ENGINEERING TRIPOS PART IIB
ELECTRICAL AND INFORMATION SCIENCES TRIPOS PART II

Wednesday 23 April 2003 2-30 to 4-00

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*

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 The axial compressor of a jet engine is being designed to have an overall stagnation pressure ratio of 15. All stages are to have a mean blade speed of 250 m/s and a stage loading coefficient of 0.4. The total to total isentropic efficiency is estimated to be 0.85. Assuming that the flow has the properties of air throughout and neglecting the mass flow of fuel.

a) Calculate the number of compressor stages needed when the inlet stagnation temperature is 290 K. Also calculate the polytropic efficiency of the compressor. [25%]

b) The compressor is driven by an axial turbine all stages of which have -20° of swirl at entry and exit from the stage, a stator exit flow angle of 70° and a rotor exit relative flow angle of -65°. If the axial velocity and mean blade speed are constant through the turbine sketch the velocity triangles for the flow through a turbine stage and calculate the relative flow angle at rotor inlet. [25%]

c) The turbine inlet stagnation temperature is 1500 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. Estimate the number of turbine stages required to drive the compressor. [30%]

d) Explain why the Mach numbers in the second stage of the turbine are higher than those in the first stage and calculate the Mach number of the flow leaving the second stator. [20%]

2 Define the term "diffusion factor" as used in axial compressor design and describe how it is used to fix the blade pitch to chord ratio. Explain why, for a fixed blade geometry and fixed pitch to chord ratio, the diffusion factor tends to increase with increasing blade inlet Mach number. [20%]

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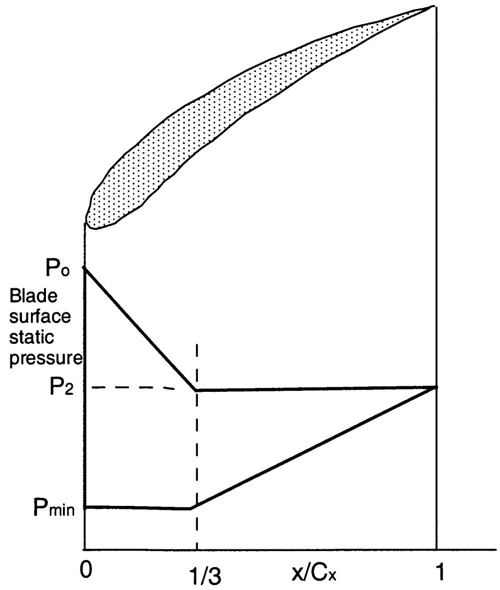

Figure 1.

Figure 1 above shows an approximation to the static pressure distribution on a typical compressor blade. The blade surface pressure is shown plotted against axial distance x expressed as a fraction of the axial chord C_x, P_o is the stagnation pressure, P_2 the blade exit static pressure and P_{min} the minimum pressure on the suction surface. Show that in loss-free incompressible flow the tangential force exerted by the blade on the flow through each blade passage is given by

$$F_\theta = \rho V_2^2 \left(\frac{1}{3(1-D)^2} - 0.75 \right) C_x$$

where D is the diffusion factor, V_2 is the blade exit velocity and ρ is the fluid density. [30%]

By equating this force to the change in momentum of the flow, obtain an expression for the blade pitch to axial chord ratio, p/C_x, as a function of the diffusion factor and the blade inlet and outlet angles. Assume that the axial velocity V_x is constant. [25%]

Use the expression for F_θ given above to obtain an expression showing how the stage loading coefficient varies with inlet angle and diffusion factor for blades with exit flow angle = 0°. [25%]

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3 A two-shaft turbojet is used to power the old Comet 4 transport plane. The plane cruises at a Mach number of 0.87 at an altitude of 10,000 m. At cruise the pressure ratio of the low-pressure compressor is 1.8, the pressure ratio of the high-pressure compressor is 20.0 and the combustion chamber exit temperature is 1200 K.

The polytropic efficiency of all compressor, fan and turbine stages is 0.9 and the propulsion nozzles may be assumed to be isentropic.

Take $\gamma = 1.4$ for air and $\gamma = 1.3$ for products of combustion and $R = 287$ J $Kg^{-1}K^{-1}$ for all gases. Neglect the mass flow of fuel, pressure losses in the combustion chamber and mechanical losses:

(a) Calculate the jet velocity and the propulsive efficiency of the turbojet. [30%]

(b) It has been decided to replace the engines of the Comet 4 with a more modern design. In order to reduce noise and improve fuel consumption the exhaust jet velocity is to be reduced. The new design of engine is a two-shaft turbofan with the low-pressure compressor replaced by a fan. If the speed of the by-pass jet at the above cruise condition is 350 ms^{-1}, show that the fan stagnation pressure ratio is approximately 1.41. [25%]

(c) The overall compression ratio of the air passing through the core of the new engine is the same as that of the turbojet. The combustor exit temperature is also the same and the speed of the core jet is 350 ms^{-1}. Assuming that the *static* temperature of the core jet remains the same as in part (a) calculate the by-pass ratio of the engine. [25%]

(d) The Comet 4 has a total net thrust of 120 kN from its four turbojet engines when cruising at the above conditions. These are to be replaced with two turbofan engines. Calculate the ratio of the mass flow rates of air passing through the two types of engine. What would be the mass flow rate through the new engines when tested on a static test bed at sea level, under dynamically similar conditions to those at cruise ? [20%]

END OF PAPER