INDIAN INSTITUE OF TECHNOLOGY, KHARAGPUR
Date of exam. : 20-02-2013 Time: **2 Hrs** Full marks: **30** No. of Students : 55
Spring Mid-Sem.,2012-2013 Dept of Aerospace Engineering 2^{nd} Yr. B.Tech (Hons.)
Sub. No: **AE21002** Subject Name: **Low Speed Aerodynamics**

Instructions: Answer **ALL THREE** Questions. Marks are indicated

Q1 (a) Establish the equivalence between the distributed vortex system as given by thin aerofoil theory and a single lumped vortex having the same circulation.

(b) Describe 2-D vortex lattice method (VLM) for flow past thin chambered aerofoil at small angle of attack α. Express the problem in non-dimensional form.

(c) Using only two elements of equal lengths find the values of C_L and $C_{M_{LE}}$ for a thin symmetric aerofoil as given by 2-D VLM. Compare these values with those obtained by thin aerofoil theory. **Marks (2+4+4)**

Q2 (a) In the element co-ordinate system of j-th element find the u_s^* & v_s^* components of velocity at the i-th control point due to uniform source distribution of unit strength on the j-th element.

(b) Find similar velocities u_v^* & v_v^* due to uniform vorticity distribution of unit strength on j-th element or panel. .

(c) Transfer these velocity components (obtained in parts a and b) to the global co-ordinate system

(d) Show that in Hess and Smith Panel method (based on source and vorticity singularity distributions) for flow past an aerofoil at an angle of incidence α, the boundary condition of zero normal flow at the i-th control point can be written as

$$\sum_{j=1}^{N} A_{ij}\sigma_j + A_{i,N+1}\gamma = b_i \qquad \text{for } i=1,2 \quad ,N$$

and that Kutta condition used in the method also reduces to a similar expression..

$$\sum_{j=1}^{N} A_{N+1,j}\sigma_j + A_{N+1,N+1}\gamma = b_{N+1}$$

Marks: 3x2+4

Q3 (a) For a general thin-cambered aerofoil at small angle of incidence α, find the expressions for C_L and $C_{M_{LE}}$

(b) Define centre of pressure for an aerofoil and find its location for a thin cambered aerofoil at small angle of incidence. Where will the centre of pressure be located if the aerofoil produces zero lift for a particular incidence and why?

(c) Define aerodynamic center of an aerofoil. Find its location as per thin aerofoil theory and also find pitching moment about this point.

(d) If the camber line of a thin aerofoil of chord 'c' is given by

$$y = -kx\{1 - a(x/c) + b(x/c)^2\},$$

find the values of the coefficients a and b such that the pitching moment about the aerodynamic center is zero. **Marks=3x2+4**