AEROSPACE ENGINEERING DEPARTMENT

INDIAN INSTITUTE OF TECHNOLOGY, KHARAGPUR

Mid-Spring Semester (2008-2009) Examination
Time: 2 Hrs.

02.03.2009 AN **AE21002 Incompressible Aerodynamics**

2nd Year B. Tech. & Dual Degree and Breadth **No of Students: 36**

Answer as many questions as you can.

Assumptions, if required, can be made with appropriate justifications.

Some important relations, which may be required:

$$\int_0^{\pi} \frac{\cos n\theta_0}{\cos\theta_0 - \cos\theta} d\theta_0 = \pi \frac{\sin n\theta}{\sin\theta}, \qquad \int_0^{c} \frac{dx_0}{x - x_0} = \ln\frac{x}{c-x}$$

Notations have their usual meaning unless specified otherwise.

1. Evaluate the zero-normal flow boundary condition for a pulsating sphere whose radius is varying such that $r(t) = a_0 + a_1 \sin\omega t$ in a fluid at rest at infinity.

   A clockwise vortex segment of strength 2.5 m^2/s is placed between the points (-1, 0, 0) and (0, 1, 1). Compute the induced velocity at the point (1, 2, 1). **Marks: 4+6**

2. A uniform stream of speed q_∞ flows over a thin aerofoil set at an angle of attack α. The camber line of the aerofoil is given by $\eta_c(x) = 4\varepsilon\frac{x}{c}\left[1 - \frac{x}{c}\right]$ with ε being its maximum height. Compute the pressure difference coefficient in terms of normalized distance from the leading edge. Also find the shift in centre of pressure as the angle of attack changes from 0° to 5°. **Marks: 8**

3. Using linearized theory calculate the pressure distribution on a symmetric aerofoil described by $\frac{\eta_t}{c} = \pm\frac{2\sqrt{3}}{9}\tau\left(\sin\theta + \frac{1}{2}\sin 2\theta\right)$, where $\frac{x}{c} = \frac{1}{2}(1 - \cos\theta)$. Express the pressure distribution in terms of $\frac{x}{c}$. Determine the trailing edge angle of the aerofoil. **Marks: 6**

4. A source distribution over a surface creates a jump in the normal velocity across the surface. If the strength of the source distribution is $\sigma(x, y)$ per unit area, find the jump in normal velocity. **Marks: 4**

5. Consider the flow past a flat elliptic planform wing at angle of attack α. A flap that covers the centre half of the wing-span is deflected such that the zero-lift angle

distribution along the span is given by $\alpha_0 = -\beta$ for $-\frac{b}{4} < y < \frac{b}{4}$ and $\alpha_0 = 0$ for the remaining part. The parameter β is a constant. Using lifting-line theory, calculate the spanwise circulation distribution, wing lift coefficient and rolling moment coefficient.

Marks: 10

6. State the characteristics of elliptic loading. Does elliptic planform ensure elliptic loading? Is it possible to have elliptic loading if the planform is not elliptic?

A uniform stream of speed q_∞ flows over a straight delta wing of aspect ratio 1.0 with root chord of 1.25 m. The wing is set at an angle of attack of 2.5°. Assume the wing and its wake are modeled by a vorticity distribution wherein the bound circulation at a station x from the leading edge is given as $\Gamma(y) = q_\infty b\alpha\sqrt{\left[1-\left(\frac{2y}{b}\right)^2\right]}$ where b is the local span and α is the angle of attack. Locate the centre of pressure and aerodynamic centre of the wing. Calculate the circulation distribution in the wake. **Marks: 4+8**