Code No: 07A42101 **R07** **Set No. 2**

II B.Tech II Semester Examinations,December 2010
AERODYNAMICS - I
Aeronautical Engineering

Time: 3 hours **Max Marks: 80**

Answer any FIVE Questions
All Questions carry equal marks

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1. Explain with neat sketch potential flow over a circular cylinder. [16]

2. Explain in detail how we can replace a finite wing by a bound vortex along with expressions and also with neat sketches. [16]

3. Write short notes on:
   (a) Stream function.
   (b) Velocity potential.
   (c) Complex numbers
   (d) Complex Potential Function. [4+4+4+4]

4. Derive how vortex panel method is used for expressing the kutta condition for panels immediately above and below the trailing edge. [16]

5. (a) At a given point on the wing of an aircraft, the pressure and temperature of the air are 1.9×10^4 pascals and 203^o K respectively. Calculate the density at this point. The gas constant may be assumed to be 287 J / (kg ^{o}K).
   (b) At a point in the test section of a supersonic wind tunnel, the pressure and density of the air are 4×10^4 PASCAL and 1.01 kg / m^3, respectively. Calculate the temperature at this point. [8+8]

6. (a) Derive the moment coefficient about the leading edge for a cambered airfoil.
   (b) Derive the expression for the distance of the centre of pressure from the leading edge of a cambered airfoil. [12+4]

7. Consider the combination of a uniform flow with a source and a sink, and obtain the expression for the stream lines. What is the significance of the dividing stream line? [16]

8. Consider a planar wing of aspect ratio 5, taper ratio unity, and swept aft by 45^o in the plane of symmetry. Develop the Vortex Lattice Method to calculate lift coefficient for this wing. Take the uniform chord of the wing as c = 1.0 unit. Divide the wing into 4 panels. [16]

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Code No: 07A42101 **R07** **Set No. 4**

II B.Tech II Semester Examinations,December 2010
AERODYNAMICS - I
Aeronautical Engineering

Time: 3 hours **Max Marks: 80**

Answer any FIVE Questions
All Questions carry equal marks

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1. Calculate the pressure coefficient distribution around a circular cylinder using the source panel technique. Make use of sketches in this regard. [16]

2. Derive and explain
   (a) Continuity equation
   (b) local derivative and substantial derivative. [8+8]

3. (a) Derive the moment coefficient about the leading edge for a cambered airfoil.
   (b) Also, derive the moment coefficient about quarter chord point for a cambered airfoil. [12+4]

4. (a) Lifting surface theory predicts better lift distribution on a wing with a low aspect ratio and of any type of given planform'. Can you demonstrate the verification of the statement?
   (b) Compare the formulation in (a) above with that in the classical lifting line theory with details. [8+8]

5. Consider a velocity field where the x and y components of velocity are given by u = cx and v = -cy, where c is a constant. Assume the flow to be incompressible, calculate the steam function and velocity potential. Using the results, show that lines of constant ϕ are perpendicular lines of constant ψ. [16]

6. A constant strength vortex panel of strength 60 units is located on the X-axis from X1=2.5 to X2=4.65. Determine the influence of this vortex panel at a point P(5.5,5.5) to evaluate V(u,w). Develop the expressions used for determining
   (a) Velocity Potential
   (b) Velocity Components. [8+8]

7. How is downwash produced on a finite wing? Explain its effect on the wing. [16]

8. (a) Explain using neat sketches, the 4 digit, 5 digit and 6 digit series of NACA profiles and the corresponding aerodynamic coefficients.
   (b) Describe the different wing geometries and the reasons for the differences. [8+8]

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Code No: 07A42101 **R07** **Set No. 1**

II B.Tech II Semester Examinations,December 2010
AERODYNAMICS - I
Aeronautical Engineering

Time: 3 hours Max Marks: 80

Answer any FIVE Questions
All Questions carry equal marks

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1. Explain in detail about the non lifting flow over a circular cylinder and also derive expression for pressure coefficient. Make use of sketches in this regard. [16]

2. What is effective aspect ratio? Why does the effective angle of attack changes at the local airfoil sections of a wing? Explain induced drag. [16]

3. Sketch a swept back wing showing sweep, taper and dihedral. Explain the utility of these features. Draw neat sketches. [16]

4. Derive the fundamental equation of thin airfoil theory, $(1/2\pi)\int[\{\gamma(\xi)d\xi\}/\{x-\xi\}] = V\{\alpha-(dz/dx)\}$, where the integration is carried out from the leading edge to the trailing edge of an airfoil and prove that the lift coefficient is proportional to angle of attack for a cambered airfoil. [16]

5. (a) State and explain Kutta - Joukowski theorem
   (b) State and explain Kelvins circulation theorem. [8+8]

6. Consider the lifting flow over a circular cylinder. The lift coefficient is 3. Calculate the peak pressure coefficient and also calculate the location of the stagnation points and the points on the cylinder where pressure equals freestream static pressure.[16]

7. Consider a plane wing of AR 5, taper ratio 04 and swept back by 45^o. Develop the vortex lattice method to calculate lift coefficient for this wing. Take the root chord of the wing as C=1.0 unit. Divide the wing into 4 panels. [16]

8. Explain Kutta - Zukovsky transformation explain how a circle can be transformed into a straight line, ellipse, a symmetric airfoil and a cambered airfoil. Explain with details. [16]

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Code No: 07A42101 **R07** **Set No. 3**

II B.Tech II Semester Examinations,December 2010
AERODYNAMICS - I
Aeronautical Engineering

Time: 3 hours **Max Marks: 80**

Answer any FIVE Questions
All Questions carry equal marks

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1. Explain the source panel method used for non-lifting flows. [16]

2. (a) An airfoil is kept at 5 degrees angle of attack in a flow. The lift and drag coefficients are 3.0 and 0.2 respectively. Find the normal and axial forces.
   (b) The normal force is acting at the quarter chord point. Find the moment on the airfoil at the leading edge of the airfoil. [8+8]

3. Explain the vortex panel method for lifting glows. [16]

4. Using the appropriate combination of elementary Flows, show that the pressure variation over a circular cylinder is given by C_{0p} = (1- 4 $\sin^2 \theta$). [16]

5. Derive the fundamental equation of thin airfoil theory,
   $(1/2\pi) \int [\{\gamma(\xi) d\xi\}/\{x - \xi\}] = V\{\alpha - (dz/dx)\}$, where the integration is carried out from the leading edge to the trailing edge of a symmetrical airfoil and prove that the lift coefficient is proportional to angle of attack for a symmetrical airfoil. [16]

6. When a finite wing is replaced by a single horseshoe vortex, it is known that down-wash at the tips become infinite. As we know this is unrealistic. Suggest and explain the remedy for this problem? Also obtain the expression for down wash. [16]

7. (a) Show that the transformation
   $\zeta = z + (b^2/z) = \xi + i\eta$ leads to $\zeta = (r + b^2/r)\cos\theta + i(r - b^2/r)\sin\theta$, where z is complex while ξ and η are real.
   (b) Hence show that $\zeta = f(z)$ transforms a circle a symmetrical airfoil if the origin of the circle is a (be, h) where b, h and e are constants. [6+10]

8. Write the advantages and disadvantages of
   (a) Extended lifting line theory
   (b) Lifting surface theory. [16]

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