# AE-2230-II Propulsion and Power

## Lecture-5 : Aero Engines - examples

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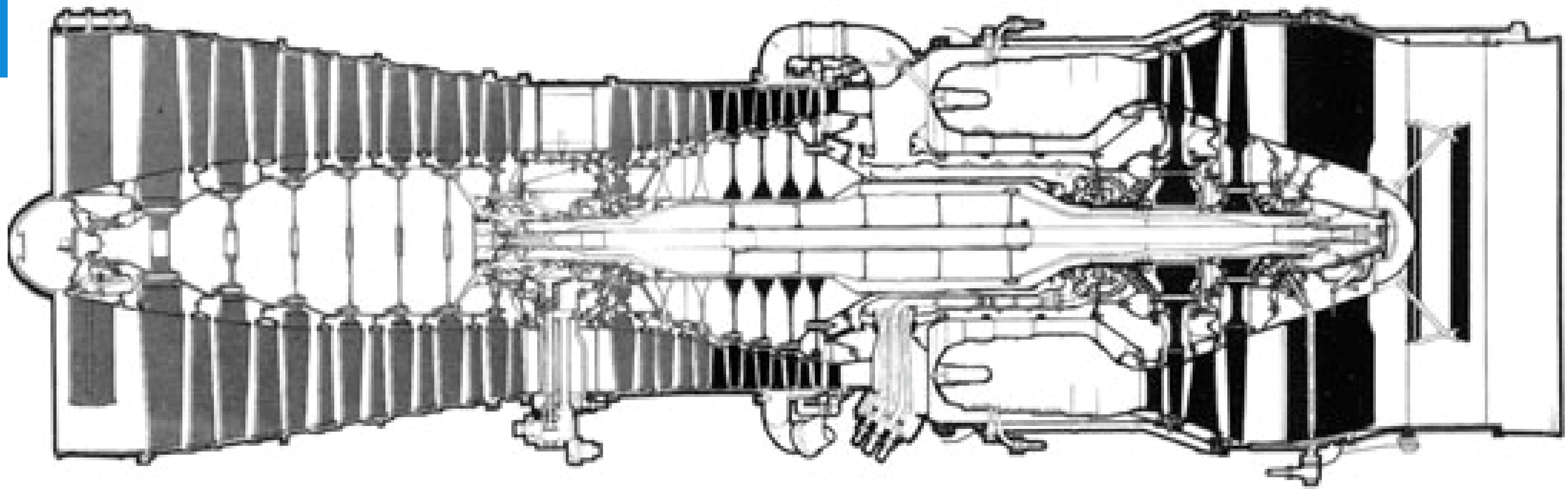
Flight Performance and Propulsion

# EXAMPLE OF A TURBOJET ENGINE

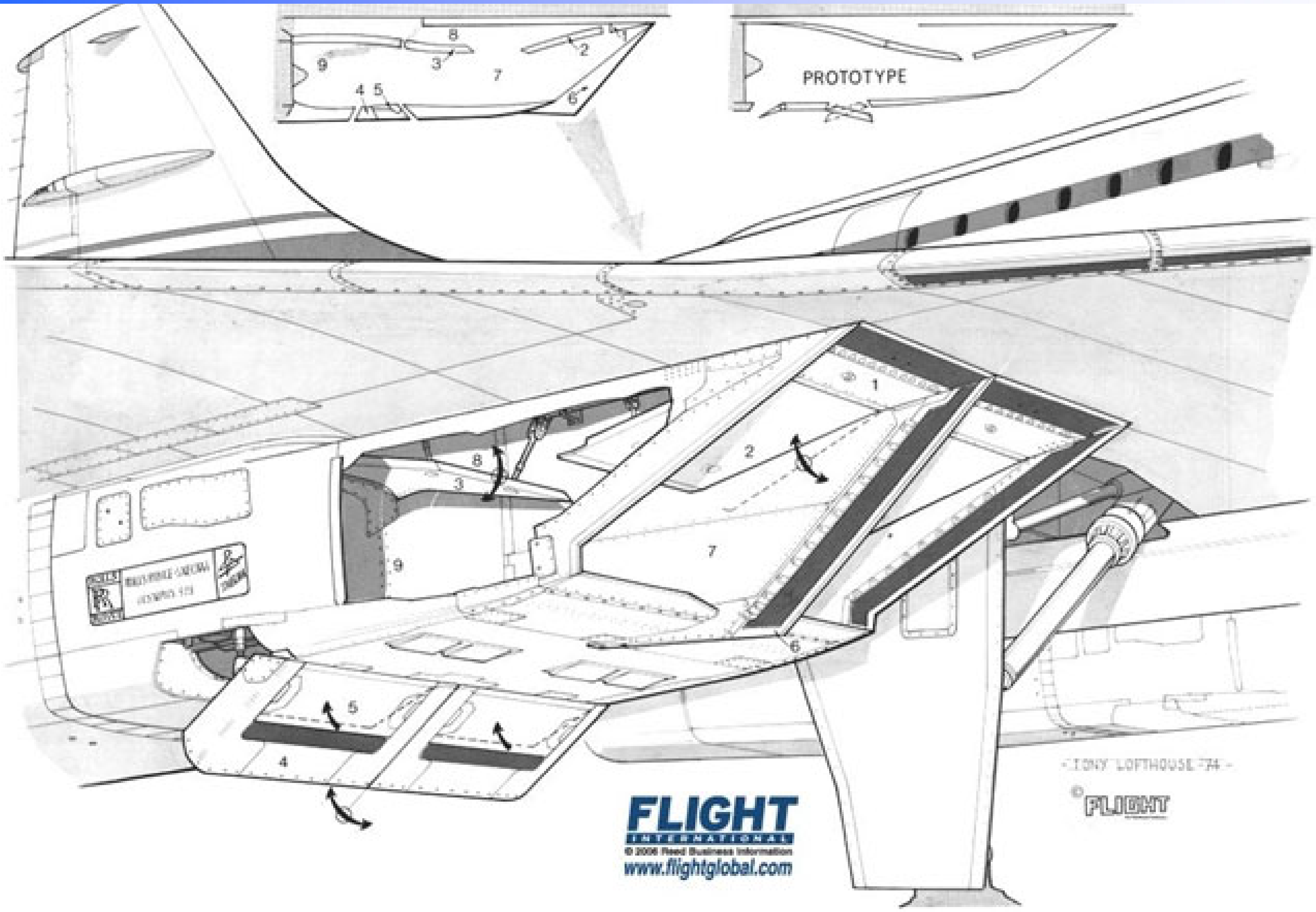
# Concorde

# Rolls Royce-Snecma Olympus 593

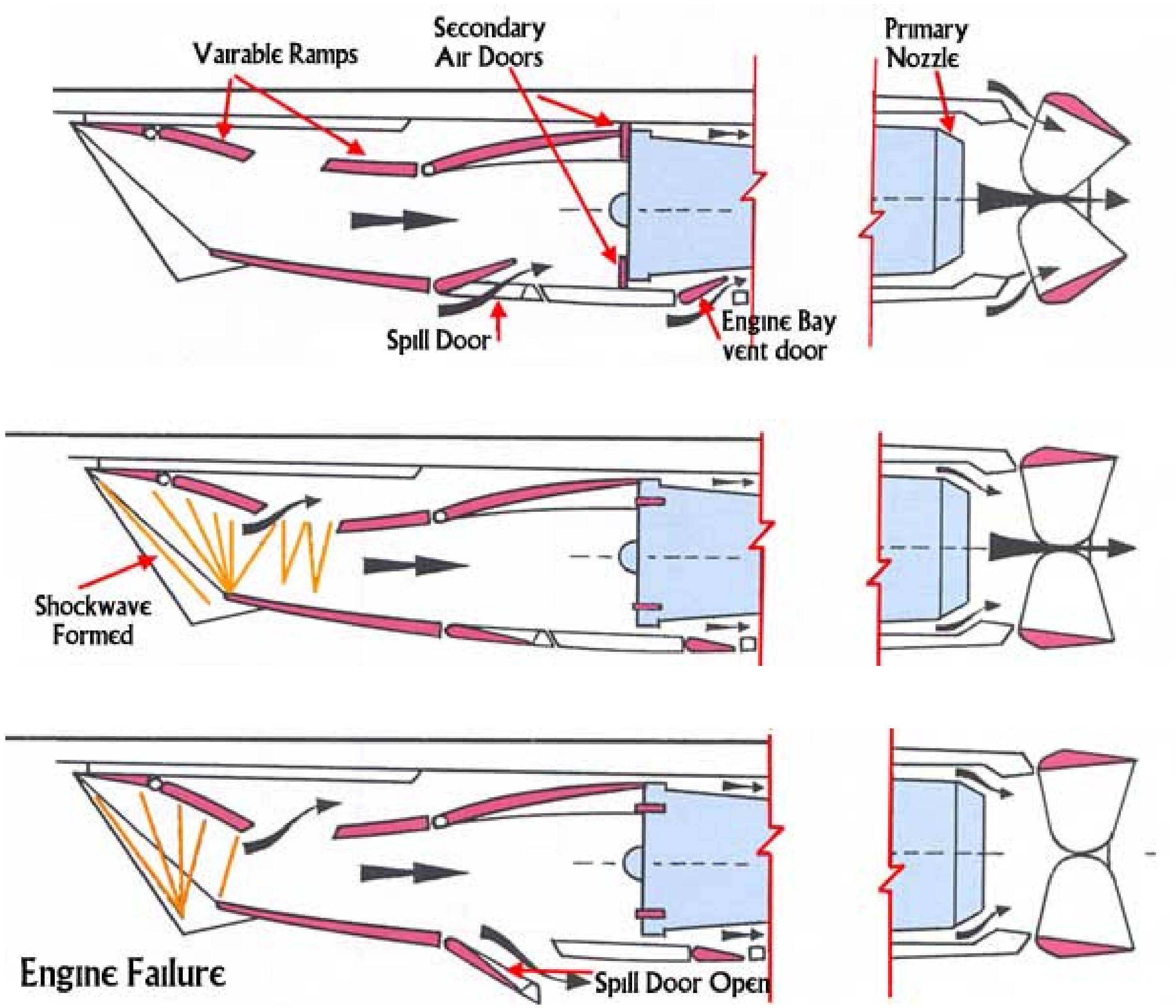


Twin spool turbojet
Compressor: Axial flow, 7-stage low pressure, 7-stage high pressure
Turbine: High pressure single stage, low pressure single stage
Overall pressure ratio: 16:1

# Variable geometry intake (concorde)



# Variable geometry intake (concorde)

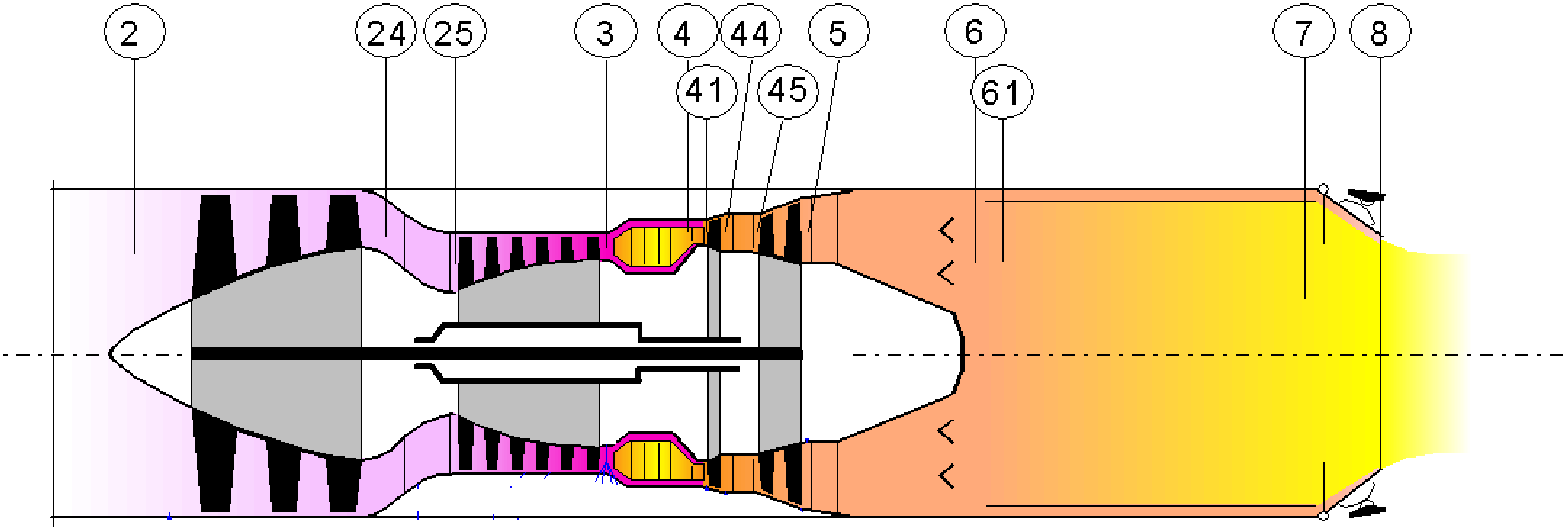



Takeoff / subsonic

Supersonic

Engine Failure

# Engine Example




Within the engine only total properties are calculated at these stations

# Olympus Engine Specifications

General characteristics at ISA take off conditions

Type: Afterburning twin spool turbojet

Compressor = 2 spool axial, 7 low pressure stages, 7 high pressure stages

Combustors = can-annular combustor

Turbine = Single stage high pressure, single stage low pressure

Nozzle = Convergent

LPC Pressure Ratio = 4.0

HPC Pressure Ratio = 4.0

Bypass Ratio = 0

Combustor Exit Temperature ($T_{0,4}$) = 1450 K

Afterburner Exit Temperature ($T_{0,7}$) = 1850 K

Engine mass flow rate= 160 kg/s

Intake Pressure ratio= 0.92 (at take off)

# Given Data & Assumptions

Compressor isentropic efficiency = 0.85
Turbine isentropic efficiency = 0.9
Mechanical efficiency = 0.99
Combustion efficiency = 0.99
Combustion chamber Pressure Ratio = 0.97
Afterburner Pressure Ratio = 0.97
Afterburner efficiency = 0.95
Nozzle efficiency = 0.95
$c_{p,air}$ = 1000 J/kg.K; $\kappa_{air}$ = 1.4
$c_{p,gas}$ = 1150 J/kg.K; $\kappa_{gas}$ = 1.33
Gas constant= 287 J/kg.K
Fuel calorific value = 43 MJ/kg
Ambient Pressure = 101,325 Pa
Ambient Temperature = 288 K

What are pressures and temperatures in all steps of the cycle and what is the thrust?

# Engine Example

General characteristics at ISA take off conditions

Compressor= 2 spool axial, 7 low pressure stages, 7 high pressure stages

Turbine= Single stage high pressure, single stage low pressure

Nozzle= Convergent

LPC Pressure Ratio = 4.0

HPC Pressure Ratio = 4.0

Bypass Ratio = 0

Ambient Temperature = 288 K

Combustor Exit Temperature ($T_{0,4}$) = 1450K

Intake Pressure ratio= 0.92 (at take off)

Combustion chamber Pressure Ratio = 0.97

Afterburner Pressure Ratio = 0.97

Afterburner efficiency = 0.95

Afterburner Exit Temperature ($T_{0,7}$) = 1850 K

Engine mass flow rate= 160 kg/s

Compressor isentropic efficiency = 0.85

Turbine isentropic efficiency = 0.9

Mechanical efficiency = 0.99

Combustion efficiency = 0.99

Nozzle efficiency = 0.95

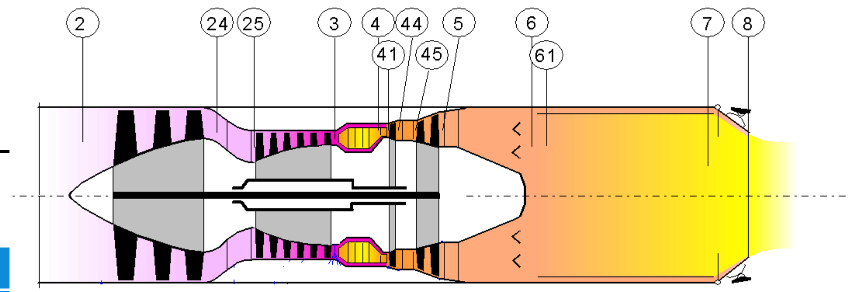
$c_{p,air}$ = 1000 J/kg.K; $\kappa_{air}$ = 1.4

$c_{p,gas}$ = 1150 J/kg.K; $\kappa_{gas}$ = 1.33

Gas constant= 287 J/kg.K

Fuel calorific value = 43 MJ/kg

Ambient Pressure = 101,325 Pa



# Olympus 593 Engine

inlet pressure ratio is 0.92 at takeoff

Thus $p_{0,2}$ = 0.92*101,325 Pa = 93,219 Pa ; $T_2$ = 288 K

$p_{0,25} = p_{0,2}$*4.0 = 372,876 Pa;

$$\frac{T_{0,25}}{T_{0,2}} = 1 + \frac{1}{\eta_{is}}\left[\left(\frac{p_{0,25}}{p_{0,2}}\right)^{\left(\frac{\kappa-1}{\kappa}\right)} - 1\right]$$

$T_{025}$ = 452.9 K; $\dot{m}_{25}$ = 160 kg/s

$p_{0,3} = p_{0,25}$*4.0 = 1,491,504 Pa;

$$\frac{T_{0,3}}{T_{0,25}} = 1 + \frac{1}{\eta_{is}}\left[\left(\frac{p_{0,3}}{p_{0,25}}\right)^{\left(\frac{\kappa-1}{\kappa}\right)} - 1\right]$$

$T_{0,3}$ = 711.85 K; $\dot{m}_3$ = 160 kg/s

# Olympus 593 Engine

Combustion Chamber

$T_{0,4}$ = 1450 K (given);

$$\dot{m}_{fuel} = \frac{\dot{m}_3 \cdot c_{pgas} \cdot \Delta T}{LHV \cdot \eta_{cc}}$$

$\dot{m}_{fuel}$ = 3.19 kg/s

$p_{0,4} = 0.97 * p_{0,3}$ = 1,446,758 Pa = 14.46 bar

$\dot{m}_4 = \dot{m}_3 + \dot{m}_{fuel}$ = 163.19 kg/s

Work done in LPC = $\dot{m} \cdot c_{p,air} \cdot (T_{0,25} - T_{0,2})$ = 160*1,000*(452.7-288) = 26.352 MW

Work done in HPC = $\dot{m} \cdot c_{p,air} \cdot (T_{0,3} - T_{0,25})$ = 160*1,000*(711.52-452.7) = 41.41 MW

# Olympus 593 Engine

$$T_{0,4} - T_{0,45} = (\dot{W}_{HPC}) / (\dot{m}_4 \cdot c_{p\,gas} \cdot \eta_{mech}) = 223.0K$$

$$T_{0,45} = (1450K - 223K) = 1227K$$

$$\frac{T_{0,45}}{T_{0,4}} = 1 - \eta_{is,HPT}\left[1 - \left(\frac{p_{0,45}}{p_{0,4}}\right)^{\left(\frac{\kappa_g - 1}{\kappa_g}\right)}\right]$$

=> $p_{0,45}$ = 679,820 Pa and $m_{45}$ = 163.19 kg/s

Work done in LPC $= \dot{m} \cdot c_{p\,air} \cdot (T_{0,25} - T_{0,2})$ = 160*1000*(452.7-288) = 26.352 MW

# Olympus 593 Engine

$$T_{0,45} - T_{0,5} = (\dot{W}_{LPC}) / (\dot{m}_{45} \cdot c_{p\,gas} \cdot \eta_{mech}) = 141.9K$$

$$T_{0,5} = (1227K - 141.9K) = 1085.1K$$

$$\frac{T_{0,5}}{T_{0,45}} = 1 - \eta_{is,LPT}\left[1 - \left(\frac{p_{0,5}}{p_{0,45}}\right)^{\left(\frac{\kappa_g - 1}{\kappa_g}\right)}\right]$$

=> $p_{0,5}$ =390,565 Pa

# Olympus 593 Engine

Afterburner
station 7 is afterburner exit

$p_{0,7} = p_{0,5} * 0.97 = 379{,}233.79$ Pa

$T_{0,7} = 1850$ (given)

$$\dot{m}_{fuel,ab} = \frac{\dot{m}_5 \cdot c_{p,gas} \cdot \Delta T}{LHV \cdot \eta_{ab}}$$

$$\dot{m}_{fuel,ab} = \frac{163.19 * 1150 * (1850 - 1085.36)}{(43 \times 10^6) * (0.95)} = 3.513 \text{ kg/s}$$

$$\dot{m}_7 = \dot{m}_5 + \dot{m}_{ab} = 163.19 + 3.513 = 166.70\, kg / s$$

# Olympus 593 Engine

## Nozzle

$$p_{0,7} / p_{critical} = \left[ \frac{1}{\left(1 - \left(\frac{1}{\eta_{noz}}\right) \cdot \left(\frac{\kappa_g - 1}{\kappa_g + 1}\right)\right)^{\frac{\kappa_g}{\kappa_g - 1}}} \right] = 1.916$$

$p_{0,7}/p_0 = 379{,}233.79/101{,}325 = 3.74 > p_{0,7}/p_{cr}$ ; Thus the nozzle is chocked

$$T_8 = T_{0,7}\left(\frac{2}{\kappa_g + 1}\right) = 1587.98 K \qquad p_8 = p_{0,7}\left(\frac{1}{p_{0,7} / p_{CR}}\right) = 197{,}644\, Pa$$

$$\rho_8 = \left(\frac{p_8}{R \cdot T_8}\right) = 0.434\, kg / m^3 \qquad V_8 = \sqrt{\left(\kappa_g \cdot R \cdot T_8\right)} = 778.56\, m / s$$

# Olympus 593 Engine

$$A_8 = A_{noz} = \left( \frac{\dot{m}}{\rho_8 \cdot V_8} \right) = 0.493 m^2$$

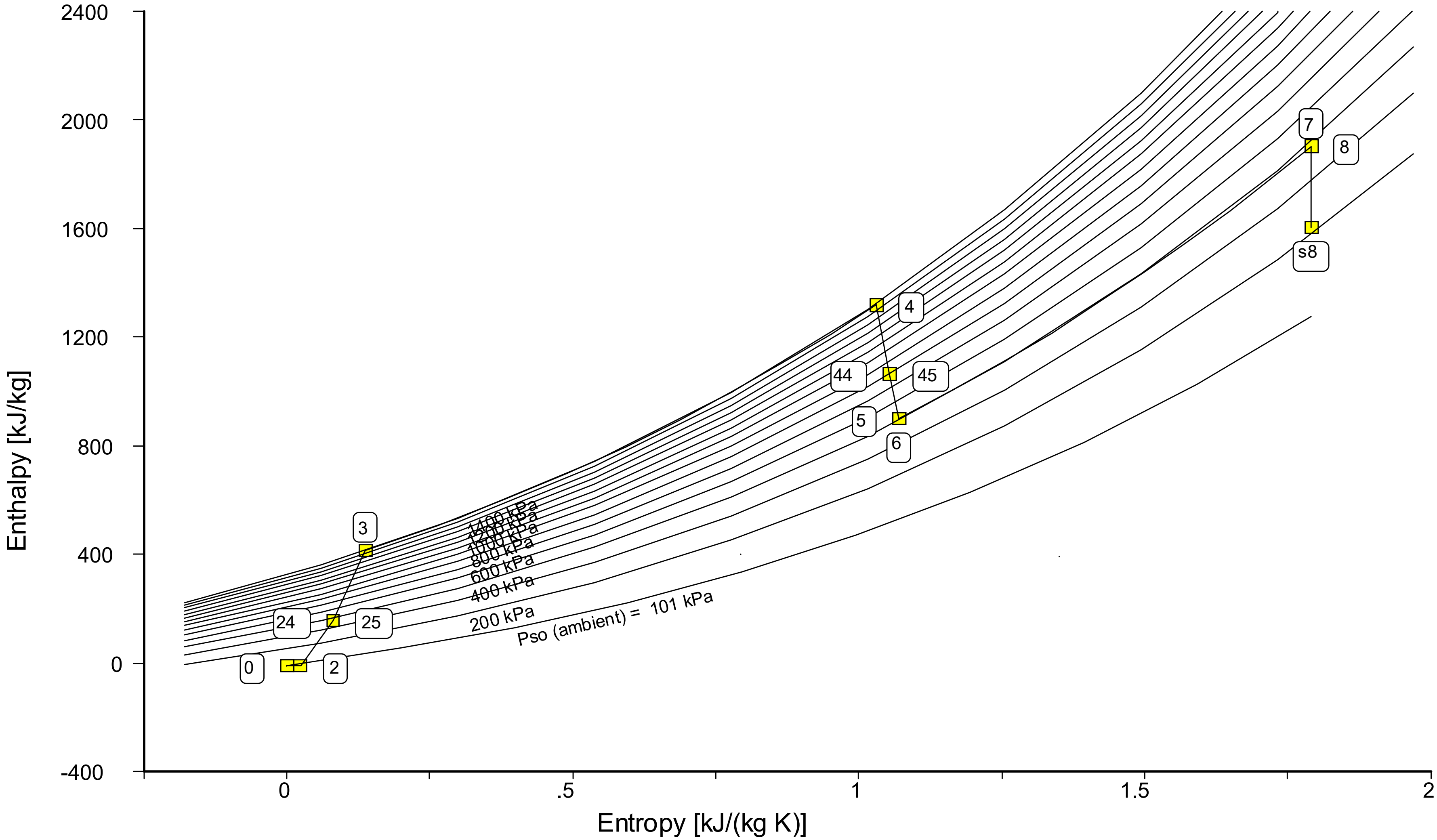
$$F = \dot{m} \cdot (V_8 - V_0) + A_8 (p_8 - p_0)$$

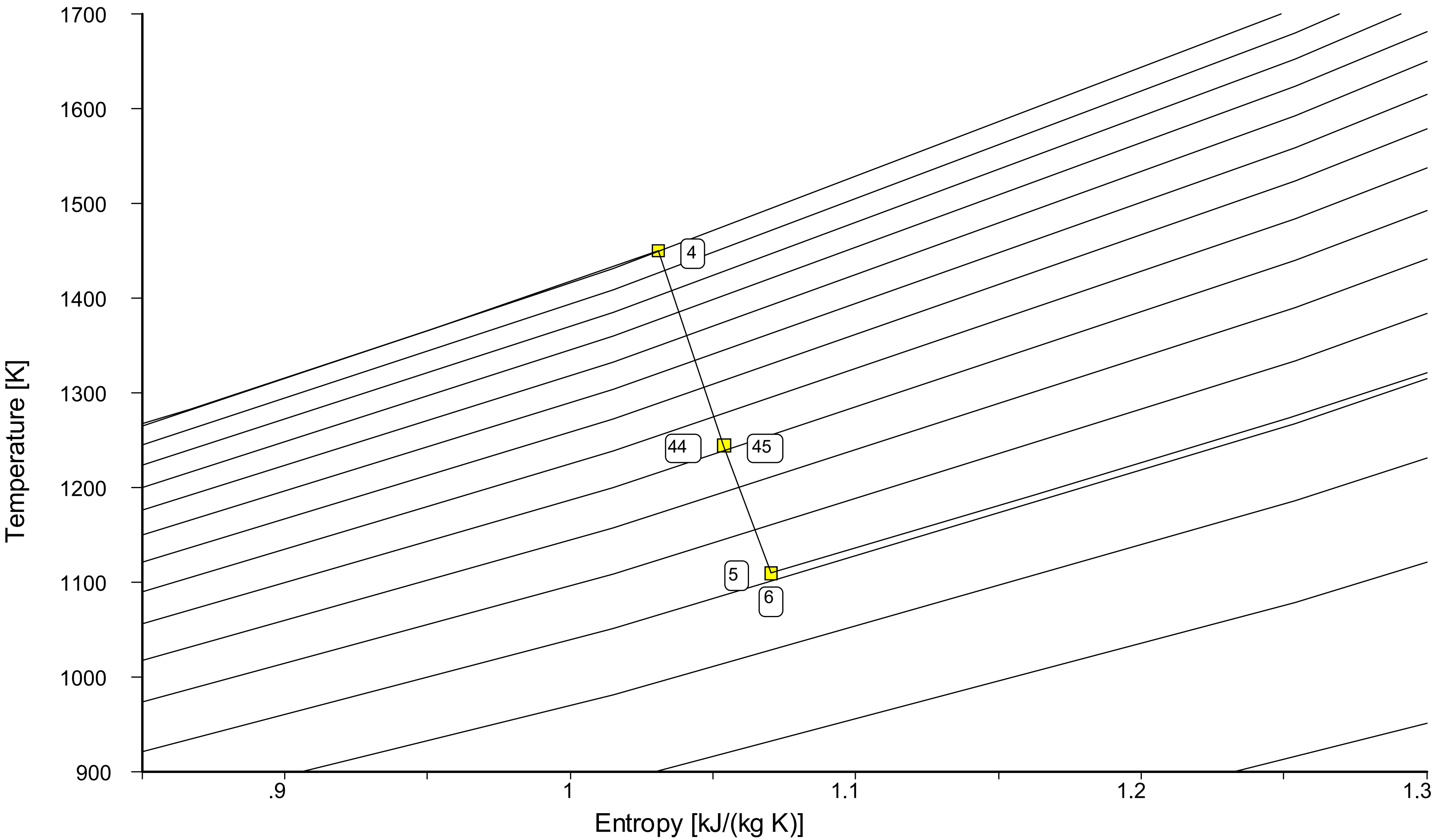
$$F = 166.70 \cdot (778.56 - 0) + 0.493 (197,929.95 - 101,325)$$

$$F = 129,231 + 47,626 = 177.85\, kN$$

$$SFC = \left( \frac{\dot{m}_{fuel}}{F} \right) = \left( \frac{3.19 + 3.513}{176.85} \right) = 37.90\, gm / kN \cdot s$$

# Olympus 593 Engine

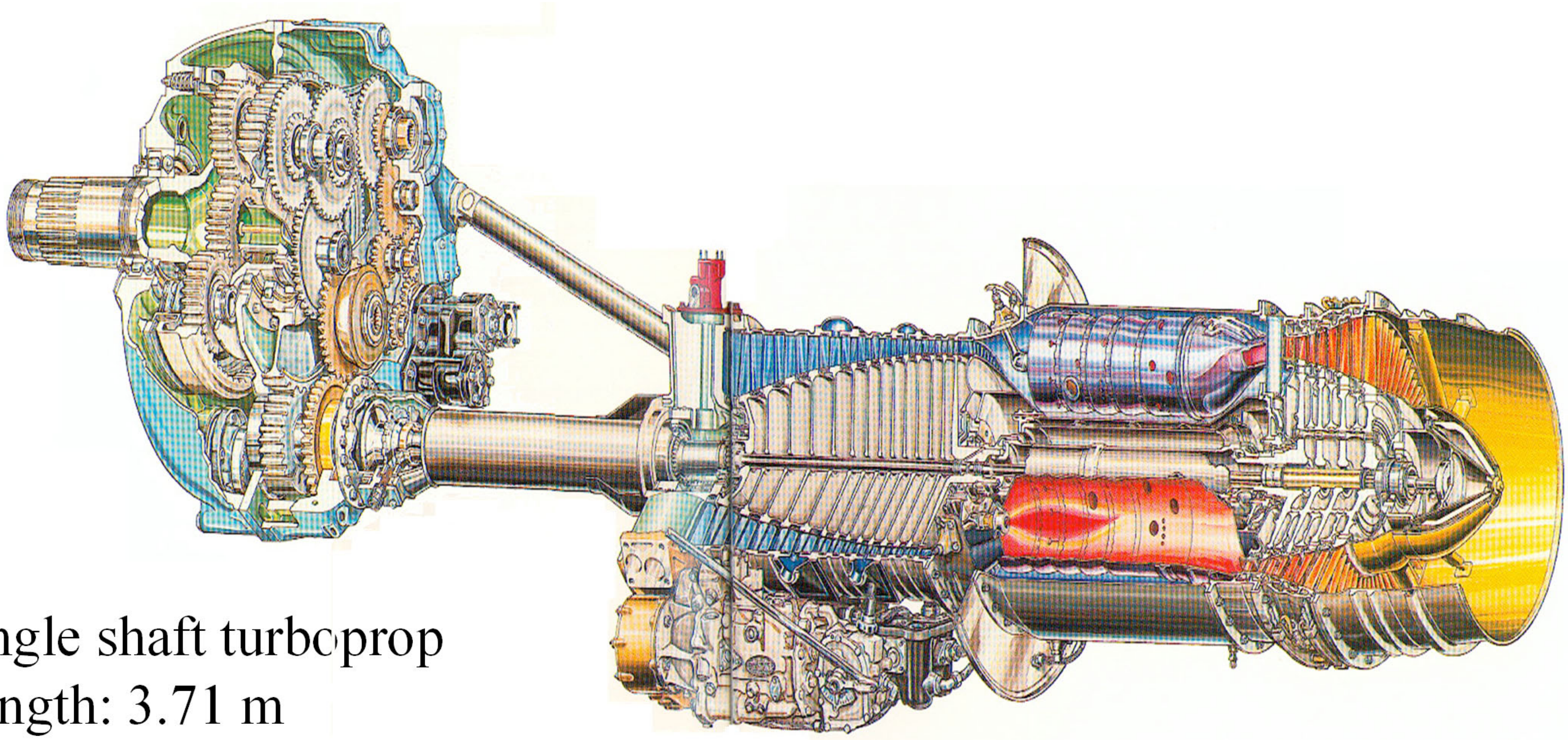



1700
1600
1500
1400
1300
1200
1100
1000
900
Temperature [K]
.9
1
1.1
1.2
1.3
Entropy [kJ/(kg K)]
4
44
45
5
6

# EXAMPLE OF A SINGLE SHAFT TURBOPROP ENGINE

# E-2C Hawkeye

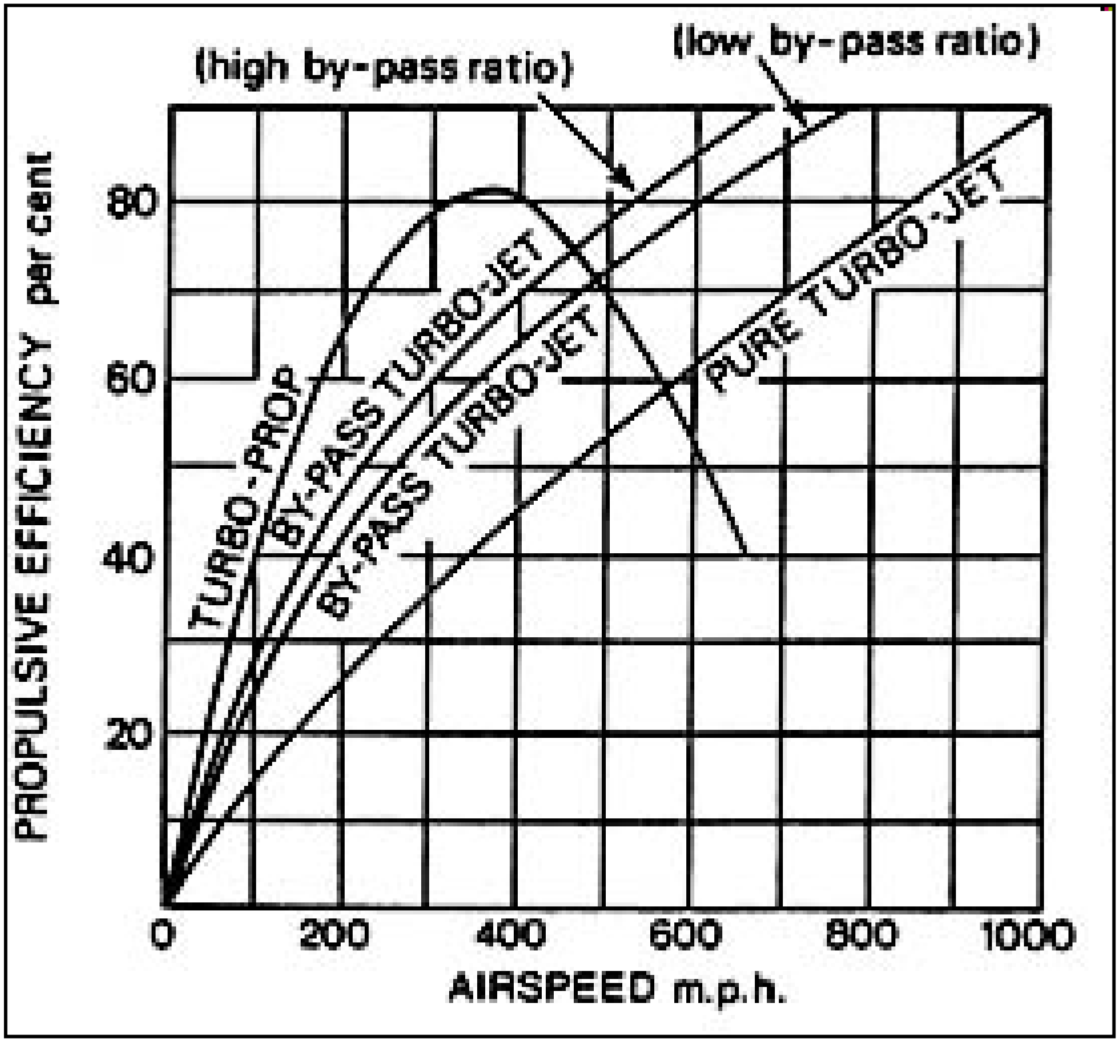
# Rolls Royce T56-A Series IV



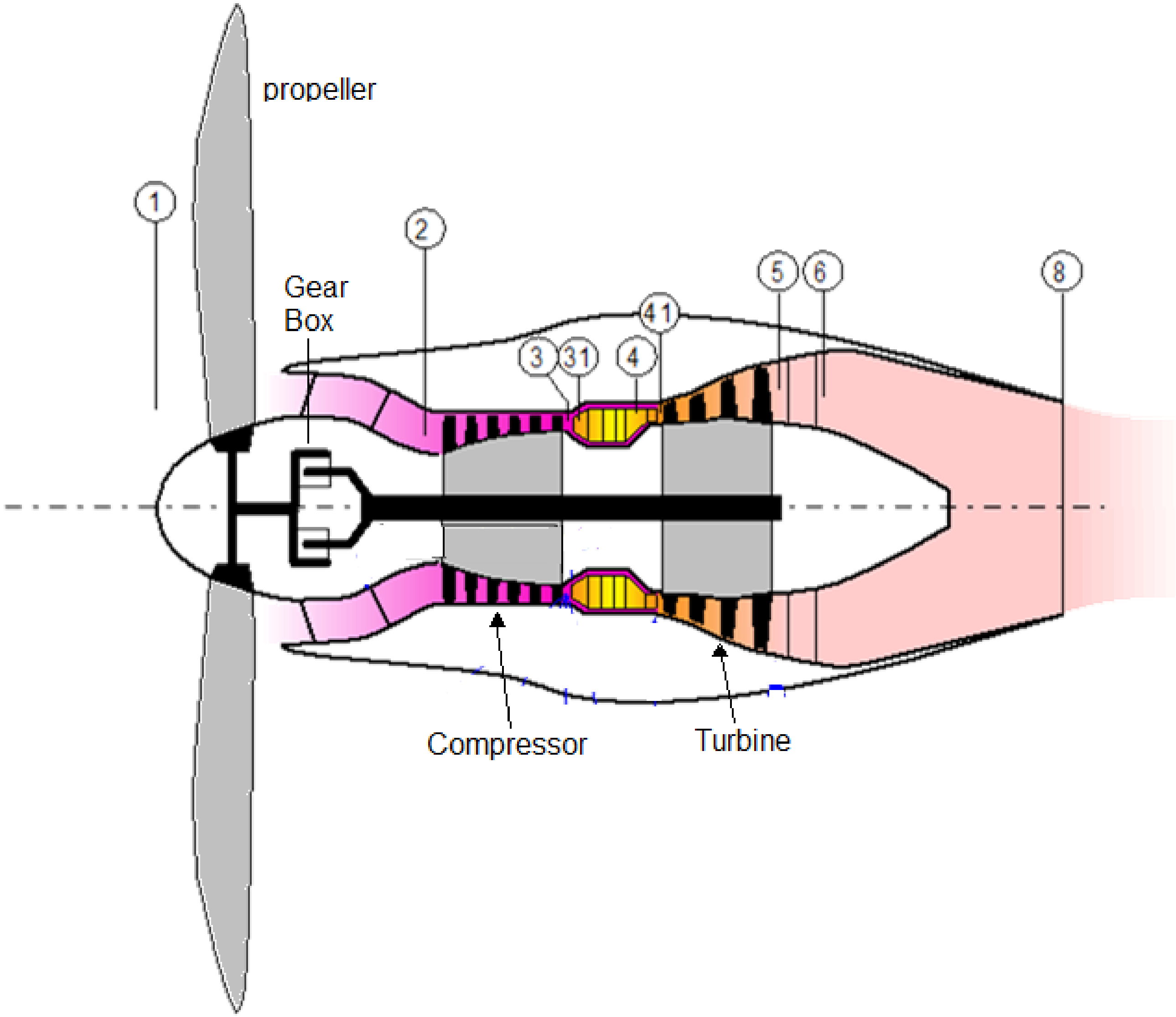
Single shaft turboprop
Length: 3.71 m
Diameter: 0.69 m
Basic weight: 1940 lb
Power-to-weight ratio: 2.75:1

Source: Rolls Royce

# Propulsive efficiency



# Engine Example



# T56-A Series IV Specifications

General characteristics at SL, ISA take off conditions, Mach = 0.5

Type: Single shaft turboprop engine

Compressor = 14 stages axial flow

Combustors = 6-cylindrical flow-through

Turbine = 4 stages axial flow

Nozzle = Convergent

Shaft power = 5,250 shp (3.91 MW)

Compressor Pressure Ratio = 11.5

Combustor Exit Temperature ($T_{0,4}$) = 1130 K

Engine mass flow rate= 35 kg/s

Intake Pressure ratio= 1.0

# Given Data & Assumptions

Propeller efficiency = 0.9

Compressor isentropic efficiency = 0.85

Turbine isentropic efficiency = 0.89

Mechanical efficiency = 0.99

Combustion efficiency = 0.995

Combustion chamber Pressure Ratio = 0.96

Nozzle efficiency = 0.95

$c_{p,air}$ = 1000 J/kg.K; $\kappa_{air}$ = 1.4

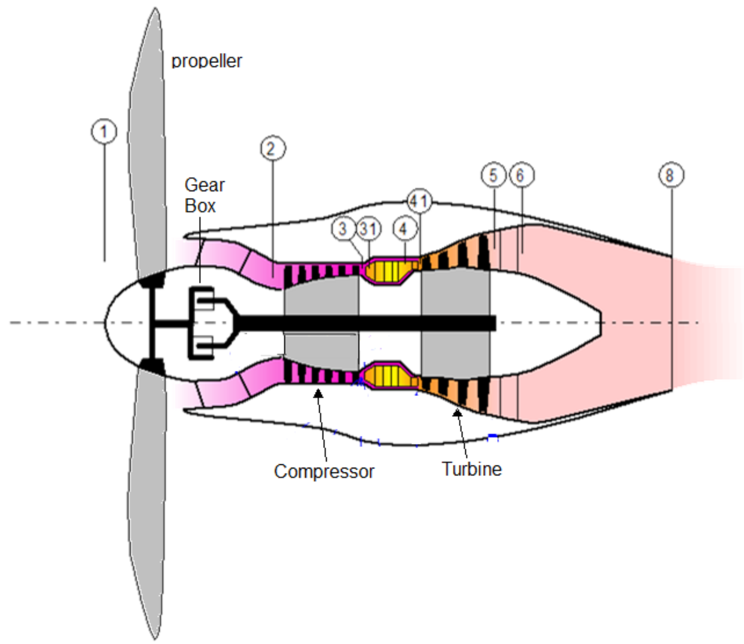
$c_{p,gas}$ = 1150 J/kgK; $\kappa_{gas}$ = 1.33

Gas constant= 287 J/kg.K

Fuel calorific value = 42.8 MJ/kg

Ambient Pressure = 101,325 Pa

Ambient Temperature = 288.15 K




Find pressure, temperature and mass flow at every station.
Calculate the total thrust

# Given Data & Assumptions

General characteristics at SL, ISA take off conditions, Mach = 0.5
Type: Single shaft turboprop engine
Compressor = 14 stages axial flow
Combustors = 6-cylindrical flow-through
Turbine = 4 stages axial flow
Nozzle = Convergent
Shaft power = 5250 shp (3.91 MW)
Compressor Pressure Ratio = 11.5
Combustor Exit Temperature ($T_4$) = 1130 K
Engine mass flow rate= 35 kg/s
Intake Pressure ratio= 1.0
Propeller efficiency = 0.9
Compressor isentropic efficiency = 0.85
Turbine isentropic efficiency = 0.89
Mechanical efficiency = 0.99
Combustion efficiency = 0.995
Combustion chamber Pressure Ratio = 0.96
Nozzle efficiency = 0.95

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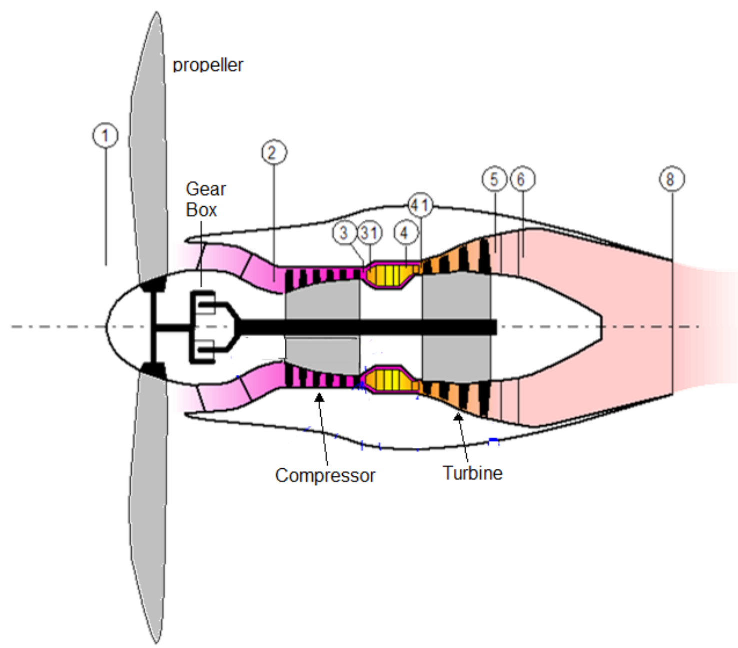
Ambient Temperature = 288.15 K



# T56-A Series IV Engine

$T_a = 288.15$ K; $p_a = 101{,}325$ Pa

Mach = 0.5

$$T_{0,a} = T_a \cdot \left(1 + \frac{\kappa_a - 1}{2} \cdot M_a^2\right) = 302.56K$$

$$p_{0,a} = p_a \cdot \left(1 + \frac{\kappa_a - 1}{2} \cdot M_a^2\right)^{\frac{k_a}{k_a - 1}} = 120{,}193\,Pa$$

$$V_0 = M_a \cdot \sqrt{\kappa_a \cdot R \cdot T_a} = 170.13\,m/s$$

inlet pressure ratio is 1.0

Thus $p_{0,2} = p_{0,a} = 120{,}193$ Pa ; $T_{0,2} = T_{0,a} = 302.56$ K

# T56-A Series IV Engine

*Compressor*

$p_{0,3} = p_{0,2}*11.5 = 1{,}382{,}219.45\text{Pa};$

$T_{0,3} = 661.84\text{ K}; \dot{m}_3 = 35\text{ kg/s}$

$$\frac{T_{0,3}}{T_{0,2}} = 1 + \frac{1}{\eta_{is}}\left[\left(\frac{p_{0,3}}{p_{0,2}}\right)^{\left(\frac{\kappa_a - 1}{\kappa_a}\right)} - 1\right]$$

*Combustion Chamber*

$T_{0,4} = 1130\text{ K (given)};$

$\dot{m}_{fuel} = 0.4425\text{ kg/s}$

$$\dot{m}_{fuel} = \frac{\dot{m}_3 \cdot c_{p,gas} \cdot \Delta T}{LHV \cdot \eta_{cc}}$$

$p_{0,4} = 0.96*p_{0,3} = 1{,}326{,}930.67\text{ Pa} = 1{,}326.93\text{ kPa}$

$\dot{m}_4 = \dot{m}_3 + \dot{m}_{fuel} = 35.4425\text{ kg/s}$

# T56-A Series IV Engine

*Work done in Compressor*

$$\dot{W}_c = \dot{m}_3 \cdot c_{p,air} \cdot (T_{0,3} - T_{0,2}) = 35 * 1000 * (661.84 - 302.56) = 12,574,800\,W$$

*Turbine*

$$T_{0,4} - T_{0,5} = (\dot{W}_C + \dot{W}_{prop}) / (\dot{m}_4 \cdot c_{p,gas} \cdot \eta_{mech}) = 408.65K$$

$$T_{0,5} = 1130 - 408.65 = 721.35K$$

$$\frac{T_{0,5}}{T_{0,4}} = 1 - \eta_{is,T} \left[ 1 - \left( \frac{p_{0,5}}{p_{0,4}} \right)^{\left( \frac{\kappa_g - 1}{\kappa_g} \right)} \right]$$

$$p_{0,5} = 162,368\,Pa = 162.37\,kPa$$

# T56-A Series IV Engine

*Nozzle*

$$\Pi_{cr} = p_{05} / p_{critical} = \left[ \frac{1}{\left( 1 - \left( \frac{1}{\eta_{noz}} \right) \cdot \left( \frac{\kappa_g - 1}{\kappa_g + 1} \right) \right)^{\frac{\kappa_g}{\kappa_g - 1}}} \right] = 1.916$$

$$p_{0,5} / p_a = 162,368 / 101,325 = 1.60 < \Pi_{cr}$$

The nozzle is not chocked and $p_{0,8} < p_{0,5}$

$$p_{s,8} = p_a = 101,325\, Pa$$

$$\frac{T_{s,8}}{T_{0,5}} = 1 - \eta_{is_{nozzle}} \left( 1 - \left( \frac{p_{atm}}{p_{0,5}} \right)^{\frac{\kappa_g - 1}{\kappa_g}} \right)$$

$$T_{s,8} = 645.8K$$

$$V_8 = \sqrt{2 \cdot c_{p,gas} \cdot (T_{0,5} - T_{s,8})} = 417.2\, m/s$$

$$F_N = \dot{m}_8 \cdot (V_8 - V_0) = 8,756.7\, N$$

*Propeller Thrust*

$$F_{prop} = \frac{\eta_p \cdot W_{prop}}{V_0} = 20,684\, N$$

*Total Thrust*

$$F_{total} = F_N + F_{prop} = 29,440\, N = 29.4\, kN$$