Hall Ticket No | | | | | | | | | | Question Paper Code: AAEB07

INSTITUTE OF AERONAUTICAL ENGINEERING

(Autonomous)

B.Tech IV Semester End Examinations (Regular/Supplementary) - July, 2021

Regulation: R18

AEROSPACE STRUCTURES

Time: 3 Hours **(AE)** **Max Marks: 70**

Answer FIVE Questions choosing ONE question from each module
(NOTE: Provision is given to answer TWO questions from any ONE module)
All Questions Carry Equal Marks
All parts of the question must be answered in one place only

MODULE – I

1. (a) Derive the equation to find out deflection and slope of cantilever beam with udl by using castiglianos theorem. **[7M]**

   (b) A joint in a fuselage skin is constructed by riveting the abutting skins between two straps as shown in Figure 1. below. The fuselage skins are 2.5 mm thick and the straps are each 1.2 mm thick; the rivets have a diameter of 4 mm. If the tensile stress in the fuselage skin must not exceed 125 N/mm^2 and the shear stress in the rivets is limited to 120 N/mm^2 determine the maximum allowable rivet spacing such that the joint is equally strong in shear and tension. **[7M]**

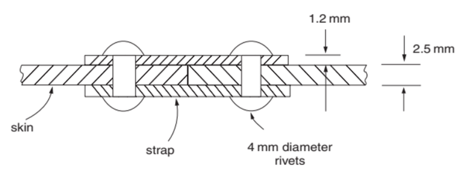

Figure 1

2. (a) Find out the vertical displacement of simply supported beam with point load at mid-point by using total potential energy method. **[7M]**

   (b) Figure 2 shows the flight envelope at sea-level for an aircraft of wing span 27.5 m, average wing chord 3.05 m and total weight 196 000 N. The aerodynamic centre is 0.915 m forward of the CG and the centre of lift for the tail unit is 16.7 m aft of the CG. The pitching moment coefficient is $C_{M,0}$ = -0.0638 (nose-up positive) both $C_{M,0}$ and the position of the aerodynamic centre are specified for the complete aircraft less tail unit. For steady cruising flight at sea-level the fuselage bending moment at the CG is 600 000 Nm. Calculate the maximum value of this bending moment for the given flight envelope. For this purpose it may be assumed that the aerodynamic loadings on the. fuselage itself can be neglected, i.e. the only loads on the fuselage structure aft of the CG are those due to the tail lift and the inertia of the fuselage. **[7M]**

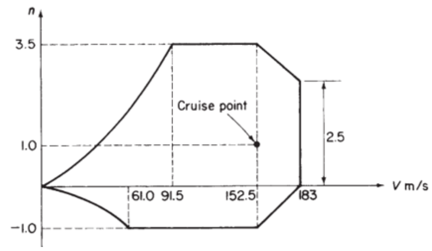

Figure 2

MODULE – II

3. (a) Derive the equilibrium equation of the plate under bending with suitable diagrams. [7M] **[7M]**

   (b) Draw the free body diagram of plate subjected to the uniform distributed load q_0. Also, clearly show the stress resultant and stress couples that are acting in plate. **[7M]**

4. (a) Determine the critical buckling stress of thin rectangular plate of dimensions axb loaded in axial direction (X direction). **[7M]**

   (b) A plate 10 mm thick is subjected to bending moments M_x equal to 10 N-mm and M_y equal to 5 N-mm. Calculate the maximum direct stresses in the plate.Take E = 20,000 N/mm^2 Poison's ratio = 0.3 **[7M]**

MODULE – III

5. (a) Differentiate between shear center and centroid of the cross-section. State the physical significance of shear centre in aircraft design. **[7M]**

   (b) Derive the Bredt Batho torsion formula for a thin walled closed section. Draw the required skecthes clearly. **[7M]**

6. (a) Write the differences between symmetric and unsymmetric beam bending with suitable equations. Support your answer with suitable figures. **[7M]**

   (b) For a T- section shown in Figure 3, determine the maximum bending stress value and its position, if the beam is subjeted to moment, M_x and M_y of magnitude 100 and 120 KN mm respectively. **[7M]**

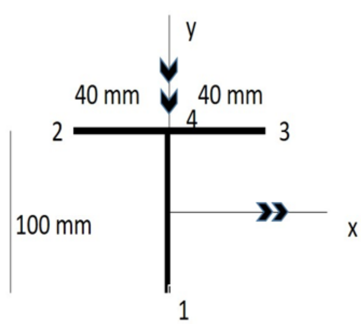

Figure 3

MODULE– IV

7. (a) Draw the actual and idealized panels of a fuselage section and explain how shear flow is distributes when vertical shear is applied **[7M]**

   (b) Determine the shear flow distribution in the idelaised multi-cell section shown in Figure 4, subjected to pure torque = 100KNmm. The thickenss of the section, t = 2 mm, is same throughout. Take G = 27 GPa, Area of each boom = 100 mm^2. **[7M]**

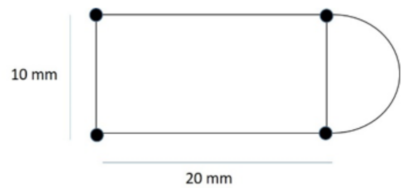

Figure 4

8. (a) Derive the equation to find out the bending stress of idealized panel of thickness t. **[7M]**

   (b) Find the angle of twist per unit length in the wing whose cross-section is shown in Figure 5, when it is subjected to a torque of 10 kN m. Find also the maximum shear stress in the section. G = 25 000 N/mm^2. Wall 12 (outer) = 900 mm. Nose cell area = 20 000 mm^2. Note: Assume that the torsional rigidity GJ of the complete section is the sum of the torsional rigidities of the open and closed portions. **[7M]**

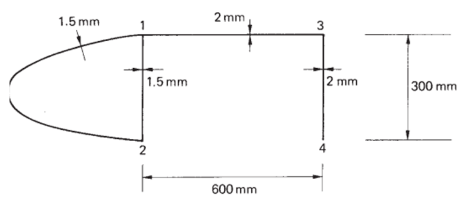

Figure 5

MODULE – V

9. (a) Explain the functions of fuselage frames and wing ribs. **[7M]**

   (b) A wing spar has the dimensions shown in Figure 6 and carries a uniformly distributed load of 15kN/m along its complete length. Each flange has a cross sectional area of 500mm^2 with the top flange being horizontal. If the flanges are assumed to resist all direct loads while the spar web is effective only in shear, determine the flange loads and the shear flows in the web at sections 1 and 2m from the free end. **[7M]**

10. (a) Derive shear flow distribution on wing section with neat sketch. **[7M]**

   (b) The doubly symmetrical fuselage section shown in Figure 7 has been idealized into an arrangement of direct stress carrying booms and shear stress carrying skin panels; all the boom areas are 150 mm^2. Calculate the direct stresses in the booms and the shear flows in the panels when the section is subjected to a shear load of 50 kN and a bending moment of 100 kNm. **[7M]** **[7M]**

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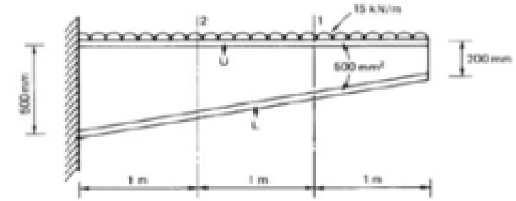

Figure 6

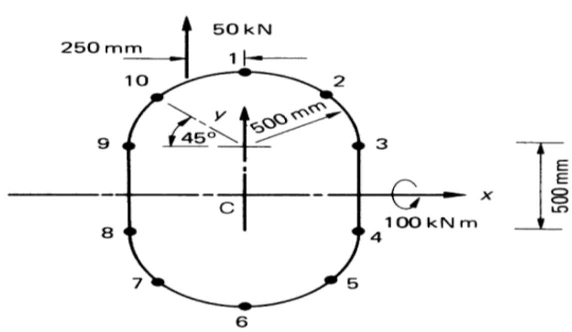

Figure 7