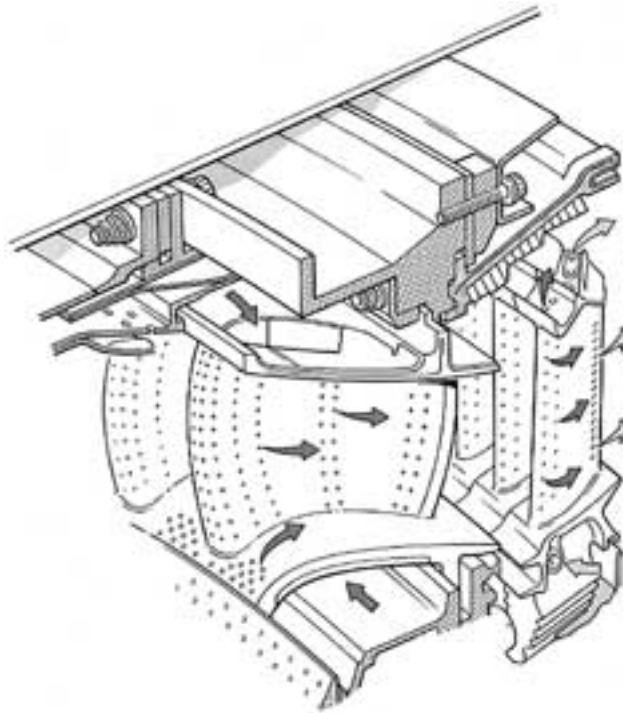


## Unified Engineering Spring Term 2004

### Problem P7. (Propulsion) (L.O. F)

For this problem, consider the design of a turbojet engine for a supersonic aircraft. Assume the engine uses conventional hydrocarbon fuels ( $h = 42.8 \text{ MJ/kg}$  and that  $\gamma = 1.4$  and  $c_p = 1000 \text{ J/(kg-K)}$ ).

There are two important material temperature limits for gas turbine engines. The first is the temperature at the inlet to the turbine ( $T_{T4}$ ) which may range as high as 2000K. The second is the temperature at the exit of the compressor ( $T_{T3}$ ) which is typically limited to less than 1000K. [The turbine can be run in conditions well above the melting temperature of the metal because air from the exit of the compressor is used to "cool" the turbine. This air is pumped through the turbine blades and forced out small holes in the surface forming a protective film (see picture below from Rolls-Royce, The Jet Engine). Due to their very small size, it is much harder to cool the compressor blades, so no active cooling is currently used.]



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- For a turbojet engine with a compressor pressure ratio of 25, and a turbine inlet temperature of 1800K, under what flight conditions (Mach number) would the compressor inlet temperature limit be exceeded? Consider performance at sea-level ( $T_{atm}=300\text{K}$ ) and at 10km ( $T_{atm} = 216\text{K}$ )
- At a constant flight Mach number of 1.2, how much do the thermal and propulsive efficiency change between altitudes of sea-level and 10km? Explain this behavior.
- If you were forced to reduce the turbine inlet temperature to 1400K because the turbine durability

group was having difficulty designing the blades to withstand temperatures of 1800K, how would it impact the performance of the vehicle (consider  $M=1.2$  and 10km only)?