MAE 5070 **Exercise Set III** March 9, 2011

D. A. Caughey

1. Use the dimensions provided in Exercise Set II for the Boeing 747 aircraft to estimate the values of $\mathbf{C}_{mq}$ and $\mathbf{C}_{m\dot{\alpha}}$ for *both flight conditions* of Exercise Set I. Refer to Solution Set I for any additional parameters needed for the $\mathbf{M}_\infty = 0.80$ flight condition. Compare your values with those given in the table below; note that the flight conditions of Exercise Set I correspond to Conditions 2 and 9 in the table below. (You should realize that you've already computed the first of these stability coefficients for Condition 1 in Exercise Set II.)

2. Show that for a straight, untapered wing (i.e., one having a rectangular planform) having a *constant* spanwise load distribution (i.e., constant section lift coefficient), simple strip theory gives the wing contribution to the rolling moment due to yaw rate as

$$(\mathbf{C}_{lr})_{\text{wing}} = \frac{\mathbf{C}_L}{3}$$

3. Show that for a straight, untapered wing (i.e., one having a rectangular planform) having an *elliptical* spanwise load distribution, simple strip theory gives the wing contribution to the rolling moment due to yaw rate as

$$(\mathbf{C}_{lr})_{\text{wing}} = \frac{\mathbf{C}_L}{4}$$

   Explain, in simple terms, why this value is smaller than that computed in Exercise 2.

4. Show that for a straight, untapered wing (i.e., one having a rectangular planform) having an *elliptical* spanwise load distribution, simple strip theory gives the *induced drag contribution* to the yawing moment due to roll rate as

$$(\mathbf{C}_{np})_{\text{wing}} = -\frac{\mathbf{C}_L}{8}$$

Note: For problems dealing with elliptic spanwise loadings the following integrals (which can be evaluated using trigonometric substitution) are useful:

$$\int_0^1 \sqrt{1-\eta^2}\,\mathrm{d}\eta = \frac{\pi}{4}, \qquad \int_0^1 \eta\sqrt{1-\eta^2}\,\mathrm{d}\eta = \frac{1}{3}, \qquad \int_0^1 \eta^2\sqrt{1-\eta^2}\,\mathrm{d}\eta = \frac{\pi}{16}$$

Condition	2	5	7	9	10
h (ft)	SL	20,000	20,000	40,000	40,000
$\mathbf{M}_\infty$	0.249	0.500	0.800	0.800	0.900
$\mathbf{C}_L$	1.11	0.680	0.266	0.660	0.521
$\mathbf{C}_D$	0.102	0.0393	0.0174	0.0415	0.0415
$\mathbf{C}_{L_\alpha}$	5.70	4.67	4.24	4.92	5.57
$\mathbf{C}_{D\alpha}$	0.66	0.366	0.084	0.425	0.527
$\mathbf{C}_{m_\alpha}$	-1.26	-1.146	-.629	-1.033	-1.613
$\mathbf{C}_{L_{\dot{\alpha}}}$	6.7	6.53	5.99	5.91	5.53
$\mathbf{C}_{m_{\dot{\alpha}}}$	-3.2	-3.35	-5.40	-6.41	-8.82
$\mathbf{C}_{L_q}$	5.40	5.13	5.01	6.00	6.94
$\mathbf{C}_{m_q}$	-20.8	-20.7	-20.5	-24.0	-25.1
$\mathbf{C}_{L_\mathbf{M}}$	0.0	-.0875	0.105	0.205	-.278
$\mathbf{C}_{D_\mathbf{M}}$	0.0	0.0	0.008	0.0275	0.242
$\mathbf{C}_{m_\mathbf{M}}$	0.0	0.121	-.116	0.166	-.114
$\mathbf{C}_{L_{\delta_e}}$	0.338	0.356	0.270	0.367	0.300
$\mathbf{C}_{m_{\delta_e}}$	-1.34	-1.43	-1.06	-1.45	-1.20

Table 1: Longitudinal aerodynamic stability derivatives for the Boeing 747 at selected flight conditions; Data from Heffley & Jewell, Aircraft Handling Qualities Data, NASA CR-2144, December 1972.