NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE 2211

AN APPROXIMATE METHOD OF CALCULATING PRESSURES IN THE TIP REGION OF A RECTANGULAR WING OF CIRCULAR-ARC SECTION AT SUPERSONIC SPEEDS

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SUMMARY

An approximate method of calculating pressures within the tip region of a rectangular wing having a symmetrical circular-arc profile is developed by means of which it is possible to determine the pressures at supersonic speeds to a higher degree of accuracy than with the usual linearized methods. The method consists essentially of applying tip corrections realized from linear theory to the exact pressures calculated for the two-dimensional flow region and then taking into account the actual region influenced by the wing tip. A comparison of some calculated pressure distributions with experimental results indicates good agreement for regions not affected by shock-boundary-layer interactions.

INTRODUCTION

Comparisons between theory and experiment indicate that, for rectangular wings of moderate thickness or at moderately high angles of attack, the pressure distributions calculated by linear theory are not in good agreement with experimental results. If viscous effects are neglected, the pressure distributions for the parts of the wings where the flow is two-dimensional can be calculated with near exactness by the method of characteristics. Other methods that provide less accuracy than the method of characteristics but greater accuracy than the linear theory are the shock-expansion technique (neglect of the reflections from initial shock that are considered in the characteristics method) and Busemann's second-order-approximation theory. For moderate Mach numbers and thickness ratios the shock-expansion method gives results which are nearly identical with those of the characteristics method and which are in very good agreement with experiment when no strong viscous effects are present. For the region influenced by the wing tip, no theories of corresponding precision are as yet available.

In reference 1, Bonney introduces a method of calculating the aerodynamic characteristics of rectangular wings with an accuracy greater than that possible with usual linear methods. The method consists essentially of computing the pressures in the two-dimensional flow regions by means of Busemann's second-order approximation and assuming that, in the region influenced by the wing tip, the differences in pressure between the wing surfaces decrease from those in the two-dimensional region to zero at the wing tip in accordance with the rate of loss in lift found by linear theory for an infinitely thin flat plate. The method is not suitable for the prediction of actual pressures in the wing tip region, however, because the increments in pressure due to wing thickness largely cancel and need not be computed for the tip region in order to determine the lift and moment characteristics of the wing.

The present paper introduces an approximate method by means of which it is possible to calculate the pressures in the tip region of a rectangular wing with a symmetrical circular-arc section to a higher degree of accuracy than is possible with the usual linear methods. The method is basically similar to Bonney's method, but the two-dimensional pressure distributions are calculated by more exact methods than Busemann's second-order approximation, and tip corrections realized from linear theory are applied to the pressure increments due to both angle of attack and airfoil thickness. In addition the actual flow field influenced by the wing tip is taken into account rather than the field between the free-stream Mach line and the wing tip as is done in linear theory and Bonney's method. A brief comparison of some calculated pressure distributions with experimental results for a 9-percent-thick wing at a Mach number of 1.62 is included.

SYMBOLS

P	pressure coefficient $\left(\frac{p - p_o}{q_o}\right)$
ΔP_α	pressure-coefficient increment due to angle of attack
ΔP_σ	pressure-coefficient increment due to airfoil thickness and shape
p	local static pressure
p_o	free-stream static pressure
q_o	free-stream dynamic pressure
x_{cp}	center of pressure, in fraction of chord from leading edge

c chord of airfoil

t/c maximum thickness ratio of airfoil

$c_{m_{c/2}}$ section pitching-moment coefficient $\left(\frac{m_{c/2}}{qc^2}\right)$

c_n section normal-force coefficient $\left(\frac{n}{qc}\right)$

$m_{c/2}$ section pitching moment about 50-percent-chord point, positive when it tends to rotate leading edge of section upward

n section normal force, positive upward

M local Mach number

M_o free-stream Mach number

α angle of attack

$\beta = \sqrt{M_o^2 - 1}$

x, y, z coordinates of mutually perpendicular system of axes

ξ, η coordinates which replace x and y, respectively, used to indicate origin of source line

u x-component of disturbance velocity, positive in flight direction

dz/dx slope of airfoil surface

K_1, K_2, K_3, K_4, K_5 constants used to determine location of point in distorted flow field (defined in fig. 5)

V free-stream velocity in flight direction

Subscripts:

1 pressure coefficient in region influenced by wing tip determined by linear theory

2 pressure coefficient in two-dimensional flow region determined by linear theory

3 pressure coefficient in two-dimensional flow region determined by use of oblique-shock theory and Prandtl-Meyer's equations for expansion of two-dimensional supersonic flow

Subscript notation for u and $\frac{dz}{dx}$ indicates the origin of source line in terms of x and y, respectively, except when noted otherwise.

Subscript b on x and y in equation (5) indicates location of boundary between two-dimensional and three-dimensional flow regions.

Primed symbols:

Primes indicate the location or coordinates of a point in the distorted flow field.

ANALYSIS

In the linearization of the three-dimensional equations of potential flow the following assumptions are made: the total-pressure losses across shocks are negligible, disturbances propagate on the airfoil along straight lines inclined at the Mach angle of the free stream, and no interaction exists between the thickness and angle-of-attack effects. In order to improve the accuracy of the pressure-distribution calculations for the tip region these simplifying assumptions must be modified to include conditions that are more nearly in agreement with actual flow conditions.

The effects of the total-pressure losses across the leading-edge shock and of the interaction between the pressures due to thickness and angle of attack are approximated by the use of pressures that are calculated for the two-dimensional-flow region by methods involving oblique-shock theory and the Prandtl-Meyer equations for the expansion of a two-dimensional supersonic flow. If the method of superposition of linear theory is followed, the two-dimensional pressures are divided into two increments. (See fig. 1.) Thus,

$$P_3 = \Delta P_{\sigma_3} + \Delta P_{\alpha_3} \tag{1}$$

where the pressure increment due to thickness ΔP_σ is taken as constant for a given airfoil and the effects due to interaction are concentrated in the pressure increment due to angle of attack ΔP_α. The pressure increment due to interaction between angle-of-attack and thickness effects may be defined as the difference in pressure found when the pressure at any point on the airfoil is computed by two methods. In the first method, which gives results that are in very good agreement with experiment, the pressure is computed by the usual shock-expansion technique in which the thickness and angle-of-attack effects are considered together. In the second method, the pressure is computed by the shock-expansion theory, but the pressure increment due to thickness is calculated separately at $\alpha = 0^{\circ}$ and to this increment is added the increment in pressure calculated for a thin flat plate at the appropriate angle of attack. The assumption that the interaction effect is concentrated in the increment in pressure due to angle of attack was made in order to reduce slightly the amount of required calculations. Actually, some calculations indicated that a more reasonable assumption, such as one in which the interaction effect was equally divided between the increments in pressure due to thickness and angle of attack, would generally lead to results in slightly better agreement with experiment. The actual differences in pressure resulting from the use of the two methods of calculation are usually negligible as regards the calculation of pressures in the tip region.

According to linear theory, the ratio of the pressure in the tip region of a rectangular wing with a symmetrical circular-arc profile to the pressure at the corresponding chordwise station in the two-dimensional flow region is given, for $\alpha = 0^{\circ}$, by the expression

$$\frac{\Delta P_{\sigma_1}}{\Delta P_{\sigma_2}} = 1 + \frac{-\frac{1}{\pi\beta}\left(\pi - \cos^{-1}\beta\frac{y}{x}\right) + \frac{2}{\pi}\frac{y}{c}\left[\cosh^{-1}\frac{x}{\beta|y|} + \frac{1}{\beta}\frac{x}{y}\left(\pi - \cos^{-1}\beta\frac{y}{x}\right)\right]}{\frac{1}{\beta}\left(1 - 2\frac{x}{c}\right)} \qquad (2)$$

where the equation is valid for $-1 \leqq \beta\frac{y}{x} \leqq 1$. (See appendix for derivation of equation and fig. 2 for definition of pertinent symbols.) The corresponding pressure ratio due solely to the angle of attack of a flat plate of rectangular plan form (reference 2) is

$$\frac{\Delta P_{\alpha_1}}{\Delta P_{\alpha_2}} = \frac{1}{\pi}\cos^{-1}\left(1 + 2\beta\frac{y}{x}\right) \qquad (3)$$

where

$$-1 \leqq \beta\frac{y}{x} \leqq 1$$

When the method of superposition of linear theory is followed but the two-dimensional pressure coefficient calculated by rigorous methods as a base is used, the theoretical pressure coefficient anywhere in the tip region is defined by

$$P = \Delta P_{\sigma_3} \frac{\Delta P_{\sigma_1}}{\Delta P_{\sigma_2}} + \Delta P_{\alpha_3} \frac{\Delta P_{\alpha_1}}{\Delta P_{\alpha_2}} \tag{4}$$

Equation (4) may be regarded as a correction of the pressure coefficient in the tip region P_1 due to wing thickness or angle of attack as calculated by linear theory by the corresponding ratio of the exact two-dimensional pressure to the linear two-dimensional pressure $\frac{P_3}{P_2}$. It may be noted that equation (2) is indeterminate at $\frac{y}{c} = 0$ (at the wing tip), but the pressures at the station actually have a finite value which varies along the chord. The equation also changes in sign from minus to plus infinity at the chordwise station $\frac{x}{c} = 0.50$ (for the circular-arc profile), and thus a discontinuity results in the pressure distributions derived from equation (4) if the exact two-dimensional pressure coefficient due to thickness does not change in sign at exactly the same chordwise point as in the linear theory. When such pressure discontinuities do occur they are ascribed to inaccuracies of the method and are neglected in the fairing of the pressure-distribution curves. Usually, the discontinuities are confined to a region less than 5 or 6 percent of the chord in length and located at the midchord point of the wing. For $\beta\frac{y}{x} = -1$ and 1, equation (2) reduces to 1 and 0, respectively. Equation (3), on the other hand, indicates a conical flow field with zero pressure ratio $\frac{\Delta P_{\alpha_1}}{\Delta P_{\alpha_2}}$ at the wing tip.

Calculation of the distortion of the field influenced by the wing tip to allow for the propagation of disturbances along Mach lines dependent upon the local velocities over the wing is somewhat more difficult and can be accomplished only by approximation. Among the methods of distorting the tip field that were investigated are: a linear distortion in a spanwise direction, with no changes being assumed in pressure at the wing tip; a linear distortion in a chordwise direction with the pressures at the trailing edge held constant; and distortions in both the spanwise and chordwise directions based on various concepts that disturbances propagate along curved Mach lines. Most of the techniques investigated gave results that were in fair agreement with experiment and thus indicated the purely arbitrary nature of the distortions, even though in some instances they appear to be linked with the basic assumptions of some of the linearized theories. The best and most consistent agreement with experiment was obtained by transforming the field by the method described in the following paragraphs.

According to Jones' linear theory (see appendix), the pressure at any point $L(x,y)$ (fig. 3) on an arbitrary spanwise station EF is determined by the pressure due to the action of the line sources and sinks representing the wing surface from the leading edge to the point $L(x,y)$ and by the decrease in pressure due to the action of the line sources and sinks representing the tip and originating between A and G. The pressure disturbances are assumed to propagate along curved lines similar to AD and the pressure decrements resulting from the tip effects calculated from the sources and sinks from A to G are assumed to be felt at the reference station at the point $L'(x',y)$. By equation (4), the pressure at the point $L(x,y)$ is corrected so that, at the same chordwise station, the pressure at the edge of the region influenced by the wing tip is equal to the exact two-dimensional pressure. The decrease in pressure due to the wing tip is taken as the difference between the exact two-dimensional pressure at the chordwise station corresponding to the point $L(x,y)$ and the corrected pressure at the point $L(x,y)$.

Inasmuch as the curved Mach line AD from the leading edge of the tip lies at the boundary of the two-dimensional flow region, its location can be determined accurately from the exact two-dimensional calculations from the following relationship derived by Frick and Boyd of the Ames Aeronautical Laboratory:

$$\frac{y_b}{c} = \int_0^{\frac{x_b}{c}} \frac{1}{\sqrt{{M_3}^2 - 1}} \, d\left(\frac{x}{c}\right) \tag{5}$$

where M_3 denotes the local two-dimensional Mach number at the point x/c. The curved Mach line GL' is assumed to be equal in slope and curvature to the corresponding chordwise section of AD. In other words, the curve GL' is found by translating AD toward the right until it passes through G. The determination of the coordinates of a large number of points $L'(x',y)$ is simplified by first finding the locus of the points $T(x,y)$ (fig. 4) that corresponds to those points $F'(x',y)$ which lie on the trailing edge of the wing DC. At each spanwise station EF the desired point is located graphically by translating AD to the right until it passes through F or F', thus locating G. From G a line is drawn parallel to the Mach line AB. The intersection of this line GT with the station line EF gives the locus of the desired point. It should be noted that $T(x,y)$ is the point required to determine the pressure at the trailing-edge point $F'(x',y)$; therefore, the procedure for locating the point is reversed from that for locating a transposed point $L'(x',y)$. Typical loci of points $T(x,y)$ are shown in figure 4. The ordinate of the point $L'(x',y)$ (fig. 5(a)) in terms of the ordinate of $L(x,y)$ and some graphically determined constants is

$$\frac{x'}{c} = 1 - \frac{K_2}{c}\left[1 - \frac{1}{K_3/c}\left(\frac{x}{c} - \frac{K_1}{c}\right)\right] \tag{6}$$

This procedure amounts to expanding the region NT linearly to N'F'. The pressure at the point $L'(x',y)$ is then found from

$$P_{at\ x',y} = \left(P_3\right)_{at\ x',y} - \left(P_3\right)_{at\ x,y} + \left[\left(\Delta P_{\sigma_3}\right)_{at\ x,y} \frac{\Delta P_{\sigma_1}}{\Delta P_{\sigma_2}} + \left(\Delta P_{\alpha_3}\right)_{at\ x,y} \frac{\Delta P_{\alpha_1}}{\Delta P_{\alpha_2}}\right] \tag{7}$$

In order to determine the pressures at spanwise stations lying inboard of the point where the line or loci of the points $T(x,y)$ intersects the wing trailing edge (for example, fig. 4, lower surface), a subterfuge must be used and pressure ratios must be computed for points behind the model. This computation is carried out by assuming that equations (2) and (3) are valid without change and by computing the exact two-dimensional pressure for the point as if the upper and lower surface extended without a break and with the same curvature beyond the actual wing trailing edge.

When a slightly greater discrepancy between the calculated and experimental pressures, probably not exceeding twice that resulting from the use of the preceeding method, can be tolerated (in general the changes in pressures and in integrated characteristics will be small), the calculations may be simplified by assuming that the points $T(x,y)$ coincide with the trailing-edge points $F'(x',y)$ and no trailing-edge distortion is present. With equal accuracy for the outboard stations and considerably better accuracy for stations approaching the two-dimensional region as compared to the previous simplified method, the calculations may be simplified by distorting the tip region linearly in a spanwise direction with the tip pressures held constant. In this case the pressures are calculated directly from equation (4) in which the ratios $\frac{\Delta P_{\sigma_1}}{\Delta P_{\sigma_2}}$ and $\frac{\Delta P_{\alpha_1}}{\Delta P_{\alpha_2}}$ are computed for the point $P(x,y)$ and are applied at the point $P'(x,y')$. The relation between the initial and transposed ordinates is

$$\frac{y'}{c} = \frac{y}{c}\frac{K_4}{K_5} \tag{8}$$

where the quantities K_4 and K_5 are defined in figure 5(b).

Because the method of calculating pressures in wing-tip regions that is presented in this paper requires the use of Mach lines, it is obviously restricted to cases in which the flow over the airfoil is always supersonic. The technique can also be applied to swept untapered wings of constant section thickness ratio, but this application has not been carried out herein because, in most practical cases, disturbances from the apex of the swept leading edge will affect part of the region influenced by the wing tip. The pressures in the field influenced by the apex of the leading edge or on wings having other than circular-arc sections can similarly be determined. In all cases only the form of equation (2) and possibly equation (3) will change. It should also be noted that the accuracy of the pressures approximated for the tip region depend directly upon the accuracy with which the two-dimensional pressures can be determined.

COMPARISON WITH EXPERIMENT

A comparison between some pressure distributions calculated by the method of this paper with experimental pressure distributions is presented in figure 6 for a rectangular wing having a symmetrical circular-arc profile, 9-percent thick. At the Mach number used for this comparison ($M_0 = 1.62$) the station $\frac{y}{c} = -0.282$ lies approximately in the middle of the region influenced by the wing tip. The agreement between the calculated and experimental values is generally good for regions not affected by shock-boundary-layer interaction or by a slight misalinement of the upper surface of the flap. At the higher angles of attack a slight discrepancy exists between the calculated and experimental pressures over the forward part of the wing theoretically not influenced by the wing tip. This discrepancy is believed to be due to the occurrence of wing twist during the tests that was not accounted for by the experimental method of measuring angles of incidence.

Additional calculated pressure distributions for other spanwise stations are presented in figures 7 and 8. The variation with spanwise location and angle of attack of the aerodynamic coefficients obtained by mechanical integration of the pressure distribution curves of figures 7 and 8 are shown in figure 9. For comparison the section coefficients determined by linear methods are included in the latter figure. At station $\frac{y}{c} = -0.282$, the values of c_n and c_m calculated by the present method for $\alpha = 4.55^\circ$ (integrated from fig. 6) were 0.174 and 0.028, respectively, and the calculated center of pressure was at the 34.0-percent chord point. The corresponding experimental values of c_n and c_m were 0.200 and 0.031 with the center of pressure at the 34.5-percent-chord point. A brief comparison of some of the aerodynamic characteristics obtained by the present method with those determined by Bonney's technique for a wing of aspect ratio 2 indicated very good agreement in normal-force coefficients and in pitching-moment coefficient at low angles of attack. At higher angles of attack, the present calculations show a forward movement of the center of pressure with α that is not predicted by Bonney's method.

CONCLUDING REMARKS

An approximate method has been derived for calculating pressures within the tip region of a rectangular wing having a circular-arc profile. Comparison of results obtained by this method and by linear theory with experimental results indicates that the approximate method is more accurate than the linear theory for predicting these pressures at supersonic speeds.

Langley Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Air Force Base, Va., July 28, 1949

APPENDIX

PRESSURE RATIOS DUE TO WING THICKNESS AND ANGLE OF ATTACK

According to linear theory, the pressure coefficient at any point on a fixed wing is given by

$$P = \Delta P_\sigma + \Delta P_\alpha \tag{A1}$$

where ΔP_σ is the pressure on the surface of the given wing at zero angle of attack and ΔP_α is the pressure on the surface of a flat plate of the given plan form at the angle of attack α. The equation can be rewritten for convenience in obtaining the derivations in the main body of this paper as

$$P = \Delta P_{\sigma_2} \frac{\Delta P_{\sigma_1}}{\Delta P_{\sigma_2}} + \Delta P_{\alpha_2} \frac{\Delta P_{\alpha_1}}{\Delta P_{\alpha_2}} \tag{A2}$$

where P denotes the pressure in the tip region as calculated by the method of this paper and the subscripts 1 and 2 indicate the pressures in the tip and two-dimensional flow regions, respectively, as calculated by linear methods. The ratio $\frac{\Delta P_{\alpha_1}}{\Delta P_{\alpha_2}}$ was first derived by Busemann (reference 2) by conical-flow methods and may be expressed as

$$\frac{\Delta P_{\alpha_1}}{\Delta P_{\alpha_2}} = \frac{1}{\pi} \cos^{-1}\left(1 + 2\beta\frac{y}{x}\right) \tag{A3}$$

where

$$-1 \leqq \beta\frac{y}{x} \leqq 1$$

For a wing of rectangular plan form and circular-arc section, the pressure due to thickness can be derived most conveniently by the use of semi-infinite line sources and sinks by the method of Jones (reference 3). Figure 2 shows the location and positive directions of the system of axes used in the derivation. The semi-infinite line sources representing the airfoil surface are assumed to originate at the spanwise station -h and extend to the right. The line sources of equal strength but opposite sign required to simulate the square wing tip are assumed to originate at station $y = 0$ and extend to the right. Station -h is taken sufficiently distant from station $y = 0$ so that the Mach cone from the point $x = 0, y = -h$ will not overlap that from the wing tip. From equation (A4) of reference 4, the disturbance velocity on the wing may be expressed by

$$u = u_{0,-h} - \frac{1}{D}u_{0,-h} - u_{0,0} + \frac{1}{D}u_{0,0} \tag{A4}$$

where the subscript notation indicates the origin of the source line and the u-expressions are given by

$$\left.\begin{aligned} u_{\xi,\eta}(x,y) &= \frac{V}{\pi\beta}\frac{dz}{dx}\left[\pi - \cos^{-1}\frac{\beta(y-\eta)}{x-\xi}\right] \\ \frac{1}{D}u_{\xi,\eta}(x,y) &= -\frac{V}{\pi}(y-\eta)\frac{d^2z}{dx^2}\left\{\cosh^{-1}\frac{x-\xi}{\beta|y-\eta|}\right. \\ &\quad \left. + \frac{1}{\beta}\frac{x-\xi}{y-\eta}\left[\pi - \cos^{-1}\frac{\beta(y-\eta)}{x-\xi}\right]\right\} \end{aligned}\right\} \tag{A5}$$

where ξ,η represents the origin of the elementary source lines. For the circular-arc airfoil,

$$\left(\frac{dz}{dx}\right)_{\xi,\eta} = 2\frac{t}{c}$$

and

$$\frac{d^2z}{dx^2} \approx -\frac{4}{c}\frac{t}{c}$$

Within the accuracy of the linear theory, the pressure-coefficient ratio is

$$\frac{\Delta P_{\sigma_1}}{\Delta P_{\sigma_2}} = \frac{\frac{-2u_1}{V}}{\frac{-2u_2}{V}} = \frac{u_1}{u_2} \tag{A6}$$

Substituting the appropriate parts of equation (A4) into equation (A6) results in

$$\frac{\Delta P_{\sigma_1}}{\Delta P_{\sigma_2}} = 1 + \frac{-u_{0,0} + \frac{1}{D}u_{0,0}}{u_{0,-h} - \frac{1}{D}u_{0,-h}} \tag{A7}$$

where the denominator of the fraction is restricted to the two-dimensional flow field outside and to the right of the Mach cone from the point $x = 0, y = -h$ and the numerator applies to the region within the Mach cone from the wing tip. Substituting equations (A5) into (A7) and simplifying give

$$\frac{\Delta P_{\sigma_1}}{\Delta P_{\sigma_2}} = 1 + \frac{-\frac{1}{\pi\beta}\left(\pi - \cos^{-1} \beta\frac{y}{x}\right) + \frac{2}{\pi}\frac{y}{c}\left[\cosh^{-1}\frac{x}{\beta|y|} + \frac{1}{\beta}\frac{x}{y}\left(\pi - \cos^{-1}\beta\frac{y}{x}\right)\right]}{\frac{1}{\beta}\left(1 - 2\frac{x}{c}\right)} \tag{A8}$$

where the equation is valid for

$$-1 \leqq \beta\frac{y}{x} \leqq 1$$

REFERENCES

1. Bonney, E. Arthur: Aerodynamic Characteristics of Rectangular Wings at Supersonic Speeds. Jour. Aero. Sci., vol. 14, no. 2, Feb. 1947, pp. 110-116.

2. Busemann, Adolf: Infinitestimal Conical Supersonic Flow. NACA TM 1100, 1947.

3. Jones, Robert T.: Thin Oblique Airfoils at Supersonic Speed. NACA Rep. 851, 1946.

4. Harmon, Sidney M.: Theoretical Supersonic Wave Drag of Untapered Sweptback and Rectangular Wings at Zero Lift. NACA TN 1449, 1947.

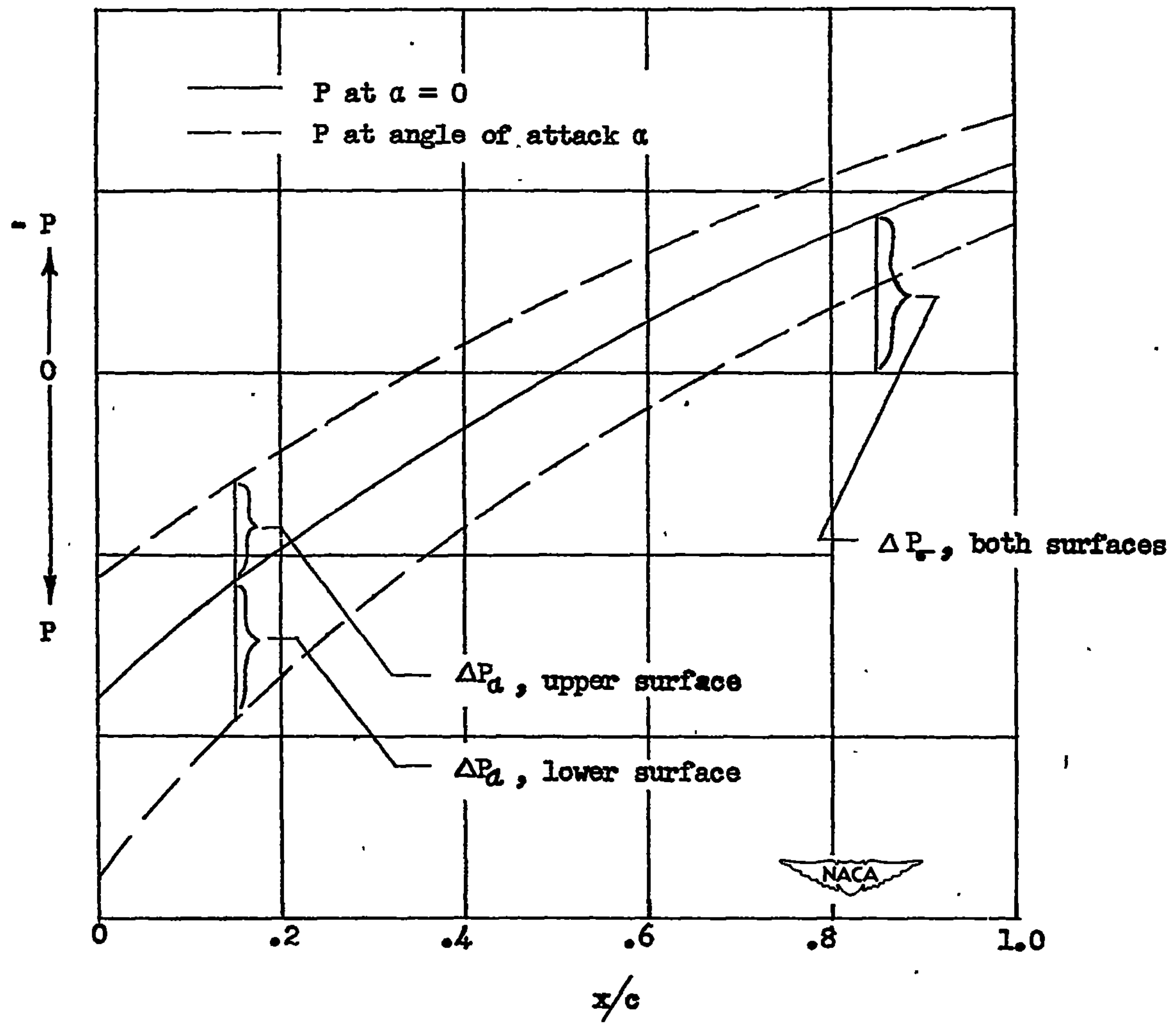


Figure 1.– Definition of pressure–coefficient increments ΔP_{α} and $\Delta P_{\overline{\sigma}}$.

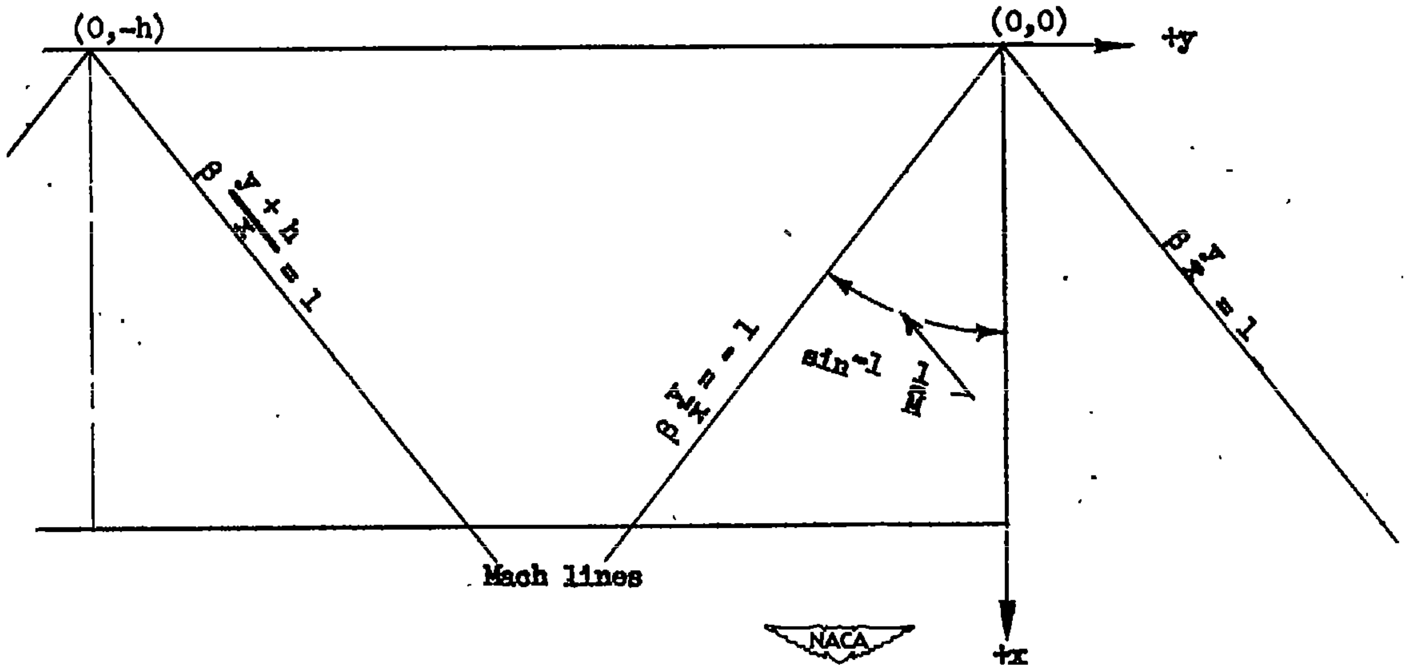


Figure 2.— Location and positive direction of axes used in derivations of pressure ratios.

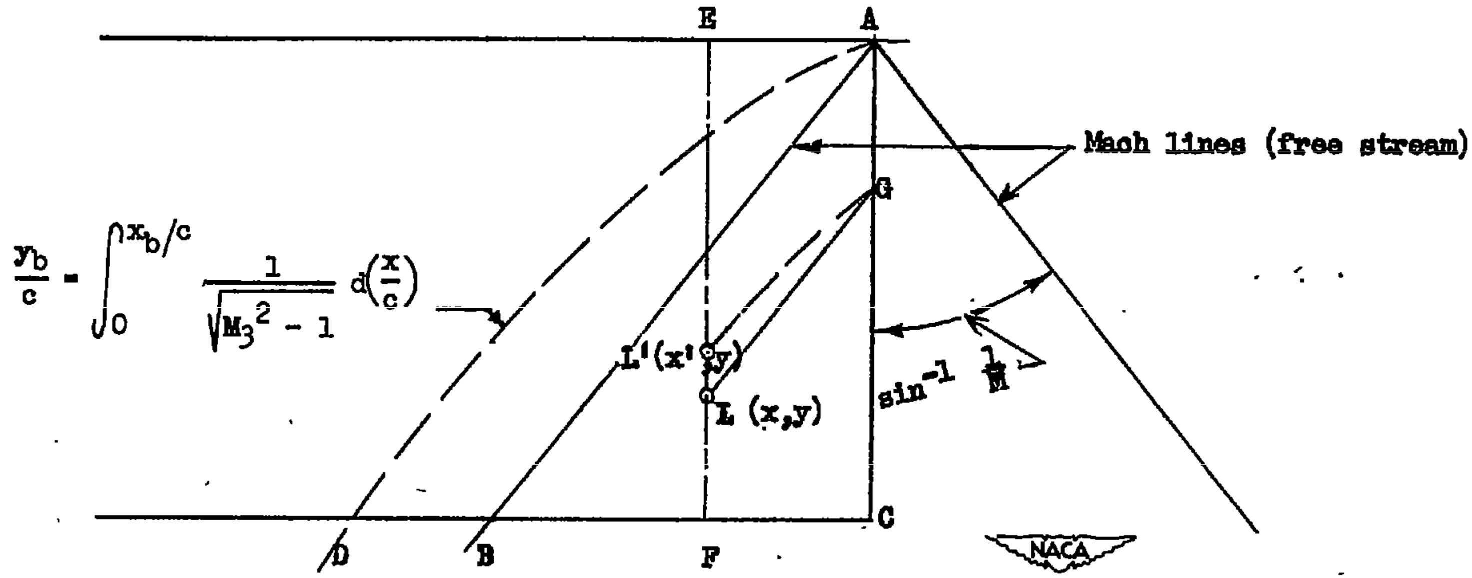


Figure 3.— Sketch showing method of transposing a point in the flow field of linear theory to the distorted flow field which allows for local velocities on the airfoil.

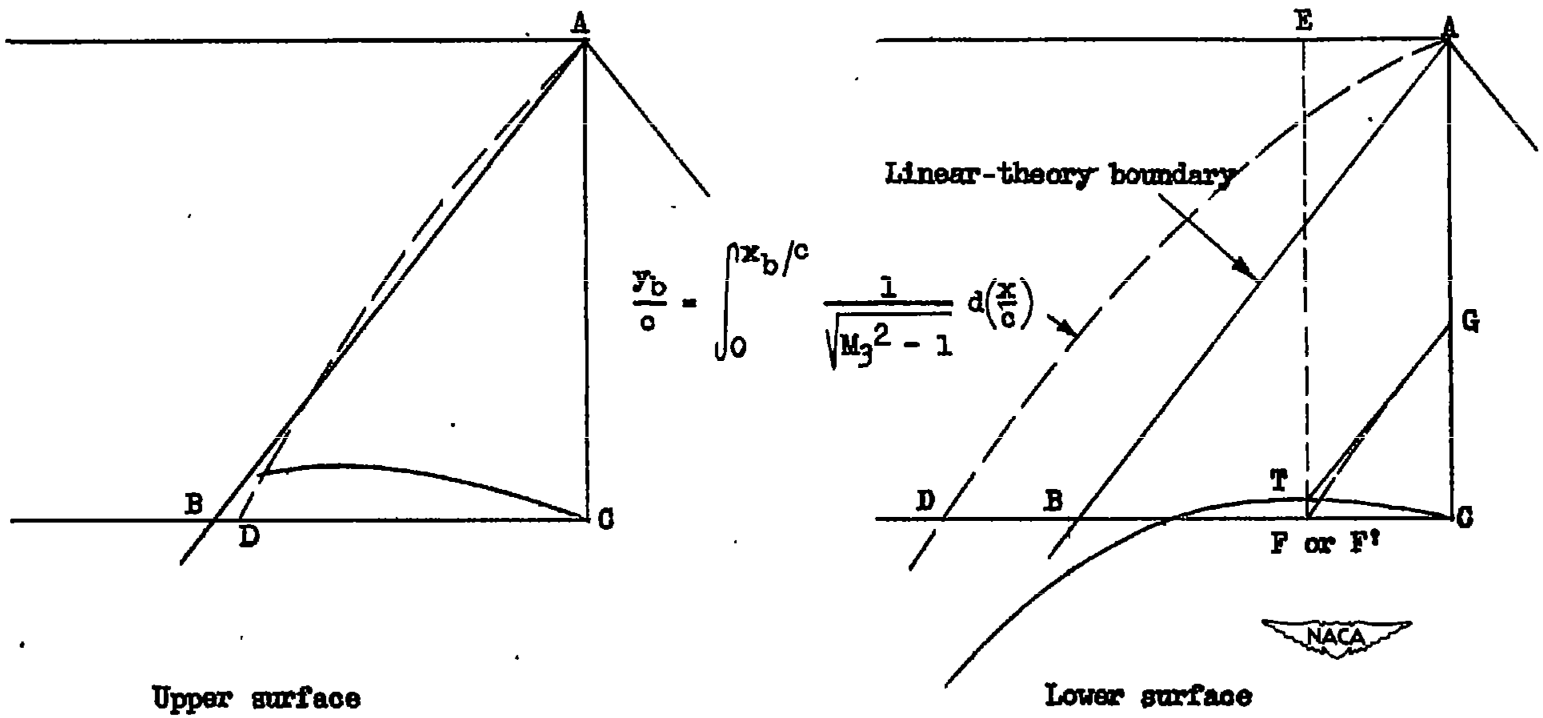
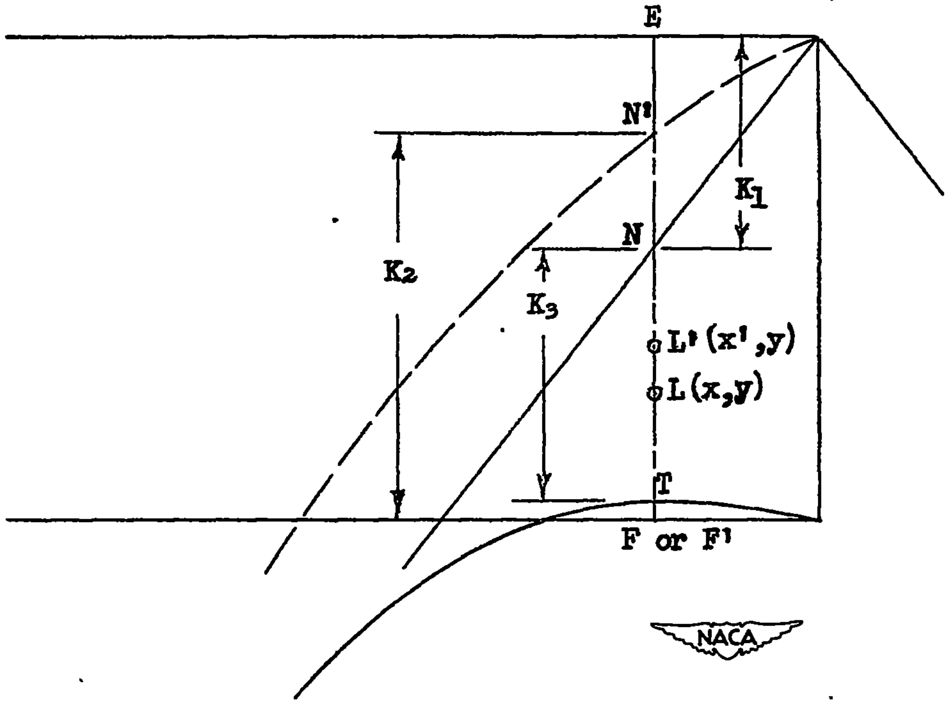
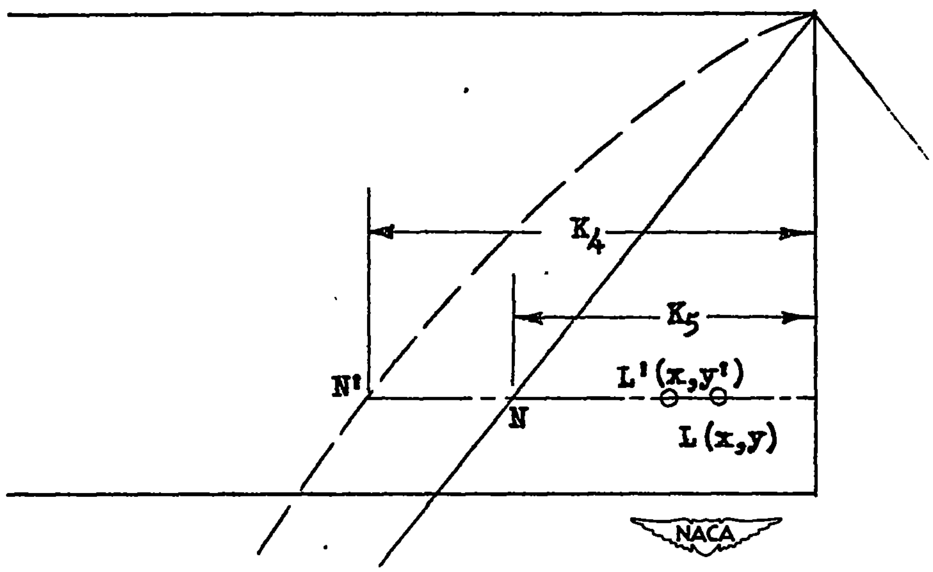


Figure 4.– Typical boundaries of region influenced by wing tip after distortion to allow for local velocities on airfoil. Symmetrical circular–arc airfoil; $\frac{t}{c} = 0.09$; $M = 1.62$; $\alpha = 3.35$.

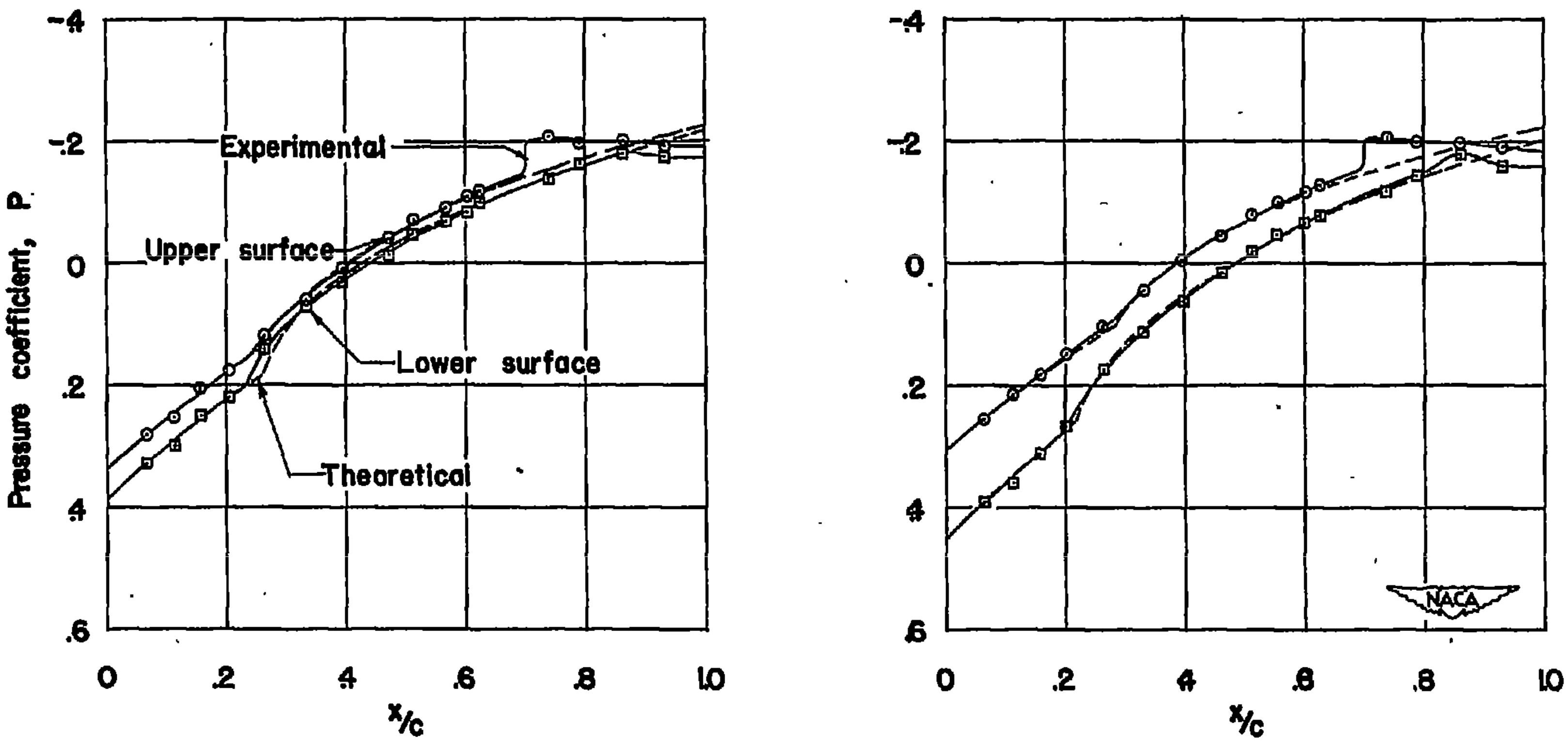


(a) Distortion based on curved Mach lines.



(b) Linear distortion spanwise.

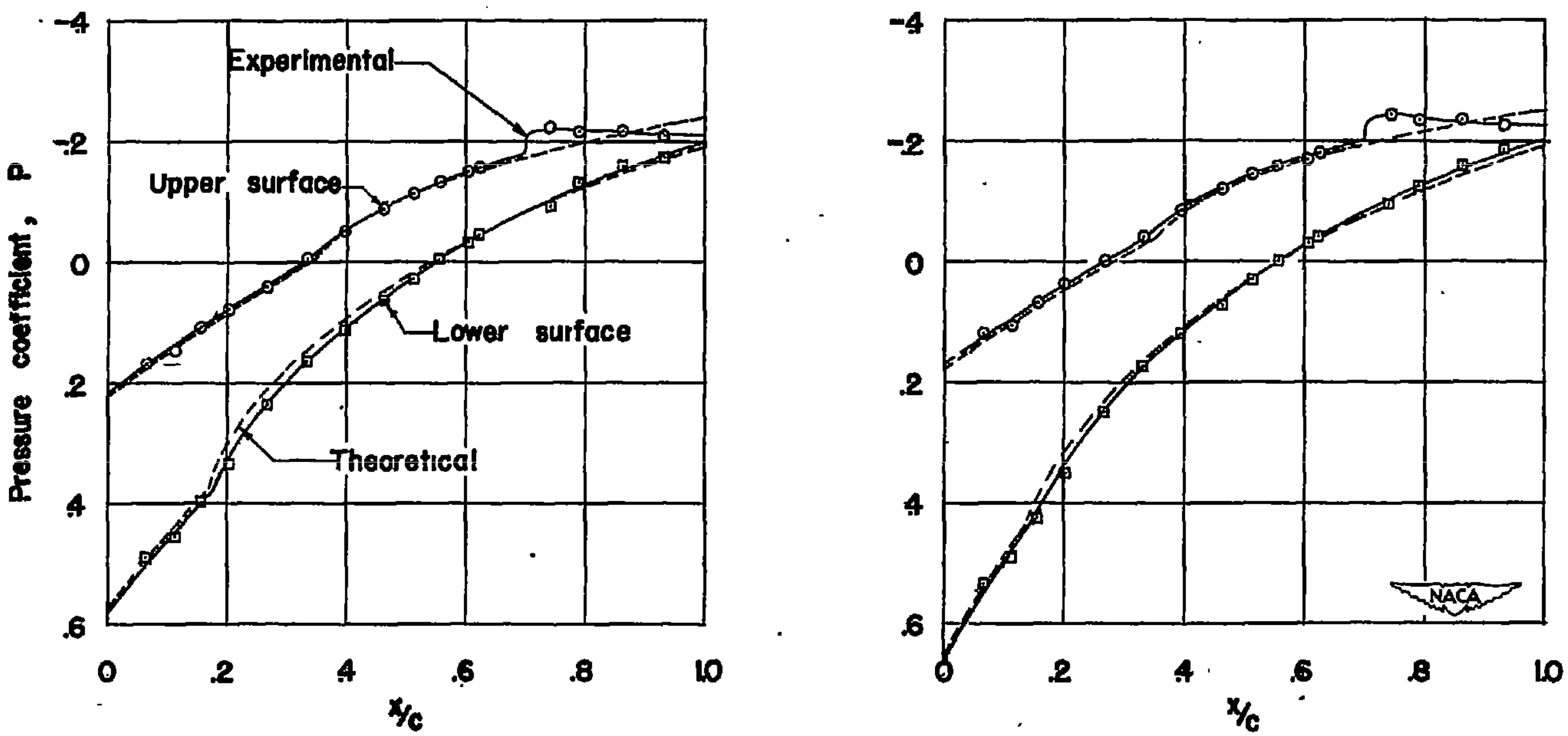
Figure 5.— Definition of constants used in determination of ordinates of point transposed from flow field of linear theory to field distorted to allow for local velocities on airfoil.



(a) $\alpha = 0.55^\circ$.

(b) $\alpha = 1.55^\circ$.

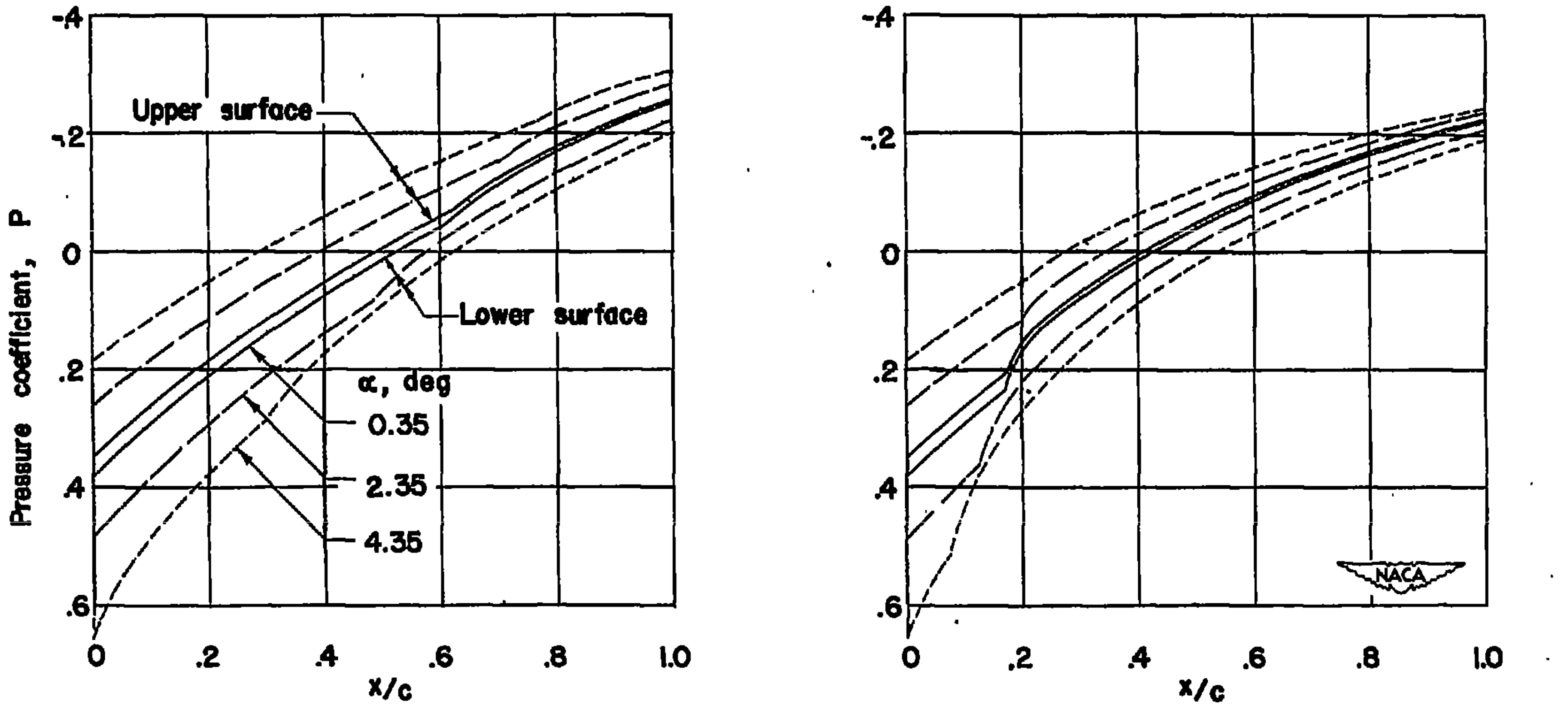
Figure 6.– Comparison of calculated and experimental pressure distributions in the tip region of a symmetrical circular-arc airfoil. $\frac{t}{c} = 0.09$; $\frac{y}{c} = -0.282$; $M = 1.62$.



(c) $\alpha = 3.55^{\circ}$.

(d) $\alpha = 4.55^{\circ}$.

Figure 6.- Concluded.



(a) $\frac{y}{c} = -0.589.$ (b) $\frac{y}{c} = -0.196.$

Figure 7.— Typical variations in calculated pressure distributions with angle of attack at stations influenced by the wing tip. Symmetrical circular-arc airfoil; $\frac{t}{c} = 0.09$; $M = 1.62$.

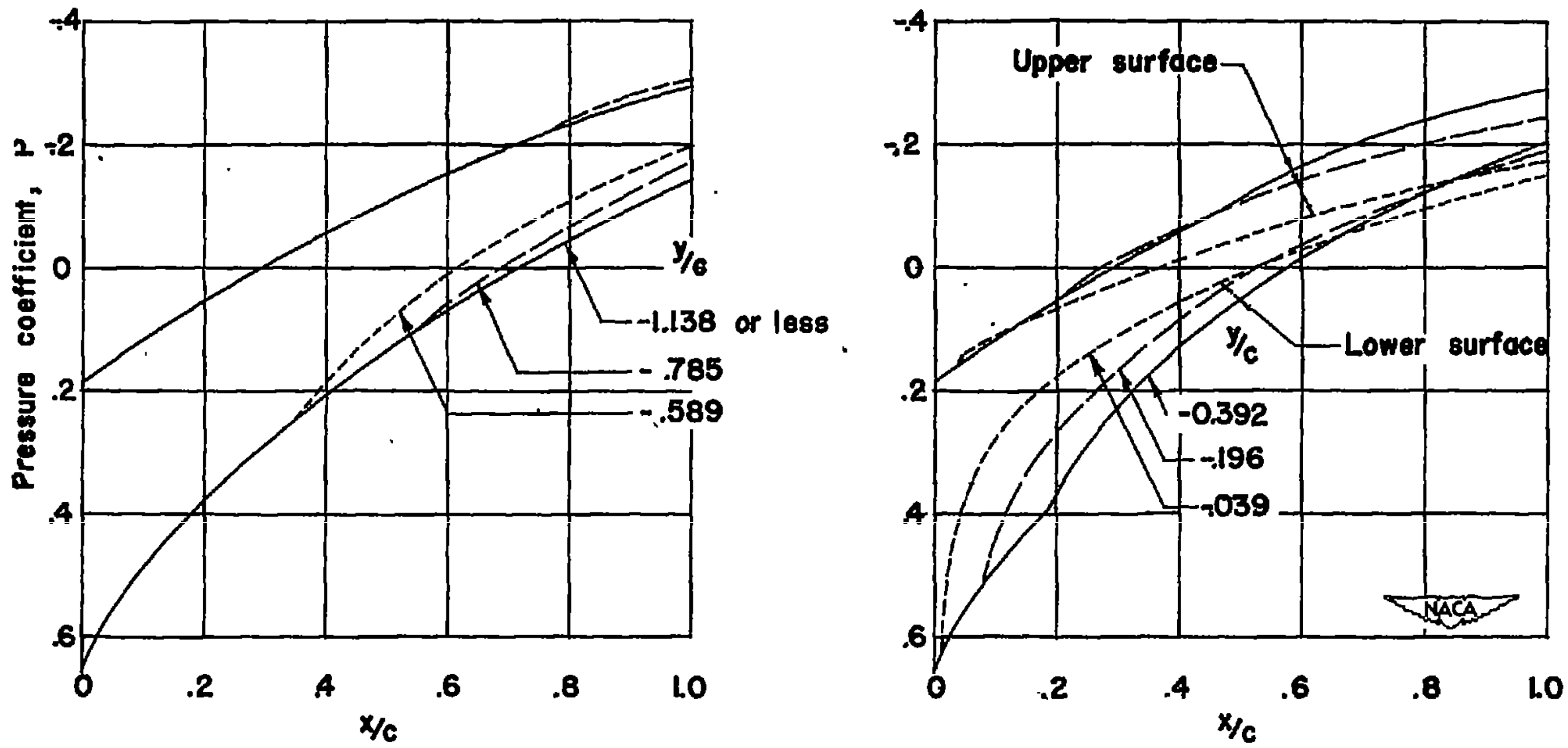


Figure 8.– Typical variations in calculated pressure distributions with spanwise location in region influenced by the wing tip. Symmetrical circular-arc airfoil; $\frac{t}{c} = 0.09$; $M = 1.62$; $\alpha = 4.35^{\circ}$.

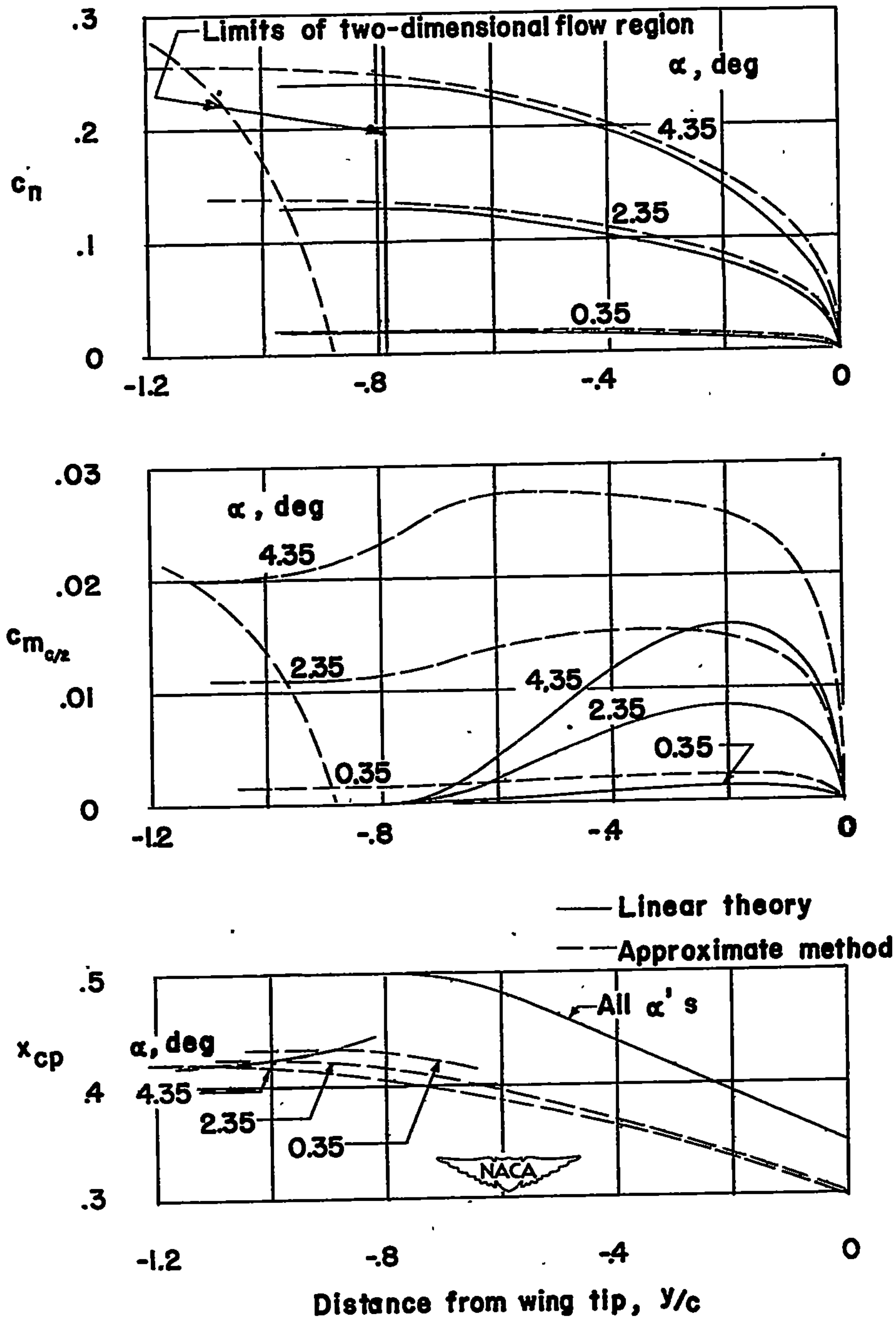


Figure 9.— Comparison of some aerodynamic characteristics of the tip region of a symmetrical circular-arc airfoil as determined from linear theory and the approximate method. $\frac{t}{c} = 0.09$.