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THE COLLEGE OF AERONAUTICS
CRANFIELD

FLIGHT TESTS OF A LAMINAR FLOW SWEPT WING
WITH BOUNDARY LAYER CONTROL BY SUCTION

by

R. R. Landeryou and R. S. Trayford

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CORRIGENDA



- Notation 8th line delete "spanwise"
 11th line " R_{al} " should read " $R_{\theta al}$ "
- Page 13, line 22 Delete ". . an upper critical . ." and insert
 "momentum thickness" after leading edge.
- Appendix 3, Page 19 5th line $\frac{W\theta}{r}$ should read $\frac{W\theta}{\nu}$
- Page 20 5th line "wave" should read "wing"
- Table 1 3rd line from bottom "16.99 x chord" should read
 Maximum thickness = 0.1699 x chord.
- Figure 7 The top of the photograph is on the left hand side of
 the page.
- Figure 38 (1) The title should read:
 "Calculated laminar profiles on infinite sheared wing".
 (2) $\frac{U_{\infty}}{V} = 1.9 \times 10^6$ per ft. should read $\frac{U_{\infty}}{\nu} = 1.9 \times 10^6$ per ft.


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SUMMARY

Flight tests have been performed on a laminar flow swept wing with 43° sweepback having slitted surfaces. The aim of the experiment was to achieve complete laminar flow at a unit Reynolds number of $1.5 \times 10^6 \text{ ft}^{-1}$ which corresponds to 10.5×10^6 based on the wing geometric chord. The test wing and installation is described and the progress of the tests so far performed is reported.

The most significant result to date has been the establishment of a critical Reynolds number above which turbulent disturbances propagate along the attachment line. It has been found that turbulent disturbances will propagate along the attachment line when conditions are such that the momentum thickness Reynolds number of a laminar boundary layer flowing along the attachment line would exceed 88. This has so far prevented the achievement of laminar flow at a unit Reynolds number of $1.5 \times 10^6 \text{ ft}^{-1}$. The critical Reynolds number of 88 corresponds to a Reynolds number based on leading edge radius of 1.15×10^5 or 8.37×10^6 based on the streamwise chord.

The progress of attempts to prevent the spread of these disturbances is reported.

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CONTENTS

	<u>Page No.</u>
Notation	
1. Introduction	1
2. The wing and laminar flow control system	1
Wing geometry	1
Wing construction	2
Suction zoning and flow control	2
Suction system	3
3. The aircraft	3
Description	3
Pressure error calibration	3
Performance	3
4. Systems operation	4
Suction engines	4
Fly protection gear	4
Traversing instrument carriage	5
5. Instrumentation	6
Manometer	6
Pitot combs	7
Microphones	7
Hot films	7
6. Results	9
6.1 Flight tests	9
Suction flow	9
Incidence measurement	9
Velocity profile results	10
Hot film results	10
Leading edge results	11
6.2 Attachment line contamination and its suppression	12
Bumps	12
Bead	13
Boundary layer fence with forward facing slot	14
6.3 Momentum thickness and drag measurement	14
7. Assessment of experimental flights	15
8. Conclusions	15
Acknowledgments	16
References	16
Appendices	
Table	
Figures	

Notation

p	local static pressure
q	free stream dynamic pressure
x	surface distance chordwise
c_{∞}	local chord in free-stream direction
U	local streamwise velocity
U_{∞}	freestream velocity
R	Reynolds number
ν	spanwise kinematic viscosity
θ	momentum thickness in a streamwise direction
θ_{al}	momentum thickness along the attachment line when boundary layer is laminar
R_{al}	Reynolds number based on attachment line velocity component and momentum thickness, both measured along the attachment line.
W	spanwise component of velocity
u	streamwise velocity component in the boundary layer
y	distance normal to the wing surface
ΔV_R	pressure error correction (knots)

1. Introduction

The original proposals for the flight testing of a laminar flow swept wing using boundary layer control were made by Handley Page Ltd., who have long been concerned with all aspects of laminar flow research. The proposal was to demonstrate full chord laminar flow under flight conditions on a wing with about 45° sweepback as the next step towards building a laminarised aircraft.

The College of Aeronautics had been engaged upon boundary layer flight research using wings mounted as dorsal fins on an "Avro" Lancaster aircraft. Ministry of Aviation contracts were granted for Handley Page to design and build a swept wing with suction applied to the wing surface and for the College of Aeronautics, in conjunction with Handley Page, to perform the flight tests on the Lancaster.

The flight tests were designed:

- (1) To demonstrate the suppression of two dimensional and three dimensional instabilities of the laminar boundary layer over the full chord or nearly the full chord of the wing, to establish the residual wake drag, the overall suction flow rate and the suction distribution required.
- (2) To investigate the tolerance of the laminar flow wing to incidence changes.
- (3) To determine the efficiency of a practical slitted surface.
- (4) To bring to light and if possible to investigate any unknown difficulties associated with boundary layer control on such a wing and with the type of slitted suction surface adopted.
- (5) To study the effects of excrescences and surface waviness.
- (6) To determine the validity of the design methods including theoretical methods for calculating the laminar boundary layer development and stability. Also to determine the success of the manufacturing techniques and materials and the design features embodied in the wing and to influence their further evolution.
- (7) To investigate the influence of acoustic disturbances on laminar flow.

2. The wing and laminar flow control system

Wing geometry

The test wing was mounted above the mid-fuselage of the Lancaster. The leading edge is swept at $42^{\circ} 35'$ and is 13 ft. long, the root chord and tip chord are 100 inches and 68 inches respectively and the span from the root fence perpendicularly to inboard of the non-laminarised tip is 100 inches (See fig. 1). The section is a symmetric RAE 103 section with a thickness chord ratio of 12.5% in the direction of flight. The suction surface on the starboard side is more conservatively designed than that on the port side, having a smaller slit spacing. It has approximately 35%

more slits (see Table 1). Because of the symmetry of the wing it is possible to make a direct comparison between the effectiveness of the two designs. Spanwise stations were defined, port and starboard, in inches normal to the root e.g. P50 and S75.

Wing construction

The wing surface is divided into a leading edge, a main panel extending from 7 per cent to 55 per cent chord and a rear panel. Thus on each side there are two spanwise joints, but there are no chordwise joints.

Primary loads are taken by a series of spanwise shear webs with angle booms, joined by cover plates forming a bolted light alloy box structure. Outside of this is added a series of close tolerance angle sections carrying a thin skin overlaid by an outer skin attached to Z members in line with the main shear webs (Fig. 2).

The main shear webs are cut away locally to receive the collector ducts which are flat in section. The Z members supporting the outer skin have raised bosses which were fettled before the final assembly of the outer skin to provide an accurate outer profile.

The leading edge (forward of 7% chord) is made of moulded glass cloth 0.25 ins thick in which spanwise grooves were milled and fitted with slitted square stainless steel tubes. Throttling tubes are situated under these metal inserts to ensure an adequate pressure drop through the surface. The air passes along a duct running parallel to the spanwise generators into a needle valve and is discharged into the central leading edge collector duct. There is a set of ten needle valves at each of three spanwise stations and the same Bowden Cable operates one of each set of ten against a return spring. The whole leading edge unit with valves and ducting is bench assembled commencing from the outer profile. The total leading edge suction air is metered through one venturi meter. No metering is provided for the individual needle valves.

Suction zoning and flow control

There are seven suction zones. The leading edge forward of 7 per cent forms one zone and aft of the leading edge the suction surfaces are divided into inboard, middle and outboard zones on both port and starboard sides. The inboard zones are from station 0 to 40, the middle zones from 40 to 70 and the outboard zones from 70 to 100 labelled respectively A, B and C (Fig. 1).

The flow is first drawn into the wing through a series of 0.005 inch wide spanwise slits and then into small channels running under the slits. To ensure a more uniform distribution of inflow it then passes through throttling holes, into a low velocity compartment and through the compartment's needle valve into the zone collector duct. In each of zones A, B and C there are 13 needle valves, the aft-most two valves being ganged together. These needle valves are installed in line spanwise and are operated by means of 13 sets of 3 concentric tubes. The outer tube of each set operates a valve in zone A, the middle tube that in zone B and the inner tube that in zone C. Thus it is possible to reduce the number of actuators to 12 and to situate the actuators remotely where they are accessible and can be

coupled to the valves in any zone to provide fine control across the chord. Each needle valve is provided with pressure tapings from which the mass flow through each compartment can be measured. The total flow from each zone is passed through a venturi meter so that the mass flow can be measured. Control of the zone mass flow is by butterfly valves, which are hand operated. They are situated downstream of the venturi meter.

Suction system

From the butterfly valve the flow passes into one of the suction engine's plenum chamber. Two sets of butterfly valves are installed so that suction can be applied from either or both of the gas turbines. If the butterfly valves were all shut too far the engines would surge or overheat but for the provision of an automatic valve which operates when the plenum chamber pressure drops below a pre-set fraction of atmospheric pressure. This admits air direct from outside the aircraft. The engines are started with the automatic valve open.

The two suction engines are standard Budworth Mark 1 units, modified to permit running with a large inlet depression. For this purpose the engines have a single stage centrifugal compressor and a single stage inward radial flow turbine, mounted back to back and surrounded by an annular return flow combustion chamber. Starting is by means of a 12 volt starter motor, and a pilot atomising jet is provided for initial ignition in conjunction with a conventional high energy system.

3. The aircraft

Description

The aircraft, an Avro Lancaster Mk. VII, Serial Number PA474, was a World War II type heavy bomber with mid-wing, twin fins and four Rolls Royce Merlin engines. The all up weight of the aircraft at take-off under test conditions was 49,800 lbs, which included 1850 lbs. of test wing.

Pressure error calibration

A static pressure error calibration using the trailing static technique was made. The result is shown in Fig. 4 where it can be seen that the operation of the suction engines did not affect the calibration curve and that changes of incidence of the test wing to within $\pm 5^\circ$ were just significant. The aircraft static to which the comparison was made was located just forward of the rear door.

Performance

The speed of the aircraft was limited to 200 Kts E.A.S. or a mean chord Reynolds number of 14×10^6 , in order to ease the design cases involving the test wing supports. All the test flights except for one special flight to be noted took place at 10,000 ft. altitude. The tests were flown in conditions of low turbulence and away from any possible effects of cloud or icing. The aircraft maintained good serviceability for the whole period of fifty flights (average duration one hour forty minutes) and aircraft serviceability was not a factor in causing flight delays whereas weather especially low ground visibility and low cloud base was such a factor.

Stall tests in various configurations were made to check any possible effects arising from the wing tip fairings which enclosed the television camera, (Fig. 5). These tests showed that the aircraft behaved normally.

4. Systems operation

Prior to the aerodynamic tests approximately ten flights were made with a dummy leading edge installed in order to test the main experimental systems. These included safety and reliability tests of the Budworth suction engines, satisfactory operation of the fly protection gear and in-flight use of the traversing carriage.

Suction engines

During the systems tests the Budworth engines would not light at equivalent air speeds greater than 110 knots. This was due to the high air speeds through the combustion chambers caused by ram pressure at higher aircraft speeds. The air intakes to the engines were modified by cutting them back so that the projected area normal to the direction of flight was considerably reduced. Some other minor replacements of fuel bellows and combustion jets had to be carried out at the same time as the engines were difficult to start or would not accelerate.

However, after the commencement of the aerodynamic tests the Budworth engines operated completely satisfactorily, and consistently gave the required mass flows without fluctuations in pressures or jet pipe temperatures. It was found that one engine was sufficient for the mass flow requirement at 150 knots if the indicated engine speed was 3700 r.p.m. (actual turbine r.p.m. being 41,800) and for most of the tests the engines were run at this speed, the butterfly valves being used to change the zone mass flows.

Fly protection gear

This gear, as originally fitted, consisted of a sheet of Melanex plastic wrapped around the leading edge to 7% chord. Between the sheet and the leading edge a small tube, sufficient to draw a cutter through, was layed from tip to root. Upon being triggered the cutter, driven by a constant tension spring motor, was pulled from the holding position at the tip to lodge at the root end of the leading edge. A razor blade in the cutter body split the sheet into two (taking about 2 secs) and the halves were shed downstream (Fig. 6).

As the flight tests proceeded it was found that the system had some intrinsic faults as well as those occasioned by the experimental nature of the tests.

It was found that holding the cutter in the ready position, by means of a split pin which was drawn out under the spring motor load, was not safe, since this could occur prior to triggering. The cutter was then free to dislodge the tube and sheet and work its way back over the wing to flail in the airstream, thus hazarding the wing surface. A cure was sought by holding the cutter in position with a solenoid. This modification was not entirely satisfactory as on one occasion the cutter was not released, thus causing the flight tests to be abandoned without any data being obtained. This was later rectified by incorporating a roller support for the solenoid pin.

In the rigging of the gear care had to be taken to see that the sheet was fitted loosely enough to allow the cutter to move along the leading edge against the imposed air loads. Also it was necessary to take care that no adhesive tape remained after shedding.

In shedding, the Melanex sheet was liable to hit and at times damage parts of the traversing gear and its pitot comb, as well as other pitot combs and static tubes disposed over the surface.

The attachment of the glass fibre slitted leading edge to the main wing necessitated the incorporation of a filler line which was subject to ageing and cracking. The presence of adhesive tape near the filler line to hold the Melanex sheet down aggravated this trouble. It was discovered that the Melanex would shred into small pieces naturally if the tape came free when the cutter had not been operated and this fact was used as the basis for a different system.

In the new system the Melanex was held down with a small amount of suction obtained from the aircraft instrument suction pumps. The suction was turned off after take-off and the Melanex then proceeded to shed. This was usually completed in about ten minutes. The advantage of this method was that the size of the pieces shed was smaller than the previous method, namely, a few square inches.

During taxiing before take-off three large strips of masking tape were used across the sheet to prevent gusts of a following wind lifting it clean away. These were removed by hand after the aircraft was lined up into wind.

While this appeared to be quite practical when applied to the leading edge without instrumentation it was difficult to hold the sheet on when the leading edge had experimental gear such as hot films fixed to the surface. On these occasions the system was dispensed with when possible.

From observations on the leading edge after landing in the summer months there was no doubt that protection against insects was necessary, but in the autumn it was found to be unnecessary.

Traversing instrument carriage

A traversing instrument carriage was provided with the wing. It could be traversed along a given chord line from 15% chord to near the trailing edge at any spanwise station from Station 15 to 90.

The carriage was tested during the systems tests and found to be unsatisfactory, as it lifted one or other of its wheels off the surface during operation in flight when carrying pitot instrumentation.

To overcome this the carriage was completely redesigned as a fully articulated structure with the minimum of restraints in its action (Fig. 7). The new design follows the contour of the wing and tracks the required chord line accurately. The attached pitot comb lifts away from the surface on the forward moving traverse to prevent the lowest pitot possibly fouling the edge of a slit. The instrument carriage can also carry two hot films.

The traversing gear allows the comb to move between approximately 15% and 73% chord. The forward limit is a little further aft for stations nearer the root. The 10 pitot tubes arranged logarithmically from 0.008 ins. to 2 ins. above the surface. The traverse chordwise position is indicated by a digital voltmeter.

The pressure field around the traversing carriage was obtained at the Handley Page low speed wind tunnel. This pressure distribution was used to choose positions for the pitot comb and hot films on the carriage, the pressure perturbation due to the carriage at the position of the pitot comb was approximately 0.02 q.

5. Instrumentation

The first main requirement for instrumentation was for pressure measurements to be made from the needle valve and venturi meters in the wing suction system and from flush static pressure holes located in the wing surface at the mid-station of the span (Fig. 8).

The second was for a satisfactory method for determining the state of the boundary layer at fixed points on the surface and along the chord traversed by the carriage. Initially velocity profiles were determined from pitot combs for this purpose but on account of its slow response such instrumentation was incapable of detecting intermittent turbulence. Moreover, where the boundary layer was very thin the profile shape could not be measured with sufficient accuracy to determine whether it was laminar or turbulent. To overcome these deficiencies hot film probes capable of detecting intermittent turbulence were developed.

Manometer

A 150 tube manometer of unusual design was constructed for the flight tests. The vertical tubes were made from cellulose acetate multi-bore strip and the connecting reservoir was made from perspex. While the design proved adequate when the fluid used was water coloured with cochineal it was difficult to find a suitable fluid of higher specific gravity which would not attack these materials.

At high suction mass flows the pressures from the tappings of the main venturi meters were equivalent to a head exceeding 60 inches of water. This was greater than the height of the manometer and a manometer fluid with a specific gravity of about 2 was required.

A mixture of iso-propanol alcohol and "Florube", a fluorinated oil, in proportions to give a specific gravity of 1.7 was found suitable and had a convenient viscosity. The mixture was coloured with Bromo-crysol green which was introduced into the alcohol before mixing. This mixture attacked the cellulose acetate by causing it to shrink but at such a slow rate that the manometer banks still had a useful life.

The manometer had a perspex front which allowed the various tubes and banks to be labelled before flight with a grease pencil. Comments could be added

in flight. The manometer together with a digital voltmeter, an air speed indicator and an altimeter was photographed with a remotely controlled 35 mm camera through a mirror placed on the opposite side of the fuselage, (Fig. 9).

Pitot combs

The condition of the boundary layer at any point on the wing was found from measurements of the boundary layer velocity profile. These were determined with the aid of pitot combs, which were made in two forms; the first, found useful from 25% chord to the trailing edge, consisted of a vertical stack of 10 stainless steel hypodermic tubes 1 mm to 2 mm (outer diameter) spaced logarithmically from 0.008" to 4" above the surface.

The second type consisted of a flat array swept to conform with the wing generators using 10 tubes of 0.012", $\frac{1}{2}$ mm and 1 mm (outer diameter) spaced in height from 0.005" to 0.18" above the surface. The heights refer to the clearance plus half the effective diameter. The lower tubes of both combs were flattened and ground. The lowest were arranged to touch the surface (Fig. 10).

In plotting the profiles obtained from these combs use was made of a suggestion by Mr. J. B. Edwards. When $\log y$ is plotted against u/U the profile can easily be recognised as being laminar or turbulent, in much the same way as when u/U is plotted against y/θ , where θ is the momentum thickness. $\log y$ could be easily assessed and a rapid method of determining u/U from the manometer photographs was available, also due to Mr. J. B. Edwards, so it was possible to analyse large numbers of profiles from each flight. (See Ref. 2 and Fig. 11).

The state of the boundary layer was however not known while the flight was in progress except for a qualitative guide obtained from viewing the manometer. Also in practice because the boundary layers near the leading edge were very thin it was not possible to obtain results forward of 15% chord with confidence. The pitot combs were displayed in sequence according to the height above the surface. If during a speed or suction change the boundary layer became laminar, the outermost water heights in two or three tubes fell to freestream values while the innermost tube heights rose slightly. This event could be correlated with a turbulent to laminar change recorded by hot films located nearby.

Microphones

In an attempt to obtain transition results during flight three microphones each provided with a boundary layer pitot-type probe were tried. The high background noise of the aircraft engines impeded listening and the method was not persevered with because of the development of the use of the hot film.

Hot films

When it was apparent that the state of the boundary layer in the forward region of the wing could not be adequately determined from pitot comb measurements another method was needed. Flow visualisation was dismissed because

of the difficulty of application, the risk of slit clogging and the further difficulty of interpretation of the pattern near the leading edge.

Hot wires were used initially but after a few flights hot films were preferred as being more robust. The use of hot films was first described by Ling in Ref. 3.

As only a qualitative knowledge of the boundary layer was required the circuit was of the simple uncompensated type fed from a clean 24 volt supply. No balancing was employed and a safe current was nominally set by fixed resistors plugged in before flight. The signal was viewed on the 1 milli-volt cent. scale of an oscilloscope.

Viewed in flight, the laminar signal was a clean straight line while the turbulent signal was of the characteristic high frequency noise type. In these tests transition signals showed characteristic rectified peaks or spikes which increased in frequency until the fully turbulent signal was obtained. An arbitrary scale from 0% to 100% was used to describe the degree of intermittency observed in transition. An intermittency of 50% was set as the point where the peaks followed one another in a regular sequence (see Fig. 12, 13).

Two patterns of hot film were made at the College of Aeronautics and are illustrated in Fig. 14. Type 1 was slightly easier to make as it did not have a bevel but type 2 gave increased amplitude and a more uniform signal and was preferred. The material used was glass microscope slides cut to size and ground with successive grades of silicon carbide and emery paper until a high degree of finish was obtained. A commercial platinum solution was painted on, air dried, and then fired to a temperature of 740°C. The firing of the platinum film and soldering of the leads was controlled to give a cold resistance of between 10 and 20 ohms.

The films were attached to the surface with cellulose tape, as more satisfactory means would involve permanent materials likely to interfere with the surface finish. Towards the leading edge the films were stuck with Araldite to phosphor bronze strips which when taped to the surface allowed the film to be sprung on to the highly curved areas near the leading edge.

While there was a steady loss of hot films due to ageing, accidental burnout and breakage, some films survived over 15 flights and on some flights there were no losses. Except at the leading edge, where the boundary layer was thin, the hot film gave clear and characteristic signals which could be interpreted with consistency. Near the leading edge the signal level was low and it was found necessary to check the operation of the hot film by the use of a trip which could be removed in flight.

An initial drawback to the use of a large number of hot films was the inability to view more than one signal simultaneously. If the transition front at various speeds was to be defined more closely up to fifty channels were needed and a simultaneous indication of the state of each signal would result in a considerable saving of flight time.

A system is now under development that will allow the indications of fifty

hot films to be presented on a bank of fifty meters scaled to show the percentage of intermittency present in each channel. Each hot film is bridge balanced to achieve the desired overheating ratio of 1.7 and the signal amplified by a packaged amplifier adjusted in frequency response (Figs. 15, 16). The signal is then rectified and at the same time amplitude limited, and is finally presented on a milliammeter in a display bank which can be photographed.

So far differences in the basic signal caused by non-uniformity in the hot films and variable positioning on the wing have not enabled the system to work as expected. From recordings of signals obtained in flight it is hoped to modify the circuit to give acceptable and reliable readings for future work.

6. Results

6.1 Flight tests

Suction flow

To set up the required design mass flows in flight in the various sections of the wing one of the Budworth engines was run up to 41,800 r.p.m. and the butterfly valves were adjusted using the venturi pressures (see Fig. 17). Venturi pressures were measured from the manometer as heights of water, and using pre-prepared charts were converted into mass flows. When the mass flows were approximately correct for each zone the mass flow of the zone to be set up on the given flight was adjusted compartment by compartment i.e. tuned. The tuning of the compartments was accomplished by actuating the needle valves located in each compartment. The mass flow passing through the needle valves was measured from pressure heights on the manometer and calculated from charts similar to those mentioned above.

To minimise the number of tuning operations the compartments requiring smaller mass flows were dealt with first. The operation was repeated until the individual compartment and total mass flows reached the required values. The tuning of a zone comprising thirteen compartments took up to twenty minutes in flight. After analysis of the flight records further slight adjustments could be made on the next flight. A typical result of this tuning process is shown in Fig. 18.

In the latter stages of the tests it was found desirable to check the distribution of flow into the leading edge slits. An attempt was made to perform this on the ground. A sliding jig carrying a hot wire and capable of traversing the full length of a single slit was made. A few slit traversing tests were conducted and these showed that no excessive variations existed.

Incidence measurement

To obtain zero aerodynamic incidence on the wing, flush static pressure holes at the mid-span and at 10% chord intervals were originally used (see Fig. 19). The pressures from the flush static pressure holes on the port and starboard sides of the wing were equalised by actuating the wing incidence mechanism to give zero incidence. The pilot's indication of aircraft yaw was obtained from a vane type instrument located on a boom mounted forward of the starboard

wing tip. The pilot flew keeping the sensitive yaw meter reading at zero throughout the run.

The incidence of the test was set approximately by an electric actuator and when finally adjusted an accuracy of $\pm 1/8^\circ$ was achieved. On occasions when the flush static pressure holes suffered interference due to the presence of experimental gear, the position of zero incidence was found by using external static pressure tubes mounted in pairs, one on each side at 7% chord.

These static pressure tubes were also used to check the presence of any span-wise variation of local incidence. Five sets of external static pressure tubes were disposed from 20% to 90%.

Apart from a single result at 90% span which indicated $\frac{1}{2}^\circ$ different from the others, all the remaining static tube pairs agreed to $\pm 1/8^\circ$ at zero incidence.

Velocity profile results

At the outset of the present work the vertical 4 in. pitot comb (see Fig. 10) was used to obtain velocity profiles. It was soon found that at 150 kt (unit Reynolds number $1.48 \times 10^6/\text{ft}$), even with suction applied, laminar flow was not present aft of 25% chord. It was necessary, therefore, to obtain profiles forward of this position and as the large combs did not have sufficient tubes within the thinner boundary layer, flat arrays of tubes were made as described in § 5.

By use of the 'log height-velocity profile' plotting method the mid-zone region of the starboard side with suction showed that at 15% chord laminar flow was present at 130 kt (unit Reynolds number $1.28 \times 10^6/\text{ft}$). At 10% chord laminar flow was present at 140 kt and at 7% chord it was possibly present at 150 kt, although due to the thinning of the boundary layer the small number of available readings made the assessment difficult at 7% chord. This method was lengthy in practice as the number of runs required to cover the necessary speed ranges and chordwise positions led to a large number of manometer photographs. Many of these were inevitably irrelevant as they were not near the transition range when analysed. Many profiles were not obviously laminar or turbulent when compared to the reference profiles. Some lay between two states but for them to be conclusively accepted as transition profiles a larger number of readings nearer the surface would have been necessary, such as points nearer than 0.008 in. to the surface. In Figs. 20 and 21 are shown typical results plotted on the 'log height' basis with suction on and suction off.

While some indication of the boundary layer going laminar as speed was decreased could be obtained by viewing the manometer bank, as described in § 5, a definite indication in flight was highly desirable to avoid delay whilst waiting for velocity profiles to become available, usually a day or two after the flight.

The majority of the readings obtained aft of 15% were taken from a pitot comb mounted on the traversing carriage.

Hot film results

To overcome the disadvantages of the velocity profile method of assessment, recourse was made to the use of hot films (§ 5).

It was found that the best method of obtaining reliable knowledge of the state of the boundary layer at a given point on the wing, was to commence at a low speed such that as the aircraft's speed was increased the state of the boundary layer changed from laminar, through transition, to turbulent. It is stressed that the reason for using this procedure was that the films varied in sensitivity and character of signal and were sometimes subject to electrical interference. It was necessary to see the laminar, transitional and turbulent signals in sequence before relying upon their interpretation. Repeated runs on a given flight established speed of onset of transition within ± 2 kt.

The traversing carriage was used to carry two films sprung on to the surface by spring strips. By this means a speed-chord envelope of transition was determined in the region covered by the traversing carriage (see Fig. 22). After each flight the spanwise position of the traverse was changed but at the same time on each flight use was made of fixed films arranged in a manner to avoid mutual interference. From all these results extending over a number of flights the transition front diagrams, Figs. 23 and 24, were established.

The results of the first forty-two flights are presented as transition contours over both surfaces of the wing (Figs. 25, 26). It can be seen that on the starboard side, laminar flow was maintained at 95% chord at 115 kt, the unit Reynolds number being 1.13×10^6 per ft, at the mid-station of zone B. In all of these flights the suction distribution was set for a speed of 130 kt and for an altitude of 10,000 ft. Overall increases of mass flow yielded no improvement. Lower suction quantities resulted in transition moving rapidly forward.

Further flight results as presented in the next section showed that the transition fronts originate at the leading edge and do not, as might be expected lay more or less along generators. Referring to Fig. 27 diagram B shows the expected result. However, if only the effect of the transition lines starting at the leading edge were to be counted the result would be as in diagram A. Combining these diagrams leads to diagram C which can be seen to be near the general shape of the transition fronts presented in Figs. 25, 26.

Leading edge results

Although it might be expected that the transition fronts would run more or less along spanwise generators near the leading edge, it has been found that they originate from the attachment line. A flight was specifically made to correlate the attachment line transition with the Reynolds number based on leading edge radius. To test dependance on Reynolds number the runs were made at three separate altitudes, 3,700, 10,000 and 22,000 ft. The result of this flight showed that the lower limit of transition occurred at a Reynolds Number of 1.15×10^5 based on the leading edge radius or 8.37×10^6 based on chord (see Fig. 28). This led to a critical momentum thickness Reynolds number at the attachment line of 88, based on the spanwise component of the freestream velocity and the momentum thickness of the spanwise flow at the attachment line. (see Appendix 3). Thus, at a particular point on the leading edge there is a speed at which the flow inboard is turbulent and the flow outboard laminar. This phenomenon on a swept back wing is caused by a disturbance originating from a point on the attachment line towards the root, propagating until it decays to laminar flow. The turbulent boundary layer on the root fence has a measured momentum thickness of 0.25 inches at a unit Reynolds Number of $1.25 \times 10^6 \text{ ft}^{-1}$ and in this case is the source of the initial disturbance.

6.2 Attachment line contamination and its suppression

To prevent disturbances contaminating the outboard flow along the attachment line various methods were suggested and they divide into two types. Firstly, complete removal of the turbulent boundary layer so that a new laminar boundary layer can develop and secondly, a local thinning of the boundary layer to give it a momentum thickness Reynolds number below its critical value. These methods include such devices as a chordwise slot or boundary layer bleed, a notch, a small region of area suction, several suction slits normal to the attachment line, a leading edge extension forming an external notch, reduced radius leading edge extension with faired ends and finally a shaped leading edge extension (a bump) designed to start a new attachment line boundary layer.

Bumps

The first device to be tried was a bump which consisted of a faired shape with a steeply rising upstream front followed by a gently sloping downstream surface (Fig. 29). This device was originated by Dr. M. Gaster of The College of Aeronautics, who expected to prevent the outboard propagation of disturbances initiated from inboard by the creation of a new boundary layer. The first bump to be tried, was positioned near the root and was larger than subsequent ones. The flow was turbulent a short distance outboard of the bump and it was presumed that its outboard surface faired too abruptly into the wing line. This was rectified on the next bump which, together with the following three were made of plasticine. On the next flight the bump was placed at 90% span and from 140 kt to 190 kt gave a signal of 10% intermittency or less, while below 140 kt the signal was laminar. Just inboard of the bump the flow was turbulent above 115 kts. This last transition speed was lower than normal because of a disturbance introduced by another device placed close inboard.

Similar results were obtained on the next two flights when small modifications to the shape and reduced roughness were tried with the object of eliminating the 10% intermittency but no definite improvement was found (Fig. 30), and no insight was gained as to the proper method for its elimination.

At this juncture a wind tunnel result obtained by Dr. Gaster on a simplified leading edge with a geometrically similar bump became available. This showed that in the same leading edge Reynolds number range as the 10% intermittency occurring in flight a fully laminar boundary layer could be obtained downstream on this type of bump. The apparent critical factor in obtaining this was surface roughness. Slight changes in roughness on successive runs led to markedly different results and only the best possible finish of the plasticine gave the above result. (The scale of the model was twice that of the test-wing).

To obtain as good a surface finish on the wing as on the model bump, a bump was made of fibre-glass paste and stuck onto the wing with commercial fillers as fairing material. A flight with this improved bump, during which the hot films were removed from the attachment line one at a time, revealed intermittent turbulence inboard of the fairing of the bump into the wing line. This result suggested that the front shaping of the bump was unsatisfactory. It is now appreciated that the bump tested in flight was not geometrically identical in shape with the bump tested

in the tunnel. An interesting observation was that the percentage of intermittency did not increase with distance along the attachment line from the bump.

Suction applied to the leading edge did not suppress the intermittent nature of the signal back to 7% chord.

A bump having a related shape to those tested can be derived in potential flow by placing a three dimensional point source on the attachment line of the basic wing (Fig. 31). It is interesting to see that no new stagnation streamline exists on the surface so that the conditions to start a new boundary layer are not realised. However it has been reported recently by Dr. Gaster, that on the bumps tested on the wing in the wind tunnel, a new stagnation point existed on the forward curved portion.

It may be that conditions did not exist to start a new boundary layer on all the bumps tested in flight. But there is little doubt that they should have produced a beneficial effect by thinning the boundary layer on account of the smaller leading edge radius on the bump.

It is possible that even if a stagnation point exists at some point on the upstream side of the bump disturbances may propagate across or around such a region. It has been found from wind tunnel tests that disturbances can propagate to the attachment line from all points in the region where the divergence of the flow direction from the attachment line is less than about 9° (Fig. 32).

From the combined wind tunnel and flight tests there is evidence to suggest that a bump, allowing a new boundary layer to form outboard, could be designed to have an upper critical leading edge Reynolds number less than that of the main wing. This could be achieved by slightly reducing the leading edge radius of the bump to compensate for its slightly increased outboard sweep.

Bead

The primary object of the bead was to obtain uncontaminated flow for a reasonable distance along the leading edge by reducing the leading edge radius to below the critical value. To prevent contaminated flow from reaching the surface of the bead, a lip was formed on the upstream end which was designed to divert rearwards the oncoming boundary layer. (Fig. 33).

On the bead, which was of constant radius, the hot films showed conclusively that laminar flow was present to 190 kts. or 1.87×10^6 unit Reynolds Number.

In order to fair the bead to the wing a small radius fillet (radius = 0.25 in.) and also a tangential fillet, both made of plasticine, were tested. The tangential fairing gave a better result and some films placed just downstream showed laminar flow up to 190 kts. However, further tests were not made as it appeared that the plasticine fillet gave variable results. The solution to this would have been to replace the plasticine with a fillet of better contour and surface finish, which would have required a fillet of a more permanent nature.

Naturally the reduction in leading edge radius leads to a more incidence sensitive nose shape and it is of importance that this be recognised when designing nose profiles to satisfy the attachment line criterion.

Boundary layer fence with forward-facing slot

The boundary layer fence was intended to isolate the attachment line flow from contamination at the root. It was mounted 4 inches from the wing root and had a forward facing slot on its outboard side. The slot, was formed by leaving a gap between the fence and a fairing super imposed on the basic wing leading edge (Fig. 34). The fairing had a constant radius approximately equal to that of the wing leading edge. The purpose of the slot was to enable the boundary layer on the outboard side of the fence to escape without contaminating the attachment line flow. To compensate for the blockage caused by the wings presence in the slot the gap was increased in depth.

The fairing was originally made of plasticine and with this the hot films showed turbulent bursts of up to 15% intermittency at 190 kts. For the same reason that the bump material was changed as explained above the fairing was changed to a constant radius metal tube section with a commercial filler fairing the downstream edge. The plasticine fillets at the sides were left unaltered. (Fig. 35).

Hot films as close to the slot as 2.5 in. on the attachment line gave 10% intermittency with the better surface finish. It is probable that the slot, which was not sucked, did not fully bleed the boundary layer. Alternatively, small disturbances in the region of the downstream end of the fairing may have grown rapidly to the quoted value.

With the radius of the fairing the same as the leading edge radius the device should not suffer from loss of laminar flow due to changes of incidence. Progress in making the device work may entail suction in the bleed or on the fairing.

6.3 Momentum thickness and drag measurement

Fig. 36 shows the band of experimentally measured momentum thicknesses for both laminar and turbulent boundary layers, compared with the theoretical predictions. A typical distribution momentum loss is shown in Fig. 37.

The ratio between the experimental turbulent and laminar values should not be taken as representative of conditions on an aircraft wing or even of expected drag reductions. The residual wake drag, represented by the momentum thickness at the trailing edge, is subject to interaction with the pump drag.

If laminar flow were maintained on an aircraft wing the increased chord would result in a greater saving in drag. This is because the turbulent boundary layer continues to grow whereas the sucked laminar boundary layer remains sensibly the same. However, the difference in the experimental and theoretical predictions of the momentum thickness for laminar flow needs further investigation.

The current flight tests have not proceeded far enough to warrant the measurement of drag at the trailing edge. Consideration of this measurement shows that because of the three dimensional nature of the wing it is necessary, if the drag of a particular spanwise zone is required, to surround the zone with a momentum "box". Therefore, not only are trailing edge wake traverses necessary but also samples of the complete boundary layer profile along both sides of the zone. This would include measurement of the cross flow profile and as can be seen from a typical theoretical profile illustrated in Fig. 38 this problem posed difficulties in measuring

both small angles and small heights from the surface. Measured cross flow profiles would also be useful for comparison with theory. While the measurement remains possible, no progress has been made to date.

7. Assessment of experimental flights

In considering the total of 50 flights to date, it should be noted that the first 20 were allocated almost entirely to proving out the aircraft and systems, and also that from the 20th to the 30th flight, systems were still handicapping aerodynamic tests. The last 10 flights were the most productive in terms of aerodynamic results. In retrospect much time was spent in early flights developing equipment and instrumentation which, although important in the long run, did not materially contribute to the discovery of the leading edge instability problem.

It may be argued that if a wind tunnel had been available for work in support of the flight tests much time could have been saved. It is at least open to question whether the attachment line contamination problem would have been discovered because if a small model of the wing had been tested it would probably have been at too low a Reynolds number for the phenomenon to occur.

8. Conclusions

In so far as the purpose of the flight tests outlined in the introduction have progressed the following has been achieved:-

1. The suppression of laminar boundary layer instability has been demonstrated over almost the full chord of the wing, but at a lower Reynolds number than the design case. The design mass flow distribution can be set up efficiently and held within the required limits. The suction system, including the Budworth gas turbine units proved to be reliable in operation.

The slitted surface at the flight speeds so far encountered appears to be satisfactory and presents no maintenance problems in the testing environment.

2. 95% chord laminar flow was obtained at a chord Reynolds number of 7.7×10^6 at the mid-section of the wing.

3. The wing as originally tested suffers from turbulent contamination propagating along the leading edge attachment line. Since the leading edge radius of the present wing decreases outboard it was found that the contamination decayed outboard and laminar flow was present when the leading edge Reynolds number was below 1.15×10^5 or can be related to the momentum thickness of the spanwise flow at the attachment line, which is equivalent to $R_{\theta_{a.1}} = 88$.

4. Leading edge devices were tested on the wing with the intention of overcoming the above limitation and it was found that although laminar flow could be maintained, over small radii beads at high speeds, other devices were relatively less successful in that turbulence of low intermittency persisted. In one flight a specially shaped bump also enabled laminar flow to be maintained up to high speeds and this was subsequently repeated in many wind tunnel tests.

Acknowledgments

Acknowledgment is made of the enthusiastic participation made by the pilot, crew and members of the Department of Flight at The College of Aeronautics and of the Departments of Aerodynamics and Design in this project. The continued effort shown by Handley Page Limited, especially the Research Department in conjunction with the division at Park Street, in meeting expeditiously all the demands made upon them is also acknowledged. Space alone prevents the names of all those associated with this research being recorded.

Grateful acknowledgment is made to Flight International magazine for permission to use the drawing appearing in Fig. 2.

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Appendix 1

Maintenance of the wing

Flying in the summer at Cranfield resulted in insects impacting on the leading edge on take off when the protective cover was not in place and on landing. The number of insects present on post flight inspection rarely exceeded a dozen and the remains were washed off by hand using a mild detergent solution. The height of the fly remains, in general, exceeded the attachment line boundary layer thickness.

The filler line between the leading edge and the main wing structure was a source of frequent trouble in that a good permanent smooth joint could not be attained. Standard techniques using rubbing down, commercial paint filler and paint were employed but the process of ageing and cracking with subsequent lifting of the paint surface continued. The problem does not appear to have a ready long term answer and it would seem that the best solution lies in designing suction wings to be free of filled gaps, if necessary, by coming to terms with small gaps or even sucking the gap.

The glass fibre leading edge had two maintenance problems during the tests. The first was the difficulty of moving the Bowden cable operated valves which after a time had a tendency to become restricted in total movement or to jam. This has since been rectified by a re-routing and shortening of the cable runs near the root of the wing. The second problem concerned small slivers of metal which were noticed to be protruding out of the slits after some flights halfway through the programme. These slivers were up to half an inch long and approximately 0.002 inches thick, thus being sufficient to cause turbulence. Their origin was traced to a burr on the inside of the slits which were cut in hollow stainless steel inserts. The manufacturing process included de-burring, but it had not been completely successful. It was believed that the possible pulsating flow and static charge build-up, brought about by the flapping and shedding of the Melinex plastic sheet during the early stages of the flight, had worked the slivers loose and brought them to the surface.

Metalised Melanex was used on subsequent flights and the problem has not been encountered again. The occurrence can be avoided in future construction.

On the two occasions when the fly protection gear cutter broke away and flailed for a short time over the rear part of the wing surface some damage in the nature of scratching of the surface occurred. This was rectified by rubbing down the small areas involved. No slits were damaged in this manner.

The cellulose tape used to attach pitot combs and hot films on to the surface left an undesirable residue behind, which itself collected dirt and eventually hardened. This had to be carefully removed with carbon tetrachloride to maintain a good surface finish. This suggests that it may be worth while to provide some fixing points on the wing surface for instrumentation.

While performing the tests there was no firm evidence that surface waviness had a detrimental effect in obtaining laminar flow, although, parts of the surface were not as free from contour discrepancies as would have been desired. This was particularly the case at the joint of the glass fibre leading edge to the main structure.

Filler was used here and was maintained throughout the test at as high a standard as possible but it is apparent that there is a flat at the joint.

APPENDIX 2

Flight operations

An assessment of the aircraft's handling qualities with a similar research wing mounted can be found in Ref. 4, and the behaviour with the present wing was not different.

The best working efficiency was found with the crew at a minimum number of five, which consisted of pilot, flight engineer, Budworth engine driver and two flight observers. Any increase restricted use of the intercommunication system and also the working space available at the rear of the aircraft.

The advantage of conducting experiments using a large unpressurised aircraft is well brought out by two of the techniques used in the flights. When the hot films apparently showed only laminar flow and it became necessary to prove that a turbulent signal was not present at the higher speeds then a trip line of 0.11 inch nylon fishing line was taped on to the wing ahead of the film or films and after turbulent signal had been observed the line was pulled off the wing into the aircraft. At the same time the change in the signal to laminar was noted on the oscilloscope. A development of this technique was used on the 50th flight when it was necessary to observe the progress of the propagation of turbulent intermittency downstream from the root along the attachment line, both ahead and behind devices designed to limit such intermittency. As flush hot films could not be used 9 films were held down on the surface with string taped to either side of the leading edge. During flight, the signal of the film nearest the root was observed and then the signal of the next film observed while the first was pulled off the surface. In this fashion the films were removed one at a time, leaving a clean surface. It may be remarked that the films, after the string had been removed, sometimes stayed firmly in position and had to be dislodged by yawing the aircraft. Also only one film of the nine was slightly damaged as the films lay back over the surface, attached by their leads. A photograph of the pre-flight arrangement is shown in Fig. 39.

APPENDIX 3

Attachment line contamination criterion

Pfenninger (Ref. 5) first suggested a criterion for the occurrence of turbulent contamination along the attachment line of a swept wing.

His suggestion was that the Reynolds number based on the momentum thickness taken normal to the leading edge at the attachment line and the spanwise velocity $\frac{W\theta}{r}$, if greater than 65 - 70 could provide conditions under which a disturbance, if sufficiently large, would propagate along the attachment line so causing turbulent contamination of large areas of the wing surface. This phenomenon had previously been detected by Gray (Ref. 6, 7) and Gregory (Ref. 8).

A more consistent definition is to take the momentum thickness of the attachment line boundary layer in the direction parallel to the attachment line. This results in a value of R_{θ} approximately 1.4 times that of the above and this is called $R_{\theta_{a.l.}}$. Edwards (Ref. 9) has expressed this criterion as a wing parameter dependent on wave profile shape, incidence, sweep and the chord Reynolds number. This parameter allows a calculation to be made on any given wing with regard to the possibility of contamination, which can be one of the following three types: laminar flow re-establishing itself on the attachment line outboard of large disturbances; laminar flow on the attachment line provided there are no large disturbances; turbulent flow on the attachment line even without large disturbances.

According to the above criterion the Handley Page wing has an $R_{\theta_{a.l.}}$ of 88. Work by Gregory (Ref. 10) and Gaster (unpublished) suggests that this may be extended by means of some of the external devices already discussed (para. 6.2). By this means $R_{\theta_{a.l.}}$ is raised to above 140 which would imply that the present test wing should exceed its design speed and also the limit set by the test vehicle. This can be inferred because

$$R_{\theta_{a.l.}} \propto \sqrt{\frac{U_{\infty} c_{\infty}}{\nu}}$$

where $\frac{U_{\infty} c_{\infty}}{\nu}$ is the streamwise chord Reynolds number, and thus for $R_{\theta_{a.l.}} = 140$, the Reynolds number based on wing chord $\frac{U_{\infty} c_{\infty}}{\nu} = 20 \times 10^6$.

TABLE 1

Compartments in wing

<u>Compartment</u>	<u>Chordal Position</u>			<u>Number of Slits</u>	
	<u>% chord</u>			<u>Port</u>	<u>Starboard</u>
leading edge	0	-	7.5	7	8
1	7.5	-	15	2	3
2	15	-	25	2	3
3	25	-	35	2	3
4	35	-	45	2	3
5	45	-	50	2	3
6	50	-	55	3	4
7	55	-	60	3	4
8	60	-	65	3	4
9	65	-	70	3	4
10	70	-	75	3	4
11	75	-	80	3	4
12	80	-	85	3	4
13	85	-	100	9	13

The compartments run in the direction of the generators. Except for the leading edge the compartments are separate from those having the same number in the other zones.

RAE 103 DATA (Normal LE)

Position of maximum thickness = .390 x chord
 " " = 16.99 x chord
 100 x Leading Edge radius = 1.82 x chord
 Trailing Edge angle = 19.19 Deg.

SUCTION ZONES AND DIMENSIONS

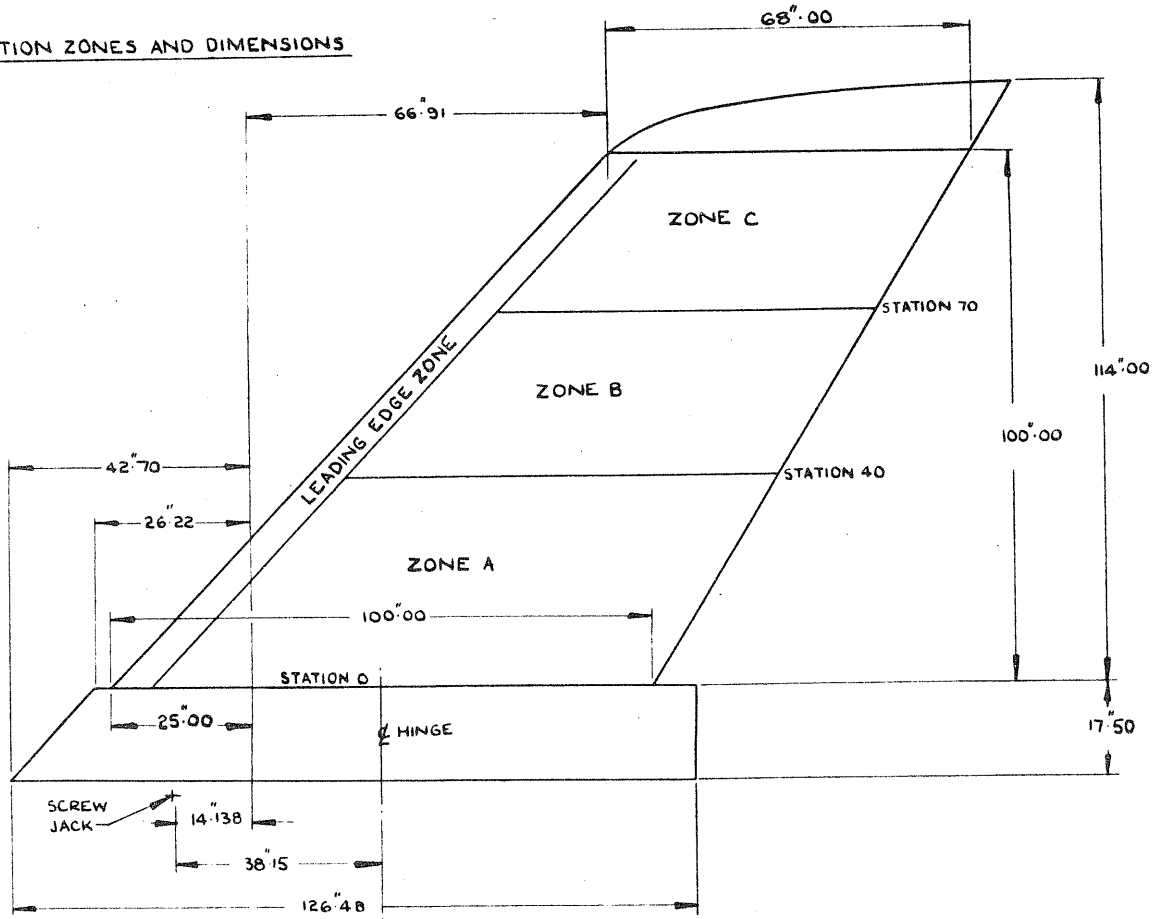
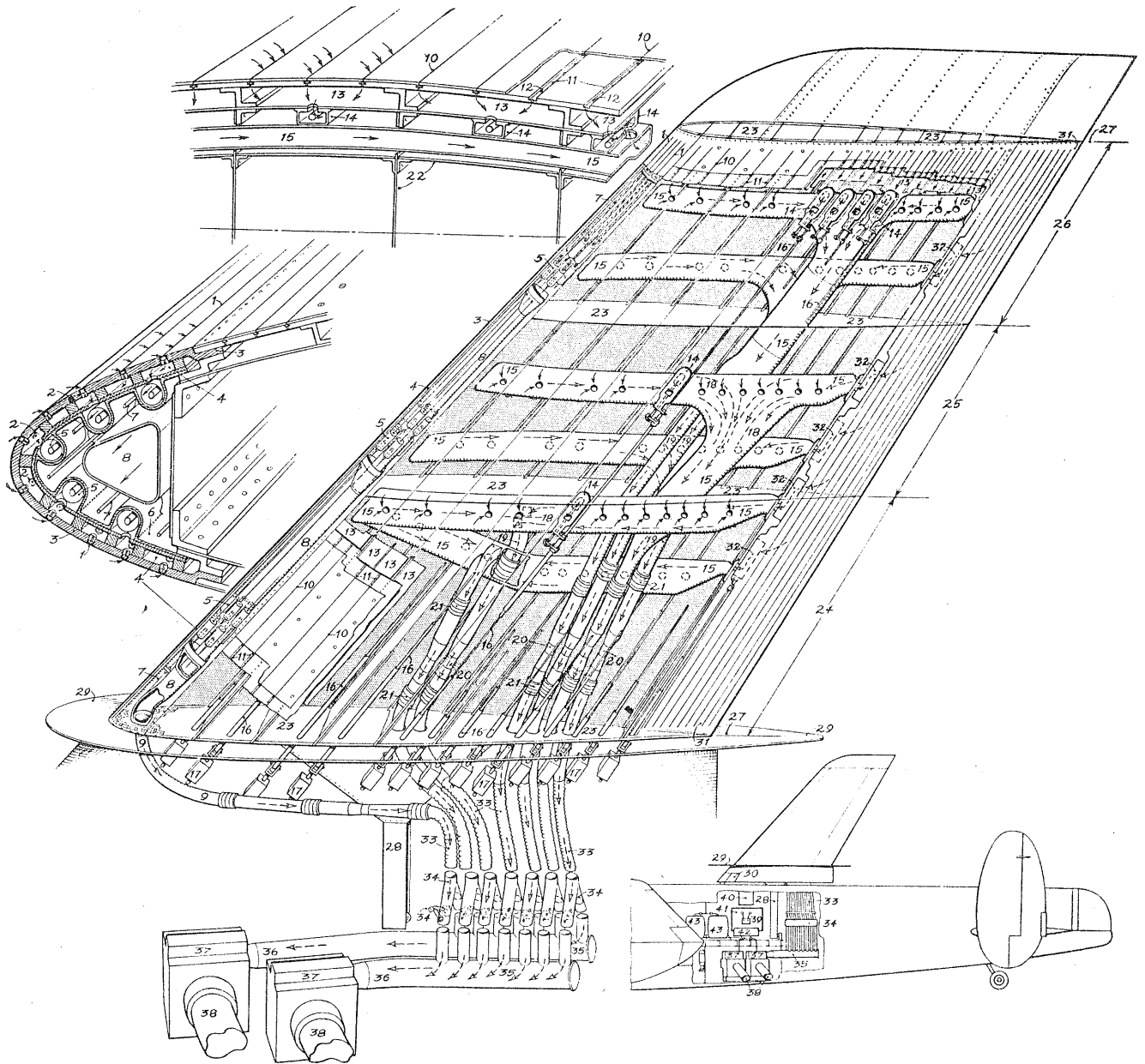


FIG. 1 SUCTION ZONES AND DIMENSIONS



Leading Edge

- 1 Suction slits 0.005in
- 2 Throttle holes in metal plugs
- 3 Glass-cloth skin
- 4 Plenum chambers
- 5 Needle valves
- 6 Glass-cloth wrap
- 7 Actuator rods for (5), with bungee return
- 8 Main collector ducts
- 9 Suction duct to turbines

Main Wing

- 10 Suction slits 0.005in in bonded outer skin

- 11 Channels in main plates
- 12 Throttle holes at 0.5in pitch
- 13 Plenum chambers
- 14 Needle valves, 13 in each zone
- 15 Collector ducts port and starboard
- 16 Concentric tubes and rod positioning (14)
- 17 Linear actuators driving (16)
- 18 Cascade vanes in (15)
- 19 Bifurcated suction pipes from (15)
- 20 Venturi meters
- 21 Flexible couplings to permit wing incidence change
- 22 Wing structure
- 23 Zone boundary ribs

Trailing edge

- 24 Zone A
- 25 Zone B
- 26 Zone C
- 27 Limit of suction
- 28 Wing attachment post, $\pm 5^\circ$ variable incidence
- 29 Boundary fence
- 30 Mass balance
- 31 Nine suction slits
- 32 Needle valves governing flow through (31)

Suction installation

- 33 Flexible ducts to suction engines
- 34 Bifurcated pipes with butterfly valves
- 35 Suction manifolds
- 36 Main suction-engine intake trunks
- 37 Plenum boxes housing Budworth gas turbines
- 38 Exhaust pipes from (37)
- 39 Engine power controls
- 40 Engine instrument panel
- 41 Engine ancillary controls
- 42 Heiland strain gauge recorder
- 43 Fuel tanks

FIG. 2

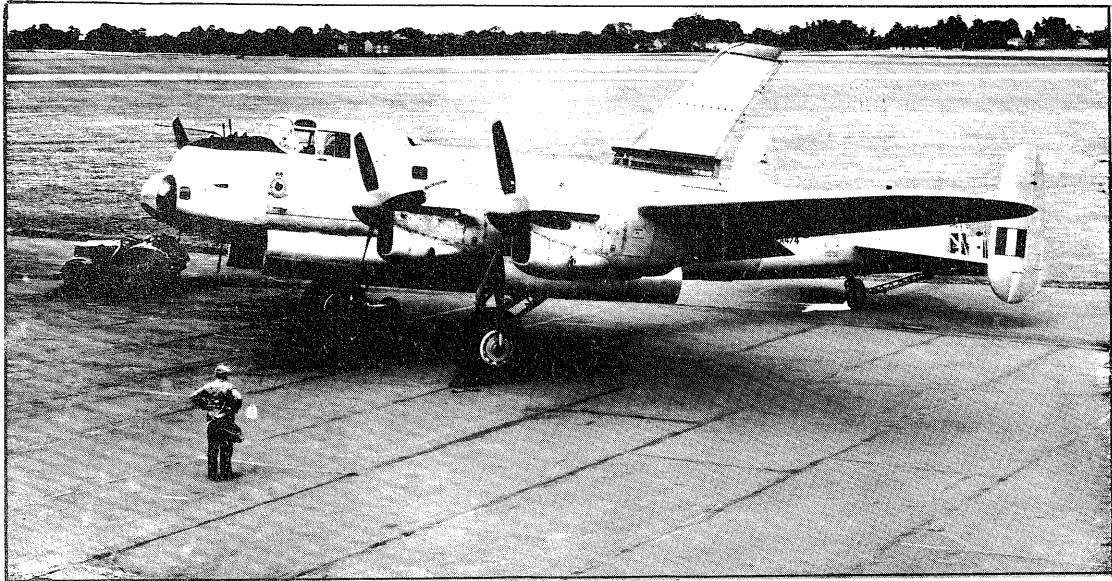


FIG. 3 HANDLEY PAGE LAMINAR FLOW SWEEP WING MOUNTED ON THE LANCASTER AIRCRAFT.

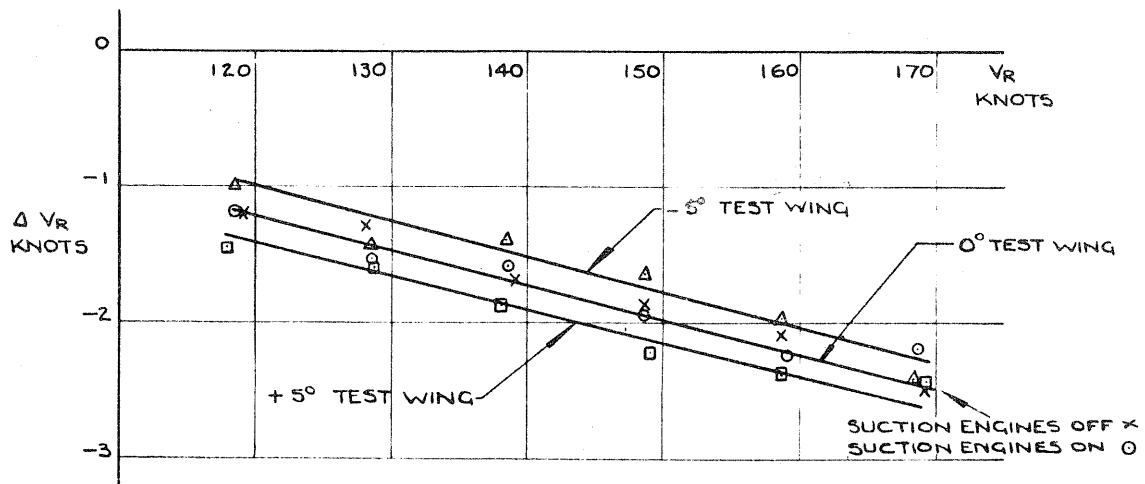


FIG. 4 STATIC PRESSURE ERROR CORRECTION

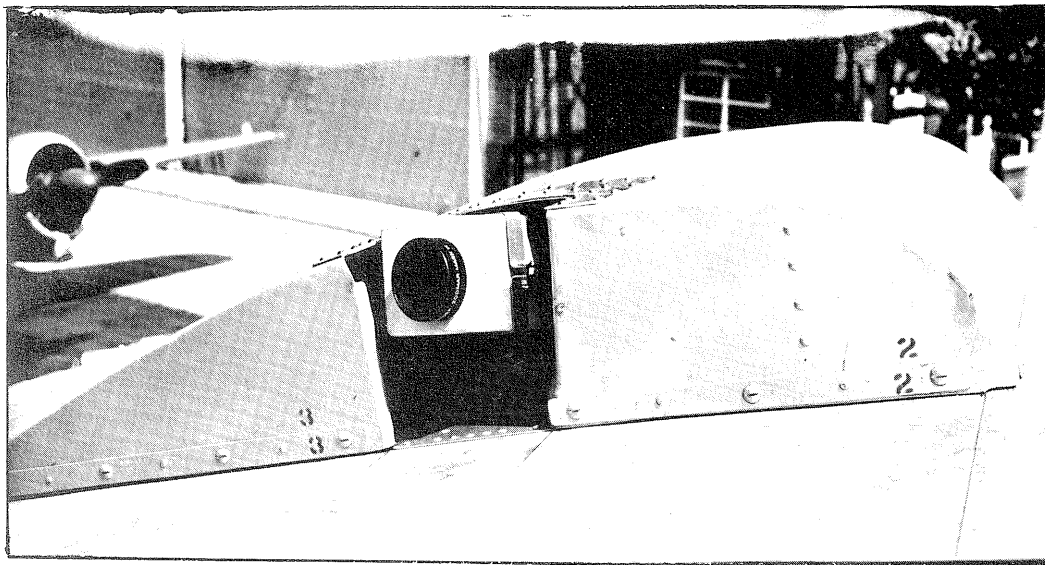


FIG. 5 TELEVISION CAMERA MOUNTED IN WING TIP FAIRING

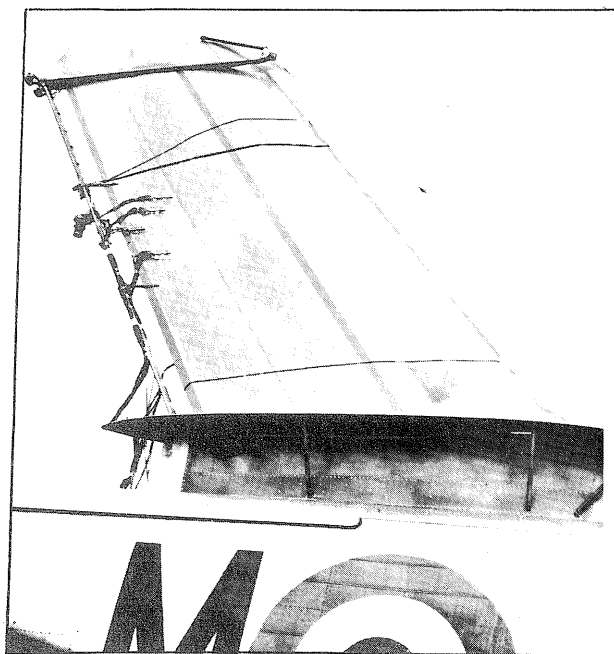


FIG. 6 WING PRIOR TO FLIGHT WITH LEADING EDGE FLY PROTECTIVE SHEET IN POSITION.

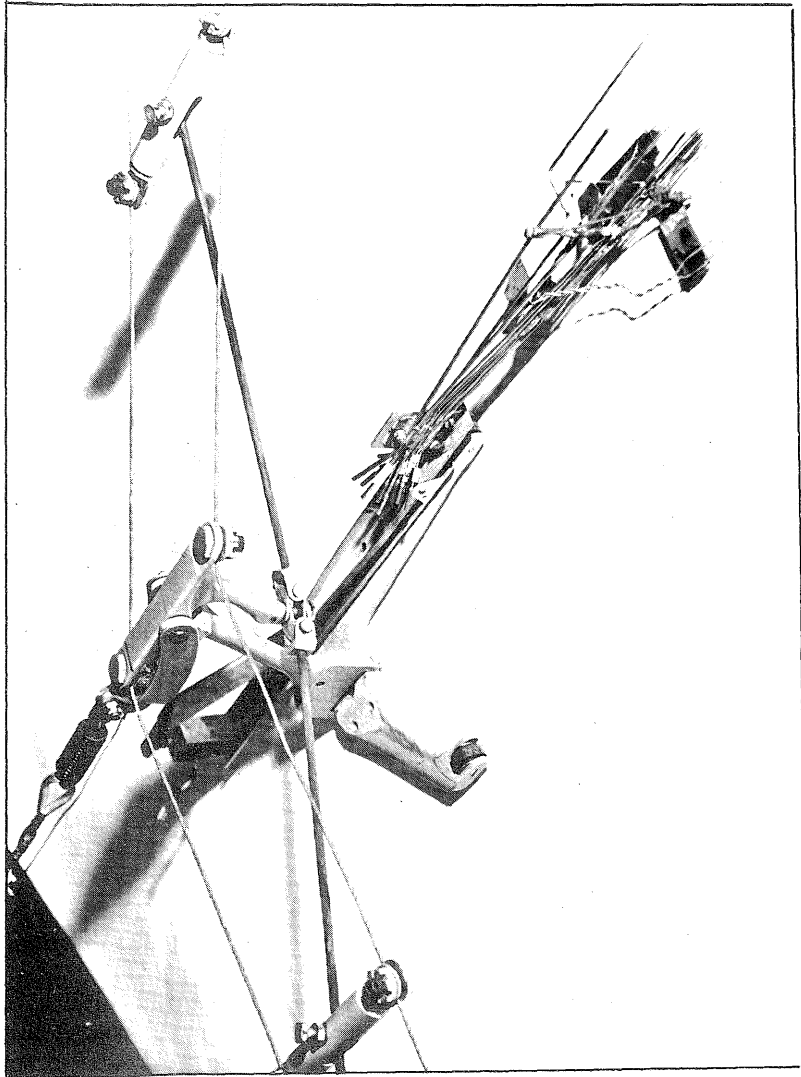


FIG. 7 ASSEMBLED TRAVELLING PITOT CARRIAGE
WITH HOT FILMS

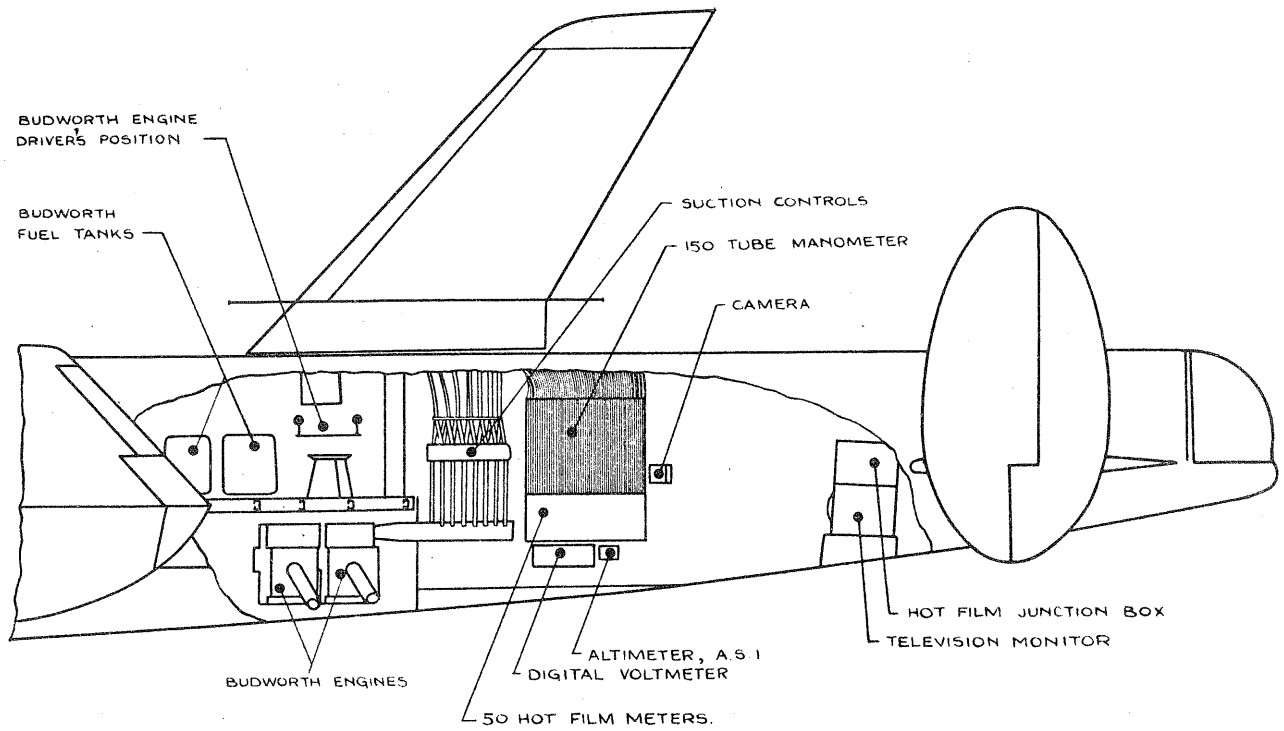


FIG. 8 LOCATION OF MAJOR EQUIPMENT IN LANCASTER AIRCRAFT.

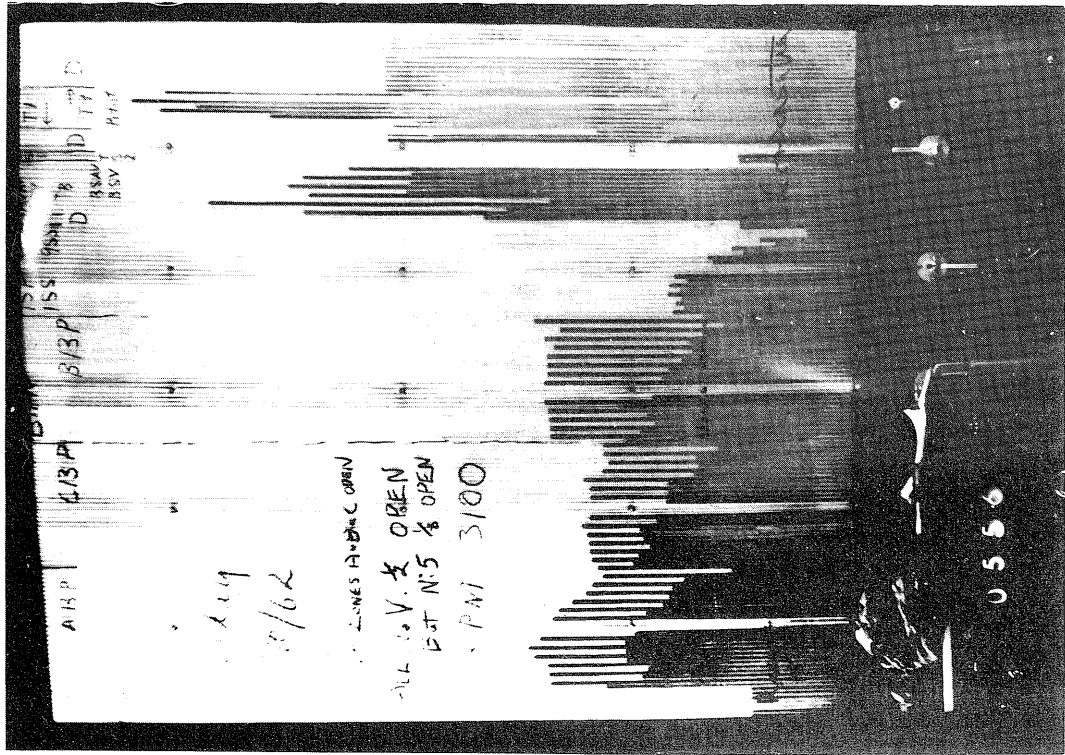


FIG. 9 TYPICAL FLIGHT RECORD OF MANOMETER

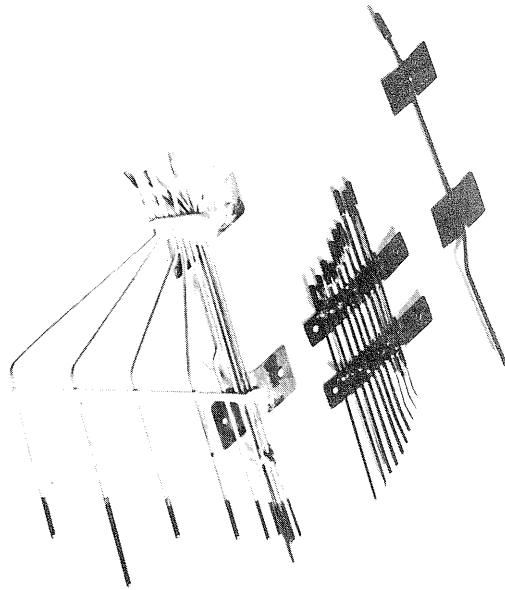


FIG. 10 BOUNDARY LAYER PITOT COMBS AND SURFACE STATIC TUBE

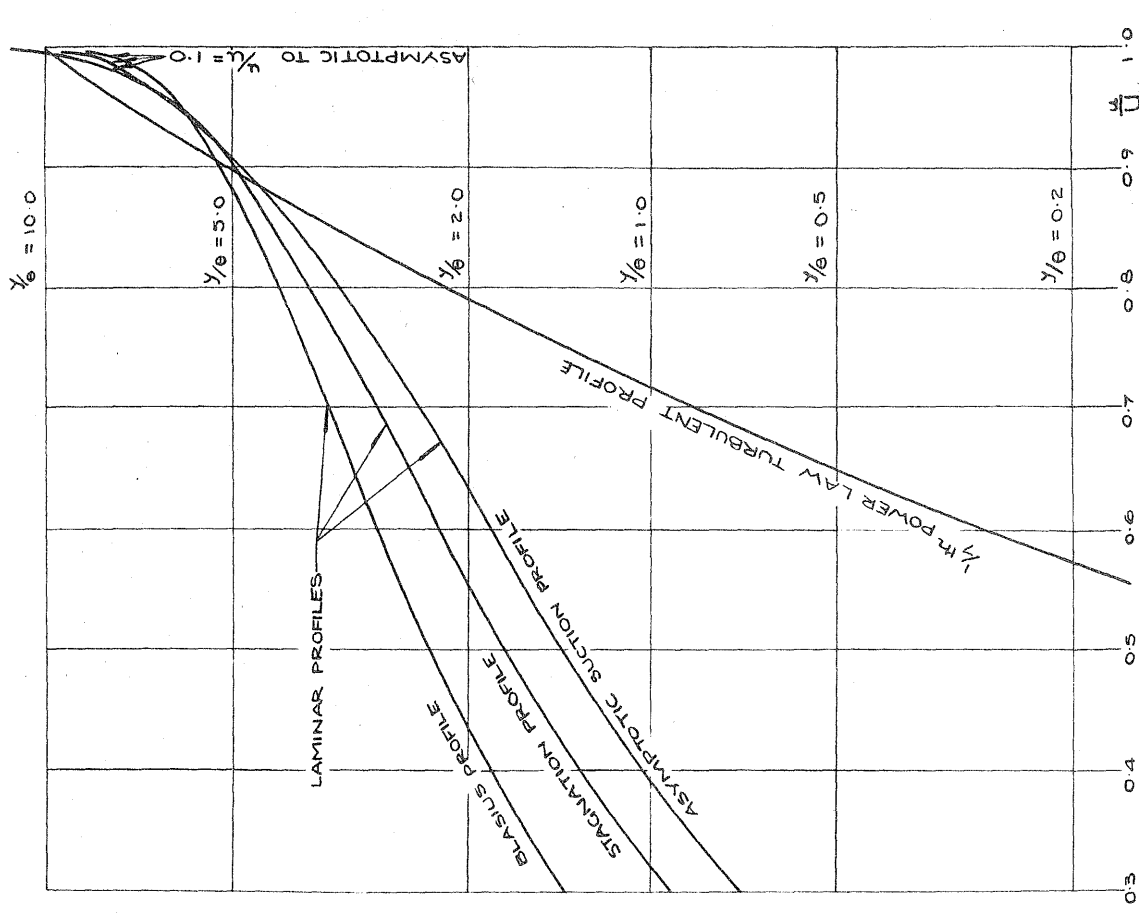


FIG. 11 THEORETICAL LOG BOUNDARY LAYER PROFILES PLOTTED ON THE "LOG-HEIGHT" BASIS

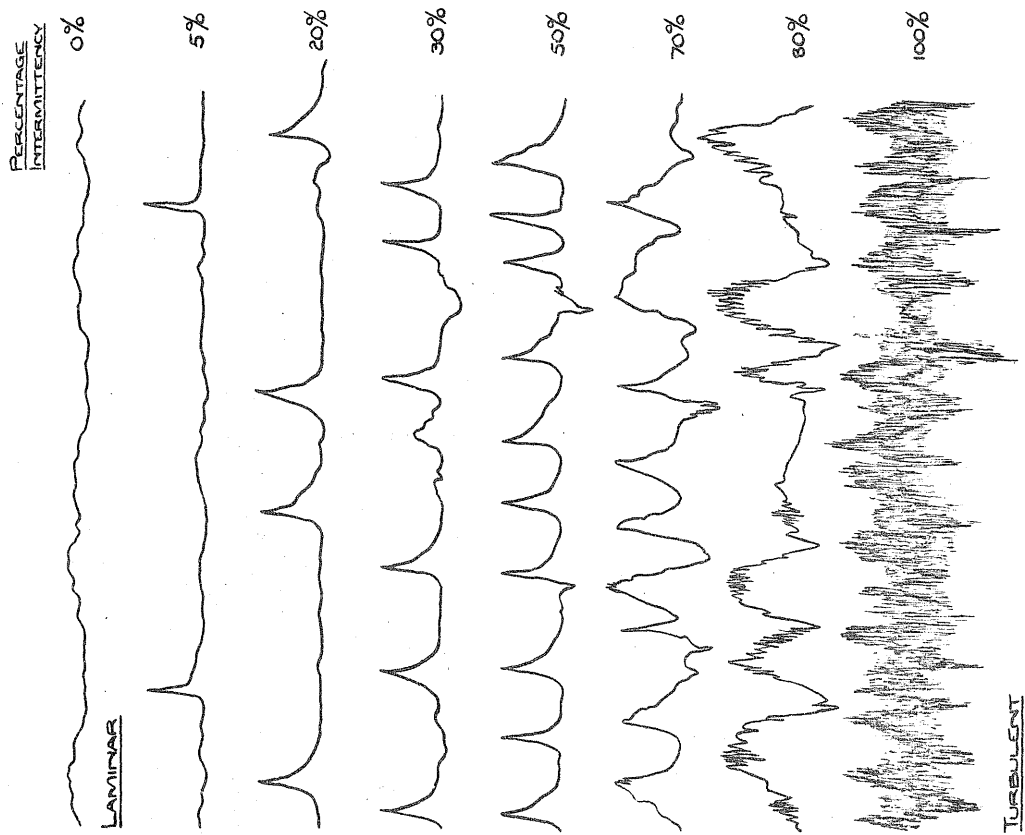
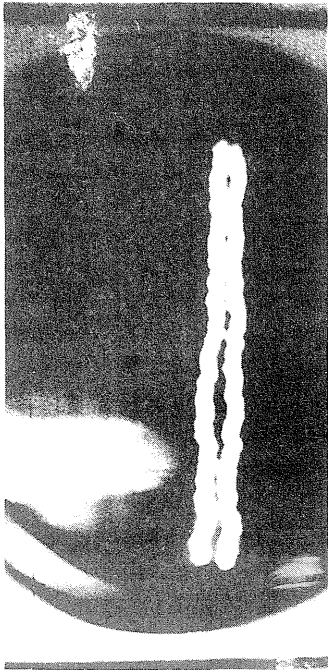
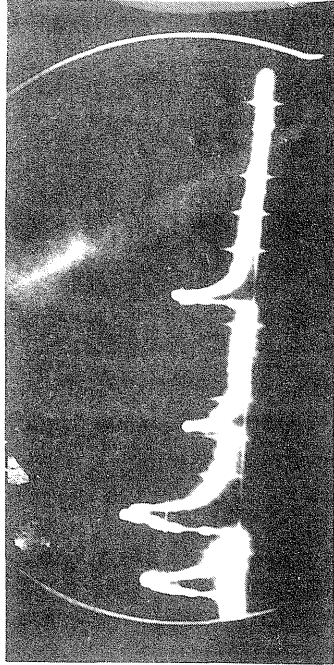


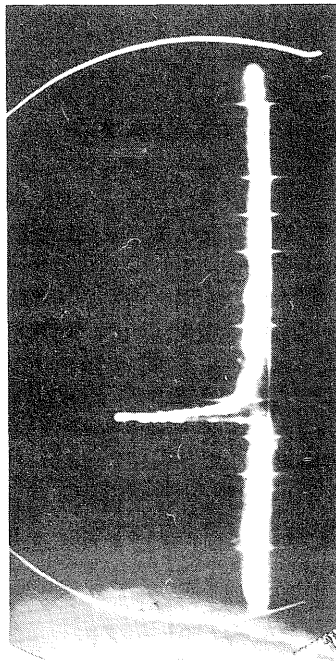
FIG. 12 ARBITRARY TURBULENT INTERMITTENCY SCALE USED FOR HOT FILM OBSERVATIONS



LAMINAR



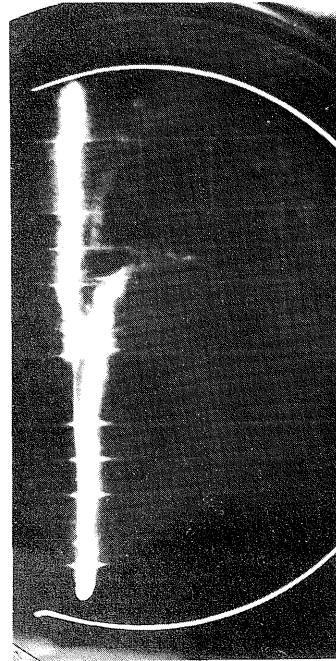
80% INTERMITTENCY



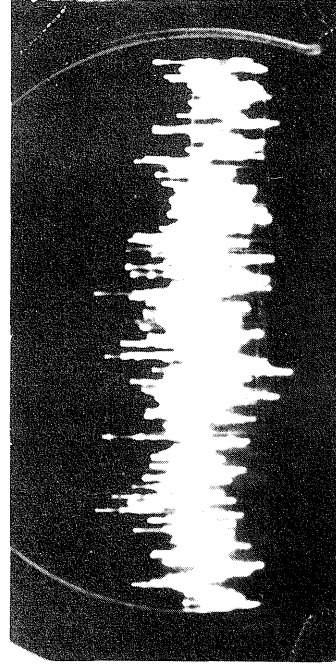
10% INTERMITTENCY



80% INTERMITTENCY

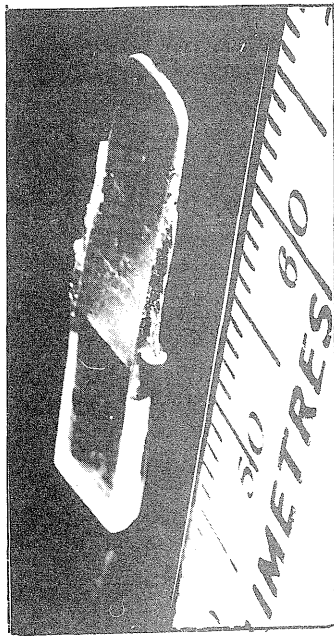


15% INTERMITTENCY



TURBULENT

FIG 13 OSCILLOSCOPE PHOTOGRAPHS USING 1/20th SECOND EXPOSURES



TYPE 1 HOT FILM - BLUNT ENDED TYPE



TYPE 2 ENLARGED CORNER OF BEVEL TYPE HOT FILM
(WIDTH OF BEVEL 0.4 MILLIMETRE).

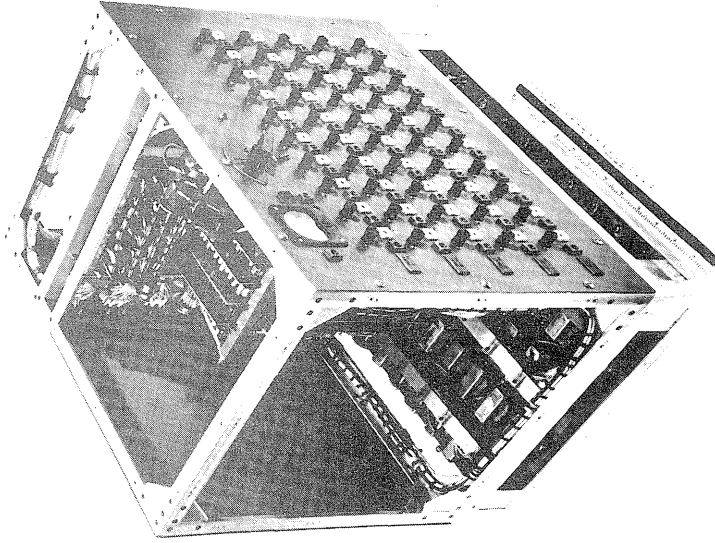


FIG. 15 HOT FILM CONTROL BOX.

FIG. 14

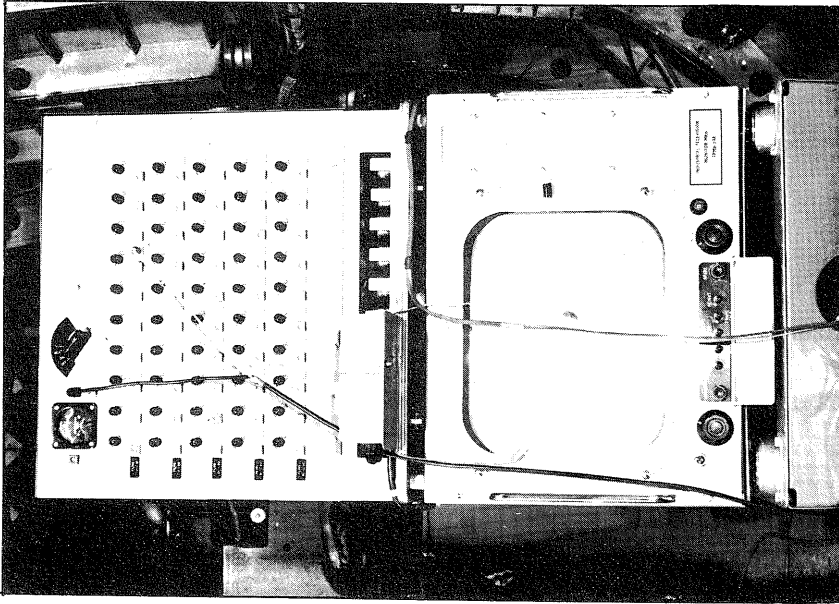


FIG. 16 TELEVISION MONITOR AND HOT FILM CONTROL BOX

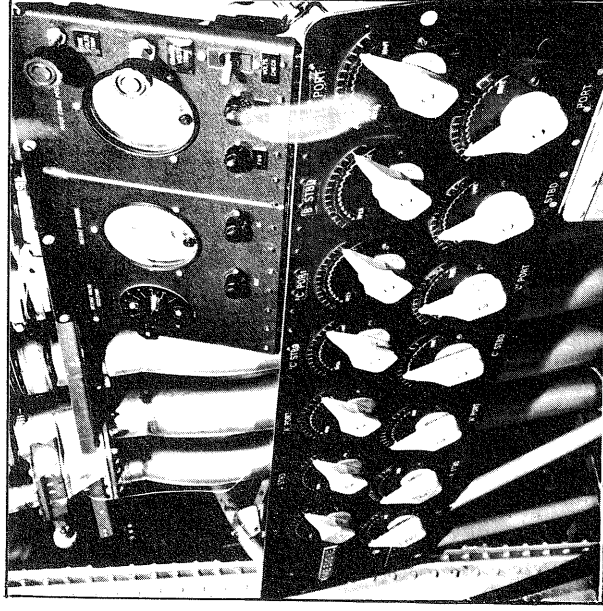


FIG. 17 FLOW DISTRIBUTION PANEL SET FOR OPERATION OF No. 1 ENGINE

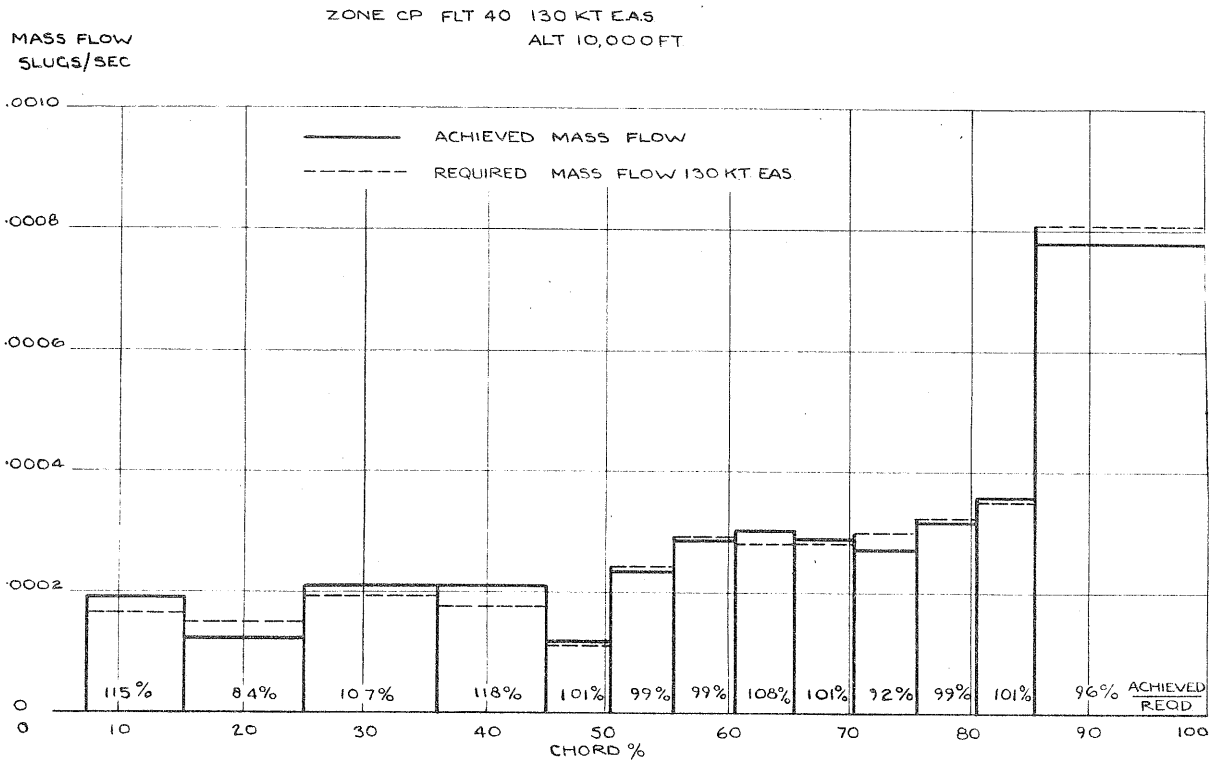


FIG. 18 TYPICAL MASS FLOW DISTRIBUTION OF SUCTION OBTAINED BY TUNING IN FLIGHT

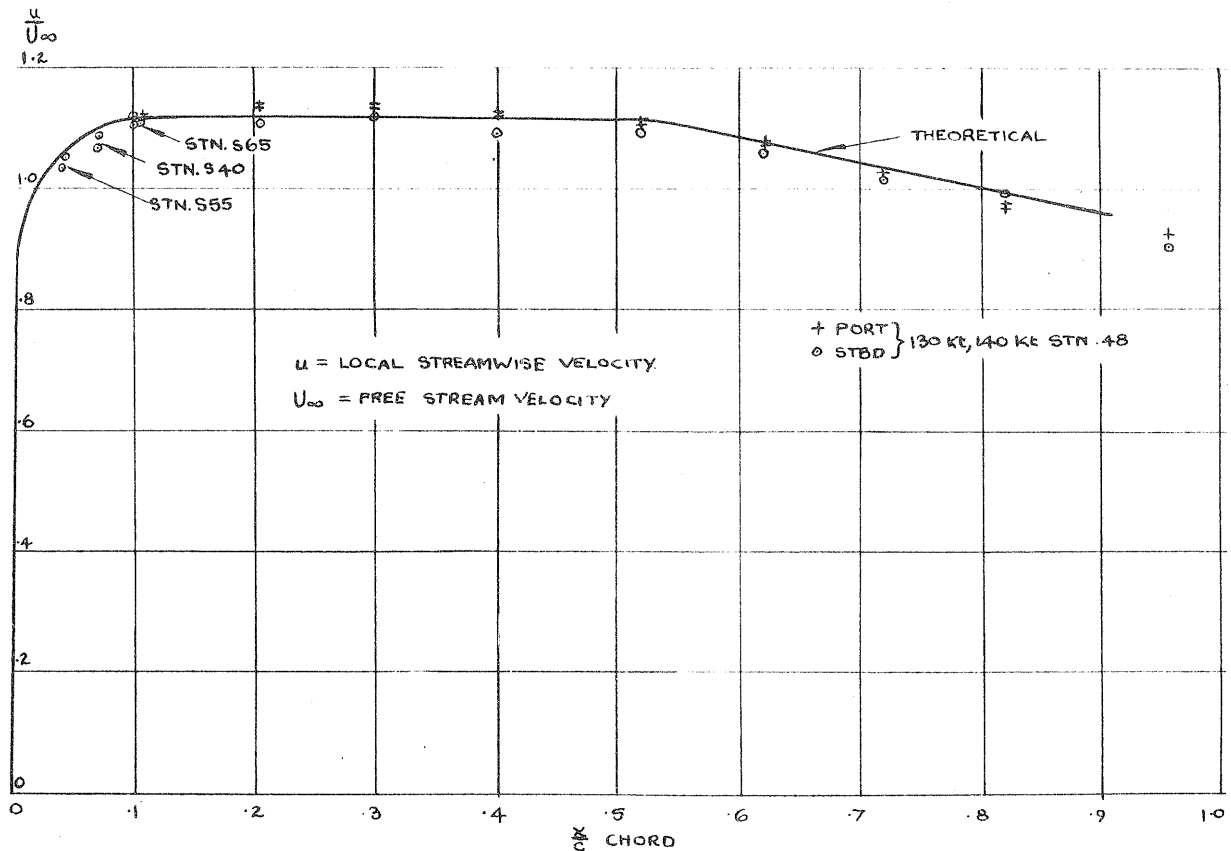


FIG. 19 TWO DIMENSIONAL VELOCITY DISTRIBUTIONS ON LAMINAR WING 12.5% t/c

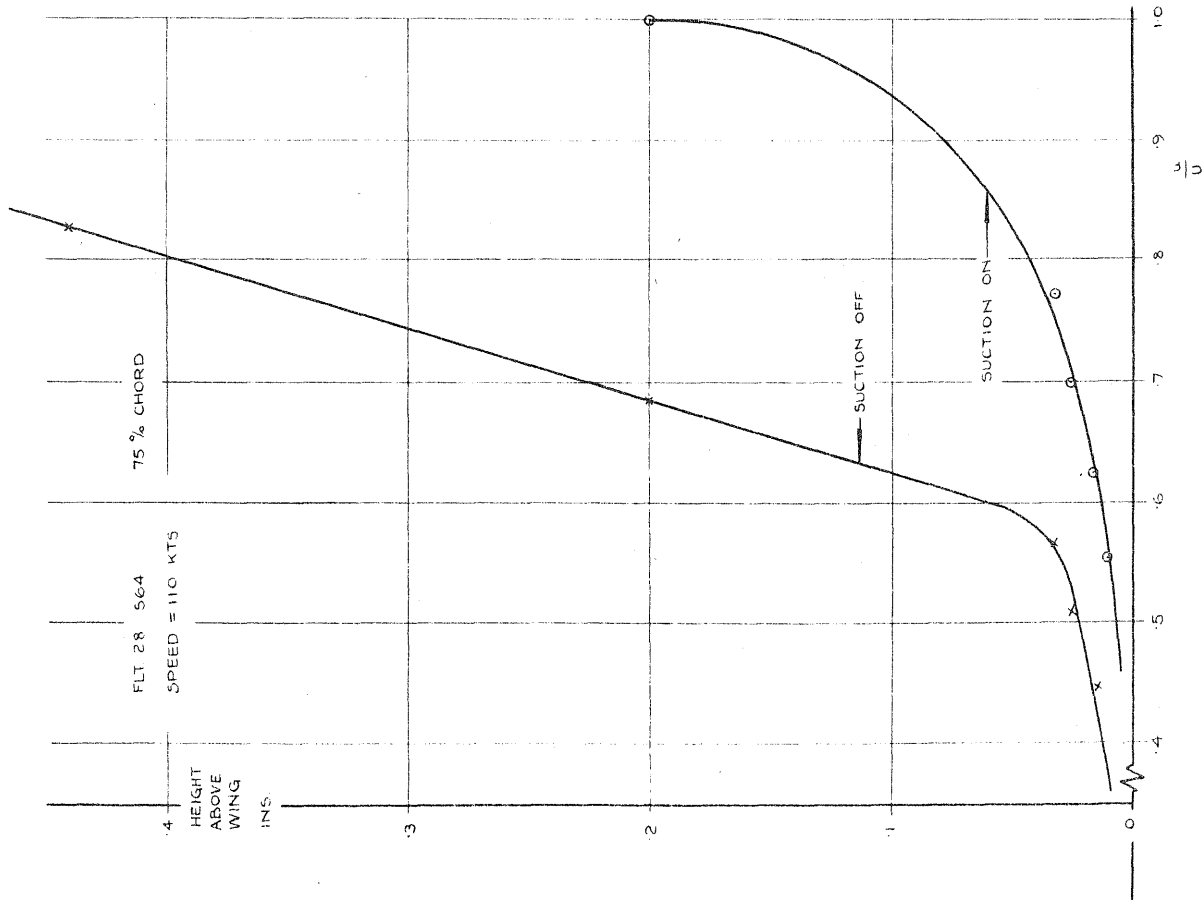


FIG. 20 VELOCITY PROFILES WITH AND WITHOUT SUCTION

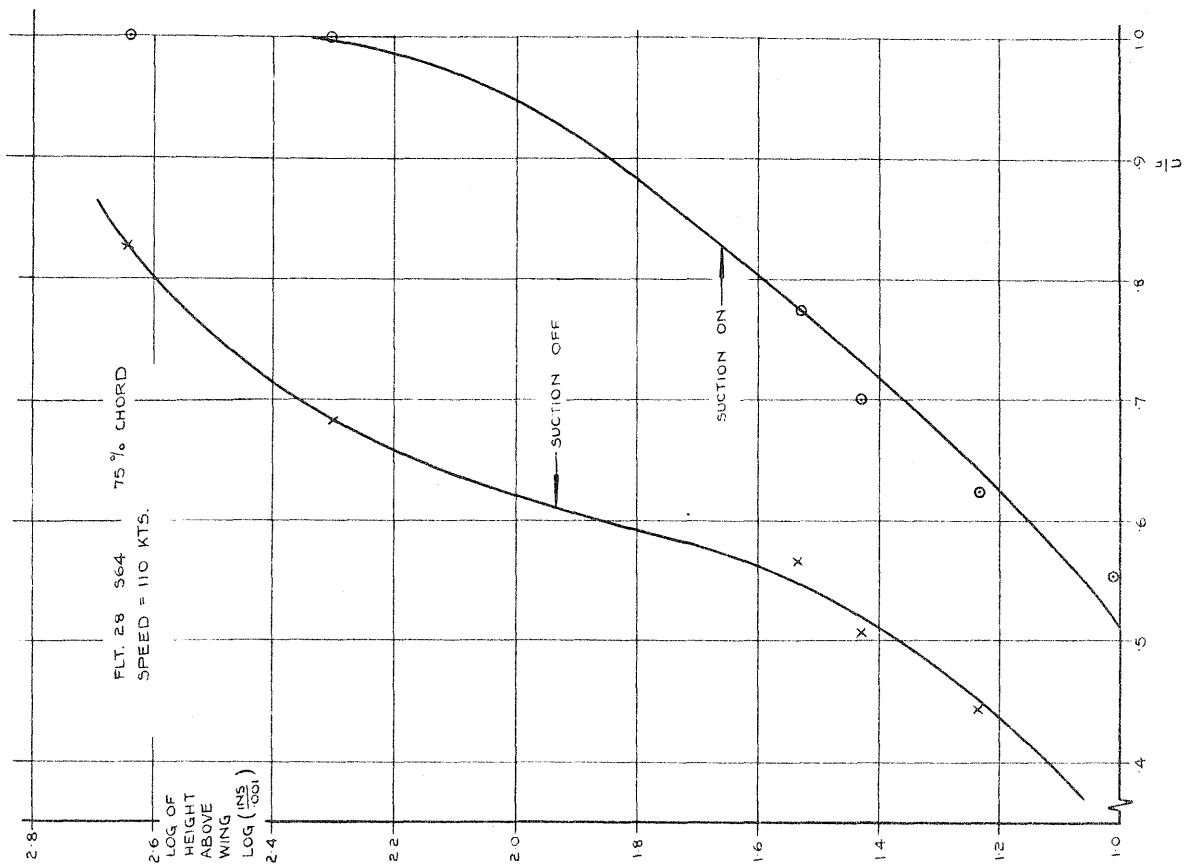


FIG. 21 VELOCITY PROFILES OF FIG. 20 PLOTTED ON A "LOG-HEIGHT" BASIS

HOT FILM RESULTS FROM CARRIAGE
 TYPICAL TRAVERSE
 PORT STN 65

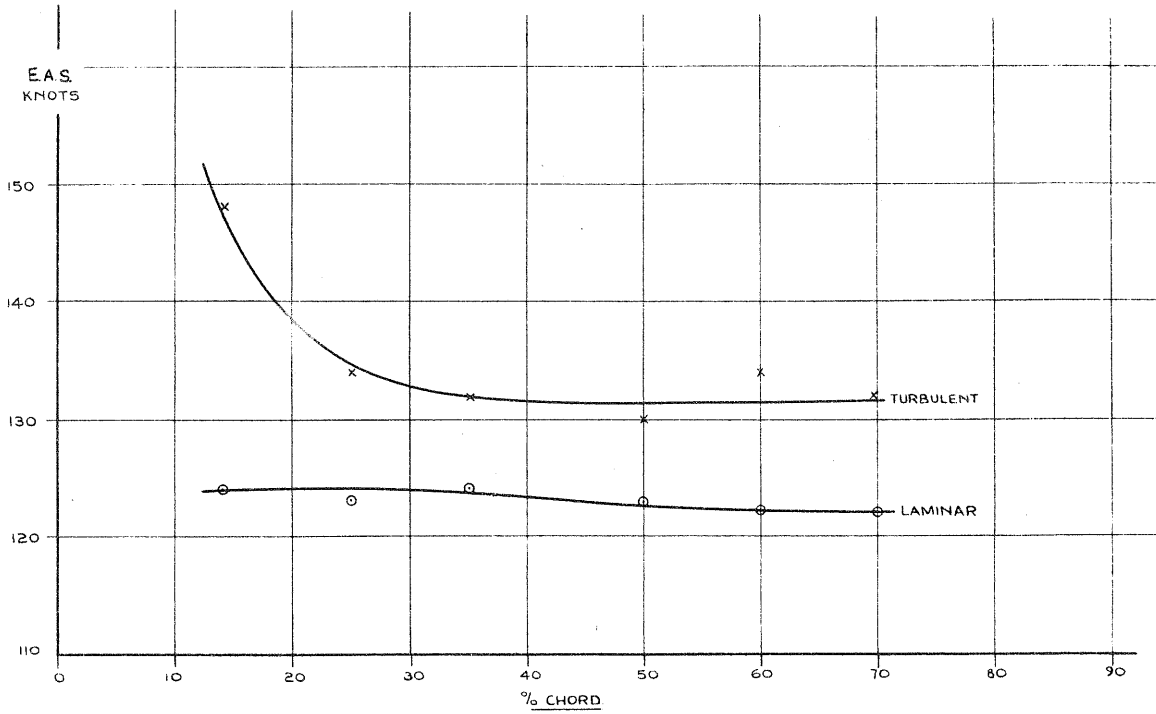


FIG. 22

UNIT
 REYNOLDS No. $\times 10^{-6}$

CHORDWISE VARIATION OF UNIT REYNOLDS No.
 AT TRANSITION FROM LAMINAR TO TURBULENT FLOW.

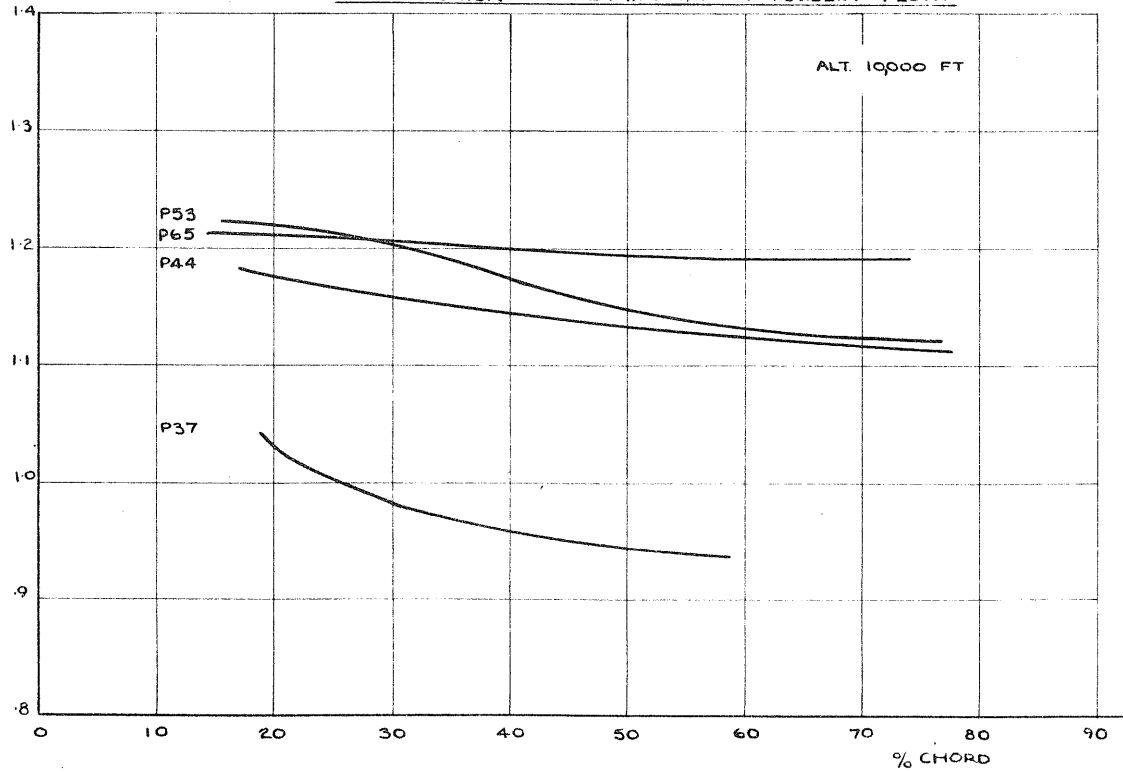


FIG. 23

REYNOLDS No. $\times 10^{-6}$ AT TRANSITION
 BASED ON LOCAL CHORD

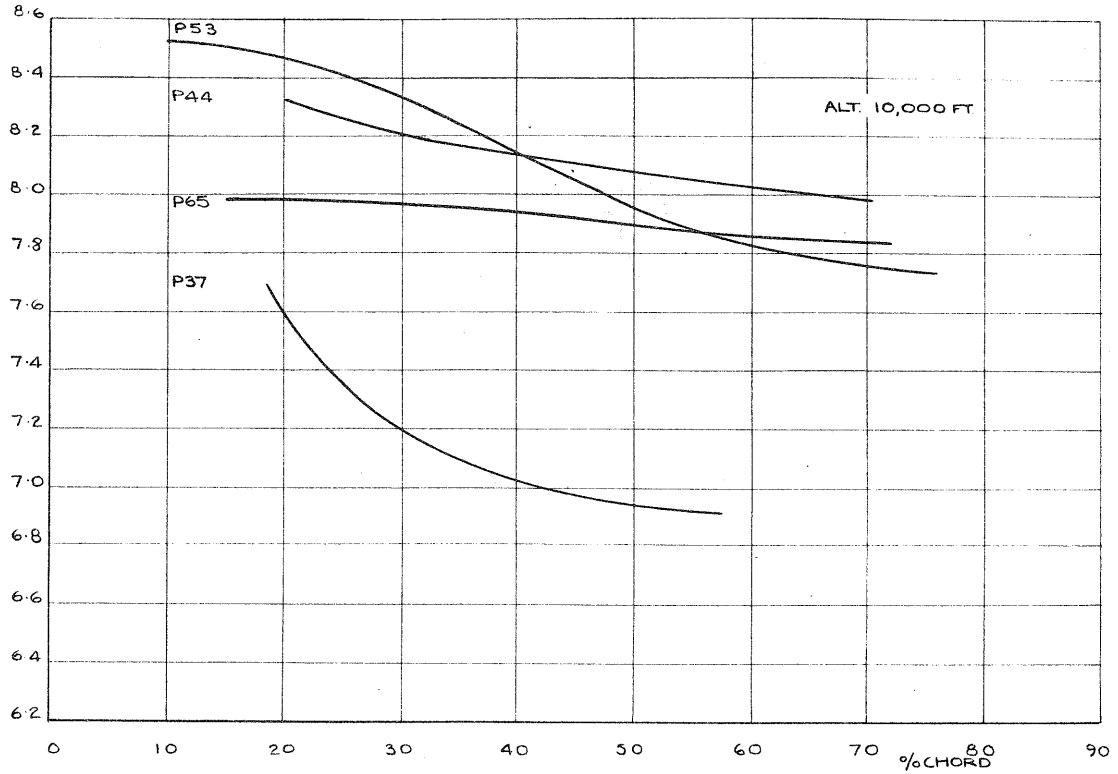


FIG. 24 TRANSITION REYNOLDS NUMBERS OBTAINED BY TRAVERSING A HOT FILM

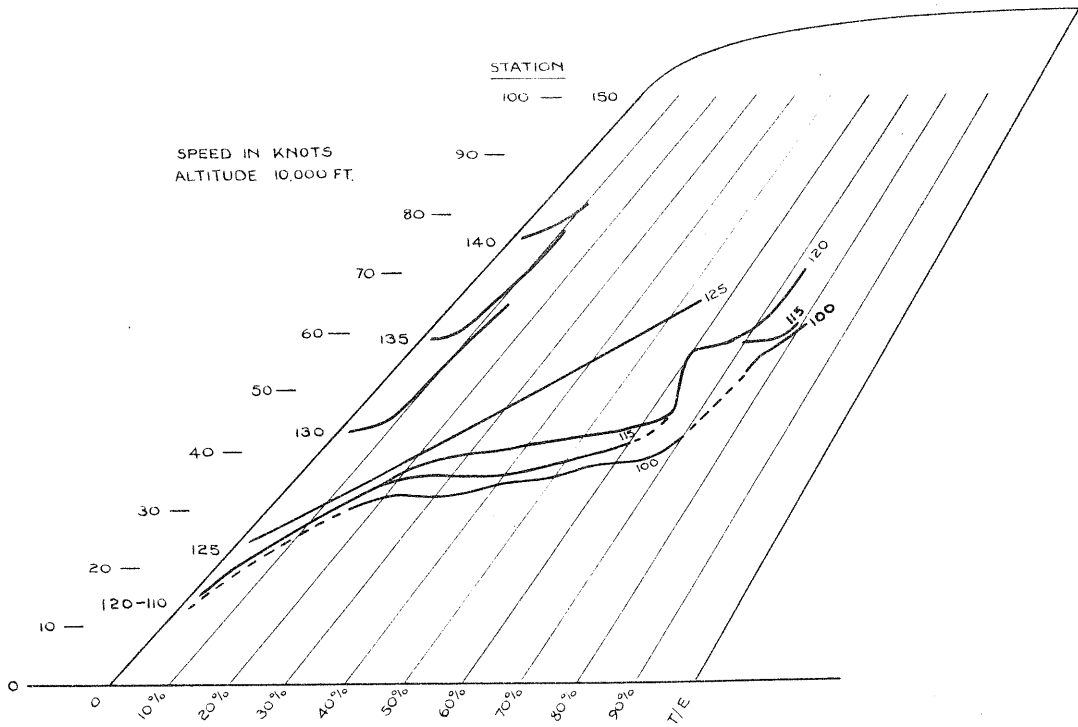


FIG. 25 TRANSITION FRONT ON LAMINAR FIN PORT SIDE

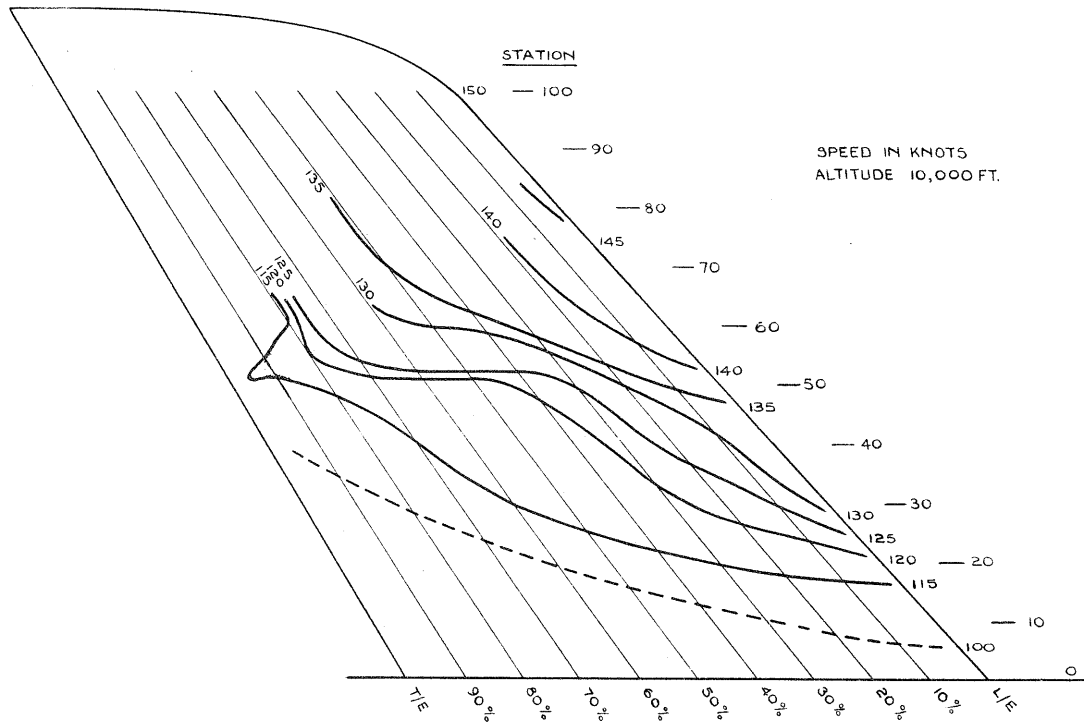


FIG. 26 TRANSITION FRONT ON LAMINAR FIN STARBOARD SIDE

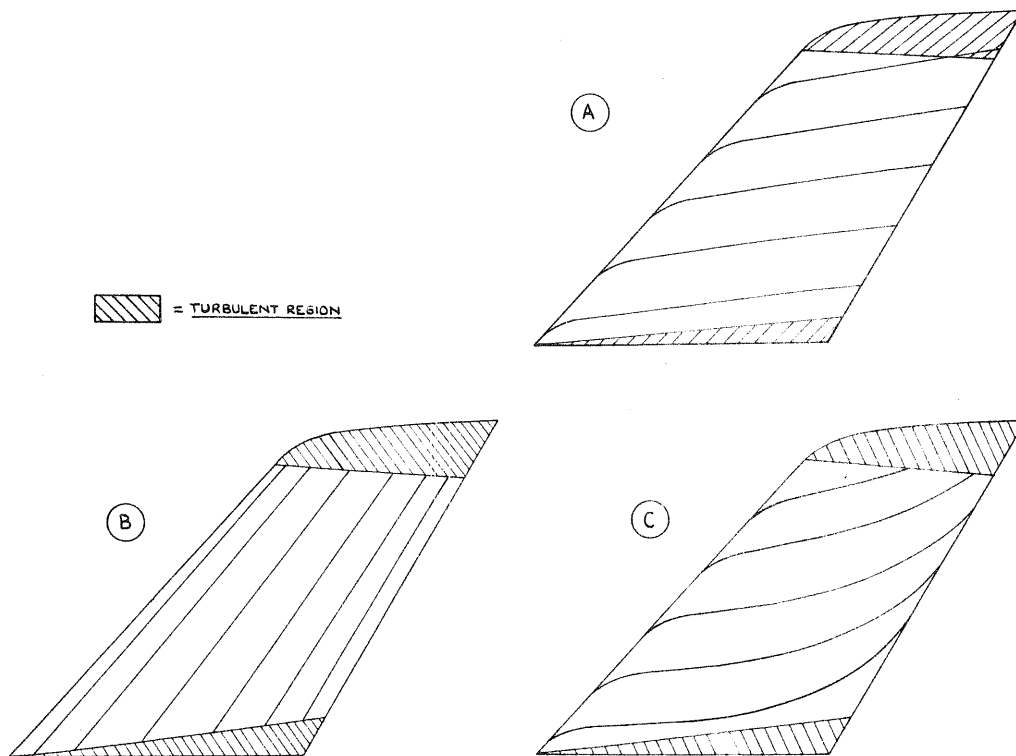


FIG. 27 IDEALISED VARIATION OF TRANSITION LINES WITH SPEED

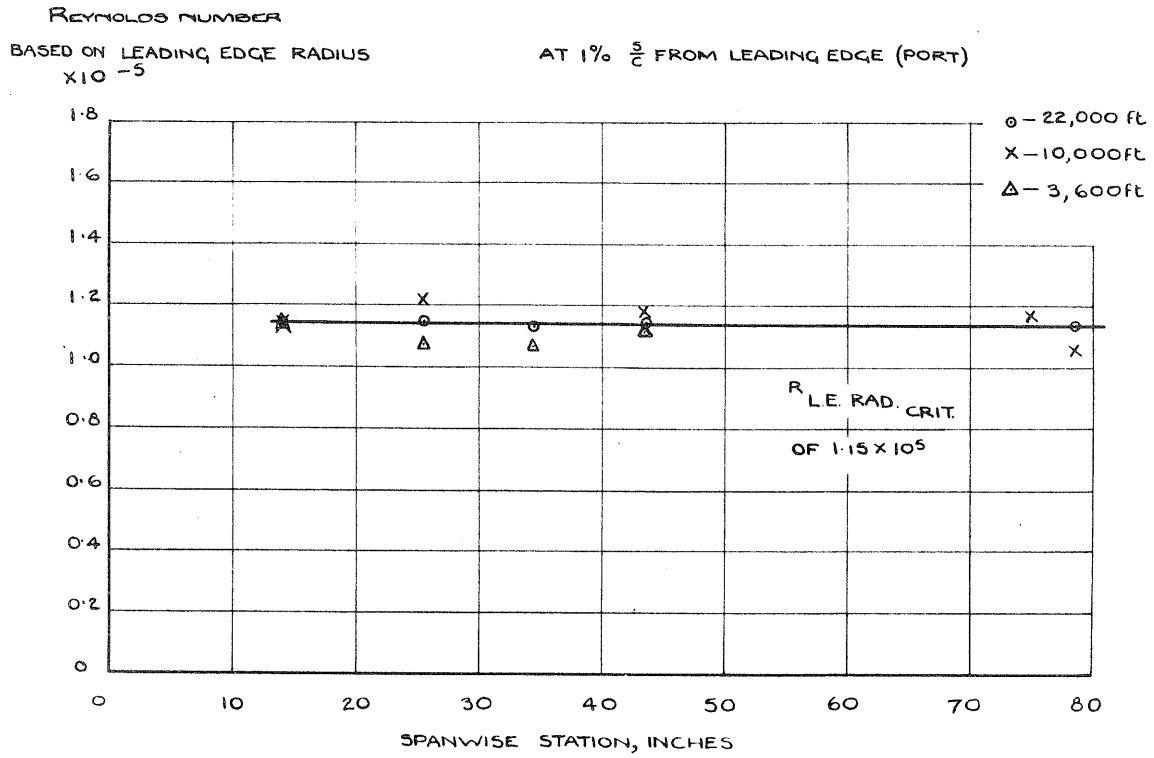


FIG. 28 EXPERIMENTAL LOWER LIMIT OF TRANSITION
 (INTERMITTENCY < 5%)

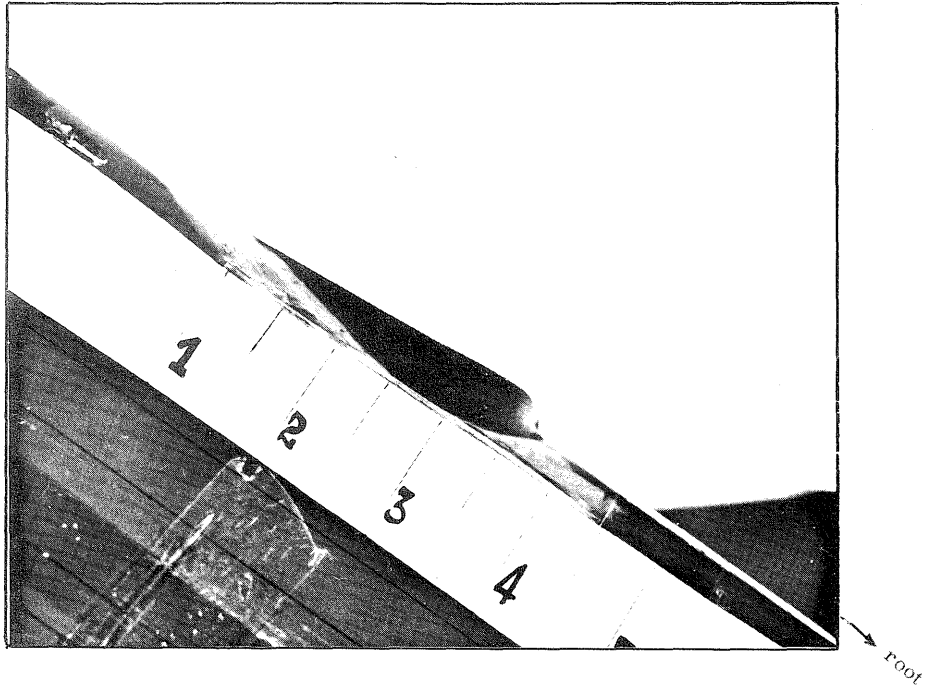


FIG. 29 EARLY PLASTICINE BUMP (SCALE IN INCHES)

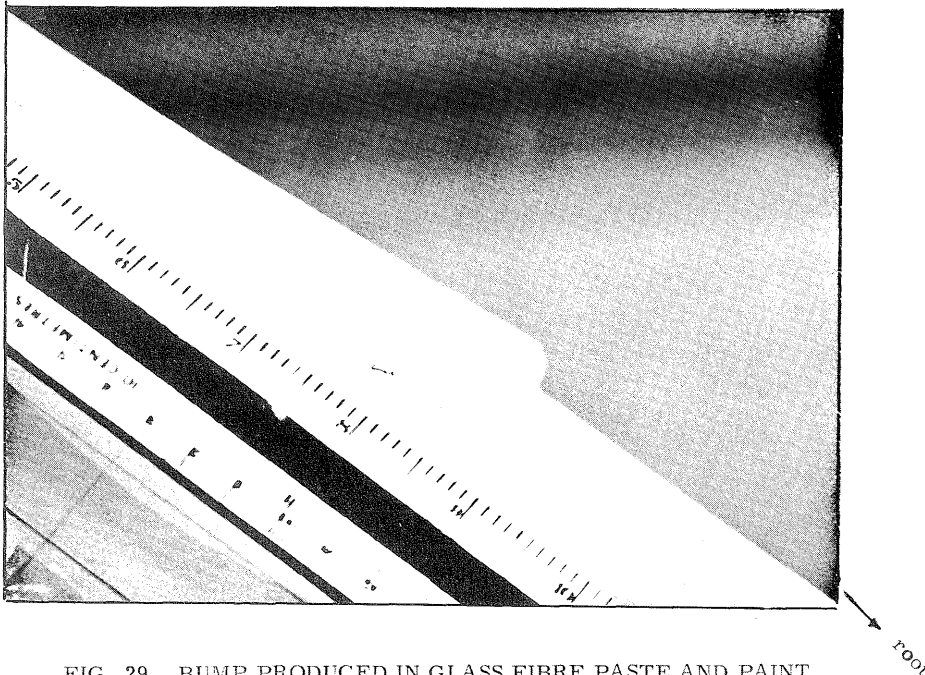


FIG. 29 BUMP PRODUCED IN GLASS FIBRE PASTE AND PAINT FILLER (MAIN SCALE IN INCHES).

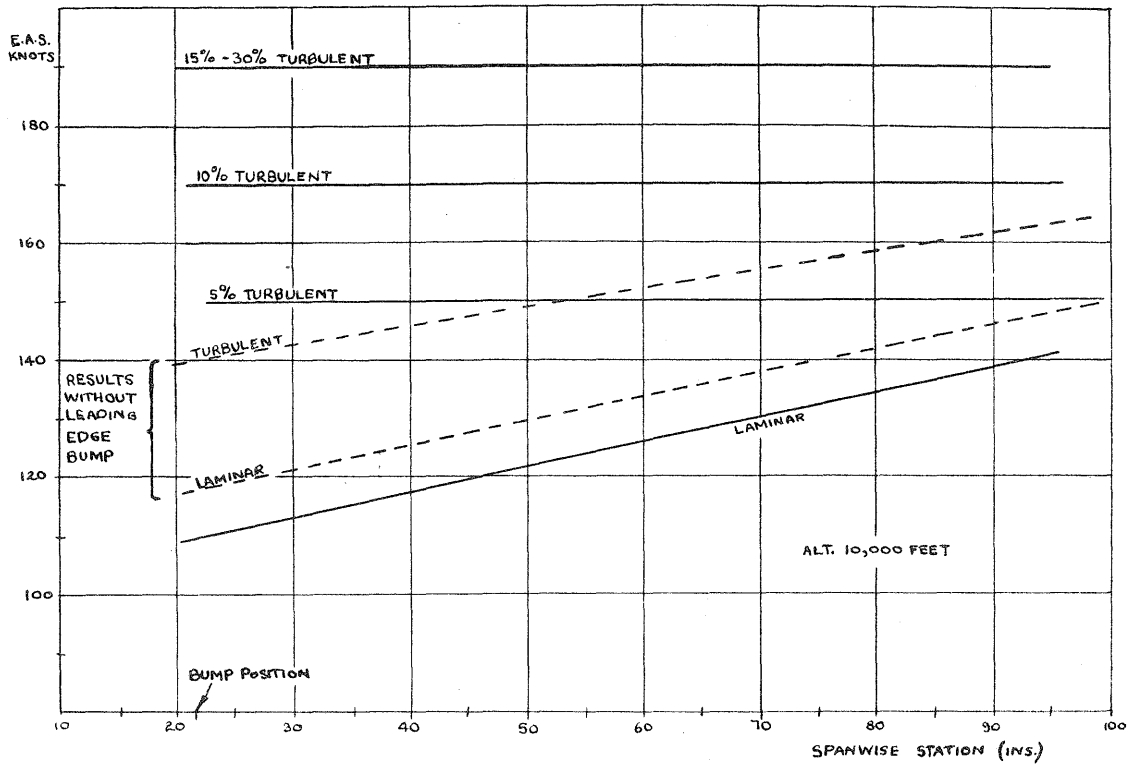


FIG. 30

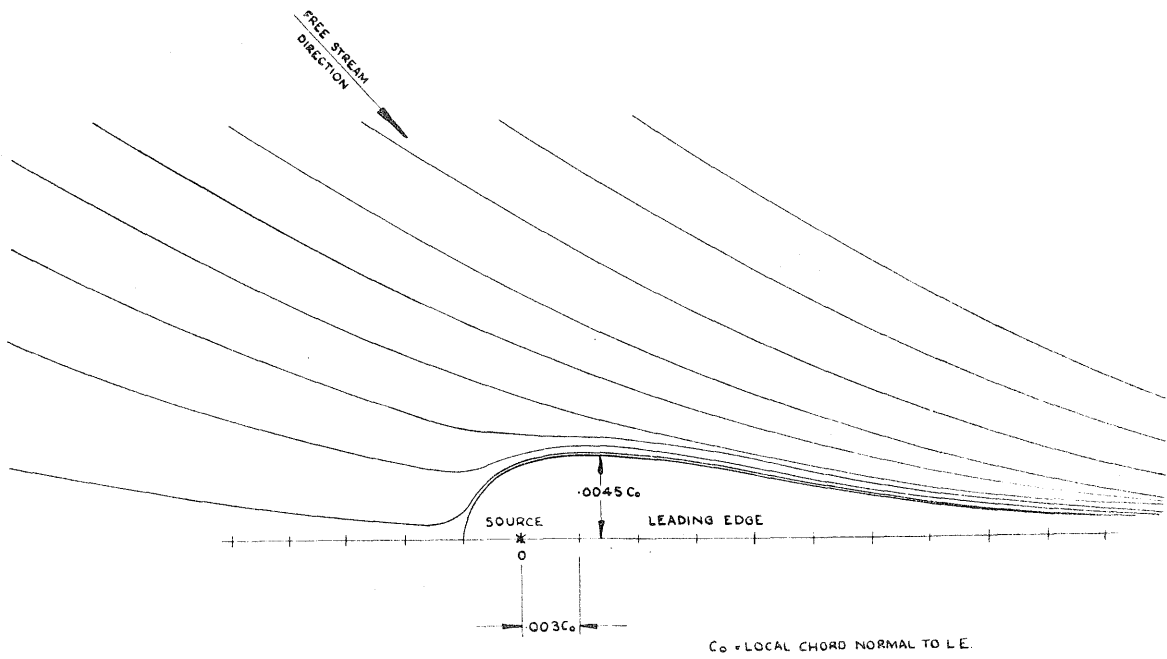


FIG. 31 STREAMLINES DUE TO THREE DIMENSIONAL POINT SOURCE ON LEADING EDGE OF SWEEP WING TO REPRESENT BUMP

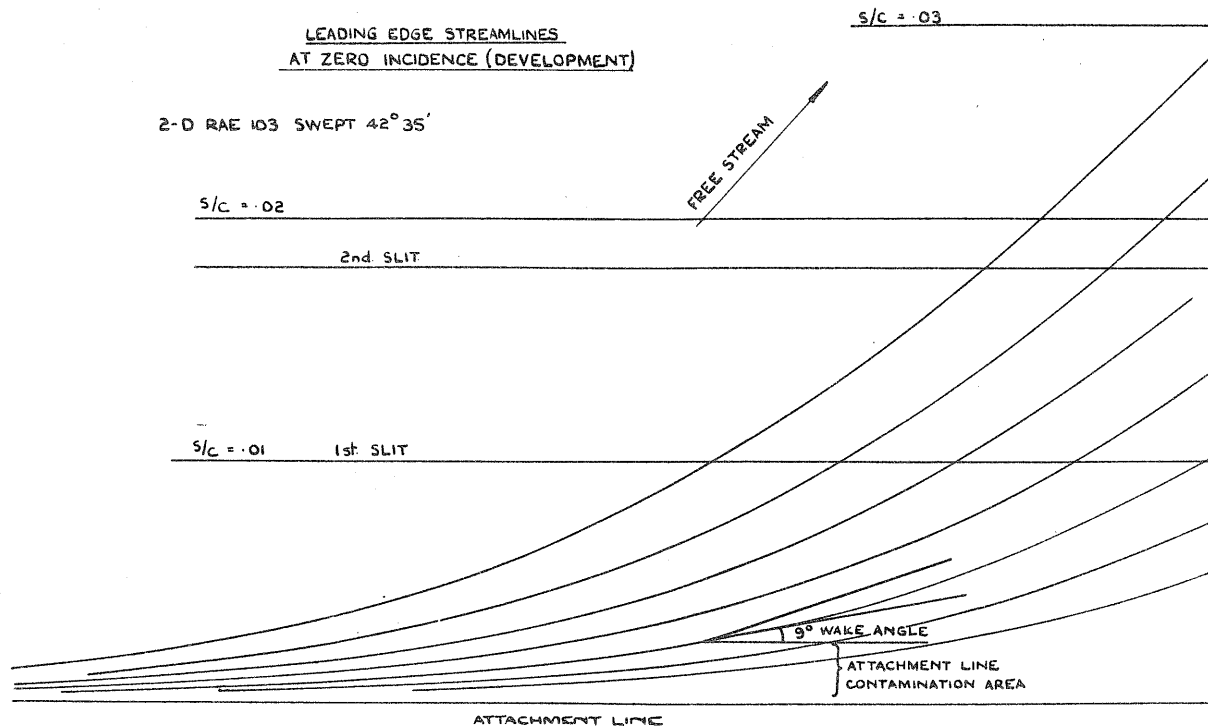


FIG. 32 LEADING EDGE STREAMLINES AT ZERO INCIDENCE (DEVELOPMENT)

CONSTRUCTIONAL DETAILS

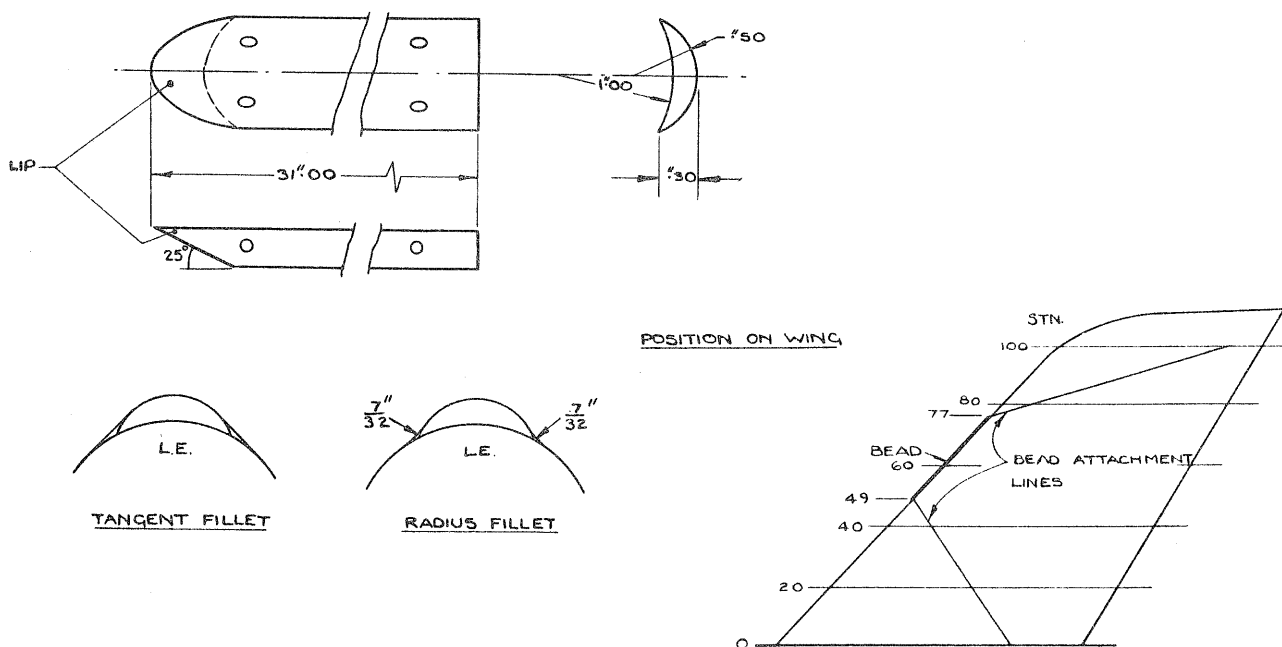


FIG. 33 THE LEADING EDGE BEAD

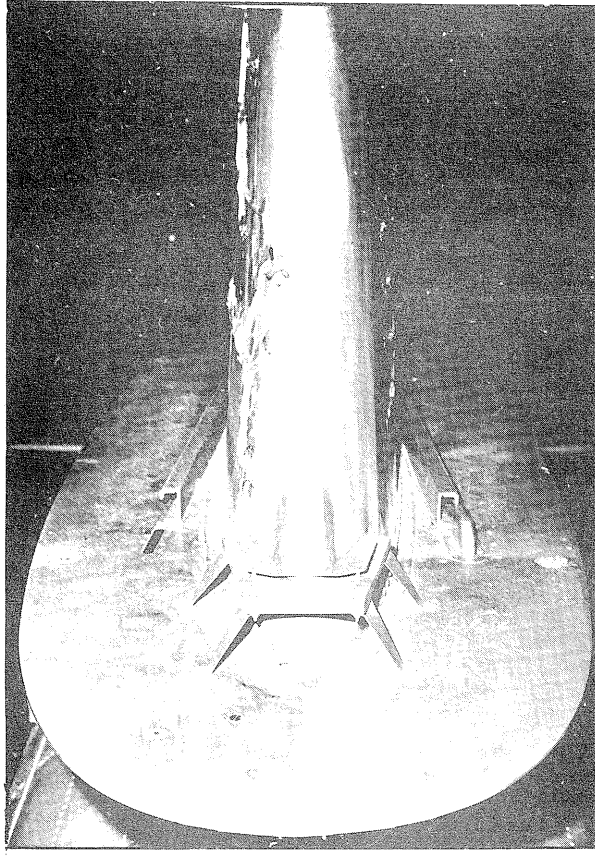


FIG 34 BOUNDARY LAYER FENCE WITH FORWARD FACING SLOT.

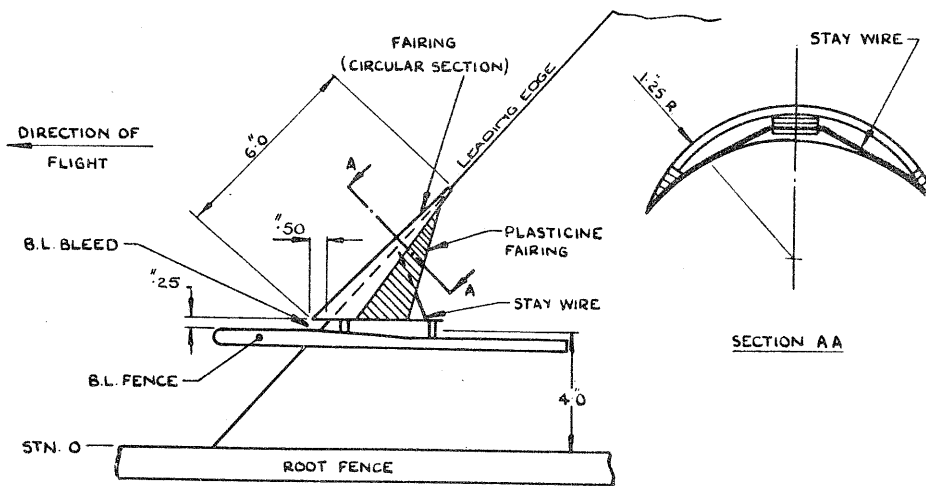


FIG. 35 BOUNDARY LAYER FENCE AT ROOT

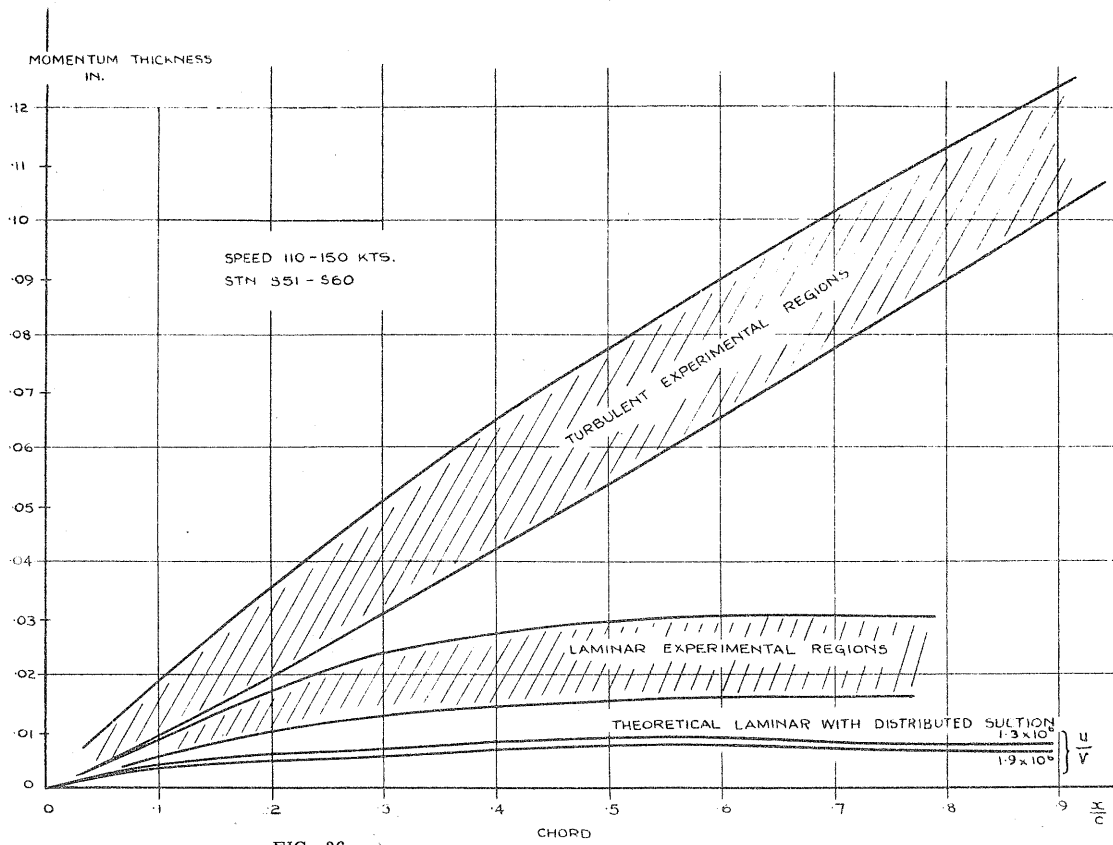


FIG. 36

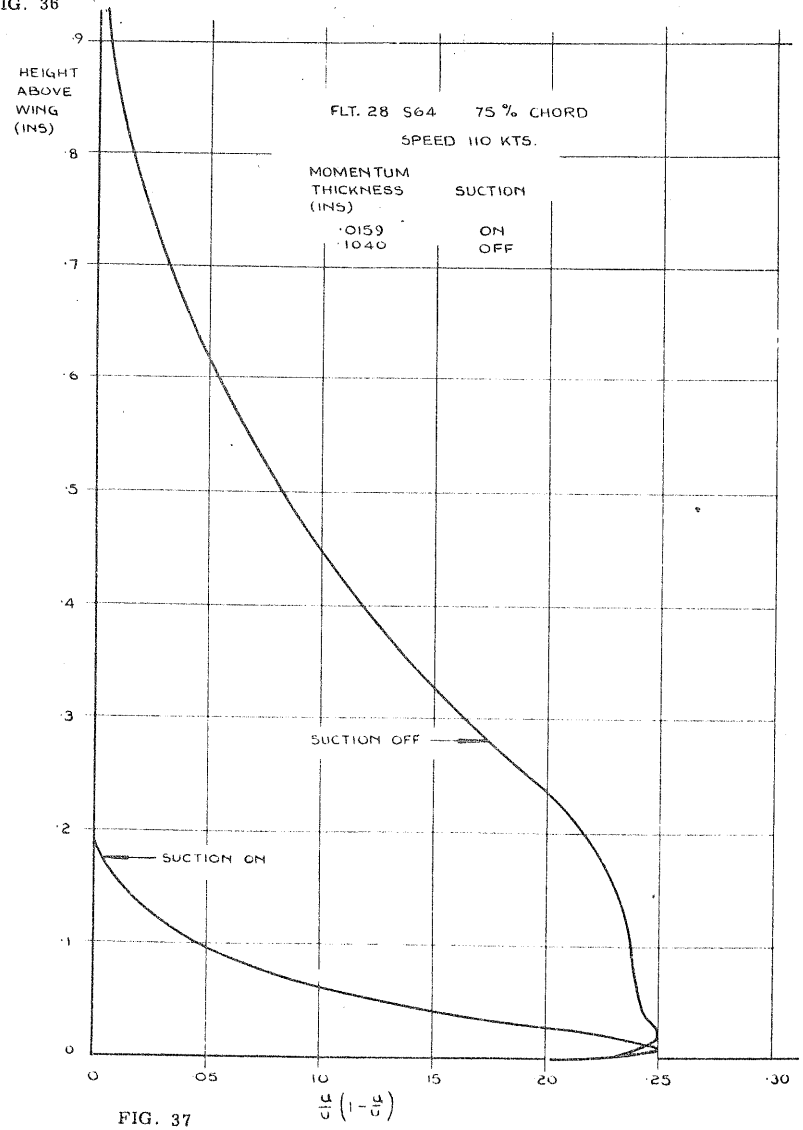


FIG. 37

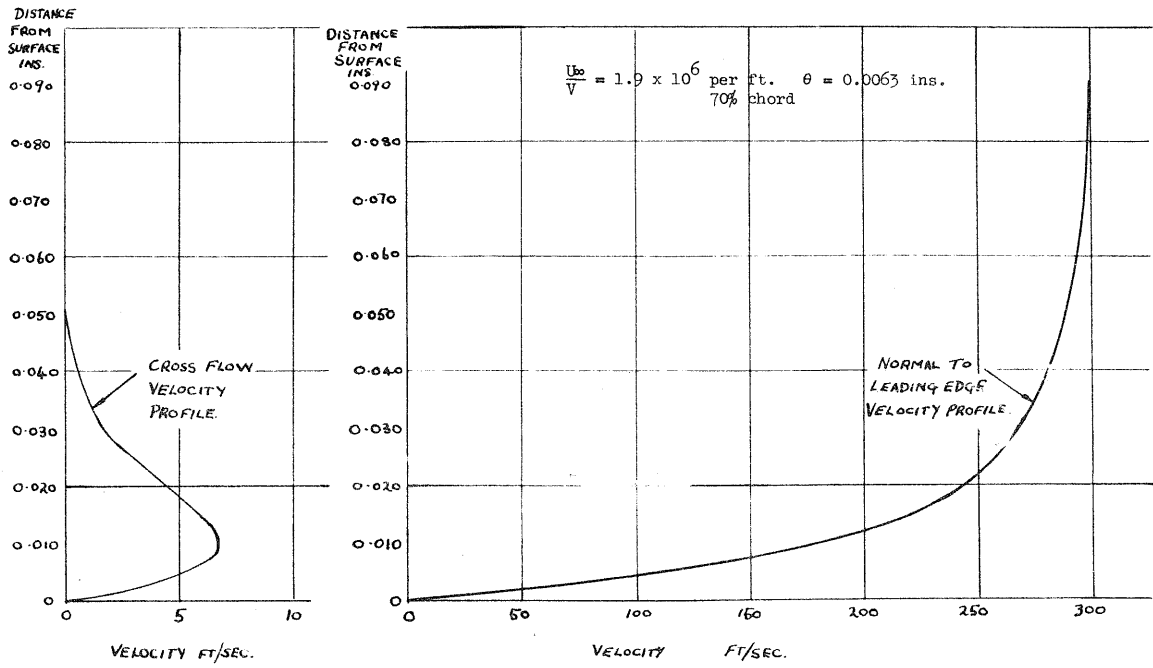


FIG. 38 TWO DIMENSIONAL SWEEP WING.

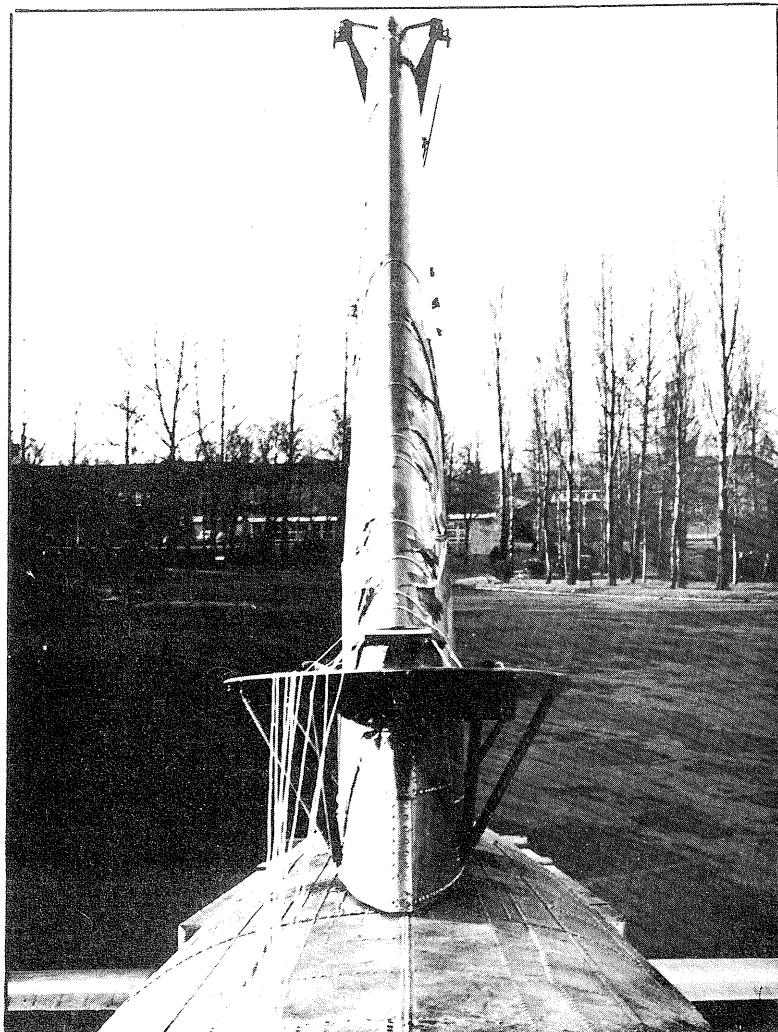


FIG. 39 ATTACHMENT LINE HOT FILMS READY FOR REMOVAL IN FLIGHT.