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Transition and Drag Measurements on the Boulton Paul Sample of Laminar Flow Wing Construction.

Part I

Measurements in the 13' x 9' Tunnel at the N.P.L.

- by -

J.H. Preston, B.Sc., Ph.D. and N. Gregory, B.A.

Part II

Measurements in the No.2 11½ ft. Tunnel at the R.A.E.

- by -

K.W. Kimber.

Part III

Joint Discussion of Results, and Note by J.H. Preston.

Joint Summary

Transition tests and drag measurements have been carried out on the Boulton Paul sample of laminar flow wing construction up to $R = 9 \times 10^6$ in the N.P.L. 13' x 9' tunnel and in the R.A.E. 11½' x 8½' tunnel up to $R = 15 \times 10^6$.

At small ± incidences < ± 2° and at R's up to 9×10^6 the agreement between measurements made in both tunnels is good. At incidences of ± 3° and ± 4° the transition moves forward with increase of speed more rapidly in the R.A.E. tunnel than in the N.P.L. tunnel and the transition front is considerably more irregular in the R.A.E. tunnel. This difference occurs in spite of the slightly lesser measured turbulence (this is to be checked) of the R.A.E. tunnel which may be expected to show up, on a wing with appreciable waviness, near the limit of the low drag range.

In the N.P.L. tunnel the theoretical low drag C_L range of ± 0.35 is maintained up to about $R = 7 \times 10^6$ with transition back to beyond 0.40C (pressure min. at 0.45C). In the R.A.E. tunnel the corresponding R is slightly less. The low drag range decreases as R is increased until at $R = 15 \times 10^6$ in the R.A.E. tunnel the low drag range of incidence is only ± 1°. This reduction is attributed to the waviness of the model, exaggerated by the turbulence of the tunnel stream. Nevertheless, the wing, considering its waviness, has performed remarkably well in both tunnels, especially when

compared with previous models of lower waviness. This is put down to:-

(a) Absence of any skin joint (except at L.E.) in regions where laminar flow is expected.

(b) The greater thickness of the aerofoil (18%) and the moderate amount (0.45C) of laminar flow aimed at, both of which help to give the wide C_L range of $\pm$ 0.35 and the very strong favourable pressure gradients at small C_L's.

(c) The new evaporation techniques for determining the transition 'front', which enable surface imperfections giving rise to turbulent wakes to be detected and removed, thus avoiding spuriously high drag readings.

The minimum drags as measured in the two tunnels are in good agreement with each other, and also with theoretical predictions using the measured transition points over a range of Reynolds number of 2.5 to 15 millions.

A tunnel-flight comparison on a wing of this thickness designed for a moderate amount of laminar flow would be very valuable. If, as the present results in the R.A.E. tunnel suggest, some relaxation of the waviness requirements may be possible on such wings, then there is a hope of present standards of construction attaining laminar flow on such sections without resort to filling.

Waviness needs to be studied further in relation to transition phenomena, and more attention needs to be paid to the form and number of waves in addition to the amplitude.

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Part I.

1 Introduction

Part I of the present report describes tests made at the N.P.L. on the Boulton Paul sample of laminar flow wing construction. The specimen is designed to achieve high standards of surface finish and low waviness which are necessary if laminar flow is to be maintained to the points of minimum pressure on the surface of a wing in flight. Three previous reports by Richards[1,2,3] on samples submitted by other firms* have already appeared. The wings so far submitted have had different profiles, with thicknesses ranging from 12% to 18% of the chord. This makes it difficult to draw conclusions as to the merits of one form of construction compared with another. Moreover in the three previous tests no reliable measurements of transition were obtained, whereas in the tests to be described, the 'china-clay' technique of Richards and Burstal[4] enabled the transition line to be accurately determined, whilst wakes arising from isolated roughnesses or from pitting of the surface by grit at top speed, could be detected and their causes removed. Hence the present drag measurements are more reliable and this should be borne in mind when comparing the results in this report with those of the three previous reports[1,2,3].

After the wing had been tested at the N.P.L., it was arranged that it should be tested in the R.A.E. $11\frac{1}{2}'$ x $8\frac{1}{2}'$ tunnel at the R.A.E. This enabled a valuable comparison between N.P.L. 13' x 9' and the R.A.E. $11\frac{1}{2}'$ x $8\frac{1}{2}'$ tunnels to be obtained. It also enabled the present wing to be tested to a Reynolds number of 15×10^6.

2 Method of Construction

A more detailed description of the wing is given in the Appendix from notes supplied by Messrs. Boulton Paul. A sketch is given in fig.10 and photographs of the complete wing are shown in fig.11.

Briefly, the wing is built in two halves, which are locked together in an ingenious manner in the final assembly. The front skin joint is at the leading edge. The main spar is well back at about 0.45c which is about the limit to which laminar flow can persist with ideal construction. The skin continues beyond this to the rear spar. Stringers of 'Z' section are attached to the 14 s.w.g. skin by thread cutting 4 B.A. screws inserted from inside, the shanks being carefully trimmed off afterwards. These stringers are in turn attached to special rib like members connecting the leading edge member and the rear spar to the main spar, thus forming a kind of double box-spar construction (see figs.10 and 11).

3 Surface Finish and Waviness of Wing as Received

3.1 Surface Finish

The wing was received in an unpainted but polished condition. No attempt had been made to fill in scratches or pits in the surface of the skin. The finishing off of the projecting screw shanks attaching the stringers to the skin could undoubtedly be improved, as many of them were slightly 'proud' of surface and could easily be detected by the eye or by touch. There is of course, a fundamental difficulty in producing a perfectly smooth surface where two materials of different hardnesses meet - in this case steel and aluminium.

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* Messrs. Armstrong and Whitworth, Messrs. Supermarine (Vickers) and Messrs. Short Bros.

3.2 Waviness

Dimpling of the surface caused by the screws attaching the stringers to the skin was visible to the eye, particularly when a straight edge was reflected in the surface. There was a bad 'flat' at the main spar position on the lower surface, which was due to an error in assembly.

Waviness measurements taken on a 3" curvature gauge at three span-wise positions are shown in figs.2 and 3. With the exception of the 'flat' on the lower surface, the waviness is comparable with that achieved on the Supermarine and Short specimens, but is not as good as that obtained by Messrs. Armstrong Whitworth[1] or as that achieved on a carefully made wooden model at the N.P.L. Comparative figures for a variety of wings are given in Table I.

4 The Section

The section was N.A.C.A. 65.3.018 and is shown in fig.1. The designation indicates that it is 18% thick and is symmetrical, with a pressure minimum at 0.5c (actually 0.45c), also it has a C_L range of $-0.3 < C_L < +0.3$ (actually $-0.35 < C_L < 0.35$). American test data from their 2-dimensional tunnel at $R = 10 \times 10^6$ shows that C_L max = 1.52 and that the lift slope for small C_{Ls} is 0.11 per degree.

A drawing of the section and the potential flow velocity distribution are shown in fig.1. The full line curve is computed by American methods and the dotted line by Goldstein's method (3rd approx.) with the theoretical lift slope. The velocity distributions by the two methods are in good agreement at small C_{Ls}. Near the limit of the C_L range Goldstein's method gives a small peak near the leading edge to the otherwise nearly constant velocity distribution. This is a feature of some American aerofoils and arises from their more approximate method of design.

5 Transition and Drag Measurements

The 'china clay' technique, developed by Richards and Burstal[4*] was used for determining the transition line on each surface. The wing was sprayed with a mixture of china clay, butyl alcohol, butyl acetate and frigilene. This was allowed to dry and it was then carefully rubbed down with super fine sand paper. The surfaces were then sprayed with nitro-benzene* thus making the coating transparent instead of opaque. The wing was set at 0° incidence and the tunnel was run at 180 ft/sec. The more rapid evaporation in the turbulent part of the boundary layer causes the china clay to dry out first, if there are no wakes from excrescences the line of transition is readily obtained (see fig.12). This was found to be the case at 0° incidence, but at 2° incidence, two wakes were formed on the upper surface (see fig.12). These were traced back to their sources which were rubbed smooth and the test repeated. If these had passed unnoticed, the 'low drag' range of incidence would have been greatly reduced as they were in line with the pitot comb. Transition lines were determined on both surfaces at each angle of incidence for a range of speeds. The drag was obtained from total head and static readings in the wake at 0.1c behind the trailing edge whilst the china clay was drying out for the next transition test, thus saving time.

Figs.4 and 5 show the variation of transition position and C_D** with

* See important footnote in Part II on toxic effects.

** Uncorrected for tunnel interference - this would reduce them by about 3%.

incidence for three Reynolds numbers. A 'low drag' range of incidence of ± 4° at R = 2.3 x 10^6 and of ± 3° at R = 6.9 x 10^6 was obtained. Using the lift slope as determined by the Americans, e.g. 0.11 per deg. and multiplying this by a factor of 1.075 to give a lift slope of 0.118 with tunnel interference gives a C_L range of $-0.47 < C_L < +0.47$ at low Reynolds numbers and $-0.35 < C_L < +0.35$ at high Reynolds numbers. This latter is in excellent agreement with the theoretical value as can be seen from fig.1. It will be noted that the N.A.C.A. C_D curve at R = 3 x 10^6 lies well below the curves obtained in the present experiments. Also note the rapid forward movement of transition on the upper surface when the wing incidence rises above the upper limit for low drag.

Figs.6 and 7 show the transition position and C_D for $\alpha = 0°$ as functions of Reynolds number. It will be noted that the transition position up to R = 9 x 10^6 is behind the position of maximum velocity at 0.45c, except for one observation at R = 9 x 10^6 on the upper surface-curve 'A'. At the lower Reynolds numbers R Ω 2.5 x 10^6 the transition position is back to 0.6c, thus laminar flow is persisting for 0.15c against a considerable adverse pressure gradient. C_D reaches a minimum value of 0.0053 at R = 8 x 10^6 falling slowly from 0.006 at R = 2.5 x 10^6. Above R = 8 x 10^6 C_D rises steeply. Theoretical estimates of the drag of this section were obtained by Tetervin's[5] method using the measured transition points. The agreement with experiment is good up to R = 8 x 10^6, but if both transition lines are assumed to remain back, then curve 'B' is the result (fig. 7). If however, whilst the traverse was being done at top speed, the transition line had moved forward on the upper surface as found on one occasion (curve A fig.6), then taking this in the theoretical calculations curve A fig.7 results which is in fair agreement with experiment. We found that at top speed there was considerable difficulty in holding the transition far back on the upper surface for any length of time. This was presumably due to abrasion by grit in the airstream. Repeat drag tests were made at R = 8 x 10^6 with the china clay removed. They gave the same C_D showing that the china clay had not masked any surface imperfections. Note that the N.A.C.A. values of C_D for a range of R between 3 x 10^6 and 10^7 are almost 20% below the present results. A possible explanation is that their model was better finished which enabled the laminar boundary layer to proceed well beyond the pressure minimum to about 0.65c.

A search was also made for the presence of laminar separation at the lower Reynolds numbers by the use of a fine probe brought near the surface from downstream and carrying sulphuretted hydrogen, which was allowed to stain lead carbonate painted on the surface in the neighbourhood of the transition line. No evidence of laminar separation was found down to R = 1.5 x 10^6.

6 Comparison of Present Tests with Previous Tests in the 13' x 9' Tunnel

In addition to the tests on the three previous specimens already referred to (references 1, 2, 3) a 6' chord E.Q.H.1260 model, both with a wooden surface and with a metal skin, has been tested by Fage[6] and an E.Q.H.1250/1550 aerofoil has been tested by Richards[7]. Comparative data are recorded in Table I and fig.8 shows the minimum drag coefficient as a function of Reynolds number for the 7 aerofoils.

It is seen that according to the curves of fig.8 the Boulton Paul specimen is the best so far tested in spite of its somewhat greater waviness*. Moreover, the rapid drag rise at R = 8 x 10^6 seems to have

* This statement does not necessarily apply to the region very close to the leading edge, where measurements have not been made on the N.P.L. models.

been preventable - in view of the fact that a far back transition position has been recorded up to $R = 9 \times 10^6$ (fig.6) and also, as will appear from Part II, the R.A.E. have succeeded in maintaining low drag up to $R = 15 \times 10^6$ at 0° and 1° incidence.

The poor performance of the wooden models can be attributed without much doubt to deterioration of the surface by abrasion due to grit particles and to the glued joints of the laminations being raised because of humidity and temperature variations. For instance, from Table I, E.Q.H.1260 has a waviness of ± 0.007", whereas E.Q.H. 1250/1550 has a waviness of ± 0.002" on a 3" curvature gauge, but its performance is the worse of the two (fig.8).

The significant feature of the Boulton Paul specimen, which distinguishes it from the previous specimens (Table I) is the fact that the skin joint is at the leading edge and the rear skin joint is well behind the farthest back position at which laminar flow can be expected. Richards in references 1, 2, 3 has called attention to the poor nature of some of the skin joints. These points may well be the chief contributing cause of breakdown of the previous specimens.

The fact that the sections are all different should not be lost sight of. For instance, the Boulton Paul specimen is the thickest (18%), the next in performance is the Short specimen 15.6%; both these sections have set out to achieve only a moderate amount of laminar flow, e.g. 0.45c. Next comes the Armstrong Whitworth section 16% thick. On the whole these thicker sections have performed better than the 12% sections. A possible explanation is that the thicker sections have a greater average curvature and a greater average favourable pressure gradient, both of which are stabilising influences.

7 The Theoretical Computation of Drag

Fig.9 shows a set of drag curves at constant transition positions as calculated by the Tetervin method compared with a similar set, interpolated for an 18% thick wing, obtained from the calculations of Winterbottom and Squire (Ref.8). Similar sets of curves have been used in previous reports for estimating the transition positions, before the advent of evaporation methods of obtaining transition fronts.

It will thus be seen that with transitions behind 50% of the chord there are discrepancies and that had transition been estimated from Squire and Winterbottom's curves at a given C_D and Reynolds number, a further backward position would have resulted than actually occurred. It is suspected that the values given in Table I and those estimated in various flight tests suffer from this defect.

It has since been demonstrated by Mr. R. Lock (Ref.9) that the Squire-Young method and the Tetervin[5]-Holt method when applied to the same aerofoil give drags which are in very close agreement over the practical range of transition movement. It was found that aerofoil shape has an appreciable affect on the estimated drag especially with far back transition. In particular the Squire-Winterbottom charts were calculated for aerofoils with appreciable trailing edge angles, whereas the Boulton Paul section is cusped. A series of charts appropriate to this latter type of aerofoil is given in Ref.9.

8 Conclusions

The waviness of the Boulton Paul test specimen, with the exception of those portions of the surface near the front spar, is comparable with that achieved on the Supermarine[2] and Short[3] specimens but it is not as good as that on the Armstrong Whitworth test specimen. The finishing off of the screw shanks projecting through the surface could be improved. Nevertheless laminar flow has been obtained to beyond 0.45c (the pressure minimum) up to a Reynolds number of $R = 9.0 \times 10^6$ (the top speed of the tunnel) at 0° incidence and the theoretical low drag range of C_L, e.g. $-0.35 < C_L < +0.35$ has been obtained at $R = 7 \times 10^6$ and exceeds this at lower Reynolds numbers. The wing has therefore the best performance of any so far tested in the 13' x 9' tunnel in spite of its waviness and the turbulence of the tunnel stream (contraction 4 : 1). This is probably due primarily to placing the front skin joint at the leading edge and the absence of any further joint in the region where laminar flow can be expected. Secondly the section being 18% thick and the pressure minimum occurring at 0.45c means that the greater convex curvature and the stronger favourable pressure gradients resulting from this may have damped out disturbances arising from waviness, roughness and tunnel turbulence. Also the new technique[4] for obtaining transition has enabled more reliable drag measurements to be obtained and the sources of isolated wakes to be removed.

It does not however follow that a wing with this waviness would have low drag properties at flight Reynolds numbers, particularly as the flight tests on the King Cobra[10] have shown the necessity for surfaces of high finish and much lower waviness. Nevertheless it remains a possibility that greater waviness and poorer surface finish may be tolerated on some profiles than on others. This is a point worthy of further investigation as the demand made on the designers' ingenuity and on production by the tolerance on waviness recommended in the 'King Cobra' report[10] is a very severe one. The R.A.E. tunnel tests should shed more light on the problem as a Reynolds number of 15×10^6 can be obtained on the present test specimen.

It may be remarked that the curvature gauge is not a very satisfactory means of specifying the waviness as it does not enable us to draw out the form of the waves. Nor is the practice of specifying waviness by its amplitude as indicated by the curvature gauge reading, really good enough, since the average pitch of the waves in relation to their amplitude and to the chord length must come into the question. Attempts are being made at the N.P.L. to design a machine for measuring up the wing profile accurately.

The method of estimating the drag described by Tetervin[5] is rapid and has given good results in the present case. Fundamentally, it is the same as the Squire and Young method but uses different relations between the momentum thickness and the skin friction to permit the integration of the momentum equation.

/Table I

Table I.

Characteristics of Low Drag Aerofoils Tested in N.P.L. 13' x 9' Tunnel.

Section & Construction	Waviness on 3" Gauge	Transition at ($\alpha = 0^{o}$)	C_D Min	R_{crit} ($\alpha = 0^{o}$)	Remarks
E.Q.H.1260 Wooden Model	± 0.007"	(Transition estimated at 0.82 C for both Models.	0.0032	3.7×10^6	
E.Q.H.1260 Wooden Model with Metal Skin	± 0.002"		0.0030	5.4×10^6	
E.Q.H.11.8/50/1550 Wooden Model	± 0.002"	Estimated (Squire and Young) 0.7 C	0.004	4.0 -4.4×10^6	R_{crit} poorly defined with slow rise of C_D and tendency to flatten out.
Armstrong-Whitworth 16% Based on N.A.C.A. 66-2-0.15. Fairing on Camber Line.	± 0.004" to 0.15 C ± 0.002" from 0.15 C to 0.6 C (Rear Joint)	Estimated (Squire and Young) 0.7 C	0.004	5.7×10^6	Front skin joint at 0.07 C from L.E. Rear skin joint at 0.60 C from L.E. Very bad bump at rear joint.
Supermarine 12% Roof Top. Metal Model in painted Condition.	± 0.010" to 0.2 C ± 0.0035" from 0.2 C to 0.6 C	Estimated (Squire and Young) 0.68 C	0.0042	4.7×10^6	Skin joints at 0.1 C and 0.75 C. Slow rise of C_D, then stationary up to $R = 6.3 \times 10^6$. Then rapid rise.
Short Bros. N.A.C.A.64,2-(2.5)15.6 $\alpha = 0.4$. Metal.	Upper Surface ± 0.003" Lower Surface ± 0.007"	0.51 C Upper 0.23 C Lower at best section by smoke filament.	0.0054	6.5×10^6 Best Section to 5.6×10^6 Worst Section	Skin joints on upper surface at approx. 0.2 C and 0.6 C. Skin joints on lower surface at 0.22 C, 0.38 C, and 0.52 C.
Boulton Paul. N.A.C.A. 65,3-018 Metal.	Upper ± 0.004" Lower ± 0.006" to 0.39 C from 0.39 C Upper + 0.008" Lower (+ 0.026" (- 0.012"	China Clay Method 0.53 C	0.00525	8.0×10^6	Bad flat on lower surface over main spar. Skin joints at L.E. and 0.74 C.

Part II.

1 Introduction

The measurements of transition and drag described in this part of the report were made in order to extend the Reynolds number of the tests from the 9 million obtainable at the N.P.L. up to 15 million. The opportunity was taken to obtain a comparison of transition in the N.P.L. and R.A.E. Tunnels using the same technique in each case ("China Clay"), and also to obtain a comparison in the R.A.E. Tunnel of transition as determined by the "China Clay"[4] and the liquid film evaporation[11] methods. In addition, the N.P.L. measurements of surface waviness were supplemented by measurements extending right to the nose of the wing, using curvature gauge lengths considerably shorter than that used at the N.P.L.

The experiments were carried out under the supervision of Mr. W.E. Gray.

2 Transition, drag and Curvature Measurements

The wing was tested in the condition in which it was received from the N.P.L., great care having been taken to protect the surface in transit; occasional slight roughnesses due to fine grit particles striking the forward part of the wing during a test were, however, removed when they occurred.

The liquid film evaporation method, developed by Gray[11] was used for determining the position of the transition "front". This method, although based on differential evaporation like the N.P.L. "China Clay" technique, does not involve covering the surface to be tested with a fine coat of pigment (over 1/3000 inch thick); instead, a very thin film of a slow-drying liquid (less than 1/50,000 inch thick) is applied directly to the wing. At low speed, ordinary paraffin was used for the film, but for the higher speeds (above about 80 ft/sec.) a less volatile liquid was needed, and as a suitable oil was not available alpha-chlorhydrin was used. This has a convenient drying time, and the viscosity is such that there is no question of the film moving rearward due to skin friction.

For a polished metal surface, as on the Boulton Paul wing the contrast in surface appearance between the dry and the wet parts was good enough to allow photographs to be taken with the ordinary tunnel lighting. Fig.17 shows three photographs of transition and the relative cleanness of the transition boundary shows that the thinness of the film was fairly uniform; interference colours are the cause of the light borders to the wet areas.

The comparison of transition by the liquid film and China Clay methods was made at 3° incidence only, as this was a sensitive case. In using the China Clay technique it was found that ordinary paraffin is much superior to the nitro-benzene recommended by Richards[4]: (1) it does not contaminate the tunnel with poisonous vapour*, (2) it does not soften the China Clay surface, as does the nitro-benzene, making it easily damaged by grit, etc. during test, (3) it renders the clay coating nearly transparent.

Measurements of drag were made at 0° incidence in the usual way by a pitot comb in the wake.

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* The use of nitro-benzene in industry is highly undesirable because of its insidious toxic effects. Some authorities assess it as being about 100 times as toxic as carbon-monoxide, with delayed action.

Curvature measurements were taken along the chord over 8 inches of profile from the L.E. at two spanwise positions on the suction side of the wing, chosen because of the large difference in transition between them. For these measurements an Ames dial curvature gauge was used, with gauge lengths of 0.6 and 1.0 inches, and the results are given in para. 3.3.

3 Results

3.1 Transition

The variations of transition "front" position, both with incidence and velocity, are shown in Fig.13. It will be seen that at incidences of 0° and 1° the transition on the upper surface remained back at all speeds, and also at 2° up to a speed of 250 ft/sec. At 3° and 4° incidence transition began to move forward rapidly at much lower speeds and was almost at the leading edge at high speed; moreover, the transition "front" became extremely irregular across the span at intermediate speeds, and this is represented by the shaded areas between the split curves of Fig.13. These irregular salients in the transition "front" are well shown in the photographs of Fig.17, and their close connection with surface waviness is discussed in para. 3.3.

The curve in Fig.13 marked "rear boundary" for $\alpha = 3°$ on the upper surface gives the only readings taken at that incidence by the liquid film method, and a comparison with the "china clay" results is given by the neighbouring 3° curve. The slight difference between these two curves was unexpected, as the china clay coating was carefully rubbed down and in fact it covered up most of the roughness due to the screw ends projecting slightly through the skin; this roughness had not caused wedges of turbulence, but the slight covering of clay had been expected to move transition rearward, if anything. No reasonable explanation can be offered at present, but the difference is not very serious.

The more important transition points from the N.P.L. tests have been added to Fig.13 for comparison (see Part III of this report).

3.2 Drag

Drag measurements were made at 0° incidence in the usual way by total head and static readings in the wake. The results obtained are plotted against speed in Fig.14; this has been done instead of the conventional plotting against log R, as that method presents a distorted picture of the wing's behaviour by cramping the important high speed end of the curves and exaggerating the low speed region.

While the drag readings were being taken a check was kept on the surface condition by the liquid film method to ensure that no "wedges" of turbulence were caused by fine grit particles damaging the forward part of the wing and thus affecting the drag readings.

To obtain a comparison between the measured drag and that deduced from transition, drag curves for constant transition, calculated at the N.P.L. by Tetervin's method[5] were used in conjunction with the transition measurements at 0° incidence. Excellent agreement was obtained, as shown in Fig.14.

3.3 Waviness

Surface waviness was noticeable to the eye, and tests at incidence have shown that large spanwise variations of transition were present (see Fig.17), also that as speed was varied the irregularities maintained a consistent pattern. This is obvious from a comparison of the photographs (a), (b) and (c) of the figure, and it pointed to spanwise variations in waviness being the cause. Since the transition at 3° and 4° incidence moved forward at the higher speeds to 2 or 3 per cent chord from the leading edge this seemed to indicate that waviness there was an over-riding cause.

Two spanwise positions were therefore chosen for curvature measurements on the suction side of the wing; at these positions curvature measurements were made from the nose to 8" round the surface; they correspond with the best and worse transition positions of Fig.17(c) and are marked A and B on that photograph.

The curvature records are plotted in Figs.15 and 16 using two different and relatively small gauge lengths because of the fundamental unsoundness of the normal curvature gauge as used for this work. It should be stressed that using a gauge length of several inches on a curved surface having waves of small length gives a very misleading indication, and can show reversals of curvature as though they were only minor variations of curvature. The design curvatures for the section have been estimated approximately and curves of what the gauge readings ought to be on a perfect wing surface are plotted for comparison.

It is obvious from the curves that section A is not as bad as section B, although for the first 2.5 per cent chord it is quite bad; in spite of this the boundary layer keeps laminar to about 8 per cent chord in photograph (c) of Fig.17; on the other hand at section B turbulence is seen from that photograph to start at about 2.5 per cent, i.e. just after the almost flat patch at 2 per cent chord. Behind this the surface is seen to be very wavy for several inches, and this might have caused transition there in the absence of the "flat" mentioned. The effect of the bad irregularities at less than 1 per cent chord from the nose cannot be assessed from these tests.

Errors in curvature close to the nose of a wing have been encountered in several earlier visual transition tests and their effects shown, firstly by the chemical method[12] in flight in 1944[13] and also in the R.A.E. No.2 11½ ft. tunnel in 1945[14].

Three examples of forward transition salients caused by local waviness around the fixing screws are indicated by arrows in the photographs (a) and (b) of Fig.17. Had roughness of the screw ends been the cause of transition these salients would have been sharp wedges with the apex of each at a screw.

3.4 Roughness

The serious waviness of this wing, and the resulting early transition on the suction side when the favourable pressure gradient was reduced by incidence, prevented the effects of roughness showing up as they might have done in the absence of waviness. Some of the roughnesses were of the same order as had caused wedges of turbulence in earlier tests and in flight tests. The very favourable pressure gradients on this wing at the lower incidences may have prevented the formation of wedges, while at the higher incidences and speeds the transition was already almost at the leading edge due to waviness; this precluded wedge formation and prevented a full study of roughness effects.

4 Conclusions

1. At incidences up to 1° there is no increase of drag, i.e. no rapid forward movement of transition, up to the highest speed tested, 400 ft/sec. (R = 15 million), despite very appreciable skin waviness near the nose and some roughness due to projecting screw ends. This is presumably due to the very favourable pressure gradients associated with the section shape and thickness and to the surface curvature. (Fig.13).

2. With decrease of favourable pressure gradient, as incidence increases, the effect of waviness in bringing transition forward on the suction side is very marked. (Fig.13).

3. Owing to the onset of turbulence due to waviness, the effect of roughness could not be studied completely. In the tests there were no wedges of turbulence due to the projecting ends of the skin attaching screws, but, had there been no waviness, wedges might well have appeared at the higher speeds and incidences. (Fig.17).

4. While the results show that the wing as constructed achieves low drag at small incidences and high speeds in the tunnel, it is uncertain by how much the low-drag incidence range might be extended in free air. As tunnel tests are being used to assess the merits of new types of laminar flow wing construction, it seems very desirable that experiments should be made to give information on this point.

5. Transition results by the liquid film evaporation and china clay methods were in substantial agreement. (Fig.13).

6. The relationship between transition and drag is found to agree with that calculated by Tetervin's method[5] for the Boulton Paul wing section.

5 Further Work

It is intended to use the specially prepared wing of the King Cobra for a tunnel-flight comparison of transition; this wing has the weakest 'favourable' pressure gradients of existing aeroplane wings of adequate surface quality, and so should be well suited for showing up any effect of tunnel turbulence on transition.

It is suggested that a tunnel-flight comparison should also be obtained with the low-drag Hurricane wing, which is thicker and has its maximum thickness further forward, thus giving more favourable gradients over the front part of the wing. Since the present tests have indicated that a wider tolerance on waviness can be allowed on such wings, some specific waviness might be added on part of the test section if the existing waviness does not already cause an irregular transition front when flying at incidences of weak gradient.

Such comparisons would be particularly helpful in interpreting the results of tunnel tests made to assess the merits of various forms of low-drag wing construction, and would put on firmer ground all future tunnel work on low-drag wings generally.

Part III.

Joint Discussion of Results - with a Note on the Influence of Shape on the Maintenance of Laminar Flow at High Reynolds Numbers by J.H. Preston.

1 Comparison of N.P.L. and R.A.E. Tests

The limiting speed and Reynolds number for comparison are 220 ft/sec. and $R = 9 \times 10^6$ as these represent the top speed condition of the N.P.L. 13' x 9' tunnel.

At incidences from 0° to 2° the movement of transition with speed on the upper surface was substantially the same in both tunnels. Moreover, the 'china clay' and 'liquid film' techniques gave substantially the same transition line. At 3° and 4° on the upper surface there are differences between the N.P.L. and R.A.E. results. It would seem that in the N.P.L. tunnel transition is further back and that the movement with speed is different. Now at these incidences the pressure gradients on the upper surface are very flat or slightly unfavourable; moreover the wing has considerable waviness and some roughness so that the laminar boundary layer must be in a very unstable state. Hence, slight difference in turbulence or even in noise might cause a considerable shift in transition 'front'; moreover this may no longer be a straight 'front', as spanwise variation of waviness will influence the result. This was in fact observed and was more marked in the R.A.E. tunnel than at the N.P.L. Preliminary hot wire measurements have shown, however, that in the N.P.L. tunnel the intensity of small scale turbulence is slightly greater than for the R.A.E. tunnel. These tests will be repeated shortly. In addition, directional steadiness and freedom from sporadic low frequency velocity fluctuations are better in the R.A.E. tunnel; these latter are features of 'large scale' turbulence. It is accordingly not easy to explain why transition at 3° and 4° incidence should be further back in the N.P.L. tunnel and the need for check measurements on the turbulence of the two tunnels is emphasised.

Comparing Figs.7 and 14, it is seen that for 0° incidence and the range of R covered, the C_D's are in agreement for the two tunnels.

The necessity of carrying the tests to as high a Reynolds number as possible is well brought out by Fig.13. In the N.P.L. tests up to $R = 7 \times 10^6$ the theoretical low drag range of $\pm 3^\circ$ was obtained (a slightly lower R was obtained for this condition in the R.A.E. tunnel). In the R.A.E. tests a low drag range of $\pm 2^\circ$ is maintained up to $R = 10 \times 10^6$ and a range of $\pm 1^\circ$ up to $R = 15 \times 10^6$. These figures of course may be influenced by the degree of turbulence present in the airstream, but there is little doubt that the waviness and finish of the specimen is now making itself felt and is restricting the low drag range at the higher Reynolds numbers. It would be extremely interesting to test a wing of the same chord and section, in which the waviness is a minimum in the R.A.E. tunnel.

It is suggested that in the future, tests on specimens should be made in the R.A.E. No.2 $11\frac{1}{2}$' x $8\frac{1}{2}$' tunnel as the maximum Reynolds number obtainable = 15×10^6 on a 6 ft. wing is not far short of flight Reynolds numbers and the tests are therefore more likely to show up any imperfections than tests at the N.P.L. at smaller R's. The turbulence of the tunnel stream will tend to give results which are pessimistic; on the other hand it may be a useful 'safety factor' and allow a margin for the increased 'scale' of flight.

2 Influence of Section Shape on Maintenance of Laminar Flow at High Reynolds Numbers

The excellent performance of the Boulton Paul specimen (in spite of appreciable waviness) in the N.P.L. 13' x 9' tunnel, as compared with previous specimens and other models, has already been commented on in Part I. Whilst the effect of waviness is showing up in the R.A.E. tests at incidences > 1° at high R's, the performance is still remarkable in a tunnel in which the turbulence level is probably fairly high compared with the reputed level of the best American tunnels. In Part I some of this increased performance was attributed to the placing of the skin joint at the L.E.

The phenomenon of transition is not yet completely understood, but the experimental work of Schubauer and Skramstadt (Ref.16) on the stability of the laminar boundary layer of a flat plate and similar work by Liepmann on the effect of curvature on stability (Refs.17, 18) which confirm the stability theories of Tollmien and Schlichting for the flat plate and that of Gortler for the concave plate, represents a big step forward. Liepmann found that the stability of the boundary layer for convex surfaces is of the type visualised by Tollmien and Schlichting for the flat plate. Briefly, oscillations occur in the boundary layer of a flat plate or aerofoil arising from external disturbances, which for a certain distance from the front stagnation point are damped - for the flat plate this is represented by a Reynolds number $R_1 \simeq 6 \times 10^4$*. After this point the oscillations are amplified and if the disturbances from which they originate are large enough, transition may occur at a Reynolds number R_T, otherwise they then pass through this amplification region to a stable region. The length of the amplification region is a function of the frequency of the disturbances. Now in the case of the flat plate, $R_T \simeq 2.8 \times 10^6$ in a tunnel of low turbulence, and for laminar flow aerofoils it may be as high as 25×10^6. It is the gap between R_1 and R_T for which there is no explanation in quantitative terms. We do know, however, that R_T is greatly increased by favourable pressure gradients - in fact the possibility of laminar flow wings depends entirely on this, - but Liepmann[17] has found that convex curvature barely affects R_T, whereas concave curvature was found to have a pronounced adverse effect. We also know that R_T is decreased by increases of stream turbulence, of waviness, and of roughness if above a certain magnitude which is a function of position on the aerofoil surface and of the Reynolds number.

As regards roughness, some unpublished experiments at the N.P.L. on isolated roughnesses, roughly cubical in shape, located at various positions on a laminar flow wing, show that turbulent wakes do not form behind these obstacles if R_t (local Reynolds number of obstacle) < 400 when they are well back from the nose. If located near the nose, where pressure gradients are very favourable, R_t > 900 for transition to be affected. Now disturbances are certainly being shed by these obstacles for R_t > 100, hence it appears that the frequency of the disturbance is such that the length of amplification region is short and that up to $R \simeq 400$ the amplified disturbance is not large enough to affect transition.

While roughnesses are easily detected and removed, waviness is a much more serious problem and constitutes the chief obstacle to the achievement of laminar flow in flight. Fage (Ref.19) has investigated the effect of isolated hollows and bulges. It appears that when the wave is sufficiently abrupt the adverse pressure gradient associated with it is sufficient to cause laminar separation, e.g. Polahusen's parameter $\lambda = \frac{\delta^2}{\nu}\frac{dU_1}{dx} = -12$. Disturbances are then propagated, which may or may not be damped out. It would seem therefore that

* More recent theoretical work by Pretsch shows that for an aerofoil (depending on the pressure gradient) R_1 may reach 2×10^6.

near the nose, where 'δ' the boundary layer thickness is small and where $\frac{dU_1}{dx}$ for the aerofoil is large and positive (in fact $\lambda = \Omega = 7.0$ near the nose), more abrupt waves might be tolerated - which is the argument for having the front skin joint there. The fact that it is the wave form which matters shows that the practice of specifying this by its amplitude is not correct, nor for that matter does the curvature gauge correctly give the amplitude. Moreover the frequency of waves per chord length must also enter the picture as it is clear that one or two large waves merely constitute a slightly different aerofoil shape. Again, a large number of small waves may have a considerably greater effect than a single wave of the same amplitude, as the boundary layer tends to be thickened by each wave and its stability thereby decreased. However, these arguments do not explain the superior performance of the Boulton Paul wing with its appreciable waviness over say the thin aerofoils with small waviness*. All that can be said is that the Boulton Paul section has a much greater C_L range and its pressure minimum is nearer to the nose than for the E.Q.H.1260 type. In general, and excluding the noses, whilst the amplitude of the waves as determined by the curvature gauge was greater for the 'B.P.' wing for E.Q.H.1260, the pitch of the waves was even greater, so that in effect the waves on the B.P. were less abrupt. This suggests that the development of a profiling machine is a necessity for further study of low drag sections.

In conclusion, it would appear that in deciding on a low drag section, with existing standards of construction, designers would be well advised to choose where possible a wing with a wide C_L range and to aim at moderate amounts of laminar flow. Further research should be carried out to check this point and see whether the allowable standard waviness recommended from the King Cobra flight tests can be relaxed.

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* The N.P.L. measurements of waviness stopped short of the nose. In the case of the E.Q.H.1260 with metal skin the R.A.E. found[14] that at the nose the waviness was much greater than further back. On the other hand, one would expect the allowable waviness tolerance to be greater at the nose.

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7	E.J. Richards, W.S. Walker & J.R. Greening	Tests on the Aerofoil E.Q.H.1250/1550 in the N.P.L. 13' x 9' Low Turbulence Wind Tunnel.	6676
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/14.

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15	F. Smith, D.J. Higton & T.A.R. Bramwell	Flight Tests on Hurricane II Z.3687 Fitted with Special Wings of "Low Drag" Design. R.A.E. Report Aero.2090. October 1945.
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17	A.W. Liepmann	Investigations of Laminar Boundary-Layer Stability and Transition on Curved Boundaries. N.A.C.A., A.C.R. A.R.C. Report 7302.
18	A.W. Liepmann	Investigation of Boundary Layer Transition on Concave Walls. A.C.R. No.4 J.28 (Not reproduced by A.R.C. yet).
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Appendix

Description of Laminar Flow Wing Test Specimen.

(Notes supplied by Messrs. Boulton Paul.)

Introduction

The specimen described in this note is one of a series being constructed, by various aircraft manufacturers, in an attempt to find a suitable form of construction for a wing possessing laminar flow characteristics.

After consultation with N.P.L. and R.A.E. an N.A.C.A.65$_3$ - 018 symmetrical aerofoil section was selected for the tests. A non tapered specimen of chord 6 ft. 0 in. x span 8 ft. 10 in. suitable in size for the N.P.L. wind tunnel was therefore constructed. The specimen has been designed to meet both wind tunnel and strength test requirements.

In order to obtain a rational set of loading conditions it has been assumed that the centre of the section would represent the geometric mean chord of a straight tapered wing, of aspect ratio 6, suitable for a fighter type of aeroplane having a maximum diving speed of 525 m.p.h. E.A.S. at 10,000 ft. A preliminary calculation showed that similar loading conditions could be obtained by using certain combinations of ultimate load factor and wing loading, therefore the following three combinations were selected as representative of the conditions which could be obtained with the type of aeroplane under consideration.

Condition	Ult. Factor (Normal Accel. x 1.5)	Wind loading lb/ft.^{2}
1	8	50
2	10	40
3	12	35

Design Considerations

It was considered that in order to obtain laminar flow a skin covering of at least 14 s.w.g. would be required, furthermore it was felt that no spanwise joints could be allowed between the leading edge at chord line and a point well aft of the theoretical transition point. As a thick covering was selected it was necessary from a structurally economic point of view to develop as high a stress in the wing covering as possible.

To avoid the possible deformation of the skin by attachment to a rigid form of internal structure, a light form of internal skin bracing was used. The normal form of heavy spar boom extrusion was therefore replaced by a light extrusion, necessary only for spar web attachment. Closely pitched skin touching stringers and upper and lower stringer touching ribs completed the structure.

As it was considered impossible to avoid local deformation of the skin when forming a rivet it was decided to attach the stringers to the skin by thread cutting screws, inserted from the inside of the wings.

From the foregoing it will be seen that a method by which the whole of the assembly could be carried out from the inside of the wing would be necessary, at the same time a suitable method of joining the two halves together had to be developed.

An over-riding consideration was that the form of construction had to be suitable for quantity production, and would fulfil the requirements of serviceability and maintenance.

It must be emphasised that no filler, or coating, was applied to the outer surface of the wing skin. The test specimen was therefore left as constructed, no attempt being made to improve the surface by such means.

From actual weights it is estimated that a wing using this type of construction could be built which would be at least as light as any conventional form.

Details of the form of construction are given in the next section.

<u>Method of Construction</u>. (See Figs.10 and 11)

The wing takes the form of a load carrying box section, built in halves (upper and lower halves, split along the chord line). The laminar flow portion extends from the leading edge to 74% chord, aft of this point the aerofoil section was completed in wood solely for the purpose of wind tunnel tests.

The skin was formed by rolling 14 S.W.G. light alloy sheets to the aerofoil contour. After forming, each skin was placed in turn in a wooden cradle such that complete access was obtained to the inside surface of the sheet for assembly purposes, the complete assembly of each half was made while held in this cradle. Light gauge stringers, "Y" section main spar extrusion and a "T" section auxiliary spar extrusion were then attached to the skin with "Shakeproof" thread cutting 4 B.A. screws. These being inserted from the inside, the shank which protruded through the skin was left to be removed at a later stage in the assembly. The leading edge extrusion was then attached with similar screws, but these were inserted from the outside of the skin.

The main spar and leading edge extrusion mentioned above, incorporates a bulbous portion through which a hole was extruded, to close limits, over its entire length. The bulbous portions were machined before assembly to form the complimentary halves of a segmented hinge extending over the entire length of the wing.

The conventional type of rib was replaced by four members of top hat section, the rims of which were bolted to the stringers. The ends of the rib members forward of the main spar were attached to the leading edge and main spar extrusions, the ends of the rib members aft of the main spar were attached to the main spar extrusion and auxiliary spar web.

The main spar web was made from flat sheet reinforced with light "Z" sections at each rib and intermediate position. A segmented bulbous extrusion was rivetted to the upper and lower edges.

The auxiliary spar web was made from flat sheet in halves split along the chord line, the upper edge being attached to the "T" section extrusions already assembled on the wing skin, the mating edges of each half were fitted with segmented complimentary extruded hinge sections. It will be seen that the internal structure of the wing was complete at this stage, the final assembly consisted of joining the two halves together by the insertion of special tubular pintles through the complimentary portions of the four hinge sections. The pintles consisted of hardened and ground tubular members which were made from spirally wound steel strip. To ensure ease of assembly this tube or pintle was approximately 0.015" less in diameter than the hole in the extruded hinge section. When finally assembling the halves of the wing, this tube is subjected to compression by means of screwed plugs at the extremities, thus the tube is expanded in diameter until a tight fit is obtained in the hinge sections. Conversely, when the plugs are removed, the tube reverts to its original diameter and can easily be withdrawn, thus facilitating major repair and maintenance.

The completed section was then removed from the cradle and the protruding shanks of the thread cutting screws were cut off in a special manner and finally milled flush with the skin surface.

Spreader plates and end fittings, etc., to R.A.E. requirements were then fitted for strength tests.

Before delivery to the N.P.L. for wind tunnel tests the bare surface of the skin was polished, no filler or protection being used.

No. 19680S

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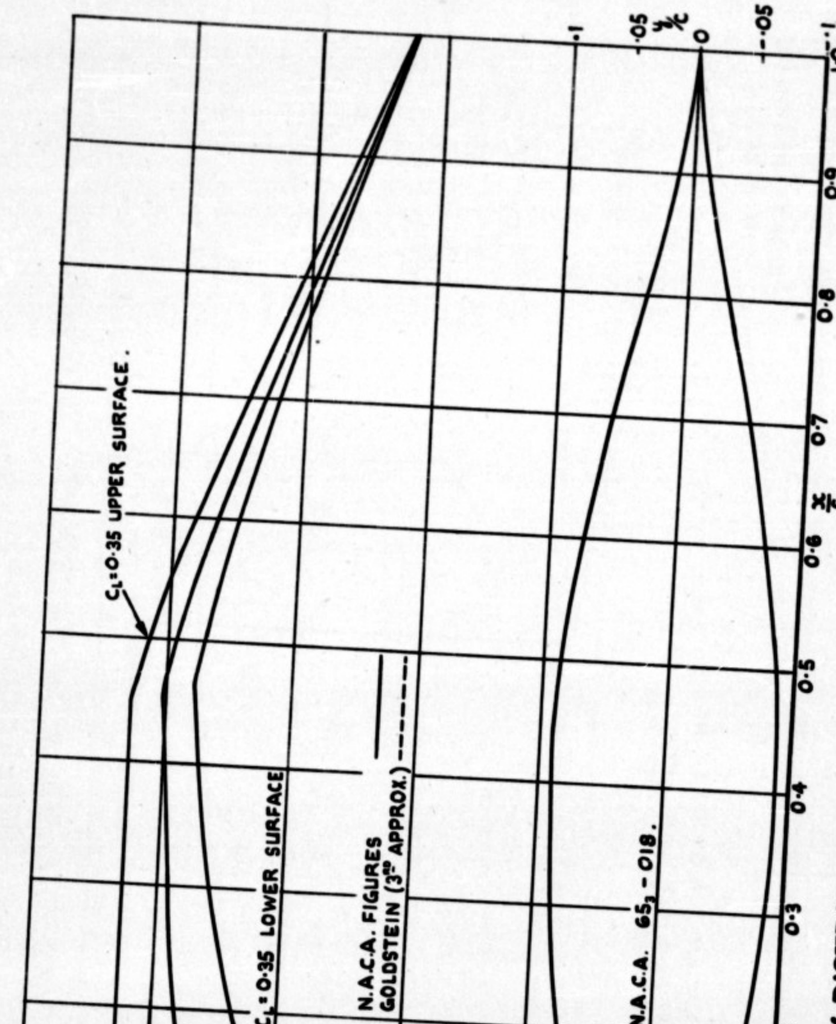
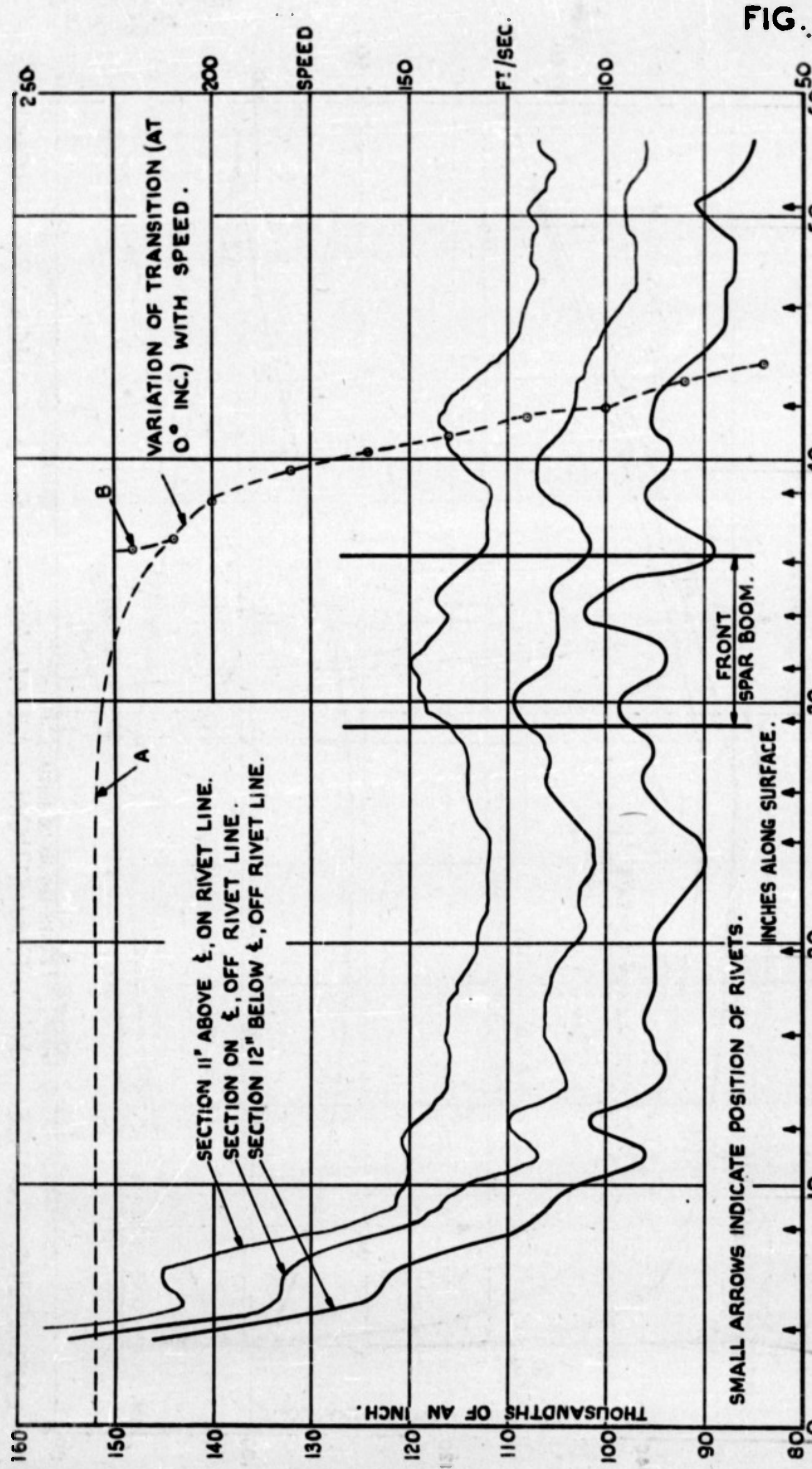
FIG. 1.



VELOCITY DISTRIBUTION AND PROFILE OF WING SECTION.

FIG. 2.



CURVATURE GAUGE AND TRANSITION MEASUREMENTS. 3" GAUGE, 72" CHORD. UPPER SURFACE.

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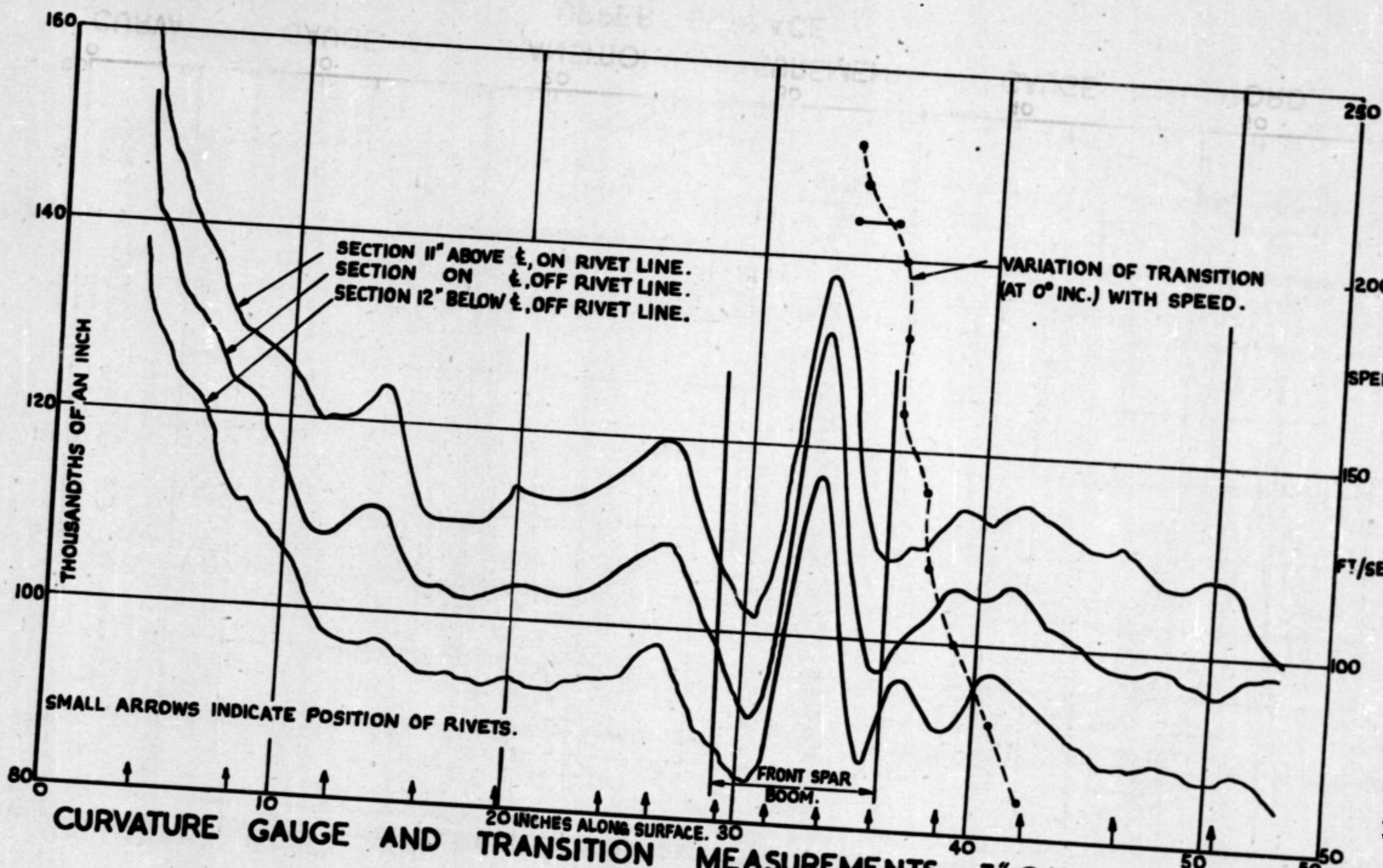
FIG. 3.



CURVATURE GAUGE AND TRANSITION MEASUREMENTS. 3" GAUGE, 72" CHORD. LOWER SURFACE.

24

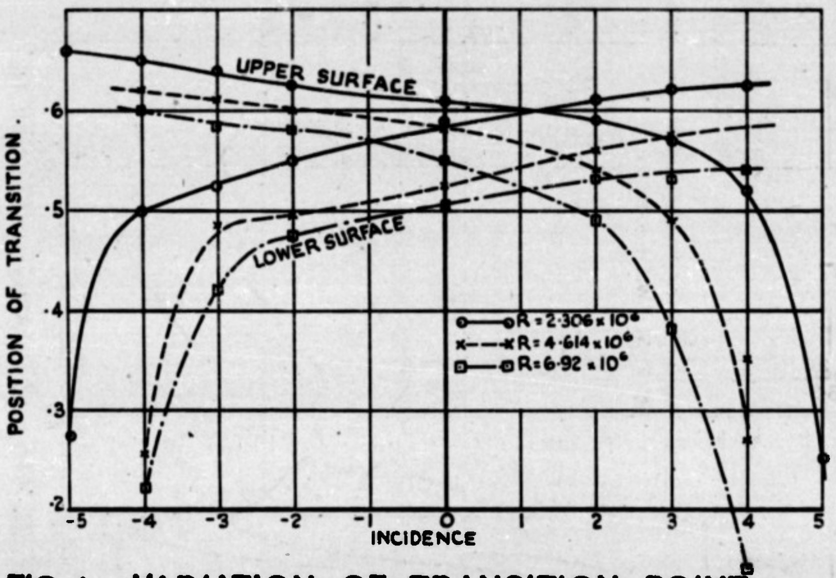


FIG. 4. VARIATION OF TRANSITION POINT WITH INCIDENCE.

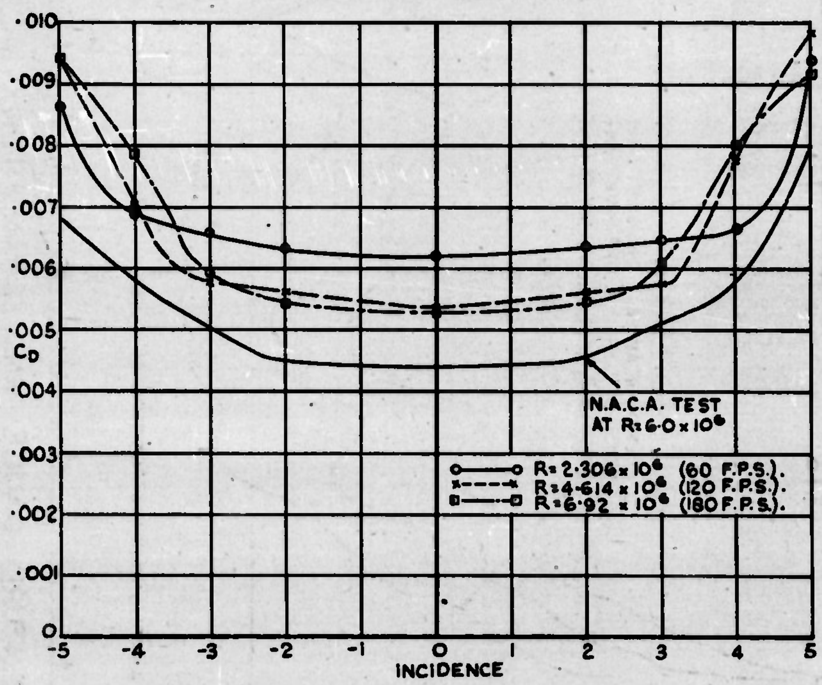


FIG. 5. VARIATION OF DRAG COEFFICIENT WITH INCIDENCE.

No. 19684 S

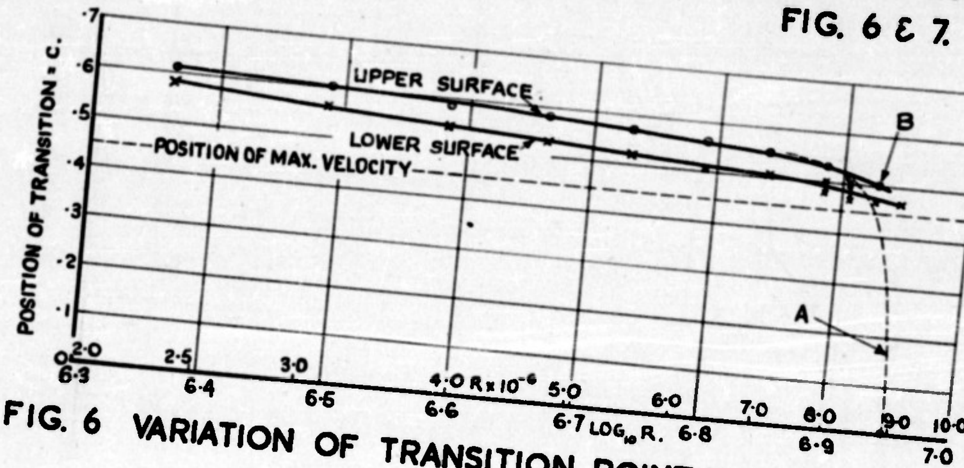


FIG. 6 VARIATION OF TRANSITION POINT WITH REYNOLDS' NUMBER AT 0° INCIDENCE.

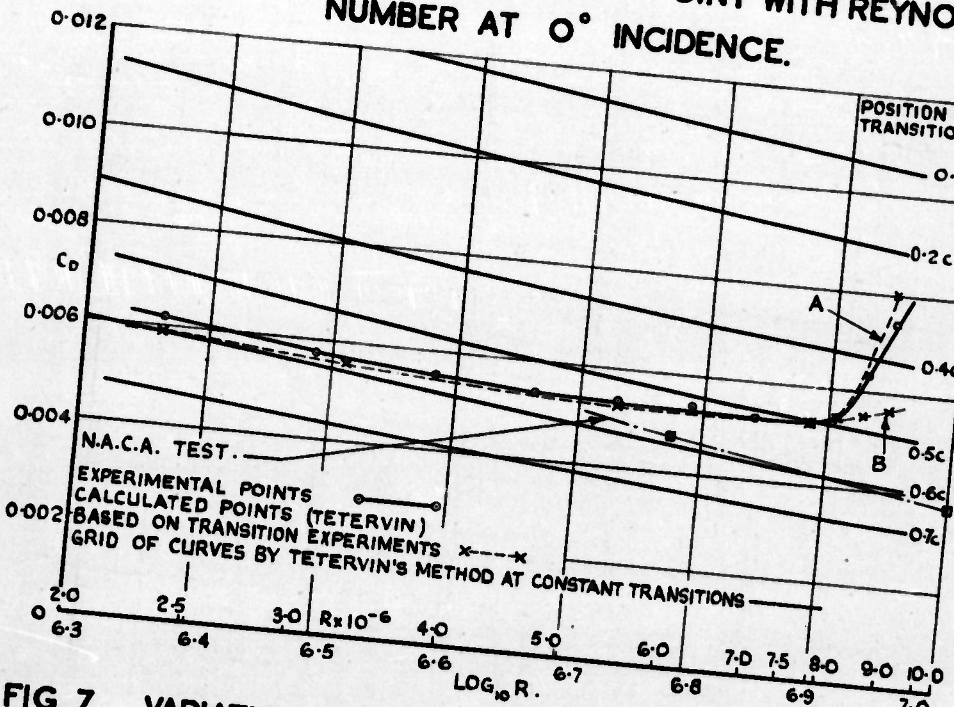
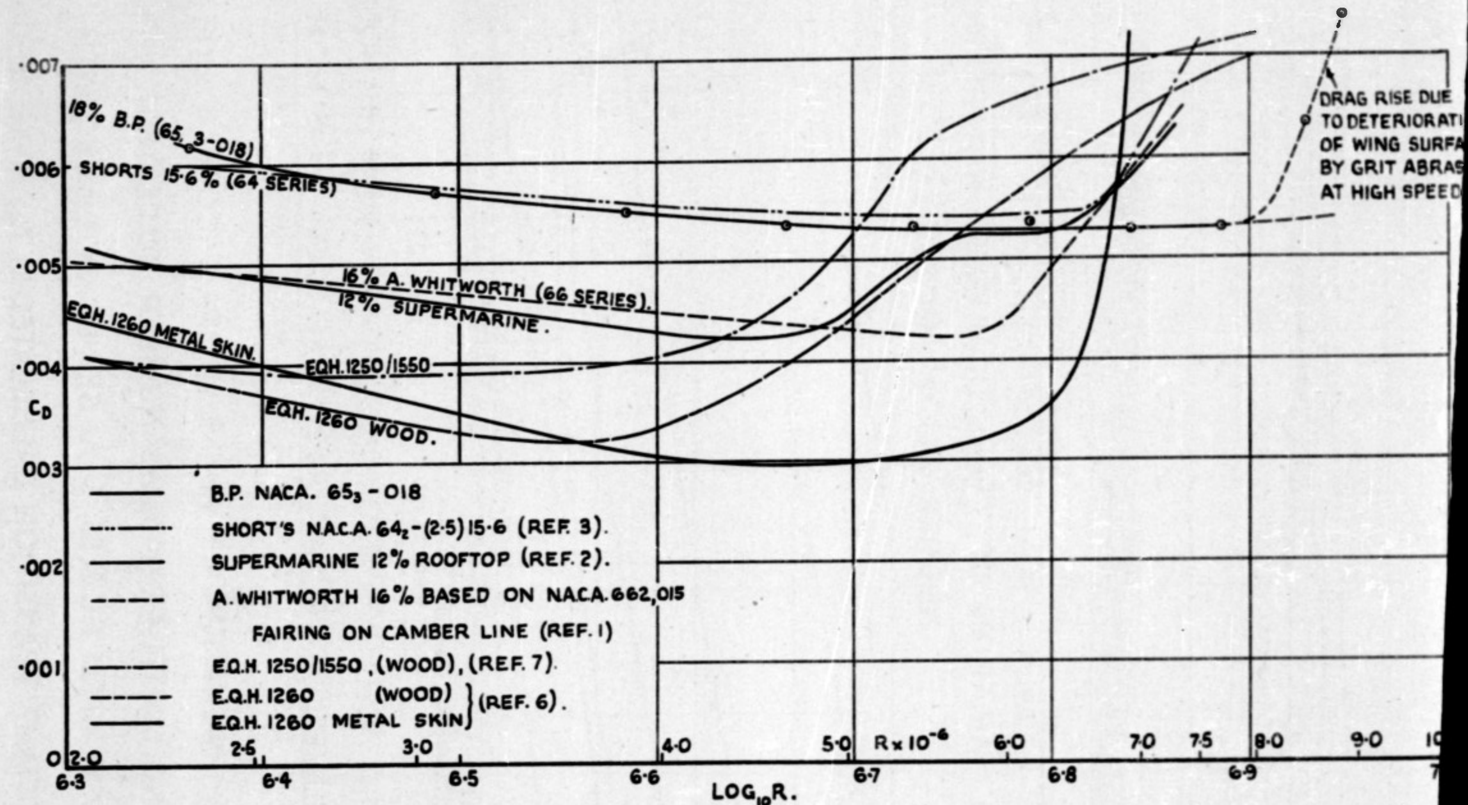


FIG. 7 VARIATION OF MINIMUM DRAG COEFFICIENT (0° INC.) WITH REYNOLDS' NUMBER TOGETHER WITH CALCULATED POINTS & A SERIES OF LINES WITH CONSTANT TRANSITION CALCULATED BY TETERVIN'S METHOD.



VARIATION OF DRAG COEFFICIENT WITH REYNOLDS' NUMBER. COMPARATIVE DATA ON LOW DRAG AEROFOILS TESTED IN THE N.P.L. 13' × 9' WIND TUNNEL.

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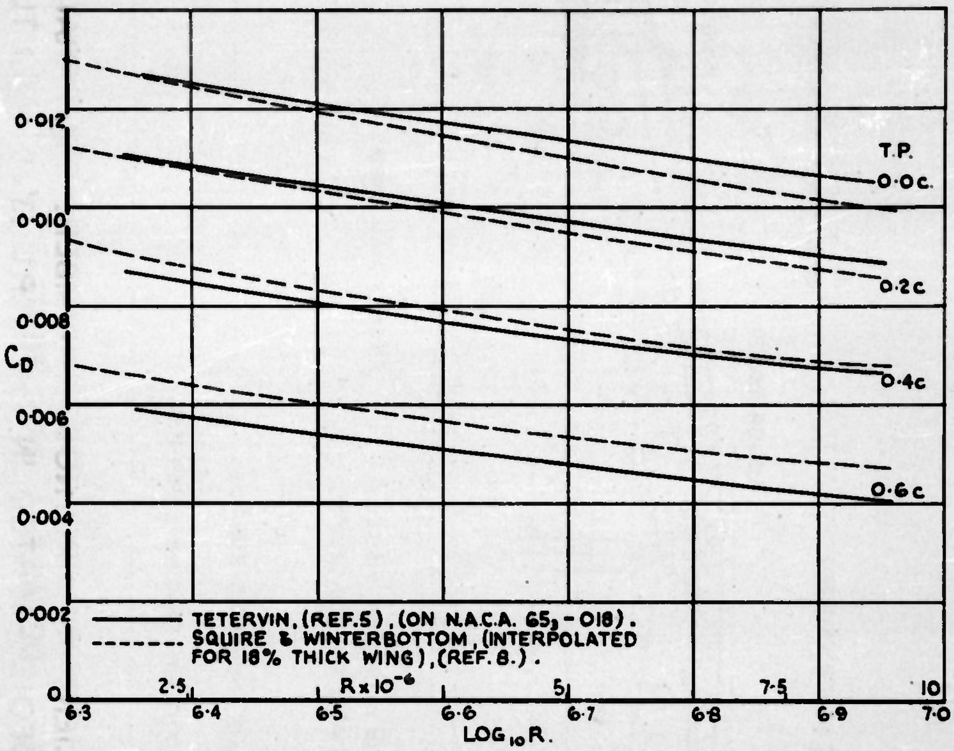
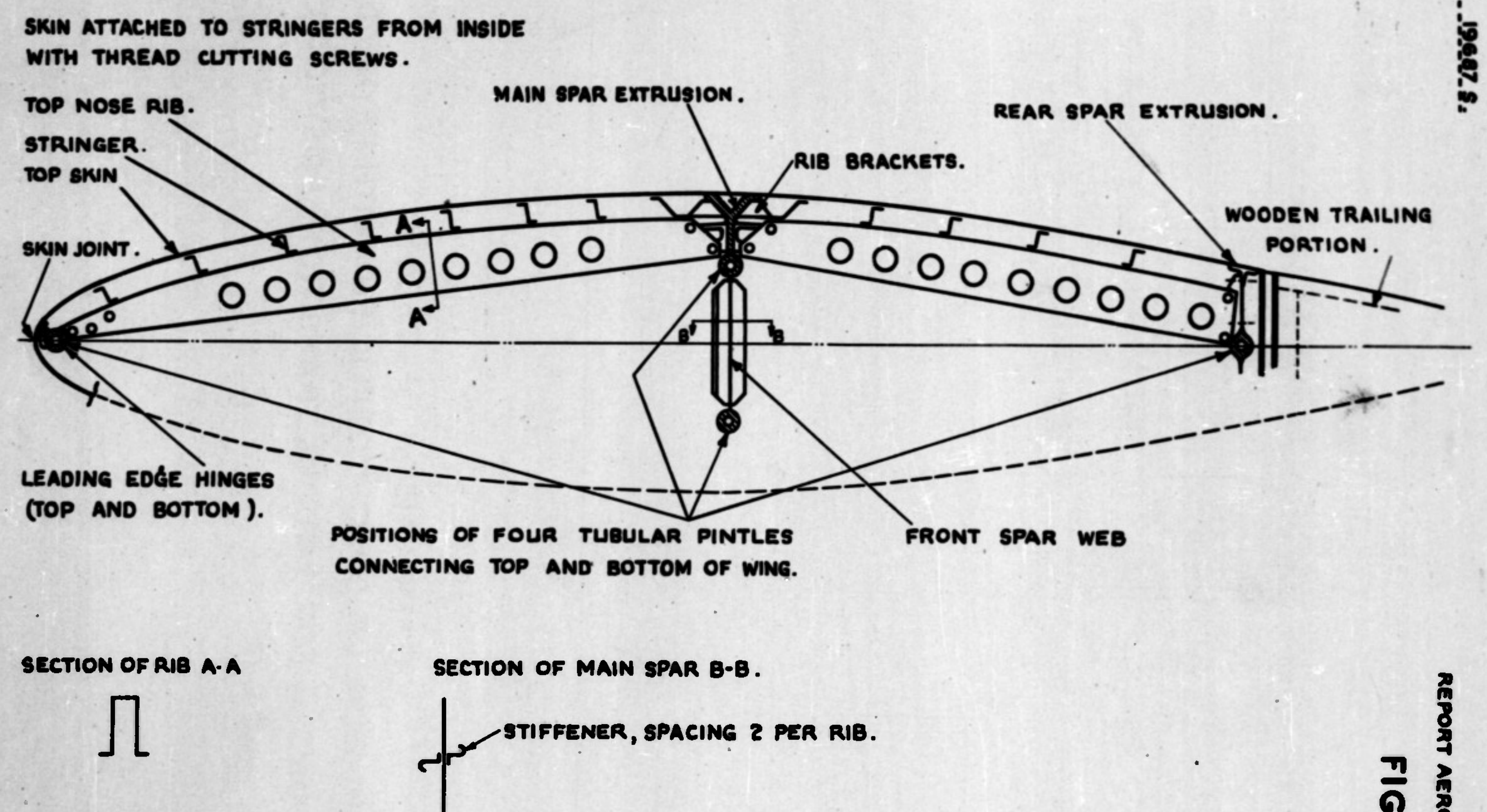
FIG. 9.



COMPARISON OF THEORY.

VARIATION OF DRAG COEFFICIENT WITH REYNOLDS' NUMBER FOR VARIOUS TRANSITION POSITIONS.

No. 19687 S.



CONSTRUCTIONAL DETAILS OF WING.

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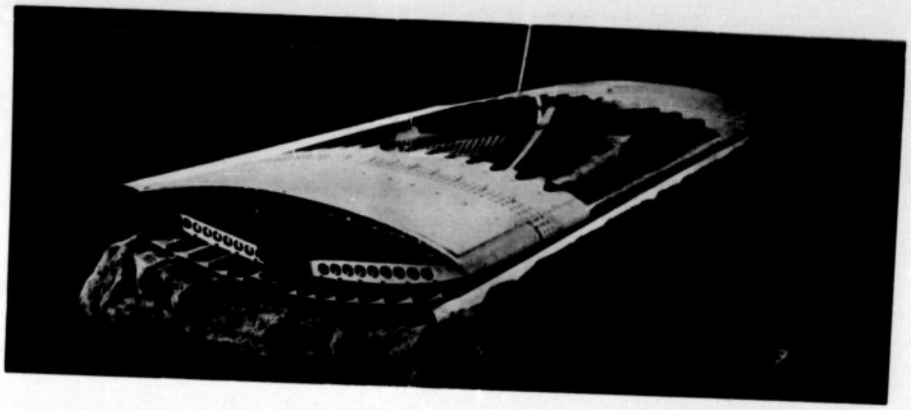
FIG. 10.



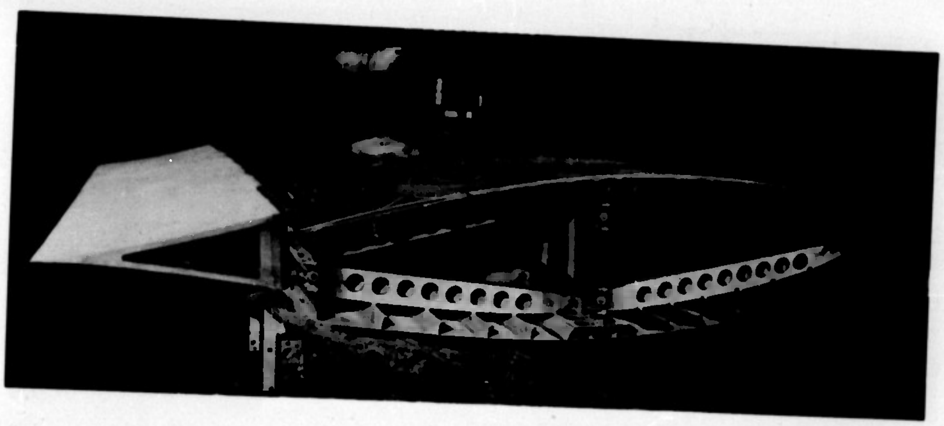
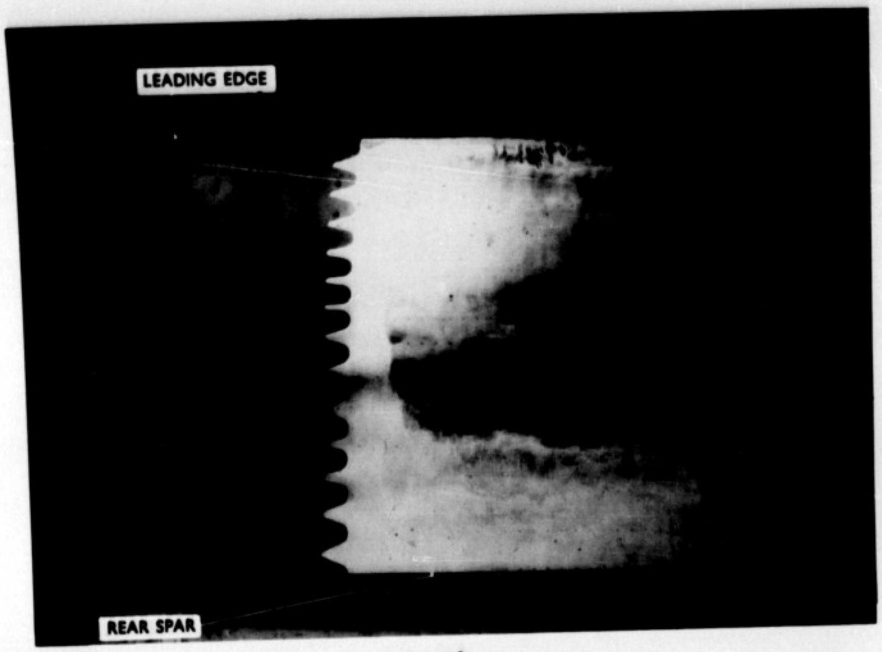


FIG. 11. TEST SECTION OF WING, SHOWING POLISHED SKIN AND INTERNAL STRUCTURE

R.A.E. NEG. NO. 71832 /16.

FIG. 12.

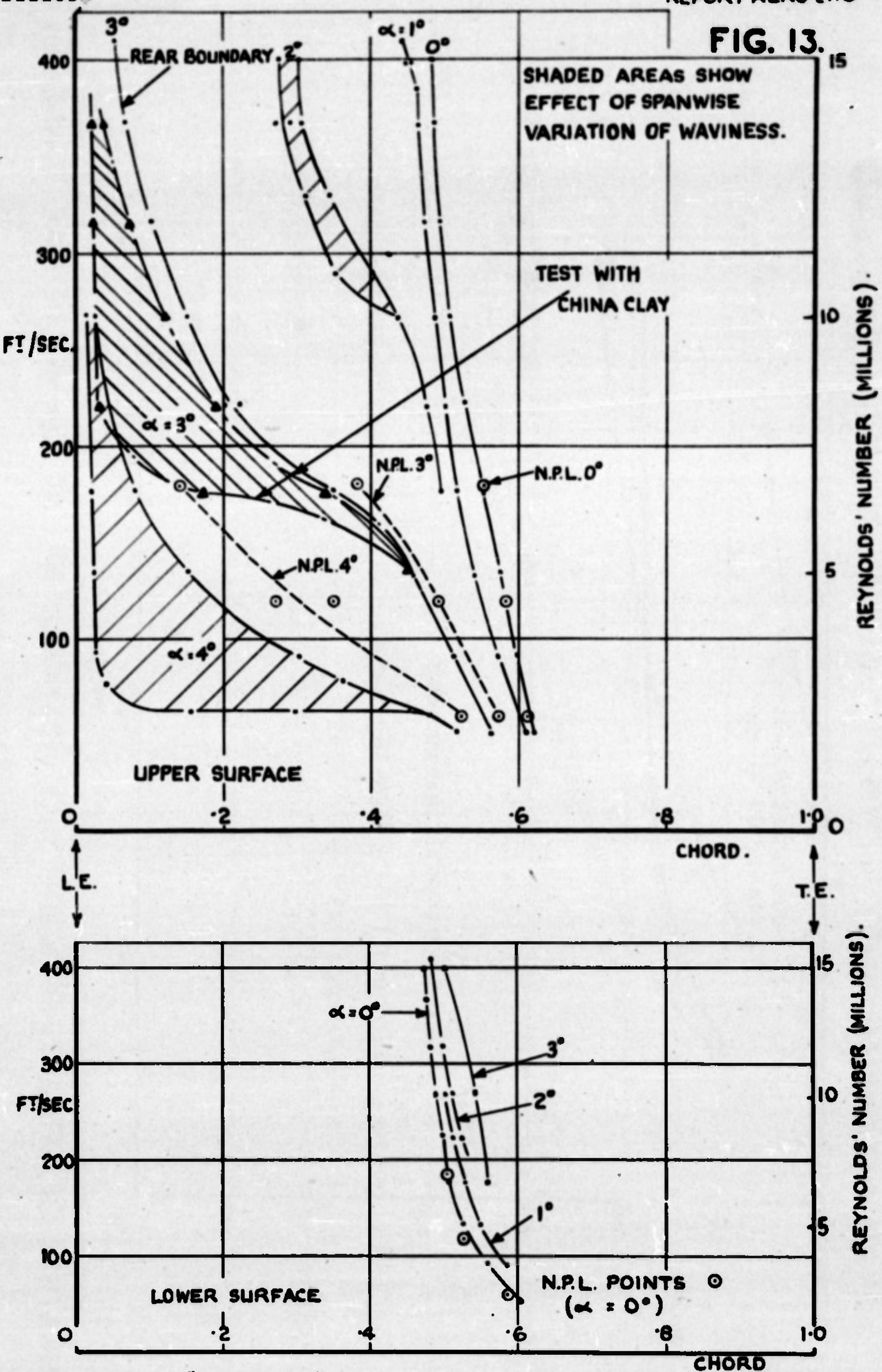


$\alpha = 0°$

$\alpha = 2°$

FIG. 12. EXAMPLES OF TRANSITION FRONTS FROM N.P.L. TESTS

FIG. 13.



TRANSITION "FRONT" ON BOULTON PAUL WING IN THE R.A.E. No. 2-11½ FT. WIND TUNNEL.

No. 19689 S.

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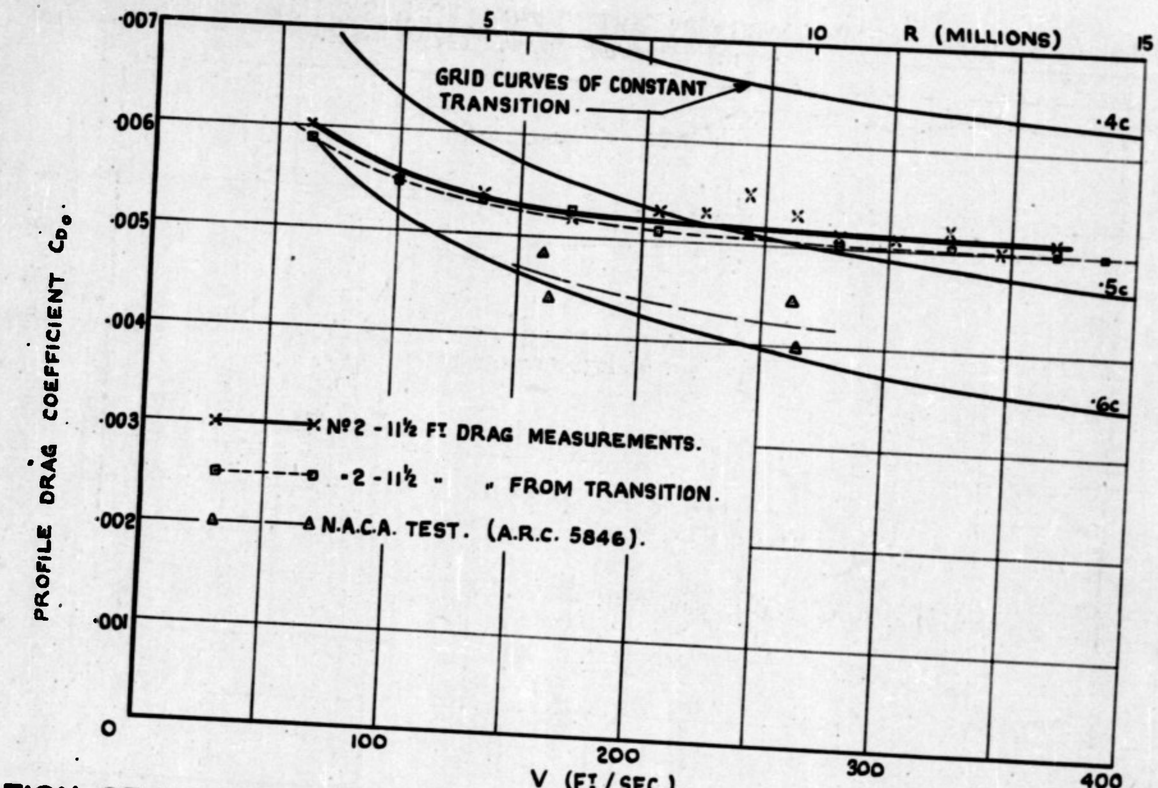
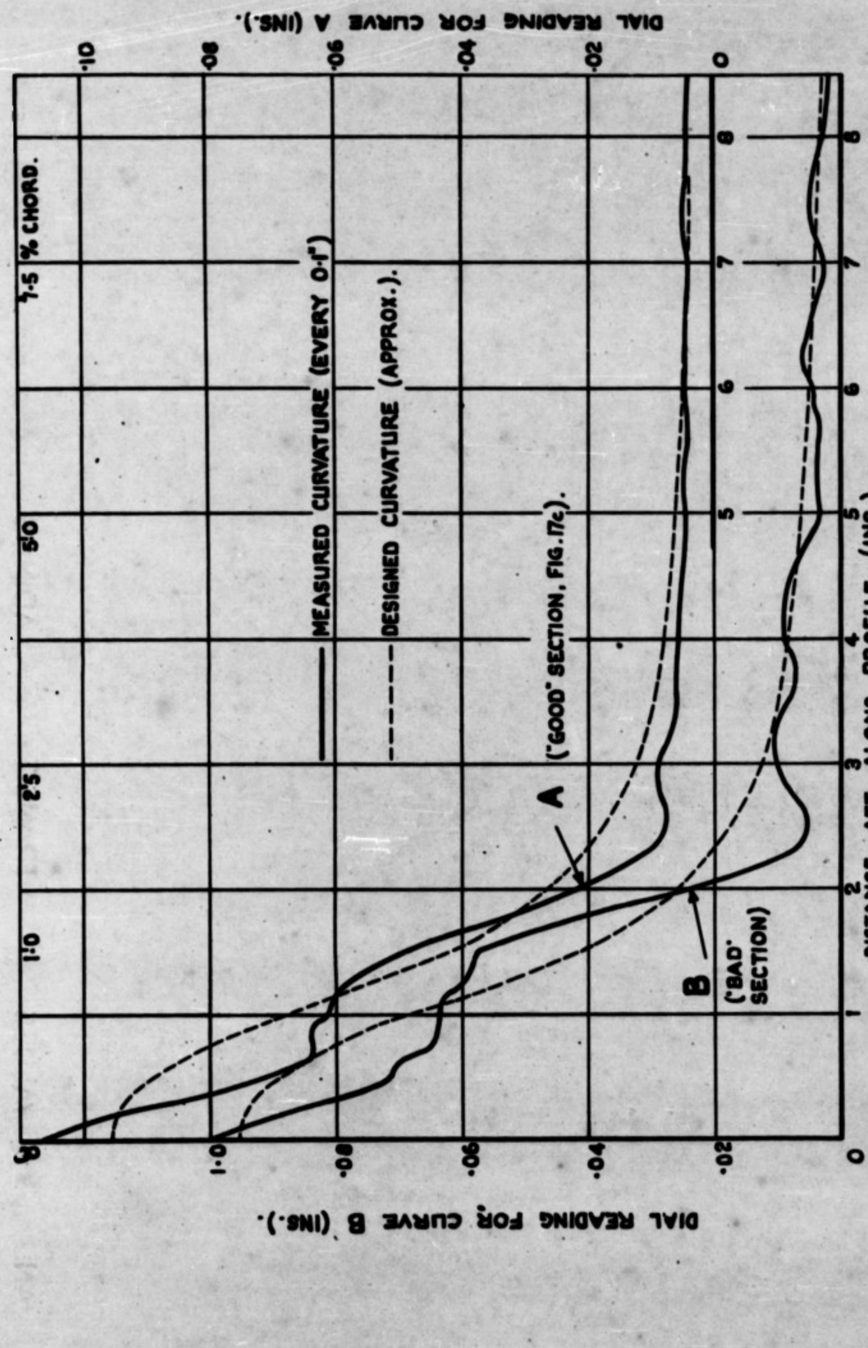
FIG. 14.



VARIATION OF DRAG COEFFICIENT WITH REYNOLDS' No AT $\alpha = 0°$ IN R.A.E. TUNNEL, AND DRAG VALUES FROM TRANSITION READINGS (ESTIMATED BY TETERVINS METHOD).

No. 19690 8.

FIG. 15.



VARIATION OF CURVATURE WITH DISTANCE FROM L. EDGE MEASURED ALONG SURFACE (UPPER).

(1·0 INCH GAUGE LENGTH).

No. 19691 S.

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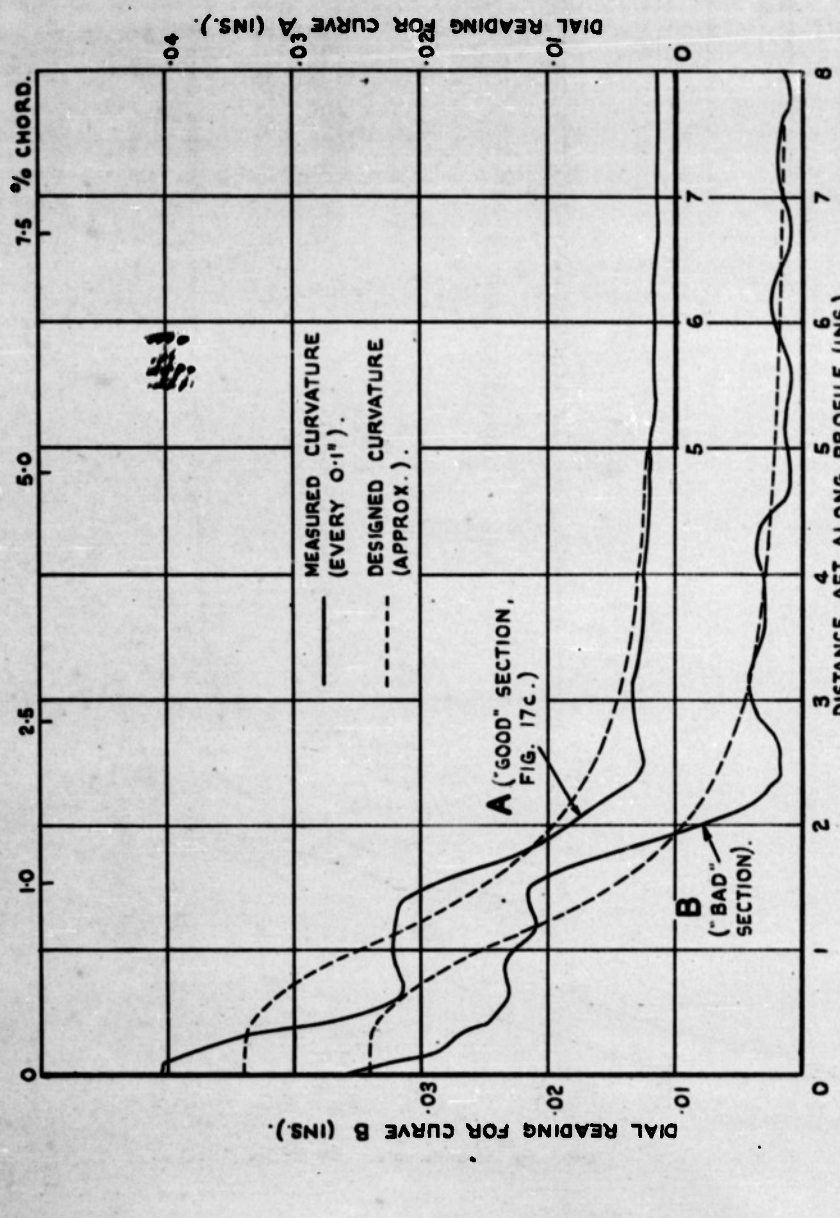
FIG. 16.



VARIATION OF CURVATURE WITH DISTANCE FROM L. EDGE MEASURED ALONG SURFACE. (UPPER).

(0·6 INCH GAUGE LENGTH).

34

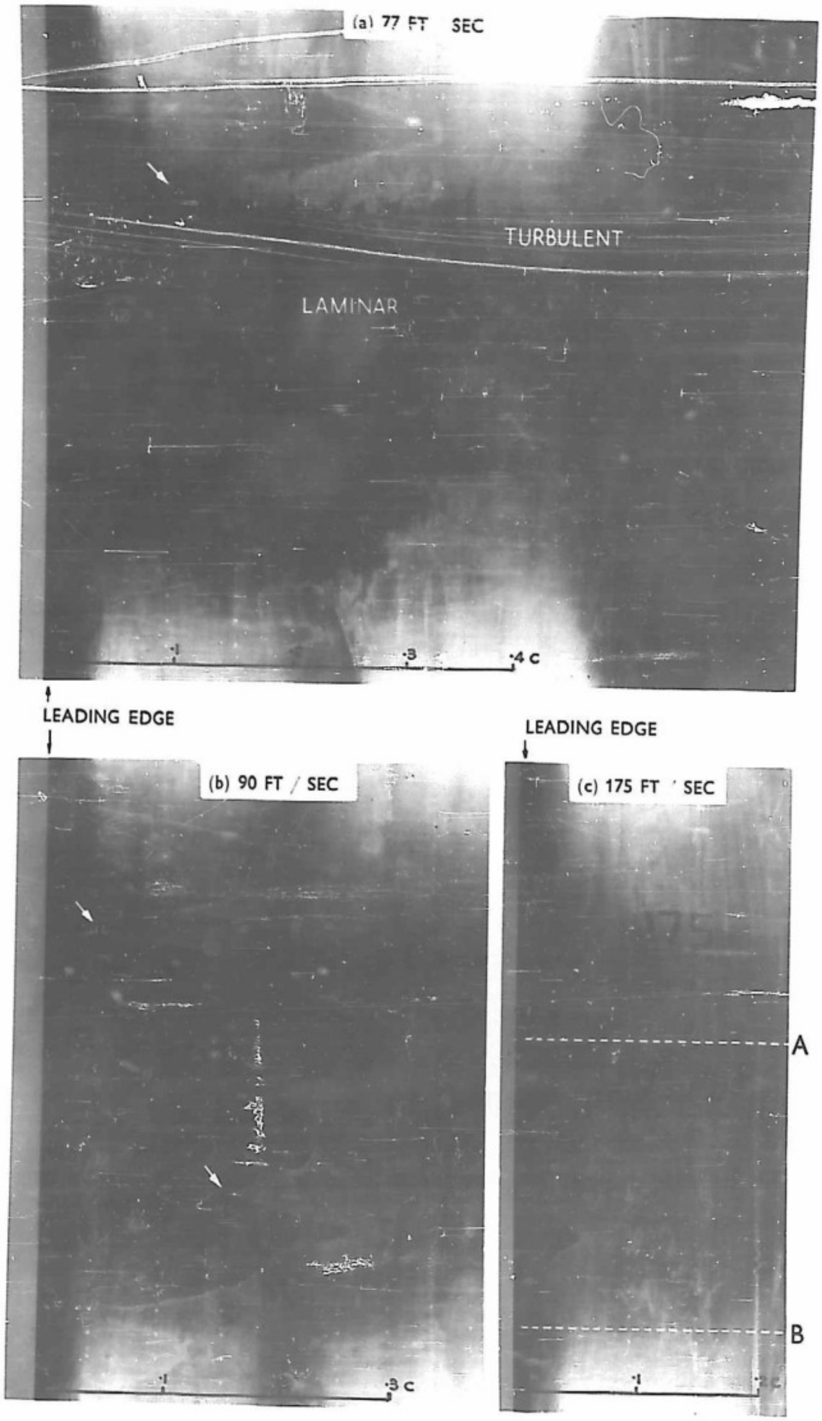


FIG. 17. R.A.E. TESTS AT $\alpha = 4°$ UPPER SURFACE TRANSITION

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9619

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ORIG. AGENCY : Royal Aircraft Establishment, Farnborough, Hants	ORIG. AGENCY NO. Aero-2170
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Nov '46	Restr.	Gt. Brit.	English	35	photos, tables, graphs

ABSTRACT:

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