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MINISTRY OF SUPPLY

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CURRENT PAPERS

Saunders-Roe Princess Flying Boat
G-Alun Air and Water Performance Tests

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MARINE AIRCRAFT EXPERIMENTAL ESTABLISHMENT, FELIXSTOWE, SUFFOLK.

SAUNDERS-ROE PRINCESS FLYING BOAT G-ALUN

AIR AND WATER PERFORMANCE TESTS

S U M M A R Y

The basic air drag characteristics of the aircraft are good, being considerably better than has been achieved on previous British flying boats.

Corrected to zero slipstream these basic characteristics are:-

C_{DZ} = 0.0179
 D_{100} = 1070 lb.
 K = 1.12
 $(L/D)_{max.}$ = 18.9

There are points on the aircraft where elimination of air leaks (e.g. between the wing tip to float junction, and control surfaces), and attention to surface roughness, particularly on the outer wings, could lead to an improvement in these already good drag characteristics.

There is a measurable change, both in basic profile drag and induced drag, with slipstream. At cruising C_L values (i.e. less than 0.65) with $T_c = 0.05$, the relevant figures are:

C_{D_0} = 0.0188
 K = 1.16

There are also indications of a change in C_{DZ} with C_L at the higher C_L values.

During take-off, with the derated engines, the minimum longitudinal acceleration at the maximum drag, (hump), speed, in up to 1 ft. sea conditions, is 2.04 ft./sec.², (0.063g), at an aircraft weight of 295,000 lb. Employing the associated mean engine output derived from the single engine torque meters, the propeller thrust has been calculated from the manufacturers performance curves, Use of this total thrust at the hump speed indicates an approximate maximum total resistance value of 0.183 W.

At 295,000 lb. commencing from a taxiing speed of 15 knots, the distance to achieve an unstick speed of 100 knots in zero wind, I.C.A.N. conditions, with a fixed elevator setting of -9 degrees is 4,750 ft.

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1. INTRODUCTION

The programme of performance tests on the Princess has, of necessity, been dictated by the fact that the aircraft could not be flight tested in a final form. The Proteus 600 series engines were installed as an interim measure in order to enable flight trials to be carried out on the airframe, and it was not intended that this engine-airframe combination should become an operational possibility. In the first instance the basic design was for more power, more economically produced than that obtaining with this mark of Proteus, which in itself was only a development engine for which no production was envisaged.

The performance test programme was therefore planned to obtain the maximum amount of information on the aerodynamic and hydrodynamic characteristics to provide a basis for future development. There have been inherent problems in analysing the results of these tests associated with the difficulty of determining the power, propeller efficiencies, and slipstream effects. These factors mean that the accuracy of the results may not be as great as could be desired, although sufficient data has been obtained to minimise the effect of scatter so as to give a reasonable picture of the performance characteristics.

2. DESCRIPTION OF AIRCRAFT

A general description of the aircraft, together with a general arrangement drawing and photographs, is given in Reference 1.

3. CONDITION OF AIRCRAFT

3.1. General

The general condition of the aircraft was as given in Reference 1. Apart from the engines themselves, as discussed in Para. 1 above, the airframe was in a representative final form, although there were one or two places where improvements in air leaks and aerodynamic cleanness could possibly be achieved. In particular the leak between the wing-tip and the float in the retracted position, would reduce the end plate effect and therefore the effective aspect ratio.

The control surface gaps were another source of air leaks, and the general aerodynamic cleanness, particularly on the outer wings, deteriorated with time owing to surface contamination.

3.2. Loadings

The all up weights for the performance measurements were between 225,000 and 295,000 lb. for take-offs, and between 250,000 and 300,000 lb. for air performance.

3.3. Design limitations

The design operating limitations in effect for the flight trials are given in Para. 3.3 of Reference 1.

3.4. Instrumentation

One of the three camera recording panels was devoted to the measurement of engine performance quantities and was photographed by an F.24 camera capable of continuous recording at about 24 frames per minute or in single shots. This panel was operated in conjunction with the handling auto-observer, (see Reference 1), for the majority of the performance tests. On take-off tests the handling observer was operated continuously at 4 frames per second to obtain a history of speed, acceleration, and attitude, and the performance

/observer

observer at 5 - 10 second intervals. In flight, under steady level or climbing conditions, the handling observer was normally operated at 10 second intervals to obtain speed and height histories, and the performance observer at 2 minute intervals or every 1000 ft.

Air temperature was obtained from a balanced bridge type of thermometer which was read visually at the master observer's panel. For reference purposes an impact type thermometer bulb was fitted, the indication from which could be selected to be presented on either the handling or performance observers.

The attitude measurements in the air were obtained by visual observation of a Watts Mk.IV drum type clinometer.

A typical record from the performance auto-observer is shown in Figure 1, the recorded quantities being as follows:-

Compressor r.p.m.	10 engine3
Propeller turbine r.p.m.	6 power units
Jet pipe temperature	10 engines
Compressor delivery static pressure	• 10 engines
Fuel flow	6 power units
Compressor air intake temperature	• Engines 1, 3A, and 3B
Compressor delivery temperature	Engines 1, 3A, and 3B
Jet pipe static pressure	Engines 1, 3A, and 3B
Jet pipe dynamic pressure	Engines 1, 3A, and 3B
Burner pressure	Engines 1, 3A, and 3B
Air intake pressure	Power Units 1 and 3
Fuel pressure at flowmeter	Power Units 1 and 3
Torquemeter oil pressure	Engines 1 and 6.
Fuel contents	
Compressor delivery static reference *	
Panel temperature	
Altitude	
Airspeed	
Air temperature	
Time	
Flap position	
Aircraft heading.	

There are two independent airspeed systems on the aircraft, both connected to Mk.8 pitot-static heads fitted to a mast on the hull, (see Figure 2).

The lower pressure head, (No.1), feeds the Captain's and Navigator's instruments, and the upper one, (No.2), the 1st Officer's, Engineer's and Flight Test instruments. A pitot in venturi was fitted adjacent to the pitot static heads, and was connected to the pressure side of the test airspeed indicator for the take-off runs.

4. SCOPE OF TESTS

4.1 Calibrations

4.1.1. A.S.I. and air thermometer calibrations

The test airspeed system static pressure error with flaps and floats retracted was determined by the aneroid method. Runs at various speeds between 130 and 245 knots I.A.S. were made at a mean weight of 250,000 lb. and at a height of 300 Pt. above the sea in order to reduce ground effect. A steady

/level

* In order to avoid overloading the airspeed system, the static side of the differential pressure gauges for compressor delivery pressure were connected to the static source of the powered flying control system feel generator. This is located inside the unpressurised mooring compartment in the nose of the aircraft.

level approach to the datum point was made over the sea for a distance of eight miles to enable a calibration of the air thermometers to be carried out at the same time. An additional check on the balanced bridge thermometer calibration was made in the course of steady speed runs at an altitude of 15,000 ft.

4.1.2. Engine performance calibration

The non-dimensional performance characteristics of the single engines, (the only engines fitted with torquemeters) were determined in the course of level and climbing flight conditions between sea level and 30,000 ft. The purpose of this calibration was to provide a basis for engine power and thrust determination on the coupled engines.

4.2. Performance tests

4.2.1. Air performance

Partial climb measurements were made at an altitude of 10,000 ft. with flaps and floats retracted, at engine compressor r.p.m. settings of 9,000 and 7,000.

Extensive performance level speed and climb measurements were made at heights up to 30,000 ft., and at speeds between 120 and 250 knots I.A.S. in such a way as to provide information for a drag analysis.

A descent was made from 30,000 ft. at a constant I.A.S. of 220 knots in order to investigate Mach number effects.

4.2.2. Lift-curve slope

In the course of performance tests clinometer readings were taken in order to determine the C_L - wing incidence relationship.

4.2.3. Water performance

Take-off performance was measured over a range of weights between 225,000 and 295,000 lb. in short sea conditions up to a 1 ft. chop. The unstick distances were obtained by cine-theodolite tracking, using two F.47 take-off cameras. These were situated at either end of a specially surveyed base line, and were synchronised manually, by means of radio contact, with the auto-observers in the aircraft. Acceleration histories were obtained from the latter records. Outside air temperature and pressure were measured on the aircraft, and the windspeeds were obtained from a hand held anemometer on the standby launch. The water speed was obtained by subtracting windspeed from airspeed.

5. RESULTS OF TESTS

5.1. Calibrations

5.1.1. A.S.I. and air thermometer calibrations

5.1.1.1. Static pressure error correction

The measured values of static pressure error with flaps and floats retracted are presented in Figure 3 as ΔV_R , the correction to be applied to the indicated airspeed, against V_R , the indicated airspeed, for a weight of 250,000 lb. at sea level. The mean correction to be applied is - 3 knots at 130 knots, zero at 166 knots, and + 1 knot at 250 knots.

/The

The A.S.I. total correction, (static pressure error plus the scale altitude term), and the altimeter correction, for speeds between 130 and 250 knots I.A.S., and altitudes up to 30,000 ft. are presented in Figure 4 for an aircraft weight of 250,000 lb.

These corrections have been derived by the method of Reference 2, and apply to No.2 pressure head, (see Figure 2), used for the test instruments.

5.1.1.2. Air thermometer calibration

The results of the air thermometer calibration measurements are shown in Figure 5. The upper of the two figures shows the results obtained at sea level for both the impact bulb and the knife edge bulb of the balanced bridge thermometer. Repeat measurements on the latter at a height of 15,000 ft. are shown in the lower plot.

The sea level value of k (the ratio of indicated temperature rise to isentropic temperature rise), was derived from the calculated best mean line through the points, and for the impact bulb is 0.951, which is somewhat low for this type. The sea level and 15,000 ft. values on the balanced bridge thermometer were found to be 0.673 and 0.701 respectively, the difference between these values constituting a difference in true temperature of less than 0.5% at the maximum airspeed attained during the performance tests.

5.1.2. Engine performance calibration

5.1.2.1. Shaft horse power determination

The measured torquemeter shaft power on the single outboard engines are plotted non-dimensionally in Figure 6. Shaft horse power is defined as:-

$$\frac{\text{power turbine r.p.m.} \times \text{torque meter pressure p.s.i.}}{2209}$$

the torquemeter constant (2209), being derived by the Bristol Aeroplane Co. Ltd. (Engine Division).

The mean line drawn through these points is the basis on which the power of the coupled engines has been determined from measurements of compressor delivery pressure, and this is discussed further in Para 6.1. and Appendix 1.

5.1.2.2. Jet thrust determination

The jet thrusts from all engines have also been derived from the compressor delivery pressure measurements using calibrations based on tailpipe pressure and temperature measurements on one single and one coupled pair of engines. These calibrations are presented in Figures 7 and 8. Further discussion is given in Para.6.1 and Appendix I.

5.2. Performance tests

5.2.1. Air performance

The partial climb performance for c.r.p.m. settings of 9,000 and 7,000 is shown in Figure 9 for I.C.A.N. conditions at 10,000 ft. at an aircraft weight of 250,000 lb.

The measured total air drags are shown in Figure 10 on a $C_D - C_L^2$ basis, and the points are identified in terms of propeller thrust co-efficient, (T_C), values. In order to determine accurate mean lines for $T_C = 0.05$ and 0.10 the slipstream correction of Figure 11, (see below), was applied to correct the measured values to $T_C = 0$, (Figure 12). The single line through these points was then used in conjunction with Figure 11 to derive the mean lines plotted in Figure 10. On the same figure C_D is plotted against Mach No., (from 0.45 to 0.55). for the one C_L value, (0.285), at which measurements were obtained.

The effect of slipstream on drag, calculated from wind tunnel tests, (Reference 3), is presented in Figure 11 as ΔC_D against T_c and C_L^2 . The relationship between these values and the evidence of previous full scale tests, (Reference 4), is discussed in Para. 6.1.

From the corrections of Figure 11 the measured total drags have been corrected to zero slipstream and are shown in Figure 12, where the original T_c values of the measured points are identified. The basic air drag characteristics have been derived from the mean line through those points.

5.2.2. Lift curve slope

The wing incidence measurements are plotted in Figure 13 against C_L , and C_L corrected to zero slipstream at constant incidence, based on the formula given in Reference 4. From the mean line through the points corrected to zero slipstream, the effect of slipstream has been plotted on the measured points, for the T_c values 0.05, 0.10, and 0.15, by a similar procedure to that adopted for the air drag measurements, (Para. 5.2.1).

5.2.3. Water Performance

The measured take-off performance data is presented in Tables 1 and 2. Table 1 gives the basic conditions of the aircraft, the meteorological and sea conditions, hump conditions, and the unstick distance and speed for each measured run. Table 2 gives the longitudinal acceleration values for water speeds from 30 to 80 knots in 10 knot increments, and for the unstick.

The variation of the minimum, (hump), longitudinal acceleration with weight is plotted in Figure 14.

In Figure 15 the take-off distance, corrected to 100 knots water speed by the method of Reference 5, are plotted against all up weight. The effect of hull trim (as defined by fixed elevator angles) on distance is shown in the same figure. These distances corrected to a constant elevator angle of -9 degrees, based on a linear variation of distance with elevator angle from Figure 15, are plotted in Figure 16.

6. DISCUSSION OF RESULTS

6.1. Air performance, drag analysis

Experimental methods of performance reduction would be of no immediate value for this aircraft as in any final form it would not have Proteus 600 series engines installed, as discussed in Para. 1. It has therefore been found more convenient to employ an analytical method of performance reduction in order to obtain data not specifically related to the power plant used.

Analytical methods have their shortcomings, particularly when applied to propeller driven aircraft with calculated, as opposed to measured, propeller efficiency, and, (in the absence of torquemeters), shaft horse power. Due to mechanical limitations torquemeters were fitted only to the single engines of the Princess, so that a direct measurement of only 20% of the total shaft power was available. The method of determining the power of the remaining engines, by measuring the compressor delivery pressure and relating this to the measured characteristics of the single engines, is discussed in Appendix 1.

This method of determining the power and residual jet thrust was quite laborious, especially when considering ten engines, but such cross-checks as have been made suggest that the values obtained were reasonably accurate and consistent. As pointed out in Appendix 1 there was evidence to suggest that the coupled engine power derived from the calibrations were slightly high, and therefore these powers have been reduced by 2% when calculating the thrust.

Propeller propulsive efficiencies, for the single and coupled propellers for all performance measurements, were derived from curves supplied by the manufacturers.

In calculating the values of the total drag coefficient, C_D , allowance has been made for the power required for auxiliary services, including cabin blowers when these were in operation. The net jet thrusts were based on the gross thrust and air mass flow, derived from tailpipe measurements, related to compressor delivery pressure (see Appendix 1), assuming the engine air intake velocity to be the same as the free stream velocity.

A correction to the full scale measured drags for slipstream effect, (Figure II), has been derived from wind tunnel tests, (Reference 3), and this correction has been compared with the formula derived in Reference 4, which is based on the changes in profile drag, variable non-induced drag, and incidence, at constant C_L . The agreement between the drag increments obtained from this formula and the tunnel tests is excellent at low values of C_L , (less than 0.4). The tunnel figures give a greater increment, by about 25%, at higher C_L values however, which over the range of T_c values covered corresponds to less than 0.001 difference in C_D . The use of these higher corrections can be justified, particularly as the net jet thrusts at high T_c values will in fact be less than the calculated values (the slipstream giving rise to a higher intake velocity than that of the free stream, the momentum drag would then be greater than calculated). (The difference in net jet thrust at a T_c value of 0.10 would be of the order of 20%, which is approximately 2% in total thrust or nearly 0.001 difference in C_D at a C_L of 0.7).

The measured drag coefficient values corrected to zero slipstream plotted in Figure 12, do not vary linearly with C_L^2 over the C_L range, which suggests that there may be a change in C_{DZ} with incidence. A linear relationship does however hold good up to cruising C_L values, (about 0.65), the equation to the curve being -

$$\begin{aligned} C_{DX} &= 0.0179 + 0.039C_L^2 \\ &= 0.0179 + \frac{1.12}{\pi A} C_L^2 \end{aligned}$$

(where A , the aspect ratio, is taken as 9.18, based on a float end plate effectiveness of 5%, as opposed to the anticipated value of 10% giving an aspect ratio of 9.62, which is not achieved owing to the leaks at the wing tip to float junction).

The basic drag and lifting efficiency values are therefore:-

$$\begin{aligned} C_{DZ} &= 0.0179 \\ D_{100} &= 1070 \text{ lb.} \\ K &= 1.12 \\ (L/D)_{\text{Max.}} &= 18.9 \end{aligned}$$

When the basic drag curve without slipstream is corrected to values of $T_c = 0.05$ and 0.10 the separation of the resulting curves shows good agreement with the separation of the measured total drags at the appropriate T_c values, (see Figure 10), thus indicating that the slipstream corrections used are of the right order.

The equations to each of these curves for $C_L < 0.65$ are as follows:-

/ T_c

-10-

$$\begin{aligned}
 T_o &= 0.05, \quad C_D = 0.0188 + 0.0402 C_L^2 \\
 &= 0.0188 + \frac{1.16}{\pi A} C_L^2, \\
 T_o &= 0.10, \quad C_D = 0.0197 + 0.0432 C_L^2 \\
 &= 0.0197 + \frac{1.25}{\pi A} C_L^2.
 \end{aligned}$$

(with A = 9.18).

The scatter of the points is somewhat greater than could be desired, although it is not excessive when taken in relation to the method of paver analysis and the number of engines involved. The degree of accuracy of these points is probably of the order of $\pm 5\%$.

The few measurements made during descent at constant C_L , while varying Mach No., (Figure 10), are inconclusive. Steady conditions were not attained until the Mach No. had reduced to 0.55 and there is insufficient evidence to say whether the higher value of C_D at this Mach No. is true or just due to scatter.

6.2. Lift curve slope

The measured relationship between lift coefficient and wing incidence has been corrected to zero slipstream by the method given in Reference 4. The correction has been applied to the measured C_L values at constant wing incidence, the ratio (C_L corrected to zero slipstream, /% measured), being taken as $1 - (1.105 T_o)$. Both the corrected and uncorrected values are shown in Figure 13, and the slope of the mean line through the corrected points gives a value of $dC_L/d\alpha = 0.093$. Extrapolating this line to $C_L = 0$ gives a no-lift angle of -1.4 degrees.

The equation for wing incidence without slipstream becomes therefore,

$$\alpha_{\text{wing}} = -1.4 + 10.7 C_L, \text{ degrees.}$$

When this mean line is corrected to given T_o values as for the drag measurements, (Para. 6.1), the separation of the resulting lines shows good agreement with the separation of the measured results, (see values on Figure 13), and confirm the order of the calculated slipstream effect. For the same no-lift angle, (-1.4 degrees), the slope for T_o values of 0.05, 0.10, and 0.15, is 0.098, 0.104, and 0.111 respectively.

The equation to wing incidence with slipstream becomes:

$$\alpha_{\text{wing}} = -1.4 + (10.7 - 11.8 T_o) C_L, \text{ degrees.}$$

6.3. Water performance

6.3.1. General

The important criterion for the take-off case of a flying boat is that there should be adequate thrust available to accelerate the aircraft through the maximum water drag, (hump), condition.

The water performance of this aircraft has been measured, in terms of longitudinal accelerations, to investigate the effect of all up weight, and to obtain a measure of the maximum value of R/W, (i.e. air plus water drag/all up weight). These measurements have been made in conjunction with records to determine take-off unstick distances and the results are discussed

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in the following paragraphs. The distances have been analysed mainly for comparative purposes to investigate the effect of attitude and all up weight, as, with the derated engines the take-off distances as such are of no real significance.

6.3.2. Longitudinal accelerations

The minimum hump longitudinal accelerations recorded on each measured take-off run are listed with the take-off data in Table 1, and are plotted against all up weight in Figure 14. Whilst the acceleration-weight relationship is not necessarily linear, the best interpretation of the measurements over the weight range covered is a linear one which has been derived by the method of least squares. The slope of this line $da/dw = -1.058 \times 10^{-6}$, where $a =$ longitudinal acceleration, 'g' units, and $W =$ aircraft all up weight, lb. The equation for the minimum acceleration, is $a_{min} = 0.3753 - W (1.058 \times 10^{-6})$, the respective values at 250,000 and 295,000 lb. then being 0.114 'g' and 0.063 'g'.

It has not been possible to analyse these measurements individually in term of drag, owing to the inability to make an accurate determination of engine power from measurements of compressor delivery pressure under take-off power conditions, (see Appendix 1). From the mean engine operation conditions a power value has however been derived from the single engine torque meter values which, when associated with the propeller performance curves supplied by the manufacturers, gives a total thrust of 72,600 lb. at the mean hump speed of 50 knots, under I.C.A.N. standard conditions. Using this figure, and the acceleration as defined by the equation above, a total drag value has been obtained which, expressed non-dimensionally, as the ratio of total drag to all up weight (R/W) is 0.177 at 250,000 lb. and 0.183 at 295,000 lb. These figures are only approximate but do give an indication of the order of the maximum R/W value.

A similar analysis, to that for the hump speed, can be applied to the acceleration measurements at other water speeds, which are presented in Table 2. The minimum acceleration is the important factor, and these results have been correlated with estimated data by Saunders-Roe Ltd. in order to provide a basis for future development. The fact that it has not been possible to measure thrust values or waterborne load, (a fact which is not peculiar to this aircraft, but may be applicable to full scale tests on any flying boat, particularly where development as opposed to research trials are in progress), has precluded any detailed water performance analysis being made at this stage.

Some of the water performance data was obtained on take-offs made primarily for handling purposes at various hull attitudes as defined by fixed elevator settings. To determine whether the changes in attitude had any measurable effect on the performance, the hump acceleration measurements were corrected to a standard weight in two groups and plotted against attitude. The weight correction was derived from the slope of the acceleration-weight relationship.

These results have been plotted against elevator angle, with the associated hull attitudes obtained from the faired take-off trim measurements presented in Reference 1. The indications are that at the lower weight, 240,000 lb. there is an increase in acceleration with a reduction in hull attitude from 9 to 8 degrees, (reducing elevator angle from 11 degrees to zero), any further reduction in attitude having no effect. At the higher weight, 285,000 lb., there is no detectable change in acceleration with attitude over the range covered. This difference between the two weights is to be expected, the effect of attitude changes decreasing with increasing load on water. At higher speeds, when the load on water decreases, there is in fact an increasing effect of attitude on acceleration at 285,000 lb., which however, is at all times less than that at 240,000 lb. which remains virtually

the same as for the hump condition. The scatter of the results for both weights, particularly at the higher speeds, is such that it is not possible to determine quantitatively the precise effect. This is discussed further in the following paragraph in relation to the effect of elevator setting on the take-off distance.

A few measurements were made of the take-off acceleration under crosswind conditions. The results obtained were not conclusive but did indicate a decrease in hump acceleration which, as would be expected, was proportional to the loss in thrust due to throttling back of the downwind outboard single engine.

6.3.3. Take-off distances

The measured take-off data is summarised in Table 1, the measured unstick distances having been corrected to a standard speed of 100 knots in zero wind by the expression

$$\frac{Z_s}{Z_a} = \left\{ \frac{U_{ts}}{U_{ta}} \right\}^2 \quad \text{- Reference 5.}$$

a and s refer to measured and standard conditions respectively. No temperature correction has been applied, as over the range of temperatures encountered, (+ 6 to + 18°C), this effect on engine performance at low forward speeds is inversely proportional to the effect on propeller performance, so that there is no significant change in thrust.

All of the measurements were made at fixed elevator settings, with the result that there was a considerable variation in unstick speed with the elevator settings used. With the exception of the measurements with an elevator angle of + 10 degrees over the hump, which have not been included in this analysis, the elevator settings divide into two groups, with mean angles of -4.5 ± 2.0 degrees, and -9.5 ± 1.5, = 3.5 degrees. The associated mean unstick airspeeds were 110 and 95 knots respectively, and, as no standard take-off technique or safety speed was determined, an arbitrary standard value for all weights, of 100 knots, was assumed for comparative purposes. This assumption does not give a representative optimum unstick distance for all weights, although the unstick speed on test was in fact 100 knots at an all up weight of 295,000 lb. and an elevator angle of approximately - 9 degrees.

When the measured distances, corrected to 100 knots, are plotted against all up weight there is a separation between the values for the two groups of elevator settings, (Figure 15). The smaller elevator angles with the corresponding lower mean hull attitude give the shortest distance, and this is consistent with the indications from the longitudinal acceleration measurements, (Err.6.3.2.). To determine the order of this difference, the low weight, (less than 260,000 lb.), distances were corrected to a standard weight of 240,000 lb., by the method outlined below, and plotted against elevator angle. From the linear relationship existing between these values, the elevator setting was found to increase the distance by 65 ft. per degree upward movement (i.e. increasing hull attitude). The distances appropriate to the mean elevator angles, corrected to higher weights, (lines in Figure 15), indicate an increase in this effect with weight, which is not however borne out by the measured values. Although at the higher weights the results are rather scattered, they suggest that the effect is of the same order as for the lower weights, (the percentage effect therefore decreasing with increase in weight). This is consistent with the indications from the acceleration measurements discussed in Para.6.3.2., namely, that the performance is less sensitive to attitude at the higher weights.

A correction to distances of 65 ft. per degree has therefore been applied to the values at all weights to bring them to a standard elevator angle of - 9 degrees, (Figure 16).

The mean line through these points, showing the variation in distance with all up weight, has been derived by first calculating the mean excess thrust at 240,000 lb. based on a distance of 2,950 ft. The change in excess thrust with weight has been derived from the longitudinal acceleration measurements at 70 knots, (Table 2). (It is assumed, (Reference 5), that mean acceleration corrections are referred to 70% of the Standard speed, which in this case is 70 knots).

The new distance was then calculated from the expression -

$$\frac{Z_s}{Z_a} = \frac{W_s}{W_a} \cdot \frac{F_{ma}}{F_{ms}} \quad (\text{Reference 5})$$

where Z = waterborne distance,

W = aircraft weight,

F_m = mean excess thrust during waterborne run, and suffices a and s refer to measured and standard conditions respectively.

This gives a distance (which is the unstick distance at the maximum weight at which records were obtained, 295,000 lb), of 4750 ft.

This distance is related to a relatively high initial speed, the average taxiing speed at the commencement of each measured run being 15 knots, (Table 1). This taxiing speed was dictated by the nature of the idling power settings required to enable the pilot to open up all engines to take-off power as quickly as possible. The distance quoted therefore would be slightly longer for any reduction in taxiing speed, being about 2% when commencing from zero forward speed. In the derivation of corrections for speed, wind, and all up weight, the error involved in not taking into account the taxiing speed would be about 2% of the correction, the effect on total distance therefore being negligible.

7. CONCLUSIONS

The basic drag and lifting efficiency values compared with previous British flying boats are as follows:-

Aircraft	A	C_{DZ}	$D_{100 \text{ lb.}}$	K	(L/D) _{max.}
Princess	(effective) 9.18	.0179	1070	1.12	18.9
Shetland (Ref.6)	8.61	.0243	760	1.12	15.7
Sunderland II (Ref.7)	7.53	.0309	603	1.14	13.2
Sealand (Ref.8)	9.86	.0345	145	1.15	13.91

The low C_{DZ} and the high (L/D) max. values are considerably better than have been achieved on previous flying boats. Improvements to the wing-tip to float junction, control gap sealing, and general aerodynamic cleanness, would contribute towards an improvement in these already good drag characteristics. The effect of slipstream at cruising C_L values, i.e. values less than 0.65, with $T_0 = 0.05$ is to increase the basic profile drag coefficient and the induced drag factor, i.e.

$$C_{D0} = 0.0188,$$

$$K = 1.16$$

/There

-14-

There are also indications of a change in profile drag (in addition to that due to slipstream) with C_L , which becomes appreciable at C_L values in excess of 0.65.

The no-lift wing incidence was found to be - 1.4 degrees and the variation of C_L with wing incidence at zero slipstream 0.093 per degree, increasing to 0.111 at $T_o = 0.15$. This increase in slope with T_o is the basic reason for the decrease in longitudinal stability, (Reference I), at high power low airspeed conditions.

On the water at the highest weight at which systematic records were obtained, (295,000 lb.), the minimum, (hump), longitudinal acceleration for a medium sea condition, (approx. 1 ft.), was 2.04 ft./sec.², (0.063'g').

Commencing from a taxiing speed of 15 knots, (a speed dictated by engine and propeller operating conditions), the distance to achieve an unstick speed of 100 knots at 295,000 lb. in zero wind I.C.A.N. standard conditions, with a fixed elevator setting of - 9 degrees, was 4750 ft.

No reliable measurements of engine power could be obtained during take-off, thus precluding any water drag analysis. However, using the engine mean output on the single engines obtained from the torque meters, the propeller thrust data was obtained from the manufacturers performance curves. From the total thrust so obtained, together with the mean hump longitudinal acceleration, the approximate values for maximum total resistance at 250,000 lb., and 295,000 lb., were calculated to be 0.179W and 0.183 W respectively.

8. ACKNOWLEDGEMENTS

Acknowledgements are made to Saunders-Roe Ltd. for preparing this report which was written by Mr G.F. Chalmers of the Flight Test Department.

/LIST OF SYMBOLS

LIST OF SYMBOLS

- A = Aspect ratio, b^2/S .
- a = Longitudinal acceleration, 'g' units.
- b = wing span, - ft.
- C_D = Drag coefficient of aircraft, with slipstream.
- C_{D0} = Drag coefficient at zero lift, with slipstream.
- C_{Dx} = Drag coefficient of aircraft, without slipstream.
- C_{DZ} = Drag coefficient at zero lift, without slipstream.
- C_L = Lift coefficient, $W \cos \gamma / \frac{1}{2} \rho v^2 S$.
- d = Propeller diameter, ft.
- D = Air drag, lb.
- D_{100} = Air drag, lb. at 100 ft./sec. at sea level corresponding to C_{DZ} (i.e. without slipstream).
- F_m = Mean excess thrust, lb.
- g = Acceleration due to gravity, 32.2 ft./sec.²
- K = Induced drag factor = $\pi A (d C_D / d C_L^2)$.
- k = Ratio of indicated temperature rise to isentropic temperature rise.
- L = Lift, lb.
- M = Mach number.
- P_1 = Compressor air intake total pressure, p.s.i.
- P_2 = Compressor delivery static pressure, p.s.i.
- R = Air plus water resistance, lb.
- S = Wing area, ft.².
- T_c = Propeller thrust coefficient = Thrust per coupled unit / $2.85 C_p v^2 d^2$.
- T_1 = Compressor air intake total temperature, degrees K.
- t = Ambient air temperature, degrees K.
- U_t = Unstick water speed, knots.
- V = True air speed, knots.
- V_i = Equivalent to air speed, $V \sqrt{\sigma}$, knots.
- V_R = Indicated air speed, knots.

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LIST OF SYMBOLS (CONTD)

W	=	Aircraft all up weight, lb.
W _e	=	Engine air mass flow, lb./sec.
X _G	=	Gross jet thrust, lb.
Z	=	Unstick distance, ft.
θ	=	Relative air temperature, t/288.
σ	=	Relative air density, ρ/ρ ₀
ρ	=	Air density, sea level value (ρ ₀) = .00238 slugs/ft. ³
α wing	=	Wing incidence to flight path, degrees.
γ	=	Flight path angle to horizontal, degrees.

/LIST OF REFERENCES

-17-

LIST OF REFERENCES

<u>No.</u>	<u>Author</u>	<u>Title</u>
1		Saunders-Roe Princess Flying Boat G-ALUN. Air and water handling tests. C.P. 257. January, 1955.
2	A.K. Weaver	The calibration of airspeed and altimeter system. M.A.E.E. Report No. Res/244. A.R.C. 12,564. August, 1949.
3	A.S. Worrall	Wind tunnel tests on a six-engined flying boat. (Saunders-Roe 10/46) with slipstream. R.A.E. Report No. Aero 2226 and Addenda.
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5	J.A. Hamilton	Methods for reducing seaplane take-off distances to standard conditions. M.A.E.E. Report No. F/Res/245. C.P. 219. April, 1954.
6	T.M. Chalmers	Work on the air drag of the Shetland D.X. 166. M.A.E.E. Report No. F/Res/198. A.R.C. 9674. October, 1946.
7	P.E. Naylor	Note on the air lift and drag of Sunderland. M.A.E.E. Report No. H/Res/181. A.R.C. 8243.
8	D.M. Ridland	Some notes on the performance of Sealand amphibian Mk.1. G-AKLN. M.A.E.E. Report No. F/Res/231. A.R.C. 15,985. January, 1953.

APPENDIX I

APPENDIX I

Engine Performance Analysis

1. INTRODUCTION

The investigation into the engine performance has been dictated by the need to determine the total thrust produced by the ten engines. The problem has been considerably aggravated by the fact that torquemeters were fitted only to the two single engines. Mechanical limitations in the coupling and contra-rotating gearboxes precluded the fitting of the existing type of torquemeter to the coupled power units, which meant that a direct measurement was available of only 20% of the total shaft horse power.

However, by carrying out measurements in the air of the shaft horse power output of the single engines, and relating this to compressor delivery pressure, a calibration has been derived which has been used as a basis for the derivation of power of the coupled units. A similar method has been applied to determine total jet thrust, jet pipe conditions being measured in only three of the ten tailpipes.

2. DETERMINATION OF SHAFT HORSE-POWER

The operation of each stage of a gas turbine engine may be expressed in the form of non-dimensional parameters which, for any given engine, are intimately related between the various stages, namely, intake, compressor, combustion chamber, turbine and jet pipe. Thus a unique relationship may be said to hold between, for example, conditions across the compressor and the turbine. The conditions across the turbine, i.e. changes in temperature and pressure of the air from entry to exit, coupled with the turbine efficiency, define the power which it is capable of delivering for any given set of conditions. It is not, at present, practicable to measure turbine conditions on normal flight engines, but, by measuring conditions elsewhere in the engine, (e.g. compressor), and relating these to known turbine conditions from the test bed, it is then possible to relate power output to compressor pressures. Going a stage further, if compressor conditions and shaft power are measured simultaneously, these two may be related directly, turbine conditions not being directly relevant.

This forms the basis from which coupled unit powers have been determined for the Protocus engines in the Princess. In the course of the aircraft performance tests simultaneous measurements were made of torquemeter pressures on the single engines, and of compressor delivery pressures on all engines. From these measurements the relationship between non-dimensional compressor delivery pressure ratio p_2/P_1 , (compressor delivery static pressure-air intake total pressure), and non-dimensional shaft power, $SHP/P_1\sqrt{T_1}$ (where T_1 is air intake temperature) is established. *

From this relationship, determined on the single units, the shaft horse-power for the coupled units was obtained using the measurements of p_2/P_1 on these units, and a typical calibration curve is shown in Figure 6.

/This

* In actual fact the true non-dimensional form of the latter is $SHP/A_1P_1\sqrt{T_1}$, where A_1 is a representative cross sectional area of the intake air flow path, which, being constant for any given engine is omitted. Dimensional analysis of the true expression serves to show the non-dimensional nature in this form, viz. Power = ML^2/T^3 , Pressure = M/T^2 , Temperature = L^2/T^2 ,

$$\therefore SHP/A_1P_1\sqrt{T_1} = \frac{ML^2}{T^3} \times \frac{1}{L^2} \times \frac{T^2L}{M} \times \frac{T}{L}$$

of which all the dimensions cancel out.

This relationship is not theoretically quite correct, there being variations due to varying pressure ratio across the engine, and differences in turbine efficiency between engines. Over the range of heights and airspeeds covered, the variation in pressure ratio was relatively small, but even 30 any effect on power was minimised by making check calibrations in conjunction with each series of performance tests. If the calibration differed significantly from previous figures it was used for analysing those particular measurements.

Any effect due to variations in turbine efficiency may be compensated for in some measure when the results are meaned out over ten engines. It could be taken fully into account by using non-dimensional total equivalent horsepower, (i.e. including jet thrust), and a higher degree of accuracy be obtained by eliminating assumptions as to intake pressure and temperature on the units for which this was not measured. This would involve substituting ambient static pressure and jet pipe temperature for these quantities and deriving calibrations for lines of constant free stream pressure ratio. Short of carrying out an extensive engine performance programme on these lines, which it was not possible to do, it is doubtful whether any appreciably higher degree of accuracy of power determination could be achieved.

No serious attempt has been made to relate the measured power to brochure figures as this would have involved a considerable amount of additional analysis work. A few spot values have however been checked, and for one set of measurements, the specific fuel consumption was calculated using the measured fuel flows, the value so derived agreeing closely with brochure figures.

Overall, the powers obtained by this method have been consistent and, together with the jet thrusts, have enabled a reasonable air drag analysis of the aircraft to be made. The exception has been under take-off power conditions on the water and at low altitudes with high air flow through the engines. The compressor delivery static pressures on the coupled engines under these conditions were rather scattered and gave mean pressure ratios higher than the single engines by about 4% for the same corrected engine r.p.m. The powers derived from the calibration curve at the appropriate compressor pressure ratios were up to 15% higher than anticipated, and were considered to be unreliable. Whilst the effect decreased rapidly with reduction in r.p.m. an investigation suggested that the coupled engine compressor pressures, and therefore the powers, were still slightly high under cruising power conditions, possibly due to small discrepancies in the measurement of intake pressure. In the light of this all the coupled engine powers were reduced by 2% when calculating the propeller thrusts.

3. DETERMINATION OF JET THRUST

Jet pipe pressures were measured in three tailpipes only, namely on one single and one coupled pair of engines. From these measurements the gross thrust and air mass flow were calculated and plotted non-dimensionally as X_G/P_1 and $\dot{M}_3 \sqrt{T_1/P_1}$ against p_2/P_1 , see Figures 7 and 8.

These calibrations were used in the same manner as that for power, the total thrust being determined using these curves and the compressor delivery static pressure ratio on the remaining engines. There are small errors involved in assuming these relationships, and similar remarks apply as far the power determination regarding the slightly higher compressor pressures on the coupled units. Any discrepancies which might exist have however been ignored, as the jet thrust is only 10% of the total thrust, and an error in Jet thrust determination of as great as 10, would only constitute a 1% error in total thrust.

TABLE 1

Take-off performance, flaps 10 degrees down

Run Ref. NO.	Aircraft Condition				Sea Conditions	Sea Level Not. Conditions			Hump Condition			Unstick		
	A.U.W. lb.	C.G. position S.M.C.	Elevator angle degrees.	Taxying Water Speed knots		Pressure p.s.i.	Temp. °C	Wind Speed Knots	Water Speed Knots	Longitudinal acceleration 'g'	Keel attitude degrees	Water Speed Knots	Uncorrected Distance ft.	Distance corrected to 100 knots and zero win
1	256,000	28.2	- 5.0	A.R.		14.85	+ 6	8	48	.113	9.0	99	N.R.	
2													3030	
3	254,000	28.1	- 5.0	15		14.85	++6.6	6	46	.107	9.0	101	2603	2884
4	250,500	28.0	- a.0	14		14.85	+ 6	6	49	.112	9.1	86	N.R.	-
5	247,500	28.0	- 8.0	14		14.85	+ 6		45	.110	9.0	86	2263	3060
6	245,000	27.9	- 2.5	16	Calm	14.85	+ 6	4	47	.125	8.0	105	3151	2858
7	242,000	27.8	- 3.5	15		14.85	+ 6	3	47	.125	8.8	184	2996	2717
8	239,500	27.0	- 9.0	15		14.85	+ 6	4	48	.126	8.8	83	2269	3216
9	236,000	27.7	- 10.5	18		14.85	+ 6	3	49	.121	8.7		1766	2564
10	234,500	27.6	- 3.5	15		14.85	+ 6	4	46	.123	8.5	109	2898	2439
11	231,500	27.6	- 3.5	15		14.85	+ 6	3	45	.128	9.2	184	2563	2080
12	247,500	28.0	- 9.0	14		14.81	+11	8	52	.108	8.6	83	2060	2920
13	244,000	27.9	- 10.0	15		14.81	+11	8	54	.118	-	81	2221	3224
14	241,000	27.8	- 9.0	17	6 inch	14.81	+11	a	52	.115	8.8	83	1985	3025
15	239,000	27.8	- 10.0	17	chop	14.81	+11.5	8	50	.128	8.2	86	2990	2693
16												83		
18	230,000	27.4	+ 10 x	N.R.	chop	14.81	+12	7	46	.145	7.8	83	N.R.	2784
19	225,000	27.4	+ 10 x	17		14.81	+12	7	48	.150	6.8		1770	2569
20	223,500	27.3	+ 10 x	15		14.81	+13		50	.142	6.9	85	1900	2630

N.R. - no record.

x - This elevator angle was maintained until approximately 65 knots, when it was moved to - 9 degrees for the remainder of the run.

TABLE 1 (CONTD.)

Take-off performance, flaps 10 degrees down

Run Ref No.	Aircraft Configuration				Sea Conditions	Sea Level Met. Conditions			Lump Condition		Unstick				
	Weight lb.	Configuration S.M.C.	Elevator angle degrees	Taxying Water Speed knots		Pressure p.s.i.	Temp °C	Wind Speed Knots	Water speed Knots	Longitudinal acceleration 'g'	Keel Attitude degrees	Water Speed Knots	Uncorrected Distance feet	Distance corrected to 100 knots and zero Wind.	
21	295,000	28.0	- 5.0	15	6 inch chop	14.79	+ 16	5	50	.063	9.5	111	5000	4058	
22	291,000	27.9	- 5.0	13		14.79	+ 16	8	50	.061	9.5	112	5250	4185	
23	287,500	27.8	- 6.5	15		14.79	+ 16	6	50	.066	9.6	101	3580	3510	
24	284,500	27.7	- 5.0	15		14.79	+ 16	5	50	.070	9.0	110	5000	4132	
25	280,500	27.6	- 6.5	14		14.79	+ 15	7	52	.083	9.2	101	4300	4215	
26	277,500	27.5	- a.5	16		14.79	+ 15	6	54	.077	9.0	95	3220	356%	
27	275,000	27.5	- a.5	N.R.		14.79	+ 15	12	48	.090	8.8	84	N.R.	-	
28	272,000	27.4	- 8.5	14		14.79	+ 14	7	50	.085	8.8	89	3140	3964	
29	270,000	27.4	- 11.0	15		14.79	+ 14	9	52	.094	8.4	85	3060	4235	
30	268,500	27.3	- 11.0	13		14.79	+ 14	8	52	.094	8.5	83	2960	4297	
31	267,000	27.3	- 13.0	16	14.79	+ 14	6	50	.098	8.6	87	2850	3765		
32	290,500	27.9	- 9.0	W.R.	9 inch chop	14.87	+ 10	a	48	.073	9.4	92	N.R.	-	
33	287,000	27.8	- 9.0	N.R.		14.87	+ 10	11	50	.079	9.9	88	N.R.	-	
34	284,000	27.7	- 9.5	N.R.		14.87	+ 11	14	46	.075	9.6	84	N.R.	-	
35	280,500	27.6	- 9.5	N.R.		12 inch chop	14.87	+ 11	16	48	.070	10.0	83	N.R.	-
36	277,000	27.5	- 9.5	N.R.		14.87	+ 11	11	50	.088	9.3	82	N.R.	-	
37	295,000	28.0	- 9.0	18		14.75	+ 17	11	52	.066	9.5	87	3320	4386	
38	291,500	27.9	- 9.5	9		14.75	+ 16	12	50	.065	9.1	89	3580	4520	
39	284,000	27.7	- 9.0	10		14.75	+ 16	12	50	.083	9.0	87	3555	4697	
40	295,000	28.0	- 9.5	N.R.		9 inch chop	14.72	+ 18	10	50	.059	9.2	91	N.R.	-
41	291,000	27.9	+ 8.5x	N.R.		to	14.72	+ 18	11	47	.073	a.2	90	N.R.	-
42	288,500	27.8	- 9.0	N.B.	12 inch chop	14.72	+ 18	10	50	.068	9.5	90	N.R.	-	
43	286,500	27.8	+ 9.0x	N.R.	14.72	+ 17	10	50	.074	8.1	92	W.R.	-		

N.R. - No record.

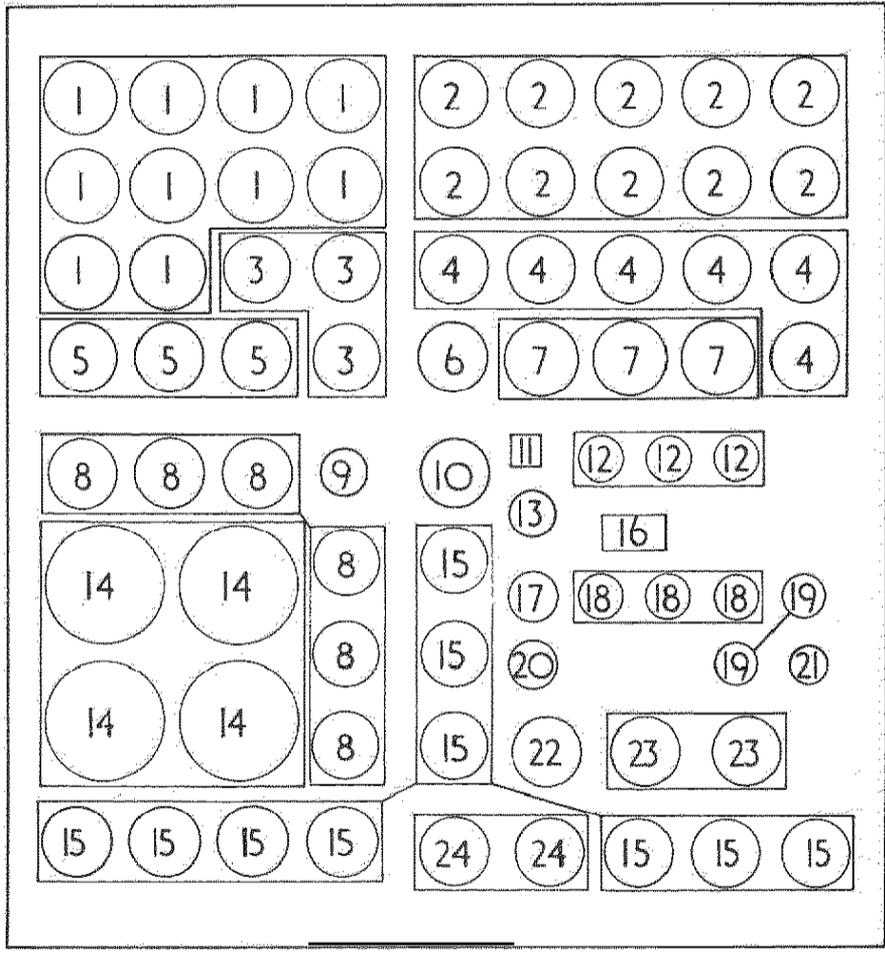
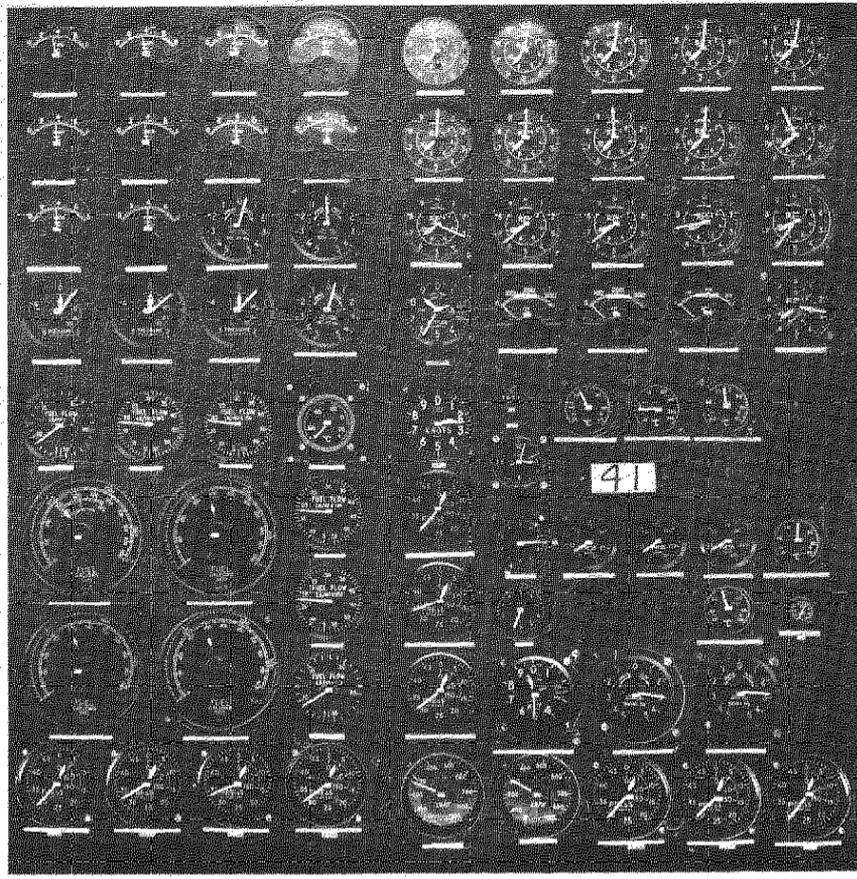
x - This elevator angle was maintained until approximately 65 knots, when it was moved to approximately - 9 degrees for the remainder of the run.

TABLE 2

Take-off Longitudinal Accelerations in
'g' units

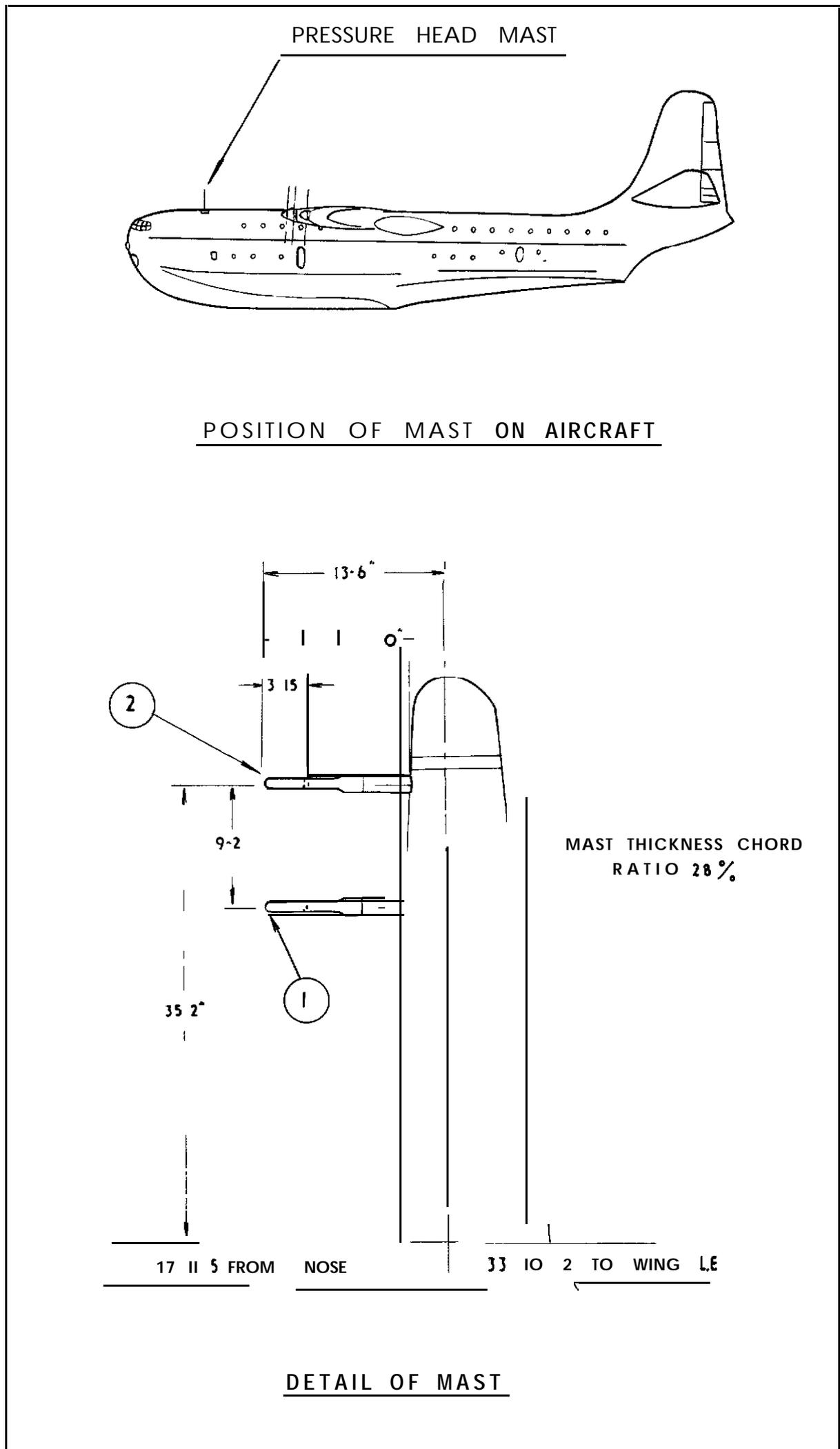
Run Ref. No.	Speed						Instick
	0 Kts.	10 Kts.	50 Kts.	60 Kts.	70 Kts.	80 Kts.	
1	.127	.128	.113	.128	.152	.166	.173
2	.116	.111	.123	.141	.151	.163	.166
3	.125	.123	.128	.150	.143	.156	.188
4	.124	.116	.116	.147	.163	.180	.180
5	.118	.111	.121	.143	.162	.173	.168
6	.133	.125	.141	.164	.168	.167	.160
7	.132	.130	.137	.159	.168	.164	.156
8	.134	.131	.128	.156	.175	.177	.163
9	.131	.123	.121	.154	.171	.172	.169
10	.128	.124	.136	.147	.158	.167	.153
11	.131	.135	.140	.162	.172	.165	.165
12	.098	.110	.108	.126	.135	.143	.141
13	.100	.130	.118	.133	.160	.153	.148
14	.093	.130	.120	.125	.158	.143	.143
15	.098	.118	.113	.133	.158	.158	.155
16	.110	.143	.130	.155	.173	.185	.178
17	.093	.135	.140	.158	.150	.158	.155
18	.108	.141	.150	.178	.175	.168	.168
19	.123	.150	.153	.178	.173	.183	.188
20	.113	.133	.142	.159	.166	.196	.176
21	.091	.072	.063	.071	.106	.106	.123
22	.070	.066	.061	N.R.	.100	.105	.120
23	.084	.071	.066	.078	.107	.104	.129
24	.091	.084	.070	.091	.113	.107	.139
25	.105	.100	.088	.099	.113	.127	.121
26	.103	.100	.081	.093	.112	.123	.131
27	.096	.092	.095	.111	.126	.136	.138
28	.097	.098	.095	.099	.118	.125	.142
29	.085	.105	.094	.112	.123	.132	.134
30	.100	.110	.098	.105	.120	.150	.150
31	.115	.112	.098	.107	.122	.143	.137
32	.081	.080	N.R.	N.R.	.114	.123	.115
33	.098	.086	.079	.108	.120	.133	.130
34	.095	.077	.089	.105	.127	.135	.125
35	.078	.102	.075	.122	.123	.128	.133
36	.089	.095	.088	.117	.131	.144	.140
37	.067	.079	.067	.086	.102	.102	.108
38	.080	.080	.065	.100	.107	.120	.118
39	.0aa	.089	.083	.109	.105	.120	.122
40	.075	.063	.059	.063	.099	.105	.118
41	.088	.078	.078	.084	.102	.106	.109
42	.091	.080	.068	.084	.101	.116	.118
43	.091	.085	.074	.106	.101	.122	.115

N.R. = no record.



- | | | |
|-----------------------------|--|---|
| 1 JET PIPE TEMPERATURE | 9 AIR TEMPERATURE | 17 FLAP POSITION |
| 2 COMPRESSOR R.P.M. | 10 AIRSPEED | 18 BURNER PRESSURE |
| 3 JET PIPE STATIC PRESSURE | 11 1/2 SEC. COUNTER | 19 FUEL TEMP. AT FLOWMETER |
| 4 PROPELLER TURBINE R.P.M. | 12 COMPRESSOR AIR INTAKE TEMP. | 20 AIRCRAFT HEADING |
| 5 JET PIPE DYNAMIC PRESSURE | 13 CLOCK | 21 PANEL TEMP. |
| 6 ALTITUDE | 14 FUEL CONTENTS | 22 COMPRESSOR DELIVERY STATIC REFERENCE |
| 7 COMPRESSOR DELIVERY TEMP. | 15 COMPRESSOR DELIVERY STATIC PRESSURE | 23 AIR INTAKE PRESSURE |
| 8 FUEL FLOW | 16 FLIGHT IDENT. | 24 TORQUEMETER PRESSURE |

PERFORMANCE AUTO OBSERVER PANEL

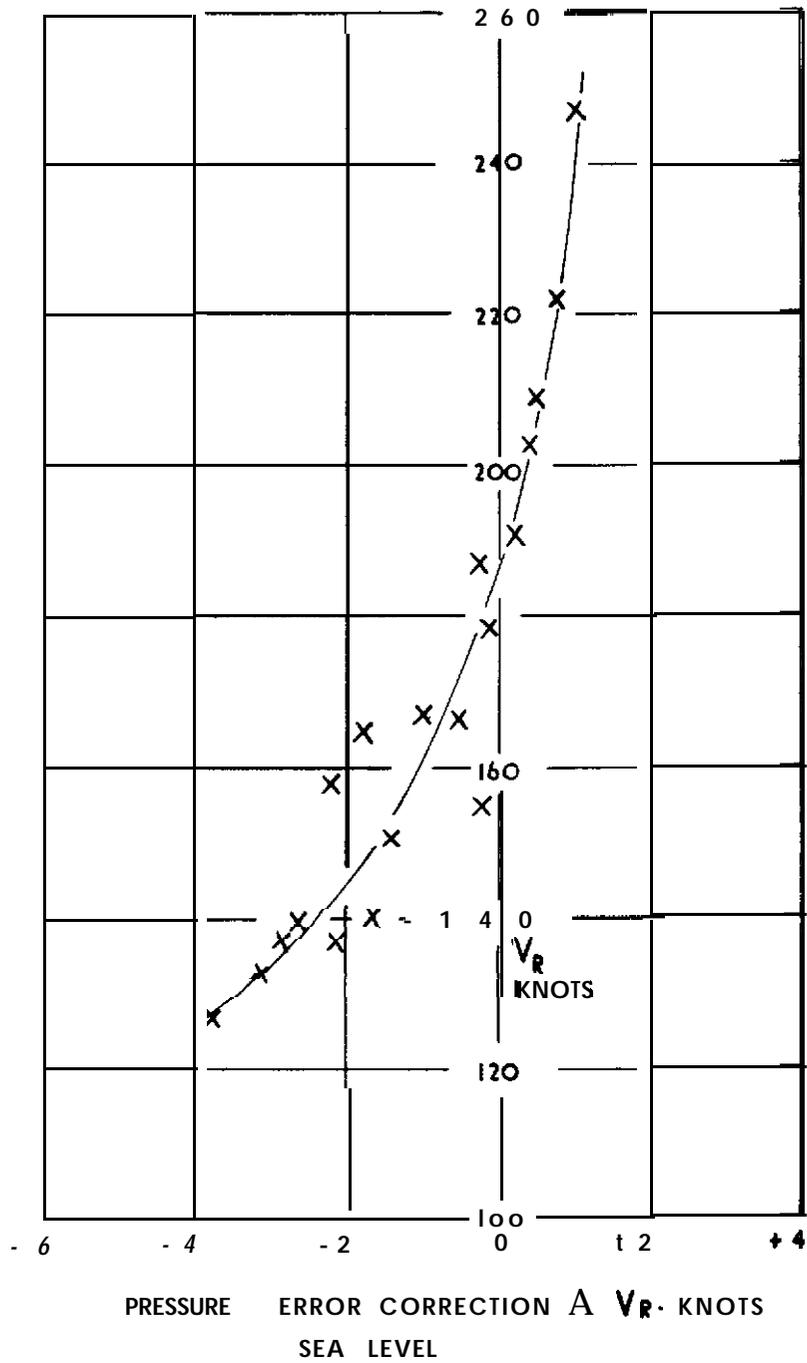


LOCATION OF PRESSURE HEADS

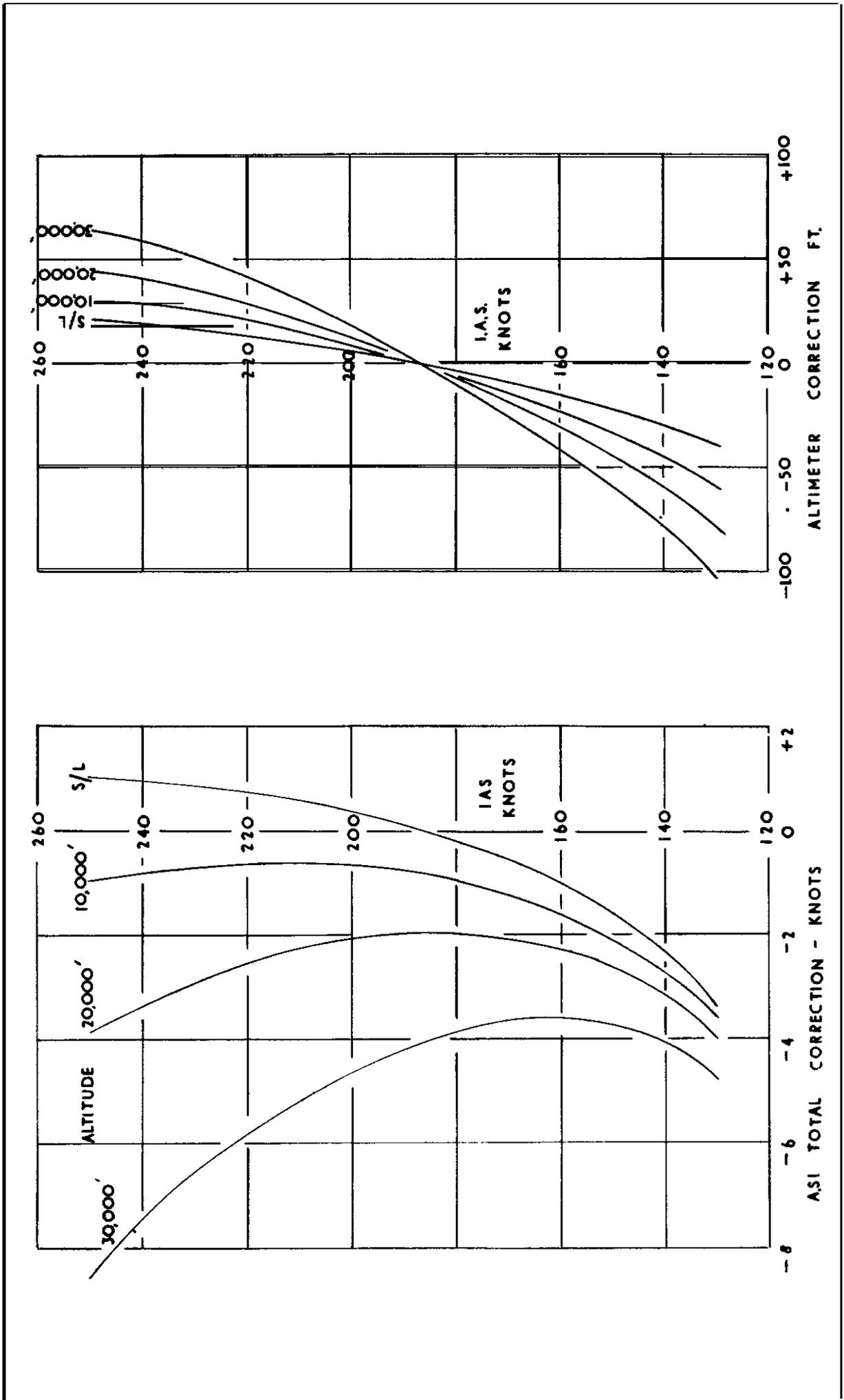
MEASURED BY ANEROID METHOD AT 300 FT.

APPLICABLE TO No2 PITOT STATIC HEAD

CORRECTED ALL-UP WEIGHT - 250,000 LB

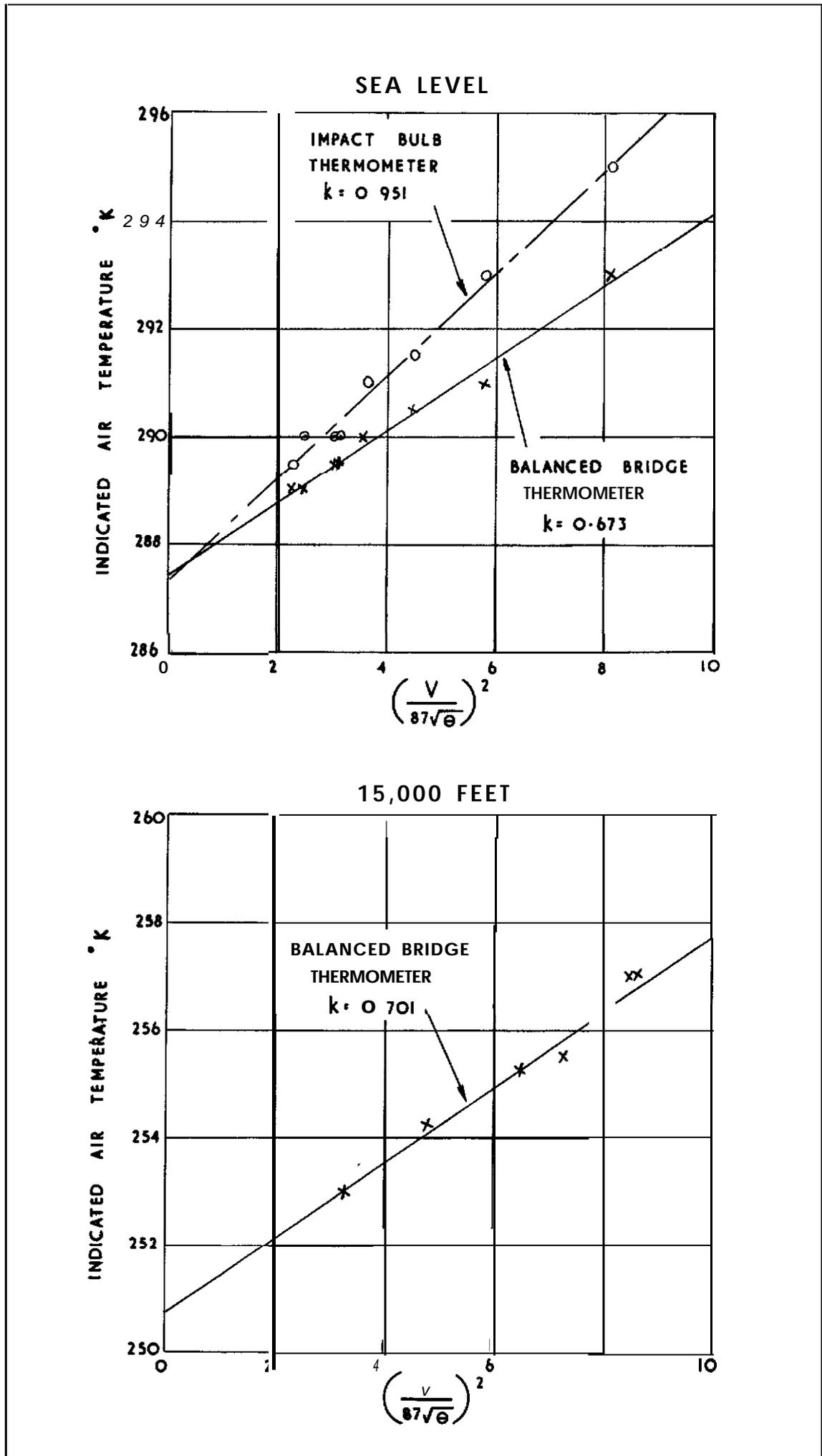


A.S.I. STATIC PRESSURE ERROR CORRECTION
 (FLAPS AND FLOATS RETRACTED)



A.S.I. AND ALTIMETER CORRECTIONS

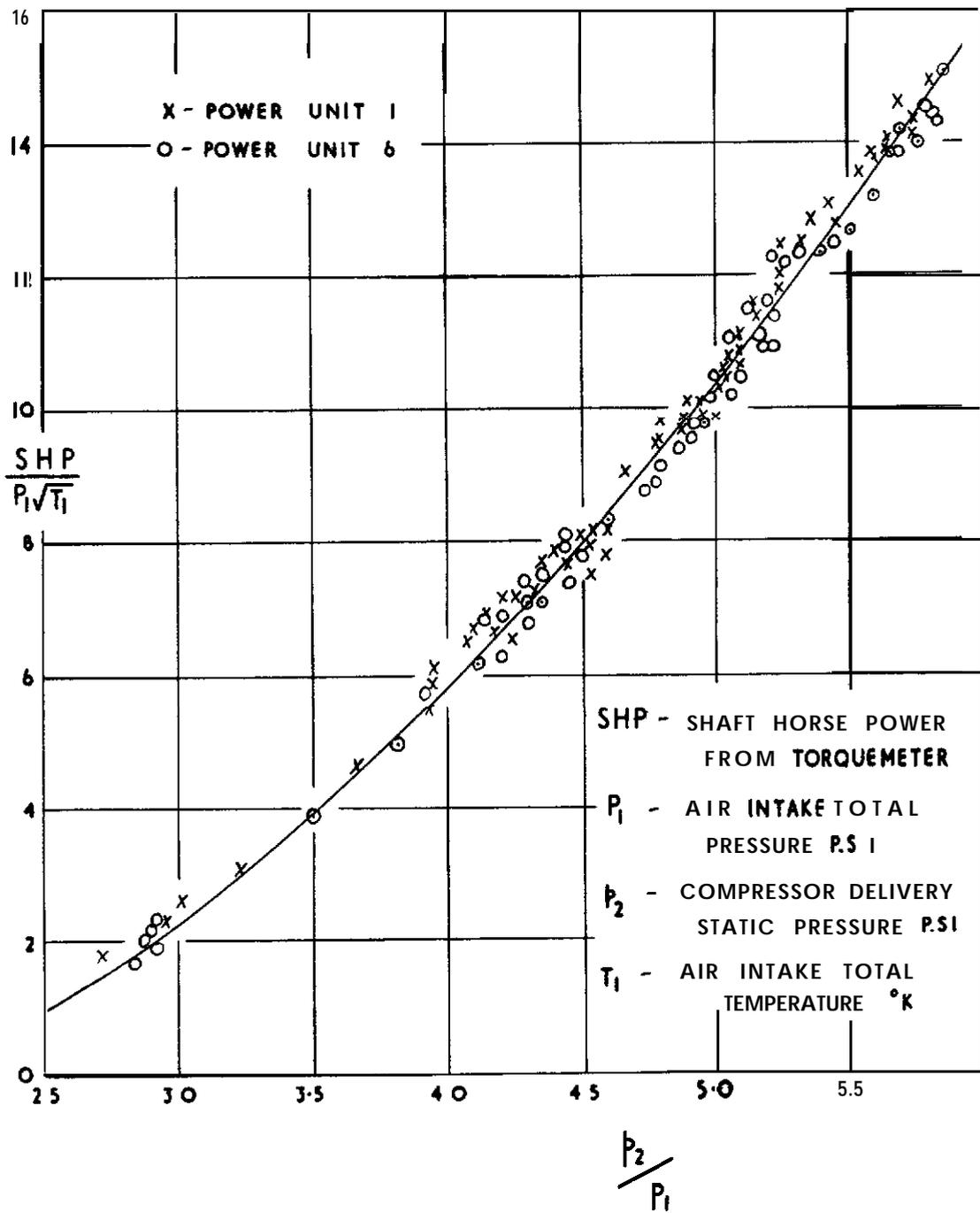
(FLAPS AND FLOATS RETRACTED 250.000 LB.)



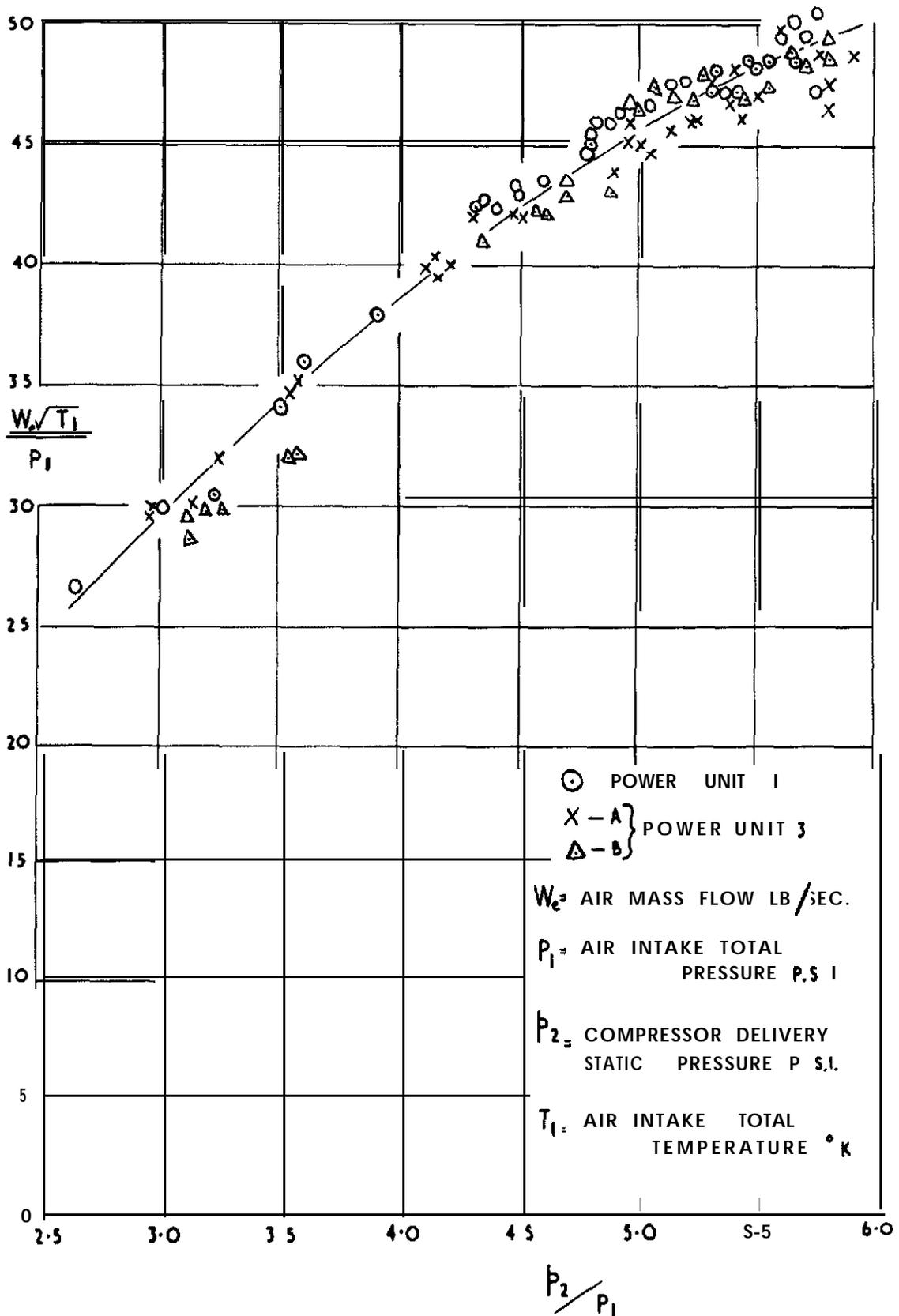
AIR THERMOMETER CALIBRATIONS

MEASUREMENTS AT - 7500 - 9500 COMPRESSOR R.P.M

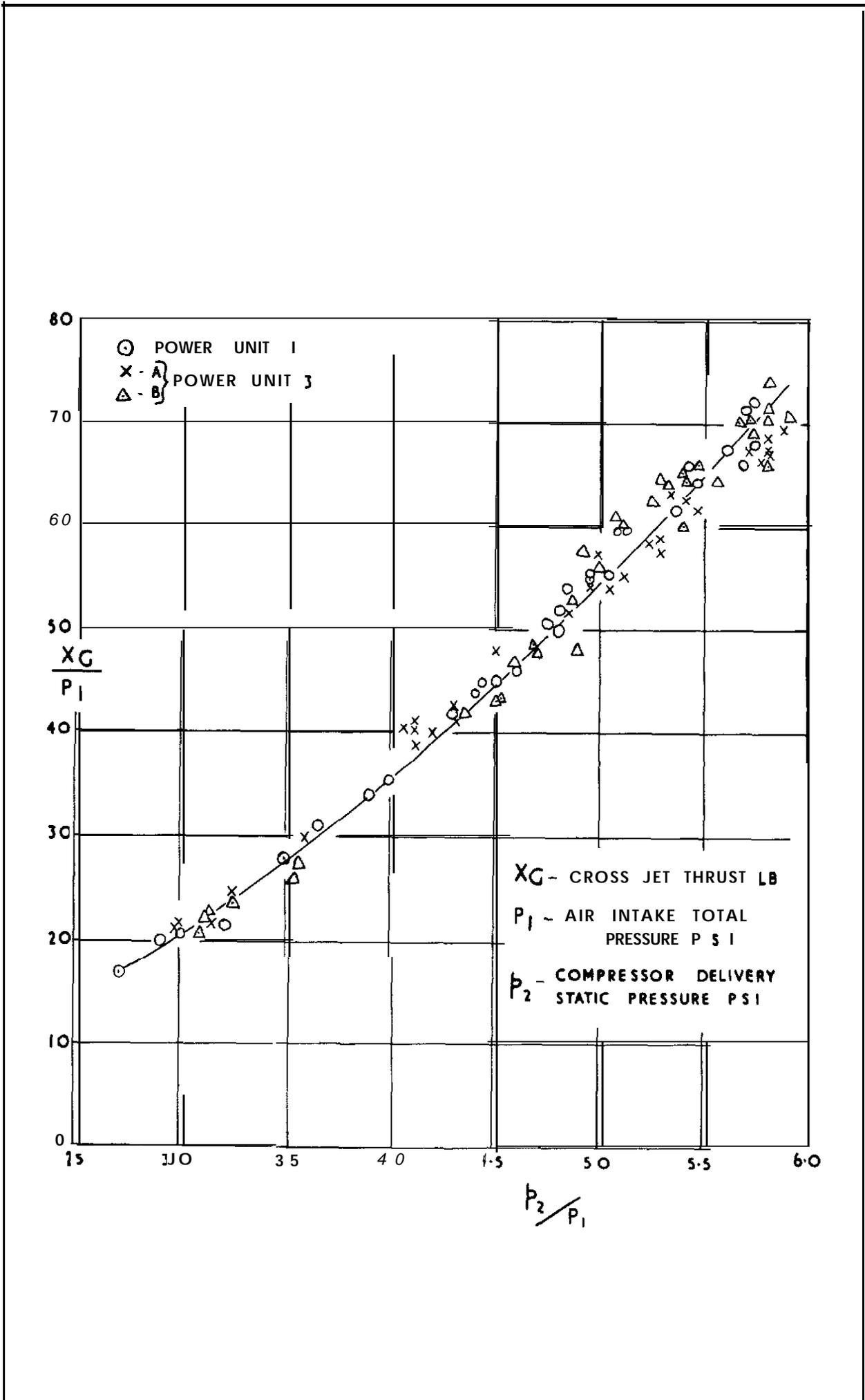
SEA LEVEL - 30,000 FT. ALTITUDE



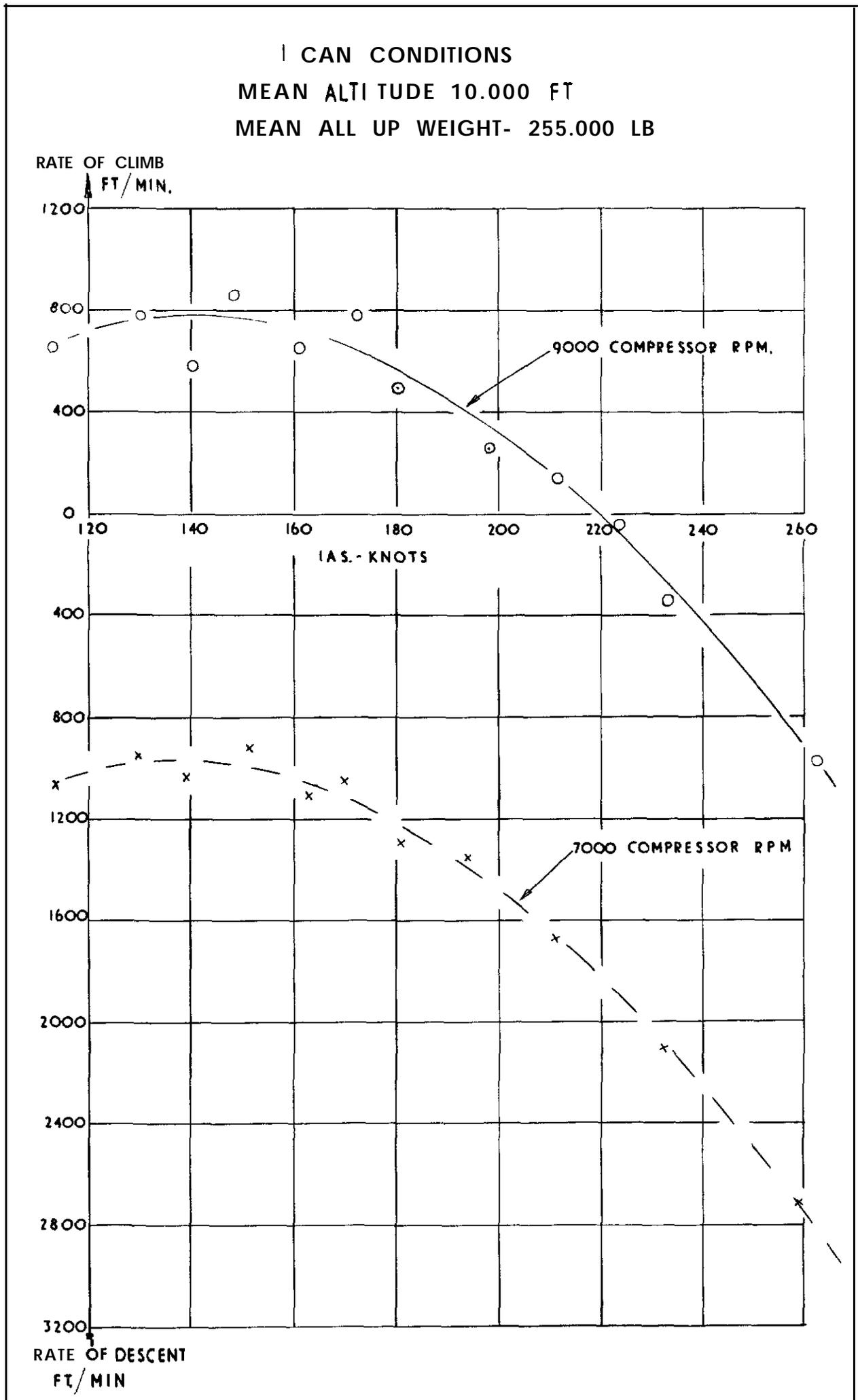
SHAFT HORSE POWER CHARACTERISTICS



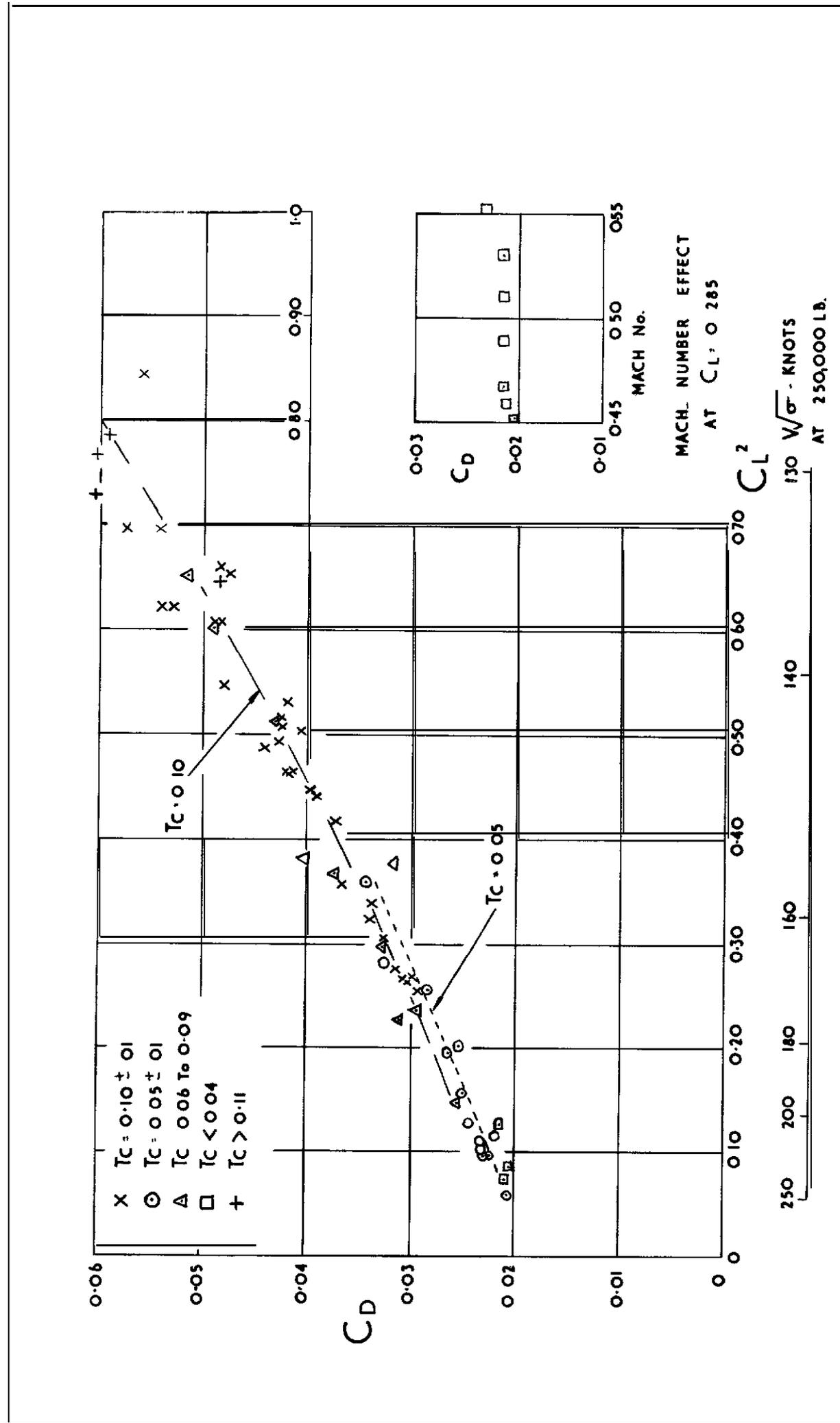
ENGINE AIR FLOW CHARACTERISTICS



GROSS JET THRUST CHARACTERISTICS



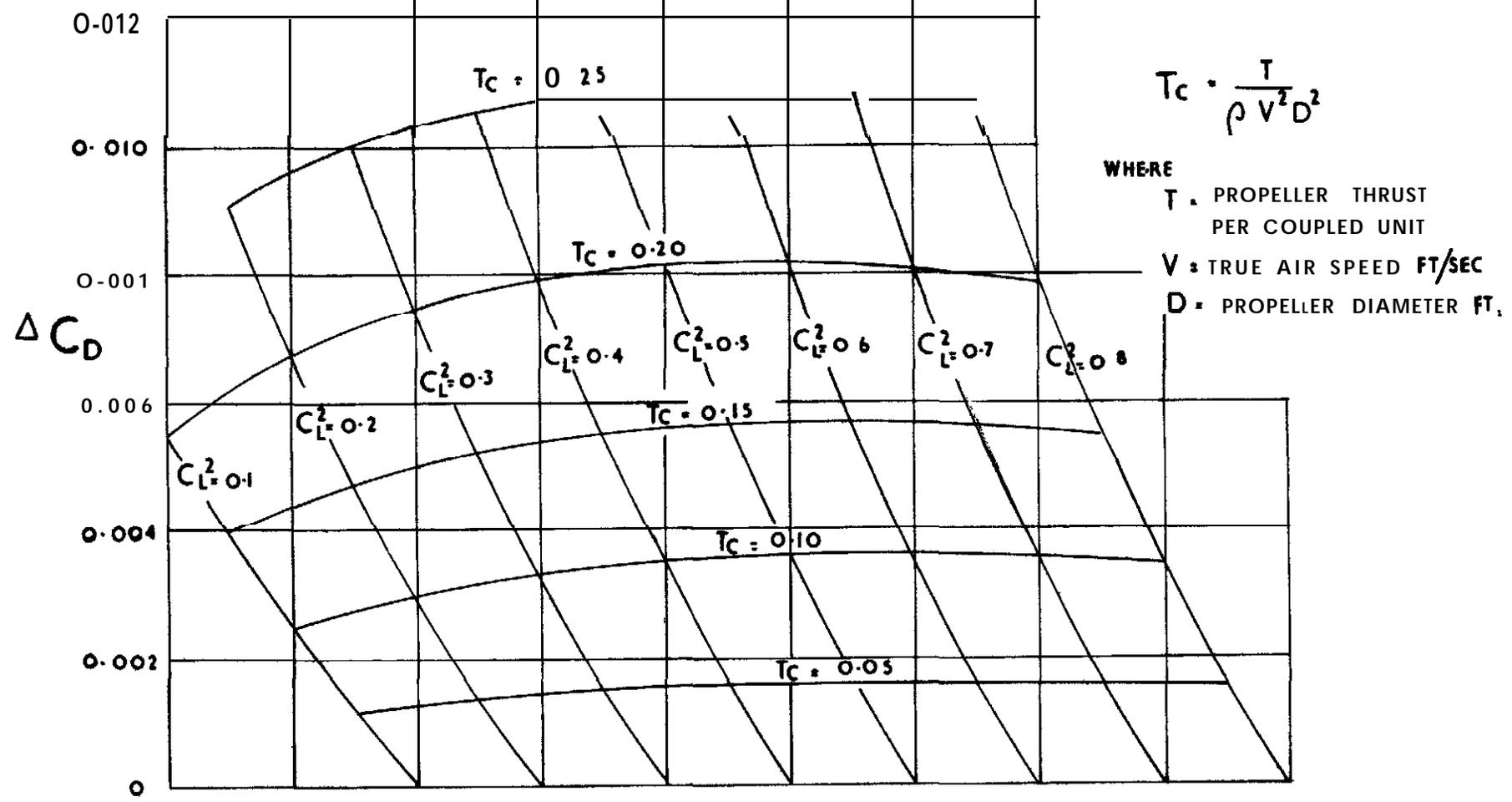
PARTIAL CLIMBS AND DESCENTS
(FLAPS AND FLOATS RETRACTED)

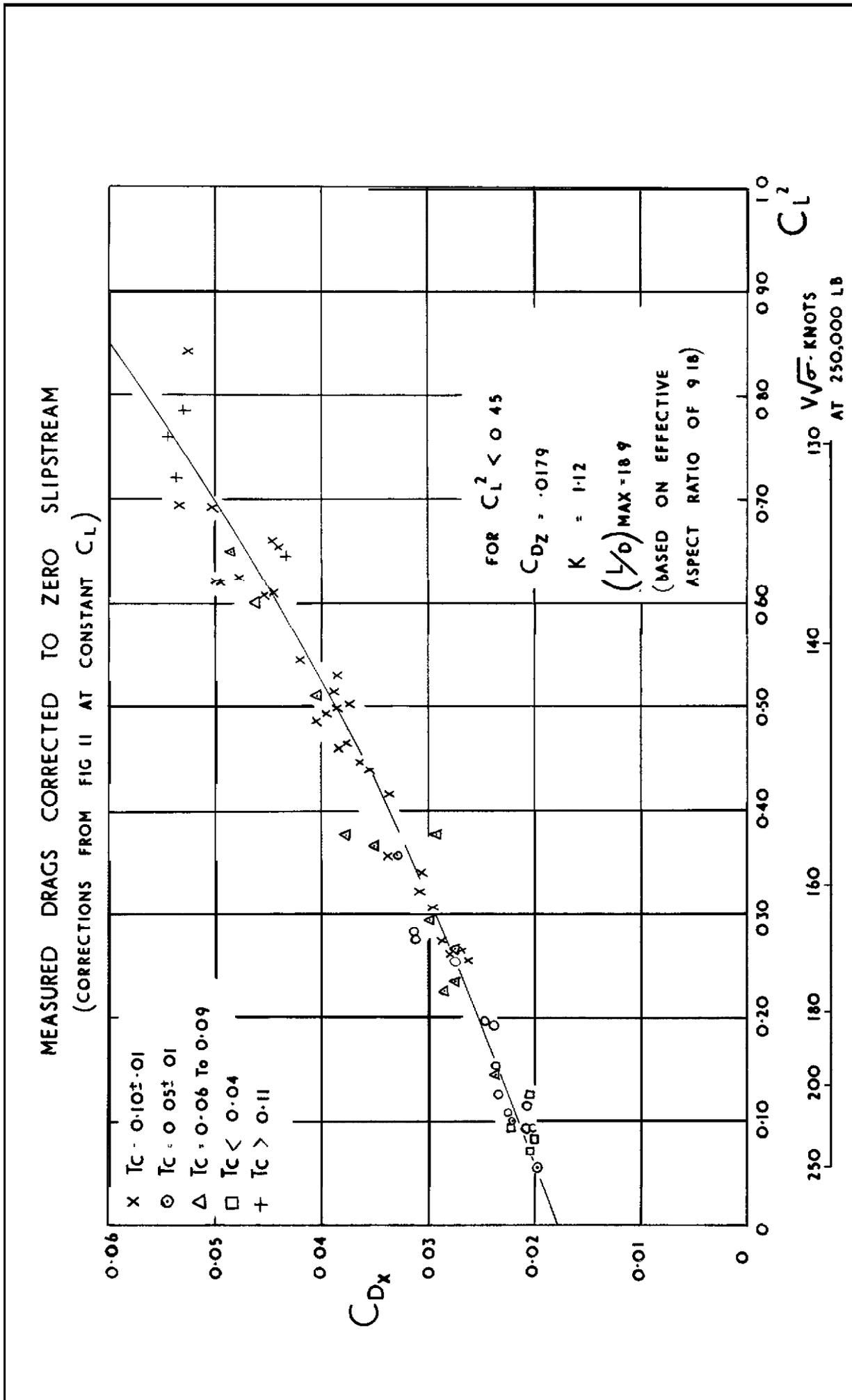


LIFT- DRAG CHARACTERISTICS
 (ZERO FLAP, POWER ON)

DRAG INCREMENT DUE TO SLIPSTREAM

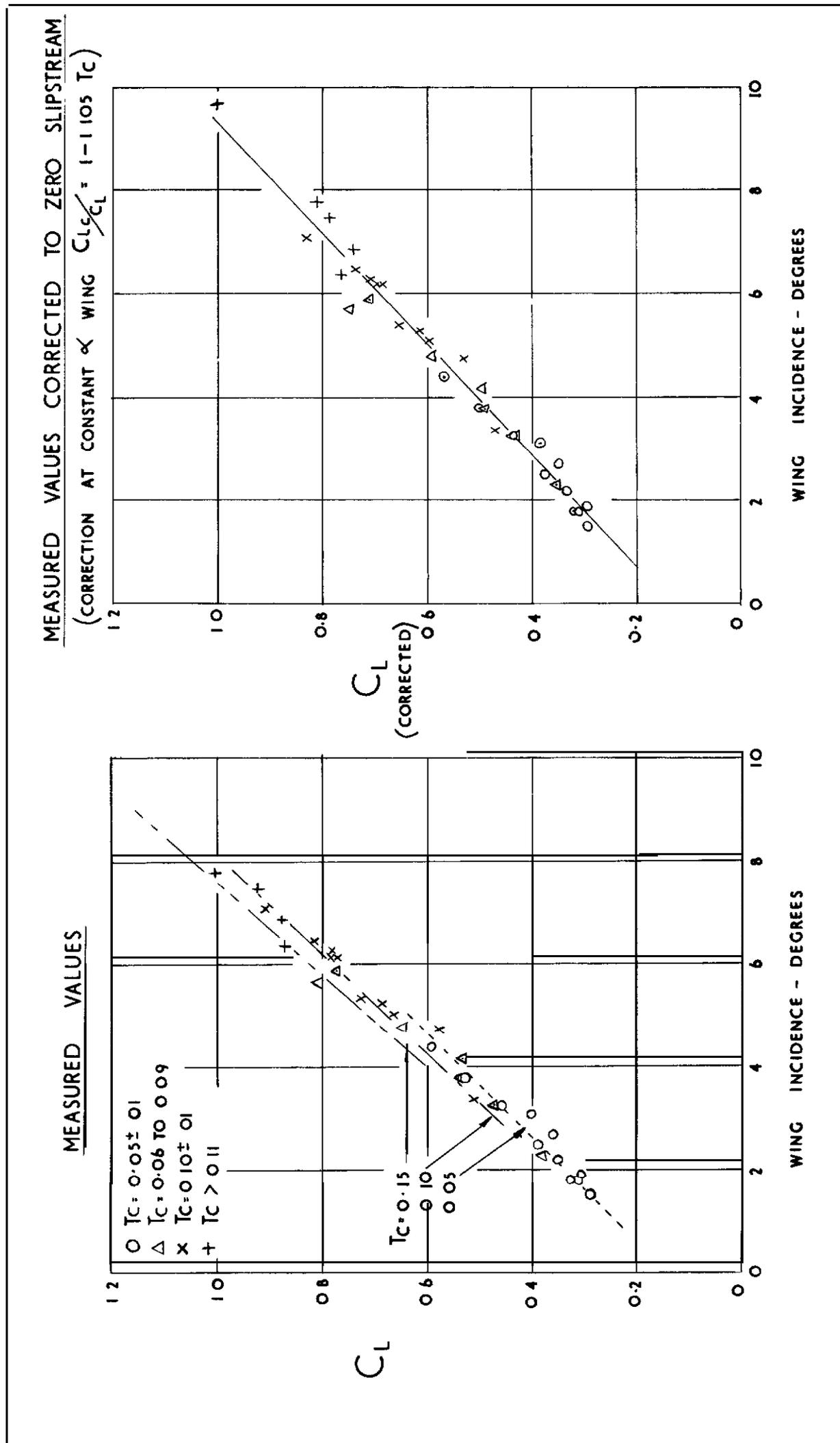
BASED ON WIND TUNNEL TESTS CORRECTED TO FULL SCALE





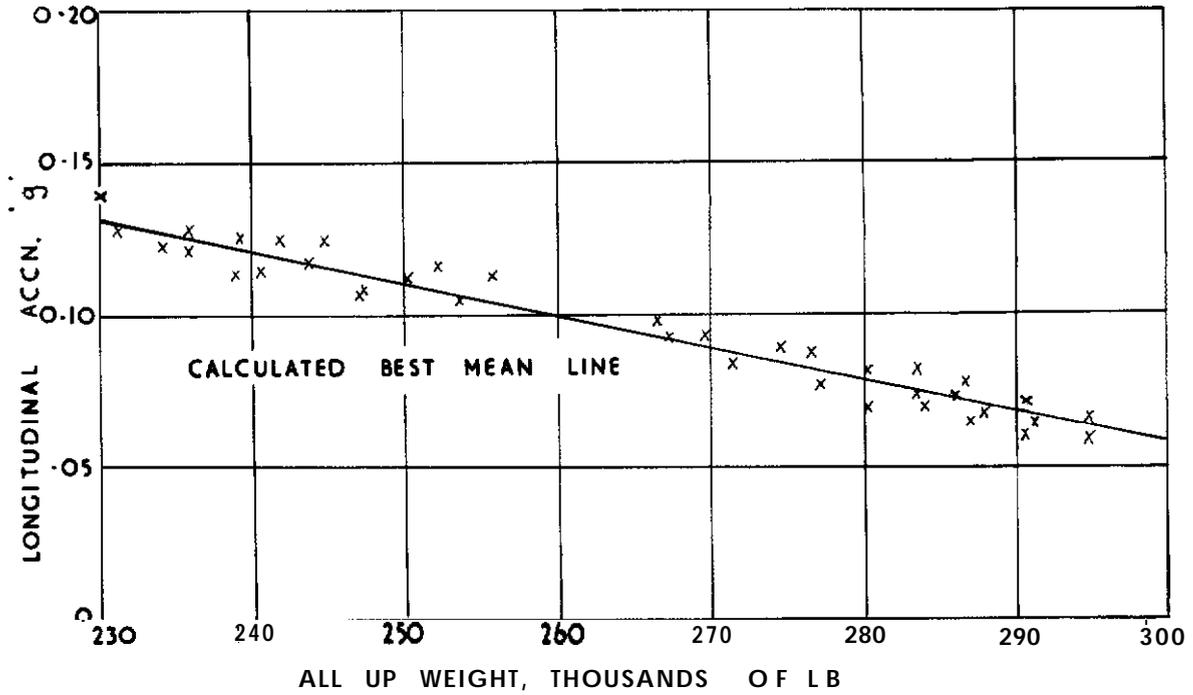
LIFT- DRAG CHARACTERISTICS

(ZERO FLAP, CORRECTED TO ZERO SLIPSTREAM)



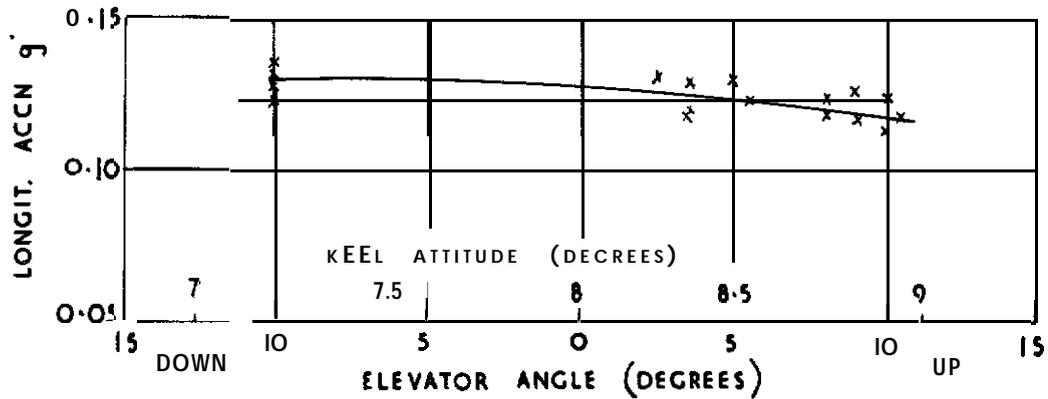
LIFT CHARACTERISTICS
 (FLAPS AND FLOATS RETRACTED)

EFFECT OF WEIGHT ON LONGITUDINAL ACCELERATION

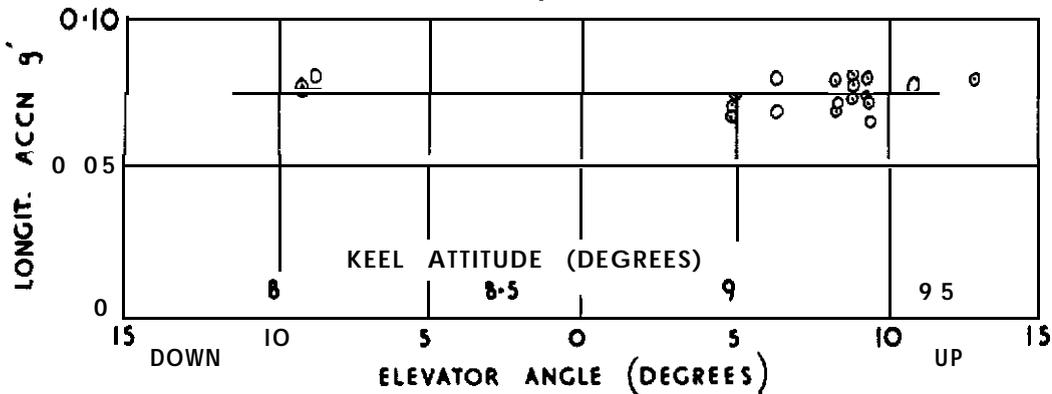


EFFECT OF TRIM ON LONGITUDINAL ACCELERATION

AT 240,000 L B



AT 285,000 LB



LONGITUDINAL ACCELERATION AT HUMP SPEED

EFFECT OF TRIM ON TAKE-OFF DISTANCE

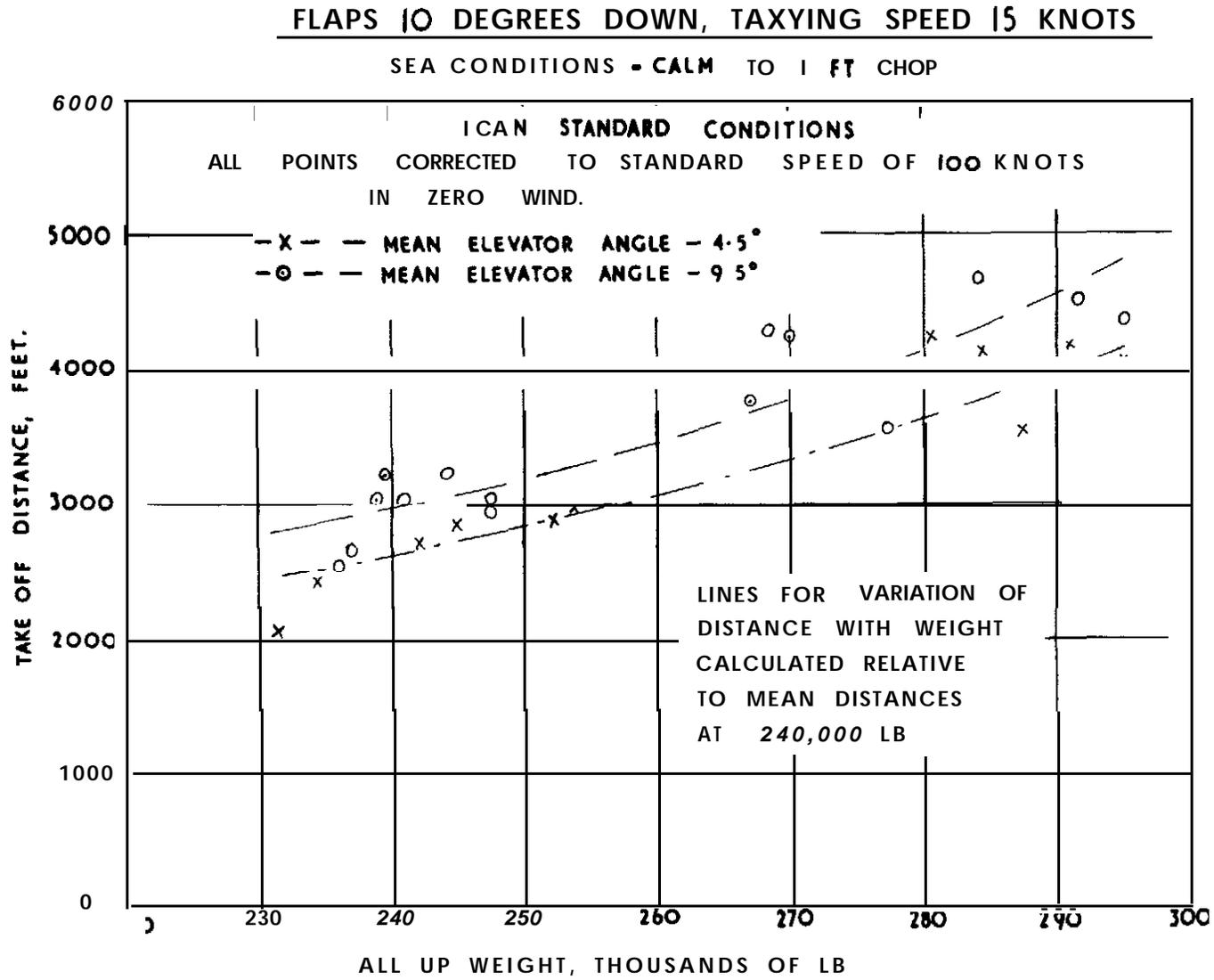


FIG. 15

EFFECT OF WEIGHT ON TAKE-OFF DISTANCE

FLAPS 10 DEGREES DOWN TAXYING SPEED 15 KNOTS

SEA CONDITIONS - CALM TO 1 FT CHOP

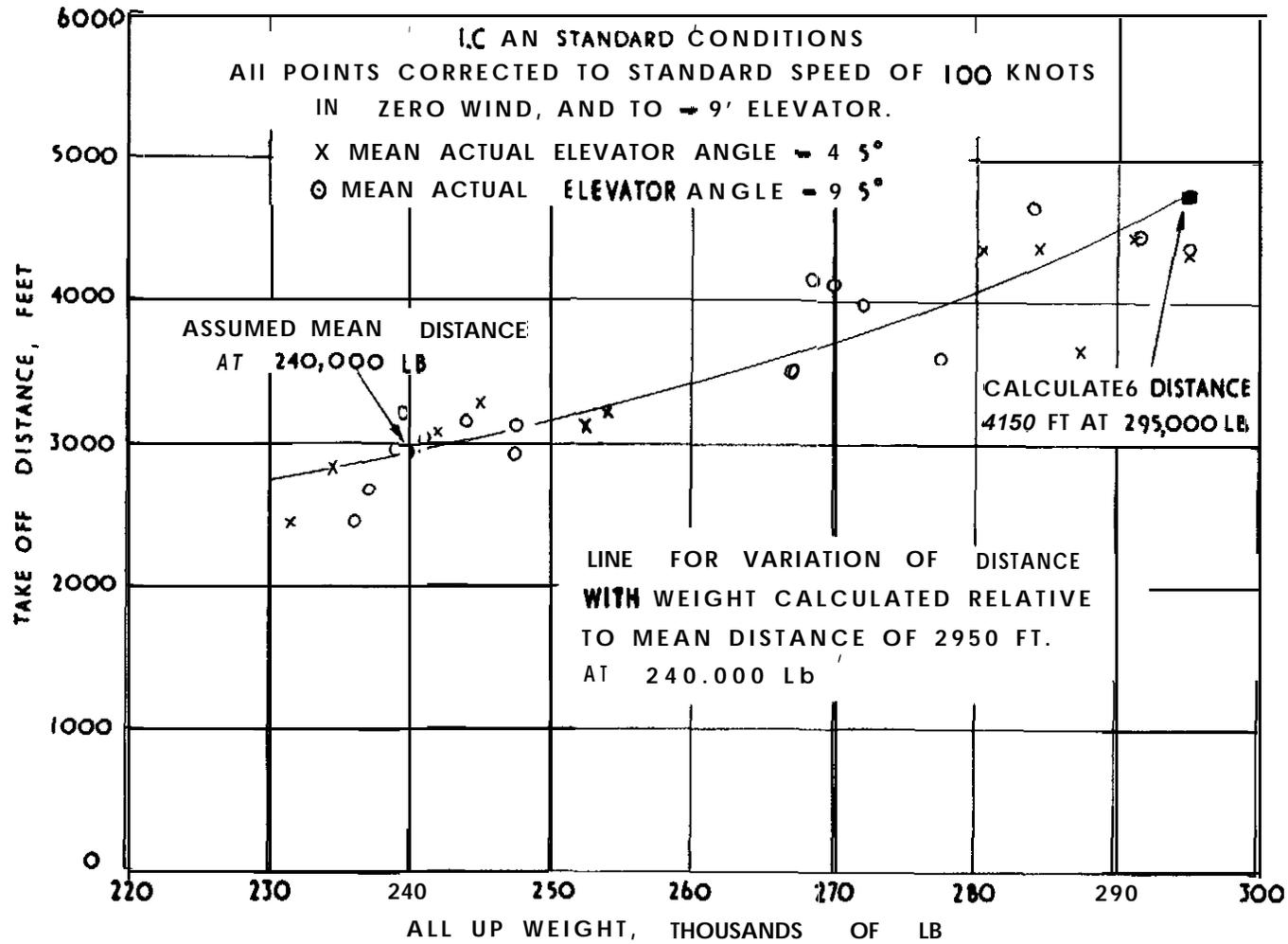


FIG. 16

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