Name: **Enrolment No:**	UPES University of Tomorrow

UPES

End Semester Examination, December 2024

Course: Propulsion-II **Semester : V**

Program: B. Tech ASE **Time : 03 hrs.**

Course Code: ASEG3004 **Max. Marks: 100**

Instructions: Make use of sketches/plots to elaborate your answer. Brief and to-the-point, answers are expected. Assume suitable data if needed. Gas Table allowed, Refer attached formula sheet.

SECTION A

(5Qx4M=20Marks)

S. No.		**Marks**	**CO**
Q 1	List the key components of an turbofan and turboprop engine with neat sketch.	**4**	**CO4**
2	Describe the differences in working principles between an ramjet and scramjet engine.	**4**	**CO2**
3	Apply the concept of A/A* ratios to analyze supersonic nozzle performance and show the plot between area ratio to Mach number with suitable example.	**4**	**CO3**
4	Describe how normal shocks affect total pressure in supersonic inlets and suggest the right method to anchor the shock.	**4**	**CO1**
5	Evaluate the impact of thermal choking on the overall efficiency of a thermodynamic cycle.	**4**	**CO2**

SECTION B

(4Qx10M= 40 Marks)

Q 6	Air is discharged from a reservoir at a pressure of 1.2 MPa and temperature of 450 K through a nozzle to an exit pressure of 0.08 MPa. If the flow rate through the nozzle is 3000 kg/h, and assuming isentropic flow, determine a) The throat area, pressure, and velocity. b) The exit area and Mach number.	**10**	**CO3**
7	Air enters a convergent -divergent nozzle with stagnation conditions of 400 kPa and 400 K. the area ratio of the nozzle is 4. After passing through the nozzle , the flow enters a duct where heat is added. At the end of duct there is normal shock wave. The stagnation temperature upstream of a shock wave is 500 K. assuming isentropic flow in the nozzle and Rayleigh flow in the duct. Calculate the heat added and the stagnation pressure loss.	**10**	**CO2**
8	A 3.5 m long well insulated duct of diameter 50 mm and average friction coeffiecient 0.005 is connected to a frictionless bell mouth entrance. Air at 110 kPa and 300 K is drawn through the entrance and flows into the duct. Find the maximum mass flow rate , the flow parameters at exit and the range of back pressure that will produce the flow.	**10**	**CO3**

	OR Differentiate between turbojet, turbofan and turboprop engines based on the following parameters. 1. Thrust generation 2. Efficiency 3. Size of the powerplant 4. Application 5. Schematic diagram		
9	An aerial vehicle is equipped with a pulsejet engine at an altitude of 1000 ft having diffuser area The vehicle is cruising at a speed of 850 m/s. consider the naphthalene as an fuel and heating value is equal to 49000 kJ/ kg Compute (a) Mass flow rate per unit area (b) thrust power. (c) TSFC	**10**	**C04**
SECTION-C **(2Qx20M=40 Marks)**			
Q 10	A supersonic wind tunnel is designed for M= 3.0. if the air in the reservoir is at a pressure of 1.4 bar and at a temperature of 27 ^{0} C, determine the mass flow rate , area of test section and the pressure , temperature and density of air at the nozzle throat and test section. Nozzle throat area is 0.1 m^2. **OR** Design an advanced turbofan engine for passenger aircraft operating at an altitude of 35000 ft at 0.6 Mach number with the following data. Bypass ratio = 0.6 Overall pressure ratio= 23 Fan pressure ratio =3.5 Fuel heating value = 46000 kJ/kg Turbine inlet temperature = 1800 K Diffuser pressure recovery factor =0.89 Axial compressor efficiency = 0.9 Axial fan efficiency = 0.84 Combustion efficiency = 0.98 Burner pressure recovery factor = 0.96 Turbine efficiency = 0.93 Nozzle efficiency = 1 Calculate the thrust per unit mass flow rate and the TSFC.	**20**	**CO4**

11	Design A single-spool afterburning turbojet engine is powering a fighter airplane flying at Mach number Ma = 1.2 at an altitude of 40000 ft. The air mass flow rate is 20 kg/s. The compressor has a pressure ratio of 7 and isentropic efficiency of 0.90. The pressure loss in the combustion chamber is 7% and the heating value of fuel is 46,000 kJ/kg. The burner efficiency is ηb = 0.97. The Turbine inlet temperature is 1300 K, and its isentropic efficiency is 0.92. The maximum temperature in the afterburner is 2000 K. The pressure drop in the afterburner is 3% and the afterburner efficiency is 0.9. The nozzle efficiency is ηn = 0.96. Calculate 1 The stagnation pressure ratio of the diffuser (rd) and its isentropic efficiency 2. The thrust force 3. TSFC	**20**	**C04**

- fuel-to-air ratio $(f) = \frac{\dot{m}_f}{\dot{m}_a}$

- $T_{momentum} = \dot{m}_a(1+f)U_e$ $\qquad T_{pressure} = (P_e - P_a)A_e$

  $T = \dot{m}_a[(1+f)U_e - u]$ $\quad$ or $T = \dot{m}_a[U_e - u]$

  $\therefore$ Thrust $= \dot{m}_a[(1+f)U_e - u] + (P_e - P_a)A_e$

- $T = \dot{m}_h[(1+f)U_{eh} - u] + \dot{m}_c(U_{ec} - u) + A_{eh}(P_{eh} - P_a) + A_{ec}(P_{ec} - P_a)$ — For turbofan & propfan

- η_p (propulsive eff.) $= \frac{uT}{uT + 0.5\dot{m}_e(U_e - u)^2} = \frac{u\{\dot{m}_a[(1+f)U_e - u] + A_e(P_e - P_a)\}}{u\{\dot{m}_a[(1+f)U_e - u] + A_e(P_e - P_a)\} + (0.5\dot{m}_a(1+f)(U_e - u)^2)}$

- $\eta_p = \frac{2uT}{\dot{m}_a[(1+f)U_e^2 - u^2]}$ → Turbofan & propfan

- $\eta_p = \frac{u(T_h - T_c)}{u(T_h - T_c) + W_h + W_c}$

  Hot thrust $(T_h) = \dot{m}_h[(1+f)U_{eh} - u] + A_{eh}(P_{eh} - P_a)$
  Cold thrust $(T_c) = \dot{m}_c[U_{ec} - u] + A_{ec}(P_{ec} - P_a)$
  wake losses $(W_h) = \frac{1}{2}\dot{m}_h(1+f)(U_{eh} - u)^2 = \frac{1}{2}\dot{m}_{eh}(U_{eh} - u)^2$
  $W_c = \frac{1}{2}\dot{m}_c(U_{ec} - u)^2$

- $\eta_p = \frac{Tu}{Tu + 0.5\{\dot{m}_{eh}(U_{eh} - u)^2 + \dot{m}_c(U_{ec} - u)^2\}}$

- $\eta_p = \frac{2uT}{\dot{m}_h\{(1+f)U_{eh}^2 + \beta U_{ec}^2 - (1+\beta)u^2\}}$; Bypass ratio $(\beta) = \dot{m}_c/\dot{m}_h$

- η_{th} (Thermal eff.) $= \frac{Tu + \frac{1}{2}\dot{m}_a(1+f)(U_e - u)^2}{\dot{m}_f Q_R}$ → Ramjet & Turbojet

- $\eta_{th} = \frac{Tu + \frac{1}{2}\dot{m}_h(1+f)(U_{eh} - u)^2 + \frac{1}{2}\dot{m}_c(U_{ec} - u)^2}{\dot{m}_f Q_R}$ → Turbofan & Propfan

- unchoked condition, $\eta_{th} = \frac{[(1+f)U_e^2 - u^2]}{2fQ_R}$; $\eta_{th} = \frac{U_{eh}^2 + \beta U_{ec}^2 - (1+\beta)u^2}{2fQ_R}$

- η_o (overall eff.) $= \frac{Tu}{\dot{m}_f Q_R}$; unchoked, $\eta_o = \eta_{th}\frac{2u}{u + U_e}$

- Specific Thrust $= T/\dot{m}_a$ $\qquad$ • $TSFC = \dot{m}_{fuel}/T = \frac{f}{(T/\dot{m}_a)}$

- Specific Impulse $(I_{sp}) = T/\dot{m}_f g$

- $f = \frac{C_{ph}T_{03} - C_{pc}T_{02}}{\eta_b Q_R - C_{ph}T_{03}}$

- Exhaust velocity $(U_e) = \sqrt{2C_{ph}T_{03}\left[1 - \left(\frac{P_a}{P_{03}}\right)^{\frac{\gamma_h - 1}{\gamma}}\right]}$

} Pulsejet & Ramjet