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Observations of the Supersonic Flow round a 6 per cent Double Wedge

By

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Observations of the Supersonic Flow round a 6% Thick Double Wedge.

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28th December, 1950

Summary and Conclusions

Surface pressure measurements and flow photographs have been taken for a 6% thick two-dimensional symmetrical double wedge at angles of incidence between 0° and 8°. The free-stream Mach number was 1.57, and the Reynolds number based on chord 0.8×10^6.

Although the results are in general in agreement with exact theory, discrepancies occur at mid-chord where the expansion is observed to be more gradual than would be expected in the absence of the boundary layer, and close to the trailing edge where separation occurs when the aerofoil is at incidence. Shock waves were observed at the nose and tail under conditions for which expansions alone would be expected. That at the nose is attributed to the fact that the leading edge was not sharp but 0.001" thick, and that at the tail may arise because the tail is blunt or because the flow separates from the other surface of the aerofoil.

1. Description of the Apparatus and Technique

The aerofoil is shown in Fig.1. It was constructed from mild steel and supported as shown in Fig.2 by tongues fitted to slots cut in the glass side walls of the tunnel. Eight 0.01" diameter pressure holes were provided at the positions shown, and were connected to a manometer by leads passing out of the tunnel through the supporting tongues. Incidence was adjusted by rotating the glass walls which were fitted in turntables. In order to obtain the greatest possible number of pressure readings, tests were made at equal positive and negative angles of incidence with the aerofoil both in its usual position, and reversed so that the trailing edge became the leading edge.

Toepler schlieren photographs were taken with an optical system based on two 9" diameter, 9' focal length spherical mirrors, and direct shadow photographs were obtained by using one of these mirrors to produce a parallel light beam. All photographs were taken with an exposure of about 1 microsecond. The tests were made in the 9" x 3" induced flow wind tunnel.

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2. Results

The measured pressure distributions are plotted in Fig.3 against position along the chord of the aerofoil. The static pressures p have been expressed non-dimensionally by dividing by the total head H_0 of the undisturbed stream. Schlieren and direct shadow photographs+ of the flow are reproduced in Figs. 5 - 9, and the lift coefficients obtained from the measured pressure distributions by integration are plotted against incidence in Fig.10.

3. Discussion and Comparison with Theory

Except in regions close to the mid-chord and to the tail, it is seen in Fig.3 that the measured pressure is, as predicted by theory, approximately constant along each of the four faces of the aerofoil. These constant values++ are compared in Fig.4 with values calculated by exact theory, and show reasonable agreement. The discrepancies may be due to errors in the profiling of the model, or to a slight error in the determination of the Mach number of the free stream. They may also arise from changes in the thickness of the boundary layer particularly in the strongly favourable pressure gradient at mid-chord.

Wave angles measured from the photographs are compared in Table I with those calculated by exact theory. Because of variations across the span of the aerofoil, and the interaction of the waves with the boundary layers on the side walls of the tunnel, the images of the waves are in many cases diffuse, and angle measurement is difficult. For this reason the tabulated values are considered to be accurate to within about $\pm 1^\circ$ only. Within these limits there seems to be fair agreement between the measured and calculated values.

TABLE I/

+The weak compression waves in the tunnel stream which strike the upper surface of the aerofoil at about 0.3 chord arose from an excrescence present on one side of the nozzle. This was removed before the pressure distribution on the aerofoil was measured.

++In cases where the pressure rises towards the rear of the upper surface, the value at 0.6 chord (i.e. ahead of the pressure rise) has been used in the comparison.

TABLE I. COMPARISON OF MEASURED AND CALCULATED SHOCK AND EXPANSION ANGLES

(a) Upper Surface

Incidence α	Nose Shock		Front of Nose Expansion		Rear of Nose Expansion		Front of Ridge Expansion		Rear of Ridge Expansion		Tail Shock	
	C	M	C	M	C	M	C	M	C	M	C	M
0	43.6	44.0	-	-	-	-	47.3	45.0	33.1	35.0	36.5	36.5
2	43.3	44.0	-	-	-	-	44.8	43.0	31.4	30.0	36.8	37.5
4	-	-	43.8	44.0	42.5	42.5	42.5	40.0	29.8	29.0	37.3	38.0
6	-	-	45.8	46.0	40.7	41.0	40.7	40.0	28.4	28.5	37.7	39.0
8	-	-	47.8	48.5	39.0	40.0	39.0	36.0	27.2	28.0	38.3	-

(b) Lower Surface

Incidence α	Nose Shock		Front of Ridge Expansion		Rear of Ridge Expansion		Tail Shock		Front of Tail Expansion		Rear of Tail Expansion	
	C	M	C	M	C	M	C	M	C	M	C	M
0	43.6	43.5	47.3	48.0	33.1	34.0	36.5	35.5	-	-	-	-
2	44.0	44.0	50.0	48.0	34.9	35.0	36.3	33.0	-	-	-	-
4	44.8	43.0	53.6	52.0	36.9	36.0	-	-	36.9	37.0	35.8	34.0
6	45.9	44.5	58.3	55.0	39.3	37.0	-	-	39.3	40.0	34.0	32.0
8	47.9	47.0	65.9	-	42.4	40.0	-	-	42.4	41.5	32.2	31.0

Notes

All angles are measured in degrees from the chord line. Calculated values are denoted by C and measured values by M. The line running along the span of the aerofoil at mid-chord is here termed the "ridge".

In/

In the photographs (Figs. 5 (a) & (b)) for $\alpha = 0°$ the flow appears to adhere to the surface back to the trailing edge, but for larger angles of incidence the boundary layer seems to thicken or separate on the upper surface upstream of the tail. This produces the rise of pressure shown at the rear of the upper surface for the $\alpha = 4°$, $6°$ and $8°$ pressure distributions plotted in Fig.3. At $\alpha = 2°$ there is evidence from Fig. 6 (a) that separation or thickening begins at about 0.95 chord, but very little pressure rise seems to occur at this point (see Fig. 3). At $\alpha = 4°$ (Fig. 7(a)) the region of separated flow appears to have moved forward to 0.9 chord, and the measured pressure at 0.95 chord is seen in Fig. 3 to have risen above that over the forward part of the second face of the upper surface. Further increase of incidence causes the separation point and the beginning of the pressure rise to move further upstream as shown in Figs. 8(a) & 9(a) and Fig. 3. The theoretical strengths (static pressure ratios across) the shock waves at the rear of the upper surface are 1.20, 1.32, 1.47, 1.65 and 1.84 at angles of incidence of $0°$, $2°$, $4°$, $6°$ and $8°$ respectively. The pressures measured at the tail of the aerofoil at $\alpha = 8°$ should be regarded with suspicion because it was not possible to move the breakdown shock in the tunnel far behind the model, and the flow at the tail may have been influenced by the interaction of this shock with the wake and boundary layers.

The pressures measured close to the middle of the chord show that the pressure fall in the expansion is not sudden, as would be expected in the absence of the boundary layer. The pressure begins to fall about 0.05 chord ahead of the ridge and, particularly for the stronger expansions (see, for example, the lower surfaces at $\alpha = 6°$ and $\alpha = 8°$), there is evidence that the pressure remains almost constant at a value a little above that over the rear face of the aerofoil for a small distance behind the ridge. The reason for this is not clear from the photographs, but it may be associated with a local separation of the flow.

The photographs taken at angles of incidence greater than that (3.47°) at which the shock wave would be expected to disappear from the nose at the upper surface show that a weak shock persists there, and is followed by the expansion required to bring the flow parallel to the surface. It is thought that this shock occurs because the leading edge is not sharp but 0.001" thick, and that the flow pattern at the nose is similar to that sketched in Fig. 11 (a). A similar phenomenon has been observed by Valensi and Pruden[1], who also found that under certain conditions a second shock occurred behind the expansion because of a local separation of the flow as sketched in Fig. 11 (b). This second shock does not appear to be present in the photographs taken during the present tests.

A shock wave occurs behind the expansion at the tail of the aerofoil when the incidence is such that an expansion only is expected. This is visible on the lower surface in Figs. 7 (a), 8 (a) & 9 (a) and may occur because the trailing edge is blunt, and the flow pattern there similar to that sketched in Fig. 11 (c). It might also arise[2], however, with a sharp trailing edge (see Fig. 11 (d)) if the flow has separated from the upper surface.

At/

At low incidence the measured lift coefficients are in good agreement (see Fig. 10) with those calculated by the exact theory, but at high incidence they fall below the calculated values because of the separation at the rear of the upper surface.

Acknowledgements

The aerofoil was made by Mr. W. Haywood of the Aerodynamics Division, N.P.L., and Miss N. A. Bunstead helped with the photography.

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No.	Author(s)	Title, etc.
1.	J. Valensi & F. W. Pruden	Some Observations on Sharp-nosed Profiles at Supersonic Speed (12th May, 1947). (A paper presented at the Sixth International Congress of Applied Mechanics, Sept. 1946). R. & M. 2482. May, 1947.
2	D. W. Holder & R. J. North	Observations of the Interaction between the Shock Waves and Boundary Layers at the Trailing Edges of Aerofoils in Supersonic Flow. Current Paper No.53. December, 1950.

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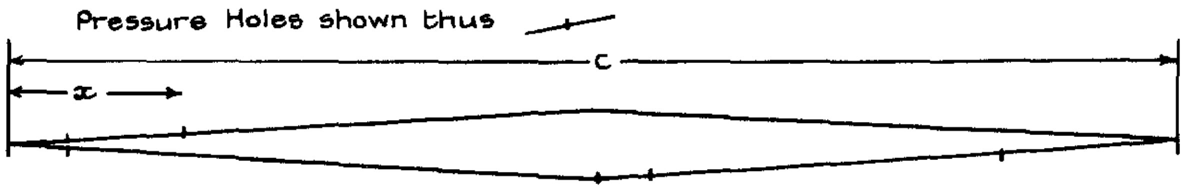
FIG. 1



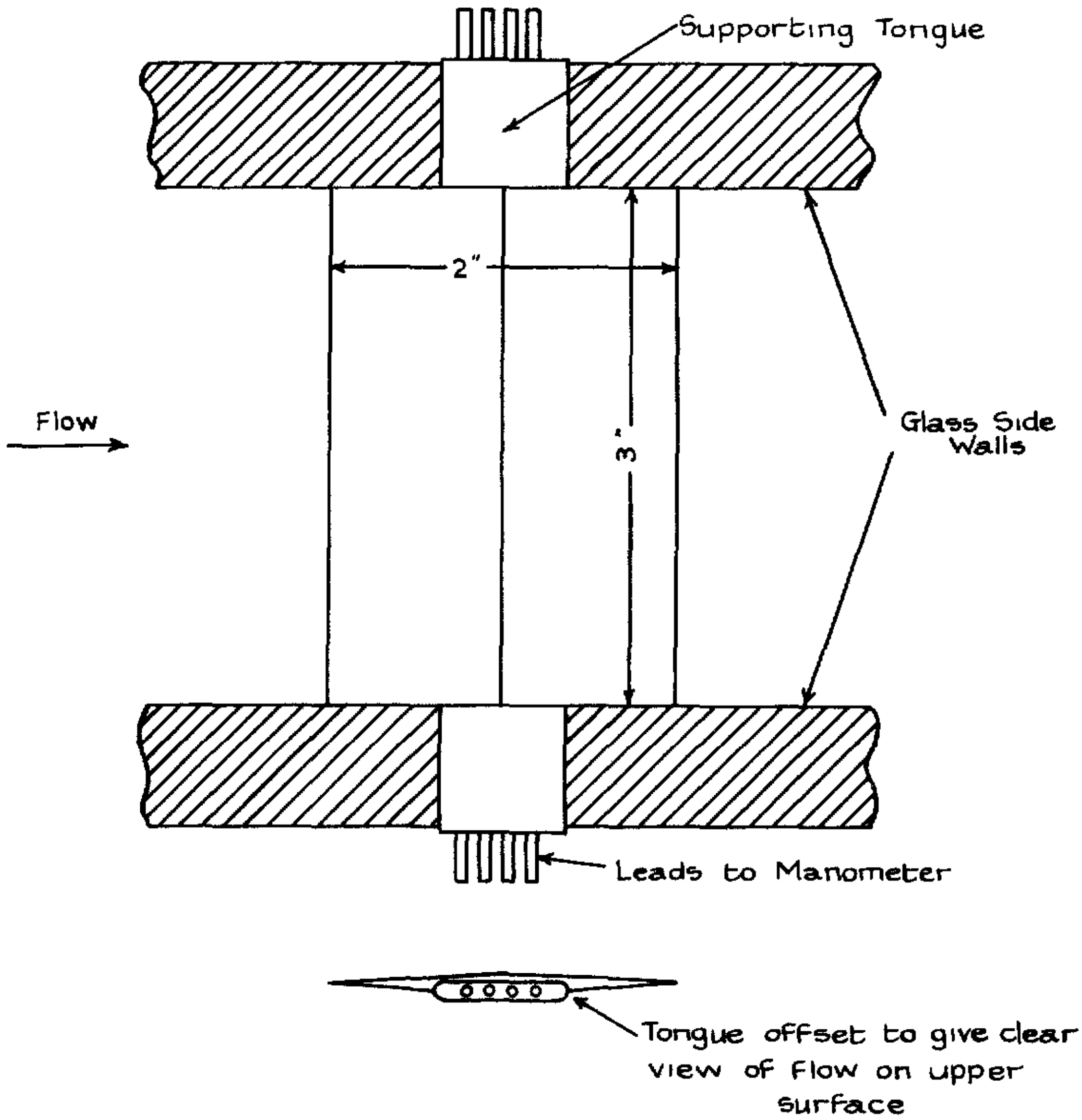
Chord 2.00"
Max thickness 0.121"

Thickness of leading and trailing edges 0.001"
Nose and tail angle = 6.93°

Positions of Pressure Holes x/c						
0.05	0.15	0.30	0.475	0.505	0.55	0.85

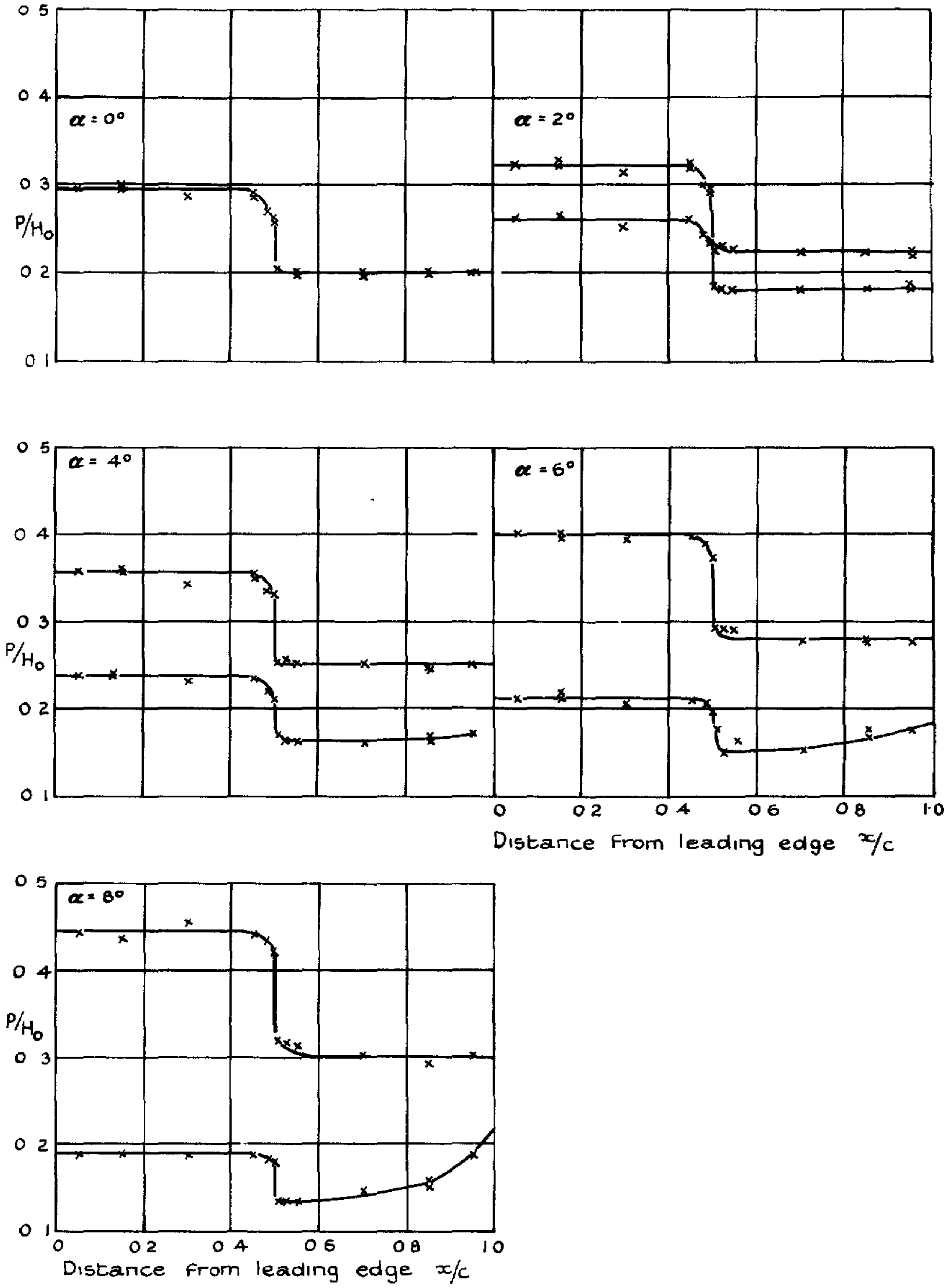
Details of the Double Wedge

FIG 2

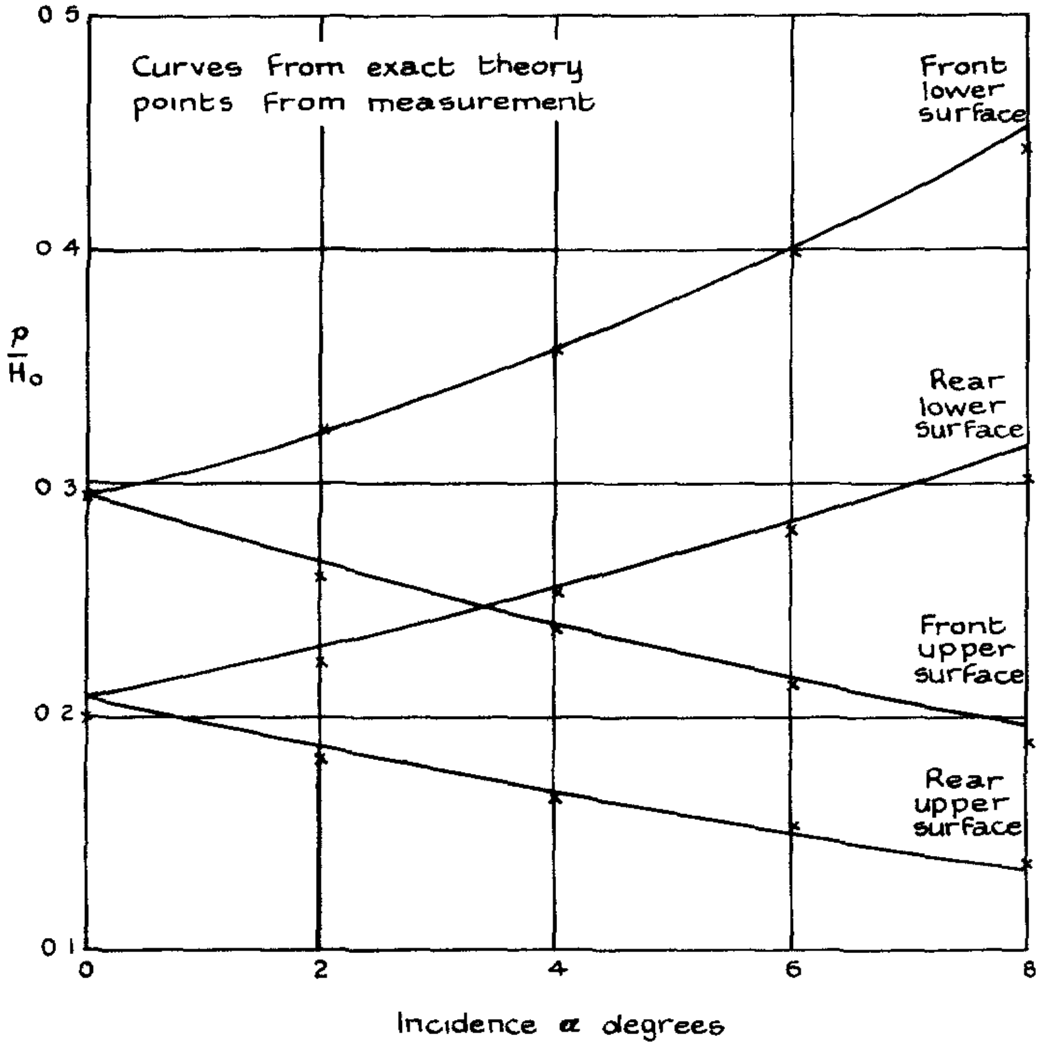


Details of the Method for Supporting the Model

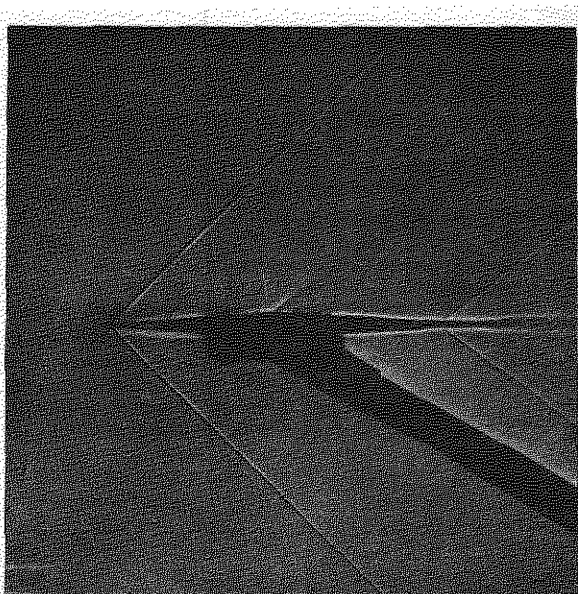
FIG 3



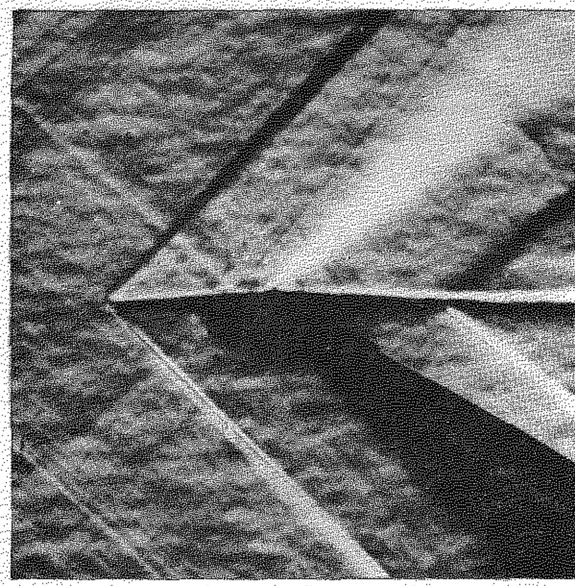
Pressure distributions on 6% double wedge



Comparison between measured and calculated pressures

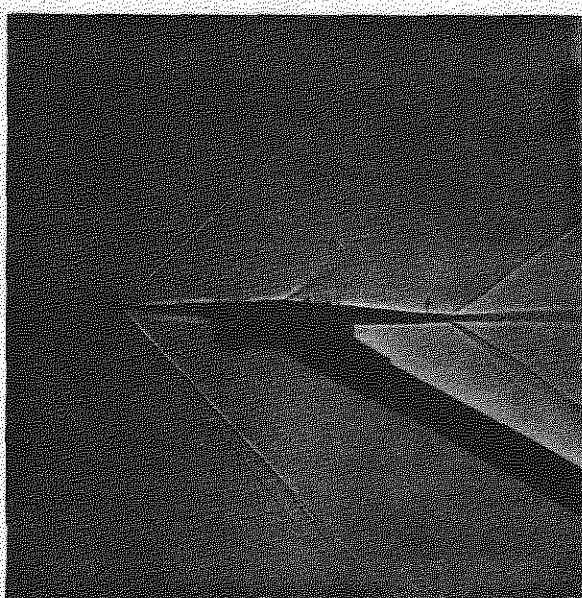


(a)

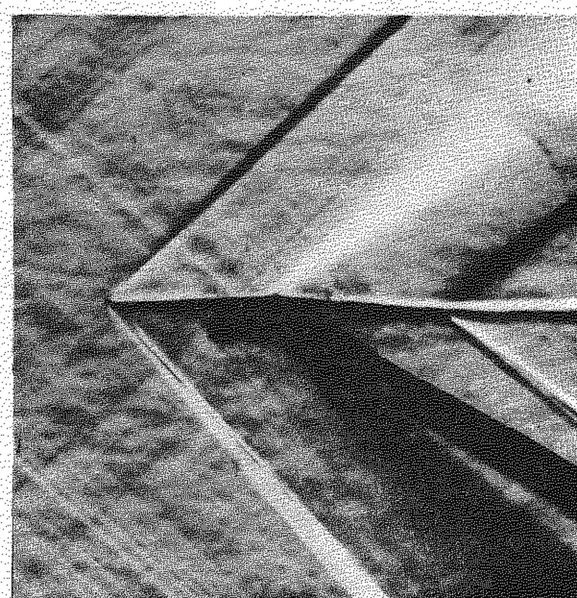


(b)

FIG. 5. $\alpha = 0°$



(a)

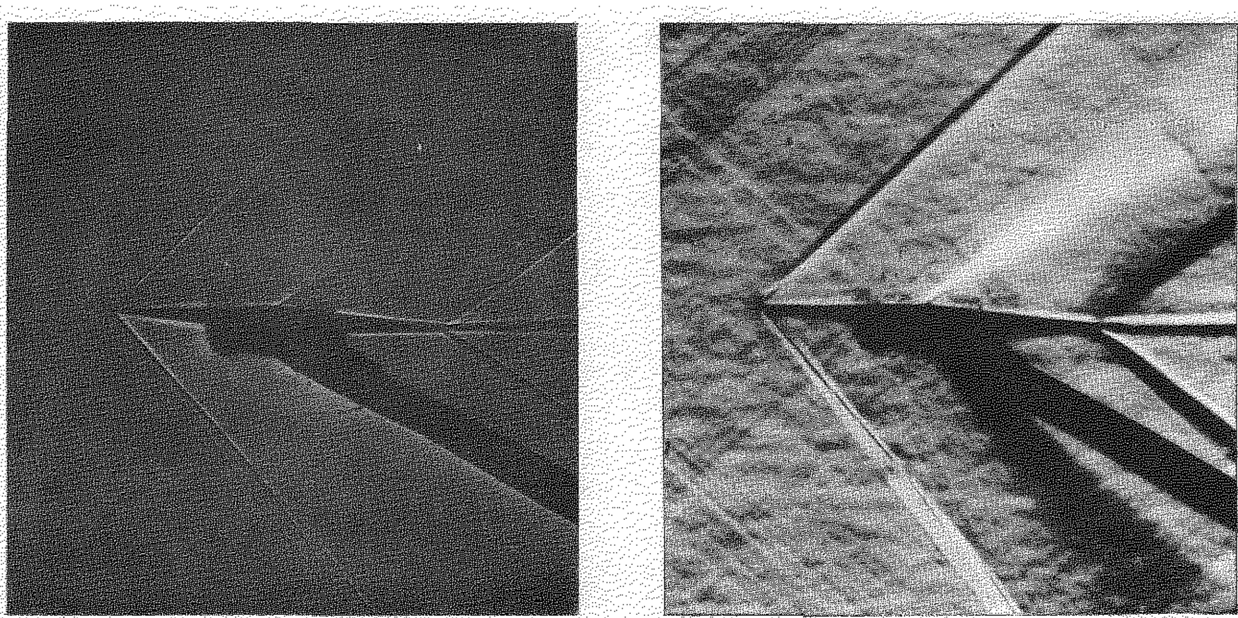


(b)

FIG. 6. $\alpha = 2°$

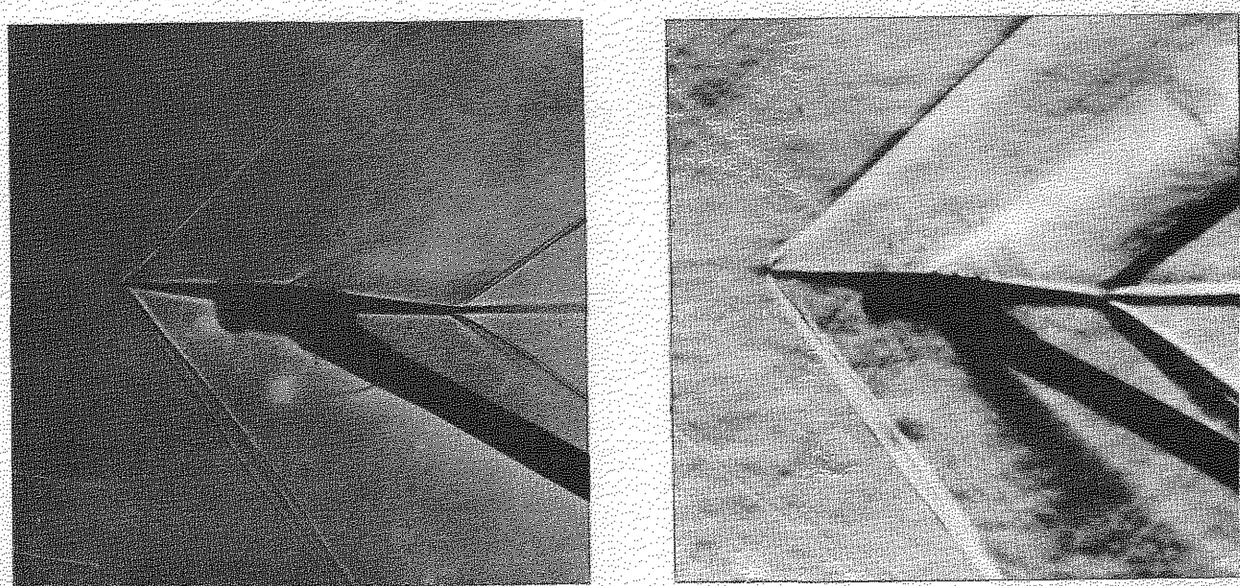
PHOTOGRAPHS OF THE FLOW ROUND A 6% DOUBLE WEDGE.

FIG.7.(a)&(b)
& FIG.8.
(a)&(b).



(a) (b)

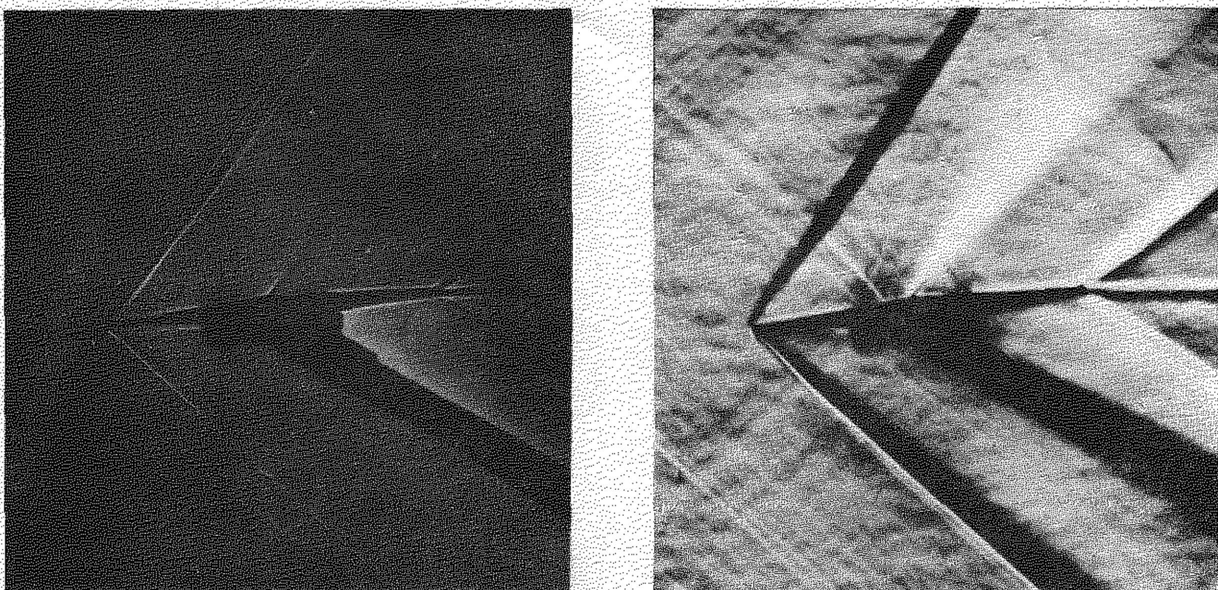
FIG.7. $a = 4.^{\circ}$



(a) (b)

FIG. 8. $a = 6.^{\circ}$

PHOTOGRAPHS OF THE FLOW ROUND A 6% DOUBLE WEDGE.



(c) (d)

FIG. 8. $\alpha = -6^{\circ}$

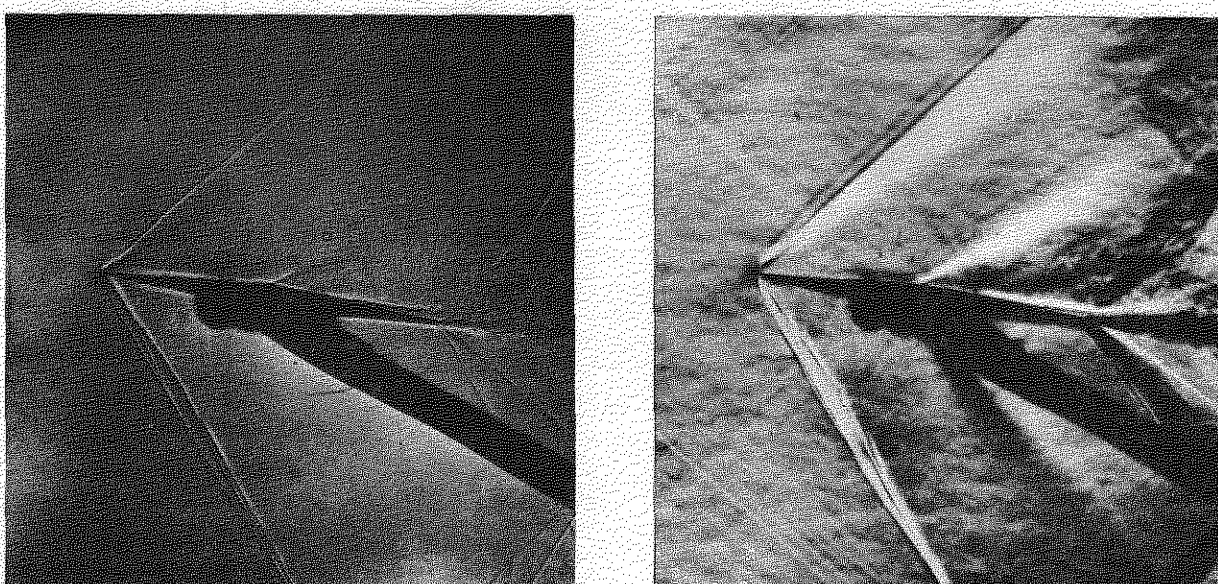
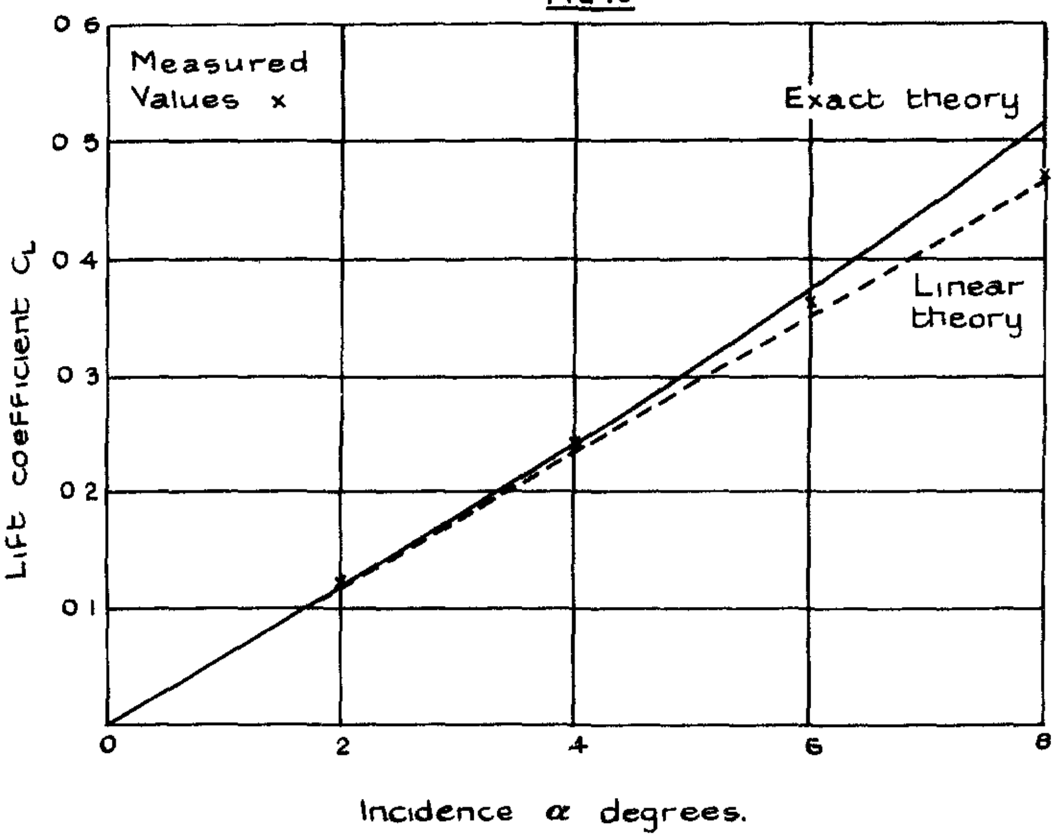


FIG. 9. $\alpha = 8^{\circ}$

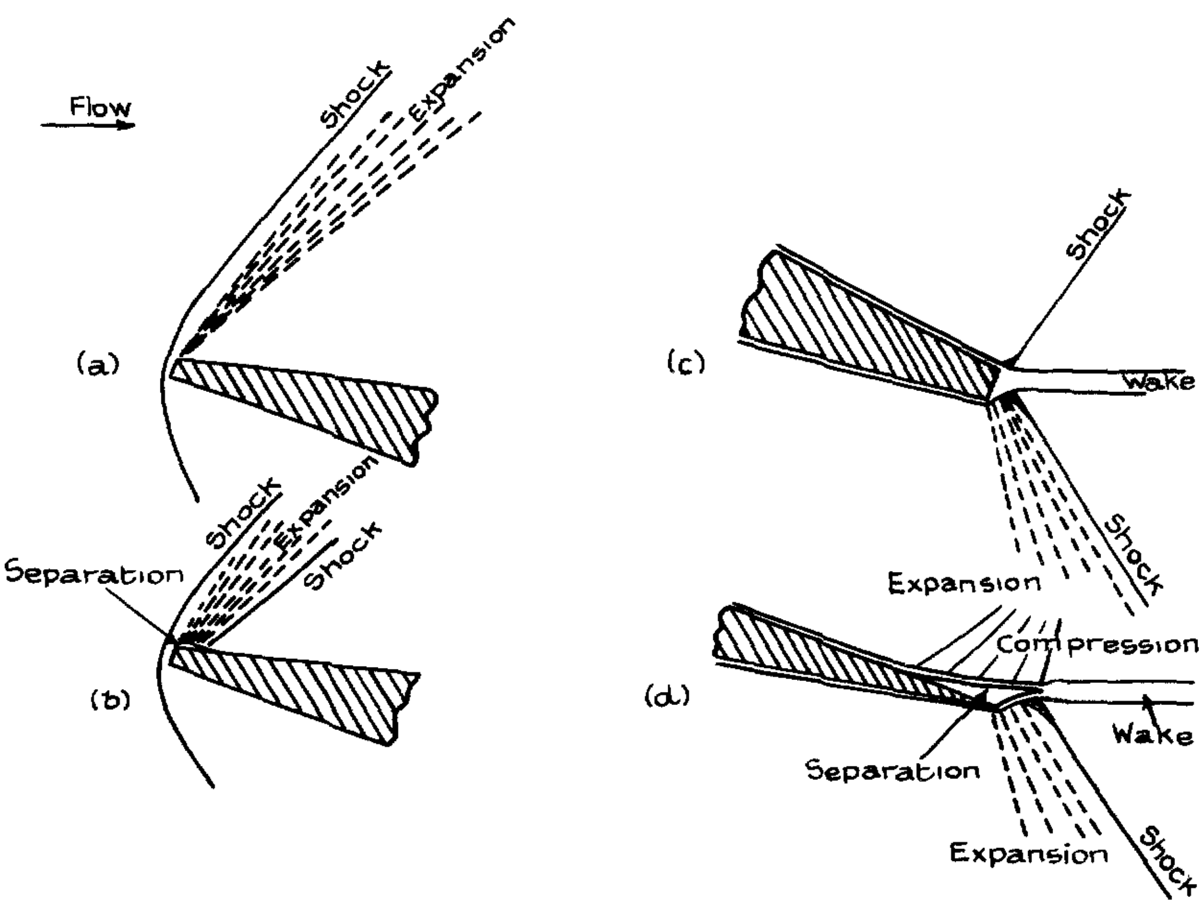
PHOTOGRAPHS OF THE FLOW ROUND A 6% DOUBLE WEDGE.

FIG 10



Variation of lift coefficient with Incidence

FIG 11



Phenomena arising from Blunt Leading and Trailing Edges.

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