AE 234 Aircraft Propulsion
Tutorial 2

1. An airbreathing engine flies at Mach $M_a = 2.0$ at an altitude where the ambient temperature is $T_a = -50^oC$ and ambient pressure is $p_a = 10$ kPa. The airflow rate to the engine is 25 kg/s. The fuel flow rate is 3% of airflow rate and has a heating value of $42,800$ kJ/kg. Assuming the exhaust speed is $V_e = 1050$ m/s, and the nozzle is perfectly expanded, i.e., $p_e = p_a$, calculate

   - ram drag ($\mathcal{D}_{ram} = \dot{m}_a V_a$) and gross thrust ($\mathcal{T}_g = \dot{m} V_e + (p_e - p_a) A_e$) in kN
   - uninstalled thrust ($\mathcal{T} = \mathcal{T}_g - \mathcal{D}_{ram}$) in kN
   - TSFC in mg/N $-$ s and kg/hr $-$ kgf
   - thermal, propulsive and overall efficiencies

   Gross Thrust: 27.04 kN, Ram Drag: 14.97 kN, Net Thrust: 12.07 kN
   TSFC is 62.15 mg/N-s, $\eta_p = 0.73$, $\eta_{th} = 0.29$, $\eta_{ov} = 0.21$

2. For a propeller driven aircraft, we have the following parameters:

$$\text{Gain in flow kinetic power: } \Delta\dot{\mathcal{E}} = 0.5\left(\dot{m}V_e^2 - \dot{m}_a V_a^2\right)$$
$$\text{Propeller Efficiency: } \eta_{prop} = \frac{\mathcal{T}_{prop} V_a}{\mathcal{P}_{s,prop}}$$
$$\text{Propulsive Efficiency: } \eta p = \frac{\mathcal{T} V_a}{\mathcal{P}_{s,prop} + \Delta\dot{\mathcal{E}}}$$

   A turbo-propeller-driven aircraft is flying at $V_a = 150\ m/s$ and has a propeller efficiency of $\eta_{pr} = 0.75$. The propeller thrust is $\mathcal{T}_{prop} = 5000\ N$ and the airflow rate through the engine is $5\ kg/s$. The nozzle is perfectly expanded and produces $1000\ N$ of gross thrust. Calculate

   - shaft power delivered to the propeller (in kW)
   - the nozzle exit velocity in m/s (neglect f)
   - Compare the gain in flow kinetic power of the exhaust with the shaft power $\mathcal{P}_s$
   - Obtain the propulsive efficiency

   Shaft Power is 1000.0 kW, Exhaust velocity is 350.0 m/s
   Gain in kinetic power is 250.0 kW, $\eta_p = 0.72$

3. A turbojet-powered aircraft cruises at $V_a = 300\ m/s$ while the engine produces an exhaust speed of $600\ m/s$. The air mass flow rate is $100\ kg/s$ and the fuel mass flow rate is $2.5\ kg/s$. The fuel heating value is $\mathcal{Q}_R = 42\ MJ/kg$. Assuming that the nozzle is perfectly expanded, calculate

   - engine ram drag, gross thrust and uninstalled thrust in kN
   - engine thrust-specific fuel consumption (TSFC) in $mg/s - kN$

- engine thermal and propulsive efficiencies
- aircraft range R for $L/D = 10$ and the $M_{init}/M_{final} = 1.25$

Ram Drag is 30.0 kN, Gross Thrust is 61.5 kN, Net Thrust is 31.5 kN
TSFC $=$ 79 mg/N-s , $\eta_p = 0.667$, $\eta_{th} = 0.129$, Range is 860.0 km

4. Derive the expression for the range of an aircraft if its weight decreases from $\mathcal{W}_{ini}$ to $\mathcal{W}_{fin}$ during cruise. Consider the following flight scenarios:
   - Constant speed, attitude
   - Constant attitude, altitude
   - Constant altitude, speed
5. The unit of lbm/lbf $-$ hr is common for TSFC. Show that 1lbm/lbf $-$ hr $= 28.3$ gm/kN $-$ s
6. Verify that the fuel-air ratio for the stoichiometric combustion of a hydrocarbon fuel of the approximate composition $C_{17n}H_{36n}$ is nearly 0.068.
7. We wish to investigate the range of a slender supersonic aircraft where its lift-to-drag ratio as a function of flight mach number is described by

$$\frac{\mathcal{L}}{\mathcal{D}} \approx 3\frac{M_a + 3}{M_a}$$

Varying the TSFC between $1.0 - 2.0$ lbm/hr$-$lbf, plot the variation of range, R, for the fligh mach number ranging between 2.0 and 4.0. You can assume that speed of sound is $333 m/s$, and the aircraft initial to final weight ratio is $\mathcal{M}_0/\mathcal{M}_f = 2$.

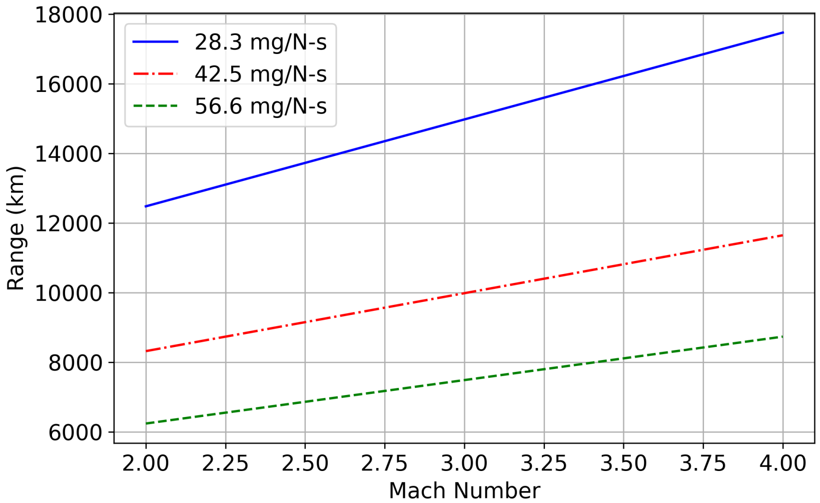

Figure 1: Problem 7: Range of supersonic bomber