Hall Ticket No | | | | | | | | | | Question Paper Code: AAEC08

INSTITUTE OF AERONAUTICAL ENGINEERING

(Autonomous)

Dundigal-500043, Hyderabad

B.Tech IV SEMESTER END EXAMINATIONS (REGULAR / SUPPLEMENTARY) - AUGUST 2023

Regulation: UG-20

AERODYNAMICS

Time: 3 Hours **(AERONAUTICAL ENGINEERING)** **Max Marks: 70**

Answer ALL questions in Module I and II
Answer ONE out of two questions in Modules III, IV and V
All Questions Carry Equal Marks
All parts of the question must be answered in one place only

MODULE – I

1. (a) Formulate expressions for stream function and velocity potential for a source flow.
[BL: Understand| CO: 1|Marks: 7]

   (b) Derive Kutta - Joukowski theorem and prove that lift is directly proportional to circulation.
[BL: Understand| CO: 1|Marks: 7]

MODULE – II

2. (a) Draw a neat sketch of airfoil and mention its nomenclature. Describe the stalling of an airfoil and the related aerodynamic phenomena [BL: Understand| CO: 2|Marks: 7]

   (b) Consider an NACA 2412 airfoil with a chord of 0.64 m in an airstream at standard sea level conditions. The freestream velocity is 80 m/s. The lift per unit span is 1264 N/m. Calculate the strength of the steady-state starting vortex. [BL: Apply| CO: 2|Marks: 7]

MODULE – III

3. (a) Illustrate the flow past finite wings and explain how are the wing tip vortices created?
[BL: Understand| CO: 4|Marks: 7]

   (b) A finite wing with an aspect ratio of 10 and a taper ratio of 0.8has a symmetric airfoil section. Calculate the lift and induced drag coefficients for the wing when it is at an angle of attack of 5 degree . Assume that $\delta = \tau = 0.055$. [BL: Apply| CO: 4|Marks: 7]

4. (a) Derive the expression for induced velocity induced by an infinite long vortex filament at any arbitrary point located at a distance 'r' from the vortex filament.
[BL: Understand| CO: 3|Marks: 7]

   (b) Consider an NACA 23012 airfoil. The mean camber line for this airfoil is given by $z/c = 2.6595[(x/c)^3 – 0.6075(x/c)^2 + 0.1147(x/c)]$ for $0 < x/c < 0.2025$ and z/c = 0.02208 {1 – (x/c)} for 0.2025 < x/c < 1.0.

   i) Calculate the angle of attack at zero lift

   ii) Calculate the lift coefficient when α= 8 degrees

   iii) Calculate the moment coefficient about the quarter chord when α= 8 degrees and

   iv) Calculate the location of the centre of pressure in terms of x_{cp}/c, when α= 8 degrees
[BL: Apply| CO: 3|Marks: 7]

MODULE – IV

5. (a) Illustrate the position of circle for tranformation of circle into flat plate, ellipse, circular arc, symmetrical and cambered airfoil in Kutta- Joukowski transformation

[BL: Understand| CO: 5|Marks: 7]

(b) Using Kutta - Joukowski transformation, transform a circle into a symmertrical airfoil.

[BL: Apply| CO: 5|Marks: 7]

6. (a) Formulate the equation for chord of circular arc airfoil using Kutta- Joukowski transformation.

[BL: Understand| CO: 5|Marks: 7]

(b) Show that transformation of an uinform flow parallel to X - axis usiong conformal transformation function $\zeta = z^2$ will result in a parabola. [BL: Apply| CO: 5|Marks: 7]

MODULE – V

7. (a) Summarize about displacement thickness and energy thickness and determine an expression for displacement thickness. [BL: Understand| CO: 6|Marks: 7]

(b) Consider a flat plate at zero angle of attack in an airflow at standard sea level conditions (p = 0.101 MPa and T = 288 K). The chord length of the plate is 2m. The planform area of the plate is $40m^2$. At standard sea level conditions, $\mu = 1.7894 \times 10^{-5}$ kg/(m - s). Assuming the wall temperature follows the adiabatic wall temperature law, calculate the friction drag on the plate when the freestream velocity is 1100 m/s. [BL: Apply| CO: 6|Marks: 7]

8. (a) Explain with neat sketch different regions of boundary layer for a fluid flowing over a horizontal flat plate. [BL: Understand| CO: 6|Marks: 7]

(b) Consider a high-speed vehicle flying at a standard altitude of 35 km, where the ambient pressure and temperature are 583.59 N/m^2 and 246.1 K, respectively. The radius of the spherical nose of the vehicle is 2.54 cm. Assume the Prandtl number for air at these conditions is 0.72, that cp is 1000 J/(kg K), and that the viscosity coefficient is given by Sutherland's law. The wall temperature at the nose is 400 K. The recovery factor at the nose is 1.0. Calculate the aerodynamic heat transfer to the stagnation point for the flight velocities of i) 2000 m/s ii) 4000 m/s. From these results, comment on the heat transfer variation with flight velocity.

[BL: Apply| CO: 6|Marks: 7]

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