TURBO MACHINES

BME IV/I

Chapter Four: Gas Turbine Nozzle

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Chapter overview

- Gas turbine nozzle
- Types of nozzle
- Thrust vectoring
- Thrust reversal
- Noise control
- Nozzle choking
- Flow through Convergent Nozzle
- Flow through Convergent Divergent Nozzle

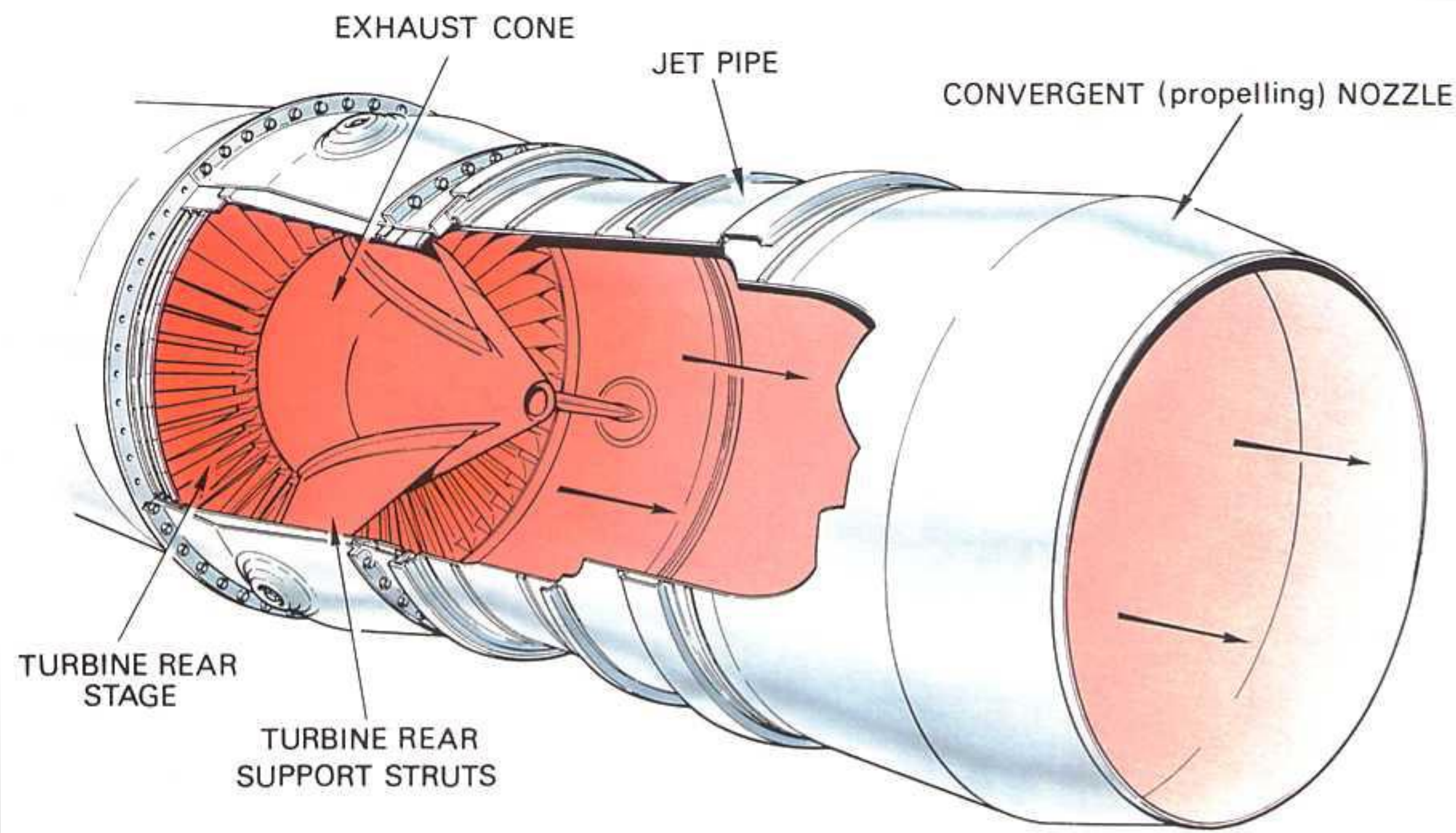
Gas turbine nozzle

A exhaust nozzle system is for the following objectives:

- 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.
- 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 maneuvers (exercises).
- Exhaust noise control

CONTD…

- Main components: tail pipe or tail cone and the exhaust duct.



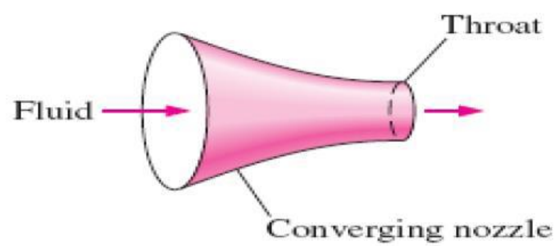
CONTD…

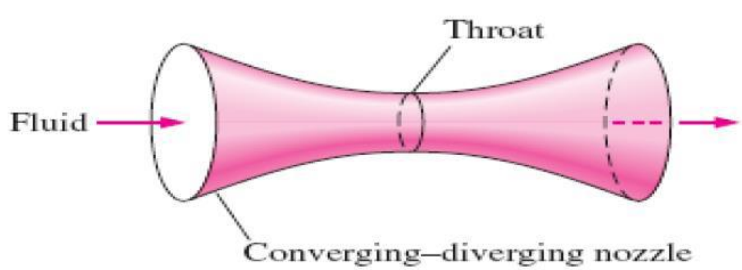
Nozzle must fulfill the following:

- 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.

Types of nozzle

- Types:
  - Convergent and Converging-diverging (CD Nozzle / De-Laval Nozzle)
  - Fixed geometry and variable geometry
- Simplest is the fixed geometry convergent nozzle which is used in subsonic commercial aircraft.
- Other nozzle geometries are complex and require sophisticated control mechanisms.





CONTD…

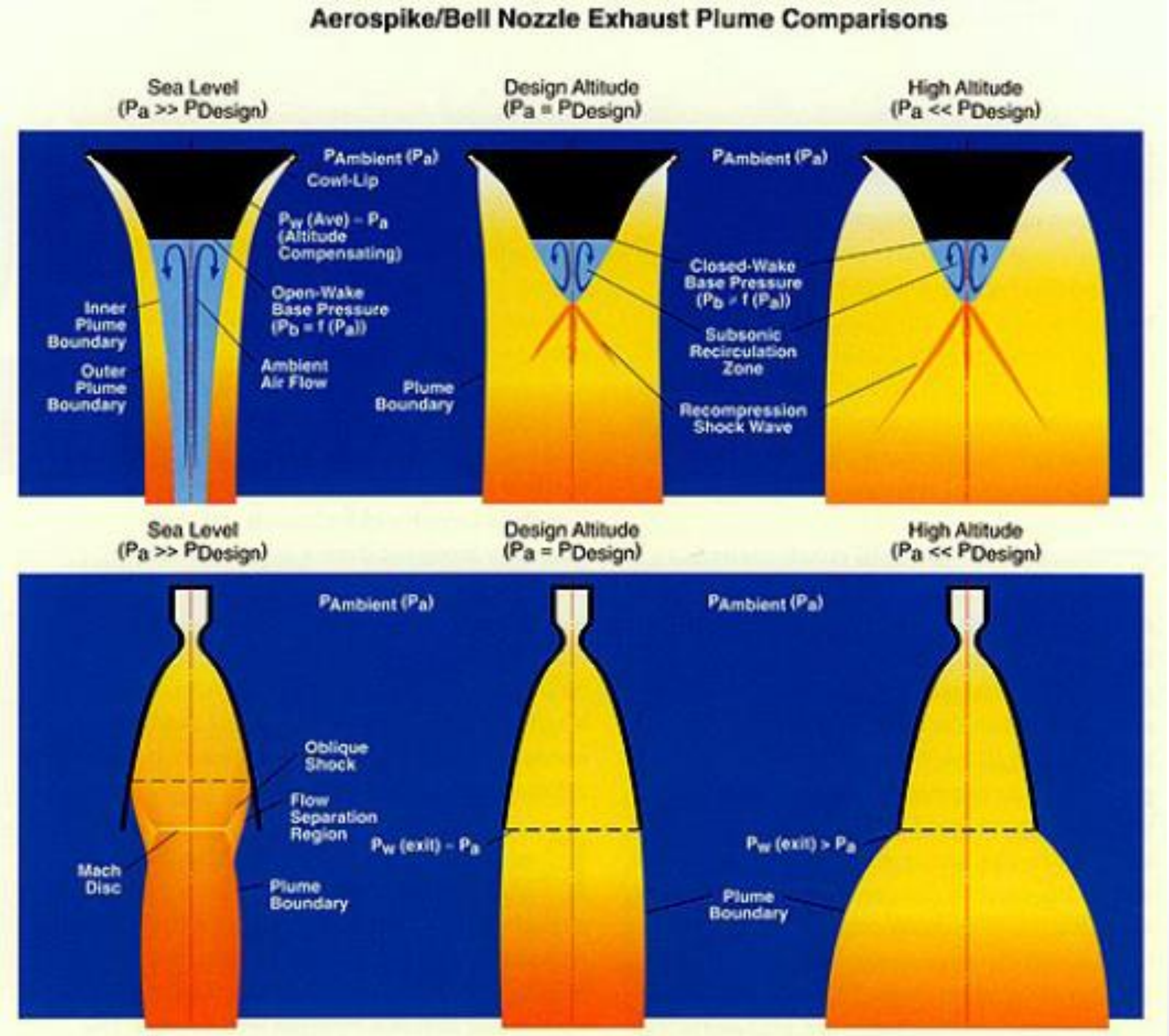
- Convergent nozzles are normally used in subsonic aircraft.
- These nozzles operate under choked condition, leading to incomplete expansion.
- This may lead to a pressure thrust.
- A C-D nozzle can expand fully to the ambient pressure and develop greater momentum thrust.
- However due to increased weight,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

CONTD…

Aerospike nozzle: fix geometry

- This is a novel design for a fix-geometry nozzle to get adapted at all altitudes.
- Instead of an outward flow in a bell-shape wall boundary, in the aerospike nozzle an annular flow issues radially inward along a decreasing-diameter inner wall (the spike), without external wall.
- The outer ambient pressure regulates the outer plume boundary so that when $p_e < p_0$ (over-expansion at low altitudes) the external pressure squeezes and makes the plume thinner, further accelerating the exhaust instead of detaching it from the walls.
- *Since ambient pressure controls the nozzle expansion, the flow area at the end of the aerospike changes with altitude*.
- The length of an ideal spike is about 150 % of a 15° conical nozzle, but performances reduce very little if the spike length is truncated to the 20 % range.

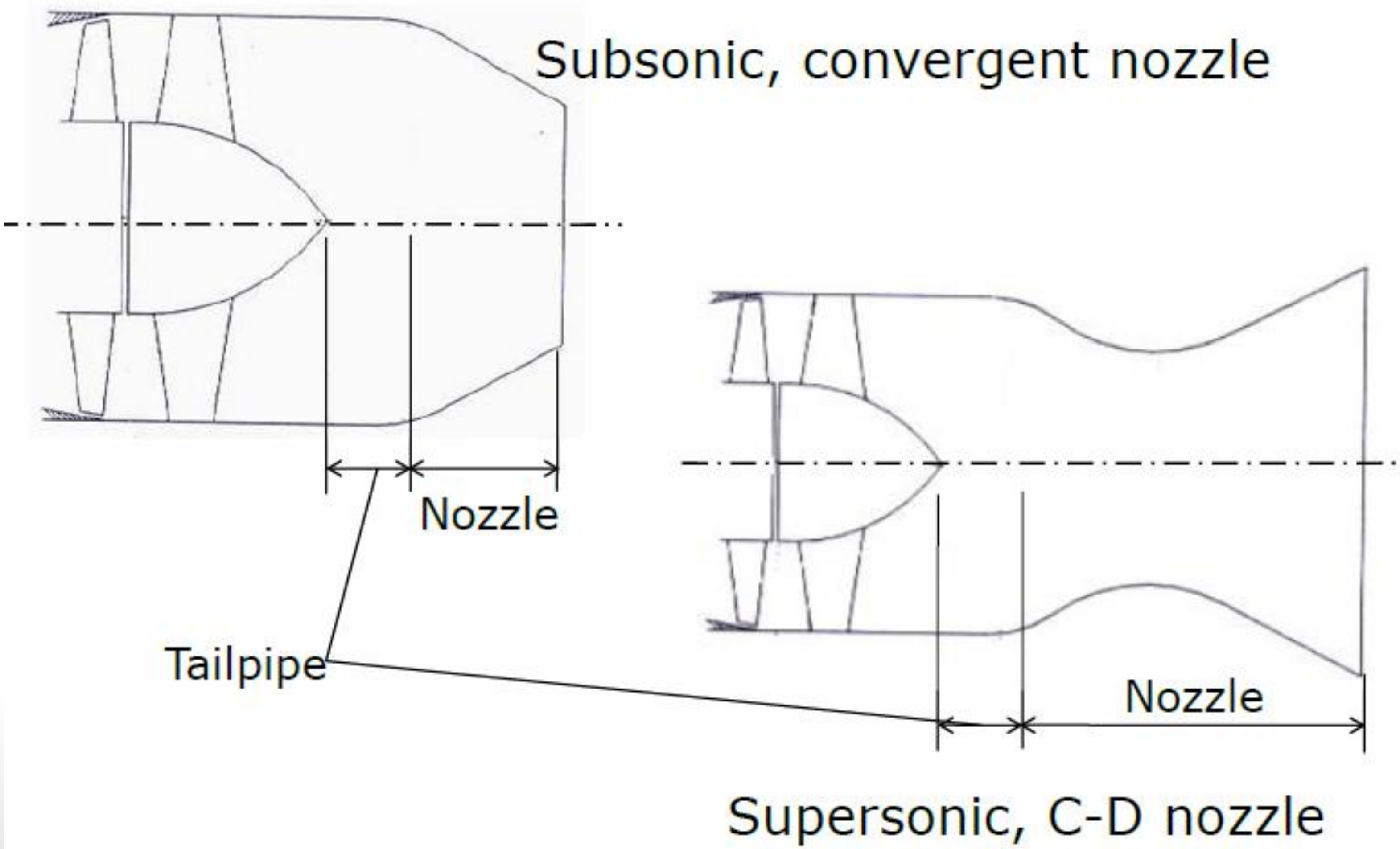
CONTD…



Aerospike/Bell Nozzle Exhaust Plume Comparisons
Sea Level
(Pa >> PDesign)
Design Altitude
(Pa = PDesign)
High Altitude
(Pa << PDesign)
PAmbient (Pa)
Cowl-Lip
Pw (Ave) ~ Pa
(Altitude
Compensating)
Open-Wake
Base Pressure
(Pb = f (Pa))
Inner
Plume
Boundary
Outer
Plume
Boundary
Ambient
Air Flow
Plume
Boundary
Closed-Wake
Base Pressure
(Pb ≠ f (Pa))
Subsonic
Recirculation
Zone
Recompression
Shock Wave
Sea Level
(Pa >> PDesign)
Design Altitude
(Pa = PDesign)
High Altitude
(Pa << PDesign)
PAmbient (Pa)
Oblique
Shock
Flow
Separation
Region
Mach
Disc
Plume
Boundary
Pw (exit) ~ Pa
Pw (exit) > Pa
Plume
Boundary

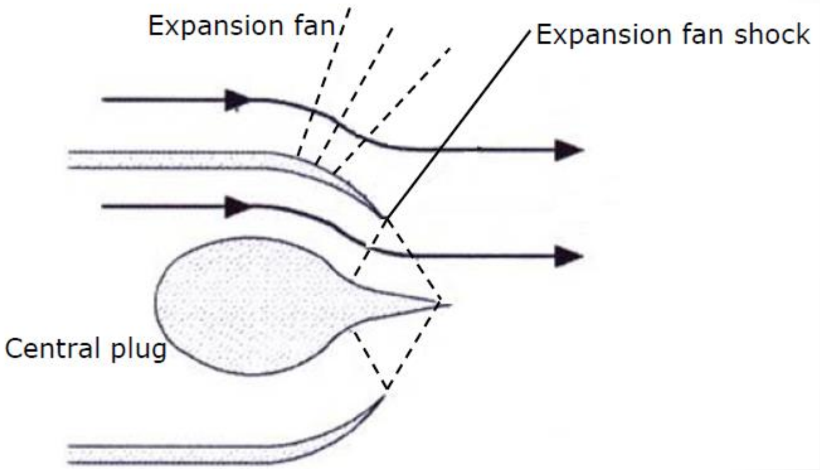
CONTD…

Exhaust nozzles: Variable geometry



CONTD…

- 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



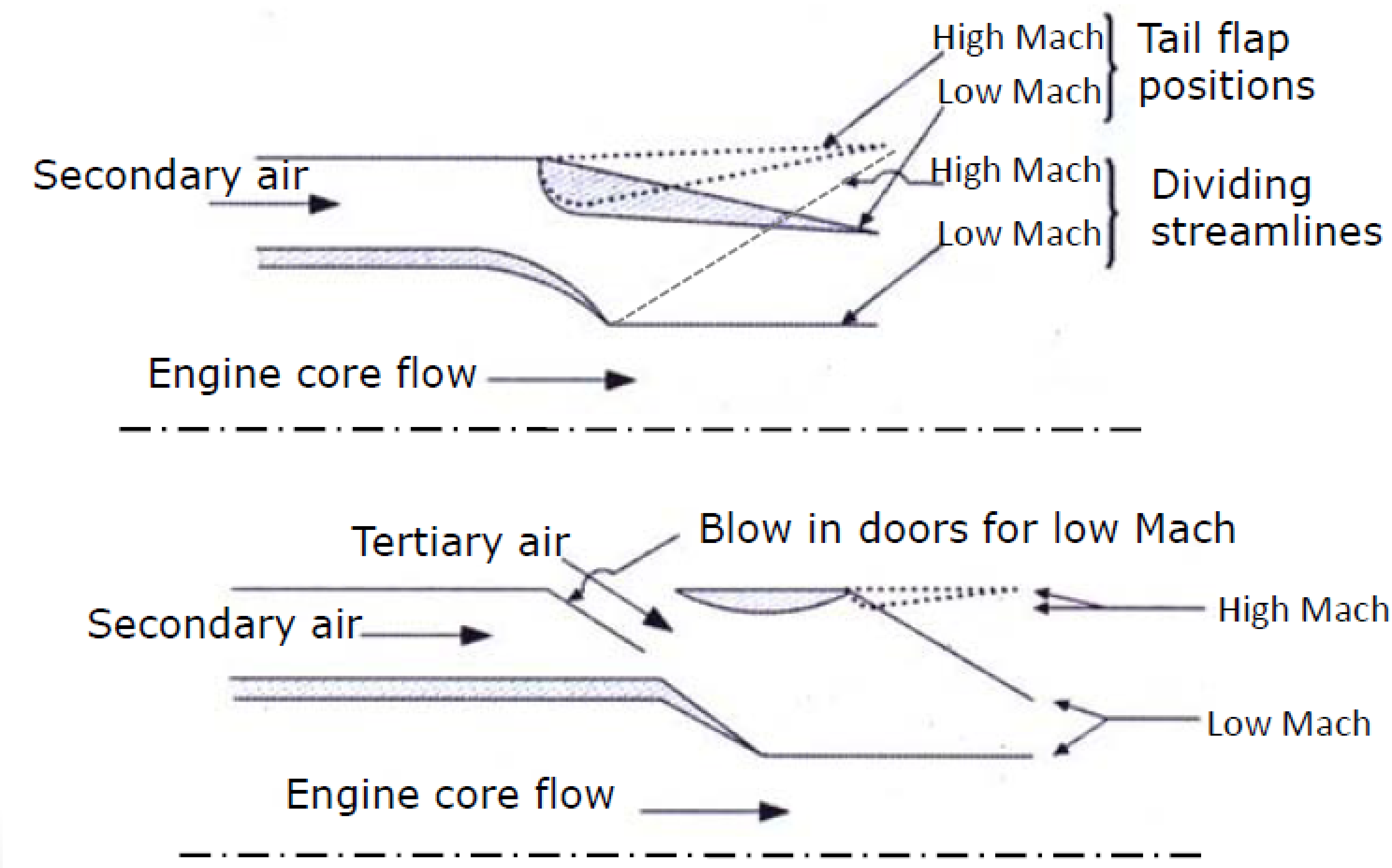
- The **Central plug** is very similar to the spike of an intake.
- Unlike intake, the central plug causes external expansion fans.

CONTD…

- **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.

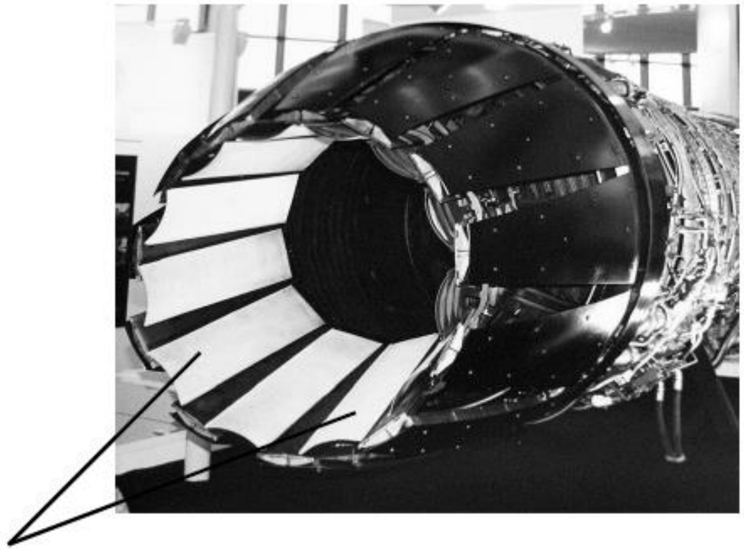
CONTD…

- Ejector nozzles



CONTD…

- **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

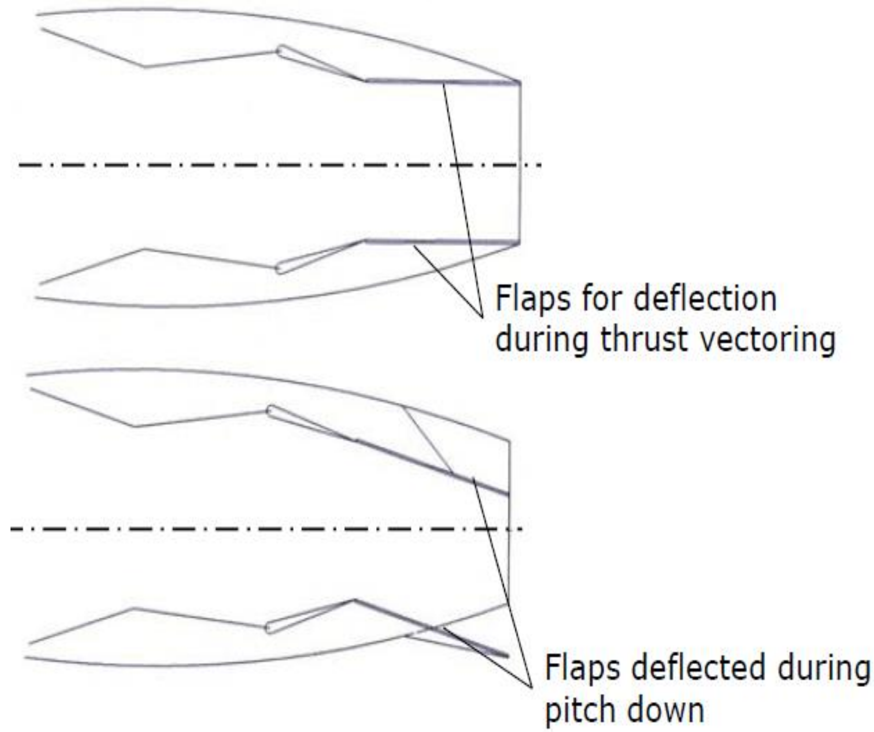
- Directing the thrust in a direction other than that parallel to the vehicles' longitudinal axis.
- This allows the aircraft to undergo maneuvers that conventional control surfaces like ailerons or flaps cannot provide.
- Provides exceptional agility and maneuvering capabilities.
- Thrust vectoring was originally developed as a means for V/STOL (Vertical or Short Take Off and Landing).
- Thrust vectored aircraft have better climb rates, besides extreme maneuvers.
- Most of the modern day combat aircraft have thrust vectoring.

CONTD…

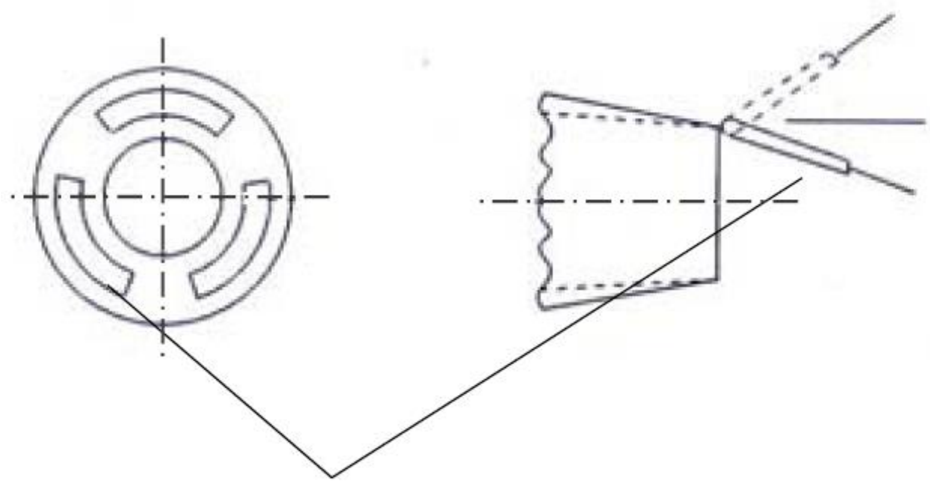
- There are two types of thrust vector controls:
  - Mechanical control-Internal and External
  - Fluidic control
- Mechanical vectoring system is heavier and complex.
- Mechanical control involves deflecting the engine nozzle and thus physically alter the direction of thrust.
  - Internal thrust vectoring permits only pitch control.
  - External thrust vectoring can be used for pitch and yaw controls.
- Fluidic vectoring involves either injecting fluid or removing it from the boundary layer of the primary jet.

CONTD…

- Internal thrust vectoring

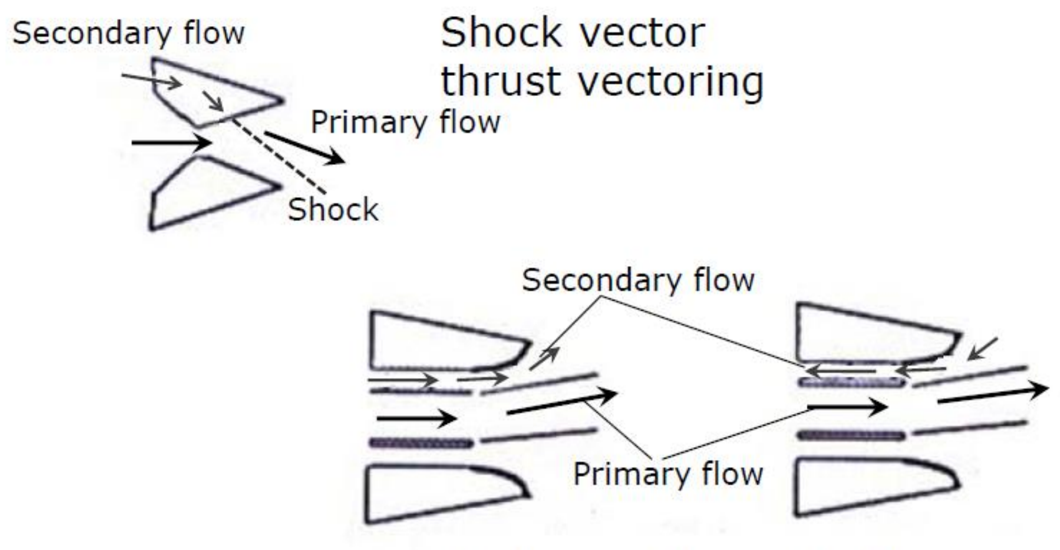


- External thrust vectoring



CONTD…

- 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.



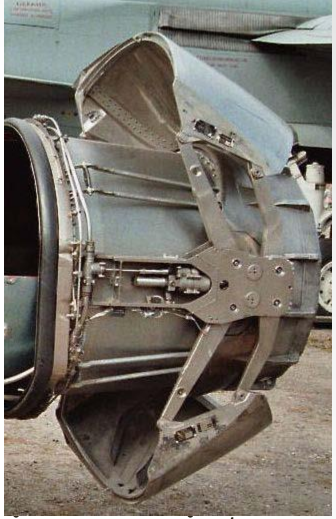
Co-flow and counter-flow thrust vectoring

Thrust reversal

- With increasing size and loads of modern day aircraft, wheel brakes alone cannot brake and aircraft.
- Deflecting the exhaust stream to produce a component of reverse thrust will provide an additional braking mechanism.
- Most of the designs of thrust reversers have a discharge angle of about 45o
- Therefore a component of the thrust will now have a forward direction and therefore contributes to braking.

- There are three types of thrust reversal mechanisms that are used
  - Clamshell type
  - External bucket type
  - Blocker doors

CONTD…



<u>Clamshell type Thrust reversal</u>

- It is normally pneumatically operated system.
- When deployed, doors rotate and deflect the primary jet through vanes.
- These are normally used in non-afterburning engines.

- **Bucket type system** uses bucket type doors to deflect the gas stream.
- In normal operation, the reverser door form part of the convergent divergent nozzle.
- **Blocker doors** are normally used in high bypass turbofans.
- The cold bypass flow is deflected through cascade vanes to achieve the required flow deflection.

Noise control


Noise control using corrugations/serrations at the nozzle exit

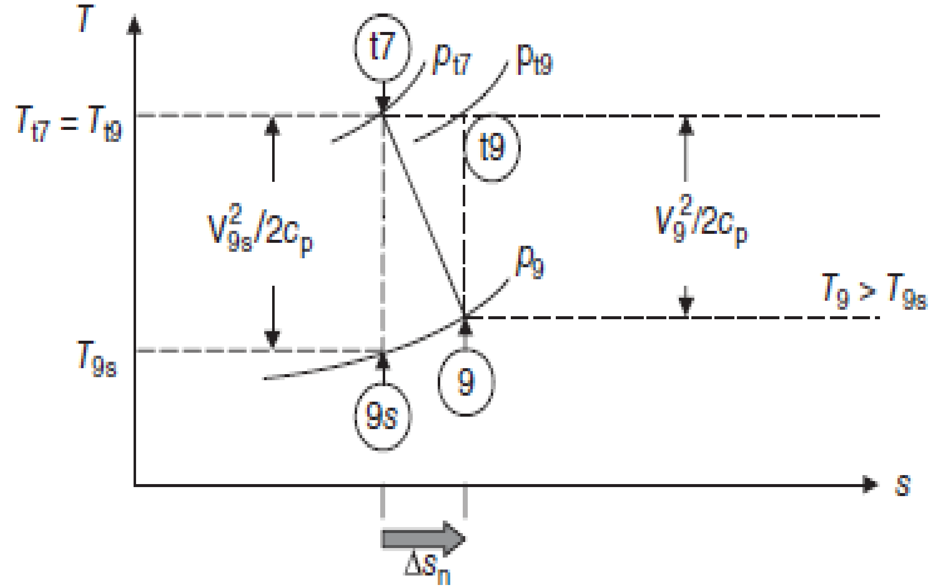
- Jet exhaust noise is a major contributor to the overall noise generated by an aircraft.
- Jet exhaust noise is caused by the turbulent mixing of the exhaust gases with the lower velocity ambient air.
- Nozzle geometry can significantly influence the exhaust noise characteristics.
- Better mixing between the jet exhaust and the ambient can be achieved by properly contouring the nozzle exit.
- Corrugations or lobes (multiple tubes) are some of the methods of achieving lower jet exhaust noise.

NOZZLE CHOKING

- The exhaust gases pass to atmosphere through the propelling nozzle, which is a convergent duct, thus increasing the gas velocity.
- In a turbojet engine, the exit velocity of the exhaust gases is subsonic at low thrust conditions only.
- During most operating conditions, *the exit velocity reaches the speed of sound in relation to the exhaust gas temperature and the propelling nozzle is then said to be 'choked'*; that is, no further increase in velocity can be obtained unless the temperature is increased.
- As the upstream total pressure is increased above the value at which the propelling nozzle becomes 'choked', the static pressure of the gases at exit increases above atmospheric pressure.
- *This pressure difference across the propelling nozzle gives what is known as 'pressure thrust' and is effective over the nozzle exit area*.
- This is additional thrust to that obtained due to the momentum change of the gas stream.

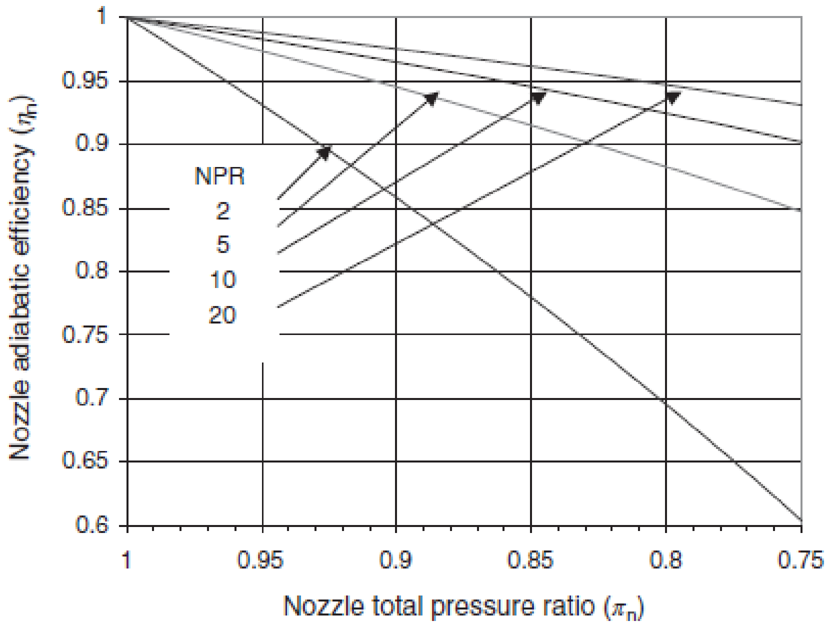
CONTD…

- The ratio of actual kinetic energy at the nozzle to the ideal kinetic energy that emerges from an isentropic expansion in the nozzle is defined as the **nozzle adiabatic (or isentropic) efficiency**.
- Actual exhaust gases has a higher temperature than the exhaust gases emerging from an isentropic nozzle.



$$n_n = \frac{h_{t7} - h_9}{h_{t7} - h_{9s}} = \frac{\frac{V_9{}^2}{2}}{\frac{V_{9s}{}^2}{2}} = \frac{\frac{V_9{}^2}{2}}{\left[\frac{V_9{}^2}{2}\right]_{ideal}}$$

CONTD…



- Nozzle total pressure recovery:

$$\pi_n = \frac{p_{t9}}{p_{t7}}$$

- Nozzle Pressure Ratio (NPR):

$$NPR = \frac{p_{t7}}{p_0}$$

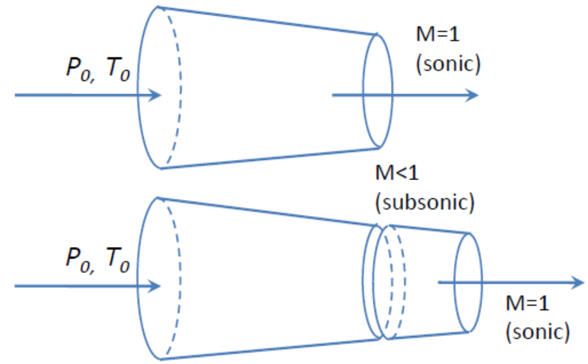
- Critical nozzle pressure ratio:
  - It is the minimum NPR that chokes the flow.

$$p_{t8} = p_8\left(1 + \frac{\gamma - 1}{2}M_8^2\right)^{\frac{\gamma}{\gamma-1}} = p_0\left(\frac{\gamma + 1}{2}\right)^{\frac{\gamma}{\gamma-1}}$$

- **Relation Between Nozzle Figures of Merit, ηn and πn**

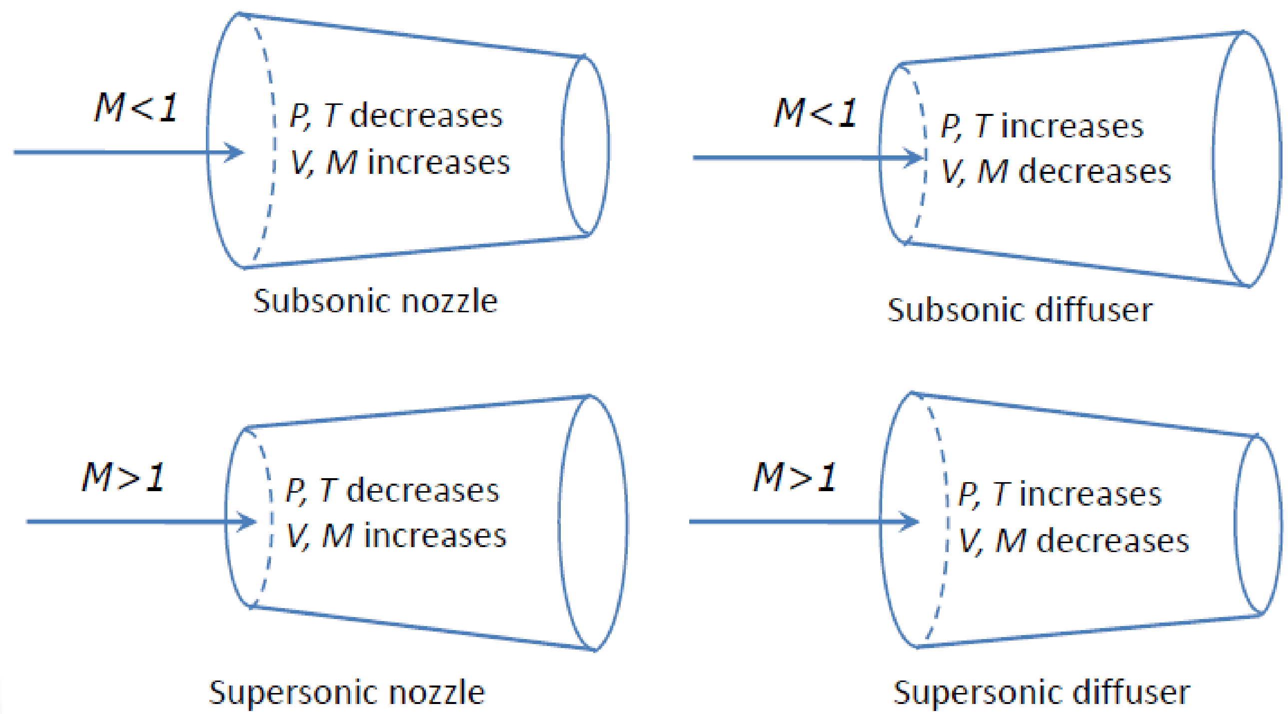
$$\eta_n = \frac{\left\{\mathrm{NPR}\left(\frac{p_0}{p_9}\right)\right\}^{\frac{\gamma-1}{\gamma}} - \pi_n^{-\frac{\gamma-1}{\gamma}}}{\left\{\mathrm{NPR}\left(\frac{p_0}{p_9}\right)\right\}^{\frac{\gamma-1}{\gamma}} - 1} \quad \text{or} \quad \eta_n = \frac{\{\mathrm{NPR}\}^{\frac{\gamma-1}{\gamma}} - \pi_n^{-\frac{\gamma-1}{\gamma}}}{\{\mathrm{NPR}\}^{\frac{\gamma-1}{\gamma}} - 1} \text{ for } p_9 = p_0$$

Convergent Nozzle



Variation of fluid velocity with flow area

- Sonic velocity will occur at the exit of the converging extension, instead of the exit of the original nozzle, and the mass flow rate through the nozzle will decrease because of the reduced exit area.



CONTD…

- Converging nozzle in a subsonic flow will have decreasing area along the flow direction.
- We shall consider the effect of back pressure on the exit velocity, mass flow rate and pressure distribution along the nozzle.
- We assume flow enters the nozzle from a reservoir so that inlet velocity is zero.
- Stagnation temperature and pressure remains unchanged in the nozzle.

CONTD…

Let us consider a calorically perfect gas flow through a nozzle.

The mass flow through the nozzle is

$$\dot{m} = \rho u A = \left(\frac{P}{RT}\right)\left(M\sqrt{\gamma RT}\right)A = (MA)\left(\frac{P}{P_0}\right)P_0\frac{\sqrt{\gamma}}{\sqrt{RT}}\sqrt{\frac{T}{T_0}}$$

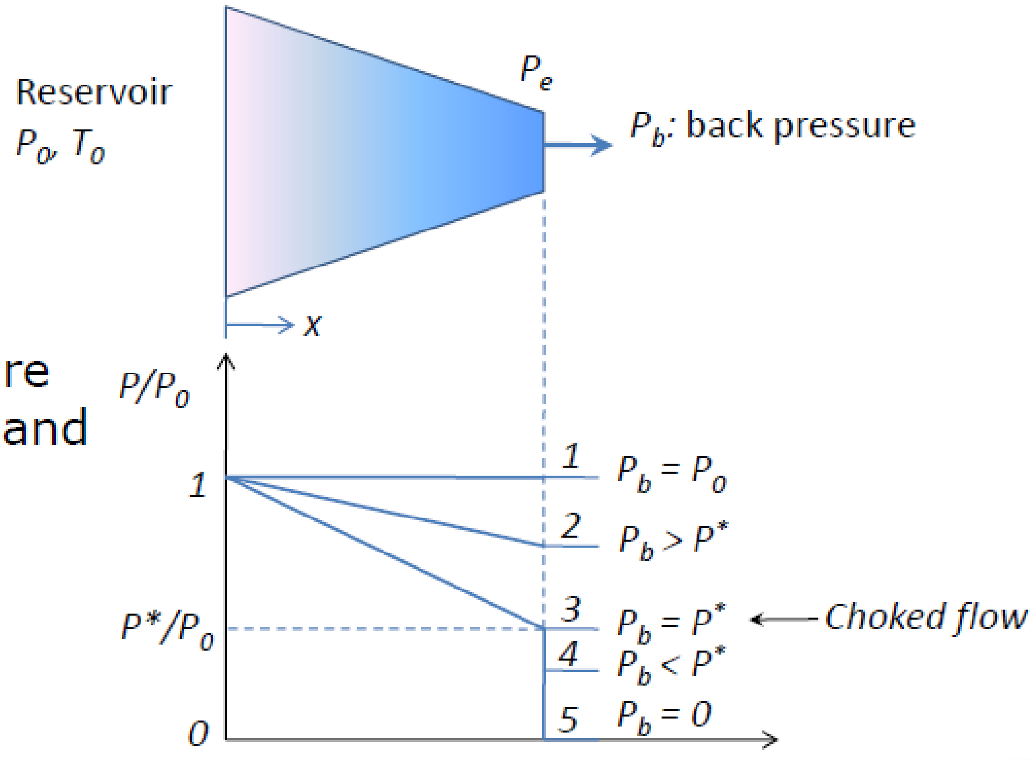
$$= \frac{\sqrt{\gamma}P_0}{\sqrt{T_0 R}}MA\frac{\left\{1+((\gamma-1)/2)M^2\right\}^{1/2}}{\left\{1+((\gamma-1)/2)M^2\right\}^{\gamma/(\gamma-1)}}$$

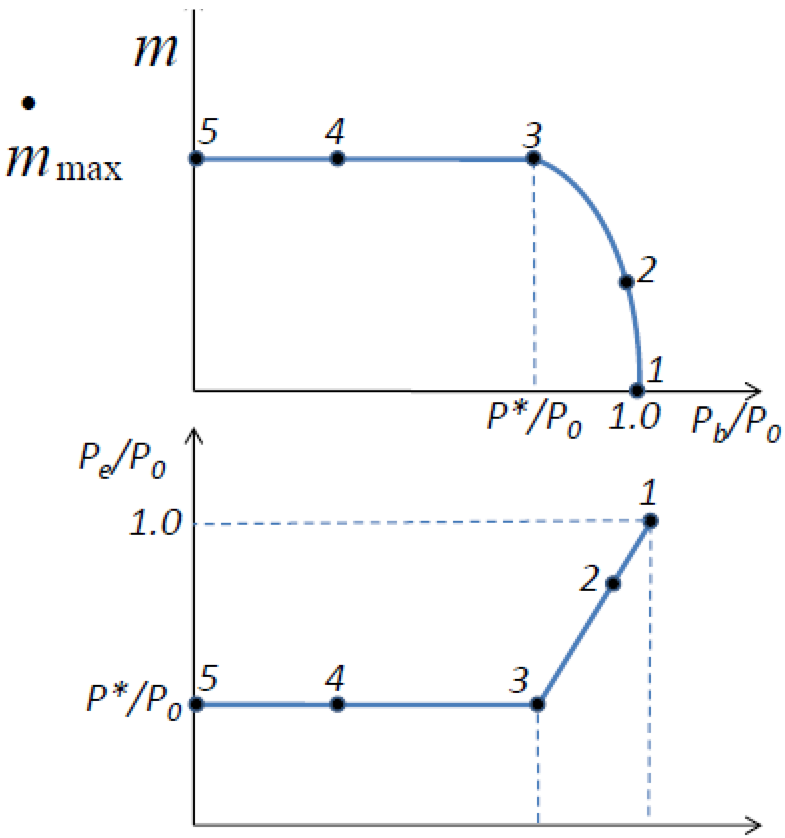
This on simplication reduces to

$$\dot{m} = \frac{AP_0}{\sqrt{T_0}}\sqrt{\frac{\gamma}{R}}\frac{M}{\left\{1+((\gamma-1)/2)M^2\right\}^{(\gamma+1)/2(\gamma-1)}}$$

CONTD…

The effect of back pressure P_b *on the* mass flow rate and the exit pressure P_e.





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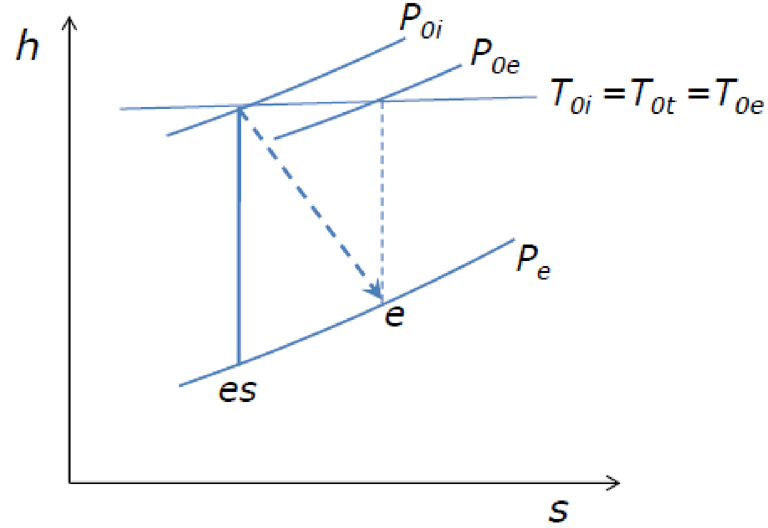

Isentropic flow converging nozzles

From the above figure,

$$P_e = \begin{cases} P_b & \text{for } P_b \geq P^* \\ P^* & \text{for } P_b < P^* \end{cases}$$

For all back pressures lower that the critical pressure, exit pressure = critical pressure, Mach number is unity and the mass flow rate is maximum (choked flow).

A back pressure lower than the critical pressure cannot be sensed in the nozzle upstream flow and does not affect the flow rate.

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CONTD…

The efficiency of a nozzle is defined as

$$\eta_n = \frac{h_{0i} - h_e}{h_{0i} - h_{es}}$$, where h_{0i} is the stagnation enthalpy at the nozzle inlet, h_e is the actual static enthalpy at the nozzle exit, h_{es} is the isentropic static enthalpy at the nozzle exit.

In terms of the corresponding temperatures,

$$\eta_n = \frac{T_{0i} - T_e}{T_{0i} - T_{es}} = \frac{1 - T_e / T_{0e}}{1 - T_{es} / T_{0i}}$$

For choked flow, $M = 1$,

$$\eta_n = \frac{1 - (2/(\gamma + 1))}{1 - (P_C / P_{0i})^{(\gamma-1)/\gamma}}$$

The pressure ratio is therefore,

$$\frac{P_{0i}}{P_C} = \frac{1}{(1 - (1/\eta_n)((\gamma - 1)/(\gamma + 1)))^{\gamma/(\gamma-1)}}$$

If $\frac{P_{0i}}{P_C} < \frac{P_{0i}}{P_a}$,

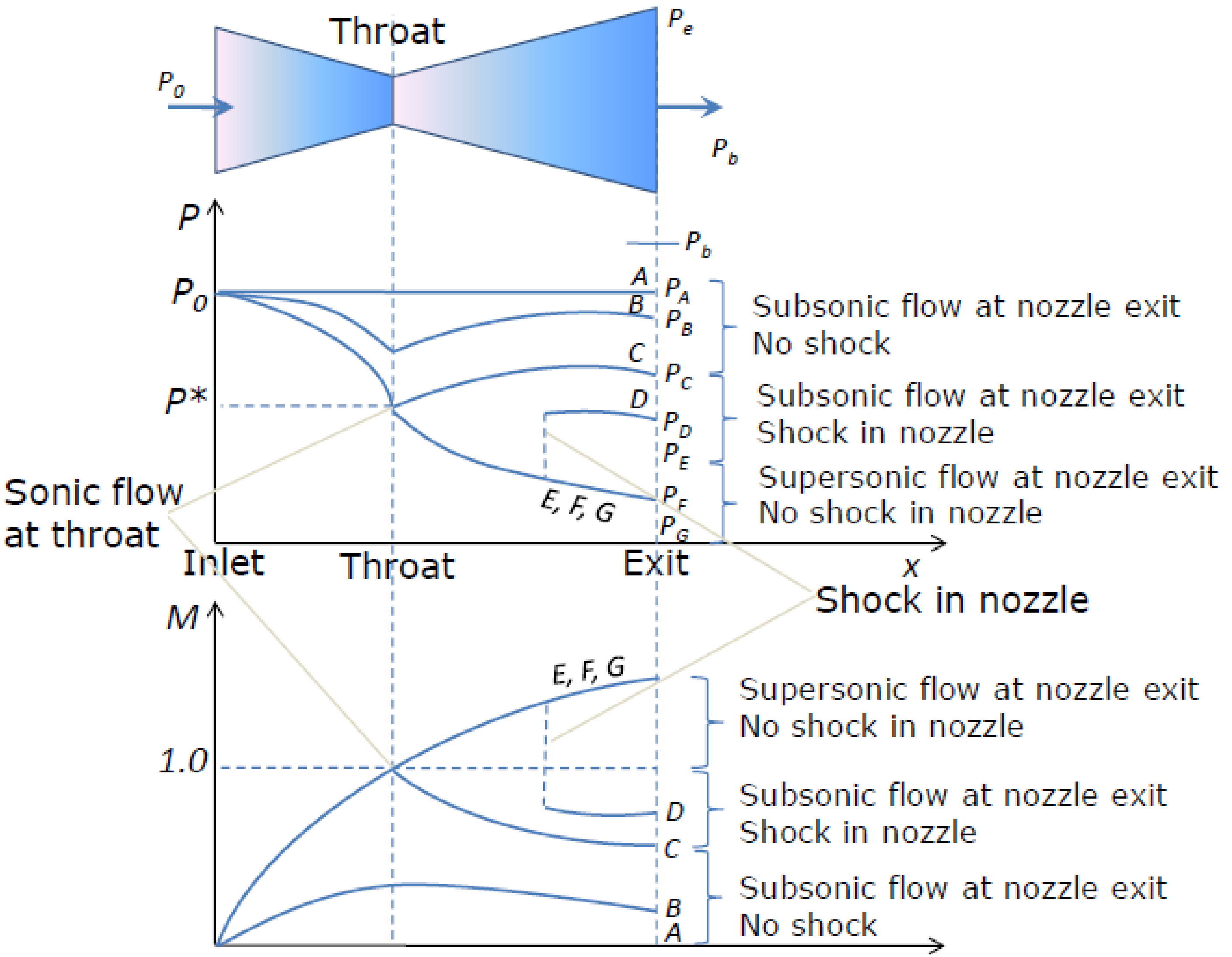
the nozzle is operating under choked condition.

- If a convergent nozzle is operating under choked condition, the exit Mach number is unity.
- The exit flow parameters are then defined by the critical parameters.
- To determine whether a nozzle is choked or not, we calculate the actual pressure ratio and then compare this with the critical pressure ratio.
- If the actual pressure ratio > critical pressure ratio, the nozzle is said to be choked.

CD NOZZLE

- Maximum Mach number achievable in a converging nozzle is unity.
- For supersonic Mach numbers, a diverging section after the throat is required.
- However, a diverging section alone would not guarantee a supersonic flow.
- The Mach number at the exit of the converging-diverging nozzle depends upon the back pressure.
- The flow through nozzles is normally assumed to be adiabatic as the heat transfer per unit mass is much smaller than the difference in enthalpy between the inlet and outlet.
- •The flow from the inlet to the throat can be assumed to be isentropic, but the flow from the throat to exit may not be due the possible presence of shocks.

CONTD…



Throat
P_0
P_e
P_b
P
P_0
$P*$
P_b
A
B
C
D
P_A
P_B
P_C
P_D
P_E
P_F
P_G
E, F, G
Subsonic flow at nozzle exit
No shock
Subsonic flow at nozzle exit
Shock in nozzle
Supersonic flow at nozzle exit
No shock in nozzle
Sonic flow
at throat
Inlet
Throat
Exit
x
Shock in nozzle
M
1.0
E, F, G
D
C
B
A
Supersonic flow at nozzle exit
No shock in nozzle
Subsonic flow at nozzle exit
Shock in nozzle
Subsonic flow at nozzle exit
No shock

CONTD…

The efficiency of a nozzle is defined as

$$\eta_n = \frac{h_{0i} - h_e}{h_{0i} - h_{es}} = \frac{T_{0i} - T_e}{T_{0i} - T_{es}} = \frac{1 - T_e / T_{0e}}{1 - T_{es} / T_{0i}}$$

$$= \frac{1 - (P_e / P_{0e})^{(\gamma-1)/\gamma}}{1 - (P_e / P_{0i})^{(\gamma-1)/\gamma}}$$

Therefore, $\left(\dfrac{P_e}{P_{0e}}\right) = \left[1 - \eta_n\left\{1 - \left(\dfrac{P_e}{P_{0i}}\right)^{(\gamma-1)/\gamma}\right\}\right]^{\gamma/(\gamma-1)}$

Since, $\dfrac{P_{0i}}{P_{0e}} = \dfrac{P_e}{P_{0e}}\dfrac{P_{0i}}{P_e} \Rightarrow \dfrac{P_{0i}}{P_{0e}} = \dfrac{P_{0i}}{P_e}\left[1 - \eta_n\left\{1 - \left(\dfrac{P_e}{P_{0i}}\right)^{(\gamma-1)/\gamma}\right\}\right]^{\gamma/(\gamma-1)}$

The exit velocity can be calculated from

$$u_e = \sqrt{2(h_{0i} - h_e)} = \sqrt{2\eta_n(h_{0i} - h_{es})}$$

$$= \sqrt{2c_p\eta_n(T_{0i} - T_{es})} = \sqrt{2c_p\eta_n T_{0i}\left\{1 - \left(\frac{P_e}{P_{0i}}\right)^{(\gamma-1)/\gamma}\right\}}$$

$$= \sqrt{\frac{2\gamma R}{(\gamma - 1)}\eta_n T_{0i}\left\{1 - \left(\frac{P_e}{P_{0i}}\right)^{(\gamma-1)/\gamma}\right\}}$$

CONTD…

- Assuming isentropic flow up to throat area, relation between throat and exit area is

$$\frac{A_t}{A_e} = \frac{P_{0e}}{P_{ot}}\frac{M_e}{M_t}\left[\frac{1+((\gamma-1)/2)M_t^2}{1+((\gamma-1)/2)M_e^2}\right]^{(\gamma+1/2(\gamma-1))}$$

$$= \frac{P_{0e}}{P_{0i}}\frac{M_e}{M_t}\left[\frac{1+((\gamma-1)/2)M_t^2}{1+((\gamma-1)/2)M_e^2}\right]^{(\gamma+1/2(\gamma-1))}$$

If the throat is choked, $M_t = 1$,

$$\frac{A^*}{A_e} = \frac{P_{0e}M_e}{P_{0i}^*}\left[\frac{(\gamma+1)/2)}{1+((\gamma-1)/2)M_e^2}\right]^{(\gamma+1/2(\gamma-1))}$$

CONTD…

The mass flow rate will therefore be,

$$\dot{m} = \frac{A^{*}P^{*}_{0i}}{\sqrt{T^{*}_{0i}}}\sqrt{\frac{\gamma}{R}}\frac{1}{((\gamma+1)/2)^{(\gamma+1)/2(\gamma-1)}}$$

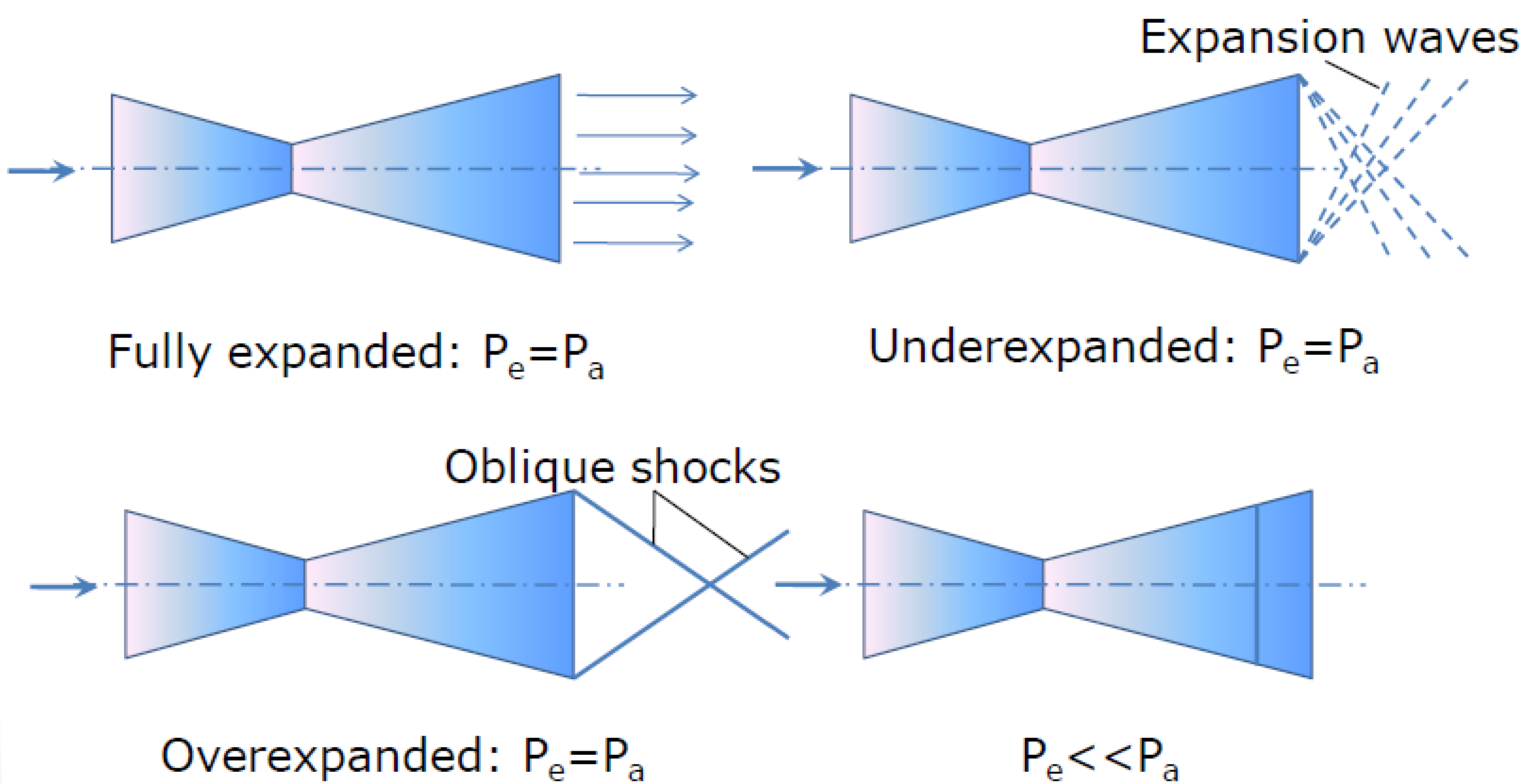
The mass flow rate is a function of the inlet stagnation pressure, temperature and throat area.

By design one would like to keep the area ratio A_i/A_e as close as possible to unity. This is to keep the external drag under control. However this may result in the nozzle exit pressure to be different from the ambient pressure : incomplete expansion.

CONTD…

- Underexpanded nozzle:
  - $P_e > P_a$
  - The flow is capable of additional expansion.
  - Expansion waves originating from the lip of the nozzle.
- Overexpanded nozzle:
  - $P_e < P_a$
  - Shock waves originate from the nozzle
- Fully expanded nozzle:
  - $P_e = P_a$
  - No shock waves/expansion waves.
- If $P_e << P_a$
  - Shock waves will occur within the divergent section of the nozzle.

CONTD…

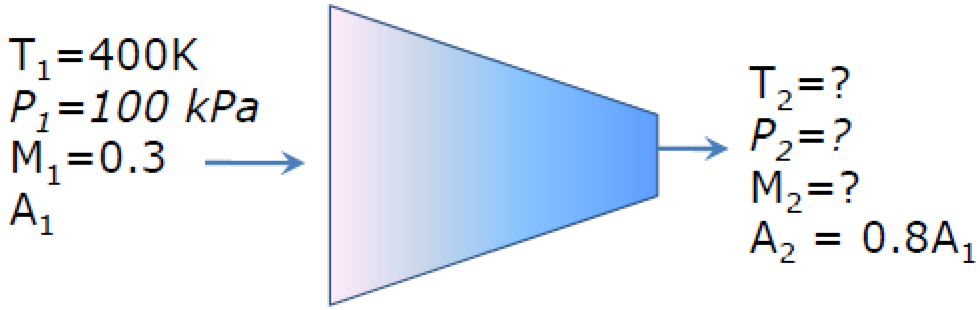


Expansion waves
Fully expanded: $P_e = P_a$
Underexpanded: $P_e = P_a$
Oblique shocks
Overexpanded: $P_e = P_a$
$P_e << P_a$

CONTD…

Numerical 1:

- 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.



CONTD…

Solution 1:

- 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 $M_2=0.391$

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

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

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

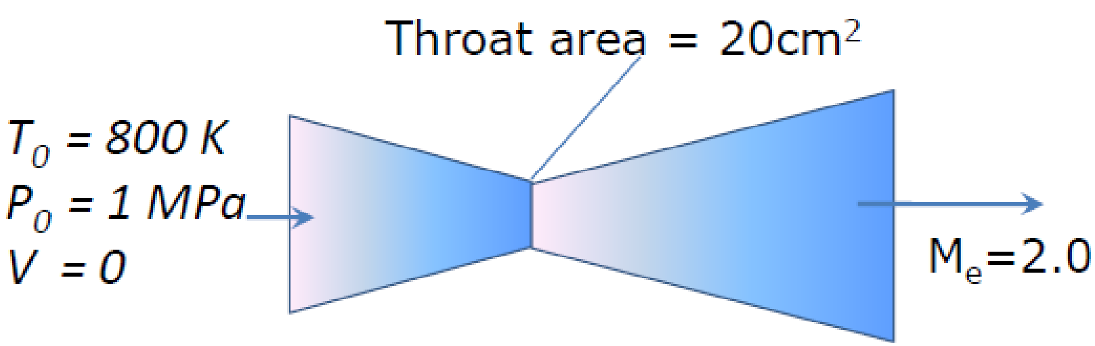
$$P_2 = 95.7 \text{ kPa}$$

- The static temp and temp drop in flow through a conversing Nozzle.
- There is an increase in Mach number.

CONTD…

Numerical 2:

- 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 cm^2, determine (a) the throat conditions, (b) the exit plane conditions, including the exit area, and (c) the mass flow rate through the nozzle.



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Solution 2:

- 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.

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

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

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

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

$$T* = 0.8333T_0 = 666.6 K$$

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

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

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(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,

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

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

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

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

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

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

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

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

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

- This corresponds to the highest possible mass flow rate through the nozzle: choking mass flow rate

THANK YOU !!!