CHAPTER 9

Control surface design

9.1. Introduction

Proper design of control surfaces requires calculation of the dynamic stability of the airplane. This is an involved task. In the preliminary design, as part of this course, we rely on the assumption that if an airplane is statically stable, it would have reasonable dynamic stability. This is valid only for conventional airplanes. We follow the approach given by Ref.1.3, chapter 1-9. His approach can be summarized as follows.

(I) For longitudinal stability and control

(a) At the rear-most location of c.g., the airplane should be at least neutrally stable, for stick-free condition.

(b) At the foremost c.g. location , the elevator must be able to provide control (i.e. trim), at C_{Lmax} in landing configuration.

(c) Adequate control for nose wheel lift-off at

$$V = 0.85\ V_{t.o}.$$

(II) Directional stability:

(a) The vertical tail should provide desirable level of directional stability.

(b) Rudder should be powerful enough to provide directional control in cross wind take-off and landing, one engine inoperative condition, adverse yaw during roll and spin recovery.

(III) Dihedral:

The dihedral angle should be such as to give adequate level of lateral stability.

Remarks:

i) Before obtaining the tail parameters it is desirable to decide the configuration of the tail. Section 6.5 describes various configurations and their merits and demerits.

ii) It is assumed that the student has already undergone the course in airplane stability and control. Reference 1.16 and chapter 15 of Ref.1.11 may be consulted to revise the background.

9.2. Estimation of horizontal tail and elevator areas

Following Ref.1.3, chapter 1-9, the expression for $(dC_m/dC_L)_{free}$, with standard notations, is as follows.

$$(dC_m/dC_L)_{free} = (x_{c.g}/\bar{c}) - (x_{a.c}/\bar{c}) + (dC_m/dC_L)_{fus,nac}$$

$$+ (dC_m/dC_L)_{power} - (a_t/a_w)\ \bar{V}\eta_t\left(1-\frac{d\varepsilon}{d\alpha}\right)\left(1-\tau\frac{C_{h\alpha t}}{C_{h\delta e}}\right) \qquad (9.1)$$

If $(dC_m/dC_L)_{free}$ is zero at the rear most c.g location then, equating L.H.S of Eq.(9.1) to zero gives:

$$\frac{(X_{c.g.})_{aft}}{\bar{c}} = \frac{X_{a.c.}}{\bar{c}} - \left(\frac{dC_m}{dC_L}\right)_{fusnac} + \frac{a_t}{a_w}\bar{V}\eta_t(1-\frac{d\varepsilon}{d\alpha_w})$$

$$(1-\frac{C_{h\alpha t}}{C_{h\delta}}) + (\frac{dC_m}{dC_{L\delta}})_{power}$$

The expression for C_{mcg} in landing configuration is:

$C_{mcg} = (x_{cg}/\bar{c} \;\; - x_{ac}/\bar{c}\;) \, C_L + C_{mflap} + C_{mfuse,\, nac} +$

$C_{mpower} - C_{m\delta} \, [\delta_{e\,max} + (\alpha_w - \varepsilon - i_w + i_t)/\tau]$.

For $C_{mcg} = 0$ at C_L equal to C_{Lmax} near ground and c.g. at the most forward position with $\delta_e = (\delta_e)_{max}$.

We get :

$$\left(\frac{x_{c.g.}}{\bar{c}}\right)_{forward} = \frac{x_{ac}}{\bar{c}} - \frac{C_{mo}}{C_{L\,max}}\left[\frac{\delta_{e\,max} + \alpha_w - \varepsilon_G - i_w + i_t}{\tau}\right] + \frac{C_{mac\,flaps} + C_{m\,fusnac} + C_{mpower}}{C_{m\delta}} \qquad (9.2)$$

Reference 1.3, p. 1-9:4 suggests that the size of horizontal tail can be obtained by equating the difference between $(x_{c.g})_{aft}$ and $(x_{c.g})_{forward}$, as given by Eqs. (9.1) & (9.2), to Δx_{cg} which is the difference between the most aft and most forward c.g locations for the airplane under consideration i.e.,

$$\Delta x_{c.g.} = (x_{c.g.})_{aft} - (x_{c.g.})_{forward} \qquad (9.3)$$

Δx_{cg} is known from calculation of c.g. shift done in chapter 8. Substituting from Eqs.(9.1) and (9.2) on the right hand side of Eq.(9.3) we get:

$$\frac{\Delta x_{cg}}{\bar{c}} = \frac{x_{ac}}{\bar{c}} - \frac{x_{acf}}{\bar{c}} - \left(\frac{dC_m}{dC_L}\right)_{fus,nac} + \frac{a_t}{a_w}\bar{V}\eta_t\left(1 - \frac{d\varepsilon}{d\alpha_w}\right)\left(1 - \frac{C_{h\alpha}}{C_{h\delta}}\tau\right)$$

$$-\left(\frac{dC_m}{dC_L}\right)_{power} + \frac{C_m\delta_G}{C_{L\max}}\left(\begin{array}{l}\delta_{e\max} + \left(\frac{\alpha_{WG} - \varepsilon_G - i_w + i_t}{\tau}\right) \\ \qquad + \frac{C_{macflaps} + C_{mfus,nac} + C_{mpower}}{C_{m\delta}}\end{array}\right) \quad (9.4)$$

In the Eq. (9.4) we know Δx_{cg} .We also know

(i) For wing x_{ac}, x_{acflap}, a_w, α_w, ε_G, $d\varepsilon/d\alpha$, i_w, $C_{mac\ flap}$.

(ii) For fuselage: $C_{mfuse,\ nac}$, $(dC_m/dC_L)_{fuse,\ nac}$

(iii) For engine: $(C_m)_{power}$, $(dC_m/dC_L)_{power}$

As regards the horizontal tail, we proceed as follows .

a) From data collection choose the location of tail on fuselage. Knowing the location , l_t, $d\varepsilon/d\alpha$, η_t can be estimated from Ref.1.3, chapter 1-9.

b) From data collection choose aspect ratio (AR) , taper ratio (λ) , Sweep (Λ), $S_{elevator}/ S_t$ and type of aerodynamic balance. Knowing these parameters estimate a_t , τ , $C_{h\delta}$, $C_{h\alpha}$ and $C_{m\delta}$. It may be added that $C_{m\delta}$ is given by ($- \bar{V}\ a_t\ \eta_t\ \tau$). At this stage $\bar{V}$ is taken based on the data collection.

This value will be refined subsequently.

(c) The tail setting (i_t) can be determined from trim at C_{Lcr} without elevator deflection.

With the above information, solve Eq.(9.4) for $\bar{V}$. Since $l_t/\bar{c}$ is known, S_t can be calculated. If the value of $\bar{V}$ assumed to get $C_{m\delta}$ in step (b) is too different from the value obtained now, then iteration will be needed till the assumed value and the calculated value are almost the same.

9.2.1 Check for nose wheel lift off

During take-off, with a nose wheel landing gear, all the three landing gears are in contact with ground initially. When the speed approaches 85% of the take-off speed, the pilot applies the elevator, the airplane rotates and achieves angle of attack corresponding to take-off. This situation is called nose wheel lift off (NWLO). The elevator must have sufficient area to enable this.

From Ref.1.3, chapter 1-9, we get the following equation for moment at NWLO.

$$(C_{mcg})_{NWLO} = C_{mcg} + C_{mcgG} = C_{macflaps} + C_{mpower} + C_L \frac{x_a}{\bar{c}}$$

$$C_{mfus,nac} - a_t \bar{V}_G \eta_t (\alpha_{wG} - \varepsilon_G - i_w + i_t + \tau\delta_e) + \frac{L - W}{qS}\left(\frac{x_G + \mu y_G}{\bar{c}}\right) \quad (9.5)$$

Equating $(C_{mcg})_{NWLO}$ to zero, we solve Eq.(9.5) and obtain $\bar{V}$.

If this value of $\bar{V}$ is higher than that obtained earlier then choose this value .

See example in Ref.1.3, article 1-9:7 and also section 7 of Appendix 10.2.

9.3. Estimation of vertical tail area

The requirement of directional stability prescribes that $dC_n/d\psi$ or $(-dC_n/d\beta)$ be negative at all speeds above 1.2 V_s.

$$C_{n\psi} = (C_{n\psi})_w + (C_{n\psi})_{fus,\ nac} + (C_{n\psi})_{power} + (C_{n\psi})_{vt} \quad (9.6)$$

Reference 1.3, section 1-9:5 gives formulae to estimate contributions of various components. It is recommended that :

$$(C_{n\psi})_{desirable} = -0.00015 \text{ per degree.}$$

Reference 1.11, chapter 16 gives $(C_{n\beta})_{desirable}$ as function of Mach number. See Fig. 9.1 .

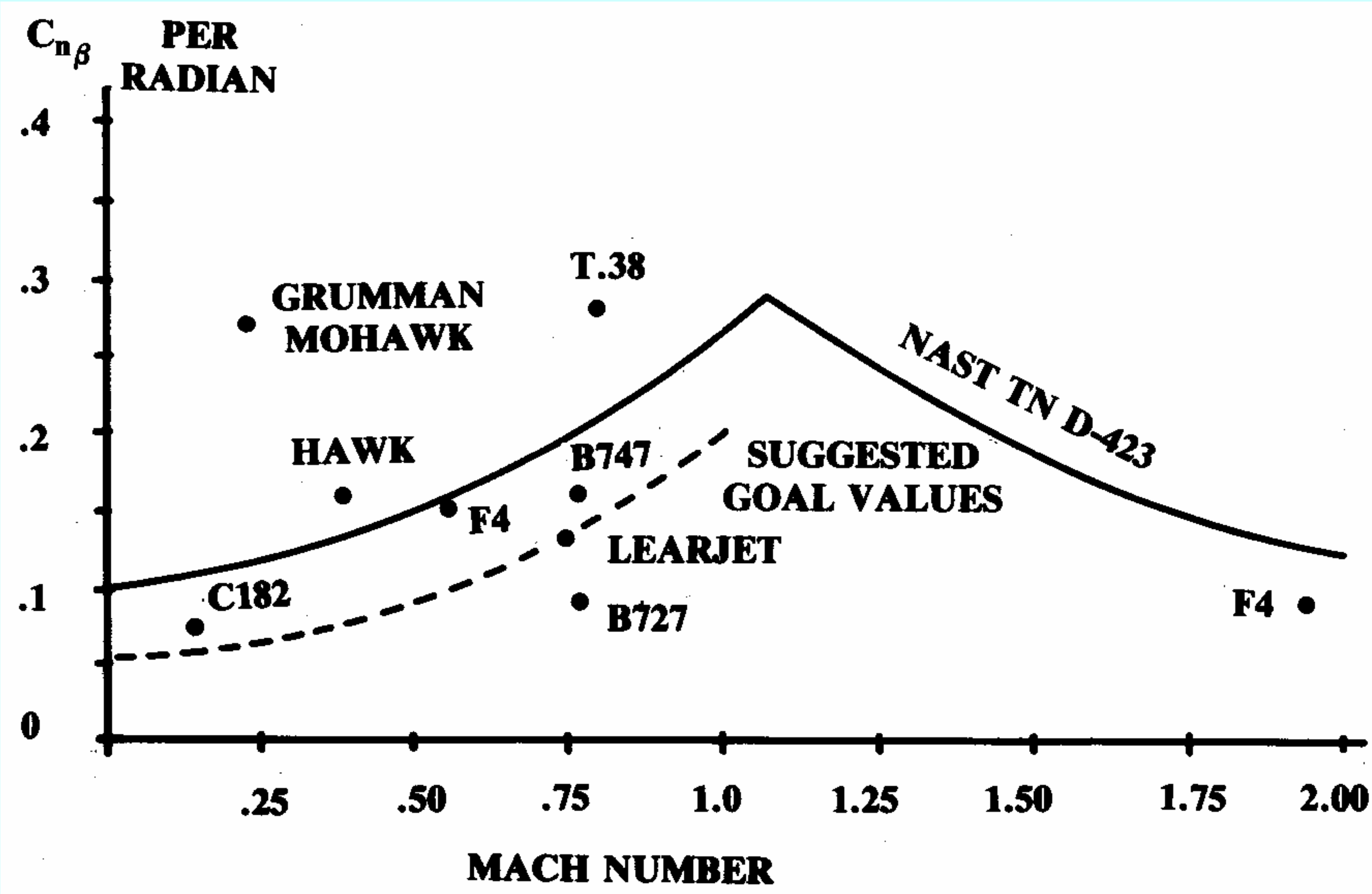

Fig 9.1 Guidelines for $C_{n\beta}$
(Adapted from Ref.1.11,chapter 16)

Following steps could be followed to obtain the area of the vertical tail :

a) Obtain $(C_{n\psi})_{wing}$, $(C_{n\psi})_{fuselage}$, $(C_{n\psi})_{nac}$, $(C_{n\psi})_{power}$. The last three terms can be grouped as $(C_{n\psi})_{f,n,p}$

$$(C_{n\psi})_{v.\ tail} = (C_{n\psi})_{desirable} - (C_{n\psi})_{wing} - (C_{n\psi})_{f,n,p} \quad (9.7)$$

b) Now $(C_{n\psi})_{v.tail} = -a_v (S_v/S)(l_v/b)\eta_v + K_i$ (9.8)

K_i = Interference factor which depends on wing location.

c) Assume vertical tail configuration from data collection and estimate a_v, η_v, l_v and b.

Now solve Eq.(9.8) to get S_v.

9.3.1 Area of rudder

Rudder must provide adequate control during the following situations.

a) Cross wind take-off and landing

b) One engine inoperative condition for multi-engined airplane.

c) Spin recovery

1) Cross wind take-off and landing:

For an airplane, the critical case prescribed is during take-off with cross wind equal to 20% of $V_{t.o}$. This gives a yaw angle(ψ_{cross}) of 11.5^0 or 0.2 radian .The moment due to cross wind equals $C_{n\psi}$ x ψ_{cross}.

This must be balanced by rudder deflection of δ_{rmax} which is taken as 20^o or 25^o. Note that moment due to rudder is given by:

$$(C_n)_{rudder} = - a_v \tau (S_v/S) (l_v/b)\eta_v \delta_{rmax} \qquad (9.9)$$

Using equation(9.9) obtain τ . Since τ depends on (S_{rudder} / S_v), the rudder area can be calculated .

2) One engine inoperative case :

For a multi-engine airplane, when an engine is inoperative, the other engine will cause a yawing moment. This moment has to be balanced by the rudder. The yawing moment due to engine will be almost constant with the flight speed. However the moment due to rudder is proportional to the square of the flight velocity. Hence there is a speed below which the maximum rudder deflection (δ_{rmax}) would not be able to balance the moment.

The requirement from this consideration is that the rudder should be effective till $V = 1.3V_s$

3) Spin recovery :

Rudder is the only control available to start recovery from a spin. During this flight a part of rudder is blanketed by the horizontal tail. The wake of the horizontal tail can be considered to be along a line 45^0 degree from the leading edge of the horizontal tail (Fig.9.2).

Reference 1.11, chapter 4 recommends that at least $1/3^{rd}$ of rudder must be left unblanketed for spin recovery. This calls for appropriate relative locations of horizontal tail and vertical tail.

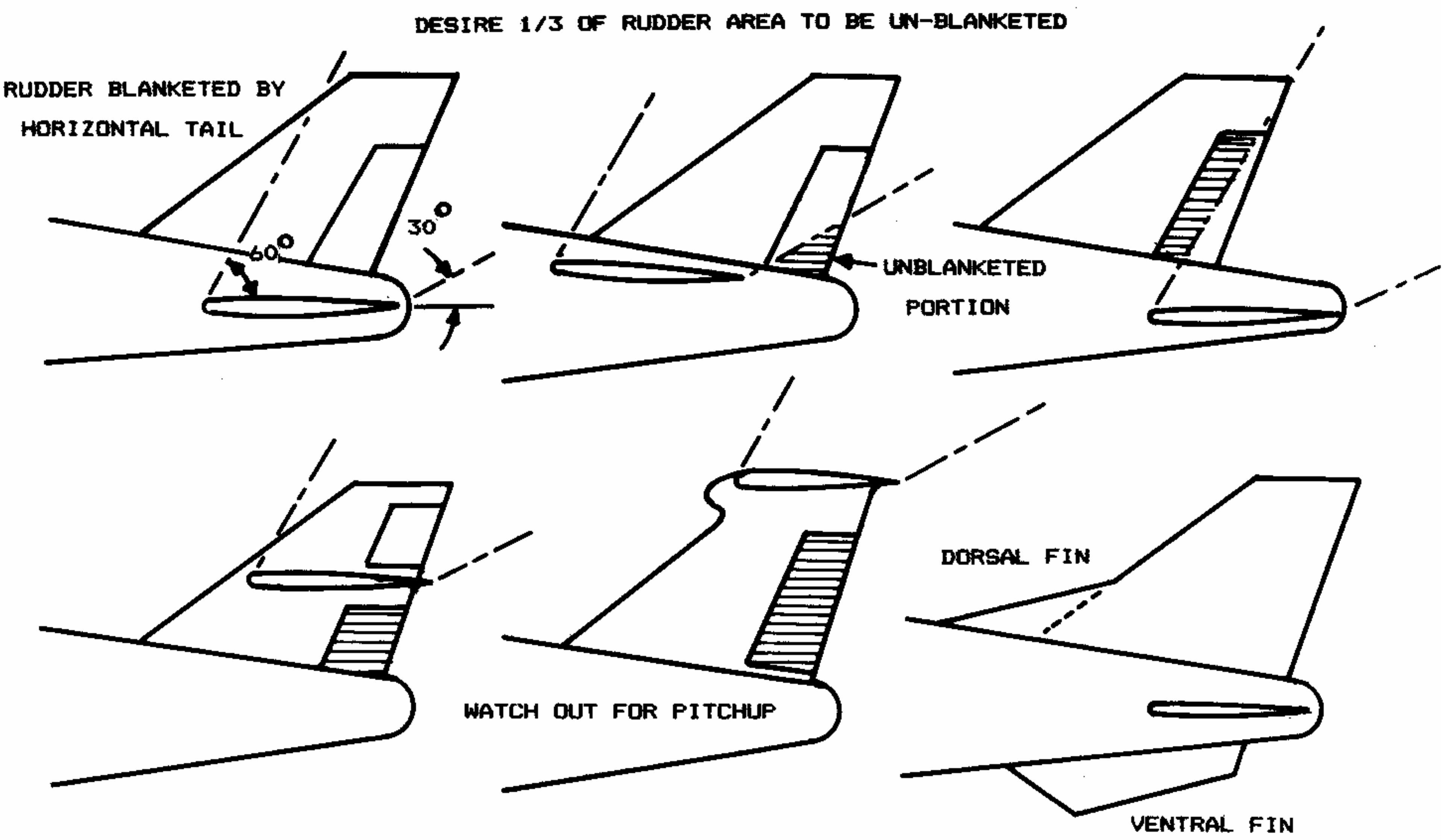

Fig 9.2 Guidelines for relative locations of horizontal tail and vertical tail from spin recovery consideration (Adapted from Ref. 1.11, chapter 4)

9.4 Dihedral angle:

When an airplane rolls it develops a side slip (β). This side slip produces a restoring rolling moment (C_l'). Hence a negative value of ($dC_l'/d\beta$) or $C'_{l\beta}$ is needed. Wing, fuselage, nacelle, power and vertical tail contribute to $C'_{l\beta}$. The dihedral angle (Γ) is chosen to provide for the difference between desired value of $C'_{l\beta}$ and the sum of contributions due to wing, fuselage, nacelle, power and vertical tail . The steps to obtain Γ are as follows.

$$\text{Calculate } C'_{l\beta} = (C'_{l\beta})_{\text{wing without dihedral angle}} + (C'_{l\beta})_{f,n,p} \qquad (9.10)$$

Ref.1.11, chapter 16 mentions that $C'_{l\beta}$ should be negative. Reference 5.2, chapter 9 suggests that:

$$(C'_{l\beta})_{desirable} = (-1/2)\, C_{n\psi} \qquad (9.11)$$

The difference between $C'_{l\beta}$ given by Eqs. (9.10) and (9.11) is provided by the dihedral. Reference 1.11, chapter 16 suggests that $C'_{l\beta}$ due to dihedral is approximately 0.0002 Γ where Γ is in degrees. This would provide first estimation of dihedral angle.

Remark:

Section 16.10, of Ref.1.11 deals with topics like handling qualities, lateral control departure parameter (LCDP) and spin recovery parameter. These are suggested as topics for self-study.

EXERCISES

9.1 Briefly explain as to how the following angles are chosen

(i) wing setting

(ii) wing twist

(iii) tail setting and

(iv) wing dihedral angle.