A02176917

MAE 5510 Flight Dynamics : Spring 2024 Final Exam Name Eric Larsen

Open book. Open notes. Open recorded lectures and all other resources provided from this course. Closed neighbor, additional internet, AI, or any other resource. All work done neatly and logically. Calculators allowed.

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

Part 1. Short Answer (2 Points Each = 100 points) Note: Circle one option when multiple options separated by the symbol | are shown.

1.1 Consider an aircraft with a single wing with zero sweep. The wing uses a simply cambered airfoil with positive camber. The aircraft does not have an empennage. In order for this aircraft to be trim, the center of gravity must be located (forward | at | aft) of the aerodynamic center of the wing.

1.2 Consider an aircraft with a single wing with zero sweep. The wing uses a simply cambered airfoil with positive camber. The aircraft does not have an empennage. In order for this aircraft to be stable in pitch, the center of gravity must be located (forward | at | aft) of the aerodynamic center of the wing.

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

1.4 (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.

1.5 Shifting the center of gravity of an aircraft forward will make the pitch-stability derivative (more positive | more negative | not change).

1.6 Shifting the center of gravity of an aircraft aft will make the static margin (more positive | more negative | not change).

1.7 The downwash angle in radians in the symmetry plane (z=0) behind an elliptic wing approaches a value of $16/\pi^3$ ____ C_{L_w}/R_{A_w} far downstream of the wing.

1.8 For a canard design, the main wing is (stabilizing | destabilizing) and the canard is (stabilizing | destabilizing).

1.9 A propeller mounted in front of the center of gravity is longitudinally (stabilizing | destabilizing).

1.10 In order for an aircraft to be stable about all three axes, the following must be true: (Circle one symbol for each)

$C_{\ell,\beta}$ (> | = | <) 0.

$C_{m,\alpha}$ (> | = | <) 0.

$C_{n,\beta}$ (> | = | <) 0.

1.11 Increasing positive dihedral makes the roll-stability derivative (more positive | more negative | not change).

more stable.

1.12 Moving the vertical tail further aft makes the yaw-stability derivative (more positive | more negative | not change).

1.13 Carrying a nacelle or tank directly below the wing tip makes the roll-stability derivative (more positive | more negative | not change).

destab. effect.

$C_{\ell,\beta} < 0$ "roll"

$C_{m,\alpha} < 0$ "pitch"

$C_{n,\beta} > 0$ "yaw"

1.14 When a traditional aircraft uses aileron deflection to roll the aircraft to the right, many aircraft experience adverse yaw, which for this scenario is the tendency of the aircraft to yaw to the (left | right).

1.15 The stick-fixed neutral point of an aircraft near the ground is (further forward | further aft | the same) than when the aircraft is not near the ground.

1.16 Frise ailerons are sometimes used to reduce or eliminate adverse yaw.

1.17 Which one of the following modes usually has the highest damping rate?
a. Short-period mode
b. Phugoid mode
c. Dutch roll mode
d. Roll mode
e. Spiral mode

For problems 1.18–1.24, use the following information: For a certain airplane, the **dimensional** eigenvalues obtained from the linearized longitudinal and lateral equations of motion are:

	Longitudinal	Lateral	
Short period	$\lambda = (-2.2000 + 3.1000i)\ \text{sec}^{-1}$	$\lambda = (-8.2000 + 0.0000i)\ \text{sec}^{-1}$	Roll mode.
	$\lambda = (-2.2000 - 3.1000i)\ \text{sec}^{-1}$	$\lambda = (-0.6700 + 2.6000i)\ \text{sec}^{-1}$	dutch roll
phugoid	$\lambda = (-0.0150 + 0.4200i)\ \text{sec}^{-1}$	$\lambda = (-0.6700 - 2.6000i)\ \text{sec}^{-1}$	
	$\lambda = (-0.0150 - 0.4200i)\ \text{sec}^{-1}$	$\lambda = (-0.0070 + 0.0000i)\ \text{sec}^{-1}$	Spiral
	$\lambda = (0.0000 + 0.0000i)\ \text{sec}^{-1}$	$\lambda = (0.0000 + 0.0000i)\ \text{sec}^{-1}$	
	$\lambda = (0.0000 + 0.0000i)\ \text{sec}^{-1}$	$\lambda = (0.0000 + 0.0000i)\ \text{sec}^{-1}$	

$W_n = \sqrt{\sigma^2 + W_d^2}$

$W_d = |imag|$

$\sigma = -|real|$

$\zeta = \frac{\sigma}{W_n}$

Compute the following and give units if any:

1.18. phugoid period $= \frac{2\pi}{W_d}$ $W_d = |imag|$ $\frac{2\pi}{0.42}$ $= \boxed{14.95996 \text{ Seconds}}$

14.95996 Seconds

1.19. phugoid damping ratio $W_n = 0.42026$ $\frac{0.0150}{0.42026} = \boxed{0.03569}$

1.20. short-period damped natural frequency W_d $\boxed{3.100\ \text{sec}^{-1}}$

1.21. short-period mode 99% damping time $\frac{-\ln(0.01)}{\sigma} = \frac{-\ln(0.01)}{2.2} = \boxed{2.09325 \text{ Seconds}}$

1.22. Dutch roll undamped natural frequency $W_n = \sqrt{\sigma^2 + W_d^2} = \sqrt{0.67^2 + 2.6^2} = \boxed{2.68493\ \text{Seconds}^{-1}}$

1.23. roll mode time constant $1/\sigma = \frac{1}{-real(\lambda)}$ $1/8.200 = \boxed{0.12195 \text{ Seconds}}$

1.24. spiral mode damping rate $\sigma = -real(\lambda) = \boxed{0.0070\ \text{Sec}^{-1}}$

1.25 The ratio $\frac{l_{np}}{\bar{c}_w}$ is called the Static Margin.

1.26 The ratio $\frac{l_{mp}}{r_{yy_b}}$ is called the Dynamic Margin.

1.27 The ratio $\frac{(\omega_n^2)_{SP}}{C_{L,\alpha}/C_W}$ is called the Control Anticipation Parameter (CAP)

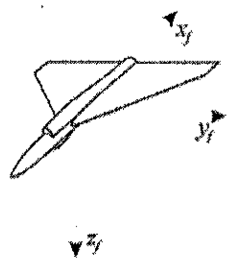
1.28 The ratio $\frac{g l_{mp}}{r_{yy_b}^2}$ is called the Control Anticipation Parameter (CAP)

1.29 The phugoid or long period (longitudinal) mode is an interchange between translational kinetic energy and potential energy.

1.30 Which of the following can vary during pure longitudinal motion? Circle all that apply.

a. axial velocity b. normal velocity c. sideslip velocity
d. roll rate e. pitch rate f. yaw rate
g. bank angle h. elevation angle i. azimuth angle

1.31 Determine the Euler angles that describe this position:



$\phi =$ 0 degrees
$\theta =$ 180 degrees
$\psi =$ 0 degrees

1.32 The derivative $C_{L,\alpha}$ is called the Lift slope.

1.33 The derivative $C_{m,\alpha}$ is called the Pitch stability derivative.

1.34 The derivative $C_{\ell,\beta}$ is called the Roll stability derivative.

1.35 The derivative $C_{n,\beta}$ is called the Yaw stability derivative.

1.36 The derivative $C_{m,\bar{q}}$ is called the Pitch damping derivative.

1.37 The derivative $C_{\ell,\bar{p}}$ is called the Roll damping derivative.

1.38 The derivative $C_{n,\bar{r}}$ is called the Yaw damping derivative.

1.39 Change in forces and moments with respect to the aerodynamic angles α and β are called Aerodynamic derivatives.

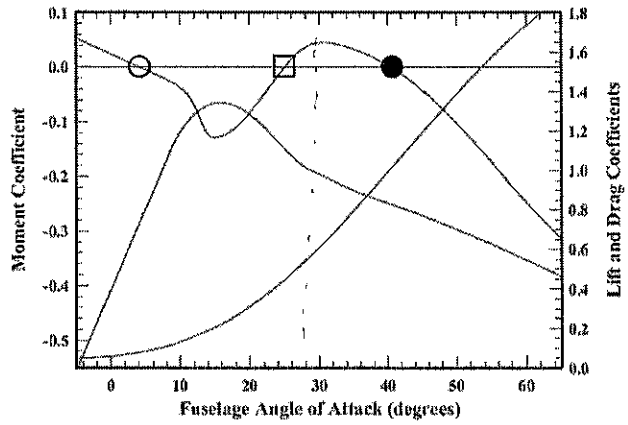
1.40 Change in forces and moments with respect to rotational rates p, q, and r are called damping derivatives.

1.41 Change in forces and moments with respect to control-surface deflections δ_a, δ_e, and δ_r are called Control derivatives.

1.42 Which stability or damping derivative could you change to increase the damping for both the spiral mode and the Dutch roll mode?

Yaw damping deriv. $C_{n_{\bar{r}}}$

For problems 1.43–1.50, refer to the following figure:



1.43 The open and filled circles on this plot denote (stable | unstable) trim points.

1.44 The square on this plot denotes a/an (stable | unstable) trim point.

1.45 The filled circle on this plot denotes a point with a specific name often referred to as a deep-stall trim point.

1.46 Assume an aircraft has the properties shown in the figure above. If this aircraft found itself at an angle of attack of 28 degrees, it would naturally rotate to a trim point denoted by the (open circle | square | filled circle).

1.47 Below stall, the lift is very nearly a (linear | quadratic) function of angle of attack.

1.48 Below stall, the drag is very nearly a (linear | quadratic) function of angle of attack.

1.49 At 90 degrees angle of attack, the lift coefficient is very nearly (0.0 | 1.0 | 2.0).

1.50 At 90 degrees angle of attack, the drag coefficient is very nearly (0.0 | 1.0 | 2.0).

if γ is not known assume and iterate on solution until true alpha is found for $C_L = C_W$ for trim flight for this I am assuming level flight so my $C_L = C_W$ to solve.

Part 2. Work-Out Problems (10 Points Each = 100 points)

For these work out problems, use the following information:

The F-16 Fighting Falcon is a single-engine aircraft with the engine aligned with the centerline of the aircraft. Hence the propulsion system produces no yawing moment or side force. Assume that the propulsion system also produces no rolling moment. For the purposes of these problems, assume the properties and operating condition for the F-16 flying at sea level can be approximated as:

$$S_w = 300 \text{ ft}^2, \quad b_w = 30 \text{ ft} \quad W = 26,500 \text{ lbf}, \quad \rho = 0.0023769 \text{ slug/ft}^3 \quad V = 240 \text{ ft/s}$$

$\alpha = 0.2431$ rad

$\delta_e = -0.06998$ rad

can solve iff $C_L = C_W$ steady flight

$$C_L = 0.105 + 5.01\alpha + 0.465\delta_e$$
$$C_D = 0.018 + 0.048C_L^2$$

← does this matter "only for thrust required"

$$C_Y = -0.525\beta + 0.135\delta_r$$
$$C_\ell = -0.028\beta - 0.140\delta_a + 0.004\delta_r$$
$$C_m = -0.380\alpha - 1.320\delta_e$$
$$C_n = 0.089\beta - 0.012\delta_a - 0.050\delta_r$$

"Assuming" and $\gamma = 0$ $C_L = C_W$

where all aerodynamic and control-surface angles are in radians.

Compute the following:

$$C_L = \frac{26,500}{\frac{1}{2}(0.0023769)\, 240^2\, 300} = 1.29039$$

2.1 $C_L =$ 1.29039

2.2 $C_{L,\alpha} =$ 5.01 $\quad \frac{dC_L}{d\alpha} = 5.01$

2.3 $C_{m,\alpha} =$ −0.380 $\quad \frac{dC_m}{d\alpha} = -0.380$

2.4 $C_{L,\delta_e} =$ 0.465

2.5 $C_{m,\delta_e} =$ −1.320

$C_{L_0} = 0.105 \qquad C_{m_0} = 0$

$$\begin{bmatrix} C_{L,\alpha} & C_{L,\delta_e} \\ C_{m,\alpha} & C_{m,\delta_e} \end{bmatrix} \begin{bmatrix} \alpha \\ \delta_e \end{bmatrix} = \begin{bmatrix} C_L - C_{L_0} \\ -C_{m_0} \end{bmatrix}$$

$$\begin{bmatrix} 5.01 & 0.465 \\ -0.380 & -1.320 \end{bmatrix} \begin{bmatrix} \alpha \\ \delta_e \end{bmatrix} = \begin{bmatrix} 1.29039 - 0.105 \\ 0 \end{bmatrix}$$

2.6 Compute the angle of attack **in degrees** required to trim at zero bank angle: $\alpha =$ 13.9286 degrees.

2.7 Compute the elevator deflection **in degrees** required to trim at zero bank angle: $\delta_e =$ −4.0095 degrees.

Computed via RREF on Calc.

For problems 2.8–2.10, assume the aircraft is now operating at the same airspeed, but at a bank angle of +5 degrees. Use the small-angle lateral trim equations to estimate the following:

2.8 Compute the sideslip angle **in degrees** required to trim: $\beta =$ 23.0216 degrees.

2.9 Compute the aileron deflection **in degrees** required to trim: $\delta_a =$ −3.4102 degrees.

2.10 Compute the rudder deflection **in degrees** required to trim: $\delta_r =$ 41.79669 degrees.

$$\begin{bmatrix} C_{y,\beta} & C_{y,\delta_a} & C_{y,\delta_r} \\ C_{l,\beta} & C_{l,\delta_a} & C_{l,\delta_r} \\ C_{n,\beta} & C_{n,\delta_a} & C_{n,\delta_r} \end{bmatrix} \begin{bmatrix} \beta \\ \delta_a \\ \delta_r \end{bmatrix} = \cancel{\begin{bmatrix} C_{y_p} \\ C_{l_p} \\ C_{n,\beta} \end{bmatrix}}^{0} + \begin{bmatrix} -C_W \sin\phi \\ 0 \\ 0 \end{bmatrix} + \begin{bmatrix} 0 \\ 0 \\ 0 \end{bmatrix}$$

Solved on spare paper and RREF on Calc.

Part 2) Work.

$$CL = \frac{W\cos\alpha}{\frac{1}{2}\rho V^2 S W} = \frac{26{,}500}{\frac{1}{2}(0.0023769)\cdot 240^2\cdot 300}\cos\alpha$$

$$CL = 1.29039\cos\alpha$$

okay I have $\alpha, \beta, \delta_e, \delta_r, \delta_a,$ that need to be solved.

$$\begin{array}{c} C_L \\ C_y \\ C_l \\ C_m \\ C_n \end{array}\begin{bmatrix} & & & & \\ & & & & \\ & & & & \\ & & & & \\ & & & & \end{bmatrix}\begin{bmatrix}\alpha \\ \beta \\ \delta_e \\ \delta_r \\ \delta_a\end{bmatrix} = \begin{bmatrix} -0.105 \\ \\ \\ \\ \\ \end{bmatrix}$$

Break into lateral and long

$$\begin{bmatrix} 5.01 & 0.465 \\ -0.380 & -1.320 \end{bmatrix} \sim \begin{bmatrix} 1.29039 - 0.105 \\ 0 \end{bmatrix} \begin{bmatrix} 1.18539 \\ 0 \end{bmatrix}$$

$\alpha = 0.2431$ rad $= 13.9286$ deg

$\delta_e = -0.06998$ rad $= -4.0095$ deg

Part 2 problems $8 \to 10$

$C_{y,\beta} = -0.525$ $\quad C_{l,\beta} = -0.028$ $\quad C_{n,\beta} = 0.089$

$C_{y,\delta_a} = 0$ $\quad C_{l,\delta_a} = -0.140$ $\quad C_{n,\delta_a} = -0.012$

$C_{y,\delta_r} = 0.135$ $\quad C_{l,\delta_r} = 0.004$ $\quad C_{n,\delta_r} = -0.080$

$\phi = 5$

$$\begin{bmatrix} C_{y,\beta} & C_{y,\delta_a} & C_{y,\delta_r} \\ C_{l,\beta} & C_{l,\delta_a} & C_{l,\delta_r} \\ C_{n,\beta} & C_{n,\delta_a} & C_{n,\delta_r} \end{bmatrix} \begin{bmatrix} \beta \\ \delta_a \\ \delta_r \end{bmatrix} = \begin{bmatrix} -C_W \sin\phi \\ 0 \\ 0 \end{bmatrix}$$

$$\begin{matrix} C_{yp} \\ C_{lp} \\ C_{np} \end{matrix} = \begin{bmatrix} 0 \\ 0 \\ 0 \end{bmatrix}$$

"problem statement."

$$C_W = \frac{26{,}500}{0.5\,(0.0023769)\,240^2\,300} = 1.29039$$

$$C_W \sin\phi = 1.29039 \sin(\overset{deg}{5}) = \underline{0.1124649}$$

$$\begin{bmatrix} -0.525 & 0 & 0.135 \\ -0.028 & -0.140 & 0.004 \\ 0.089 & -0.012 & -0.080 \end{bmatrix} \begin{bmatrix} \beta \\ \delta_a \\ \delta_r \end{bmatrix} = \begin{bmatrix} -0.1124649 \\ 0 \\ 0 \end{bmatrix}$$

RREF on calc,

$\beta = 0.4018028$ rad $= 23.0216$ deg //

$\delta_a = -0.05952$ rad $= -3.4102$ deg //

$\delta_r = 0.72949$ rad $= 41.79669$ deg //