AE 711/234 AIRCRAFT PROPULSION
Quiz 1, 05/09/2024

Constants:
Specific heat ratio of air, $\gamma = 1.4$
Gas constant for air, $R = 0.287$ kJ/kg.K
Gravitational acceleration, $g = 9.81$ m/s^2

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1. Derive the expression for the range of the aircraft. Find the range of a slender supersonic aircraft where its lift-to-drag ratio as a function of flight Mach number is given as,
$$\frac{L}{D} \approx 3\frac{M_0 + 3}{M_0}$$
The flight Mach number and TSFC are 2.2 and 0.2 kg/(N.h) respectively. The initial-to-final weight ratio of the aircraft is 3. Assume a steady cruise flight where the ambient temperature is 210 K. [4 marks]

2. A Mach 3 ramjet is flying at an altitude where the ambient temperature is 216 K. The peak temperature in combustion chamber is 2800 K. Using ideal cycle analysis, estimate the fuel-air ratio (f) if the heating value of the fuel is 43600 kJ/kg. Assume Cp of air to be 1.005 kJ/kg.K. [4 marks]

3. Sketch a real turbojet cycle. A turbojet designed for military application has a design Mach number of 1.8. It is flying in the ambient condition of 216 K with its optimum compressor ratio. The engine has a turbine inlet temperature of 1296 K. Using the ideal cycle analysis, derive and estimate its thermal efficiency. [7 marks]

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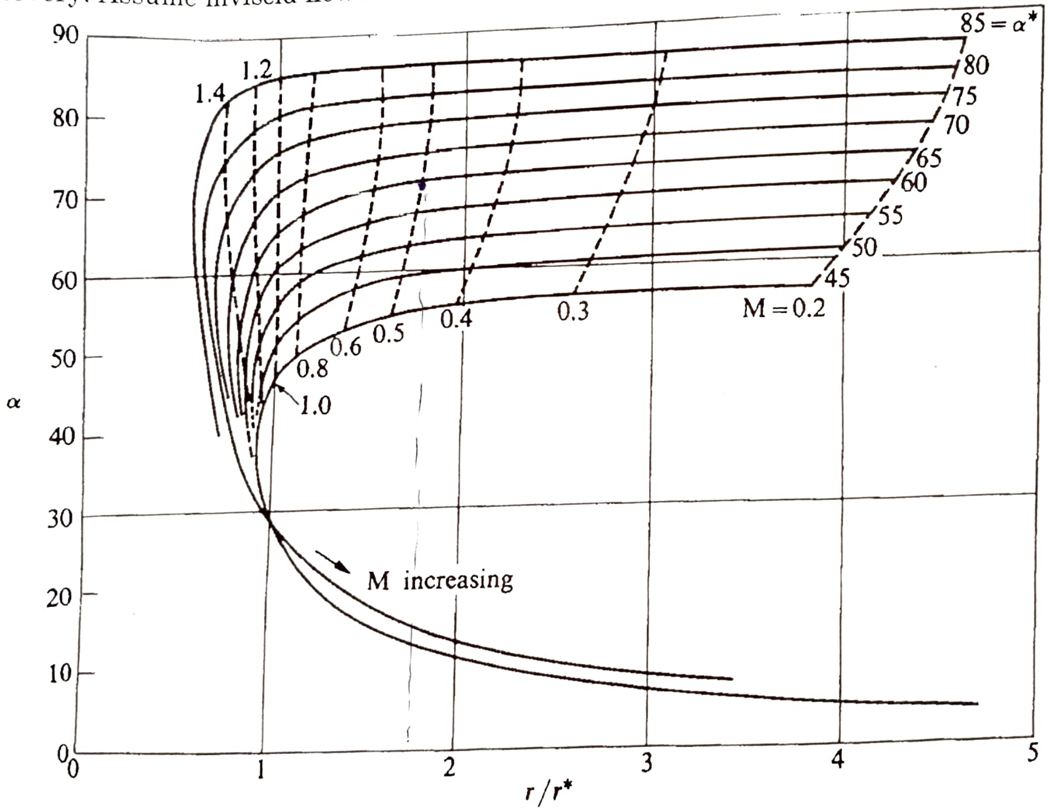
Quiz 2, 28/10/2024

A centrifugal compressor impeller has 20 straight radial impeller blades. The inlet flow condition to the compressor is purely axial with Mach number of 0.45. The shaft rotational speed is 14,400 rpm. The geometric parameters of the centrifugal compressor are: hub radius at inlet = 0.05 m, tip radius at inlet = 0.1 m, exit radius of impeller = 0.25 m. The static pressure and temperature at the inlet are 90 kPa and 250 K, respectively. The radial velocity at the impeller exit is equal to the inlet axial velocity and the compressor polytropic efficiency is 0.85. Assuming slip factor of 0.9 with gas properties $\gamma = 1.4$ and R = 287 J/kg · K, calculate (a) the mass flow rate, (b) compressor shaft power, (c) impeller absolute exit Mach number, (d) stage total pressure ratio, (e) width of the impeller at exit. **[9 units]**

For a centrifugal compressor in above question, a vaneless radial diffuser is to be designed to reduce the Mach number to 0.5. Estimate the radius of diffuser exit and the static pressure recovery. Assume inviscid flow in diffuser. Use the plot given below. **[3 units]**



Explain the effect of small increase in mass flow (while operating at same rpm) in the first stage on the flow dynamics in last stages in a multistage axial compressor. **[3 units]**

AE 711/234 Aircraft Propulsion
End semester examination, 23/11/2024, Total points: 40

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24 points

[Problem 1]

Consider Snecma M88-2 afterburning turbofan engine that powers Rafale. The engine has an axial compressor with 3 low-pressure stages, 6 high-pressure stages and axial turbine with 1 high and low pressure stage each. The overall compressor pressure ratio is 24.5 and the design Mach number is ~1.8. In order to achieve high power-to-weight ratio, supersonic performance and afterburning configuration, low bypass ratio of 0.3 is employed in the engine. The estimated air mass flow rate into the whole engine is 65 kg/s. Some of the engine parameters, which are not available in open literature, are guessed for engine analysis.

The engine is run with afterburner during transonic acceleration phase. Consider this acceleration phase when the flight Mach number is 0.9 at an altitude where ambient pressure and temperature are 10 kPa and 216 K respectively. Specific details of the aerothermodynamics in various components are given below:

i. The compression in diffuser section is adiabatic with a loss in total pressure of 5%.
ii. The compressor isentropic efficiency is 0.9.
iii. The combustor exit temperature is 1850 K and the fuel heating value is 42000 kJ/kg. The flow suffers a total pressure loss of 6% in burner. The combustion efficiency is 98.5%.
iv. The overall turbine isentropic efficiency is 0.92 and the mechanical efficiency is 0.95.
v. The compression in the fan may be assumed isentropic; however, the flow in the bypass duct (from fan exit to mixer inlet) suffers a total pressure loss of 1%.
vi. The total pressures of turbine exit and bypass stream entering the mixer are same. However, the flow at the exit of the mixer suffers from a total pressure loss of 5%.
vii. The afterburner peak temperature is 2000 K (with same fuel as the main combustor) with a total pressure loss of 10%.
viii. The nozzle isentropic efficiency is 0.95.

Estimate the thermodynamic state at the end of each station. Calculate the thrust produced and the specific fuel consumption with afterburner "ON". (The rated thrust with afterburner "ON" by the manufacturer is ~75 KN).

Gas properties are as follows:

Gas in different section	γ	Cp (J/kg.K)
Air	1.4	1004
Turbine section	1.4	1004
Afterburner exit and nozzle	1.33	1240

PTO

Problem [2] 16 points

One of the variants of MiG-21 used an afterburning turbojet (R-13-300) involving following turbomachines. Two spool axial compressor – 3 low-pressure stages and 5 high-pressure stages – with overall pressure ratio of 8.9 is used. An axial flow turbine with 1 low-pressure stage and 1 high-pressure stage runs the respective stages of the compressor. The mass flow rate into the engine is ~60 kg/s. Based on this data, answer the following questions related to the design of the turbomachines:

i. If the LP compressor has a stage pressure ratio of 1.2, what is the stage pressure ratio for HP compressor? (all the stages of LP compressor are identical and so are the HP compressor stages)

ii. The aircraft is flying at cruise Mach number of 2 with ambient conditions of 10 kPa and 210 K. The intake is designed in such a way that it brings down the flow Mach number to 0.5 adiabatically with diffuser pressure ratio of 0.88. Find out the stagnation conditions at the compressor inlet and diffuser efficiency.

iii. The inlet guide vanes take the flow at Mach 0.5 and turn it through an angle of 20^{0} with respect to the axial direction without any loss. The hub-to-tip ratio for the first stage is 0.2. Isentropic efficiency of the stage is 0.92. If we wish to design the first stage at mean blade radius, where the degree of reaction needs to be 0.5, what should be the rotational speed of the LP compressor? Also estimate the blade angles.

iv. The LP compressor is powered by an impulse LP turbine stage. Assume all LP compressor stages are identical and the overall isentropic efficiency of LP compressor is also 0.92. The nozzles provide the hot gases to the LP turbine at an angle of 72^{0} and total temperature of ~900 K. The isentropic efficiency of turbine may be assumed to be 0.95. Since the flow exiting the LP turbine is entering into the nozzle, the stage is to be designed such that the flow leaving the turbine is purely axial. Design the LP turbine and find out the LP turbine pressure ratio. (Assume that the fuel flow rate is negligibly small as compared to air mass flow rate)

Assume $\gamma = 1.4$ and $R = 287$ J/kg.K for both, compressor and turbine, for simplicity.
