THE COLLEGE OF AERONAUTICS

CRANFIELD

CHARACTERISTICS OF THE FLOW FIELD OVER THE MID-UPPER FUSELAGE OF LANCASTER P.A. 474

by

F. M. BURROWS

(Work done under Ministry of Supply Contract MoS 6/Aircraft/9807/CB/6(a).)

NOTE No. 36.
JANUARY, 1956

THE COLLEGE OF AERONAUTICS

CRANFIELD

The characteristics of the flow field over the mid-upper fuselage of Lancaster P. A. 474.

by

F. M. Burrows, D. C. Ae.

Work done under Ministry of Supply
Contract MoS 6/Aircraft/9807/CB/6(a)

SUMMARY

This note describes a series of tests conducted to determine the characteristics of the flow field over the mid-upper fuselage of Lancaster P. A. 474.

The range of the tests was to include a determination of the distributions of total head, static pressure and velocity together with the flow directional characteristics in the pitching plane for a number of aircraft flight configurations as listed in paragraph 1. 2.

Curves are presented in Figs. 9, 20 - 25, showing the flow directional characteristics and the distributions of static pressure and velocity in the region of investigation.

LIST OF CONTENTS

LIST OF SYMBOLS

The symbol $\triangleq$ denotes 'is defined to be'

H	$\triangleq$	local total head
H_o	$\triangleq$	free stream total head
U_o	$\triangleq$	free stream velocity
U	$\triangleq$	local velocity
P_o	$\triangleq$	free stream static pressure
P	$\triangleq$	local static pressure
C_p	$\triangleq$	$\dfrac{P-P_o}{\frac{1}{2}\rho U_o^2}$ non dimensional static pressure coefficient
ψ	$\triangleq$	angle of yaw (degrees)
θ	$\triangleq$	angle of flow in the pitching plane relative to the fuselage axis
Re	$\triangleq$	Reynolds number referred to wing chord
S.P.E.C.	$\triangleq$	static pressure error correction

1. INTRODUCTION

1.1. PURPOSE OF TESTS

The series of tests outlined was conducted to determine the characteristics of the airflow over the mid-upper fuselage of Lancaster P. A. 474. It is intended to mount a large scale model of a swept back half wing on the fuselage of the Lancaster in this region to conduct a series of investigations in flight of the behaviour of the three dimensional boundary layer on this model wing, and the tests described are therefore a calibration of the flow field into which the swept wing model is to be immersed.

1.2. SCOPE OF TESTS

The scope of the tests was to include measurement of the distribution of total head, static pressure, and velocity over a range of airspeeds extending from 100 knots (minimum comfortable flying speed, straight and level, with 20^{o} flap) to 200 knots (maximum straight and level speed).

The direction of the flow over the fuselage, in the pitching plane, was determined using 'Conrad' type yawmeters, and the static pressure error correction to the airspeed system measured by the aneroid technique.

For these tests several different flying configurations were considered, and these are as listed below * :-

1)	100 kts.	I. A. S.	20^{o} flap	straight and level flight
2)	115 "	"	20^{o} & 0^{o} flap	straight and level flight
3)	115 "	"	20^{o} & 0^{o} flap	sideslip : 4^{o} Port 4^{o} Stbd.
4)	130 "	"	0^{o} flap	straight and level flight
5)	130 "	"	0^{o} flap	sideslip : 4^{o} Port 4^{o} Stbd.
6)	130 "	"	0^{o} flap	straight and level, inbd. engines throttled
7)	150 "	"	0^{o} flap	straight and level flight
8)	170 "	"	0^{o} flap	straight and level flight
9)	170 "	"	0^{o} flap	sideslip 4^{o} Port 4^{o} Stbd.
10)	190 "	"	0^{o} flap	straight and level
11)	200 "	"	0^{o} flap	straight and level

* Not all these configurations were used during the aneroid runs for measurement of S. P. E. C.

2. EXPERIMENTAL EQUIPMENT

2. 1. THE AIRCRAFT AND EQUIPMENT

It is not proposed to present a detailed description of the aircraft and its equipment at this stage since this is to be fully discussed in a subsequent report. Some idea of the general layout of the equipment etc. in the aircraft may however be obtained from a study of Figs. 1 to 5.

2. 2. THE PRESSURE PLOTTING MAST

To investigate the flow characteristics over the mid-upper fuselage of the aircraft a tubular steel mast of streamlined section, fitted with pressure probes, was mounted as indicated in Figs. 2, 4 and 6. To satisfy design considerations the mast was swept back through some 30° and braced by tubular steel struts, also of streamlined section, so as to form a rigid structure. The complete mast structure could be moved to three fixed positions on the fuselage top, and these positions in relation to that of the swept wing model to be tested are shown in Fig. 6.

Provision was made on the mast itself for the attachment of nine pairs of pressure probes as required, and the pressure tubes from these were passed through the interior of the mast to its base and thence into the aircraft fuselage. Neoprene tubing (approx. 5/32" bore) was used for conveying these pressures from mast probes to the manometer inside the aircraft as this has a much greater resistance to kinking than rubber.

2. 3. THE PITOT AND STATIC TUBES

The pitot and static tubes attached to the mast were of $\frac{1}{8}$" O. D. copper tube and each extended some 10" ahead of the mast. They were arranged in pairs as indicated in Fig. 1a, there being nine pairs in all disposed at stations along the mast as shown in Fig. 6.

2. 4. CONRAD YAWMETERS

To determine the directional characteristics of the airflow over the fuselage in the pitching plane, nine 'Conrad' type yawmeters were attached to the mast, replacing the nine pairs of pitot and static tubes described in paragraph 2. 3. These were made simply of two $\frac{1}{8}$" O. D. copper tubes soldered together to form a probe of "double bubble" section. The ends of these probes were carefully shaped, using a jig, to an included angle of 70°, and each probe assembly attached to the mast as indicated in Fig. 1a.

3. EXPERIMENTAL PROCEDURE

3.1. BEHAVIOUR OF AIRCRAFT AND EQUIPMENT IN FLIGHT

Up to the present the behaviour of the aircraft and its equipment in flight has proved to be entirely satisfactory. Although the manometer and camera observer unit is attached to the aircraft via resilient mounts there is little or no noteworthy vibration of this assembly in flight. It has been found that experimental conditions can be set with a high degree of stability this being in the main due to the skilfull and accurate handling of the aircraft on the part of the pilot. This fact has been deduced from careful observation of the behaviour of the fluid columns in the manometer during each of the test runs completed so far.

During the first flight the workable speed range of the aircraft was determined. This proved to extend from 100 knots I. A. S. (with 20^{o} flap) to 200 knots I. A. S. at a test altitude of 5, 000'. * Subsequent test runs were made at suitably spaced intervals in this range.

3.2. MEASUREMENT OF TOTAL HEAD, STATIC PRESSURE, AND VELOCITY DISTRIBUTIONS OVER THE MID-UPPER FUSELAGE

With the mast positioned in turn at each of the three stations indicated in Fig. 6, the distribution of total head, static pressure and velocity over the mid upper fuselage of the Lancaster was determined for each one of the test configurations listed in paragraph 1. 2. at a test altitude of 5,000'. Photographic records of the manometer were made during each test run.

The variation of aircraft geometric incidence with forward speed was determined during one set of test runs, using a clinometer mounted on a datum surface parallel to the fuselage axis. This was purely to obtain an estimate of the change of incidence which would be encountered over the workable range of flying speeds.

3.3. CALIBRATION OF CONRAD YAWMETERS

Since it was not desired to determine the direction of the flow over the fuselage in the pitching plane to any great degree of accuracy it was considered unnecessary to individually calibrate each one of

---

* This corresponds to a Reynolds number range of from 11×10^6 to 22×10^6 referred to the effective chord (130") of the swept back half wing model to be tested.

the Conrad yawmeters. Instead, two representative yawmeters exactly similar to those mounted on the mast were calibrated in the College of Aeronautics No. 6 wind tunnel and the curves of pressure coefficient (C_p) $\sim$ angle of yaw (ψ°) so obtained may be seen in Fig. 8. From an inspection of these curves it can be seen that there is a small difference between their slopes, and that one curve has a small zero error along the ordinate at zero yaw. This latter is no doubt due to small manufacturing errors.

If due account is made for this zero error, it can be seen that the difference between the curves (in terms of the pressure coefficient C_p) is small enough to be neglected provided that the angles of yaw considered are correspondingly small. On the assumption that the directional changes of the flow field over the aircraft fuselage would also be small, it was decided that the results of the tests to be performed could be analysed to the degree of accuracy required using the calibration curve shown in Fig. 8, as obtained above. This has since been proved justifiable.

3. 4. MEASUREMENT OF FLOW DIRECTION OVER FUSELAGE

The nine Conrad type yawmeters were attached to the mast in the manner indicated in Fig. 1a. Test runs were made for each of the test configurations listed in paragraph 1. 2. and with the mast positioned at the forward and aft stations. Photographic records of the manometer were made during each run.

3. 5. MEASUREMENT OF S. P. E. C.

The pressure error correction to the static system of the aircraft was determined by the aneroid method. The correction was found to be small, varying from $\frac{1}{4}$ knot at 120 knots to approximately 1 knot at 200 knots I. A. S. The curve of S. P. E. C. against airspeed may be seen in Fig. 7.

4. TEST RESULTS

4. 1. REDUCTION OF RESULTS

The film records of test runs made were read and readings corrected as and where necessary for static pressure error. Where possible, results were reduced to non dimensional form.

4. 2. PRESENTATION OF RESULTS

All the relevant data obtained from the tests performed may be seen plotted in Figs. 9 to 25. Initially, plots were made in terms of tube hole positions on the mast (see Figs. 9 to 19) and the curves thus obtained apply to the flow field strictly in the plane of the mast only (i. e. to a plane swept back through 30° relative to the normal to the fuselage axis). This is somewhat confusing, and so to transform these results to apply to planes normal to the fuselage axis, Figs. 16 to 19 were prepared, from which both the distributions of static pressure and velocity could be obtained at any desired station in the region explored.

The curves which have been obtained using this method may be seen in Figs. 20 to 25 and these show the distribution of static pressure and velocity in planes normal to the fuselage axis at distances of 50", 100", 150" from the leading edge of the wing at the root datum (see Fig. 5).

The directional characteristics of the flow in the pitching plane may be seen in Fig. 9.

5. DISCUSSION OF RESULTS

5. 1. LIMITS OF ACCURACY

Up to the present it has been found that the only limitation upon the accuracy of the results is that encountered in the reading of the film records of the manometer. The nature of the behaviour of the experimental equipment has been discussed in paragraph 3. 1. and thus whilst true and accurate readings of pressure may be displayed upon the manometer it has so far proved impractical to attempt to read the manometer film records to an accuracy involving less than 0.05" of manometric fluid. The effects of this limitation upon the experimental readings is illustrated in Appendix I where the static pressure coefficients are considered.

During the plotting of the experimental results this limitation has been borne in mind and the results interpreted accordingly.

5. 2. DISTRIBUTION OF TOTAL HEAD

The distributions of total head over the mid upper fuselage of the Lancaster, although not shown in the figures, were found to be constant over the height range covered by the mast (i. e. from 15 to 90 inches above the aircraft fuselage). This uniformity of the flow field with respect to the total head distribution indicates that no energy losses are being incurred in this region and hence we may deduce with some degree of certainty that the region of investigation is both extraneous to the boundary layer on the aircraft and also from any wakes or separated flows.

5. 3. DISTRIBUTIONS OF STATIC PRESSURE

The distributions of static pressure in planes normal to the fuselage axis may be seen in Figs. 20 to 22. For the speed range considered it can be seen that the static pressure field has small positive values with $C_p \leqslant +0.1$. The tendency shown by these curves is for the pressure coefficient to fall off in numerical value (tending towards free stream values $C_p = 0$) as the distance above the fuselage top exceeds approximately 5'. Below this height it is sensibly constant.

The pressure gradient although very small in all cases is at its worst at the low speed end of the flight range. For most of the flight configurations used this gradient is adverse (positive) in the region to be occupied by the wing, but is so small that its presence may be neglected.
(N. B. The maximum adverse gradient is $\partial C_p/\partial x = 0.0005$/inch)

Since the static pressure field is uniform and quite small in general, it is safe to assume that this field can be linearly separated (i. e. by the principle of superposition) from that due to the presence of the swept back wing model when fitted to the aircraft. This need for separation of flows really only manifests itself in the measurement of the distribution of pressure over the swept back wing model, and the field due to the aircraft can be regarded merely as a series of correction factors to apply to the model test results. As regards the boundary layer investigation the situation is different for in this case the nature of the whole flow field (due to both wing and aircraft) must be taken into account. Fortunately the pressure gradients due to the aircraft in the field under consideration are small enough to be neglected and we may treat the investigation of the boundary layer on the wing model paying no attention to the flow field over the aircraft.

5. 4. DISTRIBUTIONS OF VELOCITY

The distributions of velocity with varying flight speeds shown in Figs. 23 to 25 show similar tendencies to the distributions of static pressure i. e. tending towards free stream values when the height above the fuselage exceeds 5'. (This is of course to be expected for constant total head distribution with height above the aircraft fuselage, see paragraph 5. 2.) It appears that in the worst case (Fig. 25, 200 knots I. A. S.) there is a departure of some 6% from the free stream velocity. More important however, is not the absolute magnitude of the velocity in the region to be occupied by the wing model with respect to the free stream, but the variation of the velocity over this region. At a maximum this is of the order of 3% (see Fig. 24 200 knots I. A. S.).

Although such a variation does constitute a change in the spanwise distribution of Reynolds number the percentage change occuring is exactly the same as that for the velocity distribution (since $R_e = V1/\nu$). Moreover the characteristics of the laminar and turbulent boundary layers are approximately dependent upon $1/\sqrt{R_e}$ and $1/\sqrt[5]{R_e}$ respectively, so that the effect of a small percentage change in Reynolds number results in a much smaller change in the boundary layer characteristics. The only case where difficulty may arise is that of the effect of Reynolds number on sweep instability, especially near to the critical Reynolds number range. Physically, this would result in an earlier state of transition occurring near the model wing tip if transition does occur due to sweep instability alone.

5. 5. FLOW DIRECTIONAL CHARACTERISTICS

Fig. 9 shows the directional characteristics of the flow in terms of tube position in the plane of the mast. It can be seen that the overall change of flow direction over the speed range (flaps 0^{o}) is from $\Theta = -2^{o}$ to $\Theta = +2^{o}$ approx. (where Θ is the angle of the flow with respect to the fuselage axis, Θ being considered positive when the flow is tending to move down towards the fuselage). This change of flow direction is inclusive of the effects of change of aircraft incidence and is tantamount to a change in the angle of sweep of the half wing. This change is however so small that it most certainly may be neglected.

5. 6. THE EFFECT OF THROTTLING THE INBOARD ENGINES

Throttling of the inboard engines seems to have only a very slight effect upon the flow characteristics in the region of investigation (see Figs. 10 to 12). It was found difficult to fly straight and level under these conditions at constant altitude (for reasons of insufficient power) and it is unlikely that any test work will be performed on the model wing with the aircraft in this configuration.

5. 7. THE EFFECTS OF STEADY SIDESLIP ON THE DISTRIBUTIONS OF STATIC PRESSURE

Although it is not intended to fly the aircraft in steady sideslip during the forthcoming tests, the effects of sideslip on the distributions of static pressure were investigated for certain flight configurations and the results obtained may be seen in Figs. 10 to 12. It may be seen that the effects are small, what departures there are from the zero sideslip cases being due to sidewash over the aircraft fuselage. This is evident from the fact that the departures from the curves for zero sideslip are greatest near to the fuselage top.

6. CONCLUSIONS

1) The behaviour of Lancaster P. A. 474, and its experimental equipment, as a test vehicle for the research project under consideration has so far proved to be entirely satisfactory. Experimental conditions can be set and maintained with a high degree of stability, this being due in the main to skilful and accurate handling of the aircraft on the part of the pilot.

2) The workable speed range of the aircraft was found to extend from 100 knots I. A. S. (with 20^{o} flap) to 200 knots at a test altitude of 5,000'. This corresponds to a Reynolds number range of from 11×10^{6} to 22×10^{6} referred to the effective chord of the swept back half wing model to be tested.

3) The distribution of total head over the mid-upper fuselage of the Lancaster was found to be constant indicating that no energy losses are being incurred and that the region of investigation (i. e. that to be occupied by the swept back half wing) is both extraneous to the boundary layer on the aircraft fuselage and also from any wakes or separated flows.

4) The distributions of static pressure in planes normal to the fuselage axis showed that small positive values of C_p ($C_p \leq +0.1$) may be expected in this region. The distributions of C_p were sensibly constant up to a height of 5' above the fuselage top, but with further increase in height tended to approach free stream values ($C_p = 0$).

5) The static pressure gradient in the region to be occupied by the model was small in all cases, the maximum adverse (positive) value

being $\partial C_p / \partial x = 0.0005$/inch.

6) The distributions of velocity showed a maximum departure from free stream values of some 6%, but the variation of U/U_o over the region to be occupied by the model had a maximum value of 3% (at 200 knots I. A. S.).

7) The flow direction in the pitching plane of the region of investigation was found to have a variation of from $\Theta = -2^{\circ}$ to $\Theta = +2^{\circ}$ (approx.) over the speed range considered. This change is inclusive of the effects of variation in aircraft incidence (geometric) and is tantamount to a change of $\pm 2^{\circ}$ in the angle of sweep of the half wing model.

8) It was found impracticable to fly the aircraft in a satisfactory manner at constant altitude with the two inboard engines throttled and hence the possibility of making use of this flight configuration has been abandoned.

APPENDIX I

Limits of Accuracy

Consider the effect of the limits of accuracy in the reading of the manometer upon the static pressure coefficient C_p.

We have:-

At 100 knots $\frac{1}{2}\rho U_o^2 \doteq 4.0''$ carbontetrachloride

At 200 knots $\frac{1}{2}\rho U_o^2 \doteq 16''$ carbontetrachloride

Now $C_p = \dfrac{P - P_o}{\frac{1}{2}\rho U_o^2}$, so that assuming a limiting accuracy of 0.1'' on the manometer, we find that the limiting accuracy on the calculated values for C_p are:-

At 100 knots: $C_{p_{Lt}} = \dfrac{0.1}{4.0} = 0.025$)
) Extreme ends of
) speed range covered
At 200 knots: $C_{p_{Lt}} = \dfrac{0.1}{16} \doteq 0.006$)

We may tabulate as follows:-

Speed (knots)	$\frac{1}{2}\rho U_o^2$'' (c. t. c.) (approx.)	C_p (corresponding to $P-P_o$ = 0.1'' c. t. c.)
100	4. 0	0. 025
115	5. 5	0. 018
130	7. 25	0. 013(8)
150	9. 5	0. 011
170	12. 0	0. 008
190	16. 0	0. 006(2)
200	16. 25	0. 006

N. B.

This table serves to illustrate the effect of an error of measurement of 0. 1'' in the manometer readings upon the static pressure coefficient C_p. For the actual boundary layer measurements to be performed on the model wing, the manometric fluid used will be distilled water. This has a specific gravity of 1. 00 whilst that for carbontetrachloride is 1. 599, and hence the limits of accuracy imposed in the interpretation of the film records will be much improved.

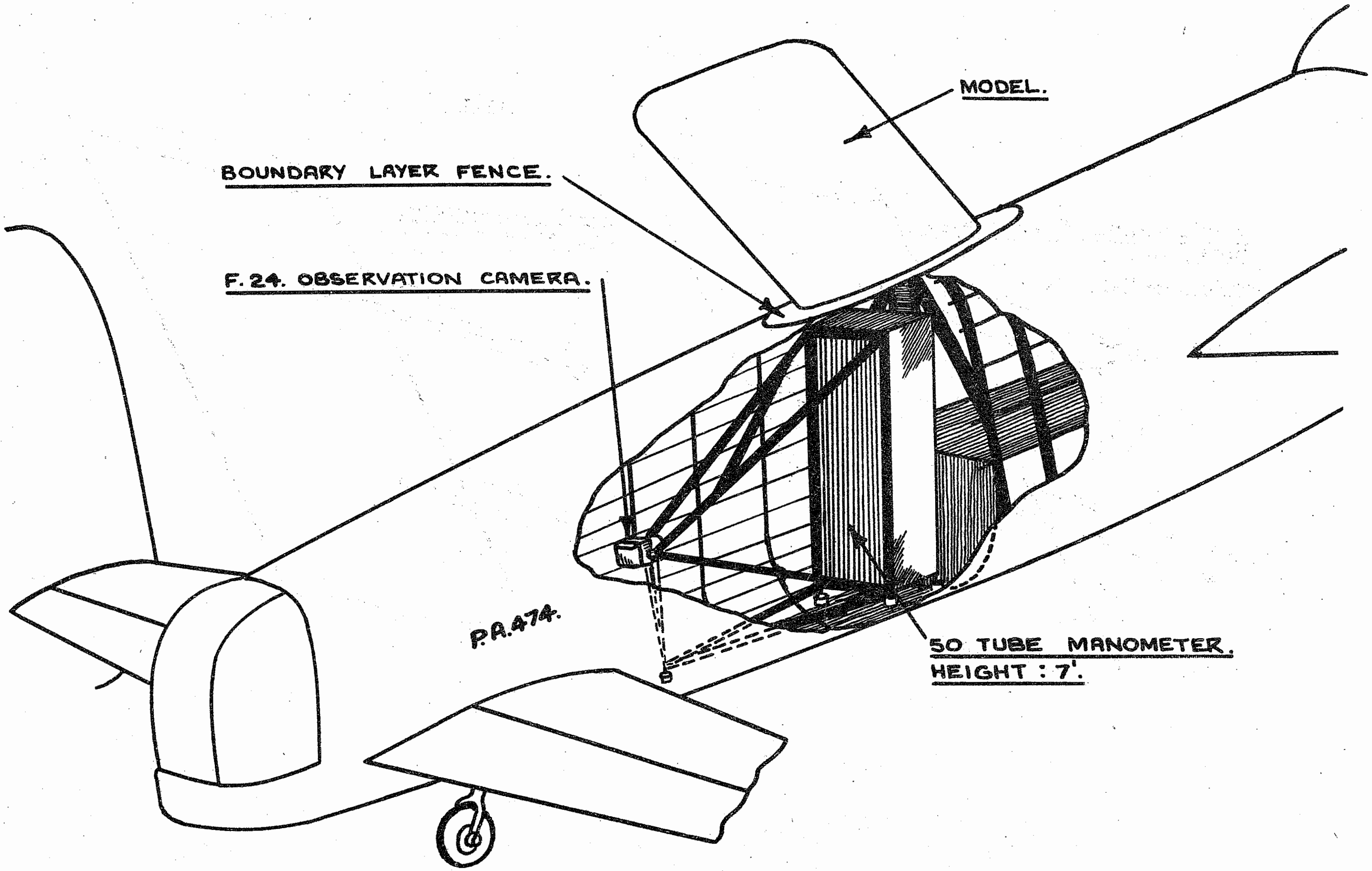

FIG. 1. LAYOUT OF EQUIPMENT IN LANCASTER.

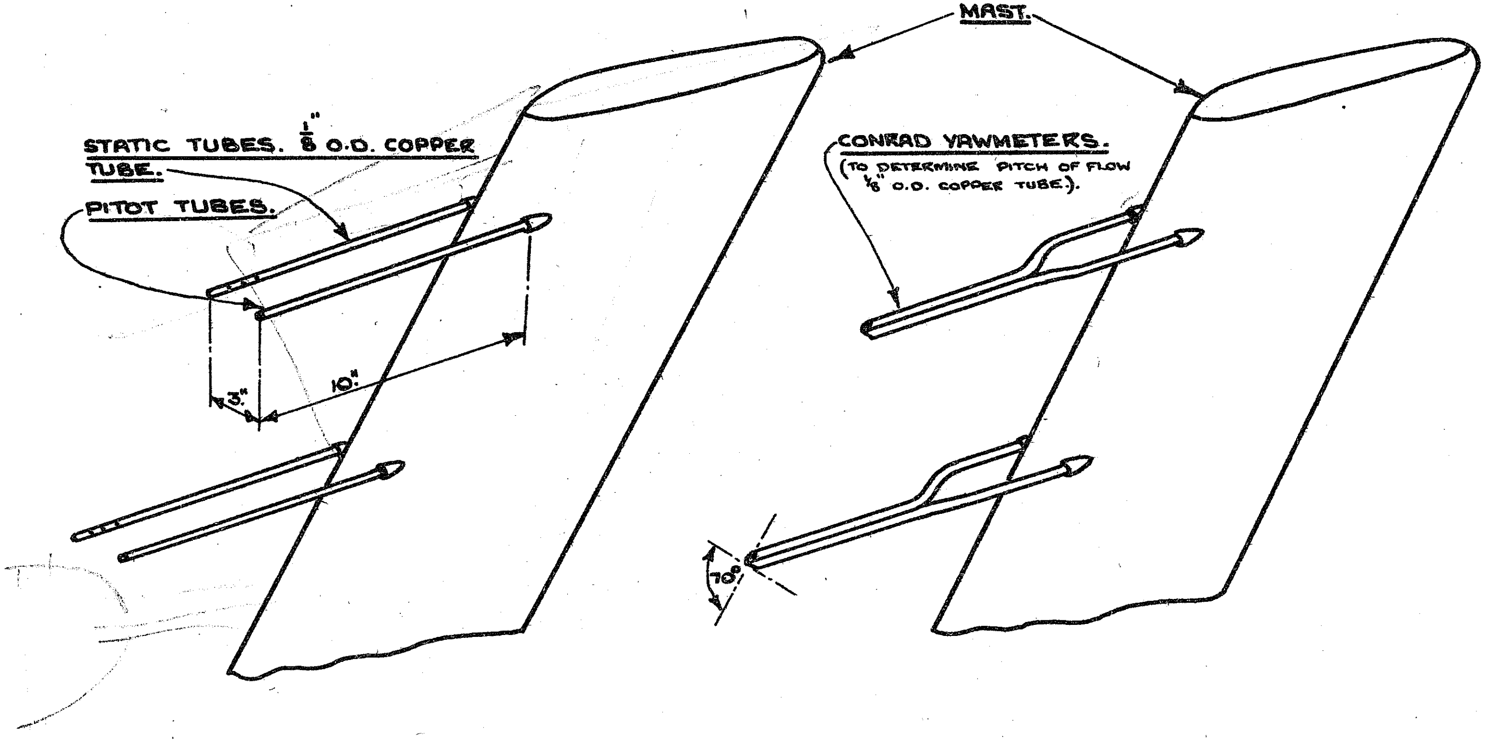

FIG. 1a. ARRANGEMENT OF PITOT TUBES, STATIC TUBES, AND CONRAD YAWMETERS ON MAST.

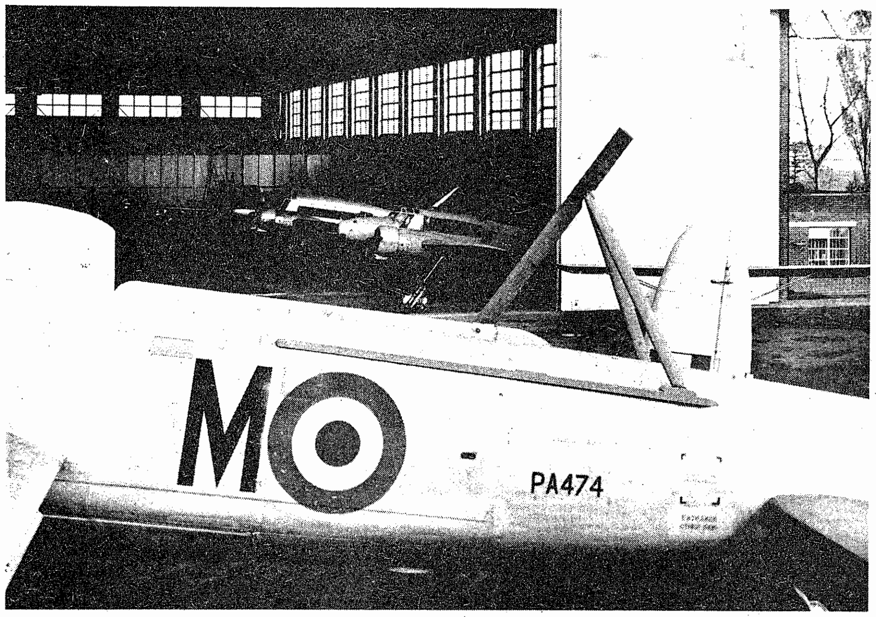

FIG. 2 PRESSURE PLOTTING MAST IN AFT POSITION.

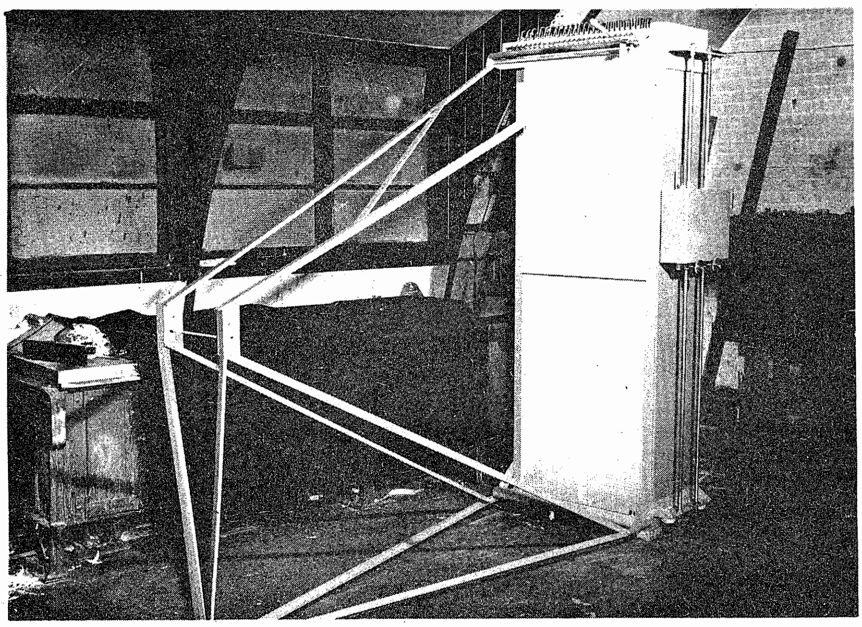

FIGURE 3. THE MANOMETER UNIT NEARING COMPLETION

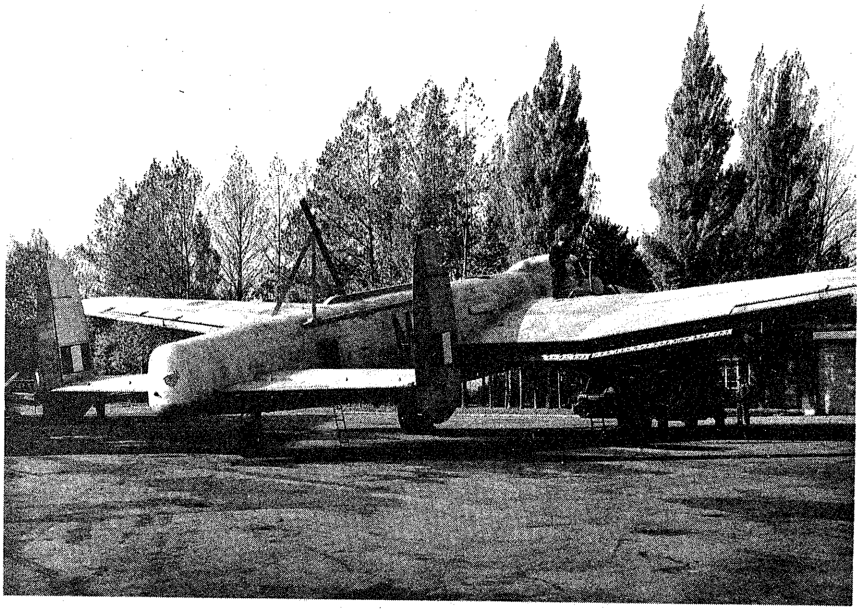

FIG. 4. AIRCRAFT WITH MAST IN AFT POSITION.

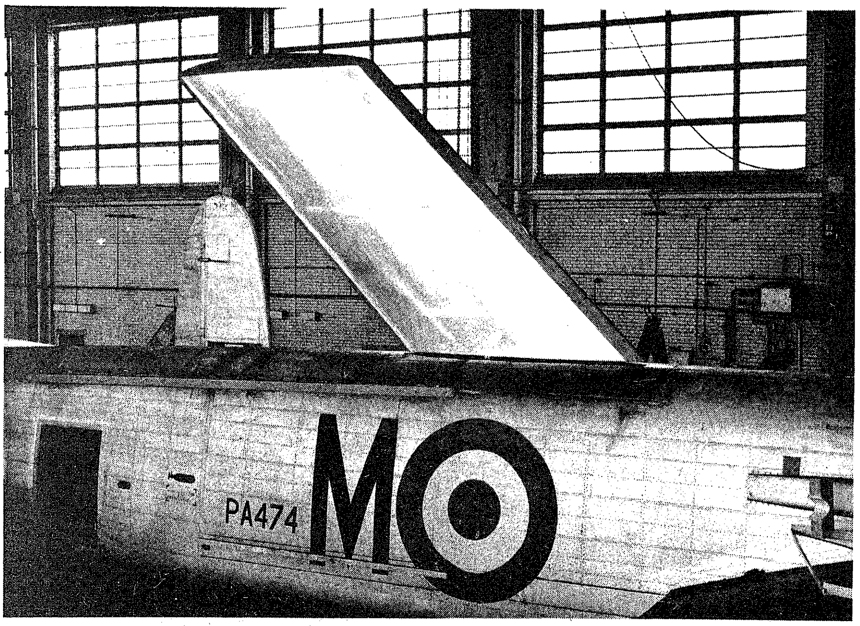

FIG. 5. SWEPT BACK HALF WING IN POSITION.

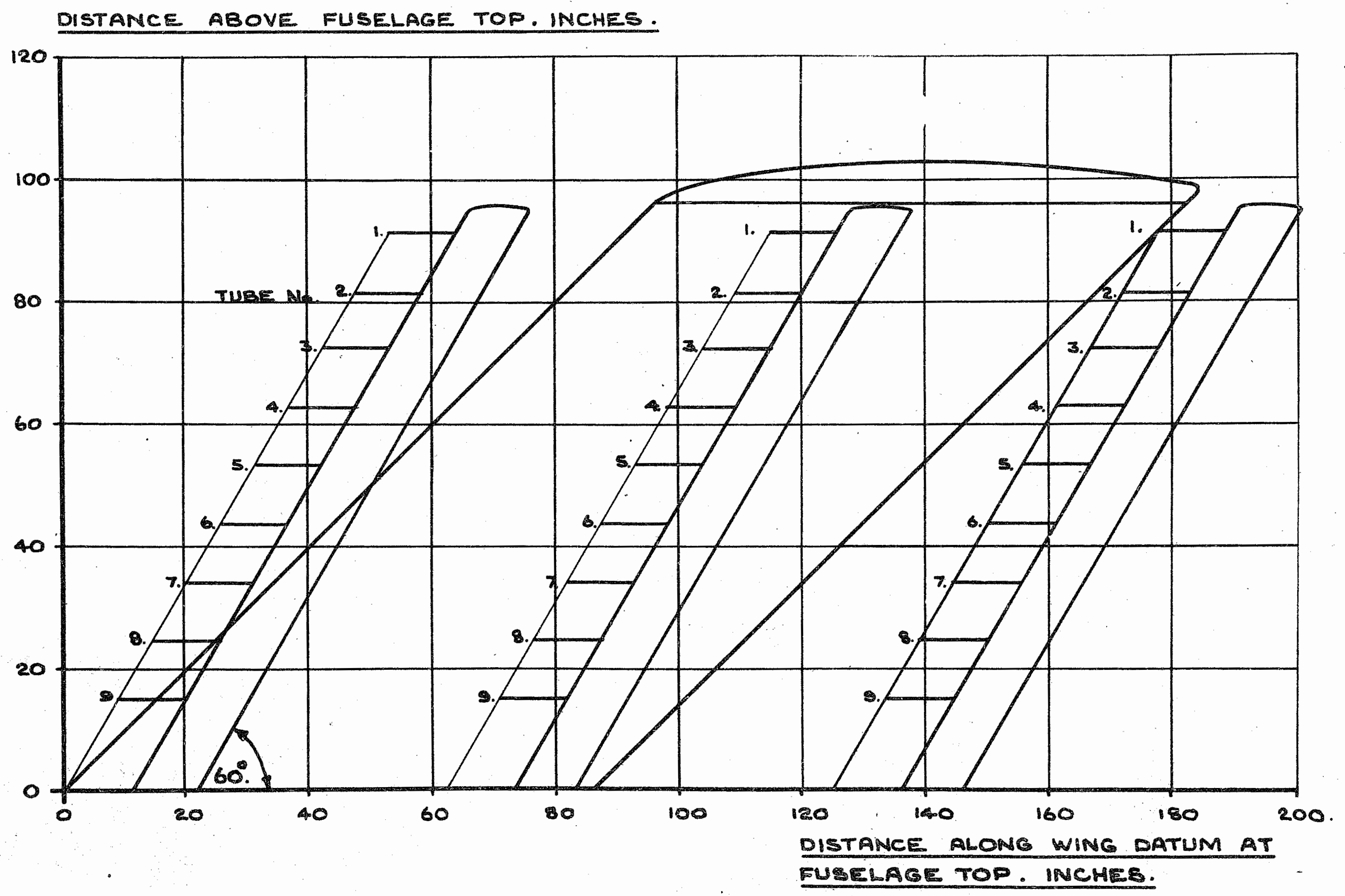

FIG. 6. RELATIVE POSITIONS OF MAST & WING.
(DIMENSIONS MAY BE SCALED).

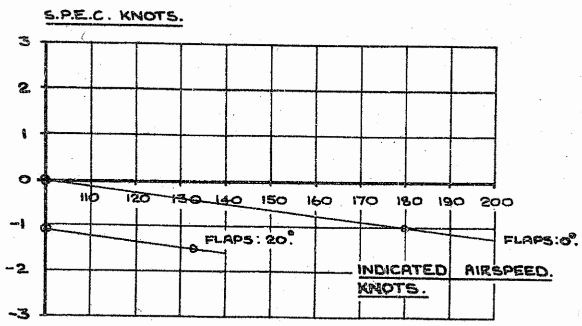

FIG. 7. STATIC PRESSURE ERROR CORRECTION CURVE FOR LANCASTER P.A.474.

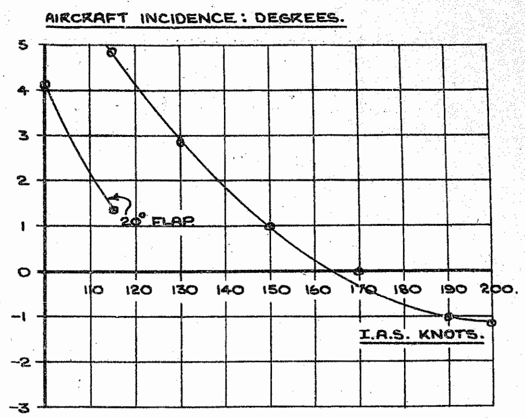

FIG. 7a. VARIATION OF AIRCRAFT INCIDENCE WITH INDICATED AIRSPEED AT 44,000 lb. A.U.W.

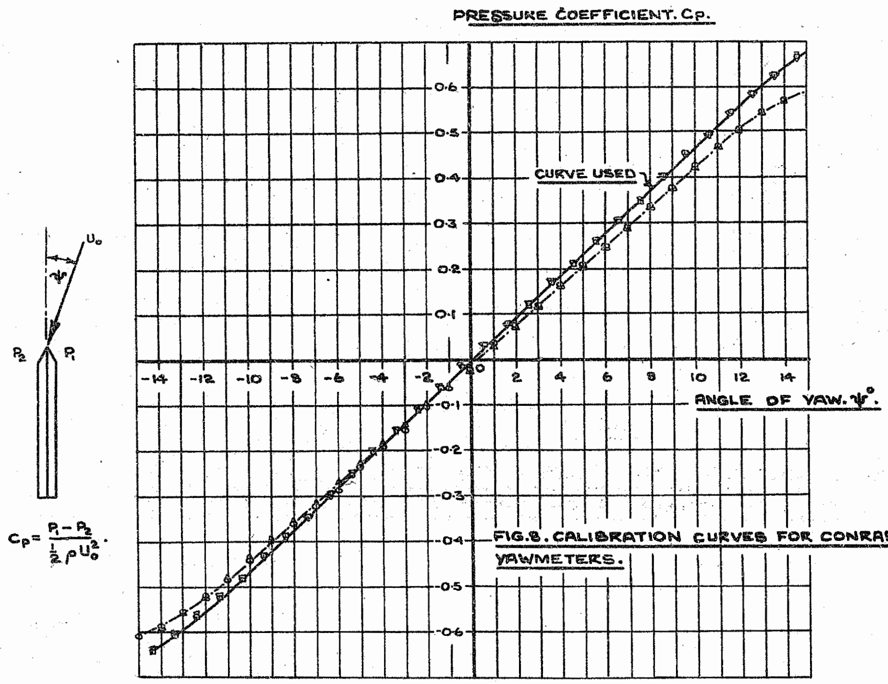

FIG. 8. CALIBRATION CURVES FOR CONRAD (70°) YAWMETERS.

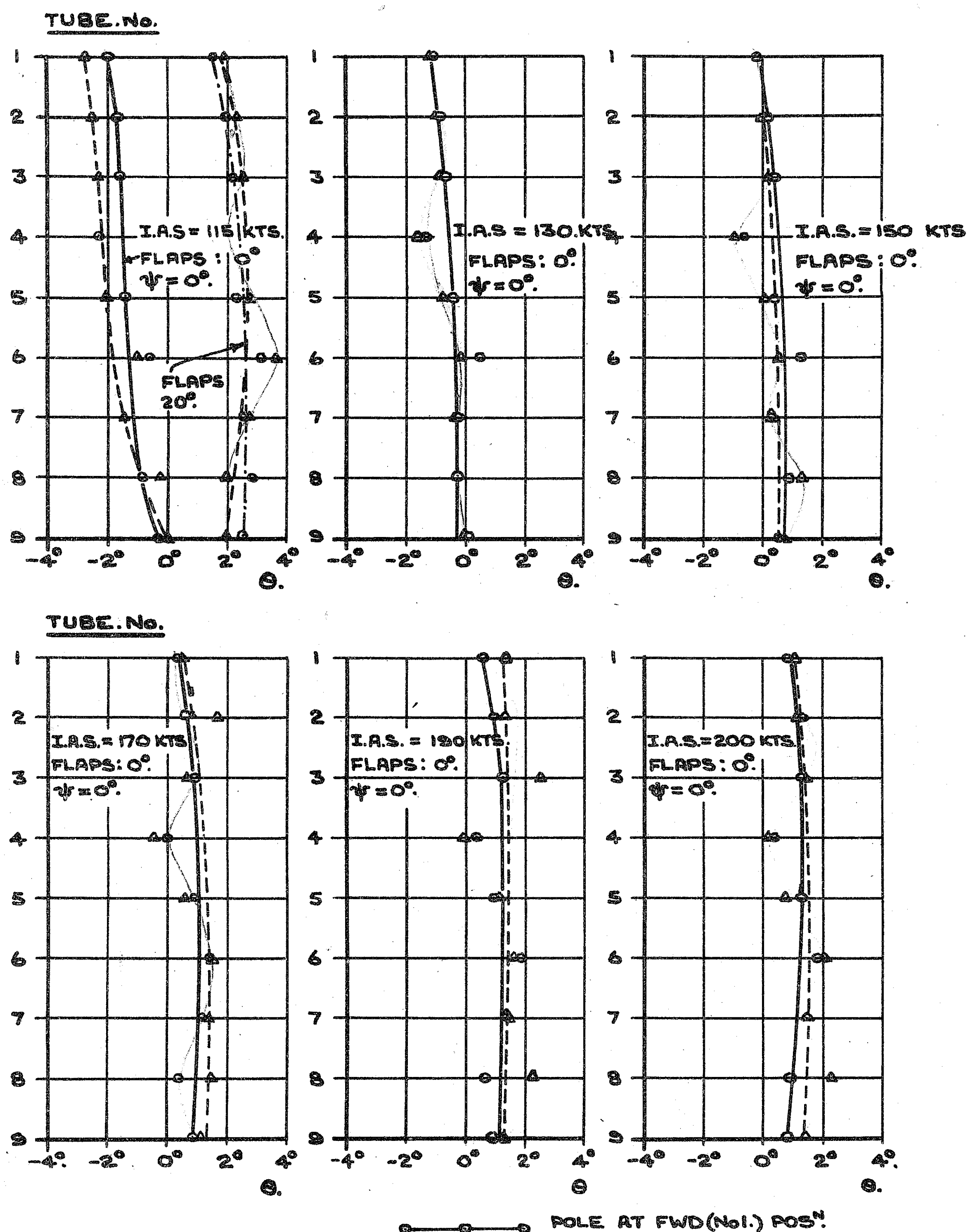

FIG.9. DIRECTION OF FLOW RELATIVE TO AIRCRAFT FUSELAGE IN TERMS OF HOLE POSITION AND IN PLANE OF POLE.

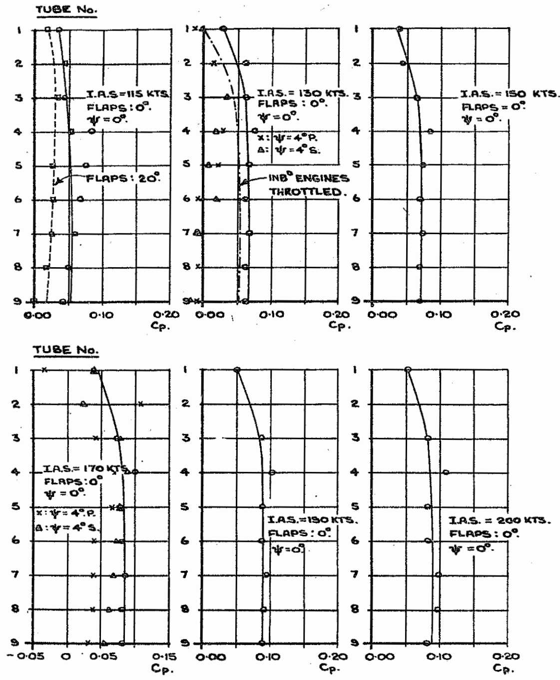

FIG. 10 DISTRIBUTION OF STATIC PRESSURE OVER AIRCRAFT FUSELAGE IN PLANE OF POLE AT FWD POS.^{N} No.1

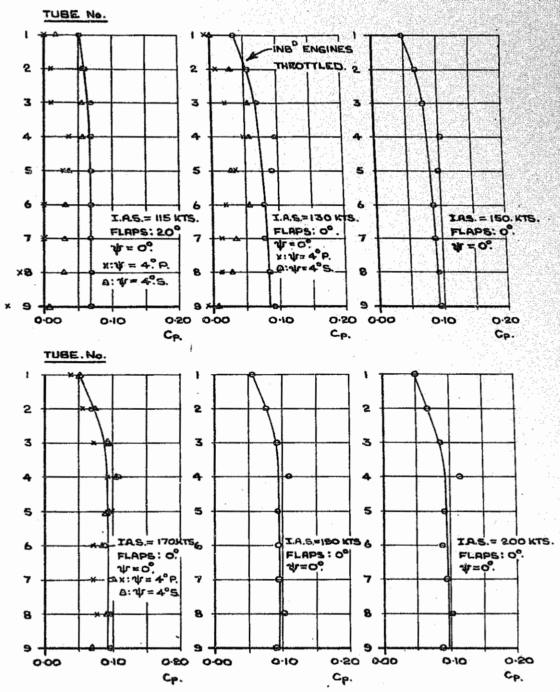

FIG.11 DISTRIBUTION OF STATIC PRESSURE OVER AIRCRAFT FUSELAGE IN PLANE OF POLE AT MID POS.^{N} No.2

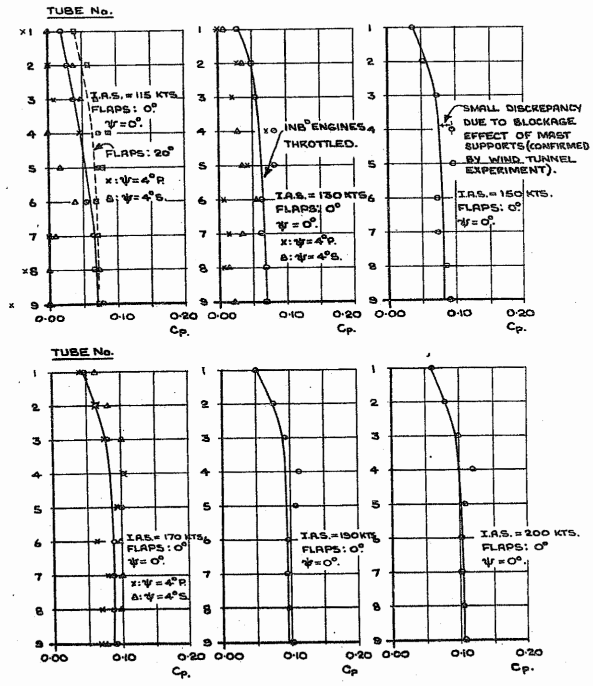

FIG. 12 DISTRIBUTION OF STATIC PRESSURE OVER AIRCRAFT FUSELAGE IN PLANE OF POLE AT AFT POS.^{N} No.3

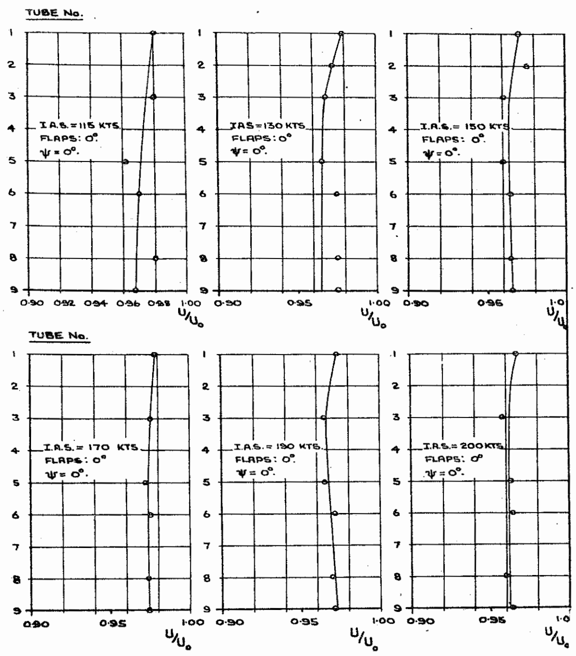

FIG.13 DISTRIBUTION OF VELOCITY OVER AIRCRAFT FUSELAGE IN PLANE OF POLE AT FWD POS.^{N} No.1

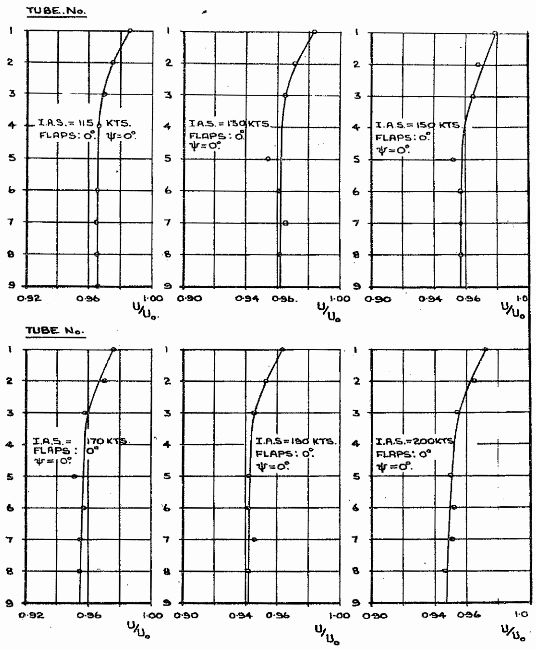

FIG.14 DISTRIBUTION OF VELOCITY OVER AIRCRAFT FUSELAGE IN PLANE OF POLE AT MID POS.^{N} No.2

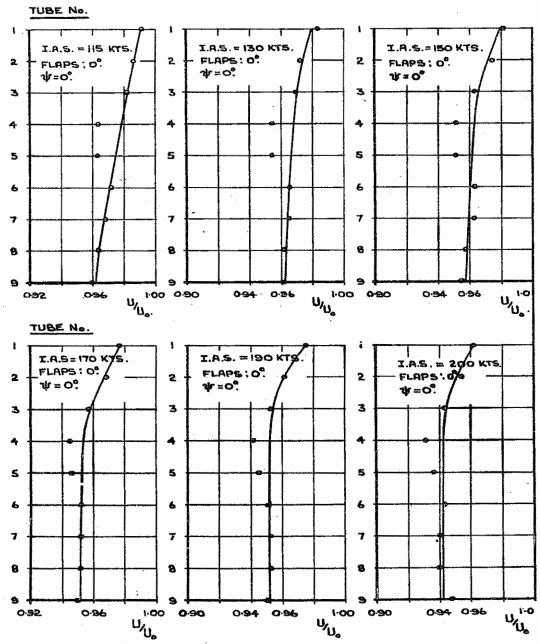

FIG.15 DISTRIBUTION OF VELOCITY OVER AIRCRAFT FUSELAGE IN PLANE OF POLE AT AFT POS.^{N} No.3

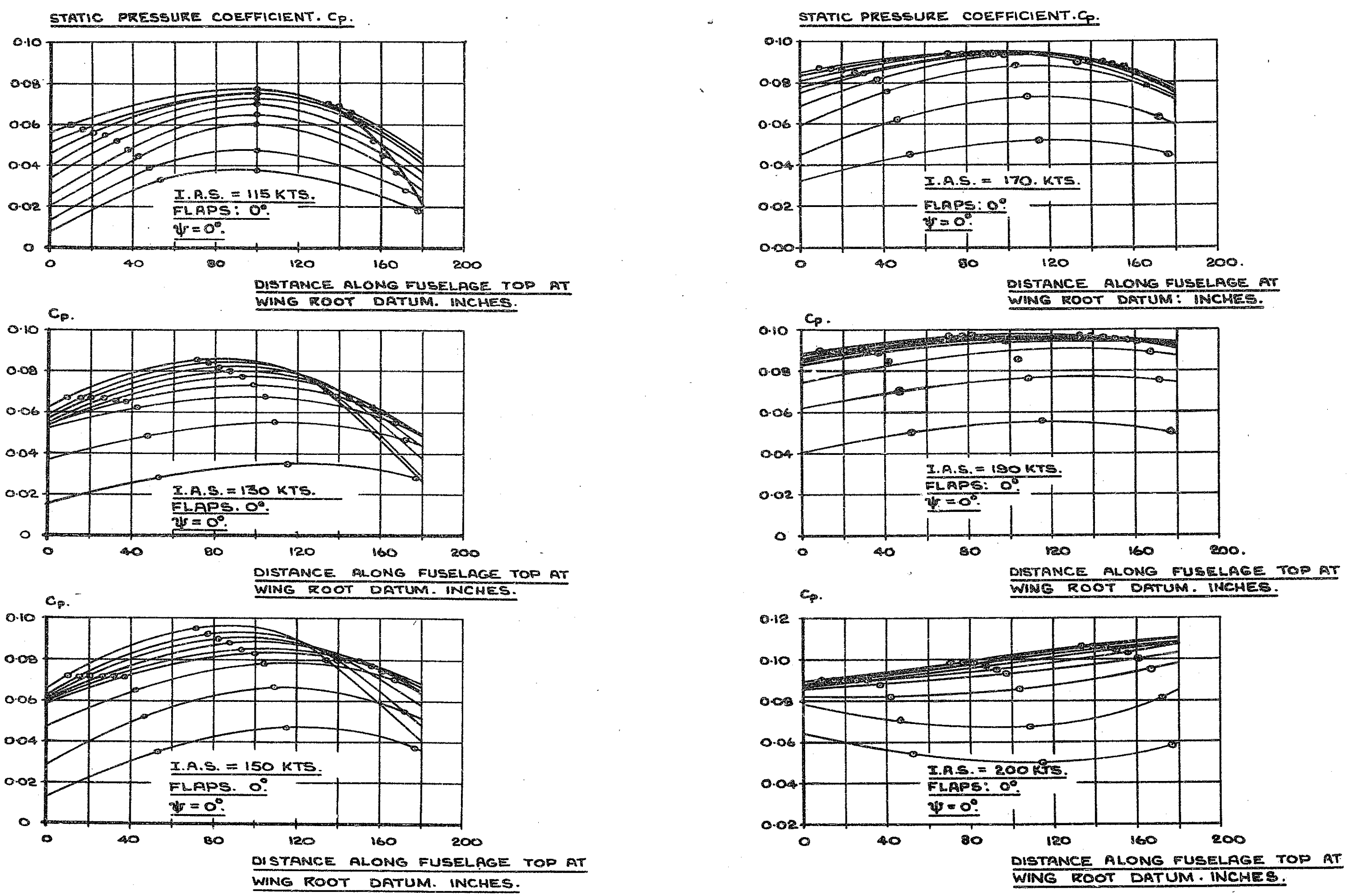

FIG.16 AND 17

STATIC PRESSURE DISTRIBUTION OVER FUSELAGE IN TERMS OF HOLE POSITION ON POLE.

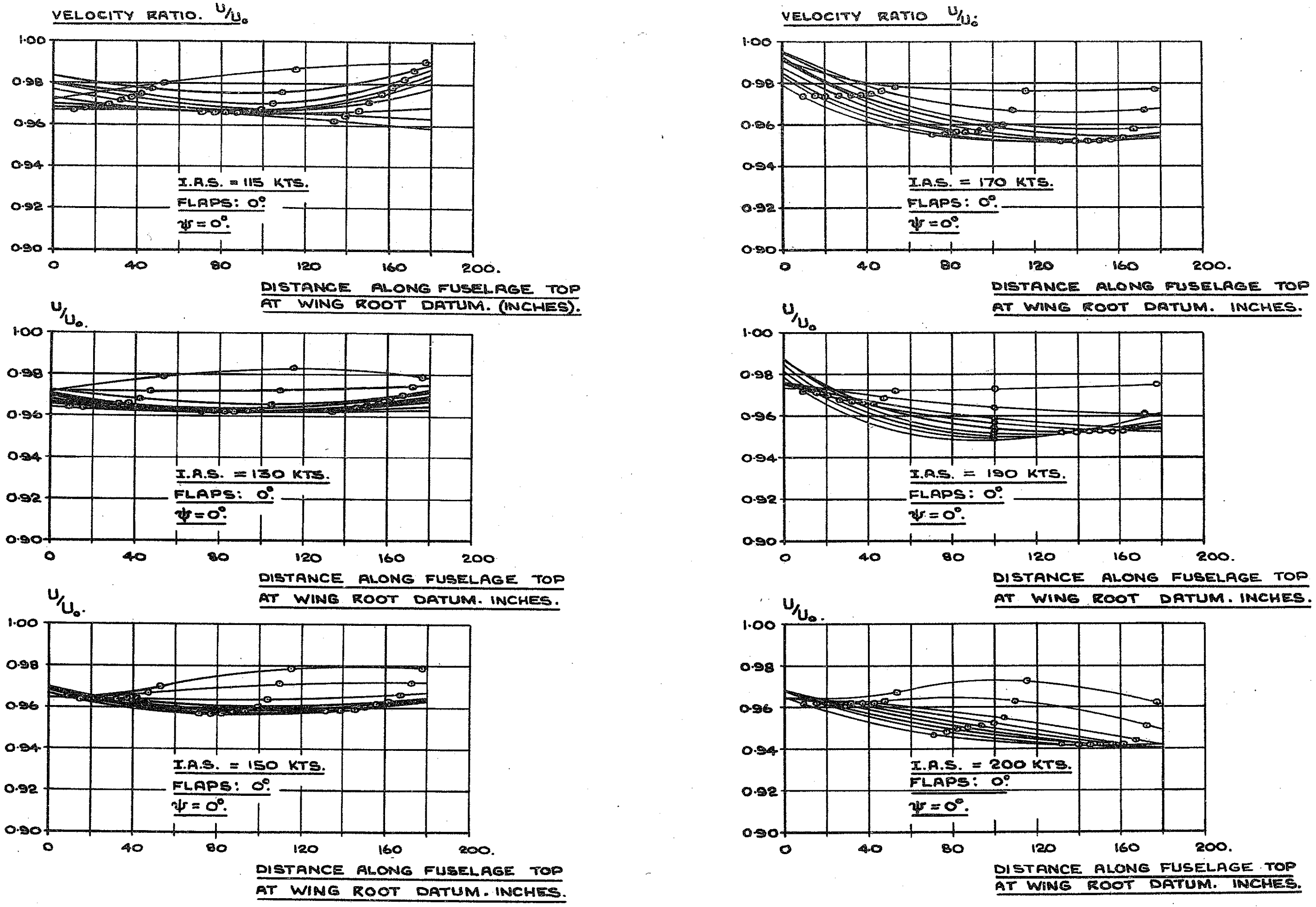

FIG. 18 AND 19

VELOCITY DISTRIBUTION OVER FUSELAGE IN TERMS OF HOLE POSITION ON POLE.

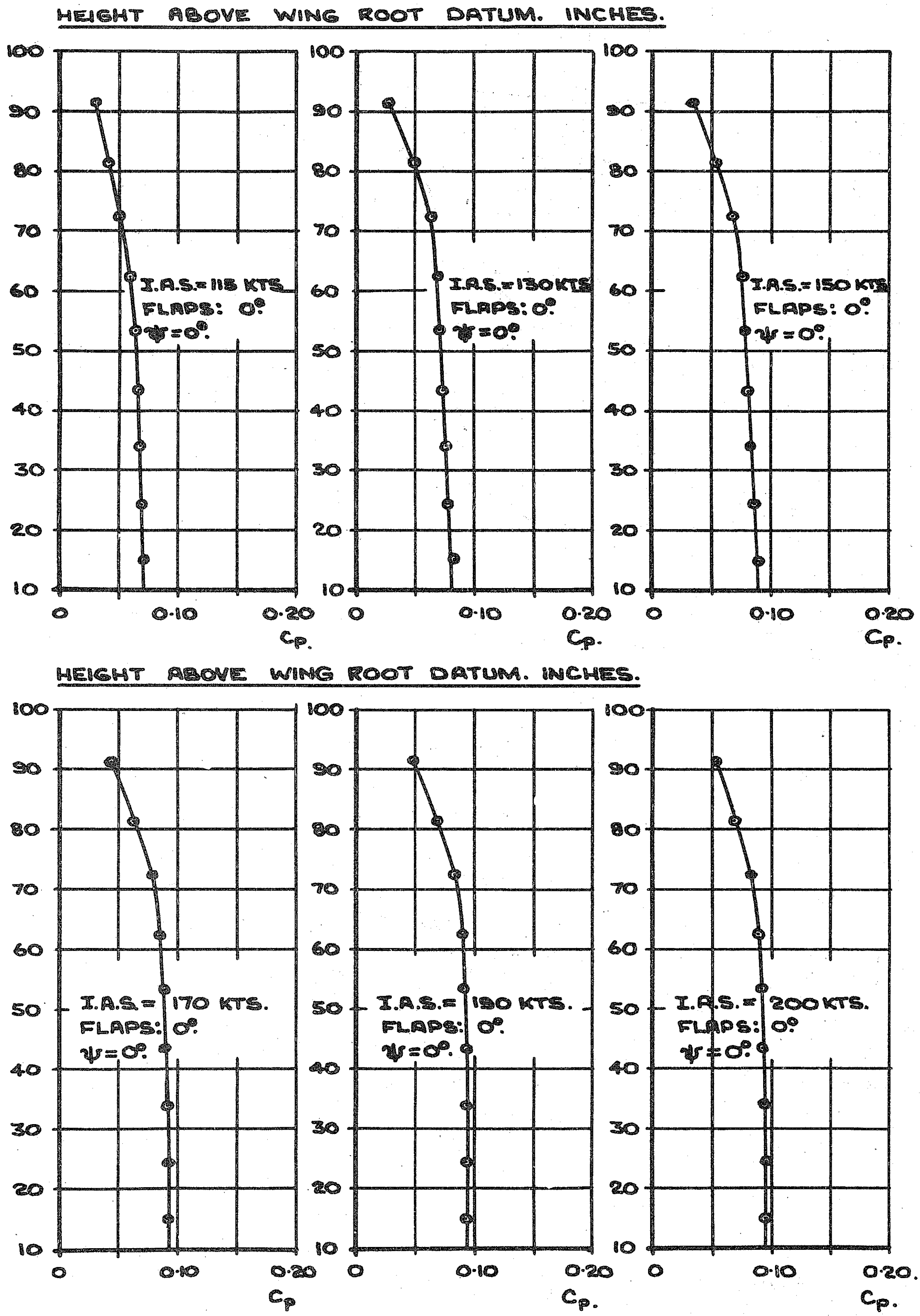

FIG. 20. DISTRIBUTION OF STATIC PRESSURE IN PLANE NORMAL TO AIRCRAFT FUSELAGE, 50" FROM WING L.E. AT ROOT DATUM.

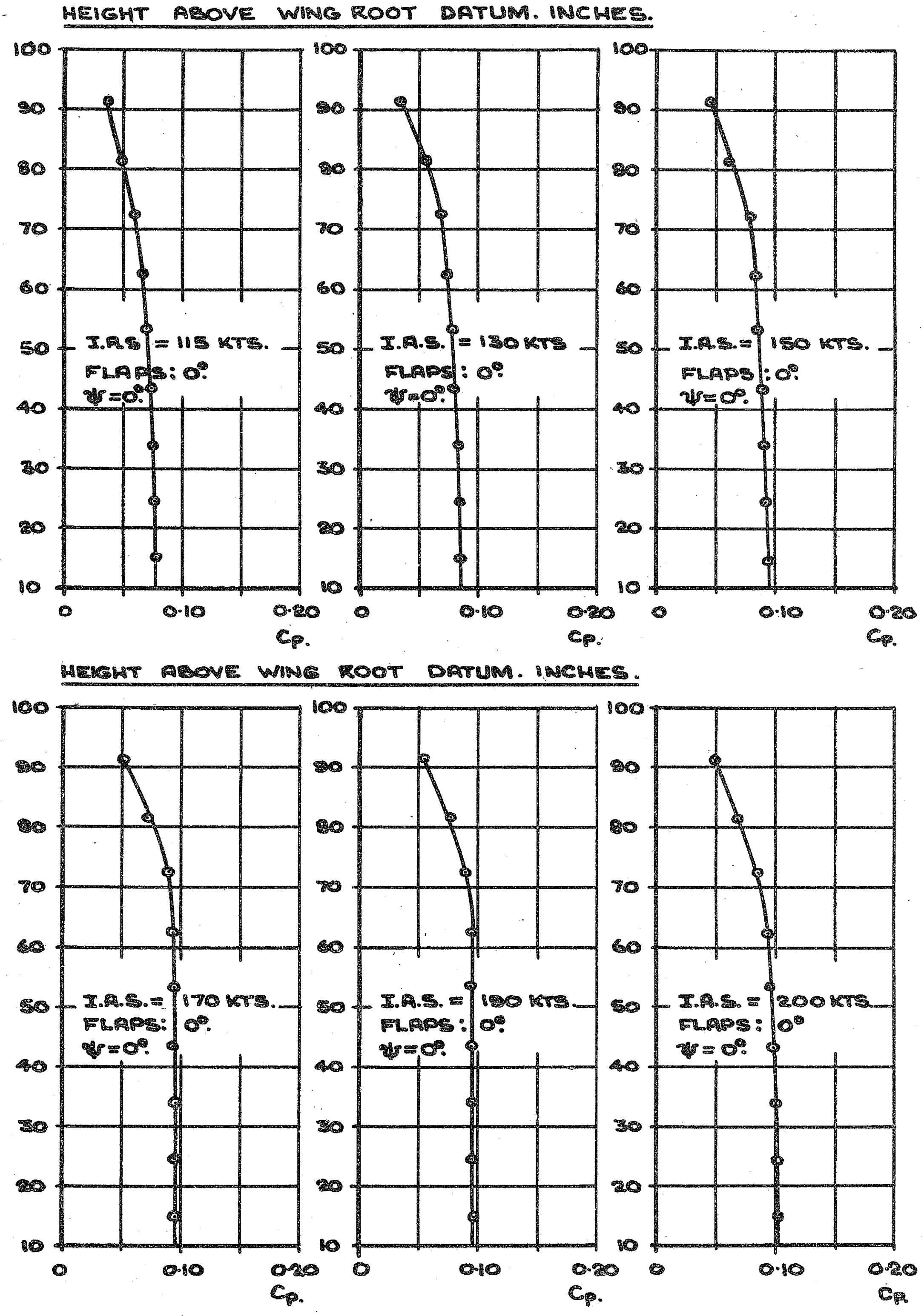

FIG. 21. DISTRIBUTION OF STATIC PRESSURE IN PLANE NORMAL TO AIRCRAFT FUSELAGE 100" FROM WING L.E. AT ROOT DATUM.

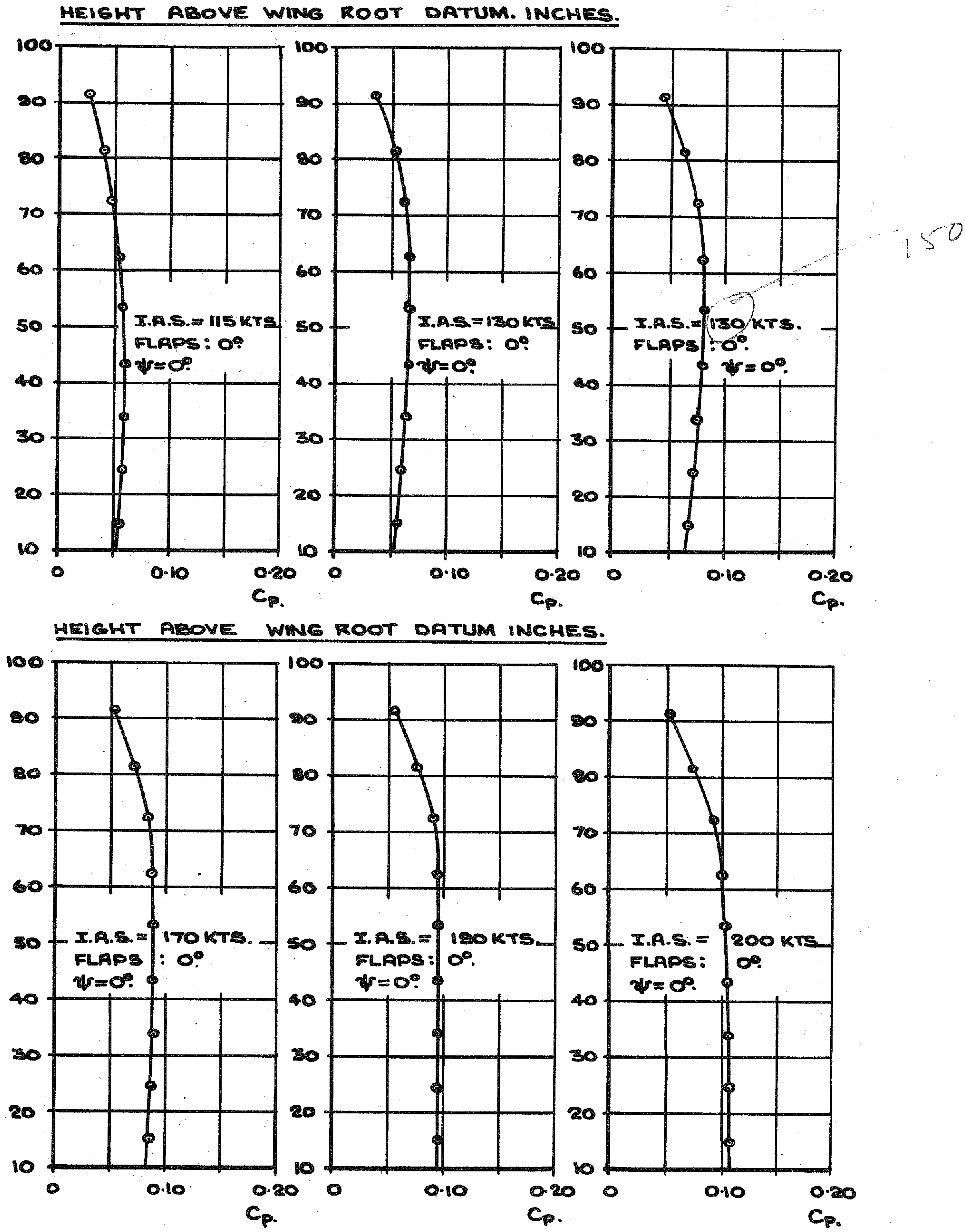

FIG. 22. DISTRIBUTION OF STATIC PRESSURE IN PLANE NORMAL TO AIRCRAFT FUSELAGE, 150" FROM WING. L.E. AT ROOT DATUM.

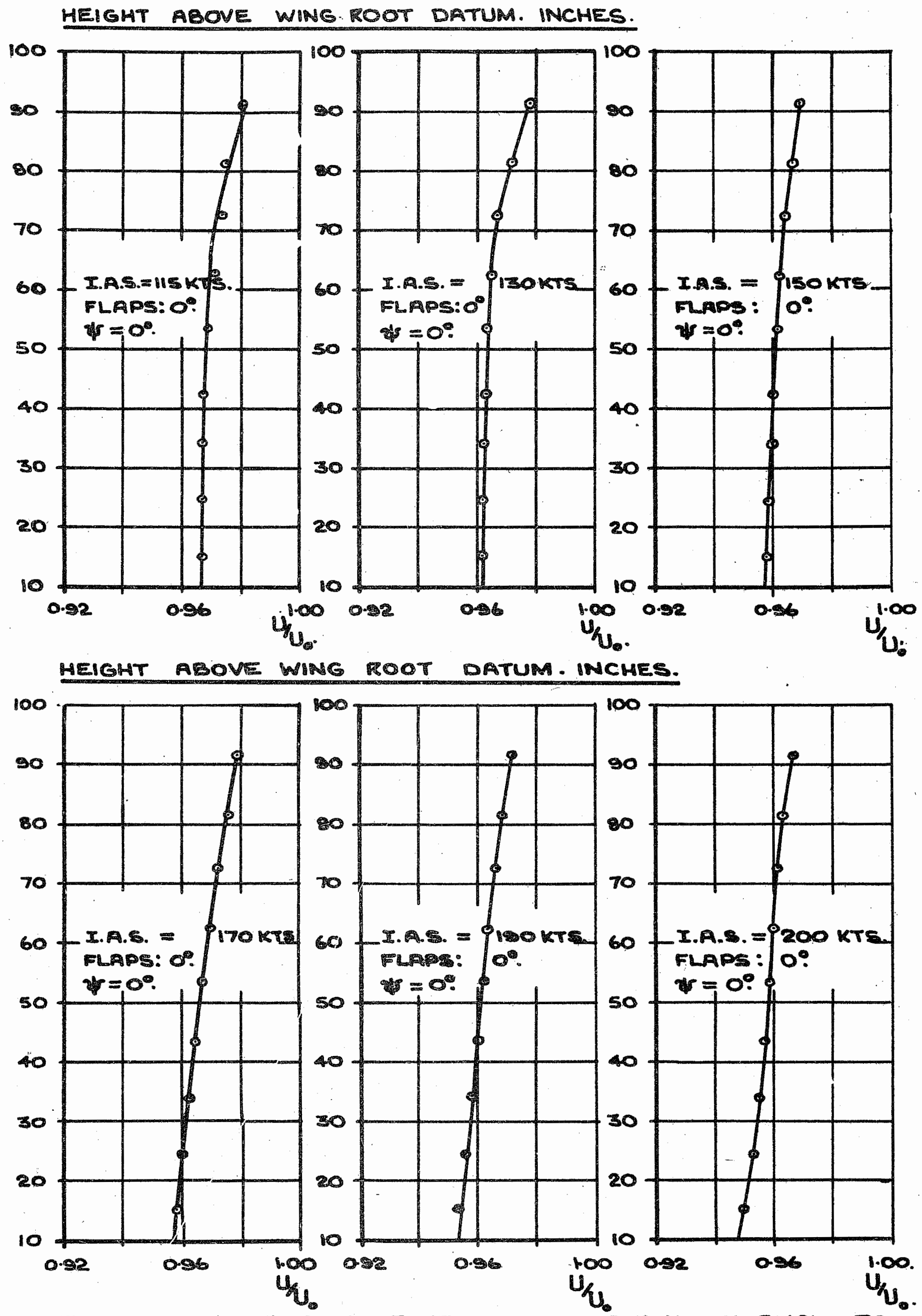

FIG. 23. DISTRIBUTION OF VELOCITY IN PLANE NORMAL TO AIRCRAFT FUSELAGE 50" FROM WING L.E. AT ROOT DATUM.

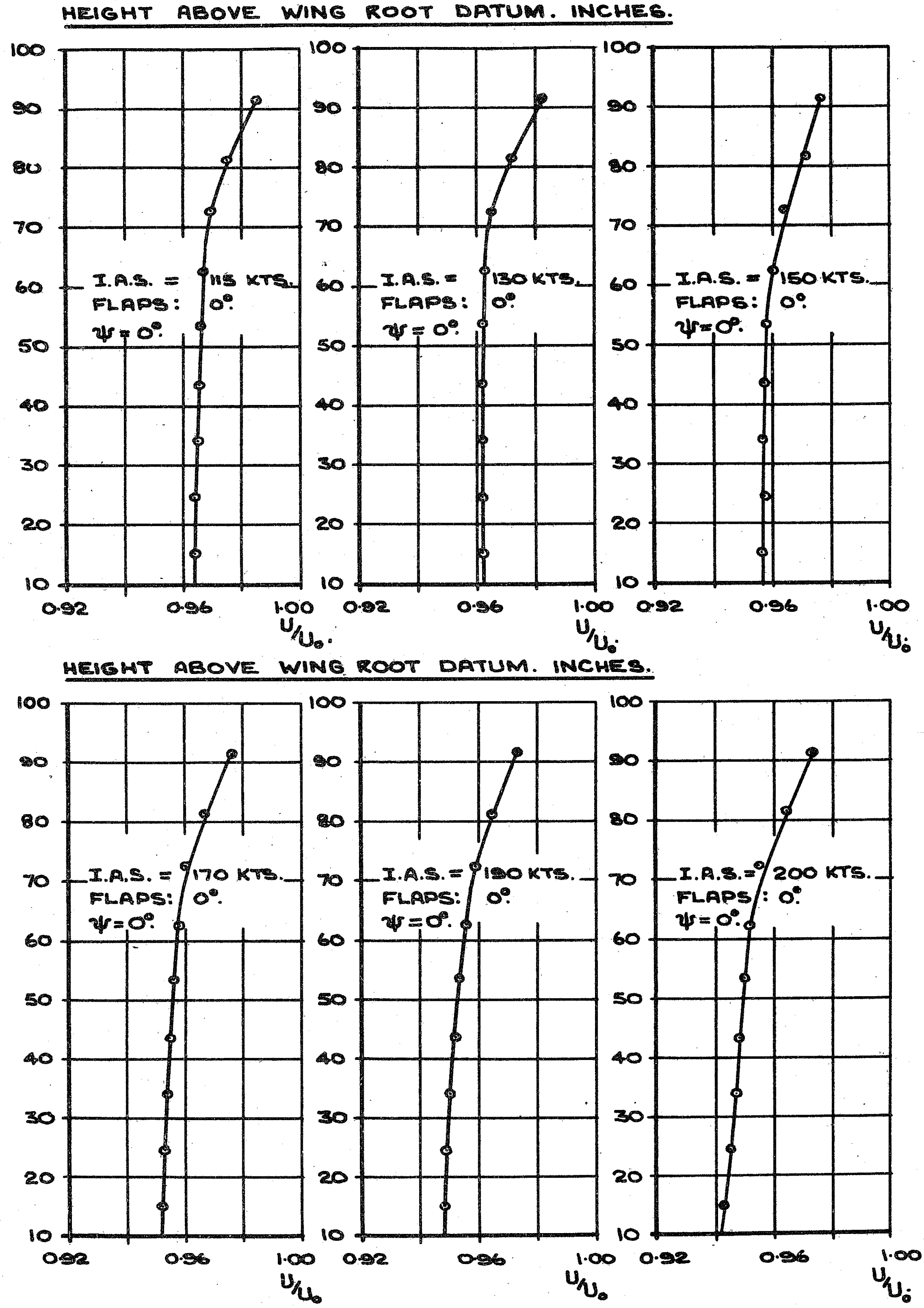

FIG. 24. DISTRIBUTION OF VELOCITY IN PLANE NORMAL TO AIRCRAFT FUSELAGE 100" FROM WING L.E. AT ROOT DATUM.

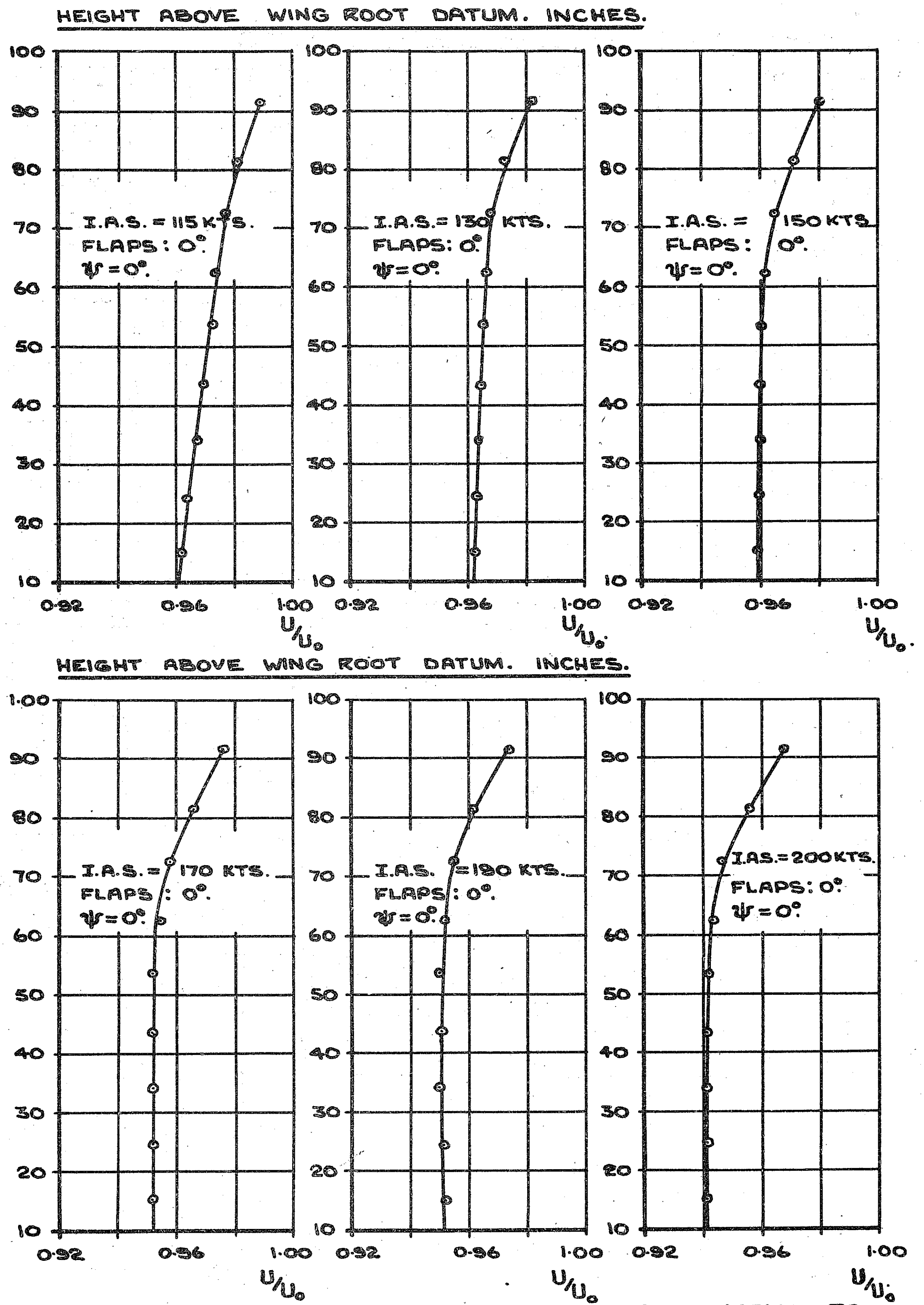

FIG. 25. DISTRIBUTION OF VELOCITY IN PLANE NORMAL TO AIRCRAFT FUSELAGE 150" FROM WING L.E. AT ROOT DATUM.