Version CAH/4

EGT3
ENGINEERING TRIPOS PART IIB

Wednesday 22 April 2015   9.30 to 11

Module 4A3

TURBOMACHINERY I

Answer not more than ***two*** *questions.*

All questions carry the same number of marks.

The ***approximate*** *percentage of marks allocated to each part of a question is indicated in the right margin.*

Write your candidate number ***not*** *your name on the cover sheet.*

STATIONERY REQUIREMENTS
Single-sided script paper

SPECIAL REQUIREMENTS TO BE SUPPLIED FOR THIS EXAM
CUED approved calculator allowed
Attachment: Compressible Flow Data Book (38 pages).
Engineering Data Book

10 minutes reading time is allowed for this paper.

You may not start to read the questions printed on the subsequent pages of this question paper until instructed to do so.

1 (a) A centrifugal compressor has a stagnation pressure ratio of 4.0 and a rotational speed of 25000 rpm. It has an inlet stagnation temperature of 300 K and an inlet stagnation pressure of 1 bar. Assume that the total-to-total isentropic efficiency of the compressor (i.e. both impeller and diffuser) is 0.8. For all parts of this question the working fluid is air with specific heat capacity at constant pressure $c_p = 1005\ \mathrm{J\ kg^{-1}\ K^{-1}}$ and ratio of specific heat capacities $\gamma = 1.4$.

(i) Calculate the stagnation temperature at the compressor exit. [10%]

(ii) The impeller blades are backswept at 45° at their exit and have a slip factor of 0.85. The absolute Mach number of the flow leaving the impeller is 1.0 and the absolute flow direction is 70° to the radial direction. Sketch the velocity triangle at the impeller exit and calculate the impeller tip speed and radius. [20%]

(iii) Discuss how a designer might reduce slip, noting any potential disadvantages. [10%]

(b) The compressor of part (a) has a vaned diffuser in order to recover the kinetic energy of the impeller exit flow.

(i) Discuss the benefits and limitations of common diffuser geometries with the help of annotated sketches. [15%]

(ii) The Mach number at the leading edge of the diffuser blades is 0.8. The velocity leaving the diffuser blades is negligible. Assuming the diffuser has a pressure recovery coefficient of 0.7, calculate the static pressure at the diffuser blade leading edge. [20%]

(iii) Explain why a pressure rise occurs in the vaneless space between the impeller trailing edge and the diffuser blade leading edge in a typical radial machine. The compression in the vaneless space may be assumed to have a polytropic efficiency of 0.7 based on the static flow conditions. Calculate the static pressure at the impeller trailing edge. [15%]

(c) Calculate the total-to-total efficiency of the impeller alone based on absolute stagnation conditions at its trailing edge. [10%]

2 The axial compressor of an industrial gas turbine engine is being designed to achieve an overall stagnation pressure ratio of 18. All stages are to have a mean blade speed of 245 m s^{-1} and a stage-loading coefficient of 0.4. You should assume that the flow has the properties of air throughout, with specific heat capacity at constant pressure c_p = 1005 J kg^{-1} K^{-1} and ratio of specific heat capacities γ = 1.4. You should neglect the mass flow of fuel.

(a) If the total-to-total isentropic efficiency of the complete compressor is 0.87, how many stages are needed to achieve the overall stagnation pressure ratio for an inlet stagnation temperature of 285 K ? Using this many stages, calculate the total-to-total isentropic efficiency required to achieve the overall stagnation pressure ratio if the inlet stagnation temperature is 310 K. [25%]

(b) The compressor is driven by an axial turbine. All stages of the turbine have −20° of swirl at entry and exit from the stage, a stator exit flow angle of 72° and a rotor exit relative flow angle of −67°. If the axial velocity and mean blade speed are constant through the turbine, sketch the velocity triangles and calculate the relative flow angle at rotor inlet. [25%]

(c) The turbine inlet stagnation temperature is 1600 K and the blade speed is chosen so that the Mach number of the flow leaving the first stator is 0.75. Calculate the turbine blade speed. Hence obtain the stagnation temperature drop across each turbine stage. The turbine drives the compressor with negligible mechanical losses. It also drives a generator, which absorbs twice the work required to drive the compressor. Estimate the number of turbine stages required when the compressor inlet stagnation temperature is 285 K and the compressor total-to-total isentropic efficiency is 0.87. [30%]

(d) Explain why the Mach numbers in the second and following stages of the turbine are higher than those in the first stage. What problems should the designer be aware of in the following stages? [20%]

3 A single shaft turbojet engine has a multistage axial compressor and a single stage axial turbine. It is operated on a stationery test-bed at sea level with an inlet stagnation pressure p_{02} = 101 kPa and an inlet stagnation temperature T_{02} = 288 K . At the design operating point the compressor pressure ratio p_{03}/p_{02} = 6.0 and the stagnation temperature at turbine inlet T_{04} = 1200 K . The compressor polytropic efficiency η_{pc} = 0.85 and the turbine polytropic efficiency η_{pt} = 0.9.

Use c_p = 1005 J kg^{-1} K^{-1} and γ = 1.4 for the air flowing through the compressor. Use c_{pe} = 1244 J kg^{-1} K^{-1} and γ_e = 1.3 for the exhaust gas flowing through the turbine and nozzle. For all operating points, assume that both the turbine nozzle guide vanes and the propulsive nozzle are choked. The stagnation pressure drop through the combustor and the mass flow rate of fuel can be neglected.

(a) Using the conditions at the design operating point, find the value of k , where $k = 1 - T_{05}/T_{04}$ and T_{05} is the turbine exit stagnation temperature. Explain why k remains constant as the engine fuel flow is varied. [25%]

(b) Show that as the engine fuel flow is varied the compressor pressure ratio and the engine non-dimensional mass flow are given by

$$\frac{p_{03}}{p_{02}} = \left(1 + k\frac{c_{pe}}{c_p}\frac{T_{04}}{T_{02}}\right)^{\eta_{pc}\,\gamma/(\gamma-1)} \qquad \text{and} \qquad \frac{\dot{m}\sqrt{c_p T_{02}}}{A_2\, p_{02}} = C\,\frac{p_{03}}{p_{02}}\sqrt{\frac{T_{02}}{T_{04}}}$$

where $\dot{m}$ is the mass flow rate of air through the compressor, A_2 is the compressor inlet area and C is a constant. [15%]

(c) Find the stagnation temperature drop through the turbine and the compressor pressure ratio when the fuel flow is reduced such that the stagnation temperature at turbine inlet is 950 K . At this condition, treating the flow in the propulsive nozzle as isentropic, find the exhaust jet velocity and the percentage reduction in gross thrust relative to the design operating point. [45%]

(d) If the fuel flow is reduced further the compressor becomes unstable. Explain the likely form and location of this instability. Describe two practical solutions that can be applied to avoid multistage compressor instability at reduced rotational speed. [15%]

END OF PAPER

4A3 Exam Paper Answers 2015

Q1. (a) 482.3 K, 581.9 m s^{-1} , 0.222 m

(b) 2.93 bar, 2.51 bar

(c) 0.925

Q2. (a) 18 stages, 0.925

(b) 47.4^{0}

(c) 351.0 m s^{-1} , 211.8 K, 6 stages

Q3. (a) 0.16

(c) 152 K , 4.463, 528 m s^{-1} , 37.2 %