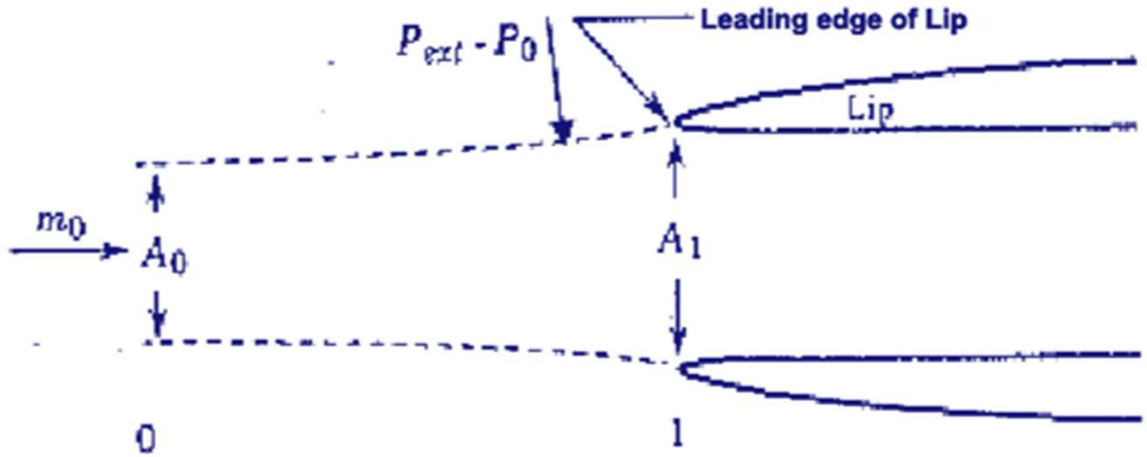
AE 658

Powerplant Component Sizing and Matching

Subsonic Intake matching



There is a positive drag acting on the stream tube which encloses the apporach air entering the engine intake (Fig.). This is known is "additive drag" (D_{add}).

$$D_{add} = \int_0^1 (P_{ext} - P_0) dA$$

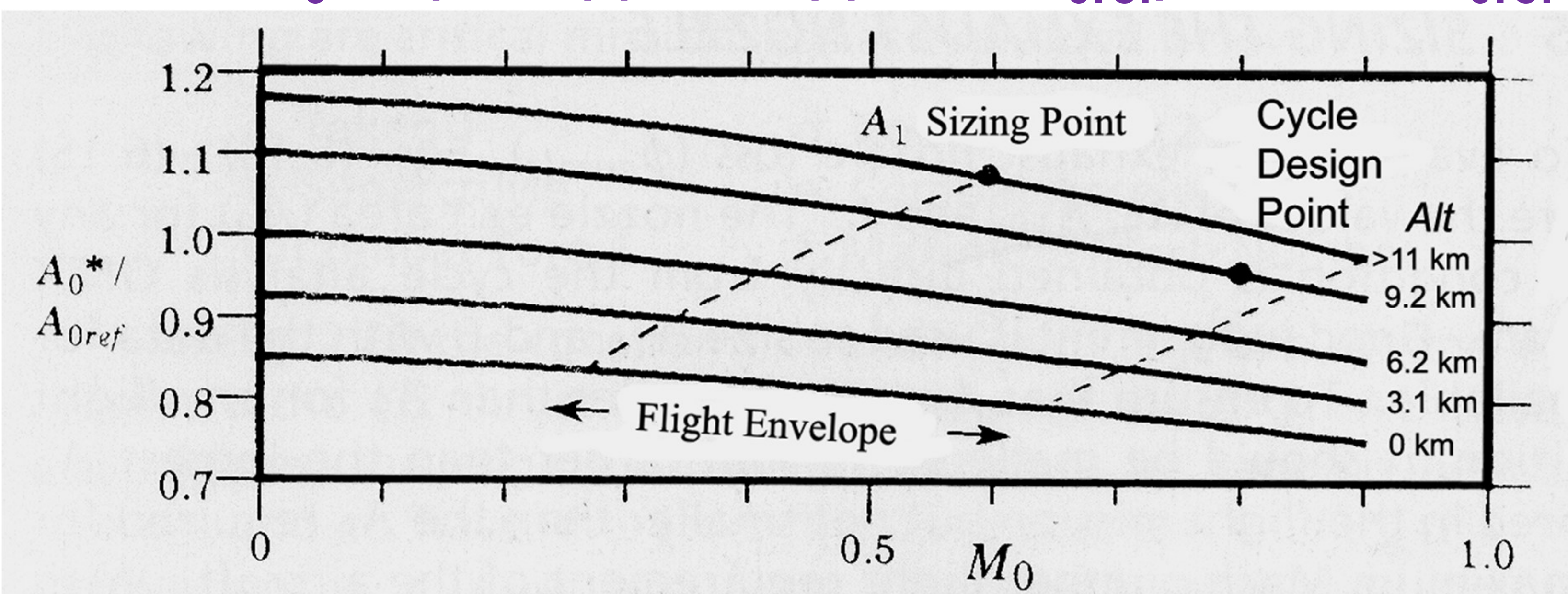
$$D_{add} = P_1 A_1 (1 + \gamma M_1^2) - P_0 A_0 \left(\frac{A_1}{A_0} + \gamma M_0^2 \right)$$

Subsonic Intake sizing

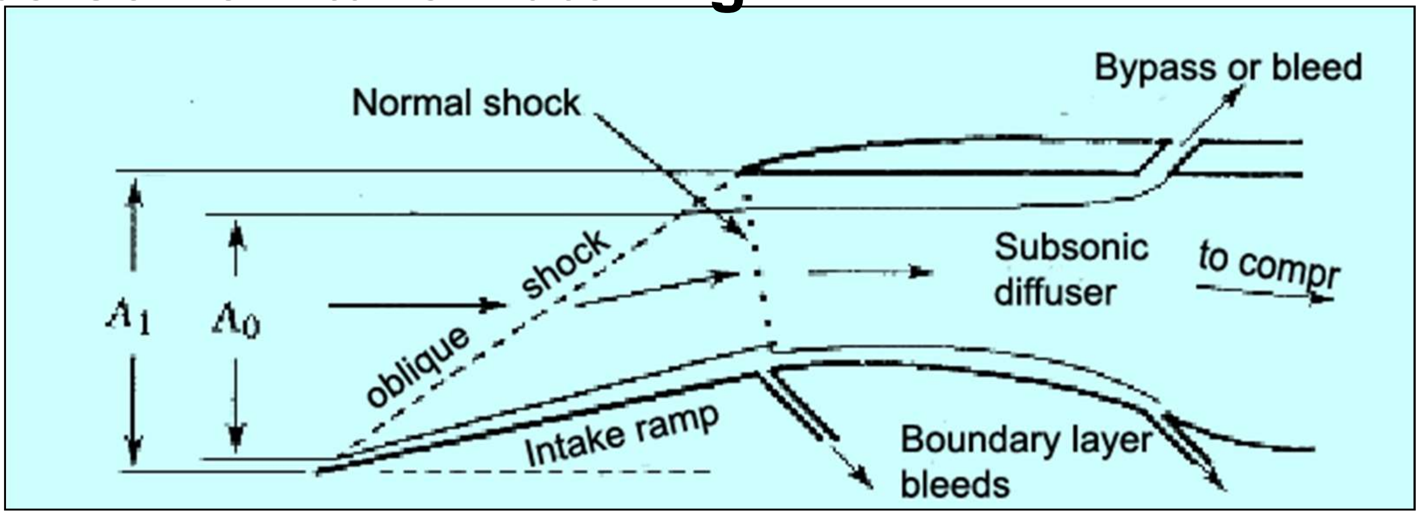
In order to prevent choking of the inlet, the inlet capture area (A_1) must be slightly larger than the area required for choking the engine flow, $A_1^*=A_0^*$. In addition to sizing the inlet for M_0= 0.8 or less to allow for boundary layer displacement, a safety margin of 4-5% is provided for any aerodynamic or mechanical effects that may restrict the flow downstream of inlet plane. Therefore, sizing A_1 for M_1=0.8 plus a 1.04 safety factor gives :

$$A_1 = 1.04(A_1/A_1^*)\ M_0=0.8$$

$$A_0^* = (1.04)(1.038)(1.07\ A_{0ref}) = 1.16\ A_{0ref}$$



Supersonic Intake Matching

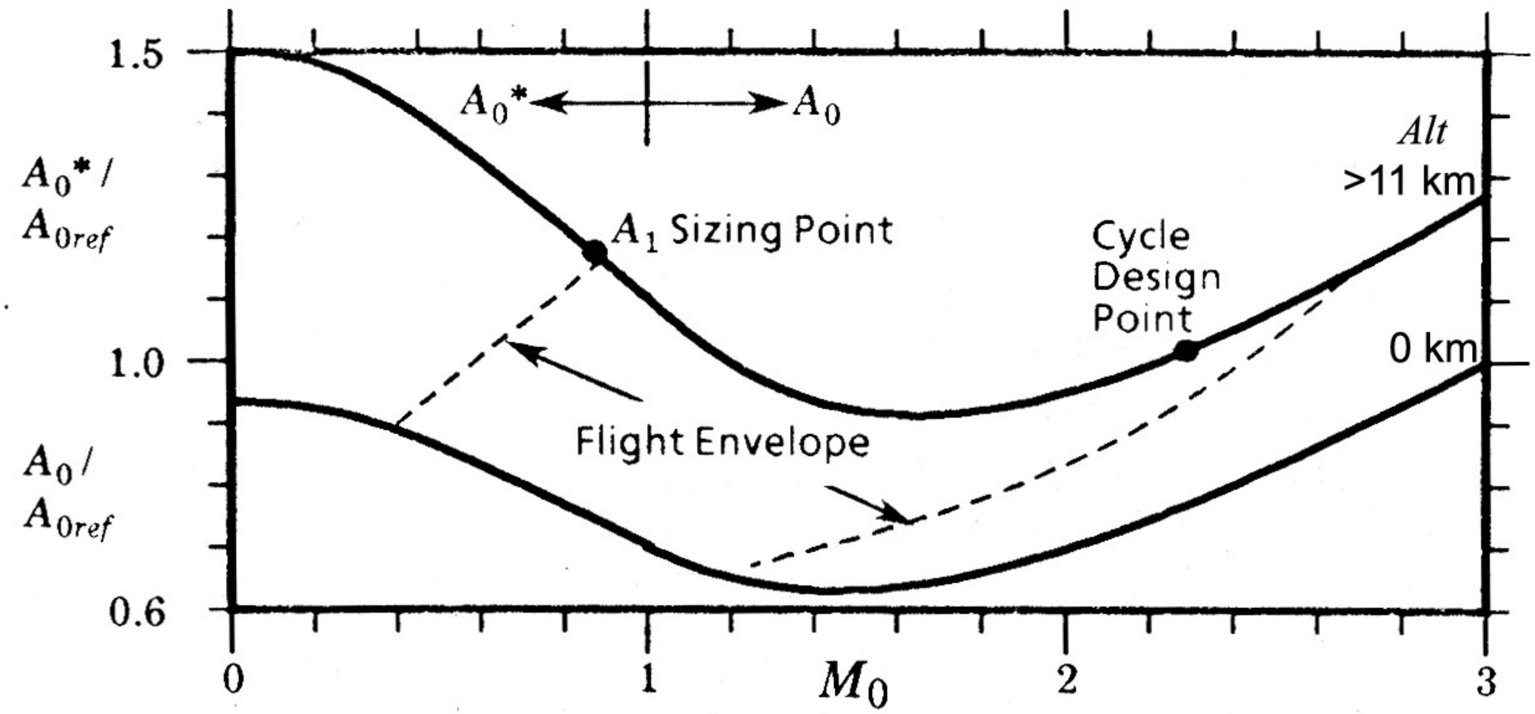


The supersonic inlet drag is estimated to equal the momentum change of the bypass and the bleed flows. The bleeds may be assumed to be under chocked flow condition

$$\Phi_{inlet} = \frac{D_{add}}{F\ (Thrust)} = \frac{\left(A_1 / A_0 - 1\right)\left(M_0 - \sqrt{\frac{2}{\gamma+1} + \frac{(\gamma - 1)M_0^2}{\gamma+1}}\right)}{\left(F.g\ / \dot{m}_0 a_0\right)}$$

This equation can be used at any given flight condition (i.e. a_0 and M_0) and engine power setting (i.e. A_0 and $F.g_c/m_0$). Note that ϕ_{inlet} approaches zero when M_0 approaches unity and when A_0 approaches A_1 (sizing point), so that ϕ_{inlet} matter only when it is evaluated far from both the conditions.

Supersonic Intake sizing

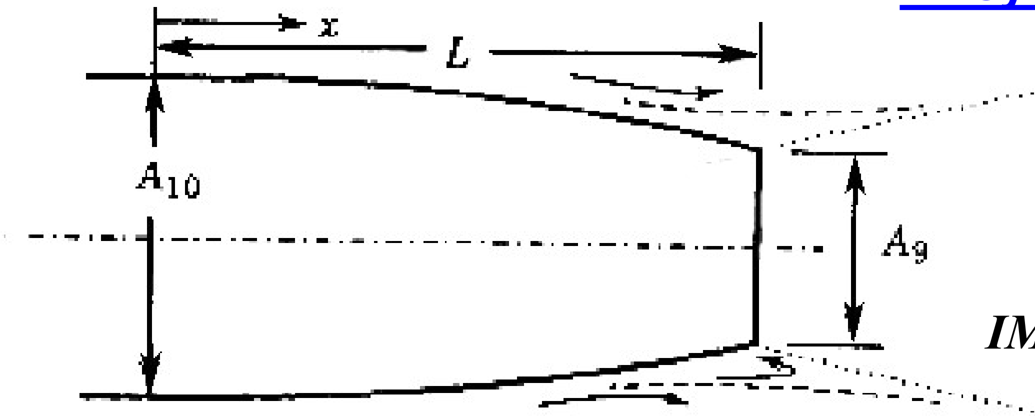


Intake capture area (A_1) must exceed the largest required engine air flow area at free stream conditions (A_0) by the minimum amount of boundary layer bleed and margin of safety

Please note that ϕ_{inlet} & ϕ_{nozzle} will vary with flight condition and throttle setting for any given engine, and may vary from aircraft to aircraft slightly even for the same engine.

The *installed thrust,* T, must therefore be the sum of *uninstalled thrust,* F and *self-drag,* D_{eng}, which is to be determined along with the engine size, or uninstalled thrust estimation. Since the very presence of the engine, its inlet, nozzle and exhaust stream actually influence the flow and pressure distribution over the entire aircraft, it is often difficult to estimate D_{eng} separately from $D_{a/c}$.

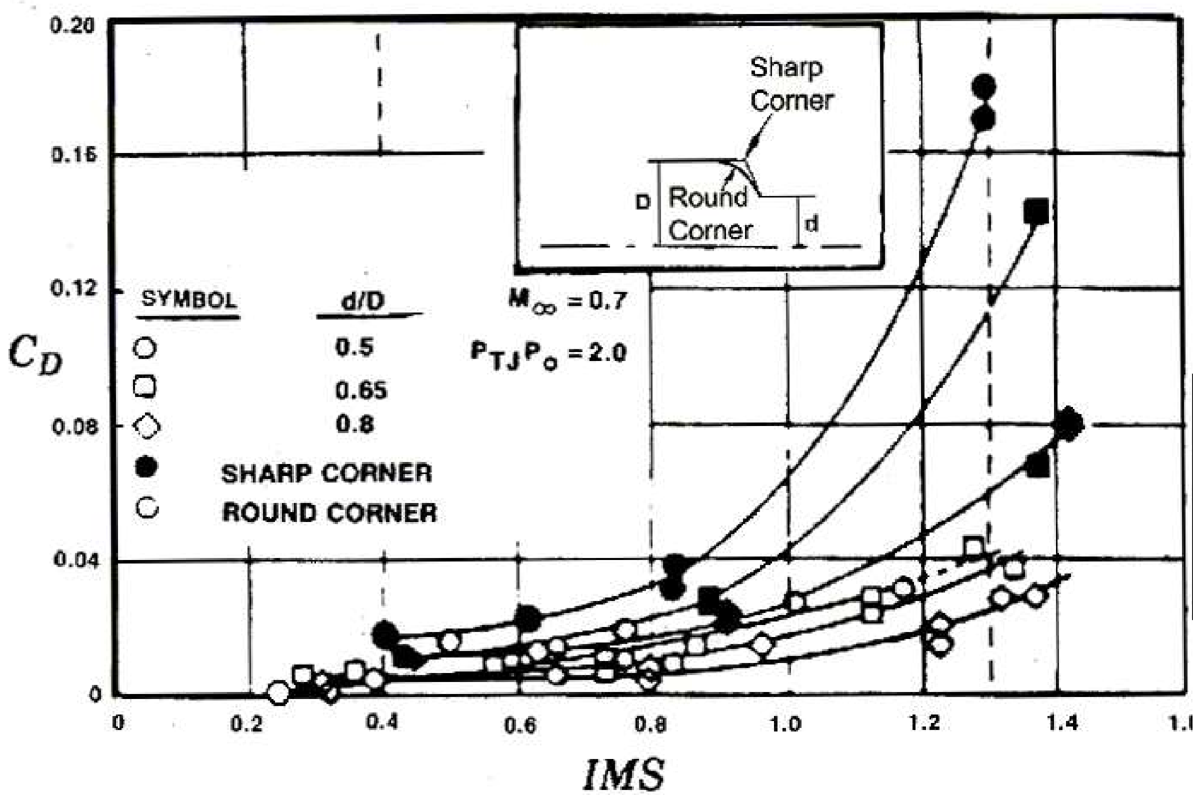
Exhaust Nozzle Drag



Axisymmetric Exhaust Nozzle Model

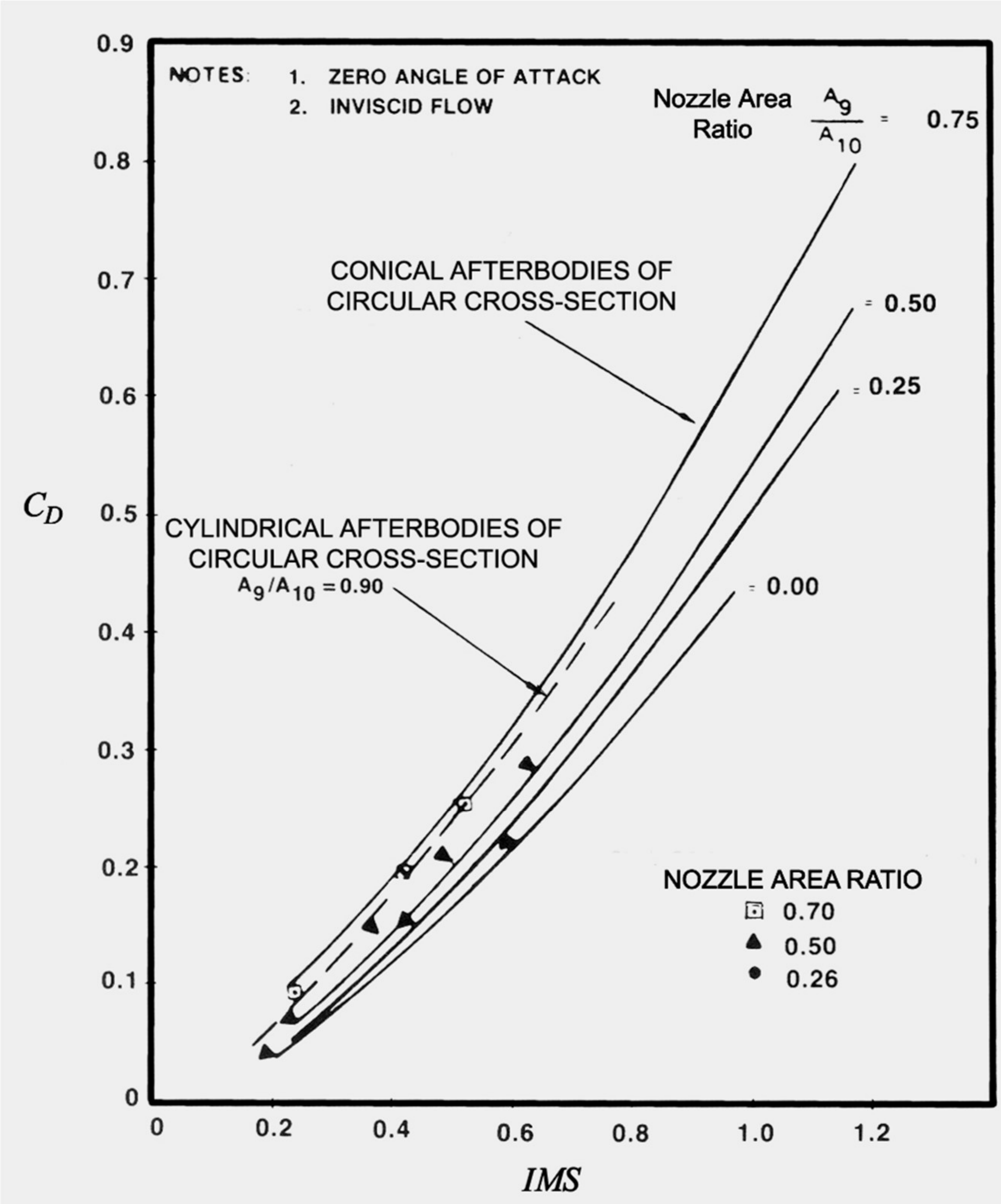
"***integral mean slope***"
is defined by

$$IMS = \frac{1}{\left(1 - \frac{A_9}{A_{10}}\right)} \int_{1}^{\frac{A_9}{A_{10}}} \frac{d\left(\frac{A}{A_{10}}\right)}{d\left(\frac{x}{R_{10} - R_9}\right)} d\left(\frac{A}{A_{10}}\right)$$



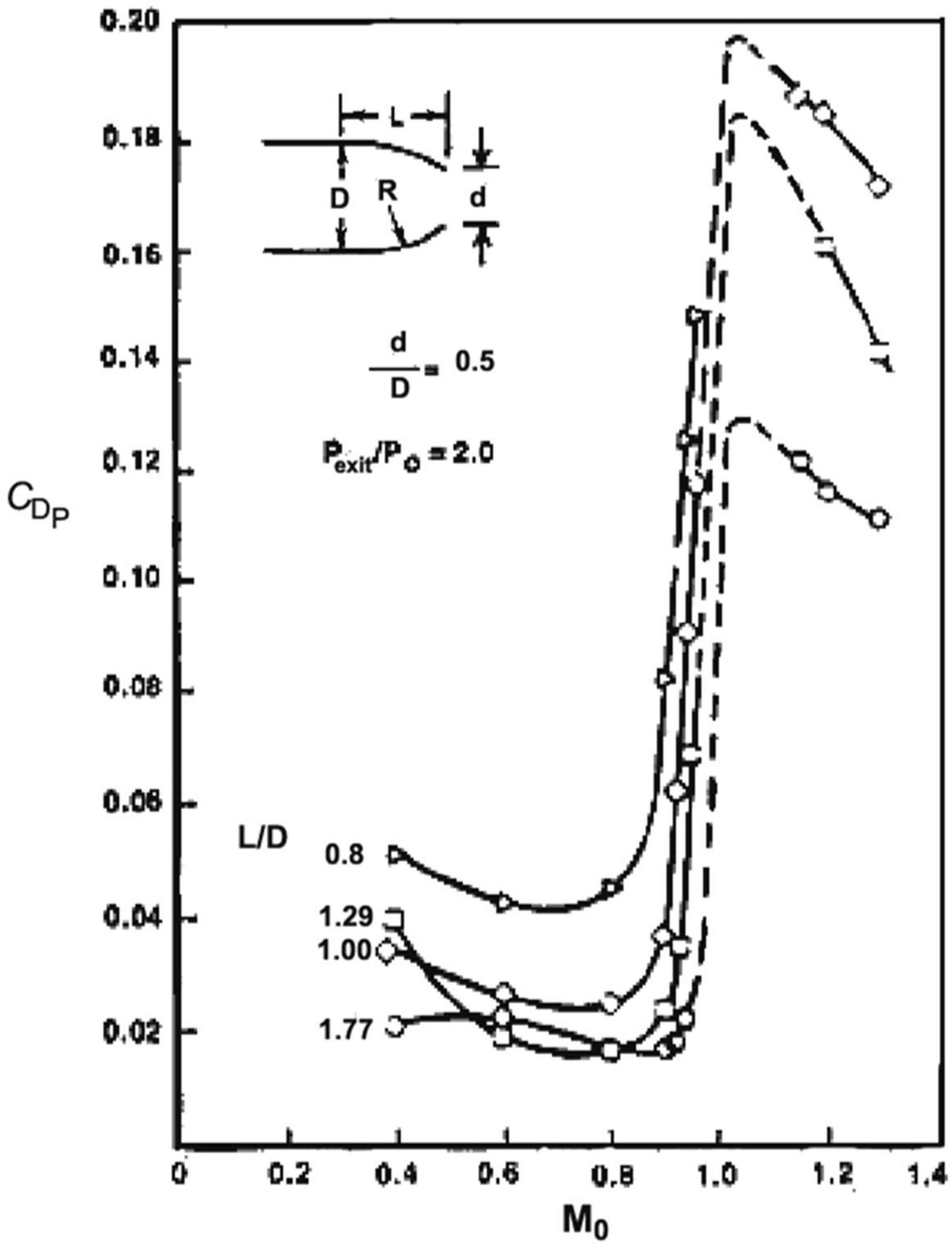
Boat-tail Pressure Drag Coefficient

when $0 < M_0 < 0.8$



IMS Correlation and Theoretical Wave Drag for Isolated Axisymmetric Afterbodies –

Mach 1.2 to 2.2



Pressure Drag Coefficients of Some Circular-Arc Boat-tails $0.8 < M_0 < 1$

Lacking actual contours (i.e. D vs x), it is impossible to evaluate the IMS using equation. One can attempt to progress by evaluating IMS for the general family of nozzle contours are described by:

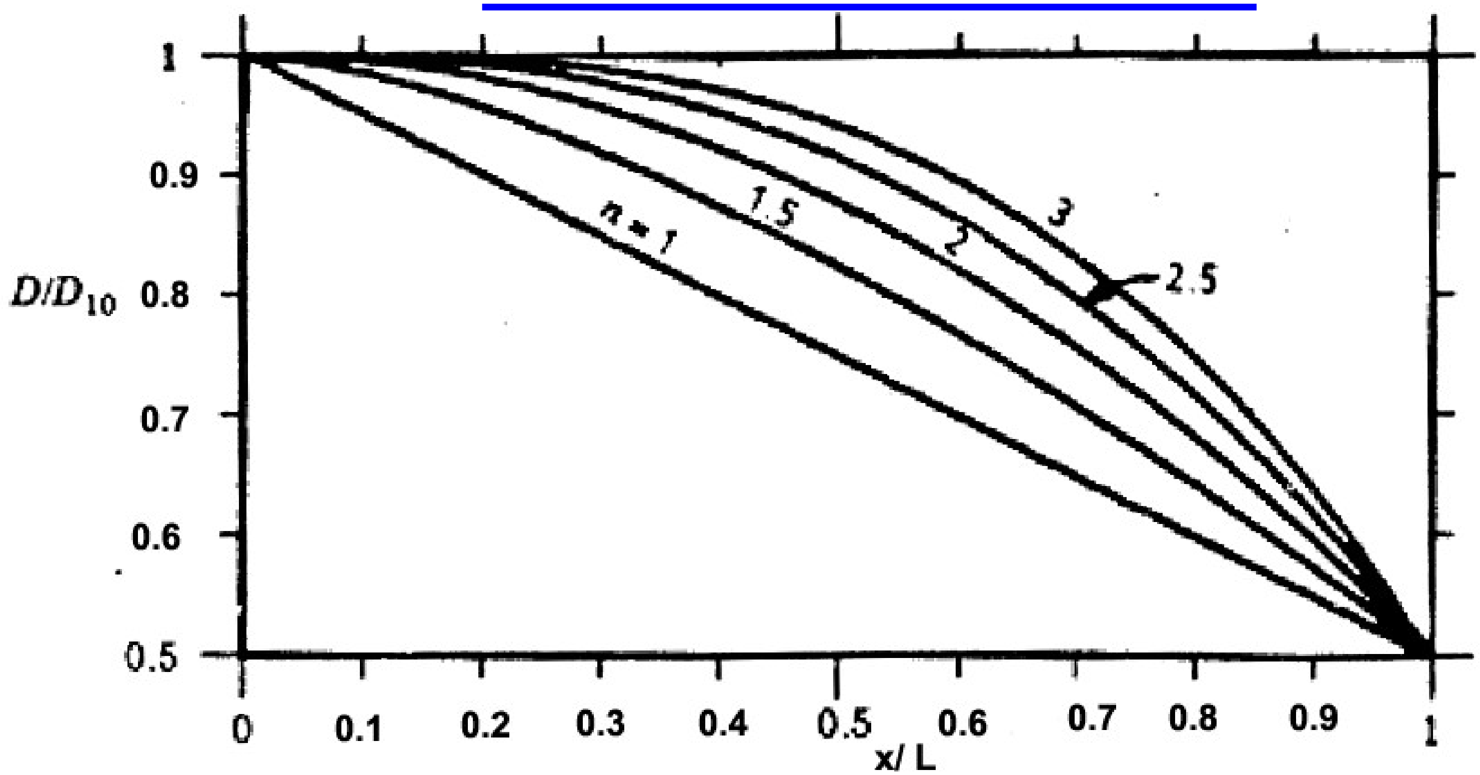
$$\frac{D}{D_{10}} = 1 - \left(1 - \frac{D_9}{D_{10}}\right)\left(\frac{x}{L}\right)^n \qquad \frac{A}{A_{10}} = \left\{1 - \left(1 - \frac{D_9}{D_{10}}\right)\left(\frac{x}{L}\right)^n\right\}^2$$

Nozzle installation penalty can be computed from

$$\Phi_{nozzle} = \frac{D_{nozzle}}{F} = \frac{q_0 C_D (A_{10} - A_9)}{\dot{m}_0 (F / \dot{m}_0)}$$

when 0.8< M0<1.2 $\quad \Phi_{nozzle} = \dfrac{M_0 \left(C_{DP} / 2\right)(A_{10} / A_0)}{F.g \ / \dot{m}_0 a_0}$

External Nozzle Contours



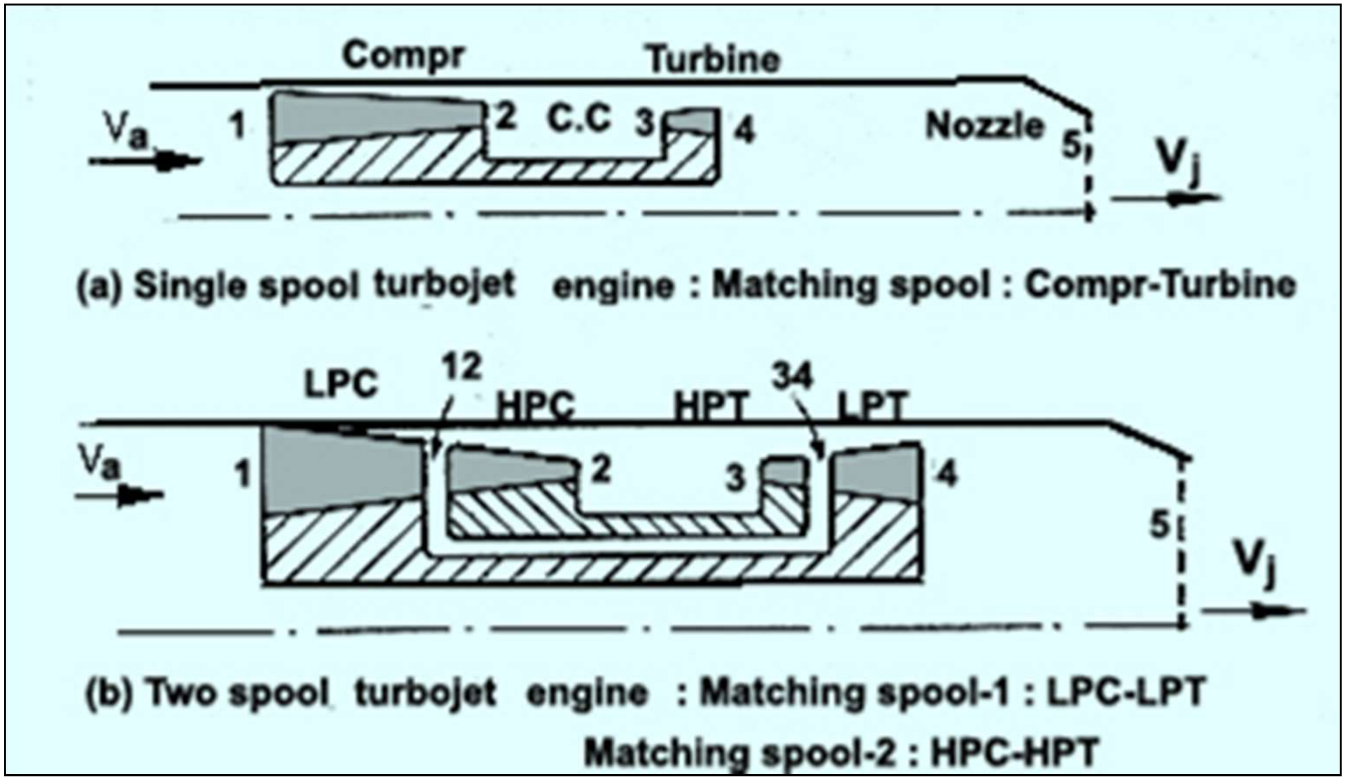
The conclusion is that one must hardly be concerned with the exact shape of the nozzle at this point in the design and that approximately

$$IMS \cong 1.8\left(\frac{D_{10} - D_9}{L}\right)\left(1 - \frac{D_9}{D_{10}}\right)$$ **will suffice**.

COMPONENT MATCHING

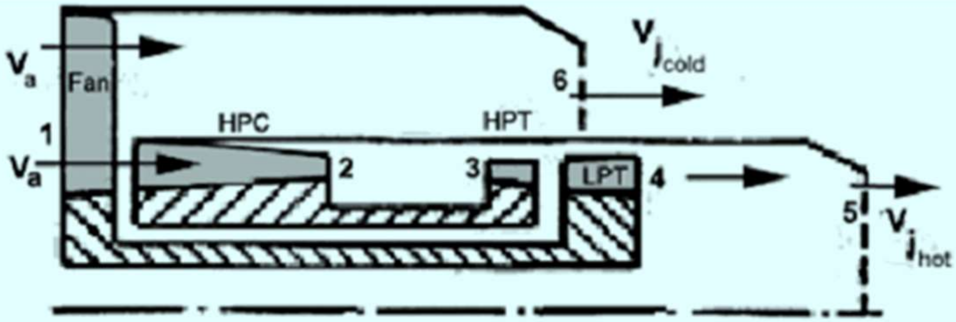
- **The *design point* of an engine is aerothermodynamically defined as the ambient operating condition (altitude, flight speed, flight condition) at which a rated cycle performance is obtainable.**
- **All the components are rendered their geometrical shape, size and structural strength to achieve the best efficiency at this operating point.**
- **The rotating components e.g. all the blades of multi-stage compressors and turbines are rendered geometric shapes to conform to the aerothermodynamics of the design point.**

- **When all the components are lined up together one after another (a gas turbine based engine) and are asked to perform, the range of possible operating conditions is considerably reduced.**
- **The off-design operation requires that for steady performance, one should be able to find corresponding operating points on the characteristics maps of all the components. Even if one component is not matched with the others the engine shall not function properly.**
- **When all the components run steadily at a matched operating condition of mass flow and rotating speed of rotating components, steady predictable performance (thrust) is obtained**

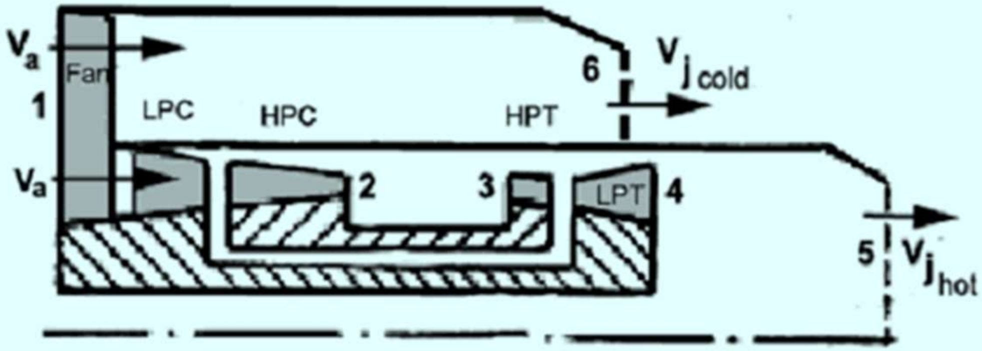


Matching assumptions

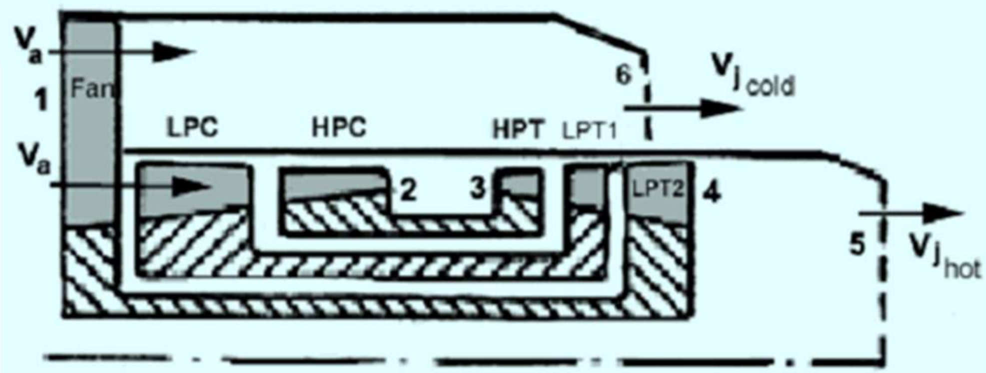
It is assumed that for most operating conditions of the engine, the propelling nozzles, - both for the core flow and for the bypass flow, are fixed at *choking (maximum flow condition*). Thus the engine responds to the inlet stagnation conditions but is 'unaware' of the forward speed, and is 'unresponsive' to the back pressure changes.



c) 2-spool engine matching- spool1 - Fan + LPT, spool2- HPC + HPT



d) 2-spool engine matching - spool 1 - Fan + LPC + LPT .
spool 2 - HPC + HPT



e) 3-spool engine matching - spool1 - Fan + LPT2,
spool 2 - LPC + LPT1 , spool 3 - HPC + HPT

In non-dimensional terms the combination of the engine parameters is

$$\frac{m_a V_j + p_j A_N}{D^2 P_{01}} = f_0 \left\{ \frac{N.D}{\sqrt{\gamma R T_{01}}} \right\}$$

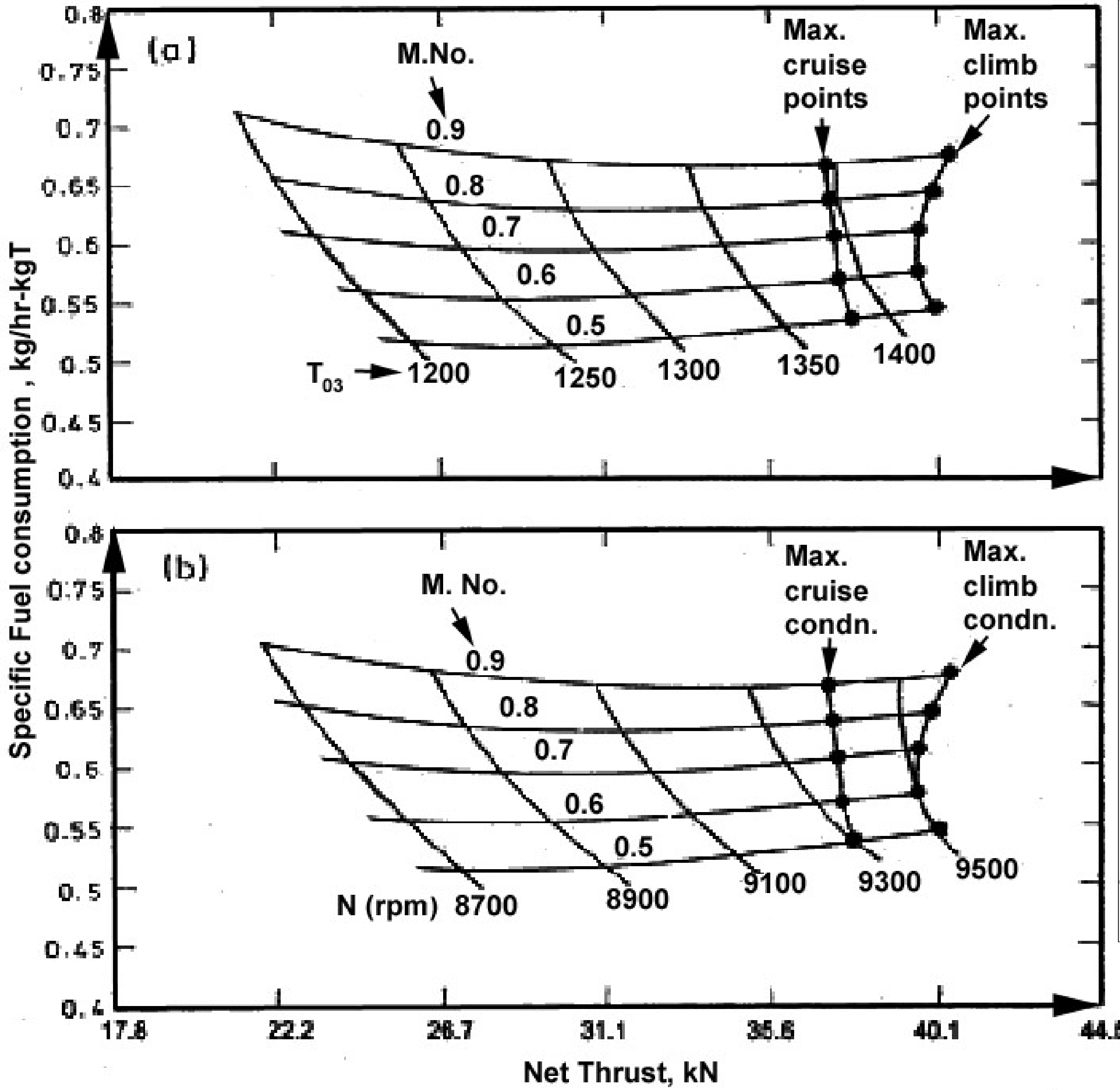
or
$$= f_1 \left\{ \frac{T_{03}}{T_{01}} \right\}$$

or
$$= f_2 \left\{ \frac{m_f \times \dot{Q}}{\sqrt{C_p T_{01}}.D^2 p_{01}} \right\}$$

Hence, in terms of gross thrust,

$$\frac{F_G + p_a A_N}{D^2 P_{01}} = f_{00} \left\{ \frac{N.D}{\sqrt{\gamma R T_{01}}} \right\}$$

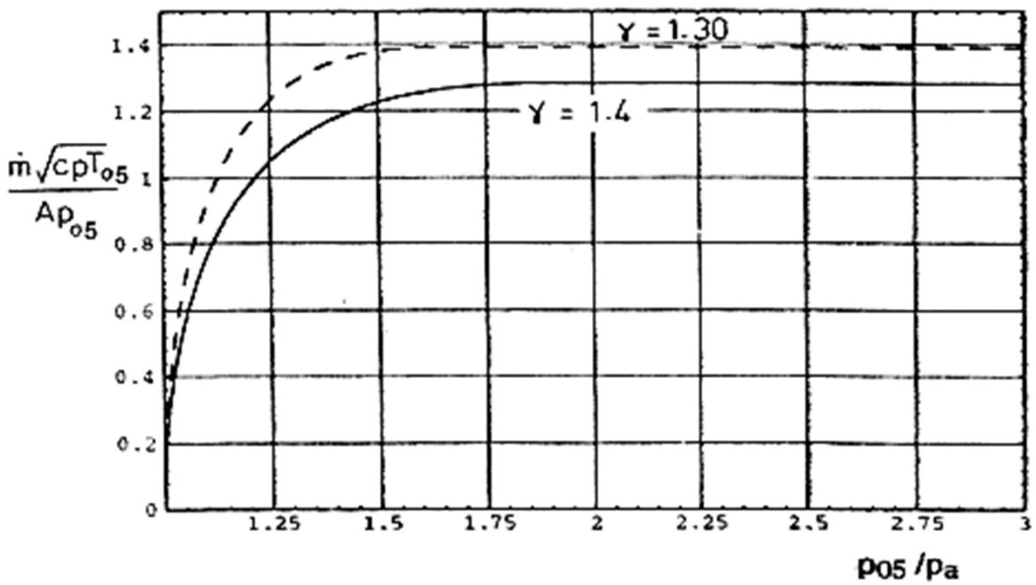
$$= f_{11} \left\{ \frac{T_{03}}{T_{01}} \right\} \qquad = f_{22} \left\{ \frac{\dot{m}_f \times \dot{Q}}{\sqrt{C_p T_{02}}.D^2 p_{02}} \right\}$$



Performance maps showing functional dependencies –

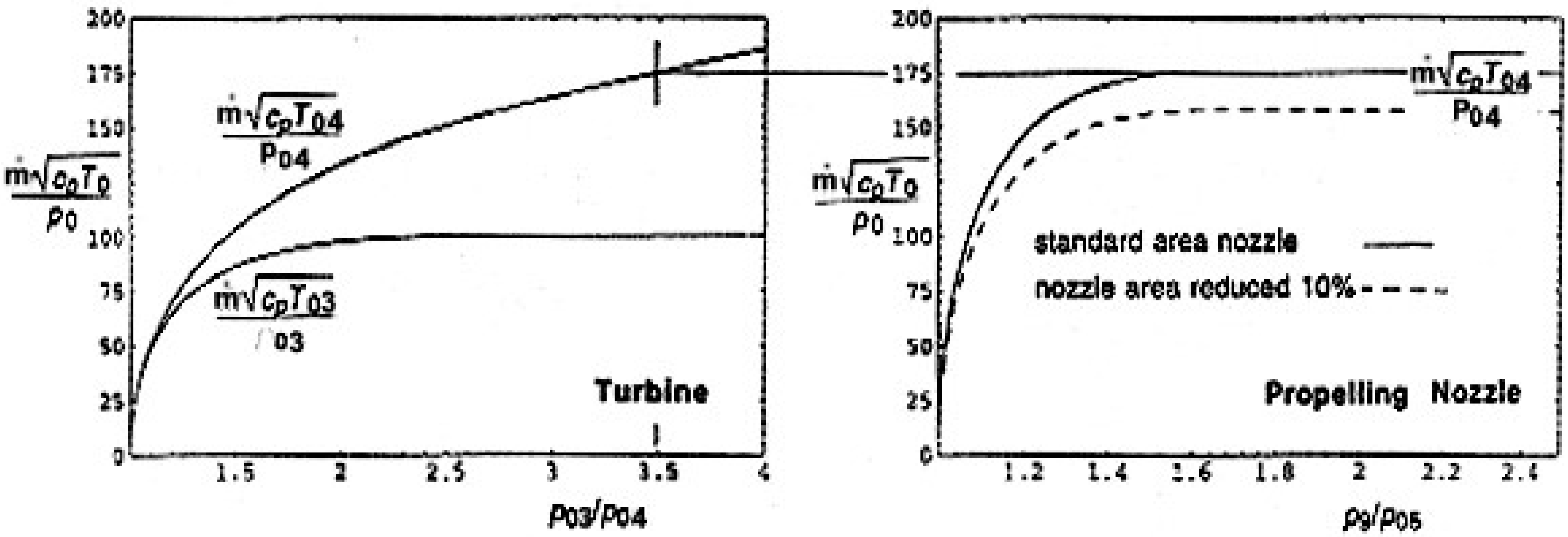
(a) variation in terms of turbine inlet temperature,

(b) in terms of HP spool rotational speed

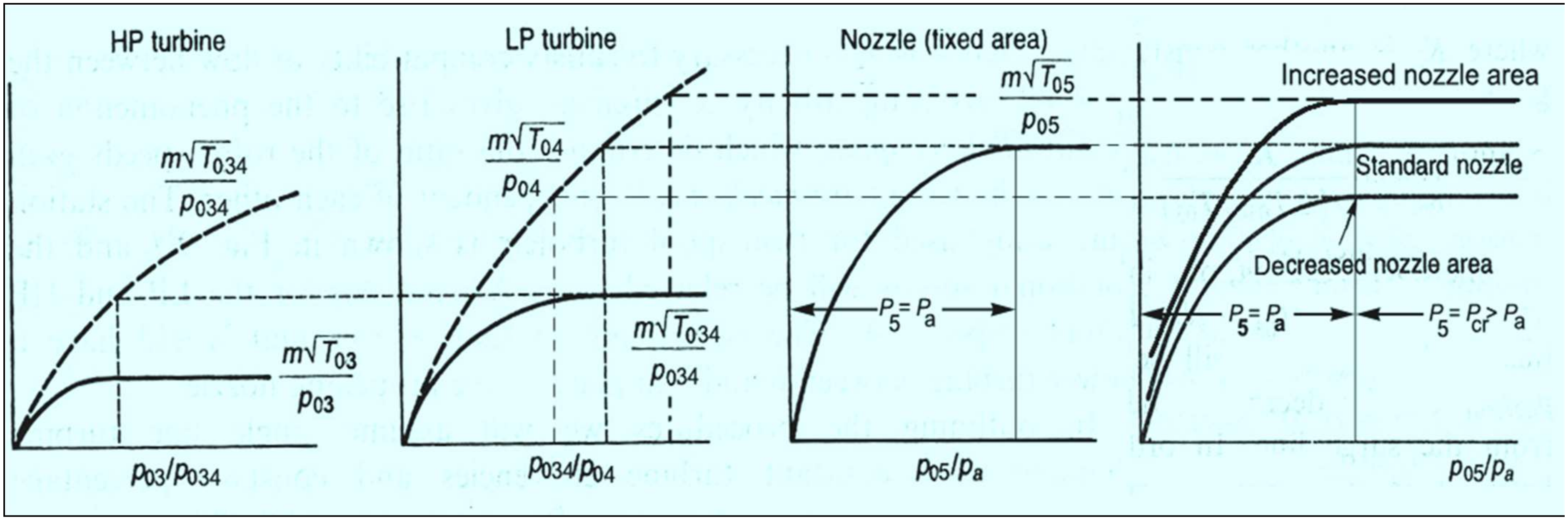


Non-dimensional mass flow variation through a convergent nozzle

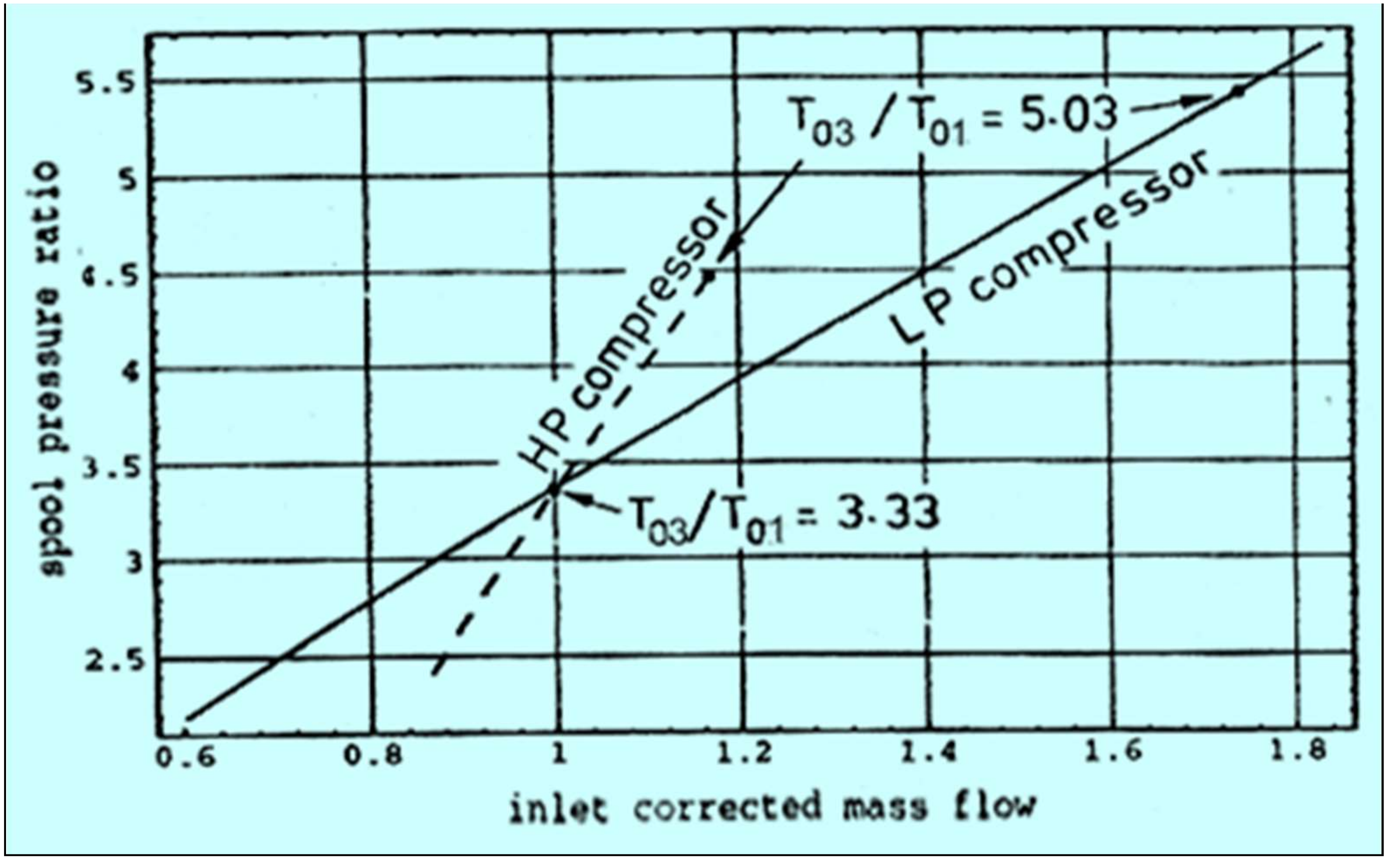
Turbine and Nozzle matching



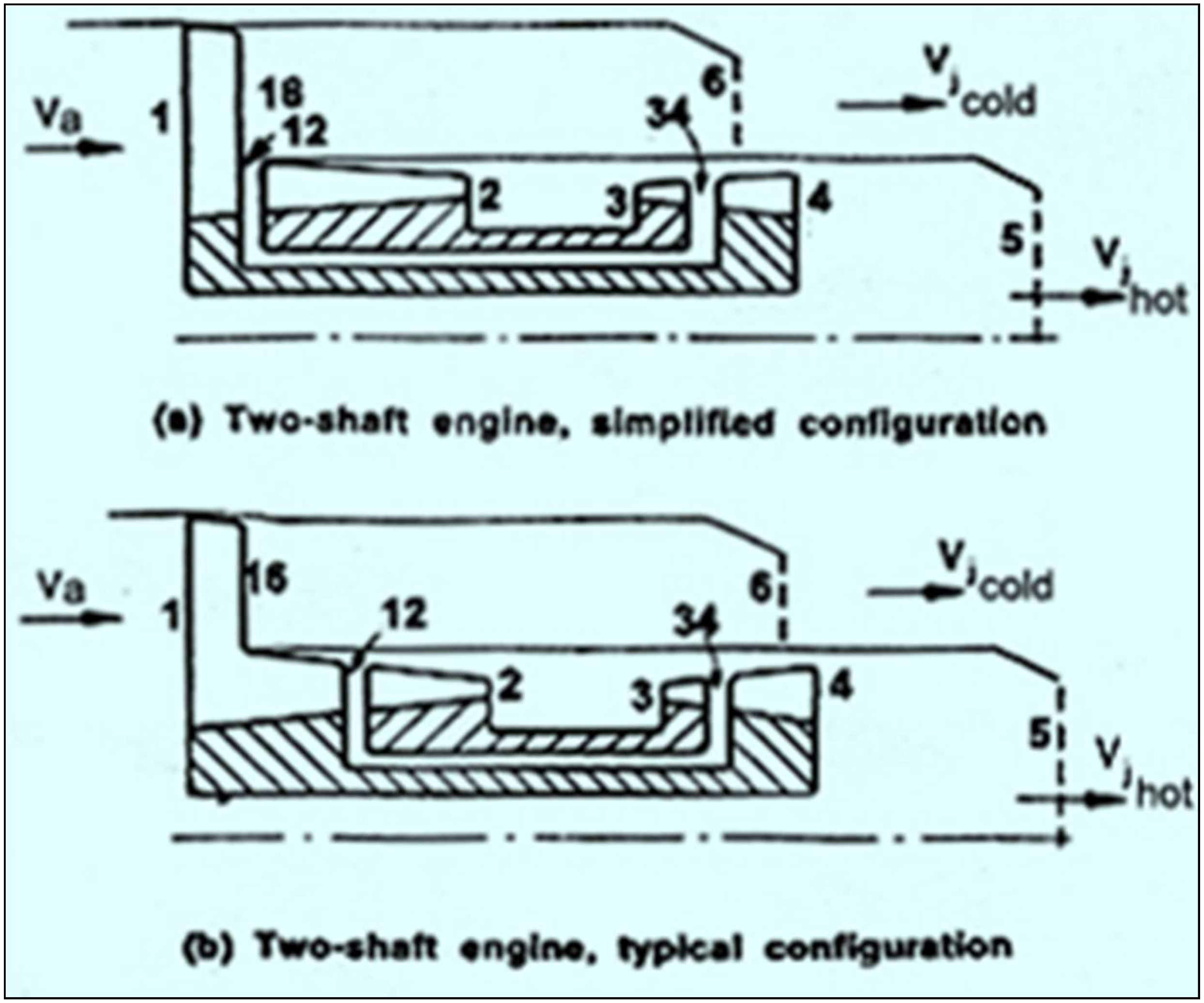
Matching of a turbine with a downstream propelling nozzle



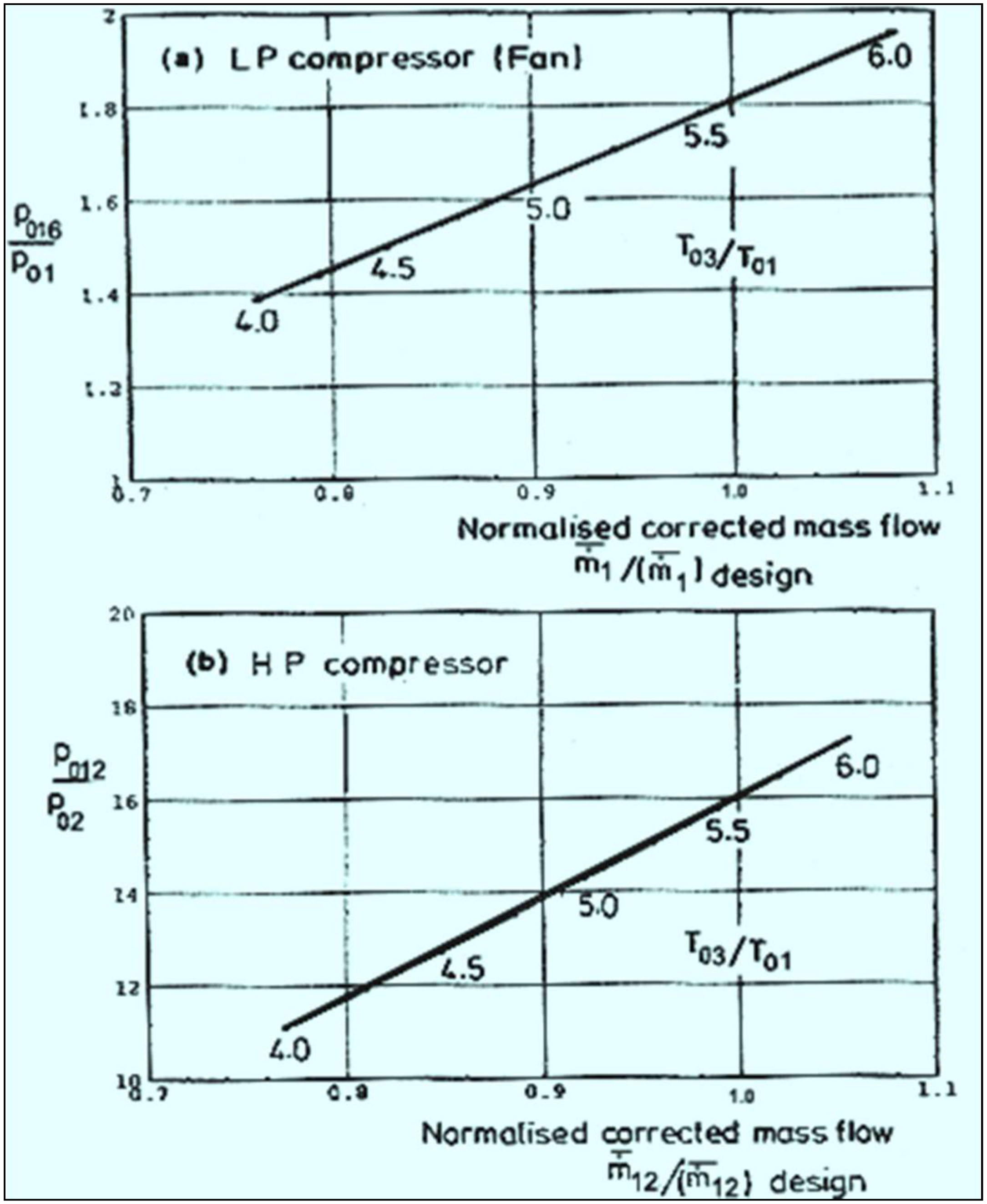
HP and LP turbine matching with fixed and variable area nozzles



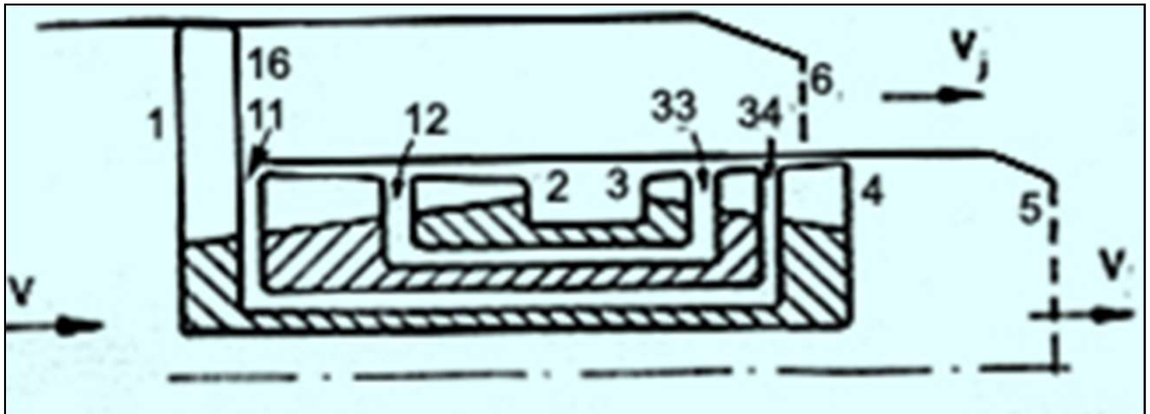
The primary means of avoiding surge line is to mount HP and LP compressors on separate shafts which rotate at different speeds. The compressors are then run at speeds at which they can meet the requirement for the safe operational non-dimensional mass flow and pressure ratio. LP compressor shows a much greater variation in mass flow than the corresponding HP compressor of the engine. Because of this the LP spool speed variation is very much greater, while HP spool operates steadily at a relatively higher speed..



Engine station numbering for high bypass engines



The operating lines for the (a) LP compressor (fan) and (b) HP compressor, for a two-shaft engine

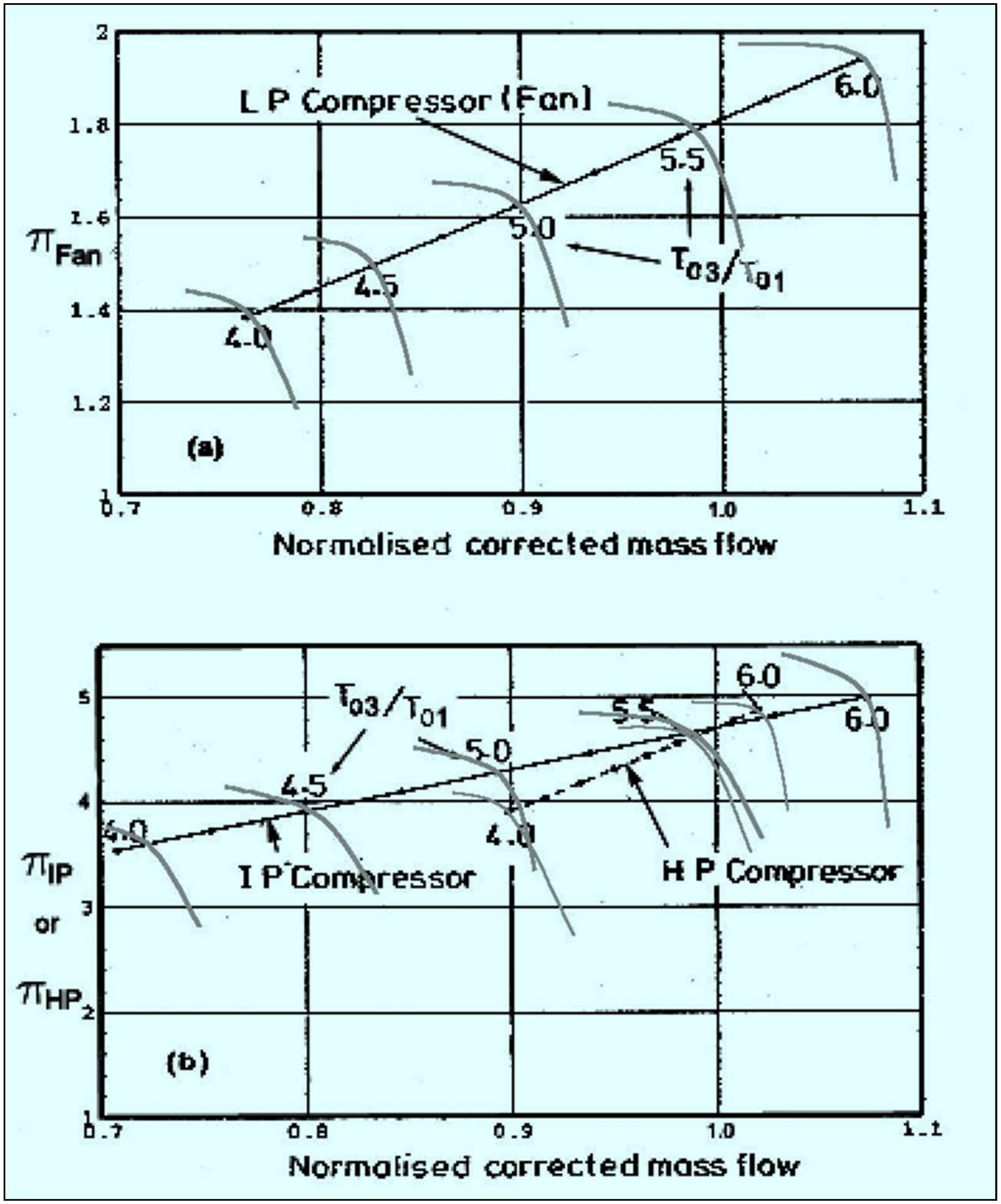


Three spool engine

Matched mass flow in IP is :

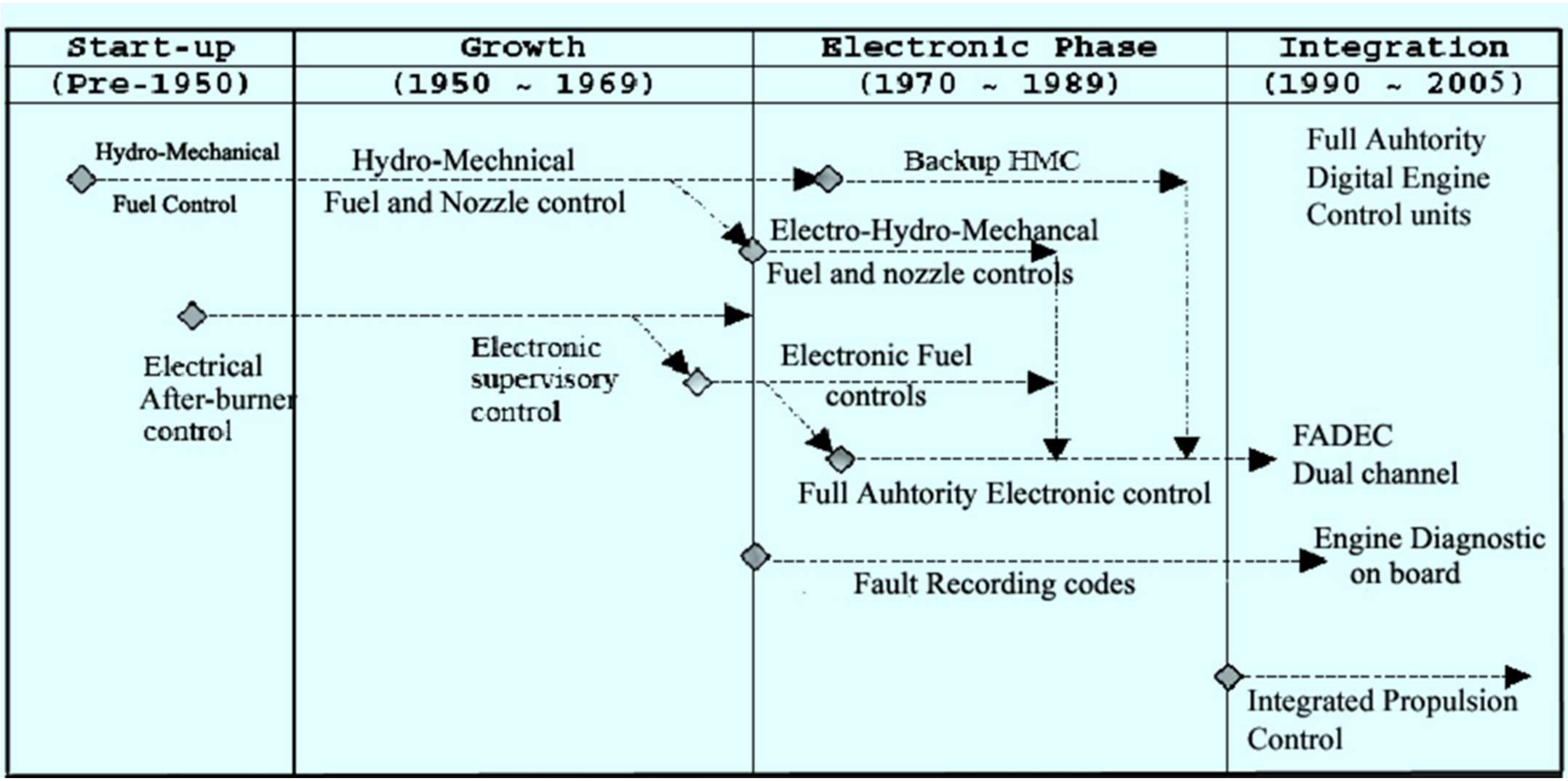
$$\overline{\dot{m}}_{air-12} = k_{IP} \frac{p_{03}}{p_{12}} \left(\frac{T_{03}}{T_{12}} \right)^{1/2}$$

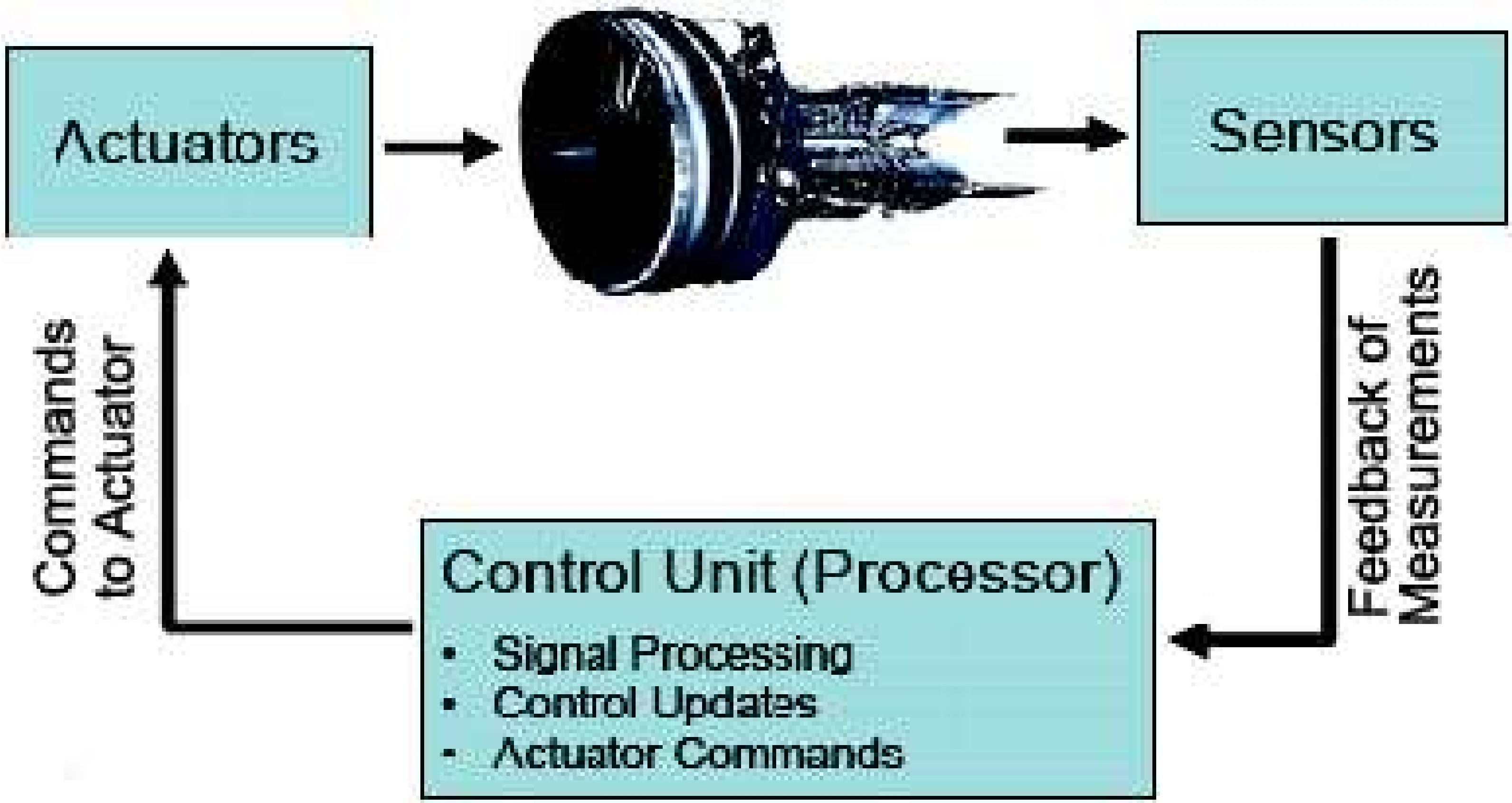
Where k_{IP} is a thermodynamic temp rise factor in IP. Similar factors may be defined for LP & HP for matching purposes.

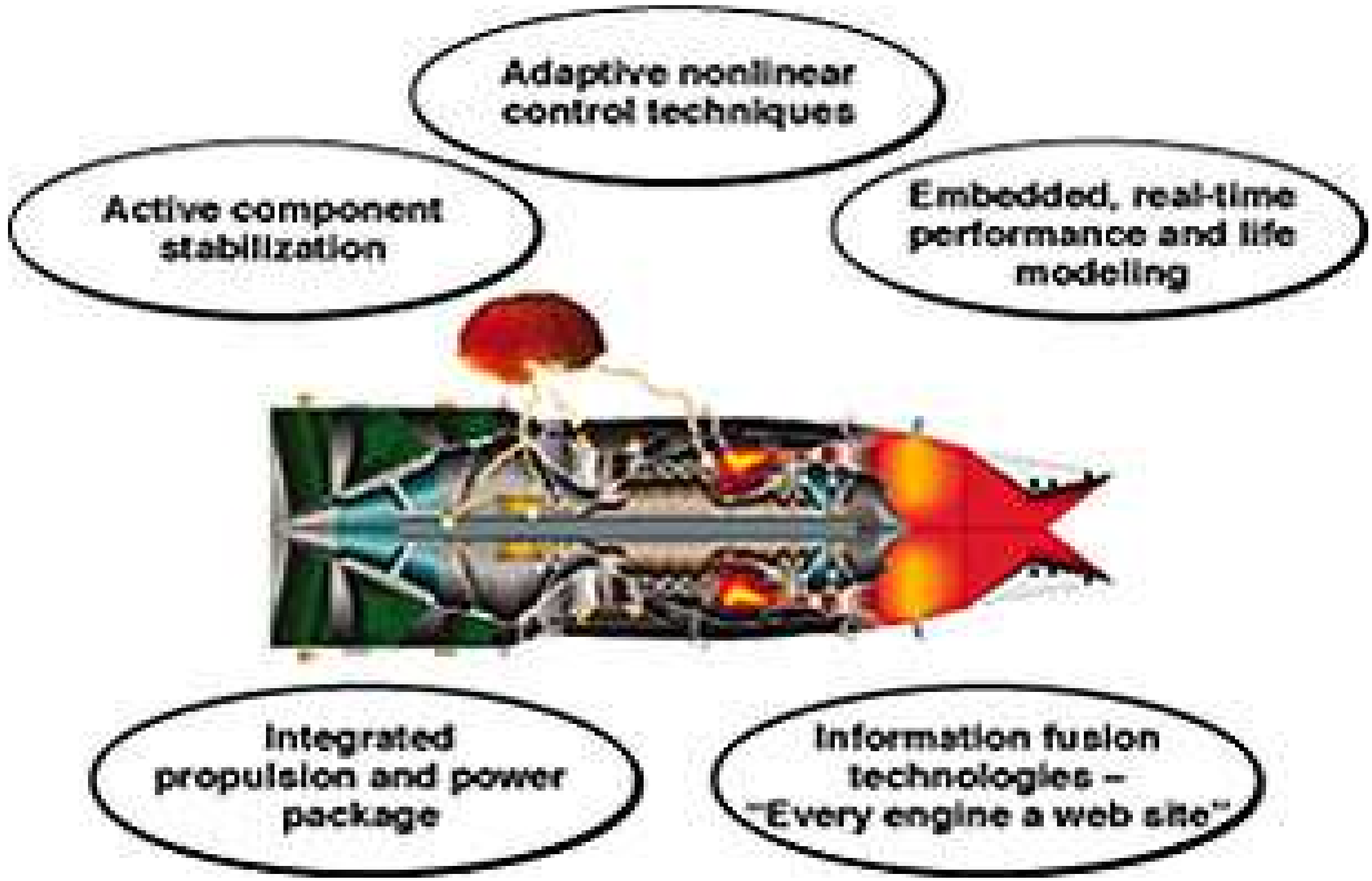


The operating lines for LP fan, IP and HP compressors for a 3-spool engine

Control of Aircraft Engines




Actuators
Sensors
Commands to Actuator
Feedback of Measurements
Control Unit (Processor)
Signal Processing
Control Updates
Actuator Commands



Intelligent Engine Technologies

FADEC is a system which controls the engine smartly for better performance. Full Authority Digital Engine Control (FADEC) is a system consisting of a digital computer (called EEC or ECU), which control all aspects of aircraft engine performance. FADEC is physically located as a "black box" on the engine casing.

The summary of its advantages is set out below:

- **Schedule Control (Operability)**
- **Multivariable Robust Control (Stability and Control)**
- **Performance Seeking Control (Efficiency and Life)**
- **Redundant Control (Reliability and Safety)**
- **Condition Monitoring (Maintainability)**