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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

REPORT No. 605

RESUME AND ANALYSIS OF N. A. C. A. LATERAL CONTROL RESEARCH

AERONAUTIC SYMBOLS

[illegible] AND DERIVED UNITS

[illegible]	[illegible]	[illegible]	[illegible]	Unit	Abbrevia-tion
Length Time Force	[illegible]	meter second weight of 1 kilogram	m s kg	foot (or mile) second (or hour) weight of 1 pound	ft (or mi) sec (or hr) lb
Power Speed	[illegible]	[illegible]	[illegible]	horsepower miles per hour feet per second	hp mph fps

[illegible]

Standard density of dry air, 0.12497 kg-m^{-4}-s^{2} at 15° C and 760 mm; or 0.002378 lb-ft^{-4} sec^{2}

Specific weight of "standard" air, 1.2255 kg/m^{3} or 0.07651 lb/cu ft

[illegible]

[illegible] where l is a linear dimension [illegible] airfoil of 1.0 ft chord, 100 [illegible] 15° C. [illegible] model of 10 cm chord [illegible] corresponding [illegible]

[illegible] leading edge to chord [illegible]

[illegible]

Angle of attack, infinite aspect ratio

Angle of attack, induced

Angle of attack, absolute (measured from zero-lift position)

[illegible]

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RÉSUMÉ AND ANALYSIS OF N. A. C. A. LATERAL CONTROL RESEARCH

By FRED E. WEICK and ROBERT T. JONES

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RÉSUMÉ AND ANALYSIS OF N. A. C. A. LATERAL CONTROL RESEARCH

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SUMMARY

An analysis of the principal results of recent N. A. C. A. lateral control research is made by utilizing the experience and progress gained during the course of the investigation. Two things are considered of primary importance in judging the effectiveness of different control devices: The (calculated) banking and yawing motion of a typical small airplane caused by a deflection of the control, and the stick force required to produce this deflection. The report includes a table in which a number of different lateral control devices are compared on these bases.

Experience gained while testing various devices in flight with a Fairchild 22 airplane indicated that, following a sudden deflection of the control at low speed, an angle of bank of 15° in 1 second represented a satisfactory minimum degree of effectiveness for this size of airplane. Some devices capable of giving this degree of control were, however, considered to be not entirely satisfactory on account of sluggishness in starting the motion. Devices located near the trailing edge of the wings had no detectable sluggishness. Lateral control forces considered desirable by the test pilots varied from 2 to 8 pounds; 15 pounds was considered excessive.

Test flights demonstrated that satisfactory lateral control at high angles of attack depends as much on the retention of stability as on aileron effectiveness.

The aerodynamic characteristics of plain sealed ailerons could be accurately predicted by a modification of the aerodynamic theory utilizing the results of experiments with sealed flaps. Straight narrow-chord sealed ailerons covering 60 to 80 percent of the semispan represented about the most efficient arrangement of plain unbalanced ailerons from considerations of operating force. The stick force of plain ailerons can be effectively reduced by the use of a differential linkage in conjunction with a small fixed tab arranged to press the ailerons upward.

INTRODUCTION

In 1931 the Committee started a systematic wind-tunnel investigation of lateral control with special reference to the improvement of control at low air speeds and at high angles of attack. Many different ailerons and other lateral control devices have been subjected to the same systematic investigation in the 7- by 10-foot wind tunnel. (See reference 1.) The devices that seemed most promising were tested in flight (references 2 and 3). In many cases, however, devices that produced what seemed to be satisfactory rolling moments and favorable yawing moments did not give satisfactory control.

An analytical study of control effectiveness was therefore made (reference 4) taking into account a number of secondary factors, including the yawing moments produced by the controls, the effect of the controls on the damping in rolling, the lateral-stability derivatives of the airplane, the moments of inertia, and the time required for the control moments to become established after the deflection of the surfaces. The computations consisted of step-by-step solutions of the equations of rolling and yawing motion for the conditions following a deflection of the controls. The results of these computations based on aerodynamic data obtained from wind-tunnel tests of wings incorporating various devices agreed satisfactorily with the results measured in flight for widely different forms of control, such as ailerons and spoilers.

The study of conditions above the stall indicated that satisfactory control could not be expected without some provision to maintain the damping in rolling and that a dangerous type of instability would arise if the damping were insufficient. Since damping in rolling depends on an increase in the lift of the airfoil with increasing angle of attack, it follows that, in order to obtain satisfactory lateral control, the outer or tip portions of the wing, which govern the rolling moments, must remain unstalled. If damping in rolling is retained, it is practically insured that control moments will be retained as well.

The progress of the investigation has thus led to a more accurate interpretation of the results of the wind-tunnel tests. In the present paper the experience gained during the course of the investigation is made the basis of a revised method of comparison of lateral control devices. Wind-tunnel measurements of control and stability factors (reference 1) are utilized in computations to show the banking and yawing motions that would be produced by the controls acting on a small typical airplane. These computations follow the method of analysis given in reference 4. In section I of the report the new basis of comparison is explained and a number of the devices that were tested in reference 1 are analyzed and compared. The principal items of comparison are collected into a table. Section II presents an analysis of the rolling, yawing, and hinge moments of plain flap-type ailerons and deals with the application of these data in the design of control systems.

I. COMPARISON OF LATERAL CONTROL DEVICES

REVISED BASIS OF COMPARISON

AIRPLANE USED IN COMPARISON

The procedure adopted in the lateral control investigation has comprised a wind-tunnel test program followed by flight tests of the different devices on the Fairchild 22 airplane. Not all of the devices tested in reference 1 have been tried in flight, however, and the present report may be considered an analytical extension of the flight-test procedure that was applied to some of the devices. The procedure employed to test lateral controls in flight is simulated by means of computation. Thus, the comparative criterions used herein are based on application of the devices to a hypothetical Fairchild 22 type of airplane, which is the type used in the flight tests.

The Fairchild 22 airplane was necessarily somewhat modified for each different flight test and wings of different moment of inertia, plan form, and section were used in some cases. The wing of the hypothetical airplane assumed in the computations represents an average of the tested wings. Furthermore, since the characteristic ratios of dimensions (tail length, tail area, radii of gyration about various axes, etc.) used agree very closely with statistical averages of these quantities, the assumed airplane may be considered to embody average stability characteristics. The principal characteristics of the assumed airplane are as follows:

Weight, W	1,600 lb.
Wing span, b	32 ft.
Wing area, S	171 sq. ft.
Wing loading, W/S	9.4 lb. per sq. ft.
Area of fin and rudder	10.8 sq. ft.
Tail length	14.6 ft.
I_X	1,216 slug-ft.^{2}
I_Z	1,700 slug-ft.^{2}

ROLLING ACTION

It is recognized that different types of airplanes require different amounts of control. At the start of the wind-tunnel investigation of lateral control devices (reference 1) a rolling criterion ($RC=C_l/C_L$) representing a conservative lower limit of rolling control for all types was assumed. The assumed satisfactory value of the rolling criterion was 0.075, which corresponds to a lateral movement of the center of pressure of 7.5 percent of the wing span. Recent experience indicates that this value is likely to be ample for any condition of flight that might be encountered and is therefore a desirable value to attain. Where a compromise must be made between the rolling moment and some other characteristic of the control system, particularly the control force, a decidedly lower value of the rolling criterion may be used. It appears that a value possibly as low as half the original one may be found reasonably satisfactory for practically all conditions of flight with nonacrobatic airplanes.

The criterion of rolling control used in the present analysis is the angle of bank attained in 1 second following a sudden deflection of the control. This criterion shows the actual amount of motion produced and depends on both the acceleration at the start and the final rate of roll. It includes the effect of yawing moment given by the control as well as the stability characteristics and moments of inertia of the airplane. The values of the criterion are found by computation and as such are applicable only to the particular type of airplane (F-22) that has been assumed.

Experience gained in flight tests of the Fairchild 22 airplane with various lateral control devices indicated a minimum satisfactory amount of rolling control corresponding to about 15° of bank in 1 second. (See fig. 1.) Ailerons capable of giving this amount of bank

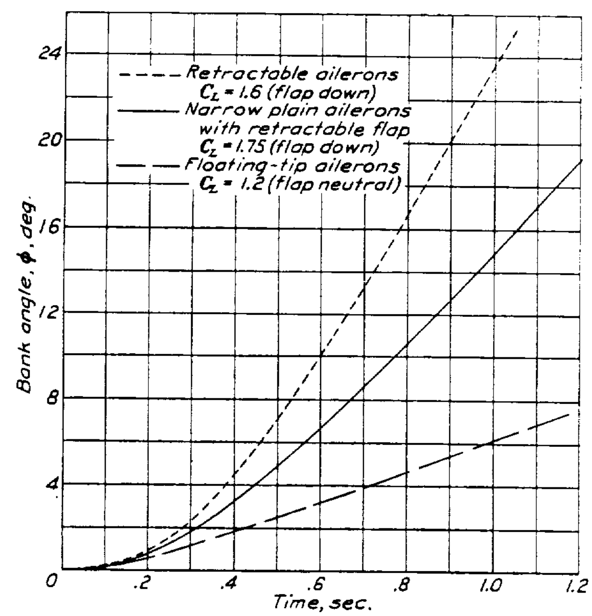


FIGURE 1.—Banking of Fairchild 22 airplane after sudden deflection of lateral control devices at low speed. (The narrow plain ailerons and the retractable ailerons were considered to give a satisfactory amount of control; the floating-tip ailerons were reported as weak.)

at low speed have been found reasonably satisfactory in practice with this type of airplane. Owing to the present general use of high-lift flaps on airplane wings, the size and deflection of ailerons are usually determined by the low-speed condition of flight with the flaps deflected. For comparative computations, in the present report, a lift coefficient of $C_L=1.8$ is assumed as representative of the low-speed condition of flight with flaps. The sizes or deflections of the lateral controls are selected in each case to give an angle of bank of 15° in 1 second at $C_L=1.8$.

In addition to providing a sufficient amount of banking motion, two further desirable characteristics of the rolling action are: (1) The response of the airplane in roll to any movement of the lateral control surface should be immediate, any noticeable delay or hesitation in the action being objectionable; and (2) the action should be so graduated that the acceleration and maximum rate of roll increase smoothly and regularly as the stick deflection is increased. Conventional ailerons or similar lateral control devices located near the trailing edge of the wing easily meet these requirements and show, in analyses of motions recorded in flight, practically instantaneous response of rolling acceleration to control-surface movement. From 0.1 to 0.2 second is ordinarily required to deflect the surfaces and, during this interval, the rolling acceleration apparently keeps pace, although only a slight amount of rolling motion is accumulated by the time of full deflection. Comparison shows that good synchronization of the calculated motion with the flight records was obtained when the assumed full deflection was taken at the instant the actual deflection reached half its ultimate value. This assumption was used in the computations for plain ailerons and other devices that gave no indication of sluggish response characteristics.

CONTROL FORCE

During the course of the lateral control investigation it became apparent that the force required to move the controls is of extreme importance in obtaining satisfactory lateral control. As shown by the flight tests of references 2 and 3, an airplane that requires a light control force is likely to seem more controllable to a pilot than one that requires a heavy control force, even though with full deflection the heavier control may be considerably more powerful than the lighter one. It seems desirable to have the control force as light as possible and yet to maintain the feeling of a definite neutral position. This characteristic is especially important in the aileron control since the effort expended in moving the stick sidewise is relatively greater than for other control movements. (See reference 5.) Correlation of test-flight reports and control-force records indicates that the forces required to operate the ailerons should not exceed about 8 pounds in order to be considered desirable. A lower limit of stick force of about 2 pounds at full deflection is apparently considered essential so that there may be a noticeably regulated increase of force with deflection. Friction of the control mechanism plays an increasingly important part as the operating force is reduced and should in no case be great enough to mask the "feel" of the control. It is probable that with sufficiently little friction a force not greatly in excess of 2 pounds would be considered most desirable. A force of 15 pounds is to be considered excessive.

As previously stated, the size or maximum deflection of the control devices compared in this paper have been selected to give an angle of bank of 15° in 1 second following full deflection and, considering the average airplane fitted with a high-lift flap and flying at a lift coefficient of 1.8, the ailerons are compared (see table I) on the basis of the stick force required to attain this angle of bank of 15° in 1 second at lift coefficients of 0.35, 1.0, and 1.8, which compose the usual flight range. The lift coefficient of 0.35 represents the conditions of high-speed and cruising flight. The lift coefficient of 1.0 is considered to represent two conditions, the first being that of low-speed flight without a flap, such as is used in an approach to a landing with an unflapped airplane, and the second being one with a flap fully deflected, which represents as high a speed as is usually attained in that condition. The value $C_L=1.8$ can be obtained only with the flap deflected and represents the low-speed flight condition with the high-lift device in use. When representative values of this nature are used, it is necessary to examine the complete original data to show that the critical values are representative of conditions throughout the flight range. Such an examination has been made for the comparisons of the present report.

The stick force for a 15° bank in 1 second is used as the basis of comparison at all flight speeds and lift coefficients even though the conventional ailerons will produce a decidedly greater bank in 1 second at higher speeds. The 15° value is taken throughout because it is considered to represent the maximum control likely to be used in ordinary flight at any speed and is therefore of greater interest as a basis for stick forces required than the maximum possible deflection, as long as the force at maximum deflection does not approach the strength of the pilot.

The data for some of the ailerons were obtained with plain unflapped wings with which a lift coefficient of 1.8 could not be attained and, in order to have all the lateral control devices on a comparable basis whether mounted on flapped or unflapped wings, their sizes and maximum deflections were selected to give essentially the same rolling effect as the others at a lift coefficient of 1.0. The analysis showed that conventional ailerons which give an angle of bank of 15° in 1 second on a flapped wing at a lift coefficient of 1.8 could, when fully deflected, give an angle of bank of 22.5° with the flap retracted at a lift coefficient of 1.0. The ailerons on the unflapped wings were therefore selected to be capable of giving 22.5° bank in 1 second at a lift coefficient of 1.0, but the values of the stick forces required were computed for partial deflections giving a 15° bank in 1 second at lift coefficients of both 1.0 and 0.35. The first aileron of table I is of the conventional unbalanced flap type on a rectangular wing of aspect ratio 6. It has a chord 0.25 c_w and a span 0.40 $b/2$ and has equal up-and-down linkage. It will be noted that, for an airplane equipped with these ailerons, the stick force computed for a 15° bank in 1 second at the cruising-flight condition is 4.7 pounds with aileron deflections of only ±3.4°. At a lift coefficient of 1.0, representing the low-speed flight condition for the unflapped wing, the same amount of control was obtained with a stick force of 3.6 pounds and aileron deflections of ±7.4°. All the stick forces are given for an assumed aileron linkage such that at the maximum deflection the control stick, which has a length of 20 inches on the Fairchild 22 airplane and is so assumed for the average airplane, is deflected 25° from neutral. The maximum aileron deflection is 11.2° and is the deflection required to produce a bank of 22.5° in 1 second at $C_L=1.0$. Here the ailerons are not being taxed to their fullest extent.

The maximum amount of control specified in a design has a predominating effect on the operating force. Figure 2 shows a calculated example of the variation of

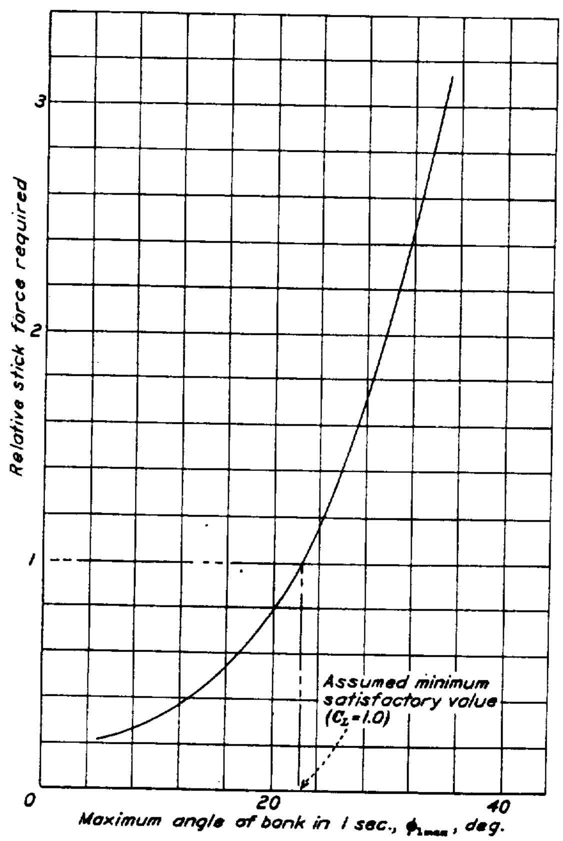


FIGURE 2.—Relation between stick force and maximum amount of control obtained. Fairchild 22 type airplane; 0.80 $\frac{b}{2}$ sealed ailerons deflected ±20°; aileron chord varied.

operating force with specified control in which it was assumed that ailerons with equal up-and-down motion and the most efficient length and deflection (±20°) were used in each case. The rate of increase of operating force with amount of control depends on the manner in which the increase of control is obtained, as will be more fully developed in a later section.

YAWING MOTION AND SIDESLIP

The effect of the yawing moment produced by the ailerons is considered in two ways. First, the secondary effect of yaw on the rolling motions is inherently included in the computed banking effectiveness. Thus, the bank in 1 second is that produced by the ailerons without aid from the rudder. If it is assumed that a sufficiently powerful rudder were used in such a way as to prevent sideslip, a given aileron device would, in general, produce a somewhat greater banking effect. This assumption is not used here, however, and the deflections of the control surfaces given in table I are those required to produce the specified angle of bank in 1 second with the particular combination of rolling and yawing moments produced by the aileron in question.

The second effect considered is the sideslip produced by the sudden use of the aileron control for banking. In flight the rudder is used to avoid sideslipping and the amount of rudder action necessary for this purpose is in direct proportion to the sideslip incurred by the ailerons alone.

The angle of sideslip accompanying a 15° bank in 1 second following the sudden displacement of the lateral controls is also given in table I. The first aileron listed, it will be noted, produces a sideslip of 7° at $C_L=1.0$ and of 3° at $C_L=0.35$ when the rudder is not used to correct for this condition.

LATERAL STABILITY

In the ordinary unstalled-flight range the effects of the lateral-stability factors on the lateral control obtained are included in the computations of the angle of bank reached in unit time. The angle of bank ϕ_1 is the angle that would be produced by the control operating on the average airplane. The effect of a given control on an airplane of greatly different lateral-stability characteristics might, of course, be considerably different than indicated in this case.

One of the most important factors in the interaction of lateral stability and control below the stall is the effect of the secondary yawing moment induced by the control and an allowance for this effect should be made in the proportioning of the airplane for lateral stability. Modifications that tend to increase spiral stability in free flight (namely, reduced vertical-fin area and increased dihedral) tend to render the airplane uncontrollable under the action of ailerons giving adverse yawing moment. The degree of "weathercock" stability should be sufficient to restore the airplane from a yawed attitude when the wings are held level by use of the ailerons. For safety in this respect the ratio of adverse yawing to rolling moment given by the ailerons should not be allowed to approach the ratio of yawing to rolling moments that naturally act on the airplane either in pure sideslipping or pure yawing motion. (See reference 6.)

One of the lateral-stability factors, the damping in rolling, has been shown by the analysis in reference 4 to have a critical effect on the controllability obtained, satisfactory lateral control requiring that positive damping exist. Since the damping in rolling depends on a positive slope of the left curve, the damping exists only at angles of attack of the outer portions of the wing that are below the maximum lift coefficient. While some semblance to control may be obtained at angles of attack above the stall if controls giving favorable yawing moments as well as sufficiently powerful rolling moments are used, the instability associated with uneven stalling and autorotation is so violent that it is necessary for the pilot to use the controls continually to keep the airplane near the desired attitude. If sufficiently rapid rolling is once started, either by the controls themselves or as the result of gusty air, it cannot be stopped. The angle of attack at which the damping in rolling becomes zero and above which autorotation takes place ($\alpha_{L_p=0}$) is used herein as an indication of the limit of the flight attitude above which satisfactory lateral control cannot be obtained. This value was given in the reports of reference 1 for both the angle of attack at which autorotation was selfstarting and the angle of attack at which the damping became zero when the wing was rotating at the rate $pb/2V=0.05$, a value representative of the rolling likely to be caused by gusty air. The latter value of α has ordinarily been found to be about 1° lower than the former value and, being therefore more decisive, is used in the present report. The difference between the angle of attack for zero damping and the angle of attack for the maximum lift coefficient of the entire wing $(\alpha_{L_p=0}-\alpha_{C_{L_{max}}})$ has been tabulated under Lateral Stability to show whether the maximum lift coefficient can be expected to be reached in flight before satisfactory lateral control is lost. It will be noted that for ailerons 3 and 4 the wing loses its damping in roll at an angle of attack 1° higher than that at which the maximum lift coefficient is reached. Thus, as far as the stability is concerned, lateral control should be possible throughout the entire unstalled-flight range, including the angle of attack for maximum lift coefficient.

WING PERFORMANCE CHARACTERISTICS

The same criterions used throughout the reports of reference 1 to show the relative performance characteristics of the wings are used in the present report and are tabulated in the last three columns of table I. The maximum lift coefficient $C_{L_{max}}$ is given as an indication of the wing area required for a desired minimum speed. The ratio $C_{L_{max}}/C_{D_{min}}$ is an indication of the speed range and, for a given minimum speed, shows the relative effects of the wings on the maximum speed attainable. The ratio L/D taken at a value of the lift coefficient $C_L=0.70$ is an indication of relative merit in climbing flight. In a series of performance computations made for airplanes of different wing loadings and power loadings and with both plain and slotted wings, this criterion was found to be satisfactory throughout the entire range. It should be noted that the comparative values used in the present report are based on tests made in the 7- by 10-foot atmospheric wind tunnel and hence do not coincide in absolute value with results of tests made at different Reynolds Numbers.

APPLICATION TO AIRPLANES OF DIFFERENT SIZES AND LOADINGS

Because the flight experience that led to the specification of a satisfactory degree of control was restricted to the Fairchild 22 type of airplane, there is some doubt about the application of this experience to other types and especially to large or very small airplanes. The Fairchild 22 type of airplane, of course, serves as well as any other when different aileron devices are simply compared among themselves. The principles governing the extension of the computations of motion to geometrically similar airplanes of different sizes and loadings are well known and can be applied here, but this extension of the computations does not definitely answer the question as to what constitutes a satisfactory degree of control for large (or very small) airplanes.

According to the principles of dynamical similarity, large or small similar airplanes of the same wing loading would show the same linear rise and fall of the wing tips $\left(\frac{\phi_1 b}{2}\right)$ during a 1-second banking motion. Large and small airplanes do actually show a tendency toward similarity in important dimensions and size of control surfaces, and it seems logical to assume that a given value of the vertical distance described by the wing tips within 1 second following a sudden control deflection that represents a satisfactory amount of control for the Fairchild 22 airplane should be satisfactory for any size of airplane.

For similar airplanes the linear distance described by the wing tips in banking $\left(\frac{\phi_1 b}{2}\right)$ is independent of the size. Figure 3 shows this distance plotted against wing loading and gives the separate effects of rolling and yawing moments of coefficient 0.01 at different lift coefficients. The banking effect of any combination of rolling and yawing moment may be found by superposition, i. e.,

$$\frac{\phi_1 b}{2}=\frac{C_l}{0.01}\left(\frac{\phi_1 b}{2}\right)_{C_l=0.01}+\frac{C_n}{0.01}\left(\frac{\phi_1 b}{2}\right)_{C_n=0.01} \quad (1)$$

The ordinates of the figure give directly the circumferential displacement of the wing tip in feet for a unit of 0.01 rolling- or yawing-moment coefficient. It is important to note that the banking effects of rolling and yawing moments can be separately considered and later added in any desired proportion to obtain the total combined effect.

The computations show that, in general, smaller values of the control-moment coefficients are required to produce a given wing-tip displacement in a unit of time for the more heavily loaded airplanes. Another point of interest in connection with the secondary adverse yawing moments produced by conventional-type controls is that these moments are more effective in hindering the control with lightly loaded airplanes than with heavily loaded ones. Note that in the usual case the banking effect of the yawing moment is to be deducted in equation (1) since this moment is usually adverse and therefore negative.

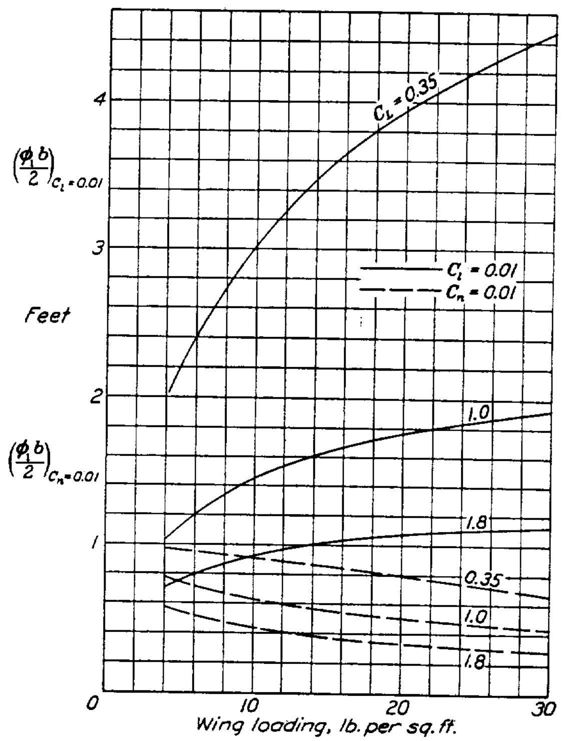


FIGURE 3.—Wing-tip displacement produced in 1 second by suddenly applied rolling and yawing moments for different wing loadings and flight speeds.
$$\frac{\phi_1 b}{2}=\frac{C_l}{0.01}\left(\frac{\phi_1 b}{2}\right)_{C_l=0.01}+\frac{C_n}{0.01}\left(\frac{\phi_1 b}{2}\right)_{C_n=0.01}$$

The variation of control force with size and loading of the airplane may be determined from general rules as in the case of the variation of the amount of rolling motion. As shown by figure 3, heavily loaded airplanes require smaller control-moment coefficients for a comparable amount of control than do lightly loaded airplanes. In general, a heavily loaded airplane that is otherwise similar to a lightly loaded one will have smaller control surfaces. On the other hand, the heavily loaded airplane will fly at a higher speed so that the dynamic pressure will be greater. Figure 4 shows a calculated example of the variation of stick force with wing loading at a given lift coefficient and for a given maximum amount of control. Here, as in figure 2, the most efficient combination of size and deflection is assumed for each point. Figure 4 shows that the stick force required to obtain a given angle of bank in 1 second is practically the same for all wing loadings up to 10 pounds per square foot but that it increases somewhat as the wing loading increases further.

With moderately large airplanes, somewhat higher stick forces are apparently tolerated by pilots without serious objection. With extremely large airplanes, however, the operating force becomes too great to be satisfactorily overcome by the pilot and either servo controls or auxiliary power is required. With auxiliary power, the pilot might presumably operate a valve or easily deflected controller governing a special power

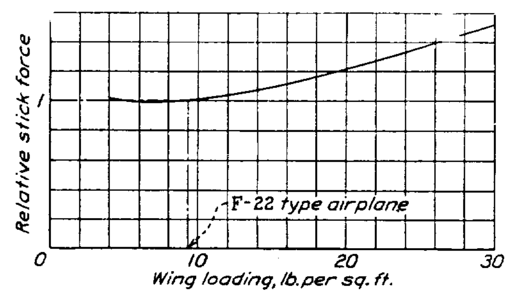


FIGURE 4.—Relation between the wing loading and the stick force required for a given amount of control ($\phi_{1_{max}}=22.5°$; $C_L=1.0$).

source that deflected the control surfaces. Under such conditions the magnitude and variation of the hinge moments would be relatively less important and the maximum deflection of the control surfaces would very likely be determined by the maximum rolling and yawing moments they could produce rather than by the hinge moments and the resultant deflecting force required. Although some indication of the relative performance of the various lateral control devices compared in this report can be obtained from the data as given, it would be desirable to reanalyze the original data given in references 1, 7, 8, 9, and 10 if a comparison on the basis of ailerons operated by auxiliary power were desired.

COMPARISONS OF VARIOUS DEVICES

PLAIN AILERONS

Effect of aileron and wing plan form.—The tests of reference 1, part I, were made with rectangular wings having ailerons of three different proportions: 0.25 c_w by 0.40 $b/2$ (which were taken as the standard for comparison throughout the series), 0.15 c_w by 0.60 $b/2$, and 0.40 c_w by 0.30 $b/2$. These sizes were selected to give approximately equal rolling moments with the same angular deflection. These ailerons are numbered 2, 3, and 4, respectively, in table I. With equal up-and-down deflection, the stick force is much larger for the short, wide ailerons than for the long, narrow ones and is, in each case, slightly less for the low-speed condition than for high speed. If a suitable differential linkage is employed, the stick forces at the low-speed condition, where the wide ailerons have the advantage of a large floating angle, are quite low for all three sizes of aileron. At the high-speed condition, however, the 0.40 c_w by 0.30 $b/2$ aileron requires a rather high stick force, even with the best differential.

The sideslip incurred by an angle of bank of 15° in 1 second is not greatly different for the different aileron plan forms either with or without differential linkages. The values are slightly lower at $C_L=1.0$ with the differential linkages than with the equal up-and-down, and with the 0.25 c_w by 0.40 $b/2$ plan form than with either of the others.

It is possible by methods to be described in section II to compute an optimum size of the aileron, i. e., the size giving the desired amount of control with the least stick force. The effect of varying the aileron span and chord is shown in figure 5, the chord for each span value being

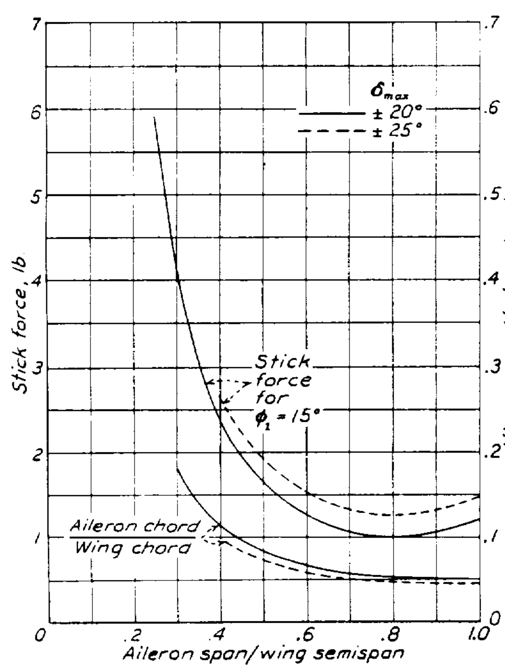


FIGURE 5.—Variation of stick force with aileron span. Aileron chord proportioned to give $\phi_{1_{max}}=22.5°$ with maximum deflection of ±25° and ±20°; rectangular wing, average airplane; $C_L=1.0$; sealed ailerons.

the smallest that will give an angle of bank of 15° in 1 second with the assumed average airplane. From this figure it is apparent that with equal up-and-down deflection an aileron span of 80 percent of the wing semispan will give the lowest stick force, but the variation is small for ailerons between 60 percent and 100 percent of the wing semispan. Other computations not shown lead to the same conclusion for ailerons having differential linkages.

The relations of aileron chord and span, considering especially that the hinge moment increases with the square of the chord while the rolling moment increases only as the square root of the chord, are such that lower stick forces are obtained with narrower chords. The narrower ailerons require greater deflections and the reduction in chord size is limited by the fact that deflections greater than about ±20° are inefficient. Marked separation of the air flow takes place at about this angle of deflection on all the conventional flap-type ailerons tested and, as shown by the typical curves of figure 6, the rolling-moment coefficients increase at a lower rate beyond 20° deflection. If it is attempted to

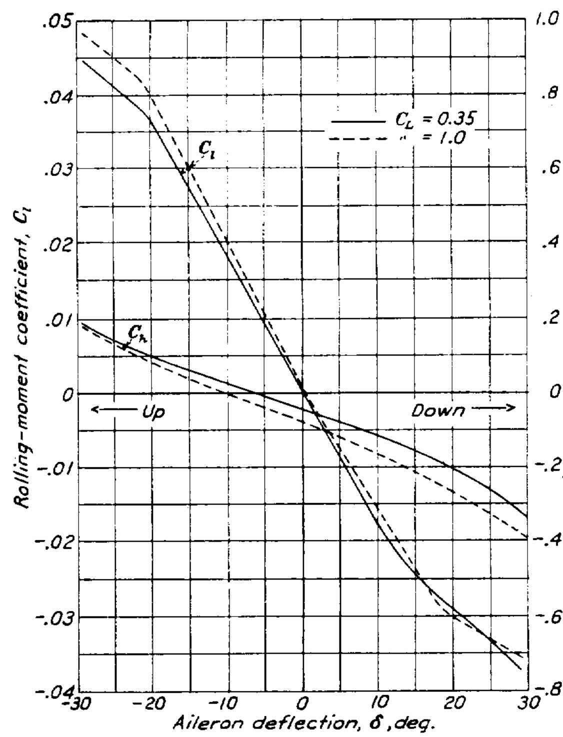


FIGURE 6.—Typical rolling- and hinge-moment coefficient curves for plain ailerons.

reduce further the chord of the aileron by extending the deflection beyond this break, the stick force will be higher because of the loss in mechanical advantage. Figure 5 illustrates this point, for when an aileron deflection of ±25° is assumed, narrower ailerons are required but the stick force is larger for all aileron spans than with a deflection of ±20°.

Aileron 5 (table I) represents the narrowest sealed aileron covering 80 percent of the wing semispan that gives the required control with a deflection of ±20°. The aileron chord in this case is only 5.3 percent of the wing chord, and the stick forces are lower than for any of the previous ailerons. If a differential motion is used, a somewhat wider aileron is required. With narrow ailerons the floating angle is very small, and a tab is required to make the ailerons float at a sufficiently high angle that the differential linkage will be effective in reducing the stick force. (See reference 11.) Aileron 6 of table I is the smallest one covering 80 percent of the semispan that will give the required amount of control with a differential motion and with suitable aileron tabs. The assumed tab covers the entire trailing edge of the ailerons, has a chord 1.5 percent of the wing chord, and is permanently bent downward 14°. For this case the entire aileron chord including the tab is 7.8 percent of the wing chord and the stick force is only 0.5 pound for the high-speed condition and 0.1 for low speed.

These values of stick force are lower than are considered desirable for the Fairchild 22 airplane but are interesting in showing the possibility of obtaining a satisfactorily low stick force in larger and heavier airplanes. For small airplanes, one satisfactory method of increasing the stick force to the value desired would be to use greater up travel than 20° with differential ailerons, thus getting into the range of inefficient stick force although obtaining the advantage of slightly smaller adverse yawing moments.

In many practical cases the chord of the aileron varies along the span. Inasmuch as the hinge moment varies as the square of the chord and the control effectiveness only about as the square root of the chord of an aileron element, the stick force required to give a certain amount of control is inherently greater if the chord of the aileron varies appreciably along the span. This relation is true in spite of the fact that the portion of the aileron nearer the tip of the wing has a greater lever arm, which suggests that it might be advantageous to increase the chord of the aileron as the wing tip is approached. Thus, it is possible to state as a general rule that to obtain the lowest stick force, ailerons should have an essentially constant chord over their entire span.[1]

On wings having rounded tips it is sometimes the practice to use ailerons having skewed hinge axes like aileron 7 in table I. This aileron corresponds in span, area, and gap to the 0.25 c_w by 0.40 $b/2$ aileron 2, but the stick force is decidedly higher for the skewed ailerons on account of the variation of the aileron chord along the span.

Ailerons 8 and 9 of table I are of tapered plan form and are mounted on tapered wings. In the computations of the rolling effect with the tapered wings the reduction in the moments of inertia due to the taper are taken into account. For example, for the wing with 5:1 taper, the value of I_X was changed from 1,216 slug-feet^2 for the original average airplane to 860, and the value of I_Z from 1,700 to 1,400 slug-feet^2. The lateral-stability derivatives were also changed to take account of the taper. (See reference 4.)

A comparison of ailerons 8 and 9 with aileron 1, which has the same relative chord size but is attached to a rectangular wing, shows that the stick force becomes lower as the taper of the wing is increased. The sideslip or adverse yawing effect is also smaller with the tapered wings than with the rectangular. The lateral-stability factor, damping in roll, is reduced to zero at an angle of attack 3° below the stall with the 5:1 tapered wing, indicating that the airplane could not be safely maintained at the maximum lift condition in flight.

The ailerons on tapered wings dealt with up to this point have had chords that were the same percentage of the wing chord at each position along the span, the ailerons tapering with the wings. It has been stated that the lowest stick force would be obtained with constant-chord ailerons. Computations have been made comparing the straight or constant-chord ailerons on a tapered wing with the ailerons that taper with the wing, and the results are shown in figure 7. The straight

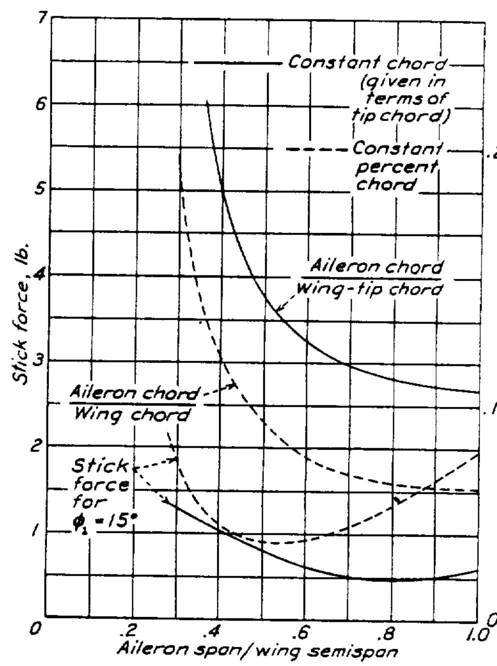


FIGURE 7.—Variation of stick force with aileron span and chord for straight and tapered ailerons on 5:1 tapered wing. Aileron chord proportioned to give $\phi_{t_{max}}$ = 22.5° with maximum deflections of ±20°; C_L=1.0; sealed ailerons.

or constant-chord ailerons require lower stick forces for any given aileron span. It is interesting to note that with tapered ailerons the aileron span giving the lowest stick force is about half the wing semispan; whereas with constant-chord ailerons the best aileron span is 80 percent of the wing semispan, as it is in the case of rectangular wings. Ailerons 10 and 11 are the optimum sizes for the tapered and straight ailerons, respectively, on a 5:1 tapered wing. With equal up-and-down deflections, the stick forces for the straight ailerons are about half those for the tapered. In either case the stick forces could be nearly counterbalanced by means of a suitable differential linkage and tab, as will be developed more fully in section II.

Effect of hinge gap.—Wind-tunnel tests have shown that even a slight gap between ordinary unbalanced ailerons and the wing upon which they are mounted

[1] The greatest taper mathematically compatible with a minimum stick force is less than about 3 percent of the aileron chord.

causes a relatively large loss in rolling moment. This loss for unbalanced flaps having a gap of one thirty-second inch on a wing of 10-inch chord was found to be approximately 30 percent. The hinge moment is also reduced by the gap but to a much lesser extent and the resultant stick force for a given amount of lateral control is greater because a larger aileron deflection is required, which necessitates a linkage having a poorer mechanical advantage. The effect on the stick force is shown in table I by a comparison of the values for aileron 2, which has a gap, with those for aileron 1, which is sealed.

BALANCED AILERONS

Balanced ailerons of the Frise and Handley Page types are widely used at the present time, the particular forms of aerodynamic balance incorporated in these ailerons giving improved yawing moments as well as reduced hinge moments. Good results are obtained with proper designs but the exact shape of these ailerons has a critical effect on the rolling and hinge moments, and each different installation is likely to require considerable individual development. Figure 8 shows typical curves of rolling and hinge-moment coefficients for Frise type ailerons. The rolling-moment coefficient for the example shown increases less rapidly with deflection after an upward angle of 7° to 10° has been reached, which is considerably lower than the 20° critical deflection for plain unbalanced ailerons (fig. 6). Thus, it is uneconomical with respect to stick force to use large up deflections and, owing to the smaller maximum deflections, larger ailerons are required for efficiency than when ailerons of the plain unbalanced sealed type are used. The break in the curve of rolling-moment coefficient against deflection is associated in the case of the Frise and Handley Page types of aileron with the downward projection of the nose of the aileron and the resultant breaking away of the flow from the under side of the aileron. This effect can be reduced or possibly eliminated by using a raised-nose portion.

The Frise and Handley Page types of aileron have gaps between the aileron and the wing, and the effectiveness of the ailerons cannot be assumed equal to that of smoothly sealed flaps.

The hinge-moment curves as shown in figure 8 have very low and even negative slopes at places, and extreme differential linkage cannot be used because overbalance would occur with medium or small deflections of the up aileron. Because the hinge-moment curves are far from straight, it is more difficult to select suitable differential linkages for ailerons of this type than for plain unbalanced ailerons. Satisfactory linkages have often been obtained in practice, however, and there are many excellent examples in which a nice balance of conditions has been obtained with satisfactory control and light stick forces.

Ailerons 12 and 13 are examples of the Frise type. A comparison of aileron 12 with the same size of plain unbalanced but sealed ailerons shows that the stick forces at the low-speed condition are about the same for both types of aileron, both with equal up-and-down and with differential motion. At the high-speed condition the Frise ailerons have somewhat lower stick forces than they have for the same control at low speed. It is worthy of note that, although the deflections are small in both cases, the Frise ailerons are apparently not greatly oversized for, in their case, substantially greater deflections would be inefficient. The plain ailerons, on the other hand, have maximum deflections well under the limiting 20° value and are decidedly oversized, considering the amount of control specified.

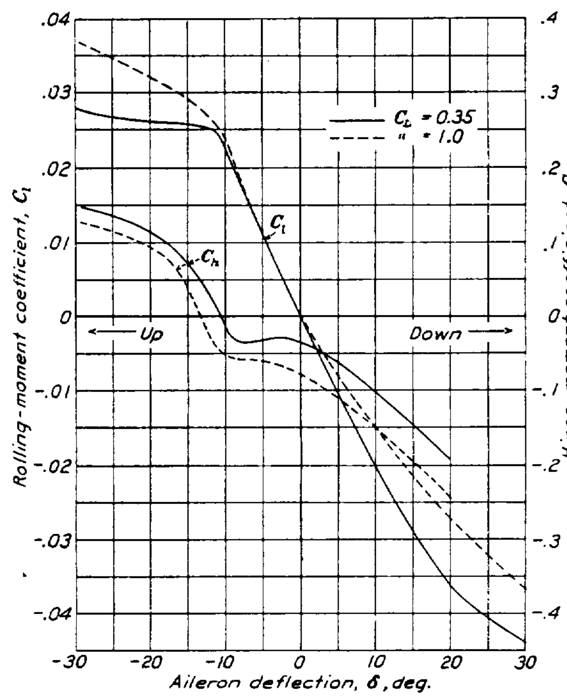


FIGURE 8.—Typical rolling- and hinge-moment coefficient curves for Frise ailerons.

If a fixed tab is used to trim the ailerons upward, lower values of stick force can be obtained with the plain unbalanced ailerons (reference 11). The tab will not give the same improvement with the Frise ailerons because of the varying slopes of the hinge-moment curves.

The 0.40 c_w by 0.30 $b/2$ Frise aileron 13 has a different sectional form than aileron 12 in that the nose portion is raised, and this aileron gives smoother curves of rolling and hinge-moment coefficients. The Frise aileron with the raised nose shows no improvement in yawing effect over the plain unbalanced ailerons of the same size, but the 0.25 c_w by 0.40 $b/2$ Frise aileron, which has the more typical Frise sharp nose, gives a slight improvement in this respect.

The drag of all commonly used forms of Frise and Handley Page ailerons is sufficiently great to be considered a serious disadvantage in connection with modern high-performance airplanes. For this reason, the development of a type of aerodynamic balance that does not add to the drag is desirable.

FLOATING-TIP AILERONS

Conventional ailerons operating on a lifting portion of the wing suffer several fundamental disadvantages. First, the production of rolling moment by a lifting wing gives rise to the adverse yawing moment; and, second, the loss of lift at the stall is accompanied by a loss of effectiveness of the ailerons. It has become apparent during the investigation, however, that the stall of the wing or, at any rate, of the outer portions of the wing, is accompanied by such a loss of stability that it is hardly an advantage to retain aileron rolling moments in this condition.

In the case of floating-tip ailerons, control is secured by surfaces that contribute no lift. This arrangement avoids both the adverse yawing moment of ordinary ailerons and the loss of rolling moment associated with stalling of the main wing; but it increases the drag of the airplane and adds to the over-all dimensions. If the airplane is designed to fulfill certain performance specifications, such as landing speed, climb, ceiling, etc., the floating-tip ailerons cannot be considered an integral part of the main wing as they do not contribute effectively to the area or span so far as induced drag and lift are concerned.

A number of floating-tip aileron devices were tested in the course of the investigation of reference 1. Apparently the most usable of these are the tip ailerons on the 5:1 tapered wing. Two methods of comparison have been followed. In one case (aileron 14) the ailerons were included within the over-all dimensions of the 5:1 tapered-wing average airplane. The values given in the table for this case (short wing) were based directly on the results of tests made in the 7- by 10-foot wind tunnel (reference 1, part XI). The criterions show the effect of reduced area and span of the lifting portion of the wing as a reduction of the climb and maximum lift.

In order to take account of the effect of simply adding a tip aileron to a normal-size wing, further calculations were made. In this case (aileron 15) it was assumed that the over-all span of the average airplane was increased by the additional span of the tip ailerons; hence, the aspect ratio of the lifting portion of the wing remained the same. The added span of the wing, although it contributed practically no lift and hardly modified other stability characteristics of the airplane, considerably increased the damping in rolling. This fact was accounted for in the computations, data on damping of the tested 5:1 tapered wing with floating-tip ailerons included in the original plan form being extrapolated for this purpose. It would be natural to assume that the floating-tip ailerons would be just as effective as the main portion of the wing in contributing damping. The tests showed, however, that the damping of the 5:1 tapered wing with floating tips was only 85 percent of that with the tips rigid.

The rolling moments produced by floating-tip ailerons can be predicted with good accuracy by the conventional aileron theory. The induced yawing moments correspond to those given by plain ailerons with an extreme uprigging or negative droop corresponding to the neutral floating positions of the tip ailerons. Ordinarily, the tip ailerons, on account of the local upwash at the end of the rigid wing, float at a negative angle of attack relative to the mean direction of flight and hence give slight favorable induced yawing moments with respect to the wind axes. The yawing and hinge moments used in table I for the long-wing airplane (aileron 15) were predicted from the results of the wind-tunnel tests on the short 5:1 tapered wing.

The tabulated results of the computations show that the stick forces required for satisfactory control are reasonably low in the case of the short 5:1 tapered wing. It will be noted that only relatively small deflections of these ailerons are required for control, a fact that can be attributed partly to the reduced damping in rolling shown by this wing. On the other hand with the long wing, when the tip ailerons were added to the regular wing span, the damping in rolling and moment of inertia were increased and, hence, larger stick forces were required to produce the given bank. The same hinge-axis location, and hence the same degree of balance of the ailerons, were assumed in both cases. It will be noted that about the same force was required to produce 15° bank at high and low lift coefficients.

Although the floating-tip ailerons give small favorable yawing moments, it will be noted that their use results in some inward sideslip during the 15° bank. The rolling motion of the wing induces a small adverse yawing effect as is indicated by the adverse sign of the yawing moment due to rolling. This cause combined with the inward acceleration due to gravity is sufficient to bring about the inward sideslip in spite of the favorable yawing moment of the floating ailerons.

It has often been suggested that tip ailerons be trimmed by tabs so as to float downward and give some lift. Such an arrangement should improve the performance characteristics but would void the advantage of these ailerons in giving favorable yawing moments. If the tip ailerons were trimmed so as to produce as much lift as the adjacent rigid portion of the wing, it is to be expected that they would show the same proportion of adverse yawing moment to rolling moment as do conventional ailerons.

At stalling angles of attack for the main wing the floating tips remain unstalled. Hence, they should be expected to aid in preventing the loss of damping in rolling at or near the stall. The only floating aileron device that effectively prevented the loss of damping in rolling in the wind-tunnel experiments was the long narrow aileron attached to a rectangular wing. (See reference 1, part XI.) In this particular case the performance characteristics were so poor that the device as tested could not be considered practical for application.

As noted in table I, the lateral-stability characteristics of the 5:1 tapered wing with the floating-tip ailerons are almost as bad as those on the conventional rigid 5:1 wing and are somewhat worse than those of the rigid rectangular wing. Inasmuch as the damping in rolling is lost at an angle of attack 2° below the angle for maximum lift, the airplane could not be safely maintained in flight above this angle even though the ailerons continue to give undiminished rolling moments. Flight tests of floating-tip ailerons on a tapered wing fitted to a Fairchild 22 airplane support this conclusion.

Wind-tunnel results with floating-tip ailerons showed a smaller adverse effect on the performance characteristics of the 5:1 tapered wing than on any of those tested. The effect of reducing the span and area of the rigid portion of a given wing is shown by the comparison of the performance criterions of the short 5:1 tapered wing, having an over-all aspect ratio of 6, with those tabulated for the conventional rigid 5:1 tapered wing, having the same over-all span and area. Here the maximum speed of the airplane will be hardly affected while the climb and maximum lift will be reduced, as indicated. Simply adding the tip portions to the normal-size wing will increase the parasite drag at high speed but, as shown by the tabulated criterions for this case, will probably slightly improve the climb.

SPOILERS

Spoilers in the form of small flaps or projections raised from the upper surface of the wing have presented attractive possibilities as lateral control devices because they give positive or favorable yawing moments and large rolling moments at the high angles of attack through the stall. (See fig. 9.) As spoilers giving apparently satisfactory rolling and yawing moments had been developed in the 7- by 10-foot wind-tunnel investigation (reference 1, part V), they were tested in flight on a Fairchild 22 airplane (reference 2). When the spoilers were first tried in flight, the pilots noticed that the airplane apparently did not react until the control stick had been given a medium amount of deflection, after which the rolling velocity suddenly built up to a much higher value than had been experienced with any previously tested control system. This characteristic made it impossible to perform smooth maneuvers requiring the coordination of the spoilers with the elevator or rudder and led to overcontrolling when an attempt was made to keep the wings level in gusty air. Closer inspection of the spoiler action, however, disclosed that for any spoiler movement there was actually an appreciable delay between the movement of the spoiler itself and the start of the desired rotation in roll of the airplane. In order to substantiate the pilot's findings, records were made of the rotation of the airplane in roll immediately following a movement of the stick and a specimen

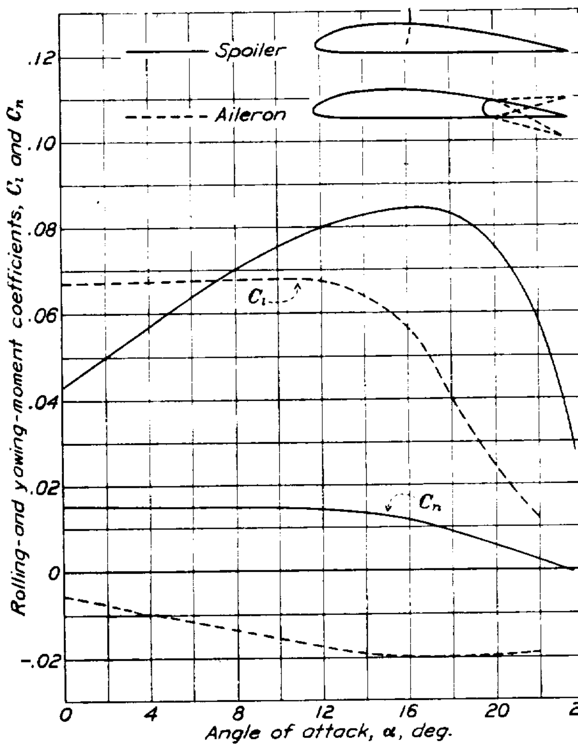


FIGURE 9.—Comparison of rolling- and yawing-moment coefficients obtained with ailerons and spoilers.

time history of the motion is shown in figure 10, together with similar information for other lateral con-

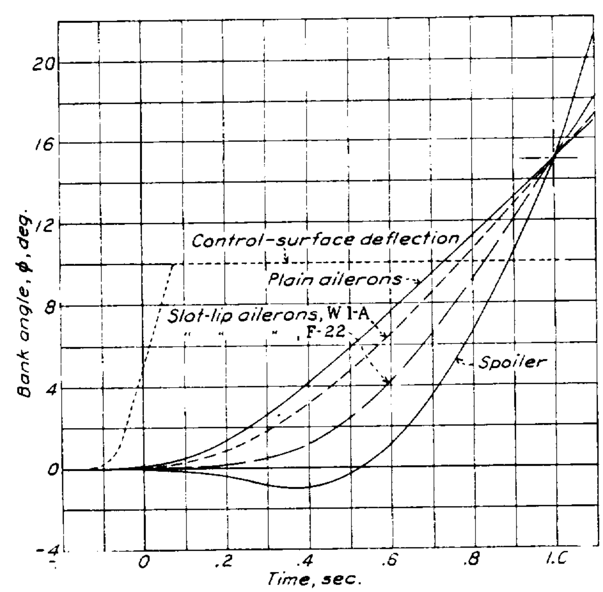


FIGURE 10.—Bank curves derived from flight records illustrating response characteristics of various lateral control devices.

trol devices including conventional ailerons. The records showed that the delay before rotation started in the desired direction was of the order of half a second. This lag seems surprisingly short to have much effect on the control obtained with spoilers, but apparently it is sufficient to prohibit the use of the spoilers close to the ground because of the danger of overcontrolling.

The lag of spoilers was then studied by means of a special hinged wing model of 4-foot chord mounted in the 7- by 10-foot wind tunnel (reference 12). This installation reproduced the conditions encountered in the flight tests. The tests with spoilers located in different positions along the chord of the wing showed that the lag was relatively large with the spoilers near the leading edge and became less after the spoiler was moved to the rear until it was zero for normal trailing-edge flap-type ailerons.

The spoiler located near the rear of the wing was found to act with a negligible amount of lag (less than one-tenth second could not be detected by the pilots) and seemed to give some promise of making a satisfactory lateral control device. Flight tests were therefore made of a retractable spoiler located 83 percent of the wing chord back of the leading edge which, because of its rearward position, was referred to as a "retractable" aileron. The aileron was made in the form of a plate curved in a circular arc to form a segment of a cylinder and was moved in and out through a slit in the upper surface of the wing and about an axis at the center of the cylinder. This arrangement produced no aerodynamic hinge moment and was found to operate satisfactorily in flight on a Fairchild 22 airplane (reference 3). The retractable aileron mounted on the assumed average airplane is number 16 in table I. The stick-force characteristic (zero force) is not the most desirable but could be brought up to a desired value either by the addition of a spring in the aileron linkage or by an off-center location of the hinge axis of the aileron. A large amount of control is available from ailerons of this type and the yawing characteristics are more satisfactory than those of conventional ailerons.

Combinations of conventional ailerons with spoilers located ahead of them and deflected simultaneously showed some promise in the wind-tunnel investigation (reference 1, part V) and were found to give satisfactory control free from lag when tested in flight on the Fairchild 22 airplane (reference 2). With the spoiler deflected in front of the aileron, the floating angle of the aileron is raised and, if properly developed, certain combinations seem very promising in regard to both yawing effect and stick force. Estimated characteristics of one such combination are given in table I, aileron 17.

Another possible combination that has been tested and may deserve further development is one in which two spoilers are located in tandem and deflected simultaneously. The tests with this arrangement (reference 12) showed that the lag of the combination was no greater than that for the rear spoiler alone, whereas the final rolling moment was the same as for the front one when used without a flap. Later tests indicate that spoilers located on the forward portion of the wing may be rendered ineffective by the action of a split flap. One other point has not yet been completely determined, namely, whether the rolling motion would get under way with sufficient acceleration immediately after the start. This point will be dealt with further in the next section on slot-lip ailerons.

SLOT-LIP AILERONS

Means for the elimination of the lag of spoilers were investigated in the 7- by 10-foot tunnel and it was found that the lag could be eliminated by providing a slot or passage through the wing back of the spoiler. This investigation has resulted in the development of what have been termed the "slot-lip" ailerons (references 8 and 12). The slot-lip aileron is a combination of a spoiler-type flap located on the upper surface of the wing and a continuously opened slot, the flap forming the upper portion or lip of the slot. The computed control performances for two arrangements of slot-lip ailerons in different positions along the chord of the wing are listed 18 and 19 in table I.

The slot-lip ailerons satisfactorily eliminate or reduce to a negligible value the actual lag intervening before the wing starts moving in the desired direction, and they give a very high maximum rate of rolling; but the rolling nevertheless increased less rapidly immediately after the start of the motion than with conventional trailing-edge flap-type ailerons. This condition is illustrated in figure 10, which includes curves from flight records of slot-lip ailerons on the Fairchild 22 airplane and slot-lip ailerons on the W1–A airplane. It will be noticed that with the W1–A the rate of roll increases nearly as rapidly as with conventional ailerons but with the Fairchild 22 the action was considerably more sluggish. The differences in the behavior of these two airplanes have been studied (reference 8) and it has been concluded that the superior response characteristics shown by the W1–A are due in large measure to the relatively great dihedral (5°) and to the smaller moments of inertia of this airplane. The secondary yawing action of the slot-lip ailerons is favorable, hence the dihedral effect increases the rolling action. Other differences favorable to improved response of the W1–A are: (1) The more rearward location of the aileron (0.30 c_w compared with 0.20 c_w tested on the Fairchild 22) and (2) the slightly greater size of the slot.

The lateral control with the slot-lip ailerons on the W1–A seemed satisfactory to the pilots, but on the Fairchild 22 it was found to be too sluggish and to give somewhat the same feeling as a slight amount of lag. This comparison, aided by several others of a pertinent nature, indicates that an additional point must be covered in a specification for a completely satisfactory lateral control dealing with the acceleration or rate at which the rolling increases during the first half second or so following the actual start. It may be stated in simple quantitative terms, applying to the conditions for the assumed average airplane, that the angle of bank one-half second after a sudden deflection of the controls should be at least one-third the angle of bank reached at 1 second. Thus, if a bank of 15° is reached in 1 second, at least 5° of this should be attained in the first half second.[2]

The sluggishness of the slot-lip ailerons is a great handicap in the method of comparison of control effectiveness used in the present report, in which a certain angle of bank must be obtained in a time of 1 second. Even though these ailerons give a high final rate of roll, excessively great deflections are required to attain an angle of bank of 15° in 1 second at a lift coefficient of 1.8, and the stick forces are excessively high. This particular disadvantage might be overcome by the use of a suitable aerodynamic balance but, even so, the sluggishness of the slot-lip ailerons might prevent them from being considered satisfactory if it were of the magnitude found on the Fairchild 22 instead of that found on the W1–A.

The sideslip accompanying a 15° bank in 1 second is negligible with the 0.55 c_w slot-lip ailerons in the usual flight range with unflapped wings. With more forward locations the yawing moment becomes decidedly positive, resulting in outward sideslip. Because of the action of the slots at high angles of attack, the damping in rolling is retained to an angle of attack beyond that for maximum lift coefficient and, for this reason, it should not be difficult to design an airplane incorporating these ailerons in such a manner that lateral control and stability would be reasonably satisfactory at all angles of attack that could be maintained in flight. The continuously open slot, however, results in a high drag, which reduces the high-speed and climbing performance to a noticeable extent. The drag is less for the rear positions of the slot-lip ailerons and a special investigation has been made in the 7- by 10-foot tunnel to develop slots with reduced drags. Some success has been attained but, considering the best results to date, these ailerons do not seem suitable for modern high-performance airplanes.

LATERAL CONTROL WITH HIGH-LIFT FLAPS

Since the inception of the research program of reference 1, wing flaps have come into very general use and have further complicated the problem of lateral control. In steady flight ordinary ailerons give rolling moments that vary almost inversely with the lift coefficient; hence, wings equipped with high-lift devices require relatively large control surfaces. The installation of an effective flap then becomes more difficult.

Another problem introduced by the use of high-lift devices concerns the adverse yawing moment of the ailerons. The ratio of induced yawing to rolling moment increases (adversely) in direct proportion to the lift coefficient. Furthermore, the effect of a given yawing moment on the rolling control is usually greater with flaps in use on account of the increased dihedral effect due to the flap. Thus it appears almost necessary to use some device that causes large changes of profile drag resulting in a favorable component of yawing moment or to use wings with washout at the tip portions (partial-span flaps) so that the induced yawing moment is reduced. Many of the devices developed in reference 1 for use with full-span flaps show satisfactory yawing moments on account of the profile-drag increments caused. Comparisons of a number of the most promising devices have been made and are listed in section B of table I.

Plain ailerons on wings with partial-span flaps.—On account of the general use of partial-span split flaps with ordinary ailerons, some tests of this arrangement were made in the 7- by 10-foot wind tunnel (reference 7). The tests were made with tapered wings because they represent the most efficient application of the arrangement and are most used in practice. The most interesting result of these tests was the small loss of maximum lift coefficient entailed by the substitution of ailerons for the tip portions of the flap, particularly in the case of ailerons 21 and 23 as listed in table I, where only 30 percent of the semispan was used for the aileron portion. The indicated reduction amounted to less than 10 percent of the maximum lift shown by the same tapered wings with full-span split flaps. The reduction was about the same for the two taper ratios tried. It will be noted that the 5:1 tapered wing gave more efficient control as regards stick forces under all conditions. In each case the stick force is slightly less for the longer ailerons, although of course the wings with shorter ailerons showed better performance characteristics. Both sizes of ailerons on the 5:1 tapered wings showed a marked diminution of effectiveness above about 10° angle of attack, presumably due to flow separation at the tip portions.

The deflection of the partial-span flap introduces a large relative washout of the aileron portions so that at a given over-all lift coefficient the ratio of yawing to rolling moments is less with flap down than with flap neutral. It will be noted that the tabulated values of sideslip remain about the same at $C_L=1.8$ as at $C_L=1.0$. The sideslip at $C_L=1.0$ would have been appreciably less than indicated if a flap-down condition had been assumed here.

Although the lateral-stability characteristics of the highly tapered wing are unfavorable, there are indica-

[2] As mentioned previously, in order to simplify the computations and to make possible a comparison with flight records, the starting time has been arbitrarily taken as the instant at which the control surfaces reached half their final deflection.

tions that the use of a partial-span flap may not aggravate the instability in every case. The results of the aileron tests, as well as visual observations of the flow by means of tufts, show that the effect of the upwash at the tips introduced by lowering the flap may be compensated by a strong spanwise flow, which inhibits the stalling of these portions. The indications are that the angle of attack for autorotational instability would be about the same with the flaps as without for the wings tested, although rolling experiments were not tried.

Plain ailerons with retractable flap.—A plain aileron with a split flap retracting ahead of it was developed as a means of control with a full-span flap. This device has been tested in flight with a modified Fairchild 22 airplane and is one of the few lateral control systems incorporating full-span flaps that has proved entirely satisfactory in flight (reference 3). This device is so designed that the retracted flap does not interfere with the ailerons in any way and hence the control characteristics with flap neutral are those of plain ailerons. With the flap deflected, however, the characteristics are similar to those of the upper-surface ailerons tested in the 7- by 10-foot wind tunnel (reference 1, part XII).

Although the deflected flap is in such a position as to shield the under surface of the ailerons entirely, it was observed in the tests that the ailerons in this condition were nearly as effective as conventional ailerons with unsealed gaps. The effectiveness of downward deflection, however, falls off rapidly at an angle of about 8°.

The rolling-moment characteristics of the plain ailerons with retractable flaps are such as to favor a differential motion, since the upgoing aileron is more effective than the downgoing one at high lift coefficients. The hinge-moment characteristics are, however, distinctly unfavorable for this mode of operation inasmuch as the ailerons show a downward floating tendency with the flap down. Relatively large deflections of the ailerons are required to meet the control requirements at low speed on account of the shielding effect of the flap, and consequently a relatively high gearing ratio of ailerons to control stick is needed. The result is that the stick forces required for the specified banking control are somewhat higher than those for conventional ailerons throughout the flight range. These forces (see aileron 24, table I) are well within the desirable range for the Fairchild 22 airplane, although they indicate undesirably high values for larger airplanes.

The yawing action of these ailerons is about the same as that of the conventional ailerons with partial-span flaps. Although the induced yawing moment of the ailerons with the full-span flap is greater than that with the partial-span flap, the ailerons cause larger compensating changes of profile drag.

Several possible means of improving the control-force characteristics of these devices suggested themselves. The device listed next in table I (aileron 25) shows the calculated effects of such improvements. First, the span of the aileron was increased to what has previously been found the most efficient value and the chord of the aileron was reduced as much as seemed practical. Second, it was assumed that a trailing-edge tab (0.02 c_w bent down 15°) was attached to the aileron so as to avoid the downward-floating tendency. It was assumed that lowering the flap caused the same change in floating angle with the tab as without. Since the deflection of the flap caused a large change in the floating position of the aileron, it was desirable to change the balancing characteristics of the differential with flap deflection. Consequently, it was assumed that the differential cranks were rotated into new positions as the flap was deflected. The resulting stick forces tabulated give an indication of the improvement that might be effected by such development of the device.

Retractable ailerons (spoilers).—Tests of spoilers (reference 12) showed that for locations behind about 80 percent of the wing chord the lag in rolling action would probably be negligible. Flight tests were subsequently made of a Fairchild 22 airplane equipped with a curved-plate spoiler that moved edgewise into and out of the wing through a narrow slit in the upper surface at 83 percent of the airfoil chord. This plate was arranged to rotate about a hinge at the center of curvature, so that the air pressure (being normal to the plate) caused no resultant hinge moment. The test airplane incorporated a full-span split flap and, inasmuch as the downward motion of the spoiler took place entirely within the wing, the flap and spoiler did not interfere.

The flight tests showed very promising results, although the feature of zero hinge moment was not found especially desirable. Angular-velocity and control-position records taken simultaneously in flight showed no definite lag or sluggishness in the response to control movements. (See reference 3.) The devices as tested (0.15 c_w by 0.50 $b/2$) were somewhat larger than necessary to give the assumed satisfactory degree of control. As is indicated in the table, a maximum deflection causing a 7.4 percent c_w projection of the spoiler should be sufficient for control in the flap-down condition.

An important advantage of the retractable ailerons (aside from their advantage in permitting the use of a full-span flap) is that they give small favorable yawing moments throughout the greater portion of the flight range. At high lift coefficients with the flap in use, however, small adverse yawing moments result. (See reference 13.)

Although the deflected spoiler causes quite an increase of profile drag, it is not expected that the incidental deflections required for control in normal flight would appreciably affect the performance. The performance criterions listed are, of course, for undeflected controls.

External-airfoil flap-type ailerons.—The external-airfoil (Junkers or Wragg) type flap has been studied as a possible means for improving the take-off and ceiling characteristics of airplanes in addition to providing the high-lift features of ordinary and split flaps. As this device showed promise of improved performance, several methods of securing lateral control with such a flap have been studied.

A simple method of providing lateral control with full-span external-airfoil flaps is to move the flaps themselves independently as ailerons. (See reference 10.) Thus the ailerons are used simultaneously as a high-lift device and to provide rolling moments without sacrificing a special part of the wing span. In order to employ these flaps to their best advantage, it is necessary to deflect them downward over the entire wing span, thereby avoiding excessive induced drag. The action of the flaps deflected downward as ailerons is similar to the action of ordinary ailerons with droop. The external-airfoil flaps show a superiority over ordinary flaps for this purpose, however, in that they retain their lift-changing effectiveness at greater downward deflections (in excess of 20°).

Aileron 27 in the table is an arrangement of these flaps whereby the entire span is deflected downward 20° and the semispan portions are moved differentially from this downward position to provide rolling control.

This arrangement was tested in flight with the Fairchild 22 airplane and was found to give unsatisfactory yawing characteristics, although the rolling moments seemed to be ample. The computations made for the average airplane indicated an adverse sideslip of 10° accompanying a 15° bank at low speed with the flaps down.

A possible way of improving the adverse-yaw characteristics of these devices is to make use of the effect of washout. This method was used in the case of aileron 28, where the flap was considered to extend unbroken over the middle portion of the wing with the parts of the flap used as ailerons covering the outer 50 percent of the semispan portions. Wind-tunnel tests (reference 10) showed that, with the inner portion down 30° and the outer, or aileron, portions down only 10°, the performance criterions were about the same as with the whole flap down 20°. This change reduced the yawing effect considerably, as shown by the table, although the sideslip is still somewhat worse than is the case with most of the other devices.

When the stick forces and deflections for these two arrangements are compared, it will be noted that the deflection required with the full semispan aileron is almost as great as that required when only half the flap is used for control. This fact is partly accounted for by the difference in yawing effects.

In the low-speed conditions ($C_L=1.8$) the ailerons are lowered 20° in one case and 10° in the other and the effective floating angles are thereby increased by these amounts. This fact introduces a difficulty into the design of a suitable differential linkage. A linkage designed to accommodate the floating tendency with flaps neutral will overbalance when the flaps are deflected. In the computations it was assumed that the additional floating tendency was neutralized by a long spring that came into action as the flaps were lowered.

The external-airfoil flaps permit high lift coefficients to be attained without excessive profile drag. The advantage over a split flap begins to be apparent at lift coefficients in excess of 0.7, aiding the take-off and the low-speed climb but hardly affecting the maximum rate of climb. Hence, in this particular case, the performance criterions listed in table I do not fully indicate the differences to be expected with these devices.

Ailerons with external-airfoil flaps.—A logical extension of the development of the slot-lip aileron has led to a device in which the aileron forms the lip of the slot between an ordinary external-airfoil-type flap and the main wing. (See aileron 29, table I.) This arrangement avoids the excessive drag entailed by other forms of slot and, on account of the rearward position of the aileron, should give good response characteristics (except, possibly, under certain conditions noted later).

The device as tested (see reference 9) comprised an aileron 0.12 c_w wide and $b/2$ long. The tests showed that, in general, the effectiveness of the aileron was reduced by the presence of the flap, in accordance with the theoretical consideration that any change in slope of the wing section ahead of the trailing edge is less effective than a corresponding change at the trailing edge itself. When the flap is lowered, however, an upward deflection of the aileron apparently causes separation of flow over the flap, thus greatly reducing the lift and developing a large rolling moment. With the flap down 30° this change occurs at the beginning of the aileron deflection, while at intermediate flap deflections the change occurs at greater up aileron angles. This more or less sudden change of conditions, in addition to giving a large increase of rolling moment, also caused a reduction or a reversal of hinge moment; hence, the device may be impracticable for use at intermediate flap settings. (See reference 9.)

In the device as shown in table I the downward deflection of the aileron is limited by the presence of the flap nose to a maximum of about 7°, and it is consequently necessary to use a differential movement. Change of setting of the flap has a pronounced effect on the floating angle of the aileron. With the flap set at 30° a differential giving no more than 7° downward deflection of the aileron will be overbalanced by this floating tendency. In the computation it was assumed that a spring tending to turn each aileron downward (with a torque of 8.7 foot-pounds acting at the aileron hinge) was brought into action by lowering the flap. With the flap neutral the floating angle of the aileron is too small for satisfactory balance, although wind-tunnel tests showed that it could be effectively increased by a tab. Consequently, the device was assumed to incorporate such a tab (0.018 c_w, down 5°) and the spring tension was adjusted to accommodate the effect of the tab with flap down.

The resulting stick forces, together with the deflections required for control, appear in the table. It will be noted that the greatest deflection required is that at $C_L=1.0$. In this condition the aileron does not produce the previously discussed change in flow over the flap. At $C_L=1.8$ the deflection required is small because a small upward movement of the aileron in the flap-down condition produces a large rolling moment. The yawing effect is adverse but is not excessive.

The performance characteristics of this wing (with the N. A. C. A. 23012 airfoil flap) are somewhat better than those of the two wings previously considered, which had flaps of Clark Y section.

II. ANALYSIS OF CONVENTIONAL FLAP-TYPE AILERONS

The practical advantages of plain ailerons are well known, and, since they are universally used in more or less modified form, the following section is devoted to an analysis of factors involved in their design.

One of the conclusions of the lateral control investigation has been that no decisive benefit was to be gained from a device that continued to give rolling moments when the major outer portions of the wings were stalled. If stalling of the aileron portions of the wing is prohibited, plain ailerons or other devices located near the trailing edge of the wing will retain their effectiveness.

If the loss of rolling effect on a stalled wing is discounted, it appears that the primary disadvantage to be associated with plain ailerons is their adverse yawing effect. For this reason the yawing action of plain ailerons will be rather fully analyzed.

ROLLING MOMENT

For the purpose of calculating the coefficients of rolling and yawing moment, the effect of a deflected aileron may be ascribed to a change of angle of attack of the wing sections comprising the aileron portions. Thus, the localized effect of the deflected aileron is measured by the change in the angle of zero lift. This change is proportional to the angle of deflection of the aileron for deflections below about ±20° and the factor of proportionality (denoted by $\Delta\alpha/\Delta\delta$) depends on the chord of the aileron. Thus, the plain flap-type aileron is considered merely as a device for changing the angle of attack. The section lift increment is not used to characterize the effect of the flap because this increment cannot, in general, be specified, being dependent on the plan form of the wing. The effective change in angle of attack per unit change of flap deflection is, however, theoretically independent of the aspect ratio and the plan form.

Figure 11 summarizes the results of a number of wind-tunnel experiments with plain flaps (references 14, 15, and 16) and shows the measure of flap effectiveness ($\Delta\alpha/\Delta\delta$) as a function of the relative flap chord. A curve predicted by wing-section theory (reference 17) is also shown for comparison. The surprisingly powerful effect of a narrow flap should be noted. Thus, deflecting a 0.20 c_w flap is about half as effective as deflecting the entire wing section.

Since the effective angle of attack of a wing section is a linear function of the camber (reference 17), the curve of figure 11 may be used to predict the effect of a multiply hinged flap, such as an aileron equipped with a balancing tab. The combined effect of a succession of bends along the wing section may be found by calculating the separate effects of each bend and adding them. Thus the effect of a 0.20 c_w aileron equipped with a 0.05 c_w tab is (using values from fig. 11)

$$\Delta\alpha=0.51\delta_a+0.21\delta_t \tag{2}$$

where δ_a is the deflection of the aileron with respect to the wing and δ_t is the deflection of the tab with respect to the aileron. This simple relation should not be expected to apply beyond ±20° deflection and, in the case of very narrow tabs, beyond about ±15°

Deflected ailerons thus cause, in effect, a discontinuous change of angle of attack across the wing span. The lift change caused by the ailerons cannot be discontinuous, however, because of the natural equalization of pressure along the span. Ailerons covering only a portion of the span influence the lift at every spanwise point and this effect appears to be satisfactorily predicted by the airfoil theory. Calculations of the effects of ailerons based on this theory have been made, the most extensive series being reported in reference 18. Figure 12 shows the rolling-moment coefficient C_l caused by a 1° difference in angle of attack of various right and left portions of a rectangular wing of aspect ratio 6. The abscissa of this diagram represents a semispan of the wing with the midspan point at the origin and the tip at the point 1.0. The ordinate gives directly the rolling- (or yawing-) moment coefficient due to a unit change of angle of attack extending from the point indicated on the abscissa out to the tip. The rolling effect of two ailerons is twice as great as that of a single one and hence the difference of the increments of equivalent angle of attack, as indicated, should be used. The rolling moment is not appreciably changed by differential deflection.

The curves give the values predicted by the theory and the points indicate values obtained in various experiments as noted on the figure. The wing-section characteristic $\Delta\alpha/\Delta\delta$ of the devices tested was determined from figure 11.

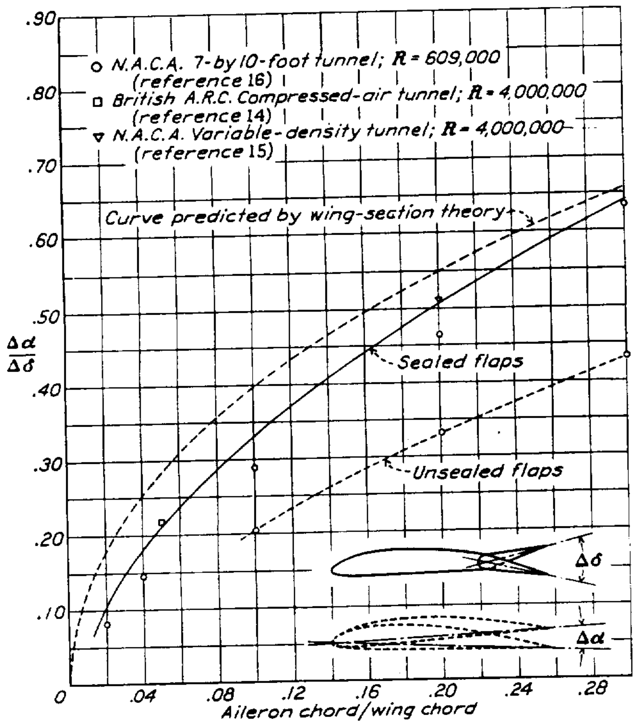


FIGURE 11.—Change of effective angle of attack of a wing section per unit change of flap angle. Plain flaps of various chords at small deflections; $\delta < \pm 20°$.

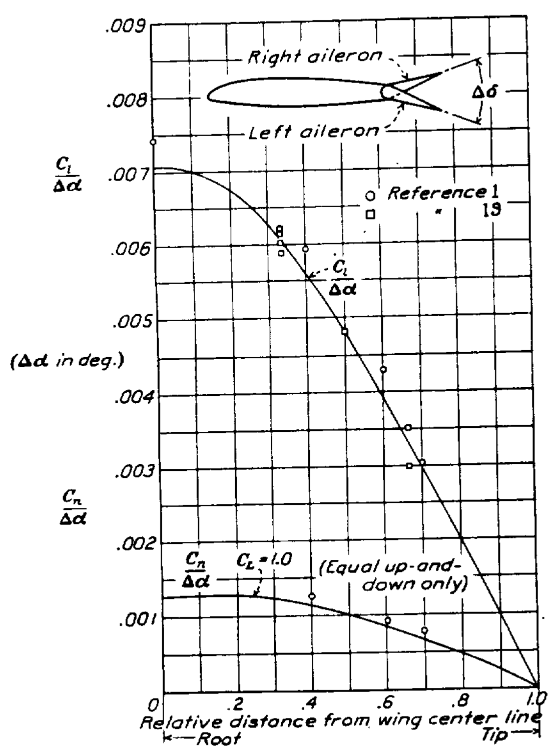


FIGURE 12.—Variation of rolling- and yawing-moment coefficients with aileron span and a comparison of theoretical and experimental values. Rectangular wings; $b^2/S=6$; $\Delta\delta < \pm 20°$.

The rolling-moment characteristics of the plain 0.25 c_w by 0.40 $b/2$ sealed ailerons (aileron 1 of table I) were calculated with the aid of figures 11 and 12. Reference to figure 11 shows that the equivalent change in angle of attack produced by a 0.25 c_w sealed flap is 57.5 percent of the angle of deflection of the flap. Thus, a deflection of $\pm 7.4°$ (see table I) is equivalent to a change in angle of attack of

$$0.575 \times 7.4° = 4.26° \tag{3}$$

or a difference of angle of the right and left aileron portions of 8.52°. According to figure 12 the rolling-moment coefficient per degree of this difference for a 0.40 $b/2$ aileron portion extending to the wing tip is 0.0039; hence, the coefficient predicted is

$$C_l = 8.52 \times 0.0039 = 0.0332 \tag{4}$$

Working charts for predicting the rolling moment of plain ailerons of any size on monoplane wings of various aspect ratios and different degrees of taper are given in figure 13. In order to use these charts it is necessary to ascertain from figure 11 the section characteristic $\Delta\alpha/\Delta\delta$, which is a function of the relative chord of the aileron. The charts may be used for differential ailerons merely by taking the difference of angle of attack of the right and left aileron portions. The theoretical rolling moment is independent of any initial washout of the wing sections along the span; hence, the rolling-moment curves are applicable to wings with partial-span flaps. The charts cannot be used with devices that change the slope of the lift curve nor for excessive deflections that introduce disturbed air flow. In this connection it appears that a deflection of plain ailerons involving disruption of the air flow is inefficient from considerations of stick force.

It will be noted that two sets of curves are given for tapered wings. The solid lines apply to ailerons that are not tapered with the wing, i. e., ailerons of constant actual chord. For this type the change of equivalent angle of attack should be calculated on the basis of the wing-tip chord (whether or not the aileron extends to the wing tip). The long-dash curves are for the particular case in which the aileron chord is a constant proportion of the wing chord along the span, in which case the change of equivalent angle of attack does not vary along the aileron portion. The additive effect of an element of aileron covering any spanwise portion of the wing may be determined from the increment of the $C_l/\Delta\alpha$ curve over that portion. Although the curves of figure 13 show increasing rolling-moment coefficients with increased aspect ratios of the wings, the control requirement (rolling-moment coefficient for a given banking effect) also increases with aspect ratio and, on account of the damping, in nearly the same way as does the coefficient. (See reference 4.) In general, it may be said that the relative proportions of the ailerons should not be reduced on account of increased aspect ratio.

YAWING MOMENT

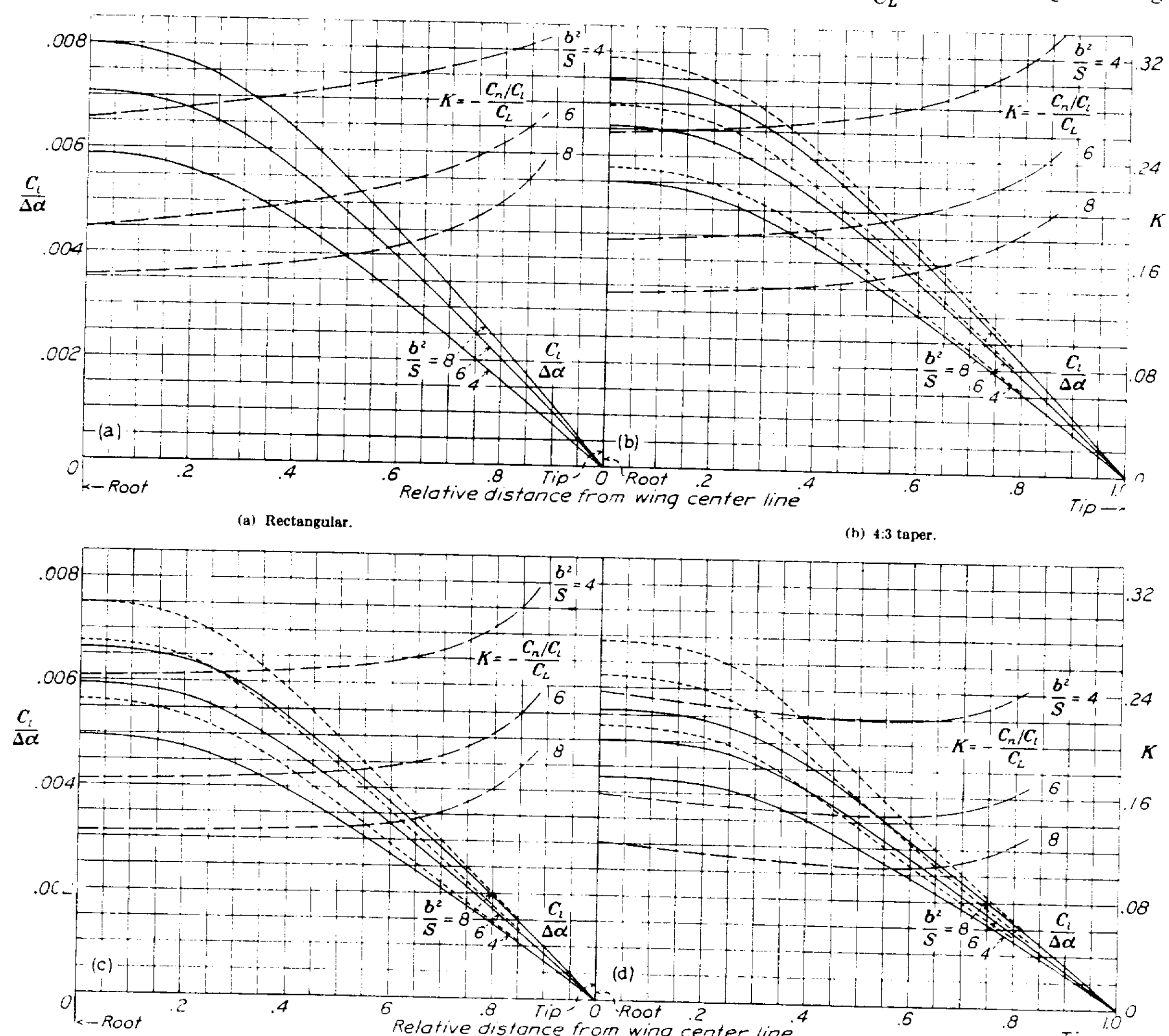
Yawing moment with equal up-and-down deflection.—The results of experiments indicate that the primary source of adverse yawing moment given by plain ailerons at small deflections is the theoretical, or induced, yawing moment. The production of rolling moment results in an induced twisting flow analogous to the downwash in direct lift. The yawing moment arises from the resultant inclination of the supporting lift vectors along the span. If the wing is supporting no lift, the production of rolling moment by equal and opposite lift increments on the two wing halves will not result in a yawing moment because the lift increment vectors are all inclined backward by the induction, resulting in a drag. Hence, only the interaction of an initial lift and a rolling moment give rise to an induced yawing moment.

A more specific treatment of this theory is given in reference 18. The formula for yawing moment that results for equal up-and-down deflections is

$$C_n = KC_L \times C_l \tag{5}$$

where K is a factor dependent on the aspect ratio and the plan form of the wing, and to some extent, on the spanwise position of the aileron. It is interesting to note that with a given equal up-and-down aileron deflection the induced yawing *moment* is the same throughout the speed range, while the rolling moments and the stabilizing factors are greatly reduced at the lower speeds.

Figure 12 gives a comparison of theoretical and experimental values of $\frac{C_n/\Delta\alpha}{C_L}$ for a rectangular wing of



(a) Rectangular. (b) 4:3 taper. (c) 2:1 taper. (d) 4:1 taper.

FIGURE 13.—Charts for calculation of rolling and yawing moments of plain ailerons, showing the effects of span and spanwise location of ailerons with straight and tapered wings.

aspect ratio 6. Deviation from the theory is to be expected at excessive deflections of ordinary ailerons and with special types of devices, since important changes of profile drag may be introduced. If complete wing section data are available, however, the profile-drag part of the yawing moment may be readily estimated.

As in the case of rolling moment, the yawing moment of an aileron at any spanwise position may be calculated by taking the difference of ordinates at abscissas corresponding to the ends of the aileron. Unlike the rolling moment, however, the yawing moment of differential ailerons is not the same as that of ailerons with equal deflections. In the general charts given in figure 13 the ratio of yawing to rolling moments at $C_L=1.0$ is given rather than $C_n/\Delta\alpha$. In this case the differences between two points cannot be used directly to give the yawing moment of an aileron extending between these two points. The yawing moment caused by an aileron ending inboard of the tip may be found, however, by taking the difference of the yawing moments given by two ailerons, one extending from the inboard end of the actual aileron to the wing tip and the other extending from the outboard end to the tip. The straight and tapered ailerons should give yawing moments in practically the same ratio to the rolling moment; hence, only a single set of values of $K=-\dfrac{C_n/C_l}{C_L}$ is given.

Referring again to the 0.25 c_w by 0.40 $b/2$ plain aileron (aileron 1) of table I, it is found that the ratio of yawing- to rolling-moment coefficients for this case is

$$\frac{C_n}{C_l}=-0.216 \tag{6}$$

at $C_L=1.0$. (See fig. 13.) At the deflection given the rolling-moment coefficient previously found is

$$C_l=0.0332 \tag{7}$$

Hence, the yawing-moment coefficient at $C_L=1.0$ is

$$C_n=-0.216\times0.0332=-0.0072 \tag{8}$$

The values of both yawing- and rolling-moment coefficients for these ailerons having been obtained, it is now possible to calculate their rolling effectiveness by means of figure 3. The wing loading of the average airplane assumed in table I is 9.4 pounds per square foot; hence, at $C_L=1.0$ the banking effect of a rolling moment of coefficient 0.01 acting for 1 second is

$$\left(\frac{\phi_1 b}{2}\right)_{C_l=0.01}=1.42 \text{ feet} \tag{9}$$

and for a rolling-moment coefficient of 0.0332

$$\frac{\phi_1 b}{2}=1.42\times3.32=4.7 \text{ feet} \tag{10}$$

The effect of the yawing moment of coefficient -0.0072 is calculated in the same way, i. e.,

$$\frac{\phi_1 b}{2}=-0.72\times0.65=-0.47 \text{ foot} \tag{11}$$

The effect of these rolling and yawing moments applied simultaneously is

$$\frac{\phi_1 b}{2}=4.7-0.47=4.23 \text{ feet} \tag{12}$$

Thus, deflecting the ailerons suddenly to $\pm7.4°$ causes a 4.23-foot displacement of the wing tips in 1 second. The angle of bank for the average airplane ($b/2=16$ feet) is

$$\phi_1=\frac{\frac{\phi_1 b}{2}}{\frac{b}{2}}\times57.3=15° \tag{13}$$

as appears in the table.

Yawing moment with differential deflection or droop.—The effect of an unequal movement of the ailerons may be taken into account by considering an equivalent equal up-and-down deflection from a mean upward position of the ailerons. Thus, deflections of 15° up and 5° down may be considered as equivalent to 10° equal up-and-down from a mean position 5° up. Inasmuch as a differential deflection of the ailerons changes the mean lift of the wing, figure 13 cannot be used without correction to calculate the yawing moment due to unequal deflection. As was brought out in the preceding discussion, the yawing moment is caused by the interaction of the wing lift and the induced flow caused by the rolling moment. Hence, the yawing moment incident to a given rolling moment depends on the distribution of the basic or symmetrical part of the lift. The basic lift distribution upon which the yawing moment depends is, then, the distribution for a wing with both ailerons raised. The adverse yawing moment will, in this case, be reduced because of the lessened lift over the tip portions. For the conditions following sudden aileron deflections the average upward movement of both ailerons will entail an actual reduction for a short time of the lift of the wing without correspondingly increasing either the flight speed or the angle of attack. The conditions will, of course, be different for steady flight with ailerons held over. For practical purposes it is sufficient to calculate an increment of C_n/C_l due to the increment of lift produced by the symmetrical droop or uprigging of both ailerons. This increment would be the yawing moment incident to a unit rolling moment when the entire lift of the airfoil was due to the droop of the ailerons. The ratio of yawing to rolling moment thus found will be a constant additive contribution to equation (5) at all lift coefficients.

Figure 14 shows the reduction of the ratio of adverse yawing to rolling moment in terms of the reduction of over-all lift coefficient for a rectangular wing of aspect ratio 6. The experimental points indicated were derived by taking the differences of yawing moment measured with equal up-and-down deflections and up-only deflections and dividing these differences by the measured reduction in total lift coefficient caused by the up-only deflection.

If C_L is the lift of the wing with ailerons undeflected and $\Delta\alpha_m$ is the equivalent angle of washout of the

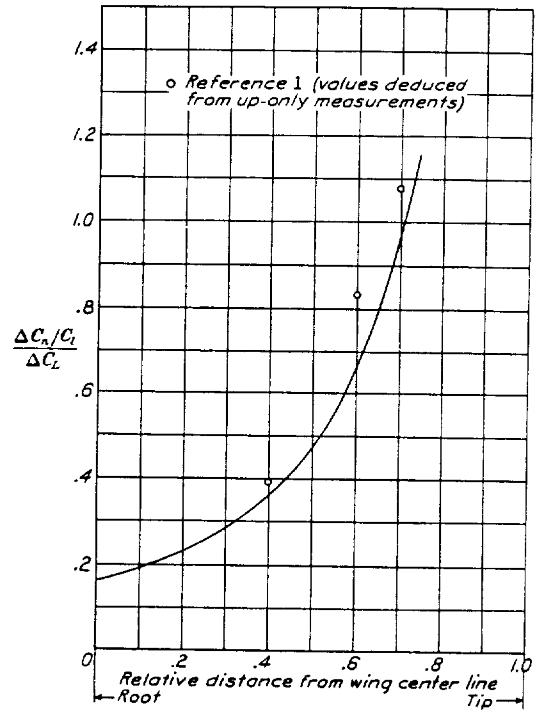


FIGURE 14.—Increment of induced yawing moment due to differential deflection of ailerons; ΔC_L is the reduction of lift coefficient due to differential deflection. Rectangular wing; $b^2/S=6$.

aileron portions introduced by the unequal aileron deflections, then

$$\Delta\frac{C_n}{C_l}=\kappa\Delta\alpha_m \tag{14}$$

since the reduction of lift is proportional to $\Delta\alpha_m$. The factor κ, like the factor K, depends on the wing plan form and the relative length of the aileron portion.

Figure 15 shows theoretical values of κ for wings of aspect ratio 6 and various plan forms. It should be remembered that C_L as used in equation (14) is the lift coefficient with ailerons undeflected. Correction of the values given in figure 15 for wings of different aspect ratio may be made by considering that κ is very nearly inversely proportional to the aspect ratio.

It is evident that the foregoing remarks apply equally as well to wings having washout at the tips or to wings with partial-span flaps. For wings with partial-span flaps $\Delta\alpha_m$ is simply the reduction of the effective angle of attack at the tips due to removal of the tip portions of the flap. It should be remembered that droop of the outer portions (negative $\Delta\alpha_m$) increases the adverse (negative) yawing moment while washout (positive $\Delta\alpha_m$) decreases it.

The increment of yawing moment due to the sum of two distributions of droop or washout is equal to the sum of the increments associated with each separate distribution. This property may be used to compute quite accurately, though not exactly, the yawing

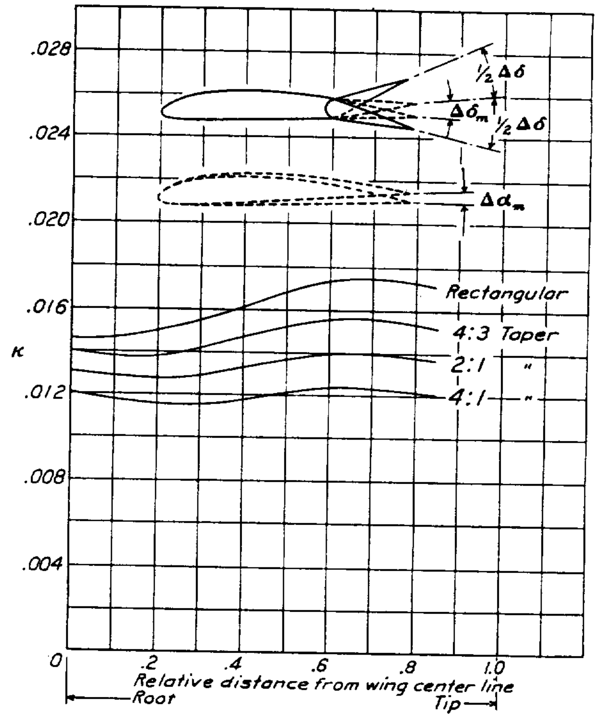


FIGURE 15.—Ratios for calculating additional induced yawing moments of differential ailerons or ailerons on wings with washout; $b^2/S=6$; $\Delta\alpha_m$ is in degrees.

$$(1+\Delta)\frac{C_n}{C_l}=-KC_L+\kappa\Delta\alpha_m$$

moment of differential ailerons that end inboard of the wing tip.

CONTROL FORCES

Hinge moment.—The available experimental data indicate that the hinge-moment coefficient C_h of an ordinary aileron can be treated with sufficient accuracy as a characteristic of the wing section, that is, as a characteristic independent of the plan form of the aileron or the wing. An average experimental value for the slope of the hinge-moment curve against deflection is

$$\frac{dC_h}{d\delta}=-0.0085 \text{ per degree} \tag{15}$$

for sealed ailerons of chord c_a and span b_a, where

$$C_h=\frac{\text{hinge moment of aileron element}}{qc_a{}^2b_a}$$

Thus, the actual hinge moment at a given deflection varies as the aileron span and as the square of the aileron chord.

Strictly speaking, the hinge moment of a deflected flap should be calculated in two parts. The primary part arises from that component of the distributed pressure change which does not contribute to the lift of the airfoil section. Since no lift is involved, this component is independent of the aspect ratio. The second component of the hinge moment, proportional to the lift change, is subject to the ordinary aspect-ratio correction. The correction is, however, small except for wide flaps.

Some additional considerations arise in the application of aileron hinge moments to the calculation of control force. The angular travel and the length of the control stick (or radius of the control wheel) are limited in practice. Thus, ailerons requiring large deflections must be geared to the control stick or wheel in a high ratio. In the case of the average airplane the total circumferential movement of the end of the control stick was assumed to be 0.73 foot in the case of each of the control devices. This value corresponds to a $\pm 25°$ deflection of a 20-inch stick corresponding to that available in the Fairchild 22 airplane.

If reference is made to the tabulated results for aileron 1, it is seen that the total deflection necessary to insure the assumed satisfactory degree of control ($\phi_1 = 22.5°$ at $C_L = 1.0$, in this case) is $\pm 11.2°$. The work of deflecting ailerons of chord c_a and span b_a is

$$\frac{dC_h}{d\delta}\,\delta\,\frac{\delta}{57.3}\,qc_a{}^2b_a = -0.0085 \times \frac{11.2 \times 11.2}{57.3}$$
$$\times 9.4 \times (0.25 \times 5.3)^2 \times 0.4 \times 16$$
$$= 1.97 \text{ foot-pounds} \tag{16}$$

The control force is equal to twice the total work divided by the linear travel of the end of the stick, or

$$\text{Stick force} = \frac{3.94}{0.73} = 5.4 \text{ pounds} \tag{17}$$

The stick force at the partial deflection required for $\phi_1 = 15°$ is

$$2.31 \times \frac{\delta 15°}{\delta 22.5°} = 2.31 \times \frac{7.4°}{11.2°} = 3.6 \text{ pounds} \tag{18}$$

These simple relations apply, of course, only to linear variation of the hinge moment and to nondifferential gearing.

Differential linkages.—It appears that a differential linkage can, when properly designed, be a very effective means of reducing the operating force of flap-type ailerons (reference 11). The reduction of operating force is accomplished by taking advantage of the upfloating tendency of the ailerons. With differential linkage the ailerons on opposite tips of the wing begin to move at different rates immediately after they are deflected from neutral, the downgoing aileron moving more slowly than the upgoing one. The upgoing aileron thus has the greater mechanical advantage at the control-stick connection. It is evident that the reduced upward pressure of the upgoing aileron is partly compensated by its increased mechanical advantage and that the increased upward pressure on the downgoing aileron is also partly compensated by its reduced mechanical advantage. At a certain deflection the downgoing aileron reaches dead center and, regardless of its aerodynamic pressure, cannot contribute to the stick force; if the upgoing aileron is then at the floating angle (i. e., angle of zero hinge moment), the stick force will be zero.

Ordinary ailerons show nearly straight-line hinge-moment curves $\left(\frac{dC_h}{d\delta} = -0.0085\right)$ and in this case the balancing effect of a given differential linkage depends only on the upfloating angle. A formula for a differential motion that gives zero operating force over a range of deflections may be obtained by writing the expression for the work of deflection of the ailerons and equating it to zero at every point.

$$\delta_d = \sqrt{(\delta_{u_f} + \delta_u)^2 - 2\delta_u{}^2} - \delta_{u_f} \tag{19}$$

where δ_u and δ_d are the upward and downward deflections of the ailerons and δ_{u_f} is the floating angle measured upward from the neutral position. A practical limitation of this formula is reached when $d\delta_d/d\delta_u$ approaches -1, for then both ailerons begin to move in the same direction and at the same rate.

It should be appreciated that a differential designed in accordance with equation (19) will give complete balance at the specified floating angle. It is, however, considered desirable not to eliminate completely the control force at any flight condition, as the pilots' feel of the control would be taken away. This condition can be avoided by designing the linkage for a fictitious floating angle somewhat higher than the maximum actually reached in flight. If $\Delta\delta_{u_f}$ is the difference between the floating angle at which the differential gives complete balance and the actual floating angle of the aileron in the given flight condition, the resultant stick coefficient C_{h_s} will be

$$\frac{\text{Stick moment}}{qc_a{}^2b_a} = C_{h_s} = \Delta\delta_{u_f}\,\frac{dC_h}{d\delta}\left(\frac{d\delta_u}{d\theta} + \frac{d\delta_d}{d\theta}\right) \tag{20}$$

where θ is the angular deflection of the control stick.

In any given case the stick force can be balanced out at only one angle of attack and, in general, the balancing effect diminishes as the angle of attack is reduced. Hence, if the stick force is made to become zero at an angle of attack above maximum lift, overbalance of the control in normal flight will be avoided.

A more or less complicated mechanical linkage that would give aileron movements approximating equation (19) could be devised. The ordinary simple linkage consisting of two properly set cranks connected by a rod can, however, be arranged to give the desired motion with close approximation, and such an arrangement will be given primary consideration.

Such a simple linkage can be made to satisfy two conditions for a minimum stick force. Figure 16 shows a type of stick-force curve that satisfies two very simple criterions. First, the slope of the curve is zero at the beginning of the deflection and, second, the resultant stick force is zero at a stick deflection corresponding to the floating angle of the up aileron. As was stated earlier, the latter condition is satisfied by arranging for the downgoing aileron to reach dead center when the upgoing aileron reaches the floating angle. Figure 17 shows geometrical arrangements of linkages that satisfy these two criterions for a minimum stick force. If the spacing of the crank centers is known in terms of the crank radius, the figure gives directly the neutral settings of the two cranks. The differential thus chosen will give what amounts to complete balance at the specified floating angle. The maximum downward

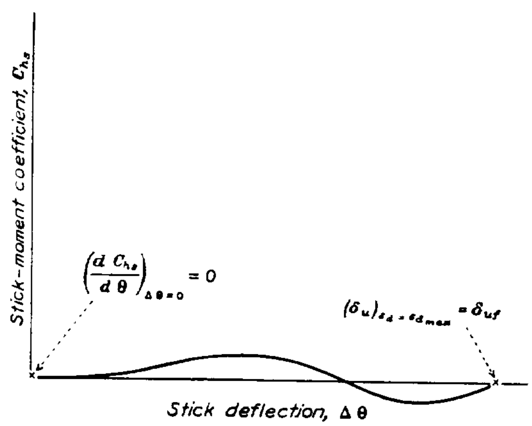


FIGURE 16.—Type of curve that satisfies simple criterions for minimum stick force.

travel of the aileron is shown in each case and it is to be noted that, if the maximum deflection of the upgoing aileron exceeds the assumed floating angle, the downgoing aileron will pass dead center and return toward neutral.

Since the floating tendency of a given aileron has a primary influence on the design of the differential linkage, it will be necessary to devote some study to this aileron characteristic. It appears that the floating angle of a plain flap-type aileron can be attributed to two effects: (1) a hinge moment proportional to the angle of attack of the wing, this moment being greater for large flap chords but independent of the shape of the wing section; and (2) a hinge moment attributed to the camber of the wing section, which remains constant as the angle of attack is changed. This second moment is primarily influenced by the camber of the aileron portion itself and is greatly affected by small changes at the extreme trailing edge. Thus, a small fixed tab can be used to introduce a large constant floating moment.

Figure 18 shows the variation of floating angle with flap chord and lift coefficient for the Clark Y wing section. The floating angles shown were indirectly computed from floating moments that were found by integration of pressure-distribution diagrams for a smooth wing (reference 20) and hence correspond to smoothly sealed flaps.

For the comparisons given in table I, infinite linkages ($R=0$ in fig. 17) were assumed to simplify the computations of control force. In most cases of differential ailerons listed, several trial computations of stick force were made to ascertain the optimum differential arrangement. These trial computations included the

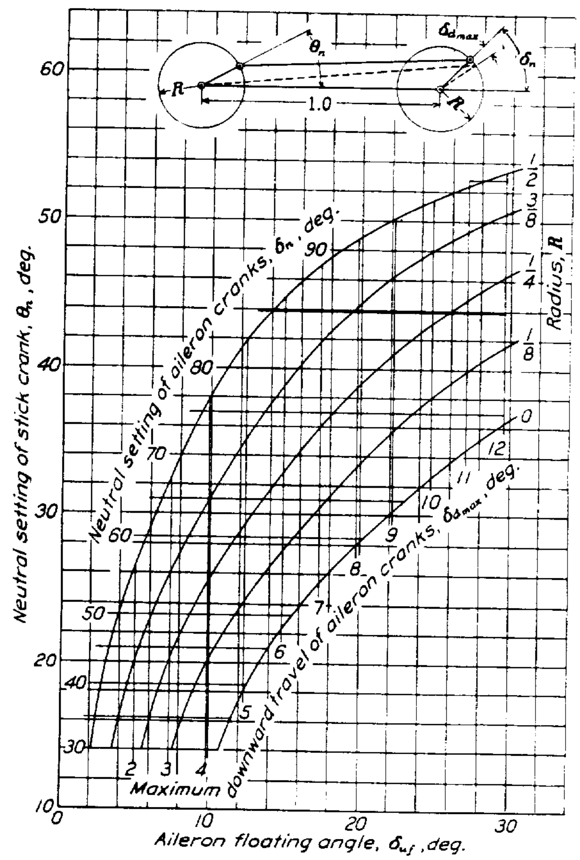


FIGURE 17.—Specifications of simple differential linkages that satisfy criterions for minimum stick force.

$\left(\frac{dC_{h_s}}{d\theta}\right)_{\theta=\theta_n}=0;\ (\delta_u)_{\delta_d=\delta_{d_{max}}}=\delta_{uf}$

determination of the curve of stick force against deflection to insure that no reversals of slope of the stick-force curve occurred at any point.

Aileron 1 may be used to illustrate the use of figure 17 in the selection of a differential. Assuming that the greatest possible reduction in stick force is desired, a floating angle only slightly higher than the maximum shown by figure 18 will be assumed. On the assumption that it is permissible to allow the control force to become zero at $C_L=1.25$ ($\delta_{uf}=11°$), the differential chosen by means of the chart will have neutral settings of $\theta_n=15°$ and $\delta_n=30°$, approximately. As indicated by figure 17, the maximum downward deflection obtainable with this arrangement will be about 4½° and this angle will be reached when the upgoing aileron reaches 11° deflection. For greater deflections the downgoing aileron will return, reaching neutral when the up aileron is at 22°.

Effect of a fixed tab used in conjunction with a differential linkage.—Figure 18 shows that the floating angles of plain ailerons are reduced as the lift coefficient is reduced. It is on this account that the balancing effect of the differential diminishes. The stick forces tabulated for the differentially linked aileron 1 show this effect as an increase of stick force at high speed. It is possible to introduce a large constant floating moment by means of a properly formed fixed tab. The effect of such a tab is to increase the floating angle at all flight speeds by a constant amount so that the percentage variation with flight speed is reduced. This effect is especially pronounced in the case of very narrow ailerons, which do not show a very great variation of floating angle with angle of attack.

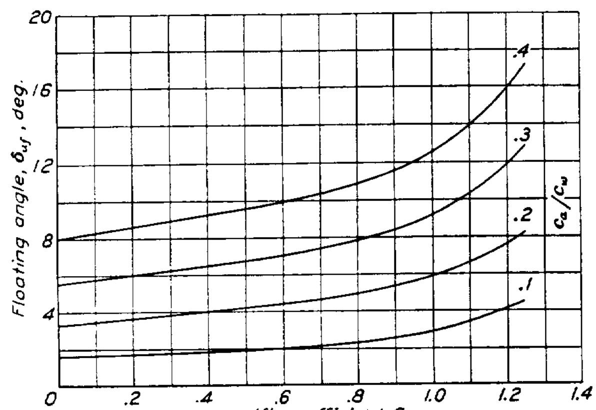


FIGURE 18.—Floating angles of sealed flaps of various chords on a Clark Y wing as computed from pressure-distribution data (reference 20).

Furthermore, the maximum floating angle shown by very narrow ailerons is not great enough to permit the use of a differential to the best advantage. Thus, if the floating angle is considerably smaller than the maximum upward deflection required to produce sufficient control, the stick force may rise considerably after this point is reached on account of the return of the downgoing aileron and the consequent extra deflection required of the upgoing aileron. Advantageous use of a differential in such cases can be accomplished by incorporating a fixed tab (or a small amount of camber) arranged to trim both ailerons upward. In order to secure satisfactory results with a tab, a reasonably smooth inset type with a sealed juncture should be used. Attached tabs or tabs set at large angles ($\delta_t > \pm 15°$) have been found to cause an adverse increase in the slope of the hinge-moment curve.

Figure 19 shows the summarized results of experiments with tabs made in the 7- by 10-foot wind tunnel. As was stated before, the tab produces an essentially constant change in floating angle. The variation of floating angle with angle of attack can be found from figure 18. Figure 19 gives the change of aileron floating angle with tab deflection. (See references 9 and 21.) The experiments indicated that this ratio depended primarily on the ratio of tab chord to aileron chord independently of the chord of the aileron, although this relation can not be expected to apply as the aileron chord is indefinitely increased. At the Reynolds Number of the tests the tabs began to lose effectiveness when deflected past 15°; hence, the ratios given should be considered applicable to tab deflections not exceeding this angle. Figure 19 may also be used to estimate the balancing effect of a movable tab.

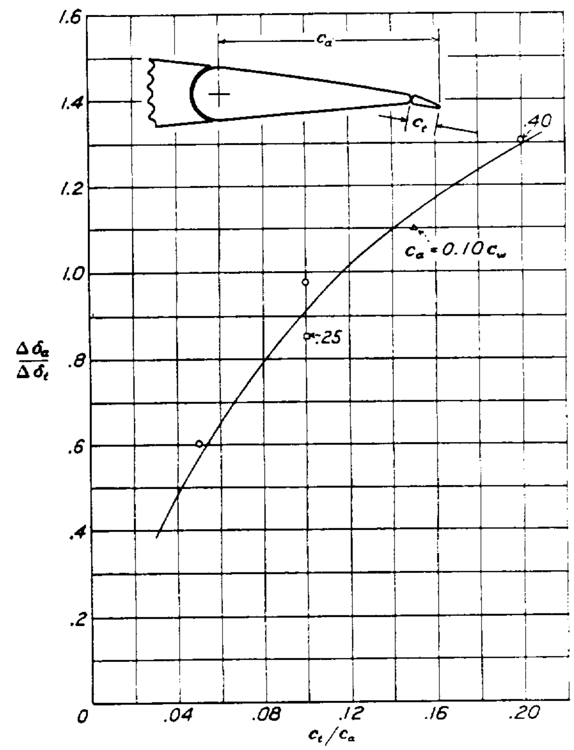


FIGURE 19.—Effect of inset tabs on aileron floating angles (references 9 and 21); $\delta_t < \pm 15°$.

It appears from figure 19 that a very large floating angle can be obtained by the use of a relatively small inset tab and deflection. Thus, the floating angle can very easily be altered to suit a given set of conditions. It has been pointed out that it is desirable to have the floating angle at least as large as the maximum upward deflection required for control so that the stick-force curve will lie reasonably near the minimum throughout the range. The smaller the percentage variation of floating angle with angle of attack, the smaller will be the variation of the actual stick force with flight speed. It would therefore appear desirable to trim the ailerons up as far as possible by means of a tab. On the other hand, inasmuch as the deflected tab is made an inherent part of the airfoil camber, the size and deflection of the tab cannot be indefinitely increased without adversely affecting the pitching-moment and drag characteristics of the airfoil.

Reference to figure 19 shows that a 0.10 c_a (2½ percent c_w) tab deflected downward 10° will change the floating angles of aileron 1 by approximately 9°, raising the maximum floating angle to about 20°. This tab on the average airplane would be only 1.6 inches wide and the deflection of 10° would displace the trailing edge of the wing section by only one-third inch and would consequently not be expected to make a noticeable change in the drag or the pitching moment of the wing as a whole. The differential linkage giving complete balance at $\alpha=15°$ with this floating angle can be found from figure 17. The neutral settings of the cranks are

$$\theta_n=28°,\ \delta_n=59° \tag{21}$$

The maximum downward deflection found on the chart is about 8°, but in this case the aileron is not required to reach this deflection (20° up and 8° down) to produce a sufficient bank. Reference to figure 18 shows that the reduction in floating angle between $C_L=1.25$ (maximum) and $C_L=1.0$ is 2.5° so that, with the tab assumed, the floating angle at $\alpha=10°$ ($C_L=1.0$) will be

$$20°-2.5°=17.5° \tag{22}$$

Similarly, the new floating angle at $\alpha=0°$ ($C_L=0.35$) will be

$$20°-4.8°=15.2° \tag{23}$$

These values indicate that the balancing effect of the differential will not be greatly reduced at the higher speeds. Table I gives the actual stick forces as computed at these lift coefficients and indicates the reduction possible with a tab. An even better degree and range of balance could be attained with narrower ailerons on account of the smaller variation of floating angle with angle of attack.

CONCLUDING REMARKS

The provision of control rolling moments at high angles of attack or beyond the stall is not sufficient to secure control in flight at these angles unless the damping in rolling is retained. This requirement necessitates that at least the tip portions of the wing remain unstalled; hence, it cannot be considered a decided advantage to retain control rolling moments far above the stall with conventional wings.

The flight-testing experience gained throughout the course of the lateral control investigation has led to more or less definitely quantitative ideas regarding the desired effectiveness of the lateral control and the desirable variation of the control forces in normal flight.

From considerations of operating force required for a given amount of control, plain narrow sealed ailerons with deflections limited to 20° seem about the most efficient. Very great taper, or change of aileron chord along the span, leads to inefficiency whether used with a straight or a tapered wing. A differential linkage can be so designed as to reduce considerably the operating force of ordinary unbalanced ailerons, especially if a small fixed tab is used to increase the floating angle.

Several devices, notably the plain ailerons with flap retracting ahead, and the retractable aileron or spoiler located at 0.80 c_w have been developed and proved in flight to be suitable for use with full-span flaps. It was found, however, that the maximum lift of a tapered wing with split flaps was reduced less than 10 percent by the removal of the outer 0.30 $b/2$ portions of the flap, so that a conventional aileron could be used over that portion of the wing without great loss.

Aerodynamic theory can be successfully applied to the calculation of rolling and yawing moments of plain ailerons provided that experimental section characteristics are used in the computation of the local changes in angle of attack along the wing span caused by the ailerons. Further calculations involving the airplane stability characteristics can be applied to the prediction of the actual resultant motions caused by a given deflection of the control, thus giving a measure of effectiveness in controlling the movements of the airplane.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., *April, 20, 1937.*

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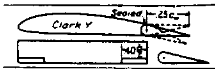
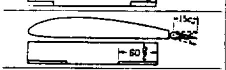
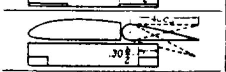
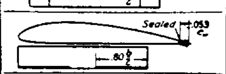
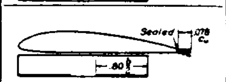
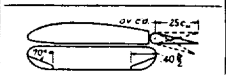
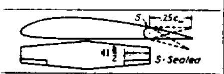
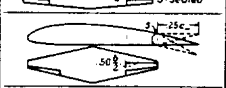
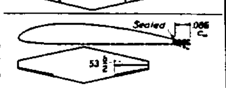
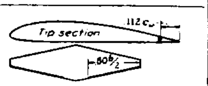
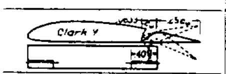
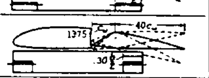
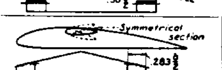
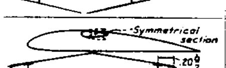
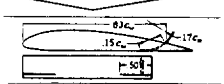
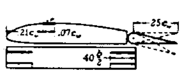
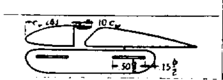
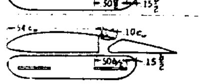
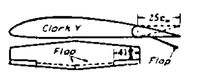
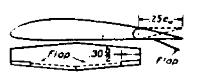
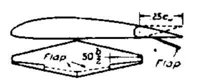
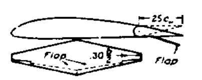
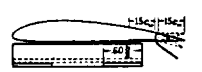
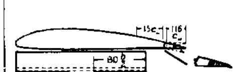
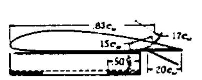
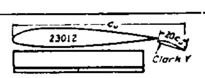
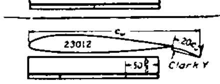
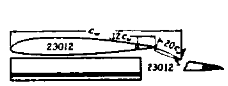
TABLE I (A).—COMPARISON OF VARIOUS LATERAL CONTROL DEVICES

Device	Criterion	Linkage	Control forces and aileron deflections to produce specified bank in 1 second: $\phi_1=15°$, $C_L=0.35$: Stick force (lb.)	$\phi_1=15°$, $C_L=0.35$: Aileron angles (degrees)	$\phi_1=15°$, $C_L=1.0$: Stick force (lb.)	$\phi_1=15°$, $C_L=1.0$: Aileron angles (degrees)	Maximum deflection (degrees) $\phi_1=22.5°$ $C_L=1.0$	Sideslip with 15° bank in 1 second (degrees): $C_L=0.35$	Sideslip: $C_L=1.0$	Lateral stability $\alpha_{L_p=0}$ minus $\alpha_{C_{L_{max}}}$ (degrees)	Performance: Maximum lift $C_{L_{max}}$	Performance: Speed range $\frac{C_{L_{max}}}{C_{D_{min}}}$	Performance: Climb $\frac{L}{D}$ at $C_L=0.7$
	1. Plain ailerons sealed [a] 0.25 c_w×0.40$\frac{b}{2}$.	Equal Diff. Diff. with tab.	4.7 2.1 .9	±3.4 3.9×2.7 3.7×3.0	3.6 1.1 .3	±7.4 11.0×4.1 9.0×6.0	±11.2 20.0×2.0 14.0×7.7	3 3 3	7 7 7	---- ---- ----	---- ---- ----	---- ---- ----	---- ---- ----
	2. Plain ailerons 0.25c_w×0.40$\frac{b}{2}$.	Equal Diff.	6.4 2.6	±3.8 4.5×3.0	5.6 1.6	±9.4 11.0×4.8	±14.5 18.0×5.0	3 3	8 7	0 0	1.27 1.27	91 91	18.7 18.7
	3. Plain ailerons 0.15c_w×0.60$\frac{b}{2}$.	Equal Diff.	3.2 2.5	±3.8 4.5×3.0	2.6 1.2	±7.9 11.0×4.8	±13.0 23.0×4.0	4 4	8 8	1 1	1.22 1.22	85 85	18.5 18.5
	4. Plain ailerons 0.40c_w×0.30$\frac{b}{2}$.	Equal Diff.	14.0 7.0	±4.2 4.8×3.5	11.0 2.0	±9.0 12.0×7.0	±14.0 18.0×8.0	4 4	9 8	1 1	1.25 1.25	87 87	18.2 18.2
	5. Optimum plain sealed ailerons [a] 0.053c_w×0.80$\frac{b}{2}$.	Equal	1.4	±6.1	1.0	±13.0	±20.0	3	7	----	----	----	----
	6. Optimum differential sealed ailerons [a] 0.078c_w×080.$\frac{b}{2}$; 0.015c_w fixed tab, down 14°.	Diff.	0.5	4.5×3.6	0.1	12.0×7.4	20.0×8.6	3	7	----	----	----	----
	7. Skewed ailerons. Rounded tip wing [b] 0.25c_w×0.40$\frac{b}{2}$; 20° skew.	Equal Diff.	10.0 5.2	±4.8 6.1×3.8	9.2 4.7	±12.0 16.0×5.0	±18.0 27.0×3.2	5 4	8 7	2 2	1.26 1.26	87 87	18.5 18.5
	8. Tapered ailerons 5:3 tapered wing 0.25c_w×0.41$\frac{b}{2}$.	Equal Diff.	4.0 2.4	±3.0 3.4×2.6	3.7 1.5	±7.5 8.4×4.8	±11.6 15.0×5.6	3 3	7 7	---- ----	1.88 1.88	125 125	19.5 19.5
	9. Tapered ailerons 5:1 tapered wing 0.25c_w×0.50$\frac{b}{2}$.	Equal Diff.	2.4 1.4	±2.8 3.1×2.5	2.2 1.2	±7.4 8.2×6.0	±11.7 13.0×7.8	3 3	6 6	−3 −3	1.81 1.81	129 129	18.2 18.2
	10. Optimum tapered sealed ailerons 5:1 tapered wing [a] 0.086c_w×0.53$\frac{b}{2}$.	Equal	1.4	±7.3	0.9	±14.5	±20.0	3	6	----	----	----	----
	11. Optimum straight sealed ailerons 5:1 tapered wing [a] 0.112c_w (at tip)×0.80$\frac{b}{2}$.	Equal	0.8	±6.8	0.5	±13.5	±20.0	3	6	----	----	----	----
	12. Frise ailerons 0.25c_w×0.40$\frac{b}{2}$.	Equal Diff	3.2 1.8	±2.6 2.6×2.5	3.8 1.1	±7.5 8.0×7.0	±14.5 18.0×12.0	3 3	7 6	0 0	1.28 1.28	85.0 85.0	18.5 18.5
	13. Frise ailerons (modified) 0.40c_w×0.30$\frac{b}{2}$.	Equal Diff.	5.1 1.7	±4.2 4.3×4.0	8.1 2.3	±10.0 11.0×8.5	±14.3 16.0×11.0	5 5	9 8	---- ----	---- ----	---- ----	---- ----
	14. Floating-tip ailerons 5:1 tapered wing 1.00c_w×0.28$\frac{b}{2}$.	Equal	2.5	4.8	2.2	8.6	13.0	2	2	−2	1.18	91.0	12.6
	15. Floating-tip ailerons 5:1 tapered special wing [a] 1.00c_w×0.20$\frac{b}{2}$.	Equal	4.8	7.4	4.9	15.8	25.0	1	2	----	1.27	84.0	19.5
	16. Retractable ailerons 0.15c_w×0.50$\frac{b}{2}$.	Up only	0	0.025c_w [f]	0	0.062c_w [f]	0.074c_w [f]	1	4	----	1.27	91.0	18.1
	17. Ailerons and spoiler: [a,d] Ailerons 0.25c_w×0.40$\frac{b}{2}$. Spoiler 0.07c_w×0.40$\frac{b}{2}$.	Equal Diff.	2.3 1.4	±2.0 2.4×2.2	1.2 .2	±4.4 6.0×4.4	±6.5 9.6×5.6	3 3	4 4	0 0	1.27 1.27	91.4 91.4	18.7 18.7
	18. Slot-lip ailerons [d] 0.30c_w location 0.10c_w×0.50$\frac{b}{2}$.	Unbalanced. Balanced	32.0 20.0	23.0×14.0 23.0×14.0	18.0 16.0	55.0×0 55.0×0	55.0×0 55.0×0	−5 −5	−2 −2	4 4	1.22 1.22	75.0 75.0	15.0 15.0
	19. Slot-lip ailerons [d] 0.55c_w location 0.10c_w×0.50$\frac{b}{2}$.	Unbalanced. Balanced.	12.0 8.5	19.0×12.0 19.0×12.0	9.3 8.5	43.0×3.0 43.0×3.0	47.0×0 47.0×0	−1 −1	2 2	4 4	1.23 1.23	83.0 83.0	16.0 16.0

[a] Computed or estimated results.
[b] Hinge moments computed or estimated.
[d] Device may not give satisfactory response characteristics.
[f] Deflection given in percentage of wing chord.

TABLE I (B).—COMPARISON OF VARIOUS LATERAL CONTROL DEVICES

Device	Criterion	Linkage	Control force and aileron deflection to produce specified bank in 1 second, $\phi_1=15°$, $C_L=0.35$: Stick force (lb.)	$C_L=0.35$: Aileron angles (degrees)	$C_L=1.0$: Stick force (lb.)	$C_L=1.0$: Aileron angles (degrees)	$C_L=1.8$: Stick force (lb.)	$C_L=1.8$: Aileron angles (degrees)	Sideslip with 15° bank in 1 second (degrees): $C_L=0.35$	$C_L=1.0$	$C_L=1.8$	Performance: Maximum lift $C_{L_{max}}$	Speed range $\frac{C_{L_{max}}}{C_{D_{min}}}$	Climb $\frac{L}{D}$ at $C_L=0.7$
	20. Tapered ailerons, sealed. 5:3 tapered wing. Partial-span split flap: Ailerons $0.25c_w\times0.41\frac{b}{2}$. Flap $0.15c_w\times0.59\frac{b}{2}$.	Equal Diff	4.0 2.4	±3.0 3.4×2.6	3.7 1.5	±7.5 8.4×4.8	3.5 .8	±12.0 15.0×5.6	3 3	7 7	8 7	1.88 1.88	125 125	19.5 19.5
	21. Tapered ailerons, sealed. 5:3 tapered wing. Partial-span split flap: Ailerons $0.25_w\times0.30c\frac{b}{2}$. Flap $0.15c_w\times0.70\frac{b}{2}$.	Equal Diff	4.0 2.1	±4.3 5.0×3.6	3.6 1.1	±9.6 13.0×5.1	4.5 1.3	±16.0 25.0×1.5	3 3	7 6	8 6	1.97 1.97	130 130	19.5 19.5
	22. Tapered ailerons, sealed. 5:1 tapered wing. Partial-span split flap: [c] Ailerons $0.25c_w\times0.50\frac{b}{2}$. Flap $0.15c_w\times0.50\frac{b}{2}$.	Equal Diff	2.4 1.4	±2.8 3.1×2.5	2.2 1.2	±7.4 8.2×6.0	1.9 .1	±11.7 13.0×7.8	3 3	6 6	6 6	1.81 1.81	129 129	18.2 18.2
	23. Tapered ailerons, sealed. 5:1 tapered wing. Partial-span split flap: Ailerons $0.25c_w\times0.30\frac{b}{2}$. Flap $0.15c_w\times0.70\frac{b}{2}$.	Equal Diff	2.4 1.5	±4.2 4.5×3.6	2.5 1.4	±12.0 14.0×18.0	2.8 1.5	±20.0 26.0×10.0	2 2	6 5	6 5	1.97 1.97	141 141	18.2 18.2
	24. Plain ailerons. Retractable flap: Ailerons $0.15c_w\times0.60\frac{b}{2}$. Flap $0.15c_w\times1.00\frac{b}{2}$.	Equal Diff	6.2 5.7	±3.8 4.0×3.5	4.7 3.7	±7.8 8.7×7.1	6.7 5.4	±25.0 28.0×11.0	4 4	8 8	8 7	2.05 2.05	143 143	18.5 18.5
	25. Plain sealed ailerons. Retractable flap: [a] Ailerons $0.116c_w\times0.80\frac{b}{2}$. Flap $0.15c_w\times1.00\frac{b}{2}$.	Diff. with tab.	1.4	3.4×4.2	0.9	8.4×6.6	2.7	35.0×0.6	3	8	6	2.05	143	18.5
	26. Retractable ailerons. Split flap: Ailerons $0.15c_w\times0.50\frac{b}{2}$. Flap $0.20c_w\times1.00\frac{b}{2}$.	Up only.	0	$0.025c_w$[f]	0	$0.062c_w$[f]	0	$0.074c_w$[f]	1	4	6	2.19	149	18.1
	27. External-airfoil flaps [e] $0.20c_w\times1.00\frac{b}{2}$.	Diff	5.5	3.2×3.0	3.1	6.0×5.5	0.2	13.0×11.0	3	7	10	1.83	172	18.7
	28. External-airfoil flap ailerons [e] $0.20c_w\times0.50\frac{b}{2}$.	Diff	0.9	3.7×3.7	0.8	7.6×7.3	0.3	16.0×9.2	3	7	8	1.80	172	18.7
	29. Slot-lip ailerons. External-airfoil flap: [e, d] Ailerons $0.12c_w\times1.00\frac{b}{2}$. Flap $0.20c_w\times1.00\frac{b}{2}$.	Diff. with tab.	2.4	10.0×6.0	2.3	25.0×6.5	1.4	14.0×6.8	3	6	7	1.92	202	19.0

[a] Computed or estimated results.
[c] C_L slightly below 1.8.
[d] Device may not give satisfactory response characteristics.
[e] Spring mechanism assumed to avoid overbalance with flap down.
[f] Deflection given in percentage of wing chord.

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1 hp = 76.04 kg-m/sec = 550 ft-lb/sec

1 metric horsepower = 1.01[illegible] hp

1 m.p.h. = 0.4470 m.p.s.

[illegible]