UNIT-2
INLETS, COMBUSTORS, AND NOZZLES

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UNIT-1 SYLLABUS

- Introduction
- Subsonic inlets
- Supersonic inlets
- Gas turbine combustors
- Afterburners and ramjet combustors
- Supersonic combustion
- Exhaust nozzle
- Numerical problems.

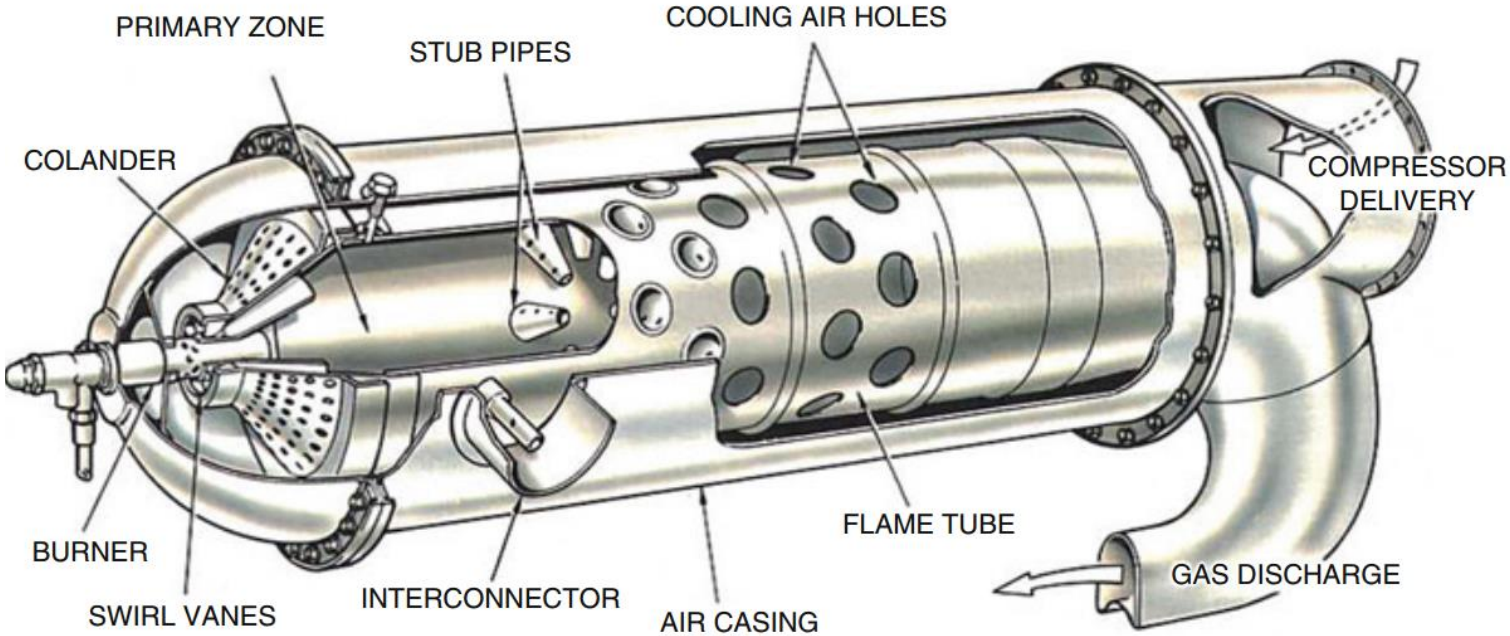
INTAKE REQUIREMENTS

The performance of an inlet must fulfill the following requirements:

- Delivers exact amount of air required for different flight phases and wide spectrum of angles of attack and sideslip.
- Diffuses air with maximum static pressure rise and minimum total pressure loss
- Recovers flow distortion or separation at large angles of attack to achieve as uniform flow as possible
- Achieves least possible external drag to the system
- Provides good starting and stability
- Achieves low signatures (acoustic, radar, etc.) for a noise suppression and stealth requirements
- Holds minimum weight and cost while meeting life and reliability goals

GAS TURBINE COMBUSTORS

The purpose of the combustion system is to increase the thermal energy of a flowing air/gas stream by combustion, which is an exothermic chemical reaction between the injected fuel and the oxygen in the flowing stream. Combustion process takes place in the main burner (or the combustor) and the afterburner (or the augmenter/ reheater) for supersonic aircraft.

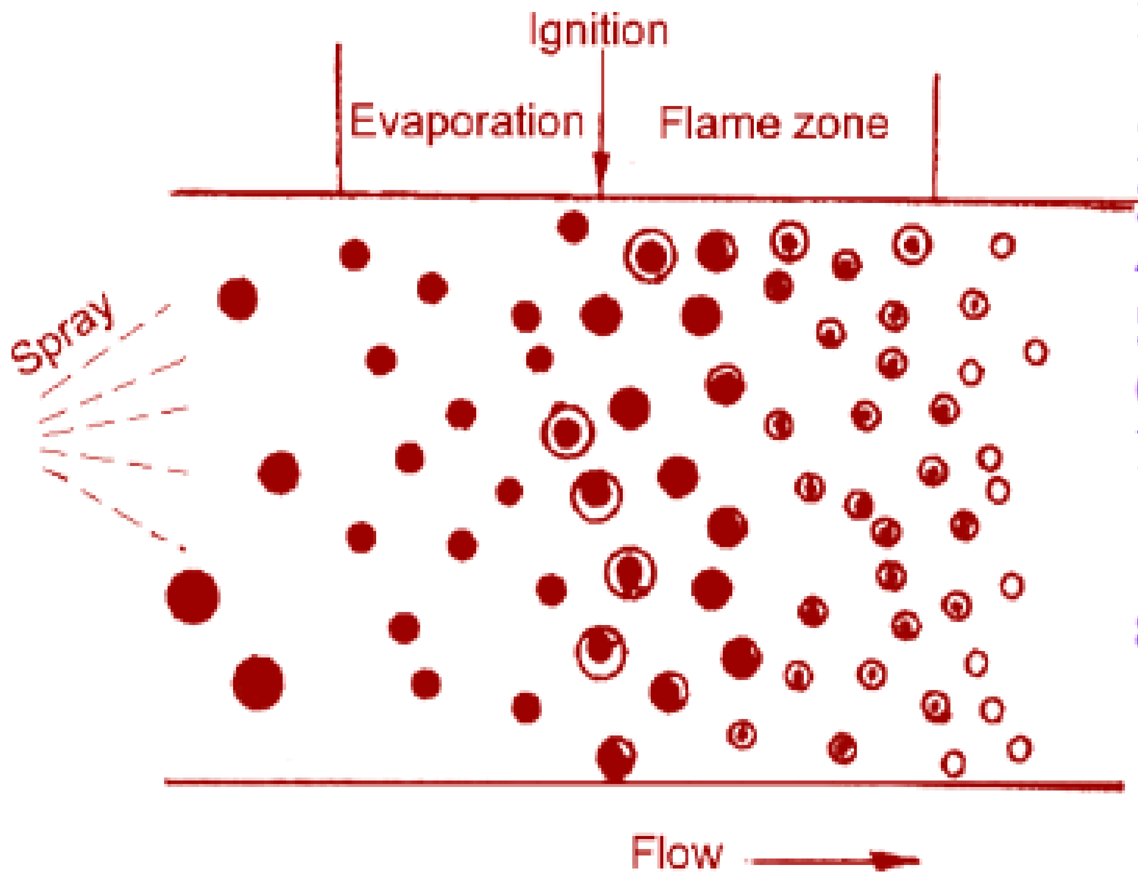


Function of The Combustion Chamber

The function of the combustion chamber is given as follows

1. Complete combustion
2. Moderate total pressure loss
3. Stability of combustion process (freedom from flameout)
4. In-flight relight ability
5. Proper temperature distribution at exit with no “hot spots”
6. Short length and small cross section
7. Wide operating range of mass flow rates, pressures, and temperatures
8. Satisfaction of established environmental limits for air pollutants

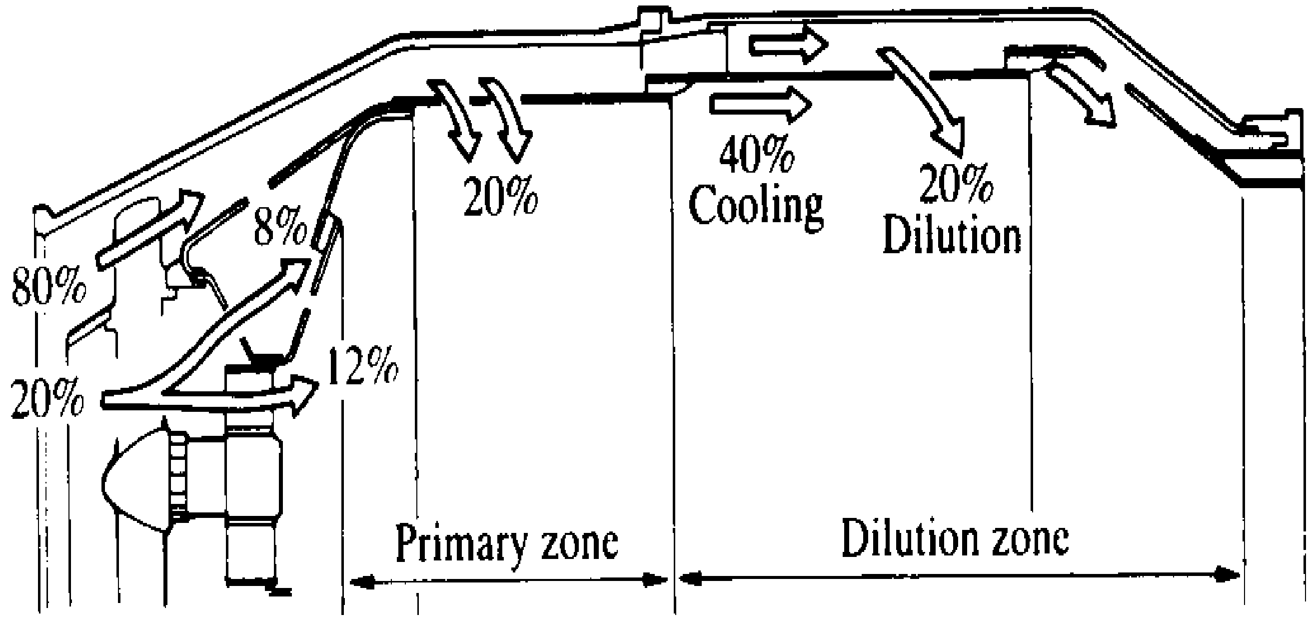
Combustion model for a gas turbine combustor



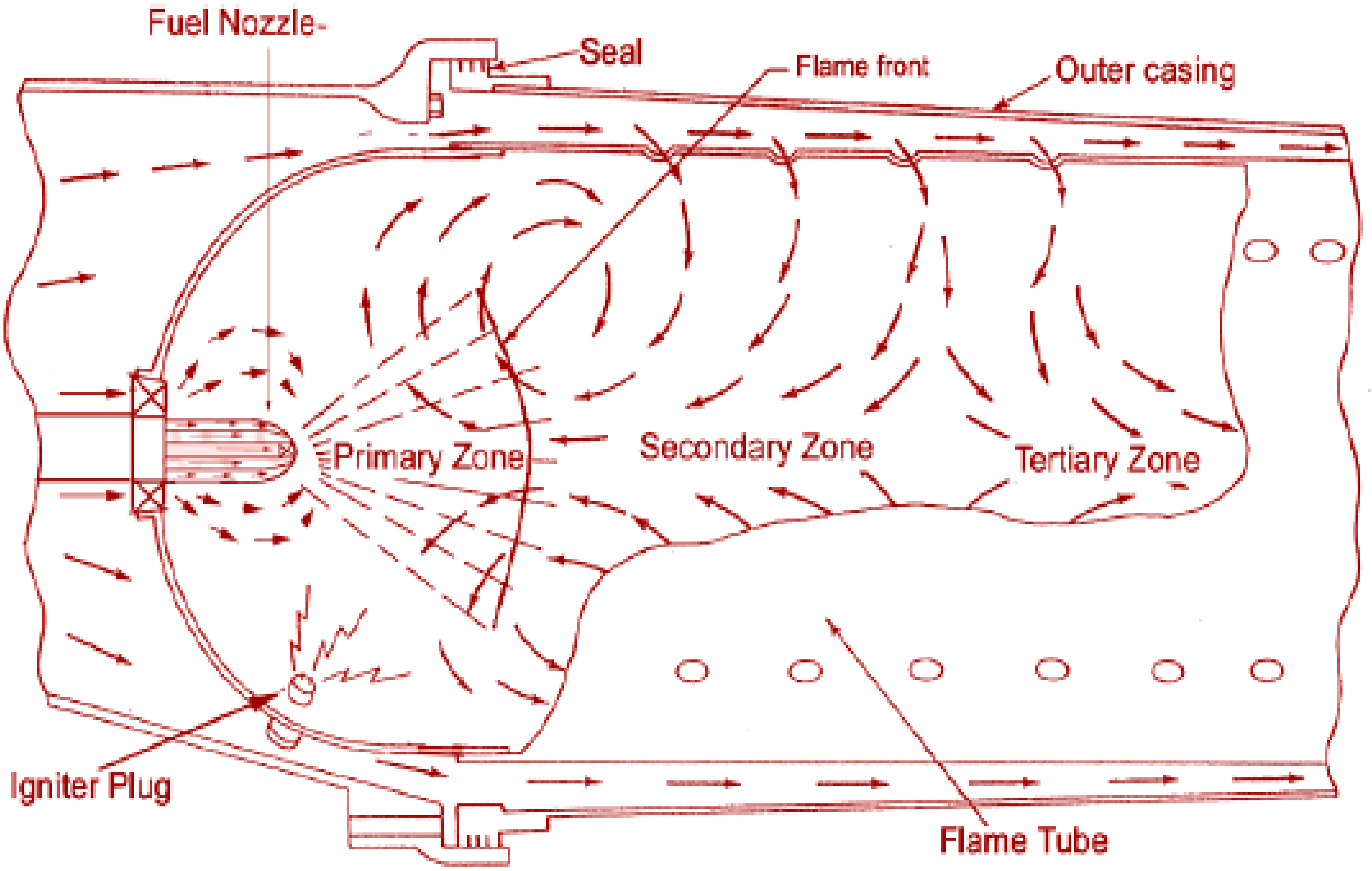
Steps :

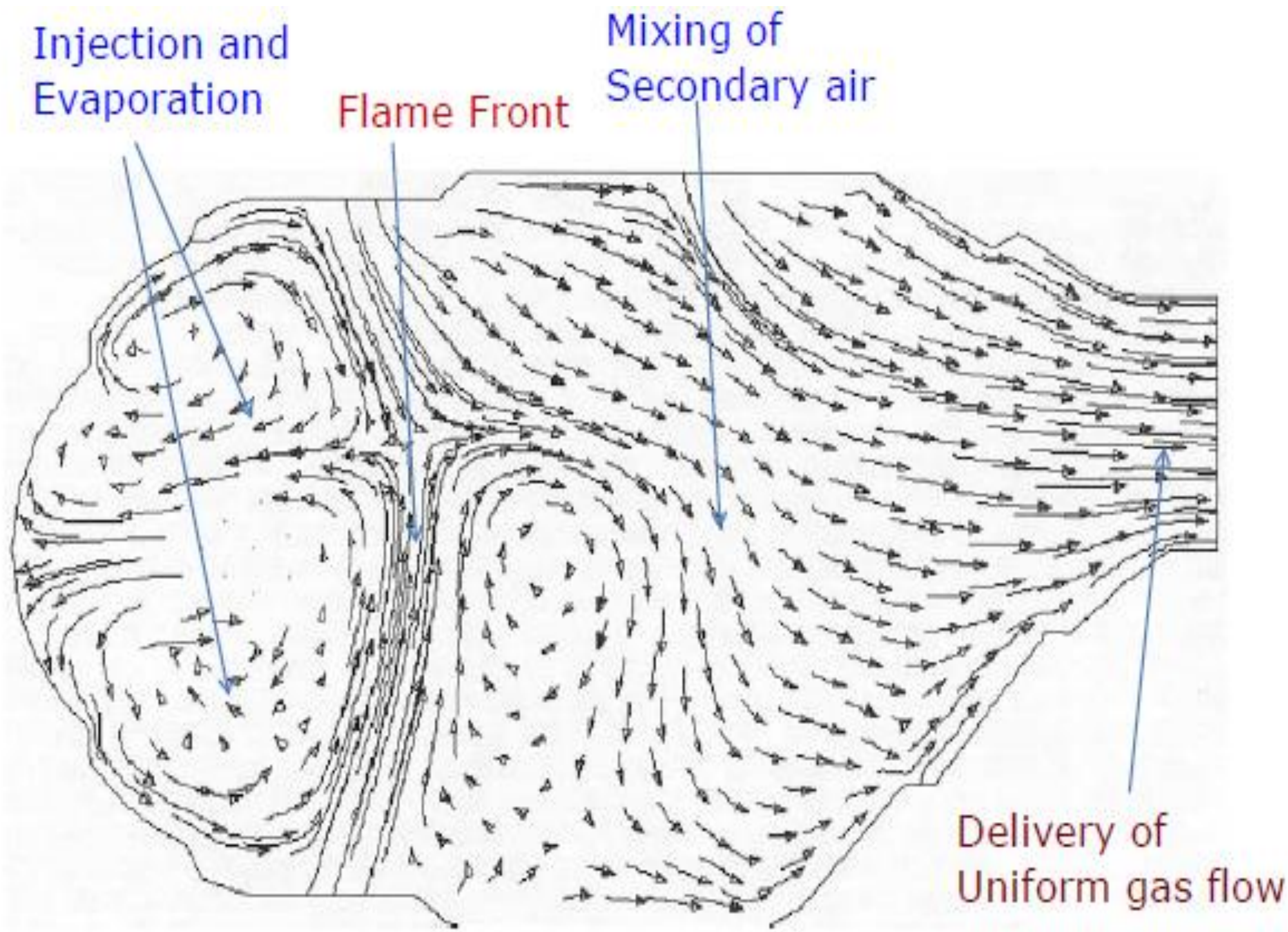
1) Spray through an Injector
2) Atomization
3) Evaporation
4) Mixing with air
5) Ignition
6) Heat release
7) Mixing of combustion products
8) Delivery to Turbine

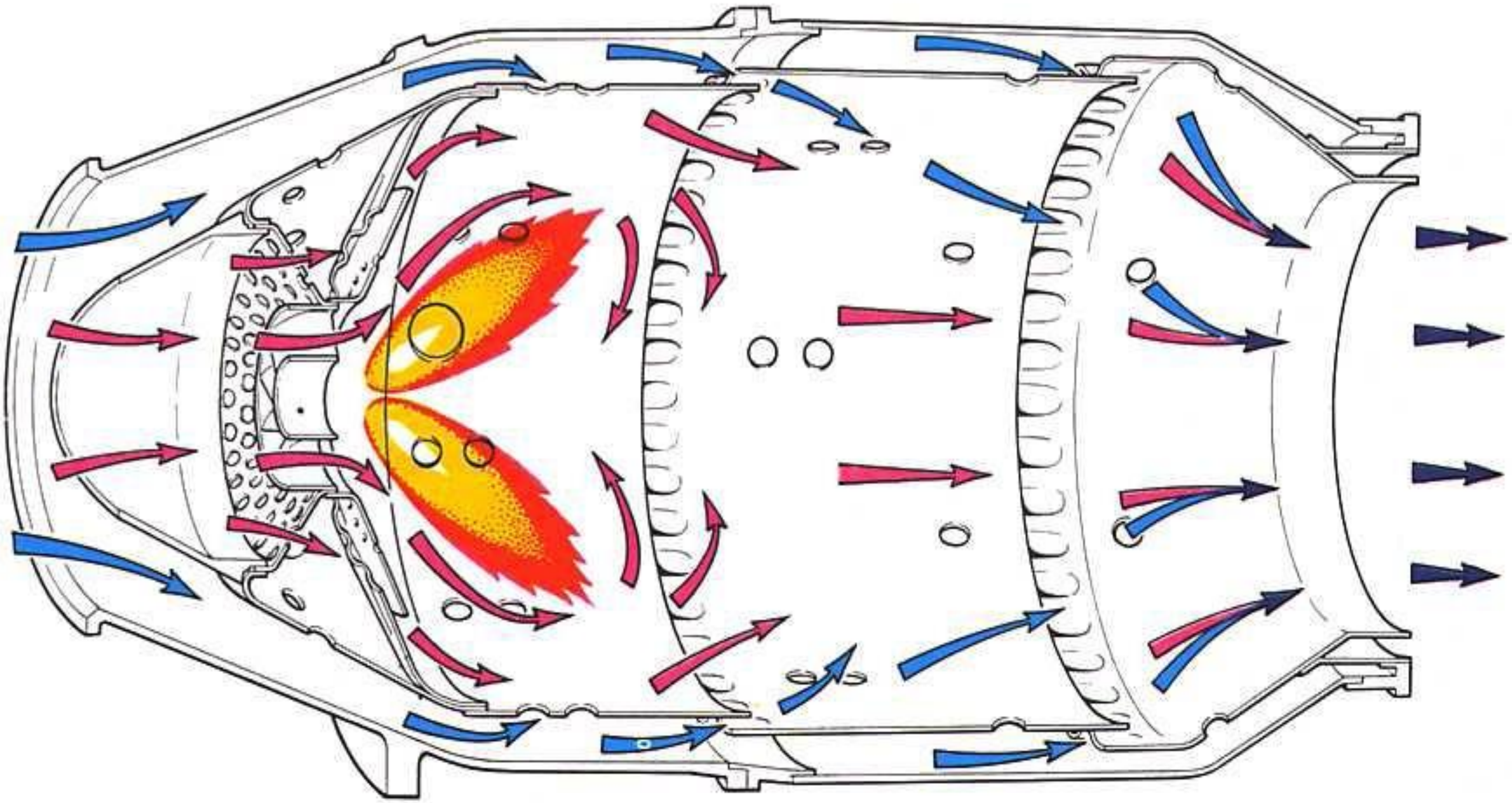
Airflow distribution in combustor



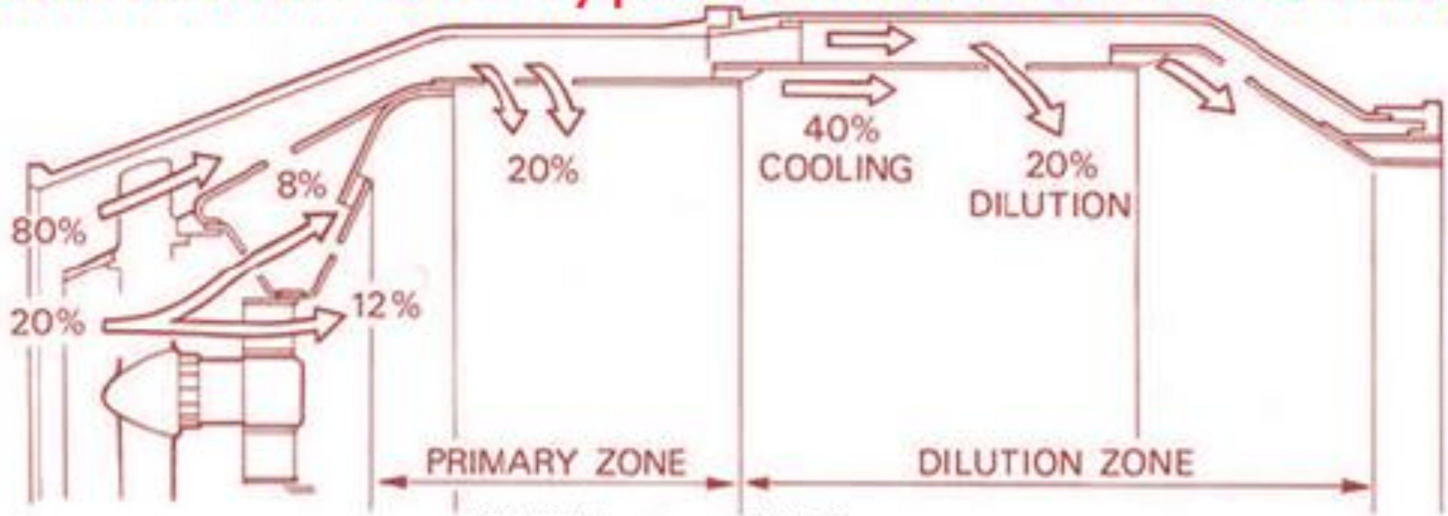
COMBUSTION SYSTEM

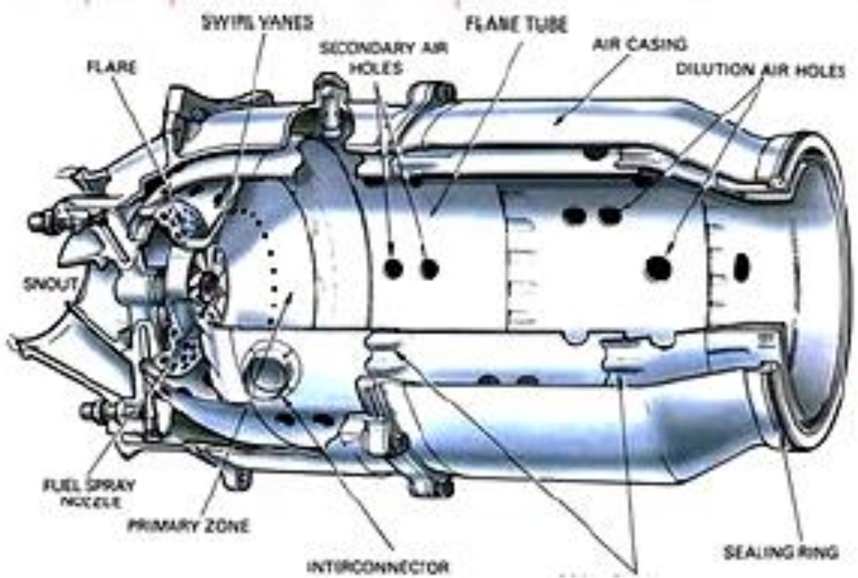



Injection and
Evaporation
Flame Front
Mixing of
Secondary air
Delivery of
Uniform gas flow



Individual Can Type combustion chamber





PRIMARY,SECONDARY AND DILUTION AIR

PRIMARY AIR

- This is the main combustion air.
- It is highly compressed air from the high pressure compressor (often decelerated via the diffuser) that is fed through the main channels in the dome of the combustor and the first set of liner holes.
- This air is mixed with fuel, and then combusted.

INTERMEDIATE AIR

- Intermediate air is the air injected into the combustion zone through the second set of liner holes (primary air goes through the first set).
- This air completes the reaction processes, cooling the air down and diluting the high concentrations of carbon monoxide (CO) and hydrogen (H_2).

DILUTION AIR

- Dilution air is airflow injected through holes in the liner at the end of the combustion chamber to help cool the air to before it reaches the turbine stages.

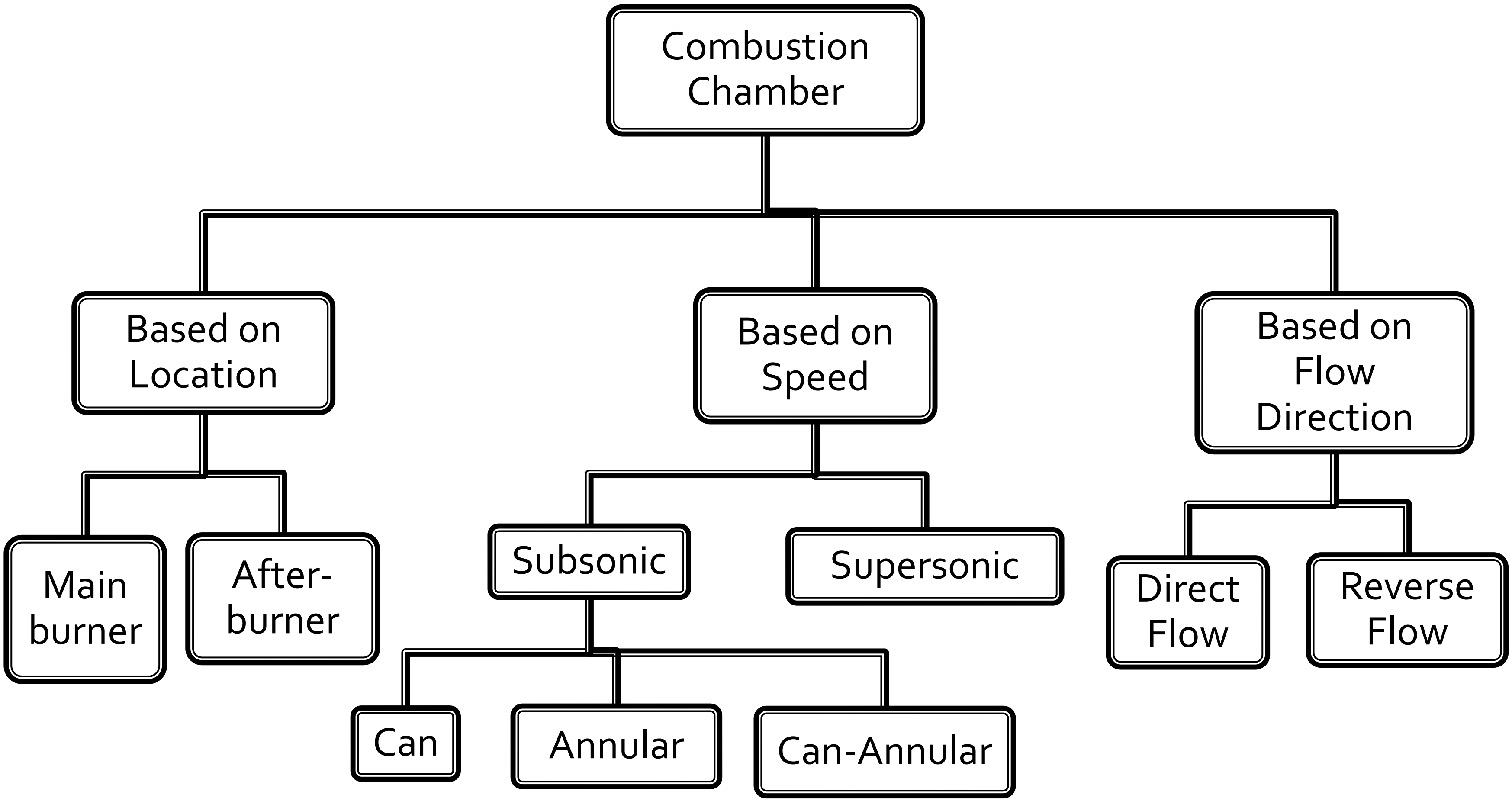
- The air is carefully used to produce the uniform temperature profile desired in the Combustor.

COOLING AIR

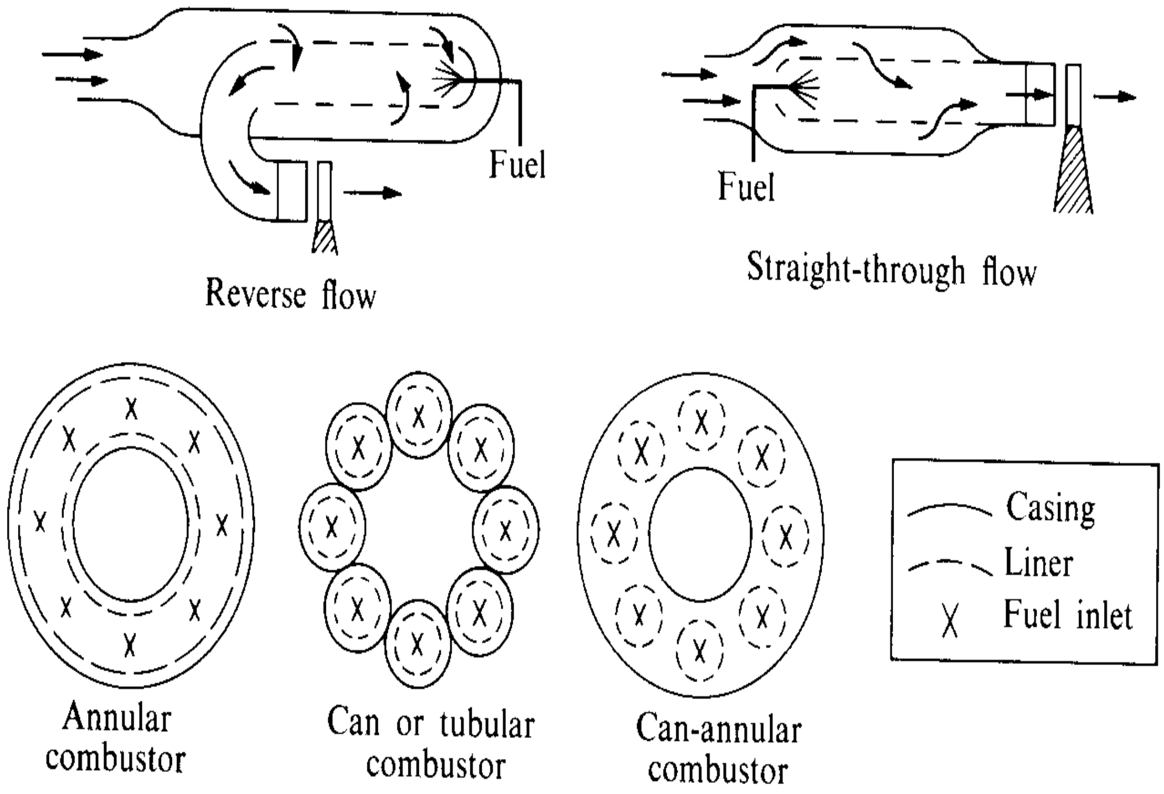
Cooling air is airflow that is injected through small holes in the liner to generate a layer (film) of cool air to protect the liner from the combustion temperatures.

. In some cases, as much as 50% of the inlet air is used as cooling air.

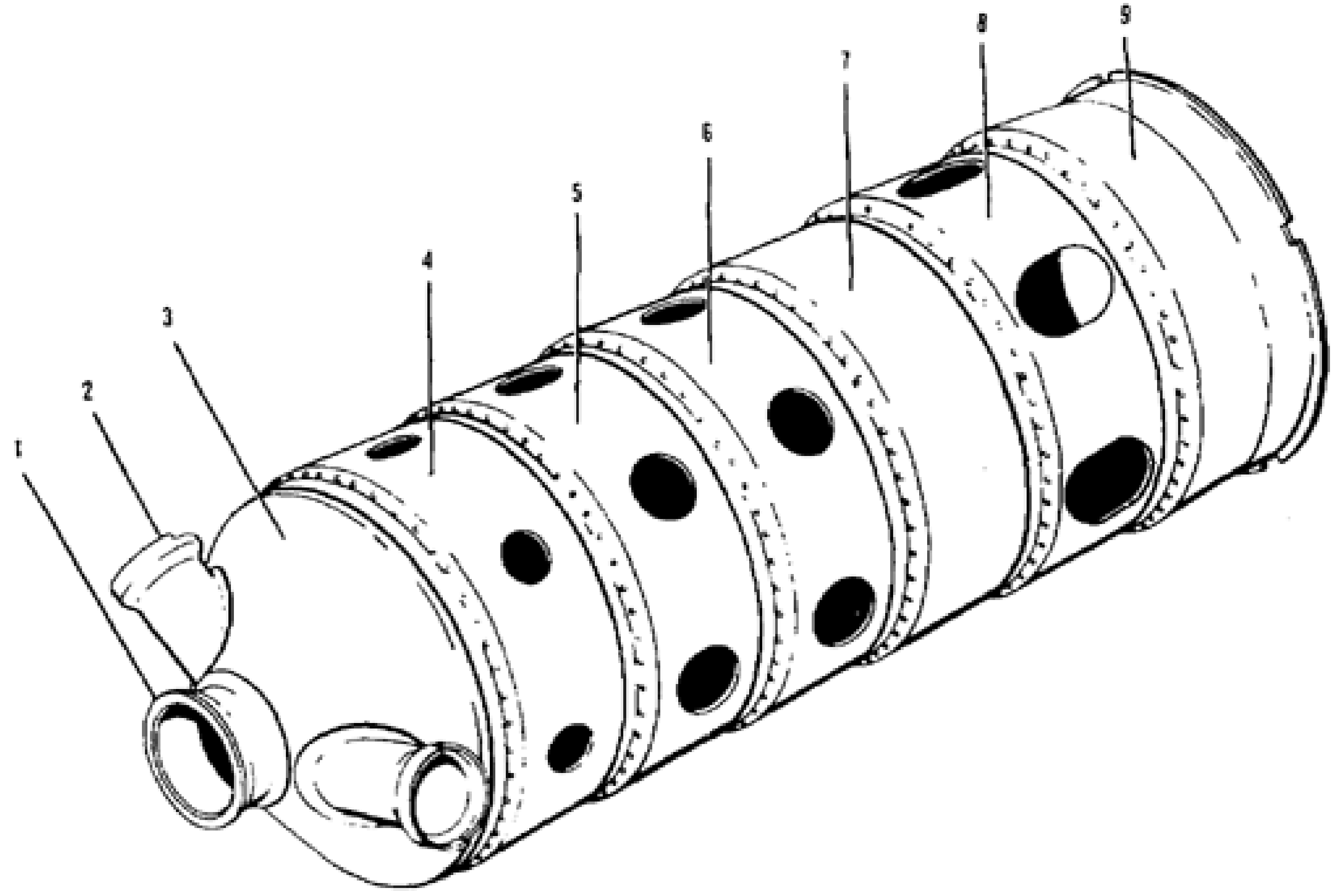
Types of Combustion chamber



TYPES OF COMBUSTOR



CAN TYPE COMBUSTOR



1. Fuel nozzle cup adapter
2. Flame tube
3. Fuel nozzle cup
4. No. 1 liner
5. No. 2 liner
6. No. 3 liner
7. No. 4 liner
8. No. 5 liner
9. No. 6 liner

- Can combustors are self-contained cylindrical combustion chambers.
- Each "can" has its own fuel injector, igniter, liner, and casing.
- The primary air from the compressor is guided into each individual can, where it is decelerated, mixed with fuel, and then ignited.
- The secondary air also comes from the compressor, where it is fed outside of the liner (inside of which is where the combustion is taking place).
- The secondary air is then fed, usually through slits in the liner, into the combustion zone to cool the liner via thin film

Advantages and Disadvantages of Can type combustor

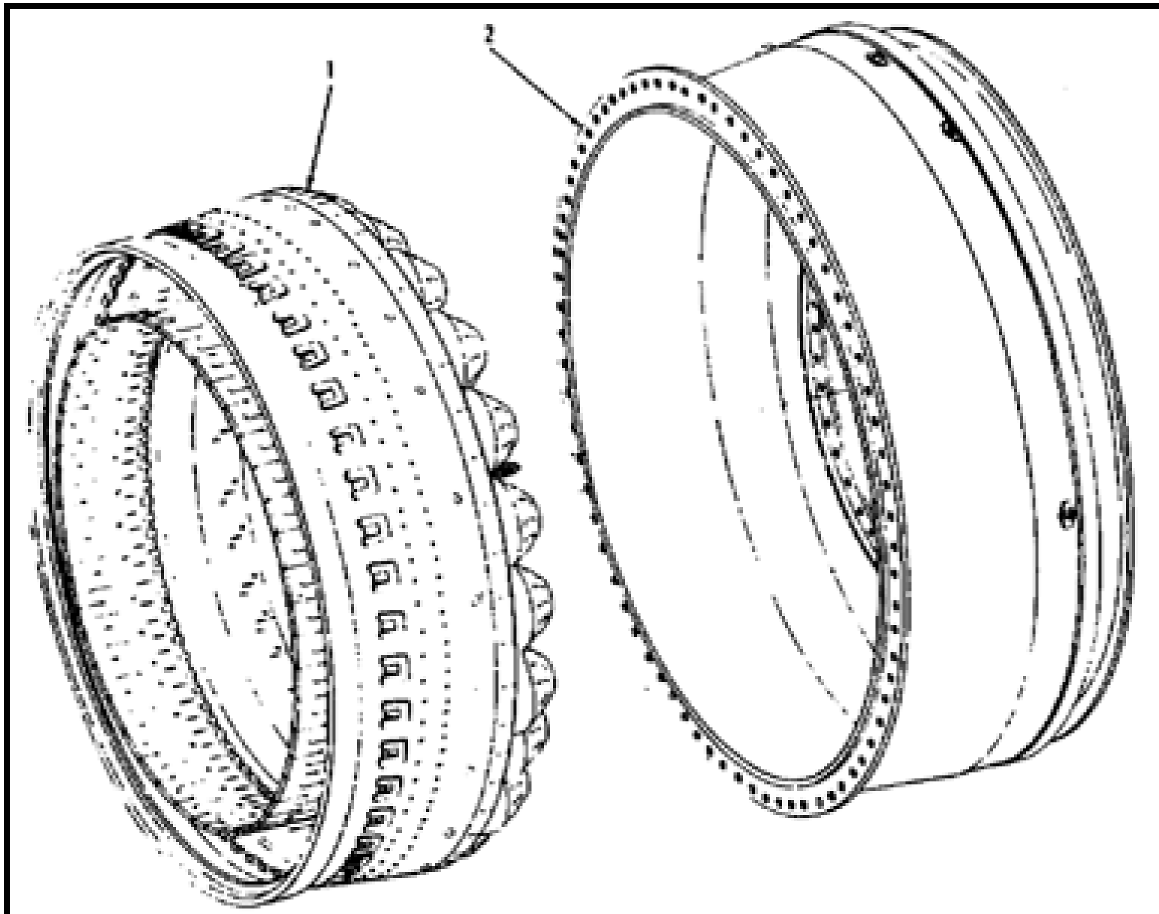
ADVANTAGES

- Mechanically robust.
- Fuel-flow and airflow patterns are easily matched.
- Rig testing necessitates only small fraction of total engine air mass flow.
- Easy replacement for maintenance.

DISADVANTAGES

- Bulky and heavy
- High-pressure loss
- Requires interconnectors that incur problem of light round
- Large frontal area and high drag

ANNULAR TYPE



Annular

- Annular combustor for a gas turbine engine, viewed axis on looking through the exhaust.
- Annular combustors do away with the separate combustion zones and simply have a continuous liner and casing in a ring (the annulus).
- There are many advantages to annular combustors, including more uniform combustion, shorter size (therefore lighter), and less surface area

- Additionally, annular combustors tend to have very uniform exit temperatures.
- They also have the lowest pressure drop of the three designs (on the order of 5%).
- The annular design is also simpler

Advantages and Disadvantages of Annular

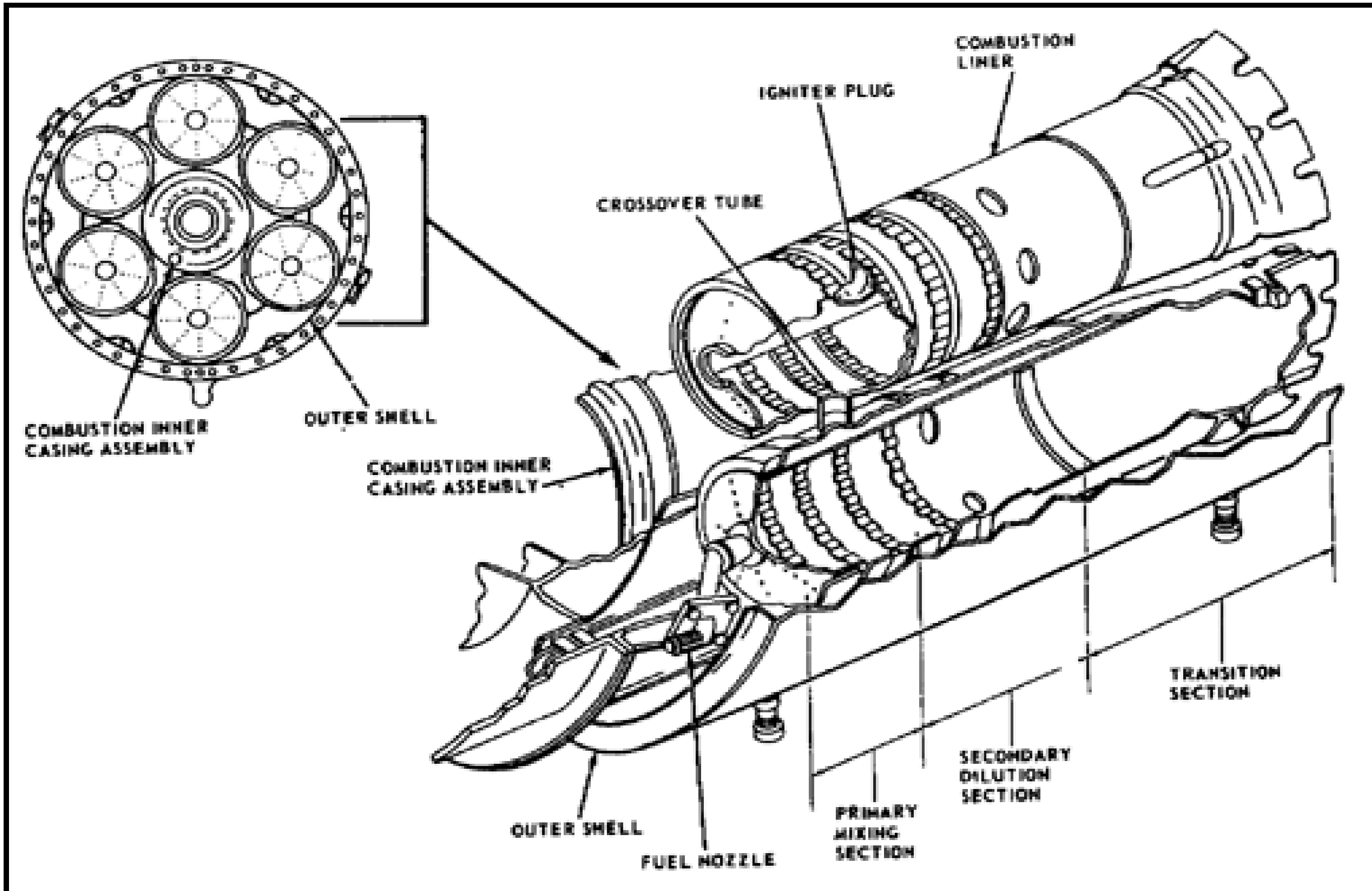
ADVANTAGES

- Minimum length and weight (its length is nearly 0.75 of cannular combustor length).
- Minimum pressure loss.
- Minimum engine frontal area.
- Less wall area than cannular and thus cooling air required is less; thus the combustion efficiency raises; the unburnt fuel is reduced and oxidizes the carbon monoxide to nontoxic carbon dioxide, thus reducing air pollution.
- Easy light round.
- Design simplicity.
- Combustion zone uniformity.
- Permitting better mixing of the fuel and air.
- Simple structure compared to can burners.
- Increased durability.

DISADVANTAGES

- Serious bucking problem on outer liner.
- Rig testing necessitates full engine air mass flow.
- Must remove the engine from aircraft to disassemble for maintenance and overhaul.

CAN ANNULAR



COMBUSTION LINER
IGNITER PLUG
CROSSOVER TUBE
COMBUSTION INNER CASING ASSEMBLY
OUTER SHELL
COMBUSTION INNER CASING ASSEMBLY
TRANSITION SECTION
SECONDARY DILUTION SECTION
PRIMARY MIXING SECTION
OUTER SHELL
FUEL NOZZLE

Can-annular

- Can annular combustors have discrete combustion zones contained in separate liners with their own fuel injectors.
- Unlike the can combustor, all the combustion zones share a common ring (annulus) casing.
- Each combustion zone no longer has to serve as a pressure vessel
- The combustion zones can also communicate" with each other via liner holes or connecting tubes that allow some air to flow circumferentially.

- The exit flow from the can-annular combustor generally has a more uniform temperature profile, which is better for the turbine section. It also eliminates the need for each chamber to have its own igniter.
- Once the fire is lit in one or two cans, it can easily spread to and ignite the others.
- This type of combustor is also lighter than the can type, and has a lower pressure drop (on the order of 6%)

Advantages and Disadvantages of Can-annular

ADVANTAGES

- Mechanically robust
- Fuel-flow and airflow patterns are easily matched.
- Rig testing necessitates only small fraction of total engine air mass flow.
- Shorter and lighter than tubular chambers
- Low-pressure loss .

DISADVANTAGES

- Less compact than annular
- Requires connectors
- Incurs problem of light round

COMBUSTION CHAMBER PERFORMANCE

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,**

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

COMBUSTION CHAMBER PERFORMANCE

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

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

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

COMBUSTION CHAMBER PERFORMANCE

Combustion intensity:-

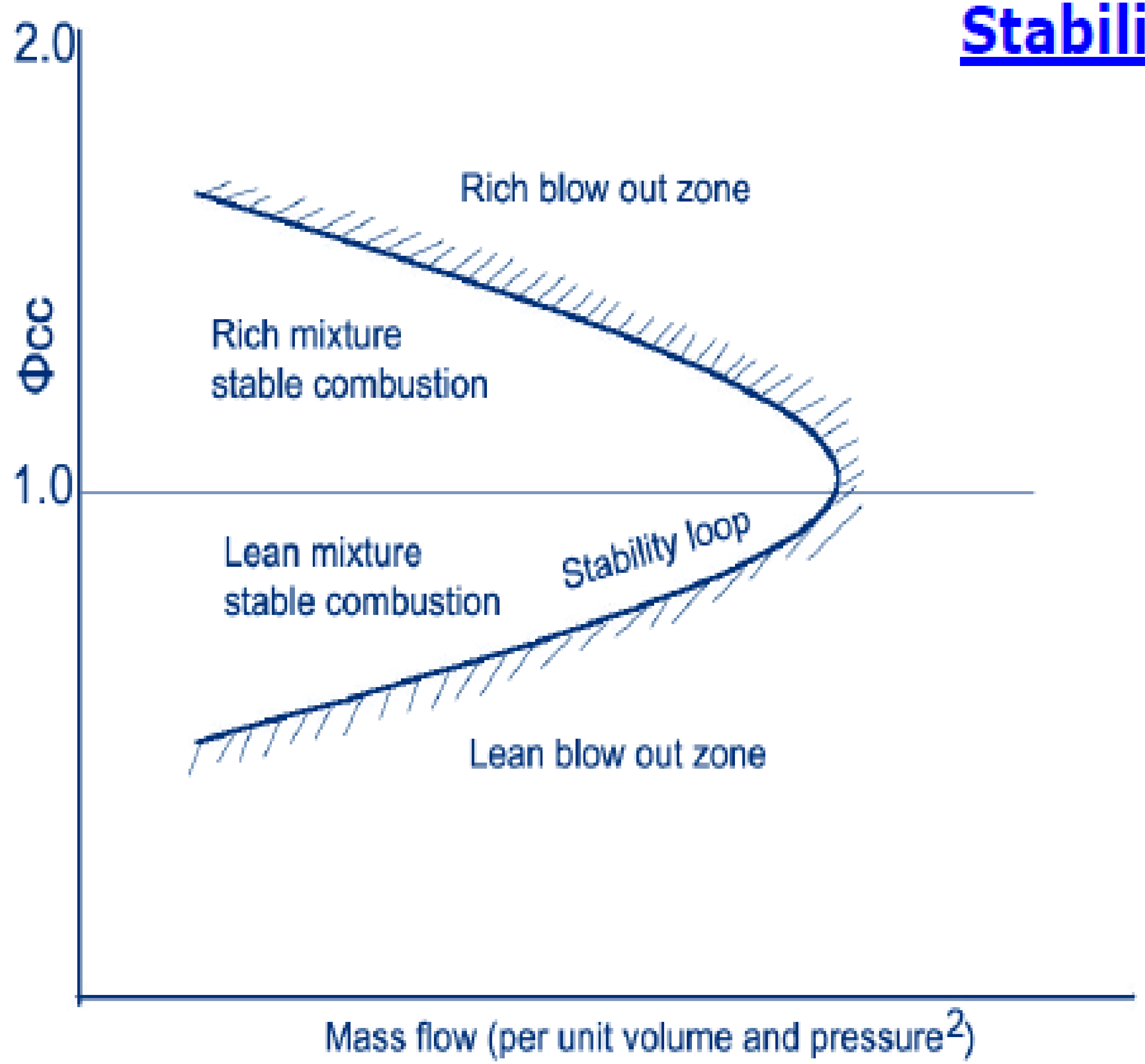
It is insufficient to characterize combustion chambers on the basis of pressure loss and efficiency. The total amount of energy it can release for useful work must be a measure of its performance. Hence, the parameter called *combustion intensity* is introduced as

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

Stability limits

- For any combustion chamber there is both a rich and a weak limit to the air/ fuel ratio beyond which flame is unstable.
- The range between rich and weak limits is narrower at higher air velocities through the combustion zone.
- The *stability loop* must cover the operating region of the gas turbine engine including all flight regimes and all transient regions of accelerations and decelerations.
- While rich limit is attained during acceleration, weak limit may occur in decelerations.
- The *fuel flow rate* needs to be controlled accordingly, which is necessary also to avoid rapid temperature changes in the turbine blades.

Stability limits



Combustion Instability

- Chemical reaction causes energy input in the engine,
- Three factors are responsible for instability :

(i) local and instantaneous air-fuel mixture ratio,
(ii) instantaneous pressure, and
(iii) Instantaneous temperature, which together decide the instantaneous heat release rate.

- Some of the root causes of instabilities are :

1) Changes in turbulent mixing rate
2) Flame area variation
3) Periodicity of the upstream air flow, induced by the instantaneous pressure field (Time unsteadyness)
4) Vortex formation and shedding (around the flame holder) due to or leading to fluid dynamic instabilities

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 :

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

Exhaust nozzles

- Nozzles form the exhaust system of gas turbine engines.
- It provides the thrust force required for all flight conditions.
- In turboprops, nozzles may generate part of the total thrust.
- Main components: tail pipe or tail cone and the exhaust duct.
- Nozzles could be either of fixed geometry or variable geometry configuration.

Exhaust nozzles

- Besides generating thrust, nozzles have other functions too.
- Variable area nozzles are used for adjusting the exit area for different operating conditions of the engine.
- For thrust reversal: nozzle are deflected so as to generate a part of the thrust component in the forward direction resulting in braking.
- For thrust vectoring: vectoring the nozzles to carry out complex manoeuvres.
- Exhaust noise control

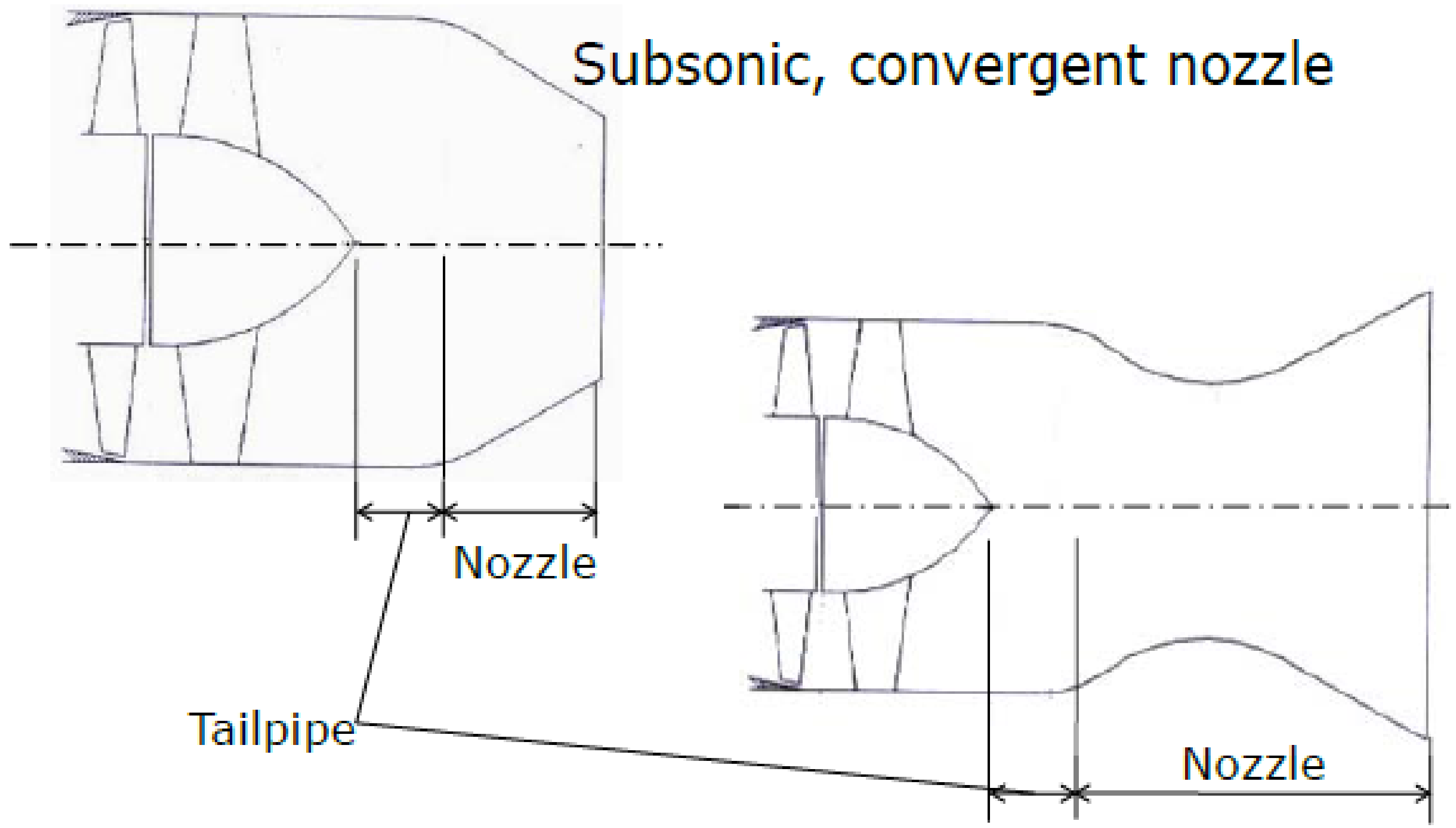
Exhaustive nozzles

- **Nozzle must fulfill the following:**
  - Be matched with other engine components
  - Provide optimum expansion ratio
  - Have minimum losses at design and off-design
  - Permit afterburner operation
  - Provide reversed thrust when necessary
  - Suppress jet noise and IR radiation
  - Provide necessary vectored thrust
  - Have minimal weight, cost and maintenance while satisfying the above.

Exhaust nozzles

- Types of nozzles:
  - Convergent or Converging-diverging
  - Axisymmetric or two-dimensional
  - Fixed geometry or variable geometry
- Simplest is the fixed geometry convergent nozzle
  - Was used in subsonic commercial aircraft.
- Other nozzle geometries are complex and require sophisticated control mechanisms.

Exhaust nozzles: Fixed geometry



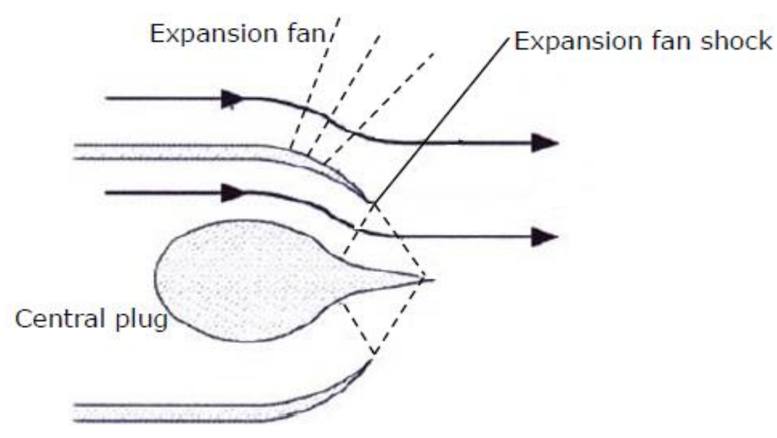
Supersonic, C-D nozzle

Exhaust nozzles

- Convergent nozzles are normally used in subsonic aircraft.
- These nozzles operate under choked condition, leading to incomplete expansion.
- This may lead to a pressure thrust.
- A C-D nozzle can expand fully to the ambient pressure and develop greater momentum thrust.
- However due to increased weight, geometric complexity and diameter, it is not used in subsonic transport aircraft.

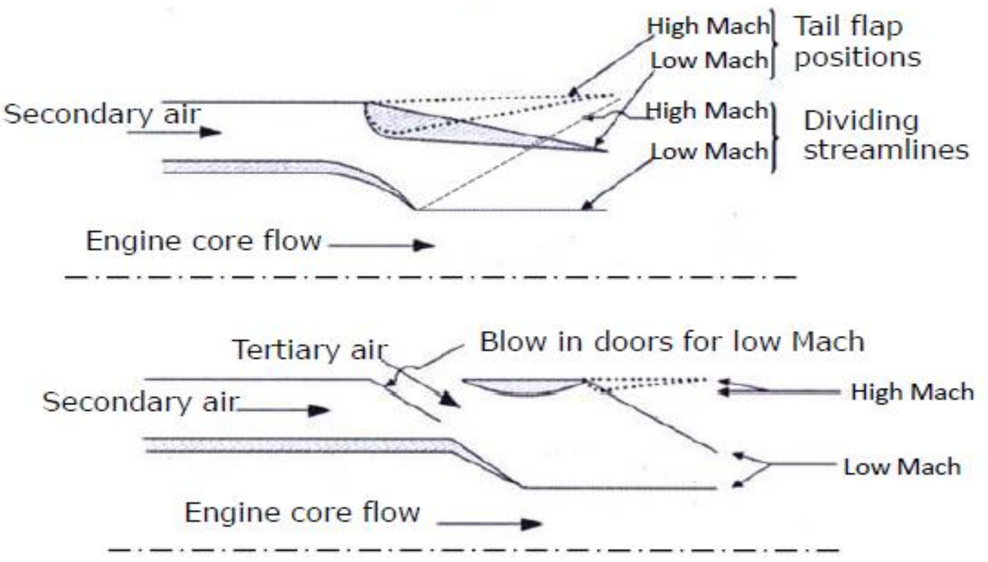
Variable geometry nozzles

- Variable area nozzles or adjustable nozzles are required for matched operation under all operating conditions.
- Three types of variable area nozzles are:
  - Central plug at nozzle outlet
  - Ejector type
  - Iris nozzle
- The Central plug is very similar to the spike of an intake.
- Unlike intake, the central plug causes external expansion fans.



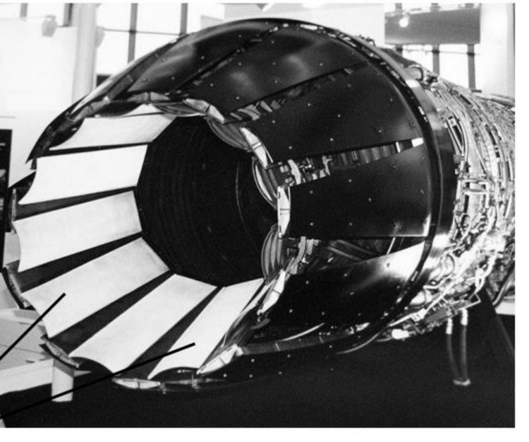
Ejector type nozzles

- Ejector nozzle: creates an effective nozzle through a secondary airflow
- At subsonic speeds, the airflow constricts the exhaust to a convergent shape.
- As the speed increases, the two nozzles dilate and the two nozzles form a CD shape.
- Some configurations may also have a tertiary airflow.
- SR-71, Concorde, F-111 have used this type of nozzle.



Variable geometry nozzles

- Iris nozzle: uses overlapping, adjustable petals.
- More complicated than the ejector type nozzle.
- Offers significantly higher performance.
- Used in advanced military aircraft.
- Some of the modern aircraft also have iris nozzles that can be deflected to achieve vectored thrust.



Iris petals for variable geometry

Thrust vectoring

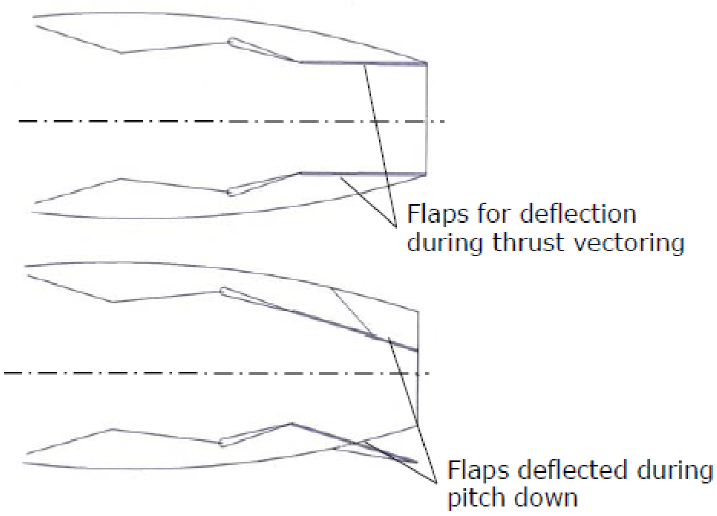
- Directing the thrust in a direction other than that parallel to the vehicles' longitudinal axis.
- This allows the aircraft to undergo maneuvers that conventional control surfaces like ailerons or flaps cannot provide.
- Used in modern day combat aircraft.
- Provides exceptional agility and maneuvering capabilities.

Thrust vectoring

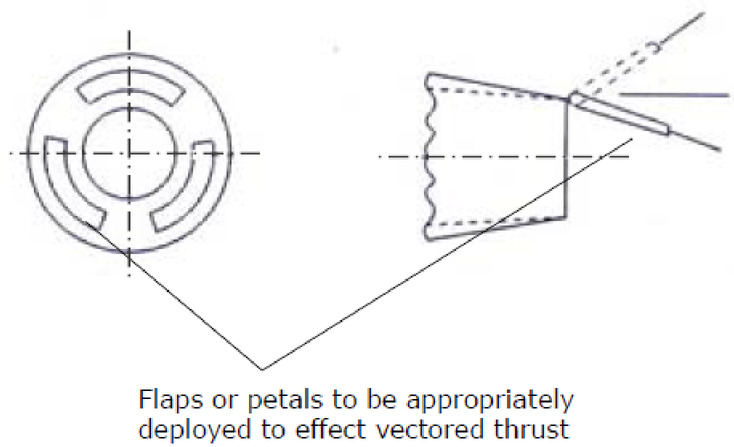
- There are two types of thrust vector controls:
  - Mechanical control
  - Fluidic control
- Mechanical control involves deflecting the engine nozzle and thus physically alter the direction of thrust.
- Fluidic vectoring involves either injecting fluid or removing it from the boundary layer of the primary jet.

- Mechanical vectoring system is heavier and complex.
- There are two types of mechanical thrust vectoring
  - Internal thrust vectoring
  - External thrust vectoring
- Internal thrust vectoring permits only pitch control.
- External thrust vectoring can be used for pitch and yaw controls.

Internal thrust vectoring

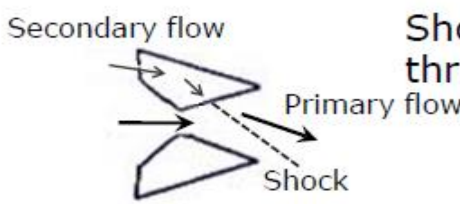


External thrust vectoring

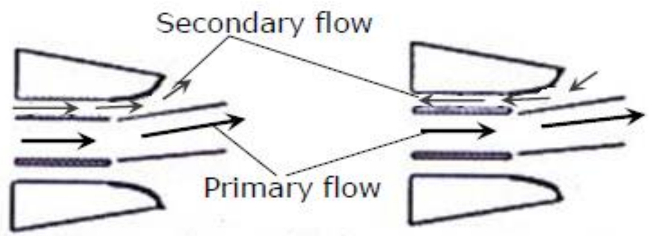


Fluidic thrust vectoring

- Fluidic thrust vectoring has been demonstrated successfully at a laboratory scale.
- This method has several advantages over the mechanical control.
- Main challenge lies in ensuring an effective control with a linear response.
- Other concepts like Shock thrust vector control, coflow and counter flow thrust vectoring concepts are also being pursued.



Shock vector thrust vectoring



Co-flow and counter-flow thrust vectoring

Mass Flow Rate and Characteristic Velocity

$$\dot{m} = \rho_t A_t V_t = \frac{\rho_t}{\rho_c} \rho_c A_t \sqrt{\gamma R T_t}$$

$$\dot{m} = \frac{P_t}{RT_t} \frac{RT_c}{P_c} \frac{P_c}{RT_c} A_t \sqrt{\gamma R T_t}$$

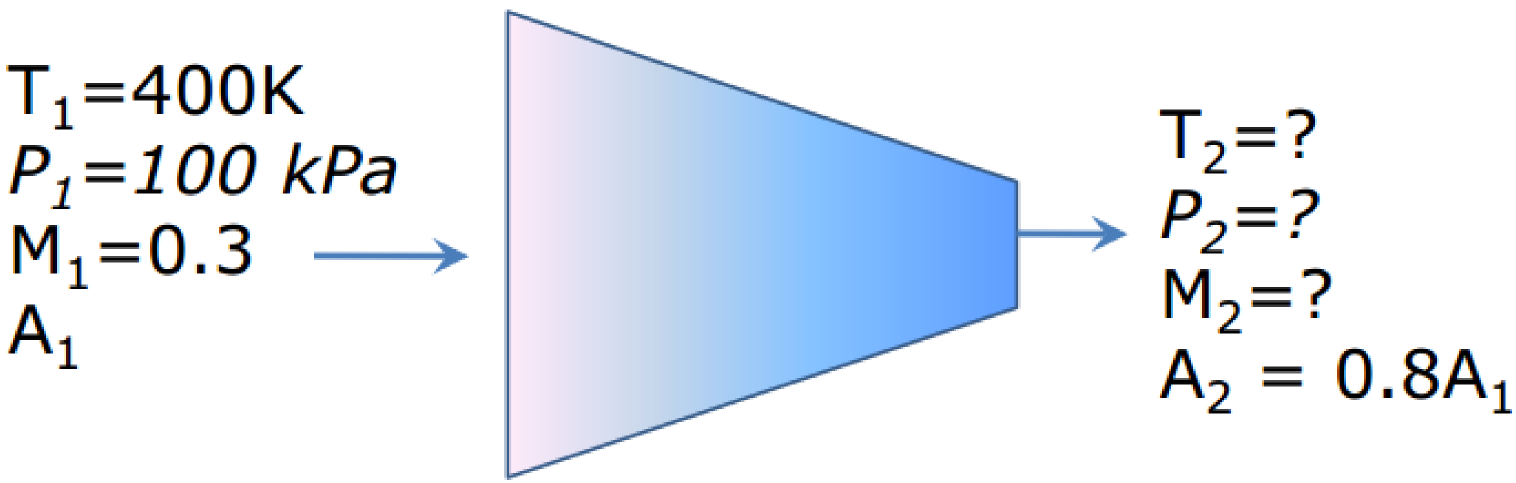
$$\dot{m} = \frac{P_t}{P_c} \frac{T_c}{T_t} \frac{P_c A_t}{\sqrt{RT_c}} \sqrt{\gamma} \sqrt{\frac{T_t}{T_c}}$$

$$\dot{m} = \left(\frac{2}{\gamma+1} \right)^{\frac{\gamma}{\gamma-1}} \left(\frac{2}{\gamma+1} \right)^{-1} \frac{P_c A_t}{\sqrt{RT_c}} \sqrt{\gamma} \sqrt{\frac{2}{\gamma+1}}$$

$$\dot{m} = \left(\frac{2}{\gamma+1} \right)^{\frac{\gamma}{\gamma-1} - 1 + \frac{1}{2}} \frac{P_c A_t}{\sqrt{RT_c}} \sqrt{\gamma}$$

$$\dot{m} = \sqrt{\gamma} \left(\frac{2}{\gamma+1} \right)^{\frac{\gamma+1}{2(\gamma-1)}} \frac{P_c A_t}{\sqrt{RT_c}}$$

Air enters a converging duct with varying flow area at T_1 =400 K, P_1=100 kPa, and M_1=0.3. Assuming steady isentropic flow, determine T_2, P_2, and M_2 at a location where the flow area has been reduced by 20 percent.



From the isentropic tables, for a Mach number of 0.3,

$A_1/A^*=2.0351$, $T_1/T_0=0.9823$, $P_1/P_0=0.9395$

With a 20% area reduction, $A_2=0.8A_1$

$A_2/A^*= A_2/A_1 \times A_1/A^* =0.8 \times 2.0351$
$=1.6281$

For this value of area ratio, from the isentropic tables, $T_2/T_0=0.9701$, $P_2/P_0=0.8993$ and therefore $M_2=0.391$

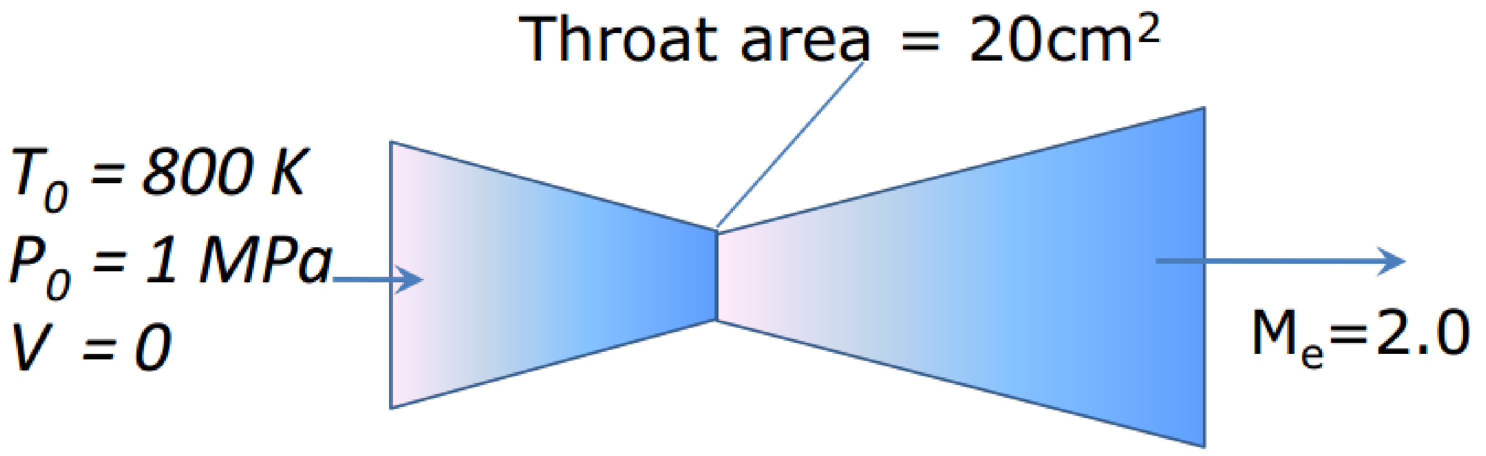
$$\frac{T_2}{T_1} = \frac{T_2 / T_0}{T_1 / T_0} \rightarrow T_2 = T_1\left(\frac{T_2 / T_0}{T_1 / T_0}\right) = 400\left(\frac{0.9701}{0.9823}\right)$$

$$T_2 = 395 \text{ K}$$

$$\frac{P_2}{P_1} = \frac{P_2 / P_0}{P_1 / P_0} \rightarrow P_2 = P_1\left(\frac{P_2 / P_0}{P_1 / P_0}\right) = 100\left(\frac{0.8993}{0.9395}\right)$$

$$P_2 = 95.7 \text{ kPa}$$

Air enters a converging–diverging nozzle, shown in the Figure, at 1.0 MPa and 800 K with a negligible velocity. For an exit Mach number of M=2 and a throat area of 20 cm², determine (a) the throat conditions, (b) the exit plane conditions, including the exit area, and (c) the mass flow rate through the nozzle.



The nozzle exit Mach number is given as 2.0. Therefore the throat Mach number must be 1.0.

Since the inlet velocity is negligible, the stagnation pressure and stagnation temperature are the same as the inlet temperature and pressure, P_0=1.0 MPa and T_0=800 K.

$$\therefore \rho_0 = P_0 / RT_0 = 4.355 \mathrm{kg} / \mathrm{m}^3$$

(a) At the throat, $M = 1$. From the isentropic tables,

$$\frac{P*}{P_0} = 0.5283, \quad \frac{T*}{T_0} = 0.8333, \quad \frac{\rho*}{\rho_0} = 0.6339$$

$$P* = 0.5283P_0 = 0.5283 \text{ MPa}$$

$$T* = 0.8333T_0 = 666.6 \text{ K}$$

$$\rho* = 0.6339\rho_0 = 2.761 \text{ kg/m}^3$$

Therefore, $V^* = \sqrt{\gamma RT*} = 517.5 \text{ m/s}$

(b) At the nozzle exit, $M = 2$. From the isentropic tables,

$$\frac{P_e}{P_0} = 0.1278, \quad \frac{T_e}{T_0} = 0.5556, \quad \frac{\rho_e}{\rho_0} = 0.2300,$$

$$M* = 1.6330, \quad \frac{A_e}{A*} = 1.6875$$

Therefore,

$$P_e = 0.1278\,P_0 = 0.1278\ \text{MPa}$$

$$T_e = 0.5556\,T_0 = 444.5\,\text{K}$$

$$\rho_e = 0.2300\,\rho_0 = 1.002\ \text{kg}/\text{m}^3$$

$$A_e = 1.6875A^* = 33.75\ \text{cm}^2$$

The nozzle exit velocity can be determined from $V_e = M_e\sqrt{\gamma R T_e} = 2\sqrt{1.4 \times 287 \times 444.5}$

$$= 845.2\ \text{m}/\text{s}$$

(c) The mass flow rate can be calculated based on the properties at the throat, since the flow is choked.

$$\dot{m} = \rho^* A^* V^* = 2.761 \times 0.0002 \times 517.5$$

$$= 2.86\ \text{kg}/\text{s}$$