EXPERIMENTAL INVESTIGATION OF THE PROFILE DRAG AND BOUNDARY LAYER OF A WING SECTION DURING FREE FLIGHT AND IN THE WIND TUNNEL

Thesis

by

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SUMMARY

A general report covering the method used in obtaining flight test and wind tunnel data for the investigation of profile drag and the thickness of the boundary layer with its transition point is presented, together with some of the results and conclusions reached.

Methods of data reduction with the difficulties encountered are given. After final results of the profile drag for flight test are obtained, they are compared with T. N. 695.

A detailed description of the apparatus with the technique used for both the flight tests and the wind tunnel is also presented.

INTRODUCTION

It has only been in recent years that the importance of profile drag has been stressed. Previously when airplanes were constructed with struts, flying wires, fixed landing gear, etc., the change in the total drag due to a small decrease in profile drag would have been of such a small order of magnitude that it would have been of little importance to the designer.

The total drag of such an airplane (C_D based on wing area) was formerly approximately 0.040 to 0.050 of which a change of 0.002 in profile drag meant a 4 to 5% change of total drag with a resulting change of 1.3 to 1.6% in the top speed. Now, due to the elimination of external bracing, the introduction of retractable landing gears, and a general cleaning up of the airplane, the total drag has fallen to around 0.015, and it can be seen that now a 0.002 decrease in profile drag results in a 5% increase in top speed. Thus with the speeds we are reaching today and expect to get in the future, it is important that the designer have accurate data on the profile drag in order to secure better performance.

The usual method of obtaining such data was by means of the wind tunnel. This method has two disadvantages: first, due to the turbulence of the air caused by the tunnel;

and second, that due to the scale effect. The change in profile drag that turbulence produces is explainable by present theories of flow in the boundary layer but although the turbulence can be measured, it appears very difficult to estimate quantitatively its effects on skin friction. Also in flight the Reynold number is going up (DC 4 approximately 35×10^6) so that most wind tunnel results have to be extrapolated.

Therefore it becomes advantageous and important for increased airplane performance, for one to be able to measure the drag of its component parts in flight. It was thus deemed advisable to return to the method suggested by Betz in 1925, i.e., that the drag on a body can be deduced by completely exploring its wake and measuring the loss of momentum of the air as it flows past.

Shrenk, in 1928, published a report describing a number of observations he had made in flight by this method. In his experiments and observations he used wing surfaces of varying degrees of roughness and his results were calculated by the Betz formula. At that time interest in profile drag was not great and although his results were interesting academically the subject was dropped and nothing more was done on it until Prof. B. M. Jones made his experiments and published his results several years ago.

His report, R. & M. 1688, thoroughly explains his method, known as the Pitot-Traverse, which merely involves an examination of the wake left behind in the air by the component part in question. This method is capable of very general application to give estimates of the energy loss due to various parts of the airplane in flight or in the wind tunnel. Its importance thus lies in the fact that parts of the airplane may be considered separately and at the present time, it appears to be the only effective way of obtaining the required measurement in free air. This method of obtaining the profile drag is accurate, quick, and convenient, and gives a means to explore the flow behind an airplane to find regions where the energy losses are excessive.

Jones also carried on boundary layer experiments, using his method, and the results were published in the Journal of Aeronautical Sciences, Vol. 5, No. 3, January, 1938.

As the profile drag depends mainly on skin friction, whose magnitude in turn depends on surface roughness and irregularities, and on the location of the transition point of the boundary layer on both upper and lower surfaces, it was deemed advisable in our work to group both the pitot-traverse and the boundary-layer surveys into one project. This was done and experiments were carried out by GALCIT,*

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*Guggenheim Aeronautical Laboratory, California Institute of Technology.

under the direction of Dr. C. B. Millikan, with the assistance of Mr. L. B. Rumph and Mr. R. Schairer. The boundary layer survey is a continuation of the thesis of Lt. D. Putt, A. C., U. S. Army, in 1938.

Aircraft for the free flight tests were made available by the Douglas Aircraft Company in the use of the DC 4, and the Lockheed Aircraft Corporation in the L 14. For wind tunnel tests there were the large highly polished wing, the 10 ft. tunnel, and other equipment of GALCIT.

The measuring equipment used was designed specially for the DC 4, and with slight changes in mounting was used on the L 14. For the wind tunnel other slight additional alterations were made.

REDUCTION OF DATA FOR PITOT-TRAVERSE

The method used in finding the profile drag of any section of a wing is that of Prof. B. M. Jones. This is thoroughly given in R. & M. No. 1688, January, 1936.

For the present free flight tests, due to only one flight being made in each airplane, the location of the pitot-traverse apparatus was restricted to a single position on each airplane. In the wind tunnel at GALCIT, observations and readings were taken at several positions along the span direction in order to calculate the drag for entire wing.

Unfortunately for the flight test of the DC 4, the pitot tubes were in the slip stream of the outboard propeller, and the results from the one run made with this propeller feathered, were inconsistent due to undetermined causes, so they had to be disregarded.

The corrections that had to be made are as listed:-

1. For the DC 4, a correction had to be made for the slip stream effect and another to correct the airplane measured static pressure. The latter was accomplished by using the airplane's static pressure calibration curve.

2. For the Lockheed 14, there were the corrections for the airplane static pressure and also the wing static pressure because of the support system. The correction

necessary for the latter was determined by making runs in the wind tunnel with and without the support system. The results were plotted as a curve from which values were taken to correct those of the flight test.

3. For the Big Wing in the wind tunnel (GALCIT Rep. 220), there was the correction for variation of dynamic and static pressure along the span. There was no correction necessary for the support system, as in the case of the Lockheed 14, due to the increased length in the rotating arm which protrudes from the box housing the operating mechanism.

The following is the formula from Jones and the method used to compute the profile drag:-

$$C_{D_p} = 2 \int \sqrt{g - p}\,(1 - \sqrt{g})\,d(\tfrac{y}{t}) \text{ ...... (1) (Jones Formula)}$$

$g = \frac{H}{H_o}$

$p = \frac{P}{H_o}$

H = total head at any point

P = static pressure at any point

H_o = free stream total head

y = vertical ordinate

t = chord

x = distance pitot tube opening is in rear of T. E.

Values measured:

B = angular displacement of arm

y = $\sin B$

H' = manometer reading in cms. of fluid. Connected to total head tube.

S' = manometer reading in cms. of fluid. Connected to static pressure ring of pitot tube.

T.S. = manometer reading connected to tunnel static ring.

H_1 = T.S. - H' fluid height proportional to total head.

S_1 = T.S. - S' fluid height proportional to static pressure at pitot tube.

$$g_1 = \frac{H_1 \times D_f}{q} \quad \text{where}$$

D_f = fluid density

q = free stream dynamic pressure.

Runs were made at constant q.

$H_1 \times D_f$ = actual pressure in $gms./cm.^2$

g_1 = dimensionless ratio (one of quantities in Jones formula).

$$p_1 = \frac{S_1 \times D_f}{q} \quad \text{same as for } g.$$

We can substitute in

$$f_1 = 2\sqrt{g_1 - p_1}(1 - \sqrt{g_1})$$

The subscript 1 refers to pitot tube No. 1.

f_1 is now plotted against $\frac{y}{t}$, Fig. 1.

The area under the curve is determined by the use of a planimeter and is equal to C_{Dp}.

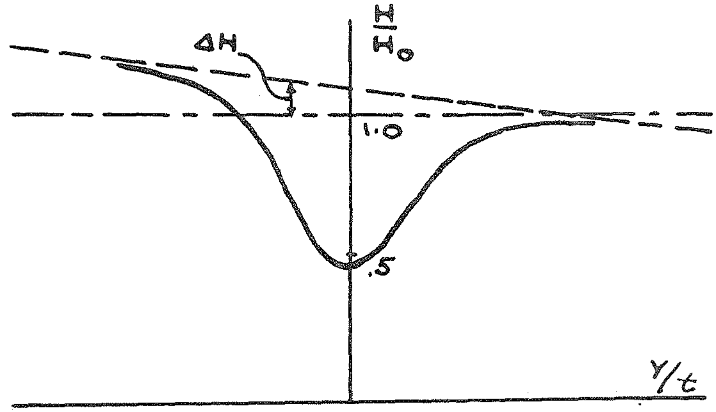
The measuring plane should be fairly close to the T.E. so that the wake measured will be from wing alone and not from some other interference, and also so that losses in the wake will be small, if any. Due to the construction of the apparatus, it is also more feasible to place pitot tubes close to the wing.

As previously stated, it was found in the DC 4 tests that the pitot-traverse tubes were in the slip stream. This made a correction necessary to secure the true C_{Dp}. In order to estimate this correction the following analysis was made:-

As the free stream total head was taken from the airplane pitot tube, it did not give the true value of the total head in front of the section in question due to the added effect of the slip stream. In order to calculate the profile drag, it was assumed that the total head outside the wake in the rear of the wing was the same as that in front of the wing.

The measured value downstream was then used instead of that from the airplane pitot-tube. On doing this, it was found that another difficulty introduced itself and this was that the total heal above the wake was not the same as that below.

In the actual computation a preliminary curve of H/H_o vs. y/t is plotted and if delta H be denoted as shown in the figure below, then the drag coefficient as expressed by equation (1) becomes



$$C_{Dp} = 2 \int \sqrt{g_c - p_c}\,(1 - \sqrt{g_c})\,d(\tfrac{y}{t}) \quad \ldots\ldots\ldots\ldots \quad (2)$$

where $g_c = \dfrac{H}{H_o + \Delta H}$, $\quad p_c = \dfrac{P}{H_o + \Delta H}$

or,

$$g_c = g\left(\frac{1}{1+\Delta g}\right), \quad p_c = \frac{1}{(1+\Delta g)}$$

$$\text{where } \Delta g \quad \frac{\Delta H}{H_o}$$

On rewriting (2) so that the slip stream correction factor appears explicitly, we have

$$c_{D_P} = 2\int \frac{1}{\sqrt{1+\Delta g}}\left(\sqrt{g} - p\right)\left(1 - \sqrt{g}\sqrt{\frac{1}{1+\Delta g}}\right) d\left(\frac{y}{t}\right)$$

The correction term $\sqrt{\dfrac{1}{1+\Delta g}}$ is a function of $\dfrac{y}{t}$ and varies with speed since the propeller thrust varies with speed. Its value was determined for each run from a preliminary plot of H/H_o vs. y/t.

In the wind tunnel we did not at first take into consideration the distribution of static pressure and of q across the span of the wing. Therefore in computing and plotting the profile drag for the entire span we obtained erroneous results as indicated by the unsymmetrical nature of the dotted curves.

To correct this and to use as many of our previous data and computations as possible the following assumptions were made: It was assumed that the static pressure was constant along the span and variation of "q" (as expressed by "a") is determined by a = g_{1F} - 0.009, where 0.009 was found from measurements outside of the wake at each spanwise position.

The details of this method of recalculating the data can be found in a report submitted by Mr. L. B. Rumph, Jr.

REDUCTION OF DATA FOR BOUNDARY LAYER SURVEY

In finding the transition point where the laminar boundary layer flow changes to turbulent flow, it should be remembered that this point fluctuates over a region of some width so that any experimentally determined location is not an exact point but rather represents a mean position.

To find this mean position two methods may be employed using the data secured by the apparatus discussed in Section VIII. The first method with small changes in the apparatus is essentially that which was developed and used by Jones, as described in the Journal of Aeronautical Sciences, Vol. 5, No. 3, January, 1938. This method involves traverses along the chord of the wing with tubes at various heights above the surface during free flight. These experiments were made in order to investigate factors which influence the drag, in particular the transition point location.

Our observations were made with four total head tubes and one static pressure tube. Thus it can be seen that for any position along the chord, the velocity profile can only be plotted from four points, one being given by each tube. However the readings do give a means by which the transition point can be found.

The method of Jones depends on the difference in laminar and turbulent boundary layer velocity profiles, in which there is a local increase in velocity in those layers near the surface in the transition region, see Fig. 3. Therefore by moving the total head tubes, which are at a fixed distance above the surface, along the chord of the wing from a point in front of the transition point, curves can be plotted showing the variation of total pressures against distance along the chord. It can be seen in Fig. 3, that a tube near the surface will show decreasing pressure as it is moved aft from the stagnation point, i.e., the laminar layer is increasing in thickness. When it enters the transition region, increasing total pressure will be shown due to the turbulent mixing, until the turbulent layer thickens, in which the pressure will decrease. This definite rise or hump in the total head tube curves for the tubes near the surface is the characteristic by which the transition point is located in this paper.

Another method, which is given by Albert E. Van Doenhoff in T. N. 671, may be employed with out data in determining the transition point. Briefly, this involves plotting the turbulent and laminar boundary layer thicknesses and then finding their intersection. The laminar layer thickness is found by using the measured static pressure gradient over the wing and a logical assumption as to the laminar velocity profile. The turbulent layer is found by using those tubes that we know are above the laminar layer. On being traversed across the chord from a point forward of the transition point, each tube will register the position where it enters the turbulent layer. These points when plotted give the curve of the turbulent layer.

Other methods may be used such as using the increase or decrease in the slope of the velocity profile normal to the surface, at sections near the transition regions. These were not used because the process is tedious, many readings at different chord points must be made, and also because our apparatus was not suited for it.

The method of reducing and presenting data is as follows:

The total head was measured at four different heights above the wing surface and also the static pressure at the point involved.

<u>Values recorded (Wind Tunnel)</u>

C.N. - counter number for identification.

d - distance in inches from L.E.

T.S. - fluid height in tube connected to tunnel static.

S_e - fluid height in tube connected to exploring static.

h_f' - fluid height in tube connected to static ring.

H_n - fluid height in tube connected to total head tube of creeper.

n - subscript for identification of total head tubes.

Values computed

$\frac{x}{t} = \frac{d}{t}$ where t = wing chord.

G_n = T.S. - H_n fluid height proportional to total head tube of creeper.

P_f = T.S. - S_e fluid height proportional to static pressure.

$$\left.\begin{aligned} g_n &= \frac{G_n}{q} \\ p &= \frac{P_f}{q} \end{aligned}\right\}$$

Dimensionless ratios proportional to total head and to static pressure.

$q = \frac{q}{h} h_f$ where

h_f = T.S. - h_f' fluid height corresponding to the total head in tunnel.

$\frac{q}{h}$ is taken from tunnel calibration curve.

($\frac{q}{h}$ is plotted as function of q)

$$\gamma = \frac{\text{Total head measured by any tube in the B.L. relative to free stream static pressure.}}{\text{Free stream total head relative to free stream static pressure.}}$$

For presentation of the data g_n is plotted as ordinate against $\frac{x}{t}$.

In the wind tunnel g_n does not become greater than one because if a tube is outside of the boundary layer, it measures the same total head which is used for reference.

In flight tests where the tubes are in the slipstream, the total head ratio, g_n, becomes greater than one because the total head in the slipstream is greater than that outside the slipstream and this latter is used for reference.

In flight the reference pressure was that measured at the static ring of the airplane pitot static tube. The value of "q" was that pressure difference which was measured by the airplane pitot static tube.

DISCUSSION OF PITOT-TRAVERSE RESULTS FOR BOTH FLIGHT TEST AND WIND TUNNEL

For a typical curve used in computing the profile drag see Fig. 1.

In the curves, Fig. 4a and 4b, there are plotted C_{D_0} vs. C_L of (1) the wind tunnel profile drag of the complete wing and (2) the flight test results which represented the profile drag of one section of the wing.

It is interesting to note that there is very little scatter in the points which fall to make smooth curves. Also the curves from flight test results are parallel to those of the wind tunnel. The wind tunnel measurements were all at one R.N. while those of flight tests were at different R.N. for different C_L.

The thickness in per cent of the wings of the DC 4 and the L 14 where measurements were made are approximately the same (0.129 compared to 0.131). Therefore the results can be compared for the different effect of roughness, i.e., flush riveting to regular, etc.

It should be borne in mind that a correction has been made for the effect of slipstream for the DC 4. Previous to the correction being made for the slipstream the curve appeared as shown in Fig. 5.

Scale and Roughness Effect on the Two Airplane Wings, Figs. 6a, 6b.

After the curves of C_{D_o} vs. C_L have been plotted we go to those of C_{D_o} vs. R.N. to compare the results for the roughness and scale effect of the two airplane wings.

To do this we have plotted C_{D_o} for the four different C_L's employed during flight test. At the same C_L we have values of C_{D_o} from the wind tunnel. These we plot also and from them we draw curves extrapolating the profile drag to full scale R.N.

Therefore the difference in drag between the flight test points and the extrapolated curve is that which is due to roughness.

It can be seen that the increment in profile drag is the same for all lift coefficient or speeds and a comparison of the two airplanes shows that the larger increment of drag is for the airplane with ordinary riveting while the airplane with flush riveting has a smaller increment.

Profile Drag of Big Wing in the Wind Tunnel

Profile drag vs. angle of attack of one section of the big wing at constant "q" can be seen in Fig. 7. This is a smooth curve showing the increase in profile drag with increasing angle of attack.

In Fig. 2, the profile drag at different wing sections is plotted. Before the correction was applied for the distribution of static pressure and "q" across the span, the drag was unsymmetrical about the center chord line. The increased drag at the wing tips and at the wire supports should be noted.

The curves before and after the correction are both plotted on the same sheet for comparison.

DISCUSSION OF BOUNDARY LAYER RESULTS FOR BOTH FLIGHT TEST AND WIND TUNNEL

The question of determining the transition point is quite important and is one that requires a large amount of research. Many tests have to be made and data collected and reduced in order to find the transition point, what causes it, and the effect it has on drag.

It is regretted that in the flight tests only one flight was available for tests in each ship. In any work of this nature, several preliminary flights should be made to eliminate any troubles before the actual measurements are taken. After both flights, it was found that the apparatus was located in the slipstream. Therefore one must view the values of the flight test results with this in mind.

In the case of the wind tunnel much information has been secured in a short time but not all has yet been reduced. These tests were made with three different types of total head supports in order to find any effect that the heads may have on the transition point. A few of these results which have been prepared to date are presented.

At the time of writing a good transition point has been found by Dr. A. C. Charters, using the small tunnel at GALCIT with a glass plate. It is hoped that it will be possible to find any effect the total head tubes have on the transition point, and in the future to construct them so that they will have a very small effect. A method of construction that has been suggested is to allow the tube to bear on the surface and when reaching its most aft position to trip up clear of the covering wing sheet, which will allow it to return to its foremost position freely. Charters' tests are made under rather ideal conditions.

Bearing in mind all these facts, i.e., the apparatus was in the slipstream during both flight tests and the possible effect of the measuring apparatus on the transition point, a discussion of the following results is presented.

On inspection of the curves obtained by using two different types of total head tubes, i.e., (1) those used in flight test and (2) those held to the surface of the wing by cellophane tape, Fig. 17, on the big wing in the wind tunnel, it was seen that the curves for the tubes held by the cellophane were jerky and jumped around and did not give smooth curves. This was believed to be due to the fact that the tunnel had to be shut down for each setting, and after resetting, the heights of the tubes were not the same due to the human element of not being able to keep them at the correct height. Extra care was taken to have the tubes at their proper distances from the wing surface at each setting.

By contrast, using the tubes which rolled or slid on the surface a smooth curve was obtained. In addition, a great many more readings could be taken in a shorter time giving many points for the curve. Thus the method of sliding or rolling the tubes over the surface has distinct advantages over the other method and the latter has little value due to its inaccuracies.

The boundary layer curve, Fig. 8, shows the comparison using the total head tube frame of the flight test with a newer one which slides instead of rolls and is also very much cleaner aerodynamically. The surface tube of the latter bears on the surface of the wing. Both were traversed over the wing by means of the same apparatus used in flight tests.

Neither of these families of curves shows transition but they do show similarity of total head measurements. The boundary layer thickness is shown plotted by points where the tubes enter the layer. "g" is negative for the surface tube as the static pressure may be negative and the dynamic pressure so small that the total is negative.

In the flight test curves, Fig. 9, it can be noticed that the total head ratio is greater than one. This is because the apparatus is in the slipstream and as the reference free stream total head was outside the slipstream, the ratio of the total head in the slipstream to reference total head is greater than one.

The boundary layer may also be plotted, Fig. 9, as before, and from the smoothness of the curves it appears that the readings were consistent and not erratic.

On the curve for airplane No. 2, the hump would seem to indicate the transition point according to Jones, as this is the characteristic type of hump which would be expected in the transition region.

COMPARISON OF PROFILE DRAG RESULTS OF FLIGHT TEST WITH T. N. 695, MARCH, 1939.

"Lockheed 14"

From Tests: Fig. 6.

$\Delta C_{D_o} = 0.005$, R.N. $10 \times 10^6 (C_L = 0.6)$ to $14 \times 10^6 (C_L = 0.28)$

$\Delta C_{D_o} = 0.006$, R.N. $8 \times 10^6 (C_L = 0.95)$

Irregularities:

2 Plain laps on top and three on bottom.

24 Rows of 3/32" Brazier Head Rivets on top.

12 Rows of 3/32" Brazier Head Rivets on bottom.

Retractable landing light close to section at approximately 25% of chord.

Rubber Plugs in Leading Edge.

Row of 6/32" Brazier Head Screws 8" from L.E., both top and bottom.

Rough approximation of increased profile drag from T. N. 695:

First Method

Due to rivets	27%
Due to laps	3%
Due to surface roughness	6%
Due to manufacturing irregularities	8%
Total Increase	44%

C_{D_0} of W.T. wing at 0.3 C_L = 0.01025

0.44 x 0.01025 = 0.0045

Estimated increase due to landing light, rubber plugs, flap gap, Phillip Head Screws = 0.0010

Therefore total increase in profile drag = 0.0055.

Second Method

Increase in profile drag of rivetted service wing over W.T. model with same rivet and lap arrangement at C_L = 0.3 for R.N. of 14 x 10^6, Fig. 28, is 0.0016, or in percentage equals 17%.

Therefore (46% - 29%) x 0.01025 = 0.0017.

Due to rivets and laps from model wing, Fig. 23, increase of 0.0015.

Estimate due to flap gap, landing light, etc. = 0.0010.

Estimate of roughness = 0.0006.

Therefore total increase = 0.0047.

From Tests: Fig. 6

ΔC_{D_0} = 0.001 for R.N. 15 x 10^6(C_L = 0.9)

to 27 x 10^6(C_L = 0.3).

Top Surface:

8 joggled laps.

62 rows of countersunk 1/8" rivets,

at 80% of chord is flap gap of 1".

Bottom Surface:

8 joggled laps.

62 rows of countersunk 1/8" rivets.

at 72% of chord is flap gap of 0.75".

De-icer on L.E.

Rough approximation of increased profile drag from T. N. 695:

<u>First Method</u>

Increase due to rivets	5%
(From curves flush rivets is approx. 1/5 that due to Brazier head rivets)	
Increase due to laps	4%
Surface roughness	6%
Manufacturing irregularities	8%
Total increase in profile drag	23%

C_{D_0} (at $C_L = 0.3$) $= 0.008$ W.T.

$0.33 \times 0.008 = 0.0026$

Adding 0.001, estimated due to de-icer, flap gap, etc.

Therefore total increase in profile drag $=$ <u>0.0036</u>.

Although the results for the Lockheed 14 were close, those for the DC 4 were not, showing the necessity for further work along these lines.

EQUIPMENT

Introduction

The description of the following equipment necessary for securing data on (1) Wing profile drag and (2) Changes in the wing boundary layer flow during free flight is given below. A brief description of the same apparatus as applied to the smooth wing (span 103.2", chord 60") in the GALCIT wind tunnel is included, along with a discussion of the investigations to be made to best utilize the equipment.

1. The fore and aft exploring device which seeks and records any change in the boundary layer flow over the wing.
2. Pitot-traverse apparatus for taking dynamic pressure readings during the traverse of the wake of the wing to determine the profile drag.
3. The installation of the above equipment on the DC 4 and Lockheed 14.
4. Instrument panel and controls.
5. Schematic diagram of hook-up and explanation of operation.
6. Brief discussion of same apparatus as applied to the large smooth wing in the GALCIT wind tunnel.

As the primary object of the apparatus was to take readings during free flight, it was constructed to be capable of easy installation on any airplane. The manner of installation was simple and required as few alterations as necessary. As far as practicable, every effort was made so that after installation, as much of the equipment as possible was in the wing or fuselage. It should be noted here, that the apparatus was designed especially for the DC 4, which made a very neat installation as compared to the Lockheed 14, where the same apparatus was installed wherever possible. The entire apparatus being electrically operated was remotely controlled at the instrument panel where every necessary reading was available. Data were secured by photos to obtain simultaneous readings, to eliminate the personal equation, to obtain a permanent record, and to speed up the recording of the data.

All photos, schematic diagrams, drawings, etc., may be found in GALCIT Rep. 220.

BOUNDARY LAYER SURVEY EQUIPMENT

The exploring device for seeking changes in the boundary layer is simply a small light frame holding the exploring tubes. This frame is held against the surface of the wing, over which it moves, by an arm attached at right angle to an electric car mounted on a track, and capable of moving fore and aft along the chord of the wing. A cable, housing the necessary rubber tubes and electric wires running from this car, is attached to the instrument panel located at any convenient place in the cabin, from which all operations are remotely controlled and readings taken.

The light frame constructed of soft iron (Fig. 15) holds four (4) exploring total head tubes and one (1) static pressure tube and is able to roll on the surface of the wing by three small discs, 3/16" in diameter and cut from fiber 3/16" thick, which serve as wheels. Two of these wheels are mounted in front while the third is at the center and rear. The tubes are held at different heights above the wing and are made of 3/32" hypodermic needles. The exploring total head tube openings are flattened to a thin rectangular shape of the same size, being made by shaping the opening about a shim. These tubes are held parallel to the surface of the wing and their openings are in a plane perpendicular to the surface and passing through the centers of the two forward wheels or rollers. The static tube is made of small brass tubing and is located higher than the highest of the exploring total head tubes.

Several of these frames with rollers were made with tubes at varying heights above the surface. This was done to facilitate quick change if during a test it was found necessary to have the tubes of different heights. Three such frames were made and it was found that No. 1 was satisfactory for all tests. It is as follows with the height of each tube opening above the surface given. Note: the height of surface tube, No. 1, varies due to irregularities of the wing.

Frame No. 1

	In Inches		In % of Chord	
	L-14	DC-4	L-14	DC-4
Pressure Tube 1	1/32	Surface	0.000305	0.000164
Pressure Tube 2	1/8	6/64"	0.00122	0.000657
Pressure Tube 3	32/64	22/64	0.00350	0.00181
Pressure Tube 4	37/64	35/64	0.00564	0.00288
Static Tube 5	2 1/16	2 1/16	0.0201	0.0201

Chord of DC-4 at measuring station 190"

Chord of L-14 at measuring station 102.5"

The tubes were calibrated at GALCIT with a standard Prandtl pitot tube as reference.

The total head frame is displaced laterally by a hollow steel arm 15" long to avoid disturbances caused by the driving car. The tube frame is held in firm contact with the surface of the wing by a coiled spring attached between the arm and the car base, and another between the arm and the rear of the frame.

The car is made of dural and runs on a track (Fig. 5) constructed of two lengths of an extruded dural bulb angle, ALCOA 8493, riveted to a narrow aluminum sheet, 0.051" by 7 1/4." The track and sheet were made to fit the wing profile of the DC 4 at the wing joint between the inner and outer wing panel, the sheet taking the place of the cover plate.

The car has two sets of four rollers each. One set is fixed and moves inside the track, rolling on the underneath side of the bulb of the stiffener. These prevent the car from ever leaving the track, and also prevent movement in a lateral direction. The other set straddles the track and is held firmly against the sheet, over which it rolls, by a separate spring to each roller. These springs allow the car to roll freely but firmly over the curved surface of the wing at any point without binding.

An electric motor is mounted on the car with its shaft parallel to the track. At one end an autosyn and a sprocket are geared to the shaft of the motor through a worm and a worm gear having a gear reduction of 216:1. The autosyn indicates remotely the position of the car, with respect to the chord, by a dial on the instrument panel. A bicycle or roller chain passes over the sprocket and is lead by means of two idling sprockets to the wing surface where it lays on the sheet between the rails of the track. By this means the car moves fore or aft through power from the motor, and is saftied at both ends of its travel by automatic limit switches. The chain is secured at both ends of the track and correct tension can be obtained by an adjusting bolt with lock nut at one end.

Three thirty-second inch rubber tubes were led from the exploring frame through the hollow arm to the car where they are raised to 3/16" tubes and there together with the two cables of five wires each from the motor and autosyn are formed into one cable 1 1/8" in diameter, and covered with a woven fabric sleeve. This cable is very flexible and when treated with ground soapstone, slid easily back and forth through a dural tube, which ran from a point at the rear of the track through or outside the wing back to the cabin or instrument panel. This allowed the cable to be let out or pulled in by a member of the crew as the car moved fore and aft on its track. In this way the exposed cable along the track, while the car was at its farthest point forward, was kept taut enough to prevent any fluttering or whipping on the surface of the wing. None of this was experienced at any time during tests made in free flight.

An inverter was used to supply the autosyn units. The type and make of the inverter, autosyn, and motor are as follows:

INVERTER

DC-AC

Type 12

Input 12 volts DC

Output 110 volts max. (adjustable) AC, 50-60 cycle

Capacity 100 watts, intermit 100 cont.

Manufactured by the American Television and Radio Co., St. Paul, Minn.

It is recommended that a rotary converter be used instead of the vibrator type due to the better reliability of the converter.

AUTOSYN MOTOR

32 volts AC, 60 cycle

Single phase

Manufactured by the Pioneer Instrument Co., Brooklyn, N. Y.

ELECTRIC MOTOR

12 volts DC

3.5 Amp.

1/50 Horsepower

1725 R. P. M.

NSH12R type

Manufactured by the Bodine Electric Co., Chicago, Ill.

WAKE TRAVERSE EQUIPMENT

The pitot-traverse apparatus (Figs. 12, 13, 18, 20) consists of an electric motor, which by means of reduction gears, rotates the pitot tubes at a known fixed radius (18.25"). An autosyn similarly connected gives the angle at all times which the pitot arm makes with a reference line, usually the trailing edge of the wing. Knowing the radius and the angle, the exact location of the pitot tubes are known. This mechanism is located out on or in the wing and is connected by a cable, composed of the necessary tubes and wires, to the instrument panel located in a convenient place, where the operation may be remotely controlled and data secured.

The pitot traverse operating mechanism was housed in a dural box approximately 12 1/2 inches long, 6 3/4 inches wide, with a trapezoidal cross section of whose respective heights being 2 inches and 4 inches. The box was made in this shape to fit into the trailing edge of the DC 4. It contained tapped holes for mounting screws. From this box extended the shaft which ended in a "T" pipe fitting, to which the pitot arms supporting the pitot tubes were fastened.

The electric motor and autosyn were the same type that were used in the boundary layer mechanism, the only difference being that the gear reduction ratio was 3360:1; worm gear reduction of 1120:1; and through spur gear of 3:1. The shaft supporting the pitot arms held two metal stops each of which tripped an electric switch when the shaft had rotated through 120° in either direction. These served as automatic stops to prevent any serious twisting of the rubber tubes in case the operator permitted the arm to rotate through an excessive angle.

Three-sixteenths inch rubber tubes from the pitot tubes were fed through the supporting section of pipe through the box to where they joined with the two electric cables, necessary for operation of the electric motor and autosyn. The four flexible rubber tubes from the pitot tubes with the two cables of five wires each, were formed into one cable 1 1/8" in diameter, and covered with a woven fabric sleeve. This gave a very flexible cable which enabled one to run it easily to the instrument panel whenever mounted.

The motor was reversible and care was taken not to revolve the pitot arms through any larger angle than necessary, operation being only through the wake.

Calibration was accomplished in the 10 ft. wind tunnel at GALCIT.

INSTALLATION OF WAKE TRAVERSE EQUIPMENT ON THE DC-4

Since the pitot traverse mechanism was designed specifically for the DC 4, it was easy to install and make an excellent installation both structurally and aerodynamically (see Fig. 12).

A section of the wing covering slightly larger than the box housing the mechanism, was removed from the trailing edge of the left wing adjacent to the aileron. This section was replaced with an exact duplicate except for a hole in the extreme trailing edge to allow the shaft supporting the pitot arms to pass through. Before placing this section on the wing, the shaft was disassembled, passed through the mentioned hole and reassembled. Then the section, with the housing box inside and pitot tubes outside, was replaced on the wing, the metal screws holding the section also secured the box to the wing. The cable from the box then was strung through the rear part of the wing up through the cabin floor, where it was connected to the instrument panel. Distance pitot tube head in rear of trailing edge of wing is 5 3/32" or 0.00268 of chord (190"); length of arms is 6.0" and 18.25". Distance of pitot tube from center line of ship is 396".

INSTALLATION OF THE WAKE TRAVERSE EQUIPMENT ON THE LOCKHEED 14

The installation on the Lockheed 14 was quite different as all the apparatus, including the cable, had to be installed on the outside of the wing (see Fig. 13). To install the box it was necessary to build a supporting structure to the rear of the wing to hold the box and its rotating arm. This was done by removing the fairing from the outer two Fowler flap supports of the left wing and fastening dural "T" members 23 3/4" long to them. Bridging between the rear of these two members was an 8" by 1 1/2" plank, which was held by bolts. On this plank the box was placed and secured in a position to allow travel of pitot tube at a proper distance from trailing edge of wing (3 1/8").

This supporting truss was strengthened by braces running from the rear of each "T" member to their respective Fowler flap support together with a 1/2" dural tube fastened lengthwise to the center of the bottom of the board to prevent spanwise vibration.

The cable, housing the tubes and wires, was run on the outside from the supporting truss along the trailing edge of the wing to the fuselage, hence up through an open window to the instrument panel. The cable was taped at 3" intervals to a 3/4" dural tube which was placed along the trailing edge expressly for that purpose. All taping, etc., was shellacked after installation. (Note only one pitot tube and arm (18.25") was used, $\frac{R_1}{t}$ = 0.178.) Average rivet head height above surface of wing = 1/32". Head of tube 193.62" from centerline of ship.

INSTALLATION OF BOUNDARY LAYER SURVEY EQUIPMENT ON DC-4

As this apparatus was also designed specifically for the DC 4, it made an excellent installation (Fig. 11). The cover plate between the inner and outer right wing panel was removed and replaced by the track, of which the dural strip took the place of the cover plate. At the rear of the track, which conformed to the wing at all points, approximately 8" of an 1 1/8" dural tube lay on the surface of the wing alongside the track, with the flanged opening forward. This tube ran from this point through the wing to where it came through the floor of the cabin next to the instrument panel. This end also being flanged to allow the woven cable, which was treated with powdered soapstone, to slide easily back and forth through the tube. After the car was placed on the track, the automatic stops were placed where desired and the chain given the right tension. To string the flexible cable through the tube a thin cord was blown through first, with which it was able to pull the woven cable through afterwards.

INSTALLATION OF BOUNDARY LAYER SURVEY EQUIPMENT ON THE LOCKHEED 14

The same track (Fig. 14) was placed on the left wing of the Lockheed 14 and since the size and curvature of the two wings are different, the track protruded a small distance above and in front of the leading edge. The rear section of the track was shimmed and the entire track fastened down with the same screws holding the cover plate. However, due to the construction of the apparatus, the exploring frame with tubes remained always on the surface of the wing. The flexible cable was carried through a dural tube which extended from the rear of the track, along the trailing edge of the wing up to the open window. The automatic stops were also reset at desired limits.

On all future installations care should be exercised to locate all apparatus out of the slip stream, if possible.

Chord "t" at boundary layer tubes = 121.5".

Distance of frame from centerline of ship = 142.5".

INSTRUMENT PANEL

The instruments were mounted on a wooden vertical panel approximately 28" high and 19" wide (Fig. 10). Instruments, manometers, adjustable flasks, etc., were mounted as shown and a suitable graph paper installed behind the manometers which were of 3/16" glass tube, 20" in height. The flasks were adjustable in a vertical direction in order to arrange the level of the fluids as required during the tests. The left set of tubes contained a fluid of specific gravity that was heavier than those on the right. Both were composed of Telegage fluid and alcohol.

For DC-4, specific gravity of heavy fluid = 2.95

For DC-4, specific gravity of light fluid = 1.61

For L-14, specific gravity of heavy fluid = 2.84

For L-14, specific gravity of light fluid = 1.25.

The fluid was placed in the apparatus just before the tests were made and taken out immediately after. For the laboratory, ethylene chlorhydrene was used for the fluid with a specific gravity of approximately 1.0.

The panel was mounted on a solid wooden frame which consisted of two by fours bolted together to form a "V" shape frame 52" long and 22" wide across the rear. Braces held the panel in position.

In front was the camera support which consisted of a clamp on a 2" dural tube 18" in height. This tube was held to the frame by flanging its bottom and secured by bolts. The light necessary for illuminating the panel was furnished by a 100 watt photo flood lamp of type No. 803, S and M Lamp Co., Los Angeles, Calif., with a semi-ground lens. The distance of lens to panel was 28". The lamp was held by a single bolt at the rear of the frame and was so arranged that its light could easily be moved as desired. Cardboard sections to fit over the airplane windows were also carried in order to shut out any light that might affect the camera.

Two Leica cameras were used, one to take photos of readings while the second was being loaded and held in readiness. It was found that holding the camera by hand in the clamp was sufficient. Super X film was used in the Leica camera with F 2 lens, summar , 50 mm. focal length and was set for 1/20 sec. and f = approx. 2.5.

In the DC 4 the panel and frame were held approximately 24" off the floor by a frame, while in the Lockheed 14 it was bolted to the floor. In both cases strips of felt were used to dampen out any vibration.

Schematic diagrams are given in Rep. 220, GALCIT.

OPERATION

After installation of all equipment, a complete check of all points was made. Tubes were checked for leaks, terminals were gone over and operation of all points made certain. The density of the fluid was taken and the fluid placed in the manometers, just prior to making tests.

Before, during, and after the take off it is important that members of the test party regulate the fluid levels in the manometers to prevent the pressure from forcing the liquid back up in the tubes, hence into altimeter and air-speed instrument cases. Also all data should be secured such as weight, fuel consumption per hour, etc., and all readings and actions synchronized and co-ordinated with pilot or engineer of the ship.

During the test, photos are taken with various settings of the equipment. The boundary layer apparatus, moving at very slow speed, was allowed to run continuously from fore to aft positions, and vice versa, while the wake survey was changed approximately one to two degrees at a time. No attempt was made to change the setting a specified number of degrees because of the time required. By this method photos were made as rapidly as the camera and counter could be operated.

Each photo is tabulated by a successive counter reading, the counter being manually operated at the instrument panel. At the same time another member of the testing group, who is in charge of the test, is keeping the chronological log and operating the apparatus by means of the two knife switches attached to the instrument frame. A third member keeps the flexible cable to the exploring car taut by pulling the cable in or playing it out as is needed. Care must be taken not to squeeze the cable as the level of the manometers are affected momentarily. By co-ordination with the camera operator no photos are taken during times when some slight squeezing of the cable is unavoidable. The fourth member of the test party has charge of the loading of the cameras and any other emergencies that might arise.

In view of the time element and the inaccuracies involved in taking visual readings during flight, it was deemed advisable to take all readings by photos. In this way a permanent record was kept of simultaneous readings, eliminating as much as possible the personal error.

INSTALLATION OF EQUIPMENT ON WING IN GALCIT WIND TUNNEL

The same boundary layer apparatus that was used on the DC 4 and Lockheed 14 was also used on the highly polished wing (103.2" span, 60" chord) in the GALCIT wind tunnel. The only difference was in substituting a shorter length of track to accomodate the smaller wing. This section of track was made straight, hence did not conform to the profile of the wing. This was overcome (Fig. 19) by placing wooden curved sections between the track and the 8" band of steel to which it was fastened. This band, separated from the wing by a thin layer of soft material to prevent scratching or marring of the highly polished surface, was held taut against the wing by adjusting tightening screws and forcing a small bar against the trailing edge of the wing. This forced the rear edge of the band from the trailing edge of the wing, giving the desired tightness. The coiled springs of the apparatus kept the exploring frame always firm against the surface of the wing.

For calibration, a set of five exploring tubes was used. These tubes were 1/8" in diameter except at the ends which were made of 1/16" hypodermic needles whose end openings were changed to thin rectangular sections. These tubes extended 12" from a 1 1/8" dural tube which held them as one unit. This tube was made so as to be a smooth fit inside another dural tube which was secured to the surface of the wing through four plates held there by cellophane tape. Graduations were made on the outside of the inner tube so it could be extended any desired distance and locked by a set screw in the outer tube. In this way the head could be moved through a limited distance fore and aft along the chord, taking as many readings as desired with minimum changes of the base, which necessitated removing the cellophane tape, relocating, and retaping. The outer dural tube was secured to the four plates by adjustable height screws and lock nuts by which the exploring head was held to the surface of the wing at all times by raising or lowering either the fore or aft pair of screws to conform to the curvature of the wing.

The wake traverse apparatus was mounted on an 8" by 1 1/2" board held at each side of the tunnel and located approximately 16" in the rear and 2" below the trailing edge of the wing (Figs. 16, 18). This board, in addition to being supported at the tunnel walls, was held at the center by 2 wires with turn buckles, one going to the top of the tunnel, the other to the floor. The housing box was clamped to the board which facilitated the movement of the apparatus on the board to get readings at various spanwise points in rear of the wing.

The only other difference in the apparatus itself was that the pitot tubes and pitot tube arms that were used in free flight tests were removed and a smaller set installed. The type used in the wind tunnel (see Fig. 18) consisted of 1/8" pitot tubes with arms 11" long. These arms, due to their size, were braced. The flexible cable housing the necessary tubes and wires was taped at intervals to the board, at the end of which it left the tunnel and was attached to the instrument panel.

The instrument panel was essentially the same except for the airspeed and altimeter which were not needed. Fluid was used in the manometer of a specific gravity of approximately 1.0.

PRELIMINARY OUTLINE OF WAKE TRAVERSE AND BOUNDARY LAYER SURVEY MEASUREMENTS FOR FREE FLIGHT

The following outline represents the investigations which should be made in order to best utilize the GALCIT equipment which has been prepared for these investigations.

1. Wake Traverse

One observation should consist of a series of measurements made at constant air speed and altitude which should occupy from three to five minutes. Observations should be taken at as large a series of air speeds as possible up to the maximum speed of flight. It may prove desirable later to put a smooth covering over the wing upstream from the pitot-traverse mechanism and repeat these measurements. However, this could not be done during the initial investigation.

2. Boundary Layer Survey

One observation consisting of measurements at constant air speed and altitude should occupy approximately five minutes.

a. Upper surface with smooth auxiliary wing cover. A series of observations at as many speeds as possible up to the maximum speed of flight.

b. Repeat with auxiliary wing cover removed, using normal wing surface.

c. Repeat b. on lower surface of wing.

d. Repeat a. on lower surface of wing.

If possible, install equipment where readings will be made in regions undisturbed by the slip stream.

As has been stated previously, this report only covers the beginning of the investigations that should be made.

REFERENCES

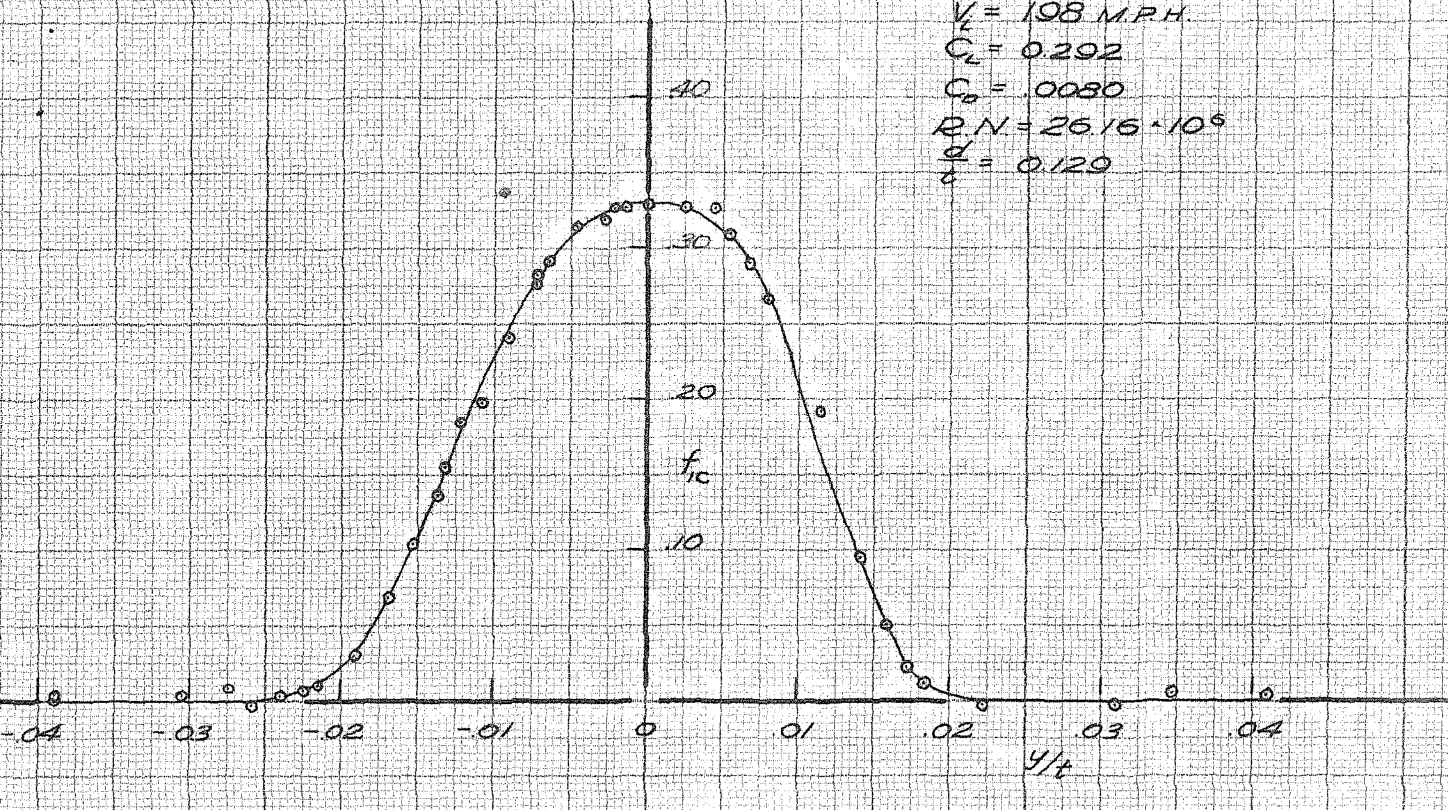
1. Jones, B. Melvill: Measurement of Profile Drag by the Pitot-Traverse Method. R. and M. No. 1688, January 1936.

2. Serby, J. E.: Loss of Momentum and Drag. Aircraft Engineering, January 1937.

3. Jones, B. Melvill: Flight Experiments on the Boundary Layer. Journal of the Aeronautical Sciences, Vol. 5, No. 3, January 1938.

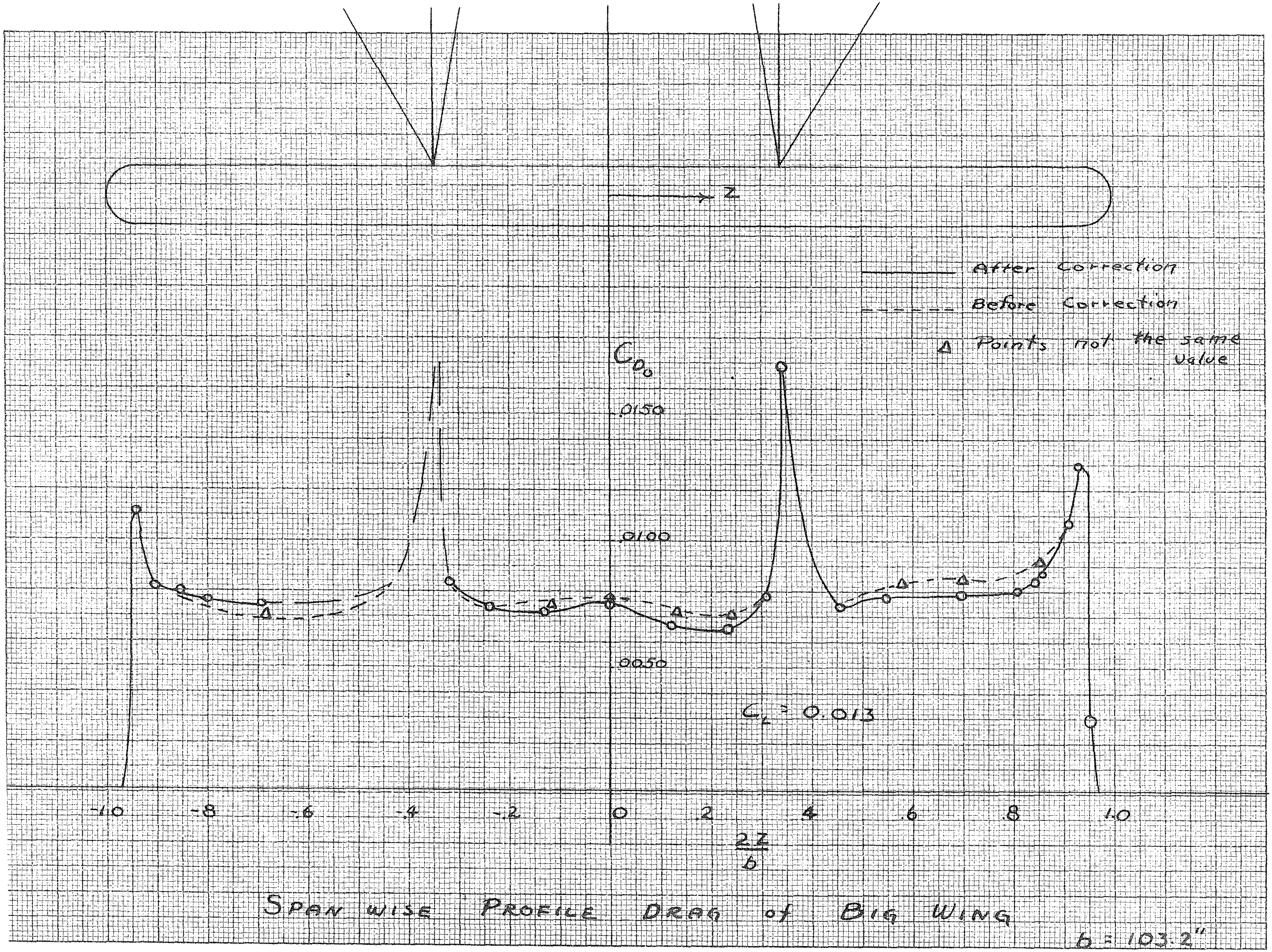
4. Putt, Donald L.: Experimental Investigation of the Thickness of the Boundary Layer and the Location of the Transitional Region along a Wing Section. Thesis, California Institute of Technology, 1938.

5. Hood, Manley J.: The Effects of some Common Surface Irregularities on Wing Drag. N.A.C.A. T.N. No. 695, March 1939.



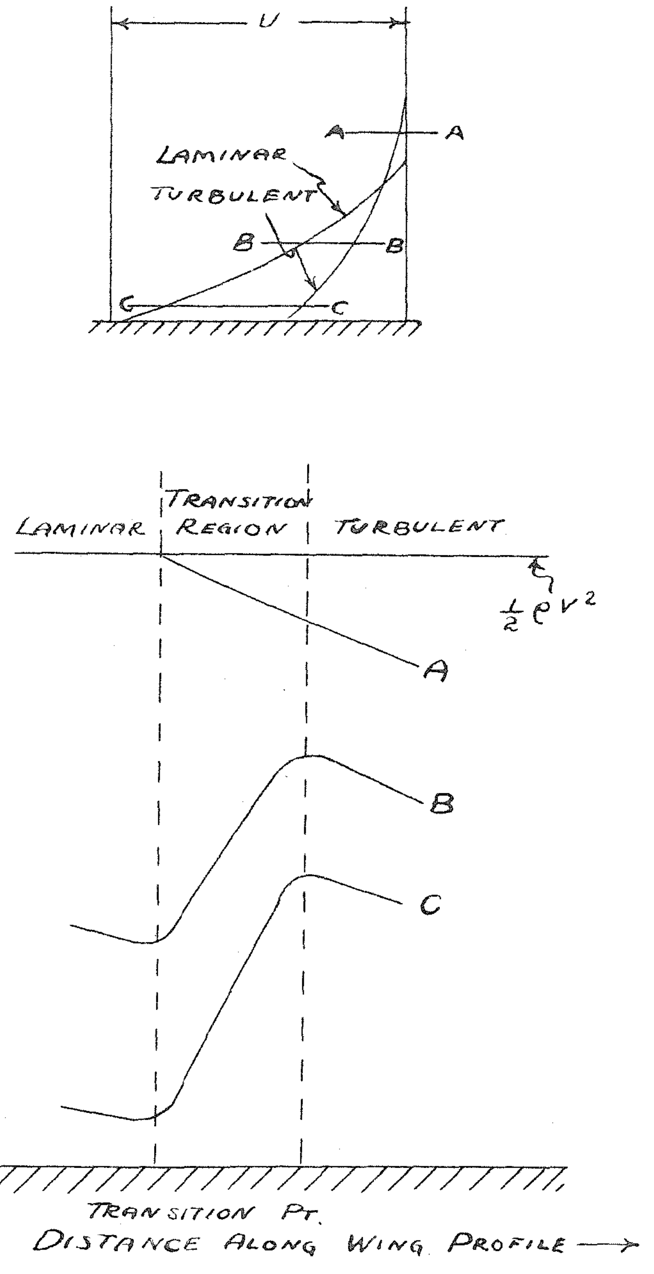
B. M. JONES' FUNCTION FOR DETERMINATION OF PROFILE DRAG

FIG. 1



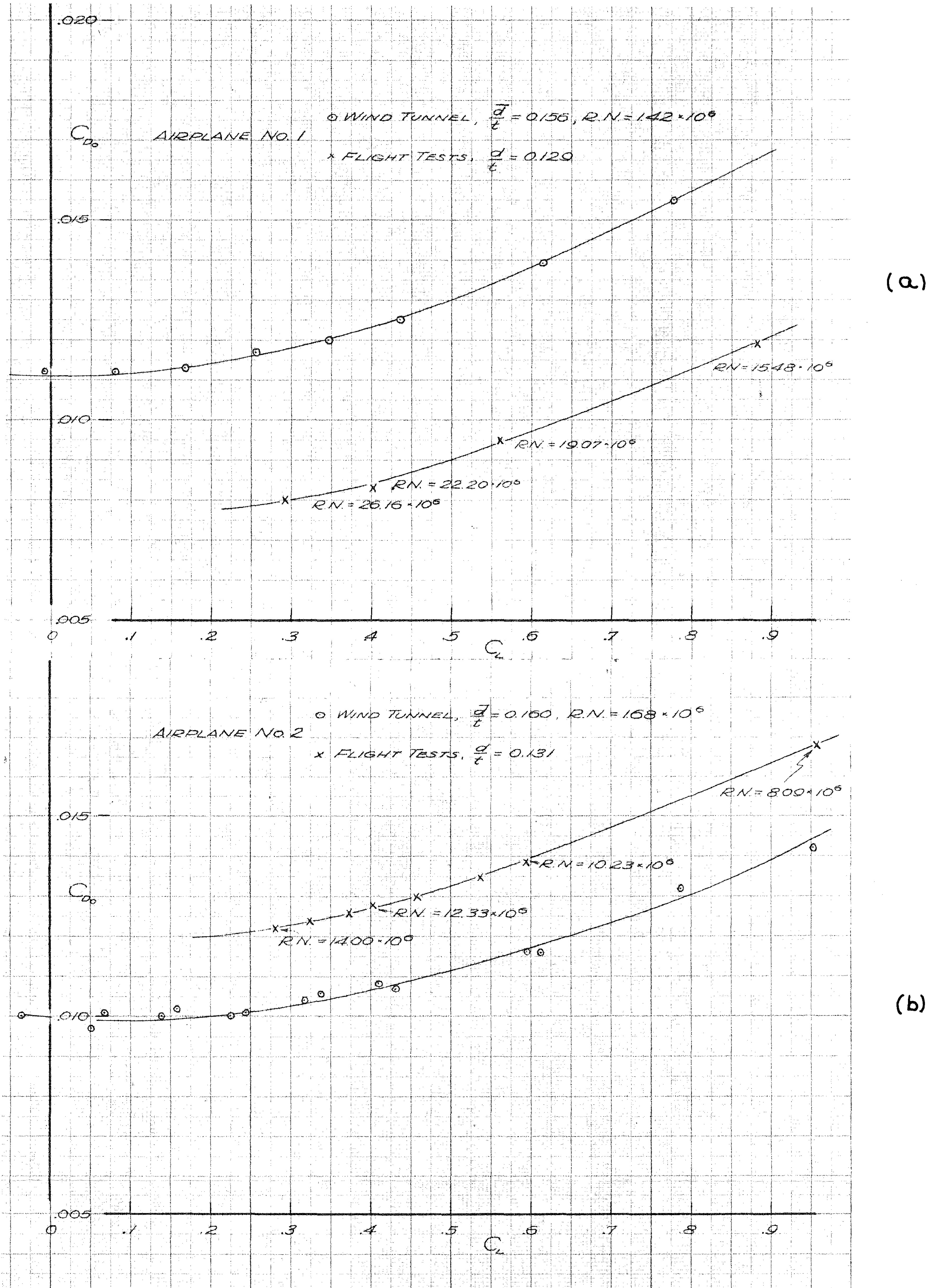
Z
After Correction
Before Correction
Δ Points not the same value
C_{D_0}
.0150
.0100
.0050
C_L = 0.013
-1.0
-.8
-.6
-.4
-.2
0
.2
.4
.6
.8
1.0
$\frac{2Z}{b}$
SPAN WISE PROFILE DRAG of BIG WING
b = 103.2"

FIG. 2



Diagrams explaining method used for locating the region of transition.

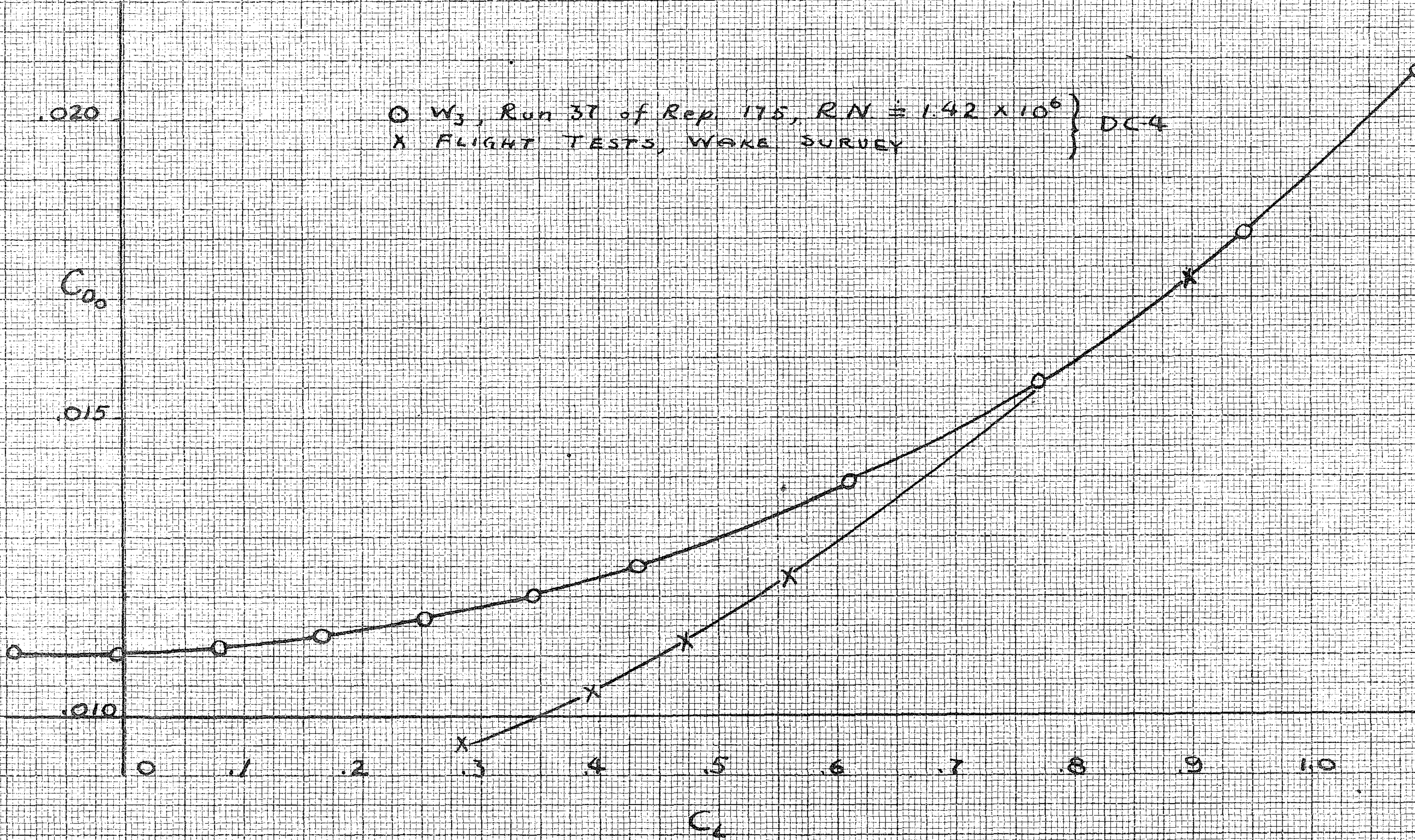
FIG. 3



WIND TUNNEL AND FLIGHT TEST WING PROFILE DRAG

MEASUREMENTS FOR TWO AIRPLANES

FIG. 4

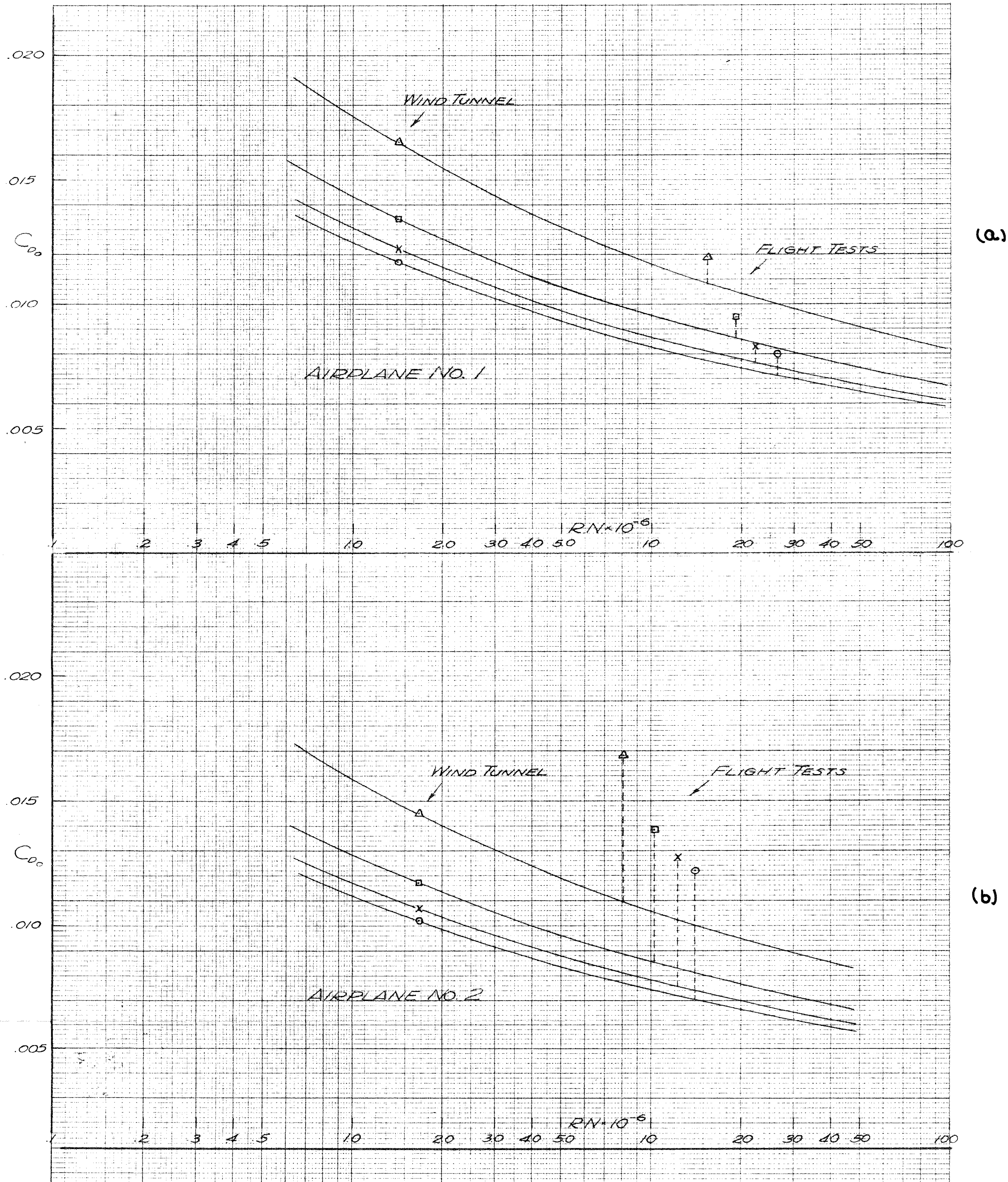


EFFECT OF SLIP STREAM BEFORE CORRECTON WAS MADE.

COMPARISON OF WIND TUNNEL RESULTS & FLIGHT TEST

DC-4

FIG. 5



SCALE AND ROUGHNESS EFFECT ON TWO AIRPLANE WINGS.

FIG. 6

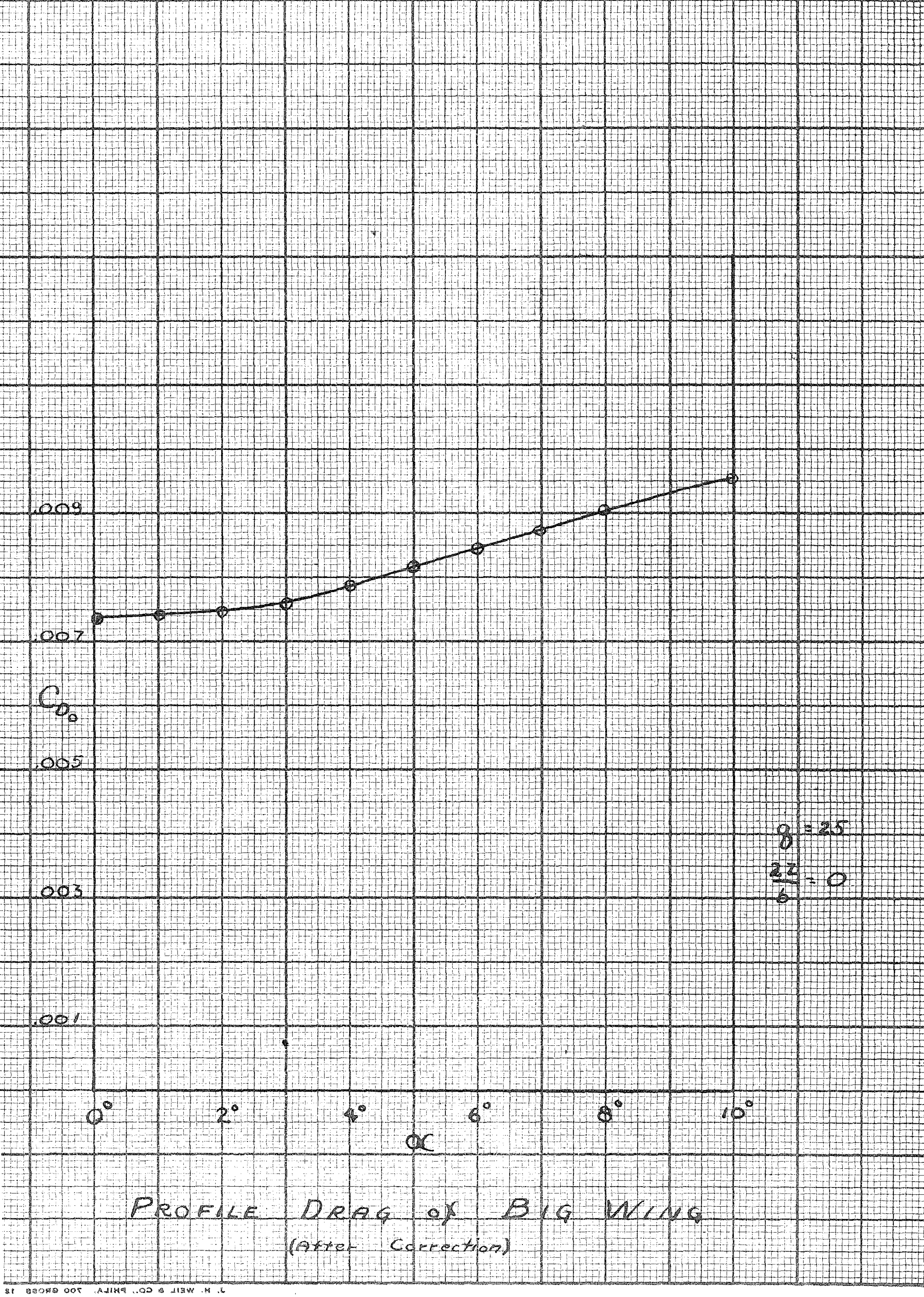


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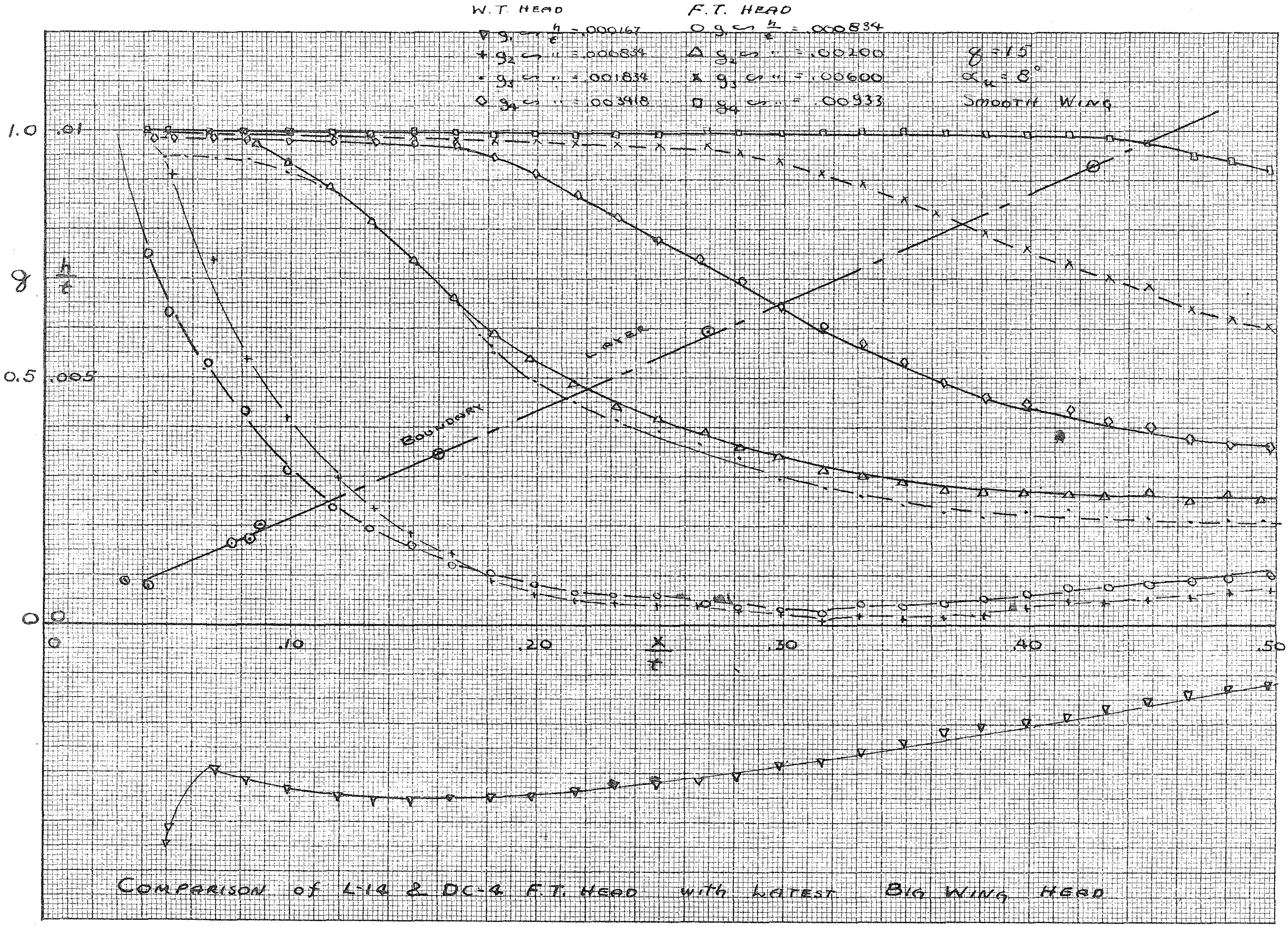
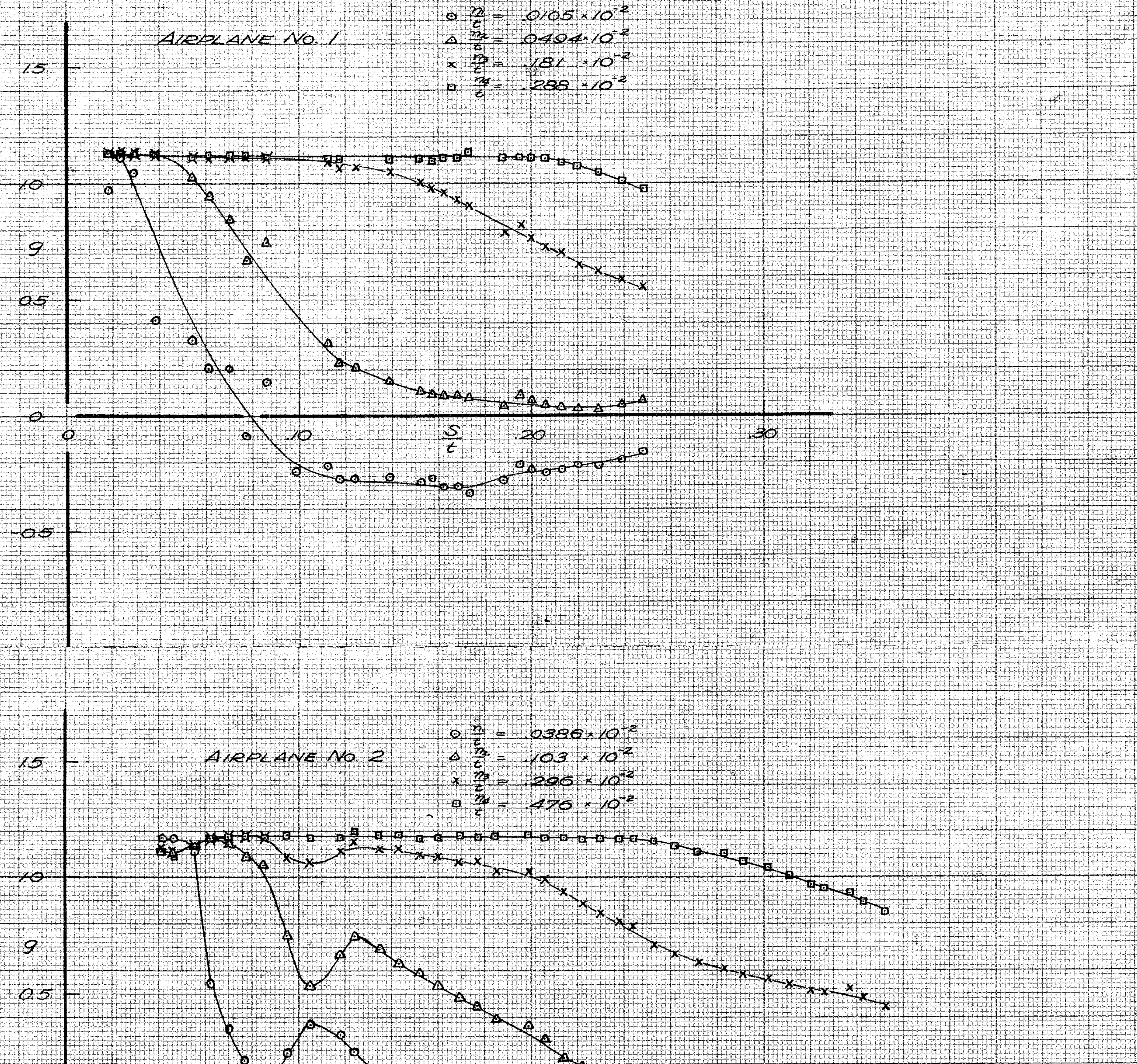
FIG. 7



W.T. HEAD
F.T. HEAD
$g_1 \backsim \frac{h}{t} = .000167$
$g_2 \backsim$ " $= .000834$
$g_3 \backsim$ " $= .001834$
$g_4 \backsim$ " $= .003918$
$g_1 \backsim \frac{h}{t} = .000834$
$g_2 \backsim$ " $= .00200$
$g_3 \backsim$ " $= .00600$
$g_4 \backsim$ " $= .00933$
$\delta = 15°$
$\alpha_u = 8°$
SMOOTH WING
1.0
.01
0.5
.005
0.0
0
g
$\frac{h}{t}$
BOUNDARY
LAYER
.10
.20
.30
.40
.50
$\frac{X}{t}$
COMPARISON of L-14 & DC-4 F.T. HEAD with LATEST BIG WING HEAD

FIG. 8



TOTAL HEAD MEASUREMENTS IN THE BOUNDARY LAYER ON TWO AIRPLANES

FIG. 9

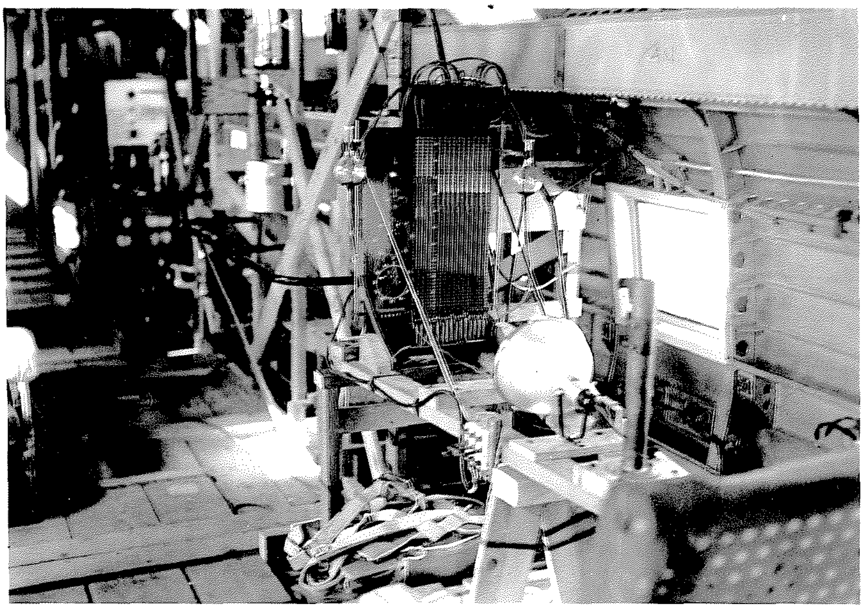


FIG. 10

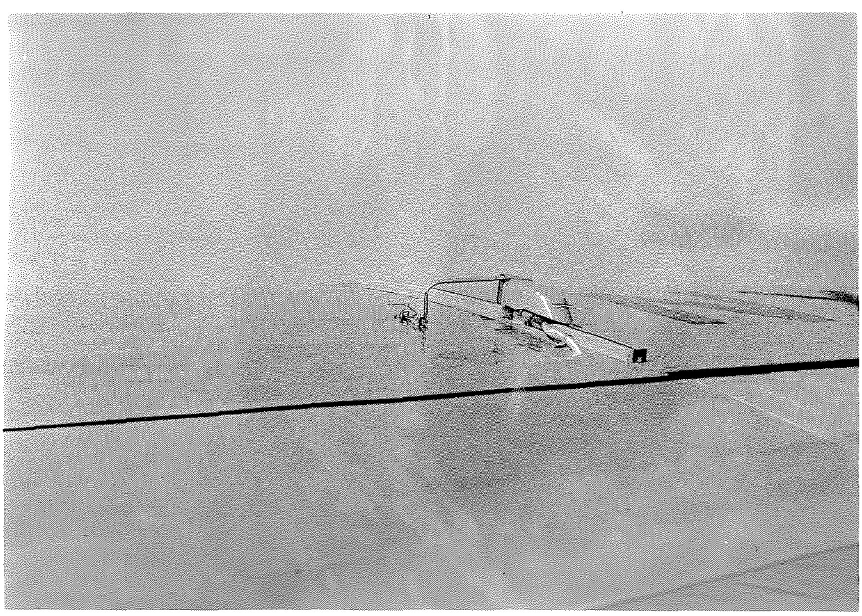


FIG. 11

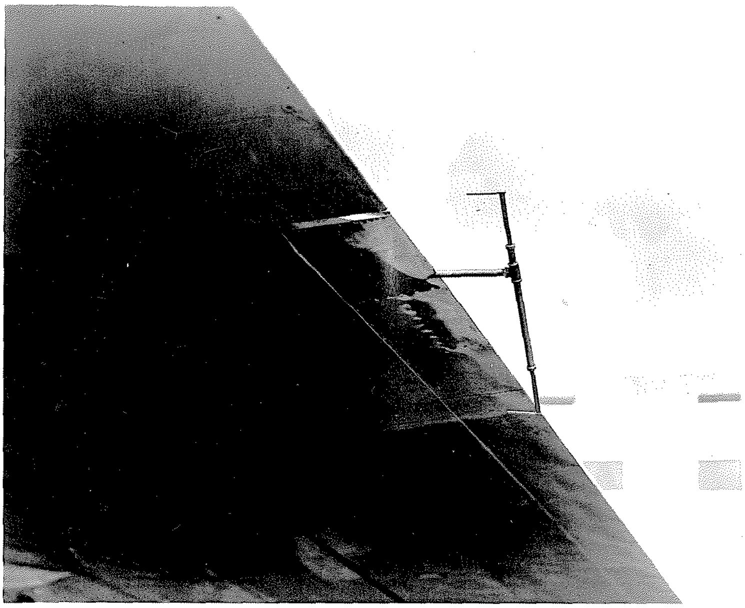


FIG. 12

FIG. 13

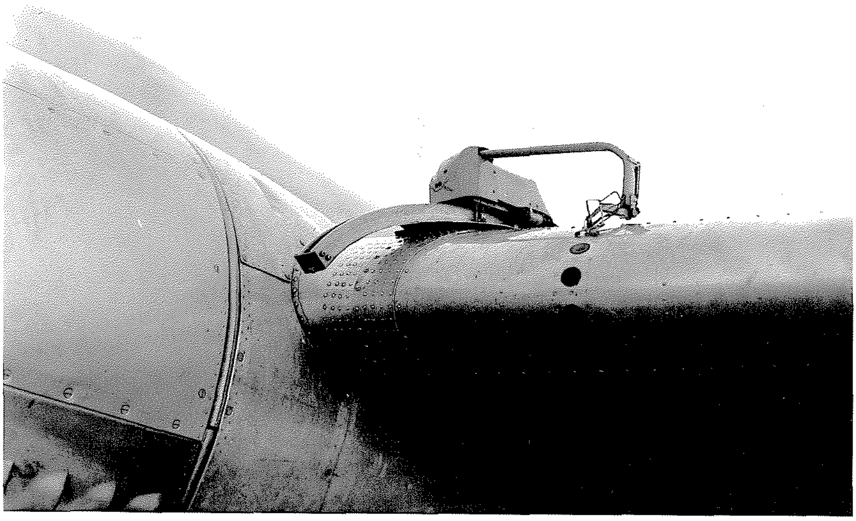


FIG. 14

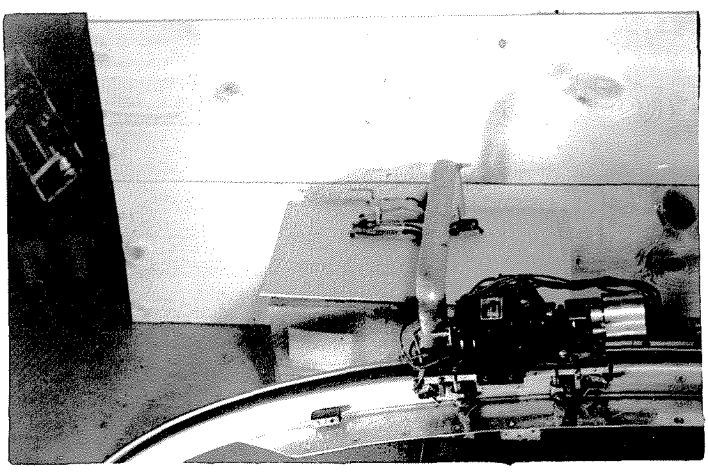


FIG. 15

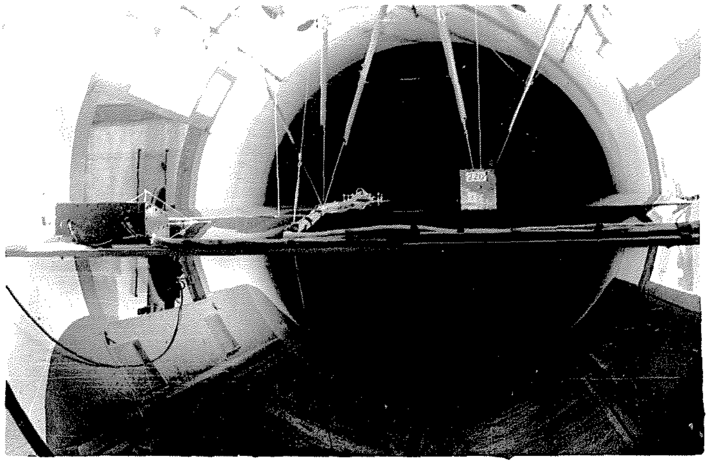


FIG. 16

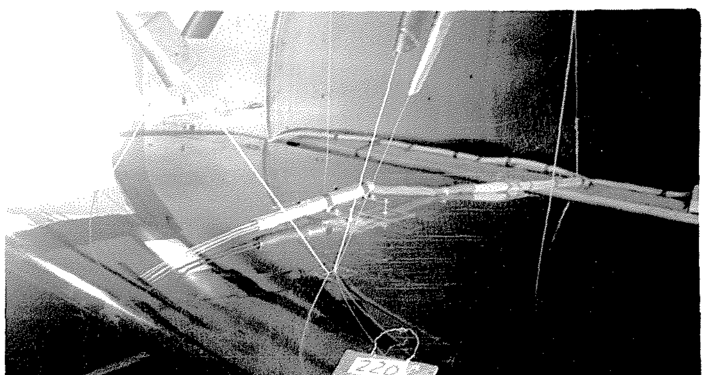


FIG. 17

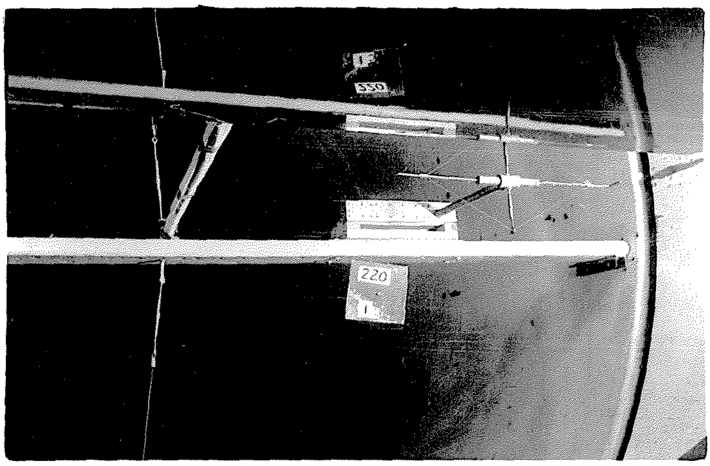


FIG. 18

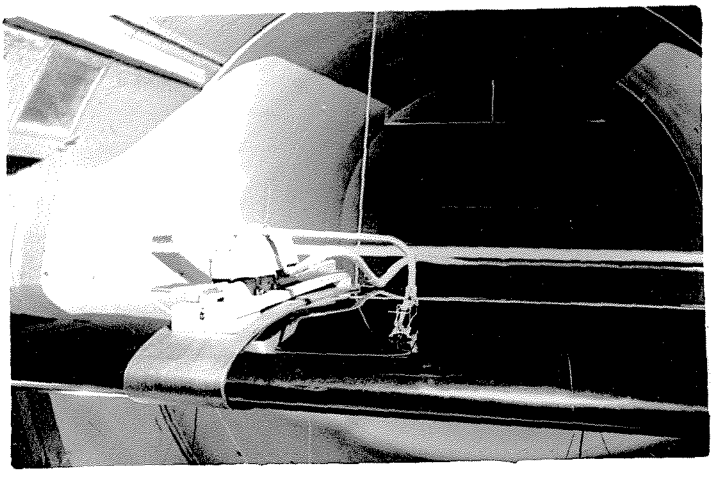


FIG. 19

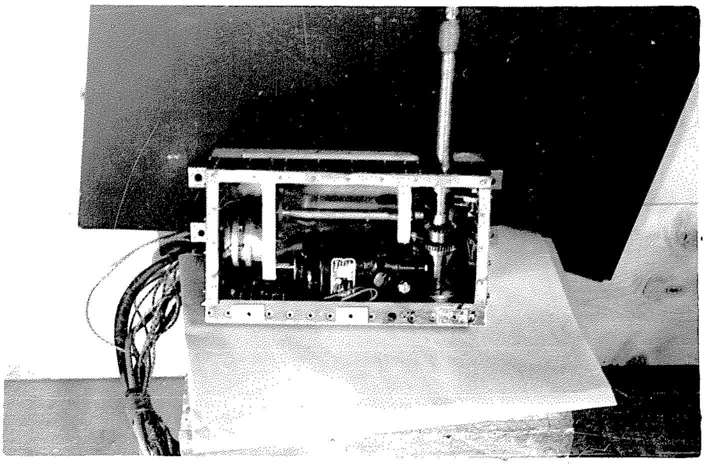


FIG. 20