Problem 1 (30 points): A turbojet is flying at $M = 1.3$ at an altitude where $T_a = 200$ K. At the exhaust $p_e = p_a$, $M_e = 1.6$, and $T_e = 1100$ K. The fuel-to-air ratio is $f = 0.06$ and the heat of reaction is $Q_R = 45,000$ kJ/kg. Assuming that the gas behaves as air at normal ambient conditions (i.e., $\gamma = 1.4$ and $R_g = 287$ J/(kg K)), compute:

- The specific thrust $T/\dot{m}_a$.
- The TSFC.
- The propulsion efficiency.
- The thermal efficiency.
- The overall efficiency.

Problem 2 (35 points): The symmetric triangular airfoil of chord $c$ and angle $\alpha = 30^\circ$ shown in the figure flies at $M_1 = 3.4$ with zero angle of attack, so that the flow on the underside of the airfoil remains unperturbed.

- Obtain the Mach number $M_2$ behind the leading oblique shock, along with the corresponding pressure relative to the ambient value $p_2/p_1$.
- Obtain the Mach number $M_3$ and relative pressure $p_3/p_1$ behind the expansion.
- Compute the lift and drag coefficients $c_l = L/(\frac{1}{2}\rho_1U_1^2c)$ and $c_d = D/(\frac{1}{2}\rho_1U_1^2c)$.

Problem 3 (35 points): A stream with stagnation pressure $p_0$ and stagnation density $\rho_0$ discharges to an open atmosphere at pressure $p_a < p_0$ through a convergent-divergent nozzle with throat-to-exit area ratio $A_t/A_e = 0.353$.

1. Obtain the value of $p_a = p_{cn}$ for which the flow is everywhere subsonic, except at the throat, where it is sonic, giving the result in the form $p_{cn}/p_0$.
2. Calculate the value of $p_a = p_{ns}$ for which a normal shock is found at the nozzle exit section, giving the result in the form $p_{ns}/p_0$.
3. Find the value of $p_a = p_{ns}$ for which a normal shock is found at the nozzle exit section, giving the result in the form $p_{ns}/p_0$.
4. For $p_a = 0.1 \times p_{ns}$, obtain the values of $M_e$ and $M_t$ as well as the mass flow rate $\dot{m}/(\sqrt{\gamma \rho_0 p_0 A_t})$. The expansion formed at the exit section is a Prandtl-Meyer expansion in the vicinity of the nozzle rim. Determine the Mach number found immediately downstream $M_d$ as well as the deflection angle $\theta$.
5. For $p_a = (p_{cn} + p_0)/2$ obtain the values of the Mach number at the exit and at the throat $M_e$ and $M_t$. Determine the mass flow rate $\dot{m}$, giving the result in the form $\dot{m}/(\sqrt{\gamma \rho_0 p_0 A_t})$.
6. For $p_a = (p_{ns} + p_0)/2$, obtain the values of $M_e$ and $M_t$ as well as the mass flow rate $\dot{m}/(\sqrt{\gamma \rho_0 p_0 A_t})$. An oblique shock is formed at the exit section. Determine the Mach number immediately downstream $M_d$ as well as the deflection angle $\delta$. 
Problem 1 (30 points): A turbojet is flying at $M = 1.3$ at an altitude where $T_a = 200$ K. At the exhaust $p_e = p_a$, $M_e = 1.6$, and $T_e = 1100$ K. The fuel-to-air ratio is $f = 0.06$ and the heat of reaction is $Q_R = 45,000$ kJ/kg. Assuming that the gas behaves as air at normal ambient conditions (i.e., $\gamma = 1.4$ and $R_g = 287$ J/(kg K)), compute:

- The specific thrust $T/\dot{m}_a$.
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- Obtain the Mach number $M_2$ behind the leading oblique shock, along with the corresponding pressure relative to the ambient value $p_2/p_1$.
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Problem 3 (35 points): A stream with stagnation pressure $p_0$ and stagnation density $\rho_0$ discharges to an open atmosphere at pressure $p_a < p_0$ through a convergent-divergent nozzle with throat-to-exit area ratio $A_t/A_e = 0.353$.

1. Obtain the value of $p_a = p_{ch}$ for which the flow is everywhere subsonic, except at the throat, where it is sonic, giving the result in the form $p_{ch}/p_0$.
2. Calculate the value of $p_a = p_{sj}$ for which the nozzle discharge occurs as a supersonic jet with $p_e = p_a$, giving the result in the form $p_{sj}/p_0$.
3. Find the value of $p_a = p_{ns}$ for which a normal shock is found at the nozzle exit section, giving the result in the form $p_{ns}/p_0$.
4. For $p_a = 0.1 \times p_{ch}$, obtain the values of $M_e$ and $M_t$ as well as the mass flow rate $\dot{m}/(\sqrt{\gamma \rho_0 p_0 A_t})$. The expansion formed at the exit section is a Prandtl-Meyer expansion in the vicinity of the nozzle rim. Determine the Mach number found immediately downstream $M_d$ as well as the deflection angle $\theta$.
5. For $p_a = (p_{ch} + p_0)/2$ obtain the values of the Mach number at the exit and at the throat $M_e$ and $M_t$. Determine the mass flow rate $\dot{m}$, giving the result in the form $\dot{m}/(\sqrt{\gamma \rho_0 p_0 A_t})$.
6. For $p_a = (p_{sj} + p_{ns})/2$, obtain the values of $M_e$ and $M_t$ as well as the mass flow rate $\dot{m}/(\sqrt{\gamma \rho_0 p_0 A_t})$. An oblique shock is formed at the exit section. Determine the Mach number immediately downstream $M_d$ as well as the deflection angle $\delta$. 

\[\text{Diagram of the airfoil with labeled angles and Mach numbers}\]