Luke Verlangieri | AA311 | 11/4/2025

5.23 Given: $S = 21.5\ m^2$ $AR = 5$

NACA 65-210 Airfoil

$e = 0.9$ $C_{d,profile} = 0.004 = C_{D,0}$

$\alpha = \cancel{8}\ 6^\circ$

Find

C_L, C_D

Properties

Profile drag → Zero-lift drag coef

$$C_D = C_{D,0} + \frac{C_L^2}{\pi e AR} \qquad e_1 \approx e$$

$$a = \frac{dC_L}{d\alpha} = \frac{a_0}{1 + \frac{57.3 a_0}{\pi e_1 AR}} \qquad C_L = a \Delta\alpha$$

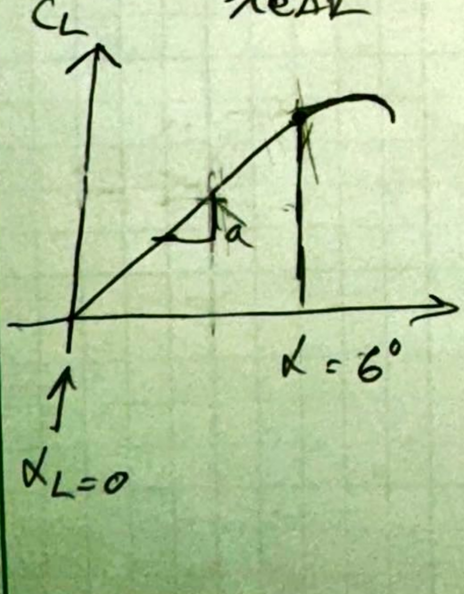
$$a_0 = \frac{dC_L}{d\alpha}, \text{ infinite} \qquad \alpha_{eo} = \boxed{\alpha - \alpha_{L=0}}$$

Analysis

From the NACA 65-210 Airfoil data

$\alpha_{L=0} \approx -1.75^\circ$ by taking the slope → $a_0 = \frac{1 - 0.15}{8^\circ} = 0.10625$

$$a = \frac{a_0}{1 + \frac{57.3 a_0}{\pi e AR}} = \frac{(0.10625)}{1 + \left(\frac{57.3 \cdot 0.10625}{\pi \cdot 0.9 \cdot 5}\right)} = 0.0743\ \frac{C_L}{\alpha}$$



$$C_L = a \cdot (6^\circ - (-1.75^\circ)) = \boxed{0.5756}$$

$$C_D = C_{D,0} + \frac{C_L^2}{\pi e AR} = 0.004 + \frac{(0.5756)^2}{\pi (0.9)(5)}$$

$$\boxed{C_D = 0.0274}$$

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5.40 Given: $L = 100$ lbf $D = 145$ lbf

$\alpha = 5^\circ$

Find

N, A

Properties

$$L = N\cos\alpha - A\sin\alpha$$

$$D = N\sin(\alpha) + A\cos(\alpha)$$

Analysis

$$\frac{(D - A\cos\alpha)}{\sin\alpha} = \frac{(L + A\sin\alpha)}{\cos\alpha} = N$$

$$D - A\cos\alpha = L\tan\alpha + A\sin\alpha\tan\alpha$$

$$D - L\tan\alpha = A(\sin\alpha\tan\alpha + \cos\alpha)$$

$$\frac{D - L\tan\alpha}{(\sin\alpha\tan\alpha + \cos\alpha)} = A = \frac{(145 - 100\tan(5^\circ))}{(\sin(5^\circ)\tan(5^\circ) + \cos(5^\circ))}$$

$$\boxed{A = 135.73 \text{ lbf}}$$

$$N = \frac{L + A\sin\alpha}{\cos\alpha} \rightarrow N = \frac{100 + 135.73\sin(5^\circ)}{\cos(5^\circ)} = \boxed{112.26 \text{ lbf}}$$

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5.28 | given: Flat surface $S = 1\ ft^2$
$V_\infty = 21.8\ ft/s$ $L = 1$ ounce
$\alpha = 3^\circ$

Find

C_L

Compare with measured

Assumptions

machine is at sea level

Properties

$q_\infty = \frac{1}{2}\rho_\infty V_\infty^2$

$C_L = \frac{L}{q_\infty \cdot S}$

$c_l = 2\pi\alpha$

$c_\ell = \frac{L}{q_\infty \cdot c_l}$

Analysis

$\rho_\infty = 0.002378\ \frac{slugs}{ft^3}$

$q_\infty = \frac{1}{2}(0.002378\ \frac{slug}{ft^3})(21.8\ \frac{ft}{s})^2 = 0.56506$ psf

1 ounce $= 1/16$ lbf $\therefore$

$$C_L = \frac{(1/16)}{(\quad)(1)} = \boxed{.11061}$$

$$c_\ell = \frac{L}{q \cdot c_l} = \frac{(1/16)}{(.56506)\cdot\left(\frac{2\pi \cdot 3\pi}{180}\right)} \boxed{= 0.33621}$$

The theory approximated using c_l is for an infinite length airfoil, the coefficient of drag for a finite airfoil differs from the theory significantly because of the airfoil wing tips and other factors of flight. In general $c_\ell \neq C_L$