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$R_A = \frac{b_w}{c_w}$ $\quad c_w = \frac{S}{b}$ $\quad \frac{b^2}{S} = R_A$

MAE 5510 : Exercise Set 2

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For the following problems, we will consider a version of the British Spitfire with the following geometric and aerodynamic characteristics:

$\bar{c}_w = 6.625$ $\quad R_A = 5.559$ $\quad R_{Ah} = 3.652$

$$S_w = 244\text{ft}^2, \quad b_w = 36.83\text{ft}, \quad \boxed{C_{L_w,\alpha} = 4.62,} \quad \alpha_{L0_w} = -2.2°, \quad C_{m_w} = -0.053,$$

$$S_h = 31\text{ft}^2, \quad b_h = 10.64\text{ft}, \quad C_{L_h,\alpha} = 4.06, \quad \varepsilon_e = 0.60, \quad C_{m_h,\delta_e} = -0.55,$$

$$W = 8,375\text{lbf}, \quad l_h - l_w = 18.16\text{ft}$$

Assume that the center of gravity lies at the quarter-chord of the main wing, the horizontal stabilizer has a symmetric airfoil, and that the main wing and horizontal stabilizer have zero twist.

2.1 The main wing of the British Spitfire has an elliptic planform. From lifting-line theory, the lift coefficient produced on an elliptic wing with zero twist can be computed from

$$C_L = C_{L,\alpha}(\alpha - \alpha_{L0})_{\text{root}} \tag{1}$$

where

$$C_{L,\alpha} = \frac{\tilde{C}_{L,\alpha}}{[1 + \tilde{C}_{L,\alpha}/(\pi R_A)]} \tag{2}$$

2π $\quad \left[1 + \frac{2\pi}{\pi(5.559)}\right]$

Assuming the main wing has a thin airfoil, compute the lift on the main wing at 5 deg angle of attack and a velocity of 200 mph at sea level.

$\tilde{C}_{L,\alpha} = 2\pi$ $\quad C_{L,\alpha} = 4.62$ $\quad V_\infty = 293.33$ ft/s

$\rho_{SL} = 0.0023769$

$C_L = 4.62(5° + 2.2°)$

$C_L = 4.62\left(7.2 \cdot \frac{\pi}{180}\right) = \underline{0.5805}$

$L = 0.5805 \cdot \left(\frac{1}{2}\rho V_\infty^2 S_w\right) = 0.5805\,(0.5 \cdot 0.0023769 \cdot 293.33^2 \cdot 244)$

$L_w = \underline{14,483.913}$ lbf

2.2 The horizontal stabilizer on the British Spitfire has an elliptic planform. Assuming is uses a thin airfoil, compute the lift on the horizontal stabilizer at 5 deg angle of attack and a velocity of 200 mph at sea level without any influence from the main wing. no down wash

$\alpha = 5$ $\quad \varepsilon_d = 0$ $\quad \alpha_{L0_h} = 0°$

$C_{L_h} = C_{L_{h,\alpha}}(\alpha - \cancel{\alpha_{0h}} - \cancel{\alpha_{L0_h}} - \cancel{\varepsilon_d}) = C_{L_h} = 4.06\left(5 \cdot \frac{\pi}{180}\right) = 0.3543$

$L_h = 0.354302 \cdot (0.5 \cdot 0.0023769 \cdot 293.33^2 \cdot 31)$ $\quad L_h = \cancel{8840.093}$

$\underline{1123.1214 \text{ lbf}}$

Acevo

2.3 Using MachUp 4, compute the lift produced on the main wing at 5 deg angle of attack and a velocity of 200 mph at sea level. Assume that the airfoil used on the main wing is thin and has a zero-lift angle of attack of −2.2°. The root chord of an elliptic wing can be computed from

$$c_{\text{root}} = \frac{4b}{\pi R_A}$$

$\frac{4 \cdot 36.83}{\pi\, 5.559} = 8.43558$

Compare this result to that in problem 2.1.

$C_L = 0.5809$

2.1

$L = 14,483$

$L = 14,483$ //

Same result.

2.4 Using MachUp 4, compute the lift produced on the horizontal stabilizer at 5 deg angle of attack and a velocity of 200 mph at sea level. Compare this result to that in problem 2.3.

$C_{root} = \frac{4 \cdot 10.64}{\pi (3.652)} = 3.7096$

$C_{L_h} = 0.3545$

−0.0369

2.3 $L_h = 1123.12$ lbf

$L_h = 1123.755$ lbf

this result is slightly higher

2.5 Using MachUp 4, compute the lift produced on the main wing and horizontal stabilizer at 5 deg angle of attack and a velocity of 200 mph at sea level when the horizontal stabilizer is placed a distance of $l_h - l_w = 18.16$ft aft of the main wing. Compare this result to that in problems 2.1 - 2.4. Discuss your results.

$C_{L_h} =$ ~~0.7987~~ $0.01054 \cdot \frac{S_w}{S_h} = 0.08296$

$C_{L_w} =$ ~~0.1~~ $\rightarrow 0.19878$

$L_w = 4958.696$

$L_h = 33.4046$

Lift is drastically wrong from previous results, much lower.

Believe that is an error w/ the code.

Should be more than main wing but less than combination due to downwash effects on H.S.

2.6 Using the simplified analysis for estimating the downwash, estimate the downwash on the horizontal as a function of the lift coefficient on the main wing.

$$\varepsilon_d(\bar{x},\bar{y},0) = \frac{k_v k_p k_s}{k_b}\frac{C_{Lw}}{R_{Aw}} = \frac{k_v}{k_b}\frac{C_{Lw}}{R_{Aw}}$$

$k_v = 1.0$

$k_b = \frac{\pi}{4}$

$k_p = 0.46$

$$\frac{1.0}{\frac{\pi}{4}} = \frac{4}{\pi}\cdot\frac{C_{Lw}}{R_{Aw}} = \frac{4}{3.1415\cdot 5.559} =$$

$\cancel{C_{Lw}}$

$$\frac{10.16}{\frac{\pi}{4}\ \cancel{391}\ 18.415} =$$

1.2556 $\bar{x}$ location

$\cancel{\varepsilon_d = 0.229\, C_{Lw}}$

$$\varepsilon_d = 0.10534\ C_{Lw}$$

$\frac{k_v k_p k_s}{k_b R_A} C_{Lw,\alpha}$

$$\varepsilon_d = \left[\varepsilon_{d_0} + \varepsilon_{d,\alpha}\,\alpha\right]$$

$$\varepsilon_{d,\alpha} = 0.10534\, C_{Lw,\alpha}$$

α_{ow} α_{oh}

2.7 Using the results of 2.6, find the mounting angle of the main wing and horizontal stabilizer required for the aircraft to be trim in steady-level flight at sea level at a velocity of 200 mph with zero elevator deflection and zero angle of attack. Compare your results to those from problem 1.13. Discuss your results.

$\delta_e = 0$ $\alpha = 0$ $\gamma = \alpha$ -0.053 200 mph $= 293.33\ ft/s$

$l_h = 18.16$

$l_w = 0$

$$C_{m_0} = -\left[C_{m_w} - \cancel{\frac{l_w}{c_w} C_{L,\alpha}(\alpha_{ow} - \alpha_{Low})}\right) - \frac{S_h}{S_w}\frac{l_h}{c_w}\eta_h C_{Lh,\alpha}(\alpha_{oh} - 0.229\, C_{Lw})$$

ε_{d_0} $\varepsilon_{d_0} = 4.62(\alpha_{ow} - 0.035)$

$$\overset{.015}{\cancel{0.11}} - 0.053 = 1.4139\left(\alpha_{oh} - \overset{0.10534}{\cancel{0.229}}\, C_{Lw}\right)$$

$C_{Lw} = 0.1774$

$\cancel{\alpha_{oh} = -0.00875\ rad}$

$\cancel{\alpha_{oh} = -0.502\ degrees}$

lowers the required angle of mounting of aircraft.

Lift.

$$0.158259 = 4.62\,\alpha_{ow} + 0.51582(\alpha_{oh} - \varepsilon_{d_0})$$

moment.

$$\frac{-0.053 + \varepsilon_{d_0}}{1.4139} = \alpha_{oh}$$

Lift: $0.067819 = 2.24\,\alpha_{ow} + 0.3182\,\alpha_{oh}$

moment: $-0.22856 = 4.62\,\alpha_{ow} + 1.4139\,\alpha_{oh}$

$\alpha_{ow} = 0.277$ $\alpha_{oh} = -1.067$

$C_{L w,\alpha} = 4.62$

2.8 Compute the aircraft static margin. Compare your results to those from problem 1.14. Discuss your results.

$$C_{m,\alpha} = \frac{-l_w}{\bar{c}_w} C_{Lw,\alpha} - \frac{S_h l_h}{S_w \bar{c}_w} \eta_h C_{Lh,\alpha}(1 - \varepsilon_{d,\alpha})$$

$$C_{L,\alpha} = C_{Lw,\alpha} + \frac{S_h}{S_w} \eta_h C_{Lh,\alpha}(1 - \varepsilon_{d,\alpha})$$

$\varepsilon_d = 0.10534\, C_{Lw,\alpha}$

$\varepsilon_{d,\alpha} = 0.10534(4.62) = \underline{0.48667}$

I need to solve.

$$C_{L,\alpha} = 4.62 + \frac{31}{244} 4.06 (1 - 0.48667) = 4.8847$$

$$C_{m,\alpha} = \cancel{4.62} - \frac{31 \cdot 18.16}{244 \cdot 6.625} \cdot 4.06 (1 - 0.48667) = -.72580$$

$$-\frac{C_{m,\alpha}}{C_{L,\alpha}} = \frac{-0.7258}{4.8847} = 0.1485 = \underline{\underline{14.86\%}}$$

Reduced the static margin greatly.

2.9 If the main wing and horizontal stabilizer both have zero mounting angles, compute the angle of attack and elevator deflection required to trim the aircraft in a steady climb at an altitude of 5,000 ft and a climb angle of 20 deg at a speed of 200 mph. Include the effects of downwash and compare your results to those from problem 1.15. Discuss your results.

$\alpha_{0w} = \alpha_{0h} = 0$

$\rho = 0.002048$

has effects of ε_d

Need ε_{d0} ??

$$\begin{bmatrix} C_{L,\alpha} & C_{L,\delta_e} \\ C_{m,\alpha} & C_{m,\delta_e} \end{bmatrix} \begin{bmatrix} \alpha \\ \delta_e \end{bmatrix} = \begin{bmatrix} C_L - C_{L0} \\ -C_{m0} \end{bmatrix}$$

$C_{L,\alpha} = 4.8847$

$C_{m,\alpha} = -0.72580$

$C_{L,\delta_e} = 0.309492$

$C_{m,\delta_e} = -0.879084$

$$\begin{bmatrix} 4.8847 & 0.309492 \\ -0.72580 & -0.879084 \end{bmatrix} = \begin{bmatrix} .0.98 \\ 0.198 \end{bmatrix}$$

$0.366066 - 0.177395 - 0.09053$

$$-C_{m0} = -\left[C_{mw} + \frac{S_h \bar{c}_h}{S_w \bar{c}_w} \eta_h C_{mh0} - \frac{l_w}{\bar{c}_w} C_{Lw,\alpha}(\alpha_{0w} - \alpha_{L0w}) - \frac{S_h l_h}{S_w \bar{c}_w} \eta_h C_{Lh,\alpha}(\alpha_{0h} - \alpha_{L0h} - \varepsilon_{d0})\right)$$

$\varepsilon_{d0} = 4.62(\cancel{\alpha_{0w}} - 0.038)$

$\varepsilon_{d0} = -0.17556$

$$-C_{m0} = -\left[-0.053 - \frac{31 \cdot 18.16}{244/36.83} 4.06(-0.177556)\right]$$

$$-\left[-0.053 \cdot +0.2510\right] = 0.198$$

$\alpha = -0.1966 \quad \delta_e = -.0628$

2.10 Using MachUp 4, compute the global inviscid lift, drag, and pitching-moment coefficients about the origin for the aircraft at angles of attack of 4, 5, and 6 deg.

Angle of Attack [deg]	C_L	C_{D_i}	C_m
4.0	0.3596	0.0138	-0.0574
5.0	0.4450	0.0181	-0.0464
6.0	0.5308	0.0234	-0.0358

$C_{L,\alpha} \approx 0.0856$ $\quad C_{m,\alpha} = 0.011$

$C_{D_i,\alpha} \approx 0.0048$

2.11 Using the results of problem 2.10, find the location of the aerodynamic center (x_{ac}, y_{ac}) using the general relations for the aerodynamic center. Compare your results to that which would be obtained from the simplified analysis in problem 2.8. Discuss your findings.

find X_{ac}, Y_{ac}

$$C_{L,\alpha\alpha} \cong \frac{C_L(\alpha+\Delta\alpha) - 2C_L(\alpha) + C_L(\alpha-\Delta\alpha)}{(\Delta\alpha)^2}$$

$$C_{L,\alpha} \cong \frac{C_L(\alpha+\Delta\alpha) - C_L(\alpha-\Delta\alpha)}{2\Delta\alpha}$$

$C_A \approx C_D$

$C_L \approx C_N$

$C_m = C_{m_o}$

$C_A = C_D$

$$C_{A,\alpha} = C_{D,\alpha} = \frac{0.0234 - 0.0138}{2(\frac{\pi}{180} \cdot 1)} = 0.275 \text{ "rad"}$$

$$C_{D,\alpha\alpha} = \frac{C_D(\alpha+\Delta\alpha) - 2C_D(\alpha) + C_D(\alpha-\Delta\alpha)}{\Delta\alpha^2} = 3.282$$

$C_{D,\alpha} = 0.27502$

$C_{m,\alpha} = 0.61879$

$C_{L,\alpha} = 4.9045$

$C_{m,\alpha\alpha} = -1.3131$

$C_{L,\alpha\alpha} = 1.3131$

$$\frac{X_{ac}}{c_{ref}} = \frac{C_{D,\alpha} C_{m,\alpha\alpha} - C_{m,\alpha} C_{D,\alpha\alpha}}{C_{L,\alpha} C_{D,\alpha\alpha} - C_{D,\alpha} C_{L,\alpha\alpha}} \approx \frac{-2.392}{15.735} = \boxed{-0.15202}$$

$$\frac{Y_{ac}}{c_{ref}} = \frac{C_{L,\alpha} C_{m,\alpha\alpha} - C_{m,\alpha} C_{L,\alpha\alpha}}{C_{L,\alpha} C_{D,\alpha\alpha} - C_{D,\alpha} C_{L,\alpha\alpha}} \approx \frac{-7.2526}{15.735} = \boxed{-0.4609}$$

not the same as before

Because I got a + $C_{m,\alpha}$ so it would be not stable at all.