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Pressure Measurements in a Supersonic Tunnel on a Two - dimensional Aerofoil of R.A.E. 104 Section

By

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28th December, 1950

Summary

Measurements of the pressure distribution over the surface of a two-dimensional aerofoil of R.A.E.104 section have been made at a Reynolds number of about 0.8×10^6 and at free-stream Mach numbers of 1.42, 1.60 and 1.79. The observations included schlieren photography of the flow, and were made at angles of incidence of 0° and 2°.

1. Description of the Apparatus

The aerofoil section[1] is shown in Fig.1; it is symmetrical and has its maximum thickness equal to 10% of the chord located at 42% of the chord behind the leading edge. Ten 0.01" diameter pressure holes were drilled in the surface at the positions shown, and these were connected to a manometer by leads passing out of the tunnel through the tongues which were used to support the aerofoil in the glass discs fitted in the sides of the tunnel (Fig.2).

The tests were made in the 9" × 3" induced-flow tunnel. Schlieren photographs were taken with an exposure of about 1 microsecond and with an optical system based on two 9" diameter, 9' focal length spherical mirrors.

2. Presentation of Results

Two different coefficients are used to express the measured static pressure p in non-dimensional form. The first p/H_1 is based on the total head H_1 of the free stream and the second p/H_2 on the theoretical total head H_2 behind a normal shock wave with the measured free-stream Mach number and free-stream total head upstream. It seems reasonable to assume that for a blunt-nosed aerofoil the total head along a stream line just outside the boundary layer will approximate to H_2, and that the pressure coefficient p/H_2 may, therefore, be used to estimate the local Mach number on the assumption that H_2 would be the pressure reached if the flow were brought isentropically to rest.

3. Results

The pressure distributions measured on the aerofoil at 0° and 2° incidence are plotted in Figs.3-5 and Figs.6-8 respectively, and photographs of the flow are reproduced in Figs.9 and 10. The lift coefficients have been obtained from the pressure distributions by integration and are shown in Fig.11.

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4. Discussion

Curves calculated for simple-wave flow over the geometrical section (i.e. neglecting the growth of the boundary layer) are included in Figs.3-8. These have been drawn to give the measured pressure at the point where the surface is parallel to the free stream. This pressure is seen to be in reasonable agreement with the static pressure of the free stream. Except in the vicinity of the nose, and close to the tail on the upper surface, the calculated curves are in fair agreement with those measured.

Both measurements and calculations show that the rate of pressure fall begins to increase at between 0.5 and 0.6 of the chord from the leading edge. This is associated with a rapid change of surface slope at this position on the aerofoil (see Fig.12). The increased rate of expansion may be seen in the Toepler schlieren photograph reproduced in Fig.10(d) as an increase of illumination above and a decrease of illumination below the aerofoil. Similar phenomena have been observed[2] on this aerofoil at high subsonic speeds when the local supersonic region extends beyond the position where the slope is changing rapidly.

After the expansion described above, the calculated curves soon become horizontal and then represent the uniform flow along the straight rear part of the aerofoil (downstream of 0.8 chord in Fig.12). Apart from those measured on the lower surface when the aerofoil is at incidence (Figs.6-8), the measured pressures tend to rise towards the tail. This seems to be associated with a local thickening or separation of the boundary layer ahead of the shock wave at the tail, and is more fully discussed in ref.3. The boundary-layer separation is clearly visible on the upper surface in the photographs reproduced in Fig.10. According to the criterion suggested[4] by Pearcey, the photographs show that the boundary layer remains laminar back to at least the point at which separation takes place.

5. Conclusions

Except in the vicinity of the nose and the tail, the measured pressure distribution on a round-nosed aerofoil moving at a supersonic speed is found to be in fair agreement with that calculated for a simple wave giving the measured pressure at the point where the surface is parallel to the free stream. This pressure approximates to the static pressure of the free stream.

The discrepancy at the nose arises from the presence of the region of subsonic flow there, and decreases as the Mach number is raised. The discrepancy at the tail is due to a local separation or thickening of the boundary layer ahead of the tail shock wave, and is particularly pronounced on the upper surface when the aerofoil is at positive incidence.

Acknowledgement

The aerofoil was made by Mr. A. J. Hewson of the Aerodynamics Division, N.P.L. and Miss N. A. Bumstead helped with the photography and in the reduction of the observations.

References/

References

No.	Author(s)	Title, etc.
1	R. C. Pankhurst and H. B. Squire	Calculated pressure distributions for the R.A.E.100-104 aerofoil sections. R.A.E. Technical Note No. Aero.2039. (March, 1950). A.R.C. 13,254 -
2	E. W. E. Rogers, C. J. Berry and R. F. Cash	Tests at high subsonic speeds on a 10% thick pressure-plotting aerofoil of R.A.E.104 section. Part II - Pressure distributions and flow photographs. (In preparation).
3	D. W. Holder and 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
4	H. H. Pearcey	The indication of boundary-layer transition on aerofoils in the N.P.L. 20" × 8" high-speed tunnel. Current Paper No.10. December, 1948.

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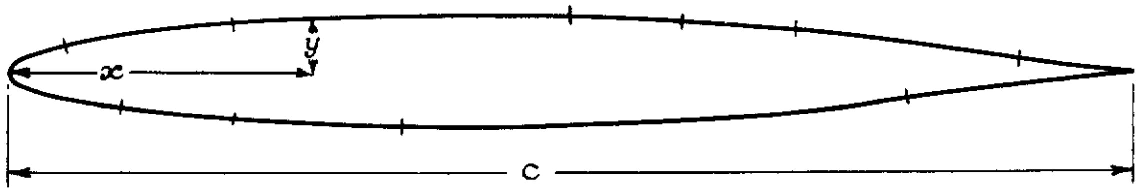
FIG 1

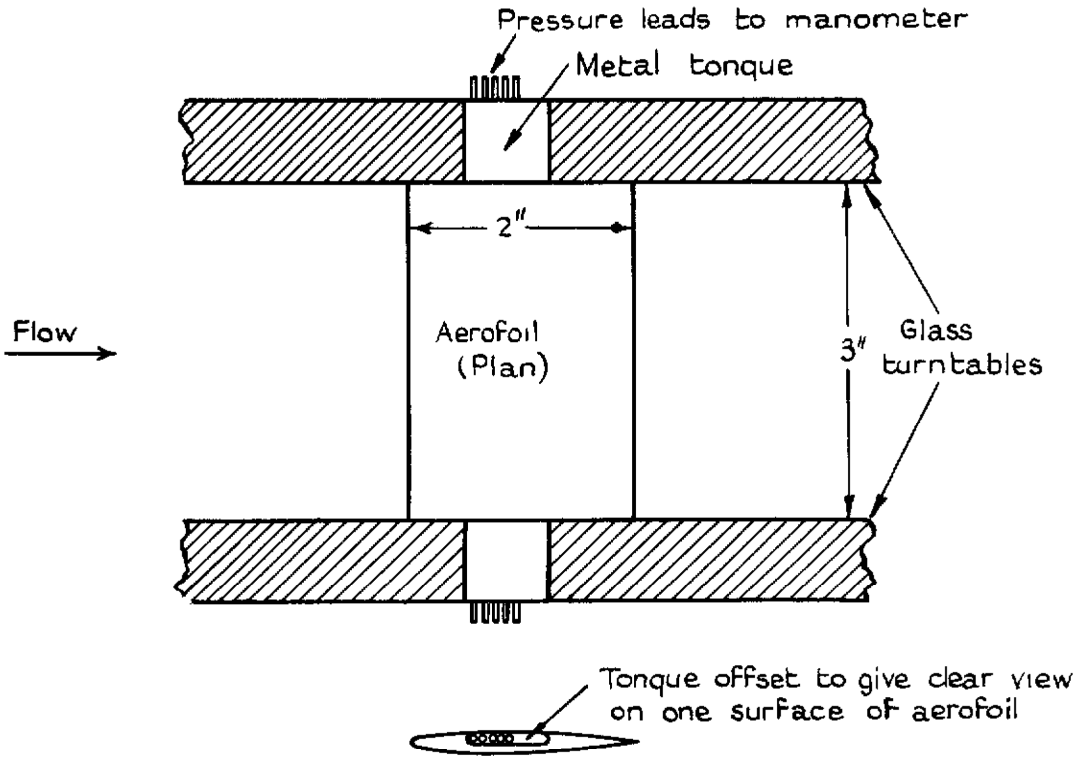
Positions of pressure holes shown thus



x/c	0.05	0.10	0.15	0.20	0.25	0.30	0.35	0.40	0.42	0.45	0.50	0.55	0.60	0.65	0.70	0.75	0.80	0.85	0.90	0.95
$100 \times y/c$	2.36	3.23	3.82	4.26	4.57	4.79	4.93	4.99	5.00	4.99	4.90	4.74	4.46	4.04	3.53	2.97	2.38	1.79	1.19	0.62

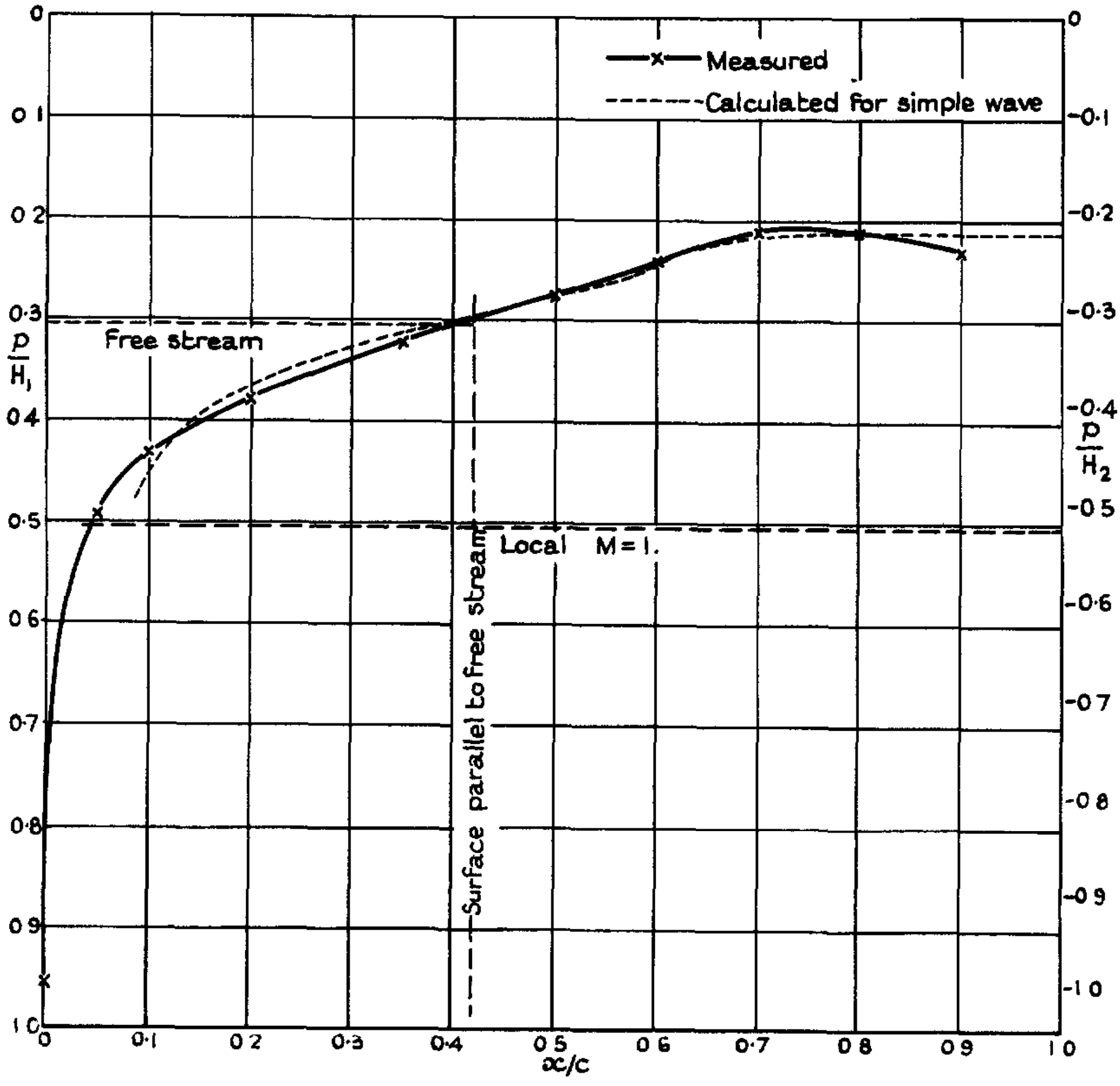
Details of the R.A.E. 104 Section

FIG 2



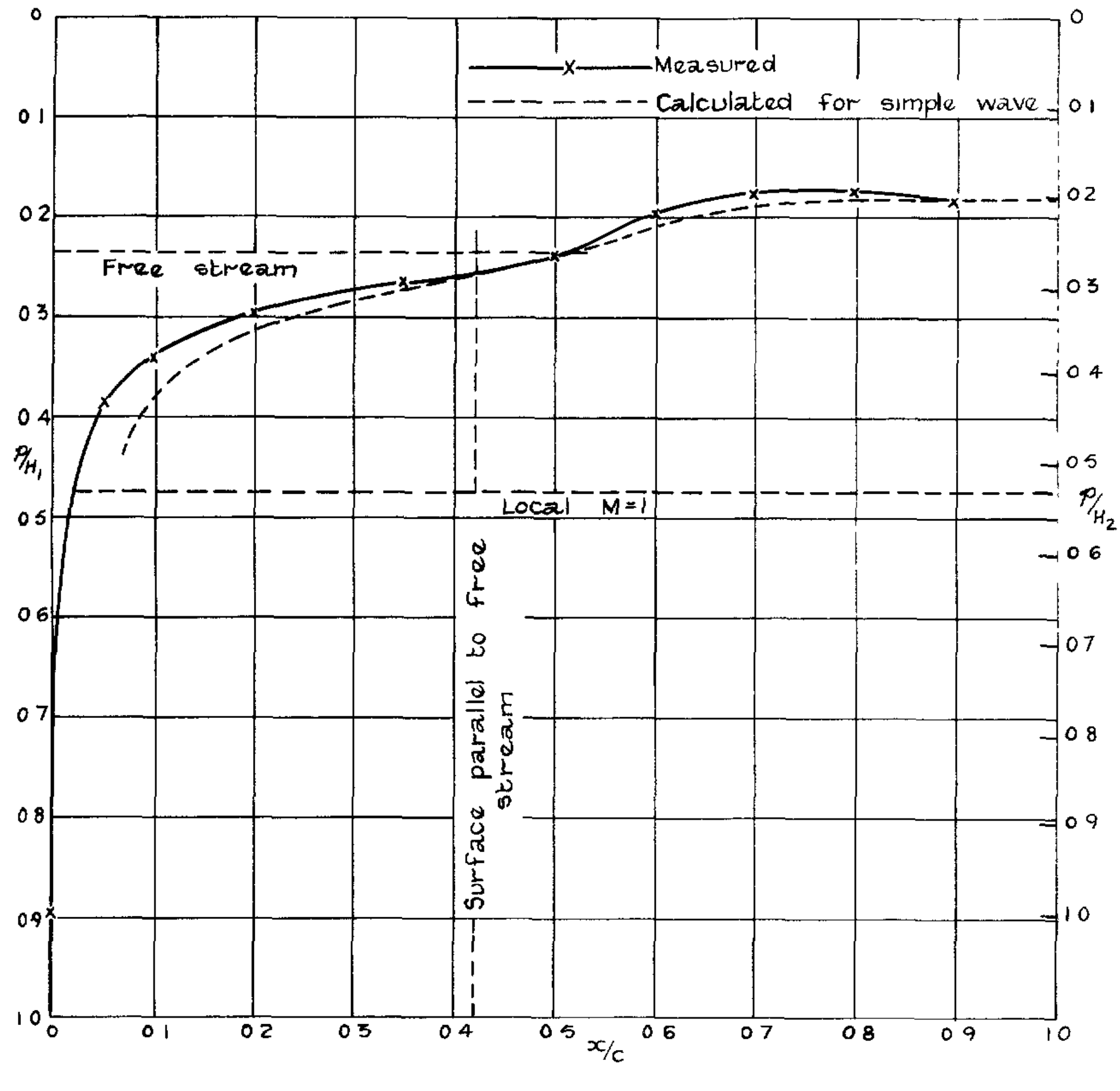
Details of the Method for Supporting the Model.

Fig. 3.

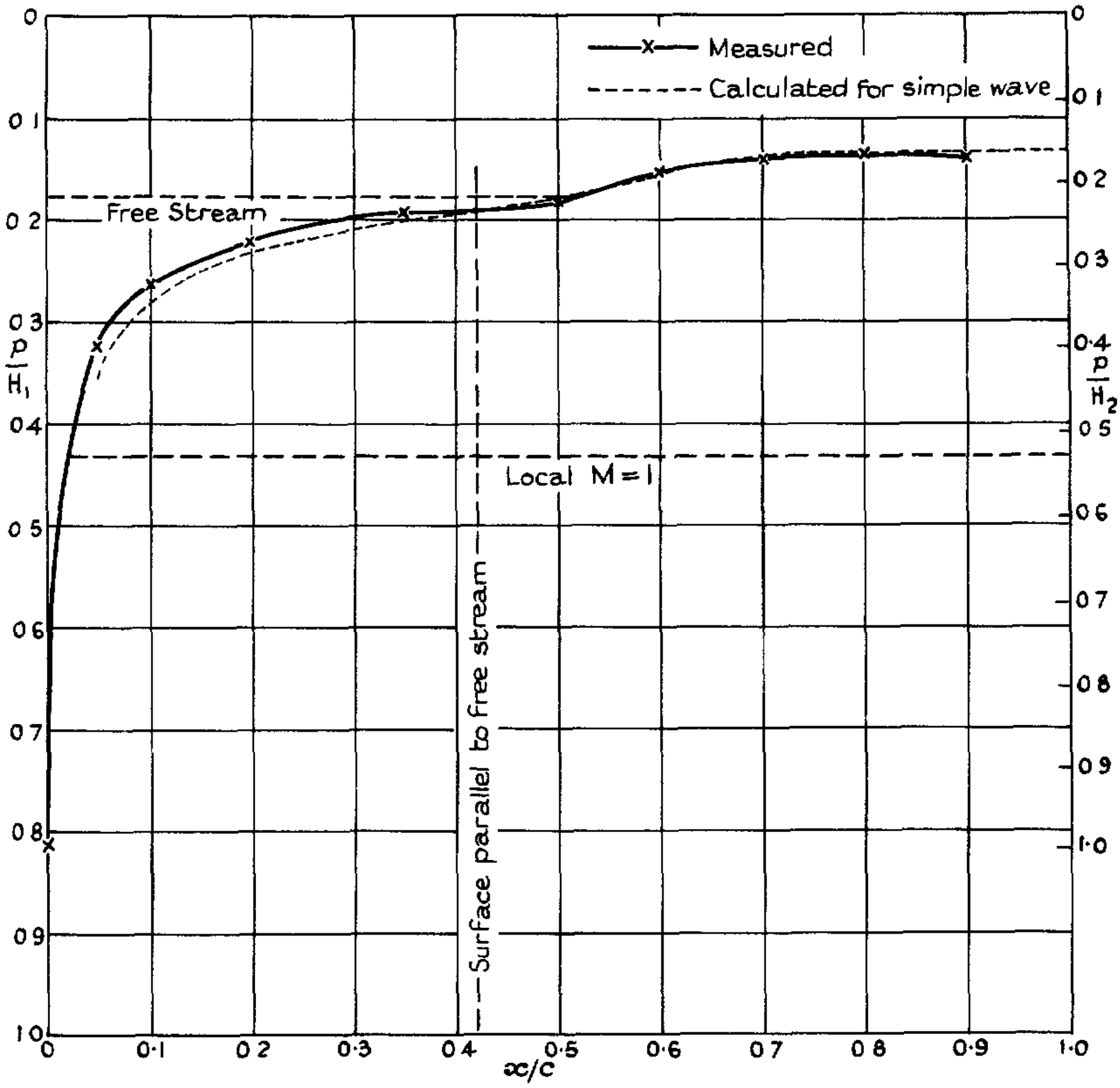


Pressure Distribution on R.A.E. 104 Aerofoil at $\alpha = 0°$ and $M = 1{\cdot}42$

FIG 4

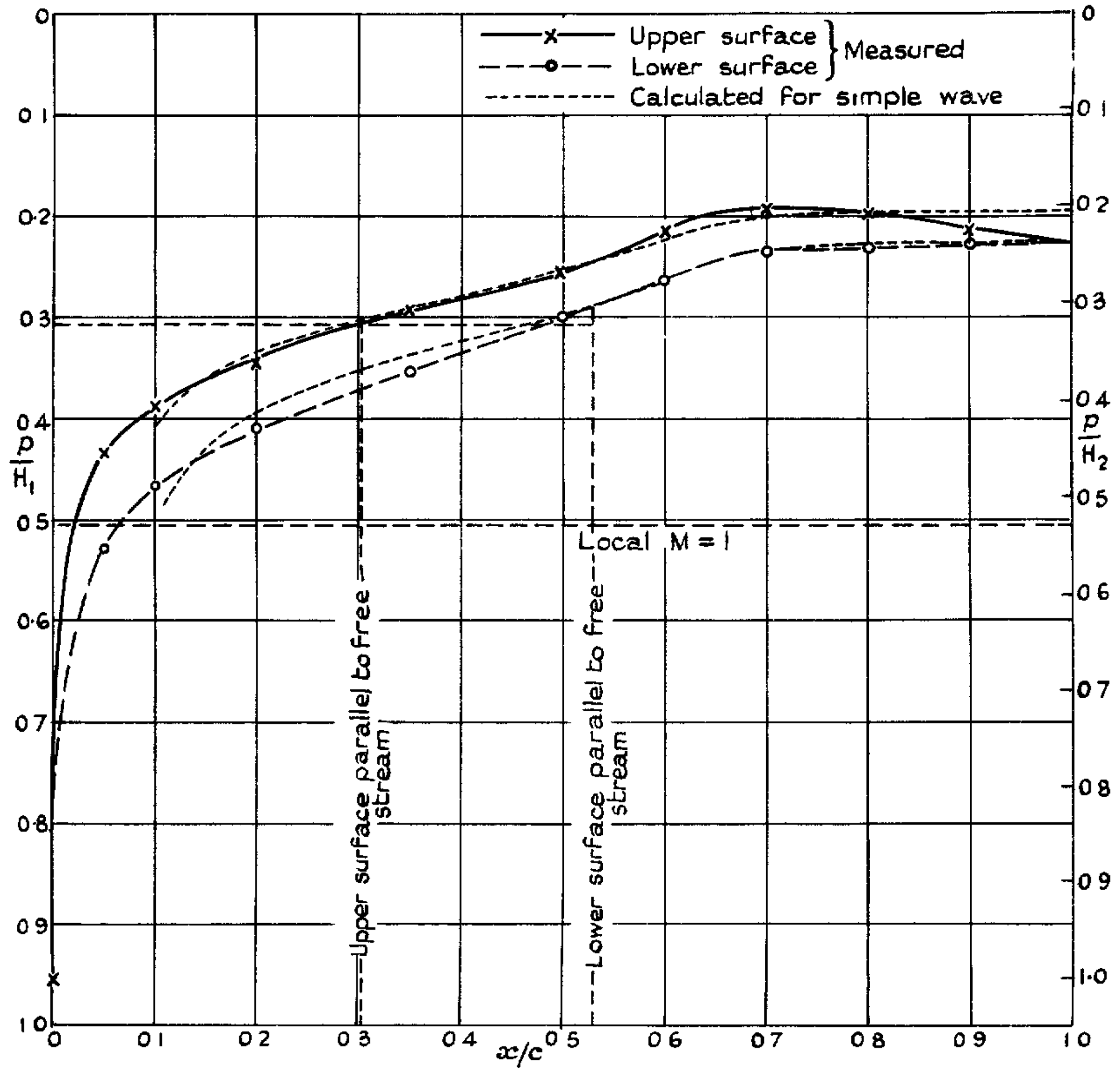


Pressure Distribution on R A E 104 Aerofoil at $\alpha = 0°$ and M = 1·60

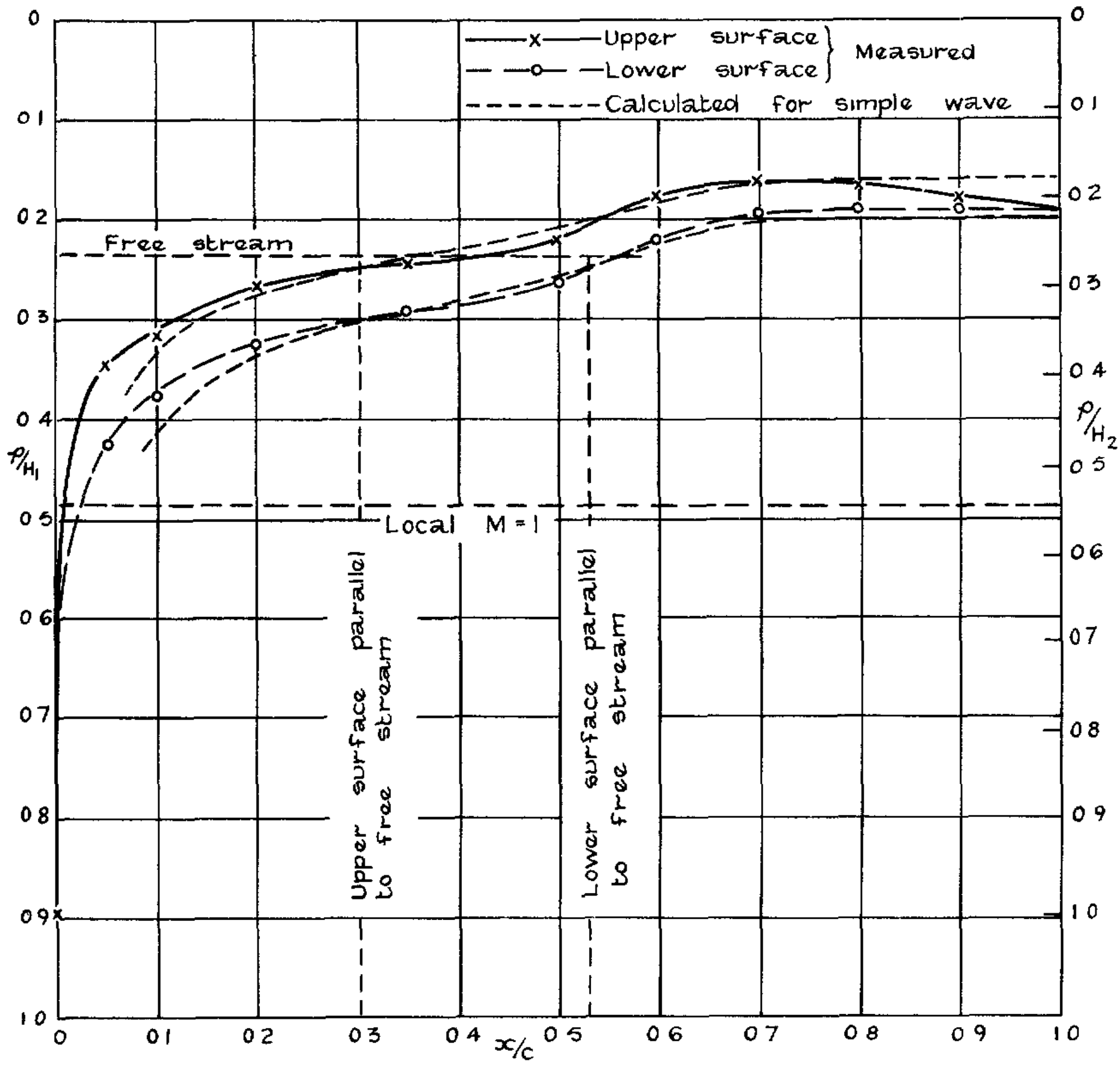


Pressure Distribution on R.A.E. 104 Aerofoil at $\alpha = 0°$ and $M = 1{\cdot}79$

FIG. 6.

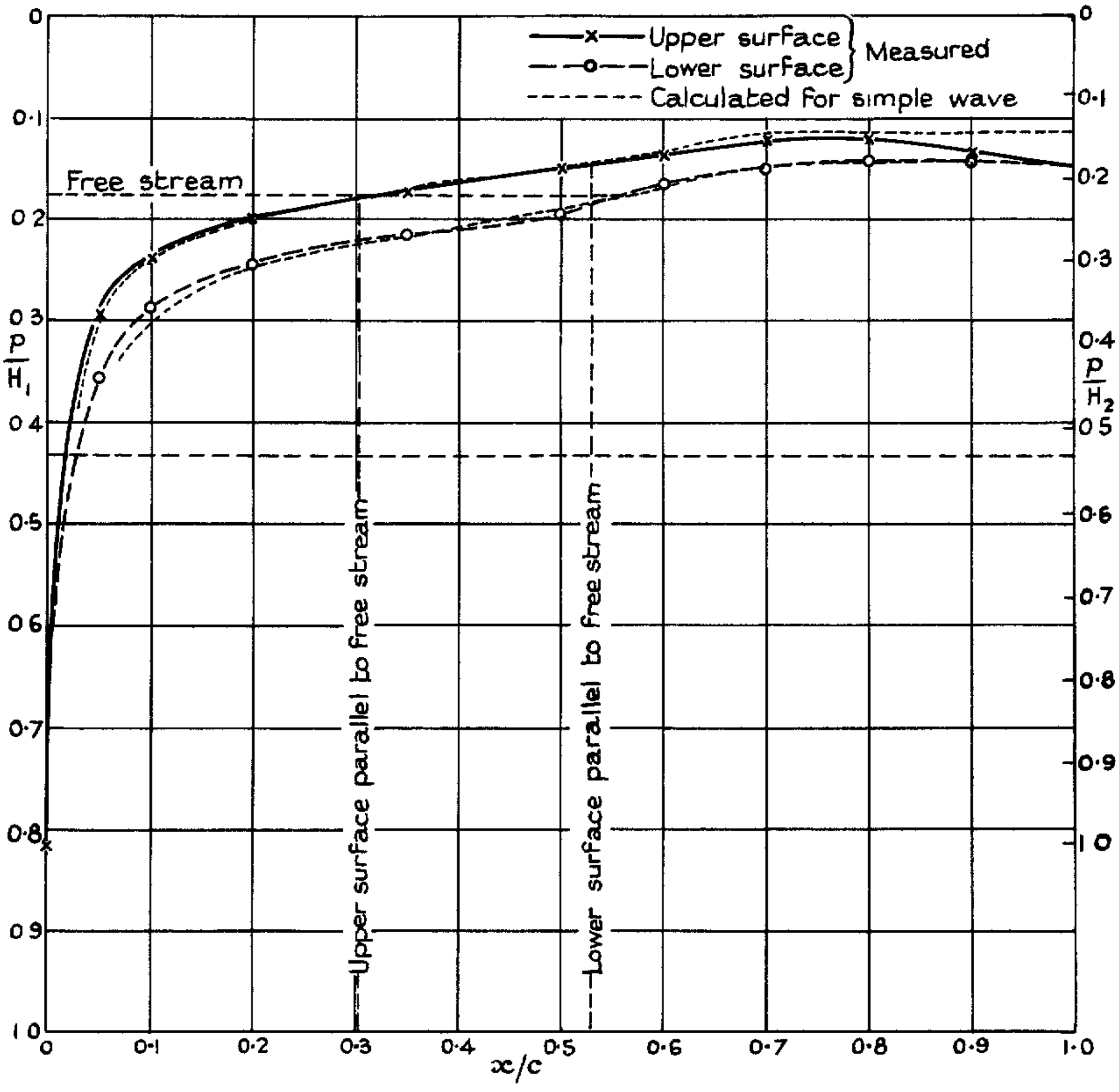


Pressure Distribution on RAE 104 Aerofoil at $\alpha = 2°$ and $M = 1{\cdot}42$

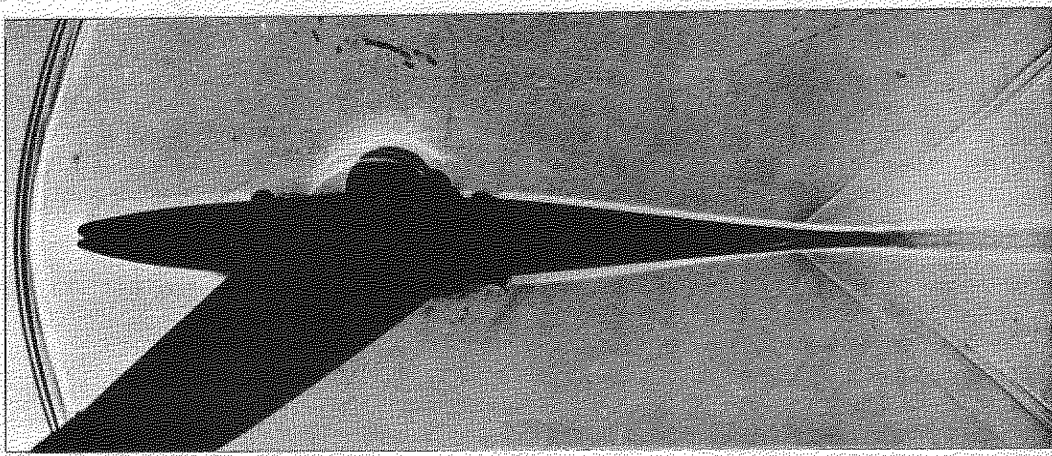


Pressure Distribution on R.A.E. 104 Aerofoil at $\alpha = 2°$ and $M = 1.60$

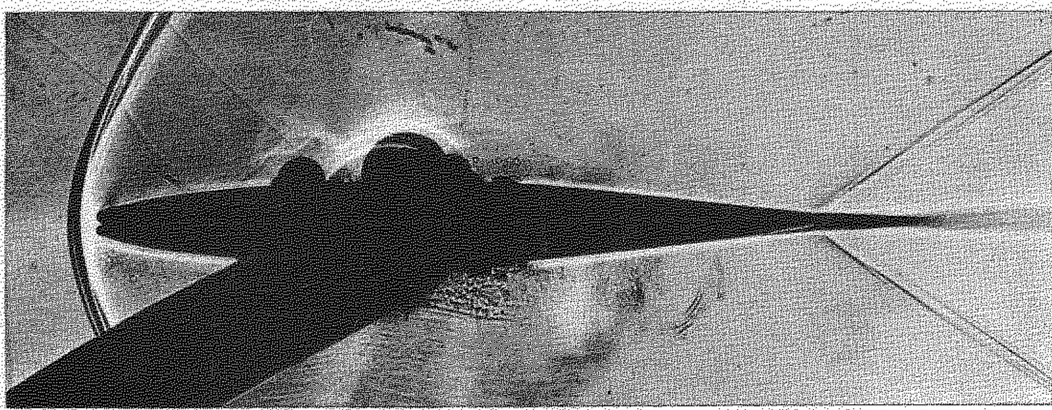
FIG. 8



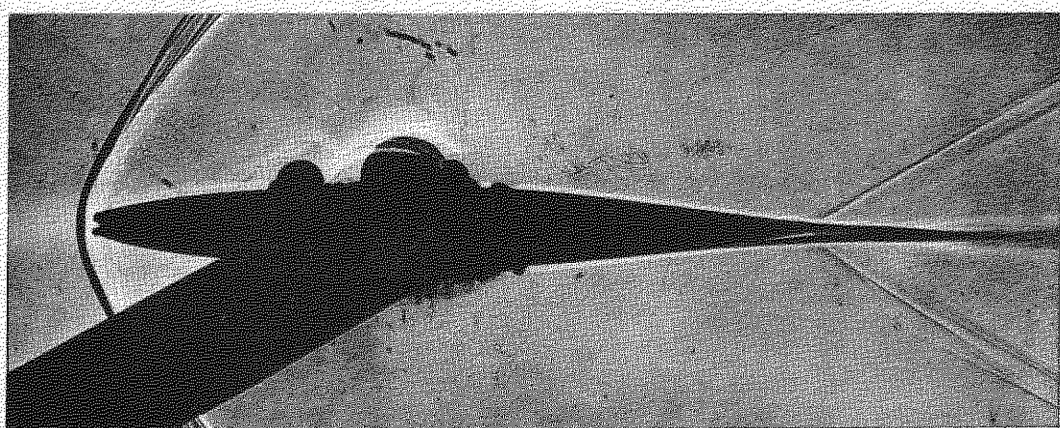
Pressure Distribution on R.A.E. 104 Aerofoil at $\alpha = 2°$ and $M = 1·79$.



(a) M = 1·42



(b) M = 1·60

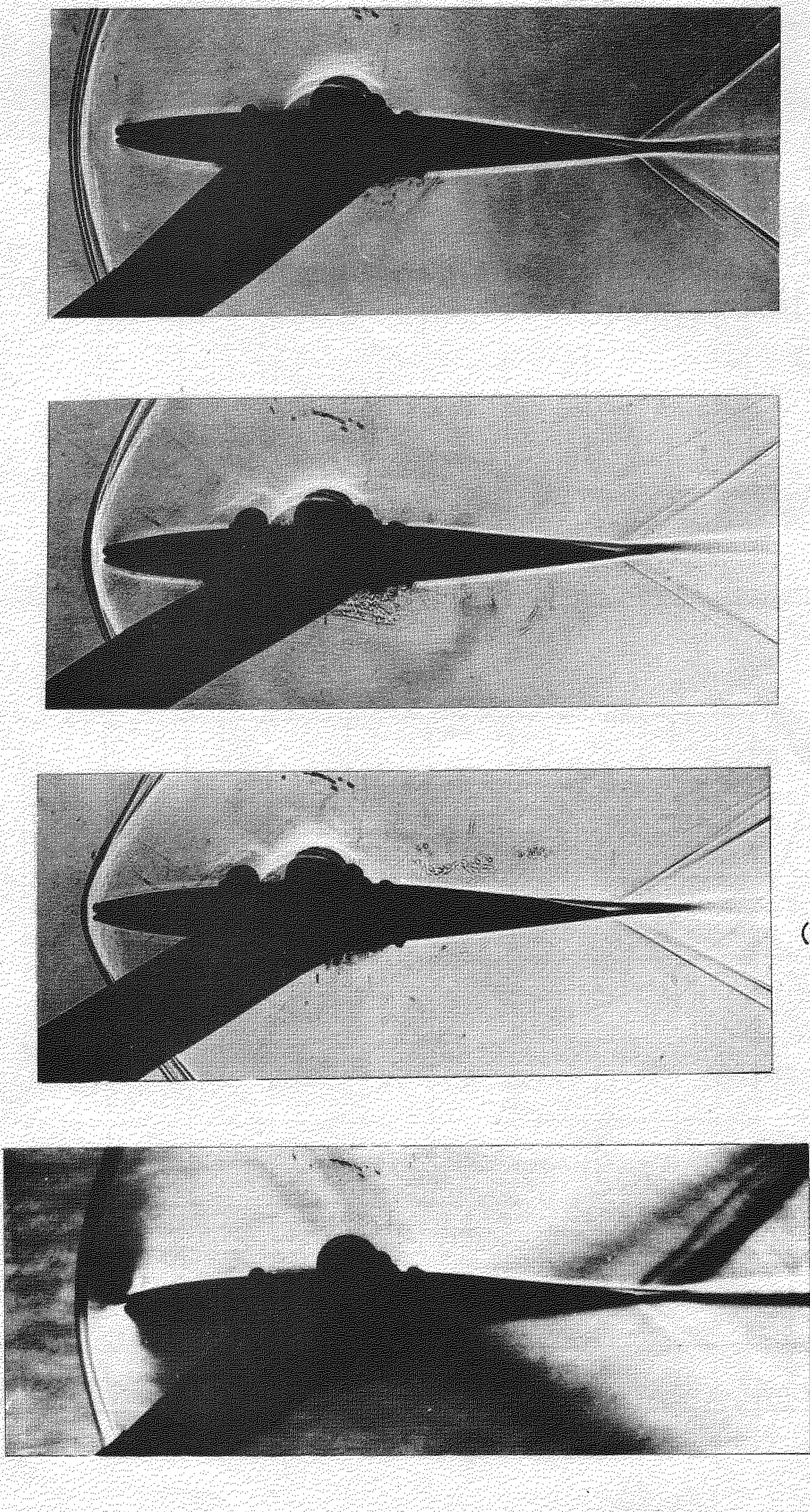


(c) M = 1·79

PHOTOGRAPHS OF THE FLOW ROUND R.A.E. 104 AEROFOIL

AT = $\alpha = 0°$

FIG. 10.



a) M=1·42

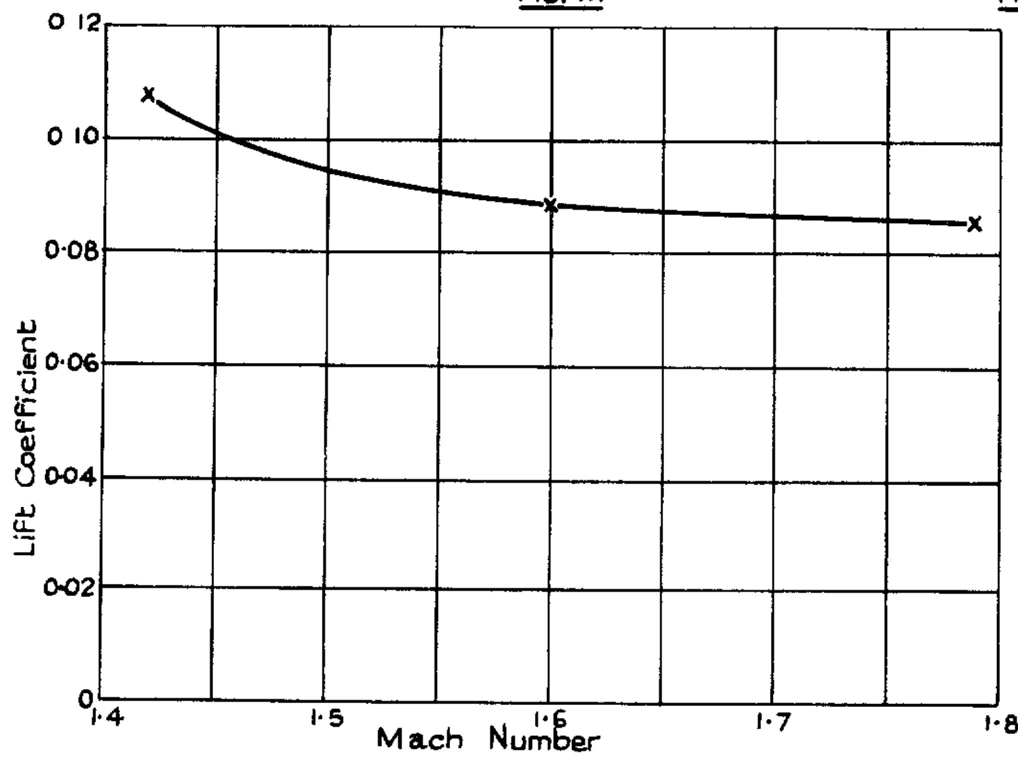
(b) M=1·60

(c) M=1·79

M=1·42

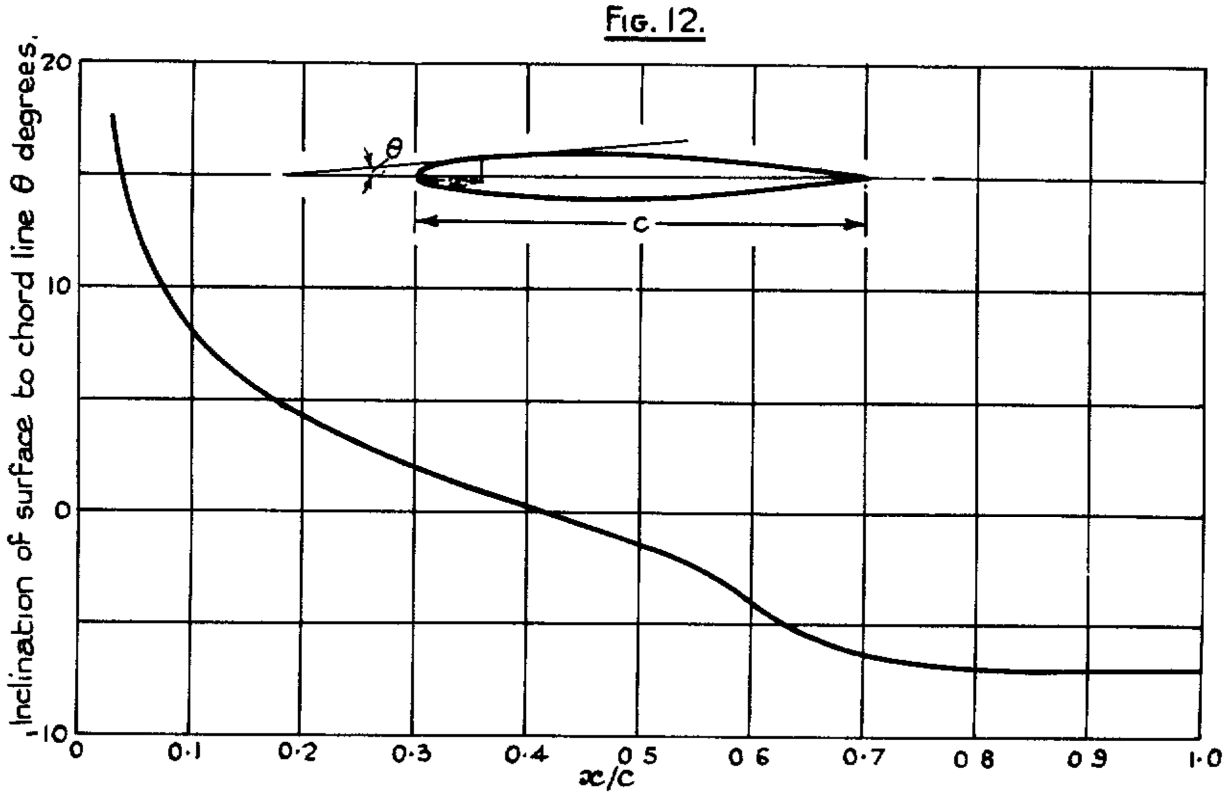
PHOTOGRAPHS OF THE FLOW ROUND R.A.E. 104 AEROFOIL AT $\alpha = 2°$

FIG. 11.



Variation of Lift Coefficient with Mach Number for R A E 104 Aerofoil at $\alpha = 2°$.

FIG. 12.



Local Slopes of the Surface of R.A.E. 104 Aerofoil.

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