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Methods of Determination and of Fixing Boundary Layer Transition on Wind Tunnel Models at Supersonic Speeds

By

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Methods of determination and of fixing boundary layer transition on wind tunnel models at supersonic speeds

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SUMMARY

An account is given of methods used in supersonic wind tunnels for observation of boundary layer phenomena, in particular of the sublimation and oil film techniques. Examples are given of the uses of these techniques.

On the fixing of transition a rough guide is given for the minimum size of wire required, with an example of the use of wires.

The results of a brief experiment on the profile of an artificially promoted turbulent boundary layer are given.

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1 Introduction

Supersonic wind tunnels are limited in size and stagnation pressure by power requirements. Consequently many model tests are made in wind tunnels at Reynolds numbers well below those of the full scale aircraft or missile. At the full scale Reynolds numbers it is likely that the boundary layers will be turbulent, whereas tests may be conducted on models with laminar boundary layers. Spurious results particularly in relation to shock induced boundary layer separation may thus be obtained. It is common in such cases to create transition by some device and so obtain a turbulent boundary layer.

In a companion paper[1] the whole question of scale effects at subsonic and transonic speeds has been discussed. In this paper, however, the main emphasis is on the techniques of observing boundary layer flow and of fixing transition, though some discussion is included on the effects of fixing transition on the aerodynamic forces on a particular model. A brief comparison is made of the flow in naturally, and artificially promoted, turbulent boundary layers.

2 Methods of studying boundary layer conditions

2.1 General

Some of the various methods which have been used in the study of boundary layer conditions, that is whether the layer is laminar, turbulent or has separated are as follows:-

(a) Optical[2,3].

(b) Acoustic.

(c) Temperature recovery.

(d) Surface pitot tube.

(e) Evaporation and sublimation techniques[4,5].

(f) Non-volatile liquid film techniques.

Of these, the last two are favoured most in general wind tunnel model testing because of their simplicity and flexibility. Their use is described in detail. If only two-dimensional models or bodies of revolution are considered optical techniques may be superior since no interference with the model is required. A brief comparison is made between the surface pitot and sublimation methods.

2.2 Sublimation technique

The evaporation technique has now been almost completely discarded in model work, in favour of sublimation methods. The latter methods are more straightforward in use and give more clearly defined and more easily photographed results. The method briefly is to spray a solution of a suitable solid on a model. Indication of the state of the boundary layer is then shown by the different rates of sublimation in different flow régimes. The requirements of the solids used are that they shall have the requisite vapour pressure, be reasonably visible and also, incidentally, be not toxic. A list of such solids is given in Ref.4. In continuous supersonic tunnels a slow indicator such as azobenzene is used, and a rapid indicator such as hexachlorethane is used in intermittent supersonic wind tunnels. A medium indicator like acenaphthene would be used in subsonic and transonic tunnels.

2.21 Use of azobenzene in the N.A.E. 3 ft x 3 ft Supersonic Wind Tunnel

Considerable use has been made of azobenzene in the N.A.E. 3 ft x 3 ft Supersonic Wind Tunnel, and the methods adopted are described in some detail.

In order to prevent excessive loads on the model whilst the tunnel shock passes it, the tunnel is started and stopped at reduced stagnation pressure, and with the model at zero incidence. During starting, the incidence is set to the required value as soon as the tunnel shock passes the model, and the stagnation pressure is then increased. The time taken to reach normal test conditions is about 15 minutes. Most of this time is taken in running up to speed at reduced stagnation pressure.

Five minutes are needed for the reverse procedure in stopping the tunnel. The indicator pattern develops in five to ten minutes at the usual stagnation temperature of 25°C*. The starting and stopping of the tunnel at reduced pressure has little effect on the pattern.

Before applying the indicator all surfaces are thoroughly degreased. Azobenzene as a 10% solution in 100/120 petroleum ether, is then sprayed to form a thin uniform film on the model surface. The model is sprayed from a distance of 12 to 18 inches using a standard half-pint spray gun with a very slow rate of feed. This produces a thin crystalline coat adhering firmly to the surface. Seven or eight successive applications are required to build up a film of the right thickness. If the gun is held too close to the surface, or if the feed rate is too fast, the azobenzene will still be in solution as it reaches the surface. When this occurs a very coarse patch of azobenzene will form as the petroleum ether evaporates. This type of deposit does not adhere to the surface and it is advisable to clean it off entirely with a solvent and respray. After spraying, the model is lightly rubbed down with cotton-wool to remove any crystals large enough to cause premature transition.

Fig. 1 is a typical photograph of the pattern produced. It is of the upper surface of an unswept wing at 6° incidence, at a Mach number of 1.61. Over most of the wing the flow is laminar. A band of turbulence spreads from the root leading edge across the wing. At the tips disturbances arising from the vortices are shown. In the middle of the span the increase in density of the deposit from leading edge to trailing edge indicates the decrease in surface shear of the laminar boundary layer.

2.22 Use of hexachlorethane in Intermittent Supersonic Wind Tunnels

The short running time of intermittent tunnels demands an indicator of high vapour pressure. Hexachlorethane is the one most commonly used. This is applied in the same manner as azobenzene except that the solvent used is acetone. It should be pointed out that hexachlorethane is slightly toxic and should not be sprayed in confined spaces.

The technique is a little more difficult than with azobenzene as the hexachlorethane sublimes fairly rapidly in free air (a coating will disappear in five minutes or so). The tunnel must be run immediately after spraying if a reliable indication is to be obtained. Results very similar to Fig. 1 can be obtained in running times of about ten seconds.

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* In tunnels in which the stagnation temperature can be controlled over a wide range, the rate of sublimation can be varied, and by reducing the temperature sufficiently it is possible to 'freeze' the indicator pattern.

2.3 Non-volatile liquid film techniques

The use of this technique is considered to be complimentary to the sublimation method. The general principle is that the liquid film moves in the direction of the surface shear on a model. It is thus of most value in showing the presence of any separation regions, which are indicated by an accumulation of oil. It can also be used to indicate the surface streamlines (which may differ appreciably from the streamlines outside the boundary layer due to secondary flow effects). Generally a thin film of oil is smeared uniformly over the model but if surface streamlines are required it is better to apply the oil with a brush.

Transition can also be seen with oil. In much the same way as with an evaporation or sublimation method, the surface shear in a turbulent region may be sufficient to remove the oil there, whilst it remains in a laminar region. Occasional globules of oil may break away from the laminar region and give rise to streamwise streaks in the turbulent region. Alternatively, during the time before the oil is swept away from a turbulent region, it is sometimes possible to see a difference in the wave pattern on the surface of the oil there, compared with a laminar region. The wavelength in the laminar region is larger than that in the turbulent region.

In the use of this technique the practice has tended to differ slightly in continuous and intermittent wind tunnels, a heavier oil being favoured in the continuous tunnel.

2.31 Use of oil in the N.A.E. 3 ft x 3 ft Supersonic Wind Tunnel

A heavy oil (Shell Nassa 87 of viscosity Redwood 1 1050 at 140°F) is used. White titanium oxide is mixed with the oil to simplify observation, and a small amount of oleic acid is added as an anti-coagulant. The oil is sufficiently viscous to be unaffected by the tunnel stopping and starting procedure.

Fig. 2 shows the indication obtained with oil on the same wing as Fig. 1 under the same conditions. The pattern is very similar to Fig. 1 but a separation at the trailing edge is now apparent. The spanwise front of oil at about 30% chord is thought not to be significant. It can be explained by the fact that the surface shear is high near the leading edge and diminishes chordwise. Consequently the oil near the leading edge is blown back faster than that downstream and an accumulation results at some point.

A different type of pattern appears in Fig. 3. It shows the very complex flow on a sweptback wing at zero incidence and Mach number of 1.61.

In Fig. 4 the streamlines produced on a body near a wing root (cropped delta wing at 4° incidence and Mach number 1.61) are shown. The oil in this case was applied by a brush.

2.32 Use of oil in Intermittent Supersonic Wind Tunnels

Much lighter oils (Redwood 1 about 200) than in the continuous tunnel are used, and the type of boundary layer flow is indicated more by the type of wave pattern on the surface of the oil. A separation is shown in the same way by an accumulation of oil or by an upstream movement. The pattern must be observed whilst the tunnel is running, and transient phenomena are shown up as variations in the pattern. Stopping the tunnel largely destroys the indication.

A typical photograph of the flow over a pair of wings, on one of which transition has been produced by a wire is given in Fig.5. The photograph is of the pressure surface of a double wedge aerofoil at an incidence of 5° at a Mach number of 2.5. On the wing with laminar flow a wave pattern is visible. On the other wing the oil has tended to form streaks.

2.4 Comparison of transition determination by surface pitot tube and azobenzene

Fig.6 gives a comparison between the transition position on a 15° cone at a free stream Mach number of 3.25 as indicated by azobenzene and by surface pitot. The results of two tests are shown, one with the cone set at zero incidence and the other with 1° incidence. Both figures show that the transition boundary given by azobenzene is the end of the laminar region, i.e. the beginning of the transition region. The lower figure shows the asymmetry in transition front which occurs for 1° mis-alignment. This points to the care which is necessary in setting up bodies of revolution on which boundary layer measurements are to be made, and also to the uncertainty involved if the transition position is determined on one generator only.

2.5 Flow on a sweptback wing

As an example of the useful information which can be obtained simply by flow observation, Figs.7 and 8 are presented. They show the flow at zero incidence on a sweptback wing at a Mach number of 1.61 and two Reynolds numbers. In Fig.7 transition is shown to occur further forward over the inboard half of the span than at the tip. The extent of the forward movement increases slightly at the higher Reynolds number.

This forward movement is associated with streamwise vortices, which can be seen more clearly in Fig.8, an enlargement of part of the lower photograph of Fig.7.

3 Methods of Fixing Boundary Layer Transition

3.1 General

The various methods which have been used for fixing transition are discussed briefly below. An analysis has been made of experiments using wires and this has led to a guide for the size of wire required. This analysis is given fully.

(a) Discrete disturbances

(i) Single wires are discussed in Section 3.2.

(ii) Multiple ridges or grooves.

In general these are to be preferred to single wires since their size can be smaller and consequently the disturbances produced in the main stream are not so severe. Ridges may consist of a series of wires stuck to the surface or of strips of plastic material formed between pieces of adhesive tape which are subsequently removed. This method is used in preference to strips of tape because it is possible to produce ridges with sharp edges. Such ridges are more effective for a given height.

(b) Distributed roughness

This is usually applied by dusting carborundum powder on a coating of adhesive. The carborundum does, however, tend to be eroded away in time. At supersonic speeds it is found that the roughness height must be increased over that estimated by low speed methods.

(c) Air injection[6]

This is probably the most elegant way of producing transition since the size of the disturbance can be varied by controlling the air flow. It is, however, of limited application to general models on account of the pipework involved.

(d) External means

This covers methods in which the tunnel stream turbulence is augmented. A single wire ahead of a model has been used both subsonically and supersonically. It is not very successful supersonically as large disturbances are propagated into the main stream whilst only a narrow wake is produced.

3.2 A guide to the size of transition wires

3.21 Method of correlation

Some investigations have been made at low speeds to find the minimum size of wire needed to fix transition[7,8], but at high speed the selection of a suitable wire size has been a matter of trial and error.

Some systematic tests on transition wires have been made at high subsonic and supersonic speeds by Gibbings at N.A.E. The results are correlated here with published results of wing tests using transition wires[9-16], and other unpublished N.A.E. results, to provide a guide to the minimum size of wire needed to fix transition at Mach numbers up to 3.0.

An approximate criterion for determining the minimum size of wire needed to fix transition in low speed flow has been given by Bryant and Batson[8]. This criterion is:

$$R_D = 600$$

where R_D is the Reynolds number based on the wire diameter, and free stream flow conditions. More recently Tani and Hama[17] have published low speed data on the effect of wires on transition on a flat plate that confirm broadly this method of correlation. These results, covering a range of wire sizes, wire locations and free stream conditions, have been analysed to find approximately the critical values of R_D at which transition first occurs at the wire. The critical values vary from 720-860.

A correlation of the results of Refs.8 - 17 and the unpublished results of Gibbings and of other N.A.E. tests is made in Fig.9 by plotting R_D against Mach number. All results are plotted irrespective of the type of test, and of wire location, which varied from 4% chord to 55% chord. Details of the test data used for correlation are given in Table I. Filled symbols are used where transition occured at the wire, and unfilled symbols where transition was downstream of the wire. In a few cases the critical value of R_D has been determined and is denoted by a half filled symbol.

TABLE I

Details of Test Data used for Correlation

Author	Type of Test	Mach number	Reynolds number	Wire position	Remarks
Bryant and Batson (Ref.8)	Two-dimensional wing (t/c = 15%)	Low speed	0.93×10^6 (chord)	0.1c - 0.53c	
Gamble (Ref.9)	25^o swept wing (t/c = 12%)	0.49, 0.82	$0.8\text{-}3.5 \times 10^6$ (chord)	0.08c - 0.37c	Transition position by sublimation method
Collingbourne and Pindar (Ref.10)	40^o swept tapered wing, (t/c = 10%)	0.4 - 0.93	1.75×10^6 (Mean chord)	0.12c - 0.28c	
Richards and Burge (Ref.11)	Two-dimensional wing with suction slots. t/c = 16.3%	Low speed	0.37×10^6 (chord)	0.55c	
Pearcey (Ref.12)	Two-dimensional wing (t/c = 18%)	0.35 - 0.8	1×10^6 - 1.8×10^6 (chord)	0.05c - 0.3c	
Ackeret, Feldmann and Rott (Ref.13)	Curved plate in stream of varying Mach number	About 0.9 at wire	1.7×10^6 (Plate length)	$0.02\,\ell$	
Cumming, Gregory and Walker (Ref.14)	Two-dimensional wing (t/c = 13%)	Low speed M = 0.2	5.7×10^6 (chord)	0.05 c	
Chapman, Wimbrow and Kester (Ref.15)	Two-dimensional wing thickness varied	1.5, 2.0 3.1	Varied up to 3.5×10^6 (chord)	0.25 c	Transition position by China clay technique
Harrin (Ref.16)	Short span aerofoil built round aircraft wing	0.71	1.7×10^6 (Mean chord)	0.04 c	
Tani and Hama (Ref.17)	Flat plate	Low speed	Varied	Varied	Only critical values of R_D shown in Fig.1
Gibbings (Unpublished)	60^o Delta wing (t/c = 4%)	0.8, 1.6 2.0	Varied	0.05c, 0.10c	Transition position by oil flow method.

3.22 The critical value of R_D

The points in Fig.9 lie either side of a fairly well defined critical curve depending on whether or not transition is fixed at the wire. It follows that R_D can be used as an approximate criterion to determine the minimum size of wire that will fix transition, independent of the wire position and the wing geometry. The critical curve is not expected to apply when the wire is located in a region of severe pressure gradient. At subsonic speeds, up to a Mach number of about 0.9, the critical value of R_D is roughly constant at 700. At supersonic speeds the critical value increases exponentially with M. There is no data available in the transonic range, but a rise in the critical value of R_D starting at about M = 0.9 is probable.

This analysis is concerned only with the size of wire needed to induce transition at the wire. It is possible that if the ratio of wire diameter to local chord is too large, undesirable separations will occur at the wire.

3.3 The effects of fixing boundary layer transition on flow separation on a double wedge aerofoil at low Reynolds number

As an example of the application of transition fixing devices some results are given of measurements of pitching moment on a pair of wings mounted on a body. It was clear from measurements on the 'clean' wings that marked changes in flow were occurring between 0° and 10° incidence, and a brief investigation of the flow pattern was made.

The effects of boundary layers on the forces on aerofoils at supersonic speeds have been discussed fully by Zienkiwicz[18]. He has shown that large forward shifts of the centre of pressure can be produced by separation occurring at low incidences. The effects diminish as incidence increases and are not very significant at large incidences when the relative contribution to the lift of the suction side is reduced.

The wings were so mounted on the body that their incidence relative to it could be varied, and the loads were measured on the wing panels only. The body diameter was 1.25 inches and each wing panel was of 1.373 inches span and chord. The wing section was symmetrical double wedge of 8% thickness/chord ratio. The hinge line, about which the pitching moment is measured, was at 40% chord.

Curves of pitching moment coefficient (referred to wing net area and chord) with body at zero incidence are plotted against wing incidence in Fig.10 at Mach numbers of 2.0 and 2.48. The corresponding Reynolds numbers based on wing chord are 0.46 and 0.37 millions respectively.

It will be seen that with the 'clean' wings there is a positive change in slope through $\alpha_W = 0$. (The pitching moment is defined as positive nose up so that a positive change of gradient means a forward movement of centre of pressure.) Observations of flow pattern were therefore made using both sublimation and oil techniques. The interesting feature of the flow was a separation at the trailing edge on the suction face and this was shown better by the oil. On the assumption that this separation was associated with a laminar boundary layer, efforts were made to eliminate it by adding roughness near the leading edge. The roughness consisted of a spanwise band of carborundum of height 0.004" to 0.006" and 0.1" wide starting 0.06" from the leading edge. At M = 2 this roughness was sufficient to eliminate the separation*

* It was also successful at M = 1.6. The results at this Mach number have not been included.

This is shown by the change in the pitching moment curve and was confirmed by observation.

At M = 2.48 the same roughness produced little apparent change in the flow pattern and only a small improvement in the pitching moment curve (centre graph of Fig.10). This agrees with Fig.9 which shows that the disturbance size must increase with Mach number. Various other types of disturbance were tried at this Mach number. Of these only wires proved successful. Three wire sizes were taken 0.005, 0.010 and 0.015 inches, the wires being attached at about 10% chord. As will be seen from the lowest graph in Fig.10 there was a progressive straightening of the pitching moment curves as the wire size increased. Correspondingly, flow observation showed a reduction in the separation. The largest wire produces an almost straight pitching moment curve and Fig.11 shows that the separation is almost completely eliminated. The photograph was taken of the suction side at an incidence of 6^{o}; the lower wing only has a transition wire.

The curve of Fig.9 indicates that a wire of 0.011 inches diameter would be required to produce transition and is thus fairly consistent with this experiment.

The transition wire is of significant size in relation to the wing being of over 1% chord diameter. It is clear from the pitching moment curves that these large wires do modify the aerofoil profile. They produce an overall change in slope of the pitching moment curve which corresponds to a forward shift of the centre of pressure of over 1% chord. This shift is in the opposite sense to that produced near zero incidence. The nose shock detaches at a lower incidence and the behaviour of the curves there is modified.

It thus appears that if reliable results on centre of pressure position are required at low incidences, and Mach numbers of the order of 2.5 and above with turbulent boundary layers, the Reynolds number, based on wing chord, should be appreciably higher than the 0.37 millions of this test.

4 Validity of the artificial promotion of turbulent boundary layers

4.1 General

No experiments have been made specifically to check the validity of the artificial promotion of turbulent boundary layers. There is some evidence from work at the N.P.L.[19] that the pressure ratios for separation are roughly the same for artificial turbulent layers at low Reynolds number as for naturally turbulent layers at high Reynolds number. In experiments where boundary layer thickness and skin friction are significant the natural and artificial boundary layers will not of course be equivalent. Coles[20] has shown that the local friction coefficient at a given Mach number depends upon the Reynolds number based on the distance from the transition point (for values of this Reynolds number greater than 2×10^5) irrespective of how transition is produced. This implies that for flows at two different Reynolds numbers the turbulent skin friction can be equivalent at one point only.

In the course of an investigation of turbulent boundary layers on a cone, some brief tests were made on the boundary layer profile with artificial transition. These tests are described. The nature of the tests was incidental to the main work and their adequacy was judged solely by the results.

4.2 The artificially promoted turbulent boundary layer on a 10° cone at M = 2.45

The primary requirement in this investigation was that the turbulent boundary layer should be extended as nearly as possible to the cone tip without separation or undue distortion.

Two methods of introducing the necessary disturbance were considered, a wire 'trip' and a roughness band. The wire was abandoned for practical reasons, and initial tests were made with a band of carborundum powder cemented near the tip. This method produced inconsistent results because some of the powder was eroded away during the course of a test. Special tips for the cone were then made with screw thread type roughness elements as shown in Fig.12. These were machined with a special tool and the thread profile was obtained using a standard screw thread projector. A number of tips were made with variations of depth of roughness, of area and of position of roughness band. However, a detailed investigation of these variations was not made, and only results with the most satisfactory roughness element are given here.

The tests were made in a tunnel of 5 inches square working section, at a free stream Mach number of 2.45, and Reynolds number per inch of 0.24 millions. Fig.12 shows the depth of the roughness band which is of the same order as that of the displacement thickness of the laminar boundary layer on a smooth cone. The Reynolds numbers based on roughness depth d and displacement thickness ∂^x are $Re_d = 1270$ and $Re_{\partial^x} \simeq 2000$.

The position of transition with the roughness band is shown in Fig.13 and compared with that on a smooth cone under the same conditions. The virtual origin of the turbulent boundary layer is moved upstream a distance of about 1.9 inches, the transition Reynolds number being reduced by approximately 40%. The roughness band does not produce immediate transition.

At the time of the tests no great importance was attached to obtaining the highest possible transition Reynolds number on the smooth cone. The cone was made of perspex with a stainless steel tip and a relatively poor joint between the two parts. In subsequent work a 15° cone of stainless steel was used and this gave larger values of transition Reynolds number (see Fig.6). The lack of smoothness on the 10° cone may well have reduced the relative effect of the roughness element.

Boundary layer velocity profiles in terms of $\frac{u}{u_1}$ (the ratio of the velocity parallel to the surface at any point in the boundary layer to that immediately outside the boundary layer), are given in Fig.14. These are plotted against $\frac{y}{\theta}$, where θ is the momentum thickness, to eliminate the effect of the additional momentum loss incurred by the roughness element. For comparison the profile on the cone after free transition is shown. The profiles close to the roughness are not accurate - the comparatively large size of pitot tubes then in use introduced size effects and distortion when measuring in a thin layer.

For distances greater than 4 inches from the tip (2.5 inches downstream of the roughness element) the velocity distributions are characteristic of turbulent boundary layers and show little difference from that for free transition. Turbulent boundary layer profiles were obtained at distances greater than 6.5 inches from the tip of the smooth cone.

It appears from these results that a finite settling length is required downstream of a disturbance before a fully turbulent boundary layer develops. Insufficient velocity profiles are available to determine this length exactly but it is between 1.5 inches and 2.5 inches. The value indicated by the surface pitot is 2.0 inches or, on a basis of Reynolds number, half a million.

Similar results have been obtained at low speeds for example the tests of Klebanoff and Diehl[21] on a flat plate. In these tests several devices were used to promote a turbulent boundary layer. Roughness produced by sandpaper glued to the surface was found to give least distortion of the velocity profile but a settling length was required. Wires attached to the surface were discarded because of the excessive distortion of the velocity profile.

5 Concluding Remarks

The usefulness of simple boundary layer flow observation techniques has been demonstrated with examples.

A rough guide has been given for the minimum size of wire to create transition. It is likely that this rule implies that the transition region starts at the wire and that fully developed turbulent flow does not occur for some distance downstream. Further experiments are required to determine the length of this region. From Fig.13 it appears to depend upon the method of creation of transition. The Reynolds number based on the length of the transition region together with considerations of the size of obstacle required to create transition should lead to a minimum scale for model tests simulating full scale turbulent boundary layers.

This note has not touched on the more difficult problem, which may have to be faced, of how to perform model tests with laminar boundary layers simulating full scale phenomena at higher Reynolds numbers.

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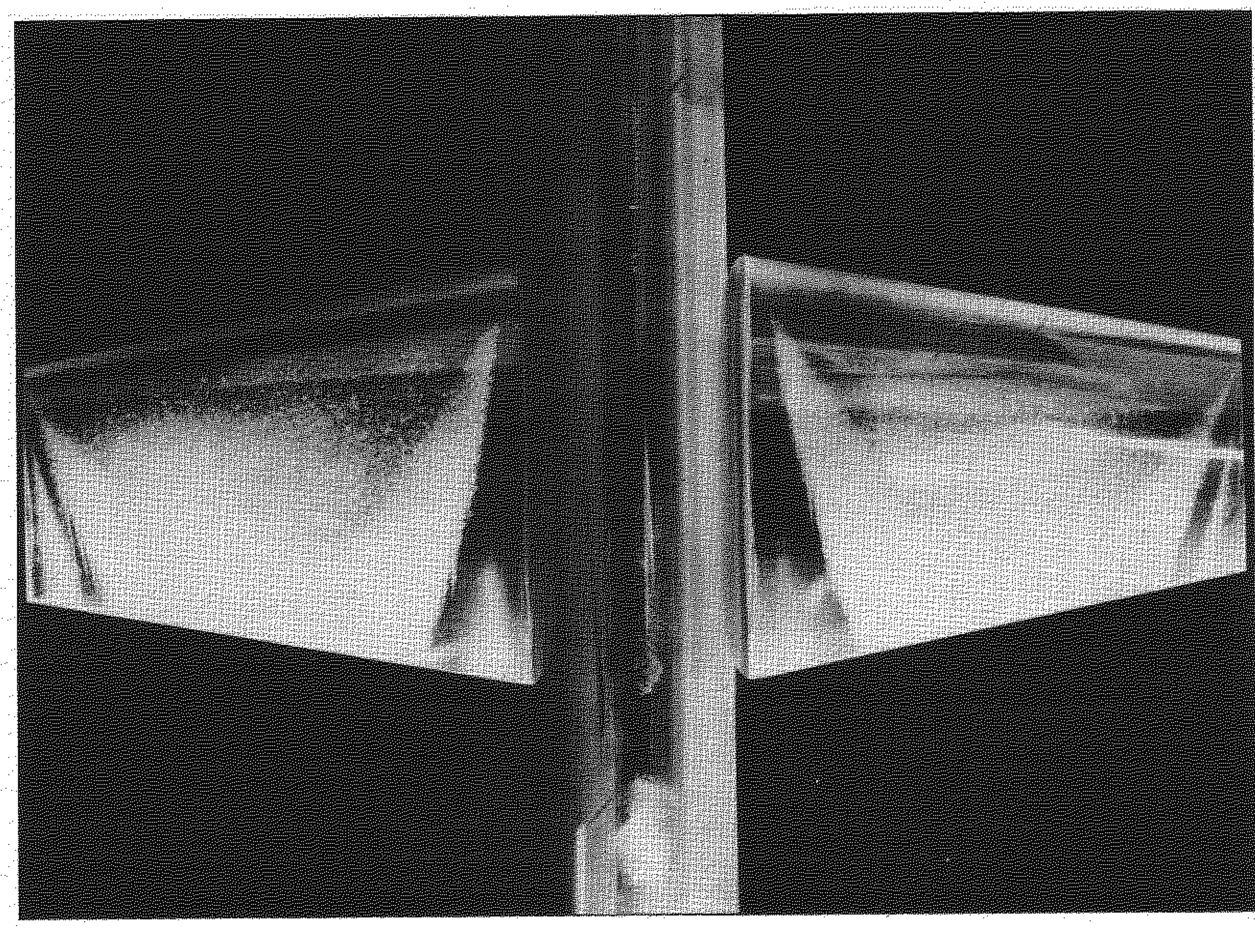


FIG.1. AZOBENZENE PATTERN $\alpha = 6°$ $M = 1.61$ UPPER SURFACE

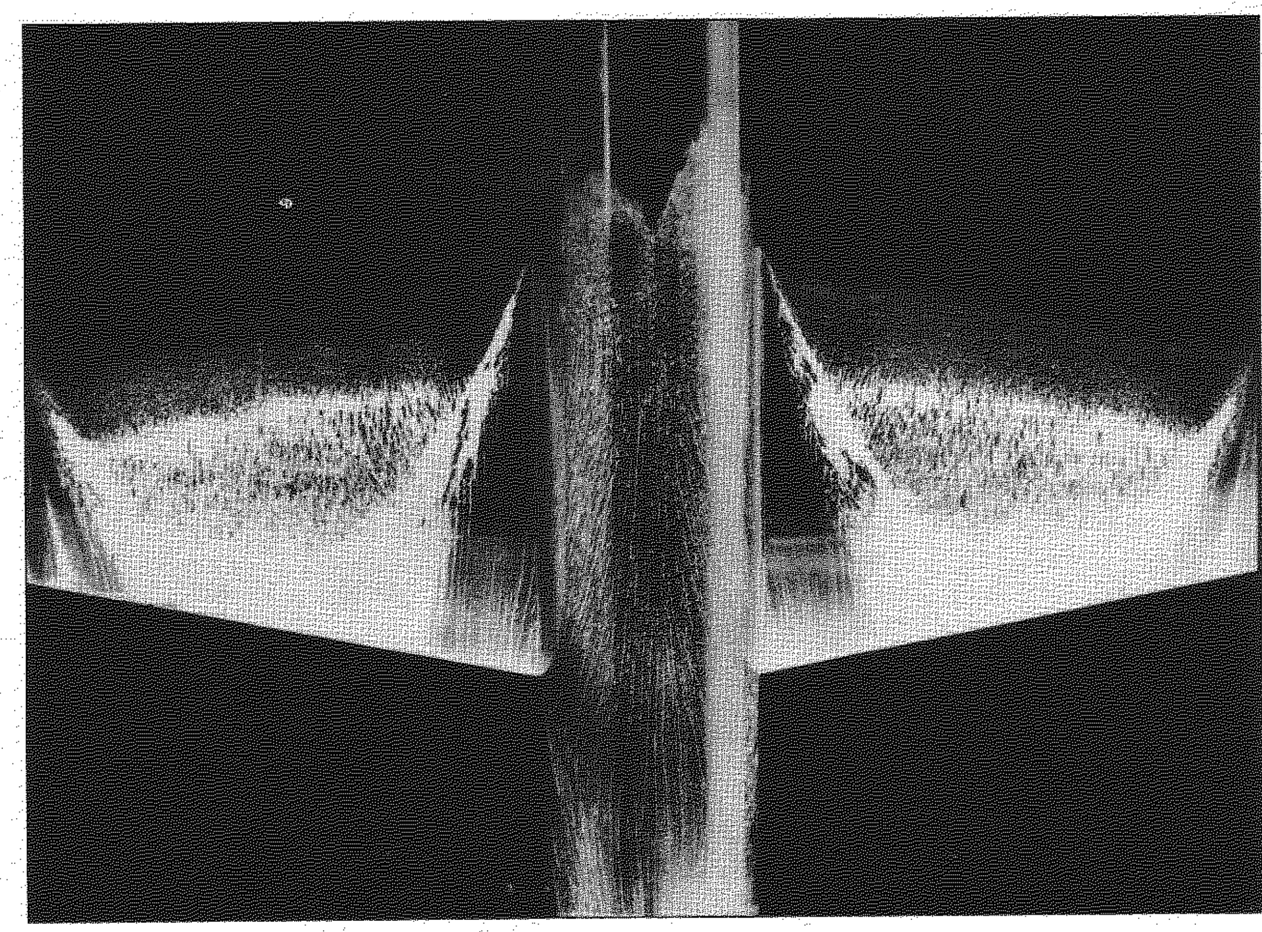


FIG.2. OIL PATTERN $\alpha = 6°$ $M = 1.61$ UPPER SURFACE

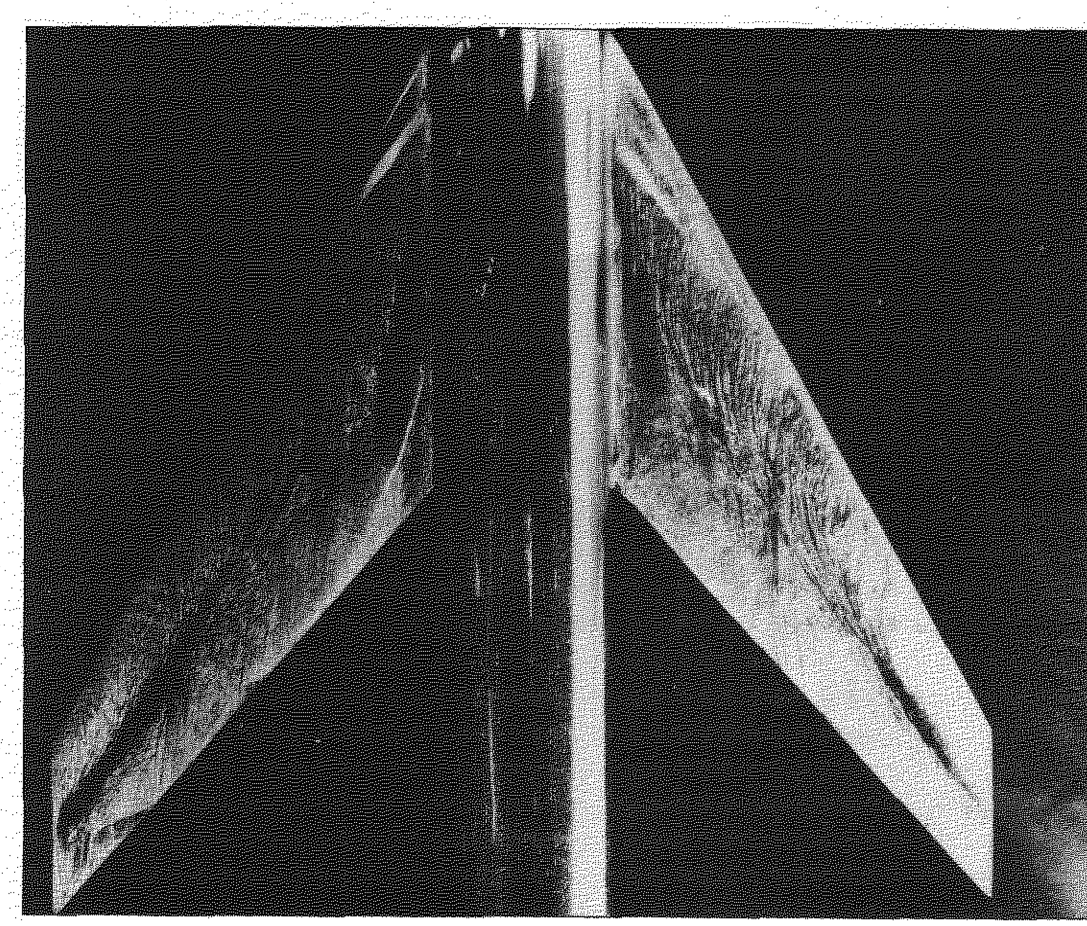


FIG.3. OIL FLOW PATTERN ON A SWEPT WING AT $M = 1.61 \quad \alpha = 0$

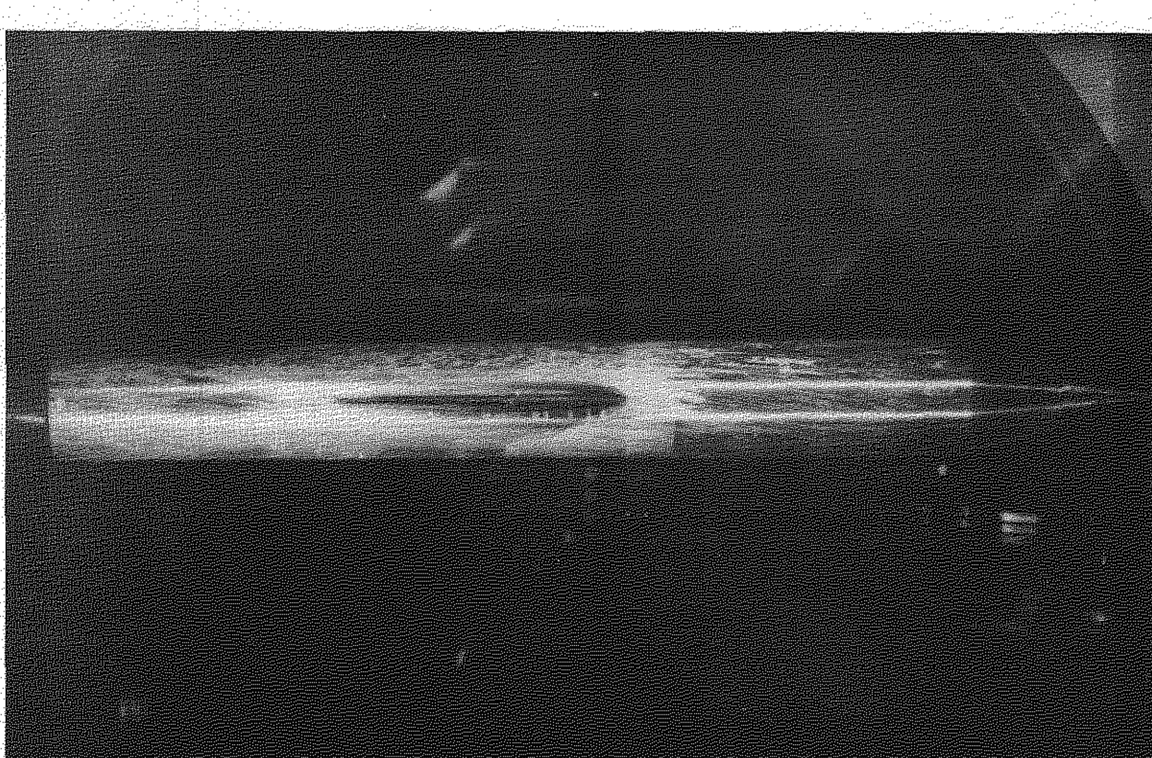


FIG.4. OIL FLOW PATTERN ON BODY ROUND WING ROOT $M = 1.61 \quad \alpha = 4$

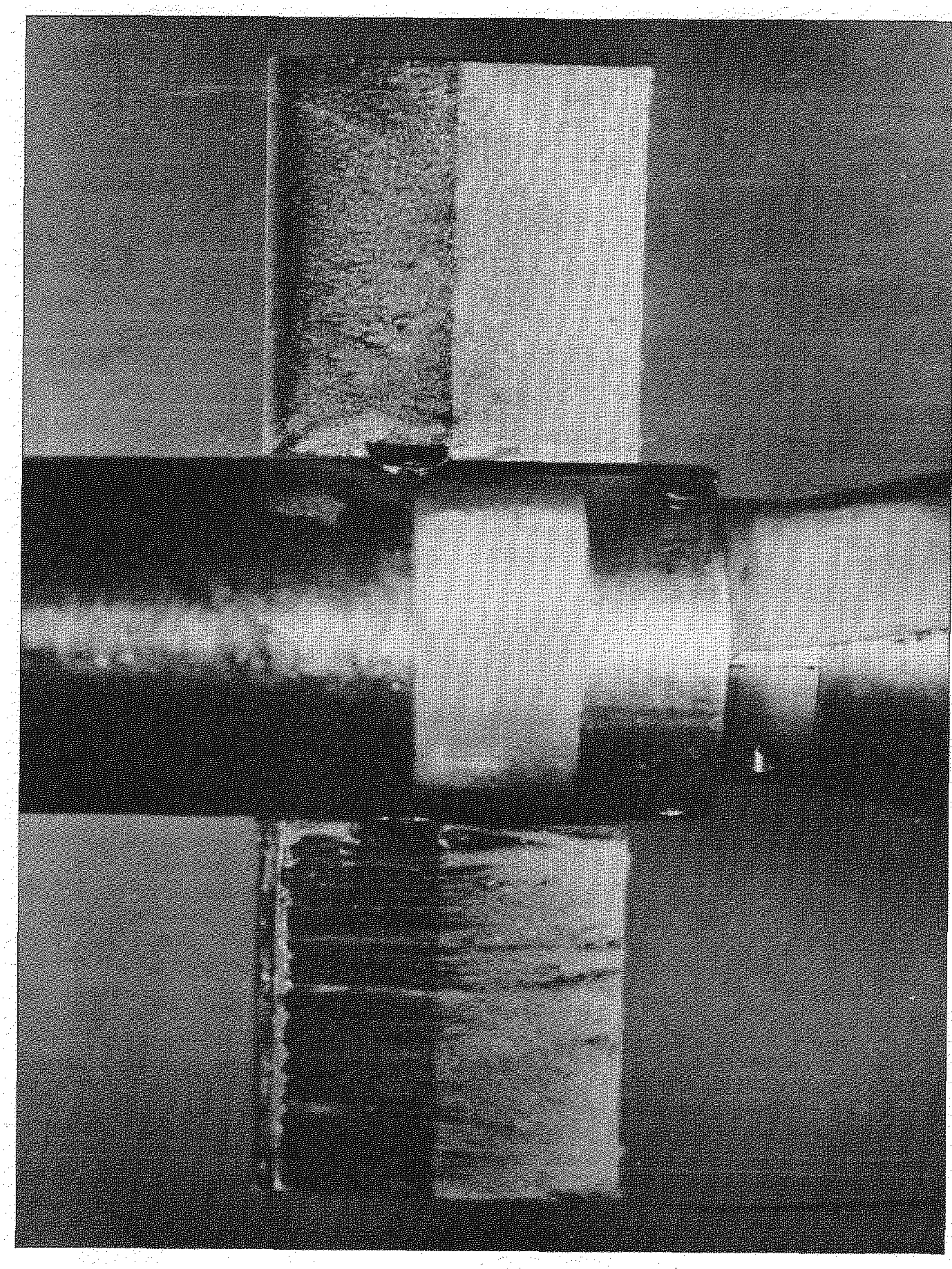
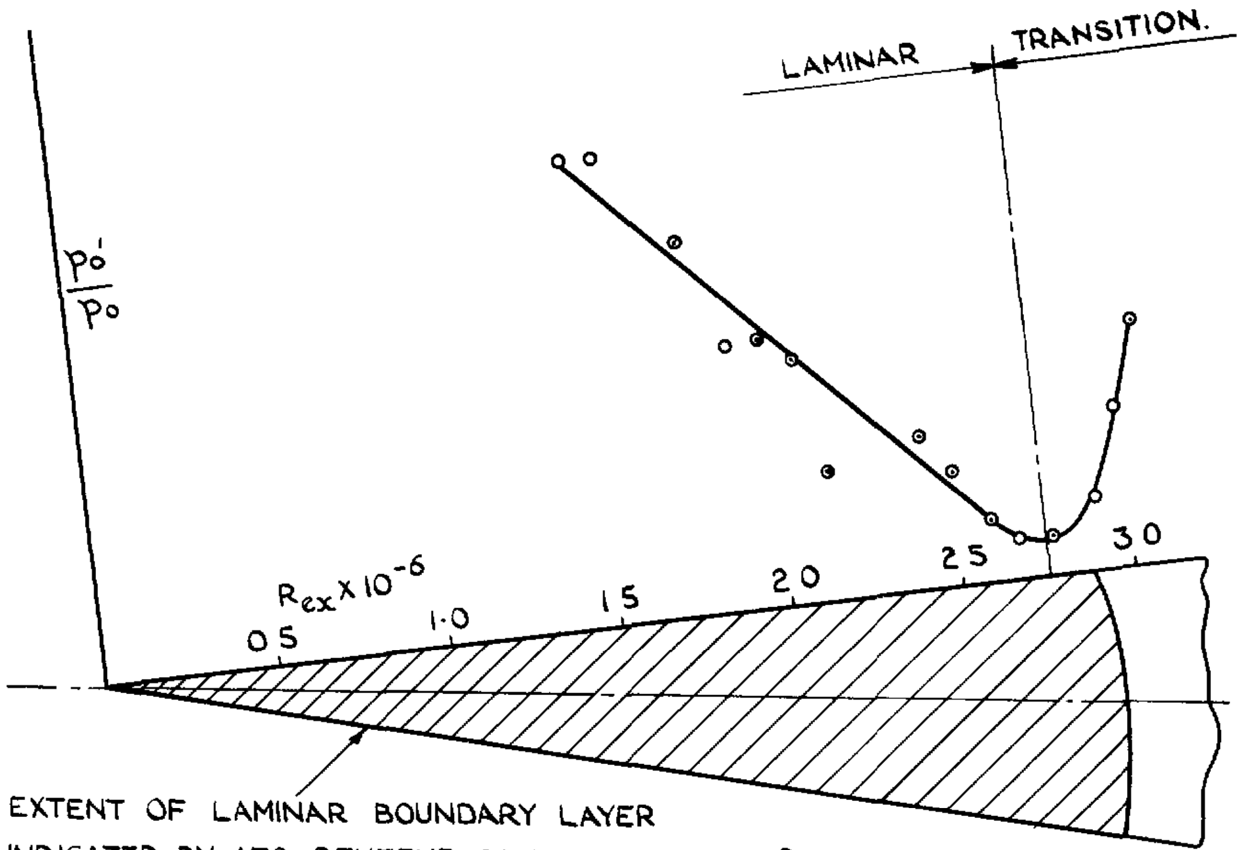


FIG.5. OIL FLOW PATTERN ON DOUBLE WEDGE AEROFOIL $M = 2.5$
UPPER SURFACE $\alpha = 5°$

(LOWER WING WITH TRANSITION WIRE)

FIG. 6.



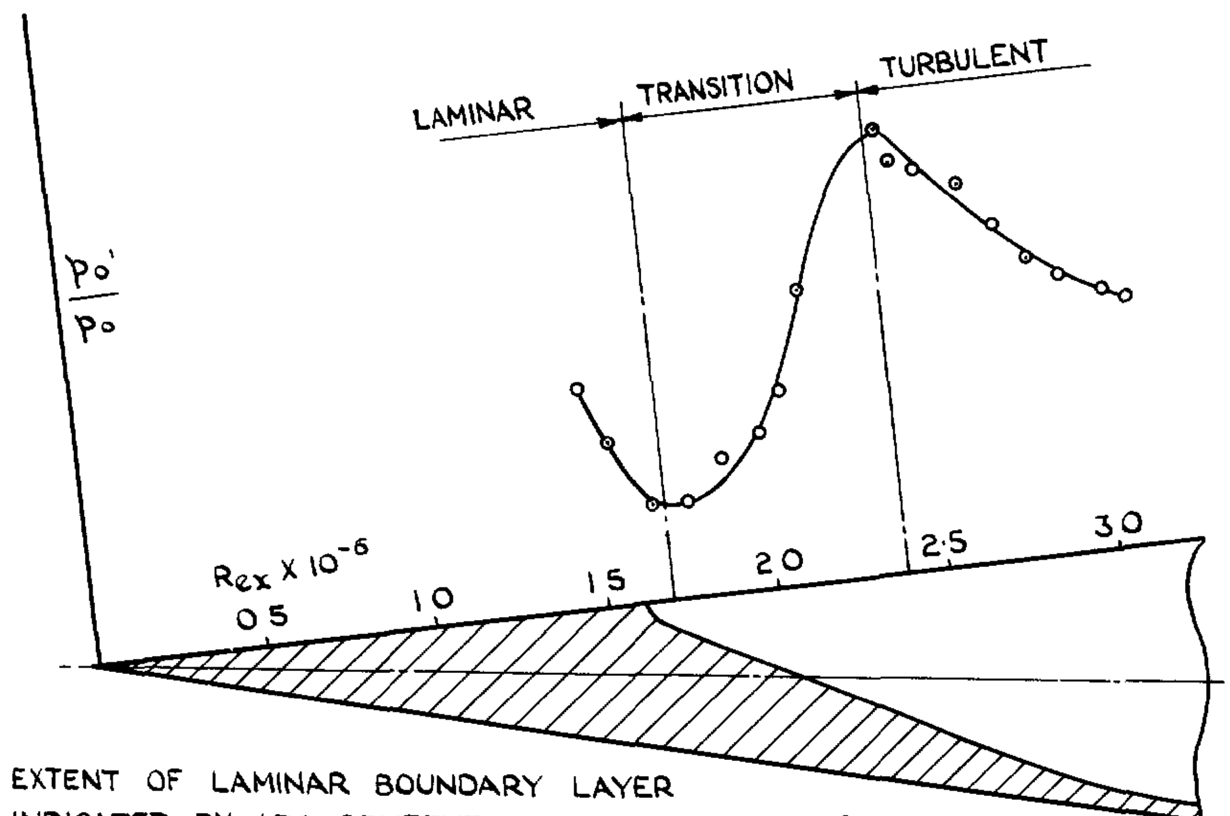
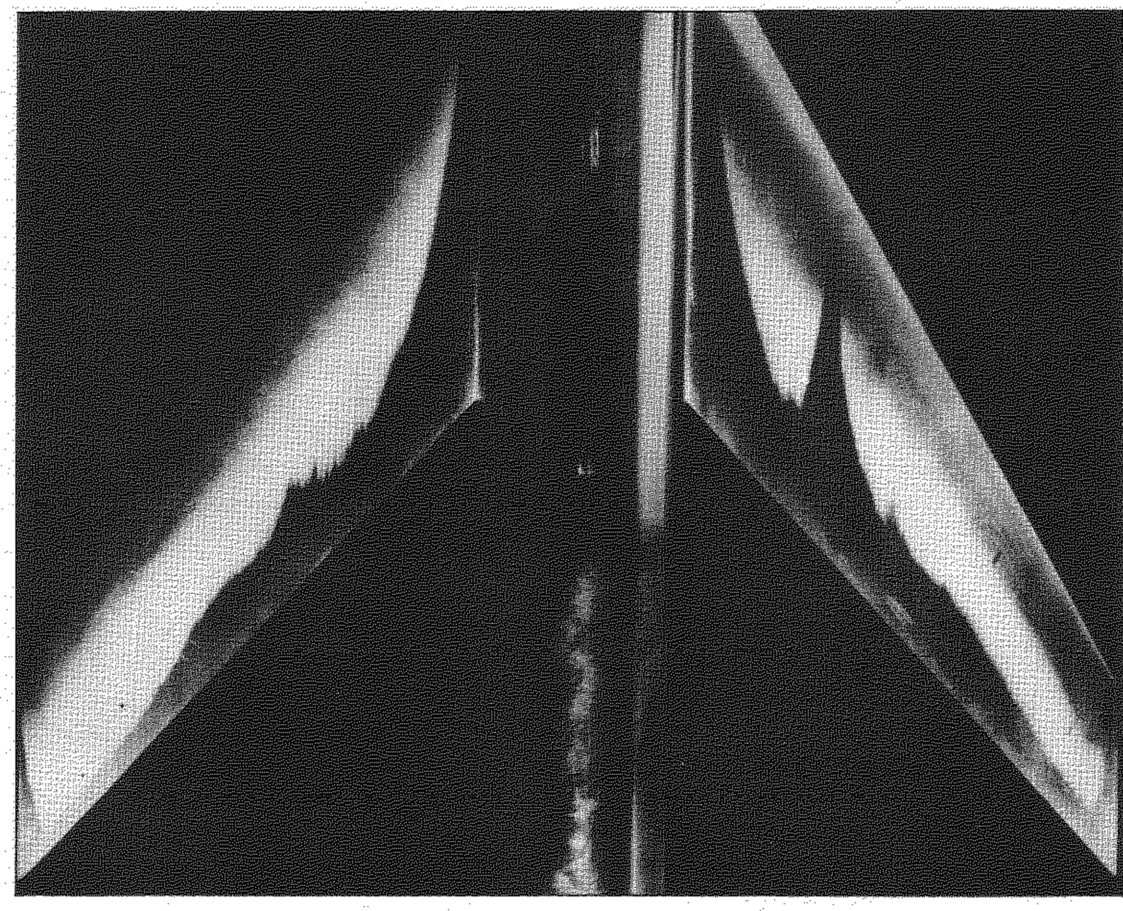
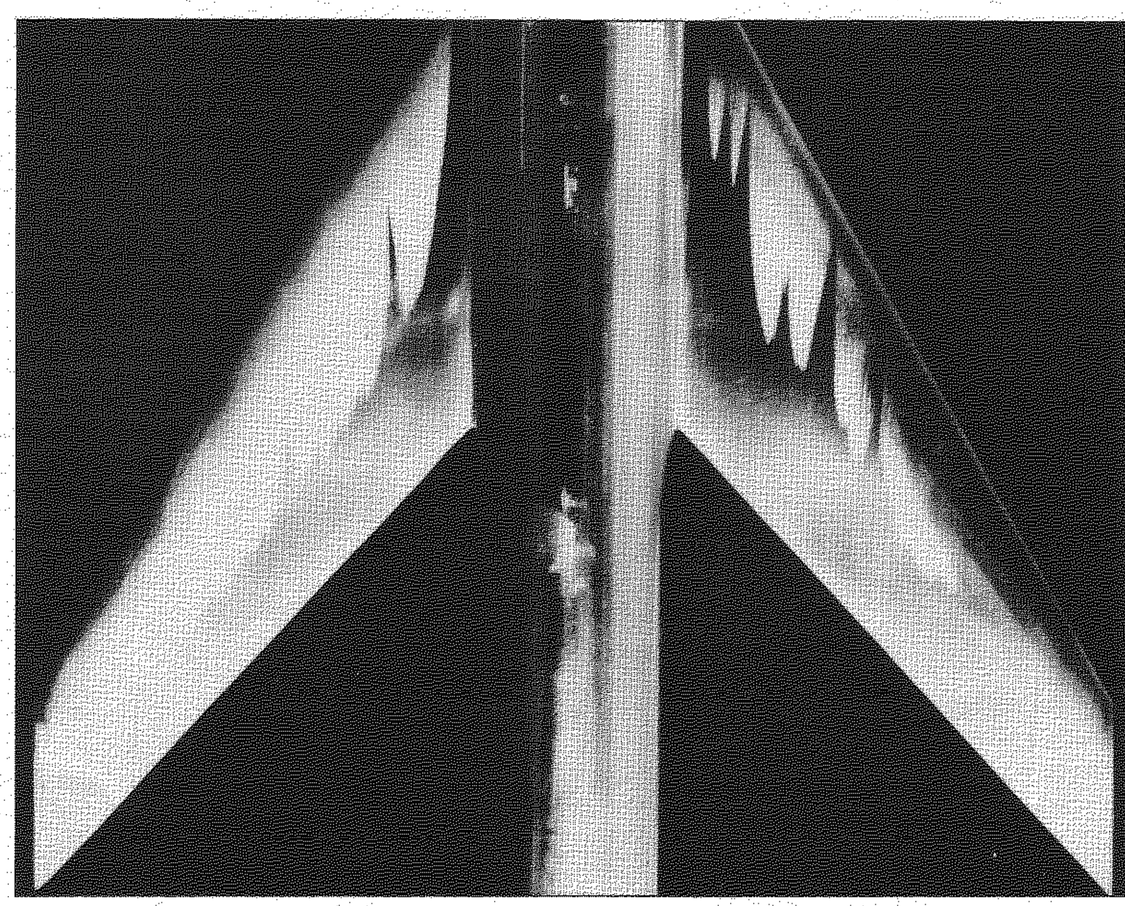


FIG. 6. TRANSITION DETERMINATION COMPARISON BETWEEN AZO-BENZENE INDICATION AND A SURFACE PITOT TRAVERSE ALONG THE TOP GENERATOR OF A 15° CONE. $M_\infty = 3.25$.



a. $R = 1.9 \times 10^6$



b. $R = 2.3 \times 10^6$

FIG.7. TRANSITION PATTERNS ON A SWEPT-BACK WING
$\alpha = 0 \quad M = 1.61$

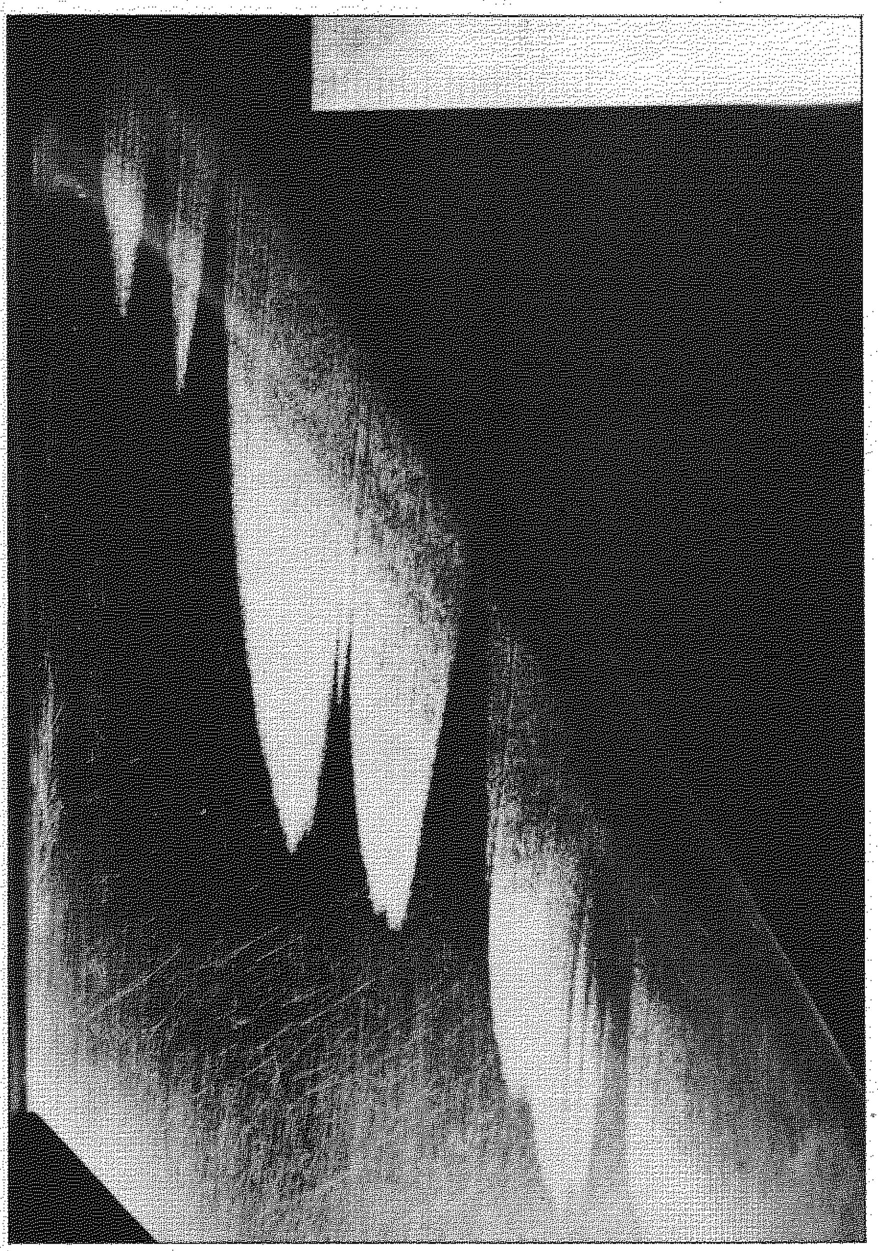


FIG.8. DETAIL OF TRANSITION PATTERN
$\alpha = 0 \quad M = 1.61 \quad R = 2.3 \times 10^6$

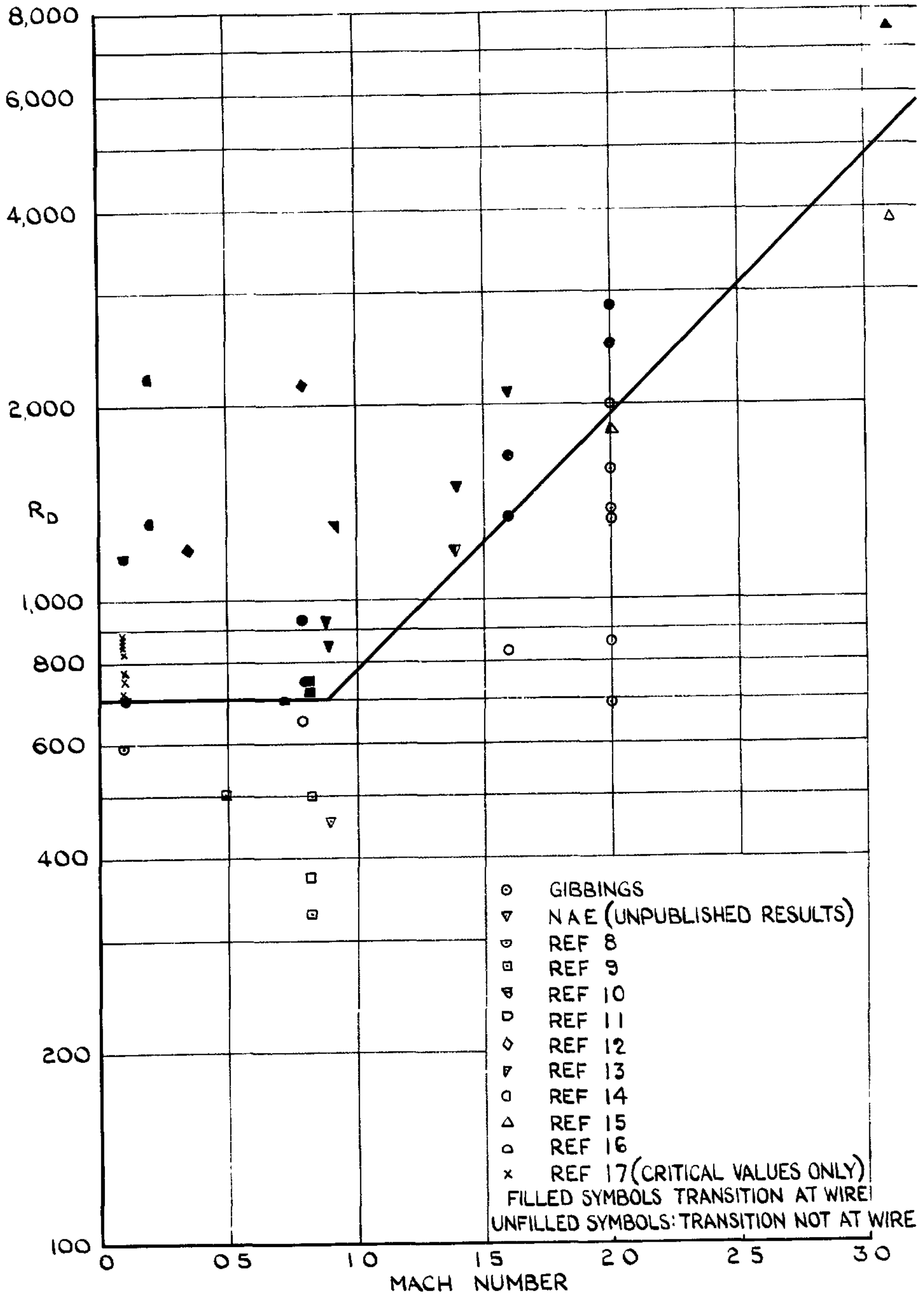


FIG. 9. CORRELATION OF RESULTS OF TESTS USING TRANSITION WIRES.

FIG. 10.

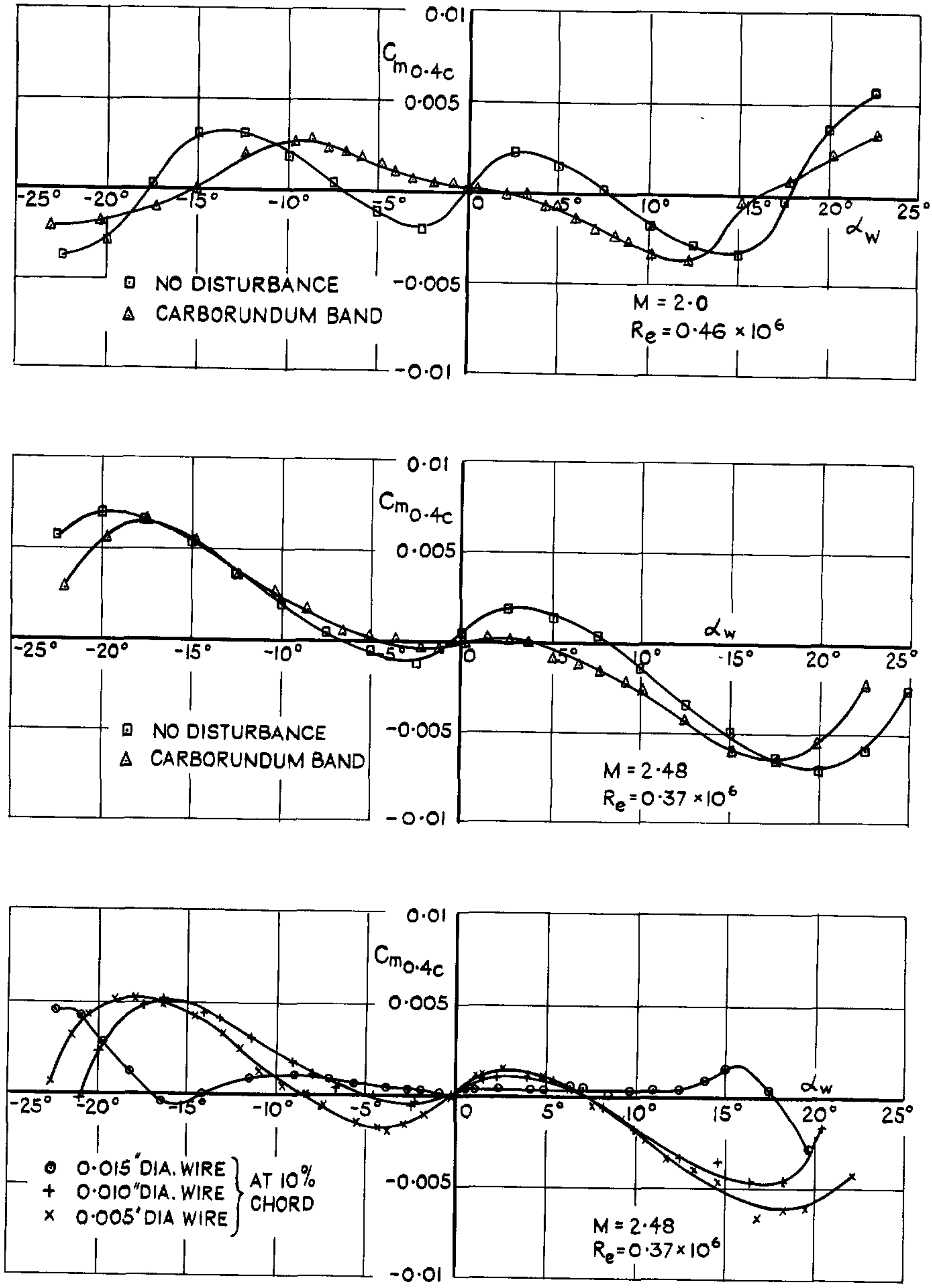


FIG. 10. EFFECT OF TRANSITION ON PITCHING MOMENT OF DOUBLE WEDGE AEROFOIL.

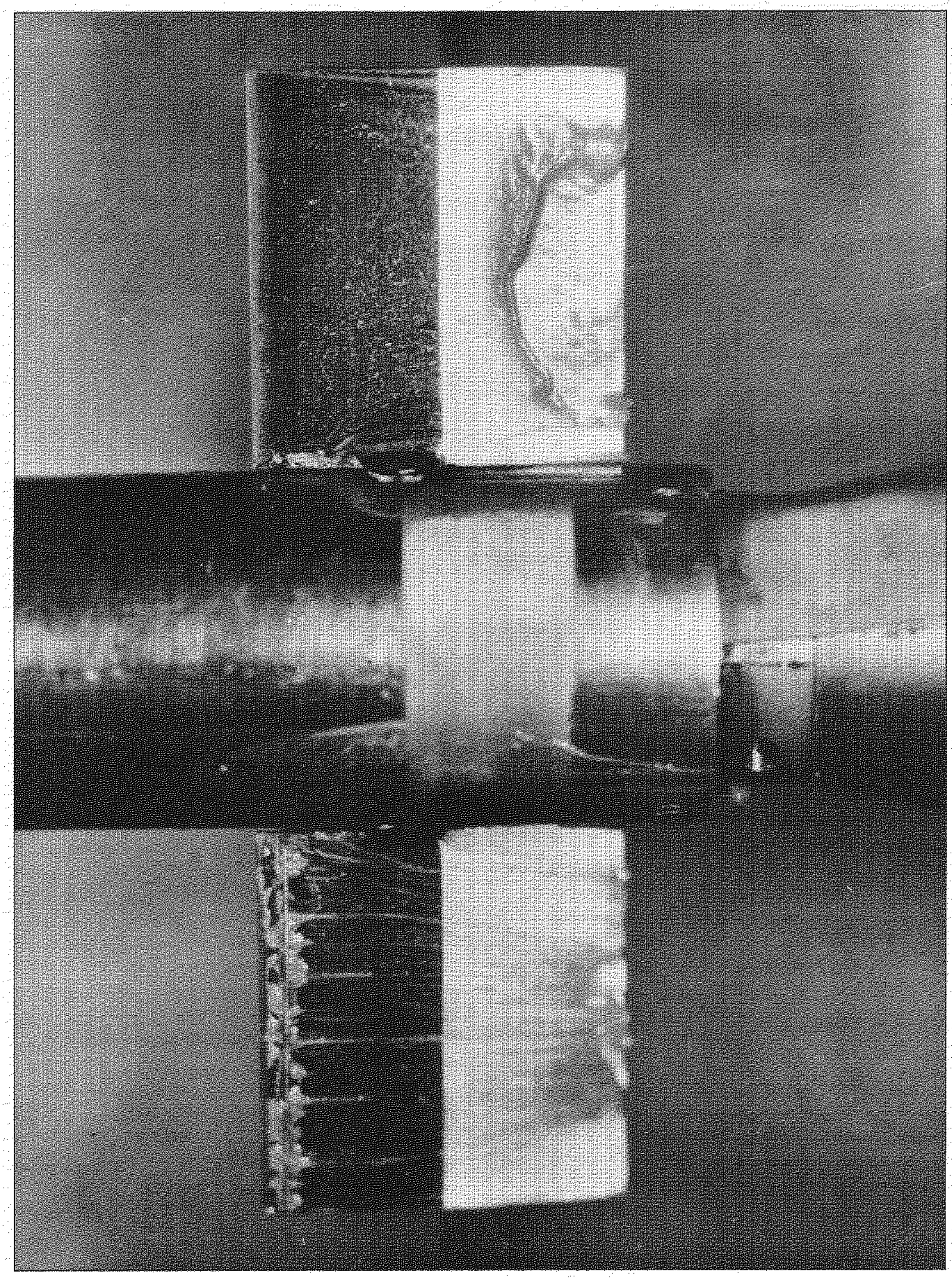
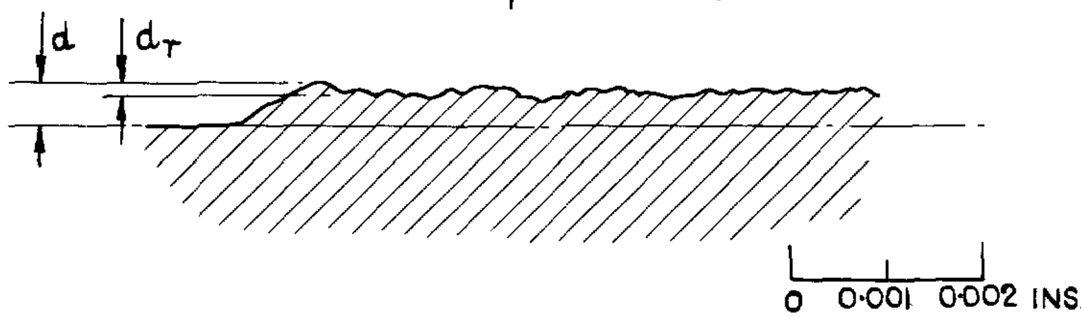


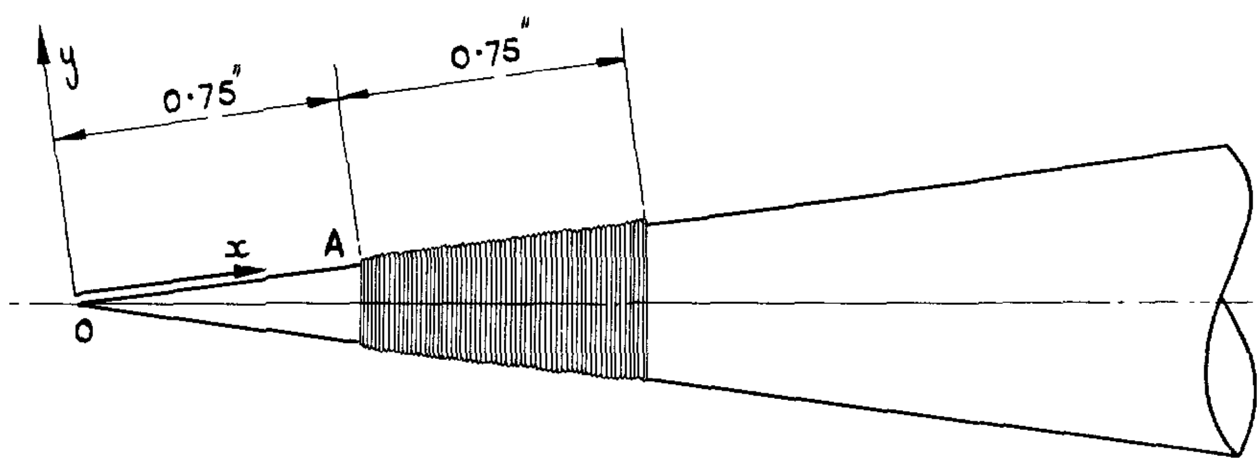
FIG.11. FLOW INDICATED BY OIL ON SUCTION SIDE OF DOUBLE WEDGE AEROFOIL AT $\alpha = 6$ AND $M = 2.48$. LOWER WING WITH 0.015 in. DIAMETER TRANSITION WIRE

FIG. 12.

DIMENSION	AVERAGE VALUE (INS)
d	0·0053
d_r	0·0020



SHADOWGRAPH PROFILE OF ROUGHNESS BAND AT 'A'



CONE TIP AND ROUGHNESS BAND.

FIG. 12. ROUGHNESS ELEMENT ON 10° CONE.

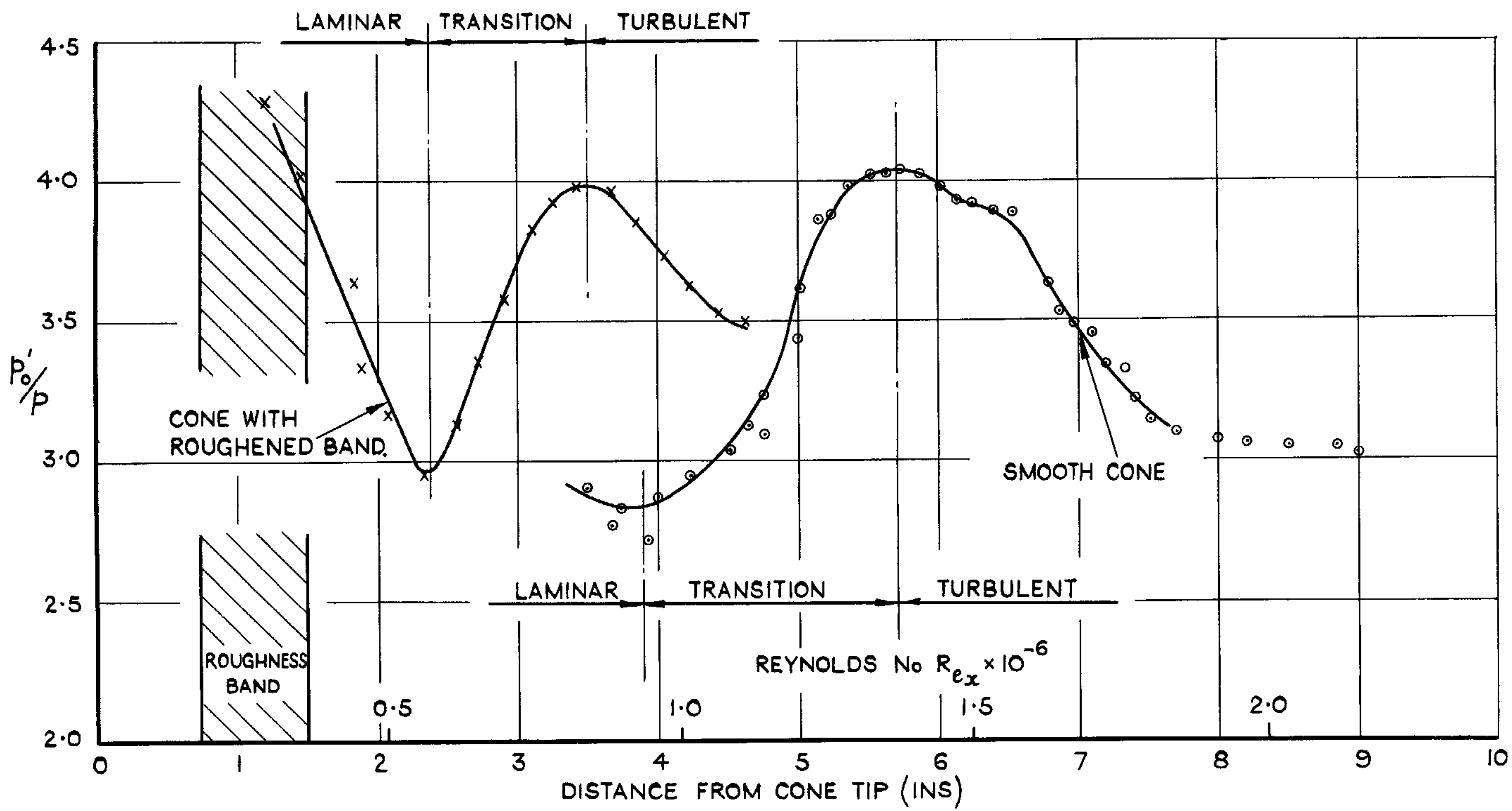


FIG. 13. TRANSITION POSITIONS ON A 10° CONE WITH AND WITHOUT A ROUGHNESS BAND.
(CREEPER PITOT TUBE MEASUREMENTS AT $M_\infty = 2{\cdot}45$).

FIG. 14.

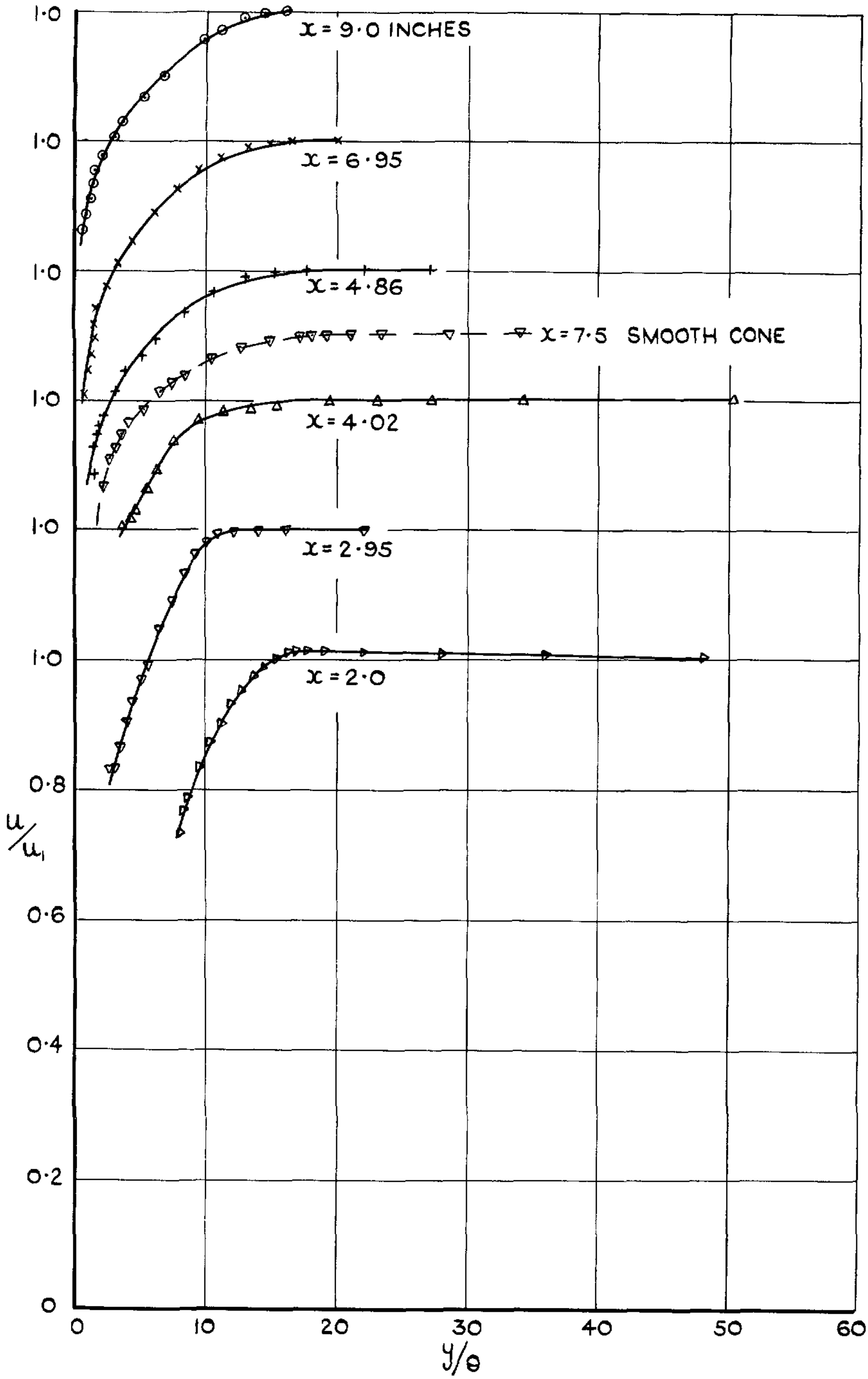


FIG. 14. BOUNDARY LAYER VELOCITY PROFILES ON A 10° CONE WITH A ROUGHNESS BAND (M_∞ = 2·45.)

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