INSTITUTE OF AERONAUTICAL ENGINEERING

(Autonomous)

Dundigal, Hyderabad - 500 043

AERONAUTICAL ENGINEERING

TUTORIAL QUESTION BANK

Course Name	:	**AEROSPACE PROPULSION I**
Course Code	:	**A52108**
Class	:	III B. Tech I Semester
Branch	:	AERO
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Course Coordinator	:	Mr. C Satya Sandeep, Assistant Professor
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OBJECTIVES

To meet the challenge of ensuring excellence in engineering education, the issue of quality needs to be addressed, debated and taken forward in a systematic manner. Accreditation is the principal means of quality assurance in higher education. The major emphasis of accreditation process is to measure the outcomes of the program that is being accredited.

In line with this, Faculty of Institute of Aeronautical Engineering, Hyderabad has taken a lead in incorporating philosophy of outcome based education in the process of problem solving and career development. So, all students of the institute should understand the depth and approach of course to be taught through this question bank, which will enhance learner's learning process.

S No	Question	Blooms taxonomy level	Course Outcomes
	UNIT – I FLIGHT PROPULSION		
	Part - A (Short Answer Questions)		
1	**Write** the history of Propulsion	Remember	1
2	**Write** the Engine Operational Limits	Understand	1
3	**Describe** the Air breathing Engines	Understand	1
4	**What** is the operating principle of gas turbine engine	Understand	1
5	**What** are the components of engine that you know	Understand	1
6	**Give** the thrust equation	Remember	1
7	**Define** thermal & propulsive efficiency.	Remember	1
8	**What** are the engine performance parameters that you know	Knowledge	1
9	**Discuss** the effect of flight conditions on engine.	Knowledge	1
10	**What** is an Engine Matching Number?	Understand	1

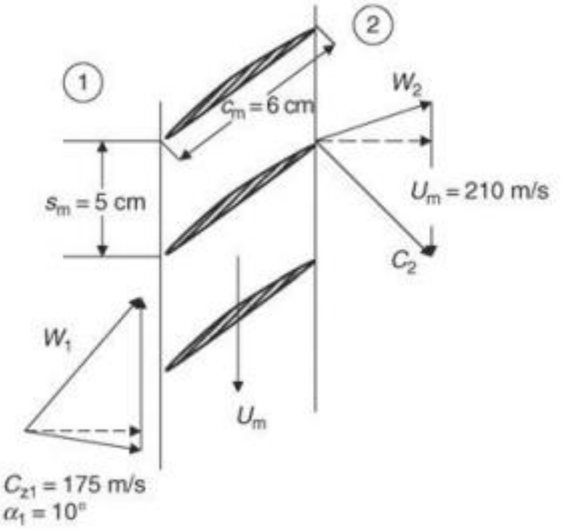
Part - B (Long Answer Questions)			
1	**Describe** the Reciprocating engines in detail	Understand	1
2	**Explain** the Airbreathing Engines	Understand	1
3	**Describe** the engine components	Understand	1
4	**Derive** Engine Performance characteristics.	Knowledge	1
5	**Describe** the criteria for Engine Selection	Understand	1
6	**Discuss** Airframe and Engine Matching	Understand	1
7	**Sketch** and explain turbojet engine	Create	1
8	**Sketch** and explain turbo fan engine	Create	1
9	**Sketch** and explain turbo jet with afterburner engine	Create	1
10	**Sketch** and explain turboprop engine	Create	1
Part - C (Problem Solving and Critical Thinking Questions)			
1	An air breathing engine has a flight Mach number of M0 =0.85 At an altitude where the speed of sound is 300 m/s. The air mass flow rate into the engine is 50 kg/s. Calculate. The ram drag (in kN) for this engine.	Analyze	1
2	A turbojet engine is flying at 200 m/s. The products of combustion achieve an exhaust velocity of 900 m/s. Estimate the engine propulsive efficiency.	Understand	1
3	An aircraft is flying at an altitude where the ambient static pressure is $p0$ = 10 kPa and the flight Mach number is $M0$= 0.85. The total pressure at the engine face is measured to be $pt2$ = 15.88 kPa. Assuming the inlet is adiabatic and γ= 1.4, calculate **(a)**the inlet total pressure recovery πd **(b)** the inlet adiabatic efficiency ηd **(c)** the non-dimensional entropy rise caused by the inlet $\Delta sd/R$	Analyze	1
4	A turbojet engine is flying at 200 m/s. The products of combustion achieve an exhaust velocity of 900 m/s. Estimate the engine propulsive efficiency	Understand	1
5	A mixed exhaust turbofan engine (JT8D from Pratt and Whitney, 1974) is described by its internal pressures and temperature, as well as air mass flow rates and the mixed jet (exhaust) velocity. Let us examine a few parameters for this engine, for a ballpark approximation. **(a)** Estimate the fuel flow rate from the total temperature rise across the burner assuming the fuel heating value is ∼18,600 BTU/lbm and the specific heat at constant pressure is 0.24 and 0.26 BTU/lbm ·◦R at the entrance and exit of the burner, respectively **(b)** Calculate the momentum thrust at the exhaust nozzle and compare it to the specified thrust of 14,000 lbs **(c)** Estimate the thermal efficiency of this engine and compare it to Problems 3.1 and 3.2 as well as a Carnot cycle operating between the temperature extremes of this engine. Explain the differences **(d)** Estimate the specific fuel consumption for this engine in lbm/h/lbf	Analyze	1
6	A ramjet is flying at Mach 2.0 at an altitude where $T0$ = − 50◦C and the engine airflow rate is 10 kg/s. If the exhaust Mach number of the ramjet is equal to the flight Mach number, i.e., $M9$ = $M0$, with perfectly expanded nozzle and $Tt9$ 2500 K, calculate **(a)** the engine ram drag Dram in kN **(b)** the nozzle gross thrust Fg in kN **(c)** the engine net thrust Fn in kN **(d)** the engine propulsive efficiency ηp Assume gas properties remain the same throughout the	Analyze	1

	engine, i.e., assume γ= 1.4 and cp = 1004 J/kg ·K. Also, assume that the fuel flow rate is 4% of airflow rate.		
7	A turbojet-powered aircraft cruises at $V0$ = 300 m/s while the engine produces an exhaust speed of 600 m/s. The air mass flow rate is 100 kg/s and the fuel mass flow rate is 2.5 kg/s. The fuel heating value is QR = 42,000 kJ/kg. Assuming that the nozzle is perfectly expanded, calculate **(a)** engine ram drag in kN **(b)** engine gross thrust in kN **(c)** engine net thrust in kN **(d)** engine thrust-specific fuel consumption (TSFC) in mg/s/kN	Analyze	1
8	A rocket engine consumes propellants at the rate of 1000 kg/s and achieves a specific impulse of Is = 400 s. Assuming the nozzle is perfectly expanded, calculate **(a)** the rocket exhaust speed $V9$ in m/s **(b)** the rocket thrust in MN	Knowledge	1
9	A rocket engine has a nozzle exit diameter of $D9$ = 2 m. It is perfectly expanded at sea level. Calculate the rocket pressure thrust in vacuum.	Analyze	1
10	A turbojet engine produces a net thrust of 40,000 N at the flight speed of $V0$ of 300 m/s. For a propulsive efficiency of ηp = 0.40, estimate the turbojet exhaust speed $V9$ in m/s.	Understand	1
UNIT – II PARAMETRIC CYCLE ANALYSIS OF ENGINES			
Part – A (Short Answer Questions)			
1	**Describe** the components of engine that you know	Knowledge	1
2	**What** arethe Performance requirements of Gas Turbine Engine	Knowledge	1
3	**What** the Under/Over Expanded Nozzle	Knowledge	1
4	**Define** Back Pressure.	Knowledge	2
5	**Define** Burning efficiency	Knowledge	2
6	**Define** the cycle of gas turbine Engine	Knowledge	2
7	**Write** about Parametric Cycle.	Knowledge	2
8	**What** is the effect of component performance?	Knowledge	2
9	**How** to compute the ideal turbojet?	Knowledge	2
10	**What** is meant by polytrophic efficiency	Knowledge	2
Part - B (Long Answer Questions)			
1	**Explain** the Engine Components in detail..	Understand	1
2	**Develop** the Engine component performance requirements	Knowledge	1
3	**Explain** Nozzle Under expansion and Over Expansion in detail with figures.	Knowledge	1
4	**What** is the gas turbine cycle? **Explain** in detail for all types	Understand	2
5	**Describe** Parametric cycle analysis clearly	Knowledge	2
6	**Derive** engine performance parameters.	Knowledge	2
7	**Describe** the engine design choices and constraints	Knowledge	2
8	**Discuss** computation of ideal turbo jet and turbo fan engines.	Knowledge	2
9	**Compare** the efficiencies of turbine and a compressor.	Knowledge	2
10	**Compare** the polytrophic efficiencies of turbine and a compressor	Knowledge	2

Part - C (Problem Solving and Critical Thinking Questions)			
1	An aircraft is flying at an altitude where the ambient static pressure is $p0$ = 10 kPa and the flight Mach number is $M0$ = 0.85. The total pressure at the engine face is measured to be $pt2$ = 15.88 kPa. Assuming the inlet is adiabatic and γ= 1.4, calculate **(a)** the inlet total pressure recovery πd **(b)** the inlet adiabatic efficiency ηd **(c)** the non-dimensional entropy rise caused by the inlet $\Delta sd/R$	Understand	1
2	A multistage axial-flow compressor has a mass flow rate of 50 kg/s and a total pressure ratio of 35. The compressor polytrophic efficiency is ec = 0.90. The inlet flow condition to the compressor is described by $Tt2$ = 288 K and $pt2$ = 100 kPa. Assuming the flow in the compressor is adiabatic, and constant gas properties throughout the compressor are assumed, i.e.,γ=1.4 and cp = 1004 J/kg ·K, calculate	Analyse	1
3	A gas turbine combustor has inlet condition $Tt3$ = 800 K, $pt3$ = 2 Mpa, air mass flow rate of 50 kg/s, $\gamma 3$ = 1.4, $cp3$ = 1004 J/kg ·K. A hydrocarbon fuel with ideal heating value QR = 42,000 kJ/kg is injected in the combustor at a rate of 1 kg/s. The burner efficiency is ηb = 0.995 and the total pressure at the combustor exit is 96% of the inlet total pressure i.e., combustion causes a 4% loss in total pressure. The gas properties at the combustor exit are $\gamma 4$ = 1.33 and $cp4$ = 1156 J/kg ·K. Calculate.	Analyse	2
4	Consider an uncooled gas turbine with its inlet condition the same as the exit condition of the combustor described in Example 4.3. The turbine adiabatic efficiency is 88%. The turbine produces a shaft power to drive the compressor and other accessories at $\wp t$ = 45 MW. Assuming that the gas properties in the turbine are the same as the burner exit in Example 4.3, calculate **(a)** turbine exit total temperature $Tt5$ in K **(b)** turbine polytrophic efficiency, et **(c)** turbine exit total pressure $pt5$ in kPa **(d)** turbine shaft power $\wp t$ based on turbine expansion ΔTt.	Understand	2
5	Consider the internally cooled turbine nozzle blade row of Example 4.5. The hot gas total pressure at the entrance of the nozzle blade is $pt4$ = 1.92 MPa, cpg = 1156 J/kg ·K, and γg = 1.33. The mixed-out total pressure at the exit of the nozzle has suffered 2% loss due to both mixing and frictional losses in the blade row boundary layers. Calculate the entropy change $\Delta s/R$ across the turbine nozzle blade row.	Analyse	2
6	An aircraft is flying at an altitude where the ambient static pressure is $p0$ = 10 kPa and the flight Mach number is $M0$ = 0.85. The total pressure at the engine face is measured to be $pt2$ = 15.88 kPa. Assuming the inlet is adiabatic and γ= 1.4, calculate **(a)** the inlet total pressure recovery πd **(b)** the inlet adiabatic efficiency ηd **(c)** the non-dimensional entropy rise caused by the inlet $\Delta sd/R$	Understand	2
7	Consider a convergent–divergent nozzle with a pressure ratio NPR = 10. The gas properties are γ= 1.33 and cp = 1,156 J/kg ·K and remain constant in the nozzle. The nozzle adiabatic efficiency is ηn= 0.94. Calculate **(a)** nozzle total pressure ratio πn **(b)** nozzle area ratio $A9/A8$ for a perfectly expanded nozzle **(c)** nozzle exit Mach number $M9$ (perfectly expanded)	Analyse	2
8	The total pressures, temperatures and mass flow rates at some stations inside a non-afterburning, mixed-flow turbofan engine, at take-off, are shown. For simplicity of analysis, assume the gas is calorically perfect with constant properties (γ= 1.4 and cp = 1004 J/kg ·K) throughout the engine. Calculate **(a)** bypass ratio, α **(b)** fuel-to-air ratio, f	Understand	2

	(c) mixer exit total temperature, T_{t6M}, in K **(d)** exhaust Mach number, M_8 (note that the exhaust nozzle is of convergent type) **(e)** (un-installed) take-off thrust, $F_{T.O.}$, in kN and lbf		
9	The T–s diagram shows the power split between the propeller and the nozzle. Assuming the mass flow rate is $\dot{m}$ = 37 kg/s with γ= 1.33 and c_p = 1,152 J/kg ·K, calculate **(a)** ideal power available in station 4.5, $\wp_i$ in MW **(b)** LPT exit pressure, p_{t5}, in kPa **(c)** LPT exit temperature, T_{t5}, in K **(d)** LPT power (actual) in MW **(e)** nozzle exit velocity, V_9, in m/s, for η_n = 0.95.	Analyse	2
10	An afterburning turbojet engine is shown in "wet mode". Calculate **(a)** fuel-to-air ratio in the primary burner, f, for Q_R = 42,000 kJ/kg **(b)** turbine exit total temperature, T_{t5} (K) **(c)** turbine exit total pressure, p_{t5}, in kPa **(d)** fuel-to-air ratio in the afterburner, f_{AB}, for $Q_{R,AB}$ = 42,000 kJ/kg **(e)** nozzle exit Mach number, M_9	Understand	1
	UNIT – III **AIRCRAFT ENGINE COMPONENTS**		
	Part - A (Short Answer Questions)		
1	**What** is Subsonic inlet?	Knowledge	3
2	**What** is Subsonic inlets- function?	Knowledge	3
3	**What** is a diffuser?	Knowledge	3
4	**Explain** Supersonic inlets	Knowledge	3
5	**Discuss** fan nozzle, Exhaust nozzles, primary nozzle	Knowledge	3
6	**Explain** nozzle-area ratio	Knowledge	4
7	**What** is thrust reversing?.	Knowledge	4
8	**What** is the effect of fuel-air mixture ratio?.	Understand	4
9	**What** is a Burner and state its types	Knowledge	4
10	**What** are the Aircraft gas turbine engine fuels?	Knowledge	4
	Part - B (Long Answer Questions)		
1	**Describe** the subsonic inlets, function and performance characteristics	Knowledge	3
2	**Describe** the supersonic inlets, function and performance characteristics	Knowledge	3
3	**Derive** performance characteristics for sub/supersonic inlets.	Knowledge	3
4	**Explain** the nozzle/ primary nozzle and fan nozzle.	Knowledge	3
5	**Discuss** the combustion process, characteristics and effects of fuel and	Knowledge	3
6	**What** is a burner, its types, function**?**	Knowledge	4
7	**What** is a combustor, its performance parameter and its design	Knowledge	4
8	**Explain** the Aircraft gas turbine fuels, its composition and commonly used fuels	Knowledge	4
9	**Differentiate** RAMJET and SCRAMJET inlets	Knowledge	4
10	**Derive** area Mach relation of a nozzle?	Knowledge	4

Part - C (Problem Solving and Critical Thinking Questions)			
1	Calculate and graph the ratio of nozzle throat area with the afterburner on and off for a range of turbine expansion parameters τt between 0.45 and 0.65. Keep $\tau\lambda$= 6.0 and vary $\tau\lambda$,AB from 6.5 to 9.0. Also, investigate the effect of flow losses on the throat area ratio of the nozzle with/without afterburner operation..	Knowledge	3
2	A subsonic aircraft flies at $M0$ = 0.85 with an inlet mass flow ratio (MFR) of 0.90. Calculate the critical pressure coefficient Cp, crit on the nacelle. Also calculate the maximum cowl (frontal) area ratio $AM/A1$ if this inlet is to experience an average surface pressure coefficient corresponding to the critical value, i.e., $\bar{C}p \approx Cp$,crit.	Knowledge	3
3	An isentropic, fixed-geometry inlet, is designed for MD = 1.5. If this inlet is to be started by over speeding, calculate the necessary Mach number for over speed	Knowledge	3
4	A normal-shock inlet is flying at a Mach number of 1.8. However, due to a non-optimum backpressure, the normal shock is inside the duct where A/Ai = 1.15. Calculate the percentage loss in the total pressure recovery due to this back pressure.	Understand	3
5	Calculate and graph the ratio of nozzle throat area with the afterburner on and off for a range of turbine expansion parameters τt between 0.45 and 0.65. Keep $\tau\lambda$= 6.0 and vary $\tau\lambda$,AB from 6.5 to 9.0. Also, investigate the effect of flow losses on the throat area ratio of the nozzle with/without afterburner operation..	Understand	3
6	An exhaust nozzle has a pressure ratio of 8, i.e., $pt7/p0$ = 8. The ratio of specific heats for the gas is γ= 1.3. Calculate the percentage increase in gross thrust if we were to expand the gas in an ideal convergent–divergent nozzle as compared with a convergent nozzle.	Understand	4
7	An exhaust nozzle has a pressure ratio of 10, i.e., $pt7/p0$ = 8. The ratio of specific heats for the gas is γ= 1.7. Calculate the percentage increase in gross thrust if we were to expand the gas in an ideal convergent–divergent nozzle as compared with a convergent nozzle	Analyze	4
8	The total temperature of the gas entering a convergent–divergent nozzle is $Tt7$ = 900 K. The ratio of specific heats of the gas is γ= 1.3 and cp = 1,243.7J/kg ·K. The nozzle pressure ratio $pt7/p0$ = 8.0, the convergent portion of the nozzle has a total pressure ratio $pt8/pt7$ = 0.98, and the divergent section total pressure ratio is $pt9/pt8$ = 0.95. For $p9$ = $p0$, calculate **(a)** $V9$ in m/s **(b)** $V9s$ in m/s **(c)** $V9i$ in m/s and **(d)** The velocity coefficient Cv	Analyze	4
9	An exhaust nozzle has a pressure ratio of 8, i.e., $pt7/p0$ = 8. The ratio of specific heats for the gas is γ= 1.3. Calculate the percentage increase in gross thrust if we were to expand the gas in an ideal convergent–divergent nozzle as compared with a convergent nozzle	Analyze	4
10	An exhaust nozzle has a pressure ratio of 10, i.e., $pt7/p0$ = 18. The ratio of specific heats for the gas is γ= 1.4. Calculate the percentage increase in gross thrust if we were to expand the gas in an ideal convergent–divergent nozzle as compared with a convergent nozzle.	Analyze	4
UNIT – IV **ROTATING MACHINERY**			
Part – A (Short Answer Questions)			
1	**Construct** Axial Flow Compressors.	Knowledge	5
2	**Discuss** the Euler's turbo-machinery equations	Knowledge	6

3	**What** are Stage parameters you know	Knowledge	6
4	**What** is a Axial flow turbines?	Knowledge	6
5	**What** is a swirl condition?.	Knowledge	6
6	**What** are the Limits on achievable performance of compressors?	Knowledge	6
7	**What** are the Limits on achievable performance of turbines?.	Analyze	6
8	**Explain** surge.	Analyze	6
9	**Write** about rotating stall.	Knowledge	6
10	**Describe** blade cooling	Knowledge	
Part - B (Long Answer Questions)			
1	**Explain** Axial Flow Compressor, construction, flow field in it in detail	Knowledge	5
2	**Explain** Axial Flow Turbine, construction, flow field in it in detail	Knowledge	6
3	**Construct** the Axial flow turbines- similarities and differences with compressors.	Apply	6
4	**Draw** the velocity diagram of compressor	Knowledge	6
5	**Discuss** the Typical blade profiles.	Apply	6
6	**Draw** the velocity diagram of turbines	Knowledge	6
7	**Describe** blade cooling techniques	Analyze	6
8	**What** is meant by blade angle	Knowledge	6
9	**What** is meant by blade velocity	Analyze	6
10	**Draw** the velocity diagram of compressor	Understand	6
Part - C (Problem Solving and Critical Thinking Questions)			
1	The absolute flow at the pitch line to a compressor rotor has a coswirl with $C\theta 1 = 78$ m⁄s. The exit flow from the rotor has a positive swirl, $C\theta 2 = 172$ m⁄s. The pitch line radius is at $rm = 0.6$ m and the rotor angular speed is $\omega = 5220$ rpm. Calculate the specific work at the pitch line and the rotor torque per unit mass flow rate.	Understand	5
2	A rotor blade row is cut at pitch line, rm. The velocity vectors at the inlet and  exit of the rotor are Assuming that $Um1 = Um2 = 210$ m⁄s and $Cz1 = Cz2 = 175$ m⁄s, $\rho 1 = 1$ g⁄m3 $\beta 2 = -25°$, and $\varpi r = 0.03$, calculate	Analyze	6

	(a) $W\theta 1$ and $W\theta 2$ **(b)** W1m and W2m **(c)** D-factor Drm **(d)** Circulation Γm **(e)** rotor lift at pitch line per unit span **(f)** lift coefficient at pitch line **(g)** rotor-specific work at rm **(h)** loading coefficient ψm **(i)** degree of reaction ◦Rm		
3	The flow coefficient to a rotor at pitch line is φm = 0.8, its loading coefficient is Cm =1.0. The inlet flow to the rotor has zero swirl in the absolute frame of reference. Assuming axial velocity Czm = constant across the rotor, calculate (a)the relative inlet flow angle β1m **(b)** the relative exit flow angle β2m **(c)** the degree of reaction ◦Rm	Understand	6
4	The relative flow across a compressor rotor is shown. The rotor rotational speed is U= 300 m/s. The axial velocity is Cz = 165 m/s and it remains constant across the rotor. Assuming the rotor solidity is σr = 0.8, calculate: **(a)** de Haller criterion **(b)** rotor degree of reaction (for a repeated stage) **(c)** rotor specific work (in kJ/kg) **(d)** rotor D-factor	Understand	6
5	An axial-flow compressor rotor at the pitch line has a radius of rm = 0.5 m. The shaft rotational speed is ω= 6000 rpm. The inlet flow to the rotor has zero preswirl and the axial velocity is constant, with Cz = 150 m/s. The stage has a 80% degree-of-reaction at the pitch line where the solidity is σm = 1.2. The stage adiabatic efficiency is equal to the polytrophic efficiency, ec = 0.92. Assuming that the inlet total temperature is Tt1 = 288 K, γ= 1.4 and cp = 1.004 kJ/kg.K, calculate: **(a)** rotor specific work at r = rm in kJ/kg **(b)** stage loading, ψ, at r = rm **(c)** flow coefficient, ϕ, at r = rm **(d)** rotor relative Mach number at the pitch line, M1r,m **(e)** stage total pressure ratio at the pitch line **(f)** rotor diffusion factor at the pitch line **(g)** is the de Haller criterion satisfied?	Analyse	6
6	An axial-flow compressor has no IGV. Its inlet conditions are: pt1 = 100 kPa, Tt1 =288 K and the mass flow rate $\dot{m}$ = 100 kg⁄s. The compressor pressure ratio is πc = 15 and the polytrophic efficiency is ec = 0.90. The axial flow in the compressor is designed to be constant at Cz = 166 m/s. Assuming the gas constant is R = 287 J/kg ·K and γ= 1.4, calculate: **(a)** absolute Mach number at the inlet to the compressor, M1 **(b)** compressor shaft power, $\wp c$, in MW **(c)** compressor inlet flow area, A1 in m2 **(d)** exit Mach number, M2 (assume the exit flow is swirl free) **(e)** compressor exit area, A2 in m2.	Knowledge	6
7	An axial-flow turbine nozzle turns the flow from an axial direction in the inlet to an exit flow angle of α2 = 70◦. The rotor wheel speed is U = 400 m/s at the pitch line. The rotor is of impulse design and the exit flow from the rotor has zero swirl, i.e., α3= 0. Calculate **(a)** the rotor-specific work **(b)** the stage loading at the pitch line	Analyze	6
8	Coolant air is bled from a compressor exit at Ttc = 800 K with cpc = 1004 J⁄kg ·K. The coolant is given a (positive) preswirl before it enters the rotor blade root in the direction of the rotor rotation. Assuming the port of coolant entry into the rotor is at rc = 42 cm with rotor angular speed ω= 12000 rpm, and the coolant enters the rotor blade root axially, as shown in Figure 10.64, calculate the	Analyse	6

	coolant relative total temperature as it enters the rotor blade.		
9	The inlet flow angle to a turbine nozzle is $\alpha 1 = 5^{\circ}$. Using correlation 10.88, estimate the induced angle, $\Delta\theta$ind, at the nozzle leading edge due to local flow curvature for a range of nozzle blade solidities from 1.0 to 2.0. Graph the induced flow angle versus the blade solidity.	Knowledge	6
10	The number of cycles to failure is an important design parameter in turbine blade material selection. Fatigue strength of conventionally cast and two directionally solidified materials. Assuming a turbine blade root experiences a stress of 80 kpsi, and the same operating condition, estimate the number of cycles to failure for the three materials shown. Note that the x-axis is in logarithmic scale.	Analyse	6
UNIT – V PERFORMANCE ANALYSIS – COMPONENT MATCHING			
Part - A (Short Answer Questions)			
1	**What** is Non-dimensionalisation?	Knowledge	7
2	**State** correction of engine and component characteristic parameters.	Knowledge	8
3	**Explain** Performance analysis of compress	Knowledge	8
4	**Explain** Performance analysis of fan	Knowledge	8
5	**Explain** Performance analysis of burner.	Knowledge	9
6	**Explain** Performance analysis of turbine	Knowledge	10
7	**Write** the pressure ratio of compressor	Understand	10
8	**What** is the off design performance of compressor	Understand	10
9	**Write** about Engine thrust rating.	Analyze	10
10	**What**'s the requirement of station numbering	Knowledge	10
Part - B (Long Answer Questions)			
1	**Explain** Non-dimensionalisation and correction of engine and component characteristic parameters	Knowledge	8
2	**Discuss** Performance analysis of compressor	Knowledge	8
3	**Discuss** Performance analysis of fan	Evaluate	8
4	**Discuss** Performance analysis of burner	Evaluate	9
5	**Discuss** Performance analysis of turbine closed section thin-walled beams.	Knowledge	10
6	**Derive** relation between compressor pressure ratio, mass flow rate, efficiency, engine speed	Knowledge	10
7	**Explain** matching of inlet and compressor	Understand	10
8	**Explain** the matching of compressor and combustion chamber	Understand	10
9	**Explain** the matching of combustion chamber turbine	Understand	10
10	**Explain** the matching of turbine and nozzle.	Understand	1
Part - C (Problem Solving and Critical Thinking Questions)			
1	**Explain** the non-dimensionalisation and correction of engine and component characteristic parameters	Knowledge	8
2	**Discuss** Performance analysis of compressor, fan, burner, turbine, exhaust nozzle	Understand	8
3	**Describe** Relation between compressor pressure ratio, mass flow rate, efficiency, engine speed	Knowledge	8

4	**What** Engine control- throttle lever setting, fuel flow, burner temperature ratio, turbine speed, flow coefficient, mass flow rate- relations	Understand	9
5	**Write** about Engine thrust ratings..	Understand	10
6	In a turbojet engine, the compressor face total pressure and temperature are 112 kPa and 268 K, respectively. The shaft speed is 6400 rpm. The air mass flow rate is 125 kg/s and the fuel mass flow rate is 2.5 kg/s. The fuel heating value is 42,000 kJ/kg and the engine produces 145 kN of thrust. Express the following engine corrected parameters: **(a)** the corrected (air) mass flow rate $\dot{m}_{c2}$ in kg/s assuming $p_{t2} = 0.99\ p_{t0}$ **(b)** the corrected shaft speed N_{c2} in rpm **(c)** the corrected fuel flow rate, $\dot{m}_{fc}$, in kg/s **(d)** the corrected thrust F_c in kN **(e)** the corrected thrust-specific fuel consumption TSFC in mg/s/N. *Note:* $p_{ref} = 101.33$ kPa and $T_{ref} = 288.2$ K	Analyze	10
7	In a gas generator, the compressor and burner performance maps are shown. The turbine adiabatic efficiency is assumed nearly constant at $\eta_t = 0.85$. The nominal operating line on the compressor performance map represents the $T_{t4}/T_{t2} = 7.0$ throttle line. The design corrected shaft speed is $N_{c2} = 10{,}000$ rpm and the compressor pressure ratio at design is $\pi_{c,D} = 13.5$ (note that the corrected mass flow rate at the compressor face is 89 kg/sat design). Assuming $\gamma_c = 1.4$, $c_{pc} = 1004$ J/kg ·K $\gamma_t = 1.33$, $c_{pt} = 1156$ J/kg ·K $f \approx 0.03$ $\eta_m = 0.995$ Calculate and graph the gas generator pumping characteristics, as percent corrected shaft speed N_{c2} (% design).	Analyze	10
8	A turbojet engine has the following design-point parameters: **1.** $M_0 = 0$, $p_0 = 0.1$ MPa, $T_0 = 15°$C **2.** $\pi_d = 0.98$ **3.** $\pi_c = 15$, $e_c = 0.90$ **4.** $Q_R = 42{,}800$ kJ/kg, $\pi_b = 0.97$, $\eta_b = 0.98$, $T_{t4} = 1485°$C **5.** $e_t = 0.80$, $\eta_m = 0.995$ **6.** $\dot{m}_{c2} = 24$ kg/s **7.** N_{c2} (rpm) = 6,000 **8.** $M_{z2} = 0.6$ **9.** $\pi_n = 0.97$, $p_9/p_0 = 1.0$ The off-design flight condition is described by $M_0 = 2.0$, $p_0 = 18$ kPa, $T_0 =$ $-15°$C $T_{t4} = 1475°$C $\pi_d = 0.88$ $p_9/p_0 = 1.0$ Assuming all other component efficiencies (except π_d that is specified) remain the same (as design) at off-design and gas properties are $\gamma_c = \gamma_t = 1.4$ and $c_{pc} =$ $c_{pt} = 1004$ J/kg · K, calculate **(a)** $\pi_{c\text{-}O\text{–}D}$ **(b)** $\dot{m}_{c2,O-D}$ (in kg/s) **(c)** $N_{c2,\ O\text{–}D}$ (in rpm) **(d)** $M_{z2,O\text{–}D}$	Analyze	10
9	Cruise flight condition is: M0 = 0.85, altitude is 12 km (U.S. standard atmosphere), with $\dot{m}_{c2} = 200$ kg/s. Assuming that the inlet total pressure recovery at cruise is 0.995, calculate **(a)** corrected mass flow rate, $\dot{m}_{c0}$, in kg/s **(b)** captured stream area, A0, in m2 **(c)** physical mass flow rate of air, $\dot{m}_0$, in kg/s **(d)** inlet throat area (assume $M_{th} = 0.70$), neglecting	Analyze	10

	the total pressure loss from highlight to throat **(e)** ram drag in kN		
10	A turbojet engine has a design corrected mass flow rate of $\dot{m}_{c2}$ = 100 kg/s at the standard sea level static condition. The design axial Mach number at the engine face is M_{z2} = 0.5. Calculate the engine face flow area, A_2, in m^2 (γ= 1.4, R = 287 J/kg ·K, p_{SL} = 101 kPa and T_{SL} = 288 K)	Analyse	10

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