Aerospace Combustion

Lecture 3:

Aero – and Rocket Engines

Aero Engines

Content

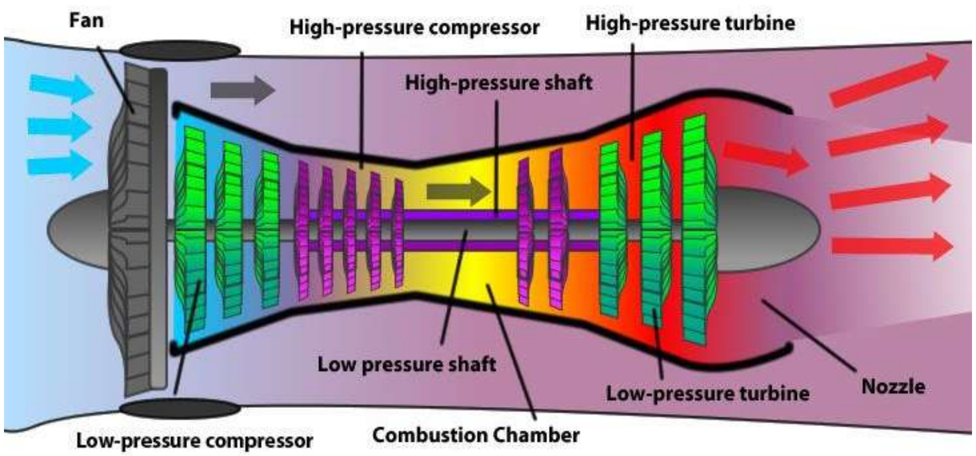
- Sample Aero Engines
- Engine Thermodynamics
- Combustor
  - Requirements and Constraints
- Engine Operation
  - Thrust
  - Emission
- Combustion Management
- Combustor Design

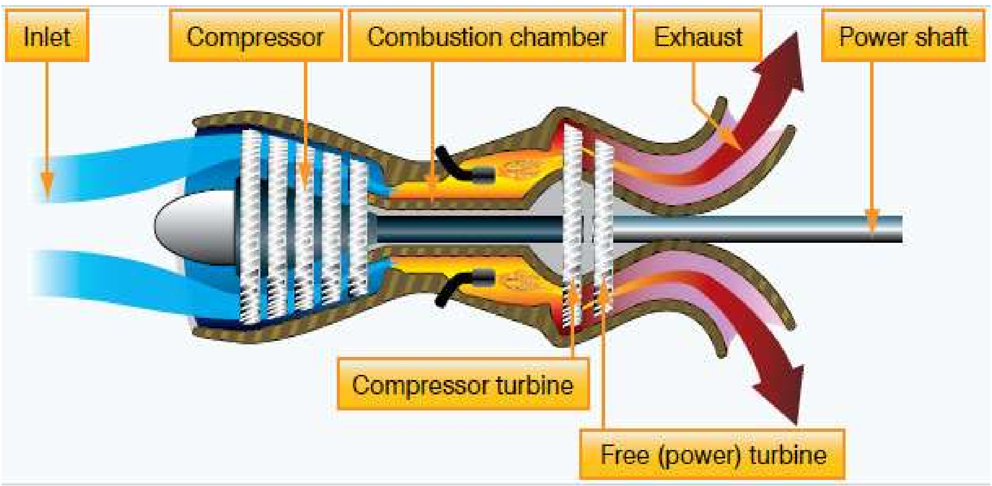
- Sample Rocket Engines
  - Solid Propellant Motors
  - Liquid Propellant Motors
- Engine Thermodynamics
- Combustor
  - Requirements and Constraints
  - Ignition
- Engine Operation
  - Thrust
  - Emission
- Combustion Management
- Combustor Design

Aero Engines

Turbo-jet(fan) / Turbo-shaft Engines

Turbo-jet(fan) engines are similar to turbo-shaft engines but optimized for jet power instead of shaft power



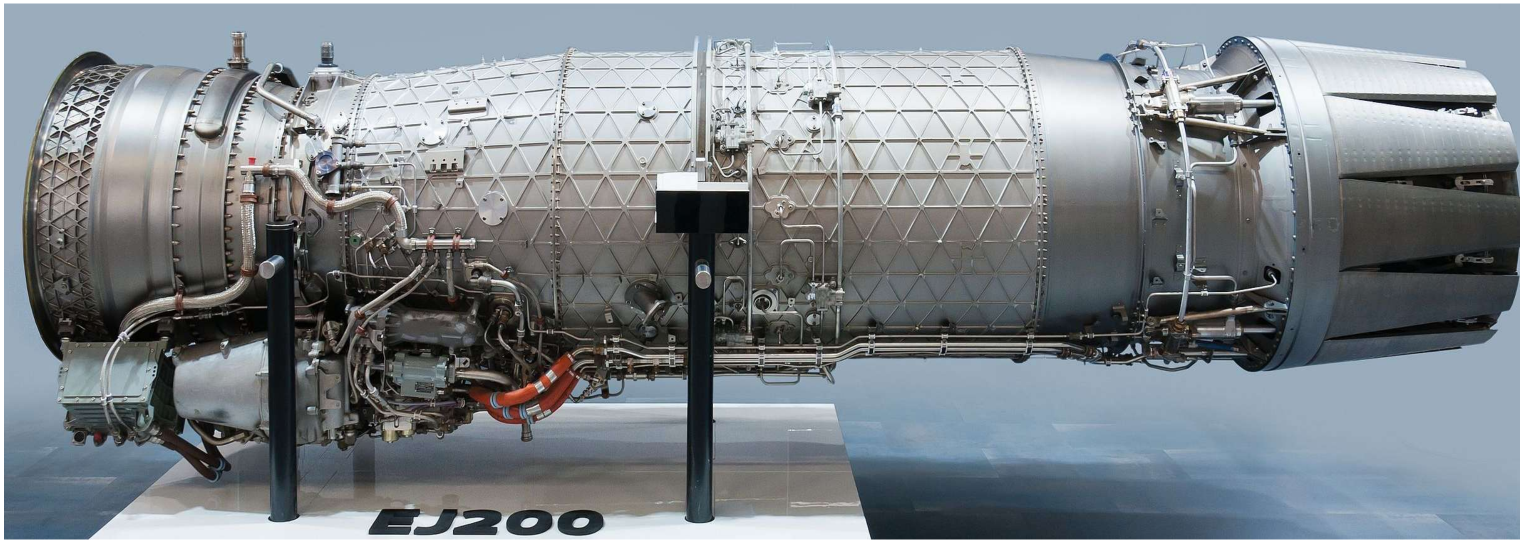


Aero Engines

Military Jet Engines (Low Bypass Ratio)

Eurojet EJ200 (Consortium: Rolls-Royce, MTU Aero Engines, Fiat Avio, ITP)

- Max. Thrust:
  60 kN (90 kN using AB)
- Mass: ca. 1000 kg
- Compressor: 3 LP +5 HP
- Pressure Ratio: 26
- Bypass Ratio: 0.4
- TET: ca. 1800 K
- Mass flow rate: 76 kg/s



Aero Engines

GEnX Family

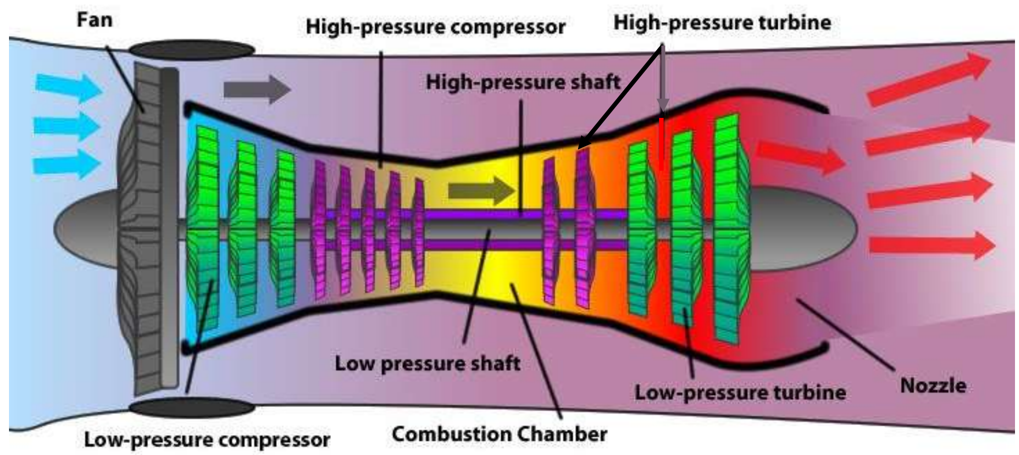
- Gear - turbo fan engine with high by-pass ratio
- single stage fan, 4 stage LP and 10 stage HP compressor
- 2 stage HP , 7 stage LP turbine
- Twin annual premixing swirl (TAPS)combustor
- 2560 rpm, 11377 rpm
- 50:1 OPR
- 8.8:1 by-pass ratio
- Thrust: 310 – 340 kN (TOT)

Aero Engines

Components in Aero Engines

Let’s consider all regions and devices in which changes of temperature, pressure and velocity appear

1) Entry diffusor
2) Fan
3) Compressor
   31) LP
   32) HP
4) Combustor
5) Turbine
   51) LP
   52) HP
6) nozzle



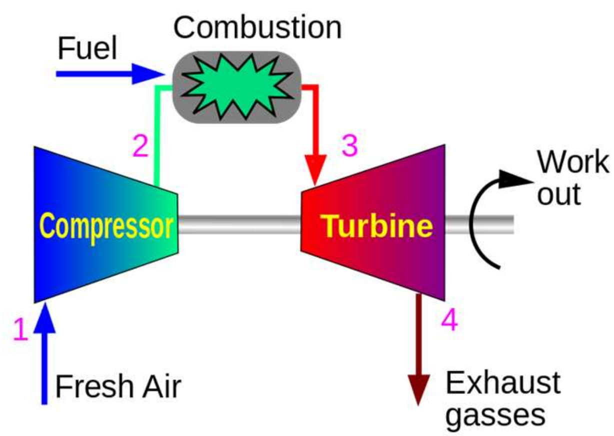
- Each of these devices has it own efficiency.
- Then, we will have 3D effects: temperature, pressure and velocity profiles aren’t 1D in any of the flow ducts.
- Additionally we may take into account thermodynamic efficiencies (increasing entropy).
- Add that the air which enters will have varying humidity or even liquid water.

Hence, it is much easier to neglect issues which are small from an engineer point of view and keep in mind that reality sooner or later will kick in.

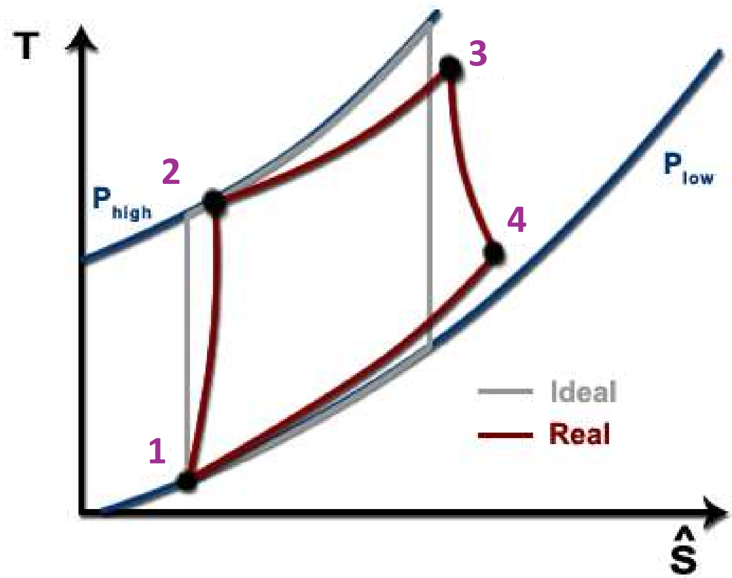
Aero Engines

Thermodynamics

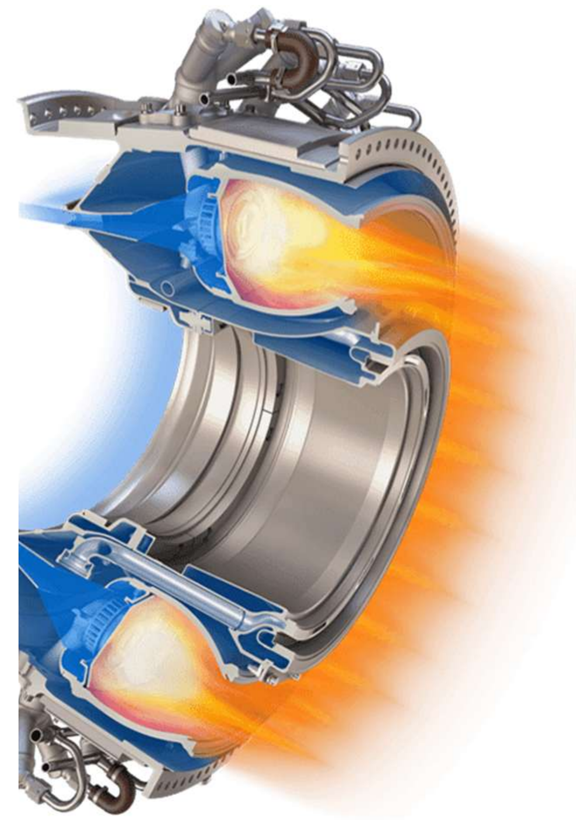
Brayton Cycle (ideal / real)



Functional principle of a jet engine



T, S diagram of a jet engine



Cut through the ring combustor of a jet engine

Cycle Analysis

For a first estimate about engine efficiency the fuel mass flow rate is frequently neglected on the analysis. Let's try to find out why.

Let's consider a turbofan engine

- with a bypass ratio of 10:1: ten times more air is generating thrust just through fan operation
- which generally operate in the lean regime: the amount of oxygen is about a factor of two higher than that of the fuel

Let's assume a total air mass flow rate of ~ 1300 kg/s at take-off (not too far away from GE90)
→ 130 kg/s go through the combustor

However, not the entire air is used for combustion but still a substantial amount is needed for cooling the engine and reducing the turbine entry temperature

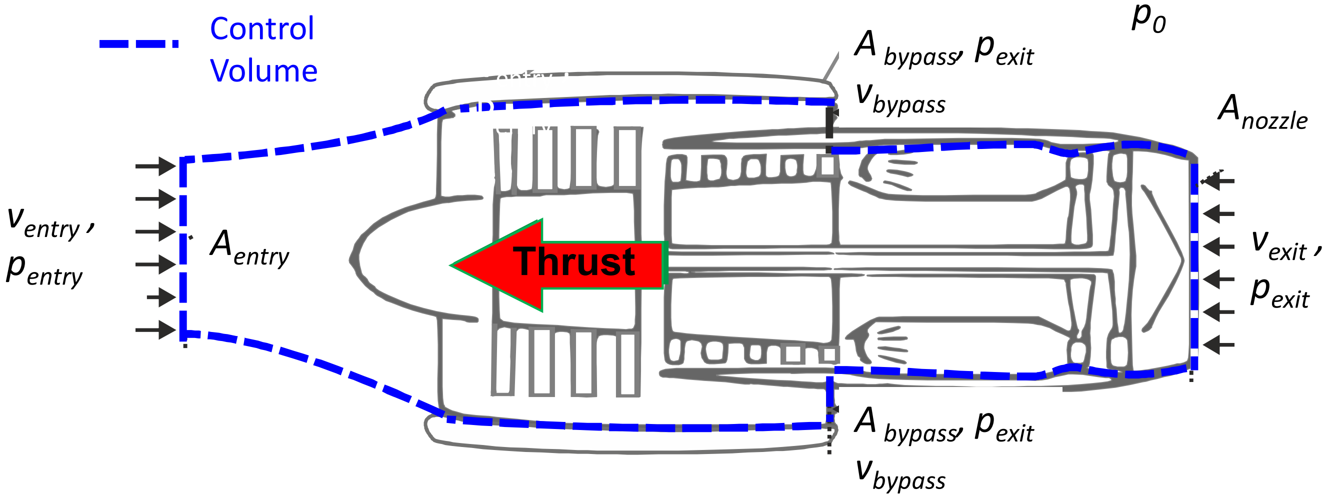
→ GE 90 runs about 3 kg/s of fuel through engine at takeoff
→ Air/Fuel ratio of ~ 43; with only 21% of oxygen → mixture ratio of ~ 9.1;
→ Stoichiometric mixture ratio would be around ~ 3.5

Fuel mass flow rate < 0.3 % of total mass flow rate through the engine → can be neglected for engineering analysis
Even for a jet engine without bypass the fuel flow can be neglected (~3%)

Turbofan Engine Thrust

$$F = \dot{m}_{bypass} v_{bypass} + \dot{m}_{core} v_{nozzle} - \dot{m}_{air,total} v_{entry} + A_{nozzle}(p_{exit} - p_{entry}) + A_{bypass}(p_{exit} - p_{entry})$$

Typically, exit pressures are equal to the ambient pressure p_0 → $F = \dot{m}_{bypass} v_{bypass} + \dot{m}_{core} v_{nozzle} - \dot{m}_{air,total} v_{entry}$



For a turbojet engine with no bypass:

$$F = \dot{m}_{nozzle} v_{nozzle} - \dot{m}_{entry} v_{entry}$$

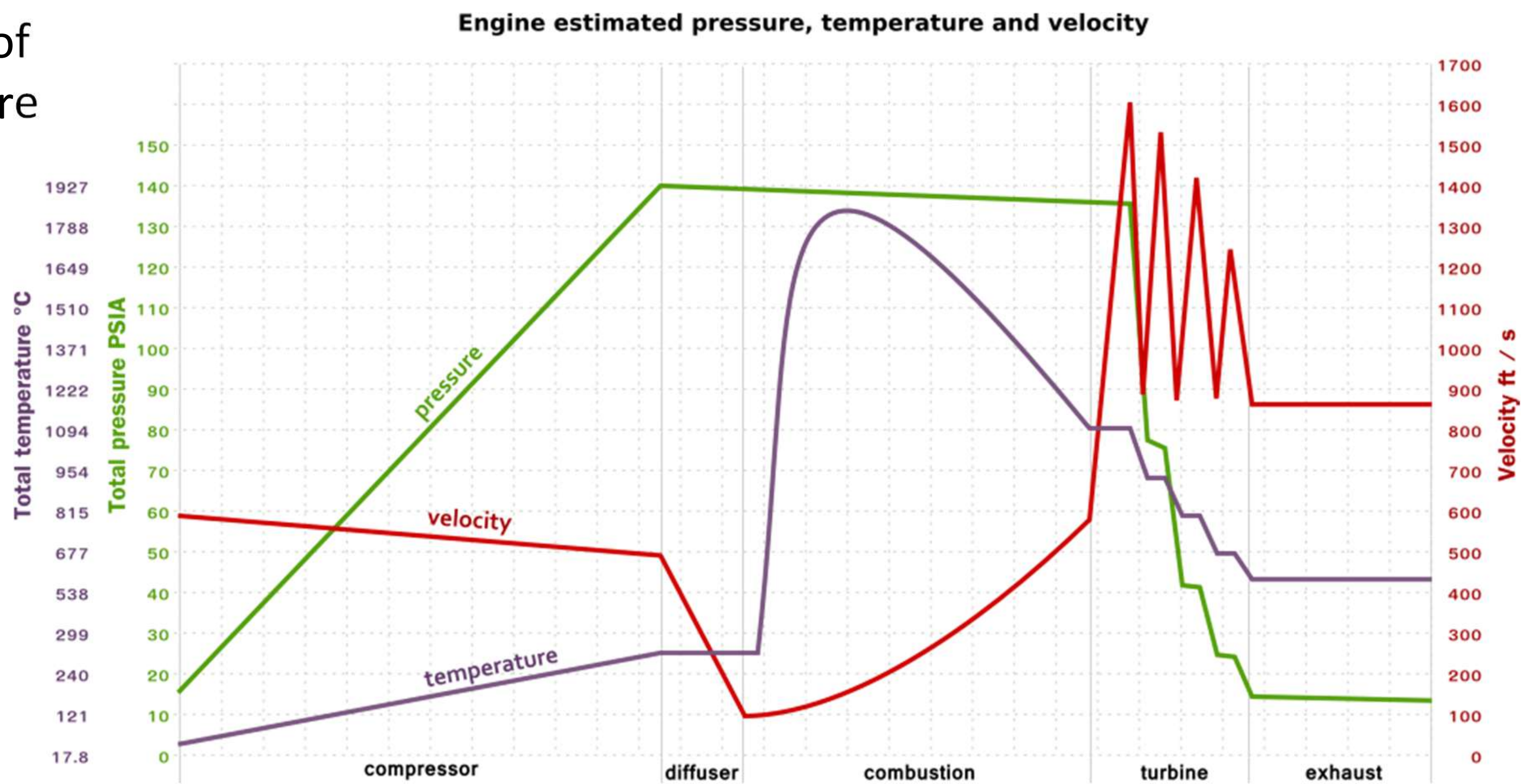
For a turbojet engine with fuel considered:

$$F = \dot{m}_{entry}(v_{nozzle} - v_{entry}) + \dot{m}_{fuel} v_{nozzle}$$

Aero Engines

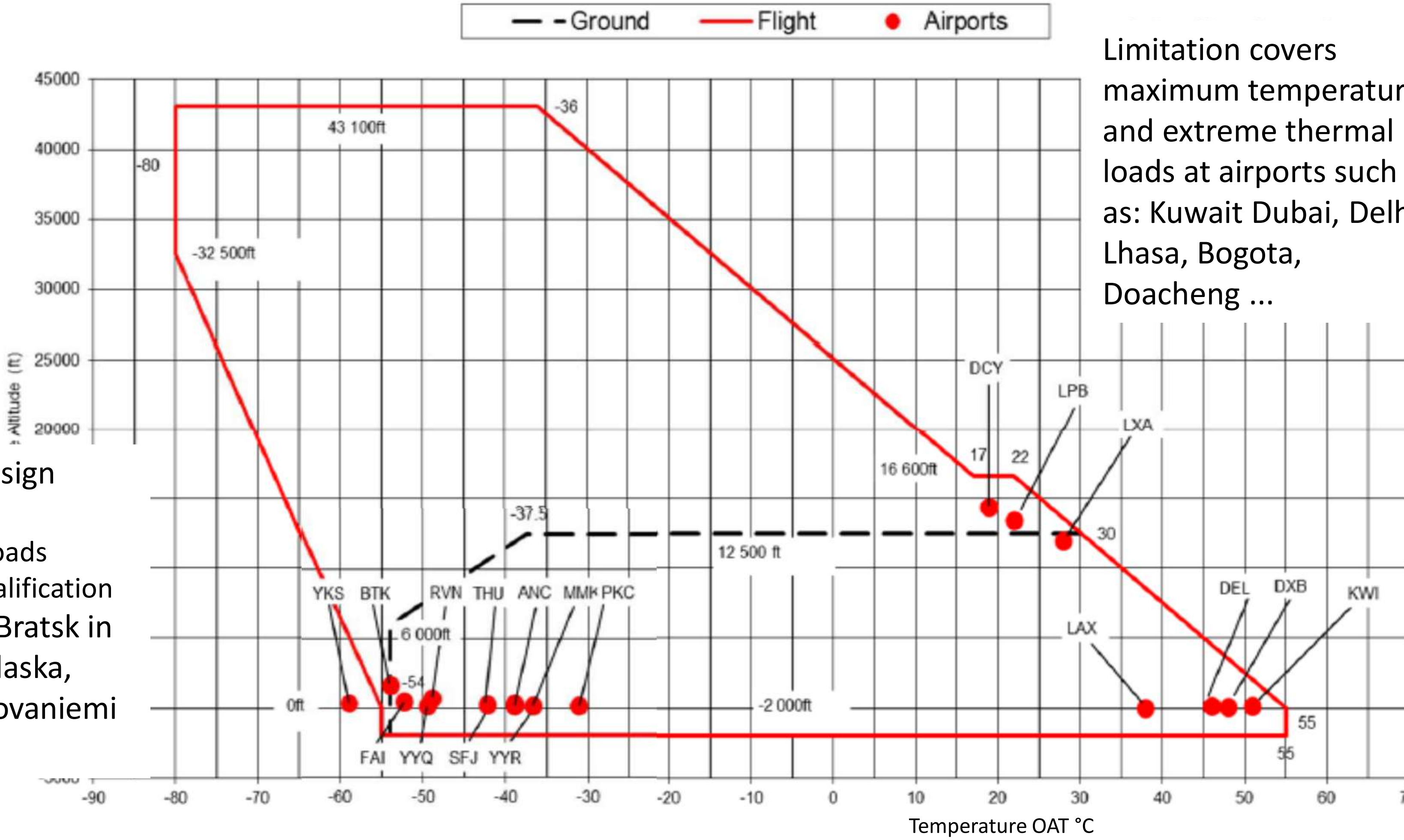
Turbojet Engine

Typical local values of temperature pressure and velocity



Aero Engines Conditions

Maximum and minimum temperature constraint



Limitation covers maximum temperatures and extreme thermal loads at airports such as: Kuwait Dubai, Delhi, Lhasa, Bogota, Doacheng ...

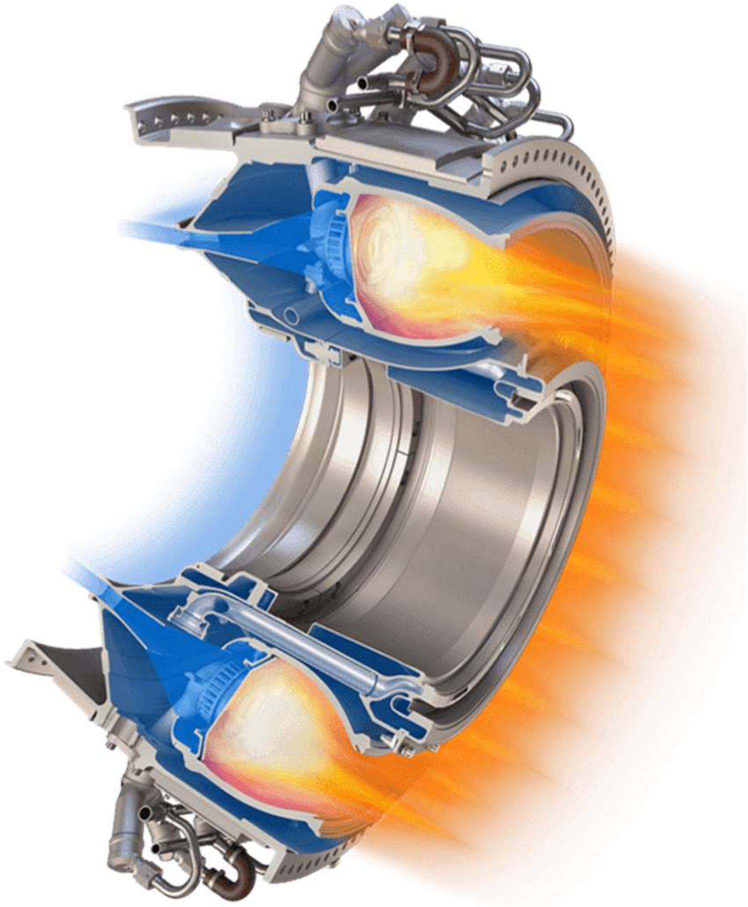
Minimum certified design temperatures:

- -54°C for structural loads
- -55°C for systems qualification

Cold airports such as Bratsk in Russia, Fairbanks in Alaska, Churchil in Canada, Rovaniemi in Finland

Aero Engines Conditions

Boundary Conditions



Lightweight combustor should be as short as possible, however

- Combustion should be complete other wise heat release in the first HP turbine
- Residence time long enough for combustion (5-8 ms)
- Average velocities 100 m/s – 200 m/s

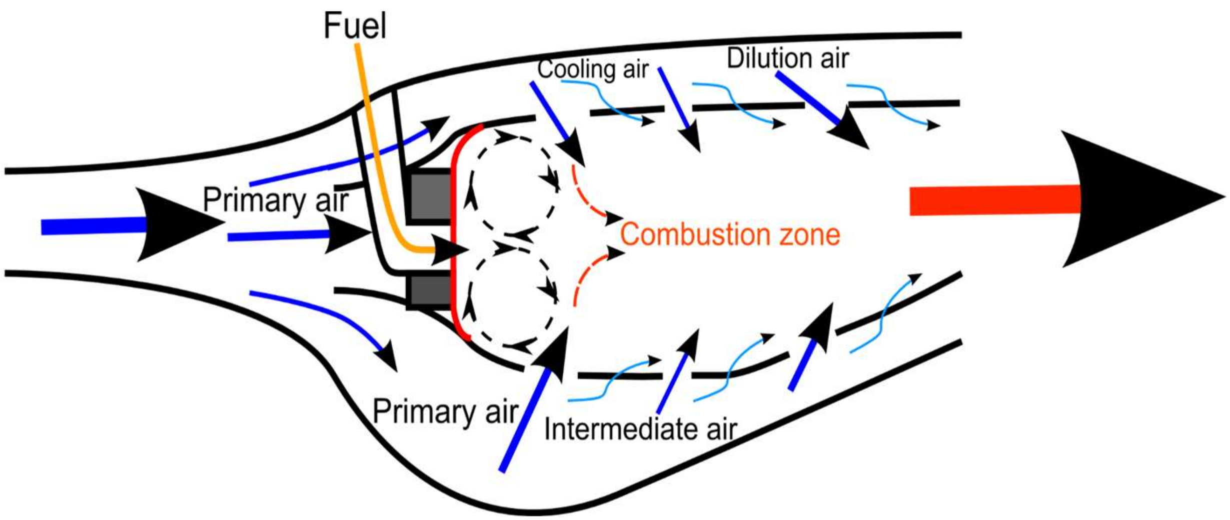
Operation Conditions (taxi – take-off) require large variations in total mass flow rates (up to 1.5 kg/s) as well as air-to-fuel (AFR) ratio

- Equivalence ratio varies with thrust level
- Combustor pressure varies with altitude
- Inlet air temperature varies drastically
- Combustor has to cope with weather conditions (rain, snow..)

Combustor

- Incoming air at about 150 m/s is slowed down to reduce velocity and pressure loss
- Fuel Injector/Burner generates a recirculation zone in with fuel is injected and which forms an almost stoichiometric primary combustion zone at sufficient low velocities to avoid flame lift-off and extinction
- Air is injected to shift from fuel rich combustion to lean combustion and cool the combustor liner

- After the compensation of the recirculation zone through an increase in liner cross section, it is contracted further downstream to reduce combustor dimension and weight
- Exhaust gas at almost homogeneous temperature exists the combustor to the turbine inlet



Aero Engine Operation

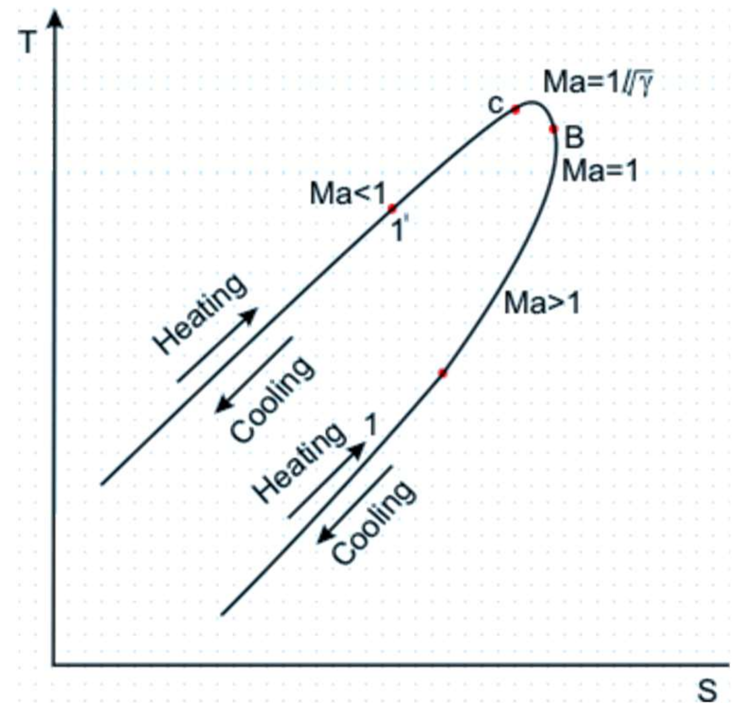
Combustor

Heat addition through chemical reaction

- In subsonic flows, heat release due combustion results in increase of specific volume and thus an increase in velocity (at constant cross section).
- In any case the speed of sound will increase.

Cooling through air injection

- Injection of coolant may depending on the thermo-physical properties of the species involved will result in a change of velocity. In any case, the speed of sound will decrease.
- It has to be avoided to come even near to the point of thermal choking in a combustor



Aero Engine Operation

Characteristic Parameters

Specific Thrust

$$F_{spec} = \frac{F_{net}}{\dot{m}_{air}}$$

Thrust to Fuel Ratio

$$F_{spec,f} = \frac{F_{net}}{\dot{m}_{fuel}}$$

Propulsive Power

$$P = \frac{1}{2}\left[\left(m_{air} + m_{fuel}\right)v_{exit}^2 - m_{air}v_{air}^2\right]$$

Thrust Specific Fuel Consumption

$$TSFC = \frac{\dot{m}_{fuel}}{F_N} = \frac{\dot{m}_{fuel}}{\dot{m}_{core}\left\{\left(1+f\right)v_{hotjet} - v_{ac}\right\} + \dot{m}_{core}\alpha\left\{v_{cold} - v_{jet}\right\}} =$$

$$= \frac{f}{\left\{\left(1+f\right)v_{hotjet} - v_{ac}\right\} + \alpha\left\{v_{coldjet} - v_{ac}\right\}}$$

Aero Engine Operation

Engine Efficiency

- Efficiency of a jet engine at high-altitudes is primary reason for operating in high-altitude environment. Specific fuel consumption decreases as the outside air temperature decreases for constant engine rpm and true airspeed (TAS).
- For efficiency, jet airplanes are typically operated at high altitudes where cruise is usually very close to rpm or exhaust gas temperature limit (EGT).
- At high altitudes, little excess thrust may be available for maneuvering. Therefore, it is often impossible for the jet airplane to climb and turn simultaneously, and all maneuvering must be accomplished within the limits of available thrust and without sacrificing stability and controllability.

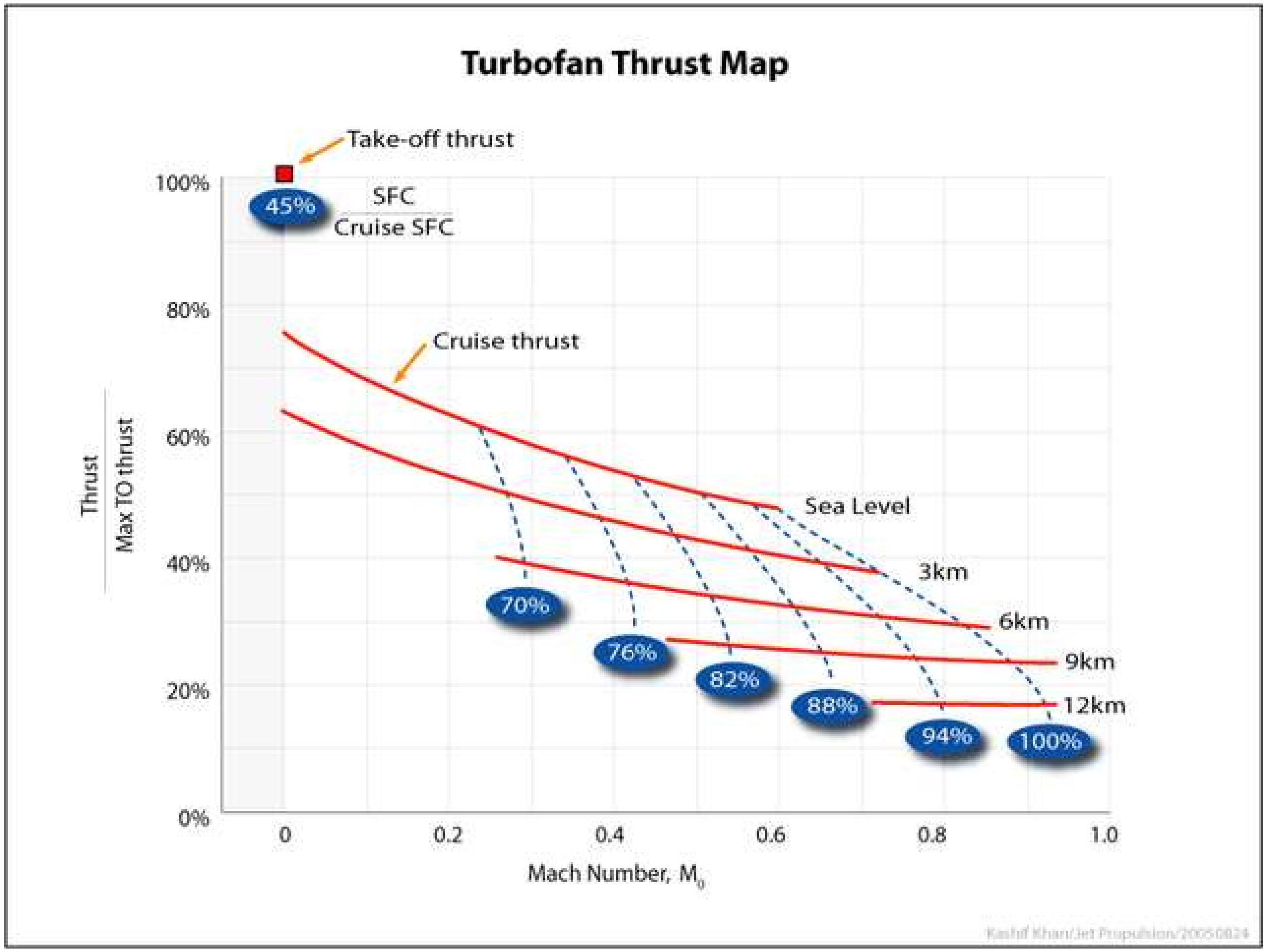
Bildquelle: MTU Aero Engines

Aero Engine Operation

Engine Efficiency

Cruise thrust and specific fuel consumption relation flight Mach number and altitude

Increasing altitude allows higher velocities and thus flight Mach numbers and yields an increase in efficiency



GE90 data sheets

Design parameters for take-off and cruise

RAMPR	inlet air stagnation pressure
FPR	fan pressure ratio (PR)
LPCPR	LP compressor PR
HPCPR	HP compressor PR
OPR	overall PR
Pa	inlet pressure
Ta	inlet temperature
Ca	inlet velocity
BPR	bypass ratio
TIT	turbine inlet temperature
ma	inlet air mass flow rate
Mf	fuel mass flow rate
SFC	specific fuel consumption

	Design Point (Cruise)	Off-Design Point (Take-off)
Height (km)	10.668	0.000
Mach No.	0.850	0.000
RAMPR	1.590	1.000
FPR	1.650	1.580
LPCPR	1.140	1.100
HPCPR	21.500	23.000
OPR	40.440	39.970
P_a (bars)	0.239	1.014
T_a (K)	218.820	288.160
C_a (m/s)	252.000	0.000
BPR	8.100	8.400
TIT (K)	1380.000	1592.000
m_a (kg/s)	576.000	1350.000
THRUST (kN)	69.200	375.300
m_f (kg/s)	1.079	2.968
SFC (mg/N-s)	15.600	7.910
Sp. Thrust (N-s/kg)	120.100	278.100

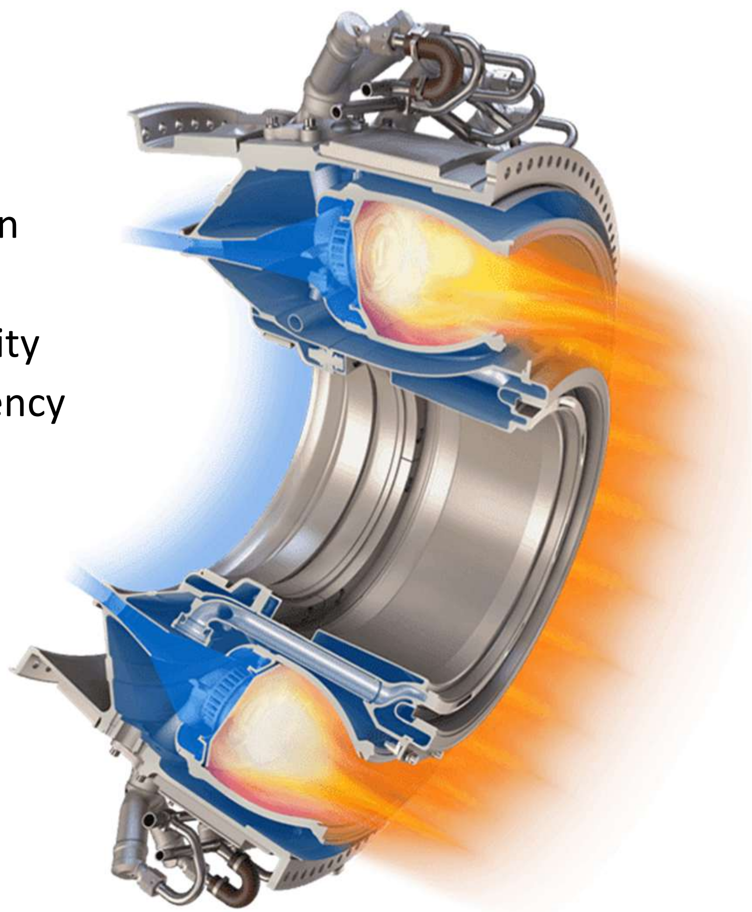
Combustor Requirements and Constraints

Restrictions of Emissions on

- NOx
- Smoke, Particles
- CO
- UHC

have consequences for engine design and operation

- Cold Start
- Altitude Re-Ignition
- Extinction Limit
- Combustion Stability
- Combustion Efficiency

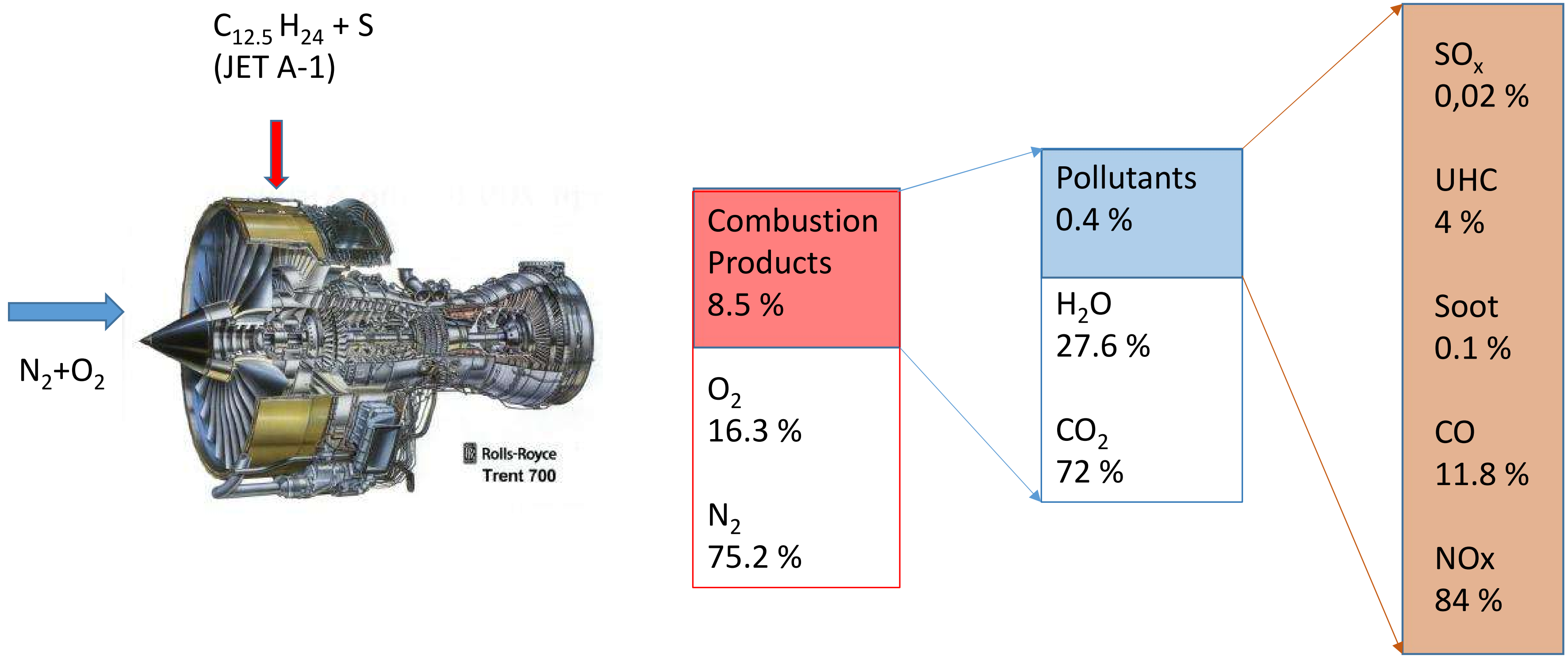


- Weight/Length
- Cost
- Durability

- Pressure Loss
- Cooling
- Exit Temperature Profile

Aero Engine Emission

Typical Composition of Aero Engine Emissions during Cruise



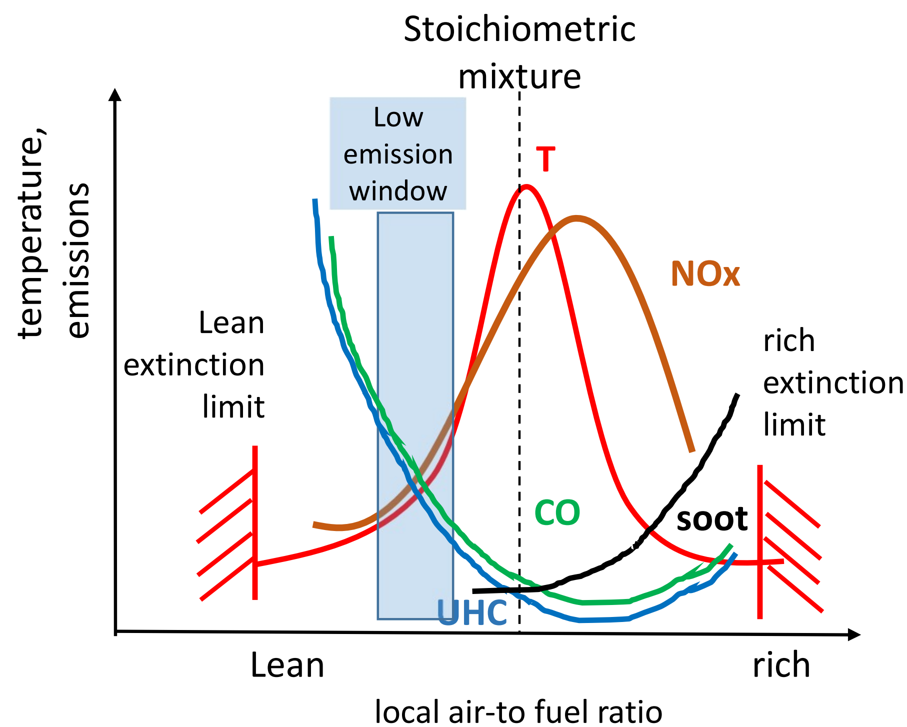
Aero Engine Emission

Competition between soot formation and NOx formation (Between a Rock and a Hard Thing)

- Burning near stoichiometric has highest margin against combustion instability but yields too high temperatures and increases NOx formation
- Too lean reduces temperatures and thus margins for flame-out during transients
- Compromise between too high NOx and too high CO,unburnt hydrocarbons (UHC) and soot.

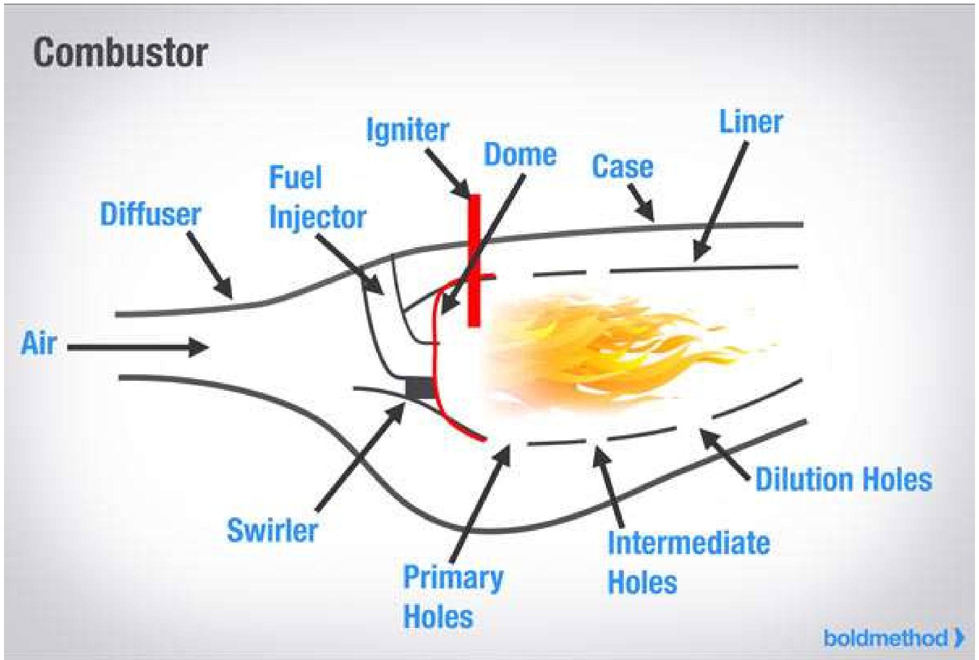
→ narrow operating range

→ During high thrust phases and take-off operation with high NOx emissions.



Engine Ignition

- Most ignition systems consist of two igniter plugs used during ground or air start. After successful ignition it goes either automatically off or is turned off.
- An engine is sensitive to the flow characteristics of the air that enters the intake of the engine nacelle. Abnormal conditions however, due to heavy rain, ice, or possible bird strike, could cause a compressor stall or flameout of the engine.
- Hence, most jet engines are equipped with a continuous ignition system which can be turned on and used continuously whenever the need arises. In many jets such a system is normally used during takeoffs and landings.
- Many jets are also equipped with an automatic ignition system that operates both igniters whenever the airplane stall warning or stick shaker is activated.



Aero Engine Design

Emission Reduction Concepts

Lean Burn Combustion (RR)

Lean burn injector

- Dominates combustion performance
- Partially premixing, up to 70% air flow through injector
- Internally staged with nested pilot injector
- Separated pilot and main stage combustion

Fuel & Control System

- Pilot/main staging
- Potentially circumferential main staging
- Stability control

Combustor

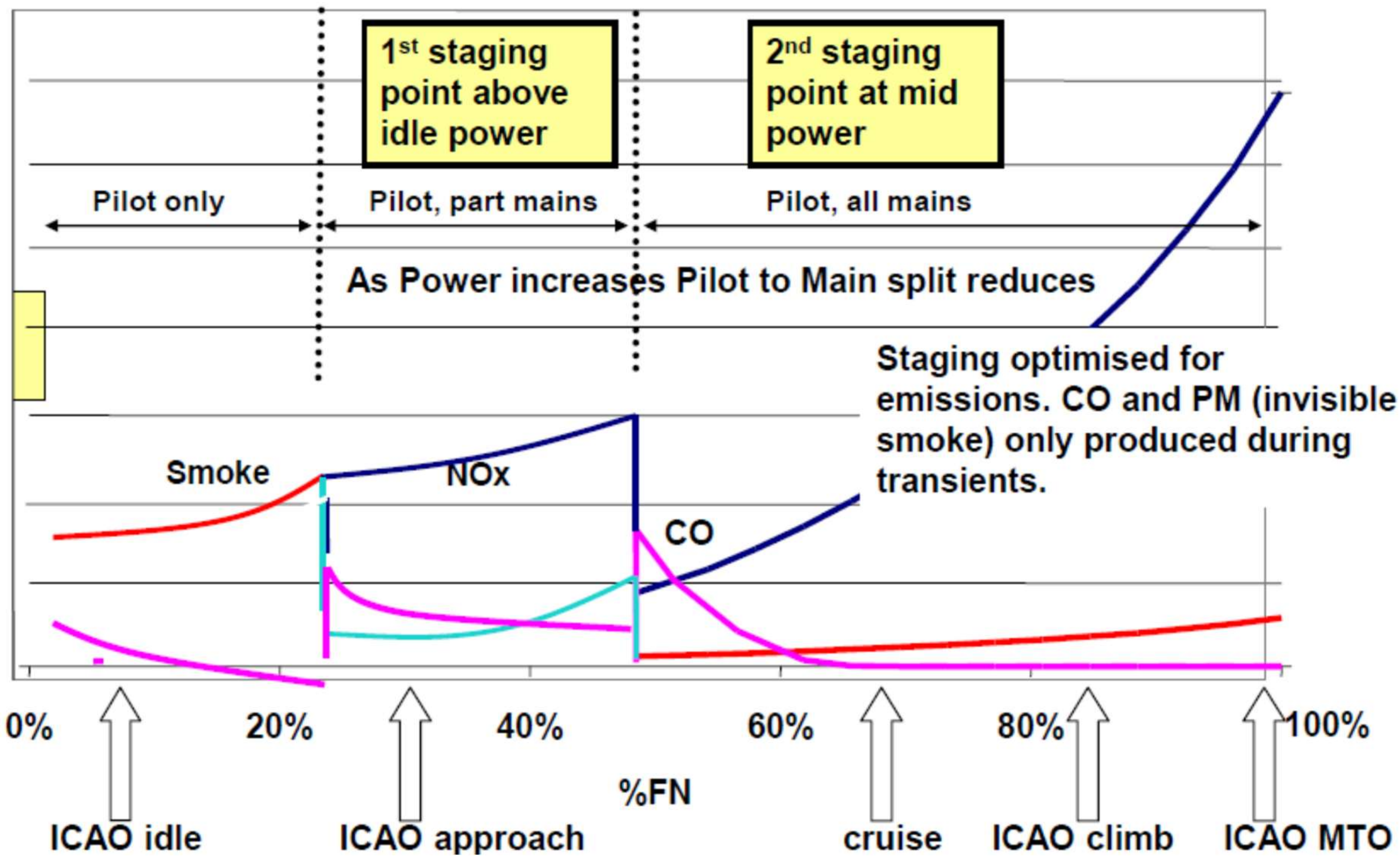
- optimized to enable lean operation and life target
- No mixing ports

Aero Engine Design

Emission Reduction Concepts

Lean Burn Combustion (RR)

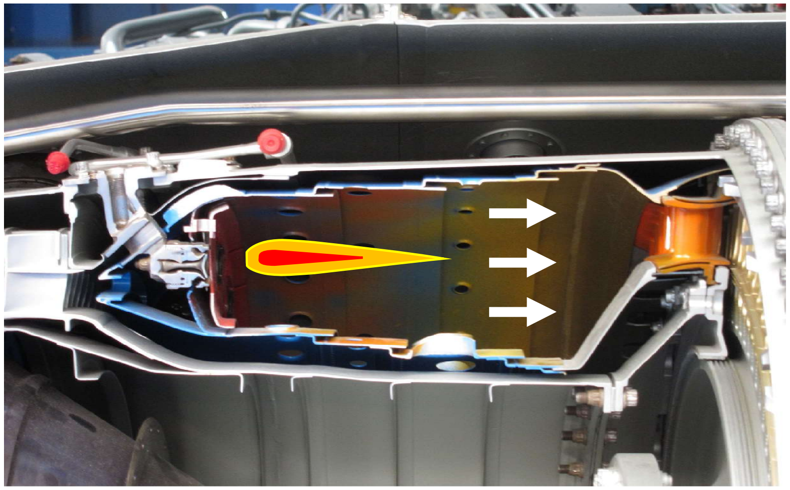
Improvements



Combustion Management

Rich burn – Quick quench – Lean burn (RQL)

- Annular combustor where in the near injector region the flame operates fuel-rich and realizes margins for flame-out at low power levels.
- Secondary air is injected quickly to quench the hot gases and bring the mixture to the lean side.
- Additional air injection to cool the liner and exhaust gas temperature adjustment (TIT)



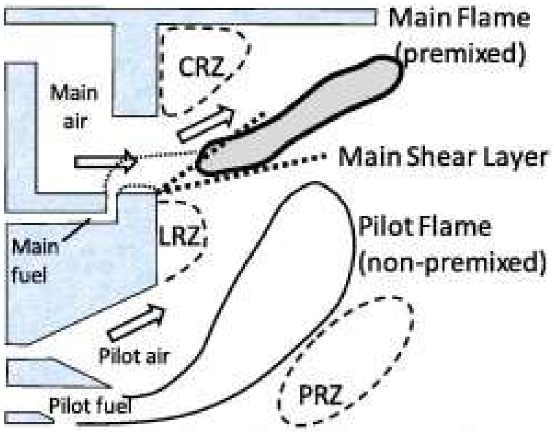
Advantages

- Complete combustion with low pollutant formation (NOx, Soot, CO, UHC)
- Good temperature distribution at exit
- Minimum volume, length and weight
- Low design and maintenance cost
- Stable combustion over a wide range of operation and humidity, rain, hail, birds …)
- Adequate re-light capability on ground and at altitude
- Low noise and acoustics (no „Buzz“ or „Rumble“)

Combustion Management

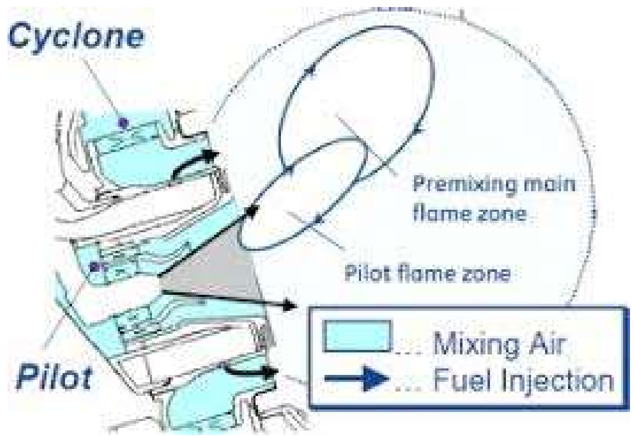
Lean Premixed Pre-vaporized (LPP, GEnx)

- Lean, premixed and pre-vaporized mixture injection in the main injector (swirl-stabilized) and classical injection in pilot burner
- Interaction of main and pilot flames → short combustion zone
- High combustion efficiency at full power with low soot and NOx emissions



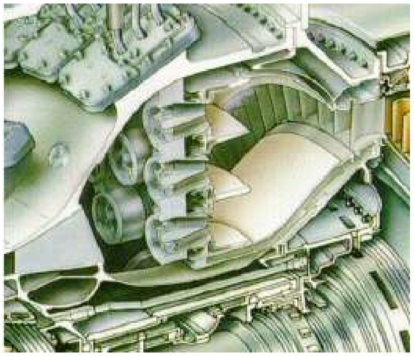
Disadvantages

- Reduced margin against combustion instabilities and lean blow-out
- Possibility of flashback and ignition in premixing zone in particular during transients
- Larger volume (weight)



Combustion Management

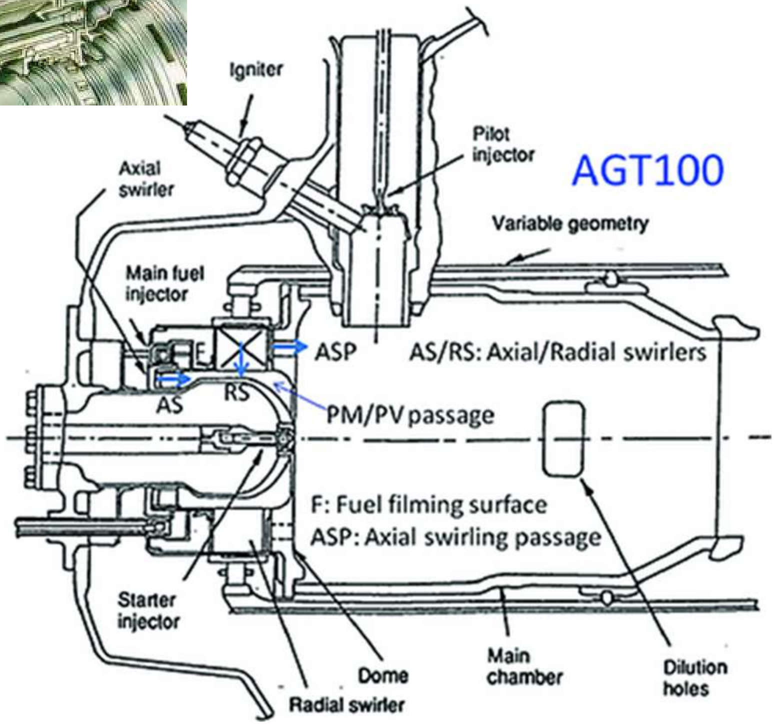
Lean Direct Injection (LDI)



- Injection of lean mixture in multiple swirl-stabilized injectors
- → extremely short combustion zone
- Combination with staged combustion
- In new generation engines (TAPS (GE), ANTLE (RR)

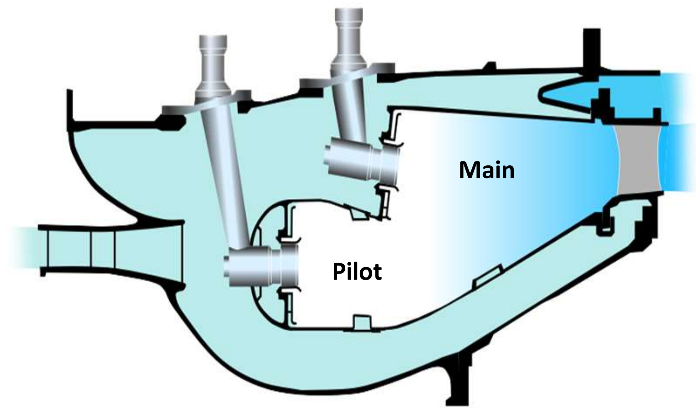
Disadvantages

- Reduced margin against combustion instabilities and lean blow-out a reduced power levels
- Reduced cooling capabilities due to high primary air mass flow rate
- Ignition at lean limit
- Air-blast atomization required
- → staging of air supply necessary



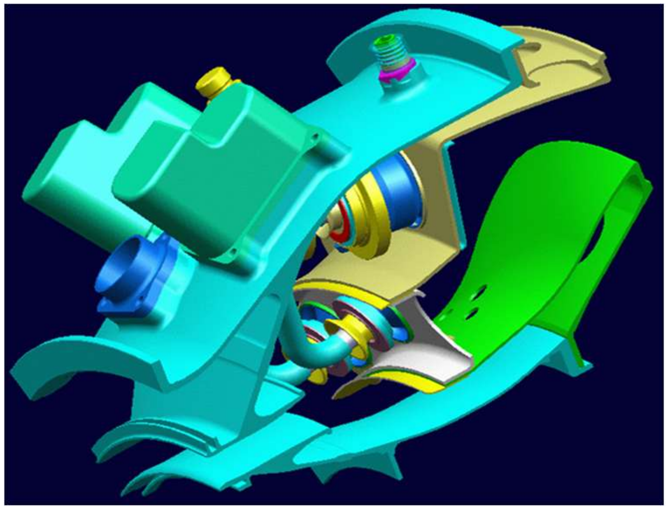
Aero Engine Design

Combustion Management



Staged Combustion

- Pilot injection for high combustion efficiency at low power levels
- main injection for minimum NOx emission at high power levels

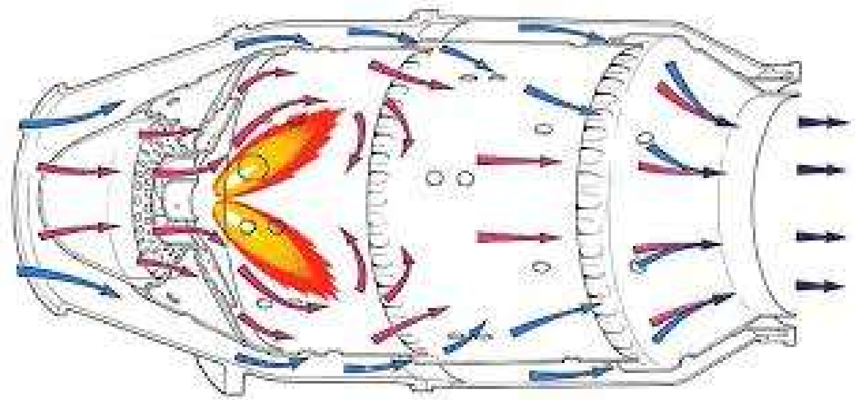


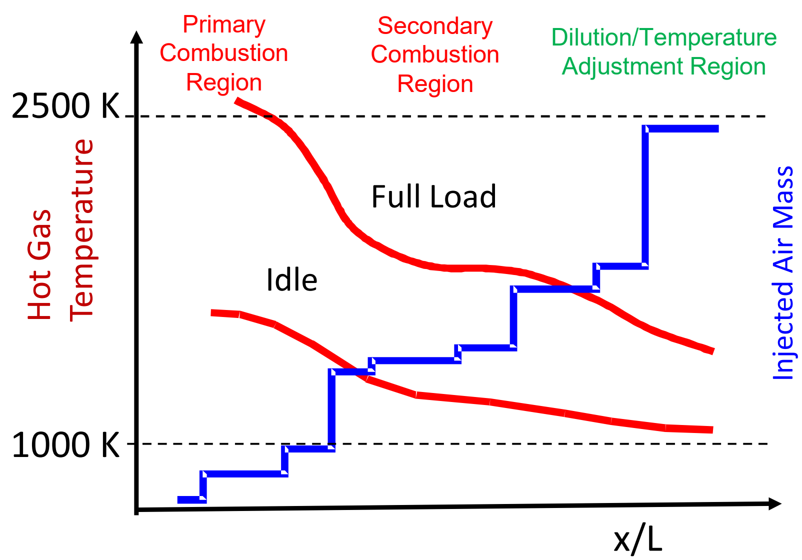
Disadvantages

- Increased complexity (control, parts not any more due to 3D printing)
- Increased CO emissions (quenching at low power levels)
- Increased weight and volume
- High cooling air mass flow rates

Combustion Management

General Concept





Primary Combustion Region:

- Combustion near stoichiometric
- Flame stabilization due to re-circulation of hot reaction products which favor fuel vaporization

Secondary Combustion Region:

- Air injection for lean combustion
- Decrease of hot gas temperature (reduction of CO and NOx) formation
- Primary temperature profile adjustment

Dilution Region:

- Final decrease of hot gas temperature to turbine entry conditions
- Adjustment of temperature profile to avoid hot spots

What you should not forget

- Sample operating conditions of a turbojet and a turbofan engines
- Ideal and real engine cycle
- Aero Engine Components
- Typical mass flow rate of air and fuel through engine and combustion chamber
- Equivalence ratio and emissions
- Methods for emission reduction
- Does an aircraft engine burn lean or rich ?

Rocket Engines

Solid Propellant Rocket Engines

Launcher and Missile Applications

- Classical missiles have only short burning times due to the limited diameter (< 10 s) and are either optimized for thrust, maneuverability or visibility
- Augmenters jet launches from an air-craft carrier
- Ballistic missile have multiple stages

Solid Fuel Ramjet Meteor (MBDA)

- Large boosters for super-heavy launchers
- Strap-on for launcher flexibility
- Stage separation
- Crew escape vehicle

Artemis (Northrop Grumman)

Solid Rocket Engines

Requirements/Challenges

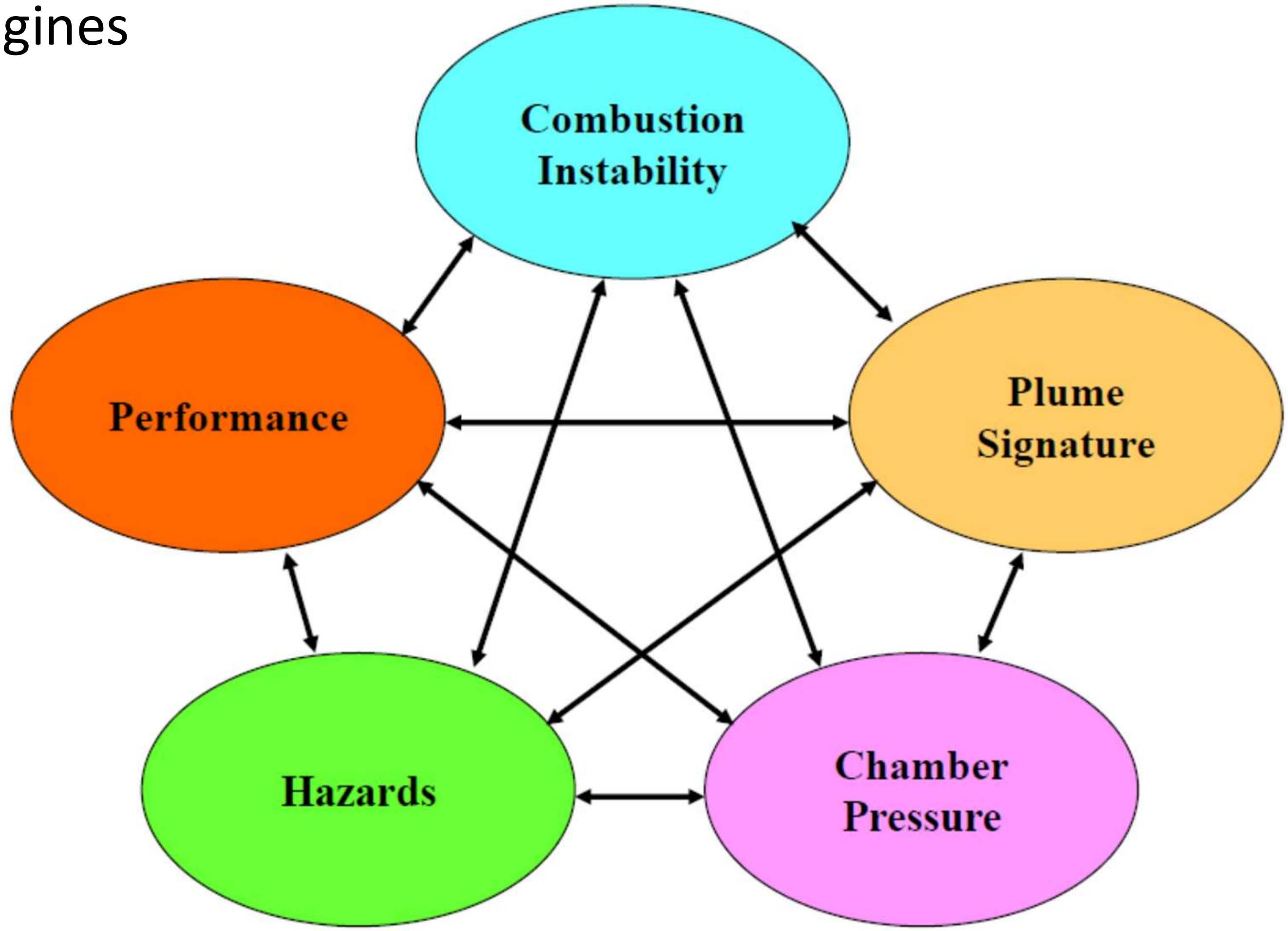
- Space Launch Booster
  - Reliability, Flight Loads, Total Impulse, Vehicle Interfaces,
  - Cost, Ground Handling Loads + ~1,000 Additional Requirements in the Commercial Item Specification
- ICBM Booster
  - Total Impulse, Mass Fraction, NH&S, Envelope, Electrical & Ordnance Interfaces,
  - Storage & Transportation, Reliability, Maintainability, Cost + ~5,000 Additional Requirements Contained in the Prime Item Development Specification
- Tactical Motor
  - Total Impulse, Envelope, Warhead & Fin Section Interfaces, Handling Loads, Hazard Classification, Environmental Loads, Vibration, Snow, Dust, Rain, Fungus, HERO + ~2,000 Additional Requirements Contained in the Weapon System Specification
- Hobby Motor-
  - Cost, Dimensions, Hazard Classification, Special Effects + Additional Requirements Mostly Contained In The Customer's Wallet

Solid Rocket Engines

Solid Propellant Rocket Engines

Design Challenges for Missile Propulsion Systems

- Explosions, toxic ingredients,
- Efficiency, visibility, aging, stability,
- Mechanical integrity, weight



*Almost every missile motor suffered from combustion instabilities during development

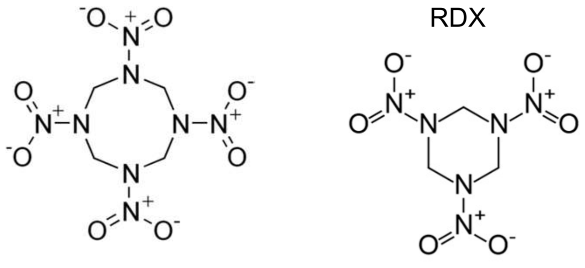
Solid Rocket Engines

Solid Propellant Rocket Engines

Propellant Combinations

Oxidizers

- Ammonium Perchlorate NH_4ClO_4 (AP)
- Nitronium Perchlorate NO_2ClO_4 (NP)
- Cyclotrimethylene-trinitramine $C_3H_6N_6O_6$ (RDX)
- Cyclotetramethylene-tetranitramine $C_4H_8N_8O_8$ (HMX)



Fuels

- Aluminum
- Magnesium
- Boron

Binders

- Hydroxyl-terminated Poly-butadiene (HTPB)
- Polybutadiene acrylic acid acrylonitrile prep-olymer (PBAN)

Typical AP-based Propellant:

- 18 % Aluminum
- 69% AP
- 10% Binder
- 2 % Isodecyl Pelargonate (Plasticizer)
- 0.5 % Iron oxide (Burn Rate Catalyst)
- 0.2 % Aziridine (Bonding Agent)
- ……….

Solid Propellants Propellants / Additional Ingredients Issues

- Processing
  - Ingredient, density, reactivity, particle size/packing fraction, solids, loading effects
  - Mix procedure, mix temperature(s), order and number of addition, blade rpm, blade times, vacuum mixing, hold/purge times
  - End-of-mix viscosity, potlife, rheological behavior, castability
  - Cast technique, cure temperature, pressure and time, cure kinetics
- Aging behavior
  - Oxidative cross-linking (hardening), hydrolytic (typically softening), decomposition, reaction with contaminants
  - Effects of temperature, humidity, vacuum, atmosphere
  - Plasticizer migration, oxidizer dissolution, or recrystallization, stabilizer depletion
  - Effect on performance, ignition, mechanical, bondline, ballistic, and hazard properties

Solid Propellants

Propellant Choice

Is application-specific and depends on

- Burn Rate
- Mechanical Properties
- Operational Temperature Range
- Hazard Classification
- Aging
- Toxicity
- Signature
- Insensitive Munitions Compliance
- Throttling Requirements
- Rheology / Processability
- Production Cost

Steady State Burn Rate $R = a\, p_c^{\,n}$

Dynamic Burning Rate R_b

$$\frac{R_b}{R_{b0}} = 1 + 2\frac{\lambda}{c_p \rho} \cdot \frac{n}{a^2} \cdot \frac{1}{p^{2n+1}} \cdot \frac{dp}{dt}$$

c_p) specific heat, ρ) density, λ) thermal conductivity,
n) pressure exponent and a) burning rate constant

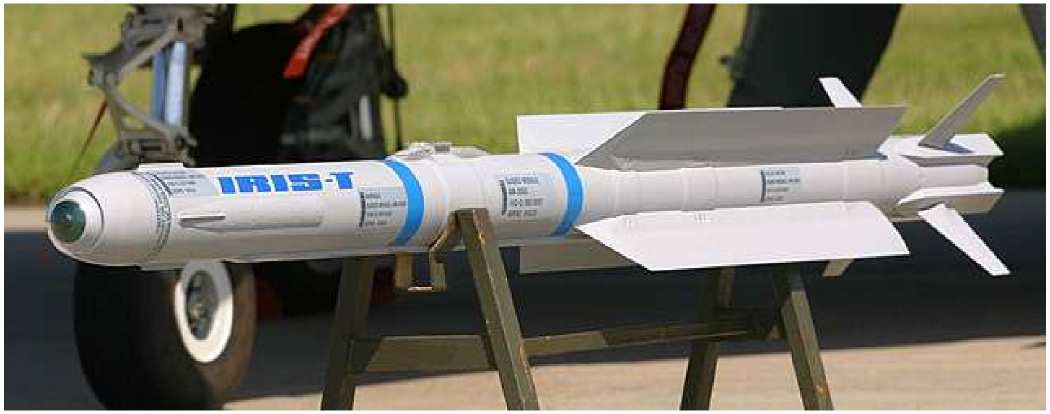
In case of fast transients the propellant temperature layer isn't in steady state and the burning rate is effected

- Low to high pressure transient results in a thicker thermal and the burning rate increases
- High to low transient yields a thinner thermal layer and the burn rate decreases

Solid Propellant Rocket Engines

Missile Propulsion

- Optimized for thrust, maneuverability and/or visibility
- Large fins and sometimes thrusters for high agility
- Propellant Grain according to mission requirement



Solid Fuel Missile IRIS-T (air-to air missile)
(Consortium led by Diehl BGT)

Solid Propulsion Basics

- Thrust ~ motor length
- Burn time ~ motor diameter

Solid Rocket Motors

Solid Propellant Rocket Engines

Launcher Propulsion

- Optimized for thrust and required thrust variation
- Minimum thrust vector control requirements

Propellant mass:	240 to
Motor mass:	26 to
Thrust :	590 to
Propellant:	AP, HTPB, Al
Burn time:	123 s
Isp:	275 s

ARIANE 5 EAP

Solid Rocket Propellants

Solid Propellant Combustion

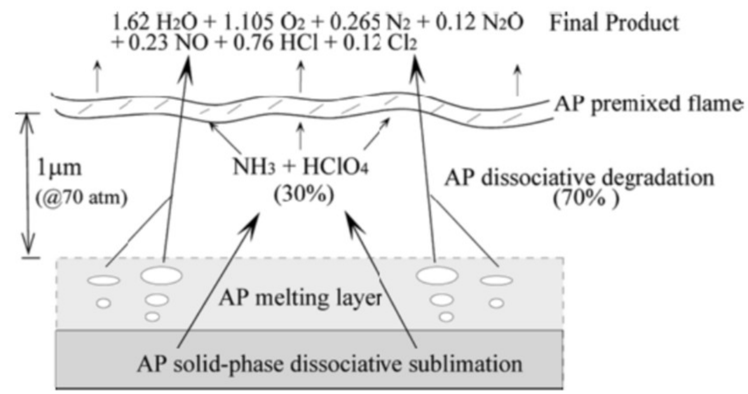
AP decomposes exothermally at pressures above 2 MPa and sufficient temperatures

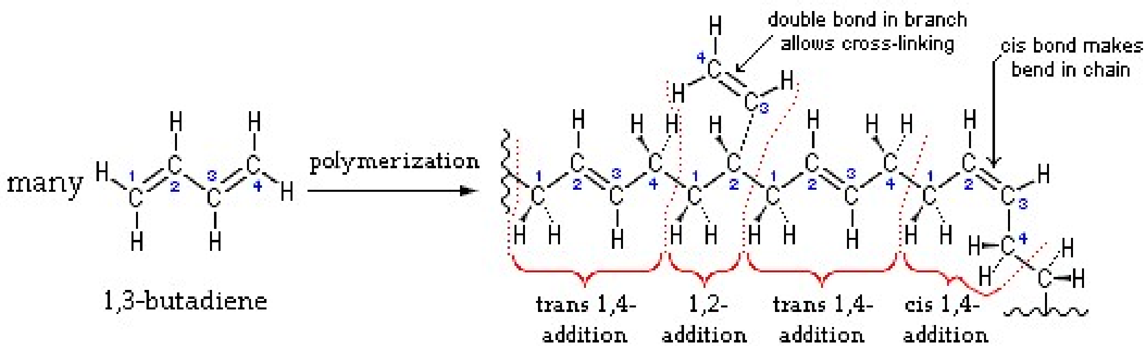
high temperature path

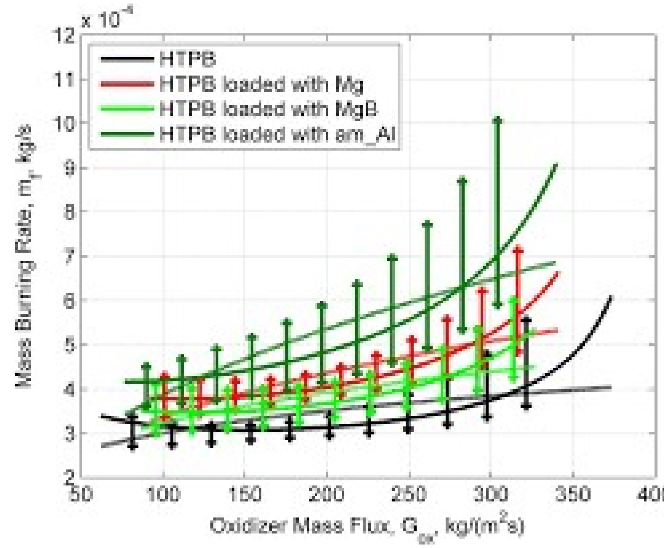
$$NH_4ClO_4 \rightarrow N_2O\ (major) + NO_2 + HCl + HNO_3 + Cl_2 + O_2 + N_2$$

$$NH_4ClO_4 \rightarrow NO_2\ (major) + N_2O + HCl + HNO_3 + Cl_2 + O_2 + N_2$$

Low temperature path







Solid Rocket Propellants

Solid Propellant Emissions

Hydrochloric acid (HCl) emissions

- One Ariane 5 launch ~ 100 to
- One VEGA launch ~ 26 to

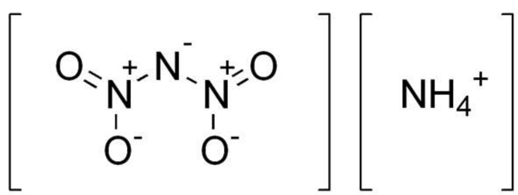
However, the number of launcher is small (< 10 / year)

Nitric acid (NO_2) emissions

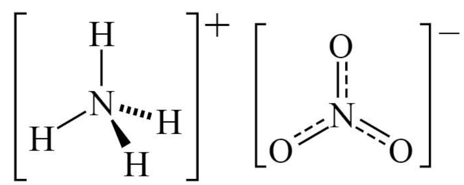
- One Ariane 5 launch ~ 40 to
- One VEGA launch ~ 10 to

Recent research aims at replacing AP with mixture of ADN and AN and keep the Al concentration almost constant

Ammonium-dinitramide, $NH_4N[NO_2]_2$ (ADN)



Ammonium-nitrate, NH_4NO_3 (AN)



There isn't a standard formulation yet since solid propellant with AND/AN mixture have typically large pressure exponents which is a problem for stable combustion and ignition.

Liquid Rocket Engines

Engine Characterization

Application
- Booster
- Core
- Upper stage
- Satellite

Engine Cycle / Propellant Supply
- Pressure-fed
- Pump-fed
  - Gas generator
  - Fuel-rich staged combustion
  - Oxidizer-rich staged combustion
  - Full flow cycle

Propellants
- Earth storable (hypergolic)
- Semi-cryogenic
- Cryogenic

Performance
- Low
- Medium
- high

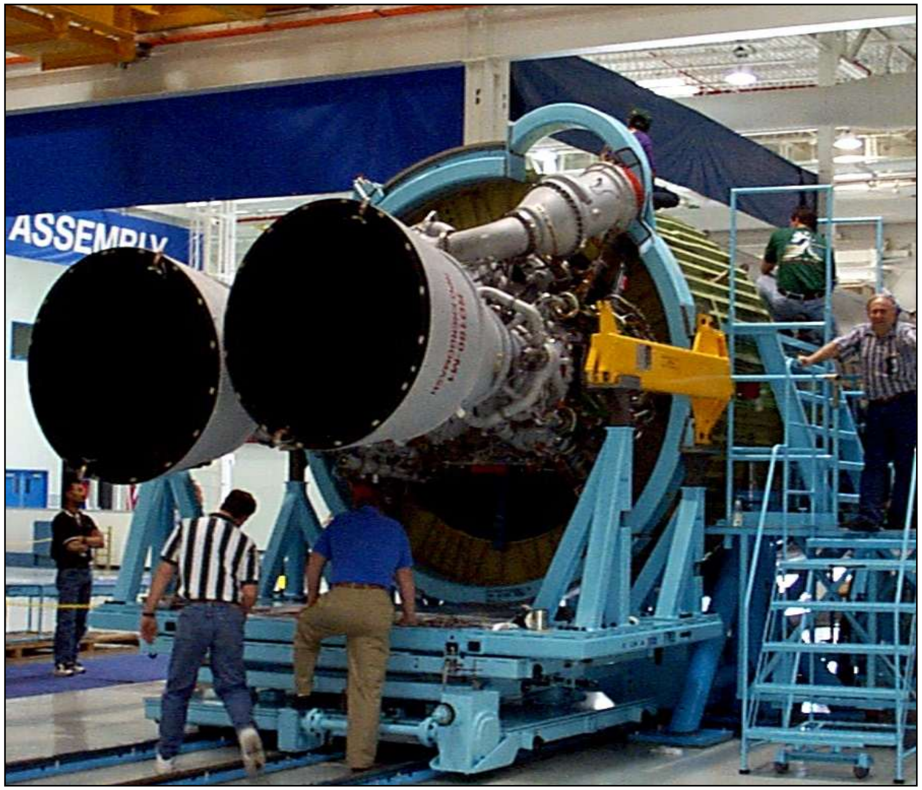
Vulcain 2: Cryogenic (LOX/LH_2)
Gas Generator Engine

Liquid Rocket Engines

Booster Applications

Optimized for high thrust with feasible specific impulse

- Quite often LOX/kerosene
- GG and staged combustion cycles
- Limited nozzle expansion ratio (~20, only very high pressure engines (~ 250 bar, RD-170 family) have expansion ratios ~ 35
- Limited burn time (< 200 s)
- Burn substantial amount of propellant (often > 70% of total propellant mass)



RD-180 engine of ATLAS 5, semi-cryogenic (LOX/kerosene); Oxygen-rich staged combustion

Liquid Rocket Engines

Core Stage Engines

Generally optimized for high specific impulse with moderate thrust levels

- Almost always LOX/LH2, seldom LOX/Kerosene (older launch vehicles, often derived from ICBMs even use storable propellants (UDMH))
- GG and staged combustion cycles
- Long burn time (> 500 s)
- Nozzle expansion ratios at the edge of flow separation and buffeting during lift-off and near sonic flight

→ large thrust penalty at lift-off due to over-expansion

Vulcain 2 engine of Ariane 5, cryogenic (LOX/LH2); Gas Generator Cycle

Liquid Rocket Engines

Upper Stage Engines

Generally optimized for highest possible specific impulse with small thrust levels

- Almost always LOX/LH2, seldom LOX/Kerosene (older launch vehicles, often derived from ICBMs even use storable propellants (UDMH))
- Pressure-fed (AESTUS) , GG (HM-7B), expander cycle (RL-10,YF 75 D) and staged even combustion cycles (RD-0124)
- Long burn time (often > 700 s)
- Very large nozzle expansion ratios (~200), sometimes expendable nozzle parts (RL-10 B, RD-0146)

RD-0146 supposed to fly on Angara 5, LOX/LH2 expander cycle engine

Liquid Rocket Engines

Idealized thermal rocket (1-D Approach)

Thrust

$$F = \dot{m} v_e + (p_e - p_a) \cdot A_e = \dot{m} c_e$$

$$F = \underbrace{p_c A_{th} \cdot \Gamma \cdot \sqrt{\frac{2\kappa}{\kappa - 1}\left[1 - \left(\frac{p_e}{p_c}\right)^{\frac{\kappa-1}{\kappa}}\right]}}_{\text{(i)} \; = \dot{m} \cdot v_e} + \underbrace{p_e A_e}_{\text{(ii)}} - \underbrace{p_a A_e}_{\text{(iii)}}$$

(i) Newton‘s thrust part based on exhaust gas mass

(ii) Pressure force in the exhaust gas at nozzle exit (positive)

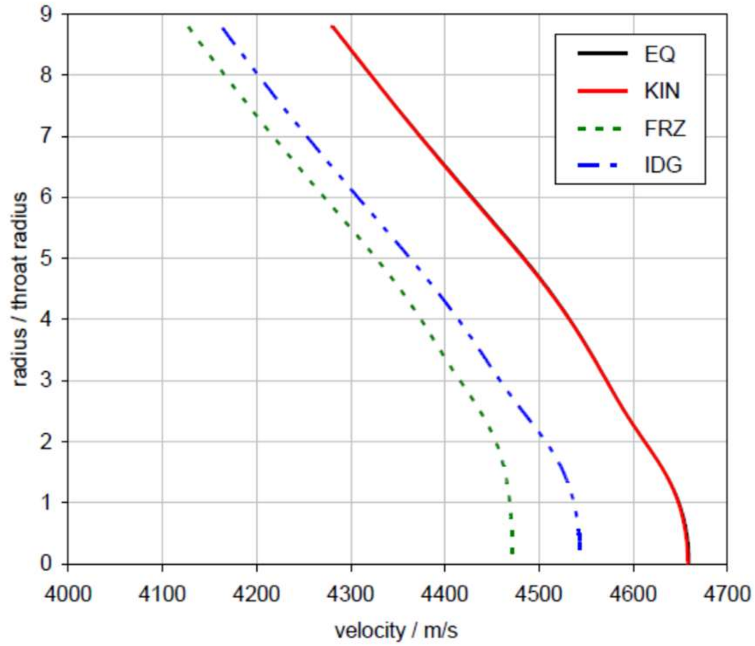
(iii) Pressure force of ambient gases at nozzle exit (negative)

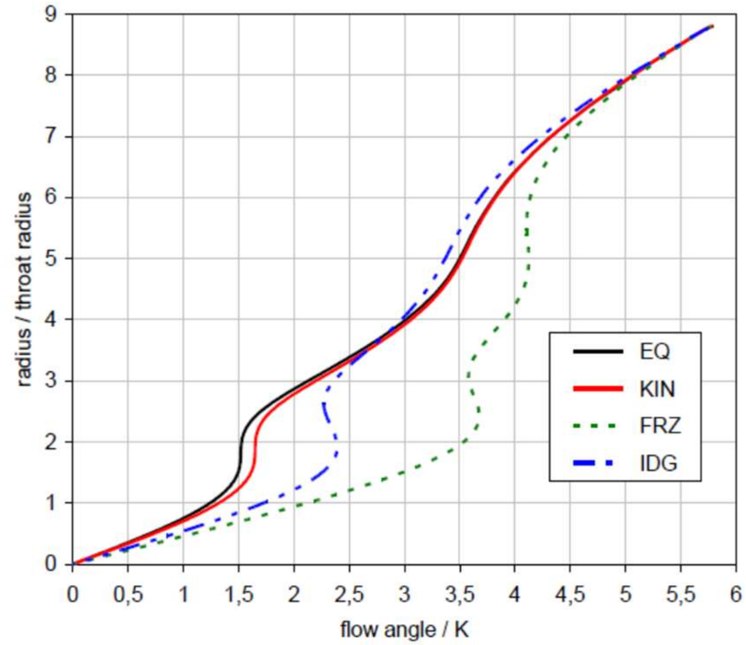
*Over-expansion ($p_a > p_e$ results in a thrust penalty

Effect of Deviations for 1-D Flow

Numerical simulation of SSME (2/2)

TDK-94, 200 bar,
Rof=6, ε=69





Variation of velocity (left) and flow angle (right) distribution at nozzle exit as a function of chemical equilibrium (EQ), finite rate kinetic (KIN), frozen (FRZ), ideal gas (IDG) assumption applied for the simulations

Propellants

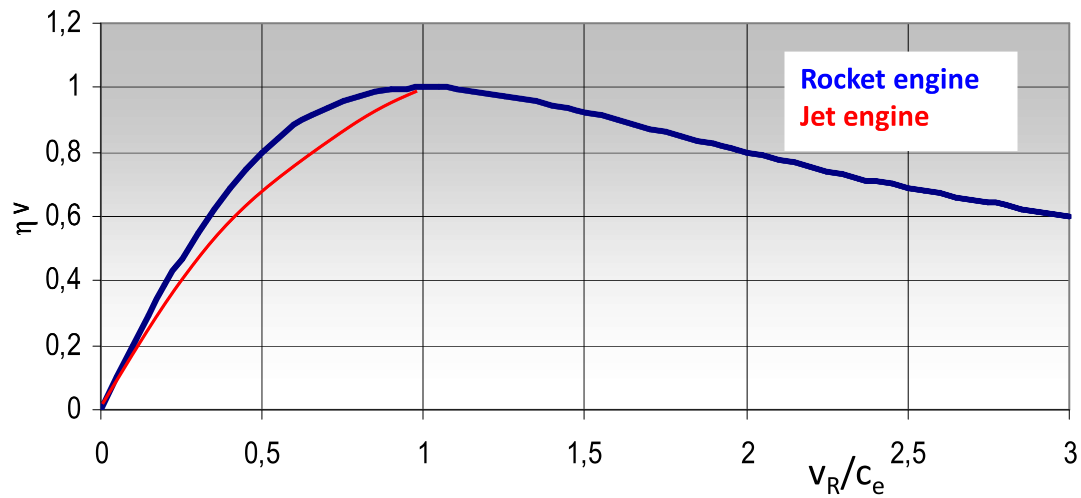
Propellant / Engine Performance Data

Engine	Propellants	O/F	p_c [bar]	Vac. thrust [kN]	I_{sp} vac. [s]	Mass [kg]
Vulcain 2	LOX/LH_2	6,1	115	1350	434	2100
SSME(Bl. 2)	LOX/LH_2	6,0	206	2278	452	3526
RS68	LOX/LH_2	6,0	96	3312	420	6800
VINCI	LOX/LH_2	5,8	60	180	465	550
RL 10A	LOX/LH_2	5,5	39	99	451	172
RD 170	LOX/Kerosene	2,63	245	8060	337	9750
RD 253	NTO/UDMH	1,90	147	1670	316	1280
Aestus	NTO/MMH	2,05	11	30	324	111

Propulsive Efficiency

$$P_U = F \cdot v_R = \dot{m} \cdot c_e \cdot v_R \qquad P_L = \frac{\dot{m}}{2}\left(c_e - v_R\right)^2$$

$$\boxed{\eta_V = \frac{P_U}{P_U + P_L} = \frac{2\frac{v_R}{c_e}}{1 + \left(\frac{v_R}{c_e}\right)^2}}$$



P_U propulsive power

v_R trajectory rocket velocity

P_L power loss due to velocity difference between launcher and exhaust gas

P_A total jet power power (P_U + P_L)

h_V propulsive efficiency

➔ is a measure for the efficiency of the transformation process of jet power to propulsive power

Rocket Engines

What you should not forget

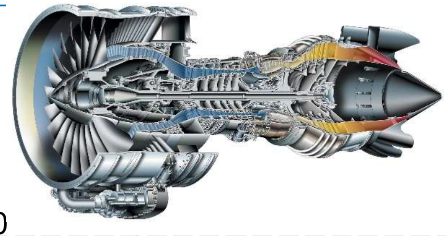
- Sample applications for solid rocket motors (SRMs)
- SRM design challenges
- Typical propellants and specific impulse of SRMs
- What would you do to increase thrust or burn time of a SRM ?
- Key issues of hybrid propulsion systems
- Applications of liquid rocket engines (LREs)
- Engine Cycles
- Ideal thrust and deviations: what are the reasons ?
- Engine Efficiencies

Annex

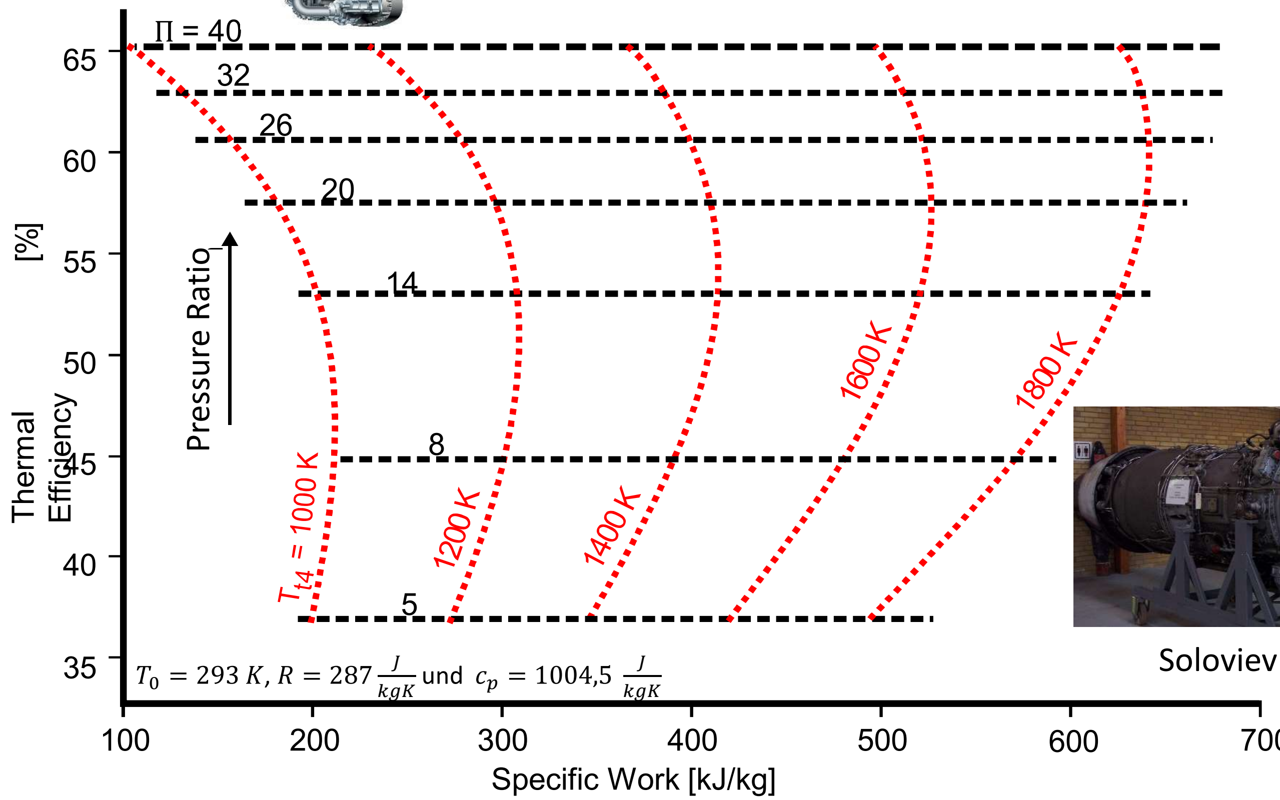
Aero Engine Operation

Cycle Analysis

- Specific work increases with turbine inlet temperature (TIT)
- Thermal efficiency increase with pressure ratio
- Civil aircraft strive for thermal efficiency
- Military aircraft for specific work



PW 6000 (A 318)

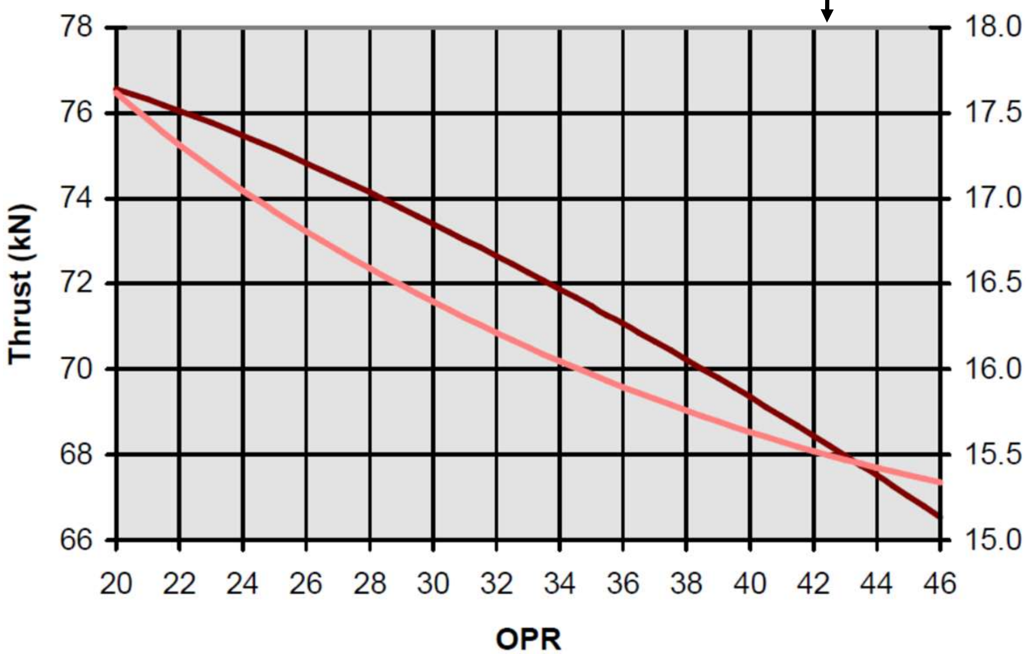


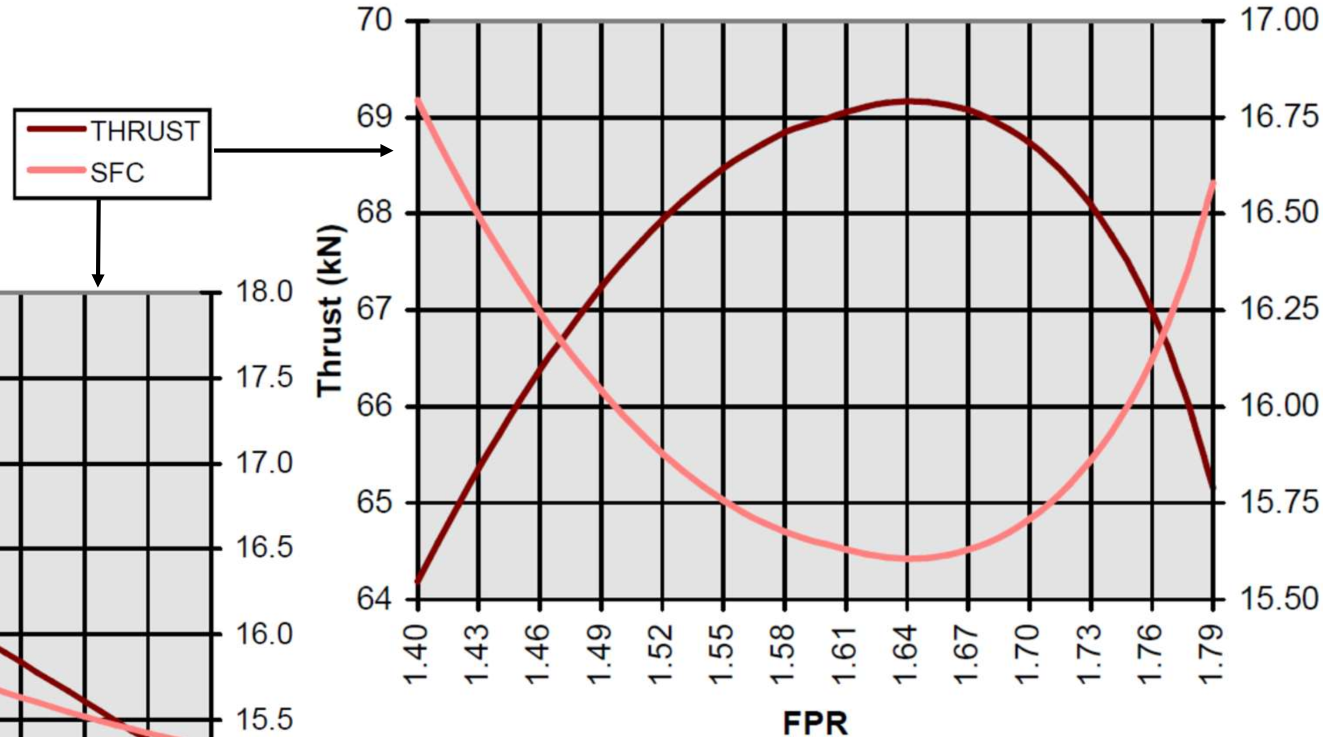
Soloviev D-30 (MIG 31)

Aero Engine Operation

GE90 data sheets

Relations between Thrust or Specific Fuel Consumption and Overall Pressure Ratio



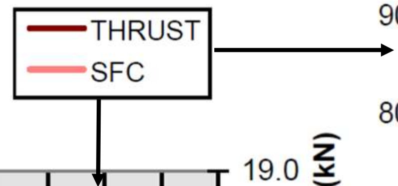


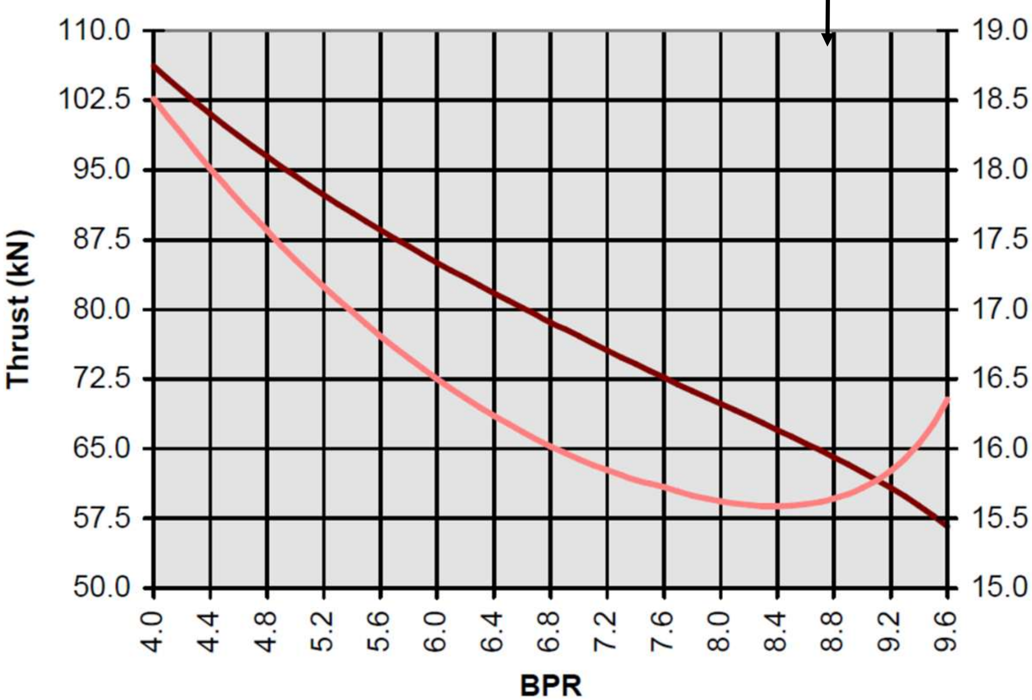
Typical Relations between Thrust or Specific Fuel Consumption and Fan Pressure Ratio

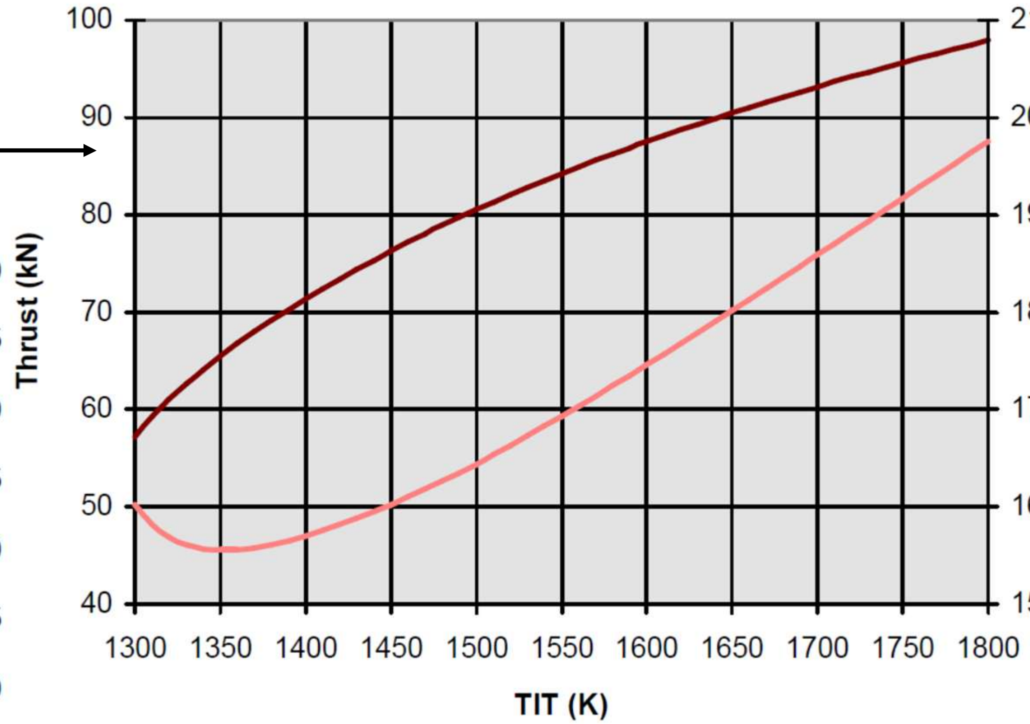
Aero Engine Operation

GE90 data sheets

Typical Relations between Thrust or Specific Fuel Consumption and Bypass Ratio



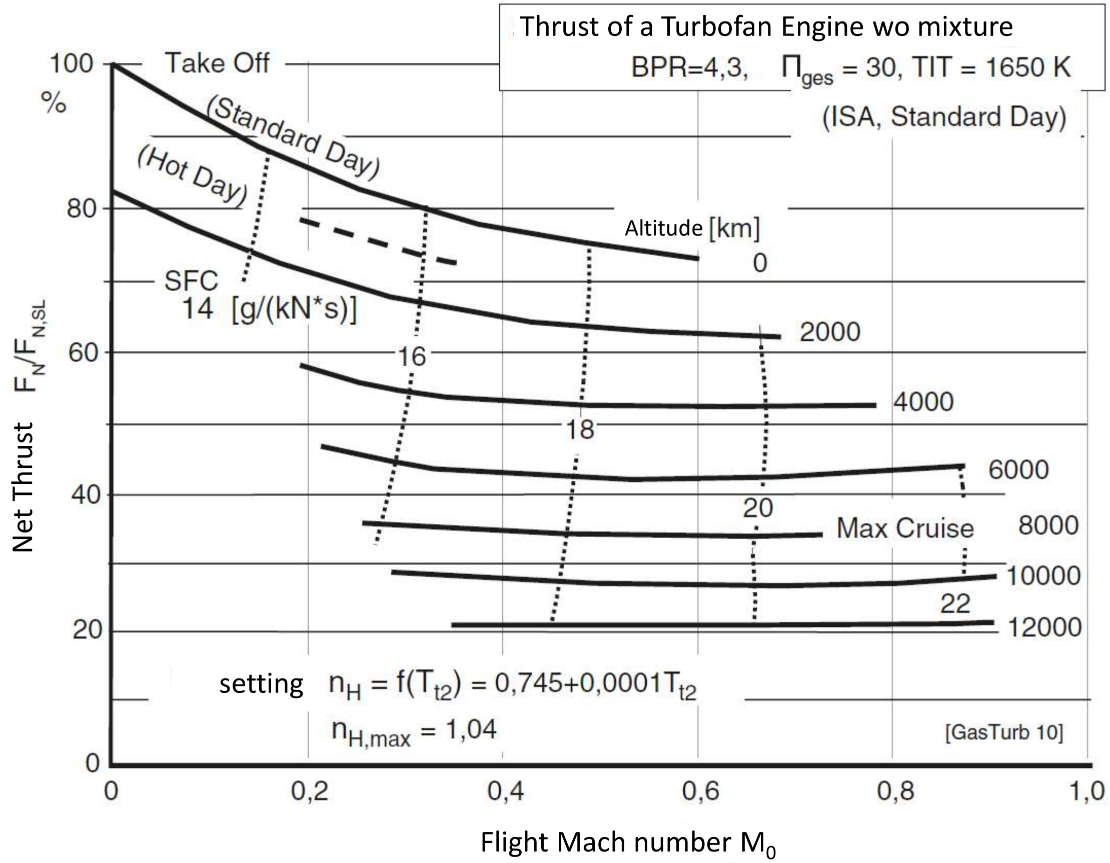




Typical Relations between Thrust or Specific Fuel Consumption and Turbine Inlet Temperature

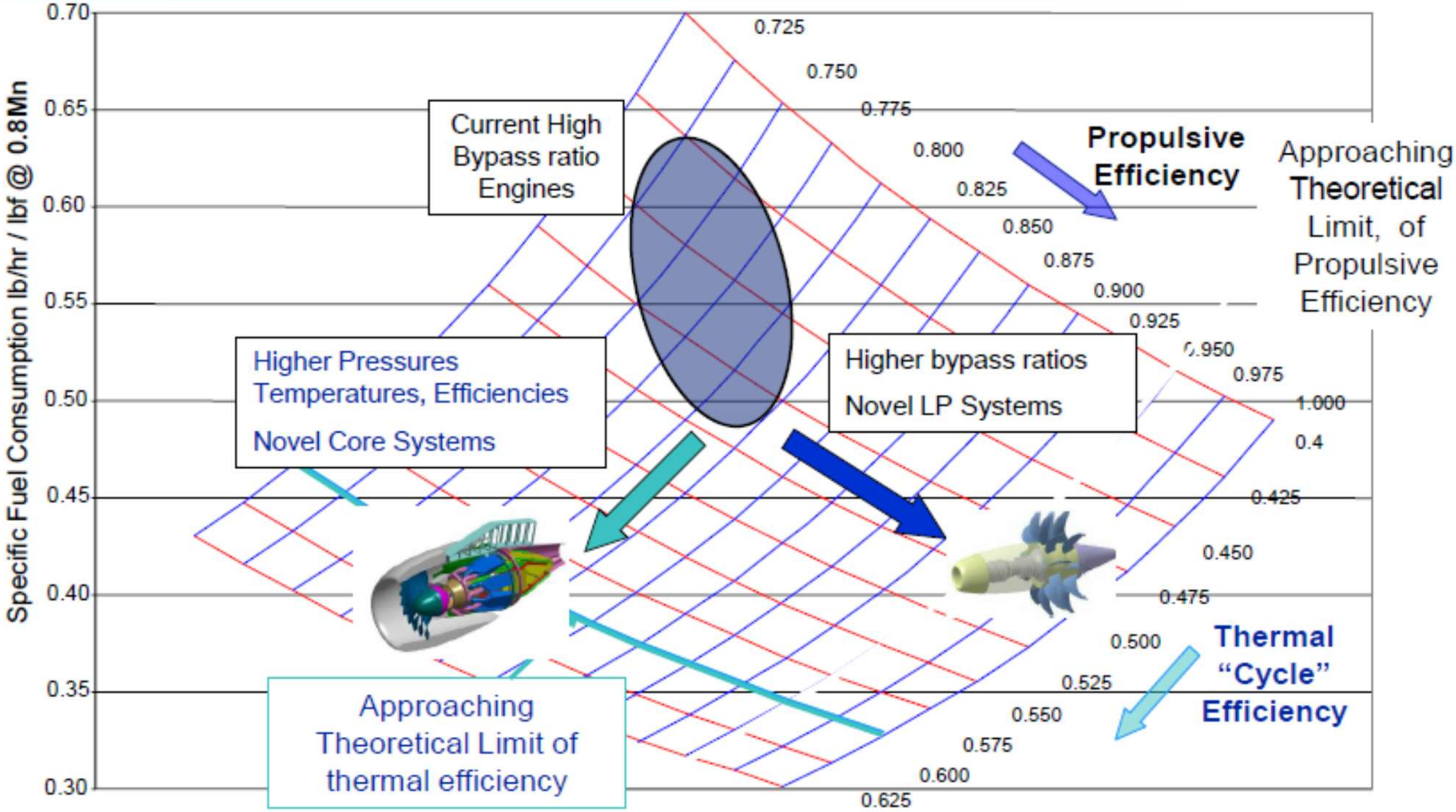
Aero Engine Operation

Turbo Fan Engine Thrust Map



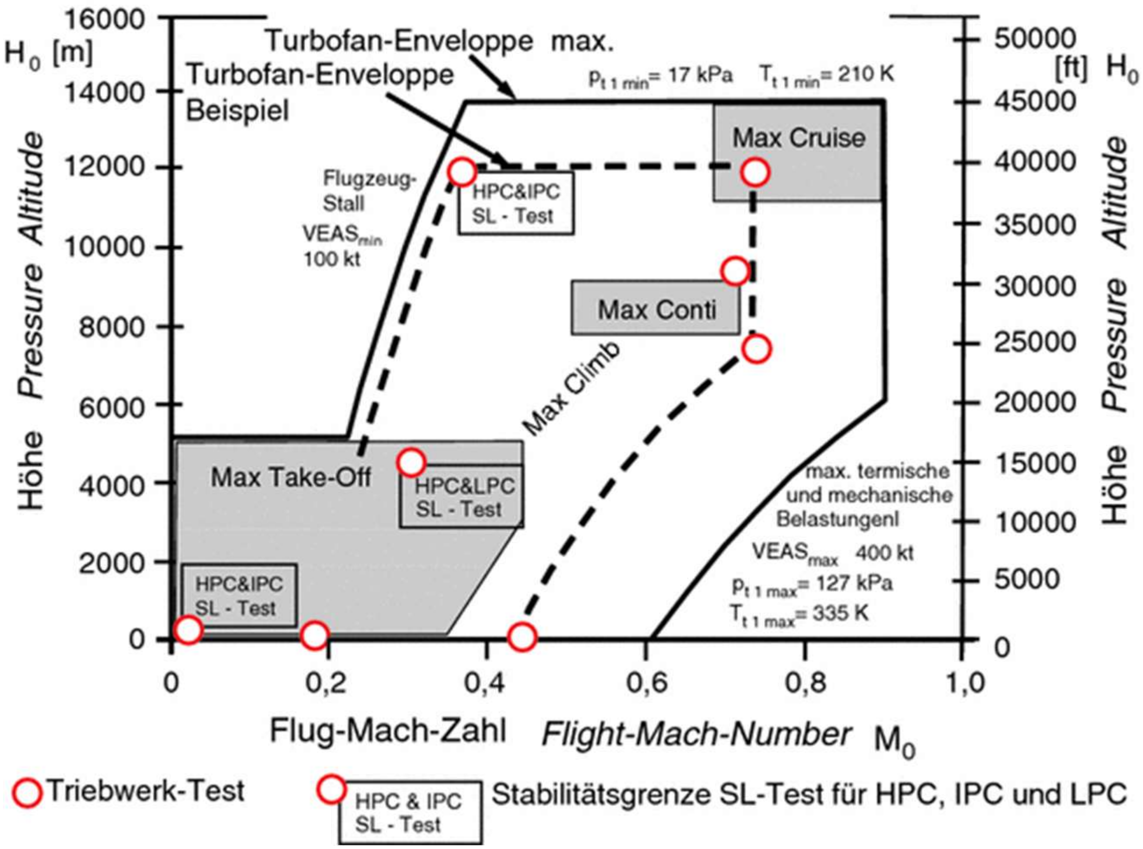
Quelle::Rick, Gasturbinen und Flugantriebe

Aero Engine Operation



Aero Engine Operation

Flug-Enveloppe *Flight Envelope* Turbofan

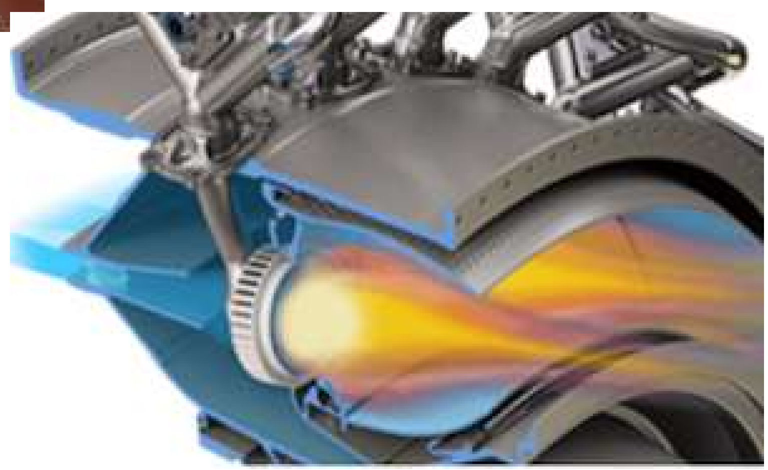


Aero Engine Design

Can Combustor → Annular Combustor → Double Annular Combustor

Old fashioned combustor exists as well as can-annular (can in an annular ring) arrangement

Different variants existing with pilot and main injectors

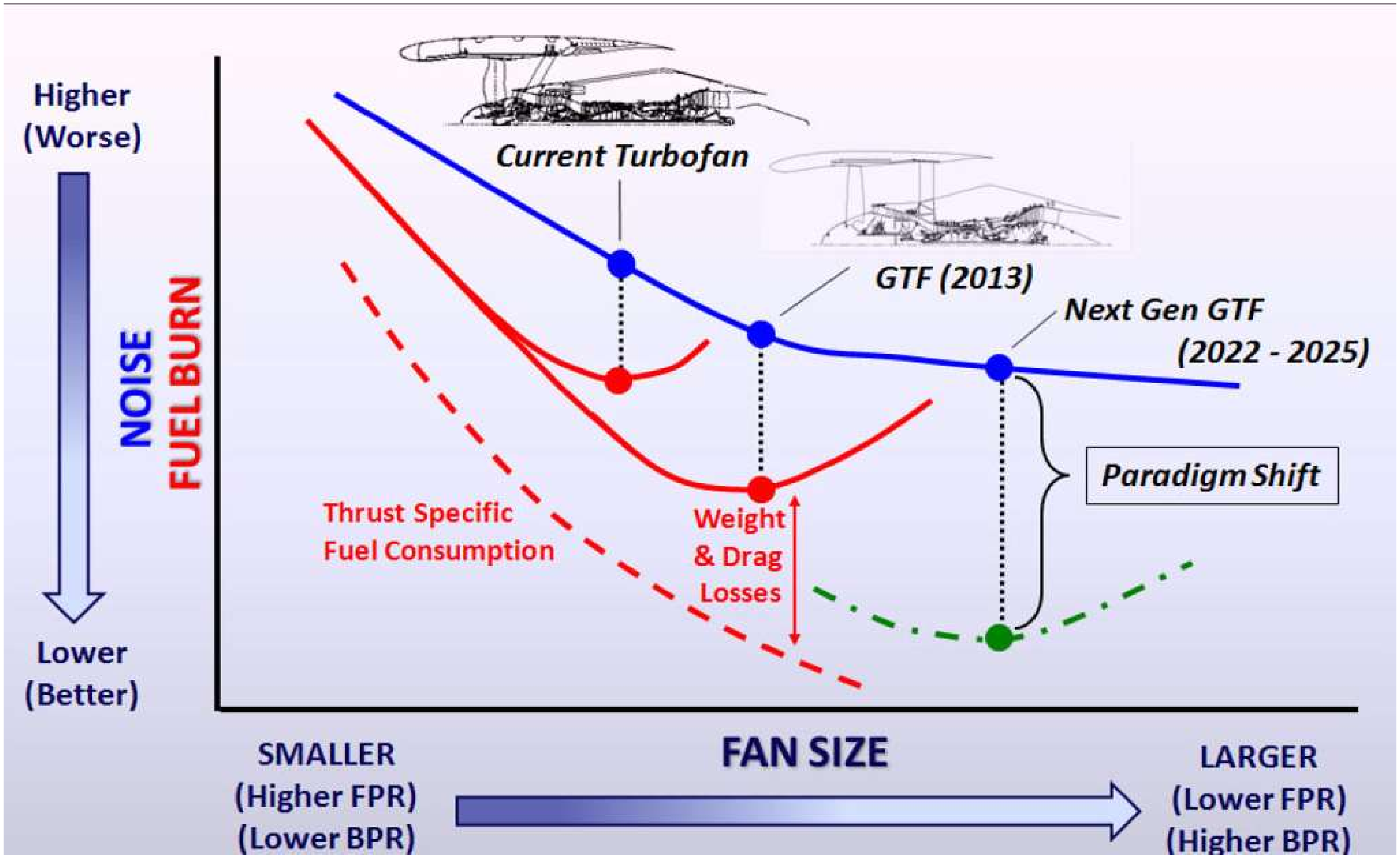


At low power levels only one annual combustor works at higher levels both operate

Engine Design

Geared Turbofan

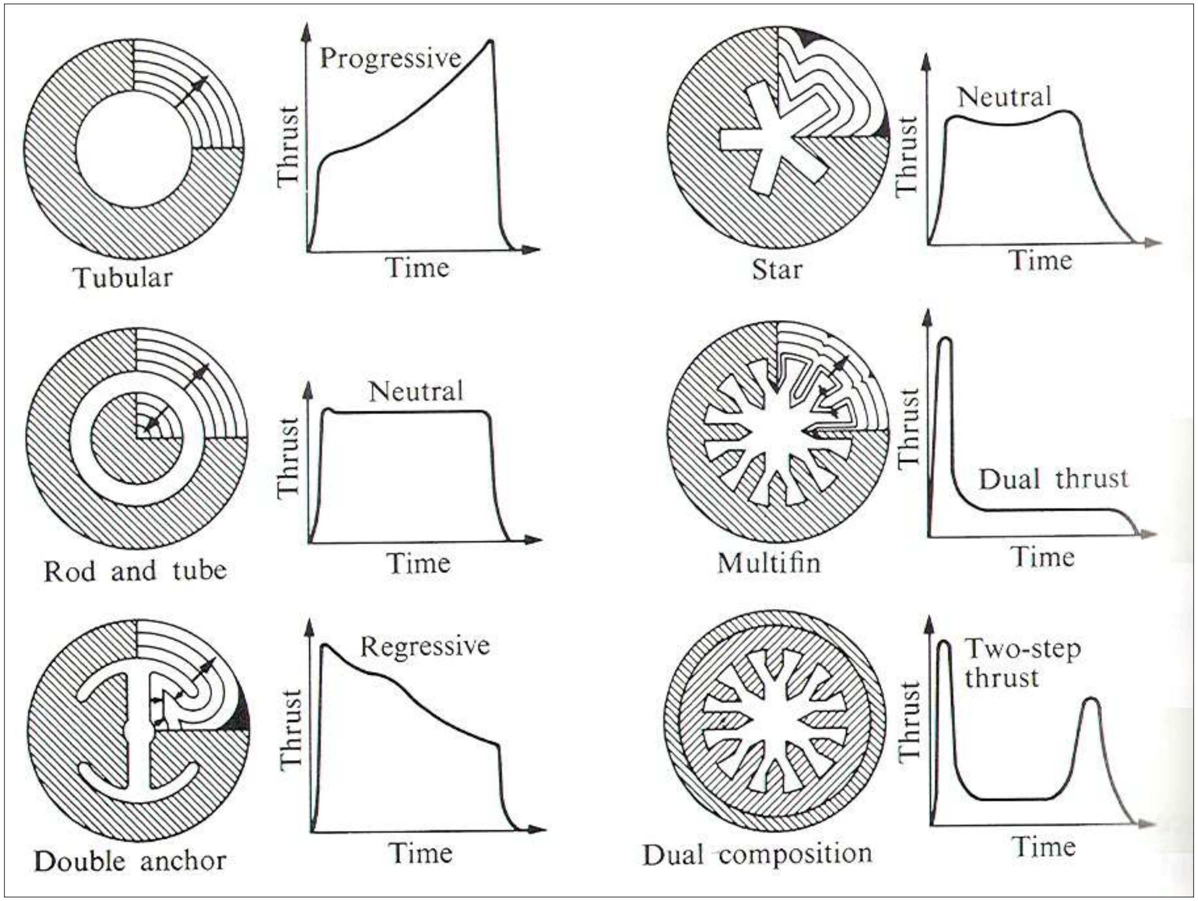
This concept is seen as a major breakthrough since it allows to optimize fan, compressor and turbine operation



Solid Rocket Engines

Propellant Grain Design

- Thrust variations can easily be implemented. However, the design once manufactured is and there is no possibility to change that again.

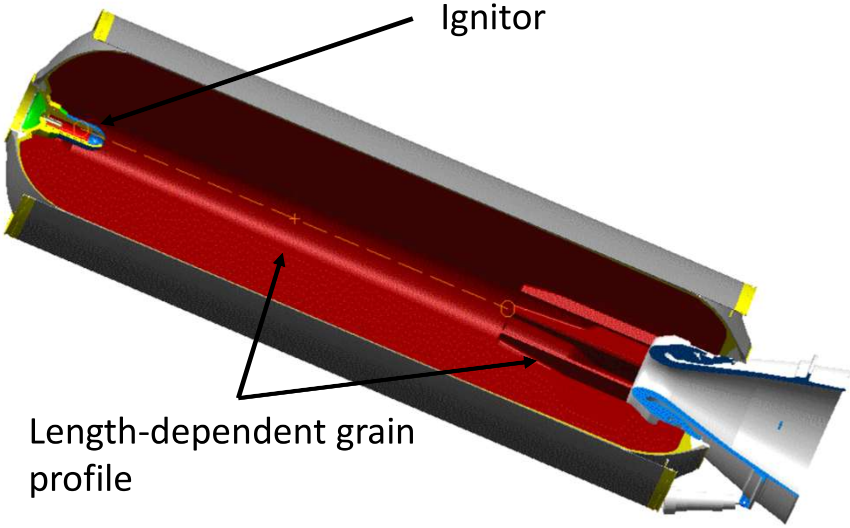


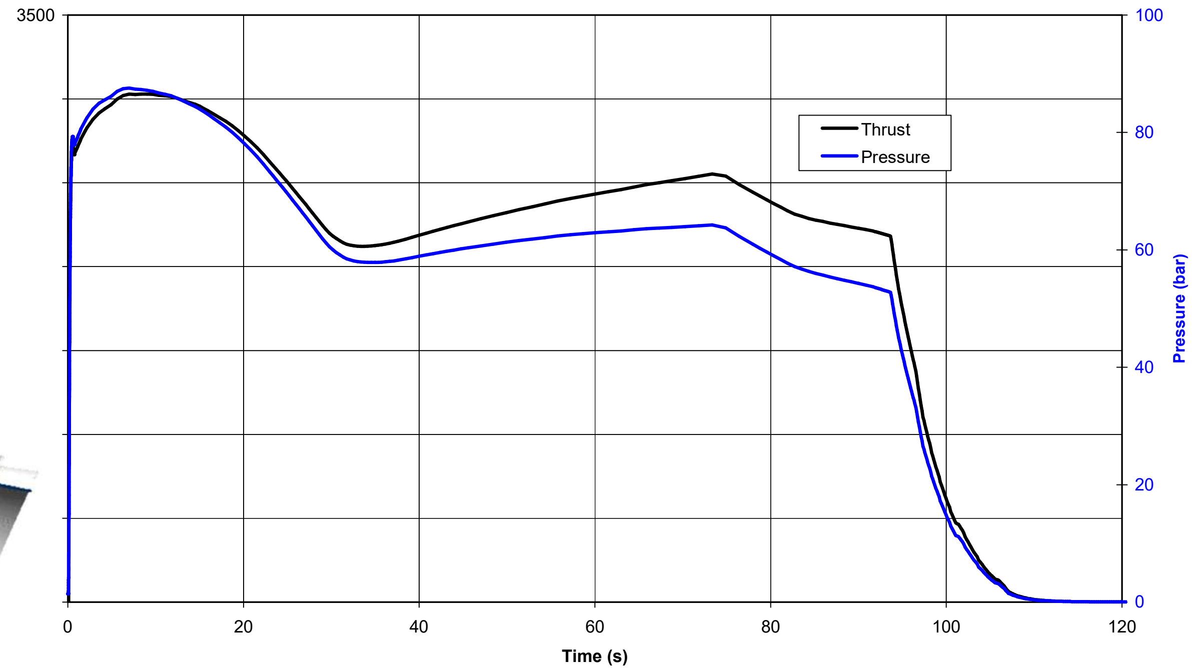
Possible configurations of pre-defined thrust variations during the propulsive mission

Solid Rocket Engines

VEGA 1. Stage

P 80 Motor





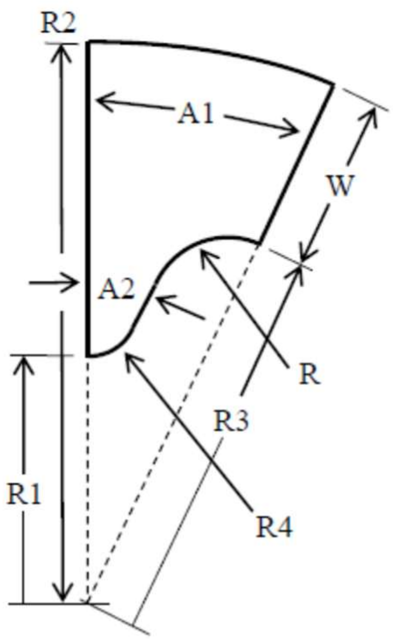
Transient Thrust and Pressure Profiles

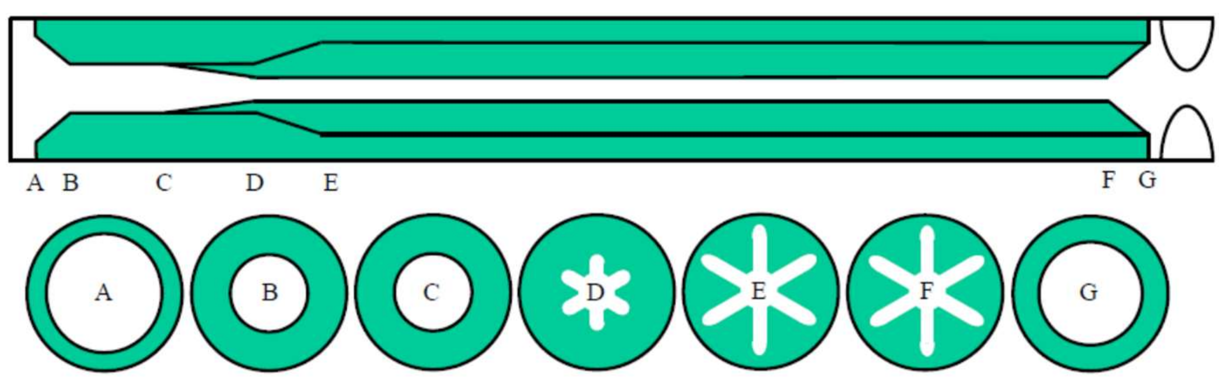
Solid Rocket Engines

Solid Propellant Rocket Engines

Missile Propulsion

Propellant Grain Cross Section Variations





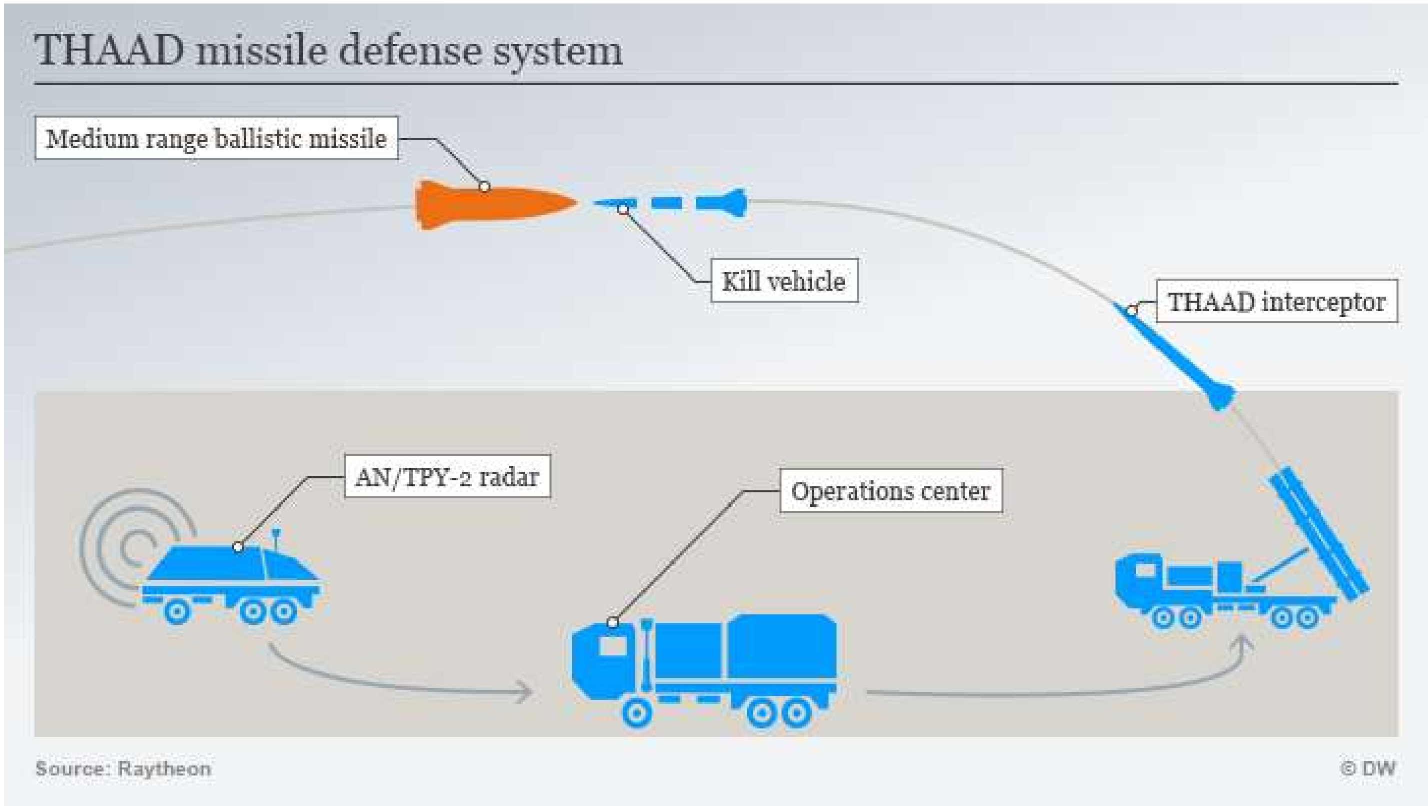
Possible configurations of programmed thrust control during the propulsive mission.

Solid Rocket Engines

Solid Propellant Rocket Engines

Interseptor Missiles

- Multi stage rocket with several flight phase specific designs
- Kill vehicle as final stage



Solid Rocket Engines

Space Launcher Applications

- Boost stages (Ariane 5, HII-A)
- Strap-on Boosters (Atlas V)
- Stage separation, fairing separation (almost all launch vehicles

Booster separation motor of Ariane 5 (NAMMO AS)

Atlas V 541lift-off:

RD-180 and four solid motors AJ-60A

Solid Rocket Motors

SLS Booster

Propellant mass:	680 to
Motor mass:	46 to
Thrust :	1500 to
Propellant:	AP, PBAN, Al
Burn time:	126 s
Isp:	269 s
Mass flow rate:	5.39 to/s

Solid Boosters

Strap-On Boosters

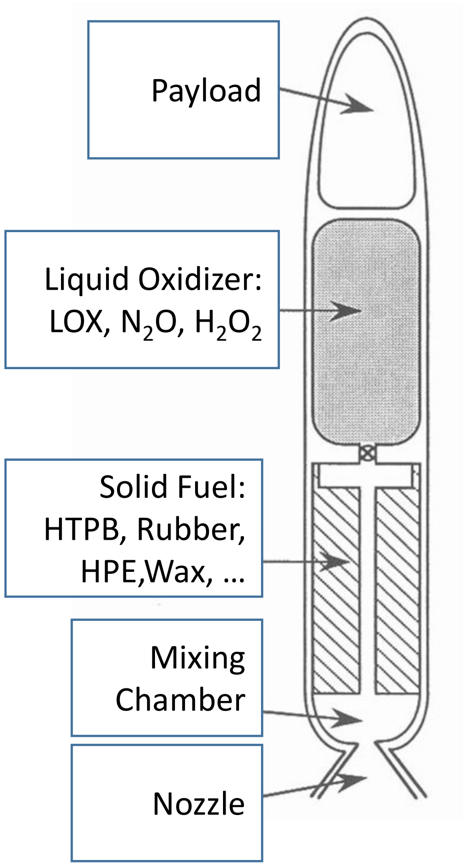
AJ-62 solid rocket motor (Atlas V (up to five motors)

- Largest single segment motor (L=17.7m, D=1.55m)
- Propellant mass: 42.63 to
- Total mass: 46.26 to
- Propellants: HTPB / AP / Al
- Thrust: 172.2 to (SL)
- Isp: 245 s (sl)
- Burn time: 90 sec.

Hybrid Propulsion

Advantages

- safety (with reservation)
- simplified throttling and shut down:
  - modulate the liquid flow rate only
  - thrust termination is simply accomplished by turning off the liquid flow rate
- grain robustness: forgiving-- cracks are inconsequential
- versatility: wide selection of propellant
- temperature insensitivity: launch variations have little effect on *Tc*
- low cost: small acreage



Disadvantages

- low regression rate: multiple ports are required
- low bulk density:
  - due to the demand on large surface area, multiple ports are required leading to the low volumetric fuel loading or bulk density
  - corners in ports lead to formation of slivers
  - reduced nozzle damping leads to poor comb stability
  - combustion efficiency: low because of diffusion flames and poor mixing
- shift of O/F mixture ratio results in thrust variation
- slow transients: response to throttling happens slowly

Hybrid Propulsion

Hybrid Propulsion Emissions

Main Performance Issues

- Limited regression rate and thus thrust and sufficient short motors
- Residual fuel mass in motor (3 % – 10 %)
- Limited thrust due to surface-based fuel mass flow rate (motor length, diameter)
- Sophisticated, inhomogeneous oxidizer supply in particular for long motors

You hardly will find experimentally determined efficiencies for hybrid rockets and the reason is:

It is rather poor (~85 %)

Poor performance results in increased fuel consumption for the same mission and thus in increased emissions of unburnt HC, CO, Soot, …

Hybrid propulsion will be limited at application niches with limited thrust and performance requirements

Liquid Rocket Engines

Engine Cycles

Consequences of Engine Cycle on Combustion

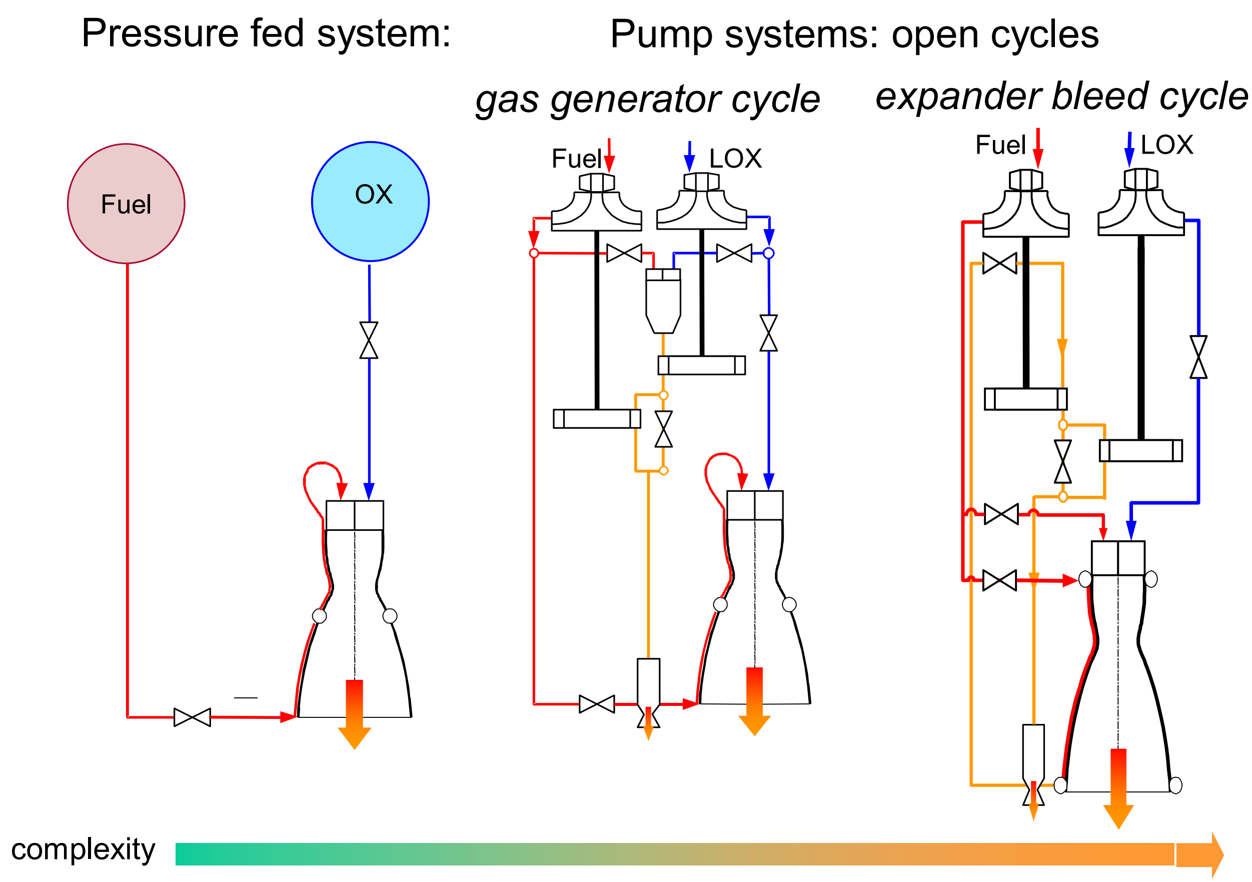
Both, choice of fuel and engine cycle define the state with which the propellants enter the combustion chamber.

1) Cryogenic fuels (LH_2, LCH_4):

- GG Cycle:
  - T_{LH2} ~ 100 K
  - T_{LCH4} ~ 200 K
- Expander Cycle:
  - T_{LH2} ~ 220 K
  - T_{LCH4} ~ 300 K

2) Ambient temperature fuels (kerosene)

- GG Cycle
  - T_{kero} ~ 600 K



Liquid Rocket Engines

Engine Cycles

Consequences of Engine Cycle on Combustion

1) Cryogenic fuels (LH_2):

- Staged Combustion Cycle:

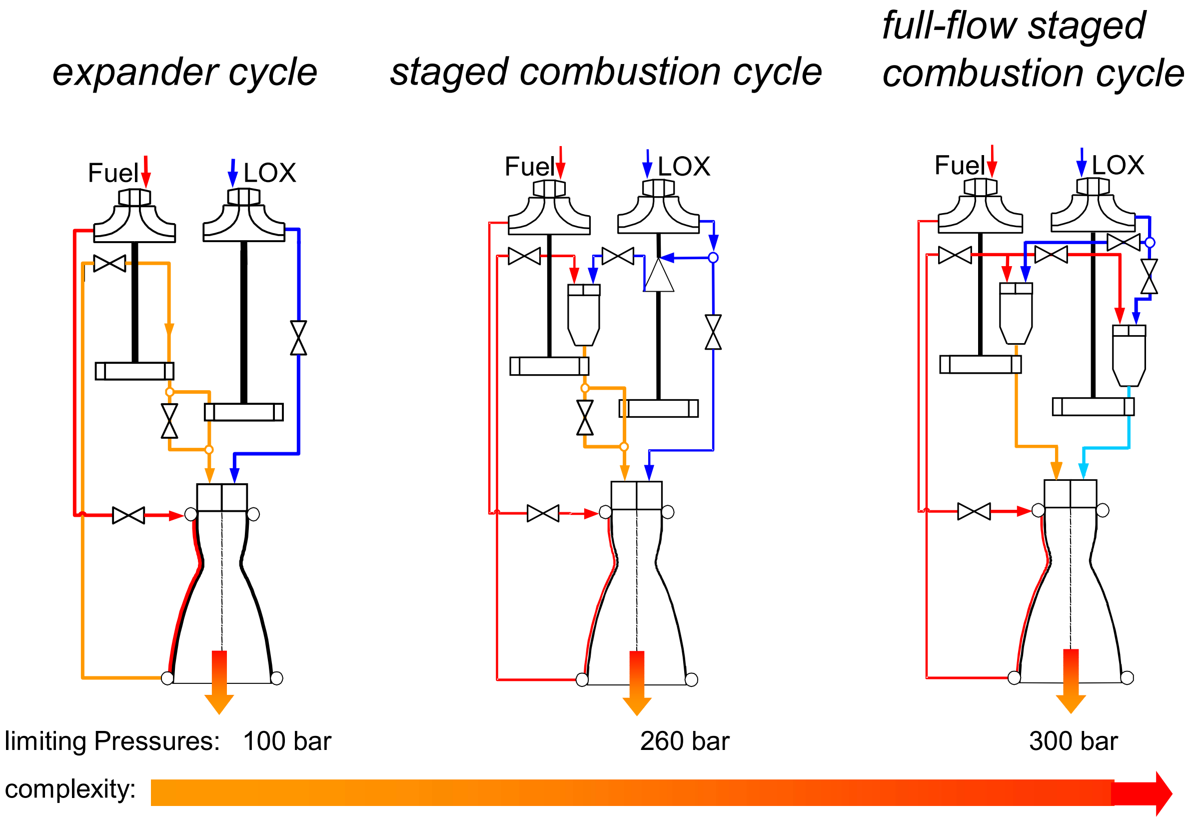
  Some H_2 is burned fuel-rich (r_{of} ~ 0.9) in the pre-burner to drive the turbine and the exhaust (H_2 content ~ 47 %, T ~ 900 K) is further mixed with H_2 from the cooling channel (130 K) and then injected.

2) Ambient temperature fuels (kerosene)

- SC Cycle

  Some kerosene is burned ox-rich (r_{of} ~ 40) in the pre-burner to drive the turbine and the exhaust mixture of CO_2, H_2O and O_2 is then injected.

Pump systems: closed cycles

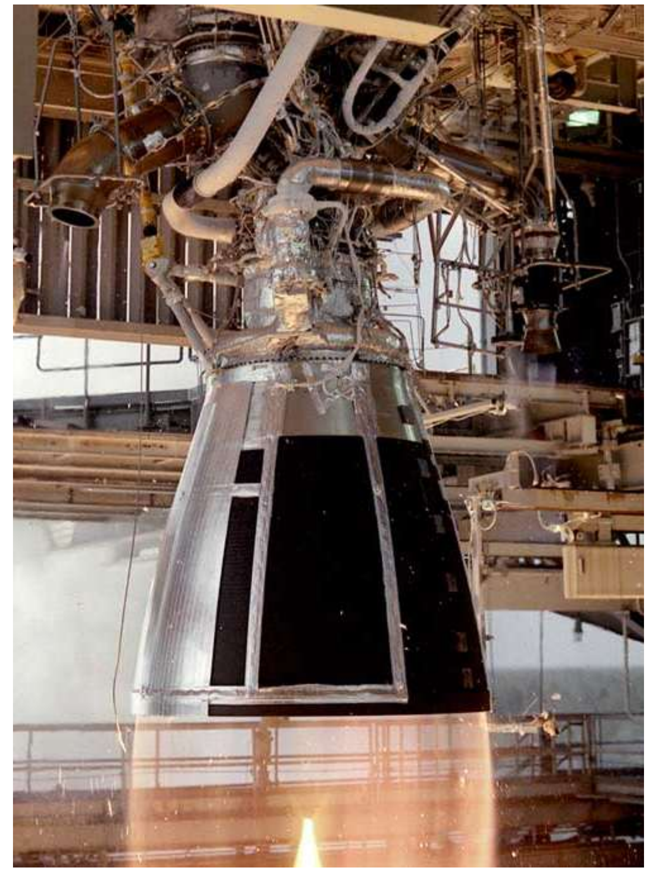


Liquid Rocket Engines

Sample Booster Engines

Thrust(sl):	755 to
Isp:	309 s
Pressure :	245 bar
Propellant:	LOX/kero
Cycle:	ox-rich SC
Appl.:	Zenit-3SL

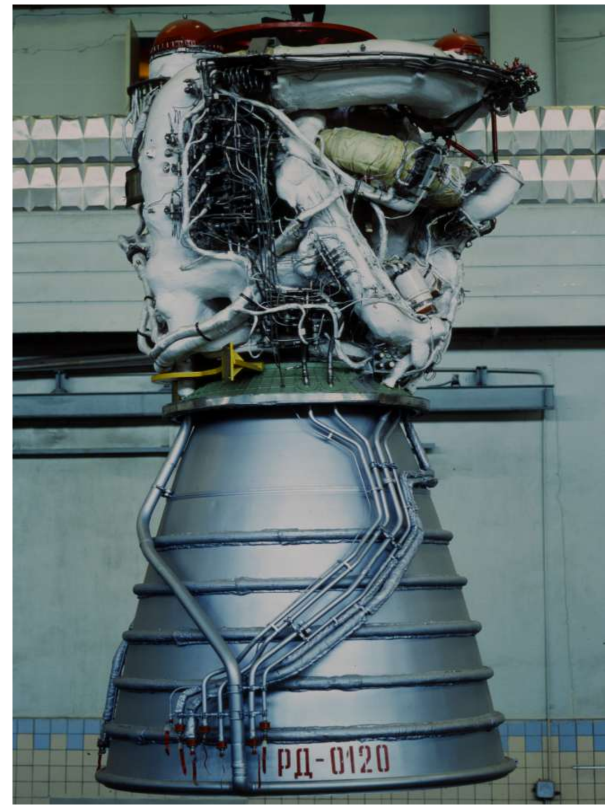
Thrust(sl):	313 to
Isp:	360 s
Pressure :	102 bar
Propellant:	LOX/LH2
Cycle:	GG
Appl.:	Delta 4



Sample Core Stage Engines

Thrust (vac):	70 to
Isp (vac):	430 s
Pressure:	102 bar
Propellant:	LOX/LH_2
Burn time:	520 s
Expansion:	49
Cycle:	GG
Appl.:	LM-5

Thrust(vac):	196 to
Isp (vac):	455 s
Pressure :	218 bar
Propellant:	LOX/LH2
Burn time:	600 s
Expansion:	85.7
Cycle:	SC
Appl.:	Buran



Liquid Rocket Engines

Sample Upper Stage Engines

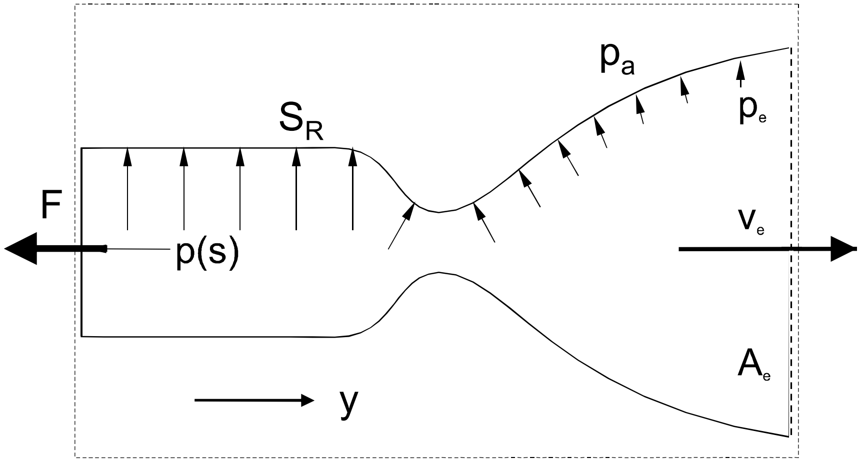
Thrust(vac): 29.4 to
Isp (vac): 359 s
Pressure : 157 bar
Propellant: LOX/kero
Burn time: 300 s
Expansion: ?
Cycle: SC
Appl.: Soyuz 2-1

Thrust(vac): 11.1 to
Isp (vac): 453 s
Pressure : 44.4 bar
Propellant: LOX/LH2
Burn time: 1125 s
Expansion: 280
Cycle: Expander
Appl.: Delta IV

Quite often Russian upper stage engine use multiple chambers but one turbo machine arrangement to reduce engine length and still have a reasonable expansion ratio

Liquid Rocket Engines

Newton's Principle (actio = reactio)



force balance in y-direction:

$$\int\int_S (\rho\vec{u}\vec{n})\, v dS + \int\int_S (p\vec{n}\vec{j})\, dS = 0$$

surface forces due to convection

pressure forces at surface

$$with \int\int_{S_R} (\rho\vec{u}\vec{n})\, v dS = 0 \rightarrow$$

$$\int\int_{A_e} (\rho\vec{u}\vec{n})\, v dS + \int\int_{A_e} (p\vec{n}\vec{j})\, dS + \int\int_{S_R} (p\vec{n}\vec{j})\, dS = 0$$

$\vec{u}$ velocity vector

$\vec{n}$ vector normal to dS

$\vec{j}$ unity vector in y-direction

S surface ($S_R + A_e$)

p pressure

ρ density

v y-component of $\vec{u}$

S_R surface

F thrust

A_e exit area

Propellants

Propellant / Engine Performance Data

Engine	Propellants	O/F	p_c [bar]	Vac. thrust [kN]	I_{sp} vac. [s]	Mass [kg]
Vulcain 2	LOX/LH_2	6,1	115	1350	434	2100
SSME(Bl. 2)	LOX/LH_2	6,0	206	2278	452	3526
RS68	LOX/LH_2	6,0	96	3312	420	6800
VINCI	LOX/LH_2	5,8	60	180	465	550
RL 10A	LOX/LH_2	5,5	39	99	451	172
RD 170	LOX/Kerosene	2,63	245	8060	337	9750
RD 253	NTO/UDMH	1,90	147	1670	316	1280
Aestus	NTO/MMH	2,05	11	30	324	111

Efficiencies

Energy fluxes within a chemical rocket propulsion system