LIQUID ROCKET PROPULSION

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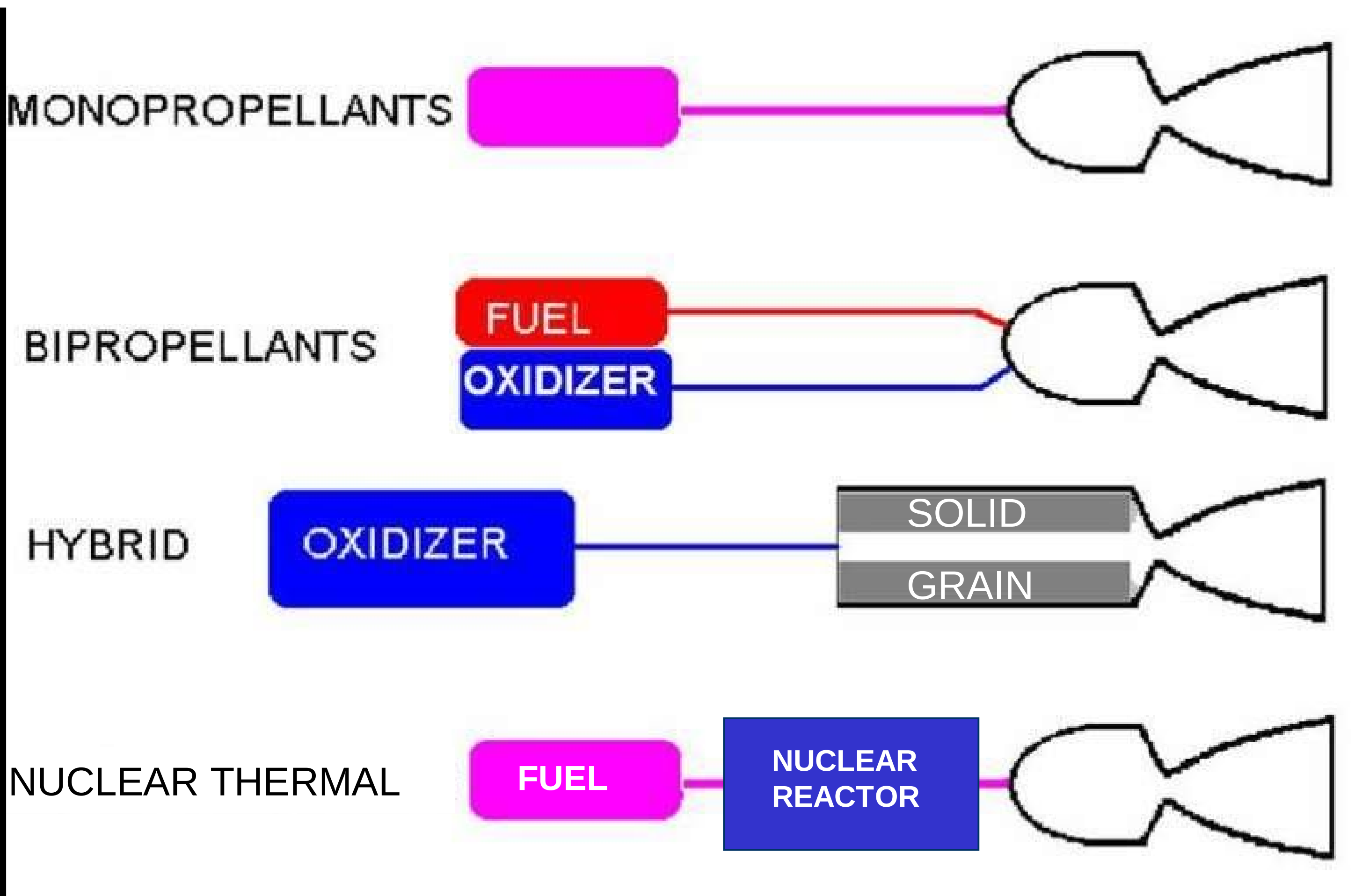
Part 1

LIQUID ROCKET ENGINES

LIQUID ROCKET ENGINE TYPES



LIQUID PROPELLANTS

MONOPROPELLANTS :

Hydrazine,

Hydrogen peroxyde

BIPROPELLANTS :

FUELS :	OXIDIZERS :
Kerosine, ethanol	Liquid oxygen,
Liquid hydrogen,	N2O4,
UDMH, MMH, Hydrazine	Hydrogen peroxyde

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HYBRID PROPELLANTS

liquid propellant : oxidizer - solid propellant : fuel

(solid oxidizers are problematic and lower performing than liquid oxidizers)

oxidizers **:**	***fuels*** **:**
gaseous or liquid oxygen	**polymers (e.g.polyethylene),**
nitrous oxide.	**cross-linked rubber (e.g.HTPB),**
	liquefying fuels (e.g. paraffin).

Solid fuels (HTPB or paraffin) allow for the incorporation of high-energy fuel additives (e.g.aluminium).

CHEMICAL REACTIONS

BIPROPELLANTS

Dimethylhydrazine (UDMH) :

$H_2N – N(CH_3)_2$ (l)+ 2 N_2O_4 (l) - > 3 N_2 (g) + 4 H_2O (g) + 2 CO_2 (g)

Hydrazine hydrate (N_2H_4,H_2O) :

2 (N_2H_4,H_2O) (l) + N_2O_4 (l) - > 3 N_2 (g) + 6 H_2O (g)

Monomethylhydrazine (MMH) :

4 $H_2N – NHCH_3$ (l) + 5 N_2O_4 (l) - > 9 N_2 (g) + 12 H_2O (g) + 4 CO_2 (g)

Kerosene (CH_2 is the approximate formula) with hydrogen peroxide :

$CH_2 + 3H_2O_2 \rightarrow CO_2 + 4H_2O$

Kerosene and liquid oxygen (LOX)

$CH_2 + 1.5O_2 \rightarrow CO_2 + H_2O$

Hydrogen and oxygen (liquids) :

2 H_2 (g)+ O_2 (g) - > 2 H_2O (g)

MONOPROPELLANTS

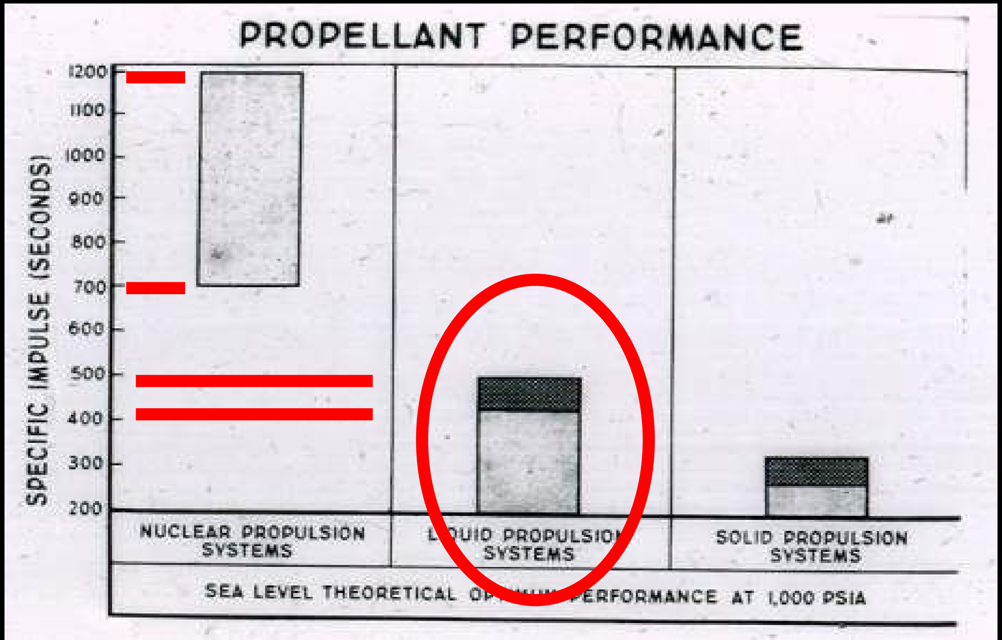
Hydrogen peroxyde (H_2O_2)

H_2O_2 (l) - > H_2O (l) + 1/2 O_2 (g)

Hydrazine (N_2H_4)

N_2H_4 (l) - > N_2 (g) + 2 H_2 (g)

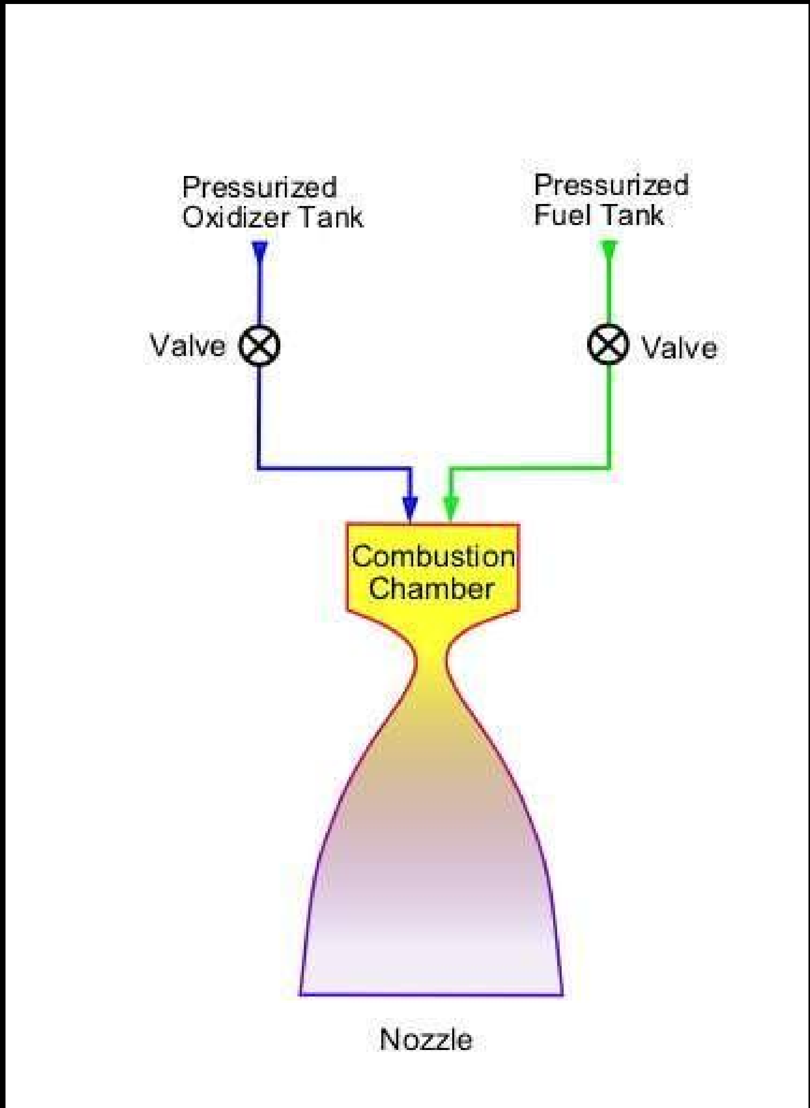
Specific impulse of various propulsion technologies



Engine type	Specific impulse
Jet engine	300 s
Solid rocket	250 s
Bipropellant rocket	450 s
Ion thruster	3000 s
VASIMR	30000 s

PROPULSION BASICS

Pressure-fed cycle



Pros :

-Simplicity

-Low complexity

-***Cons :***

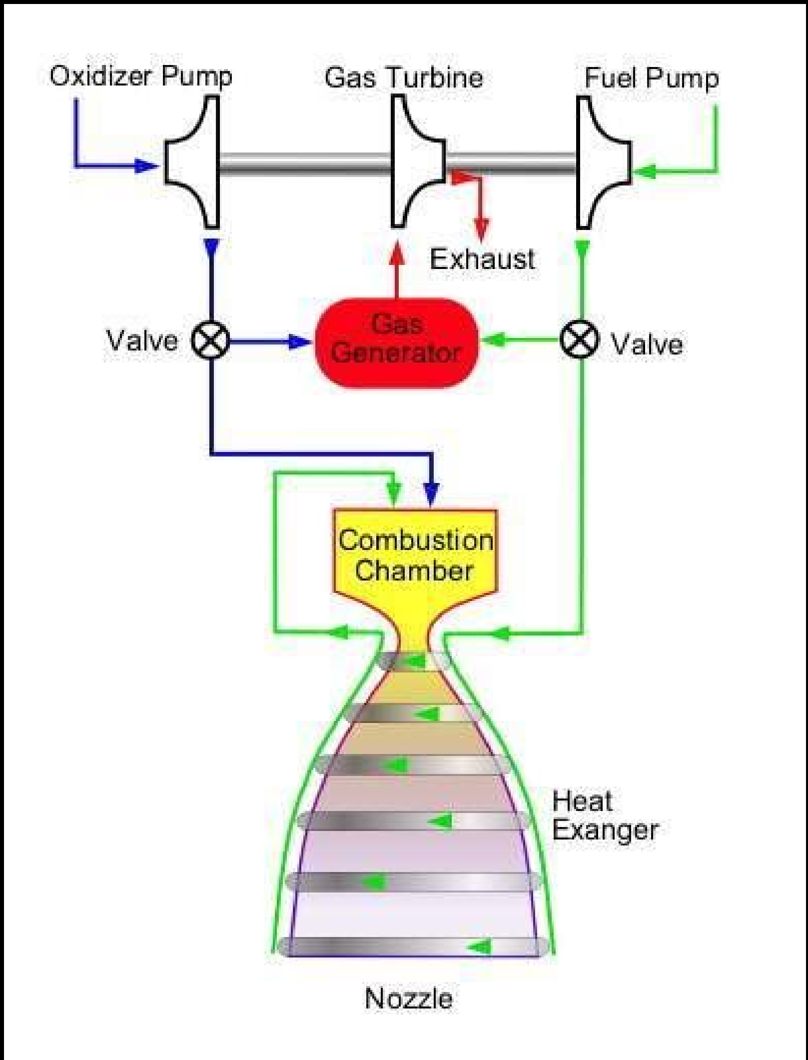
-Low performance

-Oldest and simplest cycle,

-Rarely used nowadays on launch vehicles,

-Powered France's Launch vehicle family Diamant.

Gas-generator cycle



Pros :

-Higher presssures

-Lowerturbine temperatures

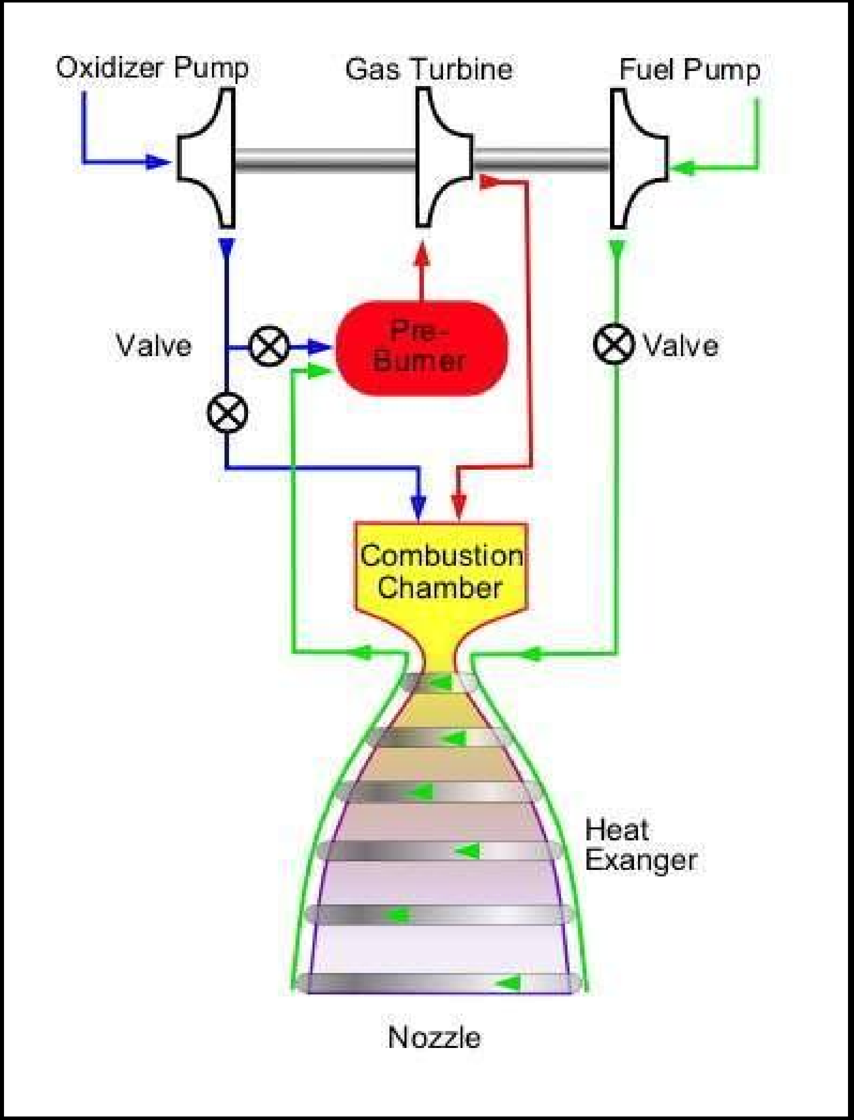
-Highest performance

-***Cons :***

-Moderate performance

-Most widely used cycle in the western world,

Preburner cycle



<u>Pros :</u>

-Higher presssures

-Lowerturbine temperatures

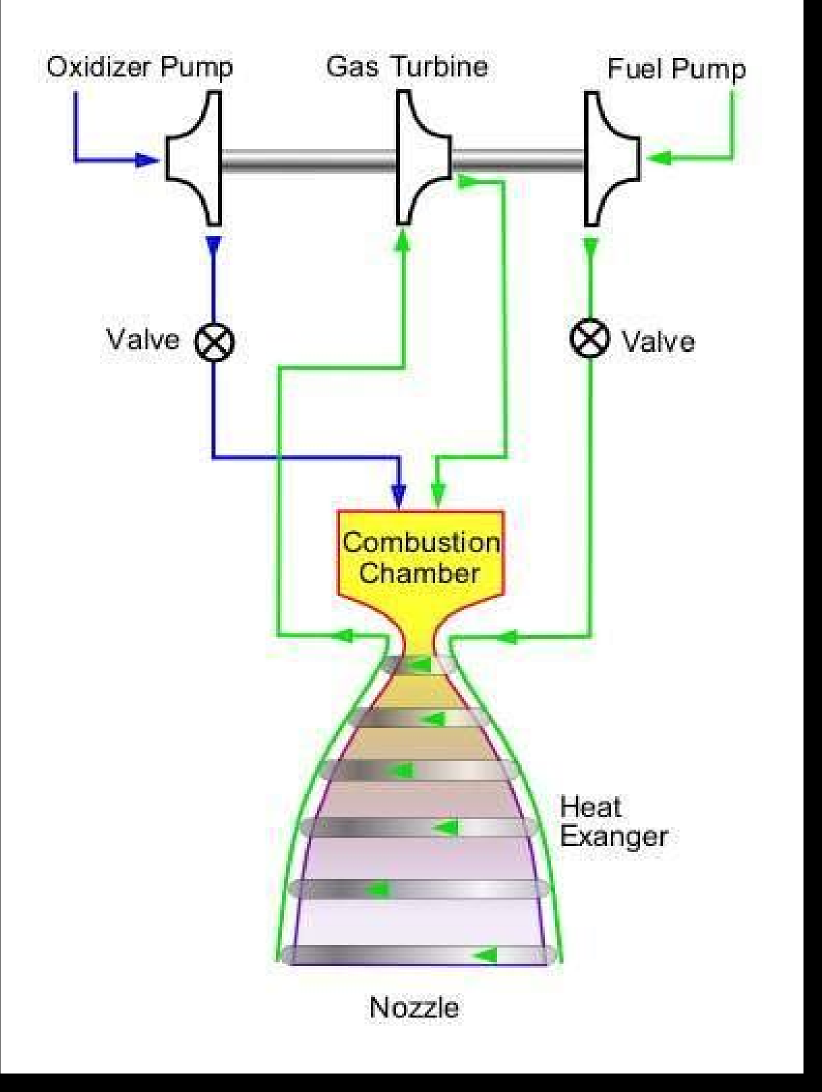
-Highest performance

*-<u>**Cons :**</u>*

-High complexity

-Used on the Shuttle and the H-2A main engine

Expander cycle



<u>Pros :</u>

-Thermally challenging

-Highest performance

-***<u>Cons :</u>***

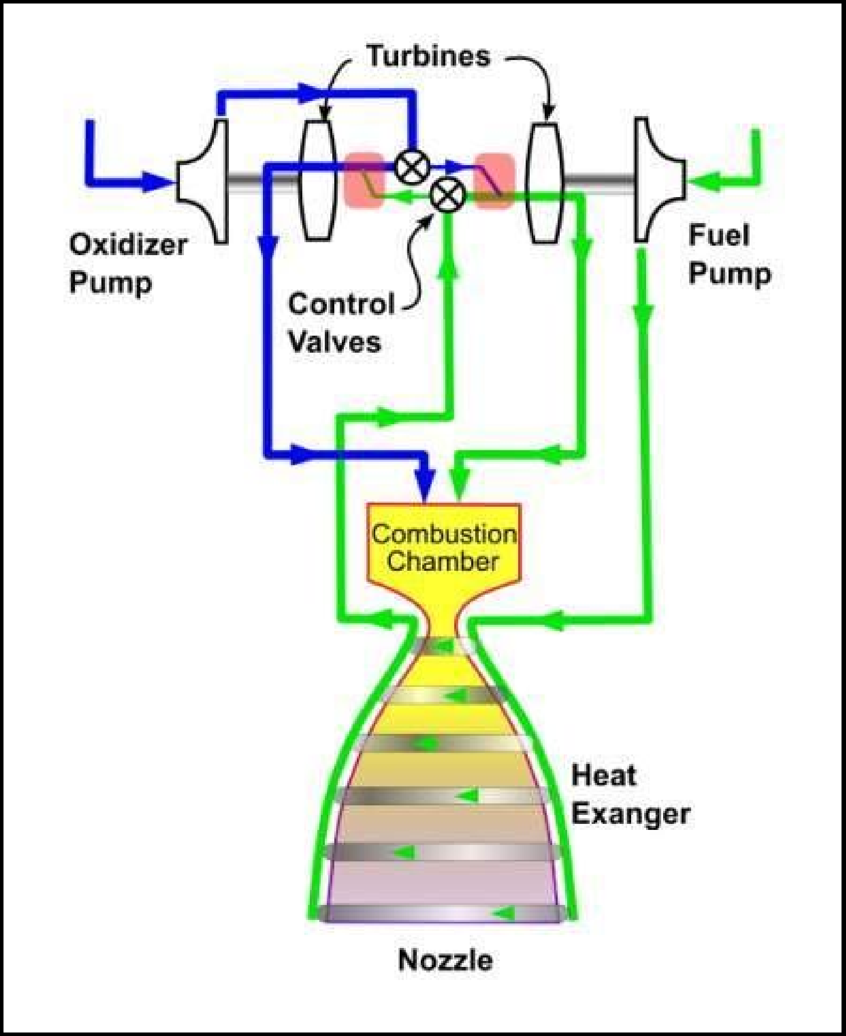
-For cryogenic engines only

-Limited power therefore not suited for high thrusts

-Oldest cryogenic engine in service (RL-10)

-Used in Japan and Europe

Full flow staged combustion rocket cycle



Pros :

-Higher presssures

-Lowerturbine temperatures

-Highest performance

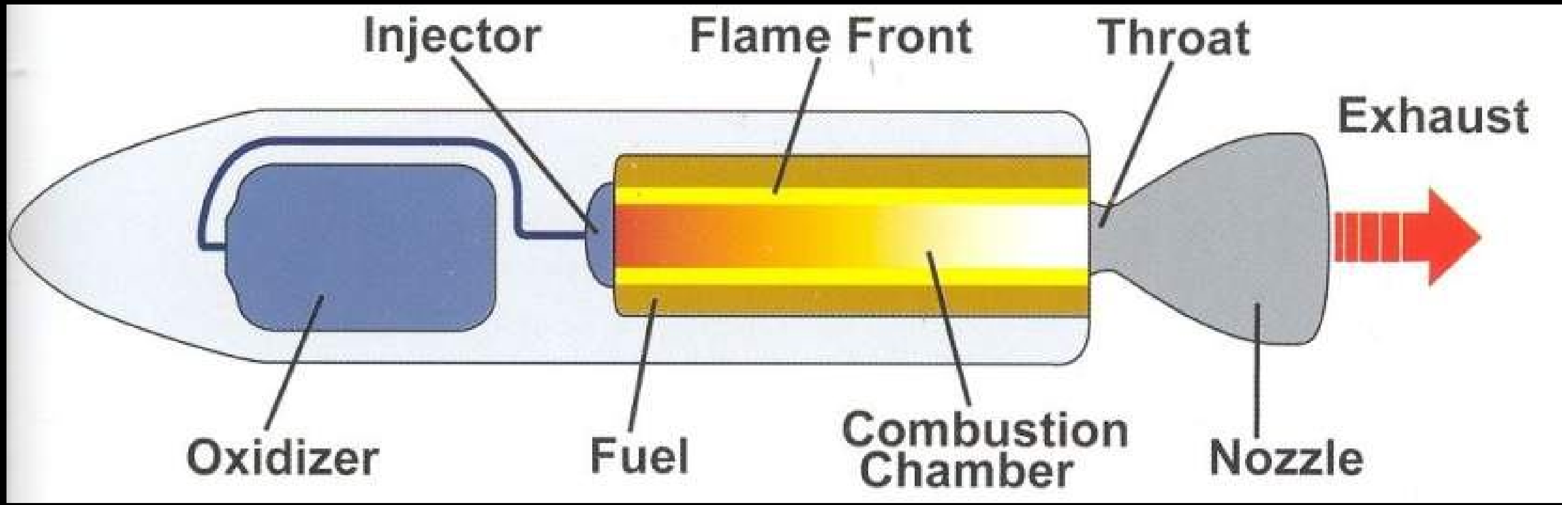
-***Cons :***

-Very high complexity

-Never flown,

-Demonstration only (IPD) so far

Hybrid rocket cycle



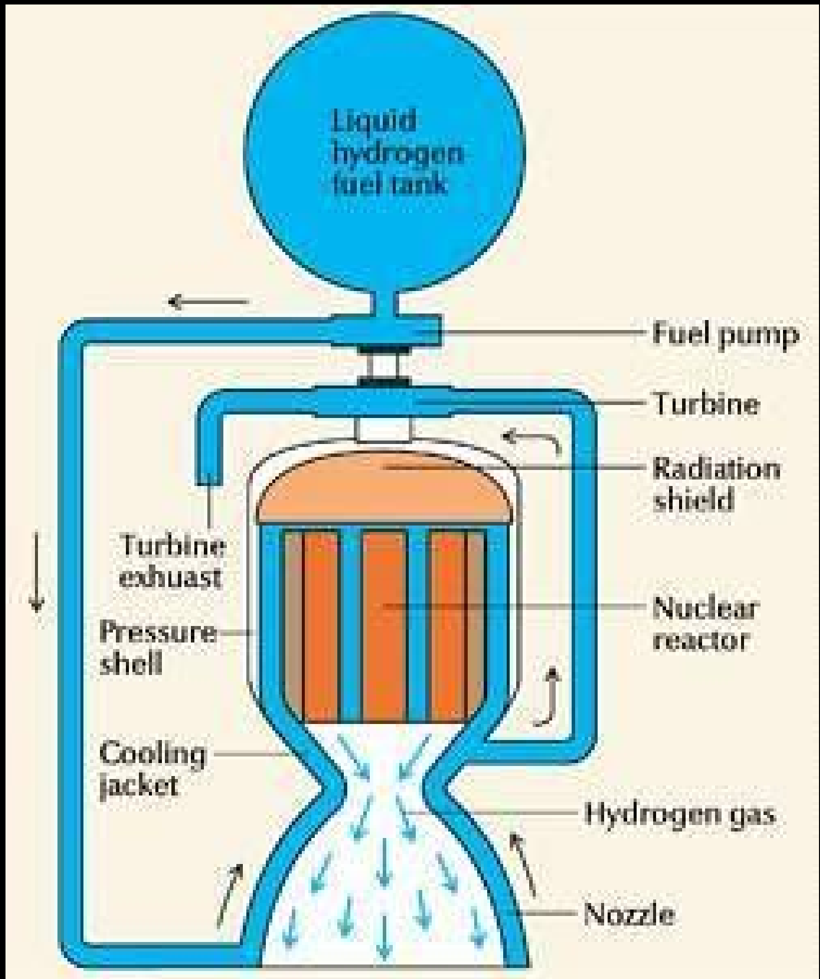
Pros :

- **Higher performance than solids**
- **Lower complexity than liquids**

Cons :

- **Lower performance than liquids**
- **Higher complexity than solids**

Nuclear thermal rocket cycle



Pros :

-Highest performance

-***Cons :***

-Very high complexity

-Radiations

-Never flown,

-Demonstration only so far

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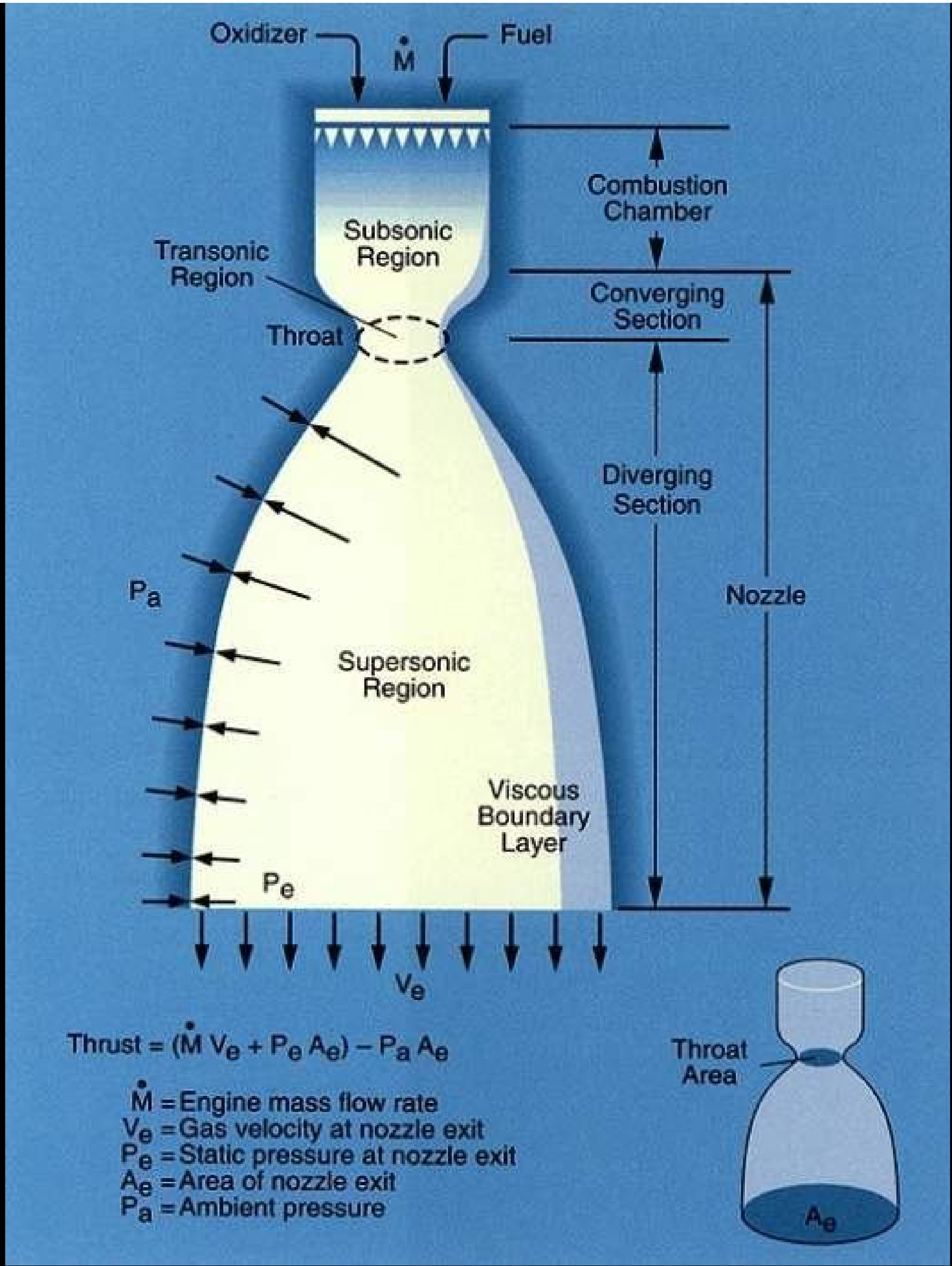
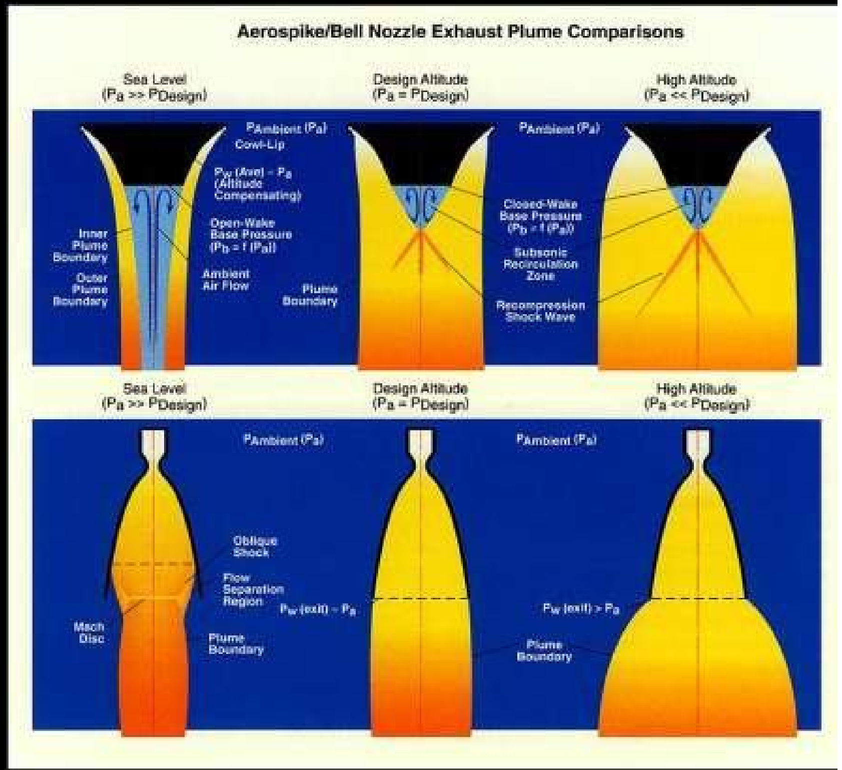
NOZZLES

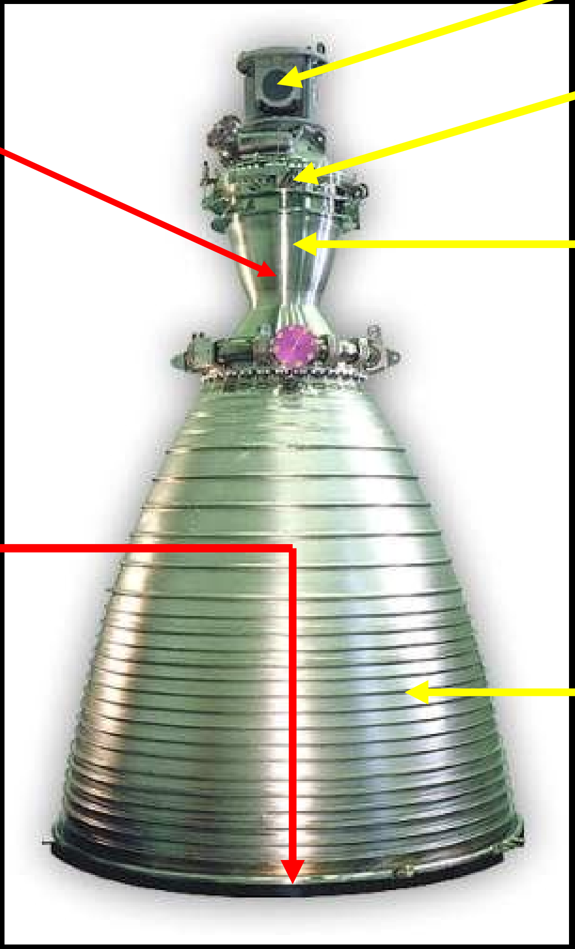

Oxidizer
Ṁ
Fuel
Combustion Chamber
Subsonic Region
Transonic Region
Throat
Converging Section
Diverging Section
Nozzle
Pa
Supersonic Region
Viscous Boundary Layer
Pe
Ve
Thrust = (Ṁ Ve + Pe Ae) – Pa Ae
Ṁ = Engine mass flow rate
Ve = Gas velocity at nozzle exit
Pe = Static pressure at nozzle exit
Ae = Area of nozzle exit
Pa = Ambient pressure
Throat Area
Ae


Aerospike/Bell Nozzle Exhaust Plume Comparisons
Sea Level (Pa >> PDesign)
Design Altitude (Pa = PDesign)
High Altitude (Pa << PDesign)
PAmbient (Pa)
Cowl-Lip
Inner Plume Boundary
Outer Plume Boundary
Open-Wake Base Pressure
Ambient Air Flow
Plume Boundary
Closed-Wake Base Pressure
Subsonic Recirculation Zone
Recompression Shock Wave
Oblique Shock
Flow Separation Region
Mach Disc

THRUST CHAMBER ASSEMBLY

Chamber Characteristics:
- Combustion
- High pressure
- High temperature
- Very low net fluid velocity

Exit Characteristics:
- Flow expands to fill enlarged volume
- Reduced pressure
- Reduced temperature
- Very high fluid velocity



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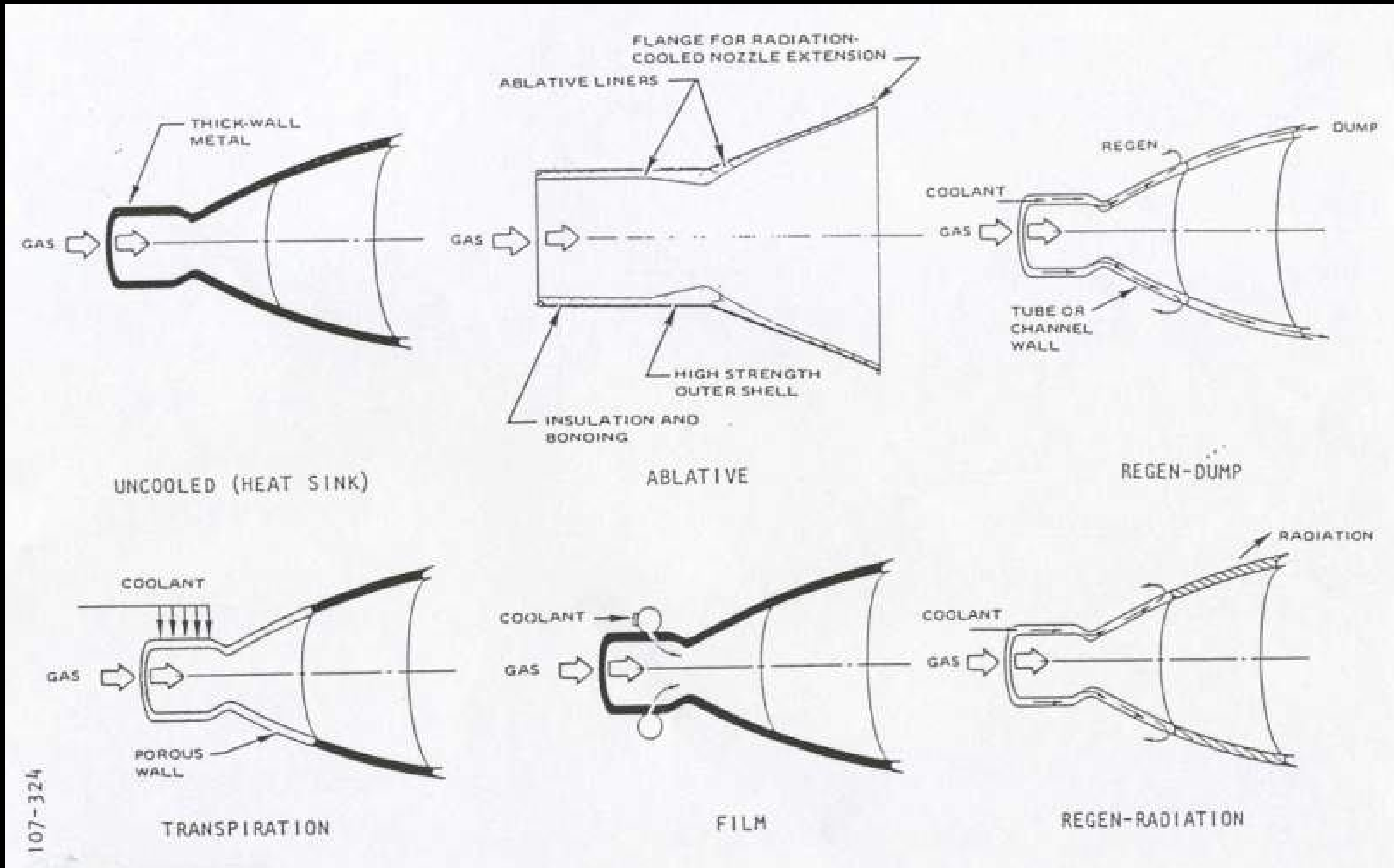
Thrust chamber cooling methods



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Nozzle design challenges

Structural factors

- Physical
  - Weight, Center of Gravity
- Service Life
  - Running time, Number of starts
- Duty Cycle/Operating Range
  - Continuous operation at one power level
  - Throttled operation
- Materials
  - Compatibility
  - Strength,
  - Heat transfer capability

Structural concerns

- Loads
  - Testing
    - Altitude Simulation or Sea Level
    - Sideloads
  - Flight
    - Lift off, shut down, restart
    - Sideloads
  - Transportation
  - Temperature

KEY REQUIREMENTS FOR SAFE NOZZLE OPERATION

Sea level :

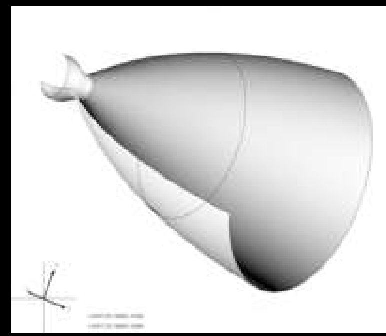
- stable operation on ground

-high performance

Vacuum :

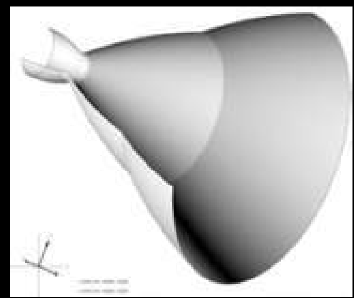
- high vacuum performance

- low package volume

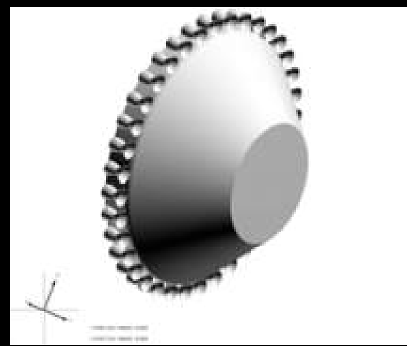


Bell nozzle

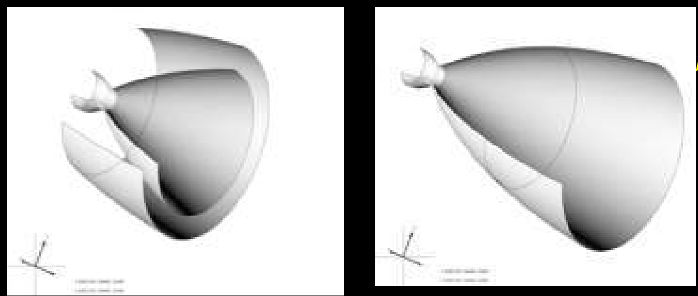
Advanced concepts with altitude adaptation



Dual-bell nozzle



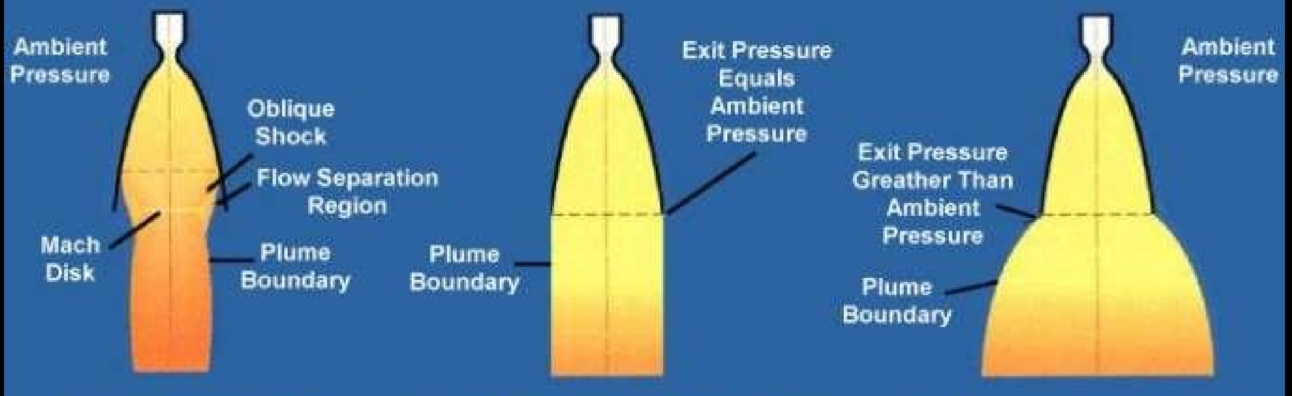
Plug nozzle ("Aerospike")



Extendible nozzle

Nozzle Types

- **Conical Nozzle**
  - **Simple cone shape - easy to fabricate**
  - **Rarely used on modern rockets**
- **Bell Nozzle**
  - **Bell shape reduces divergence loss over a similar length conical nozzle**
  - **Allows shorter nozzles to be used**
- **Annular Nozzles (spike or pug)**
  - **Altitude compensating nozzle**
  - **Aerospike is a spike nozzle that uses a secondary gas bleed to "fill out" the truncated portion of the spike nozzle**



(a) Bell Nozzle at Sea Level: the exhaust plume is "pinched" by high ambient air pressure, reducing its efficiency.

(b) Bell Nozzle at Optimum Altitude: the exhaust plume is column-shaped producing maximum efficiency.

(c) Bell Nozzle at High Altitude: the exhaust plume continues to expand past the nozzle exit reducing efficiency.

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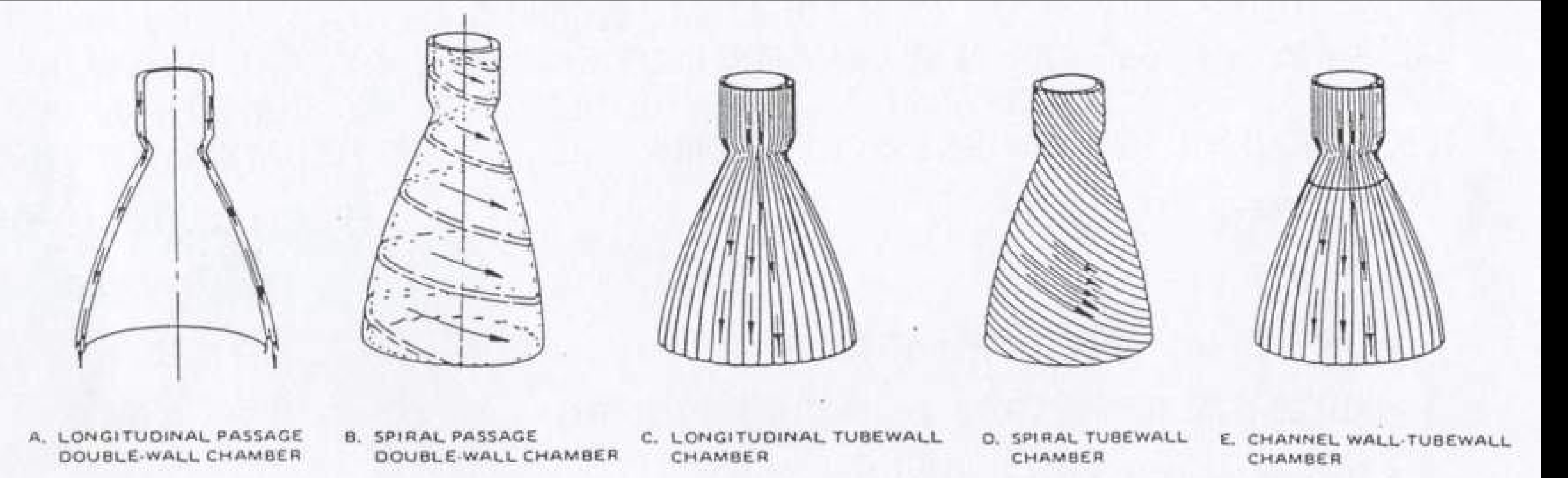
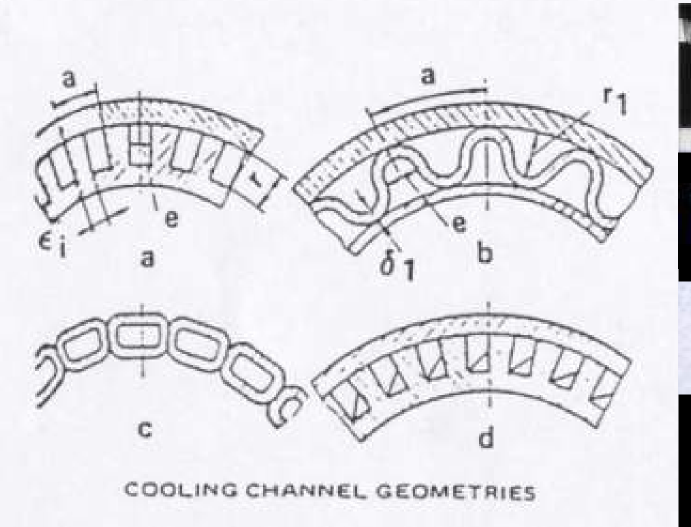
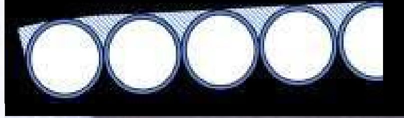
Nozzle extension cooling systems





Laser welded jacket to liner

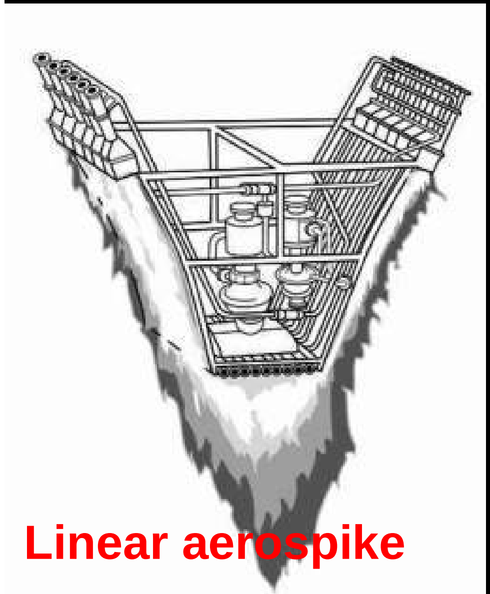


Brazed tubes inside jacket

Brazed jacket to liner

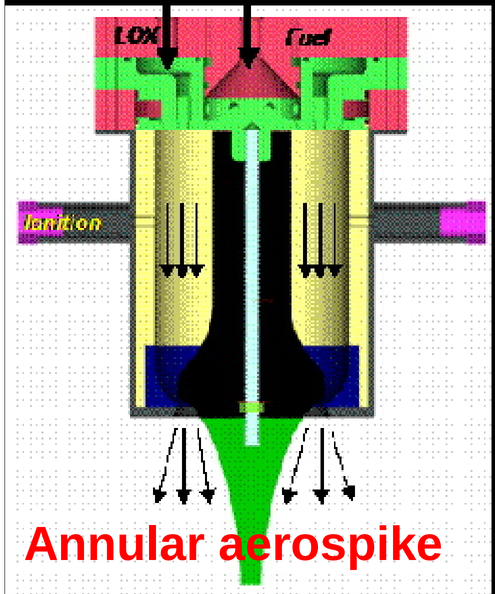
Materials : stainless steel, copper, nickel, copper/nickel

AEROSPIKE NOZZLES



Linear aerospike

Rocketdyne XRS-2200



Annular aerospike

Rocketdyne AMPS-1
1960's

World's first aerospike flight,
September 20, 2003
(California State Univ. Long Beach)

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Part 2

ANATOMY OF A ROCKET ENGINE

1 – Gas generator cycle : F-1 and Vulcain engines

2 – Staged combustion cycle : SSME

3 – Expander cycle : Vinci

4 – Linear aerospike XRS2200

5 – Nuclear : NERVA/RIFT

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APOLLO 40 YEARS

Rocketdyne F-1

ROCKETDYNE
A Division of North American Aviation, Inc.

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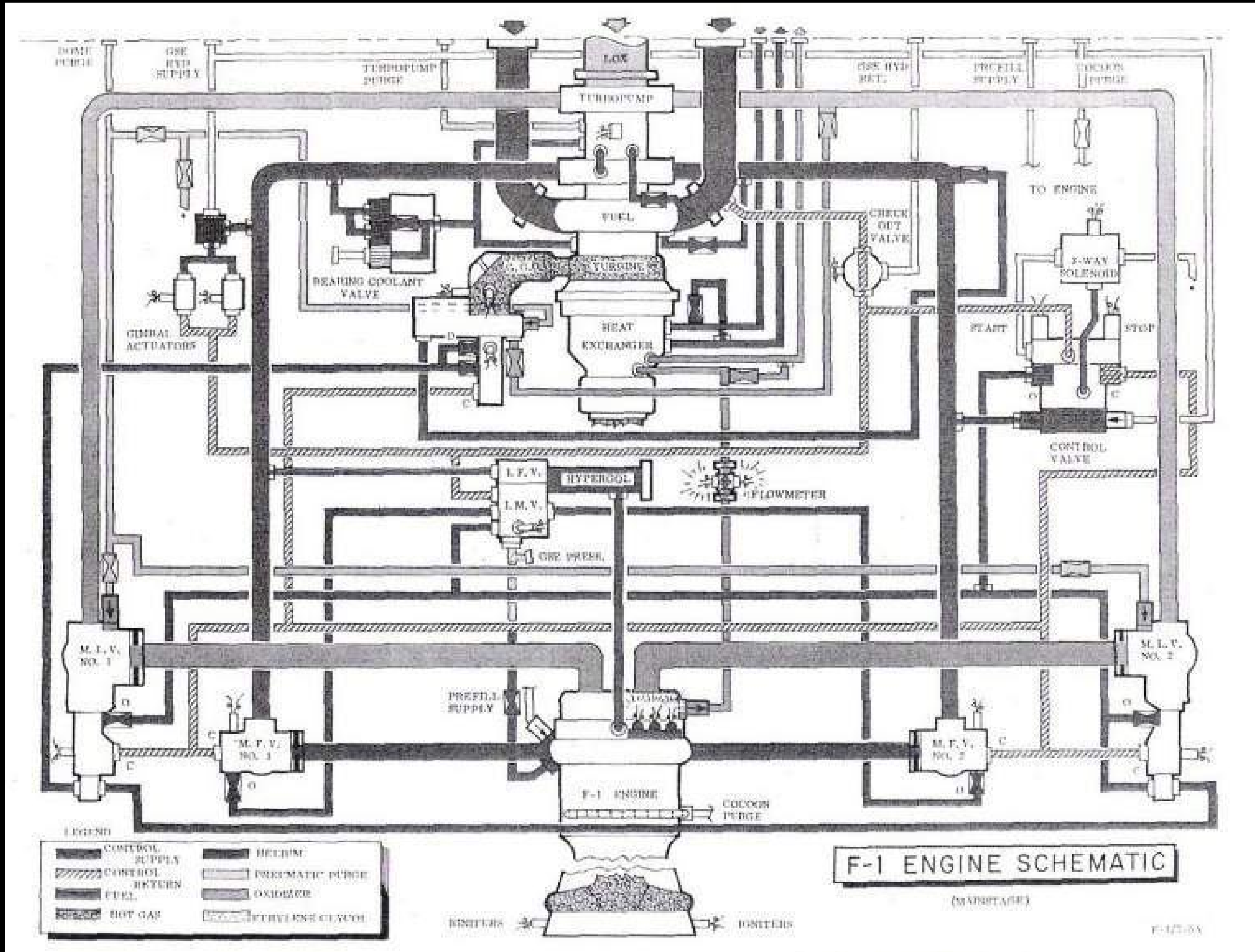
Engine flow diagram

Rocketdyne F-1



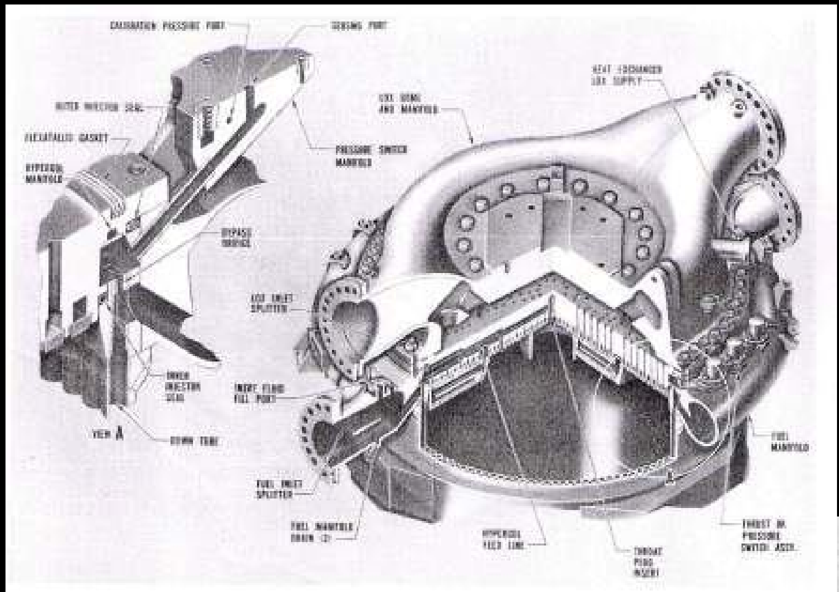
Engine characteristics

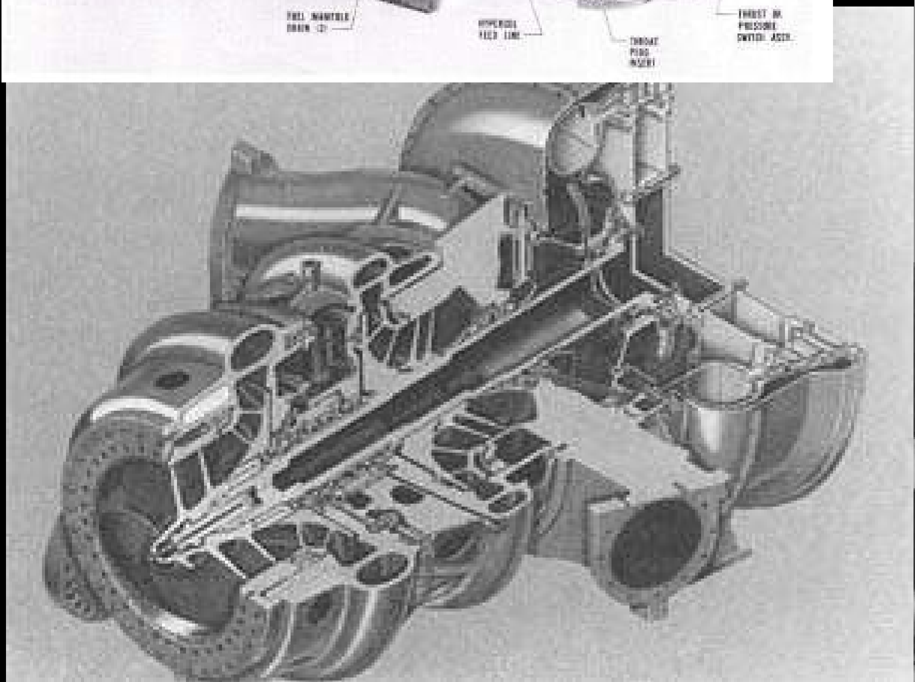
Rocketdyne F-1

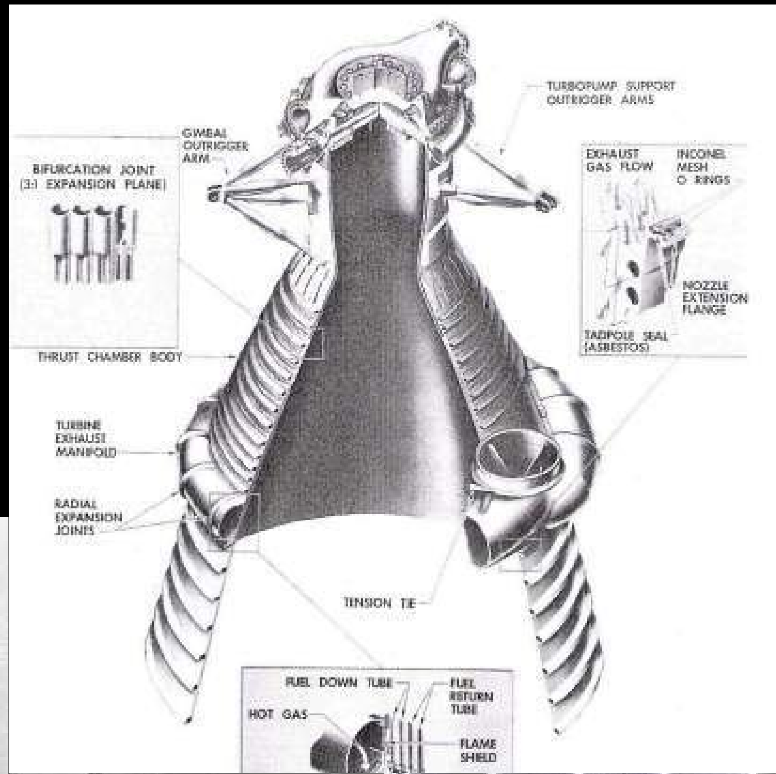
	Apollo 4, 6, and 8	*Apollo 9 on*
Thrust (sea level):	**6.67 MN**	**6.77 MN**
Burn time:	**150 s**	**165s**
Specific impulse:	**260 s**	**263 s**
Engine weight dry:	**8.353 t**	**8.391 t**
Engine weight burnout:	**9.115 t**	**9.153 t**
Height:	**5.79 m**	
Diameter:	**3.76 m**	
Exit to throat ratio:	**16 to 1**	
Propellants:	**LOX & RP-1**	
Mixture ratio:	**2.27:1 oxidizer to fuel**	

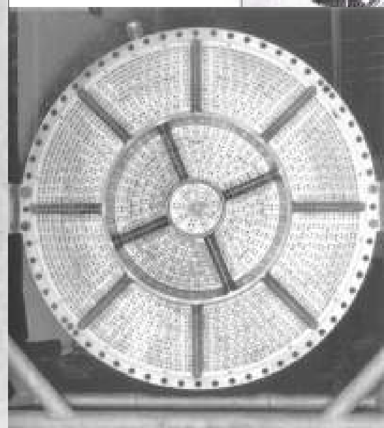
Engine components

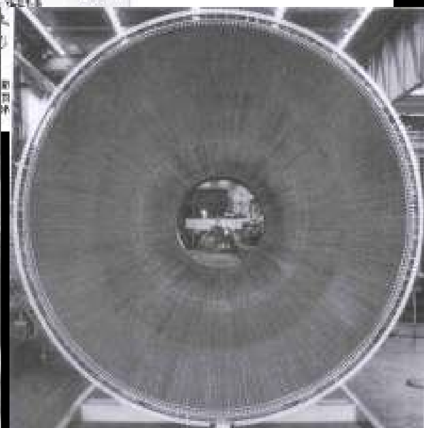
Rocketdyne F-1



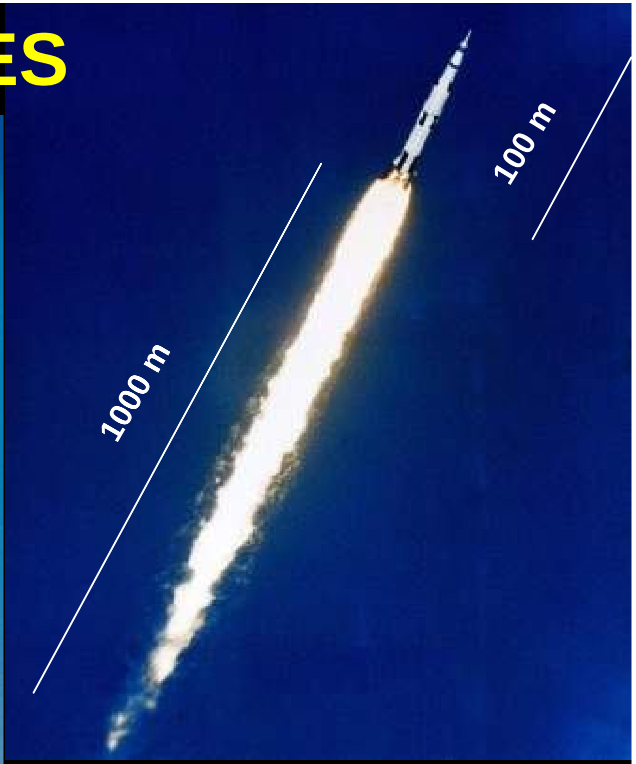






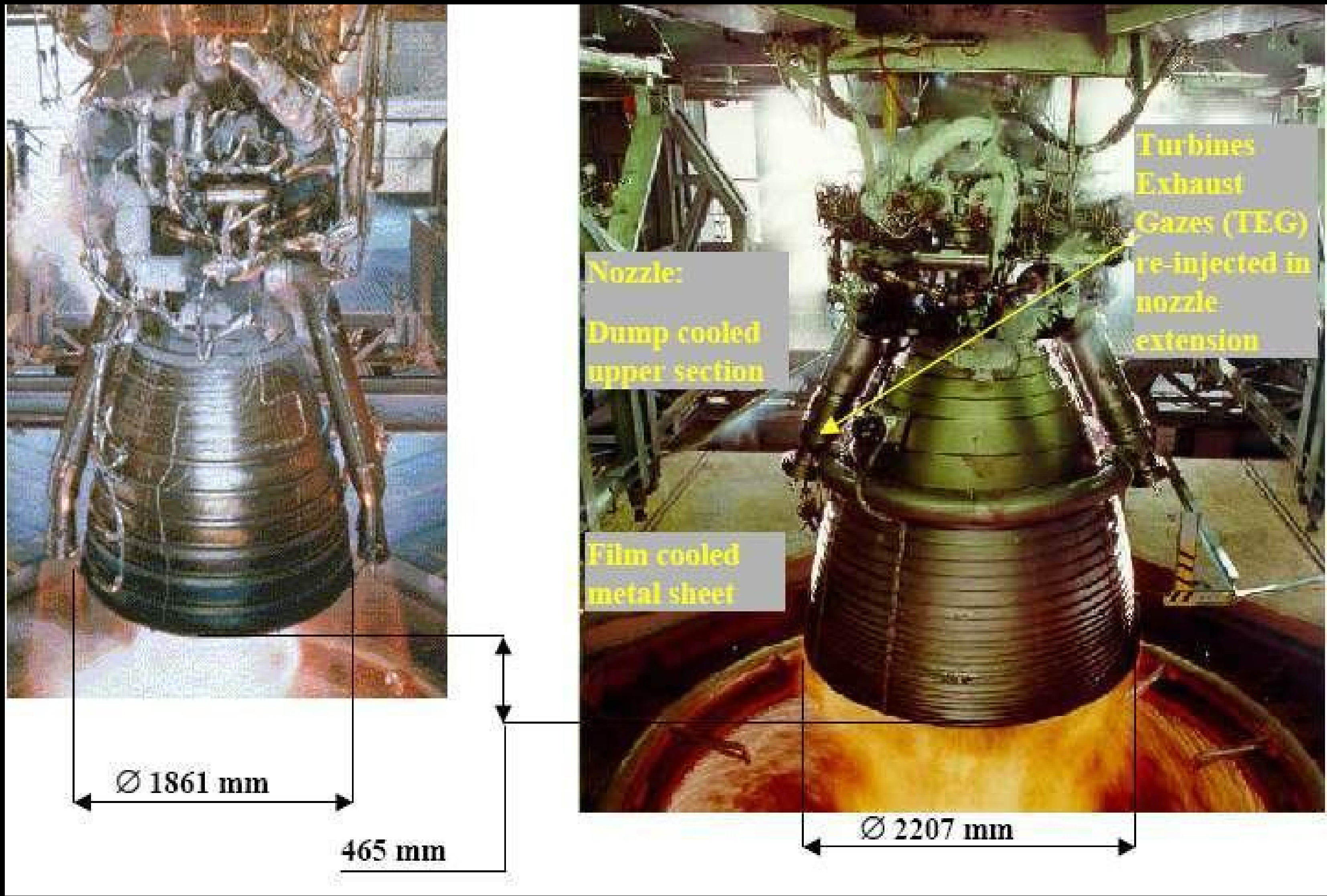


TESTS AND LAUNCHES



VULCAIN 2

VULCAIN 1 TO VULCAIN 2 EVOLUTION



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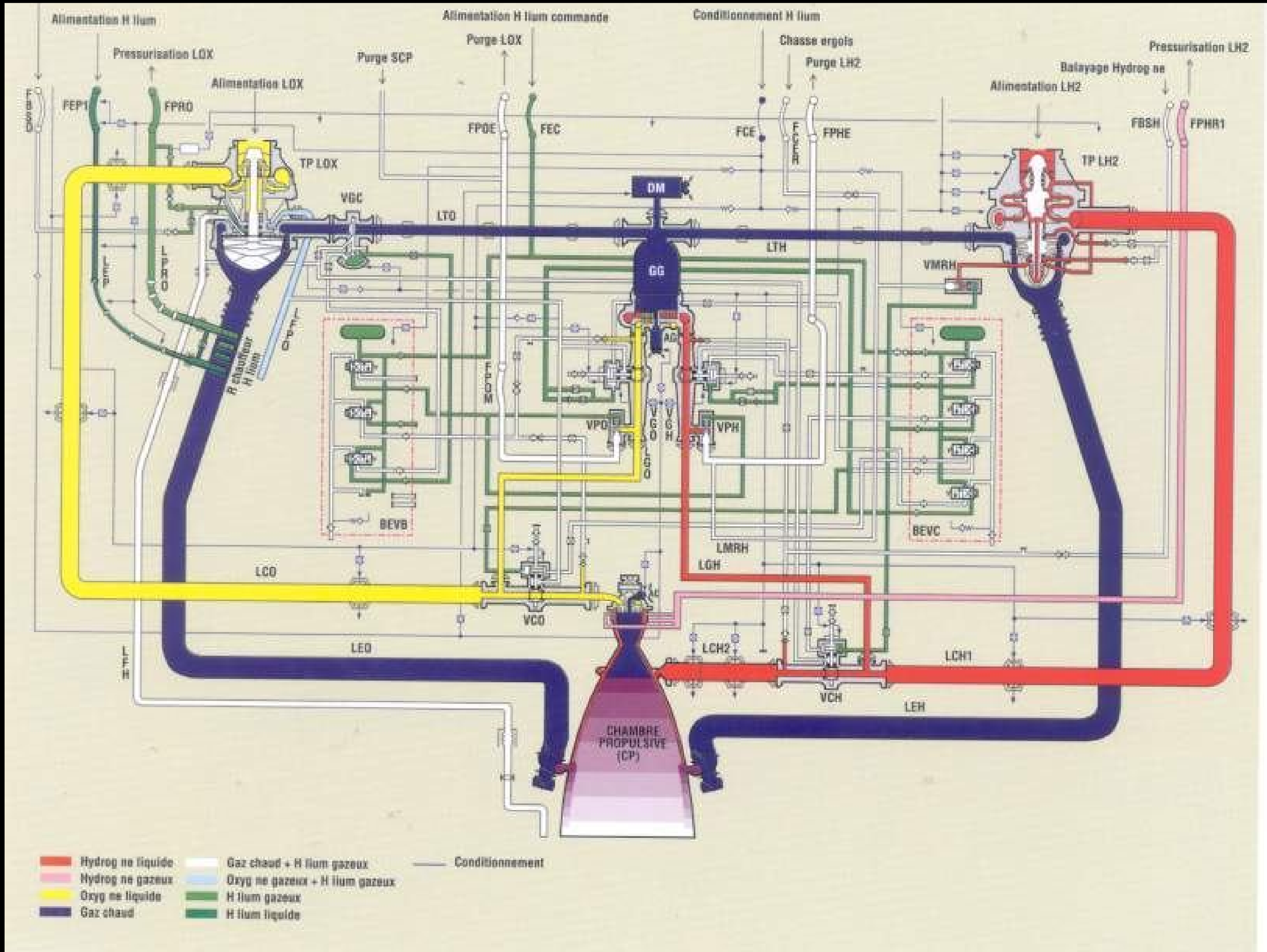
Engine characteristics

	Vulcain 1	Vulcain 2
Vacuum thrust	1140 kN	1350 kN
Mixture ratio	4,9 to 5,3	6,13
Vacuum specific impulse	430 s	434 s
Dry weight	1680 kg	2040 kg
Chamber pressure	110 bar	116 bar
Expansion ratio	45	60
Design life	6000 s 20 starts	5400 s 20 starts

Vulcain 2 Flow diagram



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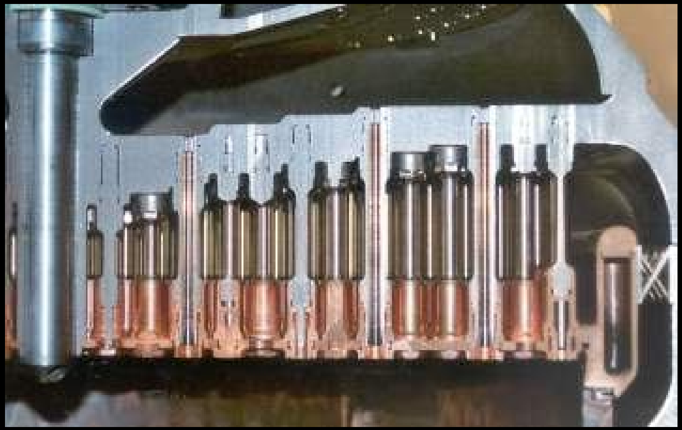
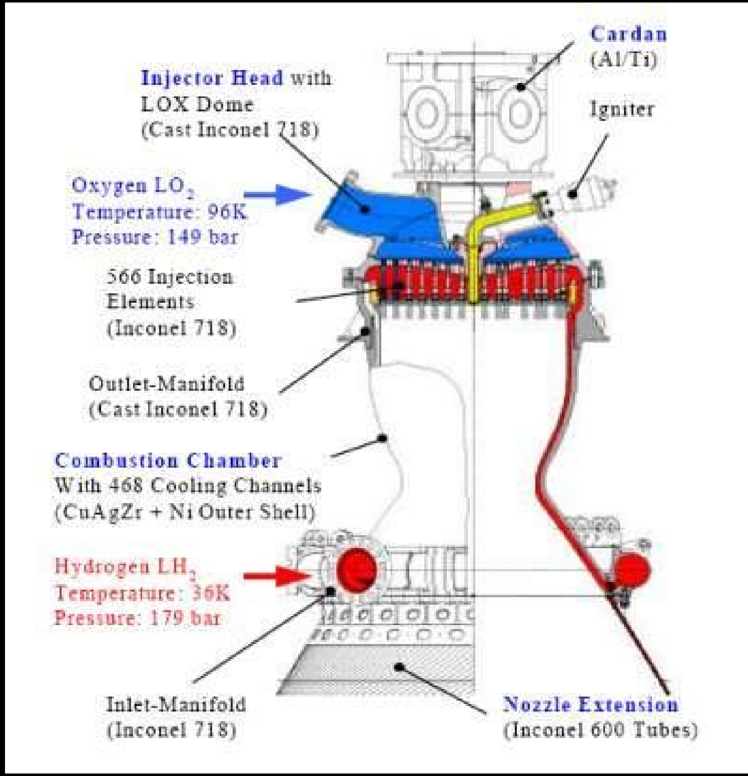
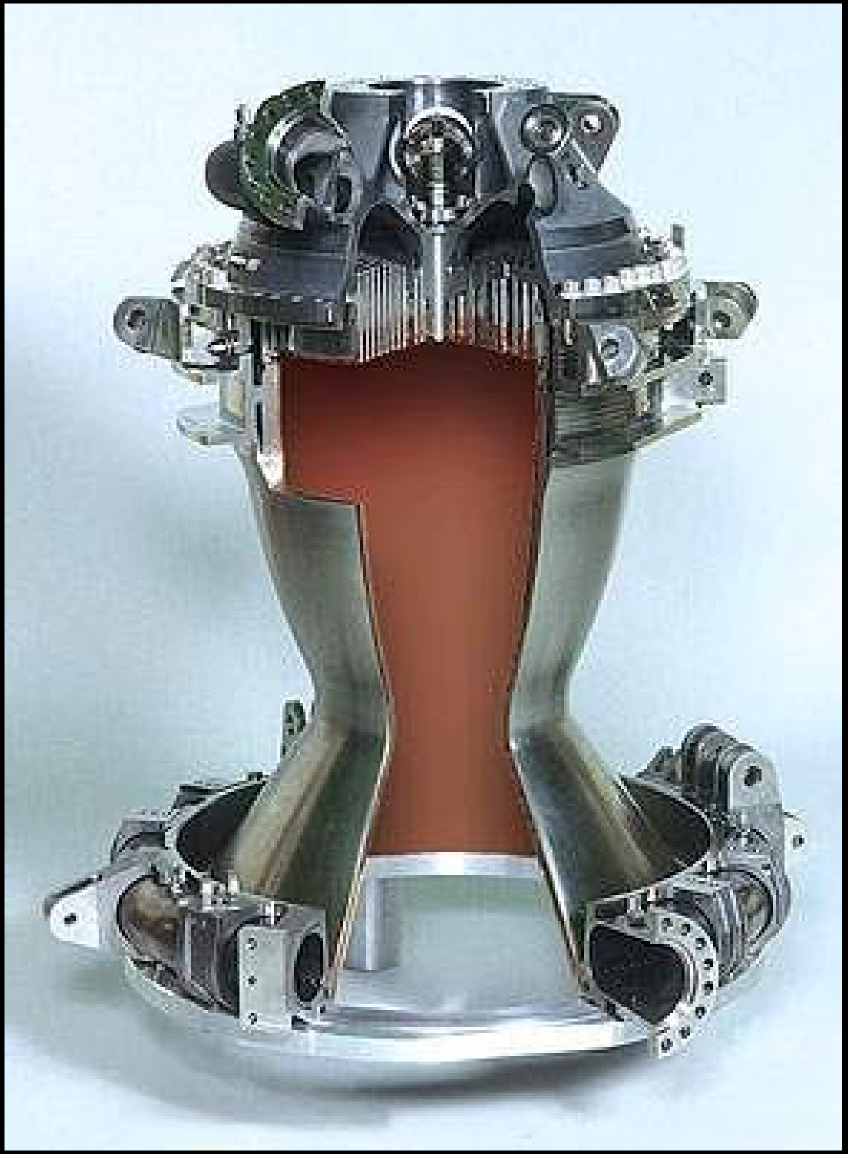
Thrust chamber

VULCAIN 2







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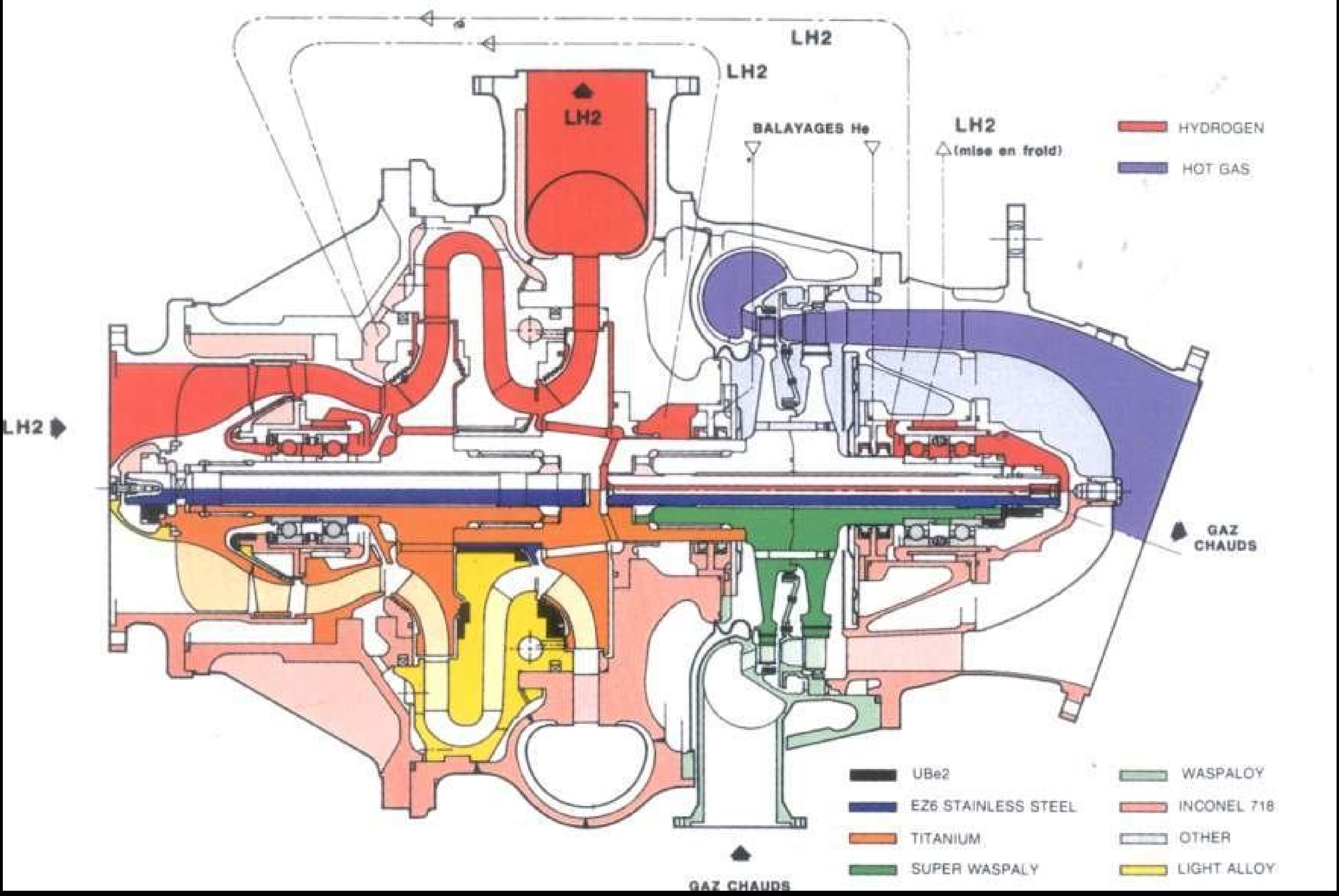
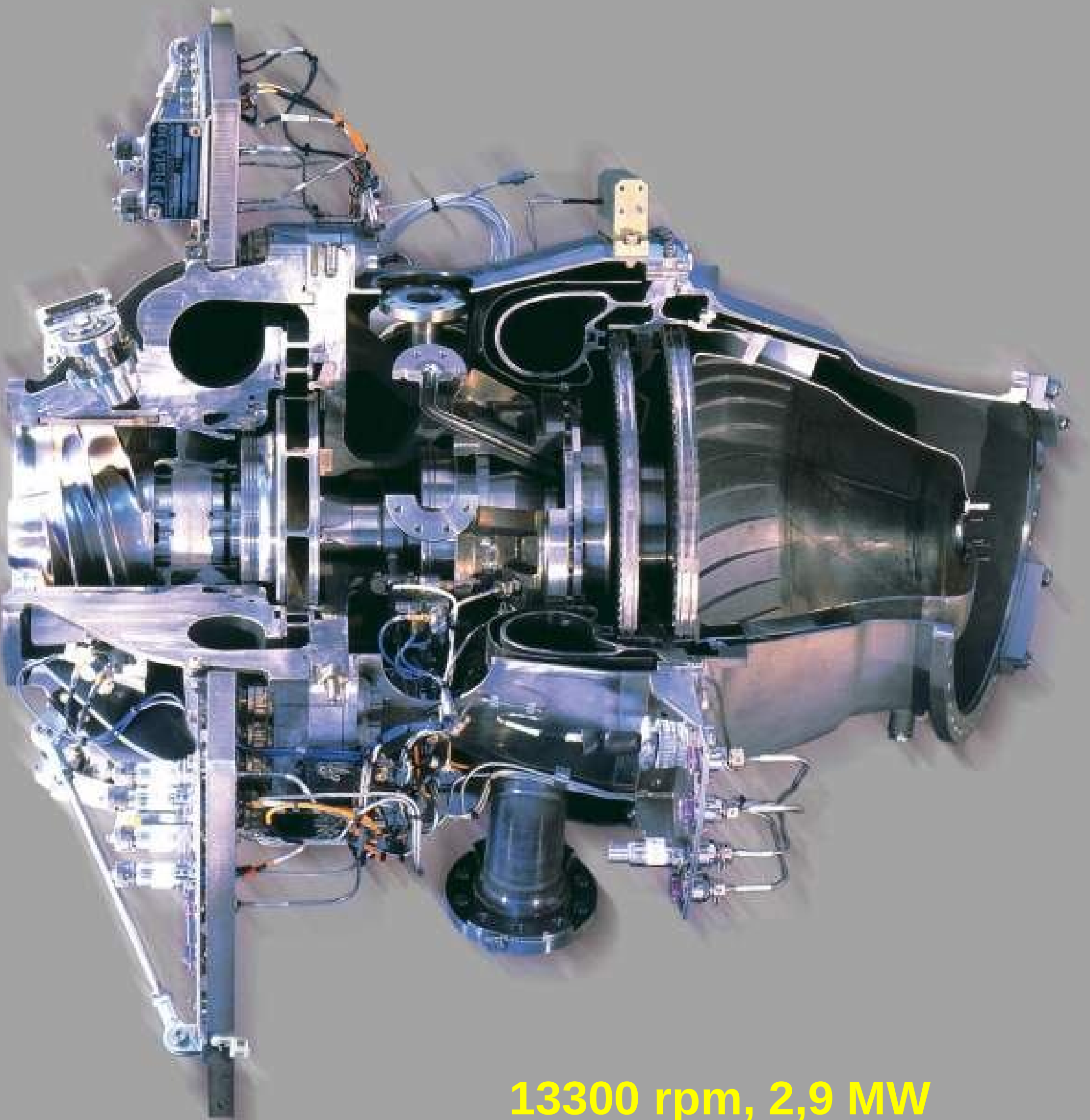
Hydrogen turbopump

VULCAIN 2



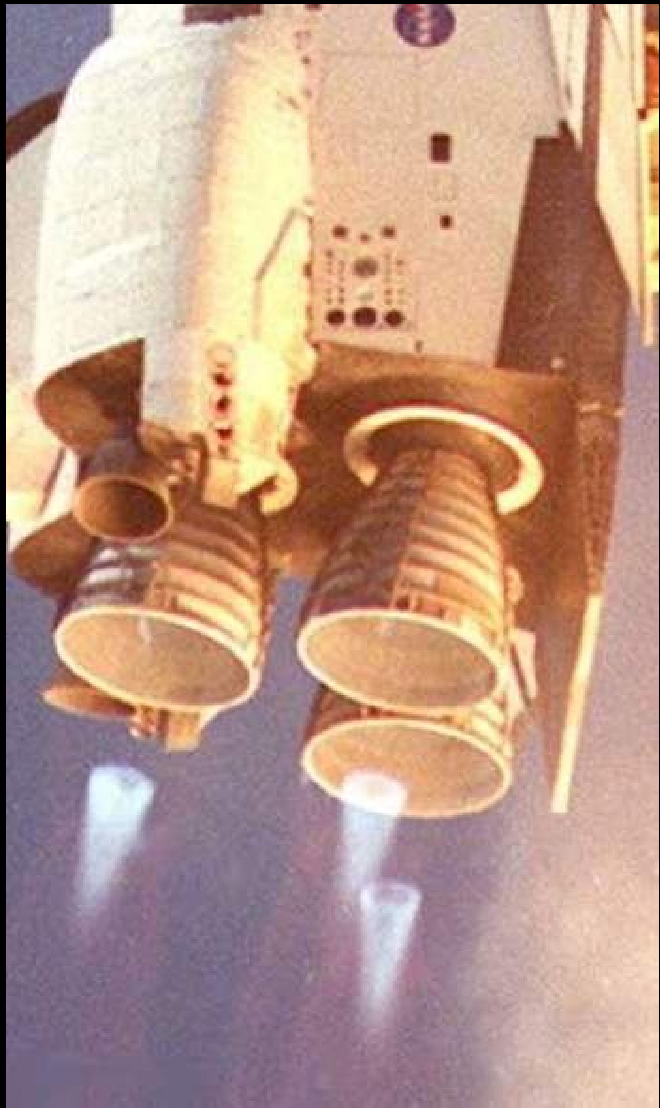
34200 rpm, 11,3 MW

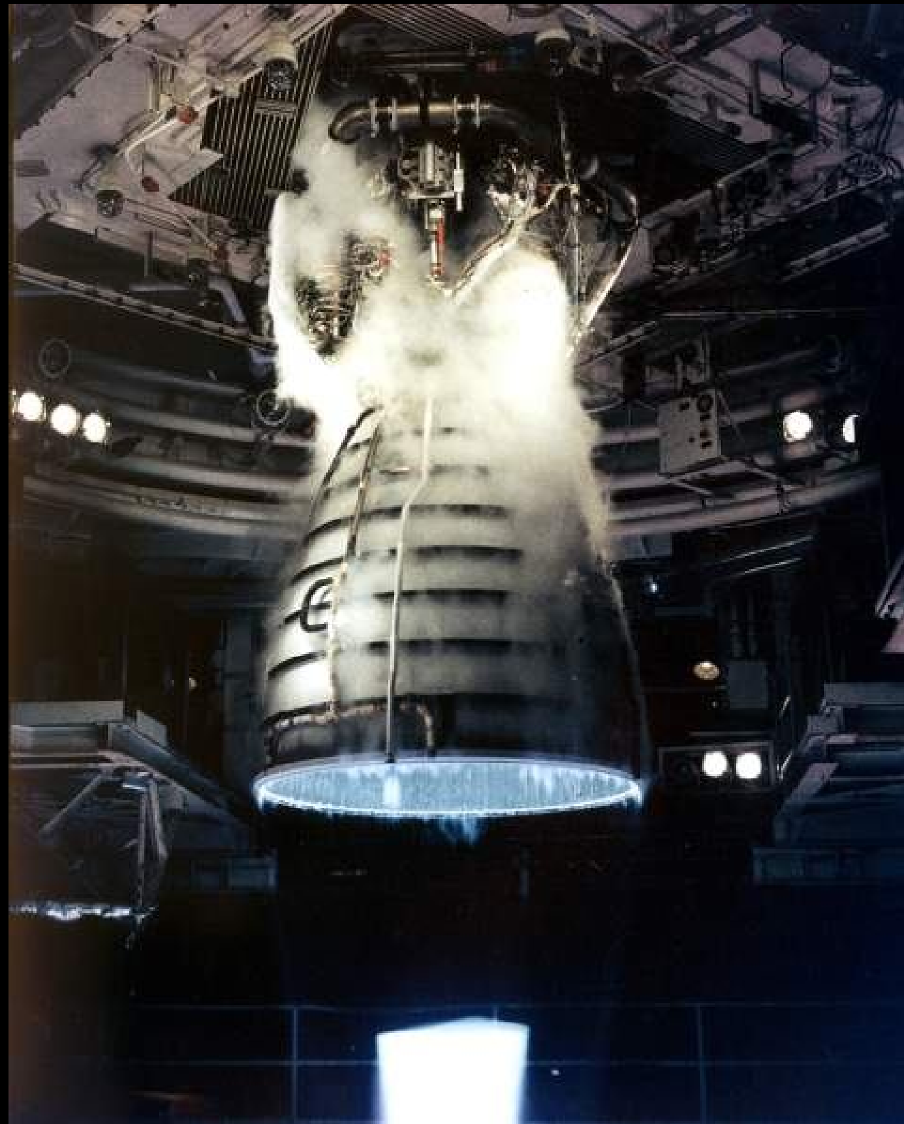
Oxygen turbopump



13300 rpm, 2,9 MW

Space Shuttle Main Engine





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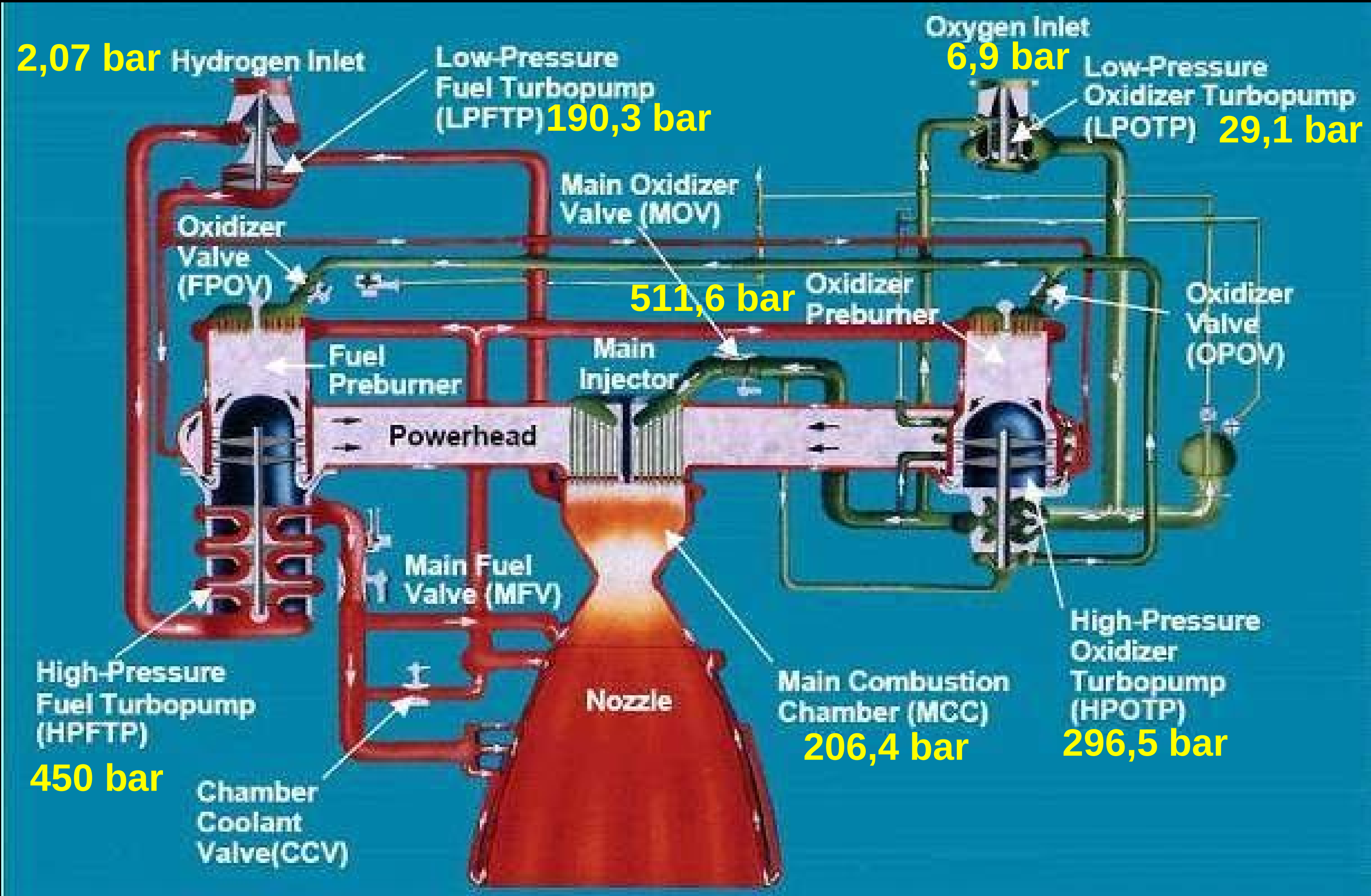
Engine characteristics

	SSME
Propellants	LOX – LH2
Vacuum thrust at 109 %	2279 kN
Mixture ratio	6
Vacuum specific impulse	452 s
Dry weight	3527 kg
Chamber pressure	206,4 bar
Expansion ratio	69
Throttle range	67 % - 109 %
Design life	7,5 hours 55 starts

SSME Flow schematic



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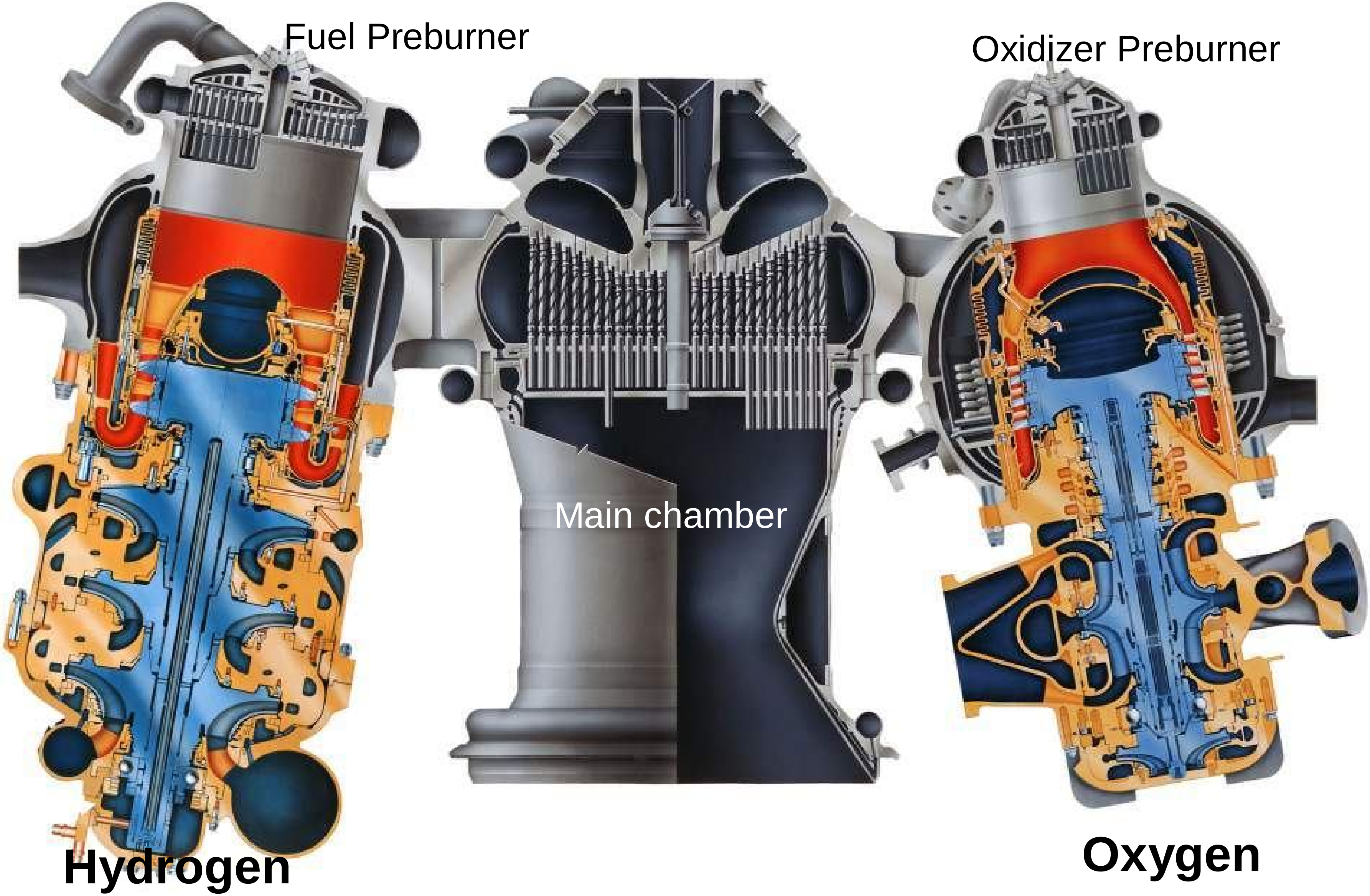
SSME Powerhead



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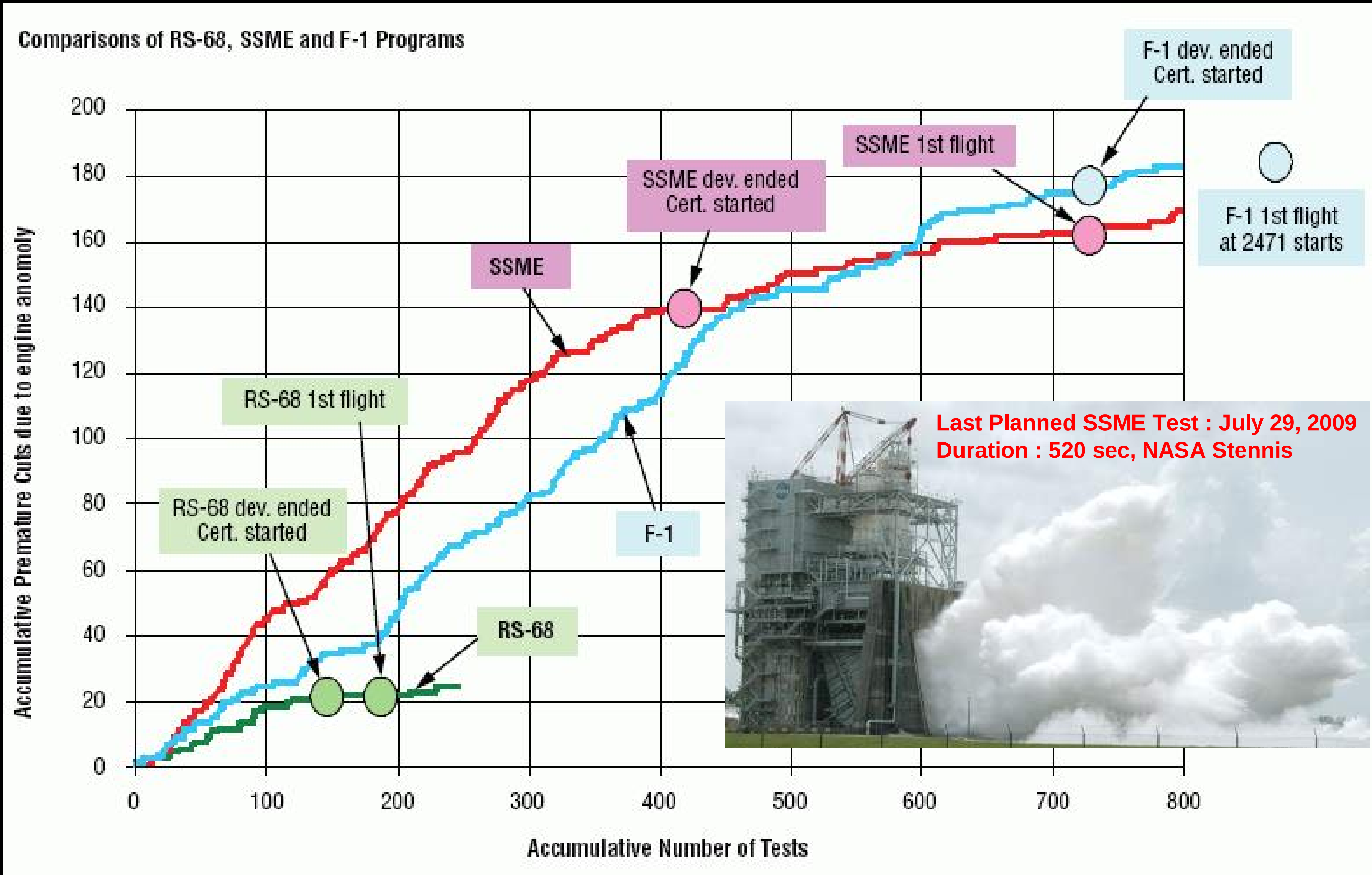
F1 vs. SSME test comparison

SSME



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VINCI

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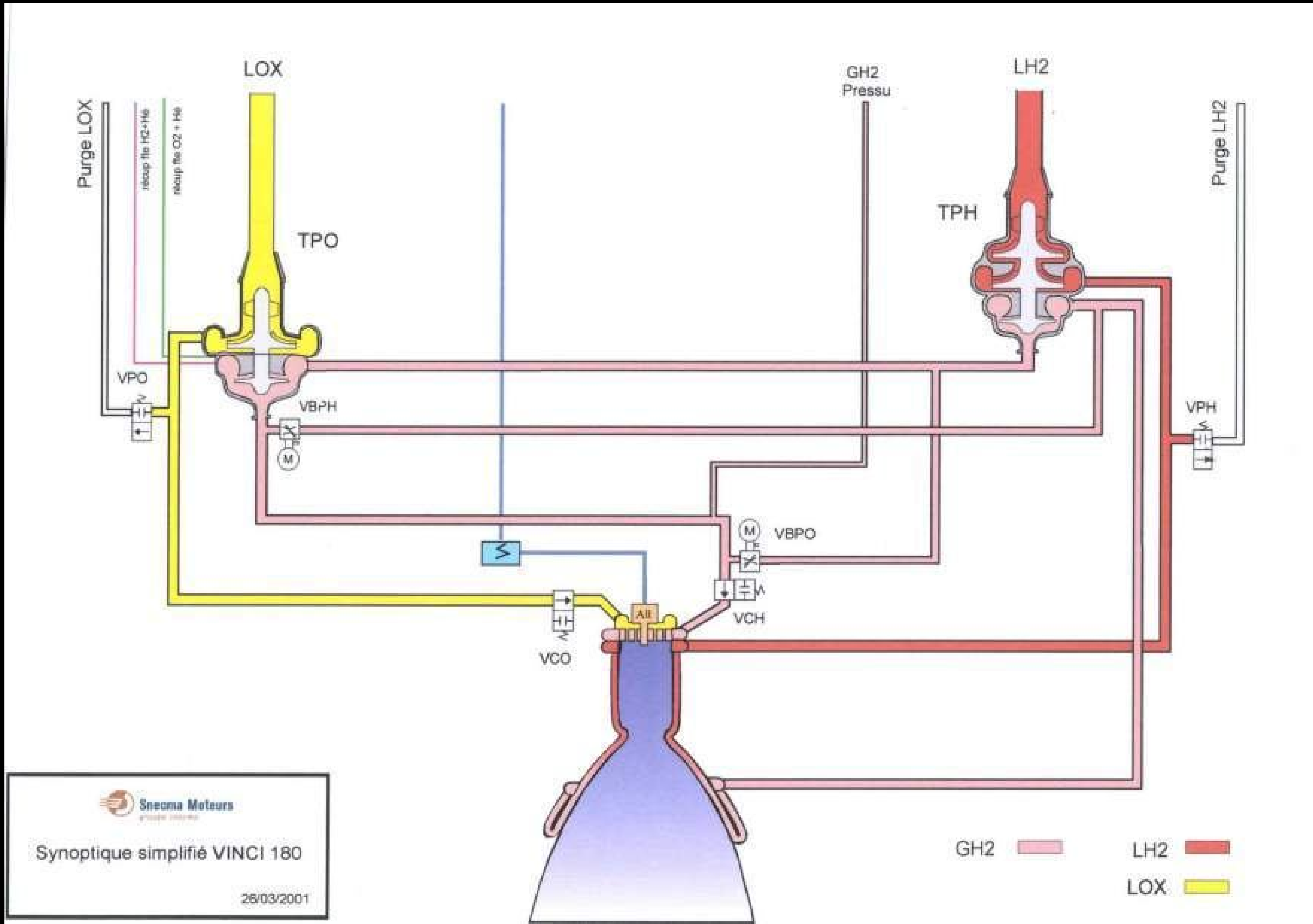
FLOW DIAGRAM



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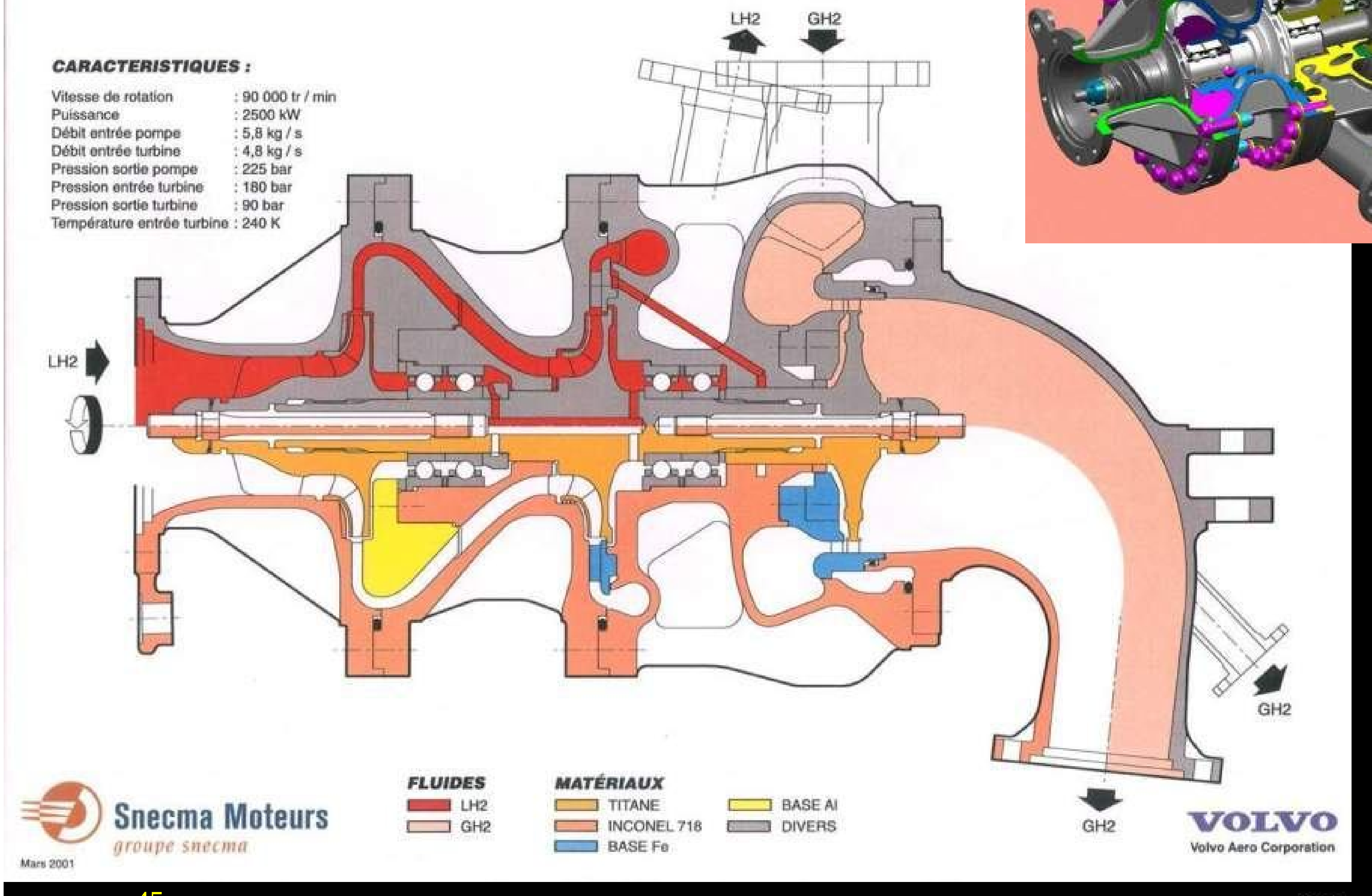
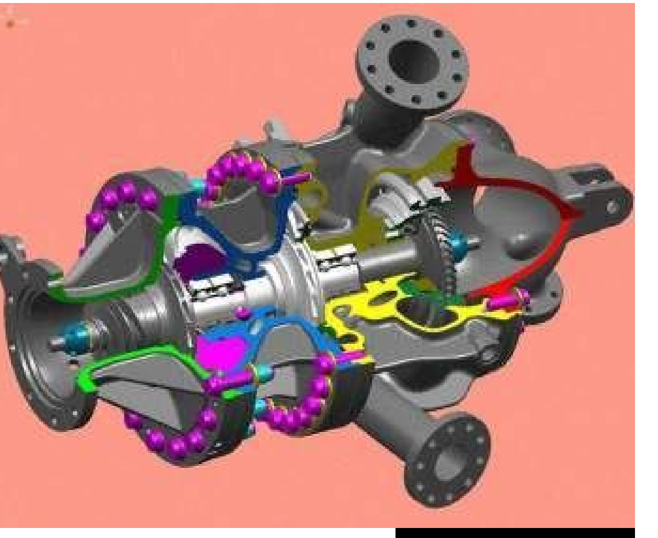
Engine characteristics

	Vinci
Propellants	LOX – LH2
Vacuum thrust	180 kN
Mixture ratio	5,80
Vacuum specific impulse	465 s
Chamber pressure	60 bar
Expansion ratio	240

Hydrogen turbopump





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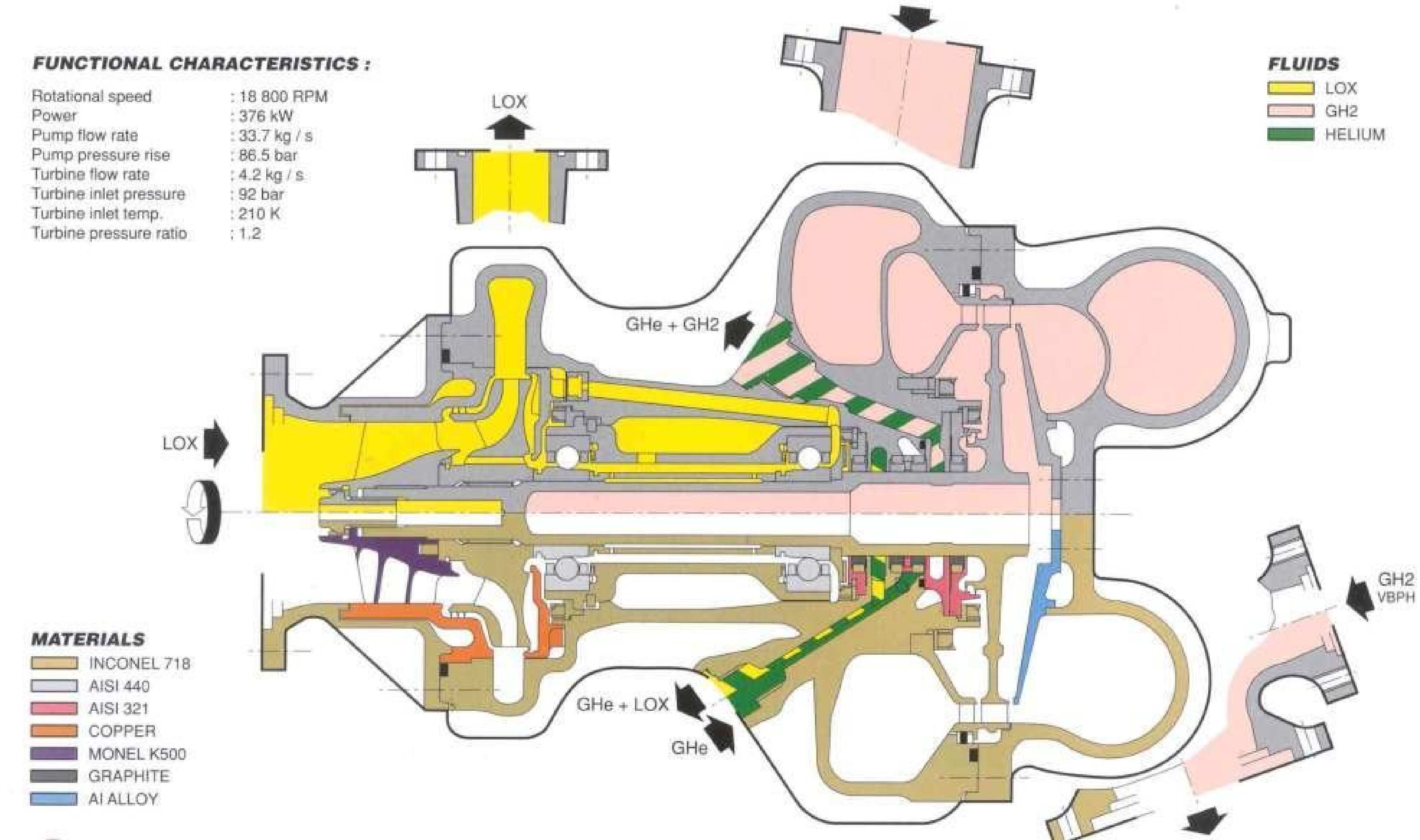
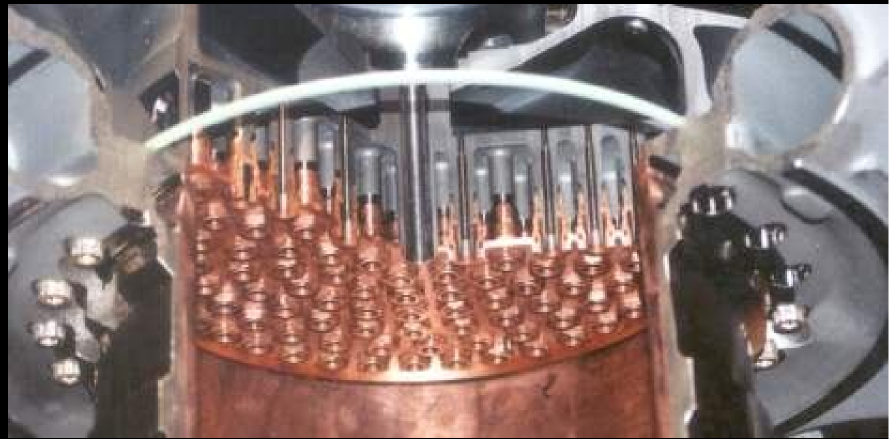
Oxygen turbopump

VINCI



Combustion chamber

VINCI



122 co-axial injection elements

228 cooling channels

Nozzle extension

Vinci test facility with altitude simulation

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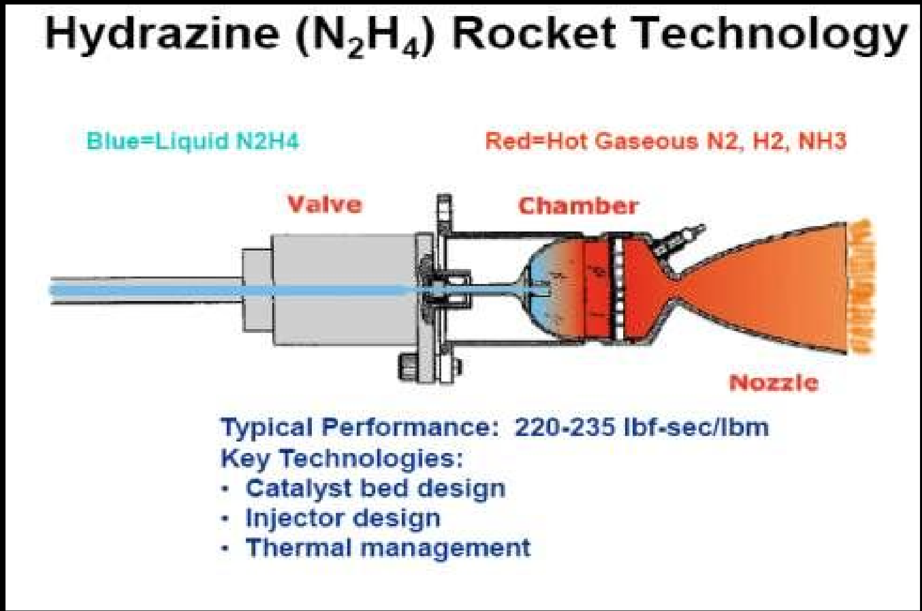
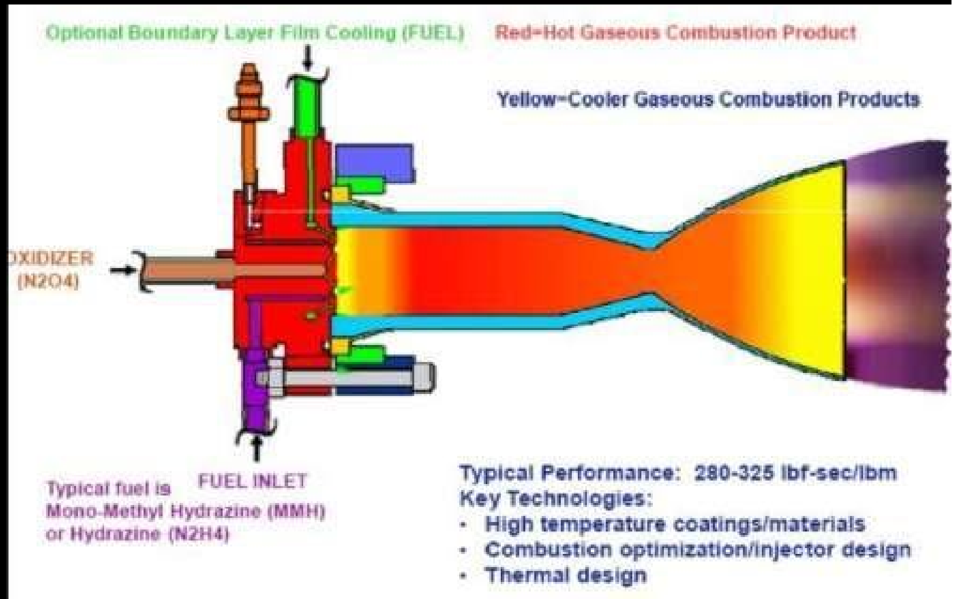
TYPICAL SMALL ENGINES



MONOPROPELLANT



BIPROPELLANT

Typical applications : **Attitude control**

Roll control

Soft landings (Moon, Mars, …)

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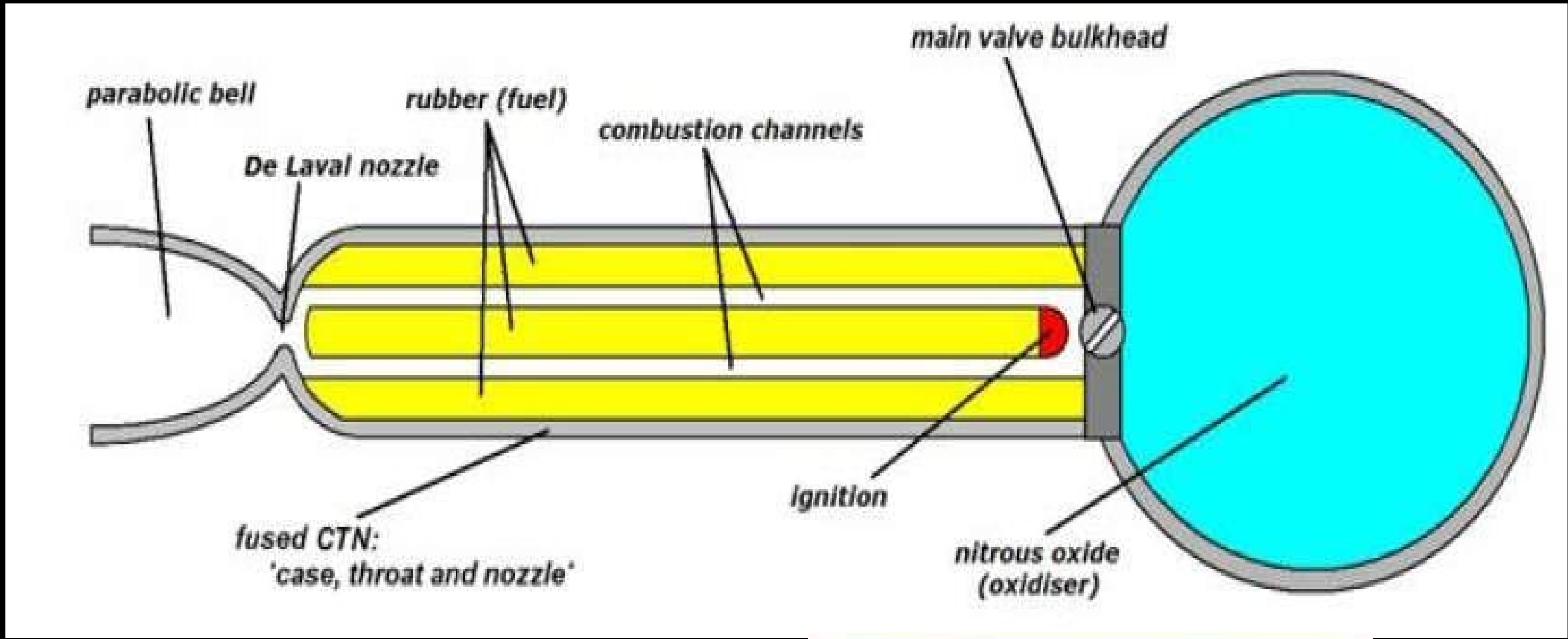
SPACESHIPONE HYBRID ENGINE



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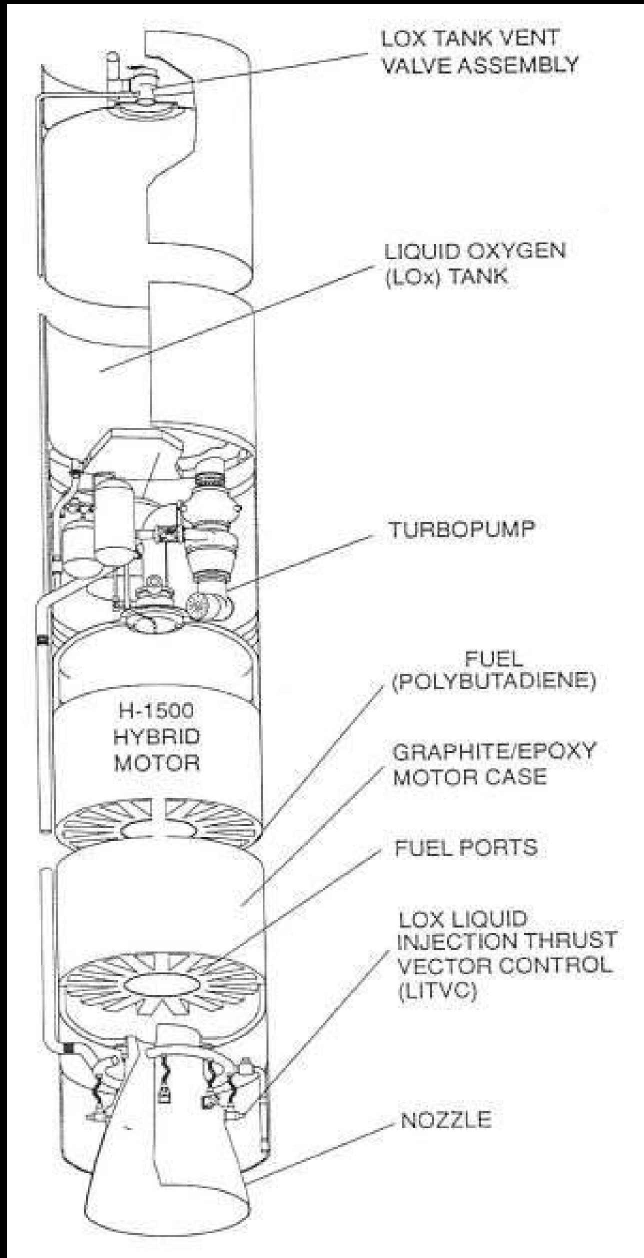
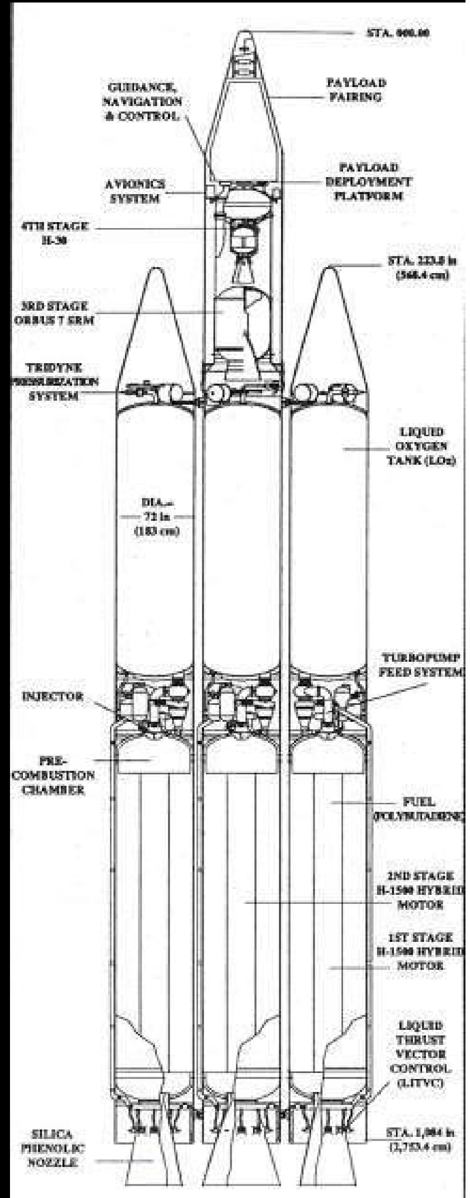
AMROC (USA)





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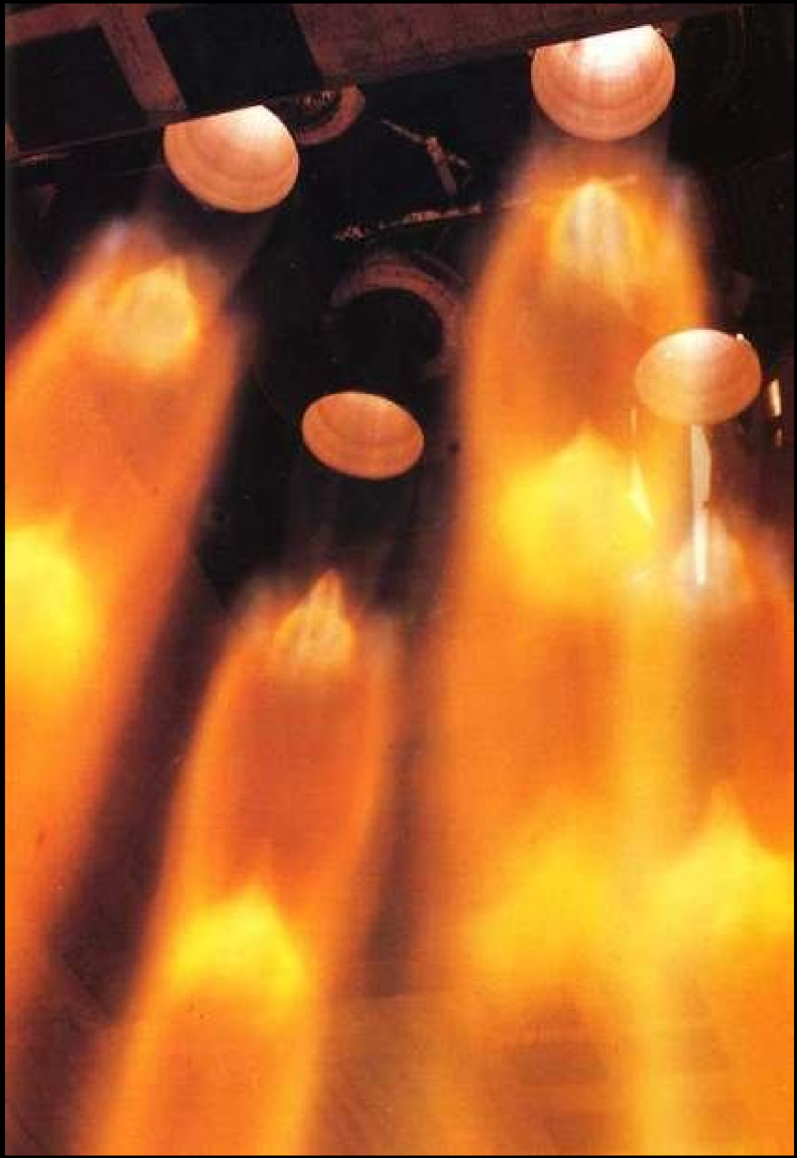
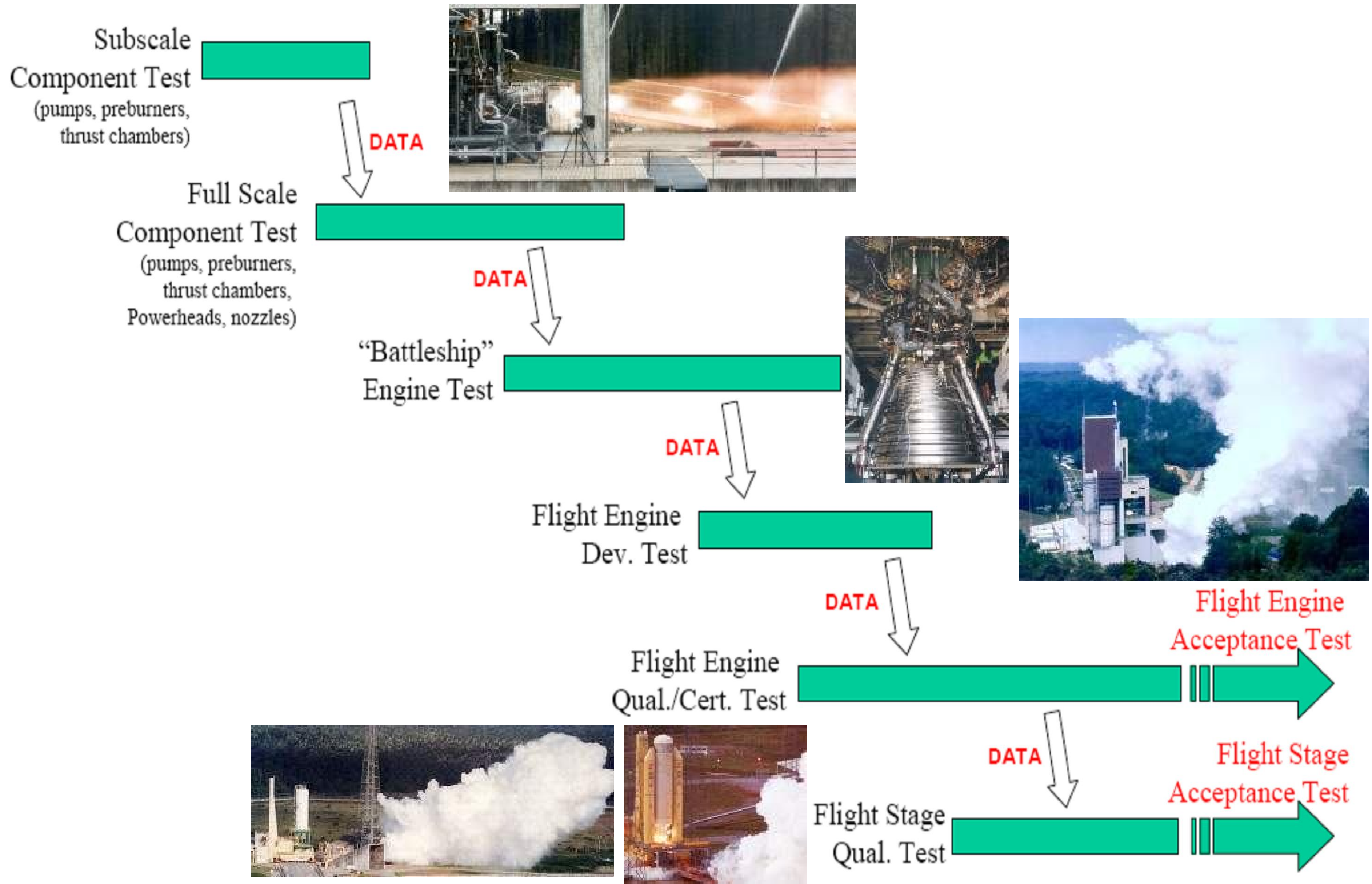
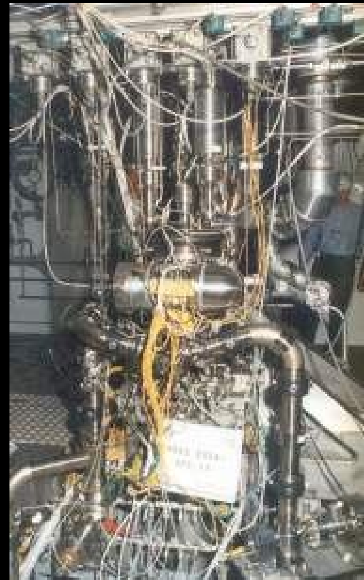
LIQUID ROCKET ENGINE TEST FACILITIES



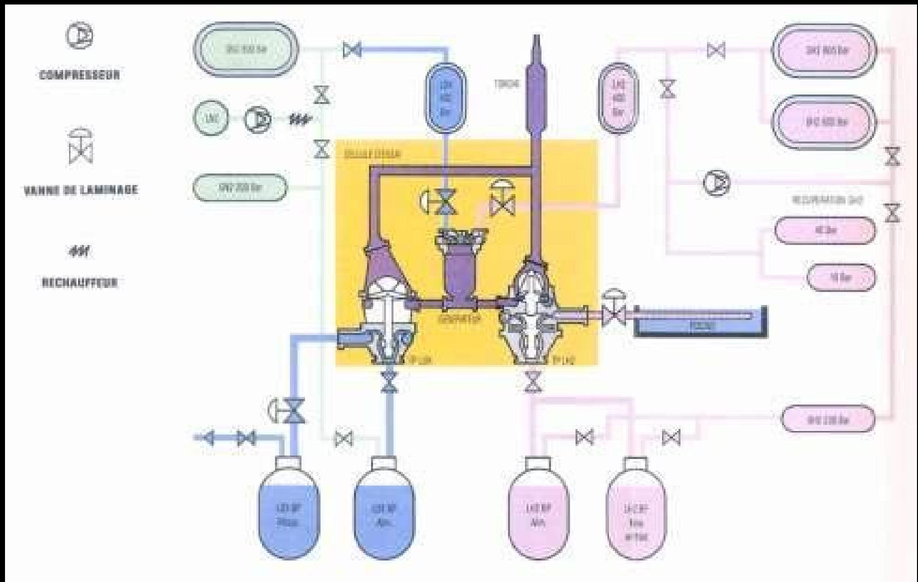
Major elements of a test program

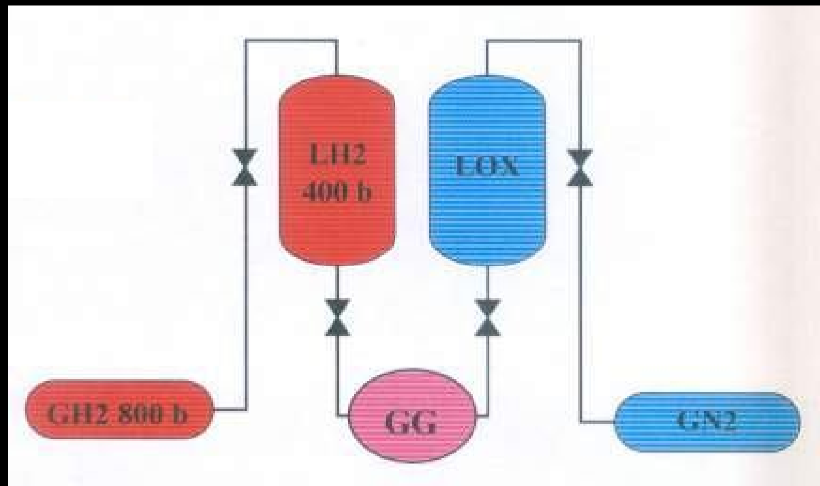


PF52 Vulcain powerpack test facility



Vulcain 1 – hydrogen TP test





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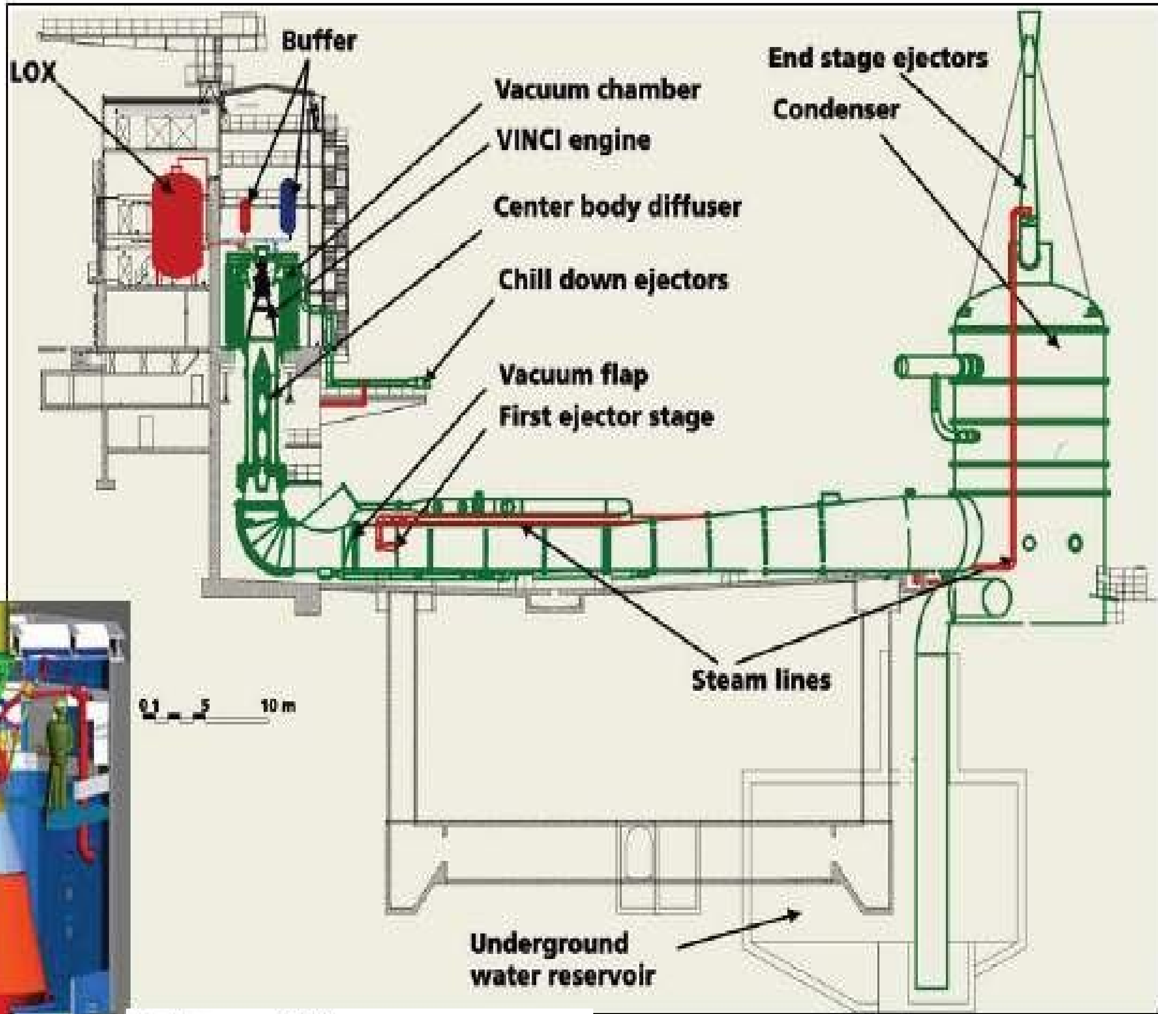
Vinci test facility with altitude simulation

ε=243

ε=22,3

ε=93



Engine ε = 22,3

Engine with fixed nozzle ε = 93

Engine with extendable nozzle ε = 243

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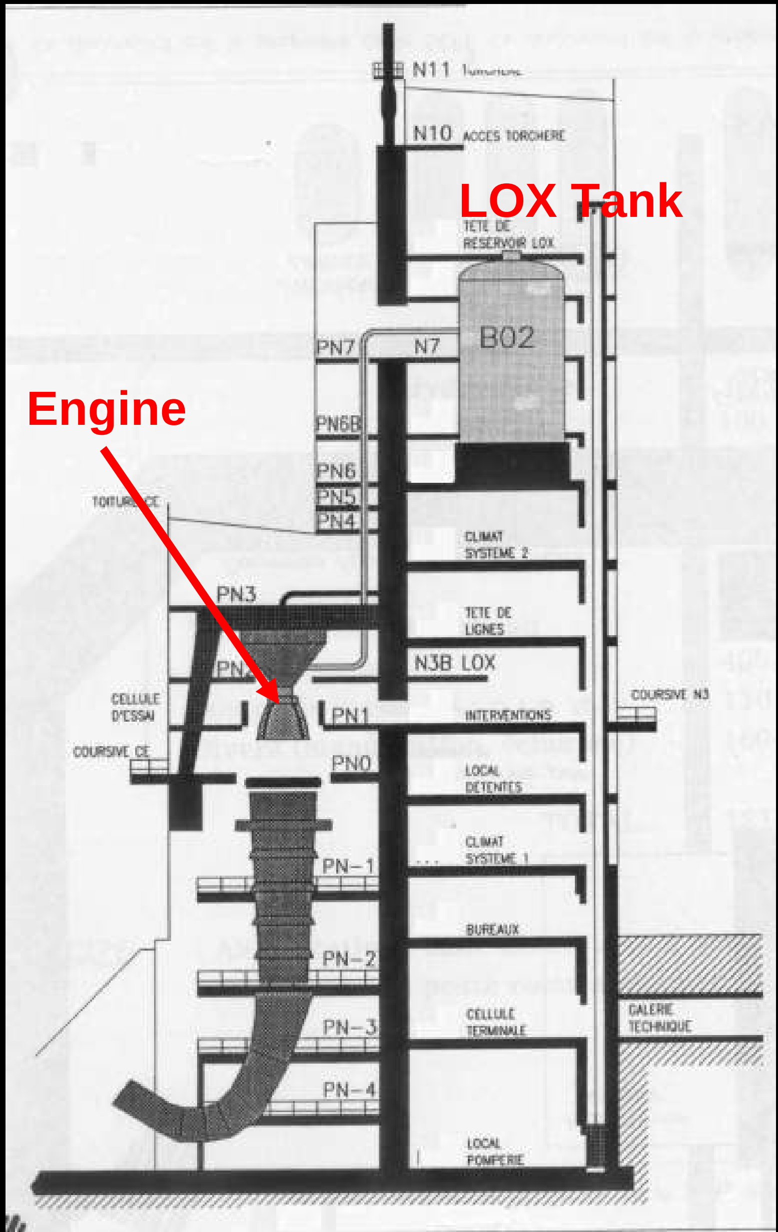
PF50 and P5 Vulcain engine test stands



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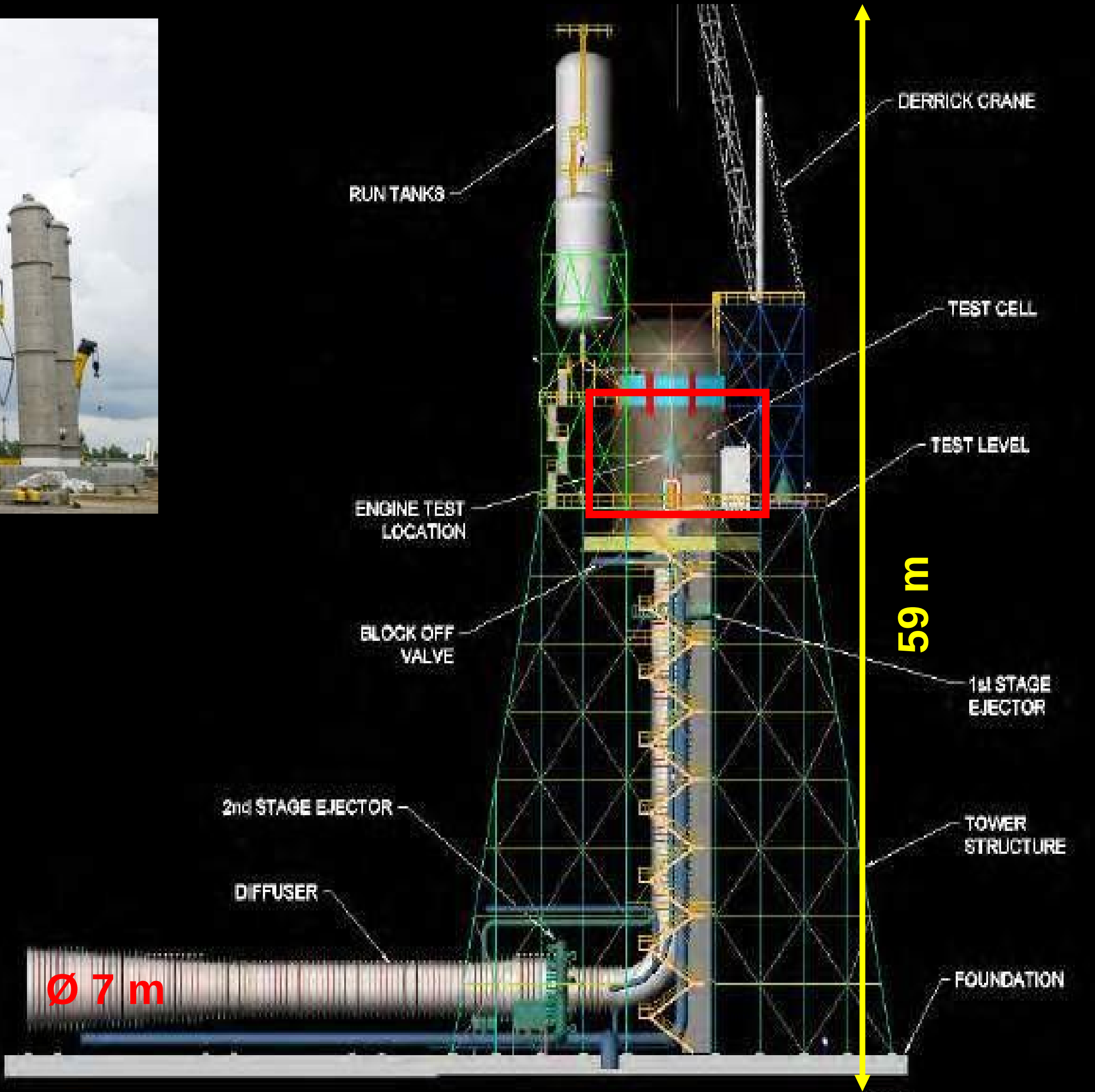
NASA A-3 J-2X altitude simulation engine test stand



simulated altitude : approximately 30 km

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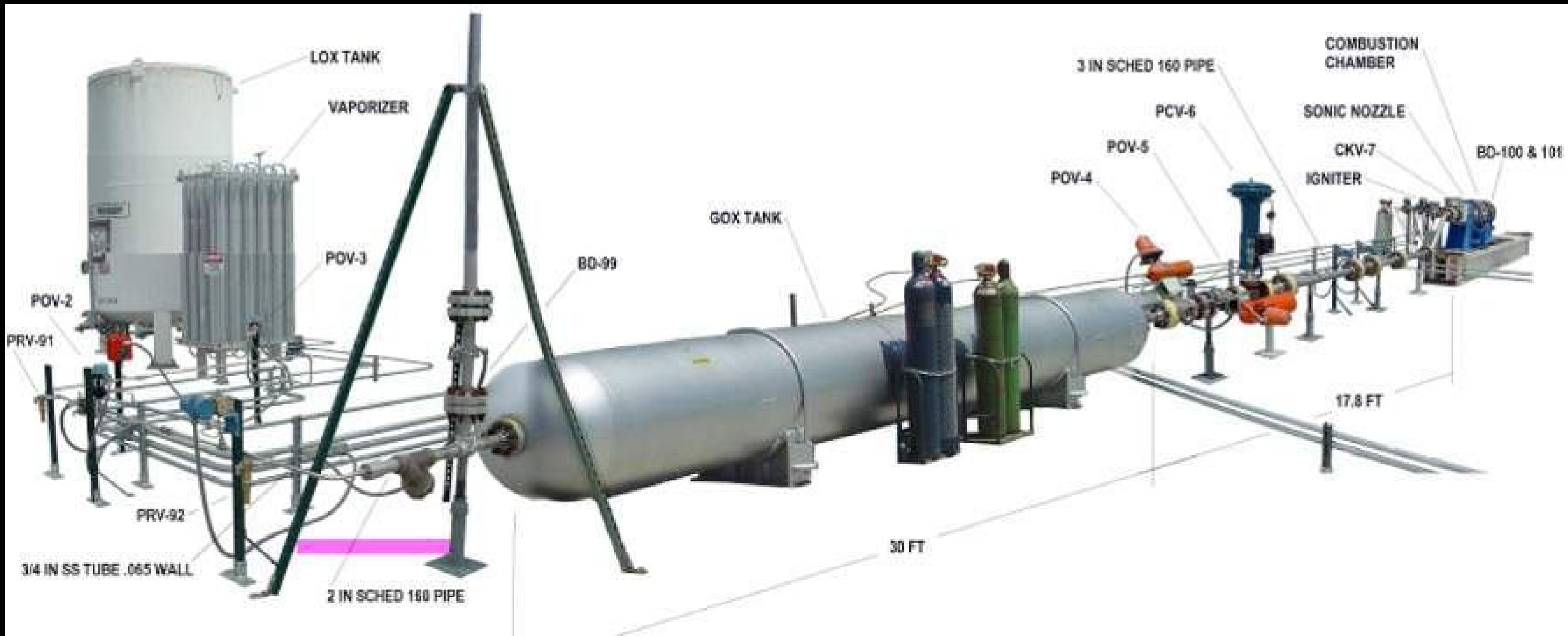
BEWARE OF HYDROGEN IMPURITIES…

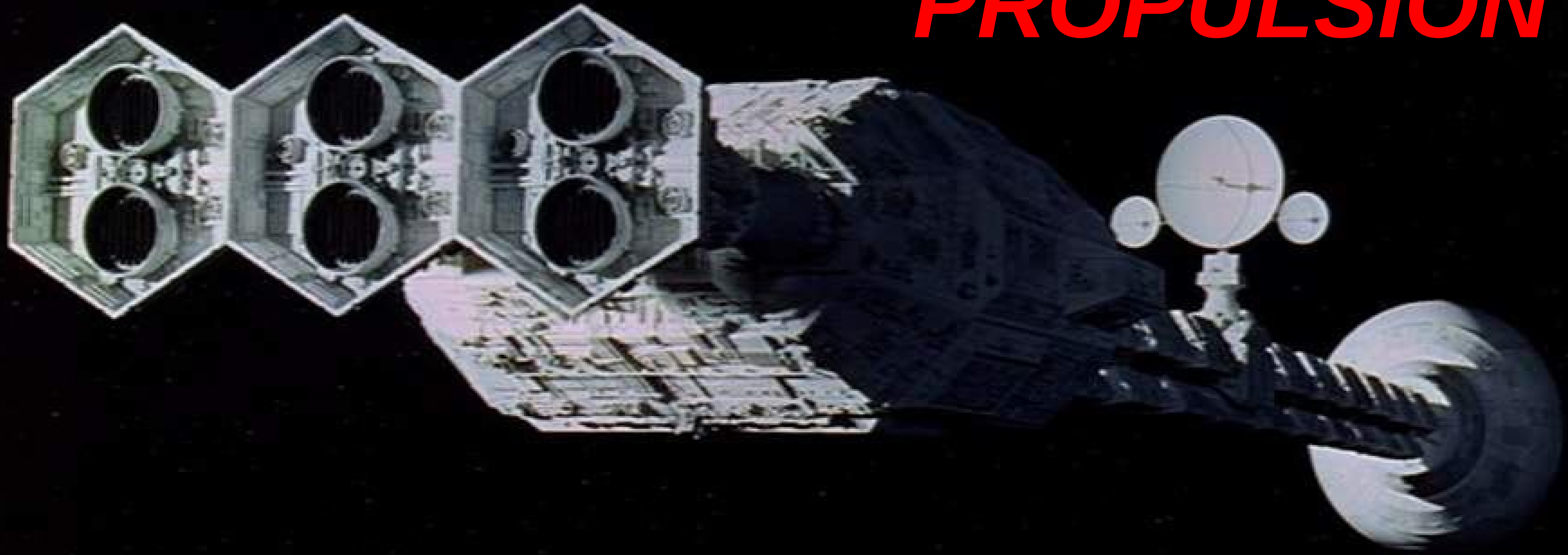
- Avoid risky configurations (e.g. low points, ...)
- Define and integrate the cleansing constraints
- Easily accessible equipment after commissioning,
- Rapid reaction instrumentation (e.g. fast pressure sensors, ...)

HYBRID ENGINE TEST FACILITY



Part 4

NUCLEAR PROPULSION



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INTEREST OF NUCLEAR THERMAL PROPULSION

Energy !

→1 kg of fissionable material (U235) contains
10 000 000 times more energy, as 1 kg of chemical fuel.

Consequences :

-Higher specific impulse - higher useful load fraction

- No oxidizer required !

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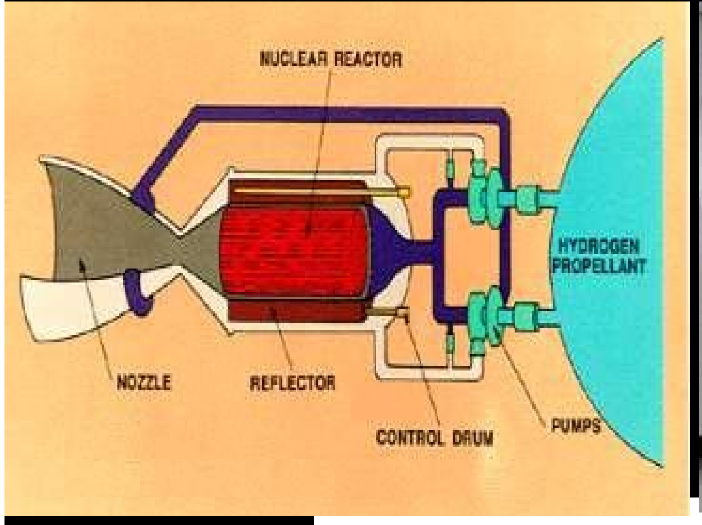
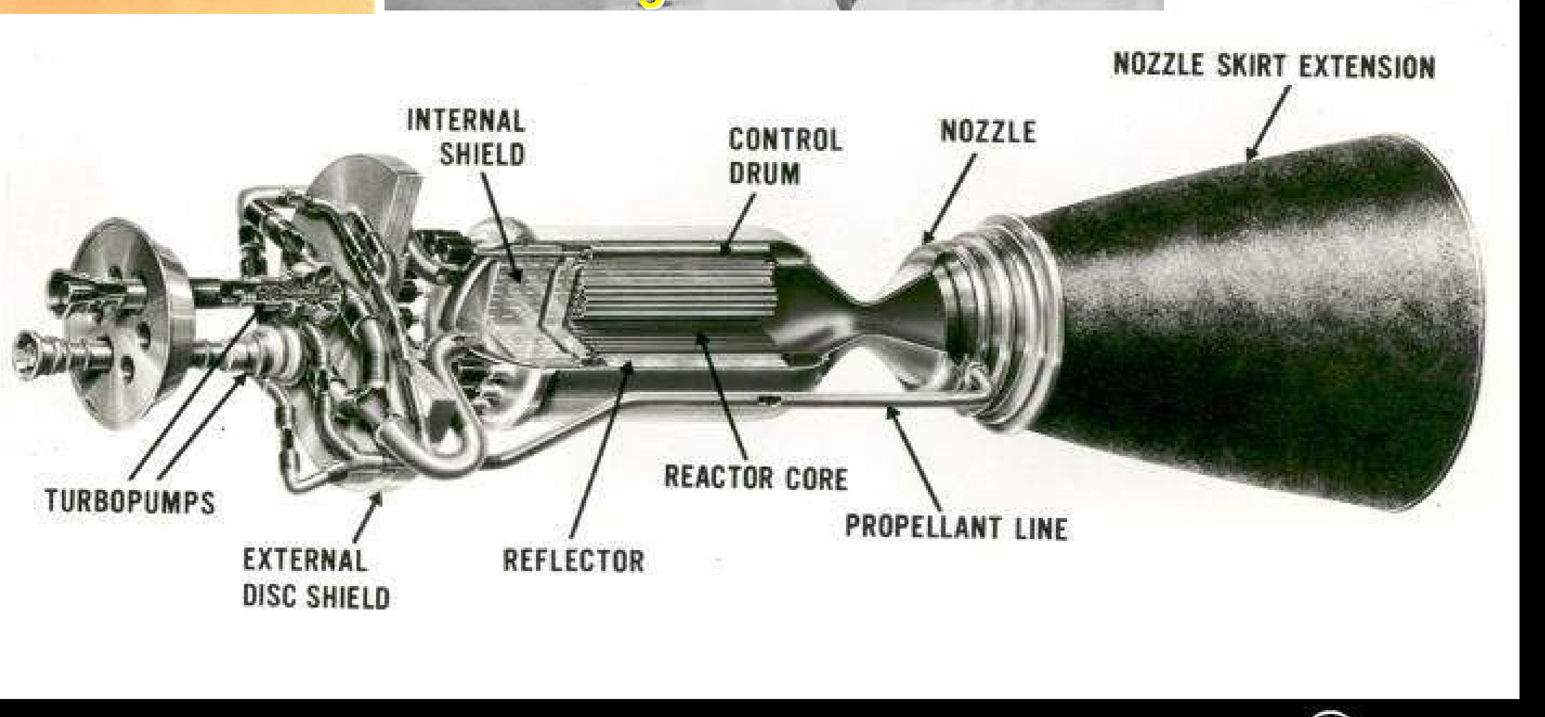
Nuclear - thermal propulsion



Isp = 888.3 s

- **F = 333.6 kN**
- **Tc = 2700 K**
- **m = 5.7 t**

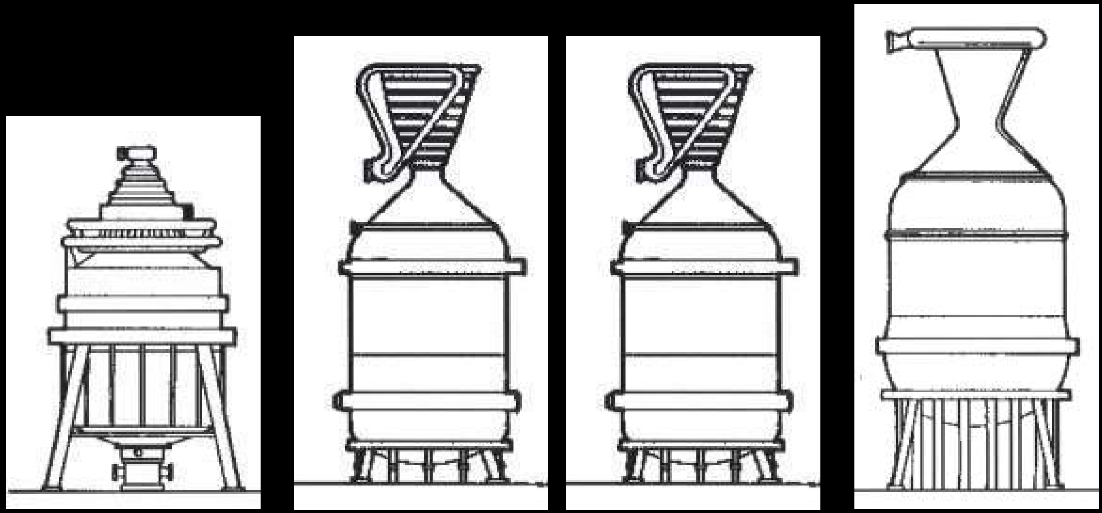


American nuclear rockets of the 1960's

Photo courtesy of National Nuclear Security Administration / Nevada Site Office

Engine and reactor in Engine Test Stand One

(Nevada Test Site)



KIWI A	KIWI B	Phoebus 1	Phoebus 2
1958-60	1961-64	1965-66	1967
100 MW	1000 MW	1000 MW	50000 MW
1.125 kN	11.25 kN	11.25 kN	56.25 kN

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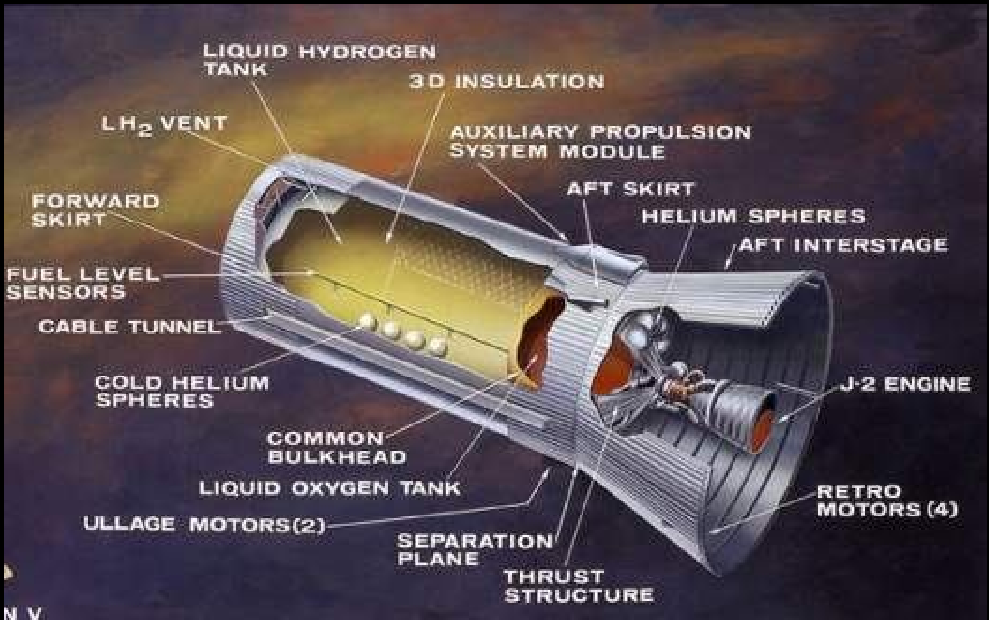
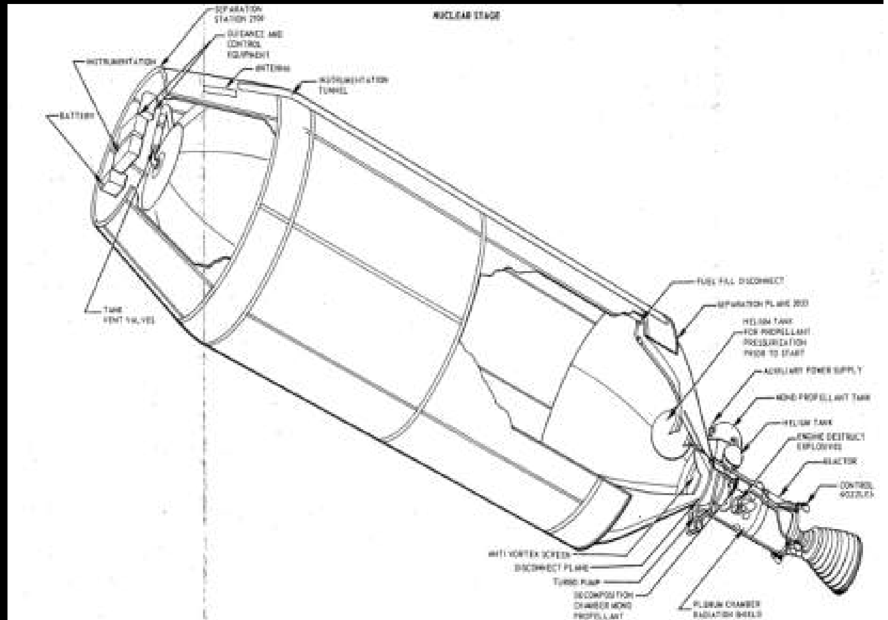
UPPER STAGE COMPARISON





	S-IV B	NERVA
Departure mass	121.2 t	53.694 t
Dry mass	12.2 t	12.429 t
Thrust	91 kN	266.8 kN
Duration	475 s	1250 s
Isp	4180 m/s	7840 m/s
Payload	39 t	54.5 t

For Saturn V

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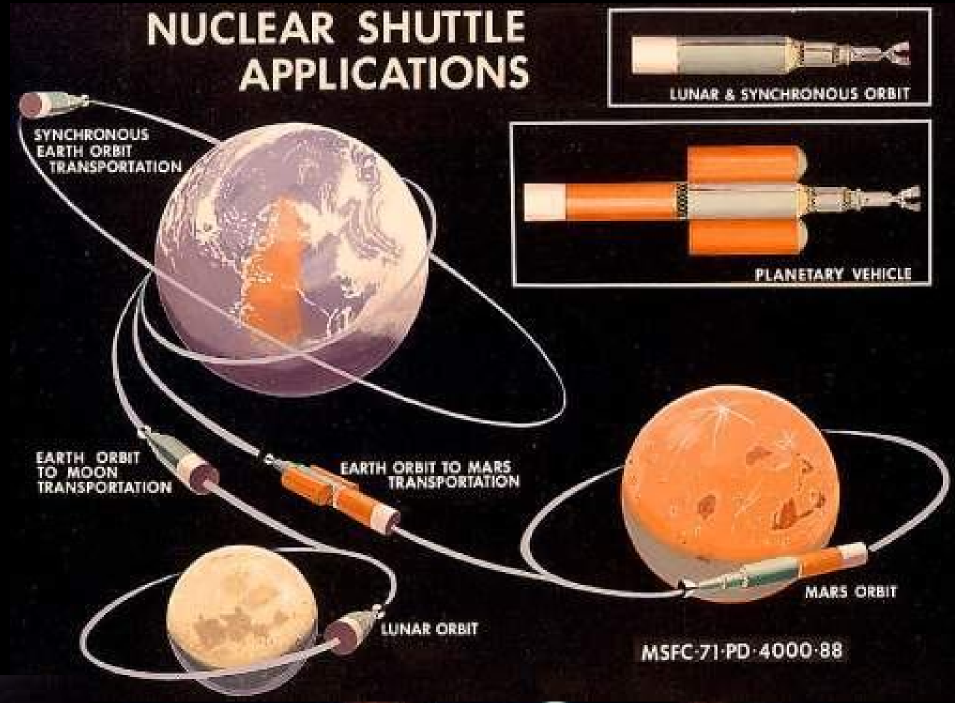
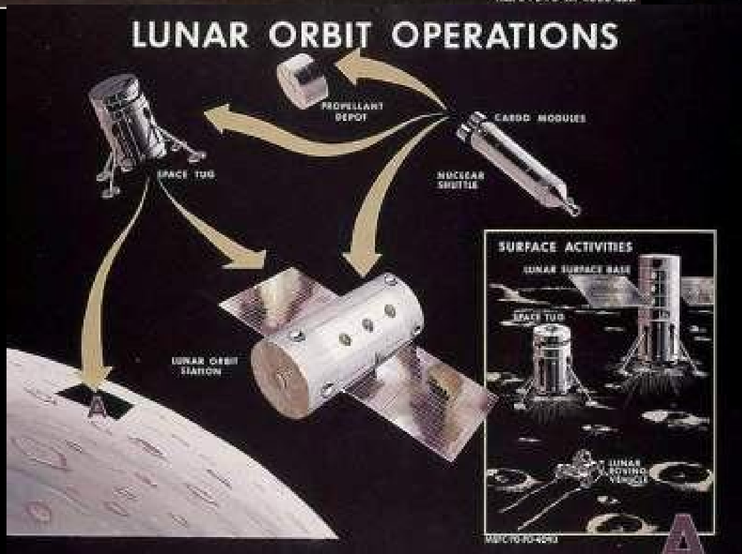
NERVA ultimate goal : manned Mars flight






ANY QUESTIONS ?
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