REPORT No. 260

THE EFFECT OF A FLAP AND AILERONS ON THE N. A. C. A. M-6 AIRFOIL SECTION

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SUMMARY

This report contains the results obtained at the Langley Memorial Aeronautical Laboratory on an N. A. C. A. M–6 airfoil, fitted with a flap and ailerons, and tested in the variable density wind tunnel at a density of 20 atmospheres. Airfoil characteristics are given for the model up to 48° angle of attack with the flap set at various angles, and also with the ailerons set at similar angles. The approximate lift distribution and the center of pressure variation along the span are determined with the model at 18° angle of attack and with the ailerons displaced 20°. Approximate rolling moment and yawing moment coefficients are determined for the various aileron settings.

A comparison of the calculated angles of zero lift and the calculated lift and moment coefficients with those observed is given in the appendix.

INTRODUCTION

The N. A. C. A. M–6 is a good airfoil section, stable in pitch and with very small center of pressure travel. Consequently in its adoption in airplane design, some knowledge of its behavior under the action of ailerons or of a flap seems desirable. This is particularly true in regard to the center of pressure travel. As no similar tests have been conducted under full-scale conditions, this series was undertaken by the Aerodynamics Staff of the Langley Memorial Aeronautical Laboratory in their variable density wind tunnel. To obtain, at the same time, the effect of controls; that is, flap or ailerons, at high angles, the range of investigation was made to extend from zero lift to an angle of attack of +48°.

THE TEST

The model was a 6-inch by 36-inch duralumin airfoil of the N. A. C. A. M–6 section. A flap and two ailerons were constructed along the trailing edge, see Figures 1 to 4. The ailerons were one-quarter span each in length; and the flap consisted of the remaining portion of the trailing edge between the ailerons. In the tests with the flap, the ailerons were set in line as if integral with the flap. In the aileron tests, the flap was set neutral and the ailerons were set both up or both down. Consequently, in these latter tests, the balance readings were approximately equivalent to double those for a semispan.

The airfoil was mounted in the tunnel in the usual manner except that slight modifications were necessary in some parts of the apparatus to obtain a range of angle of attack from −20° to +48°.

The tests consisted in setting the flap or ailerons to the desired angle and, after compressing the air in the tank to 20 atmospheres, making a normal test, recording data for lift, drag, and pitching moment. Because of the limited counterweight on the standard drag balance, 40 kilograms, the tests had to be run in two parts, the second of which was run with an additional counterweight of 50 kilograms added. Consequently, observations in which the gross drag amounted to between 40 and 50 kilograms had to be omitted.

RESULTS

The results of this series of tests are given in Tables I to XVII and in charts, Figures 5 to 20. The general airfoil characteristics, C_L, C_D, C_M (about the quarter chord), and L/D are given in Tables I to VIII, inclusive, at all angles of attack, one table for each flap setting, −20°, −10°,

$-5°$, $0°$, $+5°$, $+10°$, $+20°$, and $+25°$. Similarly, Tables IX to XV contain the characteristics for the model with different aileron settings, both ailerons at $-20°$, $-10°$, $-5°$, $0°$, $+5°$, $+10°$, and $+20°$. No corrections have been made for tunnel wall interference. Applying the Prandtl formula, the data given herein are correct for an effective aspect ratio of 7.318, whereas the geometrical aspect ratio of the model was 6.00.

Figures 5 and 6 show the true polar (lift and drag coefficient to the same scale) curves for the tests with the flap and with the ailerons, respectively; Figures 7, 8 and 9, 10 show the lift and drag coefficients plotted against the angle of attack. The effect of flap or aileron setting is seen to be uniform in regard to both the lift and the drag. The angle of zero lift varies uniformly with the flap angle through its main range. The rate of change of the lift coefficient, C_L, with angle of attack, α, $\frac{dC_L}{d\alpha}$, for the various conditions is practically constant, verifying that the $\frac{dC_L}{d\alpha}$ of an airfoil is independent, or nearly independent of the shape of the section. Figure 18 shows the variation with flap angle of the maximum lift coefficient, the angle of zero lift, and the total lift angle from zero lift to the burble point.

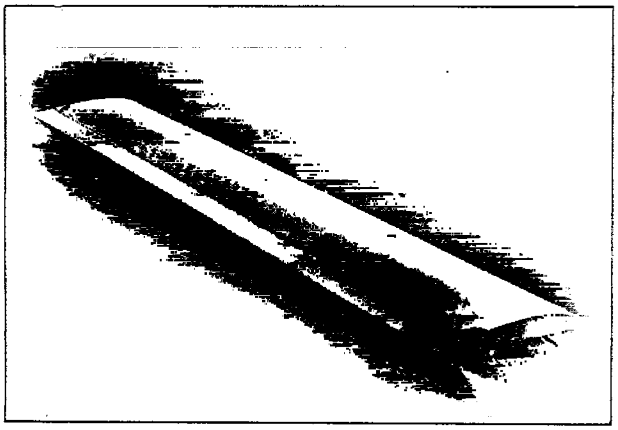

FIG. 1.—N. A. C. A. M-6 airfoil with 20 per cent flaps

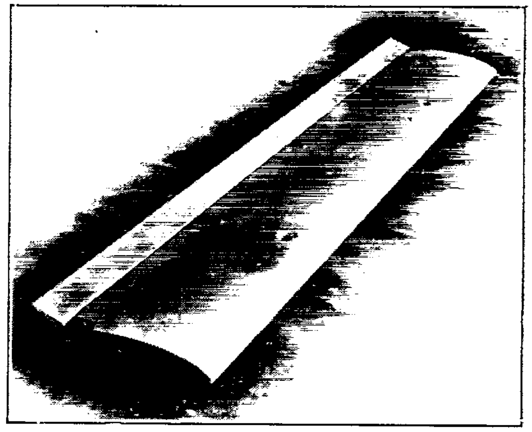

FIG. 2.—N. A. C. A. M-6 airfoil with flaps 20° up

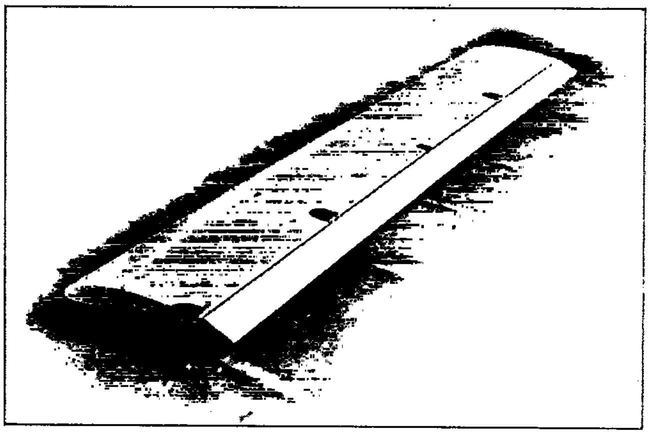

FIG. 3.—N. A. C. A. M-6 airfoil with flaps 20° down

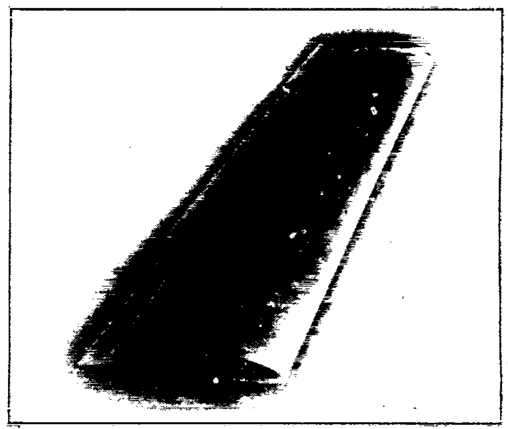

FIG. 4.—N. A. C. A. M-6 airfoil with ailerons displaced 20°

The pitching moment coefficients, about the quarter chord, and the curves of center of pressure travel are given in Figures 11, 12 and 13, 14, respectively, plotted against α for all the various flap and aileron settings.

Figure 15 shows the variation of the center of pressure across the span for the angle of attack of 18° (immediately before the burble point) and an aileron displacement of 20°. It was determined by first assuming a lift distribution along the span, see Figure 16; then computing

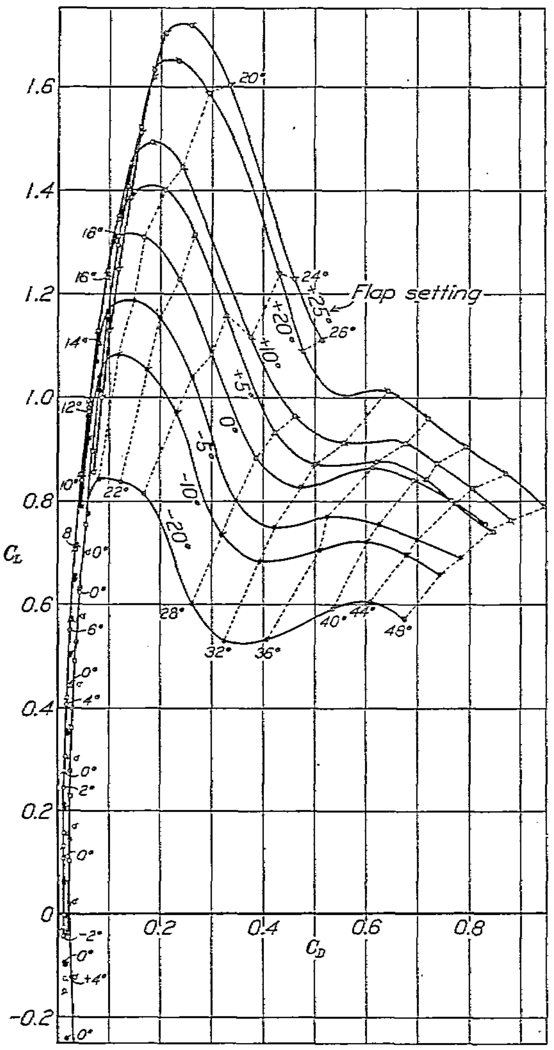

FIG. 5.—Polar curves of N. A. C. A. M-6 airfoil with different flap settings

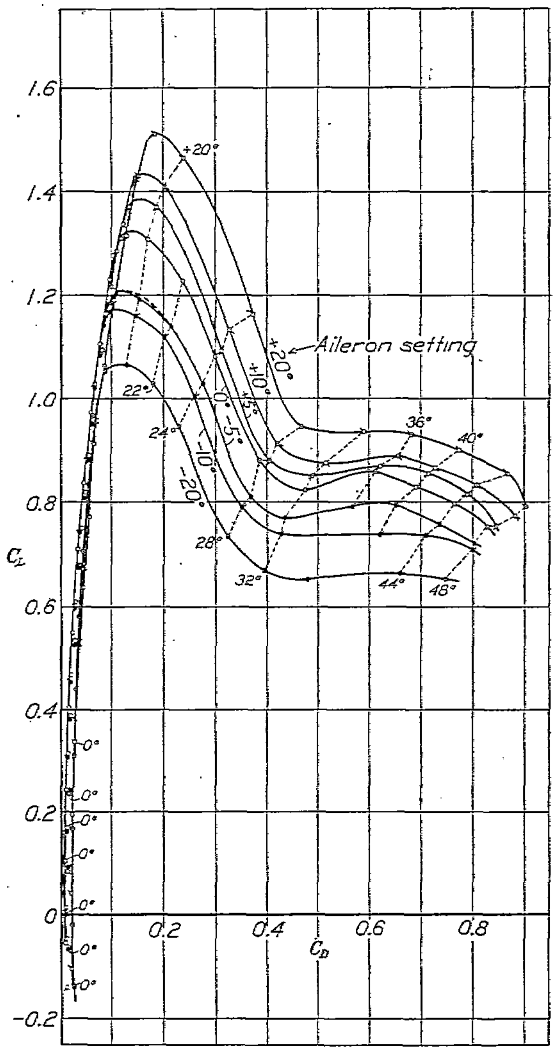

FIG. 6.—Polar curves of N. A. C. A. M-6 airfoil with different aileron settings

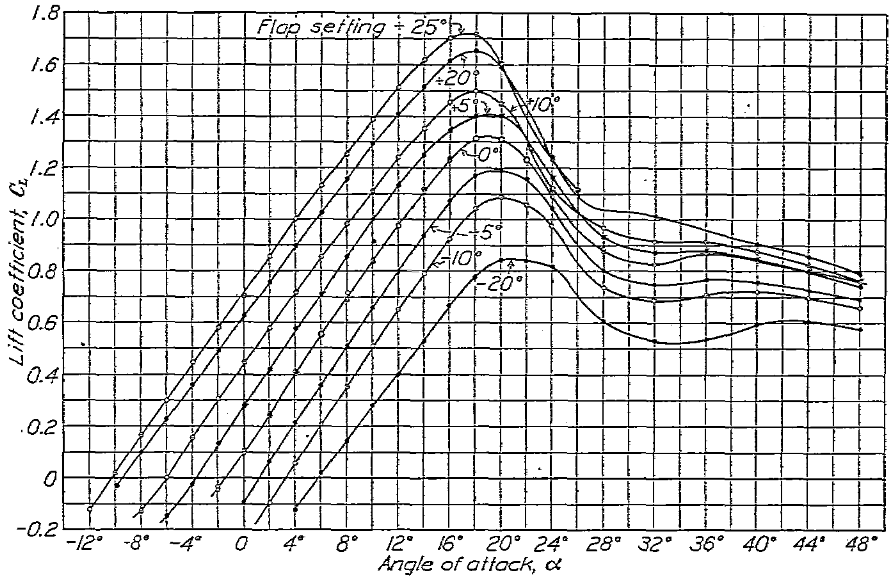

FIG. 7.—Lift coefficient versus angle of attack of N. A. C. A. M-6 airfoil with different flap settings

the lift distribution with the ailerons displaced. This was done by assuming that the total lift of the semispan equals one-half that determined from the aileron tests; and that the lift coefficient at the center is equal to the lift coefficient of the airfoil with the flap neutral, and the lift coefficient at the extreme tip equals that of the model with the flap at +20° or −20° as the case may be. The pitching moment coefficient, C_M, was also treated in a similar manner, see Figure 17. The *C. P.* curve was then determined from these curves by the use of the formula:

$$C.\ P. = 25\% - \frac{C_M}{C_L} \cdot 100\%$$

The above formula holds approximately true for the usual range of lift. The *C. P.* travel is also given (fig. 15) for neutral ailerons. The aileron effect is seen to be large at the outer portion of the airfoil.

Curves showing the variation of rolling and yawning moment coefficients with aileron angle are given in Figures 19 and 20. In each case these values were determined by taking one-half of the difference of the lift and drag values for the up and the down aileron tests and multiplying this quantity by a lever arm equal to the distance from the center of the aileron to the center of the airfoil in terms of the span, or:

$$\text{Rolling moment coefficient, } C_L' = \frac{C_L(+20°) - C_L(-20°)}{2} \times 3/8 \text{ (spans)}$$

$$\text{Yawing moment coefficient, } C_N = \frac{C_D(+20°) - C_D(-20°)}{2} \times 3/8 \text{ (spans)}$$

DISCUSSION

The N. A. C. A. M-6 airfoil section is a stable section in its original form, but when equipped with a flap, its stability characteristics are greatly altered by any change of the position of the flap. The center of pressure travel varies considerably for various flap settings and should be taken into account in designs where ailerons or flaps are used.

From the polar curves and also from the curves of rolling moment coefficient it may be seen that there is still adequate lateral control available at high angles by the use of ailerons provided the yawing tendency can be overcome.

The results given herein have been obtained at 20 atmospheres density and are therefore approximately equivalent to full dynamic scale.

APPENDIX

COMPARISON OF THE MEASURED AND COMPUTED CHARACTERISTICS OF THE M6 AIRFOIL WITH FLAPS

The preceding report contains information about the air forces on an airfoil with a flap, obtained for the first time from force measurement tests simulating full scale conditions. It is of special interest to use such particularly valuable information to throw further light on the aerodynamic theory, and in turn to use the theory to interpret the experimental information. Accordingly, the angle of attack of zero lift and angle of attack of zero moment were computed by means of Munk's integrals. (Reference 2.) These angles were then compared with the observed values.

For this computation the chord is chosen so that it passes through the trailing edge at the point $x=+1$, $\xi=0$, and its length, so that the leading edge is on the line $x=-1$. ξ denotes the ordinates of the mean curve; that is, the curve which is equidistant from the upper and lower curves of the section. The angle of zero lift is then given by:

$$\alpha_0 = -\frac{1}{\pi}\int_{-1}^{+1} \frac{\xi\, dx}{(1-x)\sqrt{1-x^2}}$$

The angle at which the moment about the origin (50 per cent chord) is zero is given by:

$$\alpha'_0 = -\frac{2}{\pi}\int_{-1}^{+1} \frac{x\xi dx}{\sqrt{1-x^2}}$$

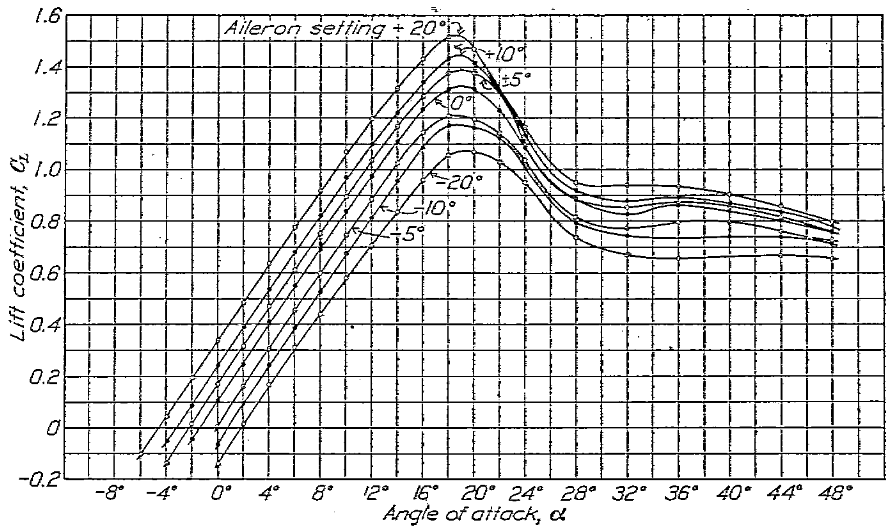

FIG. 8.—Life coefficient versus angle of attack of N. A. C. A. M-6 airfoil with different aileron settings

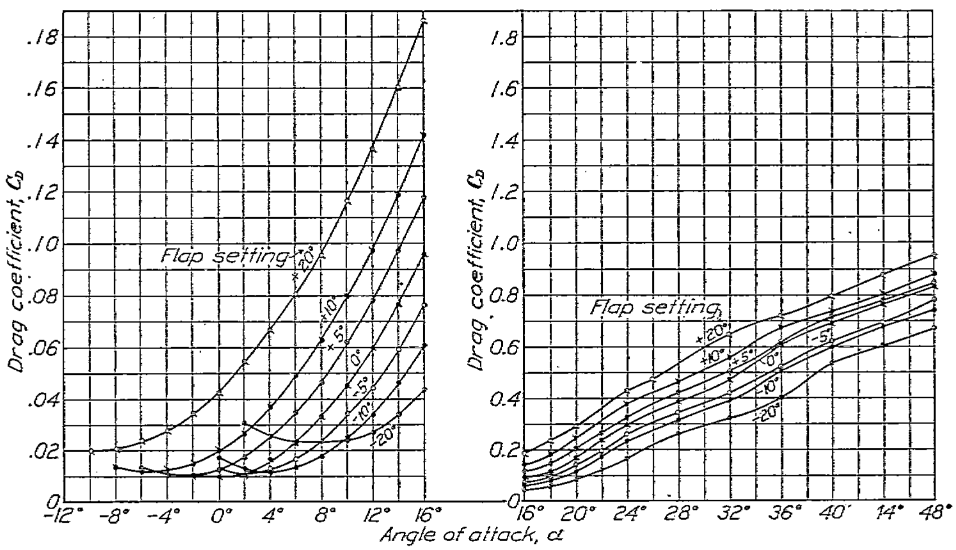

FIG. 9.—Drag coefficient versus angle of attack of N. A. C. A. M-6 airfoil with different flap settings

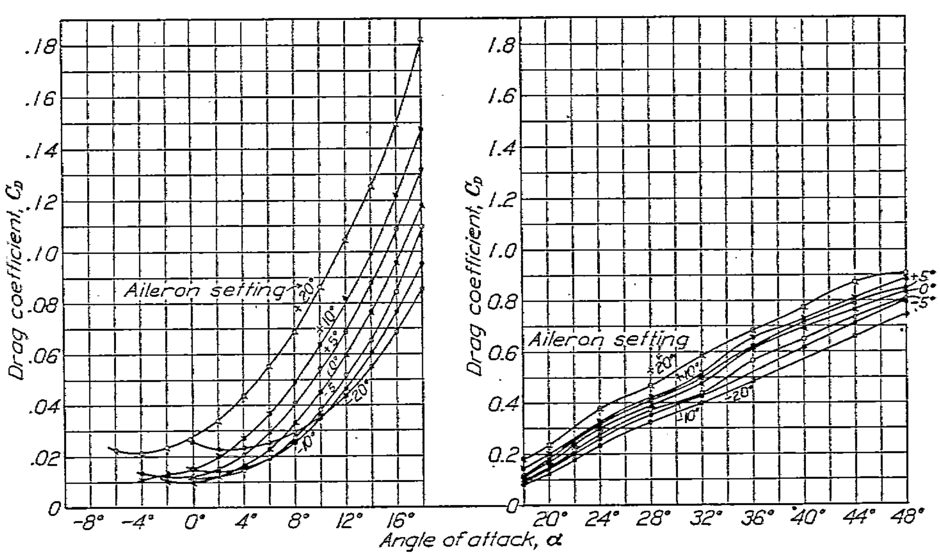

FIG. 10.—Drag coefficient versus angle of attack of N. A. C. A. M-6 airfoil with different aileron settings

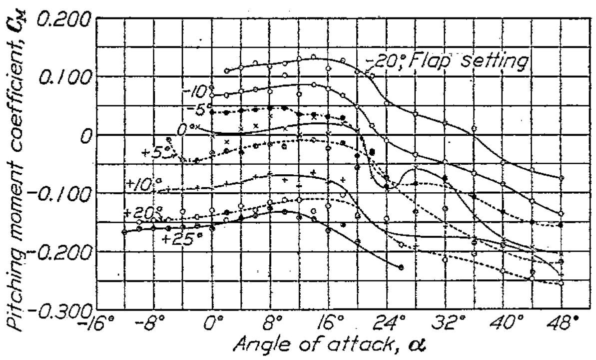

FIG. 11.—Moment coefficient versus angle of attack of N. A. C. A. M-6 airfoil with different flap settings

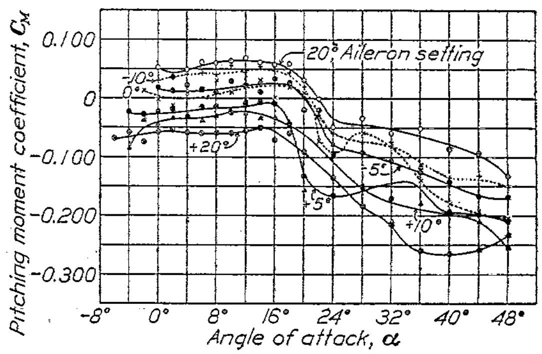

FIG. 12.—Moment coefficient versus angle of attack of N. A. C. A. M-6 airfoil with different aileron settings

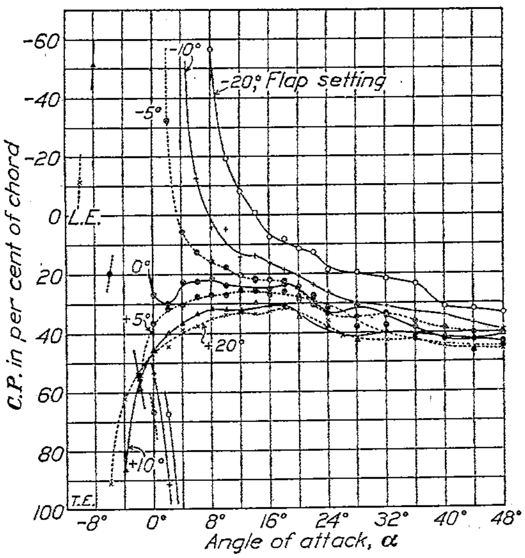

FIG. 13.—Center of pressure versus angle of attack of N. A. C. A. M-6 airfoil with different flap settings

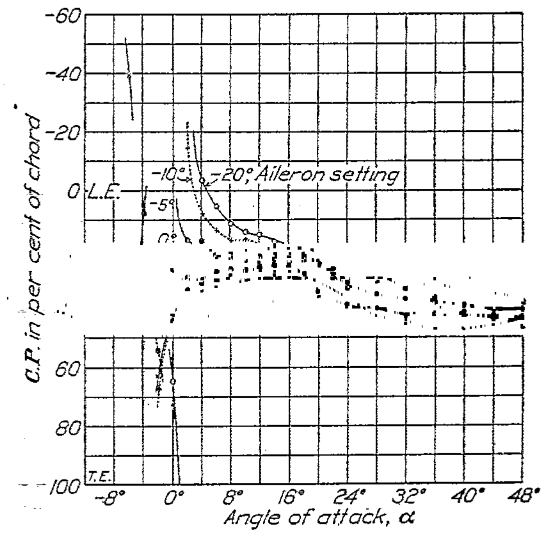

FIG. 14.—Center of pressure versus angle of attack of N. A. C. A. M-6 airfoil with different aileron settings

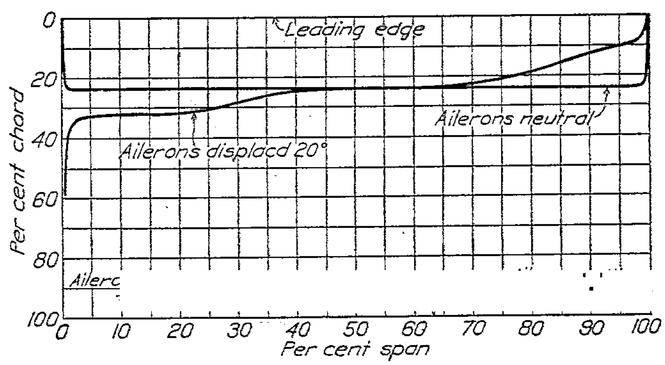

FIG. 15.—Center of pressure variation along span of N. A. C. A. M-6 airfoil with different aileron settings

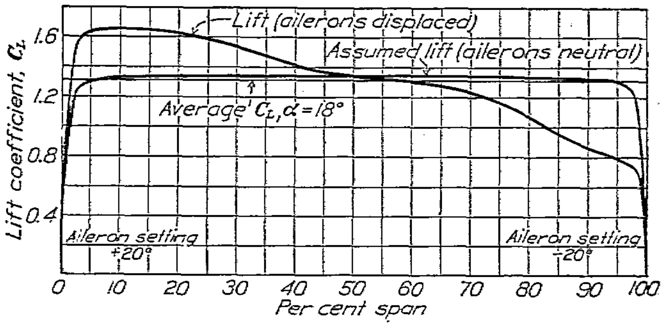

FIG. 16.—Lift distribution along span of N. A. C. A. M-6 airfoil with different aileron settings

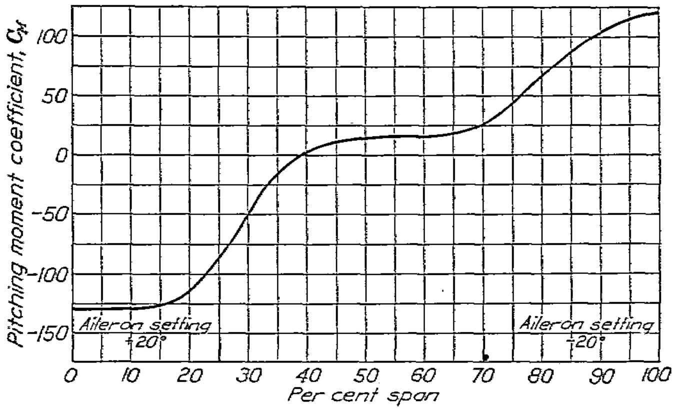

FIG. 17.—Variation of moment along span of N. A. C. A. M-6 airfoil at 18° angle of attack

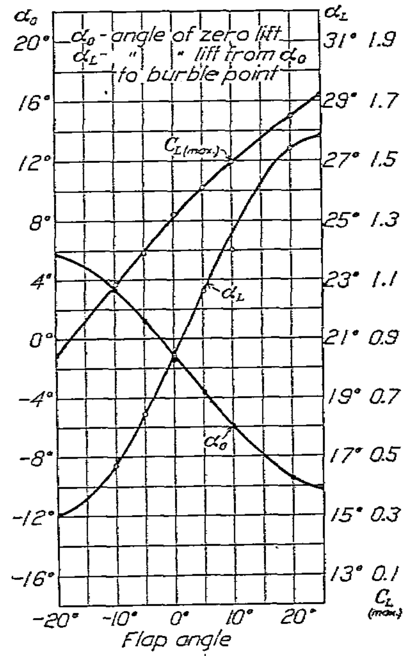

FIG. 18.—Maximum C_L, α_0 and α_L versus flap angle of N. A. C. A. M-6 airfoil

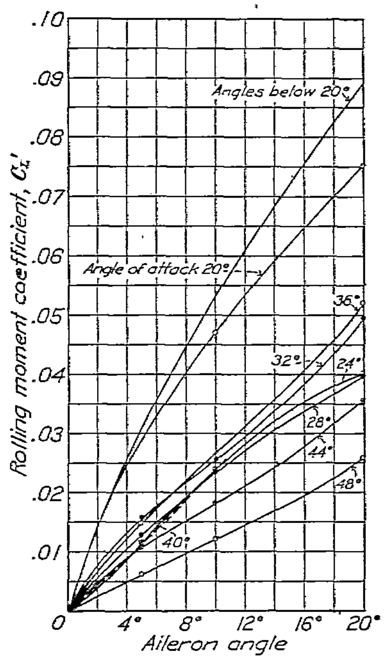

FIG. 19.—Rolling moment versus aileron angle at different angles of attack of N. A. C. A. M-6 airfoil

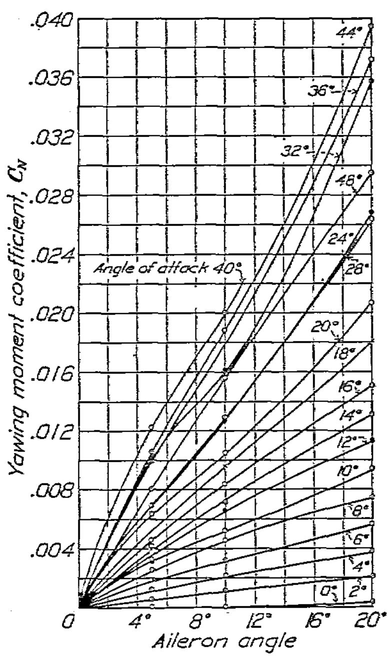

FIG. 20.—Yawing moment versus aileron angle at different angles of attack of N. A. C. A. M-6 airfoil

In the case of a symmetrical section with a flap the mean curve is a broken line as shown in Figure 21.

The value of the angles of zero lift and moment are then

$$\alpha_0 = -\frac{\cos^{-1} h + \sqrt{1-h^2}}{\pi} \tan \beta$$

$$\alpha'_0 = -\frac{\cos^{-1} h - h\sqrt{1-h^2}}{\pi} \tan \beta$$

where β is the angle of displacement of the flap and h is the abscissa of the hinge as shown in the figure, or:

$$h = \frac{S - E \cos \beta}{S + E \cos \beta}$$

where S and E are the lengths of the chords of the stabilizer and elevator, respectively, or in the present case, the fixed part of the airfoil and the flap.

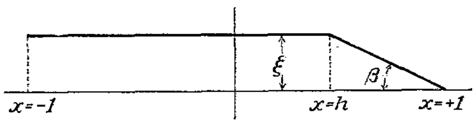

FIG. 21

If the mean curve of the base section is not a straight line it is necessary to correct the angles as given above by adding to them the corresponding angles computed for the undeformed section. These angles were determined by the method given in Reference 1.

TABLE XVIII

1 Flap displacement	2 Computed angle of zero lift	3 Measured angle of zero lift	4 K exp.	5 K exp. / K theor.	6 Computed effective angle of zero moment	7 Measured effective angle of zero moment	8 C_M (Theor.)
−20°	10.67°	5.7°	1.70	0.62	2.33°		0.229
−10°	4.99°	3.3°	2.25	.82	.96°	0.0°	.111
−5°	2.24°	1.2°	2.30	.84	.26°	−.6°	.054
0°	−0.53°	−1.4°			−.45°		−.002
+5°	−3.30°	−3.7°	2.50	.91	−1.16°		−.059
10°	−6.05°	−6.0°	2.40	.87	−1.86°	−2.0°	−.115
20°	−11.73°	−9.5°	2.08	.76	−3.23°	−2.3°	−.233
25°	−14.58°	−10.3°	1.82	.66	−3.84°	−2.7°	−.295

Table XVIII gives the computed angles of zero lift and moment, and also the measured angles, for comparison. Columns giving the value of K and the ratio of its measured to its computed value are also given. K is the coefficient which gives the effect of elevator turning (Reference 2) according to the equation:

$$\alpha \text{ effective} = \frac{E}{E+S} K\beta$$

α effective is the angle of attack of the whole undeformed airfoil which has the same effect as turning the flap by an angle β; as before, E denotes the chord of the elevator and S, the chord of the stabilizer. For small values of β, the theoretical value of K is

$$K = \frac{E+S}{E} \cdot \frac{\cos^{-1} h + \sqrt{1-h^2}}{\pi}$$

$$h = \frac{S-E}{S+E}$$

From this expression the theoretical value of K is 2.75 for 20 per cent flaps. The ratio of the value of K deduced from the experiments to the above theoretical value is given in column 5 of Table XVIII as a measure of the efficiency of the elevator. These ratios indicate that the efficiency of the elevator is greatest for small displacements and larger when the displacement is down rather than up. The fact that the elevator effect, as compared with the effect as given by the theoretical calculation, falls off rapidly when the displacement angle is over 10°, is shown by both the figures in Table XVIII and the curve in Figure 22.

From the computed angles of zero moment and zero lift, the theoretically constant value of the moment coefficient, C_M, about the quarter chord point was found. (Reference 1.)

$$C_M = \frac{2\pi}{4}\,(\alpha_0 - \alpha^1_0)$$

where 2π is the theoretical slope of the lift curve for two dimensional flow. The computed values of C_M are plotted in Figure 23 together with the values as determined from the experiments.

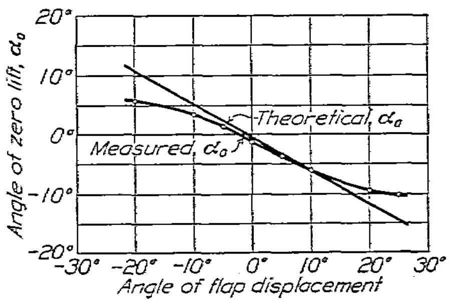

FIG. 22.—Comparison of the theoretical with the measured angles of zero lift in relation to the flap displacement angle

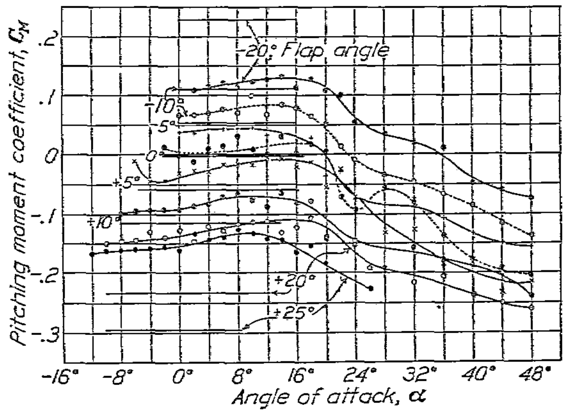

FIG. 23.—Comparison of the theoretical with the measured values of moment coefficient for various flap displacement angles

As in the case of the lift, displacing the flap produced a smaller change in the moment coefficient than the change indicated by the theoretical calculation, especially for large flap displacement angles. The moment coefficient as determined from the tests is in agreement with the theory inasmuch as it is approximately independent of the angle of attack below the burble point. Above this point the theory, of course, does not apply so it is not surprising to find that the value of the moment coefficient falls off.

In order to throw some light on what was considered a rather large discrepancy between observed and computed angles of zero lift, the flow pattern around the airfoil was studied near the angle of zero lift. The flap was set down 20° and the airfoil placed in a 6-inch wind tunnel so that the end of the wing rested on a plate which was coated with lamp black and kerosene. The airfoil was set at an angle with the air stream corresponding to the computed angle of zero lift. The resulting pattern, a photograph of which will be found in Figure 24, reveals at once the reason for the discrepancy between the observed and computed angles of zero lift.

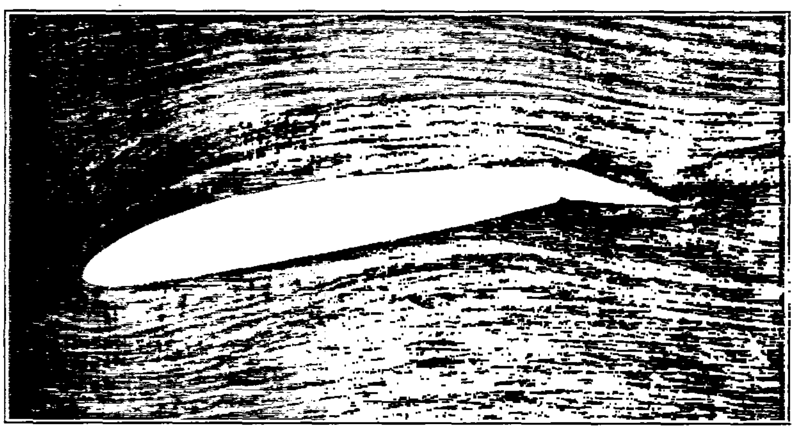

FIG. 24.—Flow pattern around airfoil at theoretical angle of zero lift

The burble region arising at the hinge and extending back over the upper surface of the flap destroys the smooth flow and the lift on the flap so that the total experimental lift is negative in this position instead of zero as indicated by the theory, which assumes a potential flow. If there were no irregularity in the upper surface curve at the hinge the flow would undoubtedly approach more nearly the potential flow assumed by the theory and a better agreement with the theory would result. In every case investigated where the drag has been low at the angle of zero lift, indicating a close approach to a potential flow, the measured angle of zero lift has been found to be very near its theoretical value. The theory then shows in this case that a considerable part of the elevator effect from the flap is lost because of the surface irregularity between the wing and the flap.

REFERENCES

1. Munk, Max M. General Theory of Thin Wing Sections. N. A. C. A. Technical Report No. 142. 1922.

2. Munk, Max M. The Determination of the Angles of Attack of Zero Lift and of Zero Moment Based on Munk's Integrals. N. A. C. A. Technical Note No. 122. 1923.

TABLE I

Span 91.44 cm.
Chord 15.24 cm.
Area .1393 m².

Test No. 202.
Airfoil N. A. C. A. M-6 (6″ by 36″) with 20 per cent c flaps.
Average tank pressure, 20.6 atmospheres.
Average dynamic pressure, 590 kg/m².
Average Reynolds Number, 3,890,000.

FLAPS SET 20° UP

α Degrees	C_L	C_D	L/D	C_M	C. P. per cent chord
2	−0.262	0.0308	−8.51	+0.110	+67.2
4	−.124	.0257	−4.82	.117	+120.9
6	+.018	.0233	+.77	.123	−586.5
8	.143	.0235	6.08	.117	−55.7
10	.277	.0238	11.64	.123	−19.4
12	.398	.0270	14.74	.131	−8.2
14	.526	.0340	15.47	.134	−.9
16	.662	.0437	15.15	.114	+7.4
18	.775	.0588	13.18	.128	8.1
20	.842	.0804	10.47	.109	11.7
22	.835	.1227	6.80	.101	12.6
24	.813	.1676	4.85	.054	18.3
28	.602	.2612	2.30	.035	19.6
32	.530	.3229	1.64	.020	21.8
36	.533	.4061	1.31	.011	23.3
40	.592	.5534	1.11	−.048	31.0
44	.605	.6062	1.00	−.060	32.0
48	.572	.6736	.85	−.074	33.3

TABLE II

Span 91.44 cm.
Chord 15.24 cm.
Area .1393 m².

Test No. 201.
Airfoil N. A. C. A. M-6 (6″ by 36″) with 20 per cent c flaps.
Average tank pressure, 20.6
Average pressure,
Average Number, 4,130,000.

FLAPS SET 10° UP

α Degrees	C_L	C_D	L/D	C_M	C. P. per cent chord
0	−0.242	0.0172	−14.07	0.068	+53.1
2	−.099	.0129	−7.67	.065	+91.0
4	+.058	.0115	+5.04	.074	−101.0
6	.204	.0136	15.00	.077	−12.7
8	.349	.0178	19.61	.073	+4.0
10	.505	.2520	20.04	.102	4.6
12	.647	.0339	19.09	.069	14.2
14	.787	.0461	17.07	.086	13.9
16	.920	.0609	15.11	.079	16.2
18	1.040	.0809	12.86	.066	18.5
20	1.084	.1162	9.33	.048	20.5
22	1.054	.1743	6.05	.015	23.6
24	.971	.2306	4.21	−.010	26.0
28	.735	.3179	2.31	−.034	29.7
32	.684	.3909	1.75	−.045	30.9
36	.705	.5088	1.39	−.067	32.8
40	.721	.6002	1.20	−.084	34.0
44	.697	.6773	1.03	−.115	36.8
48	.659	.7443	.88	−.137	38.9

TABLE III

Span	91. 44 cm.
Chord	15. 24 cm.
Area	. 1393 m².

Test No. 200.
Airfoil N. A. C. A. M-6 (6″ by 36″) with 20 per cent *c* flaps.
Average tank pressure, 20.5 atmospheres.
Average dynamic pressure, 615 kg/m².
Average Reynolds Number, 4,150,000.

FLAPS SET 5° UP

α Degrees	C_L	C_D	L/D	C_M	C. P.
0	−0. 094	0. 0130	−7. 23	0. 039	+66. 5
2	+. 064	. 0108	+5. 93	. 037	−32. 5
4	. 211	. 0131	16. 11	. 041	+5. 6
6	. 353	. 0165	21. 39	. 044	12. 5
8	. 502	. 0234	21. 45	. 046	15. 7
10	. 655	. 0341	19. 21	. 046	17. 9
12	. 795	. 0444	19. 90	. 035	20. 5
14	. 936	. 0590	15. 86	. 034	21. 3
16	1. 070	. 0761	14. 06	. 032	21. 9
18	1. 168	. 0972	12. 02	. 029	22. 4
20	1. 187	. 1426	8. 32	−. 036	28. 1
22	1. 153	. 1984	5. 81	−. 031	27. 7
24	1. 040	. 2608	3. 99	−. 088	33. 4
28	. 800	. 3472	2. 30	−. 083	34. 6
32	. 748	. 4215	1. 78	−. 074	33. 6
36	. 767	. 5242	1. 46	−. 106	36. 3
40	. 754	. 6240	1. 21	−. 132	38. 5
44	. 726	. 6977	1. 04	−. 151	40. 0
48	. 690	. 7834	. 88	−. 154	39. 7

TABLE IV

Span	91. 44 cm.
Chord	15. 24 cm.
Area	. 1393 m².

Test No. 194.
Airfoil N. A. C. A. M-6 (6″ by 36″) with 20 per cent *c* flaps.
Average tank pressure, 20.8 atmospheres.
Average dynamic pressure, 620 kg/m².
Average Reynolds Number, 4,180,000.

FLAPS SET AT 0°

α Degrees	C_L	C_D	L/D	C_M	C. P. per cent chord
−2	−0. 044	0. 0103	−4. 27	+0. 013	54. 4
0	+. 106	. 0099	+10. 71	−. 002	26. 9
2	. 243	. 0107	22. 71	−. 012	29. 9
4	. 407	. 0165	24. 67	+. 010	22. 5
6	. 550	. 0228	24. 12	. 015	22. 3
8	. 690	. 0326	21. 17	. 032	21. 4
10	. 836	. 0451	18. 54	. 010	23. 8
12	. 972	. 0597	16. 28	−. 001	25. 1
14	1. 107	. 0765	14. 47	+. 031	22. 1
16	1. 233	. 0958	12. 87	. 001	24. 9
18	1. 310	. 1178	11. 12	. 018	23. 6
20	1. 308	. 1692	7. 73	. 008	24. 4
22	1. 227	. 2374	5. 17	−. 072	30. 9
24	1. 082	. 2991	3. 62	−. 094	33. 5
28	. 882	. 3868	2. 28	−. 057	31. 0
32	. 824	. 4751	1. 73	−. 080	33. 4
36	. 863	. 6134	1. 41	−. 139	38. 1
40	. 838	. 6925	1. 21	−. 179	41. 7
44	. 800	. 7660	1. 04	−. 191	42. 3
48	. 755	. 8315	. 91	−. 203	43. 1

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TABLE V

Span .... 91.44 cm.
Chord .... 15.24 cm.
Area .... .1393 m².

Test No. 196.
Airfoil N. A. C. A. M-6 (6″ by 36″) with 20 per cent c flaps.
Average tank pressure, 20.6 atmospheres.
Average dynamic pressures, 620 kg/m².
Average Reynolds Number, 4,205,000.

FLAPS SET 5° DOWN

α Degrees	C_L	C_D	L/D	C_M	C. P. per cent chord
−6	−0.150	0.0134	−11.19	−0.008	+19.6
−4	−.027	.0103	−2.62	−.043	−145.4
−2	+.130	.0102	+12.74	−.045	+59.4
0	.277	.0123	22.52	−.032	36.5
2	.418	.0174	24.02	−.027	31.5
4	.573	.0256	22.38	−.030	30.2
6	.707	.0346	20.43	−.019	27.7
8	.851	.0464	18.34	−.016	26.9
10	.991	.0621	15.96	−.020	27.0
12	1.127	.0780	14.45	−.003	25.3
14	1.250	.0971	12.87	−.008	25.6
16	1.344	.1179	11.40	−.024	26.9
18	1.401	.1496	9.36	−.013	25.9
20	1.402	.2090	6.71	−.056	29.0
22	1.317	.2682	4.91	−.027	27.1
24	1.158	.3282	3.53	−.073	31.2
28	.931	.4227	2.02	−.132	38.0
32	.872	.5028	1.73	−.126	37.5
36	.875	.6208	1.41	−.185	42.3
40	.841	.7190	1.17	−.188	41.9
44	.794	.7793	1.02	−.235	46.1
48	.740	.8476	.87	−.216	44.3

TABLE VI

Span .... 91.44 cm.
Chord .... 15.24 cm.
Area .... .1393 m².

Test No. 197.
Airfoil N. A. C. A. M-6 (6″ by 36″) with 20 per cent c flaps.
Average tank pressure, 20.7 atmospheres.
Average dynamic pressure, 623 kg/m².
Average Reynolds Number, 4,320,000.

FLAPS SET 10° DOWN

α Degrees	C_L	C_D	L/D	C_M	C. P.
−8	−0.127	0.0136	−9.34	−0.098	−51.6
−6	−.002	.0118	−.17	−.094	+1765.0
−4	+.155	.0122	+12.70	−.094	86.3
−2	.304	.0148	20.54	−.092	55.2
0	.442	.0201	21.99	−.092	45.8
2	.574	.0264	21.74	−.085	39.8
4	.715	.0369	19.38	−.082	36.5
6	.852	.0491	17.35	−.071	33.3
8	.980	.0627	15.63	−.066	31.7
10	1.106	.0796	13.89	−.077	32.0
12	1.238	.0971	12.75	−.088	32.2
14	1.350	.1186	11.38	−.065	29.8
16	1.453	.1420	10.22	−.081	30.7
18	1.494	.1838	8.13	−.077	30.3
20	1.447	.2424	5.97	−.142	34.8
24	1.116	.3752	2.97	−.152	38.0
28	.964	.4621	2.09	−.190	42.9
32	.912	.5561	1.64	−.157	39.8
36	.910	.6757	1.35	−.176	40.5
40	.874	.7389	1.18	−.186	41.3
44	.824	.8052	1.02	−.200	42.3
48	.761	.8816	.86	−.239	40.5

TABLE VII

Span	91.44 cm.	Test No. 198.
Chord	15.24 cm.	Airfoil N. A. C. A. M–6 (6″ by 36″) with 20 per cent *c* flaps.
Area	.1393 m².	Average tank pressure, 20.8 atmospheres.
		Average dynamic pressure, 620 kg/m².
		Average Reynolds Number, 4,170,000.

FLAPS SET 20° DOWN

α Degrees	C_L	C_D	L/D	C_M	C. P. per cent chord
−10	−0.038	0.0197	−1.93	−0.150	−11.8
−8	+.100	.0203	+4.93	−.147	+178.4
−6	.226	.0237	9.54	−.147	91.0
−4	.359	.0273	13.15	−.131	62.0
−2	.490	.0344	14.24	−.141	53.8
0	.624	.0424	14.72	−.130	45.8
2	.754	.0549	13.73	−.146	44.4
4	.896	.0665	13.47	−.123	38.6
6	1.027	.0807	12.73	−.130	37.7
8	1.157	.0953	12.14	−.116	35.0
10	1.293	.1168	11.07	−.112	33.7
12	1.408	.1363	10.33	−.112	33.0
14	1.521	.1606	9.47	−.132	33.7
16	1.631	.1866	8.74	−.123	32.6
18	1.650	.2343	7.04	−.103	31.2
20	1.589	.2945	5.40	−.140	33.8
24	1.240	.4311	2.88	−.144	35.9
26	1.090	.4775	2.28	−.189	41.0
28					
32	1.012	.6410	1.58	−.214	42.7
36	.959	.7133	1.34	−.203	42.0
40	.904	.7931	1.14	−.233	44.4
44	.852	.8729	.98	−.245	45.1
48	1.787	.9463	.83	−.254	45.6

TABLE VIII

Span	91.44 cm.	Test No. 199.
Chord	15.24 cm.	Airfoil N. A. C. A. M–6 (6″ by 36″) with 20 per cent *c* flaps.
Area	.1393 m².	Average tank pressure, 20.6 atmospheres.
		Average dynamic pressure, 613 kg/m².
		Average Reynolds Number, 4,110,000.

FLAPS SET 25° DOWN

α Degrees	C_L	C_D	L/D	C_M	C. P. per cent chord
−12	−0.124	0.0327	−3.79	−0.167	−105.3
−10	+.020	.0314	+.64	−.162	+1161.0
−8	.169	.0317	5.33	−.161	+124.4
−6	.298	.0351	8.49	−.161	80.3
−4	.442	.0408	10.83	−.159	61.3
−2	.576	.0484	11.90	−.146	50.5
0	.702	.0570	12.32	−.162	48.1
2	.855	.0705	12.13	−.134	40.6
4	1.001	.0851	11.76	−.144	39.3
6	1.129	.1009	11.19	−.140	37.3
8	1.250	.1180	10.59	−.127	35.2
10	1.389	.1405	9.89	−.133	34.6
12	1.510	.1613	9.36	−.143	34.5
14	1.619	.1849	8.76	−.144	33.9
16	1.703	.2098	8.12	−.165	34.7
18	1.718	.2581	6.65	−.153	33.9
20	1.605	.3336	4.81	−.184	36.4
24	1.231	.4608	2.67	−.337	51.6
26	1.112	.5144	2.16	−.227	43.6

TABLE IX

Span 91. 44 cm.
Chord 15. 24 cm.
Area .1393 m^2.

Test No. 208.
Airfoil N. A. C. A. M–6 (6″ by 36″) with 20 per cent c flaps.
Average tank pressure, 20.4 atmospheres.
Average dynamic pressure, 602 kg/m^2.
Average Reynolds Number, 4,100,000.

AILERONS SET 20° DOWN

α Degrees	C_L	C_D	L/D	C_M	C. P. per cent chord
−6	−0. 103	0. 0223	−4. 62	−0. 068	−39. 5
−4	+. 043	. 0213	+2. 02	−. 057	+162. 5
−2	. 195	. 0233	8. 37	−. 074	63. 0
0	. 336	. 0275	12. 22	−. 057	42. 0
2	. 483	. 0337	14. 33	−. 053	36. 0
4	. 637	. 0437	14. 58	−. 058	34. 1
6	. 772	. 0552	13. 99	−. 059	32. 6
8	. 914	. 0686	13. 32	−. 059	31. 5
10	1. 063	. 0862	12. 33	−. 059	30. 5
12	1. 193	. 1044	11. 43	−. 055	29. 6
14	1. 315	. 1252	10. 50	−. 053	29. 0
16	1. 429	. 1494	9. 56	−. 071	30. 1
18	1. 512	. 1819	8. 31	−. 059	28. 9
20	1. 468	. 2384	6. 16	−. 092	31. 3
24	1. 163	. 3741	3. 11	−. 134	36. 1
28	. 948	. 4659	2. 04	−. 183	42. 5
32	. 936	. 5863	1. 60	−. 215	44. 4
36	. 932	. 6804	1. 37	−. 258	47. 4
40	. 901	. 7746	1. 16	−. 263	47. 2
44	. 856	. 8698	. 98	−. 256	46. 0
48	. 792	. 9042	. 88	−. 233	44. 4

TABLE X

Span 91.44 cm.
Chord 15.24 cm.
Area .1393 m^2.

Test No. 207.
Airfoil N. A. C. A. M–6 (6″ by 36″) with 20 per cent c flaps.
Average tank pressure, 20.6 atmospheres.
Average [illegible] pressure, 610 kg/m^2.
Average [illegible] Number, 4,130,000.

AILERONS SET 10° DOWN

α degrees	C_L	C_D	L/D	C_M	C. P. per cent chord
−4	−0. 051	0. 0108	−4. 72	−0. 084	−138. 0
−2	+. 087	. 0131	+6. 64	−. 037	+67. 3
0	. 240	. 0147	16. 33	−. 044	43. 3
2	. 390	. 0201	19. 40	−. 034	33. 7
4	. 536	. 0270	19. 85	−. 031	30. 8
6	. 679	. 0363	18. 70	−. 039	30. 8
8	. 821	. 0491	16. 72	−. 030	28. 7
10	. 969	. 0635	15. 26	−. 018	26. 9
12	1. 097	. 0816	13. 44	−. 025	27. 3
14	1. 219	. 0990	12. 31	−. 034	27. 8
16	1. 339	. 1219	10. 98	−. 031	27. 3
18	1. 431	. 1472	9. 72	−. 042	28. 0
20	1. 413	. 2011	7. 03	−. 156	36. 1
24	1. 135	. 3280	3. 46	−. 078	31. 7
28	. 916	. 4206	2. 18	−. 152	40. 2
32	. 878	. 5146	1. 71	−. 208	45. 5
36	. 891	. 6552	1. 36	−. 179	41. 2
40	. 868	. 7283	1. 19	−. 197	42. 4
44	. 835	. 8095	1. 03	−. 208	42. 9
48	. 775	. 8819	. 88	−. 252	46. 4

TABLE XI

Span	91.44 cm.	Test No. 206.
Chord	15.24 cm.	Airfoil N. A. C. A. M–6 (6″ by 36″) with 20 per cent *c* flaps.
Area	.1393 m².	Average tank pressure, 20.7 atmospheres.
		Average dynamic pressure, 622 kg/m².
		Average Reynolds Number, 4,200,000.

AILERONS SET 5° DOWN

α degrees	C_L	C_D	L/D	C_M	$C.P.$ per cent chord
−4	−0.135	0.0135	−10.00	−0.023	7.8
−2	+.015	.0122	+1.23	−.027	209.0
0	.170	.0122	13.93	−.023	38.5
2	.315	.0151	20.86	−.018	30.7
4	.469	.0207	22.66	−.031	31.6
6	.610	.0294	20.75	−.015	27.5
8	.751	.0407	18.45	−.013	26.7
10	.892	.0545	16.37	−.025	27.8
12	1.032	.0687	15.02	−.009	25.9
14	1.166	.0880	13.25	−.007	25.6
16	1.285	.1086	11.83	−.008	25.6
18	1.372	.1319	10.40	−.043	28.2
20	1.372	.1862	7.37	−.132	34.7
24	1.095	.3130	3.50	−.166	39.0
28	.882	.4075	2.16	−.149	40.4
32	.853	.4904	1.74	−.171	42.7
36	.872	.6208	1.41	−.119	36.1
40	.858	.7159	1.20	−.193	42.3
44	.816	.7902	1.03	−.198	42.4
48	.755	.8465	.89	−.210	43.5

TABLE XII

Span	91.44 cm.	Test No. 194.
Chord	15.24 cm.	Airfoil N. A. C. A. M–6 (6″ by 36″) with 20 per cent *c* flaps.
Area	.1393 m².	Average tank pressure, 20.8 atmospheres.
		Average dynamic pressure, 620 kg/m².
		Average Reynolds Number, 4,180,000.

AILERONS SET AT 0°

α Degrees	C_L	C_D	L/D	C_M	$C.P.$ per cent chord
−2	−0.044	0.0103	−4.27	+0.013	54.4
0	+.106	.0099	+10.71	−.002	26.9
2	.243	.0107	22.71	−.012	29.9
4	.407	.0165	24.67	+.010	22.5
6	.550	.0228	24.12	+.015	22.3
8	.690	.0326	21.17	+.032	21.4
10	.836	.0451	18.54	+.010	23.8
12	.972	.0597	16.28	−.001	25.1
14	1.107	.0765	14.47	+.031	22.1
16	1.233	.0958	12.87	+.001	24.9
18	1.310	.1178	11.12	+.018	23.6
20	1.308	.1692	7.73	+.008	24.4
22	1.227	.2374	5.17	−.072	30.9
24	1.082	.2991	3.62	−.094	33.5
28	.882	.3868	2.28	−.057	31.0
32	.824	.4751	1.73	−.080	33.4
36	.863	.6134	1.41	−.139	38.1
40	.838	.6925	1.21	−.179	41.7
44	.800	.7660	1.04	−.191	42.3
48	.755	.8315	.91	−.203	43.1

TABLE XIII

Span	91.44 cm.
Chord	15.24 cm.
Area	.1393 m².

Test No. 205.
Airfoil N. A. C. A. M-6 (6″ by 36″) with 20 per cent c flaps.
Average tank pressure, 20.6 atomspheres.
Average dynamic pressure, 600 kg/m².
Average Reynolds Number, 3,990,000.

AILERONS SET 5° UP

α Degrees	C_L	C_D	L/D	C_M	C. P. per cent chord
0	0.002	0.0113	0.18	0.019	−925.0
2	.162	.0122	13.28	.013	+17.0
4	.308	.0144	21.39	.023	17.5
6	.455	.0196	23.21	.014	21.9
8	.596	.0273	21.83	.010	23.3
10	.744	.0380	19.58	.029	21.1
12	.883	.0520	16.98	.021	22.5
14	1.024	.0654	15.66	.011	23.9
16	1.144	.0840	13.62	.033	22.0
18	1.207	.1098	10.99	.026	22.7
20	1.192	.1521	7.84	−.020	26.7
22	1.141	.2162	5.28	−.020	26.8
24	1.033	.2758	3.74	−.095	32.0
28	.813	.3701	2.20	−.093	35.3
32	.770	.4359	1.77	−.106	37.0
36	.793	.5673	1.40	−.126	37.9
40	.796	.6508	1.22	−.141	38.7
44	.758	.7335	1.03	−.167	40.8
48	.722	.8037	.90	−.168	40.6

TABLE XIV

Span	91.44 cm.
Chord	15.24 cm.
Area	.1393 m².

Test No. 204.
Airfoil N. A. C. A. M-6 (6″ by 36″) with 20 per cent c flaps.
Average tank pressure, 20.6 atmospheres.
Average dynamic pressure, 610 kg/m².
Average Reynolds Number, 4,150,000.

AILERONS SET 10° UP

α Degrees	C_L	C_D	L/D	C_M	C. P. Per cent chord
0	−0.065	0.0150	−4.33	0.031	72.7
2	+.095	.0140	+6.79	.037	−14.2
4	.241	.0153	15.75	.041	+8.0
6	.383	.0191	20.05	.043	13.8
8	.526	.0254	20.71	.041	17.2
10	.676	.0353	19.15	.056	16.6
12	.810	.0465	17.42	.043	19.6
14	.953	.0608	15.67	.055	19.1
16	1.086	.0769	14.12	.031	22.0
18	1.171	.0955	12.26	.043	21.2
20	1.163	.1451	8.02	.019	23.4
22	1.121	.2015	5.56	−.028	27.5
24	1.008	.2600	3.88	−.059	30.8
28	.793	.3519	2.25	−.074	33.5
32	.742	.4286	1.73	−.074	33.7
36					
40	.739	.6196	1.19	−.134	38.9
44	.738	.7091	1.04	−.131	37.8
48	.710	.7988	.89	−.150	39.0

TABLE XV

Span	91. 44 cm.
Chord	15. 24 cm.
Area	. 1393 m².

Test No. 203.
Airfoil N. A. C. A. M–6 (6″ by 36″) with 20 per cent c flaps.
Average tank pressure 20.6 atmospheres.
Average dynamic pressure, 611 kg/m².
Average Reynolds Number, 4,110,000.

AILERONS SET 20° UP

α Degrees	C_L	C_D	L/D	C_M	C. P. per cent chord
0	−0. 137	0. 0258	−5. 31	0. 054	64. 4
2	+. 018	. 0227	+. 79	. 031	205. 6
4	. 169	. 0234	7. 22	. 048	−3. 6
6	. 311	. 0252	12. 34	. 062	+5. 1
8	. 440	. 0291	15. 12	. 060	11. 4
10	. 581	. 0358	16. 23	. 063	14. 2
12	. 707	. 0438	16. 14	. 068	15. 3
14	. 835	. 0550	15. 18	. 061	17. 6
16	. 958	. 0688	13. 92	. 057	19. 0
18	1. 055	. 0853	12. 37	. 059	19. 3
20	1. 067	. 1279	8. 34	. 030	22. 1
22	1. 030	. 1800	5. 72	−. 001	25. 1
24	. 949	. 2312	4. 10	−. 050	30. 3
28	. 736	. 3254	2. 26	−. 035	29. 4
32	. 672	. 3958	1. 70	−. 059	32. 5
36	. 654	. 4821	1. 36	−. 052	31. 4
40	-----	-----	-----	-----	-----
44	. 665	. 6595	1. 01	−. 093	34. 9
48	. 654	. 7471	. 88	−. 132	38. 3

TABLE XVI

VARIATION OF C_L', C_M, and C. P. ALONG SPAN $\alpha = +18°$

Per cent span	Ailerons—Neutral			Ailerons—20°*		
	C_L'	C_M	C. P. per cent chord	C_L'	C_M	C. P. per cent chord
0. 0	0. 00	0. 14	+∝	0. 0	−0. 130	−∝
. 5	. 62	. 14	22. 74	-----	-----	-----
1. 0	. 88	. 14	23. 41	-----	-----	-----
2. 5	1. 24	. 14	23. 87	1. 56	−. 130	33. 3
5. 0	1.29_5	. 14	23. 92	1.62_5	−. 129	32. 9
10. 0	1. 32	. 14	23. 94	1. 65	−. 128	32. 7
20. 0	1. 33	. 14	23. 95	1. 62	−. 114	32. 0
30. 0	1.33_5	. 14	23. 95	1.53_5	−. 052	28. 4
40. 0	1. 34	. 14	23. 96	1. 41	+. 002	24. 9
50. 0	1. 34	. 14	23. 96	1. 34	. 014	24. 0
60. 0	1. 34	. 14	23. 96	1. 30	. 016	23. 8
70. 0	1.33_5	. 14	23. 95	1. 23	. 027	22. 8
75. 0	-----	-----	-----	1.15_5	. 046	21. 0
80. 0	1. 33	. 14	23. 95	1. 06	. 067	18. 7
85. 0	-----	-----	-----	$.95_5$	. 088	15. 8
90. 0	1. 32	. 14	23. 94	. 86	. 104	12. 9
95. 0	1.29_5	. 14	23. 92	. 79	. 115	10. 4
97. 5	1. 24	. 14	23. 87	$.73_5$	. 118	9. 0
99. 0	. 88	. 14	23. 41	-----	-----	-----
99. 5	. 62	. 14	22. 74	-----	-----	-----
100. 0	. 00	. 14	+α	. 00	. 121	+α

*Down aileron, 0 to 25 per cent of span.
Up aileron, 75 to 100 per cent of span.

TABLE XVII

MOMENTS CAUSED BY AILERONS

$$C_{L'}=\frac{L'}{qbS} \quad C_N=\frac{N}{qbS}$$

α Degrees	Rolling moment coefficient $C_{L'}$			Yawing moment coefficient C_N		
	Ailerons 5°	Ailerons 10°	Ailerons 20°	Ailerons 5°	Ailerons 10°	Ailerons 20°
0	0.0315	0.0572	0.0885	0.00017	0.00006	0.00032
2	.0287	.0553	.0870	.00054	.00114	.00206
4	.0302	.0553	.0877	.00118	.00219	.00380
6	.0291	.0555	.0862	.00184	.00322	.00562
8	.0291	.0553	.0889	.00251	.00461	.00740
10	.0278	.0550	.0904	.00309	.00529	.00945
12	.0279	.0538	.0911	.00313	.00656	.01136
14	.0266	.0499	.0900	.00424	.00716	.01316
16	.0264	.0474	.0881	.00461	.00843	.01510
18	.0309	.0488	.0855	.00414	.00965	.01810
20	.0337	.0469	.0750	.00639	.01050	.02070
24	.0116	.0238	.0401	.00696	.01275	.02680
28	.0129	.0231	.0398	.00700	.01286	.02635
32	.0156	.0255	.0495	.01020	.01610	.03570
36	.0148		.0521	.01002		.03717
40	.0116	.0242		.01220	.02040	
44	.0109	.0182	.0356	.01062	.01885	.03945
48	.0062	.0122	.0259	.00802	.01560	.02945