Chapter 5

Directional static stability and control - 3

Lecture 18

Topics

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5.8.3 Control in asymmetric power, steady flight after engine failure and minimum control speed

Control of the airplane in asymmetric power condition is critical for the design of rudder in multi-engined airplanes. The following changes take place when one of the engines of such an airplane fails (Ref.2.5, chapter 5).

(a) The engine that is operating causes a yawing moment T x y_p (Fig.5.10 a).

(b) In the case of engine propeller combination the drag (D_e) of the propeller will be large if it is held in the stopped condition. Generally the pitch of the propeller is adjusted so that it does wind milling. This change of pitch is called feathering of the propeller. In this situation, the drag due to propeller is small.

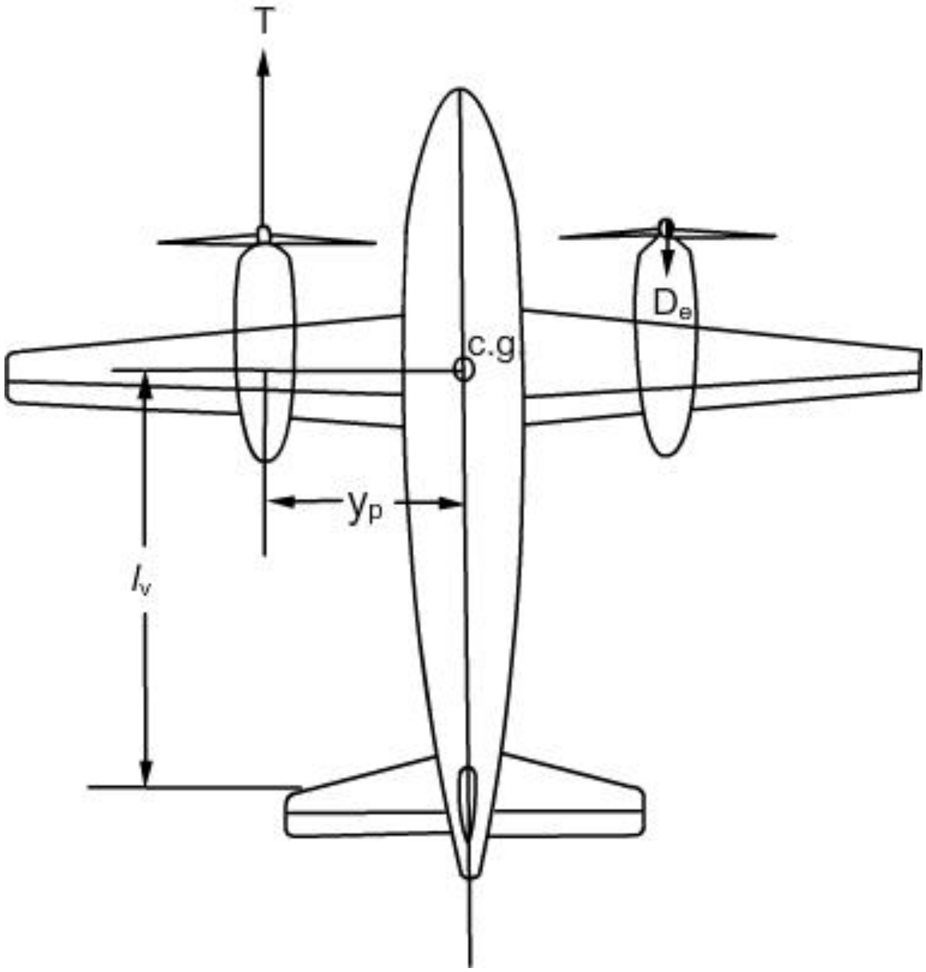

Fig.5.10a Airplane with one engine failure

(c) In the case of airplanes with jet engines, the failed engine is held in idling condition. The drag due to the failed engine causes a yawing moment which reinforces the yawing moment due to the operating engine. If the engine on the right wing has failed then the yawing moment due to the operating and the failed engines would cause a positive yawing moment (Fig.5.10 a).

$$N_e = \Delta T \times y_p \tag{5.28}$$

where, ΔT= thrust of live engine + drag of dead engine.

Or $C_{ne} = \dfrac{N_e}{\frac{1}{2}\rho V^2 Sb}$ (5.29)

(d) The engine failure may cause a small rolling moment in the case of engine propeller combination. The cause is as follows.

When the engines are on, a portion of the wing on the two wing halves is affected by the propeller slip stream. This effect (of slip stream) will be absent on the wing half with failed engine. Noting that the slip stream has a higher dynamic pressure, it is evident that when the engine on the right wing fails, the lift on it (right wing) will be slightly lower than that on the left wing.Then the airplane would experience positive (right wing down) rolling moment. The rolling moment coefficient due to engine failure can be denoted by C'_{le}.

(e) In the case of the engine failure on the right wing, C_{ne} and C'_{le} would both be positive. These cause the airplane to have a positive rate of yaw (turning to right) and positive rate of roll (right wing down). Consequently, the airplane sideslips towards the live engine ($\beta < 0$) and banks towards the dead engine ($\Phi>0$). The sideslip and the roll rate tend to increase the angle of bank (see sections 6.4 to 6.8 for rolling moment due to sideslip, $C'_{l\beta}$). If aileron is used to reduce the bank, it may cause more sideslip due to the effect of adverse yaw (see section 5.8.1). Hence, the usual practice is to counter the yawing motion by appropriate rudder deflection. Then, the ailerons are deflected to reduce the angle of bank which had developed in the meanwhile. Reference 2.5, chapter 5 may be consulted for details.

Steady flight after engine failure

Reference 2.5, chapter 5, gives various ways of achieving steady flight after engine failure. Two ways to achieve steady flight are described below.

a) The flight takes place with wings level ($\Phi = 0$). In this case, the airplane sideslips (Fig.5.10b). The side force due to the sideslip needs to be counteracted by side force from the rudder. As both the yawing moment due to engine failure (C_{ne}) and that due to sideslip ($C_{n\beta}\ \beta$) are in the same direction (Fig.5.10 b), fairly large rudder deflection is required. Reference 2.5, chapter 5, states that required β would be around 5^0.

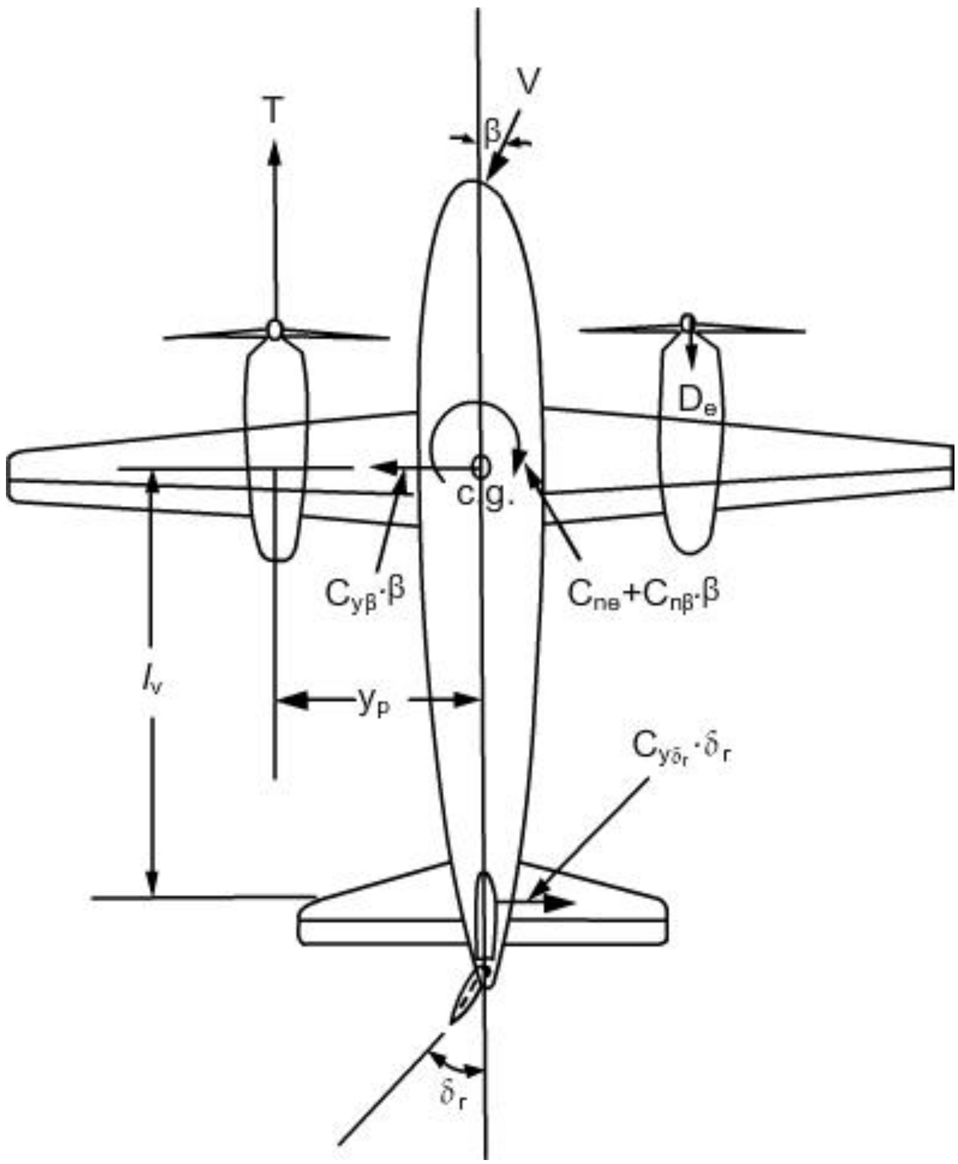

Fig.5.10 b Equilibrium with one engine failure and with wings level

b) In the second flight technique, the sideslip angle β is zero. In this case the side force is to be produced by banking the wing (live engine down). Thus, the side force on the vertical tail due to rudder is countered by the lateral component of airplane weight. The required angle of bank is within 3^0. Reference 2.5, chapter 5, recommended this procedure as it is favorable from the point of view of airplane performance.

Minimum control speed

The maximum yawing moment coefficient due to the rudder would be $C_{n\delta r}(\delta_r)_{max}$. This remains almost constant with speed. However, from Eq.(5.29) it is seen that the yawing moment due to engine (C_{ne}) increases as flight speed (V) decreases. For a jet engined airplane, if T is assumed to be nearly constant with V, then C_{ne} would increase as V^2. For an engine propeller combination when THP is nearly constant with speed, the thrust T would be proportional to 1 / V. Hence, C_{ne} would be proportional to V^3. These facts viz. ($C_{n\delta}(\delta_r)_{max}$) being constant and C_{ne} increasing as V decreases, indicate that there is a speed (V_{mc}) below which the full rudder deflection $(\delta_r)_{max}$ would not be able to control the airplane in the event of engine failure. This speed is referred to as minimum control speed (V_{mc}). Example 5.3 illustrates the procedure to calculate V_{mc}.

Example 5.3

Obtain the minimum control speed in the event of an engine failure for the following airplane:

$S = 65\ m^2$, $S_v = 6.5\ m^2$, $l_v = 10.5$ m, BHP = 880 kW (per engine),

propeller efficiency = 75%, $y_p = 4.2$ m, $dC_{Lv} / d\delta_r = 0.02\ deg^{-1}$, $(\delta_r)_{max} = 25^o$.

Solution:

Under equilibrium condition, the yawing moment due to rudder balances the moment which is due to failure of engine. Neglecting the yawing moment due to feathered propeller, the yawing moment due to operating engine is:

$T \times Y_p = \eta_p (BHP)\, y_p / V$;

where, η_p= propeller efficiency.

$$\text{Yawing moment due to rudder} = \frac{1}{2}\rho V^2\ \eta_v\ S_v\ l_v\ \frac{dC_{Lv}}{d\delta_r}\delta_r$$

For equilibrium:

$$\eta_p(BHP)\, y_p / V = \frac{1}{2}\rho V^2\ \eta_v\ S_v\ l_v\ \frac{dC_{Lv}}{d\delta_r}\delta_r \tag{5.30}$$

It is assumed that (a) flight is under sea level conditions (ρ = 1.225 kg / m^3), (b) η_p is constant, (c) δ_{rmax} = 25^0 and (d) η_v =1.0. From Eq.(5.30), V_{mc} is obtained as :

$$0.75\times\frac{880\times1000}{V_{mc}}\times4.2 = \frac{1}{2}\times 1.225\, V_{mc}^2\times 1\times\ 6.5\times10.5(-0.02)25$$

Or V_{mc} = 41.64 m/s or 149.9 kmph.

Remark:

In the above calculations, V_{mc} has been obtained in free flight. However, engine failure is more critical in take-off and landing conditions especially in the presence of the cross wind. Consequently, V_{mc} would be higher than that in the free flight. See Ref.2.5, chapter 5 for details.

5.8.4 Control for spin recovery

Spin is a flight condition in which the airplane wings are stalled and it moves downward rapidly along a helical path. The only control that is still effective is the rudder. The way to come out of the spin is to stop the rotation, go into a dive and pull out. The rudder must be powerful enough to get the airplane out of spin. Refer to section 10.1. for more information.

5.9 Need for rudder deflection in a coordinated turn

When an airplane performs a steady level turn it is going around a vertical axis with angular velocity $\dot{\psi}$ = V/R, where V is the flight velocity and R is the radius of turn. Figure 5.11 shows the flight when the airplane is turning to left. It is seen that a section on right wing at a distance ‘r’ from c.g. is moving forward with velocity $\dot{\psi}$ (R+r) or (V+$\dot{\psi}$ r).

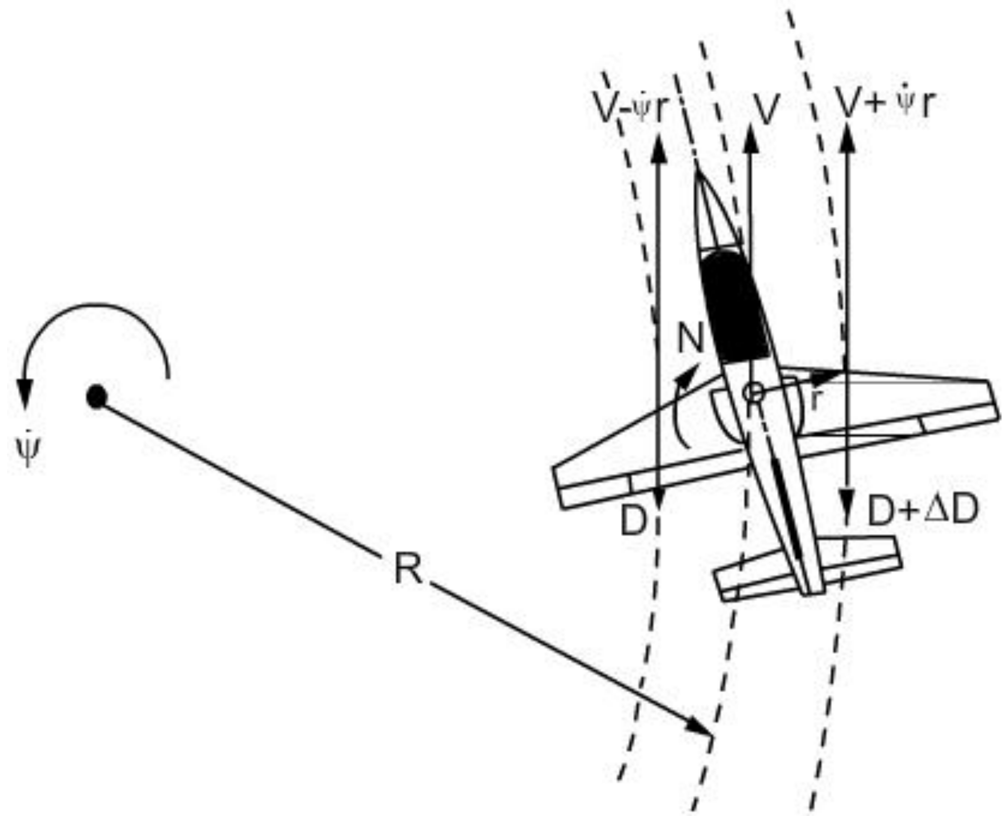

Fig.5.11 Adverse yaw during turn

Similarly, a section at a distance ‘r’ from c.g. on the lift wing is moving with velocity $\dot{\psi}$ (R-r) or (V- $\dot{\psi}$r). Thus, the section on the right wing experiences more dynamic pressure than that on the left wing. Hence, the drag of the right wing is more than that of the left wing and the airplane experiences a yawing moment. To prevent the airplane from side slipping or to execute a coordinated turn, rudder needs to be deflected. Thus, in a coordinated turn the aileron and rudder would have the following deflections.

(a) The bank angle of the wing is constant and it would appear that the ailerons should be brought to neutral after attaining the desired angle of bank. However, to compensate for the rolling moment due to yaw, the ailerons are given a small deflection.

(b) The rudder is deflected adequately to prevent sideslipping of the airplane.

5.10 Effect of large angle of side slip, rudder lock and dorsal fin

In order to understand the phenomenon of rudder lock, the following three points may be noted.

(1) At high values of β (greater than about 15^o), the vertical tail begins to stall and the following changes occur.

a) $C_{h\beta}$ and $C_{h\delta}$ change in such a way that δ_{free} is more positive than before (Fig.5.12).

b) Contribution of fuselage to C_n becomes nonlinear.

(2) The yawing moment coefficient(C_n) can be expressed as :

$$C_n = C_{n\beta}\ \beta + C_{n\delta r}\delta_r \tag{5.31}$$

where, $$C_{n\delta r} = \frac{\partial C_n}{\partial \delta_r} = -V_v\, \eta_v\, \tau_r\, C_{L\alpha v}\ ,\ \tau_r = \frac{\partial C_{Lv} / \partial \delta_r}{\partial C_{Lv}\ / \partial \alpha_v} \tag{5.31a}$$

Thus, the rudder deflection required to make C_n equal to zero or $(\delta_r)_{reqd}$ is:

$$(\delta_r)_{reqd} = -\frac{C_{n\beta}}{C_{n\delta r}}\beta \tag{5.32}$$

Note that $C_{n\beta}$ is positive and $C_{n\delta r}$ is negative. Hence, $(\delta_r)_{reqd}$ increases with β.

(3) It may be recalled from exercise 3.3 that, the control force is proportional to the difference between the control deflection required and the floating angle (δ_{free}).

Figure 5.12 shows the variations of $(\delta_r)_{reqd}$ and $(\delta_r)_{free}$ as functions of β. When β is greater than about 15^0, $(\delta_r)_{free}$ increases rapidly. It $((\delta_r)_{free})$ equals $(\delta_r)_{reqd}$ at $\beta = \beta_{rl}$ and then exceeds $(\delta_r)_{reqd}$ (Fig.5.12). In this situation, the pedal force would be reverse in direction. This phenomenon is called rudder lock as rudder may go to the mechanical stop to rudder deflection and get locked there (Ref.1.7, chapter 8).

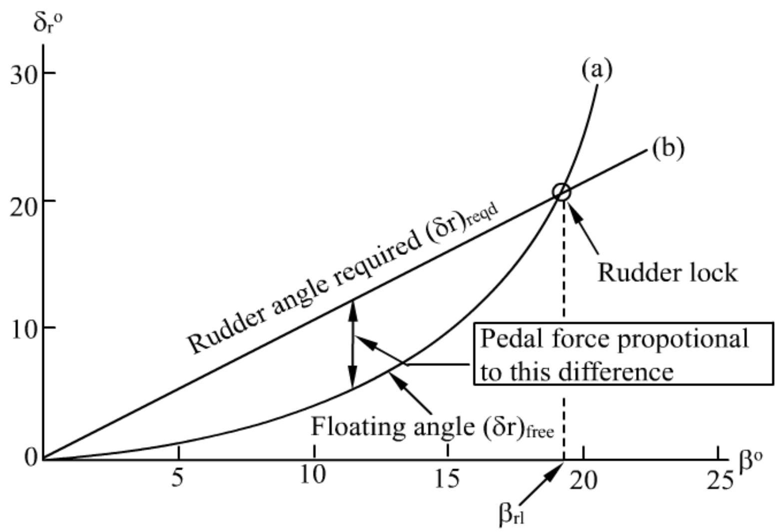

Fig.5.12 Rudder lock

Prevention of rudder lock

The rudder lock is prevented by adding a small extension, at the beginning of the vertical tail, as shown in Fig.5.13. It is called the dorsal fin (Fig.5.13). The way a dorsal fin prevents rudder lock can be explained as follows (Ref.3.1, chapter 15).

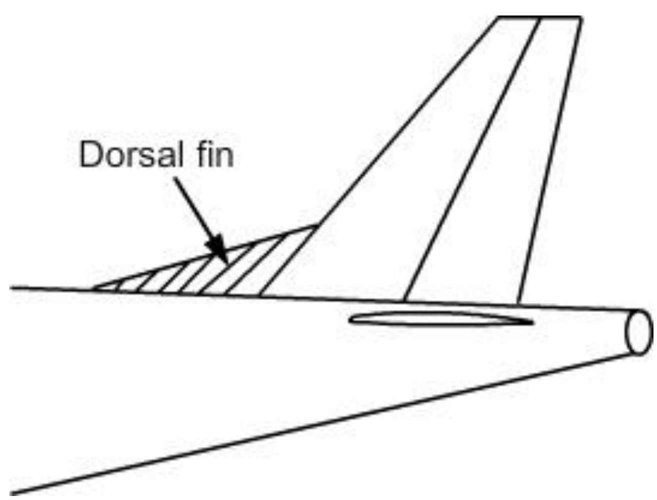

Fig.5.13 Dorsal fin

When wind tunnel tests were carried out on fuselage attached with fins at the rear, the following effects were observed.

(a) There was a change in C_{nf} vs β curve at high values of β (Fig.5.14).

(b) The stalling of the vertical tail was also delayed by dorsal fin. Reference 1.4, chapter 14 points out that the dorsal fin acts as a vertical slender delta wing which generates a strong vortex and delays separation of flow on the vertical tail. Thus, with dorsal fin added, the contribution of the fuselage plus tail does not change sign even at β values as high as 30^0 (Fig.5.14).

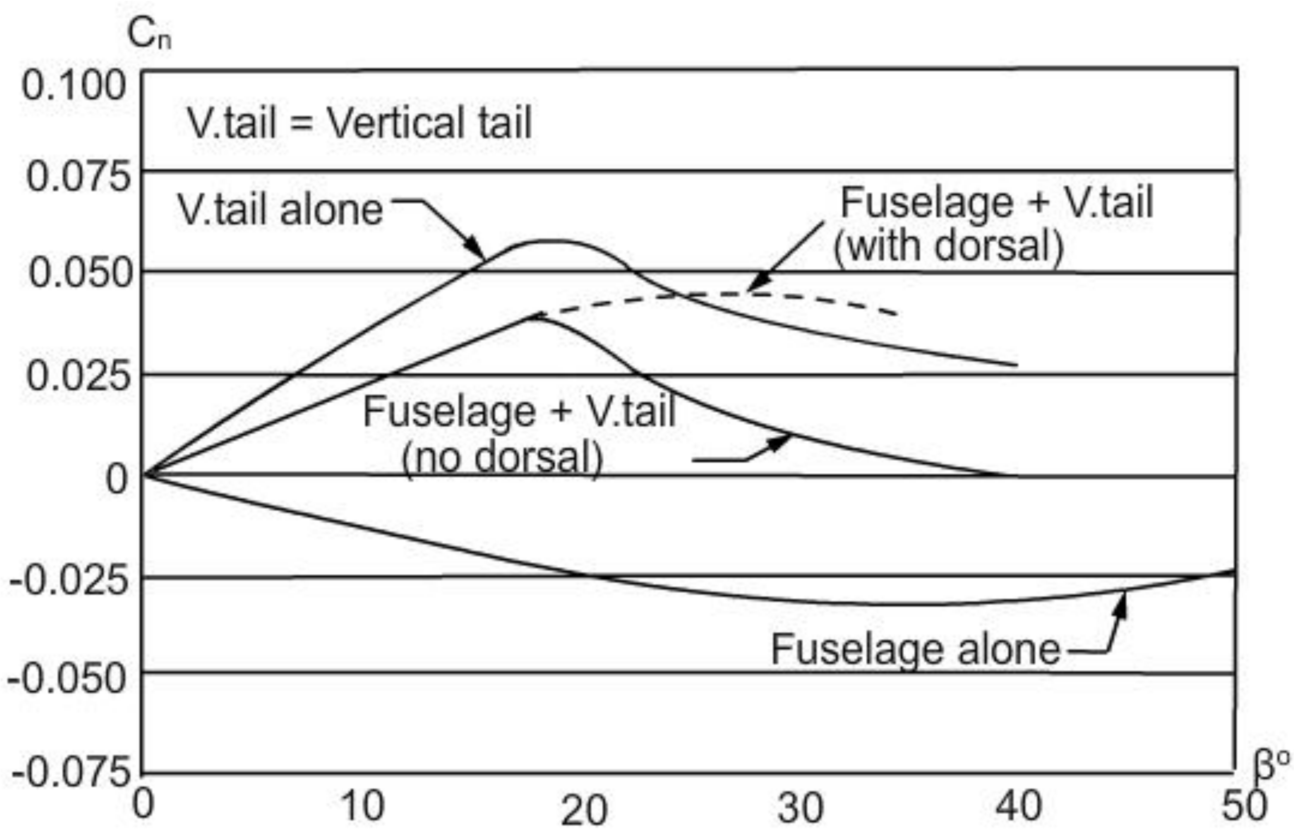

Fig.5.14 Effect of dorsal fin attached to vertical tail

(Adapted from “Dommasch, D.O., Sherby, S.S., Connolly, T.F. “Airplane aerodynamics” , Chapter 15 with permission from Pearson Education, Copyright C 1967)

The schematic variations of pedal force with and without dorsal fin are shown in Fig.5.15. As an application of dorsal fin, Fig. 5.16 shows a passenger airplane with dorsal fin.

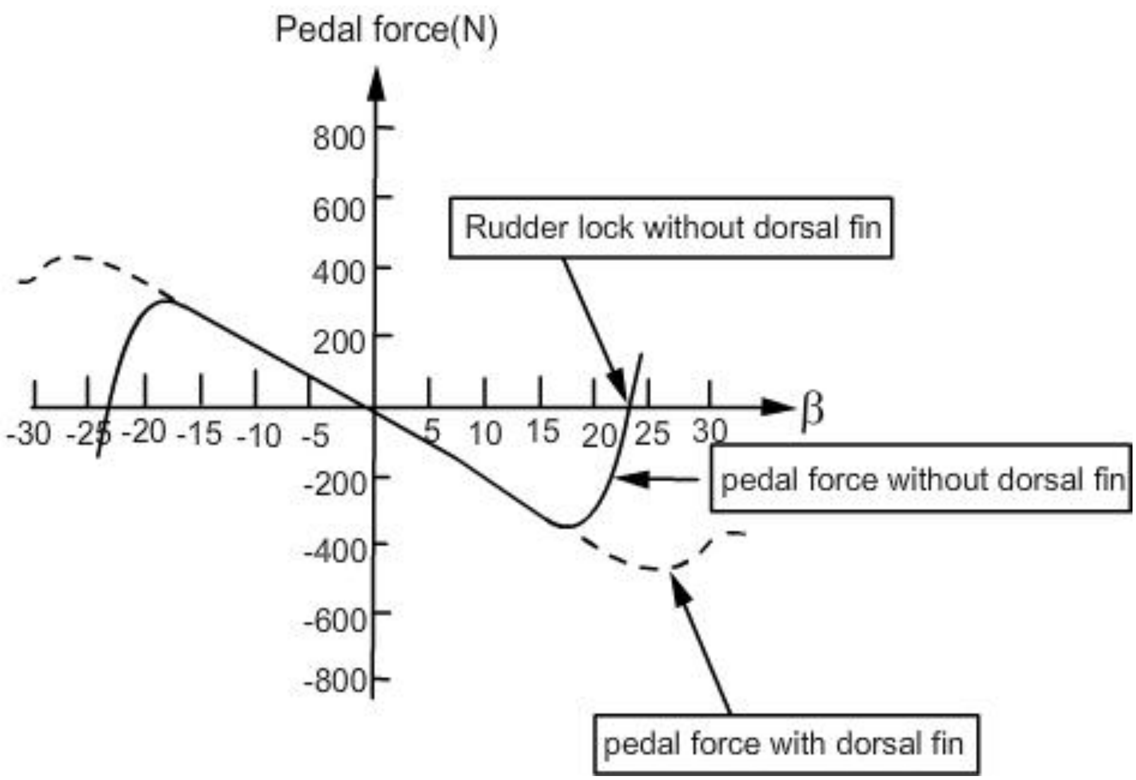

Fig. 5.15 Variation of pedal force with β (schematic)

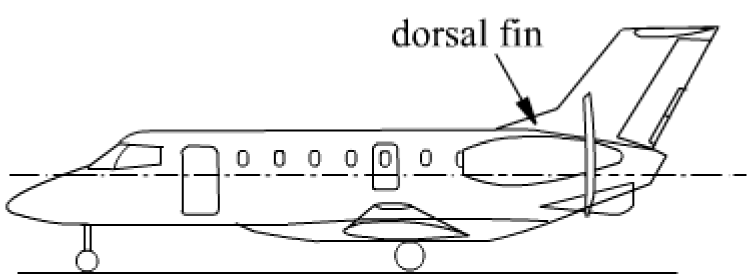

Fig.5.16 Airplane with dorsal fin

(Adapted from drawing of SARAS airplane supplied by

National Aerospace Laboratories, Bangalore, India)