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Photographs of Shock Wave Movement

By

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of the Aerodynamics Division, N.P.L.

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THREE SHILLINGS NET

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Summary.—Consecutive photographs were taken at millisecond intervals of the flow past a low-drag aerofoil at compressibility speeds. At a Mach number 0·1 above the 'pressure critical' the shock wave was found to oscillate rapidly but aperiodically, whereas the edge of the associated boundary layer remained quite steady, at least for periods of $^1/_{50}$ sec. At the critical Mach number and just below it a series of small shock waves was observed, apparently moving against the direction of flow.

Note.—The photographs reproduced in this paper were taken in January, 1944. Their issue was postponed in view of possible improvement in technique.

Introduction.—The tests described in this report may be subdivided as follows :—

1. Determination of the period and amplitude of the oscillation of the shock wave at a speed above the critical Mach number.

2. A similar determination of the oscillation of the edge of the boundary layer just behind this shock wave, which may be the cause of buffeting in flight.

3. Investigation of the small shock waves visible on single photographs taken at speeds just below the critical speed, in the region of maximum (but subsonic) velocity. It is not known whether these waves were caused by surface irregularities, or by extraneous tunnel effects, or were genuine effects. This type of flow changes gradually into that classified in (1) above, as the free stream velocity is increased.

One case of each of the above types was investigated by photography of the flow at successive intervals of the order of one millisecond.

Experimental Details.—The photographs were taken in the National Physical Laboratory 20 × 8 in. Rectangular High Speed Tunnel using the direct shadow method with parallel light as outlined in Ref. 1. The electrical circuit, shown in Fig. 1 utilises an induction coil to charge a condenser of capacity 0·0015μF to a voltage of 30 kV. This is usually discharged through a 3-cm air gap to give a single spark. Using 60 kV and a gap of length 0·5 cm an oscillatory discharge of 10 or 20 sparks was produced. The time interval between these sparks varied between $0{\cdot}5 \times 10^{-3}$ and $3{\cdot}0 \times 10^{-3}$ seconds, the whole sequence lasting about 20×10^{-3} seconds.

A set of photographs was obtained on a 20-inch strip of film wound on a drum which usually rotated at 3,000 r.p.m., or approximately 1 inch per millisecond.

The aerofoil was a 14 per cent thick low-drag section[2] (1442/1547) with Goldstein 'roof-top' distribution) of 5-in. chord designed by Mr. E. J. Richards using the method set out in Ref. 3. At the design incidence, $\alpha = 0{\cdot}5$ deg, at which these observations were made, the low speed C_L is 0·2. The theoretical critical Mach number is 0·69 and the critical Mach number deduced from pressure observations is 0·695. A few pressure holes in the surface enabled the pressure distribution to be observed simultaneously with the photographs (Fig. 2).

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A static pressure exploration was made at $M = 0{\cdot}78$ for case (1) (Fig. 3)$^{\neq}$ and at $M = 0{\cdot}69$ for case (3) (Fig. 5) to find the velocity gredients along and normal to the surface^{+}.

The flexible walls of the tunnel were shaped by the method of R. & M. 2005[6].

A full-size photograph, taken with a single spark, of the flow past the entire wing at $M = 0{\cdot}78$ ($\alpha = 0{\cdot}5$ deg) is given in Fig. 8. This shows the aerofoil and its external support and a pointed rod penetrating the main shock wave so as to demonstrate by means of the bow wave the existence of the supersonic region ahead of the wave. The wake and thickened boundary layer are also visible ; the field of view is limited by the diameter of the lens. Figs. 9 and 10 for $M = 0{\cdot}78$ and Figs. 12, 13 and 14 for $M = 0{\cdot}690$ and $0{\cdot}680$ are portions of the field of flow on the upper surface selected by appropriate masking while Fig. 11 for $M = 0{\cdot}81$ shows the trailing edge region. Figs. 9 to 14 were taken with a multiple spark discharge as described above. A fixed reference line is shown in Figs. 9 and 10.

Discussion of Results.—1. *Vibration of the Shock Wave on the Upper Surface at* $M = 0{\cdot}78$, ($0{\cdot}085$ *above the pressure* M_{crit}).—At $M = 0{\cdot}78$ the shock wave on the upper surface is formed at about $x/c = 0{\cdot}65$, as shown in Fig. 8. Figs. 9 and 10 are representative of several sets of successive photographs that were obtained. The multiplicity of shock waves apparent in these photographs is almost certainly due to a spanwise variation of their position and not to a series of successive shock waves. Since a shock wave is only visible when the incident light is tangential to its surface, the lines seen on the photographs are the maximum and minimum fore and aft positions of the wave at that instant (*see* Fig. 4).

It is thought that the most rearward waves in each frame may be near the glass walls where the shock is becoming softened by its proximity to the boundary layer on the wall. Near the aerofoil surface the wave formation was sometimes single (Figs. 8 and 10) and sometimes bifurcated (Fig. 9), $M = 0{\cdot}78$ being the speed at which the bifurcated formation begins.

The change of shape of the wave patterns in successive exposures in Figs. 9 and 10 shows that the wave form (Fig. 4) was changing with time at a rapid rate. It is evidently impossible to deduce from the photographs the actual oscillation in the direction of the stream of a particular point on the wave of Fig. 4 ; the best that could be done was to plot against time (Fig. 7) the displacement in the stream direction of the maximum of the curve of Fig. 4 which is represented by the distance forward of the datum line of the uppermost shock wave image line in each frame of Fig. 10 at a given distance ($0{\cdot}2c$ for Fig. 7a) from the surface. In set 1 (Fig. 7b) the maximum measured amplitude is nearly 0·2 in. (0·04 chord) and most of this movement can take place in less than one millisecond. In set 2 (Fig. 7a) there is little movement, the maximum variation in the whole time of observation (8 milliseconds) being less than 0·1 in. (0·02 chord). Most of the other sets of photographs did not possess enough good frames to warrant measurement.

The corresponding static pressure distribution from the holes on the aerofoil surface is shown in Fig. 3 together with the static pressures through the wave measured at various distances from the aerofoil surface with a static tube. It will be seen that there is the usual gradual rise of pressure through the wave instead of the sharp change that theory predicts for a stationary shock. In this case the pressure recovery extends over 0·2 or 0·3 in. (*i.e.*, 0·04 or 0·06 chord) and is, therefore, of the same order of size as the measured shock wave movement (Fig. 7). There are also two alternative effects that may cause a gradual rise of pressure, namely the ' softening ' effect of the boundary layer, which should not be large enough to affect the reading of a small static tube away from the aerofoil boundary layer, and the interference between the shock wave and the static tube. The possible magnitude of this interference in the present case, is not known. It appears, however, that the wave oscillation is sufficient to account for the observed pressures and is probably the most important effect.

At the end of the shock wave where its amplitude has decreased to zero a sonic region exists and a set of conditions similar to those examined in section 3 is observed.

$\neq$ This figure is almost identical with Fig. 3 of Ref. 4.

$+$ Similar explorations at high speeds for a *Mustang* wing section are given in R. & M. 2468[5].

2. *Oscillation of the Boundary Layer near the Shock Wave.*—Fig. 11 gives photographs of the thickening of the boundary layer caused by a shock wave similar to that discussed in section 1, except that the speed has been increased slightly to $M = 0{\cdot}81$. The edges of the boundary layers just behind the shock waves on each surface are seen to be fairly steady (Fig. 6) but it is possible that fluctuations in pressure are occurring, which although not sufficient to move the edge of the wake near the surface, may cause buffeting behind the aerofoil.

It is clear that a complete study of this phenomenon can only be made with the use of a small device that will measure velocity variations in a high speed stream.

3. *Shock Waves Below the Critical Speed* ($M_{\text{crit.}} = 0{\cdot}695$).—Photographs of several aerofoils taken at speeds just below the critical Mach number in the N.P.L. Rectangular High Speed Tunnel have shown a series of small wavelets apparently one behind the other in the region of maximum velocity. It had been thought that these waves were due to surface irregularities, but in the case of the present tests the waves were visible at speeds as low as $0{\cdot}05a$ below the critical Mach number, and the surface was known to be smooth.

The source of these wavelets is not obvious ; theory does not preclude the existence of moving shock waves at speeds below $M_{\text{crit.}}$, but only forbids the possibility of a stationary wave. If a source of shock wavelets existed at or behind the region of maximum speed of flow past the aerofoil, then such shock wavelets would be slowed up almost to rest while passing through this region, at forward speeds just below $M_{\text{crit.}}$ Moreover, the presence of a small velocity gradient normal to the surface would cause the wavelets to swing round. At speeds near to $M_{\text{crit.}}$ the speed of movement of the wavelets along the aerofoil surface is equal to the small difference between two quantities of the order of 1,000 ft/sec, and this swinging round effect will be most pronounced.

Photographs at $M = 0{\cdot}68$ and $0{\cdot}69$, Figs. 12 to 14, show the small waves piling up at $x/c = 0{\cdot}6$. As will be seen from the measured pressures shown in Fig. 5, the velocity of the flow is nearly constant between $x/c = 0{\cdot}35$ and $0{\cdot}55$ on the upper surface at subcritical speeds with the peak at $x/c = 0{\cdot}5$, and thus the position of the maximum flow velocity and minimum wave velocity as deduced from the measured pressures is at $x/c = 0{\cdot}50$.

The swinging round of the shock wave is visible on the photograph, but it is only about a fifth of the magnitude that would be expected from the measured pressures.

Some of the waves seen in the photographs appear to be the reflected parts of those that have changed their inclination.

The nature of the disturbance that produces these waves has not yet been ascertained. In order to check whether the disturbance could originate from the injector slot, the width of the tunnel between the injector and working section was reduced by screwing inwards the adjustable throat pieces (R. & M. 2067[7]) to such a width that the velocity of sound was attained at the throat with a Mach number of 0·68 or 0·69 at the working section. It was considered that this would prevent the transmission of disturbances upstream from the injector and the fact that there was no change in the photographs is an indication that the original disturbances did not originate from the injector. The only other obvious disturbance is that produced by the wake.

The shock wave in this instance was found to be oscillating aperiodically with displacements of as much as 0·15 in., *i.e.*, 0·03 chord, taking place in 1 millisecond. The edge of the associated boundary layer appeared to be steady but this does not preclude variations in pressure inside the breakaway. The movements of the small waves observable on aerofoil surfaces before the flow reaches the speed of sound are considered.

Conclusion.—These experiments are only the beginning of the study that must be made of vibration problems at high speeds. An apparatus is under construction that will give a longer and more regular photographic record of shock-wave movement and together with this a hot-wire technique or perhaps some form of piezo-electric device should be developed for the study of variations in high speed flow.

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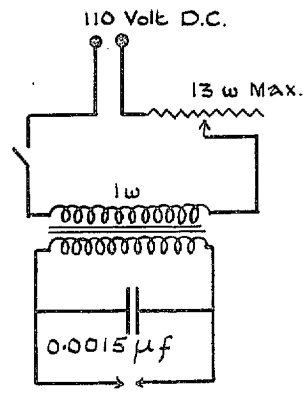

FIG. 1. Electric circuit.

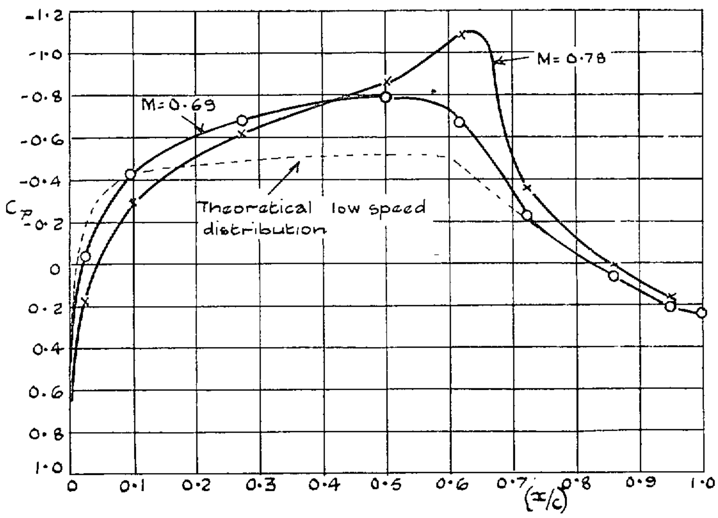

FIG. 2. Distribution of pressure on the upper surface of the aerofoil.

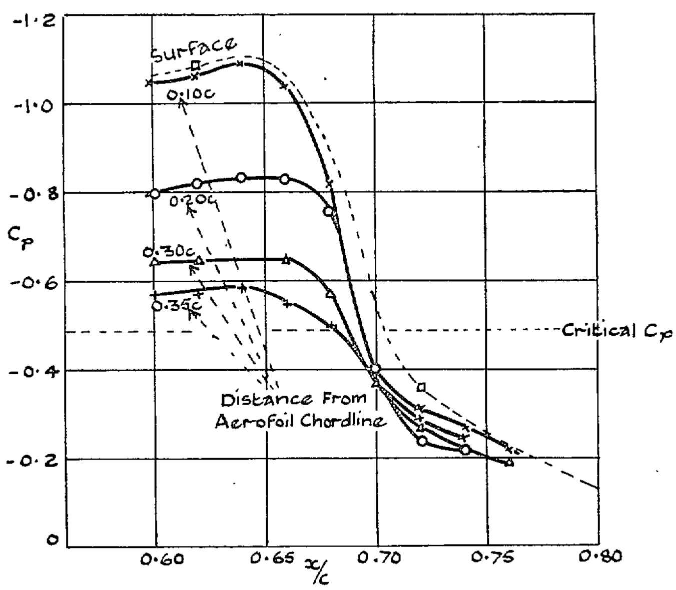

FIG. 3. Pressure distribution near upper surface at $M = 0{\cdot}78$.

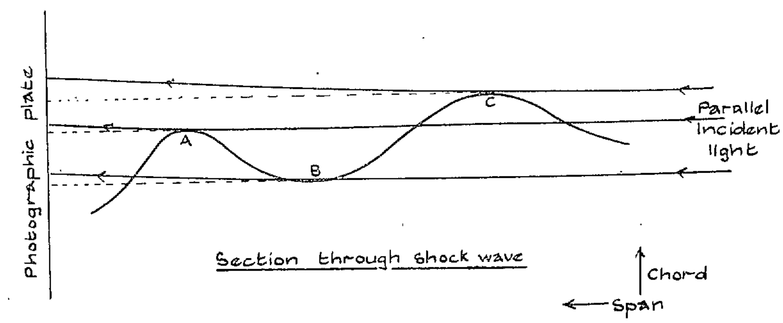

FIG. 4. The incident light is deflected at A.B.C. etc. giving rise to alternate areas of light and dark on the photograph.

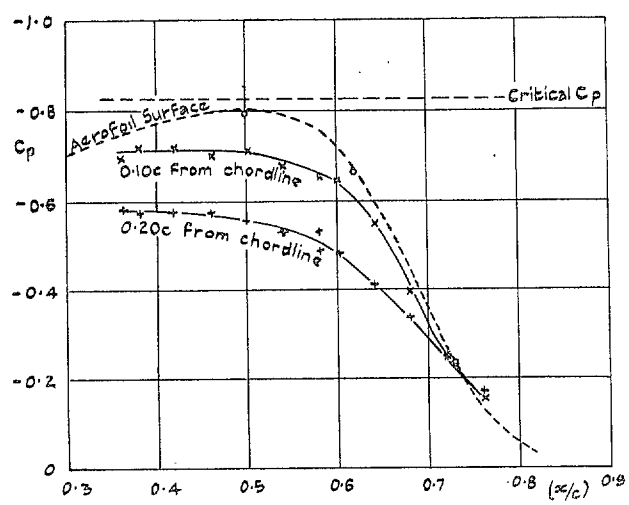

FIG. 5. Pressure distribution near upper surface at $M = 0{\cdot}69$.

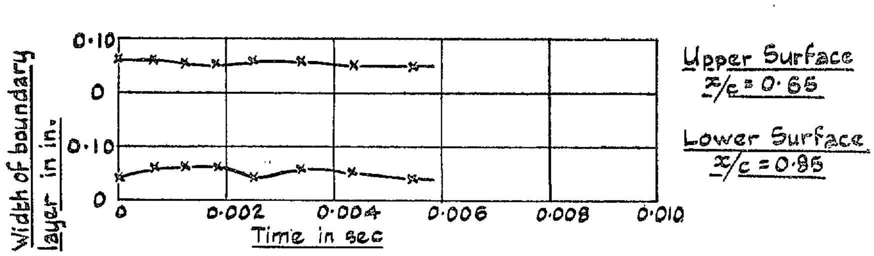

FIG. 6. Displacement of edge of boundary layer (from FIG. 11).

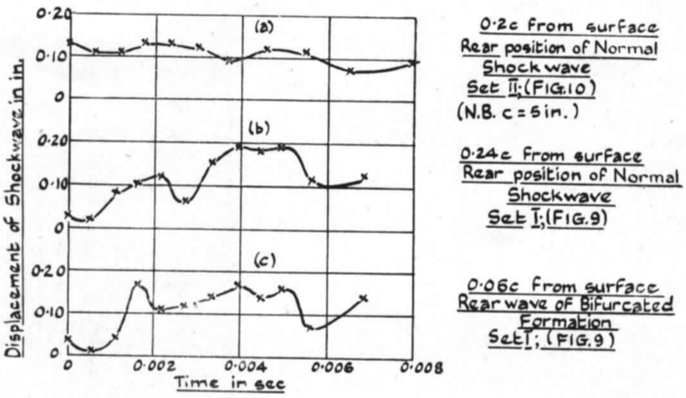

FIG. 7. Vibration of shockwaves at $M = 0{\cdot}78$.

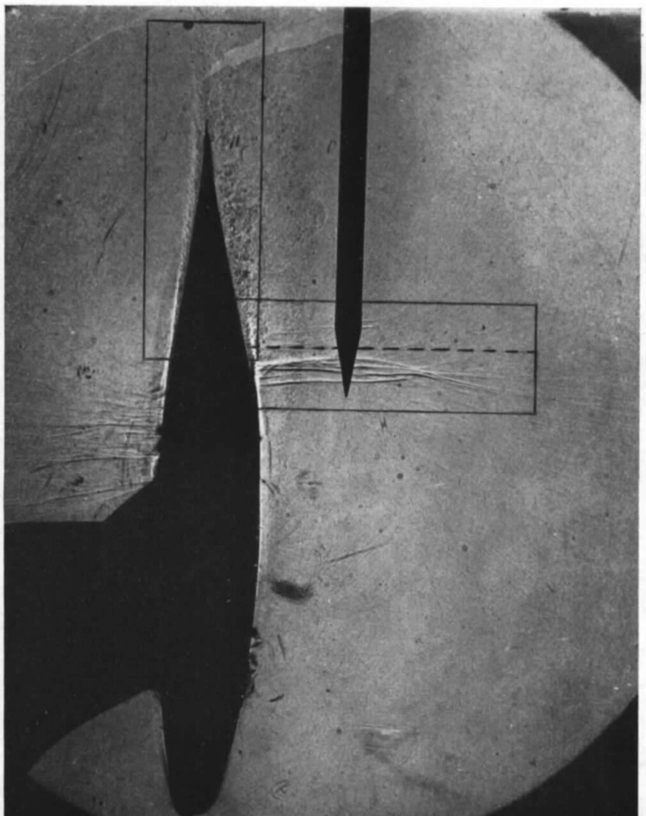

FIG. 8. Goldstein roof-top section 1442/1547 showing approximate areas of observation and position of reference line.

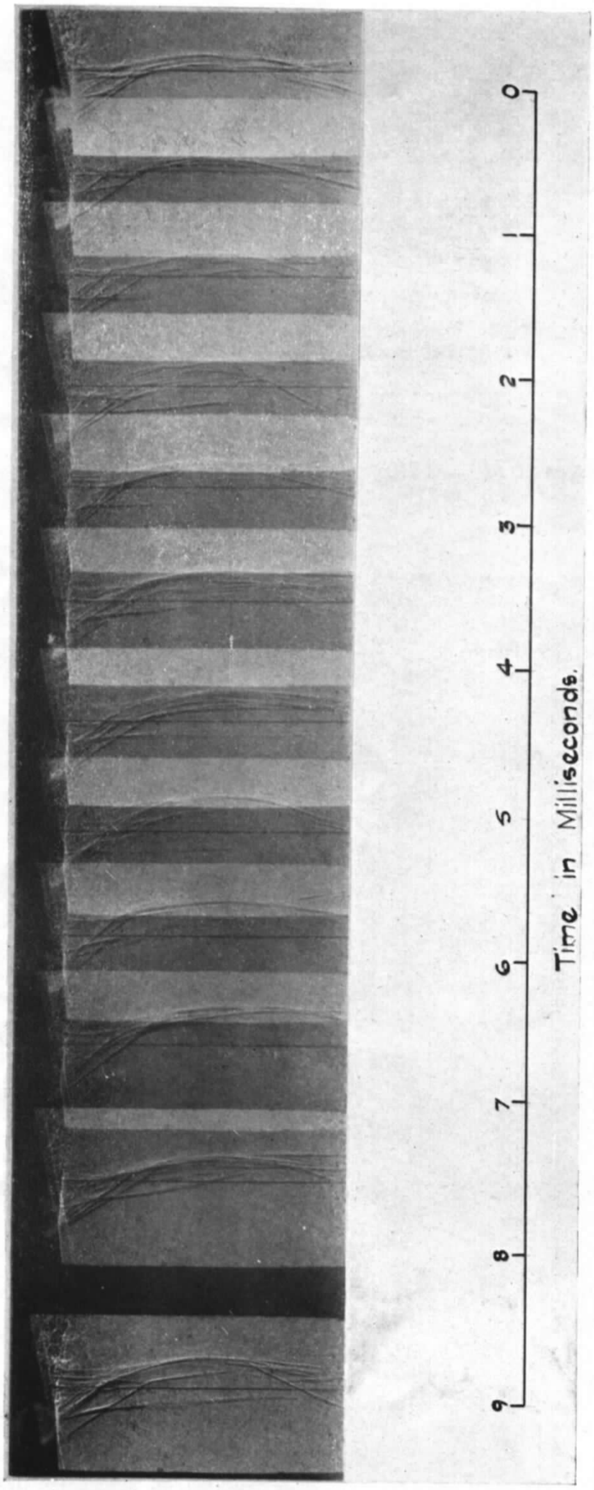

FIG. 9. Set 1. Shock wave movement at $M = 0{\cdot}78$.

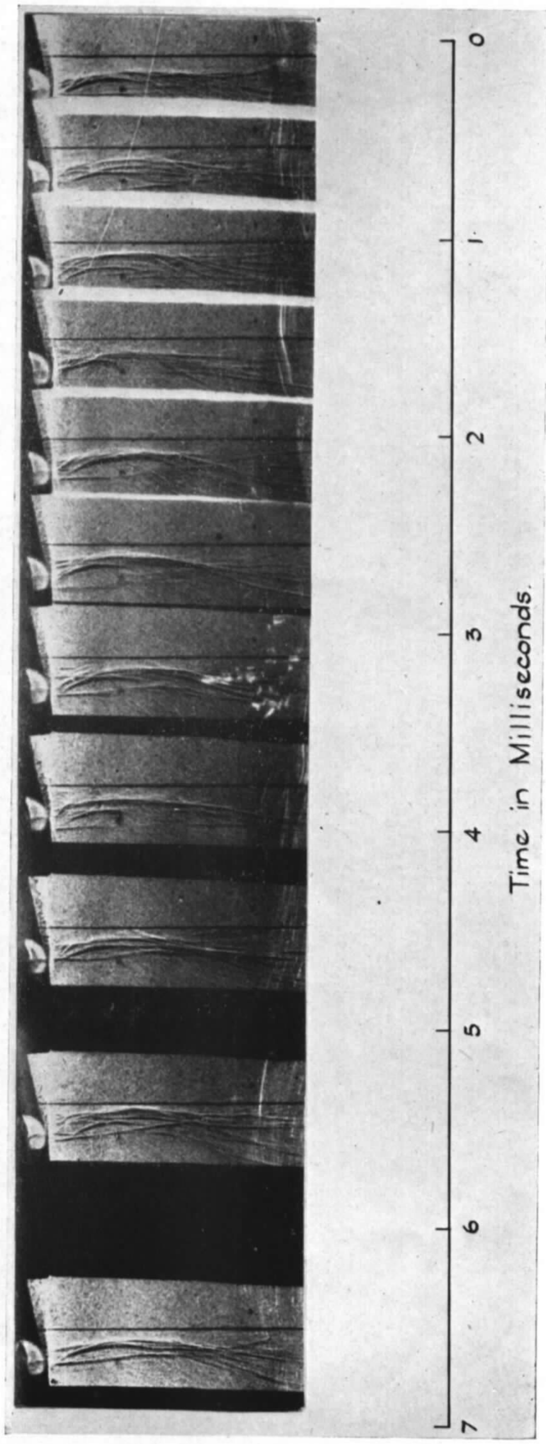

FIG. 10. Set 2. Shock wave movement at $M = 0{\cdot}78$.

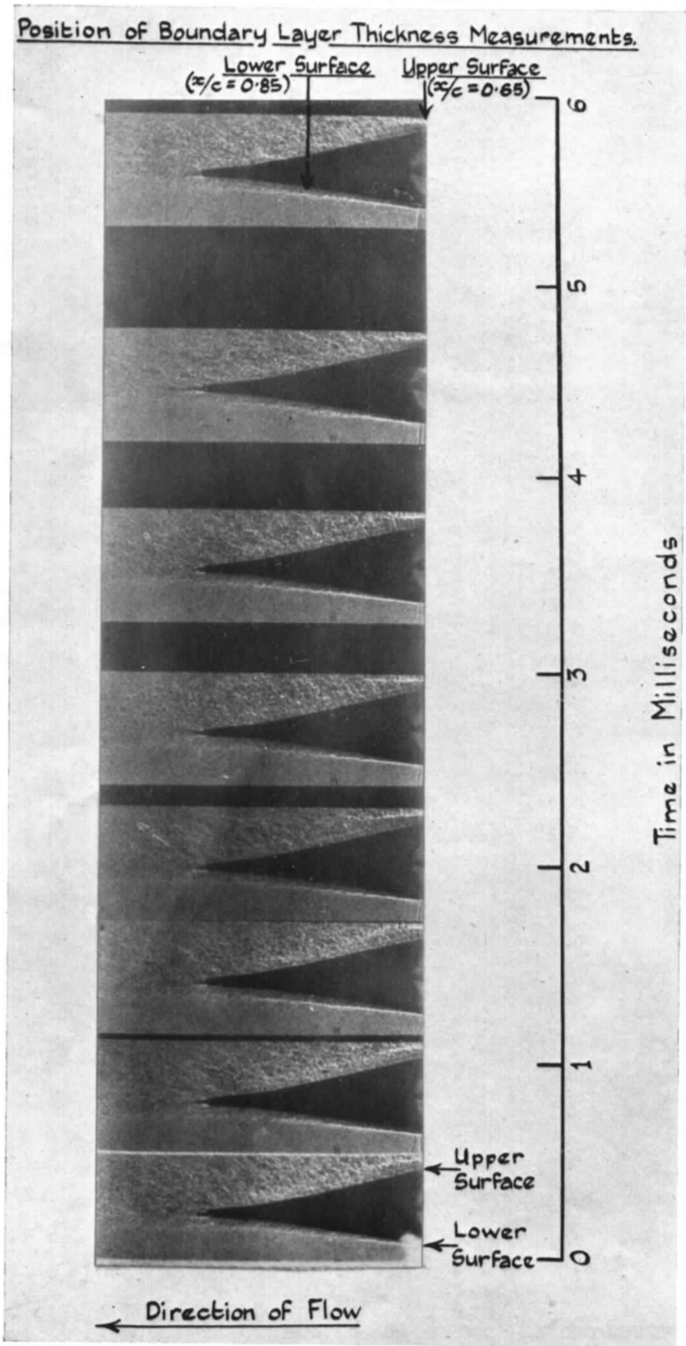

FIG. 11. Separation at $M = 0{\cdot}81$.

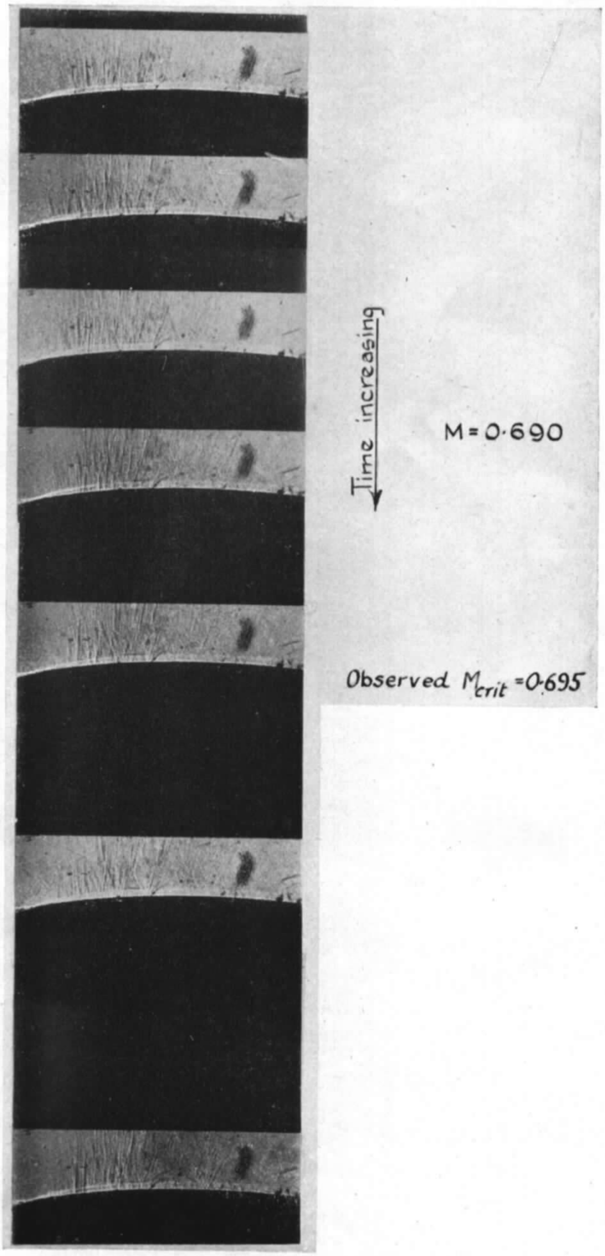

FIG. 12. Wavelets on upper surface below the observed pressure critical Mach number.

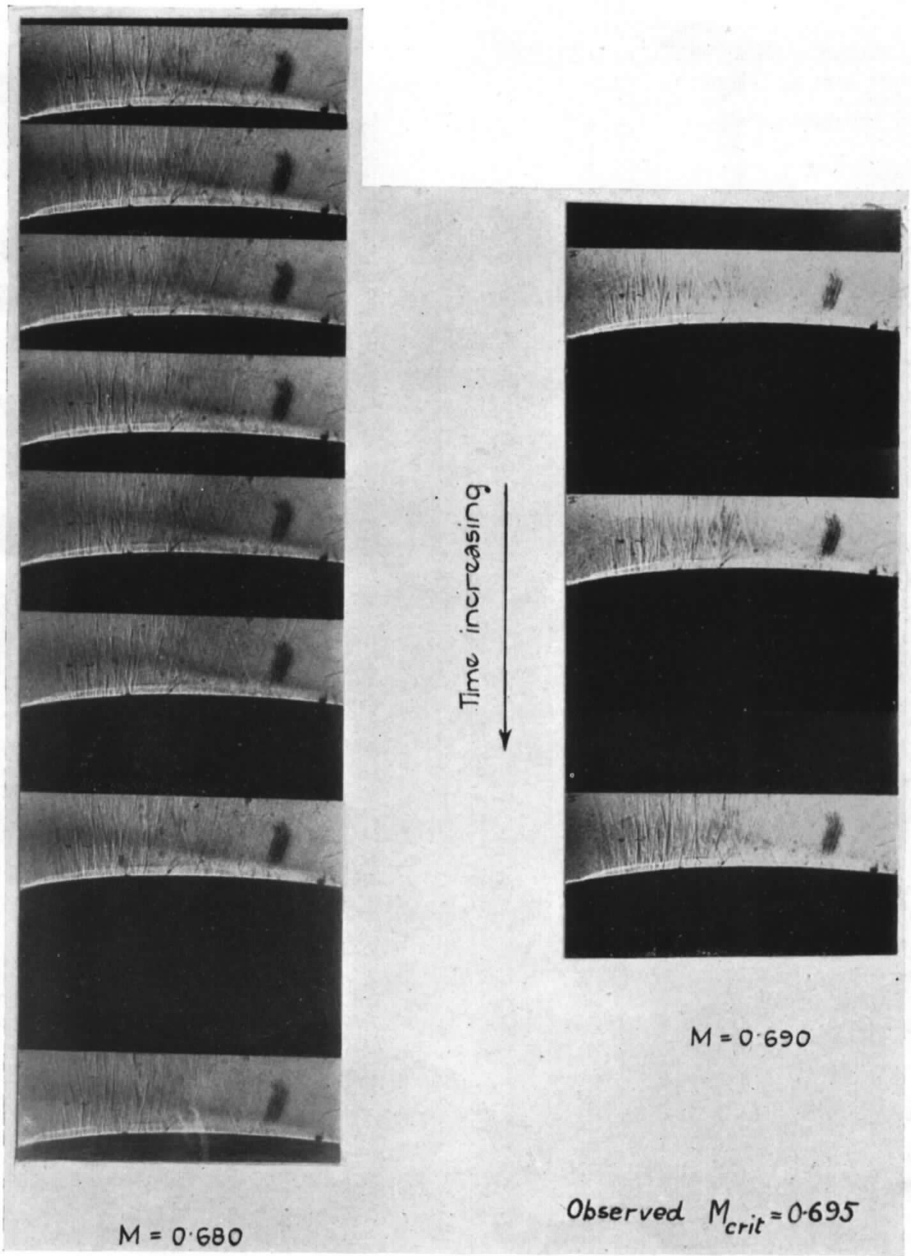

FIG. 13. FIG. 14.
Wavelets on upper surface below the observed pressure critical Mach number.

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