Stages of combustion

The overall Air Fuel ratio in the aircraft gas turbine combustion chamber is in the range of 100: 1 while the Stoichoimetric ratio is approximately 15: 1. Hence the high-pressure air coming from compressor is introduced into the combustion chamber in three stages.
(a) Primary air which is about 15% of the total air is introduced into the primary zone. This initiates combustion of the atomized fuel. The air fuel ratio in this zone is around 15: 1.
(b) About 30% of the total air called as secondary air, is then introduced through the holes in the flame tube in the secondary zone to complete the combustion. In order to achieve high combustion efficiency, this air must be carefully injected at the right points in the process, to avoid chilling the flame locally and drastically reducing the reaction rate in the neighborhood.
(c) Finally the remaining quantity of air about 55% known as Tertiary air is mixed with the products of combustion. This dilutes the high fuel air ratio mixture and cools them down from about2000°C to the temperature to safe limits at the inlet of the turbine around 1000°C.

The fuel air mixture is ignited by means of a high voltage spark at the time of starting the engine. Once a flame is established, it propagates to other sections of the combustion chamber. Various methods are .employed to stabilize the flame. The Primary Zone has a flame stabilizer.

Sufficient turbulence is promoted so that the hot and cold streams are thoroughly mixed to give the desired outlet temperature distribution.

TYPES OF COMBUSTION CHAMBERS
1) Single-Can-type combustion chamber
2) Multiple chambers at Tanged around the circumference of the unit - Common practice for aircrafts.

Single-Can-Type Combustion Chamber

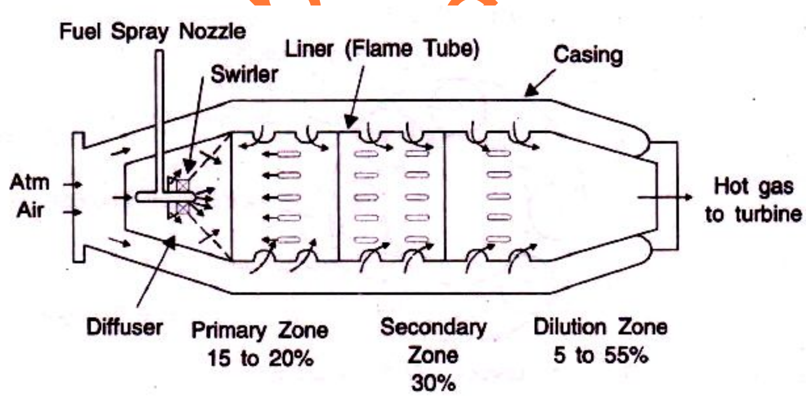

Figure 1.15

The combustion chamber has three zones: Primary Zone, Secondary Zone and Tertiary Dilution Zone.

As shown in Fig. 1.15 , fuel is injected in the same direction as the air stream and the primary air is introduced through twisted radial vanes, known as swirl vanes. This swirl vanes produce a vortex motion of the air which induces a region of low pressure along the axis of the chamber. This vortex motion is enhanced by injecting the secondary air through short tangential chutes in the flame tubes, instead of through plain holes. The net result is the production of recirculating flows i.e., the burning gases tend to flow towards the region of low pressure and some portion of them is swept around towards the jet of fuel as indicated by at Tows. A spark is initiated by a spark plug. This initiates the combustion and for this, there is an ignition system.

The liner is typically made of a Nickel alloy which can operate upto about 1100 K or of Cobalt alloy which may operate above 1200 K.

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Multiple Combustion Chambers

Advantages

• Easy to obtain the desired total combustion chamber volume with little increase in overall dimensions required.

• Individual chambers can be removed and replaced without disturbing the rest of the assembly.

• The design fits nicely in with the general diffuser and centrifugal compressor.

• Little difficulty in obtaining a fairly uniform gas temperature at exit from the chambers.

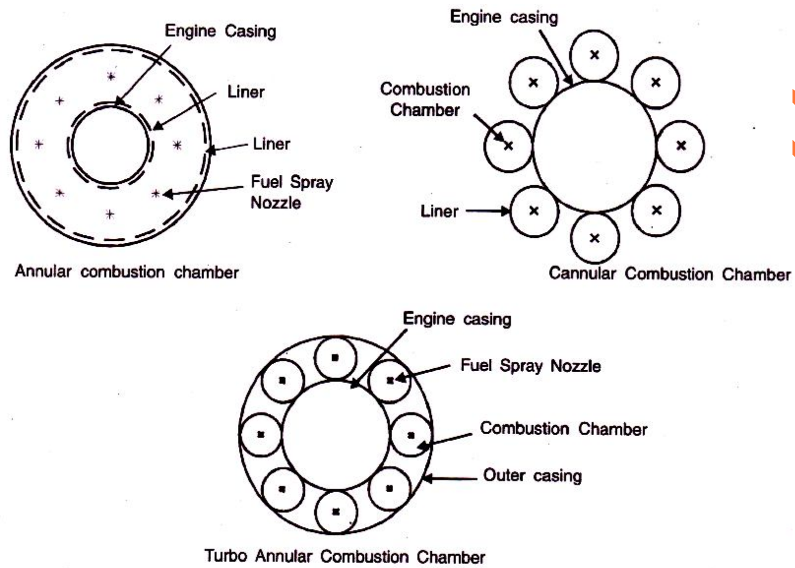

Figure 1.16

Different types

(1) Annular combustion chamber
(2) Cannular combustion chamber
(3) Turbo annular combustion chamber

Turbo Annular combustion chamber is a combination of Annular and Cannular chamber. This is best suited for Aircrafts. In this, separate combustion cans are placed in the space around the shaft connecting the compressor and the turbine, without increasing the frontal area. Each burner is a small annular chamber in that there is an inner chamber through which the air flows to cool the inside of the chamber.

In the Annular combustion chamber, the large curvature of the liner surface makes it more resistant to warping and higher strength while the Cannular arrangement gives efficient space utilization resulting in reduced diameter and weight. This type of combustion chamber is robust and has low pressure losses and is used for engines with higher pressure ratios.

solved problems

1) The diameter of the propeller of an aircraft is 2.5 m. It flies at a speed of 500 km/hr at an altitude of 8000 m (density of air 0.525 kg/m^3) with flight to jet speed ratio of 0.75. Determine (i) The air flow rate through the propeller (ii) Thrust produced (iii) Specific thrust (iv) Specific impulse (v) Thrust power.

Sol:

$$Flight\ Speed\ u = \frac{500 \times 1000}{3600} = 138.88\ m/sec$$

$$Speed\ ratio\ \sigma = \frac{u}{V_j} = \frac{Flight\ velocity}{Jet\ velocity}$$

$$\therefore V_j = \frac{u}{\sigma} = \frac{138.88}{0.75} = 185.185\ m/sec$$

Average Velocity of air through the propeller

$$V = \frac{u+V_j}{2} = \frac{138.88+185.185}{2} = 162.04\ m/sec$$

Air flow rate through propeller

$$\dot{m}_a = \varphi AV$$

$$= 0.525 \times \frac{\pi}{4}(2.5)^2 \times 162.04$$

$$= 417.58\ kg/sec$$

Thrust produced $F = \dot{m}_a\left(V_j - u\right)$

$$= 417.58(185.185 - 138.88)$$

$$= 19332.53\ N$$

Thrust Power $= F \times u$

$$= \frac{19332.53 \times 138.88}{1000} = 2685.07\ KW$$

Specific Thrust $= \frac{F}{\dot{m}_a} = \frac{19332.53}{417.58} = 46.296\ N$

Specific Impulse $= \frac{F}{\dot{w}} = \frac{specific\ thrust}{g}$

$$= \frac{46.296}{9.81} = 4.719\ sec$$

2) A Turboprop aircraft is flying at 720 km/hr at an altitude where the temperature is -18^0C Determine the specific power output and the thermal efficiency. Given specifications:-

Compressor pressure ratio 9

Maximum cycle temperature 800^0c

Intake duct efficiency 0.9

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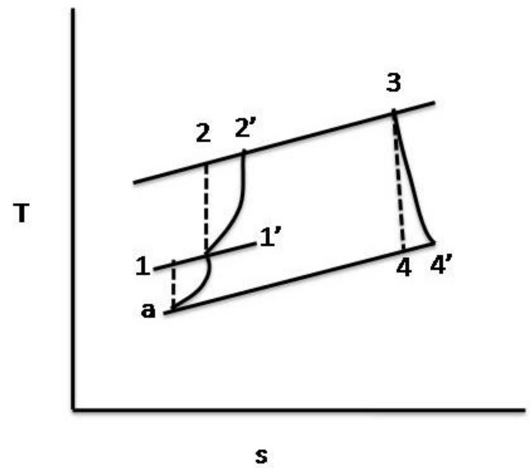
Isentropic efficiency of compressor 0.86

Isentropic efficiency of turbine 0.9

Mechanical efficiency 0.92

Neglecting the pressure loss in the combustion chamber. Assume that the exhaust gases leave the aircraft at 720 km/hr relative to the aircraft. For compression process C_p 1.005 KJ/kg K and K 1.4; for expansion process C_p 1.15 KJ/kg K and K 1.35



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Sol:

Flight speed = Air inlet velocity

$$= \frac{720\times1000}{3600} = 200\ m/sec$$

Intake duct (Diffuser) a-1

$$T'_1 = T_a + \frac{v_a^2}{2\ C_p\times1000\times Diffuser\ \eta}$$

$$= 255 + \frac{200^2}{2\times1.005\times1000\times0.9} = 274.9K$$

$$\frac{T_1}{T_a} = \left(\frac{p_1}{p_a}\right)^{\frac{k-1}{k}}$$

$$\therefore \frac{p_1}{p_a} = \left(\frac{274.9}{255}\right)^{\frac{1.4}{0.4}} = 1.268$$

Compressor $1' - 2'$

$$\frac{T_2}{T_1} = \left(\frac{p_2}{p_1}\right)^{\frac{k-1}{k}}$$

$$\therefore T_2 = 274.9\ (9)^{\frac{0.4}{1.4}} = 515\ K$$

$$\eta_c = \frac{T_2-T_1}{T'_2-T_1}$$

$$0.86 = \frac{515-274.9}{T'_2-274.9}$$

$$T'_2 = 554\ K$$

$$\text{Actual compressor work} = c_p(T'_2 - T'_1)\frac{1}{\eta_c}$$

$$= 1.005(554 - 274.9) \times \frac{1}{0.92} = 304.99\ KJ/kg\ \ air/sec$$

Turbine 3'-4'

$$\frac{p_3}{p_4} = \frac{p_2}{p'_1} = 9 \times 1.268 = 11.412$$

$$T_4 = T_3\left(\frac{p_4}{p_3}\right)^{\frac{k-1}{k}} = 1073\left(\frac{1}{11.412}\right)^{\frac{0.4}{1.4}} = 535.17\ K$$

$$\eta_t = \frac{T_3-T'_4}{T_3-T_4}$$

$$0.9 = \frac{1073-T'_4}{1073-535.17}$$

$$T'_4 = 588.95K$$

$$turbine\ work\ output = \mathrm{C_p}(T_3 - T'_4) = 1.15(1073 - 588.95) = 556.65\frac{kJ}{kg\ air/sec}$$

therefore specific power output = 251.66 kW/kg air

Combustion Chamber 2-3:

Heat supplied in the combustion chamber = $\mathrm{C_p}(T_3 - T'_2)$

H_s = 1.15(1073-554) = 596.85 kJ/kg air

$$Thermal\ efficiency = \frac{W_{Tnet}}{H_s} = \frac{251.66}{596.85} \times 100 = 42.16\%$$

3) Ramjet engine has the following data:

Altitude 6.5km; flight mach no. = 4 ; air-fuel ratio 60; calorific value of fuel used 45 MJ/kg. Diffuser inlet diameter 0.5m; ϒ=1.4; R = 287J/kg-K for both air and products of combustion . Diffuser efficiency is 0.85; combustor efficiency 0.98; Nozzle efficiency 0.95. Determine

i. ideal cycle efficiency
ii. flight speed
iii. air and fuel consumption
iv. diffuser pressure ratio
v. Maximum temperature in the engine
vi. Nozzle pressure ratio

At 6.5km altitude, air properties 0.44 atm; 246K; density 0.624kg/m^3.

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