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Observations of the Interaction between the Shock Waves and Boundary Layers at the Trailing Edges of Aerofoils in Supersonic Flow

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TWO SHILLINGS NET

Observations of the Interaction between the Shock Waves and Boundary Layers at the Trailing Edges of Aerofoils in Supersonic Flow.

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28th December, 1950

Summary

Photographs have been taken of the flow at the trailing edges of three unyawed, two-dimensional, aerofoils placed in an airstream moving with a Mach number of 1.6 and giving a Reynolds number (based on chord) of 0.8×10^6. It is found that the conditions close to the trailing edge depend on the strength of the trailing-edge shock, particularly if the boundary layer is laminar ahead of the shock.

1. Introduction

The interaction between the shock waves and boundary layers on aerofoils moving at high subsonic speeds has been the subject of several investigations, but, although there is evidence that it may produce considerable changes in the pressure distribution upstream, the interaction between the tail shock and the boundary layer on an aerofoil moving at supersonic speeds seems to have received little attention.

Two quantities which are known to play an important part in determining the nature of the interaction on aerofoils at high subsonic speeds are the strength of the shock and the state of the boundary layer. The present observations were made to obtain qualitative information on the importance of these quantities in the case of the interaction at the trailing edge of an aerofoil in supersonic flow. The observations are of a preliminary nature and are to be followed by detailed measurements on systematic families of double-wedge and biconvex aerofoils.

2. Apparatus and Technique

The observations were made in the 9" × 3" wind tunnel which for the present purpose was fitted with a nozzle designed to give a uniform flow at a Mach number of 1.6. The aerofoils tested were all of 2" chord, and were of 12% double-wedge, 6% biconvex and EC 1250 section respectively. Direct-shadow photographs of each flow were taken with the photographic plate 2" away from the working section and Toepler schlieren photographs were taken with an apparatus based on two 9' focal length f 12 mirrors. In all cases the photographic exposure was of the order of 1 microsecond.

The/

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The position of transition was observed by a sublimation technique in which the surface of the model was sprayed with a 10% solution of acenaphthene in acetone. For some of the tests transition was fixed by placing a band of Frigilene about 0.002" thick between 0.15 and 0.20 chord behind the leading edge. Separation of the boundary layer was detected by covering the rear of the model with a layer of oil. In some cases transition took place behind the trailing edge of the aerofoil; it was then detected from the direct-shadow photographs using the criterion suggested by Pearcey[1], and by observing the position at which the image of the edge of the wake ceased to be sharp.

3. Results

Typical schlieren photographs showing the general flow round the EC 1250 aerofoil and the 12% double wedge are reproduced in Figs. 1 and 2 respectively. In Figs. 1(a) and 2(a) the boundary layer was laminar back to a point close to the trailing edge and in Figs. 1(b) and 2(b) transition was fixed at about 0.2 chord. The photographs show the shock waves springing from the band of Frigilene used to fix transition.

Photographic enlargements ˣ showing the flow close to the trailing edges of the aerofoils are reproduced in Figs. 3, 4, 5 and 6. Details are set out in Table 1.

Photographs of the pattern revealed by the chemical transition indicator described in section 2 are not reproduced, but a scale drawing of a typical example is given in Fig. 7. The regions of turbulent flow at the ends of the span arise from contamination by the boundary layers on the side walls of the tunnel, and those close to mid span from dust particles or other exorescenses close to the leading edge.

Table 1./

ˣIt was difficult to keep the sensitivity of the schlieren apparatus constant throughout the experiments. The sensitivity in Fig. 6 (g) is lower than in the other photographs, and the compressions arising close to the separation point are not clearly visible.

TABLE I. DETAILS OF THE PHOTOGRAPHS

Aerofoil	Trailing-edge Angle	Incidence	Strength of Tail Shock: Upper Surface	Strength of Tail Shock: Lower Surface	Approximate Position of Transition: Upper Surface	Approximate Position of Transition: Lower Surface	Approximate Position of Separation: Upper Surface	Approximate Position of Separation: Lower Surface	FIG.
EC 1250	31°	0°	2.2 (approx.)	2.2 (approx.)	1.3 (?)	1.3 (?)	0.91	0.89	3(a)
			2.2 (approx.)	2.2 (approx.)	0.15	0.15	1.0	1.0	3(b)
12% Double Wedge	14°	0°	1.42	1.42	1.1 (?)	1.1 (?)	0.96	0.98	4(a)
			1.42	1.42	0.15	0.15	1.0	1.0	4(b)
6% Biconvex	14°	0°	1.42	1.42	1.1 (?)	1.1 (?)	1.0	0.98	5(a) & 6(a)
		0°	1.42	1.42	0.15	0.15	1.0	1.0	5(b) & 6(b)
		2°	1.63	1.28	1.1 (?)	1.1 (?)	0.92	1.0	5(c) & 6(c)
		2°	1.63	1.28	0.15	0.15	1.0	1.0	5(d) & 6(d)
		4°	1.82	1.18	1.07(?)	1.07(?)	0.89	1.0	5(e) & 6(e)
		4°	1.82	1.18	0.15	0.15	1.0	1.0	5(f) & 6(f)
		6°	2.0	1.05	1.05(?)	1.05(?)	0.86	1.0	5(g) & 6(g)
		6°	2.0	1.05	0.15	0.15	1.0	1.0	5(h) & 6(h)
			Calculated by exact (Inviscid) Theory.		Fraction of Chord Behind Leading Edge				

4. Discussion

Examination of the photographs and of the measurements set out in Table 1 shows that when the boundary layer is laminar the flow separates ahead of the trailing-edge particularly when the trailing-edge angle is high or (from the upper surface only) when the aerofoil is at incidence. It seems reasonable to suppose that this separation is caused by the pressure rise at the tail shock wave, and extends upstream when the trailing-edge angle or incidence is raised because the shock strength increases. Secondary effects arise from the effects of aerofoil section shape and incidence on the thickness and profile of the boundary layer.

The measured positions at which separation occurred are plotted in Fig. 8 against the calculated strength (neglecting viscous effects) of the tail shock. Values obtained from an analysis of Ferri's measurements[2] on a 10% biconvex aerofoil (G.U.2), and on an 8.8% aerofoil (G.U.3) with one plane surface and the other in the form of a circular arc are included. These tests were made at a Reynolds number close to that of the present work, and it has been assumed that separation is present over that part of the surface over which the measured pressure is constant.[x] The importance of the strength of the tail shock as a parameter in determining the position of separation is confirmed by the existence of some correlation between the points plotted in Fig. 8, but the scatter shows that the parameters neglected in this diagram play an important part.

When laminar separation takes place, the tail shock is softened and is replaced by several weaker shocks as shown, for example, in Fig. 3(a). The first of these appears in this case to originate at the separation point, a second from close to the trailing edge, a third from the region where the separated streams from the two surfaces meet, and a fourth from the point where the wake widens rapidly. There is some evidence from the direct-shadow photographs that this last point may be that at which transition takes place.

When the boundary layer is made turbulent, there seems to be little change in the flow pattern at the trailing edge in cases (see Figs. 4(a) and 4(b)) where little or no separation occurs for a laminar layer, but when separation is present (see Figs. 3(a) and 3(b)) it is suppressed when the spoilers are attached to the aerofoil. The tail shock then takes the form of a single compression wave and springs from a point close to the trailing edge.

When laminar separation takes place on the upper surface, the main compression at the tail seems to occur downstream of the trailing edge. The pressure just behind the trailing edge would, therefore, be expected to be lower than when separation is absent. This is thought to explain why the flow on the lower surface expands round the trailing edge (see for example Fig. 5(g)) before the shock wave occurs there. This phenomenon is more clearly visible in some photographs taken at the tail of a double wedge and reported in reference 3.

5. Conclusions/

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[x] In the absence of viscous effects, theory predicting a continuously falling pressure.

5. Conclusions

In supersonic flow it is found that particularly when the boundary layer is laminar, the flow may separate ahead of the trailing edge of an aerofoil because of the presence of the shock wave there. Separation does not seem to be extensive when the strength of the tail shock (i.e. the static-pressure ratio across the shock) is less than about $1\frac{1}{2}$, but extends forward rapidly as the shock strength is raised above this value by increasing the trailing-edge angle or the incidence of the aerofoil. For tail-shock strengths up to at least about 2, separation can be suppressed by making the boundary layer turbulent by fitting spoilers to the surfaces of the aerofoil.

When separation occurs, the tail shock wave is replaced by several weaker shocks, and the wave drag of the aerofoil may be reduced. Separation may also have an important effect on the behaviour of a trailing-edge control surface.

There is no evidence that these conclusions apply at Mach and Reynolds numbers which differ widely from those (1.6 and 0.8×10^6) of the present tests, but if the boundary layer is turbulent over the rear of the aerofoil at the full-scale Reynolds number, it seems that laminar separation may cause appreciable scale effects in tunnel tests made at low Reynolds number.

REFERENCES

No.	Author(s)	Title, etc.
1.	H. H. Pearcey	The Indication of Boundary-layer Transition on Aerofoils in the N.P.L. 20" × 8" High-Speed Tunnel. C.P. No.10.
2.	A. Ferri	Some Experimental Results obtained on Wing Profiles in the Guidonia Super-sonic Wind Tunnel. (Atti di Guidonia. No. 17, 20/9/39). M.A.P. Translation No. 1115. Translated by M. Flint. (1940). A.R.C. 4780
3.	D. W. Holder, A. Chinneck & D. G. Hurley	Observation of the Supersonic Flow Round a 6% Thick Double Wedge. (1950) 28th December, 1950. A.R.C. 13,623

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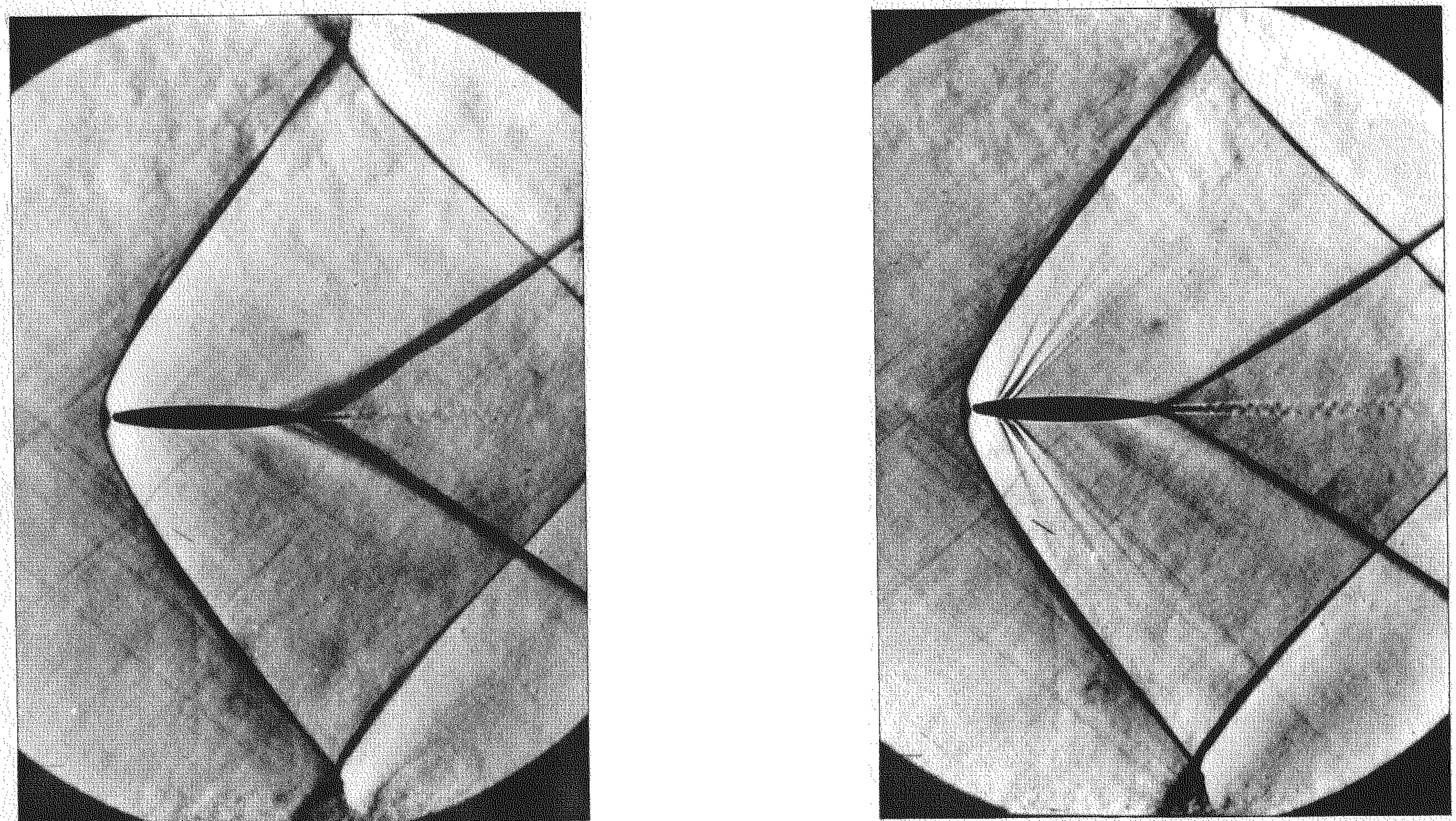
FIG. 1.



(a) TRANSITION FREE

(b) TRANSITION FIXED AT 0.15 CHORD

PHOTOGRAPHS OF THE FLOW ROUND EC1250 AEROFOIL AT M = 1.6. a = 0°

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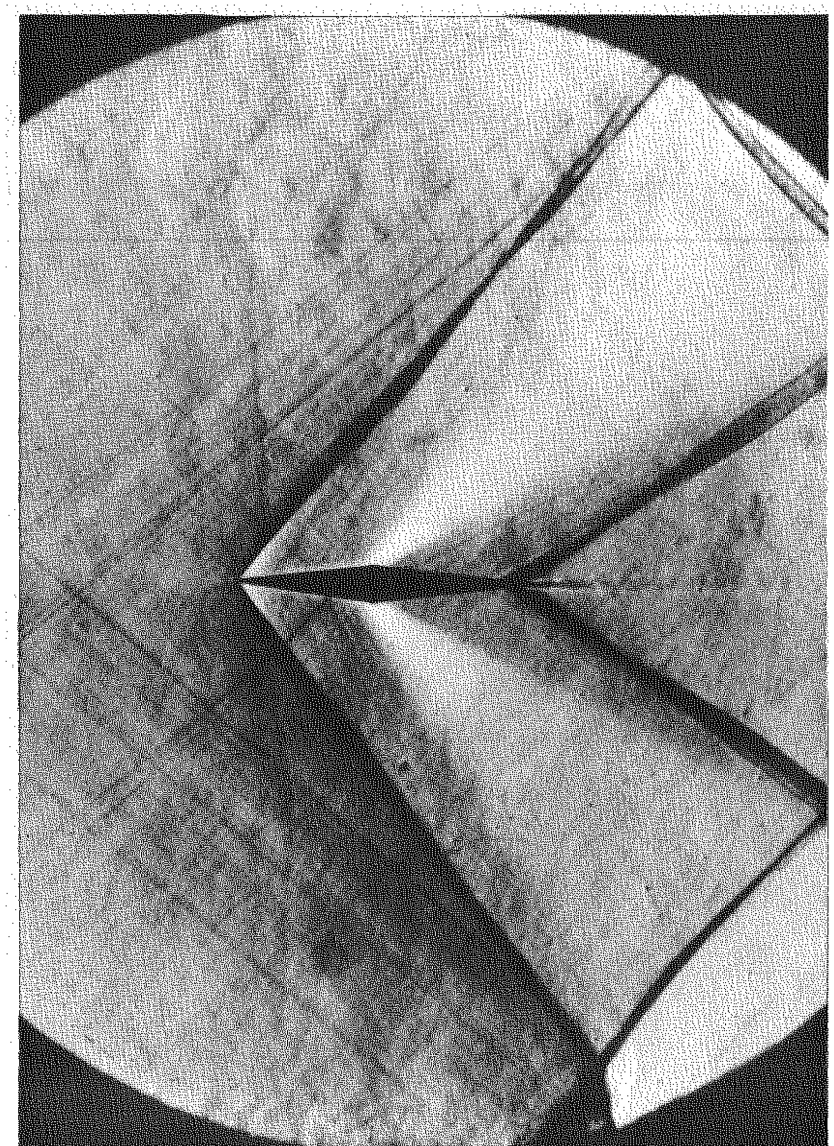
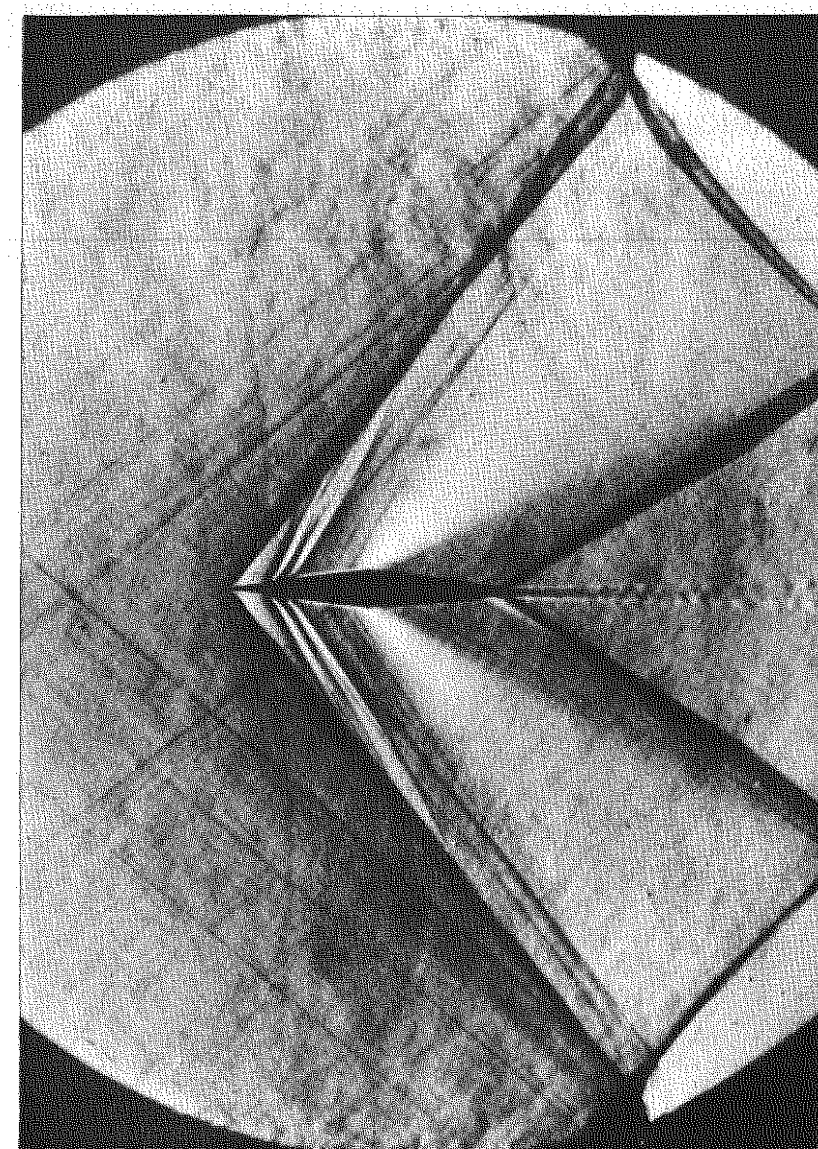
FIG. 2.



(a) TRANSITION FREE



(b) TRANSITION FIXED AT 0.15 CHORD

PHOTOGRAPHS OF THE FLOW ROUND 12½% DOUBLE WEDGE AEROFOIL AT $M = 1.6$, $a = 0°$

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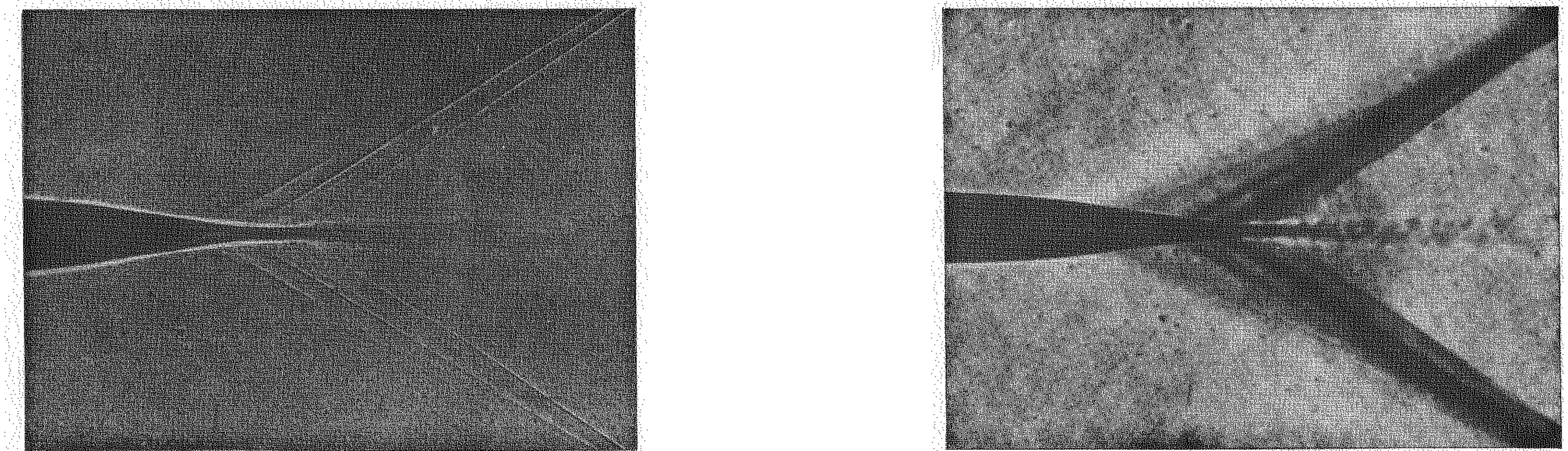
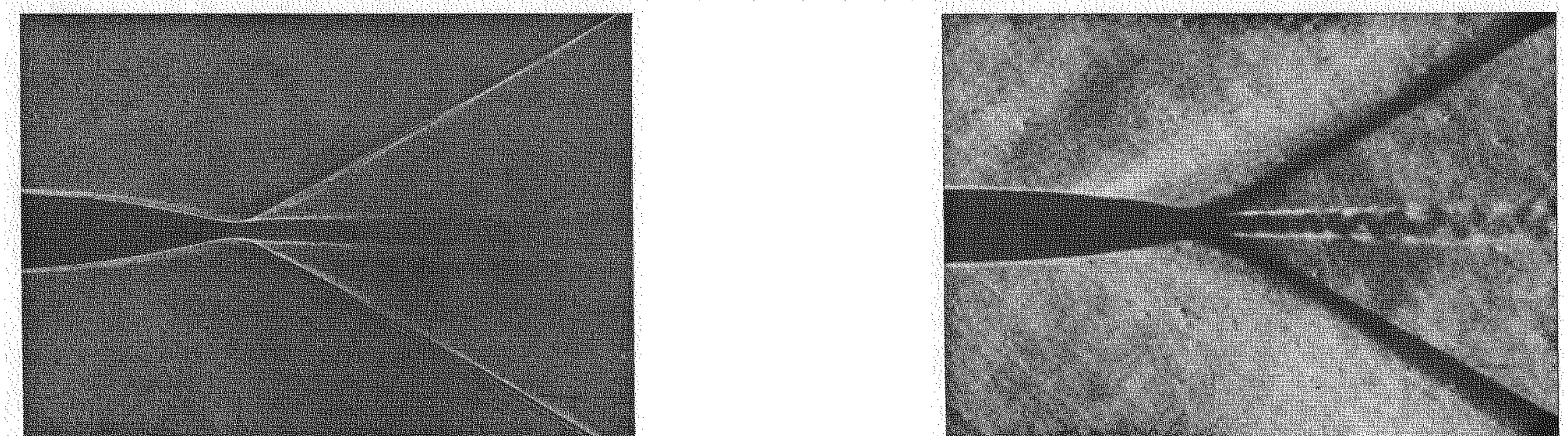
FIG. 3



(a) LAMINAR BOUNDARY LAYER

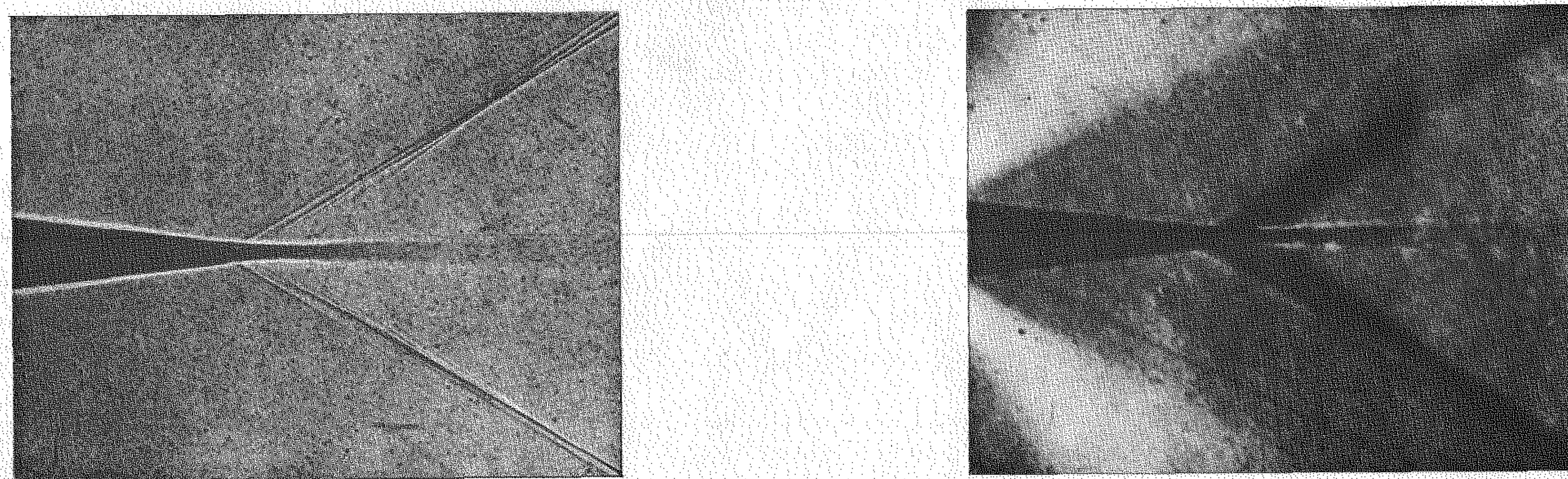


(b) TURBULENT BOUNDARY LAYER

0 0·5 1·0

Scale in Fractions of Chord

PHOTOGRAPHS OF THE FLOW AT THE TRAILING EDGE OF EC1250 AEROFOIL AT M = 1.6. a = 0°



(a) LAMINAR BOUNDARY LAYER

(b) TURBULENT BOUNDARY LAYER

0 0·5 1·0

Scale in Fractions of Chord

PHOTOGRAPHS OF THE FLOW AT THE TRAILING EDGE OF 12½% DOUBLE WEDGE AEROFOIL AT M = 1.6. a = 0°

FIG. 4.

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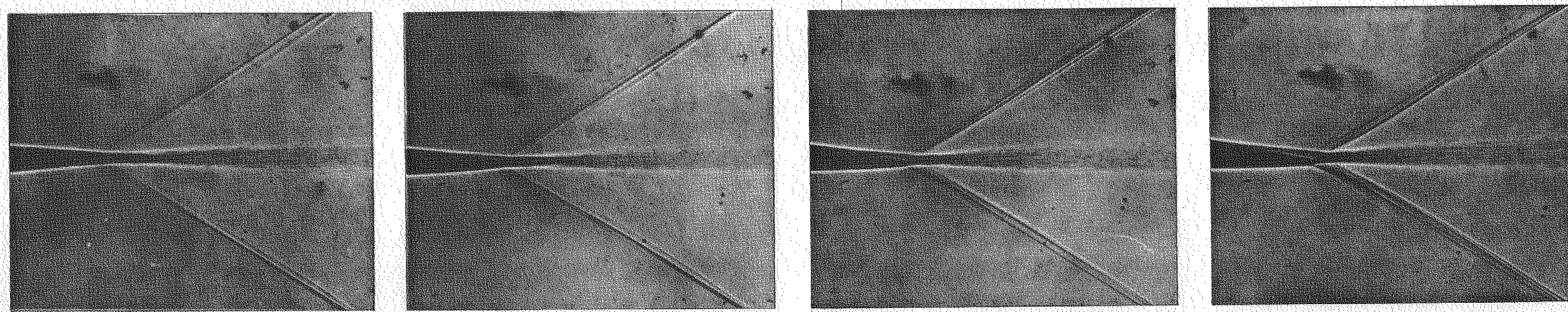
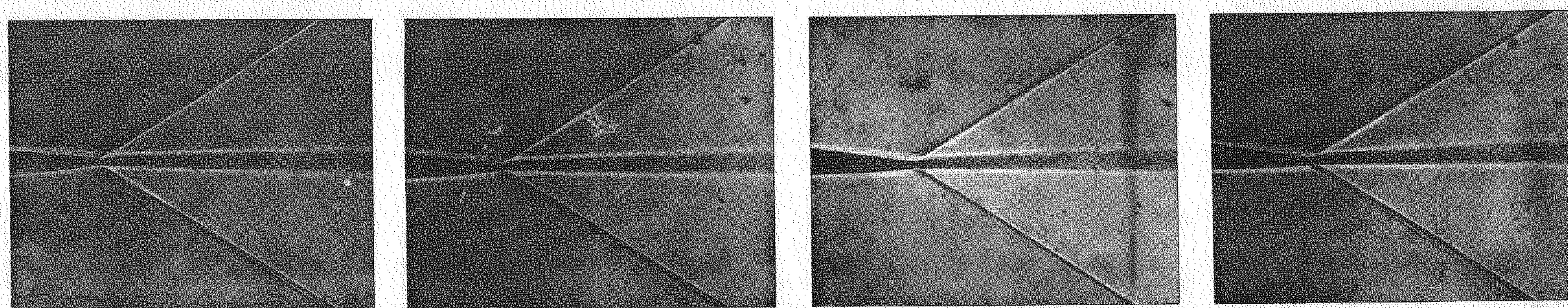
FIG. 5.



(a) a = 0° (c) a = 2° (e) a = 4° (g) a = 6°

LAMINAR BOUNDARY LAYER



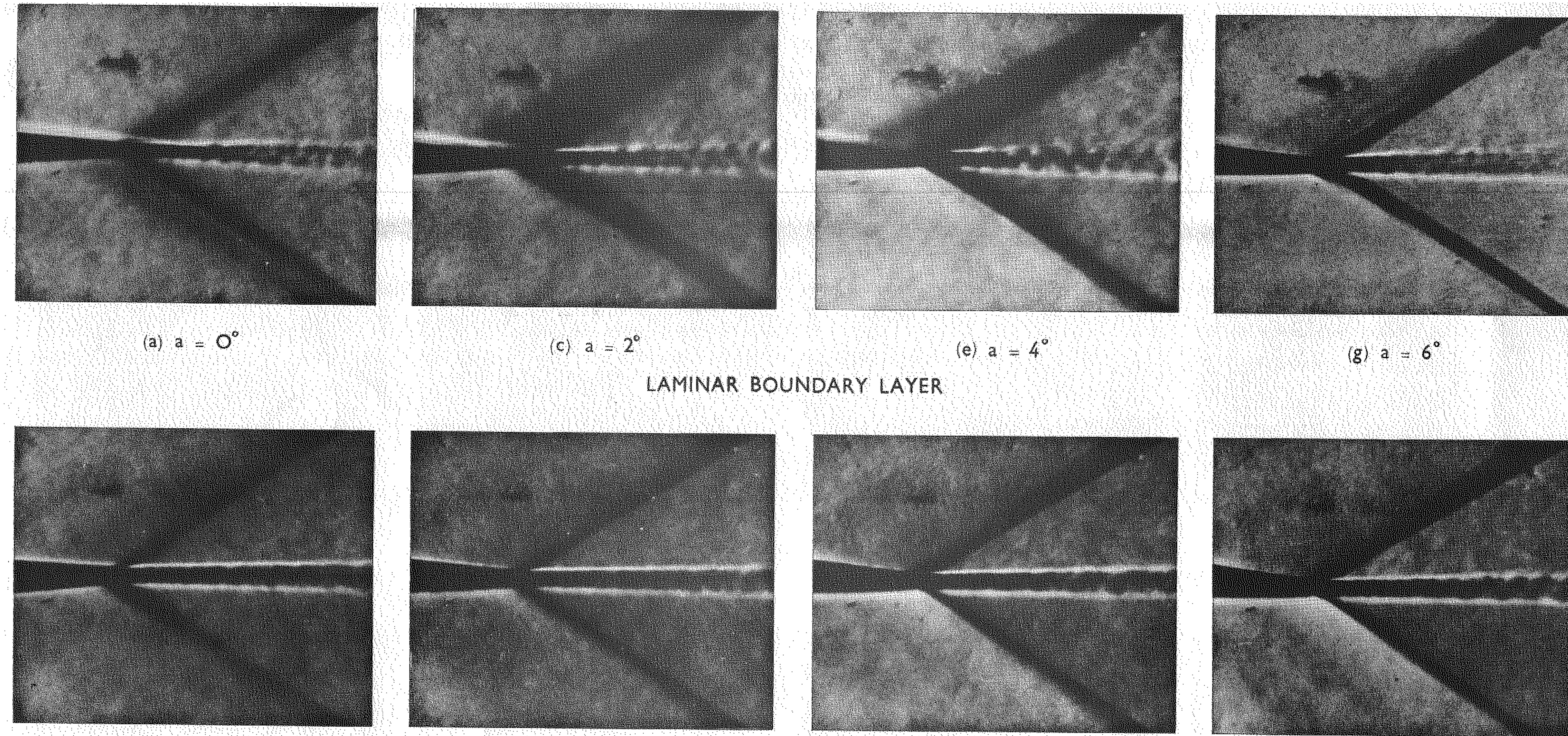
(b) a = 0° (d) a = 2° (f) a = 4° (h) a = 6°

TURBULENT BOUNDARY LAYER

0 0.5 1.0

Scale in Fractions of Chord

DIRECT SHADOW PHOTOGRAPHS OF THE FLOW AT THE TRAILING EDGE OF 6% BICONVEX AEROFOIL AT M = 1.6.



(a) a = O° (c) a = 2° (e) a = 4° (g) a = 6°

LAMINAR BOUNDARY LAYER

(b) a = O° (d) a = 2° (f) a = 4° (h) a = 6°

TURBULENT BOUNDARY LAYER

0 0·5 1·0

Scale in Fractions of Chord

PHOTOGRAPHS OF THE FLOW AT THE TRAILING EDGE OF 6% BICONVEX AEROFOIL AT M = 1.6 a = O°, 2°, 4°, 6°.

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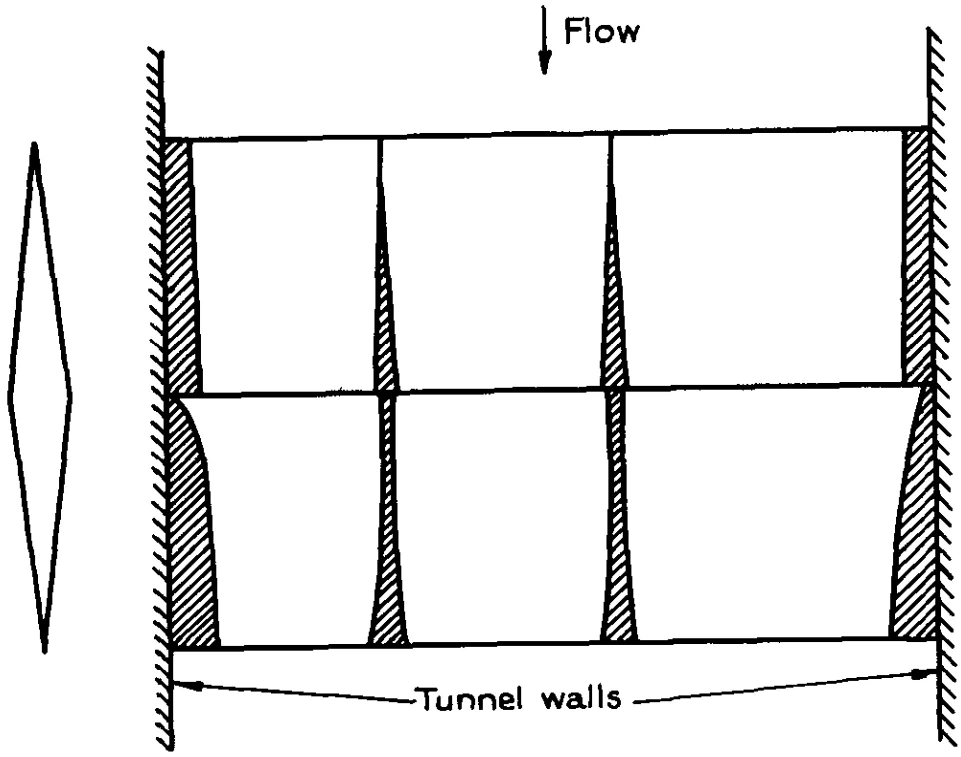
FIGS. 7 & 8.

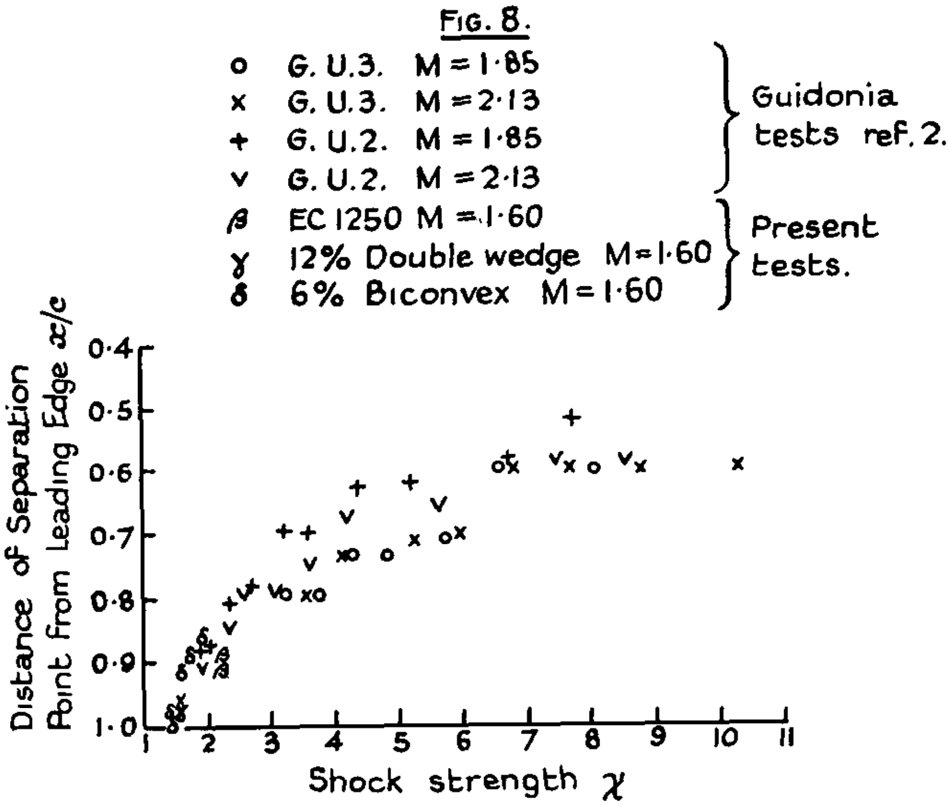
FIG. 7.



Turbulent Regions shaded thus

Pattern Revealed by Transition Indicator on 12% Double Wedge at $\alpha = 0°$.

FIG. 8.



Variation of Separation-Point Position with the Strength of the Tail Shock for a Laminar Boundary Layer

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