Jet Aircraft Propulsion

Prof. Bhaskar Roy, Prof. A M Pradeep
Department of Aerospace Engineering,
IIT Bombay

Lecture 25

Combustion Parameters

Pressure losses

Combustion chamber pressure loss is due to two causes i) Skin friction, mixing and turbulence, and ii) The rise in temperature due to combustion. The later, the "fundamental pressure loss", arises due to increases in temperature, which means decrease in density and increase in local velocity of flow. Pressure loss is proportional to (velocity)^2.

Total Pressure loss co-efficient, across stations 1 & 2, being the inlet and the outlet to combustor,

$$\overline{\omega_{cc}} = \frac{P_{02} - P_{01}}{\frac{1}{2}.\rho.C_1^2}$$

For low velocity flow (incompressible) momentum equation of one dimensional frictionless flow in a duct of _constant cross sectional area_ A_{cc} , yields _Total Pr. loss coefficient_ in the form of

$$\frac{P_{02}-P_{01}}{\frac{1}{2}.\rho.C_1^2}=\left(\frac{\rho_1}{\rho_2}-1\right)=\left(\frac{T_1}{T_2}-1\right)=\left(\frac{T_{01}}{T_{02}}-1\right)$$

Since, T_{01}/T_{02} is of the order of 2 – 3, fundamental pressure loss coefficient is of the order of 1 to 2. Due to strong vortex formation and cross flow, artificially created to aid the process of evaporation and mixing, skin friction loss is quite high - about 25% of inlet dynamic head. Thus uniform exit temperature and low pressure loss are contradictory requirements.

Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay

Overall (Total) pressure loss can be expressed by an equation of the

$$\text{Pressure Loss Coefficient, } \overline{\omega}_{cc} = \Delta P_0 \Big/ \frac{1}{2}.\rho.C_1^2 = K_1 + K_2\left(T_{02} \big/ T_{01} - 1 \right)$$

Where, K_1 and K_2 are to be found for a combustion chamber in a test rig from a <u>cold run</u> and a <u>hot run</u>, and the final version of the equation can be used for a range of mass flows, pressure ratios & fuel flows.

Typical values of at design operating point for Cannular, Can annular and Annular combustion chambers are 35, 25 and 18 respectively. However the total pressure loss is about 4 – 7 % of the inlet total pressure.

Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay

Combustion efficiency

This is defined as

$$\eta_{cc} = \frac{\text{ideal f/a ratio for } \Delta T_{023}}{\text{real f/a ratio for } \Delta T_{023}} = \frac{\text{real } \Delta T_{023} \text{for as specified f/a}}{\text{ideal } \Delta T_{023} \text{for as specified f/a}}$$

- This can be measured experimentally. Generally it is of the order of 98 – 99 % at sea level.
- At high altitude as the operating pressure falls the combustion may not be as efficient.
- As a very fast burning process high pressure and temperature provides the necessary condition for combustion. As the incoming pressure and temperature falls the combustion may become less efficient.

Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay

Combustion intensity:-

It is insufficient to characterize combustion chambers on the basis of pressure loss and efficiency. The total amount of energy it can release for useful work must be a measure of its performance. Hence, the parameter called *combustion intensity* is introduced as

$$\text{Combustion intensity} = \frac{\text{Heat release rate}}{\text{Combustion Volume x Pressure}}$$

$$= \frac{Q}{A_{cc}.L_{cc}.P_3} \ldots\ldots \left(\frac{\text{kw}}{\text{m}^3.\,\text{kPa}}\right)$$

In aircraft systems the combustion intensity is of the order of 2 – 5 x10^4 kW/m^3-kPa

Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay

- The dimensions of the combustion chamber i.e. the length and diameter are chosen on the basis of combustion intensity and volume flow rate.
- Combustion intensity is chosen on the basis of engine power needed.
- Air velocity in the combustion chamber gives some idea of the length required.
- Weight and overall diameter of the engine are other parameters are that have to be kept in mind in choosing the type of combustor e.g. cannular or annular.
- Overall layout of the engine dictates the combustion chamber type, size and shape.

Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay

The *reaction rate* of a combustion process may be given as :

$$R_{comb} \propto P^{n}.f(T).e^{-1}[-E/(R.T)],$$

Where, n depends on the number of molecules involved and is 1.8 for hydrocarbon fuels.

E is the activation energy and

$f(T)$ relates to the various forms of molecular energy (rotational, translational and vibration).

The *reaction time* is inversely proportional to the reaction rate :

$$t_{Comb\text{-}reac} \propto P_{03}^{-n}$$

The *resident time* of the gas in the burner is given as

$$t_{res} = L/C_{av\text{-}c.c} = (\rho_{cc}.A_{cc}.L_{cc})/\dot{m}_{gas}$$

Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay

whence the *length of the combustion chamber* may be written down as :

$$L_{cc} = \dot{m}_{gas}.t_{res}/\ (\rho_{cc}.A_{cc}) \propto \dot{m}_{gas}.t_{res}/(P_{cc}^{1/\gamma}.A_{cc})$$

For similar burners sizing of a burner length can be done by

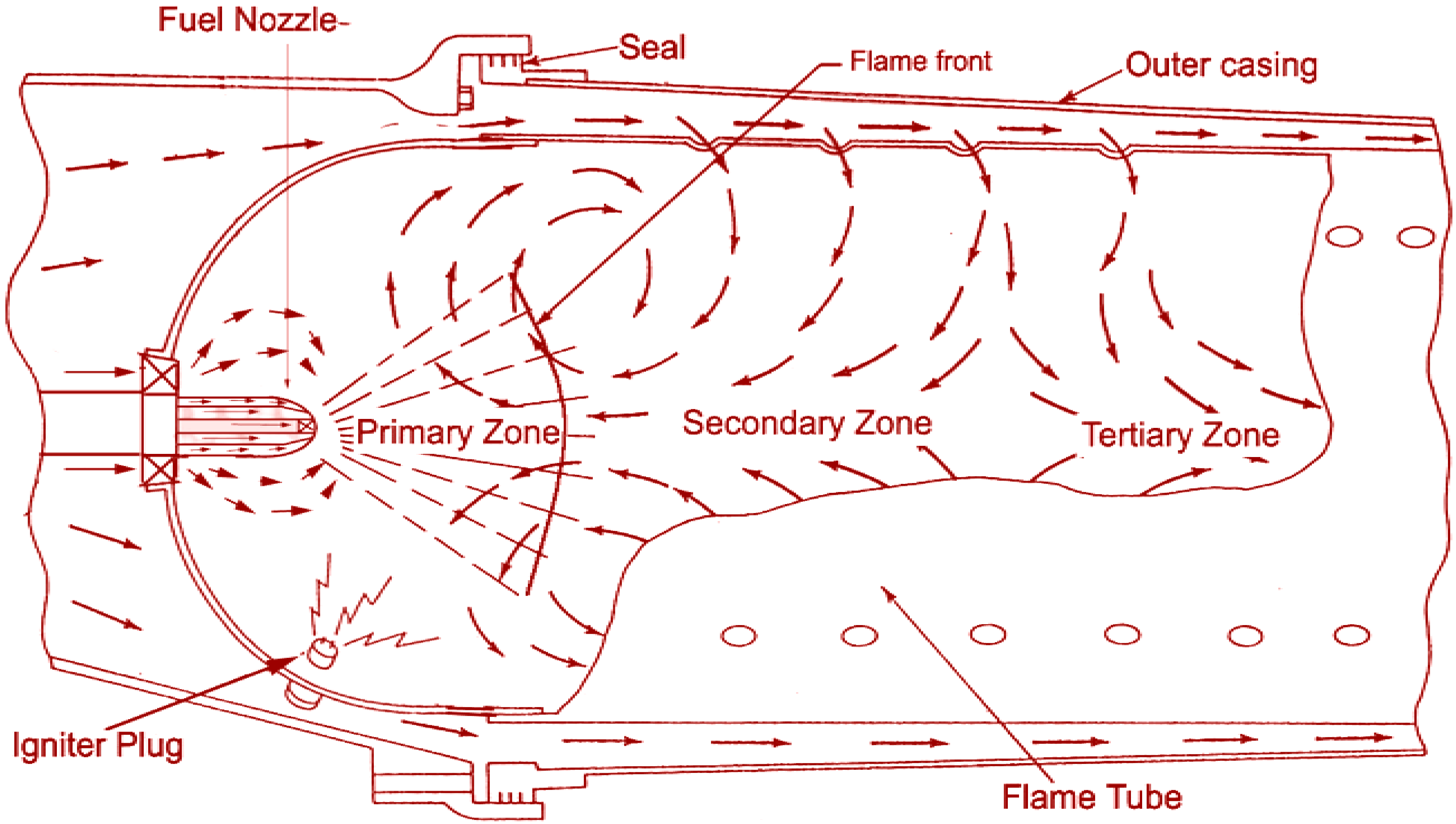
$$L \propto P_{03}^{-r} / \sqrt{T_{0\text{-}cc}} \ ,$$

where r = 1.51 for hydrocarbon fuels (n=1.8) and r =0.714 for n = 1 .

Thus with increasing compression ratio combustion chamber size and length may be reduced. Hence for large engines each can chamber size may not be large if the operating pressures are kept reasonably high.

Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay

Cutout view of a can type combustion chamber



Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay

Development of a practical combustion system

- The primary mixing needs a good circulation zone, created by placing a baffle or a bluff body in the flow.
- A swirl vane is attached to the front baffle so that a swirling flow is set up inside the chamber, this brings down the axial velocity from 150 m/s (as delivered by compressor) to above 40– 50 m/s.
- Now, the baffle, the flame tube and the swirler are combined to give a *primary combustion zone*.

Development of a practical combustion system

- The baffle creates low forward velocity region at its wake where the fuel may be injected and the flame held.
- The spiral flow together with the cross flow set up a strong vortex region that help in *atomizing* the fuel and mixing with air coming in through side entry holes when fuel air ratio may be kept around the stiochiometry value.
- For secondary mixing after combustion, a flame tube with holes on all the sides, which set up a cross flow pattern giving very good mixing inside the tube, is used.

- If the primary zone geometry (shape, size and location) has been fixed it is time for dilution of the hot combustion products.
- The flame tube is continued further with flanged outlet holes for two stage mixing -- before the hot gas is delivered to the turbine.
- An outer casing envelopes the inner can to form an annular duct for secondary and tertiary air.
- The inner can has small holes in an array in between the mixing zones to permit air entry in tangential direction to the casing wall, thus giving a cool air film which protects and extends the life of the casing material.

Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay

- The annular combustion chambers are expected to be used more and more as the gas turbines are to be applied in small aircraft engines.
- It requires less cooling air as the total combustion chamber surface area is less than in the first two cases, it is lighter and hence qualify for applications in military aircraft.
- A reheat chamber is essentially an annular combustion chamber.
- However its temperature and velocity profiles are not as uniform as in the first two cases.

Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay

Next Class

- Combustion Stability and Instability
- Fuel Injection system

Prof. Bhaskar Roy, Prof. A M Pradeep, Department of Aerospace, IIT Bombay