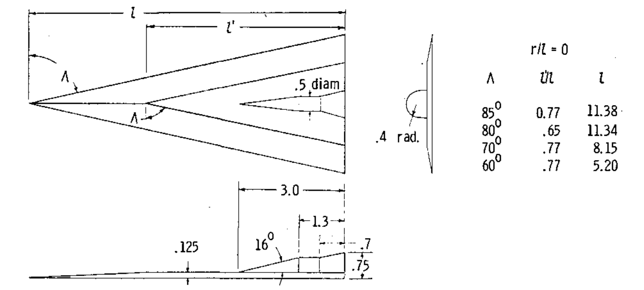

Fig. 1b Sketch of wings used to simulate zero leading-edge radius.

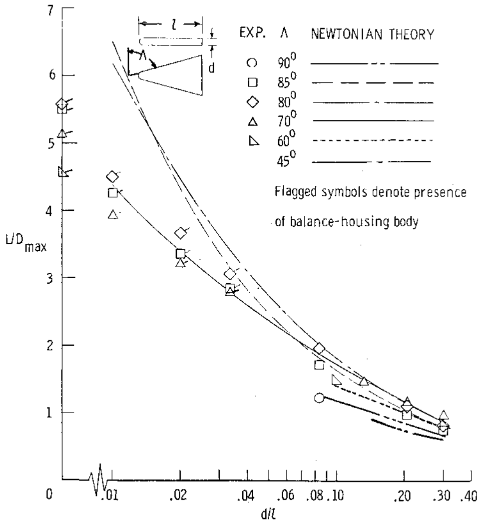

Fig. 2 Effect of thickness ratio on L/D_{max}.

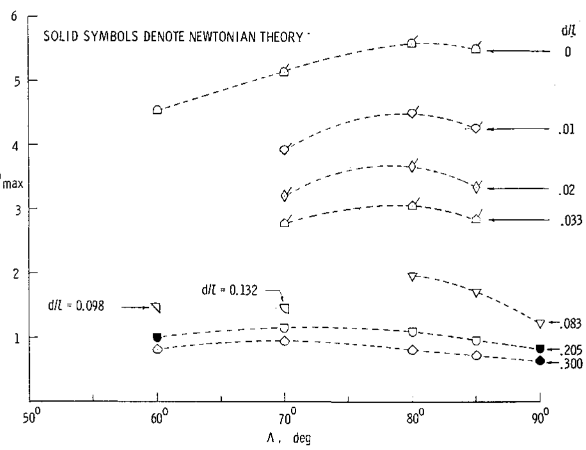

Fig. 3 Optimum sweep angle for L/D_{max}.

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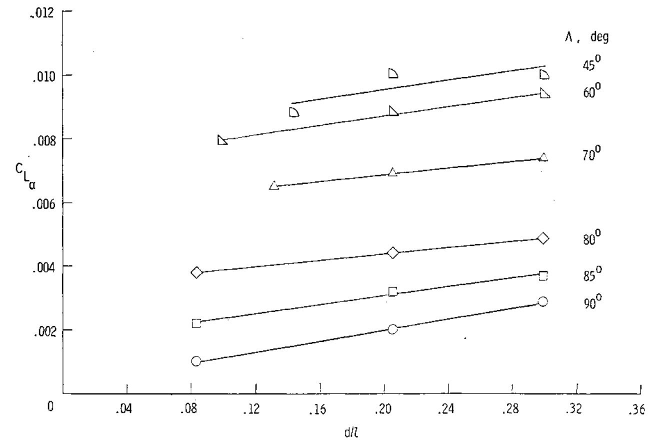

Fig. 4 Effect of wing sweep and thickness on C_{L_α}.

18% as much volume as the large balance housing body), and this resulted in only a 3% increase in $L/D_{\max}$, which is believed to be well within the accuracy of the data. Note that $L/D_{\max}$ exhibited a nearly linear decay as thickness ratio was increased logarithmically. For the more slender wings, the models with 80° sweep angles obtained the higher values of $L/D_{\max}$ for a particular thickness ratio. Optimum sweep angle is more readily obtained from Fig. 3, which indicates that as thickness ratio increased from a value of zero the optimum sweep angle for maximum lift-drag ratio decreased from an angle of 80° or slightly larger to 70° or less for the more blunt shapes tested. The peak in $L/D_{\max}$ for each constant bluntness curve suggests that, as sweep decreases from 90°, the increased lifting surface more than offsets the increase in drag initially, whereas at sweep angles less than optimum, the converse is true. Unfortunately, the angle-of-attack range did not permit many of the more blunt wings to obtain a maximum lift-drag ratio, and the optimum sweep angle for the more blunt bodies could not be definitely ascertained. Newtonian theory, which is shown as calculated, neglecting viscous effects for wings without bodies,[2] is seen in Fig. 2 to give good prediction of maximum lift-drag ratio for the thickest wings and is useful in determining the optimum sweep angle for these wings (solid symbols on Fig. 3‡).

$C_{L\alpha}$ was found to vary linearly or near linearly with thickness ratio as shown in Fig. 4. As sweep angle increased, $C_{L\alpha}$ was seen to decrease, a trend previously obtained on slender wings at lower hypersonic speeds.[3] Because the presence of balance housing bodies would tend to introduce negative lift at and near zero degrees angle of attack, the only values of $C_{L\alpha}$ presented are those for wings that had no balance housing bodies, which were the more blunt wings. Within the limitation of the data in Fig. 4, values of $C_{L\alpha}$ for arbitrary combinations of thickness ratio and sweep angle may easily be interpolated.

Of primary concern, of course, is the marked effect of even small degrees of leading edge blunting on $L/D_{\max}$. It is clear then that the achievement of high $L/D_{\max}$ during reentry (of as much as 4 or more, say) for practical configurations will require the application of much effort and ingenuity on the part of designers.

‡ As a point of interest, Newtonian theory in a modified form was used, applying a stagnation pressure coefficient given by

$$[(\gamma + 3)/(\gamma + 1)]\{1 - [2/(\gamma + 3)](1/M^2)\}$$

on the cylindrical leading edge section and the spherical segment nose sections and applying $(\gamma + 1)$ to the flat-plate portion of the wings, but it was found that classical Newtonian theory ($C_{p\max} = 2$) generally gave better predictions of all forces and moments.

References

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[2] Olstad, W. B., "Theoretical evaluation of hypersonic forces, moments and stability derivatives for combinations of flat plates, including effects of blunt leading edges, by Newtonian impact theory," NASA TN D-1015 (1962).

[3] Bertram, M. H. and McCauley, W. C., "Investigation of the aerodynamic characteristics at high supersonic Mach numbers of a family of delta wings having double-wedge sections with the maximum thickness at 0.18 chord," NACA TM L54G28 (1954).

Further Similarity Solutions of Axisymmetric Wakes and Jets

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Nomenclature

ρ = density
μ = viscosity
h = static enthalpy
H = $(h + u^2/2)$ total enthalpy
x, y = streamwise and radial coordinates with velocity components u and v

Subscripts

e = conditions at the edge of the viscous layer
x, y = partial differentiation
0 = conditions along the x axis

THE similarity equation [1–3]

$$[\eta(F'/\eta)']' + F(F'/\eta)' + \beta\eta[1 - (F'/\eta)^2] = 0 \qquad \text{(1a)}$$

subject to the boundary conditions

$$F = (F'/\eta)' = 0 \quad \text{at} \quad \eta = 0$$
$$\text{(1b)}$$
$$F'/\eta = 1.0 \quad \text{as} \quad \eta \to \infty$$

describes incompressible axisymmetric free-mixing with streamwise pressure gradients, and admits large velocity defects, where the axis velocity ratio u_0/u_e is constant.

Solutions of Eqs. (1) have been presented[2] for $\beta = 1.0$, and for the range[3] $-0.5 < \beta < 0$. For $\beta = 1.0$, it has been shown[2] that $F' = \eta(1 - 3\exp-\eta^2/4)$.

This note describes an extension to compressible flow which applies rigorously if the density ratio ρ_e/ρ can be only a function of the similarity variable η. However, it appears that ρ_e/ρ can be represented by a function of η alone only as an approximation when $(u_e^2/2h_e)[1 - (u/u_e)^2]$ is negligible with respect to unity, or when $u_e^2/2h_e$ is constant. Thus when $(u_e^2/2h_e)[1 - (u/u_e)^2] \ll 1$, the isoenergetic compressible equation reduces to the forementioned incompressible form. Exact and integral method solutions of Eqs. (1) in the ranges $-1.0 < \beta < -0.5$ and $0.5 < \beta < 4.0$ are presented herein.

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