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A Preliminary Experimental Investigation of Shock-wave Development on Aerofoils

By

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A Preliminary Experimental Investigation
of Shock-Wave Development on Aerofoils

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T. H. Moulden, Miss I. J. Cox, Miss V. A. Stringfellow

SUMMARY

The transition process from a pure subsonic type flow to a mixed flow with a steady terminating shock wave was investigated by means of simple experimental methods. Attention was restricted to the flow past a two-dimensional aerofoil for Mach numbers at, and just above, the critical.

The experiment demonstrates the way in which upstream-travelling disturbances slow down and ultimately coalesce at some point in the adverse pressure gradient. The velocity components of these disturbances along the chord line was measured for a range of free stream Mach number.

It was concluded that, while the present experiment had shed some light on the mechanism of shock wave development, a more refined experimental system was required to give a fuller understanding of the phenomena involved.

List of Contents

Notation/

- -

*Replaces NPL Aero Note 1042 - A.R.C.27 668

Notation

M	Mach number
a	sound speed
a^*	critical sound speed
U	wave speed †
P	static pressure
H	total head pressure
x, y	aerofoil co-ordinates
c	aerofoil chord length
θ	aerofoil surface slope (measured relative to chord line)
ω	Prandtl-Meyer angle

Suffices

∞	free stream conditions
l	local conditions

† For a discussion of the significance of this quantity see Section 3.2.

1. Introduction

The theoretical model usually adopted for investigation into the transonic flow past a two-dimensional aerofoil assumes steady, irrotational flow. The results obtained from this theory are not completely in agreement with experimental findings. This is particularly true in the Mach number range just above the critical condition where the steady terminating shock wave is in the process of formation. At higher Mach numbers, provided that the flow is not separated and a representative pressure rise is assumed at the shock wave (see Ref. 7), theory gives a reliable indication of the pressure distribution on the aerofoil surface.

From the practical point of view there is considerable interest in the possible achievement of isentropic compression from a supersonic local Mach number. To obtain reliable information in this Mach number range would demand a more realistic model than the steady, potential flow one usually employed and whose properties are reviewed in Ref. 8.

Two possibilities have been suggested for improving the flow model. Firstly, the presence of a shock wave in the flow renders it non-irrotational and some account should be taken of this in the theory. Secondly, the possibility of unsteadiness in the flow should be considered.

We suggest that the second of these should take precedence for Mach numbers just above the critical. This statement is defended as follows. It is assumed that the oncoming flow at infinity is wholly irrotational; then any rotation is due solely to the presence of a shock wave. By implication the shock wave formation and the breakdown of irrotational flow occurs simultaneously, and needs provoking. Admittedly, as soon as the shock has formed, the flow could be modified due to the presence of rotation. Hence the breakdown of the irrotationality condition is looked upon as a modifying factor rather than the cause of shock formation.

The main purpose of the present experiment was to make a preliminary investigation into the time-dependent nature of the flow in order to assist in formulating a theoretical model. In broad outline the experiment is similar to that of Tamaki - Ref. 2 - but makes some improvements in experimental conditions.

2. Experimental Details

2.1 Wind tunnel and model

The tests were carried out in the NPL 20" x 8" (50·80 cm x 20·32 cm) high speed wind tunnel using aerofoil NPL 1221. The tunnel is described in detail in Refs. 3. The working section was fitted with slotted liners of such geometry (1/70th open area) as to give negligible tunnel interference (see: Ref. 9). The model - which was symmetrical and of five inch (12·70 cm) chord and eight inch (20·32 cm) span - was mounted in glass windows so that the flow could be investigated by optical methods.

The model geometry (ordinates and surface slopes) is given in Table 1. The maximum thickness (11·78 %) was at 36% chord. A number of pressure tappings were incorporated on the surface of the model and the positions of these are given in Table 2. All tests were carried out with the model at zero lift.

2.2/

2.2 The optical system

A normal schlieren optical system was employed and is shown diagramatically in Fig. 1. A spot cut-off was used to give uniform definition in all directions. Two photographic recording systems were available. Static photographs were taken by focussing the image on to a plate, while time variations were recorded on a moving film using an oscilloscope camera (Cossor Model 1428, Mark II). The whole field was photographed for the static case but for the time variations only a narrow slit just above the surface of the aerofoil was filmed.

2.3 Experimental conditions

At the Mach number of the tests (of the order of 0•78) the Reynolds number - based on aerofoil chord - was just below 2×10^6. Natural transition occured on the model in the region where the shock wave forms, that is, at about 40% chord. To eliminate the premature interaction effects that tend to occur between the shock and a laminar boundary layer, transition was fixed by means of a band of caborundum (grains: 0•0019 inch (0•0048 cm) diameter) sprayed on each surface from 0 - 5% of the chord. In this way transition was promoted before 10% chord.

The experiment consisted of taking pressure distributions and photographs at a series of closely spaced Mach numbers near the critical value. Both spark (exposure of order 1 micro second) and time exposure (exposure of order 1 second) schlieren photographs were taken.

3. Discussion

The static and dynamic parts of the experiment are best discussed separately.

3.1 Attempted visualization of the characteristics

The static half of the experiment arose from an attempt to make visible the characteristic pattern in the supersonic flow region. Satisfactory visualization of the characteristics would throw light on the part played in the formation of a shock wave by the weak disturbances propagated along Mach waves within the supersonic region.

Ideally, to visualize the characteristics, it is necessary to generate an infinitesimal wave at one point on the aerofoil surface. This cannot be achieved in practice. The best that can be done is to create a system of expansion and compression waves from a finite disturbance.

The best results obtained in the present instance were from a 'line' of carborundum grains (0•0016 inches (0•0041 cm) diameter) placed along the span at 30% chord. The photographs presented in Fig. 2 show that the system was only partially successful. Obviously such a system is of little use in investigating any influence that the Mach waves may have in shock wave formation.

One point of interest, however, can be seen from Fig. 2, where the disturbance 'reflected' from the sonic line is more clearly visible on the spark than on the time-exposure photographs. If the flow were steady, a reflection of the disturbance from the sonic line would return to the surface and be visible

on/

on both spark and time exposure photographs. If, on the other hand, the flow is unsteady (forward-moving waves present as demonstrated in Section 3.2) then the position of the reflection would change as such waves pass through the flow because of the addition of a steady and a non-steady perturbation. This is also noticeable for the case shown in Fig. 2(b), where three photographs for a free-stream Mach number of 0•785 reveal that the reflected disturbance reaches the surface at different points.

An attempt was made to measure the connected points (supposing that such a concept is valid in a time dependent flow) as observed in the photographs of Fig. 2, and to compare them with the values obtained from the pressure distributions. Pearcey, in Ref. 4 showed that the connected point could be found from the pressure distribution by drawing the horizontal line between the $\theta - \omega$ and $\theta + \omega$ curves - as shown in Fig. 3(b). The comparison between the two connected-point measurements is shown in Fig. 4, where expected experimental inaccuracies are indicated. In considering Fig. 4, it should be remembered that the wave-visualization technique had produced finite waves rather than Mach waves and hence complete agreement with the pressure distributions should not be expected.

Finally, we remark that the added disturbance did not tend to magnify on propagating downstream - Fig. 2 - but rather to die away. However, the disturbance was added in the region of the favourable pressure gradient and so does not strictly compare with Busemann's conjectures in Ref. 6.

3.2 Determination of wave movement

Fig. 2(a) revealed the presence of moving waves on the surface of the aerofoil - as first observed by Hilton et al (Ref. 5). A preliminary experiment was set up to investigate these waves as follows:

The oscilloscope camera was included in the optical system and a slit 0•02 inches (0•5 mm) wide was left in a screen placed in front of the camera. The slit was parallel to the image of the aerofoil chord (here 1•10 inches (28 mm) long) and was 0•12 inches (3 mm) above the chord line. The film used was Ilford HPS 35 mm, and was traversed at a speed of 25 inches/sec (63•50 cm/sec). In this way any movement of the waves on the aerofoil surface would give rise to a trace on the film. Typical traces obtained are shown in Fig. 6, while in Fig. 5 the traces for selected free-stream Mach numbers are printed on the same scale as the spark schlieren photograph. The three cases shown in Fig. 5 are typical stages in the development of the terminating shock wave. The film movement was in the upward direction in Figs. 5 and 6, so that a positive slope on the traces indicates forward-travelling waves on the aerofoil (leading edge on the left). The film speed and the slope of the trace yield the wave velocity relative to the aerofoil. These are shown on Fig. 7 for a range of free-stream Mach numbers.

The vertical white line on the left-hand side of the film in Fig. 6 is the result of the stationary disturbance on the aerofoil surface discussed in Section 3.1 - as can be clearly seen in Fig. 5. This line is of no interest in the present investigation.

The traces shown in Fig. 6 indicate that the frequency of the waves is of the order of 1000 c/s and does not change appreciably with free-stream Mach number. This is at variance with the result of Tamaki (Ref. 2) where a rapid

decrease/

decrease in frequency was observed as the free-stream Mach number was increased. If these waves do in fact originate from disturbances in the wake it is possible that the frequency would be Reynolds number sensitive (the present Reynolds number being of the order of ten times as large as that of Ref. 2).

Much of the detail is lacking from the traces in Fig. 6 due to imperfections in the experimental set-up. However, it is possible to interpret these traces to give the following mechanism of shock-wave development.

Due to disturbances downstream of the aerofoil (the origin of which is as yet uncertain), pressure waves propagate forward over the aerofoil surface. Over the rear of the aerofoil these waves will steepen on moving into a region of lower pressure, while over the front of the aerofoil, on moving into a region of higher pressure, they will decay (see Ref. 10). The degree of steepening of the wave front depends on the magnitude of the pressure gradient (i.e., on free-stream Mach number) and hence, in general, the waves are only visible on schlieren photographs when the free-stream Mach number approaches the critical value. This is the state shown in Fig. 6(a) for $M_\infty = 0{\cdot}763$.

The situation changes little with increase of free-stream Mach number except that the steepening effect of the wave profile becomes more severe. The wave velocity 'U' measured was the component of the wave speed along the chord line relative to a fixed point (since the slit was parallel to the aerofoil chord). Insufficient data was obtained for the determination of the true wave speed. Not until the free-stream Mach number reaches 0•783 - Fig. 6(d) - do the forward-moving waves first tend to come to rest. This they then only do spasmodically. A steady shock wave develops at a free-stream Mach number of 0•788 - Fig. 6(e) and Fig. 5(c) - and the upstream moving waves merge with the shock. The hypothesis is thus suggested that the position of the shock, and its strength, is fixed to satisfy the downstream compatibility condition through the agency of the upstream moving waves.

The above outlines the mechanism of shock formation suggested by the wave traces and represents a suitable model for further investigation. Thus the actual conditions to be fulfilled at the point where the moving waves first become stationary (at least instantaneously so) are not fully understood. It is not clear whether the waves steepen to the extent of being shock waves before coming to rest relative to the aerofoil, or after. Neither is the effect of waves passing round the edge of the local supersonic region known (Tamaki suggests that these waves cause the initial instability of the 'steady' shock). Further investigation is clearly needed.

4. Conclusions

The experiment described above reveals some of the nature of a transonic aerofoil flow. The findings are in agreement with those of Tamaki in Ref. 2. In particular it is shown that the terminating shock wave develops from the coalescence of waves moving upstream.

The experiment is considered to be qualitative rather than quantitative. Thus, insight was gained into the mechanism of the flow which can be used to help formulate a more realistic flow model. Certain limitations need to be rectified, however, before the apparatus can give reliable quantitative data. Among these limitations we can mention the need to measure precisely the tunnel speed at a particular instant and also the need to know, exactly, the speed of the film through the camera. Such improvements are being made.

Acknowledgement /

Acknowledgement

The authors are indebted to Mr. H. H. Pearcey for helpful criticism of the paper.

References

No.	Author(s)	Title, etc.
1	W. R. Sears (Ed.)	General theory of high speed aerodynamics. Section F, High Speed Aerodynamics and Jet Propulsion. Vol. VI. Oxford University Press, 1955.
2	F. Tamaki	Experimental Studies on the Stability of the Transonic Flow past Airfoils. IXth Congrès International de Mécanique Appliquée. Bruxelles, 1957.
3	D. W. Holder	The high speed Laboratory of the Aerodynamics Division, NPL. A.R.C. R. & M. 2560, 1947.
	H. H. Pearcey and J. D. Regan	The Mach number distribution along the slotted walls of the NPL 20" x 8" high speed wind tunnel. A.R.C. C.P.784, July, 1958.
4	H. H. Pearcey	The aerodynamic design of section shapes for swept wings. Vol. 3-4, Advances in Aeronautical Sciences, Pergamon Press, 1961. Proceedings of the Second International Congress in the Aeronautical Sciences Zürich 12-16 September, 1960.
5	W. F. Hilton and R. G. Fowler	Photographs of shock wave movement. A.R.C. R. & M. 2692, December, 1947.
6	A. Busemann	The drag problem at high subsonic speeds. J. Aero. Sci., Vol.16, No. 6, pp.336 - 344. 1949.
7	C. S. Sinnott and J. Osborne	Review and extension of transonic aerofoil theory. A.R.C. R. & M.3156, October, 1958.
8	T. H. Moulden	Some comments on the conditions in a local supersonic flow region. A.R.C.27 322. October, 1965.

9/

No.	Author(s)	Title, etc.
9	H. H. Pearcey, C. S. Sinnott and J. Osborne	Some effects of wind-tunnel interference observed in tests on two-dimensional aerofoils at high subsonic and transonic speeds. Agard Report 296, March, 1959.
10	Y.-H. Kuo	On the stability of two-dimensional smooth transonic flows. J. Aero. Sci., Vol. 18, pp.1-6 and 54, 1951.

Table I/

Table I

Geometry of the NPL 1221 symmetrical aerofoil

x/c	y/c	$\tan\theta$
0·000000	0·000000	∞
0·002408	0·009518	1·998276
0·009607	0·018509	0·885790
0·021530	0·026350	0·502408
0·038061	0·032792	0·305264
0·059039	0·037874	0·192605
0·084265	0·041918	0·140332
0·113495	0·045677	0·117134
0·146447	0·049178	0·096357
0·182803	0·052318	0·076606
0·222215	0·054948	0·057309
0·264302	0·056969	0·039012
0·308658	0·058303	+ 0·021501
0·354858	0·058853	+ 0·001376
0·402455	0·058350	− 0·022810
0·450991	0·056660	− 0·046018
0·500000	0·053930	− 0·064525
0·549009	0·050409	− 0·078590
0·597545	0·046324	− 0·089215
0·645142	0·041869	− 0·097901
0·691342	0·037150	− 0·106526
0·735698	0·032233	− 0·115234
0·777785	0·027246	− 0·120657
0·817197	0·022464	− 0·121510
0·853553	0·018044	− 0·121510
0·886505	0·014040	− 0·121510
0·915735	0·010487	− 0·121510
0·940961	0·007421	− 0·121510
0·961940	0·004872	− 0·121510
0·978470	0·002863	− 0·121510
0·990393	0·001413	− 0·121510
0·997592	0·000538	− 0·121510
1·000000	0·000246	− 0·121510

Table II/

Table 2

Position of Pressure Holes

x/c	z/c	
0·0000	0·00000	UPPER SURFACE
0·0050	0·01410	
0·0100	0·01950	
0·0200	0·02560	
0·0500	0·03600	
0·1000	0·04410	
0·1600	0·05060	
0·2200	0·05470	
0·2800	0·05750	
0·3400	0·05890	
0·4000	0·05850	
0·4600	0·05630	
0·5200	0·05260	
0·5800	0·04790	
0·6400	0·04230	
0·7000	0·03620	
0·7600	0·02930	
0·8200	0·02200	
0·8800	0·01470	
0·9400	0·00750	
1·0000	0·00000	
0·0000	0·00000	LOWER SURFACE
0·0050	- 0·01410	
0·0100	- 0·01950	
0·0500	- 0·03600	
0·1000	- 0·04410	
0·1500	- 0·04960	
0·2000	- 0·05350	
0·3000	- 0·05820	
0·4000	- 0·05850	
0·5000	- 0·05390	
0·6000	- 0·04610	
0·7000	- 0·03620	
0·8000	- 0·02450	
0·9000	- 0·01220	
0·9600	- 0·00500	
1·0000	- 0·00000	

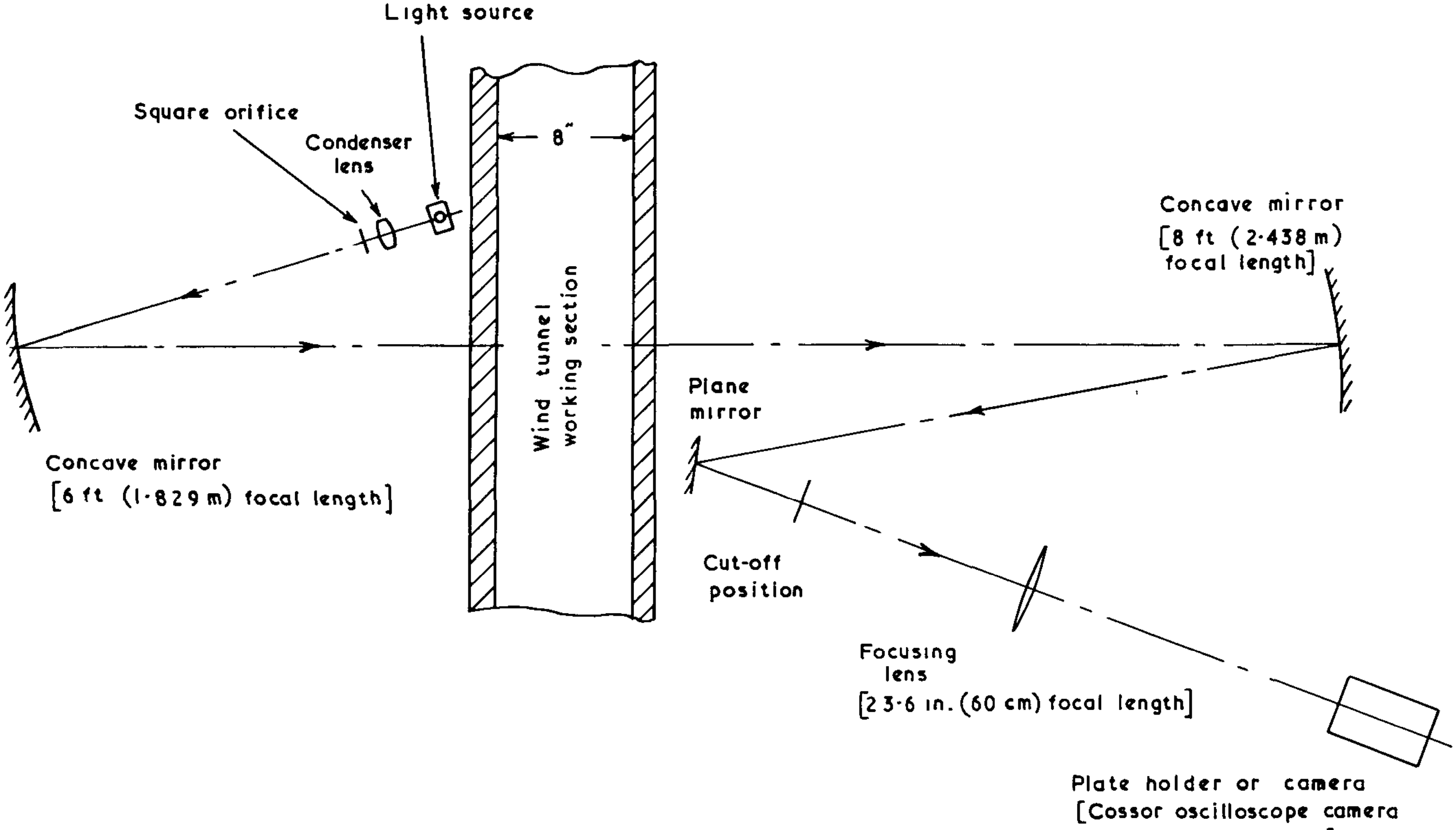
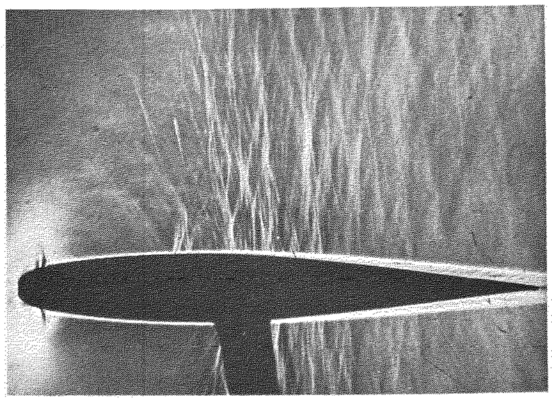


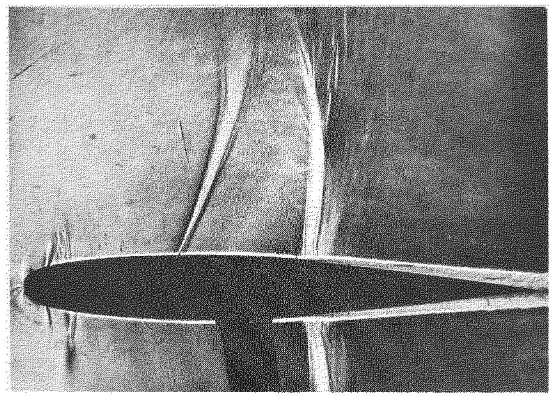
Diagram of optical system (plan)

FIG. 1

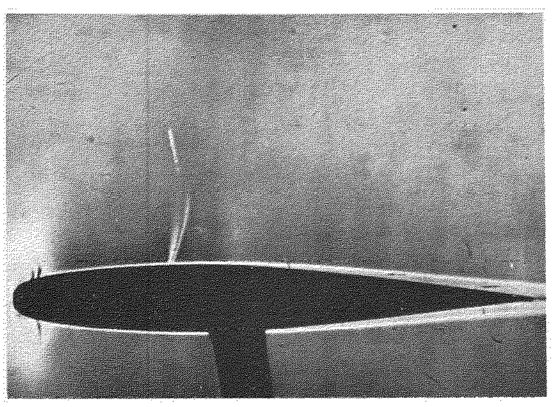
FIG. 2(a)



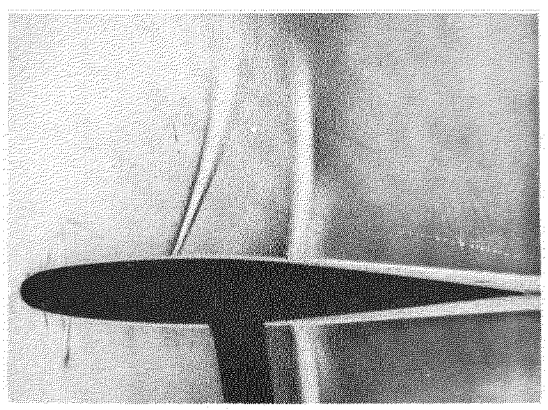
(i) $M_{\infty} = 0{\cdot}779$ (spark)



(ii) $M_{\infty} = 0\ 820$ (spark)



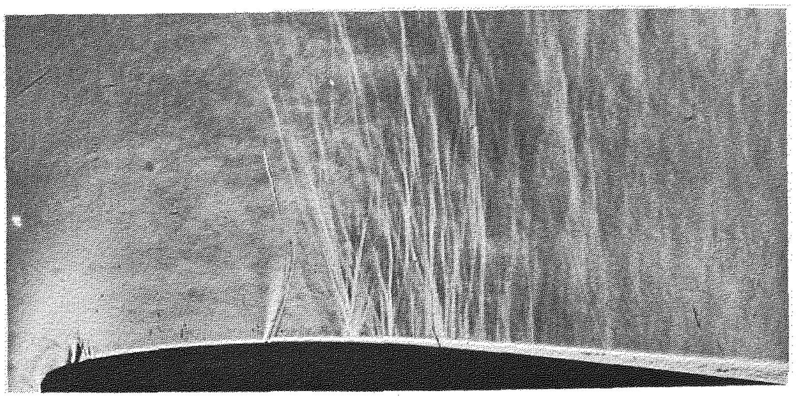
(iii) $M_{\infty} = 0\ 779$ (time exposure)



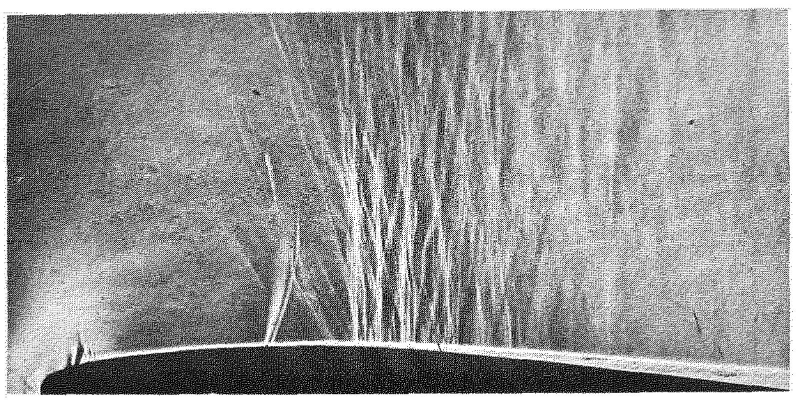
(iv) $M_{\infty} = 0{\cdot}820$ (time exposure)

Spark and time exposure photographs

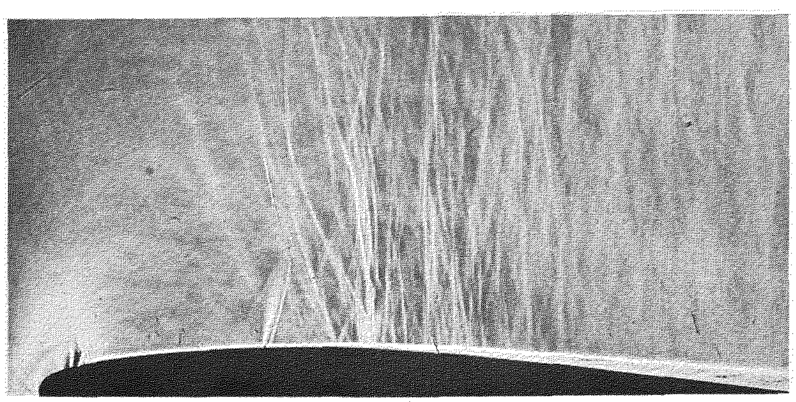
FIG. 2 (b)



(i)



(ii)



(iii)

Spark photographs at $M_\infty = 0{\cdot}785$

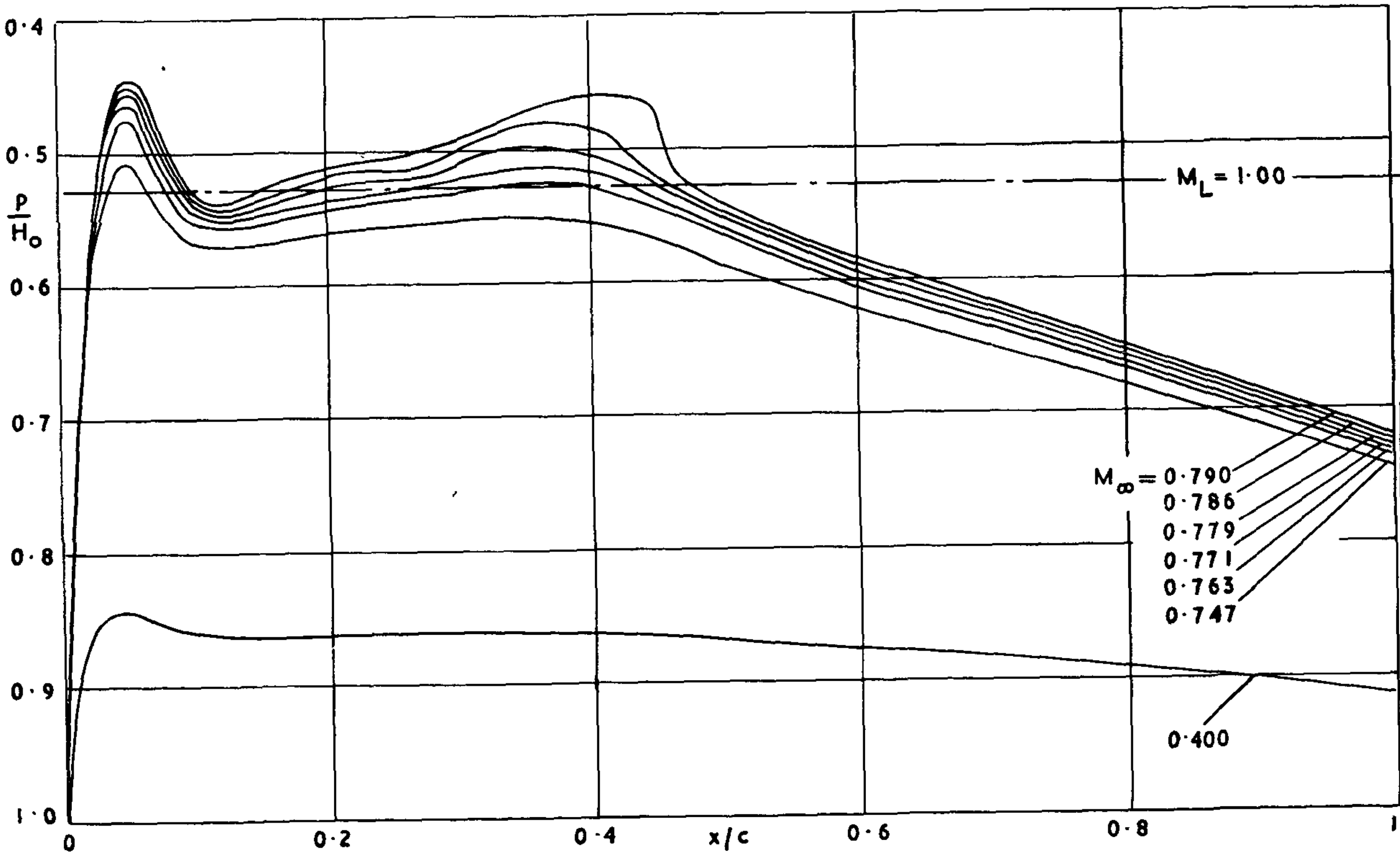
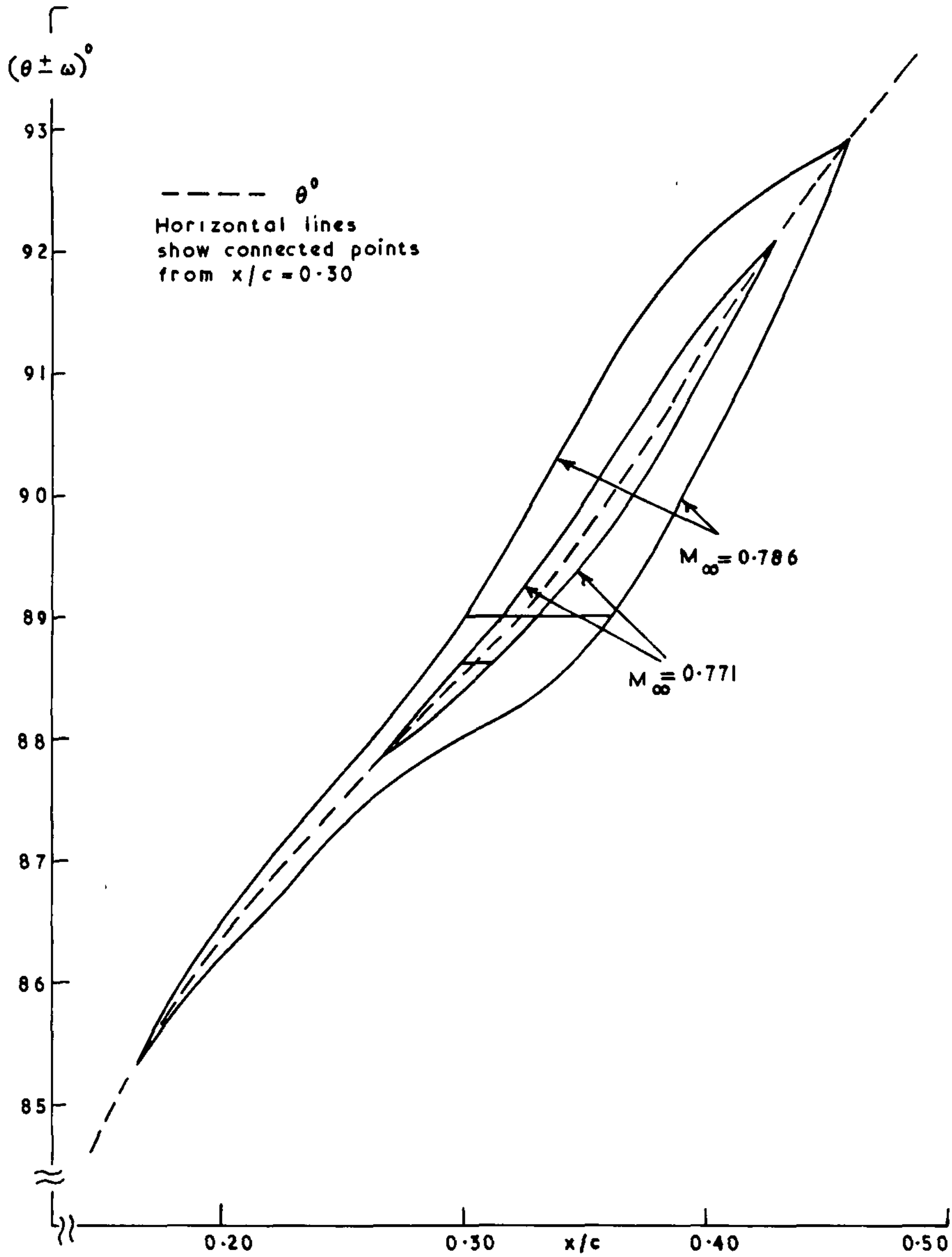


FIG. 3(a)

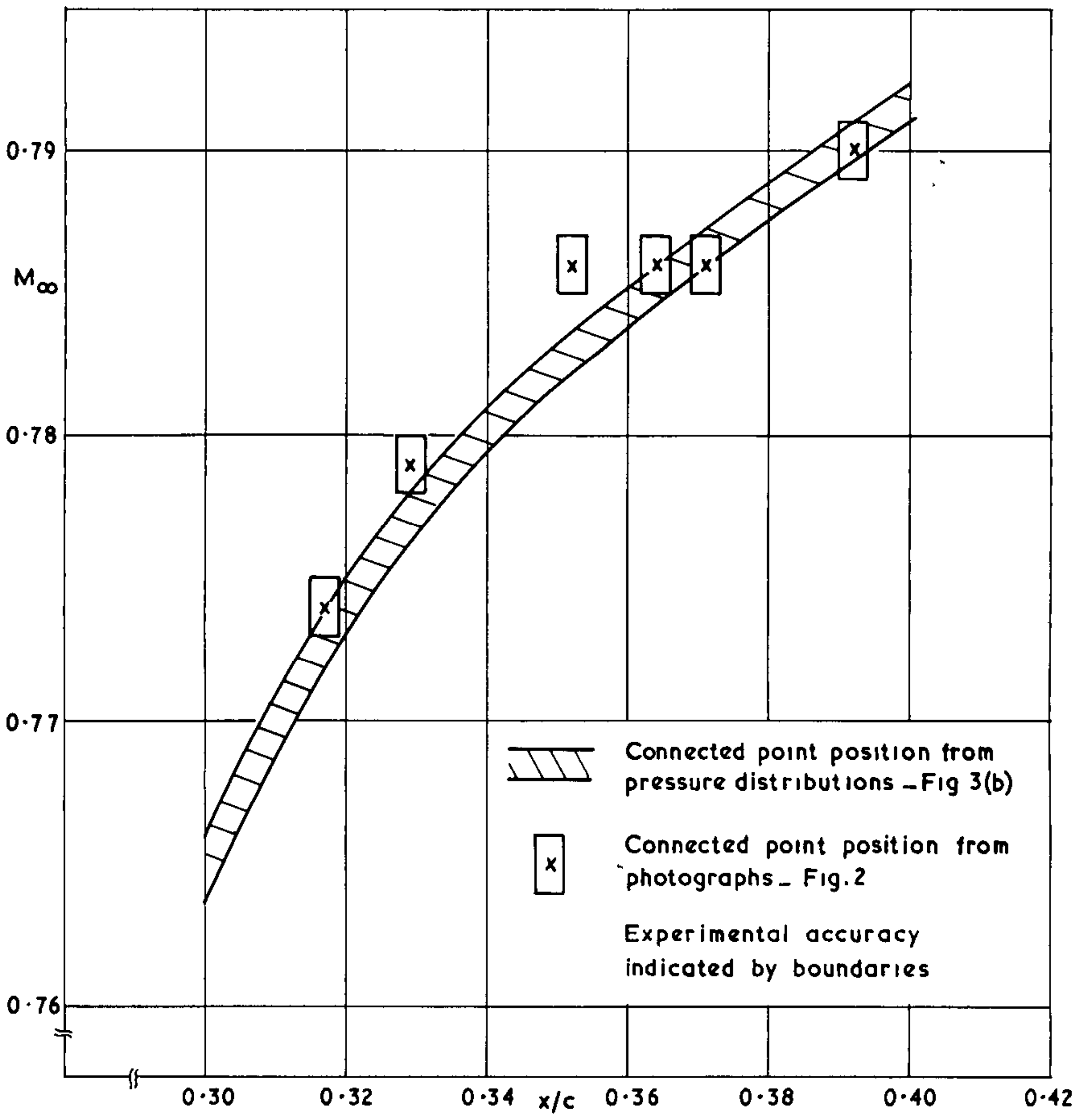
Upper surface pressure distributions

FIG. 3(b)



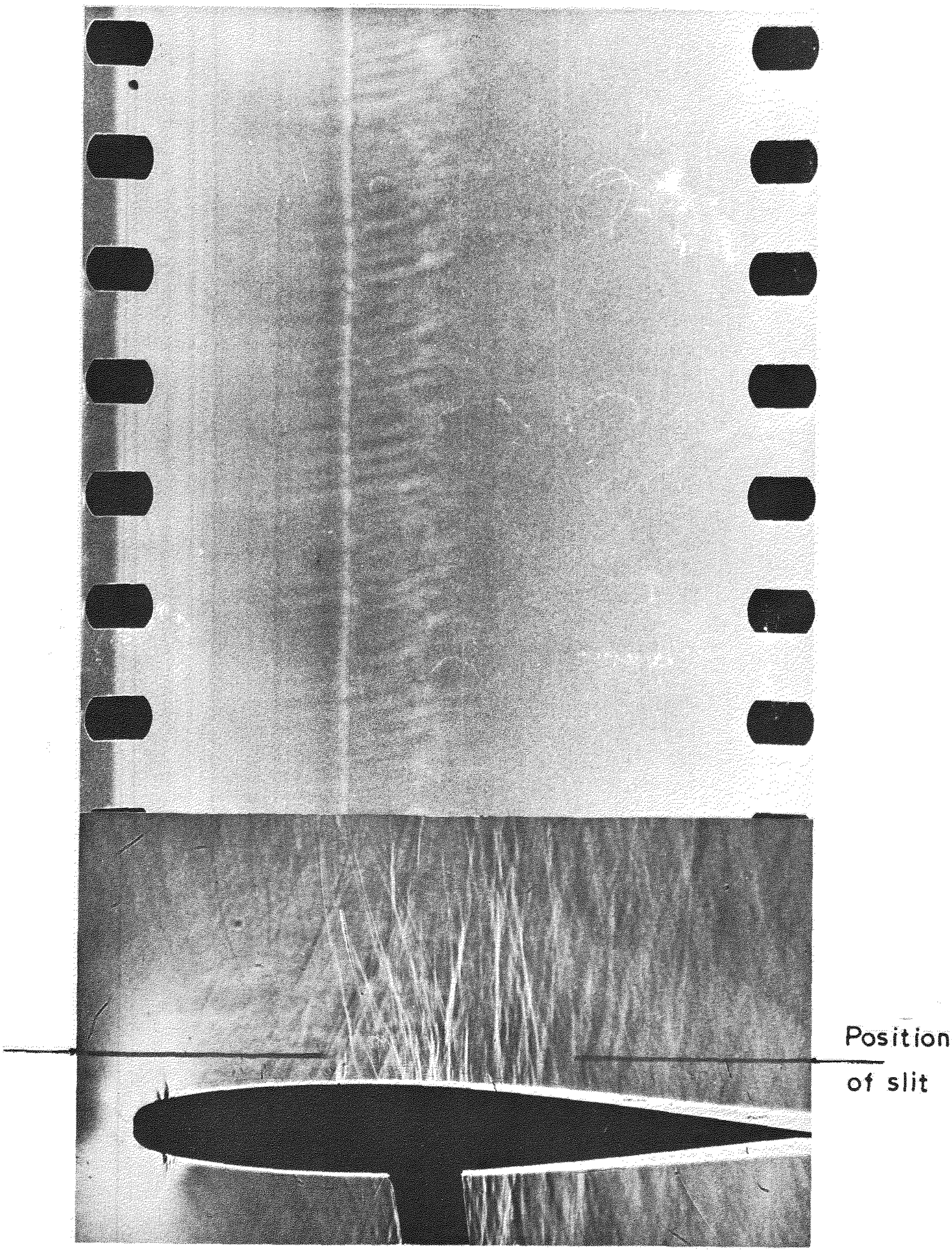
Typical curves of $\theta \pm \omega$

FIG. 4



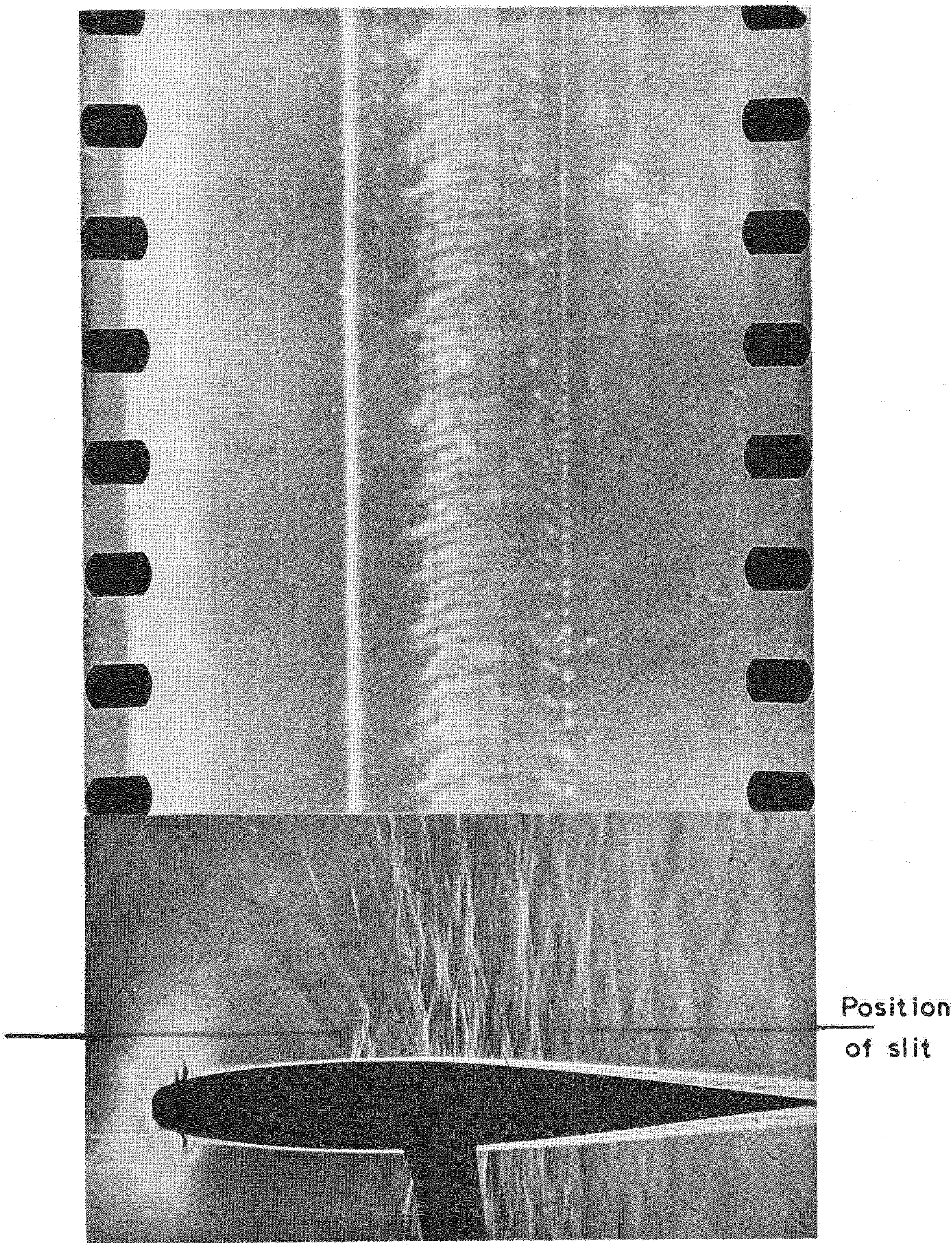
Points connected to the disturbance at x/c = 0·30 as measured by the two methods

FIG. 5(a)



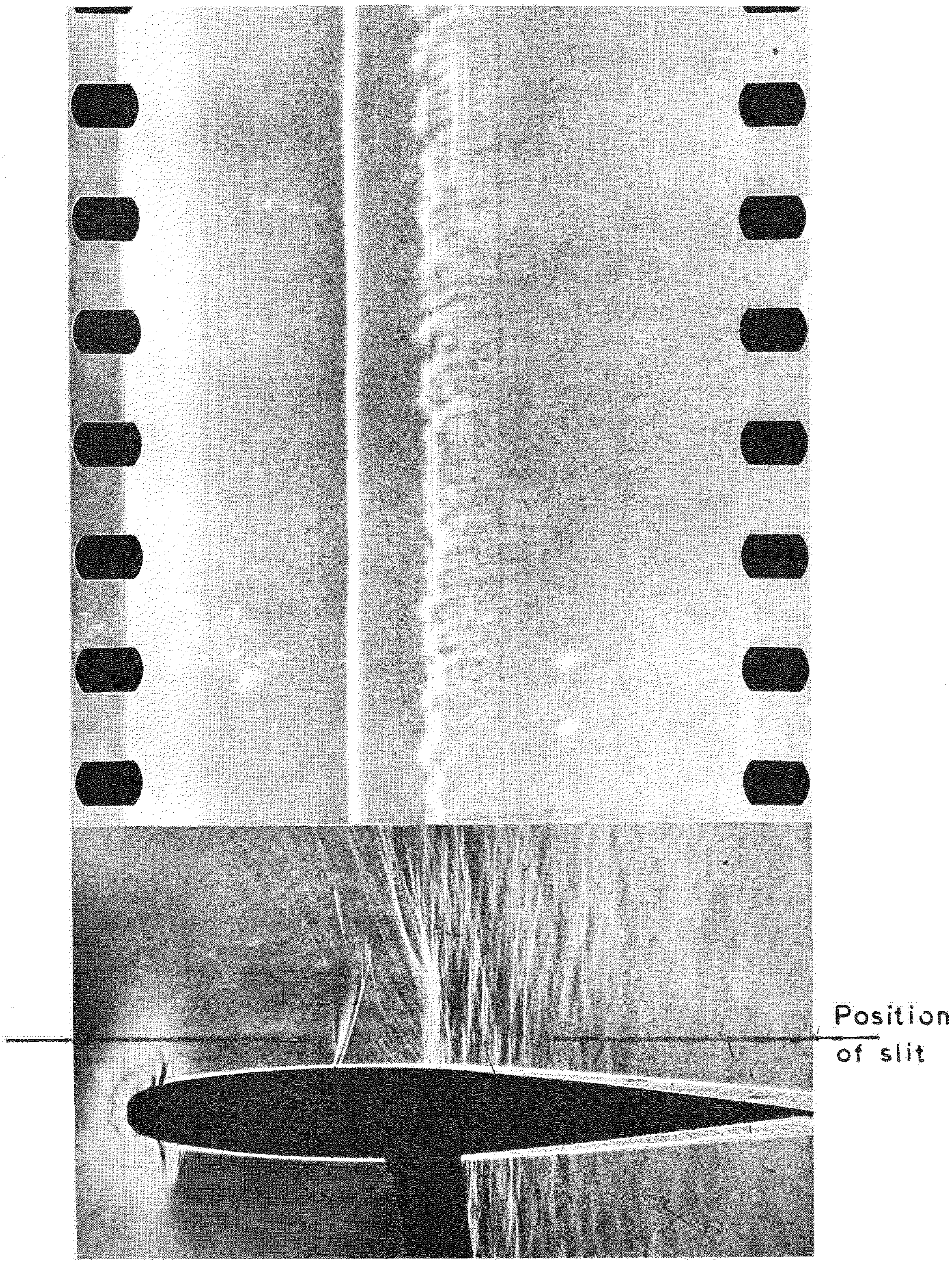
Spark schlieren photograph and wave traces at $M_{\infty} = 0{\cdot}764$. Film movement was upwards

FIG. 5 (b)



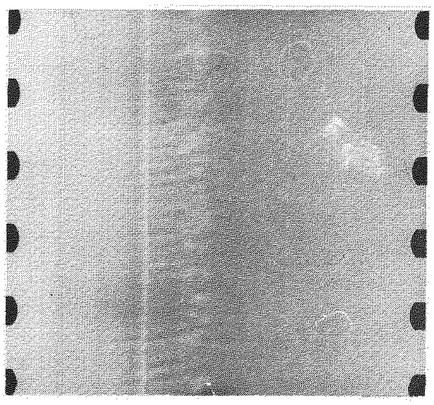
Conditions at $M_\infty = 0{\cdot}779$

FIG. 5 (c)

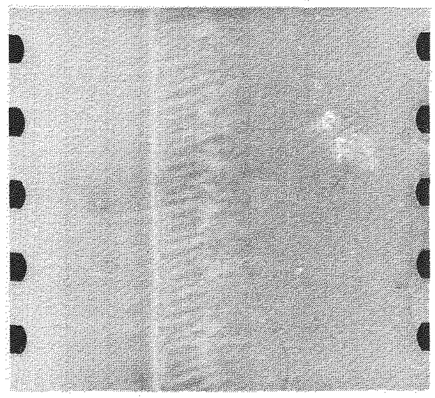


Conditions at $M_{\infty} = 0{\cdot}788$

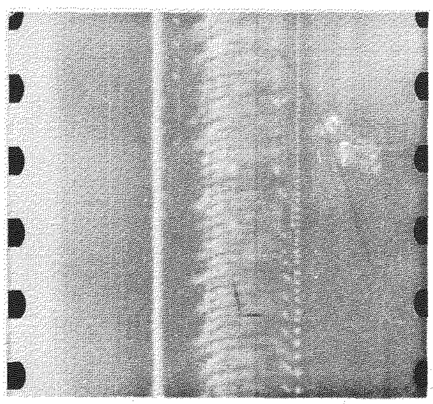
FIG. 6



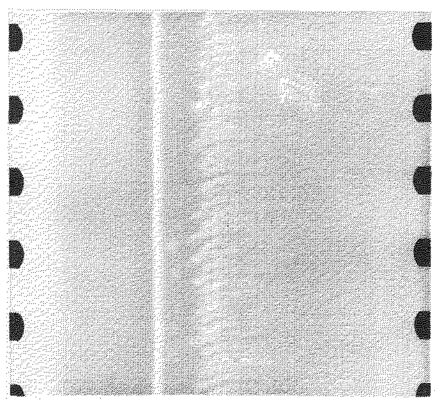
(a) $M_{\infty} = 0\ 764$



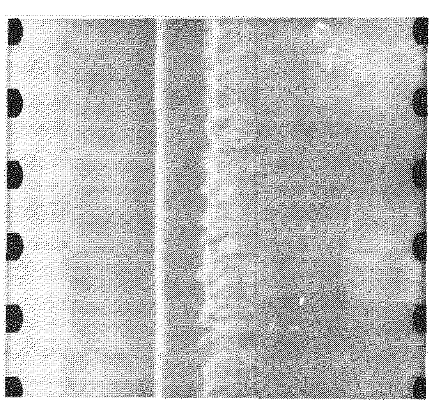
(b) $M_{\infty} = 0{\cdot}769$



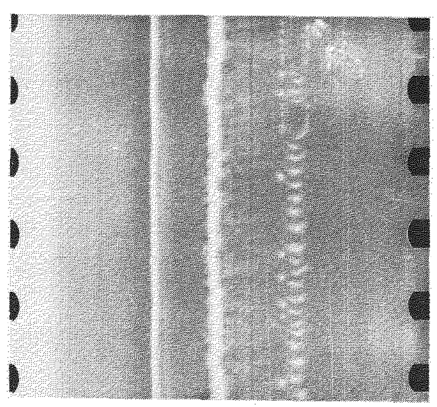
(c) $M_{\infty} = 0{\cdot}779$



(d) $M_{\infty} = 0{\cdot}783$



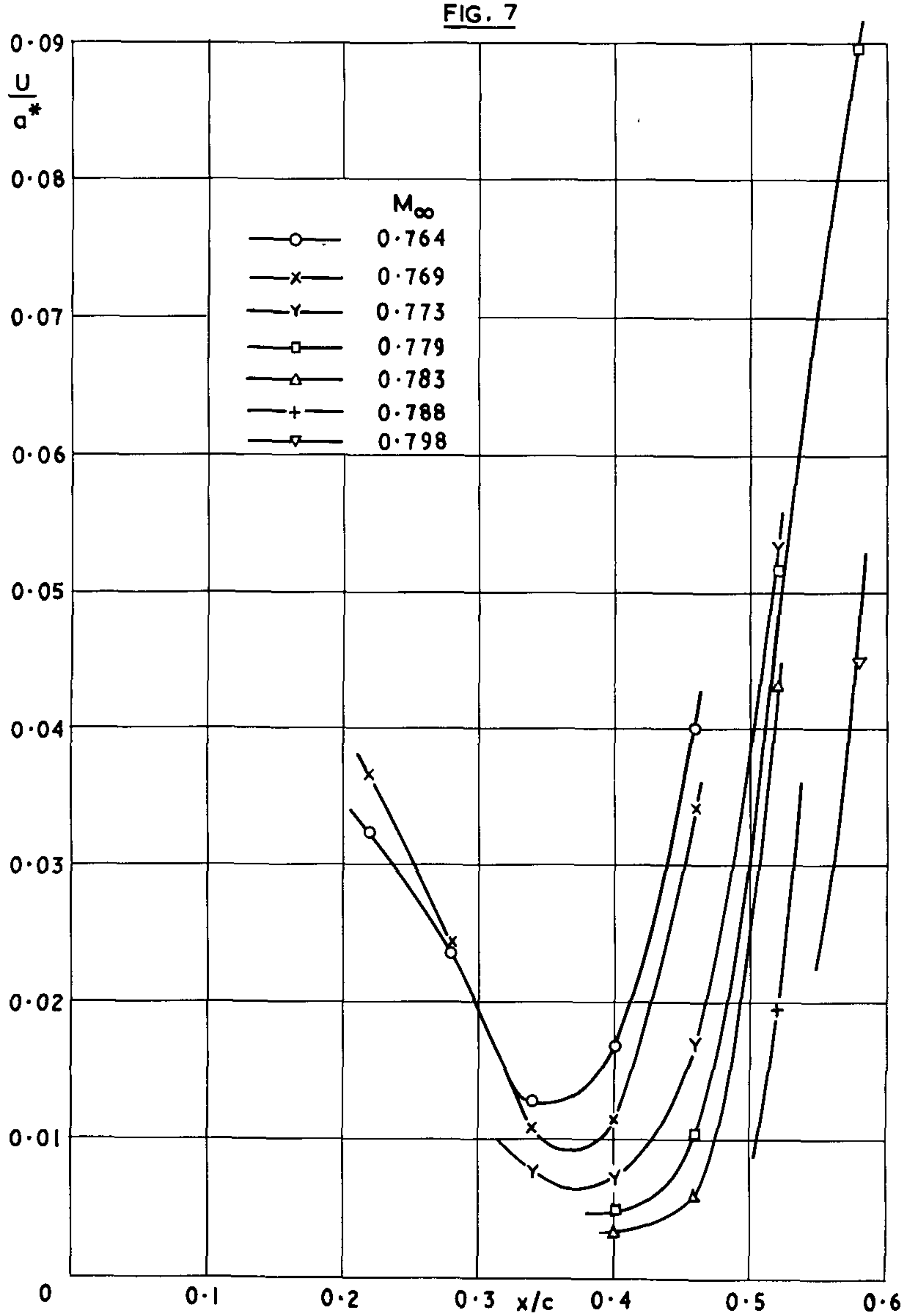
(e) $M_{\infty} = 0{\cdot}788$



(f) $M_{\infty} = 0{\cdot}798$

Wave traces. Film movement was upwards, aerofoil leading edge being on the left. (See also Fig. 5)

FIG. 7



Influence of free stream Mach number on the distribution of wave speed

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Moulden, T. H., Cox, Miss I. J., Stringfellow, Miss V. A.

A PRELIMINARY EXPERIMENTAL INVESTIGATION OF SHOCK-WAVE DEVELOPMENT ON AEROFOILS

The transition process from a pure subsonic type flow to a mixed flow with a steady terminating shock wave was investigated by means of simple experimental methods. Attention was restricted to the flow past a two-dimensional aerofoil for Mach numbers at, and just above, the critical.

The experiment demonstrates the way in which upstream-travelling disturbances slow down and ultimately coalesce at some point in the adverse pressure gradient. The velocity component of these disturbances along the chord line was measured for a range of free-stream Mach number.

It was concluded that, while the present experiment had shed some light or the mechanism of shock-wave development, a more refined experimental system was required to give a fuller understanding of the phenomena involved.

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