# Question Paper Code : 71029

B.E./B.Tech. DEGREE EXAMINATION, APRIL/MAY 2015.

Fifth Semester

Aeronautical Engineering

AE 2303/AE 1303/AE 53/10122 AE 503 – AERODYNAMICS — II

(Regulation 2008/2010)

Time : Three hours Maximum : 100 marks

Answer ALL questions.

PART A — ($10 \times 2 = 20$ marks)

1. Sound Propagation is an isothermal Process. True/False. Justify Your Answer.
2. Define Critical Mach No.
3. Define Characteristic Mach No.
4. Define Normal Shock and Oblique shocks
5. What is Shock Polar?
6. Plot the shock polar in dimensionless form for an upstream Mach No. of infinity.
7. How is a flow past a wedge different from flow past a cone?
8. What is meant by Prandtl Glauert transformation?
9. What is the need for the linear theory of supersonic flows?
10. Explain briefly the shock stall.

PART B — ($5 \times 16 = 80$ marks)

11. (a) A De Laval nozzle has to be designed for an exit Mach number of 1.5 with exit diameter of 200 mm. Find the ratio of throat area to exit area necessary. The reservoir conditions are given as: $P_0 = 1$ atm. (gauge) $T_0 = 20°$ C. Find also the maximum flow rate through the nozzle. What will be the exit pressure and temperature.

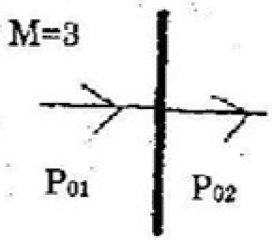
Or

(b) Discuss the performances of Nozzles under various Back Pressures.

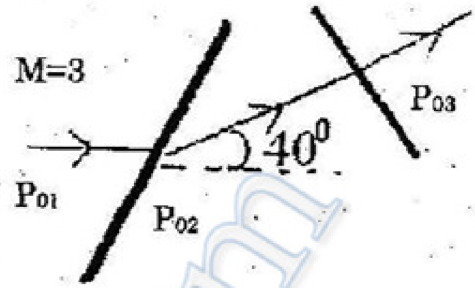
12. (a) It is required to decelerate a supersonic flow of Mach No.3 to a sub sonic speed. Consider two separate ways of achieving this

(i) The flow is slowed by passing directly through a normal shock. (4)

(ii) The flow first passes through an oblique shock with 40° wave angle and then below. Calculate the ratio $P_{03}/P_{01}$ for case 1 and 2. comment on the significance of the result. (12)



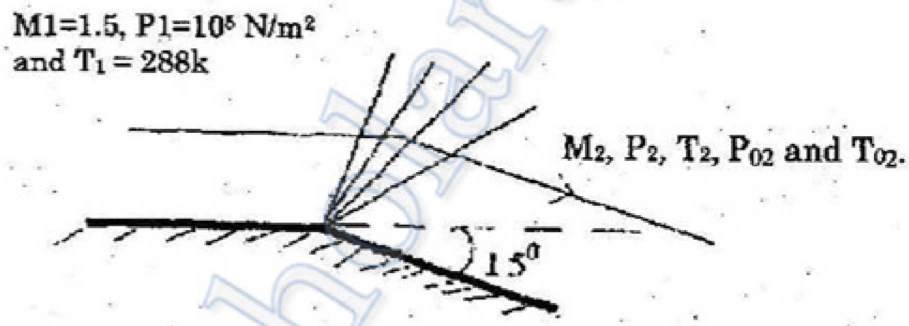

Case (1)




Case (2)

Or

(b) (i) A supersonic Flow with M1 = 1.5, P1 = $10^5$ N/m$^2$ and $T^1$ = 288k is expanded around a sharp corner through a deflection angle of 15°. Calculate $M_2$, $P_2$, $T_2$, $P_{02}$ and $T_{02}$. (12)




(ii) Distinguish Between Expansion Waves and Shock Waves. (4)

13. (a) The pressure and temperature of the air at inlet to a constant area duct are 120kPa and 150°C respectively, with an inlet Mach number of 3.0. Heat is transferred to the air as it flows through the duct leading to an exit Mach number of 1.5. Find (i) the pressure and temperature at exit. (ii) the maximum amount of heat that can be transferred to the air if no shocks occur in the flow. (iii) the exit pressure and temperature with the maximum heat transfer.

Or

(b) The exit pressure and Temperature are 125 kPa and 20°C respectively as air flows through a 25 cm diameter pipe at 1200 m$^3$/min. The pipe is 60m long. Assuming the friction factor f =0.005 estimate the inlet pressure and temperature

14. (a) Derive an expression for the $C_L$ and $C_D$ of a symmetric diamond profile in supersonic flow at small angle of attack.

Or

(b) The upper and lower surfaces of a symmetrical 2-D aerofoil are given by $Z = \pm \in_{\chi} (1 - \chi/c)^2$ where c is the chord and $\in << 1$. The aerofoil is at zero distance in a steady supersonic stream of Mach number $M_a$ in the positive $\chi$ direction.

(i) Find the velocity components according to the linear theory in the upper region of disturbance. (8)

(ii) Show that the drag coefficient of the aerofoil is given by

$$Cd = \frac{8}{15}\varepsilon^2 \frac{\varepsilon^2}{(M_\infty^2 - 1)^{1/2}}. \quad (8)$$

15. (a) Write short notes on:

(i) Drag Divergence. (5)

(ii) Transonic area Rule. (5)

(iii) Compare and contrast swept forward and swept backward wings. (6)

Or

(b) Derive the rankine — Hugoniot relation for an oblique shock. Compare graphically the variation of density ratio with the initial mach number in isentropic flow and flow with oblique shock.