AVDASI 3
(CADE 30007)
Gas Turbine Propulsion

Week 15: Lecture 3

Off-Design Conditions (1of 2)

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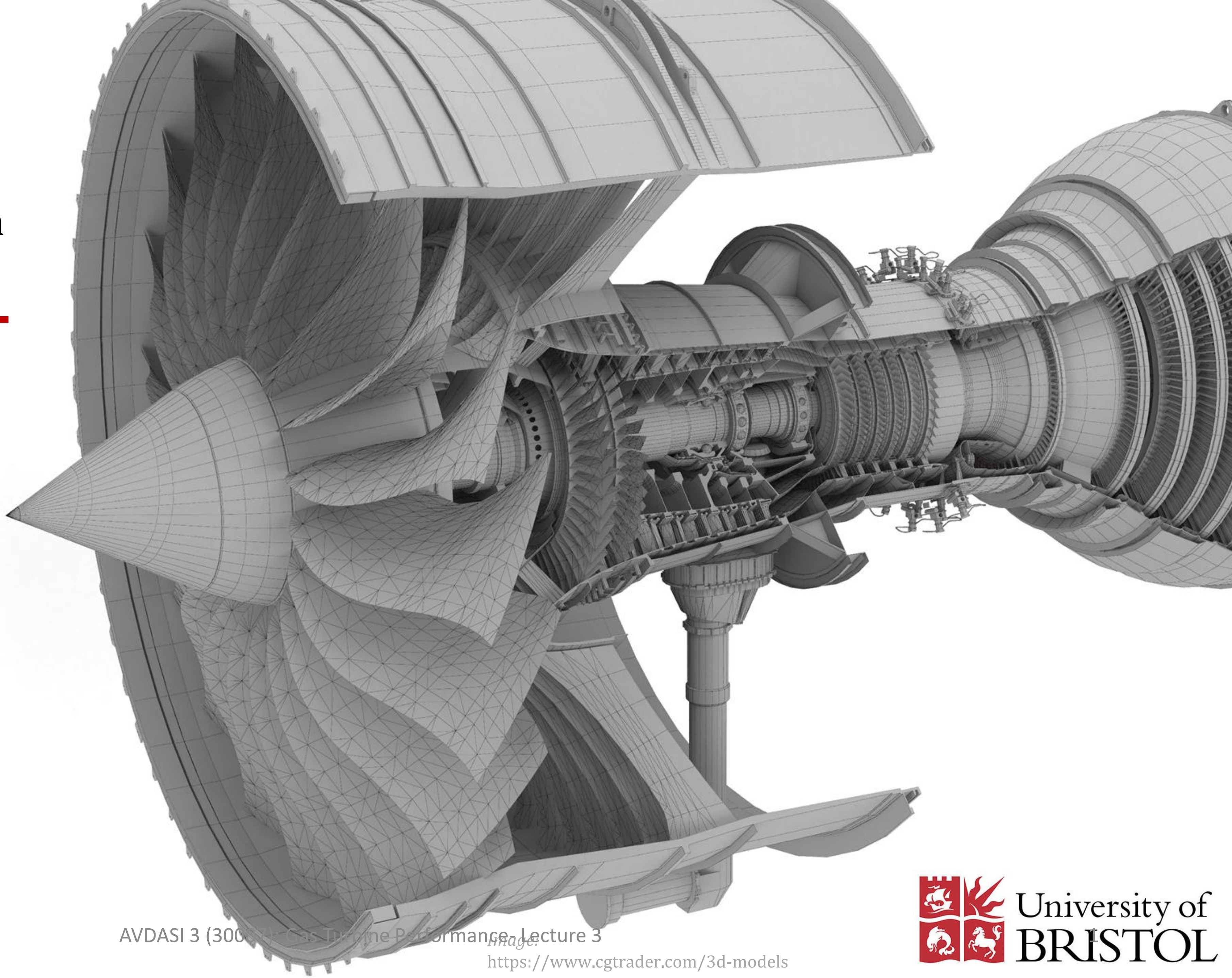

Image: https://www.cgtrader.com/3d-models

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Learning Objectives for Lecture 3

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Objective ~ Lecture 3

To outline the way that the performance of a propulsion system can be characterised

Design Point & Off Design Performance

- The Design Point of an Engine is where all of the components are matched at their *Design Conditions* - often where each unit is achieves design pressure ratio & peak efficiency at maximum flow.

  Also sometimes called *Synthesis Matching Point*

- The Altitude & Mach Number conditions are usually those which are critical for the Aircraft/Engine Requirements.

- Typically the Design Point for a Subsonic Passenger Aircraft will be at the top of Climb i.e. M = 0.7 – 0.8, 35,000 ft. – T_{01} = ~ 240K.

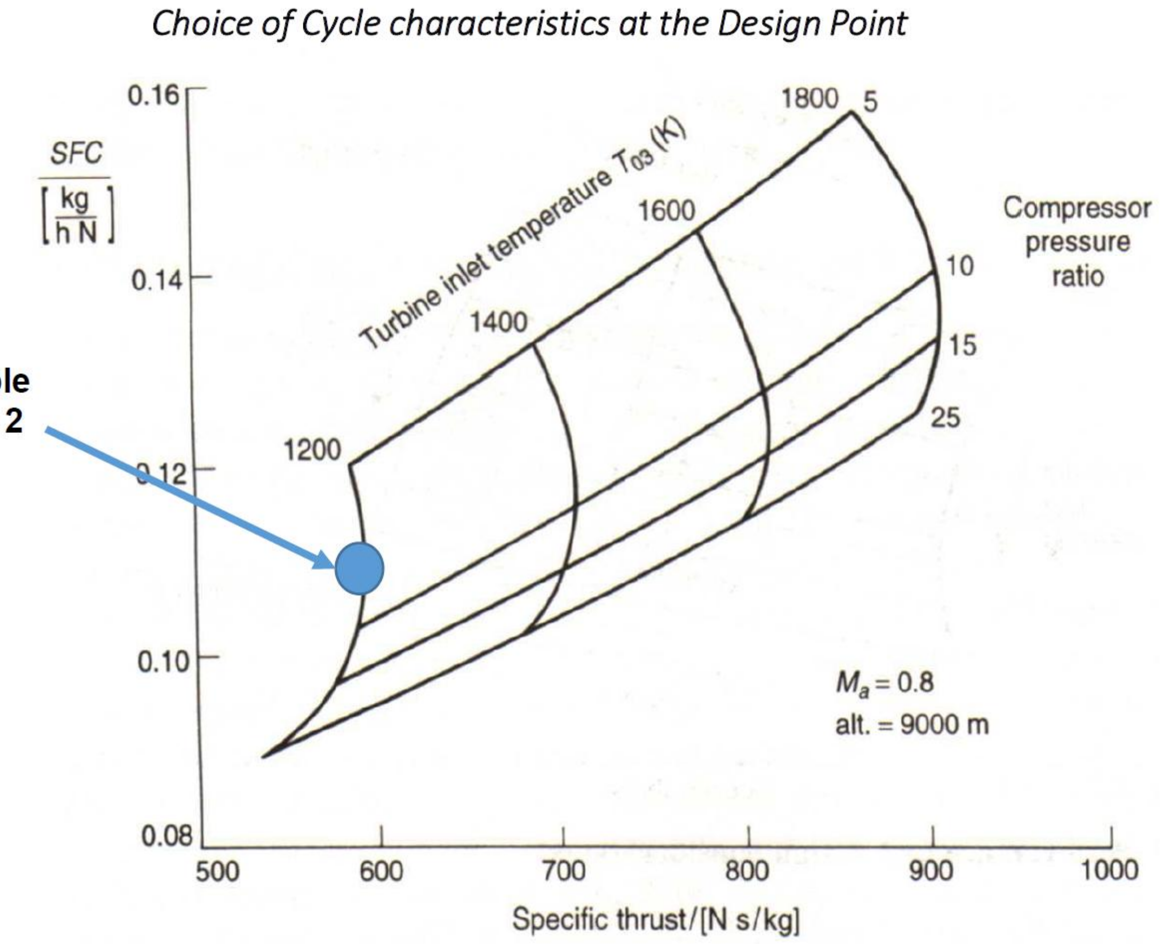
- For a Military Aircraft, the Design Point will be at the critical condition for manoeuvring in Combat i.e. M = 1.8, 40,000 ft. - T_{01} = 357K.

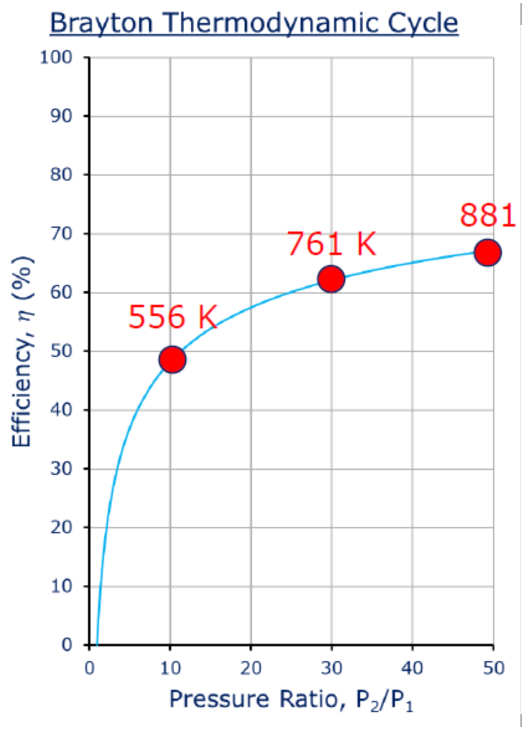
- For a Helicopter the Design Point will be at M = 0 at Sea level - T_{01} = 288 K.

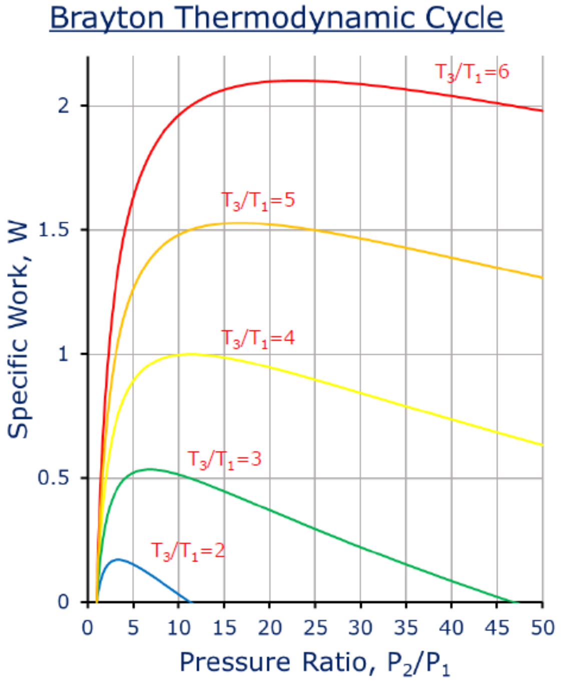
- All other conditions are *Off-Design* where the pressure ratio, efficiency and flow are different from those at the design point.

Performance of a range of Typical Turbojet Cycles

Worked Example From Lecture 2



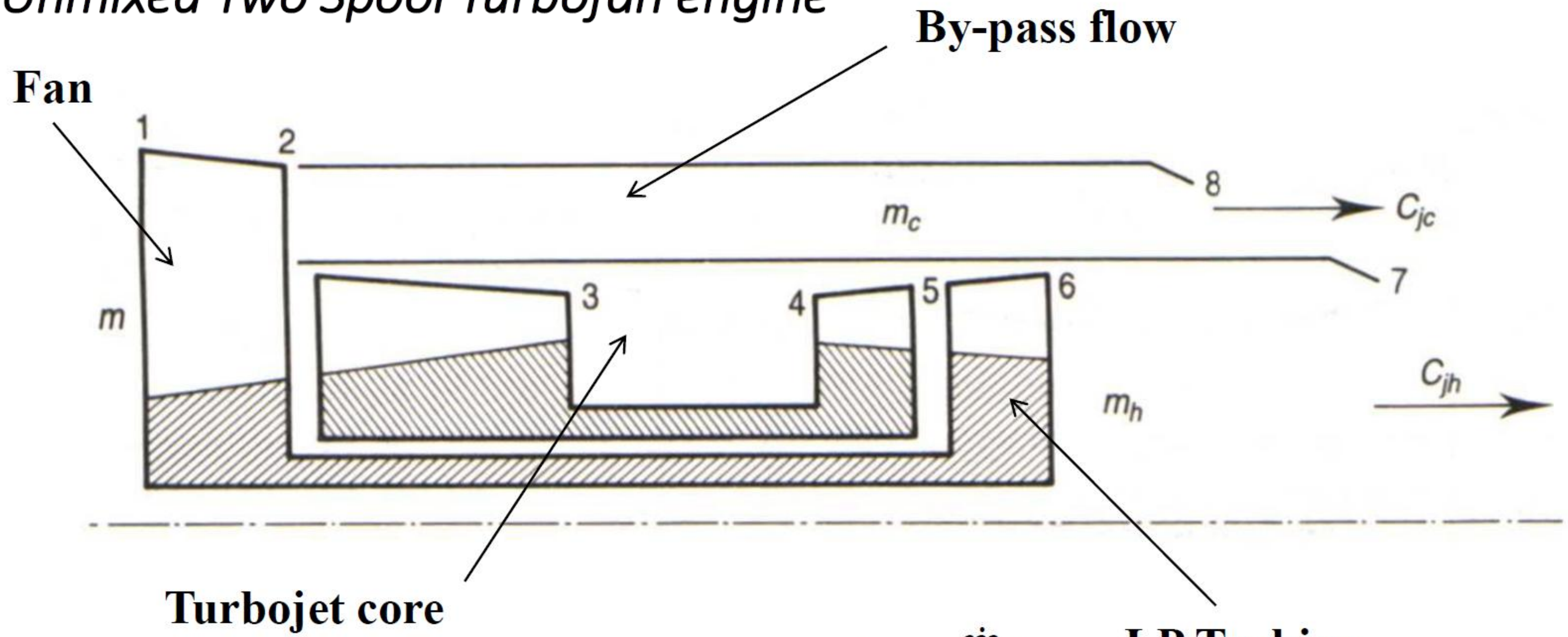




Design Point calculations ~ each point is a different engine

Most "Jet" Engines are By-pass Engines

Unmixed Two Spool Turbofan engine



Bypass ratio $\lambda = \frac{\dot{m}_c}{\dot{m}_h}$

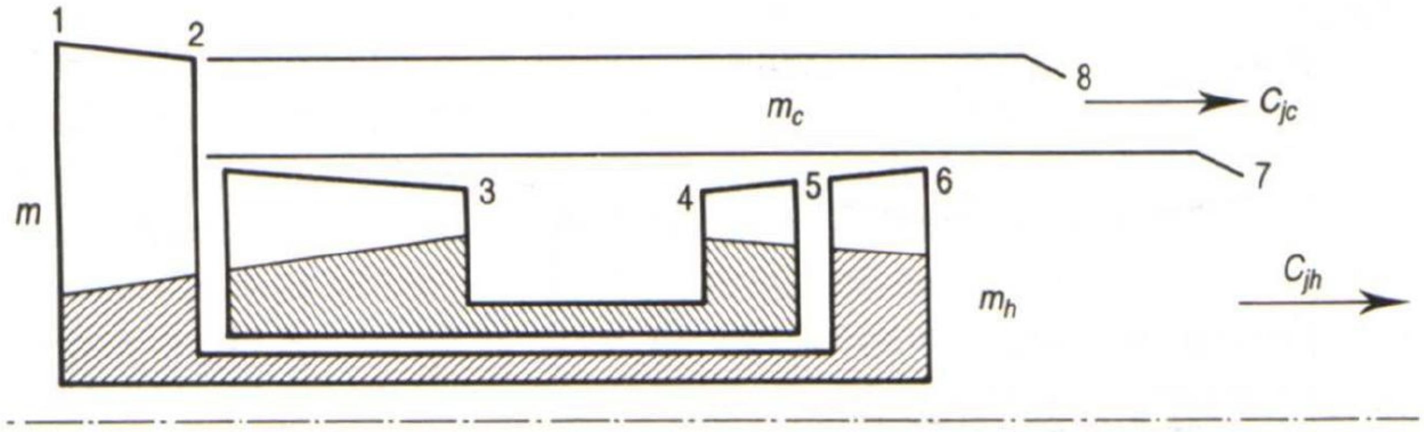
$$\dot{m} = \dot{m}_c + \dot{m}_h$$

$$F_H = \dot{m}_H(C_{JH} - C_a) + A_j(P_j - P_a)$$
$$F_C = \dot{m}_C(C_{JC} - C_a)$$
$$F = F_H + F_C$$

Most "Jet" Engines are By-pass Engines

Unmixed Two Spool Turbofan engine

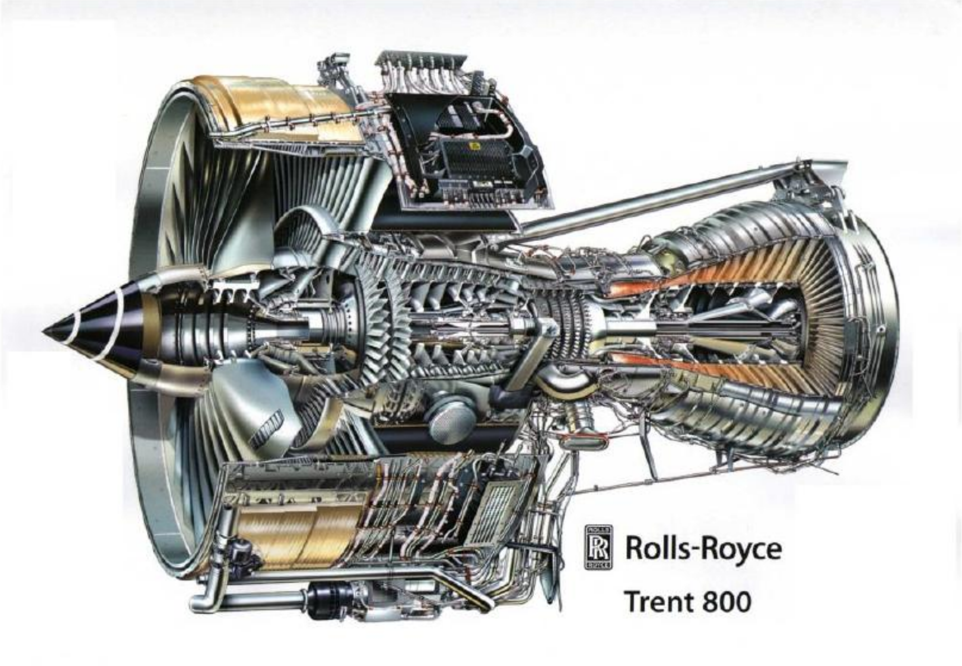
Key Cycle Parameters at the Design Point

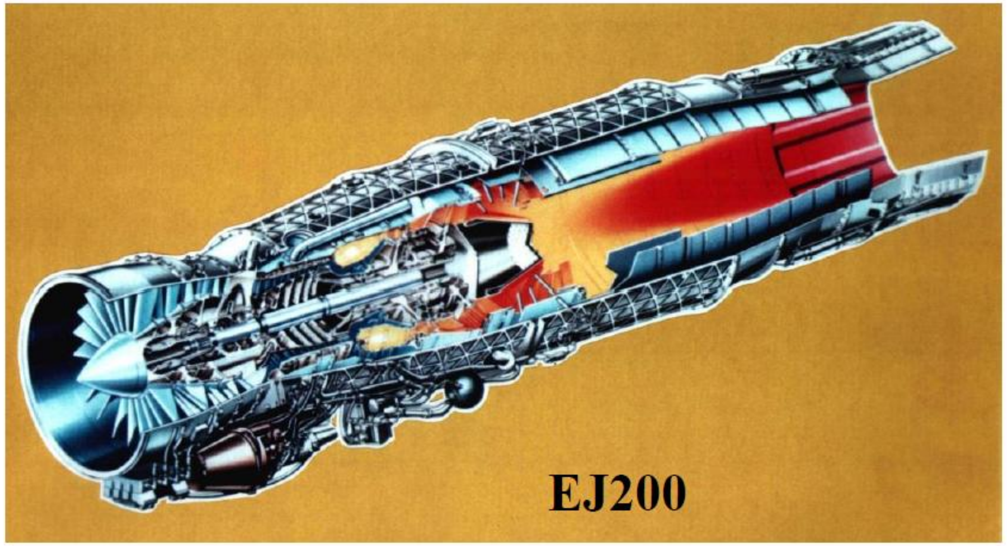


- Overall Pressure Ratio $\quad OPR = P_{03}/P_{01}$
- Fan Pressure Ratio $\quad FPR = P_{02}/P_{01}$
- Stator Outlet Temperature $\quad SOT\ K = To_4$
- Specific Thrust $\quad ST = F/\dot{m}$
- By-pass Ratio $\quad \lambda = \dfrac{m_c}{m_h}$ (by-pass flow/core flow)

Note for mixed flow engines p_{sh}/p_{sc} is also important

Propulsion Systems for Transport & Combat Aircraft





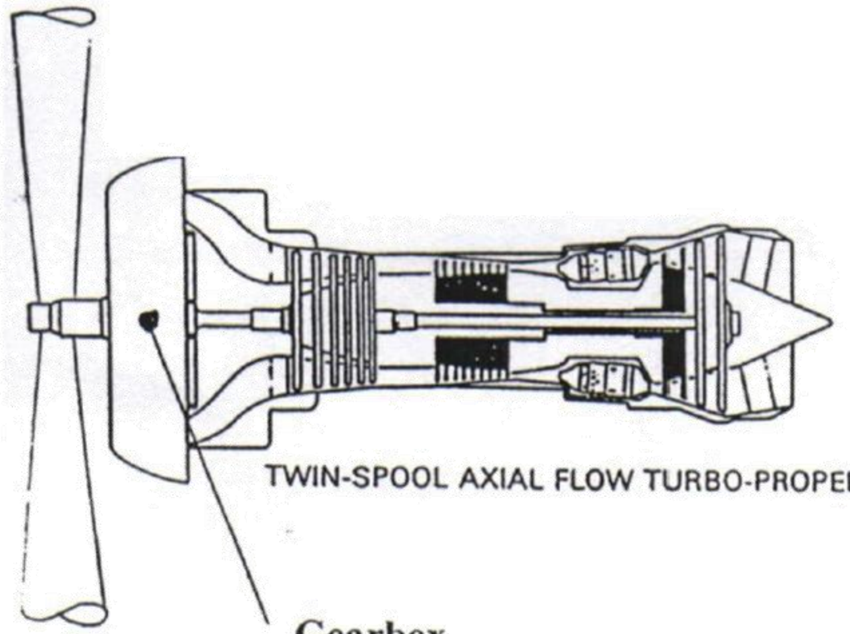
High by-pass ratio Turbofan

Thrust ~2000 to 100,000 lb
By-pass ratio 4 – 12
OPR ~ > 40
Fan PR ~ 1.9
Specific Thrust ~ 25 – 35 lb/lb/sec
~ 250 – 350 m/s

Low by-pass ratio Reheated Turbofan

Thrust ~10,000 to 40,000 lb (inc R/H)
By-pass ratio 0.3 – 1
OPR ~ 25 – 30
Fan PR ~ 3 – 5
Specific Thrust ~ 100 lb/lb/sec (inc R/H)
1000 m/s

The turboprop engine



Total Thrust:

Thrust from propeller plus jet thrust

Power:

Shaft Power x Propeller efficiency + Jet thrust x Aircraft Velocity

$$\boxed{P_{TOT} = \eta_{PROP} Q\Omega + \mathrm{FC_a}}$$

To compare with a piston engine which has no jet thrust the following is used:

Equivalent Shaft Power:

Shaft Power + (Jet thrust x Aircraft Velocity)/Propeller efficiency

Cycle Design Point: Main Parameters

Turbofan Engine

- **Overall Pressure Ratio:**
  - Total pressure at compressor delivery divided by that at entry to first compressor stage

- **Stator Outlet Temperature:**
  - Temperature of gas which does work at first turbine rotor
  . Also called Rotor Inlet Temperature or Turbine Inlet Temperature

- **Fan pressure ratio:**
  - Total pressure at fan delivery divided by that at fan entry

Cycle Design Point: Main Parameters

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Turbofan Engine

- **By-pass ratio:**
  - Ratio of mass flow of by-pass stream to that passing though combustion system
- **Component performance parameters:**
  - The characteristics of each component in terms of efficiency, flow capacity, pressure etc.
- **Pressure balance in exhaust jets:**
  - The ratio of the static pressure in the two stream should be close to unity.

Intake Performance

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- The **Intake** can be considered to be an **Adiabatic Duct** with no heat or work transfer.

- **Total Temperature remains constant**.

- The **losses** due to friction, shock waves etc. will be seen as a **reduction in total pressure**

- The main ways of defining Intake Performance are:
  - ***Inlet Pressure Ratio Factor*** (defined as the reduction in stagnation pressure in the inlet)
  - ***Ram Efficiency*** (defined in terms of pressure)
  - ***Isentropic Efficiency*** (defined in terms of temperature)

Intake Efficiency: in terms of pressure

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Inlet Pressure Ratio Factor

Intake performance is usually quoted in terms of a pressure recovery factor $\frac{p_{01}}{p_{0a}}$ i.e. the reduction stagnation pressure in the inlet.

Ram Efficiency:

$$\eta_r = \frac{P_{o1} - P_a}{P_{oa} - P_a}$$

Intake Efficiency: Isentropic

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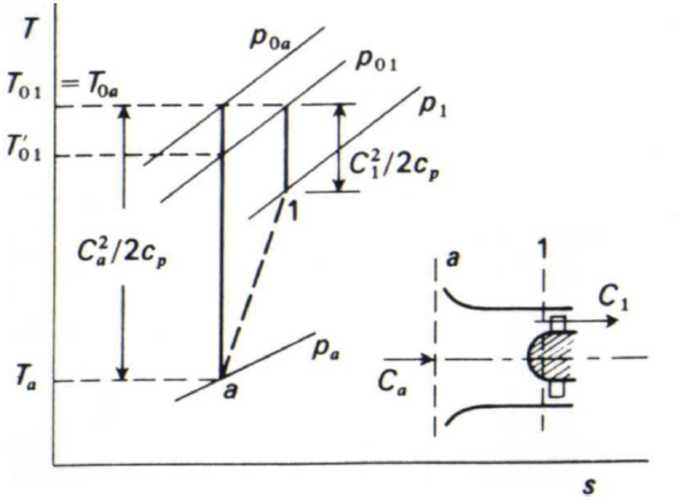


ISENTROPIC EFFICIENCY:

From T – s diagram:

$$T_{01} = T_{oa} = T + \frac{C_{FE}^2}{2C_P} \quad \& \quad \frac{P_{o1}}{P_a} = \left(\frac{T'_{o1}}{T_a}\right)^{\frac{\gamma}{\gamma-1}}$$

Where T'_{o1} is the temperature that would have been reached after an isentropic compression to P_{01}.

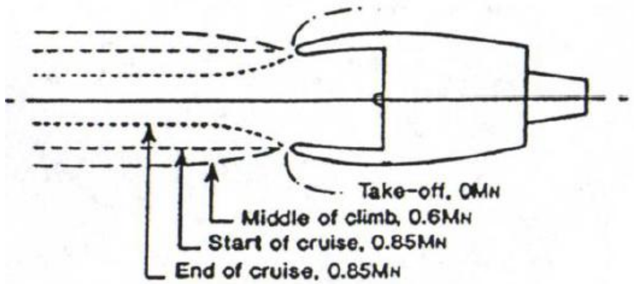
T'_{o1} can be related to T_{01} by Isentropic Efficiency:

$$\eta_{isen} = \frac{T'_{o1} - T_{o2}}{T_{o1} - T_{o2}} \quad \& \quad (T'_{o1} - T_a) = \eta_{isen}\frac{C_a^2}{2C_p}$$

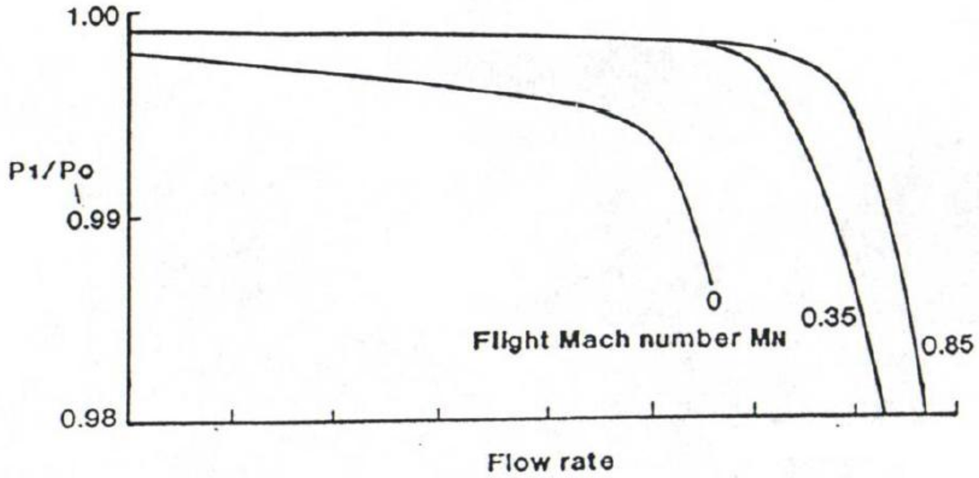
$$\frac{P_{o1}}{P_a} = \left[1 + \frac{T'_{o1} - T_a}{T_a}\right]^{\frac{\gamma}{\gamma-1}} \qquad \frac{P_{o1}}{P_a} = \left[1 + \eta_{isen}\frac{C_a^2}{2C_p T_a}\right]^{\frac{\gamma}{\gamma-1}}$$

$$M = \frac{C}{\sqrt{\gamma R T}} \quad \& \quad \gamma R = C_p(\gamma - 1) \quad \rightarrow \quad \frac{P_{o1}}{P_a} = \left[1 + \eta_{isen}\frac{(\gamma - 1)}{2}M^2\right]^{\frac{\gamma}{\gamma-1}}$$

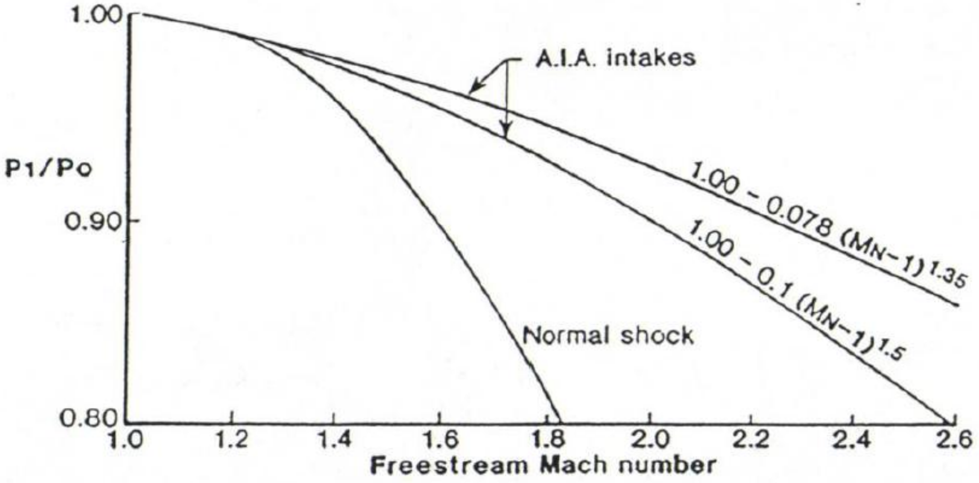
Intake Performance



Intake flow stream tubes
(typical subsonic transport)

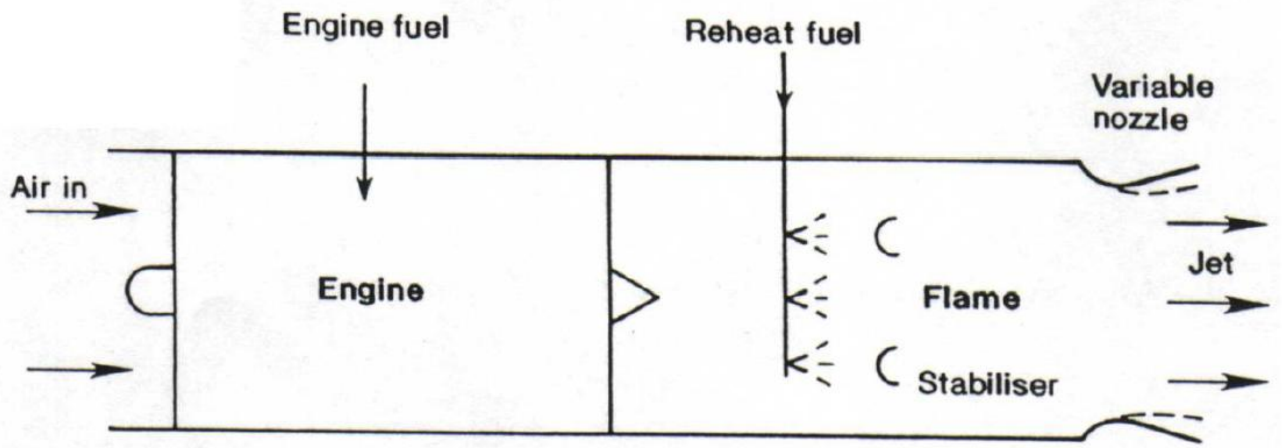


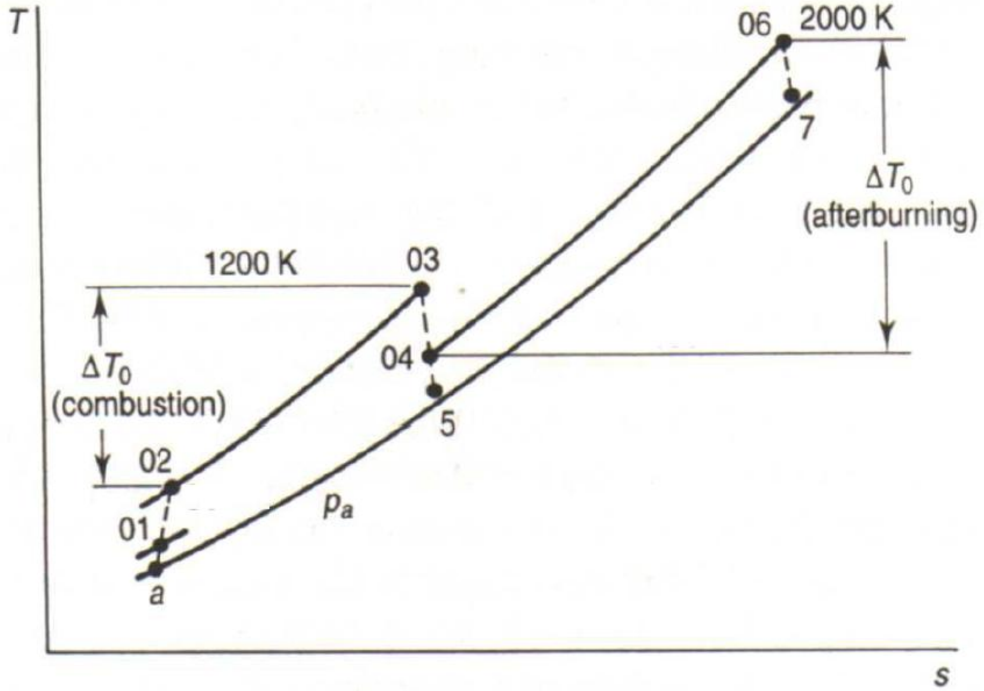
Subsonic flight intake recovery (typical) – Fixed geometry



Supersonic flight intake recovery – Variable geometry

Thrust Augmentation





Nozzle Performance

- $\boldsymbol{Net\ Thrust} = Nozzle\ Gross\ Thrust - Momentum\ Drag$

$\dot{m}C_a$

- **NOZZLE GROSS THRUST:**

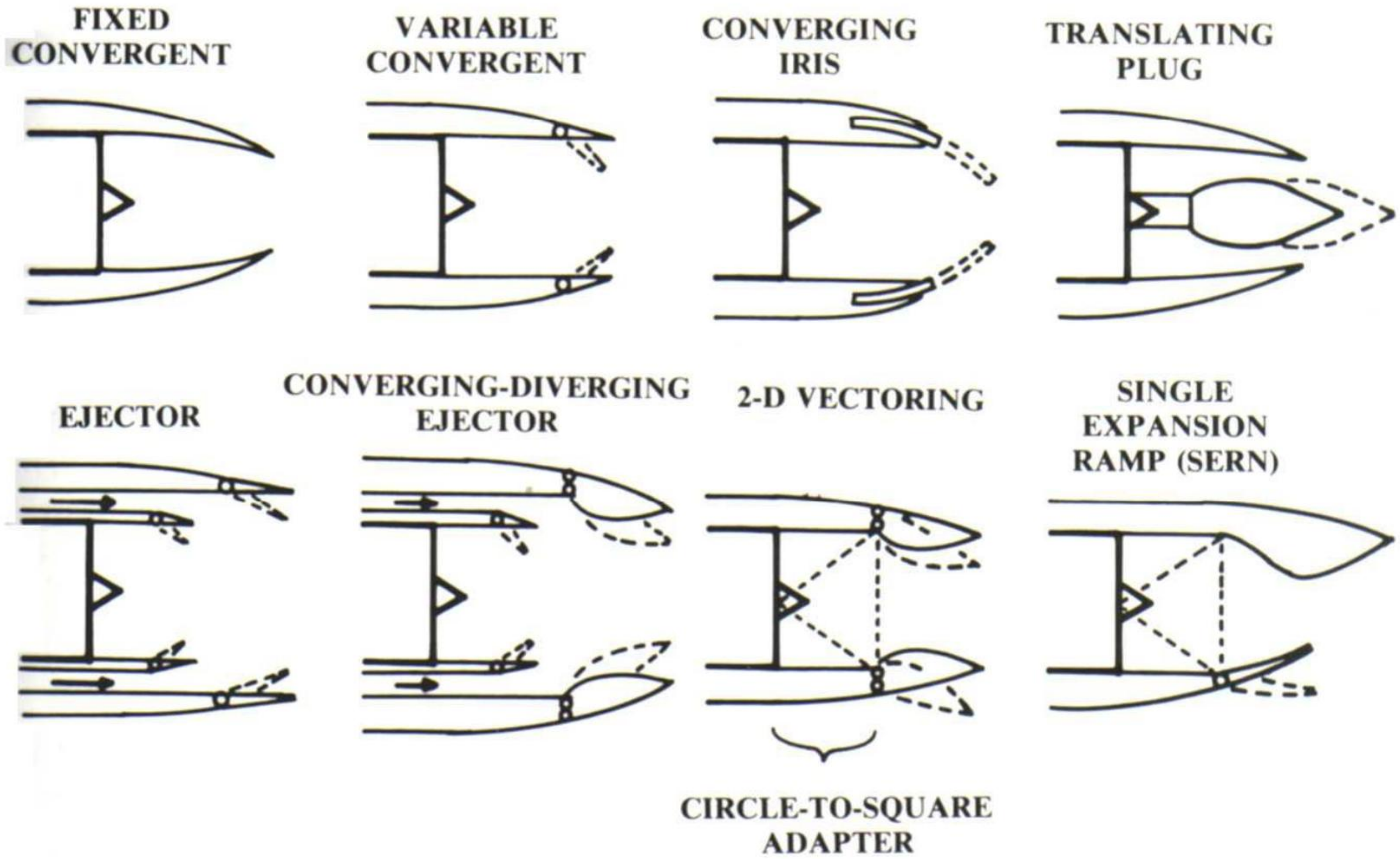
$$\boldsymbol{F_{N_{gross}} = \dot{m}_j \cdot C_j + A_j \cdot (P_j - P_a)}$$

For a Convergent-Divergent or an un-choked Convergent Nozzle, the static pressure at the nozzle exit is equal to the ambient static pressure i.e. $P_j = P_a$. Hence the pressure term is 0.

- **IDEAL GROSS THRUST:**

  - The ideal gross thrust assuming that the flow entering the nozzle is expanded adiabatically & reversibly to the ambient static pressure i.e. a convergent-divergent nozzle with no losses.

Nozzle Types



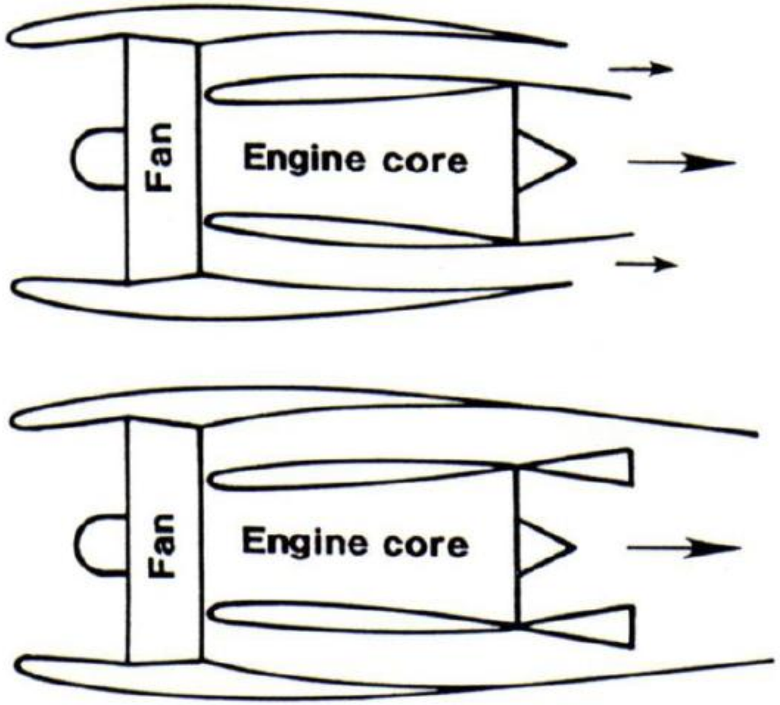
Nozzle Types: Supersonic combat aircraft

RB199 with Convergent Nozzle

EJ 200 with Con-di Nozzle

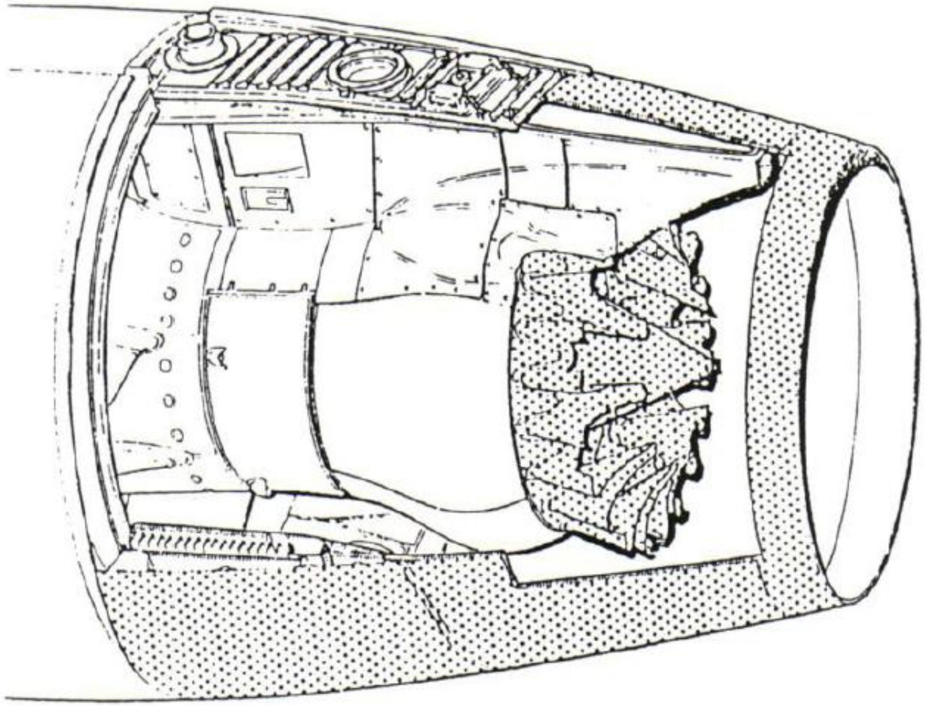
F35 with vectoring final Nozzle

Nozzle Types:
Supersonic high-bypass ratio engines



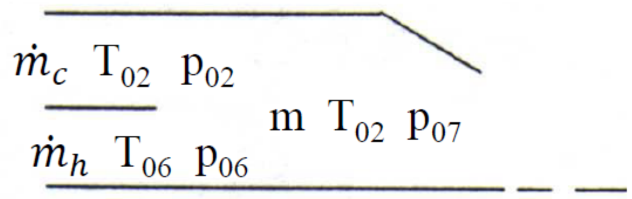
Separate jets

Mixed exhaust



Typical practical mixer

Mixing of hot and cold streams



Enthalpy Balance:

$$\dot{m}_c \cdot C_{p_c} \cdot T_{02} + \dot{m}_h \cdot C_{p_h} \cdot T_{06} = \dot{m} \cdot C_{p_m} \cdot T_{07}$$

$$\dot{m} = \dot{m}_c + \dot{m}_h$$

$$\lambda = \dot{m}_c \; / \; \dot{m}_h$$

The properties of a mixture of gases to those of its constituents can be written as:

$$C_{p_m} = \frac{\dot{m}_c \cdot C_{p_c} + \dot{m}_h \cdot C_{p_h}}{\dot{m}_c + \dot{m}_h} \quad \& \quad \left(\frac{\gamma}{\gamma - 1}\right)_m = \frac{R_m}{C_{pm}}$$

$$R_m = \frac{\dot{m}_c \cdot R_c + \dot{m}_h \cdot R_h}{\dot{m}_c + \dot{m}_h}$$

Momentum Balance:

$$(\dot{m}_c \cdot C_c + P_2 \cdot A_2) + (\dot{m}_h \cdot C_h + P_6 \cdot A_6) = \dot{m} \cdot C_7 + P_7 \cdot A_7$$

(If there is no swirl in the duct then the static pressure will be uniform i.e. $P_2 = P_6$)

Continuity gives: $\dot{m} = \rho_7 \cdot C_7 \cdot A_7$

p_{07} is required for the cycle calculation & this is best done using an iterative procedure.

Mechanical Design Parameters

Design Challenges

For a given Design Point, a number of mechanical design parameters must be kept within limits

- Shaft Speeds
- Compressor delivery temperature and pressure
- Fan flutter
- Vibration levels of rotating components
- Disc and blade stress levels
- Cycling and creep life of major components
- Oxidation levels of components
- Rotating flow instabilities

Key take-aways of Lecture 3

- The difference between Design Point & Off design Performance
- The key parameters for a Turbofan Engine
- How Intake Performance is characterized
- Nozzle types & how their performance is calculated
- Turbo prop & shaft power performance
- The data required for a full performance analysis
- The main Mechanical Design parameters

What's in Lecture 4?

- Design Points for different types of platforms
- Fundamental Dimensionless relationships
- The calculation of Off design Performance
- How thrust & fuel consumption varies with inlet conditions i.e. Altitude, Mach Number & throttle setting