THERMODYNAMIC ANALYSIS AND PERFORMANCE OF TURBOJET AND TURBOFAN ENGINES

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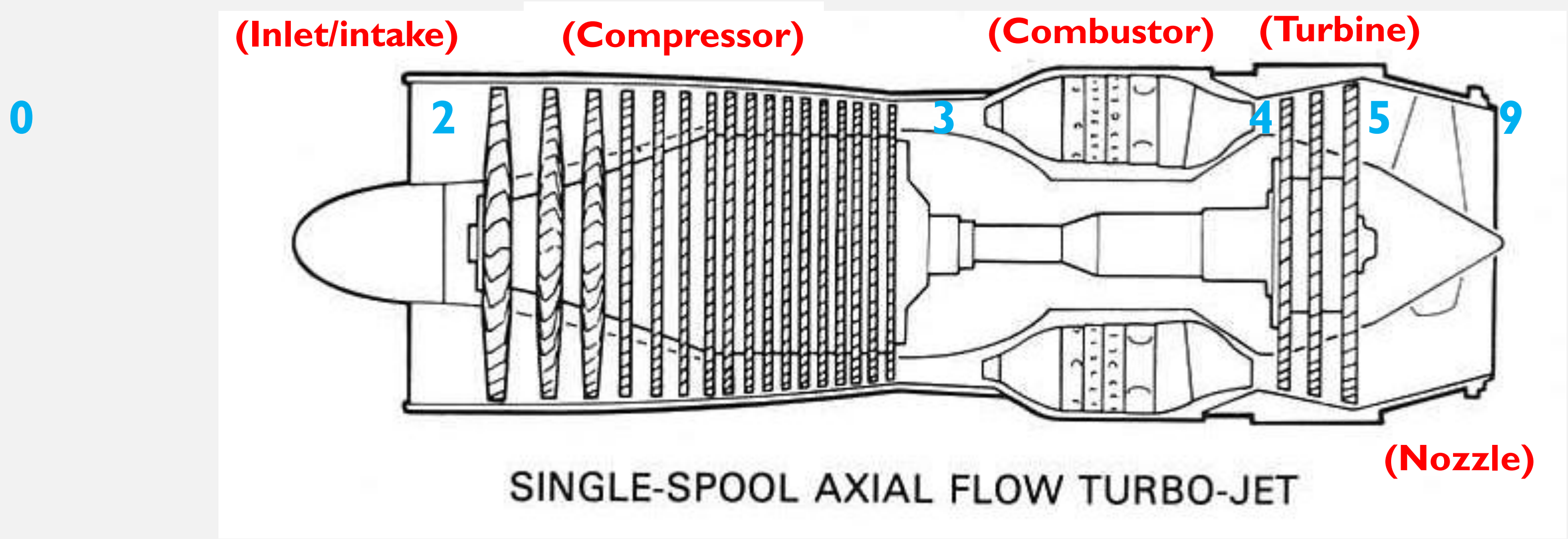
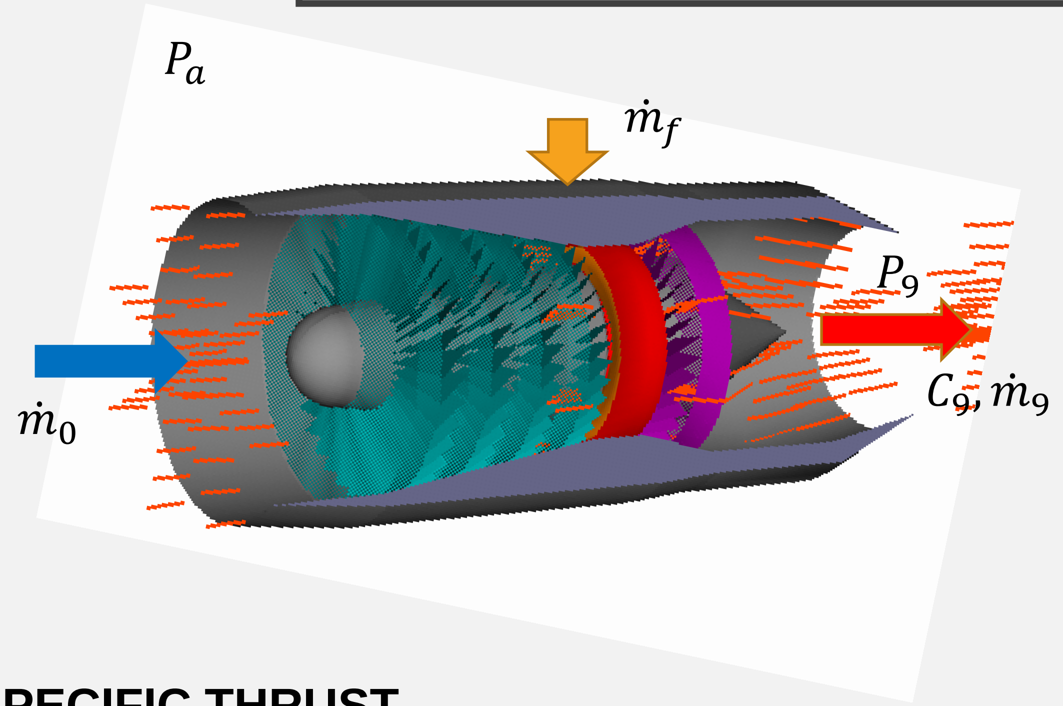
SINGLE-SHAFT TURBOJET ENGINE



Engine Components:

- INLET / INTAKE
- COMPRESSOR
- BURNER / COMBUSTOR
- TURBINE
- PROPELING NOZZLE

ENGINE THRUST AND SPECIFIC PARAMETERS



Flight speed is 0

THTUST / GROS THRUST

$$T = \dot{m}_9 V_9 + A_9(P_9 - P_a)$$

effective exhaust velocity

$$C_{eff} = C + A_9(P_9 - P_a)/\dot{m}_9$$

$$T = \dot{m}_9 C_{eff}$$

Exit pressure = ambient pressure

$$T = \dot{m}_9 C_9$$

Flight speed > 0

THTUST / NET THRUST

$$T = \dot{m}_9 C_9 + A_9(P_9 - P_a) - \dot{m}_0 V_0 = \dot{m}_9 C_{eff} - \dot{m}_0 V_0$$

Net thrust = Gross thrust –Momentum drag

SPECIFIC THRUST

$$ST = T/\dot{m}_0$$

SPECIFIC FUEL CONSUMPTION

$$\mathrm{SFC} = \dot{m}_f / T$$

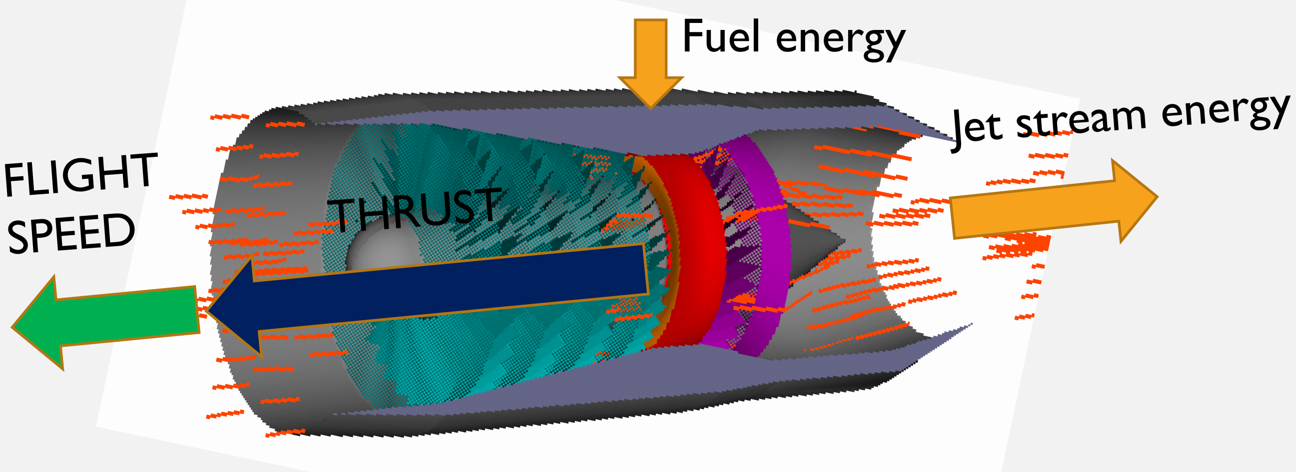
Details in: **Ahmed F. El-Sayed, Aircraft Propulsion and Gas Turbine Engines, (chapter 2.2)**

ENGINE EFFICIENCIES

Thermal efficency

$$\eta_{TH} = \frac{\text{Power imparted to engine airflow}}{\text{Rate of energy supplied in the fuel}}$$

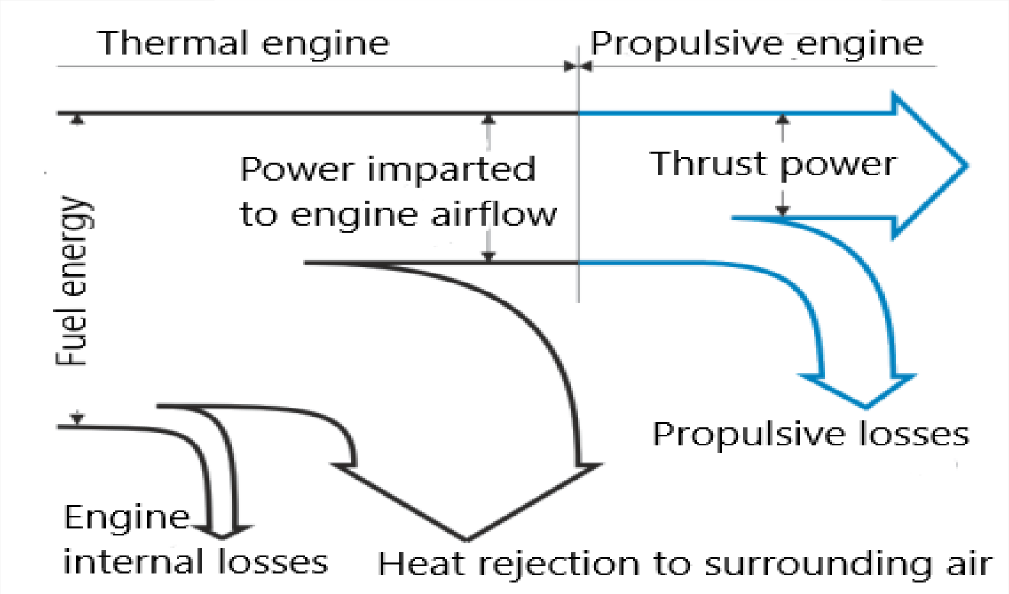
$$\eta_{TH} = \frac{0,5 * (\dot{m}_9 {C_{9e}}^2 - \dot{m}_0 {V_0}^2)}{\dot{m}_f FHV}$$



Propulsive efficency

$$\eta_P = \frac{\text{Thrust power}}{\text{Power imparted to engine airflow}}$$

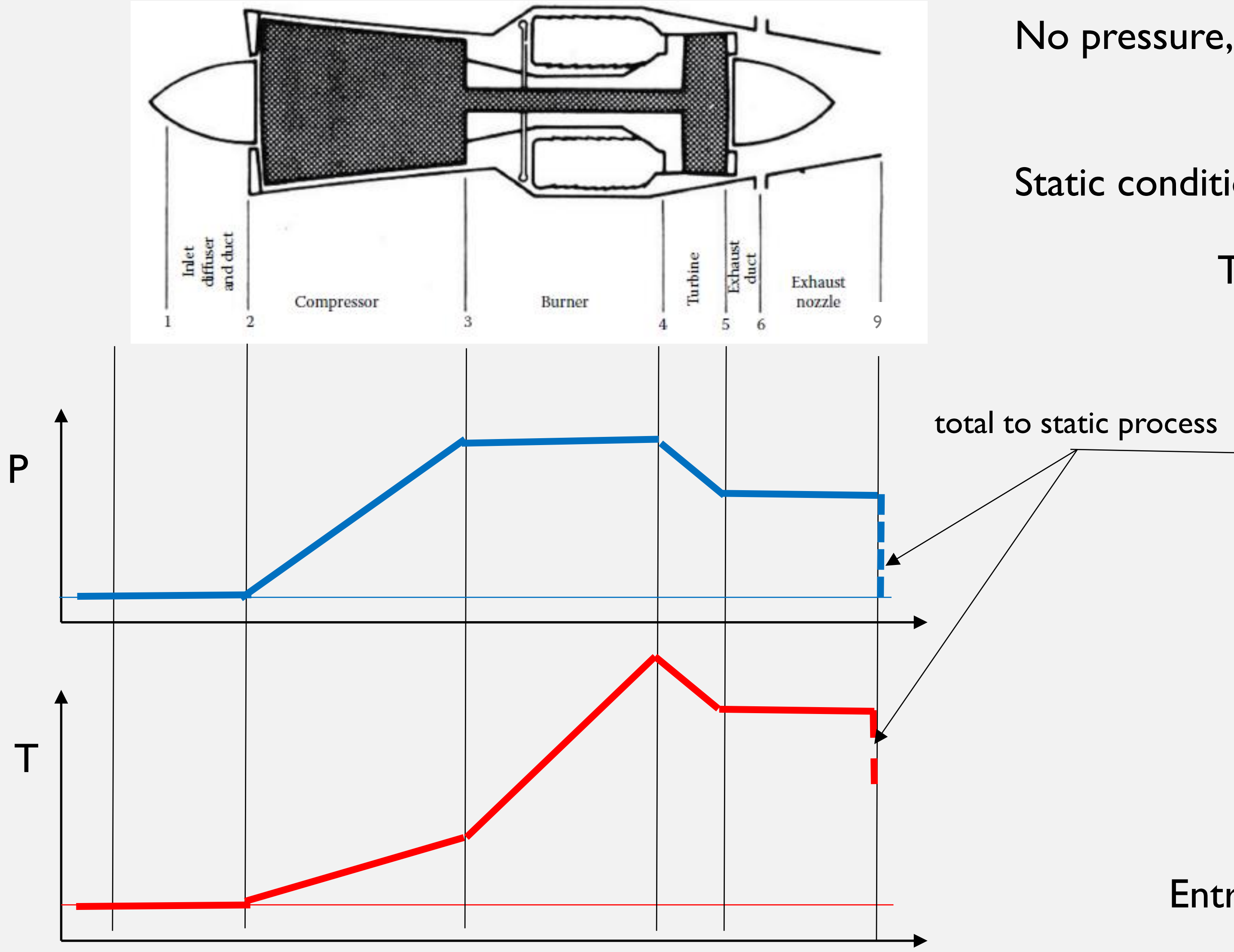
$$\eta_P = \frac{V_0 * T}{0,5 * (\dot{m}_9 {C_{9e}}^2 - \dot{m}_0 {V_0}^2)}$$



Overall efficency

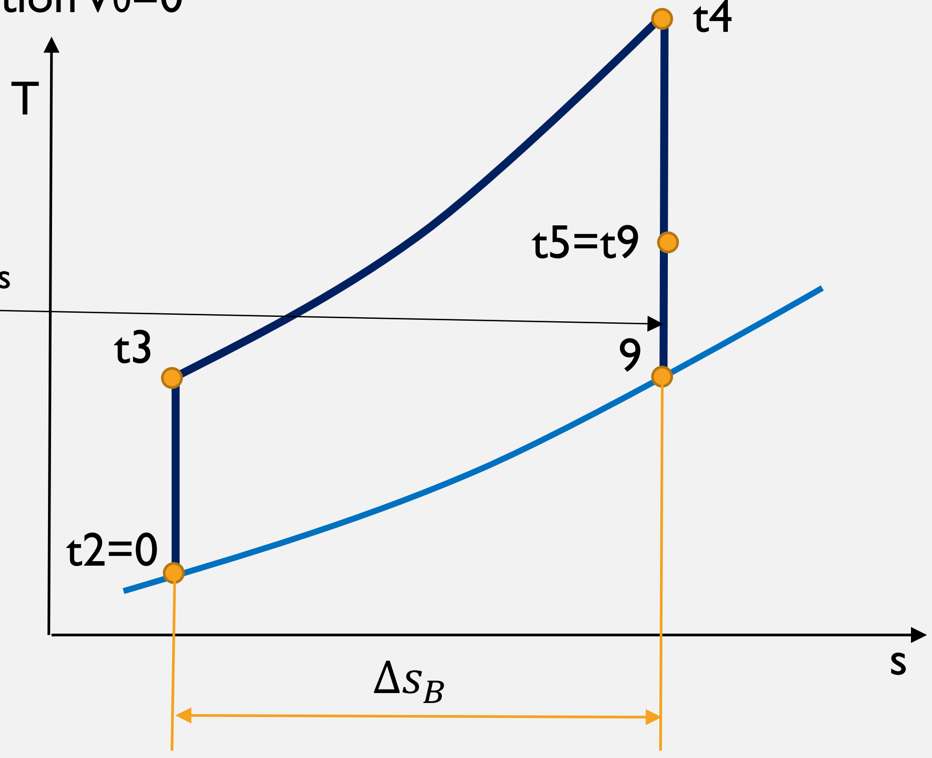
$$\eta_O = \eta_{TH} * \eta_P = \frac{V_0 * T}{\dot{m}_f FHV}$$

IDEAL TURBOJET ENGINE



No pressure, thermal and mechanical losses

Static condition V0=0



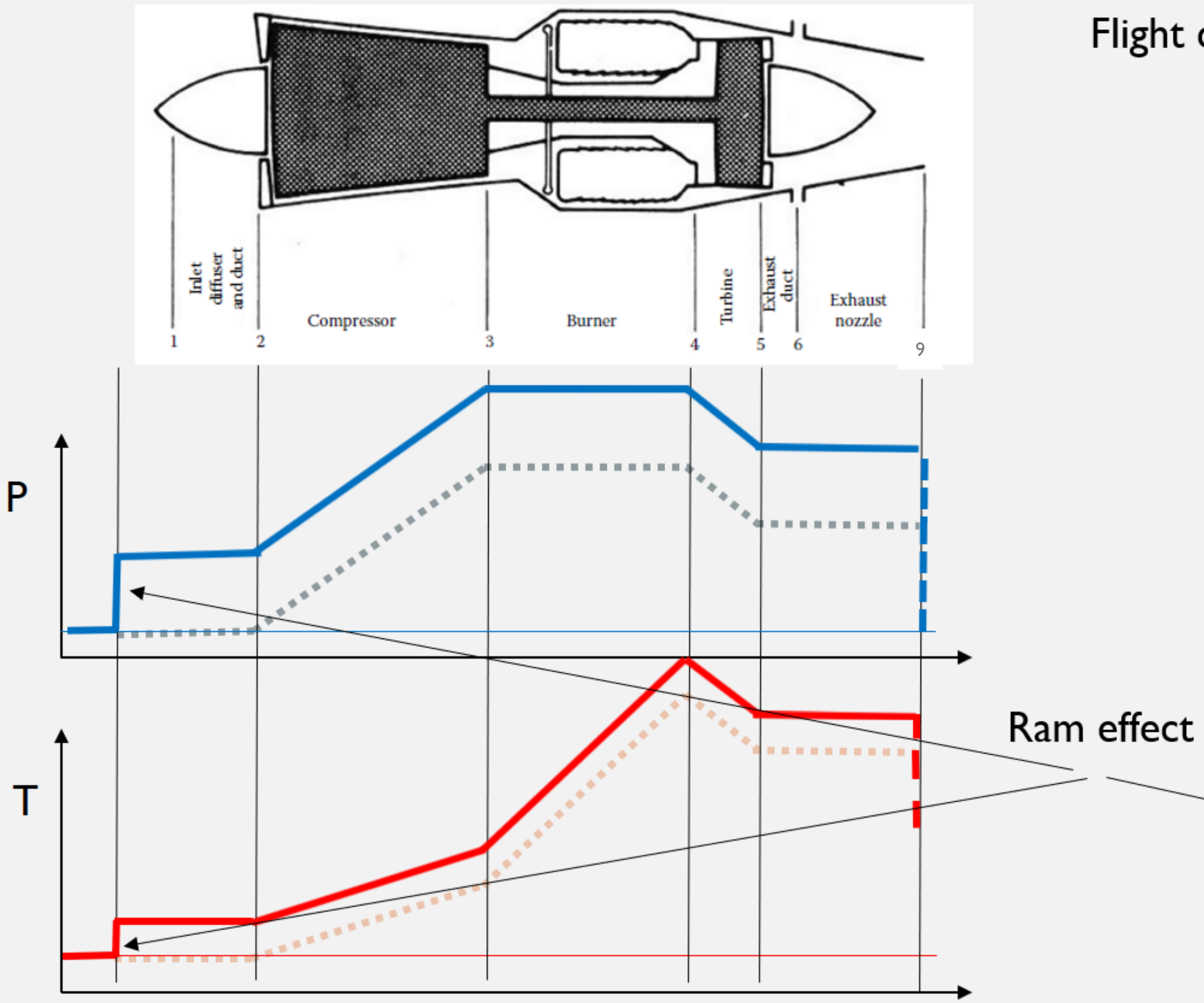
Entropy increase in a burner $\Delta s_B = cp_B ln \frac{T_{t4}}{T_{t3}}$

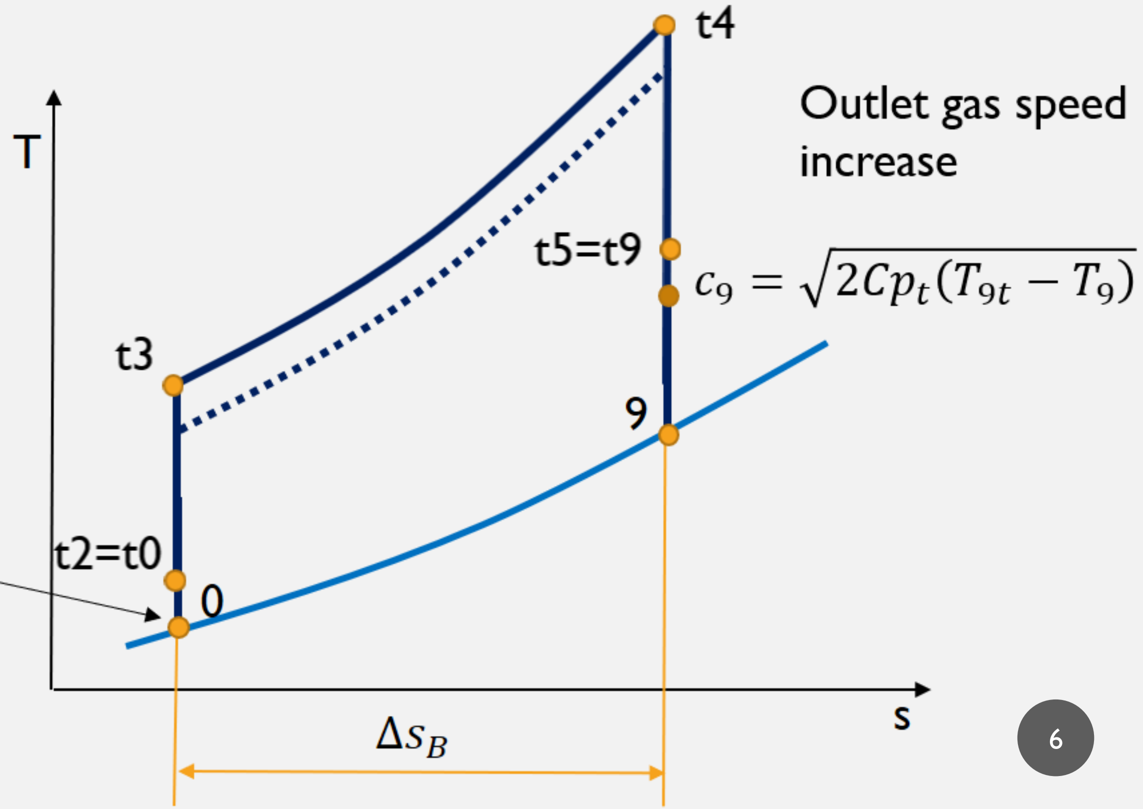
IDEAL TURBOJET ENGINE

Flight condition V0>0

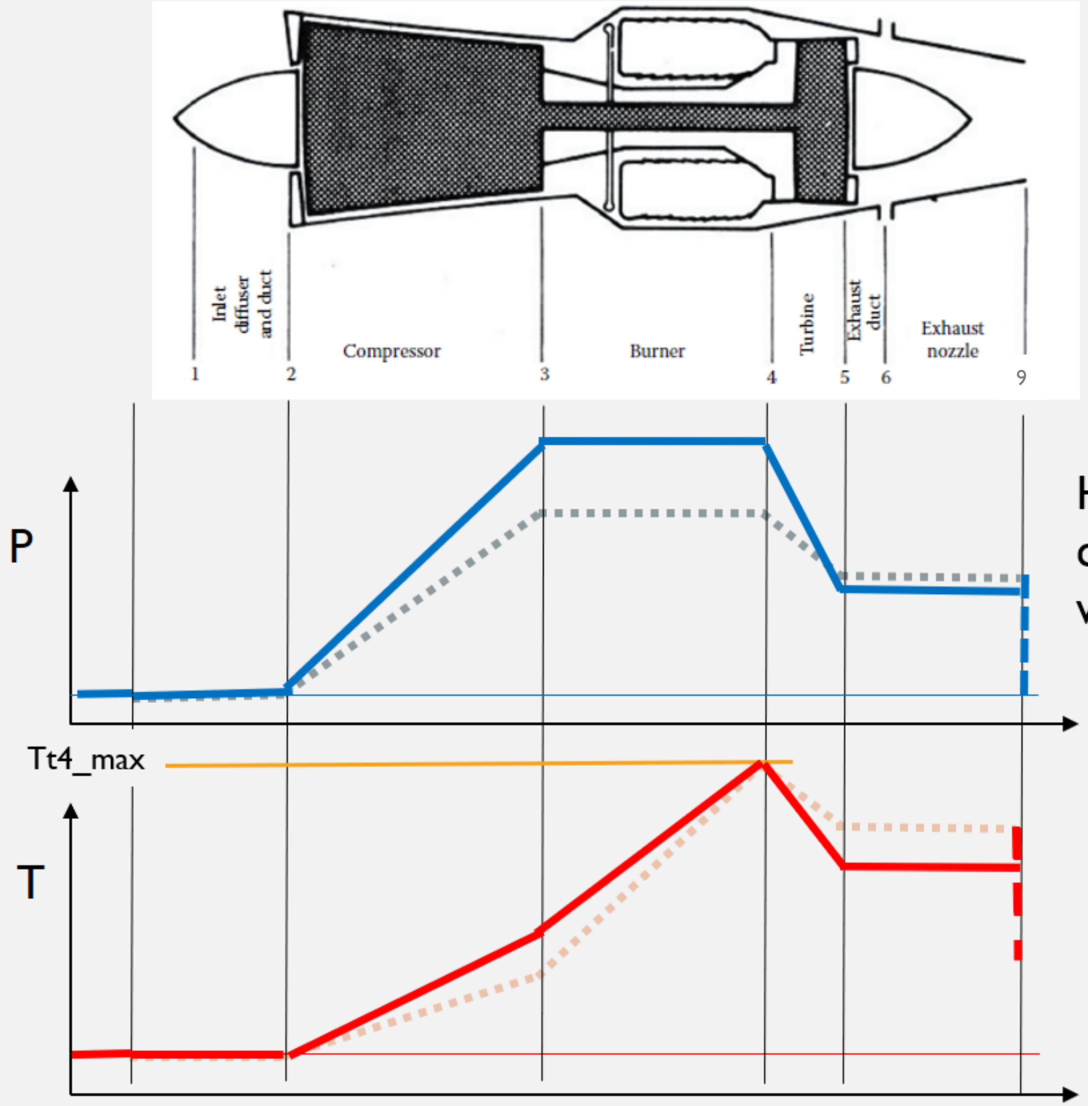
$$P_{t0} = P_0\left(1 + \frac{k-1}{2}M_0{}^2\right)^{k/(k-1)}$$

$$T_{t0} = T_0\left(1 + \frac{k-1}{2}M_0{}^2\right)$$

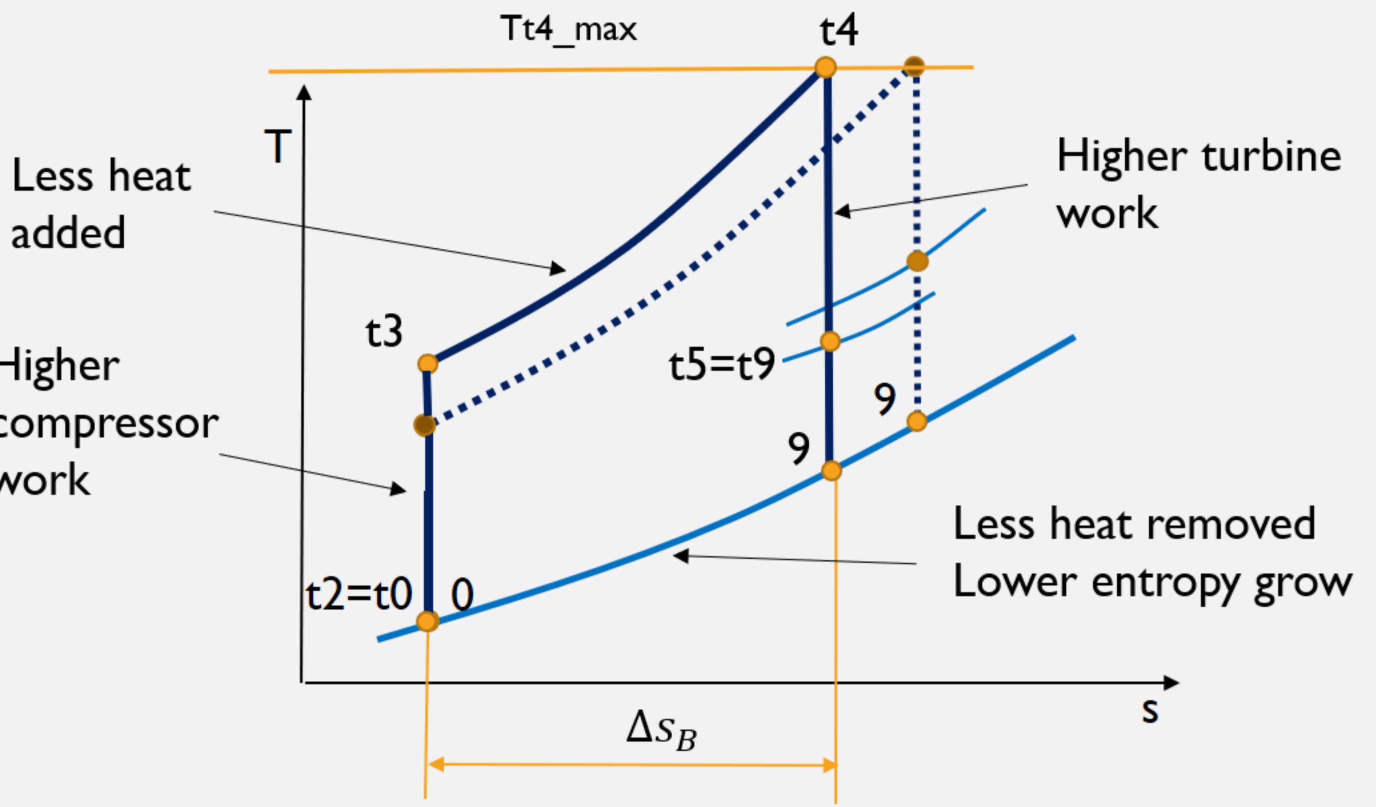




IDEAL TURBOJET ENGINE – COMPRESSOR PRESSURE RATIO INFLUENCE



CPR (compressor pressure ratio) growing and Tt4 is limited Tt4_max

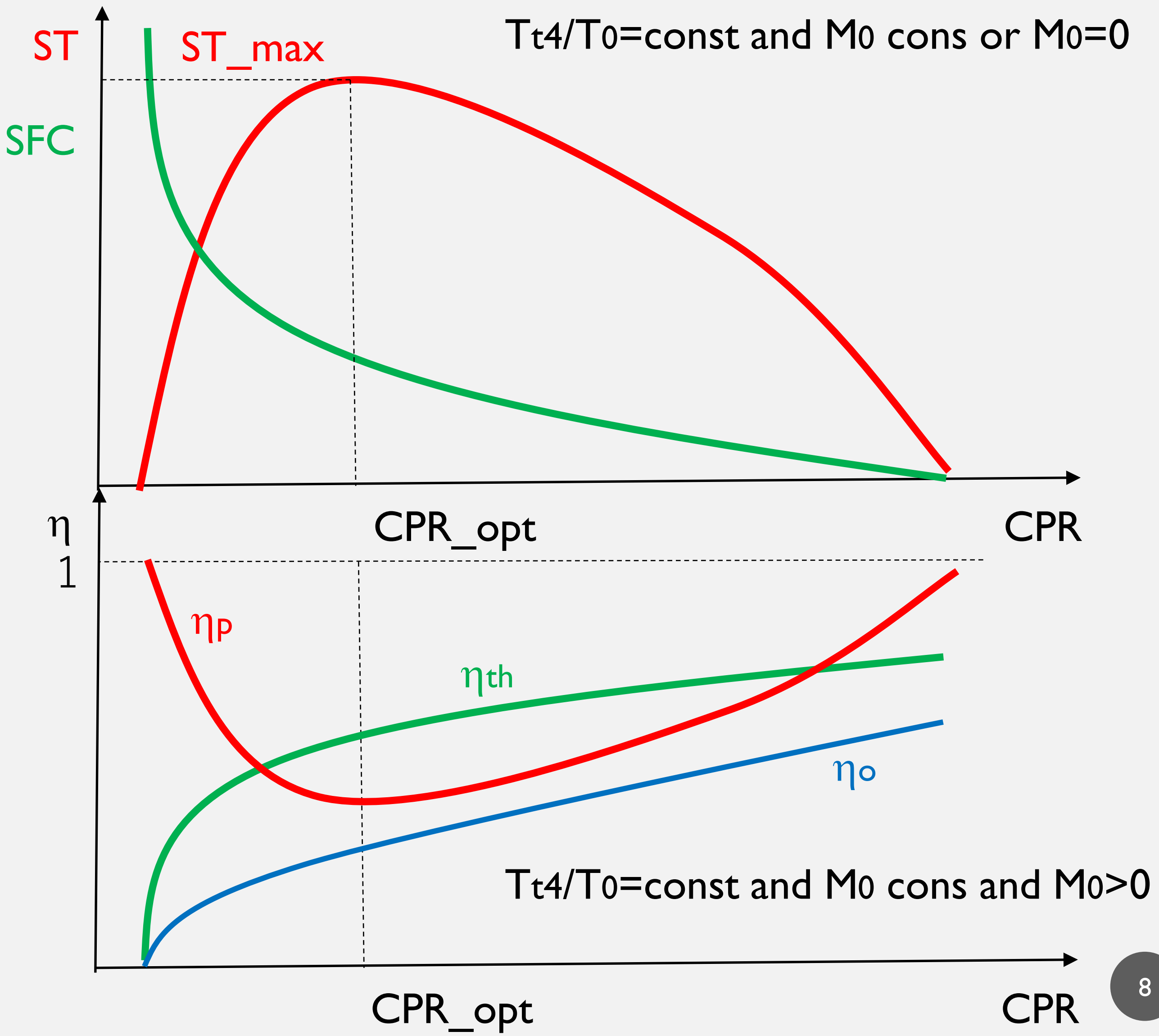


Engine thermal efficiency grow (less heat removed)
Lower fuel consumption (less heat added) and lower SFC

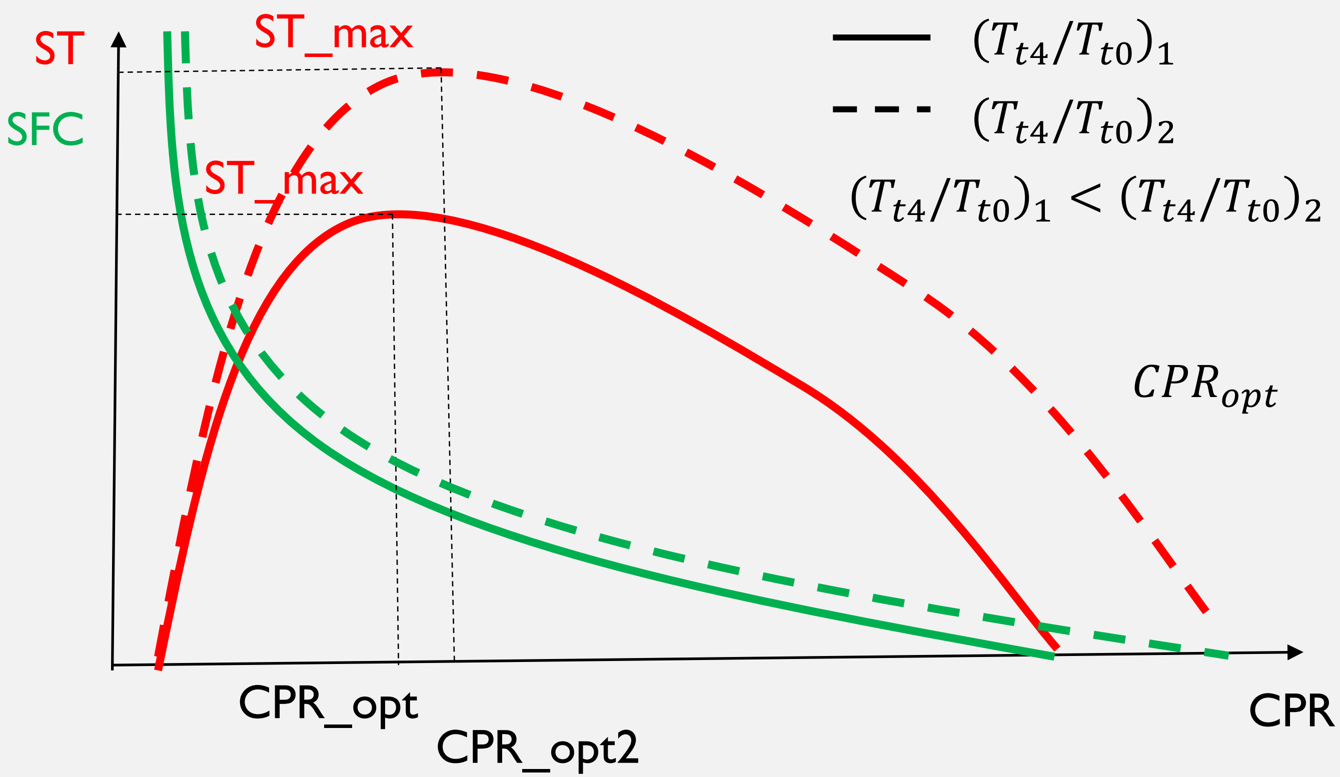
IDEAL TURBOJET CYCLE OPTIMISATION

SUMMARY:

- Specific thrust (ST) grows with compressor pressure ratio increasing, achieves maximum for optimal CPR than is goes down
- Specific fuel consumption decreases with CPR growing
- Propulsive efficiency as a function of CPR represents oposit relation to ST, it is minimal for optimal CPR and achieves 1 for ST=0.
- Thermal and overall efficiencies grow with CPR
- The presented relationship between efficiencies and CPR are right for M0>0, for M0=0, propulsive and overall efficiency are 0



IDEAL TURBOJET CYCLE OPTIMISATION FOR DIFFERENT ENGINE TEMPERATURE RATIO



For ideal cycle:

$$CPR_{opt} = T_{t4}/T_{t0}^{\frac{k}{2(k-1)}} \qquad CPR_{max} = CPR_{opt}^{\ 2}$$

SUMMARY:

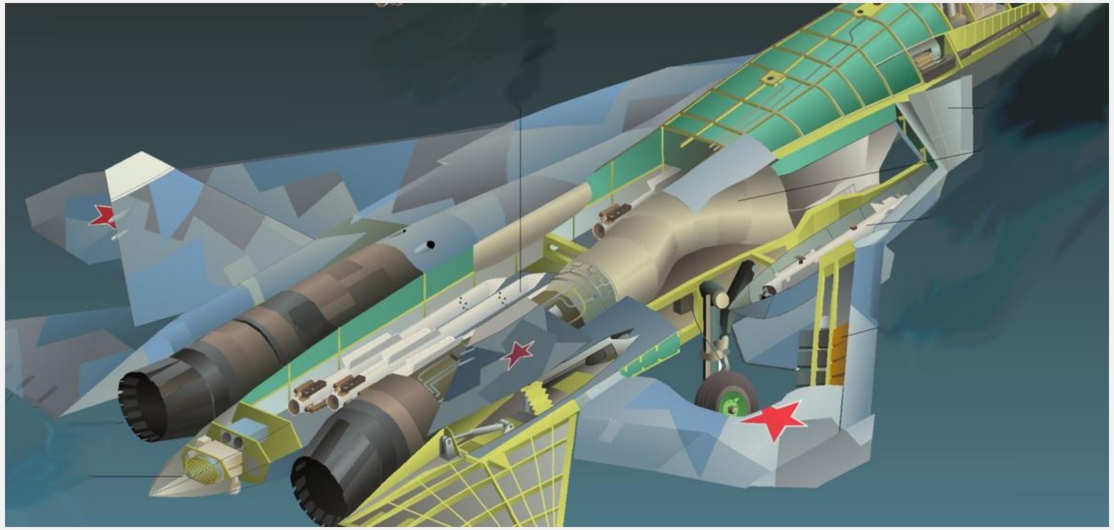
- Specific thrust (ST) is higher for higher engine themperature ratio Tt4/Tt0 and achieve ST_max for higher CPR (higher CPR_opt)
- Specific fuel consumption decreases with CPR growing, but for highrt Tt4/Tt0 is higher
- Range of available CPR increases for higher Tt4/Tt0

Tt4/Tt0	CPR_opt	CPR_max
4	11,3	128
5	16,7	279,5
6	23	529

REAL TURBOJET ENGINE

ENGINE INLET / AIR INTAKE

All **air-breathing jet engines** installed on aircraft must be equipped with an **air inlet** and a **diffuser duct**. Its role is to provide a uniform and stable air supply to ensure efficient engine operation across different flight conditions.



Key functions:

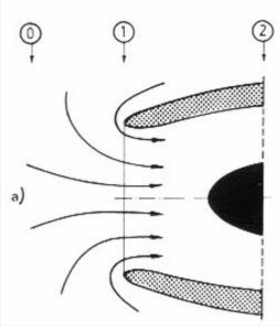
- Capture the required mass flow of air into the engine.
- Decelerate the flow with minimal total pressure loss at high flight speed.
- Ensure that a sufficient mass flow rate of air is drawn into the engine at static engine operation.
- Provide a uniform velocity profile at the compressor face.
- Ensure stable airflow under varying flight conditions.

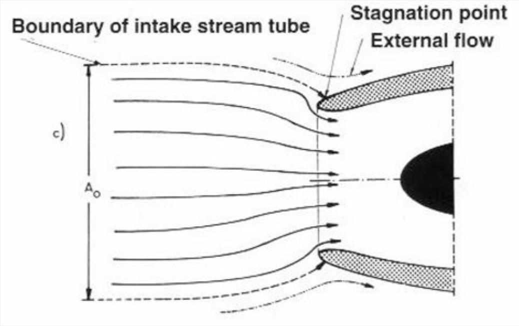
CLASSIFICATION OF JETENGINE INTAKES

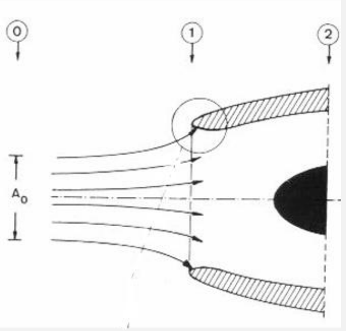
- **Subsonic intakes —** flight Mach number < 1

- **Supersonic intakes —** flight Mach number > 1

OPERATION OF A SUBSONIC INLET



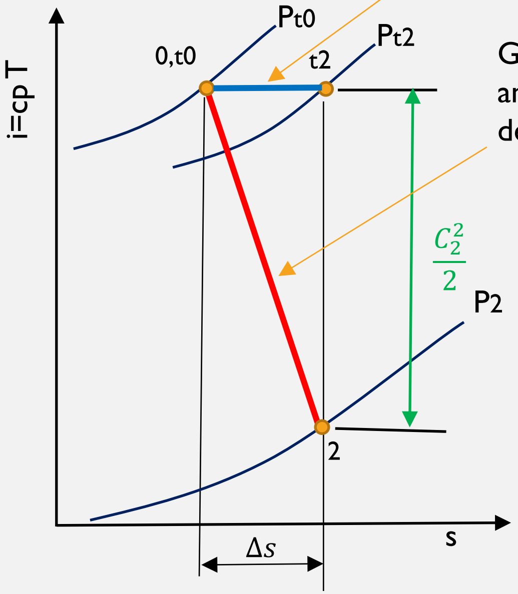




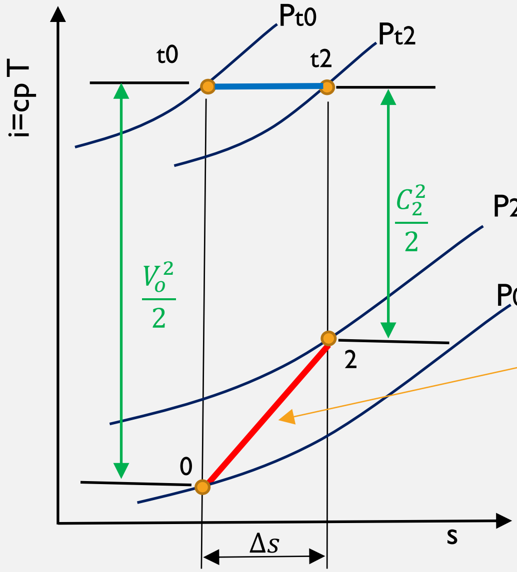
Operating regimes:

- Static operation
- Flight speed lower than design speed
- Flight speed higher than design speed

Const. enthalpy in the inlet



Gas flow acceleration and static pressure decrease

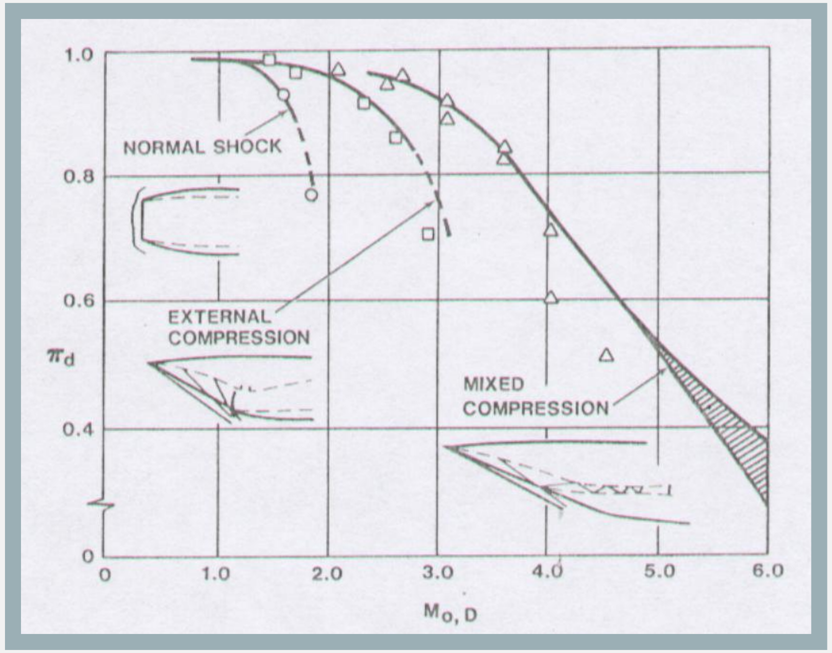


Deceleration of the flow in the inlet and an increase in static pressure

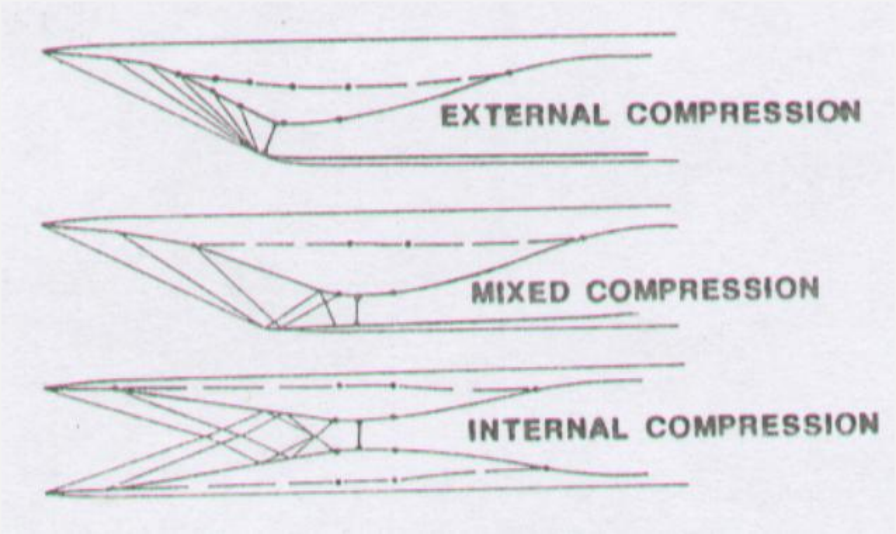
SUPERSONIC INTAKES FOR HIGHER MACH NUMBERS

These intakes generate:

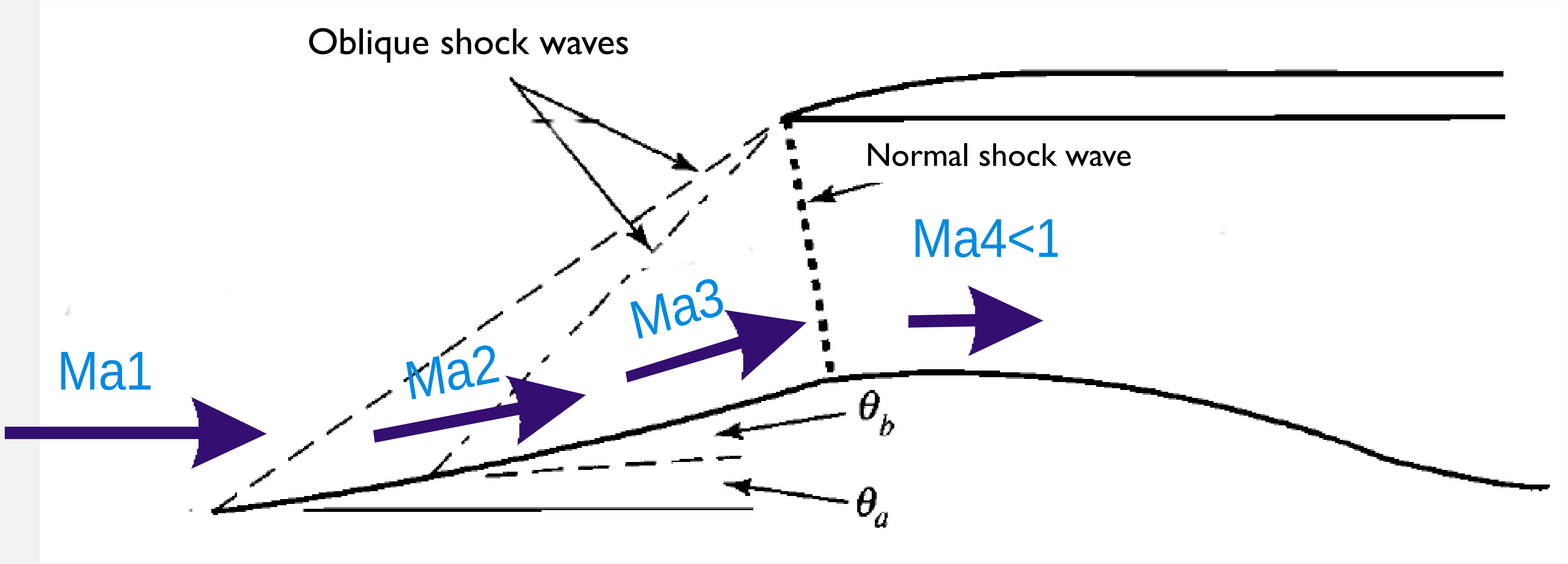
- Oblique shock waves
- A terminal normal shock at the throat
- Subsonic flow downstream of the shock system



Types of supersonic intakes:



PRINCIPLE OF OPERATION OF AN EXTERNAL-COMPRESSION SUPERSONIC INLET



Isentropic pressure rise caused by flight speed

$$P_{t1} = p_0(1 + \frac{k-1}{2} {Ma_1}^2)^{\frac{k}{k-1}}$$

$$Ma_1 > Ma_2 > Ma_3 > 1 > Ma_4$$

$$P_{t1} > P_{t2} > P_{t3} > P_{t4}$$
$$p_1 < p_2 < p_3 < p_4$$

- Deceleration of the incoming flow through a system of shock waves
- Total pressure losses across shocks
- Increase in static pressure

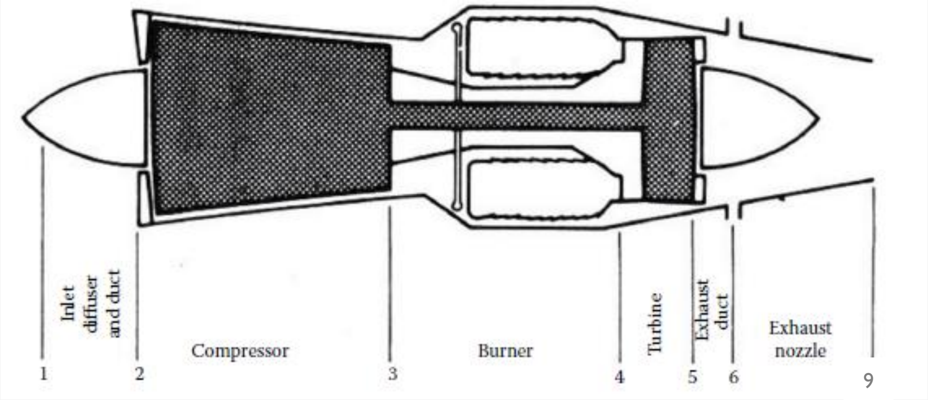
Detailed information in: Ahmed F. El-Sayed, Aircraft Propulsion and Gas Turbine Engines, chapter 9.7.1

INLET PRESSURE AND TEMPERATURE

Engine work in static conditions V0=0 → $P_{t0} = P_0, \quad T_{t0} = T_0$

Ram pressure recovery for flight condition $(M_0 > 0)$

$$P_{t0} = P_0\left(1 + \frac{k-1}{2}{M_0}^2\right)^{k/(k-1)} \quad T_{t0} = T_0\left(1 + \frac{k-1}{2}{M_0}^2\right)$$



INLET pressure losses → $P_{t2} = \pi_D P_{t0}$

No thermal losses $T_{t2} = T_{t0}$

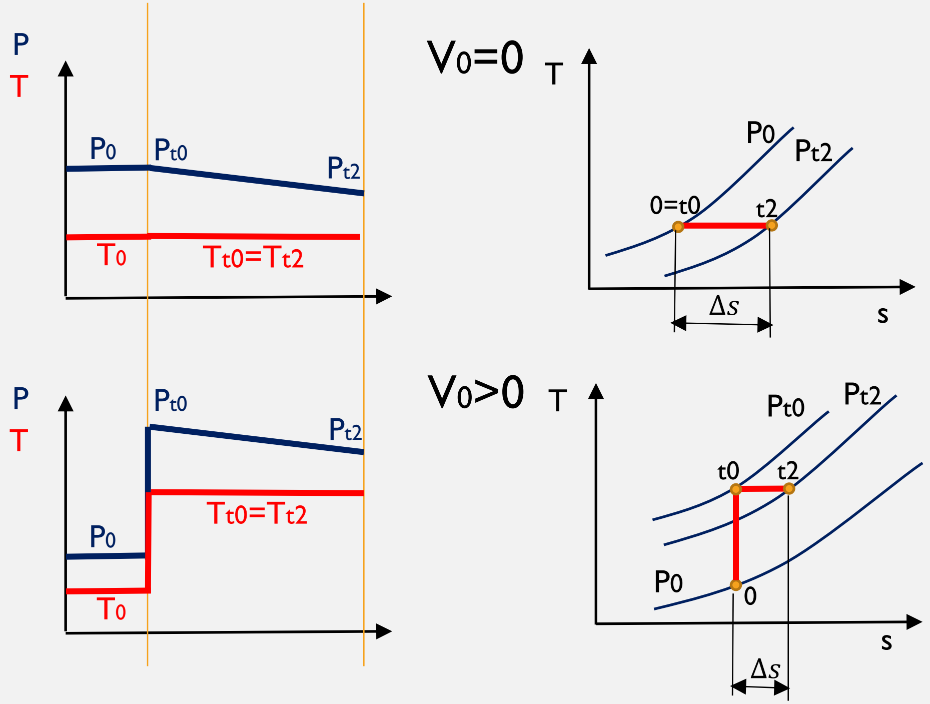
Typical range:

$\pi_D = 0{,}95 - 0{,}99, \quad \pi_D = P_{t2} / P_{t0}$

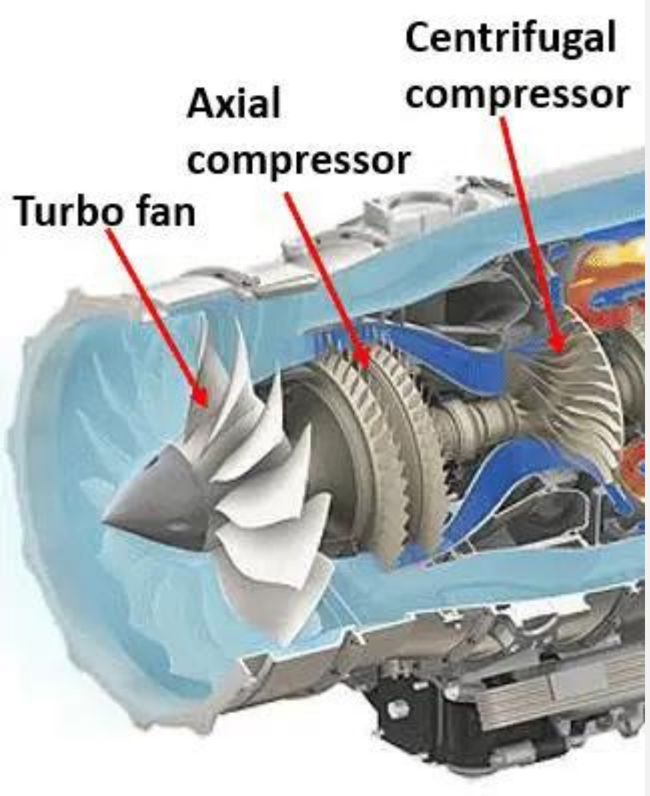
π_D is lower for high supoersonic speed

Pressure losses are visible in entropy grow:

$$\Delta s = -R\ln(P_{t2}/P_{t0}) = R\ln(1/\pi_D)$$

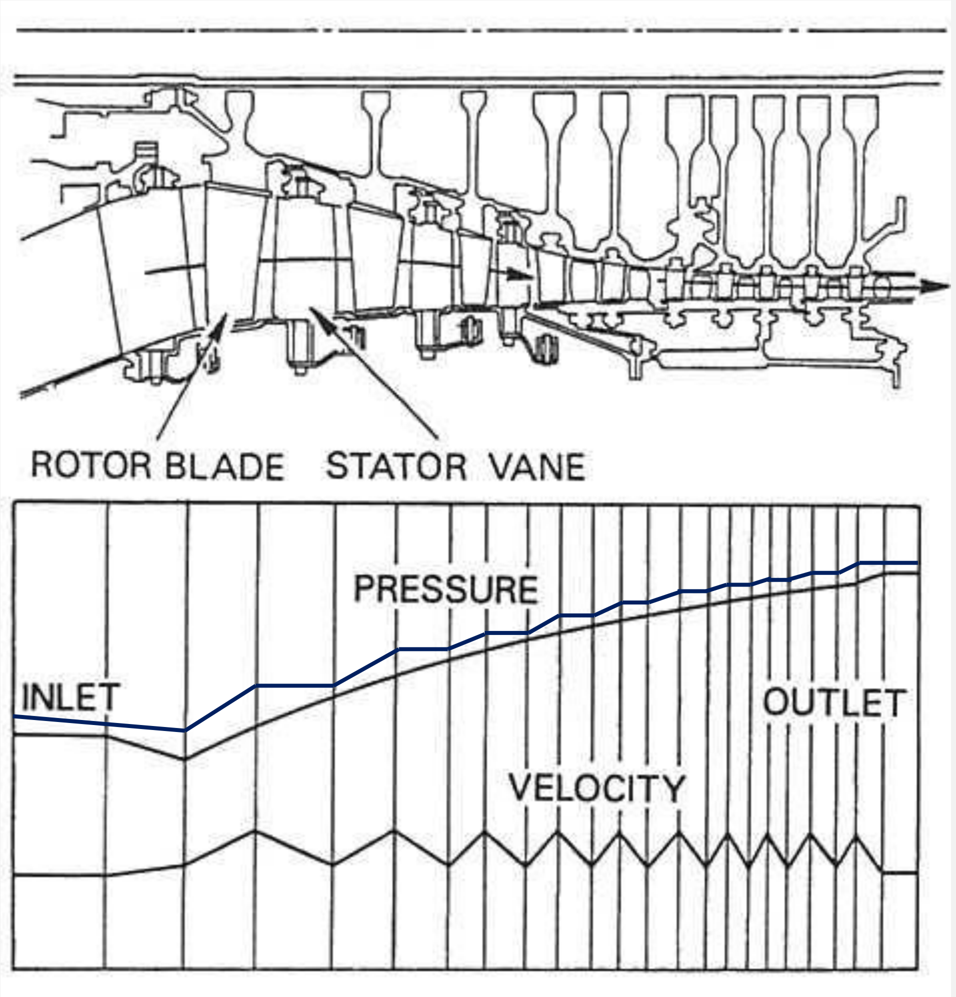


COMPRESSORS AND FANS



- The role of the compressor and fan is to compress the incoming air while consuming as little power as possible — this defines compression efficiency.
- A fan compresses both the bypass and core streams and is an axial-flow machine.
- A compressor is located in the core engine. It may be axial, centrifugal (radial), or mixed (axial-centrifugal).

PARAMETER VARIATION IN A MULTISTAGE COMPRESSOR



- Total pressure increases across the rotor and slightly decreases across the stator due to losses.
- Static pressure increases in both rotor and stator.
- Absolute velocity increases in the rotor and decreases in the stator.

AXIAL COMPRESSOR STAGE

Work of a stage

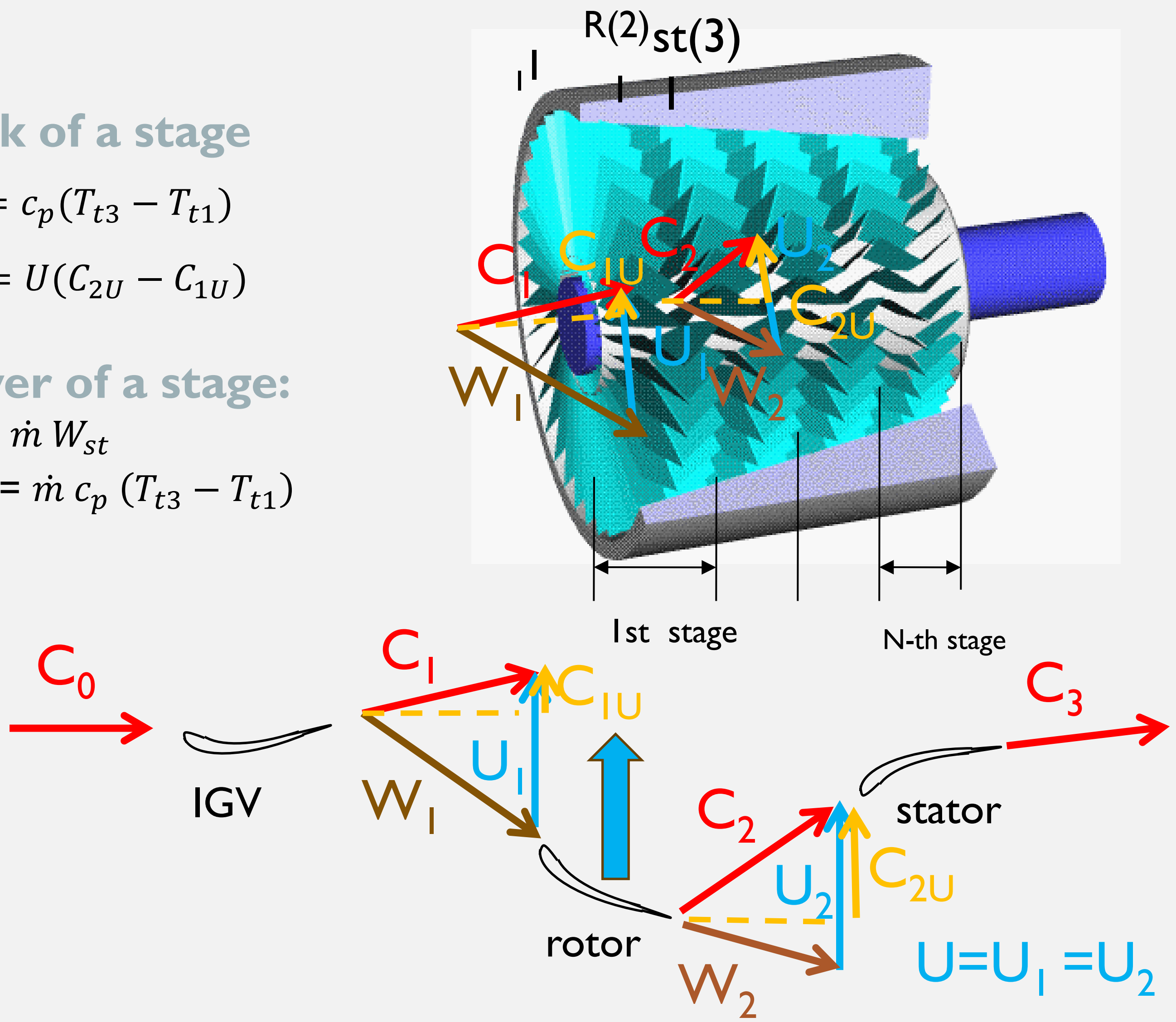
$$W_{st} = c_p(T_{t3} - T_{t1})$$

$$= U(C_{2U} - C_{1U})$$

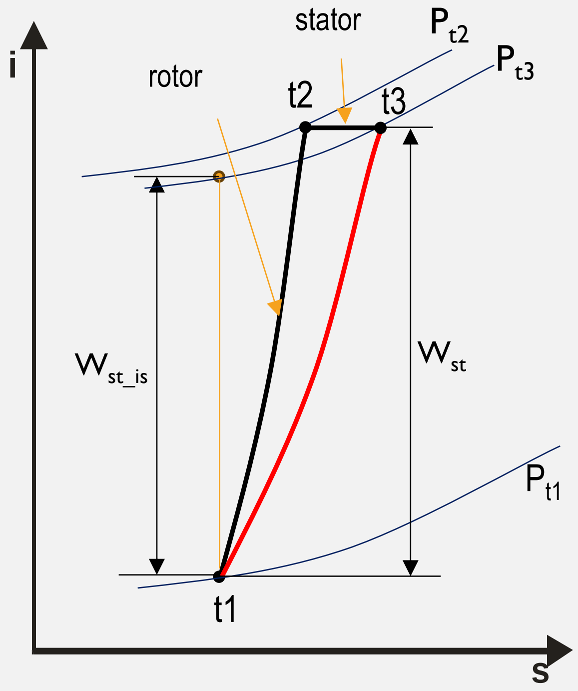
Power of a stage:

$$P_{st} = \dot{m}\, W_{st}$$

$$= \dot{m}\, c_p\, (T_{t3} - T_{t1})$$



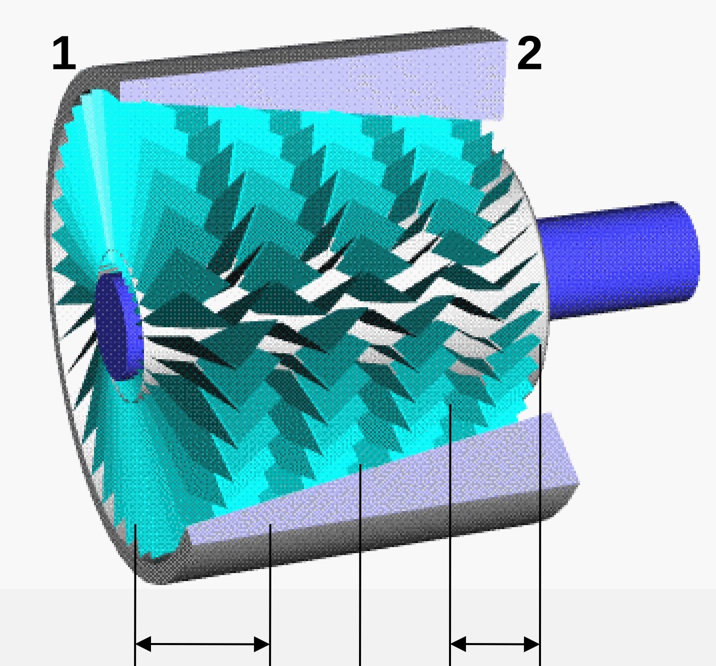
Compressor stage



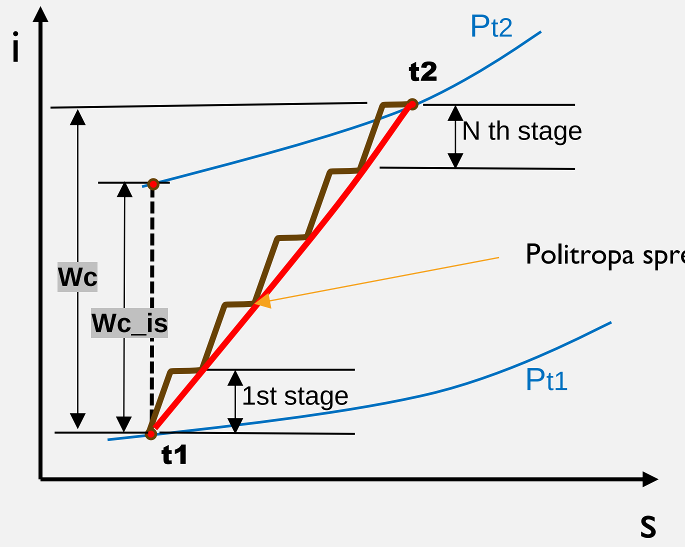
Stage efficiency:

$$\eta_{st} = \frac{W_{st\_is}}{W_{st}} = \frac{i_{t\_st\_is} - i_{t1}}{i_{t\_st} - i_{t1}}$$

MULTISTAGE COMPRESSOR



Compressor work



Compressor efficiency:

$$\eta_C = \frac{W_{C\_is}}{W_C} = \frac{i_{t2\_is} - i_{t1}}{i_{t2} - i_{t1}}$$

Compressor work:

$$W_C = \sum_{i=1}^{n} W_{st} = c_p(T_{t2} - T_{t1})$$

Compressor pressure ratio:

$$\pi_C = \prod_{i=1}^{n} \pi_{st}$$

COMPRESSOR CALCULATIONS (2–3)

COMPRESSOR (2 – 3)

CPR – compressor pressure ratio

$$\pi_C = \frac{P_{t3}}{P_{t2}} = \text{CPR}$$

Entropy rise due to losses:

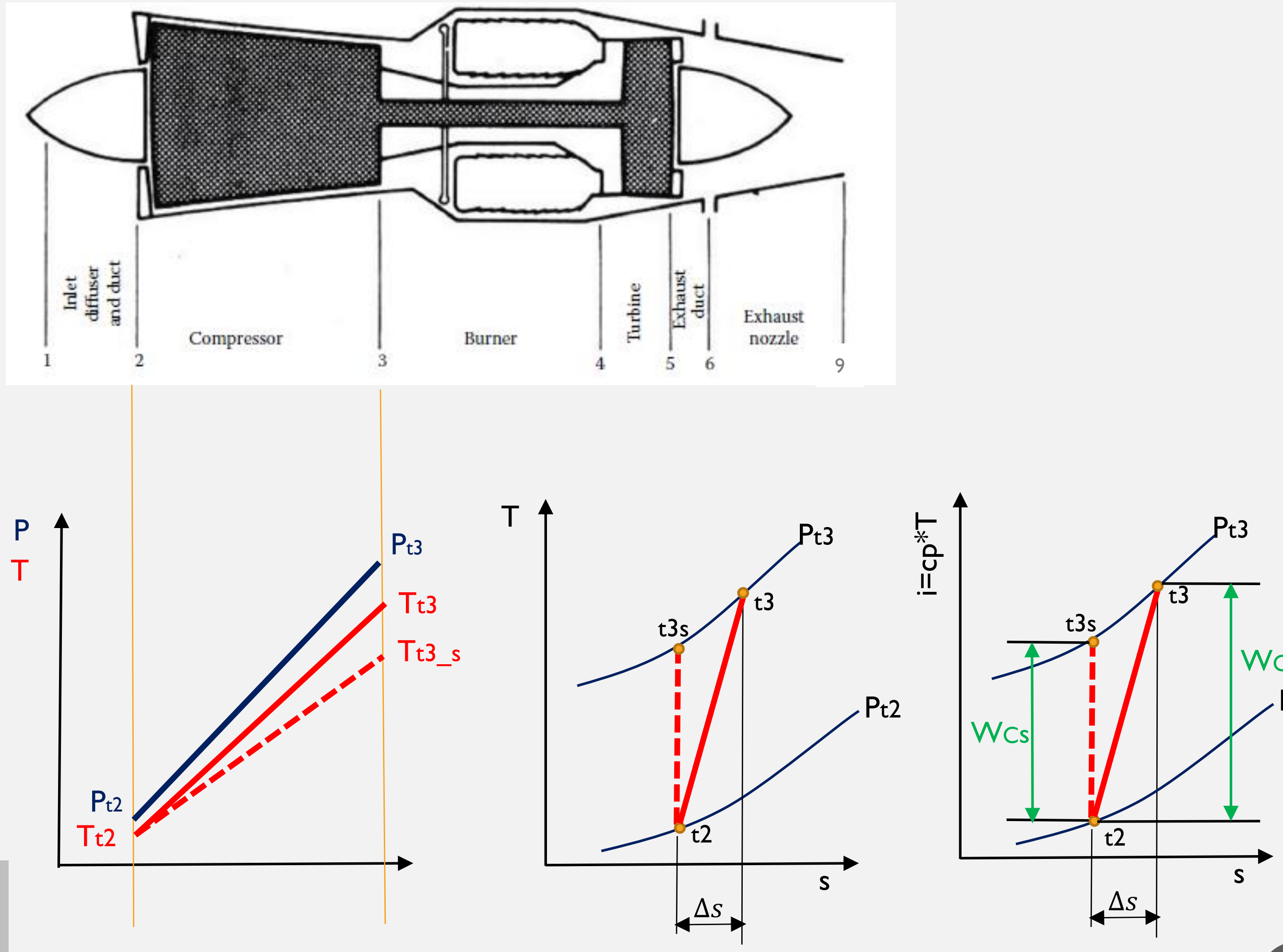
$$\Delta s = Cp * ln(T_{t3}/T_{t2}) - Rln(P_{t3}/P_{t2}))$$

Isentropic efficiency

$$\eta_C = \frac{W_{Cs}}{W_C} = \frac{Cp(T_{t3s} - T_{t2})}{Cp(T_{t3} - T_{t2})} = \frac{T_{t3s}/T_{t2} - 1}{T_{t3}/T_{t2} - 1}$$

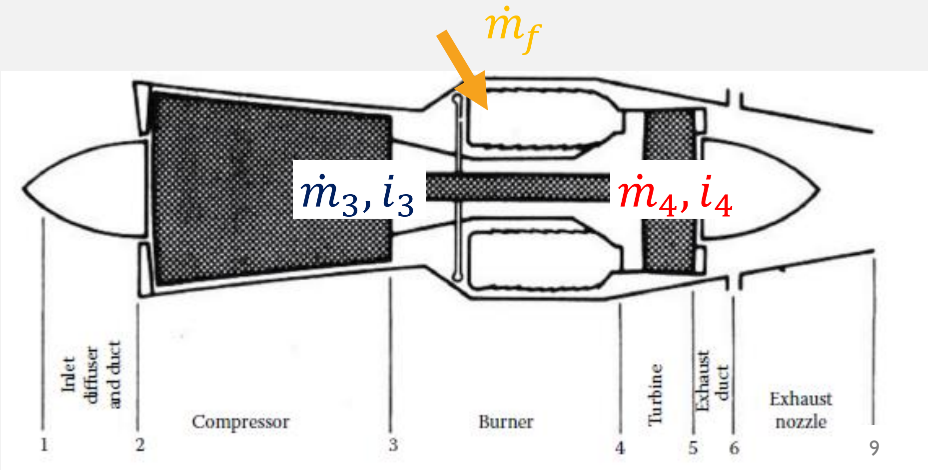
$$= \frac{(P_{t3}/P_{t2})^{\frac{k-1}{k}} - 1}{T_{t3}/T_{t2} - 1} = \frac{(\pi_C)^{\frac{k-1}{k}} - 1}{T_{t3}/T_{t2} - 1}$$

Compressor efficiency describes the relationship between the temperature rise and pressure rise across the compressor.



BURNER/COMBUSTER/COMBUSTION CHAMBER

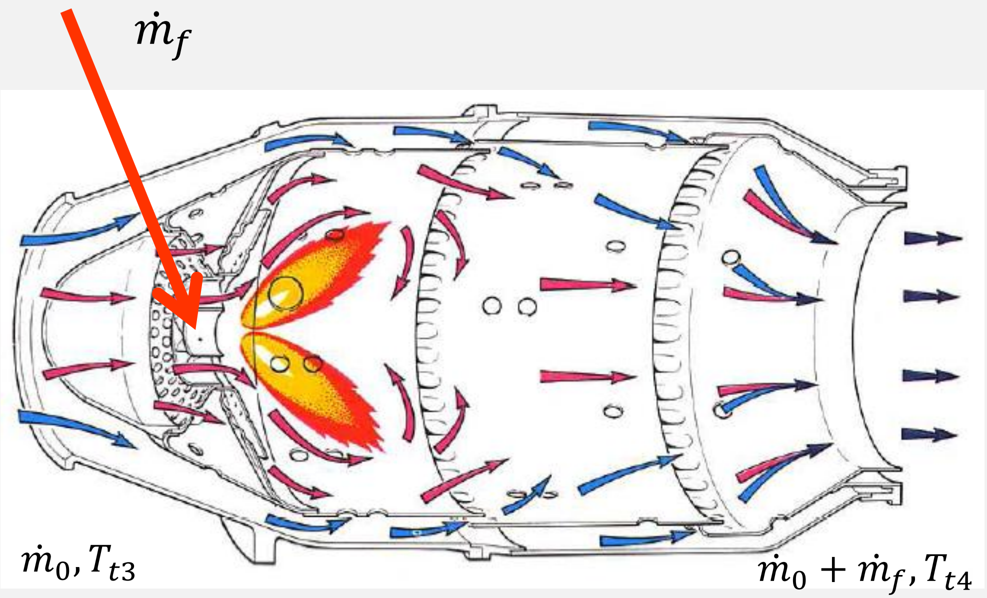
The combustion chamber in a gas turbine engine is located between the compressor and turbine assemblies.
Its role is to efficiently convert the chemical energy of the fuel into thermal energy of the working gas while minimizing flow losses (pressure losses).



ENERGY BALANCE IN THE COMBUSTION CHAMBER

Energy Balance

$$\dot{Q}_B = \Delta \dot{I}_B = \bar{c}_p\left((\dot{m}_0+\dot{m}_f)T_{t4} - \dot{m}_0 T_{t3}\right) \approx \dot{m}_0\bar{c}_p(T_{t4}-T_{t3})$$



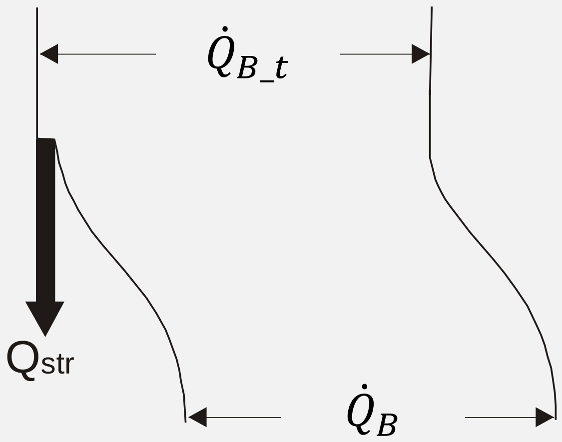
$\dot{Q}_{B\_t}$ - Theoretical heat released by the fuel

$\dot{Q}_B$ - The actual heat transferred to the airflow in the combustor

$f_B = \dot{m}_f / \dot{m}_0$ - Fuel–air ratio determines the relative fuel consumption

Thermal efficiency of the combustor defines how effectively the supplied fuel energy increases the gas enthalpy.

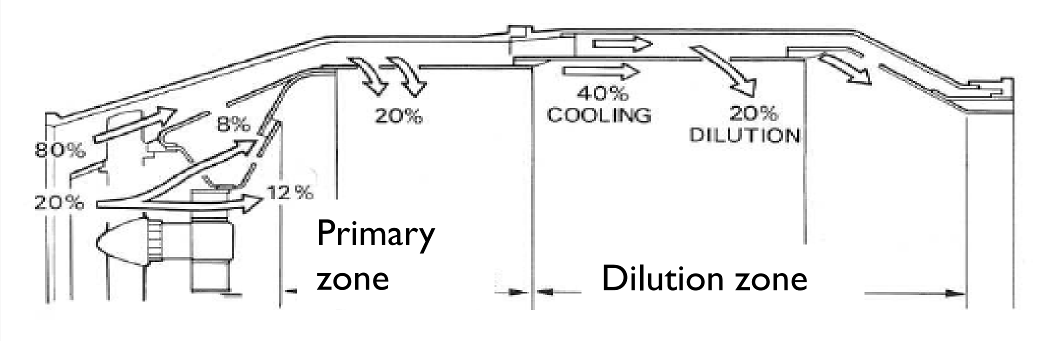
$$\eta_B = \frac{\dot{Q}_B}{\dot{Q}_{B\_t}} = \frac{\dot{m}_0\, \bar{c}_p(T_{t4}-T_{t3})}{\dot{m}_f FHV} = \frac{\bar{c}_p(T_{t4}-T_{t3})}{FHV\ f_{pal}}$$

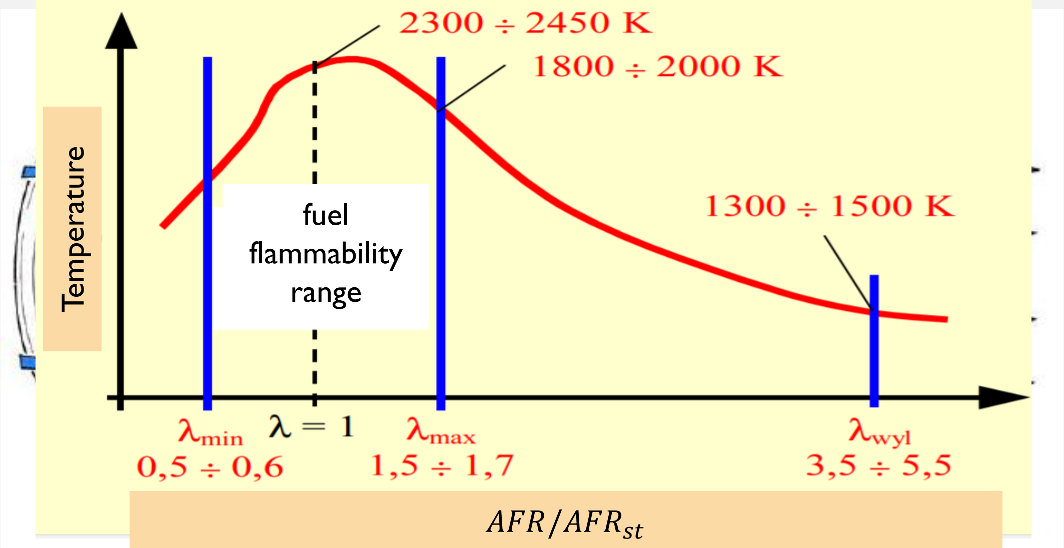


FHV - Fuel heat value

$\bar{c}_p$ - Specific heat at constant pressures

ORGANIZATION OF THE COMBUSTION PROCESS





Fuel: **aviation kerosene**

mass fractions

$$C = 0{,}86, H = 0{,}14$$

Stoichiometric oxygen requirement,

$$O_t = \frac{8}{3} \cdot 0.86 + 8 \cdot 0{,}14 = 3{,}413 [\frac{kg O_2}{kg\ fuel}]$$

Stoichiometric air–fuel ratio,

$$AFR_{st} = \frac{O_t}{0{,}232} = 14{,}7 [\frac{kg\ air}{kg\ fuel}]$$

The fuel-to-air mass ratio for a stoichiometric mixture is therefore approximately 0,067, whereas in a single-flow gas turbine engine, fB≈0,02.This means that the amount of air relative to the amount of fuel is about three times greater than required by the combustion balance. Consequently, the exhaust gases from a gas turbine engine contain approximately 30% combustion products, while the remainder consists of unreacted air.

The amount of air supplied to the primary zone is about 1/15 of the total air entering the combustor.

FLOW LOSSES IN THE COMBUSTION CHAMBER

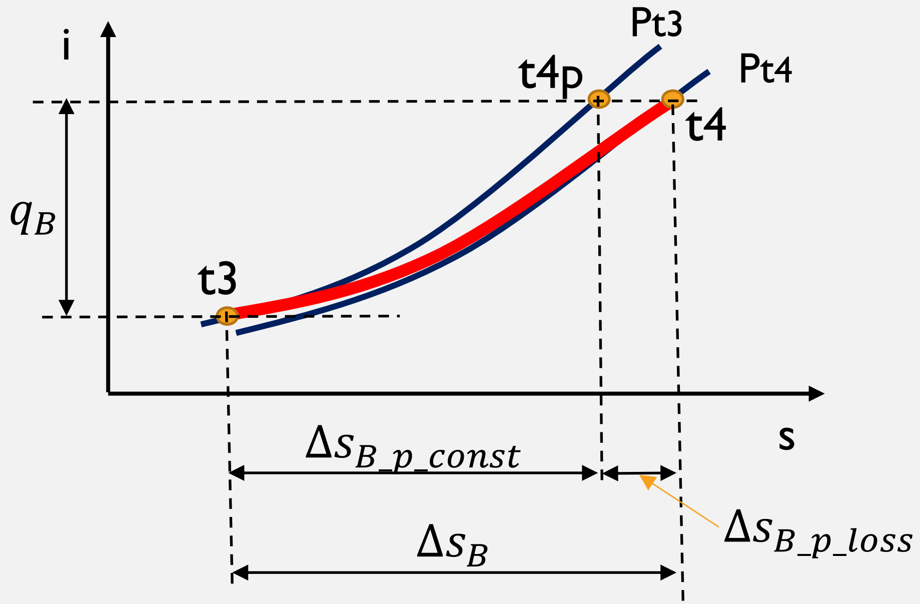
Pressure losses occur due to:

- Aerodynamic friction
- Mixing processes
- Heat addition

$\pi_B = \pi_{B\_M} \cdot \pi_{B\_T}$ - Burner pessure losses coefficient:

π_{B_M} - Mechanical lossese coefficient

π_{B_T} - Thermal lossese coefficient



Pressure losses:

$$\pi_B = \frac{P_{t4}}{P_{t3}}$$

$$q_B = \dot{Q}_B / \dot{m}_0$$

Entropy increase in a burner:

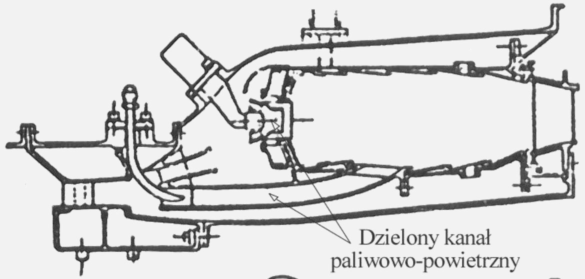
Isobaric entropy rise $\Delta s_{B\_p\_const} = cp_B ln \frac{T_{t4}}{T_{t3}}$

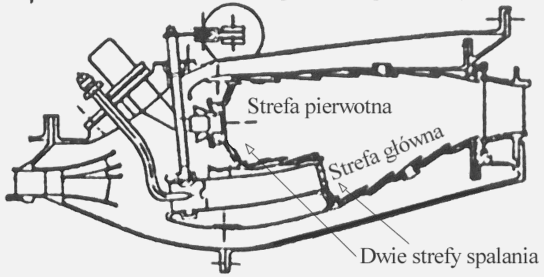
Pressure losses entropy rise $\Delta s_{B\_p\_loss} = -R_t ln \frac{P_{t4}}{P_{t3}}$

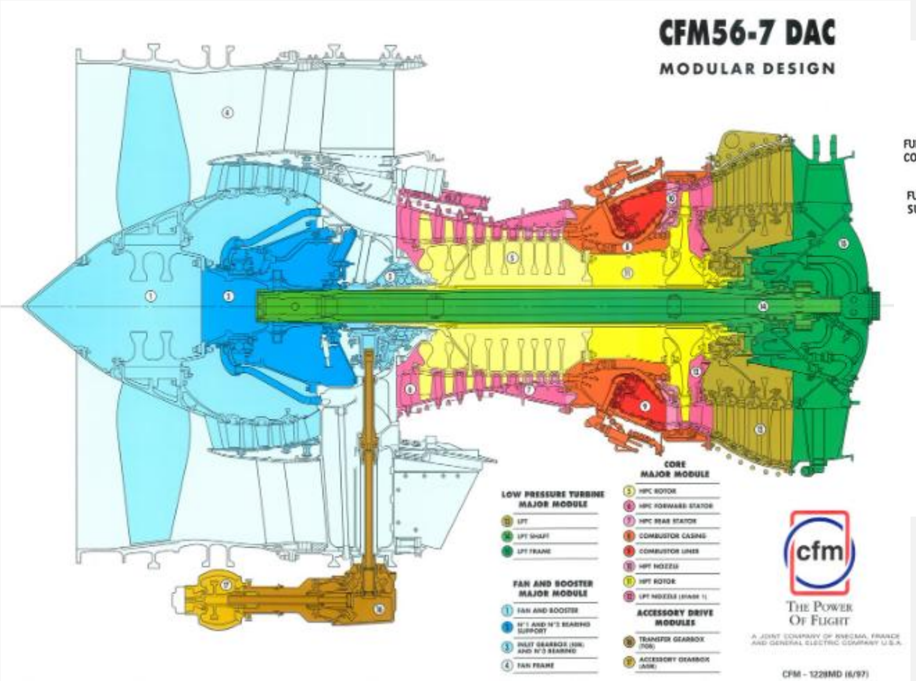
Overall entropy rise $\Delta s_B = cp_B ln \frac{T_{t4}}{T_{t3}} - R_t ln \frac{P_{t4}}{P_{t3}}$

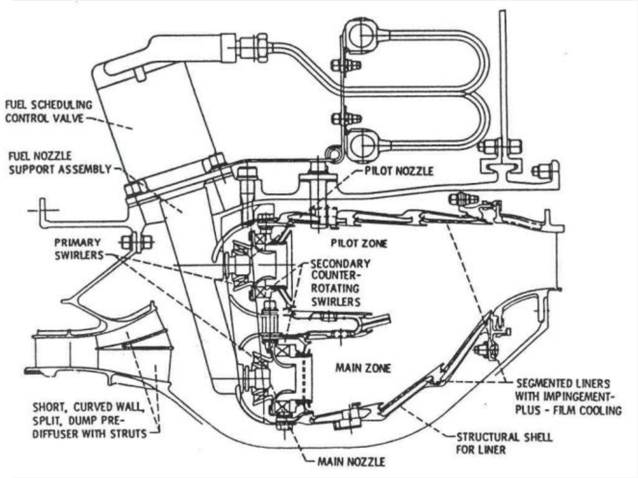
MODERN DUAL ANNULAR COMBUSTOR (DAC)

double-annular combustor arrangements:
- Series configuration
- Parallel configuration



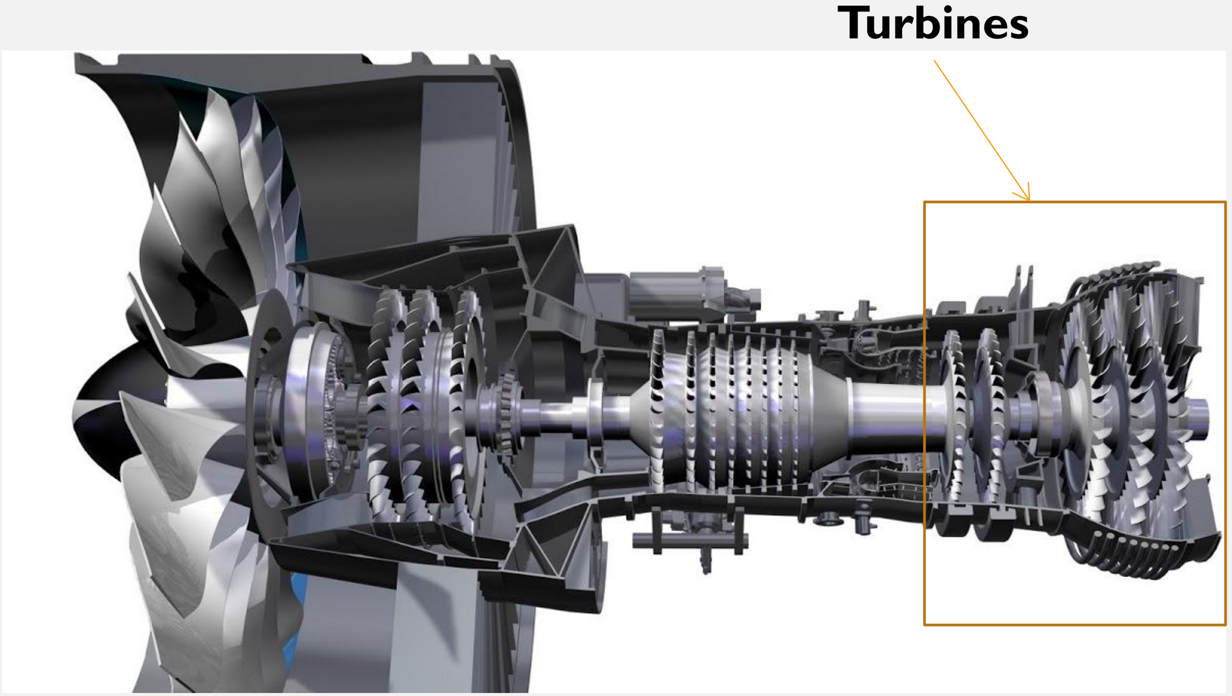






„Double-annular burner" (double-annular combustor, DAC). A dual-annular combustor contains two concentric combustion zones:an outer pilot zone operating at low thrust,an inner main zone activated at higher thrust levels.This design improves combustion stability at low power and reduces NOx emissions at high power compared to single-annular combustors.

TURBINE IN AN AIRCRAFT ENGINE



Turbine rotor blades

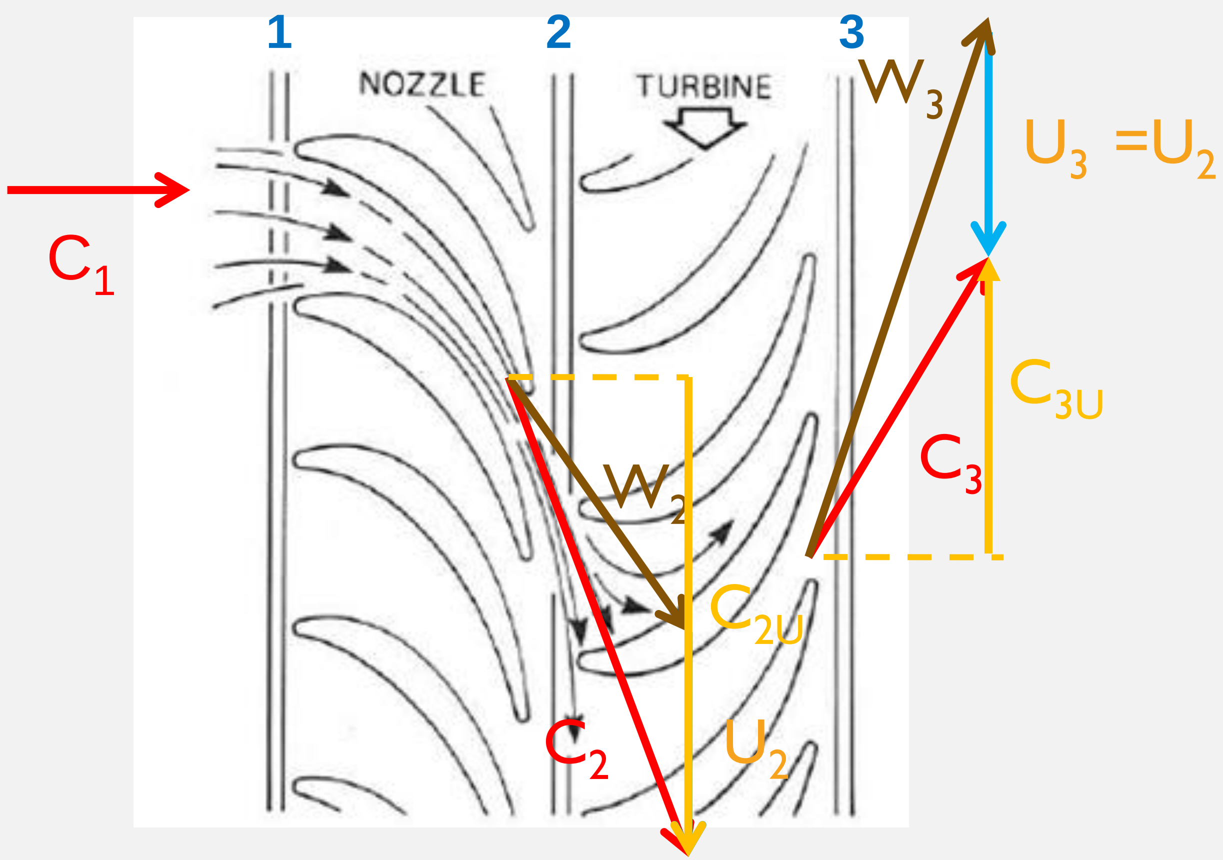
A turbine may consist of one to three turbine modules, each containing one or more stages.
Large high-bypass turbofan engines can extract 30–50 MW of mechanical power.
A single turbine blade can produce power comparable to that of a sports-car engine. Turbine inlet temperatures exceed 1800 K.

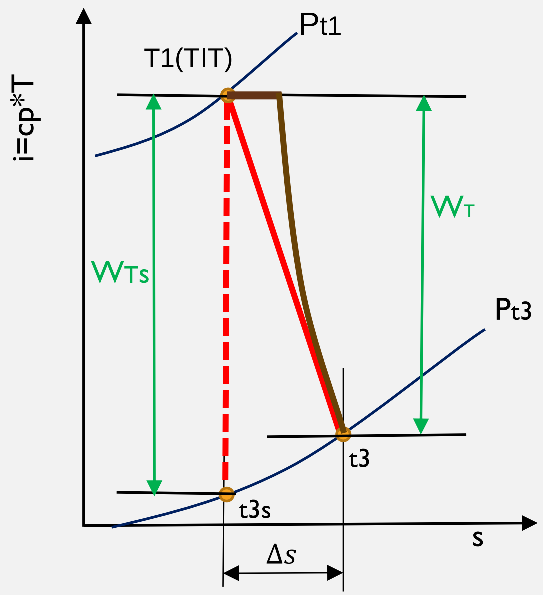
AIRCRAFT ENGINE TURBINES

Turbines convert the thermal and kinetic energy of the hot gases into mechanical power used to drive:

- **compressors,**
- **the fan,**
- **engine accessories (oil pumps, fuel pumps, generators)**
- **and, in turboshaft and helicopter engines, the main output shaft producing thrust or lift.**

OPERATION OF A TURBINE STAGE





Turbine stage work

W= $u_2(c_{2u} + c_{3u})$

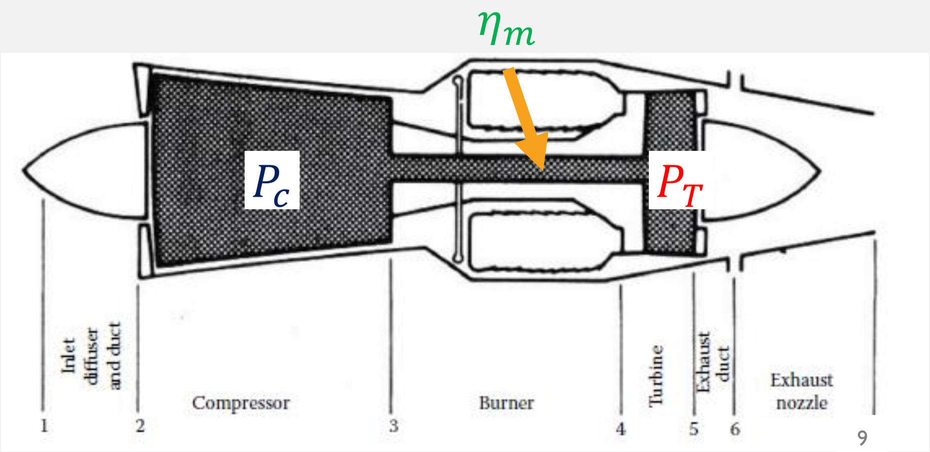
Turbine stage work

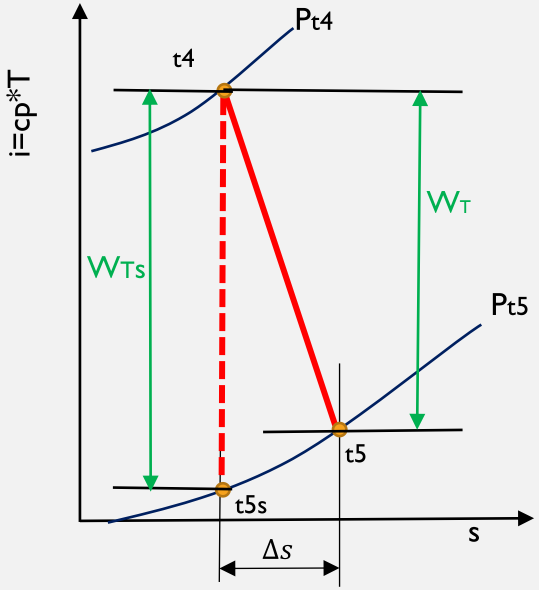
W= $c_p(T_{t1} - T_{t2})$

Turbine efficiency

$$\eta_T = \frac{W_T}{W_{Ts}}$$

TURBINE





Compressor-turbine energy balance:

$$P_T = \dot{m}_4 Cp_T (T_{t4} - T_{t5}) = \frac{1}{\eta_m} P_C$$

Turbine outlet temperature

$$T_{t5} = T_{t4} - \frac{P_C}{\eta_m \dot{m}_4 Cp_T}$$

for: $\dot{m}_2 = \dot{m}_0$ and $\dot{m}_4 = \dot{m}_0 + \dot{m}_f$

$$T_{t5} = T_{t4} - \frac{W_C}{\eta_m (1+f) Cp_T}$$

Turbine pressure ratio (TPR)

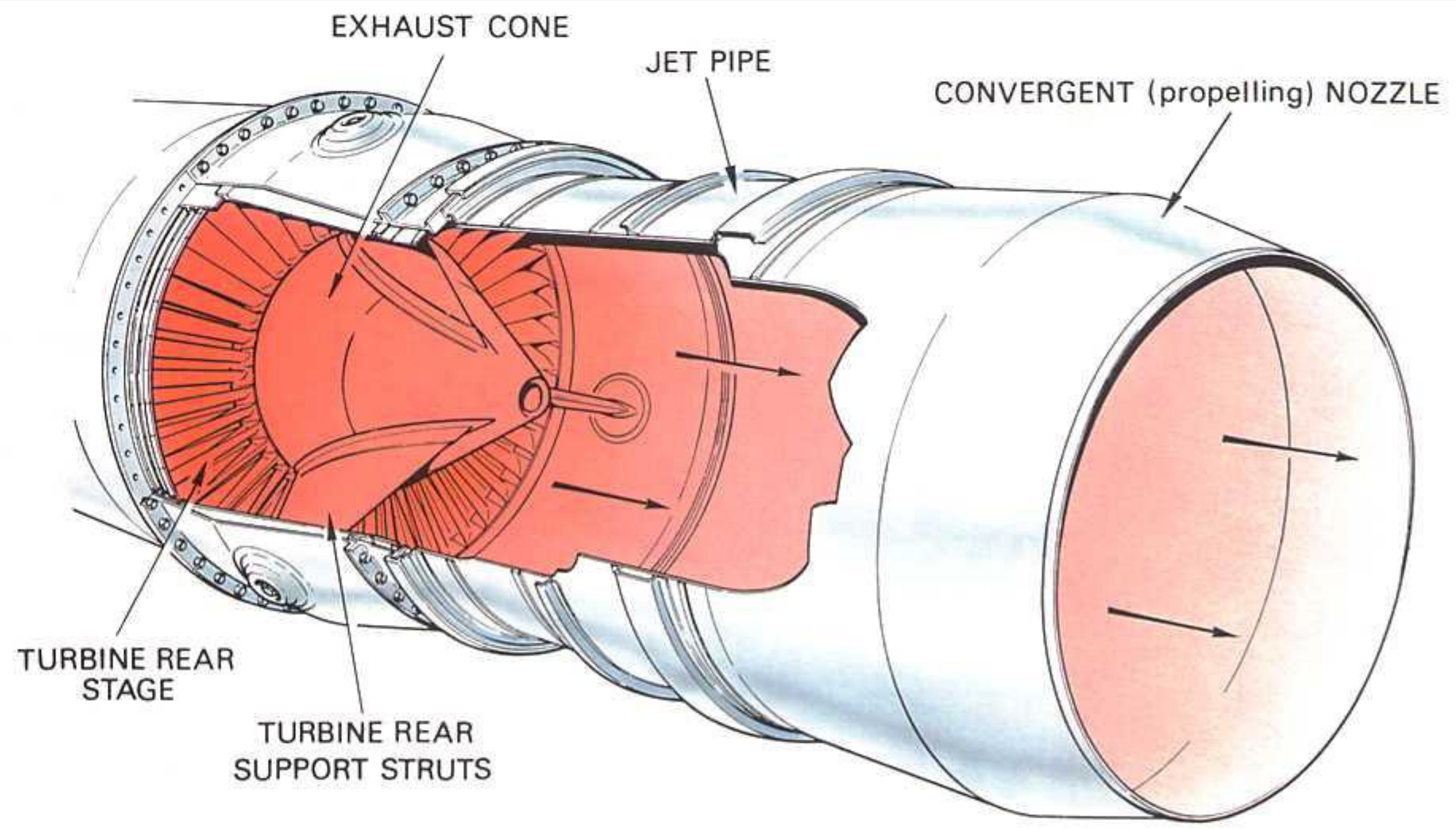
$$\pi_T = P_{t4}/P_{t5} > 1$$

$$\frac{1}{\pi_T} = \left(1 - \frac{1 - T_{t5}/T_{t4}}{\eta_T}\right)^{\frac{k_t}{k_t - 1}}$$

Turbine isentropic efficiency

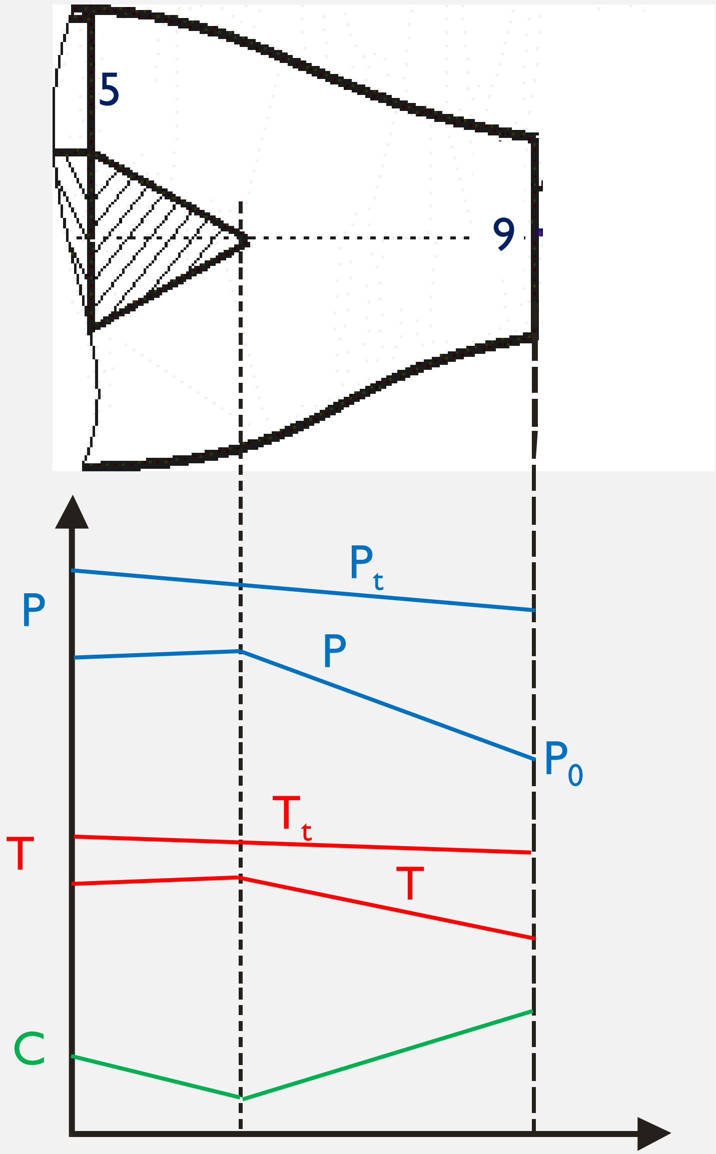
$$\eta_T = \frac{W_T}{W_{Ts}} = \frac{Cp_t (T_{t4} - T_{t5})}{Cp_t (T_{t4} - T_{t5s})} = \frac{1 - T_{t5}/T_{t4}}{1 - T_{t5s}/T_{t4}} = \frac{1 - T_{t5}/T_{t4}}{1 - (P_{t5}/P_{t4})^{\frac{k_t - 1}{k_t}}}$$

ENGINE PROPELLUING NOZZLE



Used for exhaust gas acceleration to velocity

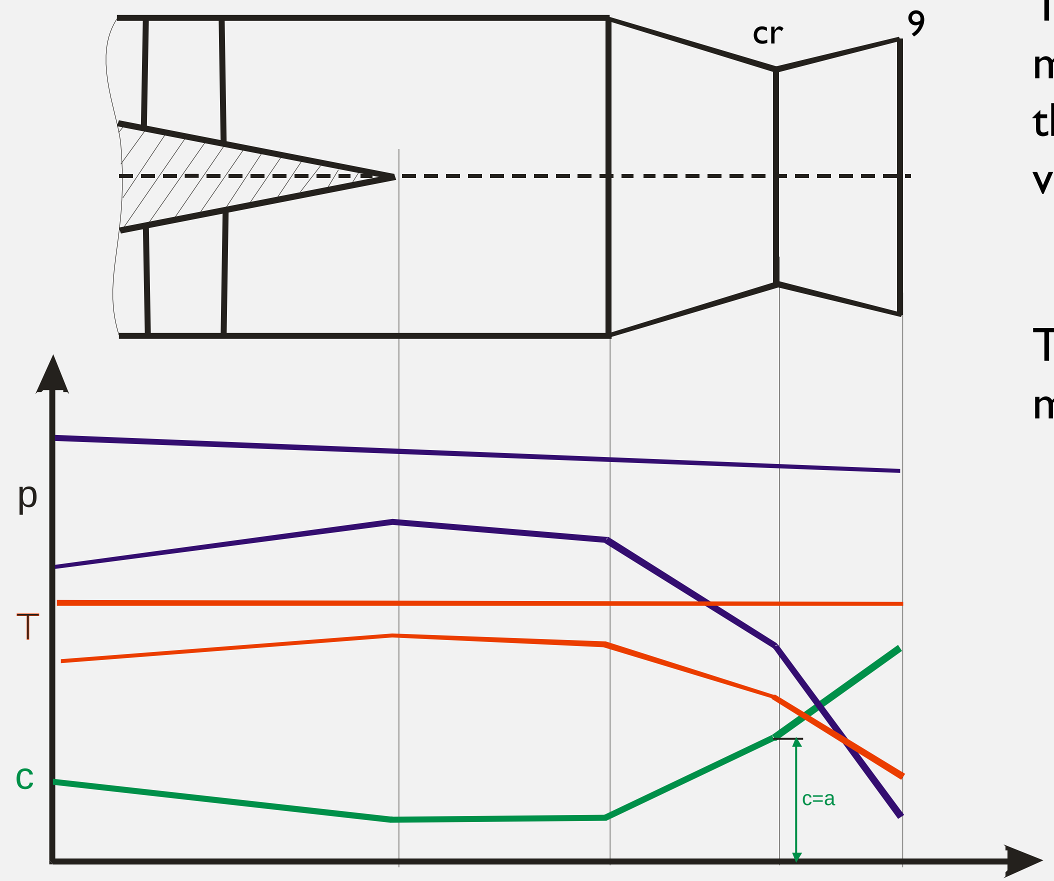
EXHAUST NOZZLE OPERATION



In the nozzle:

- Total pressure decreases due to friction losses
- Static pressure changes depending on nozzle shape
- Ideally, the nozzle expands the flow to ambient pressure
- Total temperature is assumed constant (heat transfer neglected)
- Static temperature decreases as the flow accelerates
- In a convergent nozzle, subsonic flow accelerates; in divergent sections, behavior depends on Mach number

CONVERGENT–DIVERGENT NOZZLE OPERATION



The throat is the minimum-area section where the flow reaches sonic velocity.

$$c_{kr} = a$$

The throat limits the maximum mass flow (chocked nozzle)

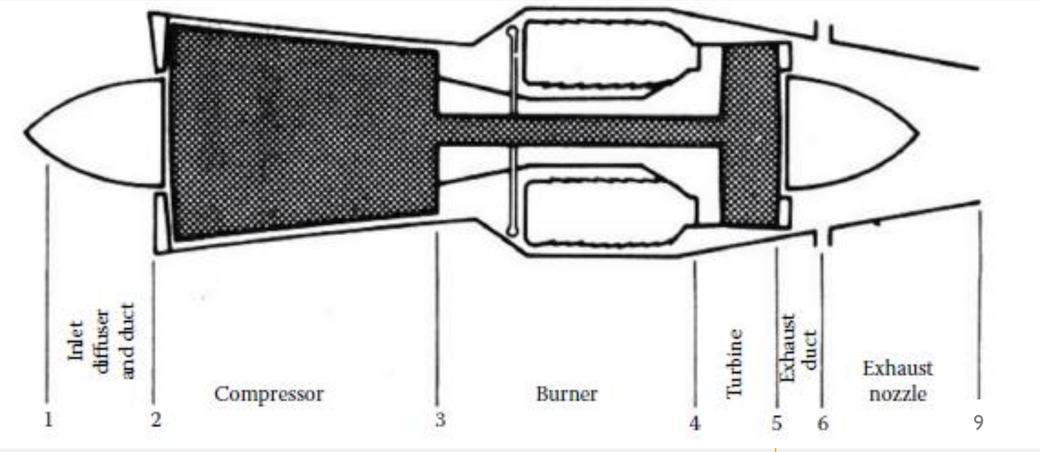
If the exit pressure matches ambient pressure, the nozzle produces fully expanded supersonic flow,

$$c_5 > c_{kr} \qquad P_9 \approx P_0$$

$$c_9 = \sqrt{2c_pT_{t5}\left(1-\left(\frac{P_9}{\pi_N P_{t5}}\right)^{\frac{k-1}{k}}\right)}$$

π_N - pressure losess in the propelling nozzle

FULL EXPANSION IN THE NOZZLE WITH LOSESS



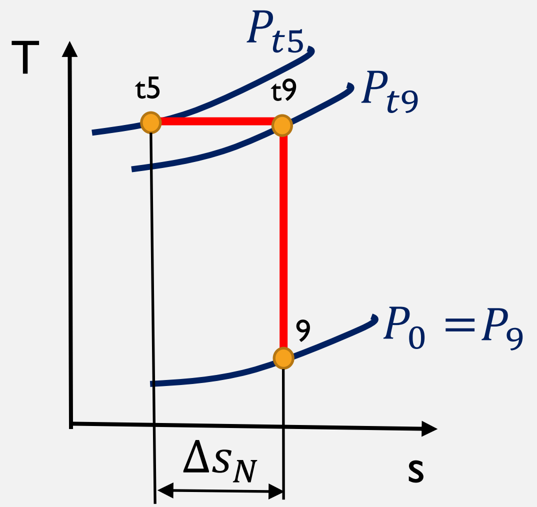
NOZZLE (5-9)

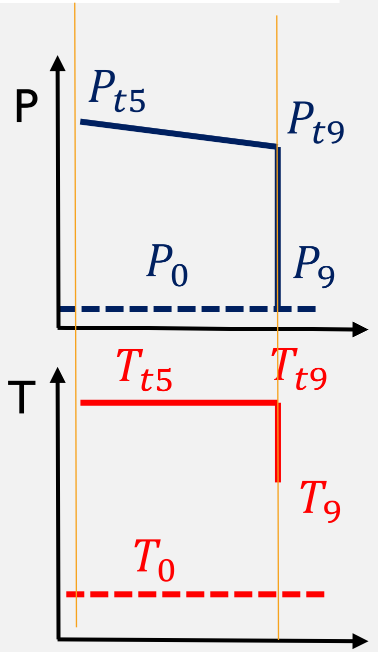
Pressure losses: $\pi_N = \frac{P_{t9}}{P_{t5}} < 1 \quad P_{t9} = \pi_N P_{t5}$

No heat losess $\quad T_{t9} = T_{t5}$

Full expansion: $\quad P_9 = P_0$

Entropy growth: $\Delta s_N = R_t ln \frac{1}{\pi_N}$



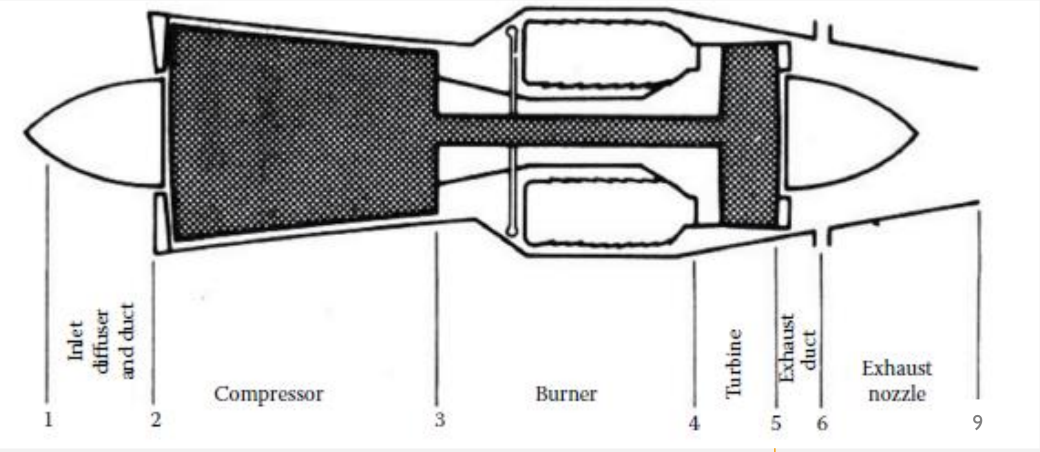


$$c_9 = \sqrt{2Cp_t(T_{9t} - T_9)}$$ - for incompressible flow

$$c_9 = a_9 M_9 = \sqrt{k_t R_t T_9} * \sqrt{\frac{2}{k_t - 1}\left(\frac{T_{t9}}{T_9} - 1\right)}$$ - for compressible flow

$$\frac{T_{t9}}{T_9} = \frac{P_{t9}}{P_9}^{(k_t - 1)/k_t}$$ - isentropic relation betwen total and static paramethers

UNDER EXPANDED FLOW IN THE NOZZLE WITH LOSESS



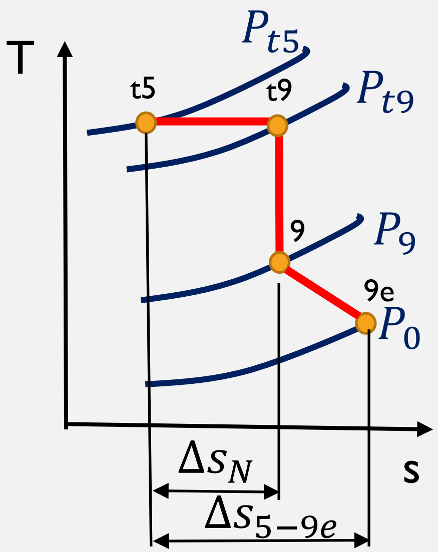
NOZZLE (5-9)

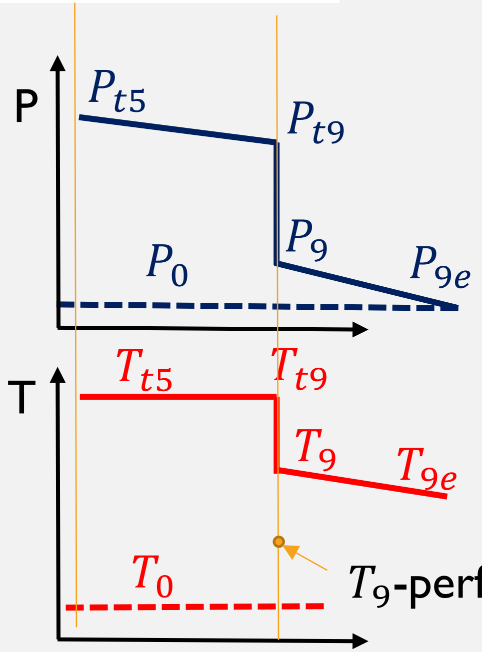
Pressure losses: $\pi_N = \frac{P_{t9}}{P_{t5}} < 1 \quad P_{t9} = \pi_N P_{t5}$

No heat losess $\quad T_{t9} = T_{t5}$

Incomploete expansion: $\quad P_9 > P_0$

Entropy growth: $\Delta s_{5-9e} = Cp_t ln \frac{T_{9e}}{T_{t5}} - R_t ln \frac{P_0}{P_{t5}}$





T_9-perfect expansion

$$c_9 = a_9 M_9 = \sqrt{k_t R_t T_9} * \sqrt{\frac{2}{k_t - 1}\left(\frac{T_{t9}}{T_9} - 1\right)}$$

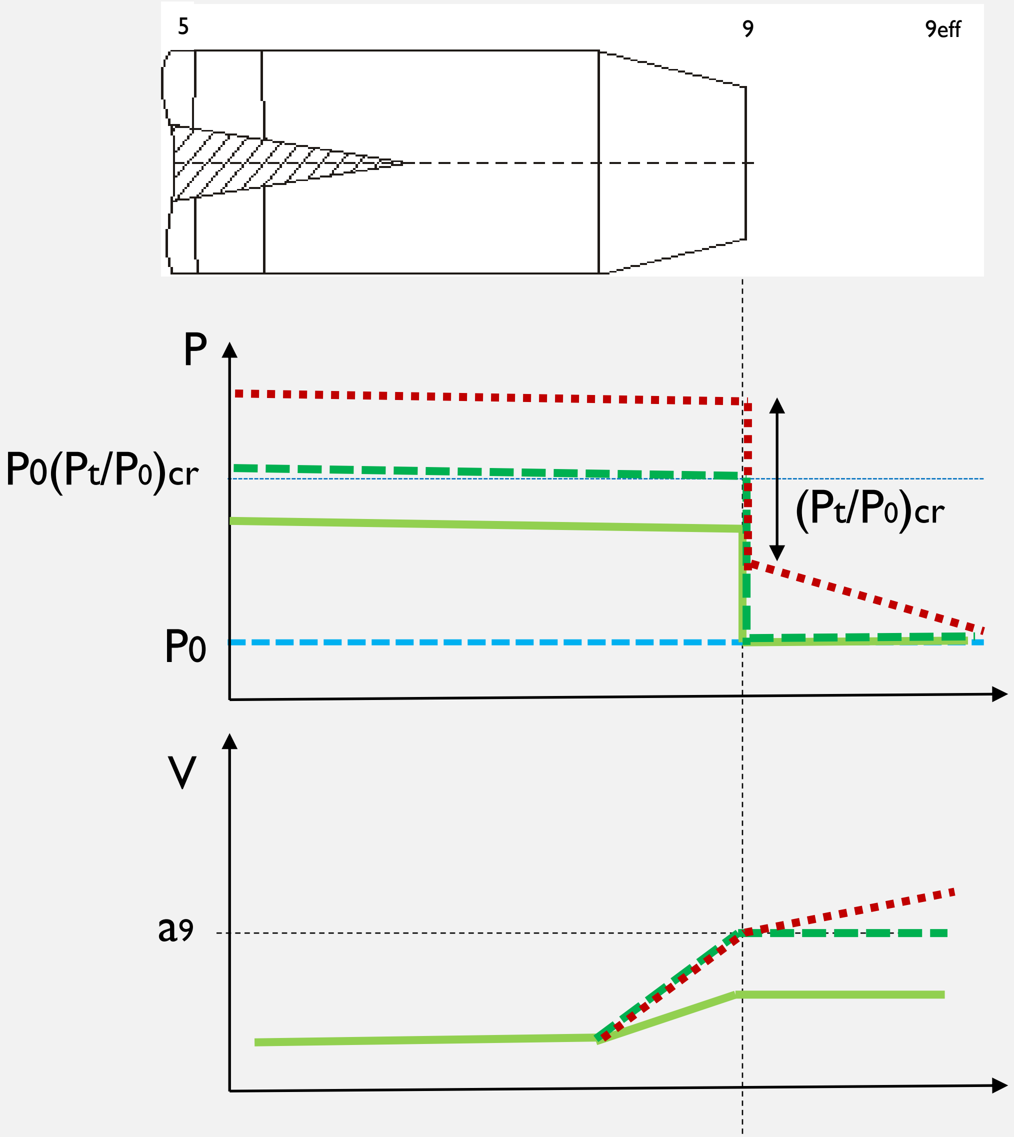
$$c_{9e} = c_9 + \frac{A_9(P_9 - P_o)}{\dot{m}_9} = c_9 + \frac{(P_9 - P_o)}{\rho_9 c_9}$$

$$= c_9 + \frac{R_9 T_9 (P_9 - P_o)}{P_9 c_9}$$

$$\frac{T_{t9}}{T_9} = \left(\frac{P_{t9}}{P_9}\right)^{(k_t - 1)/k_t}$$

continuity equation:
$\dot{m}_9 = \rho_9 \rho_9 c_9$

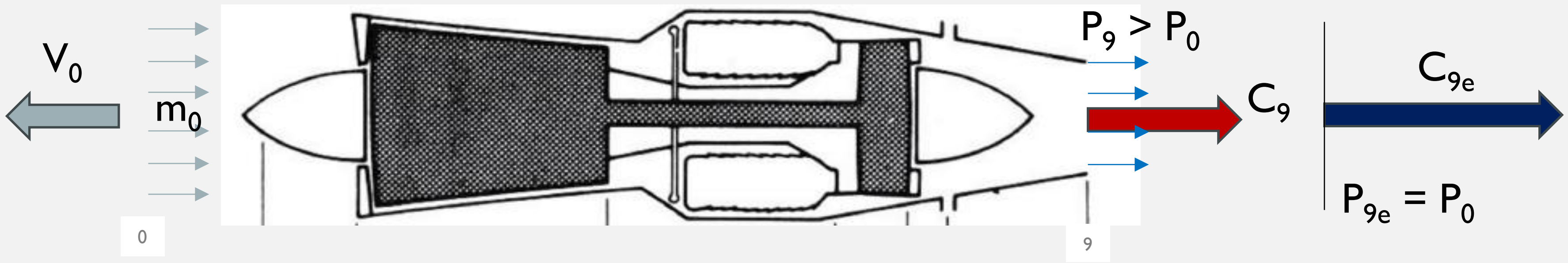
TURBOJET ENGINE WITH SUBSONIC (CONVERGENT) NOZZLE



$$\left(\frac{P_t}{P}\right)_{cr} = \left(\frac{1+k_t}{2}\right)^{\frac{k_t}{k_t-1}}$$

- Full gas expansion in the nozzle is available for subsonic flow speed ($C_9<a_9$)
- When total to static pressure ratio is critical the flow velocity in nozzle outlet is equal speed of sound
- When total to static pressure ratio is higher than critical the nozzle outlet velocity is equal speed of sound – chocked nozzle. Gas expansion process is continued outside the nozzle

TURBOJET ENGINE PERFORMANCE



$$c_{9e} = c_9 + \frac{A_9(P_9 - P_o)}{\dot{m}_9}$$

- Thrust

$$T = m_9 C_9 - m_0 V_0 + A_9(P_9 - P_0) = m_9 V_{9e} - m_0 V_0$$

- Specific Thrust

$$ST = \frac{T}{m_0} = (1 + f_B) C_{9e} - V_0$$

- Specific Fuel Consumption

$$SFC = \frac{m_f}{T} = \frac{f_B}{(1 + f_B) V_{9e} - V_0}$$

- Thermal efficiency

$$\eta_{th} = \frac{m_9 {C_{9e}}^2 - m_0 {V_0}^2}{2\, m_f\, FHV} = \frac{(1 + f_B) {C_{9e}}^2 - {V_0}^2}{2\, f_B FHV}$$

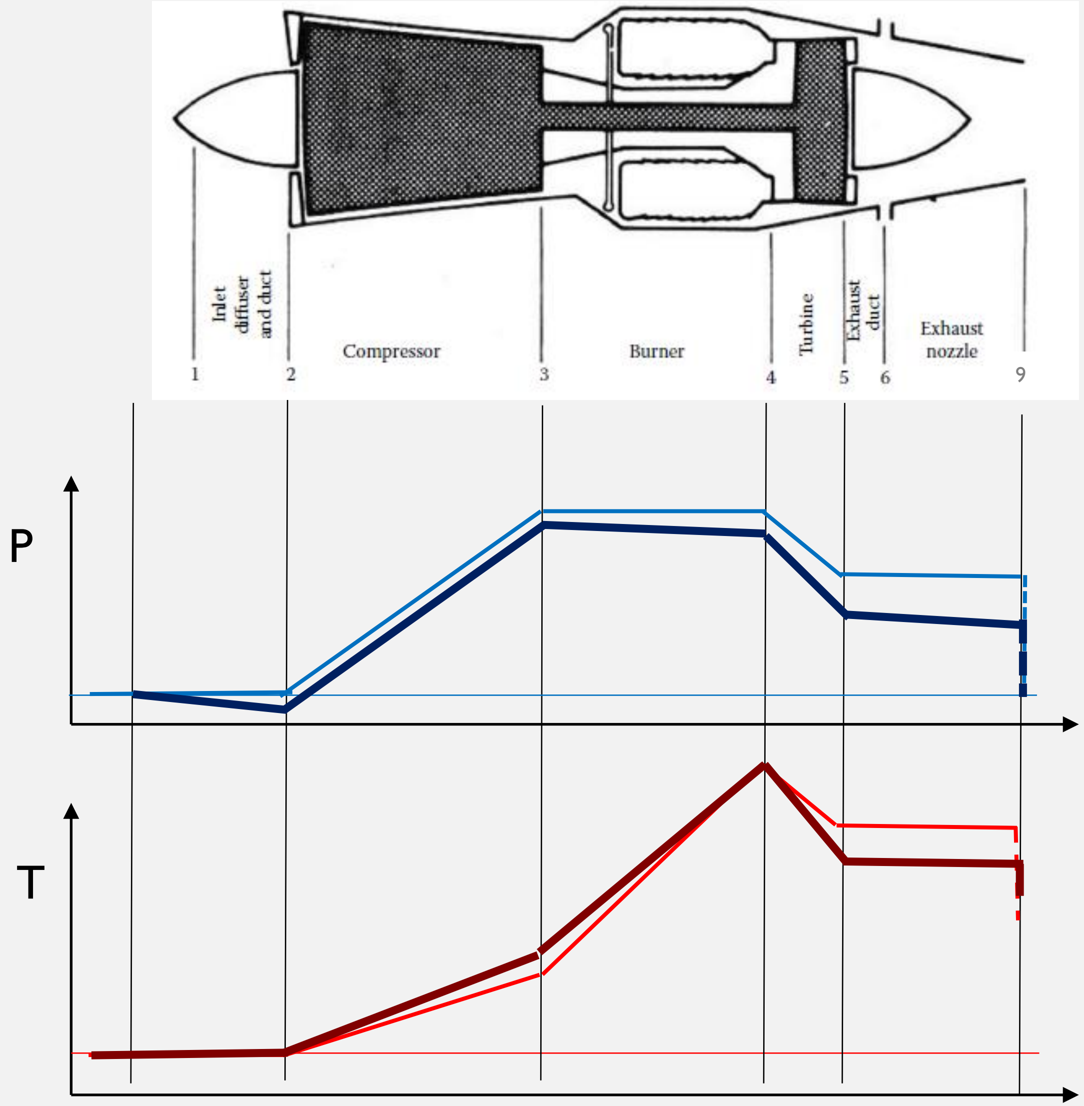
- Propulsive efficiency

$$\eta_p = \frac{2\, T\, V_0}{m_9 {C_{9e}}^2 - m_0 {V_0}^2} = \frac{2\, ST\, V_0}{(1 + f_B) {C_{9e}}^2 - {V_0}^2}$$

- Overall efficiency

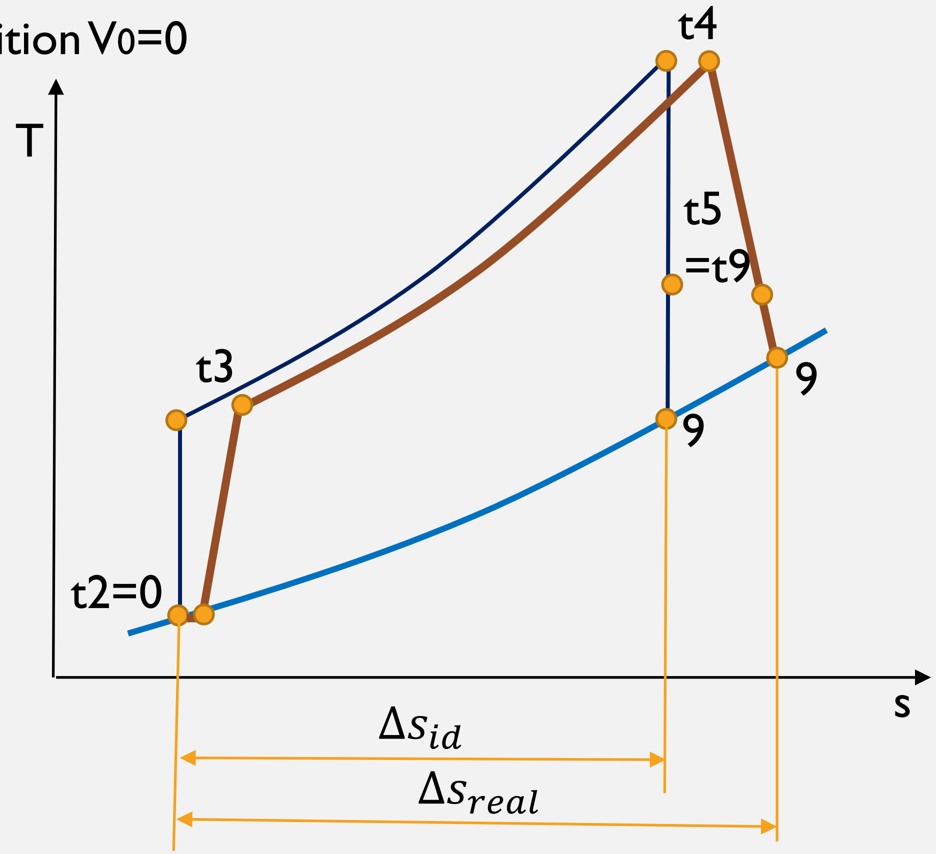
$$\eta_o = \frac{T\, V_0}{m_f\, FHV} = \frac{ST\, V_0}{f_B\, FHV} = \eta_{th} \eta_p$$

IDEAL VS REAL TURBOJET ENGINE



The same CPR and Tt4

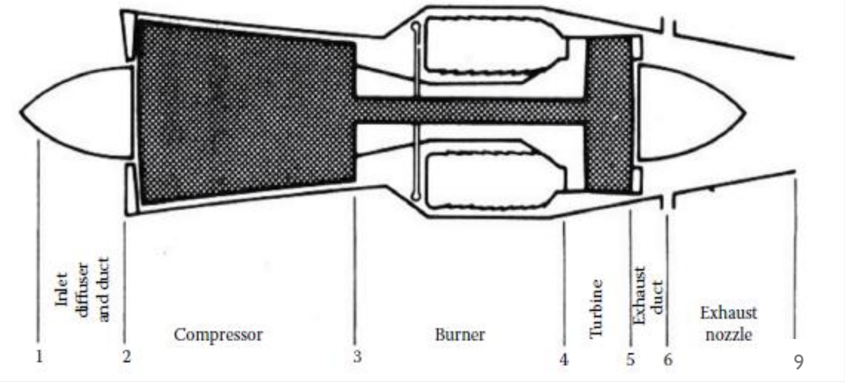
Static condition V0=0



EXAMPLE OF TURBOJET ENGINE CALCULATION

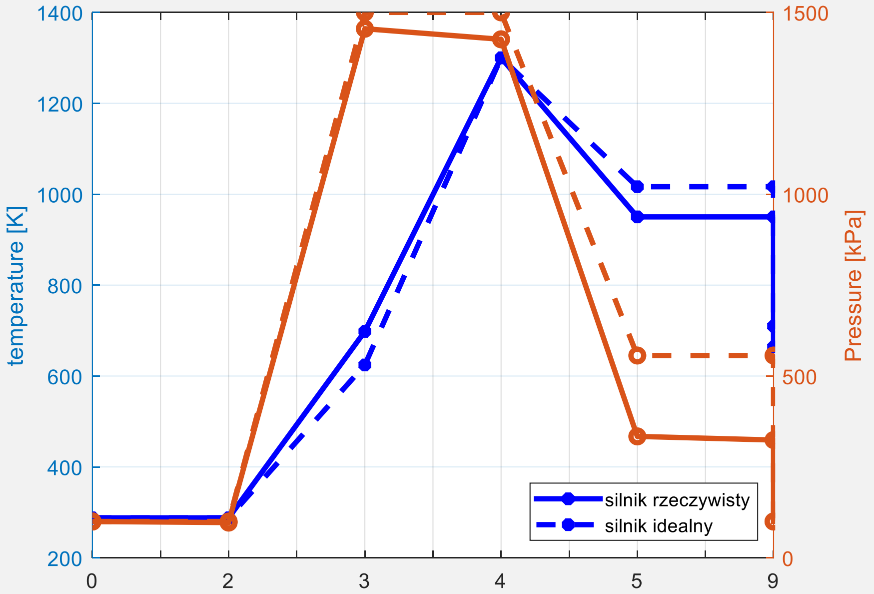
Given:

	Parameter	Value
1	'Altitude [m]'	0
2	'Flight speed Ma'	0
3	'air mass flow [kg/s]'	25
4	'CPR'	15
5	'T_t_4 [K]'	1300
6	'Inlet pressure losses \pi_D'	0.9700
7	'Burner pressure losses \pi_B'	0.9800
8	'Nozzle pressure losses \pi_N'	0.9700
9	'compressor efficiency \eta_C'	0.8200
10	'turbine efficiency \eta_T'	0.8900
11	'Burner efficiency \eta_B'	0.9800
12	'Mechanical efficiency \eta_M'	0.9900



Temperature and pressure comparison in engine cutsections:

	Section	Temp. [K] real	Temp. [K] ideal	Pressure [kPa] real	Pressure [kPa] ideal
1	'0'	288	288	100	100
2	't0'	288	288	100	100
3	't2'	288	288	97	100
4	't3'	698	624	1455	1500
5	't4'	1300	1300	1426	1500
6	't5'	950	1016	334	556
7	't9'	950	1016	324	556
8	'9'	710	664	100	100



REAL VS. IDEAL ENGINE SUMMARY

Engine performance:

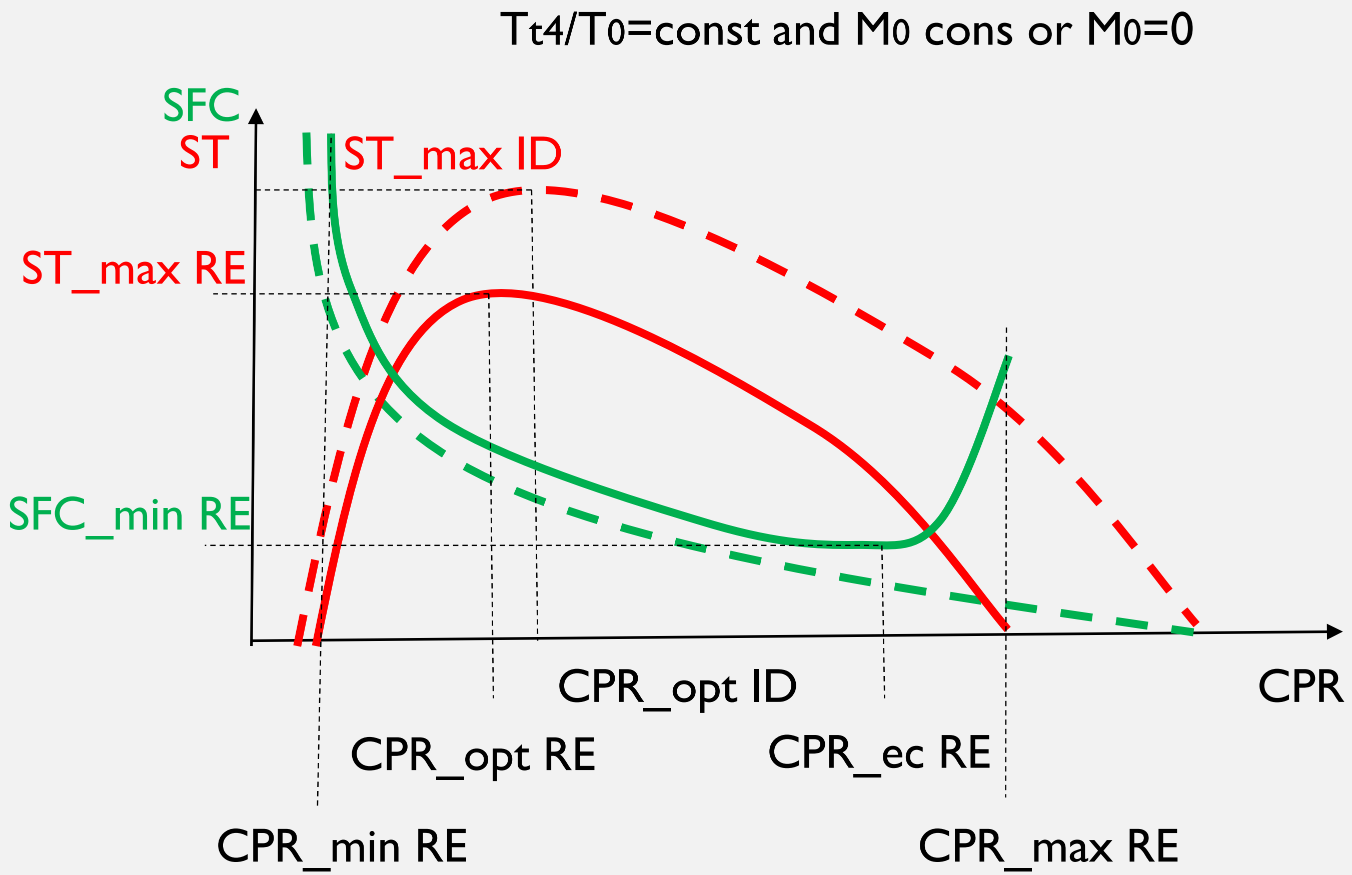
	Parameter	Unit	Real eng.	Ideal eng.
1	'Thrust'	'kN'	19.0608	23.1214
2	'Specific Thrust'	'N*s/kg'	762.4330	924.8565
3	'fuel consump'	'kg/s'	0.4285	0.4714
4	'Specific fuel consump'	'kg/N/h'	0.0809	0.0734
5	'prędkość V9'	'm/s'	749.5865	907.7404
6	'therm. efficiency'	'-'	0.3878	0.5177
7	'prop. efficiency'	'-'	0	0
8	'overall efficiency'	'-'	0	0

Real to ideal jet engine comparison shows:

- Total pressure in engine sections is lower in the real engine
- Total temperature after compressor is higher, but after turbine is lower in the real engine
- Higher temperature after compressor causes lower fuel consumption of the real engine - TIT (turbine inlet temperature) is the same in both engines
- Lower total temperature in the nozzle inlet and higher static temperature in the nozzle outlet causes lower outlet flow velocity and by this way lower thrust and specific thrust of real jet engine
- Specific fuel consumption is higher due to lower thrust
- Thermal and overall efficiencies are lower in the real engine

Example of turbojet engine thermodynamic model: real turbojet engine model

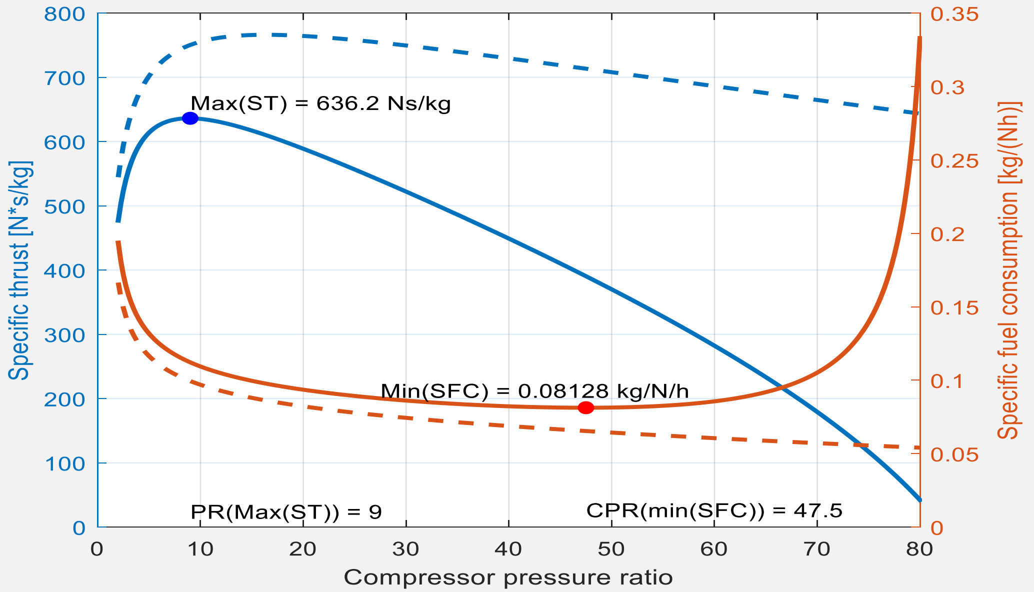
TURBOJET CYCLE OPTIMISATION REAL VS IDEAL

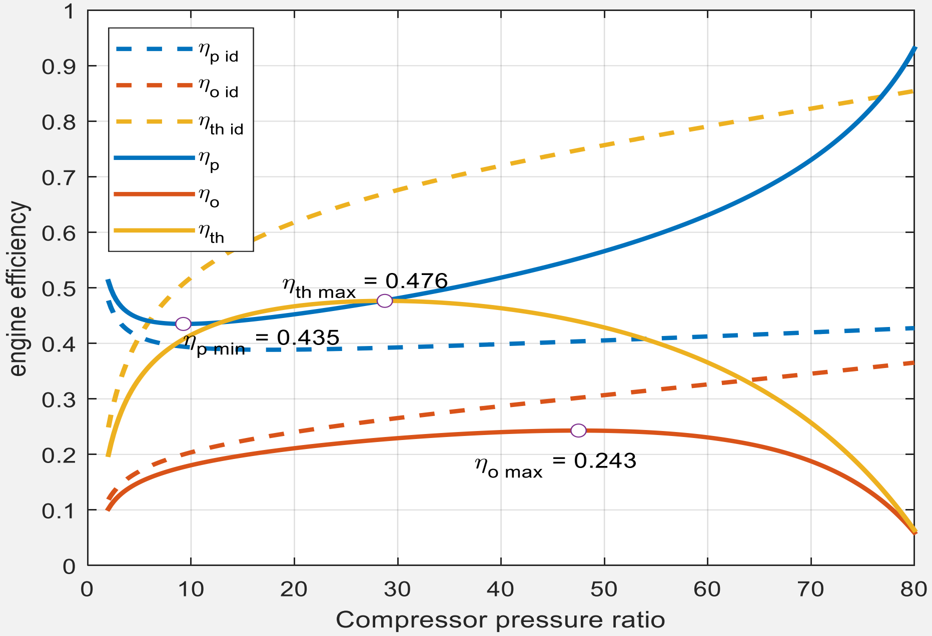


SUMMARY:

- Specific thrust (ST) of real engine looks similar to ideal engine. It grows with compressor pressure ratio, achieves maximum for optimal CPR than is goes down. Differences:
  - ST is lower in whole range of CPR than in an ideal engine
  - Pressure ratio of ST_max is lower than in ideal engine
- Specific fuel consumption (SFC) is higher than in ideal engine. It decreases with CPR growth, achieves minimum value for high CPR and then goes up. CPR of SFC_min is caled CPR_ek
- CPR available range shrinks for real engine

EXAMPLE OF ENGINE OPTIMIZATION RESULTS



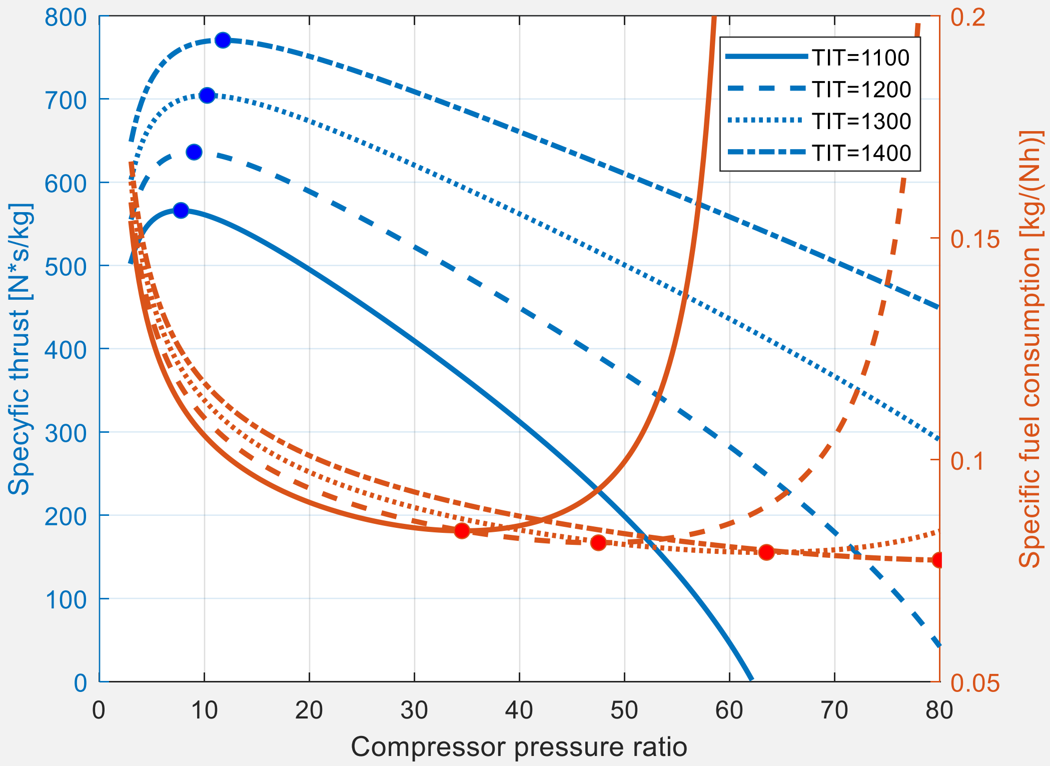


Calculation of the real and the ideal engine was prowided for the same TIT (T_{t4}) and for the same flight codition (H, M_0)

	Parameter	Value	Parameter	Value
1	'CPR(ST_max)'	9	'CPR(eta_p_min)'	9.2500
2	'CPR(SFC_min)'	47.5000	'CPR(eta_o_max)'	47.5000
3	''	NaN	'CPR(eta_th_max)'	28.7500

- Propulsive efficiency as a function of CPR represents opposite relation to ST, it is minimal for optimal CPR
- Overall efficiency represents opposite relation to SFC and it achieves maximum for minimum SFC
- For flight speed 0 thermal efficency represent opposite trend fo SFC and it achieves maximum for SFC minimum

TIT (T_4) INFLUENCES ON OPTIMAL PATAMETERS



SUMMARY:

- Specific thrust (ST) is higher for higher TIT
- Maximim ST is higher for higher TIT and is achieved for slightly higher CPR
- SFC achieves minimum value on lower level for higher TIT and for higher CPR

	TIT [K]	CPR(ST_max)	ST_max [Ns/kg]	CPR(SFC_min)	SFC_min [kg/(Nh)]
1	1100	7.7500	566.0651	34.5000	0.0840
2	1200	9	636.1510	47.5000	0.0813
3	1300	10.2500	704.2687	63.5000	0.0791
4	1400	11.7500	770.5643	80	0.0774

THANKS FOR YOUR ATENTION

Questions and Comments ?

1. …….
2. …….
3. …….