ANC-1 (2)

CHORDWISE AIR-LOAD DISTRIBUTION

WAR DEPARTMENT
ARMY AIR FORCES

NAVY DEPARTMENT
BUREAU OF AERONAUTICS

DEPARTMENT OF COMMERCE
CIVIL AERONAUTICS ADMINISTRATION

Issued by the
ARMY-NAVY-CIVIL COMMITTEE
ON AIRCRAFT DESIGN CRITERIA

Under the Supervision of the
AERONAUTICAL BOARD

OCTOBER 28, 1942

ANC-1(2)
Admt. -1*
3 Jan. 1944

(AMENDED TITLE PAGE)

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ANC-1(2), Amendment-1, as approved 3 January 1944, consists of changes in the texts which require reprinting of the following sheets:

1. Title Page
2. Pages 3-12 and 3-13
3. Pages A-2 and A-3
4. Pages A-4 and A-5
5. Pages E-4 and E-5

These sheets have been prepared with three holes for loose leaf folder use. It will be necessary to punch similar holes in the original bulletin to facilitate this and future amendments. These sheets shall be inserted in lieu of the same pages of ANC-1 (2), October 28, 1942.

NOTICE

The reader is hereby notified that this bulletin is subject to revision and amendment when and where such revision or amendment is necessary to effect agreement with the latest approved information on aircraft design criteria. When using this bulletin, the reader should therefore make certain that the latest revisions and amendments have been incorporated.

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(1) Spanwise Air-Load Distribution - Bulletin ANC-1(1), 1938.

(2) Theodorsen: On the Theory of Wing Sections with Particular Reference to the Lift Distribution. T. R. No. 383, NACA, 1931.

(3) Theodorsen: Theory of Wing Sections of Arbitrary Shape. T. R. No. 411, NACA, 1931.

(4) Jacobs and Rhode: Airfoil-Section Characteristics as Applied to the Prediction of Air-Forces and Their Distribution on Wings. T. R. No. 631, NACA, 1938.

(5) Allen: Calculation of the Chordwise Load Distribution over Airfoil Sections with Plain, Split, or Serially-Hinged Flaps. T. R. No. 634, NACA, 1938.

(6) Glauert: A Theory of Thin Airfoils, R. and M. No. 910, A.R.C., 1924.

(7) Glauert: Theoretical Relationships for an Airfoil with a Hinged Flap, R. and M. No. 1095, A.R.C., 1927.

(8) Glauert: Airfoil and Airscrew Theory; Cambridge University Press, 1930.

(9) Allen: A Simplified Method for Calculation of Airfoil Pressure Distribution. T.N. No. 708, NACA, 1938.

(10) Stack, Lindsey, and Littell: The Compressibility Burble and the Effect of Compressibility on the Pressure and Forces Acting on an Airfoil. T. R. No. 646, NACA, 1938.

(11) Von Karman: Compressibility Effects in Aerodynamics. Journal of the Aeronautical Sciences, July 1941.

SYMBOLS

c Airfoil chord

c_l Section lift coefficient, l/qc, perpendicular to undisturbed air stream.

c_d Section drag coefficient.

c_m Section pitching moment coefficient about quarterchord point.

c_{m1} Section quarter-chord pitching moment coefficient for an airfoil with plain or split flap, but with flap in neutral position. This is usually equal to c_m.

c'_m Fictitious value of c_m corresponding to c'_n distribution.

c_{m2} Section quarter-chord pitching moment coefficient for airfoil with plain or split flap displaced from neutral.

c_m Change in section quarter-chord pitching moment coefficient $(c_{m2} - c_{m1})$ resulting from displacement of flap from neutral.

$\Delta c'_m$ Fictitious change in section quarter-chord pitching moment coefficient which results with the actual normal force distribution corresponding to c_{m2} with deflected flaps plotted normal to flap-neutral chord.

$c_{m_{ac}}$ Section pitching moment coefficient about the aerodynamic center.

c_n Section normal-force coefficient, $c_l \cos\alpha + c_d \sin\alpha$

c'_n Fictitious value of normal-force coefficient for the unflapped section resulting from plotting, normal to the unflapped section chord, the actual normal forces obtained in the flapped attitude while developing an actual normal-force coefficient c_n.

c_{n_a} Section additional normal-force coefficient.

c_{n_b} Section normal-force coefficient $(c_{n_{bm}} + c_{n_{bc}})$, corresponding to basic pressure distribution P_b.

$c_{n_{bm}}$ Component of basic normal-force coefficient, $(-c_{m_{ac}})\ c_{n_{bm_1}}$, corresponding to P_{bm} distribution.

$c_{n_{bm_1}}$ Component of basic normal-force coefficient which is the integral over the chord of P_{bm_1} distribution.

$c_{n_{bc}}$ Component of basic normal-force coefficient, $(z_c)\ c_{n_{bc_1}}$, corresponding to P_{bc} distribution.

$c_{n_{bc_1}}$ Component of basic normal-force coefficient corresponding to P_{bc_1} distribution.

$c_{n_{b\delta}}$ Incremental normal-force coefficient corresponding to $P_{b\delta}$ distribution.

c_{n_f} Flap normal-force coefficient corresponding to flap angle, δ, and normal-force coefficient, c_n.

E Ratio of flap chord to airfoil chord.

G Moment arm of basic normal-force about quarter-chord point.

M Mach's (Mach) number, or ratio of velocity of airfoil (relative to the air) to speed of sound in undisturbed air.

m_o Slope of the normal-force coefficient curve against angle of attack.

p Pressure difference across the airfoil section at any station along the chord.

p_a Value of p corresponding to c_{n_a} at any station along the chord.

p_l Pressure difference across the lower surface at any station along the chord.

p_u Pressure difference across the upper surface at any station along the chord.

P Normal pressure coefficient, p/q, at any station along the chord.

P_a Value of P corresponding to c_{n_a}.

P_{a_1} Value of P corresponding to $c_{n_a} = 1.0$.

P_{ac}	Component of P_{a_1} which is a function of section aerodynamic center location.
P_{ac_1}	Value of P_{ac} corresponding to a value $x_{ac} = 1.0$.
P_{a_t}	Component of P_{a_1} which is a function of section thickness ratio and distribution for base airfoil.
$P_{a_{t_o}}$	Value of P_{a_t} for zero thickness ratio.
△ $P_{a_{t_1}}$	Rate of change of P_{a_t} with thickness ratio.
P_b	Value of P for basic part of pressure distribution.
P_{bc}	Component of P_b which varies with amount and distribution of camber.
P_{bc_1}	Value of P_{bc} corresponding to a value $z_c = 1.0$.
P_{bm}	Component of P_b, which varies with $c_{m_{ac}}$
P_{bm_1}	Value of P_{bm} corresponding to a value of $c_{m_{ac}} = 1.0$.
$P_{b\delta}$	Increment of basic normal-pressure coefficient due to flap deflection.
$P_{b\delta_1}$	Value of $P_{b\delta}$ for $c_{n_{b\delta}} = 1.0$
P_f	Surface-pressure coefficient of base airfoil at zero angle of attack.
P_l	Lower surface normal-pressure coefficient, p_l/q, at any station along the chord.
P_u	Upper surface normal-pressure coefficient, p_u/q, at any station along the chord.
P_o	Value of P corresponding to $c_n = 0$.
$P_{o\delta}$	Component of P distribution at $c_n = 0$ which results from deflection of flap from neutral.
$(P)_M$	Surface normal pressure coefficient, p/q, corresponding to a given Mach number M.
$(P)_0$	Surface normal pressure coefficient, p/q, corresponding to zero Mach number
q	Dynamic pressure, $1/2\ \rho\ V^2$

t Thickness of section in fraction of chord.

x_{ac} Fraction of chord that the aerodynamic center is forward of the quarter-chord point.

z_c Maximum ordinate of mean camber line divided by the chord.

α Angle of attack measured between relative wind and chord plane thru intersection of mean camber line with leading and trailing edges.

α_o Angle of attack at which $c_n = 0$.

α_b Angle of attack at which $c_n = c_{nb}$ (Basic angle of attack).

γ_a, γ_{ac}, γ_{bc}, γ_{bm} and $\gamma_{b\delta}$ Flap normal-force parameters.

η_a, η_{ac}, η_{bc}, η_{bm} and $\eta_{b\delta}$ Flap hinge moment parameters.

τ_m Ratio of $\Delta c'_m$ to Δc_m

δ Angular displacement of flap from neutral.

ρ Radius at leading edge of airfoil section in fraction of chord; also mass density of fluid.

0.5 INTRODUCTION TO METHOD

Reference (1) outlines a procedure for determining the airforce and moment per unit of span, acting at any airfoil section of the wing. The section lift coefficient c_l, drag coefficient c_d, and pitching moment coefficient $c_{m_{ac}}$ at any point along the span, (for any given loading condition of the complete wing) are therefore considered known; hence, also, the normal-force coefficient c_n and chord-force coefficient c_c. Before the wing ribs and other wing details can be properly designed, it is necessary to determine the corresponding distribution of air load over the chord of the section. The determination will be made in terms of the section normal-force coefficient, by giving the distribution of the latter along the chord. This distribution gives no resultant force in a chordwise direction; however, the chord-force is known and may be considered to be applied at the aerodynamic center. Furthermore, the theoretical normal-force distribution will be slightly modified in an arbitrary manner, so that it will give exactly the correct pitching moment about the aerodynamic center.

The procedure described in the following pages has essentially a theoretical-empirical basis. The "thin" airfoil theory of Glauert, references (6) and (7) as presented in a modified form by Theodorsen in reference (2), comprises the theoretical basis; while the empirical basis is essentially derived from the procedure and data given in references (4) and (5).

CHAPTER 1

DISTRIBUTION OF NORMAL-FORCE COEFFICIENT OVER AN AIRFOIL OF CONVENTIONAL SHAPE WITH FLAP NEUTRAL.

1.0 GENERAL

The normal-force intensity at any station along the airfoil chord may be found from the numerical difference between the pressures on the upper and lower surfaces at that station. When this pressure difference is plotted, normal to the chord at every point along the latter, then the area under the curve represents the normal-force; and any ordinate represents the normal-force intensity (i.e. normal-force per unit of chord). In the theory of "thin" airfoils, the wing-section of finite thickness is replaced by its median line, and the pressure difference is determined along the latter. The normal-force distribution coefficient is conveniently expressed by the ratio of the pressure difference p to the dynamic pressure q; that is, by $P = (p/q)$. With this notation, the general expression for the wing section normal-force distribution coefficient given in reference (2) is

$$P = P_b + P_{a_1} \quad (c_n - c_{n_b}) \qquad (1.1)$$

In the above equation, c_{n_b} is the section normal-force coefficient at the basic, or "ideal", angle of attack of the section. The latter is simply the angle of attack at which the stagnation point of the flow coincides with the leading edge of the equivalent thin airfoil (median line). Thus the equation states that the total normal-force distribution coefficient P (corresponding to the section normal-force coefficient c_n) is comprised of the basic distribution P_b and an additional distribution given by

$$P_{a_1} (c_n - c_{n_b}) = P_{a_1} c_{n_a}$$

where P_{a_1} is the additional distribution coefficient for $(c_n - c_{n_b})$ equal to unity. When the total normal force is zero, (1.1) becomes

$$P_o = (P_b - P_{a_1} c_{n_b}) \qquad (1.2)$$

and P_o is the distribution coefficient corresponding to this condition. The section normal-force intensity at any point along the chord is therefore given in general by

$$P = P_o + P_{a_1} c_n \qquad (1.3)$$

Theory indicates that $(P_{a_1} c_n)$ is simply the distribution which would be obtained by inclining a thin symmetrical airfoil (straight median line) at the angle of attack required to produce the total section normal-force c_n. Also, the theoretical additional normal-force acts at the quarterpoint of the chord, and therefore the moment about the quarterpoint is (theoretically) due entirely to the zero distribution P_o.

1.1 DETERMINATION OF P_b OVER AN UNCLASSIFIED AIRFOIL SECTION

This distribution depends primarily upon the shape of the median line of the section. If polar coordinates are employed with the origin at the mid-point of the airfoil chord, then the ordinates of the median line may be expressed in a Fourier series

$$z = \sum_{n=o}^{n=\infty} B_n \cos n\Theta \qquad (1.4)$$

when the abscissas along the chord are given by

$$x = \frac{1}{2}(1 - \cos\Theta) \qquad (1.5)$$

By reference (2) the theoretical basic pressure distribution is then given by

$$P_b = 4\left[\frac{d\epsilon}{d\Theta} + \frac{2(\epsilon - \epsilon_o) + (1 - \cos\Theta)(\epsilon_o - 2\epsilon + \epsilon_\pi)}{2\sin\Theta}\right] \qquad (1.6)$$

Where the so-called "distortion function" $\epsilon = \epsilon(x, z)$, at any point (x_i, z_i) along the median line is given by the integral formula (reference 2)

$$\epsilon_{\theta_i} = \frac{2}{\pi}\int_o^{\pi} \frac{z\, d\theta}{\cos\theta - \cos\theta_i} \qquad (1.7)$$

and where ϵ_o and ϵ_π are, respectively, the values of ϵ at $\theta = 0$ and $\theta = \pi$. The basic distribution occurs at the theoretical angle of attack given by

$$\alpha_b = - \frac{\epsilon_o + \epsilon_\pi}{2} \tag{1.8}$$

Substituting the value of z given by (1.4) into (1.7), and performing the indicated integration

$$\epsilon_\theta = \frac{2}{\sin\theta} (B_1 \sin\theta + B_2 \sin 2\theta + \ldots + B_n \sin n\theta) \tag{1.9}$$

The special values of ϵ at the leading edge (ϵ_o at $\theta = 0$) and at the trailing edge (ϵ_π at $\theta = \pi$) are

$$\epsilon_o = 2(B_1 + 2B_2 + 3B_3 + \ldots\ldots + nB_n) \tag{1.10}$$

$$\epsilon_\pi = 2(B_1 - 2B_2 + 3B_3 - \ldots + (-1)^{n-1} nB_n) \tag{1.11}$$

The basic pressure distribution (1.5) may now be written explicitly as (1.12)

$$P_b = 4 \left\{ \left[\frac{(\epsilon_\pi - \epsilon_o) - (\epsilon_\pi + \epsilon_o) \cos \theta}{2 \sin \theta} \right] + 2 \sum_{0}^{\infty} \frac{n B_n \cos n \theta}{\sin \theta} \right\}$$

It will be seen from (1.10), (1.11), and (1.12) that the basic distribution is zero at the leading and trailing edges of the airfoil in accordance with the definition of basic angle of attack as given by Theodorsen in reference (2).

The basic lift coefficient (at the angle of attack α_b) may be obtained by integrating (1.12) over the chord; thus

$$C_{n_b} = \int_0^{1.0} P_b \, dx$$

$$= \frac{1}{2}\int_0^{\pi} P_b \sin\theta \, d\theta$$

$$= \pi(\epsilon_\pi - \epsilon_o) \qquad (1.13)$$

The pitching moment coefficient about the leading edge at the basic angle of attack is

$$(c_m)_{LE} = -\int_0^{1.0} x \, P_b \, dx \qquad (1.14)$$

$$= -\frac{1}{4}\int_0^{\pi} P_b (1-\cos\theta) \sin\theta \, d\theta$$

Substituting the value of P_b from (1.12) into (1.14) and integrating, there is obtained

$$(c_m)_{LE} = \pi\left[B_1 + \frac{1}{4}(\epsilon_o - 3\epsilon_\pi)\right] \qquad (1.15)$$

The moment coefficient, at the basic angle of attack, about any point x_1 on the chord from the leading edge is easily shown to be

$$(c_m)_1 = \pi\left[B_1 - (x_1 - \frac{1}{4})\epsilon_o - (\frac{3}{4} - x_1)\epsilon_\pi\right] \qquad (1.16)$$

Expressing (1.10), (1.11), and (1.13) in the following forms

$$\epsilon_o = 2\left[B_1 + B_2\left(2 + \frac{4B_4}{B_2} + \frac{6B_6}{B_2} + \dots\right) + B_3\left(3 + \frac{5B_5}{B_3} + \frac{7B_7}{B_3} + \dots\right)\right]$$

$$= 2\left[B_1 + B_2\left\{P_1(B)\right\} + B_3\left\{P_2(B)\right\}\right] \qquad (1.17)$$

$$\epsilon_{\sim} = 2\left[B_1 - B_2\left(2 + \frac{4B_4}{B_2} + \frac{6B_6}{B_2} + \dots\right) + B_3\left(3 + \frac{5B_5}{B_3} + \frac{7B_7}{B_3} + \dots\right)\right]$$

$$= 2\left[B_1 - B_2\left\{P_1(B)\right\} + B_3\left\{P_2(B)\right\}\right] \qquad (1.18)$$

there results

$$c_{n_b} = -4\pi B_2\left[2 + \frac{4B_4}{B_2} + \frac{6B_6}{B_2} + \dots\right]$$

$$= -4\pi B_2\left\{P_1(B)\right\} \qquad (1.19)$$

where P_1 (B) and P_2(B) are the indicated functions of the B-coefficients.

Substituting ϵ_o and ϵ_π from (1.17) and (1.18) into (1.16) yields

$$(c_m)_i = \pi \left[4B_2 \left\{P_1(B)\right\} \left(\frac{1}{4} - x_i\right) + B_2 \left\{P_1(B)\right\} - B_3 \left\{P_2(B)\right\} \right] \quad (1.20)$$

A convenient method of computing the B-coefficients from the ordinates of the median line is given in Appendix A.

The following aerodynamic data will have been made available by tests:

$(c_m)_{ac}$, the coefficient of moment about the aerodynamic center.

m_o , the slope of the normal-force curve against angle of attack (the slope of the lift curve is sufficiently accurate and may be used).

α_o , the angle of attack of zero lift. (Note: --α must be measured as defined under Section 0.4.)

x_{ac} , the location of the aerodynamic center (in terms of chord for this development) forward of the quarter chord point; equal to $(\frac{1}{4} - x_i)$ as used in (1.20)

The theoretical and experimental values of the zero lift angle of attack are equated; then

$$\alpha_o = -\epsilon_\pi \quad (1.21)$$

The true and theoretical values of the basic lift are also equated

$$c_{n_b} = m_o (\alpha_b - \alpha_o) = \Upsilon(\epsilon_\Upsilon - \epsilon_o) \tag{1.22}$$

Introducing the test values of $(c_m)_{ac}$, x_{ac}, α_o and the calculated values B_2, $P_1(B)$, and $P_2(B)$ into (1.18), (1.19), (1.20) and (1.21); modified values are obtained for B_1 and B_3 (designated as B'_1 and B'_3) which modify the theoretical lift curve so as to correspond to the actual curve determined by test.

Assuming the relationship to hold that

$$\frac{B'_n}{B'_3} = \frac{B_n}{B_3} \tag{1.23}$$

a modified set of B-coefficients is obtained which, when employed in (1.12), yields a P_b distribution corresponding to the theoretical value of c_{n_b} which produces values of $(c_m)_{ac}$ and x_{ac} agreeing with the test data.

A scheme is presented in Appendix A for modifying the B-coefficients to correspond to the test data and also for calculating the P_b distribution from these modified coefficients. The calculated results, together with the results of the application of the method of paragraph 1.2 and Appendix B (P_{a_1} distribution), are readily inserted into equations (1.2) and (1.3) for a tabular determination of the pressure distribution at zero lift and at any desired total normal-force coefficient.

1.2 DETERMINATION OF P_a FOR AN UNCLASSIFIED AIRFOIL SECTION

The additional pressure distribution for points along the airfoil chord such that $x > o$ is given in terms of the dynamic pressure by the theoretical formula.

$$P = 4 (\alpha - \alpha_b) \sqrt{\frac{1}{x} - 1}$$

$$= \frac{4 c_{n_a}}{m_o} \sqrt{\frac{1}{x} - 1}$$

(reference 2)

Since the thin airfoil gives rise to infinite velocity at the leading edge, for all other angles of attack than the basic, the last formula cannot be used to compute the normal force distribution over the entire chord of an actual airfoil. Theoretical investigation of the flow around the nose of a wing section of finite thickness shows that the normal-force intensity reaches a maximum at a chordwise location equal to one-half the leading-edge radius ρ, and has the value

$$(P/q)_{max} = P_m = \frac{4 (\alpha - \alpha_b)}{\sqrt{2\rho}} = \frac{4 c_{n_a}}{m_o \sqrt{2\rho}}$$

(reference 2)

It has been shown that m_o may also be considered a function of ρ and hence $(p/q)_{max}$ may be semi-rationally expressed as an explicit function of the radius of curvature at the leading edge. For practical application to the calculation of the additional pressure distribution over an airfoil, it is necessary to modify the above two theoretical formulas in such a manner that the resulting expression for P will include the applicable value of P_m occuring at the proper chordwise location and, when integrated over the chord, will give a value of the normal force coefficient equal to unity and a chordwise location of the loading centroid corresponding to that of the aerodynamic center as determined by test.

n Appendix B is presented a method of determining the P_{a_1} distribution over an unclassified airfoil section for which the geometrical and aerodynamic center characteristics are known. The calculated results, together with the results of the application of the method of Appendix A (P_b distribution), are readily inserted into equations (1.2) and (1.3) for a tabular determination of the pressure distribution at zero lift and at any desired value of the total normal-force coefficient.

1.3 DETERMINATION OF P_b AND P_a DISTRIBUTION OVER A CLASSIFIED (NACA) AIRFOIL SECTION

1.31 GENERAL

Airfoils designed and/or tested by the NACA are classified according to their additional and basic pressure distributions. The character of the former distribution is designated by a capital letter and that of the latter by a number comprised of two digits. This is the Committee's "PD-Classification". (Ref. 4, Appendix B)

1.32 DETERMINATION OF P_b

When the classification is known, P_b and its integral, c_{n_b}, over the chord, are obtained from

$$P_b = P_{bm} + P_{bc} = - c_{m_{ac}} P_{bm_1} + z_c P_{bc_1} \tag{1.24}$$

$$c_{n_b} = c_{n_{bm}} + c_{n_{bc}} = - c_{m_{ac}} c_{n_{bm_1}} + z_c c_{n_{bc_1}} \tag{1.25}$$

where z_c is the maximum ordinate of the mean camber line divided by the chord, $c_{m_{ac}}$ is the measured section moment coefficient about the aerodynamic center, P_{bm} and P_{bc} are the partial distributions which depend on moment and camber, repsectively, and $c_{n_{bm}}$ and $c_{n_{bc}}$ are the section coefficients obtained by integrating P_{bm} and P_{bc} over the airfoil chord. For example, the number 12 in classification B12 indicates that P_{bm} is class 1 and P_{bc} is class 2. Values of P_{bm} and P_{bc} for various classifications are given in Table C-2 of Appendix C, rows (9) thru (13) of which demonstrate the method of calculating the basic pressure for a classified airfoil.

1.33 DETERMINATION OF P_a

The additional pressure distribution classification of the NACA is designated by the code letter (A, B, C, D, E, or F). Accordingly, the P_a distribution may be calculated from the data and by the method indicated in reference (4). Tabular data from reference (4) is presented in table C-1 of Appendix C with additional classifications as noted. By use of this data in the following relations the additional pressure for a classified airfoil may be determined directly.

$$P_{a_1} = P_{a_t} + x_{ac} P_{ac_1} \qquad (1.26)$$

where P_{a_t} represents the values of an empirically corrected theoretical partial distribution which depends essentially upon the thickness distribution of the base airfoil; P_{ac_1} represents the values of a distribution which is assumed to be identical for all airfoils, and x_{ac} is the fraction of the chord that the aerodynamic center is forward of the quarter chord point. The resulting value of P_{a_1} is that employed in equations (1.2) and (1.3). Since the additional pressure distribution due to thickness P_{a_t} has a practically linear variation with t for any particular chord station for airfoil forms similar to the NACA 0010, it may be represented by

$$P_{a_t} = P_{a_{t_o}} + t \quad \Delta P_{a_{t_1}} \qquad (1.27)$$

where $P_{a_{t_o}}$ represents the values obtained by extrapolating the data from table C-1 of Appendix C to t = 0 and $\Delta P_{a_{t_1}}$ represents the difference between the values obtained by extrapolating the data to t values of zero and 1.0. A scheme for calculating the zero lift pressure distribution incorporating this general method of P_{a_t} determination is indicated on table C-2 of Appendix C. The general procedure is applicable to use with all airfoil sections including those having values of t intermediate between the classification values.

1.34 DETERMINATION OF P_o

The values of P_b, P_{a_1}, and C_{n_b} determined by the methods of sections 1.32 and 1.33 are introduced into equation (1.2) for a calculation of P_o. By use of equation (1.3) the pressure distribution may be calculated for any desired normal-force coefficient. An extension of the tabular method of calculation shown in the example on Table C-3 of Appendix C is most desirable.

CHAPTER 2

DISTRIBUTION OF PRESSURE BETWEEN UPPER AND LOWER SURFACES OVER AN AIRFOIL - INCLUDING THE EFFECT OF COMPRESSIBILITY

2.0 INTRODUCTION.

It has been found convenient to divide the distribution of the normal-force coefficient along the chord of an airfoil into two separate distributions, one of which remains constant while the other varies directly with the value of c_n. Unfortunately, the surface pressures do not maintain their proportionality to c_n and they must therefore be determined separately for each value of c_n for which they are desired. The simplified method of reference 9 for calculating the pressure distribution, however, makes this no great handicap, expecially since the critical values of c_n can usually be estimated from the normal-force distribution alone.

2.1 REQUIREMENTS AND APPLICABILITY

This method requires a knowledge of:

(1) The chordwise normal-force distribution on the airfoil section, and

(2) The pressure distribution over the base profile of the airfoil section (i.e., the profile of the same airfoil were the camber line straight and the resulting airfoil at zero angle of attack). The method is applicable, to date, to plain airfoils and airfoils with plain trailing-edge flaps. The calculations of the normal-force distribution for ordinary airfoils has been explained in Chapter 1,

while Chapter 3 will consider the distribution for flapped airfoils. The base-profile pressure distributions would ordinarily have to be computed by the method of references 2 and 3, or be determined from pressure distribution tests made on an airfoil having the same base-profile. Fortunately, most of the N.A.C.A airfoils (as well as the Clark Y and Gottingen 398) have the same base-profile with variations in maximum thickness, as represented by N.A.C.A. 0010. The pressure distribution for this profile for thickness ratios ranging from 6 to 35 percent were computed in reference 9 and the results have been charted in figure 2-1.

2.2 INTERNAL PRESSURE

The unit air load on the covering of a wing at any point is the difference between the absolute pressure within the wing and the absolute pressure on the surface at that point. The pressure within the wing is dependent upon the location of the vents. In steady horizontal flight the internal pressure should be equal to the surface pressure at the vent. If the vent is at a location where the surface pressure changes rapidly with change in normal-force coefficient, a rapid maneuver will cause a flow of air throught the vent and the internal pressure may show considerable lag from that occurring at the outside of the vent at any instant. For this reason, it is better to locate the vents at points where the surface pressure change is small, such as near the trailing edge. Another advantage of the trailing edge location is that the surface pressure relative to the undisturbed static pressure is nearly zero and does not change appreciably with a variation in speed. It should be kept in mind that there is no vent location that will obviate the necessity of air passing in or out of the wing during a change in altitude and it is important to have sufficient vent area to allow this flow to occur without an excessive pressure drop through the vent, especially in airplanes which dive at high speeds. In the following procedure, all individual surface pressure coefficients are given relative to the static pressure coefficient in the undisturbed air. These pressures represent the true load on the covering only when the internal pressure is equal to the undisturbed atmospheric pressure. When the internal pressure is known to be different, the correct covering load may be found by adding or subtracting from all surface pressures as computed, the difference between the true internal pressure and the atmospheric pressure.

2.3 PROCEDURE

If P is the normal-pressure coefficient at any particular chord station, P_u and P_l are the upper and lower surface-pressure coefficients and P_f the surface-pressure coefficient of the base profile at zero angle of attack at the same chord station, then, designating an upward force as positive.

$$P = -P_u + P_l \tag{2.1}$$

$$P_u = 1 - \frac{\left(1 - P_f + \frac{P}{4}\right)^2}{1 - P_f} \tag{2.2}$$

and

$$P_l = 1 - \frac{\left(1 - P_f - \frac{P}{4}\right)^2}{1 - P_f} \tag{2.3}$$

It is thus seen that, with a knowledge of the P and P_f distributions, equation 2.2 or 2.3 may be used to determine either the P_u or the P_l distribution. The distribution for the other surface is then obtained by equation 2.1. A convenient chart is provided in figure 2.1 and its use is explained in paragraph C.12 for solving equation 2.2 or 2.3 for corresponding values of P and $(1 - P_f)$. A plot of P_f versus t for several chord stations for the N.A.C.A. 0010 family base profile is included in this chart so that values of P_u or P_l may be rea directly for given values of t. For airfoils which have thickness distributions differing from that of the NACA 0010 series (such as airfoils of the unclassified type for which the methods of paragraphs 1.1 and 1.2 are employed) P_f may be calculated for the base profile by the procedures described in reference (3) and P_u and P_l may then be subsequently determined with the use of

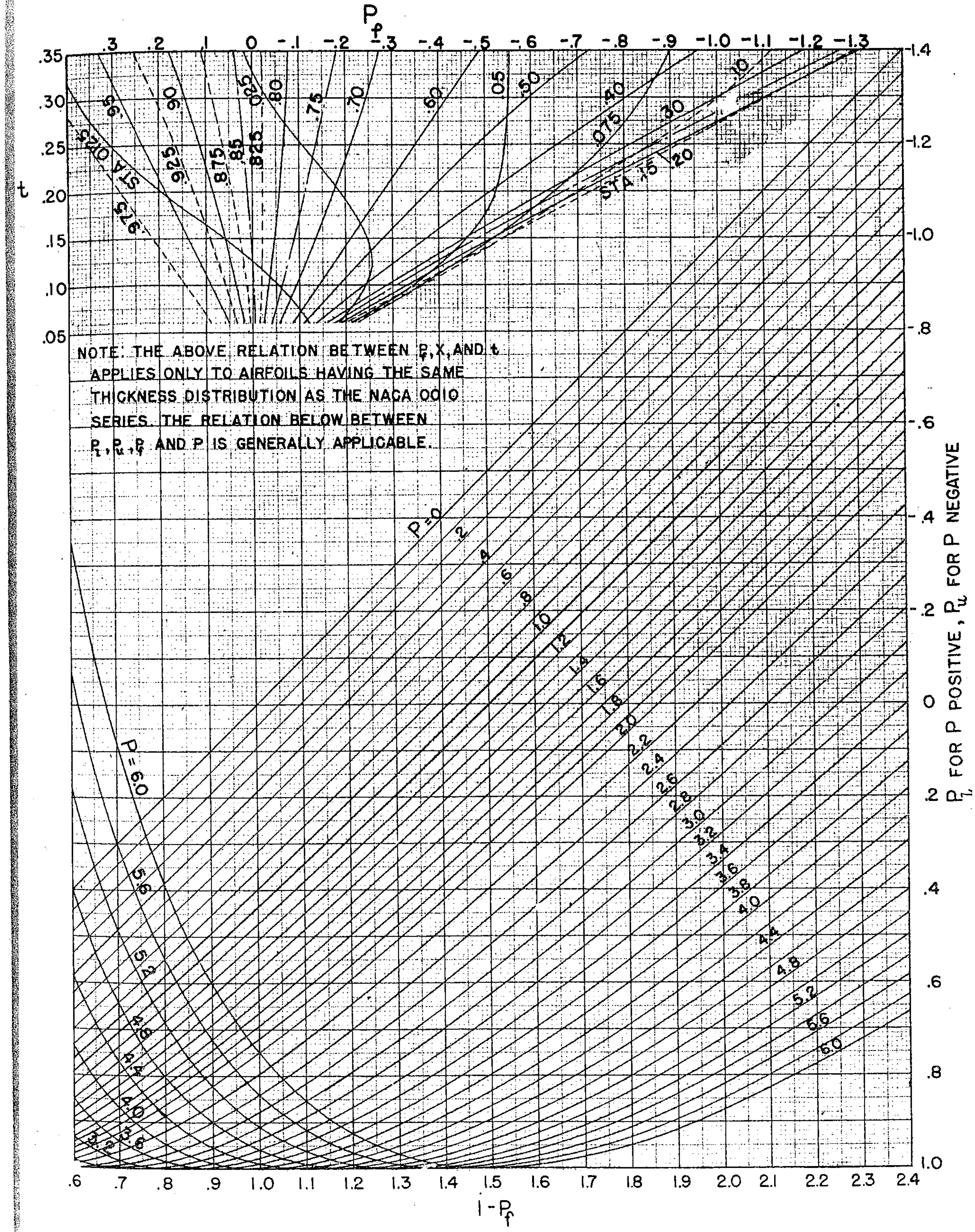


FIG. 2-1

CHART FOR DETERMINATION OF P_u AND P_l

the lower portion of figure 2-1. The actual pressure relativ to the static pressure in the undisturbed air at any particular speed of the airplane can then be found by multiplying these P_u and P_l values by the corresponding value of q.

2.4 APPLICATION TO AIRFOILS WITH FLAPS

It is pointed out in reference 9 that the method is highly accurate for thin airfoils with small amounts of camber and less accurate for thick highly cambered airfoils. Furthermore, it applies only to airfoils that are not stalled and the results become less accurate as the stall is approached. This is especially true for airfoils with flaps. It gives reasonable results for plain flaps that are deflected not more than approximately 15 degrees. For larger flap deflections, the agreement with test is still reasonable over the portion of the airfoil forward of the hinge. Since the flap is usually stalled under these conditions, the method does not give the correct distribution between the flap surfaces. This is not so important for a flap, however, because a knowledge of the pressure difference between the two surfaces is usually sufficient for design purposes. When a plain flap is stalled, the pressure on the upper surfaces is nearly constant over the flap chord. This is the condition which exists over the inner surfaces of a split flap for all flap angles. For these reasons, the method may be applied to airfoils with plain or split flaps in estimating the surface pressures forward of the hinge and the external surface pressures and total load for the section with flap displaced from neutral. This method does not, however, provide determination of the actual pressure existing between an airfoil and deflected split flap. Knowledge of such a pressure is essential for the determination of distribution of actual applied loads between the flap and airfoil structures. Estimation of the uniform pressure in this region should be based upon pertinent test data where available. NACA Technical Note 627 is suggested as a reference providing such test data.

2.5 EFFECT OF COMPRESSIBILITY

The theoretical basis of the foregoing pressure calculations includes the assumption that air acts as a perfect, incompressible fluid. However, since it is actually an elastic fluid, this characteristic must be acknowledged and taken into account in the pressure calculations. It is found that the effect of compressibility is, in general, to increase the surface pressures above those which would be developed under identical conditions of airfoil velocity and attitude in an incompressible fluid. This effect becomes greater with an increase in velocity and is primarily a function of the Mach number or ratio of the airfoil velocity (relative to the air) to the speed of sound in the undisturbed air.

Both theory and tests have indicated that the behavior of the elastic fluid can be calculated with a reasonable degree of accuracy for speeds up to a certain critical speed at which the so called "compressibility burble" phenomenon is encountered. At such a speed there occurs a compression shock which causes a breakdown of the existing regime of flow into one which, in the light of present knowledge, is not completely understood and does not lend itself to a simple calculation of the resulting pressure distribution. Although the local speed of sound may be exceeded to some extent without necessarily causing the compressibility burble, it has been found to be a reasonably accurate criterion that the critical Mach number, or Mach number at which the burble first occurs, is approximately that for which the local velocity at some point on the surface reaches the local velocity of sound.

Having calculated the pressure coefficient at any point on the surface according to the procedures described in the foregoing portions of this publication, the pressure at any value of Mach number may be estimated by use of Figure 2-2. Although these foregoing procedures have neglected the effect of compressibility, they are based upon wind tunnel determinations of airfoil characteristics and hence correspond to finite values of Mach number.

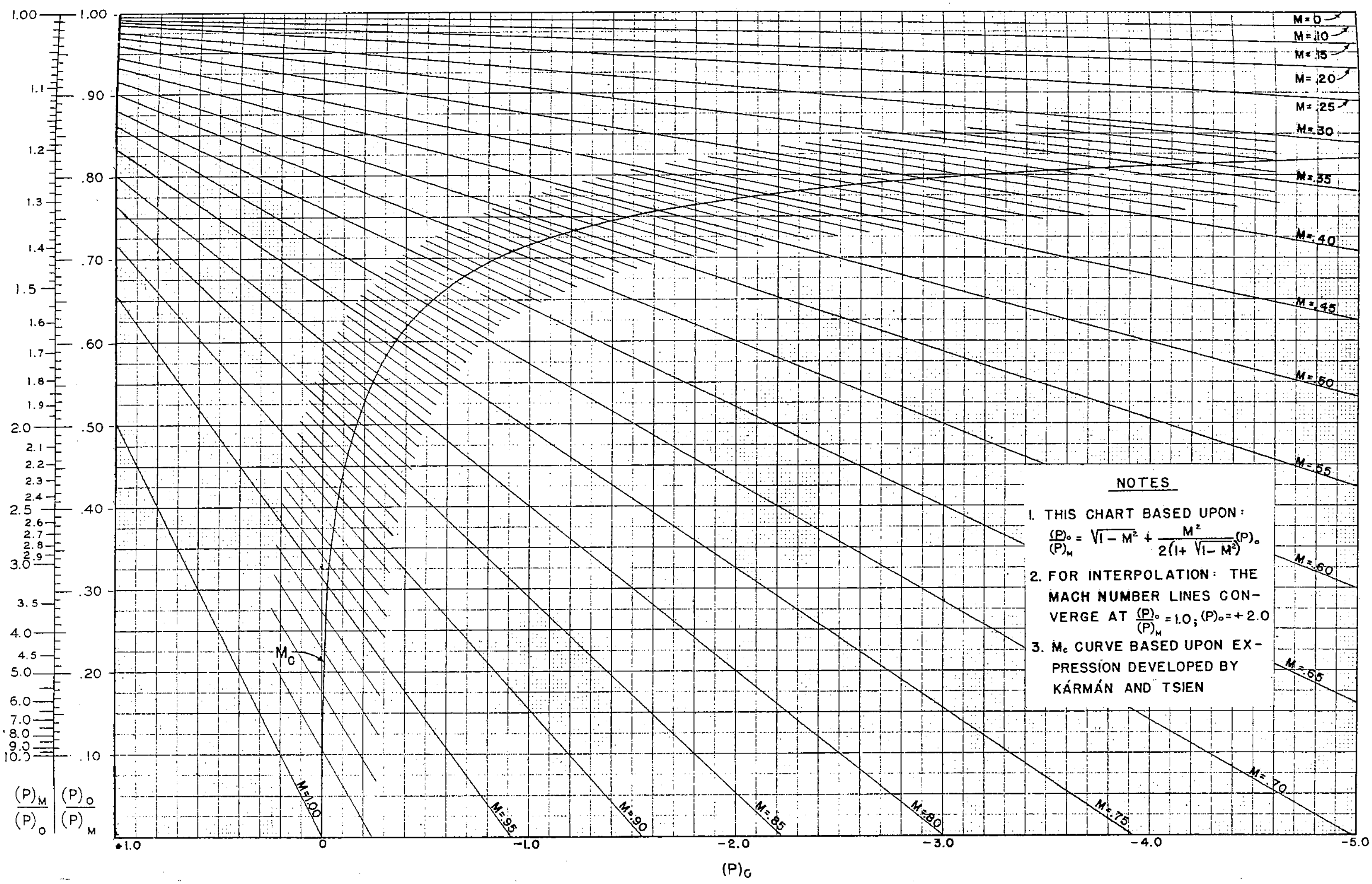


FIG. 2-2
PRESSURE COEFFICIENT VS. MACH NUMBER
INCLUDING DETERMINATION OF M_c

However, the error involved is small if these calculated pressures are interpreted as those applicable to zero Mach number.

In Figure 2-2, the reciprocal $\left(\frac{(P)_O}{(P)_M}\right)$ of the ratio of the surface pressure coefficient at finite Mach number to that at zero Mach number $\left(\frac{(P)_M}{(P)_O}\right)$ is plotted as a function of the Mach number (M) and of the surface pressure coefficient at zero Mach number $(P)_O$. The value of $\frac{(P)_O}{(P)_M}$ is equal to the ordinate corresponding to the intersection of the applicable values of $(P)_O$ and M. Alternately, the value of the ratio $\frac{(P)_M}{(P)_O}$ may be obtained directly from the conversion scale at the extreme left. This figure is based upon the following relation which is taken from reference (11).

$$\frac{(P)_O}{(P)_M} = \sqrt{1-M^2} + \frac{M^2}{2(1+\sqrt{1-M^2})} (P)_O$$

A "cut off" curve is also presented on Figure 2-2 for the determination of the critical Mach number (M_c) corresponding to any given value of $(P)_O$. M_c is interpreted as the value of Mach number at or above which the compressibility burble is likely to occur. The data upon which this curve is based are taken from the relationship developed by Kármán and Tsien and presented in Figure 19 of reference (11).

It is noted that this critical Mach number decreases with an increase in the negative value of the pressure coefficient $(P)_O$ and hence the latter is the criterion for the speed at which the compressibility burble will take place. M_c may be obtained directly from Figure 2-2 by the intersection of the cut-off curve and the $(P)_O$ coordinate corresponding to the maximum negative surface pressure calculated for any point on the airfoil. The critical velocity may then be obtained as the product of M_c and the applicable speed of sound in the undisturbed air, which latter may be determined from Figures 2-3 and 2-4. The actual pressure at any point is obtained as the product of $(P)_M$ and the applicable q.

It is noted that Figure 2-2 is applicable to the prediction of the effect of compressibility on surface pressures (and critical speeds) over bodies other than airfoils -- i.e. on windshields, cowlings, fuselages, etc.

The lift coefficient c_l is reasonably accurately represented by the net area under the curves of individual surface pressures plotted against chord. Hence c_l at any Mach number will be represented by the area under the curves of surface pressure vs. chord which have been modified by the use of Figure 2-2. The tests reported in reference (10) indicate that, at least below critical speeds, the angle of zero lift is practically unchanged. Hence, the slope of the lift curve at any Mach number will bear approximately the same relation to the slope at zero Mach number (or at low wind tunnel values) that the lift coefficient modified for Mach number as described above bears to the value at zero Mach number.

If operation of any airplane is contemplated at speeds in excess of that speed determined from analytical considerations to be the critical speed (such as by use of Figure 2-2), the effect of such supercritical speeds upon the pressure distribution and magnitudes of local pressures should be investigated in connection with the design of all structural components directly or indirectly affected. Such a determination of supercritical pressures and pressure distributions should be based upon the results of aerodynamic tests similar to those reported for the NACA 4412 airfoil in reference (10). If such data are not available, it is considered that for structural design considerations, the pressures indicated by the curves of Figure 2-2, extended beyond the apparent subcritical range to the applicable values of M, should be used as minimum values.

Figures 2-3 and 2-4 are based upon the accepted thermodynamic expressions for the velocity of sound in standard

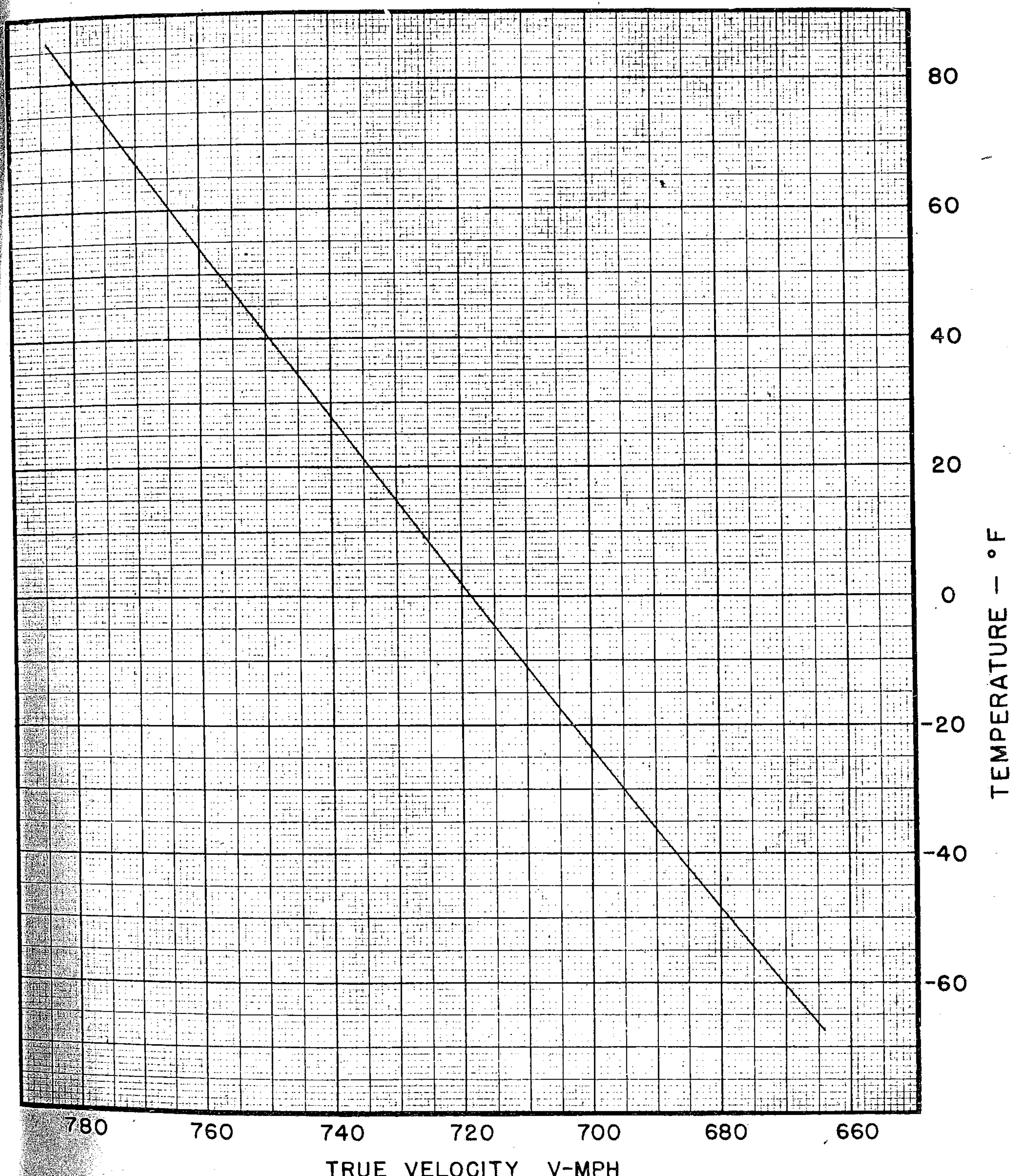


FIGURE 2-3

VELOCITY OF SOUND IN STANDARD ATMOSPHERE air expressed in miles per hour:

$$a = .81\sqrt{\frac{p}{\rho}}$$; where p and ρ are respectively the pressure and density expressed in consistent units.

$$a = 33.45\sqrt{t + 460}$$; where t is the temperature expressed in degrees Fahrenheit.

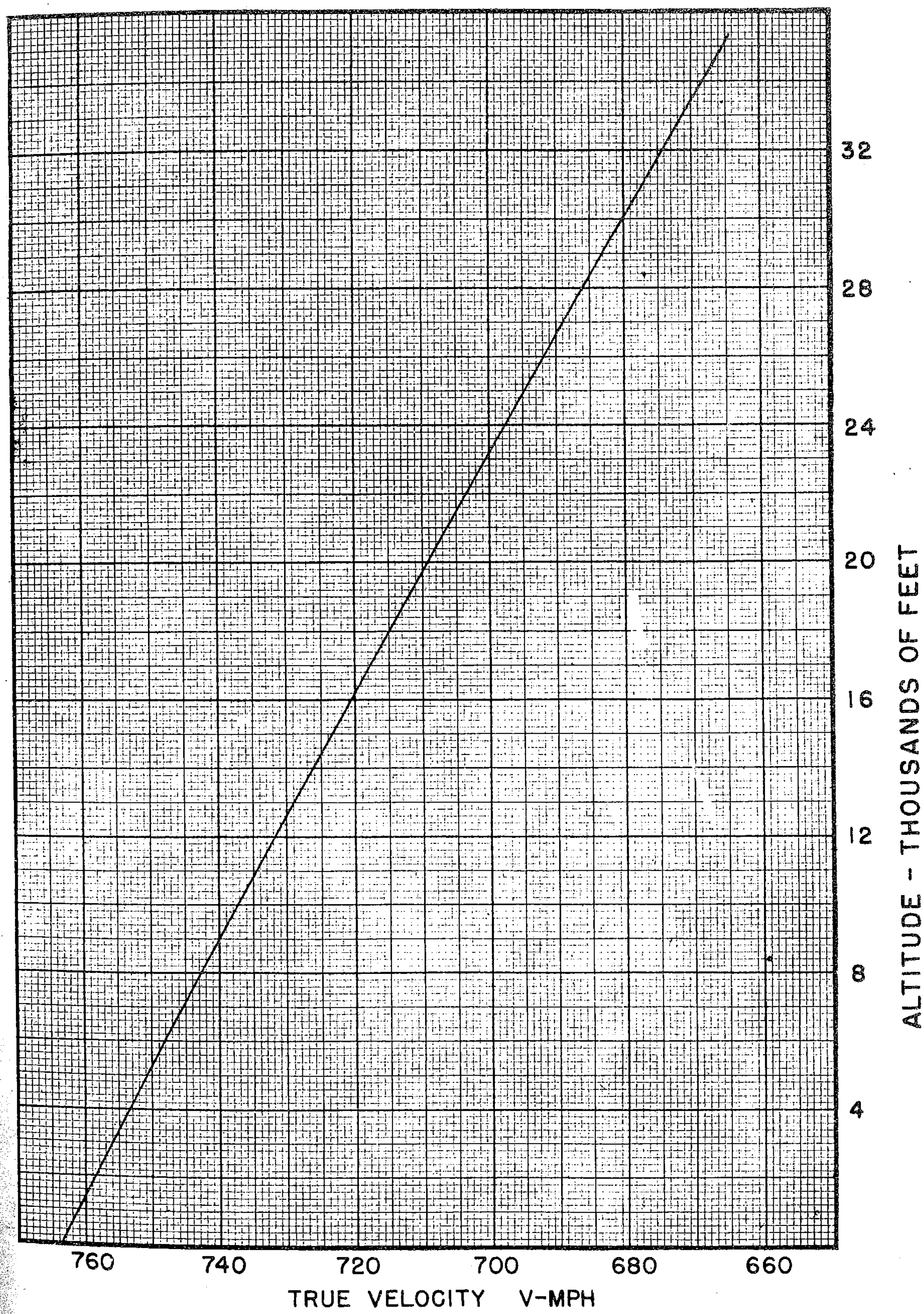


FIGURE 2-4

VELOCITY OF SOUND IN STANDARD ATMOSPHERE.

CHAPTER 3

DISTRIBUTION OF NORMAL-FORCE COEFFICIENT

FOR AN AIRFOIL WITH FLAP

3.0 INTRODUCTION

A method is presented, in reference 5, for the determination of the chordwise distribution of normal-force over an airfoil of ordinary profile and camber, with a plain trailing edge flap, a split flap, or a serially hinged flap. The form and magnitude of that portion of the normal-force distribution caused by displacing the flap from the neutral position can be determined for an airfoil and flap combination for which the section characteristics have been determined. The method was developed by correlating theoretical investigations and numberous experimental investigations employing airfoils having very small gaps between the wing and the leading edge of the flap. There is evidence that the existence of any gap between the wing and the leading edge of a plain flap has a pronounced effect upon the aerodynamic characteristics; however, the actual size of gap (between .005c and .010c) is unimportant. It is probable that this gap effect is also true for the split flap airfoils. In the absence of better information, the method outlined herein cannot be considered applicable to flaps with large gaps. Flaps having the hinge located midway between the upper and lower surfaces were used in these tests so that the data obtained (and hence the data and methods presented herein) must be considered strictly applicable only to plain flap airfoils with surface curvatures not less than half the airfoil depth at the hinge.

3.1 REQUIRED AND DERIVED PARAMETERS

The increment in quarter-chord moment coefficient (Δc_m) resulting from displacement of the flap from the neutral position should be obtained from the results of force tests of the normal airfoil and of the airfoil with flap displaced. This increment is defined as:

$$\Delta c_m = c_{m_2} - c_{m_1} \tag{3.1}$$

where c_{m_1} is the quarter-chord pitching-moment coefficient for the unflapped section or section with flap neutral, and c_{m_2} is the quarter-chord pitching-moment coefficient of the airfoil-flap combination with the fixed surface at the same attitude and with the flap displaced. When $c_{m_{ac}}$ and the aerodynamic center location are known, the quarter-chord pitching-moment coefficient, if not obtained directly from published results, may be obtained from

$$c_m = c_{m_{ac}} + c_l \quad (x_{ac}) \tag{3.2}$$

where c_l is the value of the section lift coefficient for which the distribution of normal-force coefficient is to be calculated.

It is analytically convenient, for the purposes of the following pressure calculation procedure, to treat a fictitious condition of the airfoil-flap combination in which the normal pressures actually developed with flap deflected are considered to be applied to the airfoil and flap in neutral attitude. Such a fictitious pressure distribution corresponds to fictitious values c'_{m_2}, c'_n which are functions of (1) the actual values (c_{m_2} and c_n) developed with flap deflected (2) the amount of flap deflection (δ) and (3) the flap-chord ratio (E).

Correspondingly, the fictitious value of the quarter-chord moment coefficient increment is defined by:

$$\Delta c'_m = c'_{m_2} - c_{m_1}$$

and is correlated with the actual moment coefficient increment by the relation:

$$\Delta c'_m = \tau_m \Delta c_m \qquad (3.3)$$

The component of the basic normal-force coefficient due to flap deflection is related to the fictitious moment coefficient increment according to:

$$c_{n_{b\delta}} = \frac{\Delta c'_m}{G} = \frac{\tau_m}{G} \Delta c_m \qquad (3.4)$$

where G is the net moment arm of the basic normal force distribution (corresponding to $c_{n b\delta}$) about the quarter-chord point. The factor $\frac{\tau_m}{G}$ is presented in Figure 3-1 as a function of δ and E.

The additional normal-force coefficient is determined in a manner analogous to that used in the calculation of flap-neutral pressure distribution (i.e. the difference between the total coefficient and the basic coefficient) and is equal to:

$$c_{n_a} = c_n - (c_{n_b} + c_{n_{b\delta}}) \qquad (3.5)$$

The total load on the flap (aft of the hinge) is expressed as equal to:

$$c_{n_f} \; E \; c \; q$$

which expression in turn defines the flap normal force coefficient c_{n_f}. This coefficient is determined by the relation:

$$c_{n_f} = \gamma_{b\delta}\, c_{n_{b\delta}} + \gamma_{bc}\, c_{n_{bc}} + \gamma_{bm}\, c_{n_{bm}} + (\gamma_a + \gamma_{ac}\, x_{ac})\, c_{n_a} \qquad (3.6)$$

which is based upon the assumption that the contribution of the additional normal-force distribution resulting from flap deflection is negligible. The values of the various γ factors are plotted on Figures 3-2 and 3-3.

From the value of c_{n_f} thus obtained the fictitious normal force coefficient may be calculated according to:

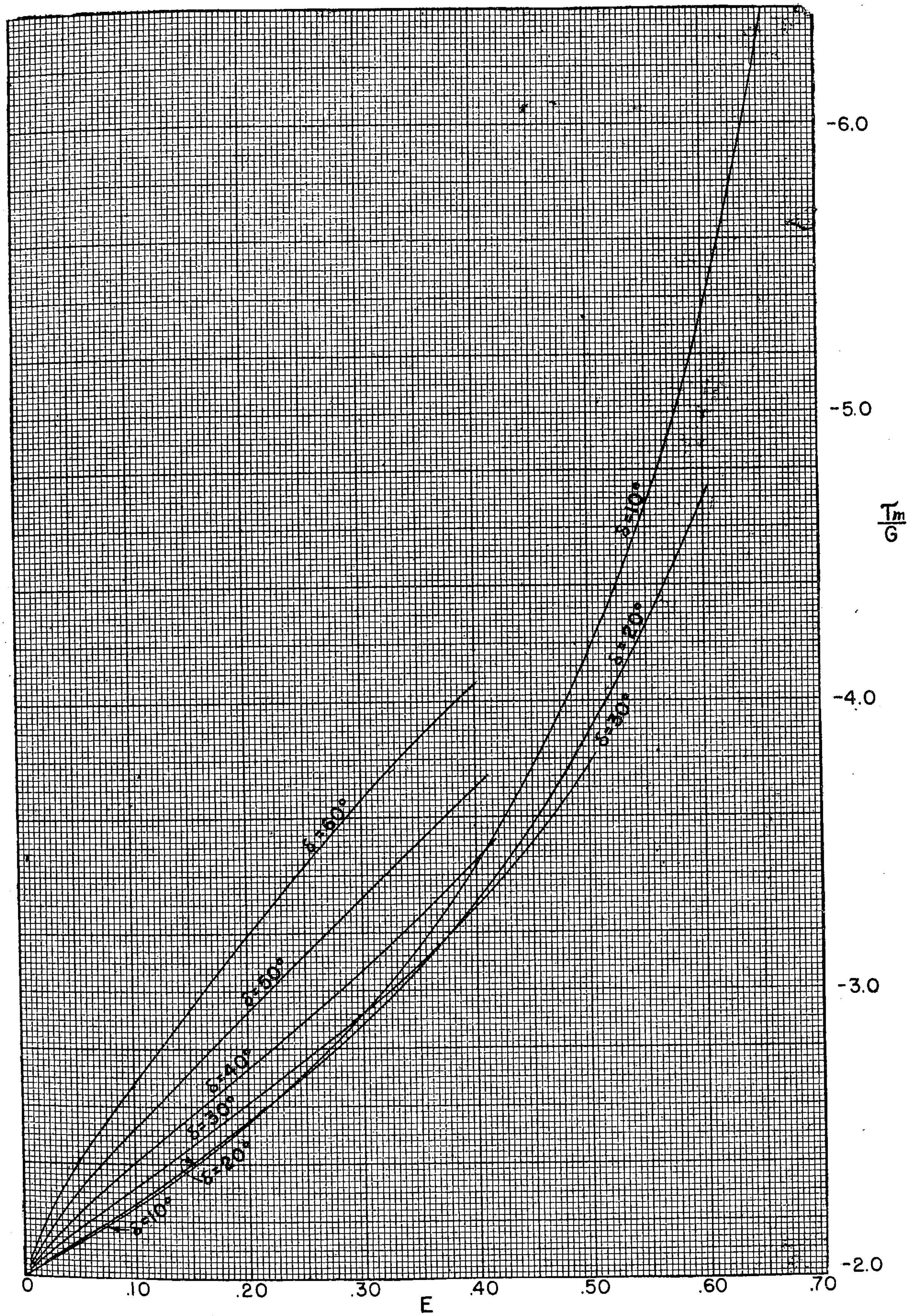


FIG. 3-1

E vs. $\frac{T_m}{G}$

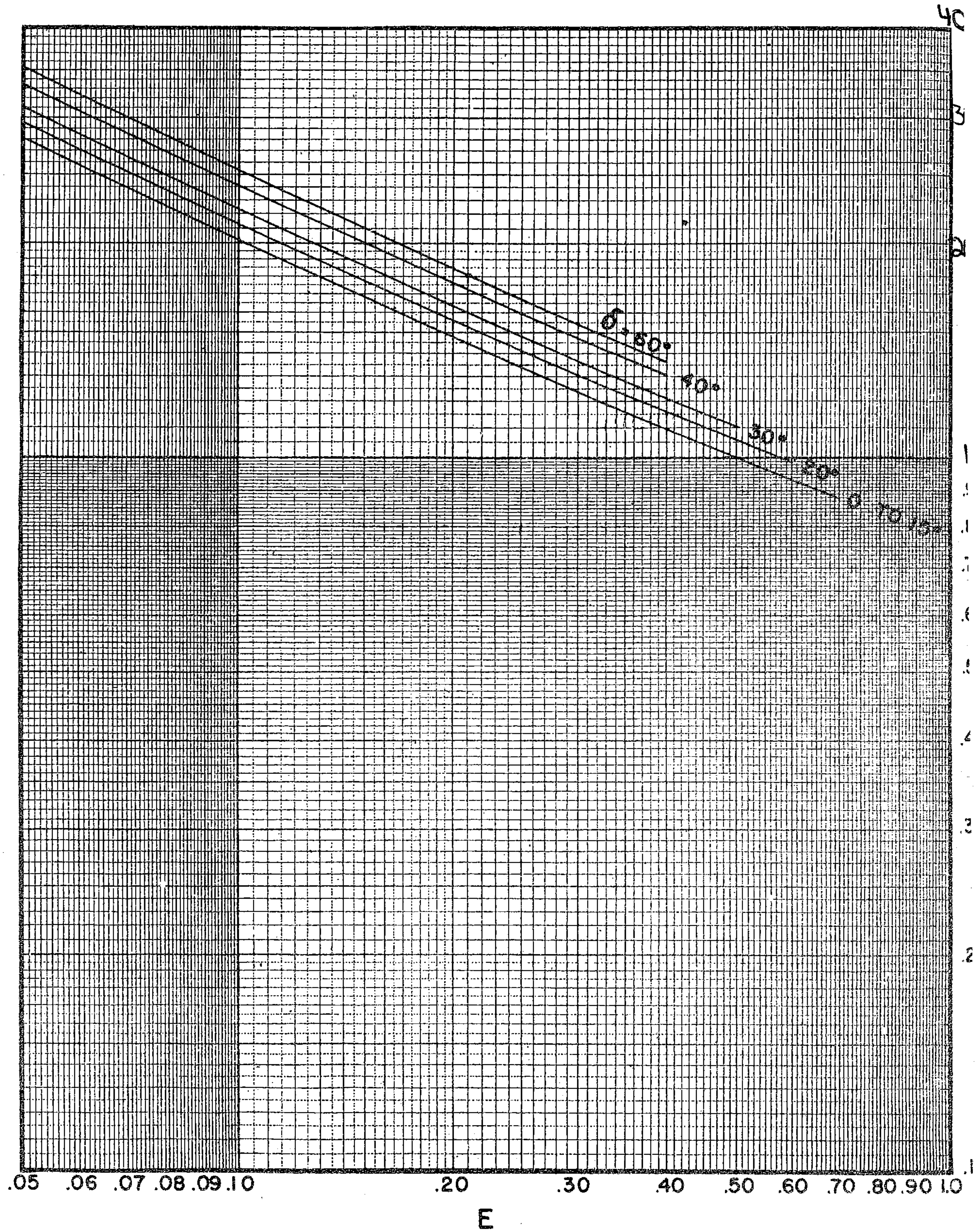


FIG. 3-2

$\gamma_{b\delta}$ VS. E

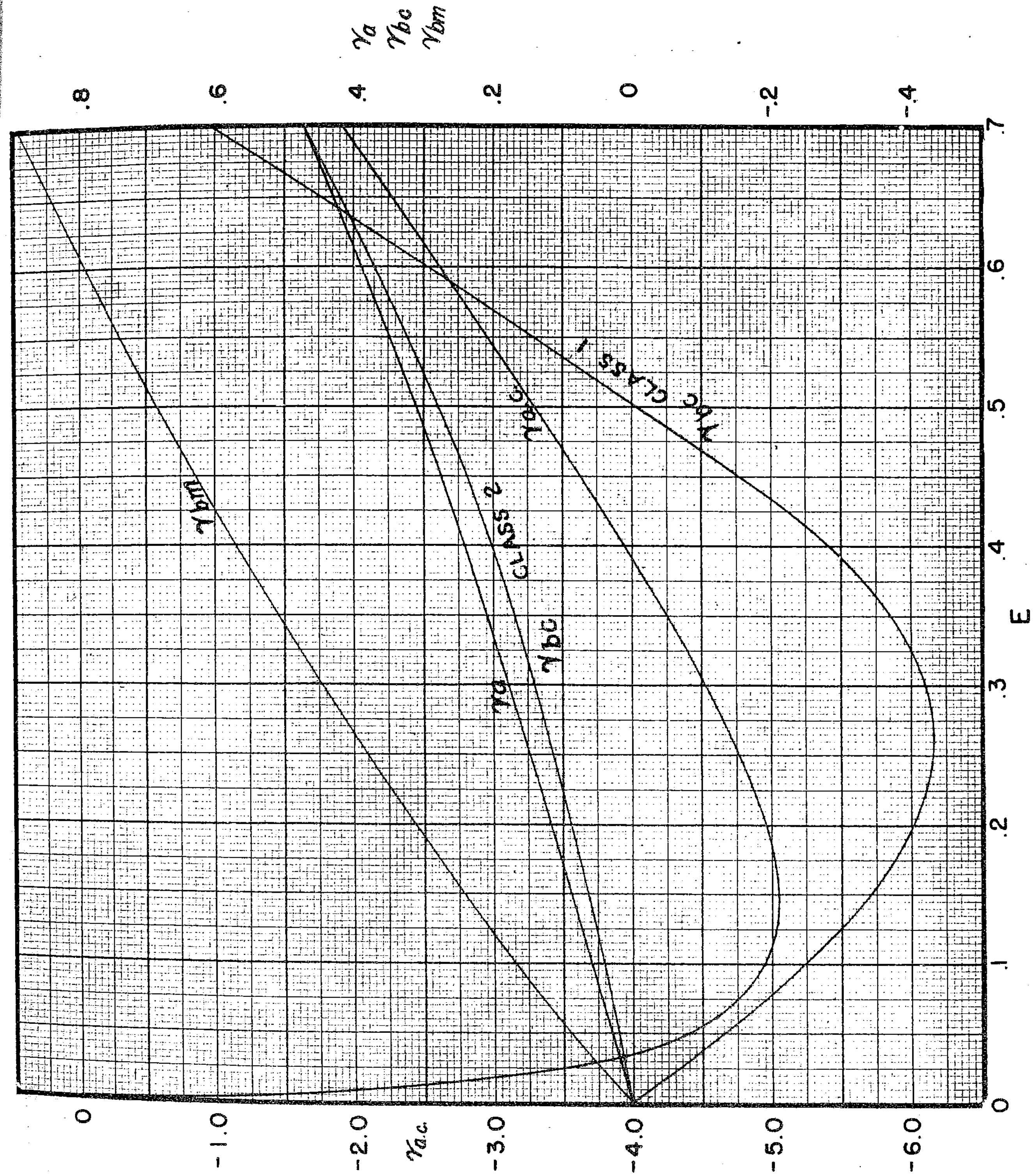


FIG. 3-3

$\gamma_{a.c.}, \gamma_a, \gamma_{bc}, \gamma_{bm}$ VS E

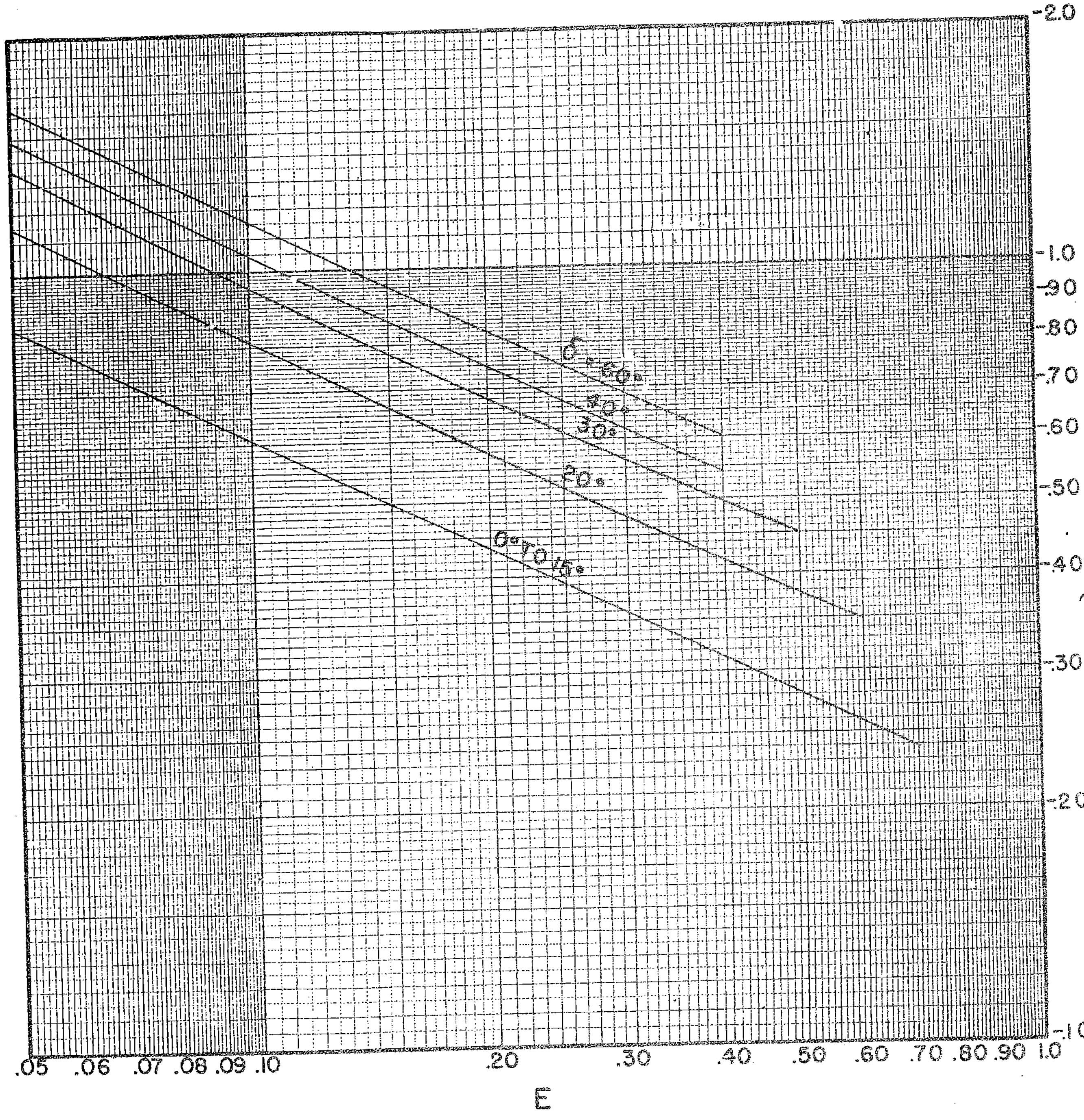


FIGURE 3-4

$\eta_{b\delta}$ VS E

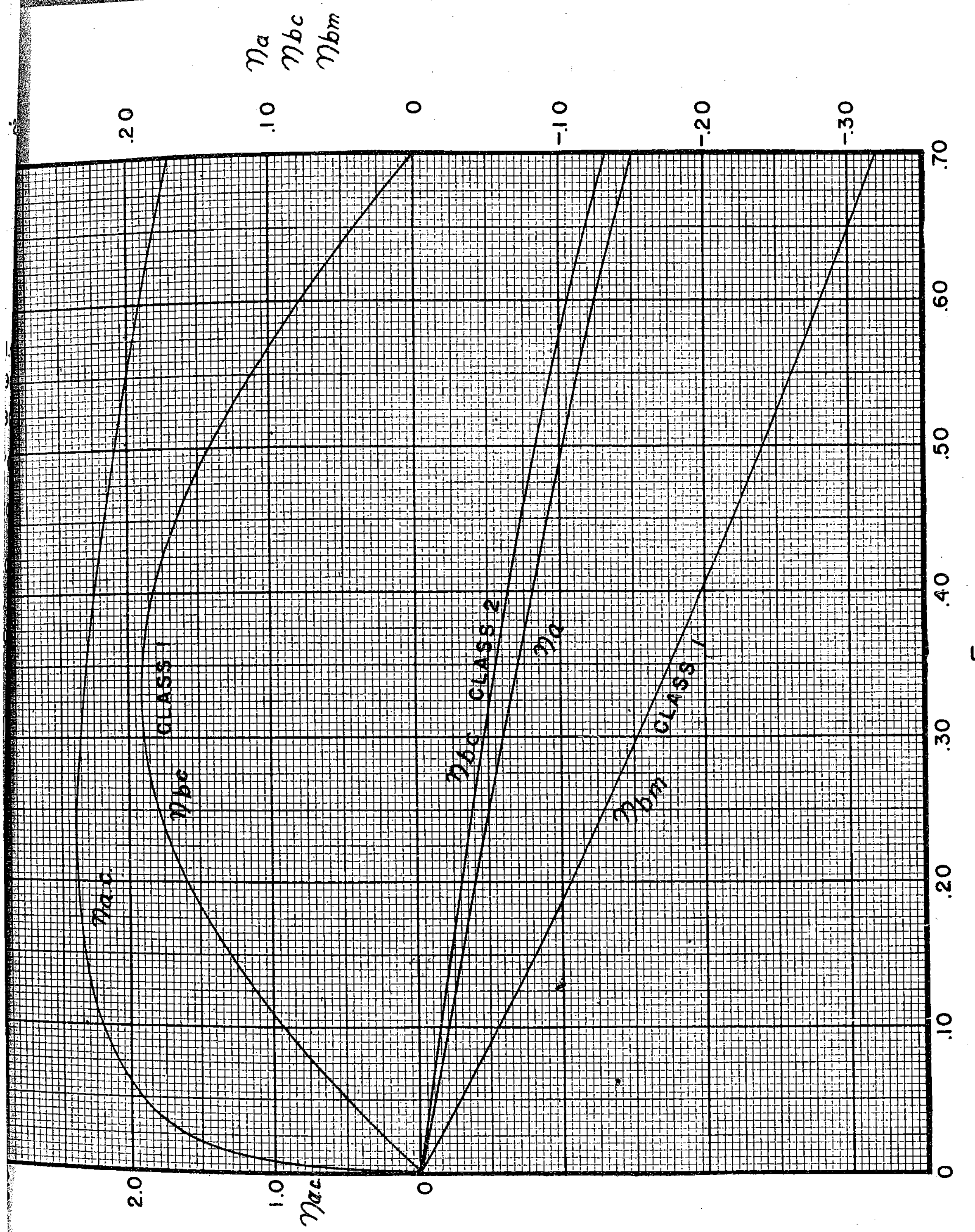


FIGURE 3-5

$\eta_{a.c.}$, η_a, η_{bc} AND η_{bm} VS E

$$c_n' = c_n + E\, c_{n_f} (1 - \cos\delta) \qquad (3.7)$$

As expressed above Equation (3.6) is applicable to the calculation of the flap normal-force coefficient for the portion of the flap aft of the hinge for NACA classified airfoil section only. For unclassified sections the following correlating relationships are employed for adaption of this equation to use.

$\gamma_{bc}\ c_{n_{bc}} + \gamma_{bm}\ c_{n_{bm}}$ = the integral, over the flap chord, of the basic pressure distribution for unflapped section divided by E, or:

$$\frac{1}{E} \int_{x = 1-E}^{x = 1} P_b \, dx$$

$\gamma_a + \gamma_{ac}\ x_{ac}$ = the integral, over the flap chord, of the additional pressure distribution for unflapped section corresponding to $c_{n_a} = 1.0$ divided by E, or:

$$\frac{1}{E} \int_{x = 1-E}^{x = 1} P_{a_1} \, dx$$

These data are obtained either analytically or graphically from the results of calculations made according to the methods of sections 1.1 and 1.2.

3.2 ZERO LIFT DISTRIBUTION

$P_{b\delta}$ represents the component of the basic normal pressure coefficient due to flap deflection and equals:

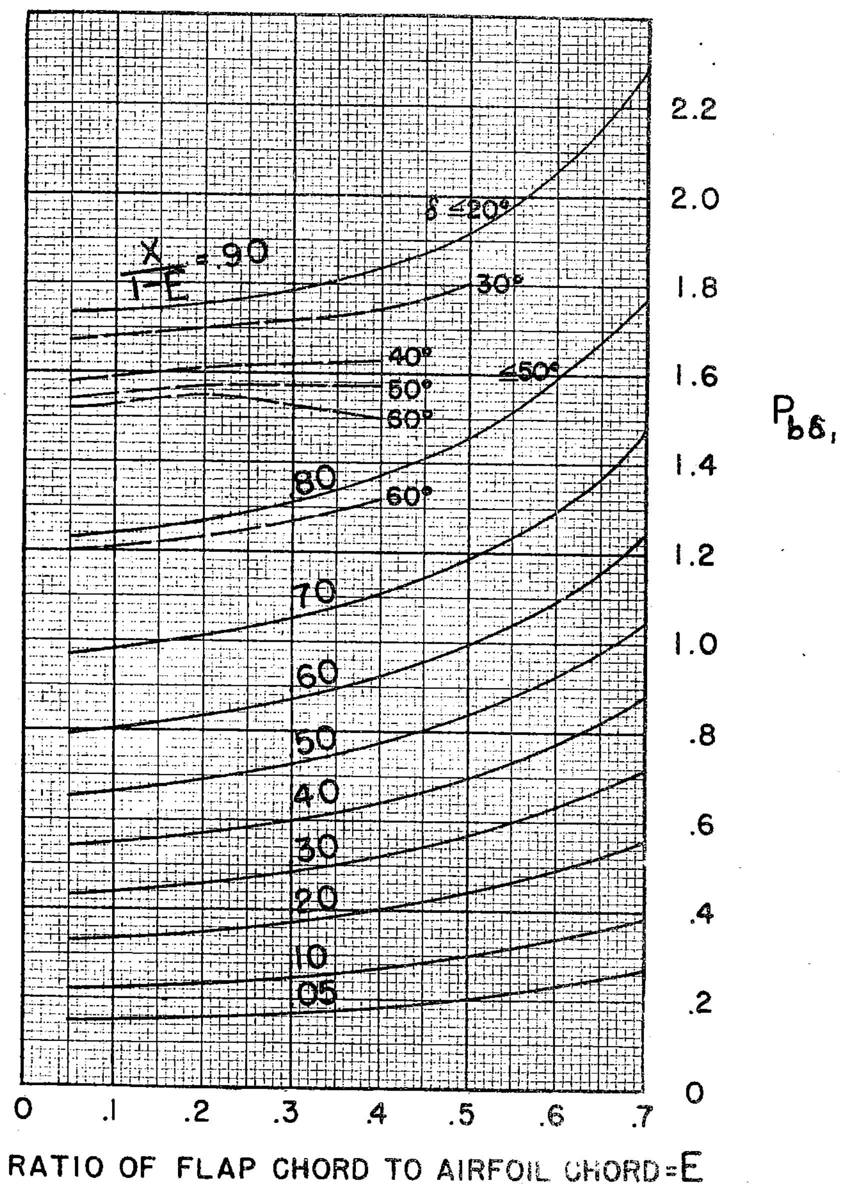


FIG. 3-6

VALUES OF $P_{b\delta_1}$, FORWARD OF FLAP HINGE FOR ALL FLAP ANGLES.

MT.-1

$$P_{b\delta} = c_{nb\delta} P_{b\delta_1} \qquad (3.8)$$

where $P_{b\delta_1}$ represents the component of basic normal-pressure coefficient for $c_{nb\delta} = 1.0$. This distribution for points forward of the hinge, presented in Figure 3-6, is independent of flap angle, except at points close to the hinge. For points aft of the hinge, this distribution is for all practical purposes independent of flap angle when the flap angle is not greater than about 15 degrees. These data are presented in Figure 3-7. The variation of this distribution with flap angle, for large flap angles, is presented in Figures 3-8 and 3-9.

The component of the zero lift distribution (P_o) due to camber and moment of the unflapped section, is obtained by equation 1.2. The component of the zero lift distribution ($P_{o\delta}$), due to deflection of flap from neutral position, may be obtained at selected chord stations from

$$P_{o\delta} = P_{b\delta} - c_{nb\delta} P_{a_1} \qquad (3.9)$$

The resultant pressure coefficient at the zero lift attitude for the airfoil-flap combination corresponding to a given flap deflection is then the sum of $P_o + P_{o\delta}$. It is usually satisfactory to assume that Δc_m corresponding to a given flap deflection does not vary with total normal-force coefficient (c_n), and that when $P_{o\delta}$

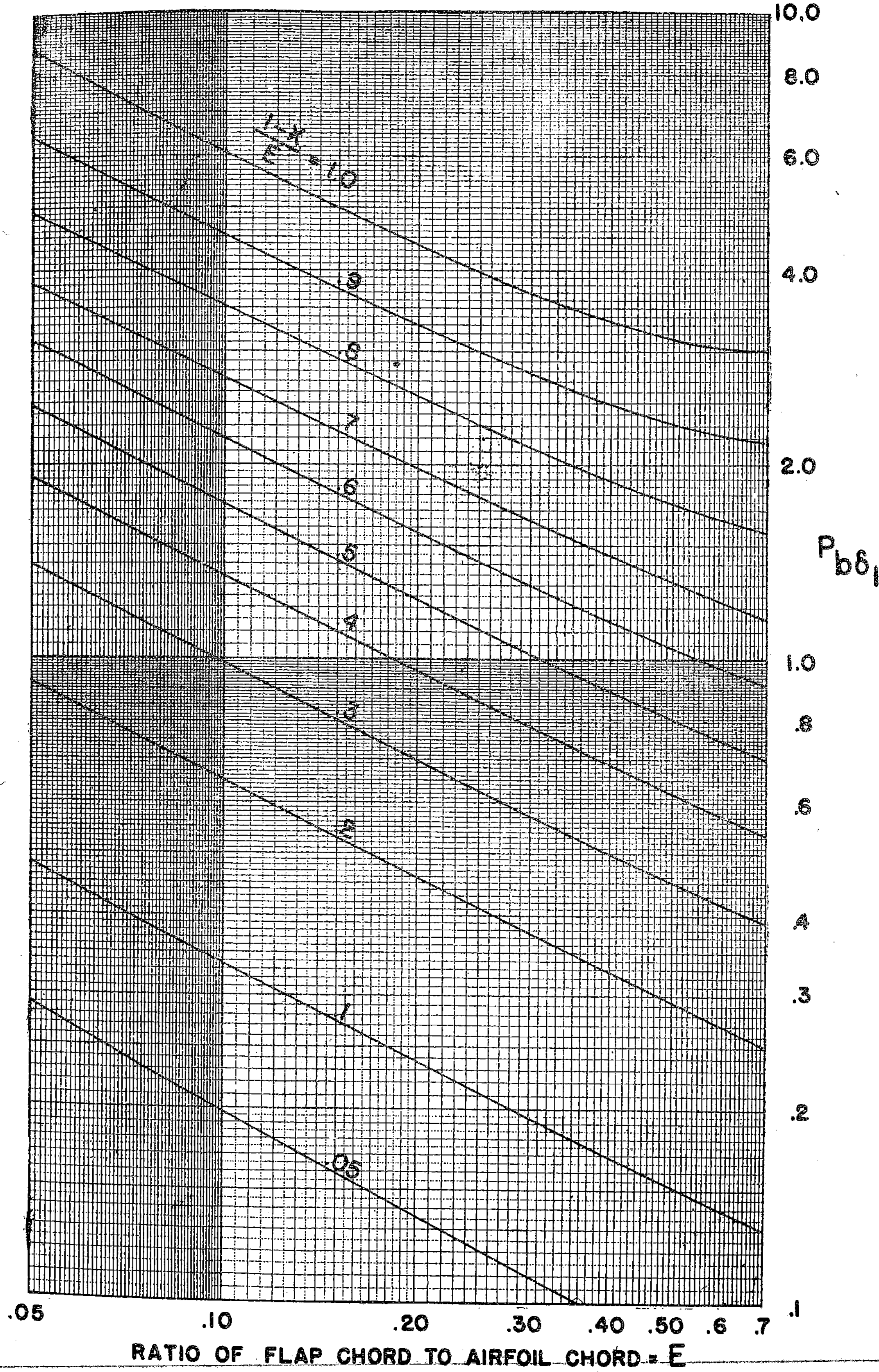


FIG. 3-7

VALUES OF $P_{b\delta_1}$, AFT OF FLAP HINGE FOR SMALL FLAP ANGLES.

is once calculated for a given flap deflection and normal force coefficient, it may be used to compute the P distribution for the same flap angle and any other value of c_n. When Δc_m does vary with c_n, and greater precision is desired, either the P distribution may be calculated as outlined in detail above for each desired value of c_n, or the Δc_m vs. c_n curve may be approximated by a straight line and $P_{o\delta}$ expressed as two partial distributions determined for $c_n = 0$ and $c_n = 1.0$.

3.3 DISTRIBUTION COMPONENTS

The distribution of total normal-pressure coefficient for an airfoil with flap displaced from neutral may now be calculated as equal to:

$$P = (P_o + P_{o\delta}) + c'_n P_{a_1} \qquad (3.10)$$

where P_{a_1} is the additional normal-pressure coefficient for the unflapped section obtained by the methods of sections 1.2 or 1.33 and $(P_o + P_{o\delta})$ is the normal-pressure coefficient for the airfoil-flap combination corresponding to the given flap deflection and $c_n = 0$.

3.4 FLAP HINGE MOMENT COEFFICIENT

Expressing the hinge moment acting on a deflected plain flap with no aerodynamic balance surface as equal to:

$$c_{h_f} \; E^2 \; c^2 \; q$$

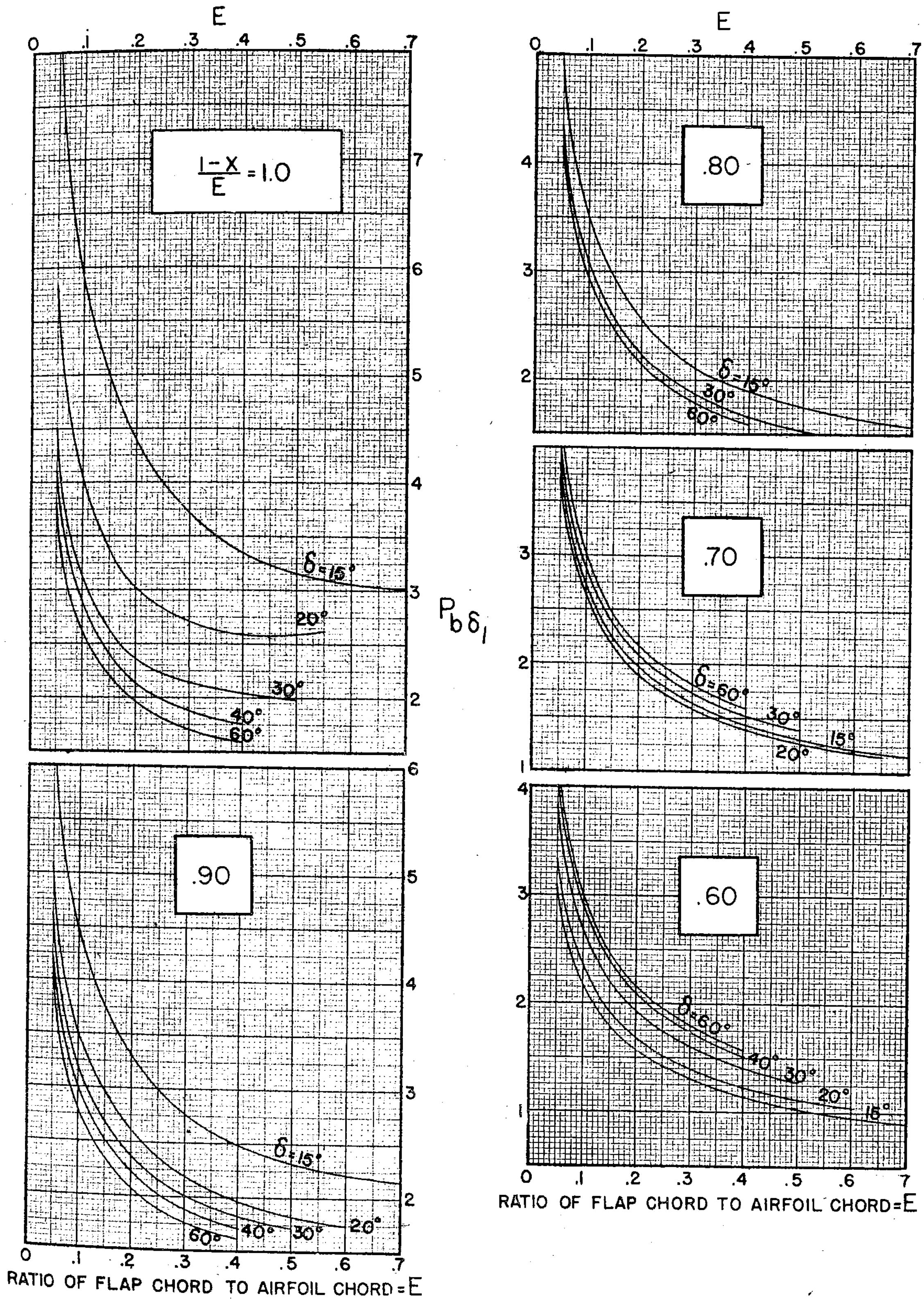


FIG. 3-8

VALUES OF $P_{b\delta_1}$, AFT OF FLAP HINGE FOR VARIOUS FLAP ANGLES.

The flap hinge-moment coefficient may be expressed as:

$$c_{h_f} = \eta_{b\delta}\ c_{n_{b\delta}} + \eta_{bc}\ c_{n_{bc}} + \eta_{bm}\ c_{n_{bm}} + (\eta_a + \eta_{ac}\ x_{ac})\ c_{n_a} \qquad (3.11)$$

The various η values are presented in Figures 3-4 and 3-5. As expressed above, equation (3.11) is applicable to the calculation of the flap hinge moment coefficient for NACA classified airfoil sections only. For unclassified sections, the following correlating relationships are employed for the adaption of this equation to use.

$$\eta_{bc}\ c_{n_{bc}} + \eta_{bm}\ c_{n_{bm}} = \frac{1}{E^2} \int_{x=1-E}^{x=1} P_b\ (x-E)\ dx$$

$$\eta_a + \eta_{ac}\ x_{ac} = \frac{1}{E^2} \int_{x=1-E}^{x=1} P_{a_1}\ (x-E)\ dx$$

These expressions are analogous to those used under paragraph 3.1 pertaining to normal-force coefficients.

The precise value of the hinge-moment coefficient for a split flap cannot be obtained in this way since the distribution of pressure difference across the flap, and therefore c_{h_f} and c_{n_f}, depends upon the surface pressure on the outer flap surface and the pressure between the flap and airfoil which pressure is uniform and approximately independent of airfoil angle of attack

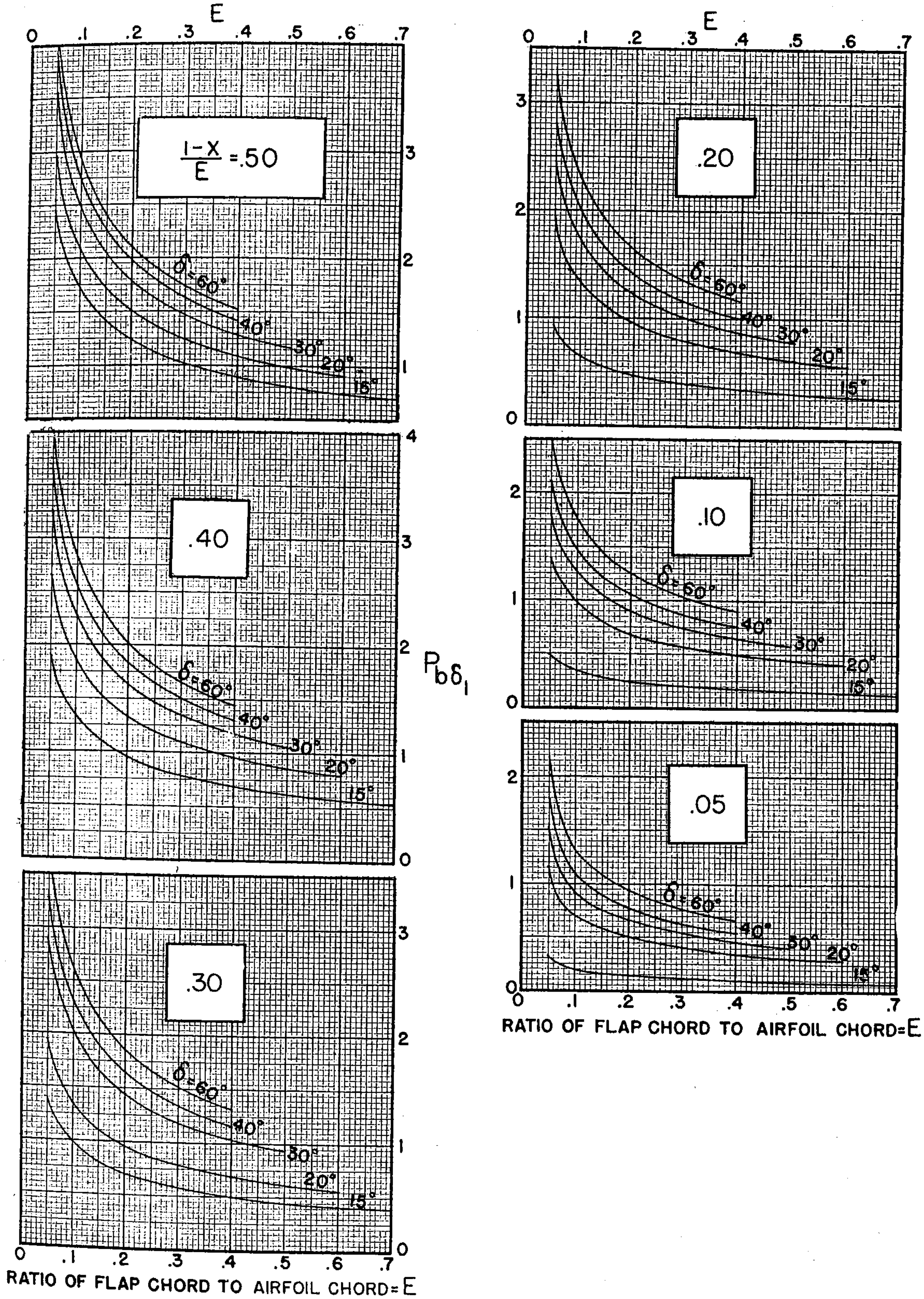


FIG. 3-9

VALUES OF $P_{b\delta_1}$, AFT OF FLAP HINGE FOR VARIOUS FLAP ANGLES.

for any given flap angle. (See section 2.4) For any type of flap, integration of the distribution of total calculated pressure difference is preferred and will give greater accuracy than can be obtained by equation 3.11. Such an integration procedure is essential to obtaining valid total forces and hinge-moments for a flap having any aerodynamic balance surface.

3.5 SERIALLY HINGED FLAPS

The distribution of normal pressure coefficient for an airfoil with a serially hinged plain flap may be obtained as follows:

(a) Obtain P_o and P_{a_1} for the normal section or for the section with all flaps in neutral position.

(b) Obtain a $P_{o\delta}$ distribution for each flap displaced the required angle, but with the other flaps in neutral, in the same manner as illustrated by table C-4.

(c) Add these various $P_{o\delta}$ distributions to the P_o distribution to obtain the zero lift distribution for the airfoil flap combination.

(d) Obtain P in the usual way as in table C-3 using the distribution obtained in (c) above for the zero lift distribution.

(e) Integrate the normal-force distribution curve to determine the several flap normal-force and hinge moment coefficients.

Experiments with airfoils having combinations of two serially hinged flaps simulating a flap and tab combination (i.e. the aft flap having a chord small in comparison with that of the forward flap) show that this method is applicable for conditions in which the flap and tab are each unstalled both when their deflections are applied separately or simultaneously. Deviations of observed experimental pressures from those calculated by the above method occur where the respective stalled or unstalled condition of the flap or the tab deflected alone is changed by the simultaneous deflection of tab and flap. Examples of such conditions are (1) the changing of the stalled flow over a down-deflected flap to unstalled flow by the opposite (up) delfection of the tab, and (2) the stalling of the flow over an unstalled tab by an excessive deflection of the flap in the same direction.

3.6 SPECIAL TYPE FLAPS

As stated above, the foregoing methods of calculating the pressure distribution over airfoil-flap combinations is limited in applicability to the plain or split types. Frequently, however, other types which are not covered by the above descriptions are employed in design. These, may include types which extend the wing chord thus increasing the area, types incorporating single or multiple slots, spoilers, speed reduction devices, etc. In such cases pressure distribution calculations for structural design purposes should be supported by aerodynamic test data.

APPENDIX A

DETERMINATION OF THE PRESSURE DISTRIBUTION OVER AN UNCLASSIFIED AIRFOIL SECTION AT THE BASIC ANGLE OF ATTACK

A.0. DETERMINATION OF B-COEFFICIENTS.

The Fourier B-coefficients for the curve of the mean camber line are determined from the known ordinates of the airfoil section. The x axis is the line drawn thru the leading and trailing edges of the mean camber line and z is measured normal to this axis. The origin of polar coordinates is taken at $x = .5$ so that equations (1.4) and (1.5) of Chapter I apply. The values of the z coordinates at the leading and trailing edges and the 11 specified intermediate chord stations are entered in Table A-1. By following the calculation procedure indicated on this table the values of B_o thru B_{12} are determined.

TABLE A-1

DETERMINATION OF B COEFFICIENTS

1	θ	0°	15°	30°	45°	60°	75°	90°	105°	120°	135°	150°	165°	180°
2	$X = \frac{1-\cos\theta}{2}$	.00000	.01703	.06698	.14644	.25000	.37059	.50000	.62941	.75000	.85355	.93301	.98296	1.00000
3	Z	Z_0	Z_1	Z_2	Z_3	Z_4	Z_5	Z_6	Z_7	Z_8	Z_9	Z_{10}	Z_{11}	Z_{12}
4	LINE 3		Z_1	Z_2	Z_3	Z_4	Z_5	Z_6			V_1	V_2	V_3	
5	LINE 3		Z_{11}	Z_{10}	Z_9	Z_8	Z_7			V_6	V_5	V_4		
6	4 + 5		V_1	V_2	V_3	V_4	V_5	V_6		P_0	P_1	P_2	P_3	
7	4 - 5		W_1	W_2	W_3	W_4	W_5	W_6		Q_0	Q_1	Q_2	Q_3	
8	MULTIPLIER	0	12	1 &	11	2 &	10	3	9	4	8	5 &	7	6
				a	b	a	b					a	b	
9	.25882				W_5								W_1	
10	.50000			W_4		Q_2				$P_1 - P_2$	$-P_1 - P_2$	W_4		
11	.70711				W_3			$W_1 - W_3 - W_5$	$-W_1 + W_3 + W_5$				$-W_3$	
12	.86603			W_2			Q_1					$-W_2$		
13	.96593				W_1								W_5	
14	1.00000	$P_0+P_1+P_2+P_3$	$P_0-P_1+P_2-P_3$			Q_0		$-W_4$	$-W_4$	$P_0 - P_3$	$P_0 + P_3$			$Q_0 - Q_2$
15	TOTAL													
16	a + b													
17	a - b													
18	DIVIDE BY 6 *													
19	B_N	B_0	B_{12}	B_1	B_{11}	B_2	B_{10}	B_3	B_9	B_4	B_8	B_5	B_7	B_6

* DIVIDE BY 12 FOR B_0 AND B_{12}

CHECK: $\Sigma B_N = 0$

ADMT.-1

EXPLANATION OF TABLE

(1) Line 1 gives the angles along the base line.
(2) Line 2 gives the x values along the base line.
(3) Line 3 gives the z values corresponding to the x values.
(4) Lines 4, 5, 6, and 7 are self-explanatory.
(5) Line 8 contains column headings corresponding to the B-coefficients to be obtained in the respective columns below. The first column contains multipliers by which all values are multiplied before entering in the columns to the right.
(6) Next fill in the spaces to right of the columns of multipliers which contain W's, P's, and Q's as follows:
   (a) For line 10, multiply .50000 by the values given in lines 6 and 7 for W_4, Q_2, $P_1 - P_2$, $-P_1 - P_2$, and W_4 and write the products in the respective spaces as indicated, being careful to use the proper signs.
   (b) Fill in lines 9, 11, 12, 13, and 14 in the same manner, using the correct multiplier given at the left for each line.
(7) Total the columns, and write the results in line 15.
(8) Obtain the sums of the quantities in line 15 for each of the three pairs of columns (a) and (b) and write the sums in line 16 in the respective columns (a).
(9) Subtract each column (b) from its preceding column (a) in line 15 and write the results in line 17 under the respective columns (b).
(10) Divide these values in lines 16 and 17, and all values in line 15 in those columns not headed by an (a) or (b), by 6 and write the results in line 18 except the values in line 15 columns (3) and (4) which should be divided by 12. The values in line 18 are the B-coefficients corresponding to the symbols in line 19.

ANC-1(2)
ADMT.-1
3-JAN-44

TABLE A-1a

DETERMINATION OF B COEFFICIENTS

ADMT.-1

1	θ	0°	15°	30°	45°	60°	75°	90°	105°	120°	135°	150°	165°	180°
2	$X = \frac{1-\cos\theta}{2}$	.00000	.01703	.06698	.14644	.25000	.37059	.50000	.62941	.75000	.85355	.93301	.98296	1.00000
3	Z	Z_0 .00000	Z_1 .00333	Z_2 .01227	Z_3 .02393	Z_4 .03438	Z_5 .03978	Z_6 .03889	Z_7 .03415	Z_8 .02639	Z_9 .01714	Z_{10} .00843	Z_{11} .00224	Z_{12} .00000
4	LINE 3		Z_1 .00333	Z_2 .01227	Z_3 .02393	Z_4 .03438	Z_5 .03978	Z_6 .03889			V_1 .00557	V_2 .02070	V_3 .04107	
5			Z_{11} .00224	Z_{10} .00843	Z_9 .01714	Z_8 .02639	Z_7 .03415			V_6 .03889	V_5 .07393	V_4 .06077		
6	4+5		V_1 .00557	V_2 .02070	V_3 .04107	V_4 .06077	V_5 .07393	V_6 .03889		P_0 .03889	P_1 .07950	P_2 .08147	P_3 .04107	
7	4−5		W_1 .00109	W_2 .00384	W_3 .00679	W_4 .00799	W_5 .00563	W_6 .03889		Q_0 -.03889	Q_1 -.06836	Q_2 -.04007	Q_3 .04107	
8	MULTIPLIER	0	12	1 & 11 a	b	2 & 10 a	b	3	9	4	8	5 & 7 a	b	6
9	.25882				W_5 .00146								W_1 .00028	
10	.50000			W_4 .00400		Q_2 -.02004				$P_1 - P_2$ -.00099	$-P_1 - P_2$ -.08049	W_4 .00400		
11	.70711				W_3 .00480			$W_1 - W_3 - W_5$ -.00801	$-W_1 + W_3 + W_5$ .00801				$-W_3$ -.00480	
12	.86603			W_2 .00333			Q_1 -.05920					$-W_2$ -.00333		
13	.96593				W_1 .00105								W_5 .00540	
14	1.00000	$P_0+P_1+P_2+P_3$ .24093	$P_0-P_1+P_2-P_3$ -.00021			Q_0 -.03889		$-W_4$ -.00799	$-W_4$ -.00799	$P_0 - P_3$ -.00218	$P_0 + P_3$ .07996			$Q_0 - Q_2$ .00118
15	TOTAL	.24093	-.00021	.00733	+.00731	-.05893	-.05920	-.01600	.00002	-.00317	-.00053	.00067	.00088	.00118
16	a + b			.01464		-.11813						.00159		
17	a − b				.00002		.00027						-.00025	
18	DIVIDE BY 6 *	.02008	-.00002	.00244	.00000	-.01969	.00005	-.00267	.00000	-.00053	-.00009	.00027	-.00004	.00020
19	B_N	B_0	B_{12}	B_1	B_{11}	B_2	B_{10}	B_3	B_9	B_4	B_8	B_5	B_7	B_6

* DIVIDE BY 12 FOR B_0 AND B_{12}

CHECK: $\Sigma B_N = 0$

EXPLANATION OF TABLE

(1) Line 1 gives the angles along the base line.
(2) Line 2 gives the x values along the base line.
(3) Line 3 gives the z values corresponding to the x values.
(4) Lines 4, 5, 6, and 7 are self-explanatory.
(5) Line 8 contains column headings corresponding to the B-coefficients to be obtained in the respective columns below. The first column contains multipliers by which all values are multiplied before entering in the columns to the right.
(6) Next fill in the spaces to right of the columns of multipliers which contain W's, P's, and Q's as follows:
  (a) For line 10, multiply .50000 by the values given in lines 6 and 7 for W_4, Q_2, $P_1 - P_2$, $-P_1 - P_2$, and W_4 and write the products in the respective spaces as indicated; being careful to use the proper signs.
  (b) Fill in lines 9, 11, 12, 13, and 14 in the same manner, using the correct multiplier given at the left for each line.
(7) Total the columns, and write the results in line 15.
(8) Obtain the sums of the quantities in line 15 for each of the three pairs of columns (a) and (b) and write the sums in line 16 in the respective columns (a).
(9) Subtract each column (b) from its preceding column (a) in line 15 and write the results in line 17 under the respective columns (b).
(10) Divide these values in lines 16 and 17, and all values in line 15 in those columns not headed by an (a) or (b), by 6 and write the results in line 18 except the values in line 15 columns (3) and (4) which should be divided by 12. The values in line 18 are the B-coefficients corresponding to the symbols in line 19.

A.1 CALCULATION OF PRESSURE DISTRIBUTION

As described previously in the bulletin, the B-coefficients must be modified before being employed in the pressure calculations so that the resulting distribution will produce a normal force curve, moment coefficient, and aerodynamic center location corresponding to observed test data. Table A-2 represents a calculation scheme whereby the pressure distribution may be determined.

The method of this appendix is applied to the calculations of the P_b distribution for an actual airfoil as shown in Tables A-1a and A-2a.

TABLE A-2
CALCULATION OF P_b DISTRIBUTION

__________ AIRFOIL

ROW	COLUMN			(0)	(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)	(15)	(16)	(17)	(18)	(19)
0			n	0	1	2	3	4	5	6	7	8	9	10	11	12	Σ (0) THRU (12)	$\frac{8}{\sin\theta}$ / (13) x $\frac{8}{\sin\theta}$	$K_1 \frac{8}{\sin\theta}$	$\cot\theta$ / $K_2 \cot\theta$	(14)+(15)+(16) = P_b	MULTIPLIER (17) x MULT'R	MULTIPLIER (17) x MULT'R
1			B_n																				
2			$\frac{B_n}{B_2}$			1.0																	
3			$\frac{nB_n}{B_2}$			2.0											$P_1(B)$						
4			$\frac{B_n}{B_3}$				1.0																
5			$\frac{nB_n}{B_3}$				3.0										$P_2(B)$						
6																							
7	θ°	X	B'_n																				
8	0	0			1.0	2.0	3.0	4.0	5.0	6.0	7.0	8.0	9.0	10.0	11.0	12.0		$\frac{8}{0}$		$\frac{1}{0}$		0	0
9	15	.01703			.96593	1.73206	2.12133	2.0	1.29410	0	-1.81174	-4.0	-6.36399	-8.6603	-10.6252	-12.0		30.90951		3.7321		1.03528	1.0
10	30	.06698			.86603	1.0	0	-2.0	-4.33015	-6.0	-6.06221	-4.0	0	5.0000	9.52633	12.0		16.0		1.7321		1.0	.86603
11	45	.14644			.70711	0	-2.12133	-4.0	-3.53555	0	4.94977	8.0	6.36399	0	-7.77821	-12.0		11.31366		1.0		2.82844	2.0
12	60	.25			.50000	-1.0	-3.0	-2.0	2.5	6.0	3.5	-4.0	-9.0	-5.0	5.5	12.0		9.23756		.57735		1.73206	.86603
13	75	.37059	n cos nθ / B'_n x n cos nθ		.25882	-1.7321	-2.12133	2.0	4.82965	0	-6.76151	-4.0	6.36399	8.6603	-2.84702	-12.0		8.28217		.26795		3.86372	1.0
14	90	.50			0	-2.0	0	4.0	0	-6.0	0	8.0	0	-10.0	0	12.0		8.0		0		2.0	0
15	105	.62941			-.25882	-1.73206	2.12133	2.0	-4.82965	0	6.76151	-4.0	-6.36399	8.6603	2.84702	-12.0		8.28217		-.26795		3.86372	-1.0
16	120	.75			-.50000	-1.0	3.0	-2.0	-2.5	6.0	-3.5	-4.0	9.0	-5.0000	-5.5	12.0		9.23756		-.57735		1.73206	-.86603
17	135	.85355			-.70711	0	2.12133	-4.0	3.53555	0	-4.94977	8.0	-6.36399	0	7.77821	-12.0		11.31366		-1.0		2.82844	-2.0
18	150	.93301			-.86603	1.0	0	-2.0	4.33015	-6.0	6.06221	-4.0	0	5.0	-9.52633	12.0		16.0		-1.7321		1.0	-.86603
19	165	.98296			-.96593	1.73206	-2.12133	2.0	-1.29410	0	1.81174	-4.0	6.36399	-8.6603	10.62523	-12.0		30.90951		-3.7321		1.03528	-1.0
20	180	1.0			-1.0	2.0	-3.0	4.0	-5.0	6.0	-7.0	8.0	-9.0	10.0	-11.0	12.0		$\frac{8}{0}$		$-\frac{1}{0}$		0	0
																						$= \Sigma_1$	$= \Sigma_2$

CALCULATIONS

$\alpha_o = -2\left[B'_1 - B_2\{P_1(B)\} + B'_3\{P_2(B)\}\right]$

$(C_m)_{ac} = \pi\left[4B_2\{P_1(B)\}X_{ac} + B_2\{P_1(B)\} - B'_3\{P_2(B)\}\right]$

SOLUTION $B'_1 =$

$B'_3 =$

$B'_n = \frac{B_n}{B_3} \times B'_3$, WHERE n IS ODD NUMBER GREATER THAN 1.

$B'_n = B_n$, WHERE n IS EVEN NUMBER.

$K_1 = -B_2\{P_1(B)\} =$

$K_2 = -8B'_1 - 8B'_3\{P_2(B)\} =$

AIRFOIL TEST DATA

$\alpha_o =$

$(C_m)_{ac} =$

$X_{ac} =$

CHECKS

1. IN COL. (17), NUMERATOR OF ROW 8 = 0, NUMERATOR OF ROW 20 = 0

2. $C_{nb} = \frac{\pi}{72}\Sigma_1 = -4\pi B_2 P_1(B)$

3. $(C_m)_{ac} = -\frac{\pi}{144}\left[\Sigma_1 - \Sigma_2\right] + C_{nb}\left(\frac{1}{4} - X_{ac}\right)$

EXPLANATION OF TABLE A-2

(1) Enter in row 1, columns 0 through 12 inclusive, the values of the B-coefficients as determined from table A-1.

(2) Divide each even numbered B-coefficient by the value of B_2 and enter the quotients in row 2.

(3) Multiply each quotient in row 2 by the value of n and enter product in row 3. Enter the sum of these products, columns 2 through 12 inclusive (as indicated), in row 3, column 13. This sum is designated as $P_1(B)$.

(4) Divide each odd numbered B-coefficient by the value of B_3 and enter the quotients in row 4.

(5) Multiply each quotient in row 4 by the value of n and enter product in row 5. Enter the sum of these products, columns 3 through 11 inclusive (as indicated), in row 5, column 13. This sum is designated as $P_2(B)$.

(6) Enter the applicable values of α_o, $(c_m)_{ac}$, and x_{ac} under "Airfoil Test Data," and, using in the equations expressed at the left, calculate B_1' and B_3'.

(7) Enter B_1' and B_3' in row 7. Also, enter the values of the even numbered unmodified B-coefficients from row 1 into row 7. Calculate the values of the odd numbered modified B-coefficients as the product of B_3' times the quotient appearing in row 4. Enter the results of this calculation in row 7 to complete the modified B-coefficients, columns 1 through 12 inclusive.

(8) Each space in rows 8 through 20, columns 1 through 12 inclusive, is divided into an upper and lower space. The upper space contains the value of $n \cos n\theta$. In each lower space should be entered the product of the modified B-coefficient times the value appearing in the pertinent upper space (i.e. $B_n' \, n \cos n\theta$).

(9) In column 13, enter the sum of the products $B_n' \, n \cos n\theta$, appearing in columns 1 through 12 inclusive.

(10) In the upper half of each space in column 14 is entered the value of $\dfrac{8}{\sin\theta}$. In the lower half of each such space should be entered the product of the value in column 13 times the value in the upper half.

(11) From the values of $P_1(B)$, $P_2(B)$, B_1, B_2, and B_3 calculate the values of K_1 and K_2 as defined by the relation at the lower left.

(12) Multiply the value of $\frac{8}{\sin \theta}$ appearing in the upper half of each space in column 14 by K_1 and enter the product in column 15.

(13) In the upper half of each space in column 16 is entered the value of cot θ. Multiply this value by K_2 and enter the product in the lower half of the space.

(14) The basic pressure coefficient, P_b, is equal to the sum of columns 14, 15, and 16 and is entered in column 17.

(15) Multiply the value of the multiplier entered in the upper half of the space in column 18 by the value in column 17 and enter the product in the lower space. The sum of the values entered in column 18, rows 8 through 20 inclusive is equal to Σ_1.

(16) Multiply the value of the multiplier entered in the upper half of the space in column 19 by the value in column 17 and enter the product in the lower space. The sum of the values entered in column 19, rows 8 through 20 inclusive is equal to Σ_2.

(17) From the values calculated as described above, execute the checking procedures indicated at the lower right.

EXAMPLE

TABLE A-2a

CALCULATION OF P_b DISTRIBUTION

UNCLASSIFIED AIRFOIL

ROW	COLUMN			(0)	(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)	(14)	(15)	(16)	(17)	(18)	(19)
0			n	0	1	2	3	4	5	6	7	8	9	10	11	12	Σ (0) THRU (12)	$\frac{8}{\sin\theta}$ / (13) × $\frac{8}{\sin\theta}$	$K_1 \frac{8}{\sin\theta}$	$\cot\theta$ / $K_2 \cot\theta$	(14)+(15)+(16) = P_b	MULTIPLIER / (17) × MULT	MULTIPLIER / (17) × MULT
1			B_n		.00244	-.01969	-.00267	-.00053	.00027	.00020	-.00004	-.00009	0	.00005	0	-.00002							
2			$\frac{B_n}{B_2}$			1.0		.02692		-.01016		.00457		-.00254		.00102							
3			$\frac{nB_n}{B_2}$			2.0		.10768		-.06096		.03656		-.02540		.01224	$P_1(B)$ 2.07012						
4			$\frac{B_n}{B_3}$				1.0		-.10112		.01498		0		0								
5			$\frac{nB_n}{B_3}$				3.0		-.50560		.10486		0		0		$P_2(B)$ 2.59926						
6																							
7	θ°	x	B'_n		.00558	-.01969	-.00541	-.00053	.00055	.00020	-.00008	-.00009	0	.00005	0	-.00002							
8	0	0			1.0	2.0	3.0	4.0	5.0	6.0	7.0	8.0	9.0	10.0	11.0	12.0		8/0		1/0		0	0
					.00558	-.03938	-.01623	-.00212	.00275	.00120	-.00056	-.00072	0	.00050	0	-.00024	-.04922	-.39376/0	.32608/0	.06786/0	.00018/0	0	0
9	15	.01703			.96593	1.73206	2.12133	2.0	1.29410	0	-1.81174	-4.0	-6.36399	-8.6603	-10.6252	-12.0		30.90951		3.7321		1.03528	1.0
					.00539	-.03410	-.01148	-.00106	.00071	0	.00014	.00036	0	-.00043	0	.00024	-.04023	-1.24349	1.25987	.25326	.26964	.27915	.26964
10	30	.06698			.86603	1.0	0	-2.0	-4.33015	-6.0	-6.06221	-4.0	0	5.0000	9.52633	12.0		16.0		1.7321		1.0	.86603
					.00483	-.01969	0	.00106	-.00238	-.00120	.00049	.00036	0	.00025	0	-.00024	-.01652	-.26432	.65216	.11754	.50538	.50538	.43767
11	45	.14644			.70711	0	-2.12133	-4.0	-3.53555	0	4.94977	8.0	6.36399	0	-7.77821	-12.0		11.31366		1.0		2.82844	2.0
					.00395	0	.01148	.00212	-.00194	0	-.00042	-.00072	0	0	0	.00024	.01471	.16642	.46114	.06786	.69542	1.96695	1.39084
12	60	.25			.50000	-1.0	-3.0	-2.0	2.5	6.0	3.5	-4.0	-9.0	-5.0	5.5	12.0		9.23756		.57735		1.73206	.86603
					.00279	.01969	.01623	.00106	.00138	.00120	-.00028	.00036	0	-.00025	0	-.00024	.04194	.38742	.37652	.03918	.80312	1.39105	.69553
13	75	.37059	$n \cos n\theta$		.25882	-1.7321	-2.12133	2.0	4.82965	0	-6.76151	-4.0	6.36399	8.6603	-2.84702	-12.0		8.28217		.26795		3.86372	1.0
			$B'_n \times n \cos n\theta$		.00144	.03410	.01148	-.00106	.00265	0	.00056	.00036	0	.00043	0	.00024	.05020	.41576	.33758	.01818	.77152	2.98094	.77152
14	90	.50			0	-2.0	0	4.0	0	-6.0	0	8.0	0	-10.0	0	12.0		8.0		0		2.0	0
					0	.03938	0	-.00212	0	-.00120	0	-.00072	0	-.00050	0	-.00024	.03460	.27680	.32608	0	.60288	1.20576	0
15	105	.62941			-.25882	-1.73206	2.12133	2.0	-4.82965	0	6.76151	-4.0	-6.36399	8.6603	2.84702	-12.0		8.28217		-.26795		3.86372	-1.0
					-.00144	.03410	-.01148	-.00106	-.00265	0	-.00056	.00036	0	.00043	0	.00024	.01794	.14858	.33758	-.01818	.46798	1.80814	-.46798
16	120	.75			-.50000	-1.0	3.0	-2.0	-2.5	6.0	-3.5	-4.0	9.0	-5.0000	-5.5	12.0		9.23756		-.57735		1.73206	-.86603
					-.00279	.01969	-.01623	.00106	-.00138	.00120	.00028	.00036	0	-.00025	0	-.00024	.00170	.01570	.37652	-.03918	.35304	.61149	-.30574
17	135	.85355			-.70711	0	2.12133	-4.0	3.53555	0	-4.94977	8.0	-6.36399	0	7.77821	-12.0		11.31366		-1.0		2.82844	-2.0
					-.00395	0	-.01148	.00212	.00194	0	.00042	-.00072	0	0	0	.00024	-.01143	-.12932	.46114	-.06786	.26396	.74660	-.52792
18	150	.93301			-.86603	1.0	0	-2.0	4.33015	-6.0	6.06221	-4.0	0	5.0	-9.52633	12.0		16.0		-1.7321		1.0	-.86603
					-.00483	-.01969	0	.00106	.00238	-.00120	-.00049	.00036	0	.00025	0	-.00024	-.02240	-.35840	.65216	-.11754	.17622	.17622	-.15261
19	165	.98296			-.96593	1.73206	-2.12133	2.0	-1.29410	0	1.81174	-4.0	6.36399	-8.6603	10.62523	-12.0		30.90951		-3.7321		1.03528	-1.0
					-.00539	-.03410	.01148	-.00106	-.00071	0	-.00014	.00036	0	-.00043	0	.00024	-.02975	-.91956	1.25987	-.25326	.08705	.09012	-.08705
20	180	1.0			-1.0	2.0	-3.0	4.0	-5.0	6.0	-7.0	8.0	-9.0	10.0	-11.0	12.0		8/0		-1/0		0	0
					-.00558	-.03938	.01623	-.00212	-.00275	.00120	.00056	-.00072	0	.00050	0	-.00024	-.03230	-.25840/0	.32608/0	-.06786/0	-.00018/0	0	0
																						11.76180 = Σ	2.02390 = Σ_2

CALCULATIONS

$\alpha_0 = -2\left[B'_1 - B_2\{P_1(B)\} + B'_3\{P_2(B)\}\right]$

$(C_m)_{ac} = \pi\left[4B_2\{P_1(B)\}x_{ac} + B_2\{P_1(B)\} - B'_3\{P_2(B)\}\right]$

SOLUTION $B'_1 = .00558$

$B'_3 = -.00541$

$B'_n = \frac{B_n}{B_3} \times B'_3$, WHERE n IS ODD NUMBER GREATER THAN 1.

$B'_n = B_n$, WHERE n IS EVEN NUMBER.

$K_1 = -B_2\{P_1(B)\} = .04076$

$K_2 = -8B'_1 - 8B'_3\{P_2(B)\} = .06786$

AIRFOIL TEST DATA

$\alpha_0 = -3.7° = -\frac{3.7}{57.3} = -.06457$

$(C_m)_{ac} = -.088$

$x_{ac} = .008$

CHECKS

1. IN COL (17), NUMERATOR OF ROW 8 = 0, NUMERATOR OF ROW 20 = 0

2. $C_{nb} = \frac{\pi}{72}\Sigma_1 = -4\pi B_2 P_1(B)$

$\frac{\pi}{72} \times 11.76180 = -4\pi(-.01969)2.07012$

$.513 = .512$ O.K.

3. $(C_m)_{ac} = -\frac{\pi}{144}\left[\Sigma_1 - \Sigma_2\right] + C_{nb}\left(\frac{1}{4} - x_{ac}\right)$

$-.088 = -\frac{\pi}{144}[11.76180 - 2.02390] + .513(.250 - .008)$

$-.088 = -.212 + .124 = -.088$ O.K.

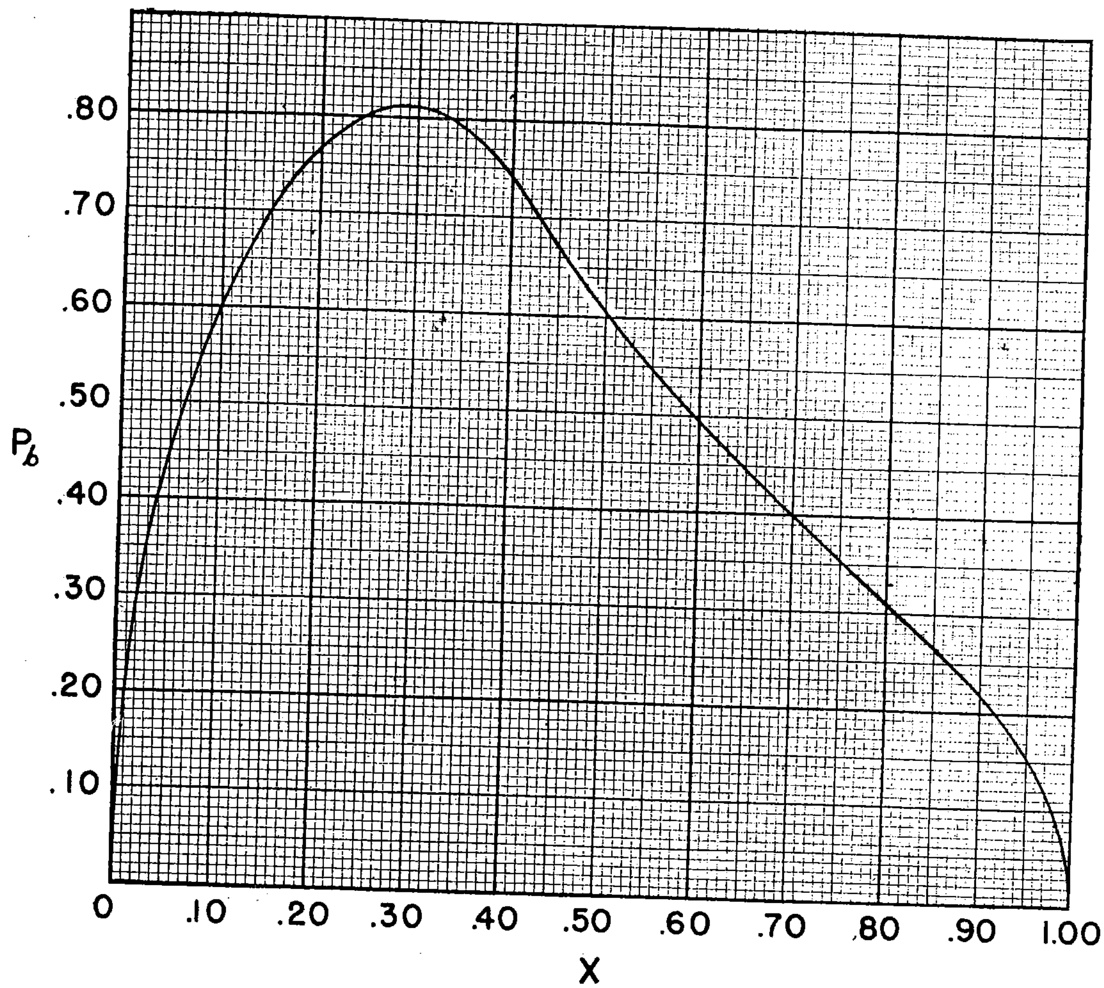


FIG. A-1

EXAMPLE: P_b DISTRIBUTION OVER AN UNCLASSIFIED AIRFOIL.

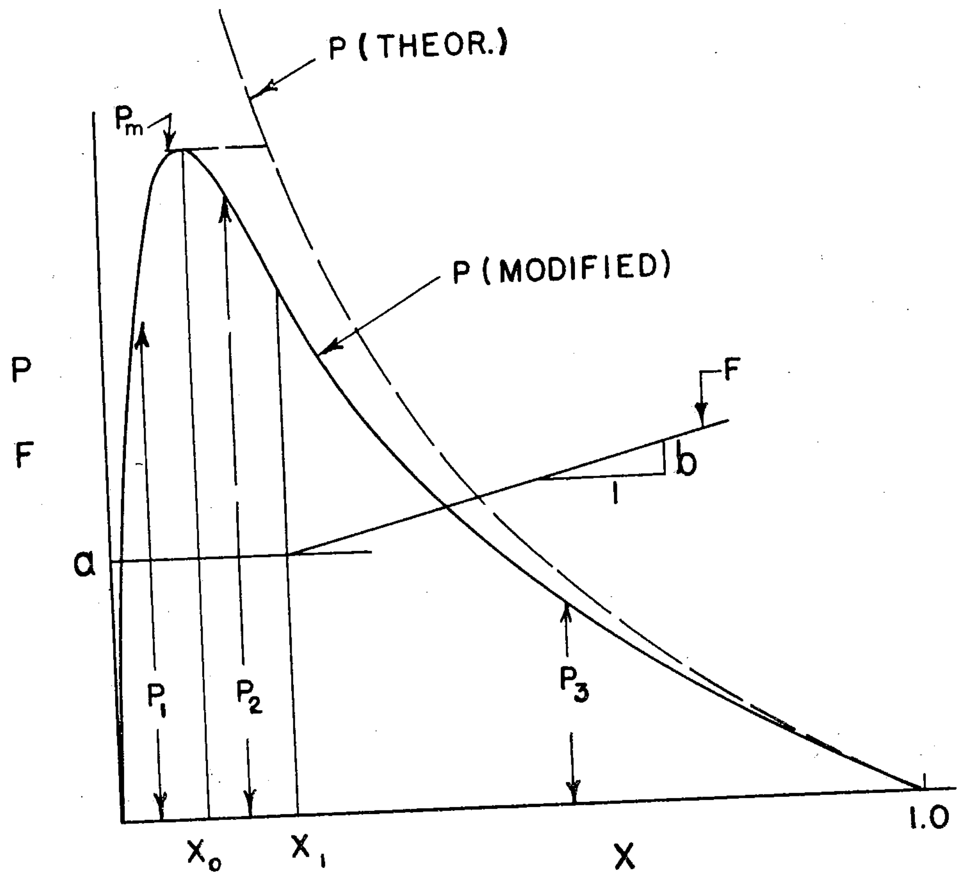
APPENDIX B

DETERMINATION OF THE ADDITIONAL PRESSURE DISTRIBUTION OVER AN UNCLASSIFIED AIRFOIL

B.0 GENERAL

The theoretical expressions of Chapter I for P and P_m are modified to produce a pressure distribution which incorporates values of P_m, x at P_m, and x_{ac} corresponding to those estimated or observed to apply to the actual airfoil in question. Figure B-1 represents the nature of the semi-empirical relationship which accomplishes this desired modification and includes an explanation of the nomenclature used.

A parabola is used between the leading edge and the station of P_m. Over the major aft portion of the airfoil the theoretical expression (i.e. $P \propto \sqrt{\frac{1-x}{x}}$) is employed as modified by a straight line multiplying factor $F= a \left[1 + b\ (x-x_1)\right]$ shown on Figure B-1. Immediately aft of x_o, and extending to tangency with the modified theoretical relationship, another parabola is used.



FOR $0 < X < X_o$ $\quad P_1 = P_m \left[1 - \left(\frac{X - X_o}{X_o}\right)^2\right]$

$X_o < X < X_1$ $\quad P_2 = P_m \left[1 - \left(\frac{X - X_o}{X_1 - X_o}\right)^2 \left(1 - a\sqrt{\frac{1 - X_1}{X_1}}\right)\right]$

$X_1 < X < 1$ $\quad P_3 = P_m \, a\left[1 + b(X - X_1)\right]\sqrt{\frac{1 - X}{X}}$

FIGURE B-1.

GENERAL EXPRESSION FOR P_a ILLUSTRATING NOMENCLATURE.

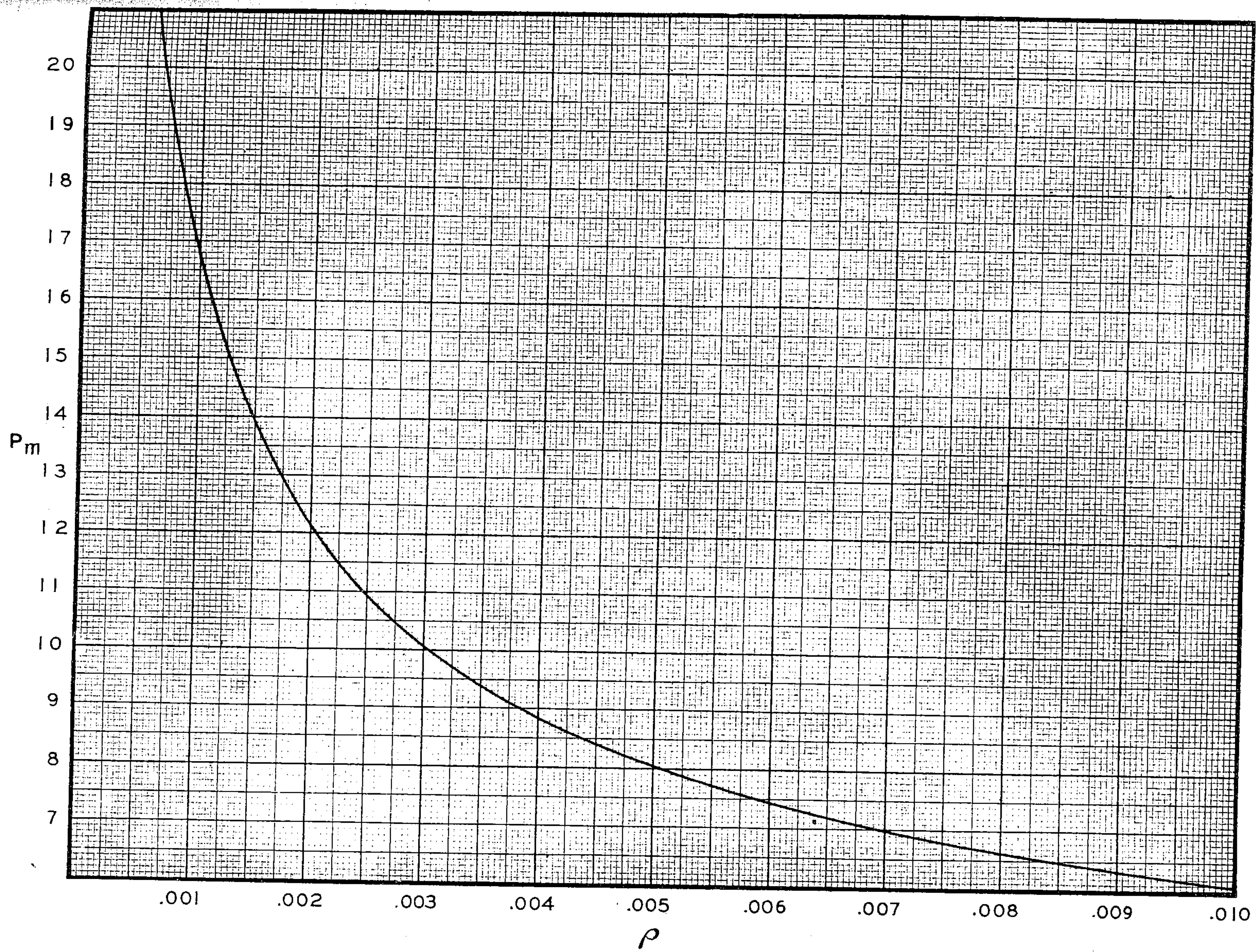


FIGURE B-2a

P_m VS ρ — FROM ρ =.001 TO ρ =.010

B.1 PROCEDURE

The applicable value of P_m may be determined from the semi-empirical function of P shown on Figures B-2a and B-2b or from observed wind tunnel data if such is available. The chordwise position of P_m, (x_o), may be taken as $x = \frac{\rho}{2}$ unless data is available substantiating another location to be applicable to the subject airfoil. In order to determine the remaining unknown variables (a, b, and the x_1 location of tangency of curves P_2 and P_3) it is necessary to use the following relations:

$$\text{at } x_o \qquad P_1 = P_2$$

$$\text{at } x_1 \qquad P_2 = P_3$$

$$\frac{dP_2}{dx} = \frac{dP_3}{dx}$$

and

$$\int_0^1 P dx = \int_0^{x_o} P_1 dx + \int_{x_o}^{x_1} P_2 dx + \int_{x_1}^{1} P_3 dx = 1$$

$$\int_0^1 P x dx = \int_0^{x_o} P_1 x dx + \int_0^{x_1} P_2 x dx + \int_{x_1}^{1} P_3 x dx = \frac{1}{4} - x_{ac}$$

Charts (Figure B-3, B-4, B-5, B-6, and B-7) have been prepared to provide a graphical solution for these variables. By entering the applicable chart (Figure

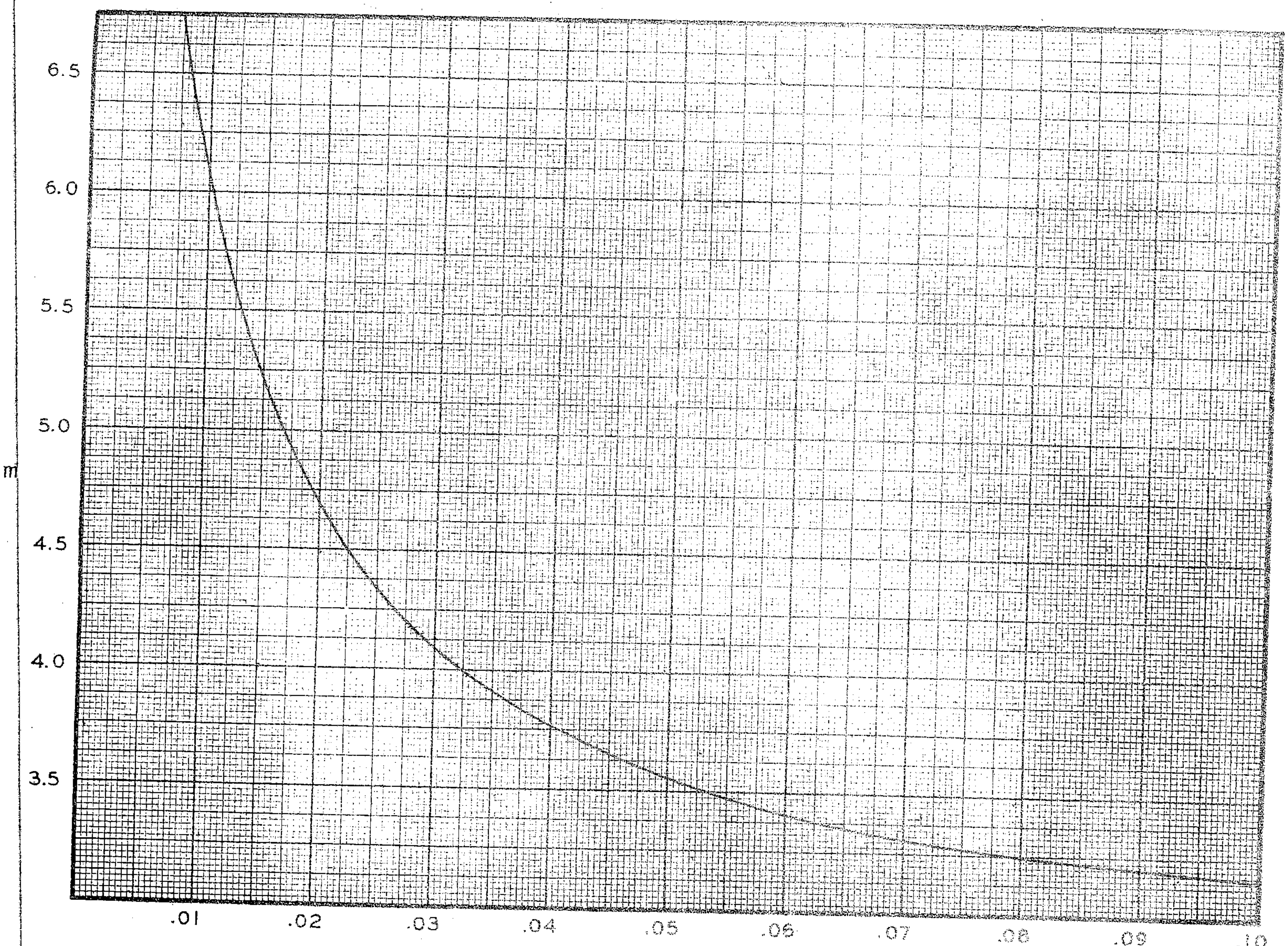


FIGURE B-2b

P_m VS ρ — FROM $\rho = .01$ TO $\rho = .10$

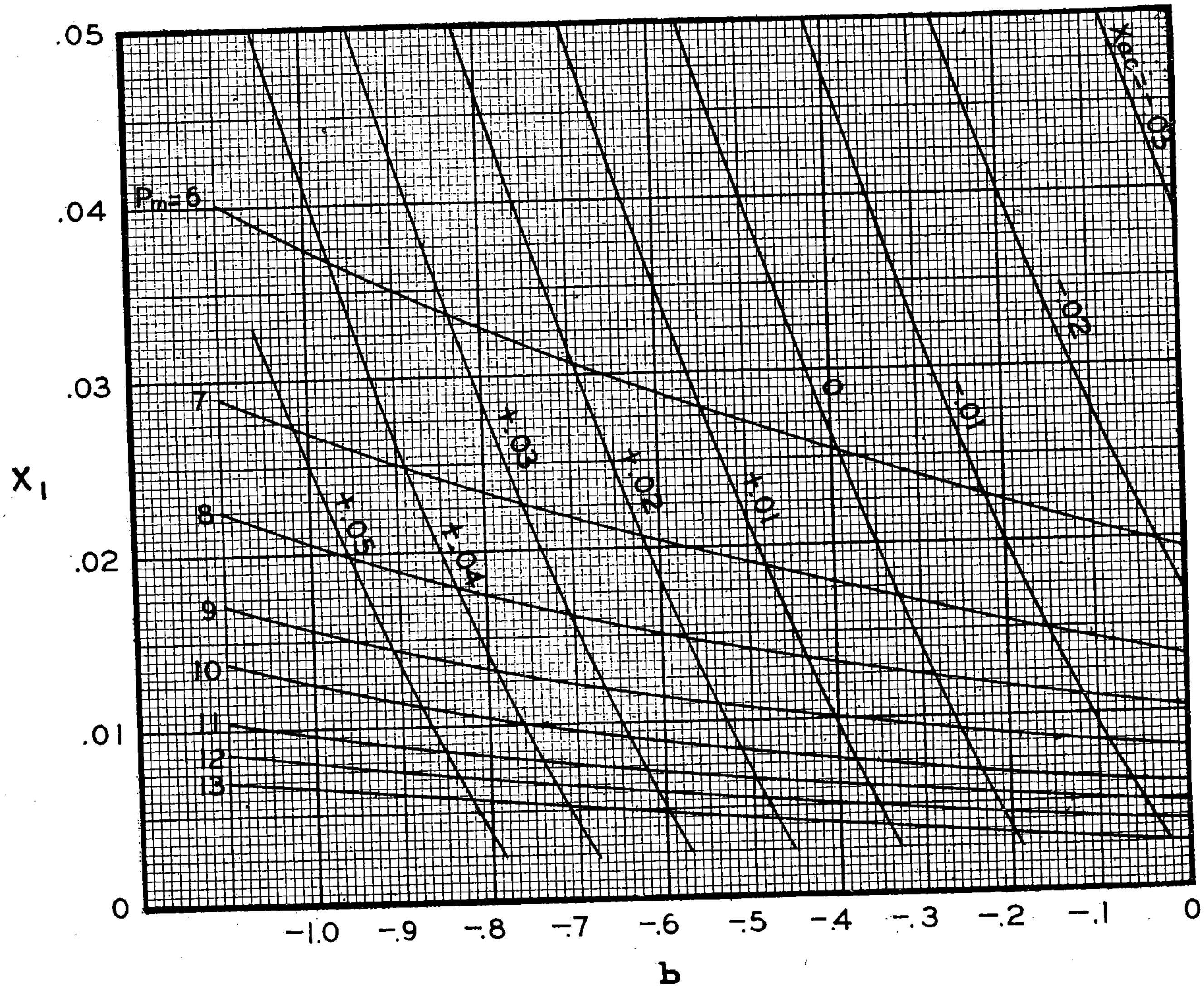


FIGURE B-3

X_1 AND b VS P_m AND X_{ac}

$X_o = .002$

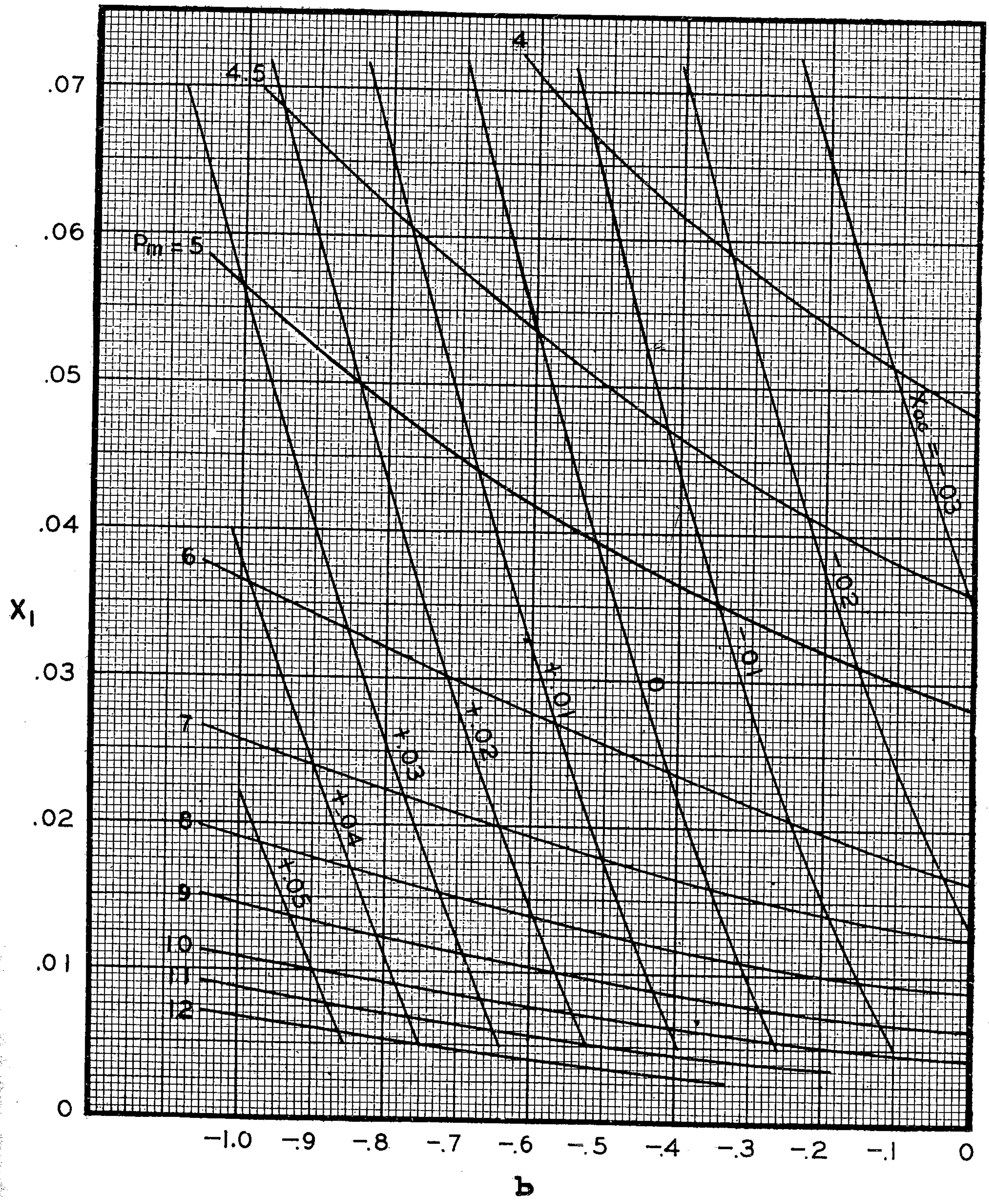


FIGURE B-4

X_1 AND b VS P_m AND X_{ac}

$X_o = .005$

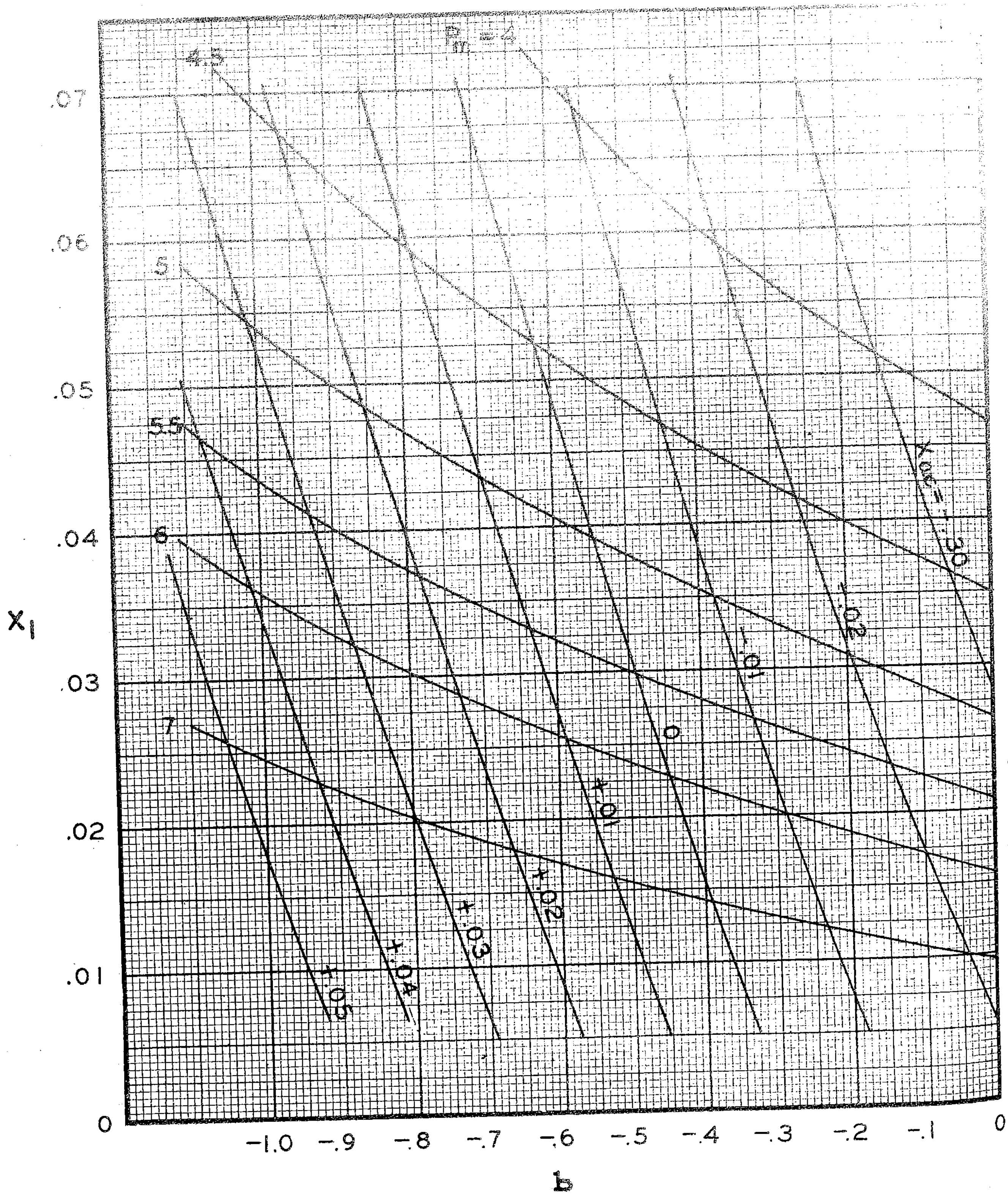


FIGURE B-5

X_1 AND b VS P_m AND X_{ac}

$X_o = .010$

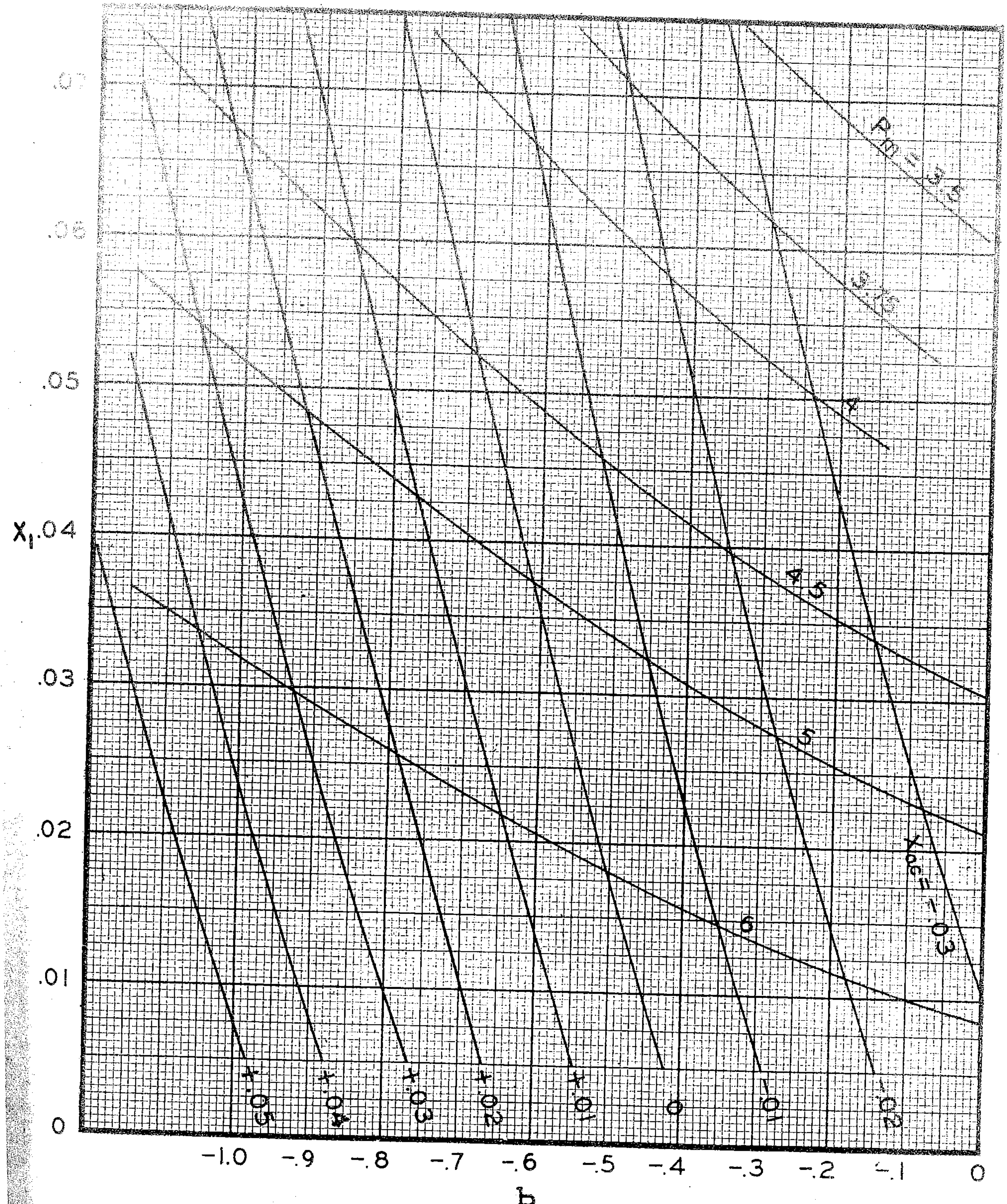


FIGURE B-6

X_1 AND b VS P_m AND X_{ac}

$X_o = .020$

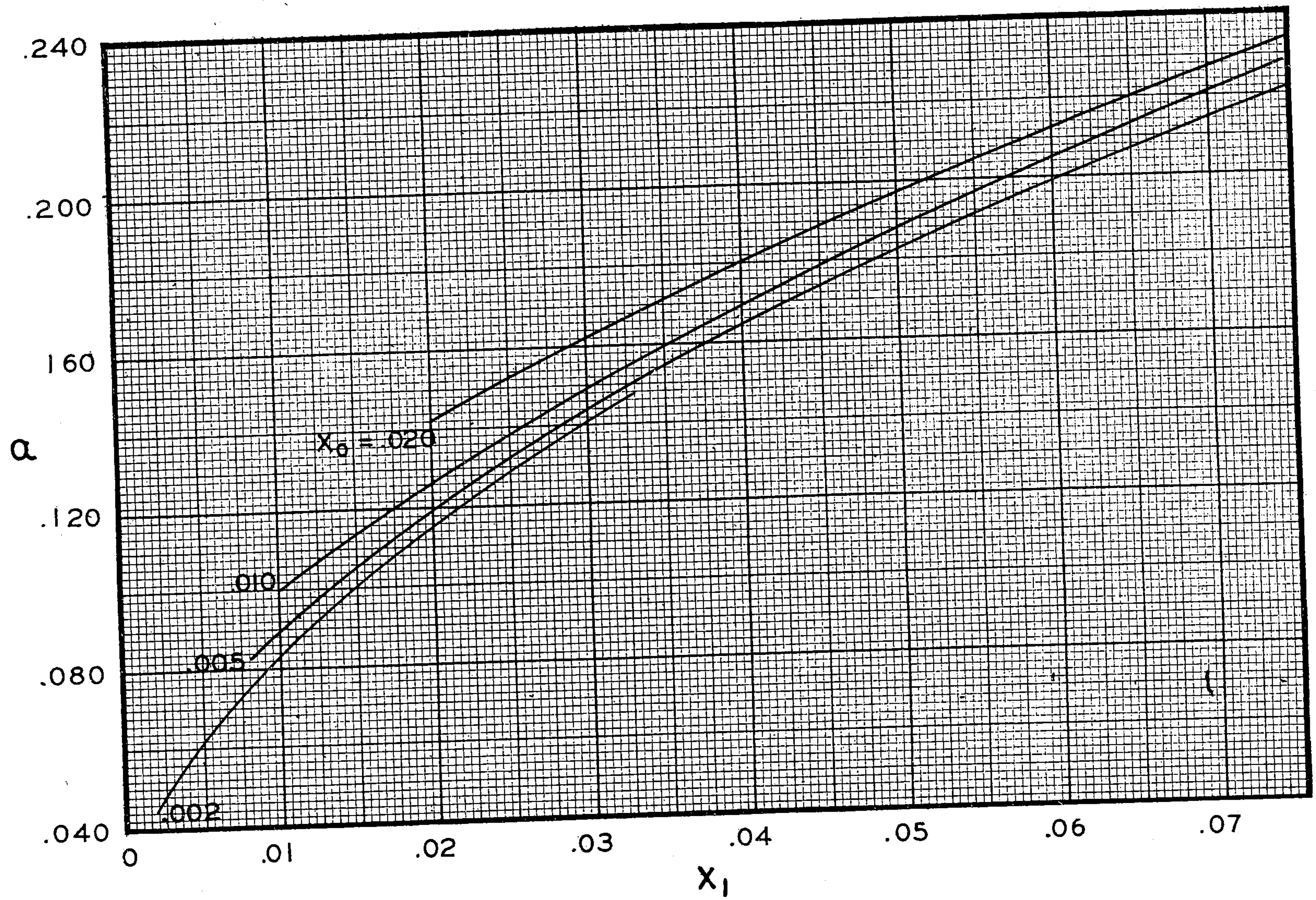


FIGURE B-7
α VS X_1 AND X_0

B-3, B-4, B-5, or B-6) with the proper values of P_m and x_o, b and x_1 are determined. For intermediate values of x_o, the results for b and x_1 should be interpolated between adjacent charts. The constant a is then determined from the chart, Figure B-7.

The resulting values of P may then be readily calculated by inserting the determined values of the constants into the expressions for P_1, P_2, and P_3 contained on Figure B-1. A tabular method of calculation is most desirable. It is noted that P as used above corresponds to Pa_1 in equations (1.2) and (1.3).

The above method is applied below to the calculation of the additional pressure over an actual airfoil for which the following data apply. (Note: This is the same airfoil for which the P_b distribution is calculated in Appendix A.)

$$\rho = .0158$$

$$x_{ac} = .008$$

From Figure B-2 $P_m = 5.12$

$$x_o = \rho/2 = \frac{.0158}{2} = .0079$$

From Figure B-4 $b = -.63$

$$x_1 = .041$$

and Figure B-7 $a = .166$

From Figure B-5 $b = -.64$

$$x_1 = .0395$$

and Figure B-7 $a = .167$

TABLE B-1

EXAMPLE: CALCULATION OF ADDITIONAL PRESSURE DISTRIBUTION OVER AN UNCLASSIFIED AIRFOIL

COL. NO.	①	②	③	④	⑤	⑥	⑦	⑧
ITEM	X	$\left(\frac{X-X_0}{X_0}\right)^2$	1−②	$P_m \times$ ③ $= P_1$				
	0	1	0	0				
	.002	.559	.441	2.25				
	.004	.244	.656	3.36				
	.0079	0	1	5.12				
ITEM	X	$\left(\frac{X-X_0}{X_1-X_0}\right)^2$	$\sqrt{\frac{1-X_1}{X_1}}$	a × ③	1 − ④	② × ⑤	1 − ⑥	$P_m \times$ ⑦ $= P_2$
	.0079	0	4.88	.8143	.1857	0	1	5.12
	.01	.0043	↓	↓	↓	.0008	.9992	5.10
	.02	.141	↓	↓	↓	.0262	.9738	4.98
	.03	.471	↓	↓	↓	.0875	.9125	4.67
	.0401	1.0	↓	↓	↓	.1857	.8143	4.17
ITEM	X	$\sqrt{\frac{1-X}{X}}$	$X - X_1$	b × ③	1 + ④	② × ⑤	$P_m \times a \times$ ⑥ $= P_3$	
	.0401	4.88	0	0	1.0	4.88	4.17	
	.05	4.36	.01	−.006	.994	4.34	3.70	
	.10	3.00	.06	−.038	.962	2.89	2.46	
	.15	2.38	.11	−.070	.930	2.21	1.89	
	.20	2.00	.16	−.102	.898	1.79	1.53	
	.30	1.53	.26	−.165	.835	1.28	1.09	
	.40	1.22	.36	−.229	.771	.94	.80	
	.50	1.00	.46	−.292	.708	.70	.60	
	.60	.82	.56	−.356	.644	.53	.45	
	.70	.66	.66	−.420	.580	.39	.33	
	.80	.50	.76	−.484	.516	.26	.22	
	.90	.33	.86	−.547	.453	.15	.13	
	.95	.23	.91	−.579	.421	.09	.08	
	1.00						0	

Interpolating:

$$b = -.63 + \left[-.64 - (-.63)\right]\left[\frac{.0079 - .005}{.010 - .005}\right] = -.636$$

$$x_1 = .041 + \left[.0395 - .041\right]\left[\frac{.0079 - .005}{.010 - .005}\right] = .0401$$

$$a = .166 + \left[.167 - .166\right]\left[\frac{.0079 - .005}{.010 - .005}\right] = .1666$$

$$P_1 = 5.12\left[1 - \left\{\frac{x - .0079}{.0079}\right\}^2\right]$$

$$P_2 = 5.12\left[1 - \left(\frac{x - .0079}{.0401 - .0079}\right)^2 \left(1 - .1666\sqrt{\frac{1 - .0401}{.0401}}\right)\right]$$

$$P_3 = 5.12\ (.1666)\left[1 - .636\ (x - .0401)\right]\sqrt{\frac{1-x}{x}}$$

A tabular calculation of P is made in Table B-1 and the results are plotted on Figure B-8.

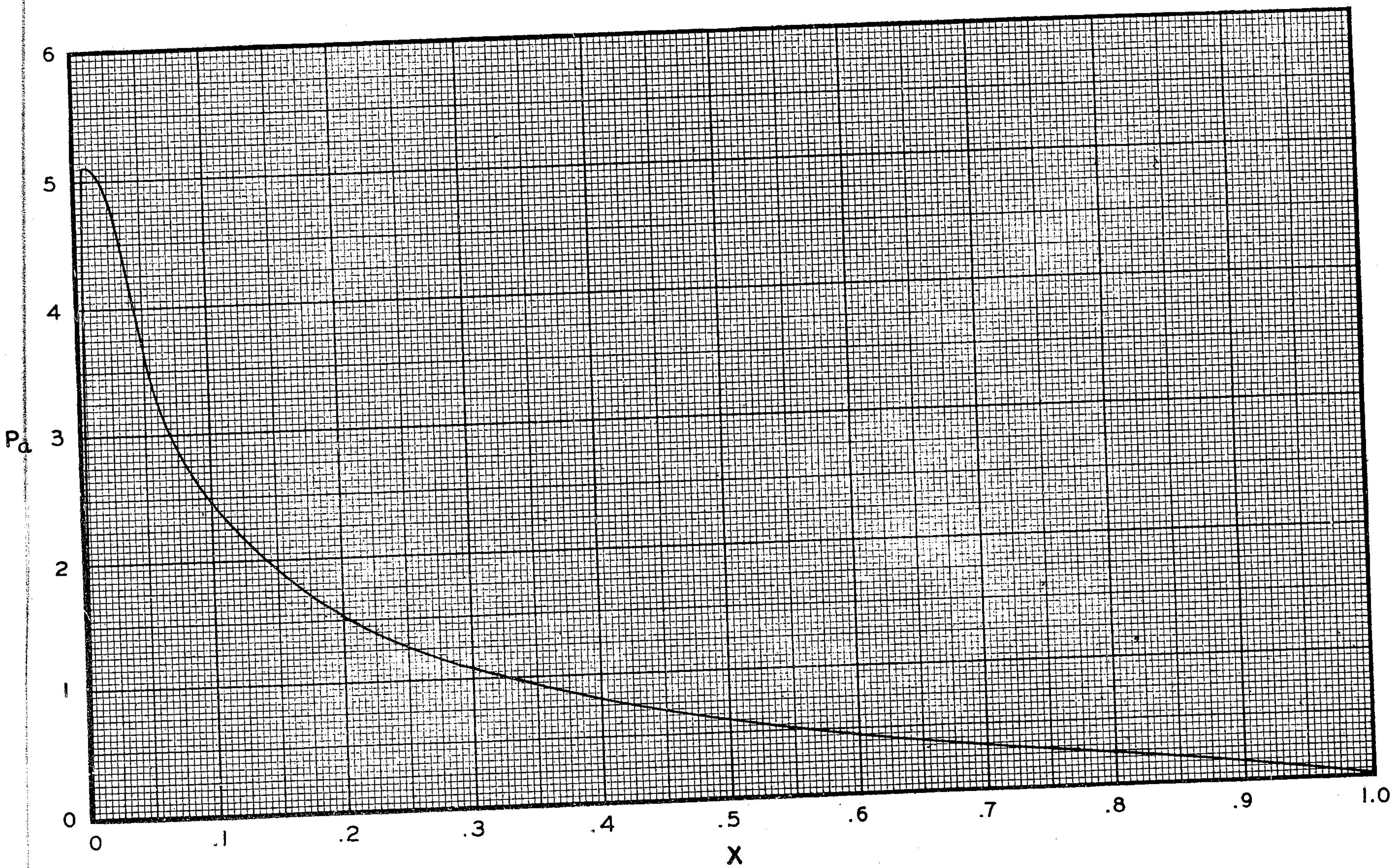


FIGURE B-8

EXAMPLE: P_a DISTRIBUTION OVER AN UNCLASSIFIED AIRFOIL

APPENDIX C

ILLUSTRATED EXAMPLES OF THE APPLICATION OF CHAPTERS 1, 2, and 3 TO THE CALCULATION OF PRESSURE DISTRIBUTION OVER THE NACA 23014.87 CLASSIFIED AIRFOIL

EXAMPLES

C.0 INTRODUCTION

Application of the procedure will be demonstrated by calculating in detail the pressure distribution for a section of a typical classified wing, first as a plain airfoil section and then with an open flap. Let it be assumed that this information is desired for a section located 60 percent of the semispan, or 285.6 inches, from the centerline of the wing used in the example of reference 1.

C.1 PLAIN SECTION

C.10 DETERMINATION OF P_o AND P_{a_1} DISTRIBUTION

The first step in computing the distributions is the determination of P_o and P_{a_1} distributions for the normal section. Those presented in figure 7-2 of reference 1 will be used as far as possible although it is evident that these data, based on an earlier report, are not identical with the latest section characteristics for the N.A.C.A. 230 airfoils as given in reference 4. The necessary data are obtained as follows:

$t = 0.1487$ (ref. 1, table IX, line 5)

$x_{ac} = 0.25 - 0.2323 = 0.0177$ (ref. 1 table IX, line 28)

$-c_{m_{ac}} = -(-0.0078) = 0.0078$ (ref. 1 table IX, line 29)

$z_c = 0.018$ (ref. 4 table I)

PD classification = D12 for 23015 (ref. 4 table I)

TABLE C-1

P_a DISTRIBUTION DATA FOR CLASSIFIED AIRFOILS

CHORD STATION	P_{ac_1}	P_{a_t} CLASS A (1)	CLASS B	CLASS C	CLASS D	CLASS E	CLASS F (1)
0	0	0	0	0	0	0	0
.0125	3.2	6.00	5.93	4.98	4.32	3.87	
.025	4.5	4.58	4.37	4.23	4.02	3.81	3.64
.05	5.5	3.18	3.20	3.22	3.25	3.27	3.29
.075	5.9	2.56	2.63	2.68	2.76	2.81	2.87
.10	5.7	2.20	2.26	2.32	2.39	2.44	2.51
.15	5.0	1.72	1.77	1.85	1.90	1.95	2.01
.20	4.3	1.42	1.47	1.54	1.58	1.62	1.68
.25	3.6	1.23	1.25	1.30	1.35	1.39	1.43
.30	2.9	1.08	1.10	1.14	1.16	1.18	1.22
.40	1.4	.85	.86	.87	.88	.89	.90
.50	0	.67	.67	.68	.68	.69	.70
.60	-1.4	.51	.51	.51	.51	.51	.51
.70	-2.9	.37	.38	.37	.37	.36	.36
.80	-4.3	.24	.25	.24	.24	.23	.23
.90	-5.7	.13	.13	.12	.12	.11	.12
.95	-5.5	.06	.06	.06	.06	.06	.06
1.00	0	0	0	0	0	0	0

NOTES (1). These distributions were not given in reference 4. Values presented here were obtained as outlined in paragraph 1.33.

2. Classes A, B, C, D, E, and F correspond to airfoils having the same thickness distribution as the NACA 0010 Series and having values of t of 6%, 9%, 12%, 15%, 18% and 21% respectively.

TABLE C-2

CALCULATION OF P_{α_1}, P_b AND P_o

AIRFOIL ________ P.D.CLASSIFICATION ____

			AIRFOIL STATION — FRACTION OF CHORD																
1	a	b	.0125	.025	.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.80	.90	.95	1.00
2		$P_{\alpha_{t_0}}$	7.85	4.96	3.13	2.44	2.07	1.60	1.32	1.15	1.02	.83	.66	.51	.38	.25	.13	.06	0
3		$\Delta P_{\alpha_{t_1}}$	-22.8	-6.30	.78	2.06	2.11	1.94	1.72	1.34	.94	.33	.17	0	-.11	-.11	-.06	0	0
4	( )×③	$(t)\times\Delta P_{\alpha_{t_1}}$																	
5	② + ④	P_{α_t}																	
6		$P_{\alpha.c._1}$	3.2	4.5	5.5	5.9	5.7	5.0	4.3	3.6	2.9	1.4	0	-1.4	-2.9	-4.3	-5.7	-5.5	0
7	( )×⑥	$(X_{ac})\ P_{\alpha.c._1}$																	
8	⑤ + ⑦	P_{α_1}																	
9	$c_{n_{bm_1}}$=6.30	P_{bm_1} CLASS 1	2.85	4.25	6.05	7.10	7.80	8.80	9.30	9.50	9.50	8.80	7.75	6.60	5.30	3.75	2.05	1.10	0
	$c_{n_{bm_1}}$=	CLASS																	
10	$c_{n_{bc_1}}$ = 0	P_{bc_1} CLASS 0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	$c_{n_{bc_1}}$ = 9.70	CLASS 1	2.5	5.5	10.0	14.5	18.0	23.0	25.0	25.8	25.0	20.5	14.0	6.0	-2.5	-5.5	-4.5	-2.5	0
	$c_{n_{bc_1}}$ = 18.75	CLASS 2	32.5	47.0	56.5	59.0	57.5	47.5	37.0	30.0	24.5	18.0	13.0	9.0	5.5	3.5	1.5	1.0	0
	$c_{n_{bc_1}}$ =	CLASS																	
11	( )×⑨	$(-c_{m_{a.c.}})\ P_{bm_1}$																	
12	( )×⑩	$(Z_c)\ P_{bc_1}$																	
13	⑪ + ⑫	P_b																	
14	$c_{n_b} = (-c_{m_{a.c.}})\, c_{n_{bm_1}} + (Z_c)\, c_{n_{bc_1}} = c_{n_{bm}} + c_{n_{bc}} =$																		
15	( )×⑧	$(c_{n_b})\ P_{\alpha_1}$																	
16	⑬ − ⑮	P_o																	

The P_o and P_{a_1} distribution are next computed in table C-2a as follows:

(1) Fill in the blank spaces indicated in the title and in parentheses in column "a", lines 4, 7, 11 and 12 from the data already tabulated.

(2) Fill in line 4 with the products obtained by multiplying line 3 by the value of t in line 4, column "a".

(3) Add lines 2 and 4 to obtain values for line 5, in accordance with equation 1.27. Note that in this case it would have been sufficiently accurate to have ignored lines 2, 3 and 4 of this table and to have inserted the data from table C-1 class D, directly into line 5.

(4) Multiply line 6 by the value of x_{ac} in column "a", line 7, to obtain the other values in line 7.

(5) Add line 5 and 7 to obtain P_{a_1} in line 8 in accordance with equation 1.26.

(6) Multiply line 9 by minus $c_{m_{ac}}$ in column "a", line 11, to obtain the other values in line 11. Note that only one class of moment-basic distribution is given in "line 9". If the first number in the PD-classification had been different from unity, the values for the distribution indicated

TABLE C-2a

CALCULATION OF P_{a_1}, P_b AND P_o

AIRFOIL 23014.87 P. D. CLASSIFICATION D12

			AIRFOIL STATION — FRACTION OF CHORD																
1	a	b	.0125	.025	.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.80	.90	.95	1.00
2		P_{at_o}	7.85	4.96	3.13	2.44	2.07	1.60	1.32	1.15	1.02	.83	.66	.51	.38	.25	.13	.06	0
3		ΔP_{at_1}	-22.8	-6.30	.78	2.06	2.11	1.94	1.72	1.34	.94	.33	.17	0	-.11	-.11	-.06	0	0
4	(.1487) × ③	$(t) \times \Delta P_{at_1}$	-3.39	-.94	.11	.31	.31	.29	.26	.20	.14	.05	.03	0	-.02	-.02	-.01	0	0
5	② + ④	P_{at}	4.46	4.02	3.24	2.75	2.38	1.89	1.58	1.35	1.16	.88	.69	.51	.36	.23	.12	.06	0
6		$P_{a.c._1}$	3.2	4.5	5.5	5.9	5.7	5.0	4.3	3.6	2.9	1.4	0	-1.4	-2.9	-4.3	-5.7	-5.5	0
7	(.0177) × ⑥	$(X_{ac})\ P_{a.c._1}$	.06	.08	.10	.10	.10	.09	.08	.06	.05	.02	0	-.02	-.05	-.08	-.10	-.10	0
8	⑤ + ⑦	P_{a_1}	4.52	4.10	3.34	2.85	2.48	1.98	1.66	1.41	1.21	.90	.69	.49	.31	.15	.02	-.04	0
9	$c_{n_{bm_1}}$=6.30	P_{bm_1} CLASS 1	2.85	4.25	6.05	7.10	7.80	8.80	9.30	9.50	9.50	8.80	7.75	6.60	5.30	3.75	2.05	1.10	0
	$c_{n_{bm_1}}$=	CLASS																	
10	$c_{n_{bc_1}}$ = 0	P_{bc_1} CLASS 0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
	$c_{n_{bc_1}}$ = 9.70	CLASS 1	2.5	5.5	10.0	14.5	13.0	23.0	25.0	25.8	25.0	20.5	14.0	6.0	-2.5	-5.5	-4.5	-2.5	0
	$c_{n_{bc_1}}$ =18.75	CLASS 2	32.5	47.0	56.5	59.0	57.5	47.5	37.0	30.0	24.5	18.0	13.0	9.0	5.5	3.5	1.5	1.0	0
	$c_{n_{bc_1}}$ =	CLASS																	
11	(.0078) × ⑨	$(-C_{m_{a.c.}})\ P_{bm_1}$	.02	.03	.05	.06	.06	.07	.07	.07	.07	.07	.06	.05	.04	.03	.02	.01	0
12	(.018) × ⑩	$(Z_c)\ P_{bc_1}$	.59	.85	1.02	1.06	1.04	.86	.67	.54	.44	.32	.23	.16	.10	.06	.03	.02	0
13	⑪ + ⑫	P_b	.62	.88	1.07	1.12	1.10	.93	.74	.61	.51	.39	.29	.21	.14	.09	.05	.03	0
14	$c_{n_b} = (-c_{m_{a.c.}}) c_{n_{bm_1}} + (Z_c)\ c_{n_{bc_1}} = c_{n_{bm}} + c_{n_{bc}} = (.0078)\ 6.30 + (.018)\ 18.75 = .049 + .338 = .387$																		
15	(.387) × ⑧	$(c_{n_b})\ P_{a_1}$	1.75	1.59	1.29	1.10	.96	.77	.64	.55	.47	.35	.27	.19	.12	.06	.01	-.02	0
16	⑬ − ⑮	P_o	-1.13	-.71	-.22	.02	.14	.16	.10	.06	.04	.04	.02	.02	.02	.03	.04	.05	0

by this number would have been inserted in the blank spaces provided in line 9 and these data multiplied by minus $c_{m_{ac}}$ to obtain P_{bm} in line 11 and $c_{n_{bm}}$ in line 14.

(7) Multiply line 10, class 2 values by the values of z_c in line 12, column "a", to obtain the other values in line 12. Similar to the moment basic data, if the second PD-classification number had been different from 0, 1 or 2, a camber basic distribution different from any of the tabulated values in line 10 would have been indicated. The values for such a distribution would then have been inserted in the blank spaces provided and these data used to obtain P_{bc} and $c_{n_{bc}}$ in lines 12 and 14.

(8) Add lines 11 and 12 to obtain the P_b values in line 13 in accordance with equation 1.24

(9) Calculate c_{n_b} in line 14 as indicated and insert this value in the blank space provided in line 15, column "a". Note that the values of $c_{n_{bm_1}}$ and $c_{n_{bc_1}}$ corresponding to the classes of moment-basic and camber-basic distribution indicated by the PD-classification are used for this purpose.

(10) The other values in line 15 are obtained by multiplying line 8 by c_{n_b} obtained in line 14.

TABLE C-3

CALCULATION OF P, P_l, P_u, p, p_l, & p_u

c_n	q	a	1	b	.0125	.025	.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.80	.90	.95	1.00
					AIRFOIL STATION – FRACTION OF CHORD																
0	1.0	TABLE C-2a LINE 16	2	P_o	−1.13	− .71	− .22	.02	.14	.16	.10	.06	.04	.04	.02	.02	.02	.03	.04	.05	0
1.0	1.0	TABLE C-2a LINE 8	3	P_{a1}	4.52	4.10	3.34	2.85	2.48	1.98	1.66	1.41	1.21	.90	.69	.49	.31	.15	.02	− .04	0
1.244	1.0	c_n × ③	4	P_a	5.62	5.10	4.15	3.55	3.09	2.46	2.07	1.75	1.51	1.12	.86	.61	.39	.19	.02	− .05	0
		② + ④	5	P	4.49	4.39	3.93	3.57	3.23	2.62	2.17	1.81	1.55	1.16	.88	.63	.41	.21	.06	0	0
		FIG. 2-1 OR ⑤+⑦	6	P_l	.96	.98	.84	.74	.65	.49	.37	.29	.22	.14	.12	.08	.06	.05	.08	.14	0
		FIG. 2-1 OR ⑥-⑤	7	P_u	−3.53	−3.41	−3.09	−2.83	−2.58	−2.13	−1.80	−1.52	−1.33	−1.02	− .76	− .55	− .35	− .16	.02	.14	0
	42.5	q × ⑤	8	p	190.8	186.6	167.0	151.7	137.3	111.4	92.2	76.9	65.9	49.3	37.4	26.8	17.4	8.9	2.6	0	0
		q × ⑥	9	p_l	40.8	41.7	35.7	31.5	27.6	20.8	15.7	12.3	9.4	6.0	5.1	3.4	2.6	2.1	3.4	6.0	0
		q × ⑦	10	p_u	−150.0	−144.9	−131.3	−120.3	−109.7	−90.5	−76.5	−64.6	−56.5	−43.4	−32.3	−23.4	−14.9	−6.8	.9	6.0	0
	1.0		4	P_a																	
			5	P																	
			6	P_l																	
			7	P_u																	
			8	p																	
			9	p_l																	
			10	p_u																	
	1.0		4	P_a																	
			5	P																	
			6	P_l																	
			7	P_u																	
			8	p																	
			9	p_l																	
			10	p_u																	

(11) The P_o distribution in line 16 is obtained by subtracting line 15 from line 13, in accordance with equation 1.2.

C.11 DETERMINATION OF P DISTRIBUTION AT A GIVEN ANGLE OF ATTACK

The distribution of P_o and P_{a_1} obtained in table C-2a may now be used to obtain the normal-force coefficient distribution P for any value of c_n. To illustrate the method the distribution will be computed for the section when the angle of attack of the wing is 13.9 degrees, which is the angle of ~~attack used~~ in the examples of reference 1. In reference 1, table X, line 6, the value of c_l was found to be 1.266 and in line 22, c_d was found to be 0.0633 at this angle of attack. Usually it will be sufficient to assume c_n equal to c_l , but precisely

$$c_n = 1.266 \cos 13.9^\circ + 0.0633 \sin 13.9^\circ = 1.244$$

The distribution of pressure-difference coefficient P for this value of c_n can now be found by means of equation 1.3. Table C-3 has been prepared as a convenient form for computing P, as well as the surface pressure coefficients P_u and P_l and the actual pressures corresponding to a given value of c_n and dynamic pressure q. Spaces are provided for other values of c_n, although the pressures for c_n = 1.244 only are actually calculated. The first column, headed by "c_n", designates the value of section normal-force coefficient for which the various lines of data are calculated; the second column designates the dynamic pressure; the third column, headed by "a", indicates how the various lines of data are derived; the fourth column gives the line numbers; and the fifth column designates by standard symbols the particular data contained in each line in the columns to the right. These columns are headed by decimal fractions corresponding to the selected chord stations for which the data is to be obtained. The procedure is as follows:

(1) Copy P_o and P_{a_1} values from table C-2a into lines 2 and 3.

(2) Multiply line 3 by $c_n = 1.244$ to obtain P_a values in line 4.

(3) Add lines 2 and 4 to obtain P values in line 5.

C.12 DETERMINATION OF SURFACE-PRESSURE COEFFICIENTS

The values of P in line 5 of table C-3 represent in terms of the dynamic pressure the differences in pressure between the upper and lower surfaces at the indicated stations along the chord. The individual surface-pressure coefficients above or below atmospheric pressure may now be found by equations 2.1, 2.2 and 2.3. Since the N.A.C.A. 230 series airfoil has the standard base form (N.A.C.A. 0010), the values of P_f may be read for t = 0.1487 from the plot of P_f vs. t at the top of Figure 2-1. It is not necessary, however, to read the values of P_f as the chart may be used to read one of the surface pressures directly. The procedure is to enter the chart at the value of t corresponding to the base airfoil. From the intersection of a horizontal line corresponding to this value of t and the curve representing the chord station for which the surface pressures are to be determined, a vertical line is dropped to the curve below corresponding to the value of P for this chord station. The surface pressure P_l (or P_u) is then read from the scale on the right corresponding to the intersection of these two curves. If the base airfoil were not of the standard (N.A.C.A. 0010) form, a knowledge of the P_f distribution for the shape should be obtained by method of reference 2 or otherwise. The procedure is then to enter the chart by the upper scale of P_f, or the lower scale of $(1 - P_f)$, with the value thus obtained corresponding to the chord station concerned. Since, in this example, all of the values of P in line 5 of table C-3 are positive except the P value for the 0.95 chord station, values of P_l are obtained from the chart for chord stations forward of station 0.95. The P_u values in line 7 for these chord stations are then found by subtracting the P_l values in line 6 from the P values of line 5 in accordance with equation 2.1. Since P = 0 for station 0.95 chord, identical values for P_l and P_u for this station are obtained from the chart and entered in lines 6 and 7.

C.13 DETERMINATION OF ACTUAL PRESSURES

The actual pressure differences/and surface pressures can now be found in pounds per square foot or in pounds per square inch for a particular speed and altitude by multiplying the respective coefficients by the dynamic pressure, in the proper units, corresponding to the speed and altitude. It will be assumed that the actual pressures are desired for a speed of 150 miles per hour at 10,000 feet above sea level. For the standard atmosphere at this altitude

$$\rho = 0.7384 \times 0.002378 = 0.001756$$

The values of p, p_l and p_u in lines 8, 9 and 10 are therefore found by multiplying the values in line 5, 6 and 7 by

$$q = \frac{1}{2} \times 0.001756 \times (150 \times \frac{88}{60})^2 = 42.5 \text{ lbs./sq.ft.}$$

for incompressible flow. The effect of compressibility on these calculated pressures may be determined by the use of Figure 2-2. The velocity of sound is determined from Figure 2-4 to be 737 m.p.h. at 10,000 feet altitude, and hence the Mach number

$$M = \frac{150}{737} = .203$$

Interpolating in Figure 2-2 at M = .203, the value of $\frac{(P)_0}{(P)_M}$ at any station on the surface for any value $(P)_0$

is read off as the value of the ordinate corresponding to the intersection of the interpolated Mach line ($M = .203$) and the pertinent $(P)_0$ coordinate which latter corresponds to either P_l or P_u (Table C-3) at $M = 0$. The actual pressure, corrected for the effect of compressibility, is then obtained by:

$$(p)_M = \frac{(P)_M}{(P)_0} \quad (P)_0 \, q$$

It is noted that in this example, such an effect is small and raises the negative peak pressure on the upper surface at $x = .0125$ from $- 150.0$ lb./ft^2 to $- 159.0$ lb/ft^2.

C.14 ESTIMATION OF THE CRITICAL SPEED

The extreme negative value of the pressure coefficient for this example is seen (Figure C-1) to occur near the leading edge on the upper surface and to have a value of -3.53 at $x = .0125$. For this value of P_u, M_c is found from Figure 2-2 to be approximately 0.360. Since the speed of sound at 10,000 feet altitude is 737 miles per hour, the local speed of sound would not be reached at a c_n of 1.271 until the airplane approaches a speed of $0.360 \times 737 = 265$ miles per hour which is well above the design speed of 150 miles per hour and justifies the application of the procedure of Chapters 1 and 2 in estimating the distribution of chordwise air load for this section under the conditions assumed.

The maximum horizontal speed as well as the terminal velocity diving speeds occur at normal-force coefficients near $c_n = 0$. The critical speed for this section is found to be approximately 465 miles per hour at 10,000 feet (480 at sea level) for $c_n = 0$, i.e., for $P = P_o$. The point where the upper surface peak negative pressure coefficient occurs, moves aft from the leading edge as c_n is decreased.

C.2 SECTION WITH SPLIT FLAP DEFLECTED

C.20 REQUIRED DATA AND DESCRIPTION OF SECTION

To illustrate the procedure of Chapter 3 the distributions of total normal-pressure coefficient P and the separate surface-pressure coefficients P_u and P_l, will be calculated for the section of the previous example with a split trailing edge flap of flap-chord ratio 0.15 (E = 0.15) and flap angle of 60 degrees ($\delta_f = 60^\circ$) for the same angle of attack ($\alpha = 13.9^\circ$) as before. Section characteristics from force tests for the particular conditions cited are not available but may be estimated as follows:

$c_l = 1.86$ ANC-1(1), page 7-44

c_l max= 1.73 for normal section ANC-1(1), page 7-3

Δc_l max= 0.80 Diehl, Walter S.: Engineering Aerodynamics (Revised edition), page 154

c_l max= for flapped section = 1.73 + 0.80 = 2.53

c_l opt= 0.096 for normal section ANC-1(1), page 7.3

Assuming $c_{l\,opt}$ for flapped section is the same as for normal section:

$$\frac{c_l - c_{l\,opt}}{c_{l\,max} - c_{l\,opt}} = \frac{1.86 - 0.096}{2.53 - 0.096} = 0.72$$ and the corresponding value of Δc_{d_o} is 0.010 ANC-1(1) Page 6-8

$c_{d_{o\,min}} = 0.008$	ANC-1(1)	Page 7-18
$\Delta c_{d_{o\,min}} = 0.112$	ANC-1(1)	Page 7-36
$c_{d_i} = 0.200$	ANC-1(1)	Page 7-44

$c_d = 0.010 + 0.008 + 0.112 + 0.200 = 0.330$

$c_n = 1.86 \cos 13.9^\circ + 0.330 \sin 13.9^\circ = 1.88$

$\Delta c_m = -.21$ ANC-1(1) Page 7-36

$\Delta c'_m = 1.28\,(-.21) = -0.27$ Equation 3.3

$c_{n_{b\delta}} = \frac{-0.27}{-0.435} = 0.62$ Equation 3.4

C.21 ADDITIONAL AND ZERO-LIFT DISTRIBUTION

The distributions P_{a_1} and P_o have been calculated in section C.10. Figure 7 of NACA T.N. 571 indicates a linear variation of Δc_m with lift coefficient (or angle of attack) which results in a consequent variation of the $P_{o\delta}$ distribution with normal-force coefficient. Since the P, P_l and P_u distributions are to be calculated for only one value of the normal-force coefficient ($c_n = 1.88$), no investigation will be made of the variation of P_o with c_n. The calculations are carried out in table C-4 in which the first column contains the line numbers; the second column, headed by "a", indi-

TABLE C-4

CALCULATION OF $P_o\delta$

AIRFOIL 23014.87 SPLIT FLAP $E = .15$ $\delta_{F_l} = 60°$

1	a	$\frac{x}{1-E}$	.05	.10	.20	.30	.40	.50	.60	.70	.80	.90	1.00
2	(1 − E) × (1)	x	.0425	.085	.17	.255	.34	.425	.51	.595	.68	.765	.85
3	SEE NOTE BELOW	$P_b\delta_1$	.15	.23	.34	.44	.56	.68	.82	.99	1.25	1.75	2.20

VALUES IN LINE 3, OBTAINED FROM FIG. 3-6 FOR CHORD STATIONS IN LINE 1, ARE PLOTTED VS. CHORD STATIONS COMPUTED IN LINE 2. FROM THIS PLOT, READ VALUES FOR LINE 5 CORRESPONDING TO CHORD STATIONS IN LINE 4.

4		x	.0125	.025	.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.80	.90	.95	1.00
5		$P_b\delta_1$	.07	.11	.16	.20	.24	.31	.38	.44	.50	.64	.80	1.00	1.34	1.96			
6	(.62) × (5)	$P_b\delta$	.04	.07	.10	.12	.15	.19	.24	.27	.31	.40	.50	.62	.83	1.21			
7	TABLE C-2a LINE 8	P_{a_1}	4.52	4.10	3.34	2.85	2.48	1.98	1.66	1.41	1.21	.90	.69	.49	.31	.15			
8	(−.62) × (7)	$(-C_{n_b\delta}) P_{a_1}$	−2.80	−2.54	−2.07	−1.77	−1.54	−1.23	−1.03	−.87	−.75	−.56	−.43	−.30	−.19	−.09			
9	(6) + (8)	$P_o\delta$	−2.76	−2.47	−1.97	−1.65	−1.39	−1.04	−.79	−.60	−.44	−.16	.07	.32	.64	1.12			

10		$\frac{1-x}{E}$	1.00	.90	.80	.70	.60	.50	.40	.30	.20	.10	.05	0
11	E × (10)	$1-x$	.15	.135	.12	.105	.09	.075	.06	.045	.03	.015	.0075	0
12	1 − (11)	x	.85	.865	.88	.895	.91	.925	.94	.955	.97	.985	.9925	1.00
13	FIGS. 3-7, 3-8, 3-9	$P_b\delta_1$	2.20	2.35	2.45	2.47	2.47	2.40	2.33	2.15	1.87	1.43	1.08	0
14	(.62) × (13)	$P_b\delta$	1.36	1.46	1.52	1.53	1.53	1.49	1.44	1.33	1.16	.89	.67	0
15		P_{a_1}	.08	.06	.05	.03	.01	−.01	−.03	−.04	−.04	−.03	−.02	0
16	(−.62) × (15)	$(-C_{n_b\delta}) P_{a_1}$	−.05	−.04	−.03	−.02	−.01	.01	.02	.03	.03	.02	.01	0
17	(14) + (16)	$P_o\delta$	1.31	1.42	1.49	1.51	1.52	1.50	1.46	1.36	1.19	.91	.68	0

cates the source of, or method of obtaining, the data in each line; and the third column indicates by standard symbols the data presented in each line in the other columns. The distribution for points forward of the hinge is obtained by the first nine lines and the distribution for the points aft of the hinge is obtained in lines 10 to 17 as follows:

(1) Lines 1, 4 and 10 are already completely filled in.

(2) Multiply line 1 by (1-E) to obtain x values in line 2.

(3) Lines 3 and 5 are to be filled in, in accordance with note under line 3.

(4) Multiply line 5 by $c_{n_{b\delta}}$, in this case equal to 0.62, to obtain $P_{b\delta}$ in line 6.

(5) Copy P_{a_1} values from table C-2a, line 8 into line 7.

(6) Multiply line 7 by <u>minus</u> $c_{n_{b\delta}}$ for values in line 8.

(7) Add lines 6 and 8 to obtain $P_{o\delta}$ in line 9 in accordance with equation 3.9.

(8) Multiply line 10 by E to obtain values in line 11.

(9) Subtract values in line 11 from unity to obtain values in line 12.

(10) Values in line 13 are obtained from Figures 3-7, 3-8 and 3-9.

(11) Multiply line 13 by $c_{n_{b\delta}}$ to obtain values in line 14.

TABLE C – 5

CALCULATION OF P, P_u, P_l, p, p_u, AND p_l

AIRFOIL 23014.87 SPLIT FLAP. $E = .15$ $\delta_{F_l} = 60°$ $\alpha = 13.9°$

c'_n	q	a	1	x	.0125	.025	.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.80	.90	.95
0	1.0	TABLE C-2a LINE 16	2	P_o	–1.13	–.71	–.22	.02	.14	.16	.10	.06	.04	.04	.02	.02	.02	.03		
		TABLE C-4 LINE 9	3	$P_{o\delta}$	–2.76	–2.47	–1.97	–1.65	–1.39	–1.04	–.79	–.60	–.44	–.16	.07	.32	.64	1.12		
		(2)+(3)	4	$P_o + P_{o\delta}$	–3.89	–3.18	–2.19	–1.63	–1.25	–.88	–.69	–.54	–.40	–.12	.09	.34	.66	1.15		
1.0		TABLE C-2a LINE 8	5	P_{a_1}	4.52	4.10	3.34	2.85	2.48	1.98	1.66	1.41	1.21	.90	.69	.49	.31	.15		
.98		c'_n × (5)	6	P_α	8.95	8.12	6.61	5.64	4.91	3.92	3.29	2.79	2.40	1.78	1.37	.97	.61	.30		
		(4)+(6)	7	P	5.06	4.94	4.42	4.01	3.66	3.04	2.60	2.25	2.00	1.65	1.46	1.31	1.27	1.45		
		FIG. 2-1, OR (7)+(9)	8	P_l	.90	1.00	.92	.83	.75	.62	.52	.44	.38	.33	.34	.34	.38	.53		
		FIG. 2-1, OR (8)–(7)	9	P_u	–4.16	–3.94	–3.50	–3.18	–2.91	–2.42	–2.08	–1.81	–1.62	–1.32	–1.12	–.97	–.89	–.92		
	36.84	q × (7)	10	p	186.	182.	163.	148.	135.	112.	95.7	82.9	73.7	60.8	53.8	48.2	46.8	53.5		
		q × (8)	11	p_l	33.2	36.8	33.9	30.6	27.6	22.8	19.2	16.2	14.0	12.2	12.5	12.5	14.0	19.5		
		q × (9)	12	p_u	–153.	–145.	–129.	–117.	–107.	–89.1	–76.6	–66.6	–59.6	–48.6	–41.3	–35.8	–32.8	–33.9		

c'_n	q	a	13	$\frac{1-x}{E}$	1.00	.90	.80	.70	.60	.50	.40	.30	.20	.10	.05	0
		1–E × (13)	14	x	.85	.865	.88	.895	.91	.925	.94	.955	.97	.985	.9925	1.00
0	1.0		15	P_o	.03	.03	.04	.04	.04	.04	.04	.05	.05	.03	.02	0
		TABLE C-4 LINE 17	16	$P_{o\delta}$	1.31	1.42	1.49	1.51	1.52	1.50	1.46	1.36	1.19	.91	.68	0
		(15)+(16)	17	$P_o + P_{o\delta}$	1.34	1.45	1.53	1.55	1.56	1.54	1.50	1.41	1.24	.94	.70	0
1.0			18	P_{a_1}	.08	.06	.05	.03	.01	–.01	–.03	–.04	–.04	–.03	–.02	0
1.98		c'_n × (18)	19	P_α	.16	.12	.10	.06	.02	–.02	–.06	–.08	–.08	–.06	–.04	0
		(17)+(19)	20	P	1.50	1.57	1.58	1.61	1.58	1.52	1.44	1.35	1.16	.88	.66	0
		FIG. 2-1, OR (20)+(22)	21	P_l	.60	.63	.65	.69	.69	.69	.69	.68	.67	.62		
		FIG 2-1, OR (21)–(20)	22	P_u	–.90	–.94	–.93	–.92	–.89	–.83	–.75	–.65	–.49	–.26		
	36.84	q × (20)	23	p	55.2	57.9	58.1	59.3	58.1	56.	53.0	49.0	42.7	32.4	24.3	0
		q × (21)	24	p_l	22.1	23.2	24.0	25.4	25.4	25.4	25.4	25.0	24.7	22.8		
		q × (22)	25	p_u	–33.2	–34.6	–34.3	–33.9	–32.8	–30.6	–27.6	–24.0	–18.1	–9.6		

(12) Values in line 15 are obtained from data in table C-2a, line 8, either by plotting P_{a1} vs. x for the trailing edge portion of the airfoil and reading values from this plot for the x stations in line 12 or by direct interpolation.

(13) Multiply line 15 by <u>minus</u> $c_{n_{b\delta}}$ to obtain values in line 16.

(14) Add lines 14 and 16 in accordance with equation 3.9 to obtain $P_{o\delta}$ values in line 17.

The zero lift distribution, which is the sum of the $P_{o\delta}$ distribution obtained as above and the P_o distribution obtained in table C-2a, line 16, is computed in table C-5.

C.22 <u>DETERMINATION OF P DISTRIBUTION AT A GIVEN ANGLE OF ATTACK</u>

The total normal-pressure coefficient P for the conditions cited in paragraph C.20 is calculated in table C-5 in a manner similar to the calculation of P for the normal section except that the fictitious parameter c'_n instead of c_n is used to calculate P_a in lines 6 and 19. The value of c'_n is obtained as follows:

c_{n_a} = 1.88 - (0.62 + 0.39) = .87 by equation 3.5

c_{n_f} = 2.12 x 0.62 + 0.07 x 0.338 + 0.24 x 0.049 + (0.09 - 5.05 x 0.0177) 0.87 = 1.34, by equation 3.6

This value of c_{n_f} is satisfactory for analytical use

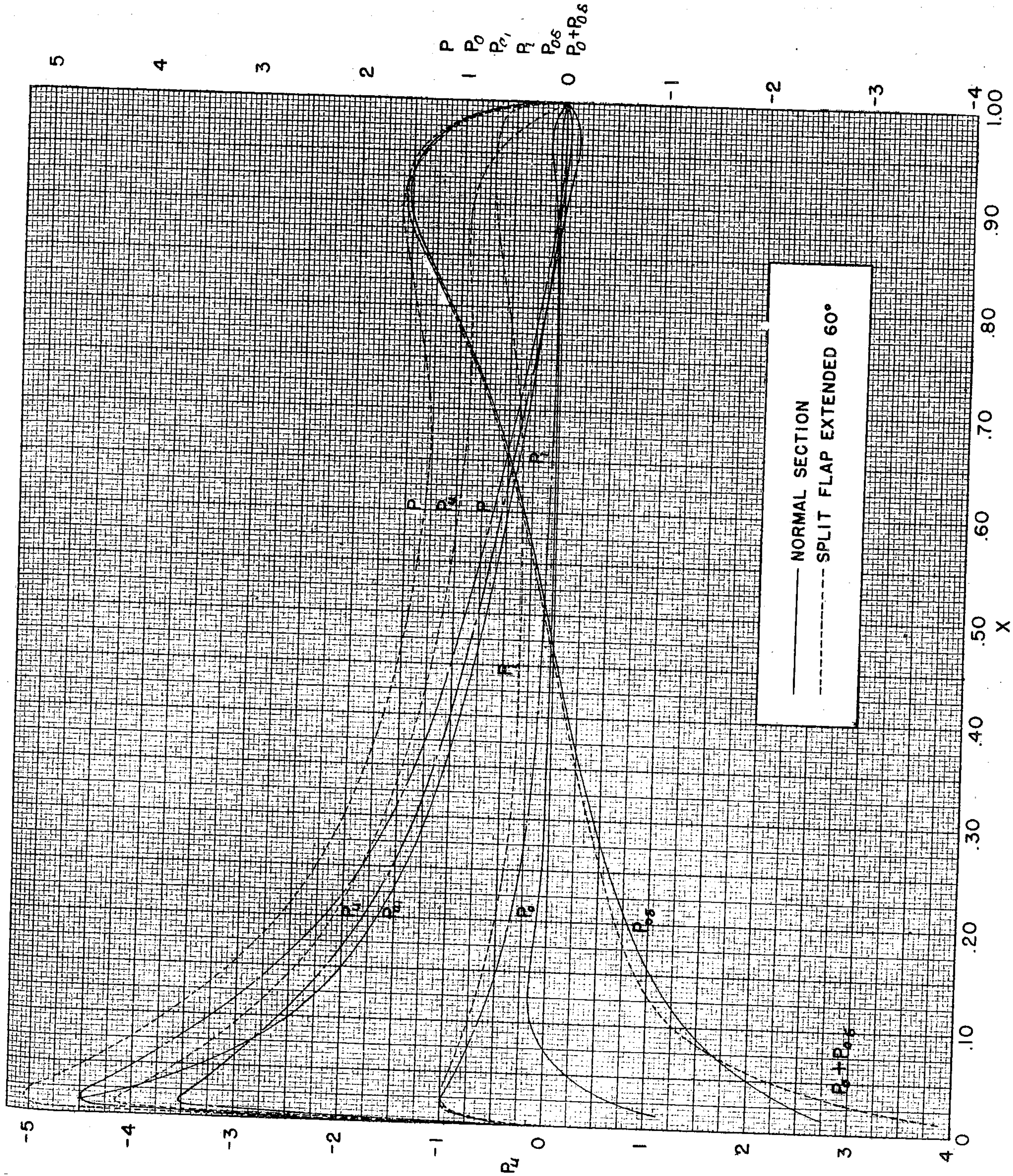


FIG. C-1

VALUES OF $P, P_0, P_{0\delta}, P_{a_1}, P_u$ AND P_η FOR THE 23014.87 AIRFOIL NORMAL SECTION AND SPLIT FLAP EXTENDED in equation 3.5 but does not necessarily represent the true normal-force coefficient for this (split) flap itself. See explanation in paragraph 3.1.

$c'_n = 1.88 + 0.15 \times 1.34 (1 - 0.5) = 1.98$, by equation 3.7

C.23 DETERMINATION OF SURFACE-PRESSURE COEFFICIENTS

The separate surface-pressure coefficients P_l and P_u are calculated in table C-5 in lines 8 and 9 for points forward of the hinge axis and in lines 21 and 22 for points aft of the hinge in the same way that P_l and P_u for the plain section, were obtained in table C-3. The data determined in line 21 represent the relative pressure on the lower surface of the flap and the data in line 22 represent the relative pressure on the upper surface of the airfoil section aft of the hinge axis. The uniform pressure coefficient in the region back of the deflected flap may be estimated to be approximately 0.70 q from NACA T.N. 627.

C.24 DETERMINATION OF ACTUAL PRESSURES

It will be assumed that at this particular flap angle and angle of attack the limit load occurs at 120 miles per hour at sea level. The values of p, p_l and p_u are obtained by multiplying P, P_l and P_u respectively by

$$q = \frac{1}{2} \rho V^2 = 0.002558 (120)^2 = 36.84 \text{ lbs./sq.ft.}$$

C.25 ESTIMATION OF CRITICAL SPEED

The extreme negative value of the pressure coefficient for the conditions of this example (Figure C-1) is seen to occur near the leading edge on the upper surface as was the case with the normal surface and to have a value of -4.16. For this value of P_u, M_c is seen to be approximately 0.333 from Figure 2-2. From Figure 2-4 the speed of sound in standard air at sea level is seen to be 764 miles per hour. The local speed of sound would not be reached at this angle of attack and flap angle until the airplane approaches a speed of 0.333 x 764 = 255 miles per hour.

APPENDIX D

CHORDWISE AIR LOAD DISTRIBUTION DATA CALCULATED FOR SELECTED CLASSIFIED AIRFOILS WITH FLAP NEUTRAL

D.0 INTRODUCTION

With a knowledge of the P_o and P_{a_1} chordwise air-load distribution characteristics of any given airfoil section, the chordwise air-load distribution for the normal wing with flaps neutral may be determined simply for any attitude or load by use of equation 1.3. Hence it was considered desirable that such necessary data be calculated for the most commonly used airfoil sections and be made more readily available to the designer so as to be of optimum aid in the design procedures.

D.1 CONTENTS

The P_{a_1} and P_o chordwise air-load distribution components have been computed at sixteen chordwise stations for the following classified airfoil sections and are tabulated herein.

Table		Airfoil					
D-1	NACA	2406 4406 6406	2409 4409 6409	2412 4412 6412	2415 4415 6415	2418 4418 6418	2421 4421 6421
D-2	NACA	23006	23009 43009 63009	23012 43012 63012	23015 43015 63015	23018 43018 63018	23021 43021 63021
		N71 (t = .1154) CYH					

D.2 METHOD

Computations were made on tabular form C-2A. Instead of computing P_{a_t} as indicated in the first five lines of the table, the values for line 5 were taken directly from Table C-1.

Tables of airfoil characteristics contained in National Advisory Committee for Aeronautics Technical Reports Nos. 610 and 628 give values of x_{ac} and $c_{m_{ac}}$ which, when plotted against the thickness-chord ratio, show considerable scatter. Values from faired curves, presented in Figure D-1, were used in computing the tabulated data.

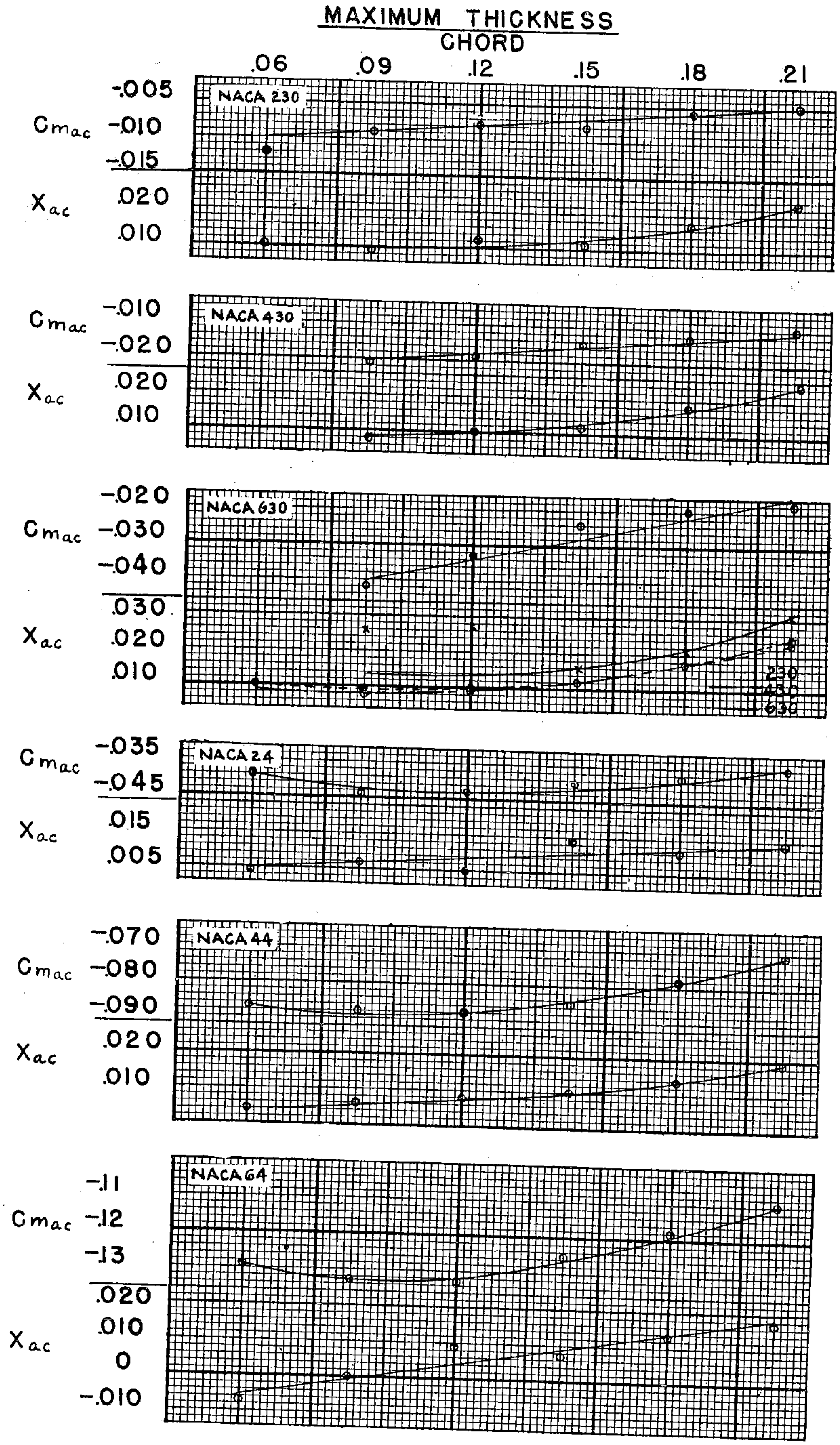


FIG. D-1

AIRFOIL SECTION CHARACTERISTICS

TABLE D-1

P_{a_1} AND P_o FOR SELECTED AIRFOILS

Airfoil Section		Chord Station x																	
		0	.0125	.025	.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.80	.90	.95	1.00
2406	P_{a_1}	0	6.02	4.60	3.21	2.59	2.23	1.75	1.44	1.25	1.09	.86	.67	.50	.36	.22	.10	.03	0
	P_c	0	-1.37	-.96	-.55	-.36	-.25	-.09	.01	.06	.10	.13	.14	.14	.12	.10	.06	.03	0
2409	P_{a_1}	0	5.95	4.40	3.24	2.67	2.30	1.81	1.50	1.28	1.12	.87	.67	.50	.36	.22	.09	.02	0
	P_o	0	-1.52	-1.03	-.63	-.43	-.30	-.11	-.01	.07	.11	.15	.15	.15	.13	.11	.07	.04	0
2412	P_{a_1}	0	5.01	4.27	3.26	2.73	2.37	1.89	1.57	1.33	1.16	.88	.68	.50	.35	.21	.07	.02	0
	P_o	0	-1.24	-.98	-.62	-.43	-.31	-.13	-.03	.05	.10	.14	.15	.14	.14	.10	.07	.04	0
2415	P_{a_1}	0	4.35	4.07	3.31	2.82	2.45	1.95	1.62	1.39	1.19	.89	.68	.50	.34	.20	.06	0	0
	P_o	0	-1.00	-.88	-.60	-.44	-.31	-.14	-.04	.03	.08	.13	.14	.14	.13	.10	.06	.05	0
2418	P_{a_1}	0	3.91	3.86	3.34	2.88	2.51	2.01	1.67	1.43	1.21	.91	.69	.49	.33	.18	.04	-.01	0
	P_o	0	-.82	-.76	-.57	-.42	-.30	-.15	-.05	.02	.07	.11	.13	.13	.12	.10	.07	.04	0
2421	P_{a_1}	0		3.70	3.37	2.95	2.59	2.08	1.74	1.48	1.26	.92	.70	.49	.32	.17	.04	-.02	0
	P_o	0		-.65	-.51	-.39	-.28	-.15	-.05	0	.05	.10	.11	.12	.11	.09	.06	.04	0
4406	P_{a_1}	0	6.01	4.60	3.20	2.58	2.22	1.74	1.44	1.24	1.09	.86	.67	.50	.36	.22	.11	.04	0
	P_o	0	-3.04	-2.15	-1.22	-.79	-.54	-.18	.02	.15	.23	.30	.30	.30	.26	.21	.12	.08	0
4409	P_{a_1}	0	5.95	4.39	3.23	2.66	2.29	1.80	1.49	1.27	1.11	.87	.67	.50	.37	.23	.10	.03	0
	P_o	0	-3.05	-2.06	-1.26	-.85	-.58	-.23	-.01	.18	.23	.29	.31	.30	.27	.20	.12	.08	0
4412	P_{a_1}	0	5.01	4.27	3.26	2.73	2.37	1.89	1.57	1.33	1.16	.88	.68	.50	.35	.21	.07	.02	0
	P_o	0	-2.53	-2.00	-1.28	-.89	-.62	-.28	-.05	.10	.20	.28	.30	.30	.28	.21	.14	.09	0
4415	P_{a_1}	0	4.35	4.07	3.31	2.82	2.45	1.95	1.62	1.39	1.19	.89	.68	.50	.34	.20	.06	0	0
	P_o	0	-2.09	-1.82	-1.26	-.91	-.65	-.30	-.08	.06	.17	.27	.30	.29	.27	.21	.14	09	0
4418	P_{a_1}	0	3.92	3.87	[illegible]	2.89	2.52	2.02	1.68	1.44	1.22	.91	.69	.49	.32	.17	.03	-.02	0
	P_o	0	-1.72	-1.59	-1.19	-.88	-.63	-.31	-.11	.03	.14	.25	.27	.28	.26	.22	.15	.10	0
4421	P_{a_1}	0		3.73	3.39	2.98	2.62	2.11	1.76	1.50	1.28	.93	.70	.48	.30	.15	.01	-.04	0
	P_o	0		-1.37	-1.09	-.83	-.62	-.32	-.13	0	.10	.20	.24	.26	.25	.20	.15	.10	0
6406	P_{a_1}	0	5.98	4.55	3.14	2.52	2.16	1.68	1.39	1.20	1.06	.84	.67	.52	.39	.27	.17	.10	0
	P_o	0	-4.49	-3.15	-1.77	-1.13	-.75	-.23	.07	.25	.37	.46	.46	.43	.36	.26	.12	.06	0
6409	P_{a_1}	0	5.93	4.37	3.19	2.62	2.25	1.76	1.47	1.25	1.10	.86	.67	.51	.38	.25	.14	.07	0
	P_o	0	-4.59	-3.09	-1.87	-1.26	-.85	-.30	.01	.21	.34	.45	.47	.45	.38	.29	.15	.09	0
6412	P_{a_1}	0	4.99	4.25	3.24	2.70	2.34	1.87	1.56	1.31	1.15	.88	.68	.50	.36	.22	.10	.04	0
	P_o	0	-3.77	-2.98	-1.90	-1.31	-.92	-.40	-.07	.16	.29	.43	.45	.45	.40	.32	.19	.12	0
6415	P_{a_1}	0	4.35	4.06	3.30	2.81	2.44	1.95	1.62	1.38	1.19	.89	.68	.50	.34	.20	.07	.01	0
	P_o	0	-3.05	-2.64	-1.83	-1.31	-.94	-.43	-.12	.10	.25	.39	.43	.43	.39	.31	.20	.13	0
6418	P_{a_1}	0	3.91	3.87	3.35	2.89	2.52	2.02	1.68	1.44	1.22	.91	.69	.49	.32	.17	.03	-.02	0
	P_o	0	-2.59	-2.39	-1.79	-1.33	-.96	-.47	-.15	.05	.21	.37	.40	.42	.39	.32	.22	.15	0
6421	P_{a_1}	0		3.73	3.39	2.98	2.62	2.11	1.76	1.50	1.28	.93	.70	.48	.30	.15	.01	-.04	0
	P_o	0		-2.11	-1.68	-1.29	-.96	-.49	-.20	.01	.16	.33	.36	.40	.37	.31	.22	.15	0

TABLE D-2

P_{a_1} AND P_o FOR SELECTED AIRFOILS

Airfoil Section		Chord Station x																	
		0	.0125	.025	.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.80	.90	.95	1.00
23006	P_{a_1}	0	6.03	4.63	3.24	2.62	2.26	1.77	1.46	1.27	1.11	.86	.67	.50	.34	.20	.07	0	0
	P_o	0	-1.80	-.97	-.22	.08	.21	.24	.17	.13	.09	.07	.04	.03	.01	.02	.02	.03	0
23009	P_{a_1}	0	5.96	4.41	3.25	2.68	2.31	1.82	1.51	1.28	1.13	.87	.67	.50	.35	.21	.08	.01	0
	P_o	0	-1.72	-.85	-.21	.06	.20	.22	.16	.13	.08	.06	.04	.02	.01	.01	.02	.03	0
23012	P_{a_1}	0	5.01	4.28	3.28	2.74	2.38	1.90	1.58	1.34	1.17	.88	.68	.50	.34	.20	.06	0	0
	P_o	0	-1.33	-.78	-.20	.06	.18	.19	.13	.10	.07	.05	.03	.02	.01	.01	.03	.03	0
23015	P_{a_1}	0	4.36	4.08	3.32	2.84	2.46	1.97	1.64	1.40	1.20	.90	.68	.49	.33	.18	.05	-.01	0
	P_o	0	-1.06	-.68	-.21	.03	.15	.17	.11	.08	.05	.04	.02	.02	.01	.02	.02	.03	0
23018	P_{a_1}	0	3.92	3.89	3.36	2.91	2.54	2.04	1.69	1.45	1.23	.91	.69	.49	.31	.16	.01	-.03	0
	P_o	0	-.86	-.58	-.20	.01	.14	.14	.10	.06	.04	.03	.02	.02	.01	.02	.04	.04	0
23021	P_{a_1}	0		3.74	3.42	3.01	2.64	2.13	1.78	1.51	1.29	.93	.70	.48	.29	.13	-.01	-.07	0
	P_o	0		-.51	-.21	-.01	.11	.11	.06	.03	.01	.02	.01	.01	.02	.03	.04	.06	0
43009	P_{a_1}	0	5.96	4.41	3.24	2.68	2.31	1.81	1.50	1.28	1.12	.87	.67	.50	.36	.22	.08	.02	0
	P_o	0	-3.63	-1.79	-.45	.12	.40	.46	.33	.25	.18	.14	.09	.05	.01	.03	.03	.04	0
43012	P_{a_1}	0	5.01	4.27	3.27	2.73	2.37	1.90	1.58	1.33	1.17	.88	.68	.50	.34	.20	.07	.01	0
	P_o	0	-2.82	-1.65	-.46	.09	.35	.39	.27	.21	.14	.12	.08	.05	.02	.04	.04	.05	0
43015	P_{a_1}	0	4.36	4.07	3.32	2.83	2.46	1.96	1.63	1.39	1.19	.90	.68	.49	.34	.19	.05	-.01	0
	P_o	0	-2.22	-1.43	-.45	.04	.29	.34	.22	.15	.11	.09	.06	.05	.01	.04	.05	.07	0
43018	P_{a_1}	0	3.93	3.89	3.37	2.92	2.54	2.04	1.70	1.45	1.23	.92	.69	.48	.31	.15	.01	-.04	0
	P_o	0	-1.81	-1.22	-.45	0	.26	.29	.17	.10	.08	.07	.04	.05	.03	.06	.08	.08	0
43021	P_{a_1}	0		3.75	3.42	3.01	2.65	2.13	1.78	1.52	1.29	.93	.70	.48	.29	.13	-.02	-.07	0
	P_o	0		-1.07	-.45	-.04	.20	.23	.11	.05	.03	.06	.04	.03	.04	.07	.06	0	0
63009	P_{a_1}	0	5.97	4.42	3.27	2.70	2.33	1.83	1.52	1.29	1.13	.88	.67	.49	.35	.20	.06	-.01	0
	P_o	0	-5.62	-2.82	-.79	.11	.51	.63	.46	.37	.28	.21	.17	.12	.06	.08	.08	.11	0
63012	P_{a_1}	0	5.02	4.29	3.29	2.76	2.39	1.92	1.60	1.35	1.18	.89	.68	.49	.33	.18	.05	-.01	0
	P_o	0	-4.29	-2.55	-.75	.09	.47	.53	.37	.29	.20	.18	.13	.11	.06	.09	.09	.11	0
63015	P_{a_1}	0	4.37	4.09	3.34	2.85	2.48	1.98	1.65	1.41	1.21	.90	.68	.49	.32	.17	.03	-.03	0
	P_o	0	-3.37	-2.21	-.74	.02	.39	.47	.31	.22	.16	.15	.11	.09	.06	.09	.10	.13	0
63018	P_{a_1}	0	3.94	3.90	3.39	2.93	2.56	2.06	1.71	1.47	1.24	.92	.69	.48	.30	.14	-.01	-.06	0
	P_o	0	-2.76	-1.88	-.73	-.02	.33	.39	.24	.14	.11	.10	.08	.09	.07	.11	.14	.15	0
63021	P_{a_1}	0		3.78	3.46	3.05	2.69	2.17	1.81	1.54	1.31	.94	.70	.47	.27	.10	-.06	-.11	0
	P_o	0		-1.64	-.72	-.10	.23	.30	.15	.06	.03	.08	.06	.08	.09	.15	.19	.21	0
N71	P_{a_1}	0	5.00	4.26	3.26	2.72	2.36	1.89	1.57	1.33	1.16	.88	.68	.50	.35	.21	.08	.02	0
	P_o	0	-.84	-.66	-.42	-.29	-.20	-.09	-.02	.04	.07	.10	.10	.10	.09	.07	.05	.03	0
CYH	P_{a_1}	0	5.00	4.26	3.26	2.72	2.36	1.89	1.57	1.33	1.16	.88	.68	.50	.35	.21	.08	.02	0
	P_o	0	-2.20	-1.73	-1.07	-.64	-.34	.06	.29	.43	.49	.47	.32	.13	-.10	-.17	-.12	-.06	0

APPENDIX E

DATA FOR USE IN DETERMINING THE CHORDWISE AIR LOAD DISTRIBUTION OVER AN AIRFOIL WITH FLAP DEFLECTED.

E.0. GENERAL

The determination of the chordwise air-load distribution over an airfoil with flap deflected is more complex than is that for the case with the flap neutral. Where in the case of a given airfoil, the latter distribution may be explicitly determined at any attitude from a knowledge of the P_o, P_{a_1}, and c_n values, the flapped distribution is in addition affected by such parameters as

(1) flap-chord ratio E

(2) flap deflection δ

(3) type of flap (plain, split, serially-hinged, etc.)

The great number of possible combinations of these parameters precludes the possibility of presenting completely calculated data applicable to any given combination of parameters in a simple tabular or graphical manner. It was considered, however, that design calculation procedure could be aided by presenting the necessary data in a more readily available component form from which the resultant chordwise air-load distribution may be computed for any particular combination of the several parameters. Such data are presented herein in Table E-1.

E.1. DESCRIPTION OF DATA

The component partial distributions presented herein have been kept as general as possible by reducing the number of variables to two which are independent of the airfoil section, and by tabulating the data in a generally applicable form.

The data presented in Table E-1 are values of $P_{b\delta_1}$, the increment of basic normal-pressure coefficient due to flap deflection when $c_{n_{b\delta}} = 1.0$. The distribution of total normal-pressure coefficient for an airfoil with flap displaced from neutral is then calculated from the formula:

$P = P_o + c_{n_{b\delta}} P_{b\delta_1} + (c_n' - c_{n_{b\delta}}) P_{a_1}$ (from equation 3.10)

The terms appearing in this equation are obtained in the following manner:

P_o from table C-2A or Appendix D

P_{a_1} from table C-2A and /or table E-1

$c_{n_{b\delta}}$ $= \frac{\Upsilon_m}{G} \Delta c_m$ (equation 3.3)

$\frac{\Upsilon_m}{G}$ is obtained from figure 3-1, and Δc_m is determined independently from results of force tests.

c_n' $= c_n + E\, c_{n_f} (1-\cos\delta)$ (equation 3.7)

c_n is determined for the desired flap deflection and angle of attack from aerodynamic force test data and c_{n_f} is found from equation 3.6 and the data contained in the pertinent charts of chapter 3.

$P_{b\delta_1}$ from table E-1, interpolating if necessary for values of E or δ lying between those given.

In order to obtain $P_{b\delta_1}$ values for the flapped airfoil, it was necessary to consider some stations aft of the hinge which do not appear on tables C-1 or C-2A. Since values of

P_{a_1} over the flap are substantially independent of airfoil thickness, average values were obtained for each flap station represented by the $P_{b\delta_1}$ data in Table E-1. The values of P_{a_1} corresponding to those for a 12% airfoil were taken as representing such an average and are given at the bottom of table E-1. They are considered sufficiently accurate inasmuch as P ordinarily varies only slightly with P_{a1} at points aft of the flap hinge.

By this method, plain and split flaps are treated in a similar manner. According to reference 5, at any given flap deflection of 40° or greater, plain and split flaps of equal flap-chord ratios give approximately the same chordwise distributions. The difference incurred at small angles have not been treated.

TABLE E-1

$P_{b\delta_1}$ DISTRIBUTION FOR VARIOUS COMBINATIONS OF E AND δ

$P_{b\delta_1}$

E	δ																			
CHORD STA. X		.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.80	.90	.93	.94	.95	.96	.98	1.00
.05	15°	.15	.19	.22	.28	.34	.40	.45	.56	.69	.85	1.06	1.37	2.60	4.06	5.35	8.74	4.96	1.90	0
	20°	.15	.19	.22	.28	.34	.40	.45	.56	.69	.85	1.06	1.38	2.68	3.95	4.64	5.83	4.23	2.65	0
	30°	.15	.19	.22	.28	.34	.40	.45	.56	.69	.85	1.06	1.39	2.13	3.07	3.63	4.50	4.34	3.25	0

CHORD STA. X		.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.80	.90	.925	.93	.94	.96	.98	1.00
.075	15°	.15	.20	.23	.30	.35	.42	.47	.58	.72	.88	1.11	1.53	3.37	7.41	5.90	4.25	2.29	1.07	0
	20°	.15	.20	.23	.30	.35	.42	.47	.58	.72	.88	1.11	1.51	3.22	4.93	4.49	3.66	2.62	1.87	0
	30°	.15	.20	.23	.30	.35	.42	.47	.58	.72	.88	1.11	1.50	2.63	3.81	3.89	3.71	3.09	2.34	0

CHORD STA. X		.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.80	.85	.88	.90	.92	.95	.98	1.00
.10	15°	.16	.21	.24	.31	.37	.43	.49	.60	.75	.91	1.16	1.66	2.26	3.50	6.04	3.52	1.72	.65	0
	20°	.16	.21	.24	.31	.37	.43	.49	.60	.75	.91	1.16	1.65	2.25	2.97	4.05	2.99	2.09	1.35	0
	30°	.16	.21	.24	.31	.37	.43	.49	.60	.75	.91	1.16	1.62	2.00	2.45	3.13	3.07	2.50	1.75	0

CHORD STA. X		.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.77	.85	.87	.88	.91	.94	.97	1.00
.15	15°	.17	.22	.25	.31	.38	.44	.50	.64	.80	1.02	1.38	1.74	5.00	3.75	2.90	1.78	1.10	.53	0
	30°	.17	.22	.25	.31	.38	.44	.50	.64	.80	1.02	1.38	1.69	2.65	2.70	2.60	2.20	1.85	1.40	0
	45°	.17	.22	.25	.31	.38	.44	.50	.54	.80	1.02	1.38	1.58	2.30	2.45	2.50	2.43	2.15	1.70	0
	60°	.17	.22	.25	.31	.38	.44	.50	.64	.80	1.02	1.28	1.54	2.20	2.37	2.45	2.47	2.32	1.87	0

CHORD STA. X		.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.70	.72	.80	.82	.84	.88	.92	.96	1.00
.20	15°	.18	.23	.27	.33	.41	.48	.54	.69	.88	1.14	1.64	1.75	4.35	3.30	2.55	1.56	.96	.47	0
	30°	.18	.23	.27	.33	.41	.48	.54	.69	.88	1.14	1.56	1.70	2.35	2.38	2.25	1.93	1.63	1.22	0
	45°	.18	.23	.27	.33	.41	.48	.54	.69	.88	1.14	1.51	1.59	2.10	2.19	2.20	2.10	1.88	1.45	0
	60°	.18	.23	.27	.33	.41	.48	.54	.69	.88	1.14	1.45	1.55	1.95	2.06	2.14	2.16	2.00	1.62	0

ADMT.-1

CHORD STA. X		.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.68	.75	.78	.80	.85	.90	.95	1.00
.25	15°	.20	.24	.28	.38	.44	.51	.58	.76	.96	1.29	1.77	4.00	3.00	2.28	1.41	.85	.42	0
	30°	.20	.24	.28	.38	.44	.51	.58	.76	.96	1.29	1.71	2.23	2.15	2.00	1.74	1.45	1.09	0
	45°	.20	.24	.28	.38	.44	.51	.58	.76	.96	1.29	1.60	1.95	1.98	1.97	1.91	1.69	1.30	0
	60°	.20	.24	.28	.38	.44	.51	.58	.76	.96	1.25	1.54	1.80	1.88	1.92	1.96	1.80	1.46	0

CHORD STA. X		.05	.075	.10	.15	.20	.25	.30	.40	.50	.60	.63	.70	.73	.76	.82	.88	.94	1.00
.30	15°	.20	.26	.30	.38	.47	.55	.54	.84	1.09	1.55	1.79	3.72	2.76	2.11	1.30	.79	.40	0
	30°	.20	.26	.30	.38	.47	.55	.64	.84	1.09	1.52	1.72	2.15	2.00	1.85	1.60	1.35	1.00	0
	45°	.20	.26	.30	.38	.47	.55	.64	.84	1.09	1.48	1.60	1.83	1.85	1.83	1.75	1.55	1.20	0
	60°	.20	.26	.30	.38	.47	.55	.64	.84	1.09	1.44	1.53	1.70	1.76	1.79	1.80	1.65	1.33	0

CHORD STA. X		.05	.075	.10	.15	.20	.25	.30	.40	.50	.54	.60	.64	.68	.76	.84	.92	1.00
.40	15°	.25	.31	.36	.46	.56	.67	.78	1.04	1.52	1.84	3.35	2.49	1.90	1.14	.69	.34	0
	30°	.25	.31	.36	.46	.56	.67	.78	1.04	1.52	1.74	2.05	1.81	1.65	1.41	1.18	.86	0
	45°	.25	.31	.36	.46	.56	.67	.78	1.04	1.52	1.50	1.70	1.68	1.61	1.52	1.35	1.04	0
	60°	.25	.31	.36	.46	.56	.67	.78	1.04	1.38	1.50	1.60	1.60	1.58	1.57	1.45	1.15	0

CHORD STA. X		.025	.05	.075	.10	.15	.20	.25	.30	.40	.45	.50	.55	.60	.70	.80	.90	1.00
.50	15°	.20	.30	.37	.44	.55	.70	.84	1.00	1.45	1.92	3.15	2.31	1.75	1.03	.62	.30	0
	30°	.20	.30	.37	.44	.56	.70	.84	1.00	1.46	1.80	2.00	1.72	1.52	1.28	1.05	.78	0

P_{a_1}

CHORD STA. X	.50	.54	.55	.60	.63	.64	.68	.70	.72	.73	.75	.76	.77	.78	.80
P_{a_1}	.68	.60	.58	.50	.45	.43	.36	.34	.30	.29	.26	.25	.24	.22	.20
CHORD STA. X	.82	.84	.85	.87	.88	.90	.91	.92	.925	.93	.94	.95	.96	.97	.98
P_{a_1}	.17	.15	.13	.10	.09	.06	.05	.04	.03	.02	.01	0	0	0	0