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The Interaction Between -Shock Waves and Boundary Layers

By

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of the Aerodynamics Division, N.P.L.

with a note on

The Effects of the Interaction on the Performance of Supersonic Intakes

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J. Seddon, Ph.D.

of the Royal Aircraft Establishment

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Summary

The interaction between shock waves and boundary layers has important effects in many problems of high-speed flow. This paper has been written as a guide to the literature On the subject, and as a critical review of the present state of knowledge concerning both the underlying physical processes and the practical applications. It will be clear to the reader that, although substantial progress has been made, cur knowledge is still far from complete and that more work both of a fundamental nature and on specific applications is needed before the problem is understood sufficiently well for design purposes.

Part I of the paper describes experiments on comparatively simple types of flow designed to provide fundamental information and to assist in the development of the theory. These experiments show that the interaction depends mainly on the Mach and Reynolds numbers and on the strength of the shock Wave. In particular, the interaction of a shock Wave with a laminar boundary layer is shown to produce much larger effects than if the boundary layer is turbulent. For most cases Where the effects of the interaction are large enough to have serious practical consequences it is found that the boundary layer separates from the surface, and the difference between the interaction With laminar and turbulent layers arises mainly because the laminar layer separates much more readily in an adverse pressure gradient. 50 details of the interaction downstream of the separation point thus depend critically on the behaviour of the separated layer, and on the conditions under which it reattaches to the surface.

Many of the features found in the fundamental experiments appear also in practical applications and these are considered in Parts II and III of the paper. Although the emphasis here is on the performance Of aerofoils and wings moving at high subsonic speeds, the importance of the interaction in other examples such as at supersonic trailing edges and in supersonic intakes is also discussed briefly. The differences between the interaction With laminar and turbulent boundary layers are often a source of serious discrepancy between model experiments and. full-scale conditions. for small-scale models it is, therefore, frequently essential to make the boundary layer turbulent by artificial means. Some of the difficulties involved in doing this, and certain of the more promising methods are briefly discussed. It is shown that experiments on models With transition fixed can be used to explain a number of aerodynamic effects encountered in transonic flight, and connected with the occurrence of shock-induced separation of the turbulent boundary layers. For both two-dimensional

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aerofoils and straight and sweptback Wings, turbulent separation occurs for shooks above a certain strength Which applios for both model and full-scale conditions. The effocts of separation arc qualitatively similar under model and full-scale conditions; differences in magnitude Would be expected if the pressure recovery along tho separated layer between tho shock and the trailing edge is affected by Reynolds number, but little information is at present available on this point.

Most of the repercussions of turbulent separation on the steady-motion characteristics of aerofoils and wings can be traced to the associated reduction in the pressure recovery over the rear of tho surface. This is because tho pressure attho trailing edge controls the inter-relation between the two surfaces (so long as the flow at the trailing edge remains subsonic), and in particular tho relative movements of the shock waves and the extents of the local regions of supersonic flow. Certain unsteady-flow characteristics such as buffeting and control surface "buzz" are also thought to be closely associated With boundary-layer separation.

Some evidence is prosentcd on the influence of section shape on the occurrence and effects of separation, but in this, as in many other respects, information relevant to turbulent boundary layers is scarce. Some notes on the further work which is required are given in Part IV of the paper.

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1. Introduction

In many aeronautical problems, such as the flow past wings and the flow through engine intakes and wind-tunnel diffusers, shock waves meet the boundary layer on the surface under consideration. In such circumstances it is well known that the shock wave and the boundary layer interact, and that a flow pattern may result which bears little resemblance to that expected in the absence of viscous effects. An example is given in Fig. 1. Here an oblique shock wave generated by a wedge held in a supersonic stream strikes the surface of a flat plate, the conditions being such that regular reflection (Fig. 1b) would be expected in the absence of the boundary layer. It is seen in Fig. 1a that the actual conditions are very different from this. Thus, for example, instead of being reflected as a single shock, the incident shock is reflected as a system of compression and expansion waves springing from regions of the boundary layer extending from a point well upstream of the incident

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shock to a point well downstream. It is also clear that there are corresponding differences between the measured static pressure distribution on the surface of the plate and that calculated for regular reflection.

Attention was first drawn to the problem in 1939 by Ferri[1], who, during tests in a supersonic tunnel observed boundary-layer separation near the trailing edge of an aerofoil in a region where a favourable pressure gradient would be expected. Soon afterwards observations in several high-speed tunnels revealed the importance of the interaction at high subsonic speeds where the supersonic flow is limited to a region close to the aerofoil. Detailed studies were initiated independently in Great Britain[2], Switzerland[3] and the U.S.A.[4,5,6] in about 1945. The British experiments were confined to the comparatively simple case of the interaction of a shock wave with the turbulent boundary layer on the flat wall of a supersonic tunnel. The Swiss[3] and American[4] experiments examined the more complicated problem of the interaction at the rear of the limited region of supersonic flow near a curved surface held in a subsonic stream. They were made with both laminar and turbulent boundary layers upstream of the shock wave and showed that the nature of the interaction depended critically on the state of the boundary layer. This observation suggested a possible source of serious scale effect in wind-tunnel tests made at low Reynolds number, and increased effort was accordingly put into the development of techniques for minimizing the scale effect by making the boundary layer on the test surface turbulent artificially.

In experiments on curved surfaces it is sometimes difficult to separate the effects of the interaction from those arising from the limited extent of the supersonic region and from the pressure gradient associated with the curvature of the surface. Thus, although useful information continues to be obtained from tests on isolated aerofoils and cascades at high subsonic speeds, most recent experiments which have been designed specifically to investigate the interaction have been made in purely supersonic flow with a boundary layer which is not subject to pressure gradients other than that occurring in the shock wave under investigation. To achieve this condition the boundary layer under test is formed either on a flat plate spanning the tunnel or on a parallel-sided body of revolution; the latter arrangement is more difficult experimentally but has the advantage that effects arising from the presence of the boundary layers on the side walls of the tunnel are minimized. The shock wave is usually produced by placing a wedge in the supersonic stream above the flat plate (Refs. 7 to 13) or by attaching a wedge or step to the surface of the plate (Refs. 14 to 16) or a collar to the body of revolution (Refs. 17 to 19).

Largely as a result of experiments of the type described above the major effects of Mach number, shock strength and boundary-layer Reynolds number on the flow in the vicinity of the interaction are known, and the physical nature of the underlying mechanism is at least partially understood. These aspects are reviewed in the first part of the present paper.

The theoretical study of the interaction presents many difficulties. For example, the usual simplifying assumption of boundary-layer theory that changes parallel to the surface are negligible compared with those normal to the surface clearly fails near the point where a shock wave meets the surface. Thus, although considerable progress has been made, the theory is still far from being complete, or adequate for design purposes. A feature which has received much attention by theoretical workers in the upstream effect of the shock wave (see, for example, Fig. 1) which would, of course, be absent in purely supersonic flow. At one time it was thought that this effect could be explained entirely in terms of the propagation of disturbances upstream through

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the region of subsonic flow which is always present within a boundary layer near the wall. It was hoped, therefore, that it might be possible to calculate the upstream influence by a comparatively simple theory which neglected the effects of viscosity other than those which produced the velocity profile in the boundary layer and the subsonic region near the wall. The original theory due to Howarth[20] was extended by Tsien and Finston[21] who considered a uniform subsonic stream parallel to the wall, by Lighthill[22] who took a linear velocity profile with zero velocity at the wall, and by Robinson[23] who considered an arbitrary profile. It was found, however, that these theories predicted upstream effects which were much smaller than those observed in practice, and that the difference between the extents of the upstream effect observed for laminar and turbulent boundary layers could not be explained[7] in terms of the difference between the thicknesses of the subsonic layers.

An alternative approach is to consider the interaction between changes of the boundary-layer thickness and the pressure changes which they produce in the general stream. For example, a positive pressure gradient (such as that in a shock wave) causes the boundary layer to thicken, and the associated curvature of the boundary layer produces a further positive pressure gradient (the flow in the general stream being supersonic). This gradient in turn causes a further thickening of the boundary layer and a further pressure gradient. The boundary layer thus grows in equilibrium with the pressure gradient caused by its growth, and the pressure gradient decays gradually as the distance upstream of the original disturbance increases. It is thought that this mechanism accounts for a large part of the upstream effect of weak shock waves, and for almost the whole upstream effect of expansion waves. The difference between the upstream effects with laminar and turbulent boundary layer arises because a laminar layer thickens much more readily in an adverse pressure gradient. Theories have been developed on this basis by Oswatitsch and Wieghardt[24] and by Lighthill[25] and appear to be in reasonable agreement with experiment for weak shook waves (see section 4).

If the shock wave is strong, the adverse pressure gradient may exceed that needed to separate the boundary layer from the surface. Separation then occurs, and the associated curvature of the edge of the boundary layer causes a further adverse gradient ahead of the shook which may still be sufficient to cause separation. The separation point may thus be considered to move upstream in a manner similar to that described in the previous paragraph, until the adverse gradient falls to a value which is just sufficient to provoke separation. In this case the equilibrium is between the pressure changes produced in the free-stream by the curvature of the edge of the separated layer, and the pressure gradient that the frictional forces in the dead-air region can withstand. Some thickening of the boundary layer may occur upstream of the separation point because of the adverse pressure gradient there. The relative ease with which a laminar layer separates in an adverse gradient leads to large differences between the upstream effects for laminar and turbulent layers. Theories dealing with boundary-layer separation ahead of the incident shock have been advanced by Lees[26], Crocco and Lees[27], Stewartson[28] and Gadd[29,30].

The details of the flow downstream of the separation point depend on several factors such as the shape of the wall, the flow deflection in the incident shock and, for initially laminar boundary layers, the position of the region where transition to turbulent flow takes place (see Refs. 12 and 26).

Separation occurs in most cases where the effects of the interaction are important in practice. These effects, which are usually undesirable, are discussed in Parts II and III of the present paper.

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Although the emphasis here is on the performance of two-dimensional aerofoils moving at high subsonic speeds, other aspects such as the conditions at supersonic trailing edges, the flow round wings of finite aspect ratio, and the flow through engine intakes are also described briefly. The important question of scale effect is first discussed, and it is shown that tests made at low Reynolds number may be of limited due for full-scale application unless the boundary layer is made turbulent upstream of the region where it interacts with the shock wave. For this reason most of the observations described in the subsequent paragraphs were made with turbulent boundary layers. Examples are included to illustrate the conditions under which boundary-layer separation may be expected to occur near a shock wave, and the effects of shook-induced separation on the characteristics of aerofoils and control surfaces. Possible remedies for the adverse effeots of separation are also mentioned, together with the ties on which further work could usefully be done.

The problem of predicting tho pressure on the base of a two- or three-dimensional body in supersonic flow is closely allied to the examples mentioned above, and. is reviewed in Ref. 31. The importance of the interaction in the performance of cascades at high speeds is discussed in Refs. 32 and 33.

Part I. Experiments Designed to Provide Fundamental Information on the Interaction

2. The Interaction Between the Boundary Layer on a Flat Plate and an Oblique Shock Wave Generated in the Main Stream

The interactions between shock waves and boundary layers which occur in practice are frequently affected by many complicating factors, and in an attempt to give a physical understanding of the problem, simpler types of interaction will be discussed in this part of the paper. As mentioned above, a type of interaction which has been studied by may experimenter-, is that between a plane incident shock produced by a wedge and the boundary layer on a flat plate. This will, therefore, be considered first as it illustrates many of the important features. Since they cover a wider range of conditions than most other tests, and give results which are in most cases in agreement with those obtained by other investigators, the experiments made recently[11,12] at the N.P.L. have been used to provide the examples included here,

The principal factors affecting tho interaction are the boundary-layer Reynolds number, the Mach number and the strength of the shook wave. The wind tunnel used in the investigations at the N.P.L. was designed to enable each of these factors to be varied independently over a wide range.

2.1 Description of the Apparatus

The wind tunnel is of the "direct-discharge" type (Fig. 2a) and works between a source of dry compressed air and a vacuum supply. The working section is 1.5 in. wide and 2.6 in. high and has glass side walls permitting photography of the flow. There is a by-pass (Fig. 2b) beneath the flat plate AB through which tho boundary layer on the tunnel wall passes, so that a new layer begins at the leading edge A of the plate. The plate is 6 in. long and spans the tunnel completely; along its centre line there are a number of static pressure tappings so that the pressure distribution on the surface can be measured. The shock wave is generated by a wedge C which in most cases fully spans tho tunnel, and can be moved along the tunnel by means of a lead screw.

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The stagnation pressure of the air entering the working section can be varied between about $\frac{1}{2}$ and 15 atmospheres absolute, giving for example at $M_0 = 2$ a range of Reynolds number from about 1.7×10^5 to 5×10^6 per inch. The Mach number can be changed by using different nozzles, and investigations have been made at Mach numbers of 1.5, 2, 3 and 4. The strength of the shock is varied by using wedges of different angles. At $M_0 = 4$ wedges of up to 20 deg. apex angle can be used, but at $M_0 = 2$, for example, the maximum shock deflection angle that can be used is only about 10 deg. because the tunnel chokes more easily at the lower Mach number.*

This apparatus is well suited for determining the overall characteristics of the interaction such as the pressure distribution at the wall and the upstream effect of the shock. Its small scale makes it less suitable, however, for the determination of details of the flow such as the boundary-layer profiles.

2.2 Preliminary Discussion of the Experimental Results

Experimental results for a number of Reynolds numbers, Mach numbers and shock strengths are reproduced in Fig. 3. In Fig. 3(a) the Reynolds number is very low and the shock is very weak so that the boundary layer remains laminar throughout the region of interaction. With a slightly greater shock strength and slightly higher Reynolds number the flow is as in Fig. 3(b) where transition to turbulent flow occurs at the position where the shock strikes the boundary layer. With strong shocks but fairly low Reynolds numbers the boundary layer is laminar at separation but turns turbulent upstream of the shock, as in Fig. 3(c). If the Reynolds number is increased sufficiently, transition takes place upstream of the region of interaction. The flow pattern is then similar to that for regular reflection if the shock deflection angle is less than about 6 deg. as in Fig. 3(d). For stronger shocksflow separation occurs even for turbulent boundary layers, however, and the flow pattern is then like that shown in Fig. 3(e).

It can be seen that large departures of the flow pattern from that predicted for regular reflection occur when there is considerable separated flow, and the principal difference between laminar and turbulent interactions is associated with the much greater readiness with which a laminar boundary layer separates.

A feature common to all interactions of the type considered here is that the ultimate pressure rise on the plate is roughly the same as for regular reflection of the shock because the combination of compression and expansion waves reflected from the boundary layer has the overall effect of turning the external flow (which has been deflected through the incident shock) back to a direction approximately parallel to the wall just as the reflected shock does in regular reflection. A further characteristic, except for weak shocks and turbulent boundary layers, is that the shock is reflected. locally where it strikes the boundary layer as an expansion wave. This is a consequence of the fact that the separated boundary layer is unable to withstand a sharp pressure gradient and thus behaves rather like a constant-pressure boundary as far as shock reflection is concerned.

The particular affects produced by different boundary-layer conditions are discussed below.

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*It is not possible to reduce the blockage by making the wedge very short because the incident shock is then followed too closely by the expansion wave springing from the rear of the wedge.

2.3 Results for Boundary Layers Laminar over the Whole Region of Interaction

(a) Flow Patterns and Pressure Distributions at the Wall

These are as in Fig. 4. If the shock is sufficiently weak (with a deflection angle less than about 2 deg.) flow separation does not occur. The pressure distribution at the wall has a single point of inflection at approximately the position where the shock strikes the boundary layer. For stronger shocks separation occurs, and the pressure distribution at the wall has three points of inflection, i.e., the curve has a "foot". This type of distribution arises in the following manner. The pressure gradient increases continuously up to separation, but downstream of separation there is a dead-air region* near the wall. The boundary layer can withstand a considerable adverse pressure gradient where the dead-air region is thin, because the rate of change normal to the wall of the friction forces acting on the dead air can then be considerable, but this is not so when the region is thick. The dead-air region *thickens* rapidly downstream of separation because the edge of the boundary layer is deflected away from the wall. The pressure gradient thus falls off downstream of separation. At the point where the shock strikes the boundary layer the flow is deflected towards the wall by the shock and the reflected expansion, so that the dead-air region becomes thinner again. Hence the pressure gradient increases once more until the reattachment point is reached. Downstream of this point the pressure gradient again falls as the boundary-layer profile returns to the undisturbed zero-pressure-gradient form, and the pressure approaches its peak value asymptotically.

It should be noted that with completely laminar boundary layers the foot on the pressure curve ends where the shock strikes the edge of the boundary layer. Also, the maximum pressure gradient downstream of the foot is of the same order as that near the separation point at the upstream end of the foot.

(b) Upstream Influence

It is found that for given shock strength the upstream distance d divided by the displacement thickness δ_o^* (see Fig. 4) increases with increasing Reynolds number R_o roughly as $R_o^{3/4}$ (this is discussed further in section 2.3(b)). The quantity $\frac{d}{\delta_o^*}\left(\frac{10^4}{R_o}\right)^{3/4}$ is, therefore, a function of shock strength and free-stream Mach number only. It is plotted in Fig. 5 for M_o = 2, 3 and 4** against peak pressure ratio $\frac{p_{max}}{p_i}$ and also against peak pressure coefficient $\frac{2}{\gamma M_o^2}\left[\frac{p_{max}}{p_i} - 1\right]$. The upstream effect for a given pressure ratio is much smaller at the higher Mach numbers, although for a given pressure coefficient the effect is approximately independent of Mach number.

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*The dead-air region is defined as being of thickness t where, in the usual notation $\int_0^t \rho u dy = 0$. The velocities in the dead-air region, although small, are not zero since there is reversed flow at the wall and forward flow further out.

**Experimental difficulties prevented boundary-layers laminar over the whole region of interaction from being obtained at M_o = 1.5.

2.4 Results for Boundary Layers Laminar at Separation but Turbulent before Reattachment

The boundary layer turns turbulent before reattachment unless the Reynolds number is very low and the shock is weak. The fully laminar type of flow described above is, therefore, much less frequently encountered than that considered here.

(a) Flow Patterns and Pressure Distributions at the Wall

These are as in Fig. 6. The position of transition relative to the shock depends on shock strength. For moderately weak shocks it may be at or slightly downstream of the shock as in Fig. 6(a). For stronger shocks it is upstream of the shock as in Fig. 6(b) and (c). When, as with very strong shocks, transition is a long way ahead of the shock (Fig. 6(c)), the pressure distribution at the wall may have five points of inflection. There is first a "foot" where the boundary layer is laminar and the pressure gradient falls off after separation because the dead-air region becomes thick as discussed in section 2.3(a). When transition occurs the boundary layer, although still separated, becomes capable of withstanding a larger pressure gradient and the pressure rises steeply. However, the dead-air region becomes very thick further downstream so that even the turbulent friction forces acting on it are not sufficient to permit a large adverse pressure gradient. The pressure gradient accordingly falls once more. Where the shock strikes the separated layer the flow is deflected towards the wall so that the dead-air region is thinned and the pressure gradient rises again. Reattachment occurs near the peak pressure position since downstream of reattachment the pressure gradient falls rapidly as the boundary-layer profile approaches the zero-pressure-gradient turbulent form.

The quintuply inflected form of pressure distribution occurs only when transition takes place a long way ahead of the shock. For weaker shocks there is a steady increase of pressure between transition and reattachment, presumably because the boundary layer does not thicken sufficiently between transition and the shock for the pressure gradient to be appreciably affected. The pressure distribution then has the triply inflected form of Fig. 6(a) and (b). It differs from the pressure distribution for a separated wholly laminar layer in that the pressure gradient after the laminar "foot" is usually considerably steeper than at the beginning of the foot. Also, the "foot" ends where transition occurs and this may be upstream of the point where the shock strikes the edge of the boundary layer.

(b) Upstream Influence

When the Reynolds number is raised with a given shock strength it is found that the separation point moves closer to the transition point, i.e., the laminar "foot" becomes shorter but the transition point remains fixed relative to the shock. Thus the total upstream distance d divided by the displacement thickness δ_o^* (see Fig. 6) decreases with increasing Reynolds number R_o for a given shock strength. Fig. 7 shows $\dfrac{d}{\delta_o^*}$ plotted against R_o at $M = 2$ for various values of the peak pressure ratio. The left-hand end of the curve for the lowest pressure ratio is in the wholly laminar regime where $\dfrac{d}{\delta_o^*}$ increases with R_o; the difference between the upstream effects for the two regimes is very clear in this diagram. The increase of $\dfrac{d}{\delta_o^*}$ for wholly laminar flow is associated with the fact that separation appears to occur at a lower pressure when R_o is

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increased. This presumably tends to make $\frac{d}{\delta_o^*}$ increase with R_o even when transition occurs, but the effect is masked by transition occurring more closely after separation and actually $\frac{d}{\delta_o^*}$ then decreases with increasing R_o. Similar effects occur in the experiments described in section 5 where minute quantities of air are injected into the boundary layer on an aerofoil in order to fix transition. When the quantity of air injected is insufficient to provoke transition ahead of the interaction, the upstream effect of the shock is nevertheless reduced because the boundary layer is less stable and turns turbulent more readily after separation. An increase in the turbulence of the free stream is found to have a similar effect (see section 5).

The upstream effect $\frac{d}{\delta_o^*}$ is found to vary approximately as $R_o^{-1/3}$ when transition occurs in the region of interaction. Hence the factor $\frac{d}{\delta_o^*}\left(\frac{R_o}{10^4}\right)^{1/3}$ is a function of shock strength and Mach number only, and in Fig. 8 curves of this factor are plotted for M_o = 1.5, 2, 3 and 4 against peak pressure ratio and peak pressure coefficient. For a given pressure ratio the upstream effect is much smaller at the higher Mach numbers, but there is much less difference for a given pressure coefficient.

2.5 Results for Boundary Layers Turbulent over the Whole Region of Interaction

At sufficiently high Reynolds number transition occurs ahead of the region of interaction and the following effects are observed.

(a) Flow Patterns and Pressure Distributions at the Wall

These are as in Fig. 9. If the shock is of only moderate strength, with a deflection angle of less than about 6 deg., flow separation does not occur. There is then a very steep rise of pressure (Fig. 9(a)) at the point where the shock strikes the boundary layer, and conditions are similar to those for regular reflection.

For stronger shocks separation occurs and the pressure distribution at the wall is similar to that shown in Fig. 9(b). The pressure rises steeply up to the separation point after which the pressure gradient falls. Downstream of the point where the shock strikes the boundary layer the pressure gradient increases, but it decreases again after reattachment which occurs close to the peak pressure position. The mechanism is physically very similar to that for wholly laminar layers discussed in section 2.3(a) but there is a very large difference in the scale of the pressure changes which occur. This is associated with the fact that the pressure rise needed to separate a turbulent boundary layer is approximately five times as large as that needed to separate a laminar layer.

(b) Upstream Influence

Over the range covered there appears to be no systematic variation of the upstream effect $\frac{d}{\delta_o^*}$ with Reynolds number R_t (see Fig. 9). Accordingly in Fig. 10 $\frac{d}{\delta_o^*}$ is plotted as a function of shock

strength/

strength and Mach number against peak-pressure coefficient. As with boundary layers laminar at separation, the upstream effect for a given pressure ratio is much smaller at the higher Mach numbers, although for a given pressure coefficient there is much less variation with Mach number.

(c) The Interaction with Boundary Layers made Turbulent Artificially

It would be expected that the details of the interaction will be substantially the same for "naturally" turbulent boundary layers and for turbulent boundary layers formed "artificially" by using some type of spoiler to provoke transition, provided that the boundary layer thickness is roughly the same in the two cases and that observations are not made too close to the spoiler. Indeed (as argued by Liepmann)[7] it may sometimes be preferable to use an artificial layer as the transition front is then uniform across the body instead of occurring irregularly as it sometimes does when transition is free. The similarity between the flow patterns and pressure distributions for natural and artificial turbulent layers is illustrated in Fig. 11; the small differences which exist arise mainly from differences of boundary-layer thickness. Also shown in Fig. 11 are the flow pattern and pressure distribution which are observed at the lower Reynolds number if transition is not fixed, and it is clear that a large scale effect would arise if tests made at the low Reynolds number without fixing transition were used to predict the flow at high Reynolds numbers or with turbulent-boundary layers. The technique of simulating high Reynolds number conditions by making the boundary layer turbulent is thus of great value in model tests and is discussed in greater detail in sections 5 and 6.

2.6 The Pressure at Separation for Laminar and Turbulent Boundary Layers

The difference in the pressure at separation with laminar and turbulent boundary layers is very marked. Thus, for a laminar layer an approximate theoretical analysis[29] gives the pressure p_s at separation as

$$\frac{p_s - p_1}{p_1} = \frac{0.780\gamma M_1^2}{[(M_1^2 - 1)R_s]^{1/4}} \left[\frac{\left(1 + \frac{\gamma - 1}{2} M_1^2\right)\left(1 - \frac{0.636 \tan^{-1}\left[\left(\frac{\gamma - 1}{2}\right)^{1/2} M_1\right]}{\left(\frac{\gamma - 1}{2}\right)^{1/2} M_1}\right)}{1 + 0.693\,(\gamma - 1)\, M_1^2} \right]^{1/2} \quad \ldots(1)$$

where p_1 is the pressure upstream of the interaction, M_1 is the free-stream Mach number, and R_s is the Reynolds number based on the free-stream conditions and the distance from the loading edge to the separation point. According to this equation, for $\gamma = 1.4$ and $R_s = 0.25 \times 10^6$, the values of $\frac{p_s - p_1}{p_1}$ at $M_1 = 1.5$, 2, 3 and 4 are respectively 0.065, 0.095, 0.175 and 0.272. Experiment confirms that these values are of the right order.

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For turbulent boundary layers, on the other hand, a semi-empirical relation[30,*] is

$$\frac{p_s - p_i}{p_i} = \left[\frac{1 + \frac{\gamma - 1}{2} M_1^2}{1 + 0.64 \frac{\gamma - 1}{2} M_1^2}\right]^{\frac{\gamma}{\gamma-1}} - 1 \qquad \ldots(2)$$

which gives the values 0.52, 0.84, 1.51 and 2.08 at M_1 = 1.5, 2, 3 and 4 respectively. These values are again in fair agreement with experiment.

Comparison of the values calculated from equations (1) and (2) shows that the turbulent layer can withstand a much larger pressure increase than the laminar layer without separating.

In section 2.3(a) it was stated that a wholly laminar boundary layer would not separate if the incidence shock was of less than about 2 deg. deflection angle. The pressure rise for regular reflection of a 2 deg. shock is roughly twice as large as that given by equation (1). This equation is, however, applicable only where separation occurs upstream of the shock. If the shock strikes the boundary layer at a position where the pressure rise is less than that given by equation (1) it is possible for separation not to occur even if the ultimate pressure rise downstream of the shock is considerably greater than that given by the equation. With wholly laminar layers it is found that the pressure increase downstream of the shock is roughly the sane as that up to the shock, so that the ultimate pressure rise for a shock just strong enough to cause separation is roughly twice the pressure increase at separation.

With turbulent boundary layers the pressure increase downstream of a shock just strong enough to muse separation is much less than that up to the shock. Accordingly, as stated in section 2.5(a), separation for a turbulent boundary layer occurs for incident shocks of greater than about 6 deg. deflection angle since regular reflection of a 6 deg. shock gives a pressure rise approximately equal to that predicted by equation (2).

This concludes the survey of the experiments performed at the N.P.L. on the interaction between a boundary layer on a flat plate and an oblique shock wave generated in the mainstream. Further details will be found in Refs. 11 and 12. A brief account will now be given of the experiments which are still in progress. Here the shock wave is generated by a wedge attached to the flat plate and the conditions thus bear certain similarities to those at the trailing edge of a supersonic aerofoil (see section 8).

3. The Interaction between the Boundary Layer on a Flat Plate and a Shock Wave Generated by a Wedge on the Plate

3.1 Description of the Apparatus

The apparatus is the same as that used in the experiments described above except that the wedge is placed on the surface of the plate as sketched in Fig. 2(c). The wedges are considerably shorter than those used to produce shocks in the mainstream*. It is, therefore, possible to use larger wedge angles without choking the tunnel. However, wedge angles nearly twice as large are needed to produce the same pressure rise in the boundary layer, and accordingly, the range of peak pressure ratios covered is no greater than with shocks generated in the mainstream.

The wedges have pressure tappings along the inclined face so that the pressures on the wedge as well as on the plate can be measured.

3.2 Discussion of the Experimental Results

As with shocks generated in the mainstream it is found that three types of boundary-layer flow can occur - wholly laminar, laminar at separation but turbulent at reattachment, and wholly turbulent. The variation of upstream effect with Reynolds number in these three régimes follows the same laws as with externally generated shocks. In all cases, however, the upstream effect is considerably smaller than for an external shock giving the same overall pressure rise and for the same Mach number and Reynolds number. This is illustrated in Fig. 12 which shows the interaction of a strong shock wave with a turbulent boundary layer at $M_0 = 4$. The upstream effect is much less with the wedge on the plate although the pressure rise in the region of interaction is actually slightly greater than for the external shock. Figs. 13, 14 and 15 also show this difference of upstream effect. Here the dotted curves give the distance from the point where the pressure begins to rise to the point where the externally-generated shock strikes the boundary layer. Although the difference is undoubtedly associated with the different ways in which the dead-air region develops and in which reattachment takes place, the details are not yet understood but it is hoped that further work now in progress will reveal the explanation.

Separation is found to occur at the same pressures with the wedge on the plate as with a shock generated in the mainstream. This is because the pressure distribution in the region of the separation point must be in equilibrium with the local rate of thickening of the boundary layer. Hence provided that the position of the agency which provokes separation is so adjusted that the separation point remains fixed, the pressure distribution in the neighbourhood of separation will be independent of the nature of the agency.

4. An Experiment to Determine the Upstream Influence in the Absence of Separation

Since the upstream influence which takes place in the absence of separation is small (of the order of ten boundary layer thicknesses for laminar, and one boundary layer thickness for turbulent layers) it is difficult to measure accurately in experiments of the type described above. Moreover, in the case of a laminar layer, it is difficult in practice to produce a shock wave which is so weak that separation does not occur. To provide data for comparision with theory an experiment[34] has, therefore, been made recently at the N.P.L. on a comparatively simple model of a boundary layer produced by the apparatus sketched in Fig. 16. Here a subsonic stream moving parallel to the supersonic mainstream ($M_0 = 1.6$) is formed by a secondary nozzle located in the wall/

*They nevertheless avoid interference effects from the expansion at the rear of the wedge equally well because the convergence of the expansion and shock cannot influence the flow in the region of interaction as it can for the externally generated shock.

wall of a supersonic tunnel. The Mach numbor of the secondary stream is adjustud by controlling its total head by means of a valvo at tho entry to the secondary-stream settling chanbor.

The flow examined was that up a wodgo attached to tho wall under the secondary stream, and the upstrean effects of the wodgo were determined by schlieren photography and pressure measurements at the wall. The Mach number profiles in the secondary stroam were determined from explorations made with Pitot and static tubes; the conditions wore, of course, not uniform across the stream because of the formation of a boundary layer near the wall and a mixing region near the outer edge. Some of the experimental results are shown in Fig. 17 where the distance upstream of the leading edge of the wodge at which tho pressure rise at the wall has fallen to 0.1 of the total pressure rise is plotted against the mean Mach number of the secondary stream.

The theory suggested[25] by Lighthill enables the upstream effoct to be calculated for the measured Mach numbor profilo in tho secondary stream, and the results thus obtained aro includod in Fig. 17 for comparison with the upstream effect observod experimentally. Tho agreement is seen to be quite good, and sinco tho upstream effect which occurs in the absence of separation does not need to bo known accuratoly this suggests that the theory is satisfactory for mst practical purposes.

Part II. The Effects of tho Interaction on Aerofoiln and Wings

These effects are found to be very important and to havo considerable influence on the aerodynamic forces and moments for aerofoils, wings and control surfaces. This is particularly so at transonic speeds when the flow is supersonic over part of the chord only and the shock-waves move along the surface to the trailing edge as the free-stream Mach number is raised and the supersonic flow extends to the trailing edge. The transonic régime is extraordinarily complex theoretically, because, in addition to tho difficulties assocwted with the mixed subsonic and supersonic flow, the strong inter-relation between the main flow and the boundary-layer flow precludes a treatment similar to those which have proved successful for flow with boundary layers at low speed& and supersonic speeds[36]. The experimental results described in the succeeding paragraphs illustrate how closely the behaviour of the mixed flow and the effects of the shook-wave boundary-layer interaction are interwoven, and how the work on the idealized flow-models discussed above (Part I) is helping towards a better understanding of both. They further demonstrate how the results of carefully controlled wind-tunnel tests can be used to explain a number of aerodynamic phenomena encountered in transonic flight and connected with the occurrence of shock-induced separation of the turbulent boundary-layers.

5. Scale Effects

As one might expect, the influence of Reynolds number on the interaction leads in certain circumstances to considerable scale effects on aerofoils. An understanding of such effects is important for the proper control of wind-tunnel experiments and reliable interpretation

These separations are the cause of the large 'upstream-influence' of the shock-waves, end also of the changes in shock pattern illustrated by the examples given in Figs. 18 and 19. The extensive 'foot' on the normal shook in Fig. 18(a) shows tho extent of laminar separation in that case, the foremost inclined compression occurring immediately upstream of the point of separation. The mechanism which produces the upstream-inclined wave in Fig. 19(a) is thought to be associated with the finite height of the supersonic region"; the gradual compression accompanying the separation produces a convergence of the 'incoming' family of characteristics (i.e., those inclined upstream with respect to the surface and along which infinitesimally mall compressive disturbances are occurring) and therefore a finite disturbance at a point above the surface, which in turn leads to the 'incoming' shock.

The pressure changes on the surface under the two shock patterns illustrated for laminar layers are the sane, and very similar to those observed for an inclined wave impinging on a flat plate (see 82.3). This type of pressure distribution seem in fact to occur almost universally when a shock-wave interacts with a laminar boundary-layer at low Reynolds numbers. The shape of the curve has the appearance of the 'laminar foot* described in §2.3 if plotted with increasing pressure upwards, or of a 'truncated peak' if plotted with decreasing pressure upwards, as is usual for aerofoils. The weak compression always occurs at, or just upstream of tho separation point. For the shock shown in Fig. 18(a) the main compression occurs through the main, or normal, branch of the foot; for that in Fig. 19(a) the main compression occurs through the shorter normal shock downstream of the other waves, the upstream-inclined wave and its reflected expansion together having a comparatively small effect on the pressure at the surface.

The effect of laminar separation on an aerofoil decreases with increasing Reynolds number in much the same way as was observed for the flat plate. This is shown in Fig. 20(a) by the pressure distributions for a typical case[38], the truncation of the peak becoming loss noticeable as the Reynolds number is increased. The effect of the laminar separation had apparently become negligible at $R = 3.5 \times 10^6$** although tho boundary layer remained laminar upstream of the shock.

An effect similar to that of increasing Reynolds number has been obtained by injecting minute quantities of air into the boundary layer and is illustrated in Fig. 21. The aerofoil was symmetrical and was fitted with a row of small holes across the span at 0.15 chord back from the leading edge. The purpose of these holes was to promote transition to turbulent flow by the method introduced by Fage and Sargent[39]. As found by them, it was necessary to inject about 0.01 to 0.02 of the mass flow in the boundary layer to cause transition, but if smaller quantities were injected a progressive change took place in the interaction between the shock-wave and disturbed laminar boundary-layer. The photograph in Fig. 21 (a) for no injection (top left-hand corner) shows nod shocks on both surfaces, each with laminar separation and gradual compression

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*It should be noted that this type of shock pattern changes to the one illustrated in Fig. 18(a) as the free-strew Mach number, and therefore the height of the supersonic region, is increased.

**The exact shape of the curve cannot of course bc found on an aerofoil where tho number of pressure holes is usually severely limited.

some distance upstream of its foot. The other photographs show the progressive change in the lower surface shock as the injection quantity was increased on this surface. The point of separation moved nearer to the main shock, just as it does with increasing Reynolds number; for the highest injection there was practically no upstream effect, the boundary layer being turbulent at the shock for this case. No air was injected on the upper surface for any of the photographs. The change in the pressure distribution for the lower surface is shown in Fig. 21(b).

It is suggested in §2.3 that the injection of increasing quantities of air has this similarity to increasing Reynolds number because the disturbance to the laminar layer becomes sufficient to affect the stability of the separated, or vortex layer, and hence the extent of the separation, before it is sufficient to cause transition upstream of separation*. This suggestion is supported by the comparison in Fig. 22(a) of four pressure distributions obtained on an aerofoil for identical conditions except for changes in free-stream turbulence**. For the lowest turbulence the 'truncated peak', typical of separated laminar boundary layers, is very marked, but its extent and presumably that of the laminar separation decreases for increasing tunnel turbulence. A direct-shadow photograph indicated that the boundary layer was still laminar back to the shock for the case with highest turbulence (the pressure gradient in the local supersonic flow upstream of the shock is of course strongly fa-able).

The pressure distributions in Figs. 20(b) and (o) suggest that the changes in flow with increase in Reynolds number when the boundary layer is turbulent- at the interaction, and also the differences with laminar and turbulent boundary-layers when the Reynolds number is as high as 3.5×10^6, are small in comparison with the effect of increasing Reynolds number up to 3.5×10^6 when tho boundary layer is laminar (see Fig. 20(a)). This is in agreement with results on the flat plate (see above), and it has also been confirmed in flight[40] on a special smooth aerofoil built round the wing of an aircraft that there is little difference between the shape of the pressure distribution curves for laminar and turbulent layers at high Reynolds number (17 to 20×10^6, based on the chord of the special aerofoil);**** a typical comparison is shown

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*A similar change, but in the reverse direction, has been observed'5 as carborundum, stuck to the surface to cause transition, was gradually blown off by the airstream.

"These observations were made in the N.P.L. 20 in. x 8 in. High Speed Wind Tunnel, for which tha turbulence is now normally low, during an early investigation of inconsistencies which mere traced to dirtiness of gauzes and poorly designed. straightener vanes. The inference that the relevant change was in the free-stream turbulence was drawn from low-speed drag measurements at another incidence. These are shown in Fig. 22(b).

Transition was fixed by a wire ahead of the leading edge.

****This experiment is noteworthy for other reasons, namely, that with transition free the boundary layer remained laminar right back to the shock, that extremely small regions of laminar separation were then detected just upstream of tho shock but had no effect on the pressure distribution, and finally, that the profilo just upstream of the shock in a boundary layer made turbulent artificially was a typical "turbulent" profile (sco Fig. 20(d)).

in Fig. 20(d). It should be noted, however, in Fig. 20(c) and (d) that the pressure recovery downstream of the shock is more complete when the boundary layer is laminar up to the shock and in particular a higher pressure is reached at the trailing edge. Results at high Mach numbers for the case represented in Fig. 20(c) suggest that the difference tends to increase with increasing Mach number[38].

The most obvious danger of making tests at low Reynolds numbers, if the results are to be applicable at high Reynolds numbers, is the probable occurrence of extensive laminar separation upstream of the shocks. This could be avoided either by making the tests at Reynolds numbers above about 4×10^6 or by fixing transition forward of the shocks. We feel, however, that even at high Reynolds numbers it is important also to distinguish between results obtained with laminar and those obtained with turbulent boundary-layers at the interaction because the differences, although less obvious, are still very real for aerofoils and wings. Separation at the shock will occur at different free-stream Mach numbers for the two cases, for example, and will have different effects on the pressure recovery downstream (see Fig. 20(c) and (d))*. That these factors can be highly significant will be realized from the discussion in §7.

Since extensive areas of laminar boundary-layer flow are not yet attained in flight, the more important practical case is that with turbulent boundary layers, and wind-tunnel tests should therefore if possible be made with the boundary layers turbulent at the shock waves**. There should then be little effect of Reynolds number on the actual interaction or flow pattern; in particular, the occurrence of separation should not be affected (see 2.6). The pressure distributions in Fig. 20(b) suggest also that in some circumstances the pressure recovery downstream of the shock may not be greatly affected, but it is emphasized that there is little information on this important point which will be referred to again in §7.

As suggested by Young[41], the interest in results obtained with laminar boundary-layers may well increase when aircraft fly at high speeds at great altitudes, because the Reynolds numbers will be low and there will be a stabilizing effect on the laminar layers due to radiation from the wing and consequent heat transfer from the boundary-layer to the wing. Results with laminar boundary layers may be applicable to compressor blades and propellers. For any application it will always be necessary, however, to take the Reynolds number into account if it is less than about 4×10^6 and also stream turbulence and condition of the surface.

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*In an experiment described in Ref. 57, Gothert found that deliberate roughening of the surface on an aerofoil led to fairly large changes in the chordwise positions of the shock-waves. These changes were caused by a reduction in the pressure recovery downstream of the shook and hence in the pressure at the trailing edge position (see §7). It is not known whether the effect of the roughness was

6. Methods of Fıxıng Boundary-layer Transition

One diffıculty when tho models are small ıs to introduco a dısturbance sufficient to causo transition without interfering too seriously with the main flow*. It is also important that the profıle and thickness of the turbulent boundary-layer created artificially should differ as little as possible from those of the turbulent layer which would occur more naturally at higher Reynolds numbers. Miss Gamble[38] observed some unsatisfactory effects with threads on tho surface which could have been due to differences of this kind, but on the other hand there is considerable evıdence to show that similar results can be obtained for 'natural' and 'artificial' layers, (see for example, §2.5).

The method of injecting small quantıties of air into the boundary layer39 has many advantages for small aerofoils. This is now in use at the N.P.L. and ıs proving satisfactory provided care is taken not to inject much air in excess of the minimum required to fix transition. Its dısadvantages are that it complicates the construction of the model and that it is difficult to use it to fix transıtıon near enough to the nose for tests at hıgh incidences.

Other methods ınclude the stretching of a fine wire ahead of and parallel to the leading edge, a technique whıch avoids any disturbance on the surface but is dıffıcult to use for three-dimensıonal models; an area of deliberate roughness such as a layer of carborundum powder stuck to the surface, which however, tends to get blown off after a time; and strips, wires or threads protruding from the surface, which, on small models at least, tend to ıntroduce larger disturbances than some of the other methods.

There is need for further ınvestıgation of these techniques, in partıcular to discover whether the pressure recovery downstream of a shock-induced separation differs for layers in whıch the turbulence was created differently or at different chordwise posıtions.

7. Shock-induced Turbulent Boundary Layer Separatıon

7.1 Effects on the Steady Flow round Two-dimensional Aerofoils

Shock waves above a certaın strength cause separation of turbulent layers on aerofoils just as they do for flat plates (see above §2.6). Recent work at the N.P.L. on aerofoils with the boundary layers made turbulent upstream of the shock waves, including systematic pressure plotting and flow photography on three sectıons**, has led to a clearer understanding of tho consequences of such separation.

The most important effects of the stoady flow arc those on circulation"" and relative shook positions; the effects on shock-wavo structure and on surface pressures in tho immediate vicinity of the shock are less spectacular than for laminar separatıon.

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In the description of the effects on shock position it is convenient to consider the movement of a normal shock along a convergent-divergent nozzle for one-dimensional, inviscid flow, (Fy. 23(a)). In spite of the obvious over-simplification, the analogy between this and the flow in the stream tube adjacent to the aerofoil is useful qualitatively. Once the speed of sound has been reached at the narrowest cross section of the nozzle, a further reduction in 'exit pressure' leads to a region of supersonic flow downstream of the 'throat', which is terminated by a shock. The pressure distribution up to the throat is unchanged. The location of the shook is such that the pressure rise through it and tho subsonic recovery downstream load to tho required exit prossure, considered here to be the independent variable. The shock moves through the nozzle as the exit pressure is still further reduced, tho only change upstream being the expansion of the supersonic flow permitted by this movement.

The features of the flow about an aerofoil which resemble those in the hypothetical nozzle develop as the free-stream Mach number is increased for a fixed angle of incidence, and the ensuing discussion relates to such a case. The point at which sonic velocity is reached locally moves forward along the surface at first but soon tends to a limiting position near tho leading edge with only small subsequent movements. The supersonic flow upstream of the terminating shock then, correspondingly, changes only very little. It is found also that this shock moves along the surface as the pressure at tho trailing edge is reduced*.

The position of the shook in a nozzle (Fig. 23(a)) for inviscid flow with constant upstream pressure is determined uniquely by the exit pressure, and the rate of shock movement is therefore fixed for a given variation in exit pressure. If, however, viscous effects are present which influence the rate of pressure recovery downstream of the shock there will be a different rate of shock movement for the same variation in exit pressure. The reduction in the rate of pressure recovery due to a boundary-layer separation could be such an effect, and if. for example, separation occurred once the shock had reached a certain point (hence a certain strength), tho rate of movement of tho shock would thereafter be reduced in comparison to that which would have occurred for tho same variation of exit prossuro in the absence of separation (see Figs. 23(a) and (b)).

For an aerofoil one can consider the pressure at the edge of the boundary layer (or wake) at the trailing-edge position as being equivalent to the exit pressure in the nozzle, and it is horo that a further important condition must be considered. This is that the prossuro at the edge of the wake at tho trailing-edge position on one surface must be related to that at the corresponding position on the other surface. For the present qualitative discussion they can be considered to be equal**. This condition controls the inter-relation between tho two surfaces and explains

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*It is believed that further justification for the use of this analogy will follow from an analysis at present being made of results obtained with spoilers on an aerofoil. Spoilers of various heights were placed at various positions along one surface of an aerofoil and a preliminary examination suggests that, for a given incidence, tho position of tho shock on the other surface depended only on the pressure at the trailing-edge (with a small amount of experimental scatter).

**This is accurate if' there is no change of static pressure across either half of the wake, or if the changes across the two halves are equal. In practice the changes are probably relatively small and any differences even smaller.

why a change locally on one surface affects the flow about the whole aerofoil or, in more mathematical terms, affects the circulation on a circuit enclosing the aerofoil and shocks (see footnote on page 18). If, in the absence of shock waves, a change occurs on one surface tending to disturb the equality of pressures at the trailing edge, the equality is maintained by a change in circulation; for example, if the disturbance tended to decrease the pressure at the trailing edge on the upper surface the change in circulation would be in the direction to give decreased velocities (i.e., increased pressures) over the upper surface and increased velocities over the lower surface, i.e., a decrease in circulation. If the equality is disturbed when shock waves are present it is maintained chiefly by movements of the shocks, in the same way as effects on exit pressure would be compensated for in a nozzle, and the supersonic flow upstream of the shocks is only slightly affected*.

The pressure at the trailing-edge position clearly plays a vital part in determining the flow about the whole aerofoil and, before finally discussing the overall effect of turbulent separation, it is relevant to consider how this pressure depends on free-stream Mach number in the absence of separation and how it is affected by separation. It is found that the coefficient $C_{p_{T.E.}}$** is almost constant, possibly increasing slightly, for increasing Mach number up to a certain value, after which it begins to decrease more or less abruptly. This is illustrated in Fig. 24 where the scale of C_p decreases upwards, i.e., increasing velocity upwards. It is fairly well established (see below, §7.3) that the abrupt decrease starts at about the same Mach number as shock induced separation occurs on one surface or the other and is due to a change in pressure recovery between the offending shock and the trailing edge, this being less complete in the presence of separation. It seems reasonable to surmise that in the absence of separation the gradual and smooth variation in $C_{p_{T.E.}}$ would continue until the shock on one surface reached the trailing edge.

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*It should perhaps be noted here, although it will be referred to again later, that once the pressure at the trailing edge position on one surface or the other (and hence on both surfaces) has fallen to the sonic value, then the equality of pressure can be maintained by an abrupt expansion or shock at the trailing-edge itself (provided the wake is not too thick). Changes on one surface then no longer affect the flow on the other even if it is subsonic upstream of the trailing edge. The trailing-edge behaves, in fact, in the same way as it does for supersonic free-stream Mach numbers and is often referred to somewhat loosely, perhaps, as a "supersonic trailing-edge".

**$C_{p_{T.E.}} = \dfrac{p_0 - p_{T.E.}}{\frac{1}{2}\rho_0 q_0^2}$, where $p_{T.E.}$ is the pressure at the trailing edge and suffix 0 refers to conditions in the free stream. $p_{T.E.}$ is usually measured at a point in the trailing-edge of the actual aerofoil and may not therefore, represent the pressure at the edge of the boundary layer precisely.

Tho fact that the occurrence of turbulent separation disturbs the smooth variation of pressure at tho trailing-edge position, when considered in conjunction with tho two features previously discussed, namely, (i) the close connection bet&on shock movement and pressure at the trailing-edge position and. (ii) the inter-relation between tho two surfaces controlled by tho condition for equal pressures at tho two sides of the wake at the trailing-edge position, is believed to be tho key to the effects of the separation on tho steady flow about an aerofoil*.

Consider an aerofoil at lift; separation will then usually occur first on the upper surface because the local Mach number is higher thoro. As the Mach number is increased, the development of the separation reduces the recovery of pressure downstream of the shock on that surface, thereby tending to give a lower pressure at the trailing edge than would otherwise have existed. The condition of equality at the trailing edge is maintained by (i) a slowing up of tho movement of the shock on tho upper surface, much as was described above for separation occurring in an isolated nozzle with a fixed rate of decrease in exit pressure (see Figs. 23(a) and (b)), accompanied by (ii) an acceleration in the development of the flow on the lower surface, much as would occur in a nozzle for a simple acceleration in the rate of decrease in exit pressure (this would affect the rate of decrease in pressure over the whole surface, or whole nozzle, when there were no shocks on the surface, and later, the rate of movement of the shock itself).

These effects are evident (together with other well-known compressibility effects) in the pressure distributions illustrated in Fig. 25 for the 6% thick RAE.104** aerofoil at 2° incidence and in the corresponding direct-shadow photographs in Fig. 26. The main shock on the upper surface starts, at a free-stream Mach number, M_o, of between 3.75 and 0.8, to move rearwards along the surface from the point very near the nose at which it was first formed***. The supersonic flow extends with this movement but also undergoes a change near the leading edge; tho small peak with fairly high local Mach numbers and the oblique waves gradually disappear. This is associated with a change in the amount of supersonic expansion around tho nose, which is very sensitive

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*The effects of separation at low speeds (i.e., not shock-induced separation) can be considered qualitatively in an essentially similar manner, except of course that thoro is no parallel to the relation between shock position and trailing-edge pressure, and that the changes occur with increasing incidence instead of with increasing Mach number. It is of historical interest that the terms "shock-stall" and "compressibility burble" were introduced, and for a time used, to describe the onset of adverse compressibility effects, presumably because the drop in lift was one of the most typical and severe features and, for tho fairly thick sections then current, Was encountered at an early stage.

""This section was chosen for the illustration because, although tho effects of separation are less drastic than on the thicker ones and occur later, the shook moves some way along the surface before separation occurs and the effects are thereforo shown in better contrast.

***The speed of sound was first exceeded locally at $M_o = 0.59$.

to small movements of the sonic point[43], and possibly also with a very small laminar-separation bubble*, but does not, we believe, greatly effect the present argument. For M_o of 0.82 and above, the pressure at any point on the surface between 0.2 chord and the main shook varies little and the qualitative analogy with the nozzle can be employed from about 0.2 chord onwards.

With M_o increasing from 0.80, the main shock is at first a single abrupt compression normal to the surface. It moves along the surface at a fairly regular rate and at $M_o = 0.86$ has reached a position at about 0.65 chord, with a local Mach number of 1.27 just upstream. It has a small bifurcated foot which is the first sign that the turbulent boundary-layer is separating, at least locally in the immediate vicinity of the foot. Thereafter the separation develops rapidly as indicated by the extending bifurcation and the changes in pressure recovery between the shock and the trailing edge; the characteristic fall in trailing-edge C_p starts immediately after $M_o = 0.86$, (Fig. 27). The consequent deceleration in the movement of the upper surface shock is obvious between $M_o = 0.88$ and. 0.92. The anticipated effects on the lower surface are also evident, namely the acceleration in the development of the flow over the whole surface at first, leading to local supersonic velocities and a shock which moves rapidly downstream. The change on the lower surface while the flow there is still subsonic is shown, Fig. 27, by the variation in the pressure coefficient for a representative point, $x/c = 0.48$, which follows closely the Kármán-Tsien law for inviscid flow up to $M_o = 0.86$ but then drops more rapidly. The marked relative movement of the shocks on the two surfaces is shown by Fig. 28, in which their chordwise positions** are plotted against M_o.

Between $M_o = 0.93$ and 0.94 the upper surface shock again moves very rapidly. It will be noticed that at $M_o = 0.94$ the lower-surface shock is at the trailing edge and sonic velocity is exceeded there, so that, as explained in the footnote on page 20, the condition for equal pressures on the two *sides* of the wake at the trailing-edge position then no longer controls the inter-relation between the two surfaces. The reason for the occurrence of the rapid movement of the upper surface shock just before this condition is reached is not fully understood, but it might be associated with the fact that once the shock has reached 0.78 chord the local Mach number upstream no longer increases with further movement but decreases slightly.

Separation does not occur on the lower surface for any Mach number in the example chosen, but does of course in some cases. This would lead to a re-acceleration of the movement of the upper surface shock and a relative deceleration of the lower surface one.

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*These phenomena occur often for leading-edges of small radii and are the subject of a special study because they are thought to play an important role for thin aerofoils, especially at higher incidences.

**The position of the shock is taken to be the forward-most point of tha 'toe' at the surface; the main re-compression occurs at or just downstream of this point.

7.2 Effects on the Force and Moment Coefficients for Two-dimensional Aerofoils and Aerofoils with Controls

The effects described in the preceding section, particularly the relative movements of the shocks and the associated development of the regions of low-pressure supersonic flow, have serious repercussions on the forces and moments. The variation of lift coefficient and $-chord pitchmg-moment coefficients for the case described (6% RAE.104 at 2° incidence) are shown repectively in Figs. 29 and 30.

C_L Fig. 29(a), rises smoothly right up to $M_o = 0.86$, the Mach number at which separation was first observed, and then falls, fairly slowly at first, but more rapidly between 0.90 and 0.93. The fall is associated with the slowing up of the movement of the shock on the upper surface and the more rapid development of the flow on the lower surface. When the upper surface shock moves more rapidly again, between Mc = 0.93 and. 0.94, C_L recovers from a trough and then, with both shocks at the trailing edge, shows the beginning of a gradual fall. This fall would be expected to continue smoothly right through M_o = 1.0 and beyond because there can be no further shock movement or change in shape of tho pressure distribution[44].

C_m, Fig. 30(a), shows little variation until M_o = 0.82 at which it starts to fall, i.e., nose-dorm tendency. This change is not due to separation but to the "natural" movement of the upper surface shock aft of the $\frac{1}{4}$-chord point. The effect of separation is not noticeable until just after M_o = 0.88 when there is a violent nose-up change. This again is undoubtedly due to the relative shock movements. The curve suffers another abrupt change, back to a nose-down tendency, at the same Mach number, 0.93, as the lift recovery occurs and is due to the re-acceleration of the uppor-surface shock. After this, the beginning of the gradual supersonic variation is shown.

The integral of tho pressure coefficient $\frac{p_o - p}{\frac{1}{2}\rho_o q_o^2}$ from the leading edge to the trailing edge has been found for each surface separately, Fig. 29(b), to demonstrate how the changes on the separate surfaces are contributing to the changes in C_L (C_L is the difference between corresponding ordinates). Both show a definite change in trend, contributing to the drop in C_L, at the Mach number at which separation was first observed. The trough in the curve for the upper surface between M_o = 0.90 and 0.925 is most marked.

The integral $\int_0^1 \left(\frac{p_o - p}{\frac{1}{2}\rho_o q_o^2}\right)\left(\frac{1}{4} - \frac{x}{c}\right) d\left(\frac{x}{c}\right)$ has similarly been found for each surface, Fig. 30(b), to indicate the separate contributions to C_m. Again, both show a change in trend after separation occurs, although the change on the upper surface is delayed, presumably because tho reduced pressures between the shock and the trailing edge compensate to some extent for the slowing up of the shock. Tho changes on both surfaces contribute to the nose-up tendency between M_o = 0.88 and 0.93.

continued, the curves would necessarily have suffered further abrupt changes and would have in fact been similar to those for the other sections through the whole Each number range. The effects occur earlier for the thicker sections and are more violent*. The influence of section shape is discussed further in f17.5, and it is sufficient at this point to note how the abrupt changes can be correlated with the occurrence of separation on the upper surface and its effect on shock mwement. The positions of the shocks for the two 10% sections are given in Fag. 32 (those for the 6% RAE.104 section are plotted in Fig. 28). The characteristic slowing up of the upper surface shook occurs in every case immediately after separation is first observed and CL begins to fall also. The abrupt nose-up tendency is delayed a little and in fact bears the same relation for all three cases to the first occurrence of the lower surface shock. A further point of interest in the curves for the RAE.102 section is that the rc-acceleration of the upper-surface shook and the associated recovery of lift and change back to nose-down tendency on Cm all occur immediately after $M_o = 0.88$, the Mach number at which separation was first observed on the lower surface.

The effects of turbulent separation on CL and C_m have so far been described in relation to their variation with Mach number at constant incidence. They clearly must also be felt on the variation of these quantities with incidence at constant Mach number; of Cm with C_L at constant Mach number; and of Cm with Mach number at constant C_L. This is illustrated for the 6% thick RAE.104 section in Fig. 33(a), CL vs a for constant M; Fig. 33(b), Cm vs CL for constant M; and Fig. 33(c), C_m vs M at constant CL. The effect of increasing Mach number in the absence of separation is to increase $\frac{\partial C_L}{\partial \alpha}$, Fig. 33(a), and to increase the nose-down tendency of the Cm vs CL curves, Fig. 33(b). The opposite effect in the presence of separation can be seen to spread over thd incidence or CL range as the Mach number is raised to 0.92, for which value separation is present even at 0° incidence (zero lift).

As far as is known no very large or very direct effects of shock-induced turbulent separation on drag-coefficient have been observed. C_D starts to rise rapidly due to the shock drag before separation occurs and continues to do so at about the same rate after separation. This is because any increased boundary-layer losses are compensated for by reduced shock losses, due to the slowing-up of its movement and possibly to its bifurcation[2]. In terms of pressure, or form drag, the shock losses can be considered to arise from the reduction of pressure at points downstream of that at which the surface is tangential to the free-stream direction as the supersonic flow develops rearwards, and the boundary-layer losses to arise from the reduction of pressure behind the shook due to incomplete pressure recovery. The variation of the form-drag coefficient for the 6% RAE.104, 2° incidence (see Figs. 25 and 26) is shown in Fig. 34. For this case, there is in fact a reduction in the rate of rise of drag when separation first occurs, and an increase again, between $M_o = 0.92$ and 0.94, when the upper surface shock re-accelerates. (The form-drag coefficient reaches a maximum when both shocks reach the trailing edge and thereafter settles down to the

The close connection between shock-induced separation and the almost complete loss of lift due to a flap-type control at a fixed setting was established for an example described in Ref. 42 (separation also caused a reversal in the sign of the hinge moment). A reversal in the effect of the flap has since been found at a higher aerofoil incidence when the separation was more severe. The mechanism producing the drop in lift, or effectiveness, is the same as on the plain aerofoil and the troughs in the curves have the same appearance. For the deflected flap, however, the adverse effect of separation on the relative shock movements is amplified when the flow at the hinge becomes supersonic; on the surface with the concave corner (lower surface) the compression at the hinge accelerates the movement of the terminating shock for a given rate of decrease in trailing-edge pressure, whereas on the surface with the convex corner the expansion at the hinge decelerates tho movement of the shock.

The additional effect due to the supersonic flow at a hinge does not, of course) occur for spoiler-typo controls and they therefore have certain advantages in the transonic régime. It has been found recontly, however, that extensive shock-induced separation upstream of a spoiler can seriously reduce its effectiveness for small heights, because it is then operating in a dead-air region.

7.3 Conditions for the Occurrence of Separation and Possibilities of its Prediction

The work on flat plates has shown that, for any given upstream Mach number, a shock wave causes separation if the strength of, or the pressure-ratio across the shock exceeds a certain value; for turbulent layers this value depends only on the Mach number, as expressed above in equation 2 (see E12.6). On aerofoils. where, until separation occurs, the shock is normal or nearly so, the pressure ratio and the local Mach number immediately upstream of the shock are related by conditions for flow through a normal shock; theoretically this is a unique relationship. Thus, if the work on flat plates is applicable, it should be possible to define the conditions for turbulent separation on an aerofoil by either the pressure ratio across the shock or the local Mach number just upstream. The upstream Mach number is more convenient in practice because the magnitude of tho pressure rise across the shock is not always easily determined with tho limited number of pressure readings available for small aerofoils. Further, tho Mach number is likely to lend itself more readily to the prediction of separation.

A certain kink in the surface-pressure curve for the flat plates was used as an 'indicator' to determine the presence of separation*. This fails for aerofoils partly because the downstream variation in pressure is very different and partly again because of the poor definition of the pressure curves with the few points available. In the following analysis and discussion the presence of bifurcation at tho foot of the shock has been used instead. The reliability of this criterion has been checked for flat plates by surface tube readings in the work of Fage and Sargent[2] and in more recent work at the N.P.L., and for aerofoils by the presence of reversed flow in oil on the surface in work at Vickers Armstrong Ltd., (Weybridge)[45] and at the N.P.L. It is possible that the separation does not always immediately extend to the trailing edge of the aerofoil and that the layer sometimes reattaches downstream of the foot of the shock. Ackeret[3] found, for example, no evidence of

separation/

*The reliability of this indication was checked by surface-tube measurements.

separation downstream of a shock with a small bifurcated foot. Again, the variation of trailing-edge pressure is not affected until just after bifurcation is first observed, Fig. 24*. If such delays occur they are, however, very small.

For low aerofoil incidences, the Mach number upstream of the shock is the maximum local Mach number on the surface. This maximum local Mach number has been plotted in Fig. 35 against free-stream Mach number for both surfaces of the 10% RAE.102 section for each of three incidences. The value of the local Mach number is indicated by a filled-in symbol when the shock was bifurcated. Bifurcation, or separation, occurred for this aerofoil when the local Mach number exceeded about 1.23**. A large number of tests have been made with spoilers and flaps on this same section. Observations of separation are available for these and are presented in Figs. 36 and 37. For each different configuration and incidence, and for both surfaces, two points have been plotted in Fig. 36. One showing the highest local Mach number observed with no separation (unfilled symbol) and the other the lowest local Mach number (immediately upstream of the shock) for which bifurcation was observed (filled-in-symbol). Results obtained at N.P.L. on 6% and 10% RAE.104 sections are also included. This presentation shows that separation always occurred if the local Mach number was above about 1.25 but never if it was below about 1.22. It suggests that there is an effect of free-stream Mach number on the local Mach number for separation, the latter varying from about 1.26 at a free-stream Mach number of 0.7 to about 1.22 at a free-stream Mach number of 0.9. The pressure-ratios across the shock just before and just after separation*** have been plotted in Fig. 37 in a similar way and for the same results. This suggests that separation occurs when the pressure ratio exceeds 1.40 and that there is little effect of free-stream Mach number on this value.

It is of interest to examine how these results compare with (i) expressions for the pressure ratio for separation on flat plates and (ii) observations in other experiments on aerofoils and wings.

(i) <u>Comparison with Expressions for the pressure-ratio for Separation of a Turbulent Layer on a Flat Plate</u>

The pressure ratio for separation $\frac{p_s}{p_1}$ is plotted in Fig. 38 against the upstream Mach number M_1, as derived from equation 2 (see page 10) and from that suggested in Ref. 16. Also plotted is the theoretical pressure ratio, p_2/p_1, across a normal shock. The

intersection/

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*Observations of shock-induced separation in flight[46] indicated a gradual spread of the extent of separation from the shock position to the trailing edge as the flight Mach number was increased.

**Once the shock waves had reached the trailing-edge, the extent of the intersection of this curve with the others would suggest separation for a normal shock with upstream Mach numbers of 1.122 and 1.129 respectively for equation 2 and the R.A.E. formula, values somewhat below those noted above. On the other hand, the pressure ratio observed on the surface of aerofoils for upstream Mach numbers of 1.25 is only 1.40 as against 1.65 expected theoretically. In fact, the value 1.40 observed for separation on aerofoils agrees fairly well with that predated by the 'flat-plate' formulae at the appropriate Mach number, about 1.25.

The discrepancy between observed and shock-theory pressure ratios for normal shocks on aerofoils arises because the pressure downstream of the shock is lower than would be expected theoretically. Two factors probably contribute to this, namely, the change in displacement thickness of the boundary layer even in the absence of separation and, probably more important, the relatively abrupt expansion which immediately follows the normal shock*. As suggested by Ackeret et al[3] this expansion results from the decreasing strength of the shock with distance above the surface (it decreases to 1.0 at the edge of the supersonic region). The decreasing strength tends to give decreasing pressures along a normal to the surface immediately downstream of the shock. The expansion which occurs near the foot can be considered as part of the re-adjustment to a normal gradient with increasing pressure outwards, a re-adjustment which, again as shown by Ackeret et al, occurs very rapidly. This explanation could account for the apparent anomally that the pressure ratio for separation is, as far as can be seen, almost unaffected by free-stream Mach number whereas the upstream Mach number for separation increases as the free-stream Mach number is reduced. The situation most probably is that separation occurs for a certain pressure ratio, but that, as the free-stream Mach number is reduced, the upstream Mach number required to give this ratio increases because the height of the shock becomes smaller and therefore the gradient in its strength steeper.

(ii) Comparison with Other Results on Aerofoils and Wings

Conditions for which turbulent separation occurred in a number of other experiments have either been noted or can be deduced from the results. These are listed in Table I.

Table I/

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*This expansion shows up in the photograph reproduced in Fig. 19(b) as a dark patch immediately following the shock.

Table I

Other Results for the Occurrence of Turbulent-separation Induced by Normal Shocks

References	Type of Experiment	Reynolds Number (based on representative length)	Natural or Artificial Turbulent Layer	Free-stream Mach number	Method of Deducing the Presence of Separation	Local Mach Number for Separation	Pressure Ratio for Separation	Remarks
Fage and Sargent (Ref.2)	Flat plate on a tunnel wall	(very thick layer)	Natural		Bifurcation and surface-tube measurements	Not measured	1.8(1)	Possible effects from moist air in the tunnel
Ackeret, Feldmann and Rott (Ref.3)	Plate parallel to wall of curved tunnel	(i) 2.7×10^6	Natural	-	Bifurcation and pressure recovery (i.e., decrease in $C_{PT.E.}$)	Between 1.26 and 1.31	Between 1.50 and 1.55	Possible effects from moist air in the tunnel
		(ii) 1.69×10^6	Artificial (produced by wire)	-	Bifurcation (Boundary layer profiles indicate no separation at a distance downstream)	Present at 1.28	Present for 1.55	
		(iii) Thick tunnel-wall layer	Natural	-	Bifurcation	Present at 1.30	Present for 1.45	
Liepmann (Ref.4)	Aerofoil	0.9×10^6	Artificial	0.86 - 0.89	Bifurcation	Between 1.21 and 1.31	Between 1.30 and 1.48	

Table I (continued)/

Table I (continued)

References	Type of Experiment	Reynolds Number (based on representative length)	Natural or Artificial Turbulent Layer	Free-stream Mach number	Me Ded Pre Se
Vickers Armstrong Ltd (Weybridge) (Ref.45)	'Bump' on a flat plate	3.1×10^6 (approx.)	Natural		Bif and oil
Wood and Gooderum (NACA TN.2801) (Ref.47)	'Bump' on a flat plate	0.6×10^6	Artificial (by two methods - wire ahead and plate ahead)	About 0.85	Bif
Zalovcik and Luke (NACA RM. No.L8C22) TIB/1865 (Ref.No.6)	Aeroplane in flight (wing specially modified)	Full scale	Natural	0.69 to 0.71	Bou lay pr and

Table I (concluded)

References	Type of Experiment	Reynolds Number (based on representative length)	Natural or Artificial Turbulent Layer	Free-stream Mach number	Method of Deducing th Presence of Separation
Cooper and Bray (NACA RM No.A51G09) TIB/2760 A.R.C.14,485 (Ref.49)	Aeroplane in flight	Full scale	Natural	0.68	Bifurcation (schlieren photograph
Hollingbourne and Pindar R.A.E. Tech Note No. Aero.2221) (Ref.50)	Half-model of a sweptback wing	1.75×10^6	Artificial (thread on surface)	(a) 0.885 (b) 0.865 to 0.885	Pressure recovery (i.e., decrease i $C_{pT.E.}$) an shock movement

(1) The value 1.8 was determined theoretically from measured shock angles for a pos 'kink' pressure to upstream pressure is taken from the curve of surface pressu done[11,12], then a value 1.51 is obtained which agrees well with equation 2 (pa number, 1.4 (see Fig.38).

(2) This pressure ratio was determined, at free-stream Mach numbers above that at wh pressure ratio across the front branch of the bifurcated wave.

The values found for upstream Mach number and pressure ratio for separation are all consistent with those indicated in Figs. 36 and 37 with two exceptions, namely the pressure ratios observed in the first two experiments listed, for both of which there are some doubts about the effects of moist air in the tunnel. The correlation with other work is therefore good and it is noteworthy that this embraces finite wings and aircraft in flight. The last entry in Table I is of special interest because not only does separation occur on the model of this fairly highly-swept wing (35° at the $-chord position) for the same conditions as on a two-dimensional aerofoil, but the characteristic effects on shock movement are present both for the model and for the full-scale aeroplane in flight (see Fig. 39).

It is thus fairly well established, at least for low angle of incidence, that shock-induced turbulent separation will occur both for model and full scale on two-dimensional aerofoils or straight wings when the local Mach number just upstream of the main shock exceeds a value varying approximately linearly from 1.26 at $M_0 = 0.7$ to 1.22 at $M_0 = 0.9$. It seems that the same criterion will. apply for swept wings if the component local Mach number normal to the shock front. is used.

It is at present impossible to predict the free-stream Mach number and lift coefficient for which this condition would be reached on any particular design because only very few section velocity-distributions have been measured with the boundary layers turbulent upstream of the shocks and because there is little information on how to apply such data to find local Mach numbers on finite wings; little help can be expected from theory in either respect.

The free-stream Mach number for which separation occurs, determined from the local Mach number criterion, is plotted against incidence in Fig. 40 for the three two-dimensional aerofoils recently tested at the N.P.L. Also plotted are the Mach number at which the rapid fall in $C_{p_{T.E.}}$ begins*, the Mach number for lift divergence*, and the Mach number at which the peak occurs in the C_L versus M curve at constant incidence. Any of these 'criticals' could be used to predict reasonably well the occurrence of serious effects of separation. The rapid change in $C_{p_{T.E.}}$ may well be more appropriate than the others if our argument is correct that the disturbance to the equality of pressure at the trailing edge is the main cause of most of the adverse effects of separation. The Mach number for change in $C_{p_{T.E.}}$ seems to agree well with that for the peak in the lift curve.

A convenient procedure for predicting the occurrence of separation under full-scale conditions might be to measure the variation of $C_{p_{T.E.}}$ on a model with turbulent boundary layers. It is obviously desirable, however, to augment the sparse information available for turbulent layers by making more complete pressure plots wherever possible.

Two cases for which prediction is more difficult and which have so far not been considered should perhaps be mentioned at this stage, namely, (a) the type of flow with high local Mach-numbers very near the leading edge and (b) the effects of interference at wing-body and other intersections.

(a)/

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*Defined by the intersection of tangents to the two branches of the curve.

**Defined as the Mach number at which the rate of rise of C_L with Mach number (constant incidence) begins to fall.

(a) Local Mach numbers considerably in excess of 1.25 often occur very near the leading-edges of aerofoils at high incidences without causing turbulent separation (see for example Fig. 25). A partial compression through an oblique shock occurs instead and the Mach number upstream of the main shock is then somewhat less than the maximum value on the surface. The lack of information for this type of distribution is even more serious than for the more normal type where the local Mach number rises monotonically to its maximum value fist upstream of the shock. It is known that it occurs at increasingly lower incidences as the section thickens (and hence leading-edge radius) is reduced. The chief uncertainties are the strength of the oblique shock and whether this is affected by the state of the boundary layer right near the nose or by Reynolds number.

(b) The super-velocities induced at a junction would lead to high local Mach numbers and therefore possibly to earlier separation locally. For example, separation on the wing of a Meteor aircraft was observed to originate from a wing-nacelle junction and to spread outwards over the wing*.

7.4 Consequences in Transonic Flight

There CM be little doubt that many of the undesirable aerodynamic phenomena encountered by present-day aircraft in transonic flight are, if not actually caused by shock-induced turbulent separation, at least aggravated by its occurrence. This would be anticipated from results on two-dimensional aerofoils and model wings, and has been confirmed in a number of flight tests.

7.4.1 Steady-flow Phenomena

Changes of stability and trim must of course occur in transition from subsonic to supersonic flight but could, we believe, be more progressive and much less violent if the 'natural' movement of the shock waves along the wing chord was not disturbed by separation. The reversals in the trend of section pitching-moment variation with increasing Mach number (see Figs. 31 and 33(c)) could, for example, cause corresponding sudden changes in trim. Loss of lift first on the outboard portion of a swept-back wing could also lead to changes in trim and even to longitudinal instability. Again loss of normal force due to separation on tailplane or fin could affect longitudinal or lateral trim and stability.

Wing dropping could quite well be caused by the onset of separation effects earlier or more severely on one wing than on the other, such as might occur in a manoeuvre. (In view of the importance of flow conditions at the trailing edge and the large effect which can be produced by a small spoiler there, very small discrepancies between the shape or condition of the trailing edges of the two wings might also be important in this respect.) In the special tests on an aircraft described in Ref. 46 separation occurred on the upper surface of a 'glove' over part of one wing but not on a corresponding glove, slightly different in profile, on the other wing; although only 20%

been suggested that, at least in some instances, wing dropping occurs beoause of an abrupt loss or maybe even reversal of aileron effectiveness. If not the actual cause of the wing dropping, simultaneous loss of lateral control certainly would greatly increase it3 seriousness. In either event the basic 'ill' is most probably turbulent separation.

Ground-launched rocket tests on the effectiveness of flap-type controls provide an important link between tunnel and aeroplanes in flight. They provide, on the one hand, results which agree well qualitatively with corresponding tests in wind tunnels and, on the other, results on model wings which confirm flight experience. The trough in the effectiveness versus Mach number curve is a characteristic feature of many results in both categories, a feature which has been shown in tunnel tests definitely to be associated with separation[42]. It is most probable also that separation adversely affects hinge moments. Some effects must also be expected on the behaviour of other types of control such as all-moving wing tips and spoilers, especially on thicker wing sections where the separation is likely to occur earlier and to produce more violent changes (see below).

Although there is as yet little information on the effects of shock-induced turbulent separation at high CL, it is likely to affect $C_{L_{max}}$ and may also be responsible for the *instability boundaries' which sometimes restrict the usable CL to a value below $C_{L_{max}}$.

7.4.2 Unsteady phenomena

Few measurements have been made of the flow fluctuations near an aerofoil moving at high speed, and. the data which are available are either for sections which are very thick by modern standards, or for low Reynolds number and laminar boundary layers. It is known, however, that when separation occurs at shock waves formed on an aerofoil moving at high subsonic speeds there may be fluctuation3 of total head and flow direction in the wake, and that there are corresponding fluctuations in the shock-wave position and pressure distribution on the aerofoil. These shock-wave movements may be considered to arise as a consequence of the fluctuations in the circulation round the aerofoil which occur when eddies are shed into the wake, or alternatively, as a consequence of pressure fluctuations at the trailing edge (see section 7.1).

It is thought that the buffeting frequently observed in high-speed flight is associated with these fluctuations; it has certainly been correlated with the occurrence of separation on the wings of 3 number Of aircraft as reported, for example, in Refs. 46, 49 and 51. Since the fluctuations behind the aerofoil are found to be confined to the mean boundaries of the wake, and these are usually arranged to clear the tailplane it seems that the pressure fluctuations on the wing (arising from the'shock-wave movements) account for most of the effect. Much more work is needed, especially observations of shock-wave oscillations, before the details cm be understood, but the following speculations which are based on work at the N.P.L. may be of interest.

It has been seen in section 7.1 that, after separating near a shock wave, the boundary layer on an aerofoil moving at high subsonic speeds sometimes reattaches to the surface and sometimes remains separated over the whole of the rear of the aemfoil. When reattachment does not occur the beginning of the wake contains two vortex sheets, one springing from the separation point on each surface. The spacing between the sheets depends on the positions of the separation points; for example, on an aerofoil at incidence and at a Mach number not too far above the critical the separation point on the lower surface may be at or near the trailing edge, and that on the upper surface may be at a shock wave located a short distance behind the position of

maximum/

maximum thickness. The spacing my, therefore, be quite large (e.g., of' the order of the aerofoil thickness) and the initial conditions in the wake thus resemble in some ways those behind a bluff body. In many cases it is found[52] that the vortex sheets formed behind an aerofoil moving at high subsonic speeds develop into a quasi-periodic type of flow not unlike a Kármán vortex sheet. This is illustrated in Fig. 41 which shows the flow round a 10% thick aerofoil at 2 deg incidence and M_0 = 0.82. This type of pattern is observed both when the boundary layer ahead of the shock is laminar and when it is turbulent (as in Fig. 41), and it is thought that for turbulent layers the turbulence in the wake near the trailing edge is of such small scale (having a length scale characteristic of the boundary layer on the aerofoil) that it has little effect on the wake, which develops its own turbulence (on a larger scale) by building up an instability of the type shown. The periodic type of flow pattern is not usually steady but occurs in 'bursts'; that is a photograph taken at one instant shows the periodic pattern whereas it is not visible in a photograph taken shortly afterwards. When periodic flow occurs, the vortex spacing ratio and frequency parameter appear to be of the same order as at low speeds.

The above is merely an example of a type of flow which could clearly lead to buffeting, and it is not suggested that buffeting will occur only when the flow is of this type. Even when the flow is not periodic, buffeting may be excited by unsteadiness in the wake and the associated fluctuations in the pressures on the aerofoil; the details must depend on many factors including the natural frequencies of the aeroplane. It is felt, however, that disturbances of a scale which is sufficiently large to have important effects are most likely to arise from the instability of vortex sheets formed as a result of separation, and that buffeting is unlikely to occur if' separation is absent. It is known[53] that at low speeds the scale, frequency and longitudinal spacing of the eddies in the wake depend on the initial lateral spacing of the vortex sheets, and on this basis it would be expected that the conditions would be most severe when separation occurs fairly well forward on the aerofoil. The situation envisaged. for a moderately thick aerofoil at low incidence is thus as follows. At free-stream Mach number below the critical there is no separation; this type of flow persists at slightly higher Mach numbers *because,* although a shock wave is present a little behind the maximum thickness position on the upper surface, it is not strong enough to provoke separation. At a higher Mach number the shock is still well ahead of the trailing edge, but is strong enough to cause separation. The spacing of the vortex sheets is thus large (see, for example, Fy. 41) and large-scale disturbances may arise and cause buffeting. As the Mach number is further raised the shock wave moves downstream so that the vortex-sheet spacing and the scale of the disturbances is reduced. When the shock reaches the trailing edge the disturbances are probably of a very small scale, and since, for a two-dimensional aerofoil, this condition is achieved at a free-stream Mach number below unity it night be expected that with zero or small sweep-back buffeting will cease before M = 1 is reached. As far as is known, this explanation is in qualitative agreement with observation.

It is known that the unsteady flow in the wake is a source of weak shock waves which are propagated upstream towards the aerofoil. This is particularly clear in Pig. 42 which shows the flow past an aerofoil with a blunt trailing edge when the Mach number is just below the critical. It may be seen that the vortex sheets from the two corners of the blunt trailing edge form a regular 'street' of eddies and that associated with these is a pattern of wavelets moving upstream. At Mach numbers above the critical the wavelets cannot pass upstream through the region of supersonic flow near the surface of the aerofoil, and hence form an envelope at the shock wave at the rear of tho supersonic region. Further out from the surface the wavelets

may/

may usually be seen[54] to pass upstream round the outside of the supersonic region, and sometimes to cause disturbances at the sonic line which propagate downwards into the supersonic region. This behaviour is observed on almost all aerofoils (whether the trailing edge is blunt or not) at least when the shock waves are strong enough to provoke separation. An example showing the wavelets on an aerofoil moving at a Mach number well above its critical is reproduced in Fig. 43. Although the upstream movement of wavelets from the wake provides a plausible explanation for the formation of the shock wave at the rear of the supersonic region, it is not yet clear whether or not they have an important effect.

In many cases small-scale eddies are observed near the surface of an aerofoil close to the point where separation occurs at a shock wave. These presumably arise from the instability of the vortex sheet formed at separation (i.e., without involving the interaction between the vortex sheets formed on the upper and lower surfaces). It is possible that the scale of disturbances of this 'type may be too small to cause buffeting, but they may, perhaps, have some connection with the phenomenon of control-surface 'buzz'. During tests at the N.P.L. on a 10% thick RAE. 102 section with a plain flap at the trailing edge, the position of the shock on the lower surface was observed, at one free-stream Mach number, to be very unsteady and to fluctuate between the hinge position and the trailing edge. A change in sign of the loading on the flap occurred. in steady flow for this shock-wave movement, and such unsteadiness could clearly lead to 'buzz' of a flap with freedom to oscillate.

The above remarks are largely confined to the unsteady flow which may result from shock-induced separation on a rigid aerofoil. The interaction between shock waves and boundary layers undoubtedly also has large effects on the characteristics of oscillating aerofoils as used, for example, in the measurement of pitching-moment damping derivatives. For low values of the frequency parameter, some information on these effects may be found. in Ref. 55.

7.5 The Influence of Section Shape

So far only a few section data have been obtained with turbulent boundary layers upstream of the shocks and such remarks as are possible on the influence of section shape are based mainly on fairly general considerations, with a few specific examples from the results for the sections tested at N.P.L.

Shock-induced turbulent separation would be avoided if the local Mach number upstream of the main shock could be kept below about 1.22 until the shock reached the trailing edge. To achieve this, the amount of supersonic expansion downstream of the sonic point must be kept small, i.e., small curvature, a condition which is fulfilled on thin aerofoils at low incidence. For example, separation occurs for 1° incidence at $M_0 = 0.817$ for both 10% RAE. 102 and 10% RAE. 104 sections but not until $M_0 = 0.880$ for the 6% RAE. 104 (Fig. 40). At higher incidences where the sonic point is extremely near to the leading edge, high local Mach numbers often occur for thin sections also and the comparison may become less favourable. This tendency is already beginning to show in Fig. 40; for the 10% and 6% RAE. 104 sections at 3° incidence, the values of M_0 at which the local Mach number for

(i) Pressure Recovery.- For present purposes the pressure recovery from just upstream of the shock to the trailing edge can be divided into (a) the recovery through the shock itself and (b) the recovery downstream of the shock. Just how important the latter can be is illustrated by the comparisons in Figs. &(a), (b) and (c) of pressure distributions at each of three free-stream Mach numbers for the 10% RAE. 104 and 102 sections at 2° incidence. Separation occurs at about the same Mach number for the two sections but the effects on C_L and Cm are much more violent for RAE. 104 (see Fig. 31).

The RAE. 104 section was designed to have its minimum pressure at 0.6 chord for low C_L compared with 0.4 chord for RAE. 102. The result of this is reflected in the subsonic pressure distribution, Fig. 44(a), in that the rate of pressure recovery over the rear of the aerofoil is more rapid; the maximum thickness occurs further aft of course and the trailing-edge angle is larger.

Fig. 44(b) shows the pressures obtained just after the onset of separation at practically the same free-stream Mach number for the two sections (0.794 and 0.795) and with the same local Mach number just upstream of the shock on the upper surface, namely, 1.27. The rate of pressure recovery downstream is as yet hardly affected and is for both practically the same as at M_0 = 0.71 (see also curves of $C_{p_{T.E.}}$, Fig. 24). The values of trailing-edge pressure are identical. The comparison of pressures on the lower surface is very similar to that for M_0 = 0.71.

The pressure distributions in Fig. 44(c) are for a still higher free-stream Mach number (0.853 and 0.855). The effects of separation are now quite prominent (Fig. 31), especially for the 104 section for which C_L has fallen considerably and C_m shown a violent nose-up tendency. The upper surface shocks are in the same relative positions as in Fig. 44(b), both having moved rearwards by about 0.03 chord only (the Mach number upstream has increased, if anything, slightly more for the 102 section than for the 104). The variation of trailing-edge pressure has been disturbed for both sections (see Fig. 24) but more for RAE. 104 than for RAE. 102. Thus the pressure at the trailing edge of the 104 section is now lower than that for the 102 section. This is an important difference and arises because the rate of recovery is now less rapid for the 104 section than for the 102 instead of more so as at earlier Mach numbers. It produces a spectacular effect on the lower surface, where for the 104 section there is now a fairly large supersonic region with the shock well aft; that for the 102 section is only just forming. This large relative change on the lower surface must be the immediate cause of the difference in magnitude of the effects on C_L and Cm which can therefore be attributed, ultimately, to the difference in pressure recovery along the separated layer downstream of the shock on the upper surface*. The relevant differences in section shape are thought to be those in the variation of surface angle between the shock position and the trailing edge and in the value of this angle at the trailing edge (i.e., half trailing-edge angle). The overall change between the shock position and trailing edge and the value at the trailing edge are both numerically smaller for the RAE. 102 section (see Table 2).

Table 2

	Angle between Tangent to Surface and Chord. Line	
$\frac{x}{c}$	RAE. 102 (10% thick)	RAE. 104 (10% thick)
0.55	-4.27°	
0.60		-3.98°
l.M)	-5 .46	-6.79°

The position of the upper surface shock might have had some small effect in the above example. It could presumably be important for widely different positions.

The effect of pressure rise through the shook did not influence the above comparison for a particular example because at corresponding Mach numbers it was from the same low pressure upstream (i.e., same local Mach number) and of approximately the sane magnitude. An affect on the total recovery and hence on the trailing-edge pressure would however be expected in general, tending to decrease the recovery with increasing Mach number upstream of the shock. The pressure rise through the front branch of the bifurcated shook is the pressure rise for separation and varies much less with upstream Mach number[47] than does the pressure rise through a normal shock (see Fig. 38). Moreover, the small normal wave downstream seems to be very weak and to produce little further compression. Thus with increasing upstream Mach number, once separation has occurred, the surface-pressure rise through the shock becomes increasingly less than the rise for a normal shock*.

It is therefore desirable to restrict the degree of supersonic expansion, or, in other words, the surface curvature, which of course is also the requirement for delaying separation. In this respect and also in that, as has been shown, the pressure recovery downstream of the shock is less affected when the curvature and slope of the surface are small, thin sections continue to ba beneficial even after the occurrence of separation.

(ii) <u>Shock Movement</u>.- A given change in trailing-edge pressure would produce different effects on the relative shock movements depending on section shape. Any feature which encouraged the deceleration of the upper surface shook or the acceleration of the lower surface

one/

*Near the point where the single shook meets the two branches of the bifurcated foot, the pressure rise through the two branches of the foot must be equal to that through the single shock, but otherwise conditions are not the same as for a uniform normal shock with bifurcated foot, for which the pressure is the same at all points behind the wave system. Instead, the strength of the shock and therefore the downstream pressure varies along its length; the pressure downstream of the point of intersection would thus depend on the height of this point above the surface and the pressure rise at the surface is not necessarily controlled by conditions at the point of intersection. Further, the single shock becomes progressively more inclined near the point of intersection (see Fig. 26, for example) and the pressure rise through this falls below that for a normal wave.

one would be undesirable. By analogy with the shock in a nozzle a slow rate of supersonic expansion, i.e., small curvature, is conducive to large shock movements and conversely. Thus camber would be expected to have an adverse effect and negative camber possibly a favourable one. An increase in curvature locally on the lower surface only, such as the cusping tried by Gothert[57], might have a favourable decelerating effect on the lower surface shock.

8. The Interaction at Supersonic Trailing Edges

For an unswept wing the shock waves on the upper and lower surfaces reach the trailing edge at a free-stream Mach number less than unity and remain there with further increase of Mach number. Shock waves are also present near the trailing edge of a swept wing provided that, to a first approximation, the inclination of the trailing edge to the undisturbed flow exceeds the Mach angle of the undisturbed flow. These shook waves interact with the boundary layer on the surface of the wing and cause it to thicken or separate upstream of the trailing edge; this in turn modifies the shock-wave pattern. As in the examples discussed abwe the effects are serious only when separation occurs; they are usually smaller than when the shock waves are located further forward mainly because the interaction does not influence the flow over such a large extent of the chord. Moreover, when the flow at the trailing edge is supersonic the conditions on one surface of the aerofoil do not influence those on the other (of. section 7.1). Nevertheless separation ahead of the trailing edge sometimes has large effects on the characteristics of the aerofoil or of control surfaces[64].

As well as depending on the strengths of the shock waves, the flow near the trailing edge is again found to depend on Reynolds number, and particularly on whether the boundary layer is laminar or turbulent. This is illustrated by Fig. 45 which shows schlieren photographs[60] of the tail of an E.G. 1250 aerofoil moving at zero incidence at a Mach number of 1.6. In Fig. 45(a) transition is free and, the Reynolds number (0.8×10^6) being low, the boundary layer is laminar over the whole surface; separation occurs ahead of the trailing edge and a diffuse shock system results in place of the single shock which would otherwise be expected. In Fig. 45(b) transition is fixed near the leading edge. No separation then occurs ahead of the trailing edge and a single shook is present there.

Supersonic wing sections are usually thinner than that (12%) discussed above, and at or near zero incidence the trailing-edge shock waves are not strong enough to provoke a very large region of separated flow. Extensive separation frequently occurs on the upper surface, however, when the wing is at incidence and the boundary layer is laminar because the shock on the upper surface is then of increased strength. For example, Fig. 46 compares the pressure distributions measured[61] on the upper surface of a 6% thick double wedge at NC = 1.6 and $R = 0.8 \times 10^6$ with those predicted by inviscid-flow theory. At low incidence the agreement is good, but at higher incidences the pressure begins to rise ahead of the trailing edge because laminar boundary layer separation occur& at the points indicated. Similar features are present in the observations described[62] by Beastall and Pallant who made tests on parallel to the free stream downstream. The conditions are thus similar to those of the experiment on the wedge attached to the plate described in section 3 although the pressure gradient associated with the curvature of the aerofoil surface my have some effect, and instead of a wall downstream of the shock there is a vortex sheet. Comparisons between the available results show reasonable agreement fcr laminar boundary layers, and it is hoped, therefore, that the results of the simple experiment will be useful in predicting the characteristics of nerofoils. Unfortunately very few experiments have been made on supersonic aerofoils with turbulent boundary layers and it is, therefore, impossible to make a detailed comparison for this case. The results described in section 2.5 suggest that turbulent boundary layer separation is unlikely to occur if the angle turned through by the flow at the trailing edge is less than about 12 deg (in contrast to about 4 deg for laminar layers). If this is so, the effect of the interaction may not be very important for supersonic aircraft; for missiles they may, however, be large not only because larger angles may be involved, but because the boundary layer may in some oases be laminar.

The difference between the behaviour with laminar and turbulent layers may be a source of scale effect in model tests as it is in the other cases discussed here. For example, because of the diffuse nature of the trailing-edge shock formed as a result of laminar separation, the drag may be reduced considerably and in some oases may be less than that calculated for inviscid flow.

Part III. The Effect of the Interaction on the Performance of Supersonic Intakes

9. The Nature of the Flow

Shock wave and boundary layer interaction is of particular significance in the design of side air intakes for aircraft and missiles. By tho term side intake is implied any intake which absorbs a proportion of external boundary layer, as from the nose of a fuselage or the under-surface of a wing. It is the interaction of the pre-entry shock system with this boundary layer which is important.

The duty of the intake is to collect a prescribed quantity of air and, under normal flight conditions, to decelerate this air from its intial relative velocity (approximately the flight speed) to the low Mach number required by the engine (0.4 or less, according to the type of engine), with as little loss of total pressure as possible. When the flight speed is supersonic, the deceleration normally takes place in three stages:-

(a) deceleration Co subsonic speed through a single shock or system of shocks ahead of the entry;

(b) subsonic diffusion between the shook system and the entry;

(c) further subsonic diffusion in the intake duct.

Boundary layer separation can occur at any of these stages but is most likely to do so at one or other of the pre-entry stages (a) and (b). Considering a normal-shock intake, if the free stream Mach number is greater than aboutl.3, the shock itself will cause the boundary layer to separate. If the free-stream Mach number is less than 1.3 and the boundary layer is turbulent, the shock may thicken and deform the boundary layer, and separation may then follow in the further adverse pressure gradient of stage (b). The magnitude of this pressure gradient depends upon the entry mass flow ratio*.

In/

*The entry mass flow ratio is defined as n = (ρV) entry/(ρV) free stream.

In either case the essential features of the flow, as affecting the ultimate pressure recovery of *the* intake, are as shown in Fig. 47. Separation occurs at some point between the shock and the entry (shown for simplicity at the foot of the shock). Behind the separation point is a dead-air region, extending into the duct and separated from the main flow by a zone of turbulent mixing. At some point inside the duct (the position depending on the further pressure gradient in the duct, and therefore to some extent controllable) the dead-air region terminates and normal boundary-layer development recommences. Thus the various losses which go to determine the final pressure recovery at the engine face are as follows:-

(1) Loss from friction in the external. boundary layer ahead of the shocks,

(2) loss from friction on the walls of the duct,

(3) shock losses,

(4) loss from turbulent miring in the zone following the separation.

Strictly each of the first three components is in some degree affected by the presence of component(4), but a convenient analysis can be obtained by assigning to the first, three components the values they would have in unseparated flow and regarding the last component as a net interference loss. A recent experimental study at the R.A.E. has shown that in the transonic speed range the interference loss is liable to be the major term. Fig. 48, showning a breakdown of loss for a typical side intake tested at Mach numbers from 0.7 to 1.8, demonstrates this result.

It will be appreciated that boundary-layer separation can occur at subsonic as well as supersonic flight speeds. The subsonic case is essentially similar to that already described, with the omission of stage (a) of the flow. The separation normally occurs ahead of the entry as before, under the influence of the adverse pressure gradient of stage (bj, and is again followed by a turbulent mixing *zone*. Fig. 48 shows, however, that the adverse effects are much less severe than in the supersonic case.

IO. The Pressure at Separation

An important property of the type of flow described is the pressure at separation. This has been measured for various initial thicknesses of the turbulent boundary layer, at Mach numbers in the range mentioned. The actual pressure measured is that at the base of the entry plane, i.e., in the dead-air region. Some results are plotted in Fyi. 49. At subsonic speeds, when the separation is produced by a wholly subsonic gradient, whose magnitude is determined by the proportions of the intake (in relation to the boundary layer *thickness),* the *critical* pressure coefficient generally lies between 0.4 and 0.6, varying in an inverse manner with boundary-layer thickness. At low supersonic speeds, when the separation is produced by a combination of shock wave and subsonic pressure gradient the critical pressure falls as Mach number rises, i.e., as the part played by the shock wave increases. At higher supersonic speeds, when the shock interaction alone is sufficient for separation, the critical pressure coefficient settles to a value around 0.35, which, so far as the experiments go, appears to be roughly independent of both Mach number and boundary layer thickness.

Independence/

Independence of Mach number is a result somewhat in disagreement with the s&-theoretical relationship advanced in section 2.6. This rule gives a lower value of the separation pressure coefficient and one decreasing slowly with increase of Mach number. The present results support rather the empirical rule advanced by Beastall[16], from experiments on the flow over a rectangular step. This is

$$\frac{p_s}{p_1} = 1 + \frac{M_1^2}{4} ;$$

from which we derive

$$C_{p_{crit}} = \frac{(p_s - p_1)}{\frac{1}{2}\rho_0 q_0^2} = \frac{M_1^2}{4} \cdot \frac{2}{\gamma M_1^2} = \frac{1}{2\gamma} \approx 0.36 .$$

The present experiments are incomplete, however, having to date been concerned mainly with obtaining results from the transonic range. It is hoped to add further evidence at higher Mach number in the near future.

As mentioned above, when separation is caused by the shock interaction alone, the pressure at separation is independent of boundary layer thickness. This is presumably because, for a given boundary-layer form, the spread of pressure rise inside the boundary layer at the shock position is proportional to boundary-layer thickness; so that the pressure gradient, in units of boundary-layer thickness, is independent of the thickness.

11. Methods of Reducing or Avoiding the Effects of Separation

Faced with the type of flow described in the preceding paragraphs, the practical problem, accepting the desirability of a continued use of side intakes, is how to avoid, or at least minimize, the detrimental effect of boundary-layer separation on intake performance. Various ways of attacking this problem my be suggested. One is to accept a pre-entry flow pattern of the kind described but minimize the effect of separation by suitable design of the intake duct. Another is to re-energize the boundary layer ahead of the shook so that separation my be avoided. A third method is to remove the boundary layer by a suitable form of bleed.

Part IV. Notes on the Further Work which is Required

The available experimental results on the interaction of a shock wave with the boundary layer on a flat plate or parallel-sided body of revolution indicate fairly clearly the general nature of the

It would also be of value to make experiments on curved surfaces to investigate the effects of the pressure gradient produced by the curvature on the interaction of the boundary layer with shocks generated both externally and by a wedge attached to the surface. An investigation of the effects of surface curvature on the reattachment of the separated boundary layer would also have important practical applications.

Sometimes it is important to know the overal effects of the interaction on the characteristics of the boundary layer (i.e., the differences between the boundary-layer thickness and profile well upstream and well downstream of the interaction). There is very little information on this point because the interacting shook is usually followed fairly closely by an expansion wave; experiments in which this difficulty was avoided would be useful.

Most of the work which has been done with interacting shook waves needs to be repeated with more gradual regions of adverse pressure gradient as would be generated, for example, by a curved surface placed in the main stream. There is also still a need for work on the interaction between expansion waves and boundary layers although the effects here are not so large, and are more amenable to theoretical treatment.

One of the greatest needs at the moment is for a detailed correlation between the results of theory and fundamental experiments of the type described in Part I, and the phenomena observed on aerofoils and wings. A preliminary attempt to do this is made in Part II and, for example, in Ref. 36, but much more work is required before it will be possible to estimate how accurately the occurrence and effects of the interaction in practice can be predicted. Correlations of this type are greatly hampered by the scarcity of detailed information in practical oases, and this is further discussed below.

Results of tests made with laminar boundary layers are not applicable to flight conditions and more work is, therefore, required to find suitable (and preferably simple) methods for fixing transition during model experiments especially for high incidence when the shocks occur very near the nose of the aerofoil.

It has been established that the conditions for the occurrence of turbulent separation are almost independent of Reynolds number, and further, that its effects in flight and in the wind tunnel are qualitatively similar. The effect of Reynolds number on the pressure recovery downstream of the shock needs to be investigated, however, to determine whether or not wind tunnel tests give reliable impressions of the magnitude of the separation effects for aerofoils at high subsonic speeds.

Further detailed explorations by pressure plotting and flow visualization on finite wings (with transition fixed) such as those at present being made by Vickers-Armstrong Ltd. (Weybridge) and at the R.A.E., are required an order to build up a better basic knowledge of the conditions for separation and of its effects for such wings. The severity of its effects. The tests should be extended* to thicknesses below 6 per cent, and to high incidences, preferably to $C_{L_{max}}$. As much of this work as possible should include pressure plotting and flow visualization, at least until the effects of separation and the influence of section shape are more fully understood. In this respect, a knowledge of the flow round an aerofoil in the absence of the boundary layer would help greatly, and it should be possible to achieve this condition by the application of area suction over the whole or most of both surfaces.

The above remarks refer to aerofoils moving at high subsonic speeds; further work is also required at supersonic speeds especially to determine the effects of the interaction on the flow near a supersonic trailing edge for turbulent boundary layers. Very few detailed observations have yet been made for this case.

Novel sections designed to alleviate the effects of separation on aerofoils moving at high subsonic speeds should be investigated, as should any promising remedy to prevent or to reduce the severity of separation.

Although, at high subsonic speeds, the effects of separation on aerofoils are closely bound up with the flow about the whole aerofoil and the interrelation of the two surfaces, certain specific features could perhaps be studied on a plate or on the wall of a wind tunnel. Examples are the influence of surface curvature or slope, or of area suction on the pressure recovery downstream of the separation point.

Studies should be made of pressure fluctuations in the wake of aerofoils with turbulent boundary layers and of the associated oscillations of the shock waves. The development of a technique which could be used in routine wind-tunnel experiments to reveal the presence of large-scale disturbances in the wake would assist in this work; it is possible that the signal from a photo-cell placed in the image plane of a schlieren apparatus might reveal this type of aerodynamic disturbance for two-dimensional aerofoils. In connection with buffeting, the conditions leading to and affecting turbulent separations at wing-body and other junctions also need to be investigated..

In the field of intake problems, more work is needed to separate out the Mach and Reynolds number effects on separation pressure and to provide a quantitative assessment of each. The ground-launched rocket model technique is suitable for high Reynolds number tests.

Further studies on side intakes are also required to determine how far the detrimental effect of shock wave and boundary-layer interaction can be offset by suitable forms of boundary-layer control, at Mach numbers above 1.5.

List of Symbols/

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*Further tests planned at the N.P.L. will include the following sections: 4 per cent thick R.A.E. 104, 6 per cant thick RAE. 102 (with and without control flap and spoilers), and 4 per cent and 3 per cent thick biconvex sections.

List of Symbols

M_o	Free-stream Mach number
M_1	Mach number just upstream of the shook wave
R_o	Reynolds number based on the distance from the leading edge of the flat plate to the point where the pressure begins to rise (see Fig. 4).
R_t	Reynolds number of the turbulent boundary layer just upstream of the interaction (see Fig. 9).
R_s	Value of R_o or R_t at the separation point.
R	Reynolds number based on aerofoil chord.
δ_o^*	The displacement thickness of the boundary layer at the beginning of the interaction (see Fig. 4).
d	The upstream influence distance of the shock wave (see Fig. 4).
c	Aerofoil chord.
x	Distance from the leading edge of the aerofoil (measured along the chord line).
u	Velocity component parallel to the surface of the flat plate,
q	Resultant velocity
p	Local static pressure at the &ace.
p_o	Static pressure in the free stream.
p_1	Static pressure at the surface upstream of the interaction.
p_{max}	The maximum static pressure at the surface
p_s	The static pressure at the surface at the separation point.
$p_{T.E.}$	The static pressure at the trailing edge of an aerofoil
$C_{p_{T.E.}}$	The pressure coefficient at the trailing edge $= \dfrac{p_o - p_{T.E.}}{\frac{1}{2}\rho_o q_o^2}$
H_o	Total head in the free stream.
H_3	Total head in intake after complete diffusion.

References

NO.	Author(s)	Title, etc.
1	A. Ferri	Experimental results with aerofoils tested in the high-speed tunnel at Guidonia. Atti di Guidonia No. 17 (1939). Translated as N.A.C.A. TM. 946.
2	A. Fage and R.F. Sargent	Shock wave and boundary-layer phenomena near a flat surface. Proc. Roy. Soc. A. vol. 190, p. 1. (1947).
3	J. Ackeret, F. Feldmann and N. Rott	Investigations of compression shocks and boundary layers in fast moving gases. Institut für Aerodynamik E.T.H. Zurich Report No. 10. (1946). A.R.C. 10,044. (Received 7th September, 1946).
4	H. W. Liepmann	The interaction between boundary layer and shock waves in transonic flow. J. Ae. Sc. vol. 13, No. 12, p. 623 (1946).
5	C. Du P. Donaldson	Effects of interaction between normal shock and boundary layer. N.A.C.A. CB. 4A27 (1944).
6	H. J. Allen, M. A. Heaslet and G. E. Nitzberg	The interaction of boundary layer and compression shock and its effect upon airfoil pressure distributions. N.A.C.A. RM A7A02 (1947).
7	H. W. Liepmann, A. Roshko and S. Dhawan	On the reflection of shock waves from boundary layer. Galcit Report.
8	F. W. Barry, A. H. Shapiro and E. P. Neumann	The interaction of shock waves with boundary layers on a flat surface. J. Ae. Sc. vol. 18, No. 4, p. 229 (1951).
9	W. F. Cope	The measurement of skin friction in a turbulent boundary layer at a Mach number of 2.5, including the effect of a shock wave. Proc. Roy. Soc. A. vol. 215, p. 84 (1952).
10	O. Bardsley and W. A. Mair	The interaction between an oblique shock-wave and a turbulent boundary layer. Phil. Mag. Ser. 7, vol. 42, p. 29, (1951).
11	G. E. Gadd and D. W. Holder	The interaction of an oblique shock wave with the boundary layer on a flat plate. Part I. Results for M = 2. A.R.C. 14,848. 24th April, 1952.
12	G. E. Gadd, D. W. Holder and J. D. Regan	The interaction of an oblique shock wave with the boundary layer on a flat plate. Part II. Interim note on the results for M = 1.5, 2, 3 and 4. A.R.C. 15,591. 30th January, 1953.

13/

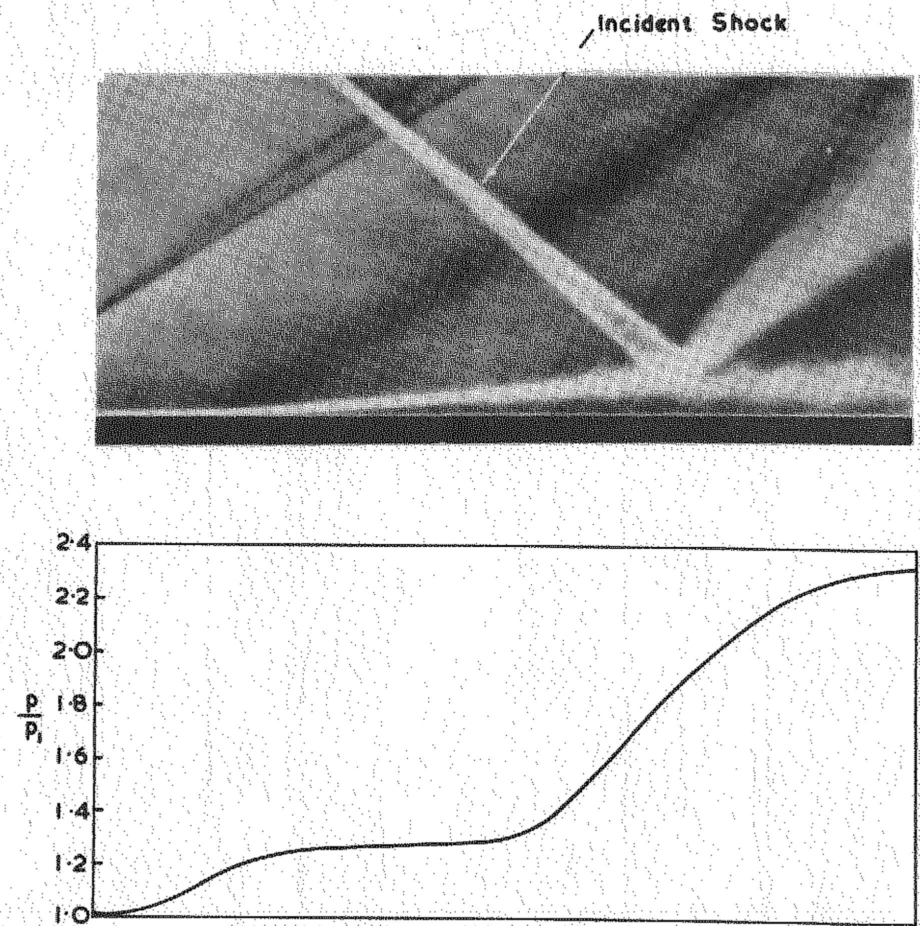
No.	Author(s)	Title, etc.
13	S. M. Bogdonoff and A. H. Solarski	A preliminary investigation of a shock-wave turbulent boundary layer interaction. Princeton University Aeronautical Engineering Laboratory Report No. 184 (1951).
14	G. Drougge	Experimental investigation of the influence of strong adverse pressure gradients on turbulent boundary layers at supersonic speeds. Paper read at the 8th International Congress on theoretical and applied mechanics, Istanbul. (1952).
15	N. H. Johannesen	Experiments on two-dimensional supersonic flow in corners and over concave surfaces. Communicated by P.D.S.R.(A), Ministry of Supply. A.R.C. 14,607. 29th January, 1952.
16	D. Beastall and H. Eggink	Some experiments on breakaway in supersonic flow. Part I. R.A.E. Tech. Note Aero. 2041 (1950). Part II, R.A.E. Tech Note Aero. 2061 (1950).
17	J. D. Lee	The influence of high adverse pressure gradients on boundary layers in supersonic flow. University of Toronto Institute of Aerophysics UTIA Report No. 21 (1952).
18	J. Lukasiewicz	Conical flow as a result of shock and boundary layer interaction on a probe. R. & M. 2669. September, 1948.
See also	D. W. Holder R. J. North and A. Chinneck	Experiments with static tubes in a supersonic airstream. Parts I and II. R. & M. 2782. July, 1950.
19	C. Du P. Donaldson and R. H. Lange	Study of the pressure rise across shock waves required to separate laminar and turbulent boundary layers. N.A.C.A. TN. 2770 (1952).
20	L. Howarth	The propagation of steady disturbances in a supersonic stream bounded on one side by a parallel subsonic stream. Proc. Camb. Phil. Soc. vol. 44, Part 3 (1947).
21	H. S. Tsien and M. Finston	Interaction between parallel streams of subsonic and supersonic velocities. J. Ae. Sc. vol. 16, No. 9, p. 515 (1949).
22	M. J. Lighthill	Reflection at a laminar boundary layer of a weak steady disturbance to a supersonic stream neglecting viscosity and heat conduction, Quart. Journ. Mech. and App. Maths. vol. 3, p. 303 (1950).

NO.	Author(	Title, etc.
23	A. Robinson	Wave reflection near a wall. College of Aeronautics Report No. 37. A.R.C. 13,252. May, 1950.
24	K. Oswatitsch and K. Wieghardt	Theoretical analysis of stationary potential flows and boundary layers at high speed. German wartime Report (1941). Translated as N.A.C.A. TM. 1189.
25	M. J. Lighthill	On boundary layers and upstream influence. Part II. Supersonic flows without separation. P.R.S. A. vol. 217, p. 478 (1953).
26	L. Lees	Interaction between the laminar boundary layer over a plane surface and an incident oblique shock wave. Princeton University, Aero. Eng. Lab. Report 143 (1949).
27	L. Crocco and L. Lees	A mixing theory for the interaction between dissipative flows and nearly isentropic streams. J. Ae. Sc. vol. 19, No. 10 (1952).
28	K. Stewartson	On the interaction between shock waves and boundary layers. Proc. Camb. Phil. Soc. vol. 47, part 3, p. 545, (1951).
29	G. E. Gadd	On the interaction with a completely laminar boundary layer of a shock wave generated in the mainstream. A.R.C. 15,100. 1st August, 1952.
30	G. E. Gadd	A semi-empirical theory for interactions between turbulent boundary layers and shock waves strong enough to cause separation. A.R.C. 15,543. 10th January, 1953.
31	K. G. Tadman	The pressure on the base of high-speed projectiles: a survey of recent research. A.R.C. 14,967. June, 1952.
32 and 33	K. W. Todd	An experimental study of three-dimensional high speed air conditions in a cascade of axial-flow compressor blades. R. & M. 2792. October, 1949.
34	D. W. Holder, A. Chinneck and G. E. Gadd	An experimental investigation of the interaction of a shock wave with a quasi-uniform subsonic stream bounded with a wall. In preparation.
35	D. A. Spence	Prediction of the characteristics of two-dimensional aerofoils. Communicated by

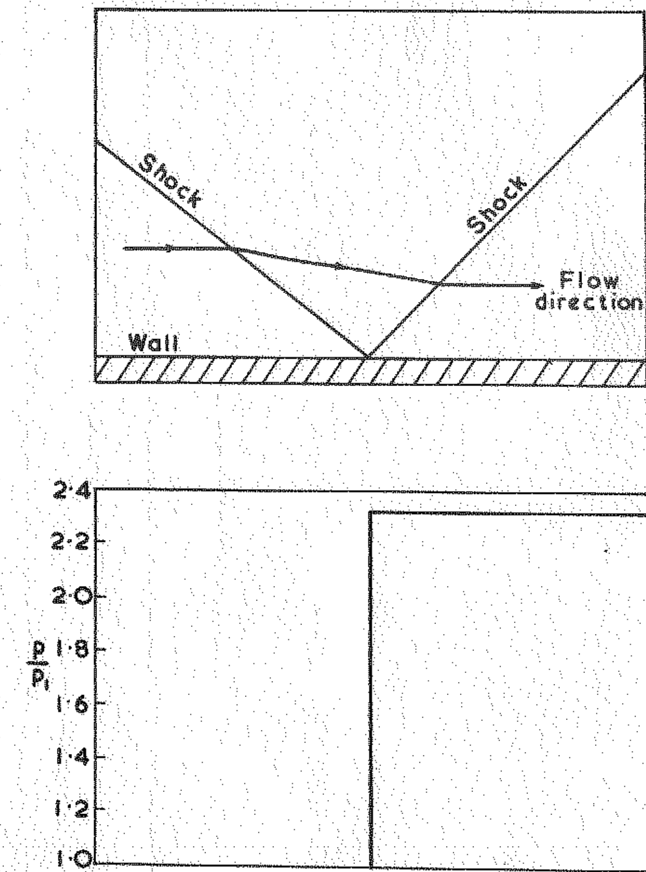
No.	Author(s)	Title, etc.
37	A. Fage	Some aerodynamic advances. Proceedings Third Anglo-American Aeronautical Conference, p. 329 (1951).
38	H. E. Gamble	Some effects of Reynolds number on a cambered wing at high subsonic Mach numbers. RAE. Report No. Aero. 2423. A.R.C. 14,448. May, 1951.
39	A. Fage and R.F. Sargent	An air-injection method of fixing transition from laminar to turbulent flow in a boundary layer. R. & M. 2106. June, 1944.
40	E. N. Harrin	A flight investigation on the effect of shape and thickness of the boundary layer on the pressure distribution in the presence of shock. N.A.C.A. TN. 2765 (1952).
41	A. D. Young	Boundary layers and skin friction in high speed flow. Journ. R. ke. Soc. May, 1951.
42	H. H. Pearcey, R. C. Pankhurst and R. F. Cash	High-speed tunnel tests on a 10% thick R.A.E. 102 two-dimensional aerofoil with 25% flap; results at zero incidence with 4° flap deflection. A.R.C. 15,176. 8th September, 1952.
43	L. F. Tanner and H. H. Pearcey	Some observations of the flow round leading edges of different radii at high incidences and high subsonic speeds. In preparation.
44	E. W. E. Rogers, C. J. Berry and R. F. Cash	Tests at high subsonic speeds on a 10%-thick pressure-plotting aerofoil of RAE. 104 section: Part III - Tests with fixed transition position. A.R.C. 13,947. 24th April, 1951. (To be R. & M. 2863.)
45	L. Hosking	Vickers Armstrong Ltd., (Weybridge) Wind Tunnel Report No. 2015 (1953).
46	J. A. Zalovcik and E. P. Luke	Some flight measurements of pressure-distribution and boundary-layer characteristics in the presence of shock. N.A.C.A. RM No. L8C22 (1948). (NACA/TIB/1865.)
47	G. P. Wood and P. B. Gooderum	Investigation with an interferometer of the flow around a circular-arc airfoil at Mach numbers between 0.6 and 0.9. N.A.C.A. TN 2801 (1952).

No.	Author(s)	Title, etc.
50	J. R. Collingbourne and A. C. S. Pindar	R.A.E. Tech. Note Aero. 2221. 1953.
51	V. Outman and A. A. Lambert	Transonic separation. J. Ae. Sc. vol. 15, p. 671. 1948.
52	D. W. Holder	Note on the wakes behind bodies moving at high Mach number. A.R.C. 14,720. 11th March, 1952.
53	A. Fage and F. C. Johansen	The structure of vortex sheets. R. & M. 1143. (1927).
54	D. W. Holder, R. J. North, W. G. Standring and J. S. T. Looms	A high-speed camera for the photography of shock-wave oscillations in a wind tunnel. A.R.C. 12,543. 31st August, 1949.
55	A. Chinneck, D. W. Holder and C. J. Berry	Observations of the flow round a two-dimensional aerofoil oscillating in a hi&-speed airstream. A.R.C. 15,141. 23rd August, 1952.
56	R. C. Pankhurst and H. B. Squire	Calculated pressure-distributions for the RAE. 100-104 aerofoil sections. R.A.E. Technical Note No. Aero. 2039. C.P. 80. March, 1950.
57	B. Gothert and li. A. Mair	German high-speed wind-tunnel results collected. by R.A.E. Tech. Note No. Aero. 1684. A.R.C. 9064. August, 1945.
58	See for example:-	
	A. Fage and R. F. Sargent	Effect on aerofoil drag of boundary-layer suction behind a shock wave. R. & M. 1913. October, 1943.
59	B. Regenscheit	Drag reduction by suction of the boundary layer separated behind shook-wave formation at high Mach numbers. N.A.C.A. TM No. 1168. (1947).
	K. G. Anderson	Preliminary investigation of boundary layer control at high subsonic speeds. U.S. Air Force Tech. Rep. No. 61%. (1950).
	H. Ludwieg	Results of experimental investigation at high subsonic speeds. A.V.A. Monograph E.8.
	Ph. Poisson-Quinton	Theoretical and experimental research on boundary-layer control - Proceedings of the VIIth International Congress of Applied Mechanics, London. September, 1948.
	K. G. Anderson	Investigation of boundary-layer control at high speeds. - U.S. Air Force Tech. Rept. No. 6344.

NO.	Author(s)	Title, etc.
59	M. Pindzola	Supersonic tests of conventional control surfaces on a double-wedge airfoil. J. Be. Sc. vol. 17, p. 204. (1950).
60	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. C.P. 53. 28th December, 1950.
61	D. W. Holder, A. Chinneck and D. G. Hurley	Observations of the supersonic flow round a 6 per cent double wedge. C.P. 63. (1951).
62	D. Beastall and R. J. Pallant	Wind-tunnel tests on two-dimensional supersonic aerofoils at M = 1.86 and M = 2.48. R.A.E. Report No. Aero. 2384. A.R.C. 13,768. July, 1950.



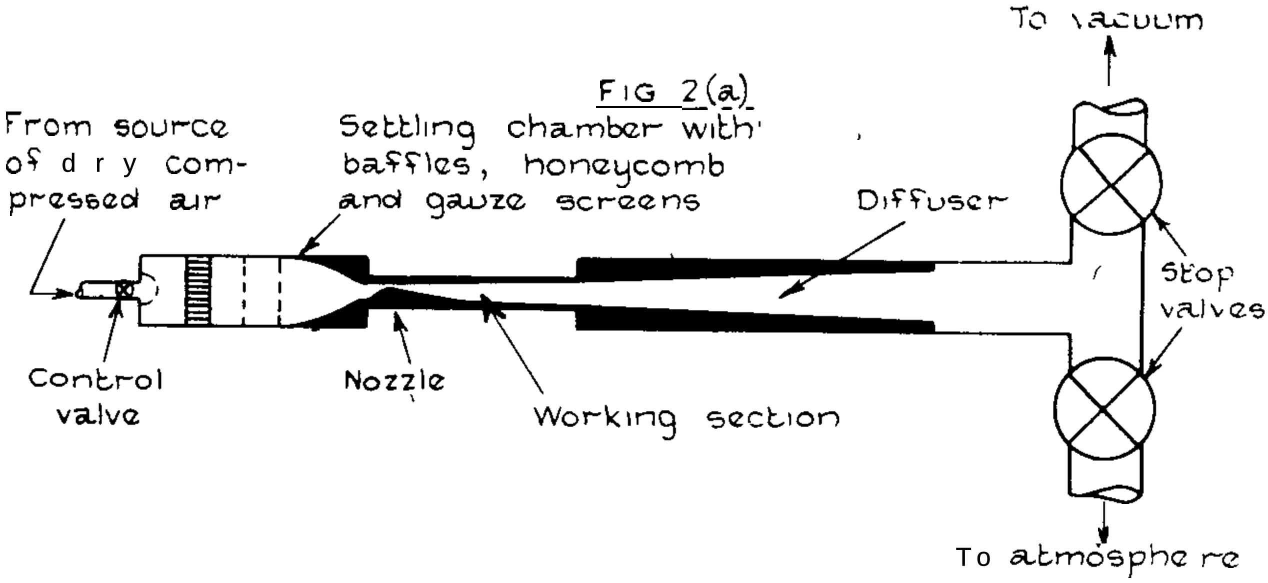
(a) $M_o=2$ Initially laminar boundary layer



(b) Regular reflection of same shock as in (a)

Fig.1. Flow patterns and pressure distributions at the wall to same longitudinal scale with and without boundary layer.

FIG 2(a)



General arrangement of the tunnel

FIG 2(b)

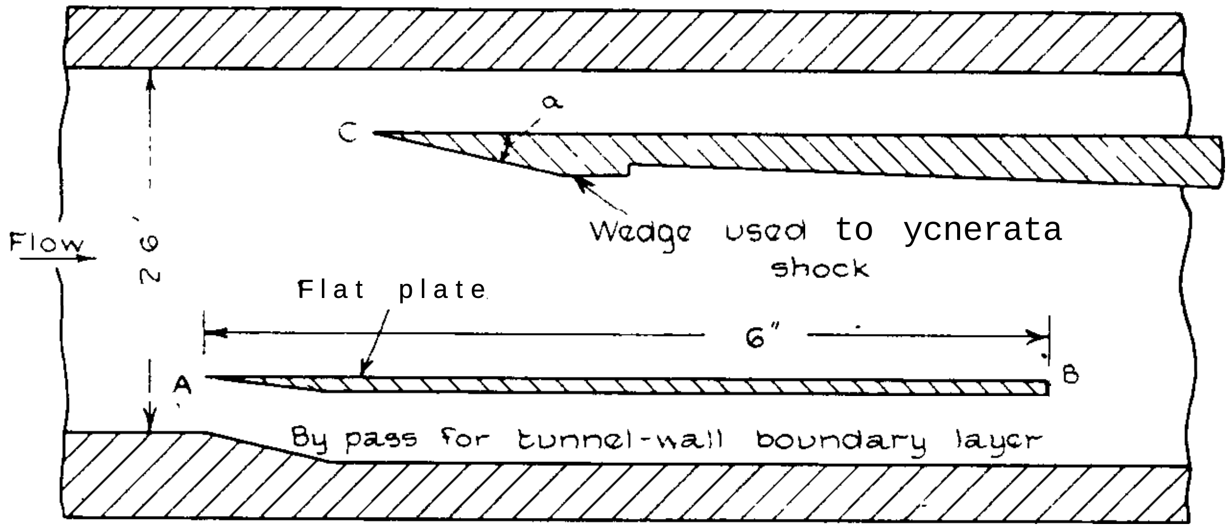
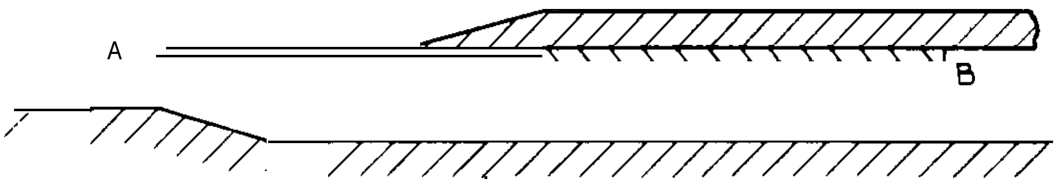


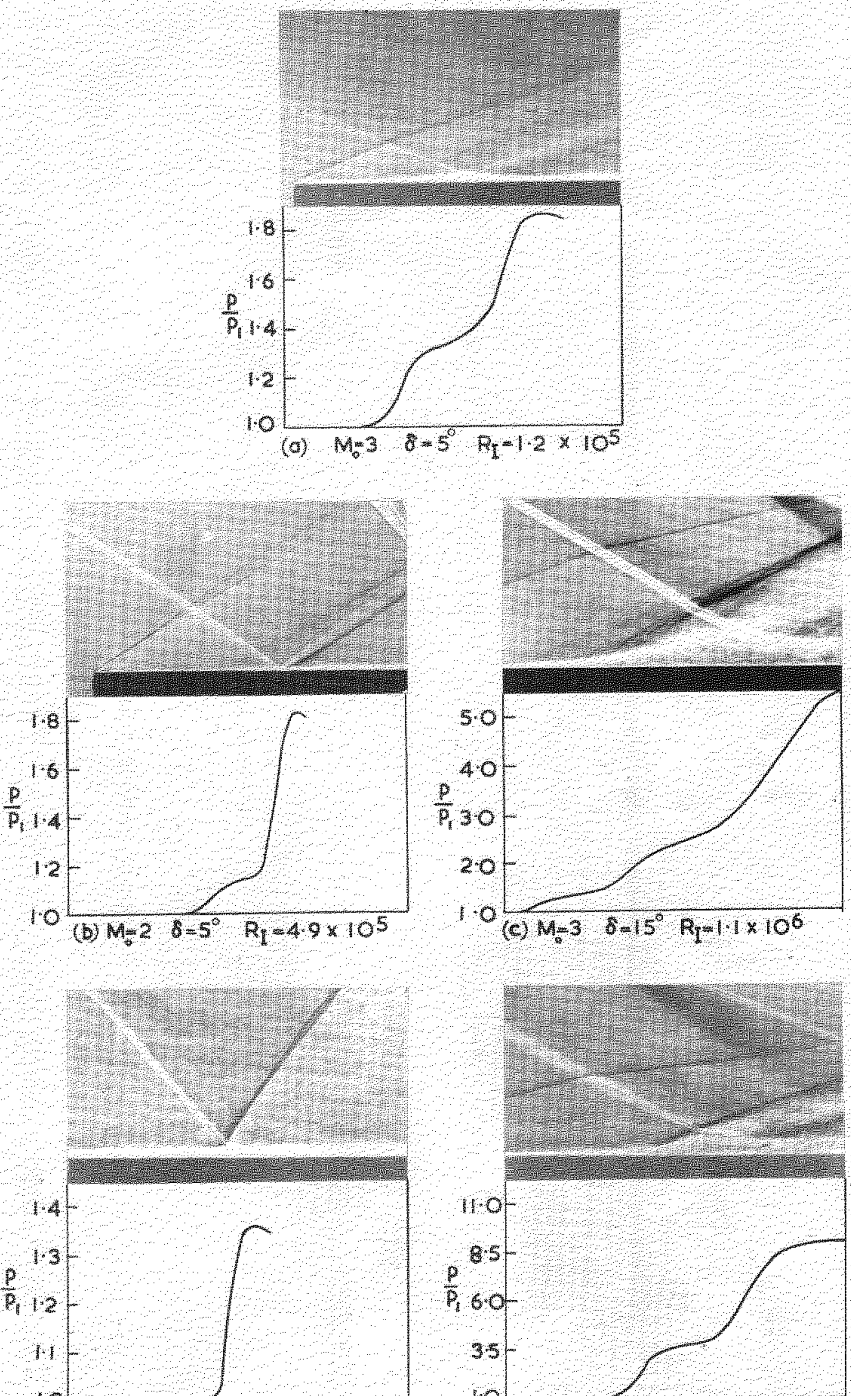
Diagram of tha working sectionwith wedge for externally generated shock5

FIG 2 (c)



Wedge on plate for shocks generated within the boundary layer

FIG 2 (a-c)
Details of tha apparatus used for investigating the interaction of a shock with the boundary layer on a flat plate


1·8
1·6
p/p1 1·4
1·2
1·0
(a) M0=3 δ=5° RI=1·2 x 10^5
1·8
1·6
p/p1 1·4
1·2
1·0
(b) M0=2 δ=5° RI=4·9 x 10^5
5·0
4·0
p/p1 3·0
2·0
1·0
(c) M0=3 δ=15° RI=1·1 x 10^6
1·4
1·3
p/p1 1·2
1·1
11·0
8·5
p/p1 6·0
3·5

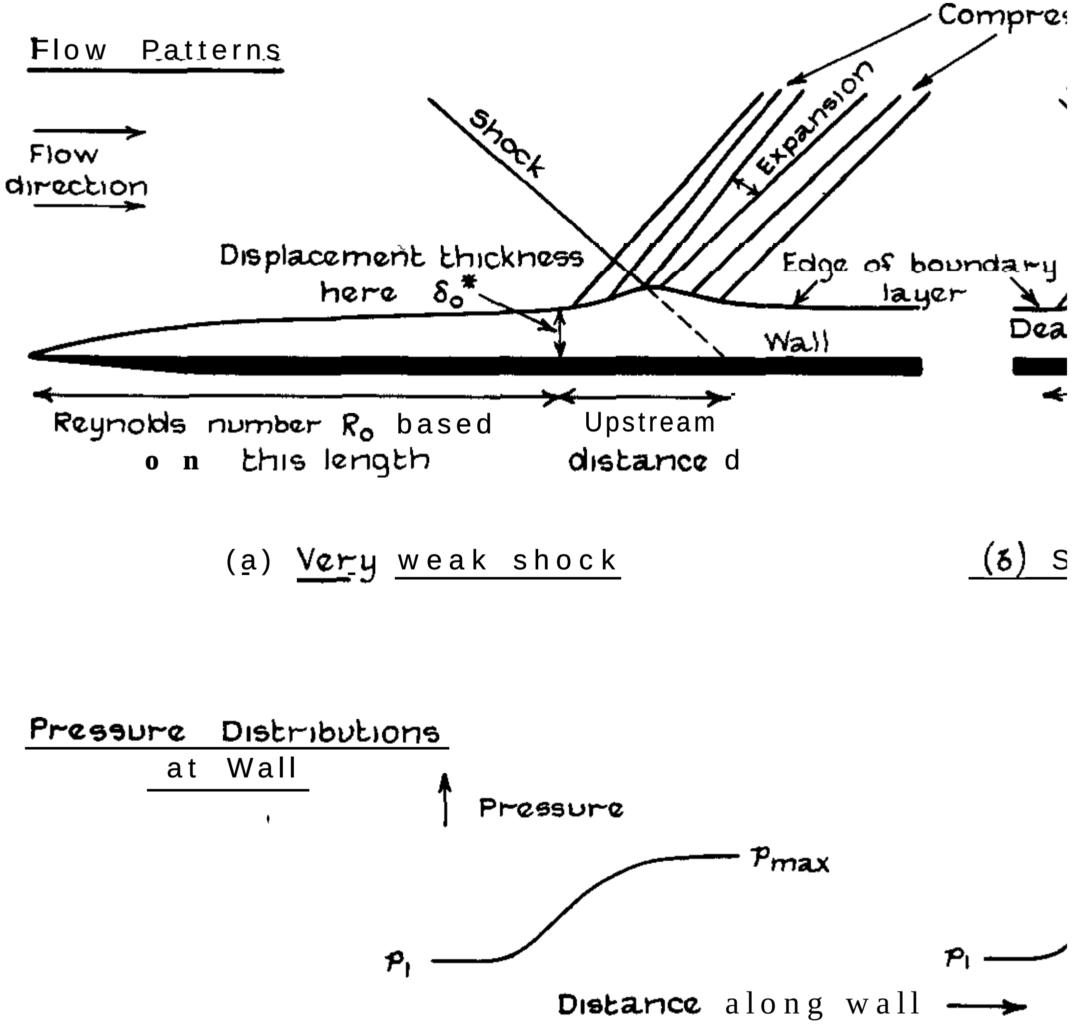


FIG. 4. Boundary Layers Laminar Over Whole Region of

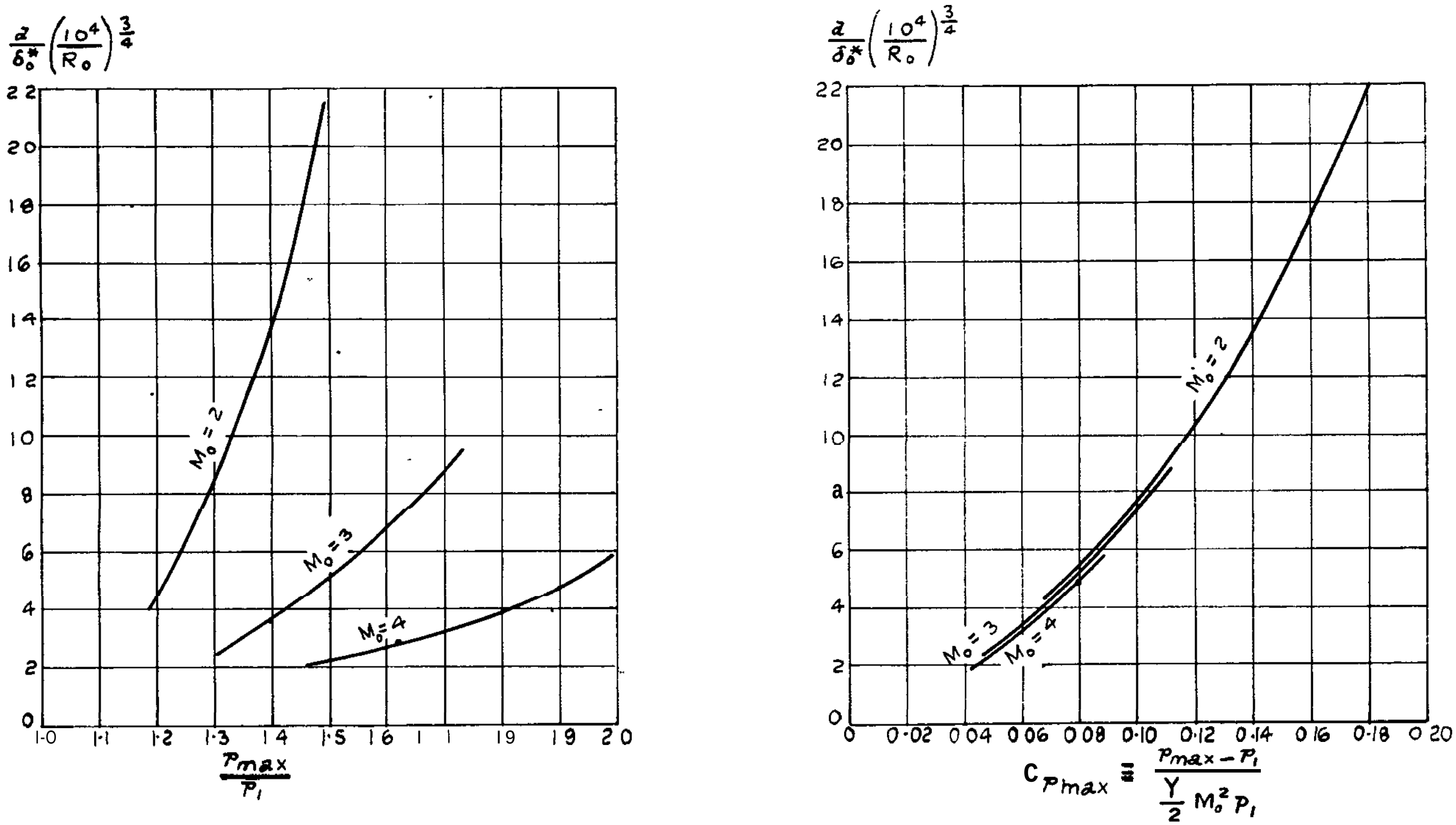


FIG. 5. Boundary layers laminar over whole region of interaction. Factor $\frac{d}{\delta_0^*}\left(\frac{10^4}{R_0}\right)^{\frac{3}{4}}$ as function of $\frac{p_{max}}{p_1}$ and $C_{p_{max}}$

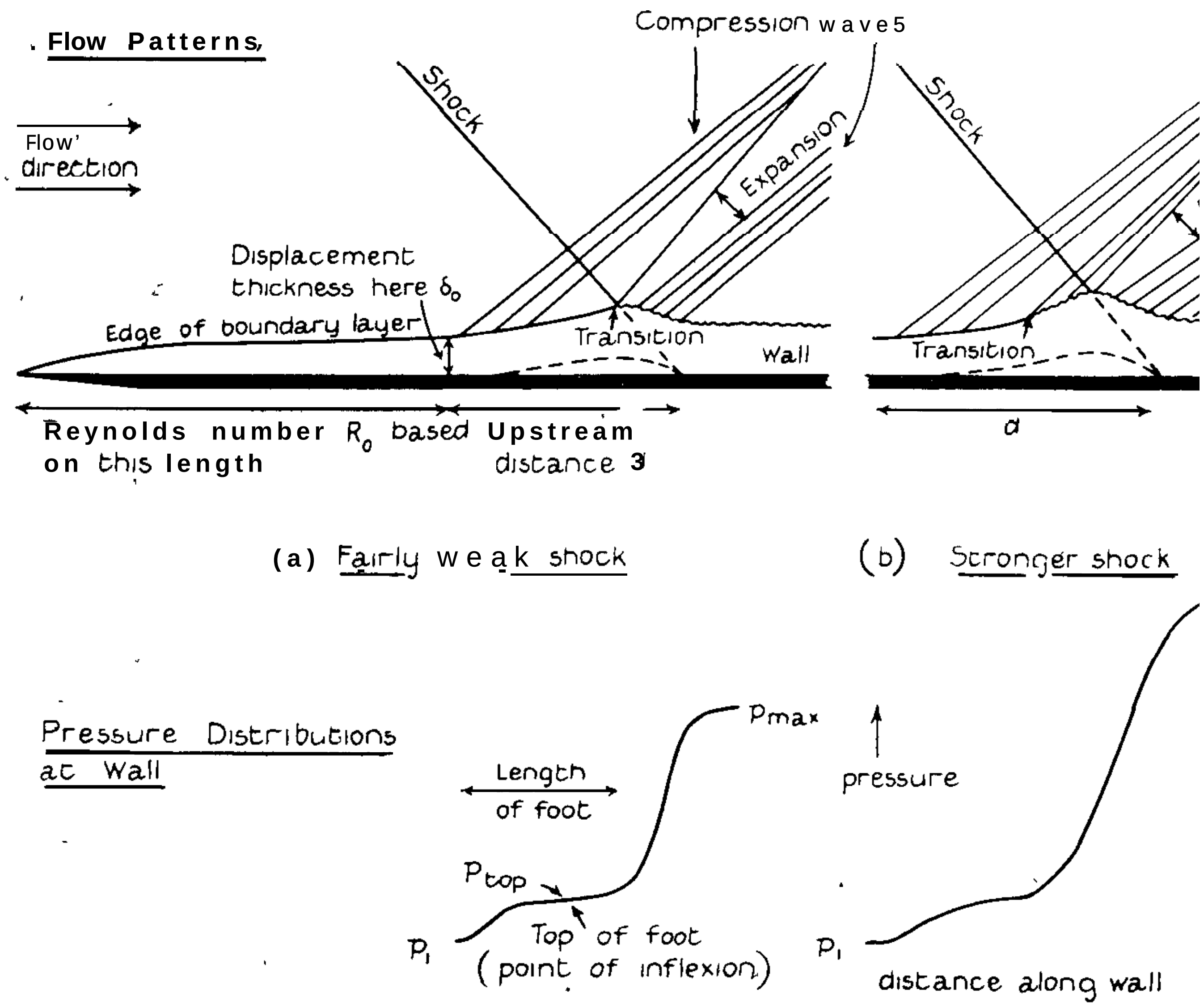


FIG. 6. Boundary layers laminar at separation but turbul

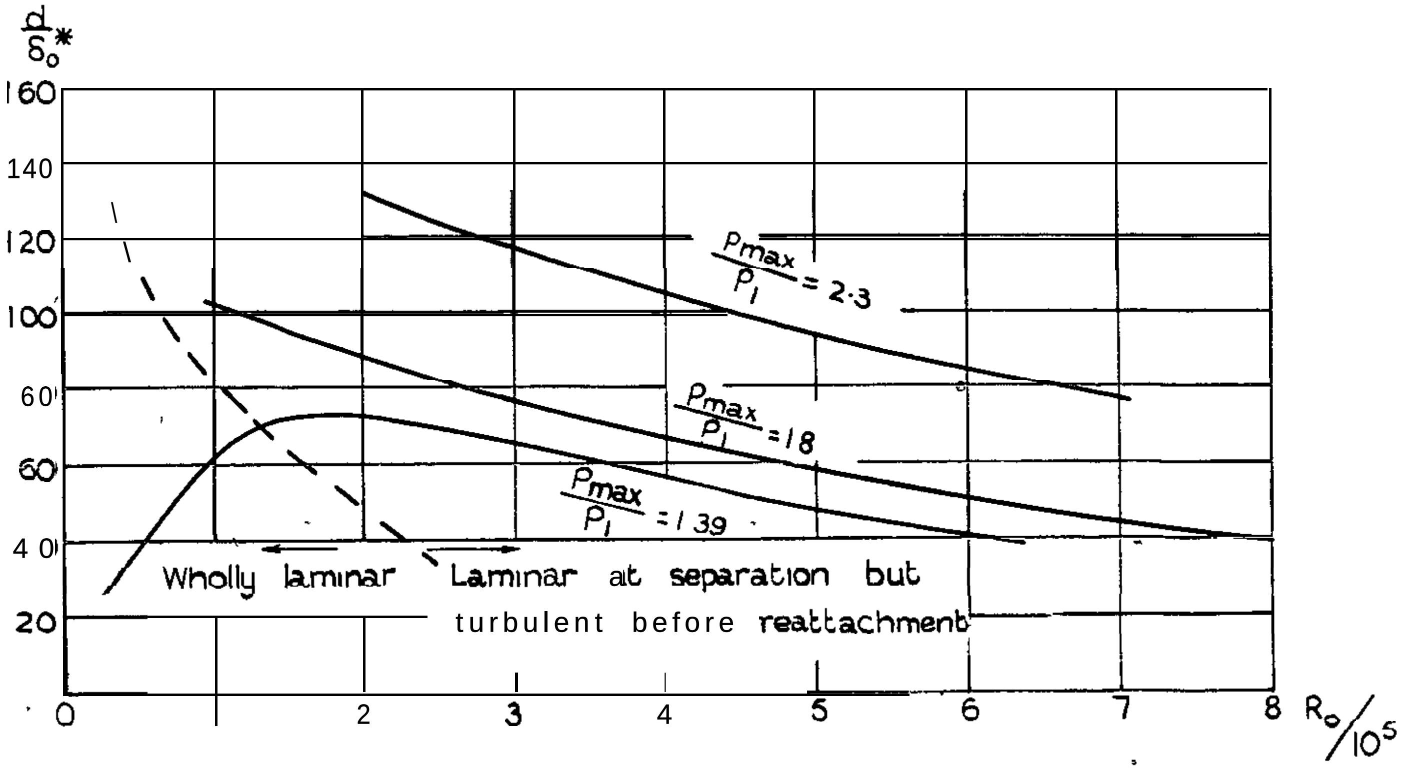


FIG 7.

Boundary layers laminar at separation. M = 2

Factor $\frac{d}{\delta_o^*}$ as function of Reynolds number R_o at various shock strength

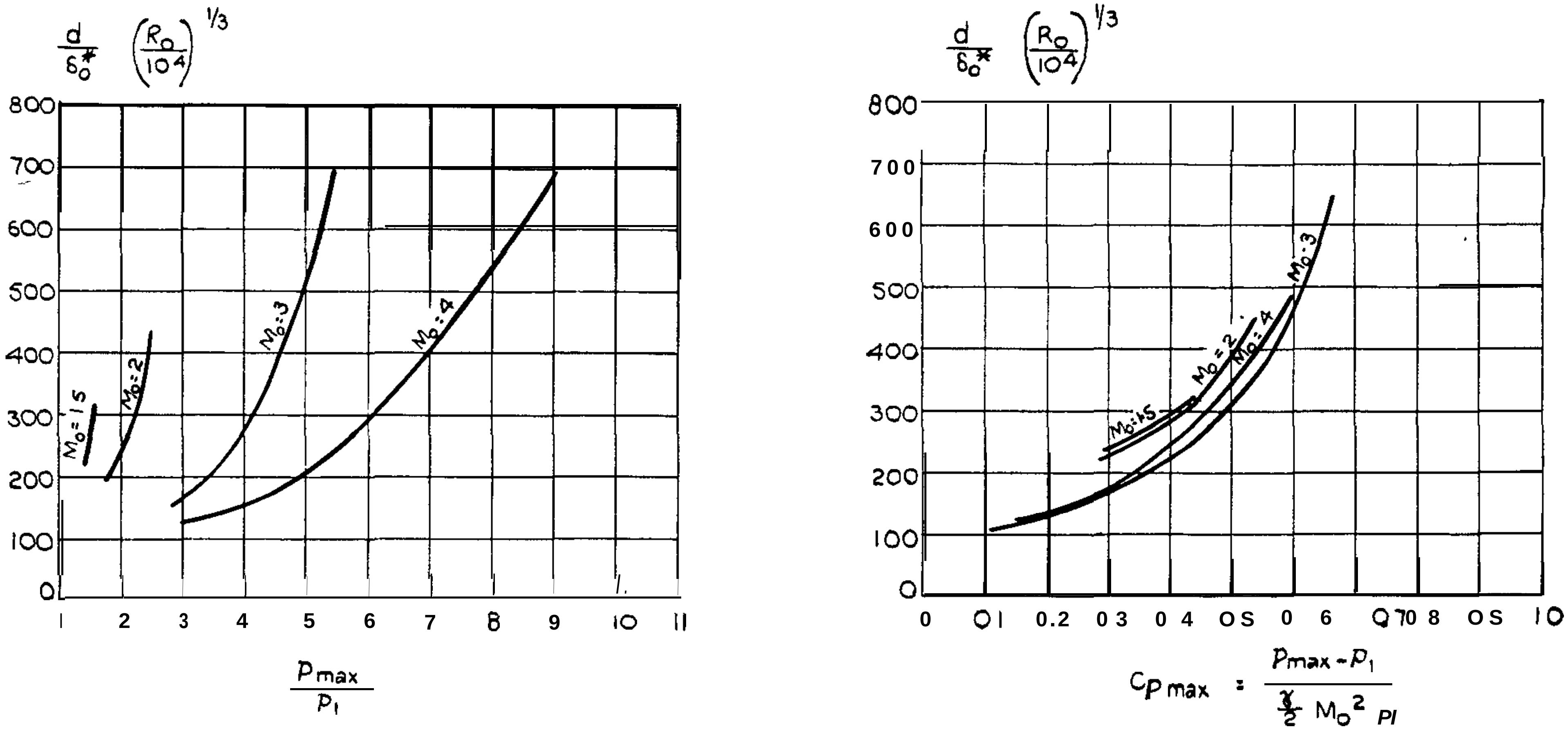


FIG 8. Boundary layers laminar at separation but turbulent before reattachment. Factor $\frac{d}{\delta_o^*}\left(\frac{R_o}{10^4}\right)^{1/3}$ as function of $\frac{p_{max}}{p_1}$ and $C_{p\,max}$

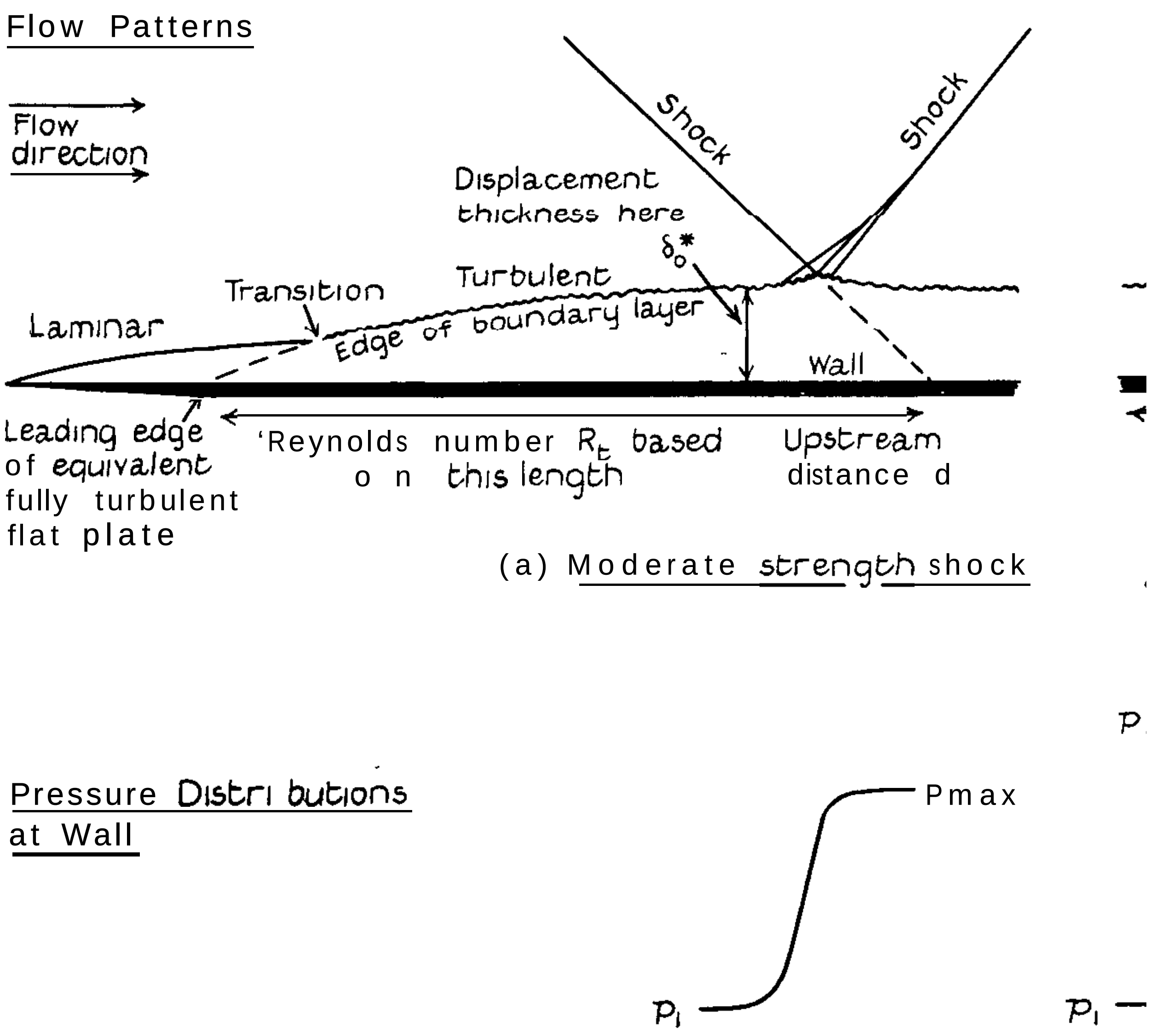
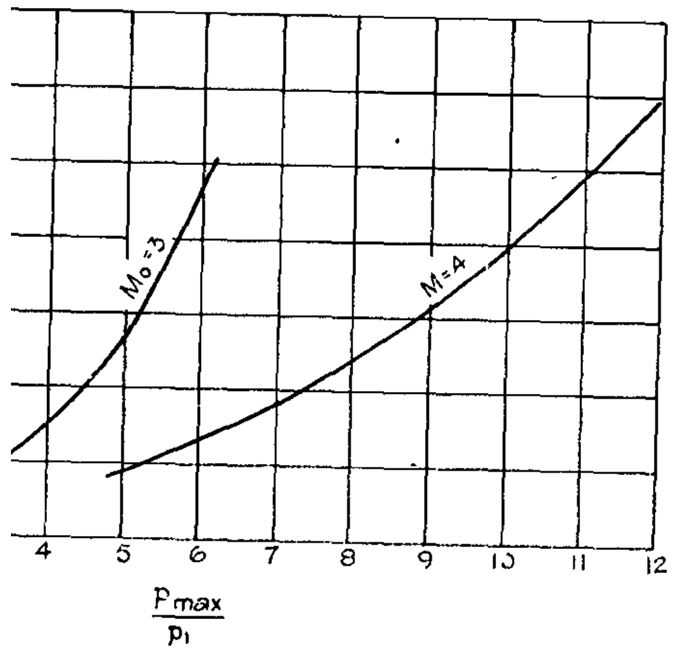
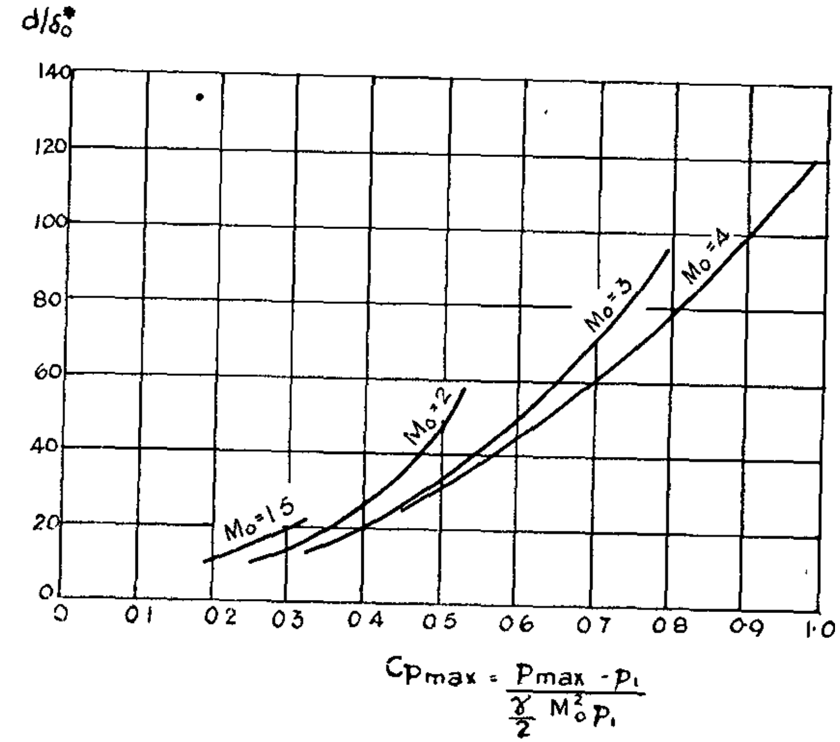


FIG 9. Boundary layers turbulent over wh





10 Boundary layers turbulent over whole region of interaction
Factor $\frac{d}{\delta_o^*}$ as function of $\frac{p_{max}}{p_1}$ and $C_{p\,max}$

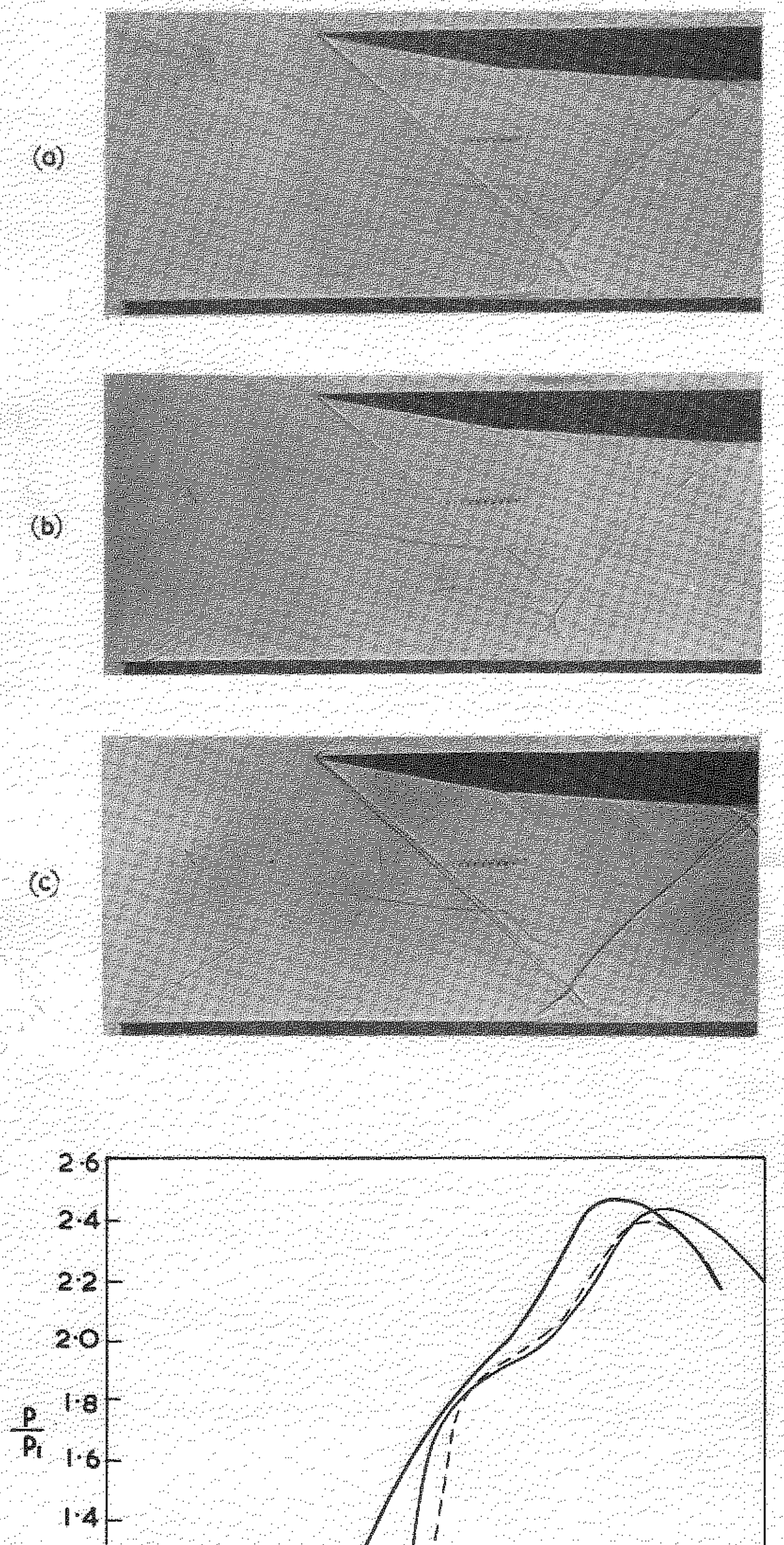


Fig.11. Comparison of effects of "artificial" and "natural" transition. Flow patterns and pressure distributions at wall at $M_0 = 2$ and :-

(a) Stagnation pressure 25" Hg abs., transition free
(b) Stagnation pressure 25" Hg abs., transition fixed
(c) Stagnation pressure 110" Hg abs., transition free

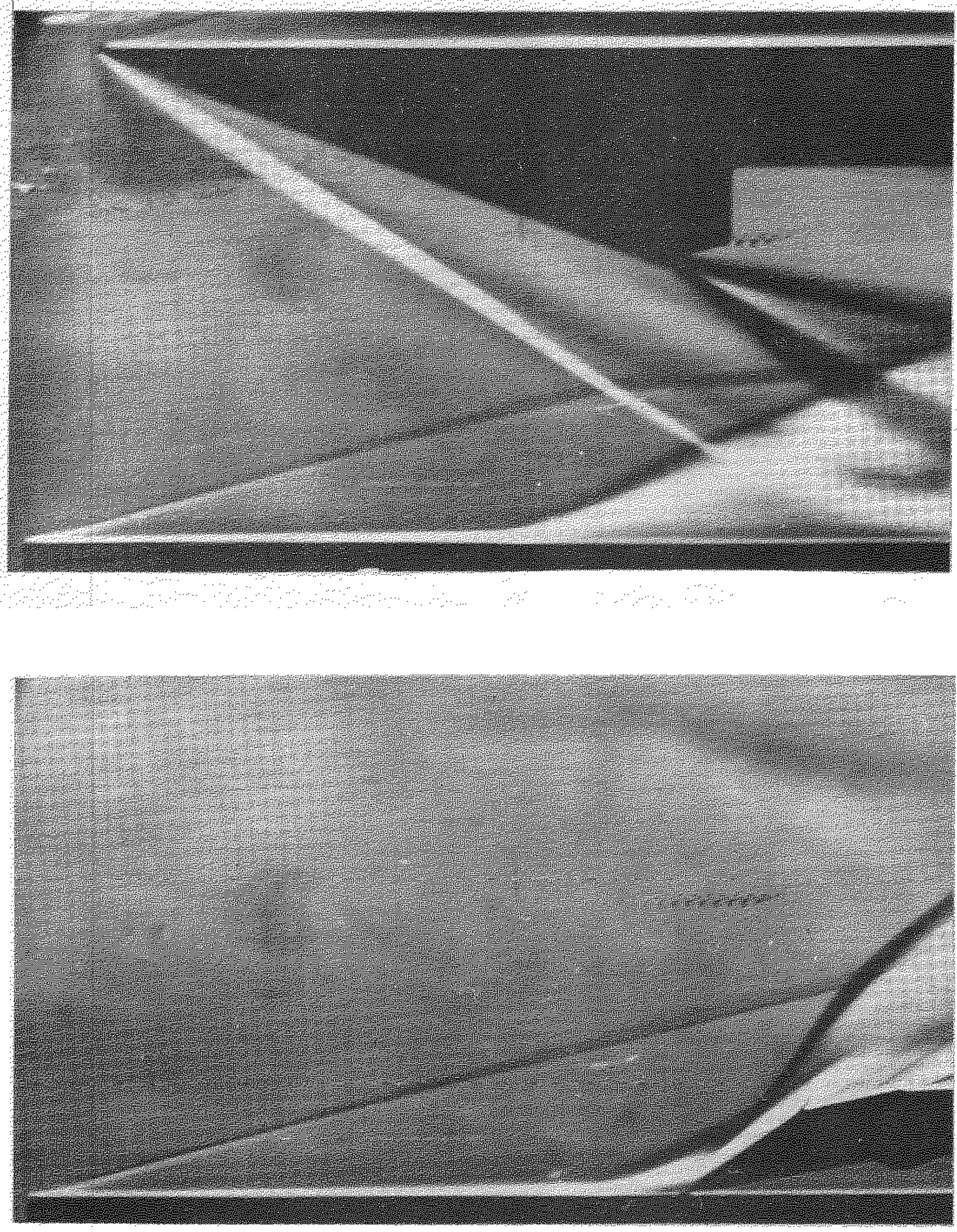


Fig.12 Turbulent boundary layers at Mach number of 4.
The pressure rise in the region of interaction is slightly greater with the wedge on the plate, but the upstream

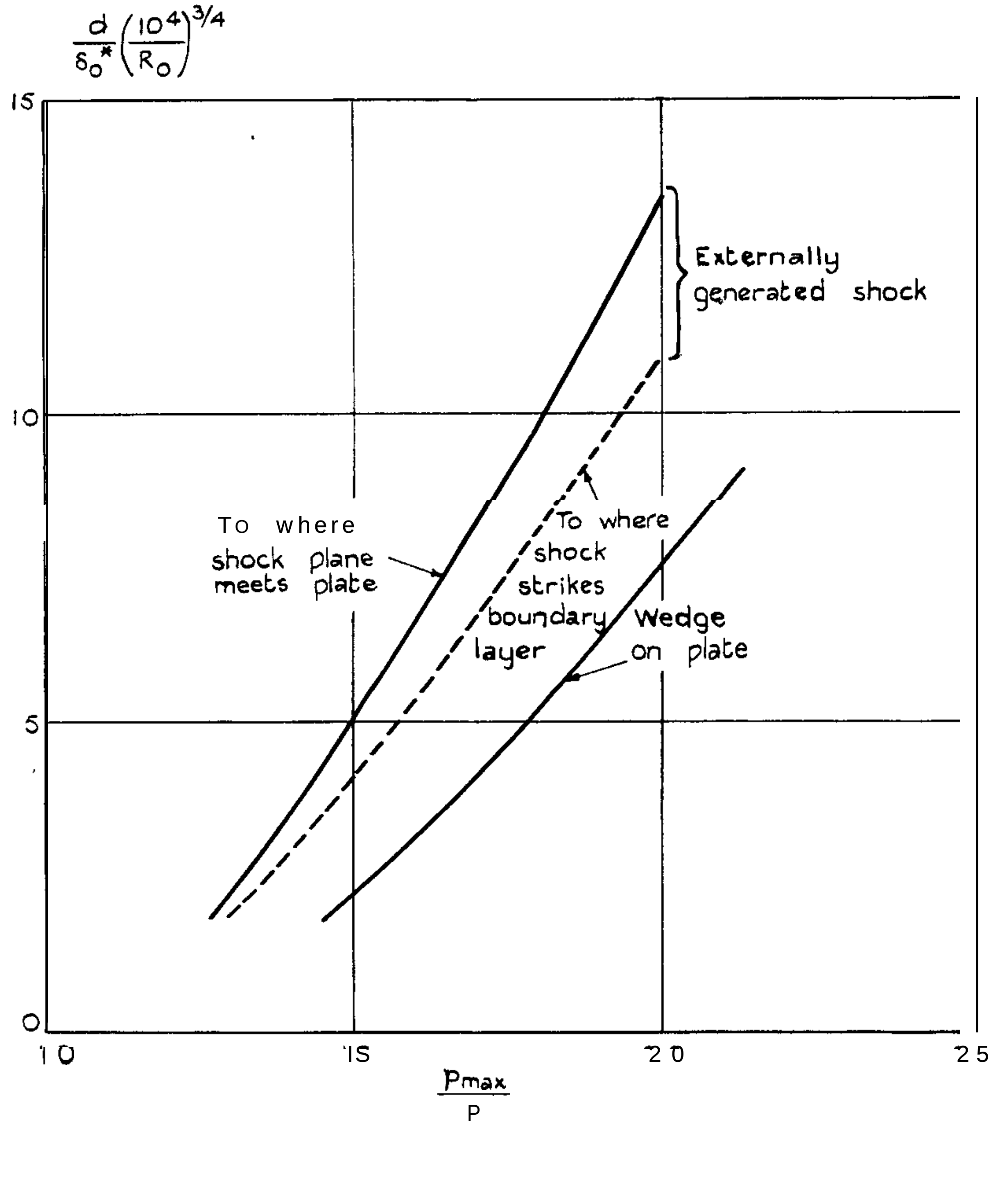


FIG 13 $M_0 = 3$ Boundary layers laminar over whole region of interaction. Comparison of upstream effects with externally-generated shock and wedge on plate

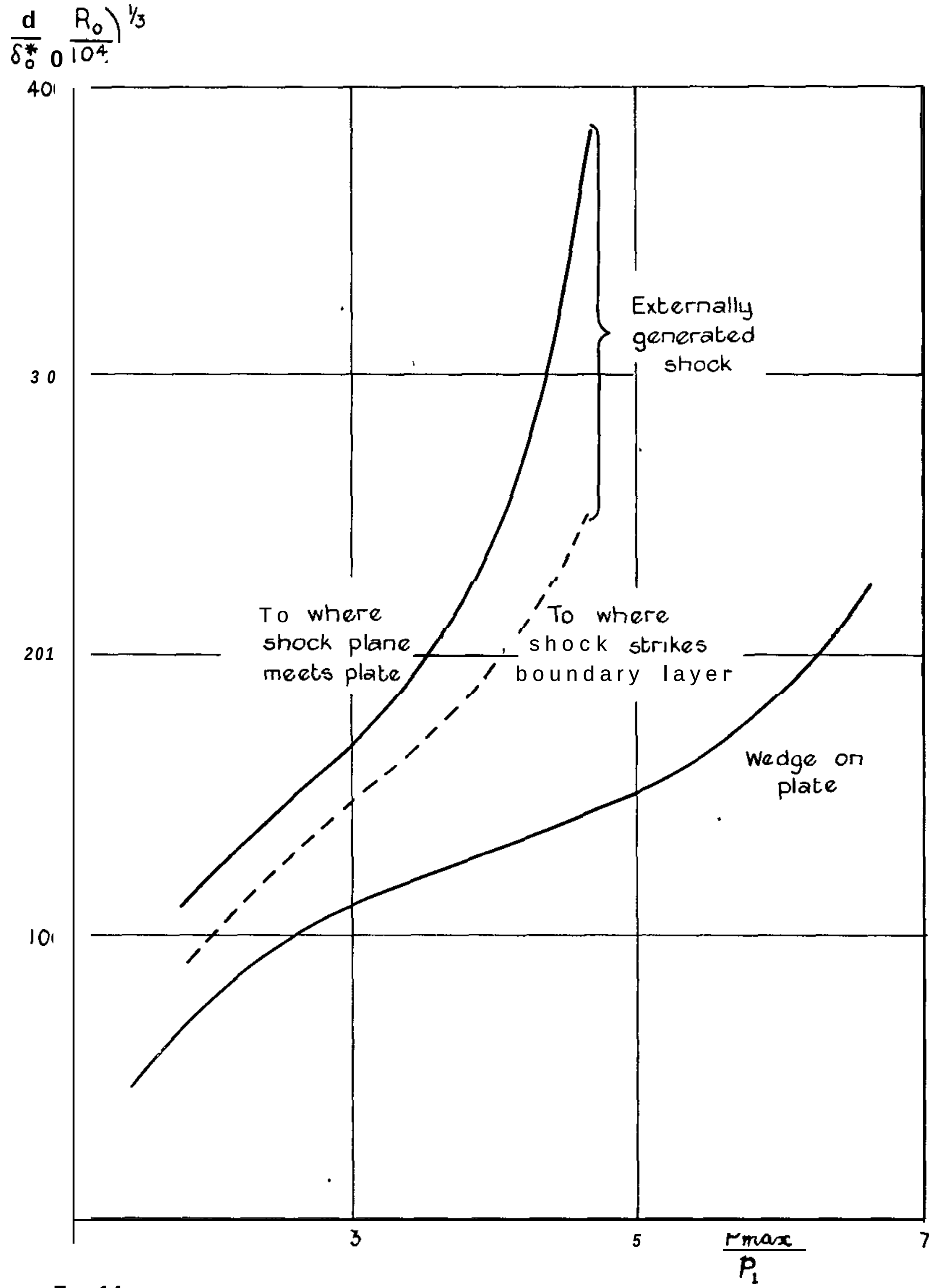


FIG. 14

$M_0 = 3$ Boundary layers laminar at separation but turbulent before reattachment Comparison of upstream effects with externally-generated shock and wedge on plate

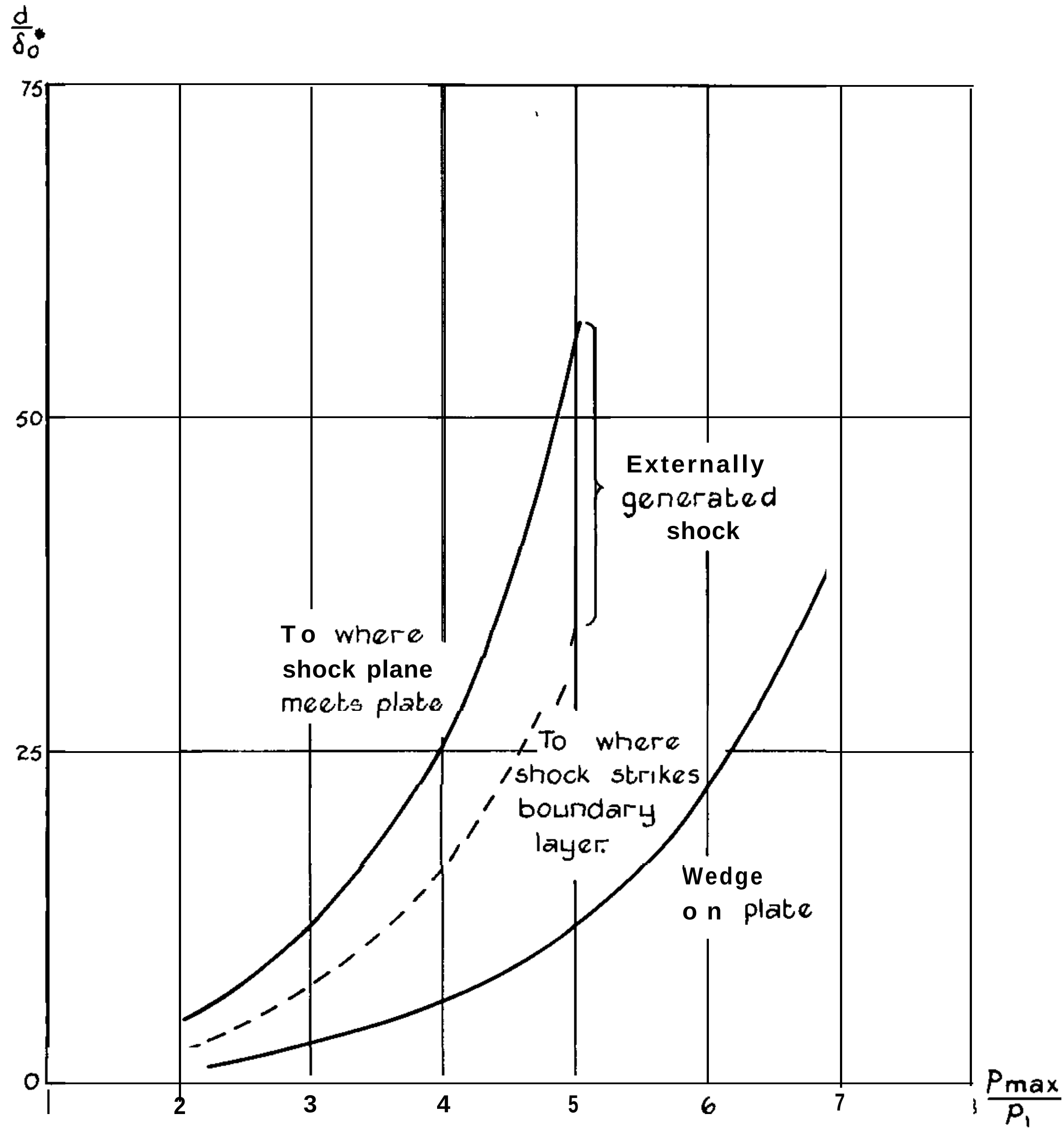
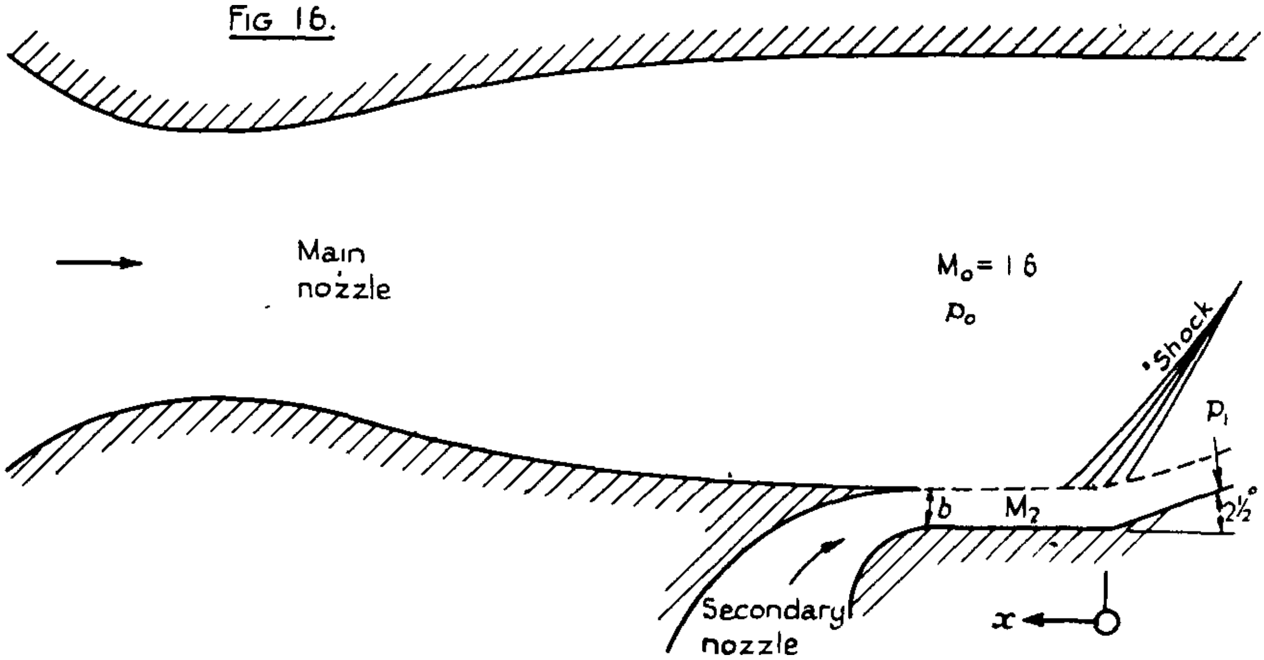


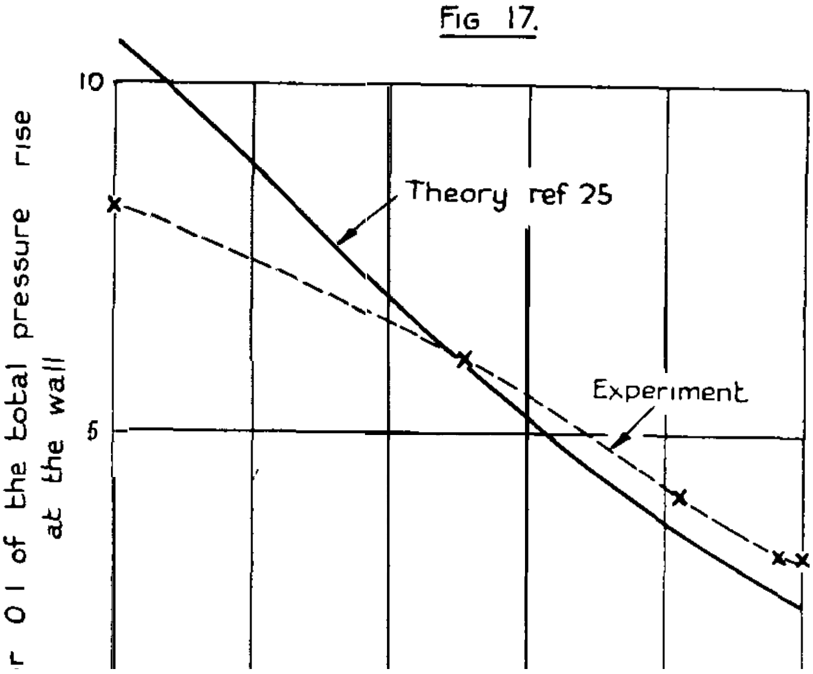
FIG. 15. $M_0 = 3$ Boundary layers turbulent over whole region of interaction. Comparison of upstream effects with externally generated shock and wedge on plate.

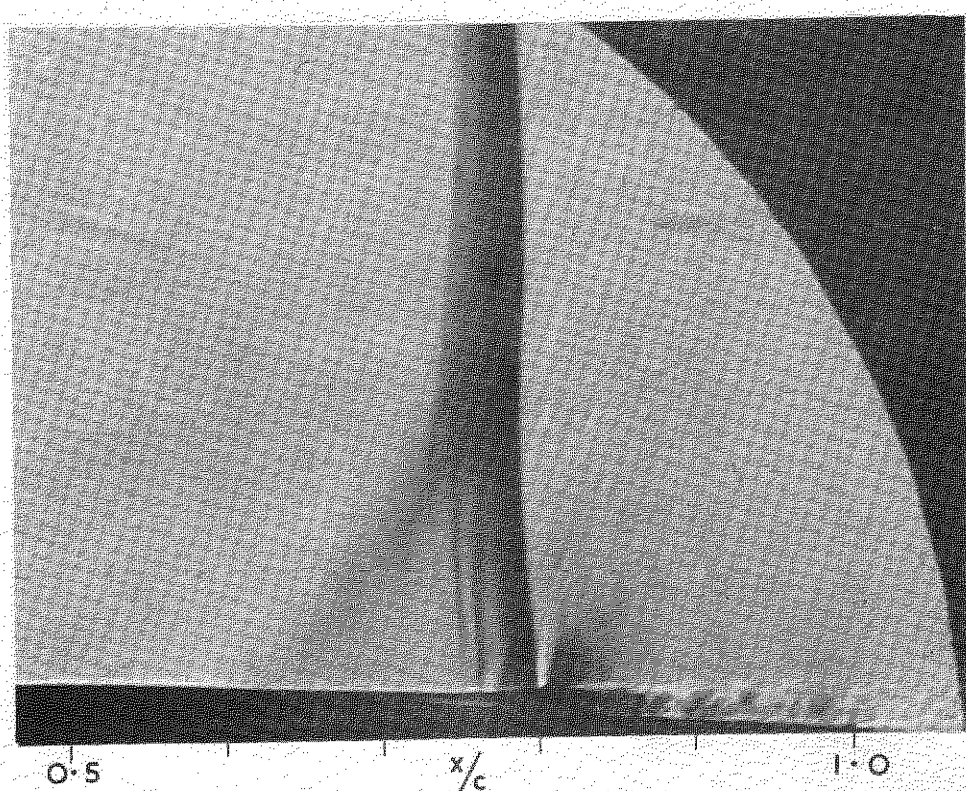
FIG 16.



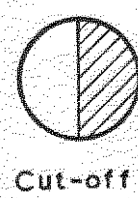
Sketch of the apparatus used for investigating the upstream effect of a wedge in a quasi - uniform subsonic stream

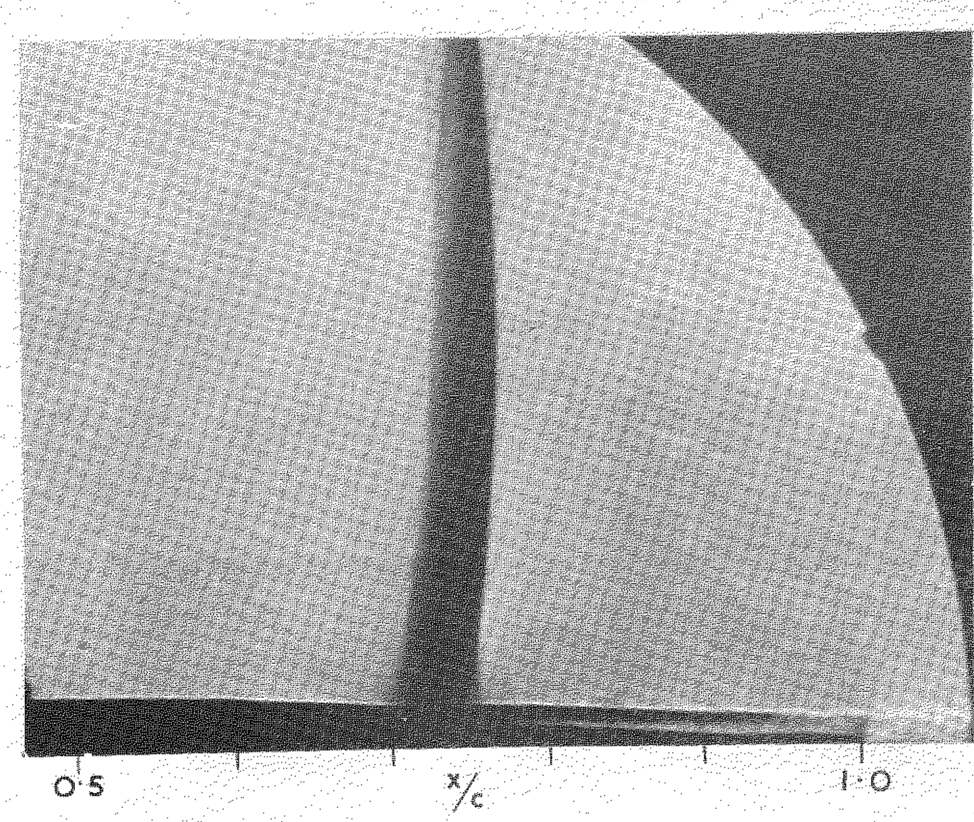
FIG 17.





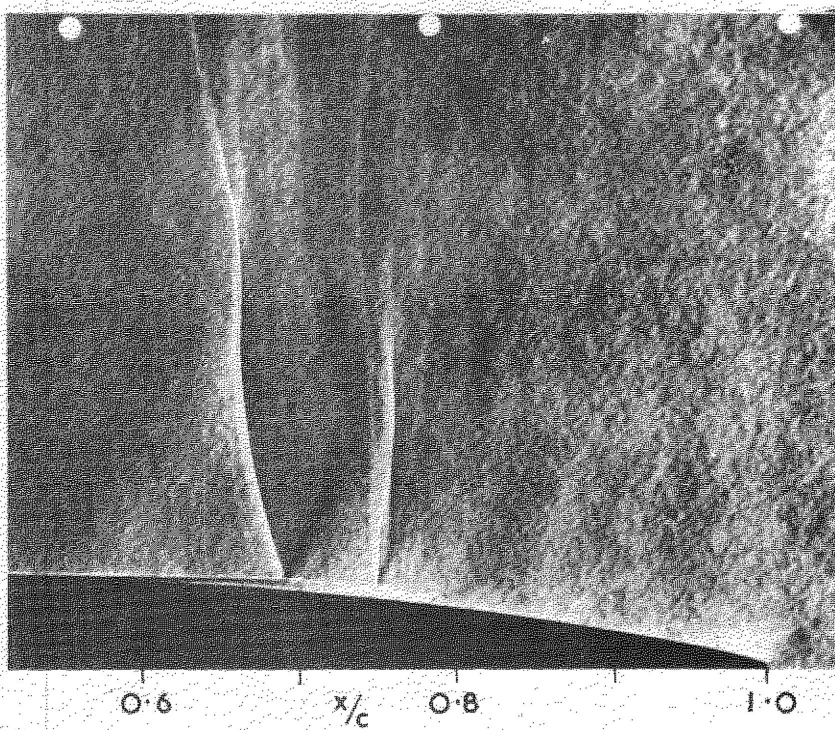
(a) With laminar boundary-layer upstream of shock.



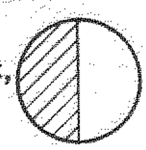


(b) With turbulent boundary-layer upstream of shock.

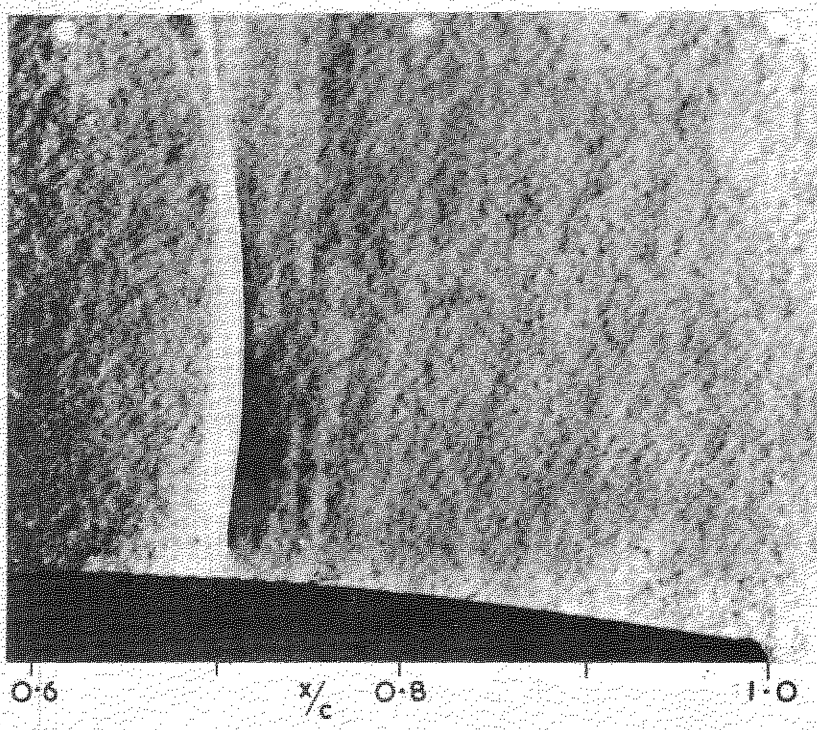
FIG. 18. Change in shock-wave pattern due to laminar boundary-layer separation upstream.
(6% thick RAE 104 Aerofoil; $M_0 = 0{\cdot}90$, $R = 1{\cdot}9 \times 10^6$, N.P.L. Tests.)



(a) With laminar boundary-layer upstream of shock, $R = 0{\cdot}8 \times 10^6$



Cut-off



(b) With turbulent boundary-layer upstream of shock, $R = 1{\cdot}7 \times 10^6$

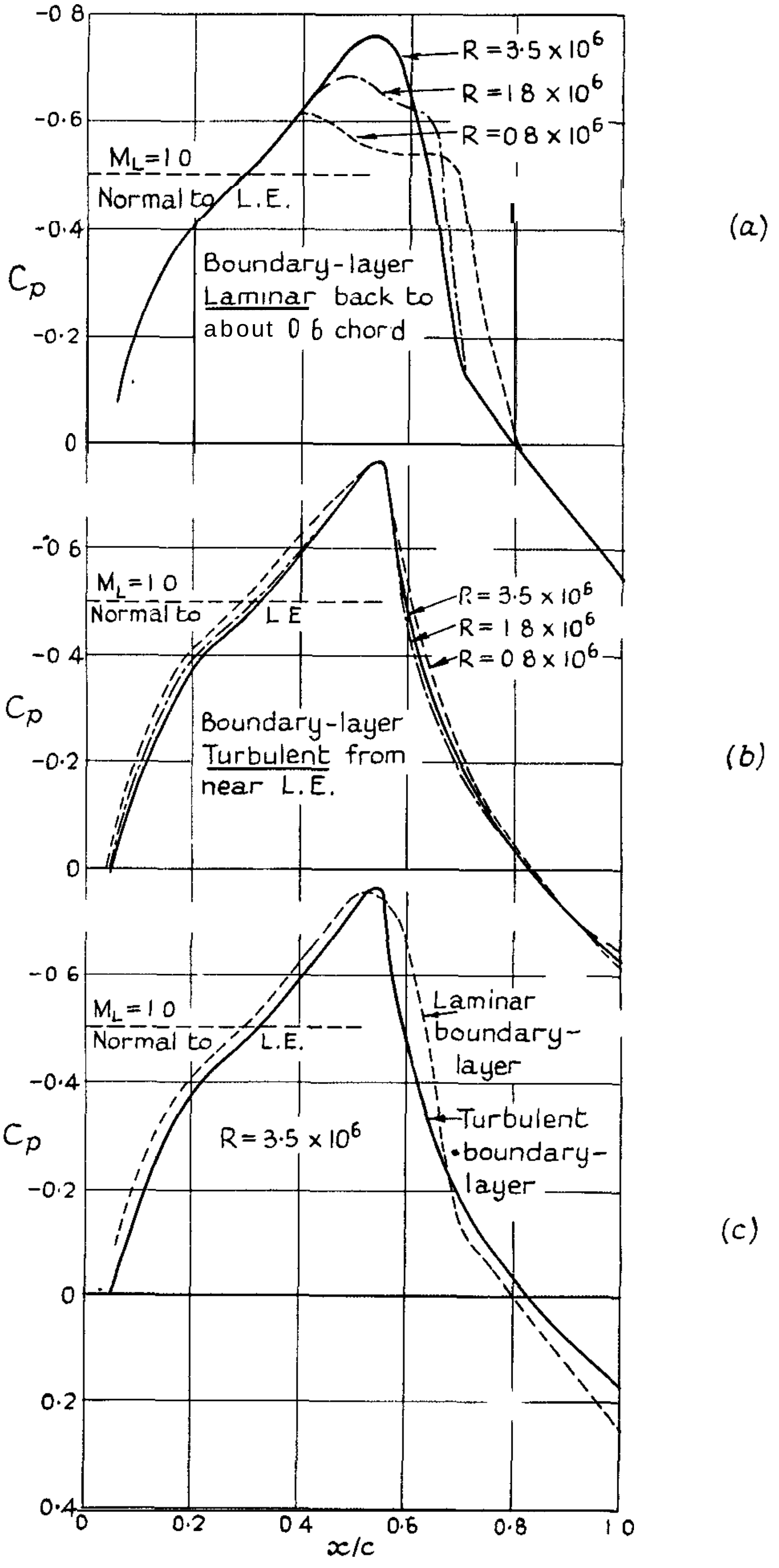
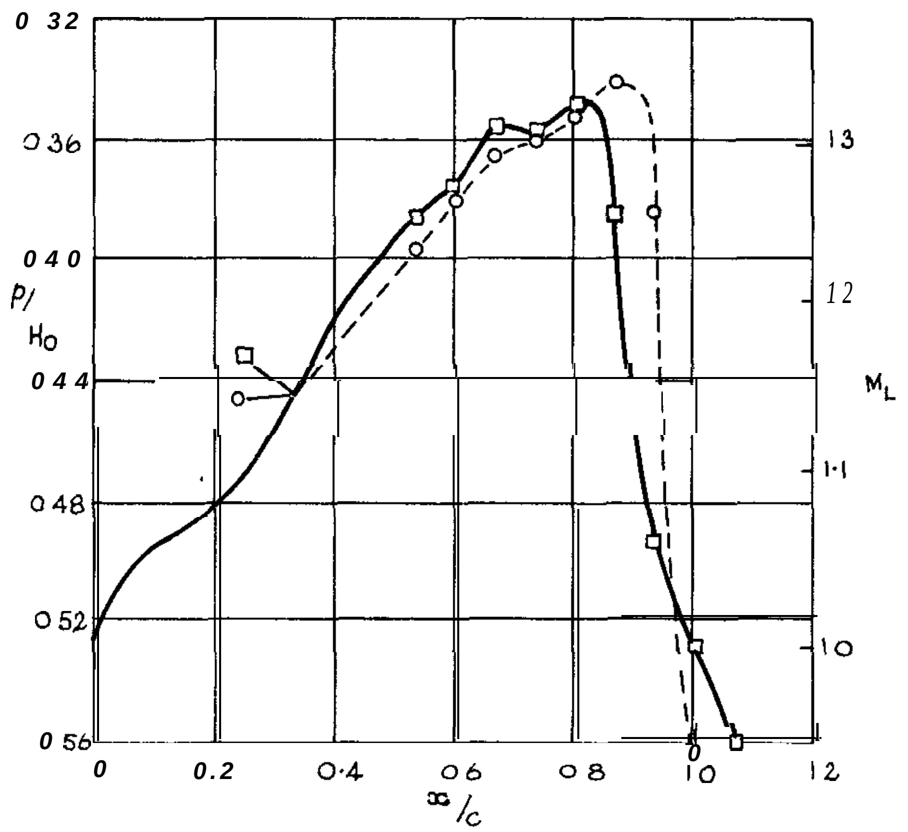


FIG. 20 (a - c).

Effect of Reynolds' number and of fixing transition in a wind - tunnel experiment (ref. 38).

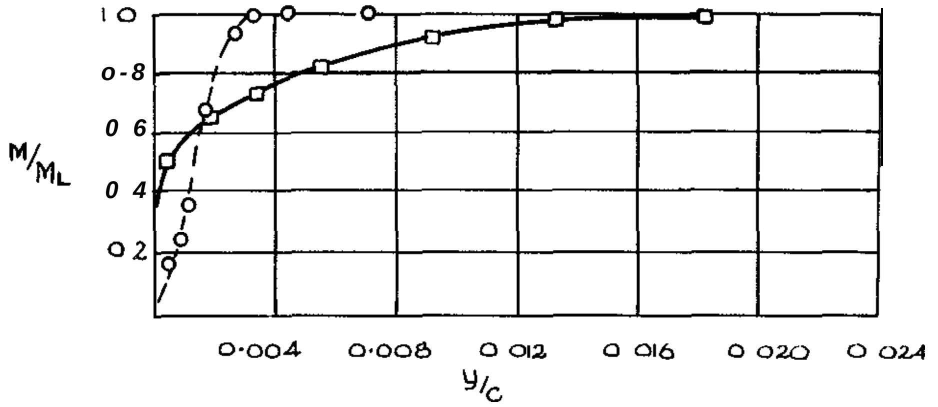
$M_0 = 0{\cdot}82$



Surface - pressure distributions

---o--- Smooth wing (laminar boundary - layer back to shock)

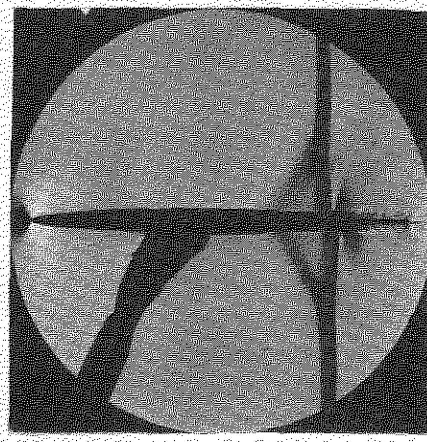
——□—— Transition fixed by wire at about 0 05 chord



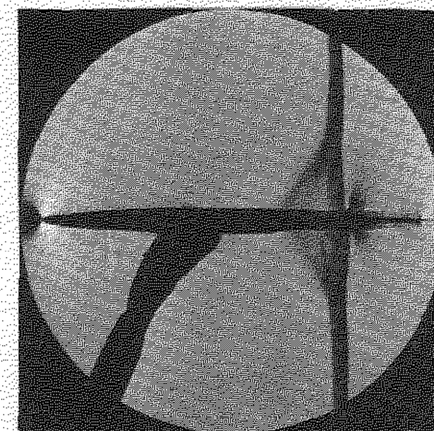
Boundary - layer profiles immediately ahead of shock

FIG. 20d Effect of fixing transition in flight, on a special smooth wing (ref. 40) $M_0 \approx 0.70$

(a) Schlieren photographs; air-injection on lower surface only.

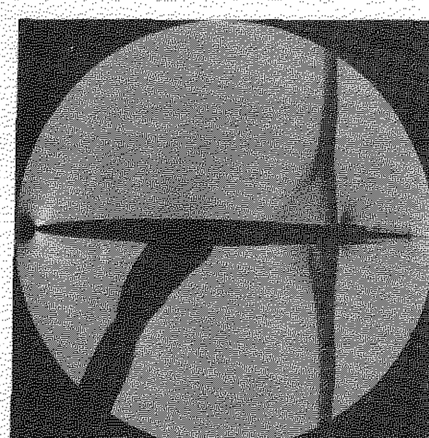


No injection (laminar)

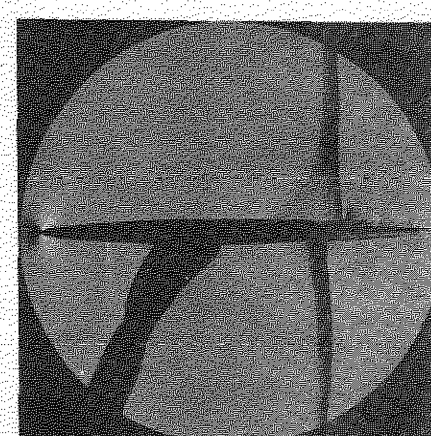


$\frac{m}{m_b} = 0{\cdot}007$

($m \equiv$ mass of air injected,
$m_b \equiv$ mass flow in boundary layer)

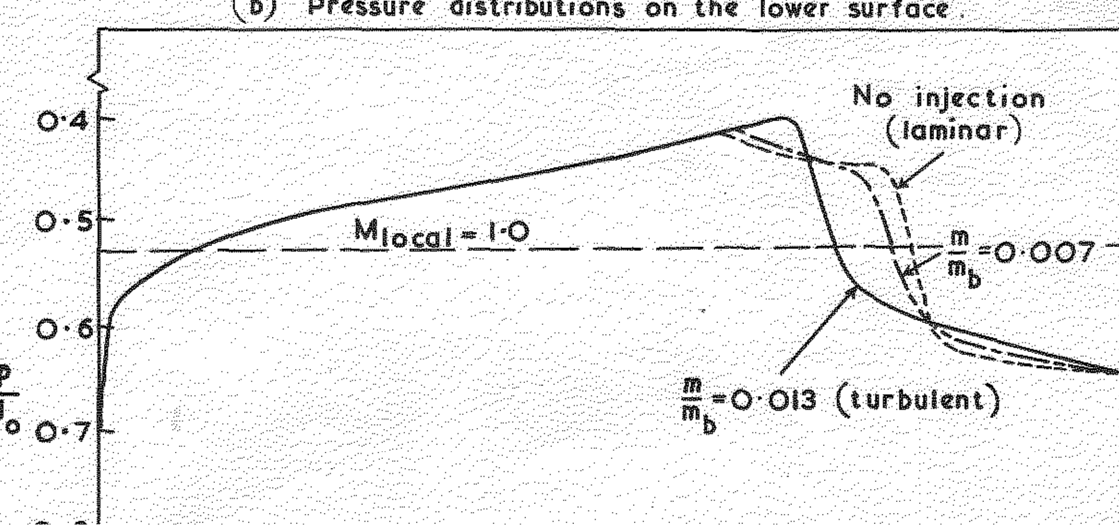


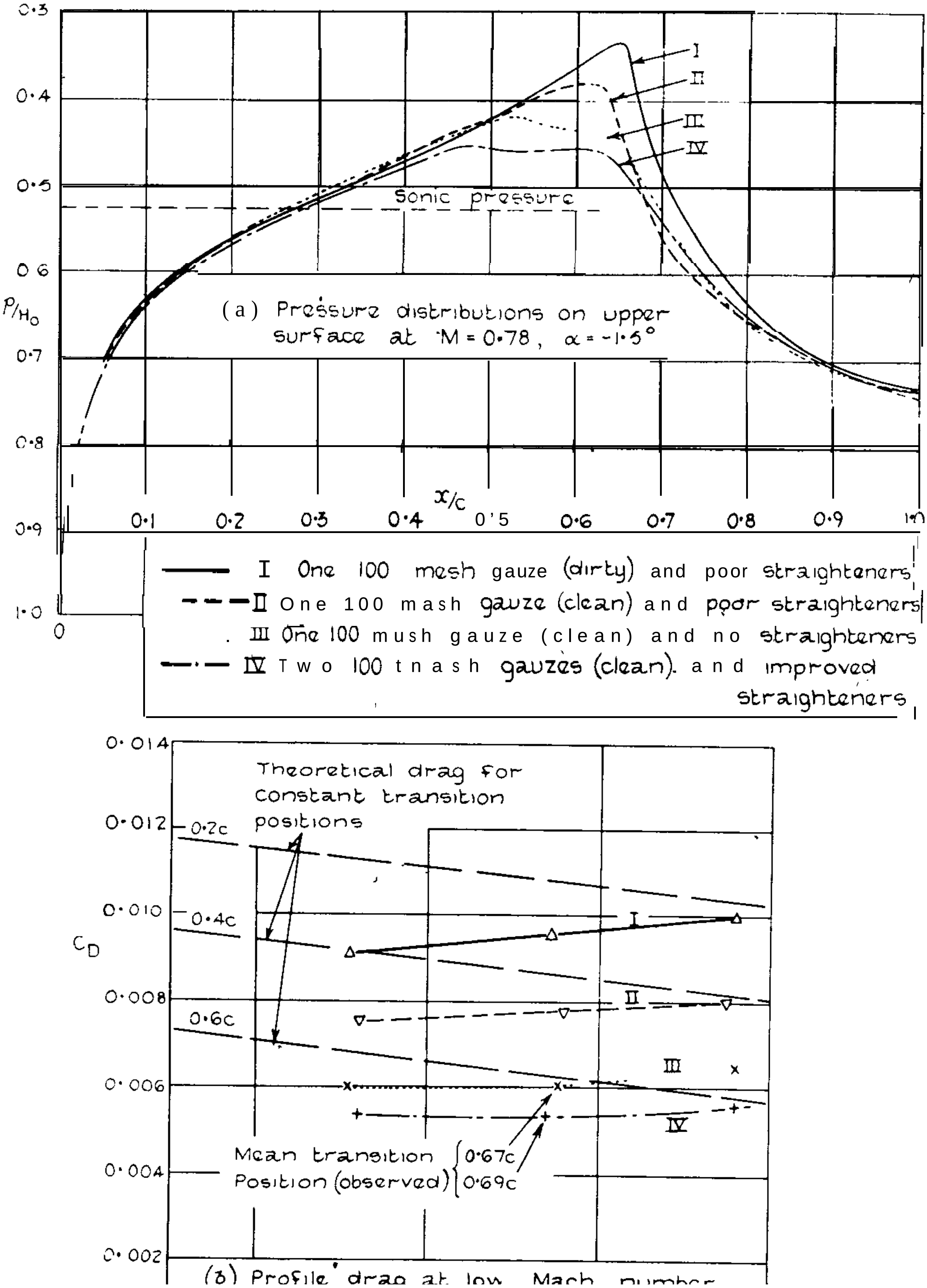
$\frac{m}{m_b} = 0{\cdot}01$



$\frac{m}{m_b} = 0{\cdot}013$ (turbulent)

(b) Pressure distributions on the lower surface.





0·3
0·4
0·5
0 6
0·7
0·8
0·9
1·0
p/H_0
I
II
III
IV
Sonic pressure
(a) Pressure distributions on upper surface at M = 0·78, α = -1·5°
x/c
0
0·1
0·2
0·3
0·4
0·5
0·6
0·7
0·8
0·9
1·0
I One 100 mesh gauze (dirty) and poor straighteners
II One 100 mash gauze (clean) and poor straighteners
III One 100 mush gauze (clean) and no straighteners
IV Two 100 tnash gauzes (clean) and improved straighteners
0·014
0·012
0·010
0·008
0·006
0·004
0·002
C_D
Theoretical drag for constant transition positions
0·2c
0·4c
0·6c
I
II
III
IV
Mean transition Position (observed)
0·67c
0·69c
(b) Profile drag at low Mach number

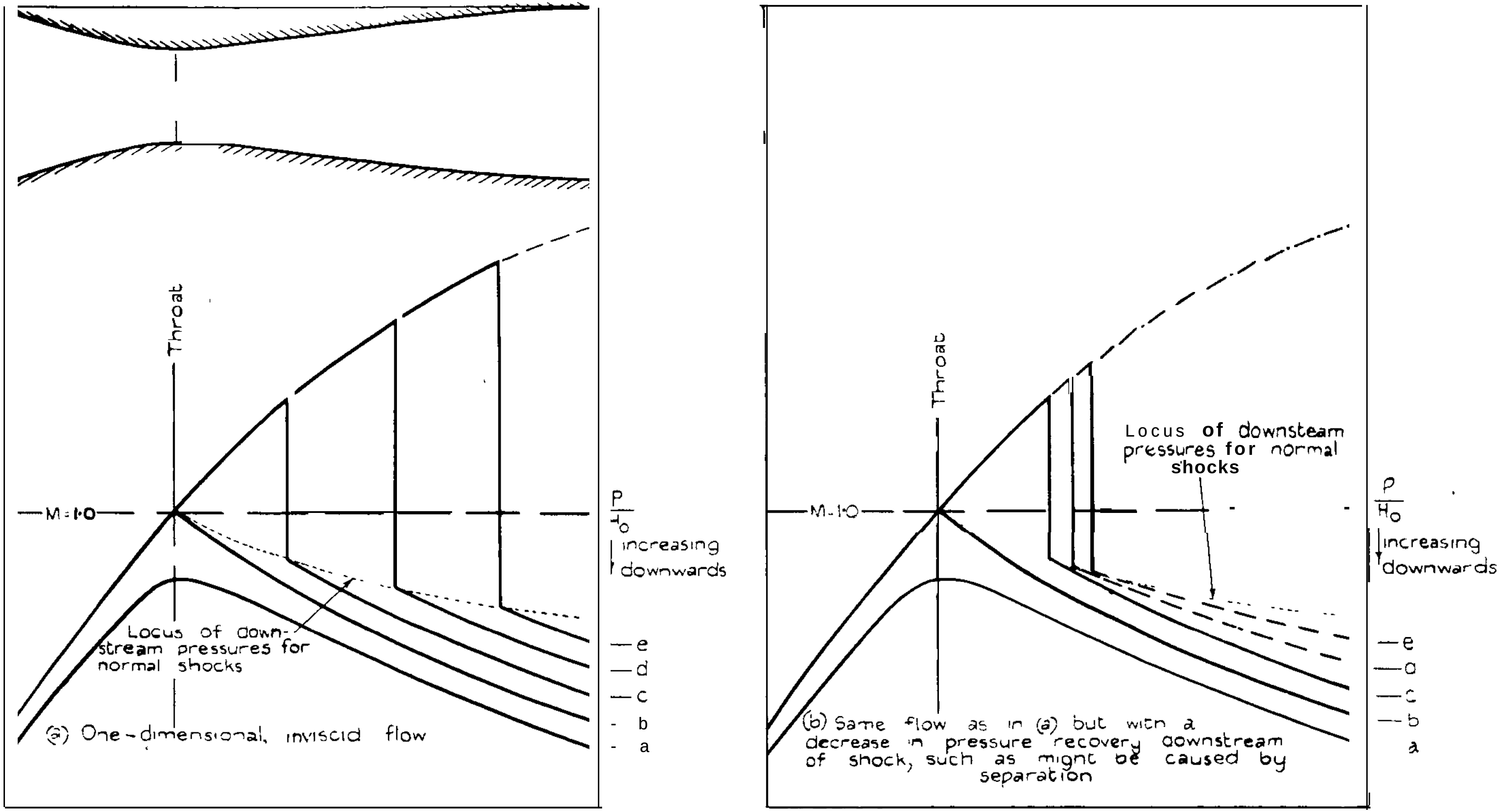
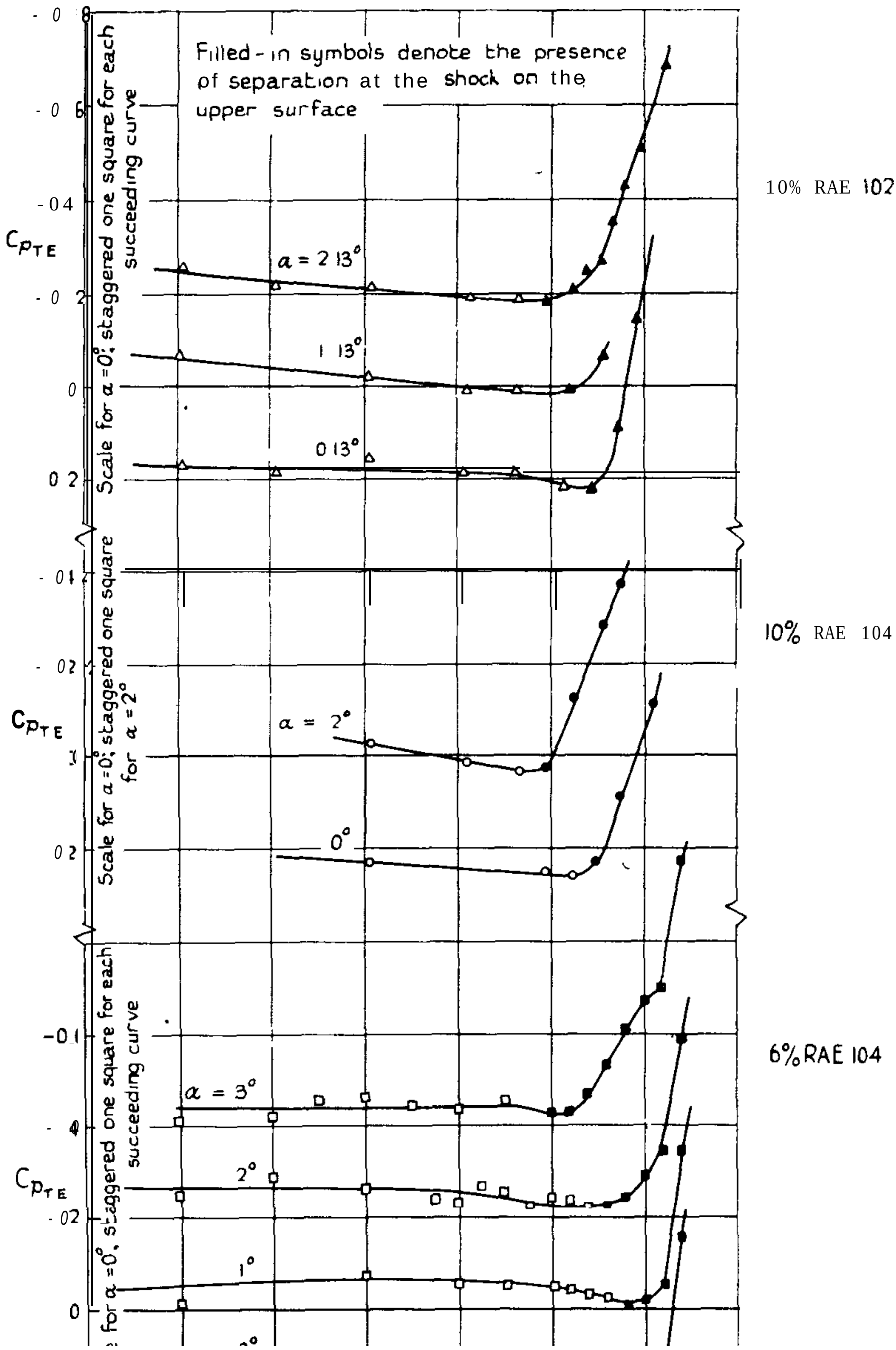


FIG 23 Shock movement through a convergent-divergent nozzle for a certain rate of decrease in exit pressure (diagrammatic)



Filled-in symbols denote the presence of separation at the shock on the upper surface
Scale for α = 0°; staggered one square for each succeeding curve
$C_{p_{TE}}$
α = 2·13°
1·13°
0·13°
10% RAE 102
Scale for α = 0°; staggered one square for α = 2°
α = 2°
0°
10% RAE 104
Scale for α = 0°, staggered one square for each succeeding curve
α = 3°
2°
1°
6% RAE 104

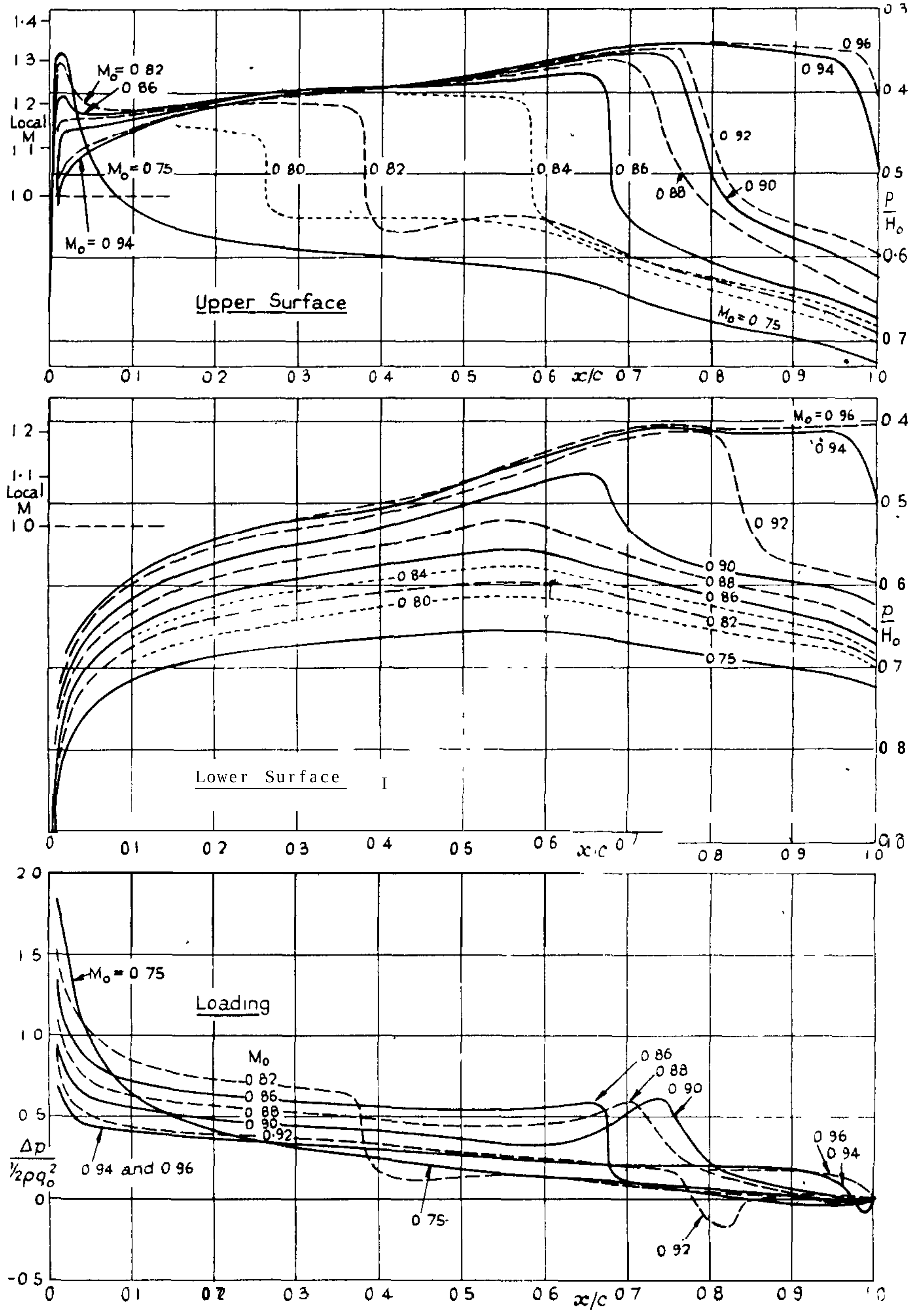
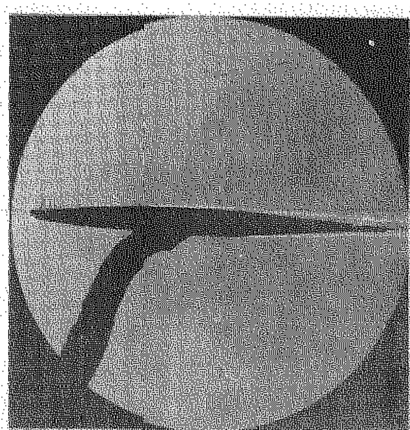


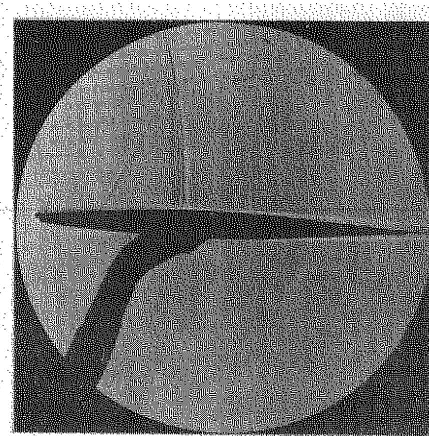
FIG 25.

Surface-pressure distributions for 6% RAE 104 aerofoil at 2° incidence

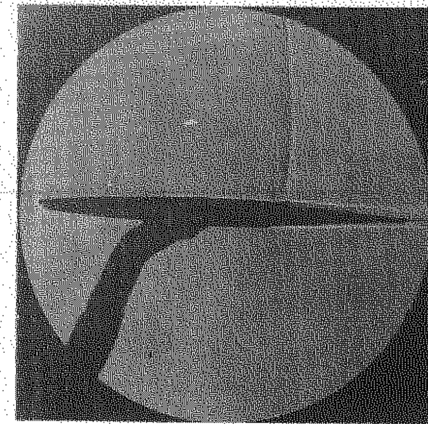
(NPL tests, transition fixed, $R \approx 1{\cdot}8 \times 10^6$)



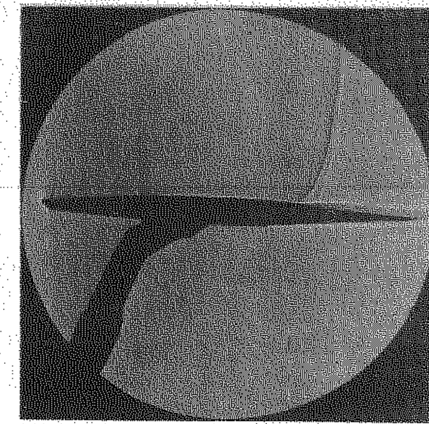
$M_o = 0{\cdot}75$



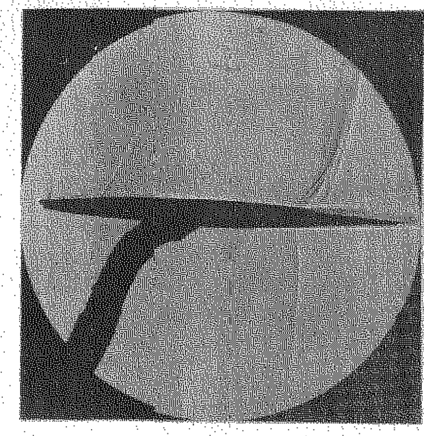
$M_o = 0{\cdot}82$



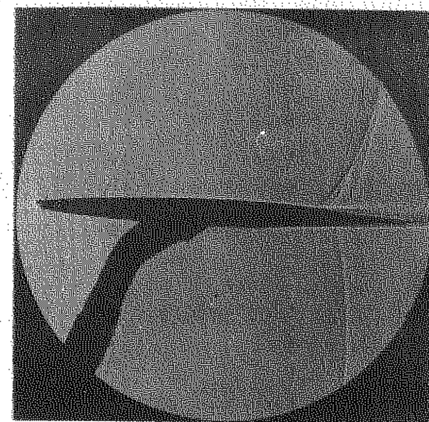
$M_o = 0{\cdot}86$



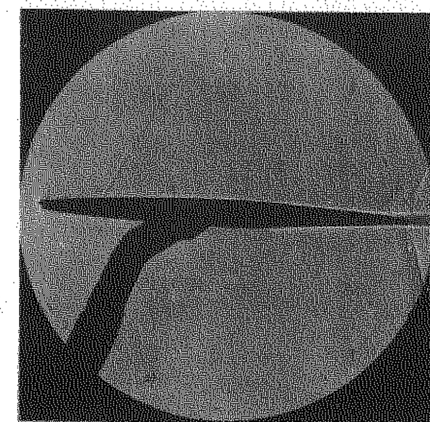
$M_o = 0{\cdot}88$



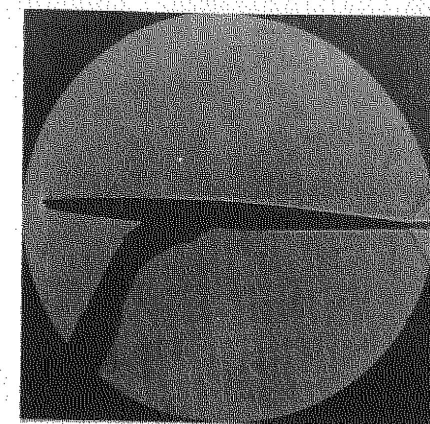
$M_o = 0{\cdot}90$



$M_o = 0{\cdot}92$



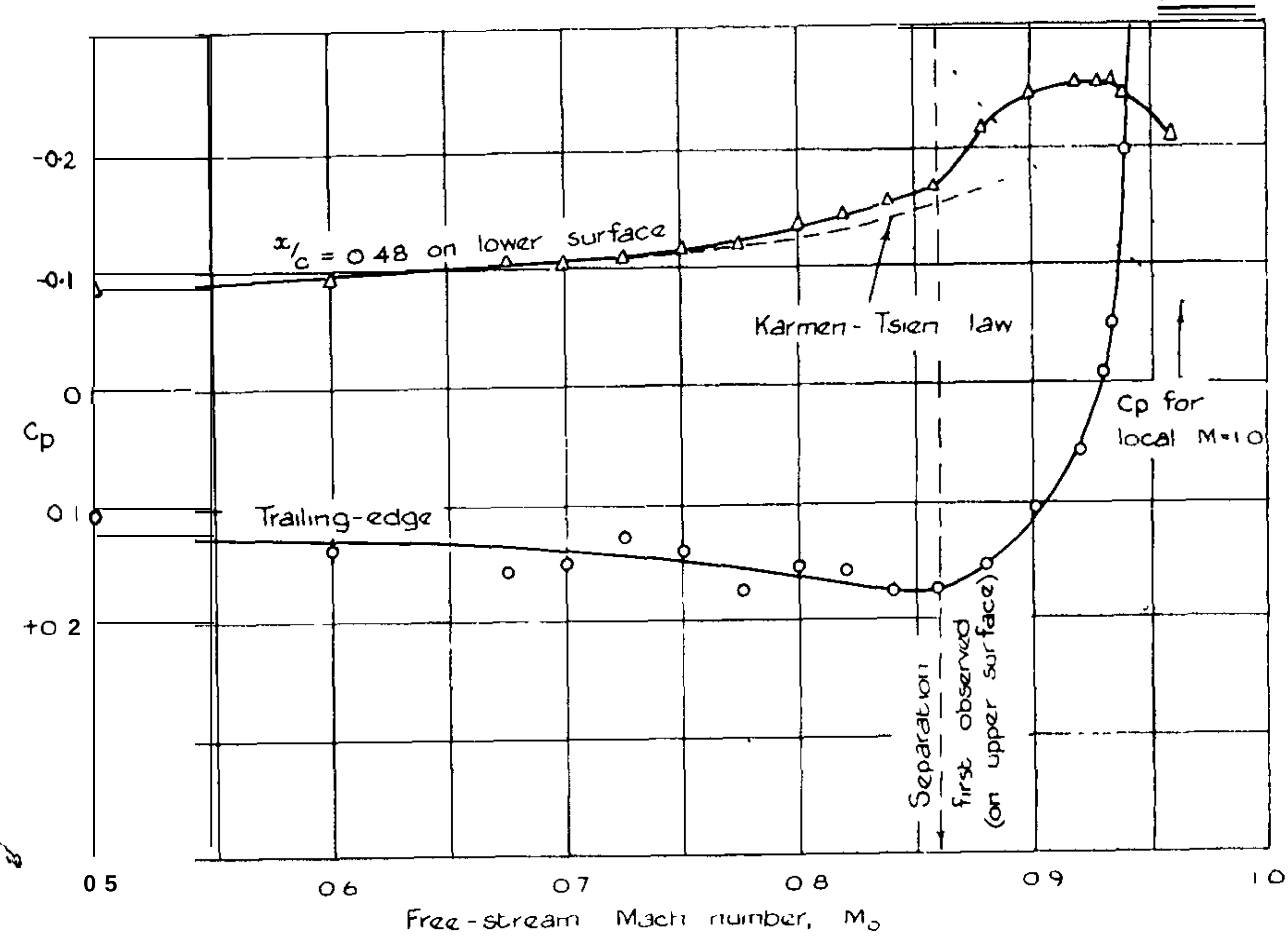
$M_o = 0{\cdot}94$



$M_o = 0{\cdot}96$

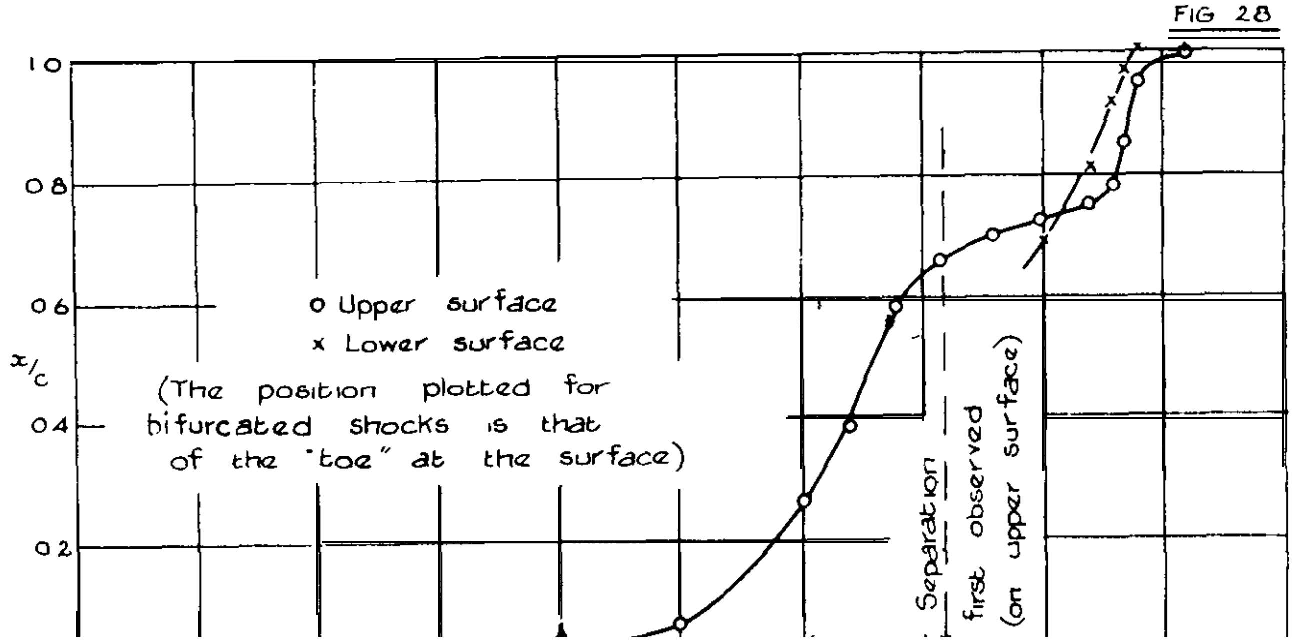
FIG. 26. Direct-shadow photographs for 6% RAE 104 Aerofoil at 2° incidence.
(N.P.L. Tests; Transition fixed upstream of shocks, $R \approx 1{\cdot}8 \times 10^6$).

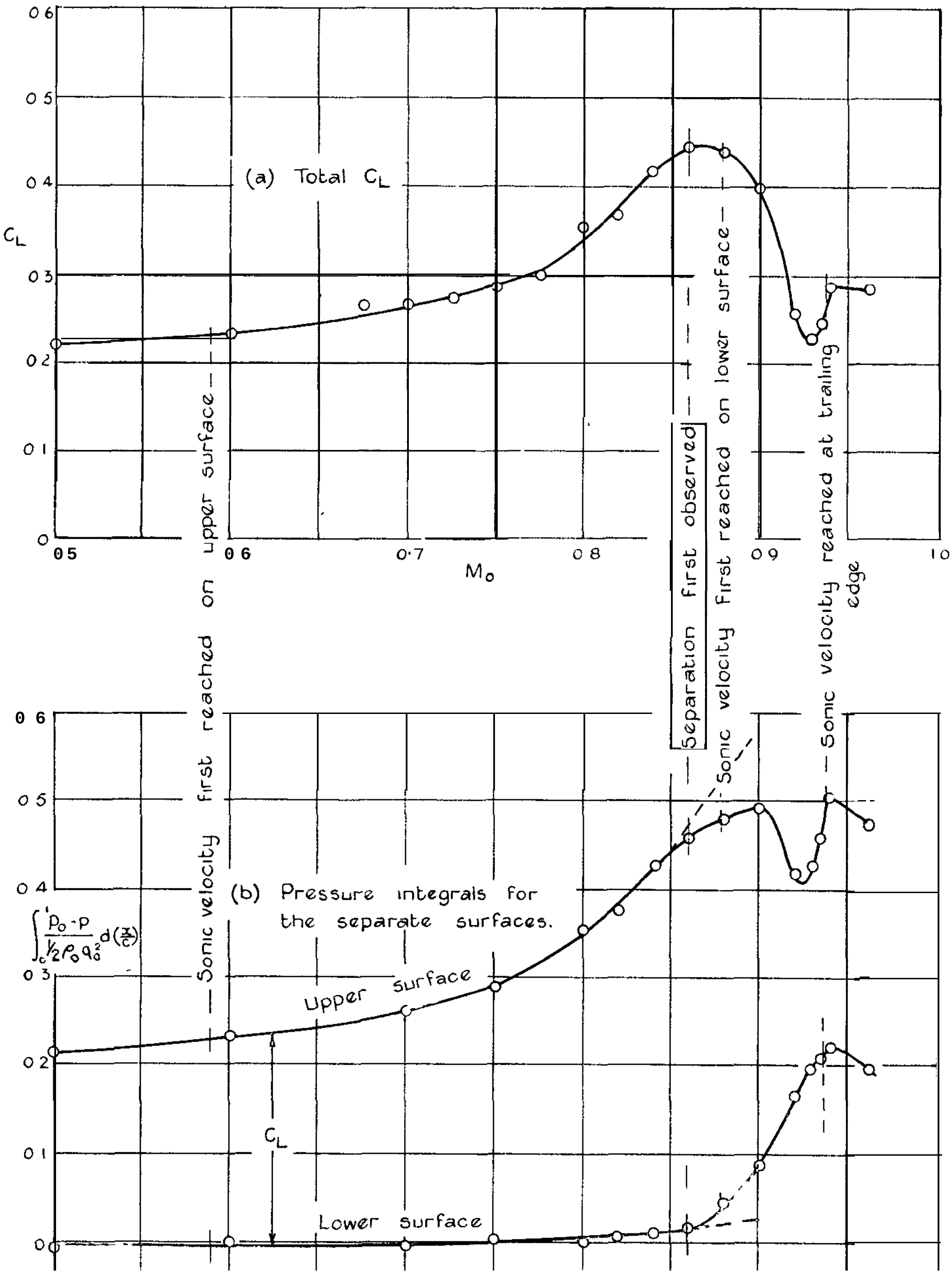
FIG 27



Variation of Cp for fixed points on the surface of 6% RAE 104 aerofoil at 2° incidence
(NPL tests, Transition fixed, $R \approx 1 \cdot 8 \times 10^6$)

FIG 28




(a) Total C_L
C_L
M_0
0.5
0.6
0.7
0.8
0.9
1.0
0
0.1
0.2
0.3
0.4
0.5
0.6
Sonic velocity first reached on upper surface
Separation first observed
Sonic velocity first reached on lower surface
Sonic velocity reached at trailing edge
(b) Pressure integrals for the separate surfaces.
$\int_c^1 \frac{p_0 - p}{\frac{1}{2}\rho_0 q_0^2} d\left(\frac{x}{c}\right)$
Upper surface
Lower surface
C_L

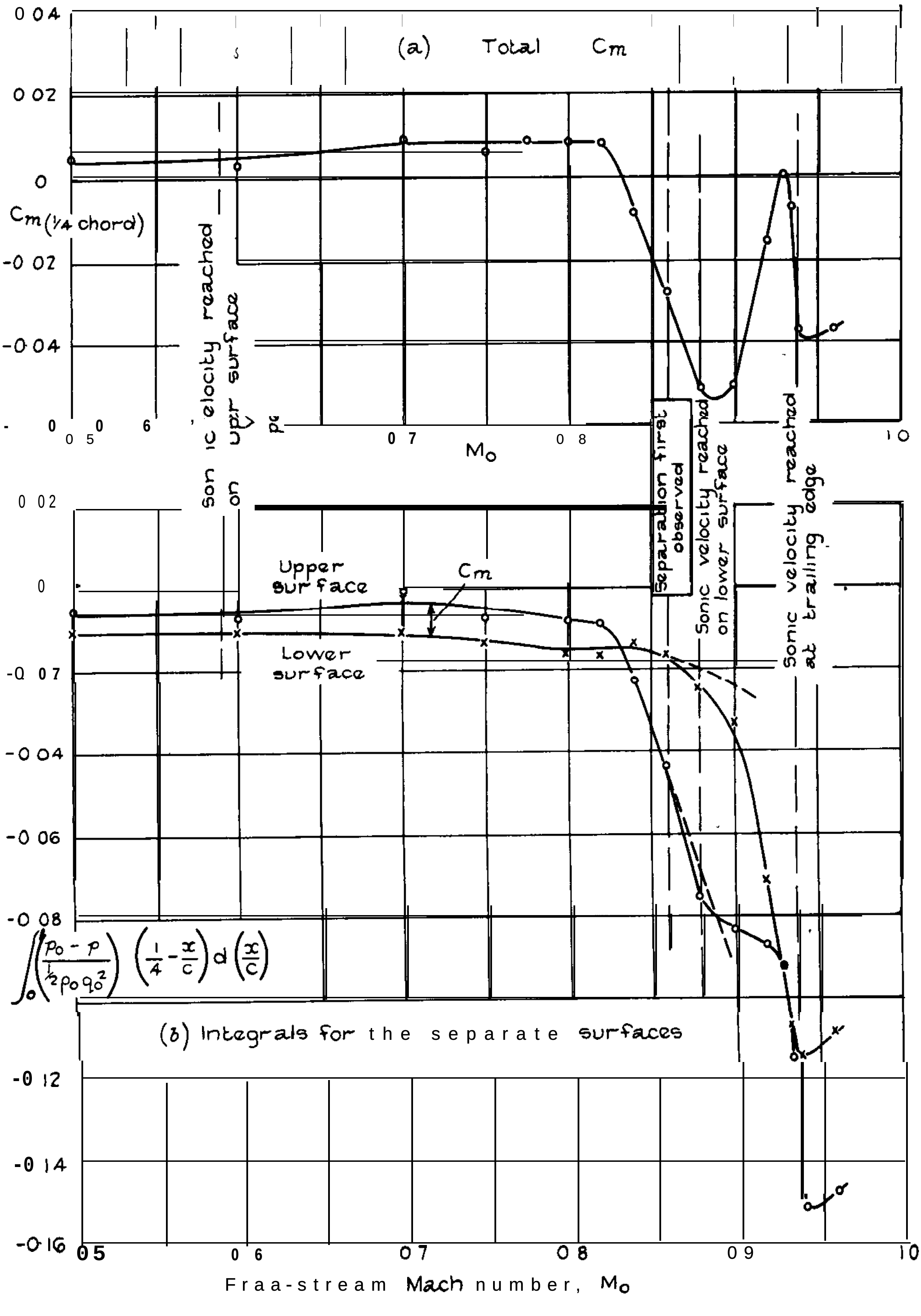


FIG 30.
Variation of Cm (1/4 chord) For 6% RAE 104 aerofoil at 2° Ix&-
(NPL tests, Transition fixed, $R \approx 1\ 8 \times 10^6$)

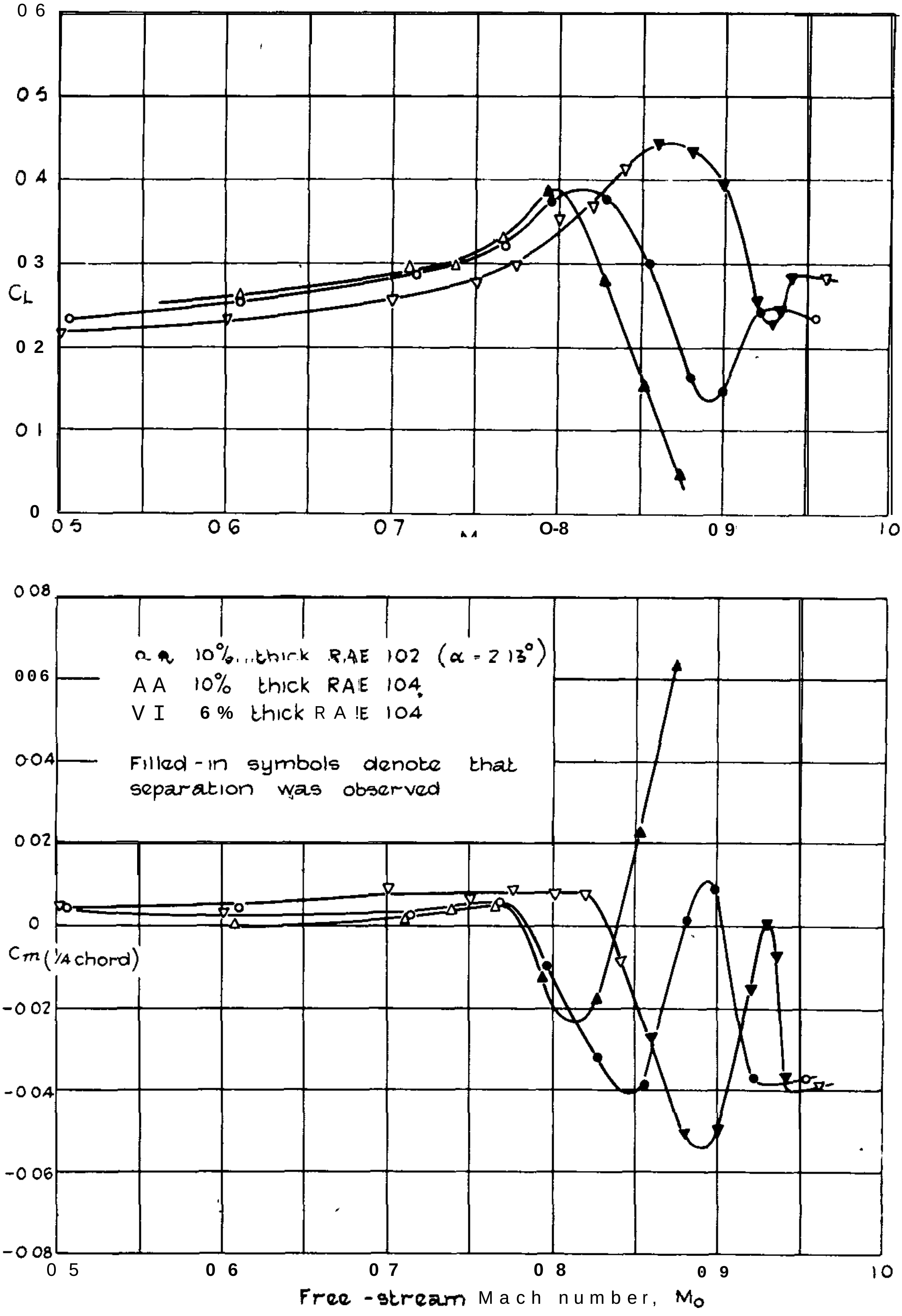


FIG 31. Variation of C_L and C_m for three aerofoil sections each at 2° incidence

(NPL tests, Transition fixed, $R \approx 1.8 \times 10^6$)

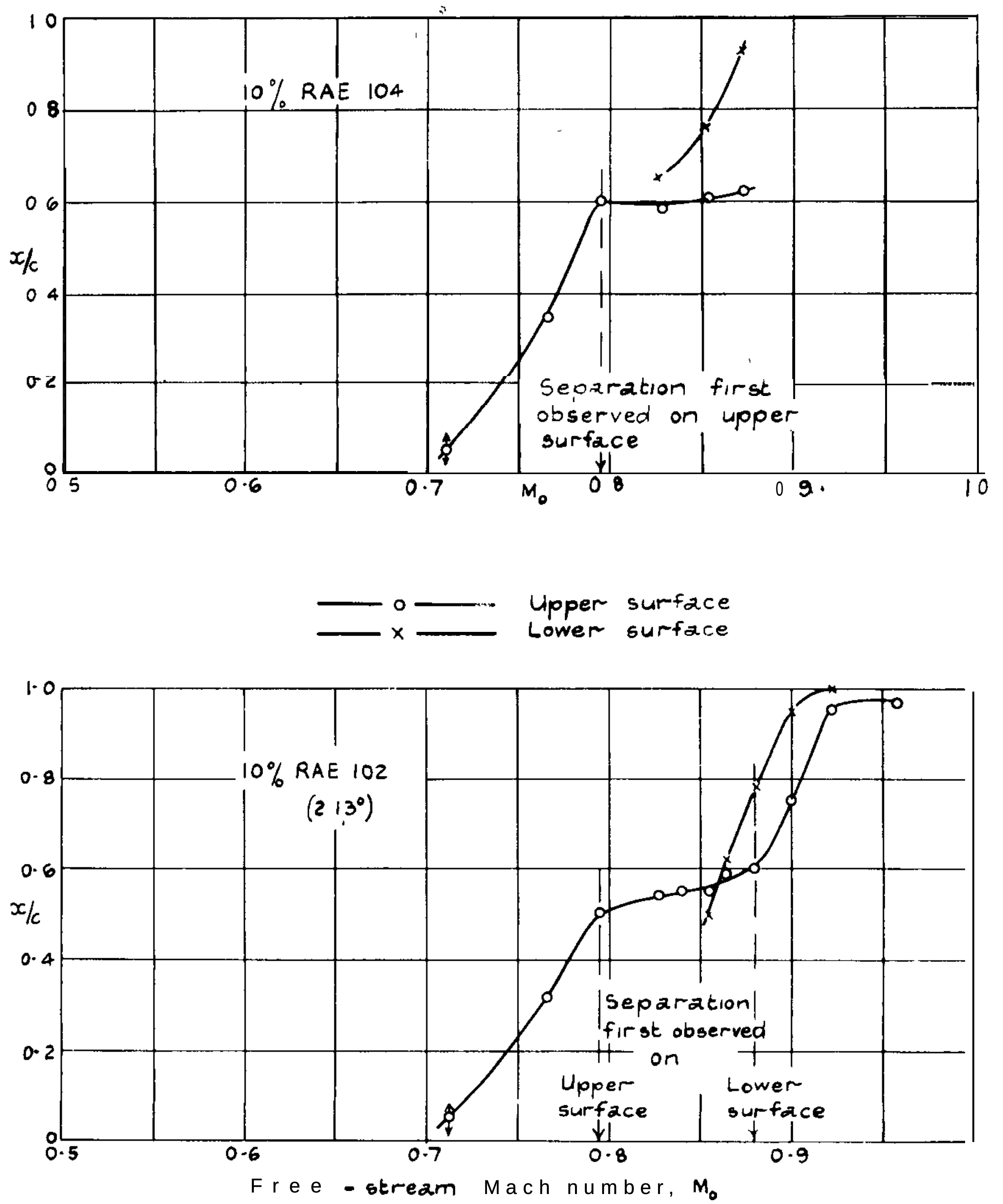
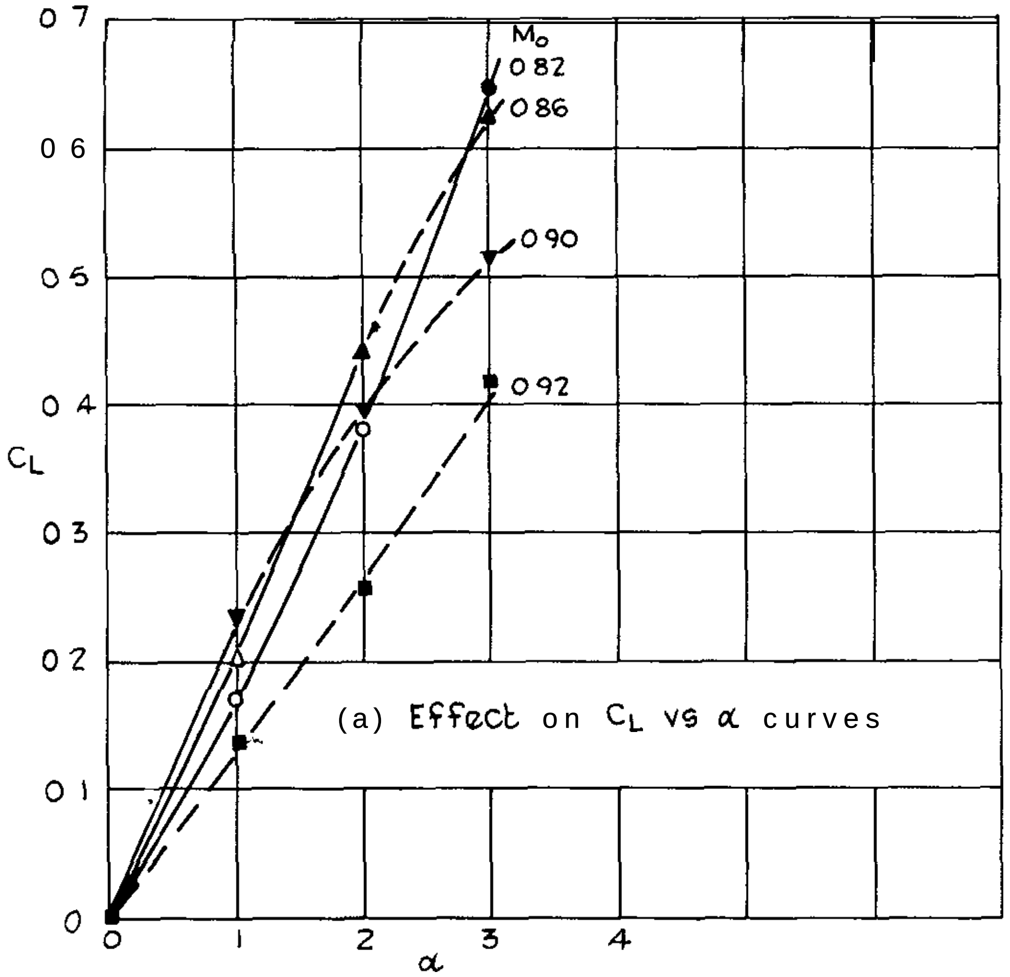


FIG. **32.** Chordwise positions of the main shock - waves for 10% RAE 102 and 104 aerofoils at 2° incidence (NPL tests, transition fixed ; $R \approx 1{\cdot}8 \times 10^6$).



Filled in symbols and broken lines denote the range of incidence or C_L for which separation was observed

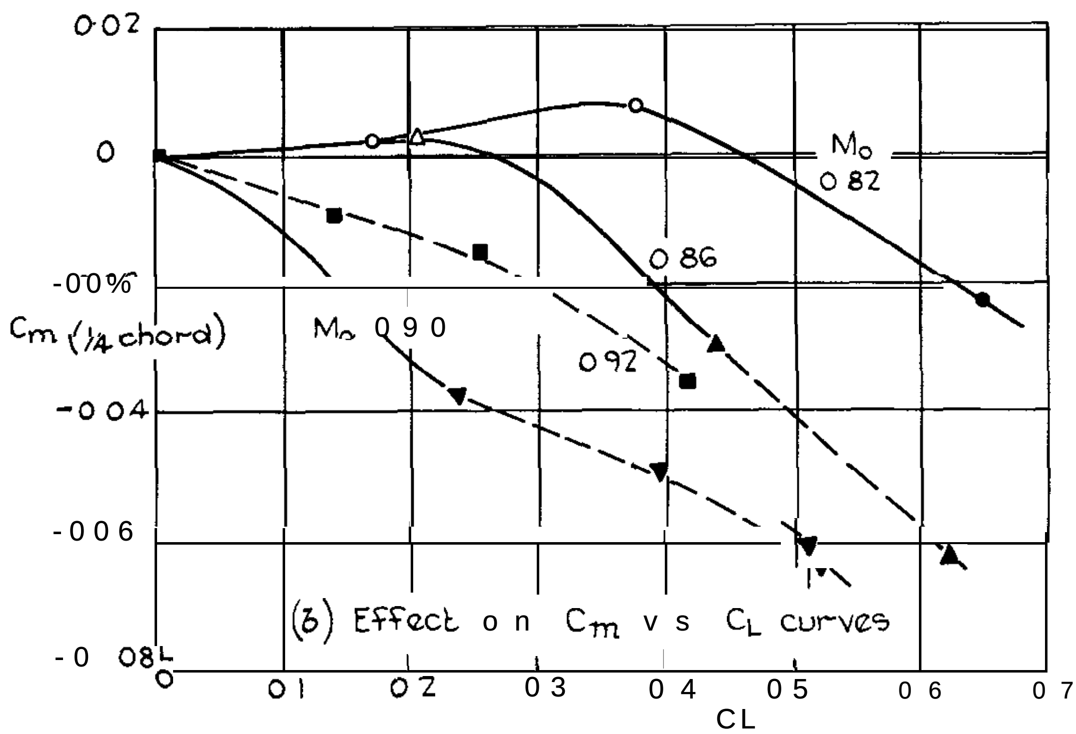
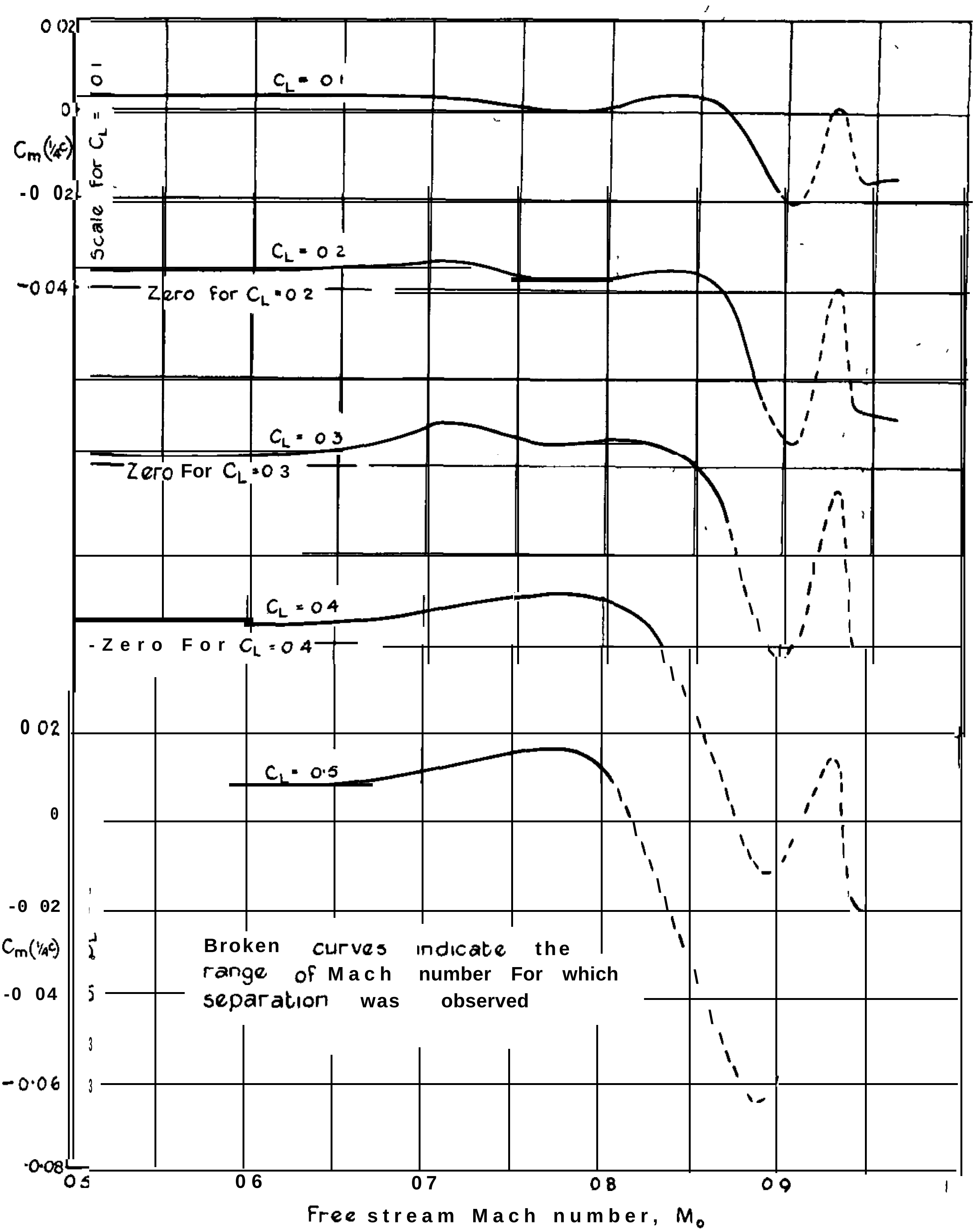


FIG. 33 (a & b).

Effects of separation for the 6 % RAE 104 aerofoil (N P L tests, Transition fixed, R $\approx 8 \times 10^6$)



(c) Effect on C_m vs M curves (const C_L)

FIG 33c. Effects of separation for the 6% RAE 104 aerofoil (N P L tests, Transition fixed, $R \approx 1 \cdot 8 \times 10^6$)

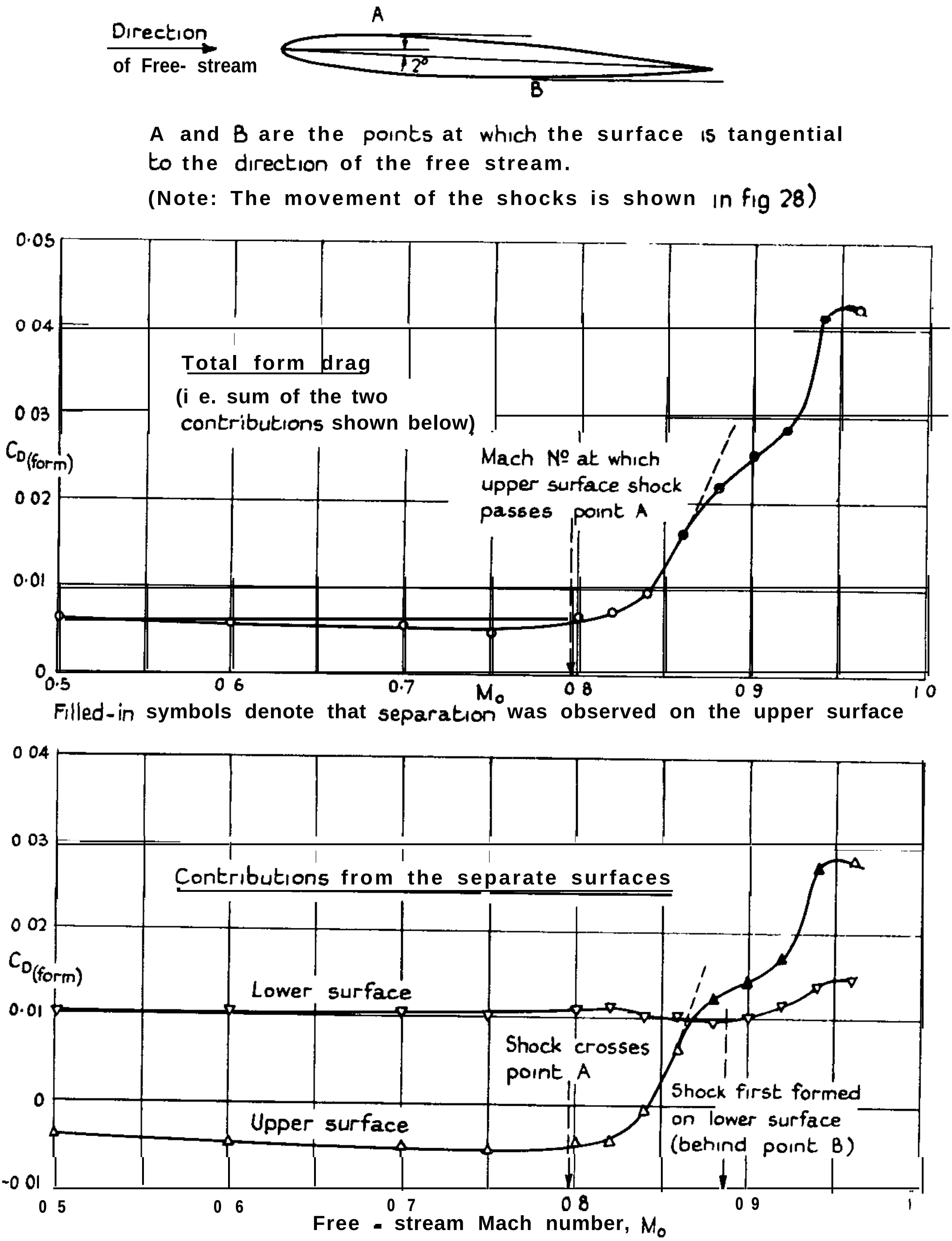


FIG. 34.

Variation of form -drag coefficient for 6% RAE 104 aerofoil at 2" incidence.

(N PL tests; transition fixed ; R ≈ 1 8 x 10^6)

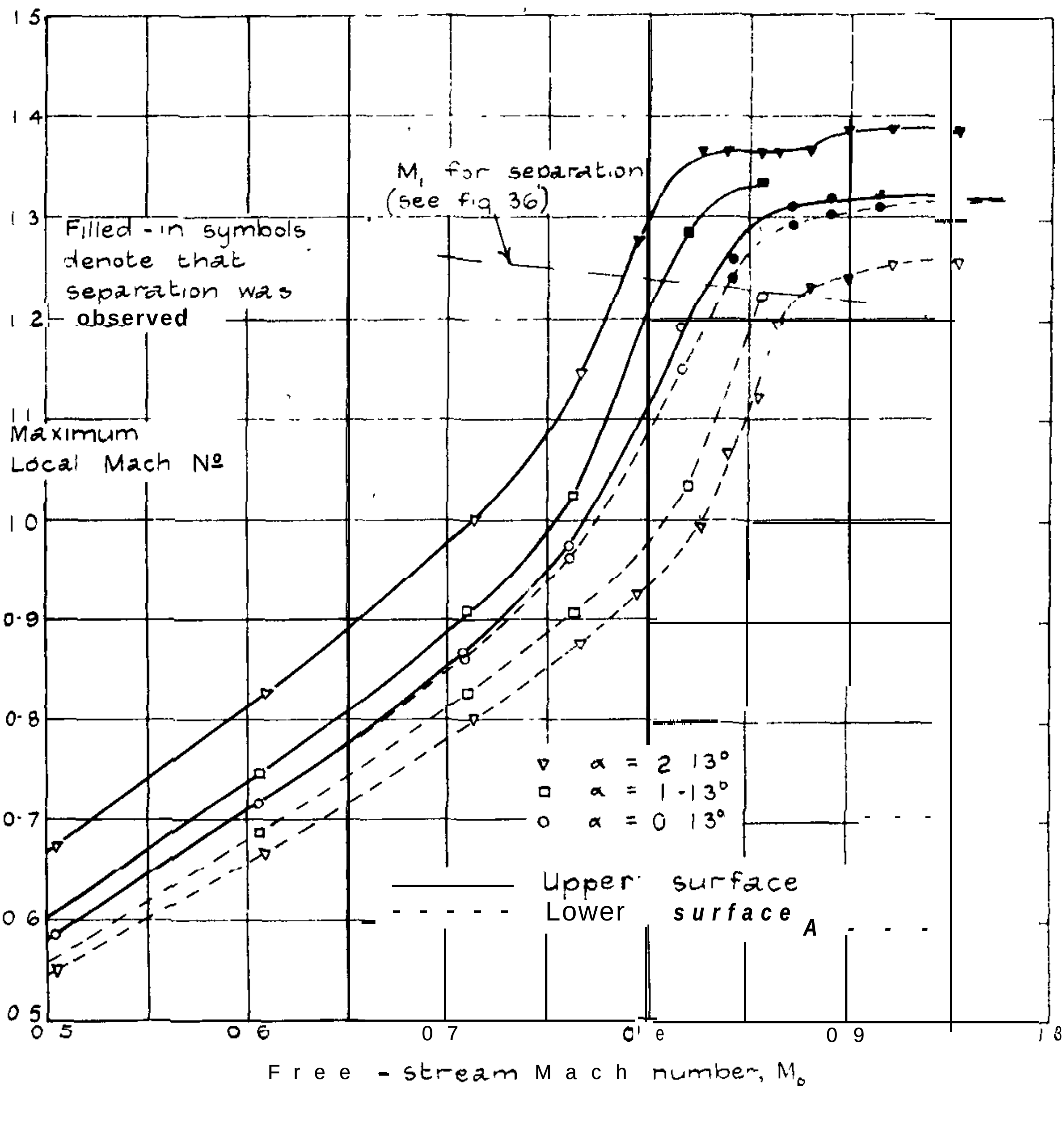


FIG 35
Maximum local Mach number reached on the surface of 10% RAE 102 aerofoil (NPL test5, transition fixed $R \approx 1{\cdot}8\times10^6$

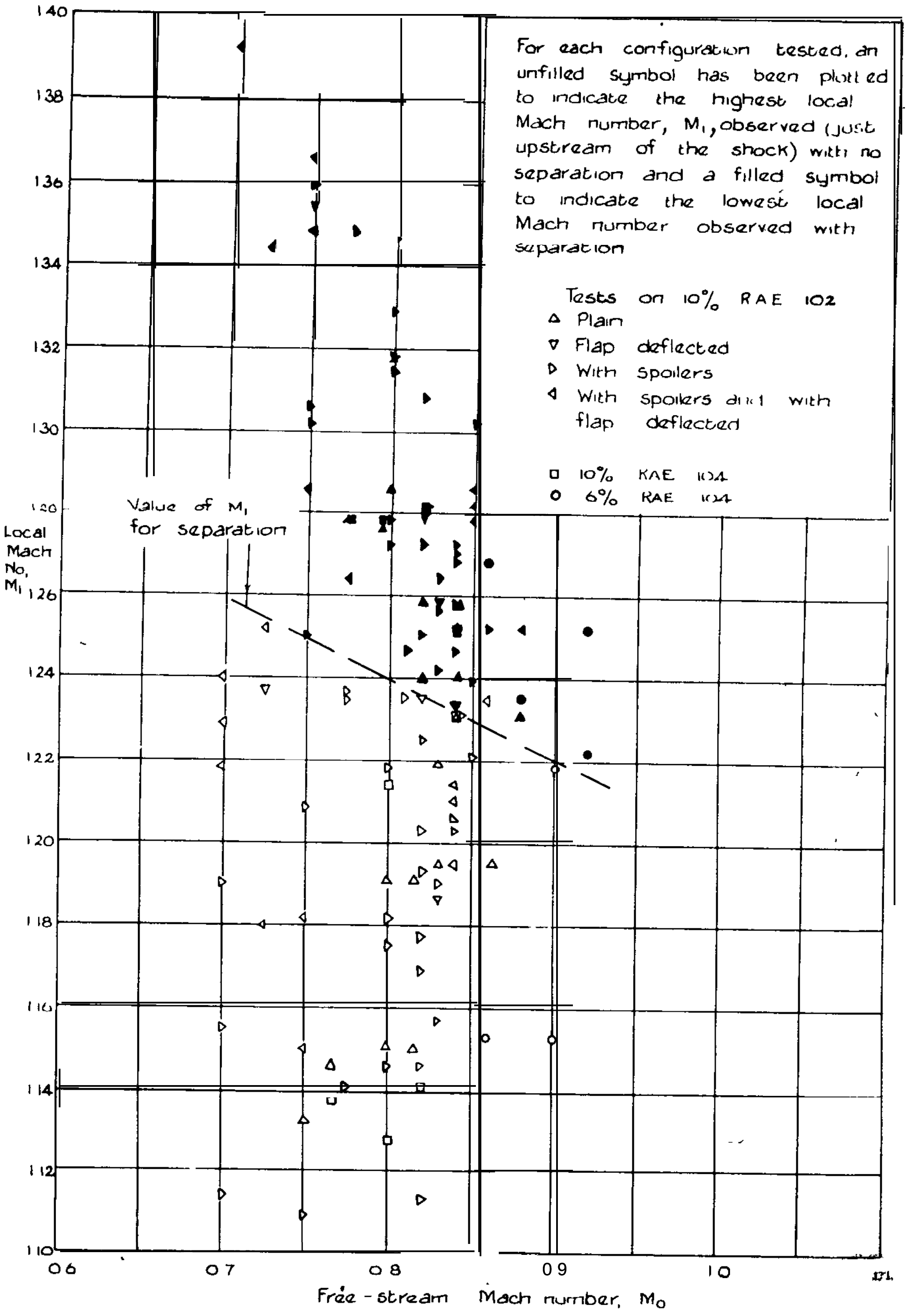


Fig 36

Observations of the local Mach number just upstream of the shock for which separation occurs on aerofoils

(NPL tests, Turbulent Boundary Layers, $R \approx 1{\cdot}8 \times 10^6$)

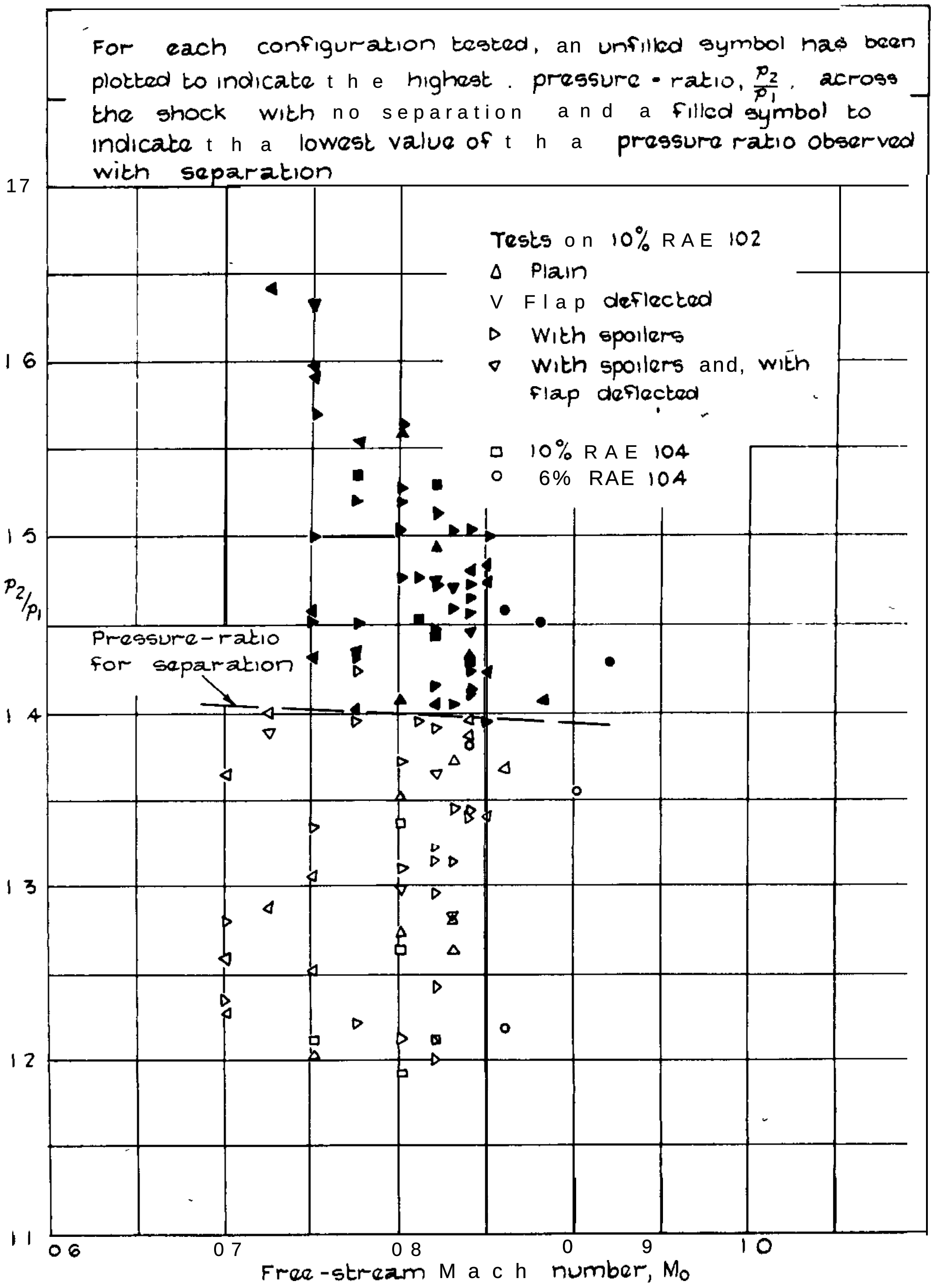


FIG. 37.

Observations of the pressure ratio across the shock for separation occurs on aerofoils

(NPL tests, Turbulent boundary layer, $R \approx 1{\cdot}8 \times 10^6$)

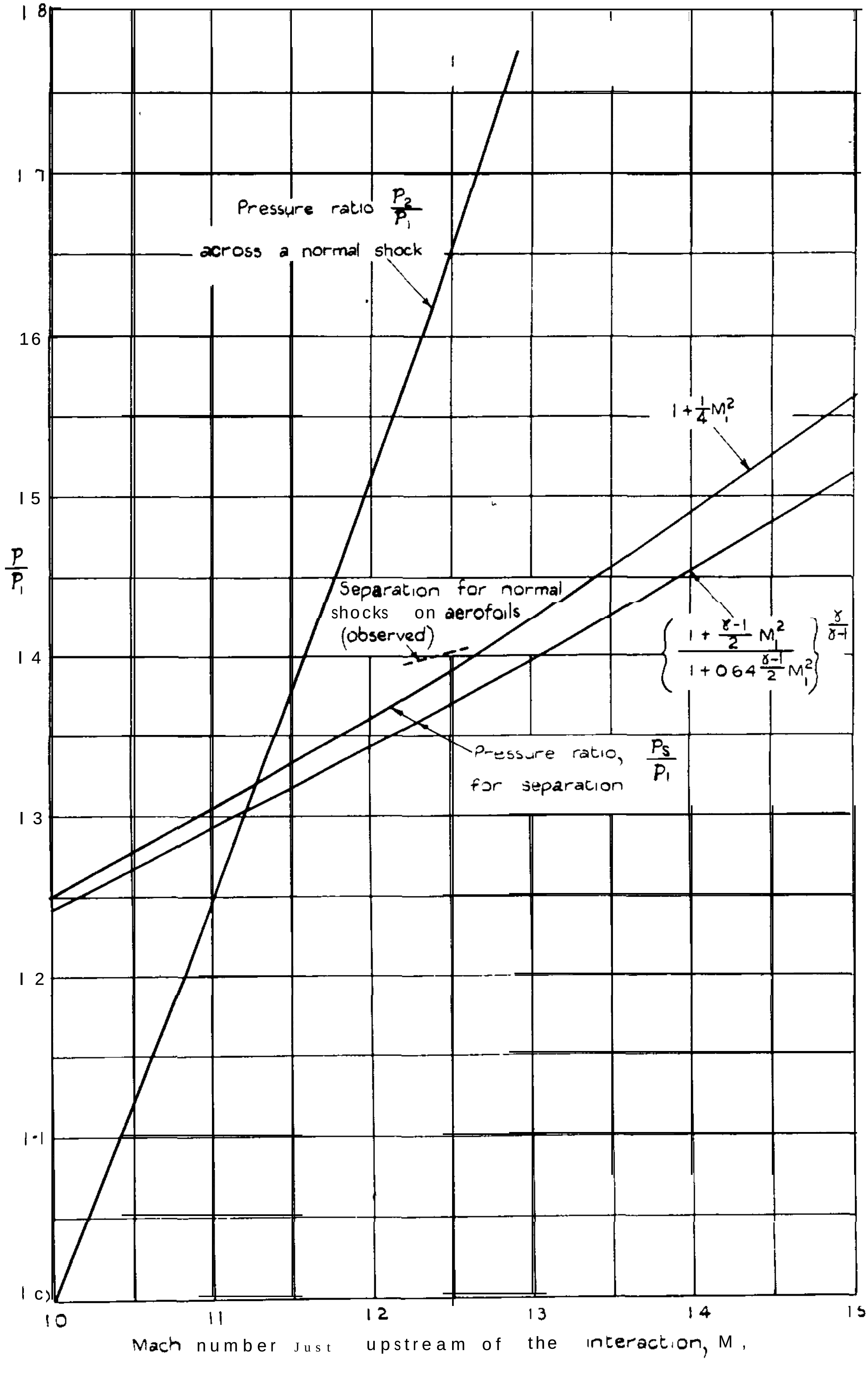


FIG 38 Comparison of pressure ratios for separation as observed and predicted

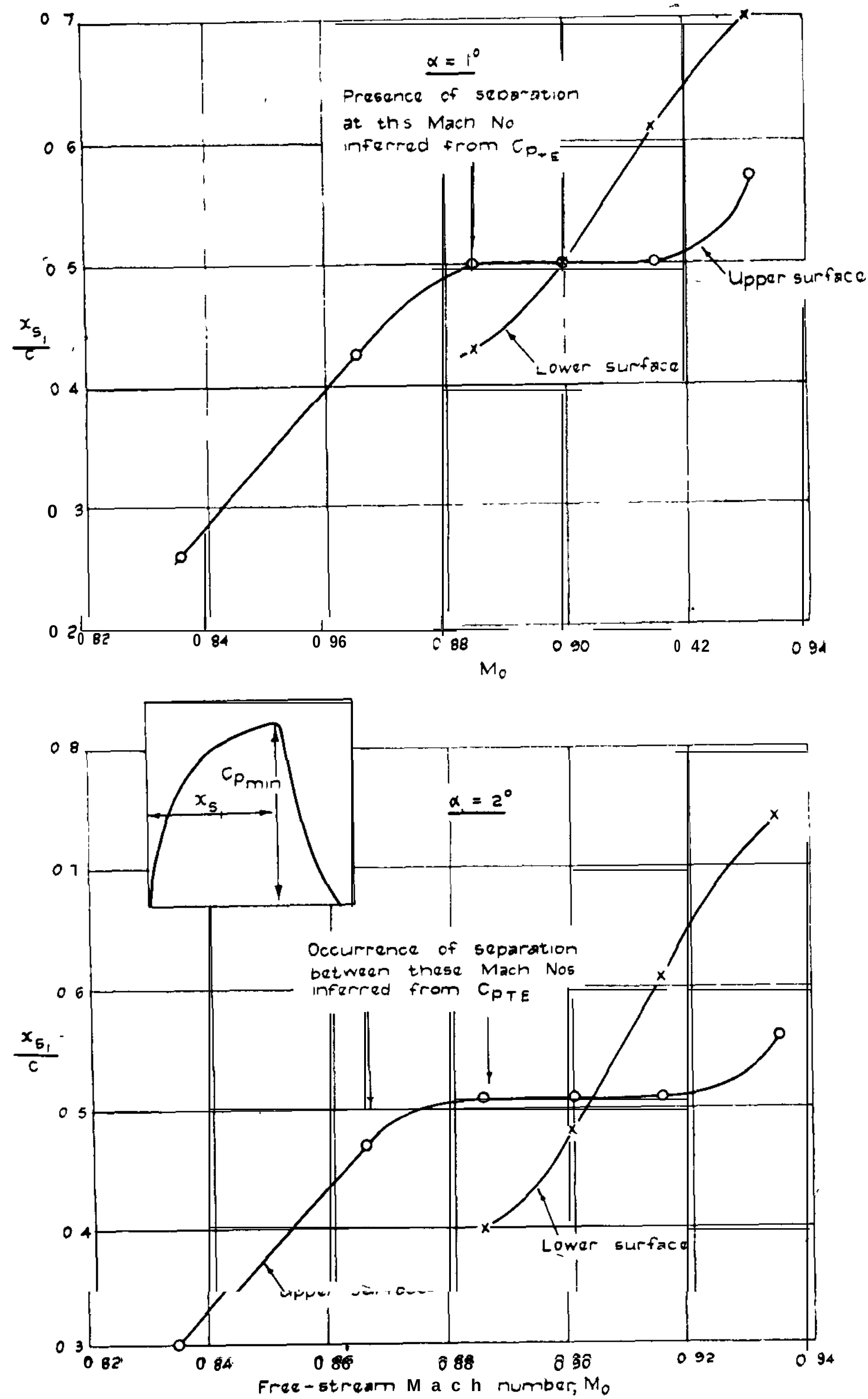


FIG. 39a

Shock-wave positions for 0·79 semispan of a model with sweptback wing (R.A.E. model tests h.f. 50), Transition fixed, $R = 1{\cdot}75 \times 10^6$.

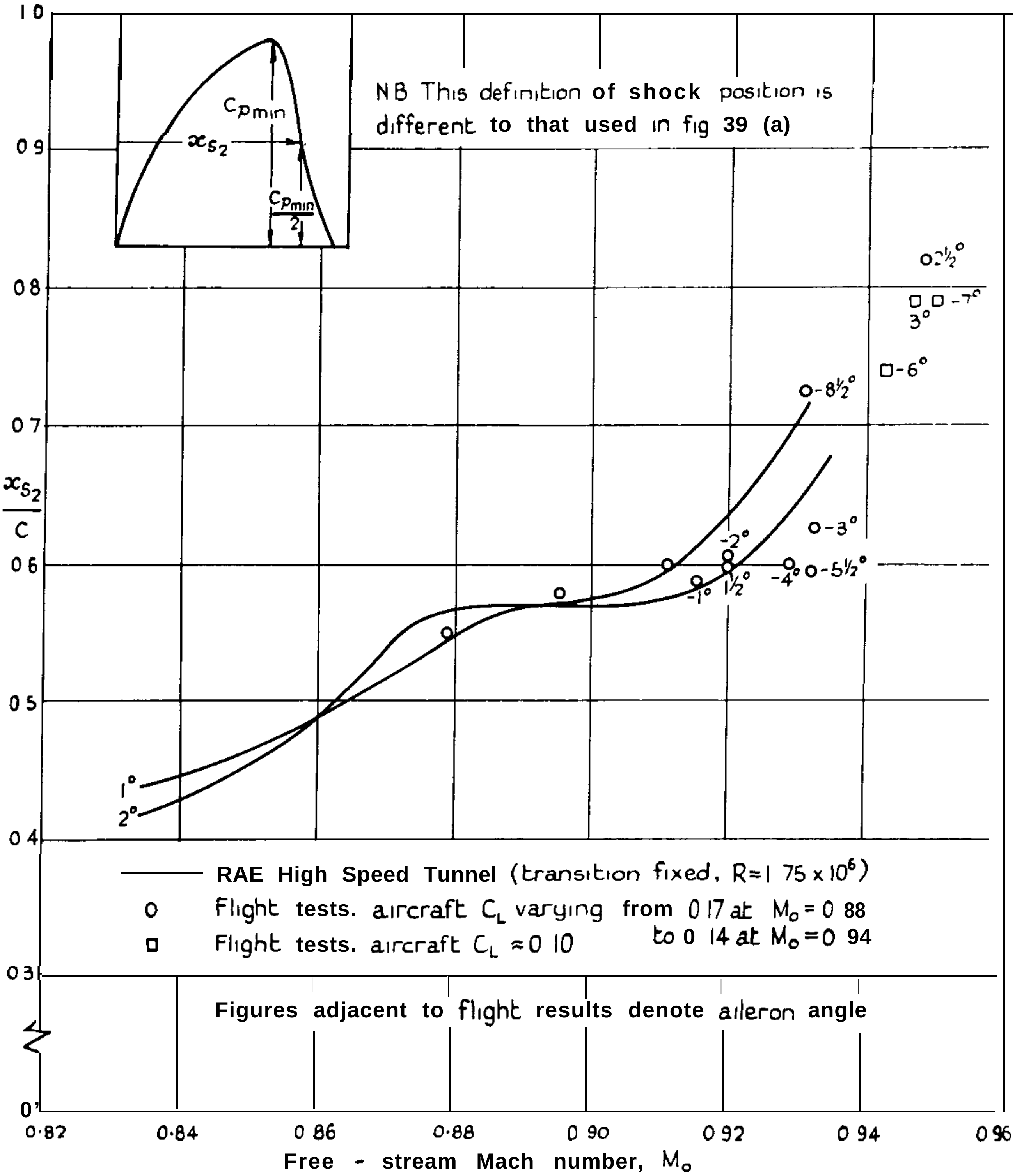


FIG. 39b
Position of upper-surface shock-wave for 0 79 semispan of a model with sweptback winq. Comparison of flight and model tests (ref 50)

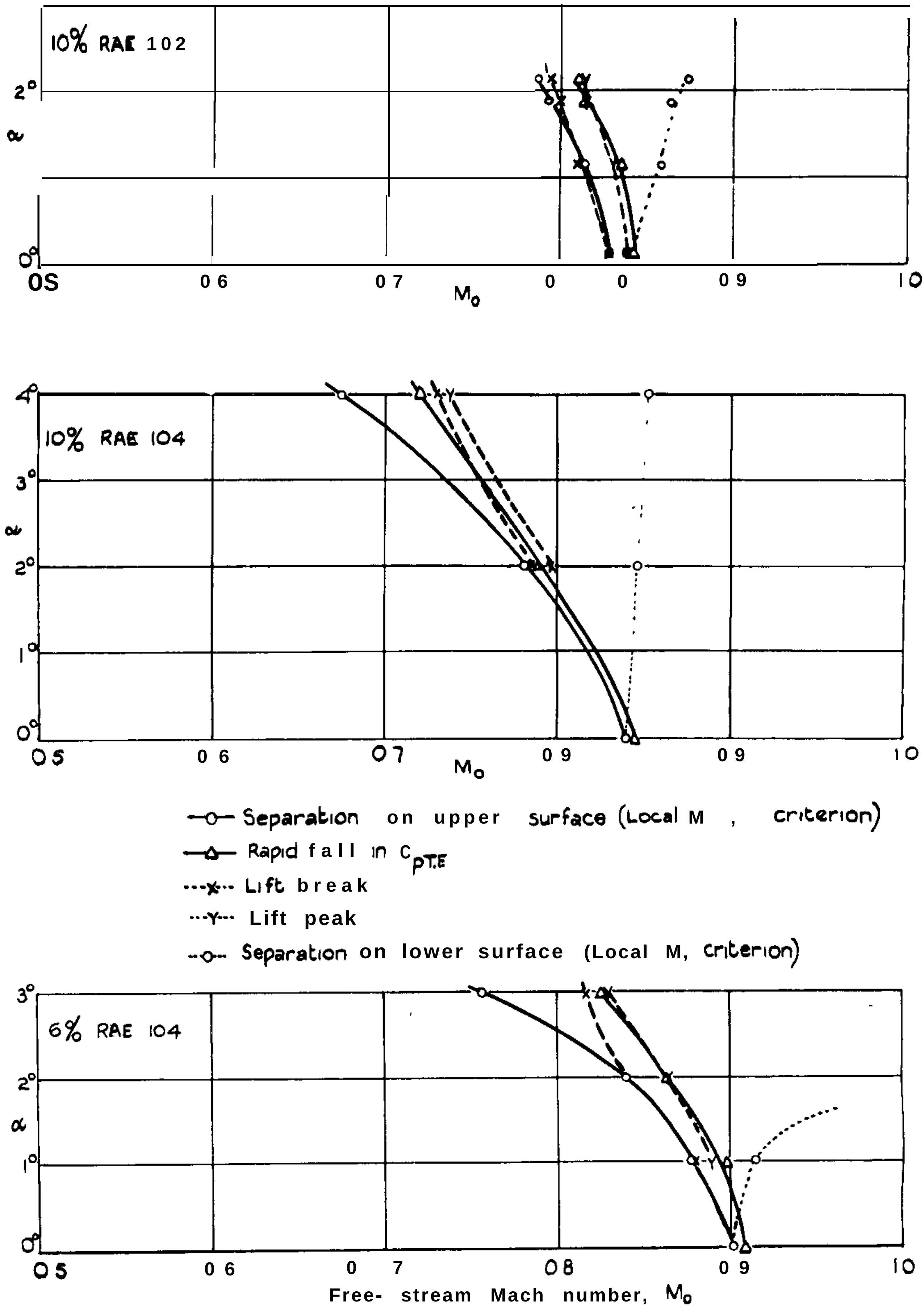


FIG 4-O. Values of free-stream Mach numbers for the occurence of shock-induced turbulent separation and the onset of certain of its effects, NPL tests on three sections

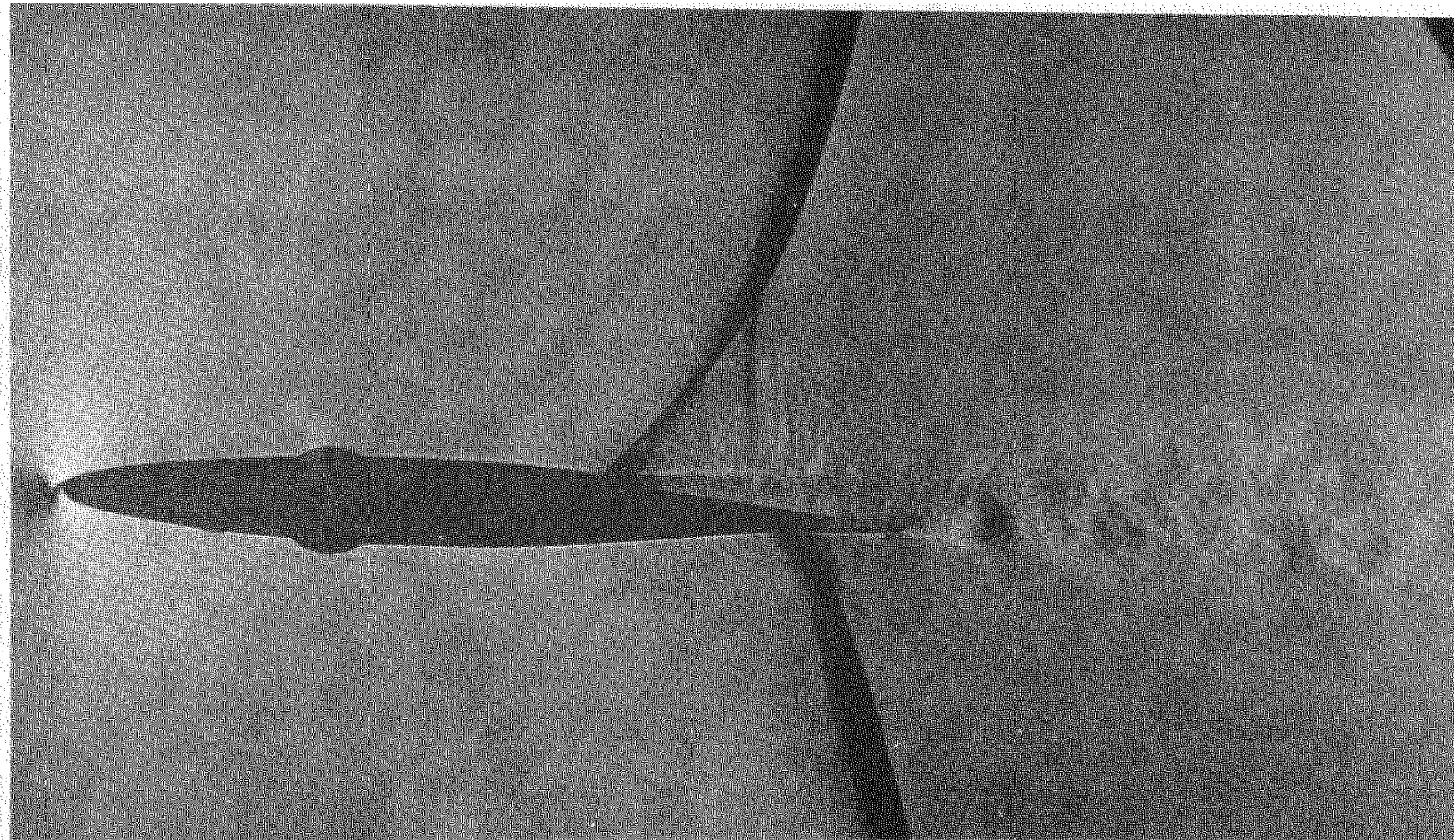


FIG. 41. Flow round a 10% thick RAE 104 Aerofoil at $M_o = 0 \cdot 82$, $\alpha = 2°$, $R = 1 \cdot 8 \times 10^6$.

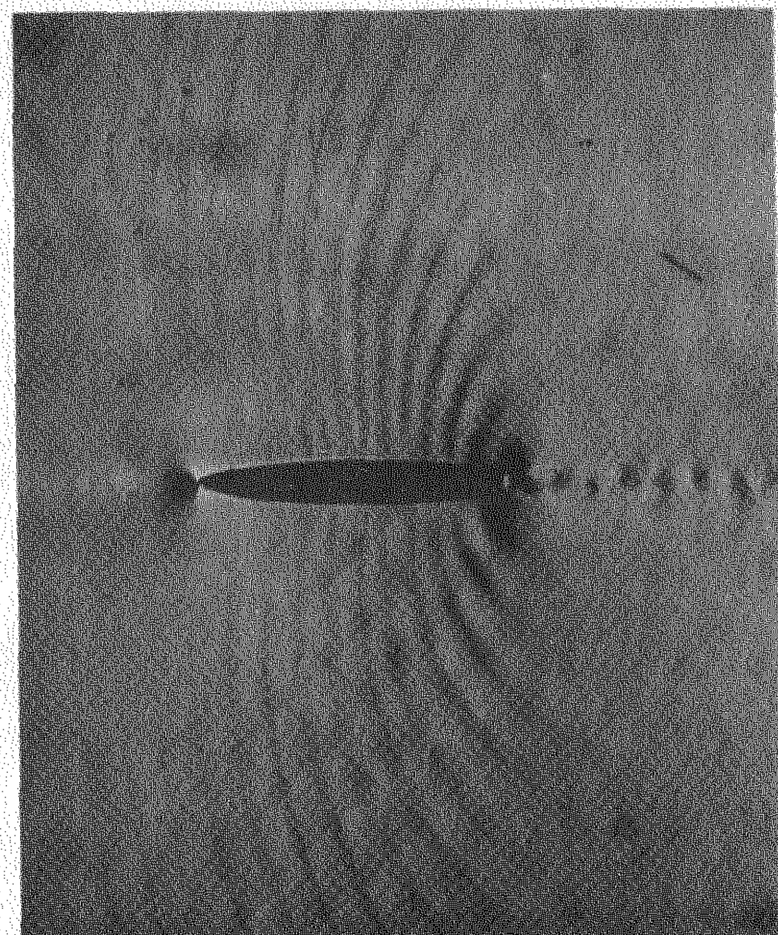


FIG. 42. Flow round a 12% thick Aerofoil with a blunt trailing edge at $M_0=0{\cdot}7$, $\alpha=0^\circ$, $R=0{\cdot}6\times10^6$ Turbulent boundary layer.

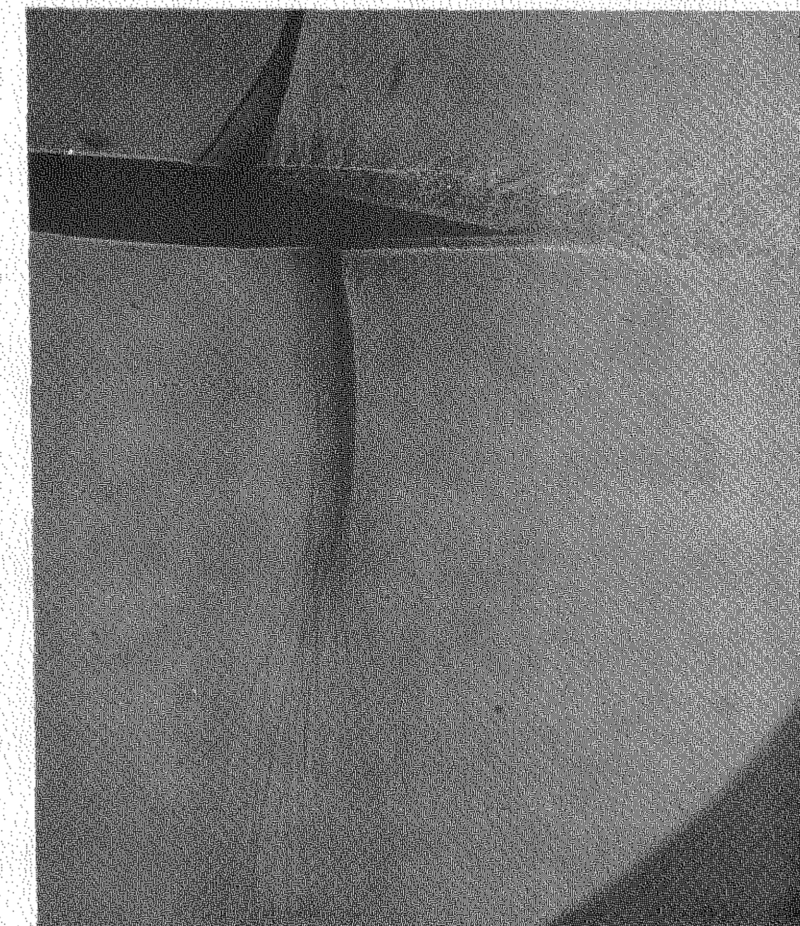
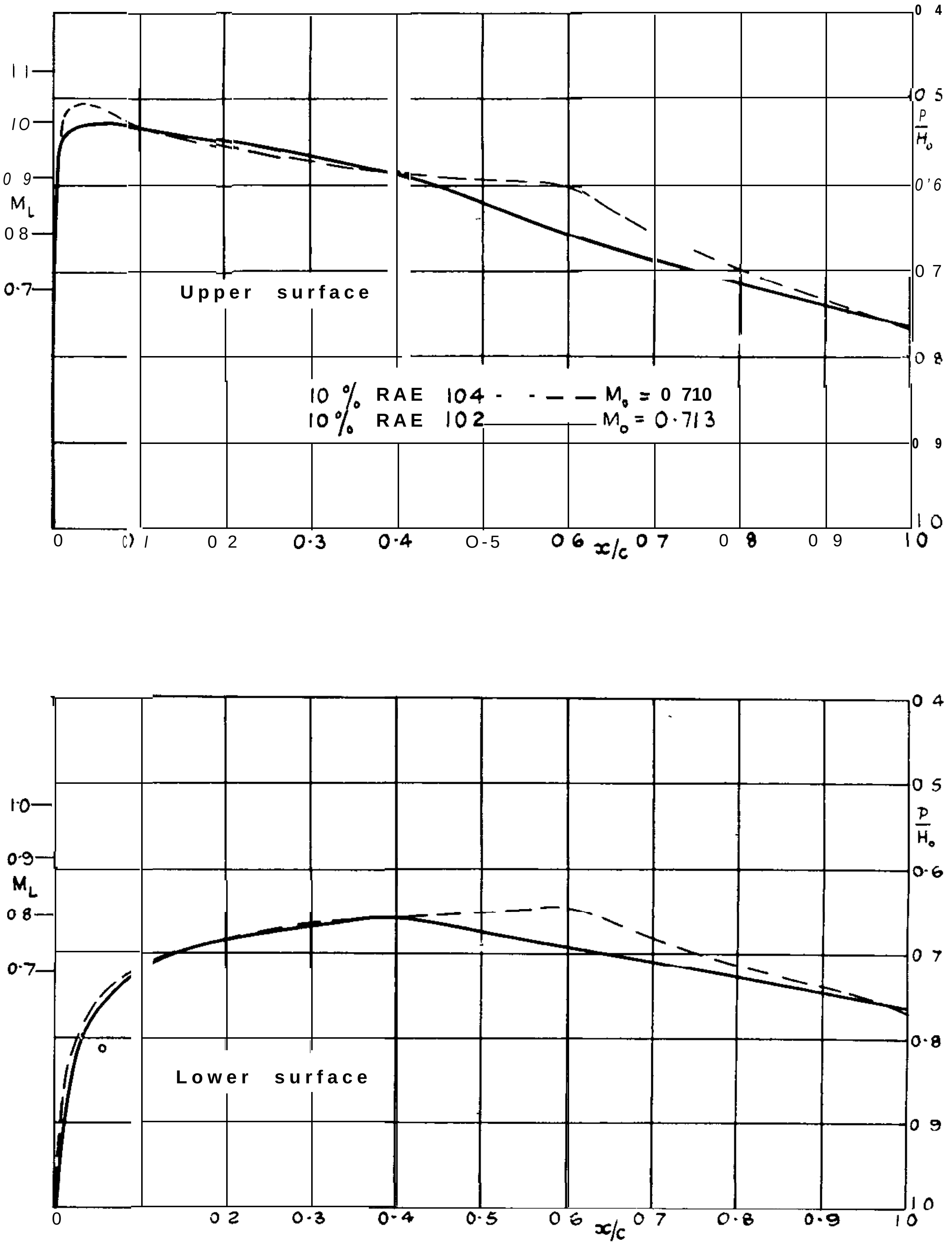
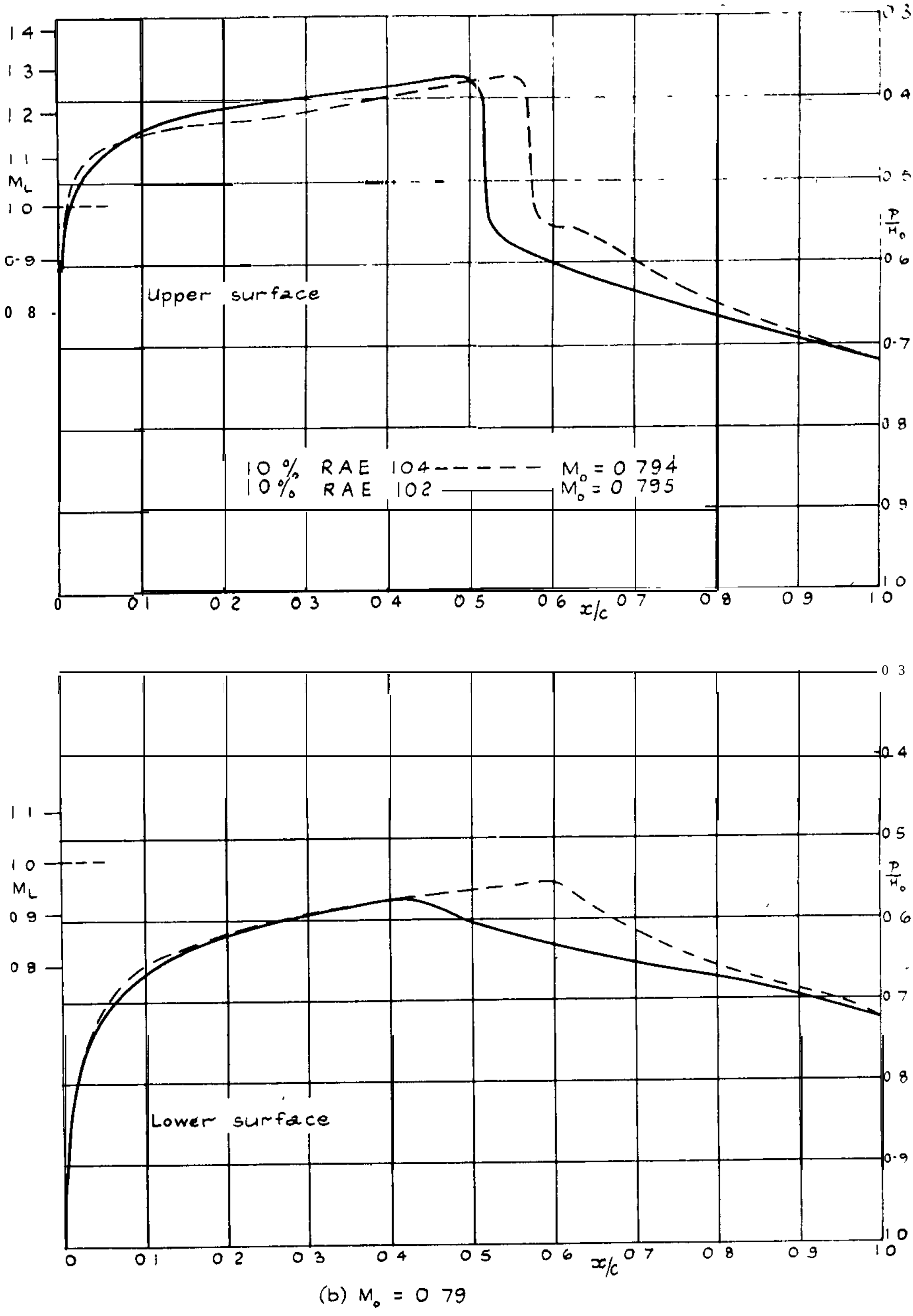


FIG. 43. Flow near the tail of a 10% thick RAE 104 Aerofoil $M_0=0{\cdot}82$, $\alpha=2^\circ$, $R=1{\cdot}8\times10^6$ Turbulent boundary layer.



(a) $M_0 = 0{\cdot}71$

FIG. 44a.
Comparison of surface-pressure distributions for the 10 % RAE 104 and RAE 102 aerofoils at 2° incidence (N.P.L. tests, transition fixed; $R \approx 1{\cdot}6 \times 10^6$)



(b) M_0 = 0 79

FIG 44b. Comparison of surface - pressure distribution for the 10% RAE 104 and 10% RAE 102 aerofoils at 2° incidence (N PL tests, transition fixed, $R \approx 1 \cdot 8 \times 10^6$)

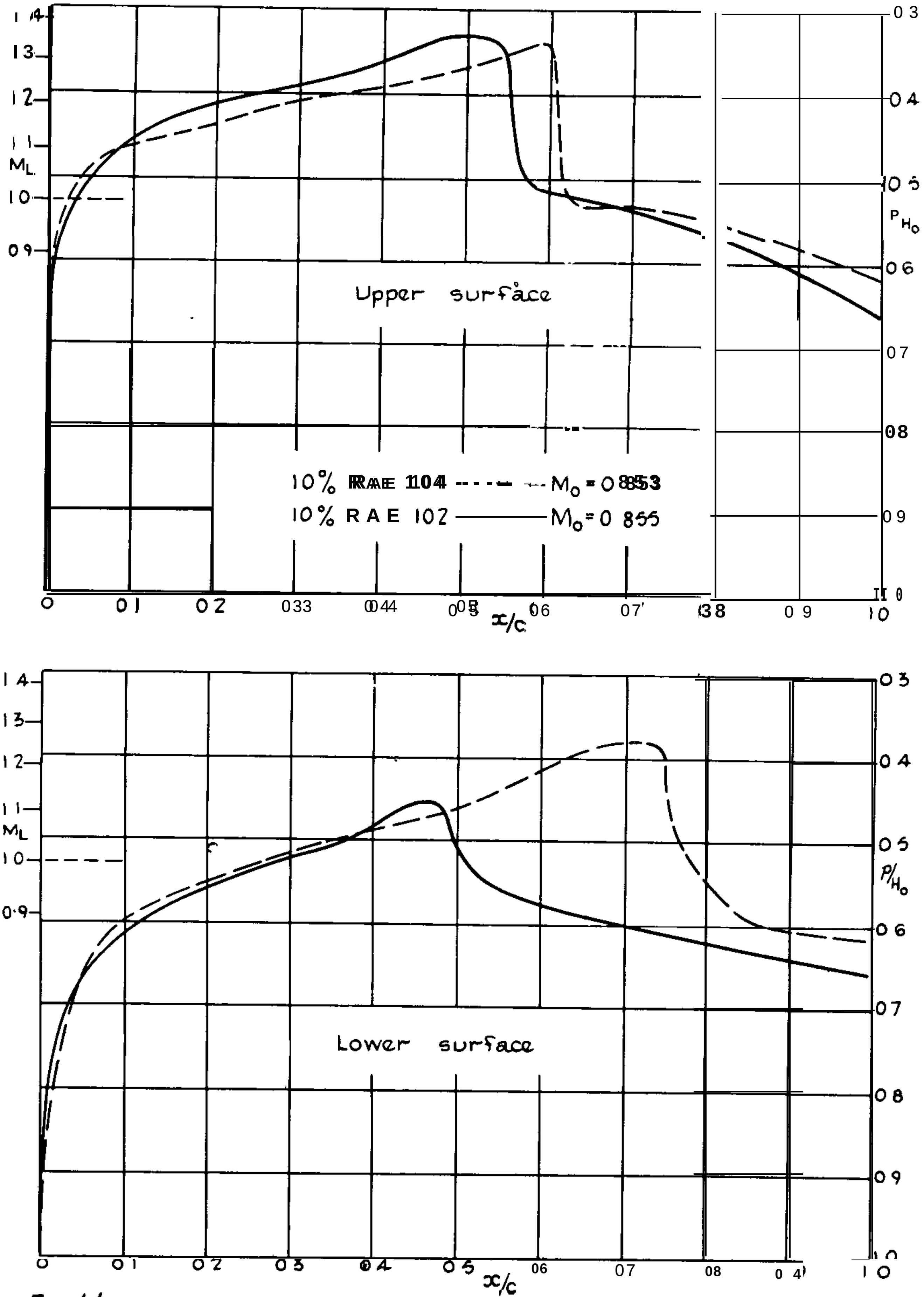
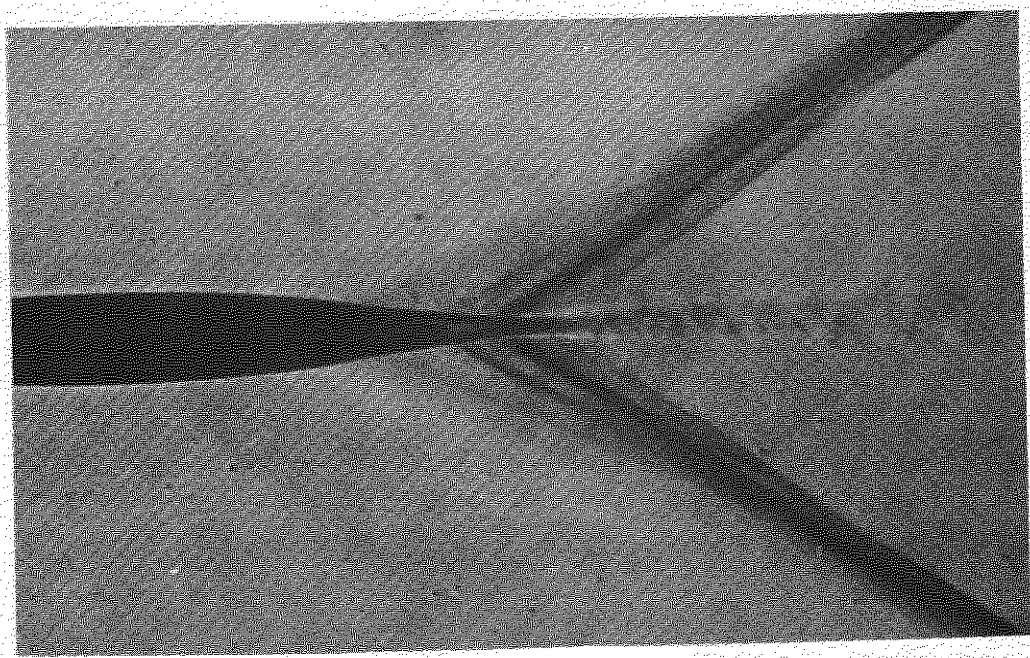


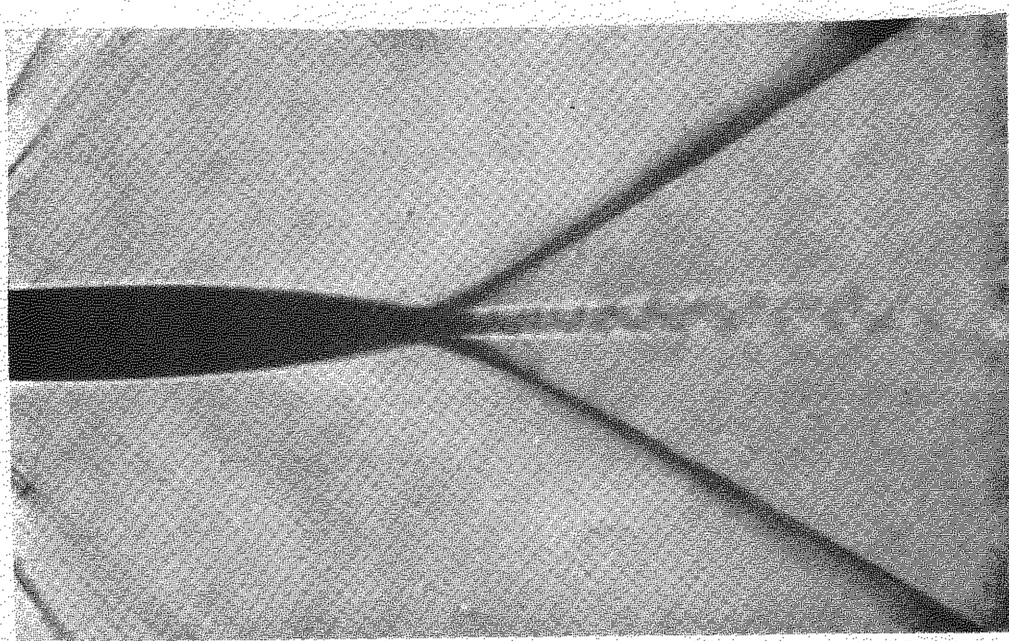
FIG 44c.

(c) M_o = 0 85

Comparison of surface-pressure distributions for the 10% RAE 104 and 10% RAE 102 aerofoils at 2° incidence (NPL tests, Transition fixed, $R \approx 1.8 \times 10^6$).



(a) Laminar boundary layer



(b) Turbulent boundary layer

FIG. 45. The flow near the trailing edge of a 12% thick aerofoil at $M_o = 1 \cdot 6$, $\alpha = 0°$, $R = 0 \cdot 8 \times 10^6$.

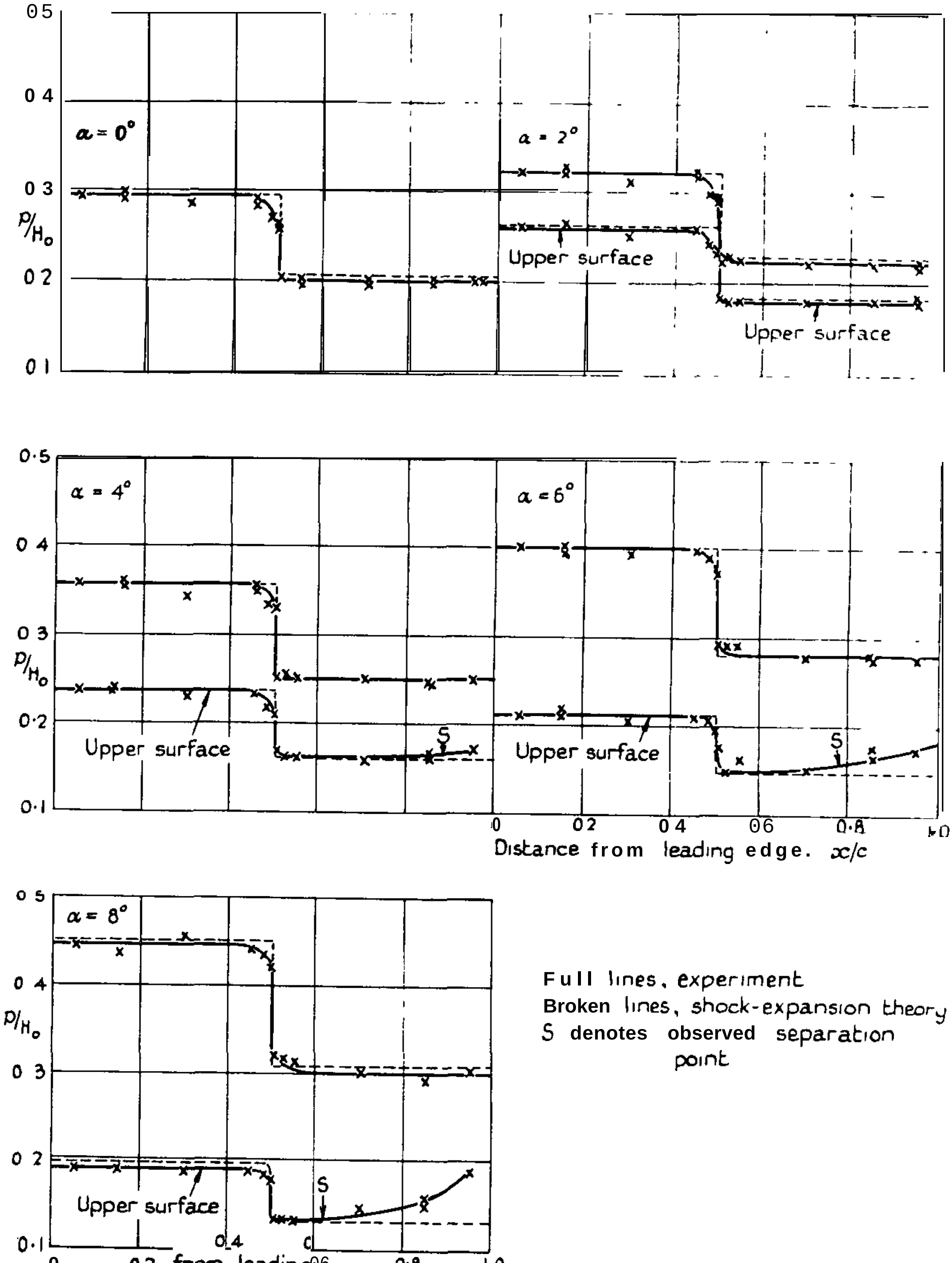


FIG. 46.
Pressure distributions on a 6% double - wedge at $M_0 = 1.57$.

R - D. 8×10^6, Laminar boundary layer.

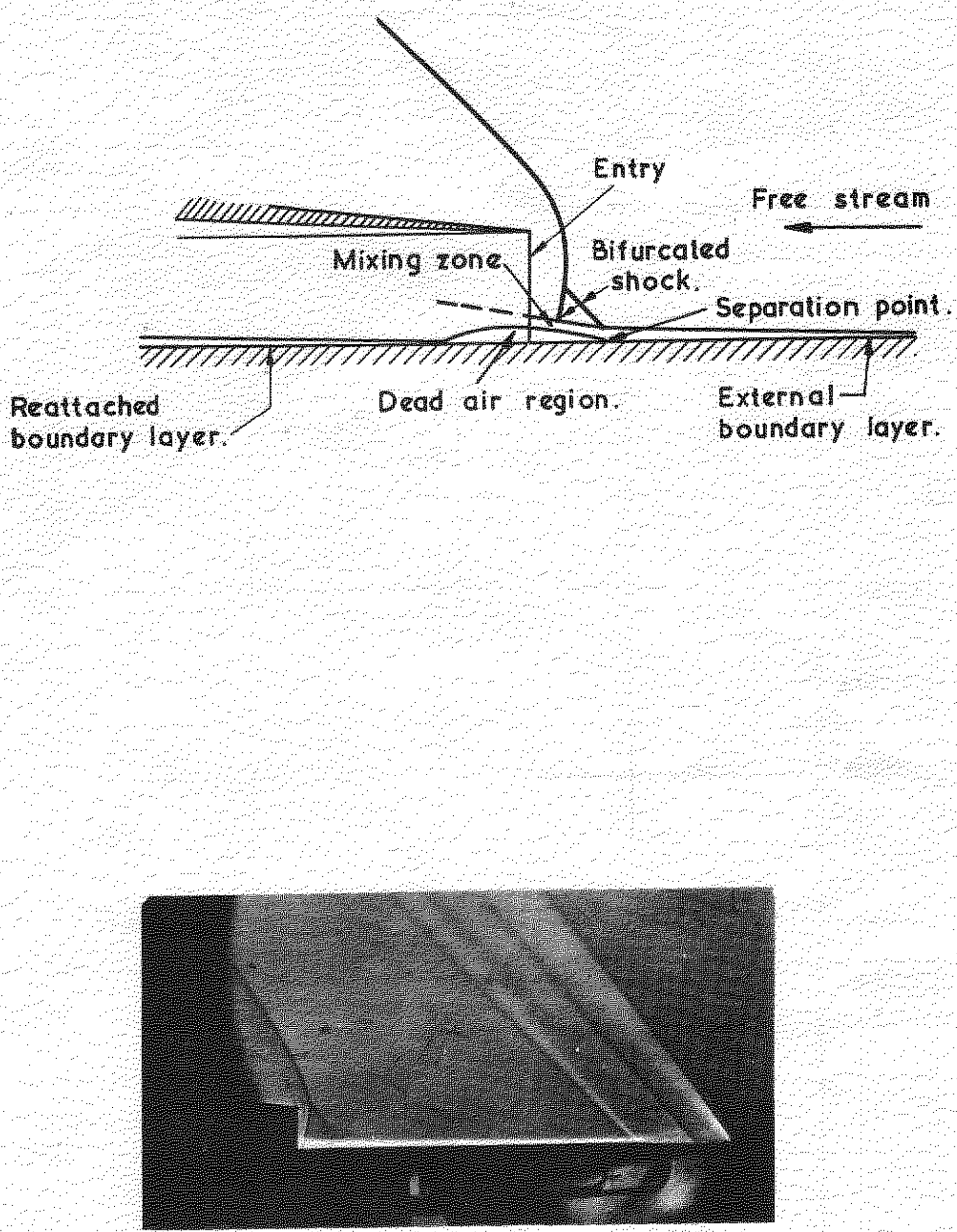


FIG. 47. Flow into a typical side intake with supersonic free-stream.

(Turbulent boundary layer)

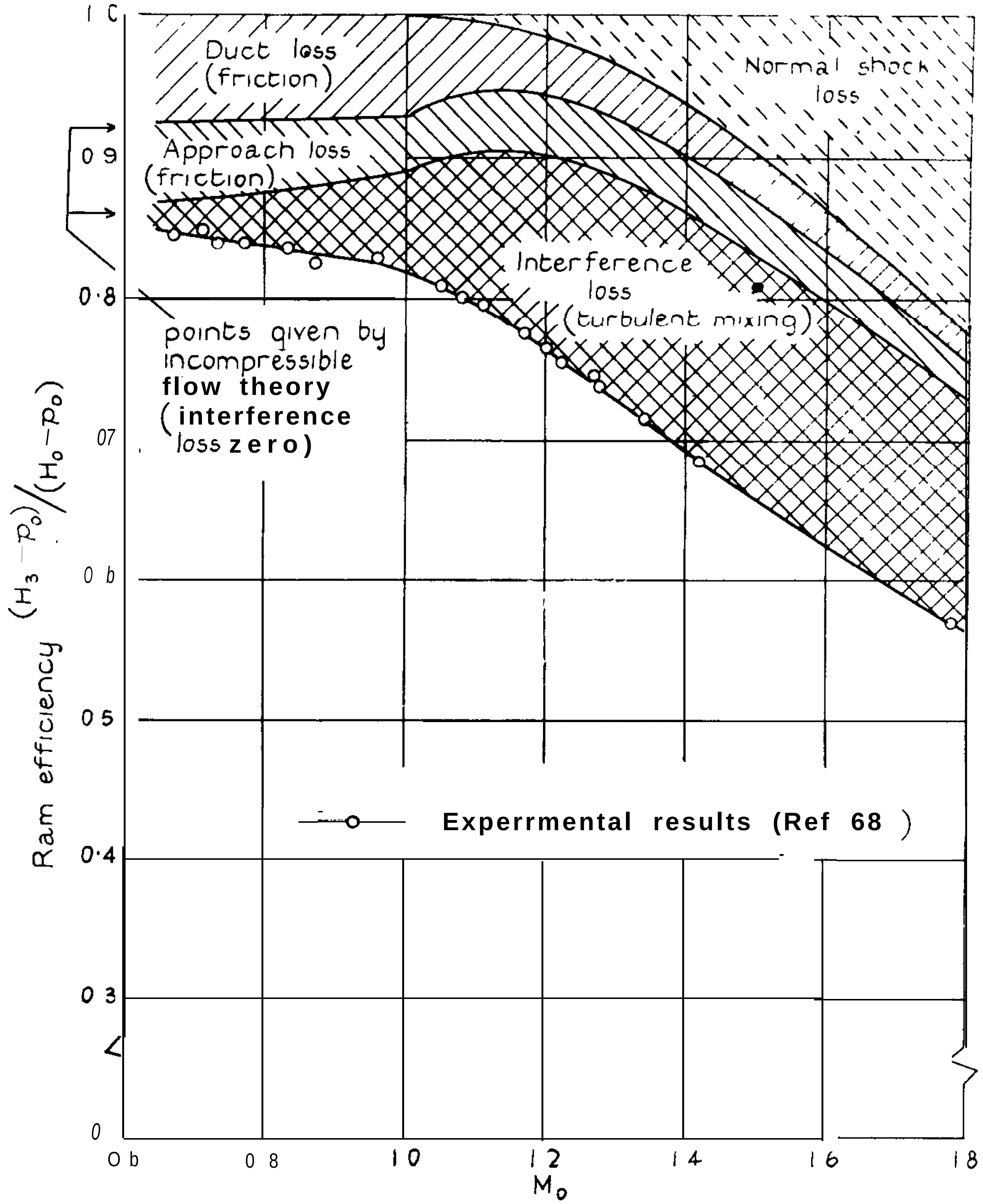


FIG. 48. Analysis of total pressure loss of a typical side intake at design mass flow ratio and transonic speeds
(Turbulent boundary layer)

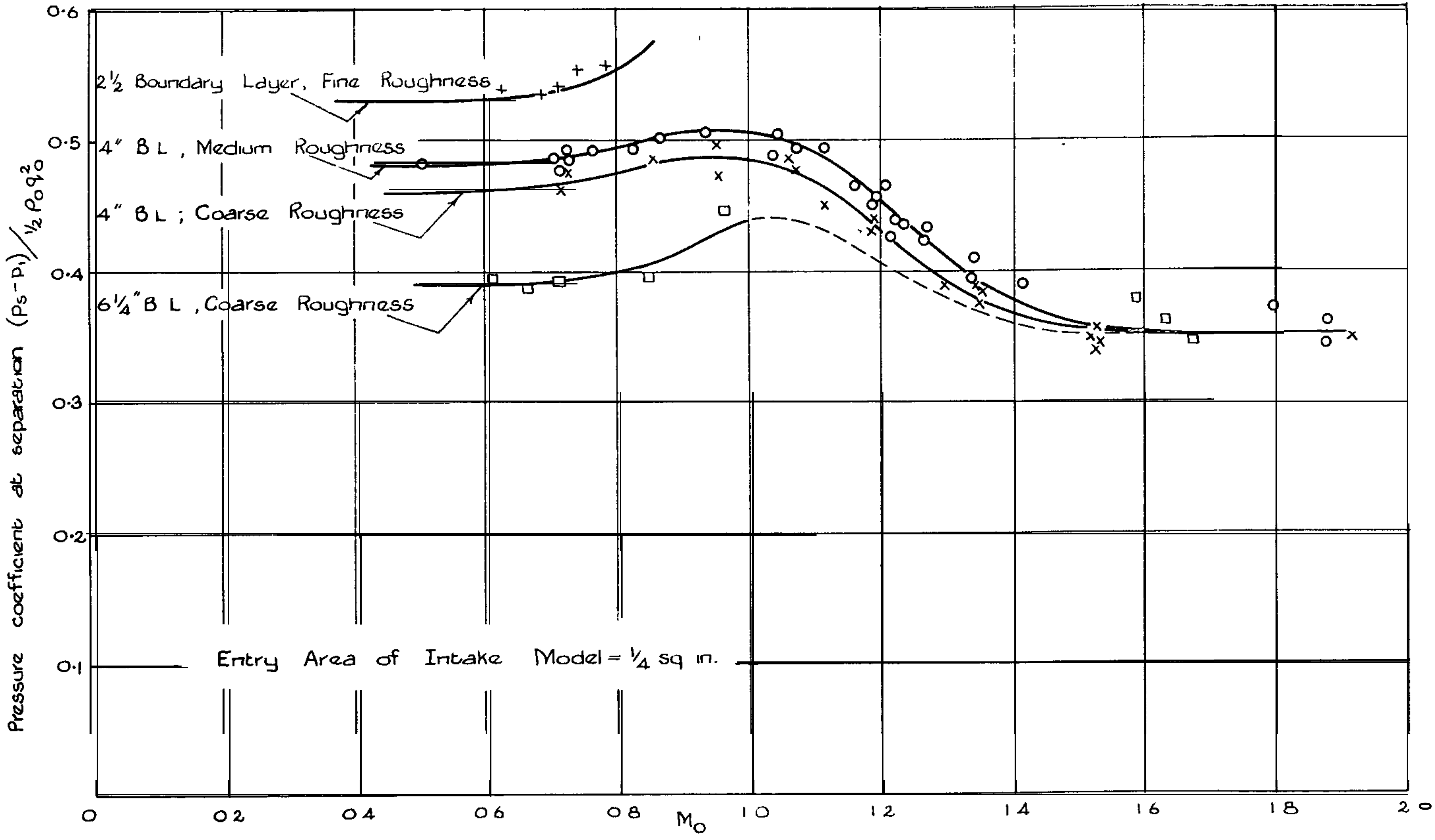


FIG 49. Pressure at separation of turbulent boundary layer ahead of side intake

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