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Universiti Tun Hussein Onn Malaysia

UNIVERSITI TUN HUSSEIN ONN MALAYSIA

FINAL EXAMINATION
SEMESTER I
SESSION 2019/2020

COURSE NAME	:	ELECTROMECHANICAL AND CONTROL SYSTEM
COURSE CODE	:	BDU 20302
PROGRAMME	:	BDC/BDM
EXAMINATION DATE	:	DECEMBER 2019/JANUARY 2020
DURATION	:	2 HOURS 30 MINUTES
INSTRUCTION	:	ANSWER **FOUR (4)** QUESTIONS **ONLY**

THIS QUESTION PAPER CONSISTS OF **ELEVEN (11)** PAGES

Q1 (a) Explain the concept of static stability and list the necessary criteria for longitudinal balance and static stability.

(2 marks)

(b) Consider a wing-body shaped model mounted in a test section of a low-speed wind tunnel. The flow conditions in the test section are assumed to be at standard sea level properties. The wing-body model is tested with an airflow velocity of 100 m/s. The wing area and chord are 1.5 m^2 and 0.45 m, respectively. Using the wind tunnel force and moment measuring balance, the moment about the center of gravity when the lift force measurement is zero is found to be -12.4 Nm. When the model is pitched to another angle of attack, the lift and moment about the center of gravity are measured to be 3675 N and 20.67 Nm, respectively. Calculate the value of the moment coefficient about the aerodynamic center and determine the location of the aerodynamic center.

(4 marks)

(c) Assume that a horizontal tail with no elevator is added to the wing-body model in **Question Q1(b)**. The distance from the aircraft's center of gravity to the tail's aerodynamic center is 1.2 m. The area of the tail is 0.5 m^2, and the tail setting angle is 2.1°. The lift slope of the tail is 0.15 per degree. From experimental measurement, $\varepsilon_0 = 0$, and $\partial\varepsilon/\partial\alpha = 0.45$. If the absolute angle of attack of the model is 5° and the lift at this angle of attack is 4135 N, calculate the moment about the center of gravity.

(6 marks)

(d) Consider the wing-body-tail model in **Question Q1(c).** Examine the stability of the model in terms of longitudinal static stability and balance. Determine the trimmed angle of attack, the neutral point, and the static margin for the wing-body-tail model. Assume that the location of the center of gravity is at $h = 0.3$.

(10 marks)

(e) Assume that an elevator is added to the horizontal tail of the configuration given in **Question Q1(c).** The elevator control effectiveness is 0.05. Calculate the elevator deflection angle necessary to trim the configuration at an angle of 6°.

(3 marks)

Q2 (a) List the characteristics of the short period and phugoid stability modes.

(5 marks)

(b) Consider an aircraft model in a wind tunnel setup where the aircraft is constrained at its center of gravity. The aircraft is free to perform a pitching motion about its center of gravity. The governing equation of this simple motion is obtained from Newton's second law and is given as:

$$\Delta\ddot{\alpha} - \left(M_q + M_{\dot{\alpha}}\right)\Delta\dot{\alpha} - M_\alpha \Delta\alpha = M_{\delta e}\Delta\delta_e$$

where $\Delta\alpha$ is the change in the angle of attack (Assumption: the change in the angle of attack and pitch angles are identical), $\Delta\delta e$ is the change in elevator angle. Derivatives, M_q and M_α are the longitudinal derivatives due to pitching velocity and angle of attack. Find the transfer function relating the change in the angle of attack, $\Delta\alpha(s)$ and the change in elevator angle $\Delta\delta e(s)$. Use the Laplace transform theorem in **Table Q2(b).**

(2 marks)

(c) Determine the solution, $\alpha(t)$ for the governing equation in **Question Q2(b)** if a step input is applied to the elevator using the following data:

$$M_q = -2.05\, s^{-1}$$
$$M_\alpha = -8.80\, s^{-2}$$
$$M_{\dot{\alpha}} = -0.95\, s^{-2}$$
$$M_{\delta e} = -5.5\, s^{-2}$$

Use partial fraction and inverse Laplace theorem to obtain the output response of the system, $\alpha(t)$ with the initial conditions of $\alpha(0) = 0$ and $\frac{d\alpha(0)}{dt} = 0$.

(14 marks)

(d) Analyze the effect of parameter $(M_q + M_{\dot{\alpha}})$ and M_α on the stability of short-period mode.

(4 marks)

Q3 A simplified pitch control system is shown in **Figure Q3** with transfer functions for each individual component in the control system are given as:

$$K(s) = K_P + \frac{K_I}{s} + K_D s$$

$$G_1(s) = \frac{10}{s + 10}$$

$$G_2(s) = \frac{3}{s^2 + 3s + 4}$$

(a) Examine the locus movement of the open-loop transfer function in a root locus plot and find the damped frequency, ω_d and gain, K, values at the imaginary axis crossing if such a situation exists.

(10 marks)

(b) Design the automatic controllers (i.e., P, PD, and PID control) for the dynamic system under consideration using the Ziegler and Nichols tuning method.

(8 marks)

(c) Compare the steady-state error performance of the compensated systems (i.e., P, PD, and PID control). Describe any problems with your design.

(7 marks)

Q4 (a) Describe the physical characteristics of Dutch Roll stability mode.

(3 marks)

(b) The Dutch Roll motion can be approximated using the following equation:

$$\begin{bmatrix} \Delta\dot{\beta} \\ \Delta\dot{r} \end{bmatrix} = \begin{bmatrix} \frac{Y_\beta}{u_0} & -\left(1 - \frac{Y_r}{u_0}\right) \\ N_\beta & N_r \end{bmatrix} \begin{bmatrix} \Delta\beta \\ \Delta r \end{bmatrix} + \begin{bmatrix} \frac{Y_{\delta r}}{u_0} \\ N_{\delta r} \end{bmatrix} \delta r$$

Assume the aircraft has the following stability derivative characteristics as follows:

$Y_\beta = -7.1$ ft/s^2	$Y_r = 2.1$ ft/s
$N_\beta = 2.9$ s^{-2}	$N_r = -0.325$ s^{-1}
$Y_{\delta r} = -4.9$ ft/s^2	$N_{\delta r} = 0.615$ s^{-2}
$u_0 = 160$ ft/s	

(i) Determine the characteristic equation of the Dutch Roll mode.

(4 marks)

(ii) Determine the eigenvalues of the Dutch Roll mode.

(2 marks)

(iii) Determine the damping ratio, natural frequency, period, time to half amplitude, and the number of cycles to half amplitude for the Dutch Roll mode.

(5 marks)

(c) Compare your Dutch Roll mode natural frequency and damping ratio calculation for this particular aircraft to the handling quality criteria in **Table Q4(c).** Assume that the aircraft under consideration is a **Class IV aircraft** (High Maneuvering Aircraft) performing **CAT A mission** (Precision Tracking). Determine the minimum state feedback gain so that the damping ratio achieved **Level 1** handling qualities. Use feedback control design based on **rudder deflection**, δr, proportional only to the **yaw rate state**, Δr, i.e.:

$$\delta r = -K^T x = -[0 \quad K] \begin{bmatrix} \Delta\beta \\ \Delta r \end{bmatrix}$$

(11 marks)

Q5 The Wright Flyer was known to be a statically and dynamically unstable aircraft. However, the Wright brothers were able to fly their aircraft successfully due to sufficient control authority incorporated into their aircraft design. Although the airplane was difficult to fly, the combination of the pilot and aircraft could be a stable system (see **Figure Q5**). If the closed-loop pilot is represented as a pure gain, K_p, and the pitch attitude to canard deflection is given as follows:

$$G_2(s) = \frac{K_p(s+0.5)(s+3)}{(s^2+0.72s+1.44)(s^2+5.9s-11.9)}$$

(a) Determine the centroid and asymptotes angle of the root locus plot of the closed-loop system.

(6 marks)

(b) Calculate the angle of departure from the complex poles.

(6 marks)

(c) Draw the root locus for the closed-loop system.

(8 marks)

(d) Determine the range of pilot gain, K_p for which the system is stable.

(5 marks)

-END OF QUESTION-

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Table Q2(b) The Laplace transform theorems.

Laplace transforms – Table			
$f(t) = L^{-1}\{F(s)\}$	$F(s)$	$f(t) = L^{-1}\{F(s)\}$	$F(s)$
$a \quad t \geq 0$	$\frac{a}{s} \quad s > 0$	$\sin \omega t$	$\frac{\omega}{s^2+\omega^2}$
$at \quad t \geq 0$	$\frac{a}{s^2}$	$\cos \omega t$	$\frac{s}{s^2+\omega^2}$
e^{-at}	$\frac{1}{s+a}$	$\sin(\omega t + \theta)$	$\frac{s \sin\theta + \omega \cos\theta}{s^2+\omega^2}$
te^{-at}	$\frac{1}{(s+a)^2}$	$\cos(\omega t + \theta)$	$\frac{s \cos\theta - \omega \sin\theta}{s^2+\omega^2}$
$\frac{1}{2}t^2e^{-at}$	$\frac{1}{(s+a)^3}$	$t \sin \omega t$	$\frac{2\omega s}{(s^2+\omega^2)^2}$
$\frac{1}{(n-1)!}t^{n-1}e^{-at}$	$\frac{1}{(s+a)^n}$	$t \cos \omega t$	$\frac{s^2-\omega^2}{(s^2+\omega^2)^2}$
e^{at}	$\frac{1}{s-a} \quad s > a$	$\sinh \omega t$	$\frac{\omega}{s^2-\omega^2} \quad s > \lvert\omega\rvert$
te^{at}	$\frac{1}{(s-a)^2}$	$\cosh \omega t$	$\frac{s}{s^2-\omega^2} \quad s > \lvert\omega\rvert$
$\frac{1}{b-a}(e^{-at} - e^{-bt})$	$\frac{1}{(s+a)(s+b)}$	$e^{-at}\sin \omega t$	$\frac{\omega}{(s+a)^2+\omega^2}$
$\frac{1}{a^2}[1 - e^{-at}(1+at)]$	$\frac{1}{s(s+a)^2}$	$e^{-at}\cos \omega t$	$\frac{s+a}{(s+a)^2+\omega^2}$
t^n	$\frac{n!}{s^{n+1}} \quad n = 1,2,3..$	$e^{at}\sin \omega t$	$\frac{\omega}{(s-a)^2+\omega^2}$
$t^n e^{at}$	$\frac{n!}{(s-a)^{n+1}} \quad s > a$	$e^{at}\cos \omega t$	$\frac{s-a}{(s-a)^2+\omega^2}$
$t^n e^{-at}$	$\frac{n!}{(s+a)^{n+1}} \quad s > a$	$1 - e^{-at}$	$\frac{a}{s(s+a)}$
$\sqrt{t}$	$\frac{\sqrt{\pi}}{2s^{3/2}}$	$\frac{1}{a^2}(at - 1 + e^{-at})$	$\frac{1}{s^2(s+a)}$
$\frac{1}{\sqrt{t}}$	$\sqrt{\frac{\pi}{s}} \quad s > 0$	$f(t - t_1)$	$e^{-t_1 s}F(s)$
$g(t) \cdot p(t)$	$G(s) \cdot P(s)$	$f_1(t) \pm f_2(t)$	$F_1(s) \pm F_2(s)$
$\int f(t)dt$	$\frac{F(s)}{s} + \frac{f^{-1}(0)}{s}$	$\delta(t)$ *unit impulse*	1 *all s*
$\frac{df}{dt}$	$sF(s) - f(0)$	$\frac{d^2f}{df^2}$	$s^2F(s) - sf(0) - f'(0)$
$\frac{d^nf}{dt^n}$	$s^nF(s) - s^{n-1}f(0) - s^{n-2}f'(0) - s^{n-3}f''(0) - \cdots - f^{n-1}(0)$		

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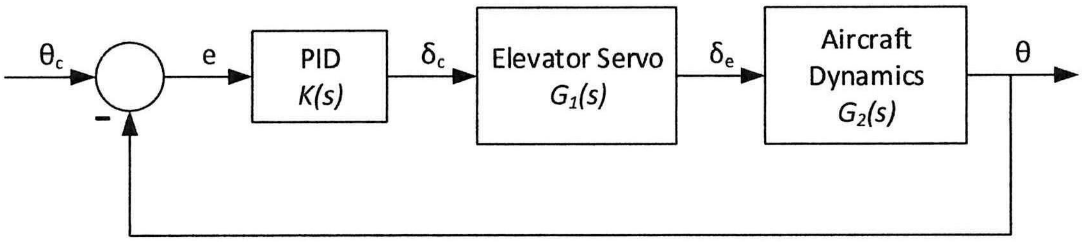

Figure Q3 Simplified block diagram for pitch angle control system.

Table Q4(c) Minimum Dutch-Roll frequency and damping

Aircraft Class	Flight Phase	Minimum Values							
		Level 1			Level 2			Level 3	
		ζ	$\zeta\omega_n$	ω_n	ζ	$\zeta\omega_n$	ω_n	ζ	ω_n
I, IV	CAT A	0.19	0.35	1.0	0.02	0.05	0.5	0	0.4
II, III	CAT A	0.19	0.35	0.5	0.02	0.05	0.5	0	0.4
All	CAT B	0.08	0.15	0.5	0.02	0.05	0.5	0	0.4
I, IV	CAT C	0.08	0.15	1.0	0.02	0.05	0.5	0	0.4
II, III	CAT C	0.08	0.10	0.5	0.02	0.05	0.5	0	0.4

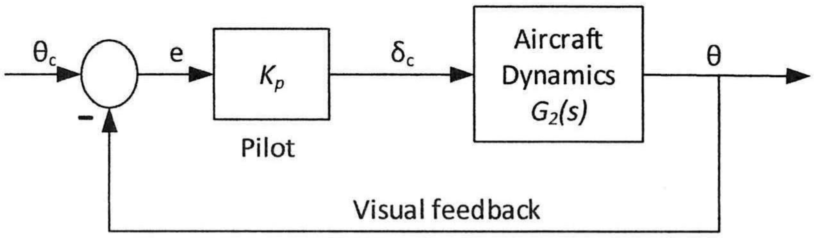

Figure Q5 The block diagram for the Wright Flyer control system.

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A Key Equations

The relevant equations used in this examination are given as follows:

1. The determinant of a 3 × 3 matrix:

$$\begin{vmatrix} a & b & c \\ d & e & f \\ g & h & i \end{vmatrix} = a\begin{vmatrix} e & f \\ h & i \end{vmatrix} - b\begin{vmatrix} d & f \\ g & i \end{vmatrix} + c\begin{vmatrix} d & e \\ g & h \end{vmatrix} \quad (1)$$

2. Partial fraction for $F(s)$ with real and distinct roots in the denominator:

$$F(s) = \frac{K_1}{(s+p_1)} + \frac{K_2}{(s+p_2)} + \cdots + \frac{K_m}{(s+p_m)} \quad (2)$$

3. Partial fraction for $F(s)$ with complex or imaginary roots in the denominator:

$$F(s) = \frac{K_1}{(s+p_1)} + \frac{K_2 s + K_3}{(s^2+as+b)} + \cdots \quad (3)$$

4. General first order transfer function:

$$G(s) = \frac{s}{s+a} \quad (4)$$

5. General second order transfer function:

$$G(s) = \frac{K}{s^2 + 2\xi\omega_n s + \omega_n} \quad (5)$$

6. The closed loop transfer function:

$$T(s) = \frac{G(s)}{1+G(s)H(s)} \quad (6)$$

where $G(s)$ is the transfer function of the open-loop system, and $H(s)$ is the transfer function in the feedback loop.

7. The final value theorem:

$$\lim_{t\to\infty} y(t) = \lim_{s\to 0} sY(s) \quad (7)$$

8. Time response:

$$T_r = \frac{2.2}{a} \quad (8)$$

$$T_s = \frac{4}{a} \quad (9)$$

$$\%OS = e^{-\left(\frac{\xi\pi}{\sqrt{1-\xi^2}}\right)} \times 100\% \quad (10)$$

$$\xi = \frac{-\ln\left(\%\frac{OS}{100}\right)}{\sqrt{\pi^2 + \left(\ln\left(\%\frac{OS}{100}\right)\right)^2}} \quad (11)$$

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$$T_p = \frac{\pi}{\omega_n\sqrt{1-\xi^2}} = \frac{\pi}{\omega} \quad (12)$$

$$T_s = \frac{4}{\xi\omega_n} = \frac{4}{\eta} \quad (13)$$

$$P = \frac{2\pi}{\omega} \quad (14)$$

$$t_{1/2} = \frac{0.693}{|\eta|} \quad (15)$$

$$N_{1/2} = 0.110\frac{|\omega|}{|\eta|} \quad (16)$$

9. Estimation of parameter q (Paynter Numerical Method)

$$q = \max\left|A_{ij}\Delta t\right| \quad (17)$$

10. Estimation of the integer value of p (Paynter Numerical Method)

$$\frac{1}{p!}(nq)^p e^{nq} \leq 0.001 \quad (18)$$

11. Numerical solution of state equation:

$$\mathbf{x}_{k+1} = \boldsymbol{M}\mathbf{x}_k + \boldsymbol{N}\boldsymbol{\eta}_k$$

with matrix ***M*** and ***N*** are given by the following matrix expansion:

$$\boldsymbol{M} = e^{A\Delta t} = \boldsymbol{I} + \boldsymbol{A}\Delta t + \frac{1}{2!}\boldsymbol{A}^2\Delta t^2 \ldots \quad (19)$$

$$\boldsymbol{N} = \Delta t\left(\boldsymbol{I} + \frac{1}{2!}\boldsymbol{A}\Delta t + \frac{1}{3!}\boldsymbol{A}^2\Delta t^2 + \cdots\right)\boldsymbol{B}$$

12. Characteristic equation of the closed loop system:

$$1 + KG(s)H(s) = 0 \quad (20)$$

13. Asymptotes: angle and real-axis intercept:

$$\sigma = \frac{[\sum Real\ parts\ of\ the\ poles - \sum Real\ parts\ of\ the\ zeros]}{n-m} \quad (21)$$

$$\phi_a = \frac{180°[2q+1]}{n-m} \quad (22)$$

14. The solution to determine real axis break-in and breakaway points:

$$\frac{dK(\sigma)}{d\sigma} = 0 \quad (23)$$

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15. An alternative solution to find real axis break-in and breakaway points:

$$\sum \frac{1}{\sigma + z_i} = \sum \frac{1}{\sigma + p_i} \quad (24)$$

16. The angle of departure of the root locus from a pole of $G(s)H(s)$:

$$\theta = 180° + \sum (angles\ to\ zeros) - \sum (angles\ to\ poles) \quad (25)$$

17. The angle of arrival at a zero:

$$\theta = 180° - \sum (angles\ to\ zeros) + \sum (angles\ to\ poles) \quad (26)$$

18. The steady state error:

$$e_{ss} = \lim_{t \to \infty} e(t) = \lim_{s \to 0} sE(s) \quad (27)$$

where the error signal is given as:

$$E(s) = \frac{1}{1 + K(s)G(s)H(s)} \times U(s) \quad (28)$$

19. Linear system (SISO) with state feedback:

$$\dot{\boldsymbol{x}} = (\boldsymbol{A} - \boldsymbol{B}\boldsymbol{K}^T)\boldsymbol{x} + \boldsymbol{B}u$$

$$\text{or, } \dot{\boldsymbol{x}} = \boldsymbol{A}_{new}\boldsymbol{x} + \boldsymbol{B}u \quad (29)$$

where $\boldsymbol{A}_{new}$ is the augmented matrix and $u = \boldsymbol{K}^T\boldsymbol{x} + \delta_{ref}$

20. The characteristic equation for the standard form of the second-order differential equation:

$$\lambda^2 + 2\xi\omega_n\lambda + \omega_n^2 = 0$$

The roots of the characteristic equation are: (30)

$$\lambda_{1,2} = -\xi\omega_n \pm \omega_n\sqrt{1 - \xi^2} \cdot i$$

$$\lambda_{1,2} = \sigma \pm \omega_d$$

21. The calculation of controller gains using the Ziegler-Nichols method:

Table 1 The Ziegler-Nichols tuning method. (31)

Control Type	K_p	K_I	K_D
P	$0.5K_u$	-	-
PI	$0.45K_u$	$1.2\,K_p/T_u$	-
PD	$0.8K_u$	-	$K_PT_u/8$
Classic PID	$0.6K_u$	$2\,K_p/T_u$	$K_PT_u/8$
Pessen Integral Rule	$0.7K_u$	$2.5\,K_p/T_u$	$3K_PT_u/20$
Some Overshoot	$0.33K_u$	$2\,K_p/T_u$	$K_PT_u/3$
No Overshoot	$0.2K_u$	$2\,K_p/T_u$	$K_PT_u/3$

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22. The contribution of the wing-body to M_{cg}:

$$C_{M,cg_{wb}} = C_{M,ac_{wb}} + C_{L_{wb}}(h - h_{ac_{wb}})$$

$$C_{M,cg_{wb}} = C_{M,ac_{wb}} + a_{wb}\alpha_{wb}(h - h_{ac_{wb}}) \quad (32)$$

23. The contribution of the wing-body-tail to M_{cg}:

$$C_{M,cg_{wb}} = C_{M,ac_{wb}} + a\alpha_a\left(h - h_{ac_{wb}} - V_H\frac{a_t}{a}\left[1 - \frac{\partial\varepsilon}{\partial\alpha}\right]\right) + V_H a_t(i_t + \varepsilon_0) \quad (33)$$

24. The equation for longitudinal static stability:

$$C_{M,0} = C_{M,ac_{wb}} + V_H a_t(i_t + \varepsilon_0)$$

$$\frac{\partial C_{M,cg}}{\partial\alpha_a} = a\left[h - h_{ac_{wb}} - V_H\frac{a_t}{a}\left(1 - \frac{\partial\varepsilon}{\partial\alpha}\right)\right] \quad (34)$$

25. The absolute angle of attack, α_a:

$$\alpha_a = \alpha + |\alpha_{L=0}| \quad (35)$$

where α is the geometric angle of attack.

26. Neutral point:

$$h_n = h_{ac_{wb}} + V_H\frac{a_t}{a}\left(1 - \frac{\partial\varepsilon}{\partial\alpha}\right) \quad (36)$$

27. Static margin:

$$SM = h_n - h \quad (37)$$

28. Elevator angle to trim:

$$\delta_{trim} = \frac{C_{M,0} + (\partial C_{M,cg}/\partial\alpha_a)\alpha_n}{V_H(\partial C_{L,t}/\partial\,\delta_e)} \quad (38)$$