B.E. (Aeronautical Engineering) Fourth Semester (C.B.S.)

# Aerodynamics - I

P. Pages : 2 **TKN/KS/16/7405**

Time : Three Hours Max. Marks : 80

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**1.** a) What is an airfoil? Explain airfoil Geometry parameters with well lable diagram. **6**

b) An aircraft is flying at an altitude of 11.5km with a speed of 655 km/hr. Calculate its equivalent air speed. **5**

c) Write down the difference between symmetrical and unsymmetrical Airfoil. **3**

**OR**

**2.** a) Draw the diagram of all types of wing planform. **3**

b) Explain the pressure distribution over unsymmetrical airfoil. **8**

c) An aircraft is flying at sea level with a velocity of 450 km/hr. The parameter of wing are, wing span = 20m average chord = 1.5m, $C_\ell = 1.056$, $C_d = 0.785$ Calculate all the forces acting on the aircraft. **3**

**OR**

**3.** a) Define following **4**
Laminar flow, turbulent flow, steady flow unifrom flow, compressible flow, rotational flow, potential function & stream function.

b) What is source-sink pair flow? Find out the expression for stream function and velocity potential function for source-sink pair. Also draw the flow pattern for source-sink pair. **9**

**4.** a) Explain in brief Kutta Joukowski's theorem. **8**

b) What is vortex flow? Derive the expression for stream function & velocity function for vortex flow. **5**

**5.** a) Write all the assumptions involved in thin airfoil theory. **5**

b) Derive the governing equation for thin airfoil theory. **8**

**OR**

**6.** Consider an NACA 23012 airfoil the mean camber line for this airfoil is given by **13**

$$\frac{z}{c} = 2.6595\left[\left(\frac{x}{c}\right)^3 - 0.6075\left(\frac{x}{c}\right)^2 + 0.1147\left(\frac{x}{c}\right)\right] \text{ for } 0 \le \frac{x}{c} \le 0.2025$$

$$\frac{z}{c} = 0.02208\left(1 - \frac{x}{c}\right), \text{ for } 0.2025 \le \frac{x}{c} \le 1$$

Calculate:

i) Angle of attack of zero lift.

ii) The lift co-efficient when $\alpha = 4^\circ$

iii) The moment co-efficient about the quarter chord.

iv) The location of the center of pressure in terms of $\frac{XoP}{C}$ when $\alpha = 4^\circ$

**7.** a) Consider a point in an airflow where the local Mach number, static pressure and static temperature are 3.5, 0.3 atm and 1800k, respectively calculate the local values of $P_o$, $T_o$, $T^*$, $a^*$, $p^*$ and $M^*$ at this point. **7**

b) What do you mean by stagnation state. Derive the relation between stagnation temperature, stagnation pressure and stagnation density. **7**

**OR**

**8.** a) The static and stagnation temperature of a stream of air are 15ºC and 50ºC respectively. **7**

i) Estimate the Mach number and flow velocity.

ii) What would be the percentage rise in pressure between stagnation and static values if the compression process is assumed to be reversible adiabatic.

b) A supersonic nozzle is to be designed for flow with Mach number 3 at the exit section which is 20cm in diameter. The pressure and temperature of air at the nozzle exit are to be 8kPa and 200k respectively. Determine the reservoir pressure, the temp. and the throat area. **7**

**9.** a) Consider a normal shock wave in a supersonic airstream where the pressure upstream of the shock is 1atm. Calculate the loss of total pressure across the shock wave when the upstream Mach number is **8**

i) $M_1 = 2$ ii) $M_1 = 4$

Compare these two results and comment on their implication.

b) Derive the Prandtl relation. **5**

**OR**

**10.** a) A supersonic airplane is flying at Mach 2 at an altitude of 16km. Assume the shock wave pattern from the airplane quickly coalesces into a Mach wave that intersects the ground behind the airplane causing a sonic boom to be heard by a bystander on the ground. At the instant the sonic boom is heard how far ahead of the bystander is the airplane? **4**

b) Consider a supersonic flow with M = 2, P = 1, atm and T = 288k. This flow is deflected at a compression corner through 20º. Calculate, M. P, T behind the resulting oblique shock wave. **9**

**11.** a) Define energy thickness. Also derive an expression for energy thickness. **5**

b) Examine whether or not the following velocity profile satisfy the essential boundary condition for velocity distribution in laminar Boundary layer on a flat plate. **8**

i) $\frac{u}{U_0} = 1 + \left(\frac{y}{\delta}\right) - 2\left(\frac{y}{\delta}\right)^2$ ii) $\frac{u}{U_0} = \sin\left(\frac{\pi y}{2\delta}\right)$

**OR**

**12.** a) 2 -D, boundary layer along a flat plate is assume to be $\frac{U}{U_0} = \sin\left(\frac{\pi y}{2\delta}\right)$, evaluate the expression for **9**

i) Rate of flow of $\delta$ as a function of x.

ii) $\tau_0$ as function of Reynold's number

iii) Total frictional force on length L and width b.

b) Define shape factor in boundary layer? Calculate the shape factor of boundary layer profile $\frac{u}{U} = \frac{y}{\delta}$. **4**

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