INVESTIGATION IN THE LANGLEY 19-FOOT PRESSURE TUNNEL OF TWO WINGS OF NACA 65-210 AND 64-210 AIRFOIL SECTIONS WITH VARIOUS TYPE FLAPS

By JAMES C. SIVELLS and STANLEY H. SPOONER
AERONAUTIC SYMBOLS

1. FUNDAMENTAL AND DERIVED UNITS

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Metric</th>
<th>English</th>
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<tr>
<td></td>
<td>Unit</td>
<td>Abbreviation</td>
</tr>
<tr>
<td>Length</td>
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<tr>
<td>Time</td>
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<td>second</td>
</tr>
<tr>
<td>Force</td>
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<tr>
<td>Power</td>
<td>P</td>
<td>horsepower (metric)</td>
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<tr>
<td>Speed</td>
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<tr>
<td></td>
<td>(meters per second)</td>
<td>mps</td>
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</table>

2. GENERAL SYMBOLS

- Weight = mg
- Standard acceleration of gravity = 9.80665 m/s² or 32.1740 ft/sec²
- Mass = W
- Moment of inertia = mk² (Indicate axis of radius of gyration k by proper subscript.)
- Coefficient of viscosity

3. AERODYNAMIC SYMBOLS

- Area
- Area of wing
- Gap
- Span
- Chord
- Aspect ratio, k²
- True air speed
- Dynamic pressure, \( \frac{1}{2} \rho V^2 \)
- Lift, absolute coefficient \( C_L = \frac{L}{qS} \)
- Drag, absolute coefficient \( C_D = \frac{D}{qS} \)
- Profile drag, absolute coefficient \( C_{D_p} = \frac{D_p}{qS} \)
- Induced drag, absolute coefficient \( C_{D_i} = \frac{D_i}{qS} \)
- Parasite drag, absolute coefficient \( C_{D_p} = \frac{D_p}{qS} \)
- Cross-wind force, absolute coefficient \( C_c = \frac{C}{qS} \)

- Angle of setting of wings (relative to thrust line)
- Angle of stabilizer setting (relative to thrust line)
- Resultant moment
- Resultant angular velocity
- Reynolds number, \( \frac{\rho V l}{\mu} \) where \( l \) is a linear dimension (e.g., for an airfoil of 1.0 ft chord, 100 mph, standard pressure at 15° C, the corresponding Reynolds number is 935,400; or for an airfoil of 1.0 m chord, 100 mps, the corresponding Reynolds number is 6,865,000)
- Angle of attack
- Angle of downwash
- Angle of attack, infinite aspect ratio
- Angle of attack, induced
- Angle of attack, absolute (measured from zero-lift position)
- Flight-path angle
REPORT 942

INVESTIGATION IN THE LANGLEY 19-FOOT PRESSURE TUNNEL OF TWO WINGS OF NACA 65-210 AND 64-210 AIRFOIL SECTIONS WITH VARIOUS TYPE FLAPS

By JAMES C. SIVELLS and STANLEY H. SPOONER

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

Headquarters, 1724 F Street N.W., Washington 25, D.C.

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INVESTIGATION IN THE LANGLEY 19-FOOT PRESSURE TUNNEL OF TWO WINGS OF NACA 65-210 AND 64-210 AIRFOIL SECTIONS WITH VARIOUS TYPE FLAPS

By James C. Sivells and Stanley H. Spooner

SUMMARY

An investigation has been conducted in the Langley 19-foot pressure tunnel to determine the maximum lift and stalling characteristics of two thin wings equipped with several types of flaps. Split, single slotted, and double slotted flaps were tested on one wing which had NACA 65-210 airfoil sections and split and double slotted flaps were tested on the other, which had NACA 64-210 airfoil sections. Both wings had zero sweep, an aspect ratio of 9, and a taper ratio of 0.4.

At a Reynolds number of 4,500,000 each type of flap increased the maximum lift coefficients of the two wings by increments which were approximately proportional to the flap neutral values of 1.21 and 1.35 for the NACA 65-210 wing and the NACA 64-210 wing, respectively. The values of maximum lift coefficient for the wings with full-span double slotted flaps were 2.48 and 2.76, which values represent increments of 105 percent of the flap neutral values. The addition of a representative fuselage or leading-edge roughness was more detrimental to the NACA 64-210 wing, but its values of maximum lift coefficient were still consistently higher than those of the NACA 65-210 wing. The values of maximum lift coefficient increased with increasing Reynolds numbers up to a value of 4,500,000.

Above this value, the test Mach number was high enough so that the effects of compressibility appeared to cause the values of maximum lift coefficient to increase less rapidly or to decrease with increasing Reynolds numbers.

The stall of the NACA 64-210 wing was somewhat more abrupt but slightly farther inboard than that of the NACA 65-210 wing. The pattern of the stall was approximately the same for all flap configurations with or without leading-edge roughness. The main effect of roughness was to make the stall progression more gradual. The fuselage, however, caused the stall to begin inboard near the wing-fuselage junction.

INTRODUCTION

The wing sections of an airplane capable of flying at high subsonic speeds must be relatively thin in order to delay the onset of the effects of compressibility. These thin sections, however, cannot normally develop as high values of maximum lift coefficient as thicker sections used on slower airplanes. More powerful high lift flaps must therefore be used on high-speed airplanes to obtain landing characteristics approaching those of lower-speed, but otherwise comparable, airplanes. In order to develop high lift flaps suitable for thin airfoils, an investigation was conducted in the Langley two-dimensional low-turbulence tunnels. (See references 1 and 2.) The most promising results of this investigation were incorporated in the design of two thin wings, the three-dimensional characteristics of which were investigated in the Langley 19-foot pressure tunnel.

One of these wings had NACA 65-210 airfoil sections and was equipped with split, single slotted, and double slotted flaps. The other wing had NACA 64-210 airfoil sections and was equipped with split and double slotted flaps. The plan form of both wings was typical of a long-range airplane in that the aspect ratio was 9 and the taper ratio was 0.4. Presently here in are the results of tests made at relatively high Reynolds numbers to determine the maximum lift and stalling characteristics of these two wings with partial-span and full-span flaps both with and without a representative fuselage and leading-edge roughness.

COEFFICIENTS AND SYMBOLS

The coefficients and symbols used herein are defined as follows:

\[ C_L \] lift coefficient \((L/qS)\)
\[ C_D \] drag coefficient \((D/qS)\)
\[ C_m \] pitching-moment coefficient \((M/qSC)\)
\[ C_{L,\text{trim}} = C_L + C_m / 3 \] (Tail length=3\(c\))
\[ C_{L,\text{max}} \] maximum lift coefficient
\[ \Delta C_{L,\text{max}} \] increment in \(C_{L,\text{max}}\) due to flaps

where

\[ L \] lift
\[ D \] drag
\[ M \] pitching moment about 0.25\(c\)
\[ q \] dynamic pressure of free stream \(\left(\frac{1}{2} \rho V^2\right)\)
\[ S \] wing area \((24.94 \text{ ft}^2)\)
\[ \bar{c} \] mean aerodynamic chord \((1.769 \text{ ft})\)
\[ \rho \] mass density of air
\[ V \] airspeed
\[ V_r \] vertical velocity in glide
\[ c \] local wing chord
\[ b \] wing span \((15 \text{ ft})\)
\[ y \] spanwise coordinate

and

\[ \alpha \] corrected angle of attack of root chord
\[ R \] Reynolds number \((\rho V \bar{c} / \mu)\)
\[ M \] Mach number \((V/a)\)
\[ \mu \] coefficient of viscosity
\[ a \] sonic velocity
MODELS AND TESTS

The two wings were constructed of solid steel and were geometrically similar except that one was contoured to NACA 65-210 airfoil sections and the other to NACA 64-210 airfoil sections. The taper ratio was 0.4 and the aspect ratio was 9. The sweep and dihedral at the 0.25-chord line were 0° and 3°, respectively. Both wings were uniformly twisted about the 0.25-chord line to produce 2° washout. A mahogany fuselage was attached to the wings for some of the tests. The wing and fuselage mounted in the Langley 19-foot pressure tunnel are shown in figure 1, and the general dimensions of the models are given in figure 2.

The wing with NACA 65-210 airfoil sections was tested with partial-span and full-span split, single slotted, and double slotted flaps. The wing with NACA 64-210 airfoil sections was tested with partial-span and full-span split and double slotted flaps. The split and single slotted flaps were, respectively, 20 and 25 percent of the local wing chord. The double slotted flap was comprised of a 7.5-percent-chord vane and a 25-percent-chord main flap. For the NACA 65-210 wing, the single slotted flap was used as the main flap of the double slotted flap. The same flap ordinates were used for both wings. The ordinates for the airfoil sections and flaps are given in tables I to V. A finite trailing-edge thickness of 1 percent of the maximum thickness was arbitrarily set for these wings. It was not possible in the construction of the wings to make the flap wells deep enough to allow the double slotted flaps to be retracted. Unpublished two-dimensional data indicated that the difference in depth and shape of the double-slotted-flap well and that of the single-slotted-flap well would not affect the test results inasmuch as the flap-well ordinates in the vicinity of the deflected vane were approximately the same. For these wings, therefore, the flap wells for single slotted flaps were constructed according to the ordinates of table VI and were not changed for the double-slotted-flap tests.

The split, single slotted, and double slotted flaps were deflected 60°, 45°, and 50°, respectively, for these tests. The flap positions used are shown in figure 3 and were determined to be optimum from preliminary two-dimensional tests. These positions do not completely conform with the final optimum values given in references 1 and 2. The partial-span flaps extended to 60 percent of the semispan and the full-span flaps, to 97.5 percent. For most of the tests of the wings without the fuselage, the flaps extended inboard to the plane of symmetry. A few tests of the NACA 65-210 wing with the fuselage off were made in which the flaps extended inboard only as far as they did when the fuselage was attached.

![Figure 1](image1.jpg)

**Figure 1.** Wing with fuselage mounted in Langley 19-foot pressure tunnel. Partial-span double-slotted-flap configuration.

![Figure 2](image2.jpg)

**Figure 2.** Wing and fuselage for tests in Langley 19-foot pressure tunnel. Root chord line at 0.25c is 2.625 inches above fuselage center line; wing area, 24.94 sq. ft.; aspect ratio, 9.02; washout, 2°; taper ratio, 0.4. (All dimensions are in inches.)
The models used for the tests reported herein were found to be smooth and fair and conformed with the true airfoil contours to within 0.003 inch over the forward 30 percent of the wing and within 0.008 inch over the rearward areas. The tests were conducted with the air in the tunnel compressed to approximately 34 pounds per square inch absolute pressure. The majority of the tests were made at a dynamic pressure of 85 pounds per square foot, corresponding to a Reynolds number of approximately 4,400,000 and a Mach number of about 0.17. Scale-effect tests were made over a range of Reynolds number from 3,200,000 to 6,400,000 corresponding to a range of Mach number from 0.12 to 0.24.

The aerodynamic forces and moments were measured by a simultaneously recording, six-component balance system. The stalling characteristics were determined from observations of the behavior of tufts attached to the upper surface of the model behind the 0.30-chord line. In order to determine the effect of leading-edge roughness, tests were made with No. 60 carborundum grains applied to the nose of each wing over a surface length of 0.08 chord measured from the leading edge on both surfaces.

**RESULTS AND DISCUSSION**

All data have been reduced to standard nondimensional coefficients. Corrections have been applied to the force and moment data to account for the tare and interference effects of the model support system. Stream-angle and jet-boundary corrections have been applied to the angle of attack and to the drag coefficients.

The lift, drag, and pitching-moment coefficients of the two wings are shown in figures 4 to 13 for a Reynolds number of 4,400,000. A comparison of the various flap configurations is made in figure 14 for the wing-fuselage combination. The effects of Reynolds number on maximum lift coefficient are given in figures 15, 16, and 17. The stalling characteristics are given in figures 18 to 29. The values of the trimmed and untrimmed maximum lift coefficients of the various flap configurations are summarized in table VII.

Some inconsistency can be noted in the values of maximum lift coefficient for the various configurations. This inconsistency appears to be a characteristic of these thin wings. Preliminary tests of these wings showed that very small errors in airfoil contour, particularly around the leading edge, could cause large changes in the stalling angle of attack and the resulting value of maximum lift coefficient. For the tests described herein, the airfoil contours were held to very close tolerances and extreme care was taken during the course of the smooth-wing tests to keep the wings in an nearly perfect condition as possible. In spite of all precautions taken, some inconsistency still appears in the results and, therefore, some of the effects of model configuration and Reynolds number may be somewhat obscured.
FLAP EFFECTIVENESS

If the values of maximum lift coefficient of the wings with flaps are expressed in percent of the flap neutral values, the flap effectiveness for both wings was practically the same at a Reynolds number of 4,400,000. Inasmuch as the flap neutral value for the NACA 64–210 wing was 1.35 as compared with 1.21 for the NACA 65–210 wing, the flap extended values for the NACA 64–210 wing were consistently higher. The increments in maximum lift coefficient due to flaps for the smooth-wing condition and for a Reynolds number of 4,400,000 are as follows:

<table>
<thead>
<tr>
<th>Flap type</th>
<th>Flap span</th>
<th>$\Delta C_{l, max}$</th>
<th>$\Delta C_{l, max}$ in percent of flap neutral value</th>
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<tr>
<td>Split</td>
<td>Partial</td>
<td>0.42</td>
<td>0.52</td>
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<td>Full</td>
<td>0.53</td>
<td>0.50</td>
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<tr>
<td>Slotted</td>
<td>Partial</td>
<td>0.66</td>
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<tr>
<td></td>
<td>Full</td>
<td>0.89</td>
<td>1.07</td>
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<tr>
<td>Double slotted</td>
<td>Partial</td>
<td>1.27</td>
<td>1.41</td>
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</table>
These increments are of the order of magnitude that would be expected from the two-dimensional data of references 1 and 2 although the flap positions were not quite the same.

The single slotted and double slotted flaps had the effect of producing an unstable break in the pitching-moment curves at $C_{l_{\text{max}}}$: This effect is a section characteristic since it was also noted in reference 2. Inasmuch as the stall of these wings tends to begin inboard with the fuselage in place, as is shown subsequently, the decrease in downwash accompanying the stall produces a positive increment of lift on the tail of a complete airplane and thereby tends to compensate for the unstable break.

In order to compare the effects of the various types of flaps on the landing characteristics of a typical airplane, contours of constant gliding speed and constant vertical (sinking) speed are superimposed on the fuselage-on drag polars in figure 14. In this figure the lift of the tail necessary to trim the airplane is taken into account in the lift coefficient presented ($C_{l_{\text{trim}}}$). For this purpose a tail length of three times the mean aerodynamic chord was assumed and the center of gravity of the airplane was assumed to be located at the quarter-chord point of the mean aerodynamic chord. For the constant-speed contours a wing loading of 60 pounds per square foot was assumed and standard sea-level conditions were used. Obviously, the drag of nacelles, landing gear, tail, and protuberances are not shown on this figure nor are
the effects of power. The relative effects of the types of flaps and the flap span, however, are readily shown. The single-slotted-flap configuration has the lowest sinking speed of any of the flapped configurations but a higher gliding speed than the double-slotted-flap configuration. Increasing the flap span from partial to full span decreases the sinking speed for the single-slotted-flap and double-slotted-flap configurations because of the lower induced drag but increases the sinking speed for the split-flap configuration because of the high profile drag of split flaps.

![Graph showing the effects of flap configuration on lift and drag coefficients.](image)

**Figure 6.** Aerodynamic characteristics of NACA 65-210 wing with and without double slotted flaps and fuselage. $R=4,000,000; M=0.17$.

**EFFECT OF FUSELAGE**

The reduction in maximum lift coefficient caused by the fuselage was approximately 0.1 for the NACA 65–210 wing and varied from 0.1 to 0.3 for the NACA 64–210 wing. The values of maximum lift coefficient, however, were still higher for the NACA 64–210 wing than for the NACA 65–210 wing. Since the tests were conducted with no fillets at the wing-fuselage junction, properly designed fillets might have minimized the loss in maximum lift.
The results of the tests of the NACA 65–210 wing with the flaps removed from that part of the wing normally occupied by the fuselage are shown in figures 4 to 6. The data in the linear lift-curve range indicate that some of the lift due to the single slotted and double slotted flaps was carried across the fuselage, whereas practically none of the lift due to the split flaps was carried across.

For all configurations the fuselage caused a destabilizing effect on the pitching moment equal to a forward shift of the aerodynamic center of about 5 percent of the mean aerodynamic chord.

**Figure 7.** Aerodynamic characteristics of NACA 65–210 wing with and without split flaps and leading-edge roughness. $Re = 4,400,000, M = 0.17$.

**EFFECT OF LEADING-EDGE ROUGHNESS**

Leading-edge roughness caused a rounding of the lift-curve peaks and a reduction in the maximum lift coefficients of both wings with and without flaps. The reduction usually amounted to about 0.2 for the NACA 65–210 wing and about 0.3 for the NACA 64–210 wing. As was true for the fuselage configuration, the maximum lift coefficients of the NACA 64–210 wing were higher than those of the NACA 65–210 wing even though the effect of roughness on the NACA 64–210 wing was greater.
At low angles of attack, the addition of leading-edge roughness usually decreased the lift coefficient slightly. For the NACA 64-210 wing with double slotted flaps (fig. 13), the lift coefficient was increased by roughness and the pitching-moment coefficient was increased negatively. An inspection of the stalling characteristics (fig. 29) indicates that this effect may be due in part to the fact that the flap was uninstalled for this condition but had some small stalled areas when the wing was smooth. Another contributing factor to this effect may have been that the support tare and interference corrections for the smooth wing were used to correct both smooth-wing and rough-wing data.
INVESTIGATION OF TWO WINGS OF NACA 65–210 AND 64–210 AIRFOIL SECTIONS WITH VARIOUS TYPE FLAPS

FIGURE 9.—Aerodynamic characteristics of NACA 65–210 wing with and without double slotted flaps and leading-edge roughness. $R = 4,000,000; M = 0.17$. 
FIGURE 10.—Aerodynamic characteristics of NACA 64-210 wing with and without split flaps and fuselage. \( R = 4,000,000; M = 0.17 \).
INVESTIGATION OF TWO WINGS OF NACA 65-210 AND 64-210 AIRFOIL SECTIONS WITH VARIOUS TYPE FLAPS

Figure 11.—Aerodynamic characteristics of NACA 64-210 wing with and without double slotted flaps and fuselage. $R = 4,000,000; M = 0.17.$
Figure 12.—Aerodynamic characteristics of NACA 64-210 wing with and without split flaps and leading-edge roughness. $R = 4,000,000; M = 0.17$. 
Figure 11. Aerodynamic characteristics of NACA 64-210 wing with and without double slotted flaps and leading-edge roughness. $R = 4,400,000$; $M = 0.17$. 
Figure 14.—Comparison of the effects of various flap configurations on the gliding characteristics of an airplane with a wing loading of 60 pounds per square foot; standard sea-level conditions.

(a) NACA 65-210 wing without fuselage.
(b) NACA 64-210 wing with fuselage.
INVESTIGATION OF TWO WINGS OF NACA 65-210 AND 64-210 AIRFOIL SECTIONS WITH VARIOUS TYPE FLAPS

SCALE EFFECT

The variation of maximum lift coefficient with Reynolds number is shown in figures 15, 16, and 17 for the various flap configurations. Although the data are not completely consistent, they show the same general trends which were indicated by the two-dimensional tests (references 1 and 2) if some allowance is made at the highest Reynolds numbers (Mach numbers about 0.2) for the effects of compressibility which are probably similar to those described in reference 3. In general, the maximum lift coefficients of both the NACA 65-210 and NACA 64-210 wings increased with increasing Reynolds number for Reynolds numbers below 4,400,000. Above this Reynolds number, the maximum lift coefficients increased less rapidly or decreased because of the effects of compressibility present for the three-dimensional tests.

FIGURE 16.—Scale effect on NACA 65-210 wing with single slotted flaps with and without fuselage.

FIGURE 17.—Scale effect on NACA 65-210 and 64-210 wings with double slotted flaps with and without fuselage.
FIGURE 18.—Stalling characteristics of NACA 65-210 wing; flaps neutral. $R = 4,400,000; M = 0.17.$

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FIGURE 19.—Stalling characteristics of NACA 65-210 wing; partial-span split flaps. $R = 4,400,000; M = 0.17.$
INVESTIGATION OF TWO WINGS OF NACA 65–210 AND 64–210 AIRFOIL SECTIONS WITH VARIOUS TYPE FLAPS

Figure 20.—Stalling characteristics of NACA 65–210 wing; full-span split flaps. $R = 4,400,000; M = 0.17$.

Figure 21.—Stalling characteristics of NACA 65–210 wing; partial-span single slotted flaps. $R = 4,400,000; M = 0.17$. 
FIGURE 22.—Stalling characteristics of NACA 65-210 wing; full-span single slotted flaps. \( R = 4,800,000; M = 0.17. \)

FIGURE 23.—Stalling characteristics of NACA 65-210 wing; partial-span double slotted flaps. \( R = 4,400,000; M = 0.17. \)
INVESTIGATION OF TWO WINGS OF NACA 65-210 AND 64-210 AIRFOIL SECTIONS WITH VARIOUS TYPE FLAPS

FIGURE 24.—Stalling characteristics of NACA 65-210 wing; full-span double slotted flaps. $R = 4,400,000; M = 0.17$.

FIGURE 25.—Stalling characteristics of NACA 64-210 wing; flaps neutral. $R = 4,400,000; M = 0.17$. 
FIGURE 26.—Stalling characteristics of NACA 64-210 wing; partial-span split flaps. $R = 4,400,000$; $M = 0.17$.

FIGURE 27.—Stalling characteristics of NACA 64-210 wing; full-span split flaps. $R = 4,400,000$; $M = 0.17$. 
INVESTIGATION OF TWO WI GS OF NACA 65-210 AND 64-210 AIRFOIL SECTIONS WITH VARIOUS TYPE FLAPS

Figure 28.—Stalling characteristics of NACA 64-210 wing; partial-span double slotted flaps. $R = 4,400,000; M = 0.17$.

Figure 29.—Stalling characteristics of NACA 64-210 wing; full-span double slotted flaps. $R = 4,400,000; M = 0.17$. 
STALLING CHARACTERISTICS

The stalling characteristics of the two wings as indicated by the tufts are shown in figures 18 to 29. The initial stall of these thin wings was characterized by an area of separated flow ahead of the 40-percent-chord line, with an area of unseparated flow behind it. An increase in angle of attack caused this area of stalled flow to extend rearward and spanwise in either direction. The unsymmetrical stall noted on many of the figures is typical of the inconsistency of the data near the maximum lift coefficient. On several repeat tests, either side of the wing was likely to stall first.

In general, the stall for the NACA 65-210 wing began between the 50-percent and 75-percent points of the semi-span, whereas for the NACA 64-210 wing the stall began slightly more inboard. The NACA 64-210 wing stalled more abruptly and with greater loss of lift than did the NACA 65-210 wing. However, because the tips remained freer of stalled area, the aileron effectiveness of this wing would probably be better maintained beyond maximum lift than for the NACA 65-210 wing.

The pattern of the stall was little affected by flaps or leading-edge roughness, but the progression of the stall was more gradual with roughness. The fuselage caused a premature stall to start near the wing-fuselage junction. This premature stall might have been eliminated by properly designed fillets, thereby increasing the maximum lift coefficient. The presence of this stall, however, might produce tail buffeting which would warn the pilot of the impending stall and also provide longitudinal stability at the stall for the single-slotted-flap and double-slotted-flap configurations.

CONCLUSIONS

From the results of tests in the Langley 19-foot pressure tunnel of a wing with NACA 65-210 airfoil sections and a wing with NACA 64-210 airfoil sections with several types of flaps, the following conclusions may be drawn:

1. At a Reynolds number of 4,400,000 maximum lift coefficients of 2.48 and 2.76, respectively, were obtained with the NACA 65-210 and 64-210 wings with full-span double slotted flaps. These values are approximately 205 percent of the flap neutral values of 1.21 and 1.35 for the respective wings.

2. Addition of the fuselage or the leading-edge roughness caused reductions of 0.1 to 0.3 in the maximum lift coefficients of the wings. The NACA 64-210 wing was affected to a greater extent than was the NACA 65-210 wing, although the maximum lift coefficients for the NACA 64-210 wing were still higher.

3. Increases in maximum lift coefficient with increases in Reynolds number were obtained at Reynolds numbers below 4,400,000. Above this value, the test Mach number was high enough so that the effects of compressibility appeared to be a contributing factor in causing maximum lift coefficients to increase less rapidly or to decrease with increasing Reynolds number.

4. The stall of the NACA 64-210 wing was somewhat more abrupt but slightly farther inboard than that of the NACA 65-210 wing. The pattern of stall was not appreciably altered by the leading-edge roughness or by the various flap configurations. The fuselage, however, caused the stall to begin inboard near the wing-fuselage junction.

REFERENCES


TABLE I
ORDINATES FOR NACA 65-210 AIRFOIL
[Stations and ordinates given in percent airfoil chord]

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<td>0.600</td>
<td>0.060</td>
<td>0.060</td>
</tr>
<tr>
<td>0.700</td>
<td>0.070</td>
<td>0.070</td>
</tr>
<tr>
<td>0.800</td>
<td>0.080</td>
<td>0.080</td>
</tr>
<tr>
<td>0.900</td>
<td>0.090</td>
<td>0.090</td>
</tr>
<tr>
<td>1.000</td>
<td>0.100</td>
<td>0.100</td>
</tr>
</tbody>
</table>

L. E. radius: 0.867
Sine of radius through L. E.: 0.984.
### Table II

**Ordinates for NACA 64-210 Airfoil**  
[Stations and ordinates given from flap chord line in percent airfoil chord]

<table>
<thead>
<tr>
<th>Station</th>
<th>Ordinate</th>
<th>Lower surface</th>
<th>Ordinate</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>25</td>
<td>0.25</td>
<td>0.78</td>
<td>0.25</td>
</tr>
<tr>
<td>50</td>
<td>1.01</td>
<td>1.52</td>
<td>1.00</td>
</tr>
<tr>
<td>75</td>
<td>1.59</td>
<td>2.17</td>
<td>1.90</td>
</tr>
<tr>
<td>100</td>
<td>2.37</td>
<td>3.76</td>
<td>2.90</td>
</tr>
</tbody>
</table>

L. E. radius: 0.78.  
Slope of radius through L. E.: 0.084.

### Table III

**Flap ordinates for NACA 65-210 Airfoil**  
[Stations and ordinates given from flap chord line in percent airfoil chord]

<table>
<thead>
<tr>
<th>Station</th>
<th>Ordinate</th>
<th>Lower surface</th>
<th>Ordinate</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>25</td>
<td>0.25</td>
<td>0.78</td>
<td>0.25</td>
</tr>
</tbody>
</table>

L. E. radius: 0.800.  
L. E. radius center: 0.240 above flap chord line.

### Table IV

**Flap Ordinates for NACA 64-210 Airfoil**  
[Stations and ordinates given from flap chord line in percent airfoil chord]

<table>
<thead>
<tr>
<th>Station</th>
<th>Ordinate</th>
<th>Lower surface</th>
<th>Ordinate</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>25</td>
<td>0.25</td>
<td>0.78</td>
<td>0.25</td>
</tr>
</tbody>
</table>

### Table V

**Ordinates for 0.675 Chord Vane**  
[Stations and ordinates given from vane chord line in percent airfoil chord]

<table>
<thead>
<tr>
<th>Station</th>
<th>Upper ordinate</th>
<th>Lower ordinate</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
<tr>
<td>25</td>
<td>0.25</td>
<td>0.25</td>
</tr>
</tbody>
</table>

L. E. radius: 1.20 (on chord line).

### Table VI

**Ordinates for Upper Surface of Flap Well**  
[Stations and ordinates given from airfoil chord line in percent airfoil chord]

<table>
<thead>
<tr>
<th>Station</th>
<th>Ordinate NACA 65-210 airfoil</th>
<th>Ordinate NACA 64-210 airfoil</th>
</tr>
</thead>
<tbody>
<tr>
<td>74.75</td>
<td>0.40</td>
<td>0.40</td>
</tr>
<tr>
<td>75.00</td>
<td>0.36</td>
<td>0.35</td>
</tr>
<tr>
<td>75.25</td>
<td>0.24</td>
<td>0.20</td>
</tr>
<tr>
<td>75.50</td>
<td>0.20</td>
<td>0.20</td>
</tr>
<tr>
<td>75.75</td>
<td>0.16</td>
<td>0.14</td>
</tr>
</tbody>
</table>

Ordinates between stations 74.75 and 84.00 connected by straight lines.

### Table VII

**Summary of Results Obtained from Tests in Langley 19-Foot Pressure Tunnel of Two Wings, One Incorporating NACA 65-210 Airfoil Sections, the Other, NACA 64-210 Airfoil Sections. $R = 4,400,000; M = 0.17$**

<table>
<thead>
<tr>
<th>Flap type</th>
<th>Smooth Wing</th>
<th>Wing with fuselage</th>
<th>Wing with roughness</th>
</tr>
</thead>
<tbody>
<tr>
<td>C_d_max</td>
<td>C_d_max_v</td>
<td>C_d_max_w</td>
<td>C_d_max_r</td>
</tr>
</tbody>
</table>

Dotted lines.

**Notes:**
- Flags neutral...
- Split...
- Double slotted...

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Positive directions of axes and angles (forces and moments) are shown by arrows.

<table>
<thead>
<tr>
<th>Axis</th>
<th>Force (parallel to axis) symbol</th>
<th>Moment about axis</th>
<th>Angle</th>
<th>Velocities</th>
</tr>
</thead>
<tbody>
<tr>
<td>Designation</td>
<td>Symbol</td>
<td>Designation</td>
<td>Symbol</td>
<td>Linear (component along axis)</td>
</tr>
<tr>
<td>Longitudinal</td>
<td>X</td>
<td>Rolling</td>
<td>L</td>
<td>u</td>
</tr>
<tr>
<td>Lateral</td>
<td>Y</td>
<td>Pitching</td>
<td>M</td>
<td>e</td>
</tr>
<tr>
<td>Normal</td>
<td>Z</td>
<td>Yawing</td>
<td>N</td>
<td>w</td>
</tr>
</tbody>
</table>

Absolute coefficients of moment

\[ C_t = \frac{L}{q_b S} \quad C_m = \frac{M}{q_c S} \quad C_s = \frac{N}{q_b S} \]

(rolling) (pitching) (yawing)

Angle of set of control surface (relative to neutral position), \( \delta \) (Indicate surface by proper subscript.)

4. PROPELLER SYMBOLS

- **Power**, absolute coefficient \( C_p = \frac{P}{\rho n^2 D^4} \)
- **Speed-power coefficient** \( C_s = \sqrt{\frac{P V^3}{n^3}} \)
- **Efficiency** \( \eta \)
- **Revolutions per second, rps** \( n \)
- **Effective helix angle** \( \phi = \tan^{-1}\left(\frac{V}{2\pi n}\right) \)

5. NUMERICAL RELATIONS

- 1 hp = 76.04 kg-m/s = 550 ft-lb/sec
- 1 metric horsepower = 0.9863 hp
- 1 mph = 0.4470 mps
- 1 mps = 2.2369 mph

- 1 lb = 0.4536 kg
- 1 kg = 2.2046 lb
- 1 mi = 1,609.35 m = 5,280 ft
- 1 m = 3.2808 ft