AE 658

Aerodynamics of

Intakes and Propelling Nozzles

Theory and Design

Requirements made on the inlets and nozzles:

On Both

1. Minimize installation drag on their own and adjacent aircraft surfaces over wide range of aircraft angle of attack and engine throttle settings
2. Maximize internal total pressure recovery (π_d, π_n) of Intake and nozzle individually under various operating conditions
3. Provide ***controllable mass flow*** matching with other engine components for all operating conditions
4. Minimize weight and cost while meeting life and reliability goals
5. Suppress acoustic and radar signatures

Intakes

- **Control inlet spatial and temporal distortion**
- **Provide good starting and stability characteristics**

Nozzles

- **Maximize energy conversion**
- **Suppress infrared (IR) signatures (military aircraft)**
- **Provide thrust reversing and vectoring as may be required (military and civil aircraft)**

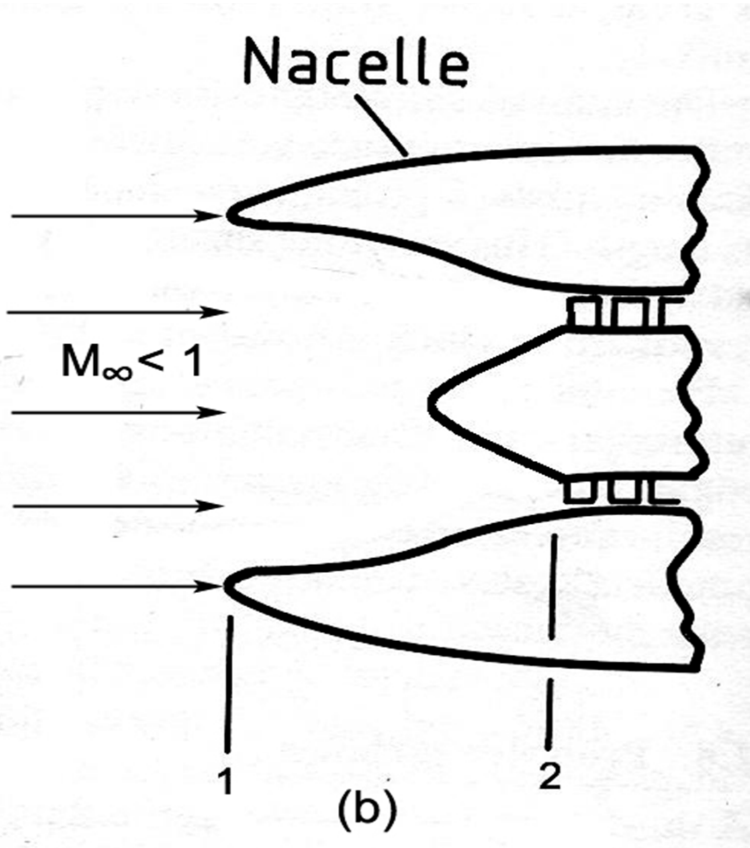
Many types of intakes and nozzles have been developed to meet these requirements

INTAKES

- The basic job of an intake of a jet engine is to (i) deliver the required mass flow at the fan/ compressor face (entry), with (ii) the highest possible stagnation pressure, with the (iii) most uniform velocity distribution possible.

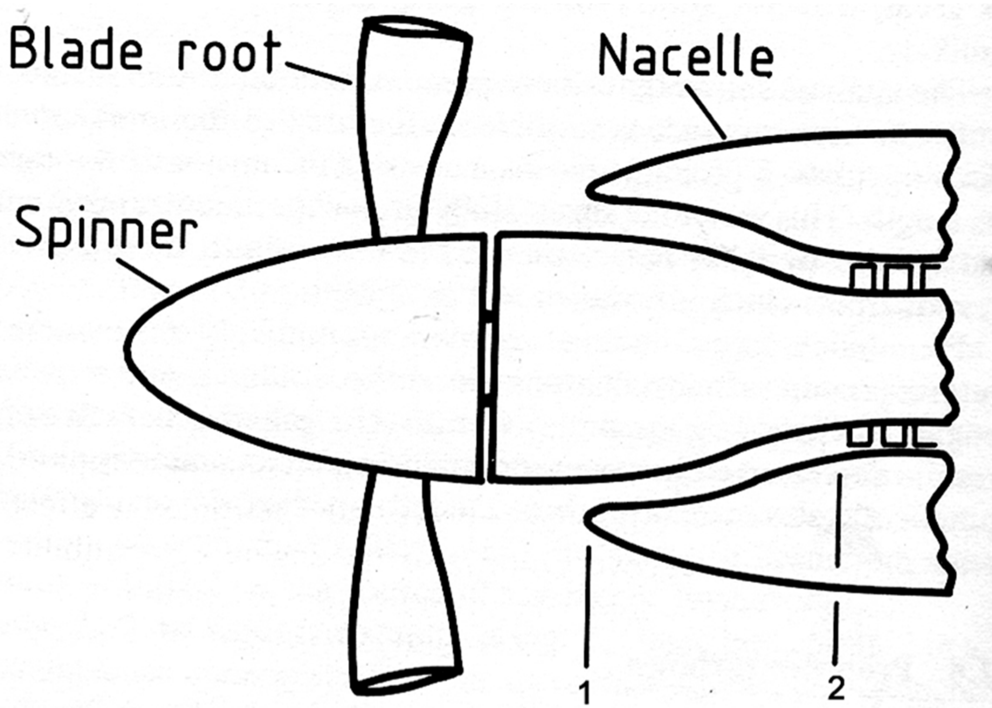
- Any loss of stagnation pressure in the intake would be reflected on the whole engine performance.

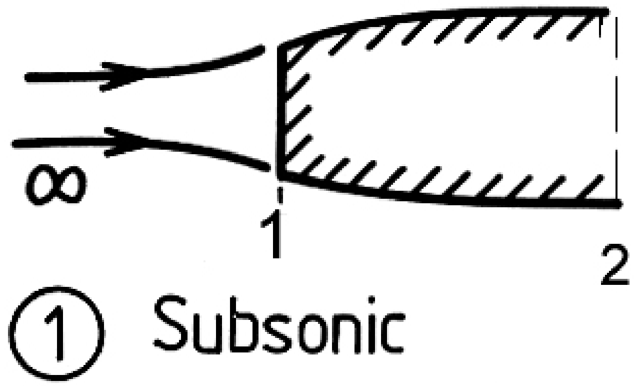
- Similarly any loss of uniformity in flow would be strongly reflected in the fan / compressor performance.



A simple ‘pitot intake’ is shown in Fig.

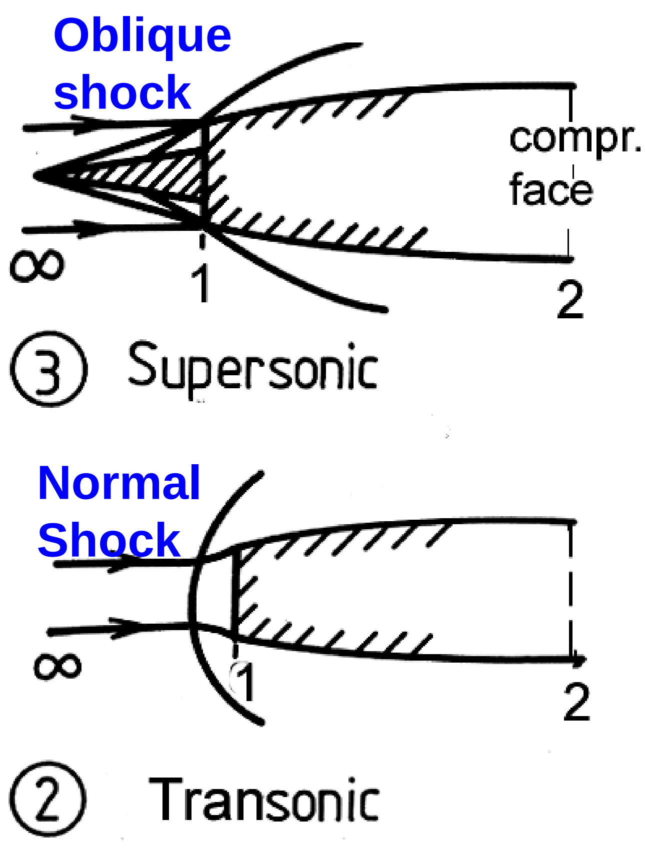
As shown in fig for turbo-prop aircraft intake air is brought in through the roots of the propeller. Thus the air suffers from stagnation pressure loss and significant loss of uniformity at the pre-entry stage.





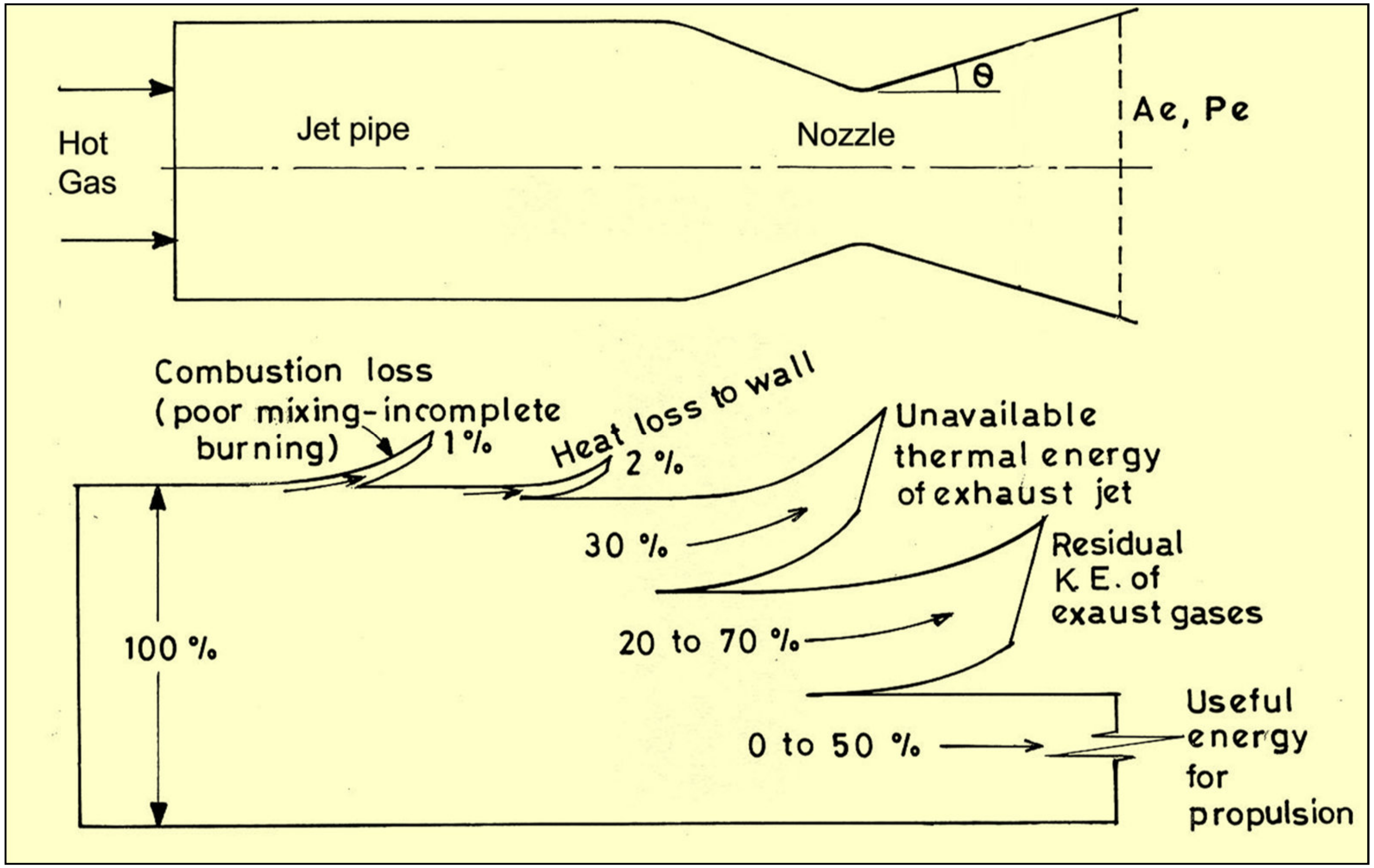
For almost all modern jet engines at the cruise flight condition, the velocity requirement at the compressor face is subsonic and in all probability be less than the aircraft flight velocity. Thus, some amount of diffusion is inevitable in the intake. Determination of the flow conditions and the intake dimension from its entry face to the compressor delivery face forms the main task of the intake designer.

An aerodynamics based optimization between various flight conditions is often required to arrive at the intake nacelle shape.

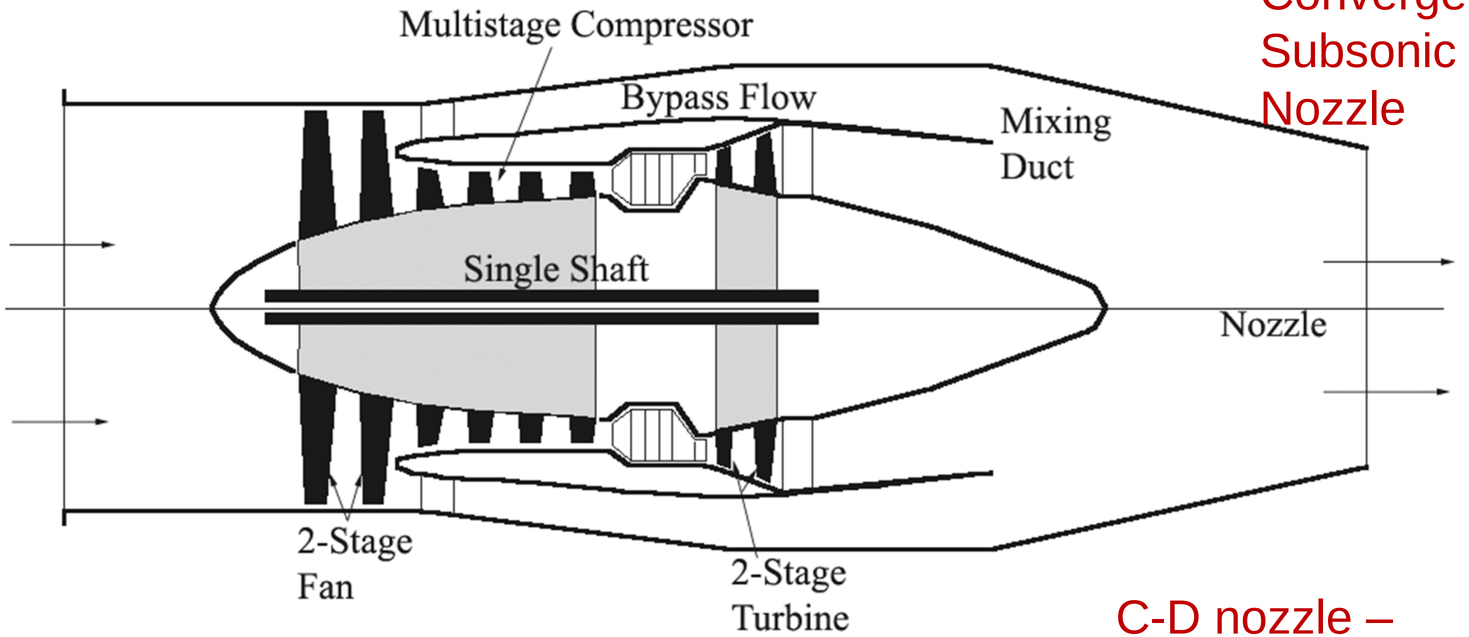


Supersonic aircraft engines are reconciled to pay high intake momentum drag caused by high flight speed. On the other hand it can benefit from high ram pressure development through a well designed and efficient intake. If the external free stream conditions are supersonic and the compressor face requirements are subsonic then deceleration through shocks is inevitable.

NOZZLES

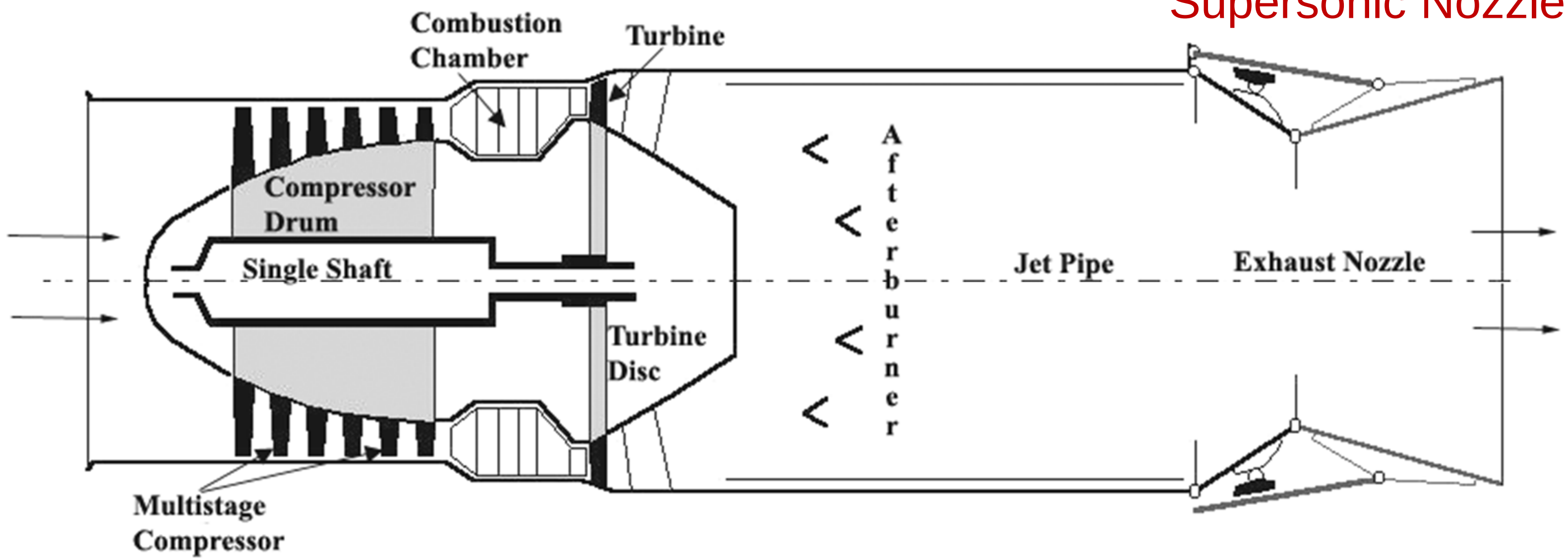


Gas coming out of turbines retains sufficiently high temperature and pressure. Approximately 40% of the total energy input in a gas turbine based powerplant is available for conversion to useful thrust/power for propelling an aircraft. Rest of the energy is used by the compressor-turbine loop. Out of the 40% available at the nozzle only 50% of it is useful for setting up a momentum change across the powerplant. The rest is normally lost completely without any chance of recovery.



Convergent
Subsonic
Nozzle

C-D nozzle –
Supersonic Nozzle



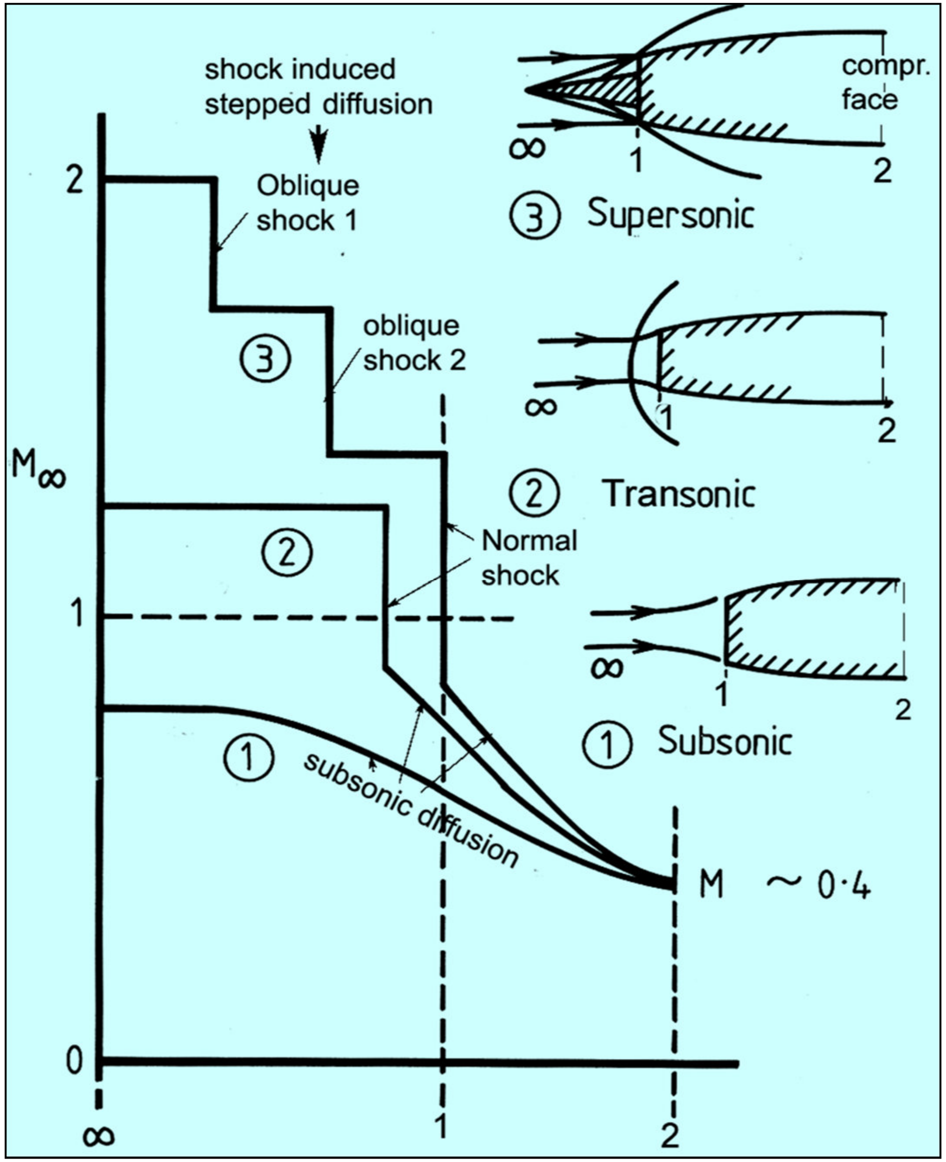
INTAKES

- The basic requirements are necessary to be met over a wide range of engine operating conditions, during aircraft and/or engine acceleration or deceleration.

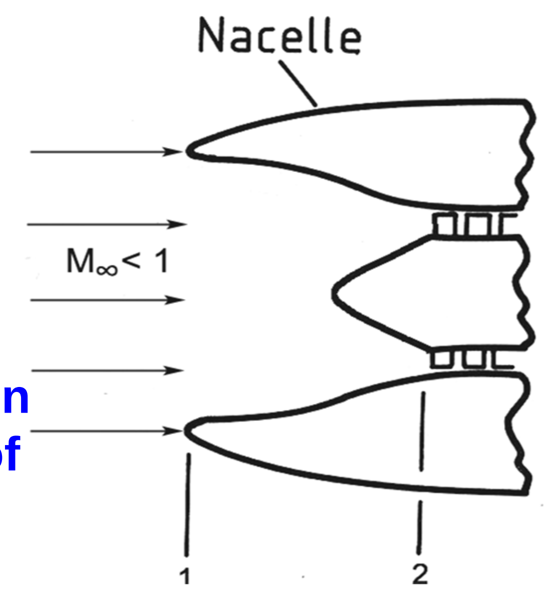
- Such requirements call for <u>variable geometry intakes</u>.

- In order to avoid mechanical complexity and weight gain <u>most jet engines intakes are fixed geometry</u>.

- For supersonic aircraft all the above tasks are to be met for both subsonic and supersonic aircraft speeds.



Subsonic Intakes



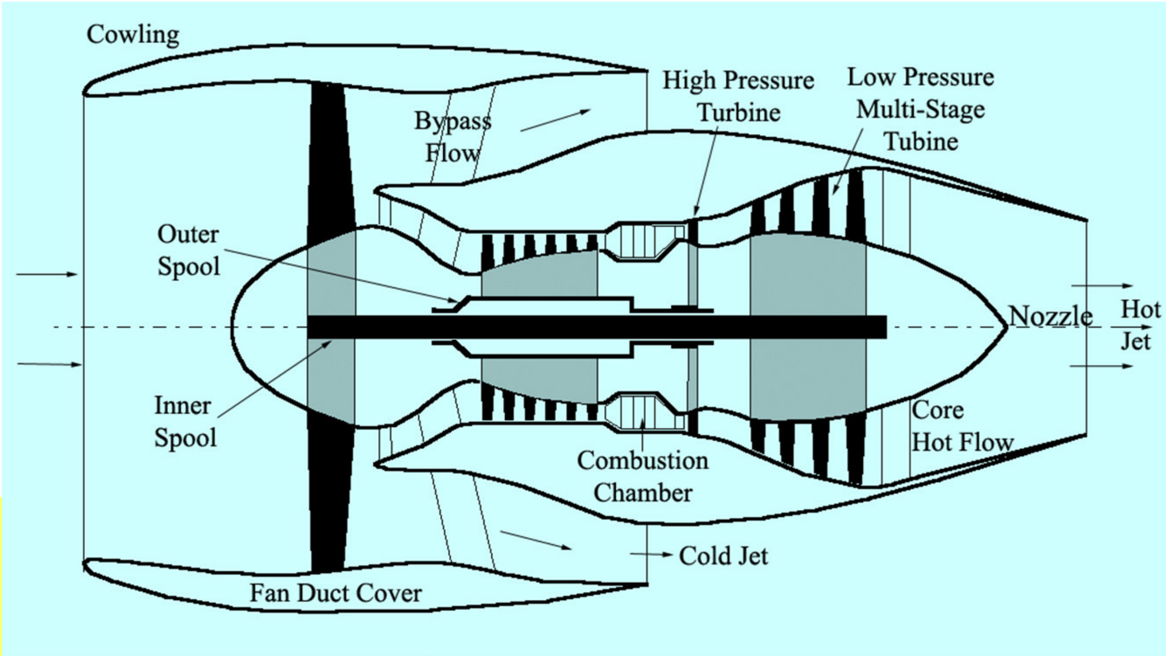
- Determination of the flow conditions and the intake dimension from its entry face to the compressor delivery face forms the main task of the intake designer.

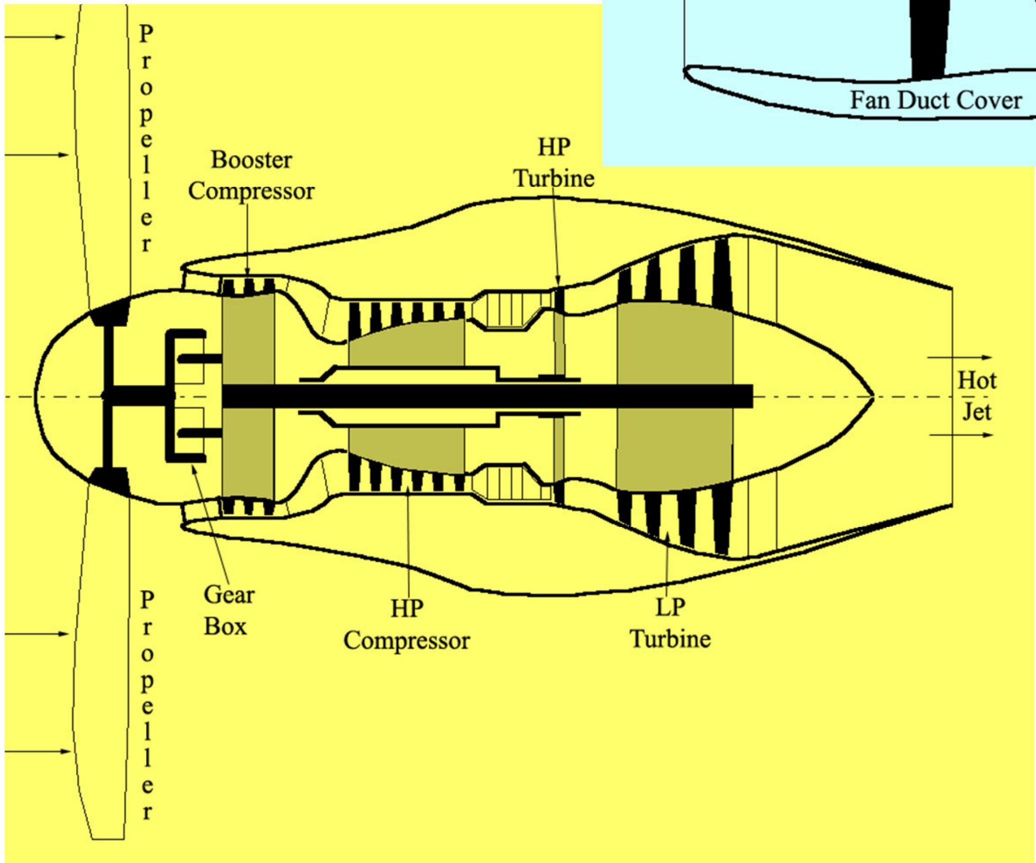
- At cruise condition, where it operates for the maximum period of time, the intake entry area is chosen such that the engine air is contained in a stream tube of the same cross section as that of the entry face of the intake (Fig.).

- If the air flow requirement is less than that in the "design" stream tube (in case of engine deceleration), the excess air is 'spilled' over the nacelle outer surface resulting in increase in outer body drag.

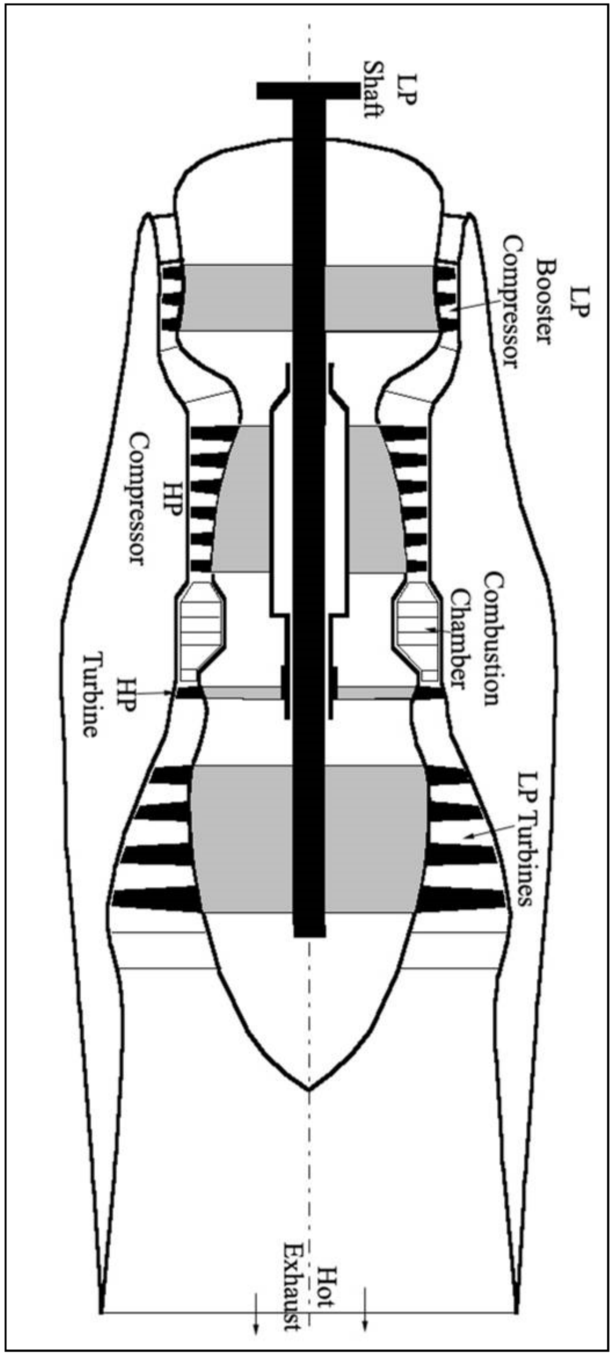
- During engine acceleration and aircraft take off the required mass flow is more than what is supplied by the 'design' stream tube.
- This ideally calls for a bell mouth shaped intake.
- Since that is prohibitive from drag-at-cruise point of view, most intake designers make the intake forward lip more rounded to meet these demands.
- An aerodynamics based optimization between various flight conditions is often required to arrive at the intake nacelle shape.

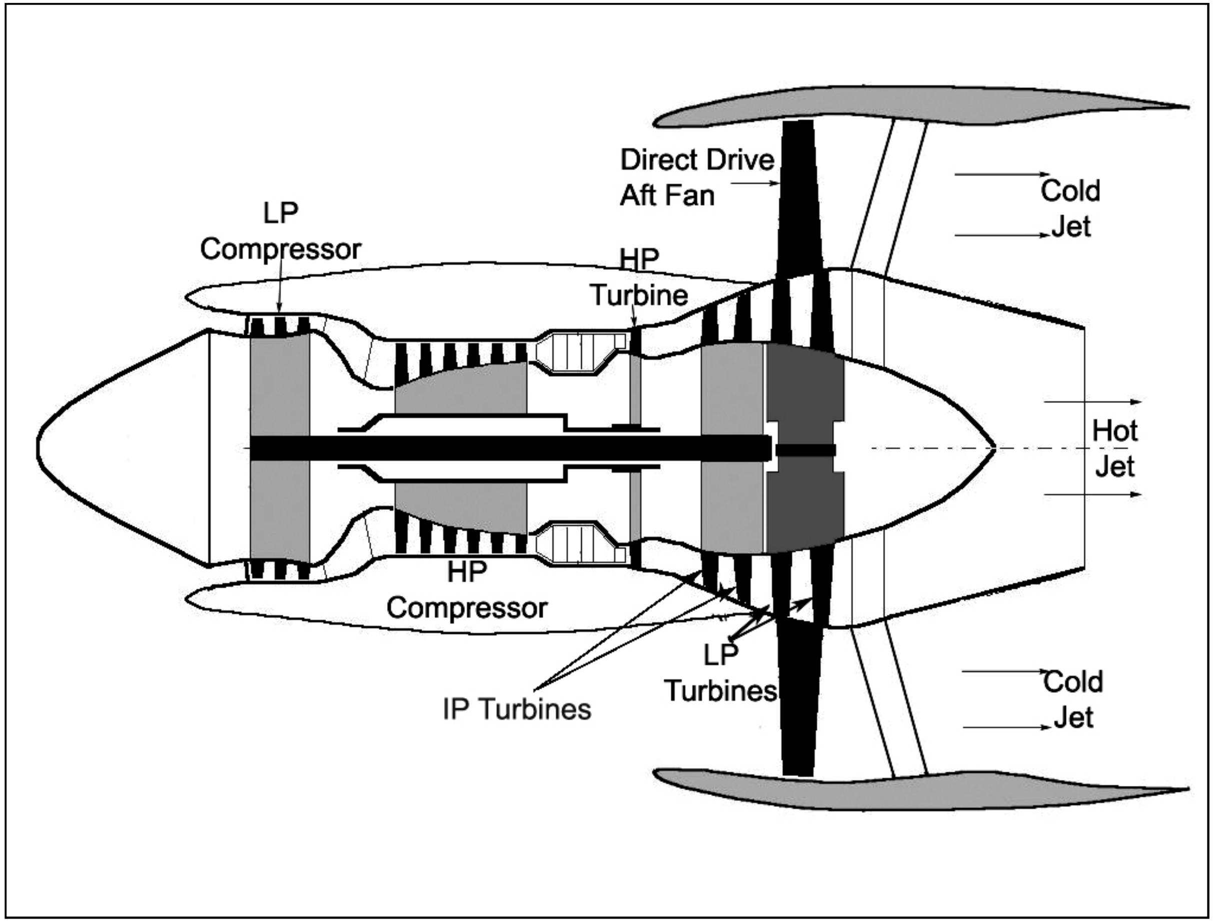
Subsonic Intakes





Subsonic Intakes





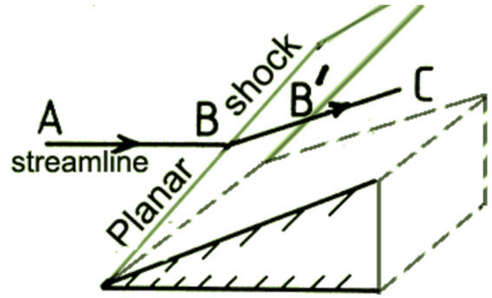
Supersonic Intakes

Supersonic Intakes

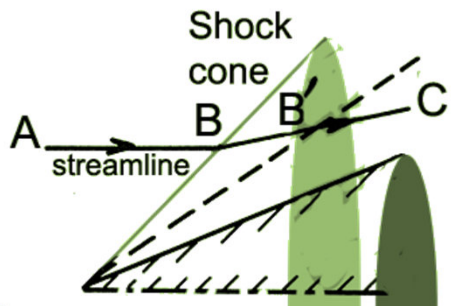
• A rectangular intake mouth with a wedge shaped fore body with sharp forward lip creates a planar oblique shock (Fig. a) that conforms to the 2-D shock theories.

• While flow prediction through such a shock system is easier this also means that the flow would require to be converted to an axi-symmetric annular flow before being delivered to the compressor / fan.

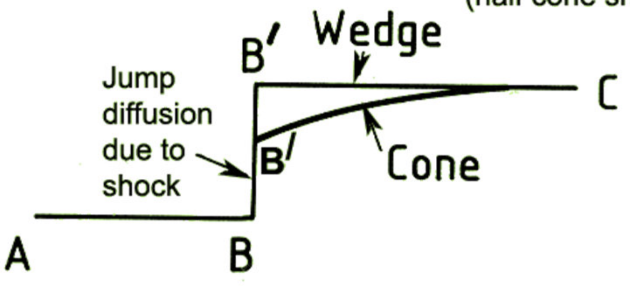
• A cone placed at the center of the intake and extended forward ahead of the intake mouth creates a (oblique) shock cone (Fig. b).



(a) Supersonic flow on Wedge



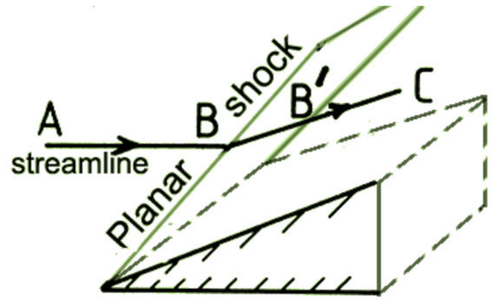
(b) Supersonic flow on Cone
(half cone shown)



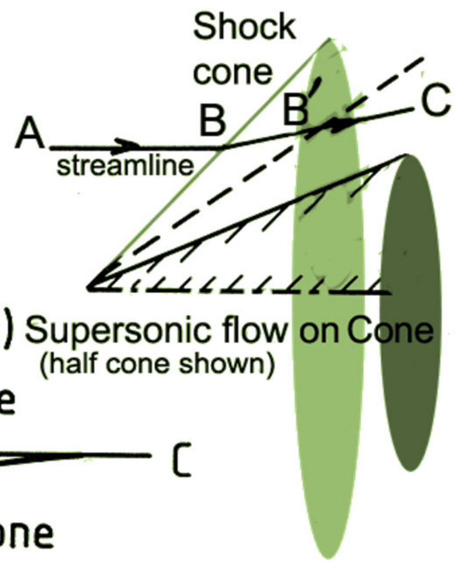
(c) Pressure variation axially along the wedge / cone

• **Transition through these shock systems (ABB/C) result in supersonic decelerations. In case of wedge (2-D shock) the deceleration is in one jump (Fig. a & c).**

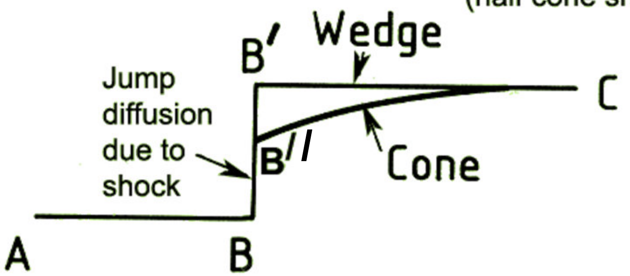
• **However, in case of a shock cone the deceleration is in a half–jump followed by steady supersonic (ABB//C) deceleration (Fig. b & c).**



(a) Supersonic flow on Wedge



(b) Supersonic flow on Cone
(half cone shown)



(c) Pressure variation axially along the wedge / cone

Supersonic Intakes

- **Supersonic aircraft engines are reconciled to pay high intake momentum drag caused by high flight speed.**
- **It can also benefit from high ram pressure development through a well designed and efficient intake.**
- **If the external free stream conditions are supersonic and the compressor face requirements are subsonic then deceleration through shocks is inevitable.**
- **At low supersonic transonic flight conditions a simple pitot type intake (slide 10 case 2) produces bow shock in front of the intake face.**
- **At higher flight mach numbers such a shock would entail high stagnation pressure loss through the shocks.**
- **To maximize the pressure recovery the flow is often taken through one or more oblique shocks before being made subsonic through a normal shock. (slide 10 case 3).**

Supersonic Intakes

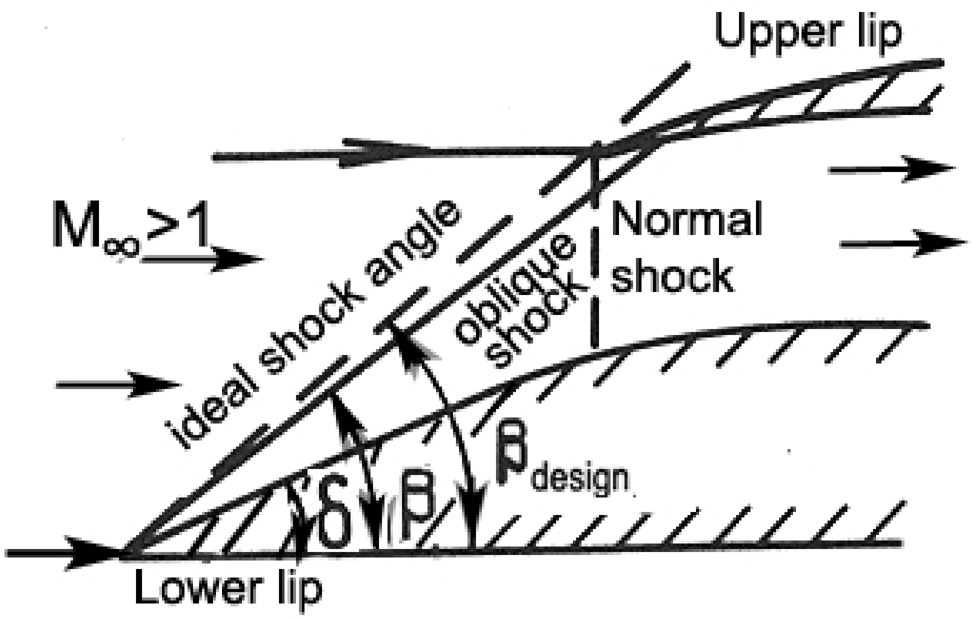
The oblique shocks are created through sharp leading edged forward nose or lip of the intake. The shape of the lip then becomes the focus of attention of the intake designer. The shock system created often typifies an intake. It is logical to go for multiple shock system as the flight Mach number keeps going up. The intake shape shall also decide whether the shocks are created outside the body of the intake (*external compression intake*) or whether they are contained within the body of the intake (*internal compression intake*). In many cases it may be a combination of the two above cases (*mixed internal and external compression intake*).

Some intake length would have to be dedicated to the subsonic diffusion prior to the compressor face.

The changes in static and total pressures and Mach numbers through the normal shocks can be estimated with the help of normal shock theories available through look up tables and / or graphs. Similar changes through oblique shocks can also be computed using shock theories.

However, as stated earlier an intake has to cater to varying operating conditions. It is quite impossible to ensure, with a fixed geometry fore body, that the shocks thus created are indeed safely anchored between the lower and the upper lips. Rise or fall in operating Mach number from the ‘design’ Mach number shall push the shock inward or outward respectively. This would result in either the shock getting swallowed in the engine, or pushed out for ‘standing off’ in front of the engine intake. Both the conditions tend to destabilize the flow in the engine. However, shock swallowing is far more dangerous and could result in engine blow up. As a result most designs cater to some amount of shock stand off under various off-design conditions. This means that most intakes are designed at maximum flight Mach number

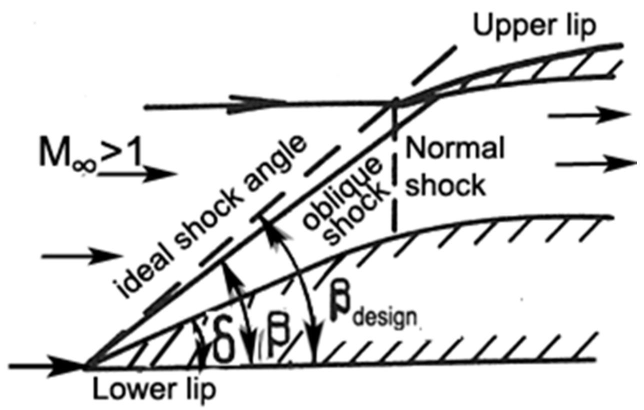
By suitable selection of the solid body wedge or cone angle, the shock can be made to meet the upper lip of the intake (Fig.). The terminal normal shock is often placed at the immediate downstream station to terminate supersonic flow.

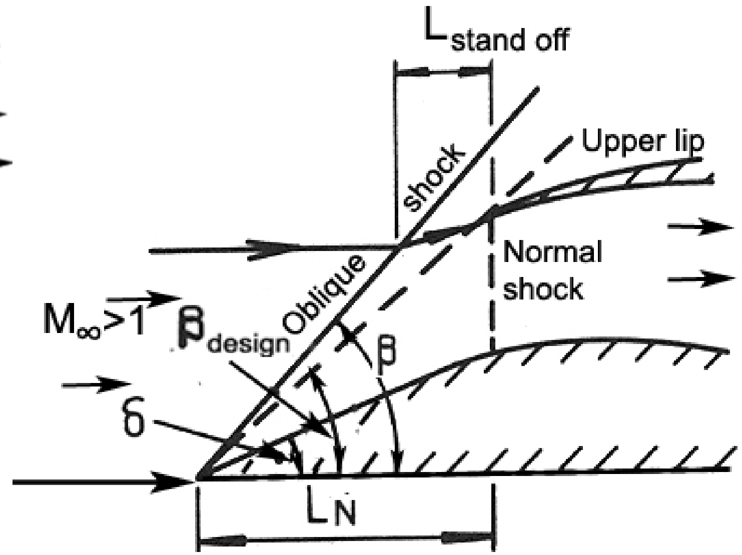


A series of low intensity shocks can be ideally created with a curved intake forward body (both wedge or cone) which could deliver flow at negligible or no pressure loss. Such a fore body is referred to as an "*isentropic spike*".

However, an intake has to cater to varying operating conditions. It is quite impossible to ensure, with a fixed geometry forebody, that the shocks thus created are indeed safely anchored between the lower and the upper lips.

Fig. illustrates that rise or fall in operating Mach number from the ‘design’ Mach number shall push the shock inward or outward respectively. This would result in either the shock getting swallowed in the engine, or pushed out for ‘standing off’ in front of the engine intake. Both the conditions tend to destabilize the flow in the engine. However, shock swallowing is far more dangerous and could result in engine blow up. As a result most designs cater to some amount of shock stand off under various off-design conditions. Most intakes are designed at maximum flight Mach number

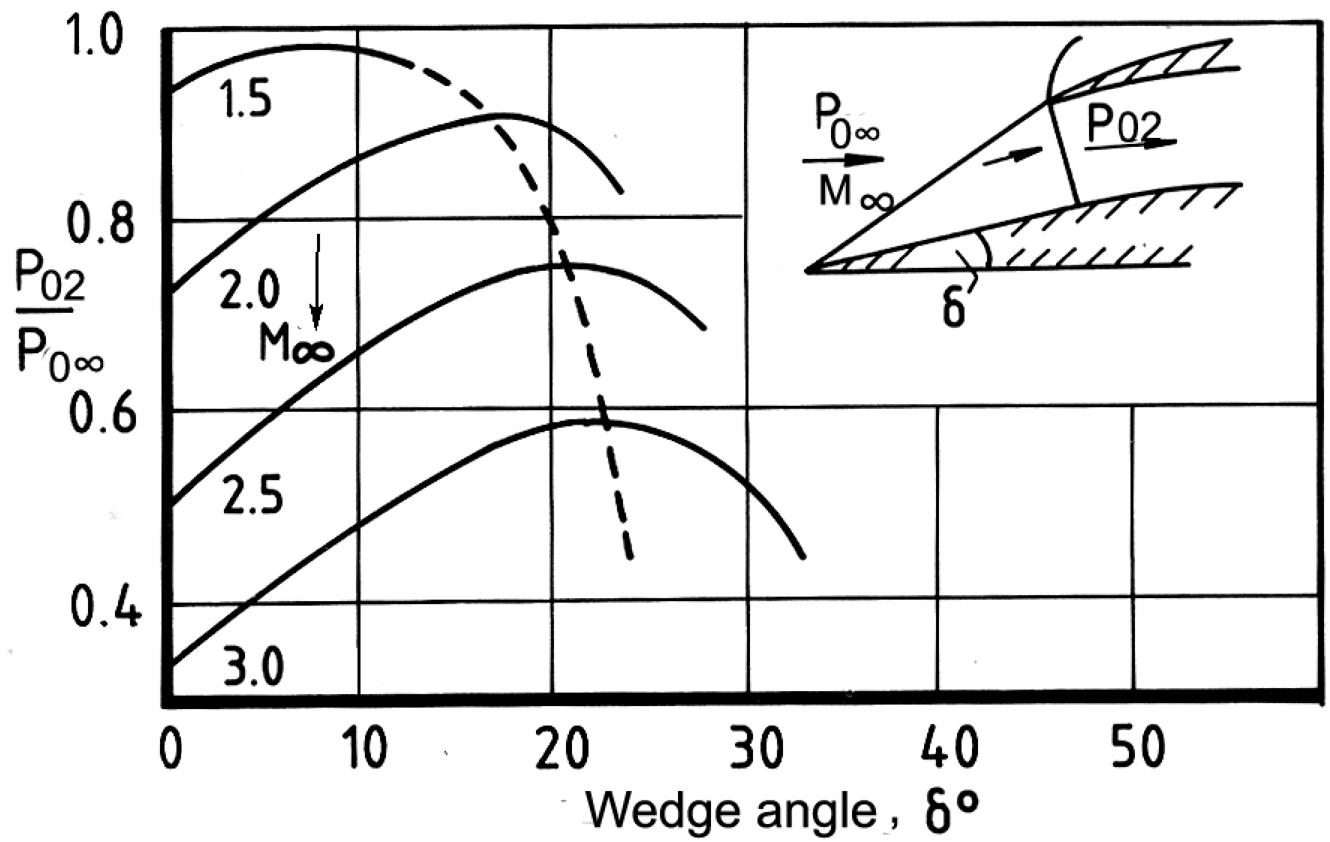




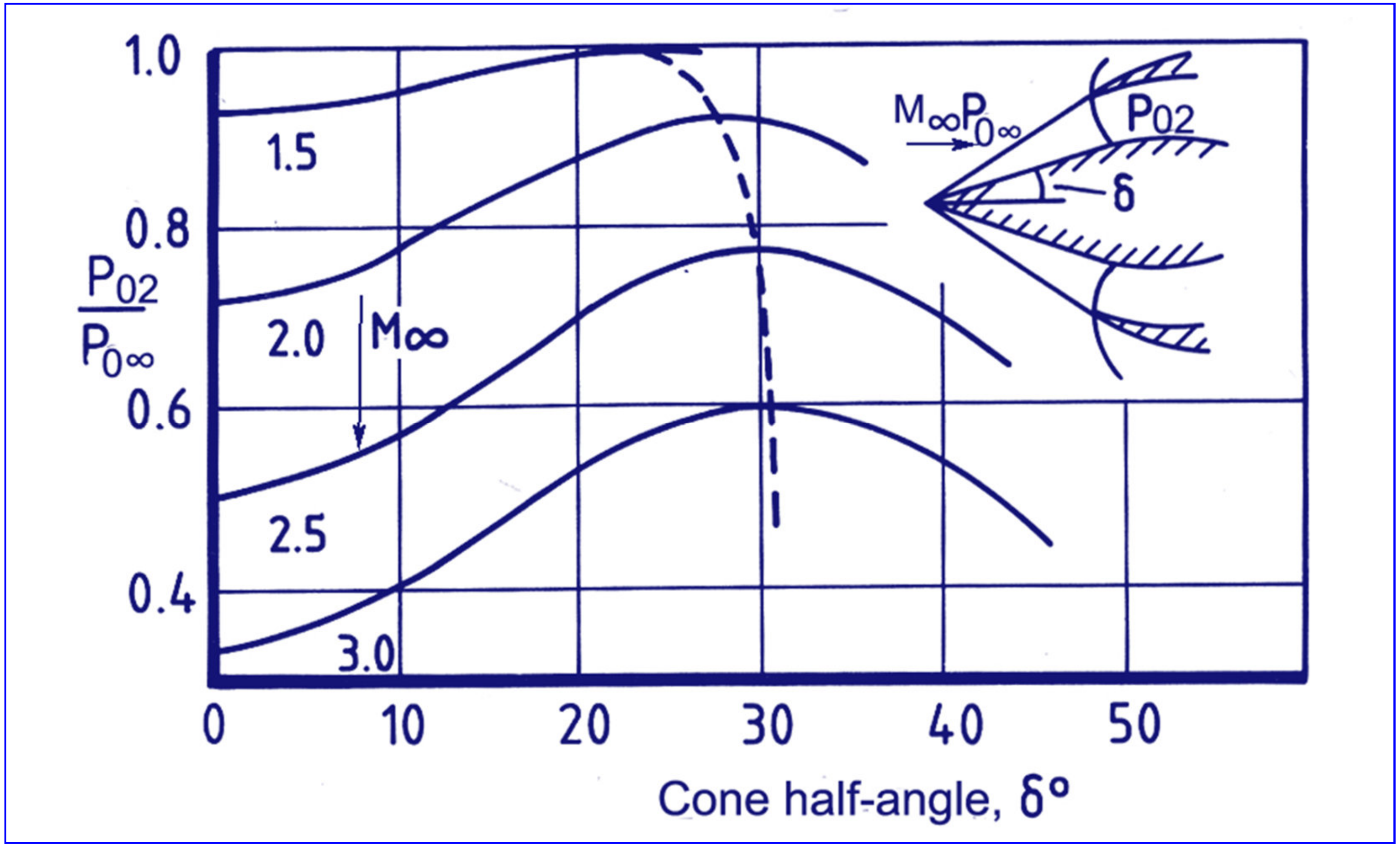
The flow going into the intake as shown in Fig., where the oblique shock is standing off at a distance $L_{stand\ off}$, some of the flow in between the oblique shock surface and the intake face may be deflected outwards on all sides.

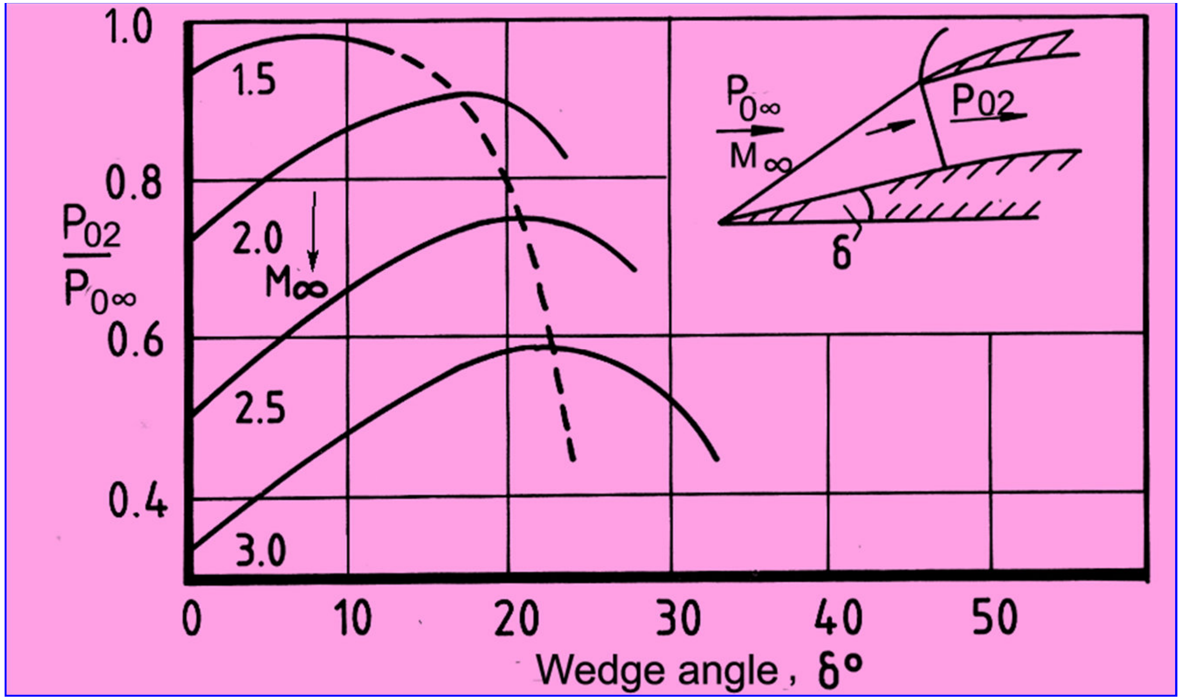
Such a spillage gives rise to aerodynamic noise, some times called ‘buzz’.

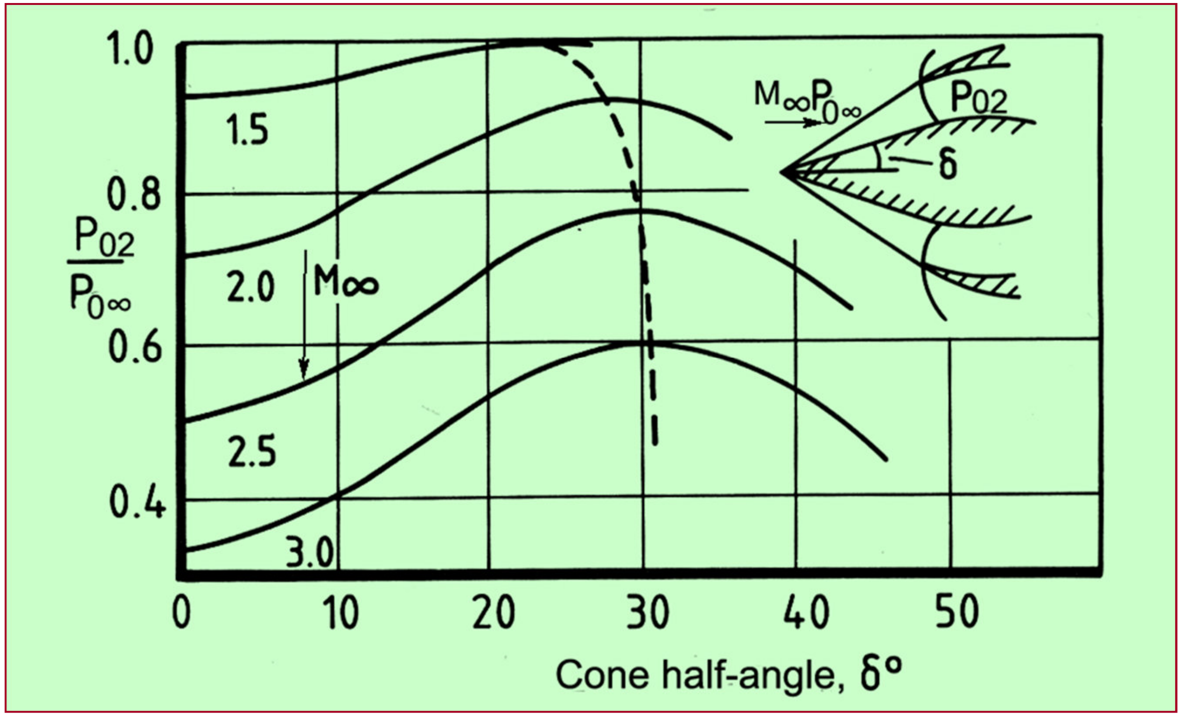
When the Intake entry Mach number rises, jump deceleration through a single shock shall require higher solid body angle, δ. In case of a wedge shaped body pressure recovery falls off after about 20^0 angle.



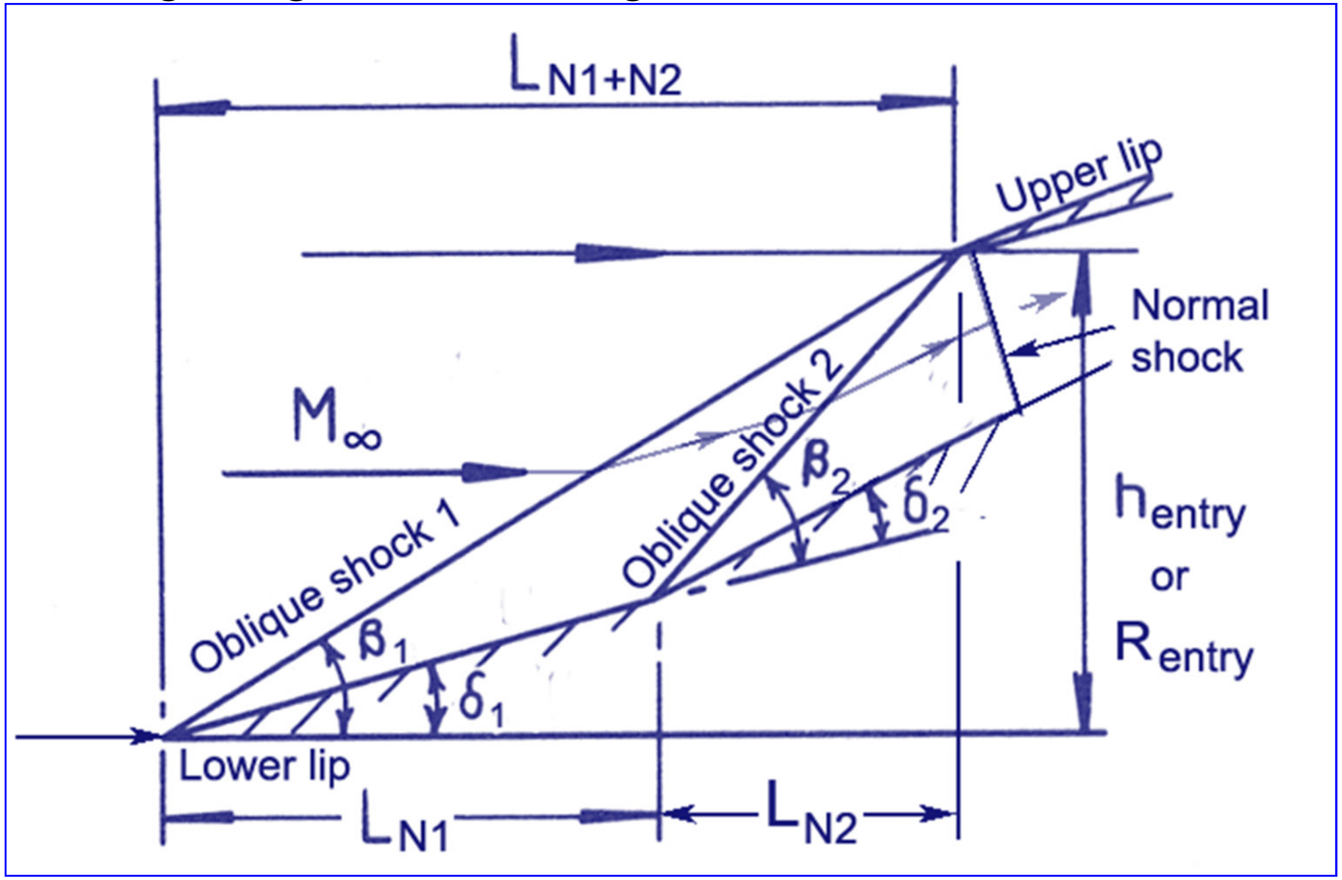
The cones continue to deliver with reasonable efficiency up to about 300. However pressure recovery of the cones is lesser at the lower solid body angles.

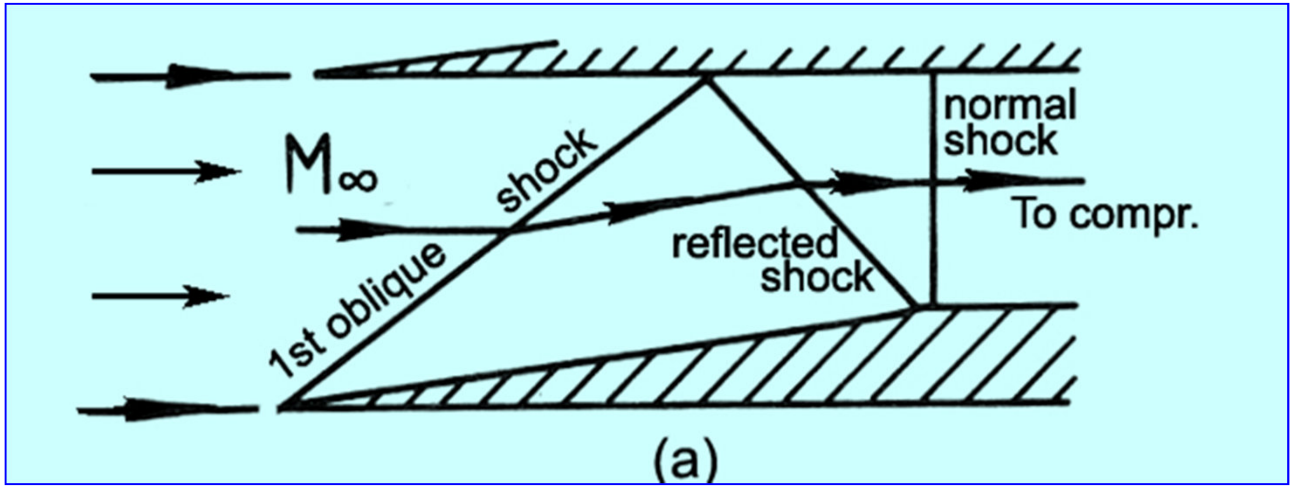


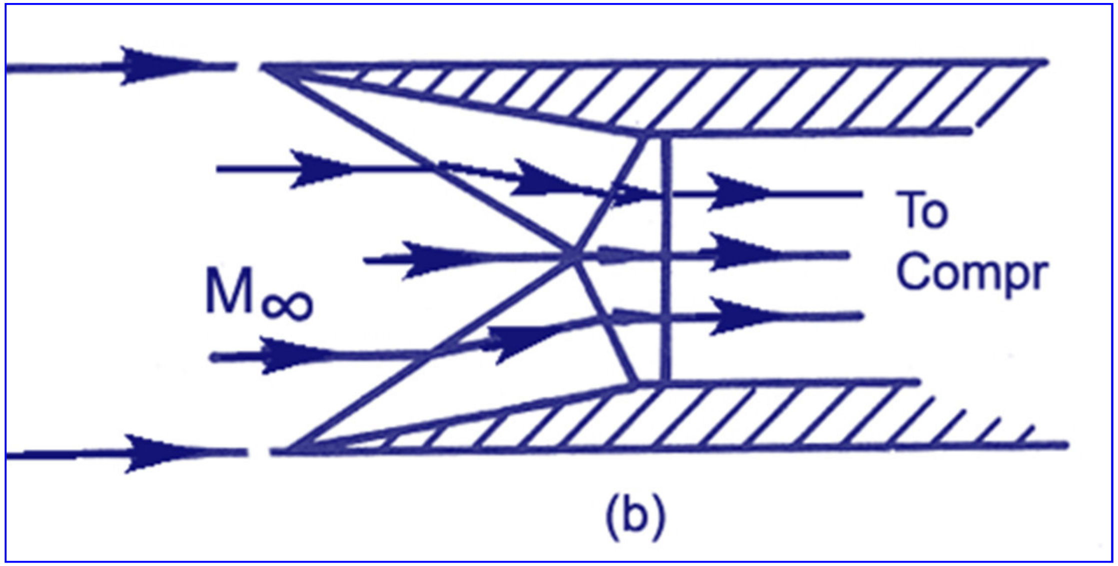

1.0
0.8
0.6
0.4
$\frac{P_{02}}{P_{0\infty}}$
1.5
2.0
2.5
3.0
M_∞
$P_{0\infty}$
M_∞
P_{02}
δ
0
10
20
30
40
50
Wedge angle , δ°


1.0
0.8
0.6
0.4
$\frac{P_{02}}{P_{0\infty}}$
1.5
2.0
2.5
3.0
M_∞
$M_\infty P_{0\infty}$
P_{02}
δ
0
10
20
30
40
50
Cone half-angle, δ°

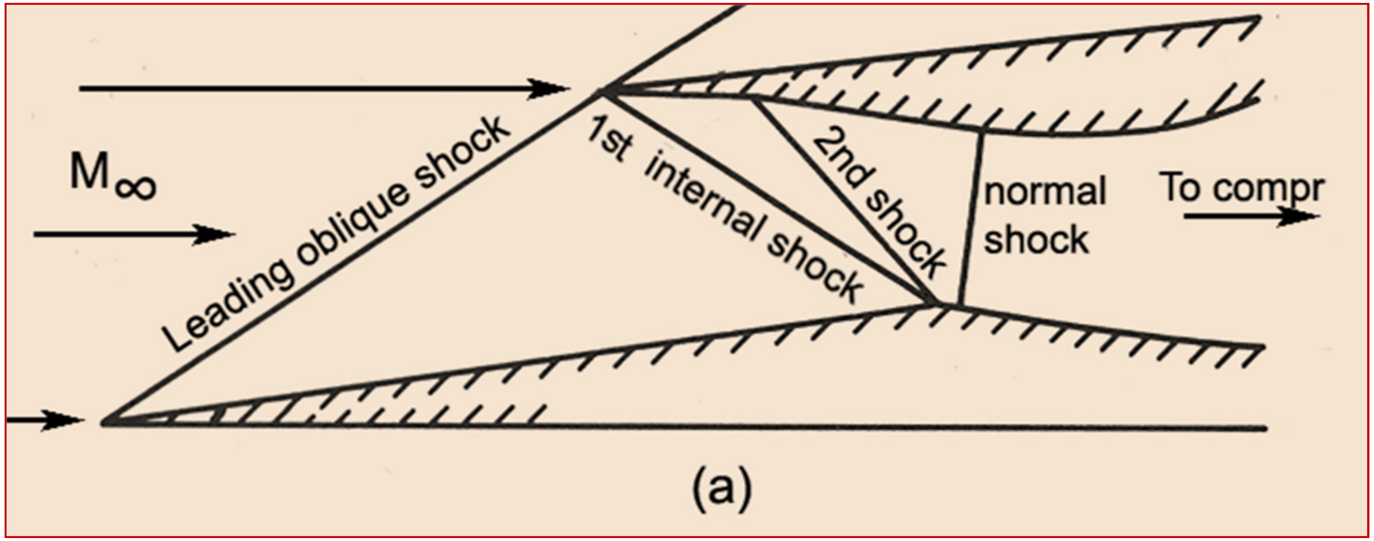
In spite of a slightly lower loss characteristics of conical forebody intakes more and more of the modern intakes are using wedge type or two-dimensional (rectangular cross section) fore bodies for intakes (Fig.). Such intakes with two oblique shocks and one terminal shock has become the most used intake configuration. The actual cross sectional shape is often morphed to cater to three dimensional flow features, including those related to high angle of attack flight conditions.

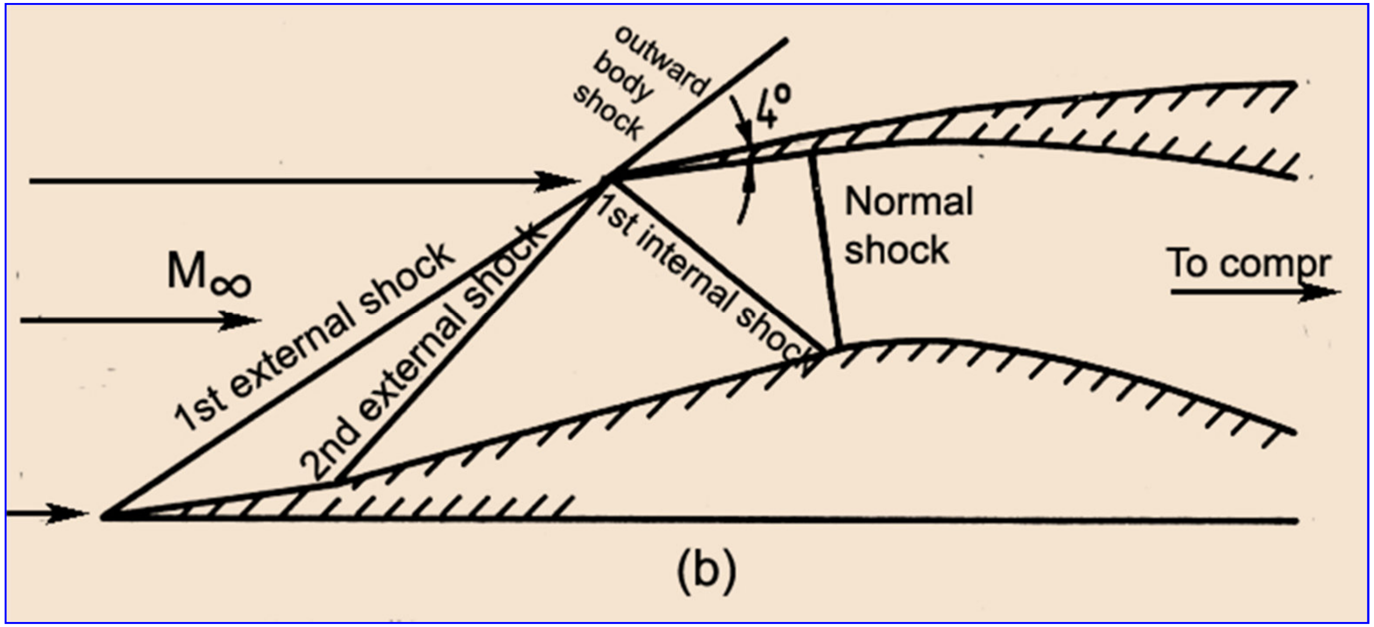






Internal shock systems in (a) asymmetric intake and (b) axi-symmetric intake





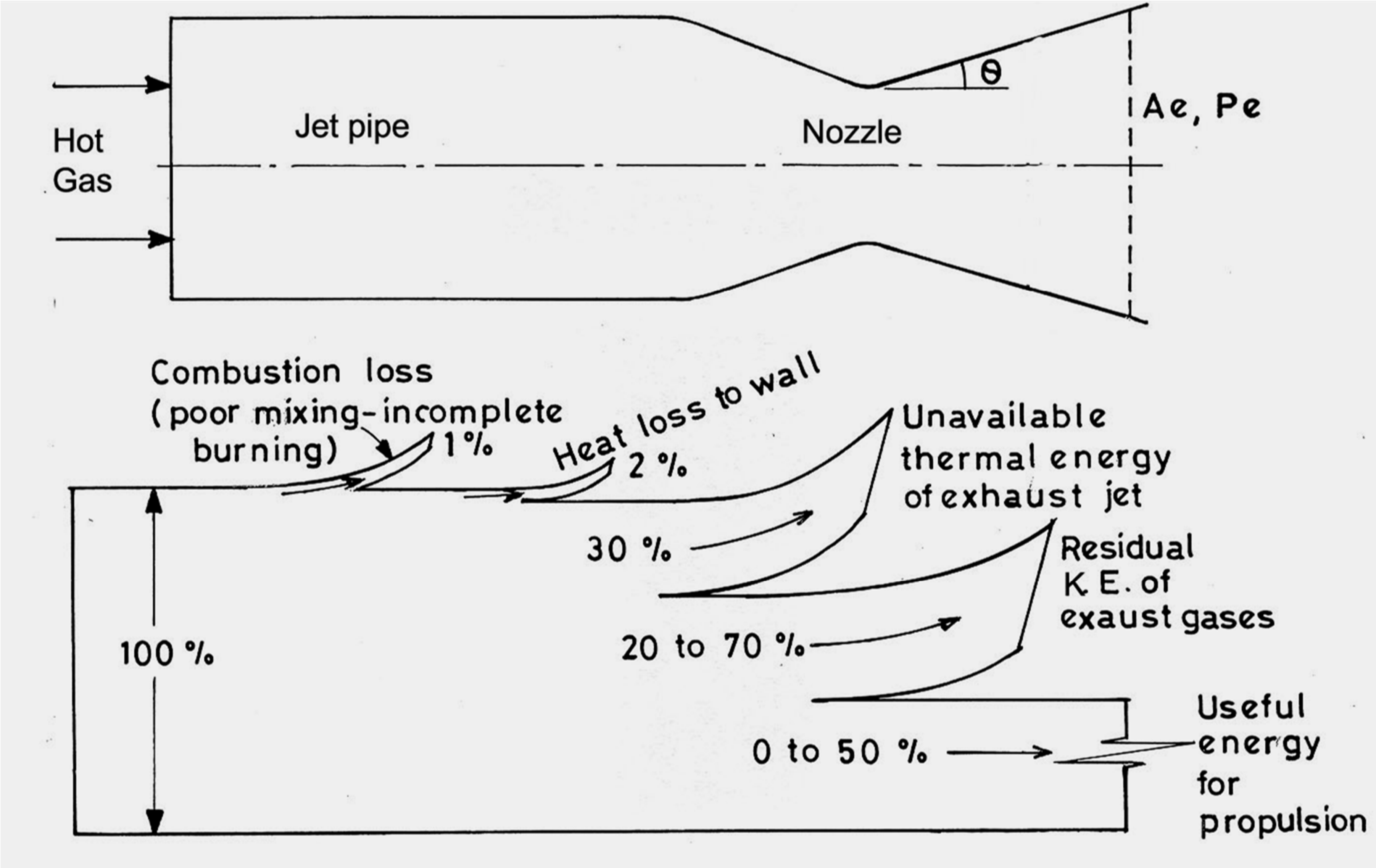
Mixed compression external and internal shock systems in intakes (a) with two internal shocks, (b) with two external shocks

- **Intakes for supersonic aircraft may be designed to contain all or some of the shocks inside the body of the intake. Shocks contained within the body (Fig.) then give off reflected shocks within the body of the intake duct. Flow through all the original and reflected shocks need to be estimated to complete the shock configuration.**

- **A more practical solution seems to be make one of the lips (normally the lower lip) move forward and create one or more shocks (Fig.) after which one or more shocks may be contained within the intake duct. Such a mixed compression intake system has been used over the years by many designers.**

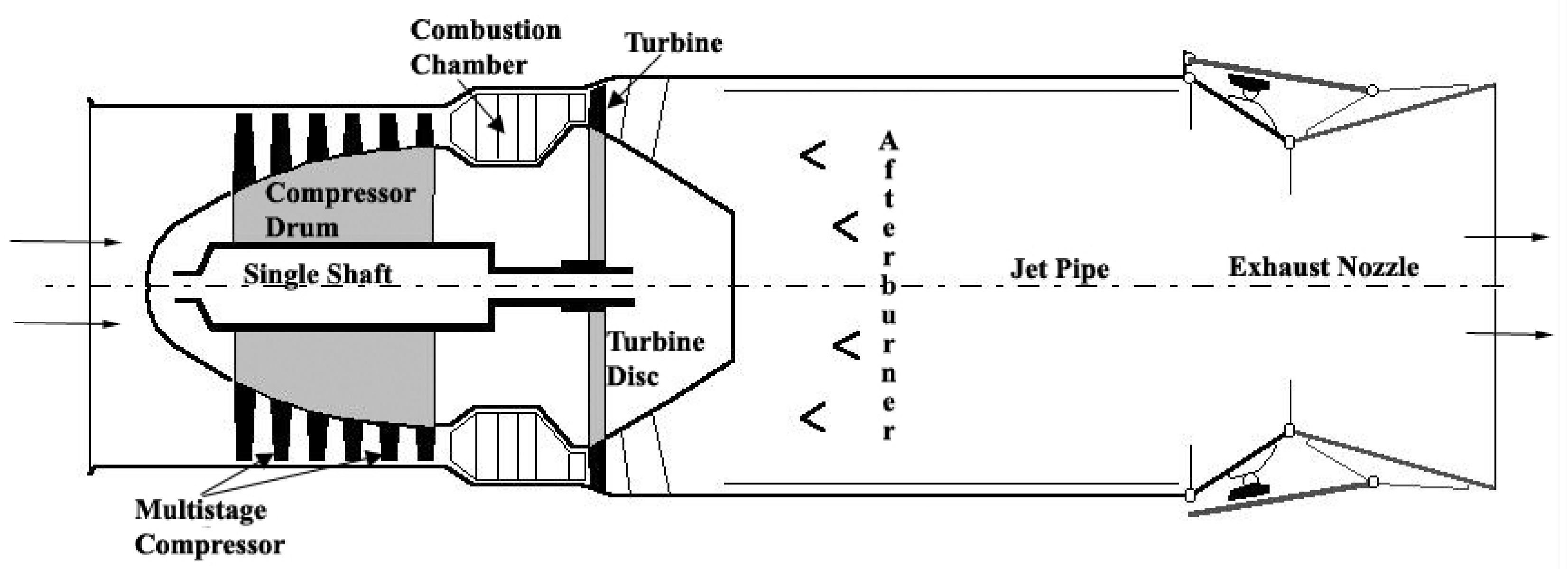
- **However, in the recent years to cater to high angle attack aircraft maneuvers the mixed compression intakes are designed with the upper lip advanced forward, and it becomes the first shock creating lip.**

PROPELLING NOZZLES

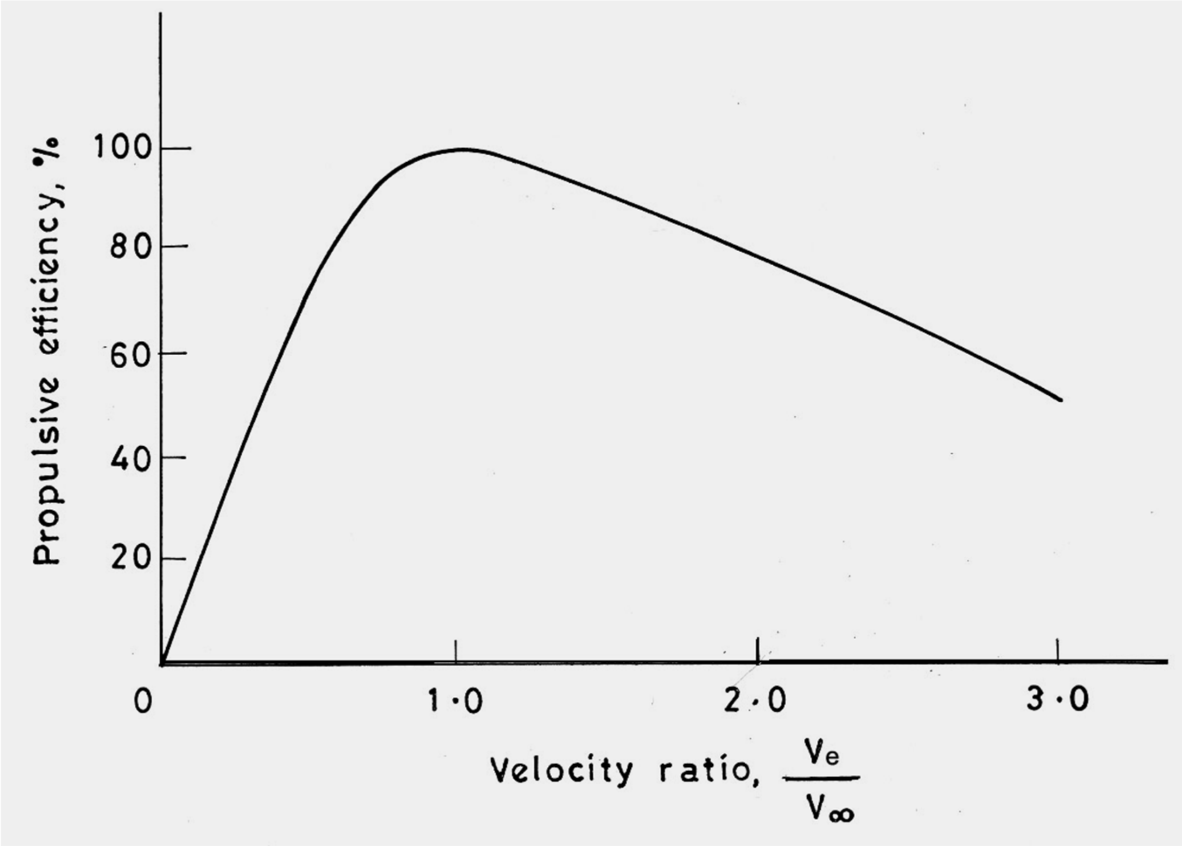


Hot Gas
Jet pipe
Nozzle
θ
Ae, Pe
Combustion loss (poor mixing-incomplete burning)
1 %
Heat loss to wall
2 %
Unavailable thermal energy of exhaust jet
30 %
Residual K. E. of exaust gases
20 to 70 %
100 %
0 to 50 %
Useful energy for propulsion

In a turbojet engine, design of nozzle is important as that is the sole means of development of maximum thrust.



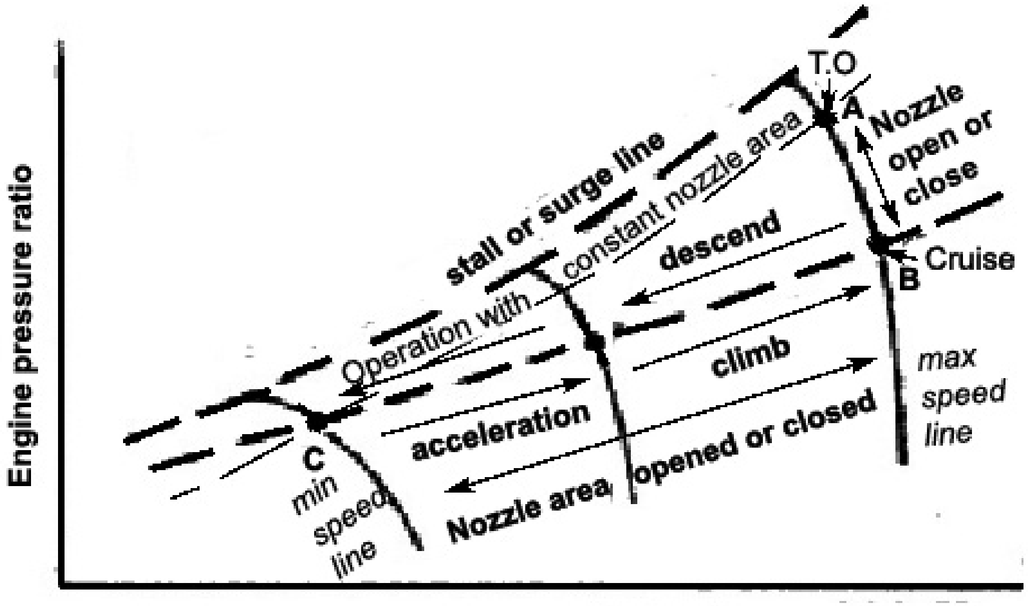
In turbo propeller engine or turbofan engine, the nozzle design is not as severe as the acting pressure ratio is much less than in the turbojet engine.



For a supersonic Mach 2.5 operation, the nozzle pressure (expansion) ratio would be around 20 where 1 - 1.5% gain in gross thrust amounts to 2.5 to 4% gain in net thrust and hence C-D nozzle may result in substantial performance gain.

$\left(\frac{p_{01}}{p_2}\right)$	4	8	12	16	20
$\frac{F(C+D)}{F(C)}$	1.007	1.062	1.09	1.12	1.13

Below a pressure ratio of 4.0 the gain in C-D nozzle would be offset by the extra length (and weight) of the powerplant.



Mass flow parameter $\bar{m} = \dfrac{m\sqrt{c_p T_{01}}}{A \cdot P_{01}}$

Consider a turbojet engine operating during take-off at the design point A . As the aircraft climbs it is necessary to continuously reduce the thrust till at cruise it reaches a point B and then during landing down to C.

The change from A to B is effected by steadily opening the jet nozzle and reducing the TET (T_{03}), maintaining rotational speed (rpm) constant. This process is continued till the jet nozzle is fully open. In this way we maintain a high air flow while reducing the TET, fuel flow and compression ratio. Point B is far removed from stall line in a stable operating zone on the maximum speed line. Further reduction in thrust can be effected only by reducing TET (i.e. fuel flow) and rotational speed simultaneously along the line BC. This operation avoids compressor stalling.

Variable Area Nozzle

During acceleration variable nozzle area is helpful for following reasons during starting.

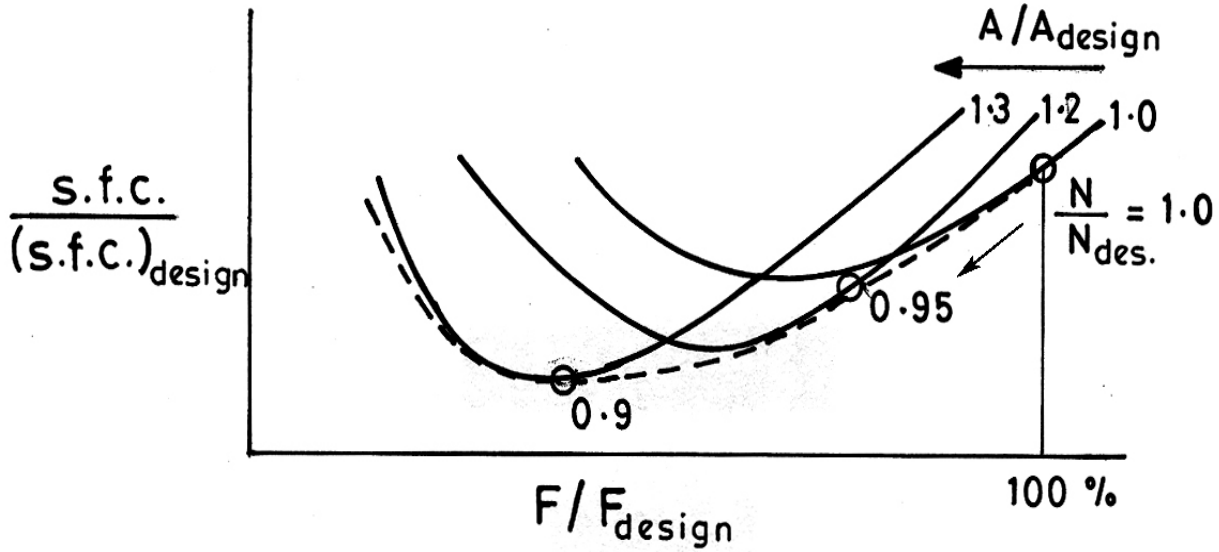
1. It permits large amounts of fuel to be added during acceleration along the line CB.
2. The thrust increase along the line BA can be accomplished in a very short time due to constant speed process, thus permitting fast acceleration.

So the advantages of a variable geometry nozzle are:

- Avoids compressor stall during starting and acceleration.
- Reduces acceleration time.
- Improves low thrust sfc.
- Reduces loss on T.O. thrust in hot climates.
- Avoids excessive or too low T_{03} under varying flying conditions.

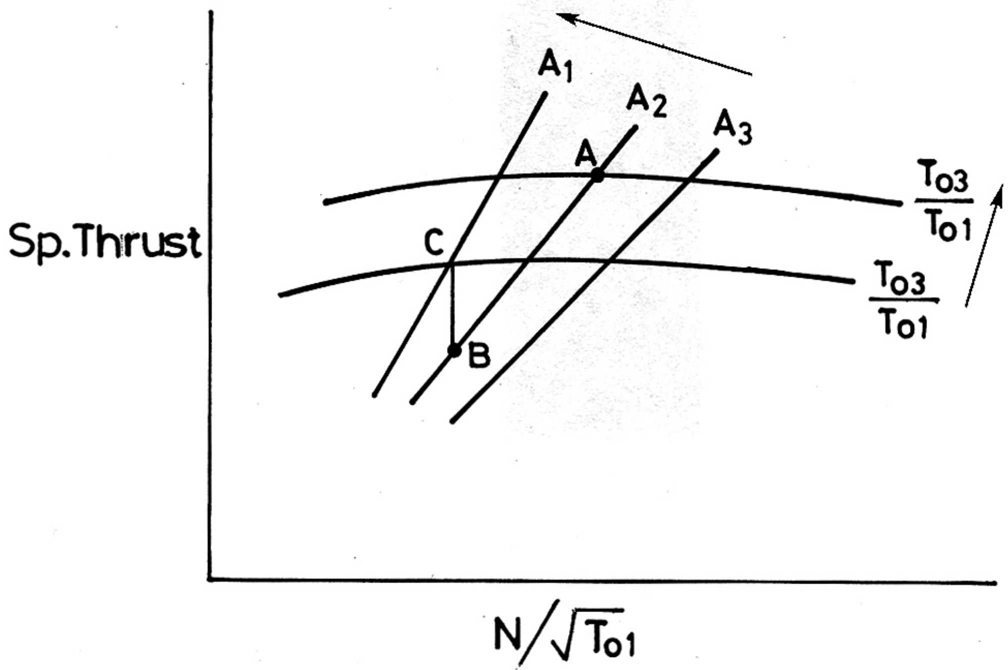
Its disadvantages are:

- Adds another control variable.
- Adds mechanical complexity.
- Adds weight.
- At cruise the air flow is high and hence duct losses are generally high.



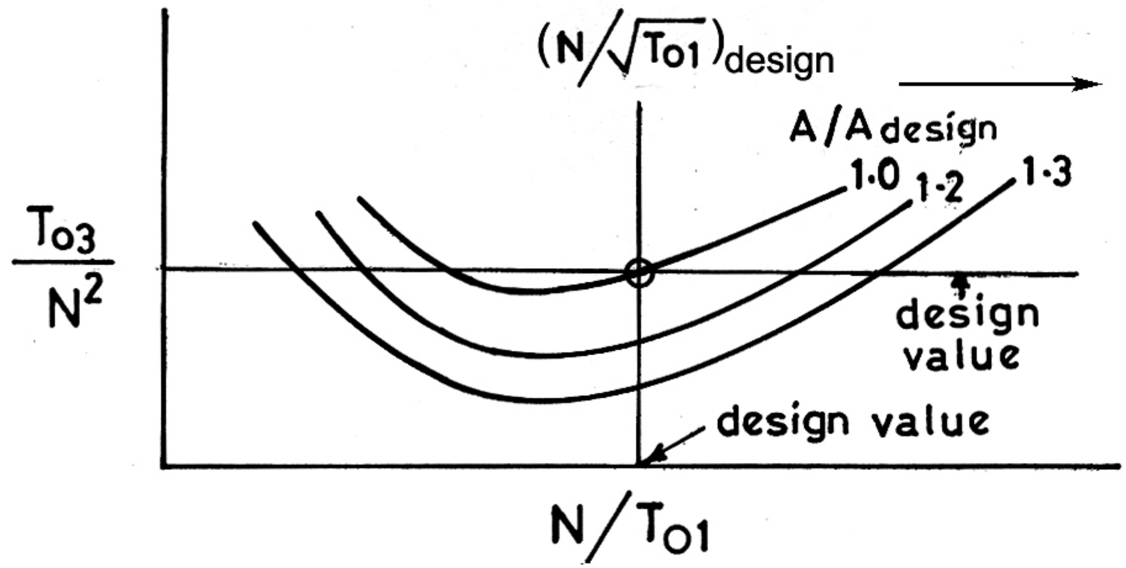
(a) Effect of Nozzle area variation on SFC

The variable area exhaust nozzle reduces the specific fuel consumption (s.f.c) at part thrust by enabling an optimum matching of the effectiveness of the compressor and turbine. The overall relative performance curves Fig.(a) show this quite clearly. For optimized operation nozzle area variation is a useful method. Such performance or control schedules are prepared for various flight operations.



(b) Nozzle area matching with rpm and TET

Another important use of variable area nozzle is operation on a warm day. The rise in T01 shifts the operating point from A to B on a constant nozzle area. Now, if it possible to vary the nozzle area it would be possible to retrieve some amount of thrust by finding the correct value of T03/T01 to coincide with A1. Fig. (b)



(c) Nozzle area matching with turbine and compressor operation

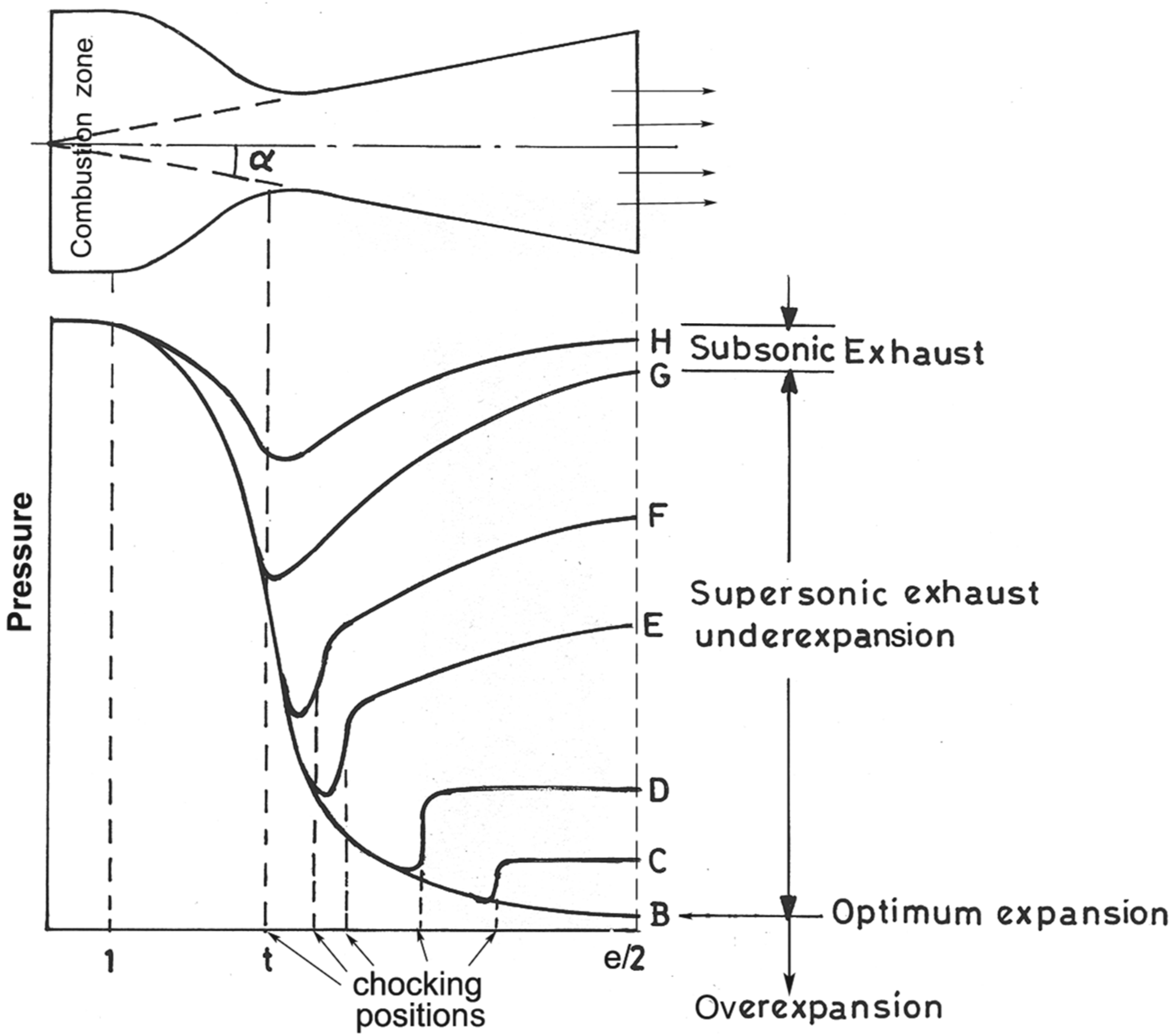
The last curve (Fig. c) suggests that on an excessive hot or cold (T_{01}) climate, nozzle can be fixed at higher values to avoid exceeding the turbine entry temperature (TET), T_{03} limit set by the turbine material.

Variable area C- D nozzle

- **Consider briefly the implications of variable area C-D nozzle. Because of the necessity of maintaining the shock at the throat during all the operational regimes, it is necessary to design a variable area throat as well. With reheat the problem becomes more acute.**

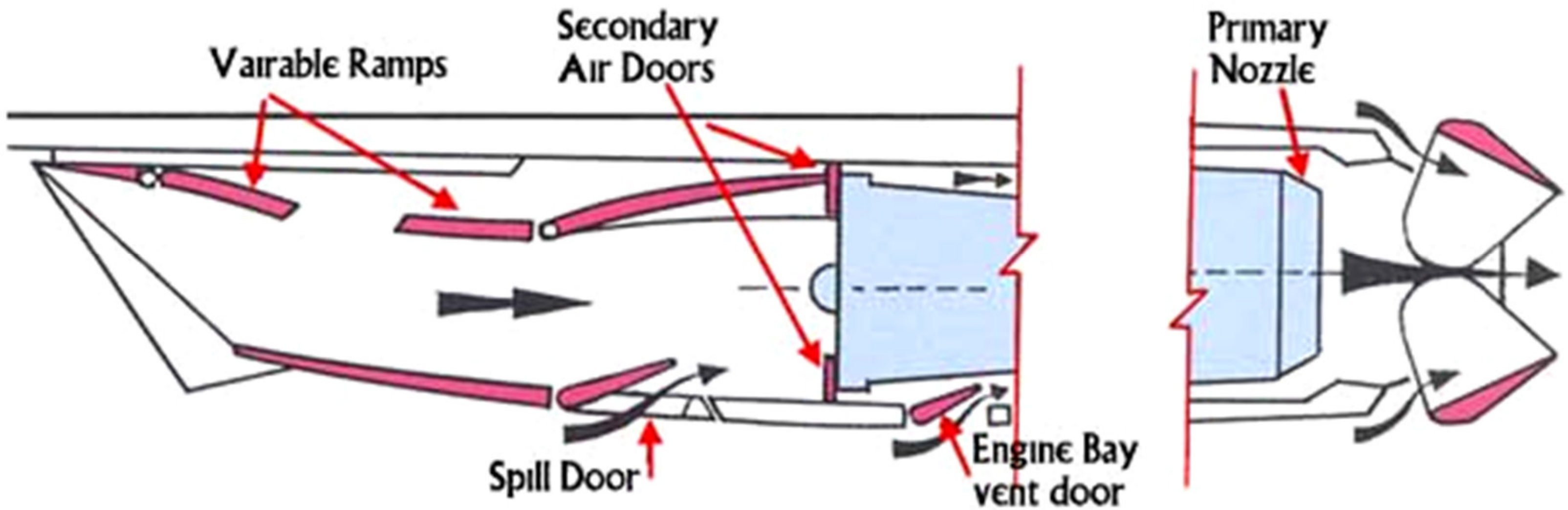
- **As a rule the C-D nozzle is more sensitive to variations in flow conditions than a convergent nozzle. Hence the mechanical design problem is quite formidable. To date except for high performance aircraft with reheat engines, the use of variable area C-D nozzle is rare.**

- **It must be remembered that extra length of power plant means more weight to be carried. Unless it is justified with respect to particular aircraft under consideration it has no useful purpose.**



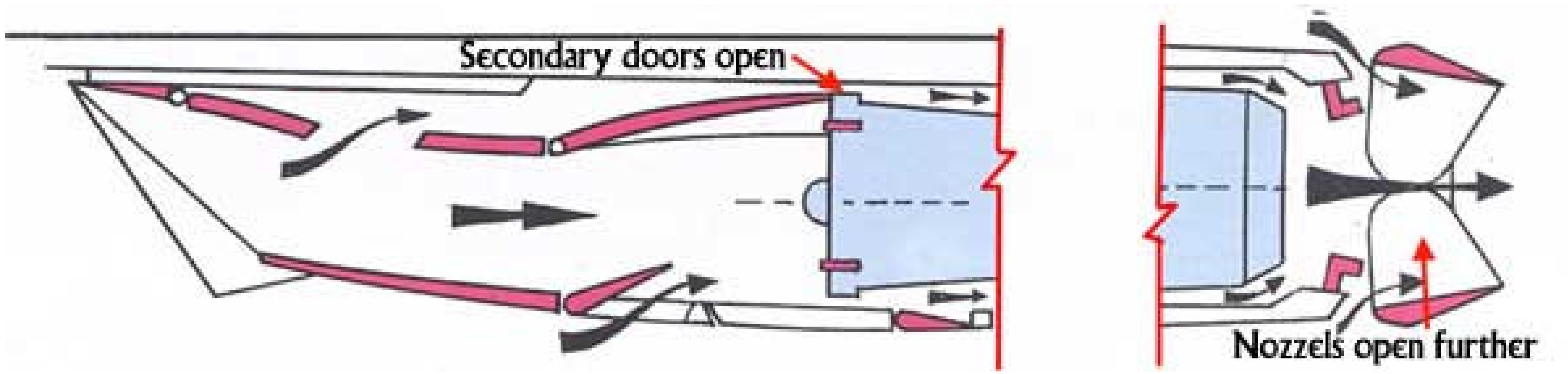
Combustion zone
α
Pressure
H
G
F
E
D
C
B
1
t
e/2
Subsonic Exhaust
Supersonic exhaust
underexpansion
Optimum expansion
Overexpansion
chocking
positions

Subsonic Speeds (take off/subsonic cruise)



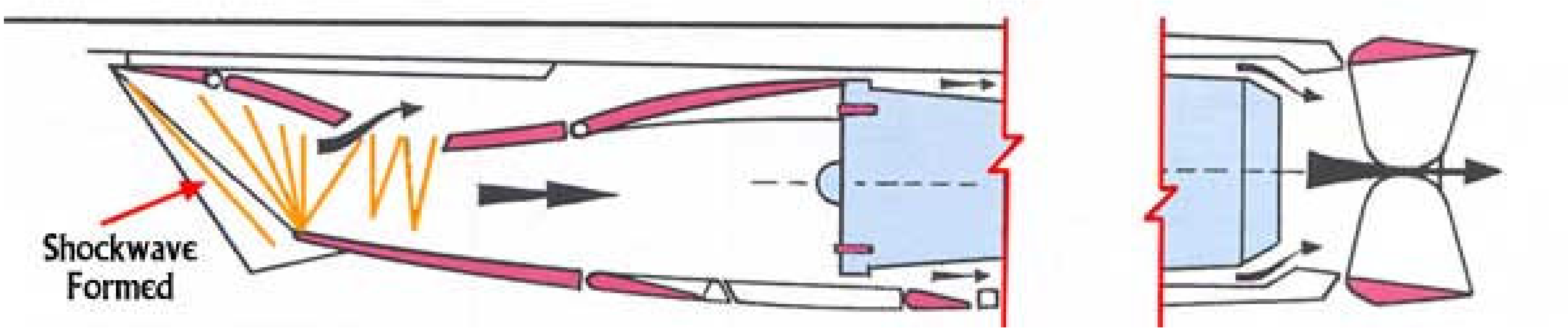
At take off the engines need maximum airflow, therefore the ramps are fully retracted and the auxiliary inlet vane is wide open. This vane is held open aerodynamically. The auxiliary inlet begins to close as the Mach number builds and it completely closed by the time the aircraft reaches Mach 0.93.

At take off and during subsonic flight, 82% of the thrust is developed by the engine alone with 6% from the nozzles and 21% from the intakes



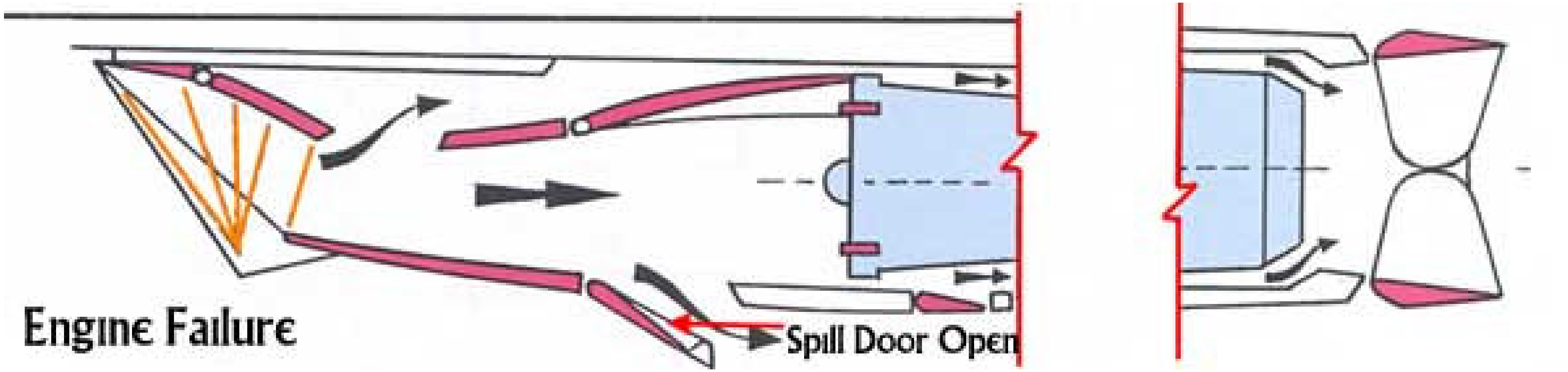
Shortly after take off the aircraft enters the noise abatement procedure where the re-heats are turned off and the power is reduced. The secondary nozzles are opened further to allow more air to enter, therefore quietening down the exhaust. The Secondary air doors also open at this stage to allow air to by pass the engine.
At slow speeds all the air into the engine is primary airflow and the secondary air doors are kept closed. Keeping them closed also prevents the engine ingesting any of its own exhaust gas. At around Mach 0.55 the Secondary exhaust buckets begin to open as a function of Mach number to be fully open when the a/c is at M=1.1

Supersonic Speeds (Supersonic cruise)



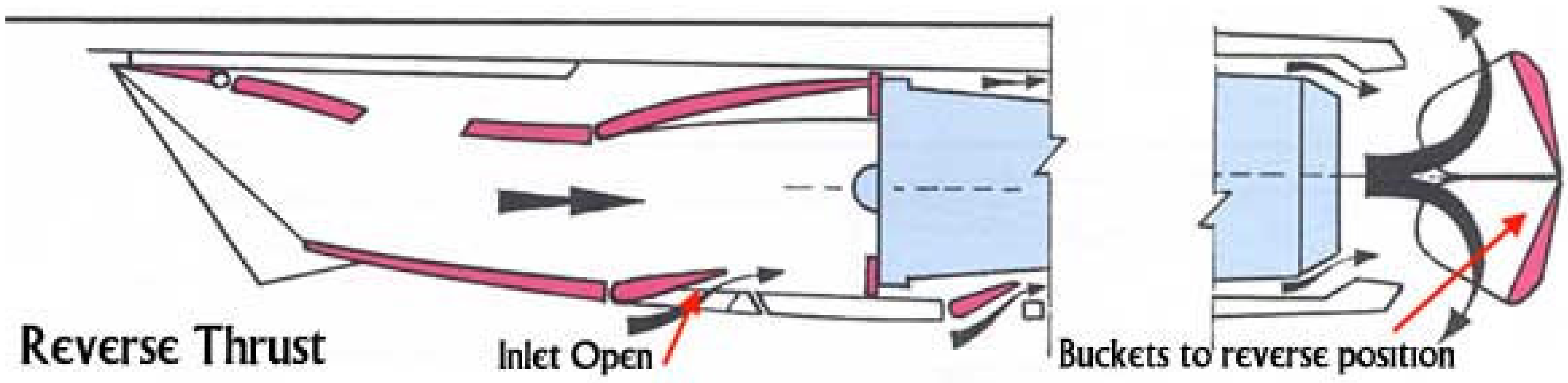
At the supersonic cruse speed of mach 2.0 the ramps have moved over half their amount of available travel, slowing down the air by producing a supersonic shockwave (yellow lines) at the engine intake lip.

During the Supersonic cruse only 8% of the power is derived by the engine with the other 29% being from Nozzles and an impressive 63% from the intakes.



When the throttles are brought back to start the decent the spill door is opened to dump out excess air that is no longer needed by the engine, this allows the ramp to go down to their maximum level of travel. As the speed is lowered the spill doors are closed and the ramps begin to move back so by M~1.3 are again fully retracted.

Should an engine fail and need to be shut down during supersonic cruise, the ramps move fully down and the spill door opens to dump out excess air that is no longer required by the failed engine. The procedure lessens the chances of surges on the engine.



After touch down the engines move to reverse power mode. The main effect of this is that the secondary nozzle buckets move to the closed position directing airflow forwards to slow the aircraft down.

Problem-1

1. A fixed geometry internal compression intake has a inlet to throat area ratio of 1.2 and a possible entry Mach number of 1.9. Using the isentropic flow tables, and shock tables find out :

Mach number at which the inlet will “start”

Mach number at throat at “starting”.

The static to total pressure ratio at the throat after starting

The Mach number at which the inlet will “unstart”

Solution **: i) Using the shock tables we can compile the variation of area ratios with critical area ratios and critical pressure ratios:**

M_0	A/A^*	P_t/P_{0a}	A_0/A_t
1.9	1.555	0.767	1.193
1.91	1.567	0.762	1.195
1.92	1.580	0.758	1.198
1.93	1.593	0.753	1.20045

using linear interpolation it can be said that the inlet will start at M_0 = 1.923

ii) The “starting” critical area ratio A/A* is 1.590 from the above analysis. Since the fixed geometry intake area ratio is fixed at 1.2 the effective operational area ratio is 1.590/1.2 = 1.325
Taking this as A/A* and using the isentropic supersonic flow tables, M_t =1.697

iii) At M_t =1.697 the static to total pressure ratio at throat is P_t/P_{0t} = 0.754

iv) The intake will unstart when M_t = 1. For A/A* = 1.2 unstart entry Mach number M_0 = 1.534

Problem -2

An axisymmetric jet engine exhaust nozzle operates with a mass flow of 75 kg/s with following parameters : Entry conditions, P01 = 350 kPa, T_{01} = 1600 K, and exit to throat area ratio 1.8, exit to throat pressure ratio, P_{09}/P_{08} = 0.98, discharge coefficient, C_d = 0.98 and ambient pressure of 40 kPa. Compute the various nozzle flow parameters and finally the thrust created by this nozzle. [Use γ = 1.33, R = 287 J]

Solution:

At throat $M_t = 1$, by applying criticality conditions $T_t = 1064$ K. Hence, V_t = at = 677 m/s.

Therefore, ideal area of the throat, $A_t^{/} = \dot{m}_{gas}.R.T_t /(P_t. V_t) = 0.215$ m2

With discharge coefficient 0.98, $A_t = 0.2194\ m^2$, Radius of the throat, r_t = 0.264 m.

Therefore, the exit area , A_e is $1.8.A_t = 0.395\ m^2$, radius at exit plane, r_e = 0.354 m

At the exit station, ideal flow area ratio is $(A_e/A_t)_i = (A_e/A_t)/C_d = 1.8367$, So that $Ae^{/} = 0.403$

For an isentropic flow,

$$\frac{A_e}{A_t} = \frac{1}{M_e}\left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}.M_e^{\,2}\right)\right]^{\frac{(\gamma+1)}{2(\gamma-1)}}$$

; whence, $M_{e\text{-ideal}} = 2.056$,

and from isentropic relations, $P_e/P_{0e} = 0.1186$, assuming $P_{0e} = P_{0t} = 350$ kPa, $P_e^{/} = 41.5$ kPa, and $T_e /T_{0e} = (P_e/P_{0e})0.2481 = 0.589$, which yields, $T_e = 942.5$ K,

Therefore the speed of sound , $a_e = \sqrt{(1.33 \times 287 \times 942.5)} = 599$ m/s

Whereby, ideal exit velocity $V_e^{/} = a_e \times M_e^{/} = 1233$ m/s

Now if we go back to the real values prescribed, $(A_{e/}A_t) = 1.8$; $A_e = 0.395\ m^2$; and from isentropic relations, mach number at the exit plane, $M_e = 2.0386$ and real flow pressure ratio, $P_e/P_{0e} = 0.1228$,

so that $P_e = (P_e/P_{0e}).C_d.P_{0e} = 42.13$ kPa

$T_e / T_{0e} = (P_e/P_{0e}) \ 0.2481 = 0.594 = 951$ K,

Therefore the speed of sound , $a_e = \sqrt{(1.33 \times 287 \times 951)} = 602.5$ m/s,

whence, $V_e = 1228$ m/s.

So, isentropic velocity coefficient of the nozzle is $\xi_N = Ve / Ve^{/} = 0.9959$

<u>Isentropic thrust coefficient</u> may be written down *for a flow which undergoes under-expansion,*

$$C_F = C_d.\xi_N.\sqrt{\frac{1-(P_e^{/} / P_{ot})^{\frac{\gamma-1}{\gamma}}}{1-(P_e / P_{ot})^{\frac{\gamma-1}{\gamma}}}}.\left[1+\frac{\gamma-1}{2.\gamma}.\frac{1-P_a/P_e}{\{P_{0e}/P_e\}^{\left(\frac{\gamma-1}{\gamma}-1\right)}}\right]$$

such that if, $P_e = P_e^{/}$, and $P_e = P_a$ then In the present case, $P_a = 40$ kPa, hence, from the above relation, $C_F = 0.9766$

***Actual thrust* produced, $F = \dot{m}_{gas}.Ve + (P_e - P_a)\ A_e = 92.5$ kN**

The *ideal thrust* produced would have been, $F^{/} = F/C_F = 94.7$ kN