"AN INVESTIGATION OF THE QUALITATIVE HIGH-LIFT CHARACTERISTICS OF THREE REPRESENTATIVE DAVIS AIRFOILS"

By Ernest M. Spear

Submitted in partial fulfillment of the requirements for the Bachelor of Science degree from the

MASSACHUSETTS INSTITUTE OF TECHNOLOGY

Signature Signature redacted

Acceptance:

Professor in charge of thesis: Signature redacted

Prof. Otto C. Koppen.

Building 33
Mass. Inst. of Techn.
Cambridge, Mass.
January9, I943.

Professor George W. Swett
Secretary of Faculty
Massachusetts Institute of Technology
Cambridge, Massachusetts.

Dear Sir:

Attached is the thesis, "An Investigation of the Qualitative High- Lift Characteristics of Three Representative Davis Airfoils," submitted in partial fulfillment ofthe requirements for the Bachelor of Science degree from the Massachusetts Institute of Technology, Aeronautical Department. It represents an attempt to determine whether certain properties in this family of uniquely-derived airfoils, observed in actual practice, are inherent in the entire series. The choice of specimens was such as to include the maximum practical range consistent with actual usage and experimental conditions.

I sincerely hope that this paper meets with your approval.

Very truly yours,

Signature redacted

Ernest M. Spear.

ACKNOWLEDGEMENTS

The author particularly wishes to express his thanks to Professor Otto C. Koppen, for the advisory and financial aid which he made possible. Professors John R. Markham and Shatswell J. Ober were of willing assistance in the details of testing technique. To Mr. James Maddox, I am grateful for his work in the construction of the models and fittings.

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SUMMARY

It may be reasonably concluded that the retention of maximum lift over a considerable range of angle of attack is not a characteristic inherent in the Davis airfoil series. It is further assumed that, should such a characteristic be exhibited by a member of the series, it represents a unique condition, or at most a condition limited to a narrow range of practical sections.

INTRODUCTION

In actual use, a Davis airfoil section has been noted to yield a retention of maximum lift over a considerable angular travel- giving a "flat-top" lift curve. Although this condition might conceivable have certain advantages in isolated instances, it is in general not a desirable arrangement- particularly as regards the conventional landing technique. In an effort to determine whether this observed characteristic was typical of the Davis family of airfoils, it was desired to test representative sections of the series, varying them by some uniform method. In an initial attempt, the obtaining of equal maximum thicknesses at different percentages of the chord seemed best.

An examination of Davis' equations and a typical section plot(see Fig.8, Appendix),shows the difficulties inherent in a mathematical analysis; the wing chord,which is the desired reference, assumes a unique orientation with respect to the plotting coordinate system for each of the endless (A)(B) combinations.

A successful mathematical expression was derived, however, but its enormity lent so little facility to direct use that it had to be abandoned for practical reasons.In its stead, 137 tedious plots of sections were made, covering a wide scope of parameters, with the resulting scaled measurements plotted and cross-plotted; involving as separate variables A, B, % thickness, chordal position. This yielded curves whose trends could be accurately predicted, and the desired sections obtained through extrapolation. This having finally been accomplished, it was discoveredthat the difference between any two airfoils, of similar thickness and in a practical region, was too slight to use a 6" chord and not create the apprehension that any similarities in results had arisen from small construction deviations. This method of approach was likewise abandoned.

The work entailed was not a complete loss, however, since the large accumulation of plotted section characteristics served as material for the selections eventually made.These selections, based on a constant B/A ratio, are given in the "Procedure", together with indications of why they were made.

APPARATUS

The tunnel employed was the M.I.T. 5' Wind Tunnel. Its principal characteristics appear in Fig.9, Appendix.

The three models were wings of 36" by 6" dimensions, giving an area of I.5 square feet and an aspect ratio of 6. Their section ordinates appear in Figs. 5,6,7, Appendix. These wings were strictly rectangular, with the ends rounded to diameters equalling the thickness at any chord station. Constructed of Philippine mahogany, they were surfaced with filler and waxed; furthermore being equipped with the fittings and sting peculiar to this type of suspension.(See Fig.IO, Appendix.)

The grid construction was simplicity itself, it consisting of 5/I6" brass rods, spaced I" on centers, in a 3' by 4' wooden frame.

PROCEDURE

With the premise being that tests were to be conducted on three Davis airfoils of constant B/A ratio, a choice was made, both of this ratio and of the magnitudes of the parameters for each wing. This understandably entailed some choice of range which was, to an extent, arbitrary. Davis' own suggestion, based on his knowledge of the subject- which is presumably the most extensive- recommends a B/A ratio lying between 18% and33%. The 25% value utilized seems justified in light of this consideration. Some slight knowledge by the author of Davis airfoils in actual use, plusexperience garnered in plotting inummerably varied sections, led to the selection of "A" values equalling 0.65, 0.75, and 0.85. It is felt that these, under the limited testing conditions, provided as extensive a practical range as possible- practical as regards both model construction and actual usage.

These three airfoils were then plotted, using the four equations derived by Davis(Consult Fig. 8,Appendix.)

These plots yielded three airfoils of varying size and with chord- lines(mathematical chords, that is, between intersections of foil with horizontal axis at nose and vertical axis at trailing edge) at angles to the plotting coordinates. To obtain usable data in their construction, these plots were greatly enlarged, with stations and ordinates then being scaled in directions perpendicular and parallel to the chord. By ratio of lengths, these numbers were all converted to airfoils of 6" chord, retaining the .00I" to which the data then become accurate, in the process. The plotting of these sections upon aluminum templates was facilitated by use of the graduations on a milling machine. (See Fig. 5,6,7, Appendix.)

In order to improve the data obtained, a turbulence grid was inserted in the tunnel about four feet before the model, yielding a higher effective Reynolds number. This tunnel is calibrated for I.A.S. vs. the static pressure differential between its interior and exterior. With such a calibration, it became obvious that it would not be correct, since the grid could not be placed in any position which would not interfere with the plate for measuring internal static pressure. With this in view, and because of the nature of the tests, a calibrated I.A.S. of 40 m.p.h. was employed- even then the motive power protesting some what over the grid's high drag. Subsequent checking with a pitot tube showed the actual speed to be 36.3 m.p.h. This value seems reliable, since it was taken over two feet behind the grid, in which the rods are of 5/I6" diameter. A further confirmation lies in the obtaining of an exact 40 m.p.h. pitot reading with the grid removed.

Each model was mounted and adjusted to obtain the parallelogram relationship typical of this Prandtl suspension. The displacement of zero balance angle and chord-line was noted.

Wind-off balance readings for all anticipated angles were recorded, as were the corresponding readings for wind-on. Following each of these, a study of the stall development across the wing was made by employing a multitude of small white threads attached lightly to the wing's upper surface. (See Fig. 4, Appendix.)

The above was repeated for all models. Tonote the effect of the grid, a second run of a similar nature was made for Wing #2 with the grid removed, and the tunnel calibration set for the speed which the pitot had previously indicated with the grid present.

DISCUSSION

This short investigation having as its purpose the determination of whether a characteristic was omnipresent in a series of airfoils, it remains sufficient to show that it does not exist in a few of that series to disprove the tacit assumption that it does.

I believe that, within the ability of these tests to do so, this negation has been accomplished. While it is true that these sections show reasonably good stalling properties throughout, even at this Reynolds number, it is apparent that none actually exhibit the characteristic in question. The Reynolds number of testing was approximately 600,000. What might be exhibitted at a higher Reynolds number is problematical. It is quite possible, as the tests with and without grid indicate, that a further increase in Reynolds number may smooth the curve out to a greater extent, but a continuous extension of lift seems doubtful.

There remain only the methods of this particular test to be considered. The grid, as Fig. 2 in the Appendix indicates, had no appreciable effect on $C_{L\,MAX}$, but it did serve to delay the stall drop-off, and create a smoother, flatter peak to the lift curve. In this comparison, the curves were displaced in position, but not slope, leading to the belief that some asymmetry in the grid or the flow had effectively altered α slightly. For this reason, an arbitrary shift was made for coincidence. The equality of slopes further justifies the accuracy of I.A.S. determinations, beyond that mentioned in the "Procedure."

For a given B/A ratio, the stalling properties seem to improve with increasing parameter, as does also the lift- all being reasonably good for these testing conditions. The progressions of stall over the surface (Fig. 4, Appendix) appear to be qualitatively quite similar, with some minor idiosyncracies- these probably attributable to fittings on the thinner wing. The high values of A and B delay the stall by several degrees. It should be noted that α measured from zero lift, and not the mathematical chord, would raise these stalling angles even higher.

The accuracy of all test measurements is about 1%- the qualitative nature of the work making this of lesser importance than is customary.

An oddity of the curves is noticed in the tendency of all to flatten, or "dimple", just before the peak. $\frac{dC_L}{d\alpha}$ is lower here, consistently, than anticipated for A.R.=6.

No further examination of this particular topic is suggested, other than tests of a similar nature at higher Reynolds numbers. The series of Davis airfoils are interesting, however, and the selection of a particular section for some express purpose may well warrant the tedious labor required to seek it out from the multitude of its counterparts.

METHODS of COMPUTATIONS

The differences in wind-on and wind-off readings of the three lift balances (3,4,5) are summed up for each angular setting. Since the constant for each balance is identical, multiplication of these summated differences by this constant yields lift in pounds.

For these wings:

$$S = \frac{36 \times 6}{144} = 1.5 \text{ FT.}^2$$

$$q = \frac{(36.3)^2}{391} = 3.355 \frac{\text{LB.}}{\text{FT.}^2} \qquad Sq = 5.04 \text{ LBS.}$$

Division of lifts by (Sq) yields the non-dimensional coefficient C_L:

$$C_L = \frac{L}{Sq}$$

Im a closed-throat tunnel of this type, the wall correction for induced change in angle of attack is positive and of the form

$$\Delta\alpha_i = \frac{\delta L}{qC}$$

where

$\Delta\alpha_i$ = induced angle

L = lift

q = dynamic pressure

C = tunnel X-sectional area

and δ is a non-dimensional factor, being a function of the tunnel shape.

For a circular cross-section

$$\delta = \frac{1}{8}\left[1 + \frac{3}{16}\left(\frac{S}{D}\right)^4 + \cdots\right]$$

Here S = *Wing span*

D = *Tunnel diameter*

For this tunnel:

$$\delta = \frac{1}{8}\left[1 + \frac{3}{16}\left(\frac{3.0}{5.0}\right)^4 + \cdots\right]$$

$$= 0.1306$$

then

$$\Delta\alpha_i \text{ (degrees)} = \frac{\delta L}{qC}(57.3)$$

$$= \frac{0.1306 \times 57.3}{3.355 \times \frac{\pi 25}{4}} L = 0.114 L$$

In plotting, it is only required to plot C_L vs. the correct angle, this being:

$$\alpha_C = \alpha_M + \Delta\alpha_i .$$

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David R. Davis----"Fluid Foil"

2. M.I.T. Lecture notes and papers- Aeronautical Dept.

3. "Pattern Making"- J. R. Stearns

APPENDIX

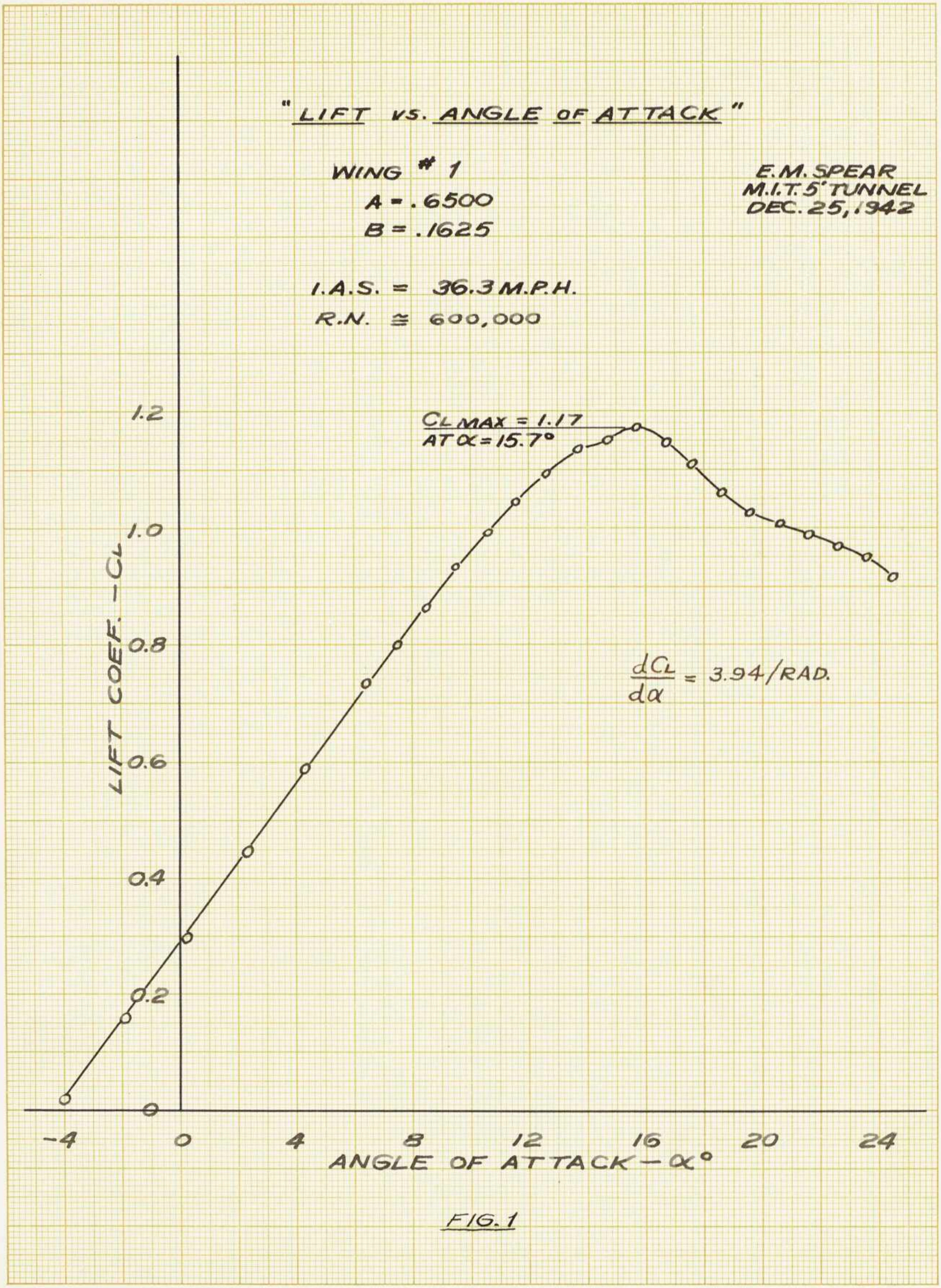

"LIFT VS. ANGLE OF ATTACK"
WING # 1
A = .6500
B = .1625
I.A.S. = 36.3 M.P.H.
R.N. ≅ 600,000
E.M. SPEAR
M.I.T. 5' TUNNEL
DEC. 25, 1942
CL MAX = 1.17
AT α = 15.7°
dCL/dα = 3.94/RAD.
LIFT COEF. – CL
1.2
1.0
0.8
0.6
0.4
0.2
0
-4
0
4
8
12
16
20
24
ANGLE OF ATTACK – α°

FIG. 1

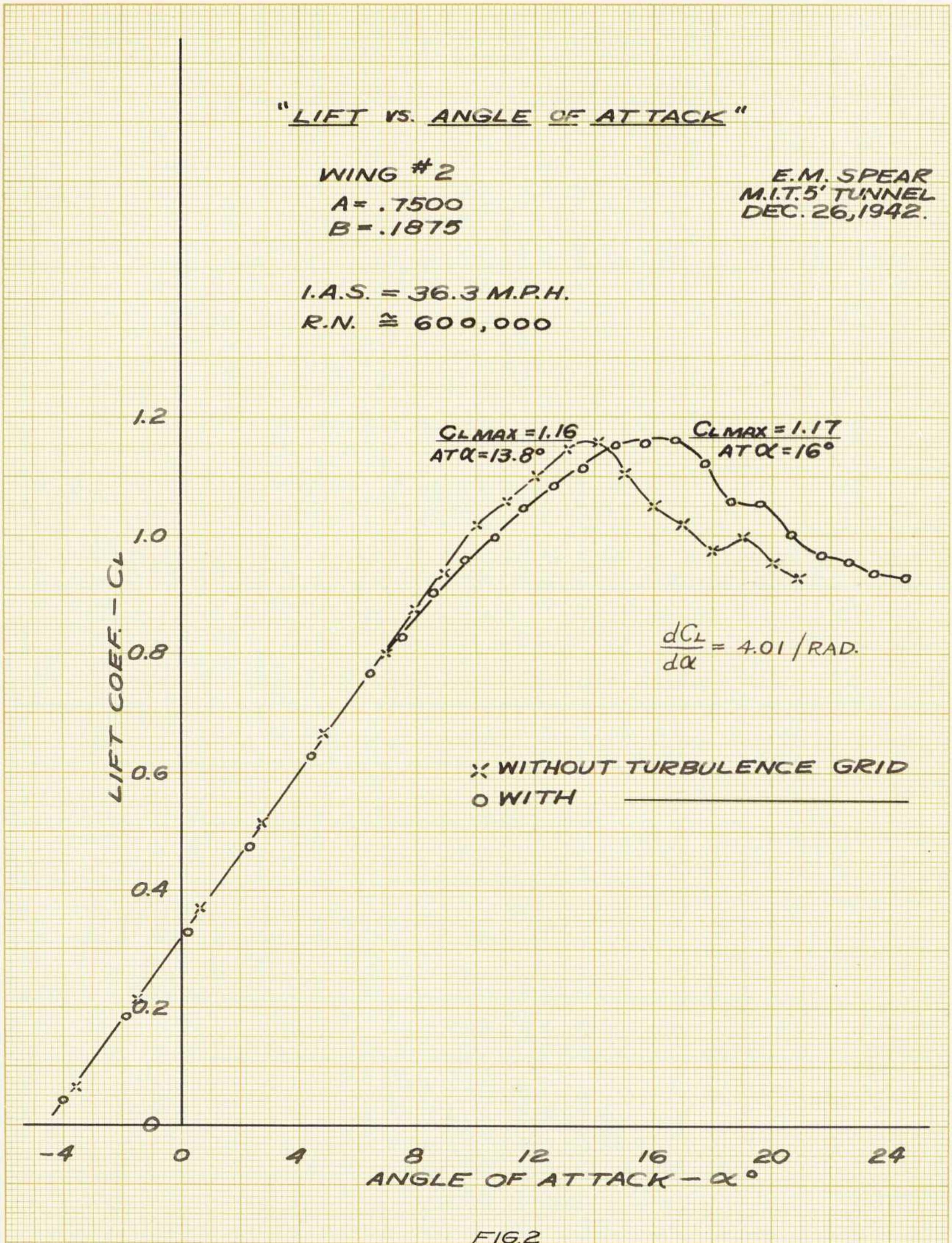

"LIFT VS. ANGLE OF ATTACK"
WING #2
A = .7500
B = .1875
E.M. SPEAR
M.I.T. 5' TUNNEL
DEC. 26, 1942.
I.A.S. = 36.3 M.P.H.
R.N. ≅ 600,000
CL MAX = 1.16
AT α = 13.8°
CL MAX = 1.17
AT α = 16°
dCL/dα = 4.01 /RAD.
× WITHOUT TURBULENCE GRID
o WITH
LIFT COEF. – CL
ANGLE OF ATTACK – α°
1.2
1.0
0.8
0.6
0.4
0.2
0
-4
0
4
8
12
16
20
24

FIG. 2

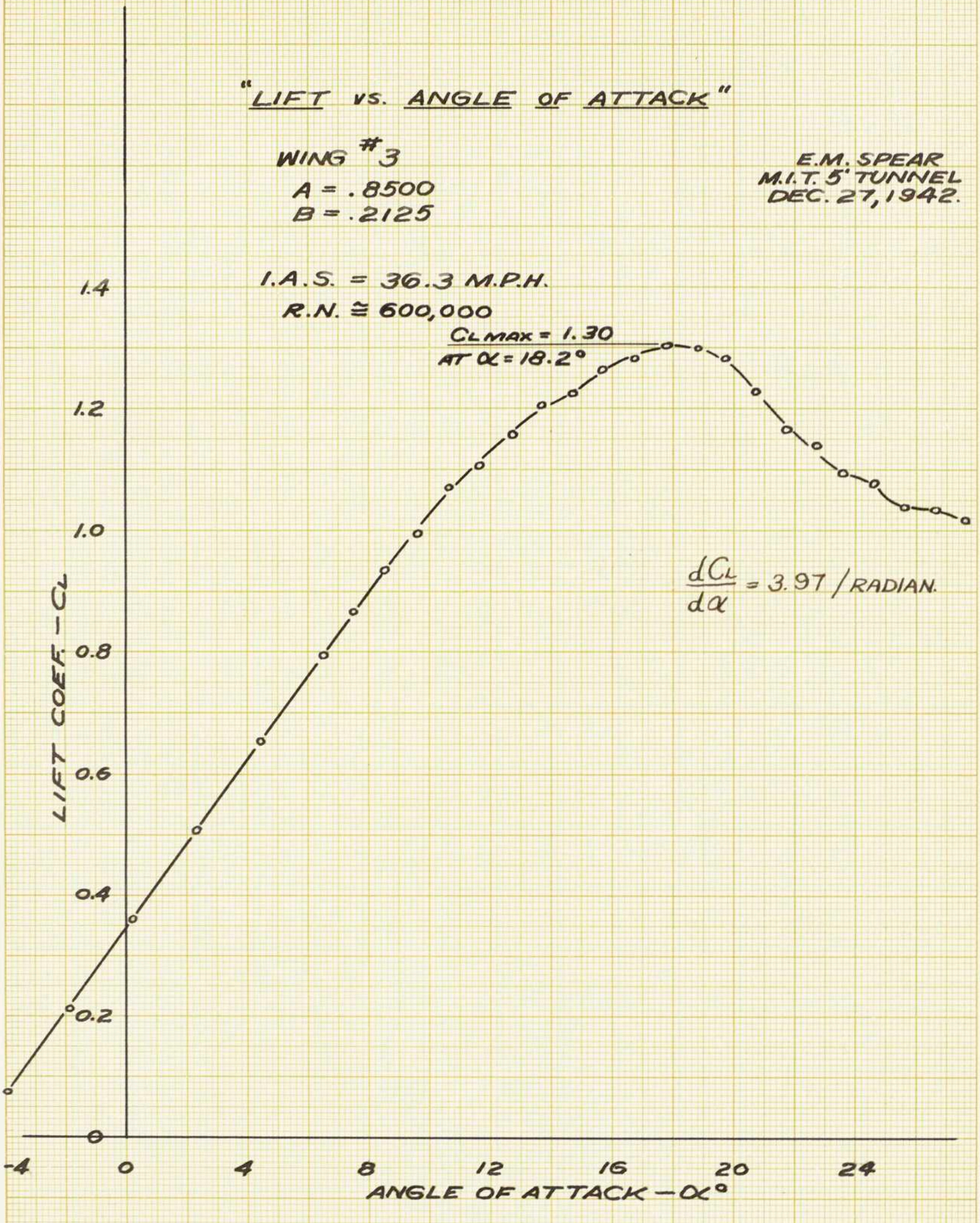
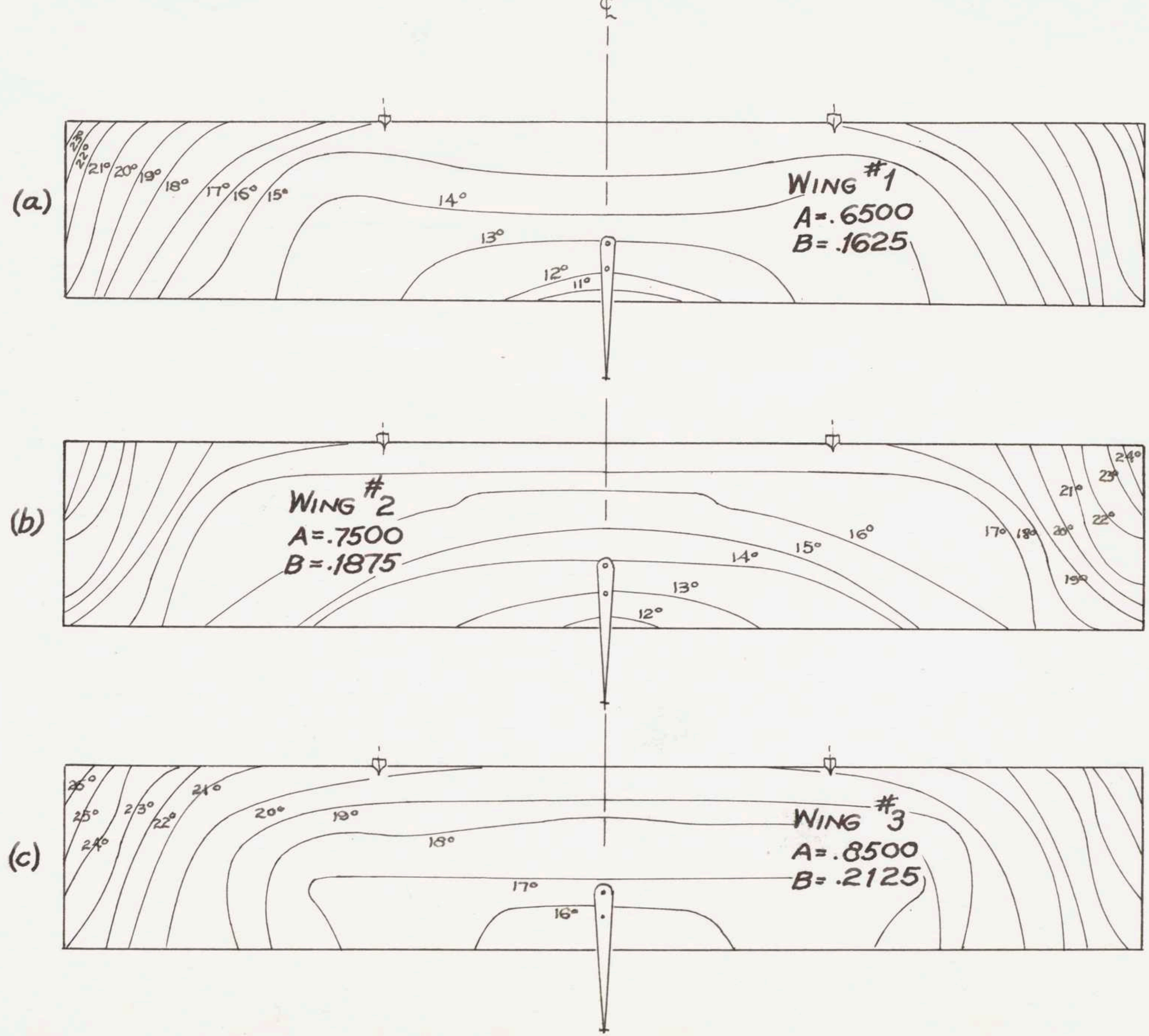

"LIFT VS. ANGLE OF ATTACK"
WING #3
A = .8500
B = .2125
E.M. SPEAR
M.I.T. 5' TUNNEL
DEC. 27, 1942.
I.A.S. = 36.3 M.P.H.
R.N. ≅ 600,000
$C_{L\,MAX}$ = 1.30
AT α = 18.2°
$\frac{dC_L}{d\alpha}$ = 3.97 / RADIAN.
LIFT COEF. – C_L
0
0.2
0.4
0.6
0.8
1.0
1.2
1.4
-4
0
4
8
12
16
20
24
ANGLE OF ATTACK – α°

FIG. 3

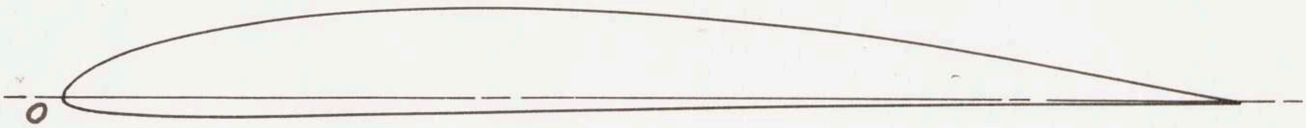


STALL DEVELOPMENT BY TUFT STUDIES

FIG. 4

WING #1

A = .6500
B = .1625

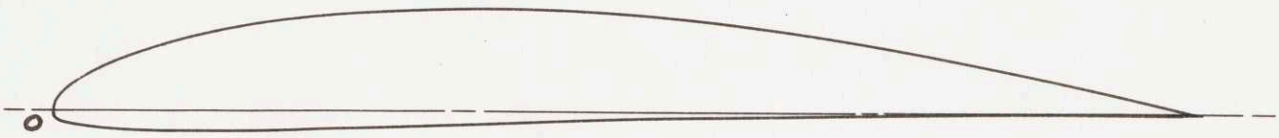


STATION (IN.)	UPPER SURFACE (IN.)	LOWER SURFACE (IN.)
0.000	0.000	0.000
0.013	0.060	0.023
0.055	0.104	0.050
0.109	0.141	0.067
0.154	0.166	0.075
0.207	0.192	0.082
0.298	0.231	0.092
0.422	0.276	0.100
0.609	0.330	0.104
0.865	0.385	0.104
1.191	0.432	0.097
1.663	0.472	0.083
2.205	0.485	0.072
2.670	0.474	0.057
3.192	0.442	0.040
3.843	0.375	0.028
4.514	0.283	0.017
5.254	0.151	0.007
6.000	0.000	0.000

FIG. 5

WING #2

A=.7500
B=.1875

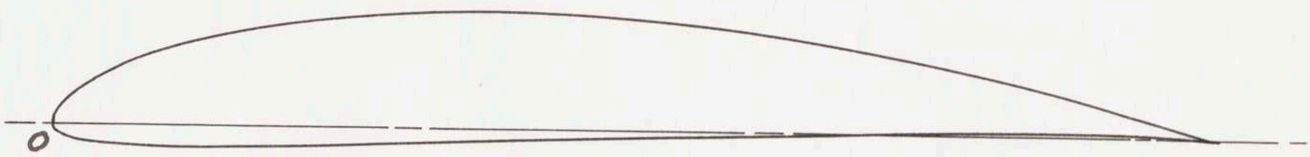


STATION (IN.)	UPPER SURFACE (IN.)	LOWER SURFACE (IN.)
0.000	0.000	0.000
0.029	0.085	0.033
0.057	0.119	0.051
0.118	0.167	0.072
0.209	0.222	0.092
0.399	0.303	0.110
0.680	0.393	0.113
1.071	0.471	0.110
1.430	0.517	0.097
1.901	0.550	0.080
2.401	0.556	0.061
2.889	0.537	0.044
3.455	0.484	0.029
4.171	0.358	0.010
5.135	0.204	0.000
6.000	0.000	0.000

FIG. 6

WING #3

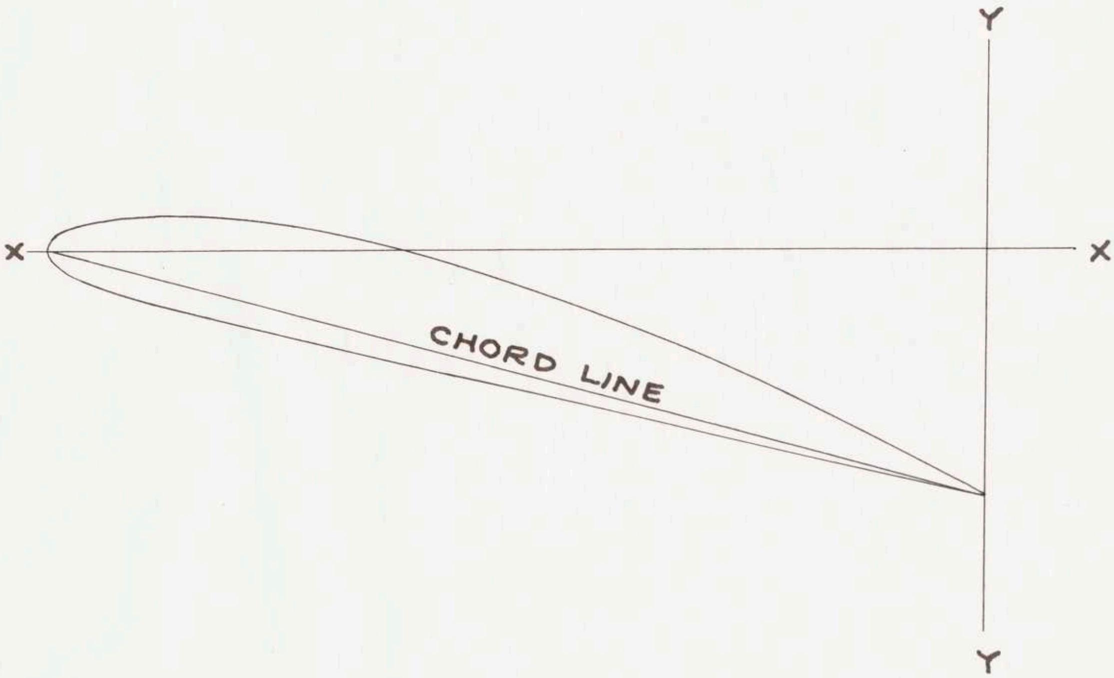
A= .8500
B=.2125



STATION (IN.)	UPPER SURFACE (IN.)	LOWER SURFACE (IN.)
0.000	0.000	0.000
0.013	0.075	0.021
0.043	0.123	0.050
0.085	0.163	0.069
0.139	0.203	0.085
0.198	0.242	0.098
0.259	0.274	0.106
0.400	0.338	0.118
0.579	0.403	0.122
0.805	0.469	0.118
1.107	0.533	0.110
1.462	0.582	0.096
1.913	0.617	0.077
2.342	0.625	0.056
2.770	0.613	0.038
3.209	0.579	0.022
3.545	0.542	0.013
3.951	0.483	0.002
4.356	0.413	-0.003
4.732	0.334	-0.006
5.266	0.218	-0.006
6.000	0.000	0.000

FIG. 7

EQUATIONS OF DAVIS AIRFOIL AND TYPICAL SECTION



$$X_S = \sin\theta\left[0.6366198(A-B)+B\right] + \tan\theta\left[1-0.6366198\,\theta\right](1-A)$$

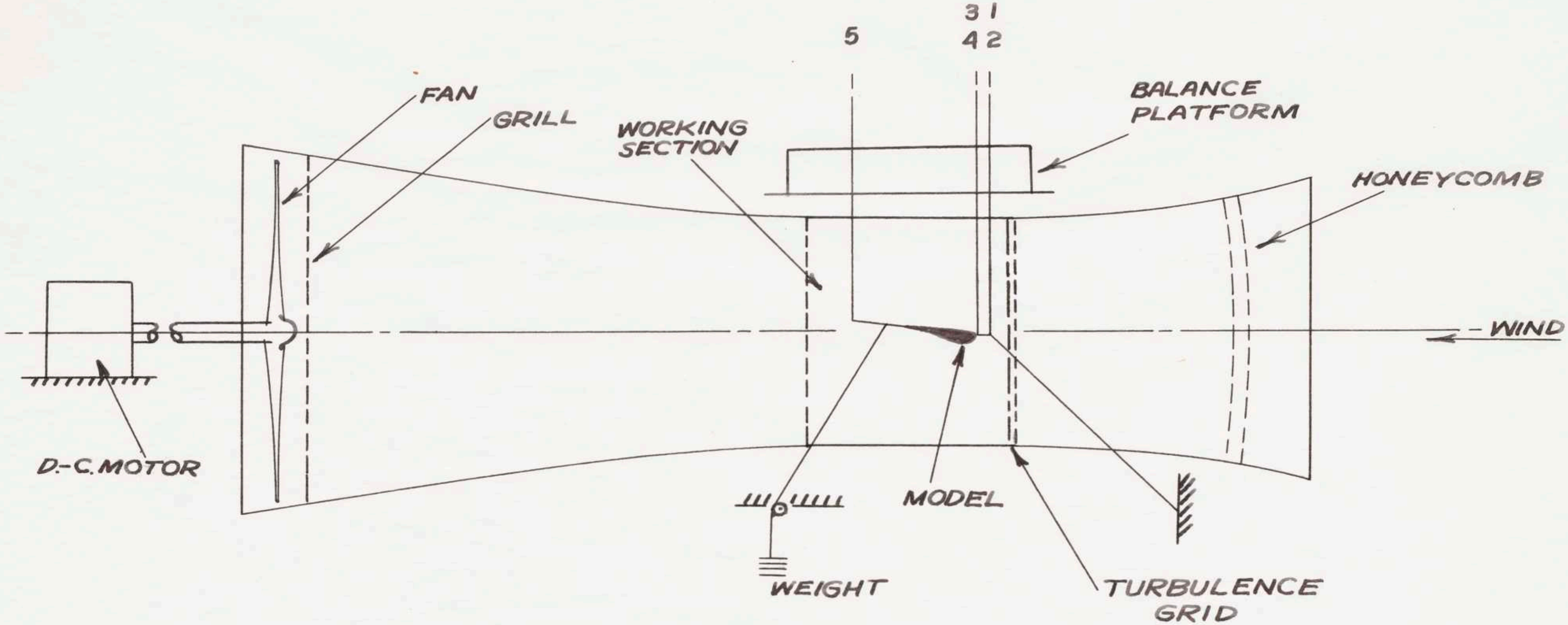
$$Y_S = \cos\theta\left[0.6366198(A-B)+B\right] - A\left[1-0.6366198\,\theta\right]$$

$$X_P = \sin\theta\left[0.6366198(A-B)+B\right] + \tan\theta\left[1-0.6366198\,\theta\right](1-A)$$

$$Y_P = \cos\theta\left[0.6366198(A-B)-B\right] - (A-2B)\left[1-0.6366198\,\theta\right]$$

θ = FROM 0 TO 1.5708

FIG. 8



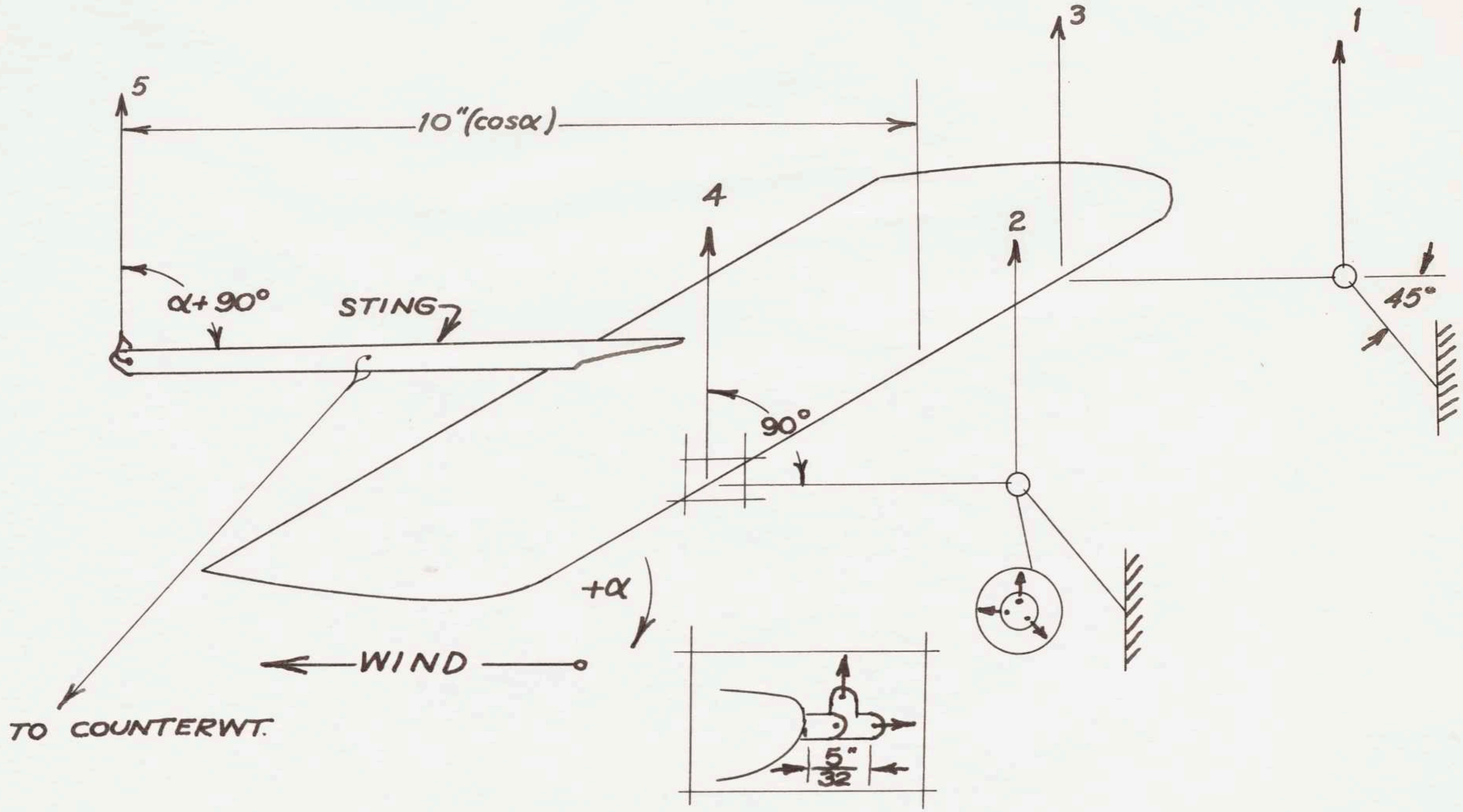
DIAGRAMMATIC ARRANGEMENT

M. I. T. FIVE-FOOT WIND TUNNEL

MOTOR: 75 H.P. - D.C. - WARD - LEONARD CONTROL

FAN: WOOD - 4 BLADE D= 9.67' b= 9"

FIG. 9



DETAILS OF MODEL
SUSPENSION

FIG. 10

M.I.T. 5' Tunnel.
Dec. 25, 1942.
I.A.S.- 36.3 m.p.h.

ORIGINAL DATA
WING #1

$\alpha_o = -4^\circ$ i.e., -4° on balance = 0° wing chord line.

Balance Readings

	Wind-off			Wind-on		
α_T	3	4	5	3	4	5
-8	153.1	185.8	408.9	152.2	170.3	431.3
-6	150.2	183.6	413.1	179.3	200.5	446.7
-4	149.0	181.6	417.6	205.4	229.7	462.4
-2	146.8	179.1	422.3	236.0	259.1	478.1
0	145.2	177.6	426.8	264.3	288.1	494.2
2	143.5	175.4	431.3	293.4	316.4	509.3
3	142.5	174.3	433.0	306.5	329.2	517.3
4	141.4	173.5	435.7	319.5	343.6	523.5
5	139.9	172.6	438.1	331.9	356.8	531.8
6	139.4	171.5	439.8	343.4	369.2	537.2
7	137.6	170.7	442.2	354.4	379.8	544.2
8	136.8	169.6	444.4	364.2	388.3	548.5
9	135.7	168.6	446.4	369.6	398.4	554.6
10	135.2	167.8	448.6	367.5	406.8	556.8
11	133.8	166.7	451.1	370.9	408.3	562.6
12	133.2	165.5	453.1	359.4	401.6	568.8
13	131.3	163.4	457.9	326.4	376.7	584.0
14	132.8	164.5	455.5	344.0	388.8	579.3
15	130.1	162.5	460.0	308.7	373.8	586.8
16	129.1	161.1	461.8	307.0	358.3	593.3
17	128.3	160.1	464.3	297.6	354.2	597.8
18	127.7	159.1	466.2	298.0	341.5	600.8
19	126.2	157.9	468.9	292.8	334.0	603.9
20	125.3	156.6	470.6	284.5	318.8	609.7

M.I.T. 5' Tunnel
Dec. 27, 1942.
I.A.S.-36.3 M.P.H.

ORIGINAL DATA - WING #2.

$\alpha_o = -4°$ i.e., $-4°$ on balance = $0°$ wing chord-line.

Balance Readings

α_T	Wind-off 3	Wind-off 4	Wind-off 5	Wind-on 3	Wind-on 4	Wind-on 5
-8	161.2	197.1	410.7	160.9	187.5	441.2
-6	159.4	195.2	415.5	189.6	217.0	455.8
-4	157.9	192.8	420.1	217.1	246.4	472.6
-2	154.9	191.0	426.1	248.5	274.9	488.2
0	153.0	189.0	429.2	277.7	306.0	504.5
2	151.1	187.2	433.5	305.6	334.2	519.0
3	150.8	186.3	436.0	316.8	347.1	526.2
4	149.2	185.2	438.5	332.2	360.8	534.4
5	148.9	184.0	440.4	342.0	374.9	540.4
6	147.6	183.2	442.7	348.0	383.1	545.2
7	147.1	182.1	444.4	361.0	391.1	550.2
8	146.1	181.2	446.8	365.3	402.0	553.9
9	145.1	180.9	448.8	368.8	408.1	564.0
10	144.2	179.2	451.2	378.6	412.6	564.0
11	143.2	178.0	453.5	374.5	414.8	566.8
12	141.6	176.9	455.5	379.1	411.2	572.2
13	140.6	176.1	458.2	367.9	393.3	578.8
14	139.3	175.1	460.0	339.7	383.6	585.2
15	138.6	173.7	462.5	337.0	380.4	591.0
16	138.1	172.6	464.5	313.6	371.5	594.3
17	136.5	171.6	466.5	301.6	359.2	600.5
18	135.9	170.6	468.9	303.4	349.0	604.7
19	134.2	169.2	471.5	292.0	348.9	606.8
20	133.7	168.4	473.3	290.8	342.4	610.8

M.I.T. 5' Tunnel
Jan. 7, 1943.
I.A.S.- 36.3 m.p.h.

ORIGINAL DATA

Wing #2- without turbulence grid.

$\alpha_o = -4^\circ$ i.e., -4° on balance = 0° wing chord-line.

Balance Readings

	Wind-off			Wind-on		
α_T	3	4	5	3	4	5
-8	152.5	184.5	408.8	163.0	176.6	438.2
-6	151.1	182.2	413.6	192.1	208.2	453.8
-4	148.7	180.1	418.7	225.1	238.2	470.8
-2	147.4	178.5	422.6	253.0	268.7	485.4
0	145.0	176.5	427.3	284.8	298.1	500.4
2	143.2	174.3	432.3	309.6	328.4	514.8
3	142.3	173.5	432.6	326.4	341.4	523.4
4	141.2	172.4	434.9	337.9	353.6	528.7
5	140.4	171.4	437.1	352.9	369.4	538.0
6	139.1	170.3	439.5	357.4	381.9	542.6
7	138.5	169.3	441.7	364.6	391.2	547.2
8	137.2	168.3	443.7	375.8	397.9	553.3
9	136.3	167.4	446.4	373.5	404.4	555.8
10	135.7	166.3	448.2	365.1	412.1	559.6
11	135.1	165.2	450.6	332.4	378.5	568.3
12	134.4	164.5	453.3	326.4	366.2	573.7
13	133.4	163.3	455.4	309.1	363.6	574.1
14	132.4	162.3	457.0	304.8	367.0	583.0
15	130.9	161.2	459.8	292.7	351.4	588.5
16	130.6	160.2	462.1	283.3	346.5	591.4

M.I.T. 5' Wind Tunnel
Dec. 26, 1942.
I.A.S. - 36.3 m.p.h.

ORIGINAL DATA

Wing #3

$\alpha_o = -5^\circ$ i.e., -5° on balance = 0° wing chord-line

Balance Readings

	Wind-off			Wind-on		
α_T	3	4	5	3	4	5
-9	168.8	199.2	408.3	173.0	192.5	448.0
-7	167.1	197.0	414.2	198.5	222.4	463.9
-5	165.0	194.7	419.0	229.6	251.3	480.7
-3	163.0	192.9	423.7	258.8	279.8	497.1
-1	161.6	191.0	428.4	287.2	309.9	512.7
1	159.2	188.9	433.3	314.8	338.7	528.1
2	158.1	187.8	435.4	330.2	352.3	536.0
3	157.2	187.0	437.4	341.1	367.3	543.1
4	156.2	185.6	439.4	354.4	378.9	549.8
5	154.4	184.8	442.2	365.0	389.6	556.8
6	154.3	183.6	443.8	374.5	402.4	562.9
7	153.0	182.7	446.4	385.7	412.1	568.0
8	152.1	181.7	449.0	389.4	419.3	572.1
9	151.5	180.7	451.2	393.7	428.7	577.4
10	150.1	179.5	453.9	403.1	435.6	582.4
11	149.0	178.4	455.6	405.0	437.8	586.3
12	148.0	177.5	457.6	402.7	444.8	591.4
13	146.4	176.4	460.2	399.6	442.9	595.1
14	146.1	175.4	462.5	393.9	438.2	598.2
15	144.3	174.2	464.8	366.6	432.2	603.8
16	144.1	173.2	467.2	354.9	406.3	610.1
17	142.8	172.0	469.2	347.0	397.7	613.5
18	141.7	170.8	471.6	318.6	400.5	616.4
19	140.0	169.7	473.4	311.0	395.7	620.7
20	139.5	168.8	474.4	313.8	366.4	625.3
21	138.4	167.7	477.0	311.1	362.3	630.5
22	136.7	166.4	480.0	309.8	349.6	636.7

SAMPLE COMPUTATIONS

For wing #3 :

At $\alpha_M = 18°$, balances- wind off- read:

3	4	5
146.4	176.4	460.2

wind on readings:

399.6	442.9	595.1

Δ readings:

253.2	266.5	134.9

$\Sigma \Delta$ readings = 654.6

- in pounds = 6.546

Sq = 5.04

$$C_L = \frac{6.546}{5.04} = 1.298$$

$$\Delta\alpha_i = 0.114\ L = 0.114 \times 6.546 = 0.8°$$

$$\alpha_c = 18° + 0.8° = 18.8°$$

WING#1

TABULATED COMPUTATIONS

α_c	Δ 3	Δ 4	Δ 5	Σ(3,4,5) (Lbs.)	C_L
-4.0	-0.9	-15.5	22.4	0.06	.019
-1.9	29.1	16.9	33.6	0.796	.158
0.2	56.4	48.1	44.8	1.493	.296
2.3	89.2	55.8	80.0	2.250	.447
4.3	119.1	110.5	67.4	2.970	.589
6.4	149.9	141.0	78.0	3.689	.732
7.5	164.0	154.9	84.3	4.032	.799
8.5	178.1	170.1	87.8	4.360	.865
9.5	192.0	184.2	93.7	4.699	.932
10.6	204.0	197.7	97.4	4.991	.990
11.6	216.8	209.1	102.0	5.279	1.046
12.6	227.4	218.7	104.1	5.502	1.092
13.7	233.9	229.8	108.2	5.719	1.135
14.7	232.3	239.0	108.2	5.795	1.150
15.6	237.1	241.6	111.5	5.902	1.172
16.6	226.2	236.1	115.7	5.780	1.147
17.6	211.2	224.3	123.8	5.593	1.109
18.6	195.1	213.3	126.1	5.345	1.060
19.6	178.6	211.3	126.8	5.167	1.026
20.6	177.9	197.2	131.5	5.066	1.005
21.6	169.3	194.1	133.5	4.969	.987
22.6	170.3	182.4	134.6	4.873	.968
23.6	166.6	176.1	135.0	4.777	.948
24.5	159.2	162.2	139.1	4.605	.914

WING # 2

TABULATED COMPUTATIONS

α_c	Δ 3	Δ 4	Δ 5	$\Sigma\Delta$ (3,4,5.)Lbs.	C_L
-4.0	-0.3	-9.6	30.5	0.206	.041
-1.9	30.2	21.8	40.3	0.923	.183
0.2	59.2	53.6	52.5	1.653	.328
2.3	93.6	83.9	62.1	2.396	.475
4.4	124.7	117.0	75.3	3.170	.628
6.4	154.5	147.0	85.5	3.870	.767
7.5	166.0	160.8	90.2	4.170	.827
8.5	183.0	175.6	95.9	4.545	.901
9.6	193.1	190.9	100.0	5.028	.997
10.6	200.4	199.9	102.5	4.840	.959
11.6	213.9	209.0	105.8	5.287	1.048
12.6	219.2	220.8	107.1	5.471	1.085
13.6	223.7	227.2	111.3	5.622	1.116
14.7	234.4	233.4	112.8	5.806	1.152
15.7	231.3	236.8	113.3	5.813	1.153
16.7	237.5	234.3	116.7	5.885	1.160
17.7	227.3	217.2	120.6	5.651	1.121
18.6	200.4	208.5	125.1	5.340	1.059
19.6	198.0	206.7	128.5	5.332	1.056
20.6	175.5	198.9	129.8	5.042	1.000
21.6	165.1	187.6	134.0	4.867	.966
22.6	167.5	178.4	135.8	4.817	.955
23.5	157.8	179.7	135.3	4.728	.936
24.5	157.0	174.0	137.5	4.685	.929

TABULATED COMPUTATIONS

Wing # 2 - without turbulence grid.

α_c	Δ 3	Δ 4	Δ 5	$\Sigma\Delta$(3,4,5.)Lbs.	C_L
-4.0	10.5	-7.9	29.4	0.32	.063
-1.9	41.0	26.0	40.2	1.072	.213
0.2	76.4	58.1	52.1	1.866	.370
2.3	105.4	90.2	62.8	2.584	.513
4.4	139.8	121.6	73.1	3.345	.664
6.5	166.4	154.1	82.5	4.030	.798
7.5	184.1	167.9	90.8	4.428	.877
8.5	196.7	181.2	93.8	4.717	.936
9.6	212.5	198.0	100.9	5.114	1.018
10.6	218.3	211.5	103.1	5.329	1.058
11.6	226.1	221.9	105.5	5.535	1.100
12.7	238.6	229.6	109.6	5.778	1.147
13.7	237.2	237.0	109.4	5.826	1.158
14.6	198.8	245.8	111.4	5.560	1.103
15.6	197.3	213.3	111.7	5.283	1.050
16.6	192.0	201.7	120.4	5.141	1.020
17.6	175.7	200.3	118.7	4.947	.973
18.6	172.4	204.7	126.0	5.031	.998
19.6	161.8	190.2	128.7	4.807	.954
20.5	152.8	186.3	129.3	4.684	.928

TABULATED COMPUTATIONS

Wing # 3

α_c	Δ 3	Δ 4	Δ 5	$\Sigma\Delta$(3,4,5)Lbs.	C_L
-4.0	4.2	-6.7	39.7	0.372	.074
-1.9	31.4	25.4	49.7	1.065	.211
0.2	64.6	56.6	61.7	1.829	.363
2.3	95.8	86.9	73.4	2.563	.508
4.4	125.6	118.9	84.3	3.288	.653
6.5	155.6	149.8	94.8	4.002	.794
7.5	172.1	164.5	100.6	4.372	.868
8.5	183.9	180.3	105.7	4.699	.932
9.6	198.2	193.2	110.4	5.018	.996
10.6	210.6	204.8	114.6	5.400	1.071
11.6	220.2	218.8	119.1	5.581	1.107
12.7	232.7	229.4	121.6	5.837	1.158
13.7	237.3	237.6	123.1	6.080	1.206
14.7	242.2	248.0	126.2	6.164	1.223
15.7	253.0	256.1	128.5	6.376	1.265
16.7	256.0	259.4	130.7	6.458	1.282
17.8	254.7	267.3	133.8	6.558	1.302
18.8	253.2	266.5	134.9	6.546	1.298
19.7	247.8	262.8	135.7	6.463	1.282
20.7	222.3	258.0	139.0	6.193	1.227
21.7	210.8	233.1	142.9	5.868	1.164
22.7	204.2	225.7	144.3	5.742	1.138
23.6	176.9	229.7	144.8	5.514	1.093
24.6	171.0	226.0	147.3	5.443	1.078
25.6	174.3	197.5	150.9	5.227	1.037
26.6	172.7	194.7	153.5	5.209	1.033
27.6	173.1	183.2	156.7	5.130	1.017