MODULE IV

STATIC STABILITY AND CONTROL

Prepared by
Dr. Deepa M S
Professor
Department of Aeronautical Engineering
SJCIT

STATIC STABILITY AND CONTROL

4.1 INTRODUCTION:

During early conceptual design, the requirements for good stability, control and handling qualities are addressed through the use of tail volume coefficients and through location of the aircraft center of gravity at some percent of the wing mean aerodynamic chord (MAC). In larger aircraft companies, the aircraft is then analyzed by the control experts, probably using a six-degree-of-freedom (6-DOF) aircraft-dynamics computer program to determine the required c.g location and the sizes of the tails and control surfaces.

Stability is simply that a stable aircraft, when disturbed tends to return by itself to its original state (pitch, yaw, roll, velocity, etc). "Static stability" is present if the forces created by the disturbed state (such as pitching moment due to an increased angle of attack) push in the correct direction to return the aircraft to its original state.

If these restoring forces are too strong the aircraft will overshoot the original state and will oscillate with greater and greater amplitude until it goes completely out of control. Although static stability is present, the aircraft does not have "dynamic stability".

"Dynamic stability" is present if the dynamic motions of the aircraft will eventually return the aircraft to its original state. The manner in which the aircraft returns to its original state depends upon the restoring forces, mass distribution and "damping forces". Damping forces slow the restoring rates.

Figure 4.1 illustrates these concepts for an aircraft disturbed in pitch. In figure 4.1a, the aircraft has perfectly neutral stability and simply remains at whatever pitch angle the disturbance produces. While some aerobatic craft are nearly neutral in stability, few pilots would care to fly such an aircraft on a long trip in gusty conditions.

Illustration figure 4.1b shows static instability. The forces produced by the greater pitch angle actually cause the pitch angle to further increase. Pitch up is an example of this.

In figure 4.1c, the aircraft shows static stability with very high damping. The aircraft slowly returns to the original pitch angle without any overshoot.

Illustration figure 4.1d shows a more typical aircraft response; the aircraft returns to its original state, but experiences some converging oscillation. This is acceptable behavior provided the time to converge is short.

In figure 4.1e, the restoring force are in the right direction so the aircraft is statically stable. However, the restoring forces are high and the damping forces are relatively low, so the aircraft overshoots the original pitch angle by a negative amount greater than the pitch angle produced by the disturbance. Restoring forces then push the nose back up, overshooting by an even greater amount. The pitch oscillations continue to increase in amplitude until the aircraft "diverged" into an uncontrolled flight mode such as a spin.

Dynamic instability is not always unacceptable provided that it occurs slowly. Most aircraft have at least one dynamic-instability mode, the spiral divergence. This divergence mode is so slow that the pilot has plenty of time to make the minor roll correction required to prevent it. In fact, pilots are generally unaware of the existence of the spiral-divergence mode because the minor corrections required are no greater than the roll corrections required for gusts.

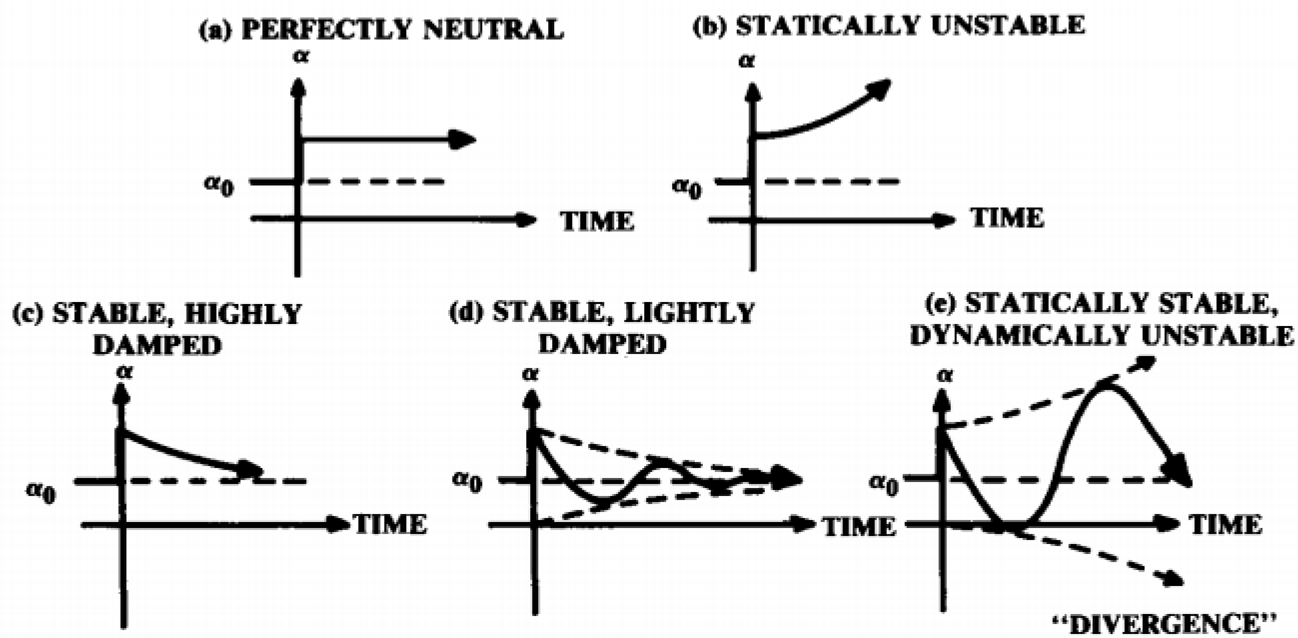


Figure 4.1 – Static and dynamic stability

4.2 LONGITUDINAL STATIC STABILITY:

Most aircraft being symmetrical about the centerline, moderate changes in angle of attack will have little or no influence upon the yaw or roll. These permits the stability and control analysis to be divided into longitudinal (pitch only) and lateral-directional (roll and yaw) analysis.

Figure 4.1 shows the major contributors to aircraft pitching moment about the center of gravity including the wing, tail, fuselage and engine contributions. The wing pitching-moment contribution includes the lift through the wing aerodynamic center and the wing moment about the aerodynamic center which is defined as the point about which pitching moment is constant with respect to angle of attack. This constant moment about the aerodynamic center is zero only if the wing is uncambered and untwisted. Also, the aerodynamic center is typically at 25% of the MAC in subsonic flight.

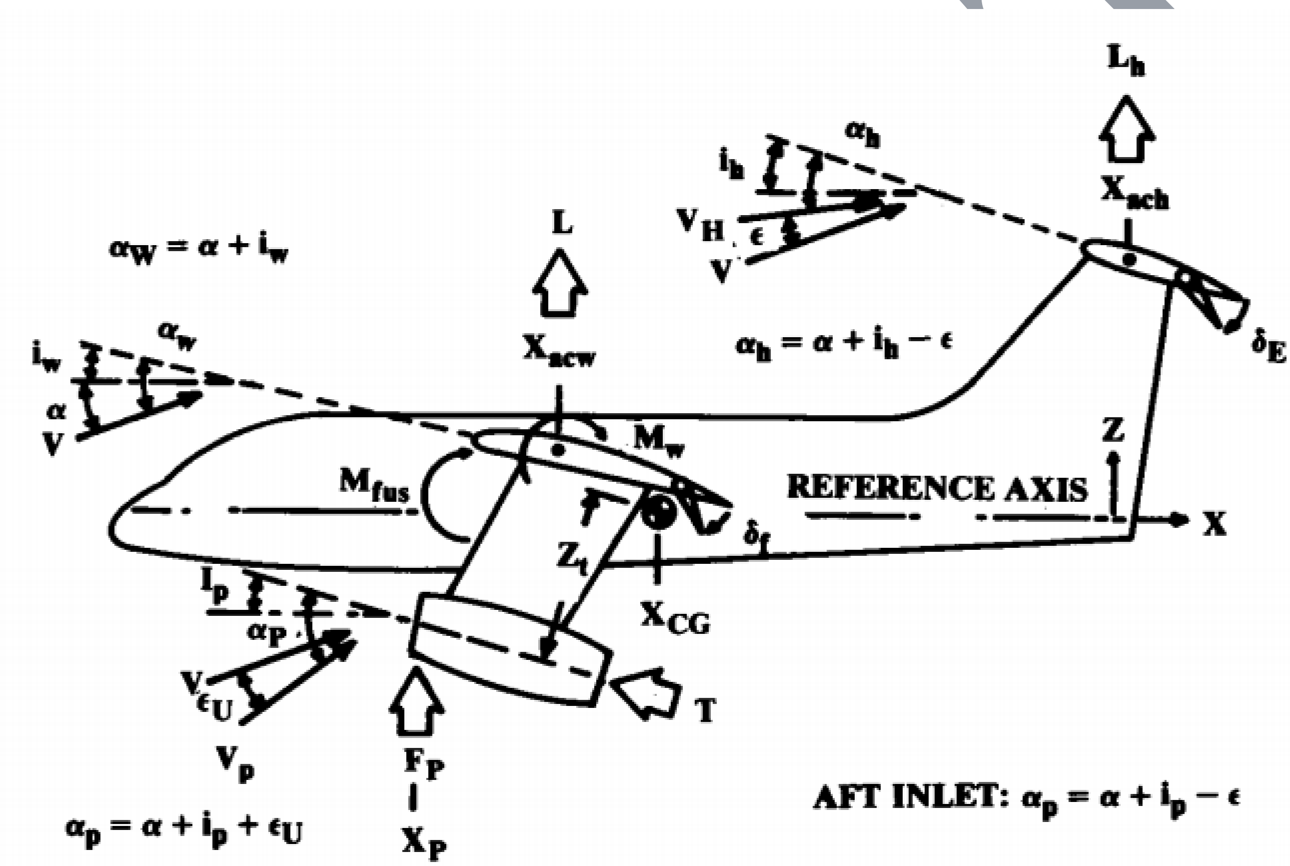


Figure 4.2- Longitudinal Moments

Another wing moment term is the change in pitching moment due to flap deflection. Flap deflection also influences the wing lift. Flap deflection has a large effect upon downward at the tail. Drag of the wing and tail produces some pitching moment, but these values are negligibly small. Also, the pitching moment of the tail about its aerodynamic center is small and can be ignored.

The wing moment arm of the tail times its lift produces a very large moment that is used to trim and control the aircraft. While this figure show tail lift upward, under many conditions the tail lift will be downward to counteract the wing pitching moment.

A canard aircraft has a "negative" tail moment-arm that should be applied in the equations that follow. If an aircraft is tailless, the wing flap must be used for trim and control. Due to the short moment arm of such a control, the trim drags will be substantially higher for off-design c.g. locations.

The fuselage and nacelles produce pitching moments that are difficult to estimate without wind-tunnel data. These moments are influenced by the upwash and downwash produced by the wing.

The engine produces three contributions to pitching moment. The obvious term is the thrust times its vertical distance from the c.g Less obvious is the vertical force F_p produced at the propeller disk or inlet front face due to the turning of the freestream airflow. Also, the propwash or jet-induced flow field will influence the effective angle of attack of the tail and possibly the wing.

Equation 4.1 expressed the sum of these moments about the c.g. the effect of elevator deflection is included in the tail lift term. Equation 4.2 expresses the moments in coefficient form by dividing all terms by (qS_wc) and expressing the tail lift in coefficient form. These equations are defined in the body axis coordinate system rather than the stability-axis system for better understanding.

$$M_{cg} = L(X_{cg} - X_{acw}) + M_w + M_{w\delta_f}\delta_f + M_{fus} - L_h(X_{ach} - X_{cg}) - Tz_t + F_p(X_{cg} - X_p) \qquad \text{-----Eq(4.1)}$$

$$C_{m_{cg}} = C_L\left(\frac{X_{cg} - X_{acw}}{c}\right) + C_{m_w} + C_{m_{w_{\delta_f}}}\delta_f + C_{m_{fus}} - \frac{q_h S_h}{qS_w}C_{L_h}\left(\frac{X_{ach} - X_{cg}}{c}\right) - \frac{Tz_t}{qS_wc} + \frac{F_p(X_{cg} - X_p)}{qS_wc} \qquad \text{Eq(4.2)}$$

(

These equations produces a term representing the ratio between the dynamic pressure at the tail and the free stream dynamic pressure, which is defined in equation 4.3 as η_h. This ranges from about 0.85-0.95, with 0.90 as he typical value.

All lengths can be expressed as a fraction of the Wing mean chord c, to simplify the equations. These fractional lengths are denoted by a bar. Thus $\bar{X}_{cg}$represents X_{cg}/c. This leads to equation 4.4.

$$\eta_h = q_h/q \quad \text{----Eq (4.3)}$$

$$C_{m_{cg}} = C_L(\overline{X}_{cg} - \overline{X}_{acw}) + C_{m_w} + C_{m_{w_{\delta_f}}}\delta_f + C_{m_{fus}} - \eta_h \frac{S_h}{S_w} C_{L_h}(\overline{X}_{ach} - X_{cg})$$

$$-\frac{T}{qS_w}\overline{Z}_t + \frac{F_p}{qS_w}(\overline{X}_{cg} - \overline{X}_p) \quad \text{----Eq (4.4)}$$

For a static "trim" condition, the total pitching moment must equal zero. For static trim, the main flight conditions of concern are during the takeoff and landing with flaps and landing gear down and during flight at high transonic speeds. Trim for the high-g pullup is actually a dynamic problem. Usually, the most forward c.g. position is critical for trim. Aft-c.g. position is most critical for stability.

Equation 4.4 can be set to zero and solved for trim by varying some parameter, typically tail area, tail lift coefficient (ie. Tail incidence or elevator deflection) or sometimes c.g. position. The wing drag and tail trim drag can then be evaluated.

STATIC PITCH STABILITY:

For static stability to be present, any change in angle of attack must generate moments which oppose the change. In other words, the derivative of pitching moment with respect to angle of attack must be negative. The wing pitching moment and thrust terms are neglected as they are constant with respect to angle of attack.

Due to downwash effects, the tail angle of attack does not vary directly with aircraft angle of attack. A derivative term accounts for the effects of wing and propeller downwash. A similar derivative is provided or inlet normal-force term F_p.

$$C_{m_\alpha} = C_{L_\alpha}(\overline{X}_{cg} - \overline{X}_{acw}) + C_{m_{\alpha fus}} - \eta_h \frac{S_h}{S_w} C_{L_{\alpha h}} \frac{\partial \alpha_h}{\partial \alpha} (\overline{X}_{ach} - \overline{X}_{cg})$$

$$+ \frac{F_{p_\alpha}}{qS_w} \frac{\partial \alpha_p}{\partial \alpha} (\overline{X}_{cg} - \overline{X}_p)$$

----Eq(4.5)

The magnitude of the pitching-moment derivative in equation 4.5, changes with c.g location. For any aircraft there is c.g location that provides no change in pitching moment as angle of attack is varied. This "airplane aerodynamic center" or neutral point X_{np} represents neutral stability and is the most-aft c.g. location before the aircraft becomes unstable.

Equation 4.6 solves the equation 4.5 for the neutral point ($C_{m\alpha}$=0). Equation 4.7 then expresses the pitching moment derivative in terms of the distance in percent MAC form the neutral point to the c.g. this percent distance, called the "static margin" is the term in parenthesis in equation 4.7

$$\overline{X}_{np} = \frac{C_{L_\alpha} \overline{X}_{acw} - C_{m_{\alpha_{fus}}} + \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} \overline{X}_{ach} + \frac{F_{p_\alpha}}{qS_w} \frac{\partial \alpha_p}{\partial \alpha} \overline{X}_p}{C_{L_\alpha} + \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} + \frac{F_{p_\alpha}}{qS_w}}$$

-----Eq (4.6)

$$C_{m_\alpha} = -C_{L_\alpha}(\overline{X}_{np} - \overline{X}_{cg})$$

----Eq(4.7)

If the c.g. is ahead of the neutral point (positive static margin), the pitching-moment derivative is negative so the aircraft is stable. At the most-sft c.g. position,a typical transport aircraft has a positive static margin of 5-10%.

Current fighters typically have positive static margins of about 5% but new fighters such as the F-16 are being with "relaxed static stability (RSS)" in which a negative static margin (zero to -15%) is coupled with a computerized flight control system that deflects the elevator to provide artificial stability which reduces trim drag substantially.

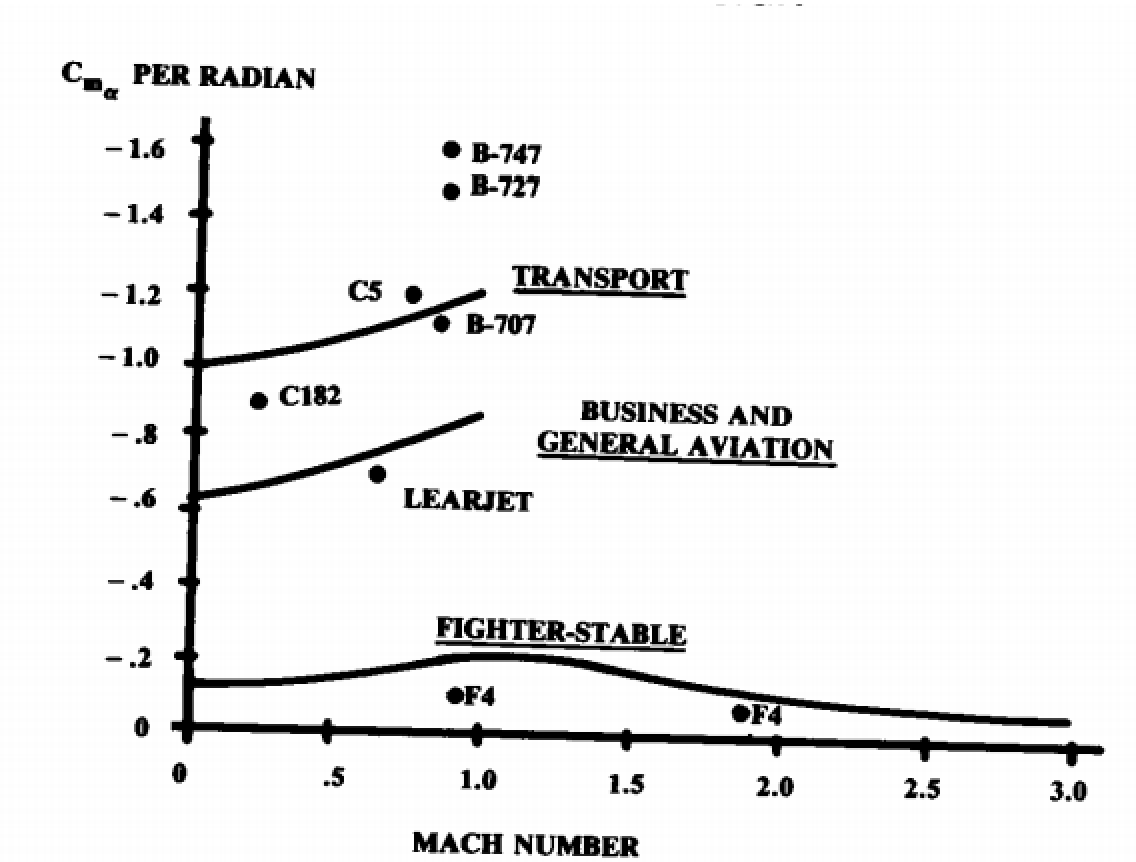


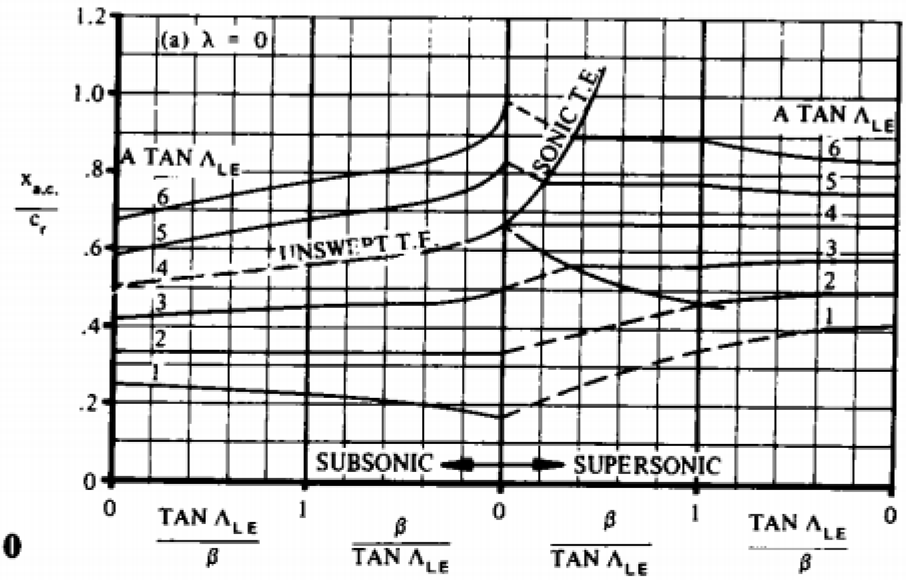
Figure 4.3: Typical pitching moment derivative values.

Figure 4.2 illustrates pitching-moment derivative values for several classes of aircraft. These may be used as targets for conceptual design. Dynamic analysis during later stages of design mat revised these targets.

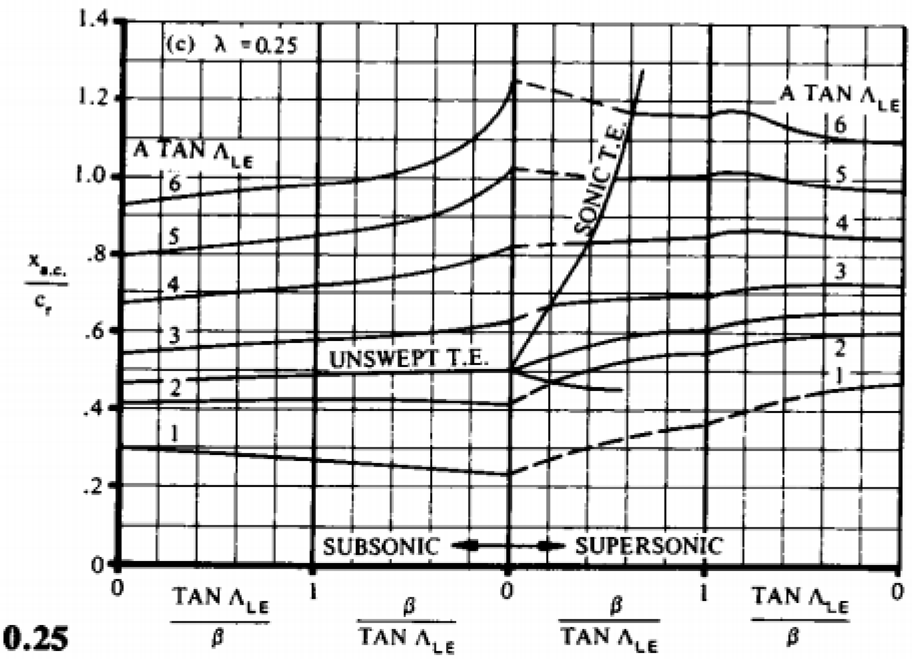
A critical term in Equation 4.4 is X_{acw}, the location of the wing aerodynamic center. For a high-aspect-ratio wing the subsonic aerodynamic center will be located at the percent MAC of the airfoil aerodynamic center. For most airfoils this is the quarter-chord point (plus or minus 1%). For supersonic speeds the wing aerodynamic center typically moves to about 45% MAC.

Wing and Tail Lift, Flaps and Elevators

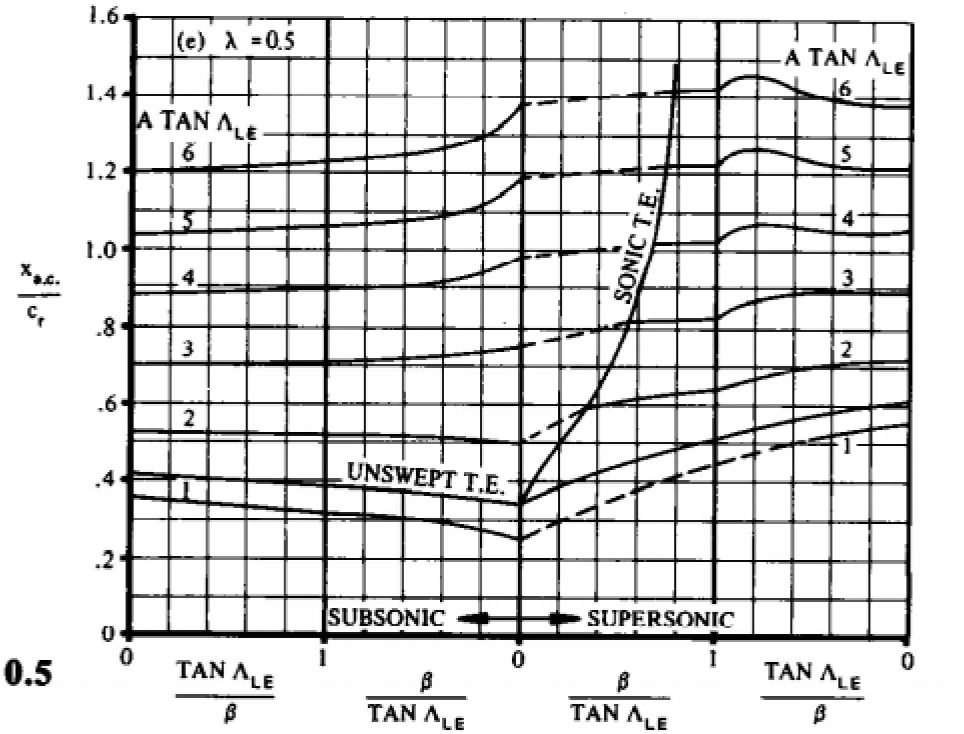
The lift-curve slopes of the wing and tail are obtained and can be reduced 20% if the elevator gap is not sealed.



a) $\lambda = 0$



b) $\lambda = 0.25$



c) λ = 0.5

Figure 4.4: Wing aerodynamic center.

The lift coefficients for the wing and tail are the lift-curve slopes times the wing or tail angle of attack (measured with respect to the zero-lift angle). These are defined in equations 4.8 and 4.9 based upon the angle of attack definitions from figure 4.1.

$$C_L = C_{L_\alpha}(\alpha + i_w - \alpha_{OL})$$ -----Eq (4.8)

$$C_{L_h} = C_{L_{\alpha_h}}(\alpha + i_h - \epsilon - \alpha_{OL_h})$$ ----Eq(4.9)

Where α_{oL} is the angle of attack for zero lift, which is a negative value for a wing or tail with positive camber and /or downwards flap/elevation deflection.

The elevator acts as a flap to increase the tail lift. Flap deflection at moderate angles of attack does not change the lift-curve slop, so the lift increment due to flaps can be accounted for by a reduction in the zero-lift angle (i.e., more negative). This reduction in zero-lift angle is equal to the increase in lift coefficient due to flap deflection divided by the lift-curve slope:

$$\Delta\alpha_{OL} = -\frac{\Delta C_L}{C_{L_\alpha}}$$ ----Eq (4.10)

For the complicated high-lift devices seen on most transport wings, the increase in lift coefficient can be approximated and the change in zero-lift angle can then be determined from equation 4.10 and applied to equation 4.8.

Plain flaps are used for a modest increase in wing lift and as the control surfaces (elevator, aileron and rudder) for most aircraft. The change in zero-lift angle due to a plain flap is expressed in equation 4.11, where the lift increment with flap deflection is expressed in equation 4.12. The 0.9 factor is an approximate adjustment for flap tip losses.

$$\Delta\alpha_{0L} = -\frac{1}{C_{L_\alpha}}\frac{\partial C_L}{\partial \delta_f}\delta_f$$ ----Eq (4.11)

Where

$$\frac{\partial C_L}{\partial \delta_f} = 0.9K_f\left(\frac{\partial C_\ell}{\partial \delta_f}\right)\frac{S_{\text{flapped}}}{S_{\text{ref}}}\cos\Lambda_{\text{H.L.}}$$ ---Eq (4.12)

Figures 4.5 and 4.6 provide the theoretical airfoil lift increment for flaps at small deflections and an empirical adjustment for larger deflections. A typical flap used for conrol will have a maximum deflection of about 30 deg. Flap deflection must be converted to radians for use in equation 4.11 and L is lift in these equations. H.L refers to the flap hinge-line sweep, $S_{flapped}$ refers to the portion of the wing area with the flap or control surface. The MAC of the flapped portion of the wing or tail is determined geometrically by considering the flapped portion as a separate surface.

If a flap, elevator, rudder or aileron has an unsealed hinge gap the effectiveness will be reduced due to the air leaking through the opening. This reduction will be approximately 15% of the lift increment due to flap deflection.

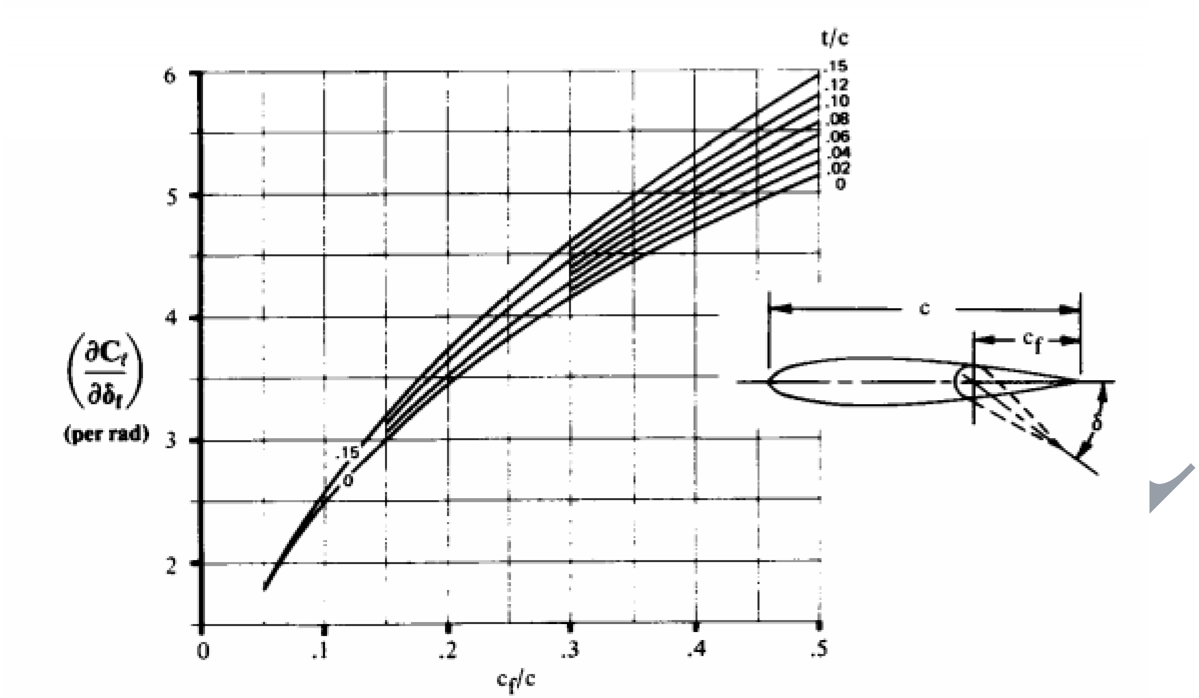


Figure 4.5: Theoretical Lift increment for plain flaps

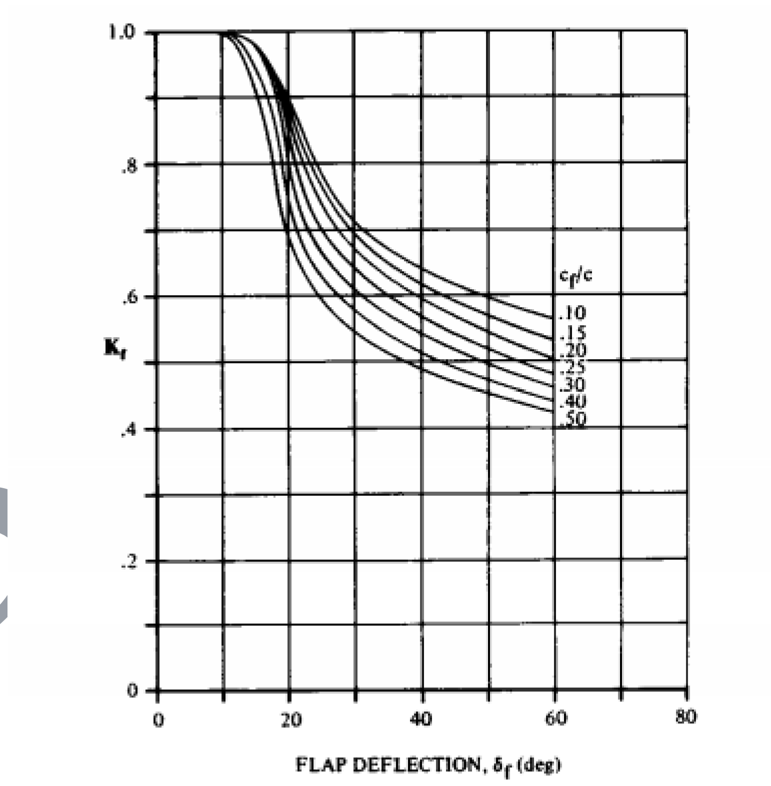


Figure 4.6: Empirical correction for plain lift increment

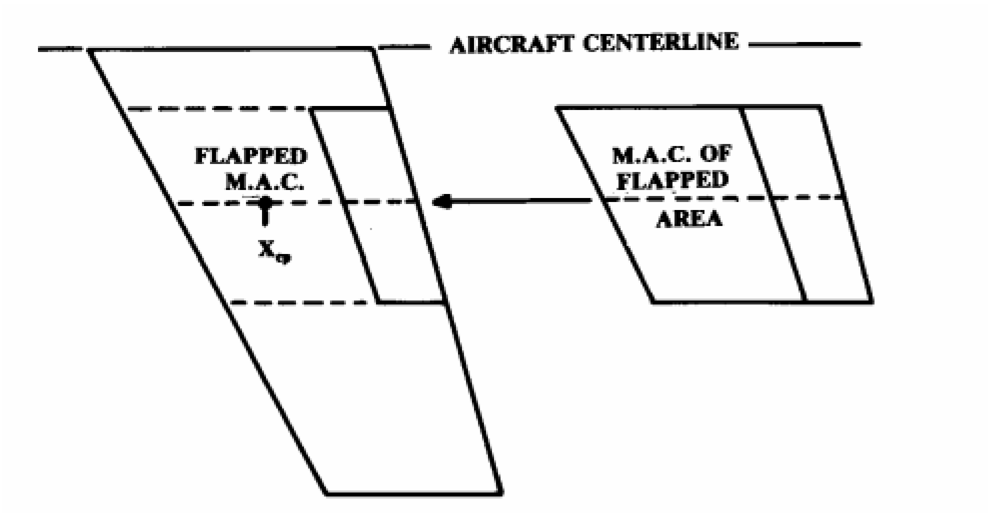


Figure 4.7: Flapped area and flapped M.A.C

4.3 LATERAL-DIRECTIONAL STATIC STABILITY & CONTROL

YAW/ROLL MOMENT EQUATIONS AND TRIM:

The lateral-directional analysis resembles the longitudinal analysis in many ways. However, the lateral-directional analysis has two closely-coupled analyses: the yaw (directional) and the roll (lateral).

They both are driven by the yaw angle β, and that the roll angle ϕ actually has no direct effect upon any of the moment terms. Furthermore, the deflection of either rudder or aileron will produce moments in both yaw and roll.

The geometry for lateral analysis is illustrated in figure 4.7, showing the major contributions to yawing moment N and rolling moment L. By definition, yaw and roll are positive to the right and most of these terms have a zero value when the aircraft is in straight and level flight. Also, by the sign conventions used for β and yaw, a positive value of yawing moment derivative with respect to β is stabilizing. However, a negative value of the rolling moment derivative with respect to β is stabilizing (dihedral effect).

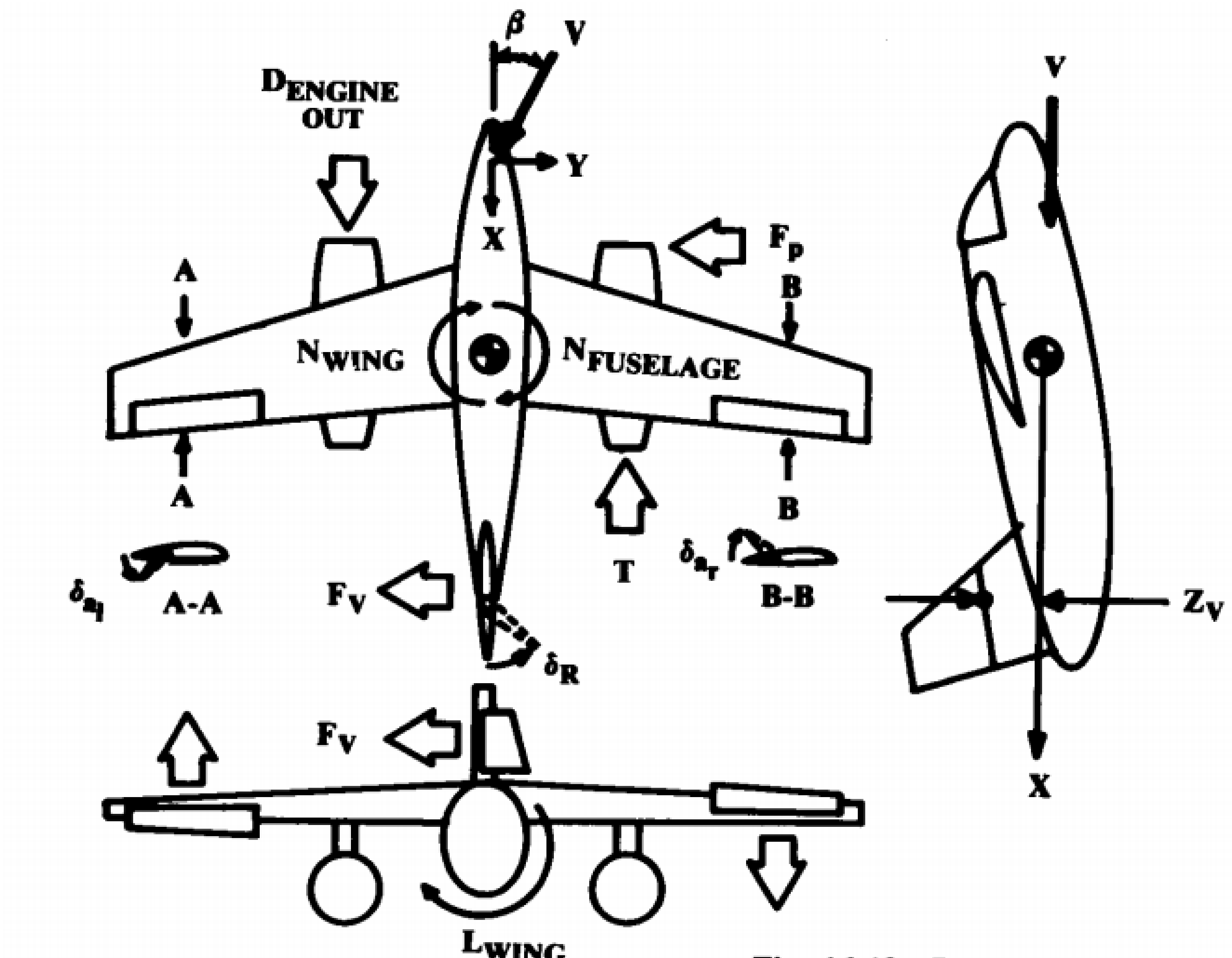


Figure 4.7 Lateral Geometry

The major yawing moment is due to the lateral lift of the vertical tail, denoted by F_v. This counteracts the fuselage yawing moment, which is generally negative to the sense shown in the figure. Rudder deflection acts as a flap to increase the lateral lift of the vertical tail.

The wing yawing moment can be visualized as an increase in drag on the side of the wing that is more nearly perpendicular to the oncoming glow. If the wing is swept aft, this yawing moment is stabilizing as shown.

Another wing yawing moment occurs with aileron deflection. The wing with increased lift due to aileron deflection has more induced drag, so the yawing moment is in the opposite direction from the rolling moment due to the aileron deflection. This is known as "adverse yaw".

The engines have the same three effects upon lateral moments that they have on longitudinal moments (direct thrust, normal force and propwash or jet-induced flowfield effects). In yaw, the thrust is balanced unless an engine fails. Then the remaining engine(s) create a huge yawing moment which is made worse by the drag of the failed engine.

In roll, the major influence is the wing rolling moment due to dihedral effect. This rolling moment tends to keep the aircraft level because it sideslips downward whenever a roll is introduced. The dihedral effect rolls the aircraft away from the sideslip direction.

The ailerons, the primary roll-control device, operate by increasing lift on one wking and reducing it on the other. The aileron deflection δ_a is defined as the average of the left and right aileron deflections in the directions shown. Positive aileron deflection rolls the aircraft to the right.

“Spoilers” are an alternative roll-control device. These are plates that rise up from the top of the wing, usually just aft of the maximum-thickness point. This disturbs the airflow and “spoils” the lift, dropping the wing on that side. Spoiler deflection also increases drag, so the wing yaws in the same direction that it rolls.

The vertical tail contributes positively to the roll stability because it is above the c.g. The moment arm for the vertical tail roll contribution is from the vertical tail MAC to the x-axis in the stability axis system. This X-axis is through the c.g. and is aligned with the relative wing. Thus, this term changes substantially with angle of attack.

$$N = N_{\text{wing}} + N_{w_{\delta_a}}\delta_a + N_{\text{fus}} + F_v(X_{acv} - X_{\text{cg}}) - TY_p - DY_p - F_p(X_{\text{cg}} - X_p) \quad \text{---Eq(4.13)}$$

$$L = L_{\text{wing}} + L_{w_{\delta_a}}\delta_a - F_v(Z_v) \quad \text{----Eq(4.14)}$$

The lateral lift force on the vertical tail appears in both equations. This is similar to the horizontal-tail lift, and must be calculated using the local dynamic pressure and angle of sideslip. The local angle of sideslip is less than the freestream sideslip angle because of a “sidewash” effect largely due to the fuselage. The tail lateral-lift-force derivative $C_{F\beta}$ is equivalent to $C_{L\alpha}$ in longitudinal notation and is calculated the same way,

$$F_v = q_v S_v \, C_{F_{\beta_v}} \frac{\partial \beta_v}{\partial \beta} \beta$$ ---Eq (4.15)

The yaw and roll moment equations are expressed in coefficient form by dividing through by ($qS_w b$) as shown in equations 4.16 and 4.18. Lengths are expressed as a fraction of wing span using the "bar" notation. The ratio between dynamic pressure at the tail and the freestream dynamic pressure is denoted by η_v. The vertical tail contributions to yaw and roll are expressed by the derivatives defined by equations 4.17 and 4.19.

Yaw:

$$C_n = \frac{N}{qS_w b} = C_{n_{\beta_w}}\beta + C_{n_{\delta_a}}\delta a + C_{n_{\beta_{fus}}}\beta + C_{n_{\beta_v}}\beta - \frac{T\overline{Y}_p}{qS_w} - \frac{D\overline{Y}_p}{qS_w} - \frac{F_p}{qS_w}(\overline{X}_{cg} - \overline{X}_p)$$ Eq14.16)

where

$$C_{n_{\beta_v}} = C_{F_{\beta_v}} \frac{\partial \beta_v}{\partial \beta} \eta_v \frac{S_v}{S_w} (\overline{X}_{acv} - \overline{X}_{cg})$$ Eq (14.17)

Roll:

$$C_\ell = \frac{L}{qS_w b} = C_{\ell_{\beta_w}}\beta + C_{\ell_{\delta_a}}\delta_a + C_{\ell_{\beta_v}}\beta$$ Eq(14.18)

where

$$C_{\ell_{\beta_v}} = -C_{F_{\beta_v}} \frac{\partial \beta_v}{\partial \beta} \eta_v \frac{S_v}{S_w} \overline{Z}_v$$ Eq (14.19)

CONTROL SURFACE SIZING:

The primary control surfaces are the ailerons (roll), elevator (pitch) and rudder (yaw). Final sizing of these surfaces is based upon dynamic analysis of control effectiveness, including structural bending and control-system effects. For initial design, the following guidelines are offered.

AILERON SIZING:

The required aileron area can be estimated from figure 4.8. In span, the ailerons typically extend from about 50% to about 90% of the span. In some aircraft. The ailerons extend all the way out to the wing tips. This extra 10% provides little control effectiveness due to the vortex flow at the wingtips, but can provide a location for an aileron mass balance.

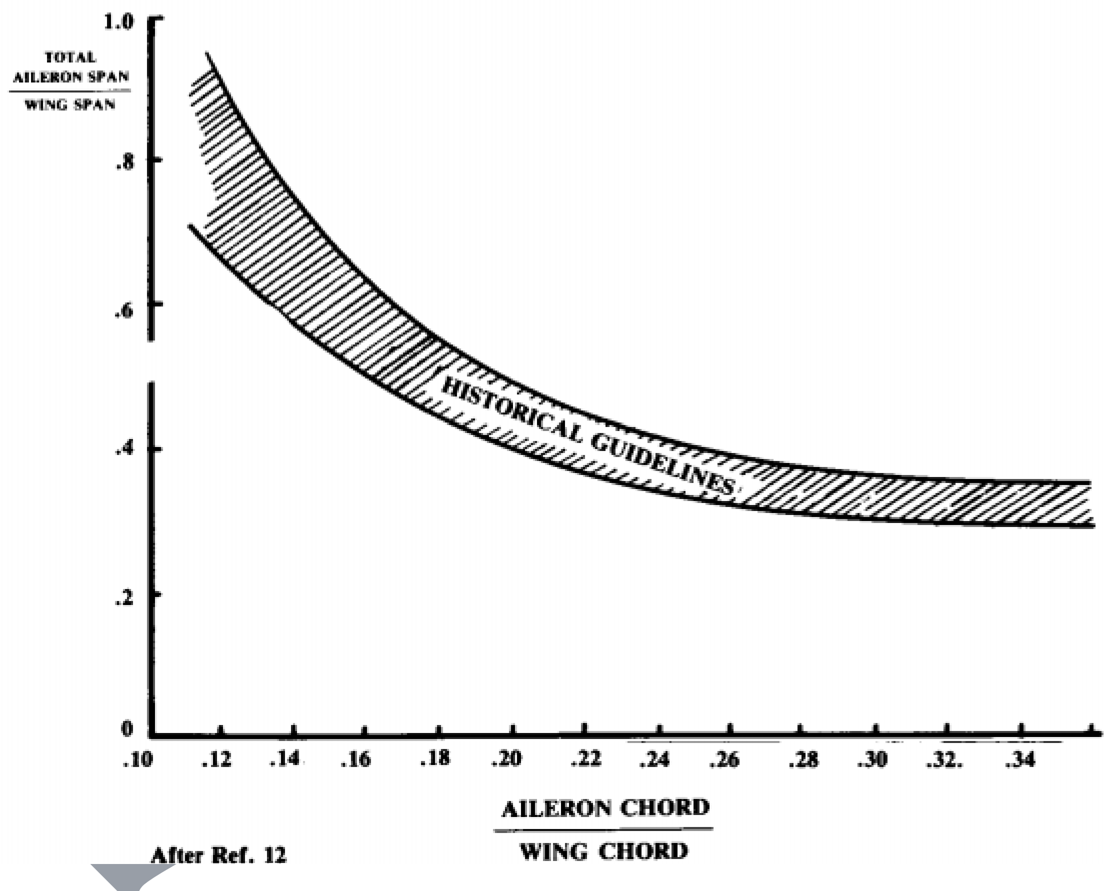


Figure 4.8: Aileron guidelines

Wing flaps occupy the part of the wing span inboard of the ailerons. If a large maximum lift coefficient is required, the flap span should be as large as possible. One way of accomplishing this is through the use of spoilers rather than ailerons. Spoilers are plates located forward of the flaps on the top of the wing, typically aft of the maximum thickness point.

Spoilers are deflected upward into the slipstream to reduce the wing's lift. Deploying the spoiler on one wing will cause a large rolling moment.

Spoilers are commonly used on jet transports to augment roll control at low speed and can also be used to reduce lift and add drag during the landing rollout. However, because spoilers have very nonlinear response characteristics they are difficult to implement for roll control when using a manual flight control system.

High-speed aircraft can experience a phenomenon known as "aileron reversal" in which the air loads placed upon a deflected aileron are so great that the wing itself is twisted. At some speed, the wing may twist so much that the rolling moment produced by the twist will exceed the rolling moment produced by the aileron, causing the aircraft to roll the wrong way.

To avoid this, many transport jets use an auxiliary, inboard aileron for high-speed roll control. Spoilers can also be used for this purpose. Several military fighters rely upon "rolling tails" (horizontal tails capable of being deflected nonsymmetric ally) to achieve the same result.

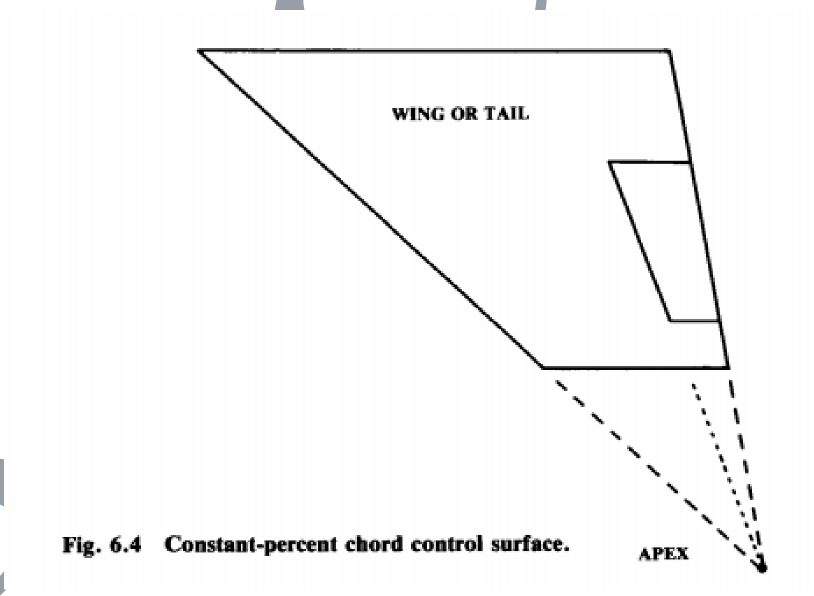


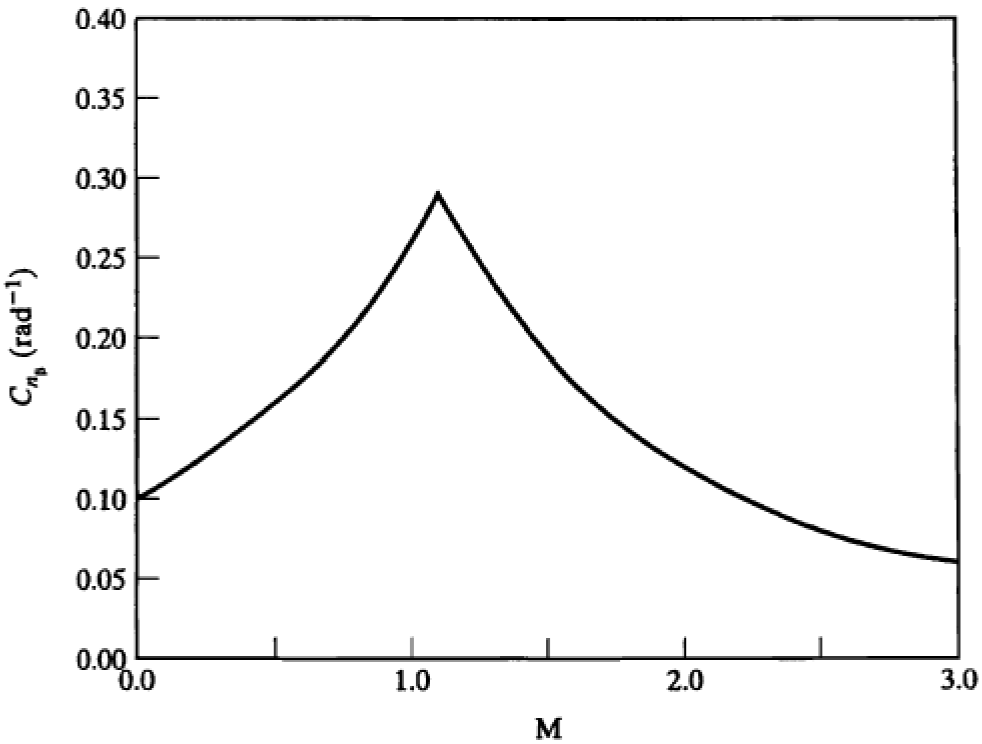
Figure 4.9: Constant-percent chord control surface

•Rolling motion is controlled by ailerons on the main wing

•They change the effective camber of the wing

•Ailerons are located at wing tip

• When the ailerons are deflected, they produce moment about fuselage longitudinal axis



$$C_{\mathcal{L}_\beta} = -C_{n_\beta}.$$

RUDDER SIZING:

Control surfaces are usually tapered in chord by the same ratio as the wing or tail surface so that the control surface maintains a constant percent chord (fig 4.9). this allows spars to be straight-tapered rather than curved, Ailerons and flaps are typically about 15-25% of the wing chord. Rudders and elevators are typically about 25-50% of the tail chord.

Control-surface “flutter”, a rapid oscillation of the surface caused by the airloads, can tear off the control surface or even the whole wing. Flutter tendencised are minimkized by using mass balancing and aerodynamic balancing.

Mass balancing refers to the addition of weight forward of the control-surface hingeline to counterbalance the weight of the control surface aft of the hingeline. This greatly reduces flutter tendencies. To minimize the weight penalty, the balance weight on a boom flush to the wing tip. Others bury the mass balance within the wing, mounted on a boom attached to the control surface.

The rudder area is sized in order to provide directional control capable of holding zero sideslip angle

- An asymmetric power condition caused by having one engine out at a velocity of 1.2 V_{TO} (equal to 1.44 V_S)
- Landing and takeoff in a cross wind of 0.2 V_{TO} and a sideslip angle of 11.5° Considering asymmetric power condition

Elevators and rudders generally begin at the side of the fuselage and extend to the tip of the tail or to about 90% of the tail span. High-speed aircraft sometimes use rudders of large chord which only extend to about 50% of the span. This avoids a rudder effectiveness problem similar to the aileron reversal.

$$C_n = 0 = -\frac{(T + D_e)}{qS_W b} + C_{n_{\delta_R}}\delta_R,$$

$$C_{n_{\delta_R}} = \frac{dC_n}{d\delta_R}.$$

$$D_e = C_d q S_e,$$

Considering the cross-wind condition at equilibrium

$$C_n = 0 = C_{n_\beta}\beta + C_{n_{\delta_R}}\delta_R$$

Rudder control is proportional to the transverse lift produced by the rudder deflection times the moment arm

$$C_{n_{\delta_R}} \simeq 0.9\,(C_{L_\alpha})_{VS}\,\bar{V}_{VS}\frac{d\alpha_{0_L}}{d\delta_R}.$$

$$\frac{d\alpha_{0_L}}{d\delta_R} = -\frac{dC_l}{d\delta_R}\frac{1}{C_{l_\alpha}}K'.$$

$$\frac{C_R}{C_{VS}} = \frac{S_R}{S_{VS}}.$$

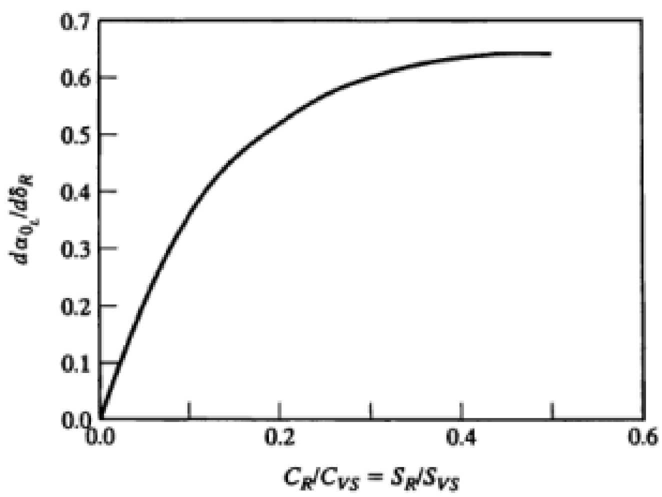
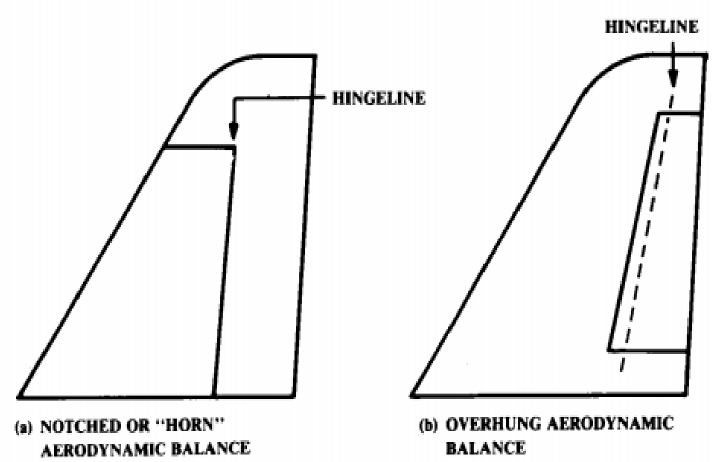


Figure 4.10: Aerodynamic balance



DESIGN SPREADSHEET

	Longitudinal Stability					
Fuselage Length						
L (f)	126					
Wing Center of Lift						
L_ctr (x/L)	0.6					
m.a.c. (ft)	27.2					
Load Summary (fuselage)						
Load Type	Magnitude (lbs)	x/L_start	x/L_end	resultant x/L	M @ C_lift f-lb (+ cw)	dw
Fuel	41261	0.4	0.6	0.5	−519888.6	8252.2
Payload	4000	0.1	0.8	0.45	−75600	266.67
Fus.Struct.	9977	0	1	0.5	−125710.2	475.1
Engine(s)	13000	0.7	0.7	0.7	163800	13000
Wing Struct.	4122	0.4	0.8	0.6	0	458
Horiz. Tail	228	0.15	0.2	0.18	−12209.4	114
Vert. Tail	334	0.8	1	0.9	12625.2	66.8
Other	16202	0	1	0.5	−204145.2	771.52
ΣL	89124			ΣM	−761.1E+3	
Tail Lift (req)	14213.41	0.15	0.2	0.18	−761.1E+3	7106.71
Center of Gravity						
X_cg / L	0.53					
X_cg (ft)	67.06	f				
Static Margin						
S.M.	0.31	stable				
Longitudinal Stability Coefficient:						
Wing Parameters:						
S_w	831	f^2				
(C_L_α)_w	0.04	(deg)^−1				
x_w	−8.54	f				
cbar	27.2	f				
Horiz. Tail Parameters:						
(C_L_α)_ht	0.03	(deg)^−1				
de/dα	0	Fig. 11.3				
η_ht	1					
l_ht	−45.01	f				
S_ht	74	f^2				
Engine Parameters						
mdot	135.4	lbm/s				
l_i	1.6	f				

FLYING QUALITIES:

Aircraft handling qualities are a subjective assessment of the way the plane feels to the pilot. Few modern pilots fully appreciate the great advances in handling qualities made since the dawn of aviation. Early fighters such as the Fokker Eindecker had handling qualities which were so poor that the pilots felt that without constant attention, the aircraft would “turn itself inside out or literally swap ends”.

A number of “goodness” criteria such as the wing helix angle is important that the aircraft have a nearly linear response to control inputs and that the control forces be appropriate for the type of aircraft. The control forces required due to flap deflection or power application should be small and predictable.

- Flying qualities of an Aircraft are related to the stability and control characteristics
- Pilot forms the subjective opinions about the ease or difficulty of controlling the airplane in steady and maneuvering flight
- The pilots impression of the airplane is also influenced by the feel of the airplane that is provided to the pilot by stick force

Level 1	Flying qualities clearly adequate for the mission flight phase.
Level 2	Flying qualities adequate to accomplish the mission flight phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists.
Level 3	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate,

Class I	Small, light airplanes, such as light utility, primary trainer, and light observation craft
Class II	Medium-weight, low-to-medium maneuverability airplanes, such as heavy utility/search and rescue, light or medium transport/cargo/tanker, reconnaissance, tactical bomber, heavy attack and trainer for Class II
Class III	Large, heavy, low-to-medium maneuverability airplanes, such as heavy transport/cargo/tanker, heavy bomber and trainer for Class III
Class IV	High-maneuverability airplanes, such as fighter/interceptor, attack, tactical reconnaissance, observation and trainer for Class IV

Cooper-Harper scale

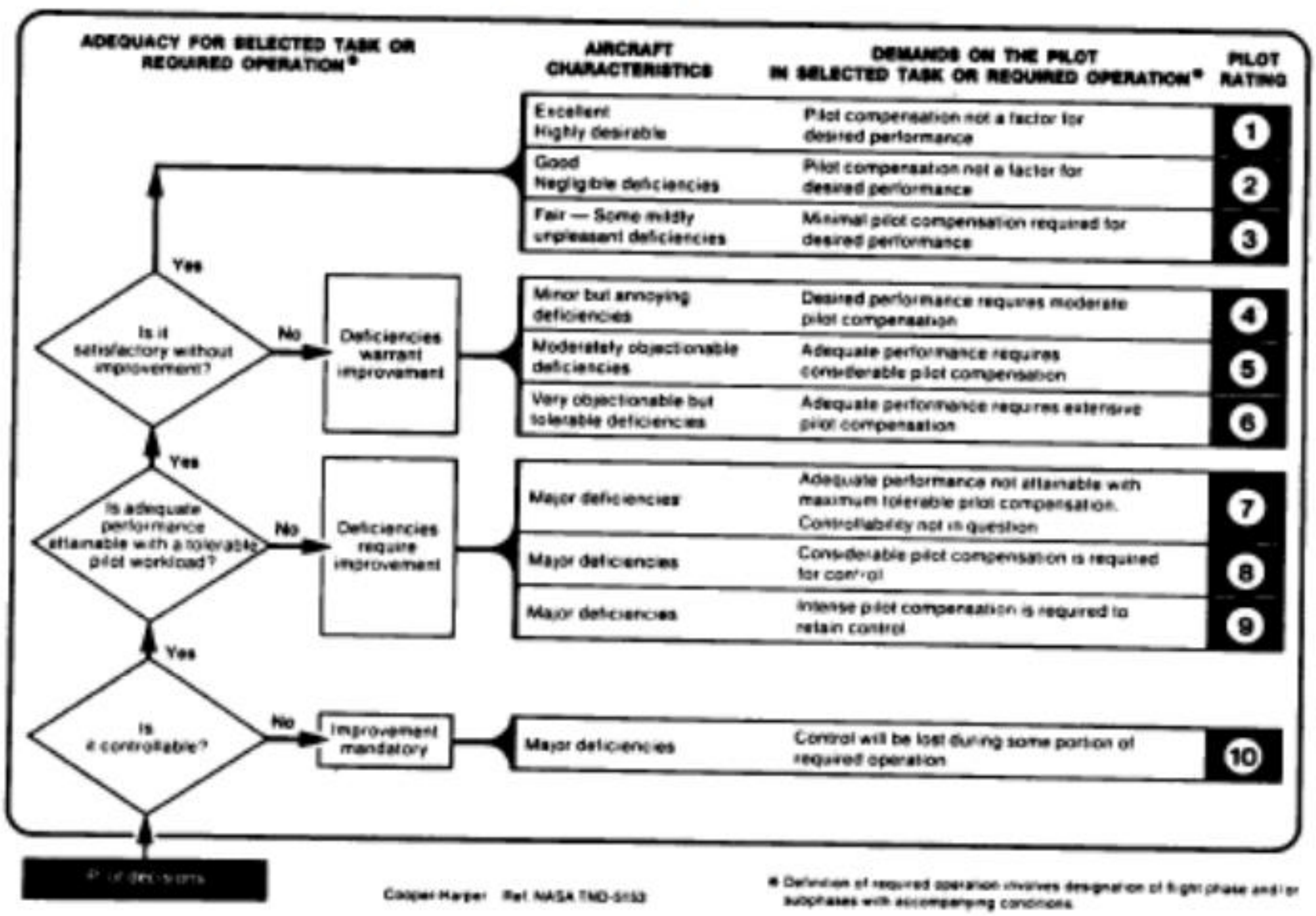


Figure 4.11: Cooper-Harper handling qualities rating scale.

ENVIRONMENTAL CONSTRAINTS:

- Since, the advent of large commercial jets in 1960, the noise profile has become an issue for residents living near Airport
- Litigation cases began to increase as a result of damages property and health
- Aircraft and Engine engineers strive to reduce noise during takeoff and landing
- Currently there are no civil Aircrafts operating at supersonic speed
- In 1980, concerns were raised on climate change to which engine emissions contribute. Again, regulatory agencies intervened to set achievable standards in order to limit pollution caused by engine exhaust gases.
- The disposal of life-expired, grounded aircraft must be considered in an early design phase