Rarefied Hypersonic Flow Characteristics of Delta Wings and Trailing Edge Spoilers

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The present experimental research provides aerodynamic characteristics of delta wings located in a rarefied hypersonic air flow at various angles of attack. Trailing edge spoiler effects are recorded and induced boundary layer separations are investigated. The flow is characterized by a Mach number of 8.1 and a freestream Reynolds number of 2200/cm, simulating roughly a flight altitude of 40 miles. Results include drag and lift coefficients at incidences up to 50°, wall pressure data along the wing center-line, and external flow probing. Visualizations, obtained by means of glow discharge and oil film deposit, allow definition of incident shock such as separation and spoiler induced shock location; the three-dimensional separation spreading is limited upstream from the spoiler for various wing incidences. Data are compared, when possible, with previous experimental results and available theoretical approaches in weak and strong interaction régimes.

Nomenclature

C	= Chapman-Rubesin viscosity coefficient
k	= leading edge drag coefficient
L	= delta wing length
M	= freestream Mach number
p, p_∞	= wall pressure and freestream static pressure
$p_{t2}, p_{t2'}$	= local and freestream stagnation pressure
Re	= Reynolds number
t	= leading edge thickness
T_0, T_w	= stagnation and freestream temperature
x	= abscissa with the origin at the apex
y_s	= incident shock ordinate
α	= wing angle of attack; positive α refers to compression flow over the surface
γ	= specific heat ratio
ε	= $(\gamma-1)/(\gamma+1)$
K_ε	= $M^3\varepsilon kt/x$
$\bar{\chi}$	= $M^3(C/Re_x)^{1/2}$
χ_ε	= $\varepsilon[0.664+1.73T_w/T_0]\bar{\chi}$

Introduction

DELTA wings have been the subject of many investigations, nevertheless most of the results are related with the continuous flow régime. The interest connected with high altitude flight at hypersonic speeds, and in particular with the space shuttle program, has led us to extend our research to low values of Reynolds numbers. It is quite useful to know the main characteristics of the flow around delta wings at transition flow régimes where viscosity and boundary displacement effects become very important. According to the required flight mission, re-entry into the atmosphere could be a high-angle-of-attack ballistic trajectory and since the maximum lift coefficient occurs at an incidence near 50° for a delta wing, most of the trajectory could be flown in the moderate angle of attack range. Vehicle control implies mechanical devices such as trailing edge spoiler; it is necessary to predict their effectiveness first as a function of the angle trajectory and second according to the rarefied flow conditions where laminar boundary-layer separation is often predominant.

The present study brings out some results concerning flat delta wings inclined in an hypersonic rarefied air stream; furthermore, trailing edge spoiler effects are analyzed, the induced separation involving changes in the vehicle stability due to the pressure variations applied on the spoiler and on the adjacent wing surface.

Received September 23, 1971; revision received February 24, 1972.
Index categories: Entry Vehicle Dynamics and Control; Supersonic and Hypersonic Flow; Rarefied Flows.

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Review of Previous Analysis and Experiments

In the strong interaction régime, Cheng[1] analyzed the effects of boundary-layer displacement, leading edge bluntness and incidence on the hypersonic flow properties around slender two-dimensional bodies. Assumption of a strong shock and of a specific heat ratio close to unity allowed development of a theoretical model. More recently, Kemp[2] modified the theory slightly, taking into account the true values of γ. The general equation governing the flow under the combined influence of bluntness, viscous interaction and angle of attack, is reduced to

$$(Z-\Gamma\xi)(ZZ')'-(ZZ')^{1/2}=1$$

where Z is given as a function of ξ.

As indicated by Kemp

$$Z=(A^4/B^2)8M\chi_\varepsilon{}^4 y_s/K_\varepsilon{}^3 x$$

$$\xi=16[(A^{7/6}/B^{1/2})\chi_\varepsilon/K_\varepsilon{}^{2/3}]^6$$

and

$$\Gamma=(B/A^2)K_\varepsilon M\alpha/2\chi_\varepsilon{}^2$$

In Cheng's theory $A=1$ and $B=1$ while in Kemp's analysis $A=(\gamma+1)/2$ and $B=\gamma$. The pressure ratio along the body is given by

$$p/p_\infty=(A^5/B^2)4\gamma\chi_\varepsilon{}^4(ZZ')/K_\varepsilon{}^2$$

Theoretical values of pressure so calculated are compared hereafter with experimental data obtained along the wing center-line.

Delta Wing without Trailing Edge Spoiler

Many papers have been concerned with flat delta wings in hypersonic flows at high Reynolds numbers. References 3–7 give some information including the effects of leading edge, sweep angle and boundary layer on the flow characteristics over the wing. When incidences are higher than 15° and viscosity effects negligible, Barber's data[7] show in particular a reasonable agreement between wall pressures measured along the wing axis and pressures calculated by the tangent cone theory.

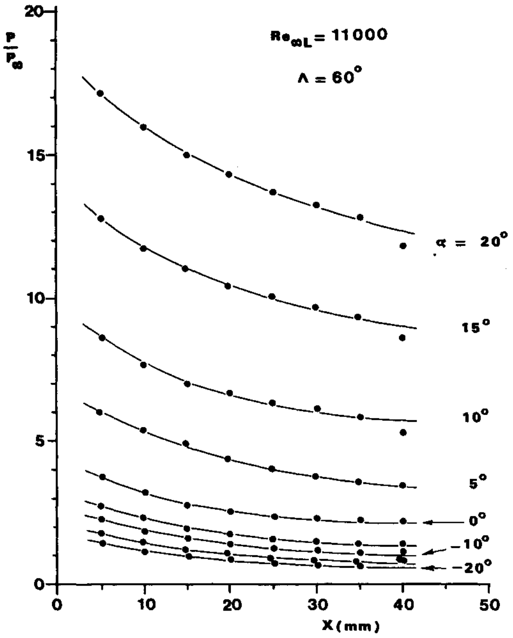

Fig. 1 Wall pressure distribution along the wing centerline (without spoiler).

At low Reynolds numbers and hypersonic flows, where viscous interaction becomes important, theoretical and experimental studies concerning delta wings are quite restricted. The present investigation extends some previous measurements with respect to delta wings located at low incidences.[8] Metcalf's paper[9] indicates, at zero angle of attack, a wall pressure distribution along the wing centerline which seems to follow a two-dimensional law. In an analytical and more recent study, Davies[10] suggests a theoretical model applied to the delta wing in combined régimes of weak and strong interaction; the analysis estimates the viscous effects on the normal pressure force acting on the wing surface; such theoretical force coefficients are to be compared with present data.

Delta Wing with Trailing Edge Spoiler

In order to control the vehicle movements during its flight, a flap must be adjoined to the main lifting surface; this flap, according to the induced interaction which results of its presence, modifies the aerodynamic characteristics of the whole body. The present experimental study is limited to the case of a trailing edge flap located normally to the wing surface. Fitted on a space vehicle, this spoiler can be controlled by means of a simple mechanism which could not have to support hinge moment variations as other adjustable deflected trailing edge flaps where it is necessary to provide adequate torque varying as a function of the separation and reattachment processes over the flap. References 11–16 give experimental and theoretical information in order to define a two-dimensional separation. Nevertheless, a trailing edge flap mounted on a delta wing generates three-dimensional separations strongly affected by any flow condition change. At hypersonic speeds and for high values of Reynolds numbers, some authors[17–19] analyzed three-dimensional effects associated with such separations. Rao's thesis[20] points out the phenomenon complexity of the three-dimensional separation which varies simultaneously as a function of the wing incidence, of the flap deflection and according to the flow viscosity. Some of Rao's results concerning force coefficients and wall pressure are compared with present experimental values where viscous interaction effects are especially important.

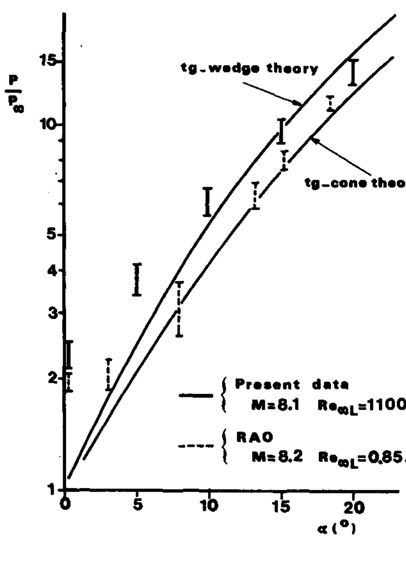

Fig. 2 Axial wall pressure level.

Test Facility and Models

The experiments were conducted in the "Laboratoire d'Aérothermique" hypersonic wind-tunnel SR2. This rarefied gas facility is a continuous and freejet type wind tunnel. The stagnation temperature and the stagnation pressure values are 700°K and 1 bar, respectively. The air flow is characterized by a Mach number of 8.1, a static pressure of 70 μm of mercury, a temperature of 50°K and a Reynolds number of 2200/cm. A contoured nozzle makes negligible the Mach number gradient through the useful core inside the test region.

Models include 60° and 70° swept delta wings 40 mm and 50 mm long. Trailing edge spoilers are located normal to the wing surface and are only mounted on a 60° swept delta wing 50 mm long and 0.5 mm thick. Wall and impact pressures are measured by means of a differential pressure transducer whereas aerodynamic forces are recorded using a three-component balance.

Flat Delta Wing Results

Wall Pressure Distribution

Along the wing axis, wall pressure distribution is shown on Fig. 1 at incidences between $-20°$ and $+20°$. The wing is 50 mm

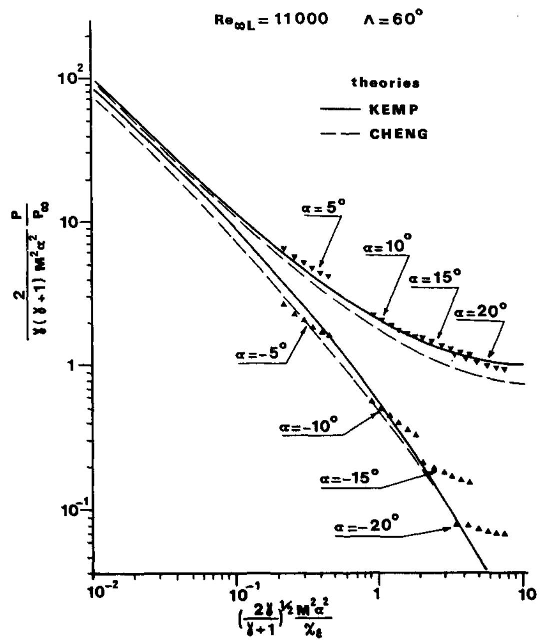

Fig. 3 Axial wall pressure correlation.

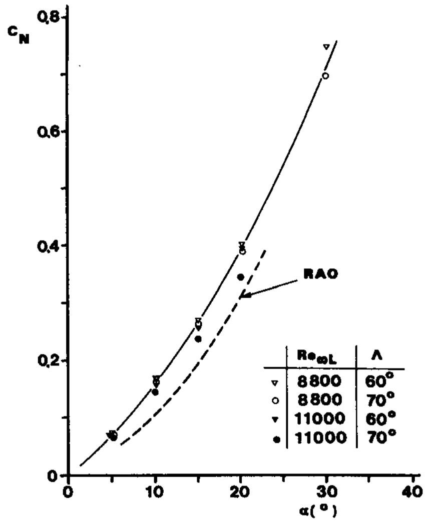

Fig. 4 Normal force coefficient results.

long and 1 mm thick; previous studies[7–8] indicated that, for delta wings, a leading edge thickness change influences only very slightly and locally the wall pressures measured along the axis, whereas the thickness effect is very sensitive in the case of a two-dimensional body like a flat plate.[8]

For abscissa higher than 25 mm downstream from the apex, pressure values are recorded on Fig. 2 vs the incidence. Results are compared with tangent cone and tangent wedge theoretical values and also with experimental data obtained at Mach number 8.2 at a higher density. In some previous work corresponding to large Reynolds numbers, in particular Barber's[7] and Rao's[20] studies, it appears that wall pressure along centerline is not very different that theoretical value deduced from the tangent cone theory when the wing incidence is more than 15°. On the other hand, in the present case where Reynolds number, based on freestream conditions and the wing length, is 11,000, the wall pressure is much higher and exceeds the tangent wedge theoretical value for angles of attack less than 15°. This high pressure level can be explained by the predominance of the viscous interaction effect.

Wall pressures measured on the wing axis between the apex and the trailing edge are compared on Fig. 3 with strong interaction calculation of Cheng[1] and Kemp.[2] The agreement between experimental data and theoretical values is quite satisfactory except for the highest negative incidences −15° and −20° where

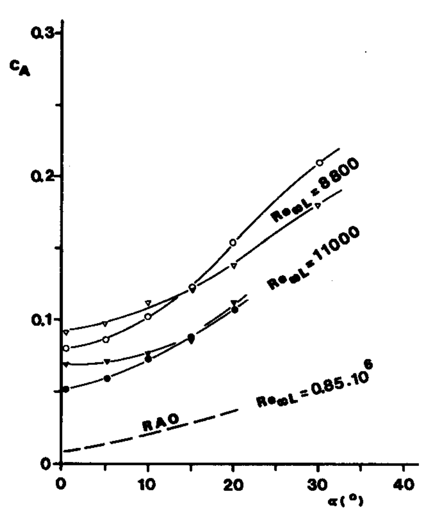

Fig. 5 Axial force coefficient results.

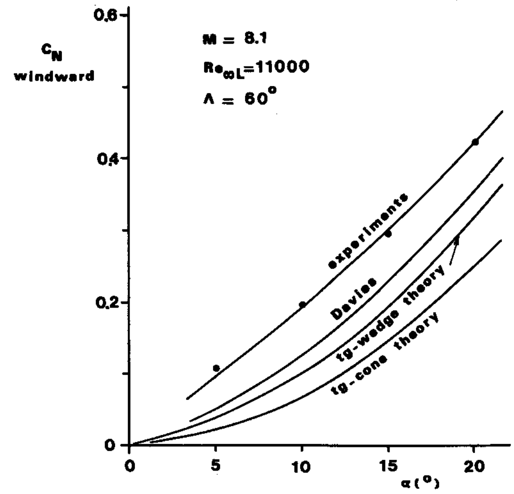

Fig. 6 Windward normal force coefficient.

scattering increases when the pressure taps are located farther from the apex.

Force Coefficients Values

Delta wings 40 mm and 50 mm long have being considered, corresponding to Reynolds numbers values of 8800 and 11,000. Force results indicated that the change in leading edge thickness affected only the total drag distribution whereas almost no effect could be systematically observed in the lift coefficient distribution. In order to compare the present data with previous data, normal and axial force coefficients have been chosen instead of drag and lift coefficients. It can be seen on Fig. 4 that the flow rarefaction effect does not affect too seriously the normal coefficient value. On the other hand, in the same incidence range up to 40°, the axial coefficient value is strongly dependent on the flow rarefaction level. A large departure can be noticed on Fig. 5 between the axial coefficient values corresponding to the low Reynolds numbers and those deduced from Rao's experiments undertaken at much higher flow density where skin-friction forces are negligible.

The normal coefficient component applying to the windward side is shown on Fig. 6; experimental values are calculated

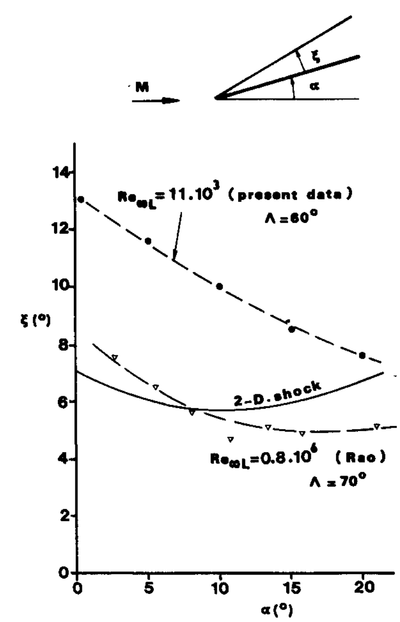

Fig. 7 Incident shock location.

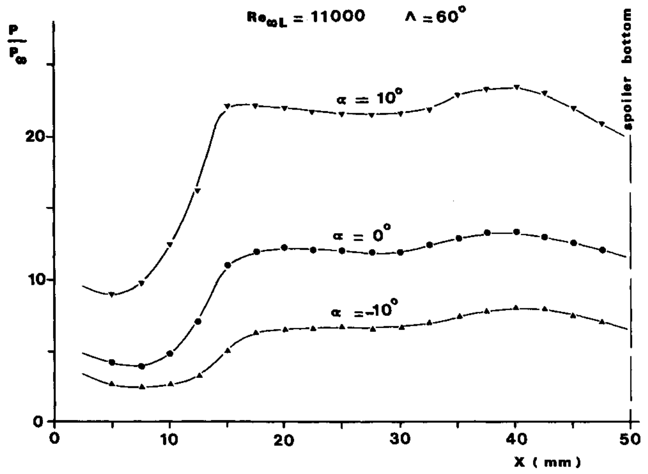

Fig. 8 Longitudinal pressure distribution along the wing centerline (with spoiler).

according to the recorded normal forces which are decomposed on each wing side, taking into account the mean wall pressure values. Tangent cone and tangent wedge curves are drawn on the same figure as also the viscous interaction law defined by Davies[10] for a cold wall model.

Incident Shock Location

The incident shock angle represented on Fig. 7 is determined from Pitot tube surveys; it is in agreement with glow discharge visualization. Angles measured by Rao are plotted on the same figure as well as two-dimensional shock theory values. The important effect of the boundary-layer displacement, which is found with low values of Reynolds numbers, involves a very sensitive slope increase of the incident shock. At zero angle of attack the shock angle almost doubles when passing from non viscous flow conditions to rarefaction conditions defined by a free-stream Reynolds number of 2200/cm. For increasing incidences corresponding to a higher correlative pressure level, the scattering between present experimental points and the two-dimensional shock theory is decreasing. Finally, when the curves are extrapolated at an incidence more than 20°, the evolution shows that true shock angle values should be less likely than those deduced from the shock theory valid for inviscid and two-dimensional flow.

Delta Wing with Trailing Edge Spoiler Results

Wall Pressure Distribution along the Wing and over the Spoiler

The flap induced separation characterizes an important increase of the wall pressure. Measurements have been limited to

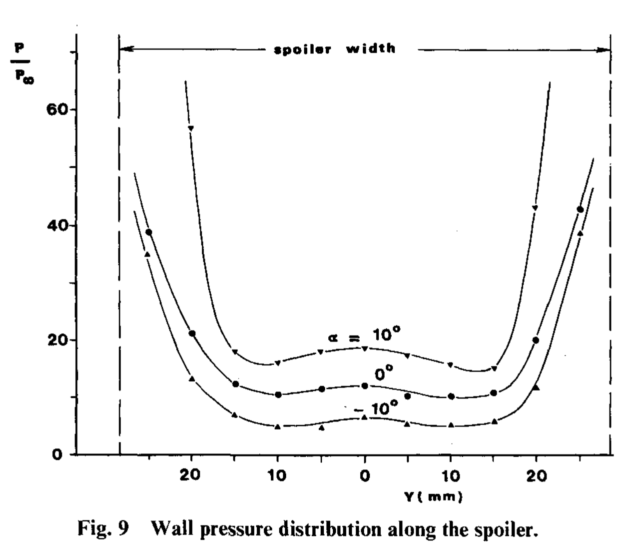

Fig. 9 Wall pressure distribution along the spoiler.

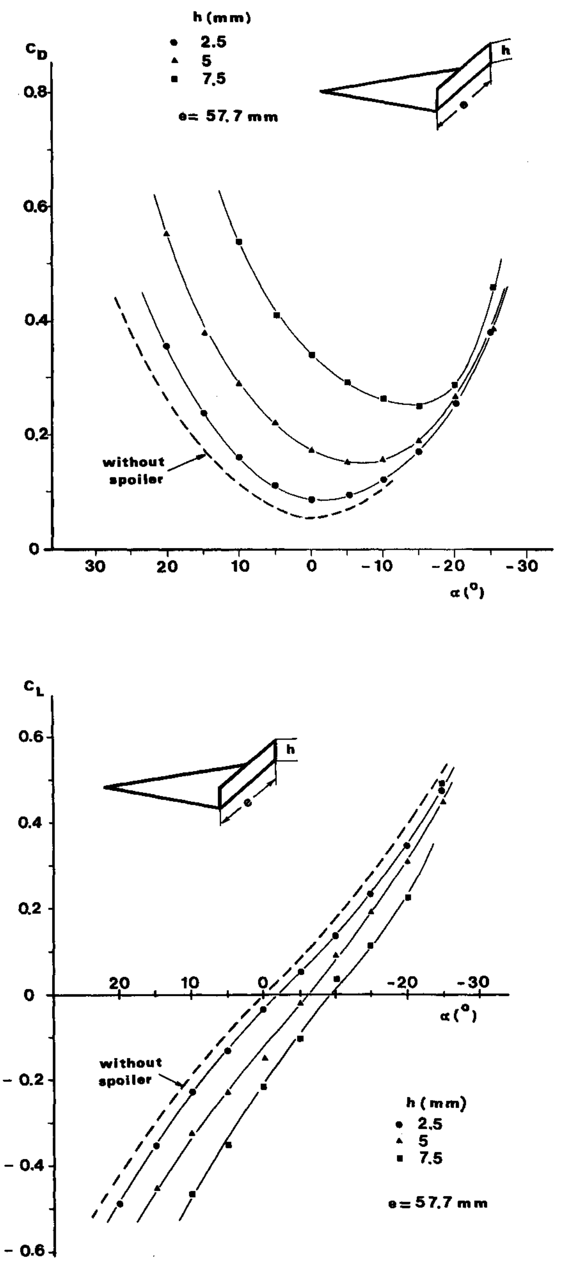

Fig. 10 Drag and lift coefficients for various spoiler heights.

the case of a 60° swept delta wing, 50 mm long, equipped with a spoiler 7.5 mm high covering the whole wing trailing edge span. At several incidences, the longitudinal pressure distribution is given on Fig. 8 from the apex until the flap bottom. The separation effect appears at approximately 10 mm downstream of the apex, then the pressure increases gradually to a maximum value through the separated region. For two-dimensional models, the pressure is generally uniform throughout the whole separation. In the present case, a slight pressure maximum is observed about 10 mm upstream from the flap bottom and was found for repeated measurements. This leaves out any error which could be due to a previous unsatisfactory outgassing of the pressure tubes or to an electrical drift of the transducer. The pressure maximum could be explained by a three-dimensional flow phenomenon.

For the same wing configuration, the transverse wall pressure distribution measured in the spanwise direction at half height of the spoiler is plotted on Fig. 9; in the centerline, pressure values are equal to those existing at the wing surface just upstream from the flap bottom. Some measurements carried out vertically, at the

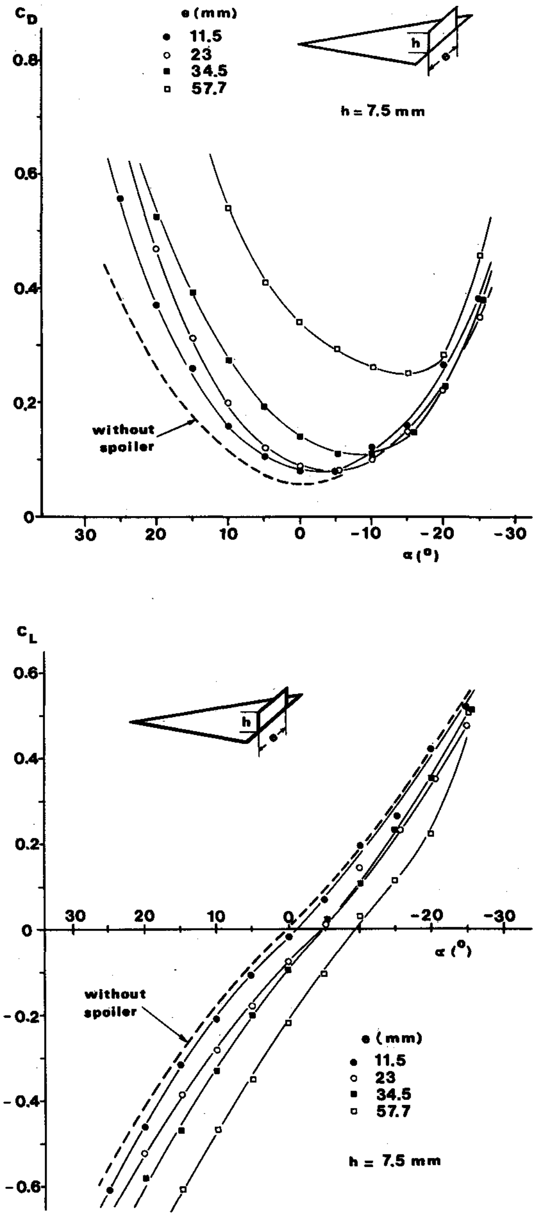

Fig. 11 Drag and lift coefficients for various spoiler widths.

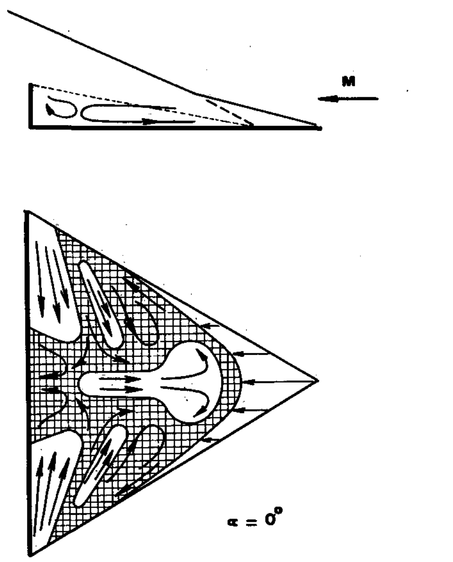

Fig. 13 Oil film visualization.

flap centerline, showed that pressure remains constant over the whole height; consequently, there is no reattachment phenomenon on the spoiler in this central region. The flap load rapidly increases in the spanwise direction to a maximum near the outboard tips. It is due very likely to the localized interaction between the spoiler outer edges, on one hand, and incident and reattachment shocks, on the other hand. At 10° incidence, the more important interaction results from a stronger three-dimensional incident shock induced from the wing leading edges; the shock collides with a larger part of the spoiler. The examination of the figure shows that, for some incidences, lateral parts of the flap have the more important effect, entailing strong modifications in total aerodynamic characteristics of the wing; this fact is to be pointed out in force measurements.

Influence of Spoiler Dimensions on Drag and Lift Coefficients

Force measurements have been performed on a 60° swept delta wing, 50 mm long and 1 mm thick; the spoiler height varies from 2.5 to 7.5 mm and its width varies from 11.5 mm to the whole wing trailing edge span. In this paragraph positive incidence refers to compression flow over the upper surface of the delta wing which supports the spoiler. For a spoiler covering the whole wing trailing edge, the distribution of drag and lift coefficients is presented vs the incidence on Fig. 10 for three flap heights. Separated regions are quite spread out and interactions between incident shocks and lateral portions of the spoiler lead to an important effect of the flap height; at incidences more than 20°, the spoiler is mainly located in the wing wake and is becoming ineffective. For a flap height of 7.5 mm, Fig. 11 shows the effect of its width change on the force coefficient values. In general, force results show the important contribution of the board portions of the spoiler in accordance with the wall pressure distribution shown previously on Fig. 9.

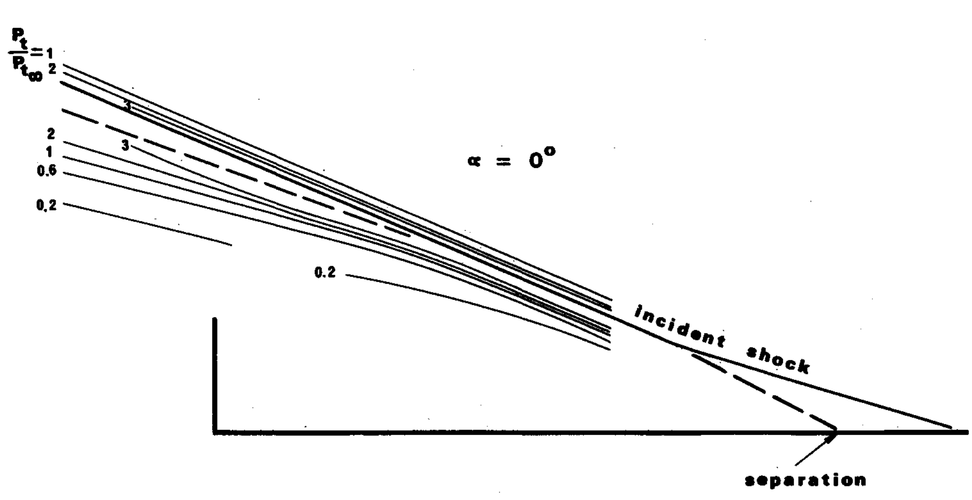

Fig. 12 External flow mapping.

External Flow Probing and Separated Region Visualization

Pitot tube surveys, glow discharge and oil film visualizations have been realized relative to the 60° swept delta wing, 50 mm long, provided with a spoiler 7.5 mm high covering the whole trailing edge length. Experiments have been performed for several angles of attack, nevertheless only the zero incidence case is presented here as an example. Impact pressure mapping on Fig. 12 corresponds to the symmetric plane containing the wing centerline. At some distance downstream from the apex, the separation induced shock merges with the incipient wing shock implying a visible direction change of the incident shock. In addition, glow discharge visualizations show two symmetrical shocks located just upstream of the spoiler around its two outboard tips. Oil film deposit allows definition of the separated regions at the wing surface; on Fig. 13, darkened surfaces indicate oil accumulations. The upstream border of the three-dimensional separation is clearly defined.

Conclusions

The present experimental study points out aerodynamic characteristics of delta wings located with incidence in a rarefied hypersonic flow. It includes measurements of wall pressure, aerodynamic forces, combined with external flow probing, glow discharge and oil film visualizations.

Flat Delta Wing

At relatively high Reynolds numbers, the tangent cone theory predicts quite reasonably the wall pressure level; on the other hand, at the present low flow density, interaction effect predominance implies a higher pressure ratio. The agreement between wall pressure data and strong interaction theories of Cheng and Kemp is quite satisfactory, except at the highest negative incidences where the scatter is the more important as the orifices on the wing axis are located farther from the apex. Force values indicate that the flow rarefaction effect seems to affect only slightly the intensity of the force component normal to the wing surface; however, it affects markedly the value of the force component directed according to the wing longitudinal axis; this confirms the importance of skin-friction forces in the present low density conditions. The large boundary-layer displacement thickness entails a more accentuated slope of the incident shock as compared with its theoretical position considering an inviscid flow; the departure is especially considerable at low angles of attack.

Delta Wing with Trailing Edge Spoiler

A very strong interaction can be noticed between the outboard tips of the spoiler and incident shocks issued from the leading edges; this involves a locally high induced wall pressure. Board effects can lead to sensible modifications in the wing aerodynamic characteristics as shown by force measurements. External flow surveys and visualizations allow prediction of precise shock structure as well as flow pattern just at the wing surface including the determination of separated regions.

References

[1] Cheng, H. K., Hall, J. G., Golian, T. C., and Hertzberg, A., "Boundary Layer Displacement and Leading Edge Bluntness Effects in High Temperature Hypersonic Flow," *Journal of the Aerospace Sciences*, Vol. 28, No. 5, May 1961, pp. 353–381.

[2] Kemp, J. H., "Hypersonic Viscous Interaction on Sharp and Blunt Inclined Plates," *AIAA Journal*, Vol. 7, No. 7, July 1969, pp. 1280–1289.

[3] Bertram, M. H. and Everhart, P. E., "An Experimental Study of the Pressure and Heat Transfer Distribution on a 70° Swept Slab Delta Wing in Hypersonic Flow," TR-153, 1963, NASA.

[4] Wetcofski, R. and Marcum, D., "Effect of Thickness and Sweep Angle on the Longitudinal Aerodynamic Characteristics of Slab Delta Planforms at a Mach Number of 20," TN-D 3459, June 1966, NASA.

[5] Cross, E. J. and Hankey, W. L., "Investigation of the Leeward Side of a Delta Wing at Hypersonic Speeds," *Journal of Spacecraft and Rockets*, Vol. 6, No. 2, Feb. 1969, pp. 185–190.

[6] Murray, W. and Stallings, R., "Heat Transfer and Pressure Distributions on 60° and 70° Swept Delta Wings having Turbulent Boundary Layers," TN-D 3644, Oct. 1966, NASA.

[7] Barber, E. A., "Some Experiments on Delta Wings in Hypersonic Flow," *AIAA Journal*, Vol. 4, No. 1, Jan. 1966, pp. 72–83.

[8] Allègre, J., Festinger, J-C., and Herpe G., "Etude expérimentale de corps portants en écoulement hypersonique à basse densité," *l'Aéronautique et l'Astronautique*, 1969-8, No. 15, Aug. 1969.

[9] Metcalf, S. C. and Berry, C. J., "Boundary Layer Interaction Effects in Low Density Supersonic Flow," *Rarefied Gas Dynamics, Fifth Symposium*, Vol. 11, Brundin, Academic Press, New York, 1966, pp. 1087–1106.

[10] Davies, L., "On the Effects of Viscous Interaction for a Flat Delta Wing at Incidence," Aero. Rept. 1315, 1970, National Physical Lab., Teddington, England.

[11] Chapman, D. R. et al., "Investigation of Separated Flows in Supersonic and Subsonic Streams with Emphasis on the Effect of Transition," Rept. 1356, 1958, NACA.

[12] Putnam, L. E., "Investigation of Effects of Ramp Span and Deflection Angle on Laminar Boundary Layer Separation at Mach 10.03," TN-D 2833, 1965, NASA.

[13] Miller, D. S. et al., "Mach 8 to 22 Studies of Flow Separation due to Deflected Control Surfaces," *AIAA Journal*, Vol. 2, No. 2, Feb. 1964, pp. 312–321.

[14] Needham, D. A. and Stollery, J. L., "Boundary Layer Separation in Hypersonic Flow," AIAA Paper 66-455, Los Angeles, Calif., 1966.

[15] Johnson, C. B., "Pressure and Flow Field Study at Mach Number 8 of Flow Separation on a Flat Plate with Deflected Trailing Edge Flap," D-4308, 1968, NASA.

[16] Cérésuela, R. and Coulomb, J., "Etude théorique et expérimentale de l'efficacité de gouvernes en hypersonique," *l'Aéronautique et l'Astronautique*, 1970-2, No. 18, Feb. 1970.

[17] Fetterman, D. E. and Neal, L., "An Analysis of the Delta Wing Hypersonic Stability and Control Behavior at Angles of Attack between 30° and 90°," TN-D 1602, 1963, NASA.

[18] Keyes, J. W. and Ashby, G. C., "Calculated and Experimental Hinge Moments on a Trailing Edge Flap of a 75° Swept Wing at Mach 6," TN-D 4268, 1967, NASA.

[19] Whitehead, A. H. and Keyes, J. W., "Flow Phenomenon and Separation over Highly Swept Delta Wings with Trailing Edge Flaps at Mach 6," *AIAA Journal*, Vol. 6, No. 12, Dec. 1968, pp. 2380–2387.

[20] Rao, D. M., "Hypersonic Control Effectiveness Studies on Delta Wings with Trailing Edge Flaps," thesis, 1970, Imperial College, London, England.