Subsonic Drag Estimation Methods
from: NADC AW-610A (Unclassified)
(Note: rev. on old brown paper & written quickly)

SW = reference wings area
FR = fineness ratio
C\textsubscript{LP} = zero lift Cp
\Delta C\textsubscript{D}\textsubscript{0} = \Delta C\textsubscript{D} due to components\n\gamma = skin friction on flat plate
\gamma = (\gamma\textsubscript{avg} + \gamma\textsubscript{tip})/2

\begin{itemize}
  \item \( C_f \text{ wings} = C_{fp} \left[ 1 + 2 \left( \frac{\gamma}{\gamma\text{avg}} \right) + 100 \left( \frac{\gamma}{\gamma\text{avg}} \right)^4 \right] \)
  \item \( C_f \text{ fuselage} = C_{fp} \left[ 1 + \frac{60}{(FR)^2} + 0.0025(FR) \right] \)
  \item \( C_f \text{ tail} = 1.1 C_{fp} \left[ 1 + 2 \left( \frac{\gamma}{\gamma\text{avg}} \right) + 100 \left( \frac{\gamma}{\gamma\text{avg}} \right)^4 \right] \)
  \item \( C_f \text{ Nac} = Q \frac{C_{fp}}{FR} \left[ 1 + \frac{0.35}{FR} \right] \quad (Q=1.5 \text{ for pylon mount}) \)
  \item \( C_{fp} = K_{fp} \times (1 + 0.2 M_a) \quad (0.467 \text{ Mach correction factor with } M \text{ up to } M=1.6) \)
  \item \( E(\text{straight wings}) = 1.76 \left[ 1 - 0.045 \left( \frac{AR}{c} \right)^{0.68} \right] - 0.64 \)
  \item \( E(\text{swept wings}) = 4.61 \left[ 1 - 0.045 \left( \frac{AR}{c} \right)^{0.68} \right] \left( \cos \left( \frac{\alpha}{2} \right) \right)^{0.15} - 3.1 \)
  \item Swet/S\text{ref} = Drag/wing x 2.7; Conventional 4.3, Transport 5.0
\end{itemize}
AERODYNAMIC-CENTER CONSIDERATIONS
OF WINGS AND WING-BODY COMBINATIONS

by John E. Lamar and William J. Alford, Jr.
Langley Research Center
Langley Station, Hampton, Va.
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SUMMARY

Aerodynamic-center variations with Mach number are considered for wings of different planform. The normalizing parameter used is the square root of the wing area, which provides a more meaningful basis for comparing the aerodynamic-center shifts than does the mean geometric chord. The theoretical methods used are shown to be adequate for predicting typical aerodynamic-center shifts, and ways of minimizing the shifts for both fixed and variable-sweep wings are presented.

INTRODUCTION

In the design of supersonic aircraft, a detailed knowledge of the aerodynamic-center movement is important in order to minimize trim drag, maximize load-factor capability, and provide acceptable handling qualities. One of the principal contributions to the movement of the aerodynamic-center position is the well-known change in load distribution with Mach number in going from subsonic to supersonic speeds. In addition, large aerodynamic-center variations are quite often associated with variable-geometry features such as variable wing sweep.

The purpose of this paper is to review the choice of normalizing parameters and the effects of Mach number on the aerodynamic-center movement of rigid wing-body combinations at low lift. For fixed wings the effects of both conventional and composite planforms on the aerodynamic-center shift are presented, and for variable-sweep wings the characteristic movements of aerodynamic-center position with pivot location and with a variable-geometry apex are discussed.

Since systematic experimental investigations of the effects of planform on the aerodynamic-center movement with Mach number are still limited, the approach followed herein is to establish the validity of the computational processes by illustrative comparison with experiment and then to rely on theory to show the systematic variations. The two theories used in this paper are for the wing alone in unseparated flow. One is a modified Multhopp subsonic lifting-surface theory developed by the senior author (unpublished), and the other is a supersonic lifting-surface theory (ref. 1). For wings experiencing separated flow these theories are not adequate per se for predicting the aerodynamic-center movement.

SYMBOLS

A aspect ratio
a distance from apex of high-sweep wing to apex of low-sweep wing (see fig. 10)
b span
C_L lift coefficient
C_p pressure coefficient, \( \frac{p_{\text{local}} - p_{\text{free stream}}}{q} \)
\( \Delta C_p \) incremental pressure coefficient, \( C_{p,\text{upper}} - C_{p,\text{lower}} \)
c local chord
c̄ mean geometric chord
c_r root chord of basic planform
c_t tip chord of basic planform
d longitudinal distance from root trailing edge to tip trailing edge
K constant
l longitudinal distance from apex to tip trailing edge
M Mach number
p static pressure
q free-stream dynamic pressure
S wing area
x chordwise distance from apex of high-sweep wing to plane-of-symmetry intercept with trailing edge of free-floating apex
\( \bar{x}_M \) chordwise distance from a reference point to aerodynamic center at any Mach number
\( \bar{x}_M=0, \bar{x}_M=0.2, \bar{x}_M=0.25 \) chordwise distance from a reference point to aerodynamic center at specific Mach number indicated by subscript
\( \Delta \bar{x} \) incremental change in aerodynamic-center location

2
A knowledge of the actual dimensional movement of the aerodynamic center is required in order to determine the out-of-trim moments which must be balanced by the control surface. Thus, in the selection of a normalizing parameter the need for a reference length which, for a given wing area, is independent of planform is considered to be of primary importance. The reference length selected herein is the square root of the wing area $\sqrt{S}$, which, of course, is independent of planform and therefore provides fractional aerodynamic-center movements that are proportional to the actual dimensional shifts.

The customary use of the mean geometric chord $\bar{c}$, although adequate for normalizing the aerodynamic-center shift for a given planform, is not convenient when comparing planforms, since the magnitude of $\bar{c}$ is dependent upon planform. As an aid in transferring aerodynamic-center shifts from one normalizing parameter to another, the relationship between $\bar{c}$ and $\sqrt{S}$ is given both algebraically and graphically in figure 1 for wings which fit within the geometry limitations shown. For composite planforms $\bar{c}/\sqrt{S}$ may be determined from

$$\frac{\bar{c}}{\sqrt{S}} = \frac{\int_0^{b/2} c^2 dy}{\sqrt{2 \left( \int_0^{b/2} c \ dy \right)^{3/2}}}$$

DISCUSSION

Comparison of Theory and Experiment

Some typical experimentally determined aerodynamic-center shifts with Mach number (ref. 2), which are useful in evaluating the theories and the previously mentioned normalizing parameters, are presented in figures 2 and 3.
The experimental shifts, together with theoretical predictions, are shown in figure 2 for a series of delta wings with aspect ratios ranging from 2 to 4. In this figure \( \Delta \bar{x} \) is the distance between the aerodynamic-center location at a Mach number of 0.25 and the aerodynamic-center location at any Mach number. The mean geometric chord \( \bar{c} \) and the square root of the wing area \( \sqrt{S} \) are used as normalizing parameters, and both \( \Delta \bar{x}/\bar{c} \) and \( \Delta \bar{x}/\sqrt{S} \) are plotted as functions of Mach number. When the aerodynamic-center shift is based on the respective \( \bar{c} \), the delta wing with the lowest aspect ratio has the smallest incremental change in aerodynamic-center location at the supersonic Mach numbers. However, when the aerodynamic-center shift is based on the respective \( \sqrt{S} \), all three wings exhibit essentially the same fractional change in aerodynamic-center location throughout the Mach number range. The theories predict reasonably well the aerodynamic-center shifts for these delta-wing—bodies.

Figure 3 presents three wing-body combinations and illustrates the effect of wing sweep and taper ratio on the aerodynamic-center shift with Mach number. The wings are of aspect ratio 3 and have planforms ranging from a trapezoidal to a delta shape. Of the three wing-body combinations shown, the delta-wing—body configuration is seen to exhibit the smallest change in aerodynamic-center location for Mach numbers greater than 1 when \( \bar{c} \) is used as the normalizing parameter. However, when \( \sqrt{S} \) is used as the normalizing parameter, the aerodynamic-center shift for the sweptback-wing—body configuration is almost as small. Again the agreement between theory and experiment is reasonable.

An indication of the aerodynamic-center movement, when the wings of figures 2 and 3 are sized to provide comparable take-off and landing performance, can be obtained by normalizing the incremental aerodynamic-center shifts previously obtained with respect to the low-speed lift coefficient of each wing at an assumed angle of attack. Such an indication is presented in figure 4 for \( \alpha = 120^\circ \). At the supersonic Mach numbers for the delta planforms, the wing with the highest aspect ratio has the lowest aerodynamic-center shift, and for the planforms of constant aspect ratio, the sweptback wing exhibits the lowest aerodynamic-center shift.

Fixed-Wing Studies

In figures discussed subsequently, the aerodynamic-center shifts have been computed by the theoretical methods. For wings which have fixed planforms, the reference length is the \( \sqrt{S} \) of each planform.

The results of one such aerodynamic-center study for a series of conventional fixed wings with planform variation in sweep and in taper and notch ratios are presented in figure 5. For a delta wing, \( d/l = 0 \) and for arrow wings, \( d/l > 0 \). For illustrative purposes both the effect of changing the leading-edge sweep and the notch ratio when the taper ratio is zero and the effect of changing the taper and notch ratios when the leading-edge sweep angle is 60\(^\circ\) are presented.

When the taper ratio is zero, a decrease in \( \Delta \bar{x}/\sqrt{S} \) of about 0.05 occurs as the notch ratio is increased from 0 to 0.5 for leading-edge sweep angles of
For a sweep angle of 70°, \( \Delta x/\sqrt{S} \) at first decreases approximately 0.01 and then increases about 0.01 above its value at \( d/l = 0 \). At any particular notch ratio, the wing with the lowest sweep shows the smallest aerodynamic-center shift.

When the wing leading-edge sweep angle is 60°, decreases in \( \Delta x/\sqrt{S} \) of 0.05, 0.09, and 0.12 occur over the range of notch ratios considered for taper ratios of 0, 0.25, and 0.50, respectively. At any particular notch ratio, the wing with the lowest taper ratio exhibits the smallest aerodynamic-center shift. When the supersonic Mach number is other than 3, different trends in the aerodynamic-center movement may occur with increasing notch ratio.

One method of minimizing the aerodynamic-center shift of an arrow wing is to reduce the sweep of the wing tip by shearing it forward. Some calculated results illustrating this technique are presented in figure 6. The basic arrow wing has a sweep of 74°, and \( \Delta x/\sqrt{S} \) is reduced to about half its original value by shearing the tip forward from 74° to 55°. The reason for this reduction is that wings with cranked tips lose some of the loading at the tip because the value of lift-curve slope on the lower swept outer portion of the wing decreases with increasing Mach number more markedly than does the value on the inner portion of the wing. This decrease in lift-curve slope on the outer portion begins at a lower Mach number and results in a change in loading which causes the aerodynamic center to move forward with increasing supersonic Mach number.

One method of reducing the aerodynamic-center shift of a delta wing is the addition of a forewing inboard. In figure 7 the effect of such an addition is presented as a function of the leading-edge-break location and apex extension. A reduction in the aerodynamic-center shift is obtained for each apex location as \( \frac{Y_b}{b/2} \) is increased from 0 to 0.5. At any particular value of leading-edge-break ratio within the range examined, the wing with the most forward apex or the longest root chord has the smallest aerodynamic-center shift, because the inboard sweeps are higher and therefore the inner panel has a lower aspect ratio which gives it an essentially invariant value of lift-curve slope with Mach number. However, the outer panel has a higher aspect ratio and lower sweep, and the value of lift-curve slope decreases with increasing supersonic Mach number. Thus, the inner panel carries proportionally more of the loading. The aerodynamic center is forced forward with increasing values of leading-edge-break ratio because of the area added inboard. Experimental substantiation of this low level of aerodynamic-center shift, with a model that had a wing which covered most of the body, is provided in reference 3. (See also ref. 4.)

In addition, wing-body combinations exhibit smaller aerodynamic-center shifts than does the wing alone because the body acts as a forewing with a very low value of leading-edge-break ratio.

Variable-Sweep-Wing Studies

For wings with variable sweep, a problem in aerodynamic-center variation, in addition to that caused by the Mach number effect, results from changes in the wing sweep. The shift resulting from wing-sweep changes must be minimized.
in order to make variable-sweep wings competitive, from aerodynamic-center considerations, with fixed wings. To illustrate this problem, the theoretical loading distributions of a variable-sweep wing with an outboard pivot (ref. 5) at a Mach number of 0.23 and at low lift is presented in figure 8. At the top of this figure the variable-sweep wing is shown in its low-sweep and high-sweep positions, and superimposed on the low-sweep planform are its theoretical and experimental chordwise pressure loadings which are seen to be in good agreement. At the bottom of the figure the theoretical longitudinal loading distributions for both sweeps have been computed at \( C_L = 0.12 \) and projected onto the plane of symmetry. As the outer panel is swept back, the inner panel carries more of the loading and thus tends to balance out the additional moments created by the reduced outer-panel loadings acting through longer moment arms. In this example, because of the outboard location of the pivot, the aerodynamic center, as given by the chordwise location of the lift vector, actually shifts slightly forward.

A study was undertaken to determine the effect that the pivot location has on the aerodynamic-center shift, and the results are presented in figure 9. In this figure and in figure 10, the reference planform area is taken for the wing in its high-sweep position.

Each pivot lies on the loci of points from which the outer panel can be swept from its high-sweep position to a low-sweep position. The relative chordwise location of the pivot determines the chordwise position of the outer panel at low sweep without changing the sweep angle or the semispan.

The results of the theoretical study show that the total aerodynamic-center shift \( \Delta \vec{x}/\sqrt{S} \) (see fig. 9) can be reduced from 0.2 to 0.1 by moving the pivot outboard. The dashed line is used as a reference to indicate that portion of the total shift caused by the change in Mach number from 0.2 to 2 at \( \alpha_0 = 70^\circ \). The remaining shift is attributed to the change in sweep from 15\(^\circ\) to 70\(^\circ\) at \( M = 0.2 \). The movement of the pivot outboard changes only the part of the shift dependent on sweep. By proper positioning of the pivot, this part of the shift can be eliminated. When the sweep effect causes the aerodynamic center to move ahead of its low-speed high-sweep position, the Mach number effect is reduced.

These results are supported by experimental data for a similar wing-body combination. Figure 9 shows that a reduction in the total aerodynamic-center shift of 0.07 occurs as the spanwise location of the pivot is moved from one extreme to the other. The characteristics of this combination and how the pivot location affects maneuverability considerations are discussed in reference 6.

As noted in reference 7, if a high inboard sweep is required for supersonic flight, then at subsonic speeds and low outer-panel sweep, devices such as the double inboard pivot (ref. 8) and the free-floating apex (ref. 9) can be used to eliminate the resulting pitch-up. These devices also provide a means of controlling the aerodynamic-center movement, as illustrated in figure 10, where they are shown to have the following two features in common: (1) When the outer panel is in its low-sweep position, the forewing or apex is either pivoted inside the fuselage or allowed to free-float carrying no load; and (2) when the
outer panel is swept back, the apex is affixed to the front of the outer panel and forms a continuous leading edge.

Lifting-surface calculations have been made to illustrate the effect of the amount of the apex which is folded or free-floated. Varying amounts of the apex have been removed to represent the aerodynamic effect of both concepts. With the removal, subsonically, of an increasingly large amount of the apex (correlated with the chordwise distance \( x \)), the total aerodynamic-center shift decreases from about 0.18 to 0. Again the dashed line represents that portion of the shift due to changing the Mach number from 0 to 3 when \( \Lambda_0 = 71.5^0 \). The effect of changing the sweep \( \Lambda_0 \) from 25\(^0\) to 71.5\(^0\) at \( M = 0 \) makes up the remainder of the shift.

When \( x/a = 0 \), the change in wing sweep has essentially no effect; consequently, almost all the aerodynamic-center shift is due to the change in Mach number. However, when \( x/a = 1.0 \), the sweep effect is large enough to cancel all the Mach number effect.

It should be noted that the aerodynamic-center shift may also be minimized by changing the supersonic Mach number or by changing the center-of-gravity location at the different sweeps and Mach numbers.

CONCLUSIONS

A general conclusion of this study is that, when comparing aerodynamic-center movements of wings of different planform, a normalizing parameter independent of planform, such as the square root of the wing area, is more appropriate than the customarily used mean geometric chord, which is dependent on planform. The following specific conclusions were reached:

1. The theoretical methods have been demonstrated to be adequate for predicting the aerodynamic-center shift with Mach number for a variety of wing planforms, but are not suitable for determining the absolute aerodynamic-center location at any Mach number since body and interference effects are not included.

2. For fixed wings, the aerodynamic-center shift can be controlled by proper selection of sweep and of taper and notch ratios and by inboard and outboard area proportioning with different degrees of sweep.

3. For variable-sweep wings the aerodynamic-center shift can be controlled by pivot location and by apex devices, such as the double inboard pivot and the free-floating apex.

Langley Research Center,  
National Aeronautics and Space Administration,  
Langley Station, Hampton, Va., May 23, 1966,  
126-13-03-22-23.
REFERENCES


EFFECT OF PLANFORM PARAMETERS ON $\frac{c}{\sqrt{S}}$

$$\lambda = \frac{c_1}{c_r}$$

$$\frac{c}{\sqrt{S}} = \frac{4}{3}\frac{1+\lambda+\lambda^2}{1+2\lambda+\lambda^2}$$

Figure 1

EFFECT OF ASPECT RATIO AND SWEEP

$\Delta x = x_{M} - x_{M} = 0.25; \lambda = 0$

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<th>$\lambda$, DEG</th>
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<td></td>
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Figure 2
EFFECT OF SWEEP AND TAPER
\[ \Delta \bar{x} = \bar{x}_M - \bar{x}_M = 0.25; \ A = 3 \]

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<th>( \Delta ), DEG</th>
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Figure 3

EFFECT OF WING SIZING FOR LOW-SPEED CONDITIONS
\[ \Delta \bar{x} = \bar{x}_M - \bar{x}_M = 0.25 \]

Figure 4
CONVENTIONAL-PLANFORM VARIATION

$\Delta x = x_M - 3 - x_M = 0$

**Figure 5**

**COMPOSITE PLANFORMS**

EFFECT OF CRANKED-TIP SWEEP; $\Delta x = x_M - 3 - x_M = 0$; $y/b/2 = 0.73$

**Figure 6**
COMPOSITE PLANFORMS
EFFECT OF LEADING-EDGE BREAK LOCATION; $\Delta \bar{x} = \bar{x}_M - \bar{x}_M = 0$

$M = 0.23; \ C_L = 0.12$

Figure 7

EFFECT OF SWEEP ON LOAD DISTRIBUTION

$K = 1.41; 1.63; 2.19$

Figure 8

THEORETICAL LOADING

CONSTANT $C_L$
EFFECT OF SPANWISE LOCATION OF PIVOT

\[ \Delta \vec{x} = (\vec{x}_M + 2) \Delta \alpha_0 = 70^\circ - (\vec{x}_M + 0.2) \Delta \alpha_0 = 15^\circ \]

THEORY EXPERIMENT

\[ \Delta \vec{x} = (\vec{x}_M + 3) \Delta \alpha_0 = 71.5^\circ - (\vec{x}_M + 0) \Delta \alpha_0 = 25^\circ \]

Figure 9

EFFECT OF VARIABLE-GEOMETRY APEX

\[ \Delta \vec{x} = (\vec{x}_M + 3) \Delta \alpha_0 = 71.5^\circ - (\vec{x}_M + 0) \Delta \alpha_0 = 25^\circ \]

Figure 10
LOW-SPEED STATIC STABILITY AND DAMPING-IN-ROLL CHARACTERISTICS OF SOME SWEPT AND UNSWEPT LOW-ASPECT-RATIO WINGS

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October 1947
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE

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SUMMARY

An investigation at low speed to determine the static stability and damping-in-roll characteristics of a number of low-aspect-ratio wings including swept wings of approximately triangular plan form has been made in the Langley free-flight tunnel and the 15-foot free-spinning tunnel. The static longitudinal stability, directional stability, effective dihedral, and damping in roll were investigated for a range of lift coefficient through maximum lift.

It was found that the unswept tapered wings showed a tendency toward decreased longitudinal stability at low angles of attack as the aspect ratio was reduced. For the swept wings the neutral point moved rearward with respect to the quarter chord of the mean aerodynamic chord as the sweepback increased. In general, the effective dihedral and directional stability increased with an increase in lift coefficient and with a reduction of aspect ratio.

The unswept wings showed no consistent variation in damping in roll with lift coefficient for lift coefficients below maximum lift; whereas the triangular and swept tapered wings in general showed a reduction of damping in roll with increasing lift coefficient and in some cases became unstable before maximum lift was reached. The damping in roll decreased as expected with aspect ratio. Experimental values of the damping in roll were generally smaller than the theoretical values.

INTRODUCTION

The recent trend toward the use of low-aspect-ratio wings for high-speed flight requires that the low-speed stability and control characteristics of such configurations be determined. Some work has
been done to determine the static stability characteristics of
unswept low-aspect-ratio wings (for example, reference 1). The
present investigation was undertaken to extend this work to include
the damping-in-roll and static stability characteristics of both
swept and unswept low-aspect-ratio wings. The swept wings were of
triangular or approximately triangular plan form.

This investigation consisted of force and damping-in-roll
tests of 16 wings having different aspect ratios, taper ratios,
and sweepback angles. Most of the wings were of low aspect ratio
(aspect ratio ≤ 3) although four wings of higher aspect ratio
were included for comparison.

SYMBOLS

All forces and moments were referred to the stability axes
which are defined in figure 1. The rolling, yawing, and pitching
moments were all referred to the quarter-chord point of the mean
aerodynamic chord. No corrections for the effects of the jet
boundaries or the support strut interference were applied to the
data. The symbols and coefficients used in the present paper are:

- \( S \) wing area, square feet
- \( V \) airspeed, feet per second
- \( b \) wing span, feet
- \( c \) chord, feet
- \( c_m \) mean aerodynamic chord, feet, \( \left( \frac{2}{S} \int_{0}^{b/2} b^2 \, db \right) \)
- \( c_r \) root chord, feet
- \( c_t \) tip chord, feet
- \( \Lambda_{c/4} \) angle of sweepback of quarter-chord line of wing, degrees
- \( \lambda \) taper ratio \( (c_t/c_r) \)
- \( \alpha \) angle of attack, degrees
- \( \psi \) angle of yaw, degrees
\[ \beta \quad \text{angle of sideslip, degrees (} \beta = \psi \text{)} \]
\[ \rho \quad \text{mass density of air, slugs per cubic foot} \]
\[ q \quad \text{dynamic pressure, pounds per square foot } \left( \frac{1}{2} \rho v^2 \right) \]
\[ A \quad \text{aspect ratio } \left( \frac{b^2}{s} \right) \]
\[ C_L \quad \text{lift coefficient } \left( \frac{\text{Lift}}{qS} \right) \]
\[ C_D \quad \text{drag coefficient } \left( \frac{\text{Drag}}{qS} \right) \]
\[ C_m \quad \text{pitching-moment coefficient } \left( \frac{\text{Pitching moment}}{qSb} \right) \]
\[ C_t \quad \text{rolling-moment coefficient } \left( \frac{\text{Rolling moment}}{qSb} \right) \]
\[ C_n \quad \text{yawing-moment coefficient } \left( \frac{\text{Yawning moment}}{qSb} \right) \]
\[ p \quad \text{rolling angular velocity, radians per second} \]
\[ \frac{p}{2V} \quad \text{rolling-angular-velocity factor of helix angle generated by wing tip in roll, radians} \]
\[ C_{tp} \quad \text{rate of change of rolling-moment coefficient with rolling-angular-velocity factor} \left( \frac{\partial C_t}{\partial \left( \frac{p}{2V} \right)} \right) \]
\[ C_{\beta t} \quad \text{rate of change of rolling-moment coefficient with angle of sideslip in degrees} \left( \frac{\partial C_t}{\partial \beta} \right) \]
\[ C_{n\beta} \quad \text{rate of change of yawing-moment coefficient with angle of sideslip in degrees} \left( \frac{\partial C_n}{\partial \beta} \right) \]
\[ C_{L\alpha} \quad \text{lift-curve slope} \left( \frac{\partial C_L}{\partial \alpha} \right) \]
APPARATUS, MODELS, AND TESTS

Force and damping-in-roll tests were made on each of the 18 wings described in table 1. In order to facilitate the wing construction, most of the low-aspect-ratio wings were of flat-plate airfoil section, for past experience has shown that at low aspect ratios (approximately 2 or less) the choice of airfoils has little effect on the aerodynamic characteristics of a wing. The geometric dihedral of the mean thickness line was zero for all of the wings except wings 4 and 6, which had -0.6° and -1.9° dihedral, respectively.

The force tests were made on the six-component balance of the Langley free-flight tunnel. (For a complete description of the balance and tunnel see references 2 and 3, respectively.) The tests consisted of measurements through a range of angle of attack from small negative angles through the angle of maximum lift with angles of yaw of 0°, 5°, and -5°. The values of the lateral stability derivatives $C_{L\beta}$ and $C_{N\beta}$ were determined from the rolling-moment and yawing-moment data at 5° and -5° yaw.

The damping-in-roll tests were made on a rolling rig in the Langley 15-foot free-spinning tunnel (reference 4) by the method described in reference 5. The values of the damping-in-roll derivative $C_{p\beta}$ were determined from the slope of curves of $C_{L}$ against $\frac{\beta}{2V}$ for several rotational speeds between $\frac{\beta}{2V} = 0.1$ and -0.1 at angles of attack ranging from small negative angles through maximum lift.

All the tests were made at a dynamic pressure of 3.0 pounds per square foot which corresponds to Reynolds numbers from 165,000 to 1,150,000 based on the mean aerodynamic chords of the wings tested. The rolling, yawing, and pitching moments were all referred to the quarter-chord point of the mean aerodynamic chord.

RESULTS AND DISCUSSION

The basic data from the low-aspect-ratio investigation are presented in figures 2 to 6 and a summary of the results prepared from the basic data is presented in figures 7 to 12.
The wings have been divided into five groups for convenience in presentation and discussion, namely:

1. Rectangular with conventional airfoil (wings 1, 2, and 3)
2. Unswept, tapered with conventional airfoil (wings 4, 5, 6, and 7)
3. Unswept, tapered with flat-plate airfoil (wings 8, 9, and 10)
4. Triangular with flat-plate airfoil (wings 11, 12, 13, and 14)
5. Swept, tapered with flat-plate airfoil (wings 15, 16, 17, and 18)

Care should be taken in interpreting the results of the present low-scale tests in terms of full-scale airplanes, although some correlation of the data for the triangular wings with full-scale tests has been obtained from unpublished force tests of a full-scale airplane of approximately triangular plan form conducted in the Langley full-scale tunnel. The static stability characteristics of the small-scale models were in good agreement with those of this full-scale airplane.

Lift Characteristics

For each group of wings the angle of attack for maximum lift increased as the aspect ratio decreased (figs. 2 to 6).

The variation of maximum lift coefficient with aspect ratio is presented in figure 7. Wing 8 with the flat-plate airfoil had a much lower maximum lift coefficient than did wing 6 which had the same plan form but a conventional airfoil section. The low maximum lift on wing 8 is attributed to a leading-edge separation at small angles of attack which is common to flat plates of moderate and high aspect ratios.

For the swept tapered wings the maximum-lift-coefficient curves were faired with the aid of additional points taken from unpublished free-flight-tunnel data on similar wings. The results of figure 7 show that the maximum lift coefficient for these wing groups reached peak values at fairly low aspect ratios (aspect ratios between 0.6 and 2.0). This result is in agreement with the data of reference 1 for straight wings and with the data of reference 6 for triangular wings with conventional airfoil sections.

The variation of the lift-curve slope \( C_l \alpha \) with aspect ratio is presented in figure 8. The theoretical variation of the values of lift-curve slope with aspect ratio for aspect ratios above 3.0 was
obtained by assuming a section lift-curve slope of 0.10 per degree and applying the calculation methods of reference 7. For aspect ratios below 1.0 the following equation

\[ C_{L_0} = \frac{1}{57.2 \pi} \frac{\pi}{A} \]

obtained from reference 8 was used. The theoretical curves were faired in for the aspect ratios between 1.0 and 3.0. Figure 8 shows good agreement between theoretical and experimental results and, in general, indicates that at the low aspect ratios the lift-curve slope is independent of plan form and at the high aspect ratios the experimental values of lift-curve slope are slightly less than those predicted by theory when the section lift-curve slope is assumed to be 0.10 per degree.

Longitudinal Stability Characteristics

The rectangular wings 1, 2, and 3 showed no change in longitudinal stability with a decrease in aspect ratio. The unswept tapered wings 4 to 10, however, showed at low angles of attack a tendency toward lower longitudinal stability with decreased aspect ratio similar to that previously reported in reference 1. Although for the low-aspect-ratio unswept wings 8, 9, and 10 there was no marked change of static margin \((-\frac{\Delta C_M}{\Delta C_L})\) with aspect ratio (fig. 4), within this range of aspect ratios (3.0 to 0.5) the sweptback wings 11 to 13 showed an increase in static margin with increasing sweepback and decreasing aspect ratio (figs. 5 and 6). This effect of sweep on the static margin is illustrated in figure 9 which indicates the rearward movement of the aerodynamic center relative to the quarter chord of the mean aerodynamic chord as the sweepback increases. The extrapolated curve for the triangular wings in figure 9 indicates that the aerodynamic center is probably located at approximately the 25-percent mean aerodynamic chord for zero sweepback and approaches the 50-percent mean aerodynamic chord (or the center of area) for the hypothetical case of a triangular wing with 90° sweepback (reference 9).

On the triangular and swept tapered wings the static longitudinal stability at the stall decreased with an increase in sweepback. Reference 10 which includes data for wing 6 and wings 9 to 17 as well as other plan forms tested at different scales shows that as the sweepback is increased low aspect ratios must be used to maintain satisfactory longitudinal stability at the stall.
Lateral Stability Characteristics

Static stability. - The effective dihedral increased with lift coefficient but, since this variation in most cases was not linear, it was not possible to compare the data for the different wings by the values of $\frac{dC_\beta}{dC_L}$ (rate of change of effective dihedral with lift coefficient). The changes in effective dihedral with aspect ratio are indicated instead in figure 10 for an arbitrary lift coefficient of 0.4. The effective dihedral increases with decreasing aspect ratio with the greatest change at the very low aspect ratios. The experimental results are compared with the equation

$$C_\beta = \frac{1}{2\pi} \frac{2C_L}{\pi L}$$

which was derived in reference 8. This equation was derived for triangular wings of aspect ratio less than 1.0 but a consideration of the assumptions involved in its derivation indicates that it should be applicable to wings of higher aspect ratios. Theory indicates the same general trend as the experimental results but the experimental values of effective dihedral are considerably less than the theoretical values.

The directional stability increased with increase in lift coefficient for all of the wings except wings 14 and 18. The variation of directional stability with aspect ratio at a lift coefficient of 0.4 is presented in figure 11. This figure indicates that the directional stability increases with decreasing aspect ratio except for aspect ratios below 1.0, at which, decreasing the aspect ratio decreased the directional stability.

Damping in roll. - The unswept wings 1 to 8 showed no consistent variation of damping in roll with lift coefficient except for angles of attack near maximum lift, at which the damping in roll decreased toward instability. Wings 9 and 10 showed an inconsistent variation of damping in roll through the lift-coefficient range and, in general, had less damping at the higher lift coefficients. The swept wings 11 to 18 showed a decrease in damping in roll with increasing lift coefficient before the maximum lift was reached. This decrease in damping in roll with lift coefficient for highly tapered sweptback wings is probably caused by a premature wing-tip stall.

A cross plot showing the variation of damping in roll with aspect ratio is presented in figure 12. For the flat-plate-airfoil wings, the damping-in-roll values at zero lift are given but, for the...
Cambered-airfoil wings, the maximum values of damping in roll are given because doubtful values are obtained at zero lift because of the possible separation from the lower surface of the wing. This figure shows the usual trend of decreasing damping in roll with decreasing aspect ratio. The theoretical variation of the values of the damping-in-roll derivative \( C_{l\alpha} \) with aspect ratio for aspect ratios above 3.0 were obtained from reference 11. A section-lift-curve slope of 0.10 per degree was assumed. For aspect ratios below 1.0 the following equation

\[
C_{l\alpha} = \frac{x_A}{32}
\]

obtained from reference 3 was assumed to be valid. The theoretical curves were faired in for the aspect ratios between 1.0 and 3.0.

**CONCLUSIONS**

The results of the tests made in the Langley free-flight tunnel and the 15-foot free-spinning tunnel to determine the static stability and damping-in-roll characteristics of low-aspect-ratio wings may be summarized as follows:

1. Although for the rectangular wings there was no change in the longitudinal stability with aspect ratio, the unswept tapered wings showed a tendency toward decreased longitudinal stability at low angles of attack as the aspect ratio was reduced.

2. For sweptback wings of approximately triangular plan form there was a rearward movement of the aerodynamic center with respect to the quarter chord of the mean aerodynamic chord as the sweepback increased. Results indicate that for triangular wings the aerodynamic center moves from approximately 25 percent to 50 percent of the mean aerodynamic chord (or the center of area) as the sweepback is varied from 0° to 90°.

3. The effective dihedral and directional stability in most cases increased with increase in lift coefficient.

4. At low lift coefficients the effective dihedral and directional stability increased with decreasing aspect ratio except that the directional stability of the wings of aspect ratio below 1.0 decreased sharply with decrease in aspect ratio.
5. Unswept wings showed no consistent variation of damping in roll with lift coefficient except for angles of attack near maximum lift where the damping in roll decreased toward instability. The triangular and swept tapered wings in general showed a reduction in damping in roll with increasing lift coefficient and in some cases became unstable before maximum lift was reached.

6. The damping in roll decreased with aspect ratio as would be expected. The experimental values of damping in roll were generally smaller than the theoretical values.

Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va., July 21, 1947
REFERENCES


### TABLE 1
DIMENSIONAL CHARACTERISTICS OF THE WINGS

<table>
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* Typical airfoil sections taken in planes parallel to the plane of symmetry are shown in the following sketches:

- **Rhode St. Genese 35° airfoil**  
  (Designated as RSG, coordinates given in reference 12)

- **Typical flat plate airfoil**  
  (Designated as FP)
Figure 1.- The stability system of axes. Arrows indicate positive directions of moments and forces. This system of axes is defined as an orthogonal system having the origin at the center of gravity and in which the Z-axis is in the plane of symmetry and perpendicular to the relative wind, the X-axis is in the plane of symmetry and perpendicular to the Z-axis, and the Y-axis is perpendicular to the plane of symmetry.
Figure 2. - Aerodynamic characteristics of rectangular wings with conventional airfoil. (Wings 1, 2, and 3 of Table 1.)
Figure 3. - Aerodynamic characteristics of unswept tapered wings with conventional airfoil.
(Wings 4, 5, 6, and 7 of Table 1.)
Figure 4.- Aerodynamic characteristics of unswept tapered wings with flat-plate airfoil. (Wings 8, 9, and 10 of Table 1.)
Figure 5. - Aerodynamic characteristics of triangular wings with flat-plate airfoil.
(Wings 11, 12, 13, and 14 of Table 1.)
Figure 6 - Aerodynamic characteristics of swept tapered wings with flat-plate airfoil.
(Wings 15, 16, 17, and 18 of Table 1.)
Figure 7.- Variation of maximum lift coefficient with aspect ratio. (Wings 1 to 18 of Table 1.)
Figure 8. - Variation of lift-curve slope with aspect ratio at $C_L = 0$. (Wings 1 to 18 of Table 1.)
Figure 9. - Variation of aerodynamic-center position with sweepback. (Wings 11 to 18 of Table 1.)
Figure 10.- Variation of effective dihedral parameter with aspect ratio at $C_L = 0.4$.
(Wings 1 to 18 of Table 1.)
Figure 11.- Variation of directional stability parameter with aspect ratio at $C_L = 0.4$.
(Wings 1 to 18 of Table 1.)
Figure 12. Variation of damping-in-roll parameter with aspect ratio at $C_L = 0$ for wings 8 to 18 and at maximum values of damping in roll for wings 1 to 7. (Wings 1 to 18 of Table 1.)
SUBSONIC CHARACTERISTICS OF
LOW ASPECT RATIOS
VOLUME II. WIND-TUNNEL TEST RESULTS

TECHNICAL DOCUMENTARY REPORT No. FDL-TDR-64-103,
VOLUME II

AF FLIGHT DYNAMICS LABORATORY
RESEARCH AND TECHNOLOGY DIVISION
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

Project No. 1366, Task No. 136613

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General Dynamics Corporation, New York 20, N. Y.;
Authors: J. K. Beamish, G. J. Klein, and E. L. Crosthwait)
### Table 1

**SUMMARY OF MODEL DIMENSIONS**

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<th>MODEL NO.</th>
<th>CONFIG. SYMBOL</th>
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<th>TAPER RATIO (NOMINAL)</th>
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<th>T.E. Sweep deg</th>
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<th>TIP CHORD ft.</th>
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<th>MEAN AERO. C. ft.</th>
<th>AIRFOIL SECTION</th>
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<td>15</td>
<td>V₁/₅</td>
<td>1.076</td>
<td>0</td>
<td>75</td>
<td>0</td>
<td>0.0208</td>
<td>1.704</td>
<td>3</td>
<td>2.082</td>
<td>1.398</td>
<td>CYM 0018</td>
<td>0.0357</td>
<td>1.0253</td>
<td>1.0556</td>
<td>8.097</td>
</tr>
<tr>
<td>16</td>
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<td>0</td>
<td>75</td>
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<td>0.0208</td>
<td>1.704</td>
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<td>2.082</td>
<td>1.398</td>
<td>CYM 0018</td>
<td>0.0357</td>
<td>1.0253</td>
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<td>8.097</td>
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<tr>
<td>17</td>
<td>V₁/₇</td>
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<td>0</td>
<td>75</td>
<td>0</td>
<td>0.0208</td>
<td>1.704</td>
<td>3</td>
<td>2.082</td>
<td>1.398</td>
<td>CYM 0018</td>
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<td>18</td>
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<td>2.082</td>
<td>1.398</td>
<td>CYM 0018</td>
<td>0.0357</td>
<td>1.0253</td>
<td>1.2009</td>
<td>8.097</td>
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<td>19</td>
<td>V₁/₉</td>
<td>1.076</td>
<td>0</td>
<td>75</td>
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<td>0.0208</td>
<td>1.704</td>
<td>3</td>
<td>2.082</td>
<td>1.398</td>
<td>CYM 0018</td>
<td>0.0357</td>
<td>1.0253</td>
<td>1.2009</td>
<td>8.097</td>
</tr>
</tbody>
</table>
Table 2
CAMBER AND SECTION ORDINATES OF THE CLARK Y MODIFIED AIRFOIL (CYM)

<table>
<thead>
<tr>
<th>CAMBER LINE</th>
<th>CYM 0010</th>
<th>CYM 0018</th>
<th>CYM 0025</th>
<th>CYM 0030</th>
</tr>
</thead>
<tbody>
<tr>
<td>Xc</td>
<td>Yc</td>
<td>dy_c/dXc</td>
<td>X</td>
<td>Y</td>
</tr>
<tr>
<td>0</td>
<td>0</td>
<td>.1998</td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>1.25</td>
<td>.25</td>
<td>.1989</td>
<td>.94</td>
<td>1.80</td>
</tr>
<tr>
<td>2.50</td>
<td>.51</td>
<td>.1950</td>
<td>2.08</td>
<td>2.65</td>
</tr>
<tr>
<td>5.00</td>
<td>.99</td>
<td>.1817</td>
<td>4.47</td>
<td>3.91</td>
</tr>
<tr>
<td>7.50</td>
<td>1.42</td>
<td>.1656</td>
<td>6.93</td>
<td>4.87</td>
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<tr>
<td>10.00</td>
<td>1.79</td>
<td>.1468</td>
<td>9.43</td>
<td>5.65</td>
</tr>
<tr>
<td>20.00</td>
<td>2.89</td>
<td>.0796</td>
<td>19.62</td>
<td>7.65</td>
</tr>
<tr>
<td>30.00</td>
<td>3.40</td>
<td>.0323</td>
<td>29.84</td>
<td>8.40</td>
</tr>
<tr>
<td>40.00</td>
<td>3.60</td>
<td>0</td>
<td>40.00</td>
<td>8.43</td>
</tr>
<tr>
<td>50.00</td>
<td>3.47</td>
<td>-.0253</td>
<td>50.11</td>
<td>7.88</td>
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<tr>
<td>60.00</td>
<td>3.12</td>
<td>-.0472</td>
<td>60.18</td>
<td>6.92</td>
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<tr>
<td>70.00</td>
<td>2.53</td>
<td>-.0638</td>
<td>70.19</td>
<td>5.61</td>
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<tr>
<td>80.00</td>
<td>1.84</td>
<td>-.0778</td>
<td>80.17</td>
<td>4.02</td>
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<tr>
<td>90.00</td>
<td>.98</td>
<td>-.0901</td>
<td>90.11</td>
<td>2.18</td>
</tr>
<tr>
<td>95.00</td>
<td>.49</td>
<td>-.0954</td>
<td>95.06</td>
<td>1.16</td>
</tr>
<tr>
<td>100.00</td>
<td>.06</td>
<td>-.0989</td>
<td>-</td>
<td>-</td>
</tr>
</tbody>
</table>

x and y in % chord along and normal to chord (reference). Leading edge radius = 110 (t/c)% chord with center on a line from x = 0 inclined 11.3 deg to chord. Trailing edge radius = 1.05 (t/c)% chord with center on line from x = 100 inclined 5.65 deg to chord.
NOTE:
CYM = CLARK Y MODIFIED
= NACA 4-DIGIT AIRFOIL SUPERIMPOSED ON THE MODIFIED CLARK Y MEAN CAMBER LINE

DIMENSIONS IN INCHES

Figure 1 - SUMMARY OF MODEL GEOMETRY; AIRFOIL STUDY
AIRFOIL ~ CYM 0018

Figure 2 - SUMMARY OF MODEL GEOMETRY; PLANFORM STUDY
Figure 3 - STABILITY AXIS SYSTEM
$W_1 \sim \#60$ CARBORUNDUM GRIT
$W_2 \rightarrow W_9 \sim \#46$ CARBORUNDUM GRIT

Figure 5 - TRANSITION GRIT PATTERNS
W_2-2CYM 0018

R = 0.75 IN
75°
36,000 IN.
-20.448 IN.

(Lowered Cover Plate)

RUNS 30, 33 & 36
REYNOLDS NO. = 1.26 x 10^6
MACH NO. = 0.094
q = 13 PSF
R = 1.07/8

<table>
<thead>
<tr>
<th>DATA SYMBOL</th>
<th>GRIT NUMBER</th>
</tr>
</thead>
<tbody>
<tr>
<td>△</td>
<td>46</td>
</tr>
<tr>
<td>●</td>
<td>60</td>
</tr>
<tr>
<td>○</td>
<td>No Grit</td>
</tr>
</tbody>
</table>

Figure 6 - FORCE AND MOMENT DATA: Grit Study at Reynolds No. = 1.26 x 10^6
FORCE & MOMENT DATA
GRIT STUDY

CVAL 355
\(W_a \sim CYM0018\)
(LOWERED COVER PLATE)

FREE TRANSITION

RUNS 31, 34 & 37
REYNOLDS NO. = 2.76 \times 10^6
MACH NO. = .201
\(q_0 = 60\) PSF
\(\frac{A}{R} = 1.0718\)

<table>
<thead>
<tr>
<th>DATA SYM</th>
<th>GRIT NO.</th>
</tr>
</thead>
<tbody>
<tr>
<td>(\Delta)</td>
<td>46</td>
</tr>
<tr>
<td>(\Box)</td>
<td>60</td>
</tr>
<tr>
<td>(\bigcirc)</td>
<td>NO GRIT</td>
</tr>
</tbody>
</table>

\(C_w\) vs. \(C_t\)

Figure 7 - FORCE AND MOMENT DATA: GRIT STUDY AT REYNOLDS
NO. = 2.76 \times 10^6
WZ-CYM 0018

RUNS 32, 35 & 38
REYNOLDS NO. = 5.14 x 10^6
MACH NO. = .368
q_e = 200 PSF
AR = 1.0718
C_L = .021
t/c = .18

Figure 8 - FORCE AND MOMENT DATA: GRIT STUDY AT REYNOLDS NO. = 5.14 x 10^6
Figure 9.- FORCE AND MOMENT DATA: REYNOLDS NO. EFFECTS
CONSTANT DYNAMIC PRESSURE

<table>
<thead>
<tr>
<th>DATA SYM.</th>
<th>RUN</th>
<th>q(PSF)</th>
<th>MODEL SYM.</th>
<th>MACH NO.</th>
<th>R N</th>
</tr>
</thead>
<tbody>
<tr>
<td>○</td>
<td>44</td>
<td>60</td>
<td>W₁</td>
<td>.201</td>
<td>$1.84 \times 10^6$</td>
</tr>
<tr>
<td>○</td>
<td>45-1</td>
<td>200</td>
<td>W₁</td>
<td>.368</td>
<td>$3.42 \times 10^6$</td>
</tr>
<tr>
<td>△</td>
<td>4-0</td>
<td>60</td>
<td>W₂</td>
<td>.201</td>
<td>$2.76 \times 10^6$</td>
</tr>
<tr>
<td>△</td>
<td>4-1-1</td>
<td>200</td>
<td>W₂</td>
<td>.368</td>
<td>$5.14 \times 10^6$</td>
</tr>
</tbody>
</table>
Figure 10 – FORCE AND MOMENT DATA: REYNOLDS NO. EFFECTS ON MODEL 2
Figure 13 - FORCE AND MOMENT DATA: REYNOLDS NO. EFFECTS ON MODEL 19
Figure 11 - FORCE AND MOMENT DATA: REYNOLDS NO. EFFECTS ON MODEL 11
**Figure 12 - FORCE AND MOMENT DATA: REYNOLDS NO. EFFECTS ON MODEL 15**

\[ C_{x_0} = 0.04 \]

\[ R = 0.70532 \]

<table>
<thead>
<tr>
<th>DATA SYM</th>
<th>RUN</th>
<th>( q ) (psf)</th>
<th>MACH NO.</th>
<th>( R_N )</th>
</tr>
</thead>
<tbody>
<tr>
<td>○</td>
<td>23</td>
<td>60</td>
<td>.201</td>
<td>2.81 \times 10^6</td>
</tr>
<tr>
<td>△</td>
<td>24</td>
<td>80</td>
<td>.232</td>
<td>3.26 \times 10^6</td>
</tr>
<tr>
<td>□</td>
<td>25</td>
<td>120</td>
<td>.285</td>
<td>4.00 \times 10^6</td>
</tr>
</tbody>
</table>

\( W_{15} \sim CYM 0018 \)

CVAL 355A
Figure 14 - FORCE AND MOMENT DATA: THICKNESS STUDY
CVAL 355
TYPICAL PLANFORM

R = 75 IN
N = 36,000 IN
20.448 IN

<table>
<thead>
<tr>
<th>DATA SYM</th>
<th>RUN</th>
<th>MODEL SYM</th>
<th>AIRFOIL SECT</th>
</tr>
</thead>
<tbody>
<tr>
<td>△</td>
<td>40</td>
<td>W2</td>
<td>CYM 0018</td>
</tr>
<tr>
<td>▽</td>
<td>54</td>
<td>W6</td>
<td>NACA 4418 M</td>
</tr>
<tr>
<td>◊</td>
<td>55</td>
<td>W7</td>
<td>NACA 653618(α=5)</td>
</tr>
<tr>
<td>▣</td>
<td>56</td>
<td>W8</td>
<td>NACA 25018</td>
</tr>
</tbody>
</table>

REYNOLDS NO. = 2.76 x 10^6   MACH NO. = 201   q = 60 PSF
\[ \frac{AR}{\lambda} = 1.0718 \]

Figure 15 - FORCE AND MOMENT DATA: AIRFOIL SECTION STUDY
Figure 16 - FORCE AND MOMENT DATA: PLANFORM NOSE RADIUS STUDY
Figure 17 - FORCE AND MOMENT DATA: TAPER RATIO STUDY AT CONSTANT SWEEP, \( \lambda = 63 \) DEG.
Figure 18 - FORCE AND MOMENT DATA: TAPER RATIO STUDY AT CONSTANT SWEET, \( \lambda = 75 \) DEG.
AIRFOIL CYM0018

MACH NO. = .201
q = 60 psf.  

<table>
<thead>
<tr>
<th>DATA SYMBOL</th>
<th>MODEL SYMBOL</th>
<th>RUN</th>
<th>AR</th>
<th>R_N</th>
</tr>
</thead>
<tbody>
<tr>
<td>●</td>
<td>W₂₆</td>
<td>1</td>
<td>1.076</td>
<td>2.74 × 10⁶</td>
</tr>
<tr>
<td>●</td>
<td>W₁₁</td>
<td>6</td>
<td>2.040</td>
<td>2.67 × 10⁶</td>
</tr>
<tr>
<td>●</td>
<td>W₁₄</td>
<td>21</td>
<td>1.619</td>
<td>2.69 × 10⁶</td>
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<tr>
<td>●</td>
<td>W₁₅</td>
<td>23</td>
<td>.713</td>
<td>2.81 × 10⁶</td>
</tr>
</tbody>
</table>

Figure 19 - FORCE AND MOMENT DATA: ASPECT RATIO STUDY
AT CONSTANT TAPER RATIO, λ = 0
Figure 20 - FORCE AND MOMENT DATA: ASPECT RATIO STUDY
AT CONSTANT TAPER RATIO, \( \lambda = .3 \)
**Figure 21 - Force and Moment Data: Sweep Study at Constant Aspect Ratio, \( \mathcal{A} \approx 0.71 \)**
### Figure 22 - Force and Moment Data: Sweep Study at Constant Aspect Ratio, $\AR \sim 1.08$

#### AIRFOIL CYM 0018

MACH NO = .201

$q = 60 \text{ psf}$

CVAL 355A

<table>
<thead>
<tr>
<th>DATA SYM</th>
<th>MODEL SYM</th>
<th>RUN</th>
<th>$\alpha_{LE}$</th>
<th>$\alpha_{NE}$</th>
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<tbody>
<tr>
<td>$\square$</td>
<td>$W_{2a}$</td>
<td>1</td>
<td>75°</td>
<td>2.74x10^{-6}</td>
</tr>
<tr>
<td>$\triangle$</td>
<td>$W_{3}$</td>
<td>12.1</td>
<td>63°</td>
<td>2.86x10^{-6}</td>
</tr>
</tbody>
</table>

$C_{L,a} = .024$

$C_{\alpha,\infty} = .024$

$$\AR = 1.0718$$

$W_{2a} = 20448$

$W_{3} = 26340$

$\AR = 1.1$
Figure 23 - FORCE AND MOMENT DATA: SWEEP STUDY AT CONSTANT ASPECT RATIO, $\mathcal{R} \sim 1.64$
A theory

Schlichting/Truckenbrodt approx.

Schlichting/Truckenbrodt approx.

Lifting line theory

Lifting surface theory

Low A theory

Hoerner data:

$\alpha = \text{Round and rounded wings}$

$\alpha = \text{Profiled delta wings}$

$1/\alpha = 10.5 + 23/A + 16/A^2$

K.D. Wood approx. $\alpha = \frac{2}{90} \frac{A}{3+A}$ for $\Lambda = 0$

X-ray approx. $\alpha = \frac{n^2}{90} \frac{A}{A+2}$

Approx. $\alpha = \frac{n^2}{90} \sqrt{A+4+2}$

$\Lambda = \alpha + \beta$

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$\Lambda = \alpha + \beta$
Please complete the attached questionnaire and return by July 8, 1977.

Route to: Susan Motley
M.S. 185

1. What is your current address?
   W. J. SMALL
   MS 180A
   LANGLEY RESEARCH CENTER
   HAMPTON, VA. 23665

2. How many copies of the Datcom do you have? ONE (1)

PART II (Use reverse side if more space is needed.)

1. Indicate the Datcom sections you have found
   (a) most useful: Sec 4 (BASIC BODY WING 3200 LIFT DRAG)
   (b) least useful: Sec 9 - NOT IN OUR AREA WHEN 250 FIXED翼
   AERODYNAMICS!

2. What sections have been found inadequate due to erroneous information?
   (List section and erroneous information.)

3. What additional subjects would increase the Datcom's utility for you?
   Present subject list adequate, see part 9.
   Subjects could be expanded with more experimental data & more than one solution method would be good.

4. Do you have a stability and control handbook of your own making?
   Yes [ ] No [ ]

ATTACHMENT 1
DATCOM QUESTIONNAIRE

Please complete and return to the following address no later than 8/31/77:
AFSC Liaison Office, Mail Stop 221
Nicholas G. Yretakis

PART 1

1. What is your current address?
   W.J. SMALL
   MS 130A
   LANGLEY RESEARCH CENTER
   HAMPTON, VA. 23665

2. How many copies of the Datcom do you have? ONE (1)

PART II (Use reverse side if more space is needed.)

1. Indicate the Datcom sections you have found
   (a) most useful: SEC 4 (BASIC BODY DRIVING ENV LIFT DRAIN)
   (b) least useful: SEC 9 - NOT IN OUR AREA WHICH IS HYPERSONIC AERODYNAMICS

2. What sections have been found inadequate due to erroneous information?
   (List section and erroneous information.)

3. What additional subjects would increase the Datcom's utility for you?
   a. Present subject list adequate. See part 9.
   b. Subjects could be expanded with more experiments.

4. Do you have a stability and control handbook of your own making?
   Yes ________; No ________

ATTACHMENT 1
5. Do you find your own methods more useful than those of the Datcom?

Yes ______; No ______. If so, why?

Our methods using high speed computers are used to supplement Datcom's.

6. Identify other non-proprietary methods or references that you use in place of or in addition to the Datcom. (Include those related to parameters of interest not presently included in the Datcom.)

- Mark III W Hypersonic Arbitrary-Body Program
  Ref. AFFDL-TR-73-157
- Subsonic Vent-Stream Program, Ref. NASA TN D-7231

7. List any sections whose value would be enhanced by increased substantiation of information related to the methods employed.

All sections dealing with subsonic prediction methods, along with more information on low aspect ratio wings, and large fuselage configurations compared to wing only.

8. Underline the extent to which you use the Datcom:

Constantly; Frequently; Occasionally; Seldom; Never.

9. Please provide additional comments which would aid us in planning future revisions.

- Section on propulsion integration would be a great help.
$M = 0$

Platfrom Area Ref. of ea., $A$

- Extended Wing $A = 1.42$
- H-T Wing $A = 9.42$
- Clip Wing $A = 0.618$

$C_l$, deg

Estimates made by fitting curve of $C_l / M$ for wings of various $A$. 
4.7 GROUND EFFECTS AT ANGLE OF ATTACK

In subsequent Sections methods are presented for estimating the effects of the ground plane on the lift in the linear range, on the maximum lift, and on the pitching moment. A brief discussion of the physical aspects of ground-plane effects is given in Section 4.7.1. A more thorough discussion, however, is given in reference 1. A list of notation used in subsequent Sections is given below.

NOTATION

- \( \Lambda_0 \) effective wing aspect ratio in presence of ground
- \( b_{eff} \) effective wing span
- \( b_f \) effective span for increment in load due to flaps
- \( b'_w \) effective span for unflapped wing
- \( C_{t,0} \) lift coefficient of unflapped wing in absence of ground plane (no slipstream effect)
- \( \Delta C_{\ell,0} \) increment in lift coefficient due to flaps in absence of ground plane (no slipstream effect)
- \( H \) height of .25-M.A.C. point of wing above ground
- \( H_h \) height of .25-M.A.C. point of horizontal tail or aft wing above ground
- \( \epsilon \) downwash angle in absence of ground plane (includes that due to slipstream for propeller aircraft)
- \( (\Delta \epsilon)_0 \) decrease in downwash angle due to presence of ground
- \( x_{ac} \) distance from leading edge of wing M.A.C. to aerodynamic center of wing
- \( x \) distance from aerodynamic center of lifting surface affected by downwash to the moment center

4.7.1 GROUND EFFECTS ON LIFT VARIATION WITH ANGLE OF ATTACK

The lift of a vehicle at a given angle of attack in the presence of the ground plane is increased as a result of the restriction of the flow by the ground plane. Generally, the change in lift of a vehicle is composed of the following increments:

1. change in wing-body lift due to the ground plane
2. change in tail-body lift due to the ground plane
3. change in lift due to the effect of the ground plane on downwash

These three increments are discussed in the following paragraphs (items 1 and 2 are treated jointly).

Change in Wing-Body or Tail-Body Lift Due to Ground Plane

There are three effects of the ground plane on the lift of a wing—the decrease in velocity of the oncoming flow in the vicinity of the wing due to the field of the reflected bound vortex, the change in the effective camber of the wing due to the distortion of the flow caused by the reflected bound vortex, and the decreased induced upwash angle at the wing due to the upflow associated with the reflected trailing vortices. The first two effects are opposite and approximately equal (reference 1); therefore the ground effect on lift can be calculated with sufficient accuracy if the last one alone is considered.
Change in Lift Due to the Effect of Ground Plane on Downwash

The change in downwash of a wing in the vicinity of a ground plane is derived theoretically in reference 1. The change in downwash is derived by considering the flow field to be represented by the real vortex system and an image vortex system reflected from the ground plane. The vortex system can be further simplified by representing it as an equivalent horseshoe vortex (with a calculated effective span). A modification to the method of reference 1 is given in reference 2, wherein certain geometric terms are redefined. This modified method is presented in this Section. The accuracy of the method compared to experimental data is shown in reference 2 to be within \( \pm 2^\circ \) of downwash angle.

Special notation used in this Section is given in Section 4.7.

**DATCOM METHOD**

Change in Wing-Body or Tail-Body Lift Due to Ground Plane

The change in lift coefficient at a given angle of attack due to the presence of the ground plane is given by the equation

\[
(\Delta C_l)_a = 57.3 \frac{C_lC_{1\alpha}}{\pi A} \left(1 - \frac{A}{A_0}\right)
\]

where

- \( C_l \) is the lift coefficient of the wing or tail—including the effects of flaps and power—at the angle of attack under consideration in free air
- \( C_{1\alpha} \) is the lift-curve slope of the wing or tail in free air
- \( \frac{A}{A_0} \) is the ratio of the actual to the effective aspect ratio and is given in figure 4.7.1-5a

Change in Lift Due to the Effect of Ground Plane on Downwash

The change in lift due to the effect of the ground plane on downwash is given by the equation

\[
(\Delta C_l)_\epsilon = C_{1\alpha} (\Delta \epsilon)_a
\]

where

\[
(\Delta \epsilon)_a = \epsilon \left[ \frac{b_{eff}^2 + 4(H_l - H)^2}{b_{eff}^2 + 4(H_l + H)^2} \right]
\]

(See Section 4.7 for notation.)

The parameter \( \epsilon \) is the downwash angle at the point of interest in the absence of a ground plane.

The effective wing span, \( b_{eff} \), is calculated from the equation

\[
b_{eff} = \frac{C_{1\alpha} + \Delta C_{1\epsilon}}{\frac{C_{1\alpha}}{b_w} + \frac{\Delta C_{1\epsilon}}{b_\epsilon}}
\]

where \( b_w \) and \( b_\epsilon \) are calculated from the equations

\[
b_w = \left( \frac{b_\epsilon}{b} \right) b
\]

4.7.1-2
The ratios \( \frac{b_r'}{b} \) and \( \frac{b_r'}{b_w'} \) are given in figures 4.7.1-5b and 4.7.1-6, respectively.

**Total Change in Vehicle Lift Due to Ground Plane**

The total change in vehicle lift due to the presence of the ground plane is given by the equation

\[
\Delta C_L = (\Delta C_L)_w \frac{q_w}{q} + (\Delta C_L)_H \frac{S_H}{S_w} \frac{q_H}{q_\infty} + (\Delta C_L)_\epsilon \frac{q}{q_\infty}
\]

where the incremental lift values are derived according to the previous paragraphs and \( \frac{q}{q_\infty} \) is obtained from Section 4.4.1. The incremental lift is calculated for a range of angles of attack over the linear lift range and the complete lift curve is then constructed. The term \( (\Delta C_L)_\epsilon \) is based on wing dimensions.

**Sample Problem**

(tail-last configuration)

Given:

- \( \epsilon = 16^\circ \)
- \( \lambda = .226 \)
- \( C_{L_w} = .85 \) (flaps up)
- \( \frac{b_r'}{b} = .44 \)
- \( (\Delta C_L)_{flaps} = .28 \)
- \( C_{L_H} = .053 \) (per deg)
- \( b_H = 11.3 \) ft
- \( \epsilon = 16.1^\circ \)
- \( S_w = 260 \) ft\(^2\)
- \( b = 27.5 \) ft
- \( \lambda_H = .225 \)
- \( A_H = 2.8 \)
- \( H = 6.2 \) ft
- \( \lambda_H = .225 \)
- \( H_H = 11.1 \) ft
- \( i_H = -12^\circ \)

Compute:

- \( \frac{2H}{b} = .451 \)
- \( \frac{\frac{A}{A_0}}{A_0} = .735 \) (figure 4.7.1-5a)

\[
(\Delta C_L)_w = 57.3 \left( \frac{C_{L_w} + (\Delta C_L)_w}{\pi A} \right) \frac{C_{L_H}}{\pi (A)} \left( 1 - \frac{A}{A_0} \right) \quad \text{(equation 4.7.1-a)}
\]

\[
= 57.3 \left( \frac{.85 + .28}{.053} \right) \left( .052 \right) \left( 1 - .735 \right) = .100
\]

- \( \frac{2H_H}{b_H} = 1.96 \)
- \( \frac{\frac{A}{A_0}}{A_H} = .985 \) (figure 4.7.1-5a)
- \( \alpha_H = \alpha + i_H - \epsilon = -12.1^\circ \)
- \( C_{L_H} = .052 \) \(-12.1\) = -.629

4.7.1-3
\[(\Delta C_L)_H = 57.3 \frac{C_{t,\text{H}} C_{l,\text{H}}}{\pi A_H} \left(1 - \frac{A}{A_0}\right)_H \quad \text{(equation 4.7.1-a)}\]

\[= 57.3 \frac{(-.629)(.052)}{\pi 2.8} (1 - .985) = -.0032\]

\[\frac{b_w'}{b} = .73 \quad \text{(figure 4.7.1-5b)}\]

\[b_t' = .62 \quad \text{(figure 4.7.1-6)}\]

\[b_w' = 21.2 \text{ ft} \quad \text{(equation 4.7.1-e)}\]

\[b_t' = 12.4 \text{ ft} \quad \text{(equation 4.7.1-f)}\]

\[b_{\text{eff}} = \frac{C_{l,\text{eff}} + \Delta C_{l,t}}{C_{l,\text{eff}} + \Delta C_{l,t}} = 17.4 \text{ ft} \quad \text{(equation 4.7.1-d)}\]

\[\frac{b_w'}{b_t'} = \frac{\Delta C_{l,t}}{\Delta C_{l}}\]

\[\frac{(\Delta \epsilon)_a}{\Delta \epsilon} = \frac{\left[\frac{b_{\text{eff}}^2}{b_{\text{eff}}^2 + 4(H_0 - H)^2}\right]}{\left[\frac{b_{\text{eff}}^2}{b_{\text{eff}}^2 + 4(H_0 + H)^2}\right]} = 4.28^\circ \quad \text{(equation 4.7.1-c)}\]

\[\frac{(\Delta C_{l})_L}{\epsilon} = .223 \quad \text{(equation 4.7.1-b)}\]

\[\frac{q_{\text{H}}}{q_{\infty}} = .95 \quad \text{(estimated)}\]

\[\Delta C_l = (\Delta C_{l,\text{w}}) + \frac{S_n}{S_w} \frac{q_{\text{H}}}{q_{\infty}} \left[(\Delta C_{l})_H + (\Delta C_{l})_L\right] \quad \text{(equation 4.7.1-g applied to tail last configurations)}\]

\[= .105 + \frac{45.85}{260} (.95) [-.0320 + .238] = .137\]

**REFERENCES**


FIGURE 4.7.1-5a EFFECTIVE ASPECT RATIO IN THE PRESENCE OF THE GROUND

FIGURE 4.7.1-5b EFFECTIVE SPAN OF WING IN THE PRESENCE OF THE GROUND
FIGURE 4.7.1-6 EFFECITIVE SPAN OF FLAPS IN THE PRESENCE OF THE GROUND
TAKEOFF/TOUCHDOWN SPEED

\( W/S = 50 \text{ lb/ft}^2 \)
\[ A = \frac{114}{61} = 1.869 \]

\[ \%e = 0.05 \]

\[ %e = 0.05 \]

\[ A = 1.8652 \]
$$\% C = 0.05$$
$$R = \frac{C}{4} + 75^\circ \approx 1.0716$$
<table>
<thead>
<tr>
<th>X</th>
<th>1</th>
<th>2+4</th>
<th>( \sqrt{3} )</th>
<th>2+4</th>
<th>( \frac{2 \pi}{5} )</th>
<th>( \frac{\text{Chord}}{A} )</th>
</tr>
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<tr>
<td>0</td>
<td>0</td>
<td>4</td>
<td>2.00</td>
<td>4.00</td>
<td>1.5708</td>
<td></td>
</tr>
<tr>
<td>1</td>
<td>1</td>
<td>5</td>
<td>2.236</td>
<td>4.236</td>
<td>1.483</td>
<td></td>
</tr>
<tr>
<td>2</td>
<td>4</td>
<td>8</td>
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<td>4.828</td>
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<tr>
<td>3</td>
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<td>13</td>
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<td>5.606</td>
<td>1.121</td>
<td></td>
</tr>
<tr>
<td>4</td>
<td>14</td>
<td>20</td>
<td>4.472</td>
<td>6.472</td>
<td>0.9708</td>
<td></td>
</tr>
<tr>
<td>5</td>
<td>25</td>
<td>29</td>
<td>5.385</td>
<td>7.385</td>
<td>0.8508</td>
<td></td>
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<tr>
<td>6</td>
<td>36</td>
<td>40</td>
<td>6.325</td>
<td>8.325</td>
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<td>53</td>
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<td>9.280</td>
<td>0.6771</td>
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</tr>
<tr>
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<td>10.246</td>
<td>0.6132</td>
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</tr>
<tr>
<td>9</td>
<td>81</td>
<td>85</td>
<td>9.220</td>
<td>11.220</td>
<td>0.5600</td>
<td></td>
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<tr>
<td>10</td>
<td>100</td>
<td>104</td>
<td>10.198</td>
<td>12.198</td>
<td>0.5151</td>
<td></td>
</tr>
<tr>
<td>11</td>
<td>121</td>
<td>125</td>
<td>11.18</td>
<td>13.180</td>
<td>0.4767</td>
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<tr>
<td>12</td>
<td>144</td>
<td>148</td>
<td>12.146</td>
<td>14.146</td>
<td>0.4435</td>
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<tr>
<td>13</td>
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<td>173</td>
<td>13.133</td>
<td>15.133</td>
<td>0.4146</td>
<td></td>
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<tr>
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<td>196</td>
<td>200</td>
<td>14.142</td>
<td>16.142</td>
<td>0.3892</td>
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<td>15</td>
<td>225</td>
<td>229</td>
<td>15.153</td>
<td>17.153</td>
<td>0.3667</td>
<td></td>
</tr>
<tr>
<td>16</td>
<td>256</td>
<td>260</td>
<td>16.125</td>
<td>18.125</td>
<td>0.3467</td>
<td></td>
</tr>
</tbody>
</table>
Relation of LE & CL Sweep, Span and Root & Tip Chords

\[ A.R. = \frac{b^2}{A} = \frac{b^2}{bC_t + bC_\Delta/2} = \frac{b}{C_t + C\Delta/2} \]

\[ \tan \Delta_{tr} = \frac{C_\Delta}{b^2} = \frac{2C\Delta}{b} \quad C\Delta = b \tan \Delta/2 \]

\[ A.R. = \frac{b}{C_t + \frac{1}{2} \tan \Delta} \]

\[ \tan \lambda_{tr} = \frac{C_t/2 - C_t/2}{b/2} = \frac{C_t - C_t}{b} = \frac{C\Delta + C_t - C_t}{b} = \frac{C\Delta}{b} \]
$0 < M < 0.6$ Airfoil $\frac{c}{d} = 0.05$, $t_{max} = 0.6c$

$0^\circ$ L.E. Sweep
$M = 0.25$

THEORY
TAKE OFF & LANDING SPEEDS

\[ \frac{W}{ft} = \frac{104,928}{1275} = 82.3 \frac{\#}{50 \text{ ft}} \text{ wing loading} \]

\[ \frac{W}{ft} = \frac{L}{\frac{1}{2} C_l \rho} = C_l q = C_l \rho V^2 \]

\[ V^2 = \frac{2L}{\frac{1}{2} C_l \rho} = \frac{164,600}{C_l (0.002379)} = 69.25 \]

**TAKE OFF**

<table>
<thead>
<tr>
<th>( \alpha )</th>
<th>( C_l )</th>
<th>( V^2 )</th>
<th>( V_{ft/sec} )</th>
<th>( V_{mph} ) (Take off)</th>
</tr>
</thead>
<tbody>
<tr>
<td>13°</td>
<td>.4</td>
<td>173,000</td>
<td>414</td>
<td>283.8</td>
</tr>
<tr>
<td>15.5°</td>
<td>.5</td>
<td>138,400</td>
<td>371.5</td>
<td>253.3</td>
</tr>
<tr>
<td>18.3°</td>
<td>.4</td>
<td>115,300</td>
<td>339.8</td>
<td>231.8</td>
</tr>
</tbody>
</table>

**LANDING**

\[ V^2 = \frac{3725 \rho (\alpha)}{C_l (P)} = \frac{59.2}{C_l \rho} = 24.900 \]

<table>
<thead>
<tr>
<th>( \alpha )</th>
<th>( C_l )</th>
<th>( V^2 )</th>
<th>( V_{ft/sec} )</th>
<th>( V_{mph} ) (Landing)</th>
</tr>
</thead>
<tbody>
<tr>
<td>13</td>
<td>.4</td>
<td>62,250</td>
<td>249.4</td>
<td>170 mph</td>
</tr>
<tr>
<td>15.5</td>
<td>.5</td>
<td>49,800</td>
<td>223.2</td>
<td>152</td>
</tr>
<tr>
<td>18.3</td>
<td>.4</td>
<td>41,500</td>
<td>203.8</td>
<td>139</td>
</tr>
</tbody>
</table>

\( C_l \)'s from TN D-3765, Edith Polhamus
\[ R_N = 8800 \text{mph} \times \text{ft} = 8800 (250 \text{ft}) = 2,190,000 \text{ft}. \]

Stratified Turb. Theory - Vel distribution to re

Liebeck, R. H. Dr. Doug.
Smith, M.D.

FAA (Bob Allen) 893 26087 ~8382

Dr. (Ed Gibson)

Hammons, A.D. 3611
Pellamne, E.C. 3711
6% of mission fuel. (FAA 7%)  

240 Min.  

30 min hold

<table>
<thead>
<tr>
<th>%</th>
<th>Sp</th>
<th>b</th>
<th>A</th>
</tr>
</thead>
<tbody>
<tr>
<td>53%</td>
<td>110</td>
<td>15.3x10^5</td>
<td>1.492 in</td>
</tr>
<tr>
<td>65.8</td>
<td>80</td>
<td>13.15</td>
<td>1.358</td>
</tr>
<tr>
<td>85.2</td>
<td>60</td>
<td>10.5</td>
<td>1.164</td>
</tr>
</tbody>
</table>

Wk Landing

39.2  
47.3  
60.0
with Cal in Ar & Phi for free

80/62 II II II

75/62 D-A 1-5 Bock

Date on 62-67 New York

Michelle Strax

NASA CR-HU
\[ 62^\circ \Delta \quad b = 2.29.24 = 58.48 \]

Planeform Area: Wing Exposed \( 59.34 \times 24.24 = 1558.0 \)

\[ F_{oee} \quad 6 \times 69.36 = 416.0 \]

\[ \frac{1}{2} R^2 (x - \sin x) \quad N = \frac{1}{2} (1.4) \left( \frac{62.4^2}{60^2} \right) = 67.08 \]

\[ \frac{1}{2} R^2 \left( \frac{16^2}{60^2} \right) = \frac{16 \times 15 \times 4}{2 \times 15^2} = 49.16 \]

\[ S_p = \frac{2090.248}{5} = \frac{1}{5} \]

Wing Plan. Area = \( 65 \times 29.24 = 1901.0 \)

\[ A = \frac{1}{5} \]

\[ K (Wing) = \frac{1.80}{1.63} = 1.10 \]

\[ K (Planeform) = \frac{1.80}{1.63} = 1.10 \]

\[ C_L A = 0.0235 \times \frac{1901.0}{2090.248} = \frac{5.0}{5.0} \]

\[ C_L A = 0.40 = 0.382 = 0.542 \]

\[ C_L A = 0.53 = 0.506 = 0.862 \]
Body Area 2000.248 - 1558 = 532.248
Wing 62° only Exposed = 1558.0

75° ∆ only Exposed, 31.6 x 3.9

Phantom Area, Sp = 2213.448

Wing Area Total 62° only = 1901.0

75° ∆ only Total, 43.3 x 5.35 = 231.7

Wing Area, S = 2132.7

A Wing \( \frac{62.181}{2132.7} \) = 0.0291
A Phantom \( \frac{158.49}{2213.448} \) = 0.071

\( \frac{S}{A} \times 2132.7 \times 0.0291 = Sp \times Aw = 0.0362 \)

C_{Lx} = 0.44

C_{Lz} = 0.61
80°/62° D0/64 Δ NACA CP-714

Body Alone = 532.25

Wing 1'' Exp. = 1558.0

80° D only Exp. 2.6 x 22.8 = 59.28

Plenum Area S0 = 2149.53

Wing Area Total 67° only = 1901.0

80° D only total 42 x 46 = 193.2

Wing Area SA = 2094.2

\[ \frac{A \text{ wing}}{2094.2} = 1.632 \]

\[ \frac{A \text{ plenum}}{2149.53} = 1.592 \]

\[ C_d \times \frac{5\pi / A}{10235 \times 2094.2 \times 1.61} = 0.3758 \]

\[ C_{d,1} = 0.41 \]

\[ C_{l,20} = 0.57 \]

\[ C_{l,20} = 0.911 \]
<table>
<thead>
<tr>
<th>Model</th>
<th>( \frac{V_{\text{pa}}}{5p} )</th>
</tr>
</thead>
<tbody>
<tr>
<td>HT-1 (Cat)</td>
<td>0.156</td>
</tr>
<tr>
<td>BBW</td>
<td>0.176</td>
</tr>
<tr>
<td>DBW</td>
<td>0.224</td>
</tr>
<tr>
<td>HRV-5 (Body) (Above)</td>
<td>0.308</td>
</tr>
<tr>
<td>MAC-1 (Body Above)</td>
<td>( \sqrt[3]{\frac{4073}{613}} = 0.74 )</td>
</tr>
<tr>
<td>MAC (1st) (Body Above)</td>
<td>( 0.295 )</td>
</tr>
<tr>
<td>MAC (Total)</td>
<td>( \sqrt[5]{\frac{4181}{934730}} = 0.271 )</td>
</tr>
</tbody>
</table>
Ames All-Body Hypersonic Aircraft

\( \lambda = 75^\circ \)

Platform Area: total. \( 39.64 \times \frac{21.24}{2} = 421.28 \)

Att. body \( 7.07 \times 21.24 = 150.24 \)

Hn. Tail \( 6.03 \times 12.94 = 78.04 \)

Cwnds. \( 7.06 \times 3.02 = 21.31 \)

Fin \( 15.31 \times 0.04(1021) = 6.25 \)

\[ \frac{2}{b/2} \]

Ref Area \( 3/2 = 48.26 \times 12.93 = 624.0 \text{ cm}^2 \)

Span total \( = 32.31 \)

\( AR \) (Platform) \( = \frac{b}{d} = \frac{(32.31)}{680.69} = 0.47 \)
\[ M = 0.1 \rightarrow 0.36 \]

\[ \alpha = 15^\circ \text{ (Measurted from } \alpha_{40} \text{)} \]

**DBW**

\[ \cos 15^\circ = 0.765 \]

\[ \frac{21.26 \text{ Int}}{33.14} \]

\[ 0.4585 \]

**BBW**

\[ 0.74 \times \frac{22.875}{29.95} = 0.5643 \]

**HT 4**

\[ 0.764 \times \frac{20}{100.5} = 0.533 \]

**HRV**

\[ \frac{1}{917} = 0.5715 \]

**Amn Body M = 0.65**

\[ 0.63 \times \frac{621}{680.69} = 0.5722 \]

\[ \alpha = 20^\circ \text{ (Measurted from } \alpha_{40} \text{)} \]

**DBW**

\[ 0.93 = 0.5964 \]

**BBW**

\[ 1.06 = 0.809 \]

**HT 4**

\[ 1.08 = 0.752 \]

**HRV**

\[ = 0.585 \]

**Area**

\[ 0.86 = 0.7884 \]
DATA FROM MISSISSIPPI STATE UNIVERSITY

1. KARMAN (TURBULENT)
2. BLASISUS (LAMINAR)
3. HORTEN IV
4. REIHER
5. RJ-5
6. TNY MITE
7. SPIRIT OF ST. LOUIS - EXPOSED CYLINDERS (INCLUDING PROP LOSSES)
8. J-3 CUB - EXPOSED CYLINDERS (GLIDE)
9. PIPER PA 23 NIDORP (GLIDE)
10. CESSNA 120 - COWL OPEN (GLIDE)
11. CESSNA 170-B COWL OPEN (GLIDE)
12. BELLANCA CRUISAIR SR COWL OPEN (GLIDE)
13. ORIGINAL LODESTAR
14. PRODUCTION LEARSTAR
15. LODESTAR MODIFIED BY LEAR INC. COWL PARTIALLY SEALED (EXPERIMENTAL)
16. WITTMAN'S TAILWIND COWL SEALED (POWER OFF GLIDE)
17. AG-14
18. CHAPPY
19. L-23 A
20. B-57 A
21. B-52 A
22. BLACK BUZZARD (CORAGYPS ATRAMUS)

\[
\frac{C_r - (0.005 - 0.05)}{C_r} = f(\text{THE METIER})
\]

\[
\overline{RN} = \left( \frac{2\overline{C_s} \text{ WING}}{(2\overline{C_s} \text{ TAIL}) + (13 \overline{C_s} \text{ FEAT})} \right) \times \frac{V}{\text{TOTAL METED}} \times 10^{-6}
\]

\[
\overline{RN} = \frac{(2\overline{C_s} \text{ WING}) + (2\overline{C_s} \text{ TAIL}) + (13 \overline{C_s} \text{ FEAT})}{(2\overline{C_s} \text{ TAIL}) + (13 \overline{C_s} \text{ FEAT})} \times \frac{V}{\text{TOTAL METED}} \times 10^{-6}
\]
The skilled model designer, whose aim it is to build models of high aerodynamic quality, will always find that even the most accurately constructed models do not attain the performance of the corresponding full-scale aircraft.

We know that this "scale effect" is due to the fact that the friction drag coefficient (C_C) is decidedly higher at the low Reynolds' numbers of the flying models, and that in addition the low RN range shows an earlier separation of the boundary layer. This phenomenon, which has been proved by wind tunnel measurements, exercises its influence mainly on the characteristics of the wing and the wing sections.

You may ask what the Reynolds' number means. The law of similarity for fluid motion, discovered by Osborne Reynolds, states that two flow conditions (for instance the flow around wing sections) are similar if the Reynolds' numbers of the two tests are the same. The RN is calculated by forming the product of a characteristic length—say the chord length—and the velocity of the flow and dividing by the kinematic viscosity of the fluid:

\[ RN = \frac{V \times L}{\nu} \]  

That means that two tests carried out in two different fluids—for instance, air and water—can be compared if we consider the different values of the kinematic viscosity.

The Reynolds' law of similarity is essentially important if we apply test results from wind tunnels to
flight conditions, and we can only expect to obtain the same performance if the RN of the wind tunnel test and the RN of the free flight condition are the same. Since this condition cannot always be realized completely, the influence of the RN—let’s call it the aerodynamic scale—was carefully investigated and it was found that the similarity is not seriously affected if the RN of the flight condition is somewhat larger than the test RN.

As we know that the friction coefficient usually decreases with increasing RN, we may get something better than indicated by the wind tunnel test. But if we apply a test result to the flight conditions of a flying model where the RN is considerably smaller than at the test, we will get something completely different and unfavorably lower in performance. At low RN flow conditions, the flow character can change so much that two different sections will give reversed performance, profile A being much better than profile B at an RN of $3 \times 10^4$ and profile B being much better than profile A at an RN of $1 \times 10^5$.

To calculate the RN for air at standard (sea) level, multiply the chord length in inches by flight speed in miles per hour and take then 800 times this product. For instance, take this as an example:

- chord length = 6 inches
- flight speed = 25 mph
- Reynolds number = $(6 \times 25) \times 800 = 120,000 = 1.2 \times 10^5$

The wing sections which have been successfully developed for modern high-speed aircraft—as for instance the laminar flow sections—are not favorable if you use these special high RN sections for your models. But even the wing sections of the full-scale gliders, the RN of which is about $1,000,000$, are not applicable for a glider model with an average RN lower than 100,000. (Cont’d on page 71)
Therefore the question arises again and again, which kind of wing sections are especially suitable for the design of flying models.

To find out how closely these early measurements would correspond to some later tests in larger wind tunnels, I compared the results given with a recent test of the 7 h ft wind tunnel of the Goettingen Institute. Fig. 2 illustrates this comparison, and we can see see a kind of ideal kind of wing section under consideration—characteristic closely corresponds with the earlier test. The difference of the \( C_{\text{D}} \) is due to the difference in \( C_{\text{CL}} \) in the test having been made at twice the RN of the test in the smaller tunnel.

The section we used was \#289 and Fig. 2 shows the measurements of two thicker sections, \#289 and \#100 of this usual type of profiles.

First of all you see in both cases the abrupt breakdown of lift at about \( C_{\text{L}} = 1.0 \) and 6° angle of attack. I show you only these two examples but there are many of the same kind indicating that the usual flat cambered thick sections have a low \( C_{\text{D max}} \) at the smaller RN.

This unfavorable behavior is caused by the fact that the flow on the nose part of the upper surface is laminar and that, since the laminar flow is very unstable, it is very prone to the point where the decrease of the local velocity begins. The separation from the upper surface is the characteristic that no recovery can be established. At higher RNs such sections have usually pretty good characteristics as shown by the measurements of \#289 at RN = 1.4 x 10^5. (Fig. 4.) The two different measurements made with a smaller wing and higher speeds, made with a larger wing with smaller speed are in agreement and prove the fact that we cannot select a good model wing section from these tests.

These stalling characteristics of the usual-shaped sections cause most of the troubles you may have with your model. This becomes evident if you have a glider model with such a section and the model enters a strong thermal. The sudden increase of angle of attack reaches the critical point and the sudden breakdown of the lift causes a dive. The smooth increase and angle of attack decreases again for a recovery of the lift. The model begins to climb again and the same oscillating movement is repeated over and over again and may in the end bring the model into such a stalling position that a spin or even a loop is produced by this instability.

We must realize that when going down with the angle of attack from a stalling position, as for instance the 8' point of the \#289 section, we have to go back along the recovery polar which is illustrated by Fig. 5, taken from another test showing similar behavior. The curve shows the results of a test where the angle of attack of the wing was increased and then decreased while still stalling. Several tests show that two branches of the polar curve can be realized—the usual one from smooth conditions up to the stalling point and further, and the recovery branch from complete stalling down to a new again. The upper branch may show a more continuous change in lift and drag, if you have a good recovery branch has usually a sudden change from stall to perfect flow at nearly the same angle of attack. So now, when your plane has reached stalling, a sudden change in lift will occur and the abrupt increase of the lift forces will bring a considerable upward acceleration. The unbalanced lift increase

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**Guides section on curves**

You may think that there is a hysteresis effect at the stalling point and you can measure two or even three different angles of attack at the same lift coefficient but at different flow conditions. If this hysteresis effect happens at lower angles of attack you will have great difficulty in bringing your plane to a steady flight condition. To avoid all these troubles we have to change the type of wing section.

Two sections of considerable high performance in the low RN range are section \#238 and \#289. The section \#238 presents the original measurements. These sections are especially suitable for glider models or larger powered models carrying high loads. At nearly 1.80 makes the models almost stall-proof, and the performance at aspect ratios above 1.3 is remarkable. It would not be of very much use to my friends if I would only show the original test results. What we need for a real evaluation of a certain section are the absolute values of the coefficients and the ordinates of the section to it.

By absolute characteristics I mean the coefficients converted to infinite aspect ratio. From these values you can calculate proper induced drag and add the individual induced drag for the aspect ratio of the wing you will design.

The slope of the Zener curve shows much more clearly what really happens at the different lift conditions. Most of the modern wind tunnel results are plotted in this way.

The angle of attack is also converted into the absolute value and indicates the direction of the neighborhood of the wing and not at far-off conditions. To obtain the angle of attack for any given AR we have to add the induced angle of attack to the value shown on the absolute polar. (We will calculate an example later, that you may see how it works)

The absolute values for section \#227 are shown in Fig. 7. On the right hand you see the polar curve, \( C_{\text{D}} \) as a function of \( C_{\text{L}} \). On the left you have the moment coefficient related here to the \( \frac{1}{4} \text{ chord point} \).

If you take now \( C_{\text{D}} / C_{\text{L}} \) the value indicates the distance in fractions of the chord length to point backward (if negative). I chose the \( \frac{1}{4} \) chord point because it is near to the theoretical focus of the section and the \( C_{\text{L}} \). Why choose this fractional constant? Below is a larger drawing of the section and on the right hand you will find the ordinates given for a chord length of 100 parts. The ordinates of most of the \#227 sections I evaluated were not counted from the chord which terminated the section at the section plan. The reason was a practical one because there are no ordinates given in the other parts of the section. We must get as exact as possible from the enlarged pictures of the sections. I measured from a line farther below, which could be drawn with better accuracy.

You can see that this evaluation and research were quite a job and in order to select the best sections from some tabled measurements, many have to be converted, plotted and compared until I could see which were the best. What we see is of even greater interest than I had expected. We can go along the polar of the section \#227.

At the larger negative angles of attack or at low lift coefficients, we have a smaller moment on the lower surface due to its camber. At about 0° or \( C_{\text{L}} = 0.2 \) the
same flow condition begins to disturb and the drag coefficient $C_{D_m}$ decreases slowly. At $C_r>0.6$ the maximum drag is attained and the flat peak in this region indicates that also at this low $RN$ we have a small laminar flow effect. You can see that faint swelling in the $C_r$ vs. $RN$ region. This is characteristic of the selected family of airfoils and sometimes the laminar bump is really pronounced.

You may say that the $C_{D_m}$ is much too large for being laminar flow, but remember first that we are working with a $RN=1 \times 10^5$ and that there can only be a fraction of the whole surface at laminar flow conditions. When we proceed now to higher lift conditions something much more surprising happens. At zero angle of attack or $C_r=0$ just in the region where the #289 and #200 section types had the complete breakdown, we can clearly observe a sudden change in the drag to some smaller value. The polar curve is shifted to the left by about $1/4$ of the $C_r$ value. When I first drew the polar curve along the points plotted I could not get a smooth curve through the points and I feared an error in the calculations. But I had been right. I checked some other measurements of similar sections and they turned out to have the same discontinuity as you can see on Fig. 6 which illustrates the results of three of the cambered sections. Sometimes later I was looking at some recent wind tunnel tests and I found that this discontinuity of the section drag coefficient appeared in many different tests even in the higher $RN$ range.

The question arises, what flow effect causes this favorable decrease in drag? I think there is only one answer, and that is a considerable increase of the laminar flow range on the lower surface of the wing. At the larger angles of attack we have on the lower surface a continuous decrease of pressure or increase of velocity. This flow condition has a stabilizing effect on the boundary layer, so that a more extended laminar flow along this surface is produced. By these means the drag decreases to a lower value and the point where this happens is usually also the point of the absolute maximum of $L/D$, in respect to $C_r/C_{D_m}$. In the low $RN$ range the discontinuity is more pronounced and we therefore get pretty high values of the absolute $L/D$ being sometimes even better than in the higher $RN$ region.

I will not go too far into the physical background of this effect, but it is most important for us to realize that the lower surface is the "sacred surface" on the wing of a model (or even a glider). To get this high performance point of $L/D$ and sinking speed, you have to keep the lower surface as clean as possible and so smooth that "even a fly would break a leg on this slick polished slide."

It is interesting to mention that the profiles showing this effect have almost an "undercamber," as they call it. The sections with a straight lower surface as the Clark Y types do not have such a pronounced discontinuity effect. That means that the velocity distribution should have a certain shape. This could be investigated more theoretically. For the model designer it is sufficient to know what he can expect and how to get it.
Wing Sections for Model Planes  
Part 2

Well-known designer, famed for his flying wing research work concludes a "lecture" on airfoils

By DR. ALEXANDER M. LIPPISCH

Editor's Note: In the April issue Dr. Lippisch introduced the subject of scale effect on models and its importance in the selection of airfoil aspect ratio and wing efficiency. He confirmed many suspicions that the so-called laminar flow airfoils do not act as laminar flow sections when reduced in scale; now he goes on to show that some full-scale non-laminar flow airfoils may assume laminar flow characteristics when scaled down.

Let us now have a look on the next section, 242, which is somewhat thicker (Fig. 9). It is interesting to see that the sharp pointed nose does not have any bad effect on the maximum lift. On the contrary, this section has the highest $c_{\text{L, max}}$. It is remarkable that the larger birds have such pointed nose sections on the root part of the wing. The tendon connecting the shoulder with the wing forms this pointed nose part of the inner wing. The drag shift is not so pronounced with this section because the "undeream" is not extended enough toward the leading edge. The drag minimum is somewhat larger than for 227 (see Part 1), but the absolute $L/D$ is about the same.

A real thick section of the bird type is section 243, which I will show you next.

Fig. 10 illustrates the original measurement. You see that again we have a sudden stalling at the 9° point but at least at a considerably higher $c_{\text{min}}$ of 1.50. Therefore, this is not very serious and could be smoothed out if we made the nose a bit sharper. The thickness ratio of this section is close to 20 percent, which means that you really can design some aspect ratio with it.

The absolute characteristics of section 243 are represented by Fig. 11. Here you see again the distinct branches of the absolute polar. The inner laminar
bump as well pronounced and shows a c p not far from section 227. The absolute L/D is higher than that of the two other sections because we have here a pronounced decrease of drag on the lower surface.

How the different branches of the polar can be distinguished is more clearly shown on Fig. 12. We see the inner laminar part and then the transition to the outer branches which could be connected by an ideal dotted line. You might say that the points measured are not sufficient to warrant this suggestion, but more detailed recent tests on other sections show the characteristic shape of the different curves. While the flow conditions on the boundary layer are very sensitive against little changes, we can expect to get the outer branches more extended if we produce a very clean design. The maximum L/D which is here about 50 proves that with some higher aspect ratio we can get a gliding angle above 1:20 with some larger model.

I thought it would be of great interest to compare this thick-cambered section with a section of about the same thickness though without the undercamber.

Fig. 13 illustrates the two sections 243 and 380 together with the polars for AR = 5 (AR, of course, meaning aspect ratio). The values of section 380 were taken from a later test in the larger wind tunnel and at somewhat higher RN (Reynolds Number). The effect of the undercamber can be seen very clearly. The break away of the flow happens at about the same angles of attack but the cambered section has at that incidence a considerably higher lift coefficient. The camber at the lower surface shifts the curve up to a higher c l region, which is most favorable for gliding and sinking speed at high AR. I think that this figure just speaks for itself.

Now you may think that I (Continued on page 85)
Wing Sections
(Continued from page 45)

am only interested in the thick-cambered sections for gliders or high-speed aircraft. Therefore, let us turn now to some thinner sections of high performance. There is for instance the section 301, of which we made measurements. This section is fairly blunt and is suitable for large angle sections. We must realize that the one and the following measurements were done with smaller chord length, the RN being smaller than with the first three sections. The RN range of 70,000 is not far from the range of medium models.

Wing sections with several models and obtained excellent results. It is important to note that the camber of the lower surface starts near the minimum. The thickness ratio of section 301 is about 1:10.

And now let us have a real fast job section. From all the tests I checked this is the section with the smallest drag coefficient measured in the test room. Fig. 15 presents very interesting absolute characteristics. The polar shows clearly the different branches according to the different boundary layer flow conditions. To demonstrate the high efficiency of this section as clearly as possible we have plotted the c⊥ values at twice the scales of the other profiles. The c⊥ mínimum is 0.012, and if we plot this value into the well-known friction drag graph we will find that this value at RN = 74,000 indicates that this section is a pure laminar section for the low RN!

We could hardly get a smaller drag in this low range. Since the camber is small and the mean line of the section has some S shape, the c⊥ is very small, indicating that this section is near to the constant C⊥ position. It could therefore be applied to tailless or flying wing type models.

The shape of the section is a really interesting one. First of all the nose is rather blunt. Then comes the main part with nearly constant thickness and small camber. The "tail" of the section is then turned upwards and has a pretty large angle at the trailing edge. This shape of section had its origin with the English fighter planes, like the Spitfire. Section 344 dates from the French.Torpedo and is listed in the report as "Pfalz 71". It is amazing to learn how this funny shape was designed. From the strength calculation somebody figured out that the two spars of the wing have a certain shape and that the rear spar should be extra strong (the rear spar getting the highest load on the biplane). The chord length was also fixed, and so they compromised as much as their French curves would allow and hence they arrived at this kind of section which is illustrated in Fig. 16. Don’t think it is a yoga because these days and know how aerodynamics were handled.

If you like to get some high performance with very thin-cambered sections, take either the 122 or the 342 (Fig. 17). They are almost the same. But it is starting how near the two different measurements fit together, even with the interesting saddle at c⊥ = 1.0. Section 123 is the one with the best L/D, absolute I could find in the report with about 54:1!

The minimum drag coefficient is nearly as low as that of section 344, and is also quite laminar, which can be seen from the course of the absolute polar. The section can in addition be used for any kind of full haging when cemented several layers on a cambered jig and sand the profile shape out. But don’t forget that the correct film is very important even with these thin sections. Make an exact pattern out of metal to work the wing out. The best method to obtain exact patterns is to get a photographic engraving as is done for printing jobs. I promised you to give an example how to do a calculation on the performance of a new design. Well, let us take wing section 301 and design a larger glider model with 500 sq. in. wing area and an aspect ratio of 14. The span is then 84” or 7 feet, and the chord length 6”. To get the RN around 70,000, the ship should have a speed of at least 15 mph or 22 ft per sec.

The following table shows the performance calculation:

<table>
<thead>
<tr>
<th>Glider Model</th>
<th>Performance</th>
<th>Section 301 G.</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aspect Ratio</td>
<td>14.</td>
<td></td>
</tr>
<tr>
<td>Parasite Drag Coefficient c⊥ = 0.010</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

(Fuselage + Tail Unit) *Induced Angle of Attack 1.822

Induced Drag Coefficient c⊥ AR = 0.0218

Target Drag Coefficient c⊥ AR = 0.0227 c⊥

Section Characteristics for Induced Drag and Angle of Attack |

<table>
<thead>
<tr>
<th>α</th>
<th>CD1</th>
<th>CD2</th>
<th>CD3</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.5</td>
<td>0.072</td>
<td>0.060</td>
<td>0.037</td>
</tr>
<tr>
<td>1.5</td>
<td>0.235</td>
<td>0.235</td>
<td>0.135</td>
</tr>
<tr>
<td>2.5</td>
<td>0.390</td>
<td>0.390</td>
<td>0.180</td>
</tr>
<tr>
<td>3.5</td>
<td>0.545</td>
<td>0.545</td>
<td>0.200</td>
</tr>
<tr>
<td>4.5</td>
<td>0.700</td>
<td>0.700</td>
<td>0.220</td>
</tr>
<tr>
<td>5.5</td>
<td>0.855</td>
<td>0.855</td>
<td>0.240</td>
</tr>
<tr>
<td>6.5</td>
<td>1.010</td>
<td>1.010</td>
<td>0.260</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Wing Characteristic</th>
<th>AR = 14</th>
<th>Total</th>
<th>Wing Coefficient</th>
</tr>
</thead>
<tbody>
<tr>
<td>c⊥</td>
<td>CD1</td>
<td>CD2</td>
<td>CD3 Total</td>
</tr>
<tr>
<td>0.5</td>
<td>0.072</td>
<td>0.060</td>
<td>0.037</td>
</tr>
<tr>
<td>1.5</td>
<td>0.235</td>
<td>0.235</td>
<td>0.135</td>
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<tr>
<td>2.5</td>
<td>0.390</td>
<td>0.390</td>
<td>0.180</td>
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<tr>
<td>3.5</td>
<td>0.545</td>
<td>0.545</td>
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<td>0.240</td>
</tr>
<tr>
<td>6.5</td>
<td>1.010</td>
<td>1.010</td>
<td>0.260</td>
</tr>
</tbody>
</table>

We take the absolute values from the graph and convert the drag coefficient and the angle of attack into the values at AR = 14 by adding the induced drag coefficient c⊥ and the induced angle of attack α⊥ to the corresponding absolute values. To many of the model designers this may look like too much theory and math. But there is only one way to get best performance and that is by the union of experience and knowledge. You cannot build as many different models as is necessary to find out how to get best performance. But if you have a method to predict the performance you are immediately at the point where you want to be.

The graph of Fig. 18 illustrates the performance of such a chip. The wing loading is according to the RN limit. You see that we would get a gliding of Y: 19 and a sinking speed of a little bit more than 1 ft per second. The angle of attack in the best performance range is between 4° (best gliding) to 7° (lowest sinking speed). To get the smallest parasite drag of the fuselage, the centerline should have about 6° to the chord of the section.

But it would be too long a story to tell everything about a special design. One word only on behalf of the CG position. The best flight is around c⊥ = 1.0 where c⊥ = 0.1 and also equals -0.1. That means the CG for unloaded elevator (low drag condition) is 10 percent behind the ¾ point of the chord, or at 35 percent chord length behind leading edge. Having a chord of 6 inches, the CG therefore lies 2.1 inches behind the leading edge.

I hope that this little lecture about model aerodynamics may give some ideas to the model designer. To fly models is a lot of fun but be always aware that many achievements in large-scale aviation made their way through aerodynamic childhoods as models. This holds true even for the fast sweeps of today.

*Values taken from test results of some wind tunnel.
THE KOCH CHART FOR ALTITUDE AND TEMPERATURE EFFECTS

TO FIND THE EFFECT OF ALTITUDE AND TEMPERATURE
CONNECT THE TEMPERATURE AND AIRPORT ALTITUDE
BY A STRAIGHT LINE.

READ THE INCREASE IN TAKE-OFF DISTANCE AND THE
DECREASE IN RATE OF CLIMB FROM STANDARD SEA
LEVEL VALUES HERE

ADD THIS PERCENTAGE TO
YOUR NORMAL TAKE-OFF DISTANCE

This chart indicates typical representative values for "personal" airplanes.
For exact values consult your airplane flight manual.
The chart may be conservative for airplanes with supercharged engines.
Also remember that long grass, sand, mud, or deep snow can easily double your
take-off distance.

EXAMPLE: The diagonal line shows that 23% must be added for a temperature of
100° and a pressure altitude of 5,000 feet. Therefore, if your standard temperature
sea level take-off distance, in order to climb to 50 feet, normally requires 1,000 feet
of runway, it would become 3,300 feet under the conditions shown. In addition, the
rate of climb would be decreased 76%. Also, if your normal sea level rate of climb
is 500 feet per minute, it would become 178 feet per minute.
RESEARCH BRIEF: VORTEX-INDUCED BOUNDARY-LAYER CONTROL

Research on methods of controlling the flow over a wing surface has been underway at Langley for many years. Past research has been oriented primarily toward maintaining attached flow over a wing by using leading-edge flaps or slats. These type of leading-edge devices, while resulting in increased maneuverability, introduce added weight and complexity to the configuration. Another approach, which has the potential for increasing maneuvering capability at a low weight penalty, is the use of a wing fuselage strake. This strake creates a vortex system which gives a direct vortex-lift increment as well as interacting with the wing flow field to maintain attached flow over the wing up to fairly high angles of attack. Research on vortex-induced boundary-layer control is continuing in the Langley high-speed 7- by 10-foot tunnel.

Figure 1 illustrates the potential gains that can be realized with the use of vortex-lift strakes. The circular symbols represent lift data for the configuration without the strake. These data show that the configuration experiences considerable wing flow separation at the higher angles of attack, as evidenced by the experimental data falling below the attached flow estimated lift curve and further substantiated by flow visualization studies. (See sketch in lower right corner of figure 1.) Adding the strake to the configuration (square symbols) results in a significant increase in lift at the higher angles of attack. This increase in lift is the combined result of the vortex lift on the strake and a favorable interference effect of the vortex on the wing flow field. This result was substantiated in several ways. First, the experimental "strake on" data agrees reasonably well with the estimated data. In this case, the estimate was made up of the attached flow wing fuselage lift and the vortex lift on the strake. Secondly, the flow visualization studies indicated that flow separation was eliminated over nearly all of the wing surface (note sketch at top right of figure 1). Thirdly, pressure measurements made on the wing surface (see figure 2) show that significant increases in the negative pressure coefficients are evident on the wing, particularly in the mid-semispan region.

The research on vortex-lift and vortex-induced boundary-layer control will continue in the Stability and Dynamics Branch, and will center around using a jet blown spanwise along the wing to create a vortex to increase the lifting capability of a wing. Accompanying the experimental effort will be an extensive analytical effort aimed at developing procedures for predicting the vortex effects.

TECHNICAL CONTACT: William P. Henderson
Stability and Dynamics Branch
High-Speed Aircraft Division

June 29, 1973
Figure 1.- Effect of vortex-lift strakes on lifting characteristics. M = 0.40
Figure 2. - Effect of vortex-lift strakes on wing pressure distribution at $\alpha = 21.5^\circ$, $M = 0.40$
RESEARCH BRIEF: AIRFOIL SLOPES OBTAINED FROM INVERSE TRANSONIC CALCULATIONS

A finite-difference relaxation method computer program has been developed for solving the nonlinear small perturbation potential equation for transonic flow about airfoils. This program can be used for either direct calculations (airfoil shape prescribed, surface pressure unknown) or for inverse studies (surface pressure given, airfoil shape to be determined). Similar inverse methods in the past have yielded unexplained discontinuities in the airfoil slope at the location of the upper surface shock wave. The figure shows results obtained from the present program with the airfoil shape given from the leading edge to the midchord and the surface pressure prescribed after the midchord. Case 1 uses the same relationship between pressure and the velocity potential in both the direct and inverse regions, and the computed shape agrees with the actual shape. Case 2, however, uses in the inverse region a different relation between the pressure and the potential function that has theoretically the same order of accuracy as Case 1, but it yields a significant discontinuity in the airfoil slope at the shock wave. It is concluded that for the transonic inverse problem that the resultant airfoil slope and shape is sensitive to the method of relating the velocity potential and the prescribed pressure distribution. Further studies of this subject are continuing.

TECHNICAL CONTACT - Leland A. Carlson, NASA-ASEE Summer Faculty Fellow
Theoretical Aerodynamics Branch
High-Speed Aircraft Division

June 28, 1973
AIRFOIL SLOPES CALCULATED WITH INVERSE PROGRAM

65 Di-convex Section, M_a=0.9

- Case 1
- Case 2
- Actual

Shock Location

Direct Solution ← Inverse Solution
RESEARCH BRIEF: A New Thick-Airfoil Theory for Designing Airfoil Families

Technical Contact
Raymond L. Burger
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3636

A method has been developed for designing families of airfoils with different thickness ratios but with the same lift and with similar type pressure distributions.

This method uses non-linear thick airfoil theory in comparison with the previous thin airfoil method, in which various thickness distributions were superimposed on the same camber line. The new theory may also be compared with an existing thick airfoil procedure which varies the thickness but inherently imposes a proportional change in lift. Often this corresponding change in lift is desirable, but in other situations a change in thickness is desired while maintaining the same lift. The new theory is applicable to such problems. Furthermore, it can be combined with the existing method to generate families of airfoils in which the thickness and lift are varied in different proportions, while a basic similarity is maintained in the shape of the pressure distribution.

Attempts to apply the theory outside of reasonable limits result in impractical or impossible airfoil shapes.
PRESSURE DISTRIBUTION FOR FAMILY OF THICK AIRFOILS

Thickness ratio = .15

T.R. = .17
(Basic shape)

T.R. = .19
A. Lift Estimation:

(1) Regular planform wings \( R \geq 2.0 \)

Determine \( R \) and \( \cos \alpha/2 \) and pick \( C_{LQ} \) from figure 1 (ref. TN D-3911).

(2) Irregular planform wings: \( R \geq 2.0 \)

Determine true aspect ratio based on total span and total area and effective \( \cos \alpha/2 \) as per equations below.

The effective value of \( \cos \alpha/2 \) may be determined by dividing the wing into \( N \) sections, each section being assumed to have constant-sweep angles within its boundary. (Ref. TM X-525.)

The span of section \( i \) is denoted by \( y_i \), the average chord of section \( i \), by \( c_{av,i} \), and the cosine of the sweep angle of the half-chord line of section \( i \), by \( (\cos \alpha/2)_i \). The weighted average or effective value of \( \cos \alpha/2 \) is therefore defined as

\[
(\cos \alpha/2)_{\text{eff}} = \frac{\sum_{i=1}^{N} (\cos \alpha/2)_i c_{av,i} \Delta y_i}{\sum_{i=1}^{N} c_{av,i} \Delta y_i}
\]

The denominator of equation (2) is the total area of one wing panel, \( S/2 \). Therefore, equation (2) may be written as follows:

\[
(\cos \alpha/2)_{\text{eff}} = \frac{2}{S} \sum_{i=1}^{N} (\cos \alpha/2)_i c_{av,i} \Delta y_i
\]

Pick \( C_{LQ} \) from figure 1. Multiply \( C_{LQ} \) by \( (S_{\text{TOT}}/S_{\text{REF}}) \) if \( S_{\text{REF}} \) is different.
(3) Regular or irregular planform wings: $\Delta R \leq 2.0$. Also lifting bodies.

Determine true $\Delta R$ and effective $\cos \alpha/2$ for potential term and use 1.2 value for $C_{d_0}$ for second order term in equation (3) given below.

$$
C_L = \frac{2\pi \alpha}{1 + \sqrt{\frac{\frac{A}{(\cos \alpha/2)_{eff}}}{2} + \frac{1}{57.3} + \cdots}}
$$

where $\alpha$ is in degrees.

This determines $C_L$ as a function of $\alpha$. (Ref. TN D-1374.)

B. Lift required and avoidance of pitch-up

(1) Select orbiter wing planform and assume maximum allowable weight.

(2) Determine $C_{L_0}$ of selected planform: assume linearity.

(3) Using $V_{MIN}$ criteria solve for $S$ planform required = $(V_{MIN}$ in ft/sec)

Since $(V_{MIN}) = \frac{\sqrt{2W}}{\rho C_{L_0} S}$

then $S = \frac{2W}{\rho C_{L_0} (V_{MIN})^2}$

(4) Calculate effective $c/4$ sweep and use figure 2 (ref. TM X-26, pg 19, fig. 3a) to insure no loss in stability with increasing $C_L$.

(5) If $S$ is too large or pitch-up occurs, alter wing geometry and iterate again.

C. Stability

- Longitudinal

(1) To determine $X_{cp}/l$ experimental for bodies see figure 3 (ref. TR-R250) and if body closely resembles one of these shapes, use the appropriate value of $X_{cp}/l$. The projected planform shape is the influencing factor. Cross-section effect on $X_{cp}/l$ is not significant.
Moment equations to calculate vehicle stability. (Ref. TM X-753.)

- Wing-Body-Tail Combinations -

No \( \epsilon \), \( C_m = C_{i_w} \cdot \frac{X_{cp_w}}{S_{ref}} \cdot S_w + C_{i_p} \cdot \frac{X_{cp_b}}{S_{ref}} \cdot S_{ref} \cdot S_{ref} + A_{b_{ref}} + C_{l_t} \cdot \frac{X_{cp_t}}{S_{ref}} \cdot S_{ref} \)

and \( C_{m_{\alpha}} = C_{i_w} \cdot \frac{X_{cp_w}}{S_{ref}} \cdot S_w + C_{i_p} \cdot \frac{X_{cp_b}}{S_{ref}} \cdot S_{ref} \cdot S_{ref} + A_{b_{ref}} + C_{l_t} \cdot \frac{X_{cp_t}}{S_{ref}} \cdot S_{ref} \)

with \( \epsilon \), \( C_m = C_{i_w} \cdot \frac{X_{cp_w}}{S_{ref}} \cdot S_w + C_{i_p} \cdot \frac{X_{cp_b}}{S_{ref}} \cdot S_{ref} \cdot S_{ref} + A_{b_{ref}} + C_{l_t} \cdot (1-\frac{\epsilon}{\alpha}) \cdot \frac{X_{cp_t}}{S_{ref}} \cdot S_{ref} \)

and therefore,

\[
\frac{3C_m}{3C_L} = \frac{C_{i_w} \cdot X_{cp_w} \cdot \frac{S_w}{S_{ref}} + C_{i_p} \cdot X_{cp_b} \cdot \frac{S_{ref}}{S_{ref}} + A_{b_{ref}} + C_{l_t} \cdot (1-\frac{\epsilon}{\alpha}) \cdot X_{cp_t} \cdot \frac{S_{ref}}{S_{ref}}}{C_{i_w} \cdot \frac{S_w}{S_{ref}} + C_{i_p} \cdot \frac{S_{ref}}{S_{ref}} + A_{b_{ref}} + C_{l_t} \cdot (1-\frac{\epsilon}{\alpha}) \cdot \frac{S_{ref}}{S_{ref}}}
\]

To determine \( X_{cp} \) of wing, tail or canard use charts on figure 4 (Ref. TN 1468) and apply in moment equations. This step is employed to locate wing relative to selected or required c.g.

- Lateral-directional

To check influence of selected planform on lateral-directional characteristics of the configuration use charts of Ref. TN-1468) figures 5 and 6.

* For canard-wing-body combinations apply influence of downwash, \((1-\frac{\epsilon}{\alpha})\) to wing, i.e., \( C_{i_w} \cdot (1-\frac{\epsilon}{\alpha}) \cdot \frac{S_w}{S_{ref}} \cdot X_{cp_w} \) for moment and \( C_{i_w} \cdot (1-\frac{\epsilon}{\alpha}) \cdot \frac{S_w}{S_{ref}} \) for lift and delete \( \epsilon \) from \( C_{l_t} \).
Figure 2. Pitch-up boundary for swept wings with variable aspect ratio.
Ref. TM X-26.
Figure 3.- Variation of center of pressure location for circular and elliptic bodies of various planforms where $n = \) body shape factor. $y = \frac{y_{max}}{2}$. 

Ref. TR R-250.
Figure 4. - Variation of aerodynamic-center position with sweepback.
Figure 5.- Variation of effective dihedral parameter with aspect ratio at $C_L = 0.4$. 
Figure 6. - Variation of directional stability parameter with aspect ratio at $C_L = 0.4$. 
CHARTS FOR PREDICTING THE SUBSONIC VORTEX-LIFT CHARACTERISTICS OF ARROW, DELTA, AND DIAMOND WINGS

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CHARTS FOR PREDICTING THE SUBSONIC VORTEX-LIFT CHARACTERISTICS OF ARROW, DELTA, AND DIAMOND WINGS

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SUMMARY

The leading-edge-suction analogy method of predicting the aerodynamic characteristics of slender delta wings has been extended to cover arrow- and diamond-wing planforms. Charts for use in calculating the potential- and vortex-flow terms for the lift and drag are presented, and a subsonic compressibility correction procedure based on the Prandtl-Glauert transformation is outlined.

INTRODUCTION

The leading-edge vortex lift associated with the leading-edge-separation vortex which occurs on slender sharp-edge wings has, during the past decade, become more than an aerodynamic curiosity with airplanes such as the Concorde supersonic transport and the Viggen fighter utilizing this flow phenomenon as a means of eliminating the need for flow control devices and high-lift flaps. (See refs. 1 to 3.) Although many analytical methods of predicting the aerodynamic characteristics associated with leading-edge vortex flow have been developed (some of which are reported in refs. 4 to 8), they have been limited primarily to delta planform wings or wings with unswept trailing edges. Because of the increased use of slender wings exhibiting leading-edge vortex flow, at least in the many off-design conditions if not at the design condition, analytical methods applicable to arbitrary planforms are needed. The leading-edge-suction analogy, described in references 8 and 9 appears to provide an accurate method of predicting the vortex-lift characteristics which, at least in concept, is not limited to delta planforms and has been shown in reference 10 to provide accurate estimates for a fairly wide range of fully tapered wings. Although the subsonic analysis was limited to incompressible flow, an appropriate application of the Prandtl-Glauert transformation should provide a subsonic compressibility correction. The purpose of this paper is to present, in chart form, the potential-flow and vortex-flow constants, including subsonic compressibility effects, for a wide series of arrow-, delta-, and diamond-wing planforms.
SYMBOLS

A \quad \text{wing aspect ratio, } b^2/s

a \quad \text{longitudinal distance from root trailing edge to wing tip station, positive rearward (see fig. 1)}

a/l \quad \text{wing notch ratio, positive for arrow wings and negative for diamond wings}

b \quad \text{wing span}

C_D \quad \text{drag coefficient}

C_{D,o} \quad \text{drag coefficient at zero lift}

\Delta C_D \quad \text{drag-due-to-lift coefficient, } C_D - C_{D,o}

C_L \quad \text{lift coefficient}

C_p \quad \text{pressure coefficient}

e \quad \text{leading-edge length of wing (see fig. 1)}

e' \quad \text{leading-edge length of transformed wing}

f_M \quad \text{compressibility factor (see eq. (5))}

K_p \quad \text{constant in potential-flow-lift term}

K_v \quad \text{constant in vortex-lift term}

l \quad \text{longitudinal distance from apex to wing tip station (see fig. 1)}

M \quad \text{Mach number}

S \quad \text{wing area}

\alpha \quad \text{angle of attack}
\[ \beta = \sqrt{1 - M^2} \]

\[ \Lambda_{le} \quad \text{leading-edge sweep of actual wing (see fig. 1)} \]

\[ \Lambda_{le}' \quad \text{leading-edge sweep of transformed wing, \( \tan \Lambda_{le}' = \frac{\tan \Lambda_{le}}{\beta} \)} \]

All primes refer to the transformed wing.

**ANALYTICAL METHODS**

In references 8 and 9 it has been shown that excellent predictions of lift and drag due to lift of sharp-edge delta wings over a wide range of angles of attack and aspect ratios can be obtained by combining the potential-flow lift and the vortex lift as predicted by the leading-edge-suction analogy. The resulting equations are

\[ C_L = K_p \sin \alpha \cos^2 \alpha + K_v \sin^2 \alpha \cos \alpha \] (1)

and

\[ \Delta C_D = K_p \sin^2 \alpha \cos \alpha + K_v \sin^3 \alpha \] (2)

or

\[ \Delta C_D = C_L \tan \alpha \] (3)

where, in equations (1) and (2), the first term represents the potential-flow contribution and the second term represents the vortex-lift contribution.

In reference 10 it was shown that equation (1) is applicable for wings of arbitrary planform providing, of course, that the constants \( K_p \) and \( K_v \) are calculated for the desired planform. The analogy method makes it possible to use potential-flow theory to predict both the potential-flow term and the vortex-flow term. For the arrow and diamond planforms of interest in this paper, any accurate potential-flow lifting-surface method, such as the methods of references 11 and 12, can be used. Since the method of reference 12 appears to offer some advantages with regard to more general planforms involving broken leading edges, it has been programmed at Langley for use in certain lifting-surface studies and was used for the present calculations of the potential- and vortex-lift constants. The constant \( K_p \) is simply the potential-flow lift-curve slope and the constant \( K_v \) is related to the potential-flow leading-edge thrust parameter. (See eq. (3) of ref. 10.)
The subsonic effects of compressibility can be accounted for by use of the Prandtl-Glauert transformation and the Goethert rule form (see ref. 13) will be used herein. This rule relates the pressure coefficient at a given nondimensionalized point on the real wing at a given Mach number to a pressure coefficient at the same nondimensionalized point on a transformed wing (stretched in longitudinal direction by $1/\beta$) in incompressible flow.

For a wing of zero thickness, the rule can be stated as follows:

$$\langle C_p \rangle_{M, \alpha, A, \Lambda_1} = \frac{1}{\beta^2} \langle C_p \rangle_{M=0, \alpha, A, \Lambda_1}$$

Application to the potential-flow-lift constant $K_p$ is well known, and the effect of compressibility can be accounted for simply by determining the incompressible value for a transformed wing having a reduced aspect ratio equal to $A\beta$ and an increased leading-edge sweep angle whose tangent is greater by $1/\beta$, and then increasing the resulting value of $K_p$ by the factor $1/\beta$. The $1/\beta$ correction to $K_p$ results from combining the $1/\beta^2$ correction to the pressure and the effect of the reduced angle of attack $\alpha\beta$. Therefore, if $K'_p$ is the incompressible value for the transformed wing, then $K_p$ for the real wing at its Mach number is given by

$$K_p = \frac{K'_p}{\beta}$$

With regard to the effect of compressibility on the vortex-lift constant $K_v$, it was assumed that the leading-edge-suction analogy can also be applied in compressible flow; therefore, the problem can be reduced to that of determining the effect of compressibility on the leading-edge suction. Although the same transformed wing is used for the leading-edge suction and the resulting vortex-lift constant $K'_v$, the compressibility factor that must be applied differs from the $1/\beta$ that is used for $K_p$. This is due to two factors. First, since the leading-edge suction increases with the square of the angle of attack, the angle-of-attack reduction associated with the transformed wing completely cancels the $1/\beta^2$ term that is applied to the pressures on the transformed wing. Second, since the method used must be equivalent to applying the transformed wing pressures along the real-wing leading edge (rather than the reference area as in the potential-flow lift case), and since the transformed leading-edge length $e'$ does not increase as rapidly as the transformed-wing area $S'$, the value of $K'_v$ must be corrected to the real wing ratio of the leading-edge length to the area. In other words

$$K_v = K'_v \frac{S'}{S} \frac{e}{e'}$$

and since $\frac{S'}{S} = \frac{1}{\beta}$ and $\frac{e}{e'} = \frac{1 + \tan^2 \Lambda}{1 + \tan^2 \Lambda / \beta^2}$.
\[ K_v = K_v' \frac{1 + \tan^2 \Lambda}{\beta^2 + \tan^2 \Lambda} = K_v' f_M \]  \hspace{1cm} (4)

PRESENTATION OF RESULTS

Lifting-surface solution values for the potential-flow-lift constant \( K_p \beta \) as a function of \( \Lambda_1 e \) are presented in figure 2 by the solid lines. Also presented as an aid in locating a particular wing are dashed lines which represent constant values of notch ratio \( a/l \). These constant notch-ratio lines are also convenient for applying the Prandtl-Glauert transformation since the notch ratio is unaffected by the transformation. Following a constant notch-ratio line removes the need for determining the sweep angles \( \Lambda_1 e \) of the various transformed wings.

Figure 3 presents values of the vortex-lift constant in the form \( K_v/f_M \) as a function of \( \Lambda_1 e \. Again, lines of constant notch ratio are presented for convenience. Values of \( f_M \) as determined from equation (4) are presented in figure 4 as a function of leading-edge sweep angle and Mach number.

For convenience in using the equations, table I presents values of the various combinations of trigonometric functions needed.

With regard to the expected accuracy of the method, reference 10 presents correlations with experimental results for the incompressible case.

CONCLUDING REMARKS

The leading-edge-suction analogy method of predicting the aerodynamic characteristics of slender delta wings has been extended to cover arrow- and diamond-wing planforms. The method of applying compressibility corrections to the leading-edge suction has been examined, and the resulting procedure applied to the vortex-lift constant. Charts for use in calculating the potential- and vortex-flow terms for the lift and drag in subsonic compressible flow are presented for a wide range of planform parameters.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., February 26, 1971.
REFERENCES


# TABLE I.- VALUES OF TRIGONOMETRIC FUNCTIONS

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Figure 1.- Sketch defining wing geometry nomenclature.

Figure 2.- Variation of potential-flow lift constant with planform parameters.
Figure 3. Variation of vortex-lift constant with planform parameters.
Figure 4.- Variation of compressibility factor with sweep and Mach number.
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— National Aeronautics and Space Act of 1958

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