AOE 3134 Stability and Control
Problem Sheet 2

Due 12 February, 2002

Read Chapter 2: 2.1, 2.2, 2,3, 2.4
Appendix B: B1, B2, B5; C: C.1, C.2, C.3, C.4

5. Consider a straight tapered wing with an areas of 2433 ft^2 and a wing span of 131 ft. The root chord is 28.6 ft and the taper ratio is 0.3. The quarter chord sweep angle is 35 deg. Estimate and plot the lift-curve slope for this wing vs Mach number, for M_a = 0, 0.2, 0.4, 0.6, 0.8. Assume that the 2-D lift curve slope of the airfoil section is 2π/rad. Use method developed in Appendix B.1. Use the equation and not the figure!

6. Although it may be physically impossible, we are going to "morph" the above wing into different aspect ratios, or sweep angles keeping all the other parameters the same. Make the following plots (use MATLAB)

a) M_a = 0.4, λ=0.3, AR = 2, 4, 6, 8, 10, 12 (C_{L_α} vs. AR)

b) M_a= 0.4, AR = AR of problem 5 (C_{L_α} vs λ)

λ = 0.0, 0.2, 0.4, 0.6, 0.8, 1.0

From the results of the plots, discuss the effects of aspect ratio, taper ratio, and Mach number on the lift-curve slope of the straight-tapered wing.

7. Problem 2.1, Etkin and Reid, Page 52. Do not do part (a). For part (c) show the position of the mean aerodynamic chord and mean aerodynamic center on the planform view of the wing. In particular locate the leading edge of the mean aerodynamic chord with respect to the leading edge of the root chord.

8. Given a glider consisting of a rectangular wing and horizontal tail with a slender boom carrying the vertical and horizontal tail surfaces. If the force and moment contributions of the boom are negligible, find the location fo the aircraft neutral point. The geometric characteristics of the glider are:

wing area = 360 ft^2	tail area = 72 ft^2	tail ac location = 3.24 MAC
wing aspect ration = 10	tail aspect ratio = 4	tail setting angle = -2.0 deg
MAC = 6 ft	tail efficiency = 0.9	tail zero lift angle = 0.0 deg
wing zero-lift angle = -3.0 deg	wing zero lift pitch moment coef = -0.02	
wing setting angle = 0.0 deg	$\frac{\partial \epsilon}{\partial \alpha}$ = 0.264	Wing aero center = 0.24MAC

Assume a thin airfoil (2-D lift curve slope = 2π)