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Gas Turbine Combustion Chamber

(Source: ‘*Gas Turbine Theory*’, Cohen, Rogers)

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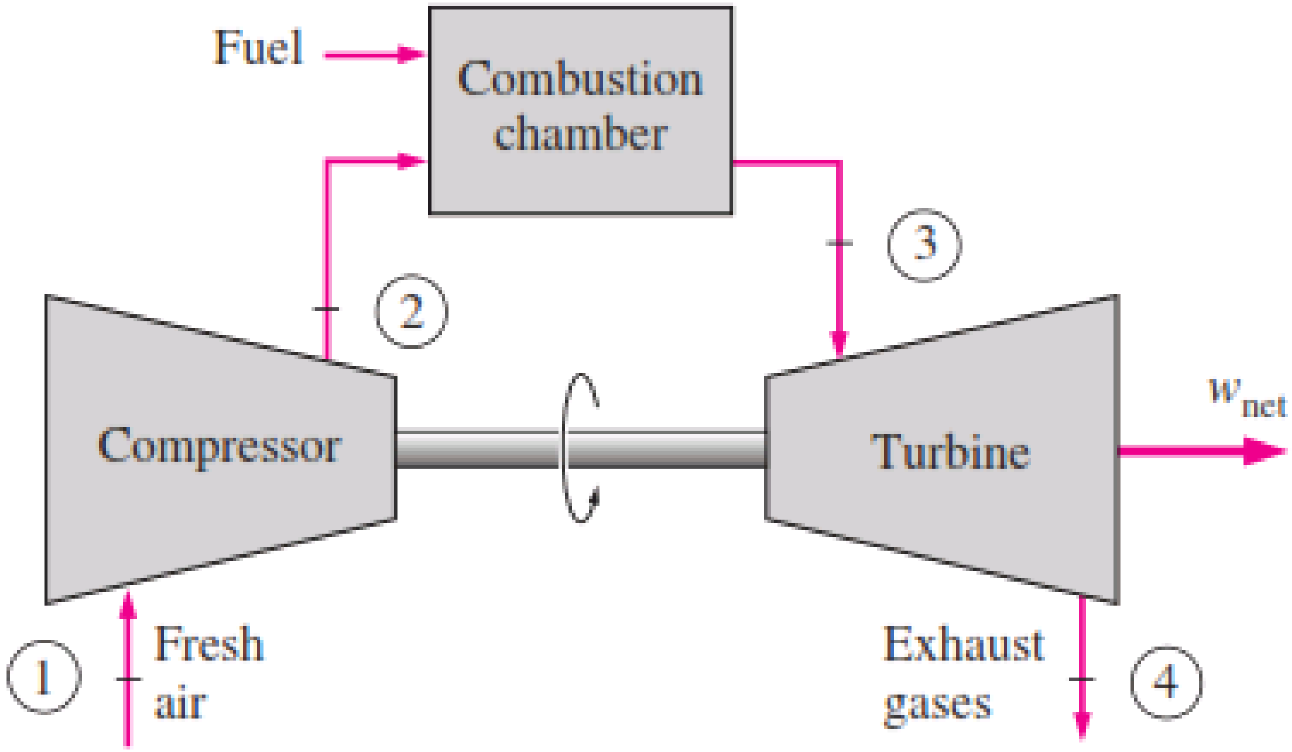


Simple Open Cycle Gas Turbine Schematic Diagram

Contents

Combustion Process

Process

- → Combustion in the normal, open cycle, gas turbine is a continuous process in which fuel is burned in the air supplied by the compressor; an electric spark is required only for initiating the combustion process, and thereafter the flame must be self-sustaining.
- → Combustion of a liquid fuel involves the mixing of a fine spray of droplets with air, vaporization of the droplets, the breaking down of heavy hydrocarbons into lighter fractions, the intimate mixing of molecules of these hydrocarbons with oxygen molecules, and finally the chemical reactions themselves.
- → A high temperature, such as is provided by the combustion of an approximately **stoichiometric mixture**, is necessary if all these processes are to occur sufficiently rapidly for combustion in a moving air stream to be completed in a small space
- → But in actual practice A/F ratio is in the range of 100:1, while stoichiometric ratio is around 15:1. This is to reduce the turbine inlet temperatures due to practical limits.

Factors influencing design

→ Low **turbine inlet temperature**

→ Uniform **temperature distribution** at turbine inlet (i.e., to avoid local over heating of turbine blades)

→ **Stable operation** even when factors like air velocity, A/F ratio & chamber pressure varies greatly, especially for aircraft engines (the limit is the '*flame-out*' of the combustion chamber) & at the event of a *flame-out* the combustor should be able to relight quickly.

→ The formation of carbon deposits ('**coking**') must be avoided. Can damage the turbine if breaks free.

→ Aircraft engines should avoid **visible smoke** as it hinders visibility in airports

→ Finally, **Pollutants** like NO_x, CO, Unburned Hydrocarbons(UHCs) etc. should be limited

Zonal method of introducing air

- Primary zone (15-20% air)
  - Air is introduced around the jet of fuel
  - burns at approximately the **Stoichiometric Ratio**
  - Therefore, **High temperature**
  - And thus, **Rapid Combustion**
- Secondary Zone (30% air)
  - Introduced through holes in the flame-tube in the secondary zone **to complete the combustion**
  - For high combustion efficiency, air must be injected carefully at the right points in the process, to **avoid chilling** the flame locally and drastically reducing the reaction rate in that neighbourhood
- Tertiary Zone (remaining air)
  - **Dilution Zone**
  - **Cooling**
  - Sufficient turbulence must be promoted so that the hot and cold streams are thoroughly mixed to give the desired outlet temperature distribution, with no hot streaks which would damage the turbine blades.

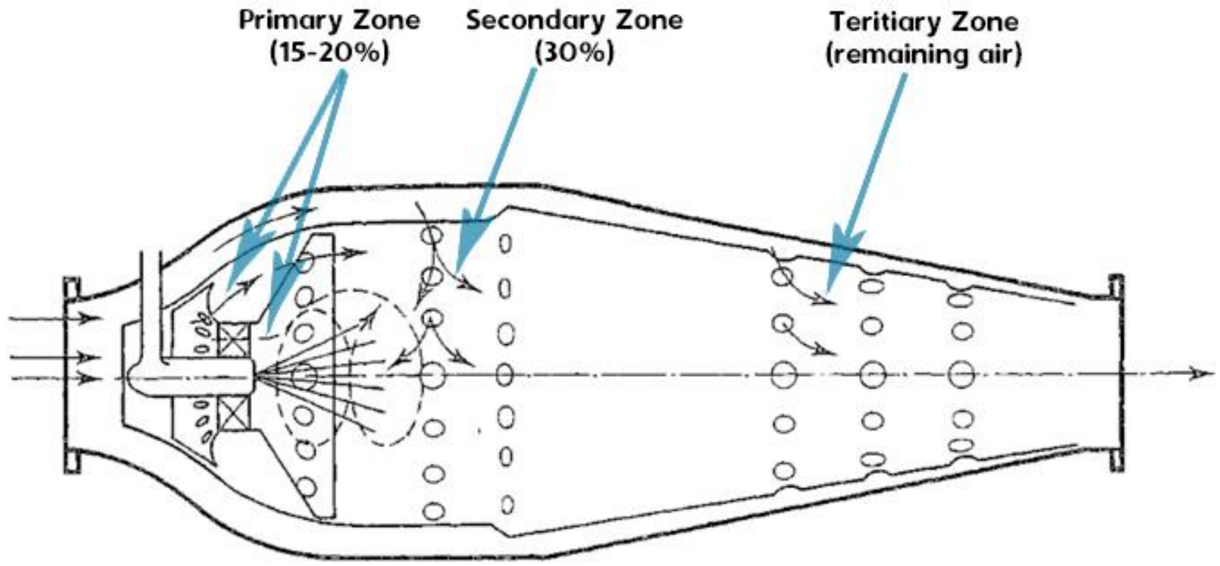
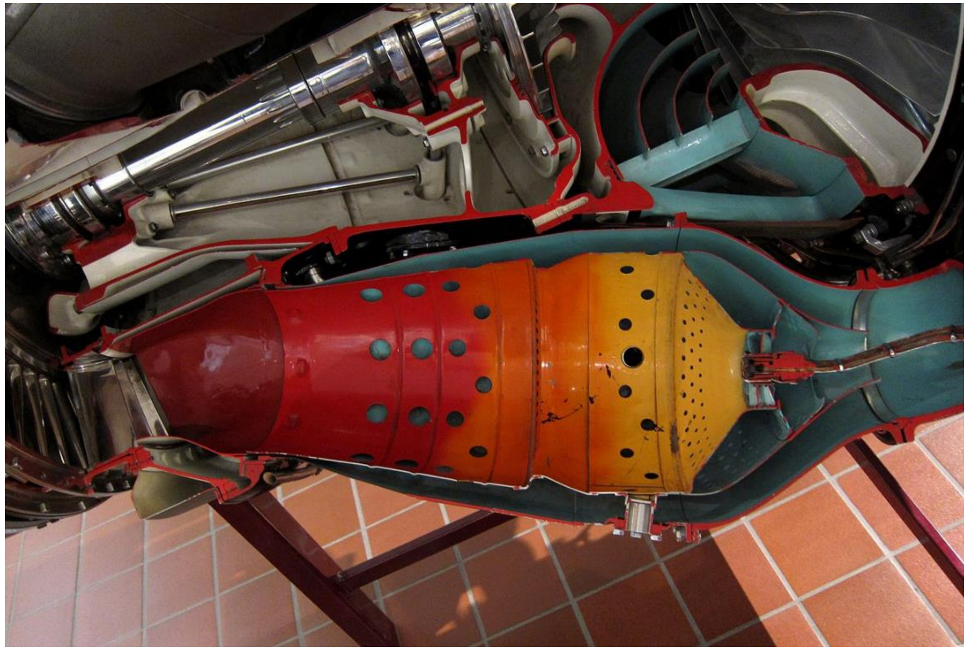


FIG. 6.2 Combustion chamber with swirl vanes



Flame stability

- The zonal method of introducing the air cannot by itself give a self-piloting flame in an air stream which is moving an order of magnitude faster than the flame speed in a burning mixture. The second essential feature is therefore a **recirculating flow pattern** which directs some of the burning mixture in the primary zone back on to the incoming fuel and air.

Recirculating flow pattern is achieved by,

1. Swirl vanes [fig 6.2]
2. Holes downstream of a hemispherical Baffle [fig 6.3 (a)]
3. Upstream injection [fig 6.3 (b)]
4. Vaporiser system (walking stick/T-shaped tubes) [fig 6.3 (c)]

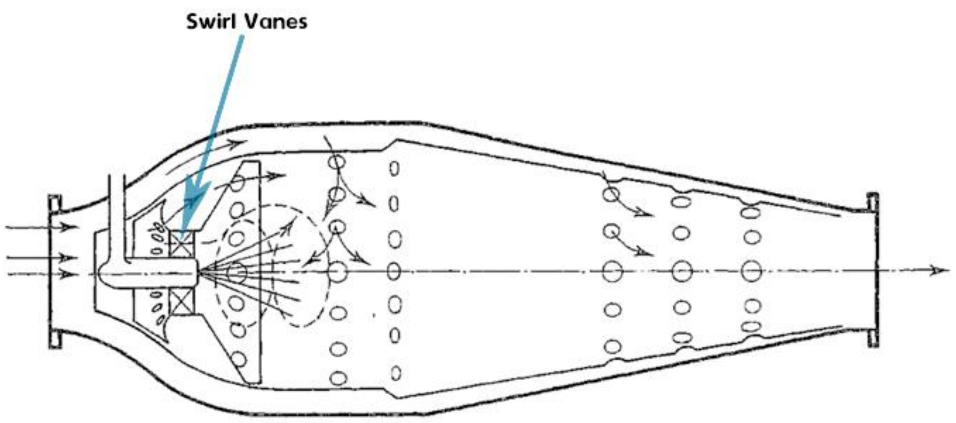


FIG. 6.2 Combustion chamber with swirl vanes

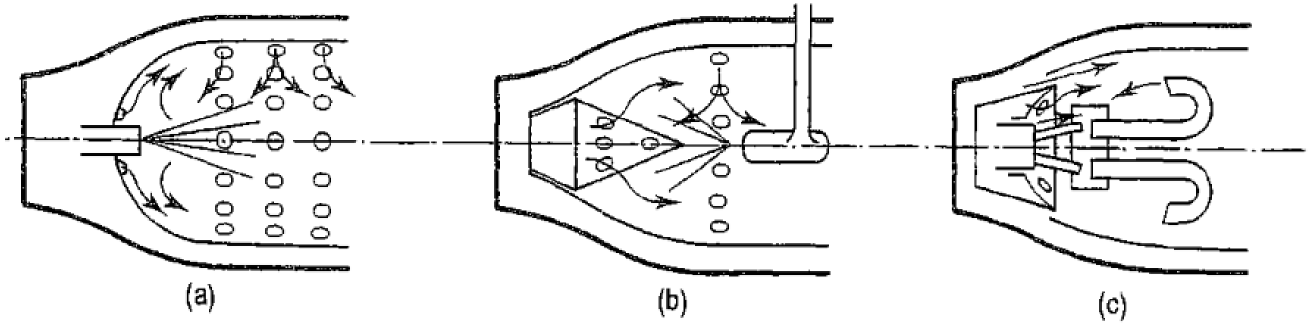


FIG. 6.3 Methods of flame stabilization

Types of Combustion Chambers

1. **Can Type (Tubular)**
2. **Cannular Type (Tubo-annular)**
3. **Annular Type**
4. **Silo Type**

1. Tubular (Can)

The earliest aircraft engines made use of can (or tubular) combustors.

Air leaving the compressor is split into a number of separate streams, each supplying a separate chamber.

These chambers are spaced around the shaft connecting the compressor and turbine, each chamber having its own fuel jet fed from a common supply line.

Well suited to engines with centrifugal compressors, where the flow is divided into separate streams in the diffuser.

- **Easier development** (could be carried out on a single can using only a fraction of the overall airflow and fuel flow)

But

- **Increased Volume, weight & frontal area**
- **Increased Pressure drop** (more surface area in contact with air/gas)

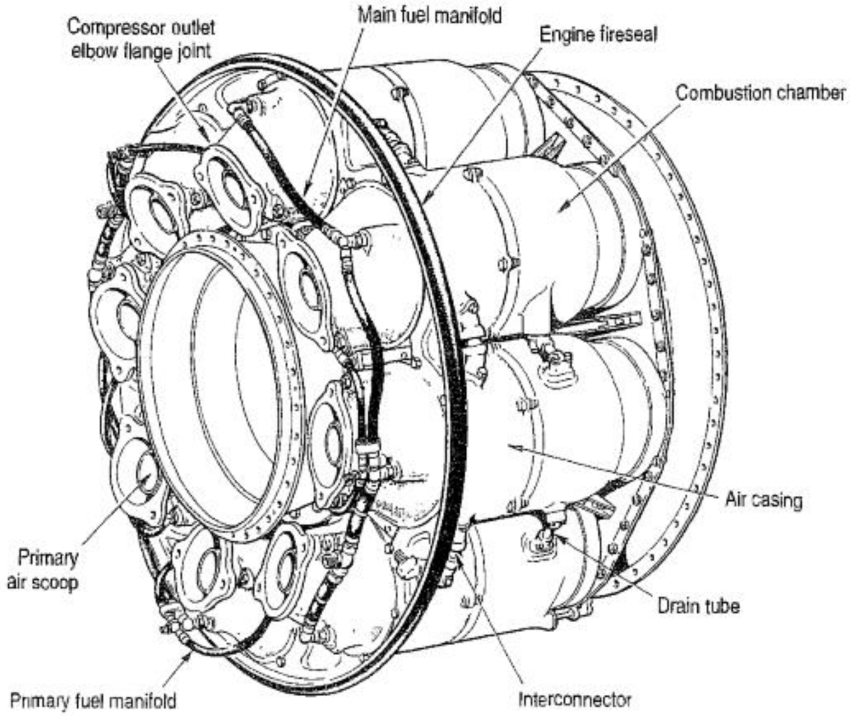


FIG. 6.1 Can type combustor [conrtesy Rolls-Royce]

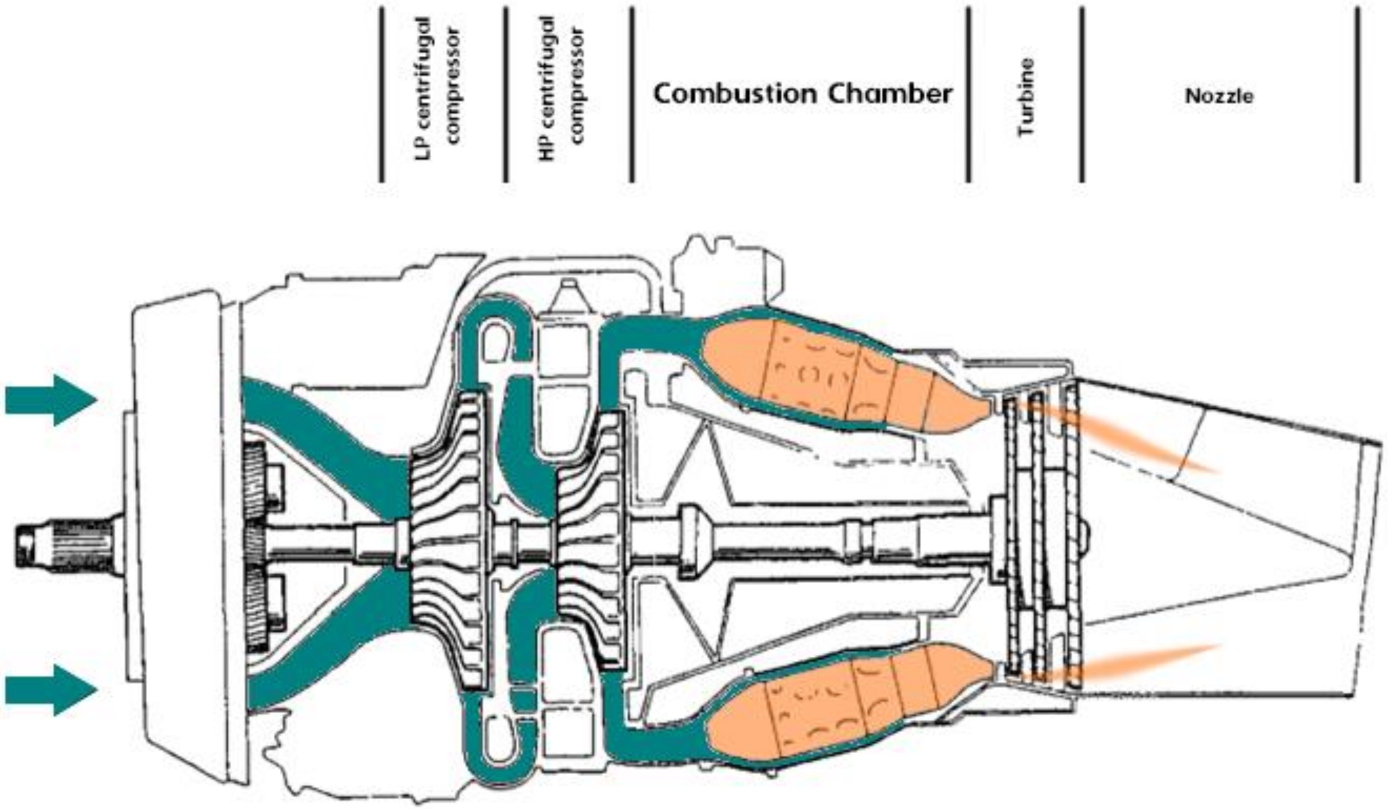
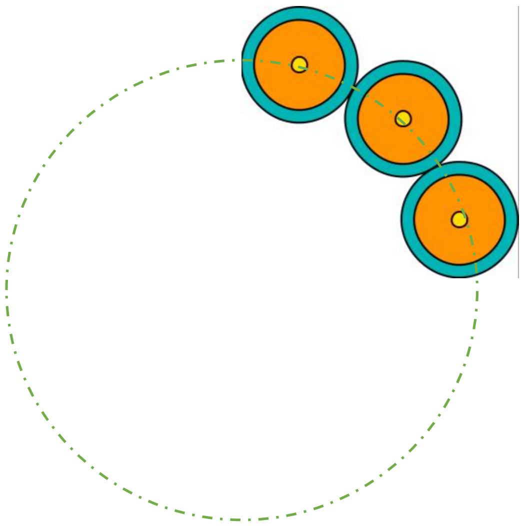


FIG. 1.10 Single-shaft turboprop engine [by conrtesy of Rolls-Royce]



Arrangement of the combustors, looking axis on, through the exhaust. The blue indicates flow path of secondary & tertiary air (outside the flame tube), the orange indicates the combustion product flow path (inside the flame tube) and the yellow indicates fuel injector

2. Tubo-annular (Cannular)

Individual flame tubes are uniformly spaced around an annular casing. Uses a reverse flow arrangement which allows a significant reduction in the overall length of the compressor-turbine shaft and also permits easy access to the fuel nozzles and combustion cans for maintenance.

I.e.,

- **Reduced shaft length**
- **Easy maintenance**

Also like Can Combustor,

- **Easier development,**

But

- **Increased Volume, weight & frontal area**
- **Increased Pressure drop**

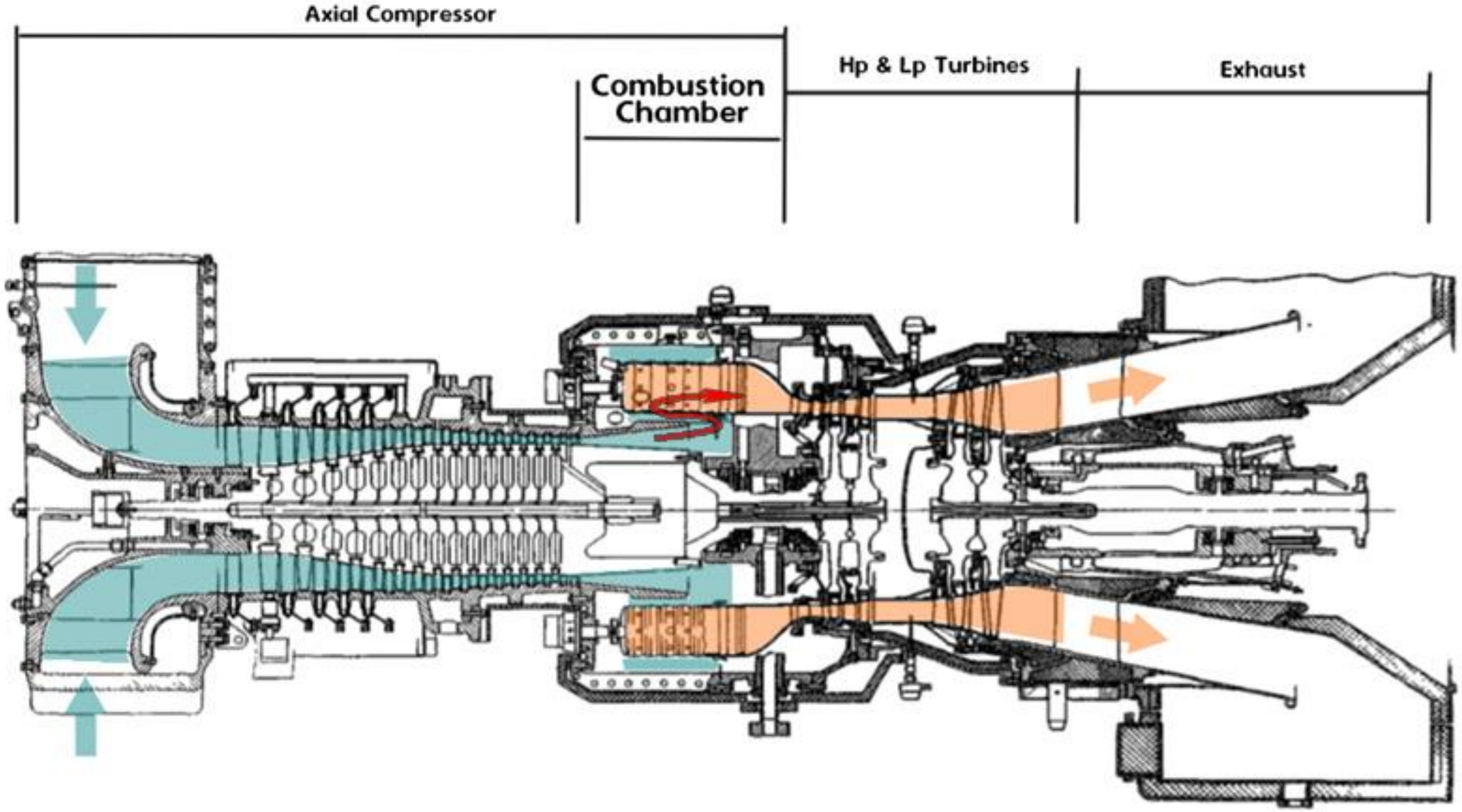
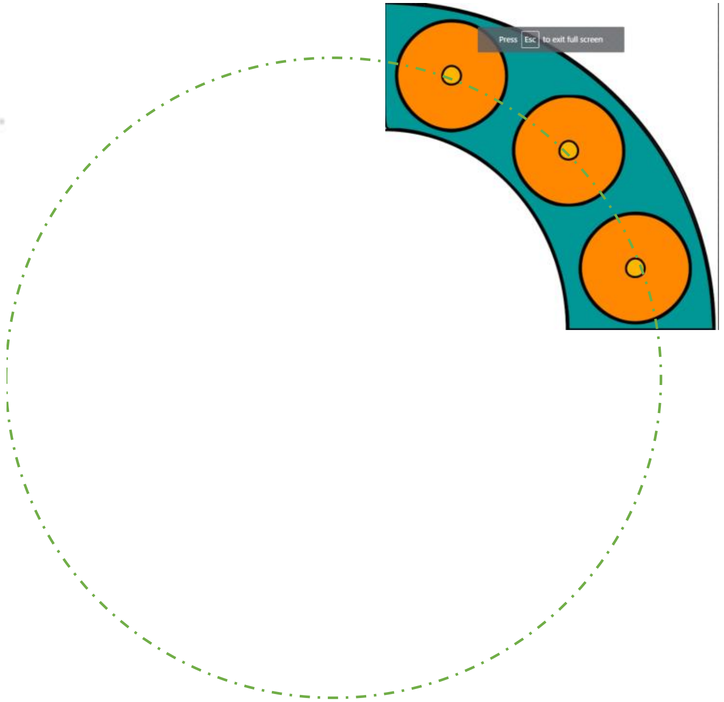


FIG. 1.13 Industrial gas turbine with separate power turbine [courtesy European Gas Turbines]



3. Annular

The ideal configuration in terms of compact dimensions is the annular combustor, in which maximum use is made of the space available within a specified diameter; this should reduce the pressure loss and results in an engine of minimum diameter. The combustion does not take place in individual flame tubes, but instead in an annular region around the engine.

Overcomes disadvantages of Can type,

- **reduces the pressure loss** (less surface exposed to air/gas flow)
- **Compact size**

But,

- **Less structural integrity**
- **Difficult to obtain even temperature distribution**
- **Difficult development (larger test facilities required)**

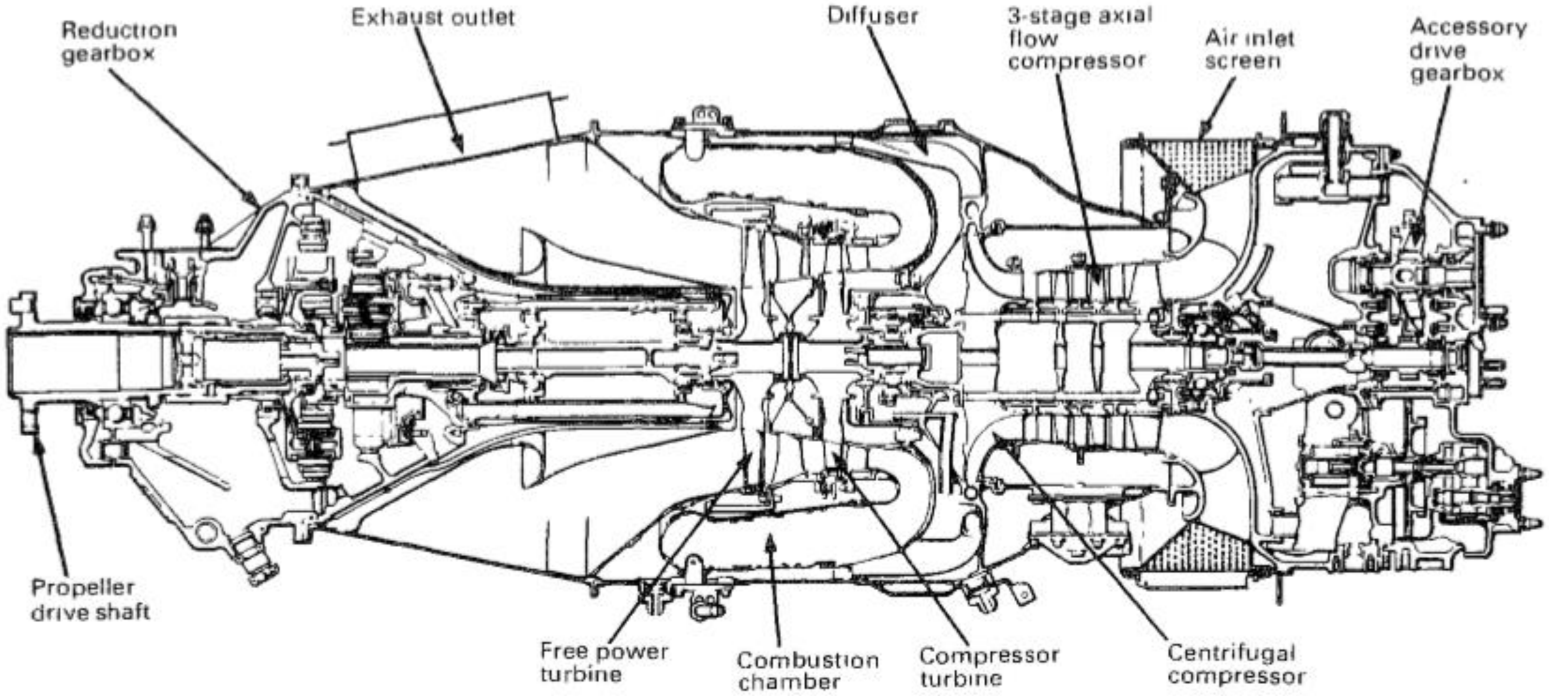
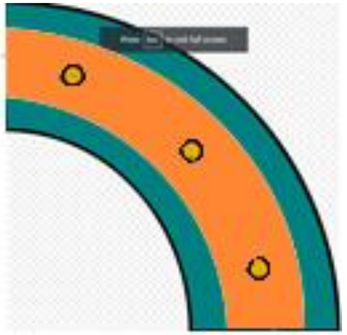
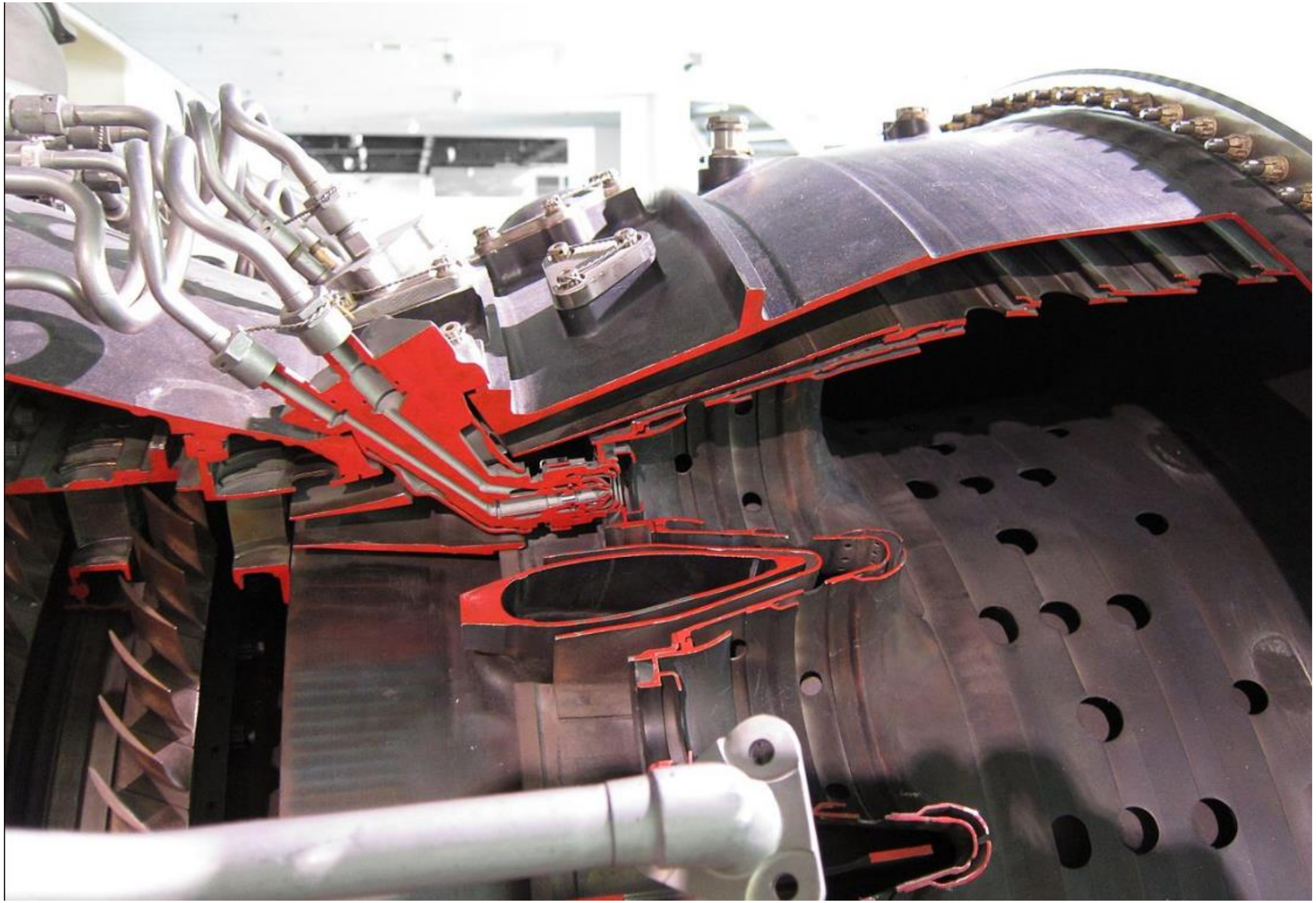


FIG. 1.11 Turboprop engine [by courtesy of Pratt and Whitney Canada]





Annular combustor on a Pratt & Whitney JT9D turbofan
(https://en.wikipedia.org/wiki/Combustor)

4. Silo type

Large industrial gas turbines, where the space required by the combustion system is less critical, have used one or two large cylindrical combustion chambers.

These large combustors allowed lower fluid velocities and hence pressure losses, and were capable of burning lower quality fuels.

I.e.,

- **Low pressure loss**
- **Can burn low quality fuel**

But,

- **Bulky**

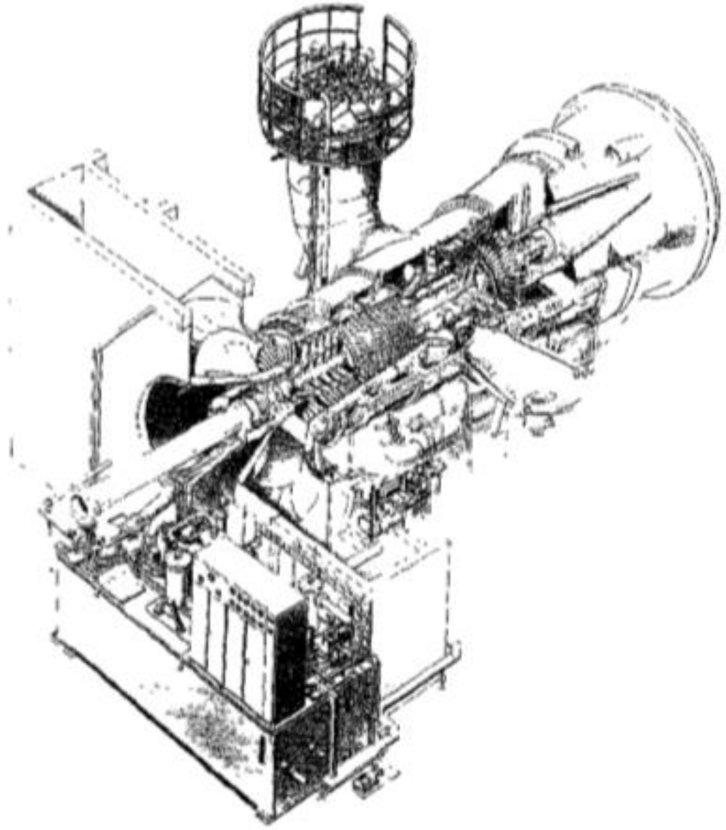


FIG. 1.14 Large single-shaft gas turbine [courtesy Siemens]

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Performance Parameters

Pressure Loss

1. Fundamental Loss

Pressure loss due to,

- **Rise in temperature during combustion.**

An increase in temperature implies a decrease in density and consequently an increase in velocity and momentum of the stream. A pressure drop ($\Delta p \times A$) must be present to impart the increase in momentum.

2. Cold Loss

Pressure Loss due to,

- **Skin friction**
- **Turbulence**

The pressure loss due to friction is found to be very much higher than that due to combustion, mainly due to turbulence, which is required for proper mixing and Temperature uniformity.

The overall pressure loss can often be expressed by an equation of the form,

$$\text{pressure loss factor, } PLF = K_1 + K_2\left(\frac{T_{02}}{T_{01}} - 1\right)$$

The pressure loss found out from a combustion chamber during a cold run (without combustion) is due to skin friction and turbulence alone (Cold loss). [*Without combustion there is no temperature ratio and hence from the above equation it is evident that K_1 represents the cold loss.*]

A hot run (with combustion) on the other hand would reveal the total loss. Hence, K_1 and K_2 are determined from a combustion chamber on a test rig from a cold run and a hot run

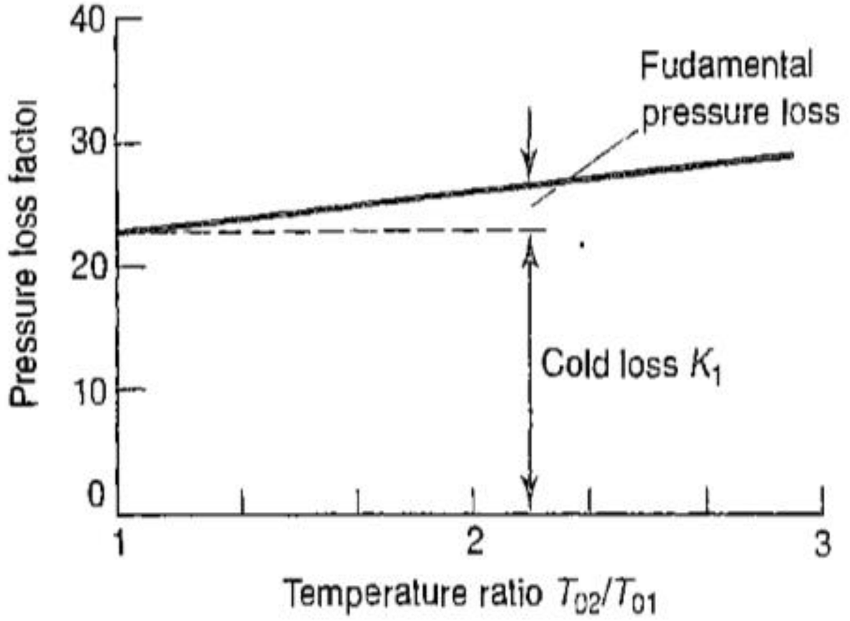

FIG. 6.5 Variation of pressure loss factor

Combustion Efficiency

Overall Combustion Efficiency is defined as,

$$\eta_b = \frac{\text{theoretical } f \text{ for actual } \Delta T}{\text{actual } f \text{ for actual } \Delta T}$$

Where *f* (Fuel/Air ratio) & Inlet and outlet mean stagnation temperatures are measured experimentally.

Stability Loop

For any particular combustion chamber there is both a rich and a weak limit to the air/fuel ratio beyond which the flame is unstable.

Usually the limit is taken as the air/fuel ratio at which the flame blows out, although instability often occurs before this limit is reached.

The range of air/fuel ratio between the rich and weak limits is reduced with increase of air velocity, and if the air mass flow is increased beyond a certain value it is impossible to initiate combustion at all.

A typical stability loop where the limiting air/fuel ratio is plotted against air mass flow is shown.

The stability loop is a function of the pressure in the chamber. A decrease in pressure narrows the stability limits.

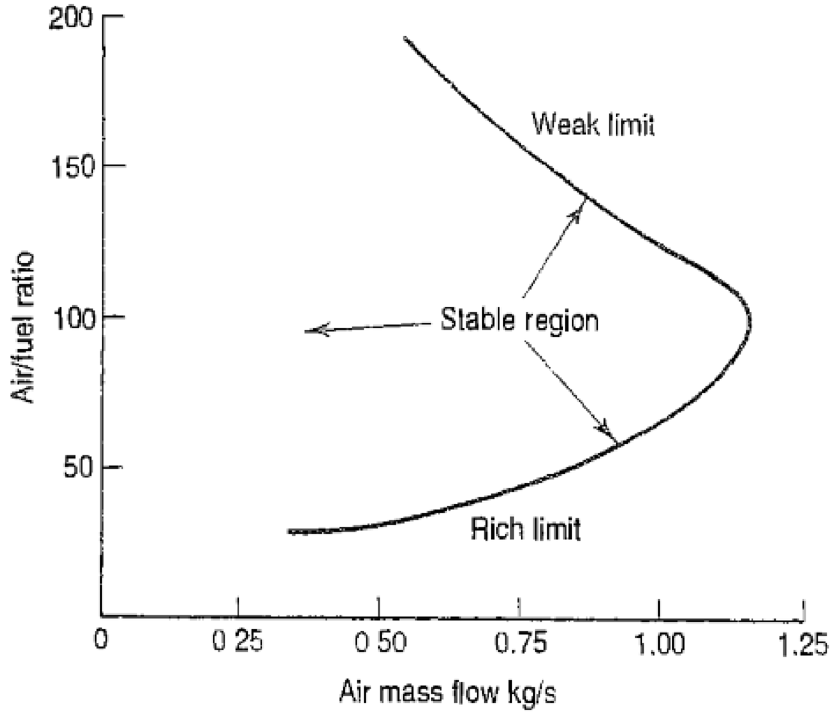

FIG. 6.7 Stability loop

Combustion Intensity

$$\text{combustion intensity} = \frac{\text{heat release rate}}{\text{comb. vol.} \times \text{pressure}} \ \text{kW/m}^3 \ \text{atm}$$

- Lower the combustion intensity easier to design a system with desired requirements
- Cannot compare to systems based on efficiency & pressure loss if they vary widely in the combustion intensity
  - Aircraft systems – 2-5x10^4 kW/m^3 atm
  - Industrial systems –much less due to large combustion systems