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EFFECT OF A TRAILING-EDGE EXTENSION ON THE CHARACTERISTICS OF A PROPELLER SECTION

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

ADVANCE CONFIDENTIAL REPORT

EFFECT OF A TRAILING-EDGE EXTENSION ON THE CHARACTERISTICS OF A PROPELLER SECTION

By Theodore Theodorsen and George W. Stickle

SUMMARY

A convenient technical method is presented to evaluate changes in the airfoil characteristics resulting from an extension of the chord at the trailing edge of a propeller blade section. The method determines the change in the angle of zero lift, the ideal angle of attack, and the difference in these angles (upon which the design lift coefficient depends) as a function of the angle and length of the trailing-edge extension. The treatment is based directly upon the thin-airfoil theory and is thus concerned only with the mean camber line of the section. Examples and detailed computations are given to illustrate the application of the method. The method is applicable to all propeller sections and is short enough to permit use in practical design.

INTRODUCTION

A flat sheet of metal is sometimes attached to the trailing edge of a propeller blade in order to increase the propeller solidity for a given blade design. The addition of the flat sheet on the trailing edge of the propeller blade changes the characteristics of the blade section. The new characteristics are dependent upon the angle of the extension, the length of the extension, and the original airfoil section. The problem of determining the angle at which the sheet should be added and the effect of this angle on the angle of zero lift, the ideal angle of attack, and the design lift coefficient is the subject of this paper.

The ideal angle of attack is defined as the angle at which the front stagnation point is at the leading edge of the airfoil. The design lift coefficient is

[redacted]

defined as the lift coefficient of the airfoil when set at the ideal angle of attack. If the design lift coefficient, so defined, corresponds to the operating condition of the propeller section, the airfoil section has a camber that gives it the highest critical speed (best pressure distribution) obtainable for the operating condition with the camber and thickness distribution which define the airfoil.

A relative change in the angles of zero lift of the airfoil sections along the propeller radius in effect changes the pitch distribution of the propeller. The angle of the trailing-edge extension thus permits some selection of the pitch distribution of the propeller. This report shows how this effect may be evaluated.

The method of this report is based on the concept of the examination of the mean camber line from thin-airfoil theory of reference 1. The method has been checked for accuracy with the more complete but more difficult methods of references 2 and 3 and found to be in good agreement for thin sections as used on propeller blades.

Experimental data on airfoil section lift coefficients as a function of angle of attack are generally used in analyzing propeller operation. Since experimental data for airfoil sections with extended flaps are not available, theoretical calculations must be used in analyzing propeller operation with trailing-edge extension flaps. Experimental and theoretical values of lift coefficient as a function of angle of attack rarely are in perfect agreement. The discrepancy increases with airfoil section camber and thickness and makes it difficult to compare results when theoretical and experimental values are used together. For this reason the difference in the airfoil characteristics between the original and the extended airfoil sections are used in the application of the results of this report to propeller analysis.

METHOD OF ANALYSIS

Calculation of the angle of zero lift, the ideal angle of attack, and the design lift coefficient is based on the examination of the mean camber line of the

airfoil section. The method is adapted from reference 1, which gives a discussion of its theoretical background and validity. The present paper applies the solution to the particular problem of extended flaps on airfoil sections and presents the method in a form that makes it easily applicable to this specific problem. The steps in the procedure are as follows:

(1) Obtain the ordinate y at an abscissa x of the camber line of the airfoil from a line joining the ends of the camber line.

(2) Calculate the function P from

$$P = \frac{y}{\sqrt{x(1 - x)}} \tag{1}$$

(3) Calculate ϵ for the nose of the airfoil:

$$\epsilon_N = \frac{1}{\pi}\int_{0.0125}^{1} \frac{P}{x}\,dx - \frac{1}{\pi}\int_{0}^{0.0125} \frac{P}{x}\,dx \tag{2}$$

The first term of equation (2) is integrated graphically. The second term, which may be denoted as $\Delta\epsilon_N$ or

$$\Delta\epsilon_N = \frac{1}{\pi}\int_{0}^{0.0125} \frac{P}{x}\,dx$$

$$= \frac{1}{\pi}\int_{0}^{0.0125} \frac{y}{x\sqrt{x(1 - x)}}\,dx \tag{3}$$

may be evaluated, however, if the term $1 - x$ is taken equal to unity. This substitution causes an error of only 0.6 of 1 percent in the $\Delta\epsilon_N$ term or approximately 0.2 of 1 percent in ϵ_N. Then,

$$\Delta\epsilon_N = \frac{1}{\pi}\int_{0}^{0.0125} \frac{y}{x^{3/2}}\,dx \tag{4}$$

[redacted]

An approximate equation for the part of the camber line from $x = 0$ to $x = 0.0125$ may be written in the form of

$$y = cx - bx^2$$

where c is the slope of the camber line at $x = 0$, and b is a constant that will make the ordinate at $x = 0.0125$ equal to the y ordinate of the camber line. (See fig. 1.)

It may be shown that the integral of equation (4) is equal to

$$\Delta\epsilon_N = \frac{2}{\pi\sqrt{x_1}}\left[y_1 + \frac{2}{3}\left(cx_1 - y_1\right)\right] \tag{5}$$

where x_1 and y_1 are the values of x and y at the common limit of the graphical and analytical integrations. The value of x at which the graphical integration is stopped and the analytical begun may be anywhere in the region of $x = 0.0125$. When extension flaps are added, the most convenient value is less than $x = 0.0125$.

(4) Calculate ϵ for the tail of the section in the same manner as ϵ_N is calculated by replacing x by $1 - x$:

$$\epsilon_T = \frac{1}{\pi}\int_0^{0.9875} \frac{P}{1 - x}\,dx + \frac{1}{\pi}\int_{0.9875}^{1} \frac{P}{1 - x}\,dx \tag{6}$$

(5) The angle of zero lift in degrees is given by

$$\alpha_{l_o} = -57.3\epsilon_T \tag{7}$$

(6) The ideal angle of attack in degrees is given by

$$\alpha_I = -28.6\left(\epsilon_T + \epsilon_N\right) \tag{3}$$

(7) The difference between α_{l_o} and α_I is the angle upon which the design lift coefficient depends. If the slope of the lift curve a is known for the section, the design lift coefficient may be calculated by

$$c_{L_I} = a\left(\alpha_I - \alpha_{l_o}\right) \tag{9}$$

As was explained in the Introduction, these computations are most accurate for values to be used in comparison of the original and the extended airfoil sections. If used in this manner, the computations give an accurate picture of the change in the operating conditions of the propeller sections when any angle or length of extension is used.

The method for two representative examples is given in the appendix and computations for four examples are given in tables I to IV. The examples are for an NACA 16-series airfoil section and a Clark Y airfoil section. The NACA 16-series section has a design lift coefficient of 0.5 and, since only the mean camber line is employed in the calculations, is designated 16-5XX, which applies to an airfoil of zero or any other thickness.

RESULTS AND DISCUSSION

Computations have been made for an NACA 16-5XX airfoil section with an extension of 20 percent of the chord. The results are given in figure 2 as a function of the angle of extension measured from a straight line joining the ends of the mean camber line of the original airfoil. The angle of zero lift, the ideal angle of attack, and the difference between the two angles of the original airfoil are shown in figure 2.[1] The angle of extension would have to be 9.7° to make the angle of zero lift equal to that of the original airfoil, 13.2° to keep the ideal angle of attack unchanged, and 8° to make the difference between the two angles the same, which would mean equal design lift coefficients for the

---

[1]It is inherent in the method that these calculated angles are measured from a straight line joining the extremities of the mean camber line of the extended airfoil section. If it is desired to refer these angles to the camber line of the original airfoil section, the following formula gives the angular difference α_1 between the two reference lines:

$$\tan^{-1}\alpha_1 = \frac{\left(\frac{\text{Extension length}}{\text{Chord}}\right) \sin\,(\text{Angle of extension})}{1 + \left(\frac{\text{Extension length}}{\text{Chord}}\right) \cos\,(\text{Angle of extension})}$$

original and extended airfoils. If the angle of extension is made 13.2° so that the ideal angle of attack remains unchanged, breakaway of the flow at low lift coefficients might be encountered.

Figure 3 gives the results for a 20-percent extension on a Clark Y airfoil of 1.83-percent camber, which corresponds to a standard Clark Y section of 6-percent thickness. The angle of the extension that gives characteristics equivalent to the original airfoil is seen to be greatly reduced over that of figure 2 because of the small amount of camber for the original airfoil.

The results are shown in figure 4 for a 20-percent extension on a Clark Y airfoil of 5.49-percent camber, which corresponds to a standard Clark Y section of approximately 18-percent thickness or to a double cambered Clark Y section of thinner section with a design lift coefficient between 0.6 and 0.7. These results were calculated to investigate the effect of camber alone.

The results for a 40-percent extension on a Clark Y airfoil of 5.49-percent camber are presented in figure 5. The angle of the extension to maintain the same design lift coefficient as the basic airfoil is approximately the same for both the 20- and the 40-percent extensions.

Figure 6 shows a comparison of the α_{l_0} and the $\alpha_I - \alpha_{l_0}$ curves for the four conditions investigated. It may be observed that the slopes of the α_{l_0} curves are all approximately equal at a value of 0.35° per degree of the angle of the extension. The reason for this condition can be seen from examination of equations (6) and (7) along with figures 7 and 8. The value of α_{l_0} is defined by ϵ_T, and from figure 8 it can be seen that the trailing edge of the airfoil is the dominating factor in determining ϵ_T. The $\alpha_I - \alpha_{l_0}$ curves are seen to vary more in slope than the α_{l_0} curves. Camber of the original airfoil section merely shifts the value of α_{l_0} for a given extension angle.

The length of the extension has the same effect as camber. The longer the extension the less camber the extended airfoil has for a given angle.

CONCLUSIONS

A convenient technical method has been presented to evaluate changes in the airfoil characteristics resulting from an extension of the chord at the trailing edge of a propeller blade section. The method determines the change in the angle of zero lift, the ideal angle of attack, and the difference in these angles (upon which the design lift coefficient depends) as a function of the angle and length of the trailing edge extension and permits the adjustment of the angle of the extension of the chord to comply with any requirements regarding the angle of zero lift or the design lift coefficient.

It was found that for the cases considered the characteristics obtained by this short method were actually in perfect agreement with data calculated by the exact method of the arbitrary airfoil theory. This agreement is, of course, not necessarily true for thick sections or extreme cases of curvature near the extremities of the chord. On the other hand, for normal cases the method is sufficiently accurate for all technical purposes.

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APPENDIX

COMPUTATIONS IN DETAIL

Tables I and II contain the detailed computations for an NACA 16-5XX airfoil section for the original airfoil and for the airfoil extended 20 percent. Tables III and IV contain the detailed computations for a standard 6-percent-thick Clark Y airfoil with and without an extended trailing edge of 20 percent. Table V gives the camber ordinates for NACA 16-series airfoil sections and table VI, the camber ordinates for Clark Y sections.

Each step in obtaining the results for the NACA 16-5XX section without trailing-edge extension is explained in detail for table I. The steps that are different because the airfoil is extended are explained for table II.

Example I - Characteristics of NACA 16-5XX Airfoil

The numbers in parentheses refer to column numbers in table I.

(1) By use of table V, select values of x to be used.

(2) Subtract values of x from unity.

(3) Multiply (1) by (2).

(4) Take square root of (3).

(5) Obtain ordinate of mean camber line from table V by multiplying values in table V by the design lift coefficient of 0.5 and convert to fractions instead of percent.

(6) Divide (5) by (4).

(7) Divide (6) by (1).

8. Plot (7) against (1) between the values of $x = 0.0125$ and $x = 0.95$ and extend the curve to $x = 1.0$ as shown in figure 7.

9. Integrate with a planimeter between the limits of $x = 0.0125$ and $x = 1.0$.

10. Calculate $\Delta\epsilon_N$ from equation (5):

$$\Delta\epsilon_N = \frac{2}{\pi\sqrt{x_1}}\left[y_1 + \frac{2}{3}(cx_1 - y_1)\right]$$

when

$$x_1 = 0.0125$$
$$y_1 = 0.00268$$
$$c = 0.62234 \times 0.5 = 0.31117$$

then

$$\Delta\epsilon_N = \frac{2}{0.1118\pi}(0.00268 + 0.00081)$$
$$= 0.0199$$

11. Calculate ϵ_N from equation (2) and the results of steps 9 and 10:

$$\epsilon_N = \frac{1}{\pi}\int_{0.0125}^{1} \frac{P}{x}\,dx - \Delta\epsilon_N$$

or

$$\epsilon_N = -\frac{0.1852}{\pi} - 0.0199$$
$$= -0.0789$$

12. Since the NACA 16-5XX airfoil has a symmetrical camber line,

$$\epsilon_N = -\epsilon_T$$

13. From equation (7) calculate angle of zero lift α_{l_0}:

$$\alpha_{l_0} = -57.3\epsilon_T$$
$$= -57.3(0.0789)$$
$$= -4.52^\circ$$

14. From equation (8) calculate the ideal angle of attack α_I:

$$\alpha_I = -28.6(\epsilon_T + \epsilon_N)$$
$$= 0^\circ$$

15. Calculate the difference $\alpha_I - \alpha_{l_o}$:

$$\alpha_I - \alpha_{l_o} = 0 + 4.52$$
$$= 4.52^\circ$$

Example II - Characteristics of NACA 16-5XX Airfoil

with 20-Percent Extension

Calculations are presented for the characteristics of the NACA 16-5XX airfoil with a 20-percent extension set at an angle of 7.38° to the line joining the ends of the mean camber line. This angle is equal to the angle formed by a straight line drawn through the y ordinate at 90 percent of the chord and the end of the mean camber line.

The numbers in parentheses refer to column numbers in table II.

(1) Select values of x_2 (abcissa of airfoil with extended additions).

(2) Obtain values of y_2 from table I (ordinate of airfoil with extended additions).

(3) Compute Δy from

$$\Delta y = -y_{2_T} \frac{x_2}{1.2}$$

where y_{2_T} is the ordinate y_2 at $x_2 = 1.2$.

(4) Add (2) and (3).

(5) Convert (4) to unit chord by dividing by 1.2.

(6) Convert (1) to unit chord by dividing by 1.2.

7. Add a station at $x = 0.9875$. The y ordinate is obtained by proportion since the extension is a straight line.

8. When the coordinates of the mean camber line are obtained from the base of a straight line joining the ends of the mean camber line of the extended airfoil, the example proceeds in the same manner as example I until ϵ_N is obtained.

REFERENCES

1. Theodorsen, Theodore: On the Theory of Wing Sections with Particular Reference to the Lift Distribution. NACA Rep. No. 383, 1931.

2. Theodorsen, Theodore: Theory of Wing Sections of Arbitrary Shape. NACA Rep. No. 411, 1931.

3. Theodorsen, T., and Garrick, I. E.: General Potential Theory of Arbitrary Wing Sections. NACA Rep. No. 452, 1933.

TABLE I.- CALCULATIONS FOR NACA 16-5XX AIRFOIL

(1)	(2)	(3)	(4)	(5)	(6)	(7)
x	$1 - x$ 1 - (1)	$x(1-x)$ (1) × (2)	$\sqrt{x(1-x)}$ $\sqrt{(3)}$	y	$P = \frac{y}{\sqrt{x(1-x)}}$ (5)/(4)	P/x (6)/(1)
0	1.00	0	0	0		
.0125	.9875	.0123	.1109	.0027	0.0242	1.936
.05	.95	.0475	.2179	.0079	.0363	.726
.10	.90	.0900	.3000	.0129	.0431	.431
.20	.80	.1600	.4000	.0199	.0498	.249
.30	.70	.2100	.4583	.0243	.0530	.177
.40	.60	.2400	.4899	.0268	.0547	.137
.50	.50	.2500	.5000	.0276	.0552	.110
.60	.40	.2400	.4899	.0268	.0547	.091
.70	.30	.2100	.4583	.0243	.0530	.076
.80	.20	.1600	.4000	.0199	.0498	.062
.90	.10	.0900	.3000	.0129	.0431	.048
.95	.05	.0475	.2179	.0079	.0363	.038
1.00	0	0	0	0		

$\epsilon_N = -0.0789$ $\quad \alpha_{l_o} = -4.52^\circ$

$\epsilon_T = 0.0789$ $\quad \alpha_I = 0^\circ$ $\quad \alpha_I - \alpha_{l_o} = 4.52^\circ$

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TABLE II.- CALCULATIONS FOR NACA 16-5XX AIRFOIL WITH 20-PERCENT TRAILING-EDGE EXTENSION

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)
x_2	y_2	$\Delta y = -y_{2_T}\left(\frac{x_2}{1.2}\right) = \frac{0.0259}{1.2} x_2$	(2) + (3)	y $\frac{(4)}{1.2}$	x $\frac{(1)}{1.2}$	$1 - x$ $1 - (6)$	$x(1 - x)$ $(6) \times (7)$	$\sqrt{x(1 - x)}$ $\sqrt{(8)}$	$P = \frac{y}{\sqrt{x(1 - x)}}$ (5)/(9)	$\frac{P}{x}$ (10)/(6)	$\frac{P}{1 - x}$ (10)/(7)
0	0	0	0	0	0	1.0000	0	0			
.0125	.0027	.0003	.0030	.0025	.0104	.9896	.0103	.1015	0.0246	2.365	0.025
.05	.0079	.0011	.0090	.0075	.0417	.9583	.0400	.2000	.0375	.899	.039
.10	.0129	.0022	.0151	.0126	.0833	.9167	.0764	.2764	.0456	.547	.050
.20	.0199	.0043	.0242	.0202	.1667	.8333	.1389	.3727	.0542	.325	.065
.30	.0243	.0065	.0308	.0257	.2500	.7500	.1875	.4330	.0594	.238	.079
.40	.0268	.0086	.0354	.0295	.3333	.6667	.2222	.4714	.0626	.188	.094
.50	.0276	.0108	.0384	.0320	.4167	.5833	.2431	.4931	.0649	.156	.111
.60	.0268	.0129	.0397	.0331	.5000	.5000	.2500	.5000	.0662	.132	.132
.70	.0243	.0151	.0394	.0328	.5833	.4167	.2431	.4931	.0665	.114	.160
.80	.0199	.0172	.0371	.0309	.6667	.3333	.2222	.4714	.0655	.098	.197
.90	.0129	.0194	.0323	.0269	.7500	.2500	.1875	.4330	.0621	.083	.248
.95	.0079	.0205	.0284	.0237	.7917	.2083	.1649	.4061	.0584	.074	.280
1.00	0	.0216	.0216	.0180	.8333	.1667	.1389	.3727	.0483	.058	.290
1.10	-.0129	.0237	.0108	.0090	.9167	.0833	.0764	.2764	.0326	.036	.391
				.0014	.9875	.0125	.0123	.1109	.0126	.013	1.008
1.20	-.0259	.0259	0	0	1.0000	0	0	0			

$\epsilon_N = -0.0871$ $\quad \alpha_{l_0} = -3.69^\circ$ $\quad \alpha_I - \alpha_{l_0} = 4.34^\circ$

$\epsilon_T = 0.0644$ $\quad \alpha_I = 0.65^\circ$ $\quad \alpha_{ext} = 7.38^\circ$

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TABLE III.- CALCULATIONS FOR CLARK Y AIRFOIL WITH 1.83-PERCENT CAMBER CORRESPONDING TO STANDARD 6-PERCENT THICK CLARK Y AIRFOIL

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)
x	$1 - x$ $1 - (1)$	$x(1 - x)$ $(1) \times (2)$	$\sqrt{x(1 - x)}$ $\sqrt{(3)}$	y	$P = \frac{y}{\sqrt{x(1 - x)}}$ $(5)/(4)$	P/x $(6)/(1)$	$\frac{P}{1 - x}$ $(6)/(2)$
0	1.00	0	0	0			
.0125	.9875	.0123	.1109	.0015	0.0135	1.080	0.014
.05	.95	.0475	.2179	.0056	.0257	.514	.027
.10	.90	.0900	.3000	.0096	.0320	.320	.036
.20	.80	.1600	.4000	.0148	.0370	.185	.046
.30	.70	.2100	.4583	.0174	.0380	.127	.054
.40	.60	.2400	.4899	.0183	.0374	.094	.062
.50	.50	.2500	.5000	.0178	.0356	.071	.071
.60	.40	.2400	.4899	.0161	.0329	.055	.082
.70	.30	.2100	.4583	.0132	.0288	.041	.096
.80	.20	.1600	.4000	.0096	.0240	.030	.120
.90	.10	.0900	.3000	.0051	.0170	.019	.170
.95	.05	.0475	.2179	.0026	.0119	.013	.238
.9875	.0125	.0123	.1109	.0007	.0063	.006	.504
1.00	0	0	0	0			

$\epsilon_N = -0.0481$ $\alpha_{l_0} = -1.82^\circ$
$\epsilon_T = 0.0318$ $\alpha_I = 0.47^\circ$ $\alpha_I - \alpha_{l_0} = 2.39^\circ$

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TABLE IV.- CALCULATIONS FOR CLARK Y AIRFOIL WITH 1.83-PERCENT CAMBER

AND A 20-PERCENT TRAILING-EDGE EXTENSION

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)
x_2	y_2	$\Delta y = -y_{2_T}\left(\frac{x_2}{1.2}\right) = \frac{0.0102}{1.2} x_2$	(2) + (3)	$\frac{y}{1.2}$ (4)	$\frac{x}{1.2}$ (1)	$1 - x$ $1 - (6)$	$x(1 - x)$ (6) × (7)	$\sqrt{x(1 - x)}$ $\sqrt{(8)}$	$P = \frac{y}{\sqrt{x(1 - x)}}$ (5)/(9)	$\frac{P}{x}$ (10)/(6)	$\frac{P}{1 - x}$ (10)/(7)
0	0	0	0	0	0	1.0000	0	0			
.0125	.0015	.0001	.0016	.0013	.0104	.9896	.0103	.1015	0.0128	1.231	0.013
.05	.0056	.0004	.0060	.0050	.0417	.9583	.0400	.2000	.0250	.600	.026
.10	.0096	.0009	.0105	.0087	.0833	.9167	.0764	.2764	.0315	.378	.034
.20	.0148	.0017	.0165	.0137	.1667	.8333	.1389	.3727	.0368	.221	.044
.30	.0174	.0026	.0200	.0167	.2500	.7500	.1875	.4330	.0386	.154	.051
.40	.0183	.0034	.0217	.0181	.3333	.6667	.2222	.4714	.0384	.115	.058
.50	.0178	.0043	.0221	.0184	.4167	.5833	.2431	.4931	.0373	.090	.064
.60	.0161	.0051	.0212	.0177	.5000	.5000	.2500	.5000	.0354	.071	.071
.70	.0132	.0060	.0192	.0160	.5833	.4167	.2431	.4931	.0324	.056	.078
.80	.0096	.0068	.0164	.0137	.6667	.3333	.2222	.4714	.0291	.044	.087
.90	.0051	.0077	.0128	.0107	.7500	.2500	.1875	.4330	.0247	.033	.099
.95	.0026	.0081	.0107	.0089	.7917	.2083	.1649	.4061	.0219	.028	.105
1.00	0	.0085	.0085	.0071	.8333	.1667	.1389	.3727	.0191	.023	.115
1.10	-.0051	.0094	.0043	.0036	.9167	.0833	.0764	.2764	.0130	.014	.156
				.0004	.9875	.0125	.0123	.1109	.0036	.004	.288
1.20	-.0102	.0102	0	0	1.0000	0	0	0			

$\epsilon_N = -0.0490$ $\alpha_{l_o} = -1.59^\circ$ $\alpha_I - \alpha_{l_o} = 2.20^\circ$

$\epsilon_T = 0.0278$ $\alpha_I = 0.61^\circ$ $\alpha_{ext} = 2.9^\circ$

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TABLE V.- CAMBER-LINE ORDINATES FOR

THE 16-SERIES AIRFOIL SECTIONS

[All values measured in percent chord from chord line; lift coefficient, 1.0]

Station	Ordinate	Slope
0	0	0.62234
1.25	.535	.34771
2.5	.930	.29155
5	1.580	.23432
7.5	2.120	.19993
10	2.587	.17486
15	3.364	.13804
20	3.982	.11032
25	4.475	.08743
30	4.861	.06743
40	5.356	.03227
50	5.516	0
60	5.356	.03227
70	4.861	.06743
80	3.982	.11032
90	2.587	.17486
95	1.580	.23432
100	0	.62234

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TABLE VI.- MEAN-CAMBER AND THICKNESS ORDINATES FOR FAMILY OF AIRFOILS BASED UPON CLARK Y SECTION

[All values are given in percent of wing chord]

Station	Mean-camber ordinate	Semithickness
0	0	0
1.25	.0822	.1504
2.5	.1597	.2150
5	.3040	.2979
7.5	.4189	.3513
10	.5185	.3923
15	.6812	.4504
20	.8062	.4842
30	.9457	.5000
40	1.0000	.4872
50	.9732	.4496
60	.8778	.3910
70	.7223	.3141
80	.5207	.2231
90	.2785	.1197
95	.1435	.0637
100	0	0
L.E. radius: $0.009t^2$. T.E. radius: 0.005t.		

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$$\Delta\epsilon_T = \frac{2}{\pi}\frac{Y}{\sqrt{1 - x_1}}$$

$$\Delta\epsilon_N = \frac{2}{\pi}\frac{Y}{\sqrt{x_1}}$$

c, slope through origin

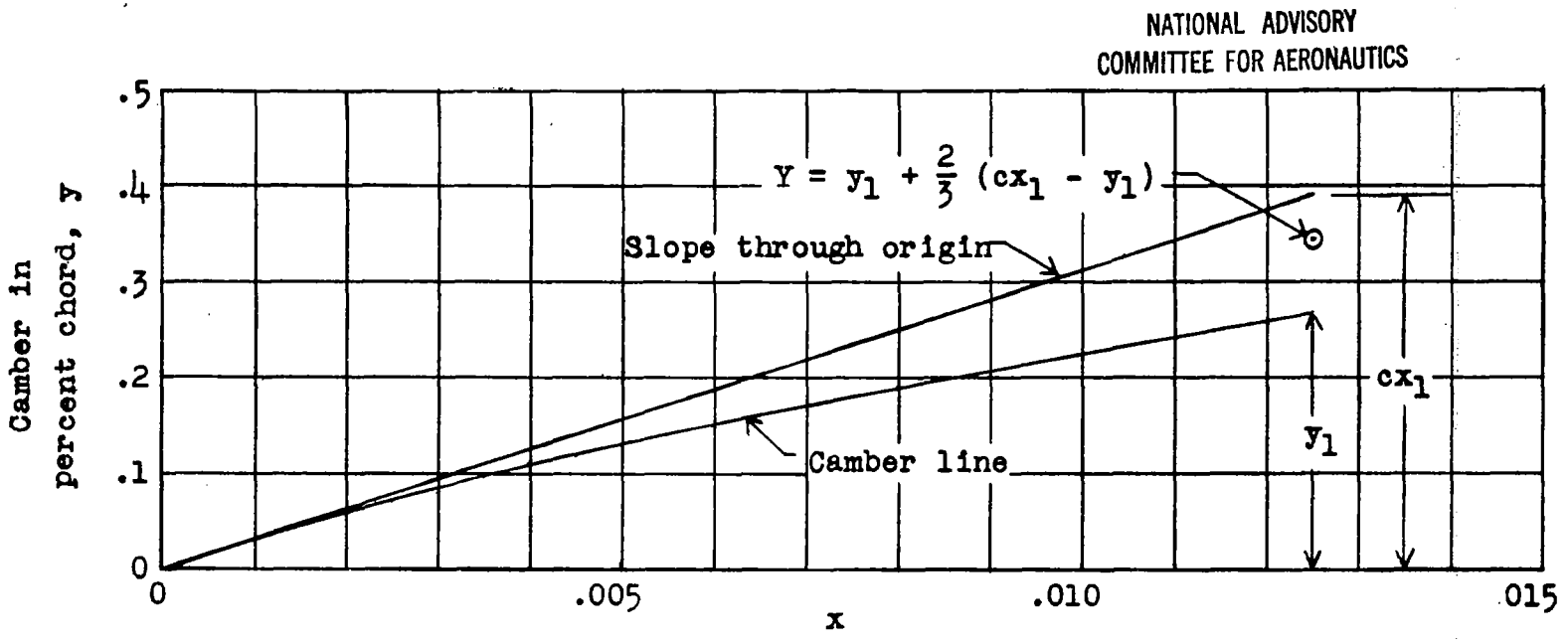

Figure 1.- Calculation of $\Delta\epsilon_N$ for $x = 0$ to $x = 0.0125$.

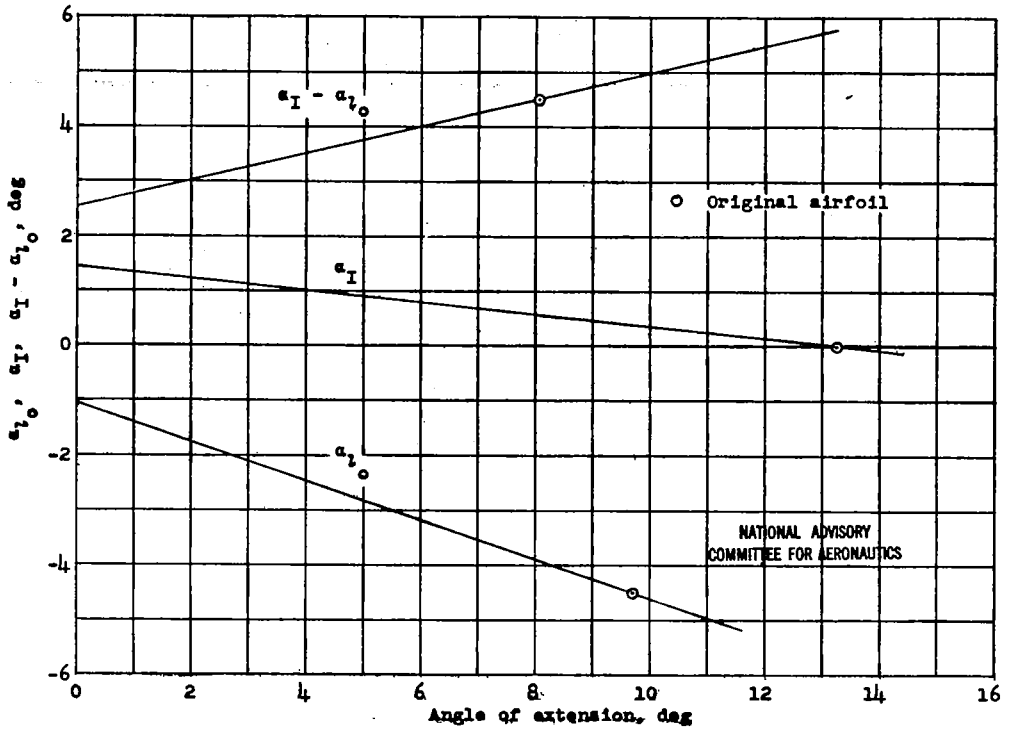

Figure 2.- NACA 16-5XX airfoil with 20-percent extension.

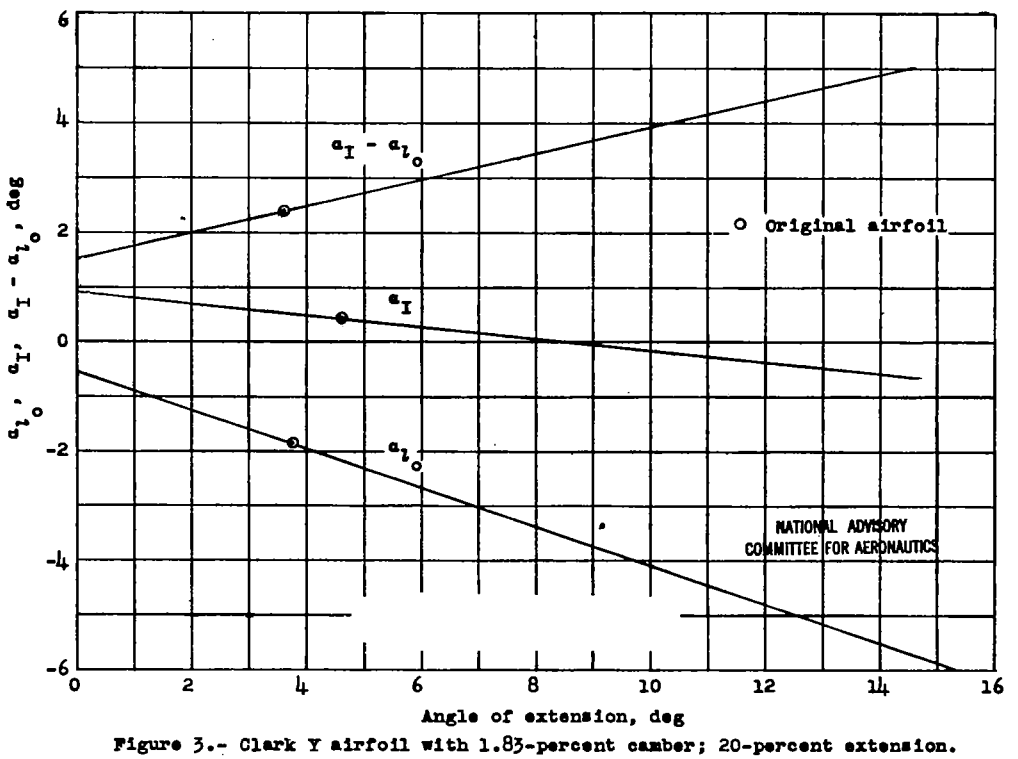

Figure 3.- Clark Y airfoil with 1.83-percent camber; 20-percent extension.

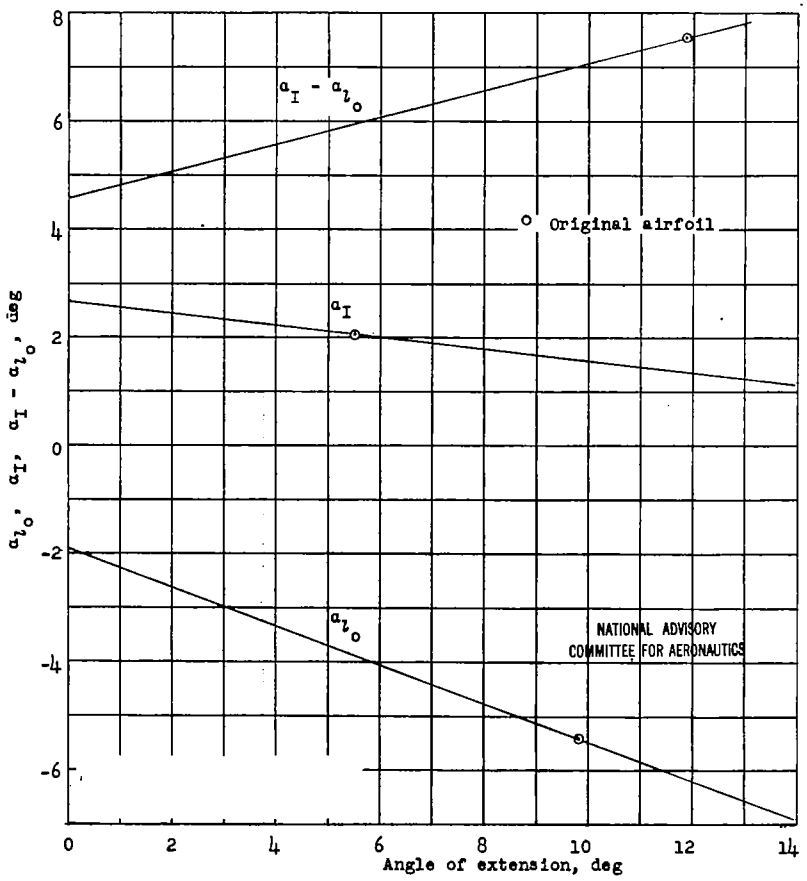

Figure 4.- Clark Y airfoil with 5.49-percent camber; 20-percent extension.

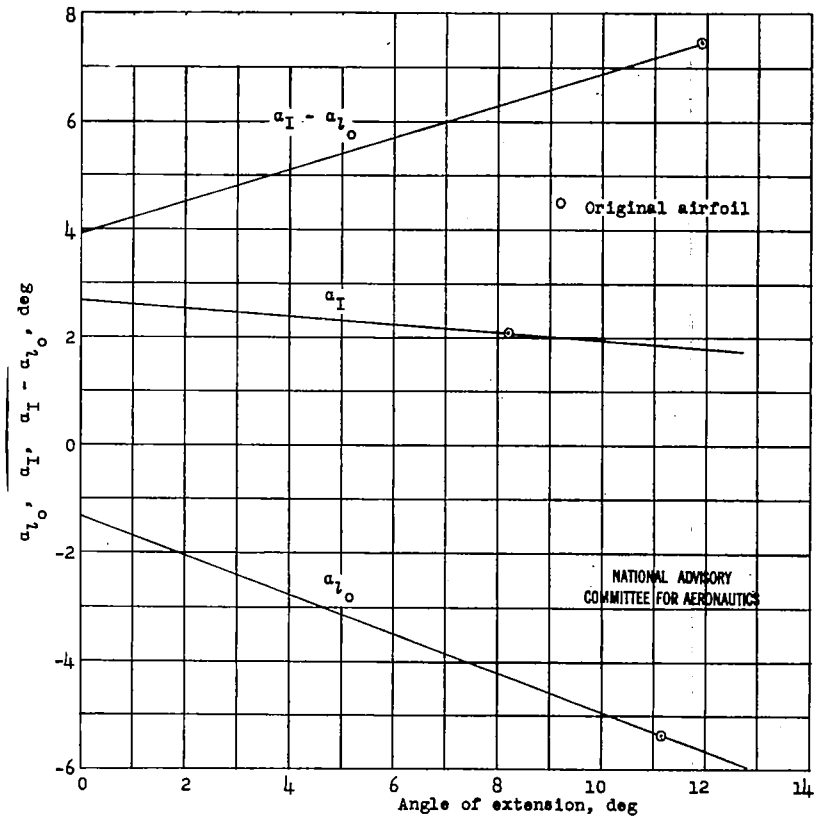

Figure 5.- Clark Y airfoil with 5.49-percent camber; 40-percent extension.

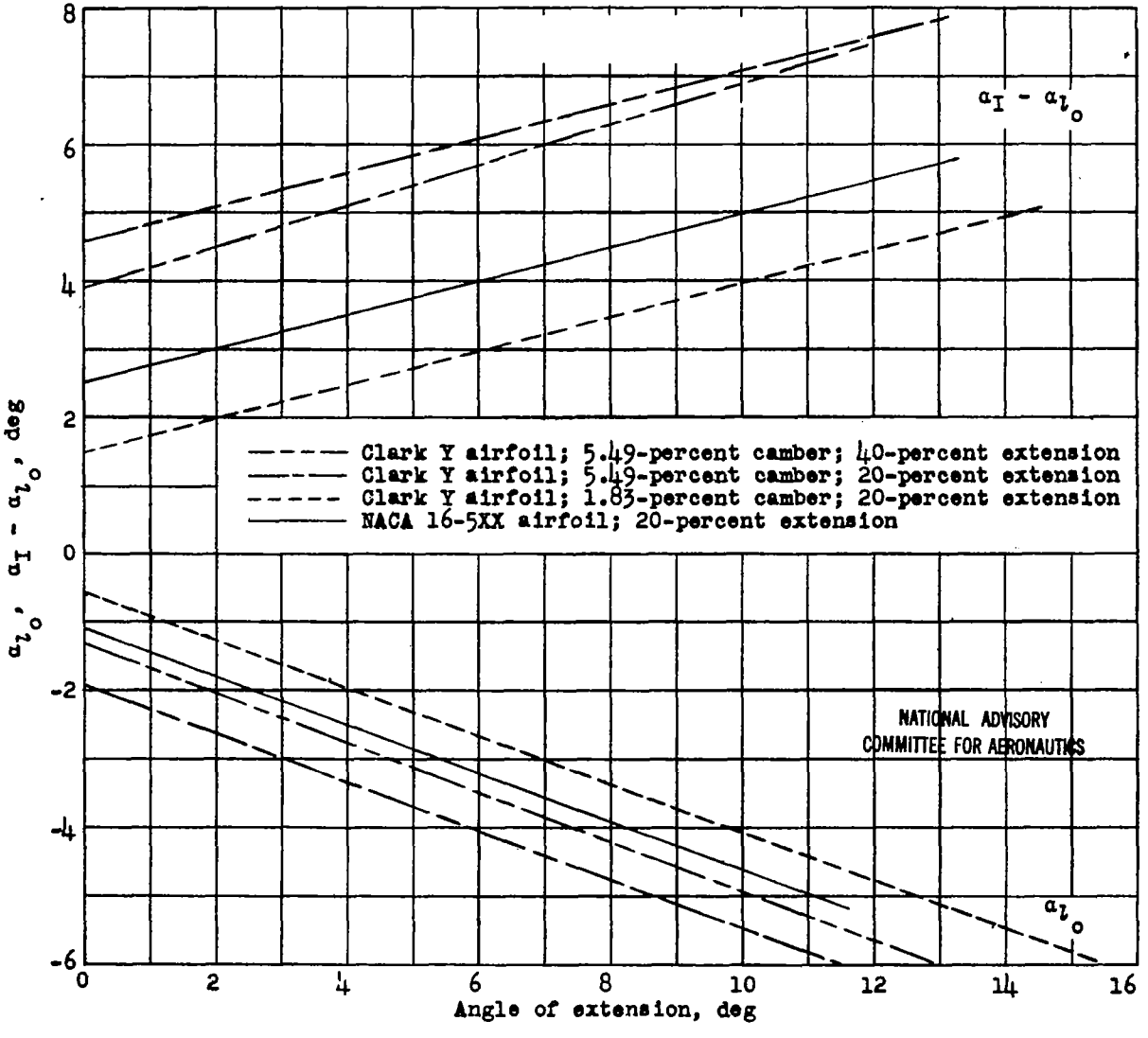

Figure 6.- Comparison chart.

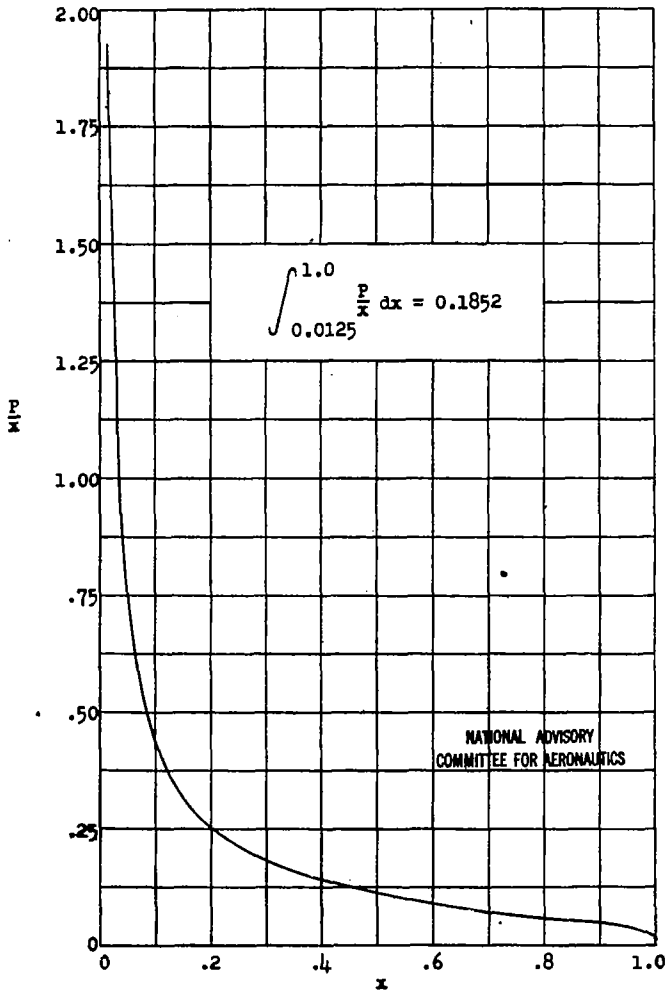

Figure 7.- Graphical integration for NACA 16-5XX airfoil.

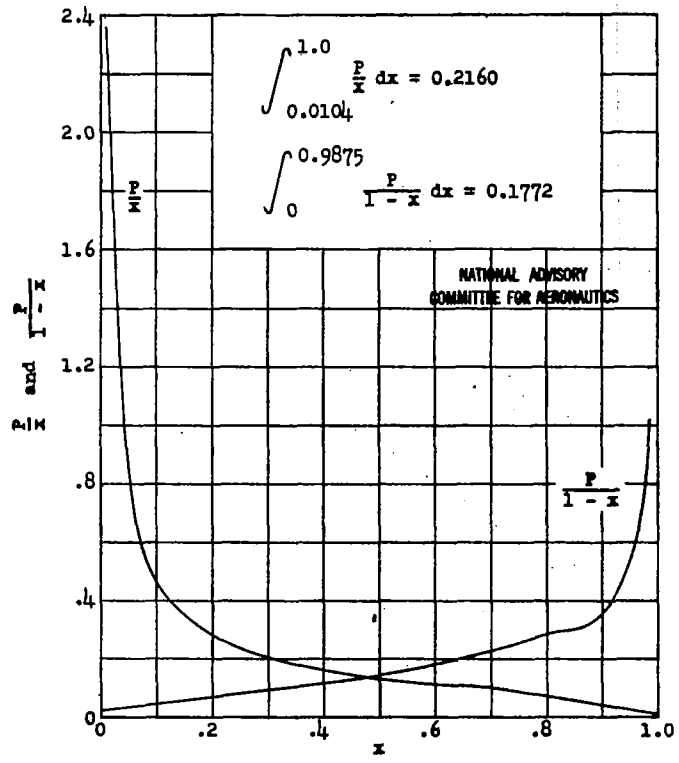

Figure 8.- Graphical integration for NACA 16-5XX airfoil with 20-percent extension.

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