ENGINEERING TRIPOS PART IIB

Wednesday 27 April 2005 2.30 to 4

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 the question is indicated in the right margin.*

There are no attachments.

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

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1 (a) Explain why it is generally considered undesirable for the exit Mach number from a turbine blade to exceed unity. What geometric features of the blades most influence the penalty associated with high exit Mach numbers? [10%]

(b) Sketch the velocity triangles for a 50% reaction and a zero reaction (impulse) axial flow turbine stage, both with zero swirl at entry and exit to the stage. If the exit Mach number from any blade row may not exceed unity which of the two designs will give the greatest enthalpy drop per stage? [15%]

(c) An axial flow turbine has an exit Mach number = 1.0 from it stator blades and a *relative* exit Mach number = 1.0 from its rotor blades. The stator exit flow angle is 75° and the flow coefficient is 0.4. The mean radius and the axial velocity are both constant through the stage. Sketch the velocity triangles and calculate the relative flow angle and relative Mach number of the flow at entry to the rotor blades. Hence obtain the ratio of the relative velocity at rotor exit to the absolute velocity at stator exit, the rotor relative exit flow angle and the stage loading coefficient. [30%]

(d) Show that for small values of the stagnation pressure loss coefficient Y_p the entropy increase in a turbine blade row operating in air with an exit Mach number of unity is given by

$$\Delta s = R\, Y_p \left(1 - \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma}{(\gamma-1)}} \right)$$

where R is the gas constant and γ is the specific heat ratio. [15%]

(e) Hence calculate the total to total efficiency of the above turbine when the stagnation pressure loss coefficient of each blade row is 0.1. Assume that the working fluid has the properties of air. [30%]

2 (a) Sketch the mass flow rate to pressure ratio characteristics of a typical high-speed axial flow compressor. Explain why the range of mass flow rates achievable at a fixed blade speed in a multistage high-speed compressor is so small. [10%]

(b) The compressor of gas turbine power plant is designed to pass a mass flow rate of 100 kg s^{-1} at a site where the atmospheric pressure is 1 bar and the atmospheric temperature is 15°C. What mass flow rate will it pass when operated under dynamically similar conditions at a site where atmospheric pressure is 0.9 bar and atmospheric temperature = 35°C? What will be the ratio of powers required to drive the compressor at the two sites? [15%]

(c) Explain how and why the useful operating range of inlet angles of a compressor blade row varies with its inlet Mach number. [10%]

(d) A high speed axial compressor operates with its first rotor blade choked at its throat which has area A_t. There are no inlet guide vanes and the *absolute* stagnation pressure P_{01} and *absolute* stagnation temperature T_{01} of the entering flow remain constant over the operating range. Neglecting all losses before the throat of the first rotor show that the mass flow passed by the compressor is given by

$$\dot{m} = F(1)\ A_t \frac{P_{01}}{\sqrt{C_p T_{01}}} (1+G)^{\frac{\gamma+1}{2(\gamma-1)}}$$

where F(1) is the dimensionless mass flow function $\dfrac{\dot{m}\sqrt{C_p T_{01}}}{A\,P_{01}}$ evaluated at Mach 1,

$G = \dfrac{U^2}{2C_p T_{01}}$ and U is the blade speed. All other symbols have their usual meaning. [30%]

By equating this mass flow rate to the inlet mass flow rate obtain an expression for the dimensionless mass flow function of the inlet flow based on M_x, the Mach number of the absolute flow at inlet to the rotor. [15%]

Hence evaluate the absolute Mach number M_x of the inlet flow and the inlet relative flow angle when the ratio of throat width to blade pitch is 0.5 and $U/\sqrt{C_p T_{01}} = 0.6$. Take the specific heat ratio γ to be 1.4. [20%]

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3 (a) Describe how and why the location at which stall in a high speed axial compressor is first initiated varies with the compressor speed. [15%]

(b) A multistage compressor is used as part of a turbojet engine. Both the turbine stator nozzle and the propulsion nozzle are choked. Show that on the working line of the turbojet the ratio of the combustor exit stagnation temperature to the turbojet inlet stagnation temperature can be written as

$$\frac{T_{03}}{T_{01}} = \frac{1}{1-\left(\frac{A_T}{A_N}\right)^k} \quad \frac{c_p}{c_{pe}}\left(\frac{T_{02}}{T_{01}}-1\right)$$

where

$$k = \frac{1}{\frac{\gamma_e}{\eta_p(\gamma_e - 1)} - 0.5}$$

T_{01}, T_{02} and T_{03} are the turbojet inlet, compressor exit and combustor exit stagnation temperatures respectively. c_p and c_{pe} are the specific heats at constant pressure of air and combustion products. A_T and A_N are the turbine stator nozzle and propulsion nozzle throat areas. γ_e is the ratio of specific heat capacities of the combustion products and η_p is the polytropic efficiency of the turbine. Neglect the mass flow of fuel. [35%]

(c) The compressor of the turbojet in part (b) has a design pressure ratio of 10 and stalls at a pressure ratio of 6. Calculate the percentage change in the compressor inlet mass flow function between the design point and the stall point. The compressor polytropic efficiency at the design point and at the stall point is 0.9 and 0.75 respectively. The turbine stators and the propulsion nozzle remain choked at all operating conditions. [25%]

(d) In order to investigate its stall behaviour, the compressor in part (c) is disconnected from the turbojet and is tested alone. The compressor is powered by a variable speed electric motor and a choked nozzle is connected to its exit. The throat area of the exit nozzle is chosen so that the compressor passes through the same stall point on its characteristic as when it is connected to the turbojet. What is the ratio of the new exit nozzle throat area to the turbojet turbine stator nozzle area? The ratio of

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turbine stator nozzle throat area to propulsion nozzle throat area of the turbojet is 0.4 and the polytropic efficiency of the turbine is 0.9. [25%]

For air take the specific heat ratio γ to be 1.4. For combustion products take γ to be 1.333. The value of the dimensionless mass flow function at M=1 for combustion products with $\gamma = 1.333$ is 1.3468 .

END OF PAPER