UNIT 5

GAS TURBINES

Course Objective:

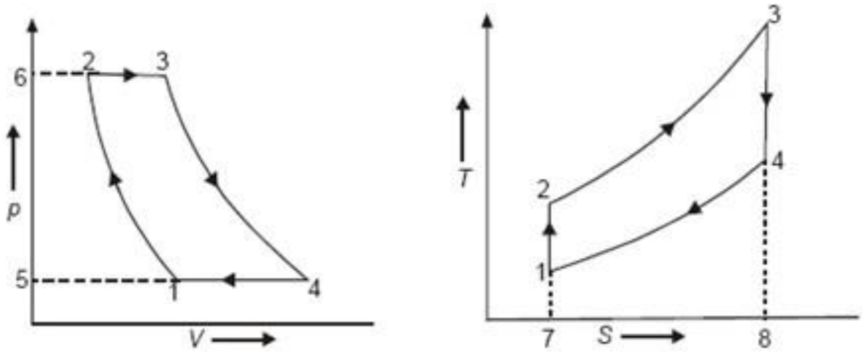
Student have knowledge of methods of analysis and design of complicated thermodynamic systems

Course Outcome:

To be able to recognize main and supplementary elements of turbines and define operational principles.

Brayton Cycle

Brayton cycle, popularly used for gas turbine power plants comprises of adiabatic compression process, constant pressure heat addition, adiabatic expansion process and constant pressure heat release process. A schematic diagram for air-standard Brayton cycle is shown in Fig. 4.1. Simple gas turbine power plant working on Brayton cycle is also shown here.



Brayton cycle on P–V and T–S diagram

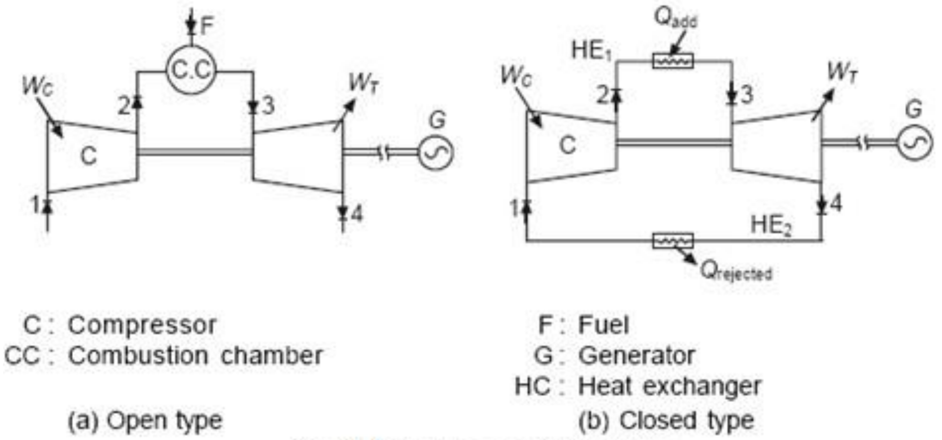


Fig. 4.1 *Simple gas turbine plant*

Thermodynamic cycle shows following processes:

1-2: Adiabatic compression, involving (–ve) work, W_C in compressor.

2-3 : Constant pressure heat addition, involving heat Q_{add} in combustion chamber or heat exchanger.

3-4: Adiabatic expansion, involving (+ve) work, W_T in turbine.

4-1: Constant pressure heat rejection, involving heat, $Q_{rejected}$ in atmosphere or heat exchanger.

In the gas turbine plant layout shown process 1–2 (adiabatic compression) is seen to occur in compressor, heat addition process 2–3 occurs in combustion chamber having open type arrangement and in heat exchanger in closed type arrangement. Process 3–4 of adiabatic expansion occurs in turbine.

In open type arrangement exhaust from turbine is discharged to atmosphere while in closed type, heat rejection occurs in heat exchanger. In gas turbine plant of open type, air entering compressor gets compressed and subsequently brought up to elevated temperature in combustion chamber where fuel is added to high pressure air and combustion occurs. High pressure and high temperature combustion products are sent for expansion in turbine where its' expansion yields positive work. Expanded combustion products are subsequently discharged to atmosphere. Negative work required for compression is drawn from the positive work available from turbine and residual positive work is available as shaft work for driving generator.

In gas turbine plant of closed type the working fluid is recycled and performs different processes without getting contaminated. Working fluid is compressed in compressor and subsequently heated up in heat exchanger through indirect heating. High pressure and high temperature working fluid is sent for getting positive work from turbine and the expanded working fluid leaving turbine is passed through heat exchanger where heat is picked up from working fluid. Thus, the arrangement shows that even costly working fluids can also be used in closed type as it remains uncontaminated and is being recycled.

Air standard analysis of Brayton cycle gives work for compression and expansion as;

$$W_C = m_1 \cdot (h_2 - h_1)$$

$$W_T = m_3 \cdot (h_3 - h_4)$$

for air standard analysis, $m_1 = m_3$, where as in actual cycle

$$m_3 = m_1 + m_f, \quad \text{in open type gas turbine}$$

$$m_3 = m_1, \quad \text{in closed type gas turbine}$$

For the fuel having calorific value CV the heat added in air standard cycle;

$Q_{add} = m_1(h_3 - h_2)$, whereas $Q_{add} = m_f \times CV$ for actual

cycle. Net work = $W_T - W_C$

$W_{net} = \{m_3\ (h_3 - h_4) - m_1(h_2 - h_1)\}$

$$\text{Air standard cycle efficiency} = \frac{W_{net}}{Q_{add}}$$

$$= \frac{m_1\{(h_3 - h_4) - (h_2 - h_1)\}}{m_1(h_3 - h_2)}$$

$$\text{Air standard Brayton cycle efficiency: } \boxed{\eta_{Brayton} = 1 - \frac{1}{r^{\frac{\gamma-1}{\gamma}}}}$$

Thus, it is obvious from the expression of efficiency that it depends only on pressure ratio (r) and nature of gas (γ). For pressure ratio of unity, efficiency shall be zero. For a particular gas the cycle efficiency increases with increasing pressure ratio. Here the variation of efficiency with pressure ratio is shown for air (γ= 1.4) and monatomic gas as

argon (γ= 1.66).

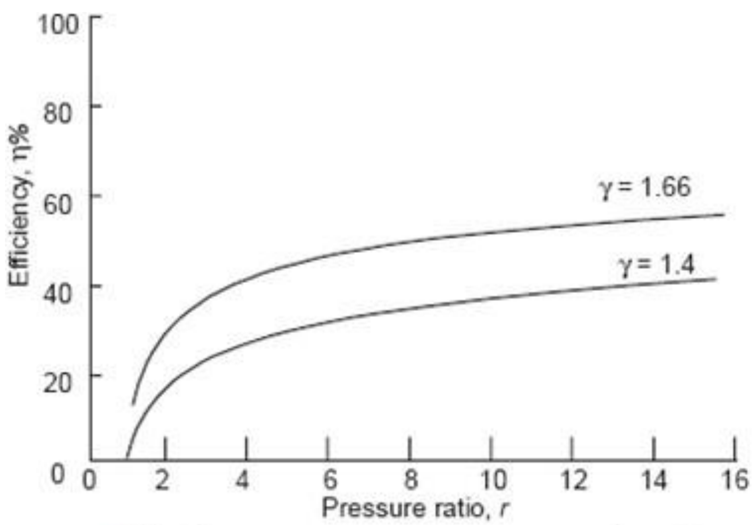


Fig. 4.2 *Efficiency vs. pressure ratio in simple cycle*

Regenerative gas turbine cycle

Regenerative air standard gas turbine cycle shown ahead in Fig. 4.3 has a regenerator (counter flow heat exchanger) through which the hot turbine exhaust gas and comparatively cooler air coming from compressor flow in opposite directions. Under ideal conditions, no frictional pressure drop occurs in either fluid stream while turbine exhaust gas gets cooled from 4 to 4′ while compressed air is heated from 2 to 2′. Assuming regenerator effectiveness as 100% the temperature rise from 2–2′ and drop from 4 to 4′ is shown on *T-S* diagram.

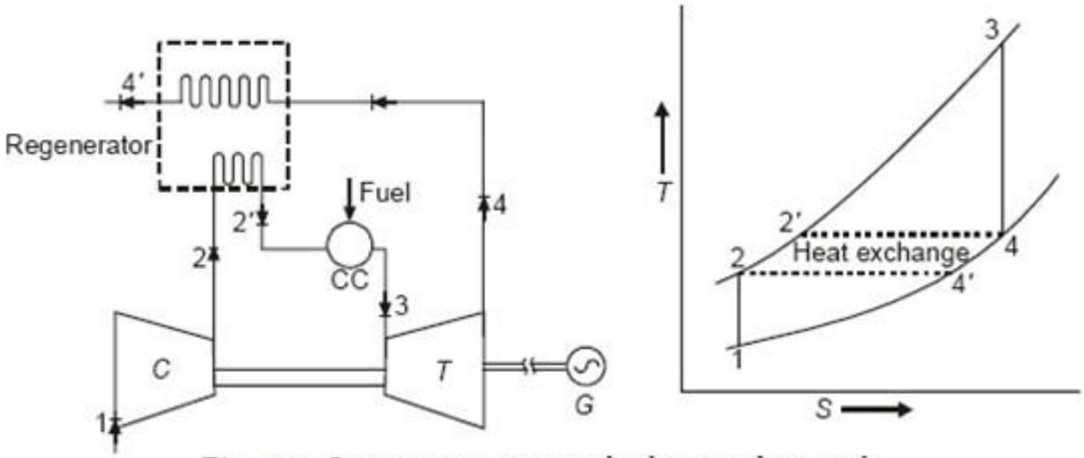


Fig. 4.3 *Regenerative air standard gas turbine cycle.*

$$\text{Regenerator effectiveness, } \varepsilon = \frac{h_{2'} - h_2}{h_4 - h_2},$$

Thus, thermodynamically the amount of heat now added shall be

$$Q_{add,\,regen} = m\,(h_3 - h_2')$$

Where as without regenerator the heat added; $Q_{add} = m\,(h_3 - h_2)$ Here it is obvious that, $Q_{add,\ regen} < Q_{add}$

This shows an obvious improvement in cycle thermal efficiency as every thing else remains same. Net work produced per unit mass flow is not altered by the use of regenerator.

$$\text{Air standard cycle thermal efficiency, } \eta_{regen} = \frac{(h_3 - h_4) - (h_2 - h_1)}{(h_3 - h_{2'})}$$

$$\eta_{regen} = \frac{c_p \cdot (T_3 - T_4) - c_p\,(T_2 - T_1)}{c_p\,(T_3 - T_{2'})}$$

eheat gas turbine cycle

Reheat gas turbine cycle arrangement is shown in Fig. 4.4. In order to maximize the
work available from the simple gas turbine cycle one of the option is to increase enthalpy of fluid entering gas turbine and extend its expansion upto the lowest possible enthalpy value.

C: Compressor HPT: High pressure turbine

CC: Combustion chamber LPT: Low pressure turbine

G: Generator RCC: Reheat combustion

chamber f: Fuel

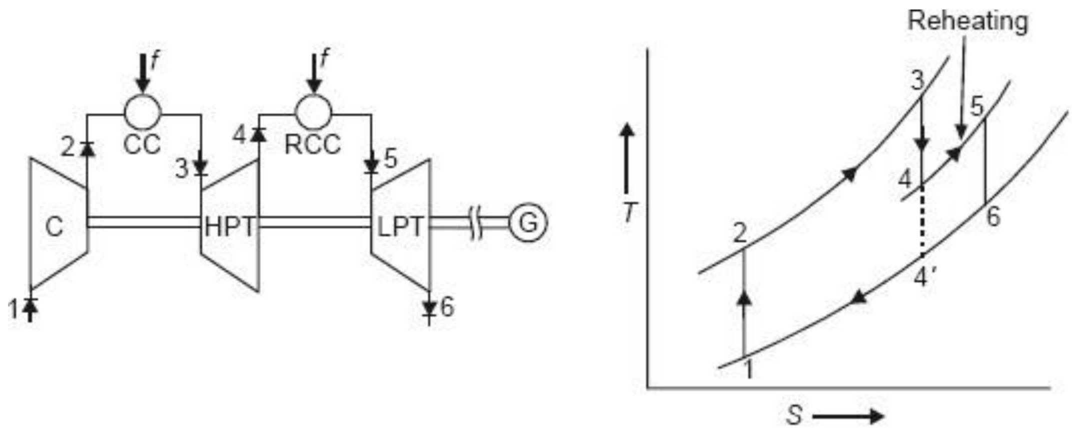


Fig. 4.4 *Reheat gas turbine cycle*

This can also be said in terms of pressure and temperature values i.e. inject fluid at high pressure and temperature into gas turbine and expand upto lowest possible pressure value. Upper limit at inlet to turbine is limited by metallurgical limits while lower pressure is limited to near atmospheric pressure in case of open cycle. Here in the arrangement shown ambient air enters compressor and compressed air at high pressure leaves at 2. Compressed air is injected into combustion chamber for increasing its temperature up to desired turbine inlet temperature at state 3. High pressure and high temperature fluid enters high pressure turbine (HPT) for first phase of expansion and expanded gases leaving at 4 are sent to reheat combustion chamber (reheater) for being further heated. Thus, reheating is a kind of energizing the working fluid.

Assuming perfect reheating (in which temperature after reheat is same as temperature attained in first combustion chamber), the fluid leaves at state 5 and enters low pressure turbine (LPT) for remaining expansion upto desired pressure value. Generally,

temperature after reheating at state 5 is less than temperature at state 3. In the absence of reheating the expansion process within similar pressure limits goes upto state 4′. Thus, reheating offers an obvious advantage of work output increase since constant pressure

lines on *T-S* diagram diverge slightly with increasing entropy, the total work of the two stage turbine is greater than that of single expansion from state 3 to state 4′. i.e.,

$$(T3 - T4) + (T5 - T6) > (T3 - T4').$$

Here it may be noted that the heat addition also increases because of additional heat supplied for reheating. Therefore, despite the increase in net work due to reheating the cycle thermal efficiency would not necessarily increase.

A plot showing variation of efficiency with pressure ratio 'r' is shown in Fig. 4.5 along with simple cycle efficiency variation. It indicates that reheating offers increase in specific work output at the

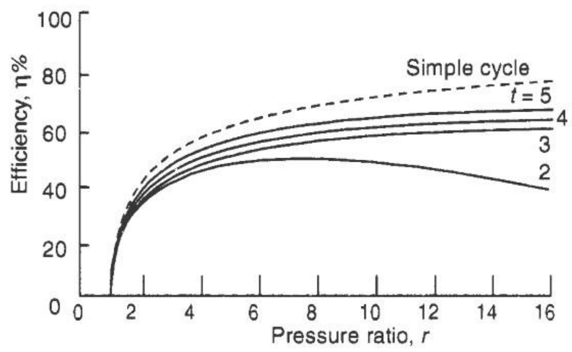


Fig. 4.5 *Reheat cycle efficiency vs. cycle pressure ratio*

cost of cycle efficiency. This reduction in efficiency may be attributed to the addition of a less efficient cycle 4564′ to a simple cycle. 4564′ is a less efficient cycle since it operates over a smaller temperature range. Variation of specific work output with pressure ratio is shown in Fig. 4.6. It shows how specific work output shows increase with increasing pressure ratio upto optimum pressure ratio. It may also be noted that in reheat cycle, the temperature of exhaust gases at exit of gas turbine gets increased as compared to simple cycle within similar limits. Therefore, reheat cycle offers potential for use of regenerator for harnessing the hotter exhaust from gas turbine.

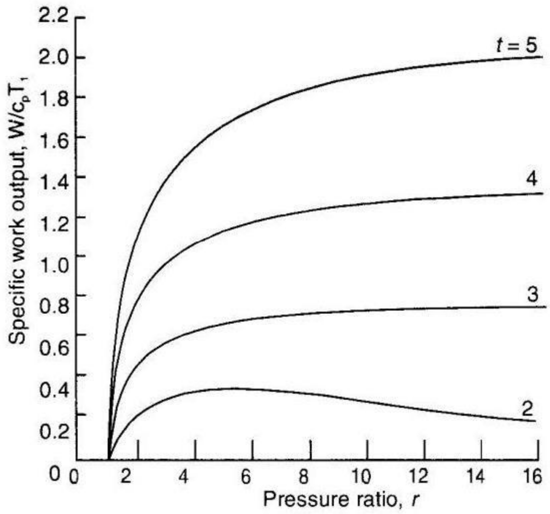
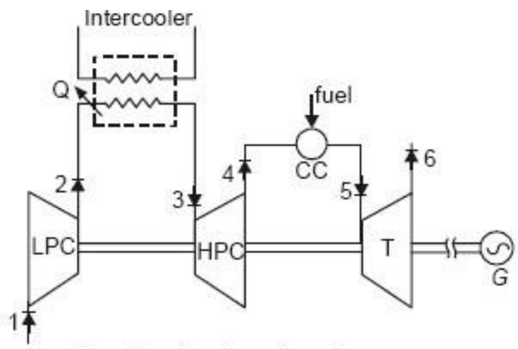


Fig. 4.6 *Reheat cycle specific work output vs. cycle pressure ratio*

GAS TURBINE CYCLE WITH INTERCOOLING

Net work output from gas turbine cycle can also be increased by reducing negative work i.e. compressor work. Multistaging of compression process with intercooling in between is one of the approach for reducing compression work. It is based on the fact that for a fixed compression ratio higher is the inlet temperature higher shall be compression work requirement and vice-a-versa. Schematic for intercooled gas turbine cycle is given in Fig. 4.7.

Thermodynamic processes involved in multistage intercooled compression are shown in Figs.
4.8, 4.9. First stage compression occurs in low pressure compressor (LPC) and compressed air leaving LPC at ‘2’ is sent to intercooler where temperature of compressed air is lowered down to state 3 at constant pressure. In case of perfect intercooling the temperatures at 3 and 1 are same. Intercooler is a kind of heat exchanger where heat is picked up from high temperature compressed air. The amount of compression work saved due to intercooling is obvious from *p-V* diagram and shown by area 2342′. Area 2342′ gives the amount of work saved due to intercooling between compression.



CC : Combustion chamber
LPC : Low pressure compressor
HPC : High pressure compressor
T : Turbine
G : Generator

Fig. **4.7** *Gas turbine cycle with intercooling*

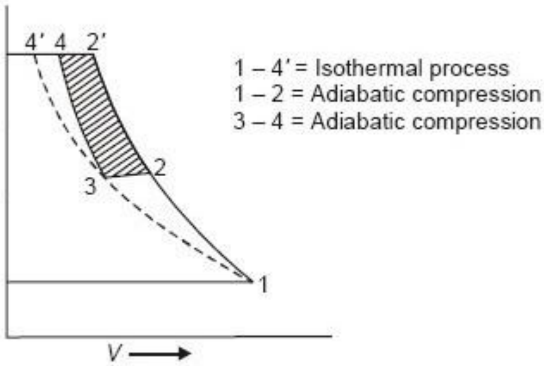


Fig. **4.8** *Intercooled compression*

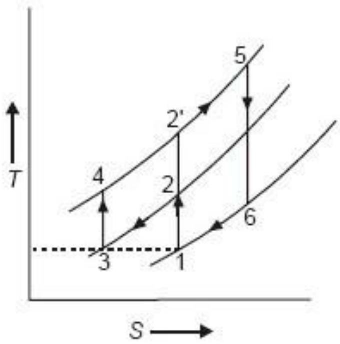


Fig. **4.9** *T-S diagram for gas turbine*

for gas turbine cycle with intercooling shows that in the absence of intercooling within same pressure limits the state at the end of compression would be 2′ while with perfect intercooling
this state is at 4 i.e., $T2' > T4$. The reduced temperature at compressor

exit leads to additional heat requirement in combustion chamber i.e. more amount of fuel is to be burnt for attaining certain turbine inlet temperature as compared to simple cycle without intercooling.

Thus, intercooled cycle thermal efficiency may not increase with intercooling because of simultaneous increase in heat addition requirement. The lower temperature at compressor exit enhances the potential for regeneration so when intercooling is used in conjunction with regeneration an appreciable increase in thermal efficiency can result.

Net work output in gas turbine cycle with intercooling;

$$W_{\text{net, intercool}} = m\{(h_5 - h_6) - (h_4 - h_3) - (h_2 - h_1)\}$$

$$W_{\text{net, intercool}} = mc_p\{(T_5 - T_6) - (T_4 - T_3) - (T_2 - T_1)\}$$

Cycle thermal efficiency;

$$\eta_{\text{intercool}} = \frac{\{(h_5 - h_6) - (h_4 - h_3) - (h_2 - h_1)\}}{\{h_5 - h_4\}}$$

GAS TURBINE CYCLE WITH REGENERATION, REHEAT AND INTERCOOLING

Regenerative gas turbine employing reheating during expansion and intercooling during compression is considered here as shown in Fig. 4.13 . This combination offers considerable increase in net work output and thermal efficiency.

C_1 : Low pressure compressor stage
C_2 : High pressure compressor stage
CC : Combustion chamber
RCC : Reheat combustion chamber
T_1 : High pressure turbine stage
T_2 : Low pressure turbine stage
G : Generator

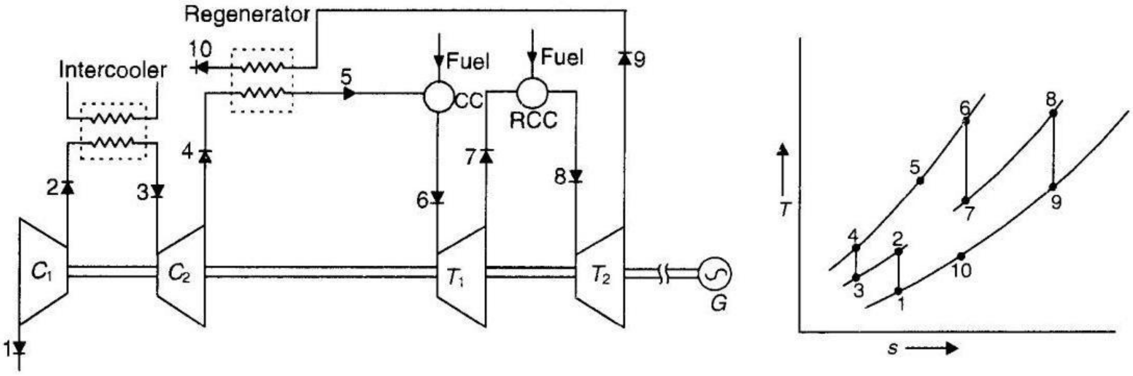


Fig. 4.13 *Schematic for gas turbine cycle with regeneration, reheat and intercooling and T-S diagram*

Based upon air standard cycle considerations thermodynamic analysis gives the Net work output from cycle,

$$W_{net} = m\ \{(h_6 - h_7) + (h_8 - h_9) - (h_4 - h_3) - (h_2 - h_1)\}$$

Heat added

$$Q_{add} = m\{(h_6 - h_5) + (h_8 - h_7)\}$$

Cycle thermal efficiency,

$$\eta_{cycle} = \frac{\{(h_6 - h_7) + (h_8 - h_9) - (h_4 - h_3) - (h_2 - h_1)\}}{\{(h_6 - h_5) + (h_8 - h_7)\}}$$

GAS TURBINE IRREVERSIBILITIES AND LOSSES

Till now the discussions have been confined to air standard Brayton cycle. But the realistic gas turbine cycle has deviations from air standard cycle due to,

(*i*) frictional effects within compressor and turbine which causes increase in specific entropy of working fluid across these components.

(*ii*) friction which shall cause drop in pressure of working fluid across the constant pressure processes.

Apart from above irreversibilities of the gas turbine power plant the irreversibilities of combustion chamber are quite significant.

Salient state points of realistic gas turbine Brayton cycle with above irreversibilities and losses are shown below:

Isentropic efficiency of turbine and compressor can be mathematically given as

$$\eta_{\text{isen},\,t} = \left\{\frac{h_3 - h_4}{h_3 - h_{4s}}\right\}$$

$$\text{i.e. } \eta_{\text{isen},\,t} = \frac{\text{Actual expansion work}}{\text{Ideal expansion work}}$$

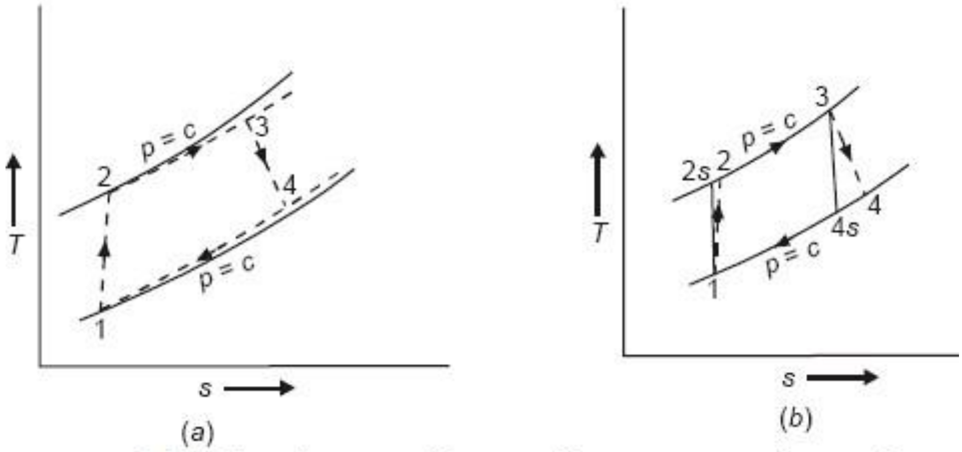


***Fig. 4.14** Effect of irreversibilities and losses in gas turbine cycle.*

Isentropic efficiency of compressor

$$\eta_{\text{isen},\,c} = \left\{\frac{h_{2s} - h_1}{h_2 - h_1}\right\}$$

$$\therefore \eta_{\text{isen},\,c} = \frac{\text{Ideal compressor work}}{\text{Actual compressor work}}$$

Other factors causing the real cycle to be different from ideal cycle are as given below:

(i) Fluid velocities in turbomachines are very high and there exists substantial change in kinetic energy between inlet and outlet of each component. In the analysis carried out earlier the changes in kinetic energy have been neglected whereas for exact analysis it cannot be.

(ii) In case of regenerator the compressed air cannot be heated to the temperature of gas leaving turbine as the terminal temperature difference shall always exist.

(iii) Compression process shall involve work more than theoretically estimated value in order to overcome bearing and windage friction losses.

Different factors described above can be accounted for by stagnation properties, compressor and turbine isentropic efficiency and polytropic efficiency.

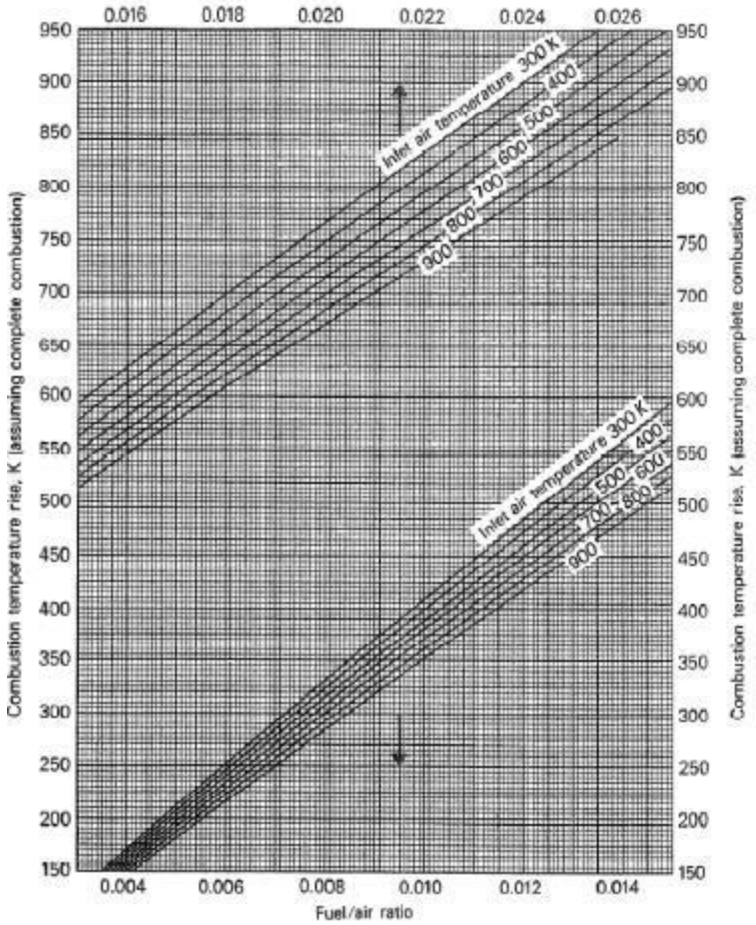
$T_{02} = 164.7 + 288 = 452.7$K, and $T_{04} = 1100-264.8 = 835.2$ K

Hence,

$T_{05} = 0.80*382.5+452.7 = 758.7$ K

For a combustion chamber inlet air temperature of 759 K and a combustion temperature rise of (1100-759) = 341 K, the theoretical fuel/air ratio required is 0.0094 (from the chart of slide 12), and thus

$$f = \frac{theoretical\_f}{\eta_b} = \frac{0.0094}{0.98} = 0.0096$$

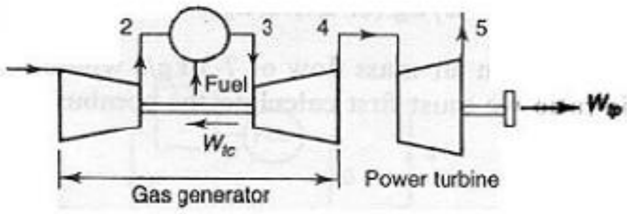


Theoretical Fuel-Air Ratio

Combustion temperature rise vs theoretical fuel-air ratio

Determine the specific work output,specific fuel consumption and cycle efficiency for a simple cycle gas turbine with a free power turbine (see figure) given the following specification:

Compressor pressure ratio	12.0
Turbine inlet temperature	1350 K
Isentropic efficiency of compressor, η_c	0.86
Isentropic efficiency of each turbine, η_t	0.89
Mechanical efficiency of each shaft, η_m	0.99
Combustion efficiency	0.99
Combustion chamber pressure loss	6 % compressor delivery pressure
Exhaust pressure loss	0.03 bar
Ambient conditions, p_a, T_a	1 bar, 288 K



Free turbine unit

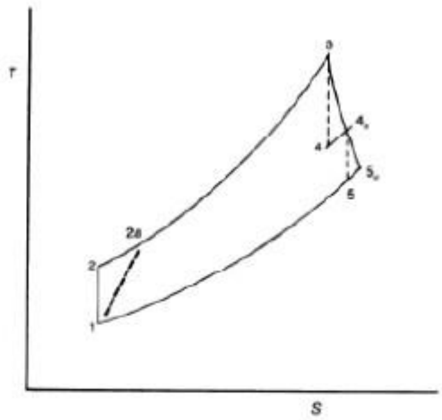
Solution:

Proceeding as in the previous example,

$$T_{02} - T_{01} = \frac{288}{0.86}\left[12^{1/3.5} - 1\right] = 346.3\,\text{K}$$

$$W_{tc} = \frac{1.005 * 346.3}{0.99} = 351.5\,\text{kJ/kg}$$

$$p_{03} = 12.0(1 - 0.06) = 11.28\,\text{bar}$$



The intermediate pressure between the two turbines, p_{04}, is unknown, but can be determined from the fact that the compressor turbine produces just sufficient work to drive the compressor. The temperature equivalent of the compressor turbine work is, therefore,

$$T_{03} - T_{04} = \frac{W_{tc}}{c_{pg}} = \frac{351.5}{1.148} = 306.2\,\text{K}$$

The corresponding pressure ratio can be found using the relation

$$T_{03} - T_{04} = \eta_t T_{03}\left[1 - \left(\frac{1}{p_{03}/p_{04}}\right)^{\gamma-1/\gamma}\right]$$

$$306.2 = 0.89*1350\left[1 - \left(\frac{1}{p_{03}/p_{04}}\right)^{0.25}\right]$$

$$\frac{p_{03}}{p_{04}} = 3.243$$

$$T_{04} = 1350 - 306.2 = 1043.8\,K$$

The pressure at entry to the power turbine, p_{04}, is then found to be

$$p_{04} = \frac{p_{03}}{p_{03}/p_{04}} = 11.28/3.243 = 3.478 \text{ bar}$$

and the power turbine pressure ratio is

$$p_{04}/p_{05} = 3.478/(1+0.03) = 3.377$$

The temperature drop in the power turbine can now be obtained

$$T_{04} - T_{05} = 0.89*1043.8\left[1 - \left(\frac{1}{3.377}\right)^{0.25}\right] = 243.7\,\text{K}$$

and the specific work output, i.e. power turbine work per unit air mass flow, is

$$W_{tp} = c_{pg}(T_{04} - T_{05})\eta_m$$

$$W_{tp} = 1.148(243.7)0.99 = 277.0 \text{ kJ/kg (or kW/kg)}$$

The compressor delivery temperature is 288+346.3 = 634.3 K and the combustion temperature rise is 1350 – 634.3 = 715.7 K

The theoretical fuel/air ratio required is 0.0202 (from the chart in slide 12), giving an actual fuel/air ratio of 0.0202/0.99 = 0.0204

The SFC and cycle efficiency, η, are then given by

$$SFC = \frac{f}{W_{tp}} = \frac{3600*0.0204}{277.9} = 0.265 \text{ kg/kWh}$$

$$\eta = \frac{3600}{0.265*43100} = 0.315$$

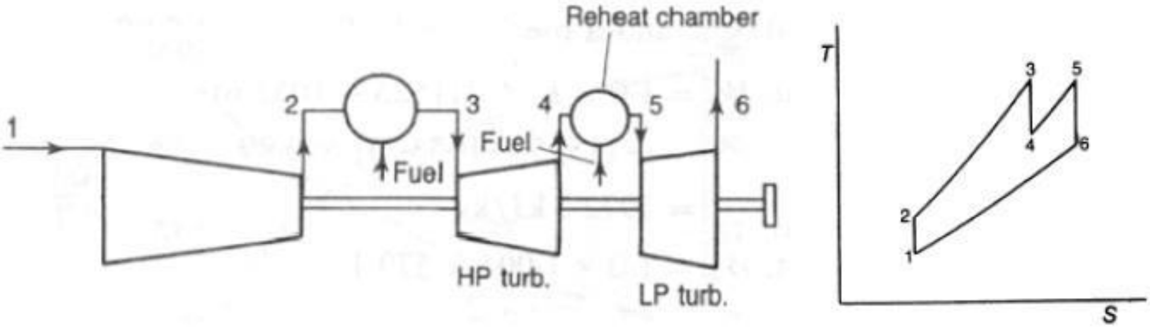
- The cycle calculations are carried out to determine the overall performance. It However, they also provide information required by other groups such as the aerodynamic and control design groups.
- *E.g.*, the temperature at entry to the power turbine, T_{04}, may be required as a control parameter to prevent operation above the metallurgical limiting temperature of the compressor turbine.
- The exhaust gas temperature (EGT), T_{05}, would be important if the gas turbine were to be considered for combined cycle or cogeneration plant.
- The temperature, $T_{05} = 1043.8\text{-}243.7 = 800.1\text{K}$ or $527°$ C, is suitable for use with a waste heat boiler.
- For a combined cycle plant, a higher T I T might be desirable because there would be a consequential increase in EGT, permitting the use of a higher steam temperature and a more efficient steam cycle.
- If the cycle pressure ratio were increased to increase the efficiency of the gas cycle, however, the EGT would be decreased resulting in a lower steam cycle efficiency.

Example 3:

Consider the design of a high pressure ratio, single-shaft cycle with reheat at some point in the expansion when used either as a separate unit,or as part of a combined cycle. The power required is 240 MW at 288 K and 1.01 bar

Compressor pressure ratio	30
Polytropic efficiency	
(compressor and turbines)	0.89
Turbine inlet temperature	
(both turbines)	1525 K
$\Delta p/p_{02}$ (1 st combustor)	0.02
$\Delta p/p_{04}$ (2 nd combustor)	0.04
Exhaust pressure	1.02 bar

Solution:



A heat Exchanger is not used because it would result in an exhaust temperature that would be too low for use with a high efficiency steam cycle.

The Assumptions are as follows

- Let us assume that the mass flow rate is constant throughout, ignoring the effect of substantial cooling bleeds that would be required with high turbine inlet temperatures specified.
- The reheat pressure is not specified. So, as a starting point we use a value giving equal pressure ratio in each turbine.
- This division of the expansion leads to equal work in each turbine and a maximum net work output for the ideal reheat cycle).

From polytropic relations:

for compression, $$\frac{n-1}{n} = \frac{1}{\eta_{\infty c}}\left(\frac{\gamma-1}{\gamma}\right) = \frac{1}{0.89}\left(\frac{0.4}{1.4}\right) = 0.3210$$

for expansion, $$\frac{n-1}{n} = \eta_{\infty t}\left(\frac{\gamma-1}{\gamma}\right) = 0.89\left(\frac{0.333}{1.333}\right) = 0.2223$$

Assuming that $p_{01} = p_a$ and $T_{01} = T_a$, we have $T_{02}/T_{01} = (30)^{0.3210}$

$$T_{02} = 858.1 \text{ K}$$

$$T_{02} - T_{01} = 570.1 \text{ K}$$

$$p_{02} = 30*1.01 = 30.3 \text{ bar}$$

$$p_{03} = 30.3(1.00-0.02) = 29.69 \text{ bar}$$

$$p_{06} = 1.02 \text{ bar, so } \; p_{03}/p_{06} = 29.11$$

Theoretically, the optimum pressure ratio for each turbine would be

$$\sqrt{(29.11)} = 5.395$$

A pressure loss of 4 % in the reheat combustor has to be considered, so a value of 5.3 for p_{03}/p_{04} could be assumed. Then,

$$\frac{T_{03}}{T_{04}} = (5.3)^{0.2223}$$

$$T_{04} = 1052.6K$$

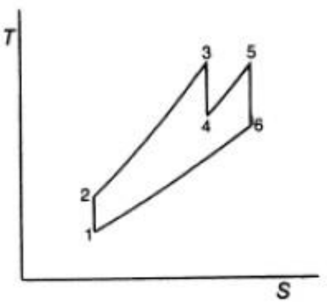
$$p_{04} = 29.69/5.3 = 5.602 bar$$

$$p_{05} = 5.602(1.00-0.04) = 5.378 bar$$

$$p_{05}/p_{06} = 5.378/1.02 = 5.272$$

$$\frac{T_{05}}{T_{06}} = (5.272)^{0.2223}$$

$$T_{06} = 1053.8K$$



Assuming unit flow of 1.0 kg/s and a mechanical efficiency of 0.99,

Turbine output, W_t= 1.0*1.148{(1525-1052.6)+(1525-1053.8)}*0.99

= 1072.3 kJ/kg

Compressor input, W_c = 1.0*1.005*570.1

= 573.0 kJ/kg

Net work output, W_N= 1072.3-573.0 =499.3 kJ/kg

Flow required for 240 MW is given by

m = 240000/499.3

= 480.6 kg/s

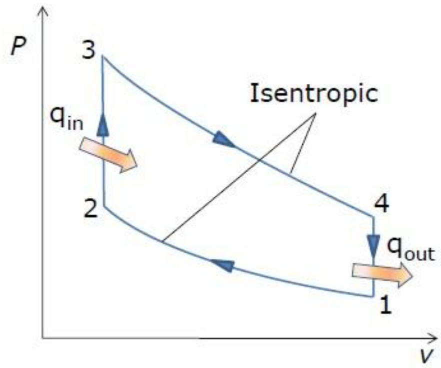
- For the first combustor, temperature rise = 1525 - 858 = 667 K, inlet temperature = 858 K and fuel/air ratio = 0.0197 (*from the chart of Slide 12*)
- For the second combustor, temperature rise = 1525-1052.6 = 472.4 K, inlet temperature = 1052.6 K and fuel/air ratio = 0.0142(*from the chart of Slide 12*)
- Actual total fuel/air ratio $f = \dfrac{0.0197 + 0.0142}{0.99} = 0.0342$
- And, thermal efficiency $\eta = \dfrac{499.3}{0.0342 * 43100} = 33.9\%$
- This is a reasonable efficiency for simple cycle operation, and the specific output is excellent.
- However, the turbine exit temperature (T_{06} = 1053.8 K or 780.8° C) is too high for efficient use in a combined cycle plant. A reheat steam cycle using conventional steam temperatures of about 550° -575° C would require a turbine exit temperature of about 600° C.

- The turbine exit temperature could be reduced by increasing the reheat pressure, and if the calculations are repeated for a range of reheat pressures, then the results obtained are as shown in the Slide 26.
- It can be seen that a reheat pressure of 13 bar gives an exhaust gas temperature (EGT) of 605° C; the specific output is about 10 percent lower than the optimum value, but the thermal efficiency is substantially improved to 37.7 per cent. Further increases in reheat pressure would give slightly higher efficiencies, but the EGT would be reduced below 600° C resulting in a less efficient steam cycle.

Problems for practice

In an air standard Otto cycle, the compression ratio is 7 and the compression begins at 35oC and 0.1 MPa. The maximum temperature of the cycle is 1100oC. Find (a) the temperature and the pressure at various points in the cycle, (b) the heat supplied per kg of air, (c) work done per kg of air, (d) the cycle efficiency and (e) the MEP of the cycle.

Solution: Problem 1



T_1=35°C=308 K
P_1=0.1 Mpa
T_3=1100°C=1373 K
$r=v_1/v_2=7$

- Since process, 1-2 is isentropic,

$$\frac{P_2}{P_1} = \left(\frac{v_1}{v_2}\right)^{\gamma} = 7^{1.4} = 15.24$$

- Hence, P_2=1524 kPa

$$\frac{T_2}{T_1} = \left(\frac{v_1}{v_2}\right)^{\gamma-1} = 7^{1.4-1} = 2.178$$

- Hence, T_2=670.8 K

- For process, 2-3,

$$\frac{P_2 v_2}{T_2} = \frac{P_3 v_3}{T_3}, \ \therefore P_3 = \frac{T_3}{T_2} P_2 = \frac{1373}{607.8} \times 1524 = 3119.34$$

- P_3=3119.34 kPa.
- Process 3-4 is again isentropic,

$$\frac{T_3}{T_4} = \left(\frac{v_4}{v_3}\right)^{\gamma-1} = 7^{1.4-1} = 2.178$$

$$\therefore T_4 = \frac{1373}{2.178} = 630.39\,\mathrm{K}$$

- Hence, T_2=630.39 K

- Heat input,

$$Q_{in} = c_v(T_3 - T_2)$$
$$=0.718(1373-670.8)$$
$$=504.18 \text{ kJ/kg}$$

- Heat rejected,

$$Q_{out} = c_v(T_4 - T_1)$$
$$=0.718(630.34-308)$$
$$=231.44 \text{ kJ/kg}$$

- The net work output, $W_{net}=Q_{in}-Q_{out}$

- The net work output,

$$W_{net}=Q_{in}-Q_{out}$$
$$=272.74 \text{ kJ/kg}$$

- Thermal efficiency, $\eta_{th,otto}=W_{net}/Q_{in}$

$$=0.54$$
$$=54 \%$$

- Otto cycle thermal efficiency,

$$\eta_{th,otto} = 1-1/r^{\gamma-1} = 1-1/7^{0.4}$$
$$= 0.54 \text{ or } 54 \%$$

- $v_1=RT_1/P_1$

$$=0.287 \text{x} 308/100=0.844 \text{ m}^3\text{/kg}$$

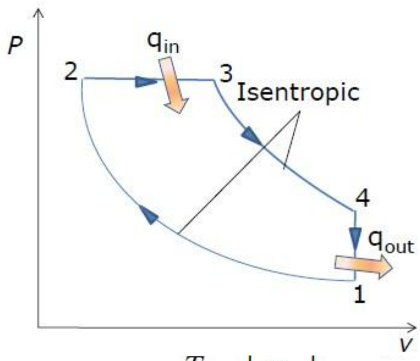
- $MEP = W_{net}/(v_1-v_2) = 272.74/v_1\,(1-1/r)$

$$=272.74/0.844(1-1/7)$$
$$=360 \text{ kPa}$$

Problem 2

In a Diesel cycle, the compression ratio is 15. Compression begins at 0.1 Mpa, 40oC. The heat added is 1.675 MJ/kg. Find (a) the maximum temperature in the cycle, (b) work done per kg of air (c) the cycle efficiency (d) the temperature at the end of the isentropic expansion (e) the cut-
off ratio and (f) the MEP of the cycle.

Solution: Problem 2



T_1=40°C=313 K
P_1=0.1 Mpa
Q_{in}=1675 MJ/kg
$r=v_1/v_2=15$

$$T_1 \quad \left(v_2 \right)$$

$$T_2 = 313 \times 2.954 = 924.66\,\mathrm{K}$$

$$Q_{in} = 1675 = 1.005(T_3 - 924.66)$$
$$\therefore T_3 = 2591.33\ \mathrm{K} = T_{\max}$$

- Hence, the maximum temperature is 2591.33 K

$$\frac{P_2}{P_1} = \left(\frac{v_1}{v_2}\right)^{\gamma} = 15^{1.4} = 44.31$$

$$\therefore P_2 = 4431\,\mathrm{kPa}$$

$$\frac{P_2 v_2}{T_2} = \frac{P_3 v_3}{T_3} \rightarrow v_3 = \frac{T_3}{T_2} v_2 = \frac{2591.33}{924.66} \times 0.06 = 0.168\,\mathrm{m^3/kg}$$

$$r_c = \frac{v_3}{v_2} = \frac{0.168}{0.06} = 2.8$$

- The cut-off ratio is 2.8.

$$T_4 = T_3 \left(\frac{v_3}{v_4}\right)^{\gamma-1} = 2591.33 \times \left(\frac{0.168}{0.898}\right)^{0.4}$$

$$= 1325.37\,\mathrm{K}$$

$$\mathrm{Q_{out}} = c_v(T_4 - T_1) = 0.718(1325.4 - 313) = 726.88\,\mathrm{kJ/kg}$$

$$\text{Net work done}, W_{net} = Q_{in} - Q_{out} = 1675 - 726.88$$

$$=948.12\ \mathrm{kJ/kg}$$

- Therefore, thermal efficiency,

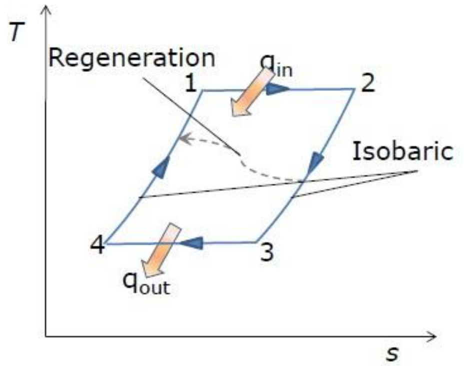
$$\eta_{th} = W_{net}/Q_{in}$$
$$=948.12/1675=0.566 \text{ or } 56.6\%$$

- The cycle efficiency can also be calculated using

Problem 3

An air-standard Ericsson cycle has an ideal regenerator. Heat is supplied at 1000°Cand heat is rejected at 20°C. If the heat added is 600 kJ/kg,find the compressor work, the turbine work, and the cycle efficiency.

Solution: Problem 3



T_1=T_2=1000°C=1273.15 K
T_3=T_4=20°C=293.15 K

Since the regenerator is given as ideal, *-Q2-3= Q1-4*

Also in an Ericsson cycle, the heat is input during the isothermal expansion process, which is the turbine part of the cycle. Hence the turbine work is 600 kJ/kg.

- Thermal efficiency of an Ericsson cycle is equal to the Carnot efficiency.

$$\eta_{th}=\eta_{th,\ Carnot}=1-T_L/T_H$$
$$=1-293.15/1273.15$$
$$=0.7697$$

- Therefore the net work output is equal to:

$$w_{net}=\eta_{th}Q_H$$
$$= 0.7697\times600=461.82\ kJ/kg$$

- The compressor work is equal to the difference between the turbine work and the net work output:

$$w_c=w_t-w_{net}$$
$$=600-461.82=138.2\ kJ/kg$$

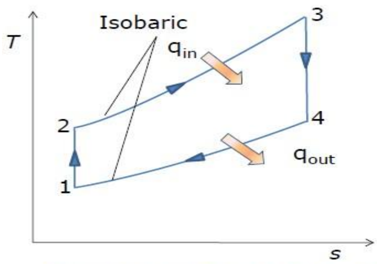
- In the Ericsson cycle the heat is rejected isothermally during the compression process. Therefore this compressor work is also equal to the heat rejected during the cycle.

Problem 4

In a Braytoncycle based power plant, the air at the inlet is at 27oC, 0.1 MPa. The pressure ratio is 6.25and the maximum temperature is 800oC.

Find (a) the compressor work per kg of air (b) the turbine work per kg or air (c) the heat supplied per kg of air, and (d) the cycle efficiency.

Solution: Problem 4



T_1 = 27°C = 300 K
P_1 = 100 kPa
r_p = 6.25
T_3 = 800°C = 1073 K

- Since process, 1-2 is isentropic,

$$\frac{T_2}{T_1} = r_p^{(\gamma-1)/\gamma} = 6.25^{(1.4-1)/1.4} = 1.689$$
$$T_2 = 506.69\text{ K}$$
$$W_{comp} = c_p(T_2 - T_1) = 1.005(506.69 - 300)$$
$$= 207.72\text{ kJ/kg}$$

- The compressor work per unit kg of air is 207.72 kJ/kg

- Process 3-4 is also isentropic,

$$\frac{T_3}{T_4} = r_p^{(\gamma-1)/\gamma} = 6.25^{(1.4-1)/1.4} = 1.689$$
$$T_4 = 635.29\text{ K}$$
$$W_{turb} = c_p(T_3 - T_4) = 1.005(1073 - 635.29)$$
$$= 439.89\text{ kJ/kg}$$

- The turbine work per unit kg of air is 439.89 kJ/kg
- Heat input, Q_{in},

$$Q_{in} = c_p(T_3 - T_2) = 1.005(1073 - 506.69)$$
$$= 569.14\text{ kJ/kg}$$

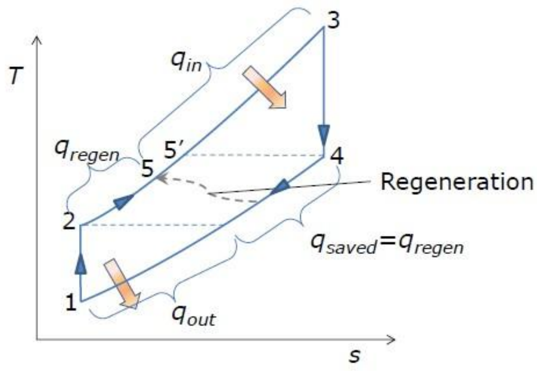
- Heat input per kg of air is 569.14 kJ/kg
- Cycle efficiency,

$\eta_{th}=(W_{turb}-W_{comp})/Q_{in}$
$=(439.89-207.72)/569.14$
$=0.408$ or 40.8%

Problem 5

- Solve Problem 3 if a regenerator of 75% effectiveness is added to the plant.

Solution: Problem 5



$$\varepsilon = \frac{T_5 - T_2}{T_4 - T_2} = 0.75$$

$$or, \frac{T_5 - 506.69}{635.29 - 506.69} = 0.75$$

$$T_5 = 603.14\,K$$

- T_4, W_{comp}, W_{turb} remain unchanged
- The new heat input, $Q_{in}=c_p(T_3-T_5)$
  $=472.2$ kJ/kg
- Therefore $\eta_{th}=(W_{turb}-W_{comp})/Q_{in}$
  $=(439.89-207.72)/472.2$
  $=0.492$ or $49.2\ \%$

INDUSTRIAL APPLICATIONS

Industrial Applications

The following are the applications of a gas turbine:

1. They are used to propel air-crafts and ships,
2. Gas turbine plants are used as standby plants for the hydroelectric power plants.
3. Gas turbine power plants may be used as peak loads plant and standby plants for smaller power units.
4. The shaft can be connected to other machinery to do various types of work such as: turning a helicopter rotor, running a compressor (which "crushes" a gas to a condensed form for use in industrial applications) or generating electric power.
5. The gas turbine is useful to our modern world because it is relatively compact in size and makes a lot of power. Gas turbines are used in backup power systems in Manhattan for example, when the grid goes down due to natural disaster, gas turbines power up and can produce power for emergency uses.
6. Gas turbines are used on oil platforms to make power. The oil platform is like a small city, isolated out on the water, so it requires a lot of power and does not have a lot of space. Gas turbines are also used in oil refineries to make power for the cracking process.

TUTORIAL QUESTIONS

UNIT – IV

Theory Questions:

1. Explain about the open cycle and closed cycle turbines with neat sketches and also draw P-V & T-S diagrams.
2. State the merits of gas turbines over IC engines.
3. Draw the gas turbine power plant with inter cooling
4. List out the advantages of open cycle gas turbine over closed cycle gas turbine.
5. List different applications of gas turbine power cycles in power sector industries.
6. Discuss the relative advantages and disadvantages of gas turbines and steam turbines.
7. What are the different methods to improve the efficiency of gas turbines?
8. What are the different types of combustion chambers in gas turbine engines? Explain them in detail with relevant sketches.
9. Draw the schematic diagram of closed cycle gas turbine and explain its working.
10. Explain the operating principle of Brayton cycle with a schematic diagram p-v and T-s diagrams.
11. Why Re-heater is necessary in gas turbine? What are its effects
12. What are the requirements of a good combustion chamber for a gas turbine?
13. Explain with neat sketch the gas turbine cycles with intercooling and reheating and what will be the condition of maximum output

Numerical Problems:

1. A simple gas turbine cycle works with a pressure ratio of 8. The compressor and turbine inlet temperatures are 300 K and 800 K respectively. If the volume flow rate of air is 250 m3/s, compute the power output and thermal efficiency
2. A constant pressure open cycle gas turbine plant works between temperature range of 15ºC and 700ºC and pressure ratio of 6. Find the mass of air circulating in the installation, if it develops 1100 kW. Also find the heat supplied by the heating chamber.

ASSIGNMENT QUESTIONS

3. In a gas turbine plant, air is drawn at 1 bar, 150 C and the pressure ratio is 6. The expansion takes place in two turbines. The efficiency of compressor is 0.82, high pressure turbine is

   0.85 and low pressure turbine is 0.84. The maximum cycle temperature is 6250 C. Calculate

   i) Pressure and temperature of gases entering the low pressure turbine.

   ii) Net power developed

   iii) Work ratio

   iv) Thermal efficiency. Work output of high pressure turbine is equal to compressor work
4. In an air standard regenerative gas turbine cycle the pressure ratio is 5. Air enters the compressor at 1 bar, 300 K and leaves at 490 K. The maximum temperature in the cycle is 1000 K. Calculate the cycle efficiency, given that the efficiency of regenerator and the adiabatic efficiency of the turbine are each 80%. Assume for air, the ratio of specific heats is 1.4. Also show the cycle on T-S diagram.
5. A gas turbine unit receives air at 1 bar and 300 K and compresses it adiabatically to 6.2 bar. The compressor efficiency is 88%. The fuel has a heating value of 44186 KJ/kg and

   the fuel air ratio is 0.017 KJ/kg of air. The turbine efficiency is 90 %. Calculate the work of turbine and compressor per kg of air compressed and thermal efficiency. Take Cp=1.005 KJ/kg K, γ=1.4 for the compression process, Cp=1.147 KJ/kg K, γ=1.33 for the expansion process.

ASSIGNMENT QUESTIONS

1. a) Describe with neat sketches the working of a simple constant pressure open cycle gas turbine.

   b) Discuss the relative advantages and disadvantages of gas turbines and steam turbines.
2. Describe with neat diagram a closed cycle gas turbine and explain advantages, disadvantages and applications.
3. Explain with neat sketch the gas turbine cycles with intercooling and reheating and what will be the condition of maximum output.
4. Explain about the open cycle and closed cycle turbines with neat sketches and also draw P-V & T-S diagrams.

UNIT 5

JET PROPULSION & ROCKETS

Course Objective:

Applications and the principles of thermodynamics to components and systems.

Course Outcome:

Develop problem solving skills through the application of thermodynamics.

5
JET PROPULSION ENGINES

5.1 Introduction

Jet propulsion, similar to all means of propulsion, is based on Newton's Second and Third laws of motion.

The jet propulsion engine is used for the propulsion of aircraft, missile and submarine (for vehicles operating entirely in a fluid) by the reaction of jet of gases which are discharged rearward (behind) with a high velocity. As applied to vehicles operating entirely in a fluid, a momentum is imparted to a mass of fluid in such a manner that the reaction of the imparted momentum furnishes a propulsive force. The magnitude of this propulsive force is termed as thrust.

For efficient production of large power, fuel is burnt in an atmosphere of compressed air (combustion chamber), the products of combustion expanding first in a gas turbine which drives the air compressor and then in a nozzle from which the thrust is derived. Paraffin is usually adopted as the fuel because of its ease of atomisation and its low freezing point.

Jet propulsion was utilized in the flying Bomb, the initial compression of the air being due to a divergent inlet duct in which a small increase in pressure energy was obtained at the expense of kinetic energy of the air. Because of this very limited compression, the thermal efficiency of the unit was low, although huge power was obtained. In the normal type of jet propulsion unit a considerable improvement in efficiency is obtained by fitting a turbo–compressor which will give a compression ratio of at least 4 : 1.

5.2 Classification

Jet propulsion engines are classified basically as to their method of operation as shown in fig. 5–1. The two main catagories of jet propulsion systems are the *atmospheric*

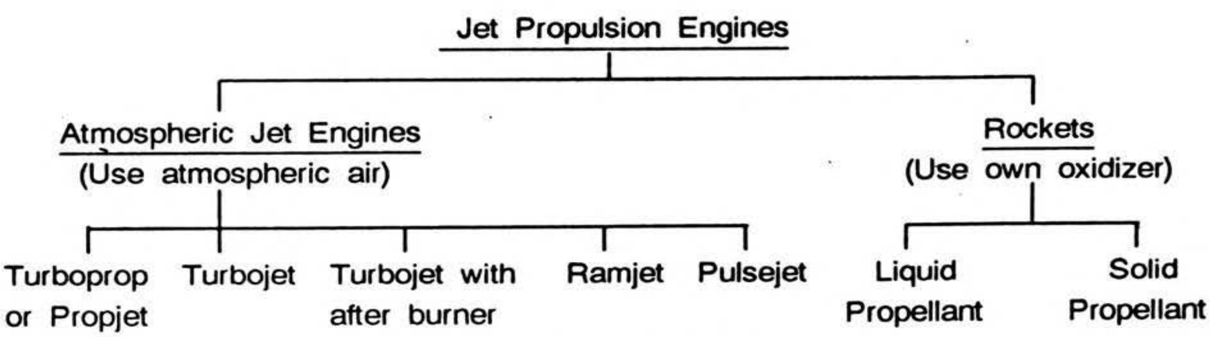


Fig. 5 – 1. Jet propulsion engines.

jet engine and *rocket*. Atmospheric jet engines require oxygen from the atmosperic air for combustion of fuel, *i.e.* they are dependent on atmospheric air for combustion.

The rocket engine carries its own oxidizer for combustion of fuel and is, therefore, independent of the atmospheric air. Rocket engines are discussed in art. 5·6.

The turboprop, turbojet and turbojet with after burner are modified simple open cycle gas turbine engines. In turboprop thrust is not completely due to jet. Approximately 80 to 90 percent of the thrust in turboprop is produced by acceleration of the air outside the engine by the propeller (as in conventional aeroengines) and about 10 to 20 percent of the thrust is produced by the jet of the exhaust gases. In turbojet engine, the thrust is completely due to jet of exhaust gases. The turbojet with after burner is a turbojet engine with a reheater added to the engine so that the extended tail pipe acts as a combustion chamber.

The ramjet and pulsejet are aero–thermo–dynamic–ducts, i.e. a straight duct type of jet engine without compressor and turbine. The ramjet has the simplest construction of any propulsion engine, consisting essentially of an inlet diffuser, a combustion chamber and an exit nozzle of tail pipe. Since the ramjet has no compressor, it is dependent entirely upon ram compression.

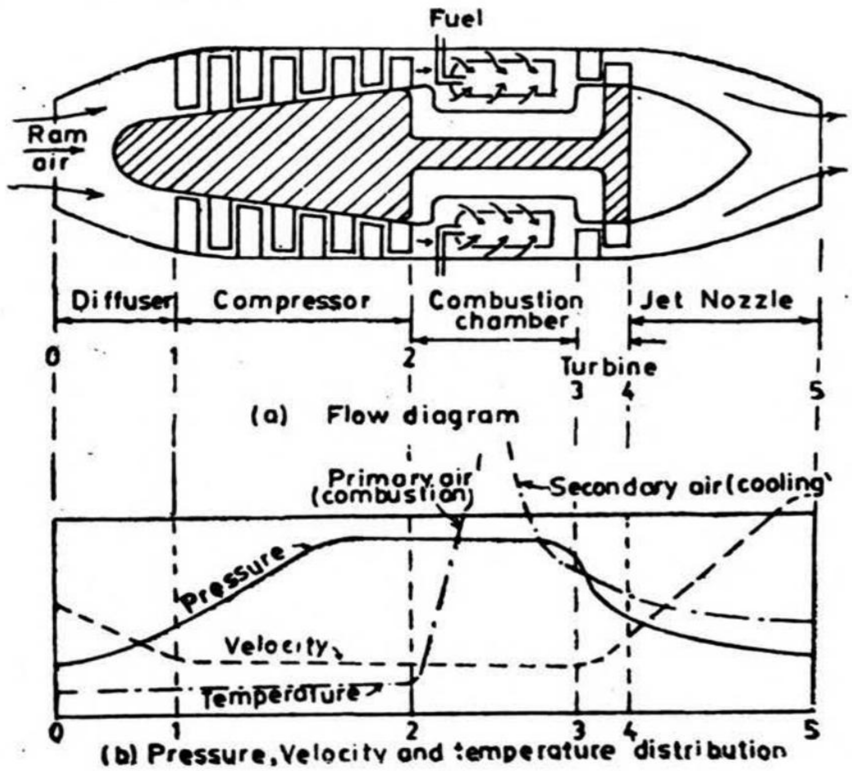


Fig. 5-2 Turbojet engine.

The pulsejet is an intermittent combustion jet engine and it operates on a cycle similar to a reciprocating engine and may be better compared with an ideal Otto cycle rather than the Joule or Bryton cycle. From construction point of view, it is some what similar to a ramjet engine. The difference lies in provision of a mechanical valve arrangement to prevent the hot gases of combustion from going out through the diffuser.

5.3 Turbojet Engine

The turboject engine (fig..5–2) is similar to the simple open cycle constant pressure gas turbine plant (fig. 4–2) except that the exhaust gases are first partially expanded in the turbine to produce just sufficient power to drive the compressor. The exhaust gases leaving the turbine are then expanded to atmospheric pressure in a propelling (discharge) nozzle. The remaining energy of gases after leaving the turbine is used as a high speed jet from which the thrust is obtained for forward movement of the aircraft.

Thus, the essential *components* of a turbojet engine are :

- An entrance air diffuser (diverging duct) in front of the compressor, which causes rise in pressure in the entering air by slowing it down. This is known as *ram*. The pressure at entrance to the compressor is about 1·25 times the ambient pressure.
- A rotary compressor, which raises the pressure of air further to required value and delivers to the combustion chamber. The compressor is the radial or axial type and is driven by the turbine.
- The combustion chamber, in which paraffin (kerosene) is sprayed, as a result of this combustion takes place at constant pressure and the temperature of air is raised.
- The gas turbine into which products of combustion pass on leaving the combustion chamber. The products of combustion are partially expanded in the turbine to provide necessary power to drive the compressor.
- The discharge nozzle in which expansion of gases is completed, thus developing the forward thrust.

A Rolls–Royce Derwent jet engine employs a centrifugal compressor and turbine of the impulse–reaction type. The unit has 550 kg mass. The speed attained is 960 km/hour.

5.3.1 Working Cycle : Air from surrounding atmosphere is drawn in through the diffuser, in which air is compressed partially by ram effect. Then air enters the rotary compressor and major part of the pressure rise is accomplished here. The air is compressed to a pressure of about 4 atmospheres. From the compressor the air passes into the annular combustion chamber. The fuel is forced by the oil pump through the fuel nozzle into the combustion chamber. Here the fuel is burnt at constant pressure. This raises the temperature and volume of the mixture of air and products of combustion. The mass of air supplied is about 60 times the mass of the fuel burnt. This excess air produces sufficient mass for the propulsion jet, and at the same time prevents gas temperature from reaching values which are too high for the metal of the rotor blades.

The hot gases from the combustion chamber then pass through the turbine nozzle ring. The hot gases which partially expand in the turbine are then exhausted through the discharge (propelling nozzle) by which the remaining enthalpy is converted into kinetic energy. Thus, a high velocity propulsion jet is produced.

The oil pump ad compressor are mounted on the same shaft as the turbine rotor. The power developed by the turbine is spent in driving the compressor and the oil pump.

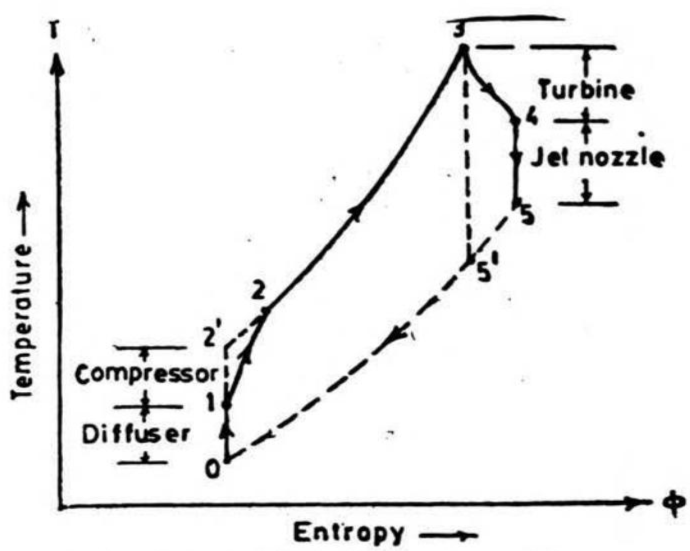


Fig. 5-3. A typical turbojet engine cycle on T – ϕ diagram.

Some starting devicé such as compressed air motor or electric motor, must be provided in the turbojet plant. Flight speeds upto 800 km per hour are obtained from this type of unit.

The basic thermodynamic cycle for the turbojet engine is the Joule or Brayton cycle as shown in $T - \Phi$ diagram of fig. 5–3. While drawing this cycle, following simplifying assumptions are made :

- There are no pressure losses in combustion chamber.
- Specific heat of working medium is constant.
- Diffuser has ram efficiency of 100 percent *i.e.*, the entering atmospheric air is diffused isentropically from velocity V_o to zero (V_o is the vehicle velocity through the air).
- Hot gases leaving the turbine are expanded isentropically in the exit nozzle *i.e.*, the efficiency of the exit nozzle is 100 percent.

5.3.2 Thrust Power and Propulsive Efficiency : The jet aircraft draws in air and expels it to the rear at a markedly increased velocity. The action of accelerating the mass of fluid in a given direction creates a reaction in the opposite direction in the form of a propulsive force. The magnitude of this propulsive force is defined as thrust. It is dependent upon the rate of change of momentum of the working medium i.e. air, as it passes through the engine.

The basis for comparison of jet engines is the thrust. The thrust, T of a turbojet engine can be expressed as,

$$T = m(V_i - V_o) \qquad \ldots (5.1)$$

where, m = mass flow rate of gases, kg/sec.,

V_j = exit jet velocity, m/sec., and,

V_o = vehicle velocity, m/sec.

The above equation is based upon the assumption that the mass of fuel is neglected. Since the atmospheric air is assumed to be at rest, the velocity of the air entering relative to the engine, is the velocity of the vehicle, V_O. The thrust can be increased by increasing the mass flow rate of gas or increasing the velocity of the exhaust jet for given V_o.

Thrust power is the time rate of development of the useful work achieved by the engine and it is obtained by the product of the thrust and the flight velocity of the vehicle. Thus, thrust power TP is given by

$$TP = T\,V_o = m(V_j - V_o)\,V_o \frac{\text{N·m}}{\text{sec.}} \qquad \ldots (5.2)$$

The kinetic energy imparted to the fluid or the energy required to change the momentum of the mass flow of air, is the difference between the rate of kinetic energy of entering air and the rate of kinetic energy of the exist gases and is called propulsive power. The propulsive power PP is given by

$$PP = \frac{m(V_j^2 - V_0^2)}{2} \text{ N.m/sec.} \qquad \ldots (5.3)$$

Propulsive efficiency is defined as the ratio of thrust power (*TP*) and propulsive power (*PP*) and is the measure of the effectiveness with which the kinetic energy imparted to the fluid is transformed or converted into useful work. Thus, propulsive efficiency η_p is given by

$$\eta_p = \frac{TP}{PP} = \frac{m(V_j - V_0)\,V_0}{1} \times \frac{2}{m(V_j^2 - V_0^2)}$$

$$\therefore \eta_p = \frac{2(V_j - V_o)V_o}{V_j^2 - V_o^2} = \frac{2V_o}{V_j + V_o} = \frac{2}{1 + \left(\frac{V_j}{V_o}\right)} \quad \ldots (5.4)$$

From the expression of n_p it may be seen that the propulsion system approaches maximum efficiency as the velocity of the vehicle approaches the velocity of the exhaust gases. But as this occurs, the thrust and the thrust power approach zero. Thus, the ratio of velocities for maximum propulsive efficiency and for maximum power are not the same. Alternatively, the propulsive efficiency can be expressed as

$$\eta_p = \frac{TP}{PP} = \frac{TP}{TP + K.E.\ \text{losses}} \quad \ldots (5.5)$$

Thermal efficiency of a propulsion is an indication of the degree of utilization of energy in fuel (heat supplied) in accelerating the fluid flow and is defined as the increase in the kinetic energy of the fluid (propulsive power) and the heat supplied. Thus,

$$\text{Thermal efficiency, } \eta_T = \frac{\text{Propulsive power}}{\text{Heat supplied}}$$

$$= \frac{\text{Propulsive power}}{\text{Fuel flow rate} \times \text{C.V. of fuel}} \quad \ldots (5.6)$$

The overall efficiency is the ratio of the thrust power and the heat supplied. Thus, overall efficiency is the product of propulsive efficiency and thermal efficiency. The propulsive and overall efficiencies of the turboject engine are comparable to the mechanical efficiency and brake thermal efficiency respectively, of the reciprocating engine.

Problem – 1 : *A jet propulsion unit, with turbojet engine, having a forward speed of 1,100 km/hr produces 14 kN of thrust and uses 40 kg of air per second. Find: (a) the relative exist jet velocity, (b) the thrust power, (c) the propulsive power, and (d) the propulsive efficiency.*

(a) Forward speed, $V_o = \frac{1{,}100 \times 1{,}000}{3{,}600} = 305{\cdot}55$ m/sec.

Using eqn. (5.1), thrust, $T = m(V_j - V_o)$

i.e., $14{,}000 = 40(V_j - 305{\cdot}55)$

$$\therefore V_j = \frac{14{,}000}{40} + 305{\cdot}55 = 350 + 305{\cdot}55 = 655{\cdot}55 \text{ m/sec.}$$

Thus relative exist jet velocity, V_j = **655·55 m/sec.**

(b) Using eqn. (5.2)

Thrust power, $TP = T \times V_o$

$= 14{,}000 \times 305{\cdot}55 = 42{,}77{,}700$ N.m/sec. or = **4,277.7 kN.m/sec.**

(c) Using eqn. (5.3),

$$\text{Propulsive power, } PP = \frac{m(V_2^2 - V_0^2)}{2}$$

$$= \frac{40\{(655{\cdot}55)^2 - (305{\cdot}55)^2\}}{2}$$

$= 6{,}727 \times 10^3$ N.m/sec = 6,727 KN.m/sec or **6,727 kW**

(d) Using eqn. (5.4),

is not effective and that there are pulsations created in the combustion chamber which affect the air flow in front of the diffuser.

Since the ram jet engine has no turbine, the temperature of the gases of combustion is not limited to a relatively low figure as in the turbojet engine. Air fuel ratios of around 15·1 are used. This produces exhaust temperatures in the range of 2000°C to 2200°C. Extensive research is being conducted on the development of hydrocarbon fuels that will give 30 percent more energy per unit volume than current aviation gasolines. Investigations are carried out to determine the possibility of using solid fuels in the ram jet and in the after burner of the turbojet engine. If powdered aluminium could be utilized as an aircraft fuel, it would deliver over 2·5 times as much heat per unit volume as aviation gasoline, while some other could deliver almost four times as much heat.

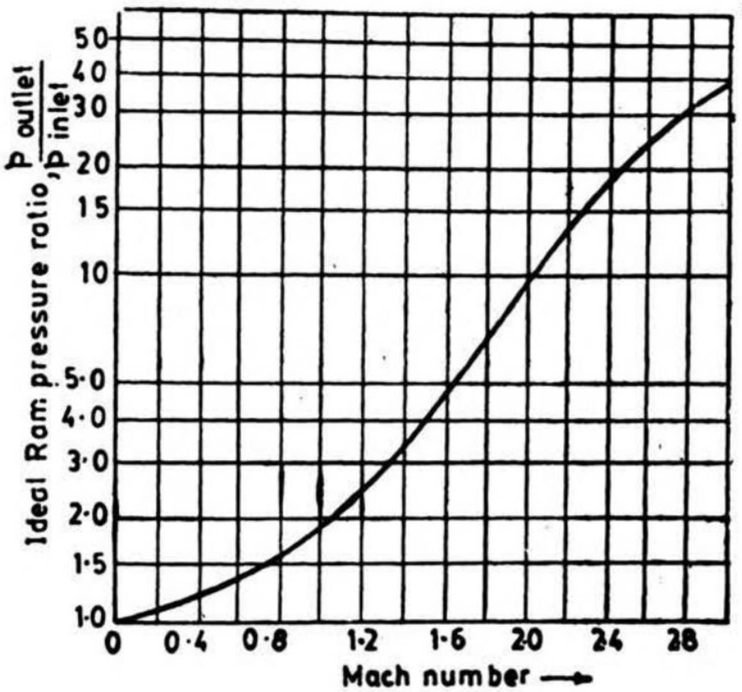


Fig. 5-5. Ram pressure ratio versus Mach number of vehicle for sea level condition.

The temperature, pressure and velocity of the air during its passage through a ram jet engine at supersonic flight are shown in fig. 5–4.

The cycle for an ideal ram jet, which has an isentropic entrance diffuser and exit nozzle, is the Joule cycle as shown by the dotted lines in fig. 5–6. The difference between the actual and ideal jet is due principally to losses actually encountered in the flow system. The sources of these losses are :

- Wall friction and flow separation in the subsonic diffuser and shock in the supersonic diffuser.
- Obstruction of the air stream by the burners which introduces eddy currents and turbulence in the air stream.
- Turbulence and eddy currents introduced in the flow during burning.
- Wall friction in the exit nozzle.

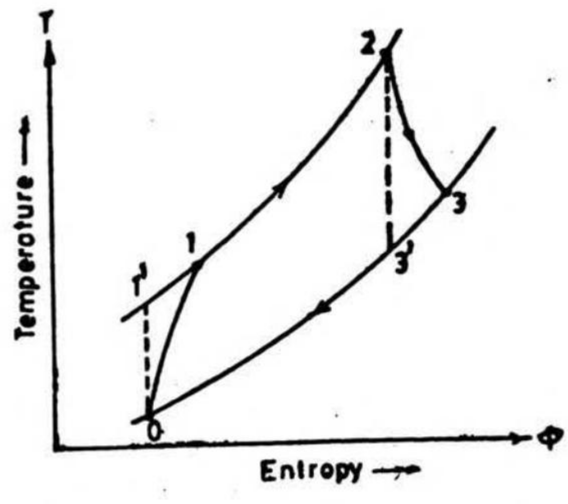


Fig. 5-6. T – ϕ diagram of Ram jet engine.

By far, the most critical component of the ram jet is the diffuser. Due to the peculiarities of steamline flow, a diffuser which is extremely

Because of the simplicity of the engine, the ram jet develops greater thrust per unit engine weight than any other propulsion engine at supersonic speed with the exception of the rocket engine. The thrust per unit frontal area increases both with the efficiency and the air flow through the engine; therefore much greater thrust per unit area is obtainable at high supersonic speeds. General performance of a ram jet engine in the subsonic range would have a specific fuel consumption between 0.6 to 0·8 kg fuel per N thrust – hr and a specific weight between 0·01 to 0·02 kg per N thrust. The supersonic ram jet engine has a specific fuel consumption between 0·25 to 0·04 and a specific weight between 0·01 to 0·04. Thus, the best performance of the ram jet engine is obtained at flights speeds of 1500 to 3500 km/hr.

5.5 Pulse Jet Engine

The pulse jet engine is somewhat similar to a ram jet engine. The difference is that a mechanical valve arrangement is used to prevent the hot gases of combustion from flowing out through the diffuser in the pulse jet engine.

Paul Schmidt patented principles of the pulse jet engine in 1930. It was developed by Germany during World–War–II, and was used as the power plant for "buzz bomb".

The turbojet and ram jet engines are continuous in operation and are based on the constant pressure heat addition (Bryton) cycle. The pulse jet is an intermittent combustion engine and it operates on a cycle similar to a reciprocating engine and may be better compared with an ideal Otto cycle rather than the Joule or Bryton cycle.

The compression of incoming air is accomplished in a diffuser. The air passes through the spring valves and is mixed with fuel from a fuel spray located behind the valves. A spark plug is used to initiate combustion but once the engine is operating normally, the spark is turned off and residual flame in the combustion chamber is used for ignition. The engine walls also may get hot enough to initiate combustion.

The mechanical valves which were forced open by the entering air, are forced shut when the combustion process raises the pressure within the engine above the pressure in the diffuser. As the combustion products cannot expand forward, they move to the rear at high velocity. The combustion products cannot expand forward, they move to the rear at high velocity. When the combustion products leave, the pressure in the combustion chamber drops and the high pressure air in the diffuser forces the valves open and fresh air enters the engine.

Since the products of combustion leave at a high velocity there is certain scavenging of the engine caused by the decrease in pressure occasioned by the exit gases. There is a stable cycle set up in which alternate waves of high and low pressure travel down the engine. The alternating cycles of combustion, exhaust, induction, combustion, etc. are related to the acoustical velocity at the temperature prevailing in the engine. Since the temperature varies continually, the actual process is complicated, but a workable assumption is that the tube is acting similar to a quarter wave length organ pipe. The series of pressure and rarefaction waves move down it at the speed of sound for an assumed average temperatures.

The frequency of the combustion cycle may be calculated from the following expression:

$$f = \frac{a}{4L} \text{ cycles/sec.} \qquad \ldots (5.7)$$

where, $a = \sqrt{\gamma RT}$ = sound velocity in the medium at temperature, T, and

L = length of engine (from valves to exit).

efficient at a given speed may be quite inadequate at another velocity.

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The frequency of the combustion cycle may be calculated from the following expression:

^ = 4 l cvc*es/sec-

where, a = VfTTT = sound velocity in the medium at temperature, *T,* and

L = length of engine (from valves to exit).

A serious limitation placed upon pulse jet engine is the mechanical valve arrangement. Unfortunately, the valves used have resonant frequencies of their own, and under certain conditions, the valve will be forced into resonant vibration and will be operating when they should be shutting. This limitation of valves also limits the engine because the gas goes out of the diffuser when it should go out of the tail pipe.

Despite the apparent noise and the valve limitation, pulse jet engines have several *advantages* when compared to other thermal jet engines.

- The pluse jet is very inexpensive when compared to a turbojet.
- The pulse jet produces static thrust and produces thrust in excess of drag at much lower speed than a ram jet.
- The potential of the pulse jet is quite considerable and its development and research may well bring about a wide range of application.

5.6 Rocket Motors

The jet propulsion action of the rocket has been recognised for long. Since the early beginning, the use of rockets has been in war time as a weapon and in peace time as a signaling or pyrotechnic displays. Although, the rocket was employed only to an insignificant extent in World War–I, marked advances were made by the research that was undertaken at that time. In World War–II, the rocket became a major offensive weapon employed by all warring powers. Rockets and rocket powered weapons have advanced to a point where they are used effectively in military operations.

Rocket type engine differs from the atmospheric jet engine in that the entire mass of the jet is generated from the propellant carried within the engine i.e. the rocket motor carries both the fuel and the oxidizing agent. As a result, this type of engine is independent of the atmospheric air that other thermal jet engines must rely upon. From this point of view rocket motors are most attractive. There are, however, other operational features that make rocket less useful. Here, the fundamentals of rocket motor theory and its applications are discussed.

Rocket engines are classified as to the type of propellant used in them. Accordingly, there are two major groups:

One type belonging to the group that utilizes liquid type propellants and other group that uses solid type propellants.

The basic theory governing the operation of rocket motor is applied, equally to both the liquid and the solid propellant rocket.

Rocket propulsion, at this time, would not be regarded as a competitor of existing means for propelling airplanes, but as a source of power for reaching objectives unattainable by other methods. The rocket motors are under active development programmes for an increasing number of applications. Some of these *applications* are :

- Artillery barrage rockets,
- Anti–tank rockets,
- All types of guided missiles,
- Aircraft launched rockets,
- Jets assisted take–off for airplanes,
- Engines for long range, high speed guided missiles and pilotless aircrafts, and
- Main and auxiliary propulsion engines on transonic airplanes.

It will be repeated again that the rocket engine differs from the other jet propulsion engines in that the entire mass of the gases in the jet is generated from the propellants carried within the engine. Therefore, it is not dependent on the atmospheric air to furnish the oxygen for combustion. However, since the rocket carries its own oxidiser, the propellant consumption is very high.

The particular advantages of the rocket are :

. . Its thrust is practically independent of its environments.

. . It requires no atmospheric oxygen for its operation.

. . It can function even in a vacuum.

. . It appear to be the simplest means for converting the themochemical energy of a propellant combination (fuel plus oxidizer) into kinetic energy associated with a jet flow gases.

Despite its apparent simplicity, the development of a reliable rocket system must be light in weight and the rocket motor must be capable of sustained operation in contact with gases at temperature above 2800° C and at appreciable pressures.The problem of materials in consequently a major one. Furthermore, owing to the enormous energy releases involved, problem of ignition, smooth start up, thrust control, cooling etc. arise.

A major problem of development of rocket is selection of suitable propellant to give maximum energy per premium total weight (propellant plus containing vessels) and convenience factors such as a safety in handling, dependability, corrosive tendencies, cost, availability and storage problems. In general, it can be stated that there is a wide variety of fuels that are satisfactory for rocket purpose, but choice of oxidizers is at present distinctly limited.

5.6.1 Basic Theory : Figure 5–7 shows a schematic diagram of a liquid bi–propellant rocket engine. It consists of an injection system, a combustion chamber, and an exit nozzle. The oxidizer and fuel burnt, in the combustion chamber produces a high pressure. The pressure produced is governed by

– Mass rate of flow of the propellants,

– Chemicals characteristics of the propellants, and

– Cross–section area of the nozzle throat.

The gases are ejected to the atmosphere at supersonic speeds through the nozzle. The enthalpy of high pressure gases is converted into kinetic energy. The reaction to the ejection of the high velocity, produces the thrust on the rocket engine.

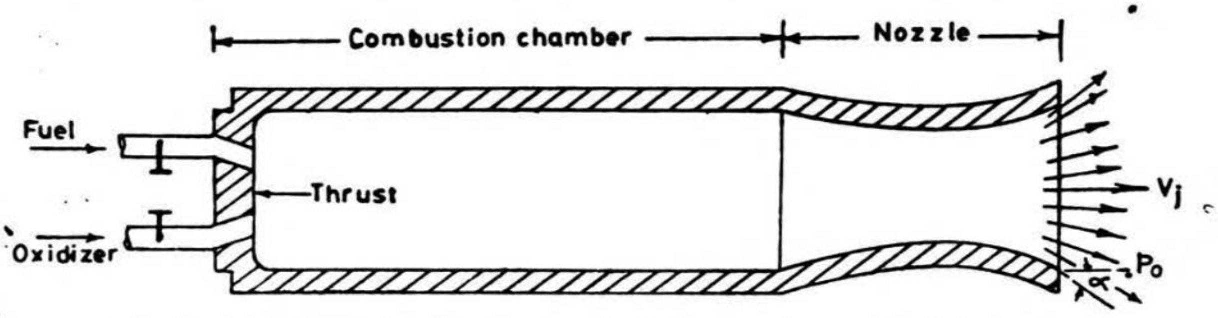

Fig. 5-7. Schematic diagram of a liquid bi-propellant uncooled rocket motor.

The *thrust* developed is a resultant of the pressure forces acting upon the inner and the outer surface of the rocket engine. The resultant internal force acting on the engine is given by

Resultant force = $m_p\ V_j + p_j\ A_j\ N$

where, m_p = Mass rate of propellant consumption, kg/sec,

V_j = Jet velocity relative to nozzle, m/sec,

V_{xj} = Average value of the x–component of the velocity of gases crossing, A_j,

p_j = Exist static pressure, N/m^2, and

A_j = Exit area of nozzle, m^2.

The resultant external forces acting on the rocket engine are $p_o A_o$, where p_o is the atmospheric pressure in N/m^2. The thrust which is a resultant of the total pressure forces becomes

$$T = m_p V_{xj} + A_j (p_j - p_o)\ N \qquad \ldots (5.8)$$

Let V_j = the exit velocity of the rocket gases, assumed constant and

let $V_{xj} = \lambda V_j$. Then, eqn. (5.8) becomes

$$T = \lambda m_p V_j + A_j (p_j - p_o)\ N \qquad \ldots (5.9)$$

The coefficient λ is the correction factor for the divergence angle *a* of the exit conical section of the nozzle. λ is given by

$$\lambda = \frac{1 - \cos 2\alpha}{4(1 - \cos\alpha)} = \frac{1}{2}(1 + \cos\alpha) \qquad \ldots (5.10)$$

Equation (5.8) shows that thrust of a rocket engine increases as the atmospheric pressure decreases. Therefore, maximum thrust will be obtained when P_o= 0, *i.e., rocket engine produces maximum thrust when operating in a vacuum.*

In testing a rocket engine, thrust and propellant consumption for a given time are readily measured. It is convenient then, to express the thrust in terms of the mass rate of flow of propellant and an effective jet velocity, V_{ej}

i.e., Thrust, $T = m_p \times V_{ej}$ $\qquad \ldots (5.11)$

The *effective jet exit velocity* is a hypothetical velocity and for convenience in test work it is defined from eqns. (5.9) and (5.11) as under :

$$V_{ej} = \lambda V_j + \frac{A_j}{m_p}(p_j - p_o)\ \text{m/sec.} \qquad \ldots (5.12)$$

The effective jet exit velocity has become an important parameter in rocket motor performance.

The *thrust power, TP* developed by a rocket motor is defined as the thrust multiplied by the flight velocity, V_o.

$$TP = T V_o = m_p \cdot V_{ej} \cdot V_o\ \text{N.m/sec.} \qquad \ldots (5.13)$$

The *propulsive efficiency,* η_p is the ratio of the thrust power to propulsive power supplied. The propulsive power is the thrust power plus the kinetic energy lost in the exhaust,

i.e., K.E. Loss $= \frac{1}{2} m_p (V_{ej} - V_o)^2$ N.m/sec.

Therefore, the propulsive efficiency may be expressed as

$$\eta_p = \frac{TP}{TP + \text{K.E. Loss}} = \frac{m_p V_{ej} V_o}{m_p V_{ej} V_o + \frac{1}{2} m_p (V_{ej} - V_o)^2}$$

$$\therefore \eta_p = \frac{2\,(V_o/V_{ej})}{1 + (V_o/V_{ej})^2} \qquad \ldots (5.14)$$

Specific Impulse, I_{sp} has become an important parameter in rocket motor performance and is defined as the thrust produced per unit mass rate of propellant consumption.

$$I_{sp} = \frac{T}{m_p} = \frac{m_p \cdot V_{ej}}{m_p} = V_{ej} \qquad \ldots (5.15)$$

Specific impulse, with the units, Newtons of thrust produced per kg of propellant burned per second, gives a direct comparison as to the effectiveness among propellants. It is desirable to use propellants with the greatest possible specific impulse, since, this allows a greater useful load to be carried for a given overall rocket weight.

5.6.2. Types of Rocket Motors : The propellant employed in a rocket motor may be a solid, two liquids (fuel plus oxidizer), or materials containing an adequate supply of available oxygen in their chemical composition (monopropellant). Solid propellants are used for rockets which are to operate for relatively short periods, upto possibly 45 seconds. Their main application is to projectiles, guided missiles, and the assisted take-off aircraft.

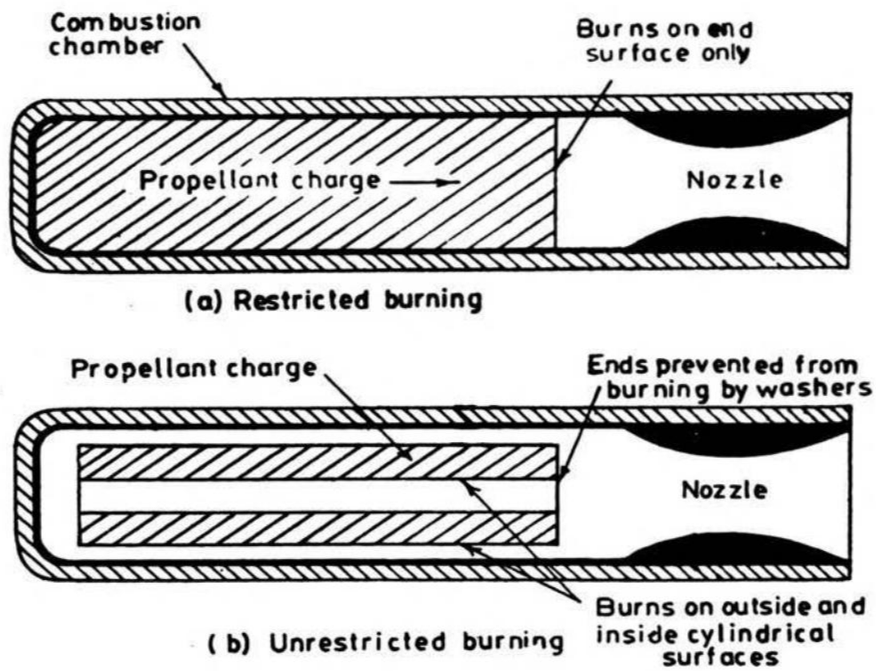

Fig. 5-8 Schematic diagram of a solid propellant rocket.

Solid propellant rockets (fig. 5–8) have been of two basic types :

. . Unrestricted burning types for projectiles and launching rockets; and

. . Restricted burning types for assisted take–off of aircraft and for propelling missiles.

In the unrestricted burning rocket [fig. 5–8(a)] all surfaces of the propellant grain except the ends are ignited; in restricted burning rockets [fig. 5–8(b)] only one surface of the propellant is permitted to burn. Liquid propellant rockets utilizes liquid propellants which are stored in the containers outside the combustion chamber. The basic theory of operation of this type of rocket is same as that for solid propellant rocket. Liquid propellant rockets were developed in order to overcome some of the undesirable features of the

which ar

solid propellant rockets such as short duration of thrust, and no provisions for adequate cooling or control of the burning after combustion starts. Here, the propellant in the liquid

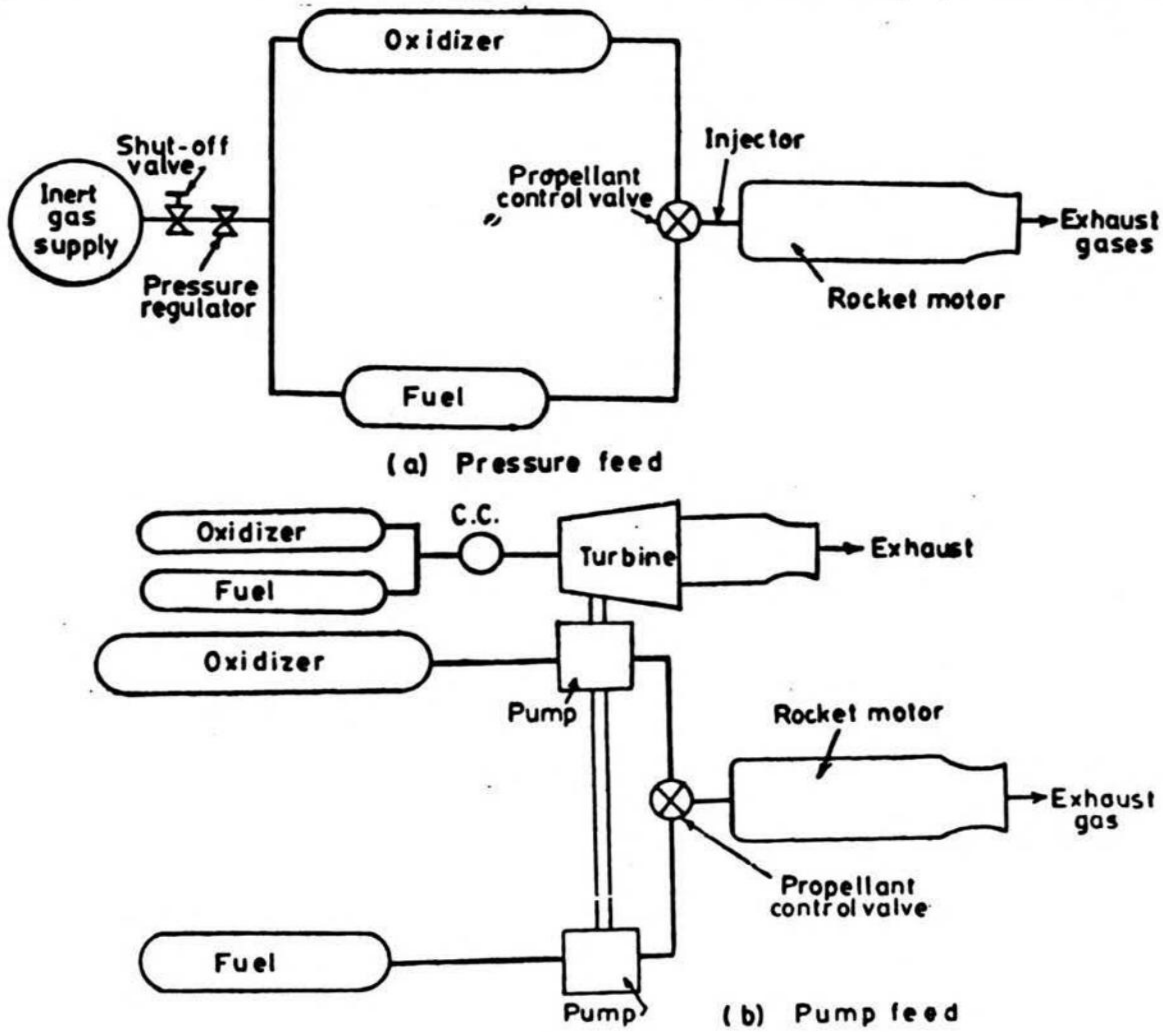

Fig. 5-9. Schematic diagrams of bi-propellant rocket system.

state is injected into a combustion chamber, burned and exhausted at a high velocity through the nozzle. The liquid propellant is also used to cool the rocket motor by circulation of fuels around the walls of the combustion chamber and around the nozzle. Certain liquid fuel, however, such as hydrogen peroxide, burn at such temperatures that no cooling is necessary. Figure 5–9 shows schematic diagrams of pressure feed and pump feed liquid bipropellant rocket systems.

Problem–2 : *The effective exit jet velocity of a rocket is 3000 m/sec, the forward flight velocity is 1500 m/sec and the propellant consumption is 70 kg per sec. Calculate : (a) Thrust, (b) Thrust power, (c) Specific impulse, (d) Specific propellant consumption, and (e) Propulsive efficiency of the rocket.*

(a) Using eqn. (5.11),

Thrust, $T = m_p \times V_{ej} = 70 \times 3{,}000 = 2{,}10{,}000$ N or **210 kN**

(b) Using eqn. (5.13),

Thrust power, $TP = T\ V_o = 2{,}10{,}000 \times 1{,}500 = \mathbf{315 \times 10^6}$ **N.m/s**

(c) Using eqn. (5.14),

Specific impluse, $I_{sp} = \frac{T}{m_p} = \frac{m_p \cdot V_{ej}}{m_p} = V_{ej} =$ **3,000 N.s/kg**

(d) Specific propellant consumption $= \frac{m_p}{T} = \frac{m_p}{m_p V_{ej}} = \frac{1}{V_{ej}} = \frac{1}{3,000}$

$$= \mathbf{3 \cdot 3 \times 10^{-4}\ kg/N.s}$$

(e) Using eqn. (5.14),

Propulsive efficiency, $\eta_p = \frac{2(V_o/V_{ej})}{1 + (V_o/V_{ej})^2}$

$$= \frac{2(1500/3000)}{1 + (1500/3000)^2} = 0 \cdot 8 \text{ i.e., } \mathbf{80\%}$$

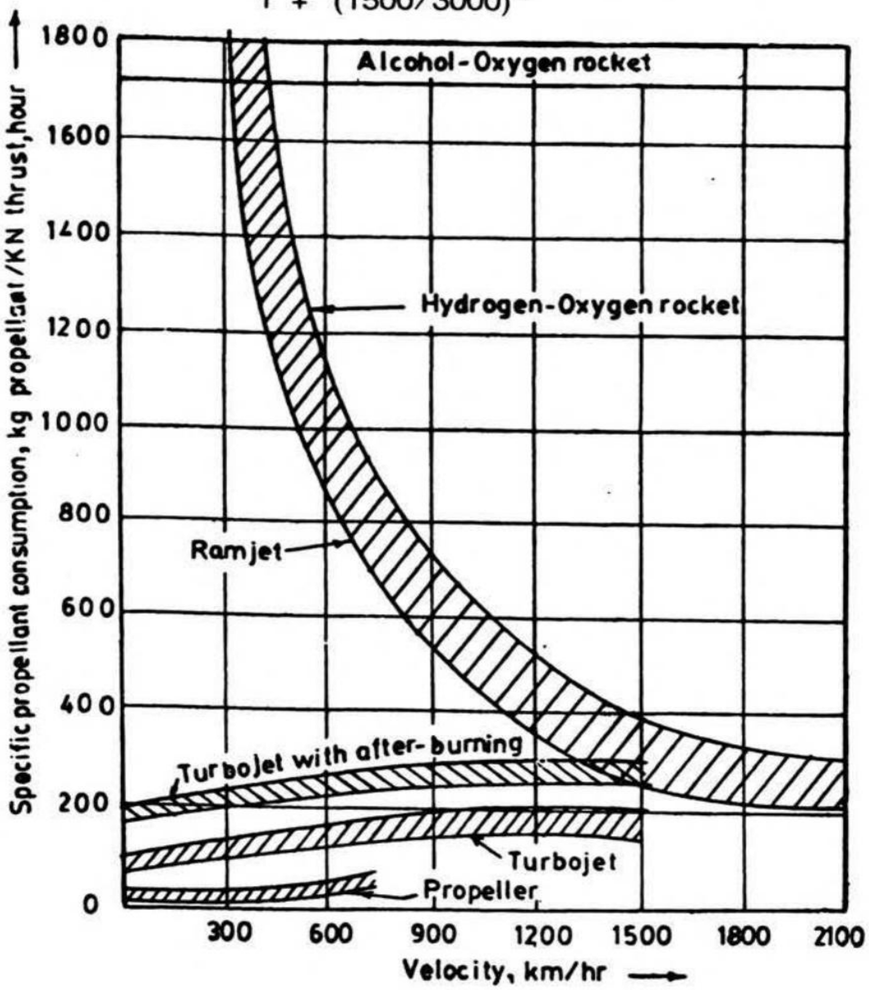

Fig. 5-10. Propellant or fuel consumption versus flight speed for different propulsion systems.

5.7 Comparison of the Various Propulsion Systems

Figure 5–10 shows the specific propellant consumption in kg per kN thrust versus speed for different engines. The curves in this figure indicate that the use of rocket engines to power air planes, as we know them today, is not feasible because of their high fuel consumption. Also, the use of ram jet engines is not economical at lower than 1500 km/hr vehicle speeds.

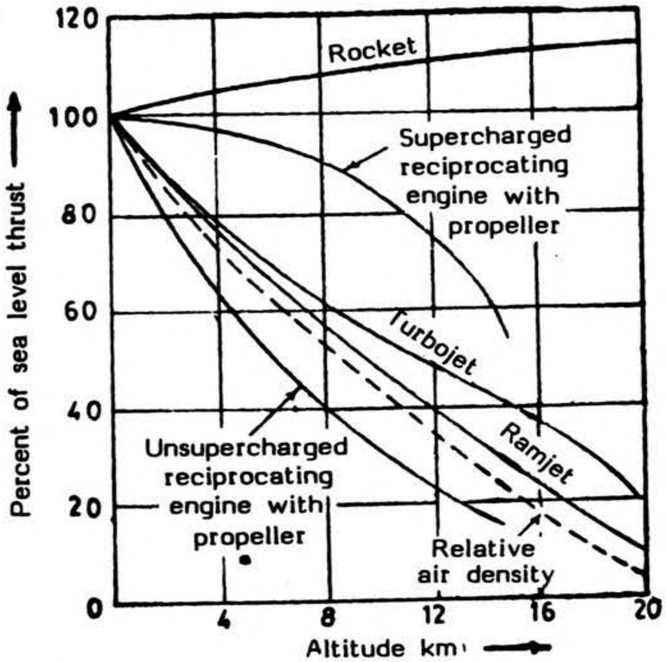

Fig. 5–11 Variation of thrust with altitude for different propulsion systems.

Figure 5–11 shows variation of thrust with altitude for different propulsion systems. It may be noted that the thrust of rocket motor increases with altitude while the thrust of other types of vehicles decreases with altitude.

Figure 5–12 gives relative picture of the probable operating envelope of the various propulsion systems.

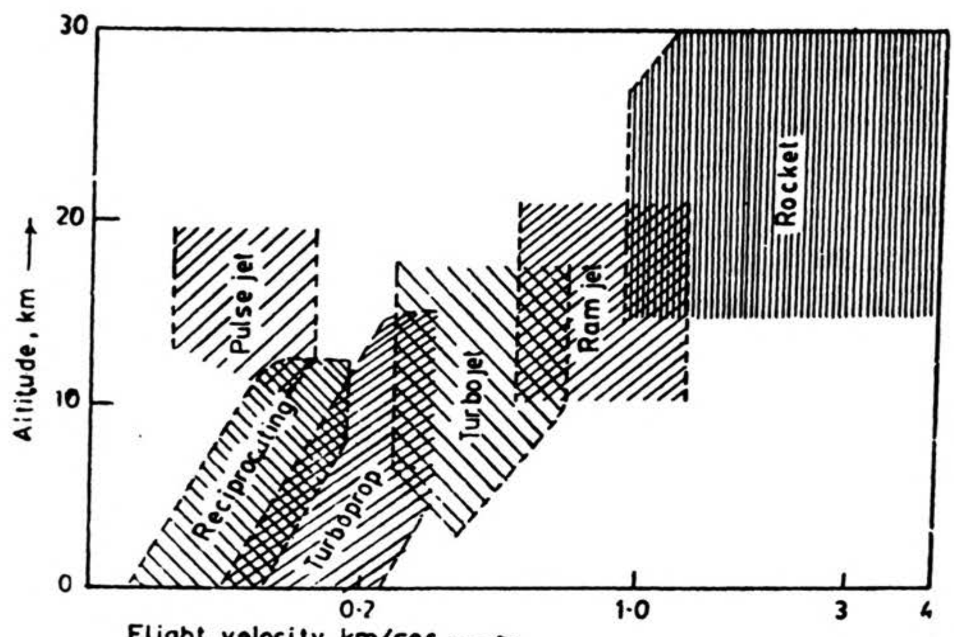

Fig. 5-12 Comparison of probable best performance for various propulsion engines.

INDUSTRIAL APPLICATIONS

INDUSTRIAL APPLICATIONS

- IN AIRCRAFT- Fighter plane, Missiles, Rocket, Airplane.
- Jet propulsion, land and sea transport, racing car.
- The first use of the jet engine was to power military aircraft.
- The General electric company used a "turboprop" jet engine to run an electric generator.
- The jet engine is not only used on aircraft but on boats, where water jets are used to propel the boat forward.
- Normal type of jet engine is used for domestic purpose i.e. Traveling, carrying goods etc.

An aircraft using this type of jet engine could dramatically reduce the time which it takes to travel from one place to another, potentially putting any place on Earth within a 90-minute flight.

Scramjet vehicle has been proposed for a single stage to tether vehicle, where a Mach 12 spinning orbital tether would pick up a payload from a vehicle at around 100 km and carry it to orbit

Rocket applications

1. Satellites in space serve air communication
2. Spacecraft
3. Missiles
4. Jet assisted air planes
5. Pilotless aircraft

TUTORIAL QUESTIONS

Theory Questions:

1. What are the different rocket propulsion systems? Brief the working differences between the propeller-jet, turbojet and turbo-prop.
2. With a neat diagram explain the working of rocket engine
3. Describe briefly about thrust augmentation method used in propulsion.
4. With a neat sketch, explain the working of turbo jet engine.
5. Differentiate between solid propellant and liquid propellant rocket engines.
6. What are the applications of pulse jet engines
7. Give the difference between ramjet and pulse jet engines
8. What are composite and homogeneous solid propellants? How do they work? State their merits and demerits.
9. What is the essential difference between rocket propulsion and turbo-jet propulsion?
10. Write a detailed classification of rockets. Explain liquid propellant rocket with a neat sketch Define and explain the terms:
    i. Thrust
    ii. Thrust power,
    iii. Effective jet exit velocity,
    iv. Propulsive efficiency related to turbojet engines.
11. What are the various applications of rockets?
12. Explain the advantages and disadvantages of bipropellants used in rocket engines over monopropellants.
2. Derive expressions for the thrust and propulsion efficiency of rockets and compare with those of turbojet

Numerical Problems:

1. A jet propulsion system has to create a thrust of 100 tones to move the system at a velocity of 700 km/hr. If the gas flow rate through the system is restricted to a maximum of 30 kg/s. find the exit gas velocity and propulsive efficiency.

2. In a jet propulsion unit, initial pressure and temperature to the compressor are 1.0 bar and 100C. The speed of the unit is 200m/s. The pressure and temperature of the gases before entering the turbine are 7500 C and 3 bar. Isentropic efficiencies of compressor and turbine are 85% and 80%. The static back pressure of the nozzle is 0.5 bar and efficiency of the nozzle is 90%. Determine (a) Power consumed by compressor per kg of air. (b)Air-fuel ratio if calorific value of fuel is 35,000 kJ/kg. Cp of gases=1.12 kJ/kg K, _ =1.32 for gases.
3. A turbo-jet engine flying at a speed of 960 km/h consumes air at the rate of 54.5 kg/s. calculate i). Exit velocity of the jet when the enthalpy change for the nozzle is 200 KJ/kg and velocity coefficient is 0.97. ii).fuel flow rate in kg/s when air fuel ratio is 75:1 iii). Thrust specific fuel consumption iv). Propulsive power v). Propulsive efficiency.
4. A simple turbine jet unit was tested when stationary and the ambient conditions were 1bar and 150C. The pressure ratio for the compressor was 4:1. A fuel consumption of 0.37kg/s was obtained for an air flow of 23kg/s. Calculate the thrust produced if the exhaust gases from the turbine were expanded to atmospheric pressure in a convergent nozzle. Assume the following data:

   Isentropic efficiency of compressor-80% Isentropic efficiency of turbine-85%

   Efficiency of nozzle-93% Transmission efficiency-98%

   Calorific value of fuel-42000kJ/kg Assuming working fluid to be air throughout.

5. In a turbojet, air is compressed in an axial compressor at inlet conditions of 1 bar and 1000C

   3.5 bar. The final temperature is 1.25 times that for isentropic compression. The temperature of gases at inlet to turbine is 4800C. The exhaust gases from turbine are expanded in a velocity of approach is negligible and expansion may be taken to be isentropic in both turbine and nozzle. Value of gas constant R and index r are same for air and flue gases.

   Determine

   i) Power required to drive the compressor per kg of air/sec

   ii) Air-fuel ratio if the calorific value of fuel is 42,000 kJ/kg

   iii) Thrust developed / kg of air / sec.

ASSIGNMENT QUESTIONS

ASSIGNMENT QUESTIONS

1. Why is thrust augmentation necessary? What are the methods for thrust augmentation in a turbojet engine?
2. A turbo-jet engine flying at a speed of 960 km/h consumes air at the rate of 54.5 kg/s. calculate i). Exit velocity of the jet when the enthalpy change for the nozzle is 200 KJ/kg and velocity coefficient is 0.97. ii).fuel flow rate in kg/s when air fuel ratio is 75:1 iii). Thrust specific fuel consumption iv). Propulsive power v). Propulsive efficiency.
3. With a neat diagram explain the working of rocket engine
4. What is turbine and classify them?