Module -3

Static lateral dynamic and longitudinal stability and control.

I. Definition of Roll stability:

An airplane possesses static roll stability if a restoring moment is developed when it is disturbed from a wings-level attitude. The restoring rolling moment can be shown to be a function of the sideslip angle β as illustrated in below figure.

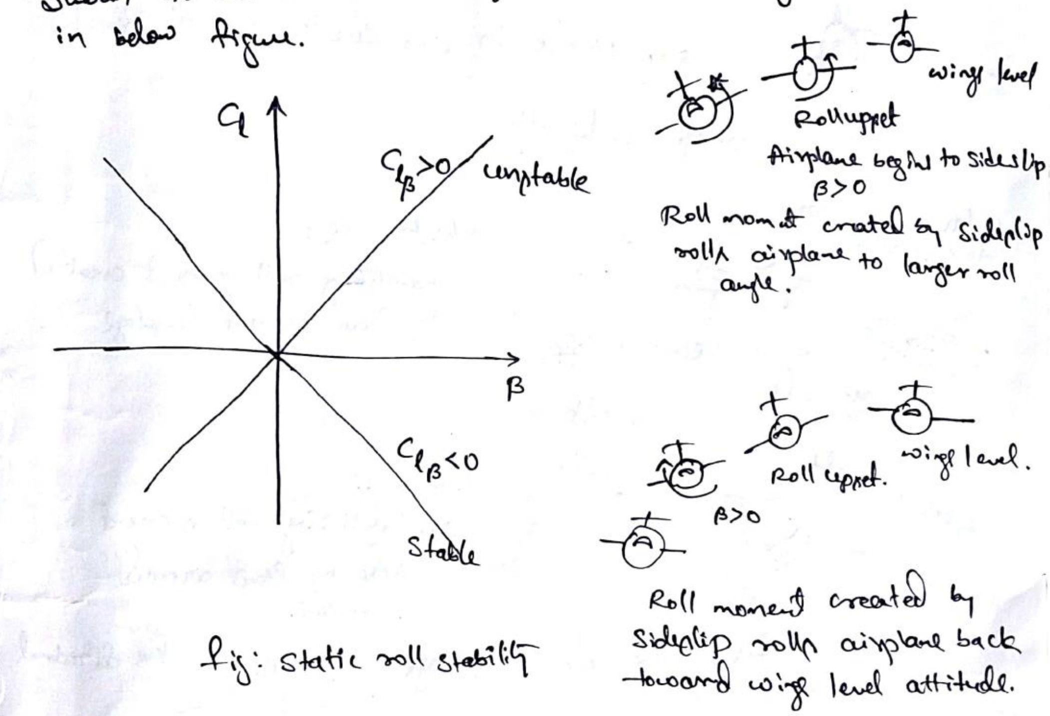


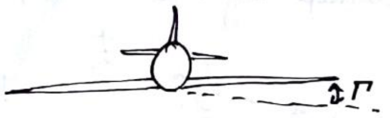
fig: static roll stability

The requirement for stability is that $C_{l_\beta}<0$. The roll moment created on an airplane when it starts to sideslip depends on the wing dihedral, wing sweep, position of the wing on fuselage, and the vertical tail. Each of these contributions will be discussed qualitatively in the following:

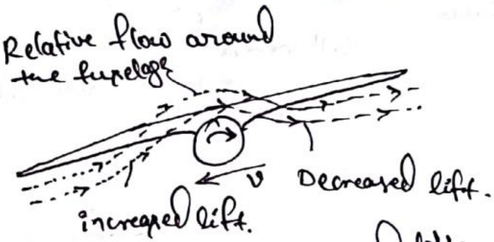
The major contributor to C_{l_β} is the wing dihedral angle 'Γ'. The dihedral angle is defined as the spanwise inclination of the wing with respect to the horizontal. If the wing tip is higher than the root section, then the dihedral angle is positive; if the wing tip is lower than the root section, then the dihedral angle is negative. A negative dihedral angle is commonly called anhedral.

When an airplane is disturbed from a wings level attitude, it will begin to sideslip as shown in below figure.

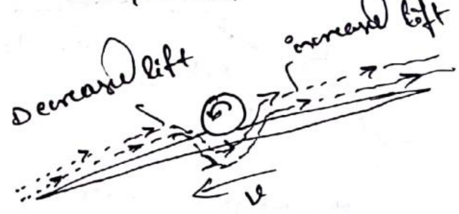
Wing contribution



fuselage contribution



high wing:
stabilizing roll moment created by flow around fuselage



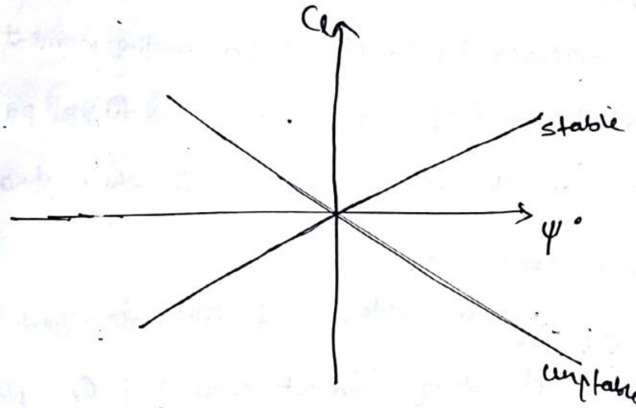
low wing:
Destabilizing roll moment created by flow around fuselage.

fig: wing and fuselage contribution to the dihedral

II. Dihedral Effect:

The problem of holding the wings level or of maintaining some angle of bank is one of control over the rolling moments about the airplane's longitudinal axis. The major control over the rolling moments is the familiar aileron system consisting of flaps on the wing outer panels which, when deflected asymmetrically, will alter the wing's spanwise lift distribution in such a way that a net rolling moment is created. A secondary control over the rolling moment can be obtained through control over the sideslip angle, as it will be shown that, for certain wing geometry, sideslip will alter the wing's spanwise lift distribution to create a net rolling moment.

The phenomenon of rolling moment due to sideslip is termed dihedral effect and is not a static stability in the true sense of the word.



where ψ = yaw angle

fig: Typical wind tunnel tests for dihedral effect.

An airplane is said to have stable dihedral effect if a negative rolling moment (left wing down) is created as the result of positive sideslip (β). This defination is somewhat arbitrary but springs from the fact that stable dihedral effect is required for complete dynamic lateral stability, and that stable dihedral effect is required for complete dynamic lateral stability, and that stable dihedral effect as defined will require the use of top rudder to pick up a wing that droops because of a gust or any other rolling disturbances.

The rolling moment will be discussed in coefficient form, with the rolling moment coefficient equal to the rolling moment 'L', ~~in pound feet~~ divided by the dynamic pressure 'q', the wing area 'S_w', and the wing span 'b'.

$$\therefore \quad C_l = \frac{L}{q S_w b}$$

The power of the lateral or aileron control will be expressed as the change in rolling moment coefficient per degree deflection of the ailerons, while the dihedral effect will be measured by the change in rolling moment coefficient per degree change in sideslip 'β', or in wind tunnel parlance, with change in yaw ψ, the negative of β when straight flight paths are considered.

The criterion of dihedral effect is the slop of the curve of rolling moment coefficient C_l, plotted against yaw and is given as the derivative $\frac{dC_l}{d\psi}$ per degree or in shorthand notation C_{l_ψ}.

III Estimation of Airplane dihedral Effect.

The rolling moment due to sideslip (dihedral effect) is mainly created by wing dihedral ~~effect~~ angle 'Γ', which is positive for tip chord above the root chord. In a sideslip the angle of attack of the forward wing will be higher than the angle of attack of the trailing wing. This will create a lift on the ~~angle of attack of the~~ leading wing that will be greater than the lift on the trailing wing, There by creating a rolling moment about the x axis.

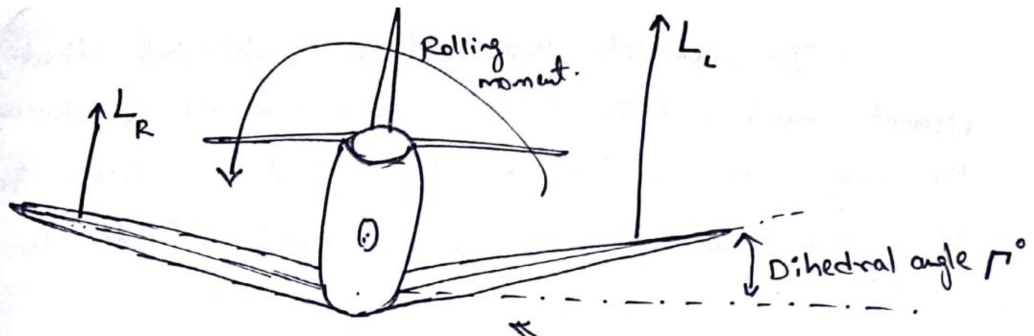


fig: Dihedral effect.

The dihedral effect, as mentioned before, is measured by the slope of the curve of rolling moment coefficient versus angle of yaw, $C_{l\psi}$. The value of this derivative varies almost directly with wing dihedral angle at the approximate rate $\Delta C_{l\psi} = 0.0002\,\Delta\Gamma^\circ$, and an airplane having $C_{l\psi}$ equal to 0.0002 is said to have 1 degree of effective dihedral.

The value of the dihedral effect $C_{l\psi}$ can be stated as follows:

$$C_{l\psi} = C_{l\psi_{\Gamma=0}} + 0.0002\Gamma^\circ$$

The difficult part of estimating the dihedral effect is to estimate the value of this derivative for zero geometric wing dihedral $C_{l\psi_{\Gamma=0}}$. It has been found as the result of considerable wind tunnel experience that this residual dihedral effect varies considerably with the position of the wing on the fuselage. It has also been found that the dihedral effect will be somewhat invarient with change in wing angle of attack for straight wings, but will change rapidly with angle of attack for swept wings.

The dihedral effect of the straight wing alone is given as

$$\left(C_{l\psi}\right)_{wing} = \frac{C_{l\psi}}{\Gamma}\cdot\Gamma + \Delta C_{l\psi_{\text{Tip shape}}}$$

To complete the analysis of dihedral effect, account must be taken of the interference effects between the wing, the fuselage, and the vertical tail. These effects are very troublesome, as they are most difficult to analyze.

If the wing is placed high on the fuselage, the interference effects will increase the effective dihedral; the interference effects for a mid wing are negligible, and for a low wing, decrease the effective dihedral. The horizontal position of the wing has only very small effects. These effects can be given as wing - fuselage interference increments as follows:

$$\text{High wing } (\Delta C_{l\psi})_1 = +0.0006$$

$$\text{Mid wing } (\Delta C_{l\psi})_1 = 0$$

$$\text{low wing } (\Delta C_{l\psi})_1 = -0.0008$$

The vertical tail, if located all above the airplane's X axis, can contribute to the dihedral effect which can be computed quite readily from the normal force on the vertical tail created by the sideslip.

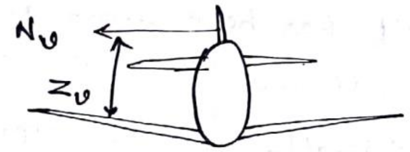


fig: The rolling moment due to vertical tail.

The rolling moment coefficient may be obtained as follows:

$$C_l = a_v \psi \eta_v \frac{S_v}{S_w} \frac{Z_v}{b} \qquad \Big| \; a_v = C_{L\alpha_v}$$

and stability contribution

$$(C_{l\psi})_v = a_v \frac{S_v}{S_w} \frac{Z_v}{b} \eta_v$$

A second interference factor, that of the wing on the vertical tail contribution to $C_{l\psi}$, may be given:

High wing $(\Delta C_{l\psi})_2 = -0.00016$

Mid wing $(\Delta C_{l\psi})_2 = 0$

Low wing $(\Delta C_{l\psi})_2 = 0.00016$

∴ The final equation for the dihedral effect of the complete airplane may be given in the following form:

$$\boxed{(C_{l\psi})_{Airplane} = (C_{l\psi})_w + (C_{l\psi})_v + (\Delta C_{l\psi})_1 + (\Delta C_{l\psi})_2}$$

IV. Effect of wing sweep, flaps and Power on dihedral effect

The method of estimating the dihedral effect in the previous section is applicable for an airplane with a straight wing, flaps up with propellers - wind milling. The variation of $C_{l\psi}$ with wing sweep, flap deflection, and high thrust coefficient is a very complex one, and one that is almost impossible to estimate quantitatively by any analytical approach.

The dihedral effect for airplanes having Swept wing planforms will become a function of the lift coefficient. Airplanes with swept back wings will have an increasing dihedral effect with lift coefficient, while airplanes with swept forward wings will have a decreasing dihedral effect with lift coefficient. Typical variations in the dihedral effect parameter $C_{l\psi}$ with airplane lift coefficient are shown in below figure for the case of the swept forward and swept back wing.

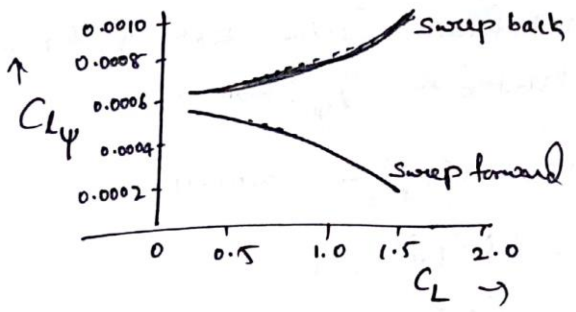


fig: Typical variation of C_{l_ψ} with C_L for swept wings.

At very low lift coefficient, the sweep of the planform has little effect on C_{l_ψ}, but if the geometric dihedral of the swept planform airplane is set to give a good value of C_{l_ψ} at high speed, then the airplane with swept back wings will be in danger of having excessive dihedral at low speeds, while the airplane with swept forward wings will probably encounter negative dihedral effect at low speeds.

The dihedral effect of the airplane can also be seriously affected by flap deflection. Experience indicates that if the hinge line of the flap is unswept, there is little difference in C_{l_ψ} flaps up or down. However, if the ~~hinge~~ hinge line of the flaps is swept back, flaps deflection will usually increase C_{l_ψ}, while if the hinge line of the flaps is swept forward, flap deflection will usually decrease C_{l_ψ}. Typical variations in C_{l_ψ} due to flap deflection are shown in below figure.

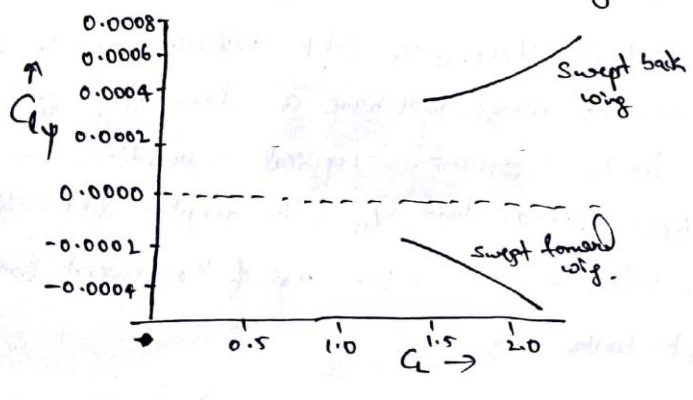


fig: C_{l_ψ} v/s C_L flaps down on swept wings

The effect of power on dihedral effect is usually serious for only the flap down condition. This effect usually arises because of the displacement of the slipstream in a sideslip, resulting in one flap being immersed in the slipstream to a greater extent than the other.

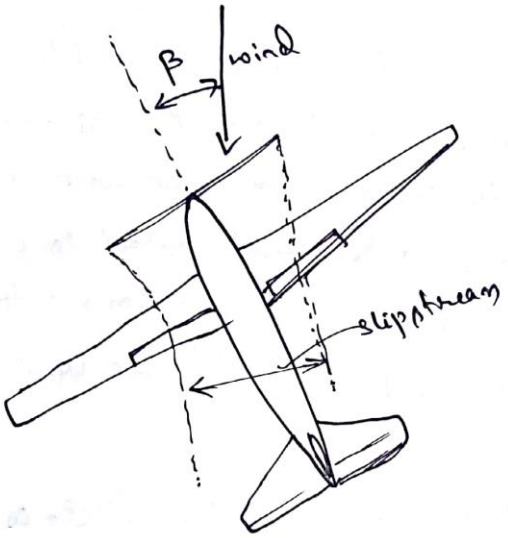


fig. Slipstream effect on $C_{l\psi}$, flaps down

These power effects are at a maximum in full power, low speed flight where the ~~r~~ratio of slipstream velocity to free stream velocity is the greatest. The effect of power on $C_{l\psi}$ with flaps deflected are greatest for flaps with swept forward hinge lines, and the least for flaps with swept back hinge lines, but in almost all cases are destabilizing.

If the airplane has a swept back planform, the stabilizing effect of sweepback at high C_L counteracts somewhat the destabilizing effects of power, while for a swept forward wing, these effects are additive.

The upper limit on dihedral effect is therefore very important and should not exceed 3 or 4 degrees of effective dihedral. The lower limit on dihedral effect varies with the usage of the airplane.

V. Lateral control

It is possible to design an airplane to operate with elevator and rudder controls only, the maneuvering possibilities of an aircraft controlled in this manner are decidedly limited, and in almost all cases totally inadequate.

It has been found necessary to provide the pilot with a powerful and definite control over the airplane's angle of bank in order to satisfy the minimum maneuvering requirements of the modern airplane. This lateral control is usually obtained through the use of plain flaps mounted on the trailing edge sections of the outer wing panels which are usually referred to as ailerons.

The ailerons on each wing deflects asymmetrically, one going up and the other going down, thereby so altering the spanwise load distribution that a rolling moment is created about the X axis. A typical aileron arrangement is shown in below figure, indicating the change in the spanwise lift due to the aileron deflection.

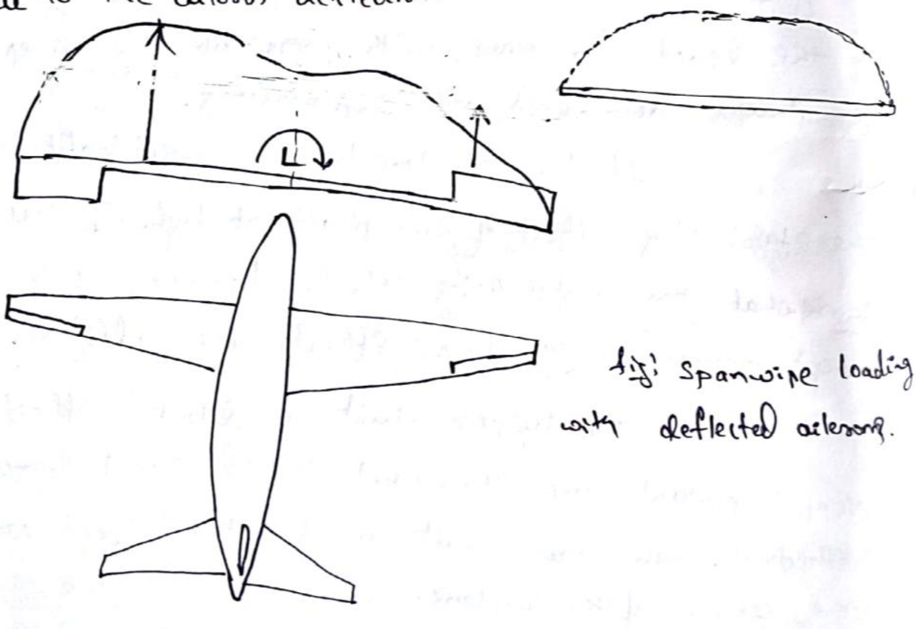


fig: Spanwise loading with deflected ailerons.

fig:

Deflecting the aileron will create a rolling moment that will accelerate the airplane in roll about the X axis. As the airplane's rolling velocity increases, a new lift distribution will be created that is a function of this rolling velocity and which opposes the rolling moment due to the aileron deflection. This moment is known as "The damping moment of the wing".

The alteration of the spanwise lift distribution due to the rolling velocity comes about as a result of the change in effective angle of attack at any wing section due to the roll,

$$\Delta\alpha_s = \frac{py}{V}$$

where p is the rolling velocity in radians per second, y the spanwise location of the section from the X axis, and V is the forward velocity in feet per second.

A typical increment in span wise lift distribution due to the rolling velocity is given in below figure.

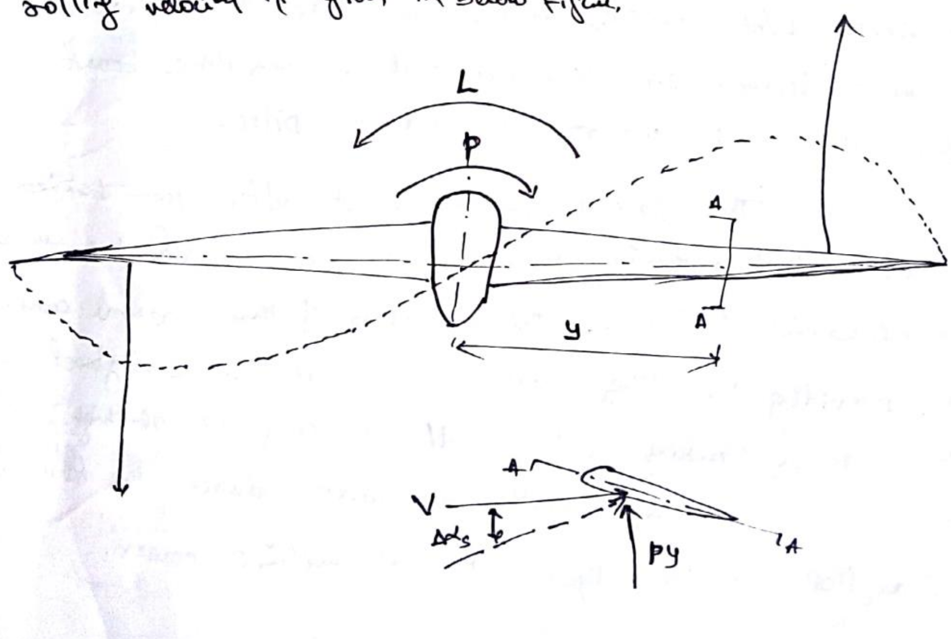


fig: load distribution due to rolling velocity.

The damping moments of a wing are very powerful, and therefore the ailerons are required to be very effective indeed if a fast rate of roll is desired. The steady state rolling velocity is determined when the increment in the rolling moment due to the ailerons is just opposed by the increment in the rolling moment due to the wing structure.

The design of the ~~rattle~~ lateral control for effectiveness and lightness is the most difficult control design problem confronting the aerodynamicist. The reasons for this are that in combat and normal flight manouvers ~~fog~~. larger deflections of the ailerons are required than for the other controls. and that the lateral control system is geared to a sidewise motion of the stick, a direction in which the pilot can exert only a small effect. In some cases, especially for large airplanes, the ailerons are geared to a wheel, where the mechanical advantage is better and the pilot can apply considerable force. For modern high speed airplanes extremely close aerodynamic balance is required for direct pilot control, and in some cases this problem has been deemed so difficult that a hydraulic boost system has been incorparated to aid the pilot.

The lateral control must fulfill two basic requirements that determine the size of the control and the amount of aerodynamic balance. The function of the lateral control, besides providing the rolling moments to maintain a wing level attitude during landing and to roll the airplane at high angular velocity at high speed. also must balance the dihedral in sideslips and in flight with asymmetric power.

VI. Estimation of lateral control power

The rolling performance of any wing-aileron system must be developed from a study of the equation of motion of the airplane in roll. This is usually done by assuming the airplane to be a single-degree of freedom system in roll about the x-axis. This is, of course, not strictly accurate, as motion about the x axis couples with the asymmetric degrees of freedom.

If the wing is considered a rigid structure, the equation of motion in roll can be written very simply. The rolling moments arising only from the aileron deflection, δ_a, and the wing damping due to angular velocity, p.

$$I_x \dot{p} = \frac{\partial L}{\partial p} p + \frac{\partial L}{\partial \delta_a} \delta_a$$

In this study only the steady-state rolling velocity is sought after, so that the assumption that $\boxed{\dot{p} = 0}$ is made at once, therefore above equation reduces to:

$$\therefore \quad 0 = \frac{\partial L}{\partial p} p + \frac{\partial L}{\partial \delta_a} \cdot \delta_a$$

$$\therefore \quad p = -\frac{\frac{\partial L}{\partial \delta_a}}{\frac{\partial L}{\partial p}} \cdot \delta_a$$

The partial derivatives $\partial L/\partial \delta_a$ and $\partial L/\partial p$ can be expressed in coefficient form by dividing both numerator and denominator by $q_\infty S b$

$$\therefore \quad p = -\frac{\frac{\partial C_l}{\partial \delta_a}}{\frac{\partial C_l}{\partial p}} \cdot \delta_a . \quad \text{---- (1)}$$

The rolling moment coefficient per degree aileron $\left(dCl/\partial\delta_a\right)$ and the rolling moment per degree per second rate of roll can be evaluated by the strip integration method or more accurately from determinations of the spanwise lift distribution. The strip method will be developed first to demonstrate the functional relationship between the variables, while the lift distribution methods will be incorporated later for actual design practice.

The rolling moment coefficient per degree aileron throw can be developed as follows.

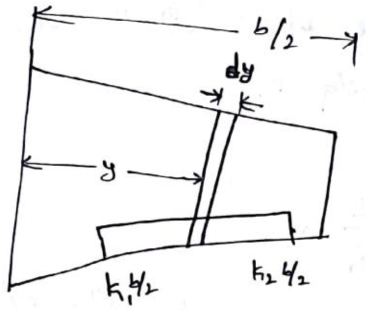


fig: Strip integration for $C_{l_{\delta_a}}$

$$\therefore \quad dC_l = \frac{c\, c_l\, y\, dy}{S_w b}$$

where, c is the wing local chord and c_l is the ~~section~~ section lift coefficient. The section lift coefficient can be expressed as,

$$c_l = a_0 \tau \delta_a$$

where τ is the section flap effectiveness factor.

Substituting c_l value in above equation and integrating between the limits $k_1 b/2$ and $k_2 b/2$ give the rolling moment coefficient for the complete wing.

$$C_l = \frac{2 a_w \tau \delta_a}{S_w b} \int_{k_1 b/2}^{k_2 b/2} c.y.dy \quad \text{------(1)}$$

where a_w and τ must be for three dimensional flow.

The derivative $\frac{\partial C_l}{\partial \delta_a}$ becomes,

$$\boxed{C_{l_{\delta_a}} = \frac{\partial C_l}{\partial \delta_a} = \frac{2 a_w \tau}{S_w b} \int_{k_1 b/2}^{k_2 b/2} C \cdot y \cdot dy \quad \text{--- (3)}}$$

The rolling moment coefficient due to the angular velocity, p. can be developed in a similar manner. The angle of attack due to the rolling velocity of any section at a distance 'y' from the center line can be given approximately as:

$$\Delta\alpha = \frac{py}{V}$$

Therefore the increment in section lift coefficient due to the rolling velocity is

$$C_l = a_0 \frac{py}{V}$$

∴ the ~~$\frac{\partial C_l}{\partial \delta_a}$ Equation become~~

the eqn (2) becomes,

$$C_l = \frac{2 a_w p}{V S_w b} \int_0^{b/2} c y^2 dy$$

$$\therefore \frac{\partial C_l}{\partial p} = \frac{2 a_w}{V S_w b} \int_0^{b/2} c y^2 dy \quad \text{--- (4)}$$

Substituting eqn (3) & (4) in (1)

$$\therefore p = \tau V \frac{\int_{k_1 b/2}^{k_2 b/2} C \cdot y \cdot dy}{\int_0^{b/2} c y^2 dy} \cdot \frac{\delta_a}{2} \quad \text{--- (5)}$$

for straight tapered wings, the chord is the following function of y, the distance from the centerline:

$$c = C_t \left[TR - \frac{y}{b/2}(TR-1) \right]$$

where C_t is the tip chord and TR the taper ratio.

If above equation substituted in eqn (5) and resulting expression integrated, the following equation is obtained

$$p = \frac{2\tau V \delta^{\circ}_{a_{Total}}}{57.3b} \left[\frac{(k_2^3 - k_1^3)(1-TR) + 3TR(k_2^2 - k_1^2)}{TR+3} \right] \quad \text{---(6)}$$

In eqn (6), the expression inside the brackets is a function of the extent & location of the aileron along the span, and the wing taper ratio (c_s/c_t). The aileron effectiveness factor τ is a constant depending only on the ratio of the aileron chord to the wing chord ahead of it. c_a/c_w.

It can be seen that for a given aileron deflection the ratio of $pb/2V$ will be a constant.

$\therefore$

$$\frac{pb}{2V} = \frac{\tau \delta^{\circ}_{a_{Total}}}{57.3} \left[\frac{(k_2^3 - k_1^3)(1-TR) + 3TR(k_2^2 - k_1^2)}{TR+3} \right] \quad \text{--(7)}$$

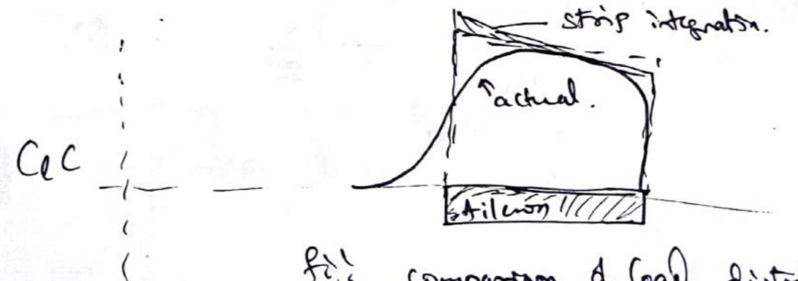


fig: comparison of load distribution.

A correction factor of 0.9 is sometimes applied to the eqn (7) and (6) to account for strip integration effect.

$$\therefore \boxed{\frac{pb}{2V} = \frac{0.9\,\tau\,\delta_a^{\circ}}{57.3}\left[\frac{(k_2^3-k_1^3)(1-TR)+3TR(k_2^2-k_1^2)}{TR+3}\right]}$$

Although the development just carried out is useful in that it demonstrates the relationship between the major variables, it is hardly ever used in practice because of its inherent accuracy. The method in nearly universal use at the present time is based on spanwise load distribution data compiled by the NACA.

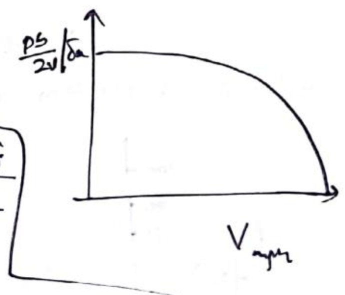
For this, similar calculations gave the damping derivative C_{l_p} defined in this case as

$$C_{l_p} = \frac{dQ}{d(pb/2V)}$$

The steady state eqn of motion in terms of these non derivatives is simply

$$I_x\dot{p} = \frac{dL}{dp}\cdot p + \frac{dL}{d\delta_a}\cdot\delta_a$$

$$\therefore 0 = C_{l_p}\left(\frac{pb}{2V}\right) - \frac{C_{l_\delta}}{\tau}\cdot\frac{\tau\delta}{2}$$



$$\therefore \frac{pb}{2V} = \frac{C_{l_\delta}}{\tau}\cdot\frac{\tau\delta_a}{114.6C_{l_p}}$$

$$\Rightarrow \boxed{\frac{pb}{2V} = \frac{C_{l_\delta}}{\tau}\cdot\frac{\tau\delta_a . K}{114.6C_{l_p}}}$$

where K = multiplying factor.

VII. Aileron control Forces (Requirements)

Once the size of the lateral control is decided upon, the next step is the investigation of the forces that the pilot must apply at the cockpit controls, in order to deflect them. In most cases the hinge moments vary as the velocity squared, and therefore the pilot's force varies essentially in the same manner.

It is obvious that for anything except 100 % aerodynamic balance, there will always be some speed above which full pilot's force may not be powerful enough to deflect the ailerons fully. For speeds in excess of this limiting speed, the aileron deflection for a given stick force will fall off rapidly. It is the duty of the aerodynamicist to provide some form of balance to insure that adequate lateral control deflections are possible in the usable speed ranges of the airplane. This is the most difficult of all control surface design problems and must be adequately solved before the design can be considered successful.

A typical variation of rolling velocity pb/2V and aileron deflection is shown in below figure

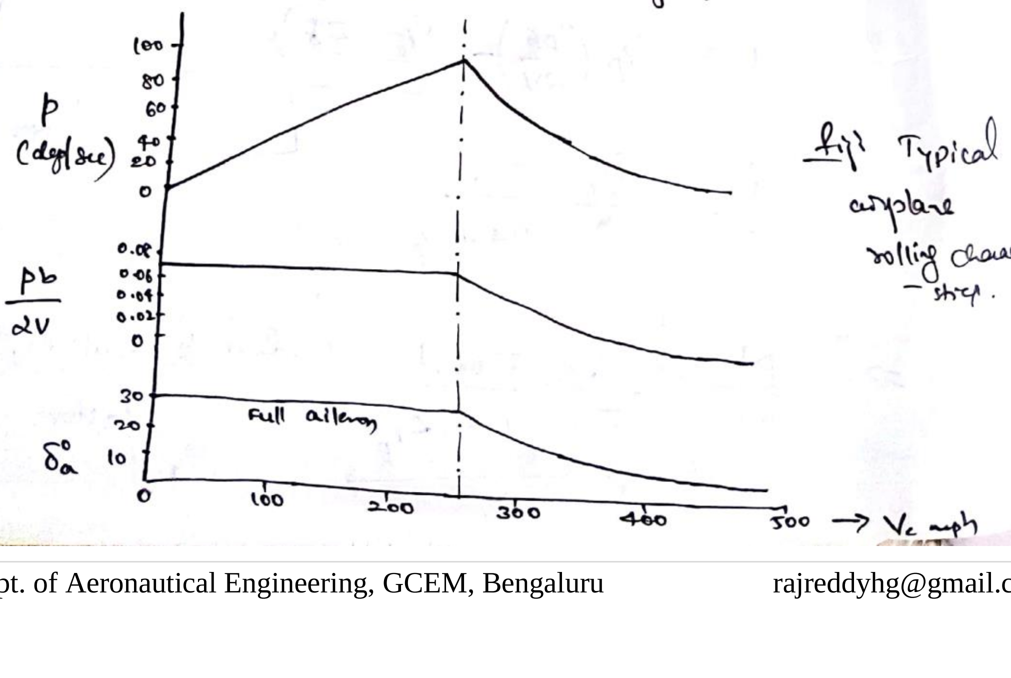


fig: Typical airplane rolling characteristics.

here V_c = Calibrated airspeed

Aileron control forces :

The aileron control forces are somewhat complicated because of the fact that there are two surfaces involved, one control going up and the other going down, and in some cases moving at different rates. because of this, it is first necessary to give some notation in order not to become confused when analyzing their combined effects on the stick forces. The term δ_a will refer to the total aileron angle in degrees and will be equal to the deflection of the up, δ_u plus the deflection of the down δ_d aileron.

$$\delta_a = \delta_u + \delta_d \quad ----(1)$$

The terms S_a and C_a are the area and root mean square chord of one aileron, respectively. An aileron stick force is considered positive for a force toward the pilot's right, and hinge moments are considered positive for trailing edge down. All aileron deflections are considered positive, as are the control gearings.

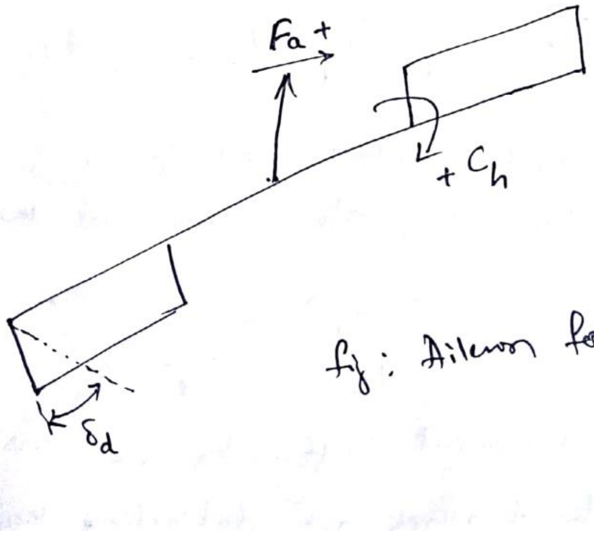


fig: Aileron force notation.

To develop the aileron control force equation, a roll to the right will be considered, requiring a positive stick force, an upward deflection of the right aileron, and a downward deflection of the left aileron. The gearing of the up aileron will be $\frac{d\delta_u}{ds}$, and the gearing of the down aileron will be $\frac{d\delta_d}{ds}$.

The stick force required at the top of the stick to hold both controls over can be expressed as:

$$F_a = -HM_d \frac{d\delta_d}{ds} + HM_u \frac{d\delta_u}{ds}. \quad ---(2)$$

The hinge moments of the down and up ailerons can be expressed in coefficient form,

$$HM_u = C_{h_u} q S_a c_a$$

$$HM_d = C_{h_d} q S_a c_a$$

Therefore eqn (2) becomes,

$$F_a = -C_{h_d} q S_a C_a \frac{d\delta_d}{ds} + C_{h_u} q S_a C_a \frac{d\delta_u}{ds}$$

$$F_a = -q S_a C_a \left[C_{h_d} \frac{d\delta_d}{ds} - C_{h_u} \frac{d\delta_u}{ds} \right]$$

For aileron systems where there is no differential, i.e., the up and down going ailerons move at the same rate:

$$F_a = -q S_a C_a \left(\frac{d\delta_a}{dS} \right) \left(C_{h_d} - C_{h_u} \right) \quad ---(3)$$

If the hinge moment coefficients are considered linear functions of angle of attack and deflections, they may be written as before.

$$C_{h_a} = C_{h_0} + C_{h_\alpha} \cdot \alpha + C_{h_\delta} \delta_a \quad \text{---(4)}$$

The angle of attack distribution across the span is altered because of the rolling velocity, the angles of attack of the wing sections on the down going wing being increased, while those on the upgoing wing are decreased. If the increment in angle of attack due to the rolling velocity is considered averaged at the spanwise location of the aileron and the distance of this station from the centroid centerline is given as y'. increment in angle of attack at the aileron is :

$$\Delta\alpha^\circ = \frac{Py'}{V} 57.3 \quad \text{---(5)}$$

The rolling velocity P can be given in term of the airplane's $\frac{Pb}{2V}$ per degree total aileron:

$$P = \frac{\frac{Pb}{2V}}{\delta_a} \cdot \delta_a \cdot \frac{2V}{b} \quad \text{---(6).}$$

Substituting Eqn(6) in (5).

$$\Delta\alpha^\circ = \frac{Pb/2V}{\delta_a} \cdot \frac{y'}{b/2} \cdot (57.3)\delta_a$$

$$\Delta\alpha^\circ = n\delta_a \quad \text{----(7)}$$

where $$n = \frac{Pb/2V}{\delta_a} \cdot \frac{y'}{b/2} \cdot (57.3)$$

The hinge moment coefficients of eqn (4) can be expressed as follows:

$$C_{h_d} = C_{h_0} + C_{h_\alpha}(\alpha - n\delta_a^{\circ}) + C_{h_\delta}\delta_d \quad \text{---(8)}$$

$$C_{h_u} = C_{h_0} + C_{h_\alpha}(\alpha + n\delta_a^{\circ}) + C_{h_\delta}\delta_u \quad \text{---(9)}$$

For the case of no differential action.

$$\delta_u = \delta_d = \frac{\delta_a}{2} \quad \text{---(10)}.$$

by considering eqn (10), substituting eqn (8) and (9) in (3)

$$\therefore \; F_a = -q S_a c_a \left(\frac{d\delta_u}{ds}\right)\left[-2C_{h_\alpha} n\delta_a + C_{h_\delta}\delta_a\right]$$

$$\therefore \; F_a = -q S_a c_a G\left[C_{h_\delta}\delta_a - 2C_{h_\alpha} n\delta_a\right]$$

where $G = \dfrac{d\delta_u}{ds}$

$$\boxed{F_a = -q S_a c_a G \, C_{h_\delta}\delta_a\left[1 - 2n\frac{C_{h_\alpha}}{C_{h_\delta}}\right]} \quad \text{----(11)}$$

Equation (11) brings out the very important part played by the floating characteristics of the ailerons. If the ailerons are aerodynamically balanced so as to float up heavily with the wind, the pilot's stick forces will be reduced materially.

here, $\left[1 - 2n\dfrac{C_{h_\alpha}}{C_{h_\delta}}\right]$ = aileron response factor.

$$\therefore \boxed{F_a = -q\, S_a c_a G K C_{h_\delta} \delta_a} \quad \text{---- (12)}$$

k = aileron response factor.

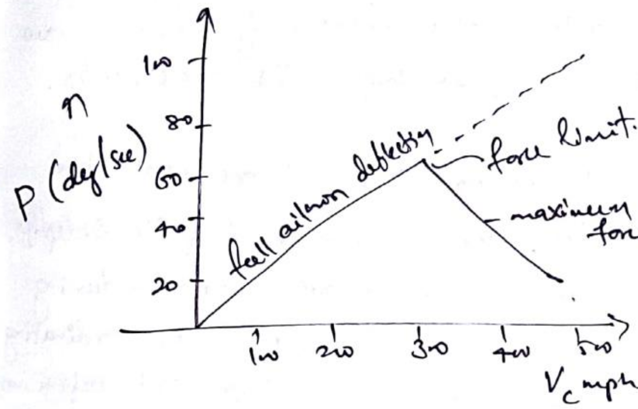


fig: typical variation of rate of roll with velocity

VIII. Coupling between rolling and yawing moments. Adverse yaw effect, Aileron reversal.

The actual rate of roll $\left(\frac{dP}{dF_a}\right)$ versus calibrated air speed falls somewhat short of that predicted by rigid wing theory, especially at the higher airspeeds. This is due to the effects of wing twist and cable stretch.

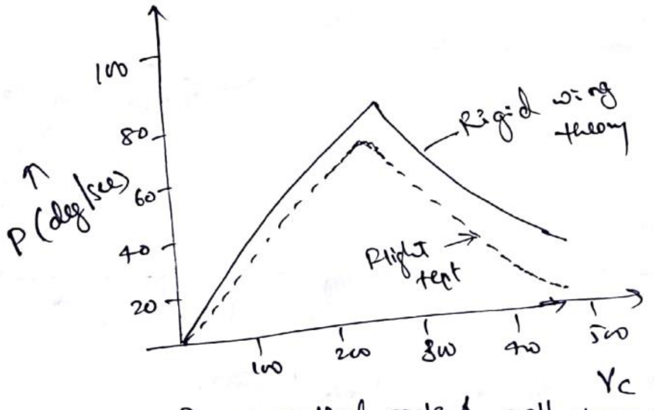


fig: Actual and theoretical rate of roll versus velocity.

The airplane's rate of roll for a given aileron deflection can be affected adversely by secondary moments in yaw developed because of the aileron deflection and the rolling motion of the wing. These secondary yawing moments are usually grouped under the general heading of ~~adverse~~ adverse yaw. These adverse yawing moments will cause the airplane to sideslip during the rolling maneuver, and if the dihedral effect is large, will create rolling moments due to the sideslip that oppose and limit the rolling velocity severely. The effects of adverse yaw are always the most severe at high lift coefficients.

In order to oppose the adverse yaw, the pilot can deflect the rudder, thereby prohibiting the sideslip. The aileron and rudder then must move together during rolling maneuvers. This techniques is known as coordinating the roll and becomes most exacting for high lift coefficients and full aileron deflection. An airplane with high directional stability and low dihedral effect will be the easiest to coordinate and for fixed rudder rolls will lose the least rolling velocity as a result of any adverse yawing moments. The time histories of two fixed rudder rolling maneuvers are given in below figure.

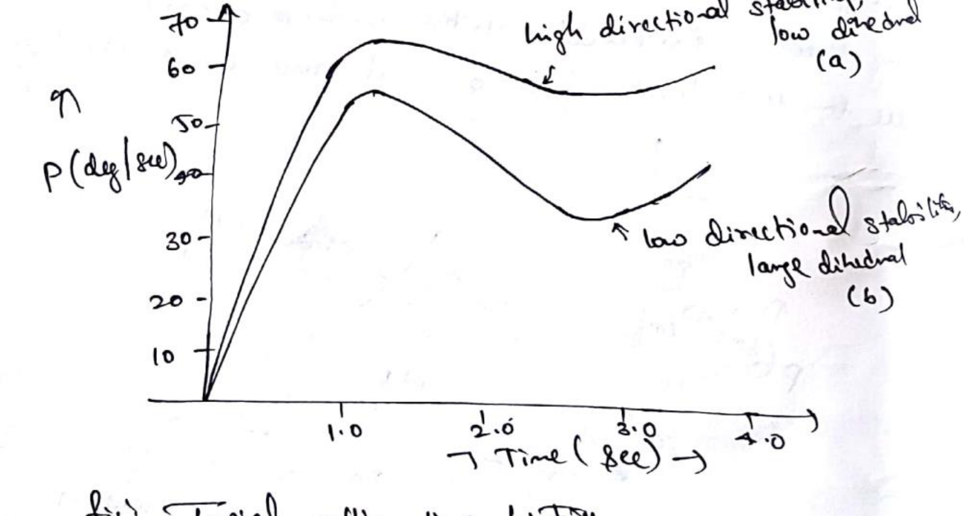


fig: Typical rolling time history.

from above figure, showing in (a) the typical falloff of rolling velocity for an airplane with low directional stability and high dihedral effect. and in (b) the more or less constant rate of roll for an airplane with high directional stability and low dihedral effect.

It should be noted in these time ~~histories~~ histories that the peak values of the rolling velocity are nearly equal. This is due to the fact that for normal aileron controls the maximum rolling velocity is established rapidly and before the sideslip has time to develop.

IX. Balancing the Aileron:

As the emphasis during the past few years has been for higher and higher rates of roll at higher and higher speed with ever-increasing airplane sizes, the problem of designing the aileron for lightness has become so difficult that in some instances airplane designers have given up all attempts at aerodynamic balance and gone over to hydraulic boost or assist.

The major lateral control is, of course, the aileron, and the development of methods for balancing the aileron took up a large percentage of the aerodynamic testing facilities.

The types of aileron aerodynamic balance can be broken down into two main classes, i.e., "nose balance and trailing edge balance."

Aerodynamic balance at the control surface nose consists of variations in nose shape, hinge line set back, shrouds, gaps, and seals, while the trailing edge types of aerodynamic balance consists of changes in ~~types~~ airfoil countour, balancing tabs, spring tabs, and trailing edge strips.

One of the most commonly used ailerons in the past and one that is still use in modified forms is the "Frise aileron." The pure frise type aileron is characterized by an asymmetrical sharp nose located on the airfoil lower surface so that it will conport as soon as the control is deflected upwards.

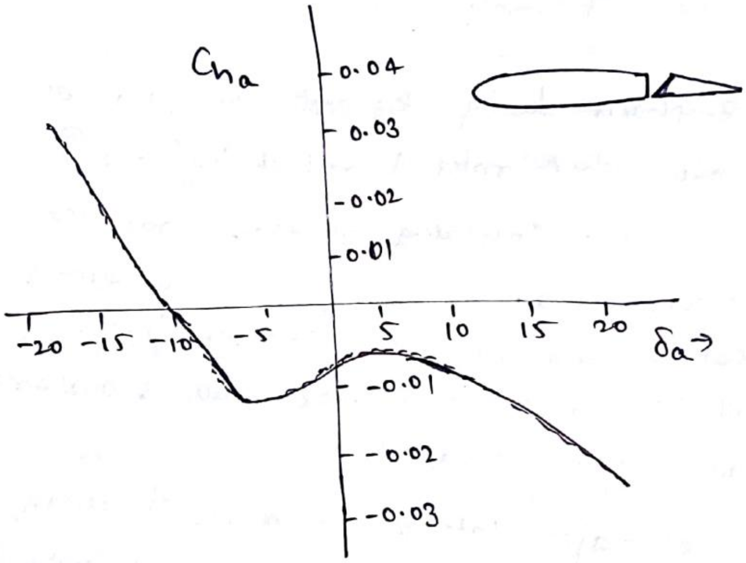


fig: Typical frise aileron hinge moments.

The major purpose of this type of control is to reduce the adverse yaw of the aileron and to provide a means of balancing the aileron for small deflections. The up going aileron is unstable, and this helps pull down the stable down going aileron; if rigged just right, excellent balance is obtained.

The advantages of this frise aileron are large balance for small hinge line set back, simplicity of construction, and reduction of aileron adverse yaw.

The disadvantage of the frise aileron are its sensitivity to rigging, the tendency of the air to separate off the lower surface of the up going aileron, causing aileron buffet and loss of effectiveness, and the tendency of the aileron to overbalance at high speed because of the aileron floating up as a result of control cable stretch.

The internal seal type of balance is another nose balance that has had widespread use in the past few years. This type of balance has essentially a very sharp

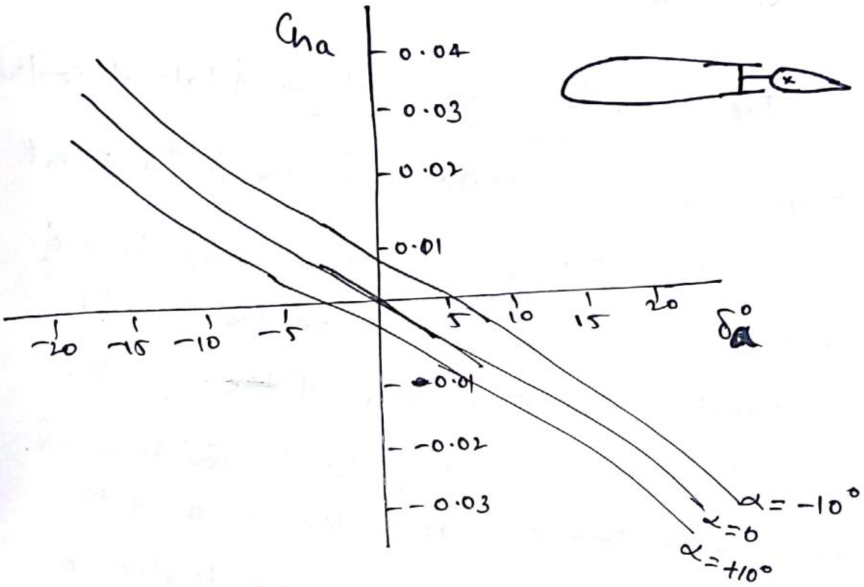


fig: Typical internal seal aileron hinge moments.

nose with a heavily set-back hinge, with curtains covering the balance area, vented close to the hinge line. This type of balance is quite popular because of the fact that its hinge moments can be maintained in production, it is not very sensitive to rigging, and it behaves quite well at high speed. Its disadvantage lies in its complicated construction and maintanance problems.

The types of trailing edge balance include such devices as the ~~beveled~~ beveled trailing edge, the balancing tab, the spring tab and strips.

The beveled trailing edge balance was developed by the NACA and has been proved to be an effective means of balance, although it had practically no tactical use during world war II.

The balancing tab can be used with discretion, but it cuts down the aileron effectiveness and tends to make the ~~aileron~~ airplane laterally unstable stick free.

The spring tab for ailerons is receiving a lot of attention at the present time. This device is a tab deflects as a function of the pilot's force only.

The spoiler type aileron of lateral control was first investigated by NACA and used for first time on the Northrop P-61 night fighter. This type of control creates a rolling moment by spoiling the lift on one wing panel. The effectiveness of the spoiler increases as its location on the wing chord moves forward. However, at the same time the lag in the action of the aileron becomes objectionably large, and the location at the present time is limited to a position about 70% of the local wing chord. Spoiler ailerons are useful as they permit more extensive use of flaps and have very low aerodynamic hinge moments.

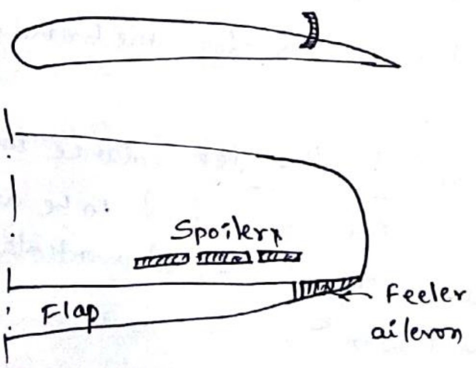


fig: A spoiler type aileron installation