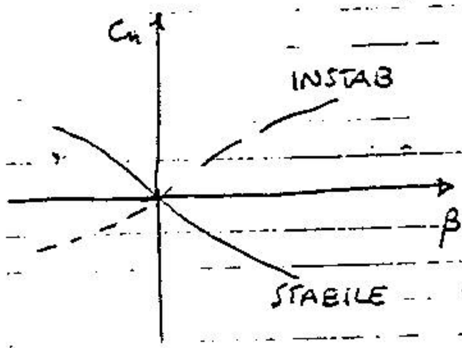
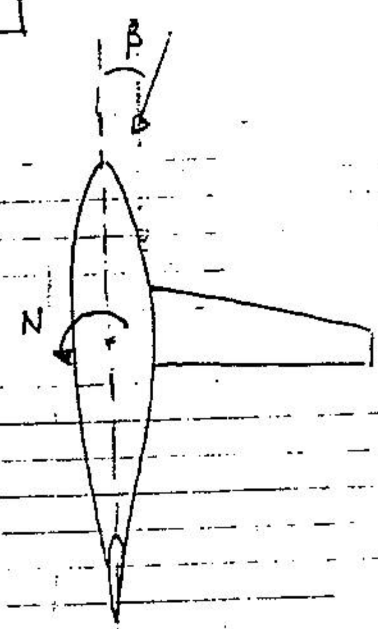
STABILITÀ DIREZIONALE

$$C_n = \frac{N}{q S b}$$





$$C_{n\beta} = \left(C_{n\beta}\right)_{wing} + \left(C_{n\beta}\right)_{FUS} + \left(C_{n\beta}\right)_{PROP} + \Delta_1 C_{n\beta} + \left(C_{n\beta}\right)_{VT}$$

<u>wing</u>

$$\left(C_{n\beta}\right)_{wing} = -\,0.00006\left(\Lambda_{c/4}\right)^{1/2}$$

<u>fusoliera</u>

$$\frac{dM}{d\alpha} = \frac{(\text{VOLUME})\, q\,(K_2 - K_1)}{28.7} = \left(\frac{dN}{d\beta}\right)_{FUS}$$

$$\left(C_{n\beta}\right)_{FUS} = \frac{\pi\,(K_2 - K_1)}{114.6\, S\, b}\int_0^l w_f^2\, dx$$

w_f diam equivalente

divido in tronchi

FORMULA empirica NASA

$$\left(C_{n\beta}\right)_{FUS} = \frac{0.96}{57.3} K_\beta \left(\frac{S_s}{S}\right)\left(\frac{L_f}{b}\right)\left(\frac{h_1}{h_2}\right)^{1/2}\left(\frac{w_2}{w_1}\right)^{1/3}$$

S_s side area

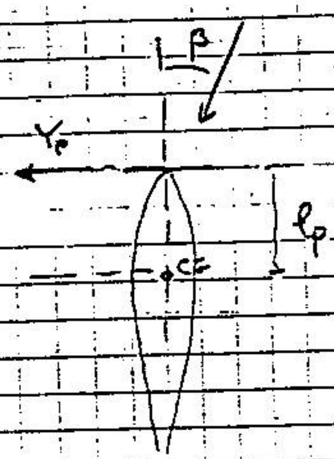
h_1 = altezza a 1/4 L_f — w_1 larghezza ad 1/4
h_2 " " 3/4 — w_2 " 3/4

K_β funz di d (posiz baricentro)

Interferenza (stabilizzante)

$\Delta_1 C_{n_\beta}$		
ALA	ALTA	− 0.0002
	MEDIA	− 0.0001
	BASSA	0

Propeller



$$C_n = \frac{Y_P \cdot l_P}{qSb}$$

$$C_{n_\beta} = \frac{\pi D^2 l_P}{4 S b}\left(\frac{dC_{Y_P}}{d\beta}\right) N$$

D = diametro

N = num di propulsori

$\left(\frac{dC_{Y_P}}{d\beta}\right)$ =	2 blades	.00165
	3	.00235
	4	.00296
	6	.00510

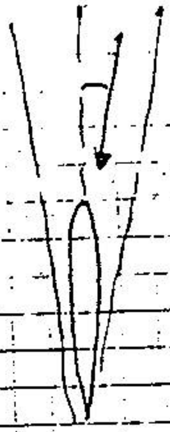
$$\left(C_{n_\beta}\right)_P = 1.5 \left(C_{n_\beta}\right)_{P_{PROP-WINDMILL}}$$

Solo un velivolo tutt'ala con elica spingente potrebbe essere stabile. Ci vuole il piano coda verticale per garantire la stabilità direzionale.

piano verticale

$$N = -L_V \cdot l_V = -\left(\frac{dC_L}{d\beta}\right)_V \beta\, q_V S_V l_V$$

$$C_n = -\left(\frac{dC_L}{d\beta}\right)_v \beta \frac{S_v}{S}\frac{l_v}{b}\frac{q_v}{q}$$



$\beta - \sigma$ B3 -

$\alpha_v = \beta - \sigma$ σ è il side-wash

$$\frac{d\alpha_v}{d\beta} = 1 - \frac{d\sigma}{d\beta}$$

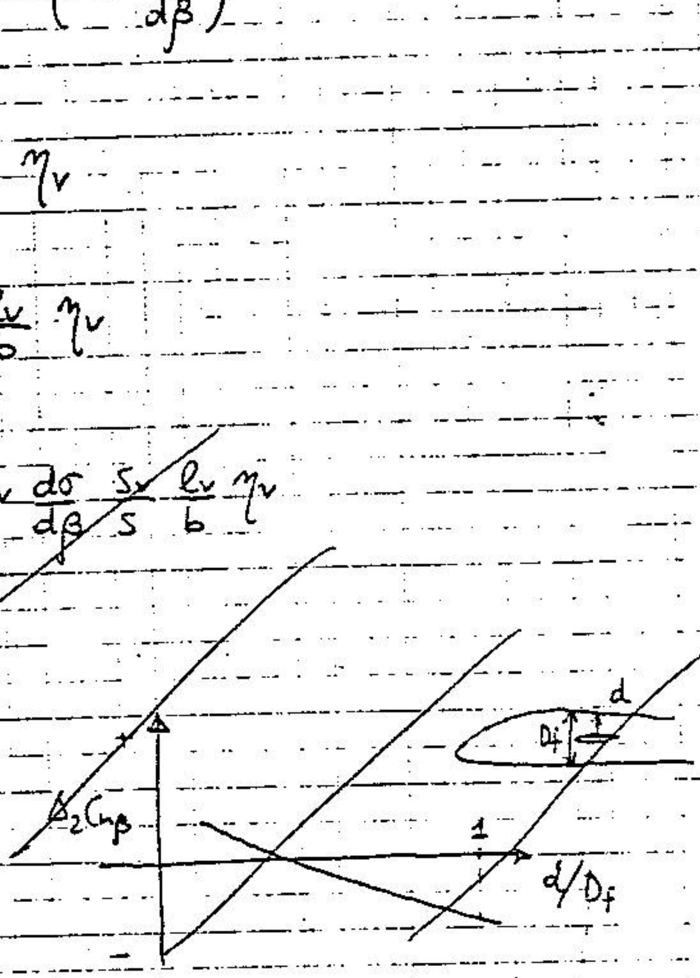
$$C_{L_v} = a_v \alpha_v \qquad \left(\frac{dC_L}{d\beta}\right)_v = a_v\left(1 - \frac{d\sigma}{d\beta}\right)$$

$$C_n = -a_v\left(1 - \frac{d\sigma}{d\beta}\right)\beta\frac{S_v}{S}\frac{l_v}{b}\eta_v$$

$$\left(C_{n_\beta}\right)_{V\,TAIL} = -a_v\left(1 - \frac{d\sigma}{d\beta}\right)\frac{S_v}{S}\frac{l_v}{b}\eta_v$$

$$= -a_v\frac{S_v}{S}\frac{l_v}{b}\eta_v + a_v\frac{d\sigma}{d\beta}\frac{S_v}{S}\frac{l_v}{b}\eta_v$$

$$= \left(C_{n_\beta}\right)_v + \Delta_2 C_{n_\beta}$$



Stimare $\frac{d\sigma}{d\beta}$ con la formula

$$\eta_v\left(1 - \frac{d\sigma}{d\beta}\right) = 0.724 + 3.06 \ldots \quad (2.81)$$

$a_v = a_v(A_e)$ vedi Perkins

$$A_e = 1.55\,\frac{b_v^2}{S_v} \qquad \text{(end plate effect)}$$

Così posso valutare $\left(C_{n_\beta}\right)_{VEL\,COMP.}$

$$C_{n_\beta} = \left(C_{n_\beta}\right)_{wing} + \left(C_{n_\beta}\right)_{fus} + \left(C_{n_\beta}\right)_{PROP} + \left(C_{n_\beta}\right)_{VT} + \Delta_1 C_{n_\beta} + \ldots$$

C_{n_β} $\quad -.0015 \quad -.0020$

MINIMO $-.0005$

Attenzione C_{n_β} dipende da b e S, mentre il contrib dell'ala è piccolo, quindi per velivoli con b ed S piccoli è giustif un valore più elevato di C_{n_β}.

$$\left(C_{n_\beta}\right)_{des} = -0.0005\left(\frac{W}{b^2}\right)^{1/2}$$

A <u>comandi liberi</u>

$$\left(C_{n_\beta}\right)_{VT_{free}} = -a_v \frac{S_v}{S}\frac{l_v}{b}\eta_v\left(1-\frac{C_{h_{\alpha v}}}{C_{h_{\delta_r}}}\tau\right)\left(1-\frac{d\sigma}{d\beta}\right)$$

CONTROLLO DIREZIONALE

$$C_n = -C_{L_v}\frac{S_v l_v}{S\, b}\frac{q_v}{q}$$

$$\frac{dC_n}{d\delta_r} = \underset{\substack{\uparrow \\ \text{POT CONTR} \\ \text{DERIVA}}}{C_{n_{\delta_r}}} = -a_v\,\tau\,\frac{S_v}{S}\frac{l_v}{b}\eta_v$$

$$\left(C_{n_{\delta_r}} \simeq -.001\right)$$

Raffica laterale

EQUILIBRIO

$$\frac{1}{2}\rho V^2 S b\, C_{n_\beta}\, \beta = \left(\frac{1}{2}\rho V^2\right)_v S_v\, a_v\, \tau_v\, \delta_v\, l_v$$

$$\delta_v = \frac{1}{\eta_v}\frac{S b\, C_{n_\beta}}{S_v\, a_v\, \tau_v\, l_v}\,\beta$$

Raffica $V_{est} = 20$ m/s V = vel crociera $= 103$ m/s

$\beta = 10°$

Ricavare δ_v per $\beta = 5°$ $\left[\begin{matrix}\delta_v < 20° \\ \text{sempre}\end{matrix}\right]$

$\beta = 10°$

PIANTATA di 1 motore

Piantata di 1 motore critico (più esterno nel caso di 4 motori)

CG MAX arretrato

esempio (TIPO P68 Partenavia)

$l_p = 2$ m

$l_v = 4.5$ m

$(X_{CG} = 34\%)$

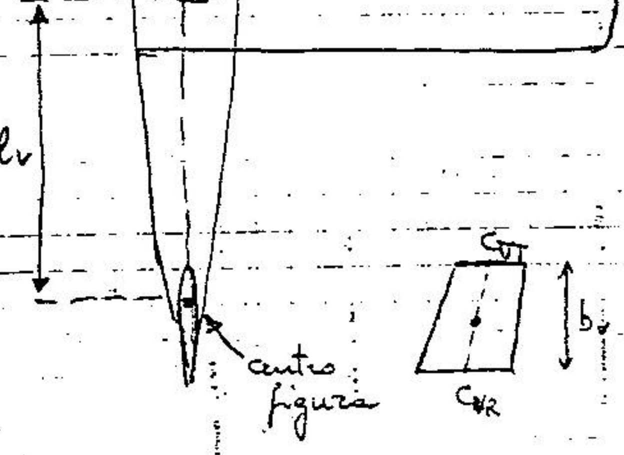
$W = 2500$ kg

$V_{s1} = 65$ kts $= 120\ \frac{km}{h}$

$\Pi_a = 300$ CV

(Per ogni motore)

$\eta_p = 0.80$ $\eta_v = 0.95$

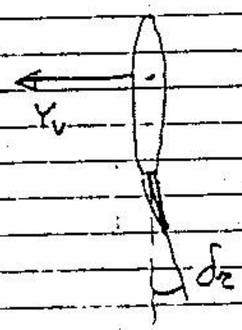


$$M = \frac{T_a \eta_p}{V} l_p + M_\beta$$

nel caso che si prevede di continuare la derapata con un certo angolo β (es 5°)

$$M_T = \frac{(300 \cdot 75)\, 0.8 \cdot 2}{V} = \frac{36\,000}{V} \qquad [kg \cdot m]$$

La deflen della deriva



$$M_v = Y_v \cdot l_v$$

$$Y_v = C_{Y_v} q_v S_v = C_{Y_v} \eta_v q S_v$$

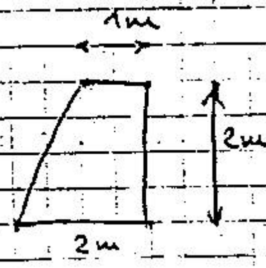
$$q = \frac{1}{2} \rho_{SL} V^2 = \frac{1}{2} \frac{1}{8} V^2 = \frac{1}{16} V^2 \qquad \left[\frac{kg}{m^2}\right]$$

$$M_v = C_{Y_v} \eta_v \frac{1}{2} \rho_{SL} V^2 S_v l_v = 0.95 \frac{1}{16} V^2 S_v C_{Y_v} \cdot 4.5$$

$$C_{Y_v} = a_v \tau \delta_v$$

$$a_v = \frac{a_{0v}}{1 + \frac{a_{0v}\, 57.3}{\pi AR_{Ev}}} \qquad AR_{Ev} = 1.55 \frac{b_v^2}{S_v} \quad \text{(end plate eff)}$$

Se suppongo di prendere certe dimensioni che mi danno un certo S_v e $\frac{b_v^2}{S_v}$



$$S_v = 3\,m^2 \qquad AR_v = \frac{4}{3} = 1.33$$

$$AR_{Ev} = 1.55 \cdot 1.33 = 2.06$$

$\tau_v \simeq 0.6$ (con $\frac{\text{area timone}}{\text{area totale}} \simeq 0.40$)

$\delta_{V\,MAX} = 25°$

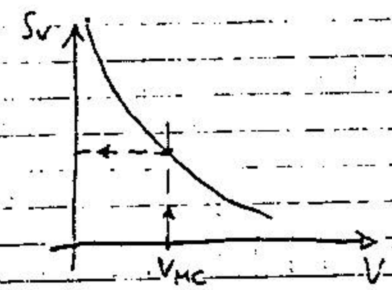
$a_{0_V} = 0.11$ (2D NACA 0012)

$a_V = 0.056 \left(\frac{1}{°}\right)$

$$C_{Y_V} = (.056) \cdot (0.6) \cdot (25) = 0.84$$

$$\frac{36000}{V} = 0.95 \frac{1}{16} \, 0.84 \, S_V V^2 \, 4.5$$

$$S_V = \frac{1.604 \cdot 10^5}{V^3}$$



$V_{MC} \leq 1.2 \, V_{S1}$ (vel stallo conf. decollo)

$V_{S1} = 33.47$ m/s

$V_{MC} = 1.2 \cdot 33.5 = 40.16$ m/s

$$\Rightarrow S_V = 2.48 \text{ m}^2$$

Quindi il piano coda disegnato prima va bene con $S_V = 3$ m²

$$V_{MIN} = \sqrt[3]{\frac{160400}{3}} \doteq 37 \frac{m}{s} \simeq 1.1 \, V_{S1}$$

il che appare conveniente.

In generale la procedura di progetto è

VELIV PLURIMOTORI

In base alle caratteristiche di velivoli simili disegnare il piano e vedere se con S_v e τ_v

→ PIANTATA MOT CRITICO a V_{mc} con $\delta_r = MAX$ (es $20 \div 25°$)

→ Raffica laterale - controllo

→ Stab. direz a com. liberi

VEL MONOMOTORI

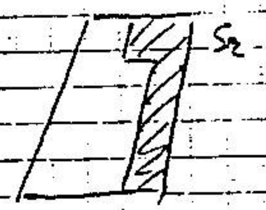
→ Raffica laterale

→ stab. direz a com liberi

Verificare anche gli SFORZI DI PEDALIERA

$$PF = G \cdot HM = G \left(C_{h\alpha_v} \beta + C_{h\delta_r} \delta_r + C_{h\delta_t} \delta_t \right) q \eta_v S_r C_r$$

$$\delta_r = \frac{d\delta_r}{d\beta} \beta$$



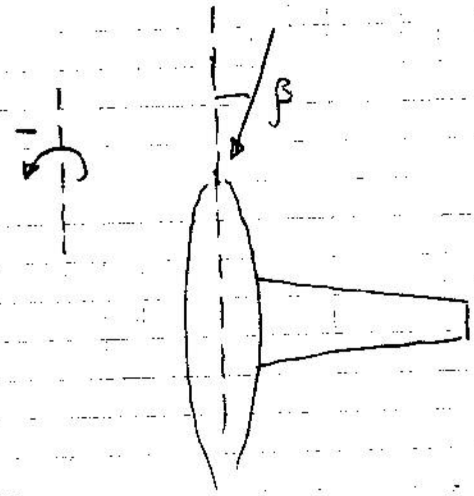
$$\frac{d\delta_r}{d\beta} = - \frac{(C_{n\beta})_{FIX}}{C_{n\delta_r}}$$

bilanc aerodinamico

$$\frac{dPF}{d\beta} = - \frac{G \, q \, \eta_v S_r C_r}{C_{n\delta_r}} C_{h\delta_r} \left(C_{n\beta} \right)_{free}$$

a $V = 150$ mph < 5 lb/deg

EFFETTO DIEDRO



$Cl_\beta < 0$ (Eff. diedro stabile)

$$Cl = \frac{L}{qSb}$$

VARI CONTRIBUTI

Ala ($\Gamma = \emptyset$)

$Cl_{\beta_{Wing}}$ usare figura 8.41

$$\left(\frac{Cl_\beta}{C_L}\right) = f(AR, \Lambda_{1/2}) \quad \text{per } \lambda = 0.5$$

Per rapp rostremaz diversi da 0.5 usare

$$Cl_\beta = -\frac{1+2\lambda}{3(1+\lambda)} C_L \tan\Lambda$$

Angolo diedro

$$Cl_{\beta_{W_\Gamma}} = -0.00021 \, K_\lambda K_A K_\Lambda \Gamma \qquad (\Gamma \text{ in } °)$$

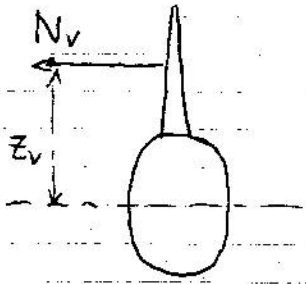
usare fig. 8.39

Interferenza ala-fus

$(\Delta Cl_\beta)_1 =$

−0.0006	ALA ALTA
0	ALA MEDIA
+0.0008	ALA BASSA

PIANO VERTICALE



$$Cl_{v} = -a_v \beta \eta_v \frac{S_v}{S} \frac{z_v}{b}$$

$$\left(Cl_\beta\right)_v = -a_v \eta_v \frac{S_v}{S} \frac{z_v}{b}$$

Interferenza (effetto ala sul p. verticale)

$\left(\Delta Cl_\beta\right)_2 =$	-0.00016	ALA ALTA
	0	ALA MEDIA
	0.00016	ALA BASSA

$$Cl_{\beta_{TOT}} = Cl_{\beta_w} + Cl_{\beta_f} + Cl_{\beta_v} + \left(\Delta Cl_\beta\right)_1 + \left(\Delta Cl_\beta\right)_2$$

L' effetto diedro non deve essere eccessivo

Un buon valore può essere

$$Cl_\beta = \frac{1}{2} C_{n_\beta}$$

where S_s is the projected side area in square feet, L_f is the over-all fuselage length, and the dimensions h_1, h_2, w_1, w_2 can be obtained by referring to the sketch in Figure 8–4. The values of the empirical constant K_β are given in Figure 8–4 as functions of fuselage fineness ratio, L_f/h, and location of the c.g. on the body, d/L_f.

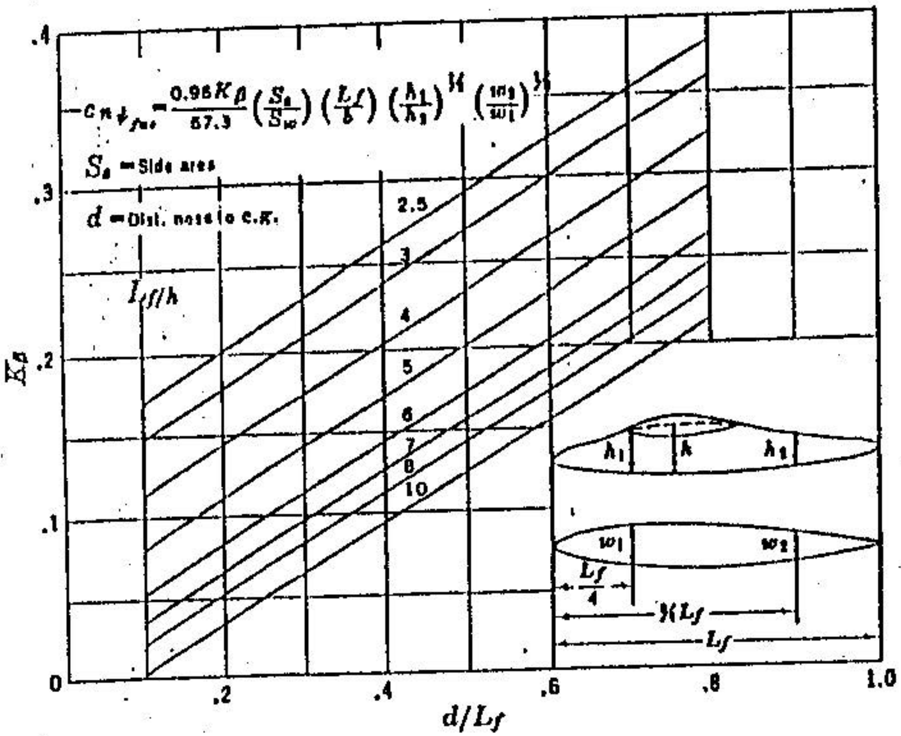


Figure 8–4. Fuselage directional stability coefficient. From unpublished data of North American Aviation, Inc.

D11

The directional stability of the combination of wing plus fuselage is usually slightly different from the sum of the two components obtained separately. This is due to the interference flow created at the wing-fuse... ...ature. This interference effect is usually slightly stabiliz-

The running propeller can have large effects on the ...tional stability, destabilizing if a tractor and stabili... propeller. The propeller contribution to directional ... from the side force component at the propeller disc created because of yaw as discussed in Chapter 5. (See Figure 8–5.)

The yawing moment coefficient produced by the side force on the propeller at some angle of yaw is simply:

$$C_n = \frac{Y_p \times l_p}{qS_w b} \qquad (8\text{–}6)$$

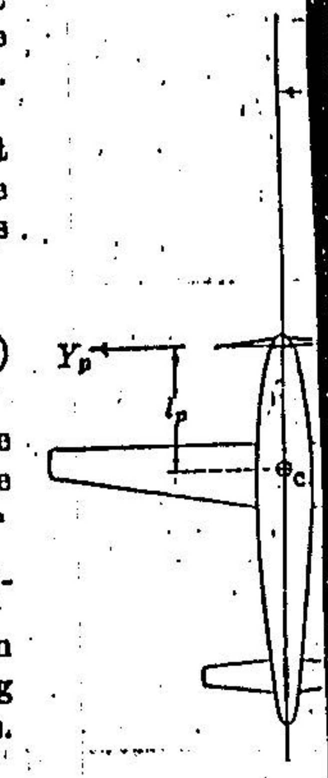


Figure 8–5. Prop... moment.

The cross wind force, Y_p, can be expressed in terms of the side force coefficient, C_{Y_p}, where $C_{Y_p} = \frac{Y_p}{q(\pi D^2/4)}$. The stability contribution of the running propeller can be obtained by differentiating (8–6) with respect to ψ in degrees.

$$C_{n\psi} = \frac{\pi D^2 l_p \left(\frac{dC_{Y_p}}{d\psi}\right) N}{4S_w b} \qquad (8\text{–}7)$$

where N is the number of propellers, and l_p the distance ... disc to the center of gravity.

The derivative $dC_{Y\psi}/d\psi$ is the same as the deriva... discussed in Chapter 5. The value of this derivative ...

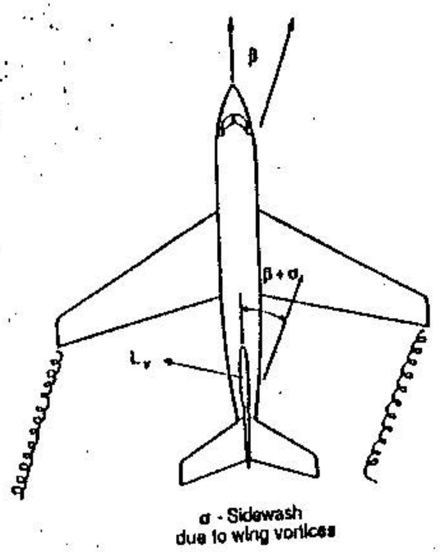


FIGURE 2.30
Vertical tail contribution to directional stability.

where the subscript v refers to properties of the vertical tail. The angle of attack α_v that the vertical tail plane will experience can be written as

$$\alpha_v = \beta + \sigma \tag{2.76}$$

where σ is the sidewash angle. The sidewash angle is analogous to the downwash angle ε for the horizontal tail plane. The sidewash is caused by the flow field distortion due to the wings and fuselage. The moment produced by the vertical tail can be written as a function of the side force acting on it:

$$N_v = l_v Y_v = l_v C_{L_{\alpha_v}}(\beta + \sigma) Q_v S_v \tag{2.77}$$

or in coefficient form:

$$C_n = \frac{N_v}{Q_w S b} = \frac{l_v S_v}{S b}\frac{Q_v}{Q_w} C_{L_{\alpha_v}}(\beta + \sigma) \tag{2.78}$$

$$= V_v \eta_v C_{L_{\alpha_v}}(\beta + \sigma) \tag{2.79}$$

where V_v is the vertical tail volume ratio and η_v is the ratio of the dynamic pressure at the vertical tail to the dynamic pressure at the wing.

The contribution of the vertical tail to directional stability can now be obtained by taking the derivative of Eq. (2.77) with respect to β:

$$C_{n_{\beta_v}} = V_v \eta_v C_{L_{\alpha_v}}\left(1 + \frac{d\sigma}{d\beta}\right) \tag{2.80}$$

...ation for estimating the combined sidewash and tail

efficiency factor η_v is presented in Ref. 2.7 and is reproduced below.

$$\eta_v\left(1 + \frac{d\sigma}{d\beta}\right) = 0.724 + 3.06\frac{S_v/S}{1 + \cos\Lambda_{c/4w}} + 0.4\frac{z_w}{d} + 0.009AR_w \tag{2.81}$$

where S = the wing area
S_v = the vertical tail area, including the submerged area to th[e] fuselage centerline
z_w = is the distance, parallel to the z axis, from wing root quarte[r] chord point to fuselage centerline
d = is the maximum fuselage depth
AR_w = is the aspect ratio of the wing
$\Lambda_{c/4w}$ = sweep of wing quarter chord.

2.7 DIRECTIONAL CONTROL

Directional control is achieved by a control surface called a rudder which i[s] located on the vertical tail, as shown in Fig. 2.31. The rudder is a hinged fla[p] which forms the aft portion of the vertical tail. By rotating the flap, the li[ft] force (side force) on the fixed vertical surface can be varied to create a yawin[g] moment about the center of gravity. The size of the rudder is determined b[y] the directional control requirements. The rudder control power must b[e] sufficient to accomplish the requirements listed in Table 2.1.

The yawing moment produced by the rudder depends upon the change i[n] lift on the vertical tail due to the deflection of the rudder times its distanc[e] from the center of gravity. For a positive rudder deflection, a positive sid[e] force is created on the vertical tail. A positive side force will produce [a] negative yawing moment:

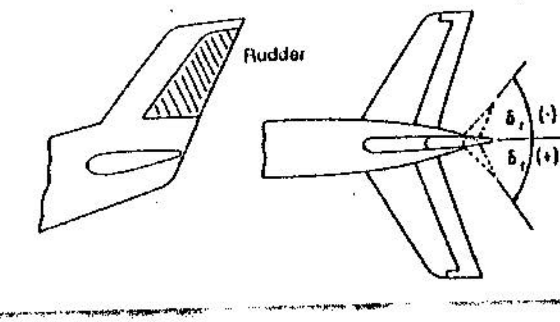
$$N = -l_v Y_v \tag{2.82}$$

where the side force is given by

$$Y_v = C_{L_v} Q_v S_v$$

Rewriting this equation in terms of a yawing moment coefficient yields

$$C_n = \frac{N}{Q_w S b} = -\frac{Q_v}{Q_w}\frac{l_v S_v}{S b}\frac{dC_{L_v}}{d\delta_r}\delta_r \tag{2.8}$$



D12

FIGURE 2.31
Directional control by means of [a] rudder.

The directional stability contribution of the vertical tail can be obtained from (8-15) by differentiating with respect to ψ:

$$C_{n\psi} = -a_v\left(1 - \frac{d\sigma}{d\psi}\right)\frac{S_v}{S_w}\frac{l_v}{b}\eta_v \tag{8-16}$$

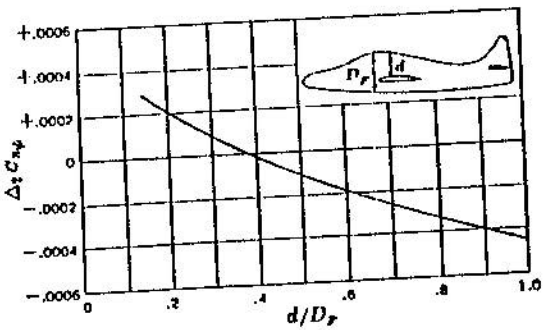


FIGURE 8-7. Interference correction to $C_{n\psi}$.

This expression can be broken down as follows:

$$C_{n\psi} = -a_v\frac{S_v}{S_w}\frac{l_v}{b}\eta_v + a_v\frac{d\sigma}{d\psi}\frac{S_v}{S_w}\frac{l_v}{b}\eta_v \tag{8-17}$$

or

$$C_{n\psi} = (C_{n\psi})_v + \Delta_2 C_{n\psi} \tag{8-18}$$

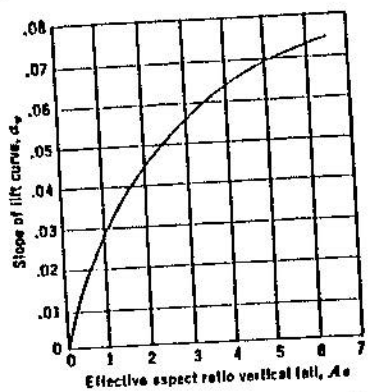


FIGURE 8-8. Slope of lift curve, vertical tail. From NACA TN 775, "Analysis of Wind-tunnel Data on Directional Stability and Control," by H. R. Pass.

The term $\Delta_2 C_{n\psi}$ is that part of the contribution of the vertical tail surface to the airplane's directional stability that arises from the sidewash or interference flow from the wing-fuselage combination. An estimate of this factor can be obtained from the curve given in Figure 8-7.

The slope of the lift curve of the vertical tail can be obtained from Figure 8-8, which is based on the effective aspect ratio of the vertical tail surface.

The effective aspect ratio of the vertical tail surface is greater than the geometrical aspect ratio, defined as b_v^2/S_v, because of the end plate effect of the horizontal tail on the vertical tail if the vertical tail is mounted over the horizontal

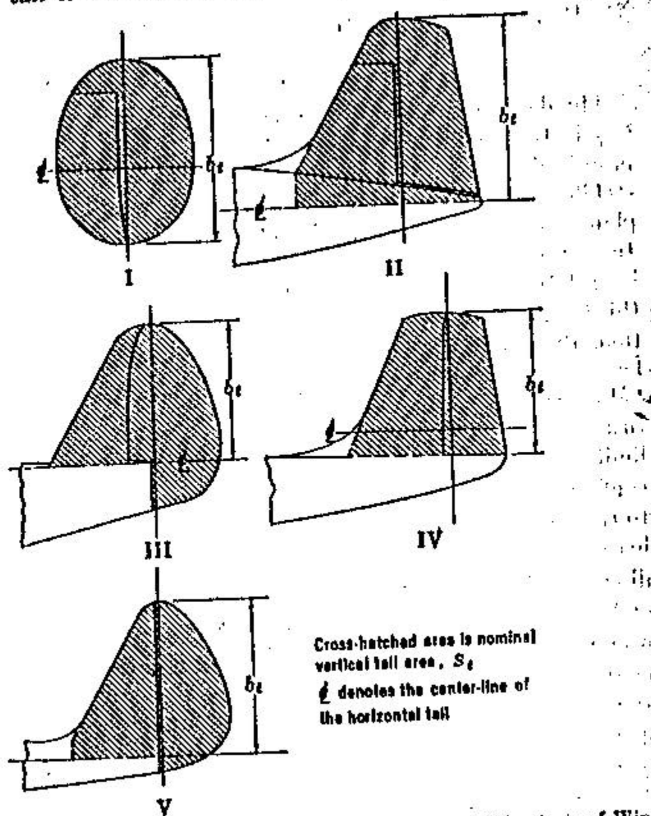


FIGURE 8-9. Vertical tail definitions. From NACA TN 775, "An... of Wind-tunnel Data on Directional Stability and Control," by H. ...

tail in the plane of symmetry. The effective aspect ... of the vertical tail, A_e, can be estimated as follows:

$$A_e = 1.55\frac{b_v^2}{S_v}. \tag{8-19}$$

The span and area of the vertical tail surface are ... ...ily quite difficult to determine because of the many different ty... ...veloped during the past years. The NACA,* however, has laid ... n somewhat arbitrary dimensions for several vertical tail ty... ...hown in Figure 8-9.

* NACA TN 775.

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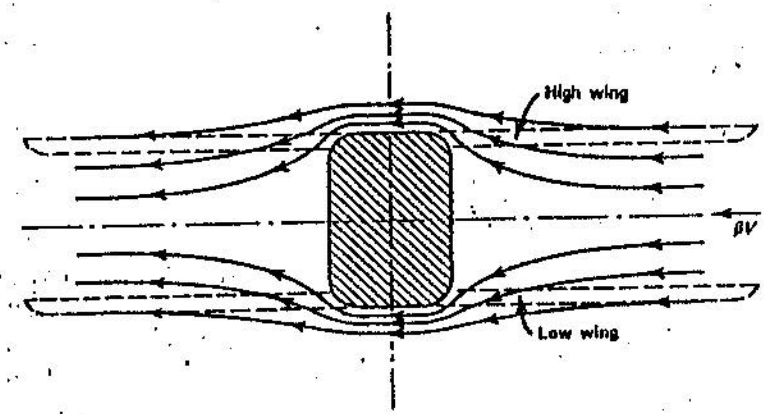


Figure 8.42 Rear view showing the cross-flow around a fuselage.

the dihedral angle for unswept low wings is generally greater than for high wings. Many high-wing airplanes do not have any dihedral angle at all.

As a rule, it is recommended that the following be added to C_{l_β} to account for the fuselage cross-flow (Ref. 8.6).

$$\text{High-wing } \Delta C_{l_\beta} = -0.00016/\text{deg}$$

$$\text{Midwing } \Delta C_{l_\beta} = 0$$

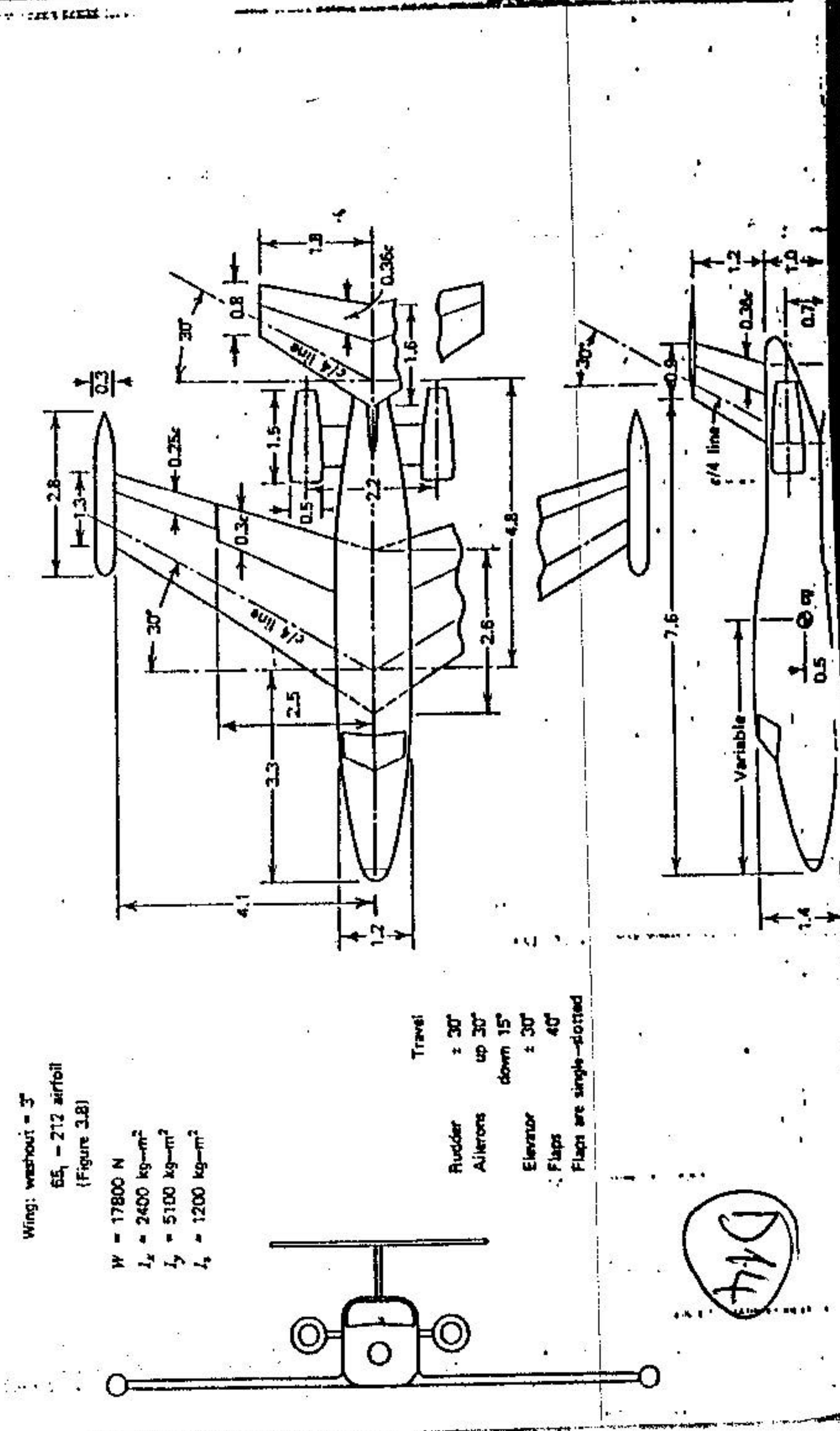
$$\text{Low-wing } \Delta C_{l_\beta} = 0.00016/\text{deg}$$

This information on C_{l_β} and the other stability derivatives is intended only as an introduction to the subject. For more complete information on these quantities, see Reference 5.5 and 8.3.

PROBLEMS

All questions refer to the airplane pictured in Figure 8.43.

8.1 Calculate the mean aerodynamic chord and location of the aerodynamic center of the wing.
8.2 Calculate the location of the stick-fixed neutral point.
8.3 What is the center-of-gravity location for a 10% stick-fixed static margin?



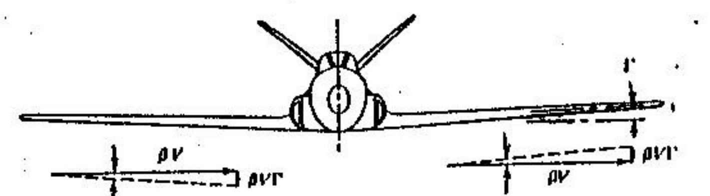


Figure 8.38 Definition of dihedral angle.

the left wing. Added to the free-stream velocity, this results in an increase in the angle of attack over the right wing equal to

$$\Delta \alpha = \beta \Gamma$$

Over the left wing, an opposite change in α occurs. This differential increment in α results in a differential rolling moment, given by

$$dl = 2qca\beta\Gamma y\, dy$$

or

$$C_l = 2a\beta\Gamma \frac{\int_0^{b/2} cy\, dy}{bS}$$

For a linearly tapered wing, this reduces to

$$C_l = \frac{a}{6}\left(\frac{1+2\lambda}{1+\lambda}\right)\beta\Gamma$$

or

$$C_{l_\beta} = \frac{a}{6}\left(\frac{1+2\lambda}{1+\lambda}\right)\Gamma \tag{8.108}$$

The derivation of Equation 8.108 has neglected induced effects that are appreciable for low aspect ratios. Its primary value lies in disclosing the linear relationship between Γ and C_{l_β}.

It is recommended that the contribution to C_{l_β} from Γ be estimated using Figure 8.39. This figure represents the departure of C_{l_β} from the following normal case and is based on graphs presented in Reference 8.11 or 5.5.

$$\lambda = 0.5$$

$$A = 6.0$$

$$\Lambda_{1/2} = 0$$

$$C_{l_\beta} = -0.00021\,\Gamma/\text{deg}(\Gamma \text{ in deg})$$

Thus,

$$C_{l_\beta} = -0.00021 k_\lambda k_A k_\Lambda \Gamma \tag{8.109}$$

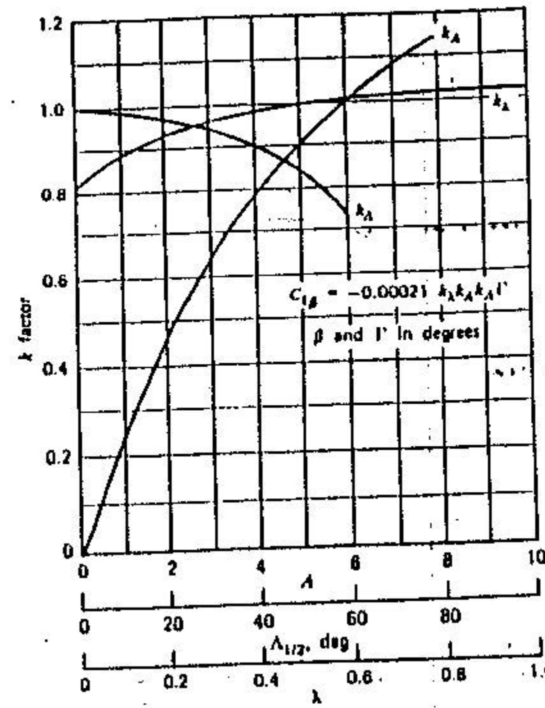


Figure 8.39 Effect of wing dihedral angle on C_{l_β}. Note that factors represent re effects of varying each parameter independently from the normal case. λ $A = 6.0$, $\Lambda_{1/2} = 0$.

where the functions k_λ, k_A, and k_Λ are obtained from Figure 8.39 as fun of λ, A, and $\Lambda_{1/2}$, respectively.

The effect of sweepback on C_{l_β} is determined with the help of F 8.40. A swept wing is shown operating at a positive sideslip angle β. Fro geometry, the velocity component normal to the leading edge of the righ is given by

$$V \cos(\Lambda - \beta)$$

The corresponding velocity on the left wing is

$$V \cos(\Lambda + \beta)$$

If C_{l_n} is the section lift coefficient based on the normal velocity "normal chord," then the differential lift on the right and left wings will

$$dL_R = q \cos^2(\Lambda - \beta) c \cos \Lambda\, C_{l_n}\, ds$$

$$dL_L = q \cos^2(\Lambda + \beta) c \cos \Lambda\, C_{l_n}\, ds$$

D15

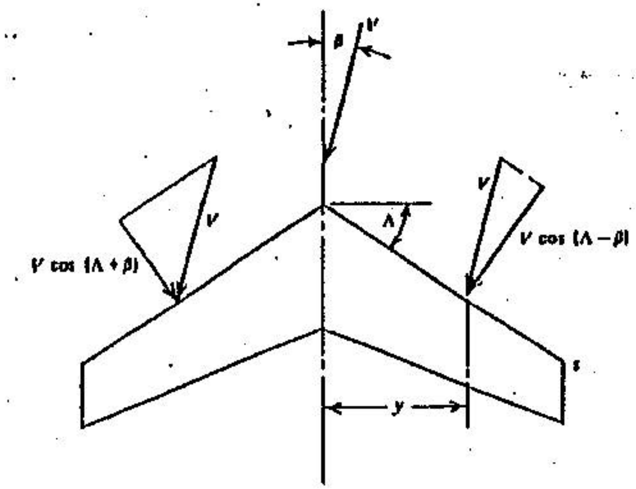


Figure 8.40 Effect of sweepback on dihedral effect.

The differential rolling moment is therefore

$$dl = qcC_{l_n}y[\cos^2(\Lambda+\beta) - \cos^2(\Lambda-\beta)]\cos\Lambda\, ds$$

But

$$y = s\cos\Lambda$$
$$dy = ds\cos\Lambda$$

Thus,

$$l = qC_{l_n}[\cos^2(\Lambda+\beta) - \cos^2(\Lambda-\beta)]\int_0^{b/2} cy\,dy$$

The total lift (for $\beta = 0$) is given by

$$L = 2q\cos^2\Lambda\, C_{l_n}\int_0^{b/2} c\,dy$$
$$= qSC_{l_n}\cos^2\Lambda$$

Thus the wing C_L and the normal section C_{l_n} are related by

$$C_L = C_{l_n}\cos^2\Lambda$$

and the rolling moment coefficient becomes

$$C_l = \frac{C_L}{\cos^2\Lambda}[\cos^2(\Lambda+\beta) - \cos^2(\Lambda-\beta)]\frac{\int_0^{b/2} cy\,dy}{Sb}$$

D16

If this equation is differentiated with respect to β and evaluated at $\beta = 0$,
following results.

$$C_{l_\beta} = -4C_L\tan\Lambda\left[\frac{\int_0^{b/2} cy\,dy}{Sb}\right] \quad (8.$$

For a linearly tapered wing, Equation 8.110 reduces to

$$C_{l_\beta} = -\frac{1+2\lambda}{3(1+\lambda)}C_L\tan\Lambda \quad (8.$$

Again, this result is only qualitatively correct. Generally,

$$C_{l_\beta} = -f(A,\lambda)C_L\tan\Lambda \quad (8.$$

Figure 8.41 (based on Ref. 5.5) presents C_{l_β}/C_L as a function of Λ f
range of aspect ratios. The variation with tan Λ is seen to hold only for
higher aspect ratios. This figure can be used with Equation 8.111 to estim
C_{l_β} for other taper ratios.

Observe that wing sweep can contribute significantly to dihedral effec
order to avoid an excessive dihedral effect on aircraft with highly sw
wings, it is frequently necessary to employ a negative dihedral angle on
wing, particularly if the wing is mounted high on the fuselage.

The effect that the wing placement on the fuselage has on C_{l_β} is seen
reference to Figure 8.42. In a plane across the top of the fuselage,
cross-flow around the fuselage is seen to go up on the right side and down
the left. Thus, for a high wing, this flow increases the angle of attack of
right wing while decreasing α on the left wing. This results in a nega
rolling moment comparable to a positive dihedral effect. For a low wing
effect is just the opposite. This is the reason, as you may have observed,

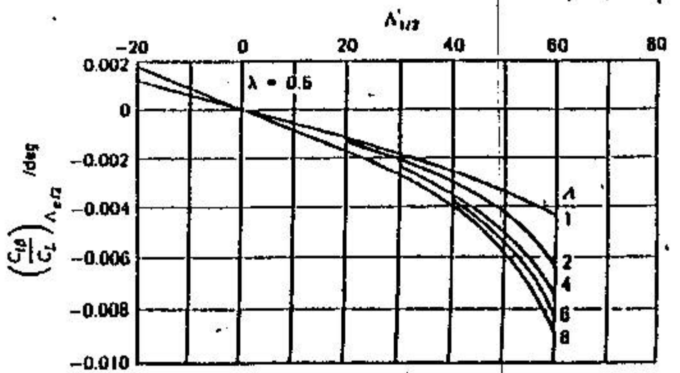


Figure 8.41 Effect of wing sweep on C_{l_β}.