Faculty of Engineering & Technology
Fourth Semester B.E. (Aeronautical Engineering) (C.B.S.) Examination
AERODYNAMICS—I
Paper— 4 AE 05
Sections—A & B

Time—Three Hours] [Maximum Marks—80

INSTRUCTIONS TO CANDIDATES

(1) All questions carry marks as indicated.

(2) Solve Q. 1 **OR** Q. 2 from Section A.

(3) Solve Q. 3 **OR** Q. 4 from Section A.

(4) Solve Q. 5 **OR** Q. 6 from Section A.

(5) Solve Q. 7 **OR** Q. 8 from Section B.

(6) Solve Q. 9 **OR** Q. 10 from Section B.

(7) Solve Q. 11 **OR** Q. 12 from Section B.

(8) Illustrate your answers wherever necessary with the help of neat sketches.

(9) Assume suitable data wherever necessary.

(10) Use of non-programmable calculator is permitted.

(11) Due credit will be given to neatness and adequate dimensions.

SECTION—A

1. (a) Explain in brief the airfoil section nomenclature (any **TWO**):
   (i) Leading Edge Radius and Chord line
   (ii) Mean Camber Line
   (iii) Maximum thickness and thickness distribution. 6

   (b) To calculate the wing geometry parameters for the space shuttle orbiter, the complex shape of the actual wing is replaced by a swept, trapezoidal wing as shown in the figure. For the reference wing of the orbiter, the root chord C_r is 17.5 m, the tip chord C_t is 3.499 m and the span b is 23.79 m. Using these values which define the reference wing. Calculate :
   (i) The wing area (S)
   (ii) Aspect Ratio (AR)
   (iii) Taper ratio (λ)
   (iv) Mean aerodynamic chord.

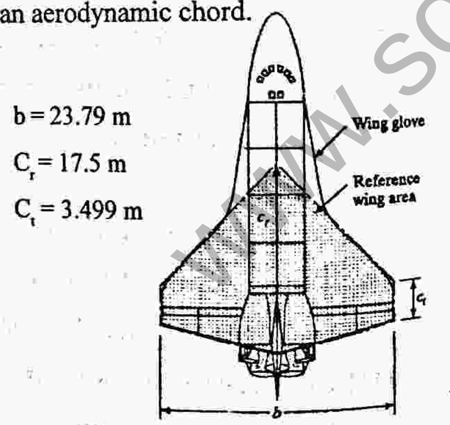

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Fig. 1(b)

OR

2. (a) Derive the relation :

$$C_d = C_n \sin \alpha + C_n \cos \alpha.$$ 6

   (b) Consider the supersonic flow over a 5° half angle wedge at zero angle of attack as shown in following fig. The free stream Mach number ahead of the wedge is 2.0 and the free stream pressure and density are 1.01×10^5 N/m^2 and 1.23 kg/m^3 respectively. The pressure on the upper and lower surfaces of the wedge are constant with distance 's' and equal to each other, namely $P_u = P_l = 1.31 \times 10^5$ N/m^2. The pressure exerted on the base of the wedge is equal to P_∞. The shear stress varies over both upper and lower surfaces as $\tau_w = 431\ s^{-0.2}$. The chord length, C of the wedge is 2m. Calculate drag coefficient for the wedge.

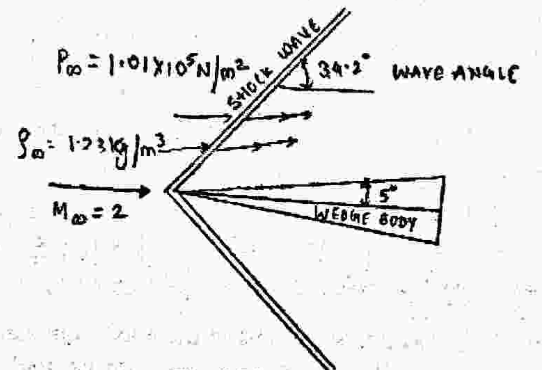

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Fig. 2(b)

3. (a) Sketch the flow pattern of an ideal fluid flow past a cylinder with circulation. 6

(b) Consider the lifting flow over a circular cylinder. The lift coefficient is 5. Calculate the peak (negative) pressure coefficient. 7

OR

4. (a) A source and sink of strength $4m^2/sec$ and $8m^2/sec$ are located at (–1, 0) and (1, 0) respectively. Determine the velocity and stream function at a point P(1, 1) which is lying on the flownet of the resultant streamline. 6

(b) A uniform flow with a velocity of 20 m/sec is flowing over a source of strength $10m^2/sec$. The uniform flow and source flow are in the same plane. Obtain the equation of the dividing streamline. 7

5. (a) Derive the fundamental equation of the thin airfoil theory. 6

(b) In NACA 2412 airfoil, the equation of the mean camber line is

$$\left(\frac{Z}{C}\right)_{fore} = 0.125\left[0.8\left(\frac{x}{C}\right) - \left(\frac{x}{C}\right)^2\right]$$

which is forward of the maximum camber position while aft of the maximum camber position is :

$$\left(\frac{Z}{C}\right)_{aft} = 0.0555\left[0.2 + 0.8\left(\frac{x}{C}\right) - \left(\frac{x}{C}\right)^2\right]$$

Calculate the section lift coefficient. 8

OR

6. (a) Show that the local jump in the tangential velocity across the vortex sheet is equal to the local sheet strength. 8

(b) Consider a NACA 2412 airfoil with a 2-m chord in an airstream with a velocity of 50m/sec at standard sea level conditions. If the lift per unit span is 1353 N. What is the angle of attack ? 6

SECTION—B

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 180 k respectively. Calculate the local values of P_0, T_0, T^*, a^* and M^* at this point. 7

(b) What is meant by normal shock wave ? Enumerate the properties of normal shock waves, across a normal. normal shock wave. 6

OR

8. (a) Consider the isentropic flow through a supersonic nozzle. If the test section conditions are given by : P = 1 atm, T = 230 K and M = 2.

Calculate the reservoir pressure and temperature. 5

(b) Derive the equation which states that the total temperature is constant across a stationary normal shock wave. 8

9. (a) Consider an oblique shock wave with a wave angle of 30° in a Mach 4 flow. The upstream pressure and temperature are 2.65×10^4 N/m^2 and 223.3K respectively (corresponding to a standard altitude of 10000 m). Calculate the pressure, temperature, Mach number, total pressure and total temperature behind the wave and the entropy increase across the wave. 8

(b) A slender missile is flying at Mach 1.5 at low altitude. Assume the wave generated by the nose of the missile is a Mach wave. This wave intersects the ground 559 ft behind the nose. At what altitude is the missile flying ? 6

OR

10. (a) Consider an airfoil in a freestream where $M_\infty = 0.4$ and P_∞ = 1 atm as shown in the figure. At point 1 on the airfoil, the pressure is P_1 which is 0.7545 atm. Calculate the local Mach no. at point 1. Assume isentropic flow over the airfoil. 6

(b) Derive the relation :

$$d\theta = \sqrt{M^2-1}\,\frac{dv}{V}.$$

Also write the implications for it. 8

11. (a) What do you mean by boundary layer separation ? What is the effect of pressure gradient on boundary layer separation ? 6

(b) For the velocity profile for laminar boundary layer

$$\frac{u}{v} = \frac{3}{2}\left(\frac{y}{\delta}\right) - \frac{1}{2}\left(\frac{y}{\delta}\right)^3$$

Determine the boundary layer thickness, shear stress. 7

OR

12. (a) How will you determine whether a boundary layer flow is attached flow or on the verge of separation ? 7

(b) Find the displacement thickness, the momentum thickness for the velocity distribution in the boundary layer is given by :

$$\frac{u}{v} = 2\left(\frac{y}{\delta}\right) - \left(\frac{y}{\delta}\right)^2$$ 6