AN INVESTIGATION OF THE FLOW BEHIND A WING WHICH COMPLETELY SPANS THE CLOSED JET OF A WIND TUNNEL

A THESIS

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by

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Preface

Meaning of Symbols Used

V = Velocity, ft./sec.

Cl = Lift coefficient.

Cn = Normal force coefficient.

p = Pressure, lbs./sq.ft.

q = Dynamic pressure, lbs./sq.ft.

t = Chord, ft.

w = Downwash, ft./sec.

b = Span of wing, ft.

AR = Aspect ratio.

Cd = Drag coefficient.

Cc = Chord force coefficient.

α = Angle of attack. (Geometric)

α_i = Induced angle of attack.

Γ = Circulation, $ft.^2$/sec.

S = Wing area.

ρ = Mass density of air.

L = Lift in lbs.

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Summary

An investigation was made to determine the flow about a wing which completely spanned the closed jet of a wind tunnel, and to determine the effective aspect ratio of the wing.

The experiment included three separate steps: (1) calculation of downwash by the slope of the circulation curve, (2) downwash measurement by yawheads, and (3) effective aspect ratio by drag curve.

The first was indeterminate, the second gave AR = 25; and the last indicated that there was induced drag. The second step was compared with Glauert's wall corrections.

For a discussion of straightening the flow in the tunnel, a necessary step before angular measurements could be made, see Appendix.

Introduction

Although many experiments had been made with pressure wings and with wings that completely spanned the jet of a wind tunnel, so far as is known, no investigation had been made with the combination, a pressure wing which spanned a closed jet, that included readings close to the wall.

The experiments were begun in June, 1938. Preliminary investigation had shown that the flow was irregular and to correct this anti-twist vanes had been installed. Refinements of the flexible trailing edges of the twist vanes and further adjustments were undertaken to smooth out the flow.

Next the pressure wing was installed and the pressure distribution read for enough points (usually four) to determine the straight portion of the lift curve. Lift curves were determined for stations every two inches from the tunnel center line to the wall with four extra stations being taken in the last inch, closest to the wall, where the flow changed most abruptly.

The lift was then plotted against the span, and the circulation variation across the span was calculated. From the slope of the circulation curve an attempt was made to apply Prandtl's downwash equation:[1]

$$w(x) = \frac{1}{4\pi} \int \frac{\frac{d\Gamma}{dx'} \, dx'}{x - x'} ,$$

to calculate the downwash and hence to get the effective aspect ratio.

The total downwash was then measured by both a large and a small yawhead. Since part of it was due to the circulation about the wing, the total downwash had to be diminished in order to determine that part, if any, due to the trailing vortices.

Finally, the chord force pressure distribution was plotted from the pressure distribution curves, and chord force and drag coefficients were calculated. Then a drag curve was plotted for a station at the center line.

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1 Hermann Glauert, The Elements of Aerofoil and Airscrew Theory, p.135.

From the behavior of this curve it was possible to draw conclusions as to whether the drag was varying with lift or not.

Apparatus

The experiments were carried out in the small wind tunnel at the Daniel Guggenheim School of Aeronautics at the Georgia School of Technology. This tunnel[2] has a square thirty inch jet and is of the return type.

Pressure distribution curves were made by means of a pressure wing that completely spanned the jet. This wing, of Clark Y section, slipped snugly through rotatable end plates in such a manner that the ring of orifices could be moved to any station across the jet from center line to the wall. The wing had a 6"chord and 50" span, which, allowing for end fixtures necessary, gave the required lateral movement. Measurements of the angle of attack were made by placing an inclinometer on two pins set into one end of the wing. The rotatable end plates had a set screw arrangement so that the wing could be turned to any desired angle and locked.

Each orifice was connected to a tube of a multiple manometer which consisted of a bank of alcohol filled tubes covered by a glass plate. This enabled the pressure distribution to be plotted directly.

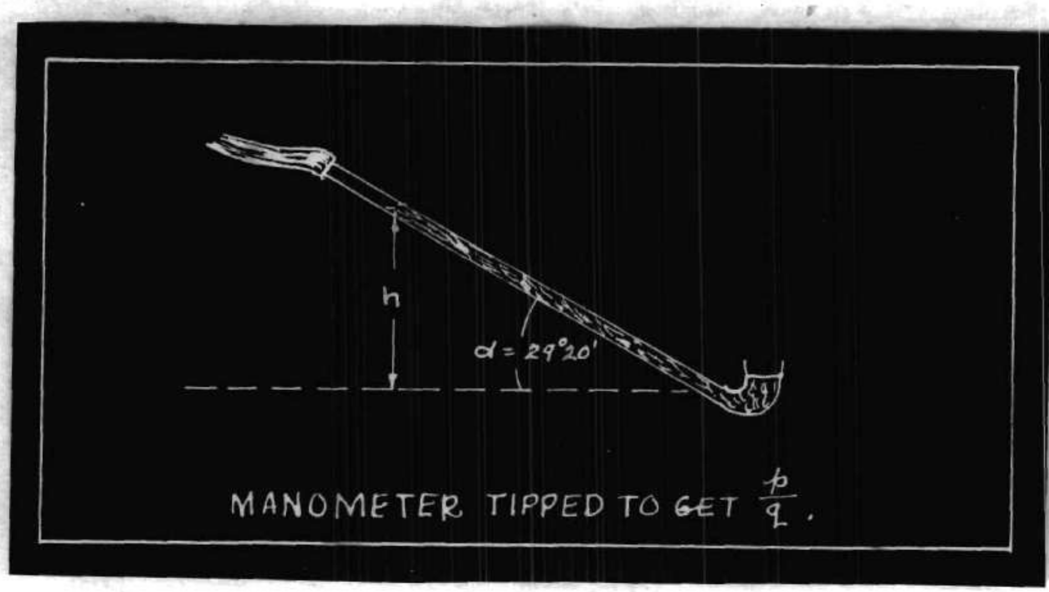
The angle of flow in the tunnel was measured by yawheads described in Figures 7 and 8. The large yawhead was a three-quarter inch tube with orificies one-half inch apart, and a built in vernier scale to read angles. The small yawhead was made of one-quarter inch tubing and had orifices 3/32" apart. Angle measurements were made with a fixed arm using an inclinometer.

2
George Van Schliestett, Experimental Verification of Theodorsen's Theoretical Jet Boundary Correction Factors.

Discussion

Operation of the Pressure Wing

The use of the pressure wing was found to be a quick, accurate method of measuring the lift. To facilitate the computations, the manometers which measured the pressures were set at an angle which was so determined as to reduce the pressure to a function p/q, conveniently arranged to make a unit p/q a unit on the graph. This enabled the lift coefficient to be simply calculated as one twenty-fourth of the area of the pressure distribution curve, as follows:



a) For a one inch strip of wing the pressure is in pounds per inch. When multiplied by the chord, this gives pounds normal force which, when divided by the dynamic pressure, q, gives the normal force coefficient Cn.

b) For convenience, the manometers were tipped to divide effectively the pressure by q. Thus, for 40 miles per hour (the running speed), q was

2.48 cms. of alcohol[4] or .979 inches of alcohol.

$$\frac{p}{q} = \frac{p}{.979} = 1.01\ p\ .$$

c) To get a large scale we double this and, using a p/q of 2.02, arrive at the equation

$$\sin\alpha = \frac{p}{2.02p} = .496\ ,$$

which value corresponds to an angle of 29°20'.

d) The normal force coefficient:

$$Cn = \frac{N}{qt} \quad \text{(for a one inch strip)},$$

$$= \frac{1}{qt}\int p\ dt\ ,$$

$$= \frac{1}{t}\int \frac{p}{q}\ dt\ ,$$

$$= \frac{1}{6}\int \frac{p}{q}\ dt\ .$$

But, in plotting the pressure distribution curve, the chord was made 12" instead of 6", and from (c) above, the ordinate was doubled, so the graph shows <u>four</u> times the actual area.

Hence,

$$Cn = \frac{1}{24}\int \frac{p}{q}\ dt\ .$$

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4 G.A. Mahoff, L.B. Rumph, and W.R. Weems, Calibration of Small Wind Tunnel at Georgia Tech, Table I, Report No.4, 1932-1933, unpublished student technical report deposited in the Library of the Daniel Guggenheim School of Aeronautics, Georgia School of Technology.

Figure 2 shows a typical pressure distribution curve, faired in, and with the area and normal force coefficients as obtained from integration with a planimeter. The normal force coefficients, as obtained in this manner, were plotted for positions across the jet from the center line to the wall, as shown in Figure 3. They were unchanged except for a correction in q necessitated by the drop in velocity close to the wall (see Figure 4). It was found that the pressure wing could be used to read the velocity, in other words, q, if the wing was set at a low angle (about $2^{\circ}30''$) and the value of the lowest (highest pressure) ordinate considered to be q. The value was from the orifice that became headed directly into the wind. An example of this is indicated on Figure 2.

Close to the wall the velocity, q, decreased. Figure 5 shows a run with lowered velocity, with the correction shown. This correction is needed as the manometer stand was set for an angle that would be right for only one speed.

The Cn, henceforth called Cl, which is well within the limits of accuracy up to an angle of attack of 8°, as read from Figure 3, and the velocity, as read from Figure 4, may be used in the following manner to get the circulation across the span.

In the following equations:

L = lift in pounds,

ρ = mass density.

$$L = \rho V b \Gamma$$

or,

$$\Gamma = \frac{L}{\rho V b} \quad \text{(for a unit span),}$$

and

$$\Gamma = \frac{\frac{\rho}{2} S V Cl}{\rho V b} ,$$

$$= \frac{VCl \quad t}{2} .$$

The circulation was plotted against the span in Figure 6 for an α of 6°, which corresponds to a Cl of 1.115. The slopes were then estimated as nearly as possible and applied in Prandtl's equation:[5]

$$w = \frac{1}{4\pi} \int \frac{\frac{d\Gamma}{dx'} dx'}{x - x'} .$$

Unfortunately, the reversal of the slope of the circulation curve near the wall was of such a magnitude that the downwash could not be calculated. A slight variation in estimating the slope would produce a considerable variation in the downwash, precluding accuracy. An example of this is given below.

From Figure 6, Circulation Against Span, the slopes were calculated first. The example was worked for an angle of attack of + 1°. The curve was broken down into a series of straight lines and the slopes were as follows:

TABLE 1

Section	Distance (Span)	Height	Slope
A)		0	0
B)	.33 to .50	-9.5 - 8.9	$\frac{.6}{.17} = 3.53$
C)	.50 to .740	0	0

5 Glauert, op.cit., p.135.

TABLE I (Continued)

Section	Distance (span)	Height	Slope
D)	.74 to 1.00	8.9 - 7.7	$\frac{1.2}{.26} = 4.62$
E)	1.00 to 1.207	7.33 - 6.6	$\frac{.73}{.207} = 3.52$
F)	1.207 to 1.24	6.6 - 8.65	$\frac{-2.05}{.033} = -62.2$

Prandtl's Relation for the downwash is

$$w = \frac{1}{4\pi} \int \frac{\frac{d\Gamma}{dx'}\ dx'}{x - x'}$$

where x is taken along the span.

Considering only half a span, this must be doubled. And considering that the change in Γ is constant for a section we then have:

$$w = \frac{1}{2\pi} \frac{d\Gamma}{dx'} \int_a^b \frac{dx'}{x - x'}$$

$$w = -\frac{1}{2\pi} \frac{d\Gamma}{dx'} (\log a - \log b) \qquad (1)$$

when $x = 0$ (at the center line).

The total downwash is the sum of the downwashes due to each section, or:

$$w = \text{Downwash due to } (A + B + C + D + E + F) \qquad (2)$$

Substituting the slopes from TABLE I in the equation (1) we have:

TABLE 2

Section	Substitution	Downwash due to Section considered.
A)	0	0
B)	$w = -\frac{3.53}{2\pi}$ (log.33 - log.50	.285
C)	$w = 0$	0
D)	$w = -\frac{4.62}{6.28}$ (log.74 - log 1.00)	.221
E)	$w = -\frac{3.52}{6.28}$ (log 1.00 - log 1.207)	.126
F)	$w = \frac{62.2}{6.28}$ (log 1.207 - log 1.24)	- .079
	Total $w =$	.553

Hence the induced angle from

$$\alpha_i = \frac{w}{V} = \frac{.553}{59.3} = .00934 \text{ radians.}$$

for a velocity of 59.3 ft./sec.

From
$$\alpha_i = \frac{Cl}{\pi\ AR}$$

we get
$$AR = \frac{Cl\ V}{\pi w}.$$

For this example the AR is $\frac{.64 \times 59.3}{3.14 \times .553} = 21.8$.

Unfortunately, the downwash as calculated for Section F is too approximate and indicates an aspect ratio below the correct value. This is due to the inaccuracies arising from trying to estimate the slope of the circulation curve, especially near the wall.

It is to be noted that the pressure distribution method gives quick, accurate results, as far as lift coefficients are concerned. The curves were straight and shifted over as progress was made towards the wall (Figure 0). At about one-half inch from the wall they showed a radical change, becoming much steeper. Observations of the pressure curves indicated that the total lift was remaining constant although the velocity was decreasing. Consequently, the circulation went up close to the wall also, and that gave an opposite sense to the tip vortices (Figure 6).

Downwash by a Large Yawhead

When it was apparent that the downwash could not be calculated by the slope method in a closed tunnel, a second method was tried. A yawhead made of three-quarter inch tubing with orifices one-half inch apart was used and stations taken every few inches back of the wing and in the plane of the wing. The angle of attack was varied and the total downwash measured at each station.

Using a yawhead with orifices at such a distance apart gave poor results. Apparently, as the stations were in the wake, there was a pressure difference between the two holes that indicated an angular relation that did not exist. Although for angles of attack between -3° and $+3^{\circ}$ the points faired into a straight line, this line did not have the correct slope, as later determined, nor did the zero lift point coincide for all stations back of the wing.

An effort was made to take stations below or above the wake, but no satisfactory results could be obtained.

Figure 7 is a drawing of this yawhead.

Downwash with a Small Yawhead

It was found that a yawhead with a small space between orifices would read better than that of the previous discussion. This type with only 3/32" between holes gave points that were substantially a straight line from zero lift to 8°, and all curves started from the same point for zero lift. The reason for the smaller yawhead being more accurate is that by reading in the wake as was done, differences present in both pressure and velocity appeared on the yawhead as angular changes.

Now the downwash, as measured, was due to two sources; the trailing vortices and the circulation about the wing. Since the induced angle and hence the effective aspect ratio depend on the part due to the trailing vortices, it was necessary to calculate and subtract the circulation effect. The variation of the circulation strength was small over the span, and the average value from Figure 6 for $\alpha = 5^{\circ}$ was found to be 14.1 ft.^{2}/sec.

The case under consideration was treated as a segment of constant strength, and the downwash due to circulation, w_c, was found as follows:[6]

$$w_c = \frac{\Gamma}{4\pi h} (\cos \theta_1 - \cos \theta_2),$$

for an $\alpha = 5^{\circ}$ we obtain the following table:

TABLE 3

h in.	h ft.	$\frac{\Gamma}{4\pi}$	$\frac{\Gamma}{4\pi h}$	tan θ	θ	cos θ	2cos θ	w_c	α_c
9.6	.80	1.12	1.40	.640	32°30'	.834	1.67	2.33	2.26°
11.4	.95	1.12	1.18	.760	37°10'	.797	1.59	1.88	1.82°
13.0	1.19	1.12	.94	.865	40°50'	.757	1.51	1.42	1.37°
17.0	1.42	1.12	.79	1.13	48°30'	.663	1.34	1.16	1.12°
20.2	1.68	1.12	.67	1.35	53°30'	.594	1.19	.79	.76°

[6] Glauert, op. cit., p.128.

$$\tan\theta = \frac{\text{dist. downstream}}{\text{half jet width}} = \frac{h(\text{inches})}{15}$$

$$\alpha_c = \text{downwash due to circulation}$$

$$= \frac{w_c}{V} = \frac{w_c\,(57.3)}{59.3} = .966 w_c$$

The values of w_c are plotted in Figure 9, as well as the values of the total downwash as read directly from Figures 10-14. The maximum difference occurs 17" back of the quarter chord point and is about 1.2°. Theoretically, half of this value has been reached at the lifting line, but practically, the value there is only 1/1.6, not 1/2.

The induced angle then becomes

$$\frac{1.2}{1.6} = .75^\circ.$$

From

$$\alpha_i = \frac{Cl}{\pi AR},$$

we get

$$AR = \frac{Cl}{\pi\,\alpha_i}$$

For the case under consideration this becomes

$$= \frac{(1.04)\,(57.3)}{(3.14)\,(.75)}$$

$$= 25.3.$$

Inasmuch as the lift distribution survey indicated that the lift vanished at the wall, it is interesting to compare the results with those obtained by assuming that the wing has its geometric aspect ratio, elliptic loading, and normal wall corrections. These are obtained from Glauert.[7]

[7] Hermann Glauert, The Interference on the Characteristics of an Aerofoil in a Wind Tunnel of Rectangular Section, p. 244.

The downwash angle is due to the aspect ratio and the tunnel wall effects.

Thus:

$$\Delta\alpha = \frac{Cl}{\pi AR} - \delta \frac{S}{C} Cl$$

where δ is the tunnel wall correction.

For Cl = 1.04, wing area, S = 180 sq.in., tunnel area, C = 900 sq.in., and AR = 5, and using the induced angle, as determined, = .75, we obtain the following:

$$\frac{.75}{57.3} = \frac{1.04}{\pi 5} - \frac{180}{900} 1.04,$$

or,

$$\delta = .254$$

In our notation this becomes .508. (C_L = 2kL)

Glauert gives the wall correction as a function:[8]

$$\bar{\eta} = \delta = \lambda F - (6) + F_1(6)$$

where

$$F_1(6) = 2\pi \sum_{m=0}^{\infty} \frac{(2m+1)!\ (2m+2)!}{m!\ (m+1)!\ (m+1)!\ (m+2)!} \left(\frac{1}{4}\right)^{2m} \sum_{p=1}^{\infty} \frac{1}{p^{2(m+1)}}$$

$$\lambda F(6) = 2\pi \sum_{p=1}^{\infty} \frac{p(q)^p}{1+q^p} \left[\frac{J_1(\pi p)}{\pi p}\right]^2$$

$q = e^{-2\pi\lambda}$ where λ is the ratio of tunnel height to width; in our case = 1.

And[9] $q = .00187$

Substituting for the series p = 1,2,3,4, etc., and m = 0,1,2, etc., and evaluating $J_1(\pi p)$ as a Bessél function, we get:

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[8] Glauert, op. cit., p. 246.

[9] Ibid., p. 242.

$$F_1(6) = .052$$

$$\lambda F(6) = \underline{.369}$$

$$\delta = .411.$$

This value compares with the value obtained by experiment of $\delta = .508$, and indicates that a wing spanning a wind tunnel jet may be considered a wing of geometric aspect ratio and elliptic loading; affected, of course, by the tunnel wall correction as given in Footnote 9.

The accuracy is within the range of the estimate of the induced angle at the wing in terms of the maximum induced angle behind the wing.

Since the lift distribution is not elliptic, the downwash is larger, but apparently the wall interference is increased in about the same ratio, and this tends to cancel out the difference.

The slope of the lift curve from Figure O is found as follows:

$$Cl = 1.04,$$

$$\alpha = 5.0^\circ \text{ or } 10.8^\circ \text{ from zero lift,}$$

$$\text{Slope} = \frac{1.04}{10.8} (57.3) = 5.5 .$$

Downwash at wing for this Cl is $.7^\circ$, hence slope for infinite aspect ratio at Reynolds' Number of 170,000 is

$$\frac{1.04}{10.8 - .7} (57.3) = 5.9 .$$

In order to get assistance in fairing the curve of downwash due to trailing vortices, a calculation of the theoretical values was made. The value of the downwash due to one vortex, as given by Glauert,[10] is:

$$w_3 = \frac{-\Gamma}{4\pi} \quad \frac{(y - \frac{b}{2})}{z^2 + (y - \frac{b}{2})^2} \left[1 + \frac{x}{\sqrt{x^2 + z^2 + (y - \frac{b}{2})^2}}\right]$$

In our case, for downwash at the center line and in the plane of the wing, y and z = 0, and the downwash for two vortices becomes:

$$w = \frac{\Gamma}{\pi b} \left[1 + \quad \frac{x}{\sqrt{x^2 \; + \; \frac{b^2}{4}}}\right]$$

For the case under consideration,[11]

$$\Gamma = 14.1$$

$$x = h$$

and b is determined from the formula,

$$AR = \frac{b^2}{S},$$

or

$$b^2 = S \; AR,$$

$$= 180 \times 28,$$

$$b = 71 \text{ inches}.$$

The several values of w work out to be:

Γ	h	w
14.1	20"	1.17
14.1	15"	1.07
14.1	10"	.96

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10 Hermann Glauert, The Elements of Aerofoil and Airscrew Theory, p.159.

11 See page 11.

When these are plotted, see Figure 9, it is seen that the curve should be very flat near the wing, and the points discarded are in all probability too close to the wing to give accurate results. Fortunately, the results depend on the maximum value of the downwash due to the trailing vortices.

Calculation of Cd.

Some work has been done by the U.S. National Advisory Committee for Aeronautics and other research agencies in measuring the drag by a pressure-distribution method. Since the pressure curves had already been plotted for the aspect ratio and lift part of the investigation, it was decided to use them and see what could be done in the way of drag calculations.

One of the first principles in this work is the understanding of the meaning of plus and minus pressures. A plus pressure on the nose of the airfoil gives drag, but on the part past the maximum thickness, it gives anti-drag. Exactly the opposite is true with minus pressures. Hence, in plotting these values against the thickness of the airfoil, it is necessary to keep in mind the meaning of each reading. To facilitate this, each orifice in the wing was numbered (see Figure 15) and the height of each marked on the ordinate axes. The values were scaled off the pressure distribution and plotted as drag or anti-drag as the case might be.

An example has been plotted for Figure 2, although this station (10" from the wall) was not used in the drag calculation. The values on Figure 2 have been numbered to identify the orifice from which they came. The number assigned to each orifice is shown on Figure 15. The drag plot, giving the area and calculated Cd, is shown in Figure 16.

The method of calculation is as follows:

$$\text{Chord force} = q\,S\,Cc,$$

$$= \int \frac{p}{q} \frac{dh}{t}$$

$$= \frac{1}{t} \int \frac{p}{q}\, dh$$

However, the chord of the wing is 6", and to make a clearer plot, the thickness was quadrupled. Since the ratio p/q was also doubled[12], the chord coefficient becomes

$$Cc = \frac{1}{6 \times 4 \times 2} \int \frac{p}{q}\, dh \,,$$

$$= \frac{1}{48} \int \frac{p}{q}\, dh \quad .$$

Thus, the chord coefficient is obtained by subtracting the anti-drag area from the drag area of the chord pressure distribution curves and dividing the result by 48.

This has been done for six points and yields the following. The angle of attack as given is uncorrected[13], and drag forces are considered plus.

TABLE 4

Chord Force Coefficients

α	Area	Cc
-4 20'	.91D	.019
-2 30'	.72D	.015
1 10'	.14D	.003
3 15'	.90A	- .019
5 10'	2.50A	- .052
7 00'	4.08A	- .085

12 See page 5.

13 See page 19.

These values are plotted in Figure 17. (Note: D signifies a drag or (+) coefficient, A an anti-drag or (-) coefficient.)

To obtain the drag coefficient, the normal force and chord force must be resolved as follows:

$$Cd = Cn \sin \alpha + Cc \cos \alpha \, .$$

A table of results is below:

TABLE 5

α (Measured)	Cn	Cc	α (true)	$\sin \alpha$	$\cos \alpha$	Cn $\sin \alpha$	Cc $\cos \alpha$	Cd
- 4 20	.14	.019	-4 30	-.0756	.997	-.0108	.019	.008
- 2 50	.28	.015	-3 00	-.0523	.999	-.0146	.015	.0004
- 1 20	.43	.020	-1 30	-.0262	.999	-.011	.020	.009
1 10	.66	.003	1 00	.0175	.999	.012	.003	.015
3 15	.86	-.019	3 05	.0539	.999	.046	-.019	.027
5 10	1.04	-.052	5 00	.0877	.996	.091	-.052	.039
7 00	1.21	-.085	6 50	.1190	.993	.144	-.085	.059

A polar plot of these values is given in Figure 18.

Since the results of these plots seemed to be open to question, it was supposed that a possible dynamic pressure variation over the chord might be the cause. This was investigated, and as the pressure change from nose to trailing edge was under .01 cm. of alcohol for a q of 2.48 cms., the error did not lie at that point.

Calculating the Cd values for four points already on the pressure curves gave an indication of the behavior of the Cd curve. (Figure 18 -- the angle of attack plotted is the measured angle) Further points were desired and so additional pressure curves were run at -2 30', - 5 45', and -6 30'. It had been previously noted that the pressure set-up seemed difficult to handle at the very low angles, and this difficulty was apparent in obtaining the new points. The Cn values fell below the straight line of the lift curve.

Assuming that the error was due to angle measurement, an effort was made to check all angles involved. Since the up flow was 1° in the tunnel, and since the pins for measuring the angle of attack were 1°10' too high, it was necessary to correct the angles of attack by subtracting 10'. As can be seen from TABLE 5, the sine of the angle is very critical, as the Cn is much larger than the Cc.

A second assumption to the effect that the angle was correct and the error, if any, was in the integration, led to new points less trustworthy than the old ones.

Conclusions

The downwash relation,

$$w = \frac{1}{4\pi} \int \frac{\frac{d\Gamma}{dx'} \, dx'}{x - x'} ,$$

cannot be used satisfactorily for a wing spanning a wind tunnel jet, as the slopes of the circulation curve close to the wall are very critical and cannot be estimated accurately.

The flow angle as measured by yawheads and corrected for circulation effect indicated that a wing spanning a wind tunnel may be treated as a wing of normal geometric aspect ratio, if the tunnel wall corrections are applied.

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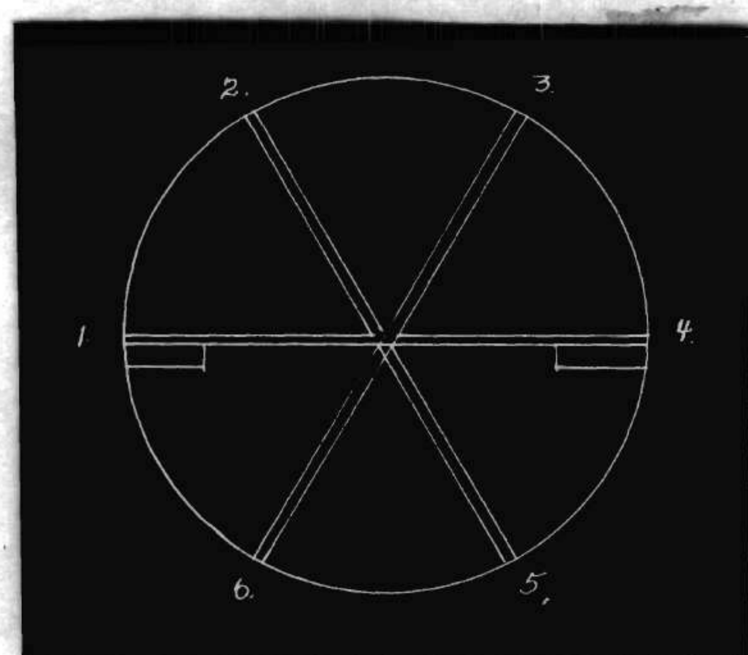
APPENDIX

APPENDIX A

Straightening the Flow in the Tunnel

The copper trailing edges of the anti-twist vanes already mentioned were found to be ample to produce a change in flow, although they were only about 20 per cent of the chord. Apparently, the horizontal vanes, Numbers 1 and 4, had the most effect on the horizontal flow across the jet. Figure 1 shows the Horizontal Survey before regulation and also after. Unfortunately, it was believed that a sharp upflow near (within 2 inches) the wall



Anti-Twist Vanes (Looking Downstream).
(Outer Flaps Down on 1 and 4)

was produced by leakage around the yawhead which was not a tight fit. Later investigation indicated that this was not the case, and the probable cause was a pair of vortices, each of nearly equal strength and opposite sense. The flow apparently does not make a turn from the propeller to the jet, because by varying the twist vane at the <u>front</u> of the tunnel (Number 1), a variation in the flow at the <u>front</u> side of the jet was produced. The variation in angularity was decreased from $\pm 2^{\circ}$ to $\pm 1/4^{\circ}$ by adjusting the flexible trailing edges, but this small variation held only for a static plate of h = 5 cms. of alcohol and decreased to about $\pm 1/2^{\circ}$ for other speeds.

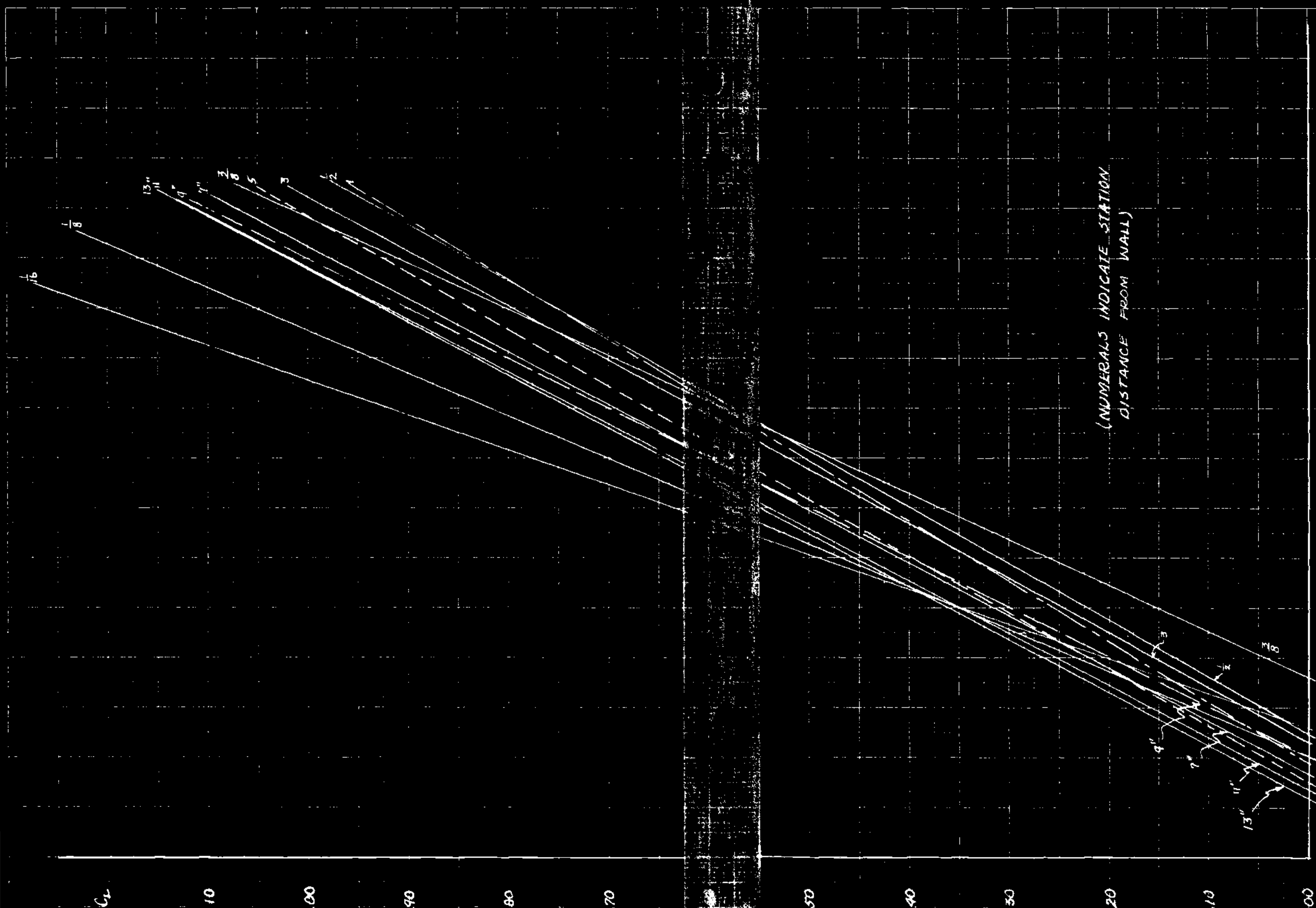


FIG 0
LIFT CURVES

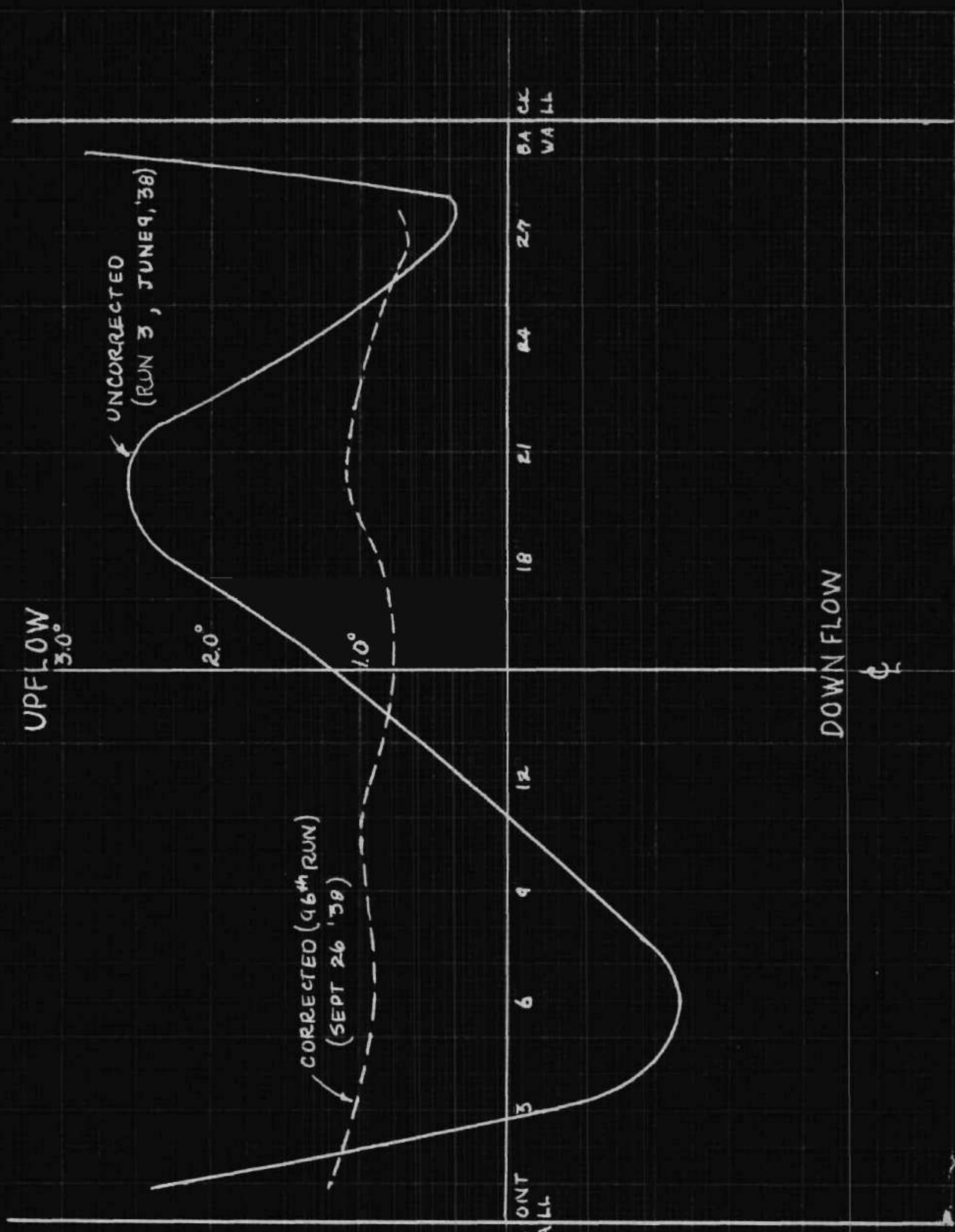


FIG. 1
HORIZONTAL SURVEY

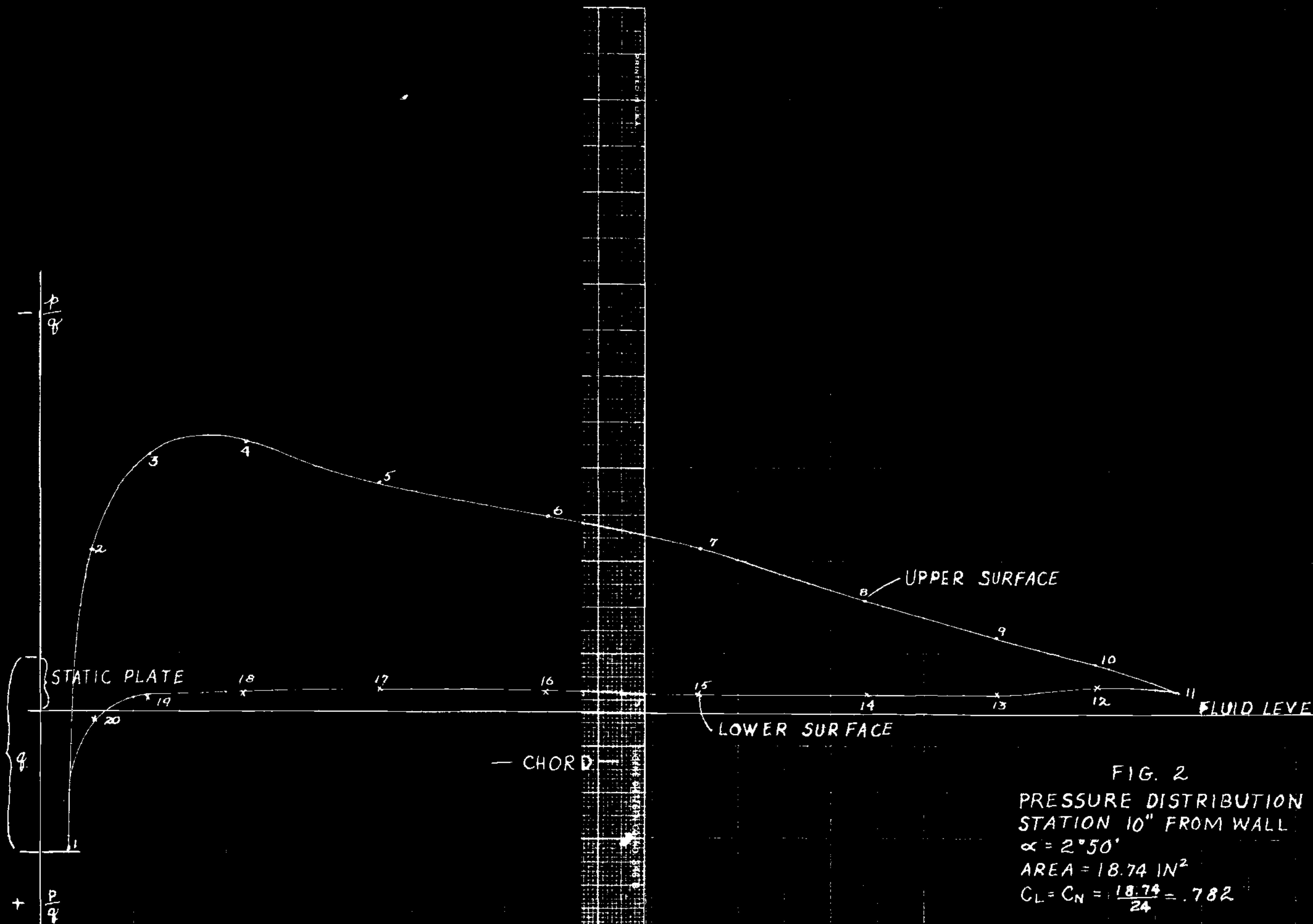


FIG. 2
PRESSURE DISTRIBUTION
STATION 10" FROM WALL
$\alpha = 2°50'$
AREA = 18.74 IN^2
$C_L = C_N = \frac{18.74}{24} = .782$

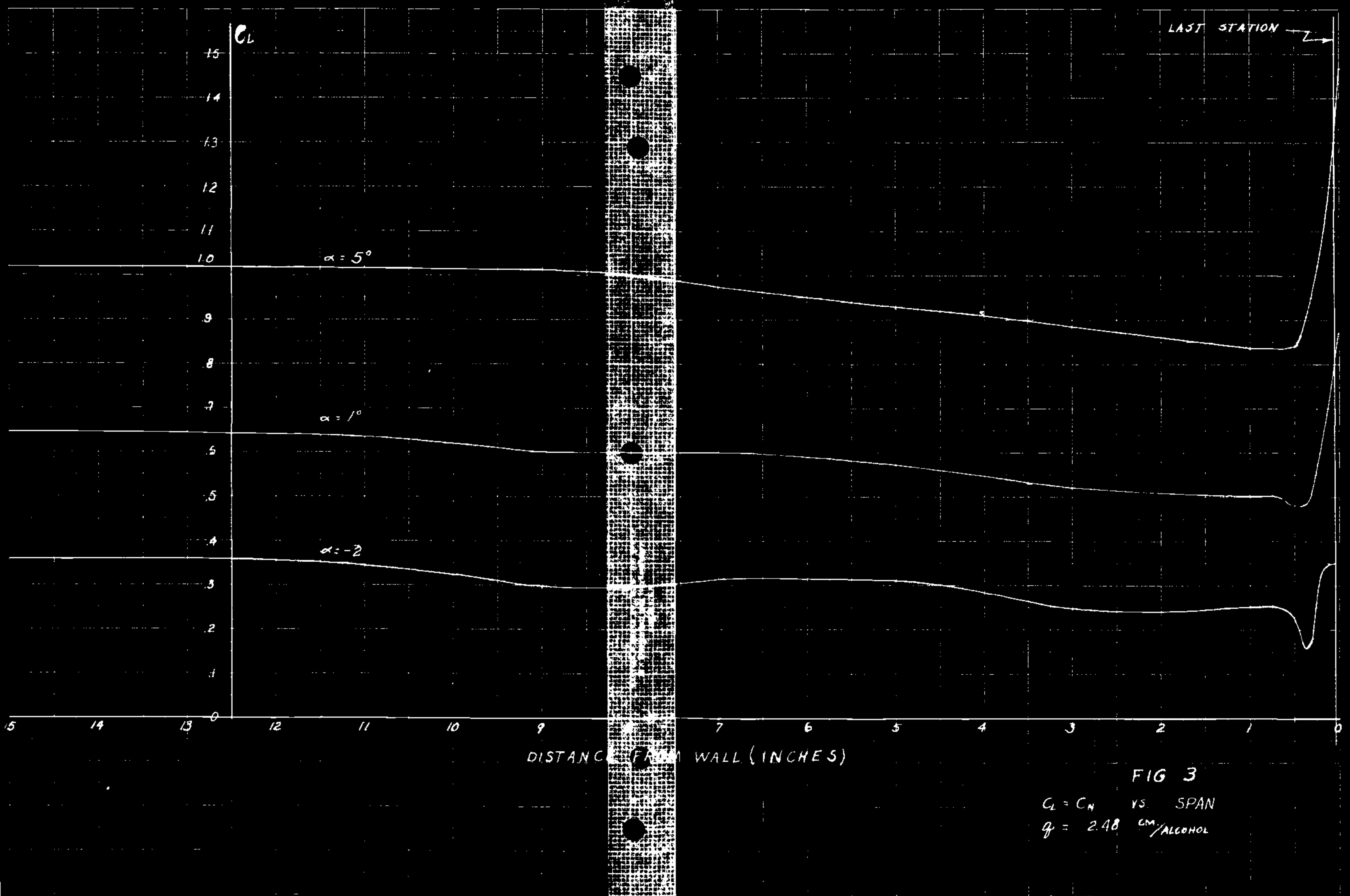


FIG 3

$C_L = C_N$ VS. SPAN

$q = 2.48$ CM/ALCOHOL

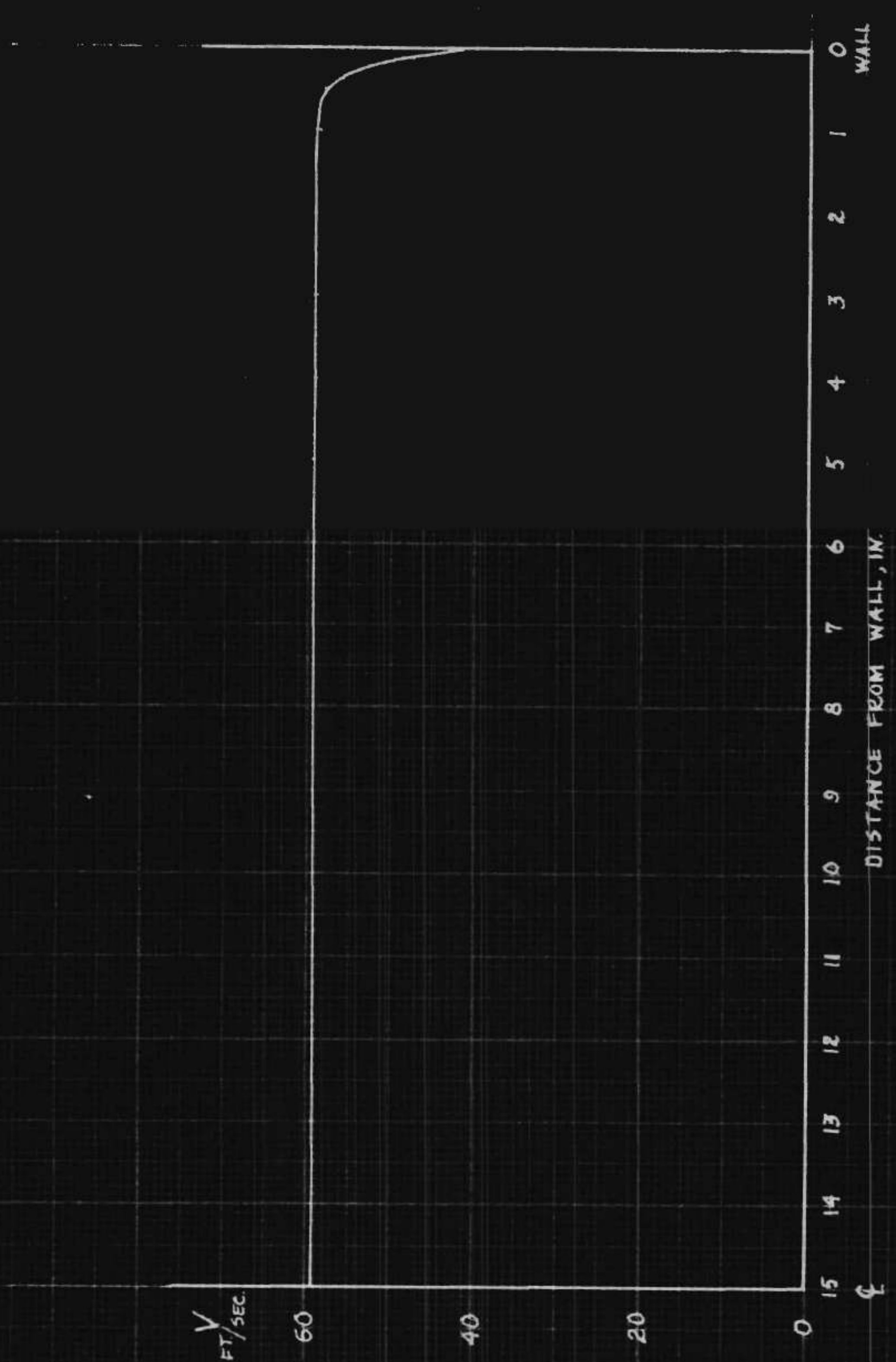


FIGURE 4
VELOCITY VS SPAN

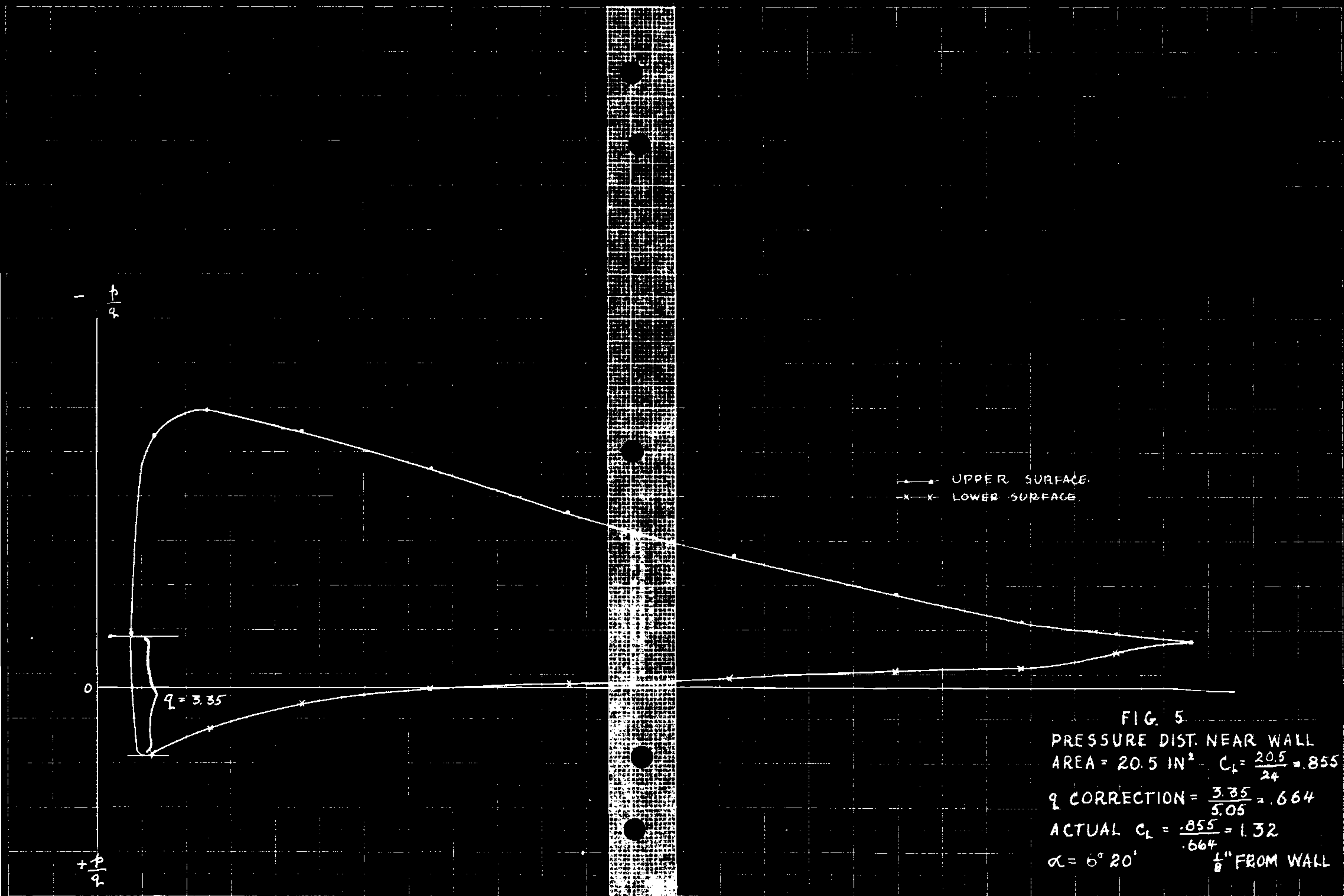


FIG. 5

PRESSURE DIST. NEAR WALL

AREA = 20.5 IN^2 $C_L = \frac{20.5}{24} = .855$

q CORRECTION = $\frac{3.35}{5.05} = .664$

ACTUAL $C_L = \frac{.855}{.664} = 1.32$

$\alpha = 6^\circ 20'$ $\frac{1}{8}$" FROM WALL

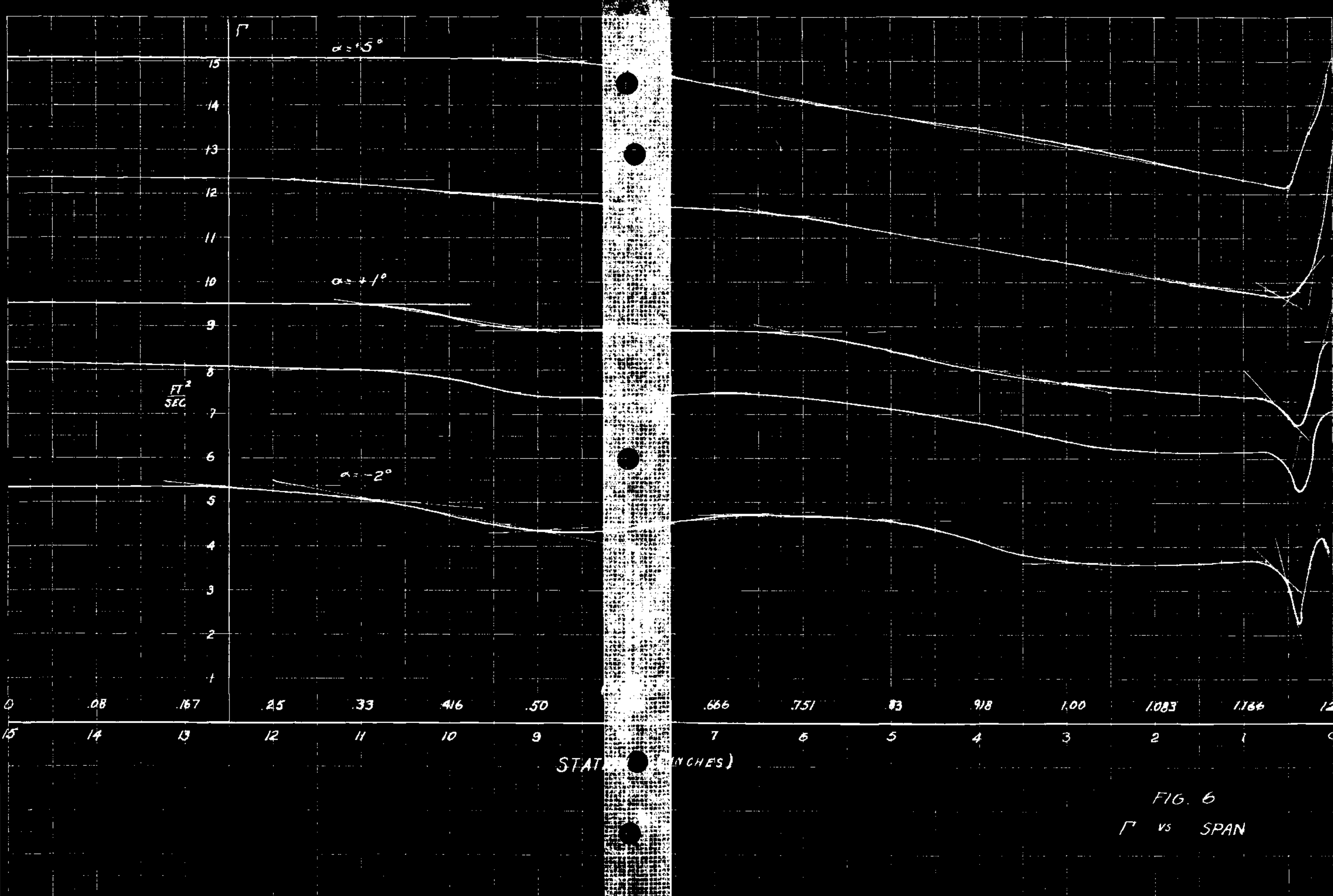


FIG. 6

Γ VS SPAN

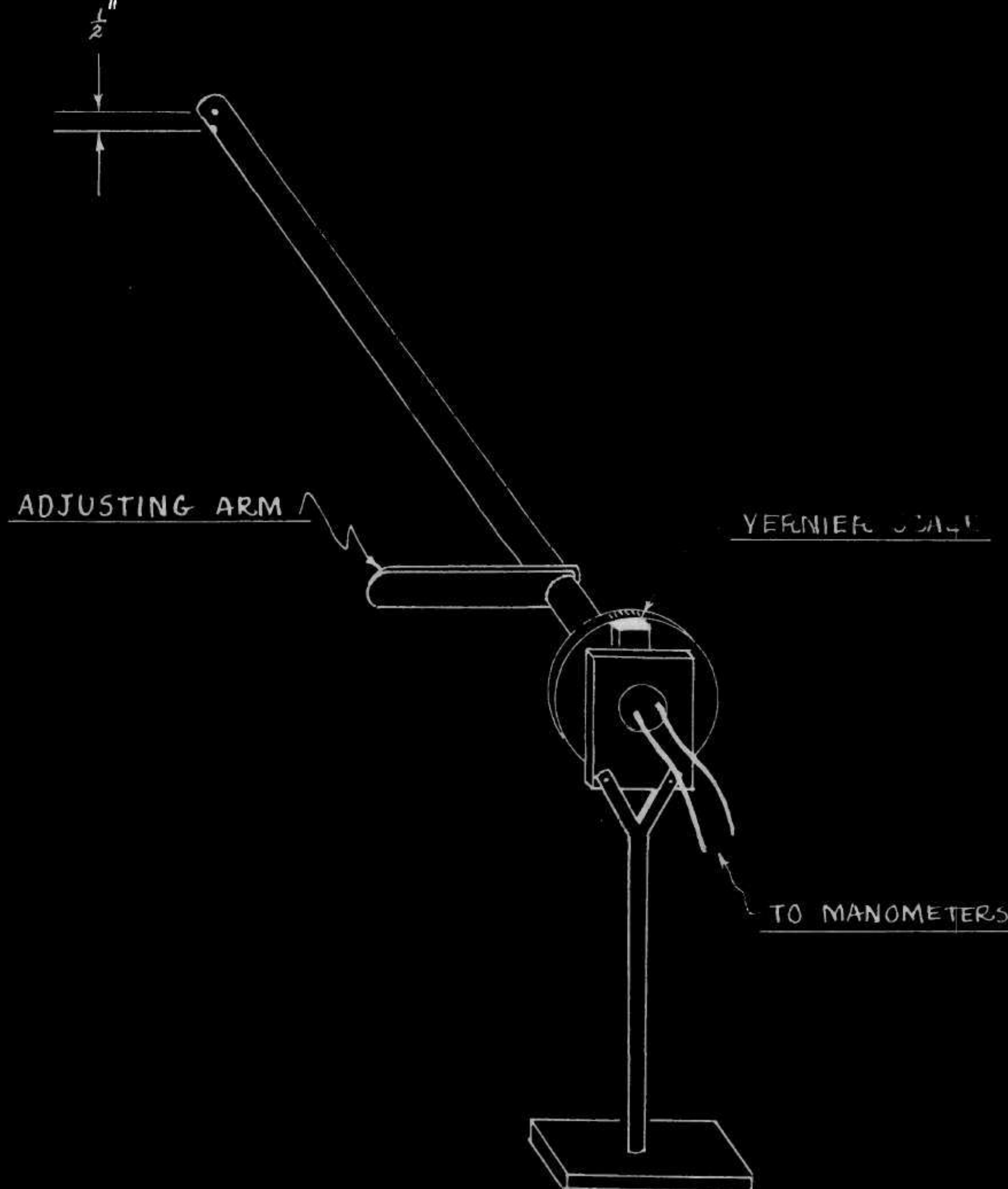


FIG 7
LARGE YAWHEAD.

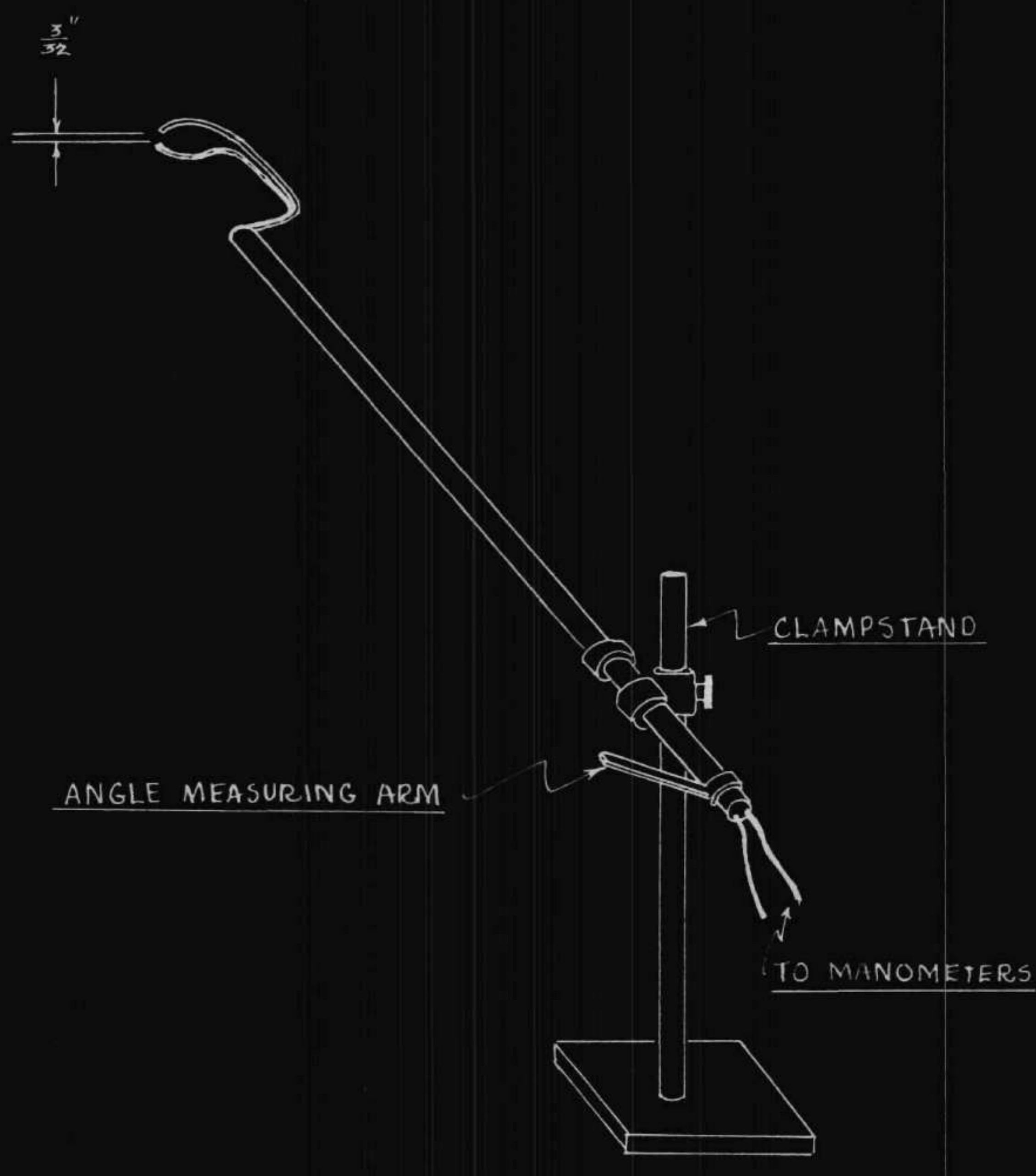
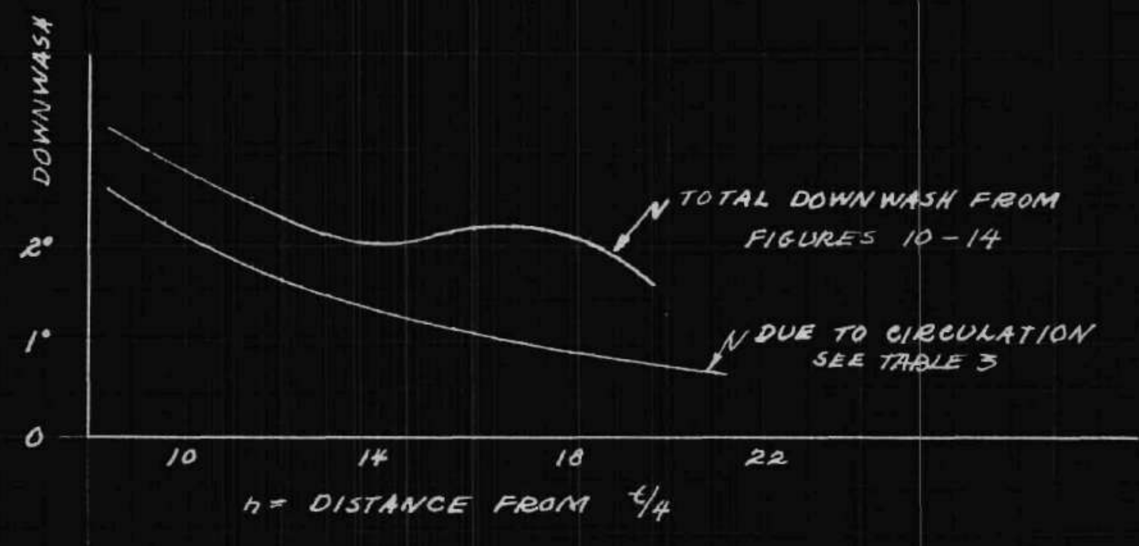


FIG 8

EIB (SMALL) YAWHEAD.



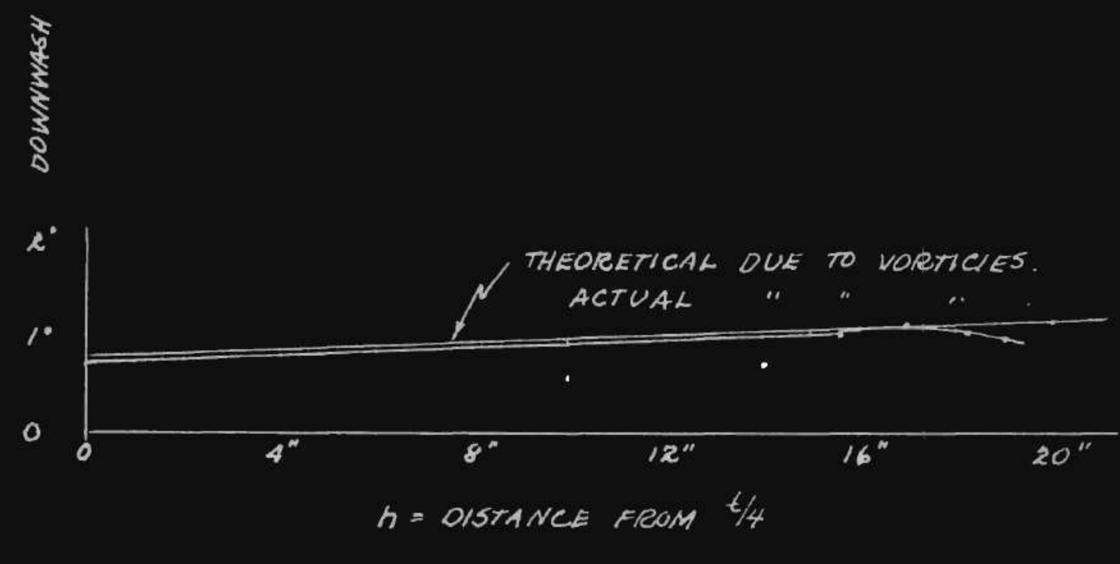


FIG. 9
DOWNWASH VS. "h."

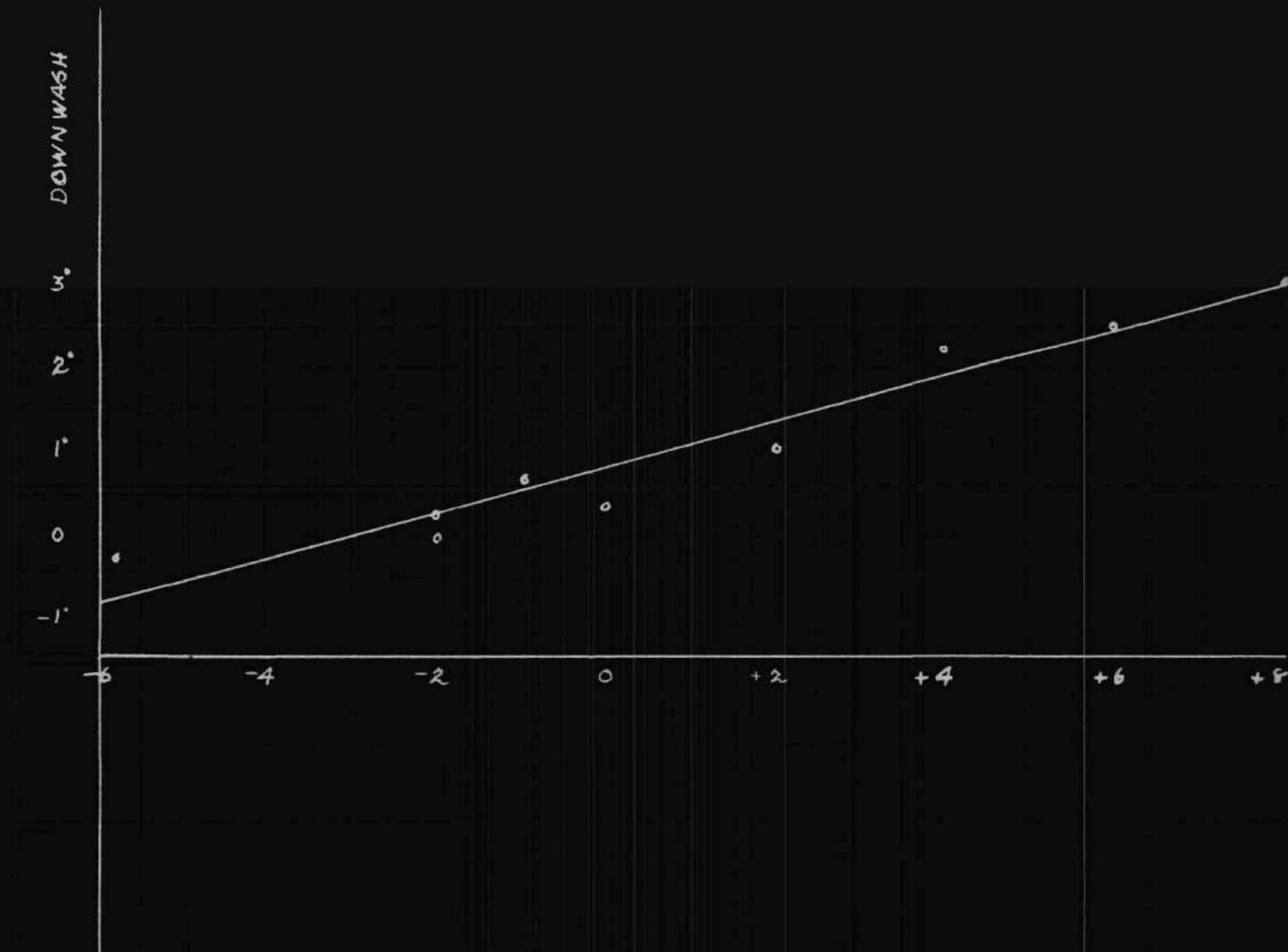


FIG 10
TOTAL DOWNWASH
h = 9 6" (FROM t/4)

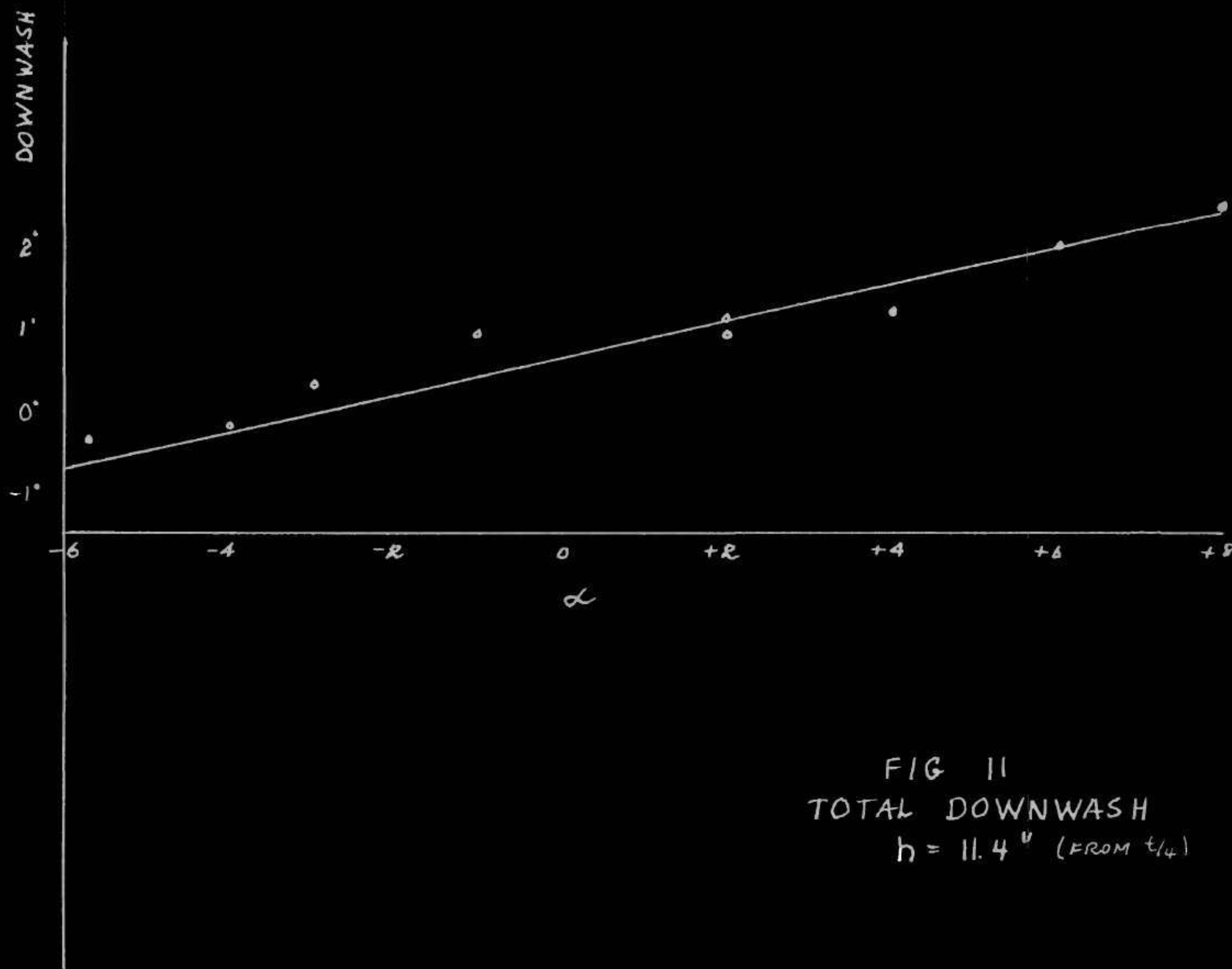


FIG 11
TOTAL DOWNWASH
h = 11.4" (FROM t/4)

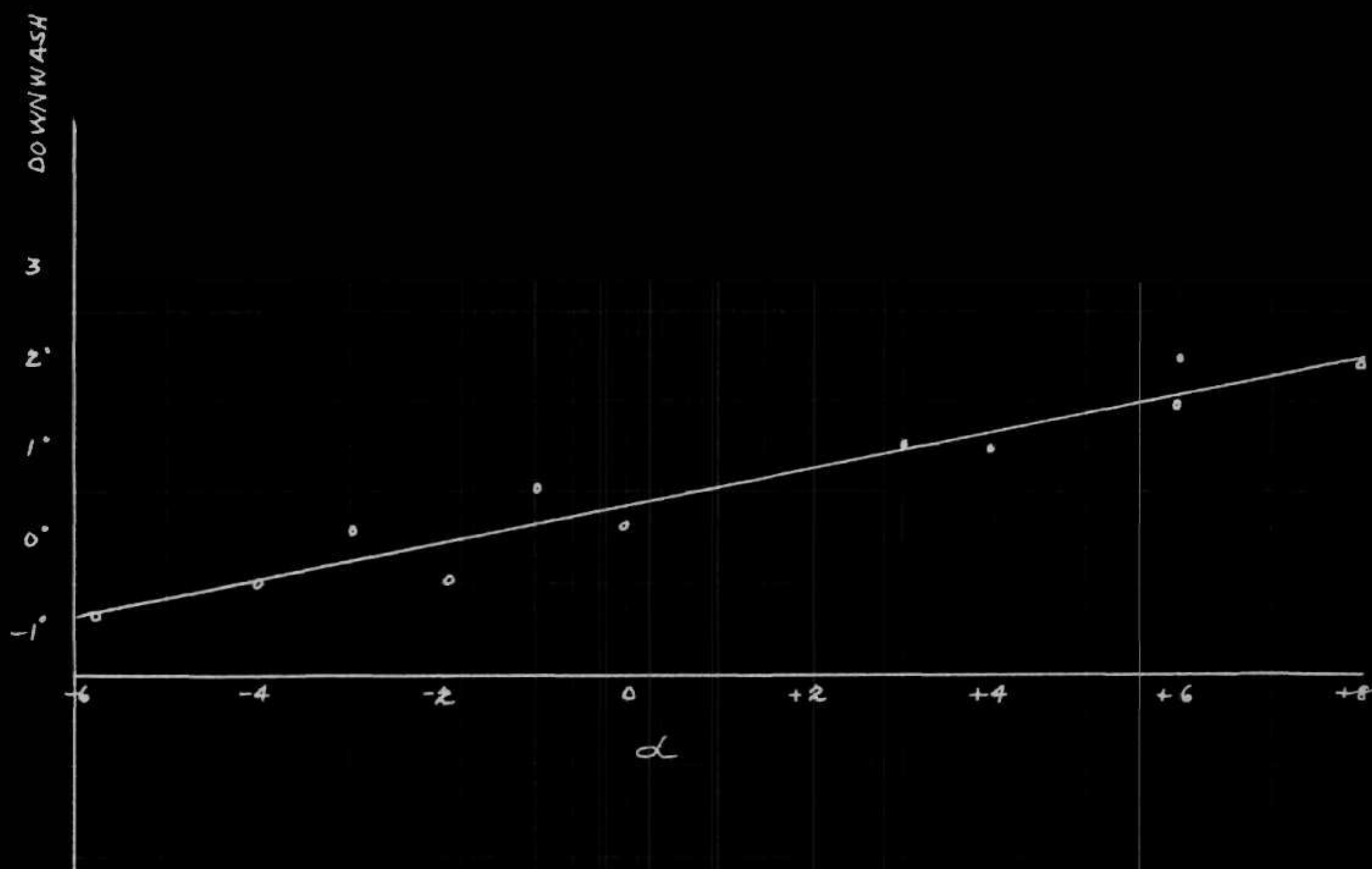


FIG. 12
TOTAL DOWNWASH.
h = 13" (FROM t/4)

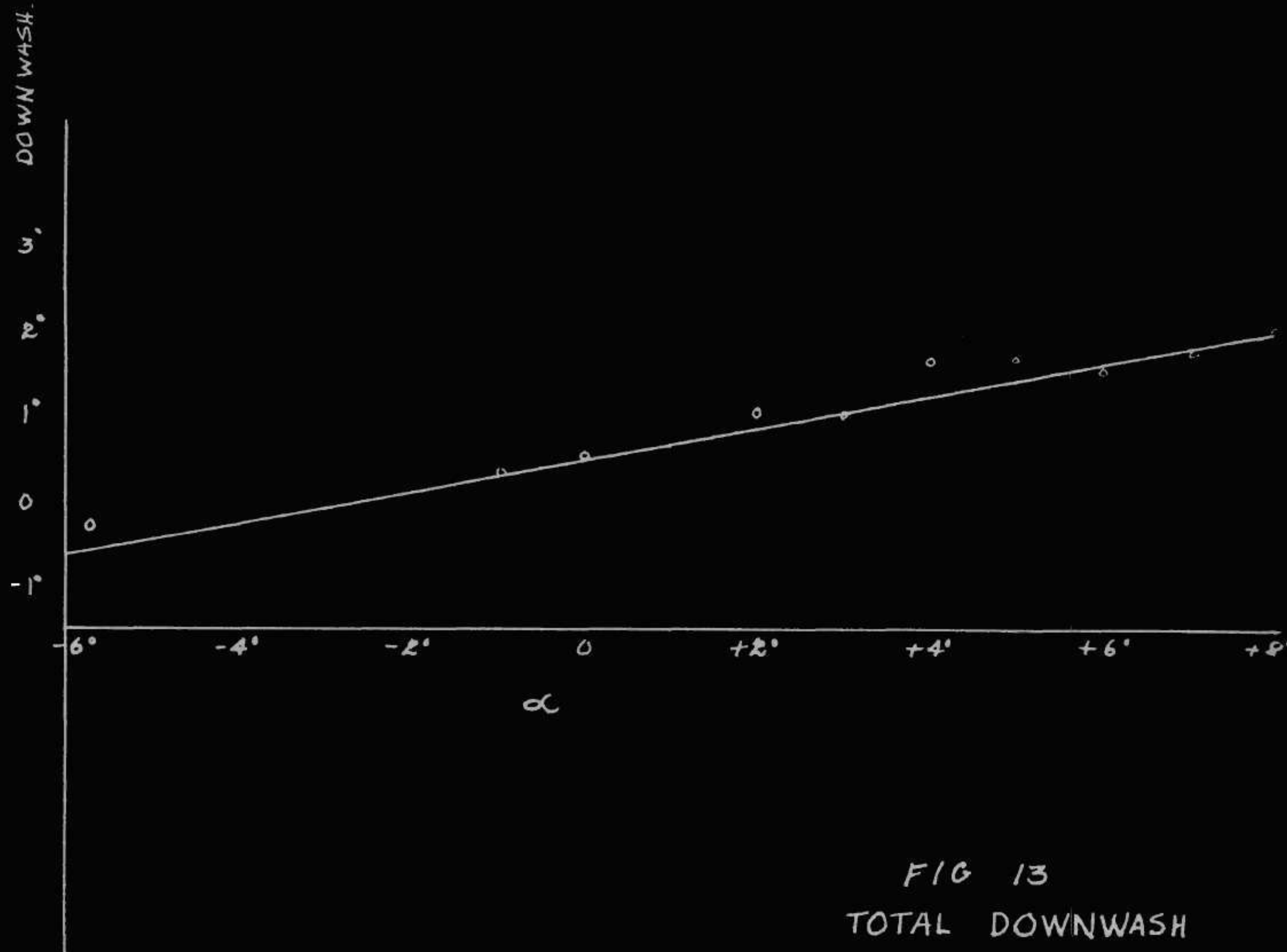


FIG 13
TOTAL DOWNWASH
h = 17" (from c/4

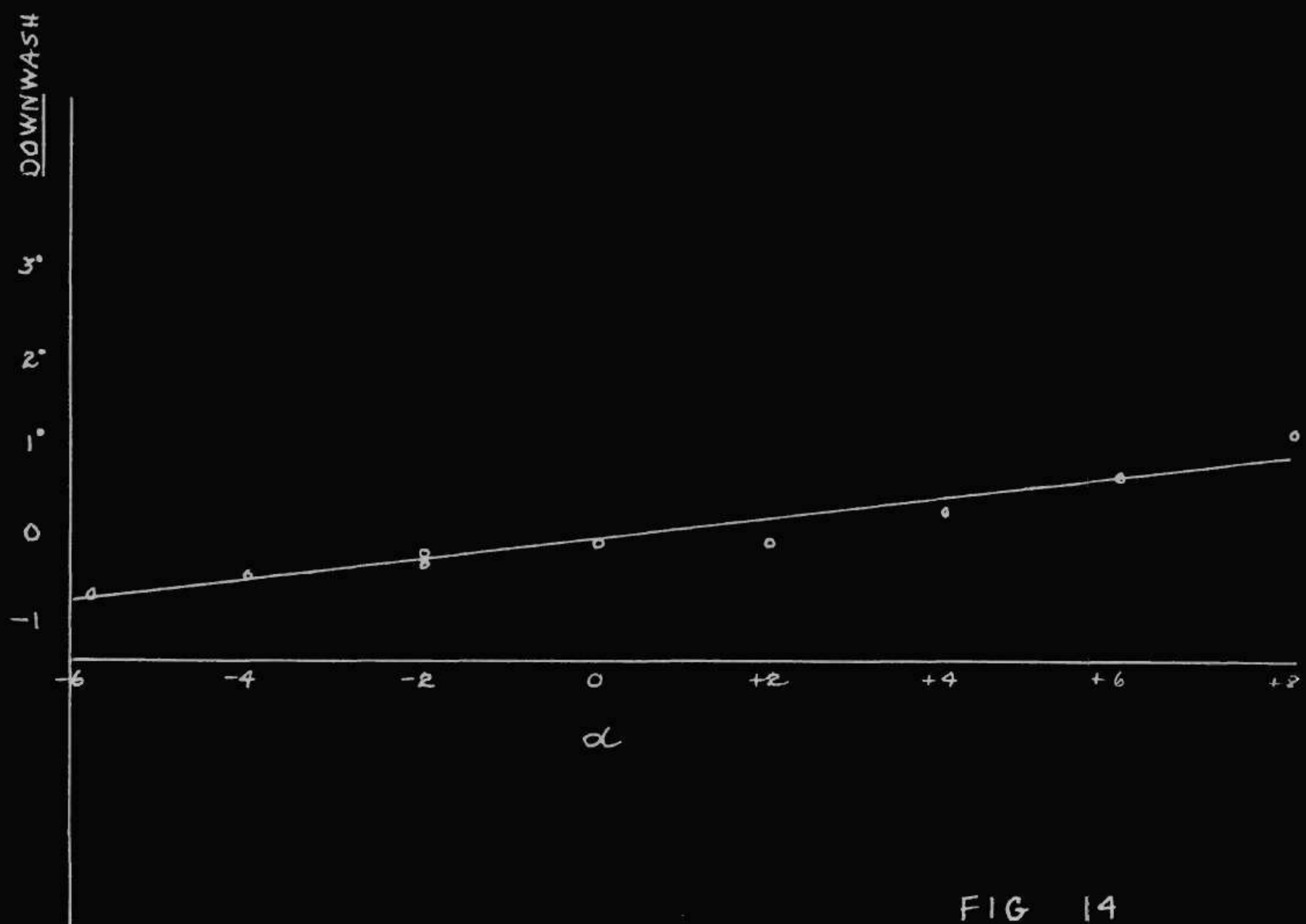
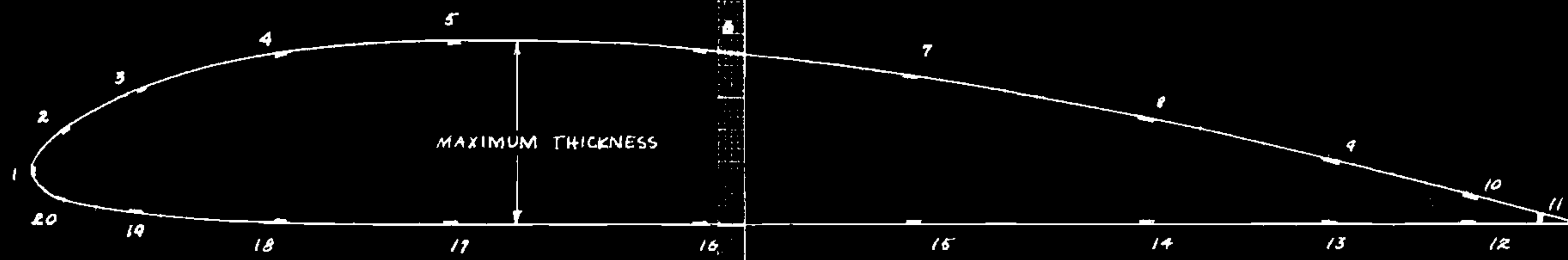


FIG 14
TOTAL DOWNWASH.
$h = 20.2''$ (FROM $t/4$)



SCALE 2" = 1"

FIG. 15
ORIFICE NUMBERS
FOR DIMENSIONS SEE DRWG AE 39
DGSA GA. TECH

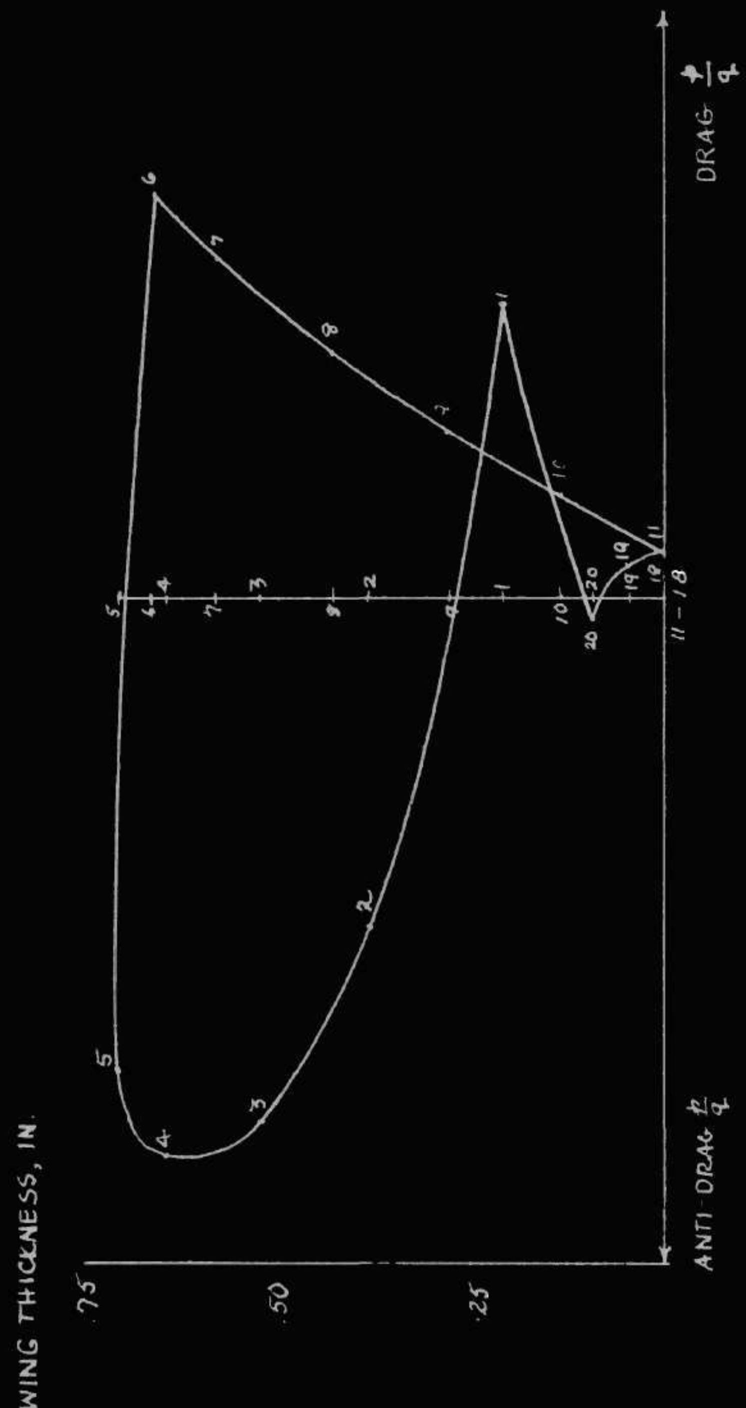


FIG. 16

CHORD FORCE PLOT OF FIG. 2

DRAG AREA 3.64

ANTI-DRAG 3.82

$C_c = \dfrac{3.82 - 3.64}{48} = .00375$

$\alpha =$ [illegible]°50′ (UNCORRECTED)

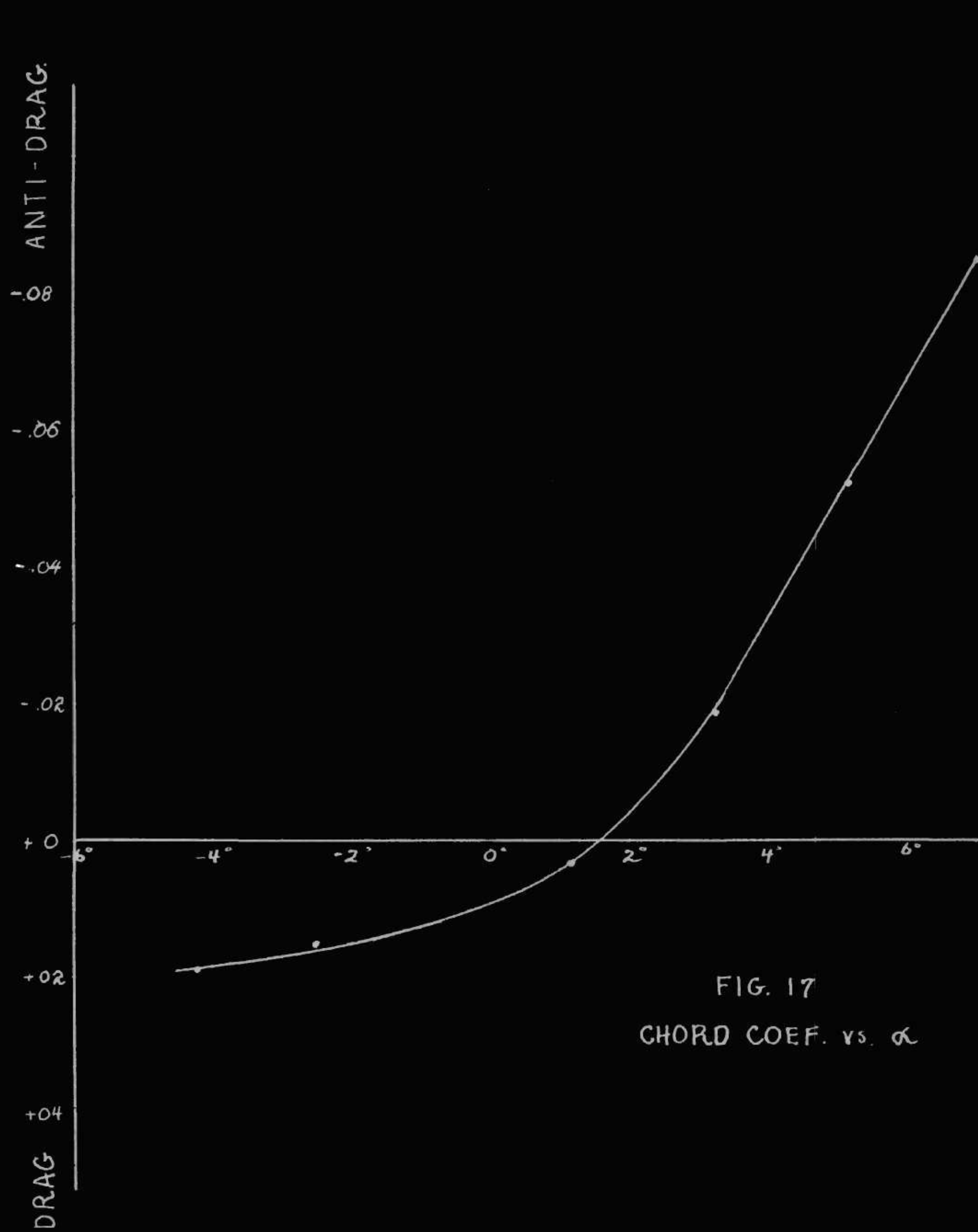


FIG. 17

CHORD COEF. vs. α

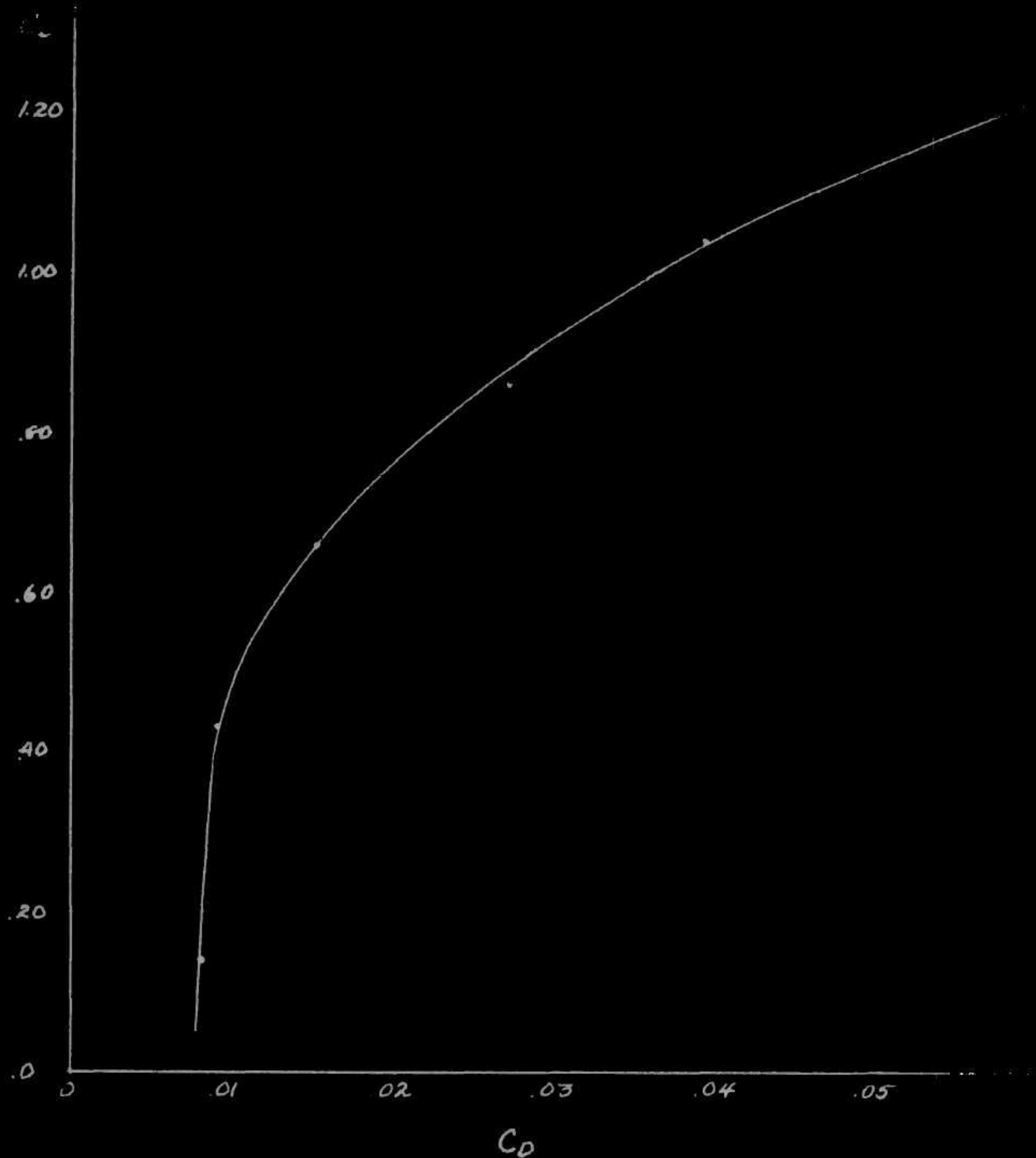


FIG 18

C_L VS C_D