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Department of Mechanical Engineering

ME 1303 GAS DYNAMICS AND JET PROPULSION

Unit – I : COMPRESSIBLE FLOW – FUNDAMENTALS

Part - A

1. Define stagnation state of a system.
2. An air jet (γ =1.4, R = 287 J/Kg-K) at 400K has sonic velocity .Determine its velocity .
3. Define compressible flow and Mach number.
4. Define stagnation temperature and stagnation pressure.
5. What is the advantage of using M*(second kind of Mach number) instead of M (Local Mach number) in some cases?
6. The wave front caused by firing a bullet gave a Mach angle of 35°. Find the velocity of the bullet if the static temperature of atmosphere is 276 K.
7. Define 'stagnation state' of a fluid.
8. Define: stagnation enthalpy.
9. Distinguish between Mach wave and normal shock.
10. Draw the Mach cone and identity its salient features.
11. What is the effect of Mach number on compressibility?
12. The static temperature of air is 300°C and velocity is 200 m/s. Find the maximum possible velocity C_{max} obtainable by air.
13. How will you classify the compressible flow based on Mach number range?
14. What is a Mach Cone?
15. Draw the disturbances wave propagation in compressible flow for M = 1 and M > 1.

16. When M* is used instead of M?

17. Express the stagnation enthalpy in terms of static enthalpy and velocity of flow.

18. When air is released adiabatically from a tyre, the temperature of air at the nozzle exit is 37°C below that of air inside the tyre. Neglecting irreversibility calculate the exit velocity of air.

PART -- B

1. (a).(i) What is the effect of Mach number on compressibility? Prove for γ = 1.4

$$\frac{P_0 - P}{\frac{1}{2}\rho c^2} = 1 + \frac{1}{4}M^2 + \frac{1}{40}M^4 + \ldots.$$

where P_0 - stagnation pressure , p – static pressure , p – density , c – velocity .

(a)(ii). Air at 200 KPa flows at a velocity of 50 m/sec . Find the Mach number at a point where its density is 2.9 kg/m^3.

(Or)

(b)(i) Speed of an aircraft is 800 km/hr. The stagnation conditions are 105 kPa & 308 K. Find static conditions and flight Mach number. (Take γ = 1.4, Cp = 1.005 kJ/kg-K).

(ii) Air flows from a reservoir at 550 kPa and 343 K. Assuming isentropic flow , calculate the velocity , temperature , pressure at a section where M = 0.6.

(iii) Velocity of an aircraft which has same Mach number at all altitudes , flying at an altitudes , flying at an altitude of 11000 m is 50 m/sec lower than that of its velocity at mean sea level. Determine its Match number.

2. (a) (i) What is the effect of Mach number on the compressibility? Prove for γ = 1.4

$$\frac{P_0 - P}{\frac{1}{2}\rho c^2} = 1 + \frac{1}{4}M^2 + \frac{1}{40}M^4 + \ldots.$$

(ii) Derive the following relations:

$$\frac{T_0}{T} = 1 + \left(\frac{\gamma - 1}{2}\right)M^2$$

$$\frac{T^*}{T} = \frac{2}{\gamma + 1} + \frac{\gamma - 1}{\gamma + 1}M^2$$

Or

(b) Air (Cp = 1.05 KJ/Kg, γ = 1.38) at $P_1 = 3 \times 10^5$ N/m² and T_1 = 500 K flows with a velocity of 200 m/s in a 30 cm diameter duct. Calculate

(i) Mass flow rate
(ii) Stagnation temperature
(iii) Mach number
(iv) Stagnation pressure values assuming the flow is compressible.

3. (a) (i) Air (Cp = 1.05 KJ/kg-K , γ = 1.38) at $P_1 = 3\times10^5$ and T_0 = 500K flows with a velocity of 200 m/s in a 0.3m diameter duct. Calculate :Mass flow rate, Stagnation temperature, Mach number and stagnation pressure values assuming the flow as compressible and incompressible respectively. (13)

(ii) An air plane travels at Mach 1.2 at an elevation where the temperature is 233K. Determine the speed of the air plane in km/hr, Assume γ=1.4. (3)

Or

(b) (i) A jet fighter at Mach number 2.5 . It is observed directly overhead at a height of 10 km. How much distance it would cover before a sonic boom is heard on the ground? (7)

(ii) An air jet at 400 K has sonic velocity. Determine: velocity of sound at 400 K, Velocity of sound at stagnation condition, Maximum velocity of jet, stagnation enthalpy.

4. (a) An aircraft is flying at an altitude of 12,000 m (T = 216 K, P = 0.193 bar) at a Mach Number of 0.82. The cross-sectional area of the inlet diffuser before the L.P. compressor stage is 0.5 m². Determine

(i) The mass of air entering the compressor per second.
(ii) The speed of the aircraft and
(iii) The stagnation pressure and temperature of air at the diffuser entry. (16)

5. Deduce expressions for P^*/P_0 and T^*/T_0 for a compressible fluid flow under isentropic conditions.

5. (i) Air flows in a duct with a velocity of 215 m/s. The temperature of air measured at a point along the duct is 30°C, and the air pressure is 5 bar. Determine (1) the stagnation pressure and (2) the Mach number at that point.

(ii) Derive an expression for T/T_0 for isentropic flow from energy equation and hence deduce the critical state to stagnation state property ratios.

6. (a) The pressure, temperature and Mach number at the entry of flow passage are 2.45 bar, 26.5°C and 1.4 respectively. If the exit Mach number is 2.5 determine for of a perfect gas (γ =1.3), R=0.467 kJ/kg-K.
(i) stagnation temperature
(ii) temperature and velocity of gas at exit, and
(iii) the flow rate per square meter of the inlet cross-section.

OR

(b) Air (γ = 1.4 ,R = 287.43 J/kg-K) enters a straight axi-symmetric duct at 300K, 3.45 bar and 150 m/s and leaves it at 277 K, 2.058 bar and 260 m/s. The area of cross-section at entry is 500cm². Assuming adiabatic flow determine:
(i) stagnation temperature
(ii) maximum velocity
(iii) mass flow rate and

(iv) area of cross-section at exit

7.(a). Air at rest at 90º C is accelerated isentropically (take $\gamma = 1.4$).

(i). What is the air speed in m/s when the Mach Number become 0.8?

(ii). What is the air speed when the flow becomes sonic?

(iii). What is the Mach Number when the air speed becomes 600 m/s. (16).

(or)

(b). What is the effect of Mach number on compressibility? Prove for $\gamma = 1.4$

$$\frac{P_0 - P}{\frac{1}{2}\rho c^2} = 1 + \frac{1}{4}M^2 + \frac{1}{40}M^4 + \ldots. \qquad (16)$$

8. Derive the energy equation

$$\frac{a^2}{\gamma - 1} + \frac{1}{2}c^2 = \frac{1}{2}c^2{}_{\max} = \frac{a_0{}^2}{\gamma - 1} = h_0$$

Unit – II : FLOW THROUGH VARIABLE AREA DUCTS

Part - A

1. Define one dimensional steady flow.
2. Show an adiabatic expansion process through a nozzle on T-S coordinates.
3. What is subsonic, sonic and super sonic flow with respect to Match number?
4. How the area and velocity vary in super sonic flow of nozzle and diffuser?
5. Draw h-S (Enthalpy-Entropy) diagram for the flow through a nozzle showing stagnation states.
6. Show graphically the variation of Mach number across a convergent-divergent nozzle.
7. Sketch on T-S plane isentropic and actual expansion of compressible fluid and give the expansion for nozzle efficiency.
8. Deduce the expression for impulse function.
9. Draw the operating characteristics for a converging nozzle with decreasing back pressure.
10. What are the difference between nozzle and diffuser?
11. Draw the curve between A/A* and Mach Number M.
12. List the condition for choking in CD nozzle?

Part - B

1. (a). A reservoir whose temperature can be varied in a wide range of temperature receives air at a constant pressure of 150 kPa. The air is expanded isentropically in a nozzle to an exit pressure of

101.5 kPa. Determine (without using gas tables) the values of the temperature to be maintained in the reservoir to produce the following velocities at the nozzle exit. (i) 100 m/sec , (ii) 250 m/sec .

(Or)

(b). A subsonic diffuser operating under isentropic conditions has inlet area of 0.15 m^2. The inlet conditions are C_1 = 240 m/sec , T_1 = 300 K , P_1 = 70 kPa. The velocity leaving the diffuser is 120 m/sec. Calculate for air (i) mass flow rate (ii) stagnation pressure at exit (iii) Stagnation temperature at exit (iv) static pressure at the exit (v) change in entropy (vi) exit Area.

2 . (a) A conical diffuser has entry and exit diameter of 15cm and 30cm respectively. The pressure, temperature and velocity of air at entry are 0.69 bar, 180 m/s respectively. Determine

(i) The exit pressure
(ii) The exit velocity and
(iii)The force exerted on the diffuser Walls

Assume isentropic flow , γ = 1.4, Cp = 1.00 KJ/Kg k

Or

(b) Air flowing in a duct has a velocity of 300 m/s , pressure 1.0 bar and temperature 290 K. Taking , γ = 1.4 and R =287 J/Kg k . Determine

(i) Stagnation pressure and temperature
(ii) Velocity of sound in the dynamics and stagnation conditions
(iii) Stagnation pressure assuming constant density.

3. (a) (i) Derive area ratio as a function of Mach number for one dimensional isentropic flow.

(ii) explain for a convergent nozzle the variation of pressure and Mach number when the back pressure is gradually lowered from stagnation pressure. (6)

Or

(b) A conical diffuser has entry and exit diameters as 0.15 m and 0.3 m respectively . The pressure, temperature and velocity of air at entry are 0.96 bar,340 K and 185 m/s respectively. Determine exit pressure, exit velocity and force exerted on the diffuser walls.
Assume γ=1.4 and Cp=1.005 KJ/Kg-k. (16)

(b) Air is discharged from a reservoir at $P_\circ$=6.91 bar and $T_\circ$=325°C through a nozzle to an exit pressure of 0.98 bar. If the flow rate is 3600 Kg/hr determine for isentropic flow (i) throat area, pressure and velocity (ii) exit area, Mach number and (iii) maximum velocity. (16)

(ii)A conical air diffuser has an intake area of 0.11 m^2 and an exit area of 0.44 m^2. Air enters the diffusers with a static pressure of 0.12 MPa, static temperature of 37°C and velocity of 267 m/s. Calculate (1) the mass flow rate of air through the diffuser,(2) the Mach number, static temperature and static pressure of the air leaving diffuser and (3) the net thrust acting upon the diffuser due to diffusion.

4. (a) (i) Deduce the expression for the change in area for isentropic flow and identity the geometric shape for nozzle and diffuser.

(ii) An air is flying at an altitude of 12,000 m (T = 216.65 K , p = 0.193 bar) at a Mach number of 0.82. The cross- sectional area of the inlet diffuser before the L.P compressor stage is 0.5 m². Determine the mass of air entering the compressor per second , the speed of the aircraft and the stagnation pressure and temperature of air the diffuser entry.

Or

(b) (i) Derive the expression for the area expansion ratio of nozzle.

(ii) A Freon-turbine has to use a maximum flow rate of 5 Kg/s of Freon. Employing a ring of convergent nozzles of total exit area of cross section of 100 cm^2 . The pressure in the nozzle entry space is 20 x 10 5 N/m^2.Taking Cp = 0.845 KJ/Kg K, γ =1.2. Calculate stagnation temperature , static pressure and temperature at the nozzle exit, and mach number at the nozzle exit.

5. (a) Air flow through a nozzle which has inlet area of 10 cm².If the air has a velocity of 80 m/s, temperature of 28°C and pressure of 700 KPa at the inlet section and a pressure of 250 KPa at the exit, find the mass flow rate through nozzle and assuming one dimensional isentropic flow, the velocity at the exit section on the nozzle.

Or

(b) A nozzle in a wind tunnel gives a test section Match number is 2, with a throat section 1000 cm² in area. The supply pressure and temperature at the nozzle inlet, where the velocity is negligible are 0.69 bar and 310 K respectively. The preliminary design is to be based on the assumption s that the flow is isentropic ,with γ =1.4 and that the flow is one dimensional at the throat and test section. Determine

(i) pressure, temperature, velocities at the throat and test section
(ii) area of cross-section
(iii) mass flow rate and
(iv) power required to drive the compressor

6. (a) Air flowing in a duct has a velocity of 300 m/s, pressure 1.0 bar and temperature 290 K. Taking γ = 1.4 and R = 287 J/kg K. Determine:
(i) Stagnation pressure and temperature
(ii) Velocity of sound in the dynamic and stagnation conditions
(iii) Stagnation pressure assuming constant density.

OR

(b) Air is discharged from a reservoir at P_0 = 6.91 bar and T_0 = 325°C through a nozzle to an exit pressure of 0.98 bar. If the flow rate is 3600 Kg/hr determine for isentropic flow:
(i) throat area, pressure and velocity
(ii) exit area Mach number and
(iii) Maximum velocity.

7. (a). Air is drawn isentropically from a standard atmosphere at sea level (101.3 KPa and 15º C) through a converging diverging nozzle. The static pressure at two different location are 80 KPa and 40 KPa respectively. Determine the Mach number at each of these locations. Also determine the velocity at each of these locations. (16).

OR

(b). A conical diffuser has entry and exit diameter of 150 mm and 300 mm respectively. The pressure, temperature and air velocity at entry are 69 KPa, 67º C and 180 m/s respectively. Determine exit pressure, exit velocity and the force exerted on the diffuser walls. Assume isentropic flow. Take γ = 1.4 and C_p = 1 kJ/kg.K.
(16).

Unit – III: FLOW THROUGH CONSTANT AREA DUCTS

Part – A

1. Give the assumptions that are used in the analysis of Rayleigh flow process.
2. Give two examples of Fanno flow in thermal systems.
3. Give two practical examples for Fanno flow and Rayleigh flow analysis.
4. What are the assumptions made in the analysis of Rayleigh process?
5. What are the assumptions made on Rayleigh flow?
6. What is the limiting Mach number in isothermal flow?
7. Write the difference between Rayleigh and Fanno flows.
8. What are the applications of Rayleigh flow?
9. Give any two assumptions regarding Fanno flow.
10. State the two governing equation used in plotting Rayleigh line.
11. What is Fanno line?
12. How do the flow properties change in Rayleigh flow?
13. What is meant by Mach reflection?
14. Explain the difference between fanno flow and isothermal flow.
15. Sketch the fanno line on the T-s plane and explain the significance of it.
16. Write down the ratio of velocities between any two sections in terms of their Mach number in a Fanno flow.
17. What is Rayleigh flow? Give two practical examples.
18. Draw Fanno curve and represent subsonic and supersonic flows.
19. Write down the assumptions made in Rayleigh flow?
20. Give four examples of Fanno flow in thermal systems.

Part - B

1 (a). Air flowing in an insulated duct with friction coefficient f = 0.002. At inlet the velocity is 130 m/sec , temperature 400 K and pressure is 250 KPa. The diameter of the duct is 16 cm. Find the length of the pipe that gives 20% drop in stagnation pressure . (ii) find the properties of air at a section 3.5 m from inlet and (iii) find the maximum length of pipe.

(OR)

(b).A combustion chamber delivers the gases at a Match number of 0.9 at 250 KPa and 1273 K. If the ratio of the stagnation temperatures at the exit and entry is 3.74 , determine the Mach number , pressure and temperature of the gas at entry. What is the amount of heat added and the maximum heat that can be added ?

2 (a) A circular duct passes 8.25 Kg/s of air at an exit Mach number of 0.5.The entry pressure and temperature are 3.45 bar and 38°C respectively and the coefficient of friction 0.005.if the mach number at entry is 0.15 , determine

(i) The diameter of the duct
(ii) Length of the duct
(iii)Pressure and temperature at the exit
(iv)Stagnation pressure loss

Or

(b) A combustion chamber in a gas turbine plant receives air at 350 K, 0.55 bar and 75 m/s. The air –fuel ratio is 29 and calorific value of the fuel is 41.87 MJ/Kg . Taking γ = 1.4 and R =0.287 KJ/Kg K for the gas determine:

(i) the initial and final mach numbers
(ii) final pressure ,temperature and velocity of gas
(iii)Percent stagnation pressure loss in the combustion chamber and
(iv)the maximum stagnation temperature attainable.

3. (a) A long pipe of 0.0254 m diameter has a mean coefficient of friction of 0.003. Air enters the pipe at a Mach number of 2.5, stagnation temperature 310 K and static pressure 0.507 bar. Determine for a section at which the mach number reaches 1.2: (i) static pressure and temperature (ii) stagnation pressure and temperature (iii) velocity of air (iv) distance of this section from the inlet and (V) mass flow rate of air.

Or

(b) The Mach number at the exit of a combustion chamber is 0.9. The ratio of stagnation temperatures at exit and entry is 3.74. if the pressure and temperature of the gas at exit are 2.5 bar and 1273 K respectively, determine : (i) Mach number , pressure and temperature of the gas at entry (ii) the heat supplied per kg of the gas and (iii) the maximum heat that can be supplied.

4. (a) A long pipe of 25.4 mm diameter has a mean coefficient of friction of 0.003. Air Enters the pipe at Mach number of 2.5, stagnation temperature 310K and static
pressure 0.51 bar. Determine for a section at which the Mach number reaches 1.2 (i) static pressure and temperature (ii) stagnation pressure and temperature (iii) Velocity of air (iv) distance of the section from the inlet and (v) mass flow rate of air (16)

OR

(b) The data for a combustion chamber employing a hydrocarbon fuel is given below: Entry: gas velocity =152 m/s, pressure =4 bar, temperature=400K Exit: mach Number = 0.8 Take gamma=1.3 & Cp = 2.144 KJ/KgK for the products of combustion caloric value of the fuel burnt = 43MJ/Kg.determine

(i) Entry mach number
(ii) Pressure, temperature and velocity of the air or exit
(iii) Stagnation pressure loss and
(iv) Air fuel ratio is required.

OR

5. (i) sketch the Fanno lines as h-p diagrams and explain how these lines are constructed?
(ii) The friction factor for a 50 mm diameter steel pipe is0.005. At the inlet to the pipe the velocity is 70 m/s, temperature is 80° c and the pressure is 10 bars. Find the temperature; pressure Mach number at exit if the pipe is 25 m long. Also determine the maximum possible length.

Or

(b) (i) distinguish between Rayleigh flow and Fanno flow .
(ii)Air enters a combustion chamber with a certain Mach number. Sufficient heat is added to obtain a stagnation temperature ratio of 3 and a final Mach number of 0.8. Determine the Mach number at entry and the percentage loss in static pressure. Take r=1.4 and Cp=1.005 KJ/Kg K for air.

6.(a) (i) Describe how the property changes in fanno flow with suitable diagram and justifications.
(ii) A circular duct passes 8.25 kg/s of air at an exit mach number of 0.5.The entry pressure and temperature are 3.45 bar and 38°C respectively and the coefficient of friction 0.005.If the mach number at entry is 0.15 , determine , the diameter of the duct, length of the duct, pressure and temperature at the exit and stagnation pressure loss.

Or

(b) (i) Outline the assumptions made in Rayleigh flow and explain their implication.
(ii) A combustion chamber in a gas turbine plant receives air at 350 K , 0.55 bar and 75 m/s . The air-fuel ratio is 29 and the calorific value of the fuel is 41.87 MJ/Kg . Taking ¥ = 1.4 and R = 0.287 KJ/ Kg K for the gas turbine the initial and final Mach number, final pressure , temperature and velocity of the gas , percent stagnation pressure loss in the combustion chamber , and the maximum stagnation temperature attainable.

8. (a) Air is flowing into an insulated duct with a velocity of 150 m/s. The temperature and pressure at the inlet are 280°C and 28 bar respectively. Find the temperature at a section in the duct where the pressure is 15.7 bar. If the duct diameter is 15 cm and the friction factor is 0.005, find the distance between the two sections.

(OR)

(b) Air flows through a constant area duct with inlet temperature of 20°C and inlet mach number of 0.5. What is the possible exit stagnation temperature ? It is desired to transfer heat such that at exit of the duct the stagnation temperature is 1180 K. For this condition what must be the limiting inlet mach number ? Neglect friction.

9. (a) The stagnation temperature of air in a combustion chamber is increased to 3.5 times its initial value. if the air at entry is at 5 bar ,105°C and Mach number of 0.25 , determine:
   (i) The Mach number ,pressure and temperature at the exit
   (ii) Stagnation pressure loss and
   (iii) Heat supplied per kg of air.

OR

(b) A long pipe of 25.4 mm diameter has a mean co-efficient of friction of 0.003. Air enters the pipe at a Mach number of 2.5, stagnation temperature 310 K and static pressure 0.507 bar. Determine the following for a section at which the Mach number reaches 1.2,
(i) Static pressure and temperature
(ii) Stagnation pressure and temperature
(iii) Mass flow rate of air
(iv) Distance of this section from the inlet.

9. (a) The stagnation temperature of air in a combustion chamber is increased to 3.5 times its initial value. If the air at entry is at 5 bar, 105°C and a Mach number of 0.25 determine:
   (i) the Mach number, pressure and temperature at the exit
   (ii) stagnation pressure loss, and
   (iv) the heat supplied per kg of air

OR

(b) A long pipe of 25.4 mm diameter has a mean coefficient of friction of 0.003. Air enters the pipe at a Mach number of 2.5, stagnation temperature 310 K and static pressure 0.507 bar. Determine for a section at which the Mach number reaches 1.2.
(i) static pressure and temperature
(ii) stagnation pressure and temperature
(iii) velocity of air
(iv) distance of this section from the inlet and
(v) mass flow rate air.

10. Properties are T_1 = 60.4° C, P_1 = 135 KPa absolute and velocity 732 m/s. Heat is added to the flow between section one and section two, where the Mach number is 1.2. Determine the flow properties at section two, the heat transfer per unit mass and the entropy change. (16).

OR

(b). A circular duct passed 8.25 Kg/s of air at an exit Mach number of 0.5, the entry pressure and temperature are 345 KPa and 311 K respectively. The average friction factor is 0.02 if the Mach number of entry is 0.15 determine,
(i). the diameter of the duct

(ii). Length of the duct
(iii). Pressure and temperature at the exit of the duct and
(iv). Stagnation pressure loss.

(16).

11. Air at P_0 = 10 bar, T_0 = 400 K is supplied to a 50 mm diameter pipe. The friction factor for the pipe surface is 0.002. If the Mach number changes from 3.0 at the entry to 1.0 at the exit determine
(i). the length of the pipe and
(ii). The mass flow rate.

Unit – IV : NORMAL SHOCK

Part - A

1. Define a shock wave.
2. What are the differences between oblique and normal shocks.
3. How is the shock formed?
4. What do you understand by strong and weak wave? Which one is preferred?
5. Give two useful applications of the shock waves.
6. Define strength of shock wave.
7. Write the Prandtle-Meyer Equation.
8. State two practical situations where oblique shock waves are produced,
9. Define the term: strength of a shock wave.
10. Sketch on oblique shock and show the angels associated with flow through it.
11. What do you understand by oblique shock?
12. Define strength of shock wave.
13. How is the shock formed?
14. Where is the shock wave advantageous?
15. Write the Prandtl – Meyer relation for normal shock.
16. Write the change across normal shock for Mach Number and static pressure.

Part – B

1. A gas (γ =1.3) at P_1 = 345 mbar, T_1 = 350 K and M_1 = 1.5 is to be isentropically expanded to 138 mbar. Determine:

(i) Deflection angle
(ii) Final Mach number and
(iii) The temperature of the gas

OR

2. A supersonic nozzle is provided with a constant diameter circular duct at its exit. The duct diameter is same as the nozzle exit diameter. Nozzle exit cross-section is three times that of its throat. The entry conditions of the gas (γ =1.4, R=0.287 kJ/kg K) are P_0 = 10 bar, T_0 = 600 K. Calculate the static pressure, Mach number and the velocity of the gas in the duct:
(i) when the nozzle operates at its design condition
(ii) when a normal shock occurs at its exit.

3. A jet of air at 275 k and 0.69 bar has initial Mach number of 2.0. if it passes through a normal shock wave, determine

(i) Mach number
(ii) Pressure and temperature ,
(iii) Speed of sound and
(iv) Jet velocity downstream of the shock.

OR

4. An air stream at Mach number of 2.0 is isentropically deflected by 10° in the clockwise direction. If the initial pressure and temperature are 98 KN/m² and 97°c, determine the final state of air after expansion.

5. Starting from the energy equation for flow through a normal shock, obtain the Prandtl- Meyer equation.

(OR)

6. Air with Mach number 2.5 enters a convergent duct with an area ratio A_2 / A_1 = 0.5. Under certain conditions, normal shock occurs at a test section where A_t/A_1 = 0.6. For this condition, find exit Mach number and pressure ratio across the duct. (A1 , A_2 , A_t – Area at inlet ,
exit and test section respectively).

7. The ratio of the exit to entry area in a subsonic diffuser is 4.0.The Mach number of a jet of a approaching the diffuser at P_0 = 1.013 bar, T = 290 K is 2.2.There is a standing normal shock wave just outside the diffuser entry. The flow of the diffuser is isentropic. Determine at the exit of the diffuser.

(i) Mach number
(ii) Temperature , and
(iii) Pressure

Or

8. A jet of air at a Mach number of 2.5 is deflected inwards at the corner of a curved wall. The wave angle at the corner is 60°. Determine the deflection angle on the wall, pressure and temperature ratios and final Mach number.

9. The ratio of the exit to entry area in a subsonic diffuser is 4. The Mach number of a jet of air approaching the diffuser at P = 1.013 bar, T = 290 K is 2.2. There is a standing normal shock wave just outside the diffuser entry. The flow in the diffuser is isentropic. Determine at the exit of diffuser, (i) Mach number (ii) temperature (iii) pressure. What is the stagnation pressure loss between the initial and final states of flow?

OR

10. Starting from the energy equation for flow through a normal shock , obtain the following relations : $C_x . C_y = a^{*2}$; $M_x^* . M_y^* = 1$ (where C_x,C_y –fluid velocity M_x^*,M_y^* -second kind of mach number at up stream and down stream of shock respectively; a* -critical velocity of sound)

11. The velocity of shock wave moving into stagnant air (P = 1.0 bar, T = 17ºC) is 500 m/s. If the area of cross section of the duct is constant, determine (i) pressure (ii) Temperature (iii) velocity of air (iv) stagnation temperature and (v) mach number imported upstream of the wave font

OR

12. An air steam at a Mach number of 2.0 is isoentropically defected by 10° in the clockwise direction. If the initial pressure and temperature are 98 kN/m² and 97°C determine the final state of air after expansion.

13. Derive the Rankine-Hergonoit relation. (6)

14. When a converging – diverging nozzle is operated at off-design condition a normal shock occurs at a section where the cross sectional area is 18.75 cm² in the diverging portion. At inlet to the nozzle the stagnation state is given as 0.21 MPa and 36°C. The throat area is 12.5 cm² and exit area is 25 cm².Estimate the exit Mach number, exit pressure and loss in stagnation pressure for flow through nozzle.

(b) (i) For flow through a normal shock deduce the relation

$$M_Y^2 = \frac{M_X^2 + \frac{2}{\gamma+1}}{\frac{2\gamma}{\gamma-1}M_X^2 - 1}$$

15. A bow shock occurs in front of a pitot tube when it is used in a supersonic flow field. It measures 16 KPa and 70 KPa for static pressure upstream of the shock and the pressure at the mouth of the tube respectively. Estimate the mach number of the supersonic flow. If the stagnation temperature is 300°C. Calculate the static temperature and total (stagnation) pressure upstream and downstream of the pitot tube.

16. Show that flow after normal shock is always subsonic. (6)

17. The ratio of the exit to entry area in a subsonic diffuser is 4.0 . The Mach number of a jet of air approaching the diffuser at P_0 = 1.013 bar, T = 290 K is 2.2. There is a standing normal shock wave just outside the diffuser entry . The flow in the diffuser is isentropic. Determine at the exit of the diffuser, Mach number, temperature, and pressure. What is the stagnation pressure loss between the initial and states of the flow?

Or

18. Plot and explain the wave pattern at the exit of a over expanded nozzle.

19. Determine the temperature and pressure field around a symmetric double wedge of 20° include angle kept at 15° angle of attack to a supersonic stream of number 2.5, by the shock- expansion theory.

20. A convergent divergent nozzle is designed to expand air from a reservoir in which the pressure is 700 KPa and temperature is 5°C and the nozzle inlet mach number is 0.2. The nozzle throat area is 46 cm2 and the exit area is 230 cm^2. A normal shock appears at a section where the area is 175 cm^2. Find the exit pressure and temperature. Also find the increase in entropy across the shock.

21. Deduce the hugoniot equation for shock heating and explain the same on p-v plane. Also give the advantage of shock heating.

22. Explain why normal shock wave causes supersonic flow to jump to subsonic flow. Substantiate your answer with governing equations, relevant reasons and using T-s diagram.

23. A converging – diverging nozzle has an exit area to throat area ratio of 2. Air enters this nozzle with a stagnation pressure of 1000 KPa and a stagnation temperature of 360 K. The throat area is 500 mm^2. The divergent section of the nozzle acts as a supersonic nozzle. Assume that a normal shock stands at a point M = 1.5. Determine the exit plane of the nozzle, the static pressure and temperature and Mach number.

24. A convergent divergent nozzle operates at off design condition while conducting air form a high pressure tank to a large container. A normal shock occurs in the divergent part of the nozzle at section where the cross section area is 18.75 cm^2. The stagnation pressure and stagnation temperature at the inlet of the nozzle are 0.21 MPa and 36º C respectively. The throat area is 25 cm^2. Estimate the exit Mach number, exit pressure. Loss in stagnation pressure and entropy increase during the flow between the tanks.

Unit – V : PROPULSION

Part – A

1. Name three commonly used aircraft engines.
2. What do you mean by mono propellant? Give examples.
3. What is a bypass engine and define bypass ratio?
4. Distinguish between monopropellant and bipropellant.
5. What is after burning in turbojet engines?
6. Why a rocket is called non–air breathing engines?
7. Define the terms ‘specific thrust’ and ‘specific impulse’.
8. List out the main advantages of ‘turbojet Engines’.
9. How will you classify the rocket engines based on applications?
10. What is ‘characteristic velocity’?
11. Give the equation of thrust for a turbojet engine.
12. How is turbofan engine different from turbo prop engine?
13. Name any two solid propellant fuels and oxidizers.
14. What are the advantages of solid propellant rockets over liquid propellant rockets? List any two.
15. Difference between impulse and specific impulse.

16. What are unique feature of jet engine combustion chambers?
17. What are the limitations of jet engine?
18. Compare the propulsive efficiency of jet engine with that of rocket engine.
19. What is the need for multi-stage rocket?
20. Sketch the thrust and propulsive efficiency variation against the speed ratio for a turbo jet engine.
21. Find the ratio of jet speed to flight speed for optimum propulsive efficiency.
22. Give any two differences between ram jet and pulse jet.
23. What is mono propellant? Give an example.
24. What are the main parts of Ram jet engine?
25. State any two differences between Ram jet and Pulse jet.
26. Write an expression for thrust of jet propulsion.
27. What is bi-propellant? Give an example
28. What is after burning in turbojet engine?
29. What are the different types of rocket engines?
30. What is after burning in turbojet engines?
31. Why rocket engines are called as non-breathing engine?

PART - B

1. (a). Explain with a neat sketch the principle of operation of a turbojet engine and state its advantages and disadvantages.

OR

1. (b). Describe the important properties of liquid and solid propellants desired for rocket propulsion and give examples for both the propellants.

2. (a). A turbo jet has a speed of 750 km/hr while flying at an altitude of 10000 m. The propulsive efficiency of the jet is 50% and the overall efficiency of the turbine plant is 16%. The density of the air at 10000 m altitude is 0.173 kg/m3 .The drag on the plane is 6250 N. Calorific value of the fuel is 48000 kJ/kg. Calculate (i) absolute velocity of the jet , (ii) diameter of the jet and (iii) power output of the unit in KW.

2. (b). (i). The effective jet exit velocity from a rocket is 2700 m/sec. The forward flight velocity is 1350 m/sec and the propellant consumption is 78.6 kg/sec. Calculate thrust , thrust power and propulsive efficiency.

(ii). Explain a solid propellant rocket engine with a neat sketch.

3. (a) Explain with a neat sketch the principle of operation of a ramjet engine and state its advantages and disadvantages

Or

3. (b) Explain with a neat sketch the working of a turbo-pump feed system used in a liquid propellant rocket.

4. (a) (i) Derive the thrust equation for rocket engines.

(ii) The diameter of the propeller of an aircraft is 2.5 m; it flies at a speed of 500 Km/hr at an altitude of 8000 m. For a flight to jet speed ratio of 0.75, determine : The flow rate of air through the propeller , Thrust produced , Specific thrust , Specific impulse and Thrust power.

4. (b) (i) Draw the sketch of a pulse jet engine. Write down its main advantages and disadvantages.

(ii) The effective jet velocity from a rocket is 2700 m/s. The forward flight velocity is 1350 m/s and the propellant consumption is 78.6 Kg/s. Calculate: Thrust, Thrust power and propulsive efficiency.

5. (a) Explain the working principle of Ramjet engine with neat sketch and list out advantages and disadvantages.

OR

5. (b) Diameter of an aircraft propeller is 4.0 m. The speed ratio is 0.8 at a flight speed of 450kmph.If the ambient conditions of air at the flight attitude are T=256K and P=0.54 bar, determine

(i) Propulsive efficiency
(ii) Thrust and
(iii) Thrust power.

6. (a) A rocket flies at 10,080kmph with an effective exhaust jet velocity of 1400m/s and propellant flow rate of 5.0 kg/s. If the heat of reaction of the propellant is 6500 kJ/kg of the propellant mixture, determine (i) propulsion efficient and propulsive power (ii) engine output and thermal efficiency and (iii) overall efficiency.

OR

6. (b) (i) Explain the working principle of liquid propellant rockets with neat sketch.

(ii) List out the important properties of solid propellant.

7. (a)(i) Determine the optimum flight to jet speed ratio that gives desirable performance for a turbojet engine.

(ii) The flight speed of a turbojet is 800Km/hr at 10000 m altitude. The density of air at that altitude is 0.17 Kg/m^3. The drag for the plane is 6.8KN. The propulsive efficiency of the jet is 60%. Calculate the SFC, Air-fuel ratio and jet velocity. Assume the calorific value of fuel as 45000kJ/Kg and the overall efficiency of the turbojet plant as 18%.

Or

7. (b) (i) compare solid and liquid propellant systems.

(ii) With neat sketches explain the constructional features and working of (1) ramjet engine and (2) turbofan engine.

8. (a) (i) Derive expression for thrust and specific impulse and propulsive efficiency of a rocket motor. (ii)Calculate the thrust specific impulse, propulsive efficiency, thermal and overall efficiency of a rocket engine from the following dada: Effective jet velocity=1250 m/s, flight to jet speed ratio=0.8, oxidizer flow rate =3.5 kg/s, fuel flow rate=1 kg/s. Heat of reaction of exhaust gases=2500 kJ/kg.

Or

8. (b) (i) Discuss in detail the various propellants used in solid fuel rockets and liquid fuel systems. Also sketch the propellant feed system for a liquid propellant rocket motor.

(ii) Explain the performance characteristics for solid propellant and liquid propellant rockets.

9. (a) (i) Explain the working of turbofan engine with a suitable sketch.

(ii) An aircraft files at 1000 KMPH. One of its turbojet takes in 40 Kg/s of air and expands the gases to the ambient pressure. The air-fuel ratio is 50 and the lower calorific value of the fuel is 43 MJ/Kg. For maximum thrust power determine jet velocity, thrust, specific thrust , thrust power and TSFC.

Or

9. (b) (i) Classify aircraft and rocket propulsion engines and identity their working principle with at least one important feature for each of them.

(ii) Diameter of an aircraft propeller is 4.0 m. The speed ratio is 0.8 at a flight speed of 450 Kmph. If the ambient condition of air at the flight altitude are T= 256 K and p= 0.54 bar .Determine the propulsive efficiency, thrust and thrust power.

10. (a) (i) Deduce the expression for the thrust and effective jet velocity of a rocket engine.

(ii) A rocket nozzle has a throat area of 18 cm^2 and combustion chamber pressure of 25 bar. If the specific impulse is 127.42 seconds and weight flow rate 44.145 N/s. Determine the thrust coefficient, propellant weight flow coefficient, specific propellent consumption, and the characteristic velocity.

Or

10. (b) (i) Derive the expression for the orbital and hence calculate the orbital and escape velocity of a rocket at mean sea level and an altitude of 300 km (assume the radius of earth = 6314.6 km).

(ii) A missile has maximum flight speed to jet ratio of 0.211 and specific impulse=2204s.determine for a burn out time of 8s, effective jet velocity, mass ratio, maximum flight speed and altitude gain during powered and coating flights.

11. (a) (i) Compare the constructional features and operating performance of turbo fan and turbojet engines.

(ii) Derive the expressions for the thrust (propulsive) power, propulsive efficiency , thermal efficiency , overall efficiency and the optimum value of flight to jet speed ratio for a turbo jet engine.

(OR)

11. (b) A turbojet propels an aircraft at a speed of 900 km/hour , while taking 3000 kg of air per minute. The isentropic enthalpy drop in the nozzle is 200 kJ/kg and the nozzle efficiency is 90%. The air – fuel ratio is 85 and the combustion efficiency is 95%. The calorific value of the fuel is 42,000 kJ/kg. Calculate :

(i) The propulsive power
(ii) Thrust power
(iii) Thermal efficiency
(iv) Propulsive efficiency.

12. (a) (i) Explain the construction and operation of a ramjet engine and derive an expression for the ideal efficiency.

(ii) A ram jet engine operates at M = 1.2 at an altitude of 6500 m. The diameter of the inlet diffuser at entry is 50 cm and the stagnation temperature at the nozzle entry is 1500 K. The calorific value of the fuel used is 40 MJ/kg. The properties of the combustion gases are same as those of air (¥ = 1.4, R = 287 J/kgK). The velocity of the air at the diffuser exit is negligible.

Calculate (1) the efficiency of the ideal cycle , (2) flight speed , (3) air flow rate , (4) diffuser pressure ratio and (5) fuel air ratio ,(6) nozzle jet Mach number. The efficiencies of the diffuser = 0.9 , combustor = 0.98 and the nozzle = 0.96.

(OR)

12. (b) (i) Explain the construction and operation of a solid propellant rocket engine. Also name any four solid propellants.

(ii) What are the advantages and disadvantages of liquid propellants compared to solid propellants. Explain in detail.

13. (a) (i) Compare the constructional features and operating performance of Turbo prop and Turbojet engines.

(ii) Derive an expression for the thrust power, propulsive efficiency, thermal efficiency and optimum value of flight to jet speed ratio for a turbo jet engine.

OR

13. (b) A turbo jet propels an aircraft at a speed of 900km/hr, while taking 3000 kg of air per minute. The isentropic enthalpy drop in the nozzle is 200 kJ/kg and the nozzle efficiency is 90%.The air-fuel ratio is 85 and the combustion efficiency is 95%. The calorific value of the fuel is 42,000 kJ/kg. Calculate:

(i) The propulsive power
(ii) Thrust power
(iii) Thermal efficiency
(iv) Propulsive efficiency

14. (a) (i) Explain with sketch a liquid propelled rock engine and its merits compared to solid propelled system.

(ii) Write brifly on Rockets Perfomance.

OR

14. (b) (i) Write about the Multi staging in rockets.

(ii) The effective jet exit velocity of a rocket is 2400 m/s, the forward flight velocity is 1200 m/s and propellent consumption is 72 kg/s. calculate thurst, thrust power and propulsive efficiency.

15. (a). Describe the working of supersonic ramjet engine with a neat sketch. List out its advantages and disadvantages.

(or)

15. (b). (i). Explain the principle of operation of liquid propellant and solid propellant engines with neat sketches. (10)

(ii). List down the advantages of liquid propellant rockets. (6)