AMRITA VISHWA VIDYAPEETHAM

MOOC
Course on Aerospace
Propulsion

Aerospace Propulsion : Fundamentals with Applications

Dr. Dr. A.R. Srikrishnan
Department of Aerospace engineering
2023

01
Course Introduction & Thrust Equation

The Power to Propel

- The Course focusses on power generation systems for propulsion
  - *"The Heart !"*
- Challenging Requirements of Propulsion Systems
  - Increasing demand for more power, more speed
  - Weight reduction
  - Fuel Efficiency
  - Operation under varying conditions – including altitude
  - Cop up with material limitations
  - SAFETY and RELIABILITY
  - *etc. …*

The Course

Need a thorough understanding of the principles, parameters and limiting factors of the performance of each component of the system

- *The rationale for conceptualization & implementation of specific engine types*

The Course..

- The requirements, some of them mutually conflicting, present a complex picture
- *Focused on The principles, parameters and limiting factors of the performance of each component of the system*
  - *The rationale for conceptualization & implementation of specific engine types*
  - *The components and the theory that govern their performance*
- ***The Requirement****: To develop the capability to analyse any propulsion system from energy perspective, based on the key performance parameters, requirements and the processes involved*

The Course Objectives

- Provide introduction to the functioning of aircraft engines and their performance characterization. Explain the components of aircraft engines and enable students to the carry out thermodynamic analysis of the engine cycles. Provide an introduction to rocket propulsion.

- Introduce the concepts of quantification of performance of propulsion systems
- Enable the students to understand and analyze the various air-breathing propulsion systems based on the key performance parameters
- Introduce the methodology of cycle analysis specific to propulsion systems
- Introduce fundamental aspects of conversion of chemical energy to propulsive thrust
- Introduce performance parameters for rockets
- Enable the students to appreciate the differences between various rocket propulsion systems and their applications

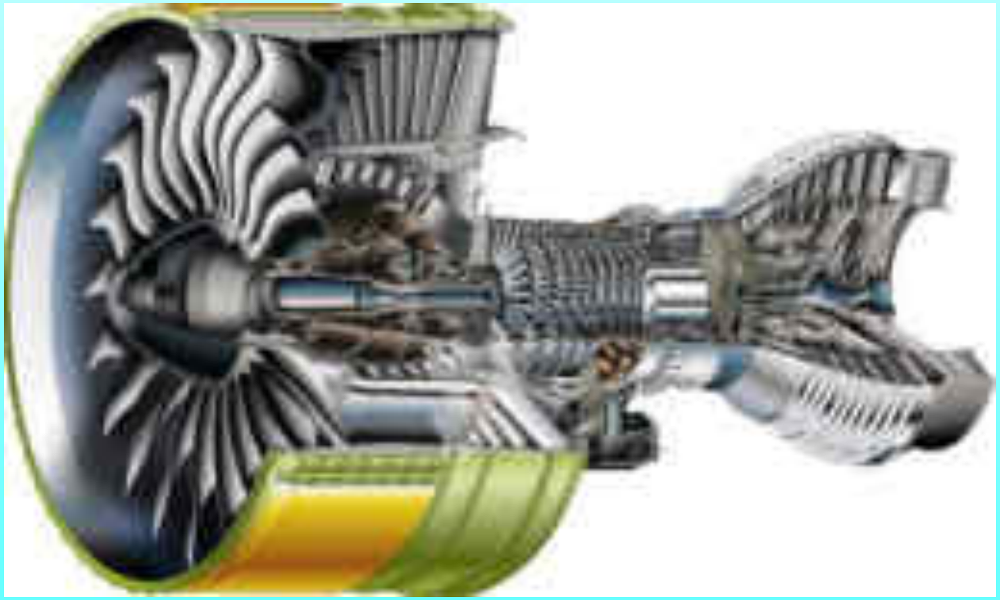
Overview of Course Contents

- *Propulsive Thrust and Propulsion Systems*
- *Thermodynamic Analysis of Components*
  - *Ideal Cycle*
  - *Real Cycle*
- *Fundamentals of Combustion*
- *Combustor Systems*
- *Rocket Propulsion*
  - *Rocket vehicle mechanics, Multistaging, Thermodynamics of the rocket engine etc.*

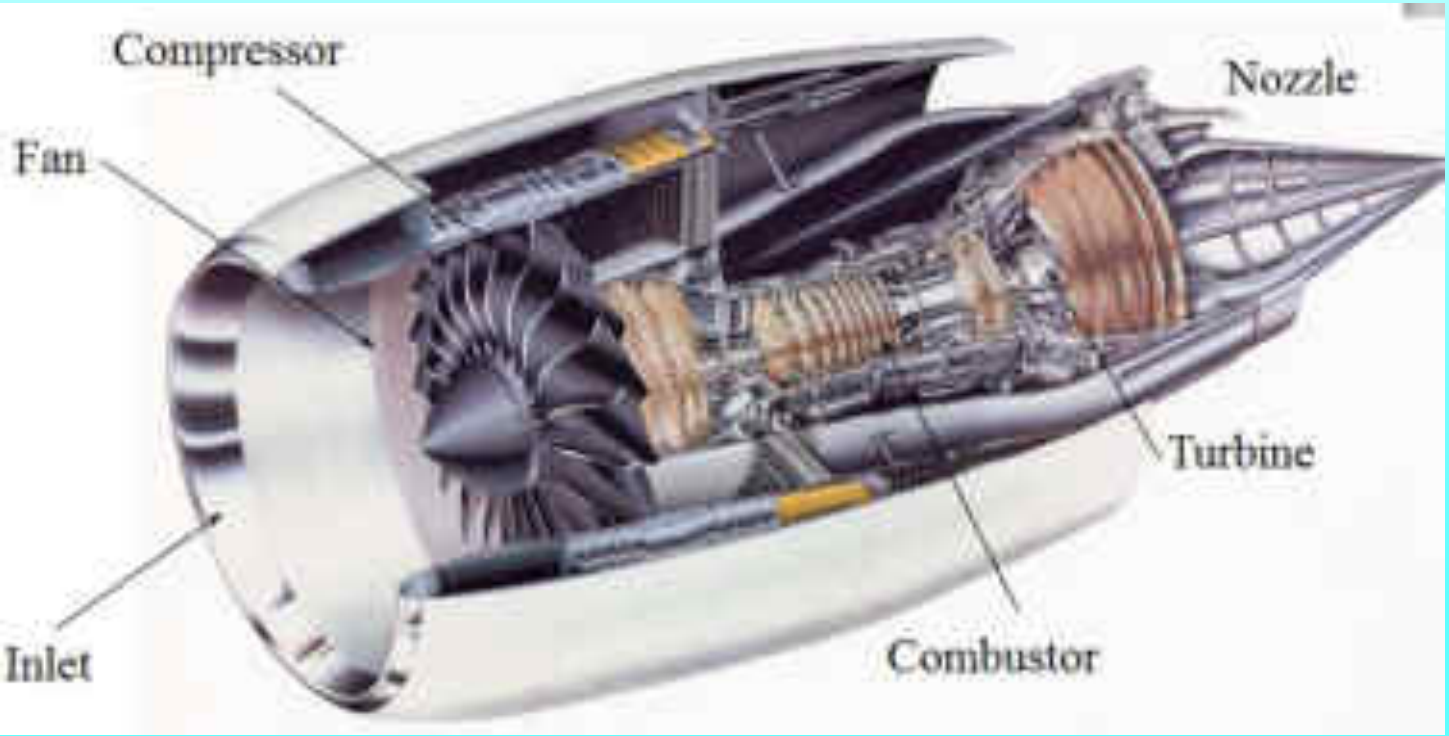
Syllabus

- ***Module I***
- Momentum analysis of thrust generation, Types of propulsion systems and their components, Performance measures, Propellers, Performance coefficients. Review of thermodynamic cycles, Ideal cycle analysis of ramjets, turbojets and turbofan engines
- ***Module II***
- Component performance, Analysis of real engines, Review of combustion chemistry, Heat of combustion, Reaction rate, Flames, Stability considerations, Application to gas turbine combustion, Introduction to aviation Fuels
- ***Module III***
- Rocket vehicle mechanics, Multistaging, Thermodynamics of the rocket engine, Rocket engine performance, Types of rocket engines, Fuels for solid and liquid propellant rockets, Rocket cooling


A380



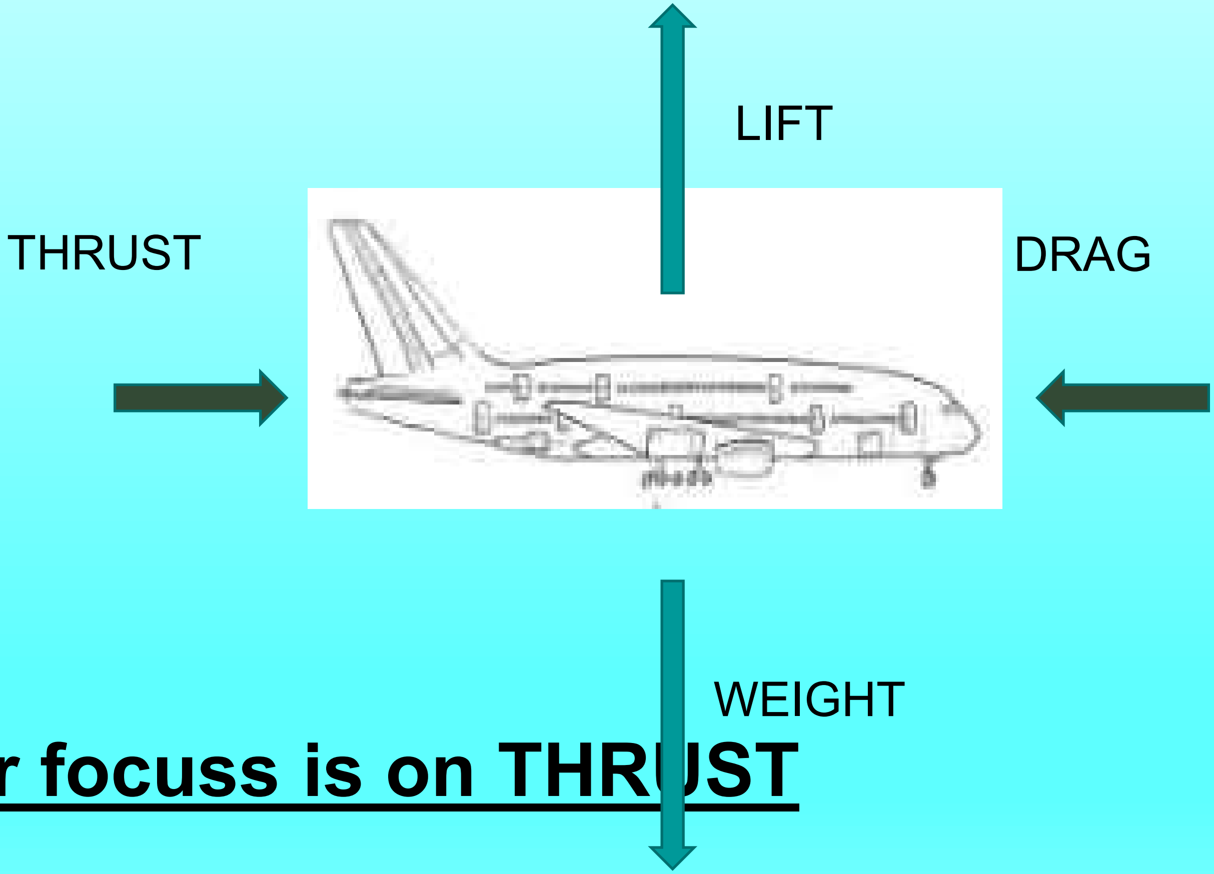
The Inside Story…



Text Books

- **Mattingly:** ***Elements of Propulsion:***
- ***Gas Turbines and Rockets***
- **Flack RD:** ***Fundamentals of Jet Propulsion with Applications***
- **Hill & Patterson**: ***Mechanics & Thermodynamics of Propulsion***
- ***Sutton & Biblarz: Rocket Propulsion Elements***
- *Etc.*

The *Familiar* Force Balance…

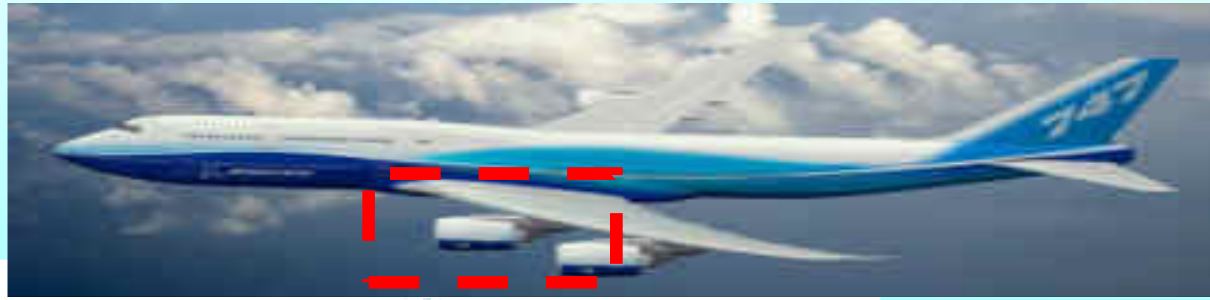


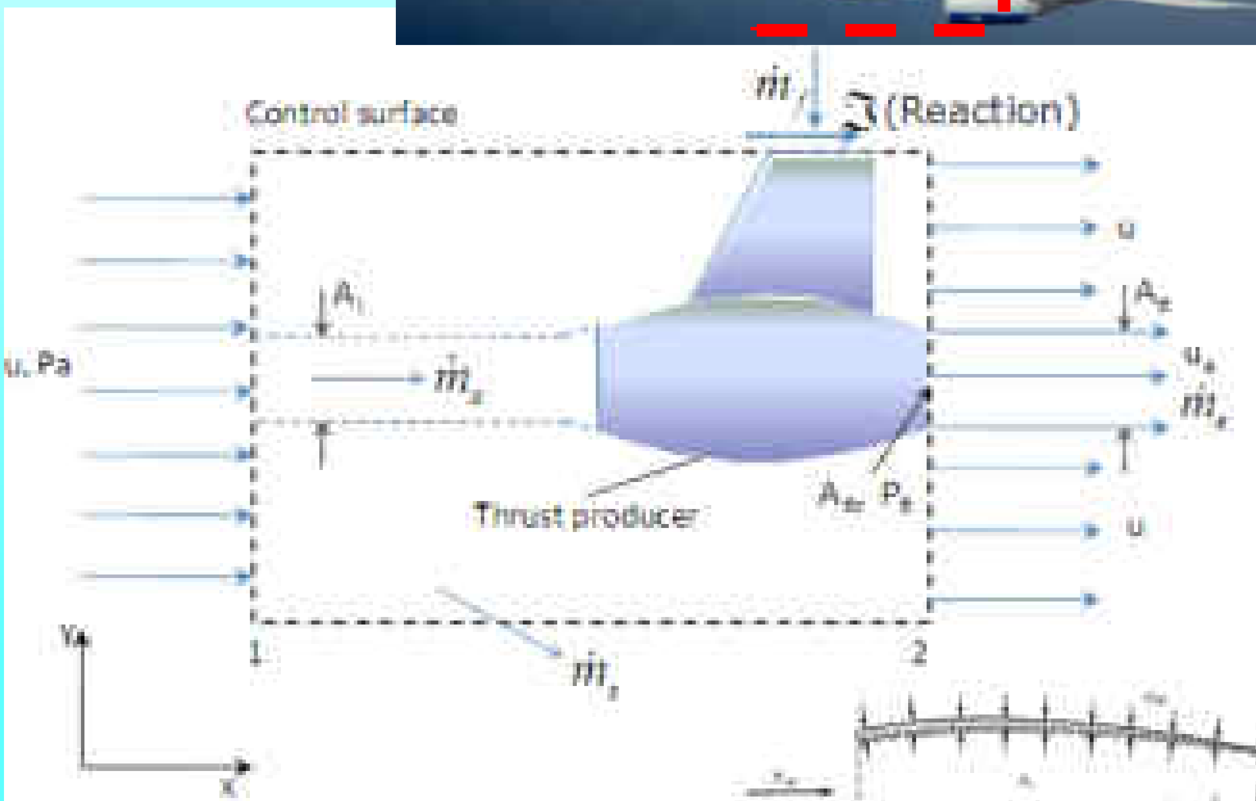
- **<u>Our focuss is on THRUST</u>**

The Propulsive Thrust

- Propulsive Thrust: The propulsive force which is generated as a reaction to the change in the momentum of a working fluid
  - Land based vehicles are "*propelled*" by the reaction to the traction force resulting from the torque transmitted to the wheels
- Can be explained and quantified using Newton's Second Law
- Derivation for an appropriate control volume

Recalling Derivation of the THRUST EQUATION



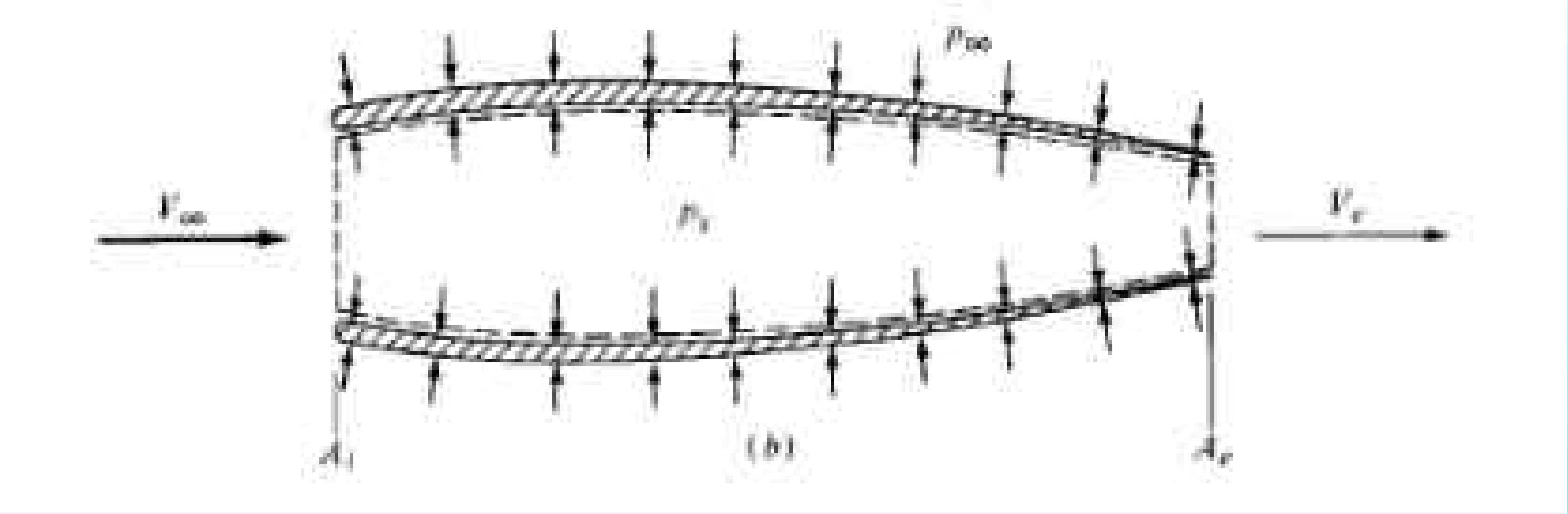


The Momentum Equation..

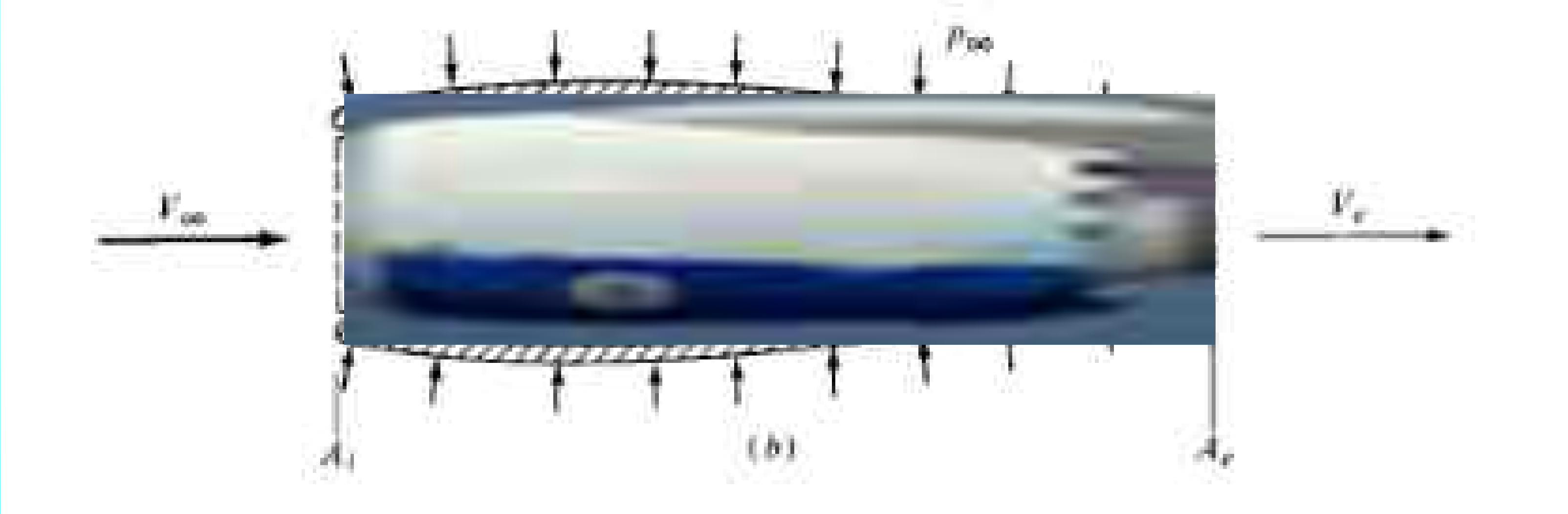
- Recalling Momentum equation as applied to a control volume:

$$\sum F = \frac{d}{dt} \oiiint_{cv} \rho u dV + \iint_{cs} \rho u(u.n) dA$$

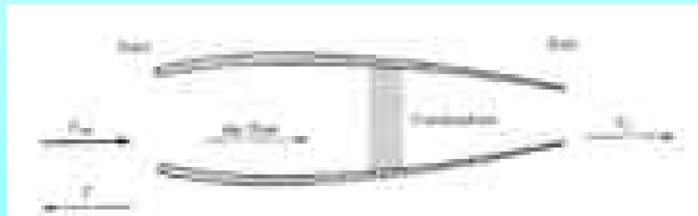
- How does it transform to a (i) Steady (ii) Uniform flow, and for forces along the x-direction ?

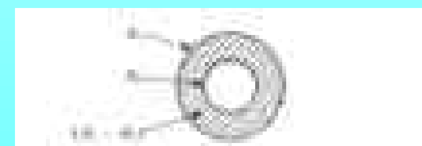


p_∞
V_∞
p_i
V_e
A_i
(b)
A_e


p_∞
V_∞
V_e
A_i
(b)
A_e

The Control Volume, Surfaces & The Pressure Forces





Mass Flow Across the CV

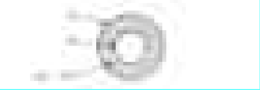
Thrust & Pressure Forces…

- Thrust: The net pressure forces acting on the engine along the direction of flight
  - Ps on the internal surface P_{inf} on the external surface

$$T = \int (p_s\, dS)_x + \int (p_\infty\, dS)_x$$

  - As Pinf acts uniformly over the surface:

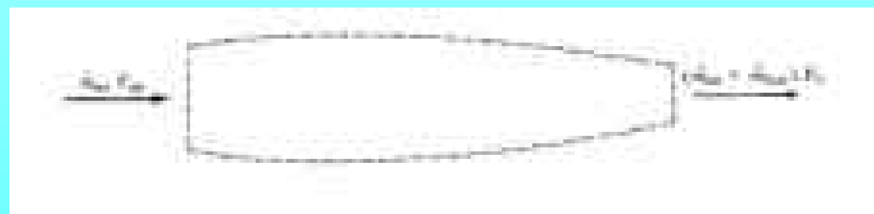
$$\int (p_\infty\, dS)_x = p_\infty \int (dS)_x = p_\infty (A_i - A_e)$$



$$T = \int (p_s\, dS)_x + p_\infty (A_i - A_e)$$

X-Component of Force on Gas Inside the CV

- Consider x-component of pressure forces acting on the CV:
  - p_s*surface area on the internal surface
    - *Equal & Opposite to the force exerted by the gas, as per 3rd law*
  - p_{inf}*frontal area on A_i
  - p_e*exit area on A_e



$$F = p_\infty A_i + \int (p_s \, dS)_x - p_e A_e$$

Rate of Change of Momentum across the CV

- 2nd law equates the force to the net rate of change of momentum across the CV

$$F = (\dot{m}_{\text{air}} + \dot{m}_{\text{fuel}})V_e - \dot{m}_{\text{air}}V_{\infty}$$

- Combining the two expressions for F

$$(\dot{m}_{\text{air}} + \dot{m}_{\text{fuel}})V_e - \dot{m}_{\text{air}}V_\infty = p_\infty A_i + \int (p_s\, dS)_x - p_e A_e$$

- Hence the integral

$$\int (p_s\, dS)_x = (\dot{m}_{\text{air}} + \dot{m}_{\text{fuel}})V_e - \dot{m}_{\text{air}}V_\infty + p_e A_e - p_\infty A_i$$

- Substituting back for the integral in the thrust equation:

$$T = (\dot{m}_{\text{air}} + \dot{m}_{\text{fuel}})V_e - \dot{m}_{\text{air}}V_\infty + p_e A_e - p_\infty A_i + p_\infty (A_i - A_e)$$

The Thrust Equation….

$$T = (\dot{m}_{\text{air}} + \dot{m}_{\text{fuel}})V_e - \dot{m}_{\text{air}}V_\infty + (p_e - p_\infty)A_e$$

The Forces

- Reaction to the thrust transmitted through the support
- Surface forces: $(p_e - p_\infty)A_e$
  - Note: *Velocity & Pressure assumed uniform over the entire control surface, except at the exit plane*
- *WHEN will p_e and p_{inf} be different ?*
- *Body forces ??*

Some **IMPORTANT** Remarks on the Thrust Equation

$$Thrust = \dot{m}_e V_e - \dot{m}_i V_i + A_e (P_e - P_0)$$

- Remember the Assumptions !
- Started from the general Continuity & Momentum Equations…
- Energy Conversion: *Chemical* → *Thermal* → *Kinetic*
  - *Will study each of these in some detail later..*
- If "*Pressure thrust*" is absent:
  - *Perfectly Expanded* Nozzle Flow

$$Thrust = \dot{m}_e \dot{V}_e - \dot{m}_i V_i$$

- If fuel mass is neglected:

$$Thrust = \dot{m}(\dot{V}_e - V_i) + A_e (P_e - P_0)$$

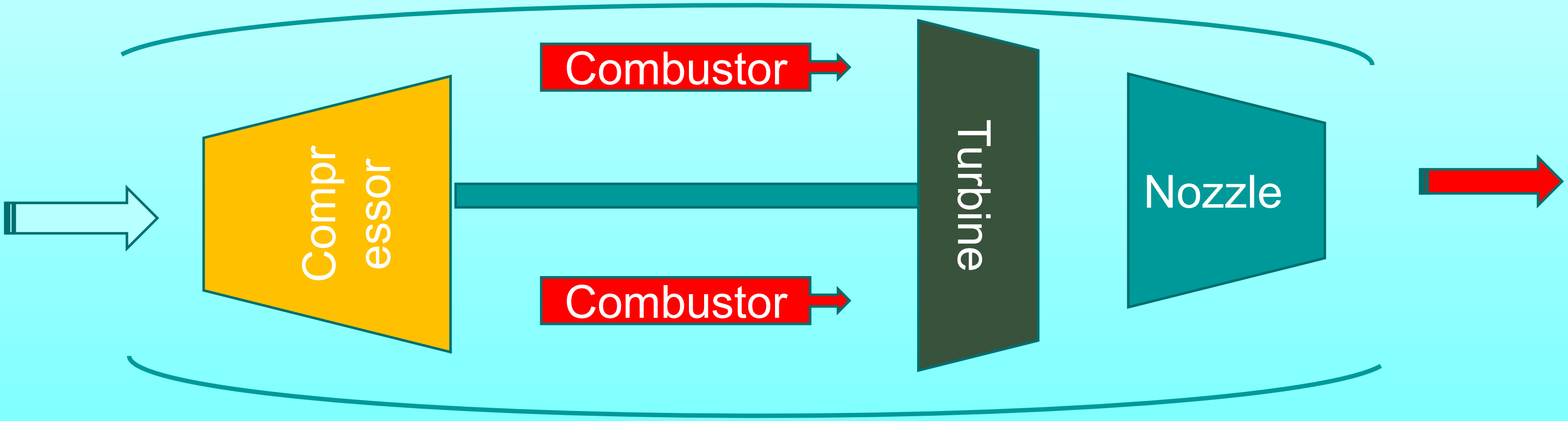
Comments on the Thrust Equation

- "*Uninstalled Thrust*" Vs "*Installed Thrust*"
  - The need to differentiate
- *Installed Thrust = Uninstalled Thrust – (Nacelle Drag + Additive Drag*)
- A "Special" Instance of Rocket (with perfect expansion in nozzle)

$$Thrust = \dot{m}_e V_e$$

WHY ?

The Components, Conceptually…



Propulsion Systems

- Objective: Generate *adequate* THRUST in a *controlled* manner
- High velocity → High Drag
  - "Need for Speed" entails need for more thrust to be generated

$$Thrust = \dot{m}_a(V_e - V_i)$$

    - Assumptions ?
  - Which one should be increased : Mass flow rate **OR** Velocity difference ?

Propulsion Systems: Performance Parameters …

- Why we need them ?
  - Quantify the expected/actual performance
    - Relevant, well-defined performance measures
    - Calculated from the available data/design variables etc.
    - Eg.: Mileage for a car/bike
  - Compare across other engines/systems/proposed configurations
    - Mission-specific analysis: *Which engine is best for the job at hand ?*
  - For a given system/engine compare the influence of various parameters on the desired performance

Performance Parameters

- TSFC: Thrust Specific Fuel Consumption
  Ξ Rate of fuel used by the engine per unit thrust produced
  = **Mass flow rate of fuel (kg/s)/Engine Thrust (N)**
  - Unit ?
  - Indicative of efficient utilization of the fuel energy
  - Reflects the importance of Fuel Weight in Aircraft performance

- Specific Thrust
  Ξ Thrust produced per unit flow rate of air
  = **Engine Thrust (N) /Mass flow rate of air (kg/s)**
    - Unit ?
  - Importance: Indicative of (i) Engine Weight (ii) Frontal Area (iii) Volume
  - Critical to maximize Sp. Thrust for
    - Military Aircraft
    - High Mach number applications

Performance Parameters ~ II

- Thrust-Weight Ratio
  - Thrust Produced/Weight of the Engine
  - Critical to Maximise for military aircrafts to improve maneuverability
    - Higher TWR -> more acceleration, ‘climb’, better maneuverability
    - Engine TWR Vs Aircraft TWR

Performance Parameters ~ III

- Thermal Efficiency
  - Ξ Net work output of the engine/Rate of thermal energy from the fuel
  - "***Thermal Factor***" of the *Overall Efficiency*
  - If the work output is *only* for increasing KE of the working fluid (as in *turbojets*)

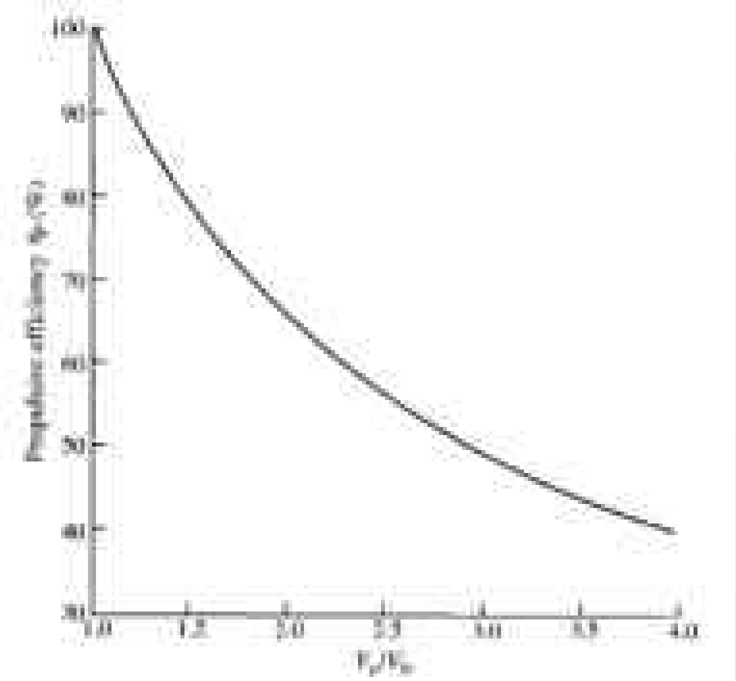
    $\eta_{thermal}$ = Rate of production of propulsive kinetic energy / fuel power

Performance Parameters ~ IV

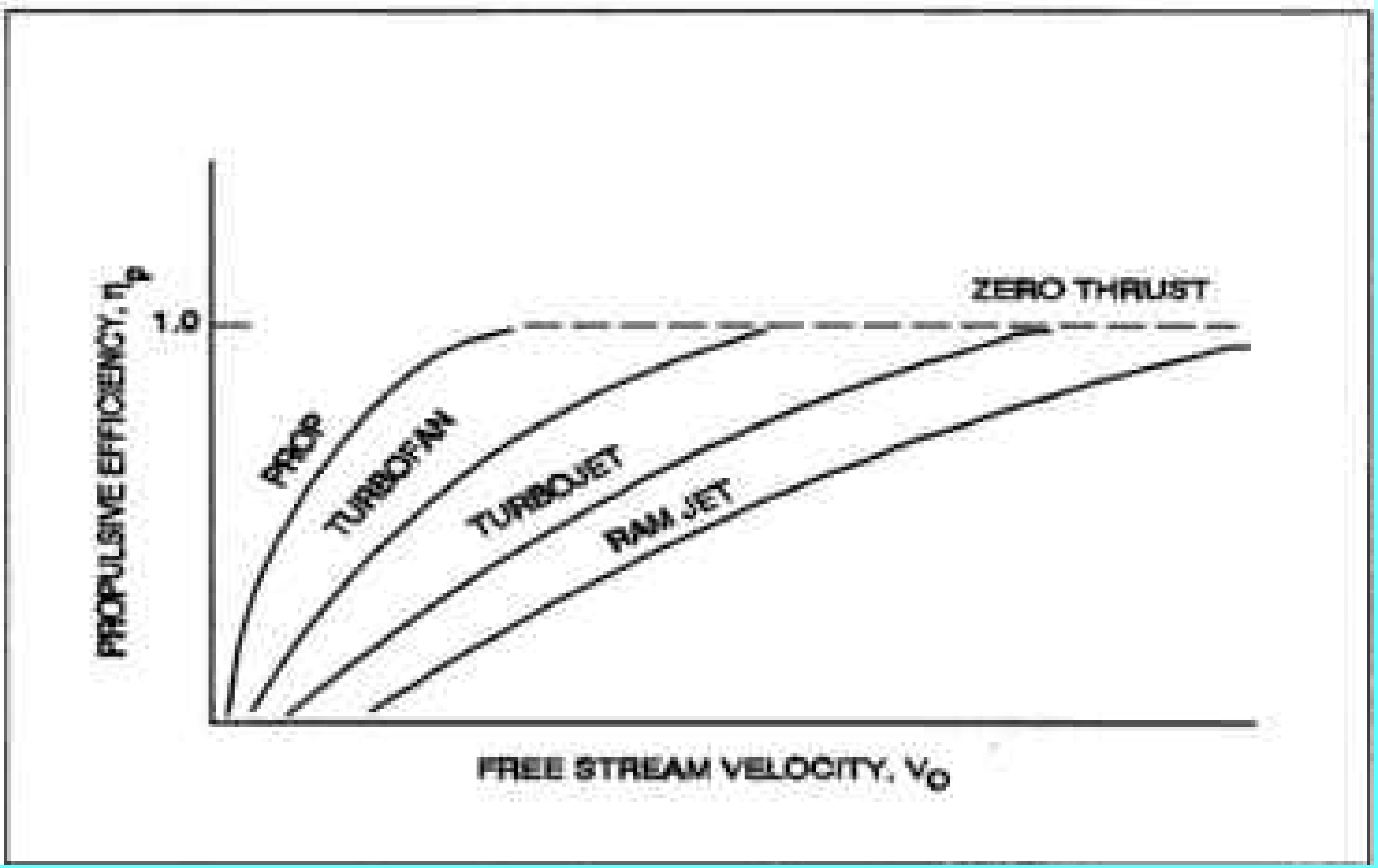
- Propulsive Efficiency
  - A measure of the effectiveness of conversion of the engine power to propulsive power
  - ***“Aerodynamic Factor*”** of the *Overall Efficiency*
  - Propulsive Power / Rate of production of propulsive kinetic energy
    - Power to the airplane / Power in the Jet

For a Turbojet with Perfect Expansion…

$$\eta_{propulsive} = \frac{TV_i}{\left(\dfrac{\dot{m}_e V_e^2}{2} - \dfrac{\dot{m}_i V_i^2}{2}\right)} = \frac{2}{1 + \dfrac{V_e}{V_i}}$$



$$\eta_{thermal} = \frac{\left(\dfrac{\dot{m}_e V_e^2}{2} - \dfrac{\dot{m}_i V_i^2}{2}\right)}{\dot{m}_f C V_f}$$


ZERO THRUST
1.0
PROPULSIVE EFFICIENCY, η_p
PROP
TURBOFAN
TURBOJET
RAM JET
FREE STREAM VELOCITY, V_O

Overall Efficiency

- Overall Efficiency
  - What you get / What you pay for
  - Propulsive Power / Fuel Power
  - ***Overall Efficiency = Thermal Efficiency X Propulsive Efficiency***

EFFICIENCY of Propulsion System From Various Perspectives

- Propulsive Efficiency
  - Propulsive Power / Rate of production of propulsive kinetic energy
  - a measure of how effectively the engine power is used to power the aircraft
    - Power to airplane / Power in Jet

$$\eta_{propulsive} = \frac{TU_o}{\left(\frac{\dot{m}_e U_e^2}{2} - \frac{\dot{m}_o U_o^2}{2}\right)} = \frac{2}{1 + \frac{U_e}{U_o}}$$

- Thermal Efficiency
  - Rate of production of propulsive kinetic energy / fuel power (*in turbojets*)
  - the net rate of shaft power or kinetic energy out of the engine divided by the rate of thermal energy available from the fuel in the engine
  - "Cycle Efficiency"

$$\eta_{thermal} = \frac{\left(\frac{\dot{m}_e U_e^2}{2} - \frac{\dot{m}_o U_o^2}{2}\right)}{\dot{m}_f h}$$

$$\eta_{overall} = \frac{TU_o}{\dot{m}_f h}$$

- Overall Efficiency
  - What you get / What you pay for
  - Propulsive Power / Fuel Power
  - Propulsive Power = TUo
  - Fuel Power = (fuel mass flow rate) x (fuel energy per unit mass)

"Thermodynamic Factor"

"Aerodynamic Facto

$$\eta_{overall} = \eta_{thermal}\eta_{propulsive}$$

Performance Parameters: Implications

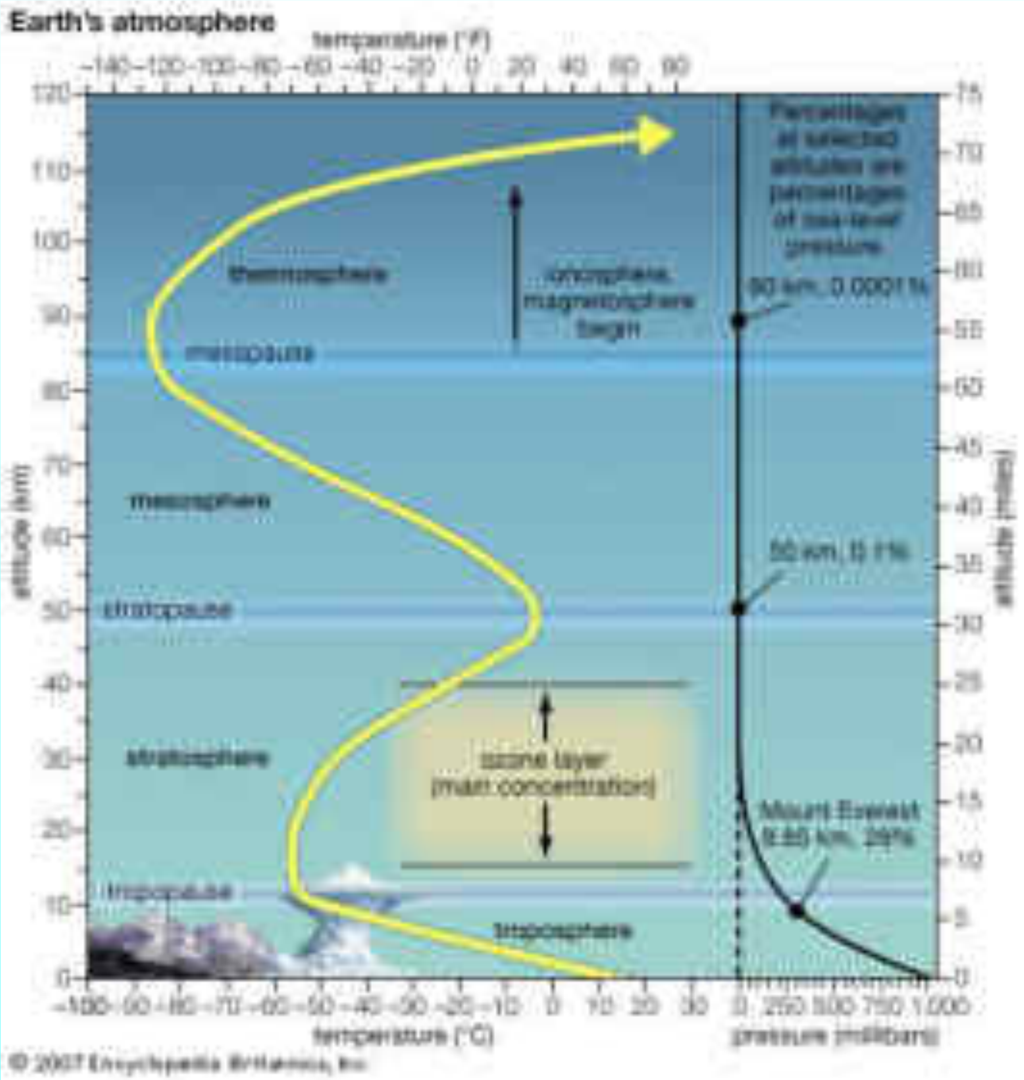
- TSFC
  - Low TSFC (desirable) → Longer range, increased payload, ability to opt for a smaller size (for the aircraft)
- Specific Thrust
  - High specific thrust (desirable) → Smaller cross sectional area (less weight, drag)

  Ξ Thrust produced per unit flow rate of air

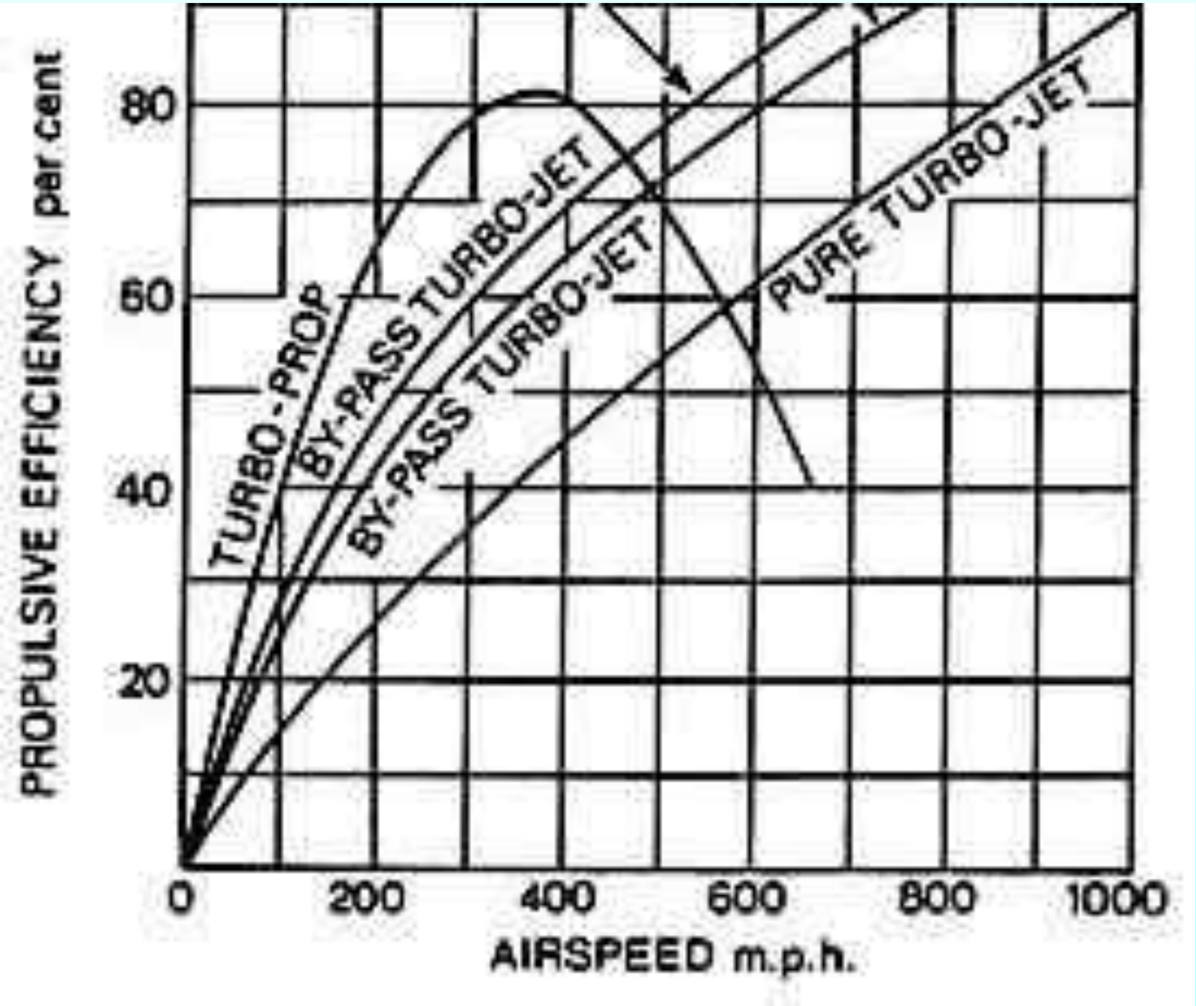
  = **Engine Thrust (N) /Mass flow rate of air (kg/s)**

- Thermal Efficiency
- Propulsive Efficiency
- Overall Efficiency

- *A turbojet flies at a Mach 1.5 at an altitude of about 18,000 ft ~ 5.5 km (T = 238 K). The nozzle-exit pressure is 154442.5634 Pa while the atmospheric pressure is 50599.8 Pa. The exit area of the exhaust nozzle is: 2330 cm^2 and the exit velocity is: 780 m/s. Air is ingested at the rate of 167 kg/s and the fuel flow rate is 2 kg/s.*
  - *Calculate Thrust developed, TSFC and specific thrust*
  - *Calculate Propulsive efficiency*
  - *Calculate the above parameters neglecting fuel flow rate.*



Earth's atmosphere
temperature (°F)
Percentages at selected altitudes are percentages of sea-level pressure
thermosphere
ionosphere, magnetosphere begin
90 km, 0.0001%
mesopause
mesosphere
50 km, 0.1%
stratopause
stratosphere
ozone layer (main concentration)
Mount Everest 8.85 km, 29%
tropopause
troposphere
altitude (km)
altitude (miles)
temperature (°C)
pressure (millibars)
© 2007 Encyclopædia Britannica, Inc.



Propulsive efficiency comparison for various gas turbine engine Prconfigurations (Rolls-Royce, 1992)

AMRITA VISHWA VIDYAPEETHAM

Aerospace Propulsion

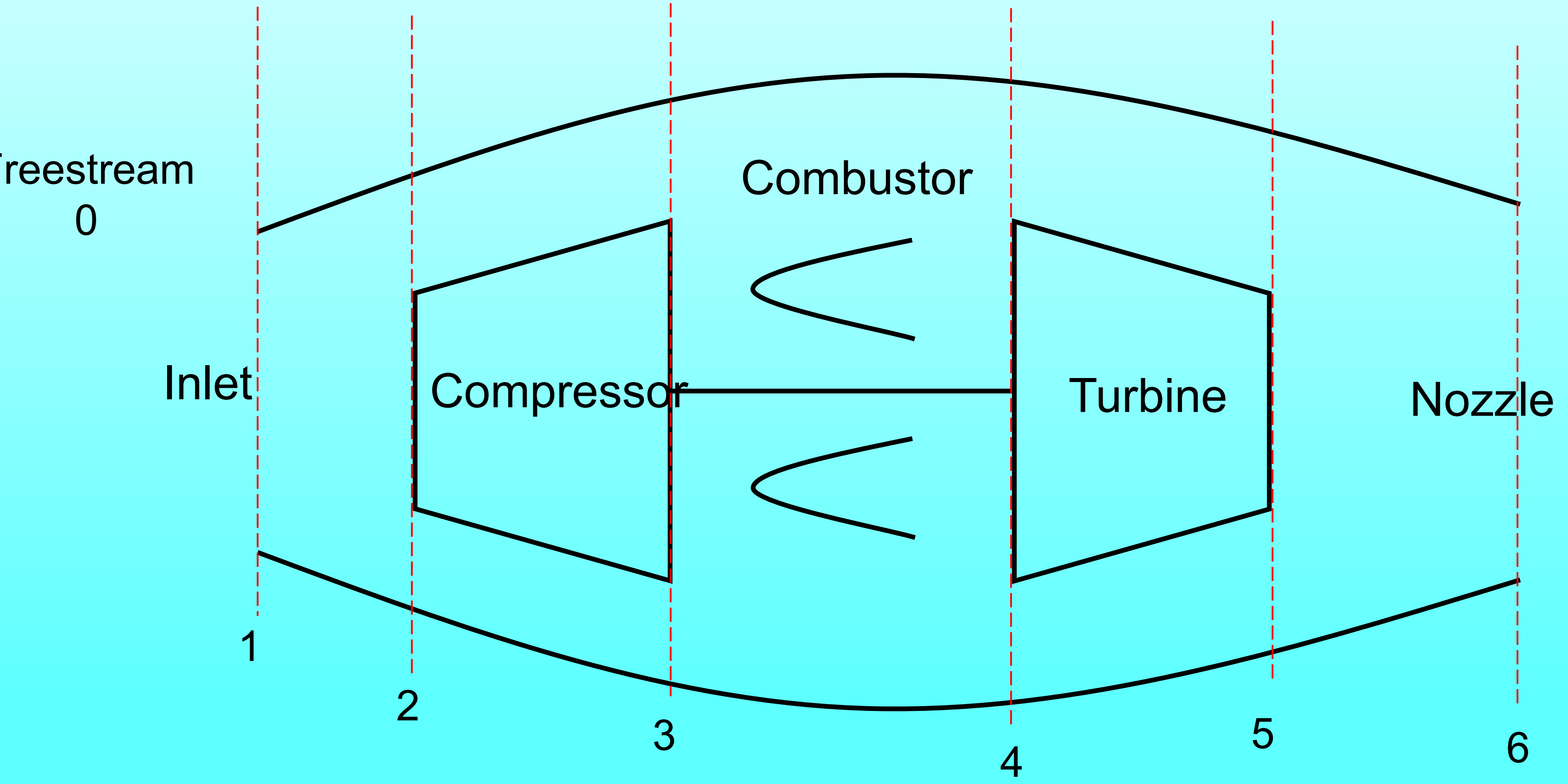
Dr. A.R. Srikrishnan
Department of Aerospace engineering
2023

The Power to Propel

- The Course focusses on power generation systems for propulsion
  - *"The Heart !"*
- Challenging Requirements of Propulsion Systems
  - Increasing demand for more power, more speed
  - Weight reduction
  - Fuel Efficiency
  - Operation under varying conditions – including altitude
  - Cop up with material limitations
  - SAFETY and RELIABILITY
  - *etc. …*

https://www.youtube.com/watch?v=kuvq-X9sdr0&t=191s

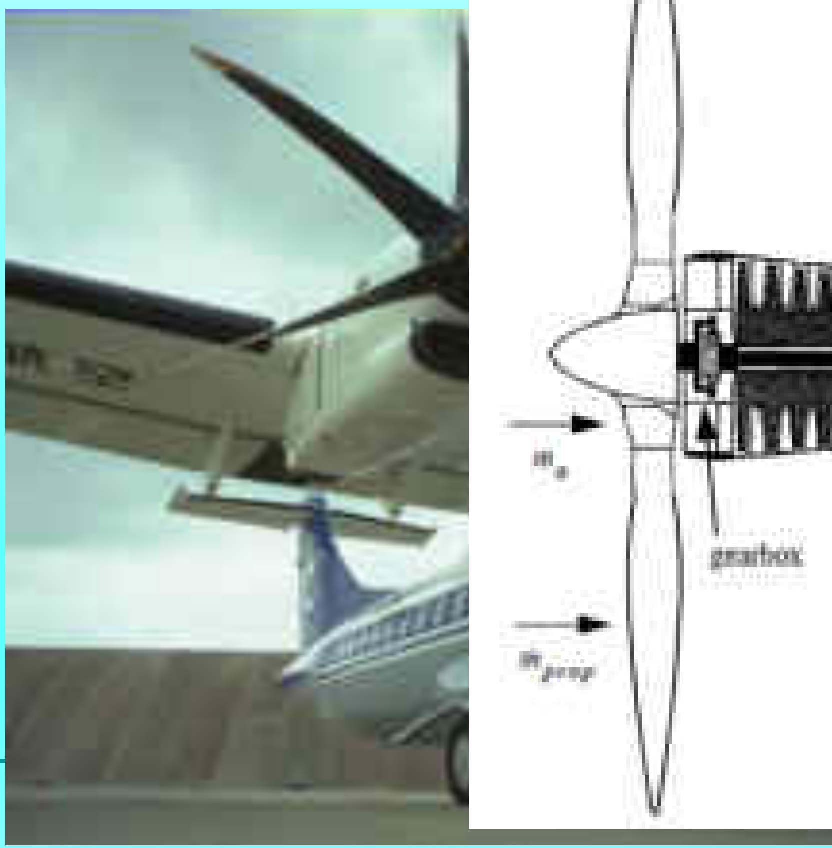
REPRESENTATION OF AN ENGINE: TURBOJET

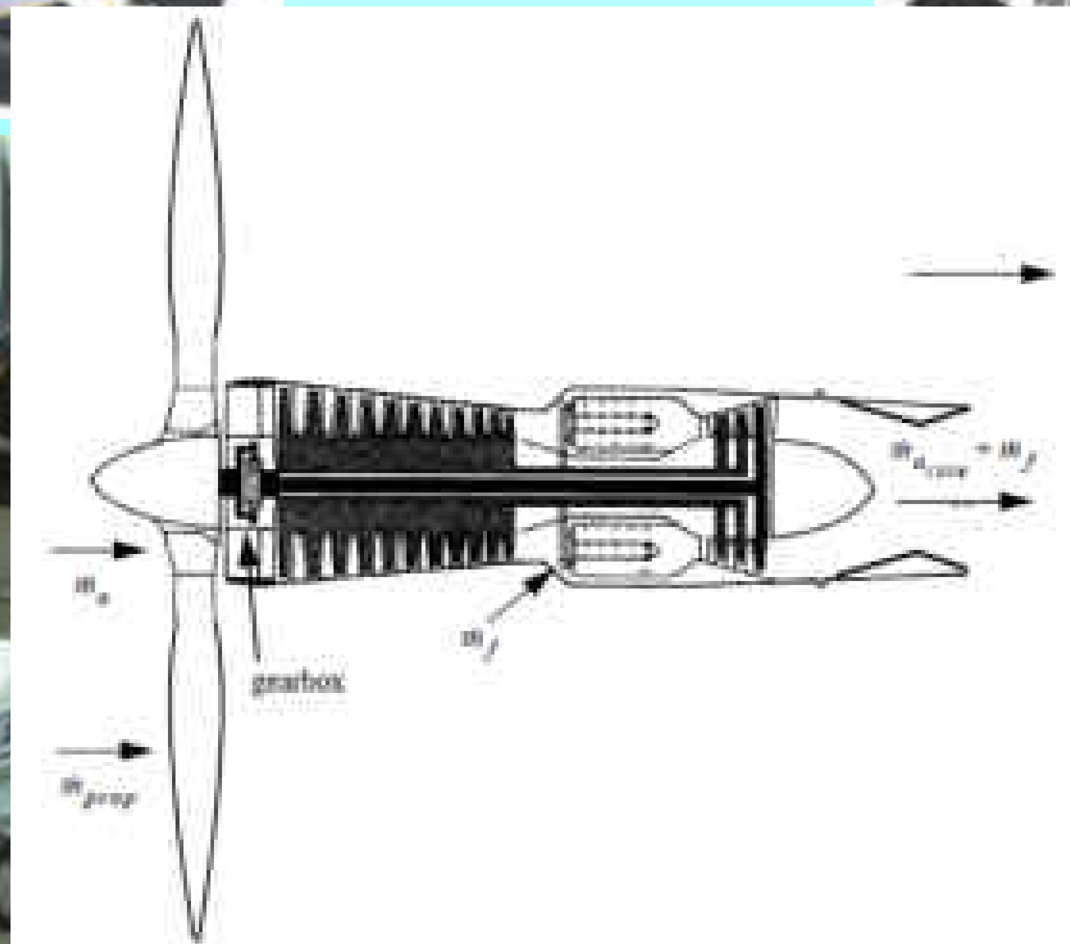


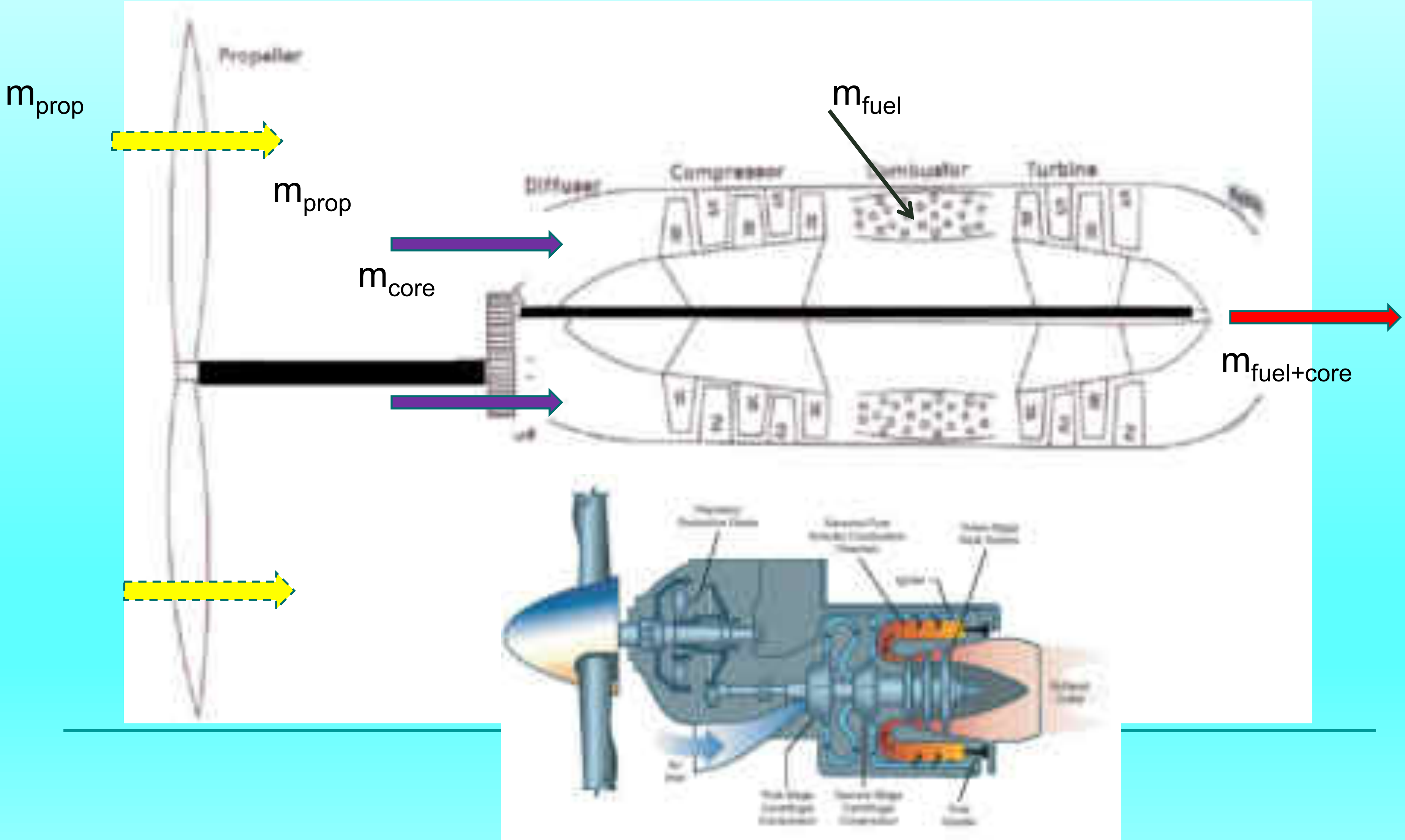
From Turbojets to Turboprops…

- High velocity of exhaust jet entails a drop in Propulsive Efficiency + increased Noise
  - ***<u>KE lost with the exhaust jet !</u>***
- The other alternative to improve thrust: Increase the mass flow rate of air
- **TURBOPROPs:**
  - *Jet engine used to turn a large propeller*
  - *The propeller ingests a **<u>large</u>** mass flow air in and imparts a **<u>small</u>** momentum to it*
  - *The propeller action produces most (90% or more) of the thrust*
  - *A small fraction of the total thrust is produced by the exhaust jet*

Turboprops






Propeller
m_{prop}
m_{prop}
m_{core}
m_{fuel}
Diffuser
Compressor
Combustor
Turbine
$m_{fuel+core}$

Turboprop: Thrust Equation

- Most of the thrust is produced by the Propeller
- Part of the total thrust (~5-15 %) produced by the core jet
- Total Thrust $T_{total} = T_{propeller} + T_{jet}$

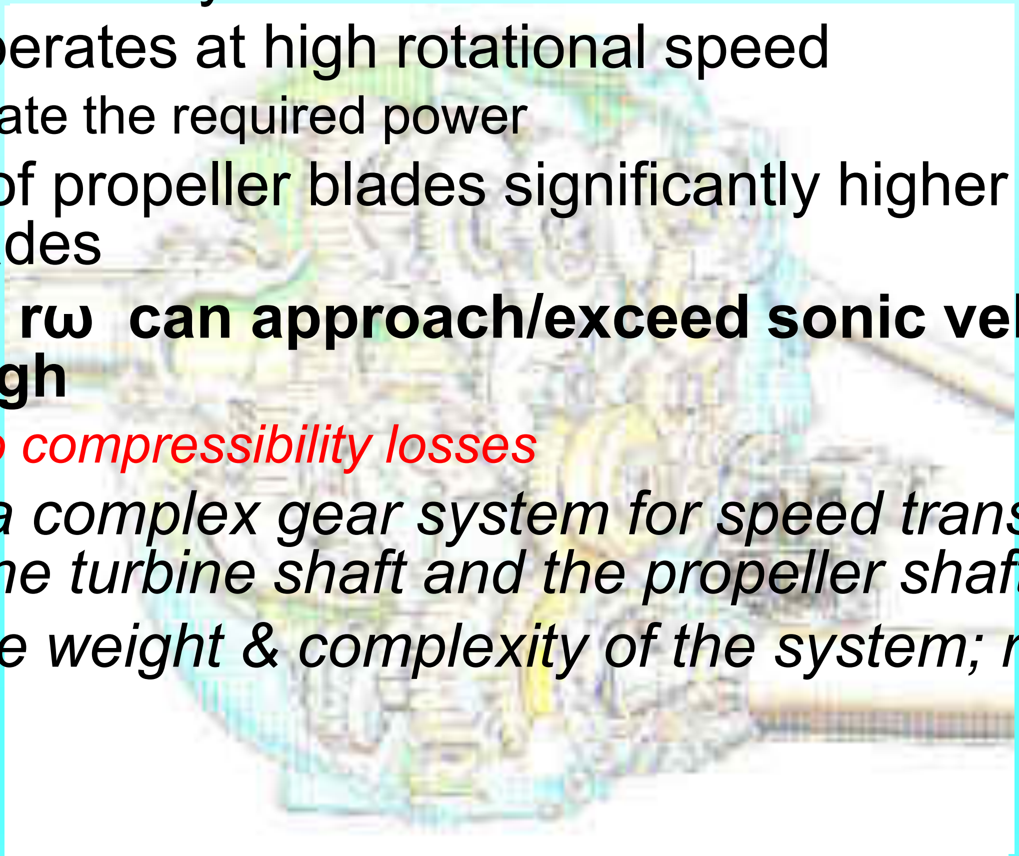
$$T_{total} = \dot{m}_e V_e - \dot{m}_i V_i + A_e(P_e - P_0) + \dot{m}_{prop}(V_{\infty prop} - V_i)$$

{Subsonic nozzle}

$$T_{total} = (\dot{m}_e V_e - \dot{m}_i V_i) + \dot{m}_{prop}(V_{\infty prop} - V_i)$$

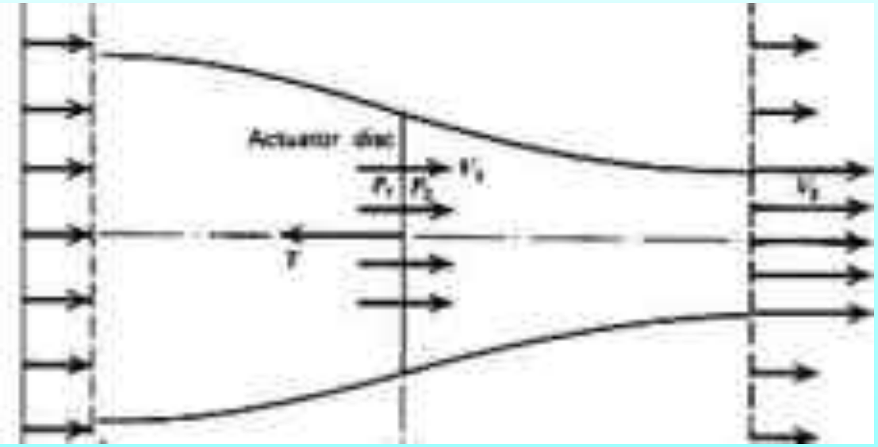
Turboprop: The Speed Reduction

- Propeller driven by turbine
- Turbine operates at high rotational speed
  - To generate the required power
- Diameter of propeller blades significantly higher than that of turbine blades
- Tip speed: **rω can approach/exceed sonic velocity if RPM is high**
  - *Leads to compressibility losses*
- *Requires a complex gear system for speed transmission between the turbine shaft and the propeller shaft*
- *Adds to the weight & complexity of the system; reliability concerns*



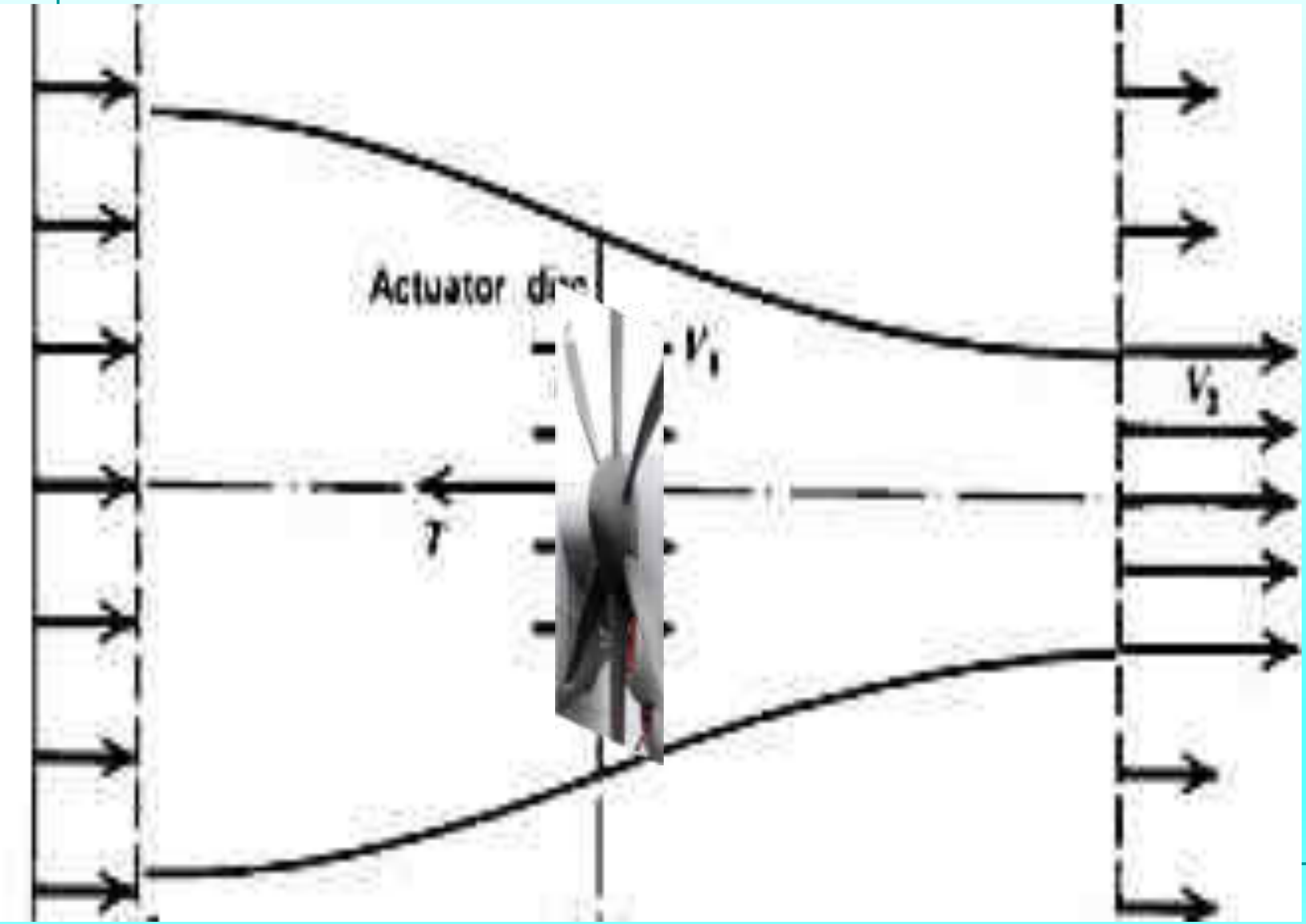
Turbojets Vs Turboprops

- The Merits:
  - Turbojets: Superior high-speed performance characteristics, ability to operate at supersonic flight speeds, less weight & frontal area, simple design, high TWR, high specific thrust
  - Turboprops: High efficiency (low TSFC) at low subsonic Mach numbers (Mach 0.3 to 0.6), ideally suited for short range applications, fairly good specific thrust
- Can the advantages be combined for better overall performance and wider range of operation ?



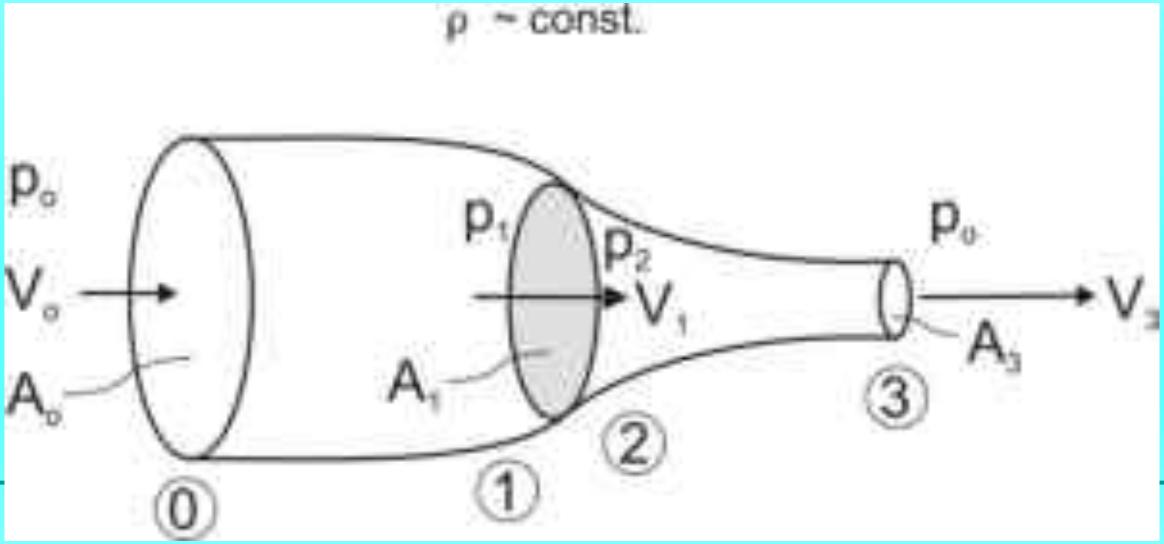
ACTUATOR DISK THEORY

A METHOD To EsTIMATE THE ideal *(minimum)* power required to drive the propeller


Actuator disc
V_1
V_2
T

Actuator Disk Theory

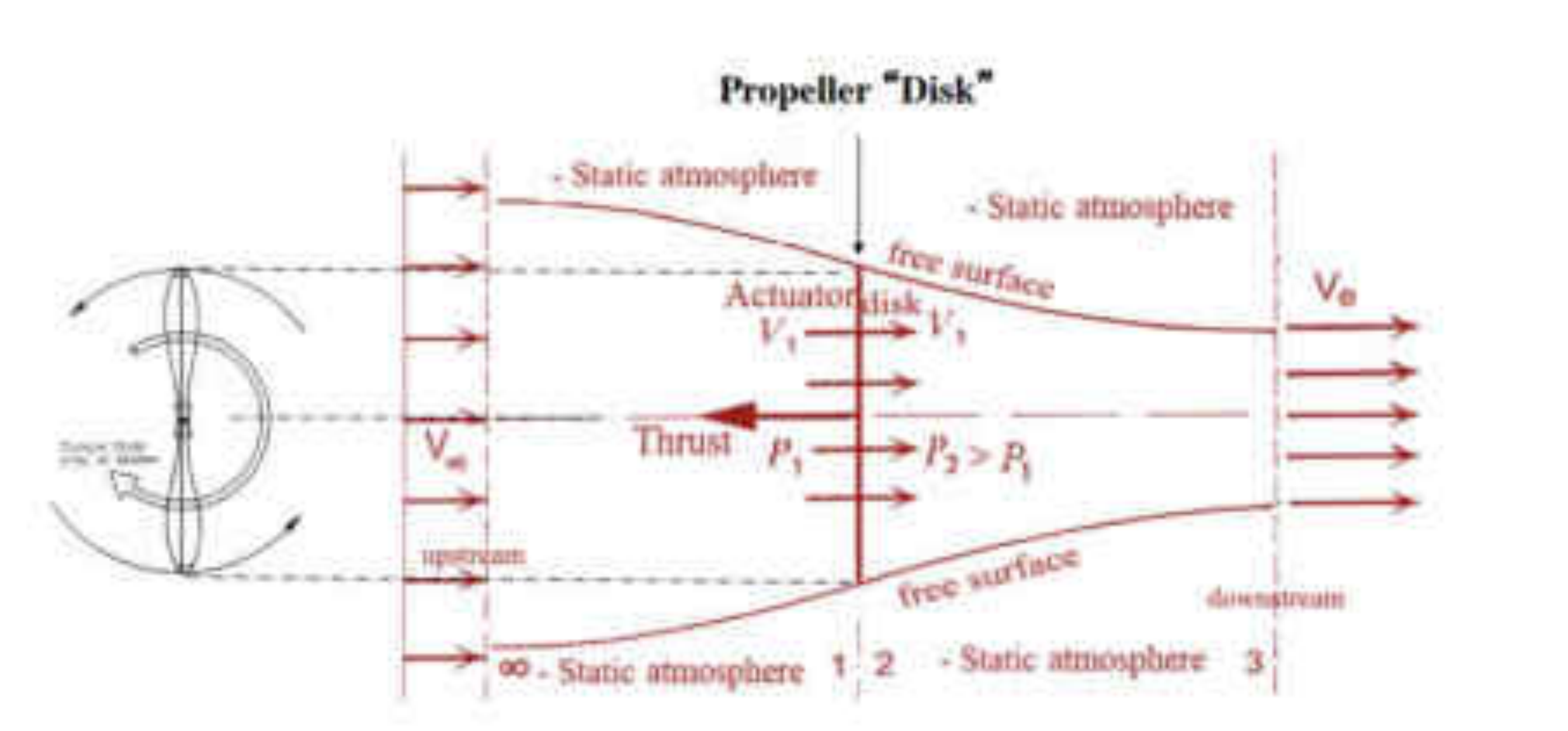
- An analytical approach
- To evaluate propeller performance
- Based on linear momentum conservation
  - Associated assumptions
- Propeller modeled as an actuator disk where momentum exchange results in is a step increase in pressure



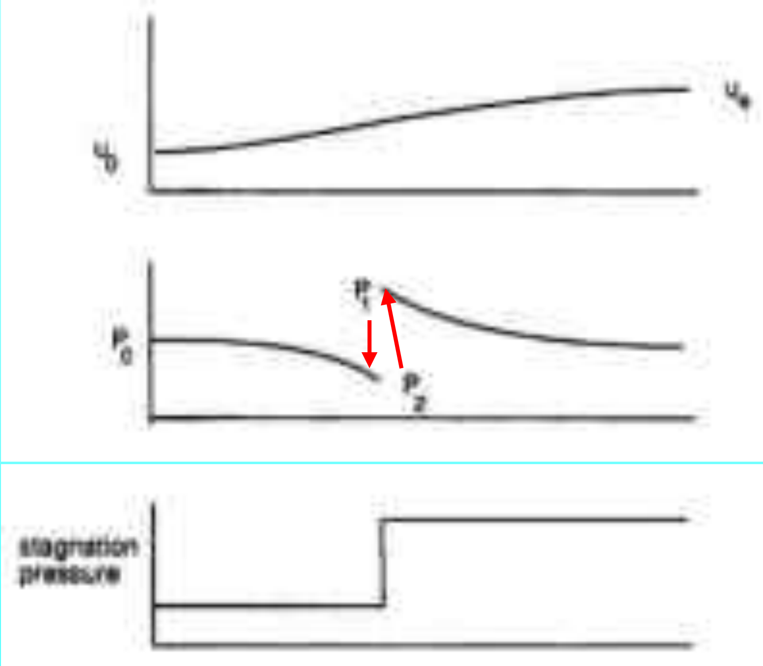
Actuator Disk Assumptions

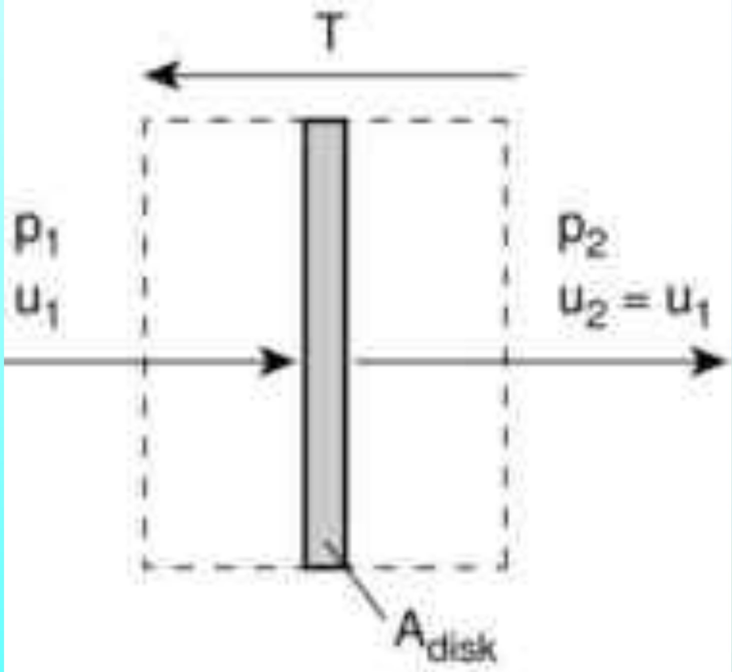
- *Rotation imparted to the flow is neglected*
- *Assume the Mach number is low so that the flow behaves as an incompressible fluid*
- *No work interaction with flow outside the stream tube*
- *The flow is steady*
- *Moving blades are conceptually modeled as one thin steady disk that has approximately the same effect on the flow as the moving blades - the ``actuator disk''*
- *Across the actuator disk, assume that the pressure changes abruptly*
- *Velocity varies in a continuous manner*

The Actuator Disk Parameters



The Actuator Disk Flow





- Thrust: from Δp (*across disk*) and ΔV (*across CV*)

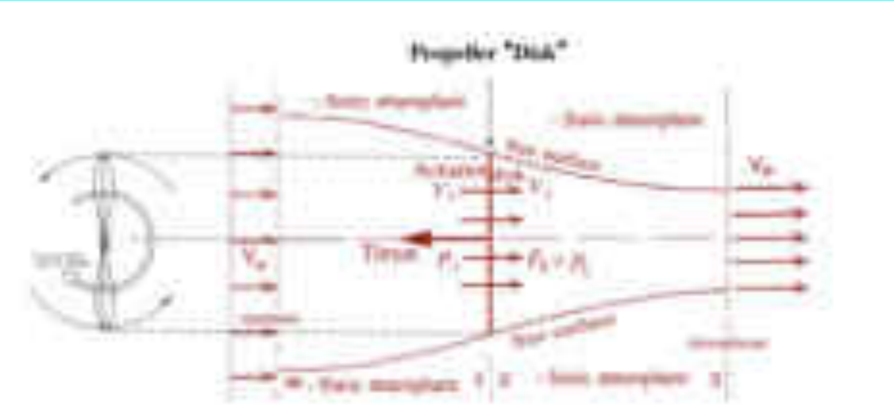
- Thrust on the disk: $$T = A_{\text{disk}}(p_2 - p_1),$$

$$\text{Power} = T u_{\text{disk}} = A_{\text{disk}}(p_2 - p_1) u_{\text{disk}} = \dot{m}(u_e - u_0) u_{\text{disk}}.$$

- From energy considerations, the power imparted at the disk:

$$\text{Power} = \dot{m}\left(\frac{u_e^2 - u_0^2}{2}\right) = \dot{m}(u_e - u_0)\frac{(u_e + u_0)}{2}.$$

- Hence the velocity at the disk:



$$u_{\text{disk}} = \frac{(u_e + u_0)}{2}.$$

$u_o = V_{inf}$ = Flight velocity

- To determine pressure difference across the disk (P_2-P_1):
  - Applying Bernauli's equation upstream and downstream of the disk:

$$p_1 + \frac{1}{2}\rho u_{\mathrm{disk}}^2 = p_0 + \frac{1}{2}\rho u_0^2$$

$$p_2 + \frac{1}{2}\rho u_{\mathrm{disk}}^2 = p_0 + \frac{1}{2}\rho u_c^2$$

$$p_1 - p_2 = \frac{1}{2}\rho(u_c^2 - u_0^2)$$

- u_e & u_o are measurable – than u_{disk}
- Mass flow rate: $$\dot{m} = \rho u_{\text{disk}} A_{\text{disk}} = \rho A_{\text{disk}} \frac{(u_e + u_0)}{2},$$
- Hence thrust: (as seen earlier)

$$T = \rho A_{\text{disk}} \frac{(u_e + u_0)}{2}(u_e - u_0) = \rho A_{\text{disk}} \frac{(u_e^2 - u_0^2)}{2},$$

- Exit velocity can be written as:

$$\left(\frac{u_e}{u_0}\right)^2 = \frac{T}{A_{\text{disk}} u_0^2 \frac{\rho}{2}} + 1,$$

- $u_{disk}/u_0 = \frac{1}{2}(1+u_e/u_o)$

$$\frac{u_{\text{disk}}}{u_0} = \frac{1}{2}\left[\frac{T}{A_{\text{disk}} u_0^2 \frac{\rho}{2}} + 1\right]^{\frac{1}{2}} + \frac{1}{2}$$

- Power

$$\text{Power} = T u_{\text{disk}} = \frac{1}{2} T u_0 \left[\left(\frac{T}{A_{\text{disk}} u_0^2 \frac{\rho}{2}} + 1\right)^{\frac{1}{2}} + 1\right].$$

The Power

- Actuator disk theory gives the minimum power required to drive the propeller, for the specified conditions
- The actual power will be significantly higher than this, by 10-15%
- Propulsive efficiency,

$$\eta_{\text{prop}} = \frac{2}{1 + \frac{u_e}{u_0}}.$$

$$\eta_{\text{propulsive}} = \frac{2}{1 + \left(\frac{T}{A_{\text{disk}} u_0^2 \frac{\rho}{2}} + 1 \right)^{\frac{1}{2}}}$$

Thrust Calculation

- Calculate the thrust produced, based on actuator disk assumptions, by the propeller of turboprop, cruising at 250 kmph. Diameter of the propeller is 1.5 m and the maximum velocity downstream of the propeller is 97 m/s. Assume sea level conditions for the flight.

V1-kmph	V1 m/s	D	V2	Rho	A	T
250	69.44	1.5	97	1.23	1.767	4964.142

$$T = \rho A_{\text{disk}} \frac{(u_e + u_0)}{2}(u_e - u_0) = \rho A_{\text{disk}} \frac{(u_e^2 - u_0^2)}{2},$$

Disk Loading…

- Propulsive Efficiency,

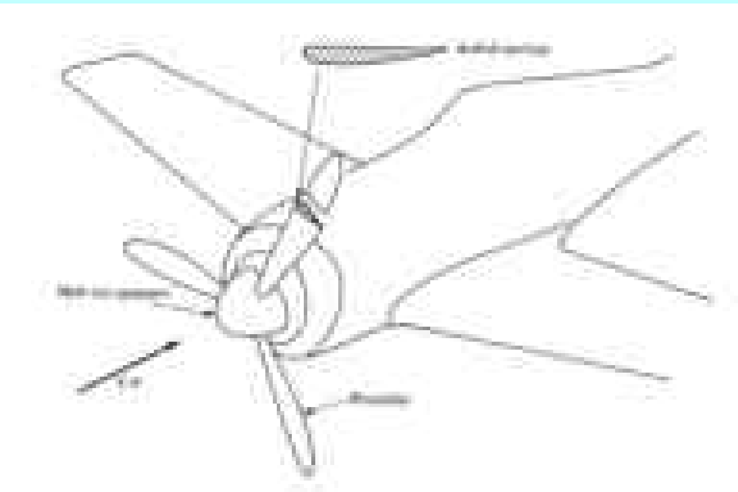
$$\eta_{propulsive} = \frac{2}{1 + \left(\frac{T}{A_{disk} u_0^2 \frac{\rho}{2}} + 1 \right)^{\frac{1}{2}}}$$

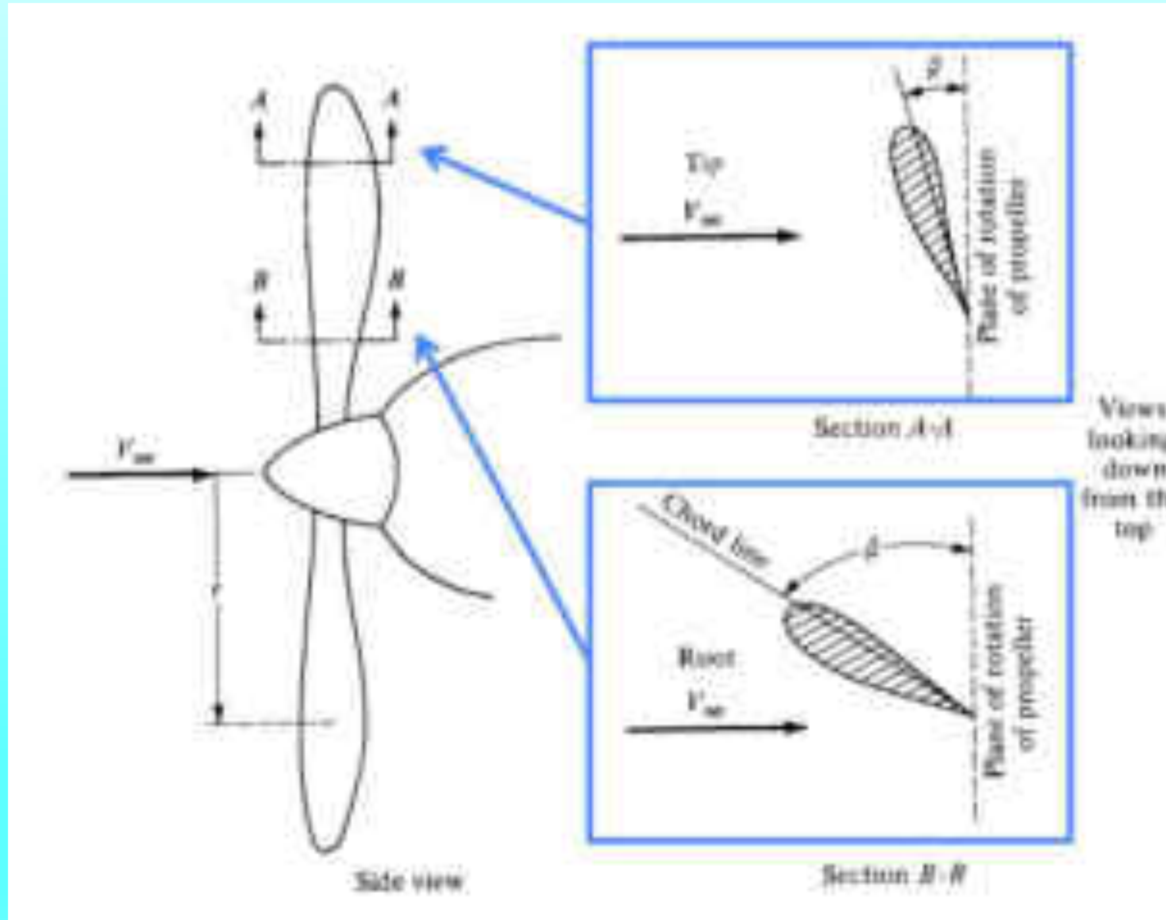
- How to improve efficiency, for a given flight speed ?
  - Keep T/A, "*Propeller disk loading*", at a low value
  - Requires large diameter for propellers
  - Limited by:
    - High tip-speed (rω)
    - Ground clearance issues
    - High Stress → Structural problems

Propeller Terminology

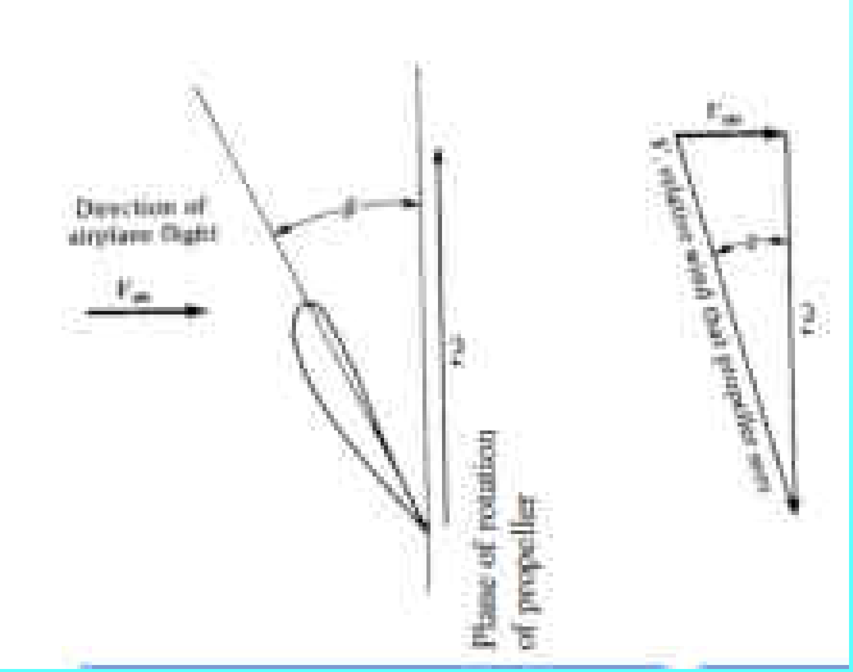
- Geometric Pitch:
  - The axial distance that the propeller will move forward for one revolution if it is moving (theoretically) through a solid object.
- Effective Pitch:
  - The axial distance that the propeller will move forward for one revolution through the air. Small blade angle results in low pitch. Large blade angle results in high pitch.
    - Because of this, the terms propeller blade angle and pitch are used interchangeably.
- Propeller slip:
  - The difference between geometric pitch and effective pitch

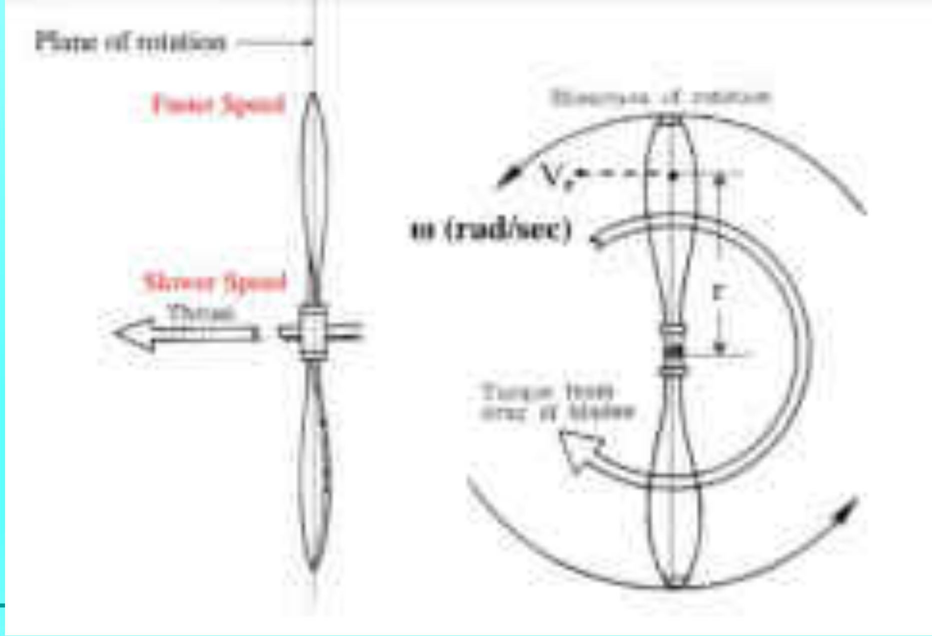
Cross-Section…



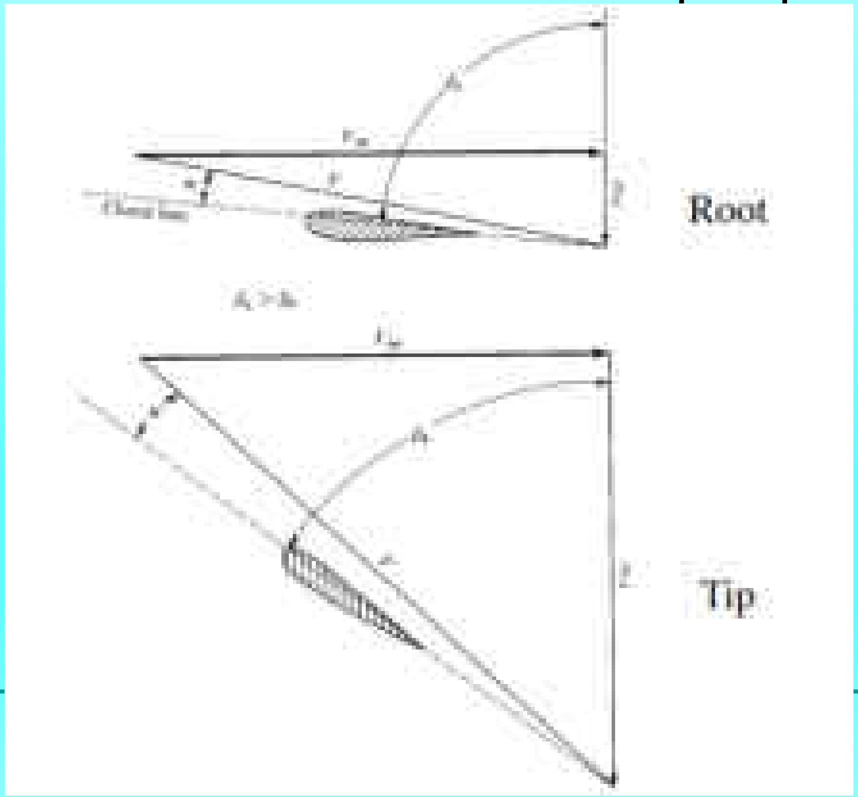


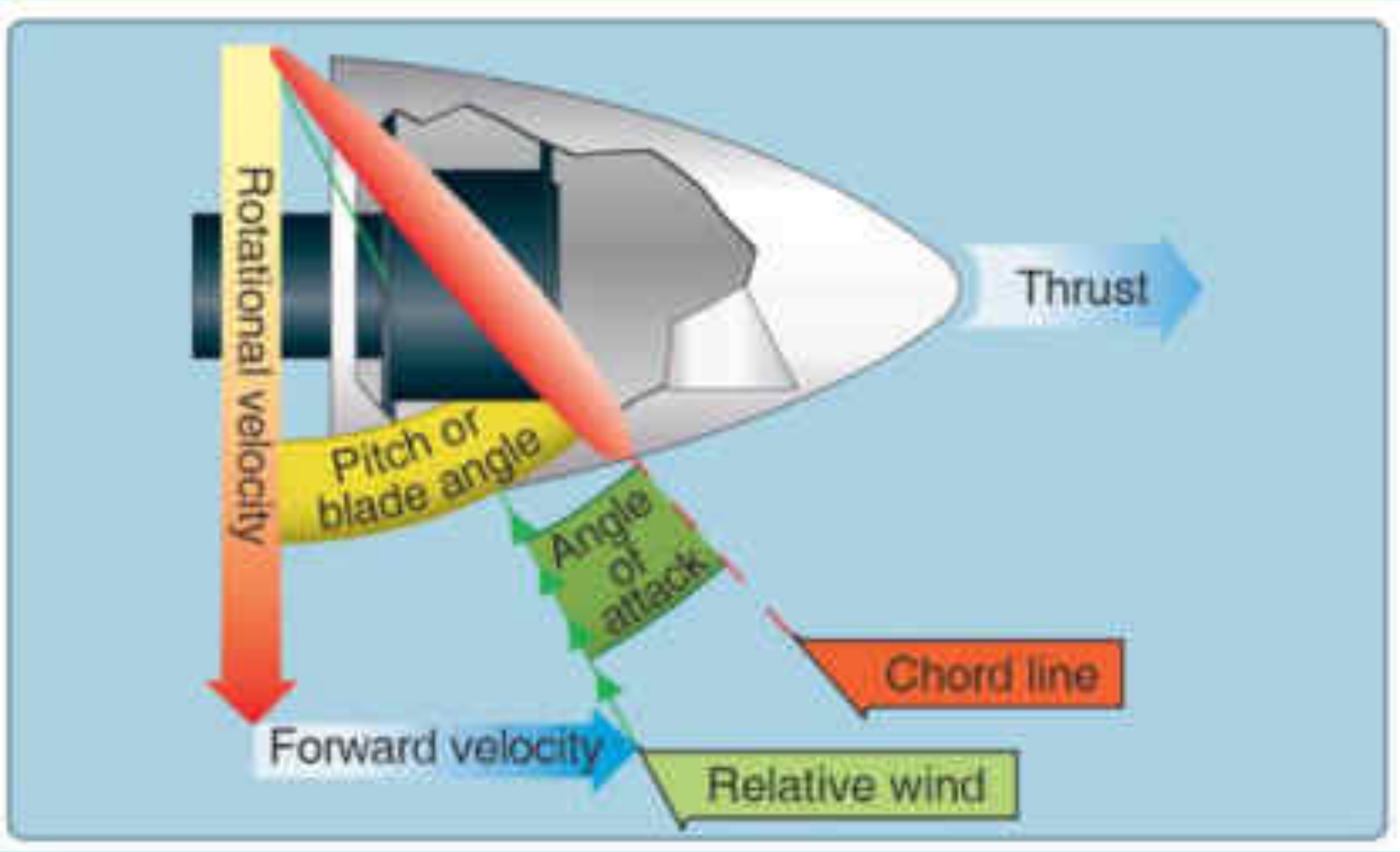
Relative Velocity, Pitch Angle





- Blade angle:
  - The angle between the propeller blade chord and plane of rotation at any blade cross-section. It has is maximum value around the blade hub and decreases towards the tip of propeller blade




Rotational velocity
Thrust
Pitch or blade angle
Angle of attack
Chord line
Forward velocity
Relative wind

Propeller Performance Characteristics

Actual Efficiency of the Propeller

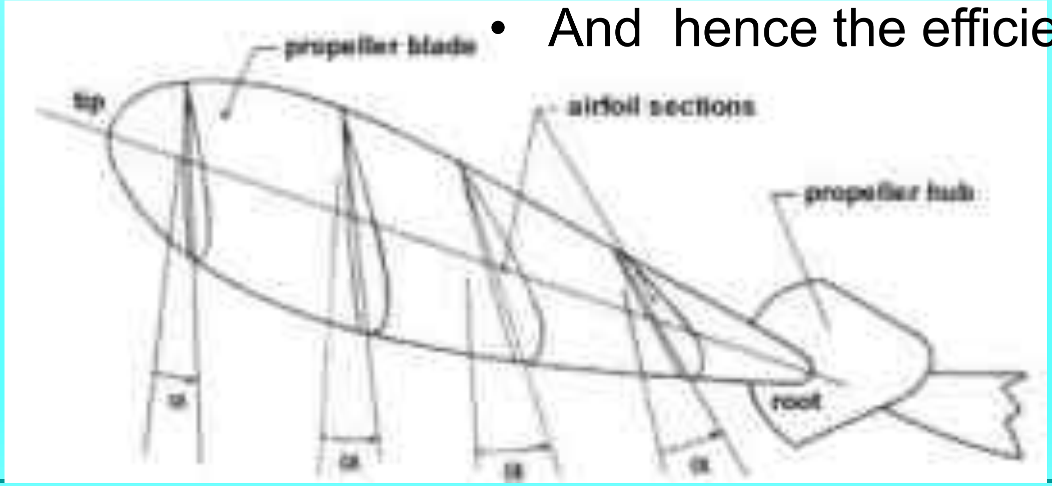
$$\eta = \frac{P_A}{P} = \frac{\textbf{Actual}\text{ Power Available }\textbf{for Propulsion}}{\text{Shaft Brake Power (Propeller Power from Engine}}$$

Since $P_A = T_A V_\infty \quad \rightarrow \quad \boxed{\eta = \frac{T_A V_\infty}{P}}$

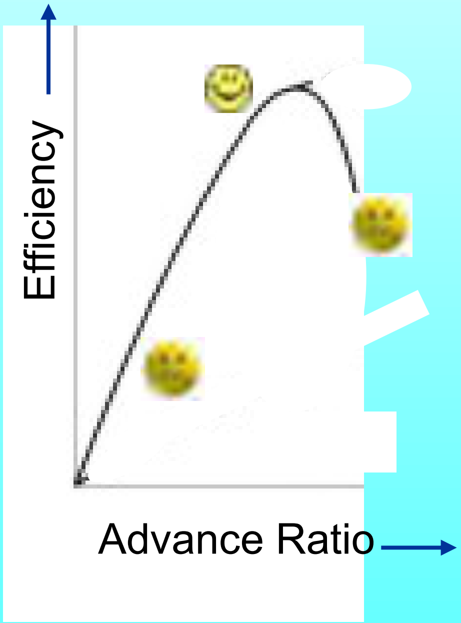
"Fixed Pitch" Propeller

Blade angle varies with radius, but cannot be changed a t a given section

- β is constant at any given r
- For a given N, rω is constant
- But as forward velocity changes, the angle of attack will change
- And hence the efficiency



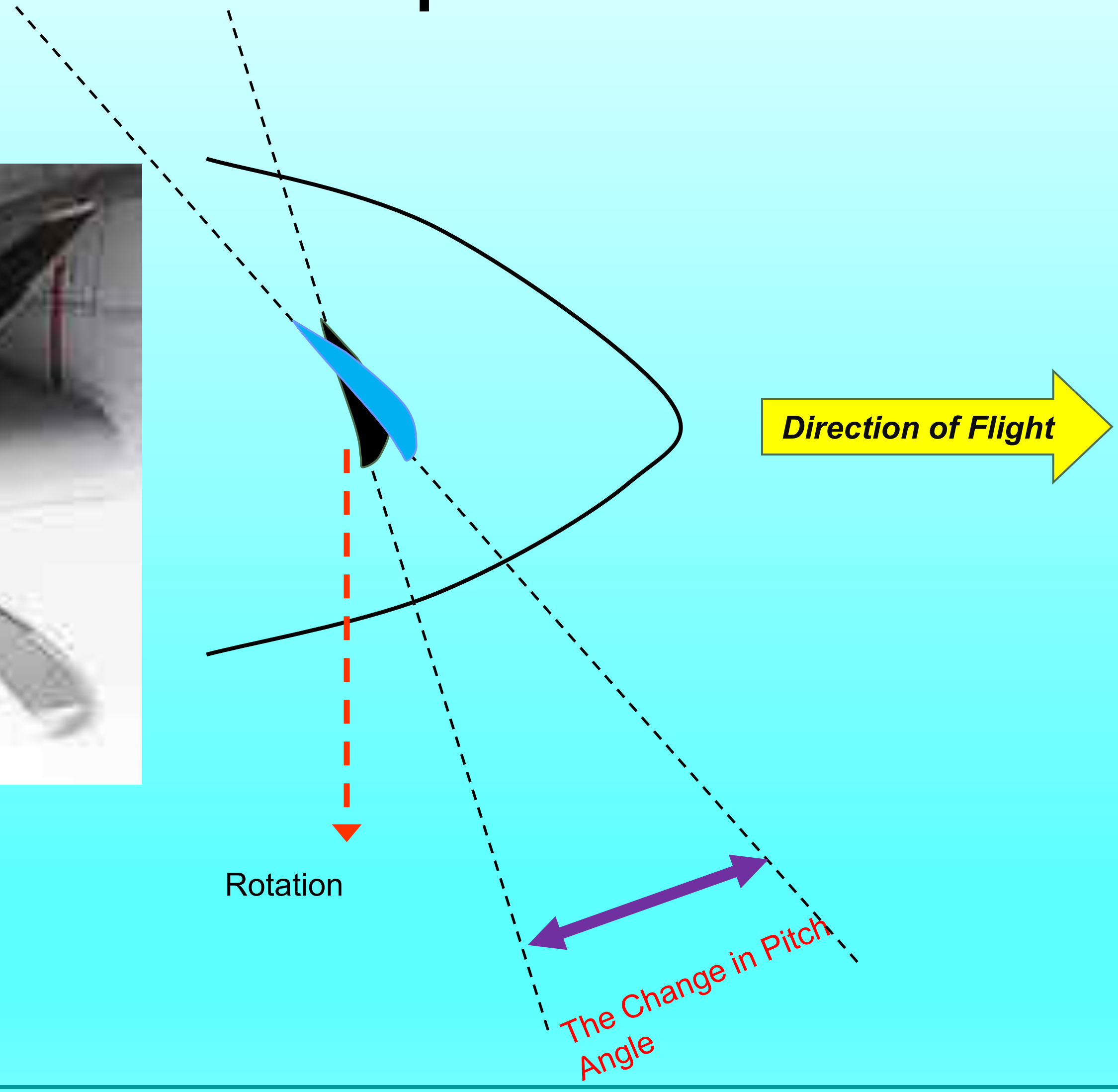
Propeller Efficiency – Fixed Pitch Propellers



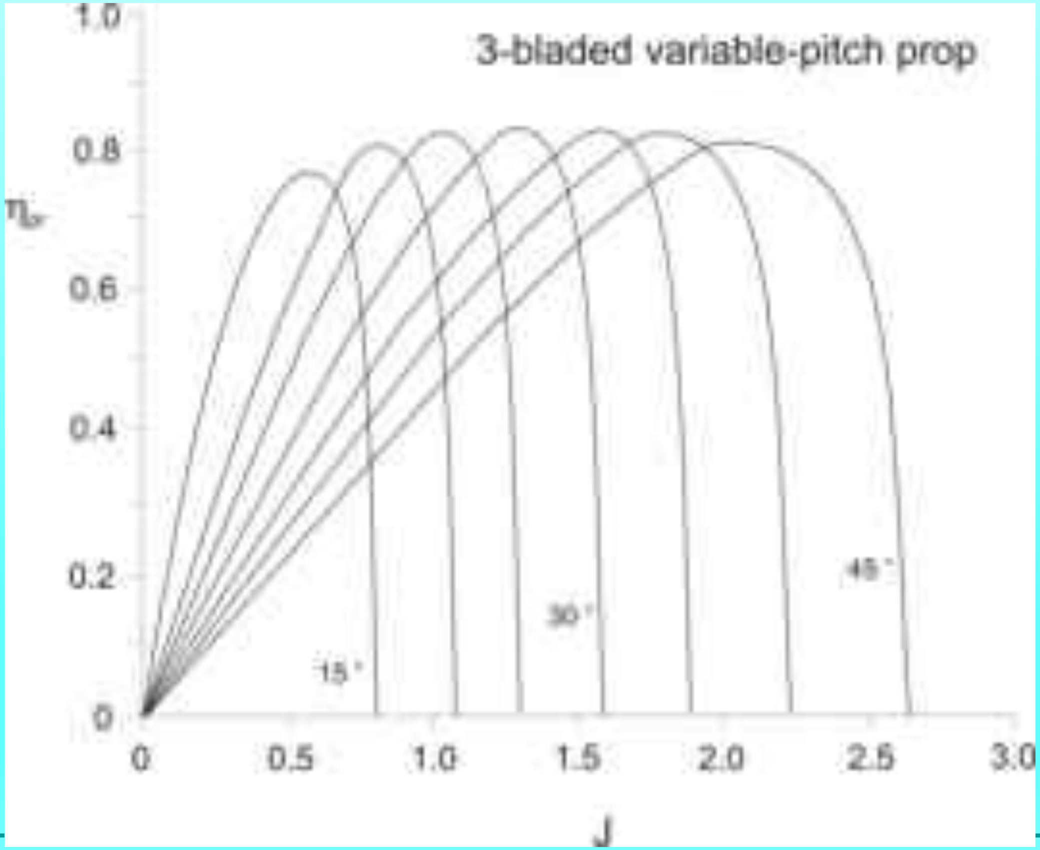
- *Can be efficient only over a narrow range of forward speed*

Variable Pitch Propeller: What it is..

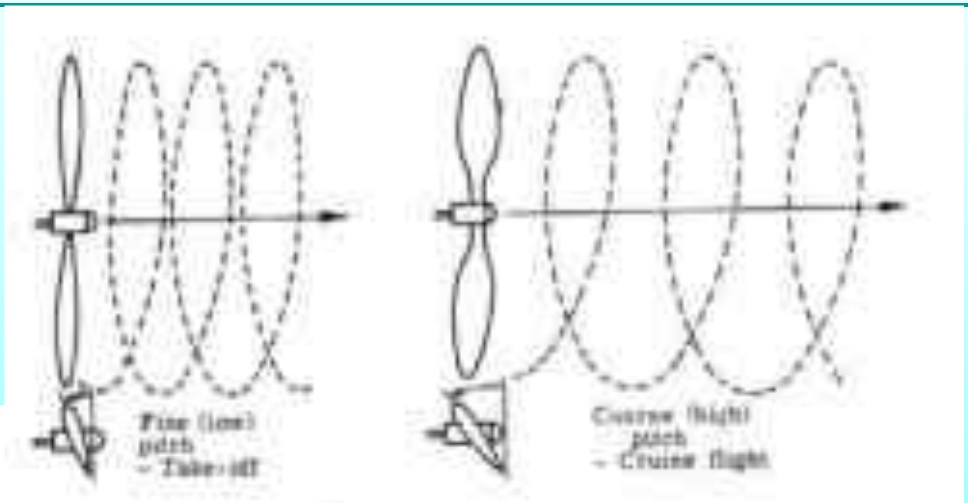
Variable Pitch Propeller: What it Does

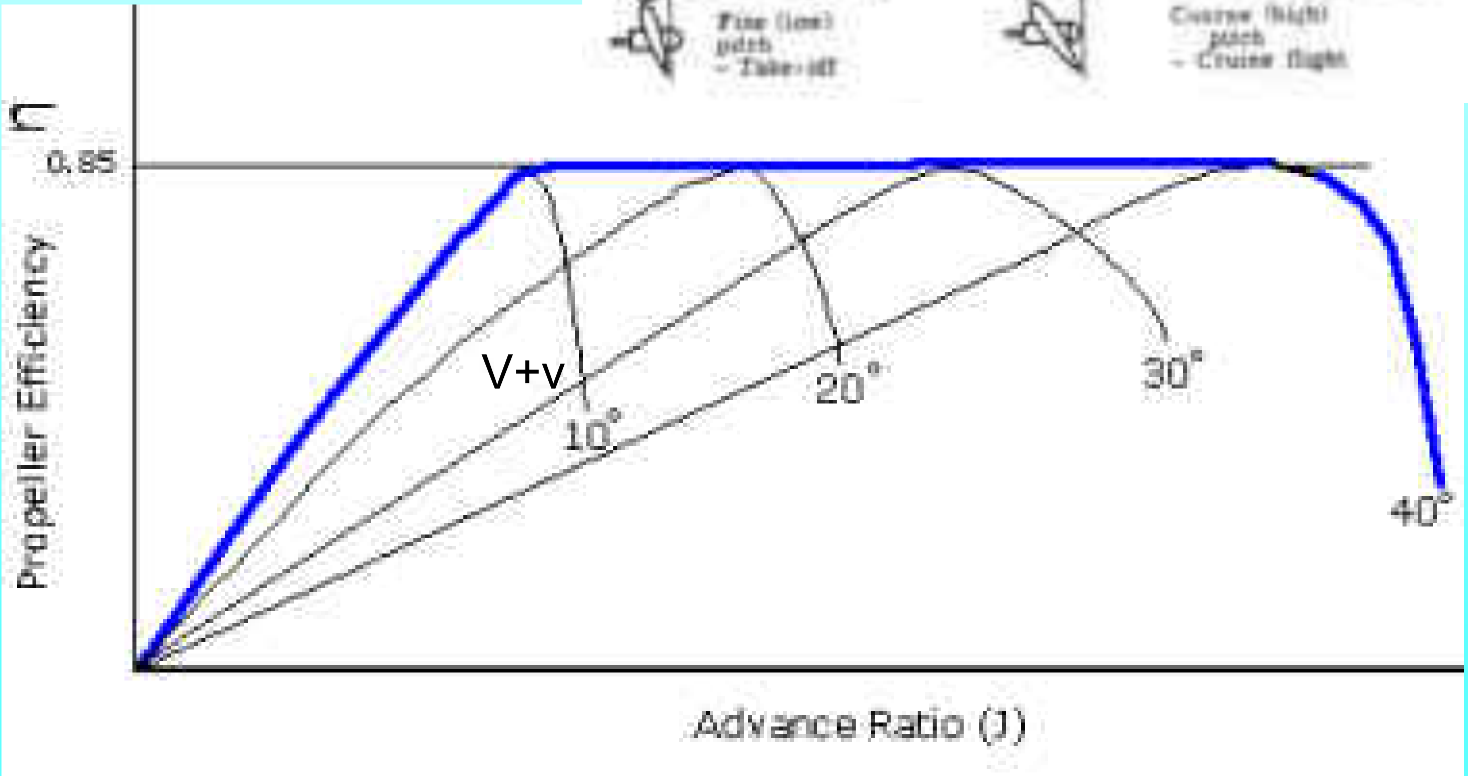


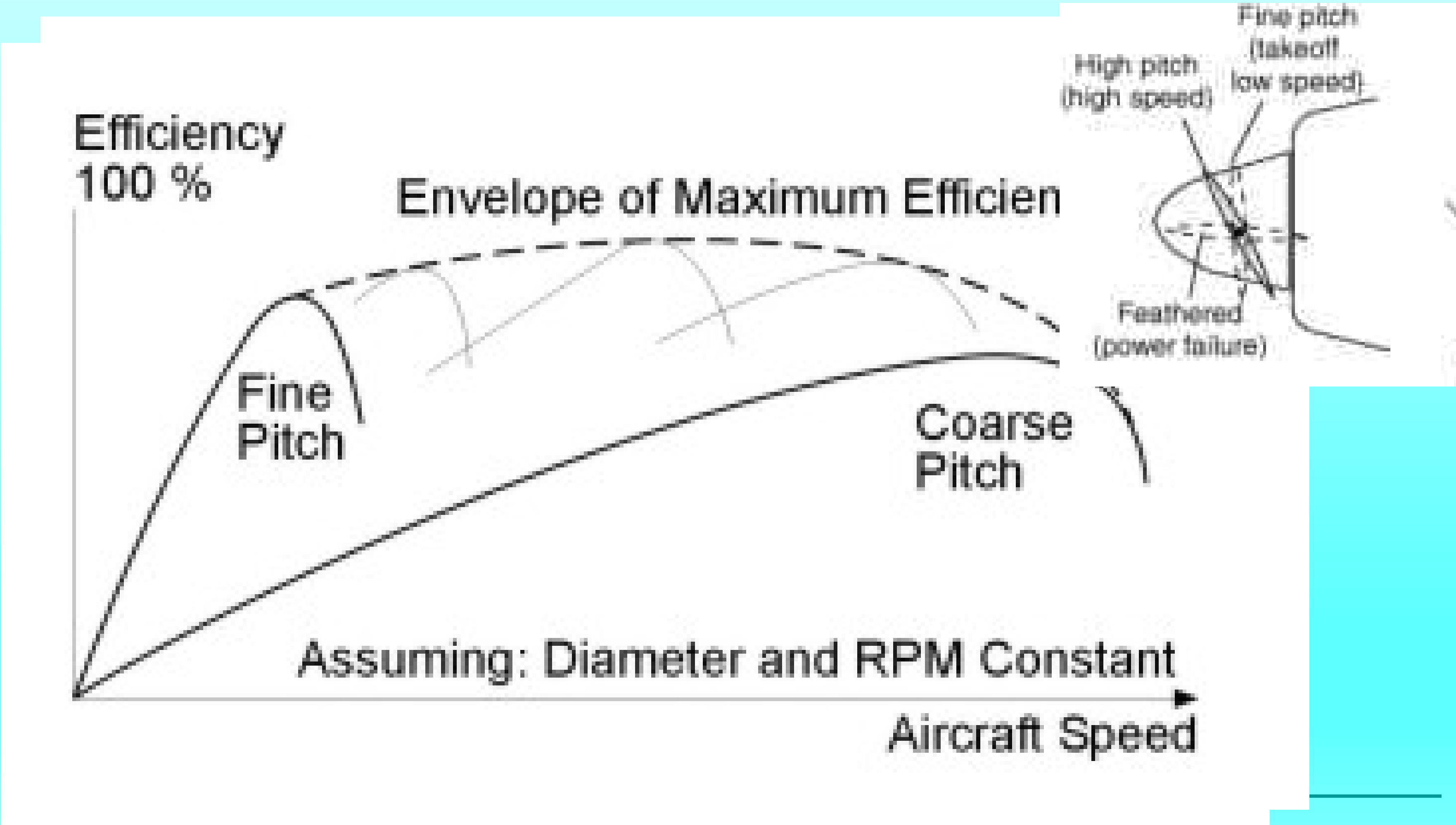
Efficiency Variation for Varying Pitch



Variable Pitch Operation






Efficiency
100 %
Envelope of Maximum Efficien
Fine
Pitch
Coarse
Pitch
Assuming: Diameter and RPM Constant
Aircraft Speed
Fine pitch
(takeoff
low speed)
High pitch
(high speed)
Feathered
(power failure)

Propellers: The Technology Progression

- “**Constant Pitch**” Propellers:
  - Blade angle the same across radius; angle of attack varies with radius
- **Fixed Pitch Propellers**:
  - Pitch varies with radius, but remains *fixed* for a given radius
  - Works efficiently at the design value of Advance Ratio (Design speed)
  - Performance poor at other speeds
- **Variable-pitch Propellers**:
  - Pilot can vary blade pitch in flight
  - Can achieve high propeller efficiency over a wide range of speeds
- **Constant speed propellers**: Pitch is controlled by a governor to maintain optimum RPM
  - Optimizes power as per requirement

Performance Analysis: The Approach

- Use of non-dimensional parameters to characterize performance
  - Cuts down the number of independent variables
  - Facilitates comparison
  - Drastic reduction in the number of experiments required
  - Easy to present performance data

Parameters: Advance Ratio, J

- Advance Ratio, $J = \dfrac{V}{ND}$
  - Unifies the flight velocity, rotational speed and the propeller diameter into one parameter
  - A measure of the advancement of the propeller in one revolution, in terms of the diameter
  - Propellers with the same *J* are *geometrically similar*

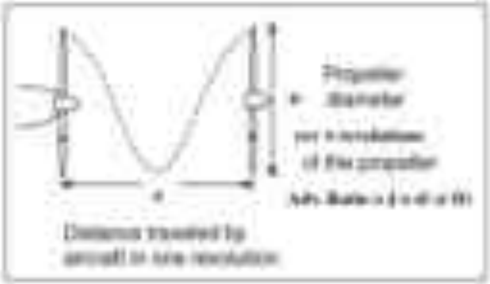
Advance Ratio

The ratio between the **distance an aircraft moves from one revolution of a propeller(s)**, under specified conditions, **and the propeller's diameter**. It is the ratio of the forward speed divided by the product of rotational speed and the diameter. All propeller performances are compared at the same advance ratio.

$$J = \frac{V_\infty}{nD}$$

Where n = propeller revolutions per sec

D = propeller diameter



$$J = \frac{d}{rev \cdot D} = \frac{V \cdot t \cdot \frac{1}{t}}{rev \cdot \frac{1}{t} \cdot D} = \frac{V}{nD}$$

Where $d = Vt$

$n = rev / t$

Efficiency

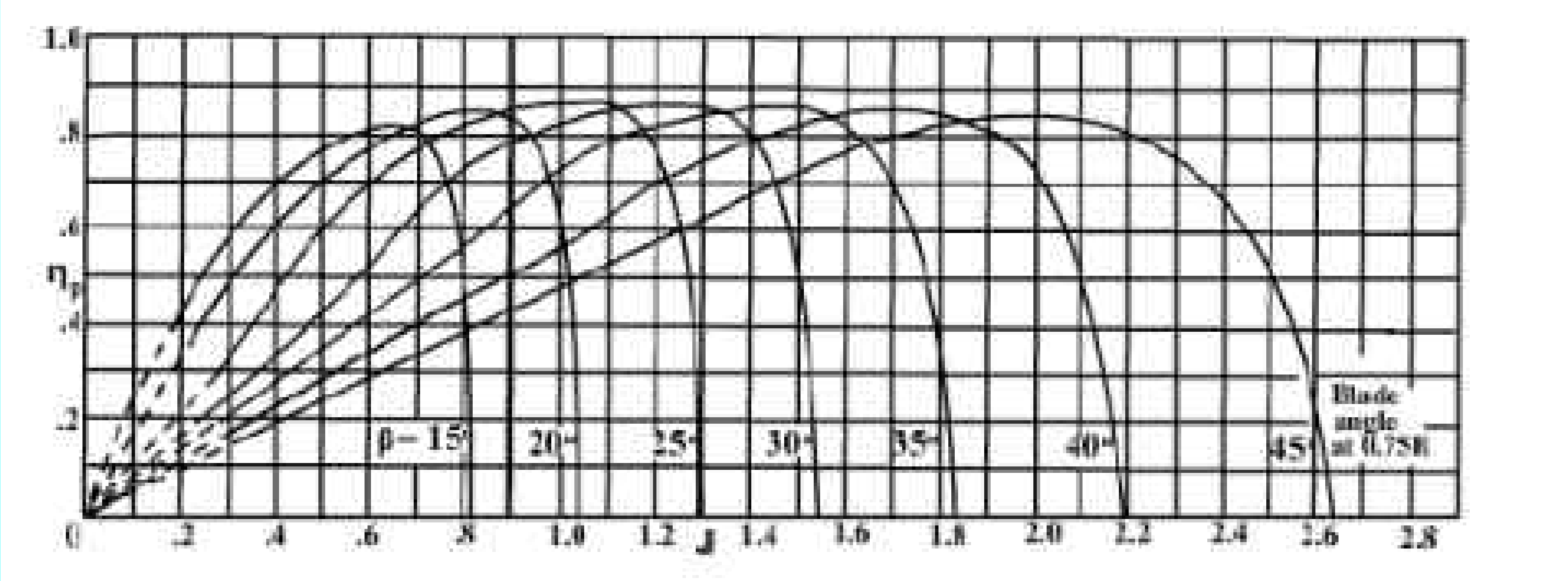
- Ratio between the propulsive power and the power input to the propeller shaft
- A measure of the effectiveness in converting shaft power to propulsive thrust
- Efficiency, $\eta = \frac{TV}{P}$

Torque & Power Coefficients

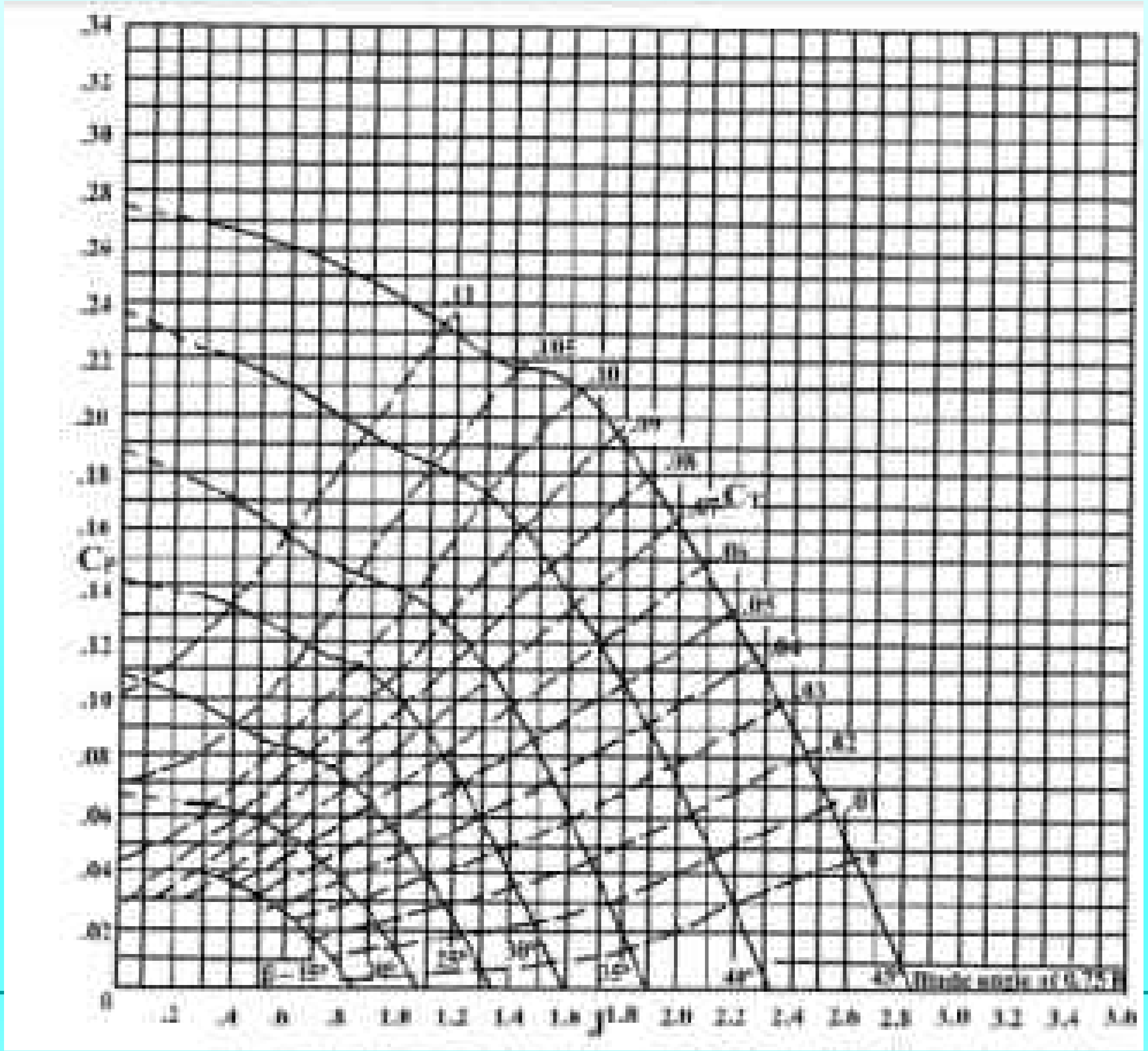
- Thrust Coefficient $C_T = T/\rho n^2 D^4$
- Torque Coefficient, $C_Q(Re, J, M_{tip}) = \dfrac{Q}{\rho N^2 D^5}$
- Power coefficient:
  - Power required to drive the propeller = Torque X Angular velocity
  - Power coefficient, $C_p = \dfrac{P}{\rho N^3 D^5}$

Performance Curves: Efficiency VS Advance Ratio, for various pitch angles

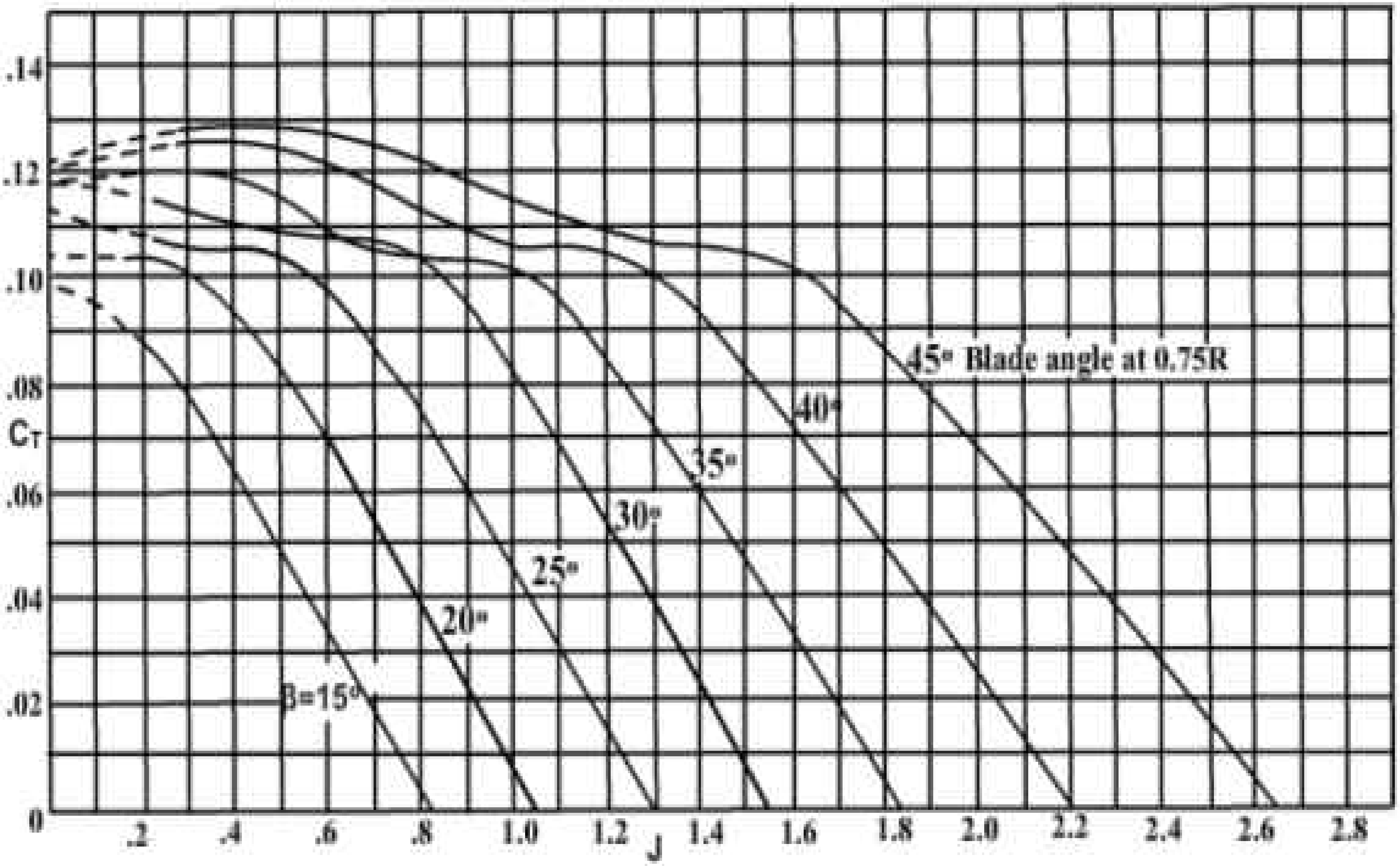
- Note that: $\eta = J\frac{C_T}{C_P}$



Power Vs Advance Ratio

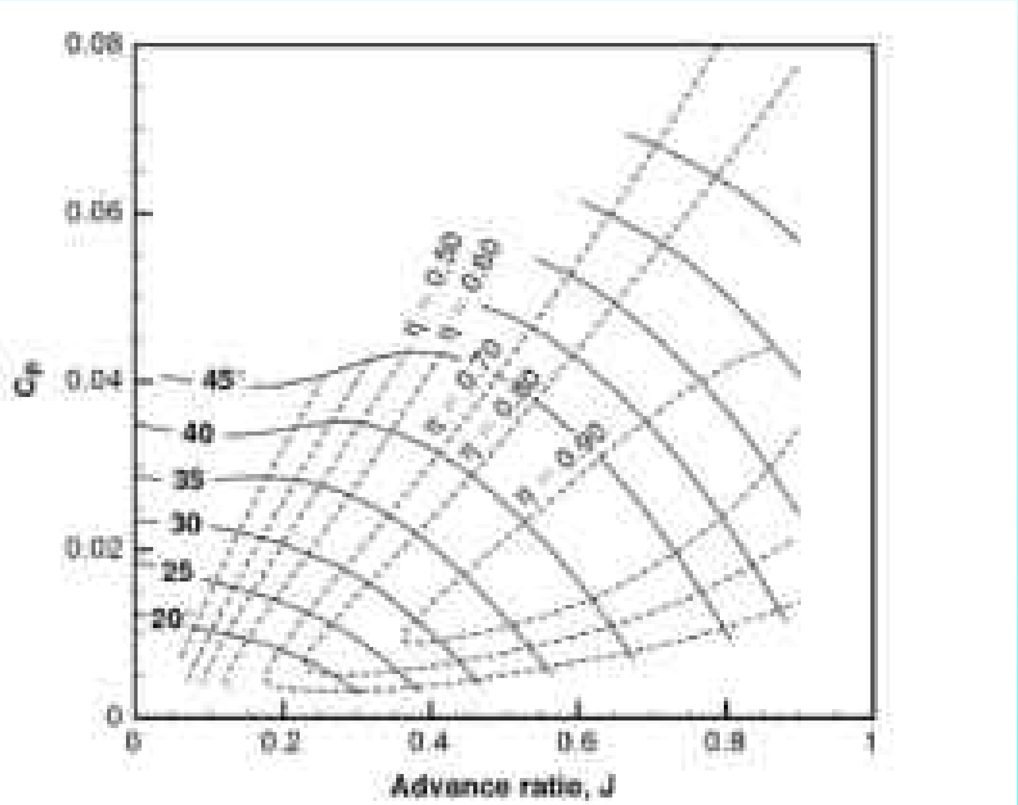


Thrust Vs Advance Ratio



Performance Characteristics, Power & Efficiency

- Solid lines for various pitch angles
- Helps to identify the pitch angle for maximum efficiency, for a given power requirement



Numerical Problem

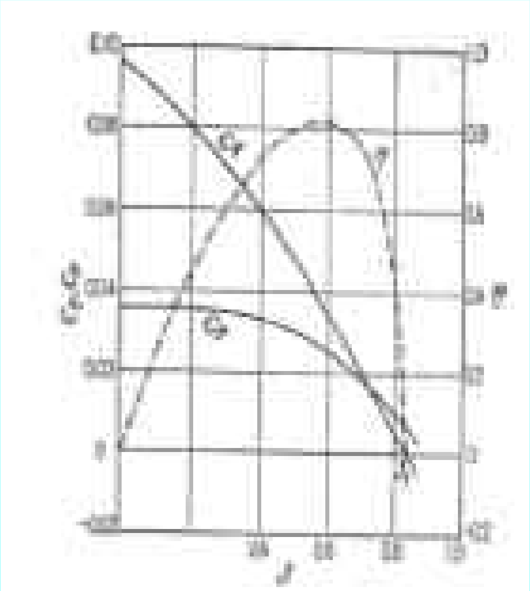
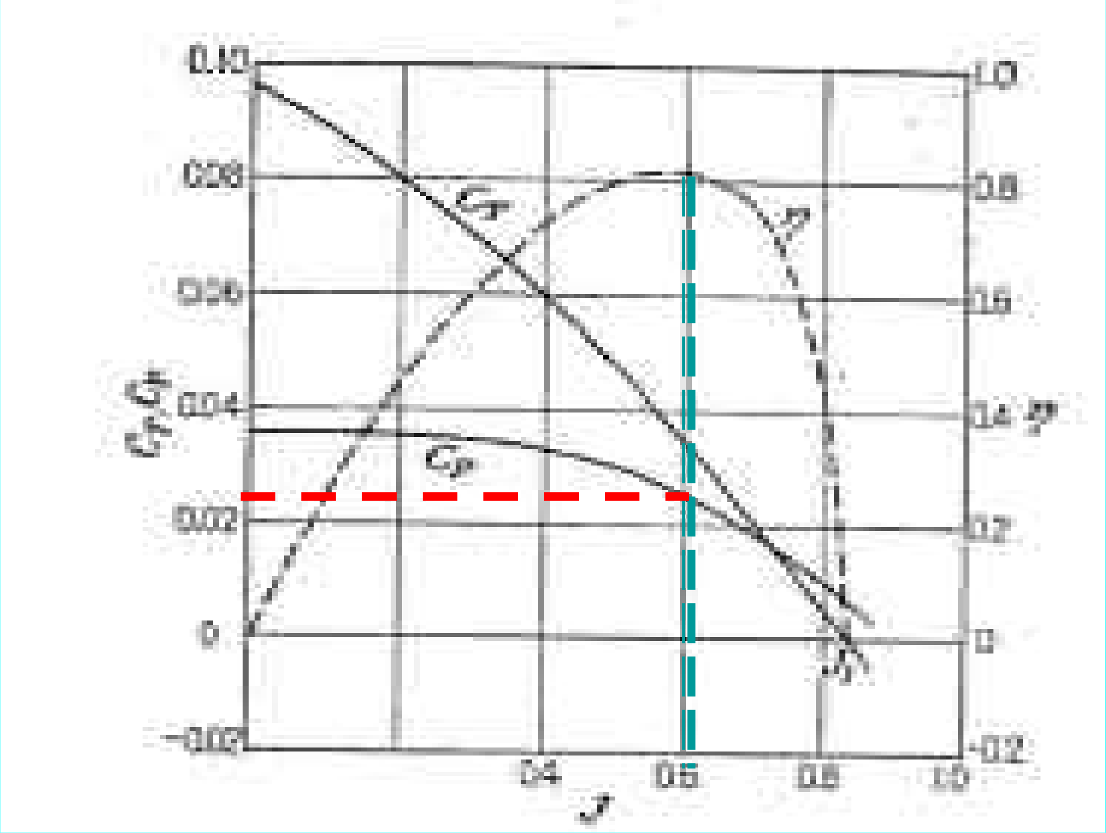


Figure shows the performance characteristics of a propeller which has a diameter of 2.5 m.

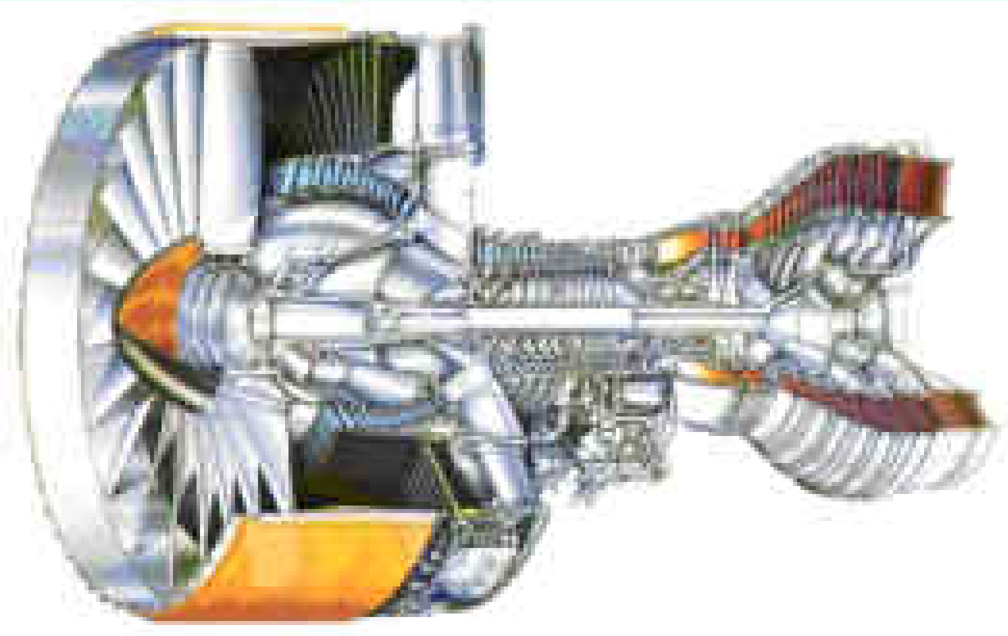
If the propeller operates at maximum efficiency while flying at 50 m/s, estimate the turbine power required to drive the propeller at that point

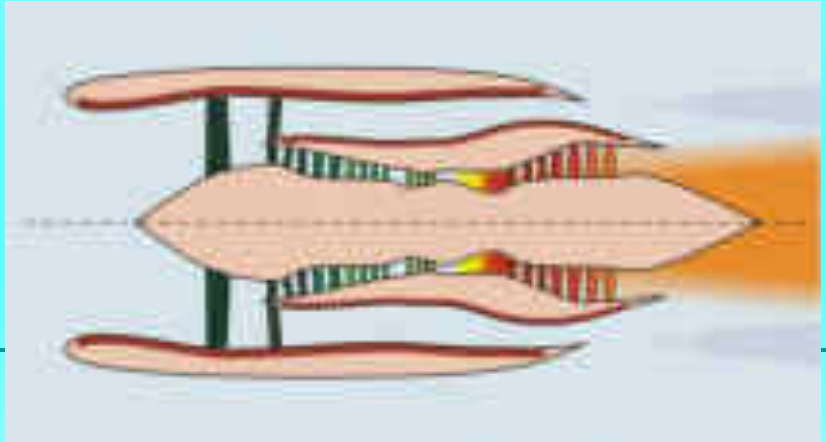

0.10
0.08
0.06
0.04
0.02
0
-0.02
1.0
0.8
0.6
0.4
0.2
0
-0.2
0.4
0.6
0.8
1.0
J
η
C_T, C_N

Turbojets Vs Turboprops

- The Merits:
  - Turbojets: Superior high-speed performance characteristics, ability to operate at supersonic flight speeds, less weight & frontal area, simple design, high TWR, high specific thrust
  - Turboprops: High efficiency (low TSFC) at low subsonic Mach numbers (Mach 0.3 to 0.6), ideally suited for short range applications, fairly good specific thrust
- Can the advantages be combined for better overall performance and wider range of operation ?

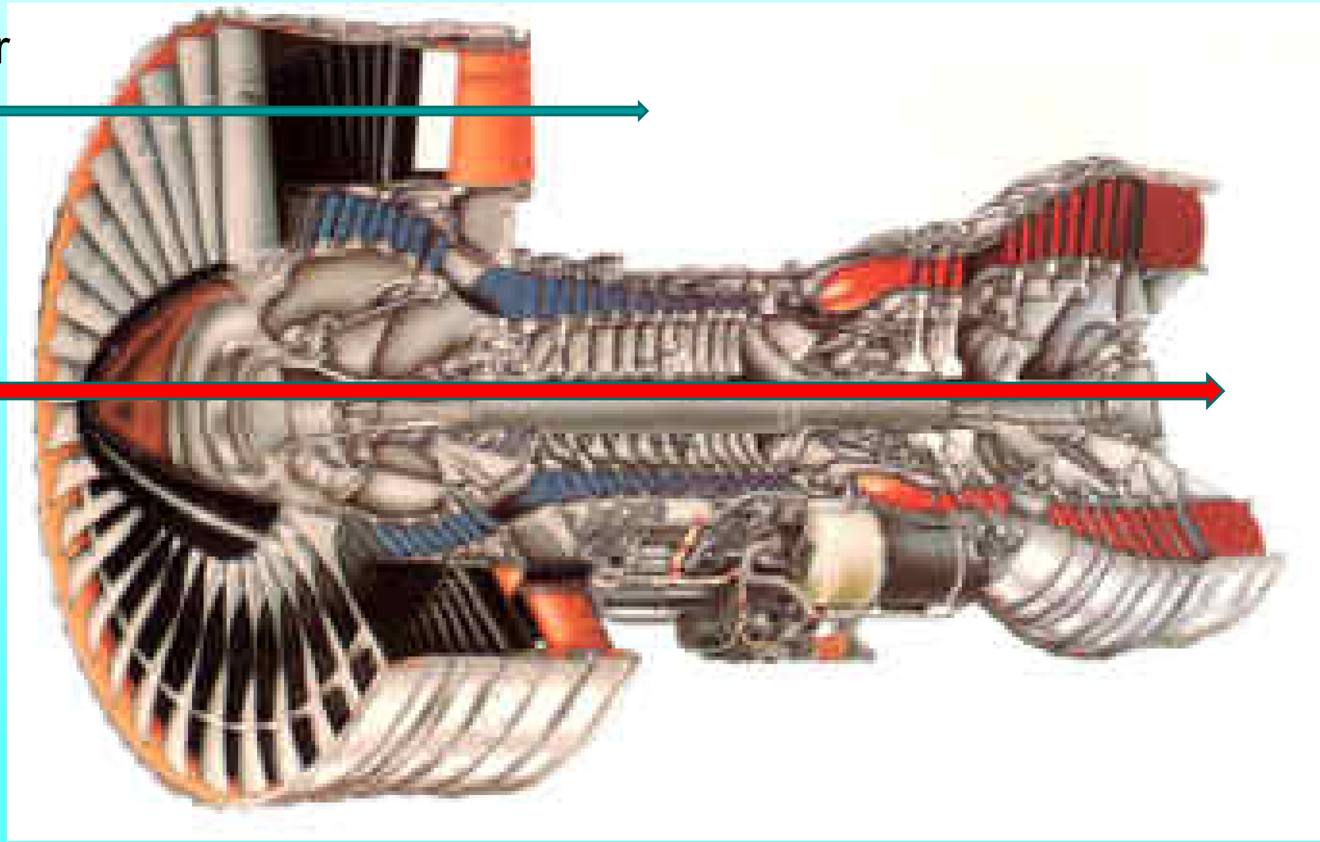
Turbofan Engine



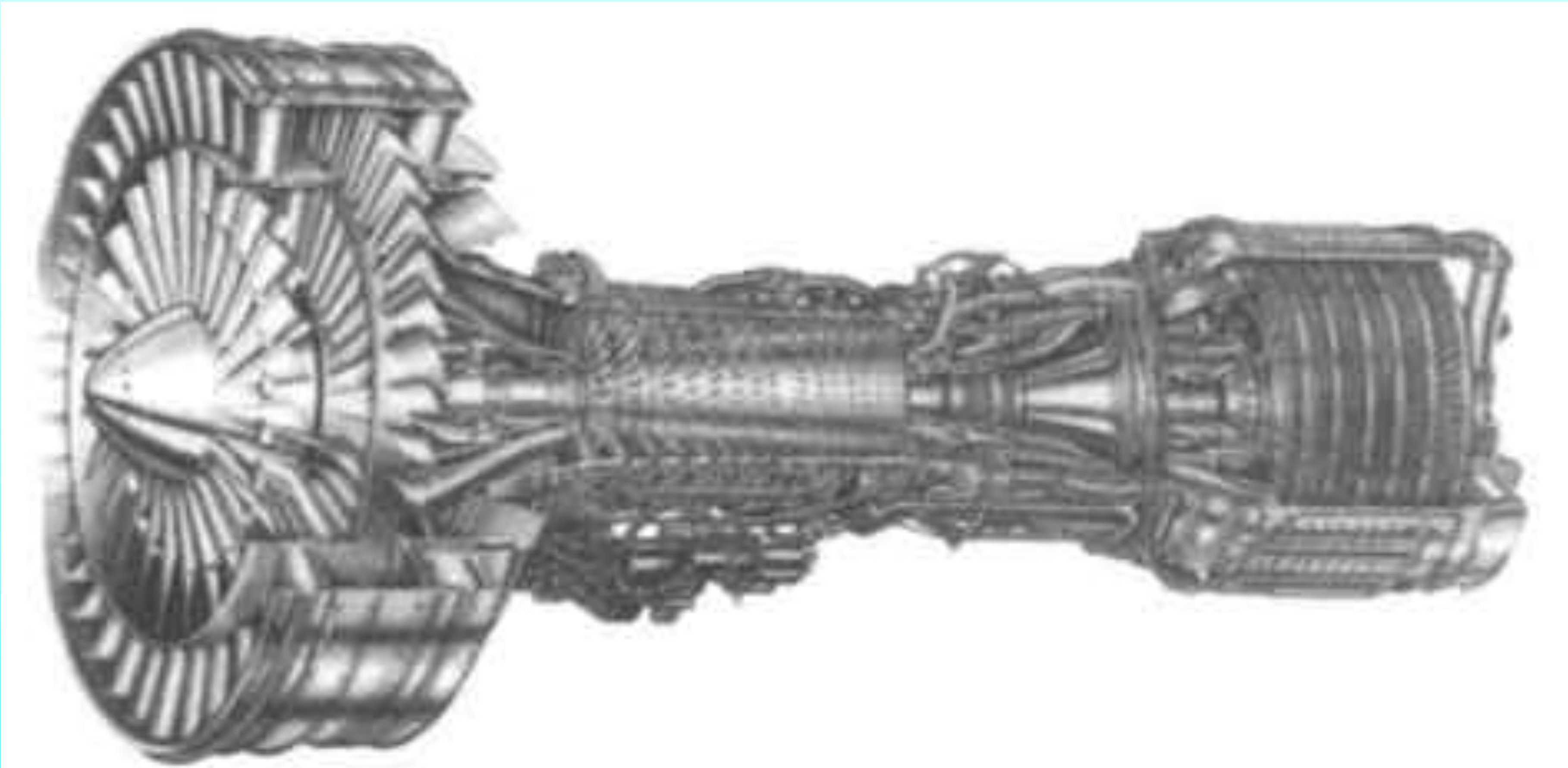


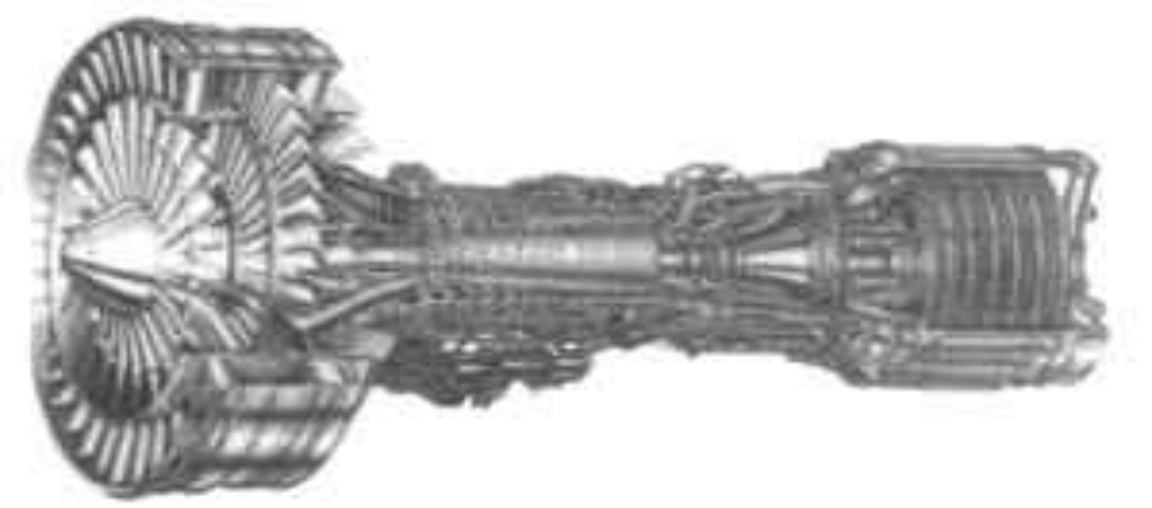
Bypass Air..

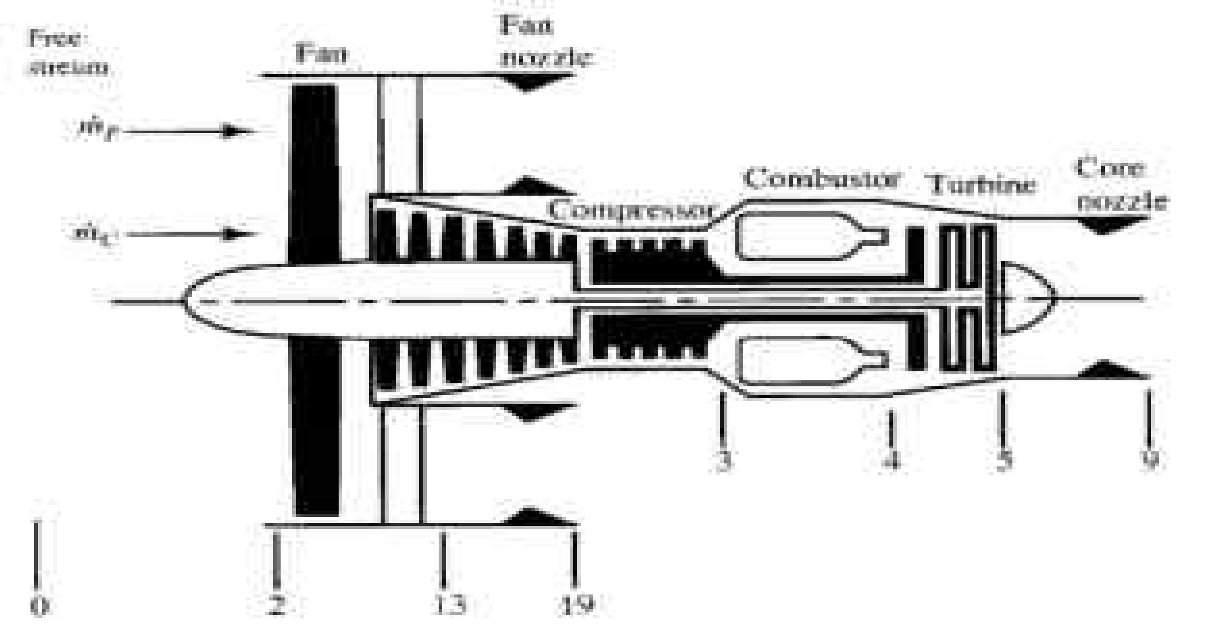
Bypass air



“Core” air

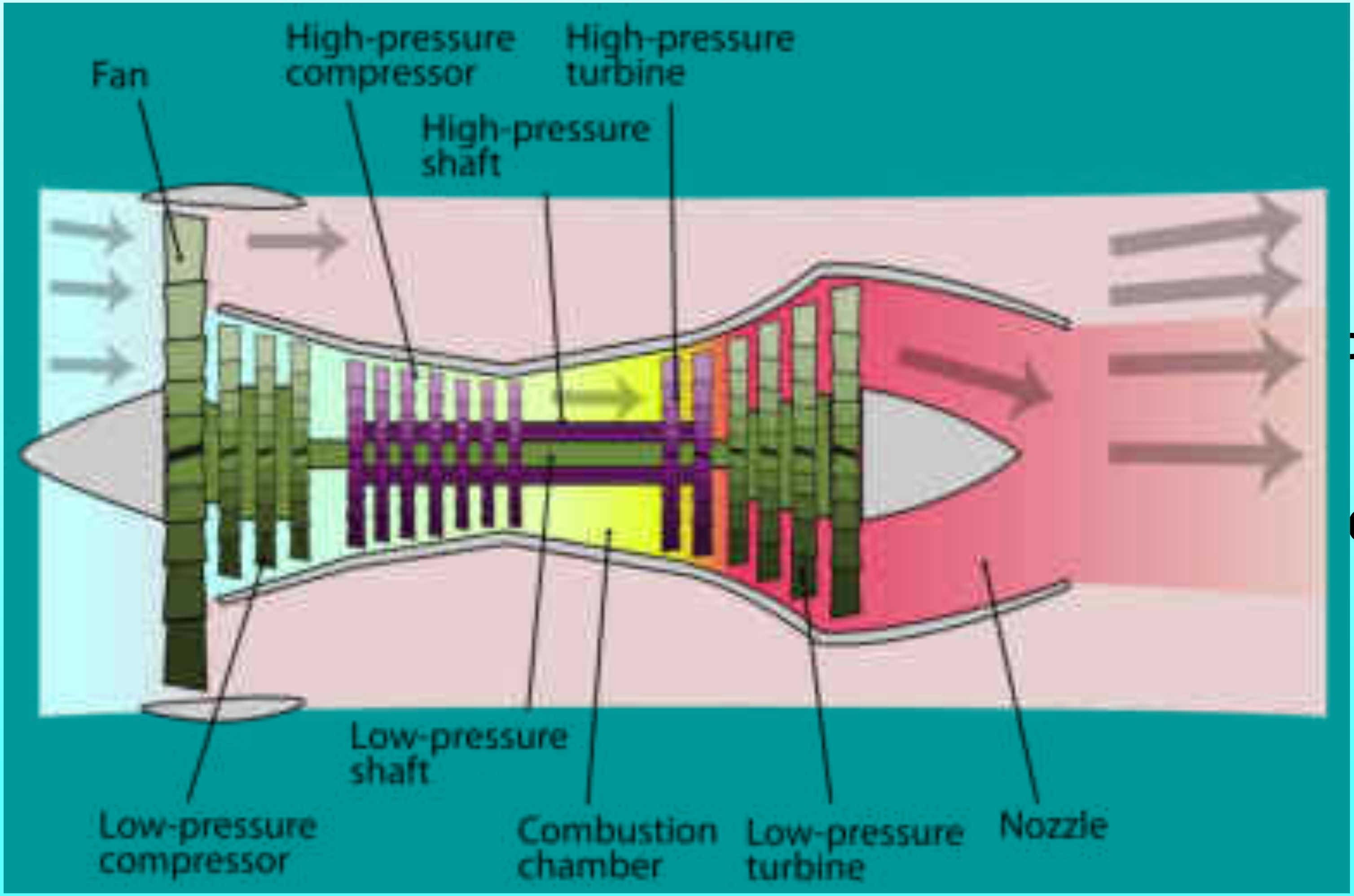






Free stream
Fan
Fan nozzle
Combustor
Turbine
Core nozzle
Compressor
0
2
13
19
3
4
5
9

Turbofan

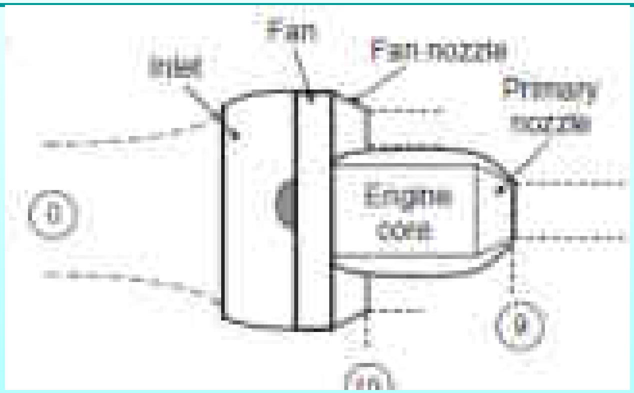


Fan
High-pressure compressor
High-pressure turbine
High-pressure shaft
Low-pressure shaft
Low-pressure compressor
Combustion chamber
Low-pressure turbine
Nozzle

■

■ ... or

Turbo Fan Thrust

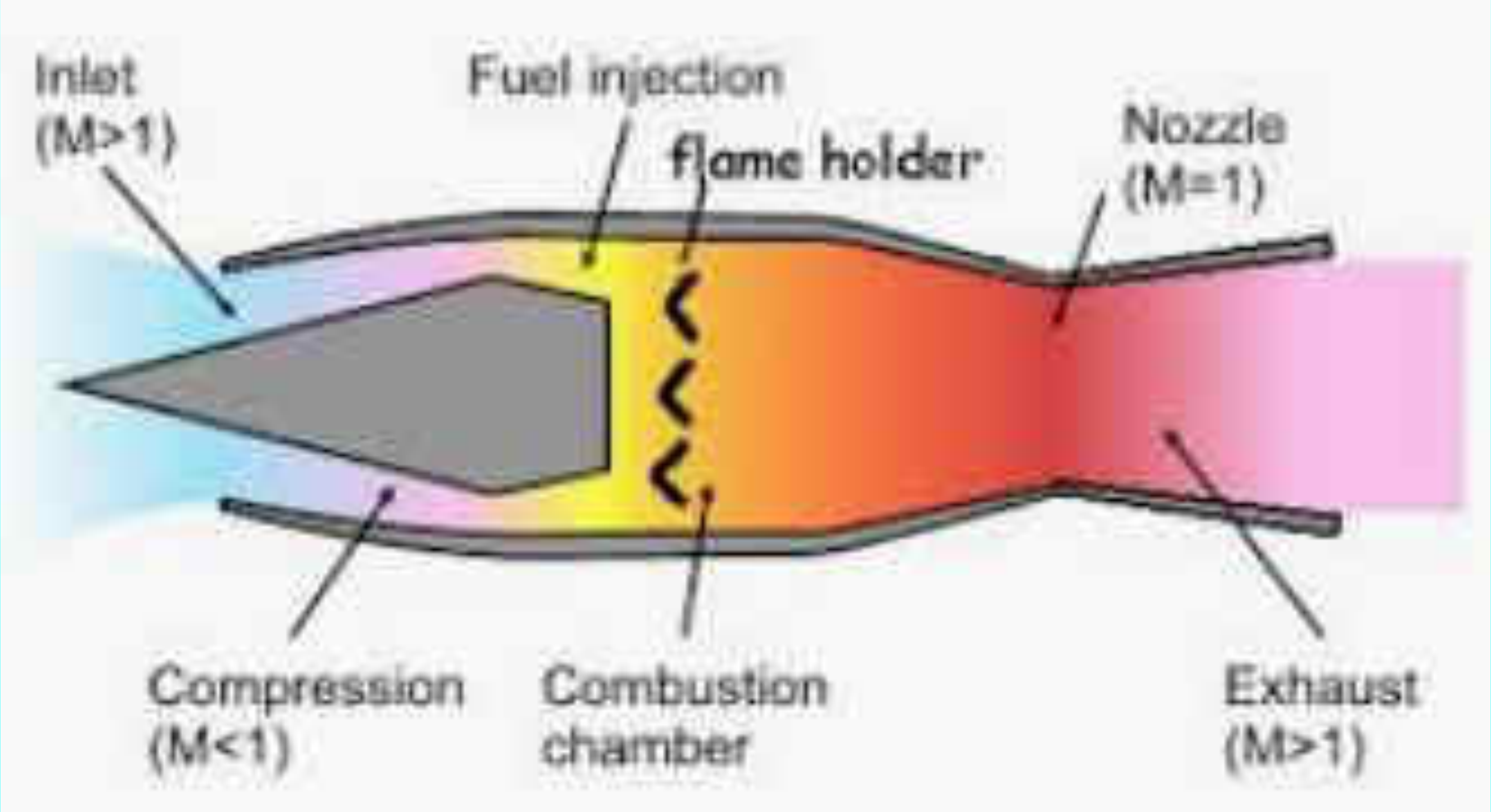


- $\boldsymbol{Thrust = fan\ thrust + core\ thrust}$

- $fan\ thrust = \dot{m}_{fan}(V_{e,fan} - V_a) + (P_{e,fan} - P_a)A_{e,fan}$
- $core\ thrust = (\dot{m}_{e,core}V_{e,core} - \dot{m}_{a,core}V_a) + (P_{e,core} - P_a)A_{e,core}$

RAMJETS

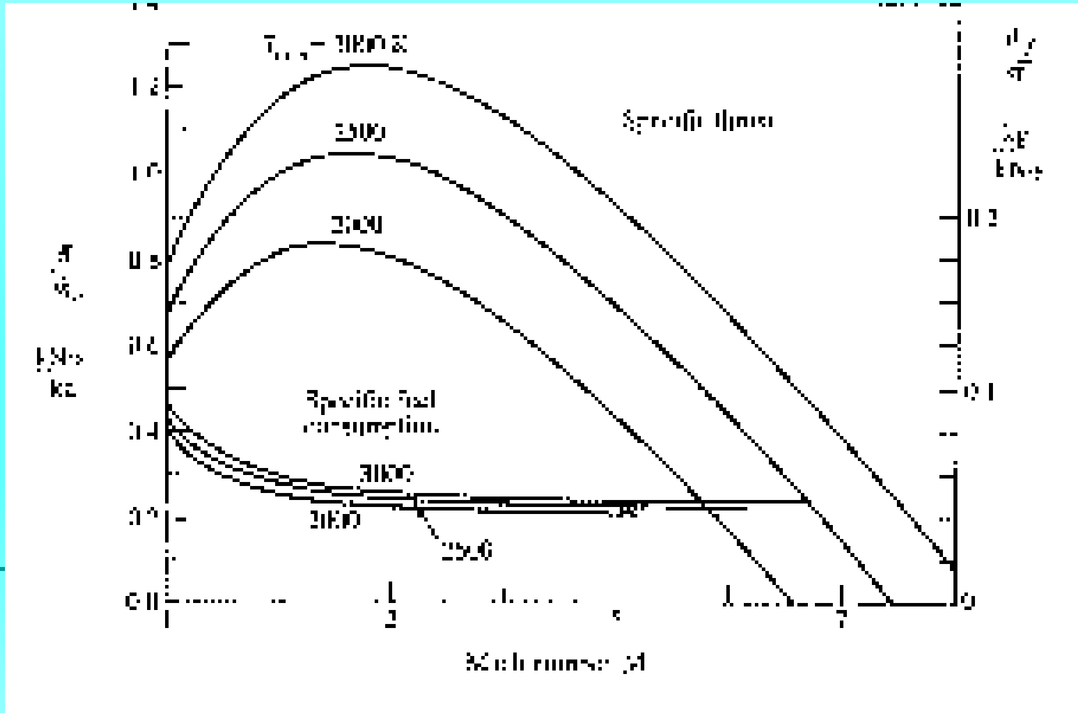
- Salient points:
  - No take-off thrust: needs an auxiliary system to propel to a supersonic Mach number
  - No major rotating parts (Compressor & Turbine)
    - The “Ram effect” of high-velocity air produces the compression
    - Mechanically the “simplest” air-breathing propulsion system
  - Can handle higher combustion temperatures
    - As there are no turbines, there is no simultaneous thermal & mechanical loading
  - Ideal for the Mach number range of about 1.5 to 3
  - Inlet/Intake: Supersonic flow makes the inlet design more complex
  - Stability issues in combustion chamber


Inlet
(M>1)
Fuel injection
flame holder
Nozzle
(M=1)
Compression
(M<1)
Combustion
chamber
Exhaust
(M>1)

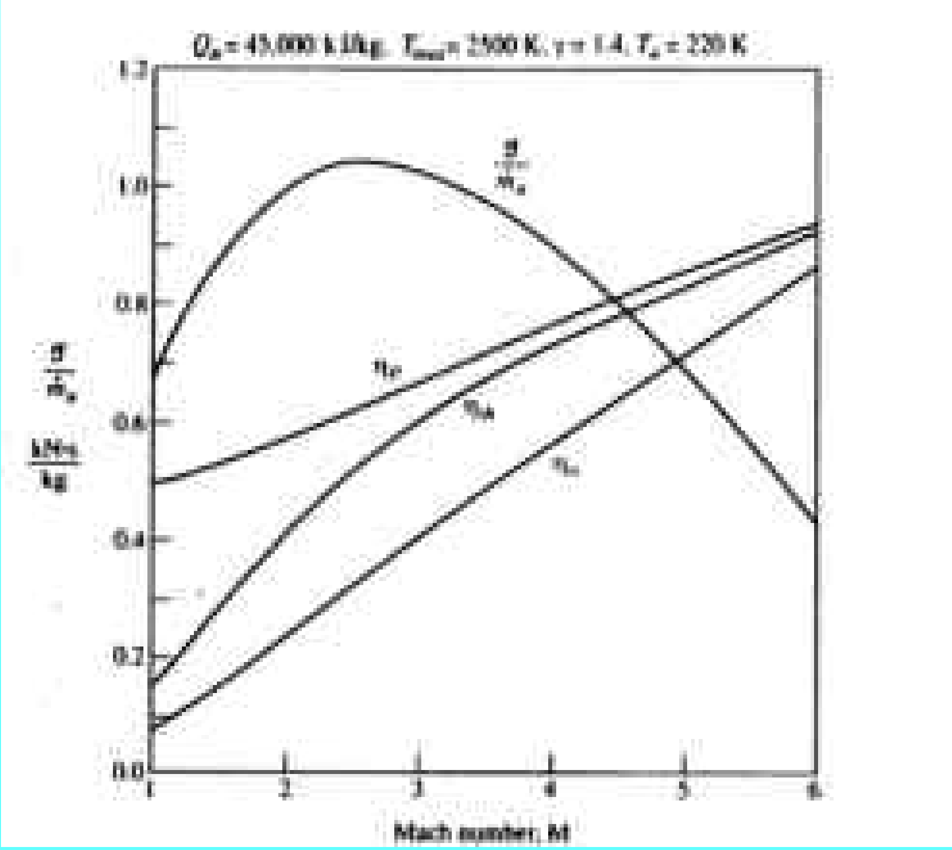

RAMJET Engine – Applications..

RAMJET Thrust

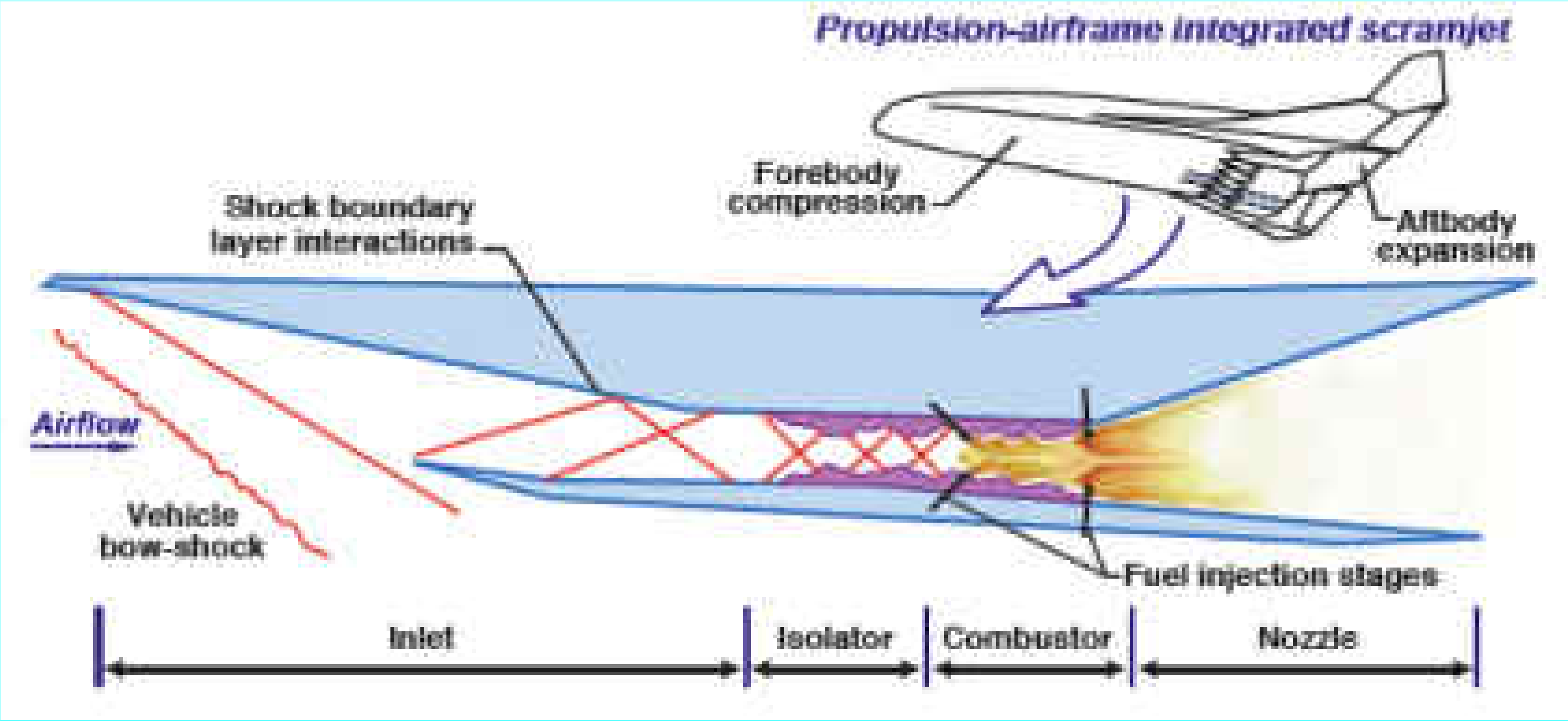
- Thrust equation is the same as that for a turbojet
- Specific thrust peaks around Mach 2.6, then falls
    - Due to the dependence on exit velocity



Thrust & Efficiencies of “Ideal” Ramjet



SCRAMJET Concept



AMRITA VISHWA VIDYAPEETHAM

Aerospace Propulsion

Dr. A.R. Srikrishnan
Department of Aerospace engineering
2023

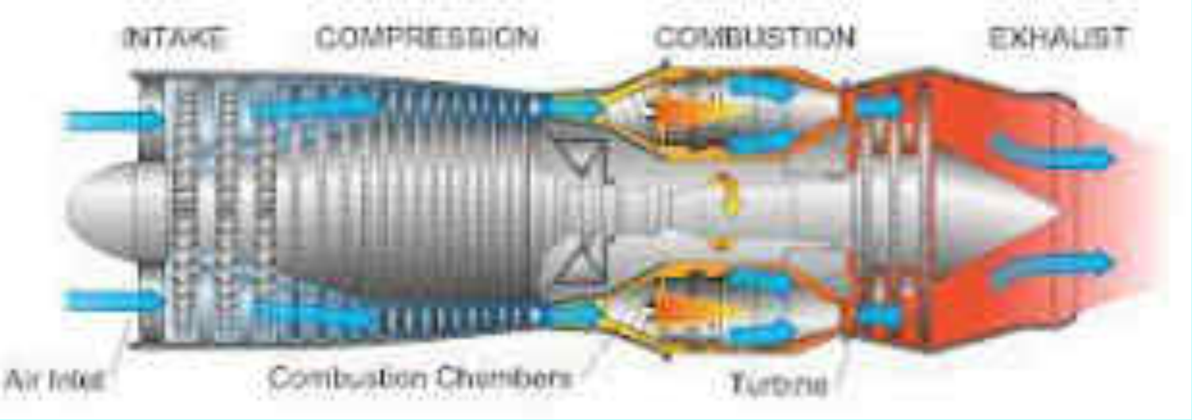
03
Aircraft Engines: Cycle Analysis- Part A: Ideal Brayton Cycle

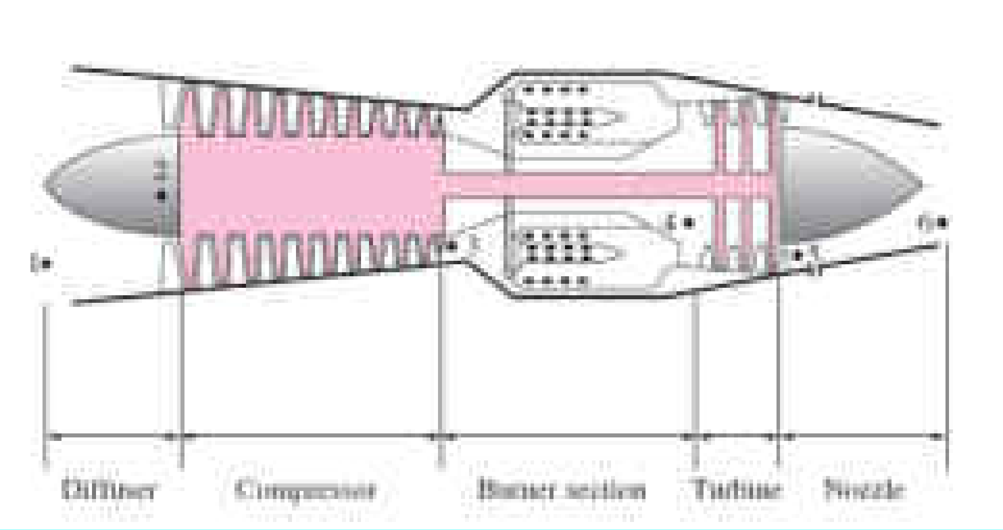
"Thermodynamic Cycle Analysis"

- What is Thermodynamics & WHY ?
- What is Cycle ?
- What is Cycle Analysis ?

- We will start with the last question first…

The Engine, The Components..





WHY Thermodynamics ?

- The Science of Energy !
  - And we are talking about energy
- Heat, Work, Energy
- …..

Cycle Analysis

- Study of the variation in properties of the working fluid as it moves through the engine
  - Does not consider the mechanical forces that drives the flow of the fluid
  - The components are not analysed, the effects that they produced are taken into account - “Rubber Analyis”..
  - *What does this mean for an aircraft engine analysis ?*
- Each component is characterized by the effect of it on the fluid

- **OBJECTIVE:** Determine which characteristics to choose for components of an engine to best satisfy a particular performance requirement
  - Relate T, η, I_{sp}, TSFC to the pertinent design parameters

The Energy Conversion That We are talking about…..

- The desired output: Propulsive power
- The input: Fuel
- In a sense, thermodynamic analysis is an attempt to link these – What you get and What you give
  - The objective is: Pay as less as possible and get as much as possible !

Energy & First Law of Thermodynamics

- Heat (thermal energy) added to a system
- Work done by a system
- The difference between the two
  - Change in overall energy of the system
- The First Law of Thermodynamics as Applied to a System
- $\Delta E_0 = Q - W$

Thermodynamic Cycle

- A *process*: *The system changes from one state to another*
- *State: A set of values of measurable properties that uniquely define the system*
- *Property: Measurables like temperature, volume, pressure, density*
- *Cyclic Process: If at the end of a process the system returns back to the properties at the beginning of the process, it has executed a Cycle*

State Function Vs Path Function

- A path function depends on the process that was followed to accomplish the change of states
  - The same end-states can lead to different processes, resulting in different values for the path functoin
- Thermodynamic Examples:
  - Temperature : A *state* function
  - Heat transfer: A *path* function

Cycle

- A series of thermodynamic *processes* that involve heat transfer and work transfer between the system and the surroundings and take the system back to the initial *state*
- *Can result in net work transfer*
  - *Positive, if System does work on the surroundings*
- Cycle analysis focusses on the work done, efficiency and the parameters that influence these

Cycle: Air Standard Assumptions

- Air is the working fluid, circulated in a closed loop
  - *Why is this just an assumption ?*
- The working fluid, air, is an ideal gas
- All cycles, processes are *internally reversible*
  - *What does this mean ?*
- Combustion process replaced by heat-addition from external source
- Exhaust is replaced by heat rejection process which restores working fluid to initial state

Ideal Vs Real Cycle

- Why idealize ?
- The "*Ideal*" assumptions
  - Cycle does not involve any friction
  - All expansion and compression processes are quasi-equilibrium processes
  - Pipes connecting components have no heat loss
  - Neglecting changes in kinetic and potential energy (except in nozzles & diffusers)

Steady Flow Energy Equation

$$\cancel{\frac{dE_{CV}}{dt}} = \dot{Q} - \dot{W}_{CV} + \sum_{in} \dot{m}(h + \frac{V^2}{2} + gz) - \sum_{out} \dot{m}(h + \frac{V^2}{2} + gz)$$

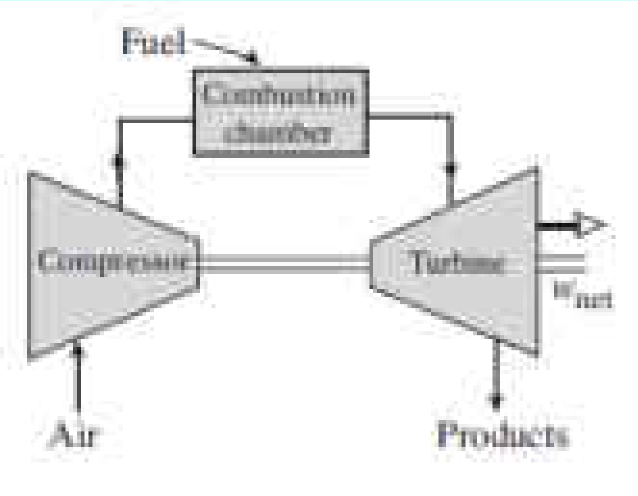
$$\sum_{outlets} \dot{m} = \sum_{inlets} \dot{m}$$

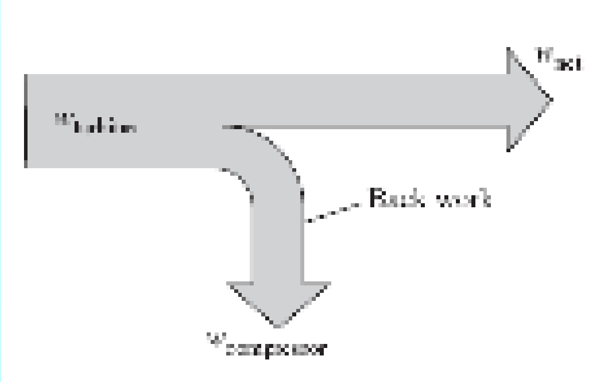
Conservation of mass

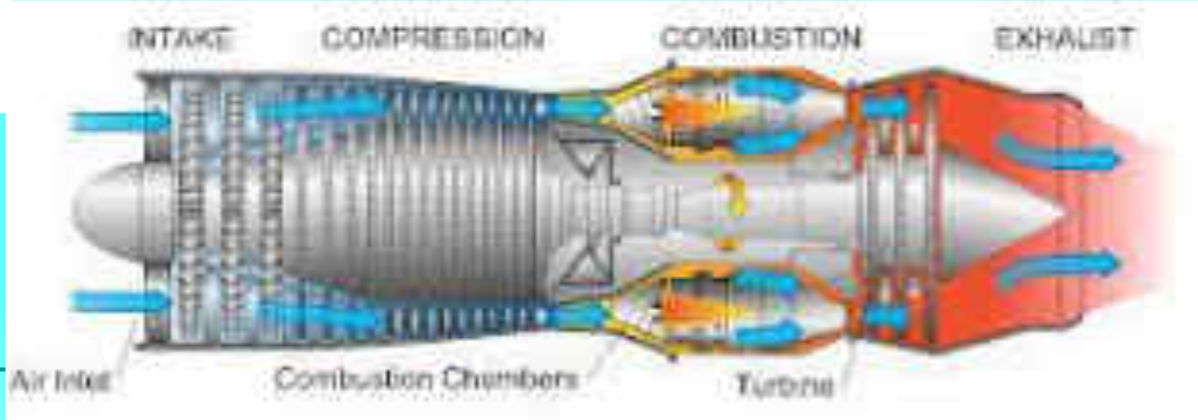
$$\sum_{outlets} \dot{m}(h + \frac{V^2}{2} + gz) = \dot{Q} - \dot{W}_{CV} + \sum_{inlets} \dot{m}(h + \frac{V^2}{2} + gz)$$

Conservation of energy

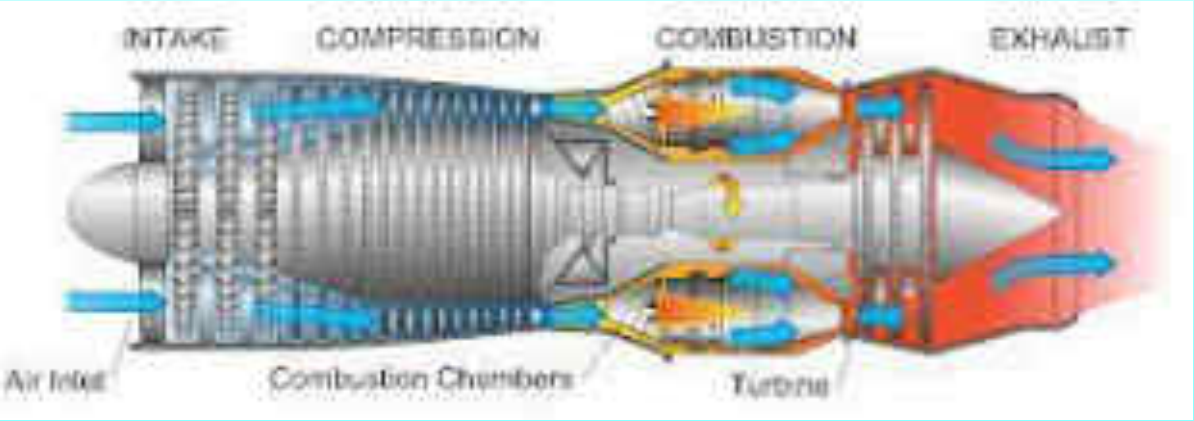
Review of Components: Aircraft Engine

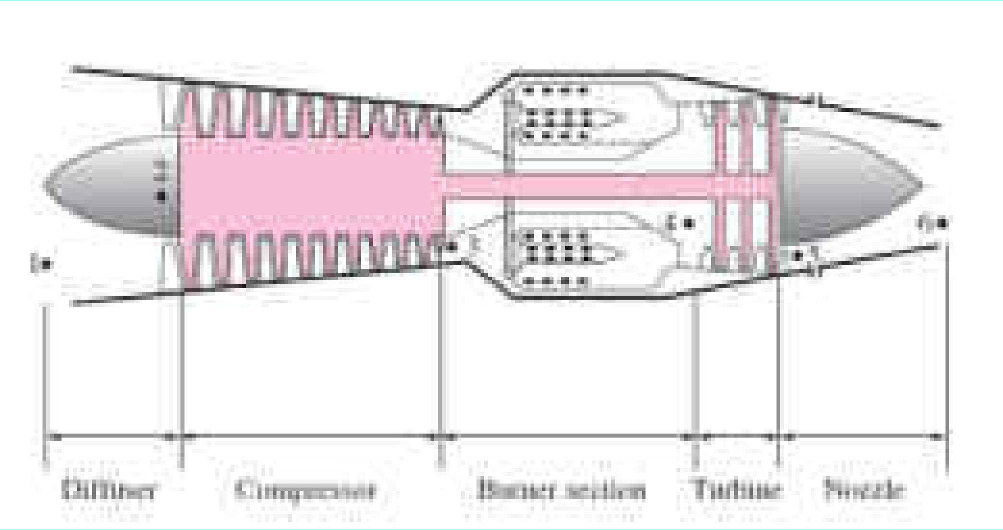




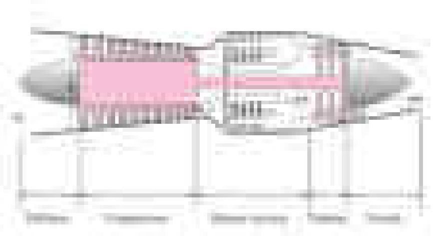


Brayton Cycle

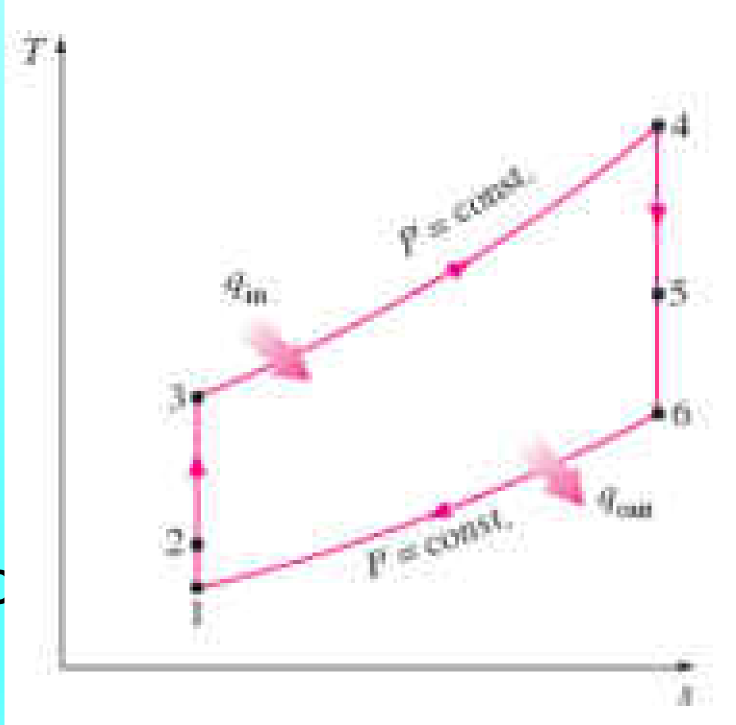




The Processes: Ideal Cycle



- The working fluid, AIR, undergoes:
  - Isentropic Compression
    - from state 1 to state 3
    - *Inside diffuser and compressor*
  - Constant-pressure heat addition
    - from state 3 to state 4
    - *Inside combustion chamber*
  - Isentropic expansion
    - From state 4 to state 6
    - *Inside turbine and nozzle*
  - 4-1 Constant-pressure heat rejec
    - From state 6 to state 1



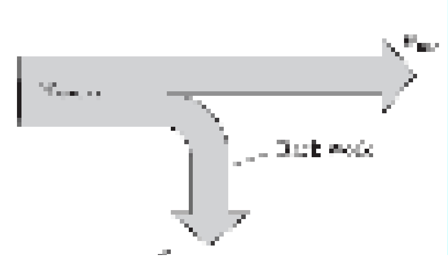
On *Net* work output

~ The Key Principle of Brayton Cycle..

- *Constant pressure lines diverge on a T-S diagram in the direction of increasing entropy*
  - What does this mean ?
- *Work input for a given compression ratio from a low temperature is significantly lower than the work output from the same expansion ratio from a higher temperature*

Tds = dh - vdp

Steady Flow Energy Equation



- Considering the entire cycle:

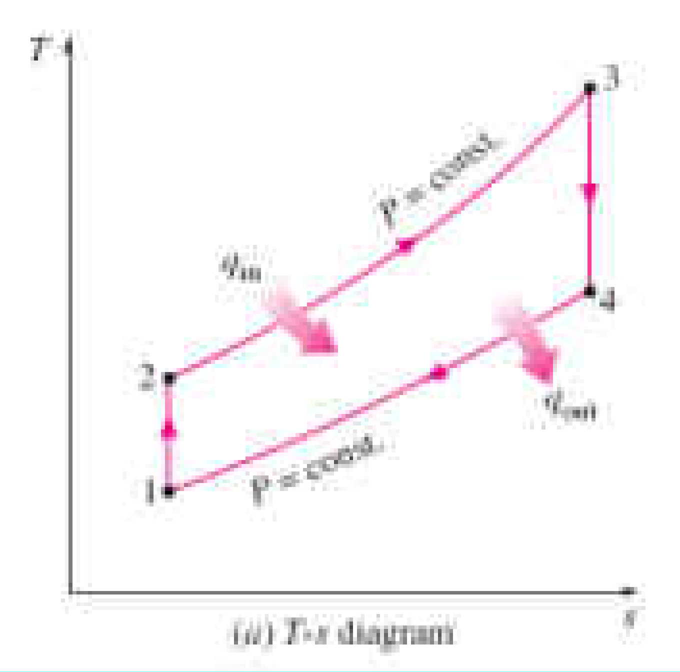
$$(q_{in} - q_{out}) + (w_{in} - w_{out}) =$$

- *Identify components in which*
  of the LHS take place

$$q_{in} = h_3 - h_2 = c_p(T_3 - T_2)$$

$$q_{out} = h_4 - h_1 = c_p(T_4 - T_1)$$

W
w



(a) T-s diagram

- *Hence:*

$$\eta_{th,Brayton} = \frac{w_{net}}{q_{in}} = 1 - \frac{q_{out}}{q_{in}} = 1 - \frac{c_p(T_4 - T_1)}{c_p(T_3 - T_2)} = 1 - \frac{T_1(T_4/T_1 - 1)}{T_2(T_3/T_2 - 1)}$$

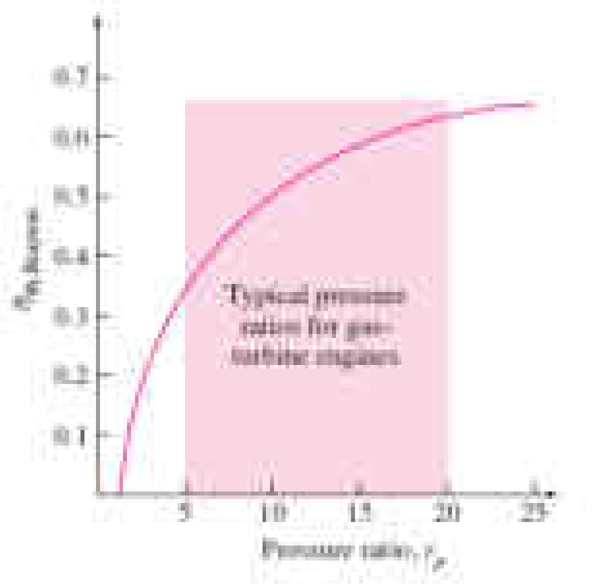
$$\frac{T_2}{T_1} = \left(\frac{P_2}{P_1}\right)^{(k-1)/k} = \left(\frac{P_3}{P_4}\right)^{(k-1)/k} = \frac{T_3}{T_4}$$

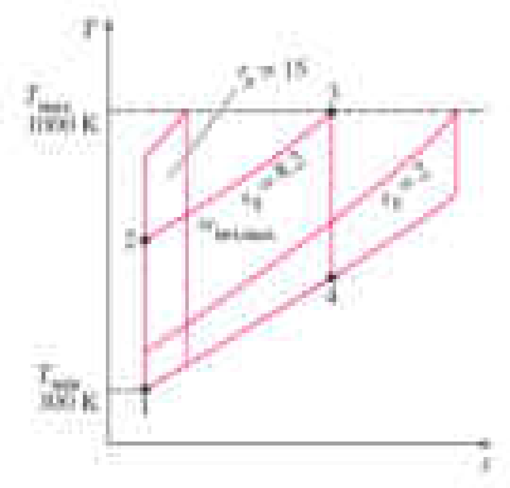
Note: k is γ (=Cp/Cv)

$$Ideal\ Brayton\ Cycle\ Efficiency = 1 - \frac{T_1}{T_2} = 1 - \left[\frac{P_1}{P_2}\right]^{(k-1)/k}$$

Observations from Ideal Cycle Analysis

- Variation of Cycle Efficiency with Pressure Ratio





AMRITA VISHWA VIDYAPEETHAM

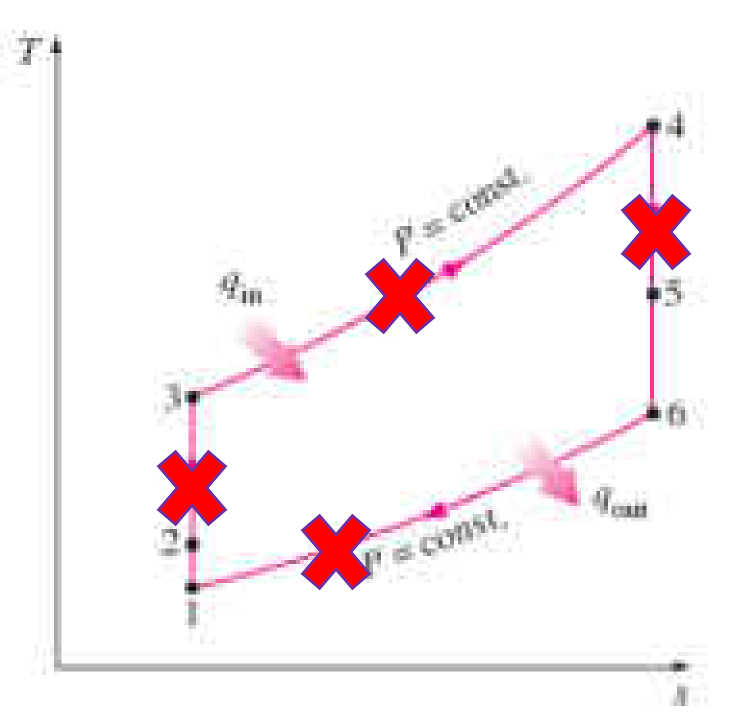
Aerospace Propulsion

Dr. A.R. Srikrishnan
Department of Aerospace engineering

Aircraft Engines: **REAL** Cycle Analysis

Analysis of REAL Cycles

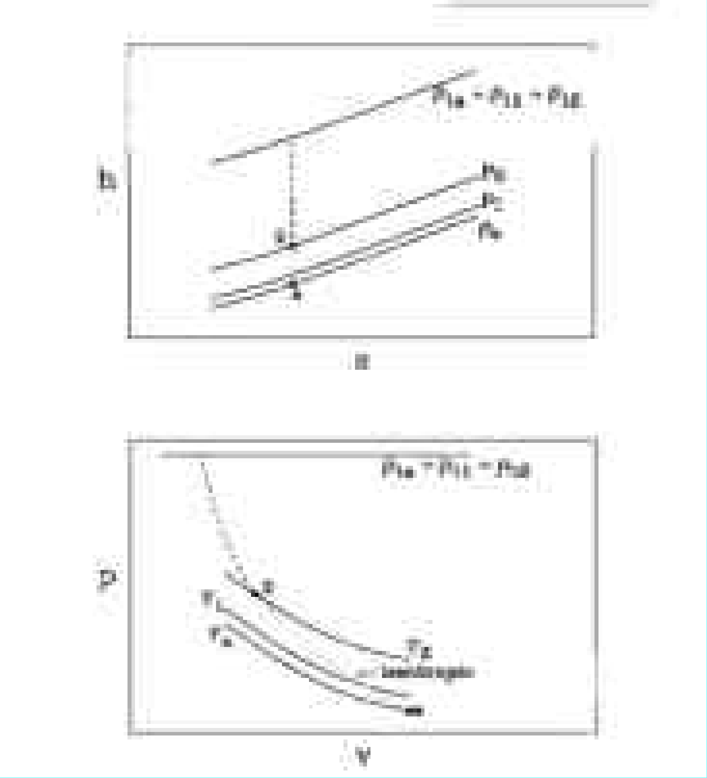
- What makes REAL different from IDEAL ?
  - Compression is not isentropic
  - Combustion is not isobaric
  - Expansion is not isentropic
  - Heat rejection is not isobaric
- Approach:
  - *We will analyze each component, each process, identify the Sources of losses and evaluate their impact on the performance*



Compression: Ideal Diffuser

- Why diffuser ?
  - Slow down the working fluid
    - Whats the velocity that it enters with ?
  - Increase the static pressure of the fluid
- Ideal assumption states:
  - Irreversibility – no friction no viscosity
  - Adiabatic: No heat transfer across the walls
  - Steady, calorically perfect (constant cp/cv)

h~S and PV Diagrams: Ideal diffuser



- SFEE for Ideal diffuser:
- Q1-w = E

Real Diffuser

- Variations from Ideal behavious:
  - Flow is viscous
  - The flow is turbulent
  - The wall may not be perfectly adiabatic
    - Not a major factor – Can be assumed to be adiabatic for practical cases
  - Air is not calorically perfect

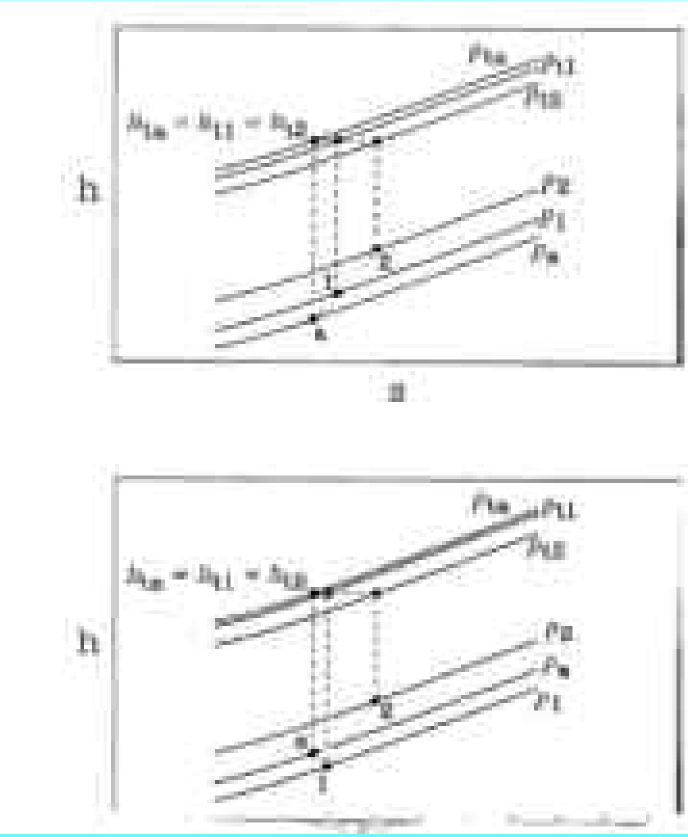
So What ?

- The impact on performance
  - Stagnation pressure loss across the diffuser
    - How about static pressure ?
  - Possible stagnation temperature loss –
    - Usually negligible
  - Shock losses – ONLY for supersonic inlets

Real Diffusion Process…

- How we quantify the deviation from ideal to real ?
  - Why quantify ?
- Total Pressure Recovery Rati

$$\pi_d = \frac{p_{t2}}{p_{ta}}$$



Diffuser efficiency

- η_D=ratio of the isentropic change in enthalpy to actual change in enthalpy
- For adiabatic process:
  - $h_{01}=h_{02}$

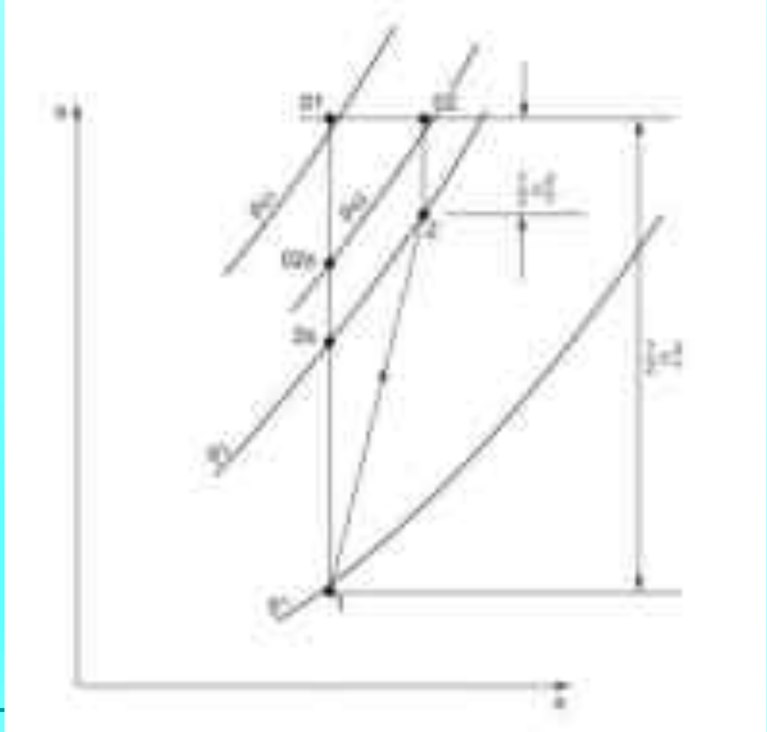
$$h_2 - h_1 = \frac{1}{2}(c_1^2 - c_2^2)$$

C1 and C2 denote air velocities at 1 & 2

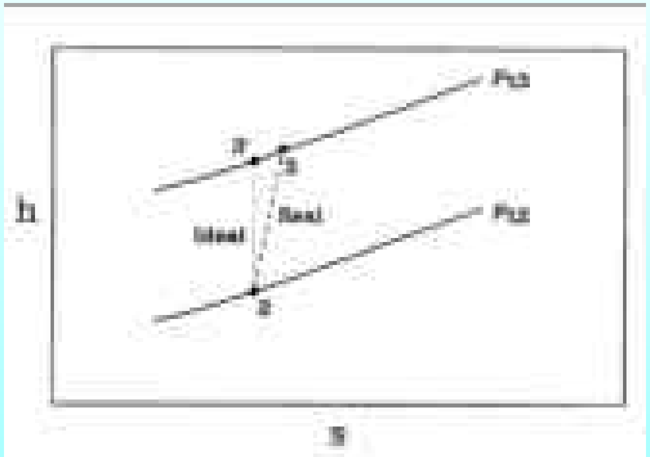
  - Isentropic

$$h_{2s} - h_1 = \frac{1}{2}(c_1^2 - c_{2s}^2)$$

$$\eta_D = (h_{2s} - h_1) / (h_2 - h_1) = (c_1^2 - c_{2s}^2) / (c_1^2 - c_2^2)$$



Compression Process



- Stagnation pressure ratio across compressor:

$$\pi_c = \frac{P_{t3}}{P_{t2}}$$

Ideal Compressor

- Assumptions
  - Isentropic Process
    - No heat loss – Compression is adiabatic
    - No viscosity/friction – Compression is reversible
  - Ideal gas assumptions
- The raise in stagnation pressure and stagnation temperature corresponds to the isentropic process values

Ideal Compressor

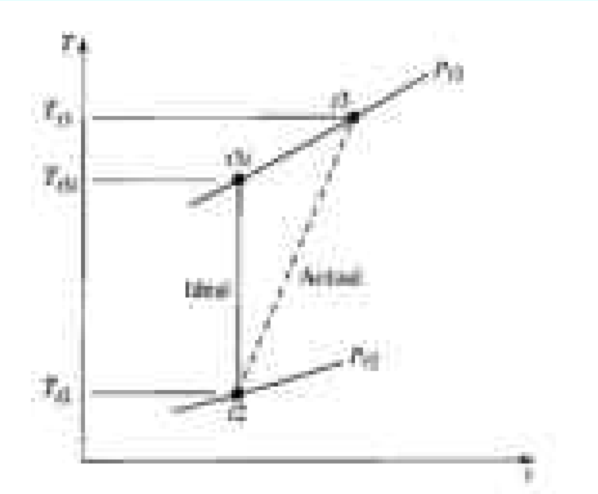
Total pressure ratio across the compressor $\pi_c = \frac{p_{t3}}{p_{t2}}$

- As compression is assumed to be isentropic,

Total Temperature Ratio across compressor,

$$\tau_c = \frac{T_{t3}}{T_{t2}} = \pi_c^{\frac{\gamma-1}{\gamma}}$$

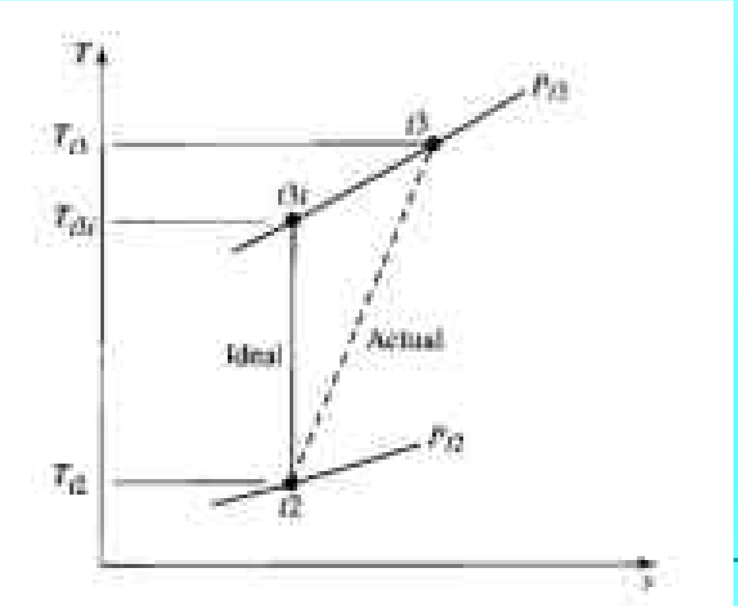
Real Compression Process



- Real compression is irreversible due
Viscosity and heat
Transfer
- The factors that lead to losses in rea
process:
  - Viscosity, boundary layer formation
  - Turbulent dissipation
  - Boundary layer separation on compressor blades, incidence losses
    - *etc.*
- Hence more power is required to drive the compressor as compared to the ideal process

Real Compression

- Thus for a ***given raise in stagnation pressure***, more power need to be given to the compressor than for the ideal (isentropic) compression to accomplish the ***same*** stagnation pressure raise



Compressor Efficiency

- Isentropic Efficiency

$$\eta_i = \left.\frac{Ideal\ Power\ Required}{Actual\ Power\ Required}\right\}\ to\ achieve\ a\ specified\ stagnation\ pressure\ ratio, \pi_c$$

$$= \frac{h_{t3,s} - h_{t2}}{h_{t3} - h_{t2}}$$

If Specific heat is assumed to be constant,

$$\eta_i = \frac{T_{t3,s} - T_{t2}}{T_{t3} - T_{t2}}$$

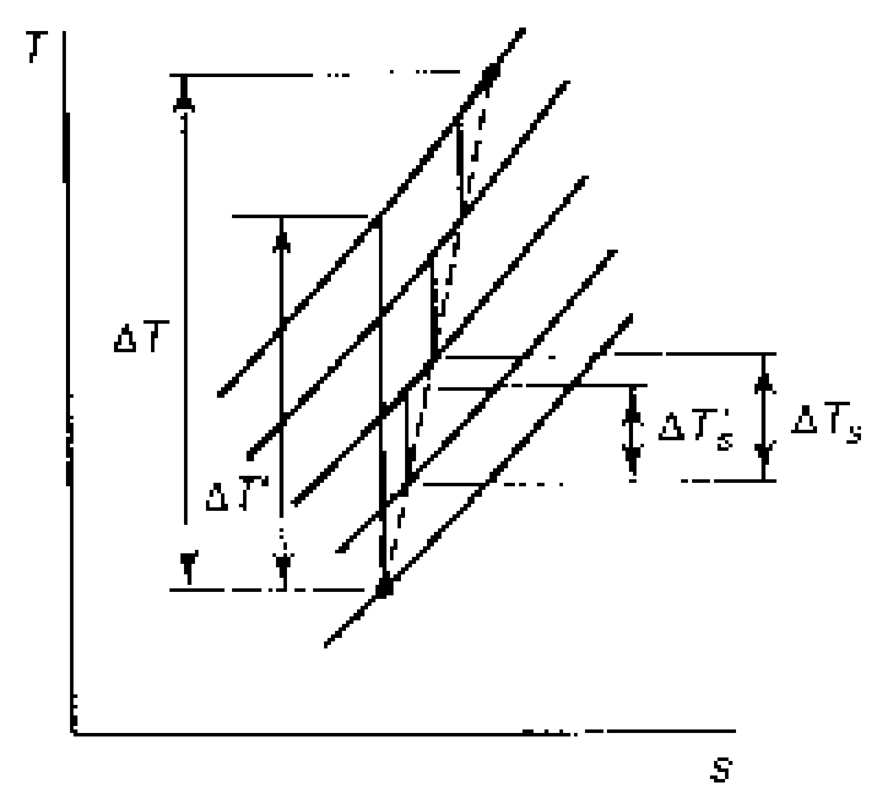
Input, Output & Efficiency

- How are π and τ related through η ?
- τ *is a measure of work input to the compressor (why ?)*
- π *is a measure of the out put of the compressor*
- *They are connected through* η*, the isentropic efficiency*

Polytropic Efficiency

- Defined as the *isentropic efficiency of an elemental stage in the process such that it is constant throughout the whole process*

- *Why this definition ?*
  - *Isentropic definition is dependent of pressure Ratio*
    - *For a finite pressure ratio relates the Input and output*



- For multistage compressors, the number of stages is a design parameter
  - Require an efficiency definition that is independent of the overall pressure ratio and remains constant for each stage

- *Polytropic efficiency is defined as* $\eta_{prop} \equiv \frac{dh_{0,s}}{dh_0}$
- *TDS equation as applied to stagnation states yields:*

$$T_0 ds = dh_0 - v_0 dP_0$$

$$= dh_0 - \frac{dP_0}{\rho_0}$$

For isentropic process $(ds = 0)$

$$dh_0 = \frac{dP_0}{\rho_0}$$

As $P_0 = \rho_0 R T_0$;

$$\eta_{poly} = \frac{\dfrac{dP_0}{(P_0/RT_0)}}{dh_0} = \frac{\dfrac{dP_0}{P_0}}{(dh_0/RT_0)}$$

$$= \frac{\dfrac{dP_0}{P_0}}{C_p\, dT_0/RT_0} = \frac{\dfrac{dP_0}{P_0}}{\dfrac{\gamma}{\gamma-1}\cdot\dfrac{dT_0}{T_0}}$$

As $R = \dfrac{\gamma - 1}{\gamma} C_p$

Hence:

$$\frac{dP_0}{P_0} = \frac{\gamma}{\gamma-1}\,\eta_{poly}\,\frac{dT_0}{T_0}$$

Integrating between the end-states of compression (2 & 3)

$$\Rightarrow \quad \frac{P_{03}}{P_{02}} = \left[\frac{T_{03}}{T_{02}}\right]^{\frac{\gamma \cdot \eta_{poly}}{\gamma-1}}$$

i.e.

$$\pi_c = \left[\tau_c\right]^{\frac{\gamma}{(\gamma-1)}\cdot\eta_{poly}}$$

$$\tau_C = \pi_C^{\frac{\gamma-1}{\gamma\,\eta_{poly}}}$$

Polytropic Efficiency

$$\eta_{poly} = \frac{Ideal\ Power\ Required\ for\ a\ differential\ pressure\ change}{Actual\ Power\ Required\ for\ a\ differential\ pressure\ change}$$

Isentropic Efficiency,

$$\eta_c = \frac{\pi_c^{\frac{\gamma-1}{\gamma}} - 1}{\tau_c - 1} = \frac{\pi_c^{\frac{\gamma-1}{\gamma}} - 1}{\pi_c^{\frac{\gamma-1}{\gamma \eta_{poly}}} - 1}$$

Tutorial – Relating Efficiencies, Pressure ratio, Temp ratio

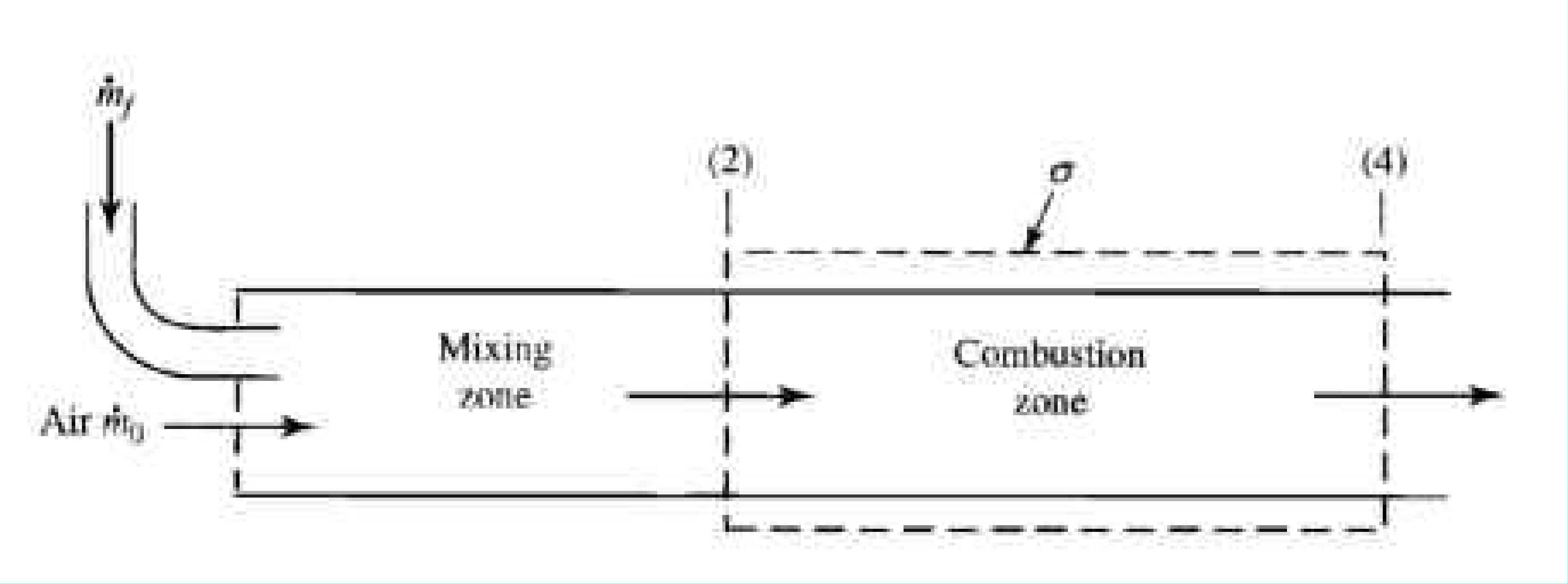
- An aircraft compressor has a specified polytropic efficiency of 91% and stagnation pressure ratio = 30. Under a given test conditions, it operates with a mass flow rate of 62 kg/s while the inlet stagnation pressure and temperature are 101325 Pa and 300 K respectively. Determine the stagnation temperature at the exit of the compressor, its isentropic efficiency and the turbine power required to drive the compressor under these conditions.

Mass flow kg/s	Pi	Eta-Poly	To2 K	Po2 Pa	To3 K	Eta Isentropic	Power, W
62	30	0.91	300	101325	872.8	0.8603683	35653259

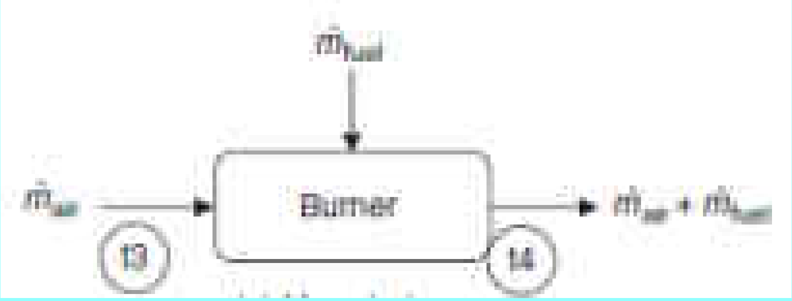
The Combustion Chamber

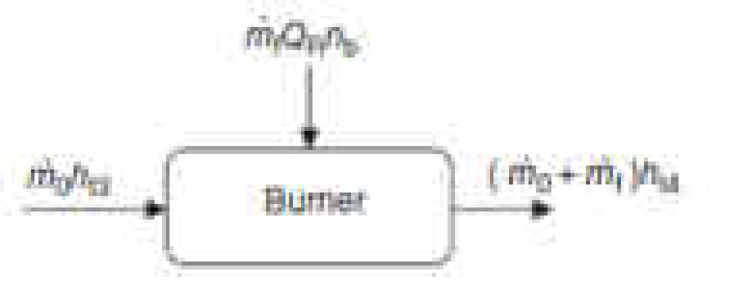
- Combustion Chamber/Combustor/Burner
- Combustion: *The process that leads to the release of thermal energy following a self-sustained chemical reaction involving fuel and oxidizer*

The Combustion System



Mass Balance & Energy Equation -1

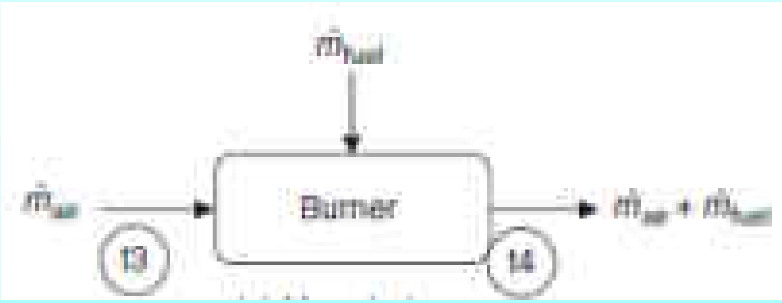


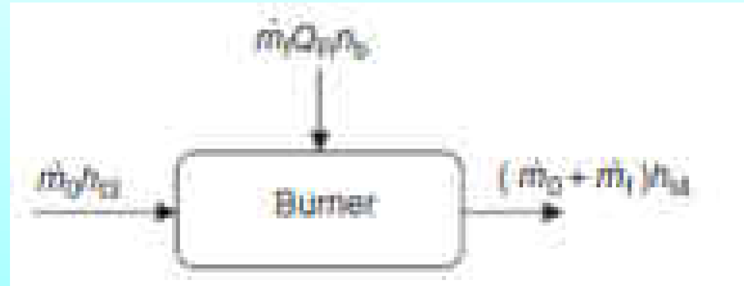


$$\dot{m}_4 = \dot{m}_a + \dot{m}_f = \dot{m}_a(1+f)$$

where f (fuel/air ratio) $= \dfrac{\dot{m}_f}{\dot{m}_a}$

Mass Balance & Energy Equation ~II





$$\dot m_a h_{03} + \dot m_f C.V.\eta_b = (\dot m_a + \dot m_f) h_{04}$$
$$\Rightarrow \dot m_a (1+f) h_{04}$$

Dividing by $\dot m_a$

$$h_{03} + f\, C.V.\eta_b = (1+f) T_{04} C_p$$

i.e. $$\frac{C_p T_{03} + f\, C.V.\eta_b}{1+f} = C_p T_{04}$$

as $f \ll 1$

$$[T_{04} - T_{03}] = \frac{f\, C.V.\eta_b}{C_p}$$

Beware of the Simplification: ***Cp is assumed to be constant across the combustor***

Steady Flow Energy Equation, Burner

- The energy balance gives:

$$\dot{m}_0 h_{t2} + \dot{m}_f h_{PR} = (\dot{m}_0 + \dot{m}_f) h_{t4}$$

- Neglecting fule flow rate in comparison to air flow rate (*why ?*), and assuming ideal gas behavior (Constant Cp):

$$\dot{m}_0 c_p T_{t2} + \dot{m}_f h_{PR} = \dot{m}_0 c_p T_{t4}$$

Ideal Combustor

- Assumptions
  - Complete Combustion
  - Adiabatic walls
  - Constant pressure process
  - Constant specific heat

Numerical Problem

- The combustion chamber of an aircraft engine operates with a fuel-air ratio of 0.018 when the mass flow arte of air is 62 kg/s. The fuel used has a calorific value of 44 MJ/kg. Air from the compressor enters the combustor at stagnation temperature = 760 K and stagnation pressure 2.2 MPa. Determine the stagnation temperature at the inlet to the turbine of the combustor efficiency is 98%.

Solution

To3 K	Po3 Pa	ma kg/s	f	CV J/kg	Etab	To4 K
760	2200000	61	0.018	44000000	0.98	1533.06773

- Re-calculate the value of T_{o4} by incorporating the variation of C_p as follows: C_p at inlet to the combustor = 1090 J/kg-K;
  C_p at exit of the combustor = 1210 J/kg-K
  - *Substitute C_{p3} and C_{p4} on the LHS and RHS respectively, of the relation*

$$\frac{C_p T_{o3} + f\, CV\, \eta_b}{1+f} = C_p T_{o4}$$

AMRITA VISHWA VIDYAPEETHAM

Aerospace Propulsion

Dr. A.R. Srikrishnan
Department of Aerospace engineering

04
Combustion Fundamentals

COMBUSTION

STOICHIOMETRY

- Mixture Fraction: Molar Basis Vs Mass Basis
- Air Fuel Ratio, Fuel Air Ratio
- Methane: $CH_4 + 2O_2 + 2*3.76\ N_2 \rightarrow CO_2 + 2H_2O + 2*3.76\ N_2$
    - Mass of Air: 2*32 + 7.52*28 = 274.56 g
    - Mass of Fuel: 12+4 = 16 g
    - Air Fuel Ratio = 17.16; Fuel – Air Ratio = 0.058
- *Stoichiometric* Ratio
    - *Fuel-Rich*
    - *Fuel-Lean*

Excess Air

- Propane: with “Theoretical” Air

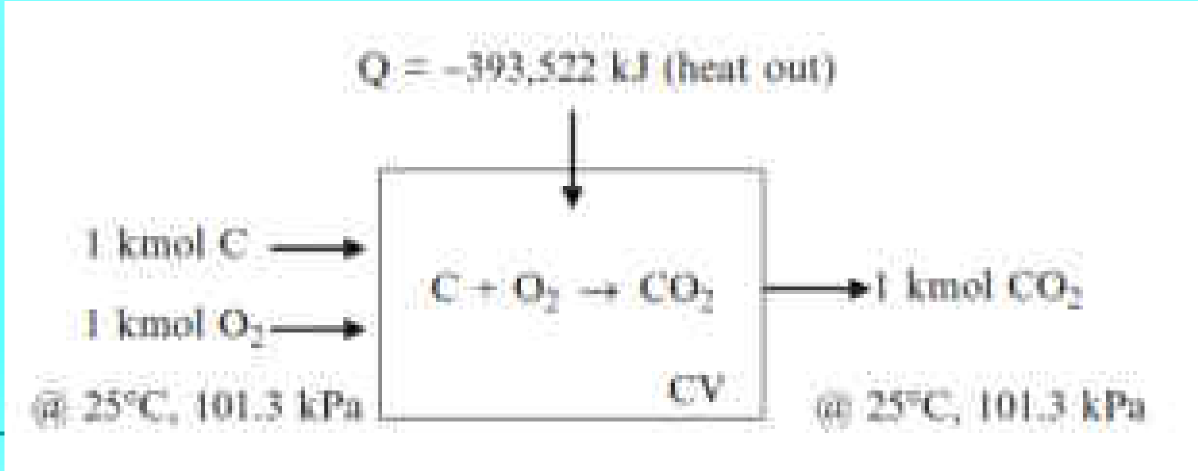
$$C_3H_8 + 5(O_2 + 3.76N_2) \Rightarrow 3CO_2 + 4H_2O + 18.8N_2$$

- The effect of using excess air
  - *%excess*: AF for 50% Excess Air ?
- Aircraft Applications

Write the balanced equation for this

Enthalpy

- Absolute Enthalpy, considered as sum of:
  - Sensible Enthalpy
  - Enthalpy of Formation
    - Rearrangement of Chemical Bonds; Energy Release
  - Definitions



Enthalpy of formation of common combustion species

Species	$\Delta \hat{h}^o$ (MJ/kmol)	Species	$\Delta \hat{h}^o$ (MJ/kmol)
H_2O (g)	−241.83	H	+217.99
CO_2	−393.52	N	+472.79
CO	−110.53	NO	+90.29
CH_4	−74.87	NO_2	+33.10
C_3H_8	−104.71	O	+249.19
C_7H_{16} (g) (n-heptane)	−224.23	OH	+39.46
C_8H_{18} (g) (isooctane)	−259.25	C (g)	+715.00
CH_3OH (g) (methanol)	−201.54	C_2H_2 (acetylene)	+226.73
CH_3OH (l) (methanol)	−238.43	C_2H_4 (ethylene)	+52.28
C_2H_6O (g) (ethanol)	−235.12	C_2H_6 (ethane)	−84.68
C_2H_6O (l) (ethanol)	−277.02	C_4H_{10} (n-butane)	−126.15

Methane (CH_4) $C_1H_4(g)$

Enthalpy Reference Temperature = T_r = 298.15 K Standard State Pressure = $p°$ = 0.1 MPa

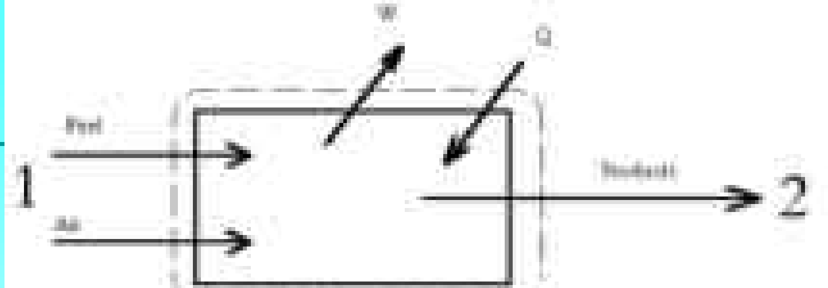
	J K^{-1} mol^{-1}			kJ mol^{-1}			
T/K	$C_p°$	$S°$	$-[G°-H°(T_r)]/T$	$H-H°(T_r)$	$\Delta_f H°$	$\Delta_f G°$	log K_f
0	0	0.	INFINITE	-10.024	-66.911	-66.911	INFINITE
100	33.258	149.500	216.485	-6.698	-69.644	-64.353	33.615
200	33.473	172.577	189.418	-3.368	-72.027	-58.161	15.190
250	34.216	180.113	186.829	-1.679	-73.426	-54.536	11.395
298.15	35.639	186.251	186.251	0.	-74.873	-50.768	8.894
300	35.708	186.472	186.252	0.066	-74.929	-50.618	8.813
350	37.874	192.131	186.694	1.903	-76.461	-46.445	6.932
400	40.500	197.356	187.704	3.861	-77.969	-42.054	5.492
450	43.374	202.291	189.053	5.957	-79.422	-37.476	4.350
500	46.342	207.014	190.614	8.200	-80.802	-32.741	3.420
600	52.227	215.987	194.103	13.130	-83.308	-22.887	1.993
700	57.794	224.461	197.840	18.635	-85.452	-12.643	0.943
800	62.932	232.518	201.675	24.675	-87.238	-2.115	0.138
900	67.601	240.205	205.532	31.205	-88.692	8.616	-0.500
1000	71.795	247.549	209.370	38.179	-89.849	19.492	-1.018
1100	75.529	254.570	213.162	45.549	-90.750	30.472	-1.447
1200	78.833	261.287	216.895	53.270	-91.437	41.524	-1.807
1300	81.744	267.714	220.558	61.302	-91.945	52.626	-2.115
1400	84.305	273.868	224.148	69.608	-92.308	63.761	-2.379
1500	86.556	279.763	227.660	78.153	-92.553	74.918	-2.609

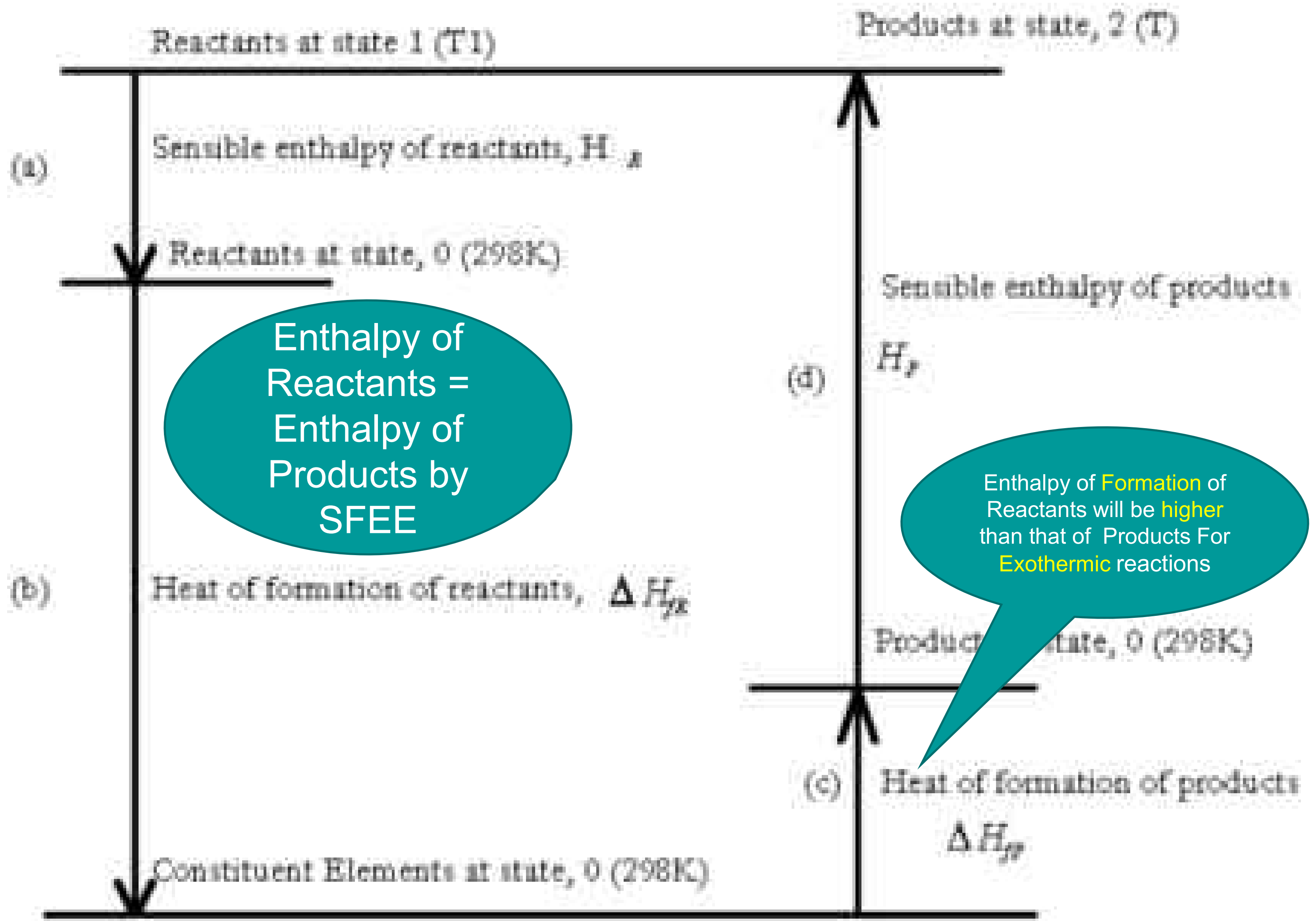
Enthalpy of Reaction

- *Defined as the energy that must be supplied in the form of heat to keep a system at constant temperature and pressure during a reaction*
  - *difference between the total enthalpy of the products of a reaction and the total enthalpy of the reactants*
  - *Note: Enthalpy of formation is a specific instance*

Combustion Can be Considered as:

- Constituting the following steps
  - The reactants are taken to a datum temperature, --- Usually 298K
  - The reactants are broken down to their constituent elements in their natural forms
  - The products are formed from their constituent elements in their natural forms
  - products are brought to their final temperature, T.





Reactants at state 1 (T1)
Products at state, 2 (T)
(a)
Sensible enthalpy of reactants, H R
Reactants at state, 0 (298K)
Enthalpy of Reactants = Enthalpy of Products by SFEE
Sensible enthalpy of products
(d)
H_P
Enthalpy of Formation of Reactants will be higher than that of Products For Exothermic reactions
(b)
Heat of formation of reactants, ΔH_{fR}
(c)
Heat of formation of products
ΔH_{fP}
Constituent Elements at state, 0 (298K)

Numerical Problem

- In a combustion chamber, Liquid propane reacts with air as per the following chemical equation:

$$C_3H_8(\ell) + 5\,(O_2 + 3.76N_2) \rightarrow 3CO_2 + 4H_2O + 18.8\,N_2$$

Determine the heat transfer during the combustion. Propane at 298 K, enters the chamber with a mass flow arte of 0.05 kg/min. Air enters at 280 K. The products leave the chamber at 1500 K. The relevant enthalpy values are listed in the table below:

Substance	$\bar{h}_f^\circ$ kJ/kmol	$\bar{h}_{280\,K}$ kJ/kmol	$\bar{h}_{298\,K}$ kJ/kmol	$\bar{h}_{1500\,K}$ kJ/kmol
$C_3H_8(\ell)$	−118,910	—	—	—
O_2	0	8150	8682	49,292
N_2	0	8141	8669	47,073
$H_2O(g)$	−241,820	—	9904	57,999
CO_2	−393,520	—	9364	71,078
CO	−110,530	—	8669	47,517

- Heat Transfer = Net Enthalpy of Reactants – Net Enthalpy of Products *(based on SFEE)*
- *Net enthalpy of Reactants:*

$$\sum_{Reactants} n_r\left(h(T) - h_o + \Delta h_f^o\right)$$

- *Net enthalpy of Products:*

$$\sum_{Products} n_p\left(h(T) - h_o + \Delta h_f^o\right)$$

- Substituting enthalpies from the table at the respective temperatures for the products (1500 K) & the reactants :
- Heat Transfer = $\sum_{Reactants} n_r \left(h(T) - h_0 + \Delta h_f^0 \right)$ - $\sum_{Products} n_p \left(h(T) - h_0 + \Delta h \right.$

$$
\begin{aligned}
&= (1 \text{ kmol } C_3H_8)[(-118{,}910 + h_{298} - h_{298}) \text{ kJ/kmol } C_3H_8] \\
&\quad + (7.5 \text{ kmol } O_2)[(0 + 8150 - 8682) \text{ kJ/kmol } O_2] \\
&\quad + (28.2 \text{ kmol } N_2)[(0 + 8141 - 8669) \text{ kJ/kmol } N_2] \\
&\quad - (2.7 \text{ kmol } CO_2)[(-393{,}520 + 71{,}078 - 9364) \text{ kJ/kmol } CO_2] \\
&\quad - (0.3 \text{ kmol } CO)[(-110{,}530 + 47{,}517 - 8669) \text{ kJ/kmol } CO] \\
&\quad - (4 \text{ kmol } H_2O)[(-241{,}820 + 57{,}999 - 9904) \text{ kJ/kmol } H_2O] \\
&\quad - (2.65 \text{ kmol } O_2)[(0 + 49{,}292 - 8682) \text{ kJ/kmol } O_2] \\
&\quad - (28.2 \text{ kmol } N_2)[(0 + 47{,}073 - 8669) \text{ kJ/kmol } N_2] \\
&= 363{,}880 \text{ kJ/kmol of } C_3H_8
\end{aligned}
$$

1 kmole of propane → 44 kg.
Hence calculate kJ/kg & multiply by mass flow rate (0.05 kg/min).

Adiabatic Flame Temperature

- The maximum temperature that can be achieved in a combustion reaction, in the absence of work transfer and heat transfer
- *The temperature that the flame would attain if the net energy liberated by the reaction is* ***fully utilized*** *in heating the products of the reaction*
- For an Adiabatic process:

$$\sum_R n_i(\bar{h}_f^\circ + \Delta\bar{h})_i = \sum_P n_e(\bar{h}_f^\circ + \underbrace{\Delta\bar{h}}_{\substack{\text{At adiabatic} \\ \text{flame temperature}}})_e$$

- Compare with "*Heat/Enthalpy of Reaction*"

Steps to Calculate AFT using Enthalpy Tables

1. Calculate the total enthalpy of reactants at the specified reactant temperature
2. Assume a Temperature for the product stream
3. Calculate the product enthalpies using values from tables
4. Compare the Reactant Enthlpy Vs Product Enthalpy
5. Repeat with new product temperature, if required.

Dissociation

$$H_2 + \frac{1}{2}O_2 \rightarrow H_2O$$

$$H_2 + \frac{1}{2}O_2 \rightarrow \alpha H_2O + \beta H_2 + \delta OH + \nu H + \varepsilon O_2 + \eta O$$

- '*wasted energy opportunity*'

Forward and Backward Reactions

$$CH_4(g) + H_2O(g) \rightleftharpoons CO(g) + 3H_2(g)$$

Chemical Equilibrium

Equilibrium Constant

1.) The relative concentration of products and reactants at equilibrium is a constant.

2.) Equilibrium constant (*K*):

- For a general chemical reaction

$$aA + bB \rightleftharpoons cC + dD$$

Equilibrium constant: $$K = \frac{[C]^c[D]^d}{[A]^a[B]^b}$$

Where:

- small superscript letters are the stoichiometry coefficients
- [A] concentration chemical species A relative to standard state

Consider the combustion reaction

$$CO + \frac{1}{2}O_2 \rightarrow CO_2$$

If the final temperature is high enough, the CO2 will dissociate. Assuming the products to consist only of CO2, CO, and O2, we can write:

$$\left[CO + \frac{1}{2}O_2\right]_{\substack{cold \\ reactants}} \rightarrow \left[(1-\alpha)CO_2 + \alpha CO + \frac{\alpha}{2}O_2\right]_{\substack{hot \\ products}}$$

α is the fraction of the CO_2 dissociated.

AFT is a function of the dissociation fraction α:

α=1, no heat released and unchanged.

α=0, the maximum amount of heat release occurs and the temperature and pressure would be the highest possible allowed by the first law

The Direction of Reaction…

- Concern of Second Law of thermodynamics
- Gibb's Function:

$$\Delta G^o = \Delta H^o - T\Delta S^o$$

- The Rationale for the above definition:

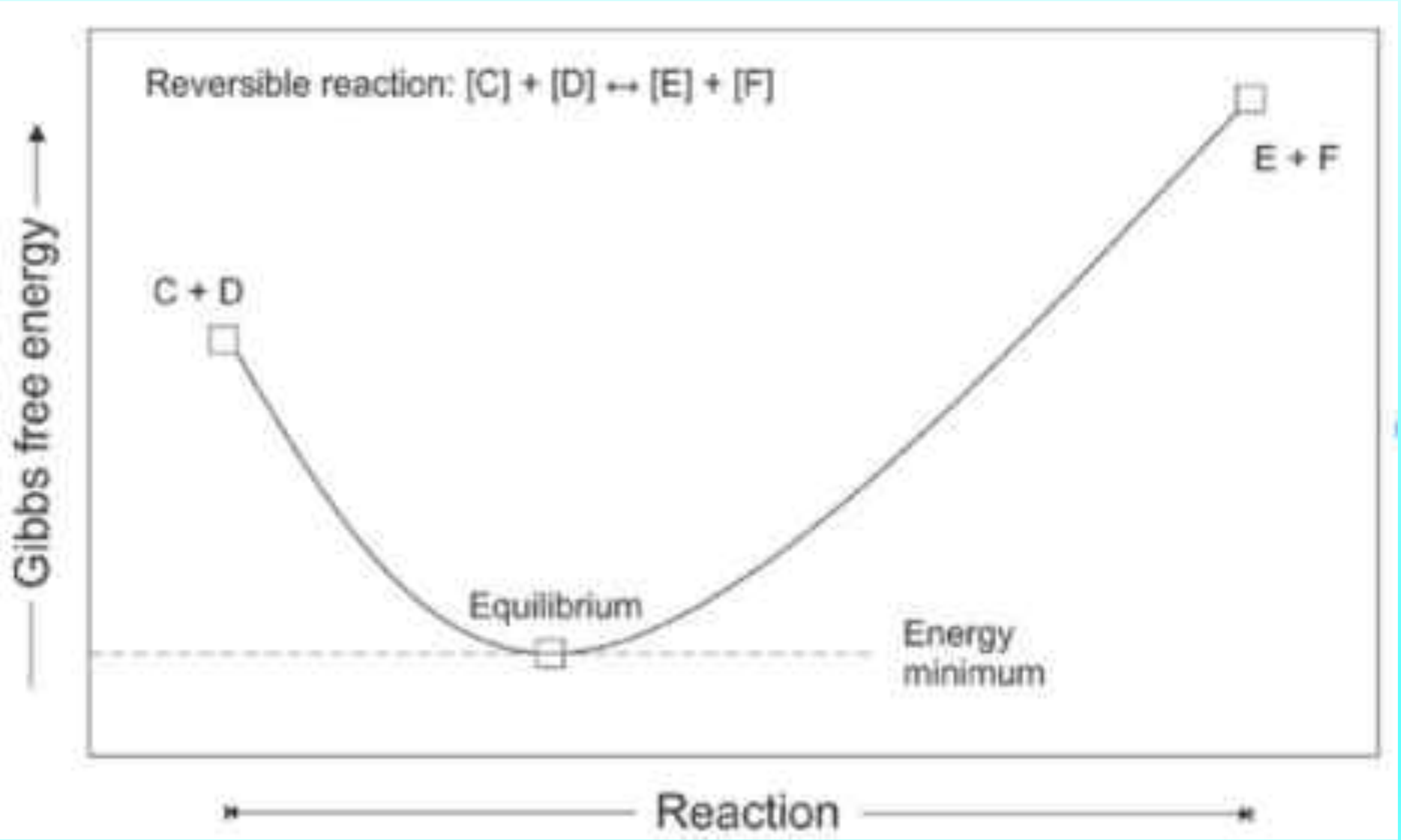
$$\Delta S_{univ} = \Delta S_{surr} + \Delta S_{sys}$$

Since = Δs_{surr} = $\Delta q/T = -\Delta q_{sys}/T$

$$\Delta S_{univ} = \frac{-\Delta H_{sys}}{T} + \Delta S_{sys}$$

$-T\Delta S_{univ} = \Delta H_{sys} - T\Delta S_{sys}$

$-T\Delta S_{univ}$ = change in Gibbs free energy for the system = ΔG_{system}


Reversible reaction: [C] + [D] ↔ [E] + [F]
Gibbs free energy
C + D
E + F
Equilibrium
Energy minimum
Reaction

COMBUSTION KINETICS

- Study of the factors that govern the speed or rates at which the combustion reactions occur

$$H_2 + O_2 \rightarrow HO_2 + H$$

$$H + O_2 \rightarrow OH + O$$

$$OH + H_2 \rightarrow H_2O + H$$

$$H + O_2 + M \rightarrow HO_2 + M$$

Reactions..

- An *ELEMENTARY* reaction is one that occurs on a molecular level exactly in the way which it is described by the reaction equation
  - *Example: $OH + H_2 \rightarrow H_2O + H$*
- *A GLOBAL reaction is an overall result of a series of elementary reactions*
  - *$2H_2 + O_2 \rightarrow 2H_2O$*

Rate Law & Order

- Rate Law:

For a reaction, $aA + bB \rightarrow cC + dD$

Reaction Rate RR = $k[A]^x[B]^y$

X and y are numbers determined experimentally.

- The exponents *x and y specify the relationships between the concentrations of* reactants A and B and the reaction rate
- *Order of Reaction: the sum of the powers to which all reactant concentrations appearing in the rate law are raised (= x + y in above reaction)*

Rate of Reaction

$$\mathbf{A} + \mathbf{B} \rightarrow \mathbf{C} + \mathbf{D}$$

$$\mathbf{Rate} = -\frac{\Delta \mathbf{A}}{\Delta \mathbf{t}} = -\frac{\Delta \mathbf{B}}{\Delta \mathbf{t}} = \frac{\Delta \mathbf{C}}{\Delta \mathbf{t}} = \frac{\Delta \mathbf{D}}{\Delta \mathbf{t}}$$

Reaction Rate RR = *k[A]^x[B]^y*

Collision theory

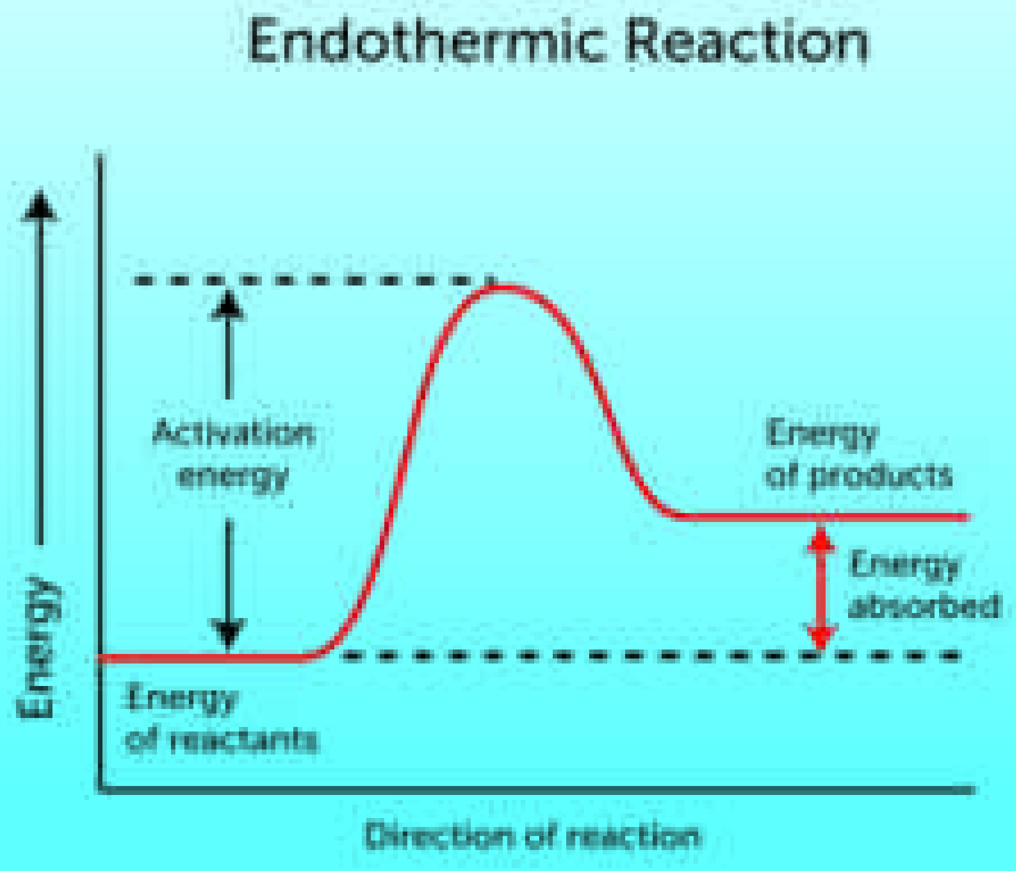
- Molecules have to collide if they are to react
- Increasing concentration increases the frequency of collisions
- Increasing pressure increases frequency of collisions
- Increasing temperature increases frequency of collision

Activation Energy E_a

Energy of the collision must be above a certain value for reactants to react

The minimum energy required for a collision to result in chemical reaction is **Ea**

Energy Diagram



Reaction Rate and Temperature

$$Rate = k[A]^n[B]^m$$

The Arrhenius equation

$$k = Ae^{-\frac{E_a}{RT}}$$

$$\ln k = \ln A - \frac{E_A}{RT}$$

Impact of Increasing Tempertaure ?

Impact of decrease in Ea ?

Example

- Calculate the Rate constant for this reaction at 323 K

Temp K	E_a kJ mol^{-1}	A L mol^{-1} s^{-1}	k L mol^{-1} s^{-1}
313	54	8.7×10^6	8.5×10^{-3}
323	54	8.7×10^6	??

T	Ea	A	lnA	R	Ea/RT	lnk	k L mol^{-1} s^{-1}
313	54000	8700000	15.98	8.31	20.8	-4.8	**0.008462**
323	54000	8700000	15.98	8.31	20.1	-4.1	**0.016087**

- Note that R to be used is the value of Universal Gas Constant = 8.314 J/mol-K (NOT the specific gas constant)

Assignment

To be submitted by: **9^{th} Jan 2023**

***h in Km** = 1+RN*0.2 where RN is the last two digits of your roll number*

- *Consider a Turbojet engine operating at an altitude of **h km** with flight Mach number of 1.5 ingesting air at the rate of 60 kg/s.*
- *The intake can be assumed to be adiabatic but entails a stagnation pressure loss of 5%. The compressor operates with a polytropic efficiency of 0.86 and stagnation pressure ratio = 25. The fuel used has a calorific value of $43X10^6$ J/kg and the combustor exit temperature = 1500 K. The turbine operates with a polytropic efficiency of 0.87.*
- *Expansion in the nozzle can be assumed to reversible and adiabatic – The nozzle is perfectly expanded.*
- *Under these conditions calculate the thrust developed, TSFC and the thermal efficiency of the engine.*

For Submission..

- *Use standard atmospheric tables for static pressure and static temperature at the corresponding altitude*
- Submit assignments with the entire calculation and a table at the end with the following format:

NAME	
Roll Number	
Altitude (km), Atm Pressure (Pa) & Atm Temperature (K)	
Thrust, N	
TSFC, kg/s/N	
Thermal Efficiency, %	

Flames..

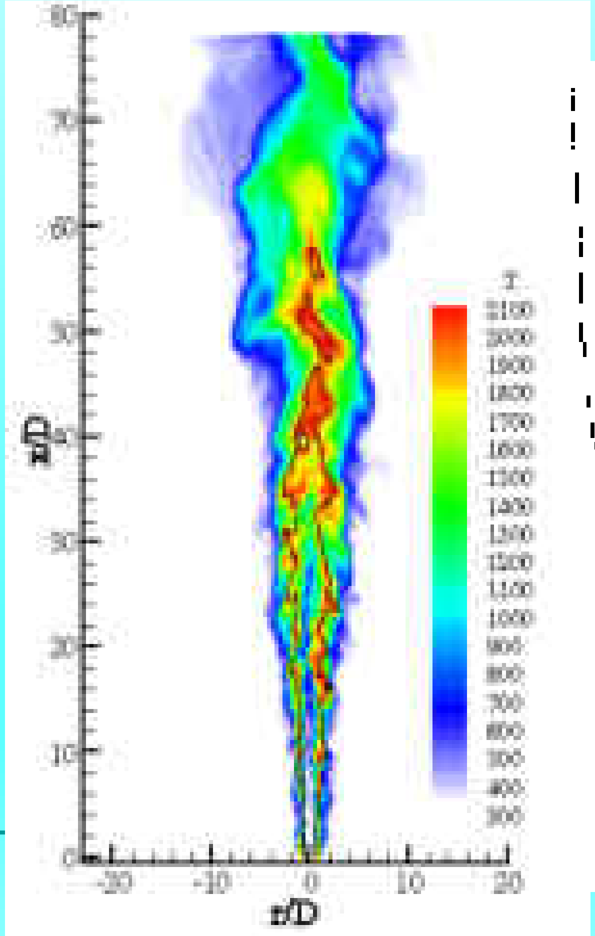
- A flame is a self-sustaining propagation of a localized combustion zone at subsonic velocities

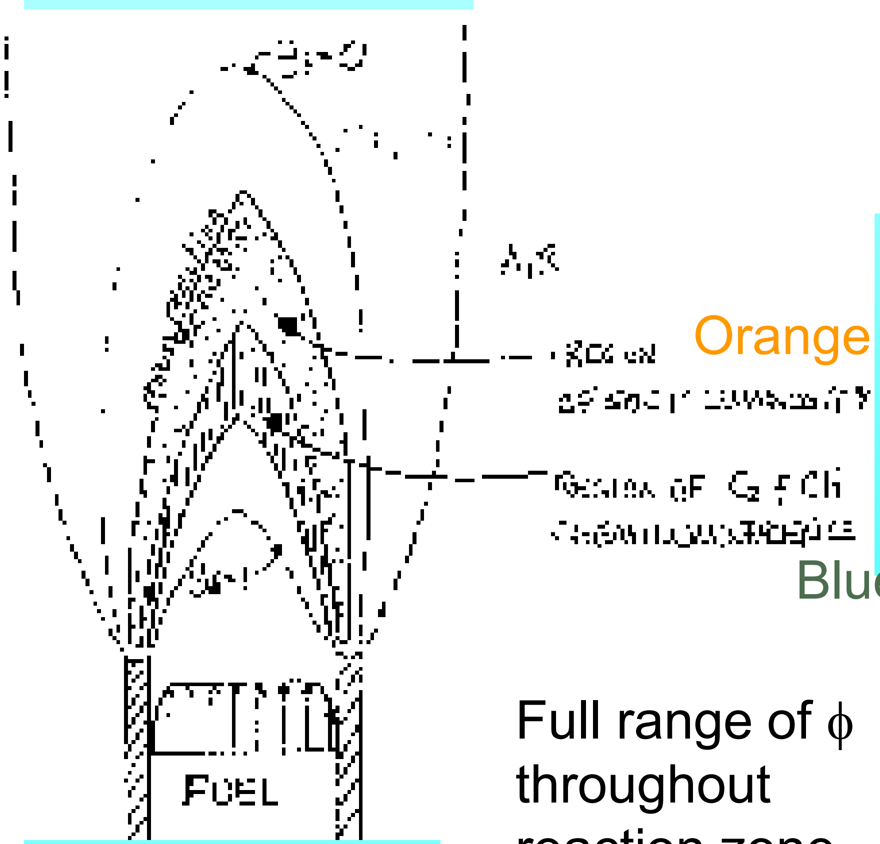
Laminar & Premixed

Fuel and oxidizer mixed at molecular level prior to occurrence of any significant chemical reaction

DIFFUSION FLAMES

- Reactants are initially separated, and reaction occurs only at interface between fuel and oxidizer (mixing and reaction taking place)
- In turbulent diffusion flames, turbulent convection mixes fuel and air macroscopically, then molecular mixing completes the process so that chemical reactions can take place





Orange

Blue

Full range of ϕ throughout reaction zone

STRUCTURE OF LAMINAR PREMIXED FLAMES

- Temperature and Mass Fraction Distribution
- Reaction: Zones within the flame
  1. Preheat zone: little heat is released
  2. Reaction zone: most of the chemical energy is released

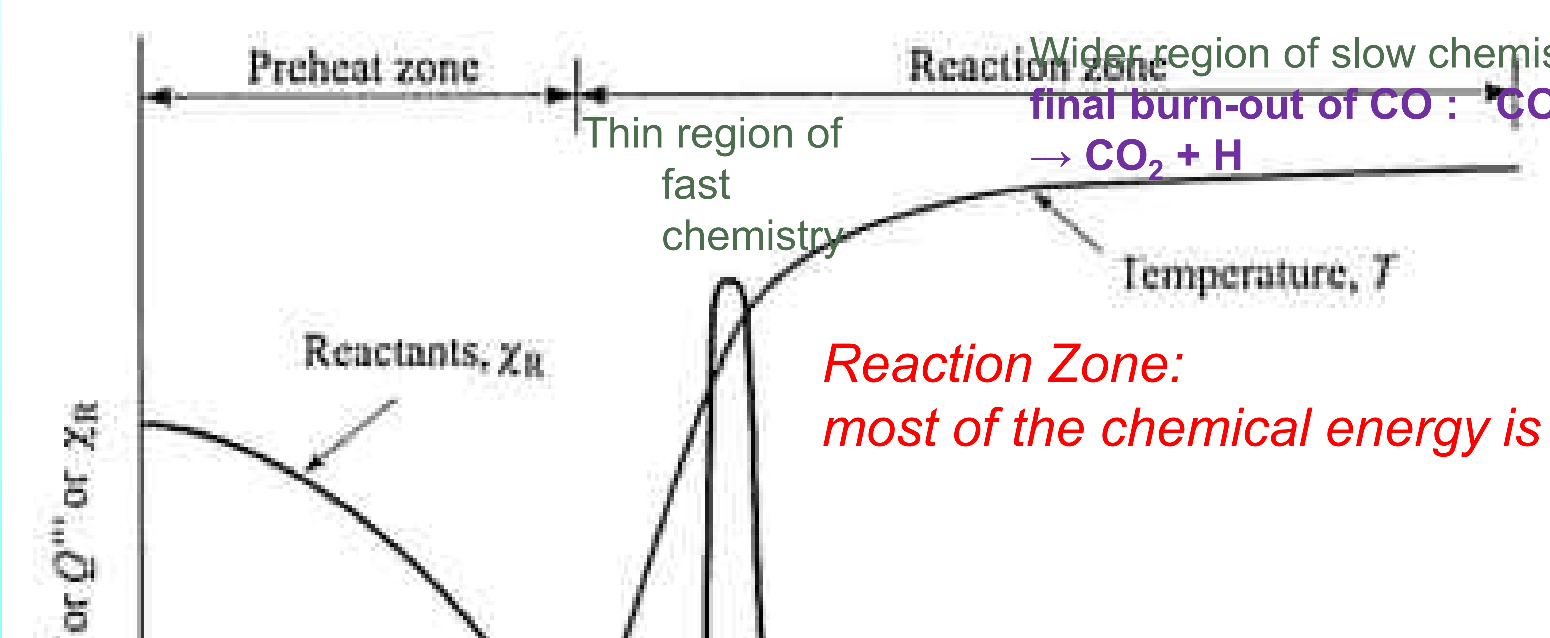
     2.a Thin region of fast chemistry

     - Destruction of fuel molecules and creation of intermediate species
     - Temperature and species concentration gradients are very large
     - The large gradients provide the driving forces for the flame to be self-sustaining

     2.b Wider region of slow reactions

     - At atmospheric pressure, this zone may extend several mm

LAMINAR FLAME STRUCTURE

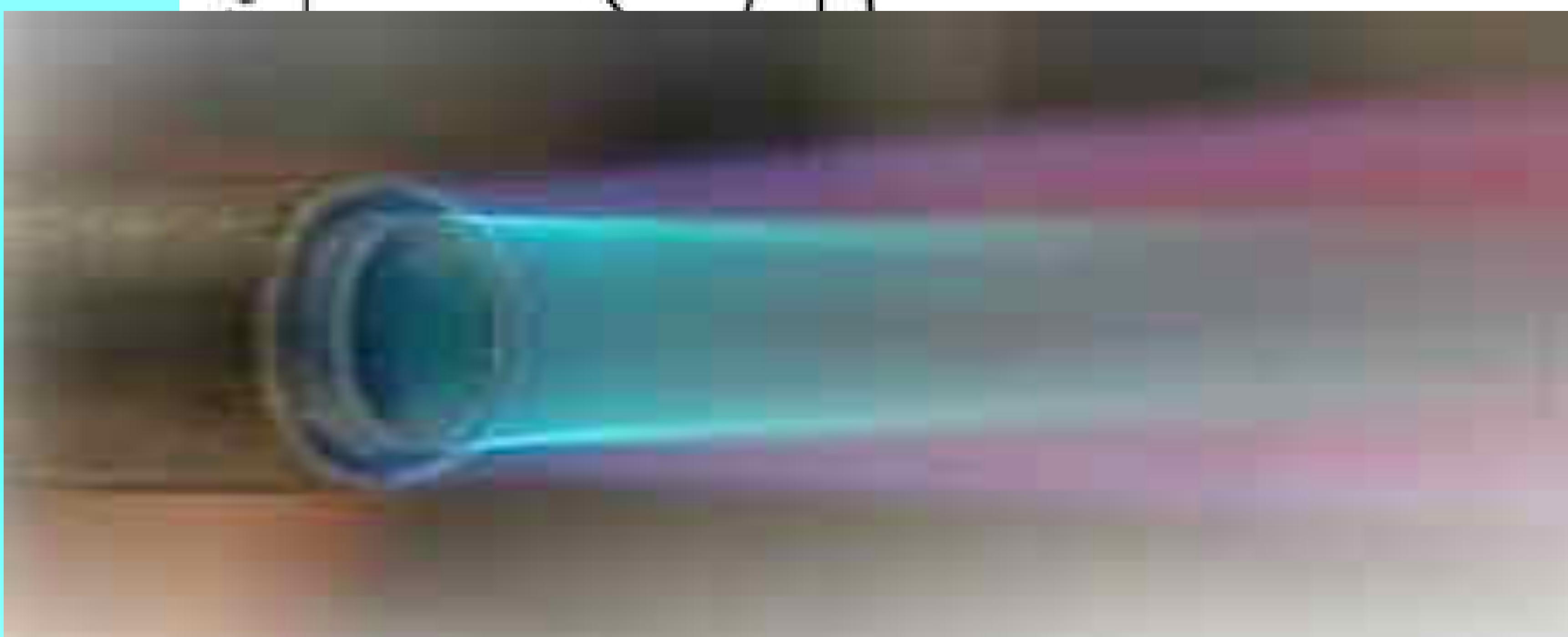


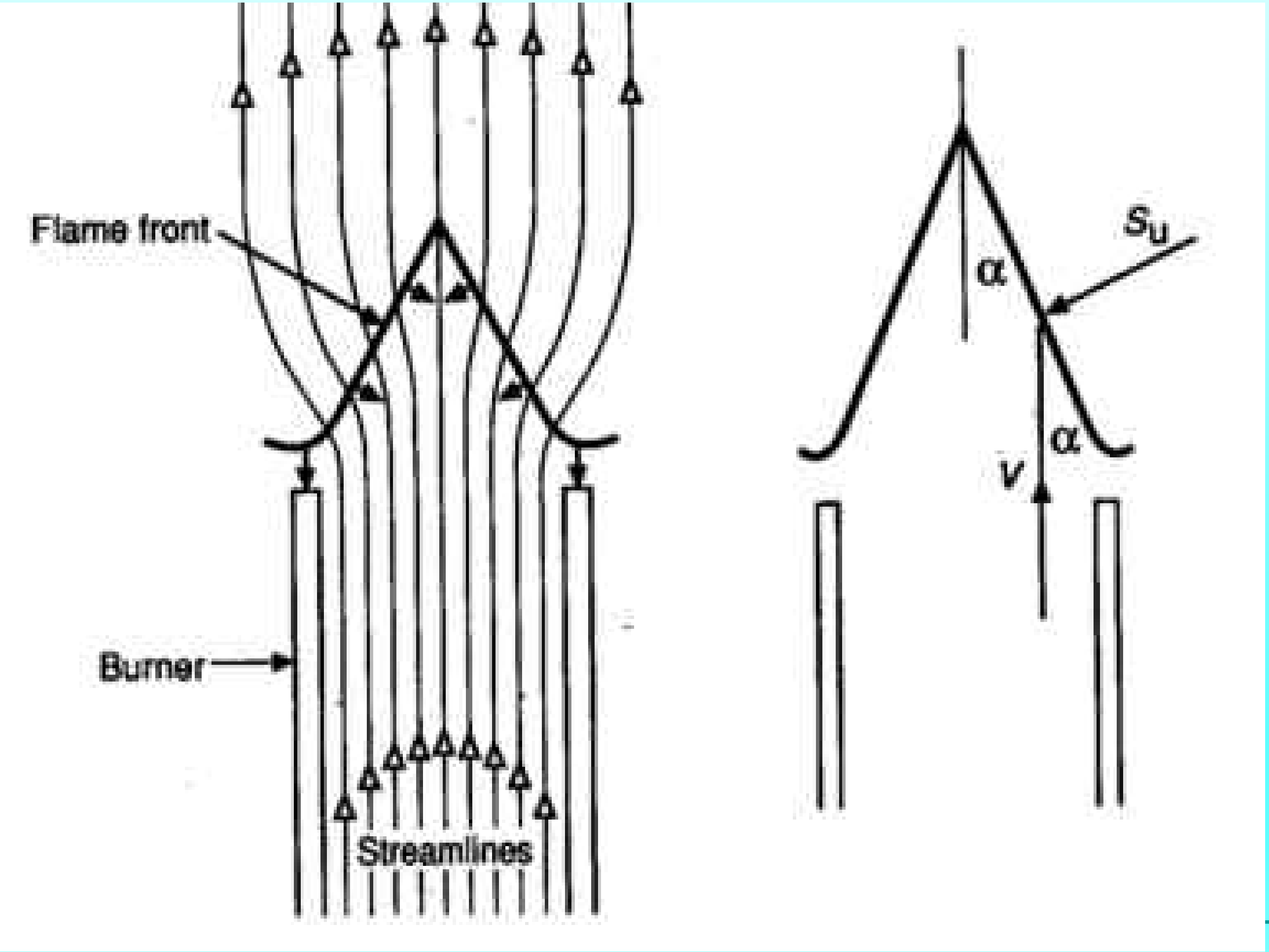
Thin region of fast chemistry

Wider region of slow chemistry

final burn-out of CO : CO + OH → CO_2 + H

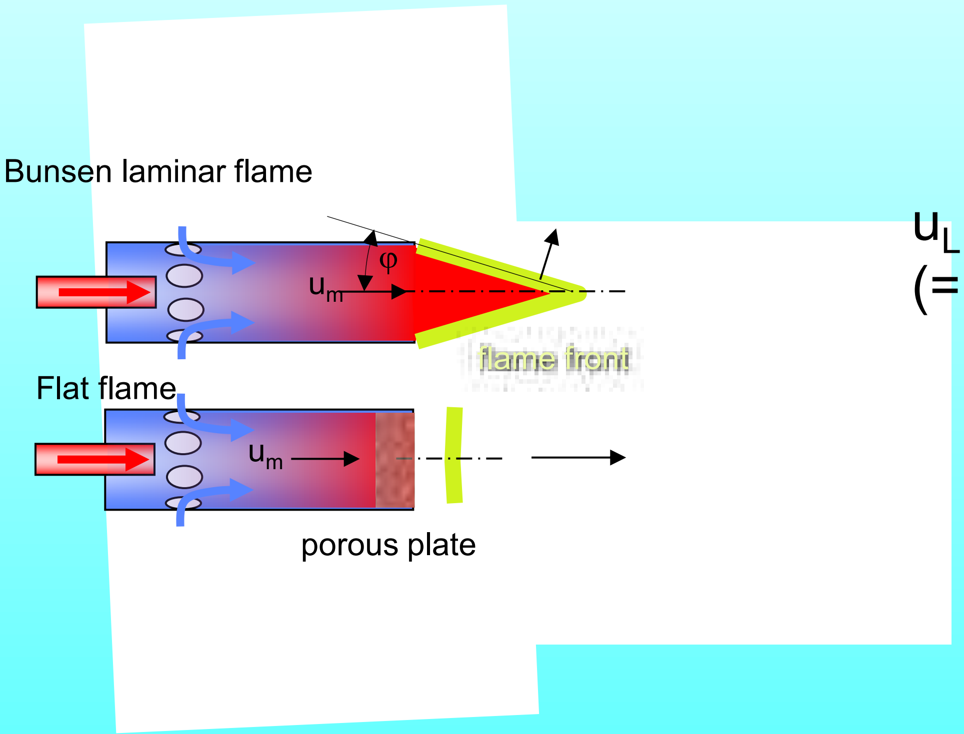
Reaction Zone:
most of the chemical energy is release



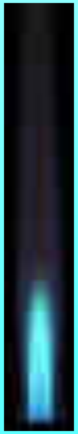


Flame front
Burner
Streamlines
S_u
α
α
V

Laminar Flame Speed



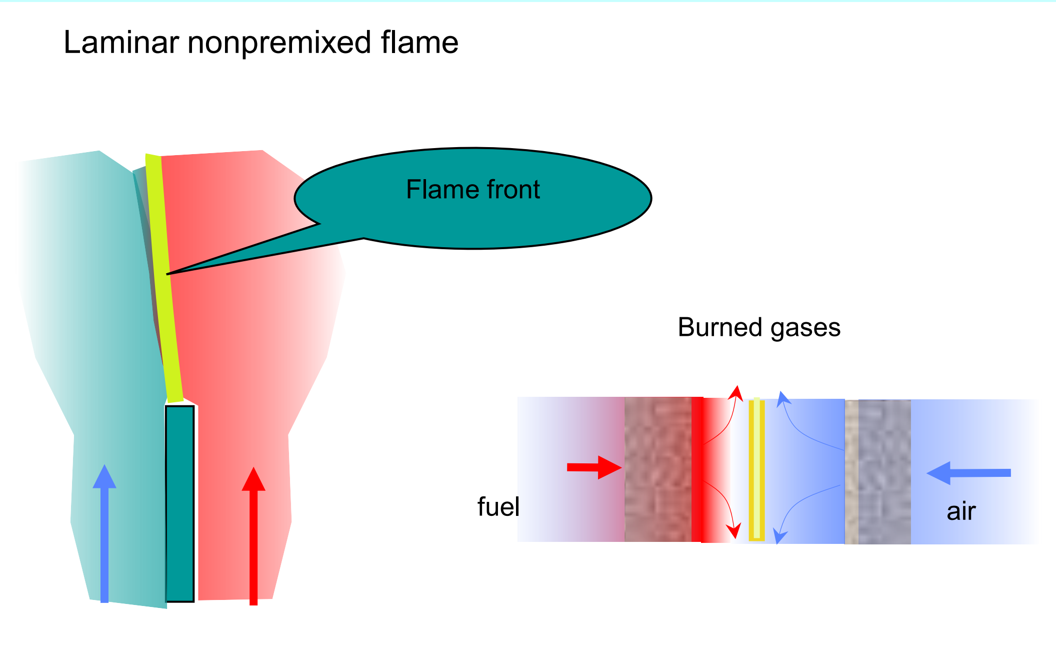
u_L burning velocity
$(=u_m \sin\varphi)$

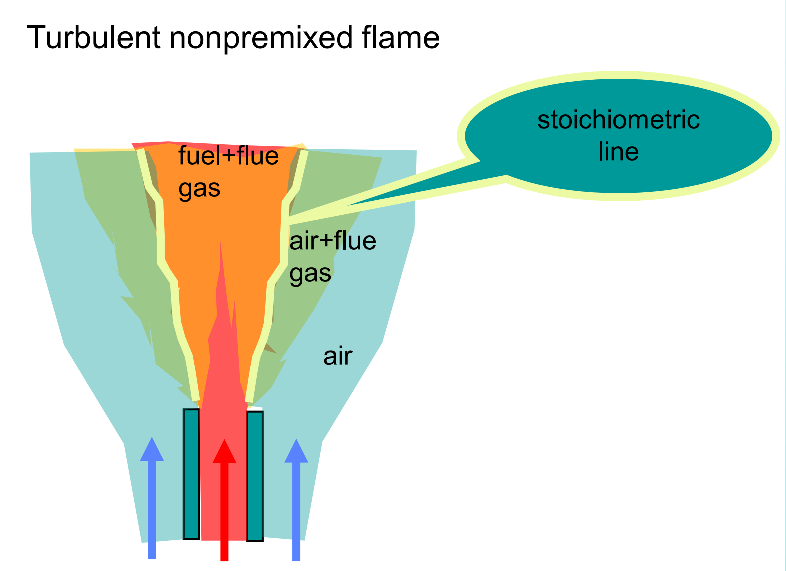


Laminar Flame Characteristics

- Shape of flame
  - Velocity profile, flame speed, heat loss to tube wall
- For the flame to remain stationary:
  - Flame speed must equal speed of normal component of unburned gas (at each location)
- Factors influence laminar flame speed and flame thickness
  - ϕ, T, P, fuel type

Diffusion *(Non-Premmixed)* Flames





Nonpremixed flames do not propagate.

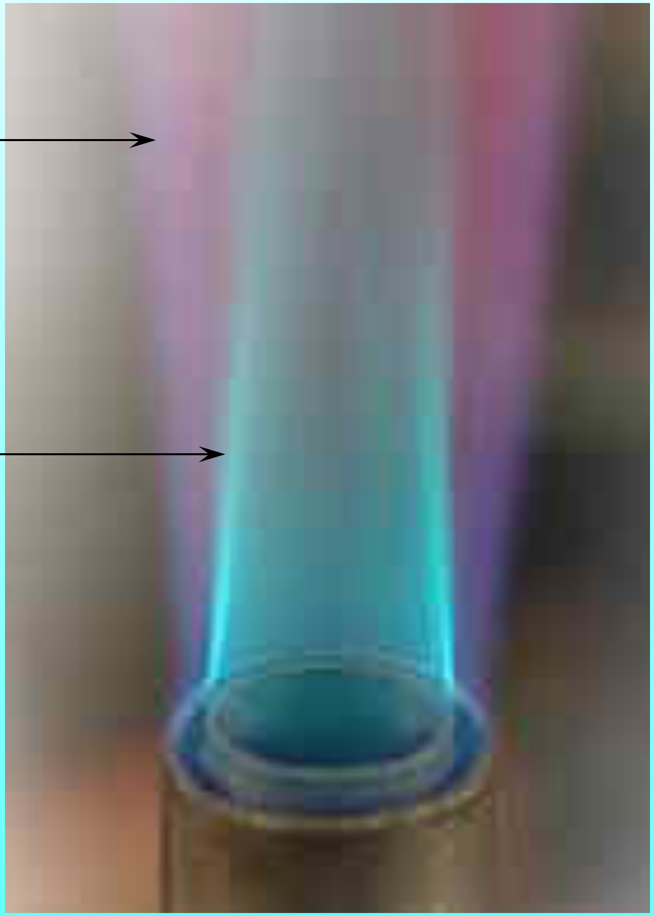
Only burned products diffuse to both sides of the flame front

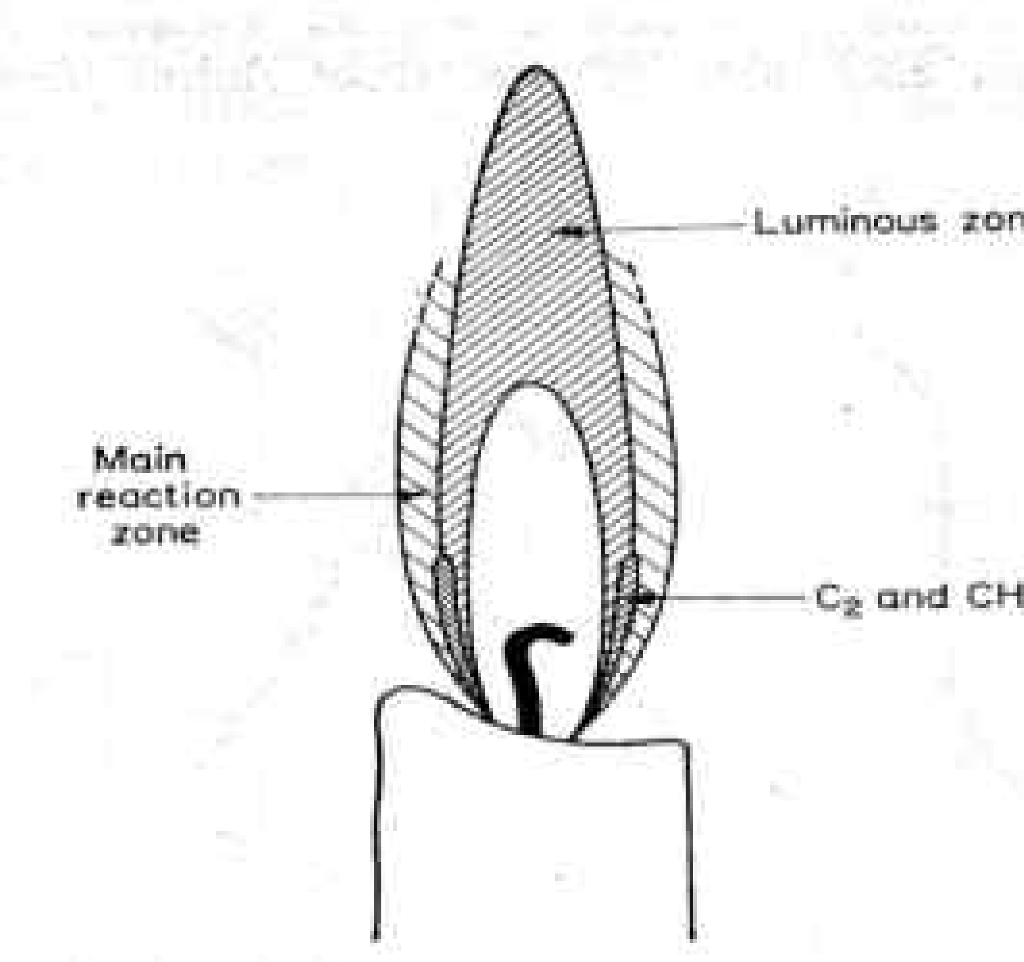
Flame front position corresponds to stoichiometric ratio fuel/air – there is also the highest temperature.

- Diffusion Flame: Combustion occurs at the interface between the fuel gas and the oxidant gas and the burning rate ***depends primarily on rate of diffusion of reactants***
  - In pre-mixed flame, burning rate depends more on the ***rates of checmical reactions*** involved

Secondary **diffusion** flame results when CO and H products from rich inner flame encounter ambient air

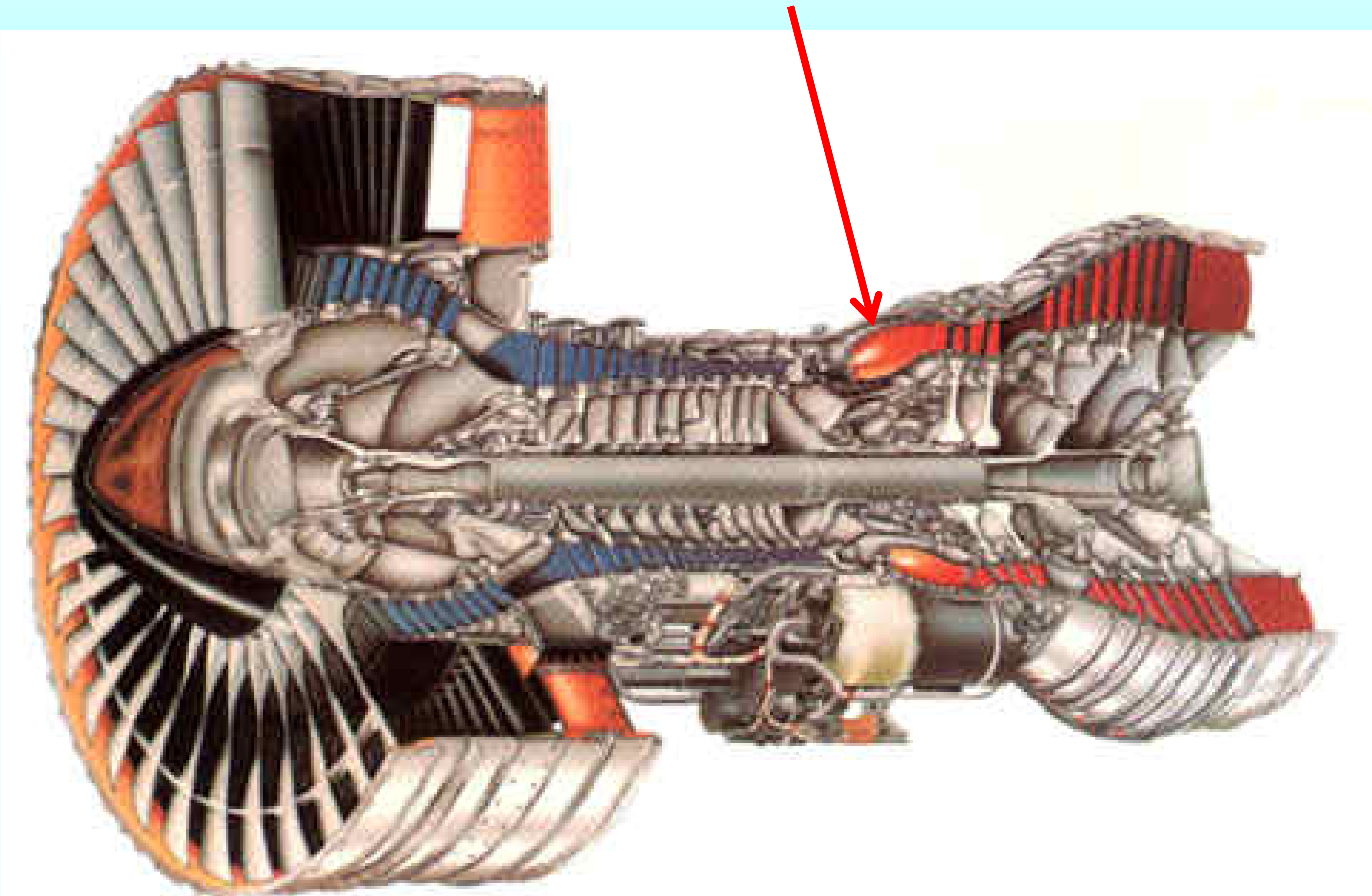
Fuel-rich **pre-mixed** inner flame



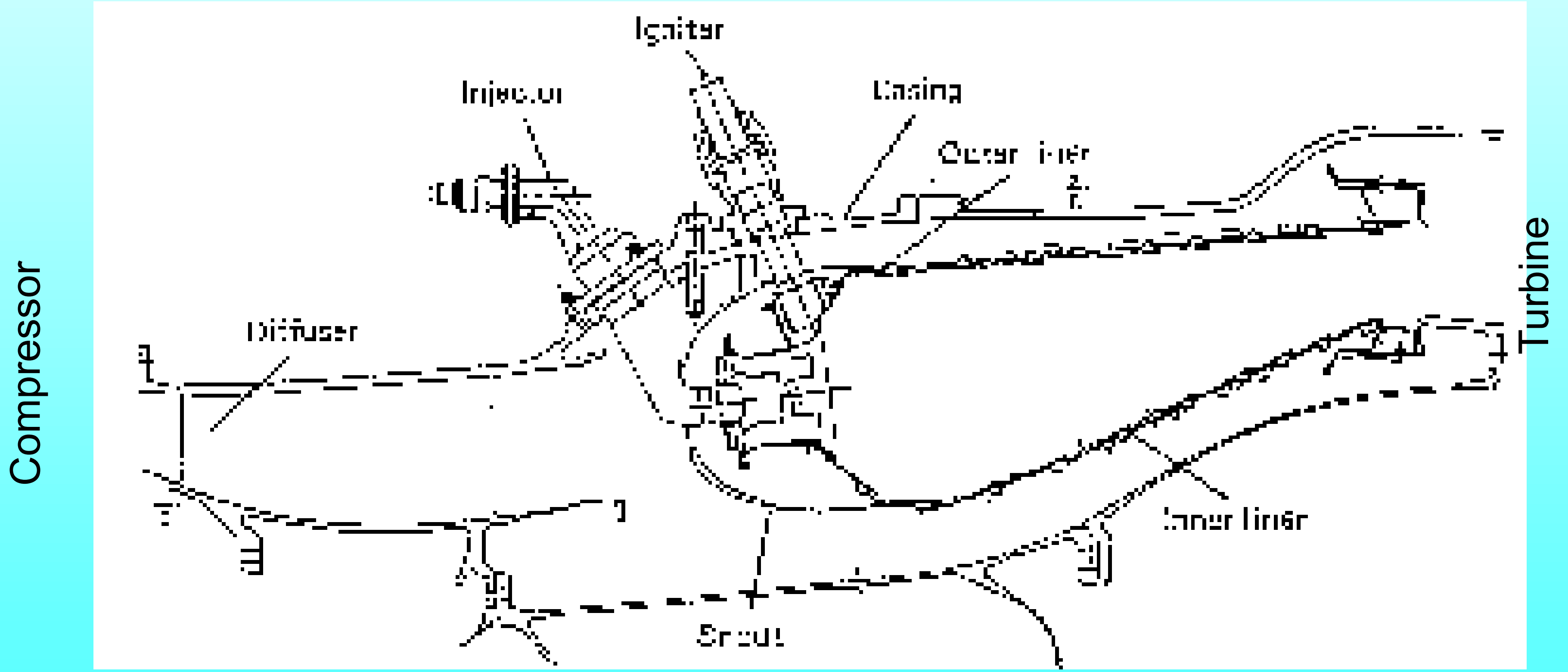


Main
reaction
zone

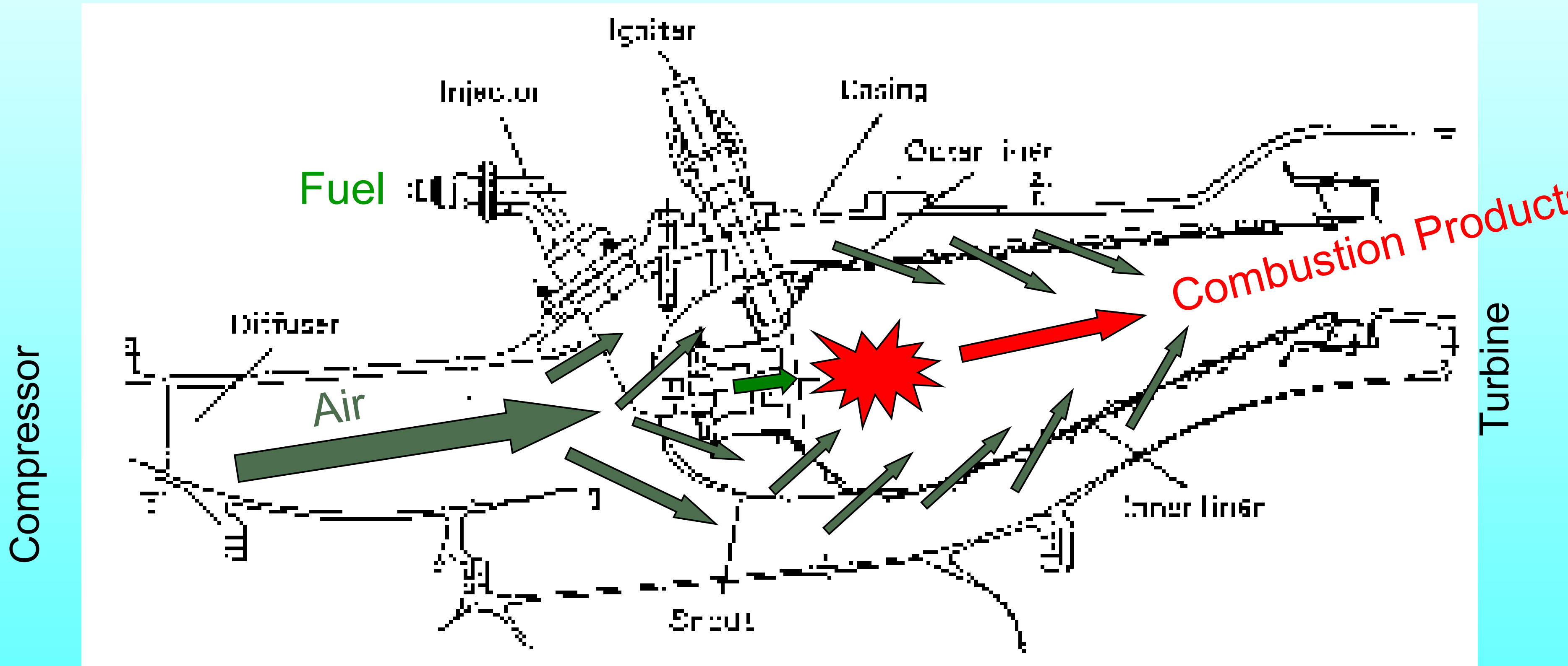
AIRCRAFT COMBUSTORS

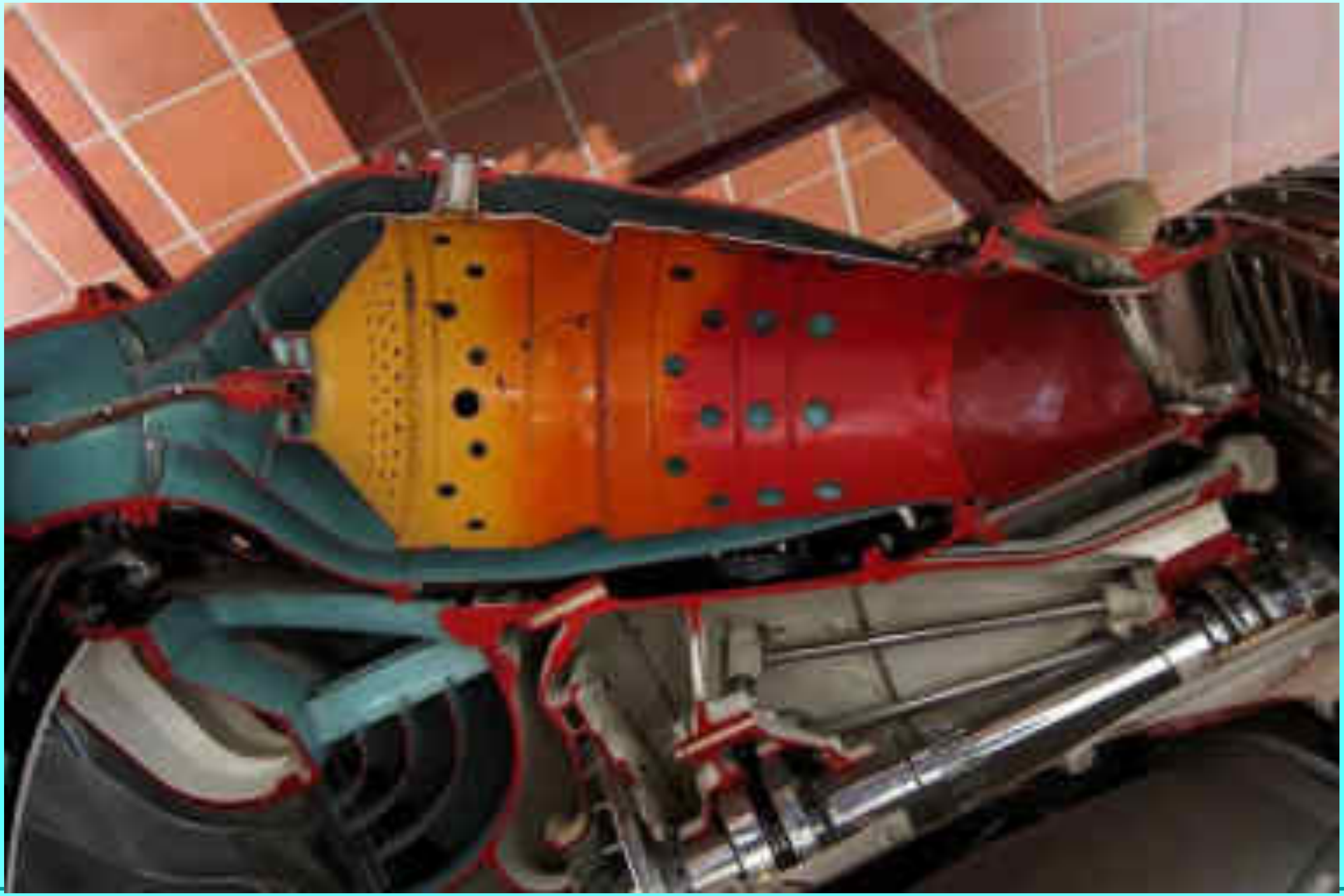


MAJOR COMBUSTOR COMPONENTS

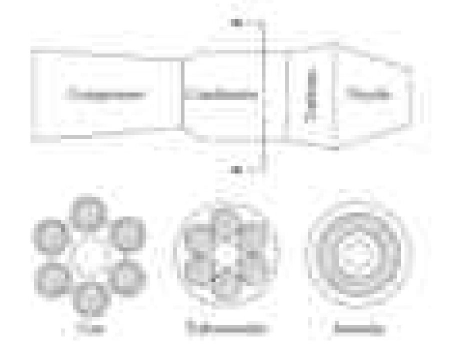


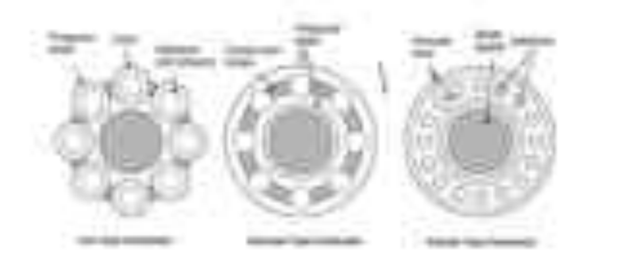
MAJOR COMBUSTOR COMPONENTS

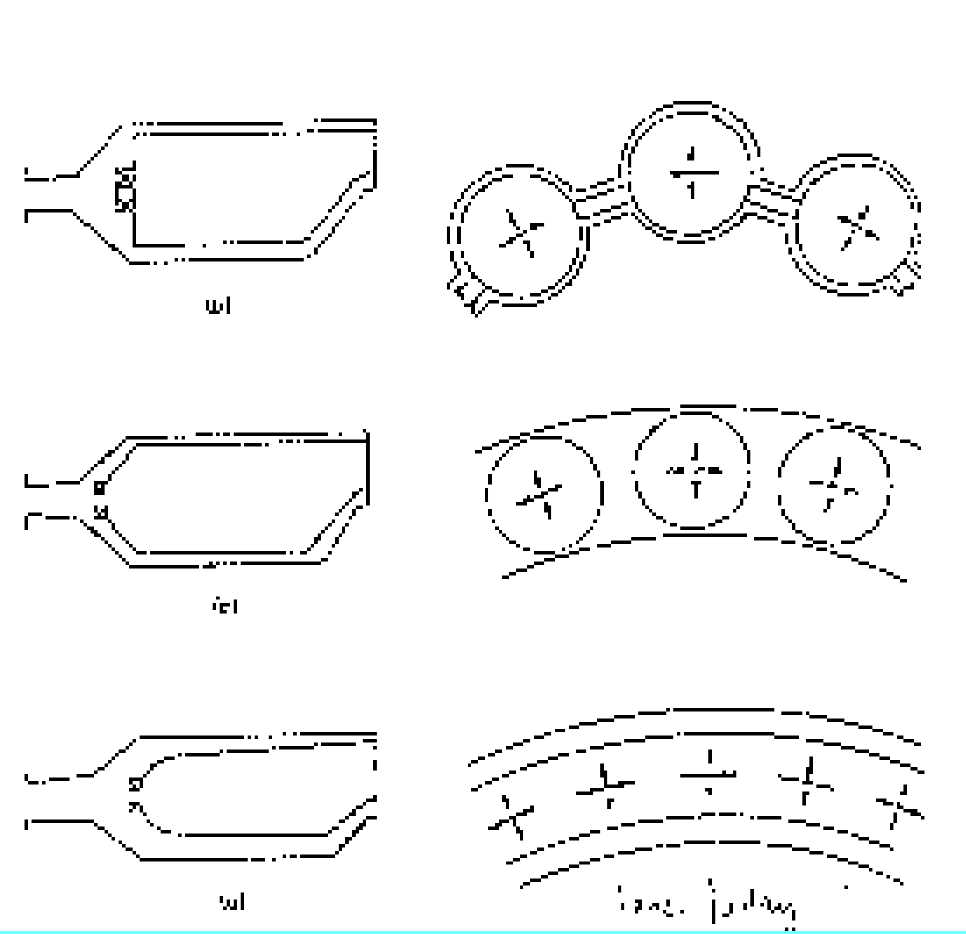




COMBUSTOR TYPES





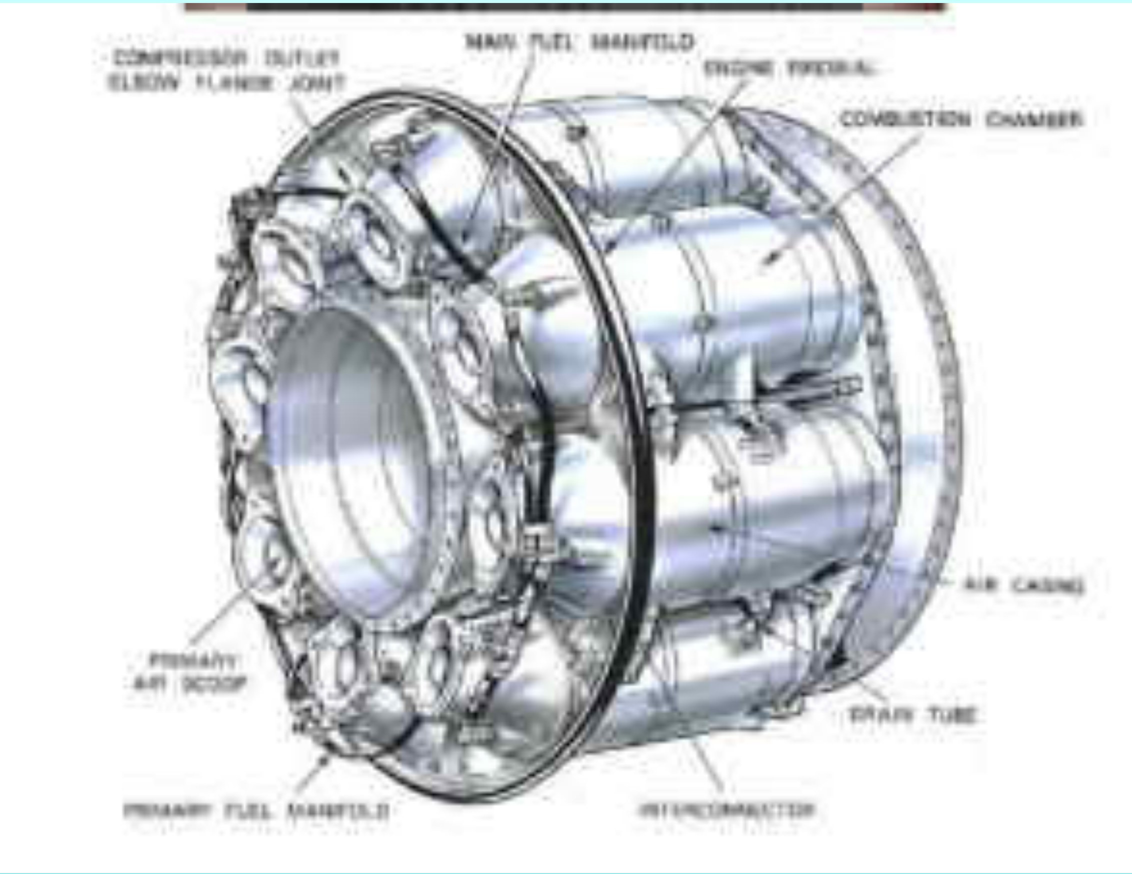


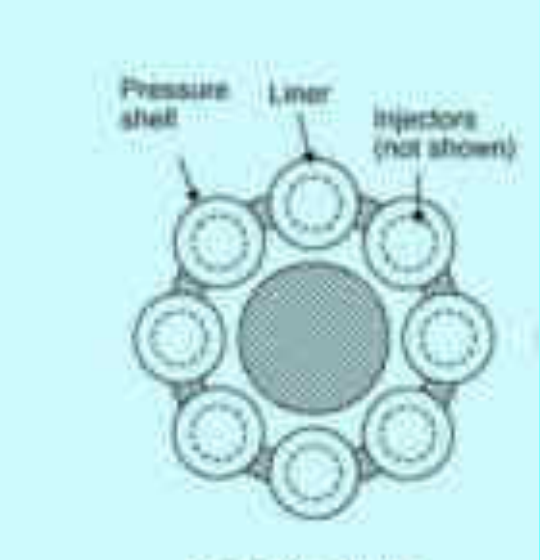
Tubular
or Multi-Can

Tuboannular
Can-Annular

Annular

Types of Combustors: Can/Tubular

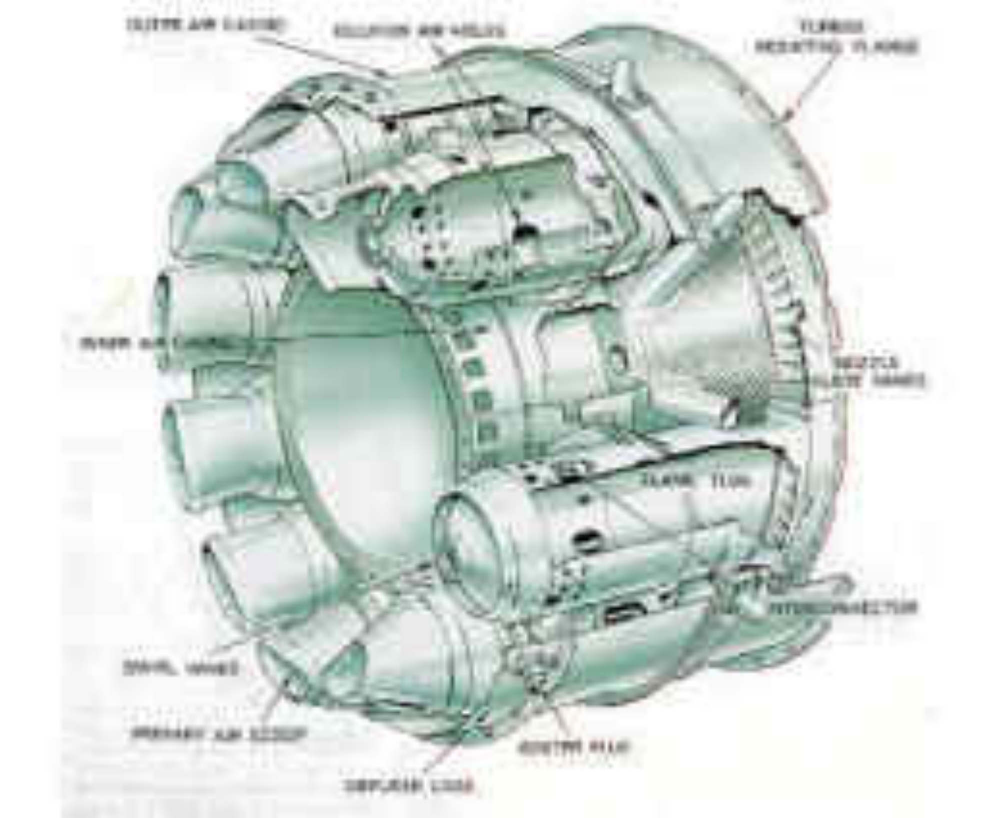


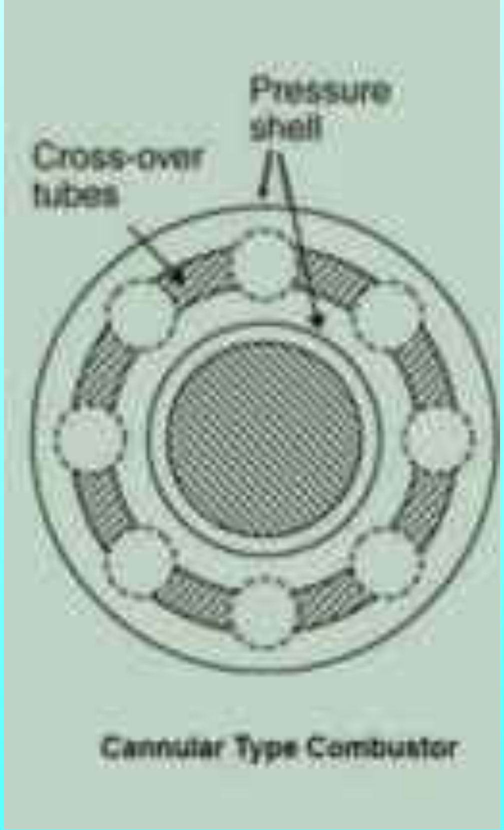


Can Type Combustor

- Easy for maintenance
- Heavy
- Less efficient
- High pressure loss

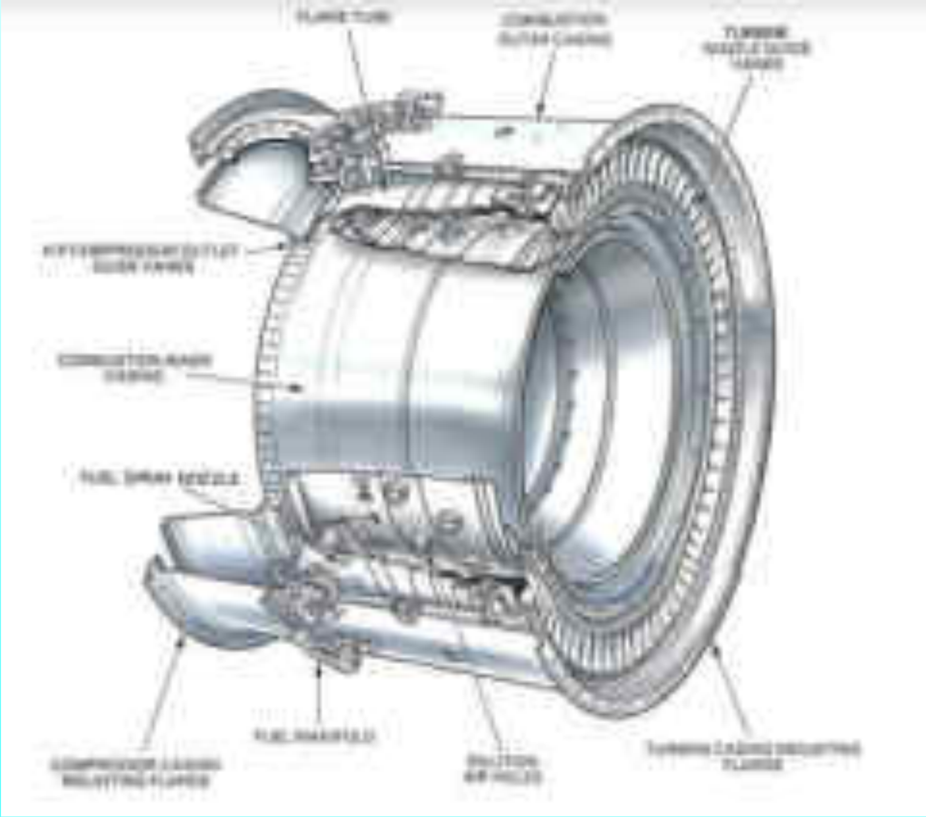
Types of Combustors: Can-Annular

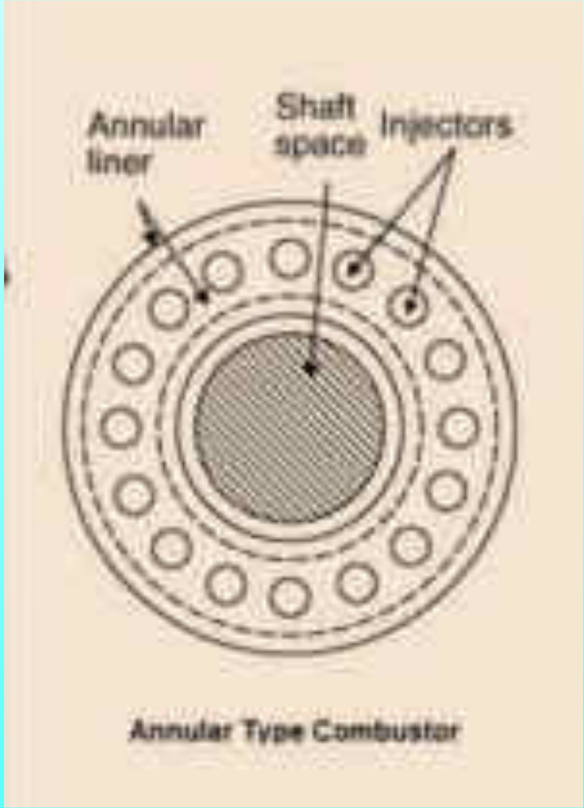




- Lighter than can-type
- Individual ignition not required

Types of Combustors: Annular



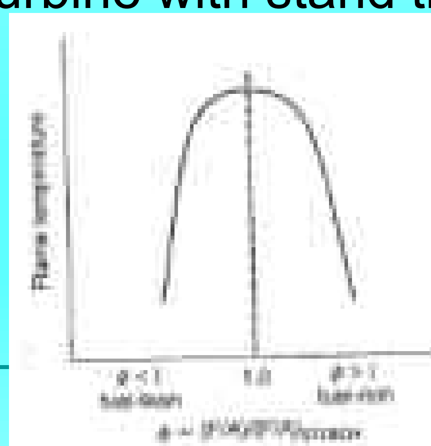


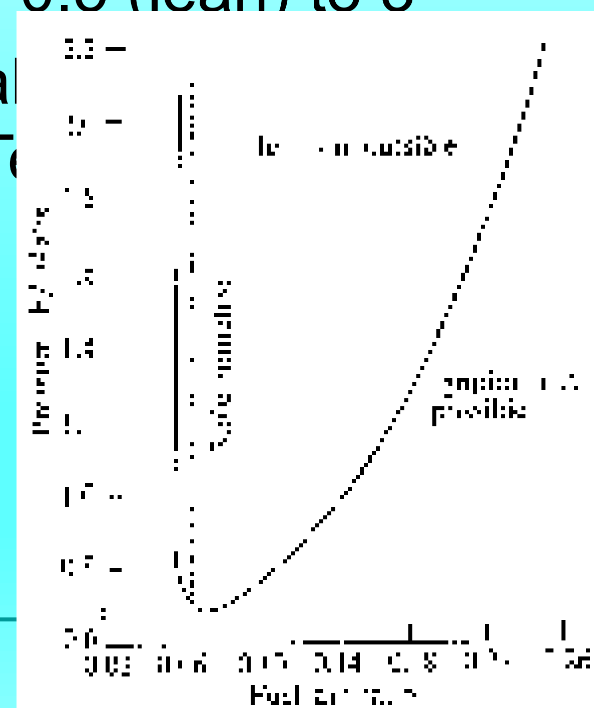
- Light, Compact
- Most efficient
- Less pressure loss
- Not easy for repair

AIRCRAFT COMBUSTOR: *THE CHALLENGES..*

Effect of Pressure On Flammability Limits

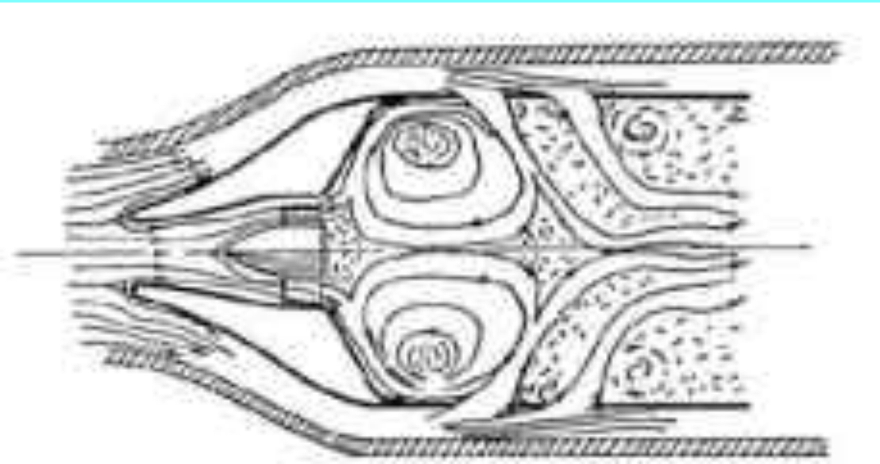
- "Flammability": flames can propagate thro air mixtures only within certain limits of compositio
- Limits of Flammability: "Lean" Limit & "Rich" Limit
- Typically within equivalence ratios: 0.5 (lean) to 3
- What if we employ a safely flammal of say 0.7 ? → Estimate resulting T
  - Can turbine with stand this ?
- *So ?!*

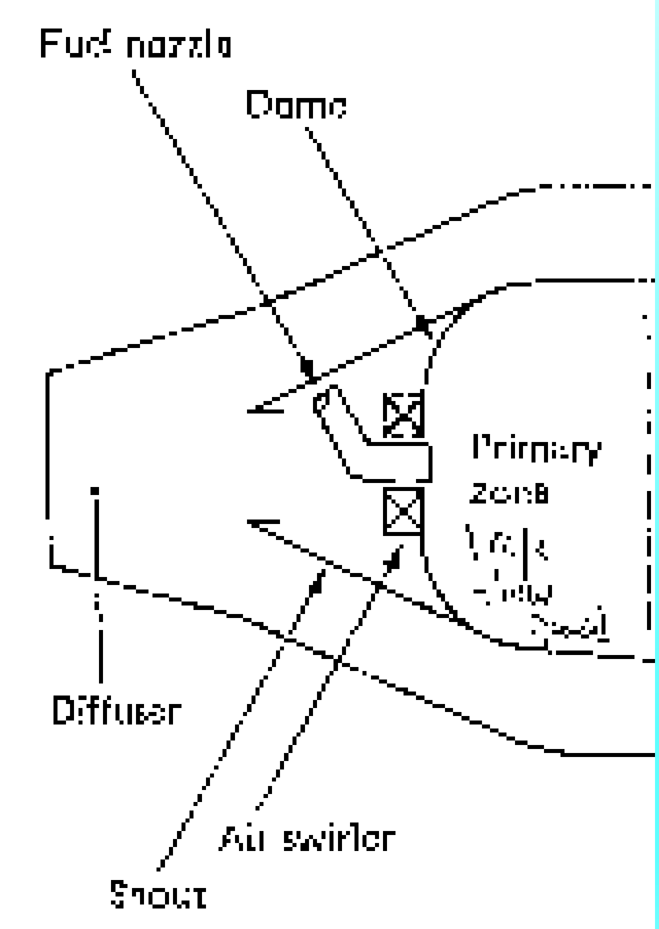


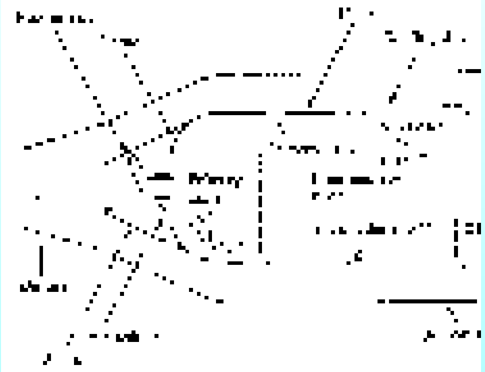


Combustor Flow Field

1. Primary Zone
   - Anchors Flame
   - Provides sufficient time, mixing, temperature for “complete” oxid
     of fuel
   - Equivalence ratio near $\phi=1$



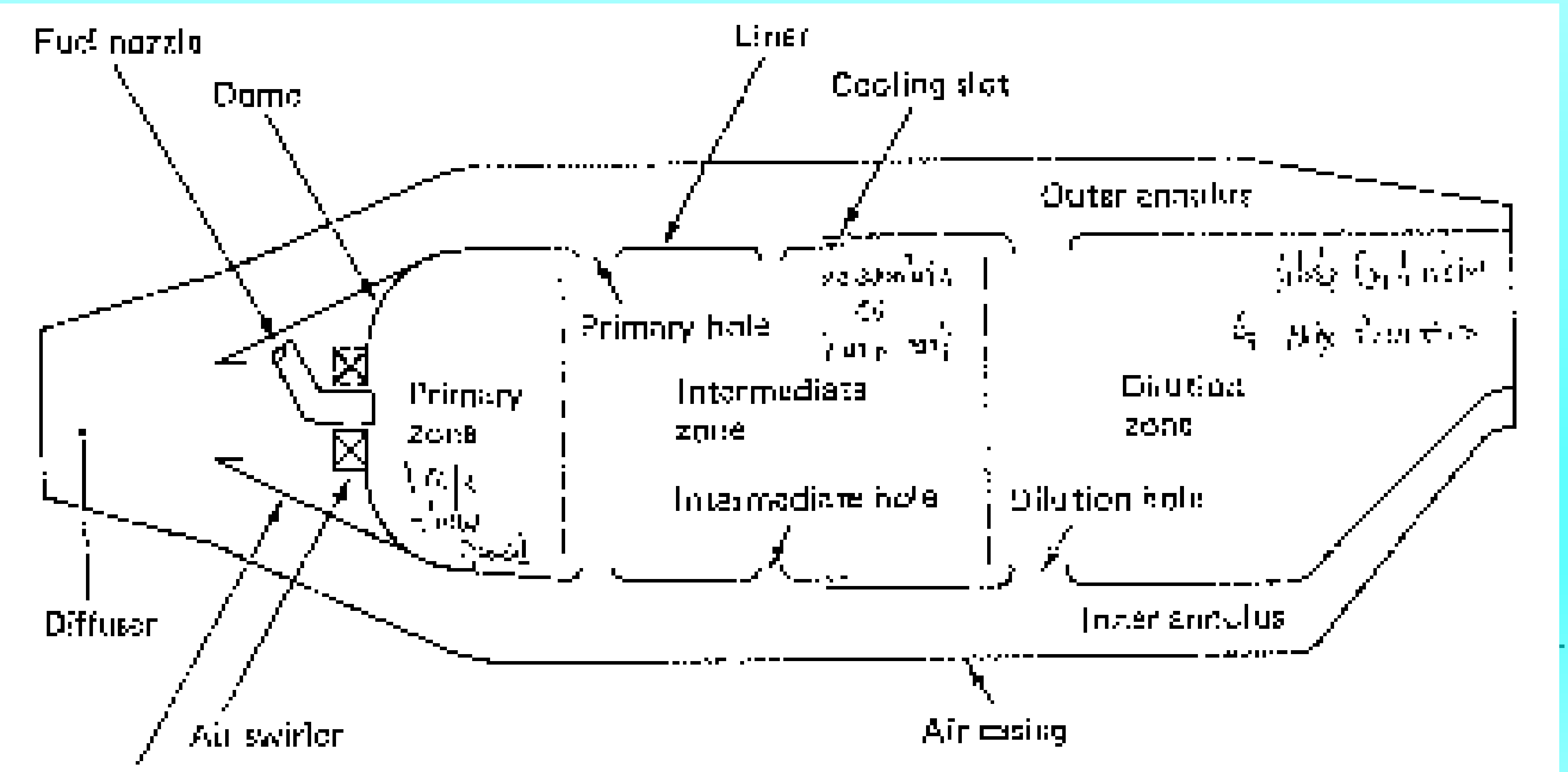




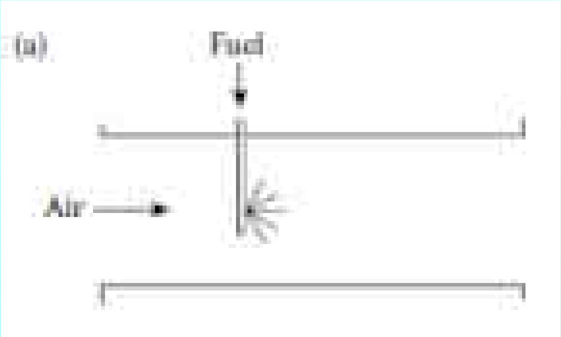
2. Intermediate (Secondary Zone)

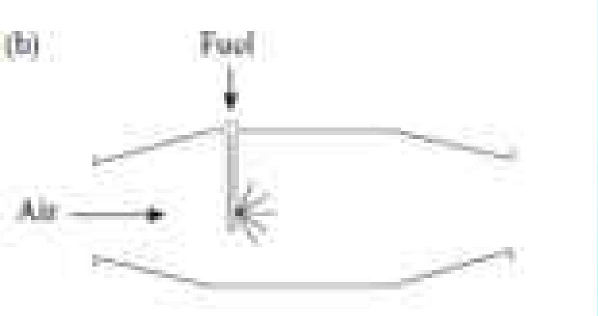
- **Low altitude** operation (higher pressures in combustor)
  - Recover dissociation losses (primarily CO → CO_2) and Soot Oxidation
  - Complete burning of anything left over from primary due to poor mixing
- **High altitude** operation (lower pressures in combustor)
  - Low pressure implies slower rate of reaction in primary zone
  - Serves basically as an extension of primary zone (increased τ_{res})
- L/D ~ 0.7

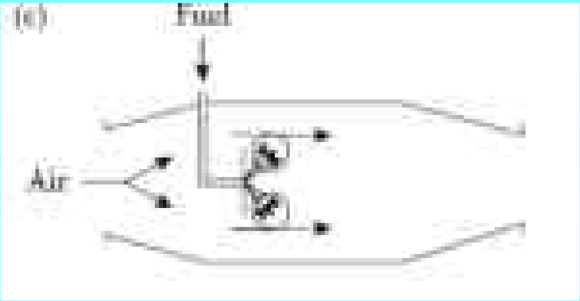
- 3. Dilution Zone: “Cools” down to suit turbine limitations

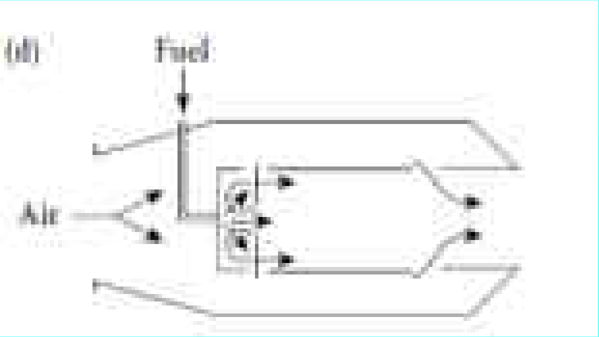


Conceptual Evolution..









AMRITA
VISHWA VIDYAPEETHAM

Aerospace Propulsion

Dr. A.R. Srikrishnan
Department of Aerospace engineering

Nozzles, RAMJET, Afterburner

Polytropic Efficiency, Turbine

- Defining polytropic efficiency for incremental expansion, for turbine:

$$\eta_{poly} = \frac{dh_t}{dh_{ts}} = \frac{dh_t}{\frac{dp_t}{\rho_t}}$$

- Deriving On the same lines as for the compressor:

$$\tau_t = \pi_t^{\frac{\eta_{poly}(\gamma-1)}{\gamma}}$$

$$\eta_t = \frac{1-\tau_t}{1-\tau_t^{1/\eta_{poly}}}$$

Numerical Problem

- Determine the polytropic efficiency and the exit stagnation temperature of an aircraft turbine under the following conditions: Specific Mechanical work produced = 820 kJ/kg; Isentropic efficiency = 0.86; Inlet stagnation temperature = 1490 K

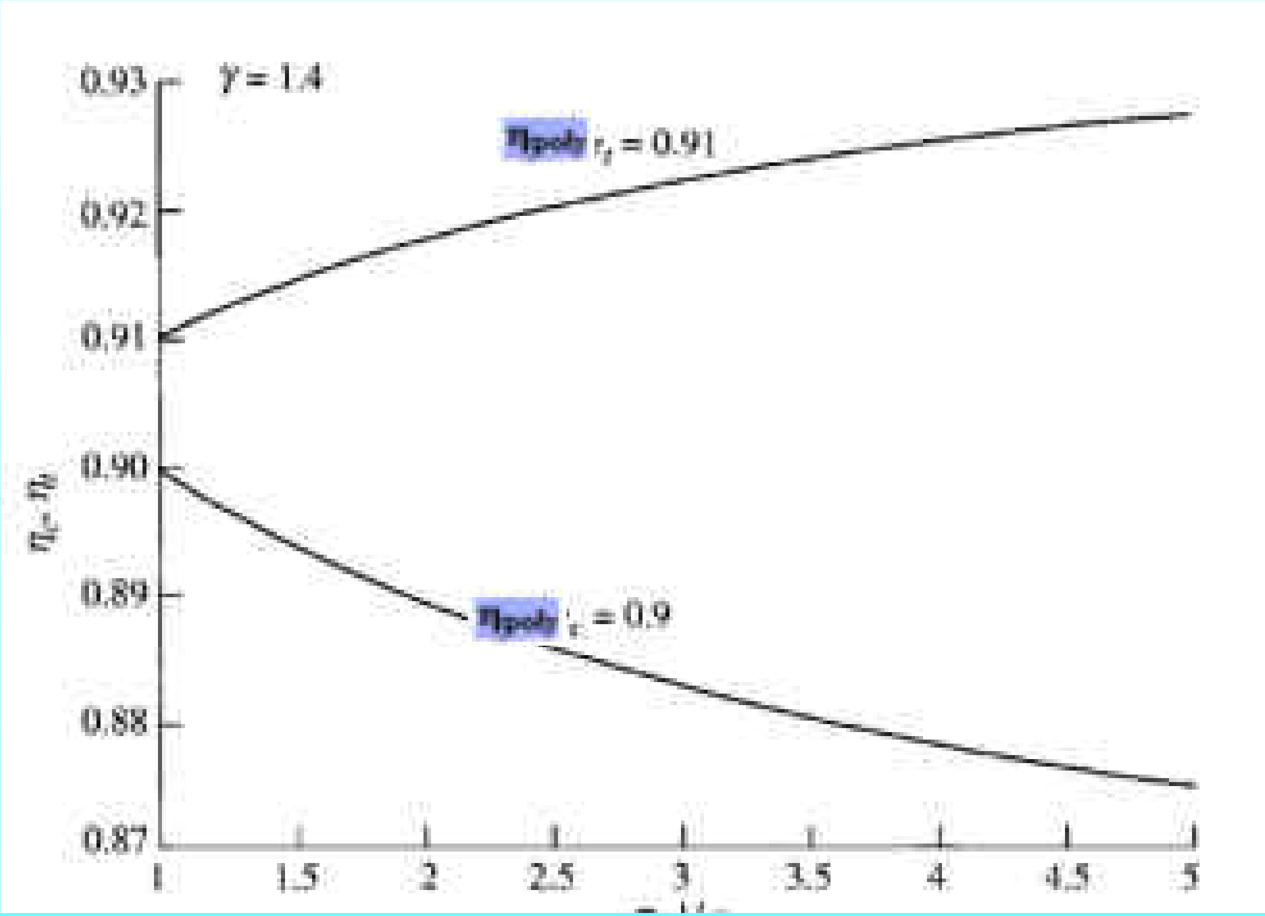
To2, Taut ← Work & To1

Determine pi_t from; →

$$\eta_t = \frac{1-\tau_t}{1-\pi_t^{(\gamma-1)/\gamma}}$$

$$\tau_t = \pi_t^{\eta_{pol}\frac{(\gamma-1)}{\gamma}}$$

Efficiency Vs Pressure Ratio for Compressors & Turbines



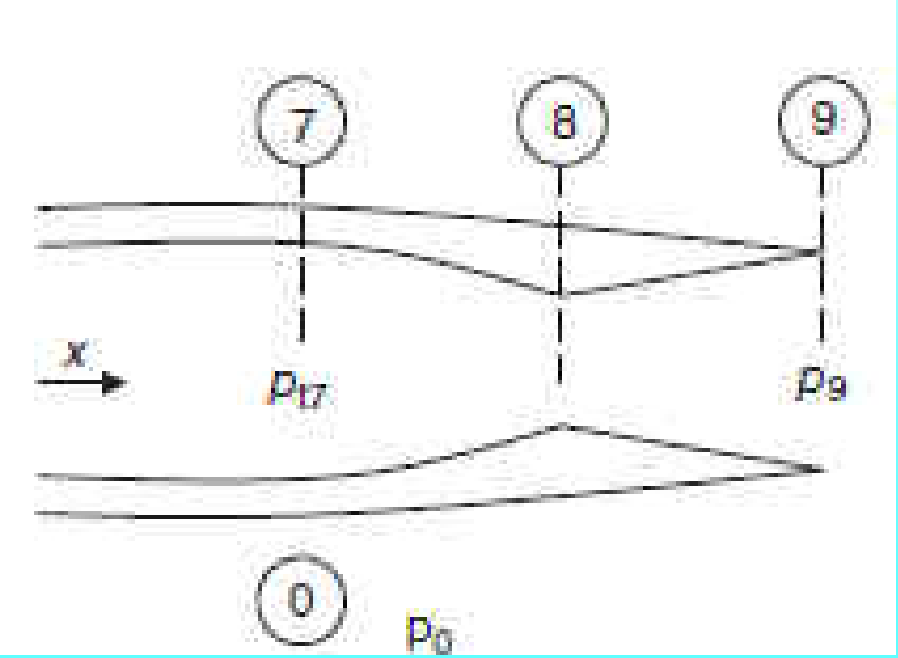
Nozzles

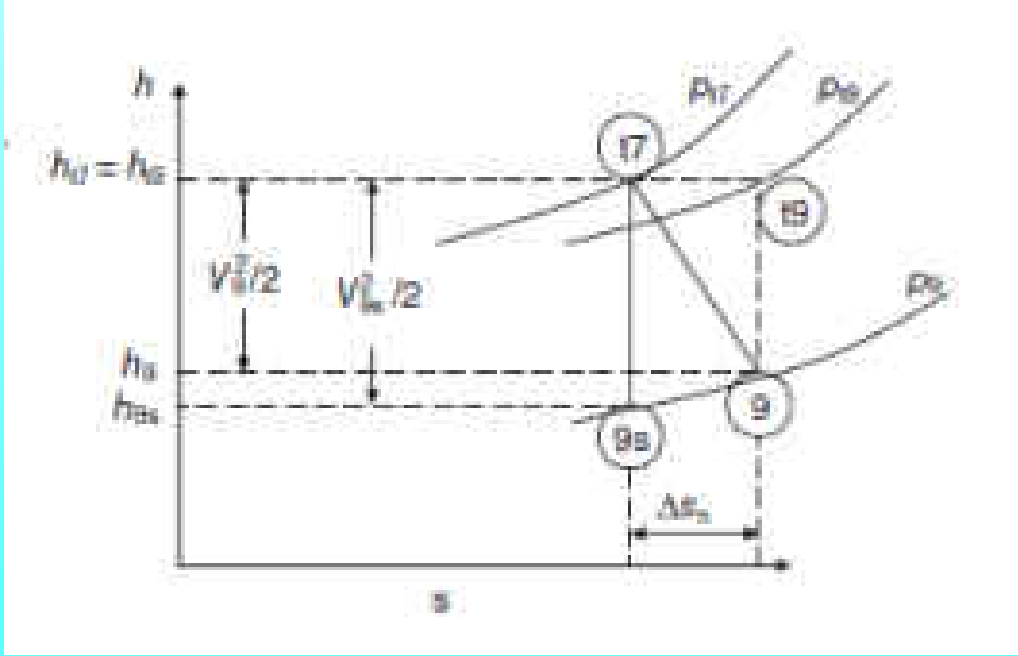
- Ideal expansion process: Isentropic
  - Stagnation pressure remains constant through the nozzle, $\pi_n = 1$
    - Inviscid flow, no flow separation
    - No wall-friction
    - No shock losses
  - Stagnation temperature remains constnat through the nozzle, $Tau_n = 1$
    - The nozzle walls are adiabatic
    - The flow is *perfectly expanded*
      - *Static pressure at the exit is the same as ambient pressure*

Real Nozzles

- Losses due to
  - Viscosity (BL, Separation,..)
  - Turbulence
  - Heat loss
  - Oblique shocks
  - Imperfect expansion
    - Under-expanded/over-expanded jet
- $\pi_n < 1$
  - The losses invariably lead to loss in stagnation pressure inside the nozzle

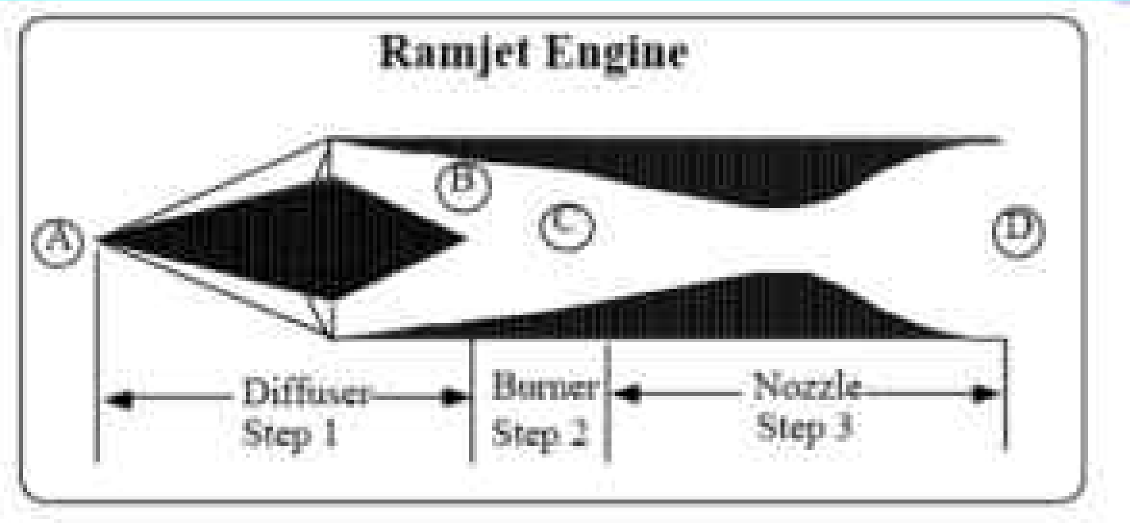
Nozzle Expansion

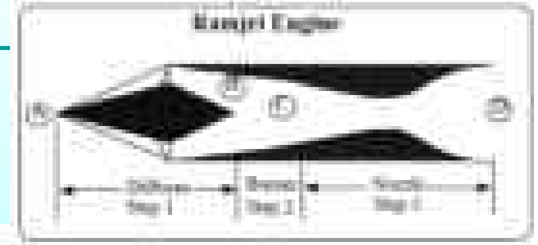




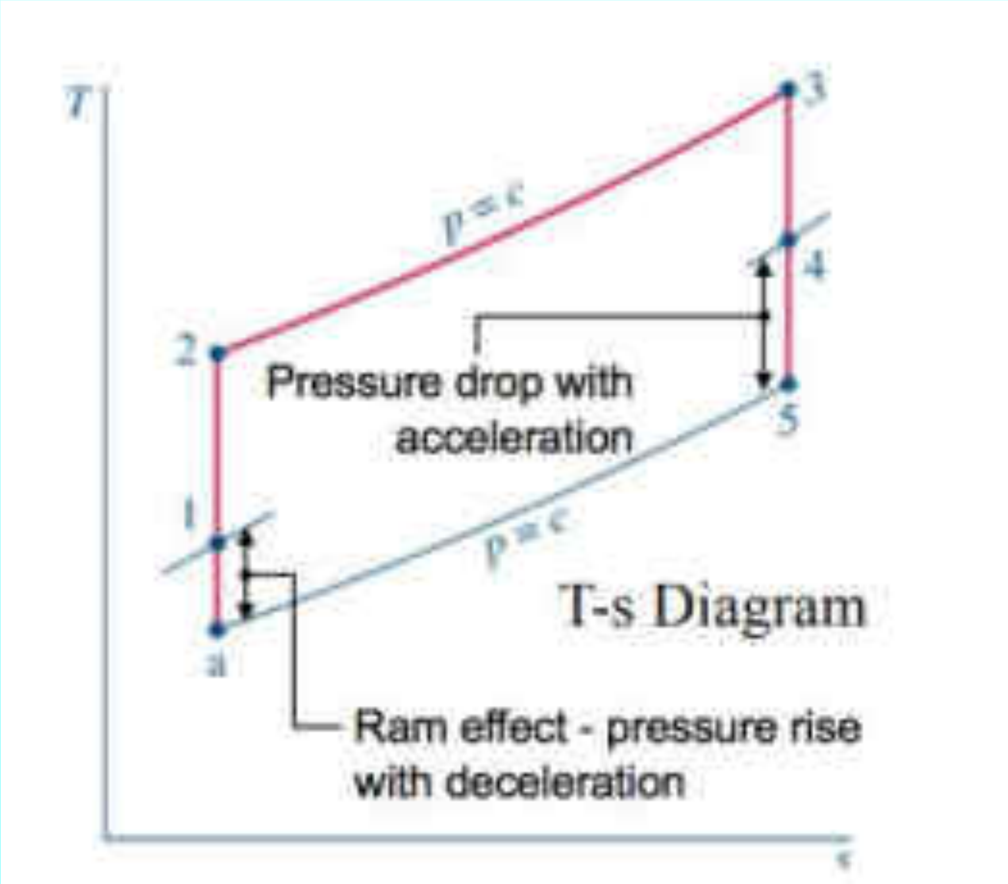
RAMJET CYCLE

- No compressor, No Turbine
- Air gets "Rammed in" by the high velocity to low velocity compression
- Entails losses





Region	Process	Ideal Behavior	Real Behavior
A to 1(inlet)	Isentropic flow	P_0,T_0 constant	P_0 drop
∞-1-2 (diffuser)	Adiabatic Compression	P,T increase P_0 drop	P_0 drop
2-3 (burner)	Heat Addition	P_0 constant, T_0	P_0 drop
3-4 (nozzle)	Isentropic expansion	T_0,P_0 constant $\Delta s > \Delta s_{rev}$	s Increase T_0 drop


T
3
p = c
4
2
Pressure drop with
acceleration
5
1
p = c
T-s Diagram
a
Ram effect - pressure rise
with deceleration
s

Ideal Ramjet *without stagnation pressure loss*

- If supersonic to subsonic deceleration is achieved through a large number of weak shocks, loss in Po may be negligible
  - Not feasible in practice
- If Po is constant & $P_e = P_a$ (Ideal expansion in nozzle): →

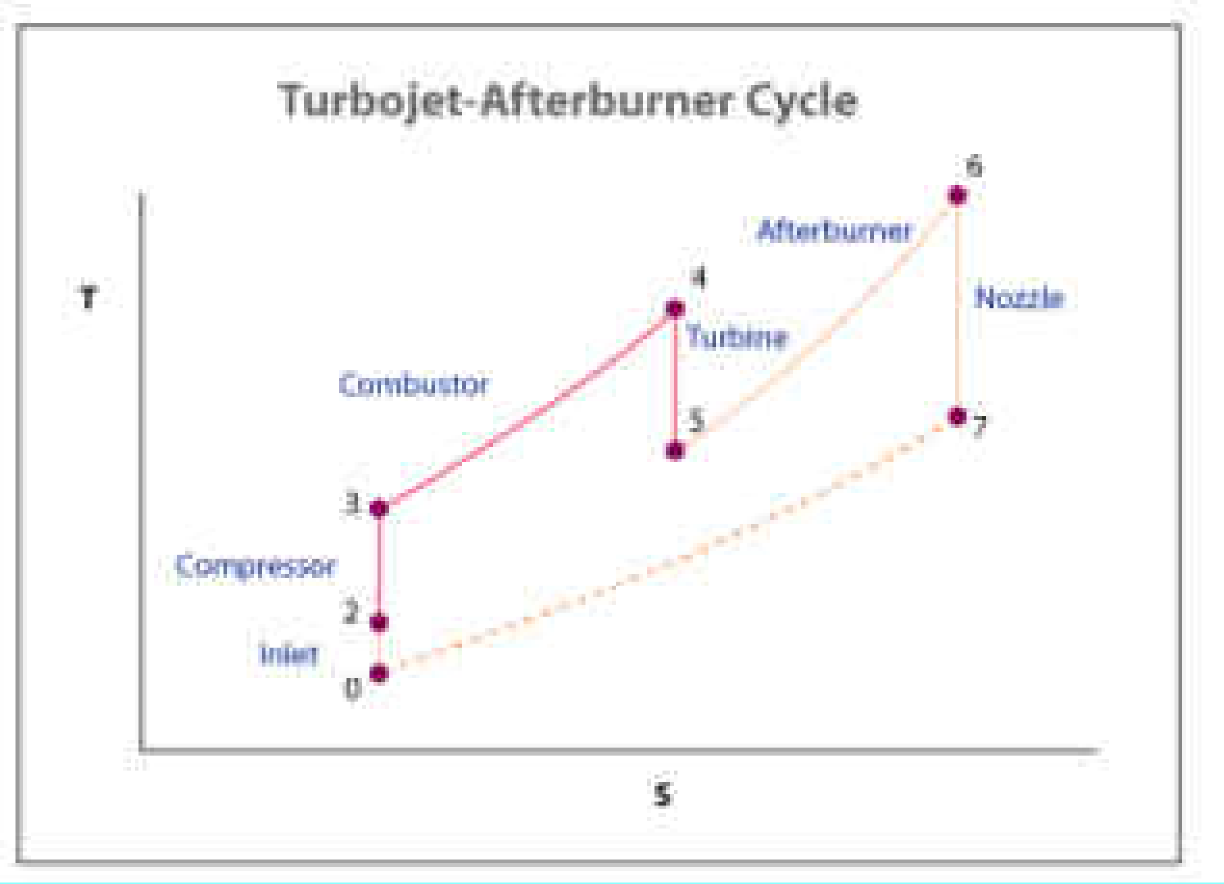
Exit Mach number = Inlet Mach number

Afterburner

- A component that provides additional thrust by burning more fuel
  - After the gases exit from the trubine
- Makes use of the excess air in combustion gases
  - Additional fuel is injected into the gas stream
- Serves for momentary increase in thrust
- Low thermal efficiency, not economic for continued operation

Ideal TS Diagram with Afterburner

- Note the afterburner combustion process after the turbine



Aerospace Propulsion

Dr. A.R. Srikrishnan
Department of Aerospace engineering
2022

07
Fundamentals of Rocket Propulsion


INDIA
PSLV

Missions..

- **Space Launch Vehicles: 1957-**
  - Type of Propellant:
    - *Storable Or Cryogenic*
    - *Liquid Or Solid*
  - *Number of Stages*
    - *1,2,3..*
    - *Single stage only for low earth orbits (Say, 150 km..)*
  - *Reuse:*
    - *Expendable or Recoverable*
  - *Mission*
    - *Manned or Unmanned*
- *Missiles*

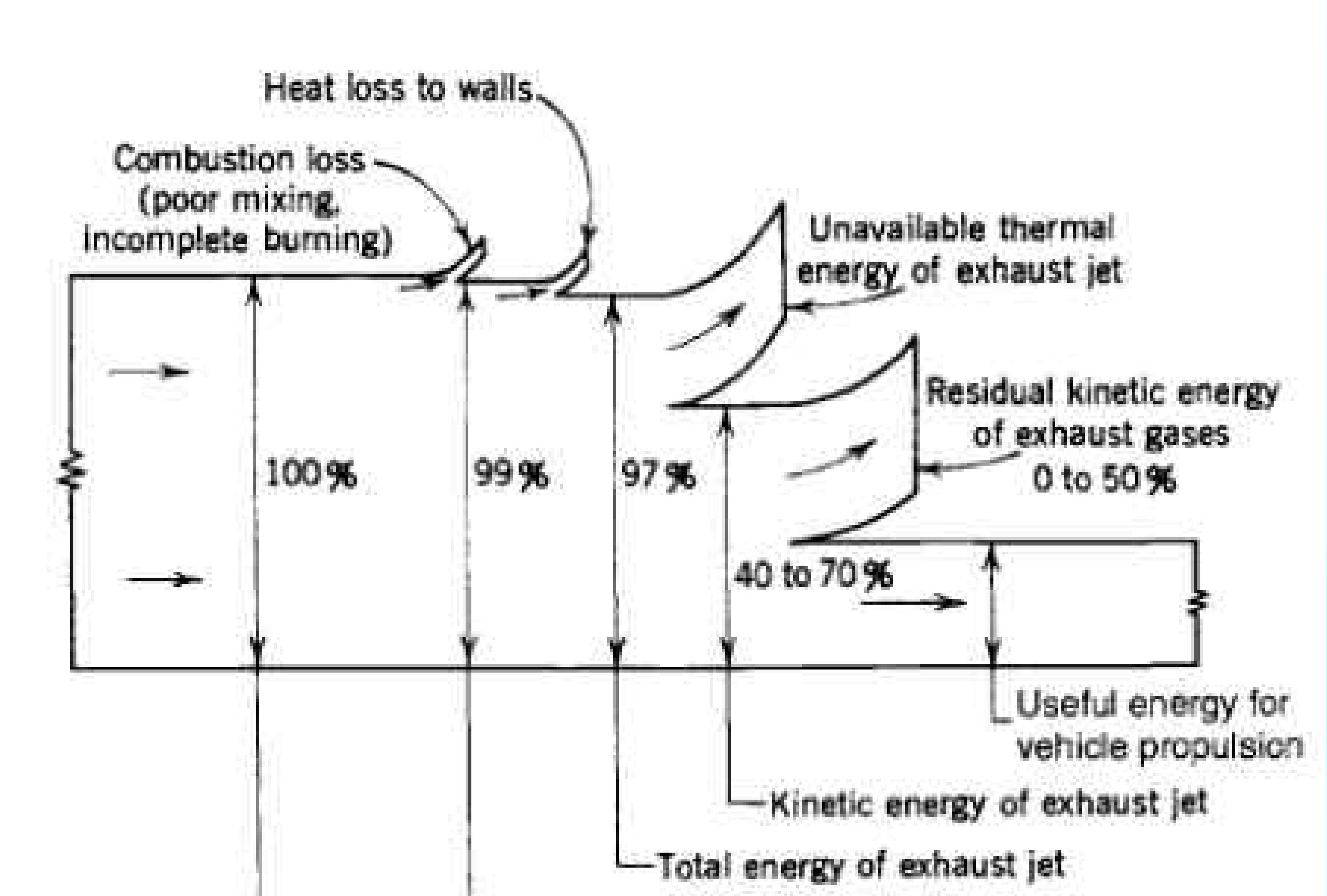
Specific Impulse

- Thrust = *Momentum thrust + Pressure thrust*
- Specific Impulse:
  - *For constant thrust & constant propellant flow rate, ISP = Thrust/Propellant mass flow rate = F/m·*
  - INTEGRATE, IF THEY VARY WITH TIME
- Mass flow or Weight Flow ?
  - *"SEconds" – not the elapsed time…*
- PROPELLANT MASS FRACTION = $M_p/M_o = M_p/(M_p+M_{final})$

Rocket Vs AB Propulsion

Feature	Rocket Engine or Rocket Motor	Turbojet Engine	Ramjet Engine
Thrust-to-weight ratio, typical	75:1	5:1, turbojet and afterburner	7:1 at Mach 3 at 30,000 ft
Specific fuel consumption (pounds of propellant or fuel per hour per pound of thrust)[a]	8–14	0.5–1.5	2.3–3.5
Specific thrust (pounds of thrust per square foot frontal area)[b]	5000 to 25,000	2500 (Low Mach at sea level)	2700 (Mach 2 at sea level)
Thrust change with altitude	Slight increase	Decreases	Decreases
Thrust vs. flight speed	Nearly constant	Increases with speed	Increases with speed
Thrust vs. air temperature	Constant	Decreases with temperature	Decreases with temperature
Flight speed vs. exhaust velocity	Unrelated, flight speed can be greater	Flight speed always less than exhaust velocity	Flight speed always less than exhaust velocity
Altitude limitation	None; suited to space travel	14,000–17,000 m	20,000 m at Mach 3 30,000 m at Mach 5 45,000 m at Mach 12
Specific impulse typical[c] (thrust force per unit propellant or fuel weight flow per second)	270 sec	1600 sec	1400 sec

The Energy Balance



Energy Limited Vs *Power Limited*

- Rockets: Develop thrust by imparting energy & momentum to the propellant being expelled
- Source of Energy: Chemical, Nuclear or Solar
  - Chemical Rockets: Energy released during combustion of propellant (Fuel + Oxidizer)

    Chemical Energy → Propellant KE

    - Limited by the Energy contained in the fuel
  - Electrical Rockets:

Nuclear/Solar Energy → Electrical Energy → Propellant KE

    - Nuclear Rockets: Heating (electrical) of the working fluid is done by energy from nuclear reactions
  - Limited by the "Rate of Conversion" of Energy (Power)

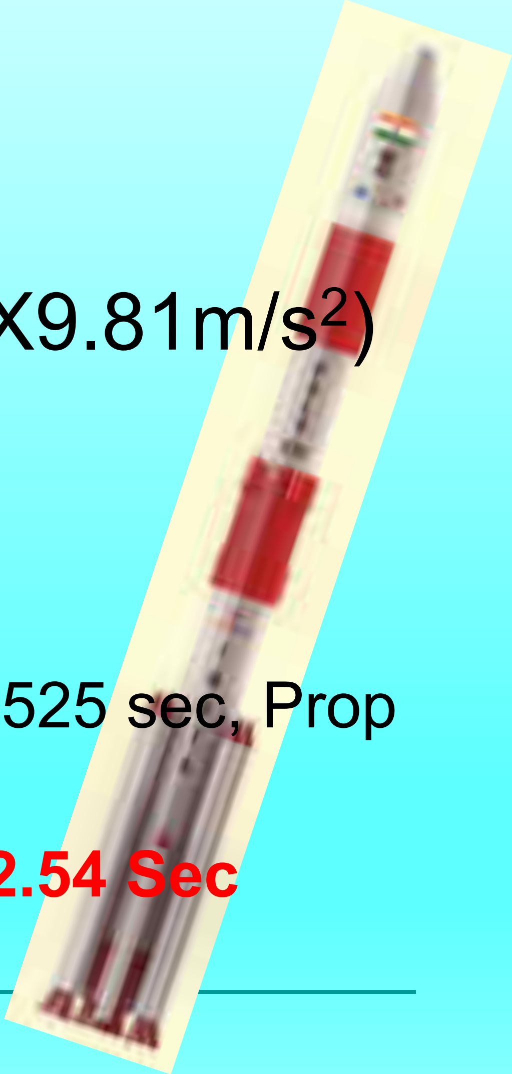
PSLV C-25: *Specific Impulse of Stage 1*

- Power Limited or Energy Limited ?
- Specific Impulse for Initial Stage:

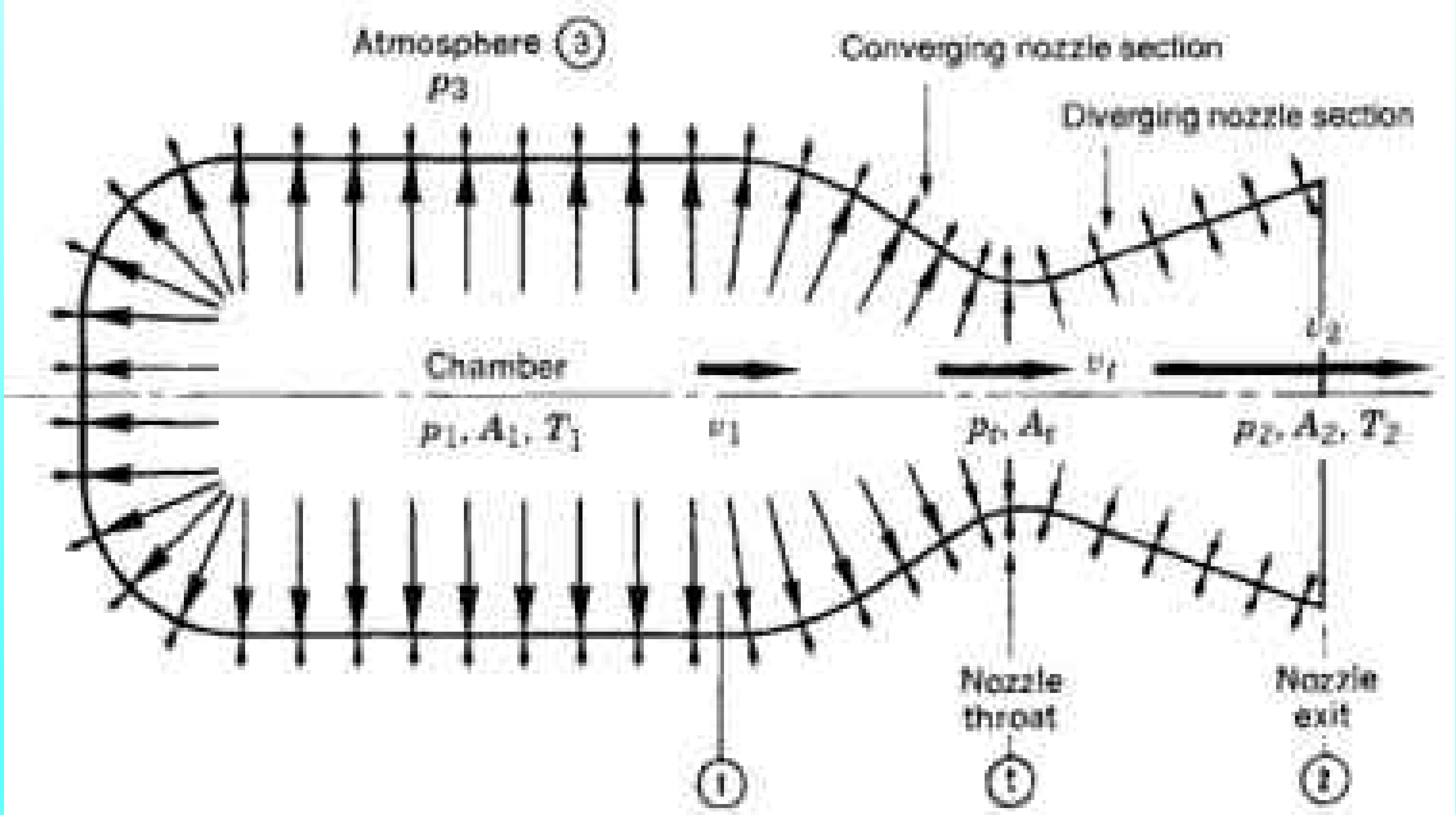
$= 4800 \text{ X } 10^3\text{N} /((138\text{X } 10^3\text{kg}/103\text{s})\text{X}9.81\text{m/s}^2)$

= 365 s

- Stage 4: Peak Thrust = 14.6 kN, Burn time 525 sec, Prop Mass = 2500 kg
- Peak ISP = $14.6*10^3/(9.81*2500/525)$ = **312.54 Sec**



The Force Imbalance



P_3 = P_a = ZERO in space

Rocket Thrust

$$\mathbf{T} = -\iint_{A_e} (\rho \mathbf{u})\mathbf{u} \cdot d\mathbf{s} - \iint_{A_e} (p - p_a)\, d\mathbf{s}$$

- In on-dimensional form:

$$T = \dot{m} u_e + (p_e - p_a) A_e$$

- Effective Velocity defined such that: T = m.V_{eff}
- *Or T = m.C where C is the effective velocity*

Effective Exhaust Velocity

- $Thrust\ F = \dot{m}_p V_e + (P_e - P_a)A_e$

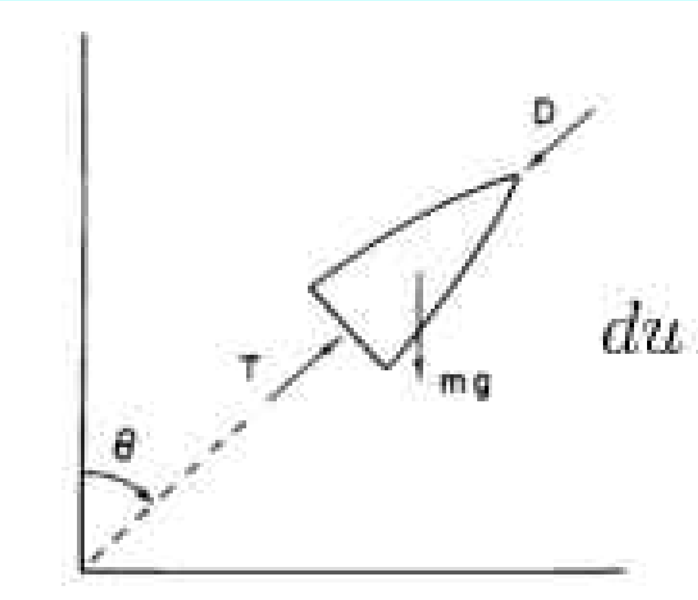
$$= \dot{m}_p V_{eff}$$

where $V_{eff} = \boldsymbol{V_e} + \frac{(P_e - P_a)A_e}{\dot{m}_p}$

Rocket Equation

- Forces on Rocket:
  - In the direction of flight:

$$T - D - g m_v \cos\theta = m_v \frac{du}{dt}$$



As $T = u_e \frac{dm_v}{dt}$

$$du = -\frac{u_e dm_v}{m_v} - \frac{D}{m_v}dt - g\cos\theta dt.$$

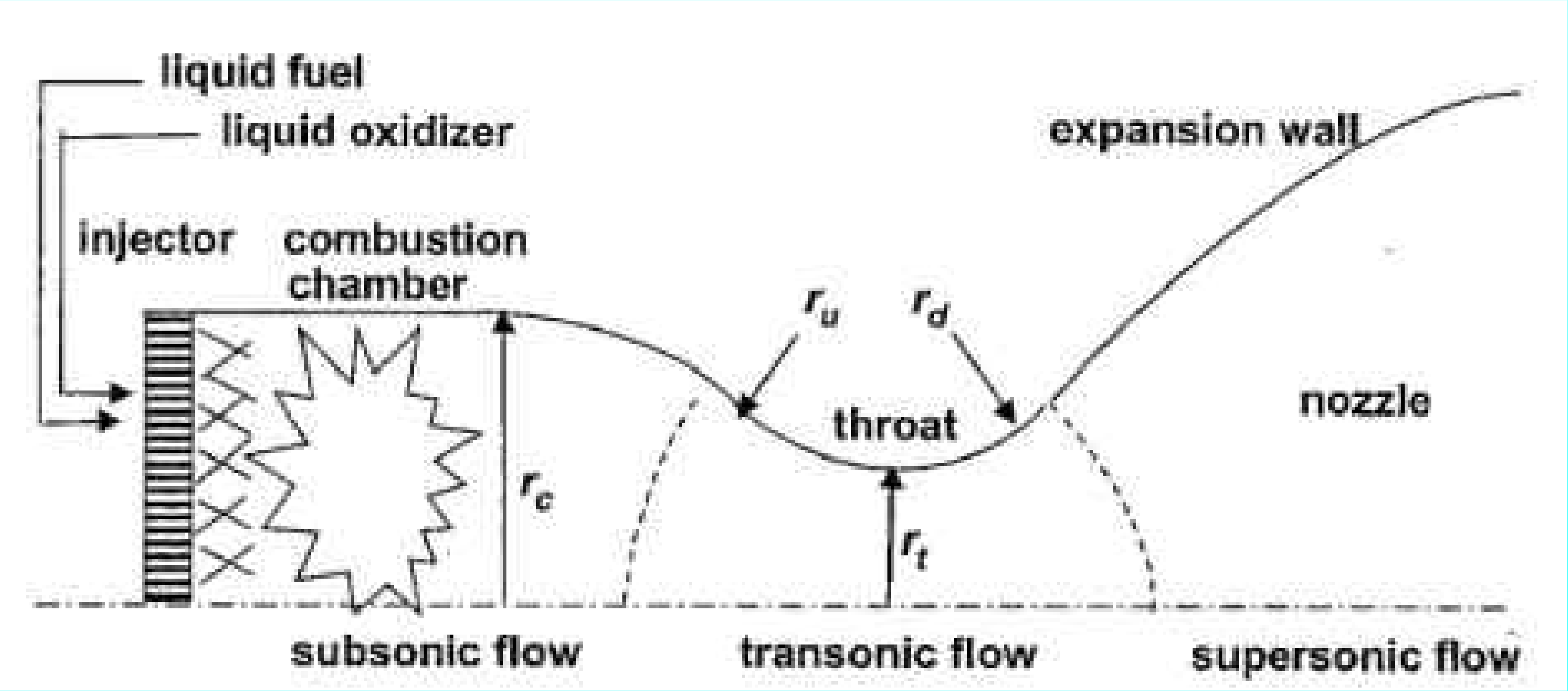
- In the absence of gravity and drag:

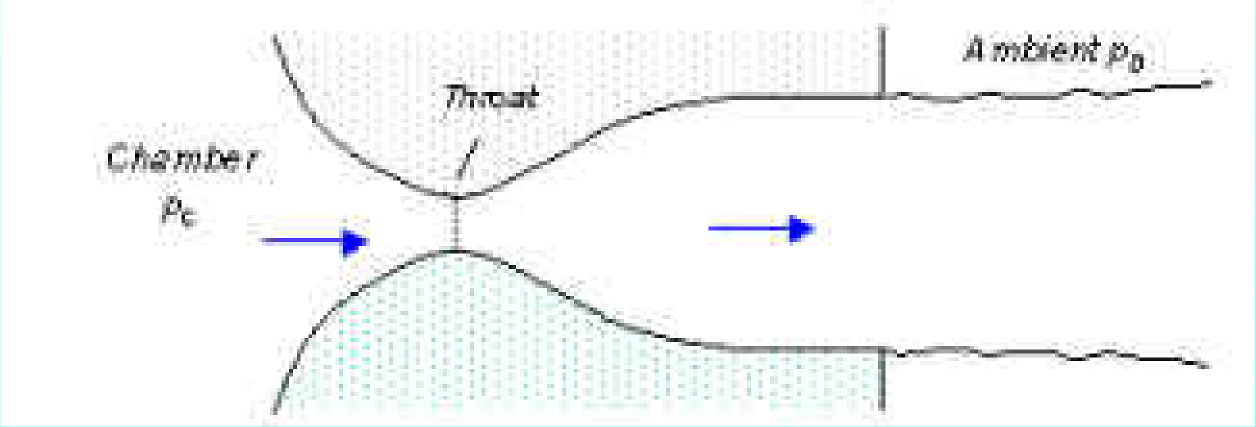
$$MR \equiv \frac{m_0}{m_0 - m_p} = e^{\Delta v / c}$$

- In the absence of gravity and drag:

$$MR \equiv \frac{m_0}{m_0 - m_p} = e^{\Delta v / c}$$

Thrust Generation





- Ideal Rocket: Perfect expansion in the nozzle
  - Stagnation temperature (and stagnation pressure remain constant through the nozzle)
  - Thrust Calculation

- Calculate Specific Impulse for an ideal rocket operating with a chamber pressure of 68 atm at sea level when the thrust chamber temperature is 1700 K. Use $\gamma = 1.3$ & Cp=1250 J/kg-K

  - *Ideal rocket implies, stagnation pressure is constant, Stagnation temperature in nozzle is constant & the expansion is perfect (Pe=Pa)*

Pay Load, Propellant and Structure

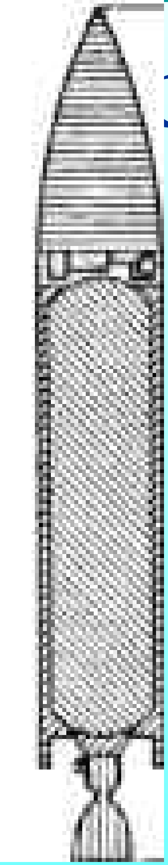
- Total mass of rocket, M_o, constitutes of
  - Payload mass, M_L
  - Propellant mass, M_P
  - Structural mass, M_S
    - Includes everything but payload and propellant
    - Engines, tanks, controls, etc.

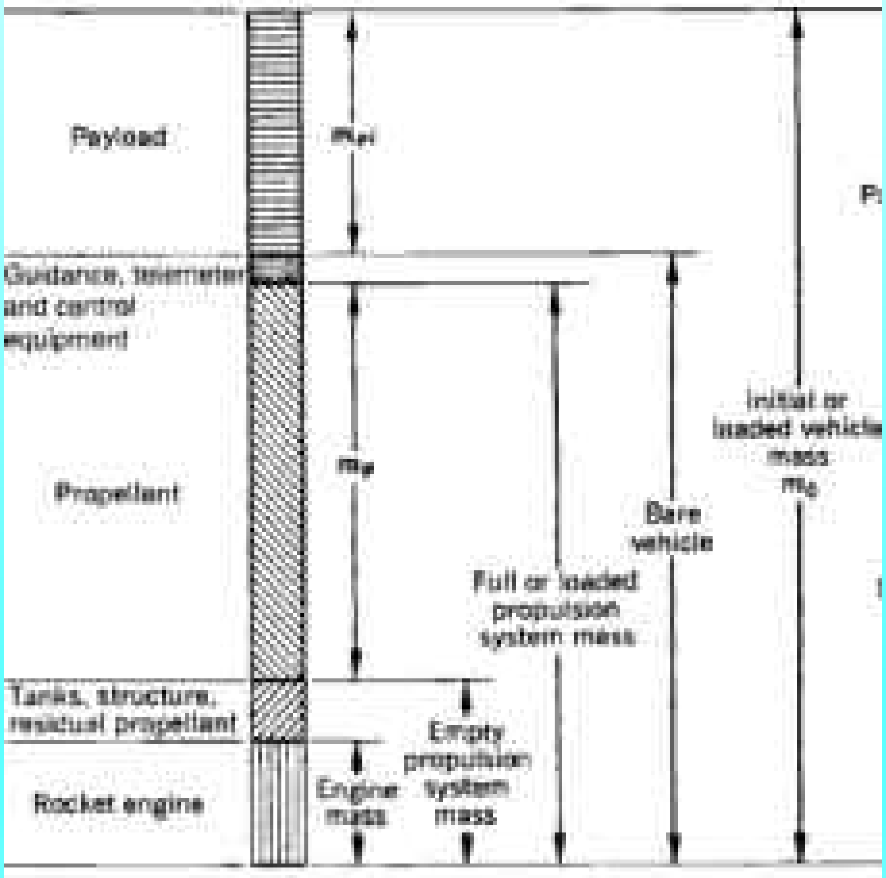
$$M_o = M_L + M_P + M_S$$

- If rocket consumes all its propellant during firing, burnout mass consists of structure and payload:

$$M_b = M_L + M_S$$

ocket Mass: Constituents





Mass Ratio, Payload Ratio, Structural Coefficient

- Mass Ratio: Initial Mass/Mass at burn out → *Should be high*
- *Pay load ratio: Mass of pay load/total mass of other constituents → Should be high*
- *Structural Coefficient: Structural … est other than payload*

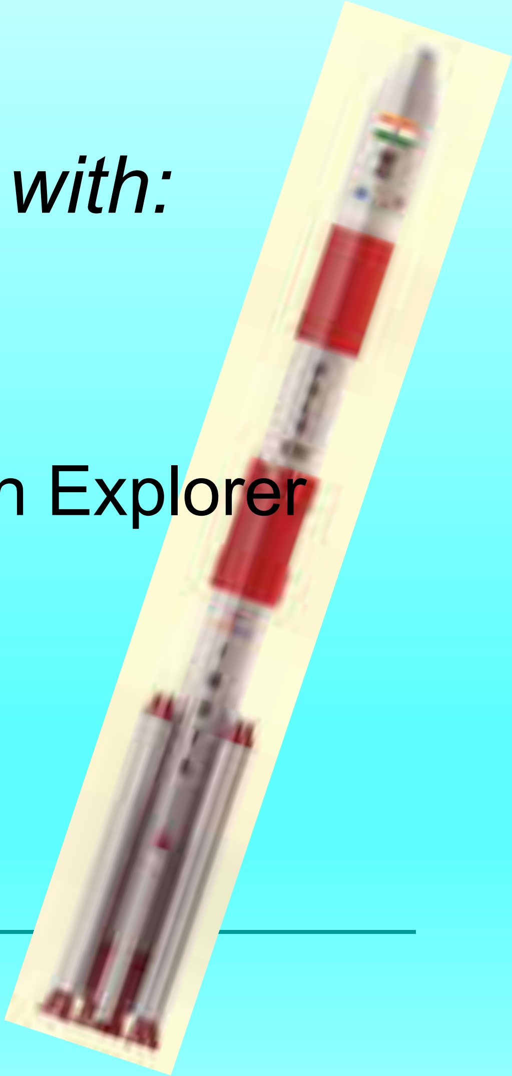
$$R = \frac{M_o}{M_b} = \frac{M_o}{M_L + M_S}$$

$$\lambda = \frac{M_L}{M_o - M_L} = \frac{M_L}{M_P + M_S}$$

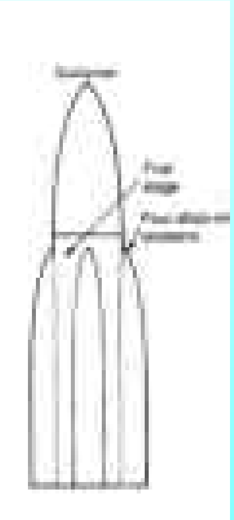
$$\varepsilon = \frac{M_S}{M_P + M_S} = \frac{M_b - M_L}{M_o - M_L}$$

PSLV C25

- The Pay Load: *1340kg: Space Craft with:*
  - Lyman Alpha Photometer (LAP)
  - Methane Sensor For Mars (MSM)
  - Martian Exospheric Neutral Composition Explorer (MENCA)
  - Mars Colour Camera (MCC)
  - TIR Imaging Spectrometer (TIS)

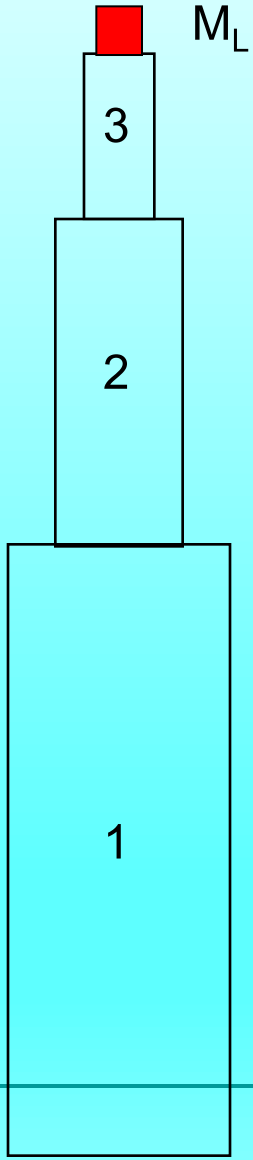


Staging of Rockets



- Discard empty tanks and extra structure as rocket travels
  - Minimize losses due to gravity
- Multistage rocket is a series of individual vehicles or stages, each with its own structure, tanks and engines
  - Typically 2-6 stages
- Launch vehicles have two to six *strap-on solid propellant motor boosters*
  - *which together form a supplementary first stage strapped on or*
  - mounted to the first stage of the launch vehicle

MULTISTAGE ROCKET EXAMPLE



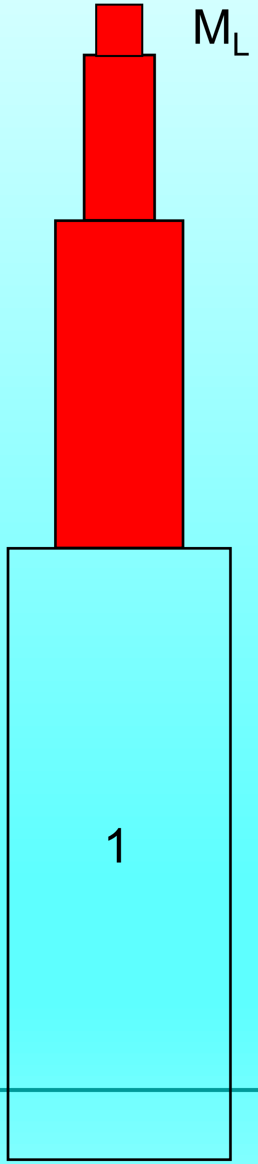
Total Mass 3: $M_{o3}=M_{P3}+M_{S3}+M_L$
Payload for Stage 3: $M_{L3}=M_L$

Total Mass 2: $M_{o2}=M_{P2}+M_{S2}+M_{o3}$
Payload for Stage 2: $M_{L2}=M_{o3}$

Total Mass 1: $M_{o1}=M_{P1}+M_{S1}+M_{o2}$
Payload for Stage 1: $M_{L1}=M_{o2}$

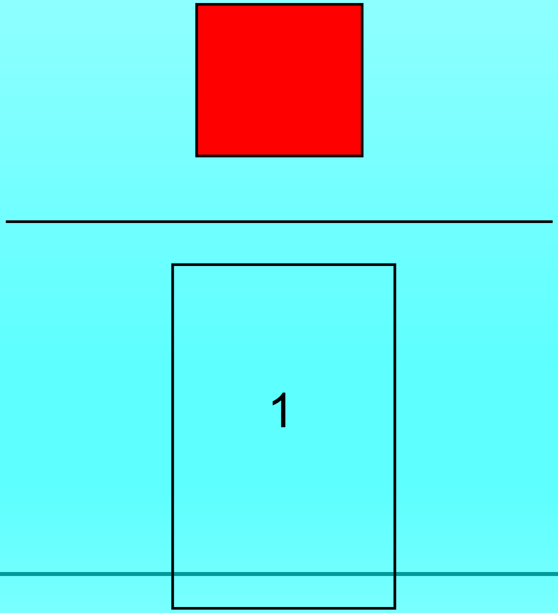
Total Mass i: $M_{oi}=M_{Pi}+M_{Si}+M_{o(i+1)}$
Payload for Stage i: $M_{Li}=M_{o(i+1)}$

PAYLOAD RATIO: MULTISTAGE ROCKETS



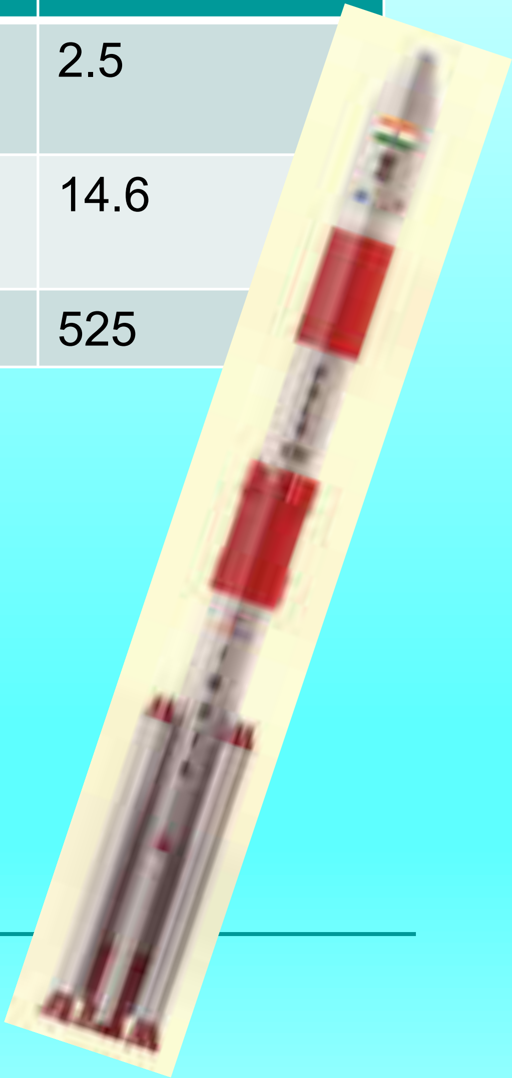
$$\lambda_i = \frac{M_L}{M_o - M_L} = \frac{M_{o(i+1)}}{M_{oi} - M_{o(i+1)}} = \frac{M_{o2}}{M_{o1} - M_{o2}}$$

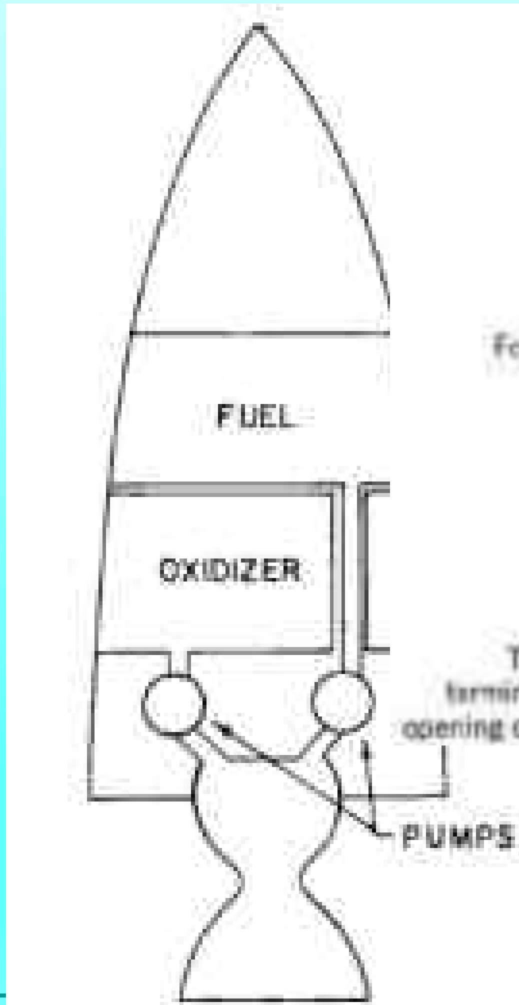
The payload ratio for stage 1 is:

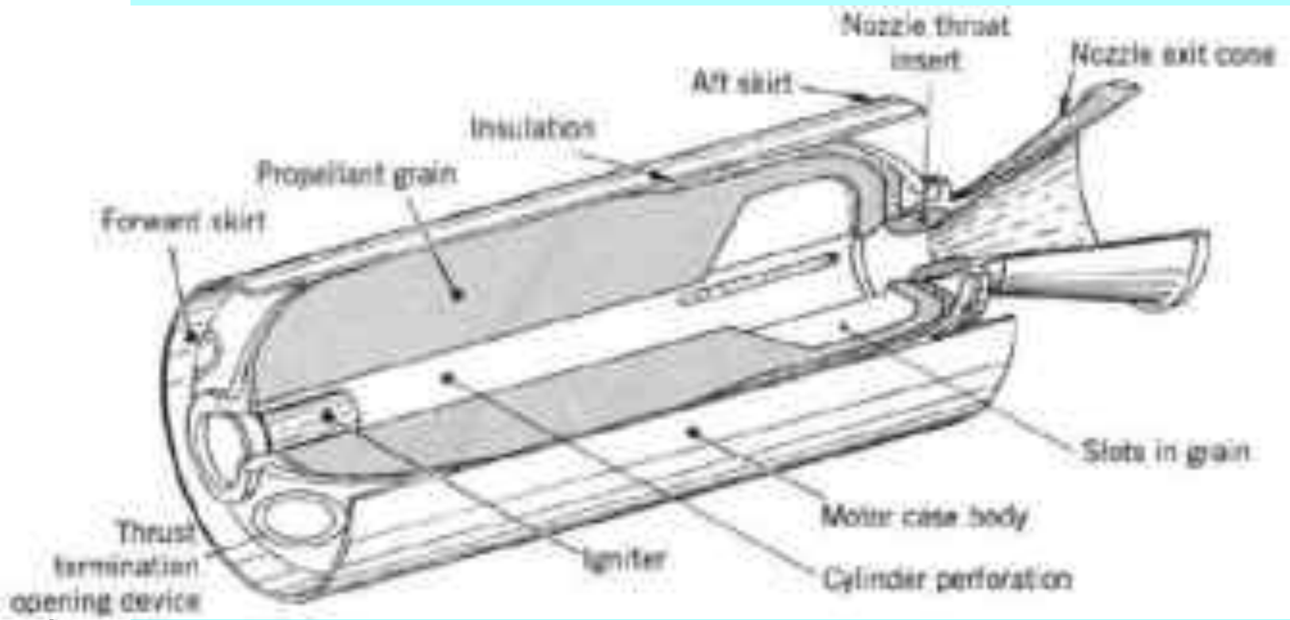


PSLV~C25

	STAGE 1	STAGE 2	STAGE 3	STAGE 4
Propellant Mass, Tonnes	138	42	7.6	2.5
Peak Thrust, KN	4800	799	247	14.6
Burn Time, S	103	148	112	525




FUEL
OXIDIZER
PUMPS


Nozzle throat insert
Nozzle exit cone
Aft skirt
Insulation
Propellant grain
Forward skirt
Slots in grain
Motor case body
Thrust termination opening device
Igniter
Cylinder perforation

Liquid Propellant Engines

- *Liquid propellants that are fed under* pressure from tanks into a thrust chamber.*
  - The liquid *bipropellant consists of a liquid oxidizer (e.g., liquid oxygen) and a liquid fuel* (e.g., kerosene). A *monopropellant is a single liquid that contains both oxidizing* and fuel species
- In the *thrust chamber the propellants react to form hot gases, which in turn* are accelerated and ejected at a high velocity through a supersonic nozzle
- A liquid rocket propulsion system requires several precision valves and a complex feed mechanism which includes propellant pumps, propellant-pressurizing device, and a relatively intricate combustion or thrust chamber.

Liquid Propellants

- *Working Fluids* of rocket engines
  - Oxidizer
  - Fuel
  - A mixture of oxidizer & Fuel
- *Monopropellant:*
  - *Contains an oxidizing agent and combustible matter in the same fluid*
  - *Ex: H2O2, hydrazine*
- *Bipropellant:*
  - *Two separate liquid propellants: Stored separately*
  - *Oxidizer & Fuel*
  - *Mixed in the combustion chamber*

Monopropellants

- Less complicated
  - Only need 1 propellant
  - Only need 1 set of drive turbines and pumps
  - Less plumbing
- Limitations: Low thrust, Low ISP

Bipropellants

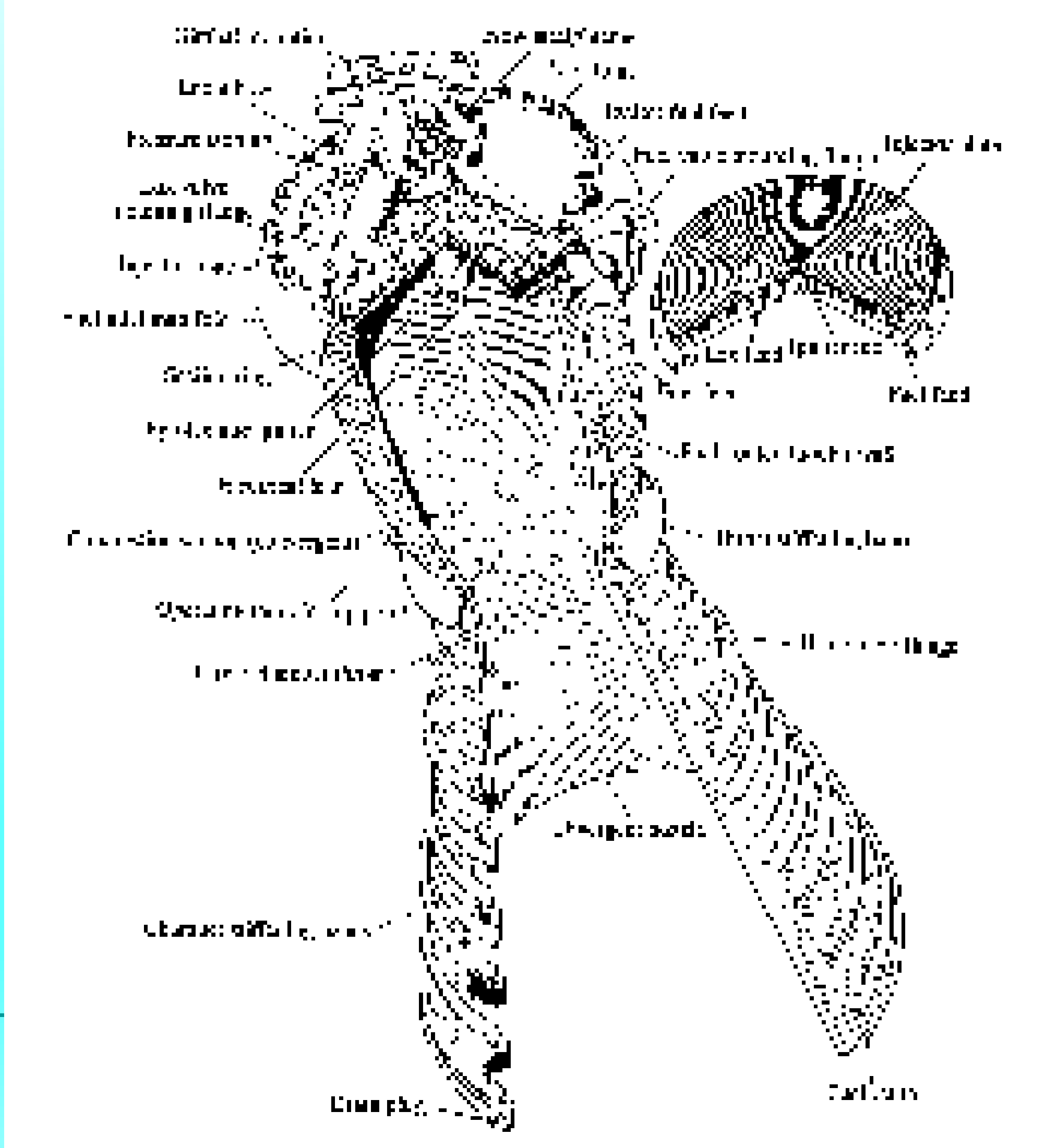
- Most liquid fuel rockets are bipropellant
  - Ex: Mixtures of liquid oxygen (LO_2) with hydrocarbons such as kerosene and RP1 or with liquid hydrogen (LH_2) are most typical
- Produce enormous thrust due to massive amount of propellant used

COMMON LIQUID ROCKET FUELS

Oxidizer	Fuel	Specific Impulse (maximum)
Liquid Oxygen	Liquid Hydrogen	391
"	RP-1	300
"	Ammonia	296
"	95% Ethyl Alcohol	287
"	Hydrazine	313
"	50% UDMH 50% Hydrazine	312
"	UDMH	310
Liquid Fluorine	Liquid Hydrogen	410
"	Hydrazine	363
"	Ammonia	357
Nitrogen Tetroxide	Hydrazine	292
"	50% UDMH 50% Hydrazine	288
"	UDMH	285
"	RP-1	276
"	92.5% Ethyl Alcohol	267
95% Hydrogen Peroxide	Hydrazine	285
"	50% UDMH 50% Hydrazine	279
"	UDMH	278
"	RP-1	273
NONE (Monopropellant)	Hydrogen Peroxide	140
NONE (Monopropellant)	Hydrazine	205
NONE (Monopropellant)	Nitromethane	180
NONE (Monopropellant)	Methylacetylene	160

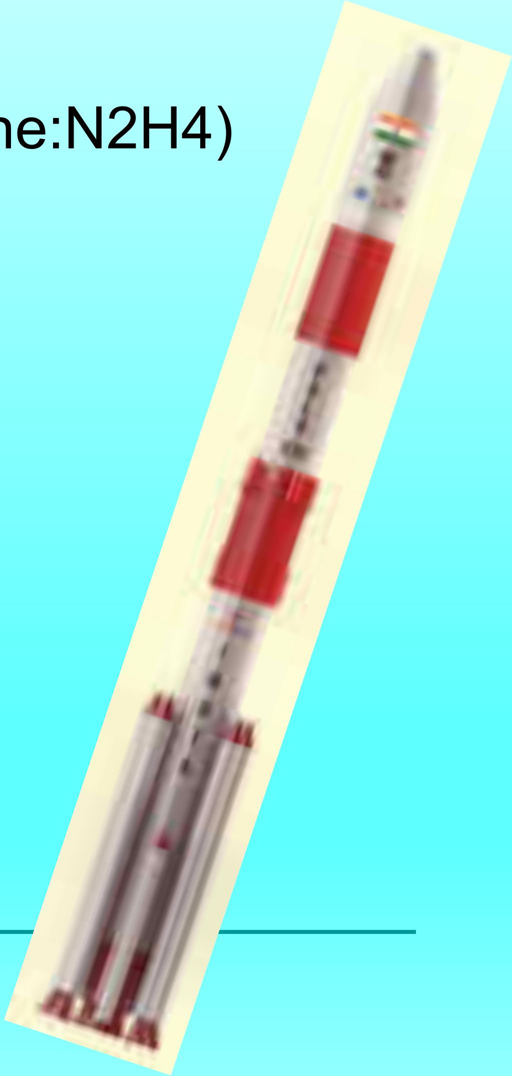
- Most Common Liquid Oxidizers
  - LOX
  - Hydrogen Peroxide
  - Nitric Acid
  - Nitrogen Tetroxide
  - Liquid Florine
- Most Common Liquid Fuels
  - Hydrocarbon fuels (RP1, kerosense, methane)
  - Liquid hydrogen
  - Hydrazine (also mono)
  - Unsymmetrical Dimethylhydrazine

BI-PROPELLANT LIQUID ROCKET



PSLV C25: Liquid Propellants

- Stage 2: UDMH + N_2O_4
  Unsymmetrical dimethylhydrazine (hydrazine:N_2H_4)
- Stage 4: MMH+MON3

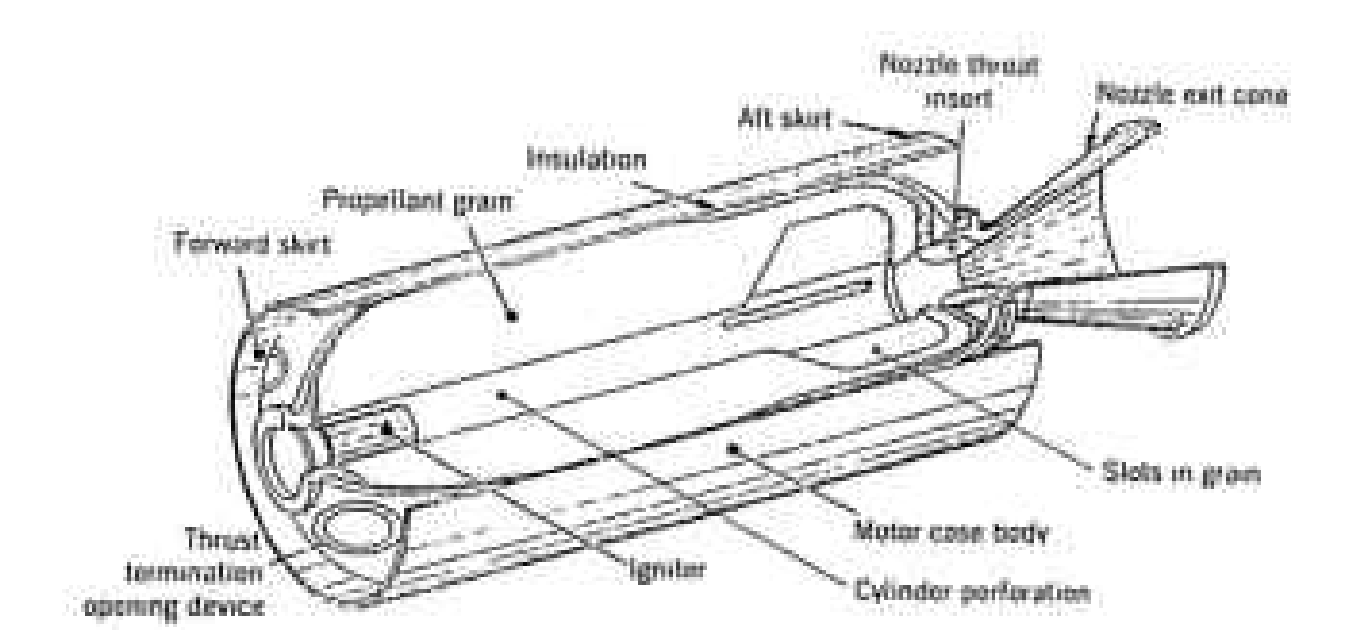


Solid Propellant “GRAIN”

- *The propellant to be burned is contained* within the combustion chamber or *case*
  - *The “Grain”*
  - *Contains all the chemical elements for complete burning.*
  - *Once* ignited, burns at a predetermined rate on all the exposed internal surfaces of the grain
- The resulting hot gas flows through the supersonic nozzle to impart thrust
  - Once ignited, the motor combustion proceedsin an orderly manner until essentially all the propellant has been consumed.

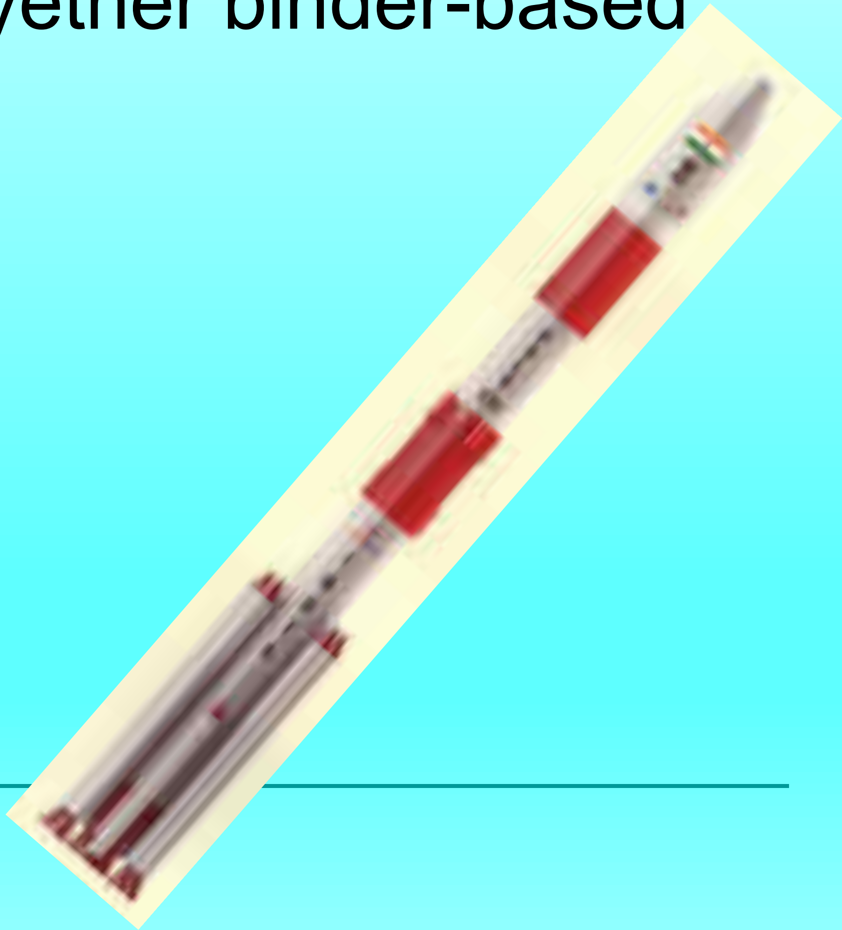
SOLID PROPELLANT ROCKETS

- Rely on controlled explosion of a mixture of substances
  - Nearly a homogeneous material that is burned
- Similar to gunpowder ~(75% potassium nitrate, 10% carbon, and 15% sulfur)
- Sample Composition of a Solid Rocket Booster:
  - Ammonium perchlorate as an oxidizer and aluminum as a fuel
  - Rest of mixture devoted to bonding two reactants
- Once a solid rocket is ignited, can not be turned off

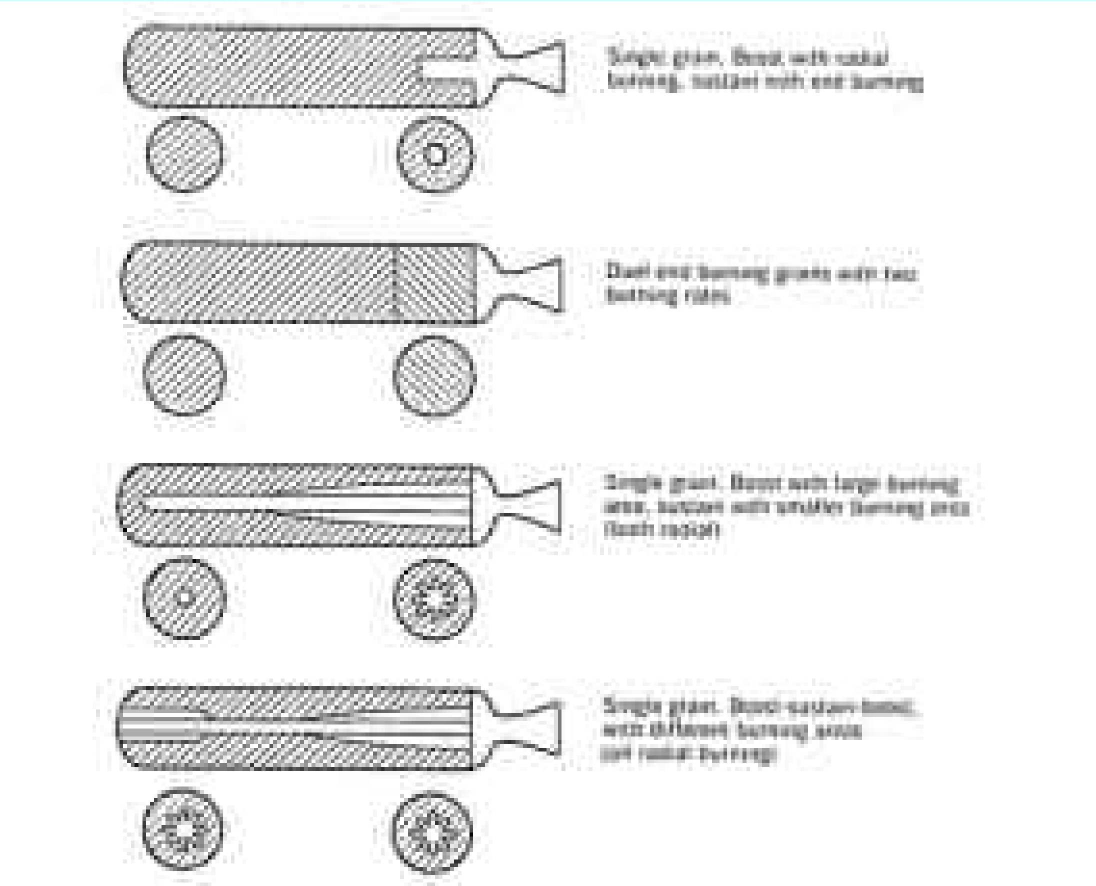

Nozzle throat insert
Nozzle exit cone
Aft skirt
Insulation
Propellant grain
Forward skirt
Slots in grain
Motor case body
Thrust termination opening device
Igniter
Cylinder perforation

PSLV C

- Stages 1 & 3 Use Solid Propellant
- Hydroxy-terminated polyether binder-based



SOLID PROPELLANT ROCKETS

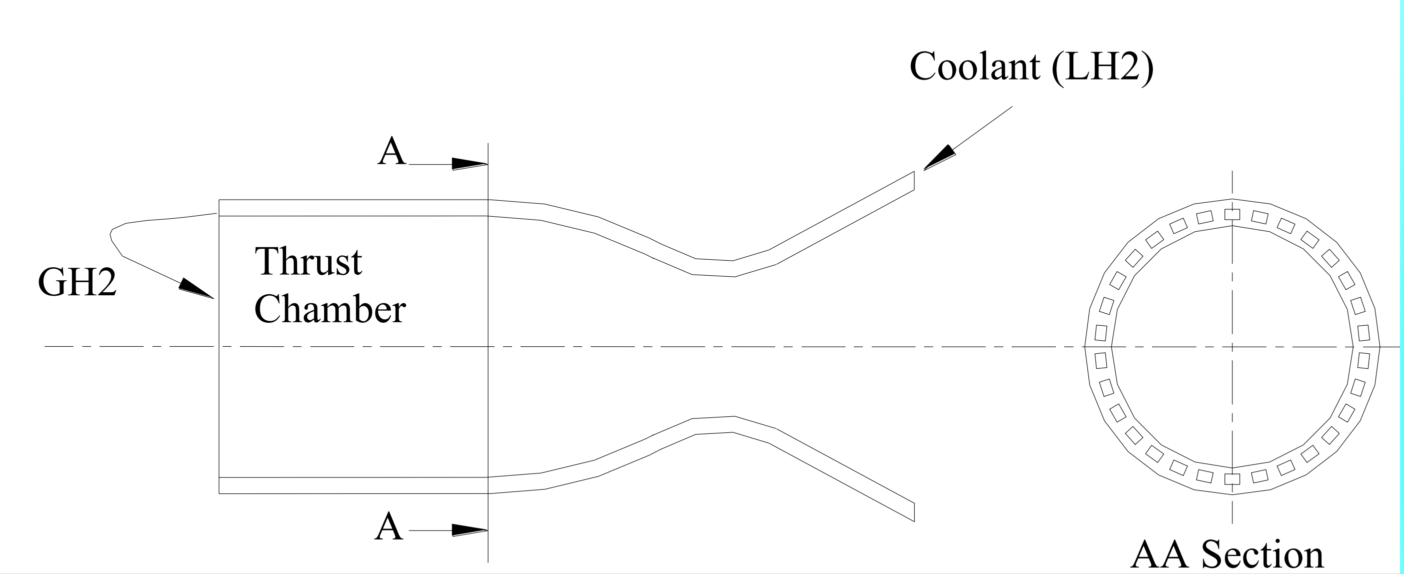


Rocket Heat Transfer: *Cooling Methods*

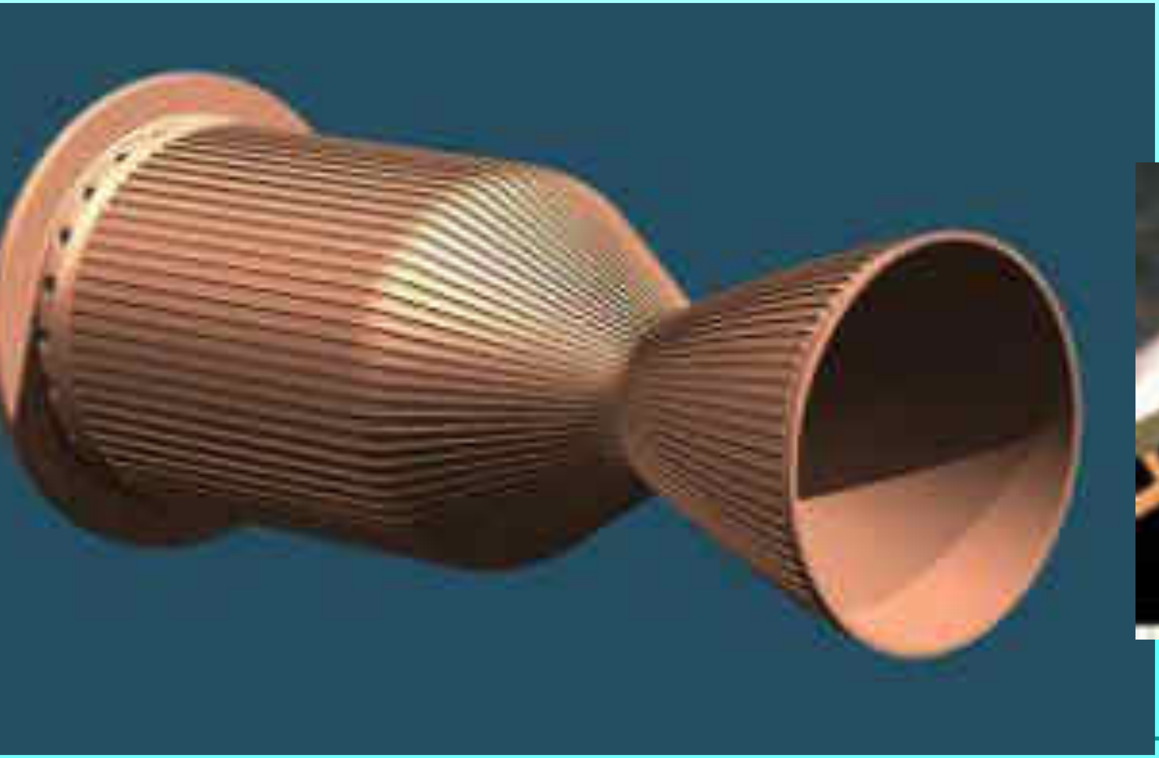
- The combustion temperature can be in the range of 2400 to 3200+ K
- Cooling required to ensure integrity of chamber and nozzle walls
- Liquid propellant rockets: The fuel/oxidizer used as coolant
- Solid propellant rockets: Provide a protective layer around nozzle walls

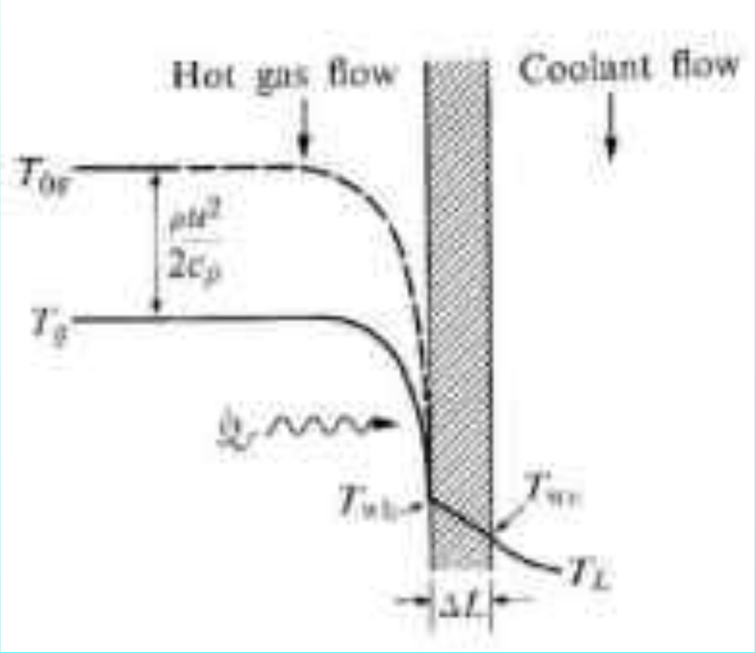
Regenerative Cooling

- Relies on convective heat transfer
- Fuel/Oxidizer flows through tubes outside the chamber & nozzle walls



Regenerative Cooling Channels





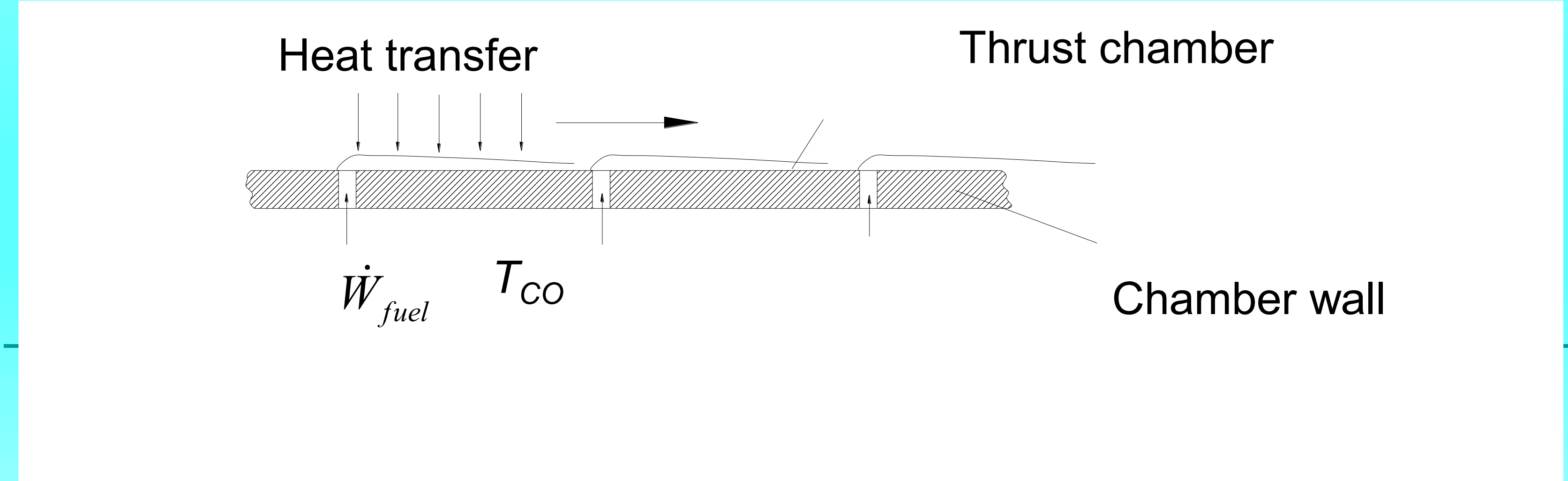
Hot gas flow
Coolant flow
T_L
ΔL

Regenerative Cooling

- Typically the liquid fuel is used as coolant
  - More often than the liquid oxidizer
- The coolant should have
  - Sufficiently high thermal capacity to absorb the energy from the walls, without undergoing chemical decomposition
  - Sufficiently high boiling point so that it does not undergo phase change during the cooling process

Film Cooling

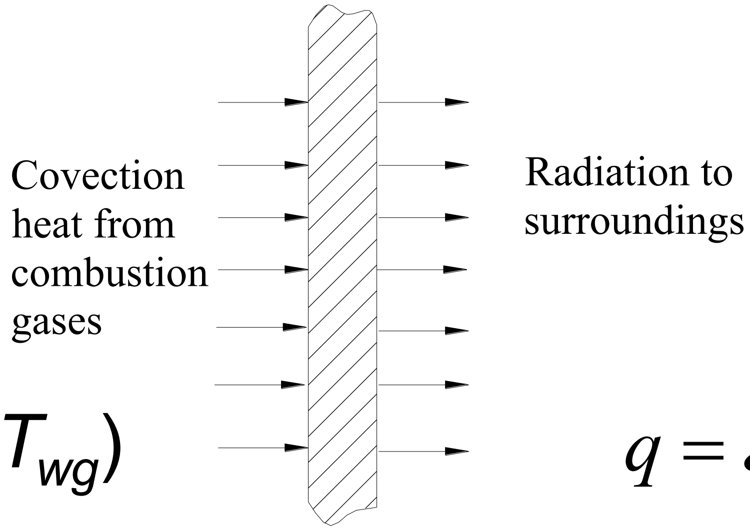
- Providing a layer of cool gas near the wall surface
  - slots and holes provided in thrust-chamber walls, serve to introduce a coolant
  - Used as supplementary to regenerative cooling



Ablative Cooling

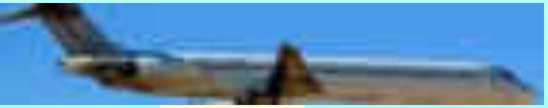
- Combustion gas-side wall material is sacrificed by melting, vaporization and chemical changes to dissipate heat
  - Relatively cool gases flow over the wall surface, thus lowering the boundary-layer temperature and assisting the cooling process
- Key phenomena:
  - Decomposition and evaporation of the material of the wall, which generates a buffer gas layer, reducing the heat transfer rate

Radiation Cooling



$$q=h_g(T_{aw}-T_{wg})$$

$$q=\varepsilon\sigma(T_{wg}^4-T_{sur.}^4)$$



HANK YOU !