Engine operates at a free stream Mach number, $M_\infty = 0.8$.
Cruise Altitude is in the stratosphere, 11 km so $T_\infty = 216.65$ K.
The design turbine inlet temperature, $T_{04} = 1944$ K.
The design compressor ratio, $\pi_c = 20$.
Relevant area ratios are $A_2 / A_4^{*} = 10$ and $A_2 / A_{1\text{throat}} = 1.2$.
Inlet throat area $A_{1\text{throat}} = 20 \text{ cm}^2$
Assume the compressor, burner and turbine all operate ideally.
Nozzle is of a simple converging type with choked throat, $A_8^{*} = A_{\text{exit}}$
Stagnation pressure losses due to wall friction in the inlet and nozzle are negligible.

**CALCULATE**

a) Correct Compressor Massflow and $M_2$ at compressor face
b) Normalized exit pressure thrust, momentum thrust, and total thrust
c) Velocity ratio across Engine $V_{\text{exit}} / V_\infty$
d) Mach number at diffuser throat, $M_{1\text{throat}}$
e) Inlet capture area
f) Total Thrust, Isp, TSFC
• Now allow an expandable **Nozzle where**, \( A_{exit} \geq A^*_{8} \)

→ **CALCULATE**

→ a) Optimal expansion ratio for nozzle \( A_{exit}/A^*_{8} \)

→ b) Velocity ratio across Engine \( V_{exit}/V_\infty \)

→ c) thrust, Isp, TSFC of optimal nozzle,

→ d) Assuming the same combustor temperature and inlet throat area as previous page

→ At what compressor demand \( \pi_c \) does the inlet throat choke (\( @ A_{1\text{throat}} \))

→ Plot the Compressor operating line → \( \pi_c \) vs corrected massflow, \( f(M_2) \) for \( 1 \leq \pi_c < \pi_c \) @ choke

→ Plot the capture area \( A_\infty \) vs corrected massflow, \( f(M_2) \) for \( 1 \leq \pi_c < \pi_c \) @ choke