MAE 5510 Flight Dynamics : Exam 1 - Longitudinal Trim and Stability   **Name** _____

Open book. Open notes. Closed neighbor, videos, internet, AI, or any other resource. All work done neatly and logically. Calculators, Excel, or personal code allowed.

I did not communicate about this exam to anyone who had already started the exam, and I will not talk to anyone about this exam until after it is due. Sign here if this is true: _____

1. Short Answer (50 Points)

1.1 For an aircraft to be trim, the forces about the center of gravity must sum to be (greater than | less than | equal to) zero.

1.2 For an aircraft to be trim, the moments about the center of gravity must sum to be (greater than | less than | equal to) zero.

1.3 For an aircraft to be longitudinally stable, the change in pitching moment with respect to angle of attack must be (greater than | less than | equal to) zero.

1.4 Another name for the aerodynamic center of an aircraft is the _____.

1.5 Rank the following in order of magnitude for a typical aircraft (1 = largest, 2 = next largest, 3 = smallest)
____ The lift slope of the airfoil section used on the main wing, $\tilde{C}_{L,\alpha}$
____ The lift slope of the main wing, $C_{L_w,\alpha}$
____ The lift slope of the entire aircraft, $C_{L,\alpha}$

1.6 The static margin can be computed by subtracting the location of the _____ from the location of the _____ and dividing by the _____.

1.7 Alternately, the static margin can be found by the negative of the ratio of the _____ to the _____.

1.8 A common rule of thumb is that the static margin should be near $\sigma \approx$ _____.

1.9 (True | False) The simplified linear longitudinal analysis, in which all components lie along the fuselage reference line, predicts that the pitch stability derivative is independent of angle of attack.

1.10 (True | False) The generalized longitudinal analysis, in which nonlinearities and vertical offsets are included, predicts that the pitch stability derivative is independent of angle of attack.

2. Work-Out Problems (50 Points)

One version of the Lockheed Martin F-16 Fighting Falcon has the following geometric and aerodynamic properties:

$$S_w = 300 \text{ ft}^2, \quad b_w = 30.0 \text{ ft}, \quad R_{T_w} = 0.2275, \quad \Lambda_{c/4_w} = 32 \text{ deg},$$
$$S_h = 63.7 \text{ ft}^2, \quad b_h = 11.6 \text{ ft}, \quad R_{T_h} = 0.39, \quad \Lambda_{c/4_h} = 32 \text{ deg},$$
$$C_{L_w,\alpha} = 3.73, \quad \alpha_{0_w} = 1.9°, \quad \alpha_{L0_w} = -1.62°, \quad C_{m_w} = -0.041,$$
$$C_{L_h,\alpha} = 3.20, \quad \alpha_{0_h} = 0.0°, \quad \alpha_{L0_h} = 0.0°, \quad \varepsilon_e = 1.0, \quad C_{m_h,\delta_e} = 0.0$$

The longitudinal components all lie directly on the fuselage reference line, and their locations are known relative to the nose of the aircraft as follows:

Component	Distance Aft of Nose [ft]
CG	26.94
Main Wing Aerodynamic Center	26.42
Horizontal Tail Aerodynamic Center	43.55

Notice that this aircraft has an all-flying tail, i.e. $\varepsilon_e = 1.0$. For the following analysis, neglect the influence of the fuselage and engine, assume the tail efficiency factor, η_h, is unity, and neglect the upwash on the main wing due to the horizontal tail.

2.1 Draw a free-body diagram of this aircraft using the simplified model discussed in Section 4.3 of the book. Include all major forces and moments for the longitudinal components.

2.2 Using the simplified analysis given in Section 4.5 of the book, estimate the downwash angle on the aft tail as a function of angle of attack, $\varepsilon_d = \varepsilon_{d0} + \varepsilon_{d,\alpha}\alpha$. Compute values for

$$\varepsilon_{d0} =$$

$$\varepsilon_{d,\alpha} =$$

2.3 Estimate the static margin of this aircraft.

$$\sigma =$$

For the following two problems, assume the aircraft is operating near take-off conditions at a velocity of 225 ft/s, a weight of 20,500 lbf, in steady-level flight at sea level.

2.4 Compute the lift coefficient for the complete aircraft at this operating condition.

$$C_L =$$

2.5 Compute the angle of attack and elevator deflection **in degrees** required to trim the aircraft at this operating condition.

$$\alpha =$$

$$\delta_e =$$