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by

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Some Instabilities Arising from the Interactions between Shock Waves and Boundary Layers

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February, 1958

Summary

A brief review is made of the available information concerning the flow fluctuations and instabilities arising from shock-induced separation in the flow over aerofoils and wings. The influence this phenomenon has on the oscillatory behaviour of aerofoils and control surfaces is also briefly discussed.

A more detailed consideration is devoted to a recent investigation at the N.P.L. into the part played by shock-induced separation in the instability of a control surface.

1. Introduction

Shock waves which, under some conditions of high-speed flight form close to wings and bodies can lead to the flow separating from the surface. Such shock-induced separation can have a number of serious consequences concerning aircraft performance and control, including the possibility of oscillatory instabilities of control surfaces and flow fluctuations giving rise to buffeting.

The interaction between a shock wave and a boundary layer in steady flow has been the subject of numerous investigations, for instance Refs.1 and 2, whilst various authors[3,4] have discussed its effects on aerofoil characteristics and on aircraft performance and the means whereby they can be prevented. Flow fluctuations arising from shock-induced separation have been measured for a variety of aerofoils and wings, mainly at the NACA. Measurements of the aerodynamic forces on oscillating aerofoils or their controls, a knowledge of which is required for flutter prediction, have shown that at high subsonic speeds large changes in the values often occur, and that sometimes unstable regions exist. It has been suggested that some of these changes can be attributed to separation.

Although much information is available on the unsteady effects of shock-induced separation, it appears that little attention has so far been paid to the reciprocal problem of the effects of unsteady flow on shock-wave boundary-layer interaction.

The purpose of the present paper is firstly to provide on the basis of available information a very brief review of the part played by shock-induced separation in instabilities and other conditions of unsteady motion; secondly to describe some results from a recent investigation at the N.P.L. in connection with an oscillatory instability of a control surface arising from shock-induced separation.

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The paper has been written with structural effects in mind, but no attempt has been made to survey the occurrences of actual flight. Indeed much of the available information relates to wind tunnel tests of aerofoils and for simplicity the discussion will refer mainly to two-dimensional conditions of flow.

2. The Development of Shock Waves Close to an Aerofoil and their Effects on the Flow

The formation and subsequent development of shock waves close to a wing is conveniently outlined by considering a symmetrical aerofoil at a small positive incidence in a flow of gradually increasing Mach number. At a certain critical value of the free-stream Mach number a small region of flow in which the local velocities are supersonic is formed, firstly in the neighbourhood of the position of the maximum thickness on the upper surface. This region is terminated by a shock wave (see Fig.1).

With increase of Mach number, the rearward limit of the supersonic region tends to move towards the trailing edge accompanied, for the usual type of aerofoil, by an increase in the strength of the shock wave as measured by the pressure ratio across it.

The formation and growth of a similar region at the lower surface of the aerofoil is delayed to a higher Mach number depending on the incidence.

At some stage in the flow development the shock wave at the upper surface will be strong enough to separate the boundary layer either locally near the foot of the shock or more extensively over the rear of the aerofoil (see Fig.2). The separation induced by the shock has considerable effect on the further development of the flow and on the pressure distribution over the aerofoil. A marked falling-off (or divergence) of the pressure occurs at the trailing-edge when the separation is extensive enough to affect the flow there[3,4]. As will be mentioned again later, this is found to coincide with the onset of many of the consequences of separation.

In the absence of separation the positions of the shock waves at the upper and lower surfaces would be expected to vary smoothly with increasing Mach number, the shock wave on the upper surface reaching the trailing-edge first (see Refs.4 and 5). However, when the shock wave interacts with the boundary layer, which for the moment will be regarded as turbulent, both the individual and the relative rates of movement of the shocks towards the trailing edge are modified. As shown in Fig.3, after separation occurs over the upper surface, the movement of the shock at this surface is slowed down whilst that of the lower surface shock is hastened.

The photographs of Fig.5 show the following stages in the flow development as the Mach number is increased towards unity:-

Fig.5 (a) A series of weak shock waves has formed at the upper surface; probably the waves seen in the photographs are not stationary but travelling upstream, later at a higher Mach number, they coalesce into a single stationary shock.

(b) The shock on the upper surface is now strong enough to cause the flow to separate, but probably only locally near the foot of the shock; a series of weak shocks now exists at the lower surface.

(c) There has been only a very slight rearward movement of the upper surface shock but the separation now extends over the rear of the aerofoil; a single shock is present at the lower surface but is not strong enough to cause separation.

(d)/

(d) The upper surface shock has been overtaken by the shock at the lower surface which is now causing slight separation.

(e) With the free-stream close to sonic speed, both shocks have reached the trailing edge.

An important feature in the variation of the position of the shock wave with incidence is shown up by considering an aerofoil initially at low incidence and at a Mach number sufficient to produce a small region of supersonic flow over the upper surface. When the Mach number remains constant and the incidence is gradually increased, as shown in Fig.4, the shock moves rearwards over the surface until separation occurs, after which the direction of movement is reversed. If for some reason separation did not occur, the rearwards movement of the shock would be expected to continue as indicated by the dashed line in the diagram.

The variation of the position of the shock with incidence which has just been described relates to quasi-static changes. A full understanding of the influence of shock-induced separation on the flow over oscillating aerofoils would require information on the effects of rapid changes of incidence or of flap deflection on the shock-wave boundary-layer interaction. For cyclic variations of incidence at relatively low values of the reduced frequency ($\omega c/V \simeq 0.1$) measurements[6] have shown that the shock movement is slightly delayed and that for any particular instantaneous incidence the general flow pattern is similar to that for steady motion but at a different incidence. However in a more recent, and as yet unpublished, investigation at the N.P.L. in connection with the problem of a high-speed aircraft meeting a gust, Chinneck has observed the behaviour of the shock on the upper surface of an aerofoil when the incidence is rapidly increased. A preliminary result for a rapid variation of incidence from $0°$ to $9\frac{1}{2}°$, Fig.5, shows that in comparison with the quasi-static variation, (1) the formation of the shock is delayed to a higher incidence and, (2) for any incidence less than about 7 degrees the shock is situated more to the rear of the aerofoil.

Before discussing the instabilities arising from shock-induced separation a general comment must be made regarding the state of the boundary layer. This is that the type of interaction which occurs depends on whether the boundary layer has become turbulent or has remained laminar up to the point of separation (see Fig.2). In the latter case the effects of the interaction are also more dependent on Reynolds number. The differences between the conditions under which laminar and turbulent boundary layers separate and behave after separation necessitate caution in applying the results obtained with small models with extensive laminar boundary layers to the conditions of flight where turbulent boundary layers are more common[7].

3. Flow Instabilities with Rigid Aerofoils

The flow separation caused by a shock wave can lead to flow unsteadiness involving fluctuations in the pressure distribution over the aerofoil and in the wake sometimes accompanied by considerable movements of the shock. It is clear from a consideration of the pressure change across a shock-wave, (see Fig.1), that, should any oscillatory motion of the shock occur, any point on the surface over which it passes would experience severe pressure fluctuations. There is evidence from flight[8,9,10,11,12], that buffeting encountered at high subsonic speeds can be associated with shock-induced separation and can be attributed to fluctuating loads on the wing. That the onset of the fluctuations coincides with the effects of the separation reaching the trailing-edge is suggested by the observed close correlation between the buffet boundary and the falling-off of the trailing-edge pressure[11,12]. If a shock wave forms in the flow over a fuselage, even a separation of very limited extent could possibly lead to excessive noise being transmitted into the interior.

Most/

Most of the available information concerning flow fluctuations has been obtained in wind tunnel tests and mainly with two-dimensional models. Some of the earliest recorded examples of instabilities arising from shock-induced separation on aerofoils were those encountered by Daley and Humphries[13] in investigating the flow over a series of propeller sections which had high (very high by wing standards) thickness to chord ratios. At Mach numbers above the critical for shock formation, a violent high-frequency oscillation of the flow was encountered which was characterized by a relatively regular out-of-phase backwards and forwards motion of the shock waves at the upper and lower surfaces. This motion was accompanied by an upwards and downwards deflection of the wake. The significance of this large scale type of oscillation is not quite clear since the investigations of Eggink[14] and of Liepmann and Ashkenas[15], in which similar oscillations were found, showed that the tunnel itself, unless it had a sonic throat downstream of the model, could be a source of disturbance which produced the unstable flow. On the other hand a similar type of instability was observed by Holder and North[16], even after the disturbances originating from the tunnel were suppressed. A schlieren photograph, Fig.7, taken using a short duration spark shows the shock waves and wake at one instant during the oscillation. It was found, however, that this instability could be suppressed by introducing rough turbulence into the stream. A general conclusion that can be drawn from the investigations already mentioned is that, in addition to instabilities arising from the wind tunnel itself, a violent, large scale, flow oscillation can occur with aerofoils having high thickness to chord ratios and laminar boundary layers.

A different type of oscillatory flow has been found to occur due to a shock wave interacting with a turbulent boundary layer[3,17]. This consists of bursts of periodic formations of eddies in the wake behind the aerofoil, rather like a Kármán vortex street (see Fig.8). The reduced frequency ($\omega c/V$) for the violent type of instability occurring with a laminar boundary layer was in the range 1 to 2; that for the instability occurring with the turbulent boundary layer was very much higher, being in the region of 50.

A large number of measurements of the fluctuations in the pressures and forces acting on aerofoils at high subsonic speeds has been made at the NACA (Refs.18 to 25). Some of these investigations included pressure measurements in the wake. Although information concerning the flow was not always obtained, the fluctuations of large magnitude which were encountered occurred under conditions for which shock-induced separation would be expected. The wide range of Reynolds numbers covered in the tests indicates that the occurrence of fluctuations is not dependent on the precise nature of the shock-wave boundary-layer interaction. Particular reference will now be made to an investigation of Humphries[21] in which measurements were made on four aerofoils having thickness to chord ratios of 4, 6, 9 and 12 per cent. Although not explicitly stated in the report, the schlieren photographs indicate that the boundary layer was laminar up to the point of the shock interaction. As an example the distributions of the magnitude of the pressure fluctuations over the upper surface for several Mach numbers are shown in Fig.9. In general for any given Mach number the magnitude of the fluctuations increases with distance from the leading-edge, reaches a maximum and then decreases as the trailing edge is approached. The position of the maximum fluctuation is probably close to the foot of the shock and thus the rearwards movement of the peak with increasing Mach number accords with a rearward movement of the shock. It will be noted that the magnitude of the maximum fluctuation decreases as the free-stream Mach number approaches unity. For a 12 per cent thick aerofoil near to the zero lift condition there appeared to be bursts of fixed frequency fluctuations in the surface pressures, Fig.10(a), whilst at high angles of incidence the fluctuations were more random, Fig.10(b).

Whether regular or random fluctuations occur would also appear to depend on the thickness to chord ratio of the aerofoil. From an analysis of the results of the various investigations of Refs.18 to 24 involving 17 different aerofoils and wings, Table I has been compiled.

Table I/

Table I

(1) Thickness to chord ratio	(2) Reference	(3) Indication of only random fluctuations	(4) Indication of definite frequencies
0.02	23	X	
0.04	"	X	
0.04	21	X	
0.06	22	X	
0.06	21	X	
0.08	23	X	
0.09	24	X	
0.09	"	X	
0.09	"	X	
0.09	"	X	
0.09	21	X	
0.10	18		X
0.12	23	X	
0.12	21		X
0.13	19		X
0.13	20		X
0.13	20		X

In the above table the aerofoils are listed in column 1 in order of maximum thickness. A cross placed either in column 3 or 4 indicates whether regular or random fluctuations were encountered. The disposition of the crosses clearly indicates that definite frequencies are only associated with aerofoils whose maximum thickness is more than about 10 per cent. The definite frequencies encountered in the investigations covered in this survey are shown in Fig.11 as a plot between reduced frequency $\omega c/V$ and Mach number. It will be noticed that the frequencies lie within the range $0.5 < \omega c/V < 2.0$ which suggests that these fluctuations could be important sources of structural vibration.

A simple formula giving an estimate of the oscillation frequency of a shock in terms of its chordwise position and the Mach number was put forward by Erickson and Stephenson[19], originally in connection with control surface instabilities. In some cases[19,20] this was found to give values in fair agreement with the measured frequencies.

4. Instabilities Involving Oscillation of the Aerofoil or a Control Surface

Measurements of the pitching moments for aerofoils performing pitching oscillations[26,27] have shown that both the inphase (or stiffness) and the out-of-phase (or damping) components undergo large changes as the Mach number is increased above the critical. Under some conditions regions of oscillatory instability or negative damping have been encountered which, it has been suggested[26], can probably be associated with shock-induced separation, although, as far as is known, no physical explanation of the instability has been advanced.

Fig.12 shows some unpublished measurements made by Bratt at the N.P.L. for a 10 per cent thick aerofoil performing pitching oscillations about an axis 0.445c from the leading-edge. The stiffness and the damping curves show large changes of slope at the higher Mach numbers including a sudden change in the damping from a stable to an unstable value. Although the onset of shock-induced separation was not measured in these particular tests, other measurements with a similar aerofoil show that at this incidence separation would probably first occur when the Mach number is about 0.81, which is close to the Mach number at which there is a steep fall of damping.

Rather/

Rather more attention has been paid to the oscillatory instabilities of control surfaces at high subsonic speeds known as single degree of freedom flutter, or "buzz". Although the present discussion refers only to what which depends on flow separation, two-dimensional potential flow theory[28,29], suggests that other types of single degree-of-freedom oscillations of control-surfaces can occur at transonic speeds.

The early investigations[19,30,31,32] into control surface instabilities at high Mach numbers showed that they occurred in the presence of shock waves which executed a backwards and forwards motion over the surface of the wing or aerofoil in synchronism, but not in phase, with the motion of the flap. Erickson and his associates[19,30] showed that the cyclic movement of the shock wave lagged behind the motion of the flap and this phase lag featured prominently in several of the theories put forward at the time. Shock-induced separation was known to be involved. The precise rôle that this was playing in the instability was, however, not clear although the results of an experiment made at low speed showed that a similar type of instability could occur in the presence of a separation of varying severity artificially provoked and coupled with the flap motion[33].

Although regions of negative damping and self-excited oscillations of control surfaces have been encountered and their prevention studied in wind tunnel and flight tests (Refs.34 to 38), little further investigation appears to have been made into the mechanism of the instability. Recently some attention has been given to the subject at the N.P.L. during an investigation which will be briefly described in the following paragraphs.

Table II

Data for Aerofoil-Flap Combination used in N.P.L. Experiments

Aerofoil section	RAE 102
Thickness to chord ratio	0.10
Position of max. thickness	0.356 chord
Aerofoil span	14 in.
Aerofoil chord	9 in.
Flap to aerofoil chord ratio	0.25

Hinged flaps	Stiffness	Natural frequency
"Free"	approx 9 lb in./radn	approx 13 c.p.s.
Spring constrained	330 lb in./radn	76 c.p.s.

5. Recent Work at the N.P.L. on Control Surface Buzz

5.1 Description of experiments

The experiments, which were designed to throw some light on the part played by shock-induced separation in the buzz instability, were made with a two-dimensional rigid aerofoil spanning the working section of a transonic wind tunnel. Data concerning the aerofoil and the flaps are given in Table II, whilst Fig.13 shows a photograph of the model supported by tongues in the glass windows of the tunnel so that optical observations of the shock waves and boundary layers could be made. The leading-edge of the aerofoil was roughened with a band of carborundum grains to provide a turbulent boundary layer which is more representative of flight conditions. Two alternative hinged flaps could be fitted; these were nominally identical in their extenral shapes but differed in their elastic hinge stiffnesses, one being almost freely hinged. Both flaps had spring hinges with very low

inherent/

inherent dampings and each had snubbers set to limit the amplitudes to approximately ±7 degrees. The hinged flaps could be replaced by a solid tail-piece to provide a plain aerofoil which was used for comparative observations.

The onset of a flap instability was determined by gradually increasing the Mach number until the flap was seen to oscillate, the value at which this occurred being called the "buzz critical Mach number". The onset of buzz was found to be very sensitive to Mach number, the oscillation growing rapidly to a sustained amplitude of several degrees, as soon as the critical value was reached. On lowering the Mach number again, the oscillation persisted until the value was only about 0.005 less than that at which the buzz started. The buzz critical Mach number for the flap which was elastically constrained was, as shown in Fig.14, slightly higher than that for the freely hinged flap, although it is possible that some of this difference can be attributed to small differences in the external shape or setting of the two flaps. Although the critical Mach number for buzz was not radically changed, it is probable that the amplitude of the oscillation was reduced by the increase in the wing stiffness.

Optical observations of the flow by the schlieren and the direct shadow techniques showed that, for incidences less than about 7 degrees, over a small range of Mach numbers just before the buzz critical was reached the usual features occurred indicative of a shock wave interacting with a turbulent boundary layer, namely bifurcation of the foot of the shock wave accompanied by a separation and thickening of the boundary layer, (Fig.2(b). A slight fluctuation in the position of the shock and an intermittent flicking of the boundary layer away from the surface downstream of the shock could sometimes, but not always, be detected just before buzz commenced. The photographs of Fig.15 for incidences 0°, 2° and 4° show the flow conditions immediately before the critical condition was reached. It will be noticed that the appearances of the bifurcation at the foot of the shock (or shocks for 0° incidence) and of the boundary layer separation are almost identical in each of these photographs.

Comparative observations of the flow over the plain aerofoil in Mach number regions below and above the buzz critical showed that similar small scale fluctuations were present. Analysis of high-speed ciné films of the shock waves showed that these inherent flow fluctuations were of a random nature and that the total excursion of the foot of the shock along the aerofoil surface was only about 0.005 chord, and thus, as will be seen later, very much smaller than that occurring during the buzz instability. It seems reasonable to regard the control surface instability, not as a consequence of the flow fluctuations, but rather as an independent effect arising from shock induced separation.

Observations of the behaviour of a thin film of oil on the upper surface of the plain aerofoil, Fig.16, indicated that buzz did not occur until the Mach number had been raised sufficiently for a region of reversed flow to extend over the flap. As seen in the diagram, the flow was then no longer two-dimensional, the area of reversed flow originating near, and expanding outwards from, the centre of the span. Also, unless the incidence was small, the region of reversed flow near the centre of the span was accompanied by a bowing forward of the shock wave. It has already been mentioned that a change in the pressure measured at the trailing-edge of an aerofoil has been found to coincide with the onset of other effects of shock-induced separation such as buffeting. As shown in Fig.17 the Mach number corresponding to the falling off of the trailing-edge pressure measured for the plain aerofoil agrees fairly well with the critical Mach number for buzz.

In agreement with the usual type of separation boundary[8,11] for an aerofoil at high subsonic speed, the buzz critical Mach number falls with increasing incidence as shown in Fig.14. This diagram also shows the frequencies of the buzz oscillation just after onset. However, the frequency

variation/

variation for the spring constrained flap, particularly at the higher incidences, may not be significant owing to the occurrence during the tests of changes in the elastic stiffness whose magnitudes are unknown. That a difference between the buzz frequencies for the two flaps does exist is evidence that the buzz is not a response of the flap to an inherent flow oscillation. It seems likely that the buzz frequency is determined simply by the moment of inertia of the flap and the sum of the aerodynamic and elastic hinge-stiffnesses. The results of such a calculation in which the aerodynamic hinge-stiffness was based on measurements of the static hinge moments for a similar aerofoil-flap combination is shown in Fig.18. The variation of the calculated frequency shown in this diagram, which refers to a constant stagnation pressure of 1 atm. as in the tests, is due solely to the variation of the quantity $\frac{1}{2}\rho V^2$.

For the various buzz critical conditions with increasing incidence, the shock at the upper surface and thus the point of separation moved towards the leading-edge. Between 7° and 8° incidence the shock was so close to the leading-edge that the precise nature of the separation was not clear. Above this incidence, although small shock waves continued to be present near the forward part of the aerofoil, the separation appeared to be similar to that which occurs near a leading-edge at low speed. Since the buzz critical Mach number and the buzz frequency vary smoothly with incidence in spite of the change from a shock-induced to a leading-edge separation, it may be concluded, in agreement with Ref.33, that the essential feature of the instability is the separation and not the shock wave.

The region corresponding to Mach numbers above the critical for the onset of buzz was found to be extremely difficult to explore systematically since a very violent behaviour of the flap accompanied the passage of shock waves across its surface. In addition to oscillatory instabilities, large destabilizing steady hinge moments occurred which caused the flap to move to one or other of the limits set by its snubbers, often in a manner which was not repeatable from one run to the next. Observations close to a free-stream Mach number of unity could only be made for low incidences or otherwise the loadings would have been excessive. But these consistently showed that when both the upper and the lower surface shocks reached the trailing-edge and thus no extensive regions of separated flow existed, see Fig.19(b), the flap remained stable. Between the initial onset of buzz and this stable condition close to sonic speed the existence of two kinds of oscillatory instability, either in separate regions or overlapping, according to the incidence, would seem likely. An intermediate region of stability in the presence of shock-induced separation was consistently encountered at incidences between -1° and +1° when the Mach number was raised sufficiently for both shocks to move rearwards to positions just forward of the flap hinge, (Fig.19(a)). A possible reason for this stable condition is discussed later in Section 5.3.

5.2. Analysis of the buzz oscillation

High-speed ciné films obtained using the schlieren method of flow visualization have shown the cyclic behaviours of the flap, the shock waves and the boundary layers during a buzz oscillation. Sequences from films of the oscillation for 0° and 4° incidence are shown in Figs.20 and 21 respectively, whilst the results of an analysis of another portion of the film for 4° incidence are shown in Fig.22.

In each of Figs.20 and 21 the photographs are prints of alternate frames of the film and, although the amplitude of the oscillation was growing very slightly, it has been possible to arrange them in a cyclic order corresponding to a single cycle of the motion. In the left-hand pictures the flap is moving upwards, in the right-hand pictures it is moving downwards, and as far as possible the two pictures of each horizontal pair show the flap deflected by about the same amount. (It is not possible to match the pairs exactly because the picture frequency in the camera was not an exact multiple of the oscillation frequency).

Referring firstly to the sequence for 4° incidence, it is obvious from the photographs that the oscillation of the flap is coupled with large variations in the position of the shock, in the strength of the shock and in the severity of the separation. A close examination of the sequence of shock positions reveals that the shock is in the process of moving forwards in frames (b), (c) and (d) and moving rearwards in frames (f), (g) and (h). It is very probably close to its extreme rearward position in frame (a) and to its extreme forward position in (e). It is then clear that when moving forwards the shock is of high strength and induces severe separation, but on moving to the rear it becomes weaker and no longer causes separation. Indeed in frames (g) and (h) the shock has weakened to such an extent that it is hardly visible. Fig.22 relates to an oscillation of smaller amplitude at the same incidence (4°). In this diagram, against a common time axis, are plotted the deflection of the flap ξ, the position of the shock x_{sh}/c, and the severity of the separation as measured in a rather arbitrary manner by the height δ of the shear layer above the aerofoil surface at a position slightly ahead of the flap. As mentioned previously the shock wave, particularly when separation occurs, is sometimes bowed forwards at the centre of the span of the aerofoil; the two curves showing the shock position correspond to the forward and rearward extent of the base of the shock as seen in the photographs. It will be noticed that the flap deflection and the shock position both vary approximately sinusoidally, and that there is only a small phase lag in the motion of the shock with respect to the flap. As already shown by the sequence of photographs, there is a large phase lag in the severity of the separation so that the flap performs its up-stroke mainly under conditions of shock-induced separation and its down-stroke mostly under conditions of attached flow.

The photographs of Fig.20 show the instability occurring with the aerofoil at zero incidence. For this incidence, under stable conditions before buzz commences, the shock waves are arranged symmetrically at the upper and lower surfaces. It is clear from the photographs that during a buzz oscillation the two shocks execute a backwards and forwards motion out-of-phase with one another and that each has a phase displacement with respect to the flap. Examination of the sequence reveals that the upper-surface shock is definitely moving forward in frames (c), (d), (e) and (f) and definitely moving rearwards in frames (i), (j) and (a). It probably is close to its extreme forward position in either frame (g) or frame (h) and to its most rearward position in frame (b). In comparison with the photographs for an incidence of 4° the cyclic changes in the appearances of the shock and its effect on the boundary layer are not so marked. But it can be seen, for instance, by comparing frames (c) and (i) for which the upper-surface shock is in approximately the same position, that, in agreement with the previous observations, the forward moving shock, frame (e), is stronger and has more influence on the boundary layer than the rearward moving shock, frame (i).

5.3. Considerations concerning the mechanism of the buzz oscillation

In attempting to provide an understanding of the instability it is convenient to consider separately, (1) the effects on the flow of an oscillation of the flap and, (2) the changes in the aerodynamic loading on the flap arising from changes in the character of the flow.

Before considering the effects on the flow of a rapid oscillation of the flap, it is necessary to discuss the relations between the various quantities for the quasi-static condition of a slowly deflecting flap under flow conditions similar to those existing just before buzz occurs. For simplicity we shall consider the aerofoil-flap combination to be at a small angle of incidence with shock-induced separation over the upper surface only; the Mach number will be such that the shock wave remains ahead of the flap hinge for all values of the flap deflection.

The/

The deflection of the flap can be regarded as a source of pressure disturbance which is propagated into the field of flow. In the present simple approach it is justifiable to assume that the shock is the upstream limit to the propagation of this disturbance over the upper surface; thus the flow which remains ahead of the shock is unaltered by the flap deflection. The disturbance, which according to the direction of the flap deflection will be either a compression or an expansion wave, will when it reaches the shock represent an alteration in the pressure ratio across it. In consequence of this change of shock strength and in order to restore the equilibrium of the flow, some change, either in the position of the shock, or in the severity of the separation, or both, must occur.

Starting with the flap in an undeflected condition we now imagine it to be given a downward (i.e. a positive) displacement. In the absence of viscous effects this would be expected to lead to a rearward extension of the supersonic region and thus to a rearward displacement of the shock which would be accompanied by an increase in its strength. As can be seen from the observations of the flow for a statically deflected flap shown in Fig.23(b) and (c), under actual conditions of shock-induced separation, although there is a rearwards extension of the supersonic region, the foot of the shock remains in almost the same position and the severity of the separation increases.* It will be noted that in addition to the increased severity of the separation at the foot of the shock, the thickness of the "dead-air" region over the surface of the flap is also increased by the convex corner at the hinge.

With an upward (i.e. a negative) deflection of the flap the effects are contrariwise (see the change from (b) to (a) in Fig.23); the shock is displaced forwards and becomes weaker; in consequence the severity of the separation is first diminished and eventually suppressed.

Thus if the flap were moved slowly through a cycle, the shock displacement (regarded as positive towards the trailing-edge) and the severity of the separation would both be in-phase with the deflection of the flap.

Proceeding now to a consideration of what happens when the flap undergoes a more rapid oscillation, we must consider two additional factors; these are:-

1. The shock wave does not immediately respond to the movement of the flap owing to the time necessary for the disturbance to travel the distance between the flap and the shock.

2. The strength of the shock is closely related to its speed of propagation in the fluid and thus to its rate of movement over the aerofoil surface.

The first of the factors mentioned above, as pointed out by Erickson and Stephenson[19] and by Smilg[31], would cause a phase lag in the motion of the shock with respect to the flap which the present and previous observations[19] have shown to exist. The relation between the rate of shock movement and the shock strength for the simple case of a normal shock in a parallel stream is obtained immediately from the Rankine-Hugoniot shock-wave equations. As shown in Fig.24, for shock velocities small in comparison with the velocity of the free-stream, the increase in shock strength is directly proportional to the upstream velocity of the shock. For the flow

over/

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*The flow observations and hinge moment measurements with statically deflected flaps which are referred to in this discussion were obtained by Pearcey and Pankhurst during a steady motion investigation with a similar aerofoil and flap combination.

over an aerofoil it is then reasonable to find, and as the observations have shown, that the shock wave in moving forward over the aerofoil surface in response to an upward deflection of the flap has its strength increased and is thus capable of inducing more separation than a similarly situated shock which is moving rearwards and is therefore weaker. Thus it is to be expected that the severity of separation would depend on the angular velocity of the flap. Combining these additional dynamic effects with the quasi-static effects of the flap oscillation on the flow, we should expect to find a tendency for the shock-induced separation to be most severe when the shock is displaced rearwards and moving upstream, and to be least severe when the shock is displaced forwards and is moving to the rear. That this is not completely in accord with the observations suggests that further factors will need to be considered.

We now turn to the effects of changes in the flow on the hihge moment acting on the flap. Measurements of the hinge moments for static deflections of the flap indicate that, for the incidence and Mach number conditions under discussion (i.e. with the shocks remaining ahead of the hinge line), the onset of shock-induced separation on either surface leads to an incremental change in the hinge moment which tends to turn the flap towards the separation; that is, the additional hinge moment tends to turn the flap upwards when the separation occurs on the upper surface. If this remains valid under oscillatory conditions, then it is clear from the observed behaviour during the buzz that the incremental hinge moment due to separation is almost always acting in the direction of motion of the flap; thus it can perform the work necessary to amplify or maintain the oscillation.

It was mentioned earlier, in Section 5.1, that for incidences near zero the flap became stable when the Mach number was increased sufficiently for both the upper and the lower surface shocks to have reached the hinge line of the flap. Observation of the flow with a statically deflected flap reveals that at this Mach number the positions of the shocks are almost unaffected by the flap deflection. Thus with no movement of the shocks, we can no longer associate a variation in the severity of the separation with the angular velocity of the flap. The absence of this dependence might well account for absence of an instability.

It will be remembered that at high incidences an oscillation of the flap occurred in the presence, not of shock-induced separation, but of separation from the leading-edge. A film sequence of this oscillation is shown in Fig.25, from which it can be seen that the upward motion of the flap occurs under conditions of flow separation, whilst for much of the downward motion the flow is attached to the upper surface. To this extent then, the phasing of the separation cycle is similar to that for the previous examples in which the separation was shock-induced. If the assumption is again made that the separation causes an incremental hinge moment tending to turn the flap towards the separation and thus in the direction of its motion, it is clear that this additional hinge moment has the ability to amplify the oscillation.

It seems reasonable to expect that a consideration of the effects of unsteady motion on the behaviour of a boundary layer could predict the observed phasing of the separation cycle. Furthermore it would seem that the cyclic behaviour of the shock-wave boundary-layer interaction, which has already been discussed, is a particular example of the general phenomenon. Motion of the flap leads generally to time rates of change of the velocities at points on the aerofoil surface. Based on existing knowledge of the sensitivity of the boundary layer to an unsteady external stream[39,40] it is reasonable to suppose that a boundary layer in an accelerating stream (i.e. a stream in which the velocities are everywhere increasing with time) is less likely to separate than one in a decelerating stream. Thus the tendency, generally, observed, for separation or attachment to occur on the upper surface according to whether the flap is moving upwards or downwards is to be expected.

It/

It must be admitted that the preceding discussion is only of a tentative and rather speculative nature and is based on inadequate knowledge of the effects of the various factors entering into the buzz instability. Since some of the difficulties encountered concern the separation of cause from effect, there is a need for further experiments designed to examine each factor independently.

6. Concluding Remarks

A brief review has been made of the available information on the flow fluctuations and instabilities arising from shock-induced separation and on the influence this phenomenon has on the oscillatory behaviour of aerofoils and control surfaces. A more detailed discussion has been devoted to a recent investigation at the N.P.L. into the part played by shock-induced separation in the instability of a control surface.

Although for a variety of reasons extensive shock-induced separation would not be tolerated under normal conditions of flight, it would seem that local regions of separation might occur during manoeuvres or under other special conditions. For this reaon some attention should continue to be given to the instabilities themselves as well as to the means of preventing the occurrence of separation.

No attempt has been made in this paper to discuss the prevention of shock induced separation or its effects, although the subject can hardly be left without mentioning that for wings, a low thickness to chord ratio, a small trailing edge angle and sweepback have all been shown to be beneficial, and that methods of boundary layer control such as vortex generators have been used effectively to delay the onset of the effects. Without attempting to modify the actual separation characteristics, control surface instabilities can sometimes be prevented by the use of a high circuit-stiffness or of damping.

Acknowledgement

This paper is presented by permission of the Director of the Laboratory.

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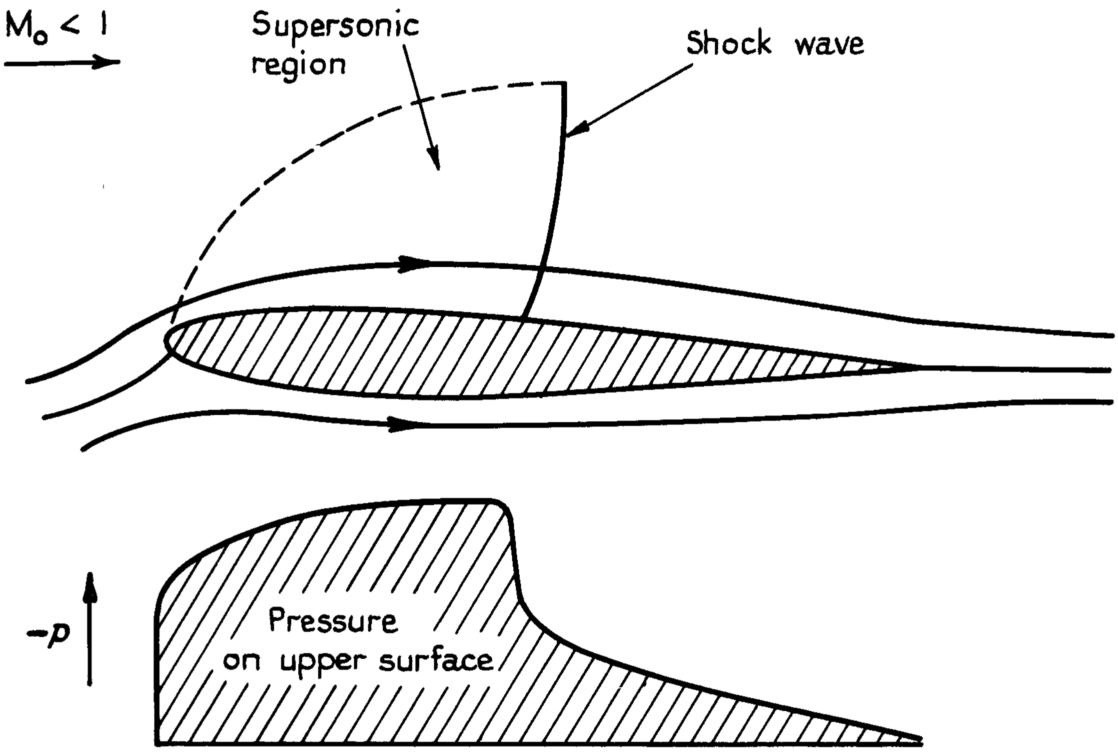
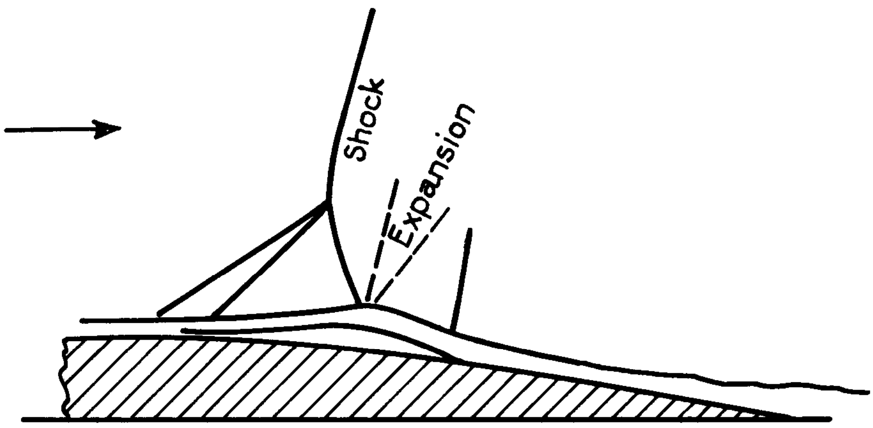
FIG. 1.

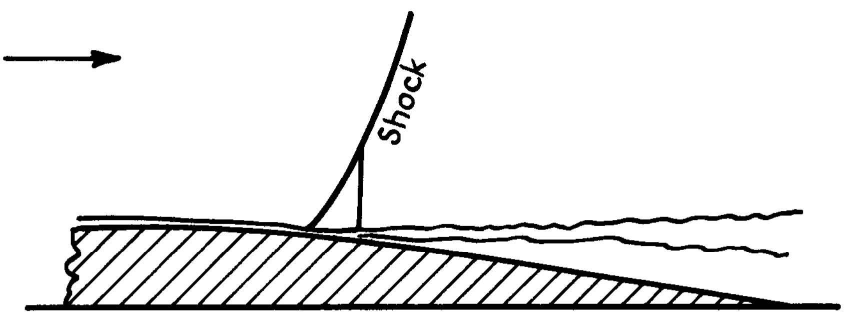


Flow over aerofoil with shock wave

FIG. 2.



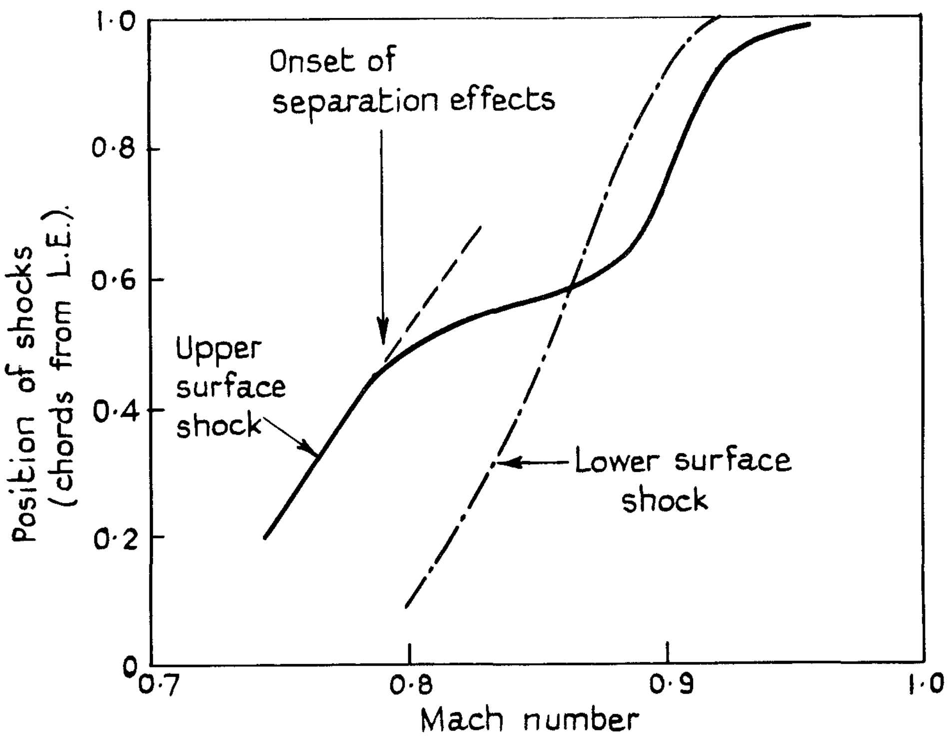
(a) Laminar boundary layer



(b) Turbulent boundary layer

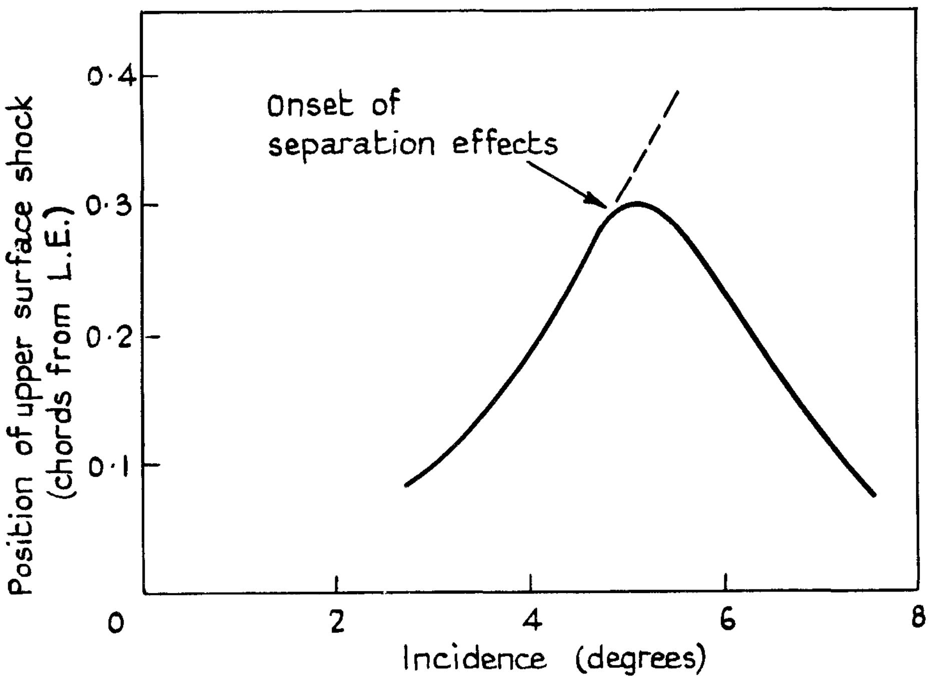
Sketches showing the types of shock-wave interaction with laminar and turbulent boundary layers (From ref. 7)

FIG. 3.

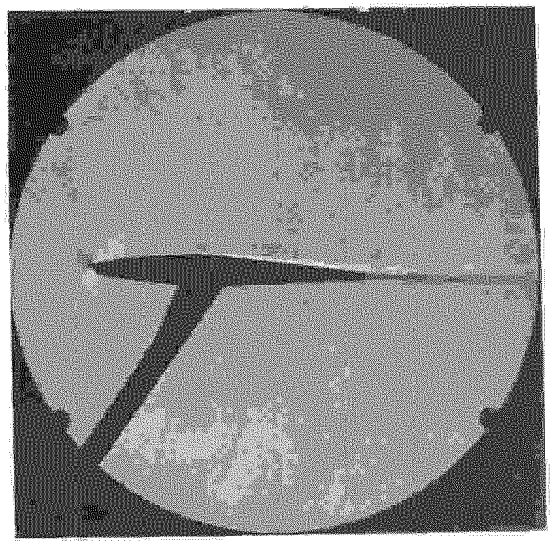


Variation of shock position with Mach number for constant incidence (after ref. 4).

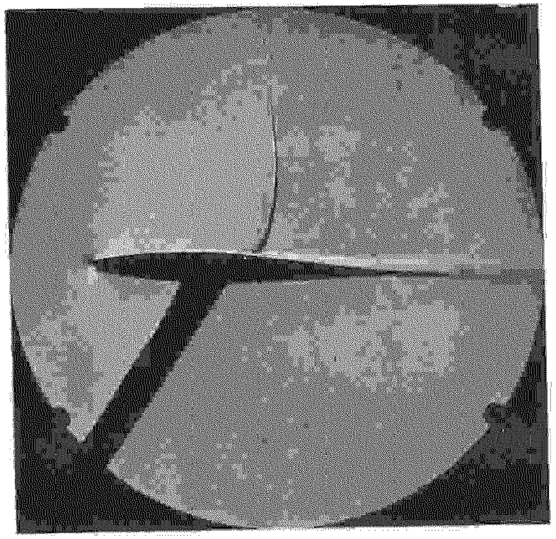
FIG. 4.



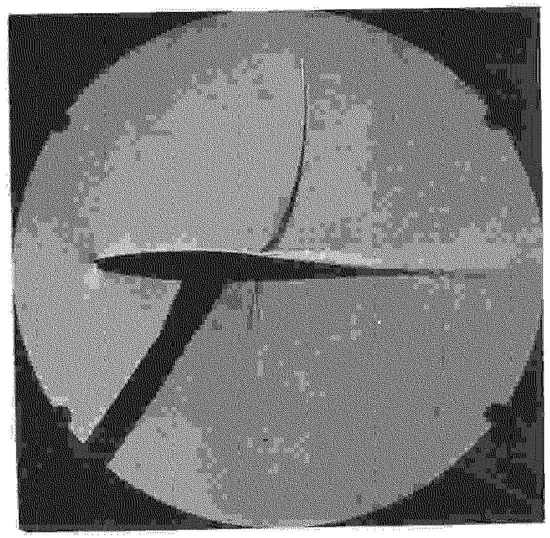
Variation of shock position with incidence for constant Mach number (after ref. 4).



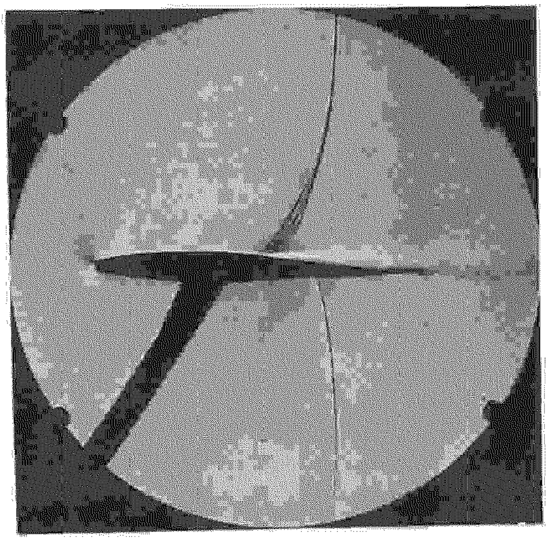
(a)



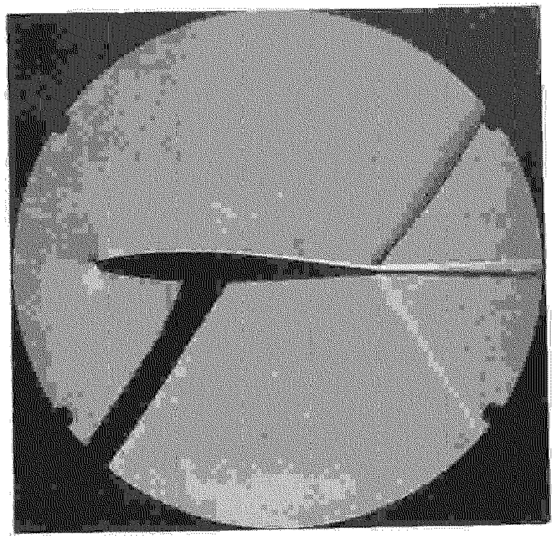
(b)



(c)



(d)

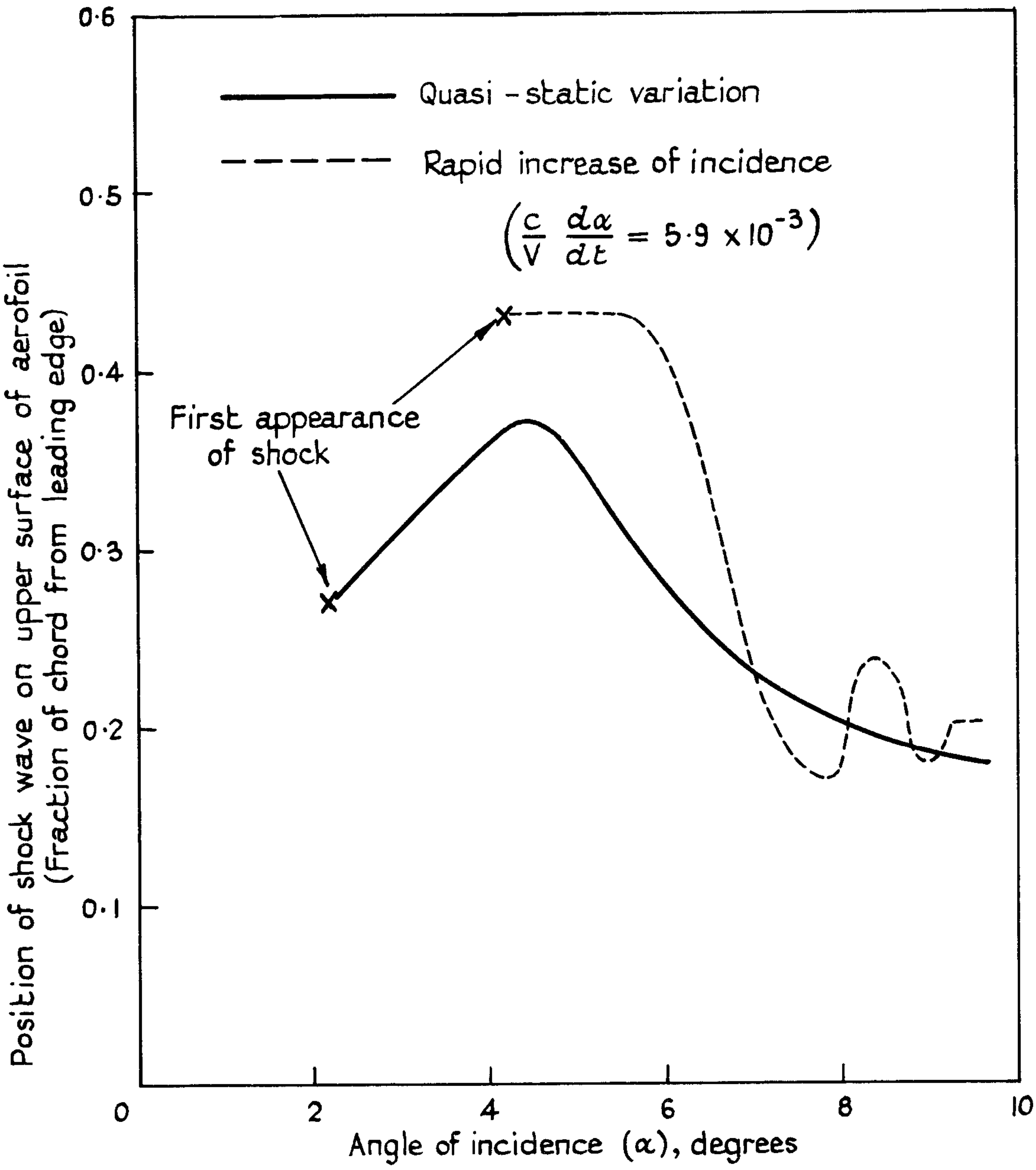


(c)

The development of flow for an aerofoil at a small angle of incidence with Mach number increasing towards unity.

FIG. 5.

FIG. 6.



Effect of rate of change of incidence on movement of upper-surface shock. $M = 0{\cdot}75$ (Chinneck, NPL).

FIGS. 7 & 8.

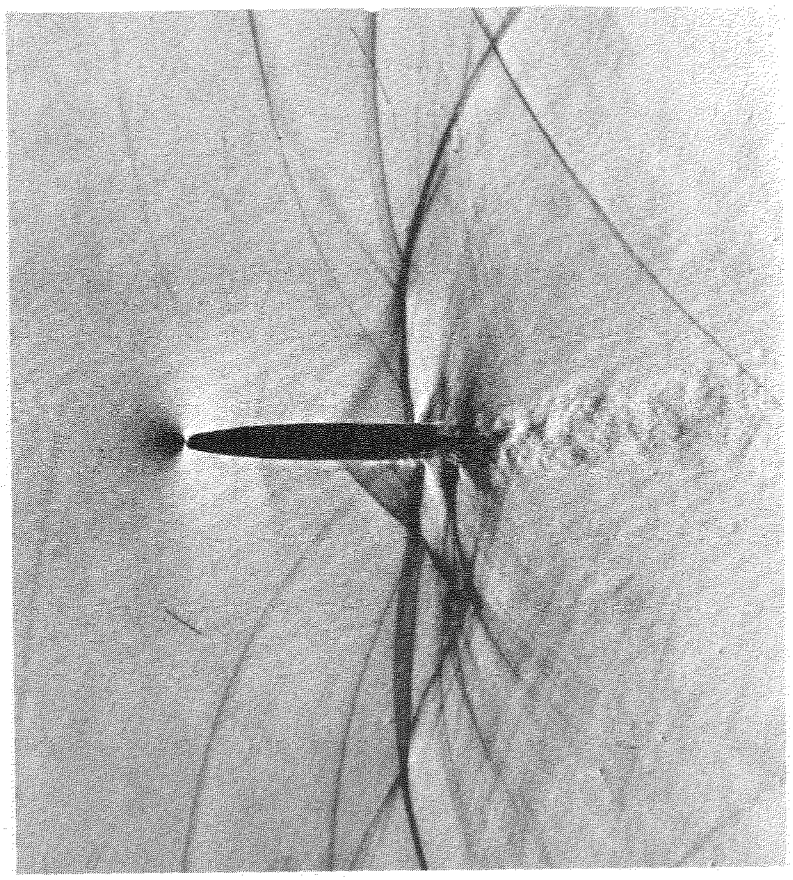


FIG. 7.

Oscillatory flow due to shock-induced separation of laminar boundary layer. E.C.1250 aerofoil 12% thick. $\alpha = 0^{\circ}$, $M = 0{\cdot}82$ (Ref. 17)

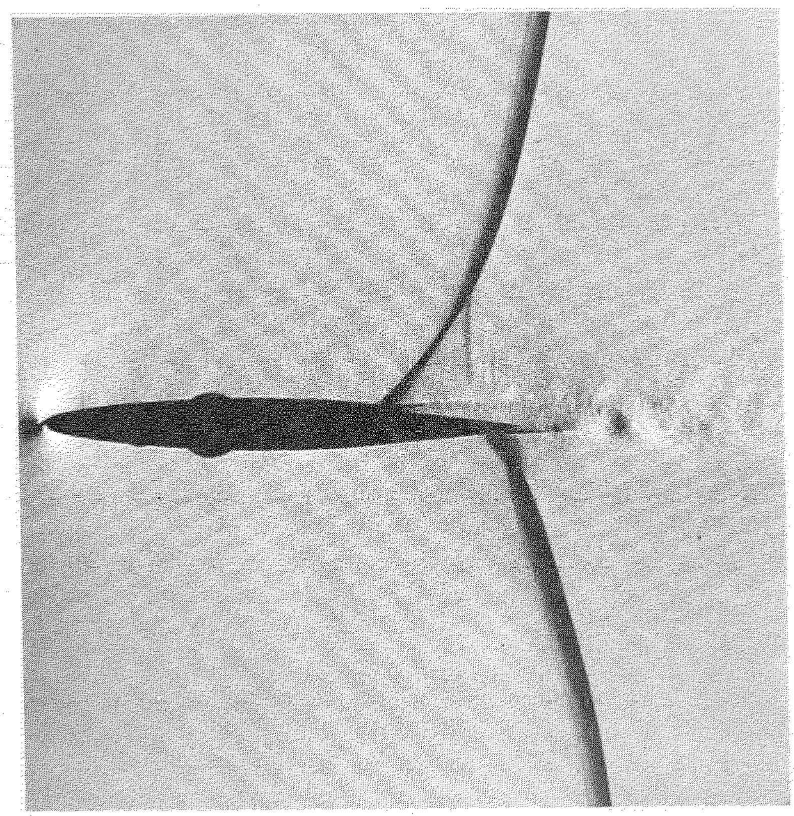
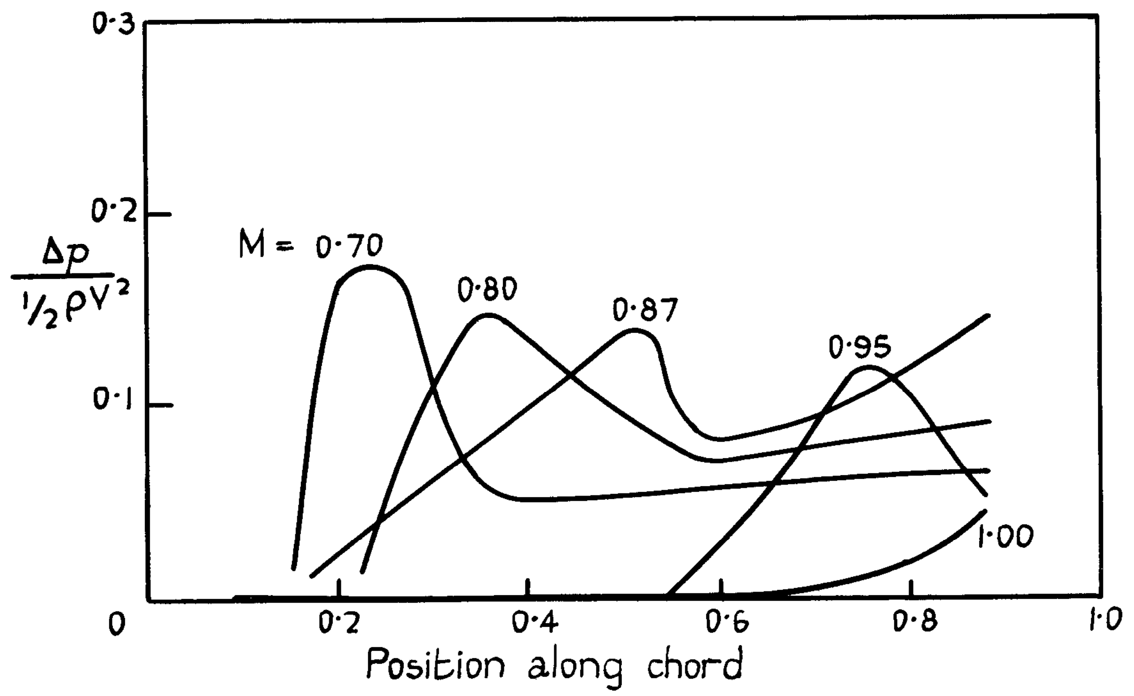


FIG. 8.

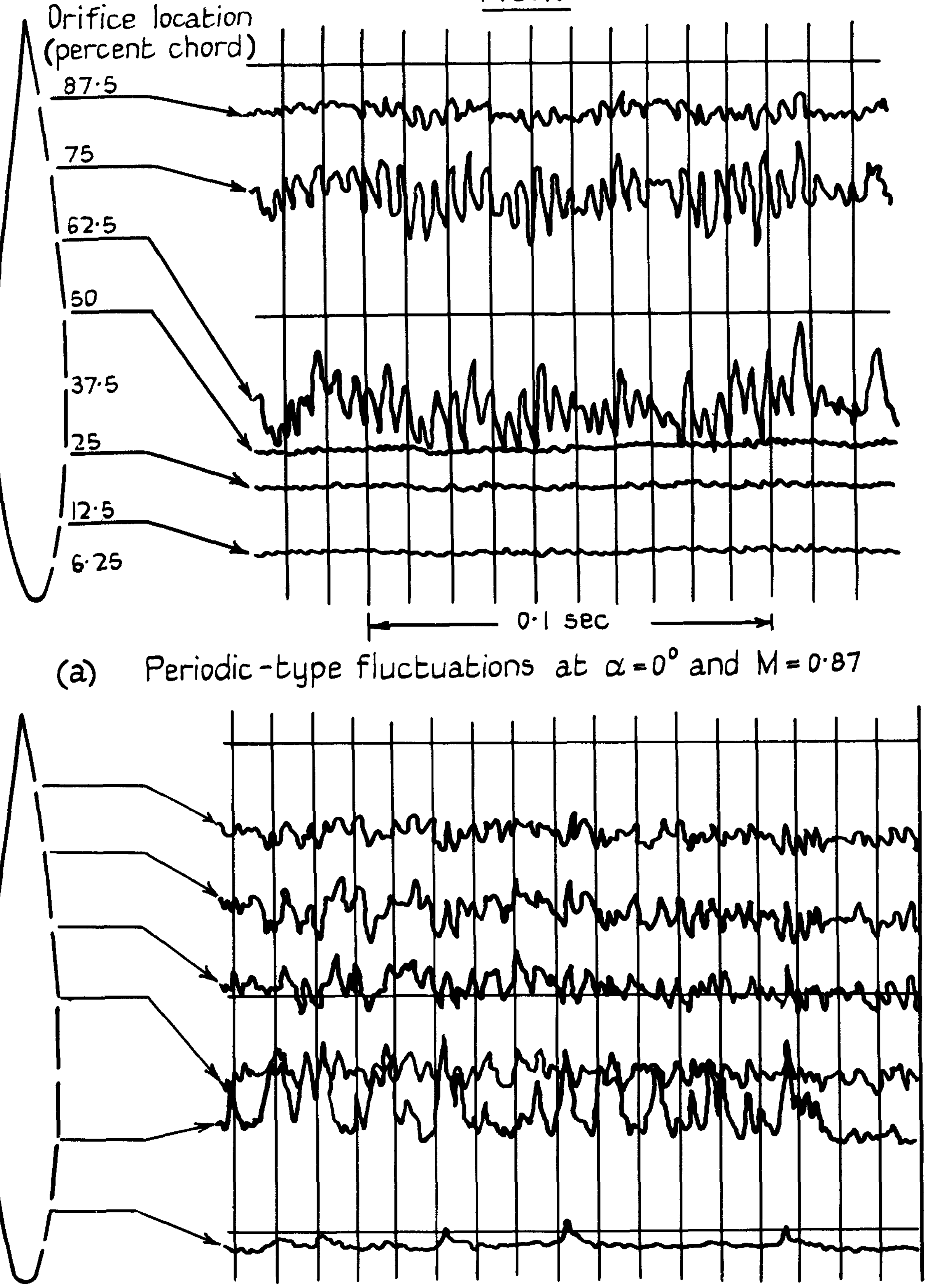
Eddying wake downstream of shock-induced separation of turbulent boundary layer. RAE 104 aerofoil. 10% thick. $\alpha = 2^{\circ}$, $M = 0{\cdot}87$ (Ref. 17)

FIG. 9.



Magnitude and distribution of pressure fluctuations over upper surface of 9% thick aerofoil at 6·4° incidence for various Mach numbers (from ref. 21)

FIG. 10.

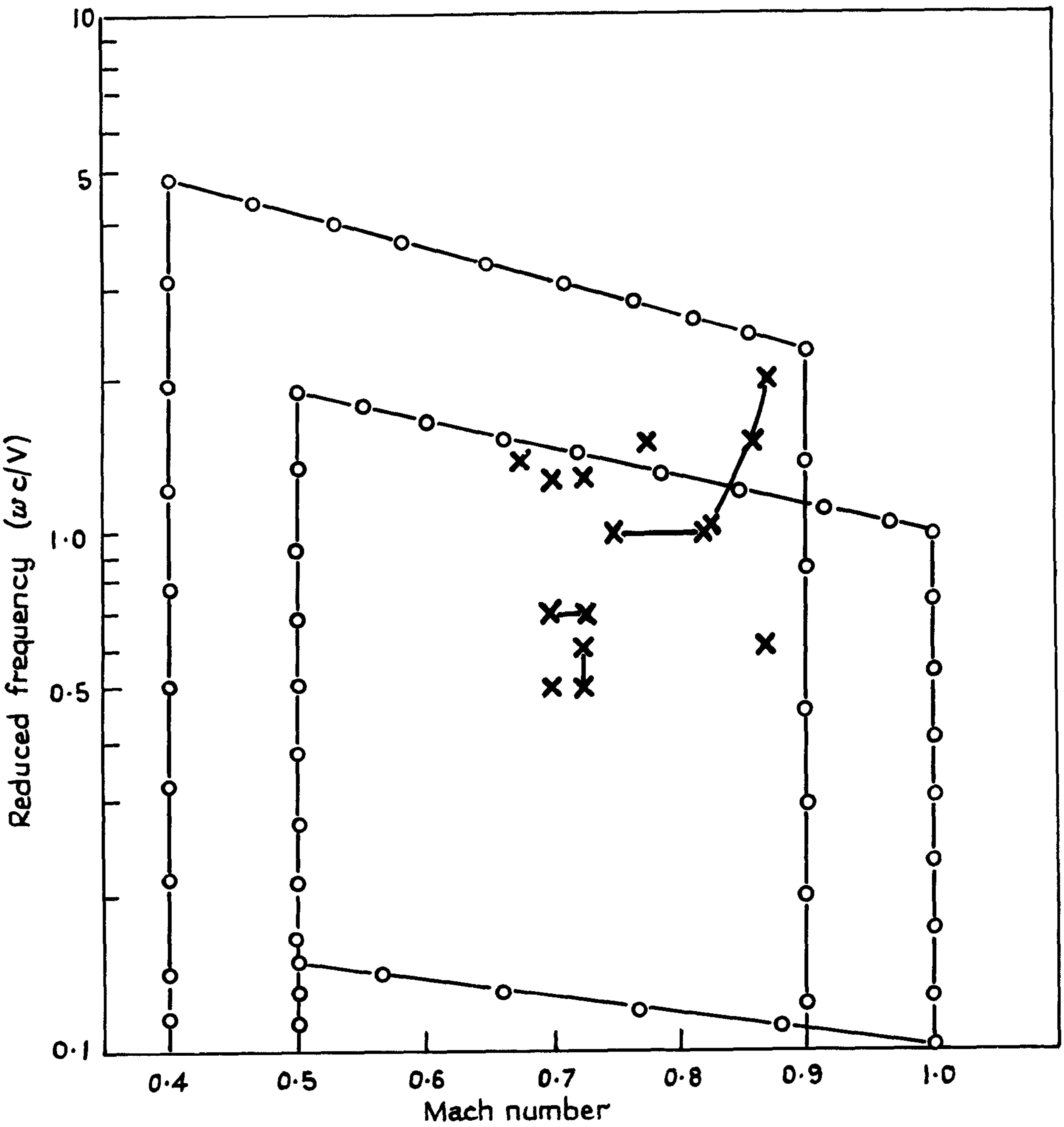


(a) Periodic-type fluctuations at $\alpha = 0^{\circ}$ and $M = 0{\cdot}87$

(b) Random-type fluctuations at $\alpha = 8^{\circ}$ and $M = 0{\cdot}80$

Examples of pressure fluctuations at surface of 12% thick aerofoil (from ref. 21)

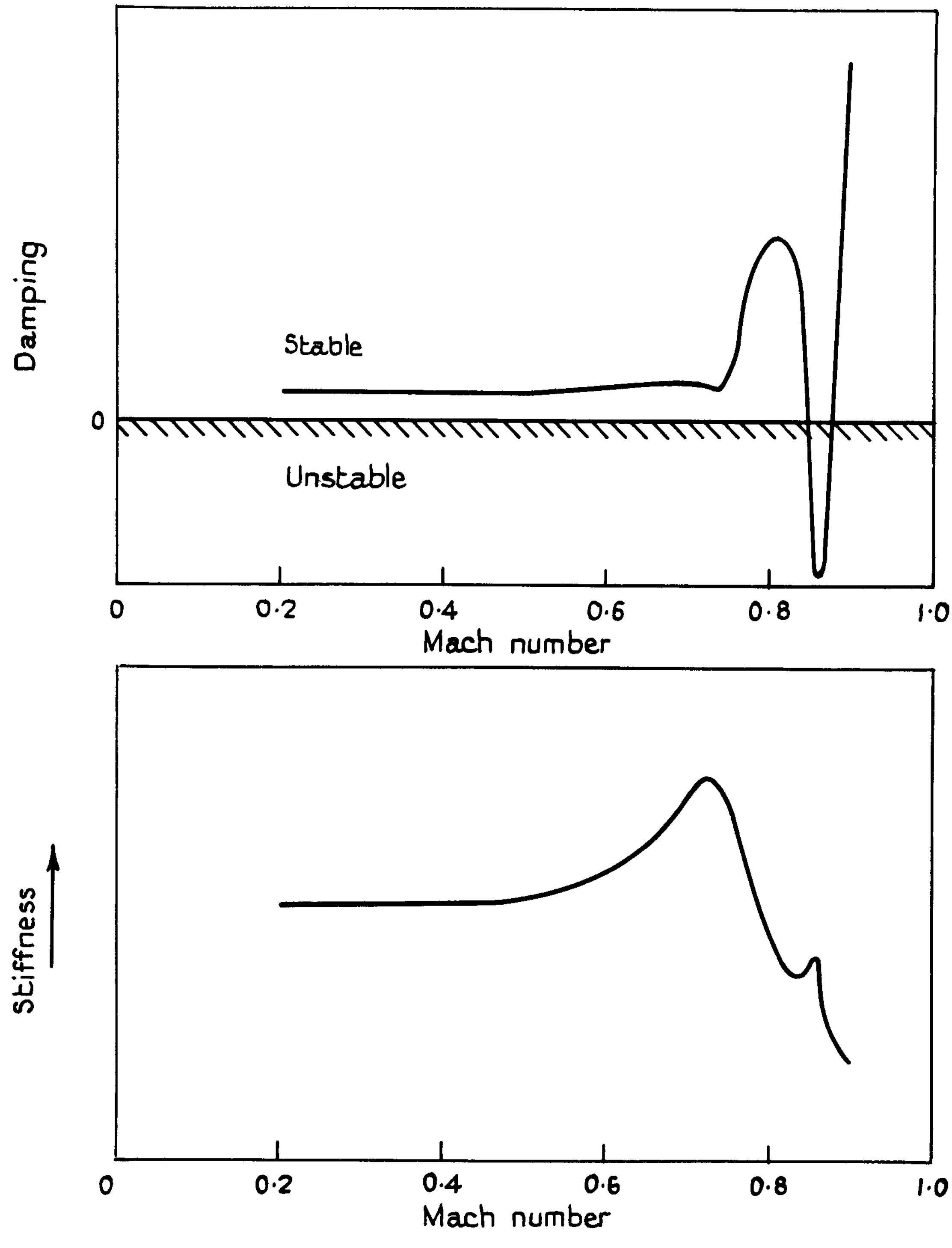
FIG. 11.



Flow fluctuations with rigid aerofoils

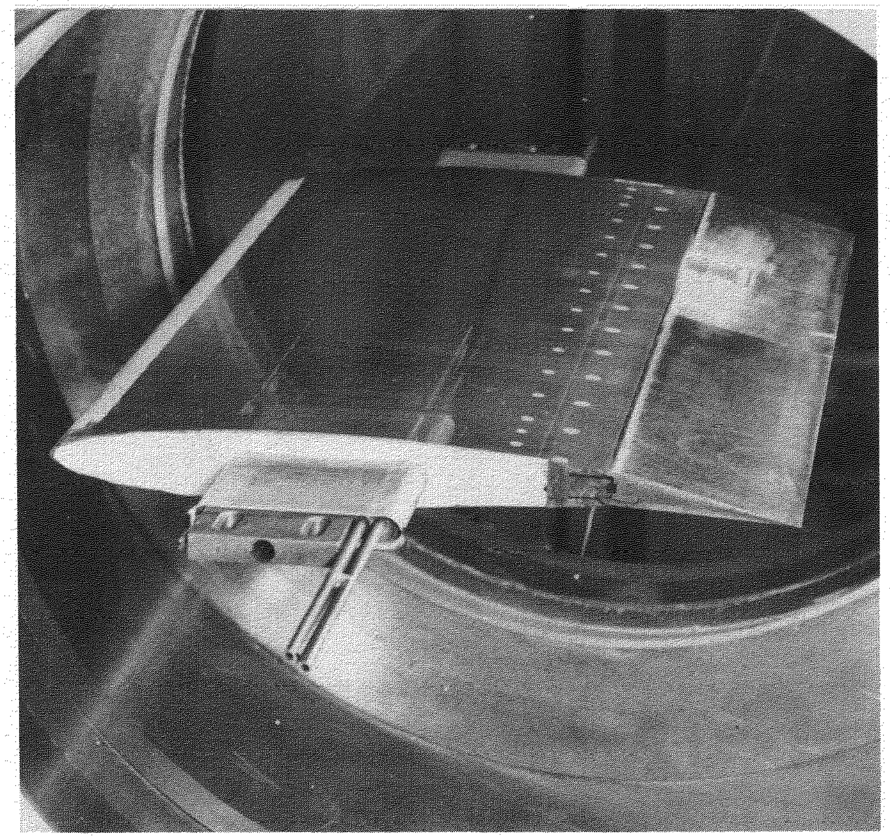
Definite frequencies indicated by x
Areas covered in searches enclosed by o—o—o

FIG. 12.



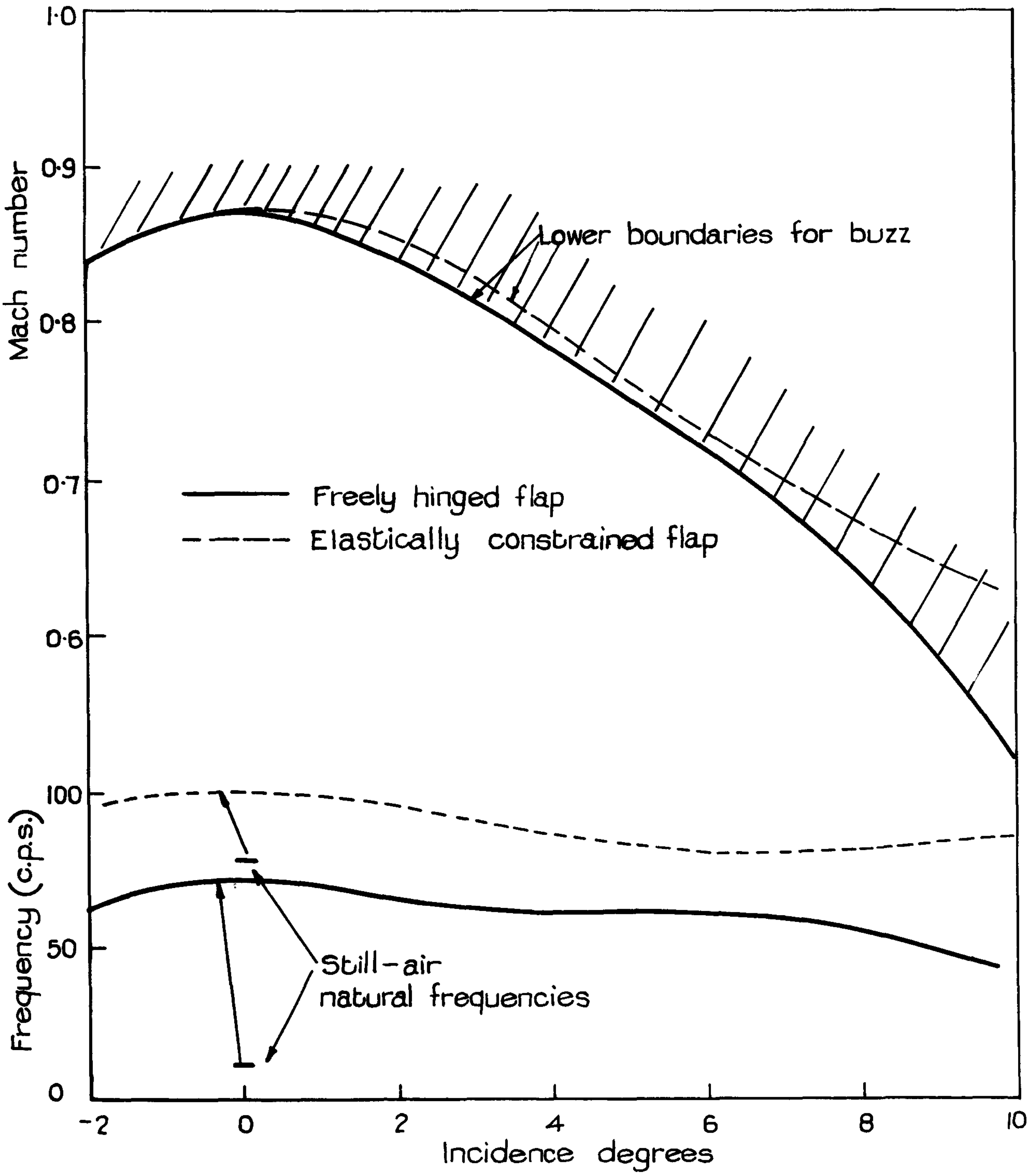
The aerodynamic stiffness and damping for an aerofoil pitching about an axis 0·445c from leading-edge. Aerofoil thickness 10%, mean incidence 3°, amplitude 2°, $(\varpi c/V)$ varies with M between 0·04 and 0·01 (Bratt, NPL)

FIG. 13.



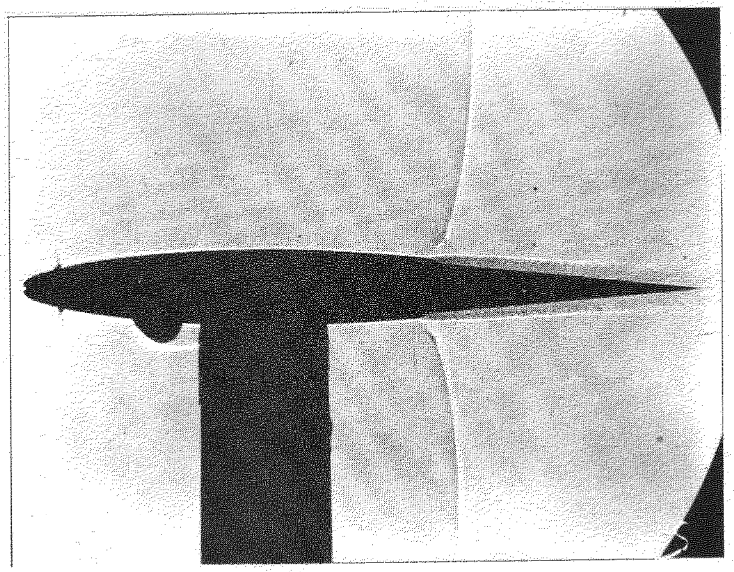
Aerofoil-flap combination mounted in tunnel for buzz experiment

FIG. 14.

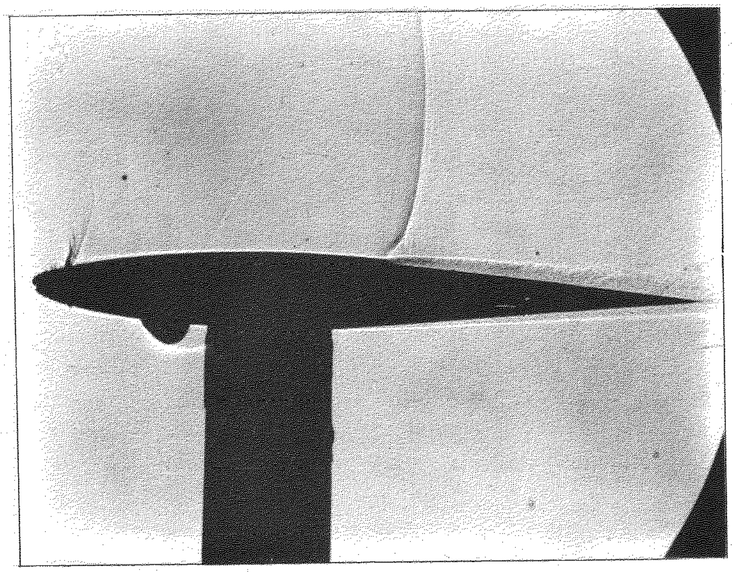


Variation with incidence of critical Mach number for buzz and buzz frequency, (constant stagnation pressure = 1 atm.)

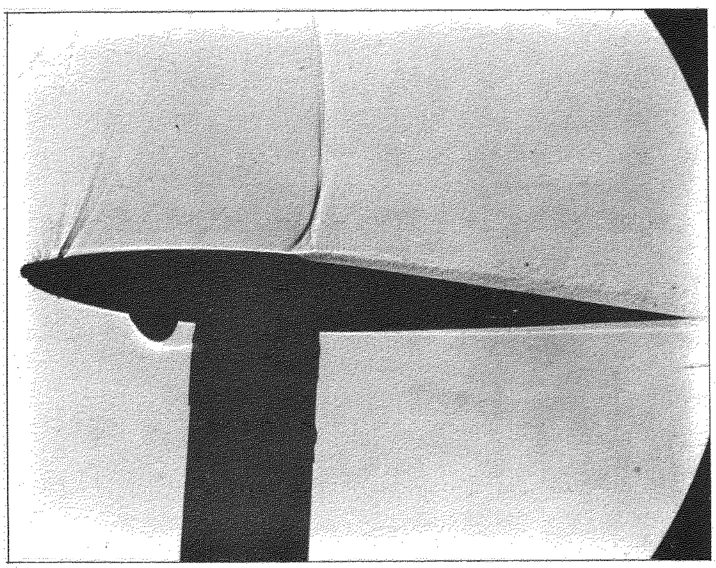
FIG. 15.



(a) Incidence 0° M=0·867



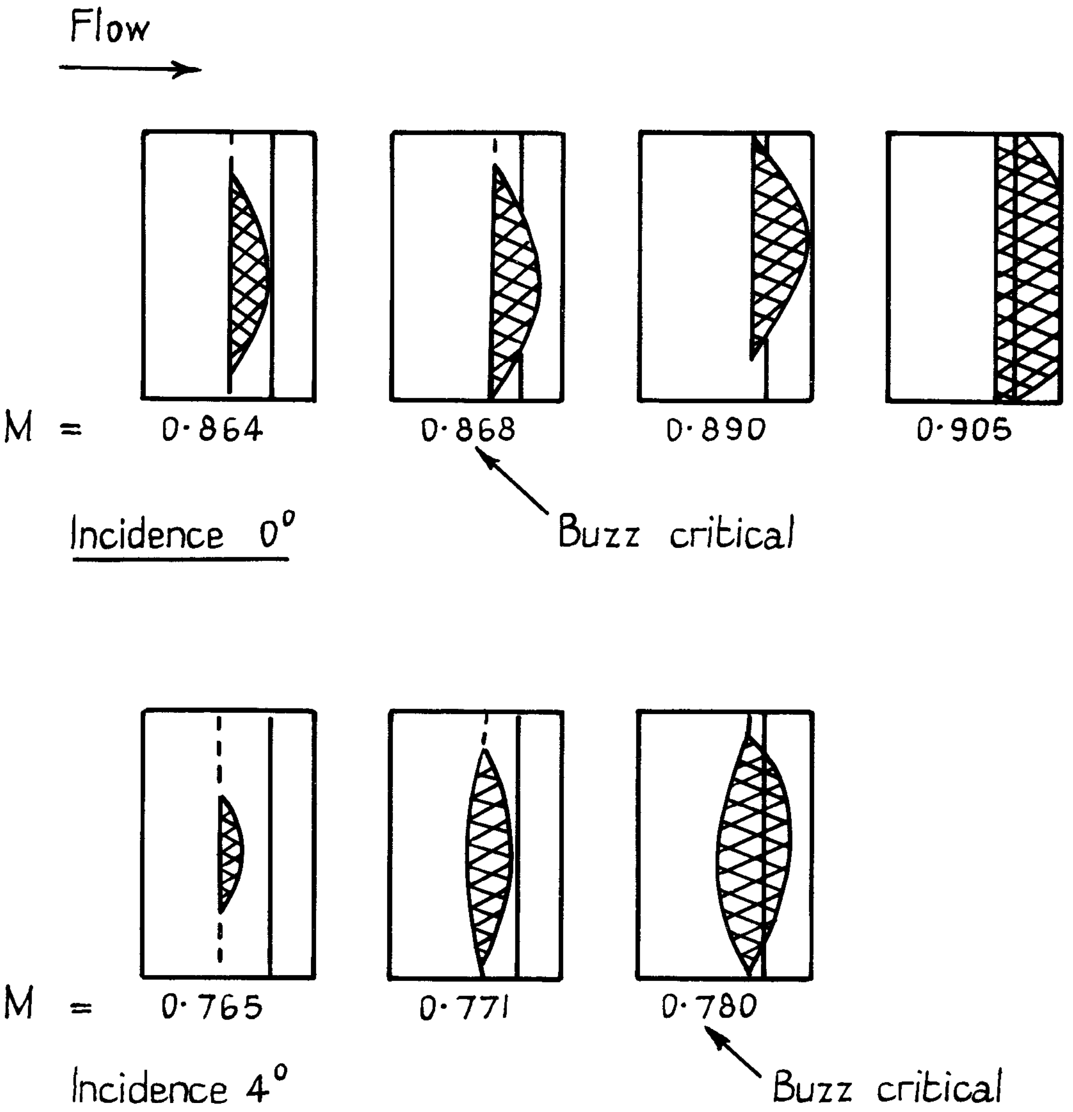
(b) Incidence 2° M=0·835



(c) Incidence 4° M=0·780

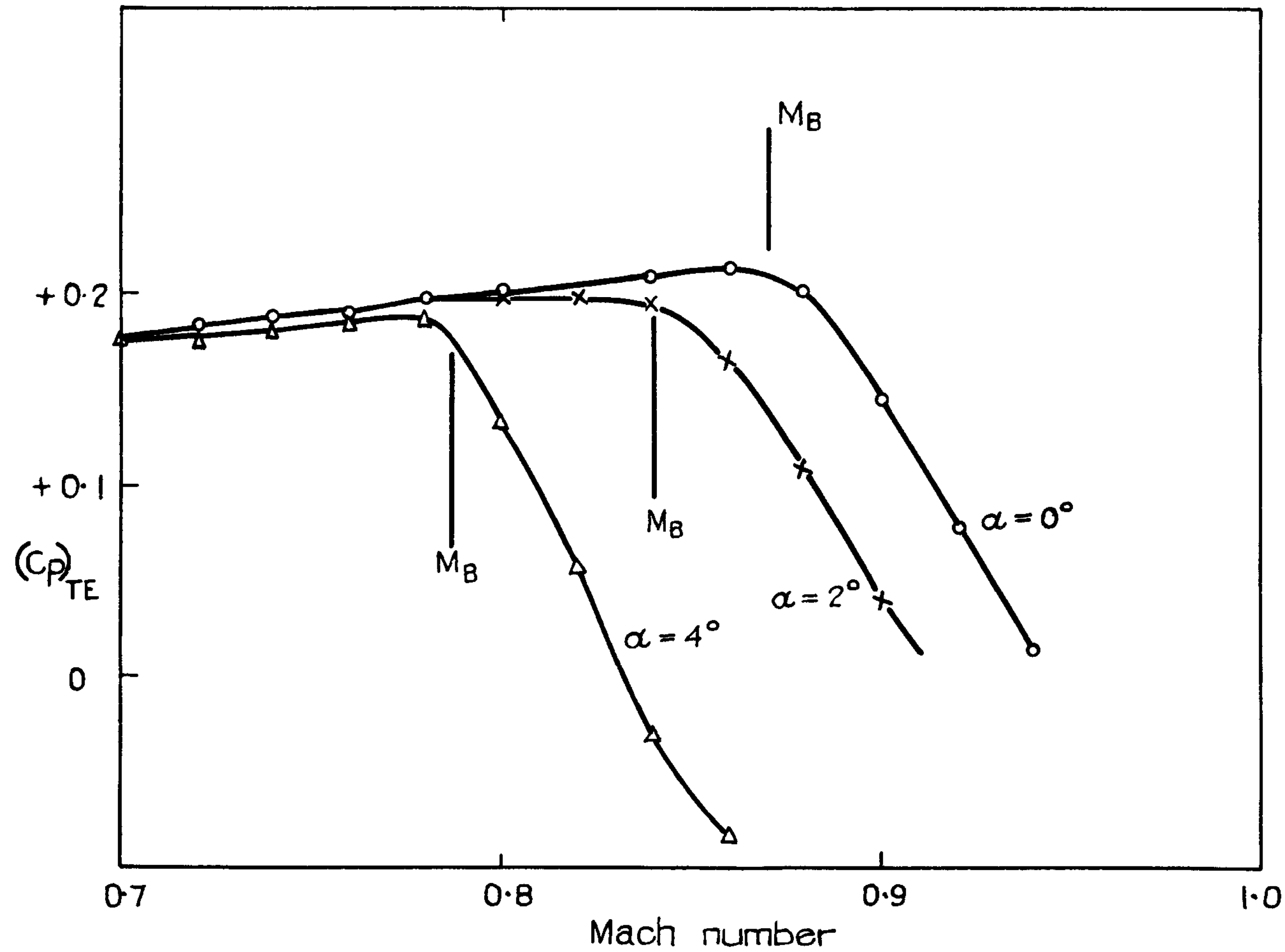
The flow just below the critical Mach number for buzz. Direct-shadow technique. (An apparent distortion of the aerofoil surface is inherent in the method of visualization.)

FIG. 16.



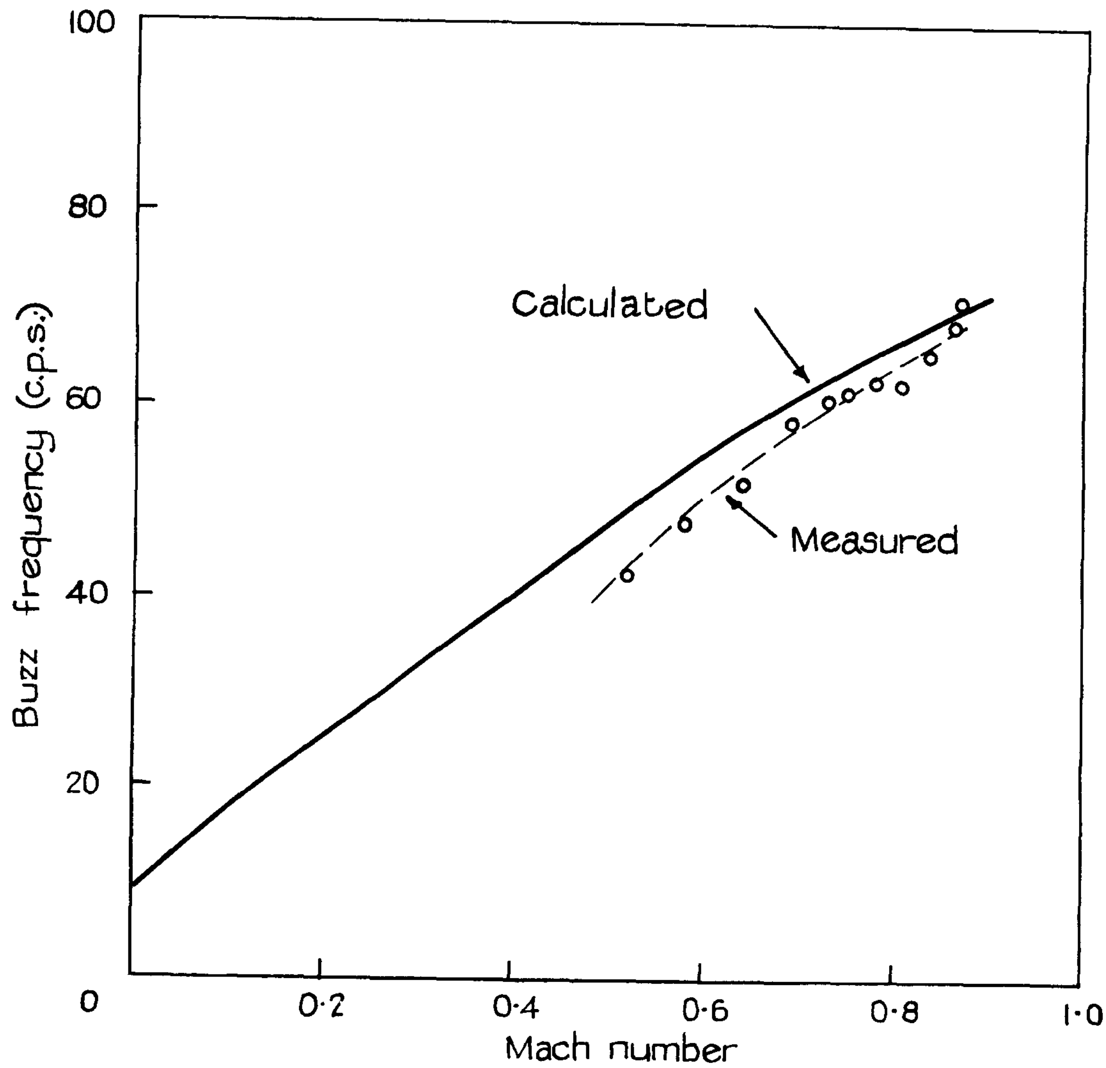
Development with Mach number of reversed flow regions on upper surface of plain aerofoil and correlation with buzz onset for freely hinged flap.

FIG. 17.



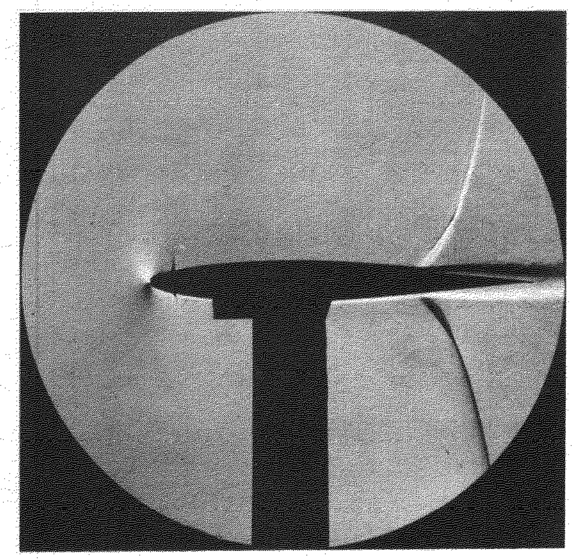
Variation of trailing-edge pressure coefficient with Mach number for plain aerofoil and correlation with buzz critical Mach number M_B

FIG. 18.

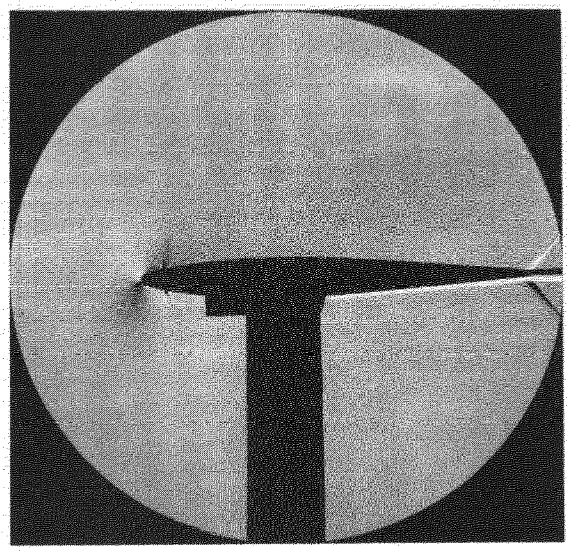


Comparison between measured and calculated buzz frequencies.
(Constant stagnation pressure = 1 atm.)

FIG. 19.



(a) $M = 0{\cdot}925$

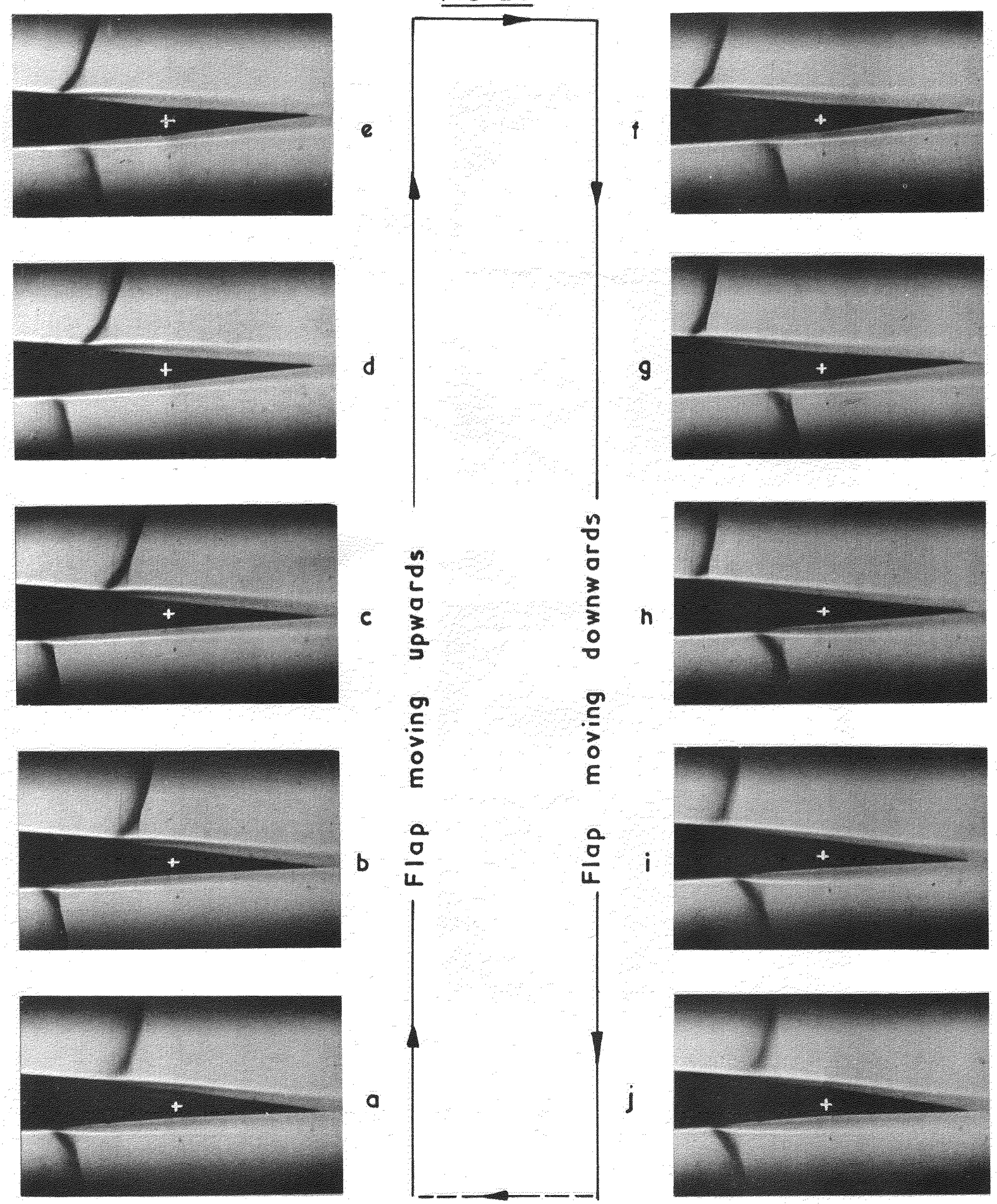


(b) $M \doteq 1{\cdot}0$

Flows corresponding to stable conditions of the flap

Incidence 0°

FIG 20

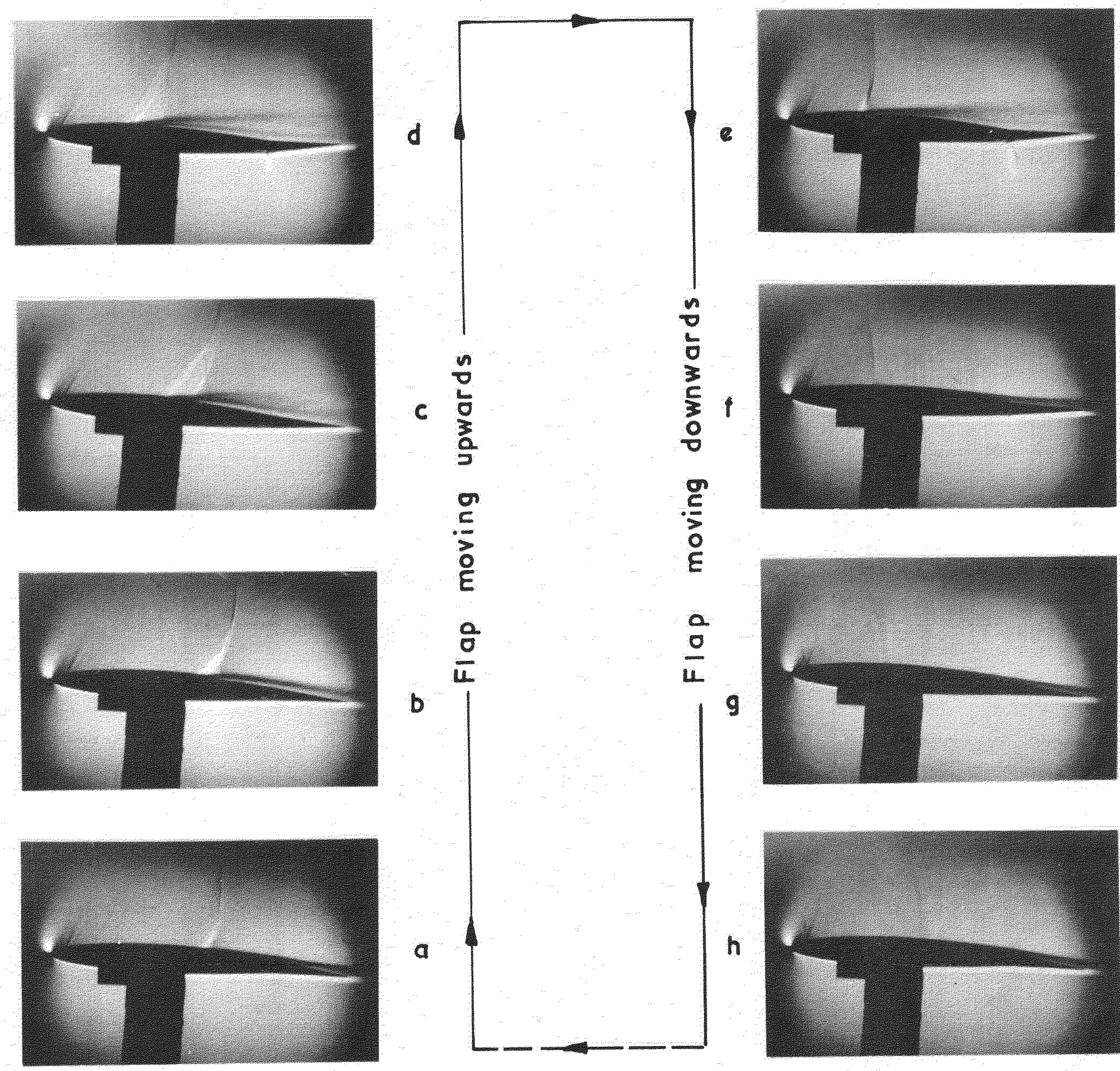


Alternate frames from a film sequence showing the flow over the rear of aerofoil during a cycle of the buzz oscillation.

Incidence 0°. Film speed 1500 frames per second (approx.)

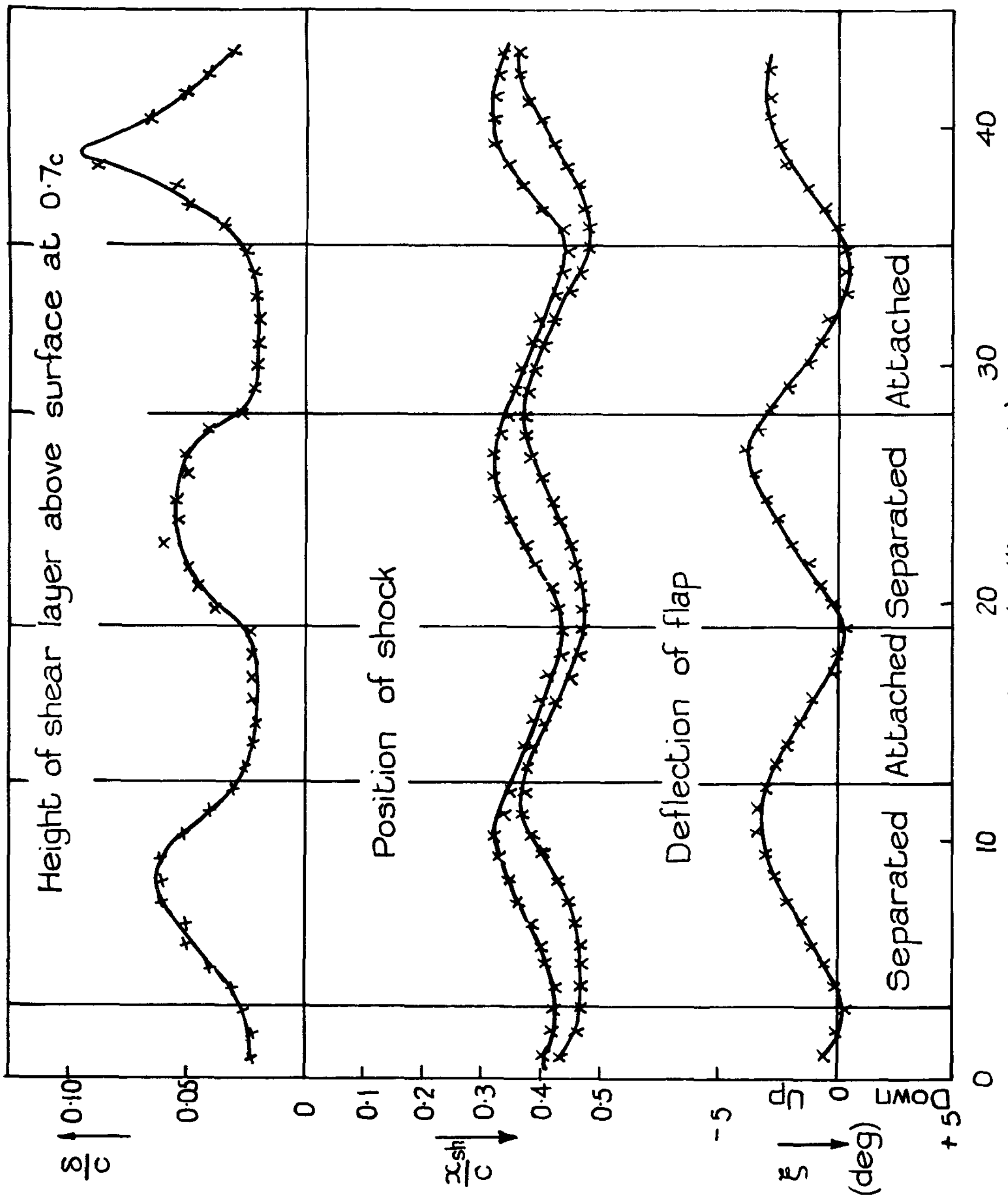
The position of the flap hinge is shown by a cross.

FIG. 21



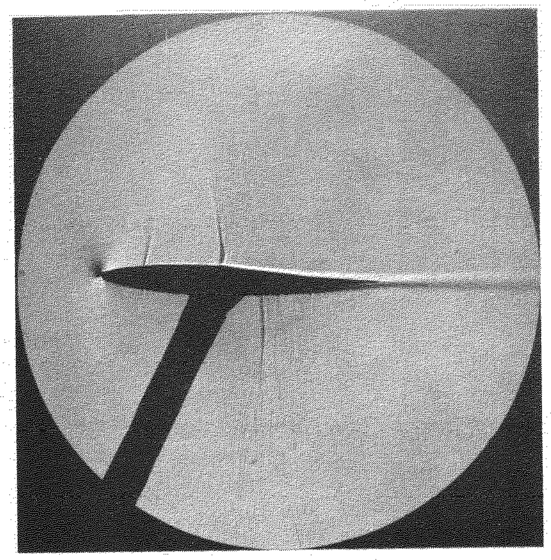
Alternate frames from a film sequence showing the flow during a cycle of the buzz oscillation. Incidence 4°. Film speed 1000 frames per second. (approx.)

FIG. 22

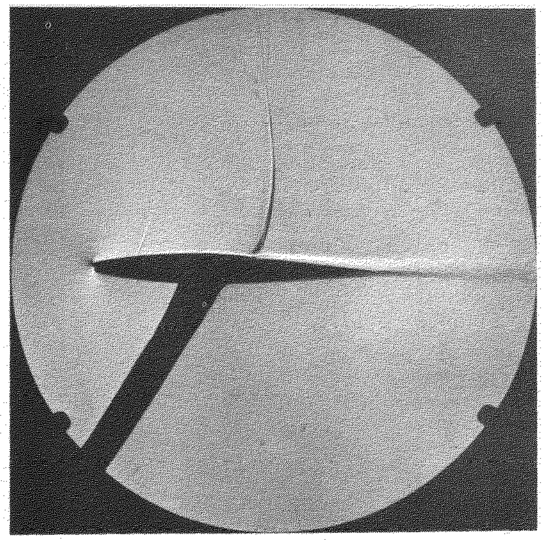


Analysis of buzz oscillation. Incidence 4°, M = 0·780.

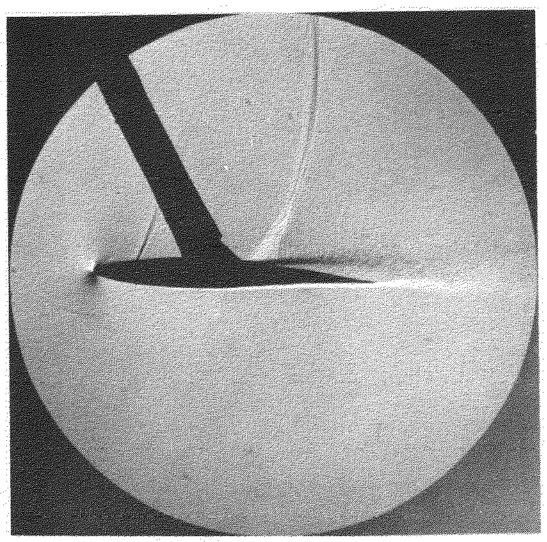
FIG 23



(a) Flap deflected 4° upwards



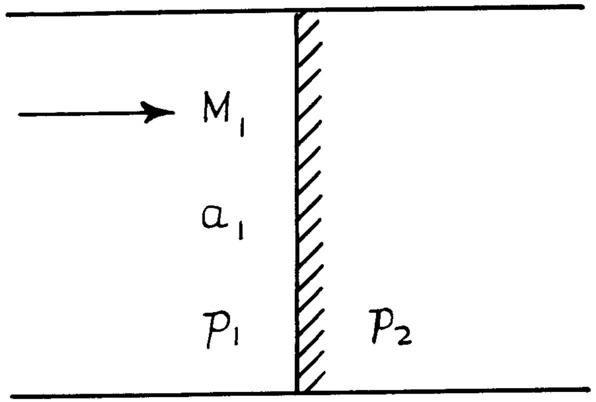
(b) Zero flap deflection



(c) Flap deflected 4° downwards

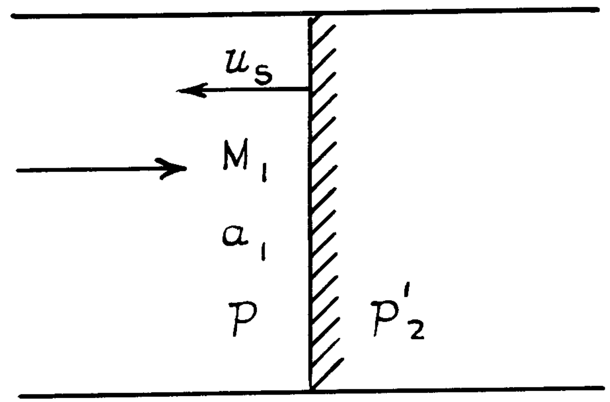
The flow over an aerofoil with statically deflected flap at a Mach number for which buzz would occur. Incidence 2°.

Stationary shock



$$\frac{p_2}{p_1} = \frac{2\gamma}{\gamma+1} M_1^2 - \frac{\gamma-1}{\gamma+1}$$

Shock moving upstream



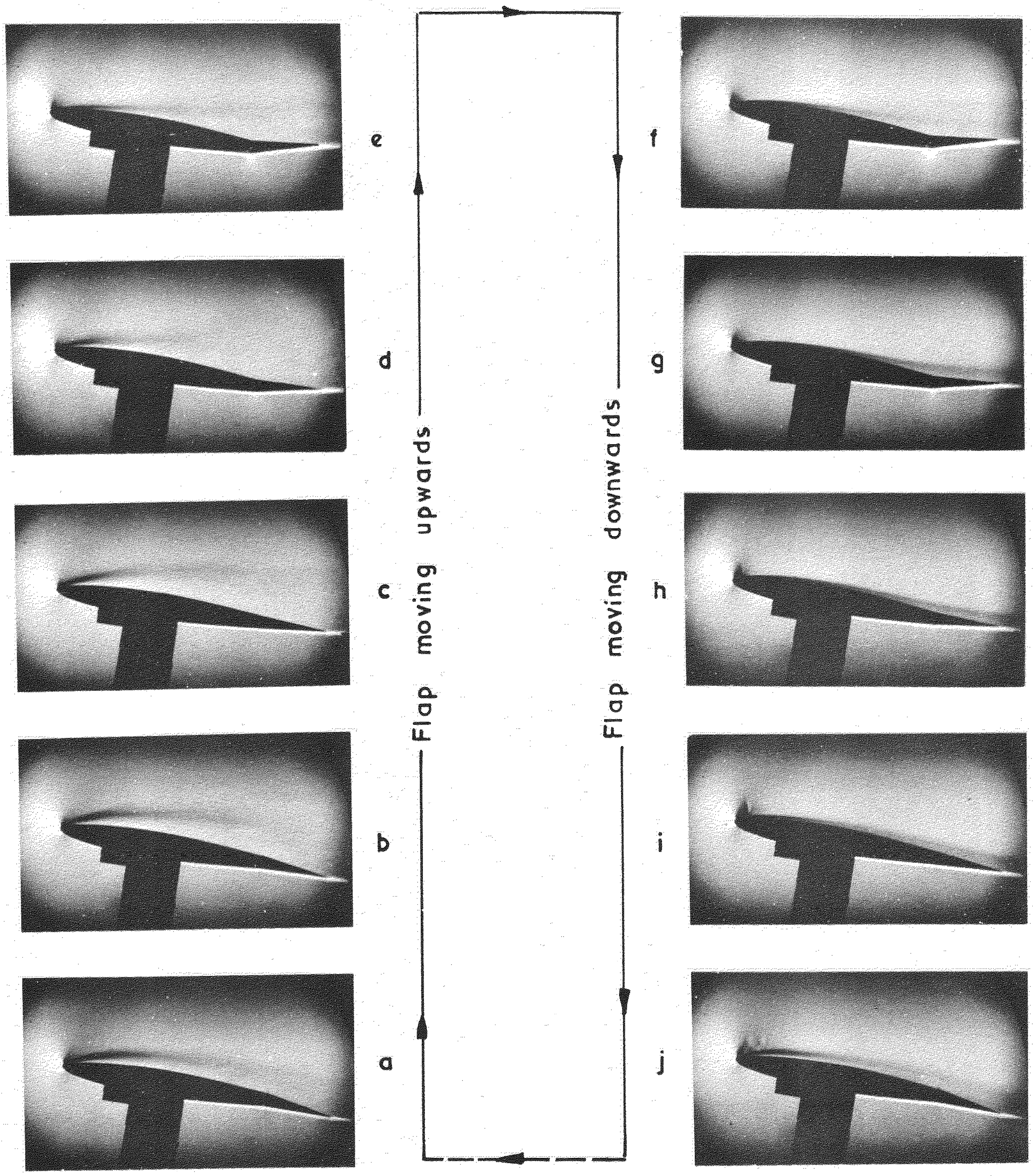
$$\frac{p_2'}{p_1} = \frac{2\gamma}{\gamma+1}\left(M_1 + \frac{u_s}{a_1}\right)^2 - \frac{\gamma-1}{\gamma+1}$$

FIG. 24.

Change in shock strength $= \left(\frac{p_2'}{p_1}\right) - \left(\frac{p_2}{p_1}\right) \simeq \frac{4\gamma}{\gamma+1} \cdot M_1 \cdot \frac{u_s}{a_1}$, when $\frac{u_s}{a_1} \ll M_1$

The effect of shock movement on the shock strength

FIG 25



Film sequence for a flap oscillation at 10° incidence $M \doteq 0.5$
Separation from near the leading-edge.

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