Question Paper Code : 71027

B.E./B.Tech. DEGREE EXAMINATION, APRIL/MAY 2015.

Fifth Semester

Aeronautical Engineering

AE 2301/AE 51/10122 AE 501— FLIGHT DYNAMICS

(Regulation 2008/2010)

Time : Three hours Maximum : 100 marks

Answer ALL questions.

PART A — (10 × 2 = 20 marks)

1. Why is the Newton's Law of Inertia valid only in Inertial Frame of Reference?

2. Common aerodynamic practice is to work with non-dimensional forms of the lift and drag, how do you non-dimensionalize these parameters?

3. As the altitude increases, temperature decreases and its get cold at higher altitude, why?

4. What makes the difference in coefficient of lift in infinite wing and finite wing?

5. What are the conditions for maximum range of a jet powered airplane?

6. Explain briefly the term rate one turn and coordinated turn.

7. Why the propellers have not been used for transonic and supersonic airplanes?

8. What is the criterion for static longitudinal stability?

9. Explain briefly why a body-fixed reference frame is more appropriate for the analysis of flight dynamics rather than an Earth fixed reference frame.

10. Define static margin and neutral point.

PART B — (5 × 16 = 80 marks)

11. (a) Consider an aircraft with following features :

Wing Loading = 700 kg/m^2 ; Wing Span = 29 m; Wing area = 70 m^2; MTOW 19950 kg; Thrust to Weight Ratio = 0.2; Max. Operating altitude = 4.6 km (density ratio = 0.6295); Power plant: Two Turboprops, each flat rated at 1864 kW (2500 Shaft Horse Power) at sea level. For cruise condition it can be assumed Maximum lift-to-drag ratio = 16; Oswald Efficiency factor = 0.85; Propeller Efficiency = 0.8.

(i) Calculate the aspect ratio, AR

(ii) Calculate the drag co-efficient at zero lift

(iii) Calculate the aircrafts equivalent power at sea level conditions. Assume the ratio of jet thrust to propeller thrust to be 0.15.

(iv) Calculate the equivalent power at maximum operating altitude.

(v) Find the minimum thrust required for straight and level flight and the corresponding airspeeds at sea level.

(vi) Find the minimum power required and the corresponding true airspeeds for straight and level flight at sea-level.

Or

(b) Find the time to climb from sea-level to 20,000 feet for a jet aircraft using a "maximum rate of climb" schedule. The aircraft parameters are:

W = 12000 lbs; C_D = 0.02 + 0.05C_L^2 ; S = 230 ft^2; and the thrust at sea level is 2000 lbs. Density at sea level is 0.002377 slug/ft^3 and Density at 20,000 feet is 0.001267 slug/ft^3. Lift coefficient corresponding to maximum rate of climb at sea level and 20,000 feet are 0.280 and 0.462 respectively. Calculate the service ceiling where the maximum rate of climb is 500 feet per minute.

12\. (a) (i) An aircraft is equipped with a wing of symmetrical airfoils. The lift curve slope of the total aircraft is estimated to $\partial C_L / \partial \alpha = 0.8 * 2\pi(1/\text{rad})$. The stall angle of attack (AOA) is 12°. Wing area is 20 m^2. Use g = g_0 and $\rho = \rho_0$. What is the aircraft's mass during a flight on which a stall speed of 35.72 m/s was observed?

(ii) Calculate the flattest possible gliding angle of an aircraft having maximum lift to drag ratio 8 when the power is completely cut-off. Calculate the horizontal distance it can cover if it is at a height of 1800 m. If the gliding speed is 120 m/s, calculate the sinking speed.

(iii) While flying at an indicated airspeed of 135 km/hr at 3000 m altitude (rho = 0.9096 kg/m^3), an aircraft weighing 6000 kg makes a complete turn (steady horizontal) in one minute. If the wing loading is 130 kg/m^2 and lift to drag ratio during turn is 15, calculate the radius of turn and thrust horsepower required.

Or

(b) Consider an aircraft with the following features :

W = 6,00,000 lbs; maximum thrust = 180,000 lbs ; S = 5128 ft^2 ; C_{DO} = 0.017; K = 0.042 ; Maximum list coefficient = 2.2 ; TSFC = 0.85 (lbs/hr)lb ; Fuel weight = 180,000 lbs. Find the range flown under maximum range flight conditions, assuming the flight starts at 20,000 feet (0.001267 slug/ft^3), for the case where

(i) h = constant; angle of attack = constant

(ii) V = constant; angle of attack = constant and endurance for a constant angle of attack flight schedule and for a constant speed schedule.

13. (a) Consider the aircraft with the following features :

Gross weight = 10,000 lbs; Wing span = 40 feet; Maximum lift coefficient = 1.8 (normal flight) and 2.4 (with flaps); Wing area = 200 fl^2; C_D = 0.02 + 0.05C_L^2; Thrust = 1000 lbs/engine (two engines); $K_{uc} = 5.81 * 10^{-5}$(no flaps) and $3.16 * 10^{-5}$ (full flaps).

(i) During ground roll, the wing is 5 feet above the ground. The coefficient of friction on the runway is 0.02. Calculate the takeoff ground roll distance assuming no flaps are used for takeoff and that minimum takeoff distance conditions are used during the ground roll.

(ii) How would the takeoff distance equation be modified if the runway were on a slope? Determine the take-off distance for the case where the runway is sloped upward 7 degrees in the direction of takeoff. What is the % increase over the normal takeoff distance?

Or

(b) (i) An aircraft model is tested in a low-speed wind tunnel at an angle of attack of 20 degree, sideslip of 10 degree, and a bank angle of 10 degree. An internal strain gauge balance was used to measure the aerodynamic forces acting on the model, which gives components of force in the body axes system. The measurements are axial force, F_x = 10 kg, side force, F_y = -15 kg and normal force, F_z = – 41 kg. Determine the lift, drag and side forces acting on the model w.r.t wind axes system.

(ii) Obtain the characteristics equation and its roots for the disturbed motion, given that : $u_0 = 53.64\, m/s$, $Z_w = -1.978 S^{-1}$, $M_w = -0.1602\, m^{-1} S^{-1}$, $M_{\dot{w}} = -0.0167\, m^{-1}$, $M_q = 2.006 S^{-1}$

14. (a) Consider an aircraft with the following features :

W = 12232.6 N ; Velocity = 53.64 m/s; Wings Area (S) = 17.09 m^2 ; Span (b) = 10.18 m ; $C_{root} = 1.98$ m; M.A.C = 1.737 m; $i_W = 1$ deg; $C_{mac} = -0.116$; $C_{1\alpha W} = 0.097$ deg^{-1}; $\alpha_{0LW} = -6°$; a.c. locations = 0.25 $\overline{C}$. Fuselage : The portion of the fuselage ahead of the root chord is divided into four equidistant portions each of length 0.4573 m. The portion of fuselage aft of the root chord is divided into fve equidistant sections rach of length 0.8841 m. The value of $(K_2 - K_1) = 0.82$. The values of $(d\varepsilon / d\alpha)$ for the portions ahead of the root chord are 1.20, 1.34, 1.56, and 3.20. Length = 8.23 m; Width of fuselage at maximum cross section = 1.4 m; Height of fuselage at maximum cross section = 1.6 m; $l_h = 3.17$ m .

C_{m0} $W_f(m)$	1.098	1.402	1.402	1.402	1.25	0.945	0.701	0.457	0.244
C_{ma} $W_f(m)$	0.915	1.036	1.159	1.280	1.158	0.945	0.701	0.457	0.243

Obtain the contribution of fuselage to the moment about C.G.

Or

(b) Consider a model of a wing-body shape mounted in a wind tunnel. The flow conditions in the test section standard sea-level properties with a velocity of 100 m/s. The wing are and chord are 1.5 m^2 and 0.45 m respectively. The distance from the airplane's center of gravity to the tail's aerodynamic center is 1 m. The area of the tail is 0.4 m^2, and the tail-setting angle is –2.0 degree. The lift slope of the tail is 0.12 per degree. The moment about the center of gravity when the lift is zero is found to be –12.4 N-m. From experimental measurement $\varepsilon_0 = 0$ and $d\varepsilon / d\alpha = 0.42$. If the absolute angle of attack of the model is 5 degree and the lift at this angle of attack is 4134 N, calculate the moment about the center of gravity when $(x_{cg}/\bar{c}) - (x_{ac,W}/\bar{c}) = 0.02$.

15. (a) From the given following equations :

$$(13.78\ s) + 0.088\ u(s) - 0.392\ \alpha(s) + 0.740\ (s) = 0$$

$$1.48\ u(s) + \_\ (13.78\ s + 4.46)\,\alpha(s) - 13.78s\ \theta(s) = -0.246\ \delta_e(s)$$

$$(0.0552s + 0.619)\ \alpha(s) + (0.514s^2 + 0.192s)\,\theta(s) = -0.710\,\delta_e(s)$$

Find the transfer function of $\frac{u(s)}{\delta_e(s)};\frac{\alpha(s)}{\delta_e(s)}$.

Or

(b) The 3-DOF longitudinal transfer functions are approximated by

$$\frac{u(s)}{\delta_e(s)} = \frac{6.312s^2 - 4927s - 4302}{s^4 + 2.0284s^3 + 8.0766s^2 + 0.1277s + 0.06625}$$

$$\frac{\alpha(s)}{\delta_e(s)} = \frac{0.746s^3 + 208.3s^2 + 2.665s + 1.39}{s^4 + 2.0284s^3 + 8.0766s^2 + 0.1277s + 0.06625}$$

$$\frac{\theta(s)}{\delta_e(s)} = \frac{205.1s^2 + 136.9s + 2.380}{s^4 + 2.024s^3 + 8.0766s^2 + 0.1277s + 0.06625}$$

Find the natural frequency, damping ratio, damped frequency, time constant and period of oscillation for the short period and phugoid modes. Use the short and phugoid 2-DOF approximations to estimate the natural frequency, damping ratio, damped frequency and period of oscillator. Compare the approximation results.

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