APPLIED THERMODYNAMICS

Gas Turbine Engines
(Module IV)

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List of Topics

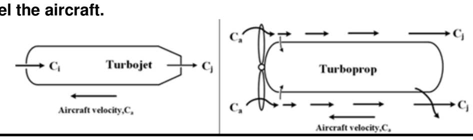
Lecture 6

Aircraft Propulsion Cycle – II

- Jet propulsion
- Ramjet propulsion
- Efficiencies of turbojet engine
- Pressure thrust on a turbojet engine
- Combustion in gas turbine engine

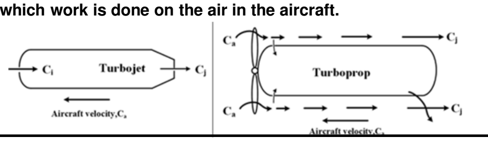
Jet Propulsion

- The family of propulsion engines are, piston engine, turbo-prop engine, turbo-jet engine, turbo-fan engine, ram-jet engine and scram-jet engine in the order of increasing speed (subsonic M < 1 to hypersonic M > 5).
- Broadly, the aircraft propulsion is achieved by using a heat engine to drive a propeller or by allowing high-energy fluid to expand and leave the aircraft in a rearward direction as a high-velocity jet.
- In propeller type engine, large slug of air mass is used by the propeller and moderate velocity is achieved in backward direction relative to the aircraft.
- In a jet engine, the aircraft induces comparatively small air flow and gives high velocity jet backward relative to the aircraft.
- For both cases, the rate of change of momentum air provides reactive thrust to propel the aircraft.



Jet Propulsion

- Consider a simple propeller/jet engine, for which the velocity of jet backwards relative to aircraft is C_j and velocity of aircraft is C_a.
- If the jet leaves the engine at atmospheric pressure, then there is no thrust due to pressure forces. So, the available propulsive force is mainly due to momentum of the stream.
- In order to keep the aircraft moving at constant velocity, the thrust equivalent work must be done against frictional resistance and drag.
- The net work output from the engine is given by the increase in kinetic energy and it is used in two ways: (a) to provide thrust work; (b) to impart kinetic energy (air which is initially at rest, receives absolute velocity of C_j-C_a).
- Now, the propulsive efficiency can be defined as the ratio of thrust work to the rate at which work is done on the air in the aircraft.



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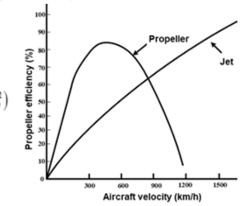
Jet Propulsion

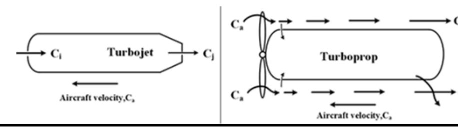
- It can be noticed that the propulsive efficiency increases with increase in aircraft velocity.
- In a propeller driven aircraft, the propulsive efficiency is higher for propeller driven aircraft in the initial stage up to sonic velocity.
- Jet driven aircraft, becomes superior at higher speed range (above 850km/hr; Mach number 0.7).

Thrust power per unit mass flow rate, $F = C_a\left(C_j - C_a\right)$

Work output from engine, $W = C_a\left(C_j - C_a\right) + \frac{1}{2}\left(C_j - C_a\right)^2 = \frac{1}{2}\left(C_j^2 - C_a^2\right)$

Propulsive efficiency, $\eta_p = \frac{2C_a\left(C_j - C_a\right)}{C_j^2 - C_a^2} = \frac{2C_a}{C_j + C_a}$





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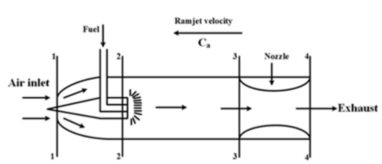
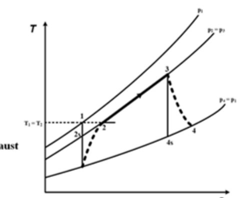
Ram Propulsion

- One of the simplest from of jet engine is “ramjet” in which compressed air is used for conversion of kinetic energy of atmospheric air relative to the aircraft. It is commonly known as “ram effect”.
- Fuel is burnt in the compressed air stream at constant pressure and the hot gas is allowed to expand in the nozzle giving rise to high velocity jet backwards relative to aircraft.
- The intake duct of ramjet engine acts as a diffuser and the kinetic energy of air is used for stagnation temperature rise after diffusion.





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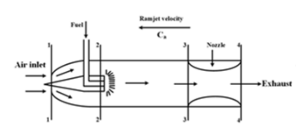
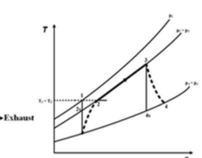
Ram Propulsion

- The air attains stagnation pressure when the process is isentropic while for irreversible adiabatic process, the air receives less pressure than its stagnation value. But, the stagnation temperature does not change.
- An aircraft powered by ramjet engine requires auxiliary power supply to attain velocity necessary for starting of engine through ram compression.
- The total pressure is the pressure that the air attains when the diffusion process is isentropic. When the pressure is irreversible (but still adiabatic), the total pressure will be less than the isentropic value.
- Since, the available kinetic energy remains the same irrespective of the process, the temperature change remains the same.





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Ram Propulsion

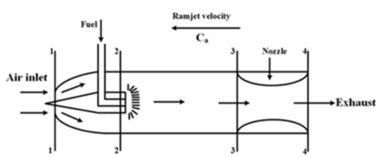
Thrust power per unit mass flow rate, $F = C_a\left(C_j - C_a\right)$

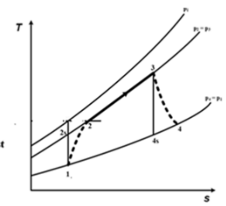
Work output from engine, $W = C_a\left(C_j - C_a\right) + \frac{1}{2}\left(C_j - C_a\right)^2 = \frac{1}{2}\left(C_j^2 - C_a^2\right)$

Propulsive efficiency, $\eta_p = \frac{2C_a\left(C_j - C_a\right)}{C_j^2 - C_a^2} = \frac{2C_a}{C_j + C_a}$

Isentropic flow : $c_p T_{01} = c_p T_1 + \frac{C_a^2}{2} \Rightarrow T_{01} - T_1 = \frac{C_a^2}{2c_p}$

$\frac{T_{01}}{T_1} = \left(\frac{p_{01}}{p_1}\right)^{\frac{\gamma-1}{\gamma}}$; $T_{02} = T_{01}$ & Intake efficiency, $\eta_i = \frac{T_{02s} - T_1}{T_{02} - T_1}$





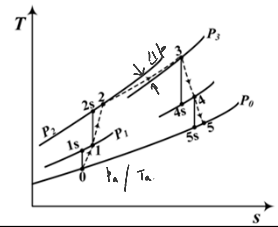
Efficiency of Turbojet Engine

- In a jet engine (turbojet), the kinetic energy of the incoming air can be used to obtain ram compression in the intake duct, thus raising the overall efficiency.
- For gas turbine concepts, it is important work on stagnation parameters because the velocity changes can not be neglected. Moreover, the measuring instruments of pressure and temperature detect the stagnation values.
- So, all the component efficiencies can be redefined w.r.t. stagnation values.
- For adiabatic flow, the total temperature remains constant for intake duct and jet pipe. There is a loss of pressure in the combustion chamber from 2 to 3.

Intake duct, $\eta_i = \frac{T_{01s} - T_0}{T_{01} - T_0}$; Compressor, $\eta_c = \frac{T_{02s} - T_{01}}{T_{02} - T_{01}}$

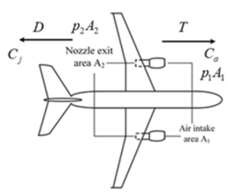
Turbine, $\eta_t = \frac{T_{03} - T_{04}}{T_{03} - T_{04s}}$; Jet pipe, $\eta_i = \frac{T_{04} - T_5}{T_{04} - T_{5s}}$

For intake duct and jet pipe, $T_0 = T_{01}$ & $T_{04} = T_{05}$



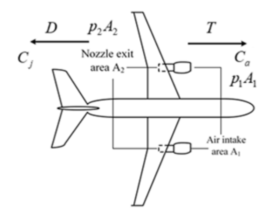
Pressure Thrust on a Turbojet Engine

- In jet engines, the gases expand to ambient pressure in a jet nozzle. But the back pressure is lower than the pressure of the gas at nozzle outlet. This phenomena is known as ‘under-expansion”.
- Due to the pressure difference, the aircraft experiences an additional thrust known as “pressure thrust”.
- Consider a schematic turbojet, with air intake of certain area A_1, inlet air pressure p_1, nozzle exit area A_2, pressure p_2 and atmospheric pressure p_a. By Newton’s law, the net force can be calculated through rate of change of momentum of working fluid in the direction of motion of the fluid.



Pressure Thrust for a Turbojet Engine

- The aircraft experiences, reactive force exerted from the fluid, drag force due to air resistance and the pressure force due to atmospheric pressure acting on the projected area in the direction of flight
- When the aircraft is flying at constant velocity, the net force will be zero and the total thrust required to overcome the total drag can be calculated.



Newton's law, $F = \dot{m}\left(C_j - C_a\right) - p_1A_1 + p_2A_2 = R$

Net force, $R - D + p_a\left(A_1 - A_2\right) = 0 \Rightarrow D = R + p_a\left(A_1 - A_2\right)$

Total thrust, $T = D \Rightarrow T = \dot{m}\left(C_j - C_a\right) + A_2\left(p_2 - p_a\right) - A_1\left(p_1 - p_a\right)$

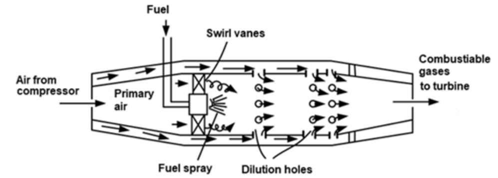
For subsonic aircraft, $p_1 = p_a \Rightarrow T = \dot{m}\left(C_j - C_a\right) + A_2\left(p_2 - p_a\right)$

Pressure Thrust for a Turbojet Engine

- At sea-level condition with same exit area of propulsion nozzle, the thrust produced will higher since the mass flow rate through the unit is increased.
- At higher altitude, the variable area nozzle (convergent-divergent) nozzle or under-expanded convergent nozzle is preferred.
- Another method of thrust boosting is achieved through "afterburning". Thermodynamically, it is equivalent to 'reheat'.
- Here, the fuel is sprayed into the gases leaving the turbine, thus increasing the jet velocity leaving the nozzle.
- About 50% increase in thrust can be achieved but at the cost of extra fuel. So, afterburners are normally incorporated during starting and as a reserve power source for thrust augmentation over a short periods.
- The other variants of aircraft engines to improve propulsive efficiency are, ducted fan engine and bypass engines by regulating core air flow through the main engine and through auxiliary component.

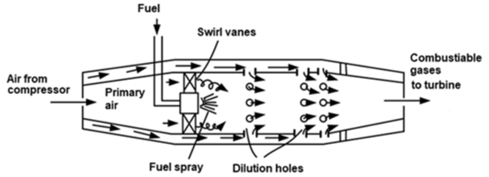
Combustion in Gas Turbine Engine

- In a closed cycle gas turbine unit, heat is transferred to the air in a heat exchanger, but in an open cycle, fuel is sprayed into the air continuously.
- The combustion in a gas turbine is continuous process unlike cyclic combustion in IC engines.
- One of the method of combustion system is 'can-type' in which air stream leaving from compressors is split into several streams and supplied to the combustion chamber.
- Combustion is initiated by electrical ignition and once the fuel starts burning, a flame is stabilized in the chamber.



Combustion in Gas Turbine Engine

- Some of the air from compressor is introduced directly into the fuel burner (called as 'primary air') and it represents 25% of the total air flow.
- The remaining air enters the annulus round the flame tube, thus cooling the upper portion of the flame tube and then enters to the combustion zone directly through dilution holes.
- The primary air forms a rich mixture and temperature is high in this zone while the air entering the dilution zone completes the combustion and helps to stabilize the flame in the high temperature zone.



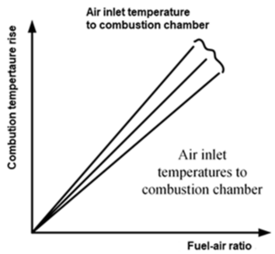
Combustion in Gas Turbine Engine

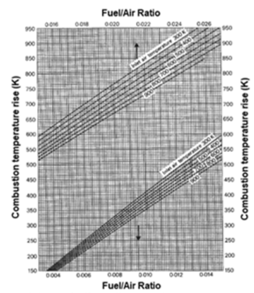
- The air-fuel ratio is of the order of 60 to 120 and the air velocity at the entry of combustion chamber is limited within 75 m/s.
- The rich and weak limit of flame stability is referred as flame blow-out. This instability results in rough running (known as 'screech phenomena') and consequent effect on life of combustion chamber.
- Since, high values of air-fuel ratio is used, the gases entering HP turbine contains high percentage of oxygen. So, reheating is performed between turbine stages or afterburning of fuel is used, for satisfactory usage of additional fuel requirement.
- The pressure loss in the combustion chamber is mainly due to non-adiabatic flow, turbulence and friction. However, at a given cross-section, the pressure is almost constant.
- This pressure loss due to friction is also refereed as "cold loss" and the pressure loss due to heating process is known as "fundamental loss".

Combustion in Gas Turbine Engine

Combustion efficiency and combustion intensity:

- For a given rise in temperature, "combustion efficiency" is the ratio of "theoretical fuel-air ratio to the actual fuel-air ratio".
- During a combustion process, the theoretical temperature rise is a function of calorific value of the fuel, fuel-air ratio and initial temperature of air. It can be plotted graphically.





Reference: Gas Turbine Theory, HIH Saravanamuttoo, H. Cohen and GFC Rogers, Pearson, India, 2012

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Combustion in Gas Turbine Engine

Combustion efficiency and combustion intensity:

- The comparison of combustion chambers of different size operating under different operating conditions is compared though the parameter "combustion intensity". Lower its value better is the design.
- In aircraft engines, its value is about 2 kW/m².atm while industrial plant has combustion intensity of 0.2 kW/m².atm.
- The aircraft gas turbine units use light petroleum distillate (known as high grade kerosene) having gross calorific value of 46.7 MJ/kg. The conventional gas turbine power generation units use natural gas (~ 42.4 MJ/kg).

$\eta_c = \dfrac{f_{theory}}{f_{actual}}$; f_{theory} : theoretical air-fuel ratio; f_{actual} : actual air-fuel ratio

$C_I = \dfrac{Q_R}{V_c\, p_{in}}$; Q_R : heat release rate; V_c : volume of chamber; p_{in} : inlet pressure

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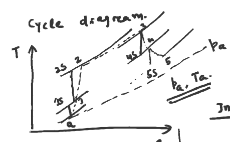
Numerical Problems

Q1. An aircraft driven by turbojet engine is flying at 220 m/s at an altitude where ambient conditions are 0.24 bar and -50ºC. The compressor pressure ratio is 10 and the maximum cycle temperature is 800ºC. Calculate, the thrust developed by the engine and specific fuel consumption by using the following data: Isentropic efficiency: intake (0.9), compressor (0.9), turbine (0.92), propelling nozzle (0.92), mechanical and combustion efficiency: 0.98, combustion chamber pressure loss: 0.15 bar, nozzle outlet area: 0.08 m².

Soln

Cycle diagram.



$\underline{K.E.} = \frac{1}{2} C_a^2 = 24200$ J/kg

$c_p(T_{01} - T_a) = 24200$

$\Rightarrow T_{01} = 247$ K

$T_a = -50°C = 223$ K

$C_a = 220$ m/s

$c_p = 1.005$ kJ/kg

$\gamma = 1.4$

$p_a = 0.24$ bar

Intake $\eta_i = \dfrac{T_{01s} - T_a}{T_{01} - T_a} = 0.9 \Rightarrow T_{01s} = 476$ K

$\dfrac{p_{01}}{p_a} = \left(\dfrac{T_{01s}}{T_a}\right)^{\frac{\gamma}{\gamma-1}} \Rightarrow p_{01} = 0.3317$ bar

Compressor

$\dfrac{p_{02}}{p_{01}} = 10$

$\Rightarrow \left(\dfrac{T_{02s}}{T_{01}}\right) = \left(\dfrac{p_{02}}{p_{01}}\right)^{\frac{\gamma-1}{\gamma}} \Rightarrow T_{02s} = 476$ K

$\eta_c = \dfrac{T_{02s} - T_{01}}{T_{02} - T_{01}} = 0.9$

$\Rightarrow T_{02} - T_{01} = 256$ K

$T_{02} = 503$ K

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Numerical Problems

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Turbine

$p_{03} = p_{02} - \Delta p_c = 3.317 - 0.15 = 3.167$ bar

$\underline{w_t} = C_{pg}(T_{03} - T_{04}) = \dfrac{w_c}{0.98} = \dfrac{C_{pa}(T_{02} - T_{01})}{0.98}$

$\Rightarrow T_{04} = 844.3$ K

$\eta_t = \dfrac{T_{03} - T_{04}}{T_{03} - T_{04s}} = 0.92 \Rightarrow T_{04s} = 824.4$ K

$\dfrac{p_{03}}{p_{04}} = \left(\dfrac{T_{03}}{T_{04s}}\right)^{\frac{\gamma}{\gamma-1}} \Rightarrow p_{04} = 1.095$ bar

$C_{pa} = 1.005$ kJ/kg

$C_{pg} = 1.148$ kJ/kg

$T_{03} = 800°C = 1073$ K

$\gamma = 1.33$

NPR $\dfrac{p_{04}}{p_a} = \dfrac{1.095}{0.24} = 4.55 \rightarrow$ Choking flow

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Numerical Problems

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Choking condition for nozzle.

$$\frac{p_c}{p_{04}} = \left(\frac{2}{\gamma+1}\right)^{\gamma-1} = 0.54. \qquad \frac{T_c}{T_{04}} = \frac{2}{\gamma+1} = \frac{2}{2.33}.$$

$$\Rightarrow p_c = \;= 0.5913 \text{ bar}. \qquad T_c = T_5 = 725 \text{ k}.$$

$$\eta_j = \frac{T_{04} - T_5}{T_{04} - T_{05s}} = 0.92. \Rightarrow T_{05s} = 714.3 \text{ k}. \qquad \gamma = 1.33.$$

$$\left(\frac{p_{04}}{p_5}\right) = \left(\frac{T_{04}}{T_{05s}}\right)^{\frac{\gamma}{\gamma-1}} \Rightarrow p_5 = 0.558 \text{ bar}.$$

$$R = 287 \text{ J/kg k}.$$

$$\rho_5 = \frac{p_5}{R T_5} = 0.268 \text{ m}^3/\text{kg}. \qquad c_5 = \sqrt{\gamma R T_5} = 526 \text{ m/s}.$$

$$\dot{m} = \rho_5 A_5 c_5 = 11.28 \text{ kg/s}.$$

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Numerical Problems

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Momentum thrust, $\dot{m}(c_j - c_a) = 11.28(528 - 220) = 3362 \text{ N}.$

Pressure thrust. $(p_5 - p_a) A = (0.558 - 0.24) \times 0.08 = 2544 \text{ N}.$

$$T = 5906 \text{ N}.$$

$$Q = \dot{m} c_{pg} (T_{03} - T_{02}) = 11.28 \times 1.148 (1073 - 503) = 7382 \text{ kJ/s}.$$

$$Q_{cv} = 43300 \text{ kJ/kg}.$$

$$\eta_c \dot{m}_f Q_{cv} = Q. \Rightarrow \dot{m}_f = 0.174 \text{ kg/s}.$$

$$SFC = \frac{\dot{m}_f}{T} = \frac{0.174}{5906} \times 3600. = 0.105 \text{ kg/hr.N}.$$

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Numerical Problems

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THANK YOU

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