3.11 Example of An Airplane Load Calculation

Airplane data

Geometry (See Fig. 3.11.1).

Wing:

- Area, $S = 1200$ sq ft
- Span, $b = 98$ ft
- Aspect ratio, AR (or A) $= 8.0$
- Taper ratio, $\lambda = \dfrac{C_T}{C_R} = 0.4$
- Root chord, $C_R = 17.5$ ft
- Tip chord, $C_T = 7.0$ ft

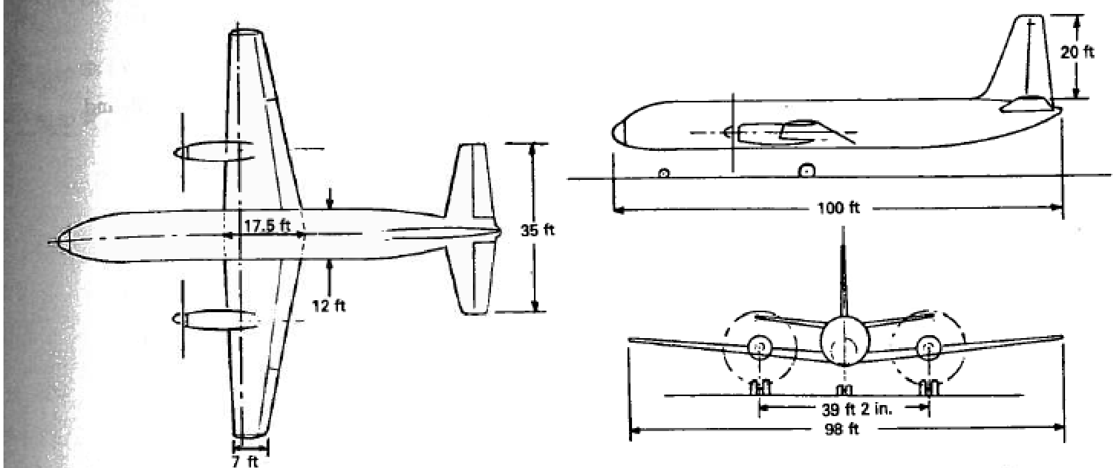

Fig. 3.11.1 Three view diagram of the airplane.

- Mean aerodynamic chord, $\bar{c} = 13.0$ ft

Vertical Tail:

- Area, $S_{vt} = 220$ sq ft
- Tail length, $\ell_{vt} = 45$ ft
- Rudder area, $S_y = 60$ sq ft
- Rudder mean chord, $\bar{c}_y = 3.5$ ft

Horizontal Tail:

- Area, $S_{ht} = 200$ sq ft
- Tail length, $\ell_{ht} = 48.0$ ft

Weight Data

- Maximum take-off gross weight = 108000 lb
- Landing design gross weight = 88000 lb
- Maximum zero fuel weight = 84 000 lb

Aerodynamic Data

- $C_{L\,max}$ (complete airplane) $= 1.3$
- $C_{L_{\alpha A}} = 1.10\ C_{L_{\alpha W}}$
- $C_{h_{\delta_y}} = 0.01$ per degree
- Rudder hinge moment output = 7520 ft-lb

Landing Gear Data

- Strut stroke, $X_S = 18$ in
- Tire defection, $X_t = 4.5$ in
- Tire spring constant = 160000 lb/ft
- Tire efficiency, $\eta_t = 0.8$

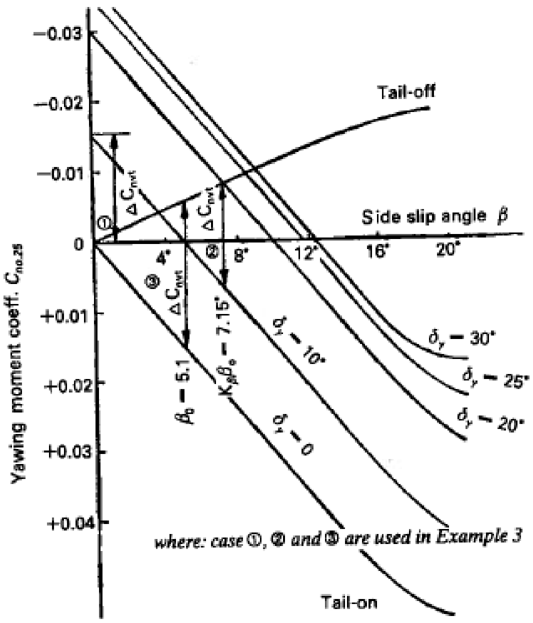

Fig. 3.11.2 *Vertical tail yawing moment coeff. vs. side slip curves.*

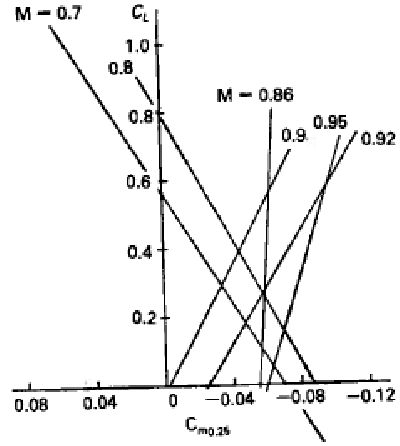

Fig. 3.11.3 *Tail-off pitch stability.*

Example 5

Estimate the balancing horizontal tail load for a steady maneuver at $n = 2.5$, gross weight = 88000 lb, c.g. @ 12% MAC, $V = 400$ KEAS, $M = 0.95$. Make use of the pitch curve of airplane-less-tail. (Assume fuselage lift and moment are equal to zero for this case, see Fig. 3.11.8).

Aerodynamic pressure

$$q = \frac{V_e^2}{296} = \frac{(400)^2}{296} = 540 \text{ lb/ft}^2$$

The distance between the 25% MAC and c.g. at 12% MAC

Wing, $\overline{C} = 13$ ft (from airplane data)

Therefore, $\Delta X = (25\% - 12\%)\,\overline{C}$
$= (13\%) \times 13 = 1.69$ ft

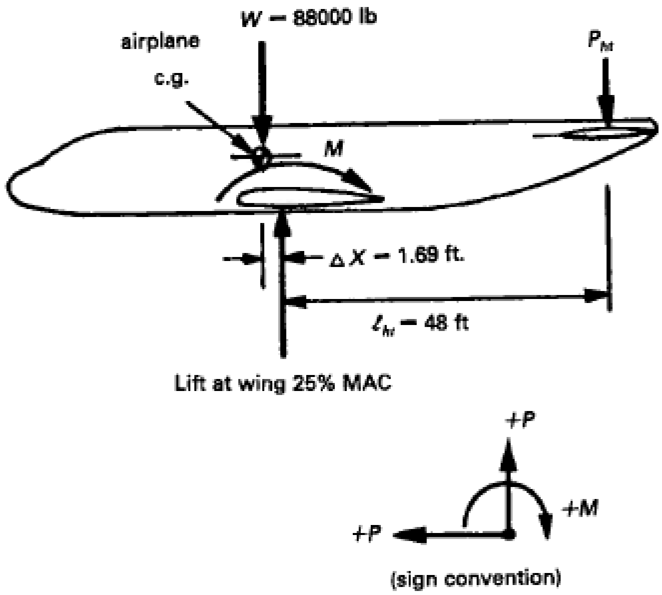

Fig. 3.11.8 Airplane pitch loads.

The airplane lift coefficient calculation is

$$C_{LA} = \frac{nW}{qS} = \frac{(2.5)(88000)}{(540)(1200)} = 0.34$$

with $C_{L_A} = 0.34$ and from curve of $M = 0.95$ (Fig. 3.11.3) obtain

$$C_{m0.25} = -0.082$$

The wing pitch moment at 25% MAC is

$$\begin{aligned} M &= (C_{m0.25})(q)(S)(\overline{C}) \\ &= (-0.082)(540)(1200)(13) \\ &= -690768 \text{ ft-lb} \end{aligned}$$

Take moment center at wing 25% MAC, then

$$-(nW)(\Delta X) + M + P_{ht}(\ell_{ht}) = 0$$
$$-(2.5 \times 88000)(1.69) - 690768 + P_{ht}(48) = 0$$
$$P_{ht} = 22137 \text{ lb (balancing horizontal tail load)}$$

W = 88000 lb (Landing design gross weight), 108000 lb (MTOW), 84000 lb (MZFW)

V, keas	X cg, %	n	q	nW, lb	qS	CLA	Cm0.25	Mw	dX, ft	Pht, lb	Lw, lb	alfa, deg	d alfa
400	12	2.5	541	220000	648649	0.339	-0.082	-691459	1.69	22151	242151	4.50	
400	40	2.5	541	220000	648649	0.339	-0.082	-691459	-1.95	5468	225468	4.19	**0.31**
400	12	2.5	541	270000	648649	0.416	-0.082	-691459	1.69	23912	293912	5.46	
400	40	2.5	541	270000	648649	0.416	-0.082	-691459	-1.95	3437	273437	5.08	**0.38**
245	12	2.5	203	220000	243345	0.904	0.0518	163868	1.69	4332	224332	11.11	
245	40	2.5	203	220000	243345	0.904	0.0518	163868	-1.95	-12351	207649	10.28	**0.83**
145	12	1	71	108000	85236	1.267	0.0894	99062	1.69	1739	109739	15.51	
145	40	1	71	108000	85236	1.267	0.0894	99062	-1.95	-6451	101549	14.35	**1.16**
200	12	1	135	84000	162162	0.518	-0.0118	-24876	1.69	3476	87476	6.50	
200	40	1	135	84000	162162	0.518	-0.0118	-24876	-1.95	-2894	81106	6.03	**0.47**