WIND TUNNEL TEST OF TANDEM WING AIRPLANE
MODEL WITH HIGH ASPECT-RATIO

By

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WIND TUNNEL TEST OF TANDEM WING AIRPLANE
MODEL WITH HIGH-ASPECT-RATIO

Summary

The tests in this report were conducted for the purpose of determining the stability and efficiency characteristics of tandem wing arrangements, with high aspect-ratio on airplane models. The model used in this test was equipped with trailing edge flaps.

Lift and Drag determinations were made on a model of this type, with various combinations of front and rear wing flap settings. For all tests the ailerons were kept in neutral position except for one combination of the flaps settings. When both flaps were down 20 degrees a test was run with the flaps in neutral and one for both ailerons down 15 degrees. The model was equipped with Clark Y 18 wings.

Tests were also made on a single wing of the Clark Y wing section, 24" x 4" aspect-ratio 6.

Introduction

The testing of model wing sections and airplane models, in wind tunnels, has been recognized since the early development of the airplane, as a means of determining the characteristics and behavior of the full scale airplane in flight. Since types of aircraft vary greatly in their qualities of stability and safety, it is important that one have a reasonable assurance that the plane he intends to design and build will possess favorable performance characteristics.

Since this particular type of airplane is of unconventional design and construction, it becomes increasingly important that something be known regarding its stability and efficiency characteristics, before it
is constructed. By stability is meant the tendency of the plane to return to its original altitude of steady motion, after any disturbance therefrom. Efficiency is the ratio of the amount of load the plane will carry to the drag created when carrying that load. These values of course vary with the speed of the plane and the angle of inclination of the wing to the relative wind.

It has been shown in wind tunnel tests that the airfoil of high aspect ratio has a higher maximum lift coefficient, and that a tandem wing combination with a separation of two or three chord lengths may be distinctly superior in aerodynamic performance to a biplane of the same aspect ratio.\footnote{pp.91 and 128 Warner} The consideration of these facts has prompted this unusual design in airplanes.

**Description of Tunnel**

The tests were conducted in the five foot atmospheric wind tunnel at the University of Kansas. It is shown in longitudinal section in Figure I. The tunnel is of the open circuit, closed throat type, and consists of a parabolic entrance and a straight side exit cone connected by a cylinder. The overall length is approximately sixty-three feet and the cylinder portion ten feet. The cross-section is everywhere circular, the diameter at the throat being 5 feet. A more complete description together with standardization tests will be found in a Thesis by George O'Mara, submitted to the University of Kansas in partial fulfillment of the requirements for a Master of Science Degree.

**Description of Model**

The model was furnished by Ben F. Brown who has been conducting some full scale tests on planes with wings of high aspect ratio.
Model No. I was made 1/24 full scale, and represents a wide departure from the conventional type of airplane design. A three view drawing of this model is shown in Figure 23 which represents the actual size of the model. The span is 24", height 4.75" length over all length 8.5".

The plane, a pusher type, with four place open cockpit construction may be thought of as having four wings. The two upper wings are placed one behind the other in the same plane, parallel to the thrust line and extending the full length of the span. They are connected at the tips presenting a diamond shape arrangement above the fuselage. The two inclined or lower wings, one on either side are connected to the landing gear and slope upward towards the tip, as struts, being connected by small struts 3 inches from the tip. These wings have the same chord length as the two upper wings, this being 1". This arrangement overcomes the structural difficulties involved in the tandem combination with parallel wings which are due to the heavy stresses to be transmitted along the body which connects the two wings. Overcoming of this problem alone has probably accounted for, in a great measure, the slow development of this type of aircraft.

Trailing edge flap gears of 30% of the chord, on both upper wings, gives the necessary longitudinal control without horizontal tail surfaces, and furnishes a means of varying the lift on either wing in any ratio desired, thereby contributing to the longitudinal stability, and decrease in the landing speed, both of which increases safety in flying. These flaps are designed to work differentially to give pitching moment, and in unison to control the stalling speed and vary the effective angle of incidence of the wings. The length of each
The flap is 84% of the half span or 10 inches, commencing at the center line of the fuselage and continuing outward to within 2 inches of the wing tip. Ailerons of 30% of the chord, fitted to the lower wings are designed to give the necessary lateral control. These continue the full length of the lower wing.

A full scale airplane has been constructed with a single upper wing but with lower wing construction similar to the tandem wing model. It was found that the ailerons fitted to these sloping lower wings did not give the proper lateral control without excessive stick force and binding of the control system. The ailerons therefore were moved out on the tips of the upper wing, and flaps were substituted for the ailerons on the lower wings. With the ailerons on the wing tips very little stick force was required, and the ship responded quickly to the controls.

Larger wind tunnel models of the tandem wing construction have been built using flaps on both the lower and upper wings, and placing the ailerons on the wing tips.

The undercarriage is of the three wheel construction without the usual type of shock absorber; air wheels being used to reduce the shock on landing. The wheels are all well stream-lined to cut down the wind resistance. The streamlining is such that shock absorber struts with the ordinary airplane tires and wheels could be incorporated in the design instead of airwheels without changing the streamline effect materially or increasing the resistance. The strut connections from the landing gear to the fuselage, on either side are all housed in one large stream-line form, resembling a short portion of a wing. Near the point where the lower end of these struts are attached.
to the wheel, the lower wing is connected.

Attached to the fuselage at the rear is a third wheel which is really the lower part of the fin and rudder.

Behind the rudder is the propeller. The motor being placed in the body of the fuselage with an extension shaft running back to the propeller, affords better stream-lining of the fuselage, thereby decreasing the parasite resistance.

The Clark Y wing section was rectangular in plan-form having a span of 24" and chord of 4" making an aspect-ratio of 6. This wing section was not equipped with flaps. The test of this wing section was made for a comparison of the data obtained in the Wind Tunnel at the University of Kansas, and other wind tunnel data.

**Method of Mounting**

Simple wire balances were used for measuring the lift and drag. These balances are shown schematically in Figures 2 and 3. They consist of three independent, simple beams on knife edges. Two are used for measuring the lift components and one for measuring the drag. A photograph of the arrangement of the balances in the test is shown in Figure 4. Damping of the balances was accomplished by attaching to the bottom of the scale pan a small rod about four inches in length with a 1-1/2" disc soldered to the lower end. The disc was emersed in a vessel containing a medium weight grade of oil. This set-up gave very satisfactory results.

In mounting the model in the wind tunnel, two small steel rods 3/16" in diameter and 12-1/2" long were fastened rigidly to the wings. These rods were placed 20" apart and parallel to the thrust line. The rods were so located longitudinally that the front ends were
Figure 2
Schematic Diagram Showing
Wire Type Wind Tunnel Balance
To Drag Balance

To Lift Balance

Detail below

TUNNEL WALL

TUNNEL WALL

Tandem Wing Model

Wind Direction

Note:—
Not drawn to Scale.

Adjustment

To rear lift balance

Counter Weight

METHOD OF VARYING
ANGLE OF ATTACK

METHOD OF MOUNTING MODEL
IN WIND TUNNEL

Figure 3

To Model
in line with the farthest point of the leading edge of the front wing, as shown in Figures 2 and 5. A photograph of the model as it was mounted in the tunnel is shown in Figure 5. In this view most of the wires used for measuring the lift and drag are shown distinctly.

In testing the Clark Y Airfoil Section the same rods were used and the model wing was mounted with the leading edge 2 inches behind the point of attachment of the front lift wires to the rods.

This type of mounting is very simple and affords a convenient method of determining the drag of the rigging. In determining this drag, two counter weights were substituted for the single one. One counter weight being attached to each rod. This arrangement allowed the rods to remain parallel to the wind direction after removing the model from them. After the model was removed from the tunnel a series of tests was run on the rigging at each angle of attack as when the model was in place. These values were then subtracted from the Drag and lift values obtained before removing the model. It was necessary with this arrangement that the drag be measured at every angle of attack that was used when the model was in place, since at all angles except that of 0 degrees, a component of the drag on the two rods is measured on the two lift wires. This method of course did not take into account the possible interference effect which would be produced when the model was in place. It is necessary in this arrangement that the drag wires be set at the proper angle. Looking at Figure 3 we can readily see that if the angle of the wire D is not 45° then the wire A will not measure the true drag force. At the point E, we have a system of three concurrent forces in equilibrium as shown below.
Figure 4
Wire Balances in 5-Feet Wind Tunnel
University of Kansas

Figure 5
Tandem Wing Airplane Model
Asmounted in Wind Tunnel
Then $D = T \sin \theta$, and $F = T \cos \theta$

If the drag force is to be measured direct then $F$ must be equal to $T \sin \theta$, and this can be true only when $\sin \theta = \cos \theta$ which is when $\theta = 45^\circ$

**Method of Testing**

Tests on the model plane and the Clark Y wing section were all made at a speed of about 80 miles per hour, varying of course with the temperature and pressure at the time the tests were made.

The angle of attack was varied by raising or lowering the rear lift wires. This was accomplished by providing two small aluminum strips with holes drilled at equal intervals as shown in Figure 3.

**Test Results**

The coefficients are based on the true dynamic pressure which was determined for each observation.

The absolute system of coefficients have been used throughout.

These are defined by the relations:

$$L = C_L q S$$

$$D = C_D q S$$

Pitching moment $M_A = C_M q S$.

where:

$L$ = lift in pounds

$D$ = Drag in pounds

$M_A$ = Moment about the leading edge of the front wing at the center line of the fuselage.

$C_L$ = Lift Coefficient (Absolute)

* See Appendix
\[ C_D = \text{Drag Coefficient (Absolute)} \]
\[ C_M = \text{Moment Coefficient (Absolute)} \]
\[ q = \text{Dynamic pressure} = \frac{1}{2} \rho V^2 \text{ lb per sq ft} \]
\[ S = \text{Wing area in sq ft} \]
\[ c = \text{The distance from the leading edge of the front wing to the trailing edge of the rear wing, measured at the center line of the fuselage, along a line connecting the chords, of the two upper wings, measured in inches.} \]

The absolute center of pressure coefficient \( C_p \) is the fraction of the distance \( c \) as above, measured from the leading edge of the front wing to the line of action of the resultant force. This is equal to the moment coefficient divided by the normal force coefficient \( C_N \).

\[ C_p = \frac{C_{p*}}{c} = \frac{C_M}{C_N} \]

where:

\[ C_{p*} = C_L \cos \alpha + C_D \sin \alpha \]

\[ C_{p*} = \text{center of pressure location from leading edge of front wing} \]

\[ \alpha = \text{Angle of attack} \]

**Tunnel Wall Corrections**

The corrections of induced drag and angle of attack due to tunnel wall interference, have been applied, to the drag coefficients and angles of attack, by the use of Prandtl's formula (NACA Technical Report No. 275).

\[ \Delta C_{D1}^* = \frac{C_L^2 S}{2 \pi D^2} \left(1 + \left(\frac{b}{D}\right)^{\frac{4}{3}} + \ldots\right) \]

\[ \Delta \alpha^* = \frac{57.5 \cdot C_L S}{2 \pi D^2} \left(1 + \left(\frac{b}{D}\right)^{\frac{4}{3}} + \ldots\right) \]

where:

\[ \Delta C_{D1} = \text{induced drag correction} \]

\[ \Delta \alpha = \text{induced angle of attack} \]

* See Appendix.
\[ S = \text{Wing area of model} \]
\[ b = \text{span of model} \]
\[ D = \text{diameter of tunnel} \]

In these equations \( \frac{b}{D} \) is very small and is therefore neglected.

**Summary of Tables and Figures**

The following summary of tables and figures is given for convenience:

Test data on tandem wing model with rear flap set at 0 degrees and front flap setting varied from 0 degrees to 20 degrees down in 5 degree intervals. Tables I to V inclusive.

\( (C_L & C_M) vs \alpha \) Figure 6

\( (C_D & C_p) vs \alpha \) Figure 11

\[ \frac{C_L}{C_D} vs \alpha \) Figure 16

Pitching moment vectors Figures 1a to 1a inclusive.

Test data on tandem wing model with rear flap set 5 degrees down and front flap setting varied from 0 degrees to 20 degrees down in 5 degree intervals. Tables VI to X inclusive.

\( (C_L & C_M) vs \alpha \) Figure 7

\( (C_D & C_p) vs \alpha \) Figure 12

\[ \frac{C_L}{C_D} vs \alpha \) Figure 17

Pitching moment vectors Figures VIIa to XIIa inclusive.

Test data on tandem wing model with rear flap 10 degrees down and front flap setting varied from 0 degrees to 20 degrees down in 5 degree intervals. Tables XI to XV inclusive.

\( (C_L & C_M) vs \alpha \) Figure 8

\( (C_D & C_p) vs \alpha \) Figure 13
$C_L/C_D \; \text{vs} \; \alpha \; \text{Figure 18}$

Pitching moment vectors Figures XIa to XVa inclusive.

Test data on tandem wing model with rear flap 15 degrees down and front flap setting varied from 0 degrees to 20 degrees down in 5 degree intervals. Tables XVI to XX inclusive.

$(C_L \; \& \; C_M) \; \text{vs} \; \alpha \; \text{Figure 9}$

$(C_D \; \& \; C_P) \; \text{vs} \; \alpha \; \text{Figure 14}$

$C_L/C_D \; \text{vs} \; \alpha \; \text{Figure 19}$

Pitching moment vectors Figures XVIa to XXa inclusive.

Test data on tandem wing model with rear flap set 20 degrees down, and front flap setting varied from 0 degrees to 20 degrees down in 5 degree intervals, and also one run with both flaps down 20 degrees and both ailerons down 15 degrees. Tables XXI to XXVI inclusive.

$(C_L \; \& \; C_M) \; \text{vs} \; \alpha \; \text{Figure 10}$

$(C_D \; \& \; C_P) \; \text{vs} \; \alpha \; \text{Figure 15}$

$C_L/C_D \; \text{vs} \; \alpha \; \text{Figure 20}$

Pitching moment vectors Figures XXIa to XXVIa inclusive

Test data on Clark Y Wing Section, Table XXVIIa

$(C_{L-c_D-c_M-c_P} \; \& \; C_L/C_D) \; \text{vs} \; \alpha \; \text{Figure 21}$

Pitching moment vectors Figure XXVIIa.

**Discussion of Data**

In plotting the curves from this data it was thought advisable to use as large a scale as possible in order to separate the curves and give a clearer representation of the variation under the different test conditions. To do this the lift and moment coefficients were plotted on one sheet, the drag and center of pressure on another, and the lift-
Drag ratio on another.

It will be observed from the lift coefficient curves that $C_{\text{Lmax}}$, for all different settings of the rear flap, occurs at about 21.2 degrees angle of attack when the front flap is down 15 degrees or 20 degrees. For the other conditions of the front flap settings (that is 0 degrees, 5 degrees, and 10 degrees); irrespective of the rear flap setting, $C_{\text{Lmax}}$ occurs at about 14 degrees angle of attack. The lift coefficient appears to reach its maximum value at a slightly smaller angle of attack as the flap angle is increased. If more values had been taken in this vicinity the results would have shown this more clearly. The maximum value of the lift coefficient varies from 0.7623 at 14 degree angle of attack with both flaps at 0 degrees to 0.9981 at 21.25 degrees angle of attack when both flaps are down 20 degrees and both ailerons down 15 degrees. However the slope of the lift curves do not change appreciably, although there is a tendency to increase in slope as the flap angles are varied from 0 degrees. The fact that at large flap angles $C_{\text{Lmax}}$ occurs at near 21.2 degrees would seem to convey the idea, that the slope of the lift curve decreased as the flap angles were increased. But near 14 degrees angle of attack the lift curve flattens out and becomes almost horizontal, with a slight increase in the coefficient up to 21.2 degrees and then a small decrease to 28.8 degrees angle of attack. The value of $C_{L}$ at 28.8 degrees attack angle in most cases, with the front flap down 15 degrees or 20 degrees was about equal to or more than the value at 14 degrees angle of attack. This gives what is known as a flat top lift curve for these flap settings.

For the front flap setting of 0 degrees, 5 degrees and 10 degrees, the lift coefficient falls quite rapidly until the attack angle of about
20\ degrees\ is\ reached\ when\ the\ curve\ flattens\ out\ and\ falls\ very
gradually\ to\ 22.8\ degrees\ angle\ of\ attack.\ The\ maximum\ decrease
in\ the\ lift\ coefficient\ from\ maximum\ value\ was\ about\ 15.9\%\ of\ \(C_{\text{max}}\)
which\ occurred\ with\ the\ front\ flap\ at\ 0\ degrees\ and\ the\ rear\ flap
down\ 10\ degrees.

Continuation\ of\ \(C_{\text{max}}\)\ through\ a\ large\ range\ of\ angles\ of\ attack
is\ a\ very\ desirable\ quality\ for\ an\ airplane,\ since\ it\ lessens\ the
possibility\ of\ stalling\ the\ ship\ and\ falling\ into\ a\ spin.

Wind\ tunnel\ tests\ have\ shown\ that\ tandem\ wing\ arrangements
cause\ the\ center\ of\ pressure\ to\ move\ back\ as\ the\ angle\ of\ attack\ is
increased,\ although\ for\ single\ wing\ tests\ the\ center\ of\ pressure\ moves
forward.\ This\ is\ accounted\ for\ by\ the\ relative\ increase\ in\ lift\ of
the\ rear\ wing\ to\ the\ front\ one\ as\ it\ is\ moved\ out\ of\ the\ downwash\ from
the\ front\ wing.\ Although\ the\ rear\ wing\ furnishes\ less\ than\ half\ the\ lift,
the\ relative\ increase\ of\ lift\ for\ the\ rear\ wing\ is\ a\ little\ larger
than\ for\ the\ front\ one,\ and\ the\ center\ of\ pressure\ of\ the\ combination
therefore\ tends\ to\ move\ back\ towards\ the\ rear\ airfoil.\ A\ fair\ comparison
of\ the\ results\ of\ above\ tests\ with\ this\ type\ of\ arrangement\ could\ not
be\ made\ due\ to\ unsimilarity\ of\ the\ models.\ The\ former\ being\ parallel
tandem\ wing\ arrangements\ only\ and\ the\ latter\ a\ complete\ model\ plane
with\ tandem\ wings\ attached\ at\ the\ tips\ and\ separated\ by\ 4\ chord\ lengths
at\ the\ connections\ to\ the\ fuselage.\ It\ is\ interesting\ to\ observe\ that
the\ general\ outline\ of\ each\ center\ of\ pressure\ curve\ is\ practically\ the
same\ for\ each\ front\ flap\ setting,\ while\ the\ rear\ flap\ is\ varied.\ The
only\ change\ is\ a\ shift\ of\ the\ curve\ toward\ the\ rear\ wing.\ This\ can\ be
noted\ by\ comparing\ the\ curves\ in\ Figure\ 11\ and\ Figure\ 15\ when\ both
1\ p.\ 129,\ Warner
flaps are set at 0 degrees and when the front one is at 0 degrees and the rear one at 20 degrees down. The shift in this case amounts to about 5%. The rest of the curves have about the same amount of variation when the front flap is kept at a constant angle while the rear flap is varied. A condition of this kind would be expected since by lowering the rear wing flap the angle of attack with the relative wind is increased and therefore would produce a larger proportional part of the lift at this point. In general when the rear flap is set at a constant angle the center of pressure moves forward continuously to angle of attack of 28.8 degrees for the front flap at 0 degrees and 5 degrees. For the front flap setting of 10 degrees, 15 degrees and 20 degrees it moves forward to about 7 degrees and then backward to about 18 degrees. Angle of attack. From here on the curves move forward again. The movement in each case is fast up to about 14 degrees and then the change is very gradual being almost constant.

A study of the vector diagrams Figures Ia to XXVIa inclusive, show the center of pressure travel quite clearly for each combination of flap setting. The center of pressure curves seem to indicate at low angles of attack that the rear wing furnishes most of the lift, but this can be accounted for, by realizing that the lift is small at these angles and the drag has a decided influence in moving the center of pressure back towards the rear wing. Although the drag is measured at the center of rotation the wind reaction on the fuselage has a tendency to produce a higher lift on the rear wing in relation to the lift on the front wing.

The drag curves Figures 11 to 15 inclusive show no unusual characteristics, except in the case where the front flap is set down 15 degrees. With any rear flap setting, with the front flap at this
angle the drag values at low angles of attack drop materially below the values for any other flap setting. Whatever the cause of this unusual condition it does not seem to have affected the values of the lift coefficient. The particular type of construction of the model may have influenced the drag values under these particular conditions.

It is noticed that the lift and drag coefficient curves in these tests are smooth and continuous with no abrupt changes as appears in the test on the single wing Clark Y-18 Airfoil of aspect-ratio 6. (N.A.C.A. Technical Report No. 286). Although the latter test was run at much higher Reynolds Number, being about 137,200 as compared to 63,820 for the tandem wing model. This low value of Reynolds number makes it difficult to make any general comparison with other model tests.

The low values of the lift coefficients and relative high values of the drag coefficients in these tests can be accounted for in part to scale effect or the low value of Reynolds number. However, the roughness of the model probably has the greater influence. When using the chord for "L" in determining Reynolds number an average value of 63,860* is obtained. This value is arrived at by using the value of L at standard temperature and pressure, and an average velocity of 81.92 mi. per hr. Since the chord is only 1 inch, it becomes quite difficult to construct the wings and get them to the proper smoothness. Later tests have shown that the smoothness of the model affects the results materially. Recently a larger model, 1/16 full scale, has been constructed, and particular attention has been given to making the wings as smooth as possible. The chord and span have been increased 50% of the former value. The results of the test on this model show lift coefficients in the neighborhood of 1.2 and Cl/Cp values as high as 11.

* See Appendix
This size of model increases Reynolds' Number to 95,790 for the same average wind velocity. Part of this increase in lift coefficient and lift drag ratio may be attributed to the increase in Reynolds' number but the major portion is most likely due to better model construction, and smoothness of the wings.

For thick wing sections high Reynolds' Number tests seem to indicate that the maximum lift decreases as the value of Reynolds number is increased. However, for the cruising speed of the plane we are not concerned with the maximum lift but with the lift at the angle of attack of 3 degrees or 4 degrees. In this range the lift decreases very slowly with increase in Reynolds Number. When maximum lift is desired we also desire slow speed for landing. For this condition a high drag is not undesirable provided the action line of the resultant force is such as to give a negative moment, tend to depress the nose of the ship.

Wind tunnel experiments have shown that with thick wing sections such as the Clark Y-18 the U.S.A. 35A and others, the drag decreases with increase in Reynolds number. This is desirable since in the cruising range the drag will be less than is indicated in the wind tunnel test at low Reynolds Number.

Where it is possible to test models at full scale Reynolds Number it is desirable that the test be run at several different values in order that the test data may be comparable to the different conditions of flight. For example, suppose that for this particular model the cruising speed is 100 miles per hr. Since the model is 1/24 full scale and has a chord of 1 inch the full size ship will have a chord of 2 ft. At standard temperature and pressure conditions Reynolds number would be $9554.4 \times 2 \times 100 = 14970,880$. But for landing at a speed of 40 miles per hour this would decrease to 740,352.
With certain types of airfoils the lift and drag values may change considerably within this range of Reynolds Number. Then again suppose this same plane to be equipped with wings of different aspect-ratio having the same span, say instead of a 1' chord a 3 inch chord were substituted the range of Reynolds Number for full scale conditions would be from 5,612,640 to 2,245,056. Some interesting experiments could be run showing the effect of Reynolds Number with changing aspect-ratio. This ratio could be varied from about 4 to 24.

The $\frac{C_L}{C_D}$ curves, Figures 16 to 20 inclusive, show two unusual characteristics. The relatively high value for the front flap setting of 15 degrees down at all other rear flap settings and the approach of all the curves except the ones just mentioned, to a constant value of 1.43 (Tables I to XXVI inclusive) at about 23.8 degrees angle of attack. The first of these conditions is the result of the unusual drag curves for the same flap settings. Since the lift coefficients seemed to be normal for this particular data, the unusual characteristic would show up again in the $\frac{C_L}{C_D}$ values. The relatively low maximum values of the rest of the curves are probably due to the high drag and low lift coefficients caused by the roughness of the model and its extremely small scale.

Figures 1a to XXVIa inclusive, show quite clearly the stabilizing moments of the model. Although the magnitude of the vector is not shown its direction indicates the sign of the moment. As long as the vectors at very low angles of attack tend to cause a rotation about the C.G. in a counter clockwise direction the resultant force tends to bring the ship out of a dive. At angles of attack of 7 degrees and above, if the vector tends to rotate clockwise about the C.G. the resultant force tends to bring the nose of the ship down. It will be noticed
that as the angle of attack is increased the C.G. moves forward but
the vector position remains unchanged. If the C.G. is rotated to the
highest angle of attack it will be observed that in nearly every case
the vector for this condition is in the proper position to cause the
nose of the ship to fall.

The Clark Y wing section (Figure 21) which was also tested was
made of bass wood and polished until it was made very smooth. This wing
section is different to the Clark Y-18 wing section which was used for
the wings on the tandem wing model. Since the model wing had an aspect-
ratio of 6 it furnished a means of comparing the results obtained in
this wind tunnel to those obtained elsewhere. Reynolds number in this
case was \( \frac{9354.4 \times 82.34}{2} = 365,120 \). This value corresponds closely
with the curves "C" for the Clark Y wing section as shown on page 6
of N.A.C.A. Technical Report No. 331. The curves there represented were
taken from a test made at McCook Field on a 6" x 36" wing model at a
Reynolds number of approximately 374,000. A comparison of the lift
coefficient curves from that test and the one made at the University of
Kansas shows them to be almost identical for all points along the curve.
The drag curves are about the same shape but the test at this laboratory
shows the values of the drag coefficient to be slightly lower. In the
case of the \( \frac{C_L}{C_D} \) ratio the maximum value occurs at 0 degrees angle of
attack in both cases but in this wind tunnel the value of the ratio is
considerable higher, even exceeding that obtained for Reynolds number
of approximately 3,610,000 when tested in the variable density wind
tunnel of the N.A.C.A.

Moment and center of pressure coefficients are somewhat higher
in the test at this University but they follow the same general outline.
General Conclusions

1. Data from tests in the five-foot wind tunnel at the University of Kansas is reliable if care is taken in obtaining the measurements, and in preparation of the models for testing.

2. With the C.G. located as shown a tandem wing airplane such as herein described has inherent static longitudinal stability at angles of attack within the range of the test, with all combinations of flap settings between 0 degrees and 20 degrees down, except in a few cases at angles between 4 degrees and 7 degrees.

3. More experimental data is needed on airfoil sections of high aspect-ratio at various Reynolds Numbers.
TANDEM WING AIRPLANE MODEL
WITH TRAILING EDGE FLAPS

$C_L$ AND $C_m$ VS ANGLE OF ATTACK
WITH DIFFERENT FRONT FLAP SETTINGS
REAR FLAP SET DOWN $-0^\circ$

MODEL NO. I.

FIGURE 6

Angle of Attack in Degrees $\alpha$
Tandem Wing Airplane Model with Trailing Edge Flaps

$C_L$ and $C_M$ vs Angle of Attack with Different Front Flap Settings Rear Flap Set Down + 5°

Figure 7

Angle of Attack in Degrees $\alpha$
Tandem Wing Airplane Model with Trailing Edge Flaps

$C_L$ and $C_m$ vs Angle of Attack with Different Front Flap Settings Rear Flap Set Down $+10^\circ$

Model No. I

Figure 8

Angle of Attack in Degrees $\alpha$
Tandem Wing Airplane Model with Trailing Edge Flaps

$C_L$ and $C_m$ vs Angle of Attack with Different Front Flap Settings

Rear Flap Set Down - 20°

Model No. 1

Figure 10

Front Flap Symbol

<table>
<thead>
<tr>
<th>Setting</th>
<th>Symbol</th>
</tr>
</thead>
<tbody>
<tr>
<td>+20°</td>
<td>O</td>
</tr>
<tr>
<td>+15°</td>
<td>⊙</td>
</tr>
<tr>
<td>+10°</td>
<td>⊙⊙</td>
</tr>
<tr>
<td>+5°</td>
<td>⊙⊙⊙</td>
</tr>
<tr>
<td>0°</td>
<td>●</td>
</tr>
</tbody>
</table>

Angle of Attack in Degrees

Millimeters, 10th figure heavy
TANDEM WING AIRPLANE MODEL WITH TRAILING EDGE FLAPS

$C_D$ AND $C_p$ VS ANGLE OF ATTACK WITH DIFFERENT FRONT FLAP SETTINGS
REAR FLAP SET DOWN 0°

MODEL No. I.

$C_D$, $C_p$, and center of pressure vs angle of attack in degrees for different front flap settings.

---

FIGURE II

FRONT FLAP SETTING

$+20°$  $+15°$  $+10°$  $+5°$  $0°$

SYMBOL

To 1.65

Angle of Attack in Degrees
Tandem Wing Airplane Model with Trailing Edge Flaps

Coefficient of Drag $C_d$ and Coefficient of Pressure $C_p$ vs Angle of Attack

With different front flap settings, rear flap set down 5°

Model No. 1

Front Flap Setting | Symbol
--- | ---
$+20^\circ$ | 0
$+15^\circ$ | 0
$+10^\circ$ | 0
$+5^\circ$ | 0
0° | 0

Figure 12

Angle of Attack in Degrees
Tandem Wing Airplane Model with Trailing Edge Flaps

$C_D$ and $C_p$ vs Angle of Attack with Different Front Flap Settings
Rear Flap Set Down +10°

Model No. I.

Figure 13

Front Flap Settings
+20°
+15°
+10°
+5°
0°

Center of Pressure Coefficient (Absolute) $C_p$

Drag Coefficient (Absolute) $C_D$

Angle of Attack in Degrees $\alpha$
Tandem Wing Airplane Model
With Trailing Edge Flaps

$C_D$ and $C_p$ vs Angle of Attack
With Different Front Flap Settings
Rear Flap Set Down $+15^\circ$

Model No. I.

Figure 14
Tandem Wing Airplane Model with Trailing Edge Flaps

$C_D$ and $C_p$ vs Angle of Attack
With Different Front Flap Settings
Rear Flap Set Down +20°

Model No. 1

Figure 15

Angle of Attack in Degrees, $\alpha$
Tandem wing airplane model with trailing edge flaps

$C_L/C_D$ vs Angle of Attack with different front flap settings
Rear flap set down $+0^\circ$

Model No. 1

Figure 16

Angle of Attack in Degrees
<table>
<thead>
<tr>
<th>FRONT FLAP SETTING</th>
<th>SYMBOL</th>
</tr>
</thead>
<tbody>
<tr>
<td>+20°</td>
<td>○</td>
</tr>
<tr>
<td>+15°</td>
<td>○</td>
</tr>
<tr>
<td>+10°</td>
<td>○</td>
</tr>
<tr>
<td>+5°</td>
<td>○</td>
</tr>
<tr>
<td>0°</td>
<td>●</td>
</tr>
</tbody>
</table>

**Tandem Wing Airplane Model with Trailing Edge Flaps**

**$C_L/C_D$ vs Angle of Attack with Different Front Flap Settings**

Rear Flap Set Down +10°

**Model No. I**

**Figure 18**

Angle of Attack in Degrees
TANDEM WING AIRPLANE MODEL WITH TRAILING EDGE FLAPS

$C_L/C_D$ VS ANGLE OF ATTACK WITH DIFFERENT FRONT FLAP SETTINGS
REAR FLAP SET DOWN 15"

MODEL NO. I

Angle of Attack in Degrees

Figure 19
TANDEM WING AIRPLANE MODEL
WITH TRAILING EDGE FLAPS

$C_L/C_D$ VS ANGLE OF ATTACK
WITH DIFFERENT FRONT FLAP SETTINGS
REAR FLAP SET DOWN $+20^\circ$

MODEL NO. 1

FIGURE 20

Angle of Attack in Degrees
CHARACTERISTIC CURVES FROM WIND TUNNEL TEST ON CLARK Y WING SECTION 2.4 x 4
AIR VELOCITY 82.34 M. PER HR.

Angle of Attack in Degrees

FIGURE 21
Pitching Moment Vectors
Tandem Wing Airplane
Model No. 1.

Front Flap Set Down 0°
Rear Flap Set Down 0°

Scale of Model 1/24

Note: Vectors are not drawn to scale.

Figure 1a
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.
Front Flap Set Down 5°
Rear Flap Set Down 0°
Scale of Model 1/24.

Note:
Vectors are not drawn to Scale.

Figure IIa
Pitching Moment Vectors
Tandem Wing Airplane

Model No. I.

Front Flap Set Down 10°
Rear Flap Set Down 0°

Scale of Model 1/24

Note:
Vectors are not drawn to scale.

Figure III
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.

Front Flap Set Down 15°
Rear Flap Set Down 0°

Scale of Model \( \frac{1}{24} \)

Note: Vectors are not drawn to scale

Figure IVa
Pitching Moment Vectors
Tandem Wing Airplane
Model No I.

Front Flap Set Down 20°
Rear Flap Set Down 0°

Scale of Model 1/24

Note: Vectors are not drawn to scale

Figure Vα
Pitching Moment Vectors
Tandem Wing Airplane
Model No. 1.

Front Flap Set Down 0°
Rear Flap Set Down 5°

Scale of Model 1/24

Note:
Vectors are not drawn to scale.

Figure VI
Pitching Moment Vectors
Tandem Wing Airplane
Model No I.
Front Flap Set Down 5°
Rear Flap Set Down 5°
Scale of Model 1/24

Note: Vectors are not drawn to scale.

Figure VIIa
Pitching Moment Vectors
Tandem Wing Airplane
Model No. 1

Front Flap Set Down 10°
Rear Flap Set Down 5°

Note:
Vectors are not drawn to scale.

Figure VIII
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.

Front Flap Set Down 15°
Rear Flap Set Down 5°

Scale of Model 1/24

Note:
Vectors are not drawn to scale

Figure IXa
Pitching Moment Vectors
Tandem Wing Airplane
Model No I.
Front Flap Set Down 20°
Rear Flap Set Down 5°
Scale of Model 1/24

Note: Vectors are not drawn to Scale.

Figure Xa
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.
Front Flap Set Down 0°
Rear Flap Set Down 10°
Scale of Model 1/24

Note:-
Vectors are not
drawn to Scale

Figure XI.a
Pitching Moment Vectors
Tandem Wing Airplane
Model No I.

Front Flap Set Down 5°
Rear Flap Set Down 10°

Scale of Model \( \frac{1}{24} \)

Note: Vectors are not drawn to Scale.

Figure XII
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.

Front Flap Set Down 10°
Rear Flap Set Down 10°

Scale of Model $\frac{1}{2}$ A

Note: Vectors are not drawn to scale.

Figure XIIIa
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I
Front Flap Set Down 15°
Rear Flap Set Down 10°
Scale of Model 1/24.

Note:
Vectors are not drawn to Scale.

Figure XIV.
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.

Front Flap Set Down 20°
Rear Flap Set Down 10°

Scale of Model 1/24

Note: Vectors are not drawn to scale.

Figure XVa
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.

Front Flap Set Down 0°
Rear Flap Set Down 1.5°

Scale of Model 1/24

Note:
Vectors are not drawn to Scale.

Figure XVIa
Pitching Moment Vectors
Tandem Wing Airplane
Model No. 1.

Front Flap Set Down 5°
Rear Flap Set Down 15°

Scale of Model 1/24

Note:
Vectors are not drawn to scale

Figure XVIIa
Pitching Moment Vectors
Tandem Wing Airplane
Model No I.
Front Flap Set Down 10°
Rear Flap Set Down 15°
Scale of Model 1/24

Note: Vectors are not drawn to scale

Figure XVIIIa
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.
Front Flap Set Down 15°
Rear Flap Set Down 15°
Scale of Model 1/4 A.

Note: Vectors are not drawn to Scale.

Figure XIX a
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.
Front Flap Set Down 20°
Rear Flap Set Down 15°
Scale of Model 1/24.

Note: Vectors are not drawn to scale

Figure XXa
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.
Front Flap Set Down 0°
Rear Flap Set Down 20°
Scale of Model 1/24.

Note:
Vectors are not drawn to Scale.

Figure XXIa
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I
Front Flap Set Down 5°
Rear Flap Set Down 20°
Scale of Model 1/24

Note: Vectors are not drawn to scale

Figure XXIIa
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.
Front Flap Set Down 10°
Rear Flap Set Down 20°
Scale of Model 1/24.

Note: Vectors are not drawn to scale.

Figure XXIII.
PITCHING MOMENT VECTORS
TANDEM WING AIRPLANE
MODEL NO. 1.
FRONT FLAP SET DOWN 15°
REAR FLAP SET DOWN 20°
Scale of Model 1/24.

Note: Vectors are not drawn to scale.

Figure XXIVa
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.

Front Flap Set Down 20°
Rear Flap Set Down 20°

Scale of Model 1/24.

Note: Vectors are not drawn to scale.

Figure XXV a
Pitching Moment Vectors
Tandem Wing Airplane
Model No. I.

Front Flap Set Down 20°
Rear Flap Set Down 20°
Both Ailerons Down 15°

Scale of Model ¼.

Note:
Vectors are not drawn to Scale.

Figure XXXVI a
Pitching Moment Vectors
CLARK Y Airfoil Section
Model 24" x 4"

Note: Vectors are not drawn to scale

Figure XXVIIa
### TABLE I

**Front Flap Down 0 Degrees**  
**Rear Flap Down 0 Degrees**

<table>
<thead>
<tr>
<th></th>
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<th></th>
<th></th>
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<th></th>
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<tbody>
<tr>
<td>-3.42</td>
<td>.0102</td>
<td>-1.432</td>
<td>-1.42</td>
<td>+.0240</td>
<td>.027</td>
<td>-3.44</td>
<td>.1011</td>
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<td>-.01</td>
<td>.0085</td>
<td>.0750</td>
<td>0.85</td>
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<td>.768</td>
<td>0</td>
<td>.0985</td>
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<td>.460</td>
<td>28.69</td>
<td>.4587</td>
</tr>
</tbody>
</table>

*Uncorrected for tunnel wall effect.*

\[ p = \text{Density} = 0.002396 \text{ lb.-sec}^{-2}\text{-ft.} \]
\[ q = \text{Dynamic pressure} = 17.159 \text{ lb./sq.ft.} \]
\[ V = \text{Velocity} = \frac{681.8}{\sqrt{p}} = 81.72 \text{ miles/hour} \]

### TABLE II

**Front Flap Down 5 Degrees**  
**Rear Flap Down 0 Degrees**

<table>
<thead>
<tr>
<th></th>
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<th></th>
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<td>.0869</td>
<td>.1455</td>
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<td>-.0761</td>
<td>.523</td>
<td>0</td>
<td>.0868</td>
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<td>.449</td>
<td>.6.99</td>
<td>.1035</td>
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<td>.7936</td>
<td>4.73</td>
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<td>.465</td>
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<td>.2523</td>
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<td>.3377</td>
<td>.7000</td>
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<td>.467</td>
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<td>28.69</td>
<td>.4594</td>
</tr>
</tbody>
</table>

* Uncorrected for tunnel wall effect.

\[ p = \text{Density} = 0.002396 \text{ lb.-sec}^{-2}\text{-ft.} \]
\[ q = \text{Dynamic pressure} = 17.159 \text{ lb./sq.ft.} \]
\[ V = \text{Velocity} = \frac{681.8}{\sqrt{p}} = 81.86 \text{ miles/hour} \]
### TABLE III

Front Flap Down 10 Degrees  
Rear Flap Down 0 Degrees

<table>
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<th></th>
<th></th>
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<tr>
<td>a</td>
<td>C_D</td>
<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
<td>C_P</td>
<td>*a1</td>
<td>*C_{D1}</td>
</tr>
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<td>-.416</td>
<td>.453</td>
<td>28.69</td>
<td>.5261</td>
</tr>
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</table>

*Uncorrected for tunnel wall effect

\[
p = \text{Density} = \cdot002332 \text{ lb.-sec}^2-\text{ft}^{-2}
\]
\[
q = \text{Dynamic pressure} = 17.115
\]
\[
V = \text{Velocity} = \cdot6818/2q_p = 82.60 \text{ miles/hour}
\]

### TABLE IV

Front Flap Down 15 Degrees  
Rear Flap Down 0 Degrees

<table>
<thead>
<tr>
<th></th>
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<th></th>
<th></th>
<th></th>
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</thead>
<tbody>
<tr>
<td>a</td>
<td>C_D</td>
<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
<td>C_P</td>
<td>*a1</td>
<td>*C_{D1}</td>
</tr>
<tr>
<td>-3.43</td>
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<td>.1942</td>
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</table>

*Uncorrected for tunnel wall effect

\[
p = \text{Density} = \cdot002366 \text{ lb.-sec}^2-\text{ft}^{-2}
\]
\[
q = \text{Dynamic pressure} = 17.056
\]
\[
V = \text{Velocity} = \cdot6818/2q_p = 81.56 \text{ miles/hour}
\]
### TABLE V

Front Flap Down 20 Degrees  
Rear Flap Down 0 Degrees

<table>
<thead>
<tr>
<th></th>
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<td>396</td>
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<td>5.747</td>
</tr>
</tbody>
</table>

*Uncorrected for tunnel wall effect.*

\[ p = \text{Density} = 0.02334 \text{ lb}-\text{sec}^2-\text{ft}^2 \]
\[ q = \text{Dynamic pressure} = 16.874 \text{ lb/sq. ft.} \]
\[ V = \text{Velocity} = \frac{6218}{p} = 81.99 \text{ miles/hour} \]
### TABLE VI
Front Flap Down 0 Degrees  
Rear Flap Down 5 Degrees

<table>
<thead>
<tr>
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<th></th>
<th></th>
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<td>0.382</td>
<td>0.471</td>
<td>23.69</td>
<td>4.6200</td>
</tr>
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</table>

*Uncorrected for tunnel wall effect.

\[
p = \text{Density} = 0.002380 \text{ lb.-sec}^2\text{-ft.}^{-1}\]
\[
q = \text{Dynamic pressure} = 17.16 \text{ lb./sq.ft.}

\[
V = \text{Velocity} = \frac{8818/2q}{p} = 61.67 \text{ miles/hour}
\]

### TABLE VII
Front Flap Down 5 Degrees  
Rear Flap Down 5 Degrees

<table>
<thead>
<tr>
<th></th>
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<td>0.0836</td>
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<td>2.77</td>
<td>0.386</td>
<td>0.436</td>
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<td>22.683</td>
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<td>0.384</td>
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<td>3.524</td>
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<td>0.7117</td>
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<td>0.402</td>
<td>0.466</td>
<td>23.69</td>
<td>4.955</td>
</tr>
</tbody>
</table>

*Uncorrected for tunnel wall effect.

\[
p = \text{Density} = 0.002320 \text{ lb.-sec}^2\text{-ft.}^{-1}\]
\[
q = \text{Dynamic pressure} = 17.020 \text{ lb./sq.ft.}

\[
V = \text{Velocity} = \frac{8818/2q}{p} = 62.59 \text{ miles/hour}
\]
### TABLE VIII
Front Flap Down 10 Degrees
Rear Flap Down 5 Degrees

<table>
<thead>
<tr>
<th></th>
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<th></th>
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<th></th>
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</tr>
</thead>
<tbody>
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<td>a</td>
<td>$C_D$</td>
<td>$C_L$</td>
<td>$C_L/C_D$</td>
<td>$C_M$</td>
<td>$C_P$</td>
<td>$\alpha_1$</td>
<td>$\alpha_{CD}$</td>
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<td>46</td>
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<td>-0.260</td>
<td>2.65</td>
<td>-1.34</td>
<td>0.514</td>
<td>0</td>
<td>0.935</td>
</tr>
<tr>
<td>6.99</td>
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<td>-3.10</td>
<td>0.469</td>
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<td>2.227</td>
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<td>3.076</td>
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<td>-4.12</td>
<td>0.479</td>
<td>21.10</td>
<td>2.799</td>
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<td>28.81</td>
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<td>-0.774</td>
<td>1.45</td>
<td>-4.39</td>
<td>0.467</td>
<td>29.69</td>
<td>3.354</td>
</tr>
</tbody>
</table>

* Uncorrected for tunnel wall effect.

$p = \text{Density} = 0.002330 \text{ lb-sec}^2\text{-ft}^{-4}$
$q = \text{Dynamic pressure} = 17.103 \text{ lb/sq.ft.}$
$V = \text{Velocity} = 6618\frac{\text{ft}}{\text{sec}} = 81.52 \text{ miles/hour}$

### TABLE IX
Front Flap Down 15 Degrees
Rear Flap Down 5 Degrees

<table>
<thead>
<tr>
<th></th>
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<th></th>
<th></th>
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<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>a</td>
<td>$D_D$</td>
<td>$C_L$</td>
<td>$C_L/C_D$</td>
<td>$C_M$</td>
<td>$C_P$</td>
<td>$\alpha_1$</td>
<td>$\alpha_{CD}$</td>
</tr>
<tr>
<td>-3.43</td>
<td>-0.366</td>
<td>-0.357</td>
<td>41</td>
<td>-0.365</td>
<td>1.204</td>
<td>-3.44</td>
<td>0.366</td>
</tr>
<tr>
<td>-04</td>
<td>-0.309</td>
<td>-0.257</td>
<td>3.13</td>
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<td>0.486</td>
<td>0</td>
<td>0.307</td>
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<td>6.35</td>
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<td>0.482</td>
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<td>3.541</td>
</tr>
<tr>
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<td>-4.949</td>
<td>-0.424</td>
<td>1.67</td>
<td>-4.53</td>
<td>0.472</td>
<td>29.69</td>
<td>4.931</td>
</tr>
</tbody>
</table>

* Uncorrected for tunnel wall effect.

$p = \text{Density} = 0.002366 \text{ lb-sec}^2\text{-ft}^{-4}$
$q = \text{Dynamic pressure} = 17.056 \text{ lb/sq.ft.}$
$V = \text{Velocity} = 6618\frac{\text{ft}}{\text{sec}} = 81.52 \text{ miles/hour}$
### TABLE X
Front Flap Down 20 Degrees  
Rear Flap Down 5 Degrees

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>a</td>
<td>$C_D$</td>
<td>$C_L$</td>
<td>$C_L/C_D$</td>
<td>$C_M$</td>
<td>$C_P$</td>
<td>$*a_1$</td>
<td>$*C_{D_1}$</td>
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<td>522</td>
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<td>1.091</td>
<td>.3395</td>
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<td>454</td>
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<td>21.10</td>
<td>4.294</td>
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<td>.8546</td>
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<td>466</td>
<td>22.69</td>
<td>5.969</td>
</tr>
</tbody>
</table>

* Uncorrected for tunnel wall effect.

- $p$ = Density = 0.0236 lb.-sec.$^{-2}$-ft.$^{-4}$
- $q$ = Dynamic pressure = 17.056 lb/sq.ft.
- $V$ = Velocity = 82.01 miles/hour

### TABLE XI
Front Flap Down 0 Degrees  
Rear Flap Down 10 Degrees

<table>
<thead>
<tr>
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<th></th>
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</thead>
<tbody>
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<td>a</td>
<td>$C_D$</td>
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<td>$C_L/C_D$</td>
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<td>$C_P$</td>
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<td>$*C_{D_1}$</td>
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<td>-.0204</td>
<td>+292</td>
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<td>.1008</td>
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<tr>
<td>.02</td>
<td>.0947</td>
<td>.1219</td>
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<td>-.0955</td>
<td>733</td>
<td>0</td>
<td>.0947</td>
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<tr>
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<td>2.492</td>
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<td>472</td>
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<td>5.160</td>
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</table>

* Uncorrected for tunnel wall effect.

- $p$ = Density = 0.0236 lb.-sec.$^{-2}$-ft.$^{-4}$
- $q$ = Dynamic pressure = 17.056 lb/sq.ft.
- $V$ = Velocity = 82.01 miles/hour
### TABLE XII
Front Flap Down 5 Degrees
Rear Flap Down 10 Degrees

<table>
<thead>
<tr>
<th></th>
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<th></th>
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<th></th>
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</thead>
<tbody>
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<td>a</td>
<td>$C_D$</td>
<td>$C_L$</td>
<td>$C_L/C_D$</td>
<td>$C_M$</td>
<td>$C_P$</td>
<td>$\delta_a$</td>
<td>$C_D$</td>
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<td>.22</td>
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<td>.599</td>
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<td>.0943</td>
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<td>.1906</td>
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<td>.2363</td>
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<td>.475</td>
<td>28.69</td>
<td>.5045</td>
</tr>
</tbody>
</table>

* Uncorrected for tunnel wall effect.

$$
p = \text{Density} = .002318 \text{ lb-sec}^{-1}\text{-ft}^{-1}
q = \text{Dynamic pressure} = 16.998 \text{ lb/sq-ft}^2
V = \text{Velocity} = 6318\sqrt{\frac{q}{p}} = 82.55 \text{ miles/hour}
$$

### TABLE XIII
Front Flap Down 10 Degrees
Rear Flap Down 10 Degrees

<table>
<thead>
<tr>
<th></th>
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<th></th>
<th></th>
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<tbody>
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<td>a</td>
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<td>$C_L/C_D$</td>
<td>$C_M$</td>
<td>$C_P$</td>
<td>$\delta_a$</td>
<td>$C_D$</td>
</tr>
<tr>
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<td>.0666</td>
<td>.65</td>
<td>-.0604</td>
<td>1.003</td>
<td>-3.44</td>
<td>.1029</td>
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<td>.2825</td>
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<td>-.151</td>
<td>.535</td>
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<td>.0974</td>
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<tr>
<td>6.99</td>
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<td>-.331</td>
<td>.486</td>
<td>6.39</td>
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<tr>
<td>14.01</td>
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<td>.3300</td>
<td>3.66</td>
<td>-.416</td>
<td>.484</td>
<td>15.88</td>
<td>2.247</td>
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<td>.6103</td>
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<td>.489</td>
<td>17.46</td>
<td>3.119</td>
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<td>.7985</td>
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<td>.478</td>
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<td>5.463</td>
</tr>
</tbody>
</table>

* Uncorrected for tunnel wall effect.

$$
p = \text{Density} = .002337 \text{ lb-sec}^{-1}\text{-ft}^{-1}
q = \text{Dynamic pressure} = 16.968 \text{ lb/sq-ft}^2
V = \text{Velocity} = 6318\sqrt{\frac{q}{p}} = 82.16 \text{ miles/hour}
$$
### TABLE XIV
Front Flap Down 15 Degrees  
Rear Flap Down 10 Degrees

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>a</td>
<td>C&lt;sub&gt;D&lt;/sub&gt;</td>
<td>C&lt;sub&gt;L&lt;/sub&gt;</td>
<td>C&lt;sub&gt;L&lt;/sub&gt;/C&lt;sub&gt;D&lt;/sub&gt;</td>
<td>C&lt;sub&gt;M&lt;/sub&gt;</td>
<td>C&lt;sub&gt;P&lt;/sub&gt;</td>
<td>*a&lt;sub&gt;1&lt;/sub&gt;</td>
<td>*C&lt;sub&gt;D&lt;/sub&gt;</td>
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<td>-0.0728</td>
<td>0.920</td>
<td>-3.44</td>
<td>0.0352</td>
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<tr>
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<td>0.535</td>
<td>0.0795</td>
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</tr>
<tr>
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<td>0.1045</td>
<td>0.7046</td>
<td>8.76</td>
<td>-0.341</td>
<td>0.479</td>
<td>6.39</td>
<td>0.1032</td>
</tr>
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<td>-0.444</td>
<td>0.510</td>
<td>13.88</td>
<td>0.2050</td>
</tr>
<tr>
<td>17.59</td>
<td>0.2308</td>
<td>0.5337</td>
<td>3.04</td>
<td>-0.469</td>
<td>0.522</td>
<td>17.46</td>
<td>0.2789</td>
</tr>
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<td>2.41</td>
<td>-0.463</td>
<td>0.516</td>
<td>21.10</td>
<td>0.3589</td>
</tr>
</tbody>
</table>

*Uncorrected for tunnel wall effect

\[ p = \text{Density} = 0.002382 \text{ lb/ft}^2 \text{sec}^2 \text{ ft} \]
\[ q = \text{Dynamic pressure} = 17.03 \text{ lb/sq ft} \]
\[ V = \text{Velocity} = 6913.2 \text{ ft/sec} \]

\[ \rho = \frac{p}{q} = 81.70 \text{ miles/hour} \]

### TABLE XV
Front Flap Down 20 Degrees  
Rear Flap Down 10 Degrees

<table>
<thead>
<tr>
<th></th>
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<td>C&lt;sub&gt;L&lt;/sub&gt;/C&lt;sub&gt;D&lt;/sub&gt;</td>
<td>C&lt;sub&gt;M&lt;/sub&gt;</td>
<td>C&lt;sub&gt;P&lt;/sub&gt;</td>
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*Uncorrected for tunnel wall effect

\[ p = \text{Density} = 0.002382 \text{ lb/ft}^2 \text{sec}^2 \text{ ft} \]
\[ q = \text{Dynamic pressure} = 17.067 \text{ lb/sq ft} \]
\[ V = \text{Velocity} = 6918.2 \text{ ft/sec} \]
\[ \rho = \frac{p}{q} = 81.65 \text{ miles/hour} \]
### TABLE XVI

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<td>C_L/C_D</td>
<td>C_M</td>
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*Uncorrected for tunnel wall effect

\[
p = \text{Density} = .002365 \text{ lb.-sec.}^2/\text{ft.}^{-4}
\]
\[
q = \text{Dynamic pressure} = 17.124 \text{ lb/sq.ft.}
\]
\[
V = \text{Velocity} = .6818/\sqrt{p} = 82.05 \text{ miles/hour}
\]

### TABLE XVII

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<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
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* Uncorrected for tunnel wall effect

\[
p = \text{Density} = .002319 \text{ lb.-sec.}^2/\text{ft.}^{-4}
\]
\[
q = \text{Dynamic pressure} = 15.817 \text{ lb/sq.ft.}
\]
\[
V = \text{Velocity} = .6818/\sqrt{p} = 82.11 \text{ miles/hour}
\]
### TABLE XVIII
Front Flap Down 10 Degrees
Rear Flap Down 15 Degrees

<table>
<thead>
<tr>
<th></th>
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<td>C_D</td>
<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
<td>C_P</td>
<td>a_1</td>
<td>C_D_1</td>
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<td>0.920</td>
<td>-3.44</td>
<td>1.100</td>
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<td>0.1013</td>
<td>0.2392</td>
<td>2.96</td>
<td>-0.153</td>
<td>0.529</td>
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<td>0.505</td>
<td>6.89</td>
<td>1.297</td>
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<td>0.504</td>
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<td>0.481</td>
<td>23.69</td>
<td>5.577</td>
</tr>
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</table>

*Uncorrected for tunnel wall effect

\[ p = \text{Density} = 0.002366 \text{ lb} \cdot \text{sec}^2 \cdot \text{ft} \]
\[ q = \text{Dynamic pressure} = 17.129 \text{ lb/sq ft} \]
\[ V = \text{Velocity} = 6818/22 = 31.57 \text{ miles/hour} \]

\[ \sqrt{\frac{q}{p}} \]

### TABLE XIX
Front Flap Down 15 Degrees
Rear Flap Down 15 Degrees

<table>
<thead>
<tr>
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</thead>
<tbody>
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<td>a</td>
<td>C_D</td>
<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
<td>C_P</td>
<td>a_1</td>
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<td>0.505</td>
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<td>0.492</td>
<td>23.69</td>
<td>0.5199</td>
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</table>

*Uncorrected for tunnel wall effect

\[ p = \text{Density} = 0.002405 \text{ lb} \cdot \text{sec}^2 \cdot \text{ft} \]
\[ q = \text{Dynamic pressure} = 17.212 \text{ lb/sq ft} \]
\[ V = \text{Velocity} = 6818/22 = 31.57 \text{ miles/hour} \]
\[ \sqrt{\frac{q}{p}} \]
**TABLE XX**

Front Flap Down 20 Degrees  
Rear Flap Down 15 Degrees

<table>
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<td>C_D</td>
<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
<td>C_P</td>
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<td>*C_D_1</td>
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<td>13.89</td>
<td>0.2860</td>
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<tr>
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<td>0.9441</td>
<td>2.45</td>
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<td>0.506</td>
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*Uncorrected for tunnel wall effect*

\[ p = \text{Density} = 0.002409 \text{ lb-sec}^2\text{-ft}^{-4} \]
\[ q = \text{Dynamic pressure} = 17.061 \text{ lb/sq ft} \]
\[ V = \text{Velocity} = 681.8 \left( \frac{2g}{p} \right)^{1/2} = 81.15 \text{ miles/hour} \]

**TABLE XXI**

Front Flap Down 0 Degrees  
Rear Flap Down 20 Degrees

<table>
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<td>C_D</td>
<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
<td>C_P</td>
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<td>*C_D_1</td>
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<td>0.0</td>
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<td>0.1135</td>
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<td>0.611</td>
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<td>0.1359</td>
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<tr>
<td>17.60</td>
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<td>0.520</td>
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*Uncorrected for tunnel wall effect*

\[ p = \text{Density} = 0.002363 \text{ lb-sec}^2\text{-ft}^{-4} \]
\[ q = \text{Dynamic pressure} = 17.181 \text{ lb/sq ft} \]
\[ V = \text{Velocity} = 681.8 \left( \frac{2g}{p} \right)^{1/2} = 82.13 \text{ miles/hour} \]
### TABLE XXII
Front Flap Down 5 Degrees
Rear Flap Down 20 Degrees

<table>
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<td>3.974</td>
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<td>1.154</td>
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<td>1.081</td>
<td>0.2510</td>
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<td>-0.353</td>
<td>6.10</td>
<td>0</td>
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<td>-0.357</td>
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<td>2.083</td>
<td>0.5655</td>
<td>4.15</td>
<td>-0.451</td>
<td>5.06</td>
<td>13.93</td>
<td>2.065</td>
</tr>
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<td>4.92</td>
<td>28.69</td>
<td>5.526</td>
</tr>
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</table>

*Uncorrected for tunnel wall effect

\[ p = \text{Density} = 0.002346 \text{ lb.-sec.-ft.} \]
\[ q = \text{Dynamic pressure} = 16.048 \text{ lb/sq.ft.} \]
\[ V = \text{Velocity} = \frac{6818}{2} = 22.05 \text{ miles/hour} \]

### TABLE XXIII
Front Flap Down 10 Degrees
Rear Flap Down 20 Degrees

<table>
<thead>
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<tr>
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* Uncorrected for tunnel wall effect

\[ p = \text{Density} = 0.002346 \text{ lb.-sec.-ft.} \]
\[ q = \text{Dynamic pressure} = 17.077 \text{ lb/sq.ft.} \]
\[ V = \text{Velocity} = \frac{6818}{2} = 22.05 \text{ miles/hour} \]
### TABLE XXIV

Front Flap Down 15 Degrees  
Rear Flap Down 20 Degrees

<table>
<thead>
<tr>
<th></th>
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<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
<td>C_P</td>
<td>*a_1</td>
<td>*C_D_1</td>
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<td>0.495</td>
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<td>5.369</td>
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</tbody>
</table>

* Uncorrected for tunnel wall effect

\[ p = \text{Density} = 0.002457 \text{ lb} \cdot \text{sec}^2 \cdot \text{ft}^{-3} \]
\[ q = \text{Dynamic pressure} = 17.467 \text{ lb/sq} \cdot \text{ft} \]
\[ V = \text{Velocity} = 8818/17.467 = 51.30 \text{ miles/hour} \]

### TABLE XXV

Front Flap Down 20 Degrees  
Rear Flap Down 20 Degrees

<table>
<thead>
<tr>
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<th></th>
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<tbody>
<tr>
<td>a</td>
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<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
<td>C_P</td>
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<td>*C_D_1</td>
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<td>0.611</td>
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</table>

* Uncorrected for tunnel wall effect

\[ p = \text{Density} = 0.002401 \text{ lb} \cdot \text{sec}^2 \cdot \text{ft}^{-3} \]
\[ q = \text{Dynamic pressure} = 17.057 \text{ lb/sq} \cdot \text{ft} \]
\[ V = \text{Velocity} = 6818/17.057 = 51.54 \text{ miles/hour} \]
**TABLE XXVI**

Front Flap Down 20 Degrees

Rear Flap Down 20 Degrees

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
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</thead>
<tbody>
<tr>
<td>a</td>
<td>C_D</td>
<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
<td>C_P</td>
<td>*a_1</td>
<td>C_D1</td>
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<td>1.393</td>
<td>4.715</td>
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<td>3.305</td>
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</tr>
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<td>0.659</td>
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</table>

* Uncorrected for tunnel wall effect

\[ \text{p = Density} = 0.002401 \text{ lb} \cdot \text{sec}^2 \cdot \text{ft}^{-2} \]

\[ \text{q = Dynamic pressure} = 17.067 \text{ lb/sq ft.} \]

\[ \text{V = Velocity} = \frac{6318 \text{ ft}}{22} = 81.34 \text{ miles/hour} \]

**TABLE XXVII**

Clark "Y" Wing Section

24" x 4"

<table>
<thead>
<tr>
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<tbody>
<tr>
<td>a</td>
<td>C_D</td>
<td>C_L</td>
<td>C_L/C_D</td>
<td>C_M</td>
<td>C_P</td>
<td>*a_1</td>
<td>C_D1</td>
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<td>0.396</td>
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<td>1.717</td>
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\[ \text{p = Density} = 0.002366 \text{ lb} \cdot \text{sec}^2 \cdot \text{ft}^{-2} \]

\[ \text{q = Dynamic pressure} = 17.254 \text{ lb/sq ft.} \]

\[ \text{V = Velocity} = \frac{6318 \text{ ft}}{22} = 82.54 \text{ miles/hour} \]
Appendix

Methods used in calculating the different coefficients and locating the action line of the resultant force for the pitching moment vectors.

Formula used:*  
Lift Coefficient \( C_L = \frac{L}{q S} \)

Drag Coefficient \( C_D = \frac{D}{q S} \)

Pitching Moment Coefficient \( C_M = \frac{M_A}{q S \cdot \text{sec}} \)

where:

- \( S \) = Area of wings = 4194 sq. ft.
- \( c = 6'' \)

The velocity of the flow of air may be given by the relation;

\[ v = \sqrt{\frac{2gh_a}{\rho}} \]  

(1) = velocity in feet per sec.

where:
- \( g \) = acceleration of gravity = 32.174 ft. per sec.²
- \( h_a \) = Dynamic head in feet of air.

In these experiments the dynamic head was measured in inches of water.

Then \[ h_a = \frac{h_w g_w}{12^4 a} \]  

(2)

Where:
- \( h_w \) = dynamic head in inches of water.

\[ g_w \] = specific weight of water = 62.4 lb. per cu. ft.

\[ g_a \] = specific weight of air under operating conditions.

Since \[ q \] = The dynamic (or impact) pressure \( = \frac{1}{2} \rho v^2 \)  

(3)

Where:
- \( p \) = density of air (mass per cu. ft.) under operating conditions = \[ \frac{g_a}{g} \] (lb. - sec.² - ft. -⁴)

Substituting the values of equations (1), (2), and (4) in equation (3) we have:

* For meaning of terms used see page 11.
\[ q = \frac{2g h_w x_w}{12 x_a} \quad x = \frac{h_w x_w}{x_a} = \frac{62.4 h_w}{12} = 5.2 h_w \quad \text{(5)} \]

The true air velocity is therefore
\[ V = \frac{2q}{p} = \frac{10.4 h_w}{p} \]

From the laws of perfect gases,
\[ \frac{P_1 V_1}{T_1} = \frac{P_2 V_2}{T_2} \]

\[ V = \frac{1}{r} \quad \text{and} \quad r = pg \]

Therefore:
\[ \frac{P_1}{T_1} = \frac{P_2}{T_2} \quad \text{and} \quad \frac{P_2}{T_2} \]

Where: \( P_1 \) = absolute pressure at standard conditions.
\( P_2 \) = absolute pressure under operating conditions, in inches of mercury.
\( T_1 \) = absolute temperature at standard condition
\( T_2 \) = absolute temperature under operating condition in degrees F.
\( p_1 \) = density of air (mass per cu. ft.) at standard conditions =
\( \frac{0.002378 \text{ lb. sec.}^2 \cdot \text{ft.}^{-4}}{29.921 x T_2} \)
\( p_2 \) = density of air under operating conditions.
\[ p_2 = \frac{P_2 518.4 x 0.002378}{29.921 x T_2} = 0.04120 \quad \text{(6)} \]

The standard values are taken from the N.A. C.A. Technical report No. 218 by Walter S. Diehl (page 5).

Center of pressure coefficient \( C_p = \frac{C_p}{C_s} \)

Referring to Figure 22 it is seen that the center of pressure distance, \((C.P.)\) from the leading edge is the distance from "A" to the
\[ M_A = 12.5 L_R \cos \gamma = L \times d \]
\[ d = \frac{M_A}{L} \]
\[ a = d - (2.5 \cos \alpha - 2.125 \sin \alpha) \]
\[ b = (2.125 \cos \alpha + 2.5 \sin \alpha) \]

Moment about C.G. = \[ D \times b - L \times a = D \times b - \left[M_A - L (2.5 \cos \alpha - 2.125 \sin \alpha)\right] \]

**Figure 22.**

**Illustrating Method of Obtaining Pitching Moments About C.G.**
point where the action line of the resultant force intersects the rod \( A·B \). If the point of application of the resultant is moved to this intersection, and the lift and drag forces resolved into components along the rod and perpendicular to it, the force then along the rod will be \( (L \sin \alpha + D \cos \alpha) \), which has no moment about \( A \) since it passes through the point. The force perpendicular to the rod will be \( (L \cos \alpha + D \sin \alpha) \), where \( \alpha \) is the angle of attack in each case.

The moment about \( A = M_A = 12·1/2 L_R \cos \gamma = (L \cos \alpha + D \sin \alpha) (C.P.) \)

\[ M_A = C_{MA} q S \cos \alpha, \quad L = C_L q S \quad \text{and} \quad D = C_D q S \]

Then:

\[ C_{MA} q S \cos \alpha = (C.P.) q S (C_L \cos \alpha + C_D \sin \alpha) \]

but,

\[ (C_L \cos \alpha + C_D \sin \alpha) = C_N \]

therefore,

\[ C_{MA} = (C.P.) C_N \]

and

\[ \frac{C.P.}{C_N} = C_{MA} = C_p \]

Induced drag correction \( \Delta C_D = \frac{C^2}{2 \pi D^2} \)

\( D \) = Diameter wind tunnel = 5 ft.

\( S \) = Area of wings = 4.194 sq. ft. for tandem wing model and 2/3 sq. ft. for the single wing Clark Y airfoil.

\( \Delta C_{D_1} = C_L^2 S = \frac{C_L^2 S}{2 \pi D^2} = \frac{C_L^2 S}{\pi D^2} = 0.006366 \quad \frac{C_L^2 S}{2 \pi D^2} \)  

For Tandem wing model.

\( \Delta C_{D_1} = 0.006366 \times 4.194 \times C_L^2 S = 0.0067 \)

For Clark Y airfoil.

\( \Delta C_{D_1} = 0.006366 \times 2/3 \times C_L^2 = 0.006366 \times \frac{2}{3} \)  

Induced angle of attack correction \( = \frac{57.3 C_L S}{2 \pi D^2} \)

\( \Delta \alpha = \frac{57.3 C_L S}{157.03} = \frac{35477 S C_L}{157.03} \) in degrees.
For the tandem wing model,

\[ \cdot36473 \times 0.4194 = 0.153 C_L \text{ in degrees} \]

For the Clark Y airfoil,

\[ \cdot36477 \times 2/3 = 0.2432 C_L \text{ in degrees}. \]

These corrections are to be added algebraically to the drag coefficient and angle of attack respectively.

In determining the moment and center of pressure coefficients for the Clark Y airfoil, the wing was mounted as shown below.

\underline{METHOD OF MOUNTING CLARK Y AIRFOIL}
For this mounting

(1) \( M_A = \frac{12-1/2}{L} \cos \alpha \cdot (C_{p*} + a)(L \cos \alpha + D \sin \alpha) \)

(2) \((C_{p*}) (L \cos \alpha + D \sin \alpha) = M_{C} = \text{Moment about the leading edge of the wing.}\)

\[
C_{p*} + a = \frac{M_A}{L \cos \alpha + D \sin \alpha} = \frac{C_{M_A} C}{C_N}
\]

\[
\frac{C_{p*}}{C} = \frac{C_{M_A}}{C_N} - \frac{a}{C} = C_p \text{ measured from the leading edge of the wing.}
\]

\[
M_{C} = M_{A} - a(L \cos \alpha + D \sin \alpha)
\]

\[
\frac{C_{M_{C}}}{C} = \frac{C_{M_A} - aC_N}{C} \]

In this particular set-up, \( a = 2 \) inches and \( c = 4 \) inches

which gives the following relations for these values:

\[
C_p = \frac{C_{M_A}}{C_N} - .5
\]

\[
C_{M_{C}} = \frac{C_{M_A}}{C_N} - .5 \frac{C}{C_N}
\]

From (2) we have also the relation:

\[
(C_{p*}) \frac{C}{C_N} = C_{M_{C}}
\]

Therefore, \( C_{M_{C}} = C_{p} \frac{C}{C_N} \)

With the above equations and relations it is possible to determine values for full scale conditions for any particular angle of attack.
Bibliography

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Airplane Design by Edward P. Warner.
Measuring the Flow of Gases, by Thomas G. Estep