EXPERIMENTAL INVESTIGATION OF THE THICKNESS OF THE BOUNDARY LAYER AND THE LOCATION OF THE TRANSITIONAL REGION ALONG A WING SECTION

Thesis

by

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ACKNOWLEDGMENT

The author is indebted to the staff of the Guggenheim Aeronautics Laboratory, California Institute of Technology, and to the Douglas Company, for their able assistance in carrying out the investigation and development herein reported.

In particular, he wishes to thank Dr. Th. von Karman, Director of the Laboratory, Dr. C. B. Millikan and Dr. E. E. Sechler under whose supervision the problem was undertaken, and Mr. George Tsubota who was responsible for the detail design of much of the apparatus.

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SUMMARY

A general review of the investigation of the thickness of the boundary layer and the location of the transition point on an airplane wing in free flight that have been published to date is presented, along with some of the conclusions that have been reached from the results of the investigations.

The development and design of apparatus for the experimental determination of the thickness of the boundary layer, and the location of the point of transition from laminar to turbulent flow in the boundary layer which can be readily attached to the surface of a metal airplane wing is described.

Also the method and technique developed for recording the data and the interpretation of the test results are described in detail.

INTRODUCTION

The present investigation and development was started upon the suggestion of Mr. Arthur E. Raymond, chief engineer of the Douglas Aircraft Company, and because of the interest created in the subject by the Wright Brother's Lecture presented before the Institute of the Aeronautical Sciences on December 17, 1938, by Professor B. Melvill Jones of Cambridge University, England (Ref. 1).

Everyone associated with aeronautics is cognizant of the general knowledge concerning the subject: that the resistance to motion of aircraft moving through the air, arises chiefly from the friction developed between the air and the exposed surface of the moving body. This skin friction, is developed in a relatively thin layer of air immediately surrounding the exposed surface, and the air flow in this "boundary layer" can take one of two forms, a smooth "laminar" flow in which there is no mixing or interchange of momentum between the laminar layers, or a "turbulent" flow in which there is mixing and interchange of momentum between the layers.

The flow in the boundary layer is laminar over the forward portion of the exposed surface, changes to turbulent flow over a comparatively short distance known as the "transition region," and remains turbulent over the after portion of the surface. The friction associated with the two types of boundary layer flow differs considerably, that produced when laminar flow is present being much less than when turbulent flow occurs. Hence, the mean drag coefficient of the whole exposed surface depends upon the percentage of total area over which the two types of boundary layer flow are present, or the position on the surface at which transition from laminar to turbulent flow takes place. For minimum drag then, it is necessary to maintain laminar flow over as great a portion of the exposed surface as is possible.

Years ago, when the drag of airplanes was produced largely by wires, struts, landing gear and supports, and engine nacelles, the profile drag of a wing was but a small per cent of the total drag, and reduction in the drag coefficient of the order of magnitude possible by changing the location of the transition point would have been of little or no importance to an airplane designer.

Today, however, the picture has changed and aerodynamic and structural refinement has reduced the drag items of modern planes to the wings, fuselage, and tail surfaces. With the continual demand for increased performance, small reductions in the drag coefficient become of increasing importance, especially in the case of long range aircraft where a slight reduction in the drag means a considerably less fuel load for a given range, or a greater range with a fixed fuel load.

Little or no experimental work has been done on large commercial production airplanes operating at a much higher Reynolds number than is obtainable in a wind tunnel or than has been obtained in free flight experiments. It was with this in mind that Mr. Raymond kindly offered the Douglas DC-4 to the Guggenheim Aeronautics Laboratory, California Institute of Technology, for a boundary layer investigation. The DC-4 has a flush riveted wing operating at a Reynolds number of approximately 35×10^6, and the test results, which will be the subject of a later report, should reveal many interesting facts. The apparatus, the design of which is the subject of this report, was designed for use in the DC-4 investigation.

REVIEW OF PREVIOUS INVESTIGATIONS

The scientific and technical literature on skin friction, turbulence, boundary layer flow, and the transition region is very extensive. By far the greater portion of the experimental work has been done in wind tunnels in an attempt to correlate theoretical conclusions and experimental results. Likewise the few free flight investigations of boundary layer flow, concerning which information has been published, have been made for the purpose of correlating theory and experiment, as a check on the results obtained in wind tunnel experiments, and to find, if possible, the factors or parameters which may control transition from laminar to turbulent flow. These tests have been made on experimental wing surfaces which have been very carefully built, smoothed, and polished, and were conducted at Reynolds numbers far below those obtainable in our present day transport airplanes. Very few data have been published on the boundary layer thickness or the location of the transition point on modern production airplanes.

The information concerning the subject which is presented below is largely a review of the work of Cuno (Ref. 2) and Stüper (Ref. 3) in Germany, and Jones and his colleagues (Ref. 1) in England.

All three experimenters used essentially the same method for measurement of the thickness of the boundary layer and the location of the transition point: that of measuring the pressure as recorded by a total head pitot tube located at varying distances above the surface of the wing, similar measurements being taken at several stations along the wing chord. This method is described in detail under TEST PROCEDURE.

Cuno, who was interested only in the boundary layer thickness, found this to vary on the upper surface from 0.16 inch at the leading edge, to 2.36 inches at the trailing edge, for a wing of 72 inch chord operating at a Reynolds number of 4.5×10^6 (Fig. 1). The flight tests showed that the boundary layer thickness thus determined agreed with the wind tunnel data, and with the theoretical calculations from the velocity distribution up to 70 per cent of the chord, from which point there was considerable departure, the thickness increasing more rapidly than would be indicated by theory (Fig. 2). Cuno attributed this discrepancy to the effect of the increasing vortex formation at the trailing edge of the airfoil upon the boundary layer thickness.

Stüper, who attained greater accuracy in his investigation, likewise found close agreement between theory and experiment. The boundary layer thickness varied from 0.08 inch on the suction side at the leading edge to about 1.5 inches at the trailing edge for a wing of 72 inch chord and operating at a Reynolds number of 2.82×10^6 to 4.88×10^6.

He also found that the transition point is in close agreement with wind tunnel tests and occurs at about 10 to 30 per cent of the wing chord from the leading edge. The rearward motion of the transition point due to decreasing angle of attack was less than the forward motion due to increasing Reynolds number; hence the transition point shifted slightly forward with increasing velocity and decreasing C_L. In general, the transition on the pressure side occurs sooner than that on the suction side. An attempt was made to find the effect of Reynolds number on transition, but the scatter was so great as to indicate that the Reynolds number effect was being overshadowed by other effects.

Many investigations have been devoted in recent years to a study of the flow conditions around bodies held at rest in a stream of fluid, with a view to throwing some light on the factors which influence drag. Jones, in his investigation on the boundary layer, has been able to conduct similar experiments in free flight to show the effect of the location of the transition point on the wing drag coefficient. The drag coefficient was determined in flight by the method in which small pitot and static pressure tubes are made to traverse the wing wake. The location of the transition point was determined by a method similar to that described in the next section.

Figure 3 illustrates the variation of drag coefficient of a smooth wing which takes place as the point of transition from laminar to turbulent flow moves backward or forward along the wing profile. These curves were obtained from theoretical calculations and were closely checked by the experimental results. It will be noted that the decrease in drag coefficient with rearward travel of the transition point is considerably greater for the thicker wings. This is of particular importance in long range aircraft which have relatively thick wings, and also emphasizes the desirability of maintaining laminar flow over as great a portion of the wing as possible.

In Figure 4 is shown the point of transition plotted against the lift coefficient of the airplane as a whole for a wing thickness of 18%. The location of the transition point is measured from the front stagnation point around the wing profile. These curves are characteristic of all the wings tested up to 30% thickness. On the upper surface the transition point moves forward with increasing lift coefficient and the point is very definitely established. On the under surface the movement of the transition point is to the rear with increasing lift coefficient, but it is less definitely located and appears to be very sensitive to airspeed. In general, the transition point on the under surface is located further forward than that on the upper surface at low values of lift coefficient. It should be noted that the direction of travel of the transition point with increasing speed as observed by Jones is just opposite to that observed by Stüper. It is possible that this discrepancy may be due to the use of different wing sections with correspondingly different pressure distributions. From wind tunnel and free flight tests it has been found that the pressure distribution is an important factor in transition from laminar to turbulent flow.

An interesting point observed by Jones is that test data could be checked on many different days, indicating that there is no small-scale turbulence present in the atmosphere which influences the system of flow around an airplane. In wind tunnel tests, the turbulence of the free stream is known to have a marked influence of transition.

The conclusions to be drawn from the investigation carried on by Jones are: (1) That with the transition point known, the drag of smooth wings of moderate thickness can be calculated fairly accurately from the known values of skin friction on a smooth plate. (2) That at wing Reynolds numbers of five to ten million, transition can be delayed to distances of greater than 30% of the chord, this delay being accompanied by drag reductions of 30-35% from that obtained when the flow over the entire surface is turbulent. (3) That the thickness of the laminar boundary layer just before transition is of the order of 0.05 inch. (4) That very small imperfections of the wing surface have a marked influence on the transition point and consequently on the drag. It has been found that transition can be made to occur at any desired point along the wing chord by placing a wire of 0.01 inch diameter at that point. Also a piece of paper 0.002 inch thick will appreciably alter the point of transition.

TEST PROCEDURE

There are three methods generally used for detecting transition from laminar to turbulent flow and determining the boundary layer thickness. These methods are: (1) the hot-wire method, (2) measurement of velocity gradients at the surface, and (3) measurements made in the boundary layer with small total-head tubes.

The first method depends upon the variable cooling effect of eddying air currents upon a small electrically heated wire placed in the airflow. The varying current produced by the changes in resistance can be amplified, and by means of an oscillograph the velocity fluctuations can be studied. The apparatus required for this method is delicate and of such size as to make its use difficult in free flight experiments.

The second method depends upon measurements of the velocity distribution along a number of normals to the surface of the wing with a small total-head tube. This method can be used in free flight experiments, but the process is tedious and laborious as many readings at various wing chord locations must be made. Characteristic velocity distribution curves obtained by this method are shown in Figure 5. It will be noted that the slope of the curves at the origin varies according to the distance of the test section from the front stagnation point. As the distance aft increases the slope decreases, until a position is reached when, over a comparatively short distance, it undergoes a marked increase in magnitude, indicating that the velocity of the air nearest the surface has been increased. This point is called the transition point. The increased velocity is due to the mixing and interchange of momentum between the layers of the turbulent flow. Hence, the velocity of the air nearest the surface is increased by being mixed with higher velocity air from the upper layers, and likewise, the velocity of the outer layers is decreased by being mixed with the slower moving air from near the surface.

The third method depends upon the characteristic noted in the discussion above: that there is a local increase in velocity in those layers lying nearest the surface in the vicinity of the transition point. To determine transition it is only necessary to move a total-head tube along the chord of the wing at a fixed distance from the surface. A plot of total pressure against distance along the chord will exhibit characteristics from which the transition point can be determined. Depending upon the distance of the tube from the surface of the wing, this curve will display somewhat different trends, as explained below.

Typical velocity cross-sections just before and after transition are shown in the left-hand diagram of Figure 6.

The right-hand diagram of this figure shows the changes that take place in the pressure as recorded by pitot tubes moved through the transition region along lines parallel to the wing surface, such as AA, BB, CC, shown in the left-hand diagram. From the sketch it will be seen that a tube in the laminar boundary moving along BB or CC aft along the chord, will register a slowly decreasing pressure as the laminar layer thickness increases. Passing into the transition region, it will register an increasing pressure until a peak is reached, and then a slowly decreasing pressure as the turbulent layer thickens. Conversely, a pitot tube moving along AA outside the laminar layer, will register a falling pressure as it passes through the transition region. The transition point is assumed to be at the point where the rise or fall of pressure starts, as shown in the figure.

Strictly speaking, transition from laminar to turbulent flow does not occur suddenly, but develops gradually over an appreciable distance; hence one should speak of a transition region rather than a transition point. It has also be shown by Dryden (Ref. 4) and others, that the transition region does not remain stationary but fluctuates rapidly along the chord. However, the apparatus generally employed does not respond to the rapid pressure fluctuations accompanying the movement of the transition region, but indicates a mean position for transition.

A manometer of the multiple "U" type, schematically shown and connected as indicated in Figure 7, is used to measure the various pressures, which may be recorded on sensitized paper photographically or with pencil. By connecting two manometer tubes to the total-pressure tube of the aircraft pitot static head, a base line is established from which to measure the other pressures. If the total-pressure loss and the static pressure are divided by the impact pressure ($1/2 \rho V^2$), the values obtained will be independent of the tilt of the airplane, fluid or air densities. The static pressure tube is employed to determine the pressure distribution over the wing, but is not necessary for the determination of the transition point or boundary layer thickness.

In order to determine the boundary layer thickness it is only necessary to employ several pitot tubes placed at various distances from the wing surface and moved along the chord. The record obtained will be as shown in Figure 8, and the point along the chord at which each tube enters the boundary layer can be determined and the boundary layer profile plotted as shown.

The spacing of the tubes along the wing surfaces will depend upon the maximum thickness of the boundary to be measured. The thickness of the layer at the trailing edge of a wing may be calculated by the formula, (Ref. 5)

$$\delta = 0.38t(C_f)^{1/2}$$

where t is the wing chord and C_f is the coefficient of skin friction of a smooth plate as shown in Figure 9. The effective thickness of the laminar layer just before transition upon a smooth wing is seldom much greater than 0.05 inch.

Hence, to determine the transition point and the boundary layer thickness in free flight it is only necessary to traverse the wing chord with several pitot tubes while flying under steady conditions. In addition to recording the manometer readings, pressure altitude and free air temperature should also be recorded, from which the air density and hence the Reynolds number can be calculated.

An alternate method is to fix the pitot tube head to the wing surface at the estimated location of the transition region, and then by varying the air speed of the airplane the transition region will move past the pitot tubes. In this case the total-head pressures are plotted against the lift coefficient of the airplane as a whole to determine the transition. Since the transition region moves aft with increasing lift coefficient, the slopes of the pressure curves are reversed as shown in Figure 10.

DESCRIPTION OF APPARATUS

Keeping in mind the test procedure outlined in the previous section, an attempt was made to design equipment which would fulfill the following requirements:

(1) The apparatus must be capable of installation on any airplane.

(2) Experiments on commercial airplanes being contemplated, attachment of apparatus to wing must be simple and require as few alterations as possible.

(3) All equipment must be installed outside of wing with the exception of tubing connecting pitot tubes to manometer in fuselage.

(4) Device carrying pitot head must be capable of traversing the wing from leading to trailing edge on either upper or lower surface while in flight, and position must be remotely controlled from the cabin.

(5) Position of pitot tube along the chord must be readily determined from the cabin.

The equipment finally evolved is shown in the photographs, Figures 11, 12, and 13, and in the assembly drawing, Figure 14.

The track is constructed of two lengths of an extruded dural stiffener section riveted to a narrow aluminum sheet. The stiffeners and sheet were formed to fit the wing profile, and in the case of the DC-4, replaced the cover plate over the wing joint between the inner and outer wing panels.

The car for carrying the pitot arm is likewise constructed of dural and is held rigidly against the track by the four springs shown. These springs allow the car to roll around the curved portion of the leading edge without binding. Movement in a lateral direction is prevented by the grooved bearings rolling along the bulb of the stiffener section.

Movement of the car fore and aft along the wing chord is accomplished by means of a roller chain lying between the tracks, fixed at both ends and running over the idler sprockets to the sprocket mounted on the motor shaft. An autosyn mounted on the rear of the car is also geared to the motor shaft through a worm and worm gear and provides a means of remotely indicating the position of the pitot head in the cabin.

The pitot arm shown in Figure 15 extends to the side and forward, and is held in firm contact with the wing surface by means of the coil spring attached to the arm and the car base.

The five flexible rubber tubes connecting the pitot tubes with the manometer, and the two five-conductor electrical cables to the motor and autosyn, were formed into one cable and covered with a woven fabric sleeve. This cable proved to be very flexible and was allowed to slide back and forth, through a metal tube installed in the trailing edge of the wing, as the car moved along the chord.

As it was desired that all data be recorded by means of a motion picture camera, the instrument was limited to a maximum size of 15 x 20 inches. This necessitated the use of a manometer fluid of greater density than is usually employed and one that would not attack the rubber tubing used in the construction of the manometer. It was found that a sixty per cent solution of zinc chloride having a specific gravity of 1.5 was satisfactory for all pressures except the exploring static pressure. Rather than sacrifice accuracy by using fluid of a greater density than 1.5 for all the pressures, a separate four-tube manometer was constructed using Telegage fluid with a specific gravity of 2.8 to measure the static pressure along the wing. The two outer tubes were connected to the aircraft pitot and one other tube to the static bomb to give a reference line from which to measure the exploring static pressure. The instrument panel is shown in Figure 16.

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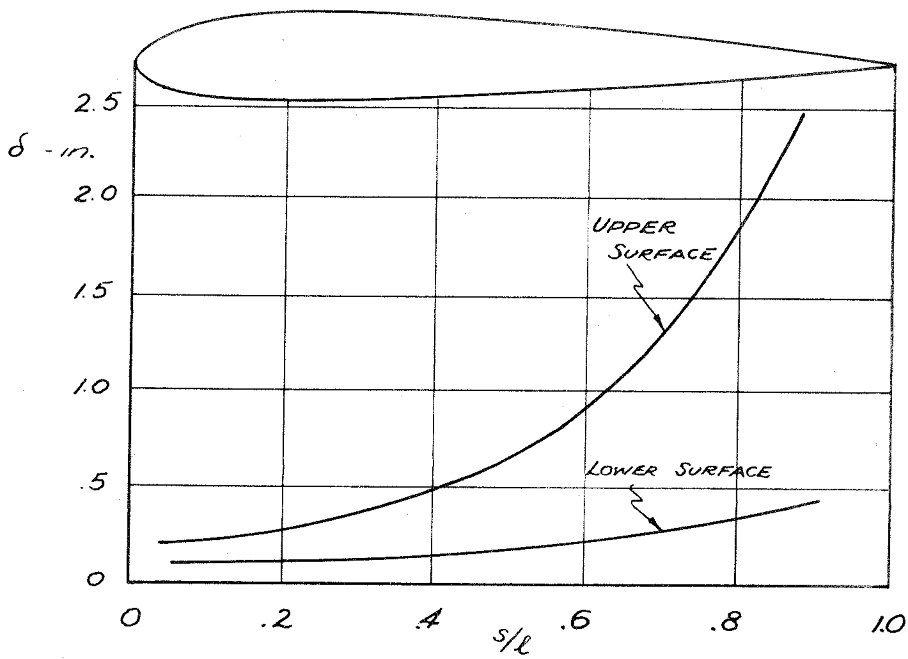


Fig. 1 Thickness of boundary layer along wing chord.

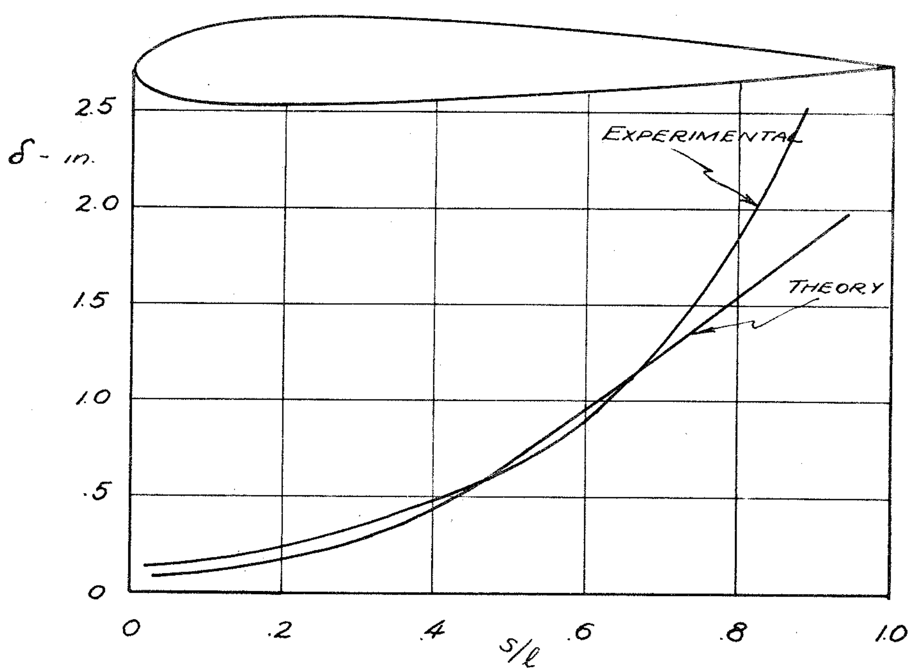


Fig. 2 Theoretical and experimental boundary layer thickness on upper surface of wing.

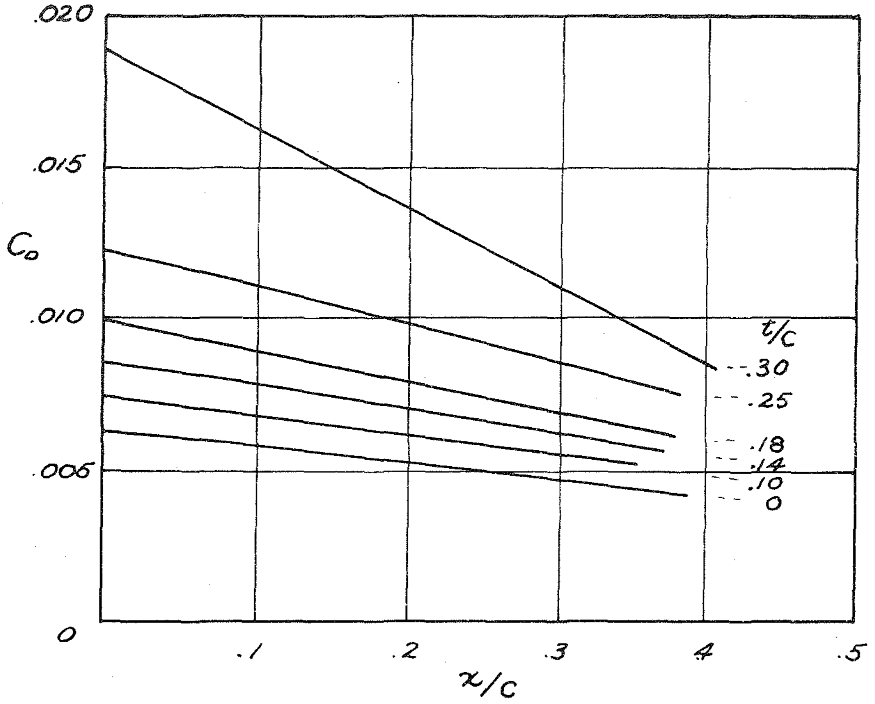


Fig. 3 Variation of drag coefficient with location of point of transition.

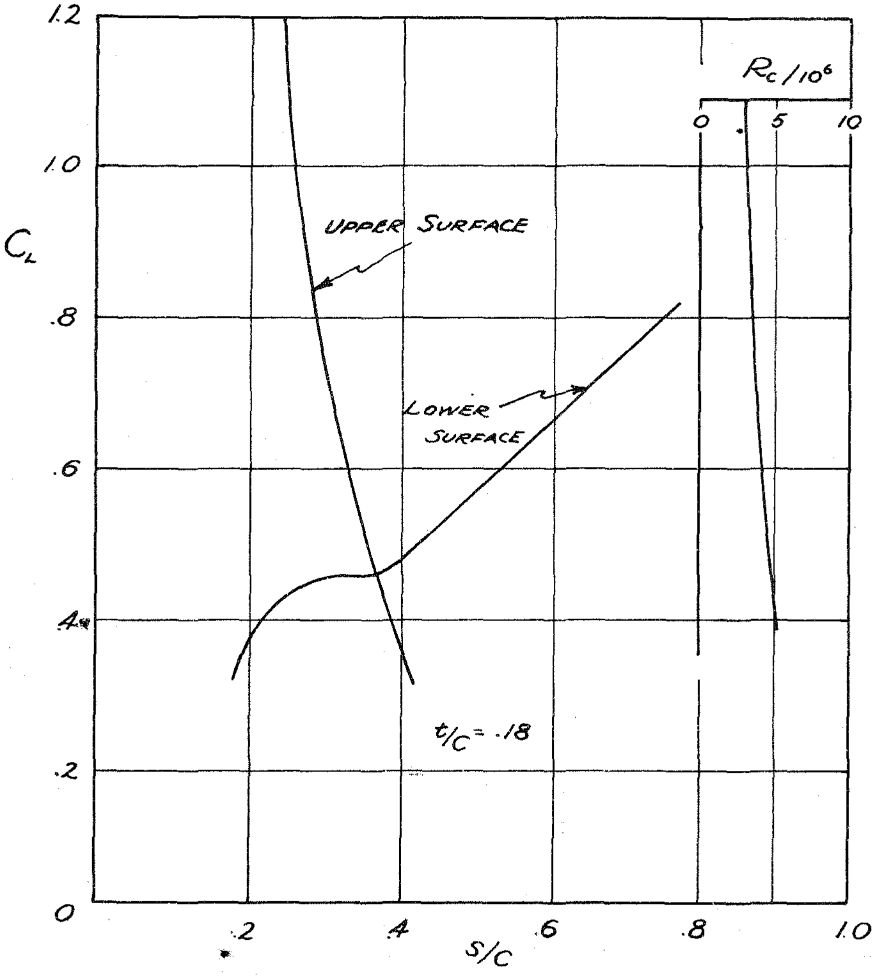


Fig. 4 Variation of location of transition point with lift coefficient.

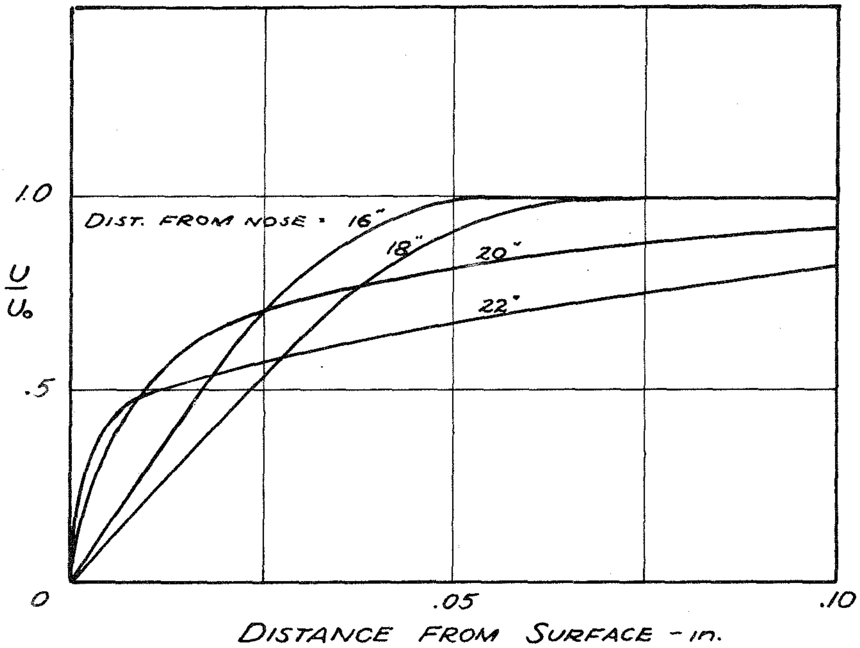


Fig. 5 Velocity distribution normal to the surface at sections near the transition point.

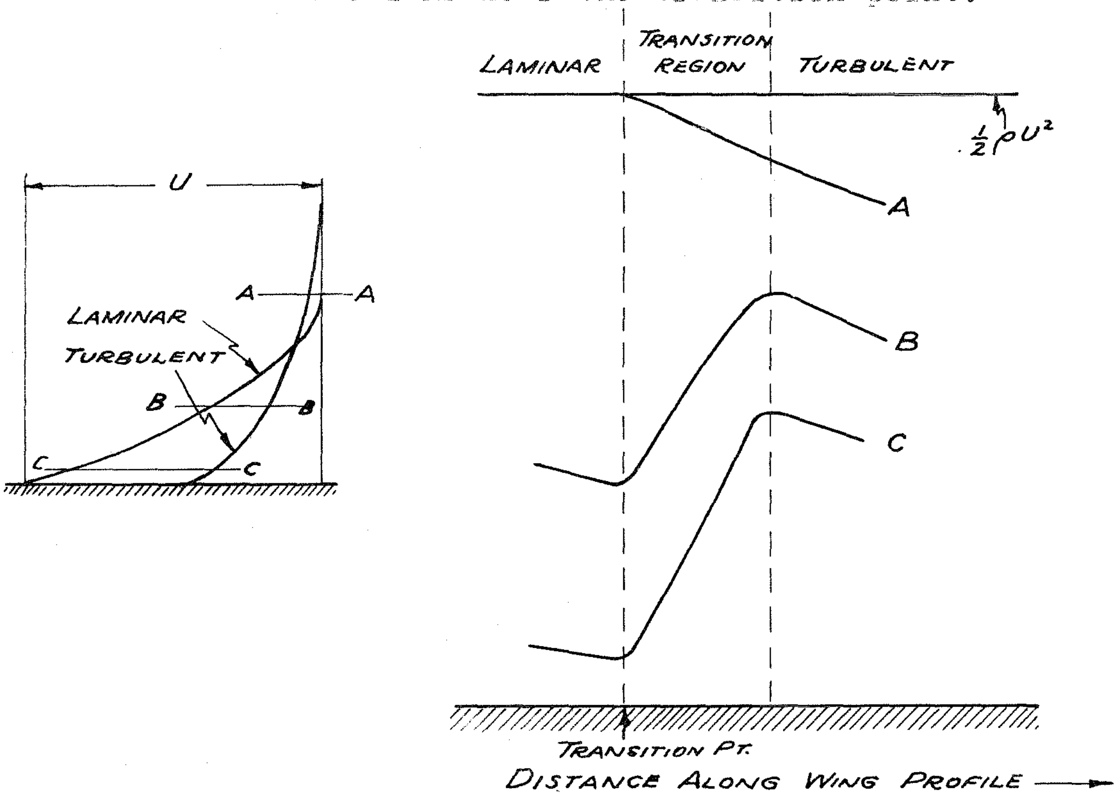


Fig. 6 Diagrams explaining method used for locating the region of transition.

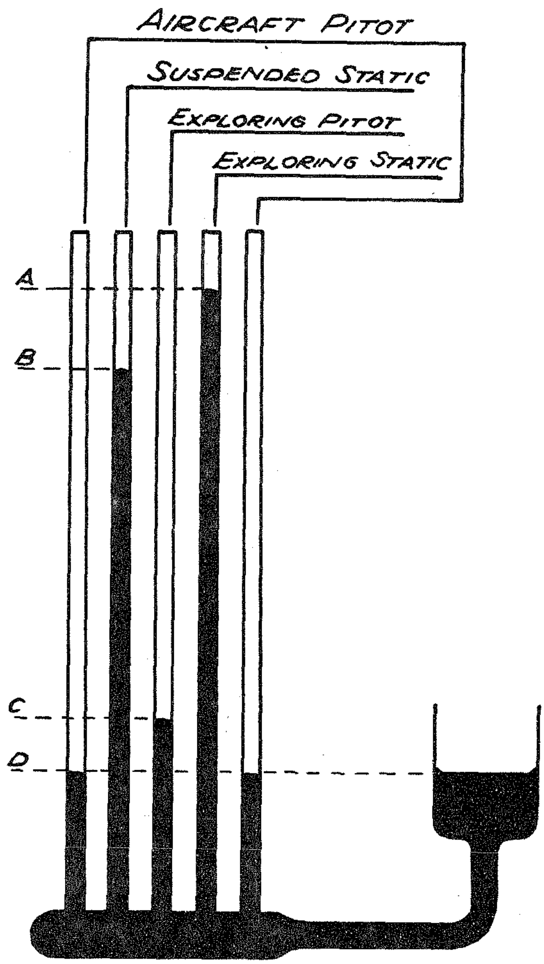


Fig. 7 Schematic diagram of manometer connections.

BD indicates the "impact pressure" ($\frac{1}{2}\rho U^2$), CD the total-pressure loss at the exploring pitot tube, and AB the static pressure (below atmospheric) at the exploring static pressure tube. Expressing total-pressure loss and static pressure in coefficient form by dividing by the impact pressure makes the values obtained independent of airplane tilt, acceleration, fluid or air density.

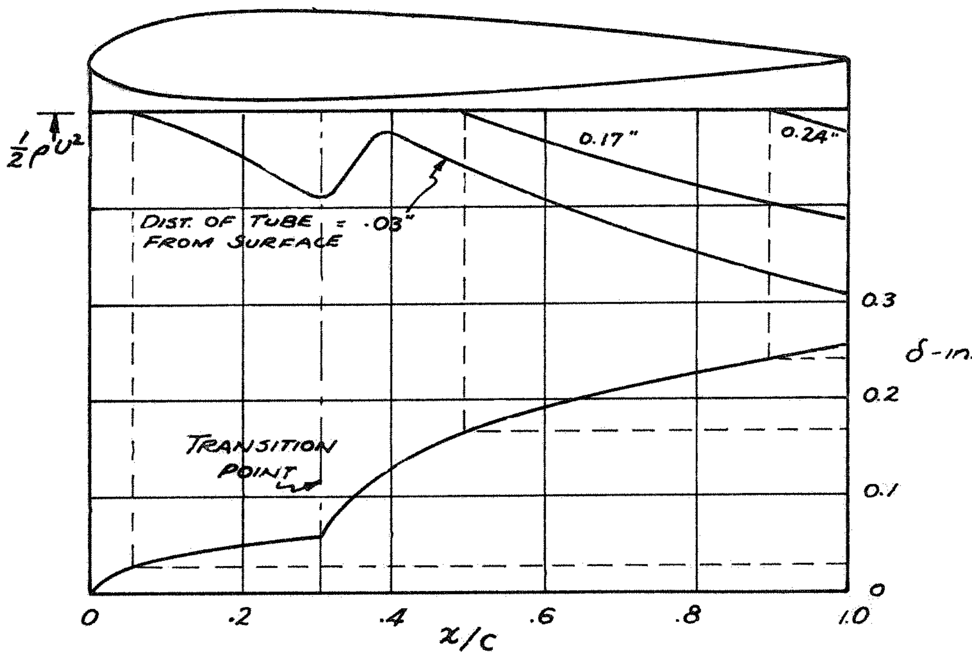


Fig. 8 Diagram explaining method of determining the boundary layer thickness and profile.

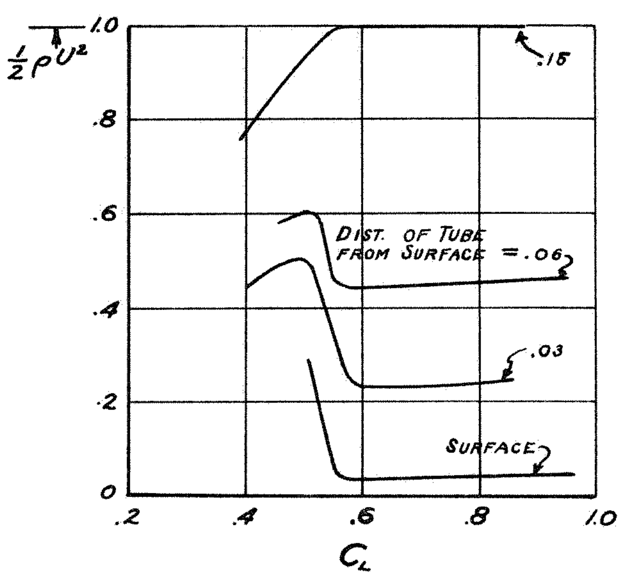
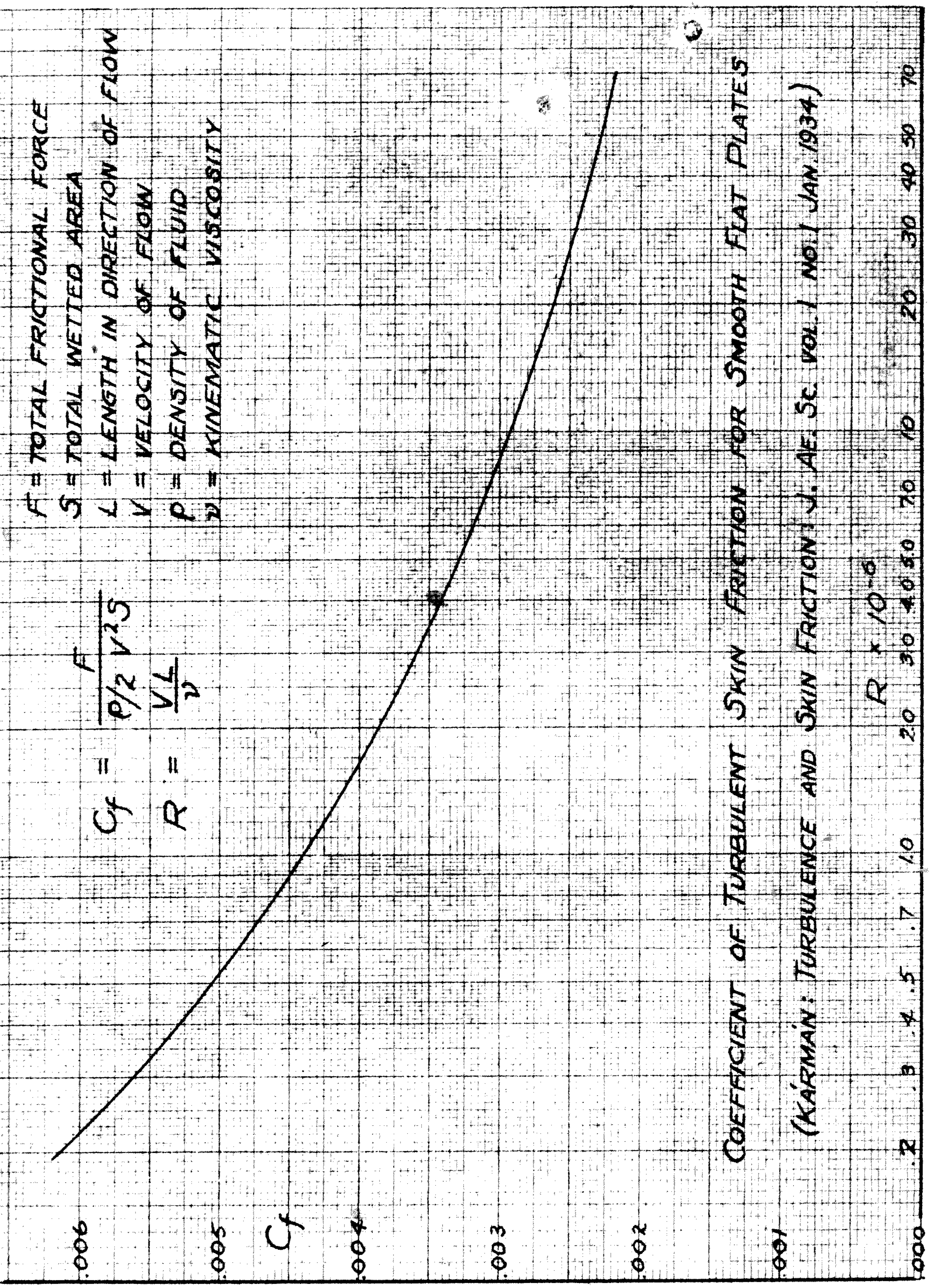
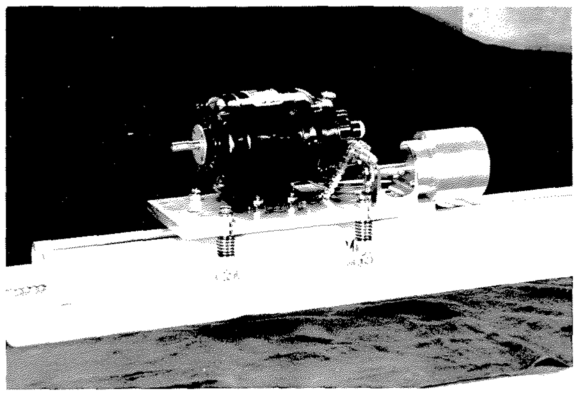


Fig. 10 Variation of total pressure with lift coefficient as recorded by pitot tubes fixed to wing.



COEFFICIENT OF TURBULENT SKIN FRICTION FOR SMOOTH FLAT PLATES

(KÁRMÁN: TURBULENCE AND SKIN FRICTION: J. AE. SC. VOL. I NO. I JAN. 1934)



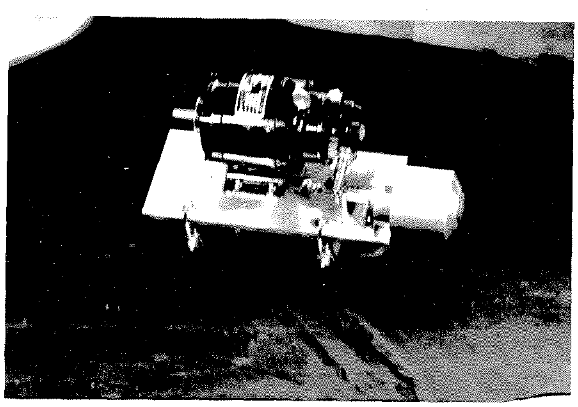
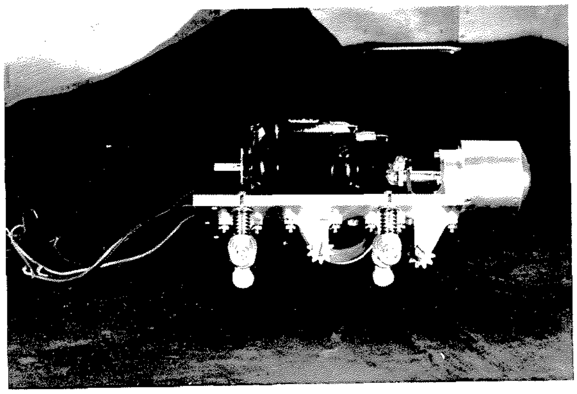


Figure 11



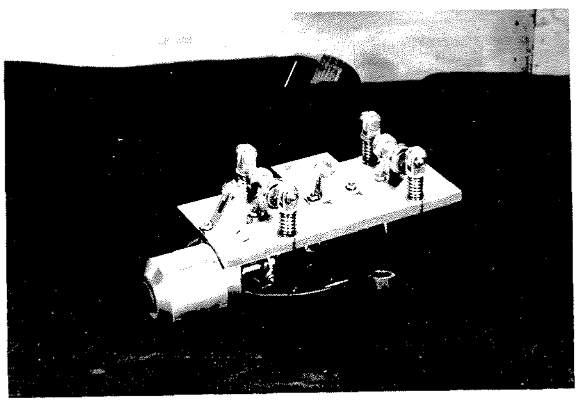
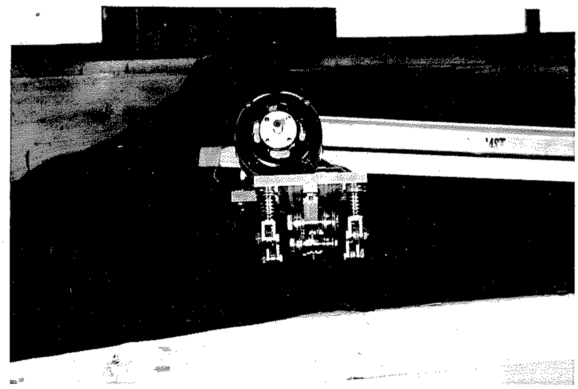


Figure 12



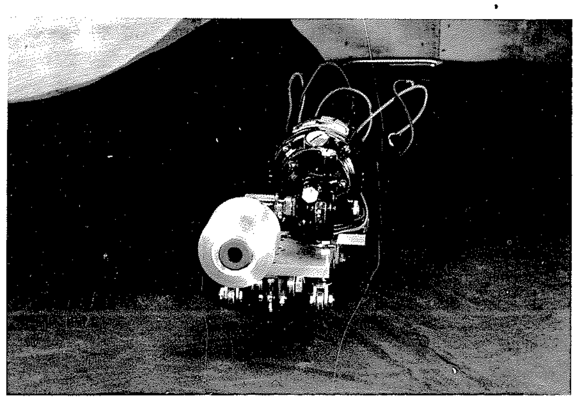
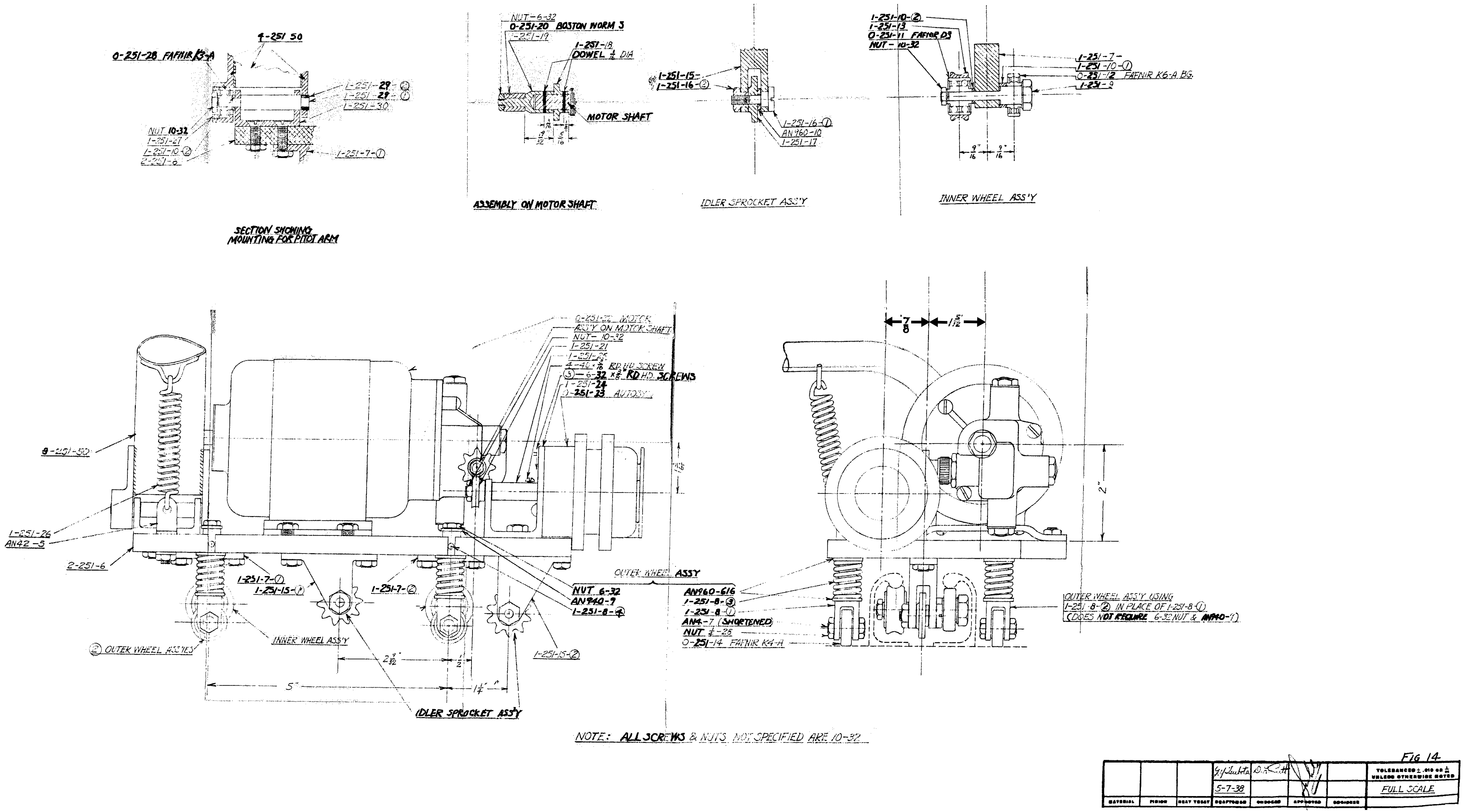
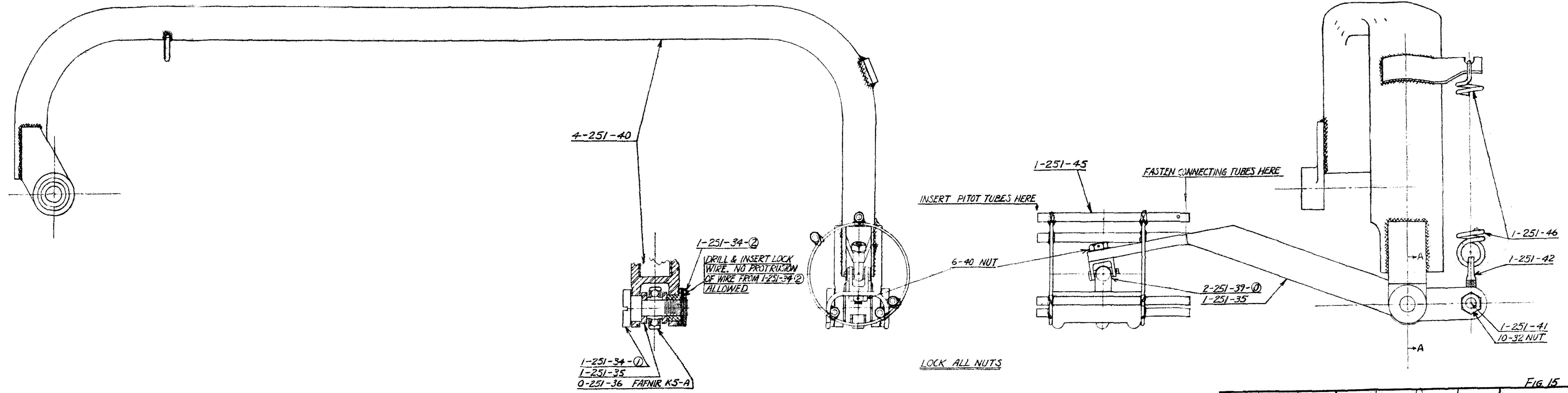


Figure 13



SECTION SHOWING MOUNTING FOR PITOT ARM
ASSEMBLY ON MOTOR SHAFT
IDLER SPROCKET ASS'Y
INNER WHEEL ASS'Y
OUTER WHEEL ASS'Y
NOTE: ALL SCREWS & NUTS NOT SPECIFIED ARE 10-32
FULL SCALE
5-7-38

FIG 14



SECTION A-A

ASSEMBLE ①

FIG. 15

				R.S.S.			TOLERANCES ± .010 OR 1/64 UNLESS OTHERWISE NOTED
			6-14-38				FULL SCALE
MATERIAL	FINISH	HEAT TREAT	DRAFTSMAN	CHECKED	APPROVED	ENGINEER	
GUGGENHEIM AERONAUTICAL LABORATORY CALIFORNIA INSTITUTE OF TECHNOLOGY			PITOT ARM ASSEMBLY				3-251-50
			NAME				DRAWING NO

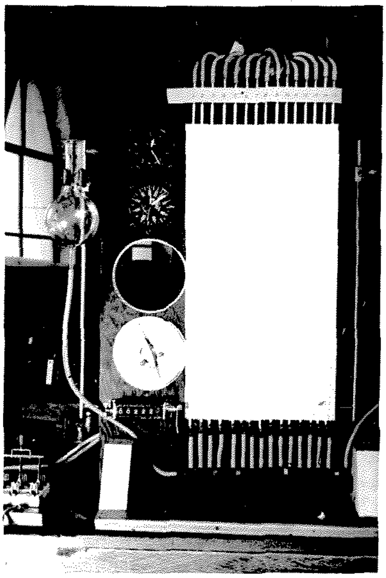


Figure 16