AE 658

Aerodynamics of Propelling Nozzles

Requirements made on the nozzles:

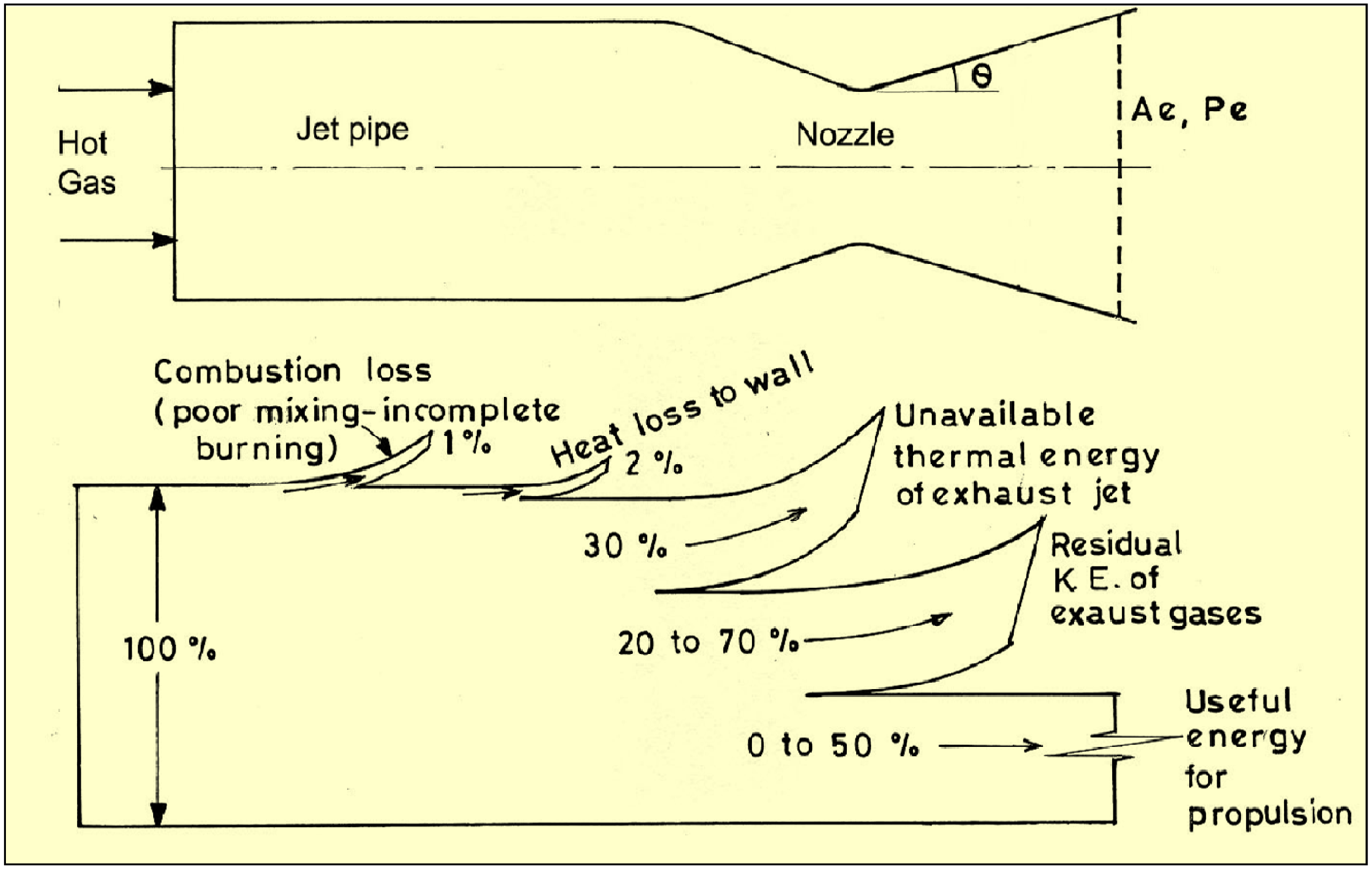
1. **Minimize installation drag on their own and adjacent aircraft surfaces over wide range of aircraft angle of attack and engine throttle settings**
2. **Maximize internal total pressure recovery (π_d, π_n) of nozzle under various operating conditions**
3. **Provide *controllable mass flow* matching 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**

Special requirements : Nozzles

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

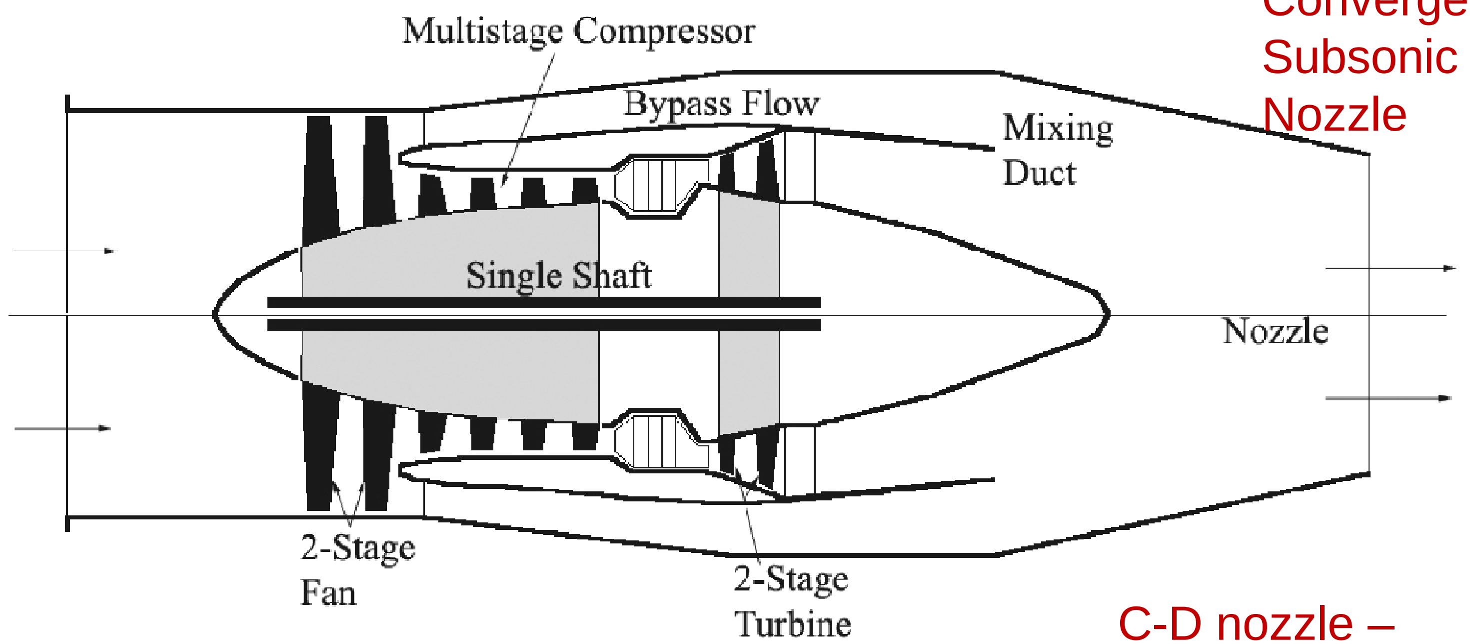
Many types of nozzles have been developed to meet these requirements

NOZZLES

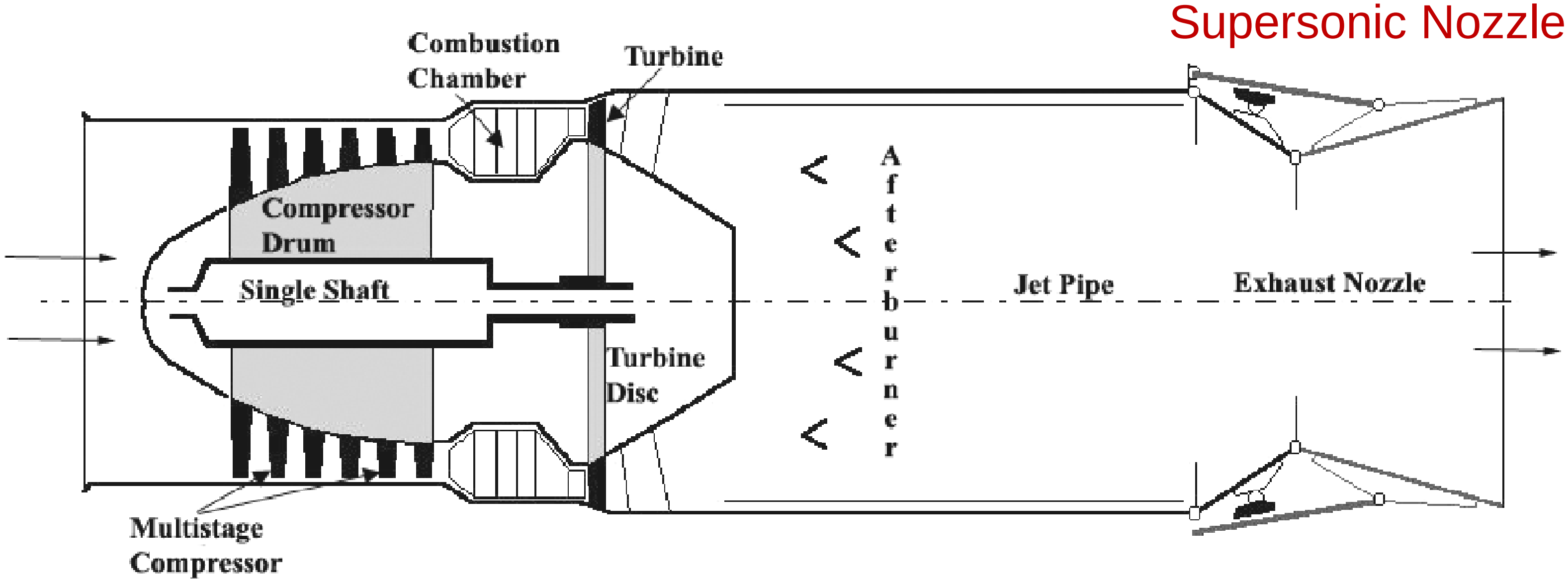


Gas coming out of turbines retains sufficiently high temperature and pressure. Approximately 40% of the total energy input in a gas turbine based powerplant is available for conversion to useful thrust/power for propelling an aircraft. Rest of the energy is used by the compressor-turbine loop. Out of 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.

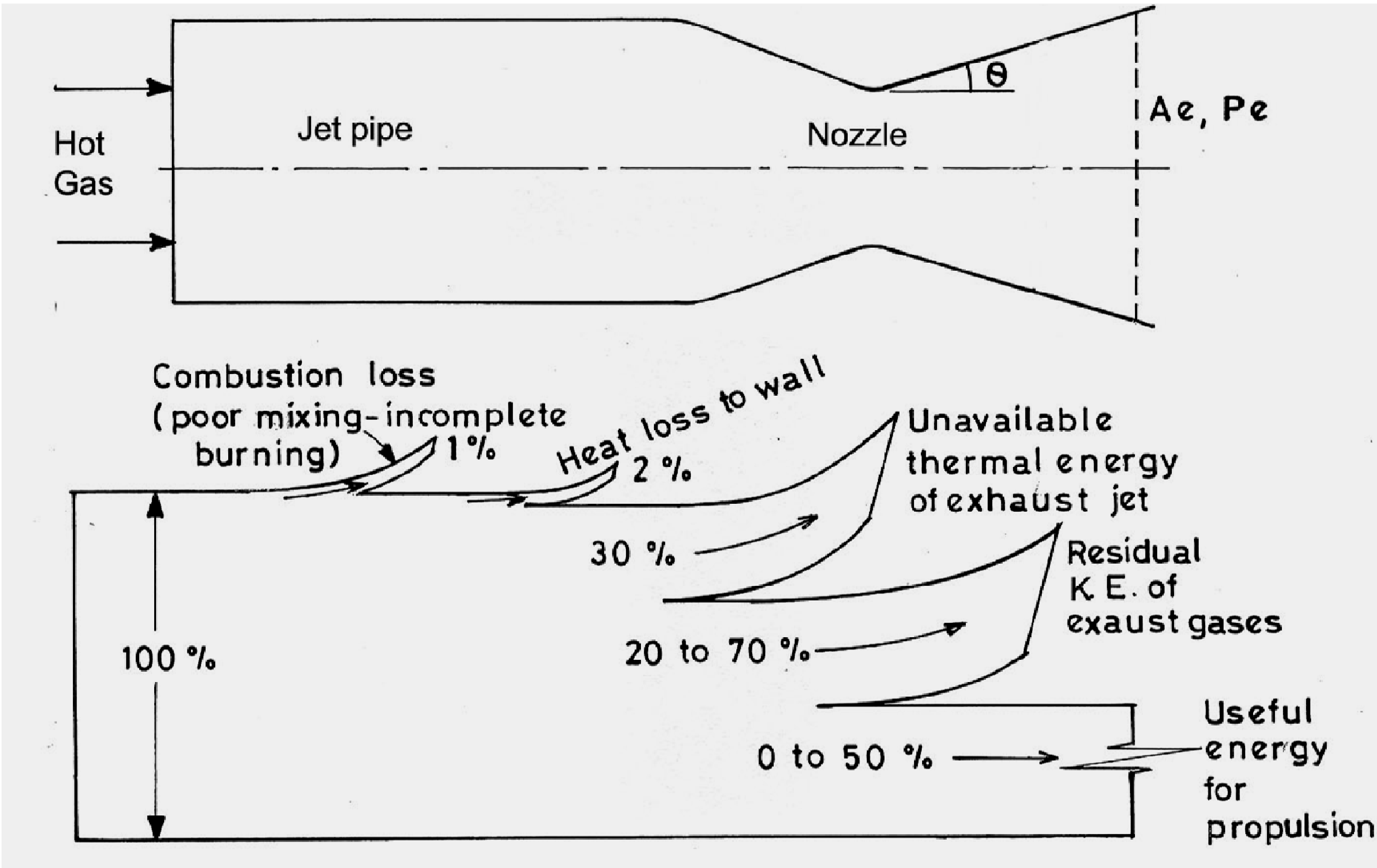
Convergent
Subsonic
Nozzle



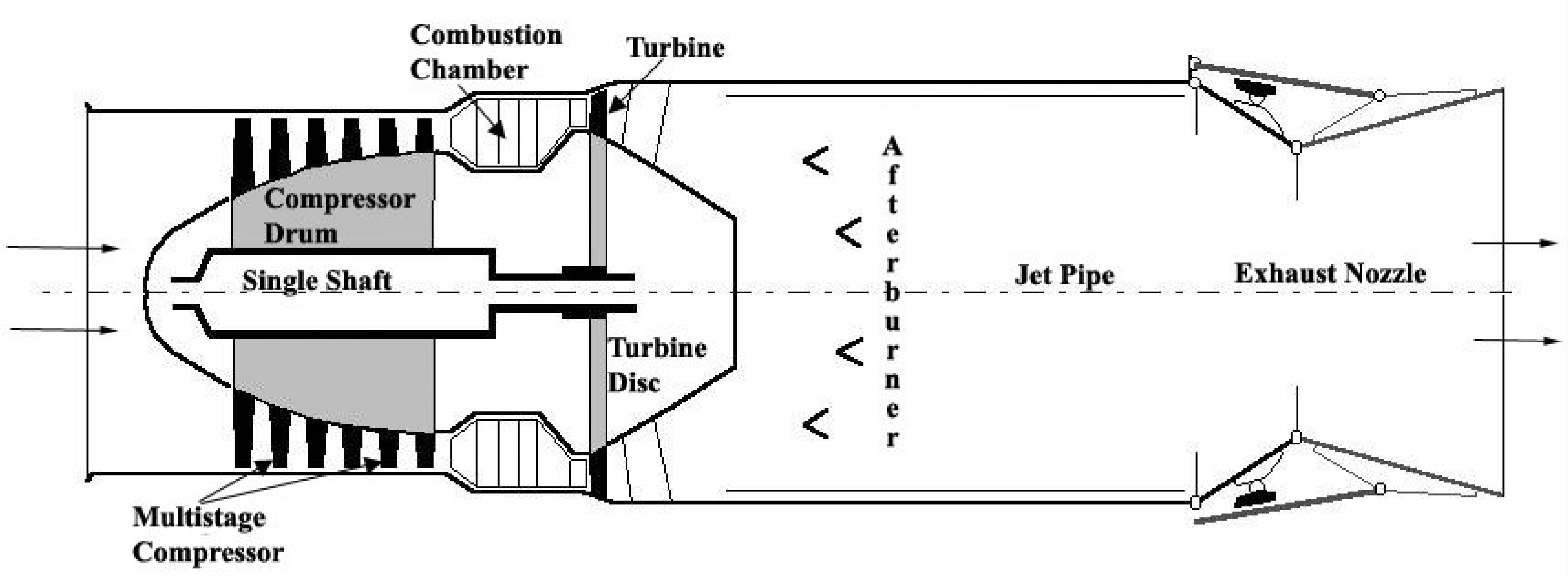
C-D nozzle –
Supersonic Nozzle



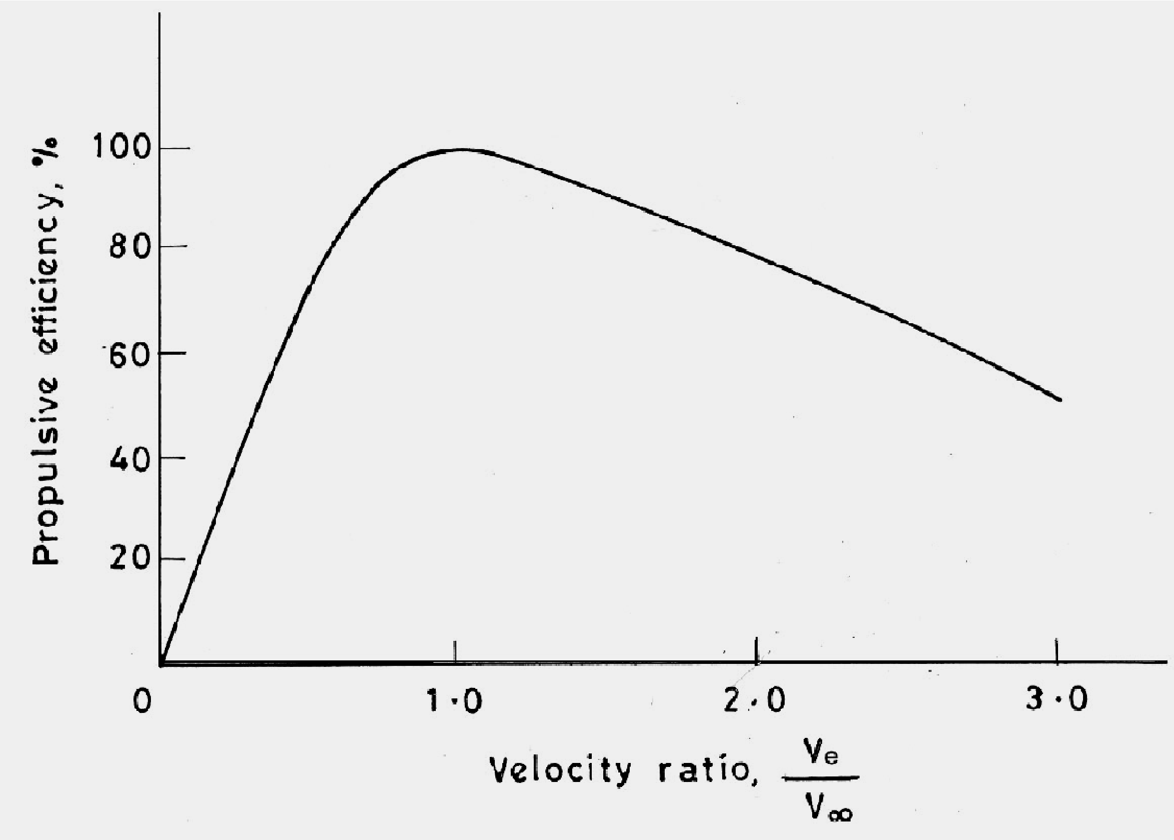
PROPELLING NOZZLES


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.



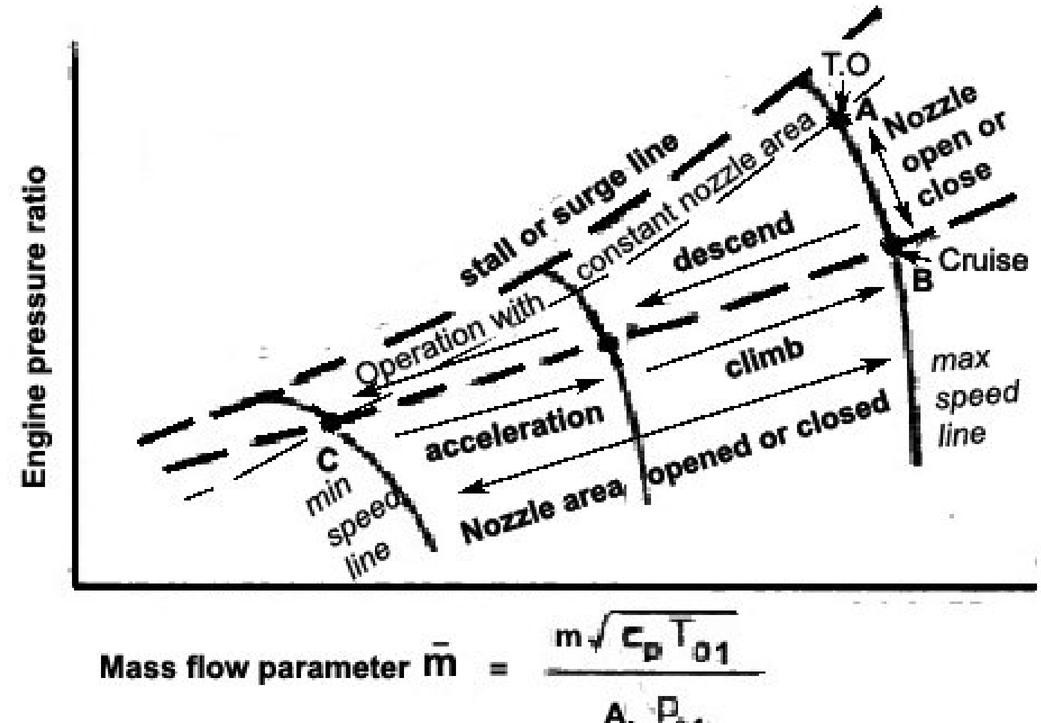
In turbo propeller engine or turbofan engine, the nozzle design is not as severe as the acting pressure ratio is much less than in the turbojet engine.



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.



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.

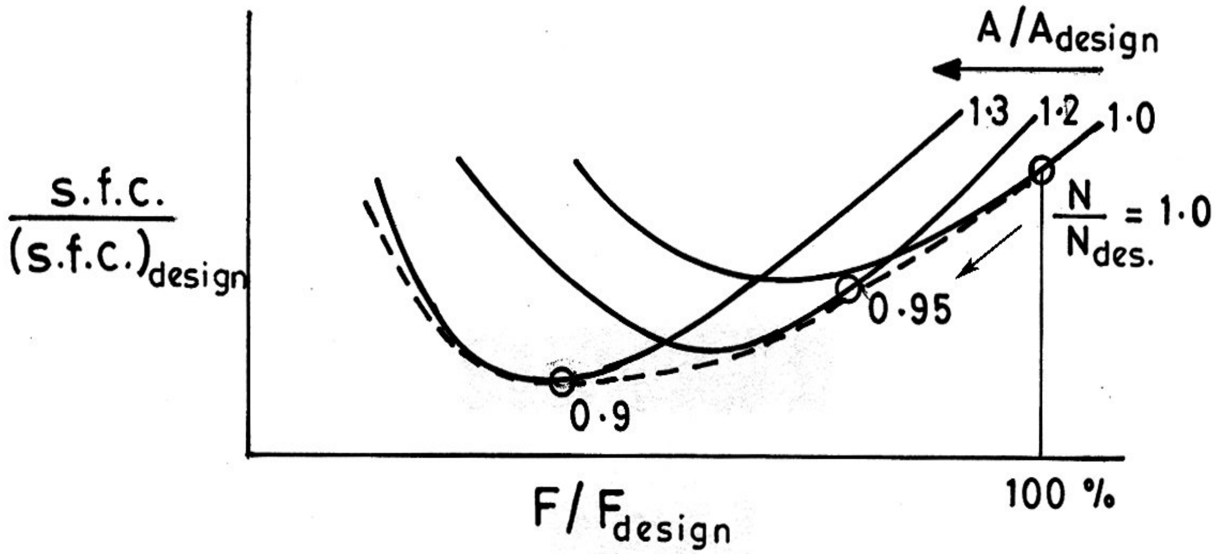
1. It permits large amounts of fuel to be added during acceleration along the line CB.
2. The thrust increase along the line BA can be accomplished in a very short time due to constant speed process, thus permitting fast acceleration.

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 T03 under varying flying conditions.

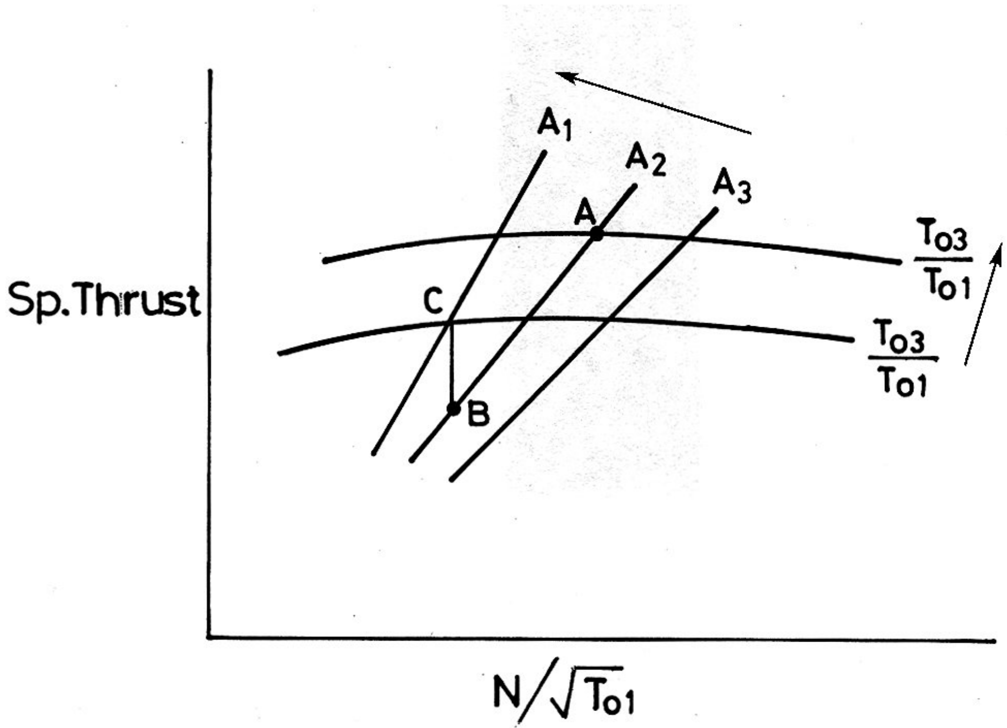
Its disadvantages are:

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



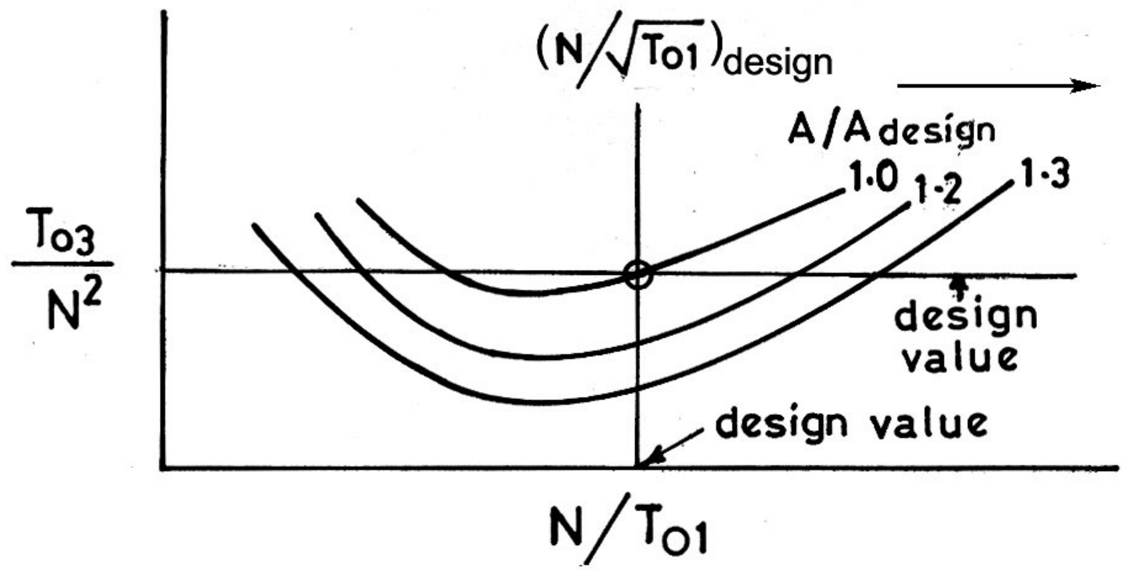
(a) Effect of Nozzle area variation on SFC

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

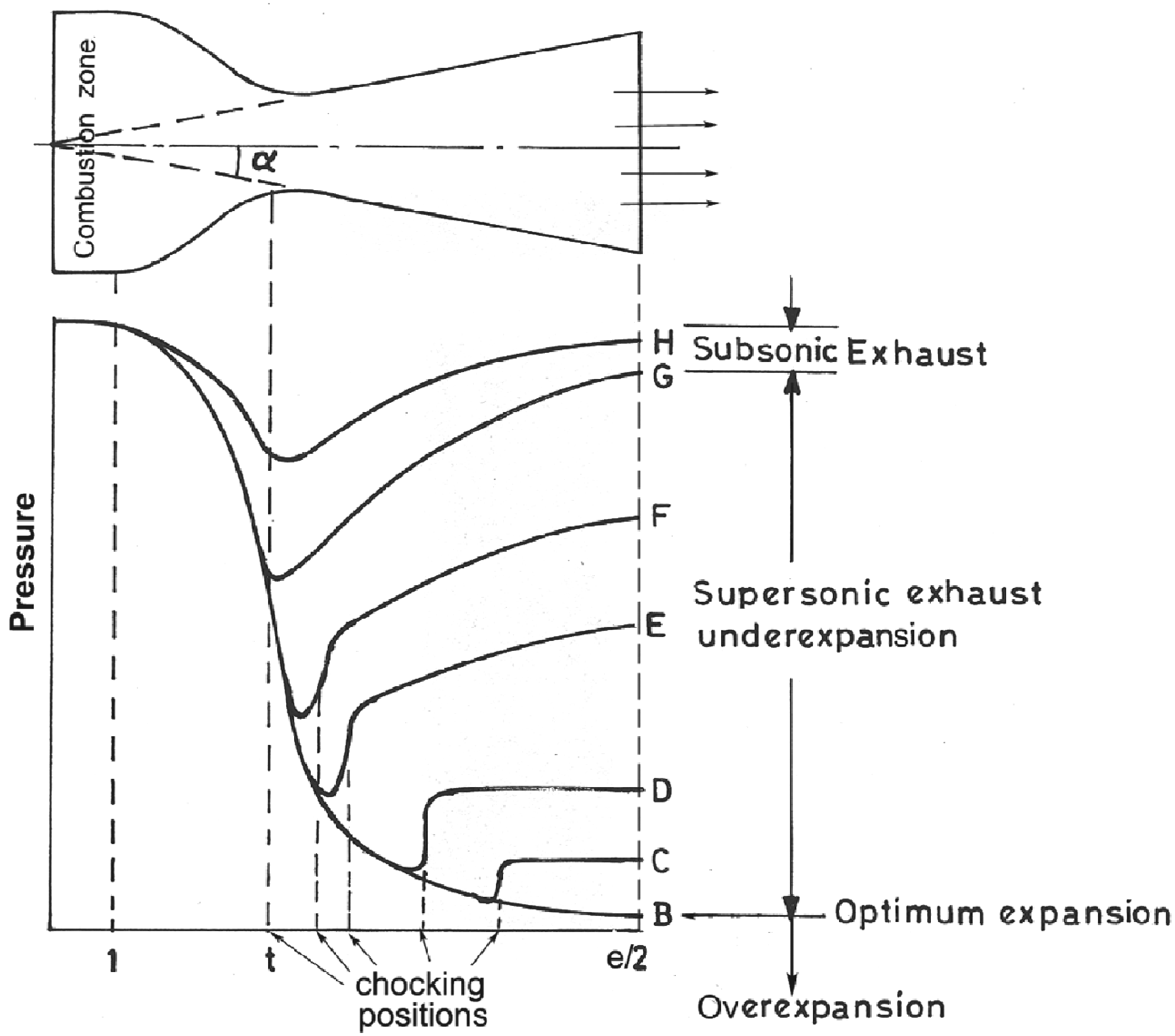
The last curve (Fig. c) suggests that on an excessive hot or cold (T_{01}) climate, nozzle can be fixed at higher values to avoid exceeding the turbine entry temperature (TET), T_{03} limit set by the turbine material.

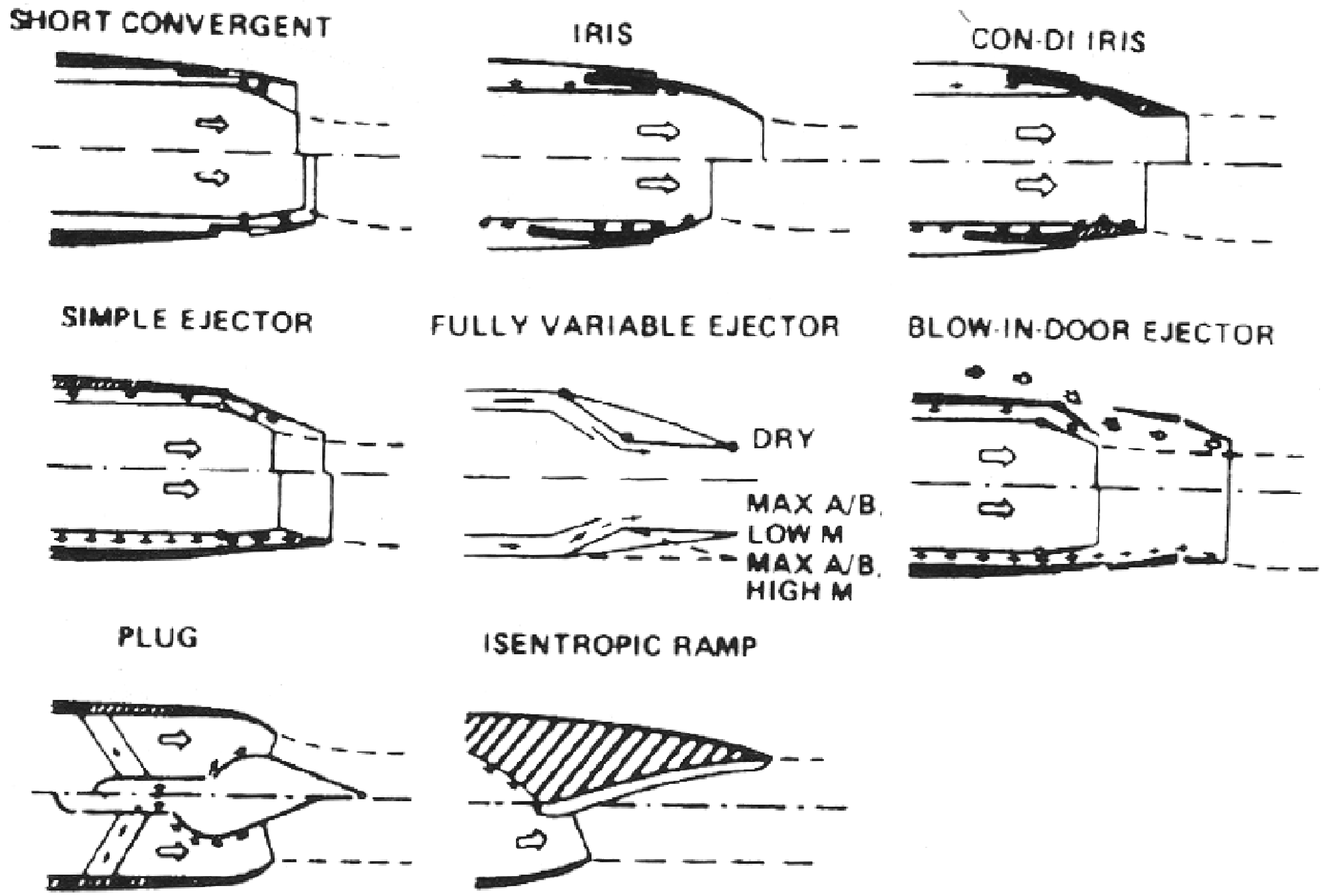
Variable area C- D nozzle

• 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.

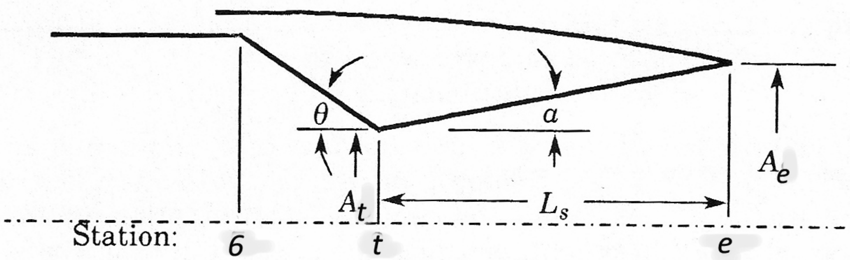
• It must be remembered that extra length of power plant means more weight to be carried. Unless it is justified with respect to particular aircraft under consideration it has no useful purpose.


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



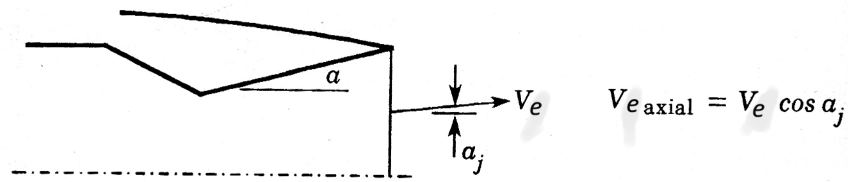
(Short arrows indicate the presence and flow direction of nozzle cooling air which is usually inlet bleed air, compressor bleed air, or fan discharge air)

Typical Nozzle Concepts for Afterburning Engines

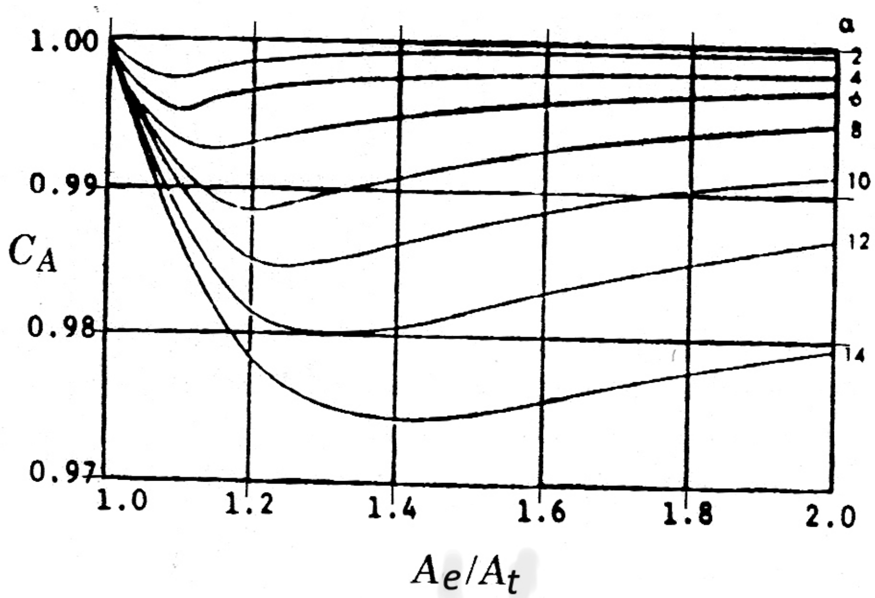


A_t - Primary nozzle throat area
A_e - Secondary nozzle exit area
α - Secondary nozzle half angle
θ - Primary nozzle half angle
L_s - Secondary nozzle length

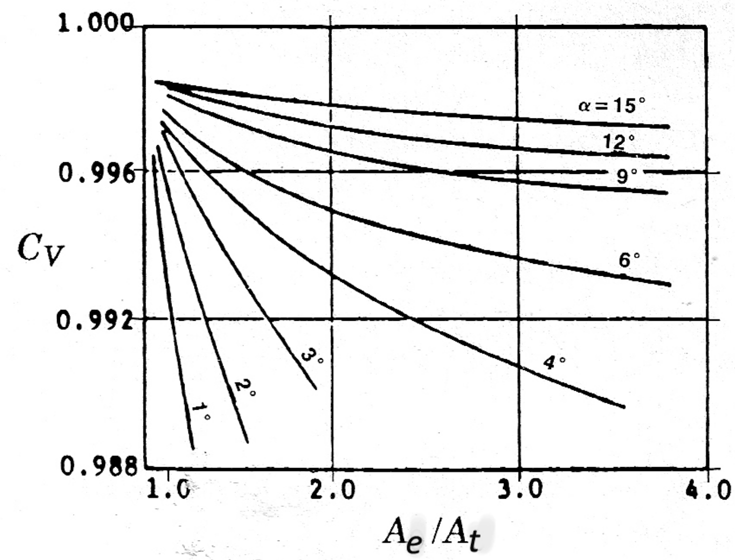
Nozzle Geometric Parameters



Local Angularity Coefficient

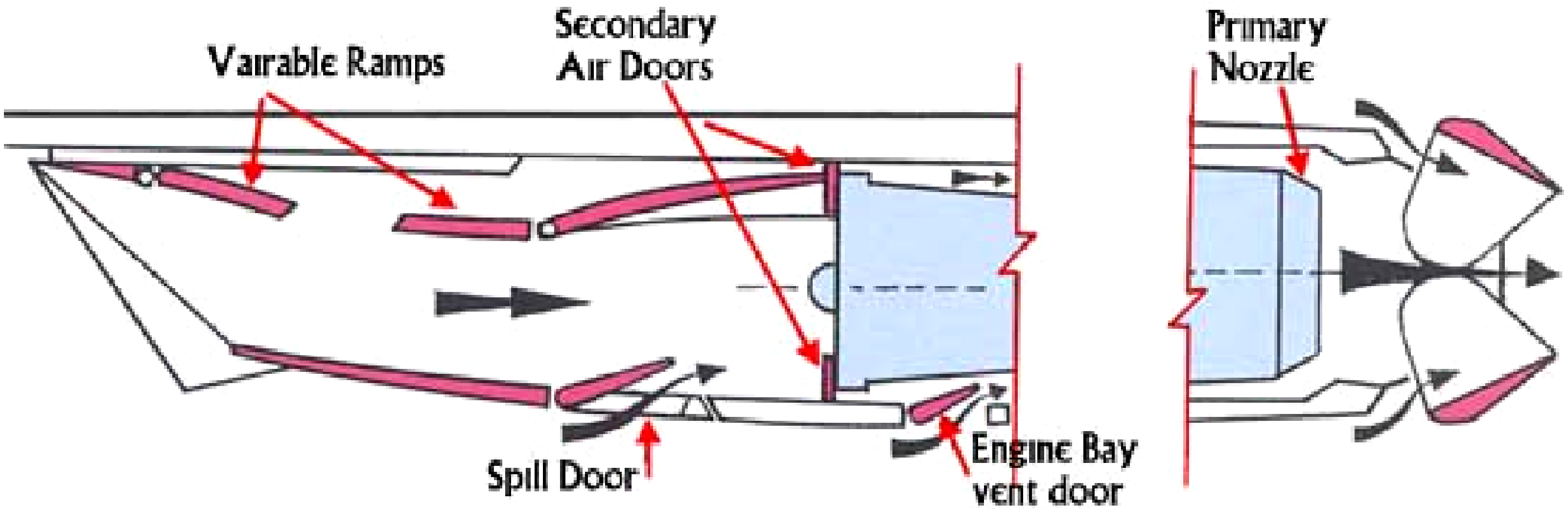


C-D Nozzle Angularity Coefficient



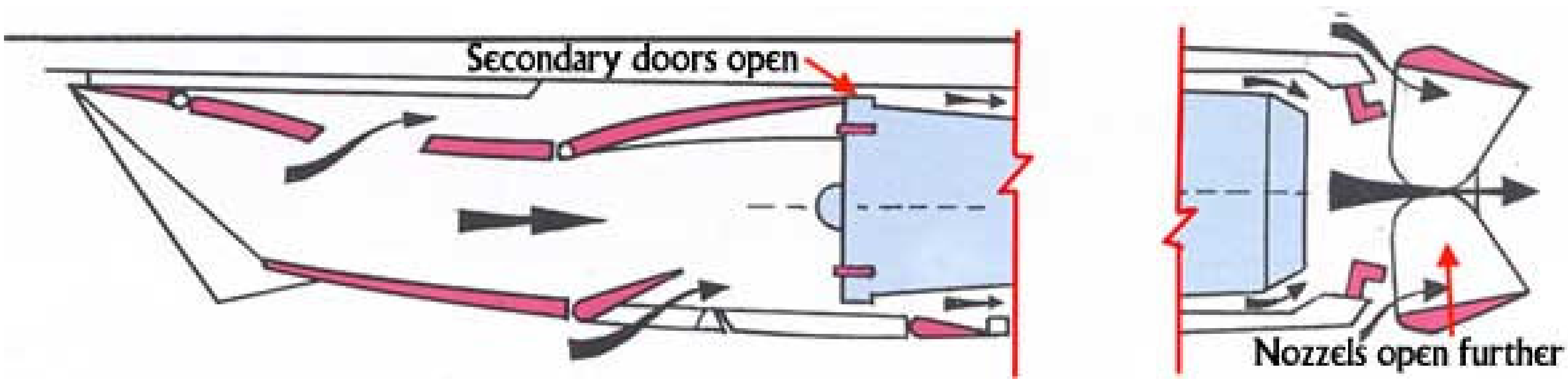
C-D Nozzle Velocity Coefficient

Subsonic Speeds (take off/subsonic cruise)



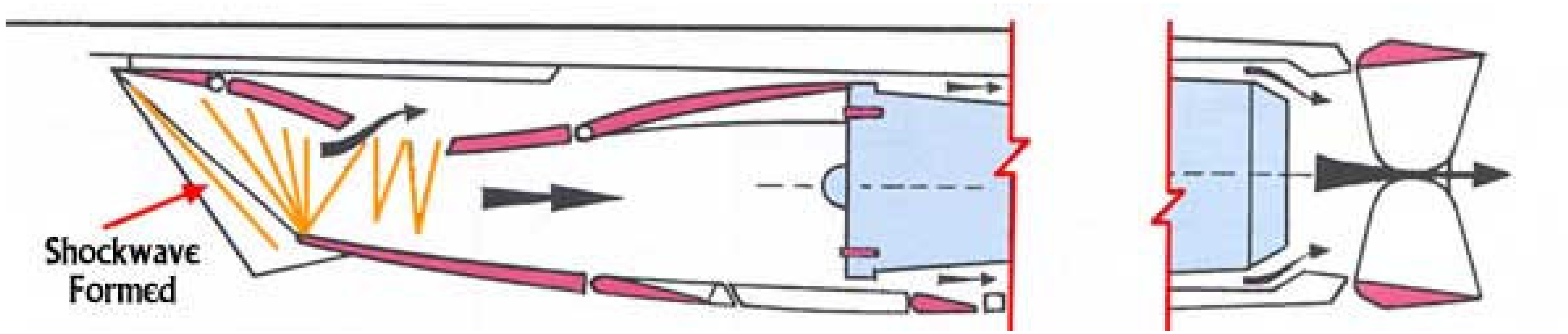
At take off the engines need maximum airflow, therefore the ramps are fully retracted and the auxiliary inlet vane is wide open. This vane is held open aerodynamically. The auxiliary inlet begins to close as the Mach number builds and it completely closed by the time the aircraft reaches Mach 0.93.

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



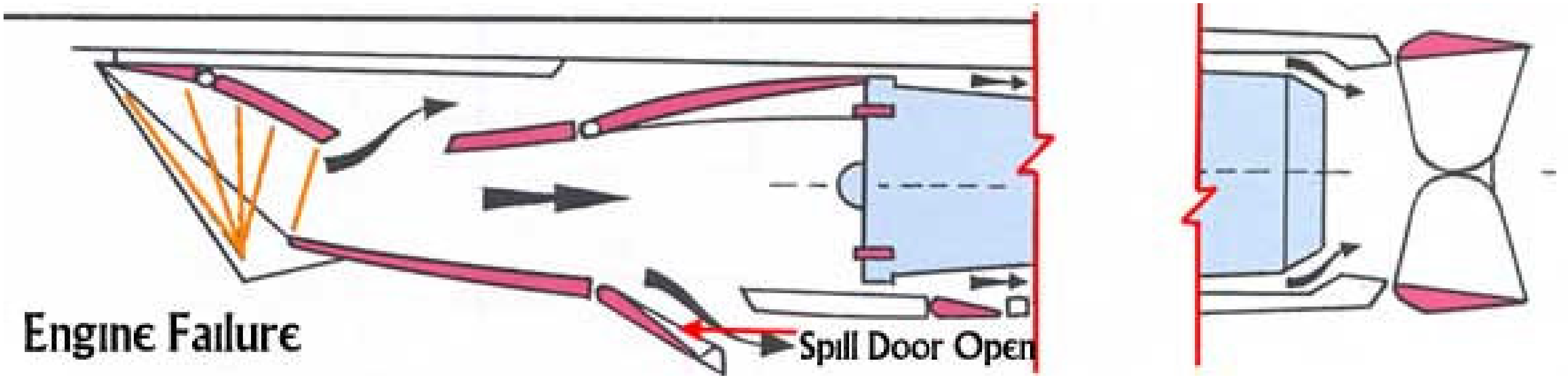
Shortly after take off the aircraft enters the noise abatement procedure where the re-heats are turned off and the power is reduced. The secondary nozzles are opened further to allow more air to enter, therefore quietening down the exhaust. The Secondary air doors also open at this stage to allow air 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)



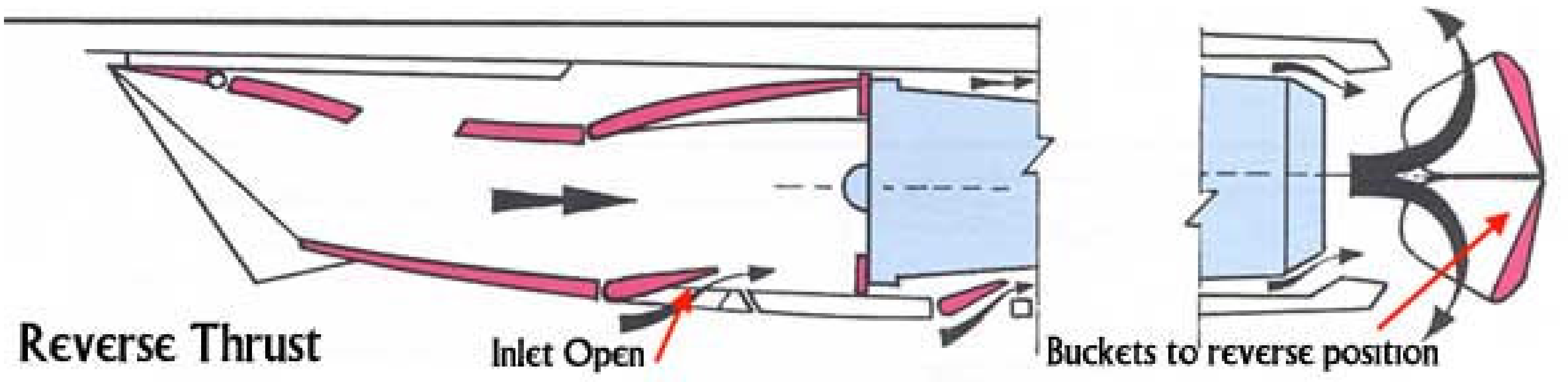
At the supersonic cruse speed of mach 2.0 the ramps have moved over half their amount of available travel, slowing down the air by producing a supersonic shockwave (yellow lines) at the engine intake lip.

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,*

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

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