TECHNICAL NOTES

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

No. 219

THE COMPARISON OF WELL-KNOWN AND NEW WING SECTIONS TESTED IN THE VARIABLE DENSITY WIND TUNNEL.

By George J. Higgins,
Langley Memorial Aeronautical Laboratory.

May, 1925.

48

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

TECHNICAL NOTE NO. 219.

THE COMPARISON OF WELL-KNOWN AND NEW WING SECTIONS TESTED IN THE VARIABLE DENSITY WIND TUNNEL.

By George J. Higgins.

Introduction

There have been tested up to now in the Variable Density Wind Tunnel of the National Advisory Committee for Aeronautics three groups of airfoils, a total of thirty-seven sections in all. The first group (Reference 1) contains three airfoils: N.A.C.A. 97, 98, and 99. The first two are sections of equal camber and thickness, with a sharp and a rounded trailing edge respectively, and the third is a symmetrical section of about the same thickness. The second group is a systematic series of twenty-seven airfoils: N.A.C.A. M-1 to M-27, characterized by a small travel of the center of pressure. The third group consists of several frequently used wing sections: U.S.A. 5, U.S.A. 27, U.S.A. 35A, U.S.A. 35B, R.A.F. 15, Clark Y, and Göttingen 387. The first and third groups were tested at five tank pressures each. The second group was tested at twenty atmospheres only.

Representation of the Results

The complete results of these tests will be published as N.A.C.A. Technical Reports. The discussion in this paper is based on the data of these reports. The following symbols will be used:

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

$$C_D = \frac{D}{qS}$$

$$C_M = \frac{M}{qcS}$$ (Moments taken about a point at 25% of the chord.)

$$C.P. = 0.25c - \frac{C_M}{C_L \cos\alpha + C_D \sin\alpha}$$

$$C_{D_i} = \frac{C_L^2\ S}{\pi\ b^2}$$

$$R.N. = \frac{\rho V l}{\mu}$$

where

C_L = Absolute lift coefficient.

C_D = Absolute drag coefficient.

C_M = Absolute moment coefficient.

C.P. = Center of pressure.

C_{D_i} = Induced drag coefficient

R.N. = Reynolds number.

L = Lift.

D = Drag.

M = Moment about a point at 25% of the chord.

q = Dynamic pressure, $\frac{1}{2}\rho V^2$.

V = Velocity.

ρ = Density of air.

c = Chord of airfoil.

b = Span of airfoil

S = Area of airfoil.

α = Angle of attack of airfoil.

l = Arbitrary length = c.

μ = Coefficient of viscosity.

The drag coefficients and the angles of attack have not been corrected for the influence of tunnel walls, if any.

The characteristics of a few of these airfoils are given as obtained from the tests made at twenty atmospheres tank pressure (See Figs. 1-10 inclusive, and Table IX). They are plotted in two different ways, as polar curves and with coefficients against the angle of attack.

Power coefficients versus speed range are plotted on Figs. 11, 12, 13 and 14, for several airfoils for comparison.

The effect of change of scale or Reynolds number upon C_L (max.), C_D (min.), and maximum L/D ratio on ten airfoils is shown in Figs. 15 to 20 inclusive. Cross sections of the airfoils are shown on all charts.

Tables I to VIII give the C_L (max.), C_D (min.), maximum L/D ratio, and $C_L(\text{max.})/C_D(\text{min.})$ ratio values on these sections at a low and a high $\frac{\rho V l}{\mu}$, with the airfoils arranged in the order of their merit.

Discussion of the Results.

The airfoils tested at five tank pressures give interesting information about the influence of the scale or of the Reynolds number (See Figs. 15-20 inclusive). The minimum drag coefficient is materially decreased at Reynolds numbers higher than obtained in an atmospheric tunnel. This minimum drag coefficient possibly approaches a constant value toward the upper end of the range of the Variable Density Tunnel. The decrease is more marked on thicker sections. The scale effect on the lift/drag ratio is erratic and is of lesser amount; in general, the lift/drag ratio increases in magnitude. This increase is more uniform for thick sections than for thin ones. The change of the maximum lift coefficient with scale does not seem to follow any definite law, but is uncertain, differing with each section. The result shown by the curve C_L(max.) vs. R.N. for the N.A.C.A. 99 airfoil (Fig. 16), is particularly interesting in regard to the peculiar hump. Tests in other tunnels have indicated the rise in the curve at low Reynolds number (R.& M. No. 928 - R.A.F. 30).

From Tables I to VIII, it can be seen that an airfoil section which is good at a low Reynolds number may be decidedly inferior at a high $\frac{\rho V l}{\mu}$. Full scale tests have also shown this to be true.

The power coefficient changes with scale as well as the other characteristics. This can be seen from the curves of power coefficient versus speed range shown in Fig. 14 for a R.A.F. 15 airfoil, tested at one and twenty atmospheres. A big decrease in power per unit weight is seen to occur at the higher speed ranges.

In addition to the series of frequently used sections, there are several good sections in the group of the 27 wing sections. The most notable of these are: the N.A.C.A. M-4, M-6, and M-12. The M-4 airfoil is of the thin type having a slight amount of S-shape. It compares favorably with the R.A.F. 15 as a high speed section (Fig. 11). The drag is very low and the lift/drag ratio is high. Its pitching moment about the quarter chord point is constant and small. The center of pressure travel is consequently small. The M-6 airfoil is a moderately thick section with both upper and lower surfaces convex, having its mean camber line slightly S-shaped. The drag is low, the lift, good, and the lift/drag ratio, high; the moment is constant and small. The center of pressure travel is likewise small. Both this section and the M-4 are stable in pitch. It compares very favorably with the R.A.F. 15 and because of its greater thickness is better adapted to internal bracing. M-6 is apparently better than the U.S.A. 35B, Clark Y, and the U.S.A. 27 wing sections of about the same thickness. (See Figs. 12 and 13). The M-12 airfoil is also a moderately thick section, but with a slight mean camber and no S-shape. Its characteristics are good though slightly poorer than the M-6 and about the same as the U.S.A. 35B and Clark Y. Its moment about the quarter chord point is practically zero, making the center of pressure travel very small.

Conclusion

The few tests made up to now in the Variable Density Wind Tunnel with wing sections show the great value of this type of wind tunnel for gaining information about wing sections valuable for use in aeronautical practice. Several new and good wing sections have been found. It has further been demonstrated that similar research work in ordinary atmospheric wind tunnels gives results less reliable. The results on scale effect show the need of extensive systematic research along that line.

Reference

1. Max M. Munk: Preliminary Wing Model Tests in the Variable Density Wind Tunnel of the N.A.C.A. Technical Report No. 217. 1925.

Table I.

Maximum Lift Coefficient.

R.N. - 200,000			R.N. - 3,600,000		
Order	Name	C_L (max.)	Order	Name	C_L (max.)
1	U.S.A. 35A	1.57	1	U.S.A. 27	1.39
2	Gott. 387	1.42	2	N.A.C.A. 97	1.38+
3	N.A.C.A. 98	1.40	3	Clark Y	1.38
4	N.A.C.A. 97	1.36	4	N.A.C.A. 98	1.38-
5	U.S.A. 27	1.31	5	U.S.A. 35B	1.37
6	U.S.A. 35B	1.24	6	Gott. 387	1.33
7	Clark Y	1.21	7	R.A.F 15	1.24
8	U.S.A. 5	1.12	8	U.S.A. 35A	1.21
9	R.A.F. 15	1.07	9	U.S.A. 5	1.19
10	N.A.C.A. 99	0.75	10	N.A.C.A. 99	0.98

Table II.

Name	R.N.	C_L
N.A.C.A. M-4	3,680,000	0.98
N.A.C.A. M-6	3,660,000	1.22
N.A.C.A. M-12	3,800,000	1.30

Table III.

Minimum Drag Coefficient.

R.N. - 200,000			R.N. - 3,600,000		
Order	Name	C_D (min.)	Order	Name	C_D (min.)
1	R.A.F. 15	0.010	1	R.A.F. 15	0.009
2	Clark Y	0.012	2	U.S.A. 35B	0.009
3	U.S.A. 5	0.014	3	N.A.C.A. 99	0.010
4	N.A.C.A. 99	0.017	4	Clark Y	0.010
5	U.S.A. 27	0.018	5	U.S.A. 5	0.011
6	U.S.A. 35B	0.019	6	N.A.C.A . 97	0.011
7	N.A.C.A. 97	0.022	7	U.S.A. 27	0.012
8	U.S.A. 35A	0.022	8	Gott. 387	0.012
9	Gott. 387	0.024	9	U.S.A. 35A	0.014
10	N.A.C.A. 98	0.027	10	N.A.C.A. 98	0.015

Table IV.

Name	R.N.	C_D(min.)
N.A.C.A. M-4	3,680,000	0.0075
N.A.C.A. M-6	3,660,000	0.0085
N.A.C.A. M-12	3,800,000	0.0090

Table V.

Maximum L/D Ratio.

R.N. - 200,000			R.N. - 3,600,000		
Order	Name	Max. L/D	Order	Name	Max. L/D
1	U.S.A. 5	26.0	1	R.A.F. 15	26.3
2	Clark Y	25.2	2	U.S.A. 5	25.0
3	R.A.F. 15	25.0	3	Clark Y	23.0
4	U.S.A. 27	21.6	4	U.S.A. 35B	22.0
5	U.S.A. 35A	18.9	5	U.S.A. 27	21.9
6	N.A.C.A. 97	17.7	6	N.A.C.A. 97	21.4
7	U.S.A. 35B	17.0	7	N.A.C.A. 99	19.8
8	Gott. 387	15.4	8	U.S.A. 35A	19.6
9	N.A.C.A. 98	15.1	9	Gott. 387	19.2
10	N.A.C.A. 99	13.4	10	N.A.C.A. 98	18.4

Table VI.

Name	R.N.	Max. L/D
N.A.C.A. M-4	3,680,000	25.8
N.A.C.A. M-6	3,660,000	23.6
N.A.C.A. M-12	3,800,000	22.0

Table VII.

Ratio - $\frac{\text{Maximum Lift Coefficient}}{\text{Minimum Drag Coefficient}}$

R.N. - 200,000			R.N. - 3,600,000		
Order	Name	C_L (max.) / C_D (min.)	Order	Name	C_L (max.) / C_D (min.)
1	R.A.F. 15	107.0	1	U.S.A. 35 B	152.1
2	Clark Y	100.8	2	Clark Y	138.0
3	U.S.A. 5	80.0	3	R.A.F. 15	137.9
4	U.S.A. 27	72.7	4	N.A.C.A. 97	125.6
5	U.S.A. 35A	71.3	5	U.S.A. 27	115.9
6	U.S.A. 35B	65.2	6	Gott. 387	111.0
7	N.A.C.A. 97	61.7	7	U.S.A. 5	108.1
8	Gott. 387	59.1	8	N.A.C.A. 98	91.8
9	N.A.C.A. 98	51.8	9	U.S.A. 35A	86.4
10	N.A.C.A. 99	44.1	10	N.A.C.A. 99	65.4

Table VIII.

Name	R.N.	C_L(max.)/C_D(min.)
N.A.C.A. M-4	3,680,000	130.8
N.A.C.A. M-6	3,660,000	143.7
N.A.C.A. M-12	3,800,000	144.4

Table IX.

Table of Principal Airfoil Characteristics.

Name	R.N. $\times 10^5$	Thickness % c		
		Max.	10% c	65% c
M-4	36.8	6.16	4.68	4.64
M-6	36.6	11.94	8.94	8.86
M-12	38.0	11.94	8.94	8.86
U.S.A. 5	36.3	6.28	5.63	4.84
U.S.A. 27	35.7	10.98	9.17	8.59
U.S.A. 35A	35.2	18.18	14.7	11.94
U.S.A. 35B	34.7	11.58	9.40	7.56
R.A.F. 15	35.8	6.38	6.05	4.45
Göttingen 387	34.7	15.14	12.27	9.83
Clark Y	36.1	11.7	9.2	8.3

Note: All airfoils - 5 in. x 30 in.

(12.7 cm x 76.2 cm)

Tested at twenty atmospheres tank pressure.

Table IX (Cont.)

Table of Principal Airfoil Characteristics.

Name	R.N. $\times 10^5$	Max. C_L	Angle Max. C_L	Min. C_D	Angle Min. C_D
M-4	36.8	0.94	13°	.0066	0°
M-6	36.6	1.22	18°	.0080	0°
M-12	38.0	1.29	18°	.0090	-1.5°
U.S.A. 5	36.3	1.19	11°	.0116	-1.5°
U.S.A. 27	35.7	1.39	16°	.0120	-4.0°
U.S.A. 35A	35.2	1.21	14°	.0140	-7.0°
U.S.A. 35B	34.7	1.37	15°	.0088	-5.0°
R.A.F. 15	35.8	1.21	15°	.0083	-2.0°
Göttingen 387	34.7	1.33	15°	.0125	-7.0°
Clark Y	36.1	1.38	16°	.0107	-4.5°

Table IX (Cont.)

Table of Principal Airfoil Characteristics.

Name	R.N. × 10^5	Max. L/D	Angle Max. L/D	$\frac{d\,C_L}{d\,\alpha}$	Angle zero lift	Average $C_{M\left(\frac{c}{4}\right)}$
M-4	36.8	25.8	4°	.073	0°	+.020
M-6	36.6	23.0	4.5°	.073	0°	+.020
M-12	38.0	21.4	4.5°	.073	-1°	-.002
U.S.A. 5	36.3	25.2	1°	.075	-4°	-.090
U.S.A. 27	35.7	21.8	1.5°	.070	-4.5°	-.070
U.S.A. 35A	35.2	19.4	-1.5°	.071	-8°	-.120
U.S.A. 35B	34.7	22.4	0°	.073	-5°	-.060
R.A.F. 15	35.8	25.2	3°	.075	-2°	-.050
Göttingen 387	34.7	19.2	-1.5°	.075	-7°	-.080
Clark Y	36.1	22.4	0°	.074	-5°	-.070

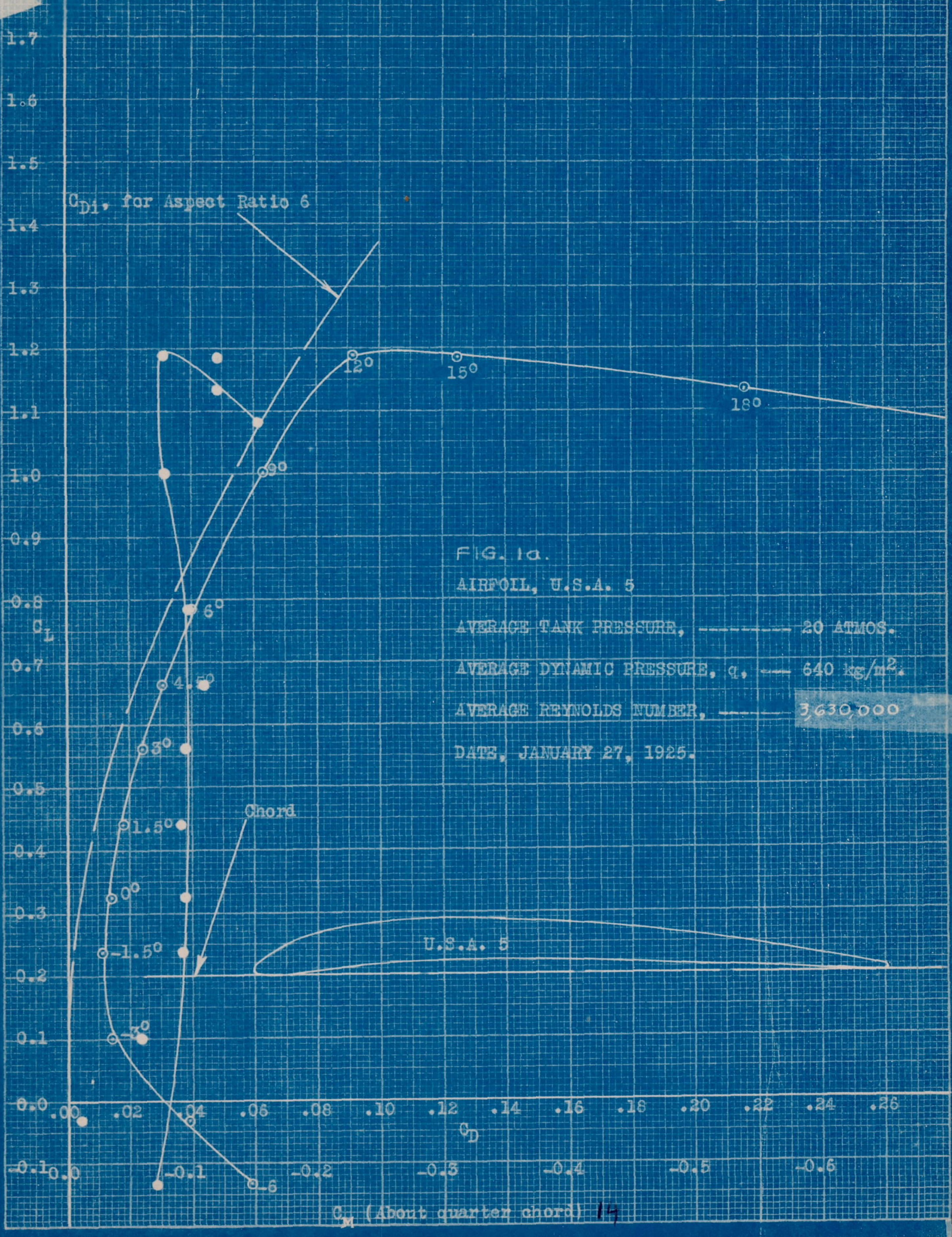


FIG. 1a.

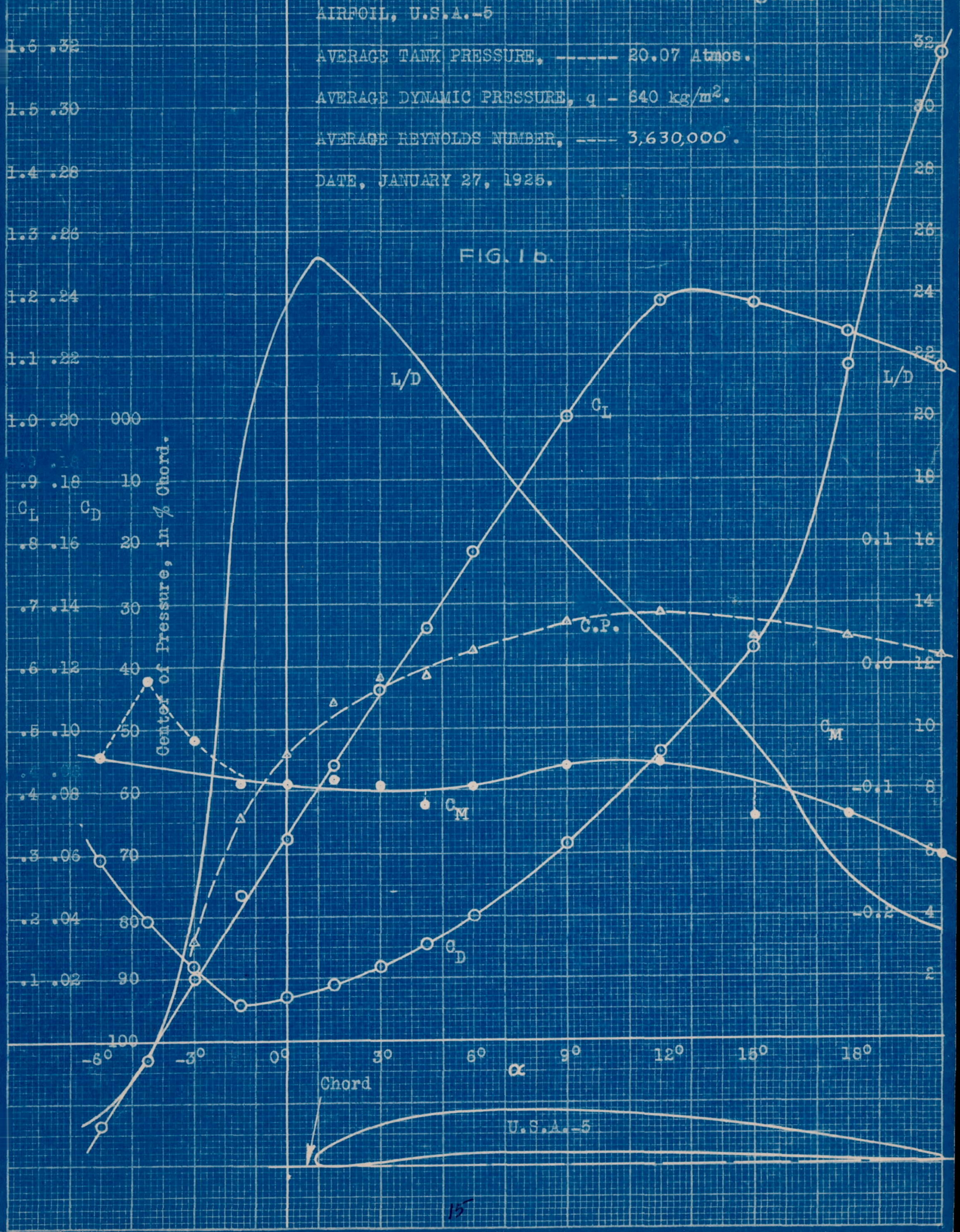
AIRFOIL, U.S.A. 5

AVERAGE TANK PRESSURE, ---------- 20 ATMOS.

AVERAGE DYNAMIC PRESSURE, q, ---- 640 kg/m².

AVERAGE REYNOLDS NUMBER, -------- 3,630,000

DATE, JANUARY 27, 1925.


AIRFOIL, U.S.A.-5
AVERAGE TANK PRESSURE, ------ 20.07 Atmos.
AVERAGE DYNAMIC PRESSURE, q - 640 kg/m².
AVERAGE REYNOLDS NUMBER, ---- 3,630,000.
DATE, JANUARY 27, 1925.
FIG. 1b.
L/D
C_L
C.P.
C_M
C_D
C_L C_D
Center of Pressure, in % Chord.
α
-6° -3° 0° 3° 6° 9° 12° 15° 18°
Chord
U.S.A.-5

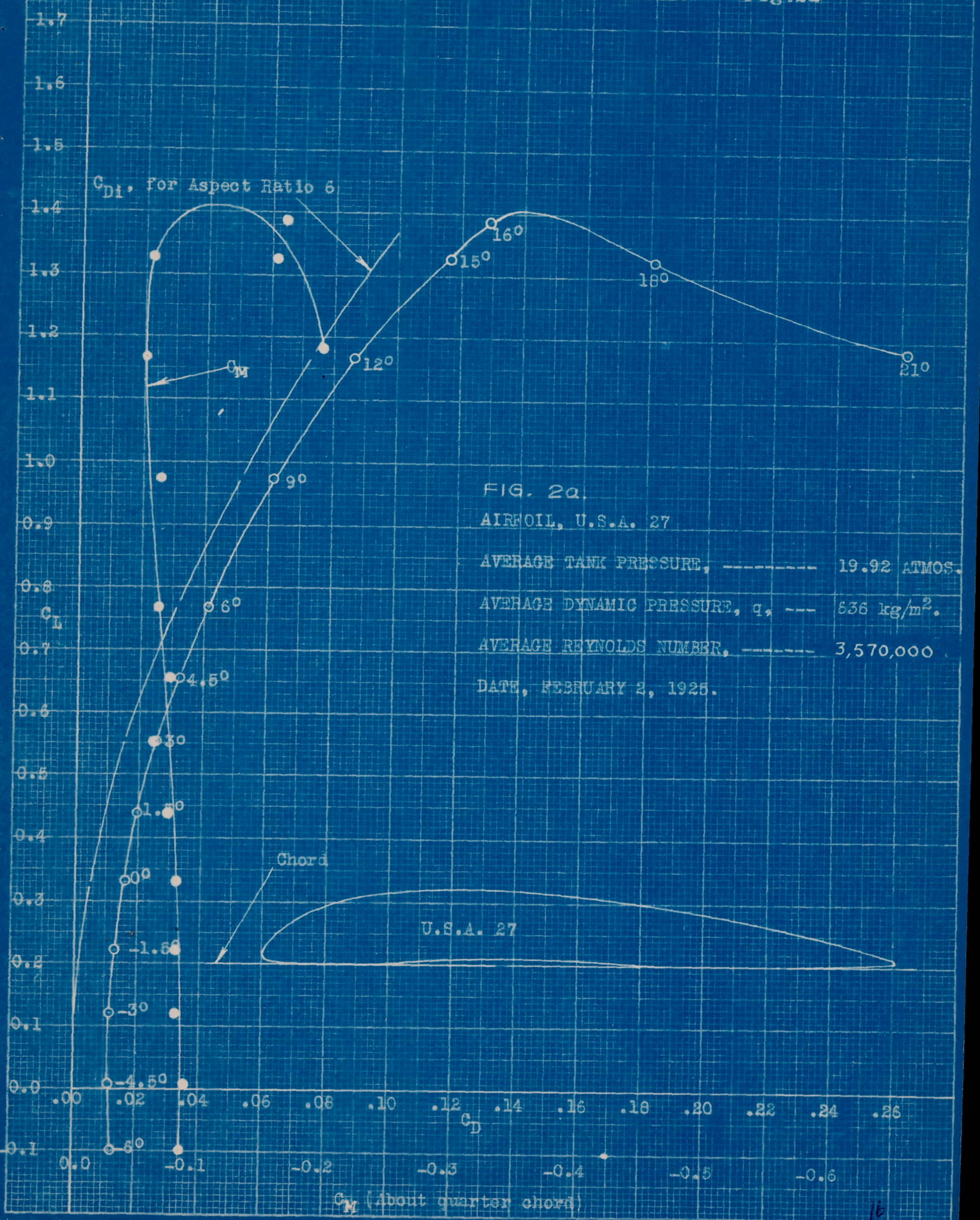


FIG. 2a.

AIRFOIL, U.S.A. 27

AVERAGE TANK PRESSURE, --------- 19.92 ATMOS.

AVERAGE DYNAMIC PRESSURE, q, --- 536 kg/m^2.

AVERAGE REYNOLDS NUMBER, ------- 3,570,000

DATE, FEBRUARY 2, 1925.

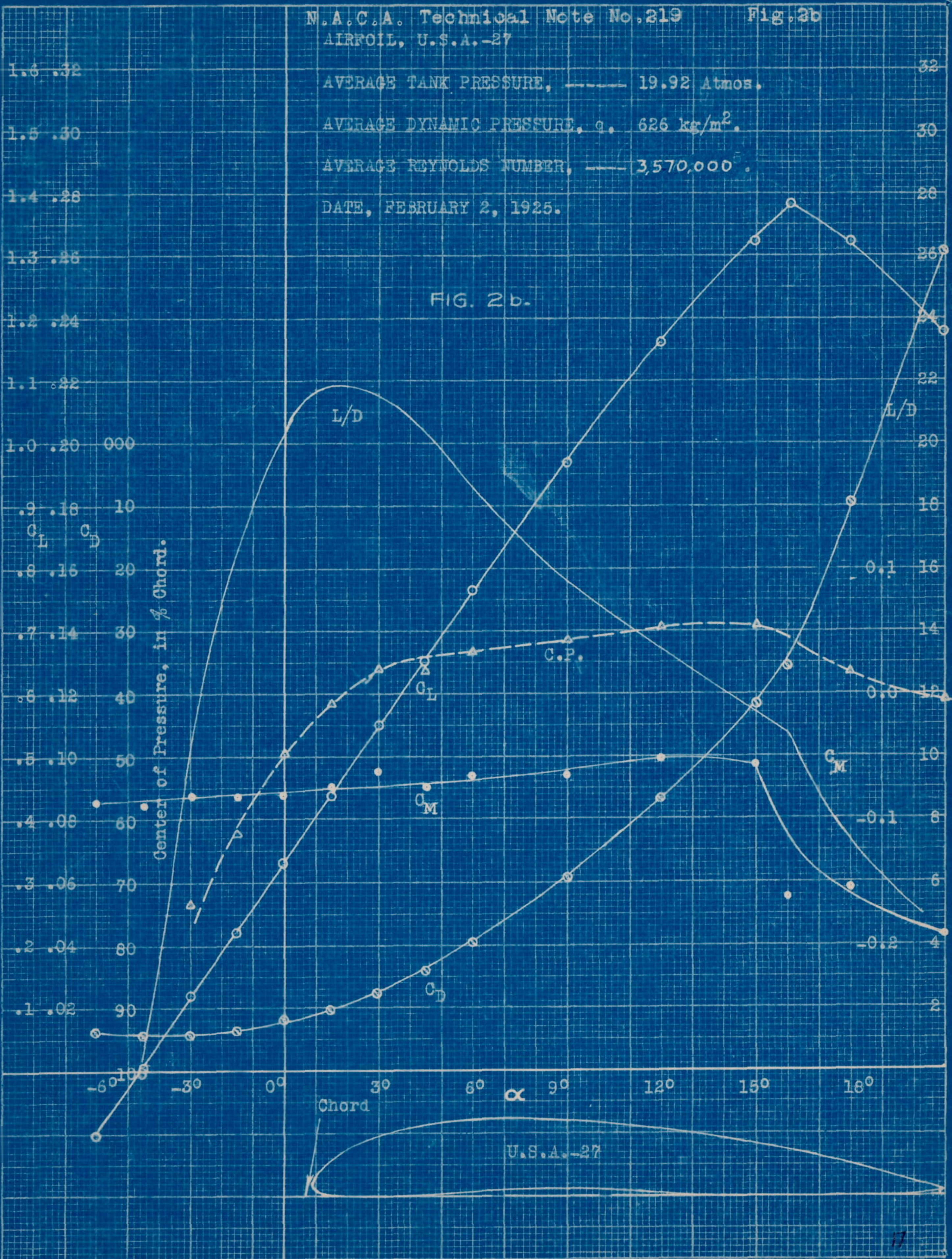

AIRFOIL, U.S.A.-27
AVERAGE TANK PRESSURE, ------ 19.92 Atmos.
AVERAGE DYNAMIC PRESSURE, q. 626 kg/m2.
AVERAGE REYNOLDS NUMBER, ---- 3,570,000.
DATE, FEBRUARY 2, 1925.
L/D
C.P.
CL
CM
CD
CL CD
Center of Pressure, in % Chord.
α
Chord
U.S.A.-27
-6° -3° 0° 3° 6° 9° 12° 15° 18°
1.6 .32
1.5 .30
1.4 .28
1.3 .26
1.2 .24
1.1 .22
1.0 .20 000
.9 .18 10
.8 .16 20
.7 .14 30
.6 .12 40
.5 .10 50
.4 .08 60
.3 .06 70
.2 .04 80
.1 .02 90
100
32 30 28 26 24 22 20 18 16 14 12 10 8 6 4 2
0.1 0.0 -0.1 -0.2

FIG. 2b.

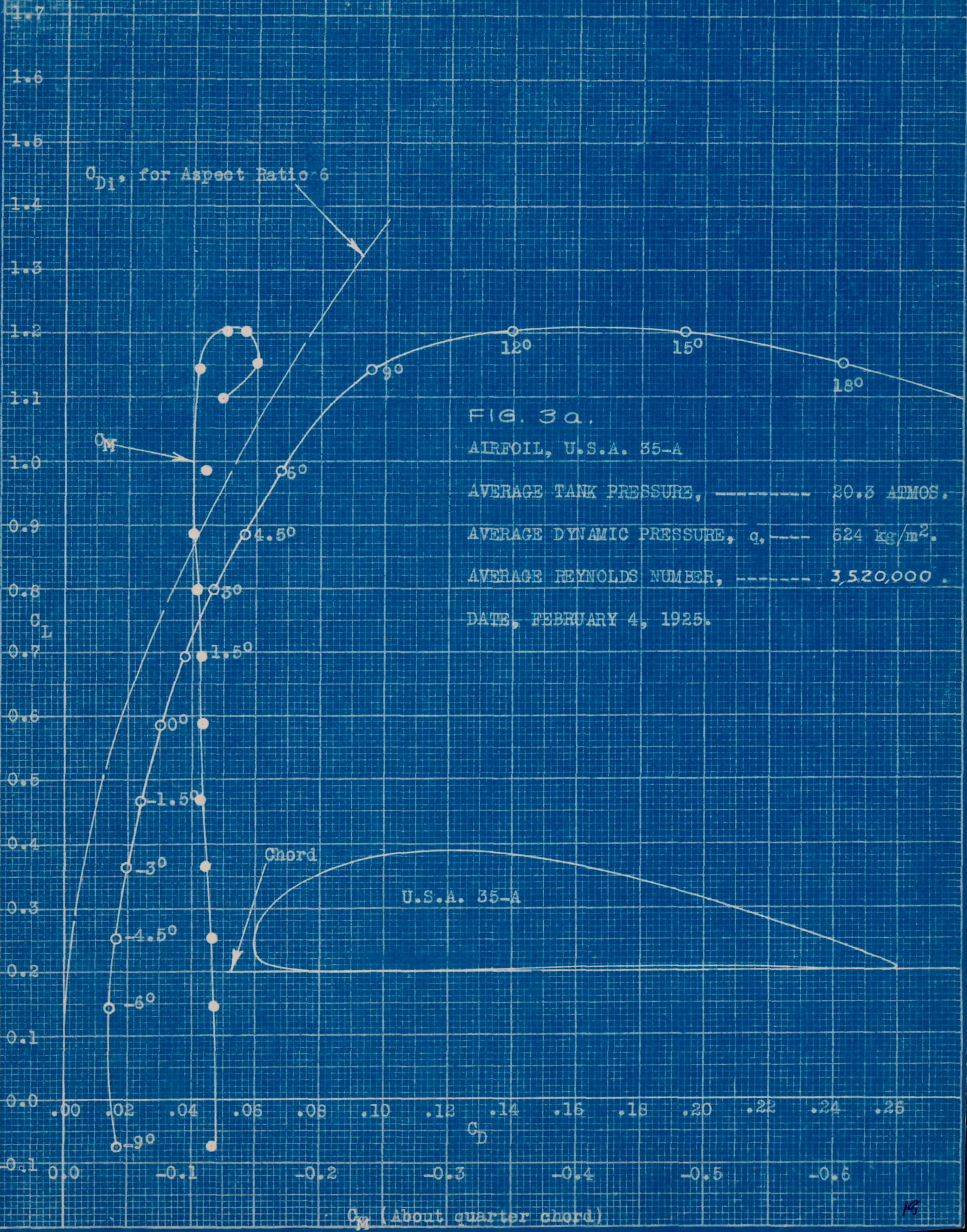


FIG. 3a.

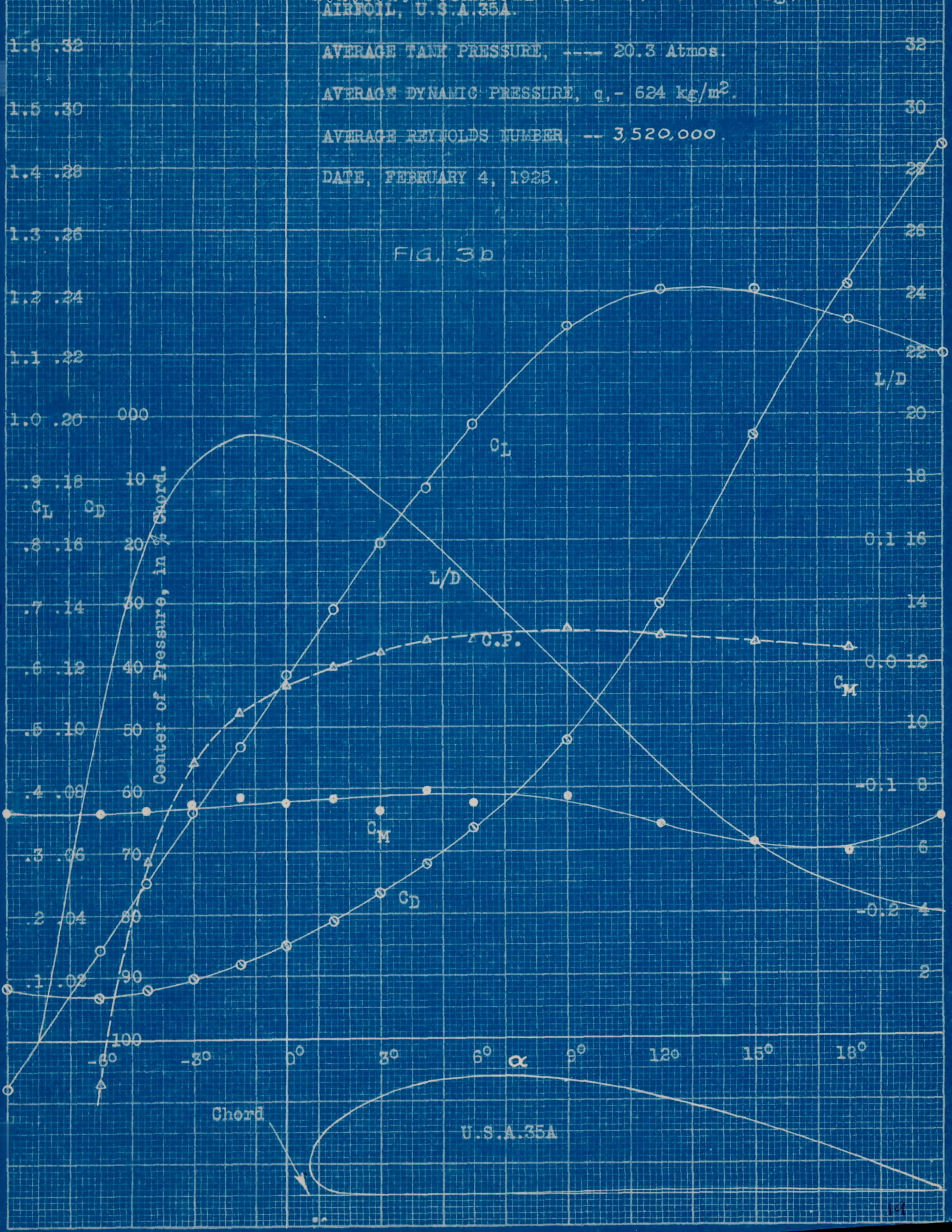
AIRFOIL, U.S.A. 35-A

AVERAGE TANK PRESSURE, ---------- 20.3 ATMOS.

AVERAGE DYNAMIC PRESSURE, q, ---- 624 kg/m^2.

AVERAGE REYNOLDS NUMBER, ------- 3,520,000 .

DATE, FEBRUARY 4, 1925.


AIRFOIL, U.S.A.35A.
AVERAGE TANK PRESSURE, ---- 20.3 Atmos.
AVERAGE DYNAMIC PRESSURE, q, - 624 kg/m2.
AVERAGE REYNOLDS NUMBER, -- 3,520,000.
DATE, FEBRUARY 4, 1925.
FIG. 3b
CL
CD
L/D
C.P.
CM
Center of Pressure, in % Chord.
α
Chord
U.S.A.35A

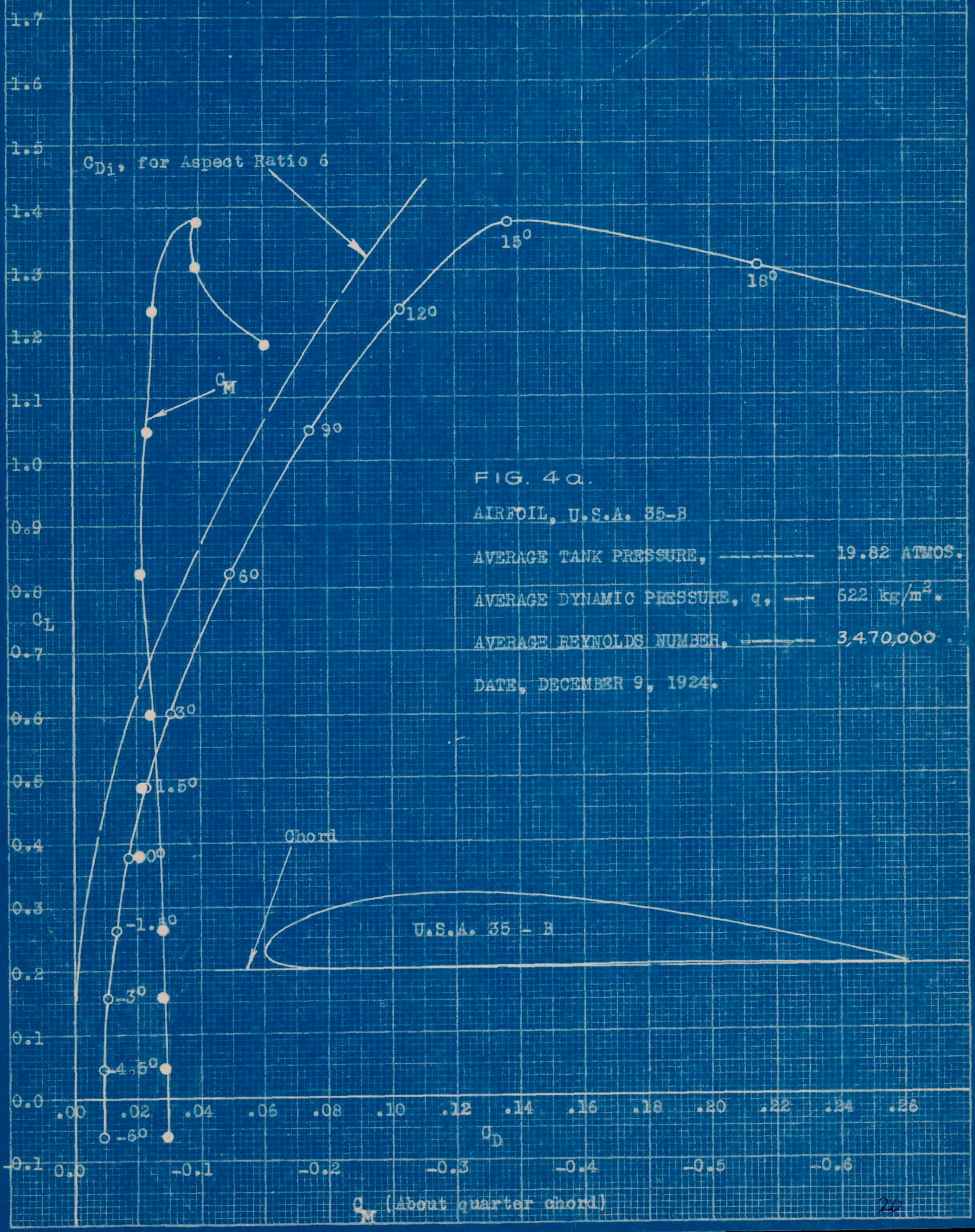
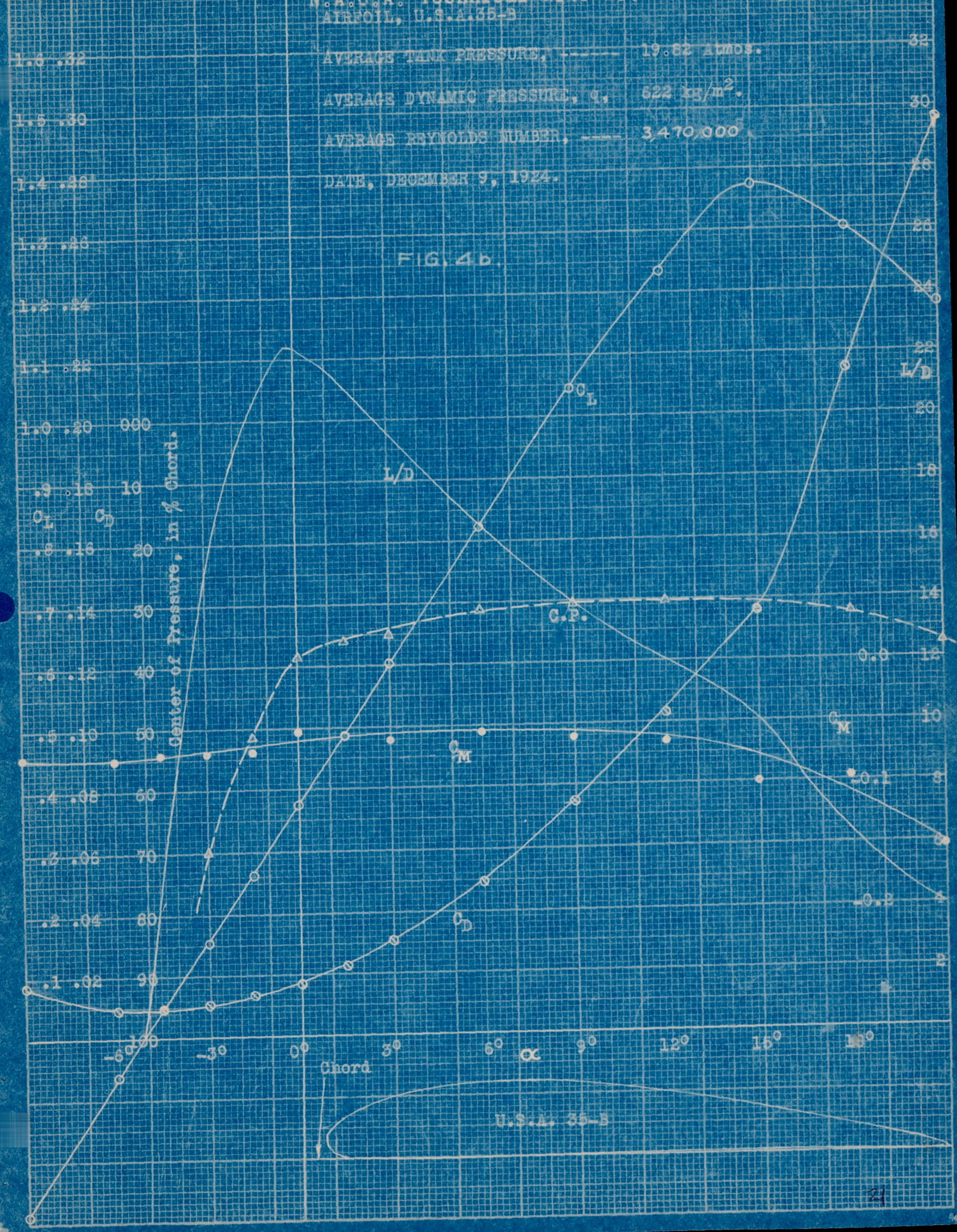
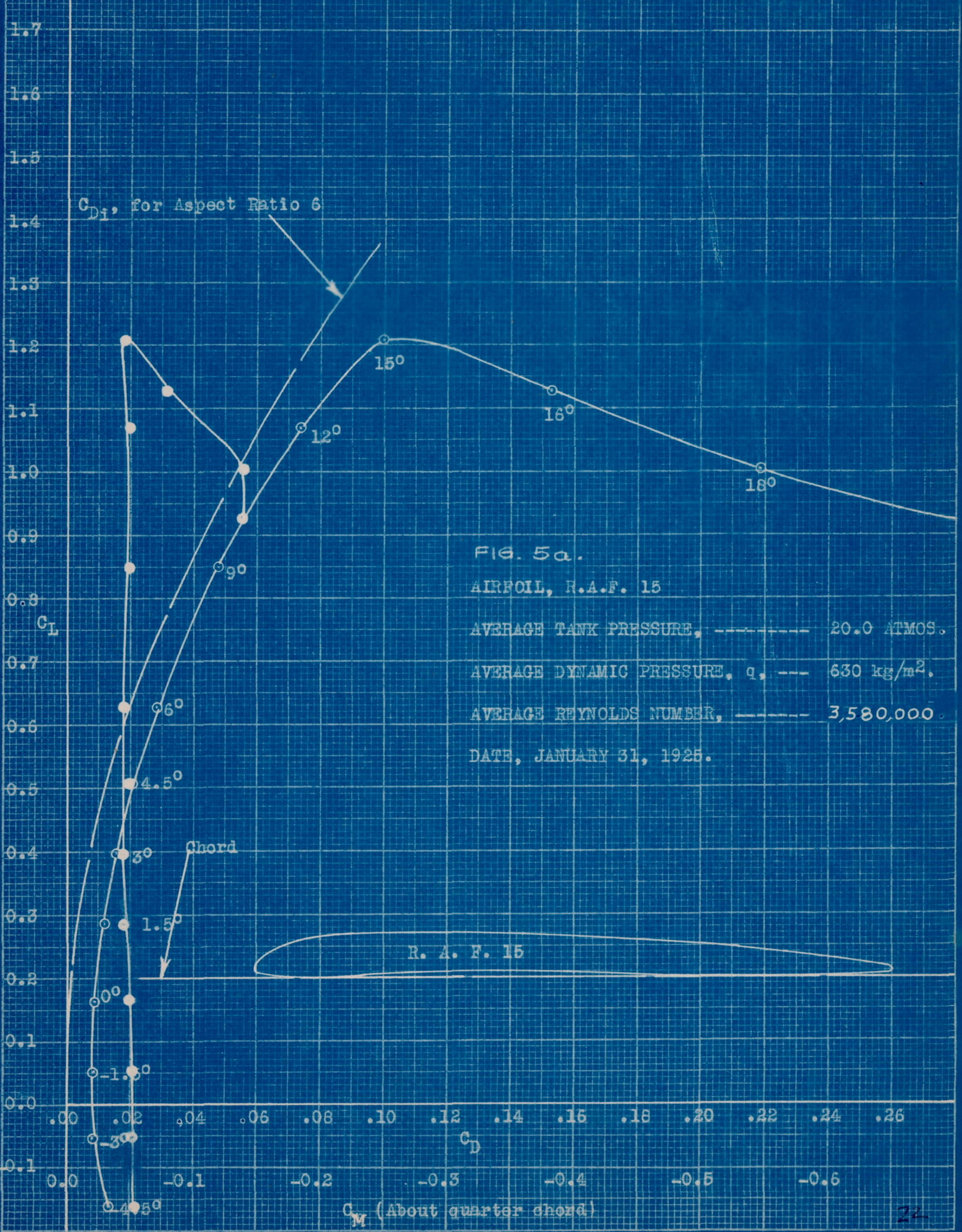


FIG. 4a.
AIRFOIL, U.S.A. 35-B
AVERAGE TANK PRESSURE, ---------- 19.82 ATMOS.
AVERAGE DYNAMIC PRESSURE, q, --- 622 kg/m².
AVERAGE REYNOLDS NUMBER, ------- 3,470,000.
DATE, DECEMBER 9, 1924.
C_Di, for Aspect Ratio 6
C_M
C_L
Chord
U.S.A. 35 - B
C_D
C_M (About quarter chord)
15°
18°
12°
9°
6°
3°
1.5°
0°
-1.5°
-3°
-4.5°
-6°
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0.0
-0.1
.00
.02
.04
.06
.08
.10
.12
.14
.16
.18
.20
.22
.24
.26
0.0
-0.1
-0.2
-0.3
-0.4
-0.5
-0.6


AIRFOIL, U.S.A.35-B
AVERAGE TANK PRESSURE, ------- 19.82 Atmos.
AVERAGE DYNAMIC PRESSURE, q, 822 kg/m².
AVERAGE REYNOLDS NUMBER, ---- 3,470,000
DATE, DECEMBER 9, 1924.
FIG. 4b.
L/D
C_L
C.P.
C_M
C_D
Center of Pressure, in % Chord.
Chord
α
U.S.A. 35-B

21



C_Di, for Aspect Ratio 6
15°
16°
12°
18°
9°
6°
4.5°
3°
1.5°
0°
-1.5°
-3°
-4.5°
Chord
R. A. F. 15
C_L
C_D
C_M (About quarter chord)
FIG. 5a.
AIRFOIL, R.A.F. 15
AVERAGE TANK PRESSURE, ---------- 20.0 ATMOS.
AVERAGE DYNAMIC PRESSURE, q, --- 630 kg/m².
AVERAGE REYNOLDS NUMBER, ------- 3,580,000
DATE, JANUARY 31, 1925.

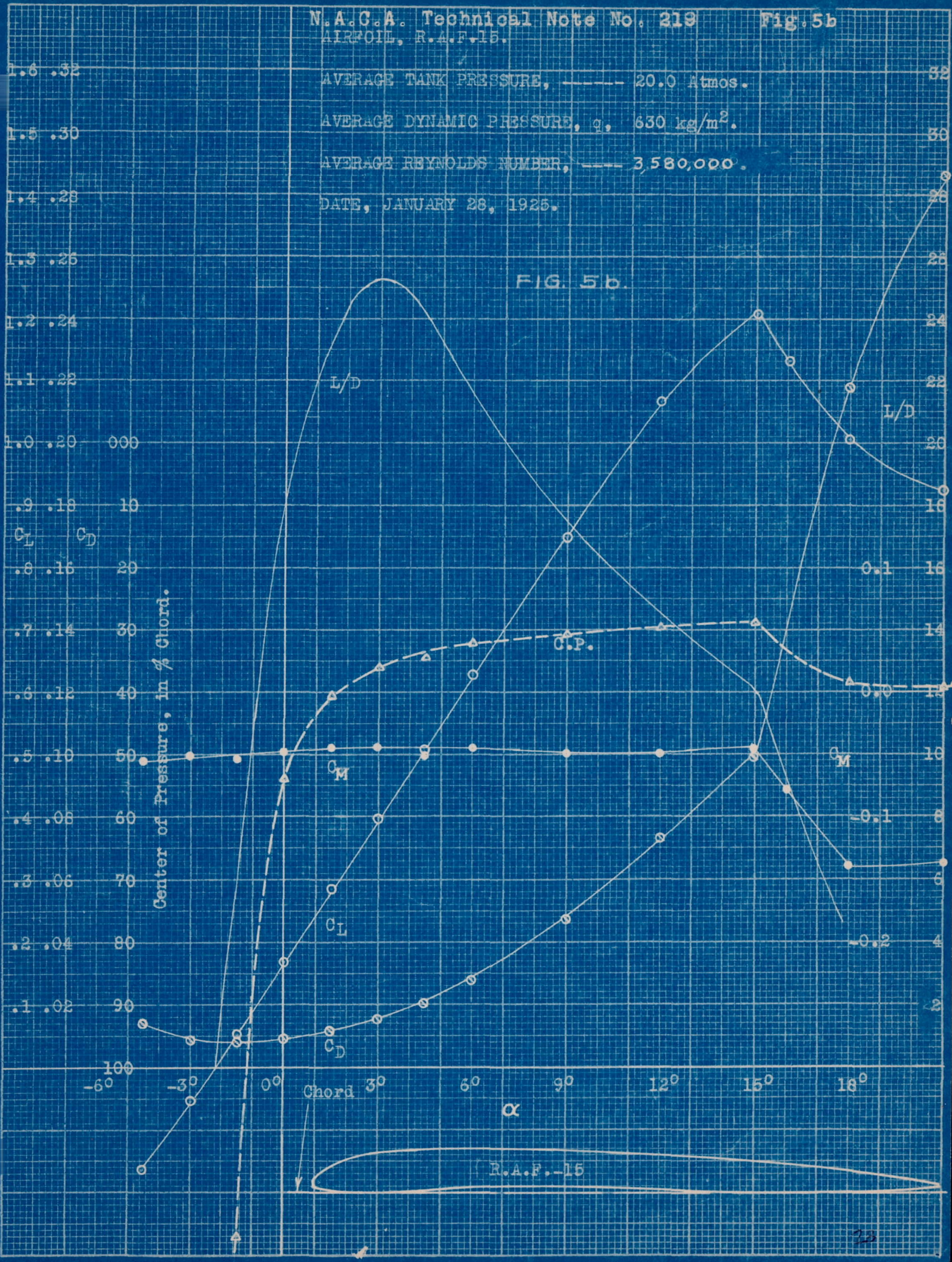
AIRFOIL, R.A.F.-15.

AVERAGE TANK PRESSURE, ------ 20.0 Atmos.

AVERAGE DYNAMIC PRESSURE, q, 630 kg/m².

AVERAGE REYNOLDS NUMBER, ---- 3,580,000.

DATE, JANUARY 28, 1925.



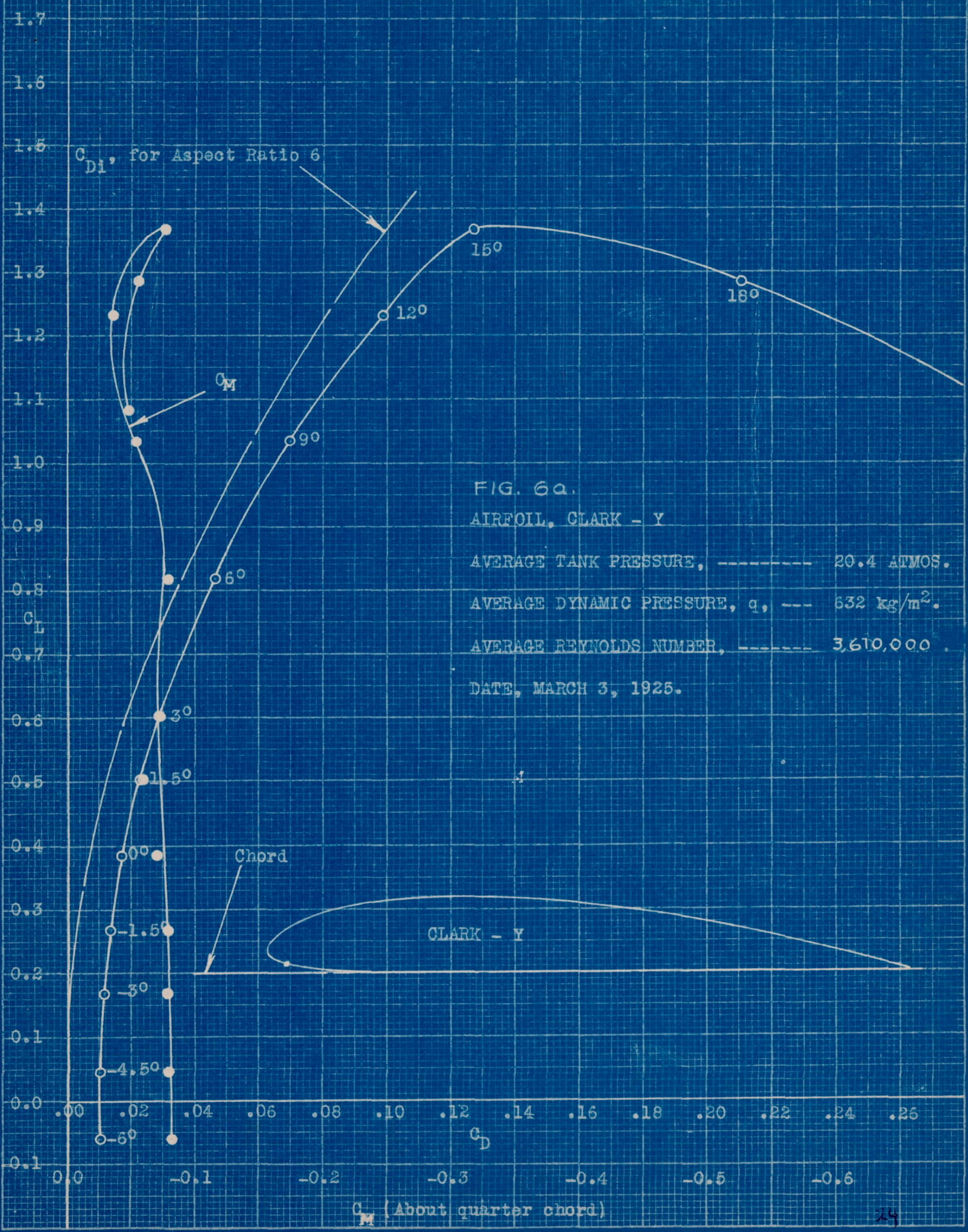


FIG. 6a.

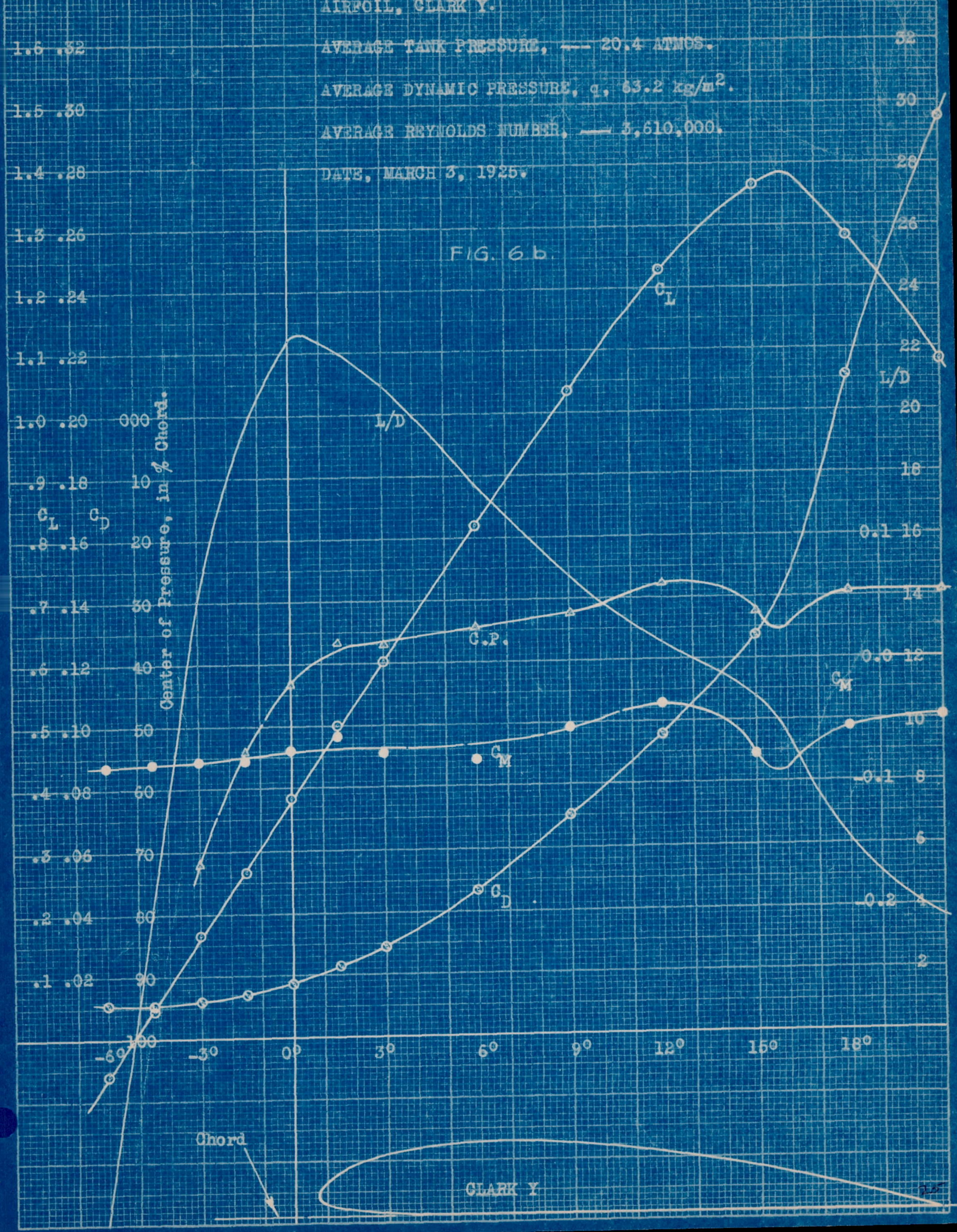
AIRFOIL, CLARK - Y

AVERAGE TANK PRESSURE, --------- 20.4 ATMOS.

AVERAGE DYNAMIC PRESSURE, q, --- 632 kg/m^2.

AVERAGE REYNOLDS NUMBER, ------- 3,610,000.

DATE, MARCH 3, 1925.


AIRFOIL, CLARK Y.
AVERAGE TANK PRESSURE, —— 20.4 ATMOS.
AVERAGE DYNAMIC PRESSURE, q, 63.2 kg/m².
AVERAGE REYNOLDS NUMBER, —— 3,610,000.
DATE, MARCH 3, 1925.
FIG. 6 b.
C_L
C_D
L/D
C.P.
C_M
Center of Pressure, in % Chord.
Chord
CLARK Y

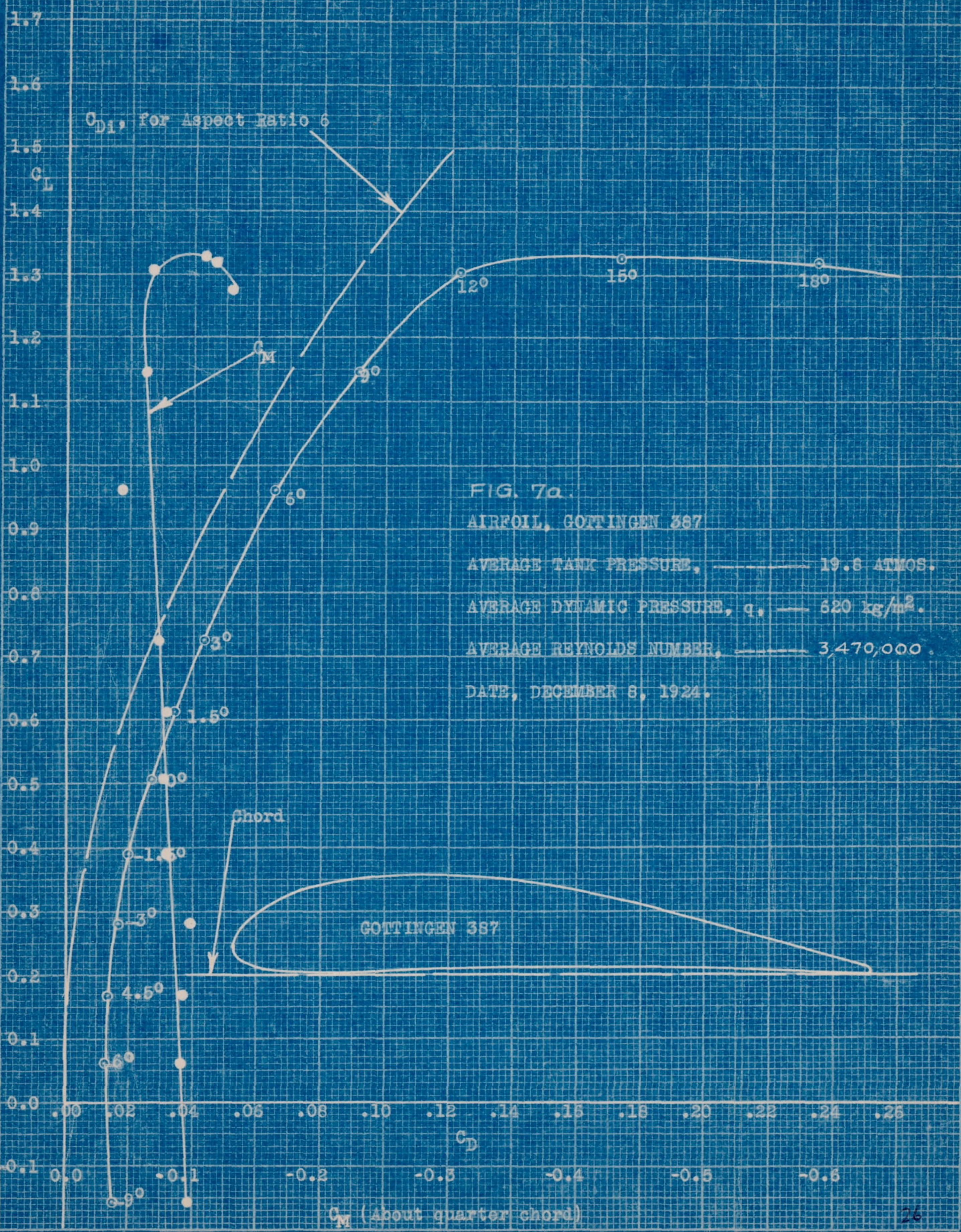


FIG. 7a.

AIRFOIL, GOTTINGEN 387

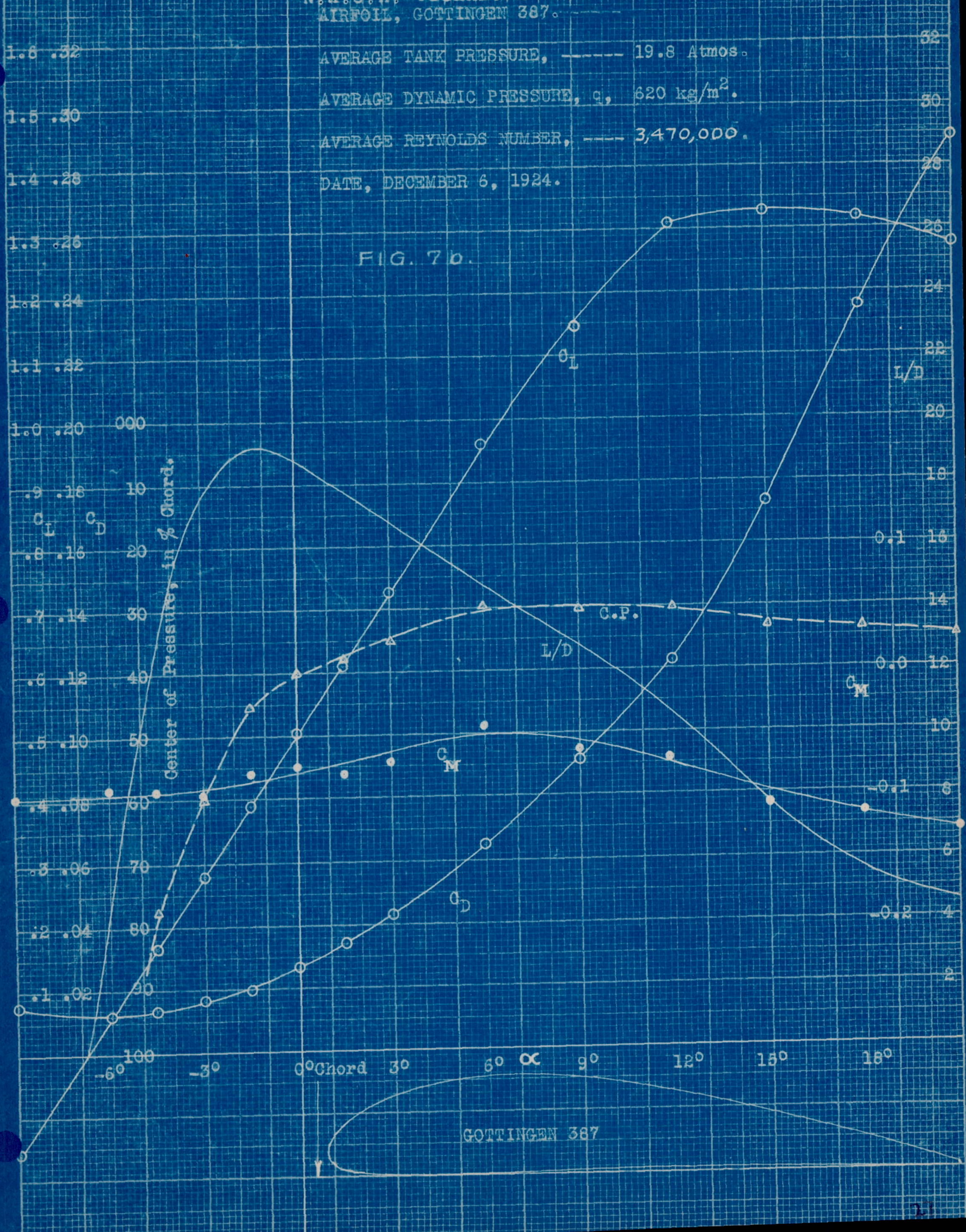
AVERAGE TANK PRESSURE, ———— 19.8 ATMOS.

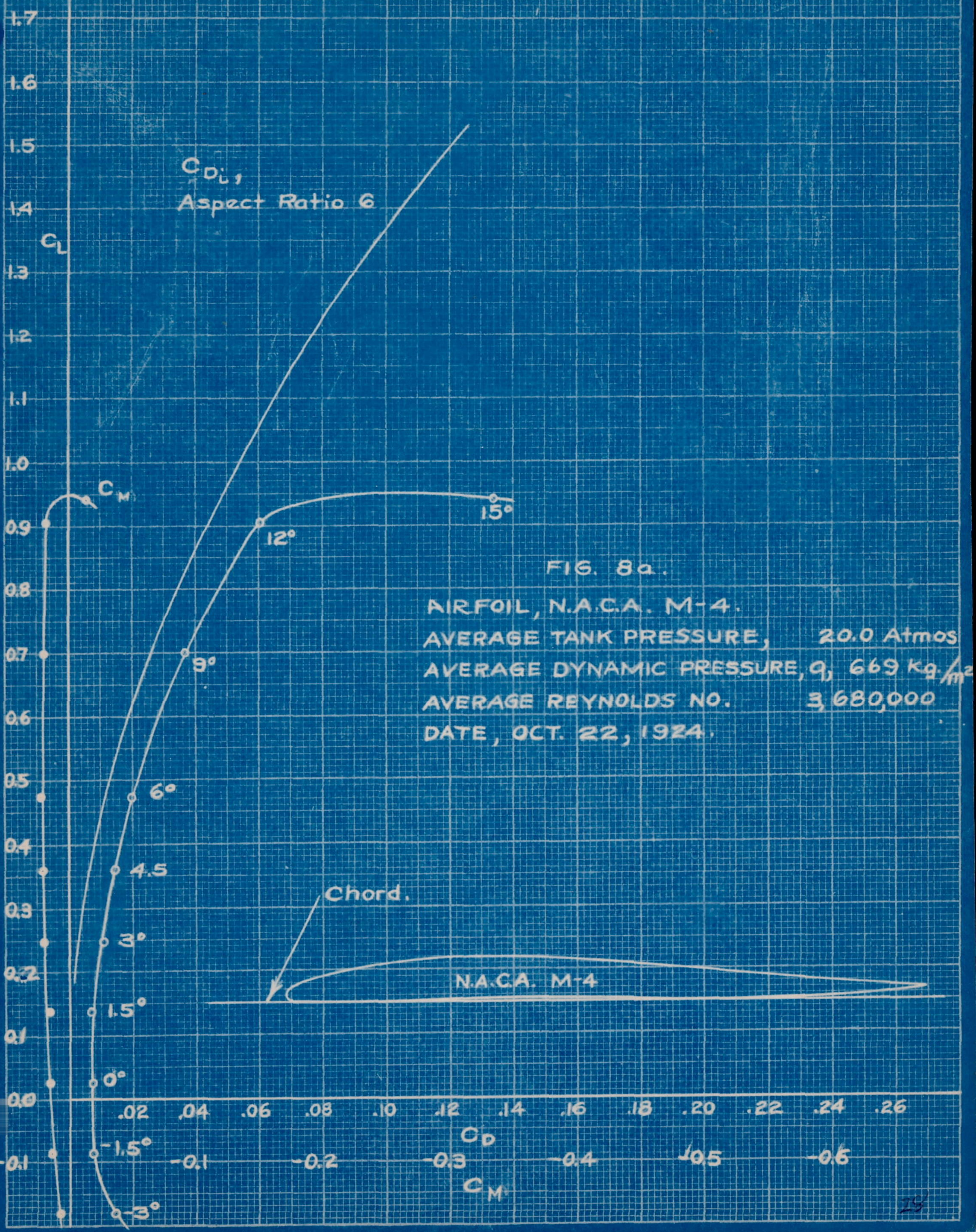
AVERAGE DYNAMIC PRESSURE, q, —— 620 kg/m².

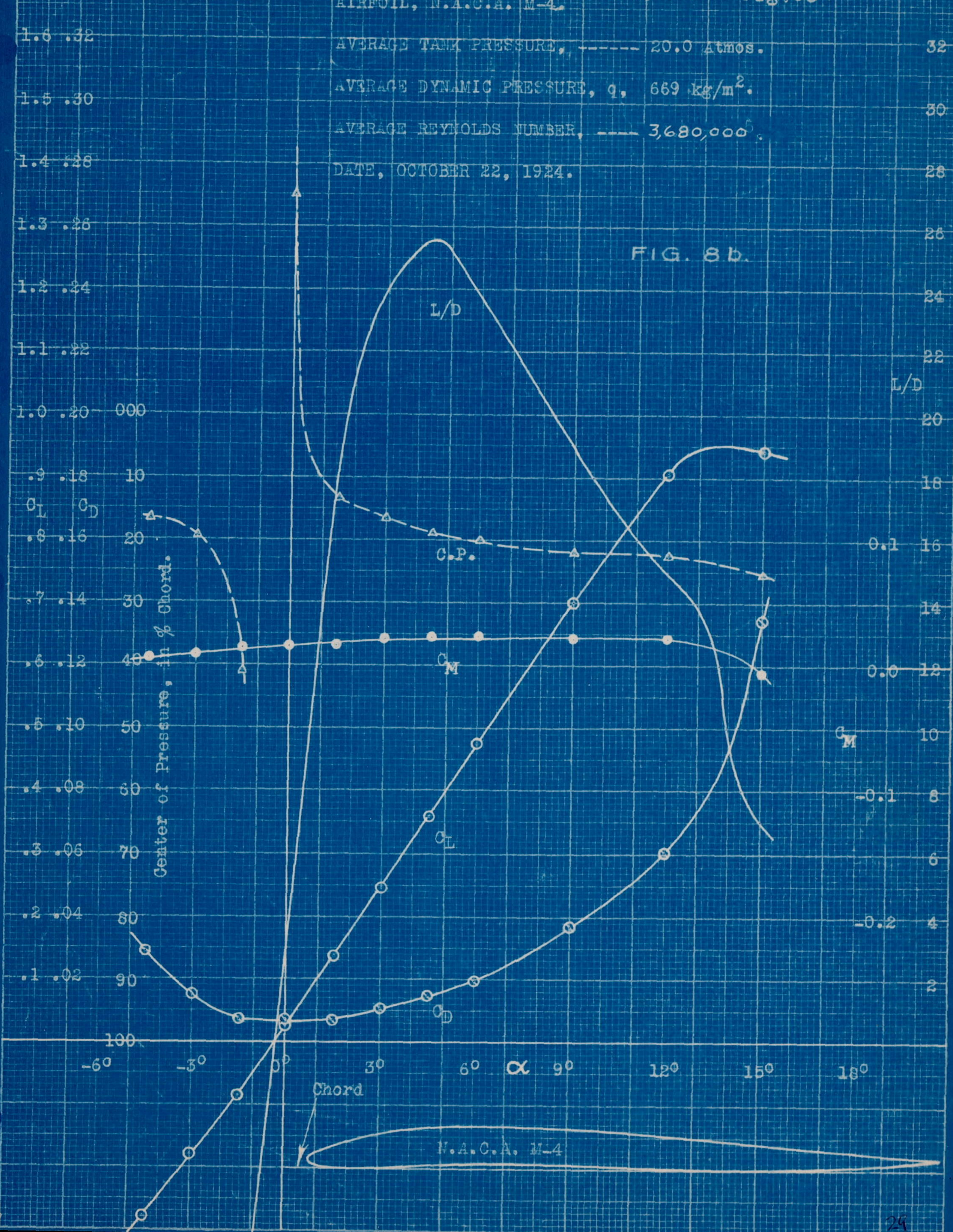
AVERAGE REYNOLDS NUMBER, ———— 3,470,000.

DATE, DECEMBER 8, 1924.

N.A.C.A. Technical Note No.219 Fig.7b




1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0
0.9
0.8
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0.0
-0.1
C_L
C_Di,
Aspect Ratio 6
C_M
15°
12°
9°
6°
4.5
3°
1.5°
0°
-1.5°
-3°
FIG. 8a.
AIRFOIL, N.A.C.A. M-4.
AVERAGE TANK PRESSURE, 20.0 Atmos
AVERAGE DYNAMIC PRESSURE, q, 669 Kg./m²
AVERAGE REYNOLDS NO. 3,680,000
DATE, OCT. 22, 1924.
Chord.
N.A.C.A. M-4
.02 .04 .06 .08 .10 .12 .14 .16 .18 .20 .22 .24 .26
C_D
-0.1 -0.2 -0.3 -0.4 -0.5 -0.6
C_M


AIRFOIL, N.A.C.A. M-4.
AVERAGE TANK PRESSURE, ------- 20.0 Atmos.
AVERAGE DYNAMIC PRESSURE, q, 669 kg/m².
AVERAGE REYNOLDS NUMBER, ---- 3,680,000.
DATE, OCTOBER 22, 1924.
FIG. 8b.
L/D
C.P.
CM
CL
CD
Center of Pressure, in % Chord.
α
Chord
N.A.C.A. M-4

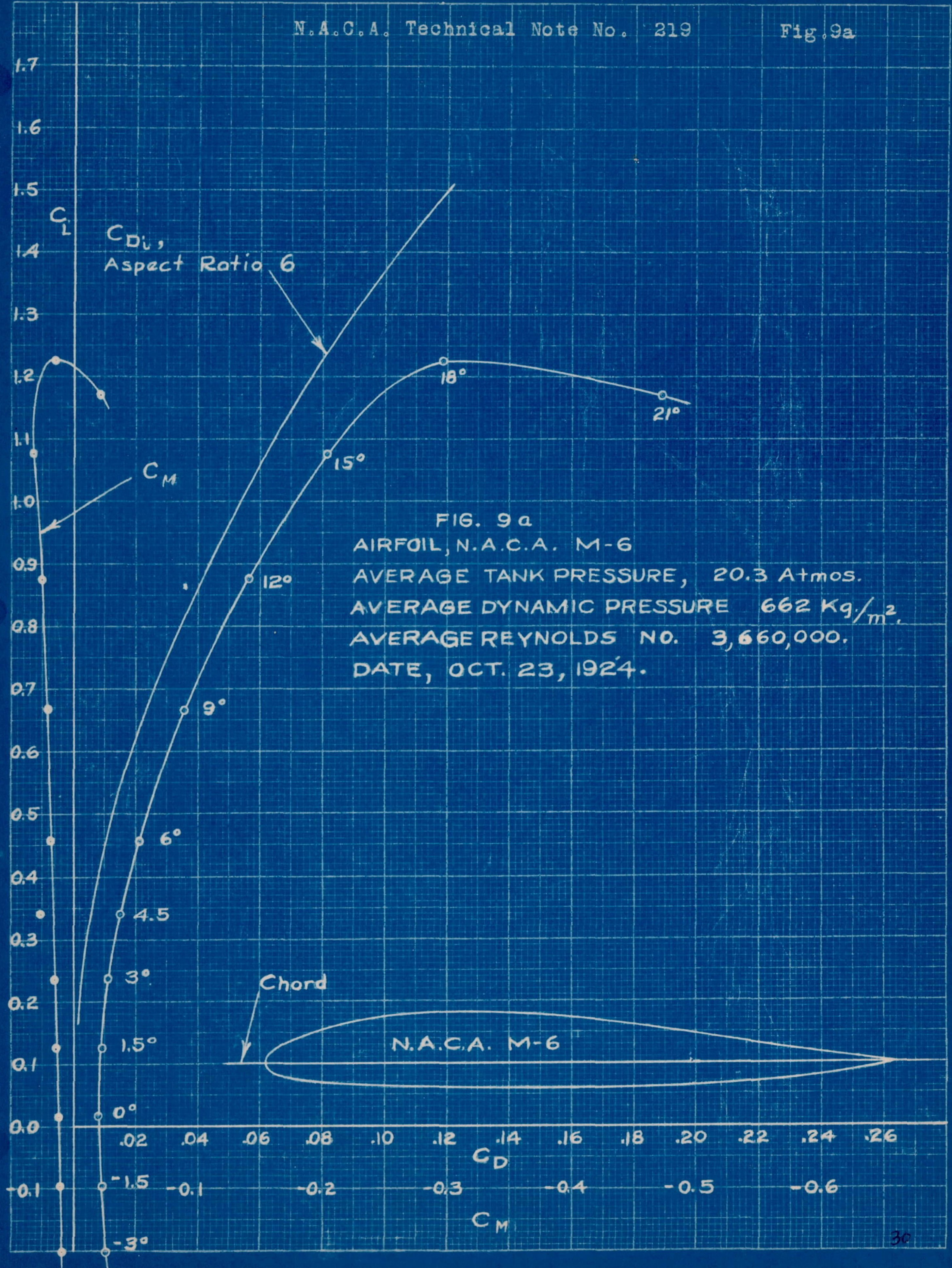
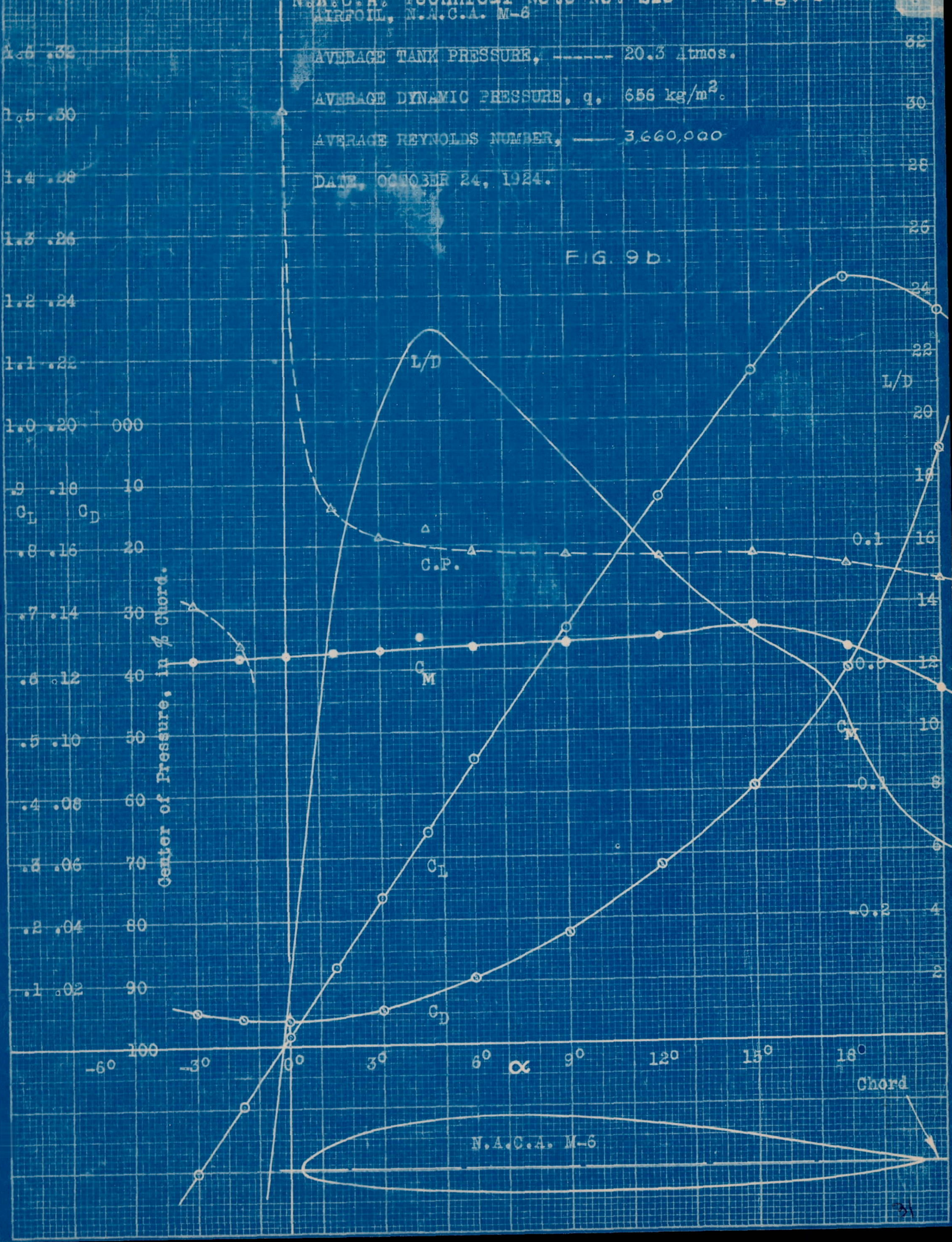


FIG. 9a
AIRFOIL, N.A.C.A. M-6
AVERAGE TANK PRESSURE, 20.3 Atmos.
AVERAGE DYNAMIC PRESSURE 662 Kg/m².
AVERAGE REYNOLDS NO. 3,660,000.
DATE, OCT. 23, 1924.


AIRFOIL, N.A.C.A. M-6
AVERAGE TANK PRESSURE, ------ 20.3 Atmos.
AVERAGE DYNAMIC PRESSURE, q. 656 kg/m².
AVERAGE REYNOLDS NUMBER, ---- 3,660,000
DATE, OCTOBER 24, 1924.
FIG. 9b.
L/D
C.P.
C_M
C_L
C_D
Center of Pressure, in % Chord.
α
Chord
N.A.C.A. M-6

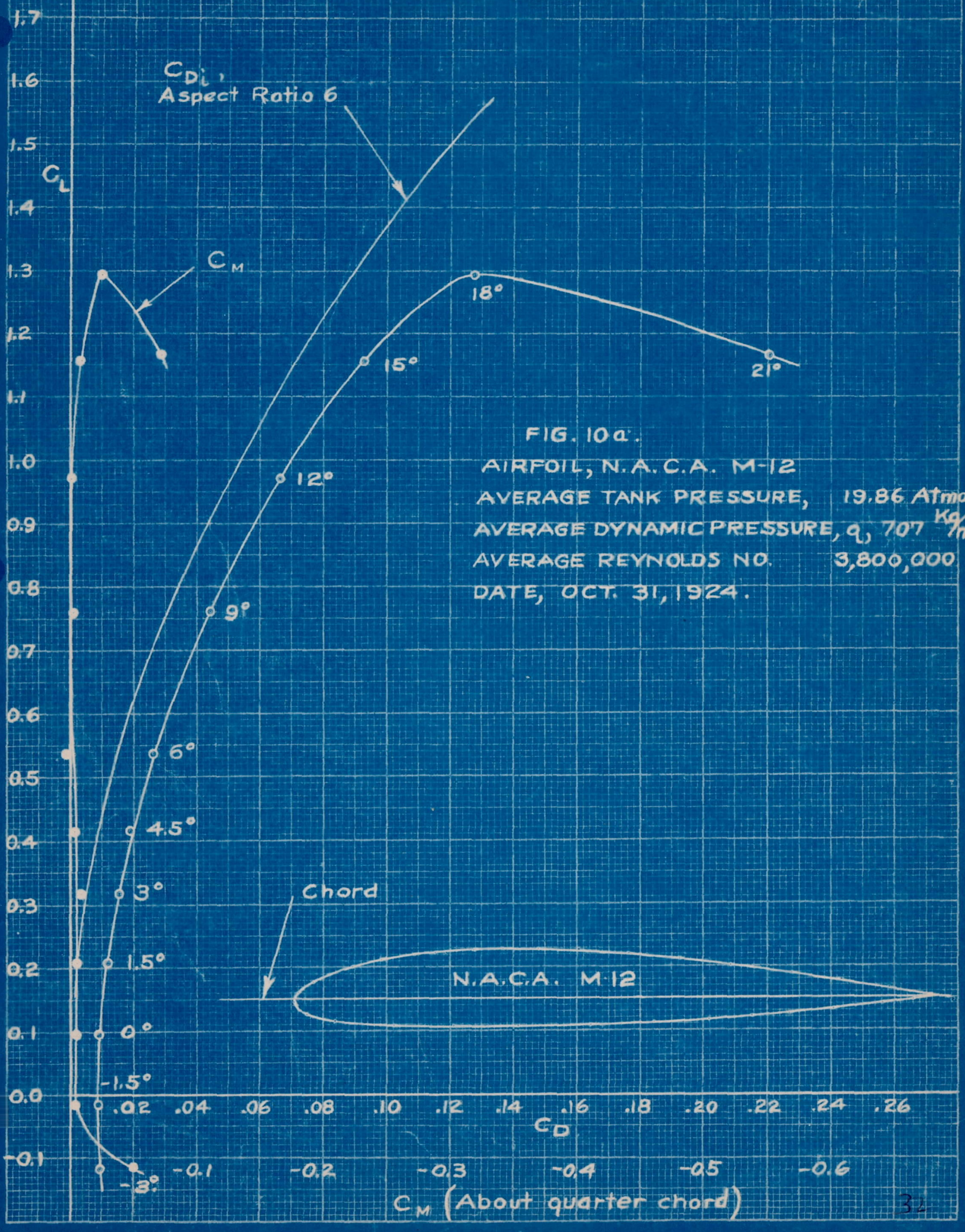
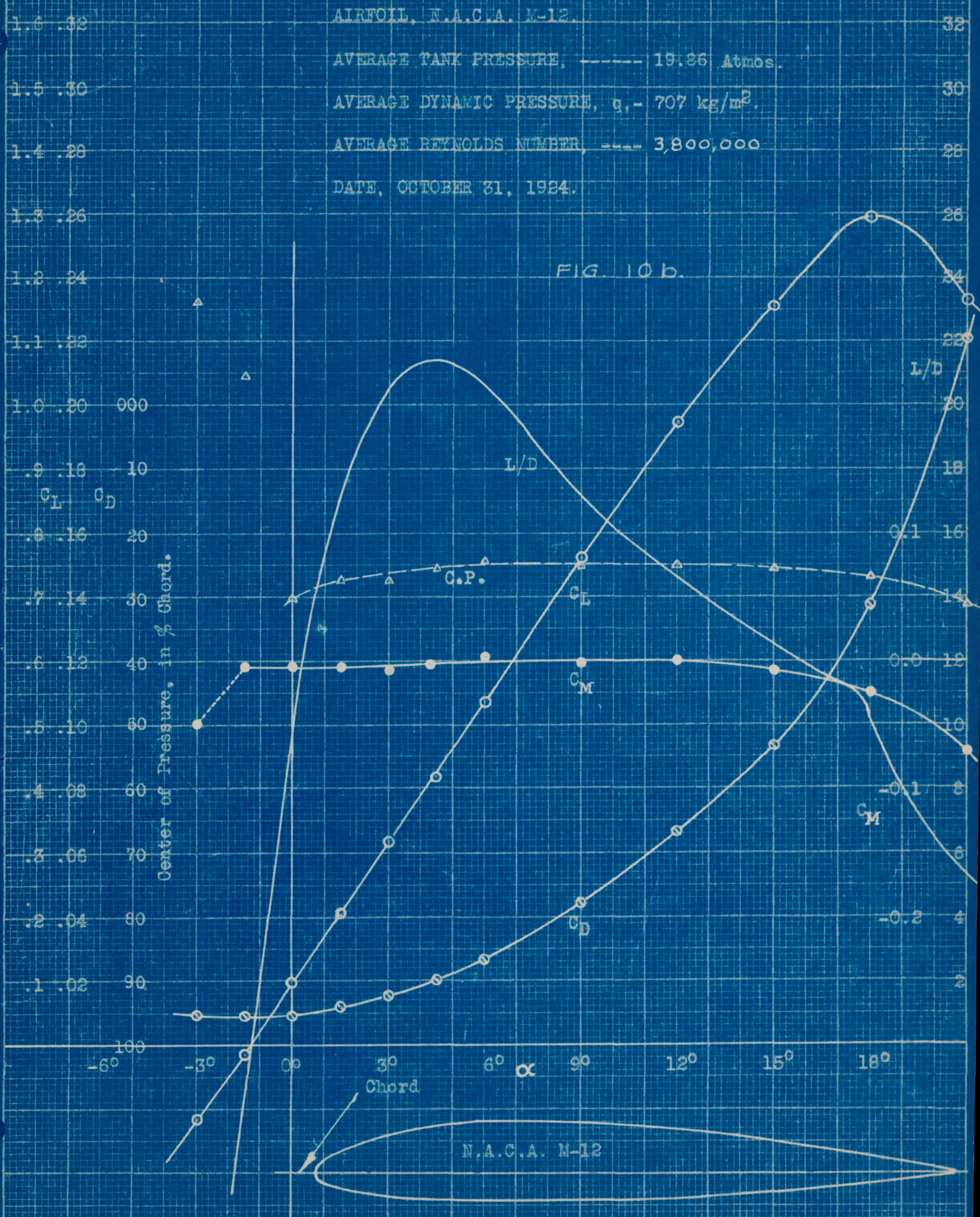
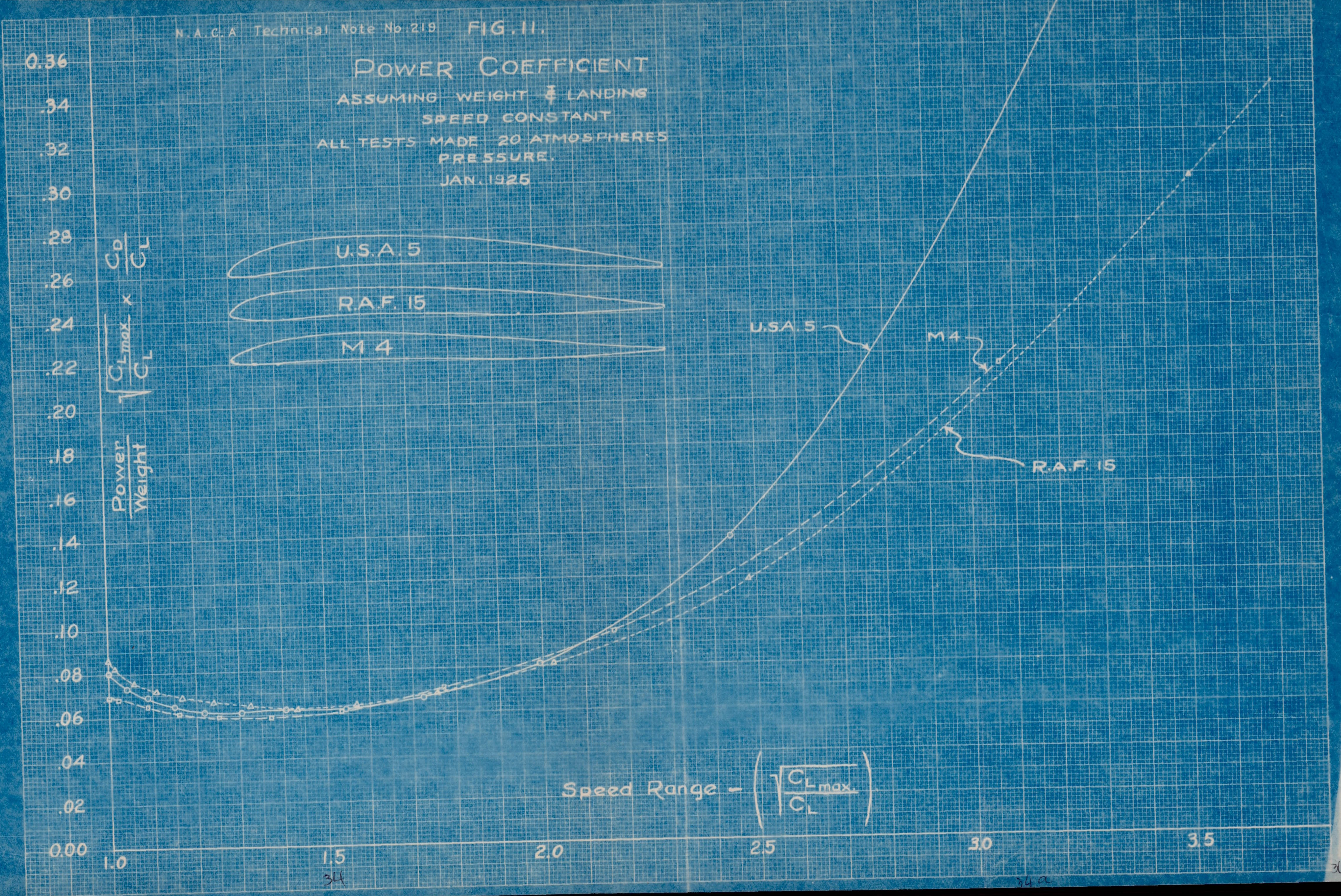


FIG. 10a.
AIRFOIL, N.A.C.A. M-12
AVERAGE TANK PRESSURE, 19.86 Atmos
AVERAGE DYNAMIC PRESSURE, q, 707 Kg/m²
AVERAGE REYNOLDS NO. 3,800,000
DATE, OCT. 31, 1924.


AIRFOIL, N.A.C.A. M-12.
AVERAGE TANK PRESSURE, ------ 19.86 Atmos.
AVERAGE DYNAMIC PRESSURE, q,- 707 kg/m2.
AVERAGE REYNOLDS NUMBER, ---- 3,800,000
DATE, OCTOBER 31, 1924.
FIG. 10 b.
CL
CD
Center of Pressure, in % Chord.
L/D
C.P.
CL
CM
CD
α
Chord
N.A.C.A. M-12
-6° -3° 0° 3° 6° 9° 12° 15° 18°
1.6 .32
1.5 .30
1.4 .28
1.3 .26
1.2 .24
1.1 .22
1.0 .20 000
.9 .18 10
.8 .16 20
.7 .14 30
.6 .12 40
.5 .10 50
.4 .08 60
.3 .06 70
.2 .04 80
.1 .02 90
100
0.1
0.0
-0.1
-0.2
32 30 28 26 24 22 20 18 16 14 12 10 8 6 4 2


N.A.C.A. Technical Note No. 219
FIG. 11.
POWER COEFFICIENT
ASSUMING WEIGHT & LANDING
SPEED CONSTANT
ALL TESTS MADE 20 ATMOSPHERES
PRESSURE.
JAN. 1925
U.S.A. 5
R.A.F. 15
M 4
$\frac{\text{Power}}{\text{Weight}} \sqrt{\frac{C_{L_{max}}}{C_L}} \times \frac{C_D}{C_L}$
Speed Range - $\left(\sqrt{\frac{C_{L_{max}}}{C_L}}\right)$
0.36
.34
.32
.30
.28
.26
.24
.22
.20
.18
.16
.14
.12
.10
.08
.06
.04
.02
0.00
1.0
1.5
2.0
2.5
3.0
3.5

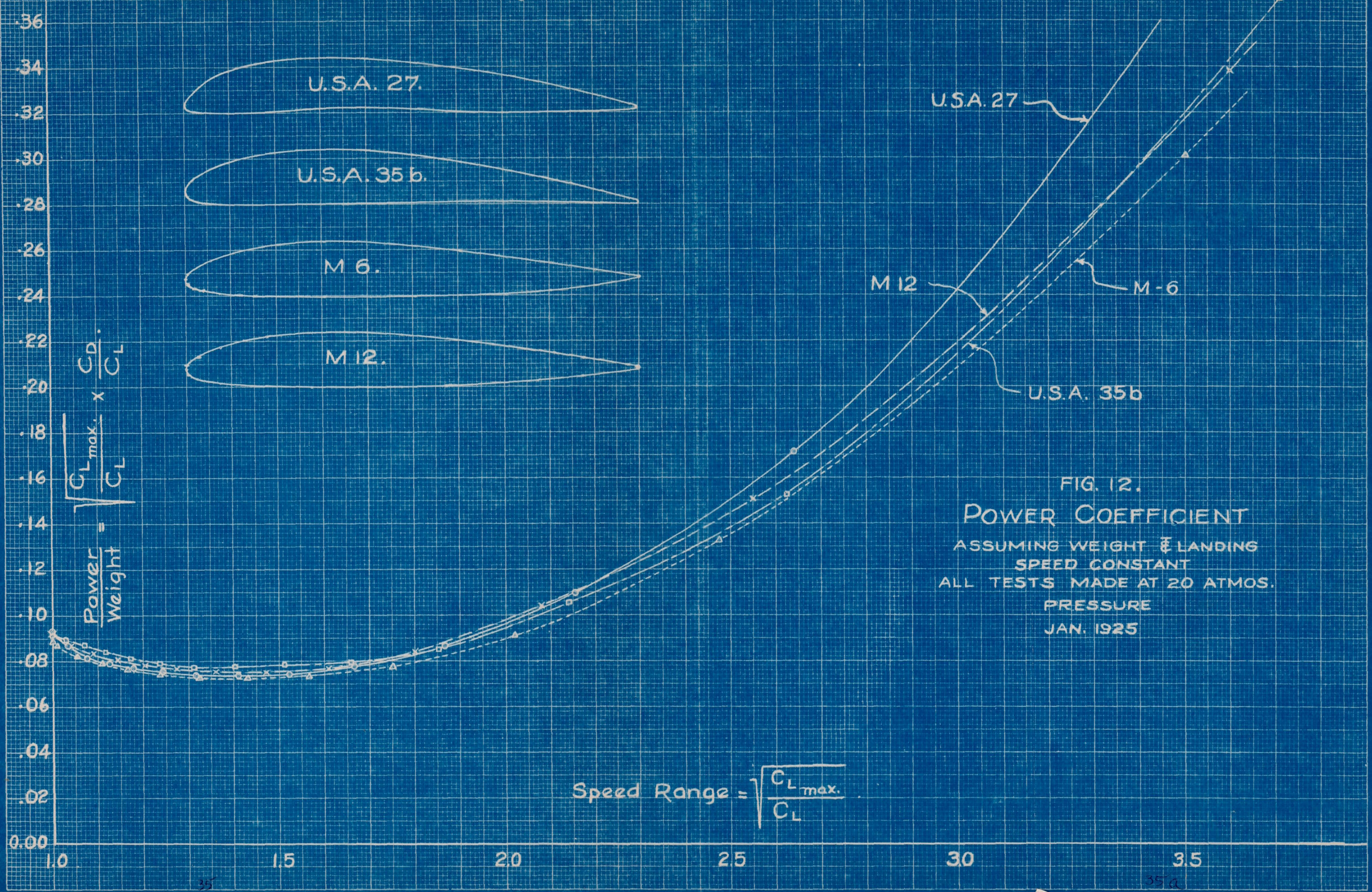


FIG. 12.

POWER COEFFICIENT

ASSUMING WEIGHT & LANDING SPEED CONSTANT

ALL TESTS MADE AT 20 ATMOS. PRESSURE

JAN. 1925

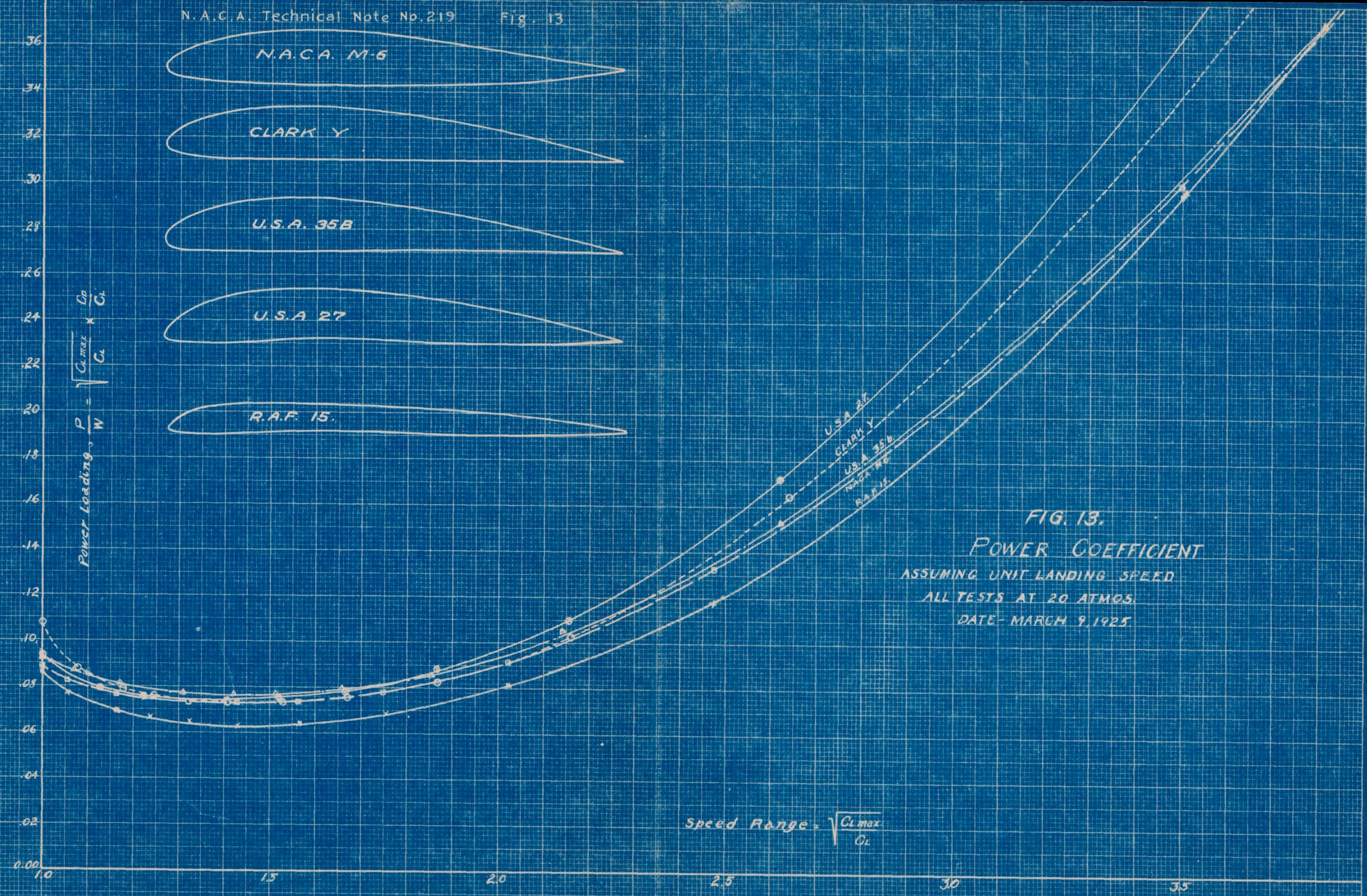


FIG. 13.
POWER COEFFICIENT
ASSUMING UNIT LANDING SPEED
ALL TESTS AT 20 ATMOS.
DATE - MARCH 9, 1925

36

36a

E.N.J.

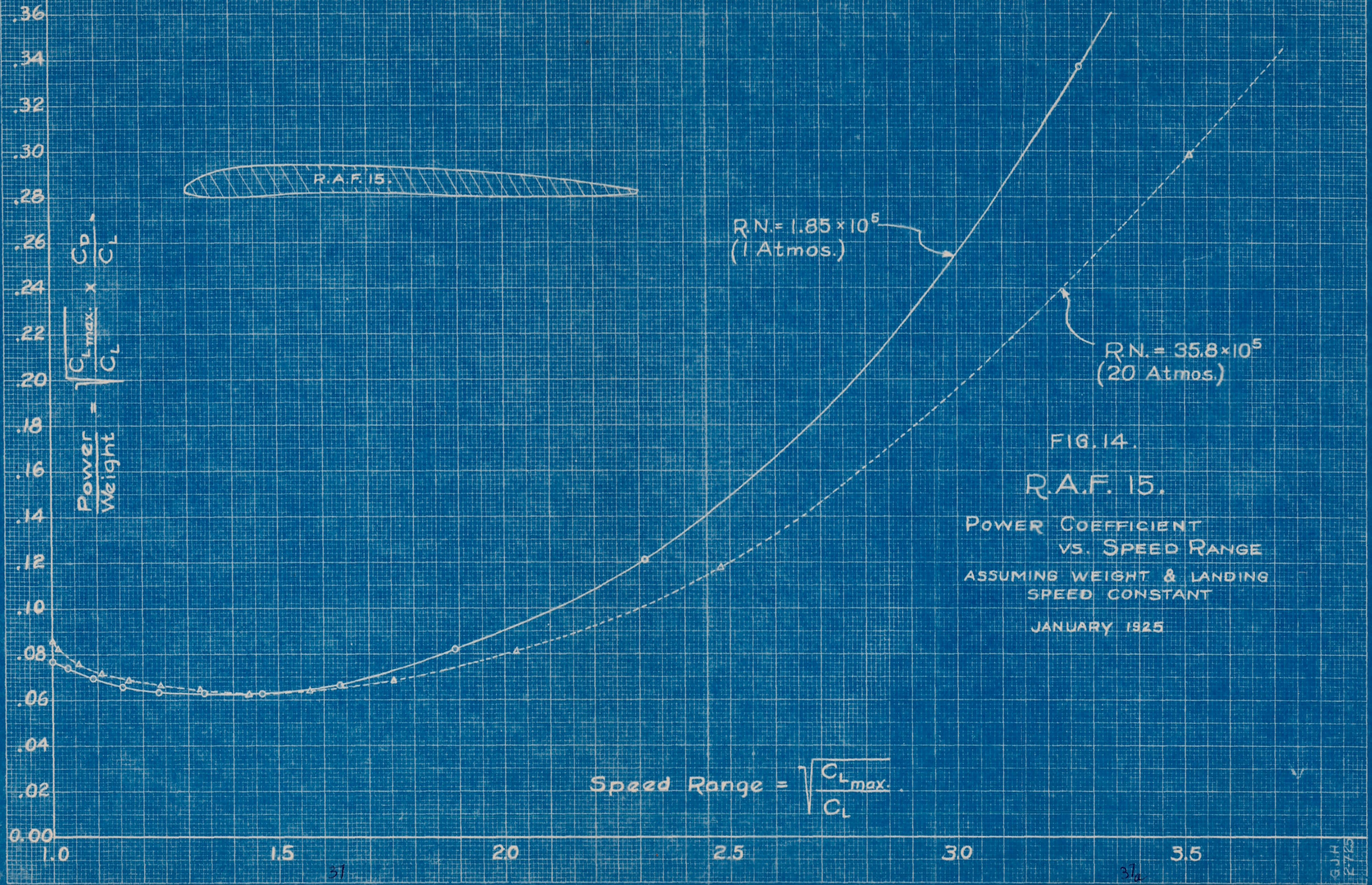


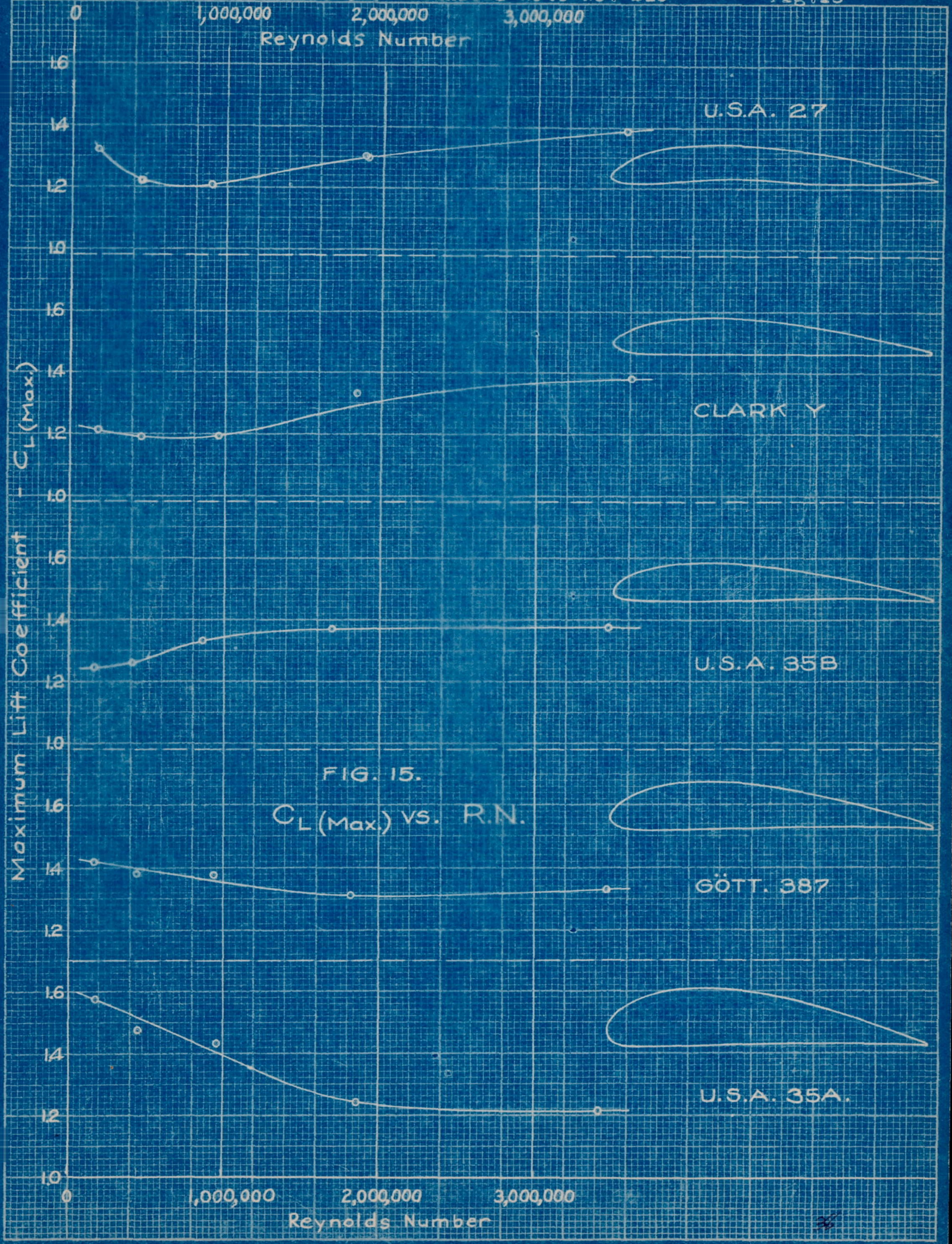
FIG. 14.

R.A.F. 15.

POWER COEFFICIENT
VS. SPEED RANGE

ASSUMING WEIGHT & LANDING
SPEED CONSTANT

JANUARY 1925


0
1,000,000
2,000,000
3,000,000
Reynolds Number
Maximum Lift Coefficient - C_L (Max.)
1.6
1.4
1.2
1.0
U.S.A. 27
CLARK Y
U.S.A. 35B
GÖTT. 387
U.S.A. 35A.
FIG. 15.
C_L (Max.) VS. R.N.

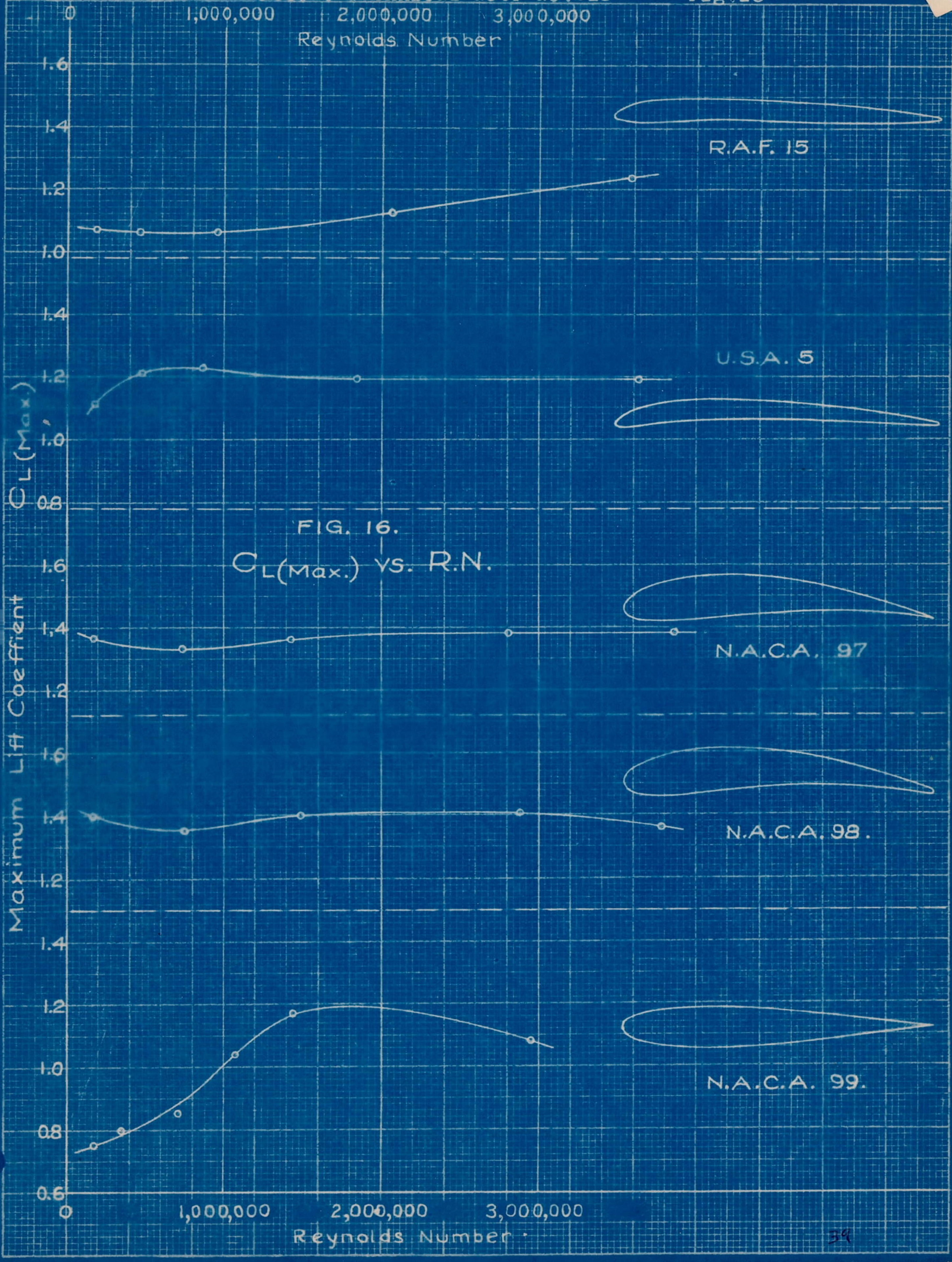


FIG. 16.
$C_{L(Max.)}$ vs. R.N.

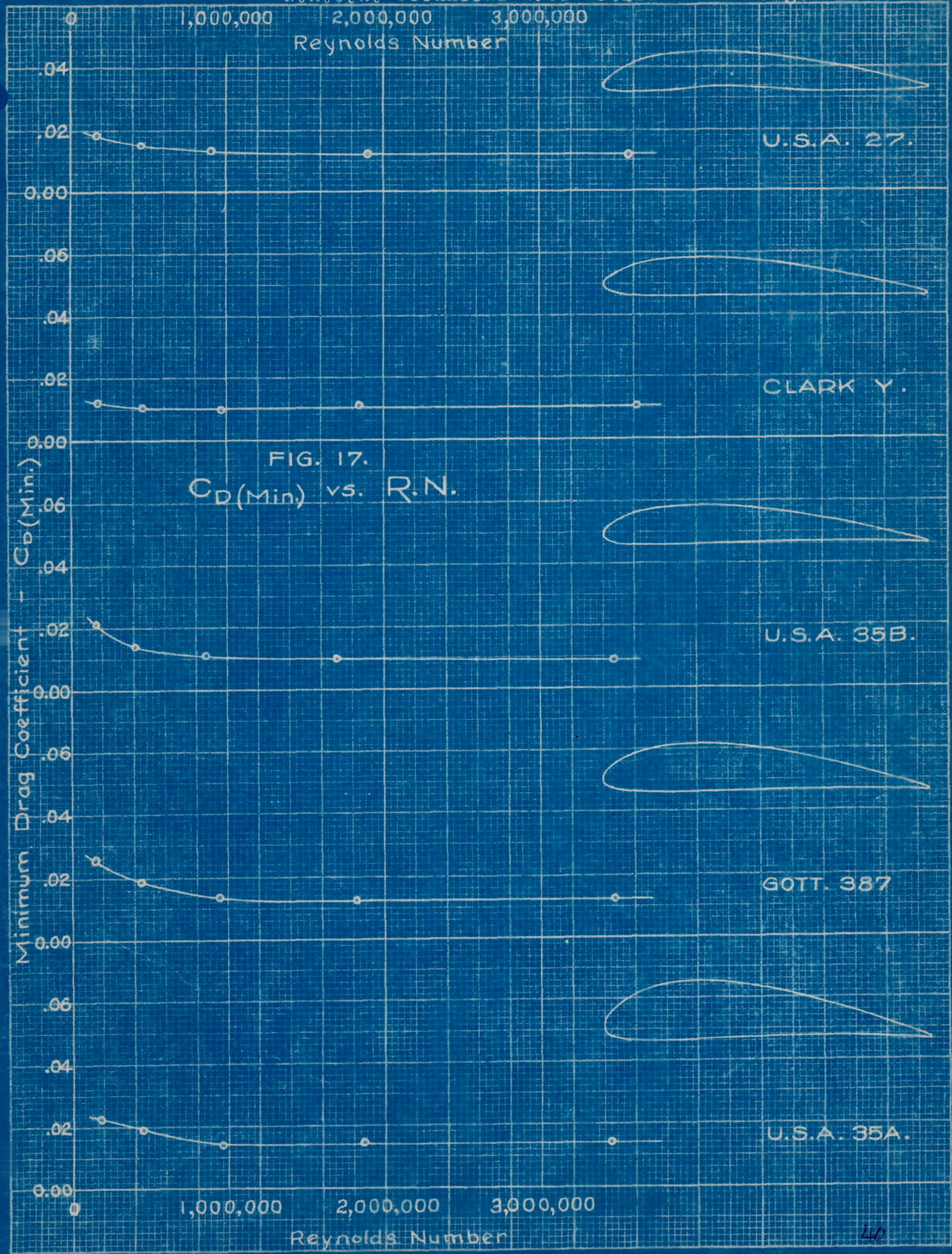
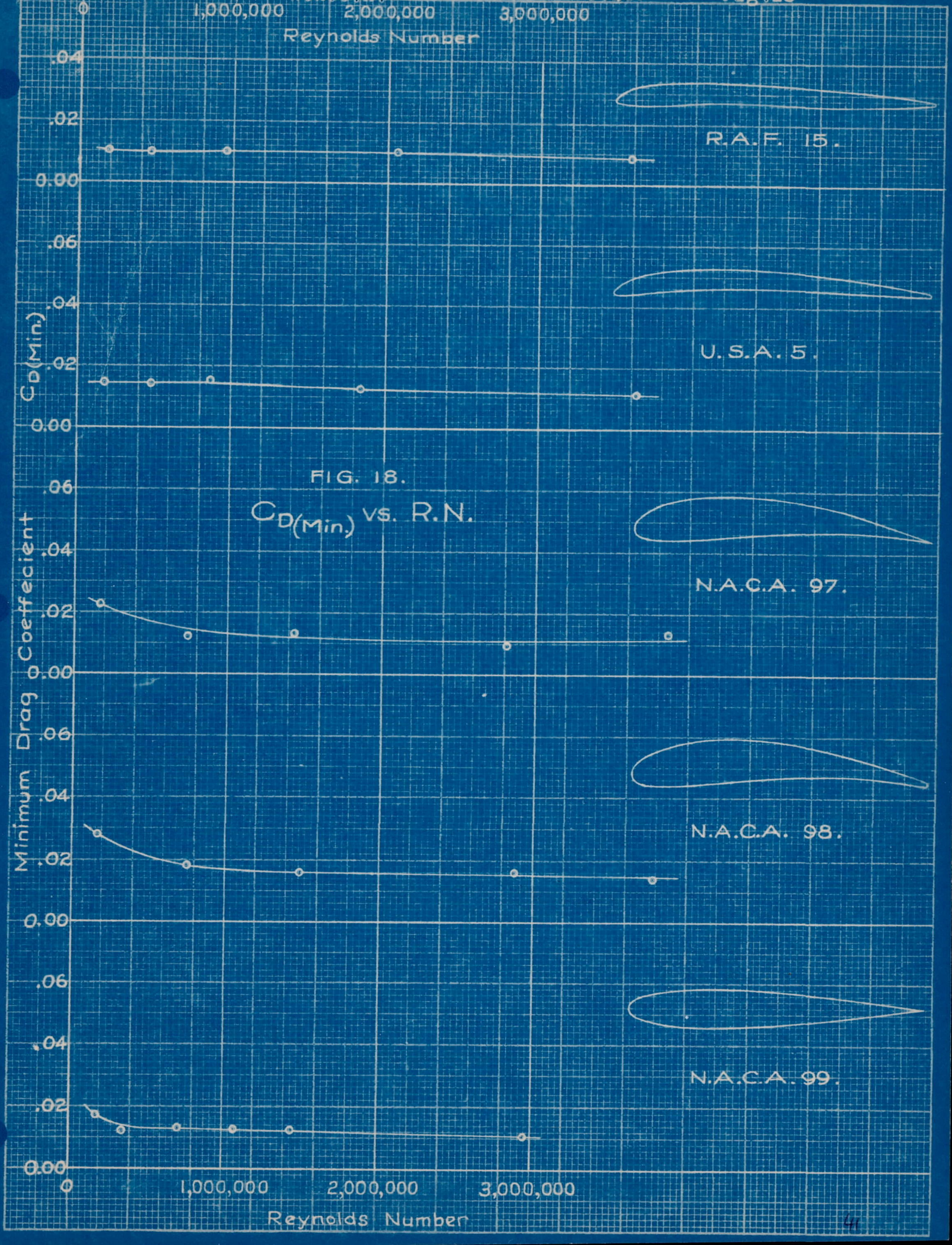


FIG. 17.
$C_{D(Min.)}$ vs. R.N.


0
1,000,000
2,000,000
3,000,000
Reynolds Number
.04
.02
0.00
.06
.04
.02
0.00
.06
.04
.02
0.00
.06
.04
.02
0.00
.06
.04
.02
0.00
R.A.F. 15.
U.S.A. 5.
N.A.C.A. 97.
N.A.C.A. 98.
N.A.C.A. 99.
FIG. 18.
$C_{D(Min.)}$ vs. R.N.
$C_{D(Min.)}$
Minimum Drag Coeffecient
0
1,000,000
2,000,000
3,000,000
Reynolds Number

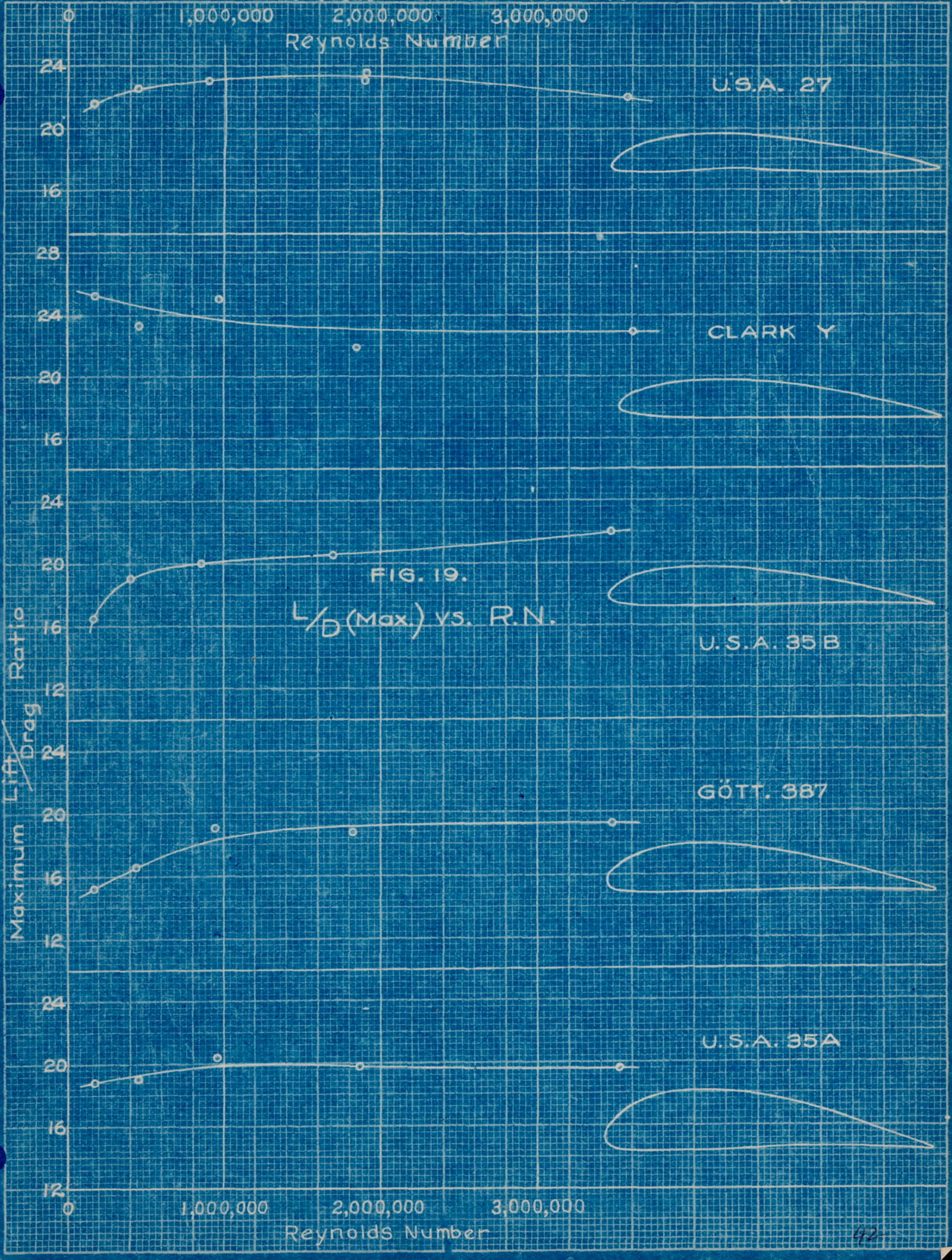
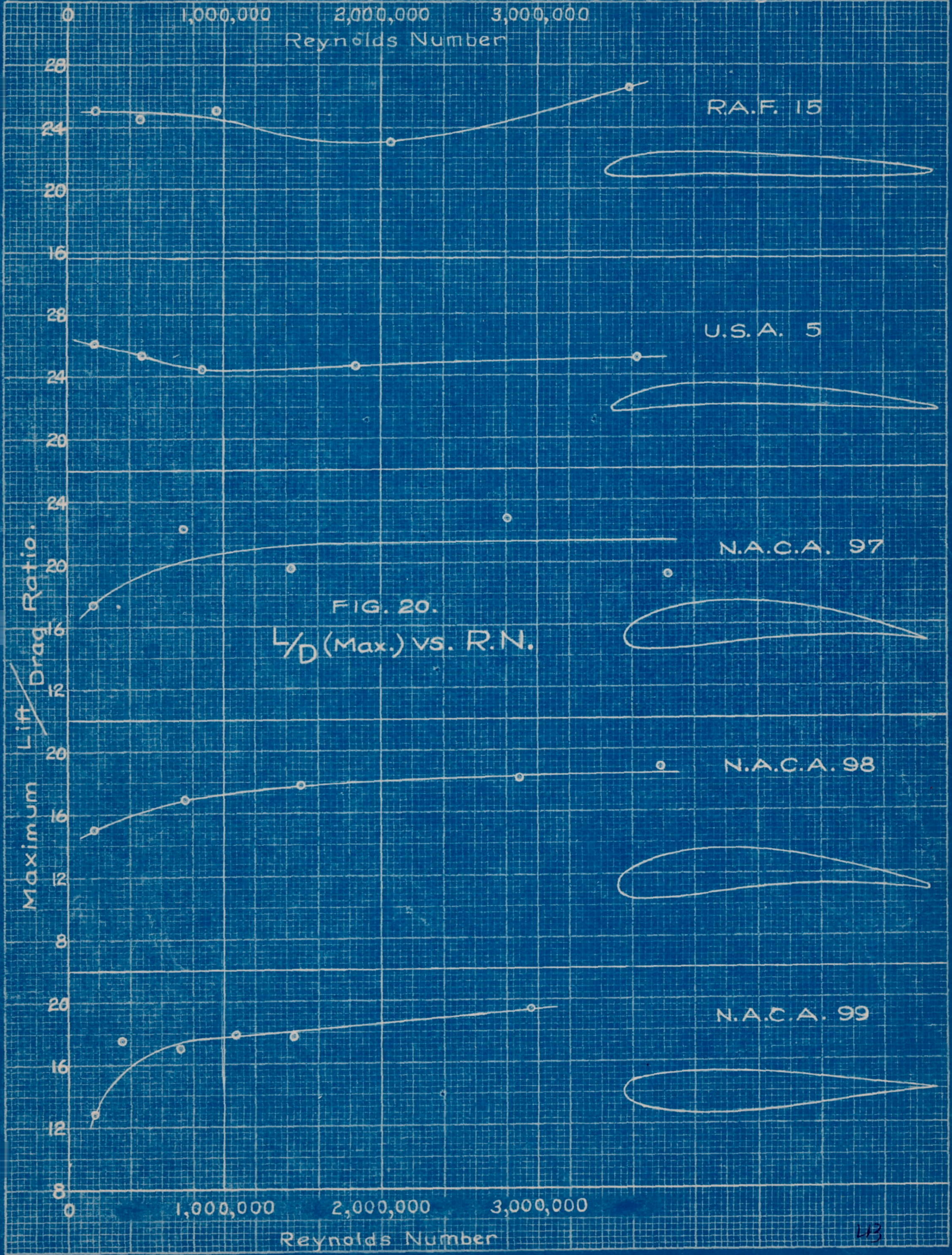


FIG. 19.
L/D (Max.) vs. R.N.


0
1,000,000
2,000,000
3,000,000
Reynolds Number
R.A.F. 15
U.S.A. 5
N.A.C.A. 97
N.A.C.A. 98
N.A.C.A. 99
FIG. 20.
L/D (Max.) vs. R.N.
Maximum Lift/Drag Ratio.
28
24
20
16
12
8

43