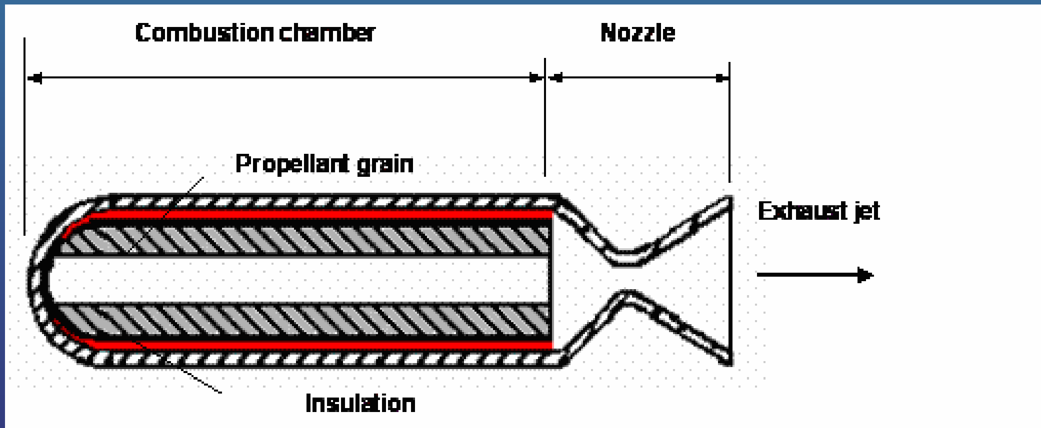
Space Propulsion

Solid rockets

The earliest recorded mention of gunpowder comes from China late in the third century before Christ. Bamboo tubes filled with saltpeter, sulphur and charcoal were tossed into ceremonial fires during religious festivals in hopes the noise of the explosion would frighten evil spirits. Certainly by the year 1045 A.D. the use of gunpowder and rockets formed an integral aspect of Chinese military tactics.

Rocket-like weapons were being used by the Mongols against Magyar forces at the battle of Sejo which preceded their capture of Buda Dec. 25, 1241. Not later than the year 1300, rockets had found their way into European arsenals, reaching Italy by the year 1500,



Propulsion force is delivered by a controlled explosion, heating up the reaction gases and expelling them through a nozzle

$$I_{sp} = V_e \propto \sqrt{\frac{2\kappa}{\kappa - 1}\frac{RT_c}{M}}$$

$\kappa = c_p/c_v$

The same exhaust – speed equation is valid for all “thermal” rockets (solid-, liquid-, cold gas-, resistojet-, arcjet thrusters) which produce their momentum out of the thermal motion of molecules.

cont’d

Space Propulsion

Performance data of typical solid booster motors (s.l./vac. = sea level/vacuum)

	Ariane 5 230P	Ariane 4 PAP	Titan 3 SRB	Space Shuttle SRB	Castor 4A	H2 SRB
Average thrust [kN]	6360 (vac.)	650 (vac.)	6227 (vac.)	11520 (vac.) 11.5 MN	436,7 (s.l.)	1560 (s.l.)
Specific impulse [s]	273 (vac.)	241 (s.l.)	265,2 (vac.)	268,4 (vac.)	237,6 (s.l.)	273 (vac.)
Total mass [ton]	268,1	11,6	238	569,9	11,6	70,4
Propellant mass [ton]	237,1	9,64	210,6	501,7	10,1	59,2
Mass fraction [-]	0,884	0,833	0,885	0,883	0,874	0,840
Burn time [s]	132,6	33	113,7	123,7	53,1	94
Diameter [m]	3,049	1,07	3,11	3,71	1,016	1,81
Length [m]	26,77	11,328	27,57	38,5	9,199	19,26
Av'ge pressure [bar]	?	?	?	45	46,6	46
Expansion ratio	9,6	8,3	10	7,72	8,58	10
Exit diameter [m]	2,826	0,958	3,05	3,801	0,82	1,69

$$I'_{sp} = \frac{I_{sp}[m/s]}{g} \qquad [s]$$

alternate definition of specific impulse

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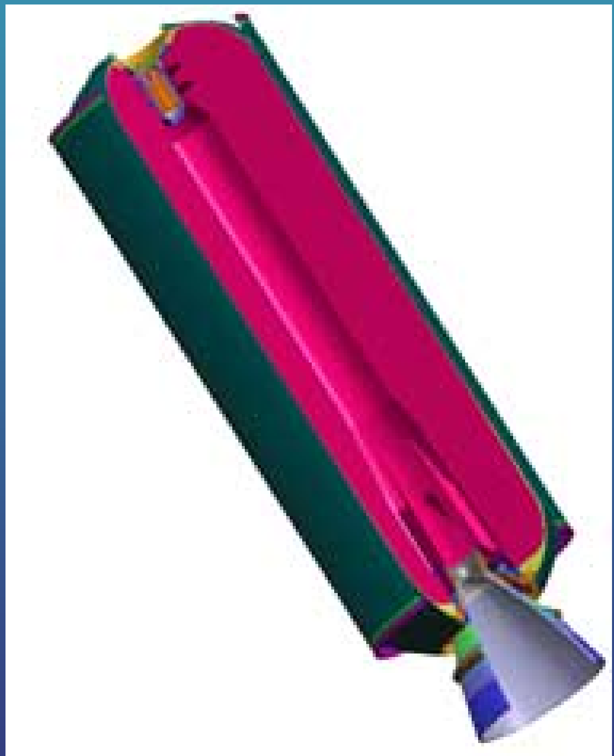
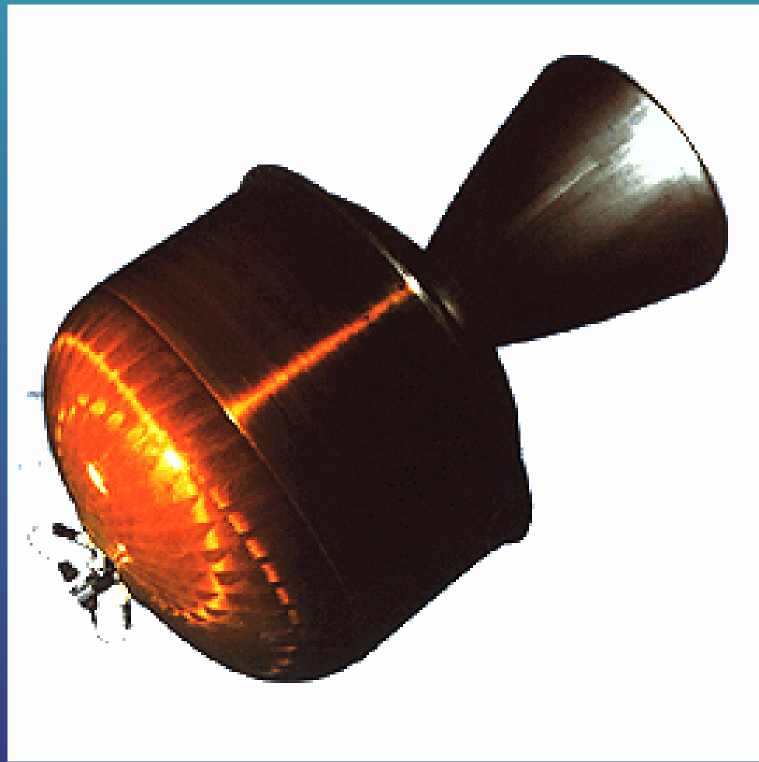
Space Propulsion

Space Propulsion



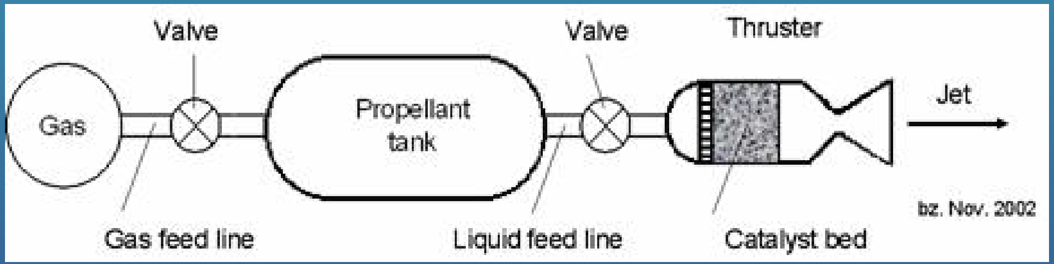


VEGA first stage solid rocket motor (courtesy ESA)& MAGE solid upper stage rocket motor

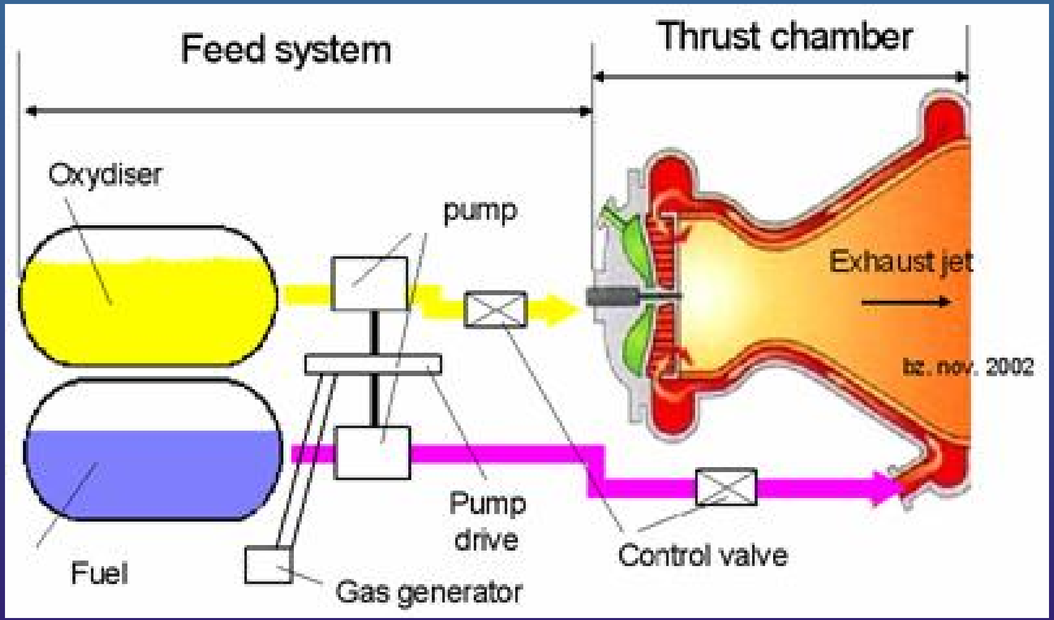
Space Propulsion

Liquid propellant rockets

monopropellant



bipropellant



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Space Propulsion

$$I_{sp} \propto \sqrt{\frac{2\kappa}{\kappa-1} \cdot \frac{RT_c}{10^{-3} M}}$$

Specific impulse of Advanced Chemical Propellants

Propellant Combination	Specific Impulse	
	[s]	[m/s]
Hydrogen-Oxygen ($0_2/H_2$)-space shuttle	**480**	**4709**
$0_2/H_2$ (ideal)	**528**	**5180**
Hydrogen-Fluorine (F_2/H_2)-ideal	**528**	**5180**
F_2/Li-H_2	**703**	**6896**
O_2/Be-H_2	**705**	**6916**
$0_3/H_2$	**807**	**7917**
Unproved Exotic Chemical Concepts		
Free Radicals (H + H) → H_2	**2,130**	**20,895**
Metastable Atoms (e.g. Helium)	**3,150**	**30,902**

cont'd

Space Propulsion

Thrust chamber or thruster: generally consists of a cylindrically or spherically shaped reaction or combustion chamber and a convergent-divergent outlet referred to as nozzle. In the reaction chamber, the chemical energy from the liquid propellant is converted into thermal energy, thereby creating a hot gas mixture. This hot gas mixture is accelerated by pressure forces to a high exhaust velocity in the specially shaped nozzle. To allow high gas temperatures in the thrust chamber, most combustion chambers and nozzles are cooled to some extent.

The liquid propellant: Most liquid propellants are **bipropellants**, which consist of a separate fuel and oxidizer. Fuels include liquid hydrogen (H_2), hydrazine (N_2H_4), ammonia (NH_3), hydrogen peroxide (H_2O_2), unsymmetrical di-methyl-hydrazine or UDMH, Rocket Propellant 1 (RP-1, some kind of kerosene devised by the nitro-methane, and methyl-acetylene. Oxidizers include liquid oxygen or LOX (O_2), fluorine (F_2), nitrogen tetra-oxide (N_2O_4), and hydrogen-peroxide. Some liquid propellants have both fuel and oxidizer properties in a single molecule. These propellants are known as **monopropellants**. Typical such propellants are hydrazine and hydrogen peroxide.

The feed system (including the propellant tanks): Because of the high pressure in the combustion chamber, a feed system is needed to pressurise and to transport the propellant from the propellant tank to the thrust chamber. Propellant pressurization is accomplished by either (turbo)-pumps or by a high pressure gas that is released into the propellant tanks. In case of high total impulse, short duration launcher missions, the choice is almost exclusively for pump-fed systems, whereas for low total impulse, long duration (typically years), orbital missions the choice is for using a high pressure gas.

cont'd

Space Propulsion

A control system: ensures the proper flow of propellants to the thrust chamber. It includes amongst others:

- On/off valves that control whether the propellant is flowing or not;
- Check valves that prevent the fluids from flowing in the wrong direction;
- Fill and drain valves that allow for filling and emptying of the tank(s) when on ground;
- Pressure regulators that control the pressure in the tank(s);
- Filters that filter out contaminants;
- Transducers that provide information on pressures, and temperatures.

distinction for liquid rockets is with regard to application:

Launcher propulsion: Attention is on achieving high thrust (to overcome gravitational acceleration) and high specific impulse to reduce propellant mass. For space launchers, because of the short thrust duration and because of the limited importance of short launch preparation time, cryogenic propellants may be used. For non-space (e.g. military)applications, storable propellants are used, like the hydrazine and nitric acid combination, as this allows for short launch preparation times (a few minutes to a few hours).

Spacecraft propulsion: Attention is on storable and hypergolic propellants (self – igniting upon contact) to attain a high reliability and to allow for long mission life (order of years). Hypergolic fuels commonly include hydrazine, monomethyl hydrazine (MMH), and unsymmetrical dimethyl hydrazine (UDMH). The oxidizer is typically nitrogen tetroxide (N_2O_4) or nitric acid (HNO_3).

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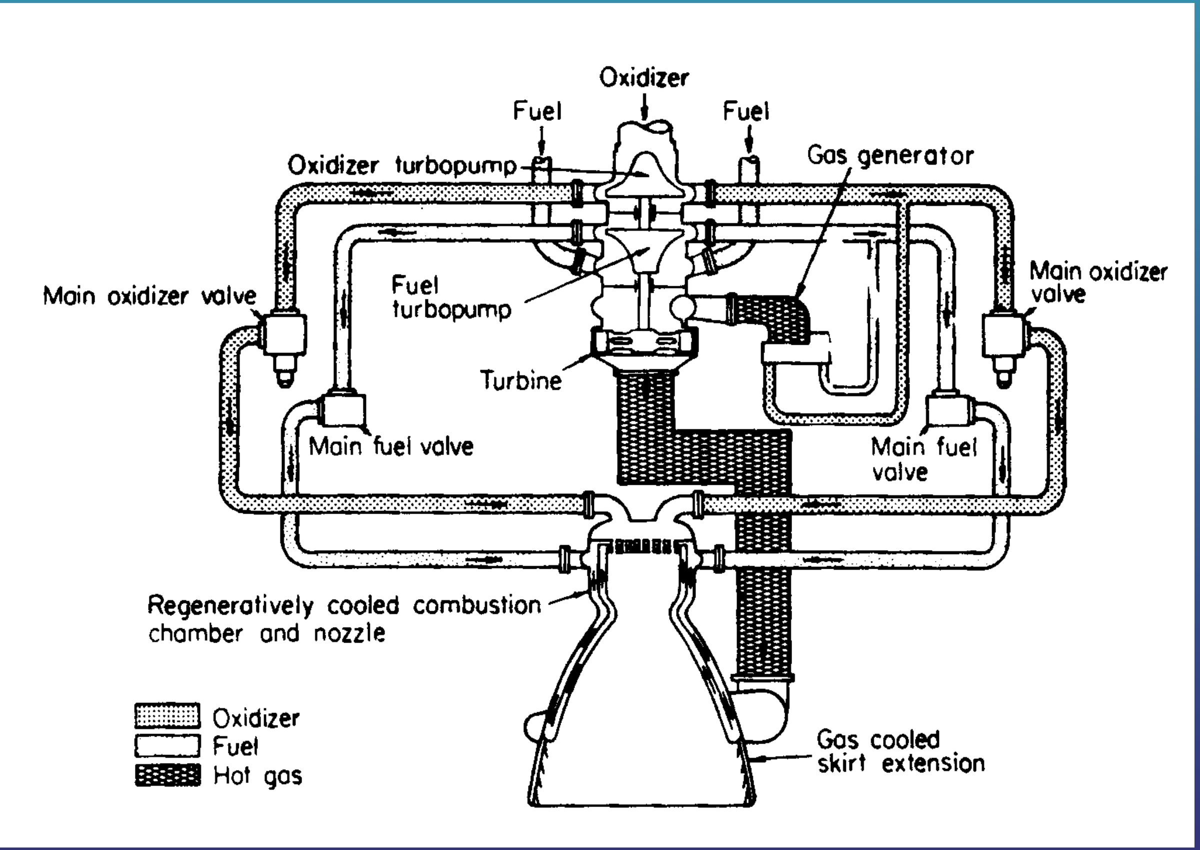
Space Propulsion



Space Propulsion

Specific mass characteristics of space launcher stages [Jane's]

Propellant type	Launcher	Stage designation	Dry mass [ton]	Propellant mass [ton]	Total mass [ton]	Dry mass fraction [-]
Earth storable	Ariane 4	L220	17.515	227.1	244.615	0.072
	Ariane 4	PAL	4.49	39.28	43.77	0.103
	Ariane 5	L9.5	1.19	9.7	10.89	0.109
	Long March 3/CZ-3	Stage 1	9	142	151	0.060
	Long March 3/CZ-3	Stage 2	4	35	39	0.103
	Long March 3/CZ-3	Stage 3	2	8.5	10.5	0.190
	Proton K/SL-12	Stage 1	32.5	420	452.5	0.072
	Delta 7920	Stage 2	0.924	6.006	6.93	0.133
	Cosmos/SL-8	Stage 1	5.3	82	87.3	0.061
	Cosmos/SL-8	Stage 2	1.443	19	20.443	0.071

Space Propulsion

Specific mass characteristics of space launcher stages [Jane's]

Propellant type	Launcher	Stage designation	Dry mass [ton]	Propellant mass [ton]	Total mass [ton]	Dry mass fraction [-]
Cryogenic	Saturn V	S-II	38	427	465	0.082
	Ariane 5	H155	12.6	156.2	168.8	0.075
	Saturn V	S-IVB	9.9	104.4	114.3	0.087
	H2	Stage 1	11.9	86.2	98.1	0.121
	Titan IV	Centaur	3	23	26	0.115
	H2	Stage 2	3	16.7	19.7	0.152
	H1	Stage 2	1.8	10.6	12.4	0.145
	Ariane					

Space Propulsion

Performance characteristics of specific high total impulse rocket engines

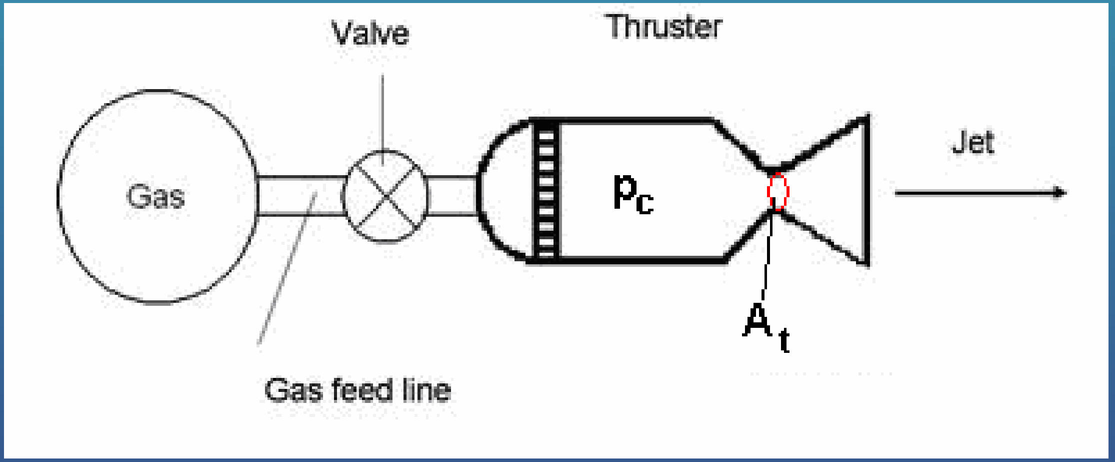
Characteristic	SSME	HM60	LE-7	RS-68 Boeing	RD-170 ENERGIA	RD-180	RS-2200
Propellants	LOX / LH_2	LOX / LH_2	LOX / LH_2	LOX / LH_2	LOX / Kerosene	LOX / Kerosene	LOX/LH_2
Engine cycle	SCC	GGC	SCC	GGC	SCC	SCC	GGC
Vacuum thrust (kN)	2090	1075	1078	3310	7910	4152	2200
Specific impulse (s)	455.2	430	446	410	336-337	338	455
Overall mass mixture ratio	6.0	5.3	6.0		2.63	2.72	5.5
Propellant density[2] (kg/m^3)	333	346	333		1008	1011	354
Length (m)	4.24	3.00	3.40	5.18	4.0	3.8	
Total dry mass (kg)	3170	1300	1714	6597	9750	5393	2670
Mission duty cycle (s)	480	600	346		140-150	150	
Life span (s)	27000		>2000				
Max number of restarts	>100	2	>20				
Thrust/weight ratio	67	84	64	51	83	79	84
Throttle capability (%)	67-109		No	60-100	56-100	50-100	Yes
First flight in (year)	1981	1996	1994	2001?	1985	1999?	
Reliability	0.999	0.9927	0.9935		0.999		

Space Propulsion

Manufacturer Name: RD-170. *Government Designation*: 11D520.
Designer: Glushko. *Developed in*: 1981-93. *Application*: Energia strap-on. *Propellants*: Lox/Kerosene. *Thrust(vac)*: 7,903.000 kN (1,776,665 lbf). *Thrust(sl)*: 1,887.500 kN (424,327 lbf). *Isp*: 337 sec. *Isp (sea level)*: 309 sec. *Burn time*: 150 sec. *Mass Engine*: 9,750 kg (21,490 lb). *Diameter*: 4.02 m (13.17 ft). *Length*: 3.78 m (12.40 ft). *Chambers*: 4. *Chamber Pressure*: 245.00 bar. *Area Ratio*: 36.87. *Oxidizer to Fuel Ratio*: 2.60. *Thrust to Weight Ratio*: 82.66. *Country*: Russia. *Status*: Out of Production. *First Flight*: 1987. *Last Flight*: 1988. *Flown*: 8.00.

Space Propulsion

Cold gas thrusters



Cold gas thrusters are often used on spacecraft as **attitude control systems**. These systems are mostly used in cases requiring low total impulse of up to 4000 Ns or where extremely fine pointing accuracy or thrust levels must be achieved or the use of chemical propellants is prohibited for safety reasons.

cont'd

Space Propulsion

$$I_{sp} \propto \sqrt{\frac{2\kappa}{\kappa-1}\frac{RT_c}{10^{-3}M}}$$

Characteristics of some candidate gases for cold gas thrusters

gas	M	κ	I_{sp} [m/s]
H_2	2	1,4	2845
He	4	1.659	1727
N_2	28	1.4	746
NH_4	17	1.31	1030
N_2O	44	1.27	657
Freon-14	88	1.22	481

cont'd

Space Propulsion

thrust is that fraction of the wall pressure force in the thruster chamber which is not counteracted by the walls

$$T \cong p_c A_t \quad [N]$$

propellant mass required to produce a total impulse **I** over mission time t_m is

$$m_p = \frac{I}{I_{sp}} = \frac{T.t_m}{I_{sp}} \quad [kg]$$

tank volume V_t to store the propellant mass for the mission at a tank pressure of p_t is

$$V_t = \frac{m_p R T_t}{10^{-3} M p_t}$$

minimum impulse bit I_{min} is determined by chamber pressure and switching time t_s of the on / off valve

$$I_{\min} = T.t_s = p_c A_t t_s \quad [N.s]$$

cont'd

Space Propulsion

Example 10 *A cold – gas system has to meet the following requirements*

Total impulse: 4000 N.s
Minimum impulse bit: $5x10^{-3}$ N.s

Further limiting factors are:

Thruster and tank temperature: 20 °C
Thruster gas: Nitrogen
exit nozzle diameter: 1mm
Valve response time: 20 ms
Maximum tank pressure: 300 atm ≈ $3x10^{7}$ N/m^2

What is the thrust level, thruster pressure, propellant consumption, and tank volume?

$$T = I_{\min} / t_s = 5x10^{-3} / 2x10^{-2} = 0.25 \;\; [N]$$

$$p_c = \frac{T}{A_t} = \frac{0.25}{3.14 * 10^{-6} / 4} = 3.18x10^5 \;\; [N/m^2] \cong 3.18 \;\; [atm]$$

$$I_{sp} \propto \sqrt{\frac{2\kappa}{\kappa - 1} \frac{RT_c}{M}} = \sqrt{\frac{2 * 1.4 * 8.317 * (273 + 20)}{(1.4 - 1) * 28 / 1000}} = 780 \;\; [m/s]$$

$$m_p = \frac{I}{I_{sp}} = \frac{4000}{780} = 5.13 \;\; [kg]$$

$$V_t = \frac{m_p R T_t}{10^{-3} M p_t} = \frac{5.13 * 8.3144 * 293}{10^{-3} * 28 * 3 * 10^7} = 0.0146 \;\; [m^3] = 14.6 \;\; [l]$$

cont'd

Space Propulsion

Table: Characteristics of some specific cold gas thrusters; Nitrogen propellant

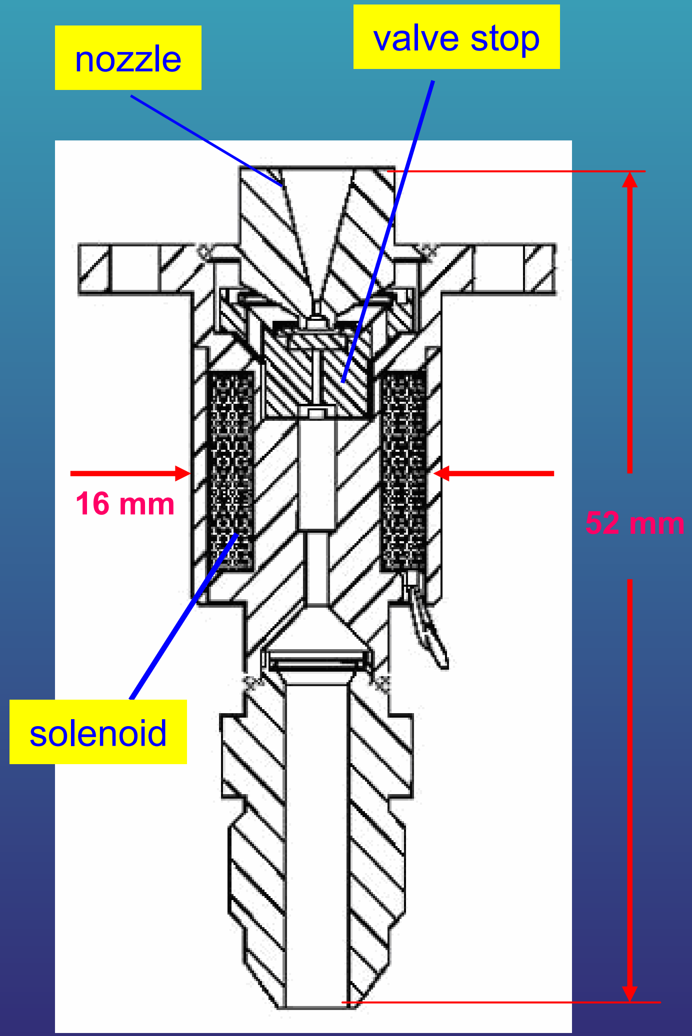
Engine	Manufacturer	Vacuum Thrust [N]	Vacuum Specific impulse [s]	Cycle life [Cycles]	Engine mass [kg]	Inlet pressure [bar]	Input power [Watt]	Voltage range [volt]	Envelope [mm] (LxD)
CGT1	DASA	0.02	67		0.120	7.0			64 (L)
	Sterer	1	68	250,000	0.174	3.5	5-6	24-32	66 x 31
58-102	Moog	1.11		10,000	0.015	8.8-6.3	30	24-32	24.7 x 14.5
58-112	Moog	1.11		10,000	0.015	7.4-4.9	30	24-32	24.7 x 14.5
58-103	Moog	5.55		10,000	0.015	8.8-6.3	30	24-32	24.7 x 14.5
50-673	Moog	44.5		5,000	0.231	10.5-4.9	6-12	24-32	86.6 x 79.7 x 64.2
58-126	Moog	266		10,000	0.181	10.5-4.9	30	24-32	70 x 63

cont'd

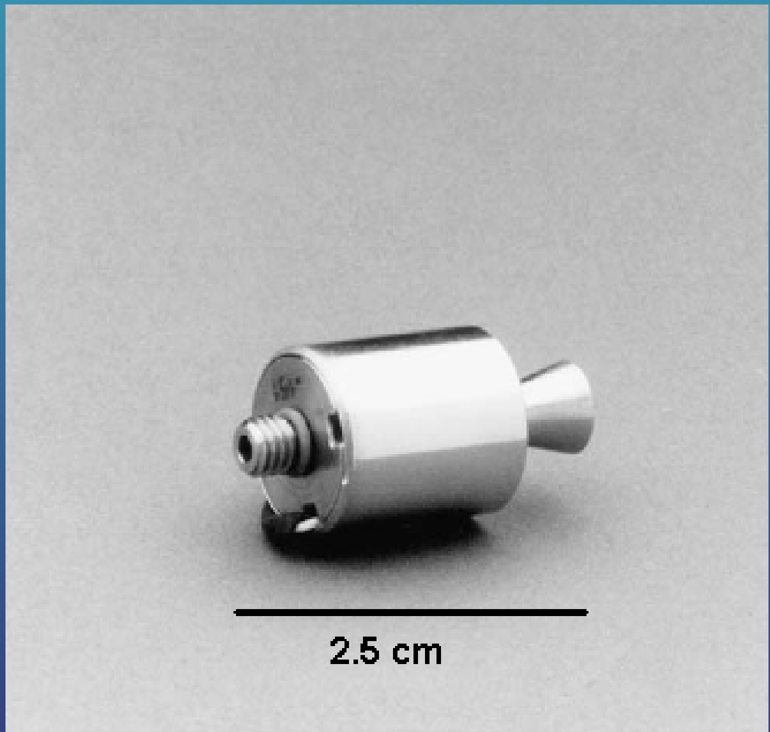
Space Propulsion

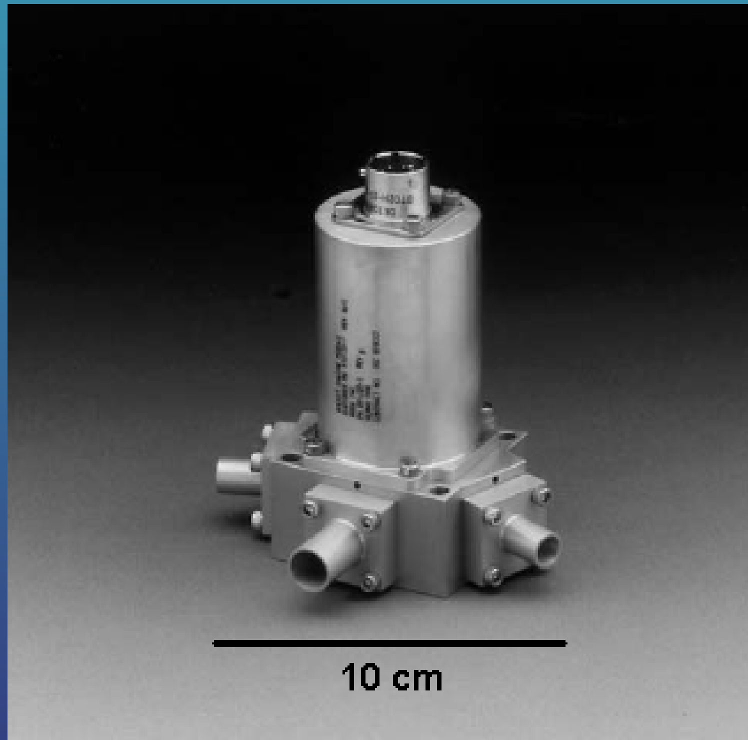
performance characteristics

operational temperature	-35°C to +65°C	
operating pressure	2.5 bar	
vacuum thrust	10 – 40 mN (± 5%)	
power consumption	< 3.5 W (pull-in) < 0.7 W (holding)	
response time	< 4.0 ms (opening and closing)	
mass	< 75 g	
cycle life	>2,000,000	
operating fluid	GN2, Xe	



Space Propulsion





single cold gas thrusters (left) and thruster triad (right) from MOOG / USA

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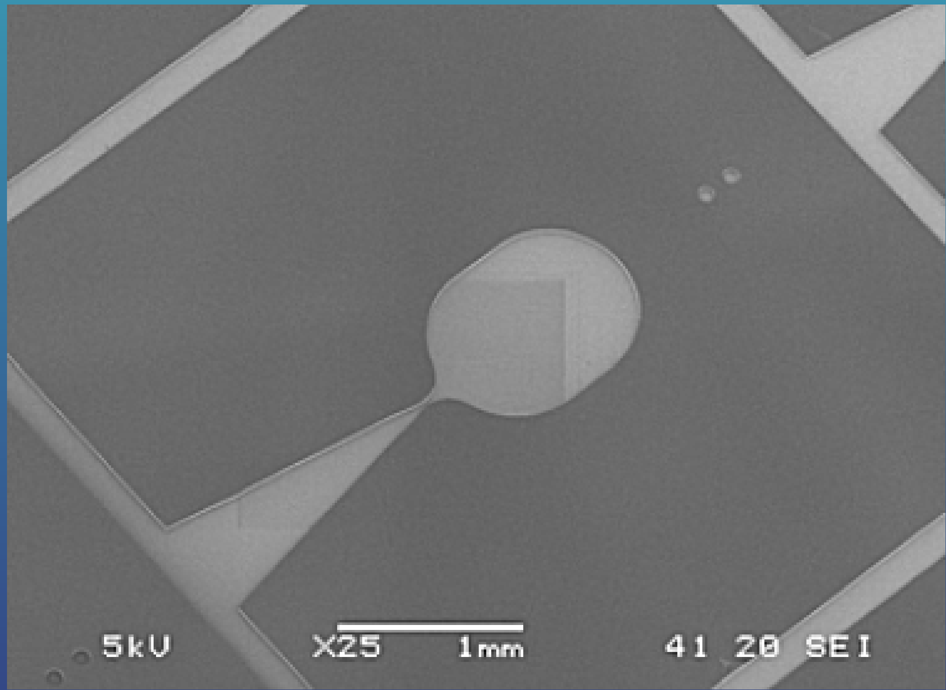
Space Propulsion



miniaturisation of cold gas thrusters using silicon technology

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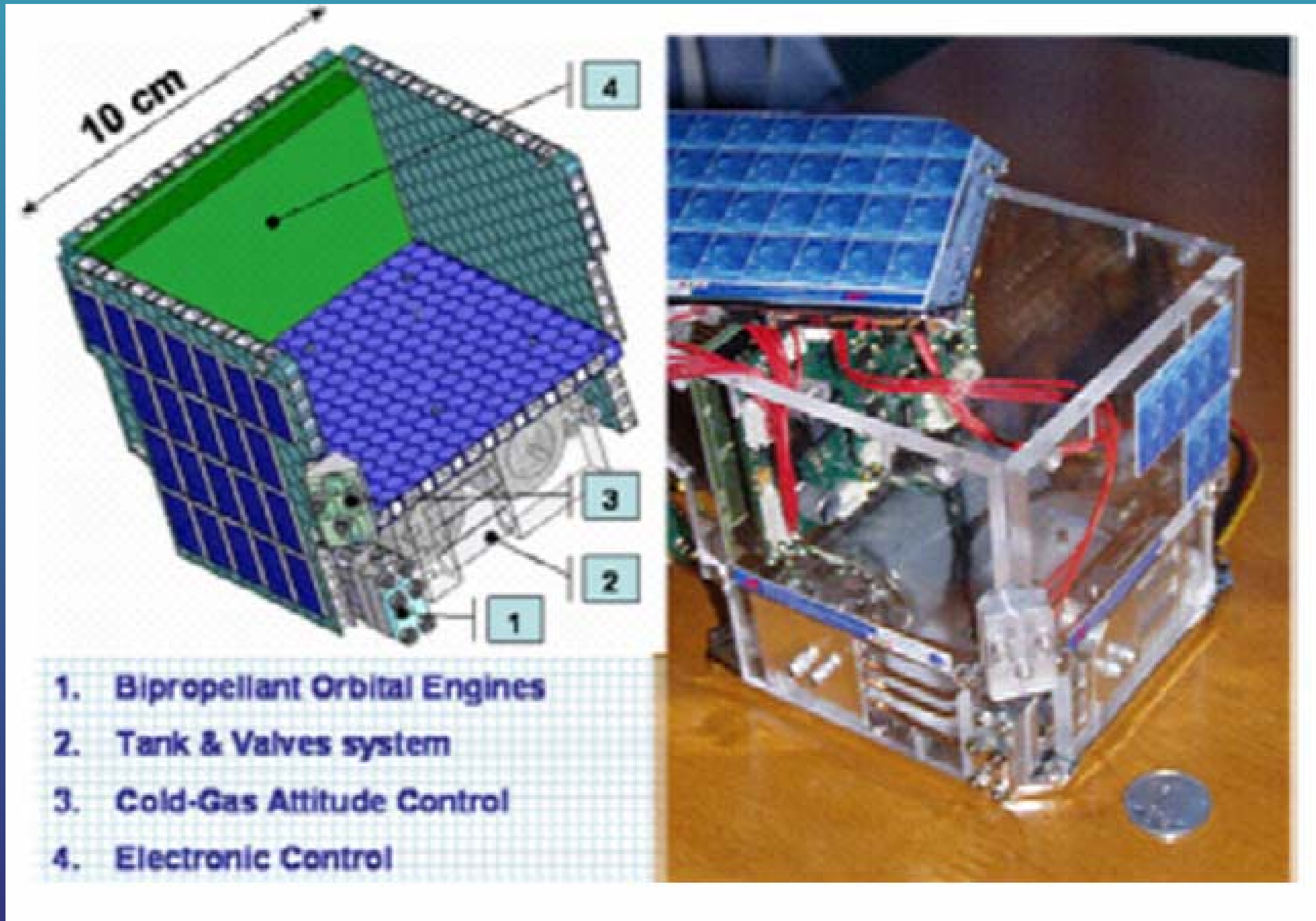
Space Propulsion



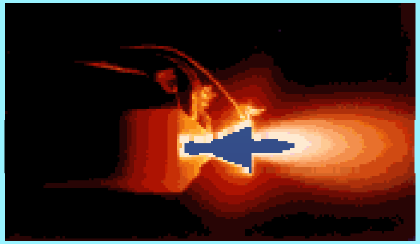
CUBESAT with micropropulsion system; 3D model (left)and functional Mock-Up (right)

Electric Propulsion Overview

ELECTRIC PROPULSION

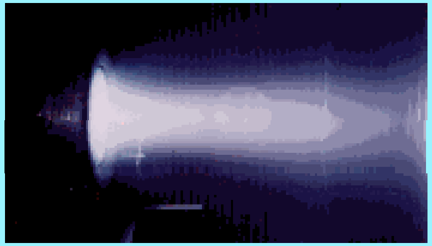
THREE CLASSES OF CONCEPTS

ELECTROTHERMAL
- GAS HEATED VIA RESISTANCE ELEMENT OR ARC AND EXPANDED THROUGH NOZZLE



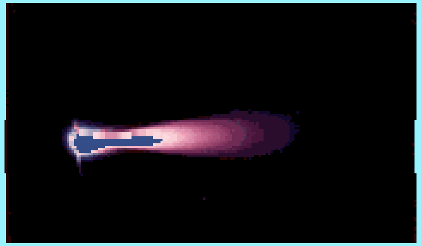
- RESISTOJETS
- ARCJETS

ELECTROSTATIC
- IONS ELECTROSTATICALLY ACCELERATED



- ION
- HALL
- FEEP
- colloid

ELECTROMAGNETIC
- PLASMA ACCELERATED INTERACTION OF CURRENT AND MAGNETIC FIELD



- PPT
- MPD

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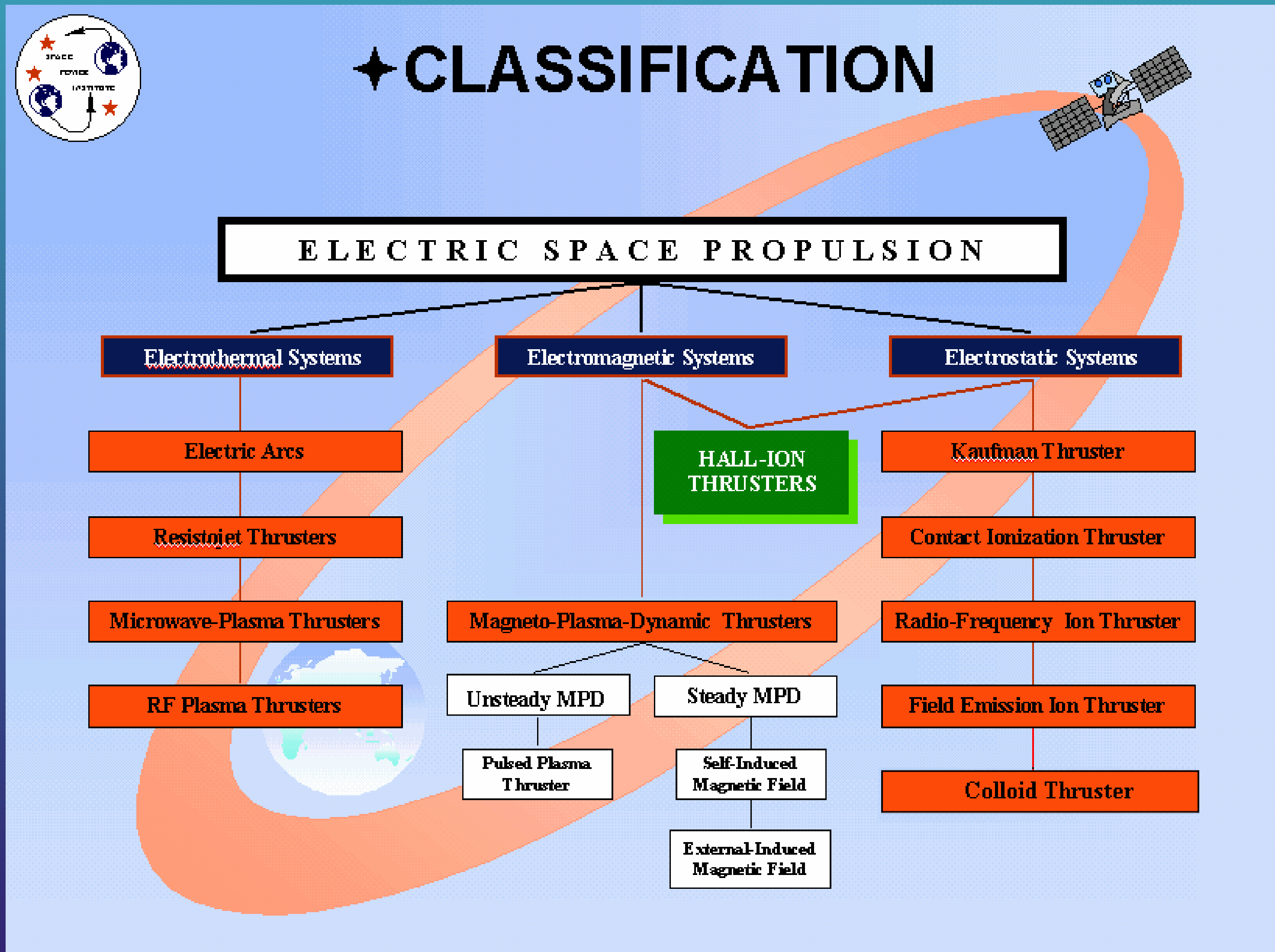
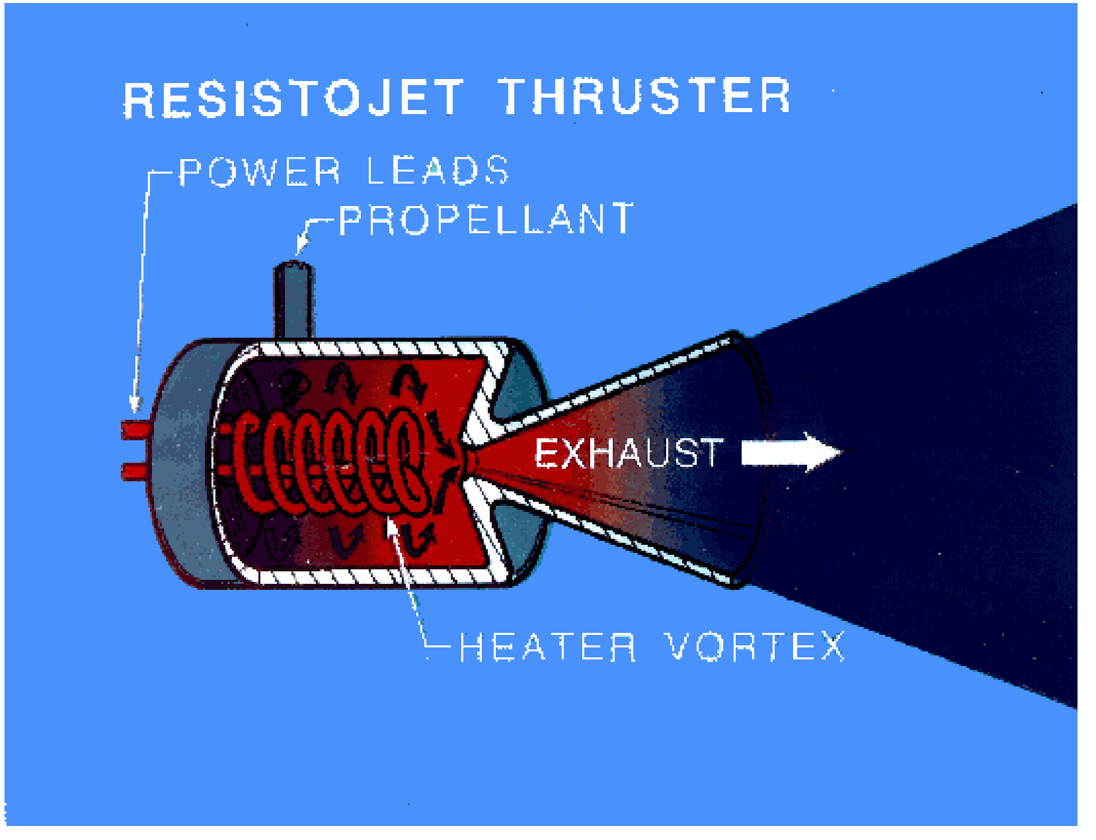
Space Propulsion



Space Propulsion

Electric Propulsion Overview

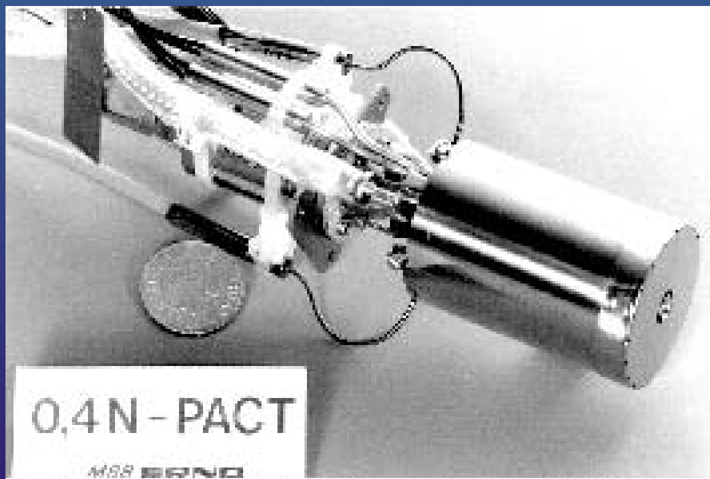
RESISTOJET THRUSTER ELEMENTS



Space Propulsion

Typical Reported Performance Parameters for Resistojets

Thruster	Propellant	Input power [W]	Isp [s]	Thrust [mN]	Operat. Lifetime [h]	Manufacturer	status	Total impulse [kN]	Thruster mass [kg]	Thruster size [cm]
MR501 resistojet	Hydrazine	350 – 510	280 - 304	180 – 330	>389 (500000 pulses)	Primex	Several flown	311	0,816	20 L x 10 dia
MR502A Resistojet	Hydrazine	610 – 885	299	360-500	>370	Primex	Several flown	525	0,871	19,8 L x8,8 dia
HPEHT Resistojet	Hydrazine		295	220-490		TRW	Several flown	160		

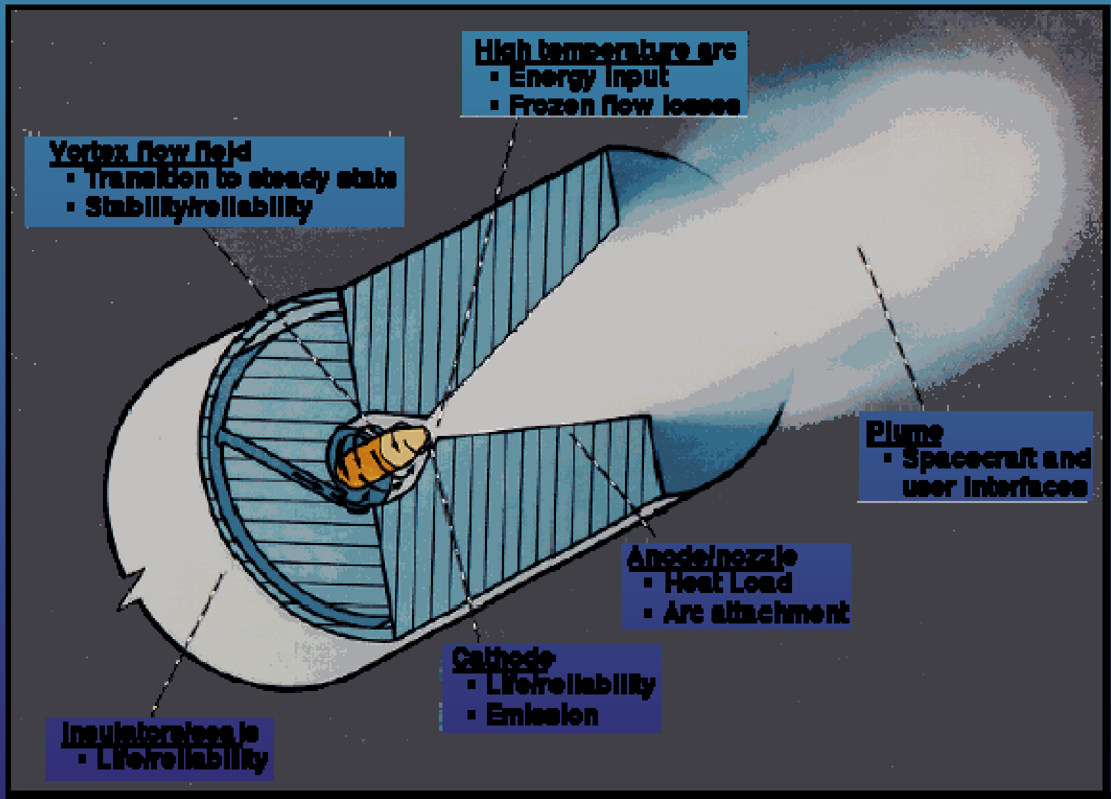


Power Augmented Catalytic Thruster (courtesy MBB-Erno)

Space Propulsion

Electric Propulsion Overview

ArcJet Elements



Space Propulsion

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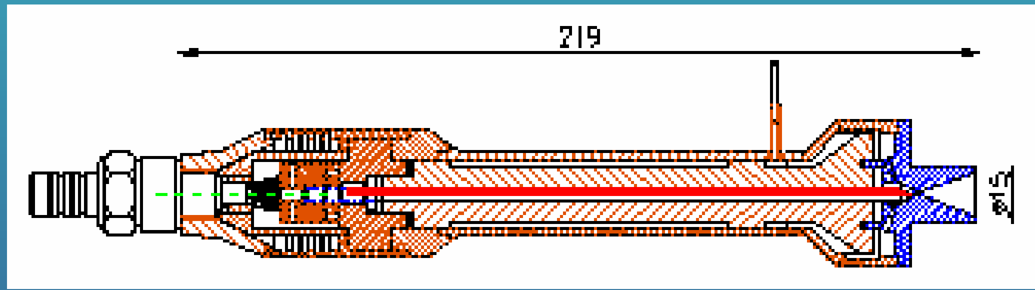
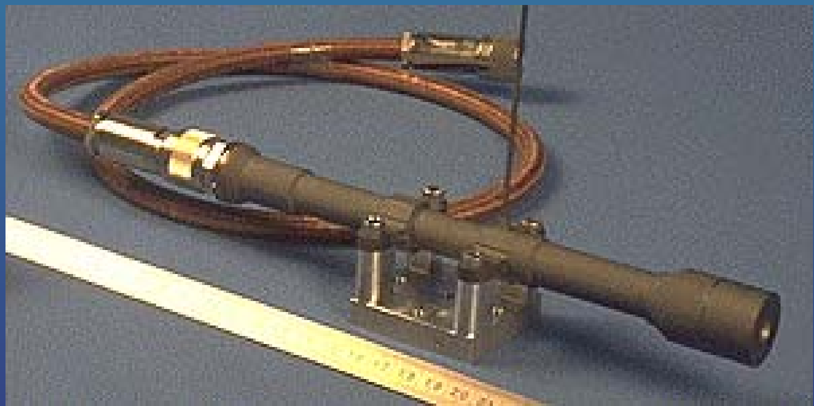
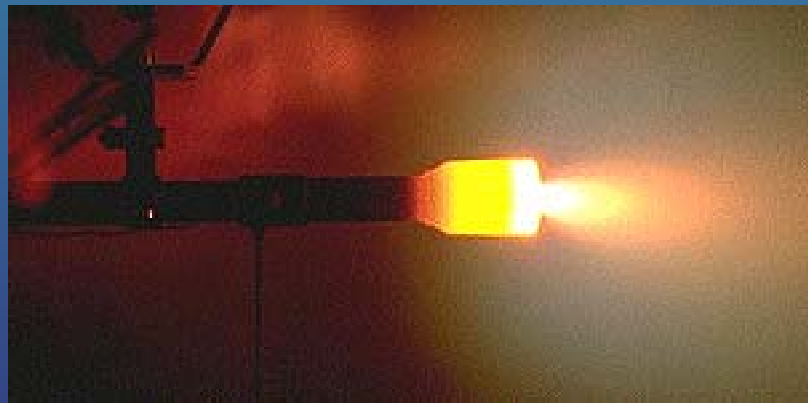
Stuttgart



ARTUS
arcjet





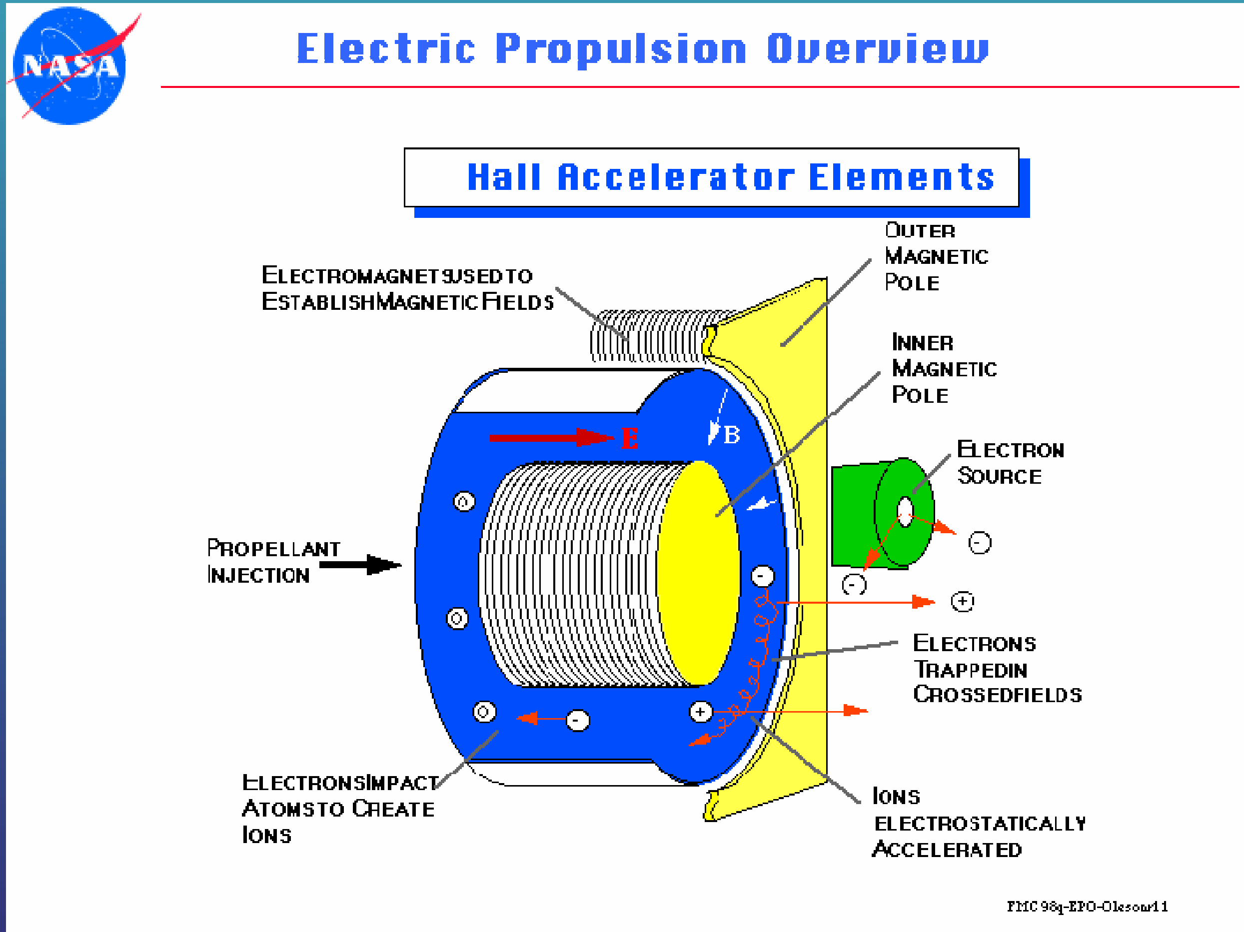
power input:	1 kW
propellant:	hydrazine, ammonia
thrust:	0.1 – 1 N
Isp:	< 7000 m/s

Space Propulsion

Typical Reported Performance Parameters for Arcjets

Thruster	Propellant	Input power [W]	Isp [s]	Thrust [mN]	Thrust efficiency [%]	Operat. Lifetime [h]	Manufacturer	status	Total impulse [kN]	Thruster mass [kg]	PCU mass [kg]	Thruster size [cm]
MR506 Arcjet	Hydrazine	1500	>502	200-231	>30	>830 (1hr on 1/2hr off)	Primex	Flown	>634	1,36	4,1	23 x 11 x 11
MR-509 Arcjet	Hydrazine	1500	>502	209-249	>31	>1050 (1hr on 1/2hr off)	Primex	Several flown	>557	1,47	4,1	24 x 13 x 9
MR510 Arcjet	Hydrazine	2.170	>580	213-245	>31	>1050 (1hr on 1/2hr off)	Primex	Flown	>812	1,431	15,77 powers 4 thrusters	24 x 13 x 9
ATOS Arcjet	Ammonia	748 [1]	480	114	36	1010 (1hr on, 1hr off)	IRS (D)	Qual flight 97/98	>400	0,480	2,5	

Space Propulsion



Hall thruster

Space Propulsion

- originally developed in the USSR (1950ies) as ion source for particle accelerators
- from the 1990ies flown on many Russian satellites for station keeping
- end of 1990ies tested by ESA and NASA
- 2003 flown on SMART – 1 mission of ESA (common development of SNECMA / F and FAKEL / Ru)

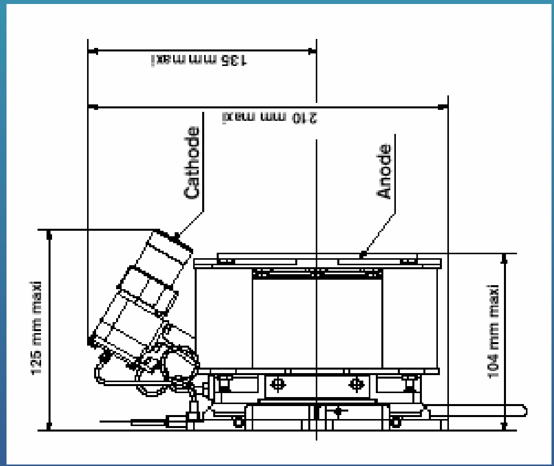
Space Propulsion

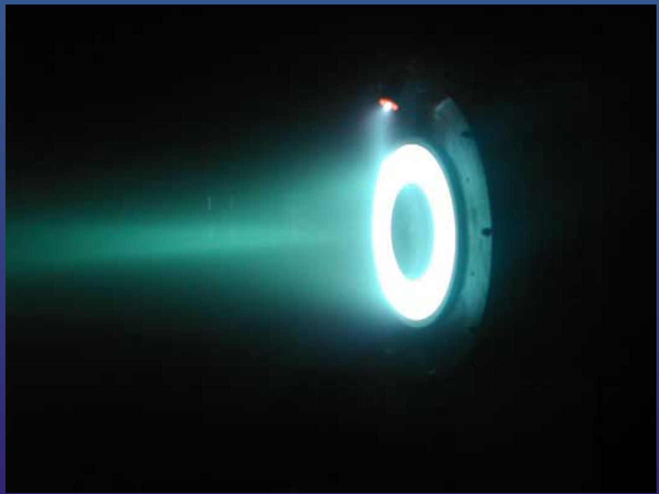
Typical Reported Performance Parameters for Hall thrusters

Thruster	Propellant	Input power [W]	Isp [s]	Thrust [mN]	Thrust efficiency [%]	Operat. Lifetime [h]	Manufacturer	status	Total impulse [kN]	Thruster mass [kg]	PCU mass [kg]	Thruster size [cm]
SPT100 HET	Xe	1350	1600	83	45	>7424 (50 min on, 30 min off)	Fakel (Ru)	Several flown	>2000	3,5	8	15 x 22 x 12,5
SPT70 HET	Xe	640 – 660	1550	40	48	9000 (est.)	Fakel (Ru)	Several flown		1,5		
T-100SPT HET	Xe	1350	1630	83	50	>8000, 3000 restarts (est.)	NIITP (Ru)	Ground tested in RHETT1		3	10 (est.)	23 x 10 x 13
D-55 TAL HET	Xe	1350, 1600	1600	80	60	>5000 (est.)	TsNIIMASH (Ru)	Exper. Flight 97				

Space Propulsion

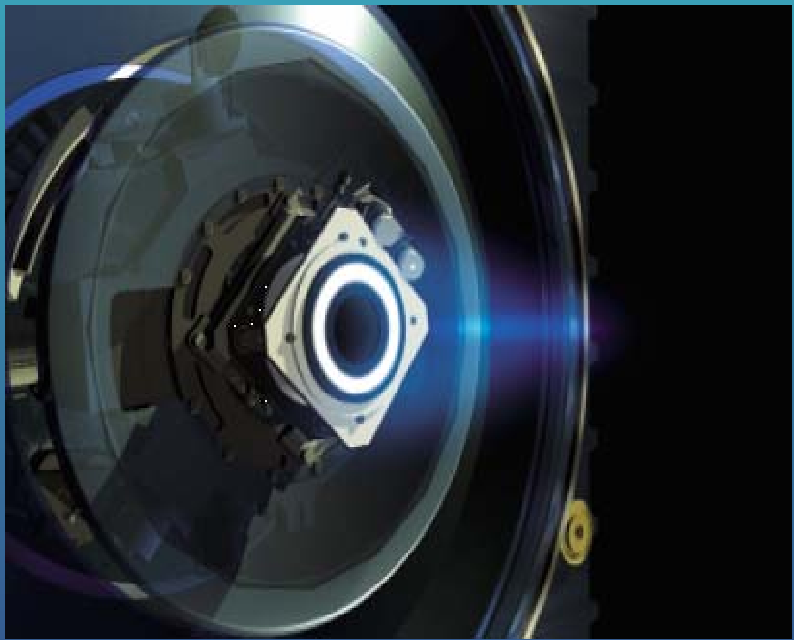
SNECMA PPS-1350 ion engine





CARACTÉRISTIQUES	
• Puissance nominale (W)	1 500
• Poussée (mN)	88
• Impulsion spécifique (s)	1 650
• Impulsion totale délivrée (N.s)	3.10^6
• Nombre de cycles	8 200
• Courant de décharge (A)	4,28
• Rendement (%)	55
• Tension d'alimentation (V)	350
• Pression d'alimentation xénon (bar)	2,50 à 2,80
• Classe de xénon	haute pureté
• Masse (2 systèmes de contrôle de débit Xe inclus) (kg)	5,30

Space Propulsion



SNECMA PPS-1350 ion engine on SMART 1

Comparison of propulsion technologies

	Chemical		Electric
	Small monopropellant thruster	Fregat Main Engine (S5.92M)	SMART-1 Hall Effect Thruster (PPS-1350)
Propellant	Hydrazine	Nitrogen tetroxide / Unsymmetrical dimethyl hydrazine	Xenon
Specific Impulse (s)	200	320	1640
Thrust (N)	1	1.96×10^4	6.80×10^{-2}
Thrust time (s)	1.66×10^5	877	1.80×10^7
Thrust time (h)	46	0.24	5000
Propellant consumed (kg)	52	5350	80
Total Impulse (Ns)	1.1×10^5	1.72×10^7	1.2×10^6

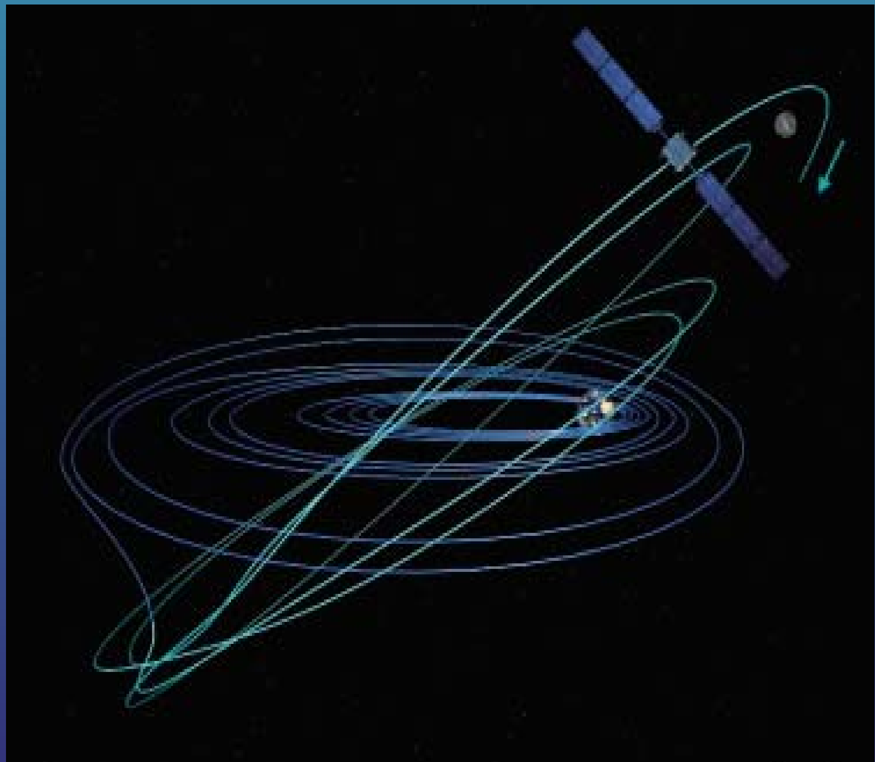
Fregat produces ~ 14 times the total impulse of SMART-1's engine, but uses nearly 70 times more propellant mass to do so. The hydrazine thruster produces less than a tenth as much total impulse while using 65% of the propellant mass.

Electric Propulsion System operations history	
Number of Pulses	844
Total number of hours fired (h)	4958.3
Xenon at launch (kg)	82.5
Remaining Xenon (g)	280
Remaining useable Xenon (g)	~ 60

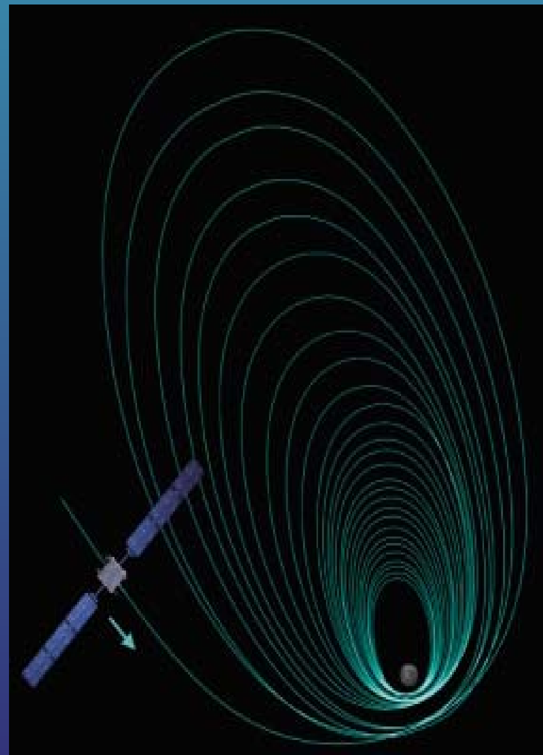
Space Propulsion

esa SMART-1

European Space Agency



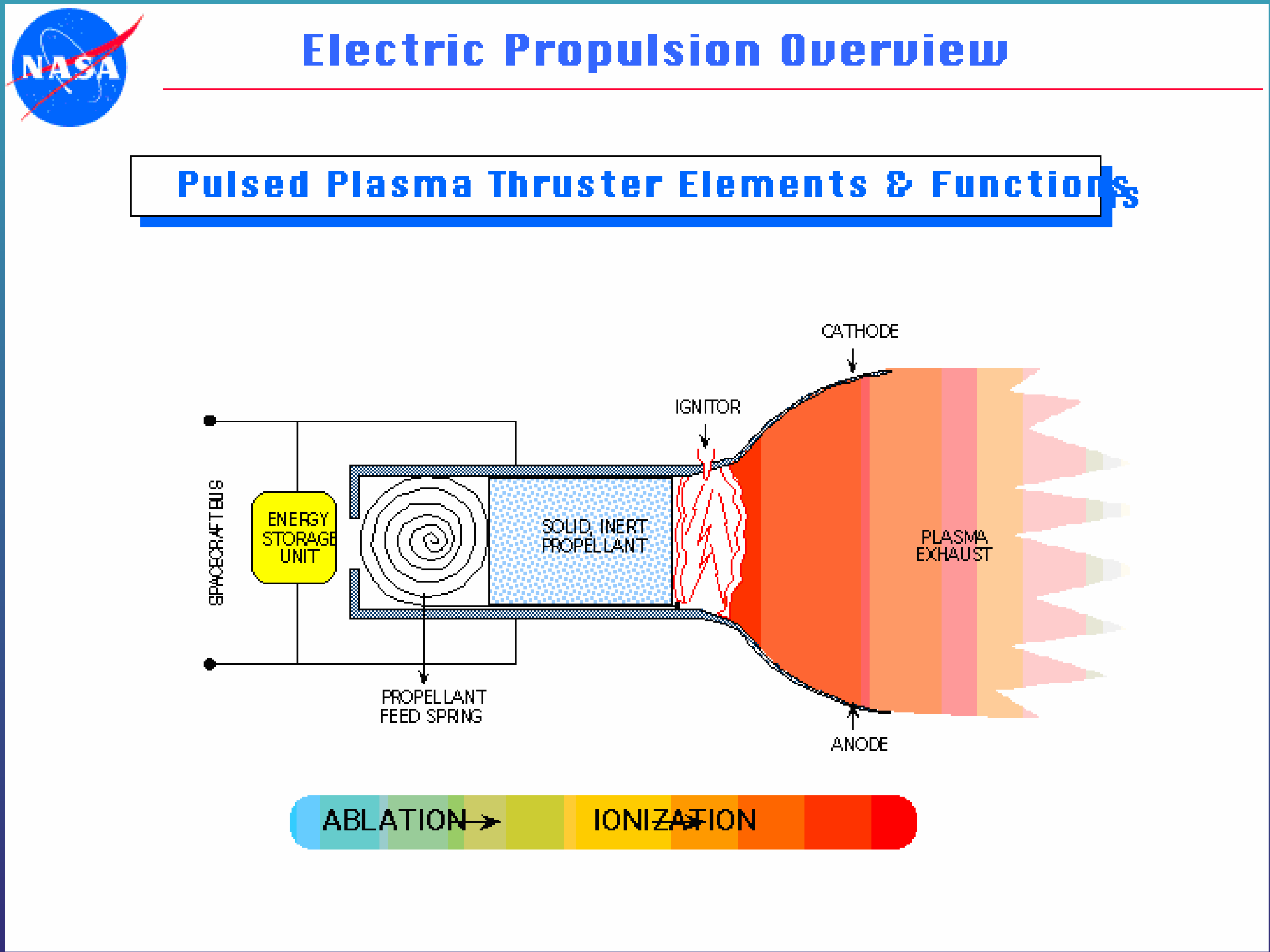
spiraling up and lunar plane insertion



spiraling down to moon

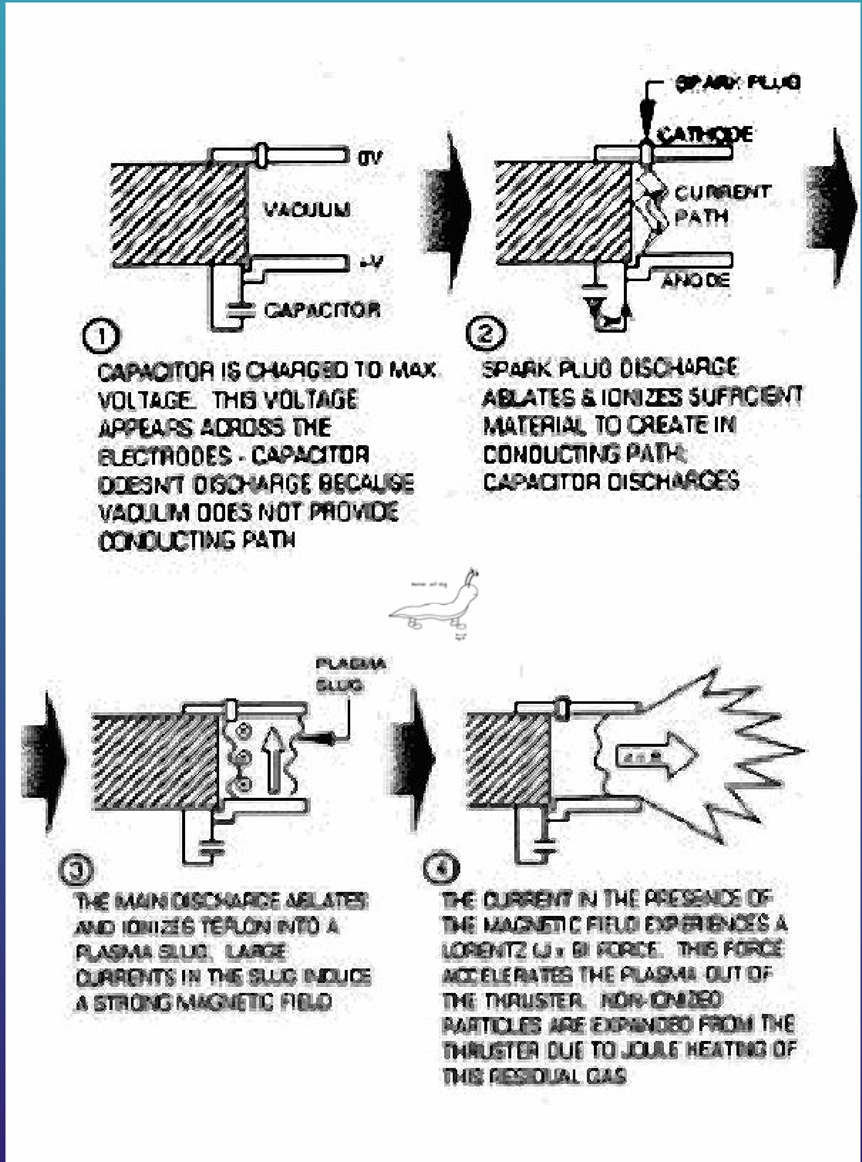
Launch:	Sept. 27 2003
Moon impact	Sept. 3, 2006

Space Propulsion



pulsed plasma thruster (PPT)

Space Propulsion



Space Propulsion

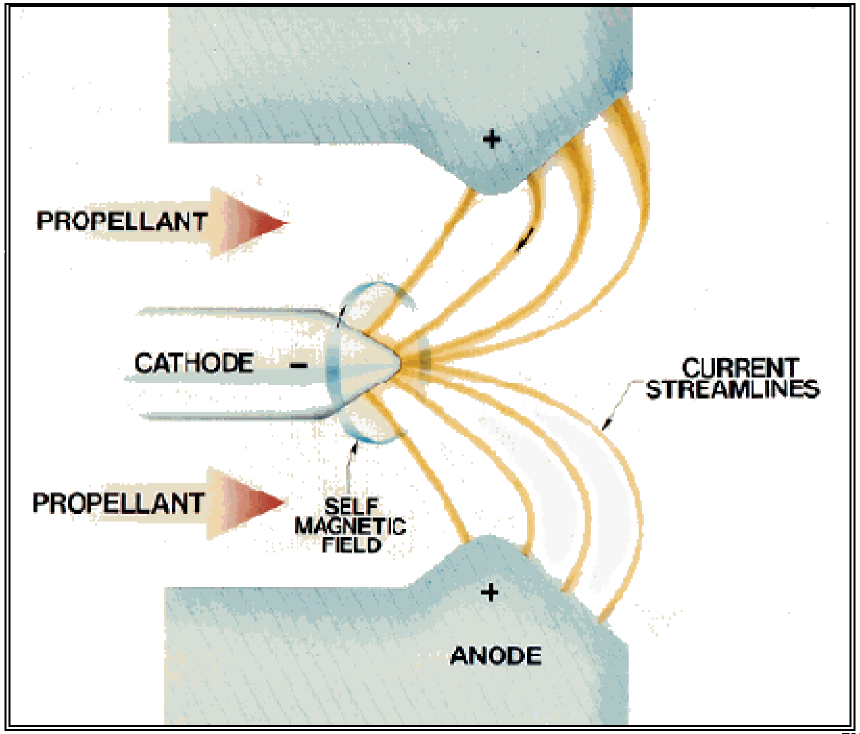
PRIMEX PPT – thruster flown on EO-1 (2000)

thrust	860 μN
I_{sp}	1.37×10^4 m/s
power consumption	70 W

Space Propulsion

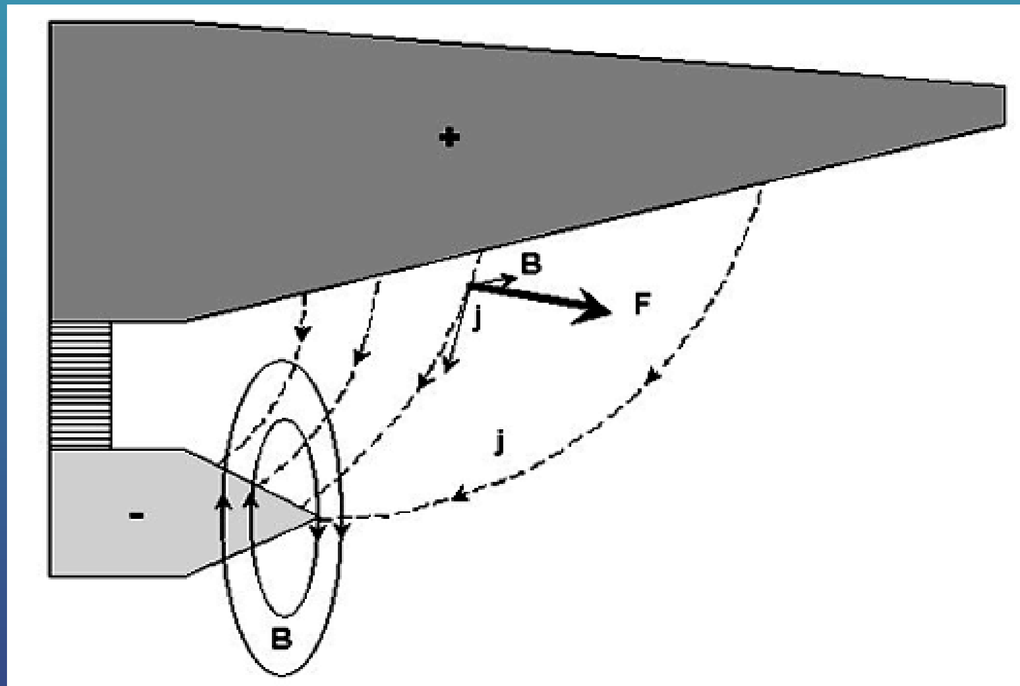
Electric Propulsion Overview

Magnetoplasmadynamic Thruster Elements



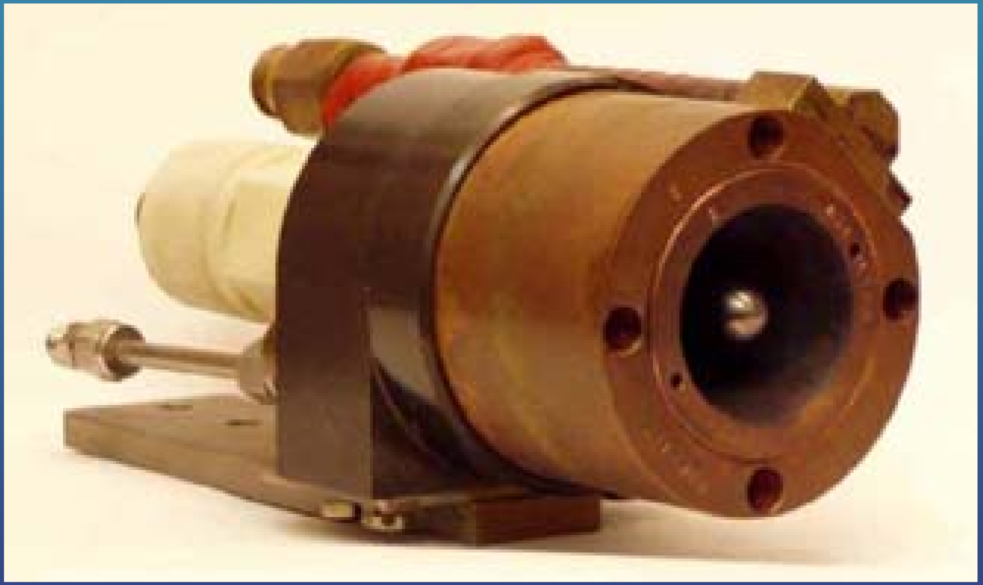
PMC 98q-EPO-Olesonr17

Space Propulsion



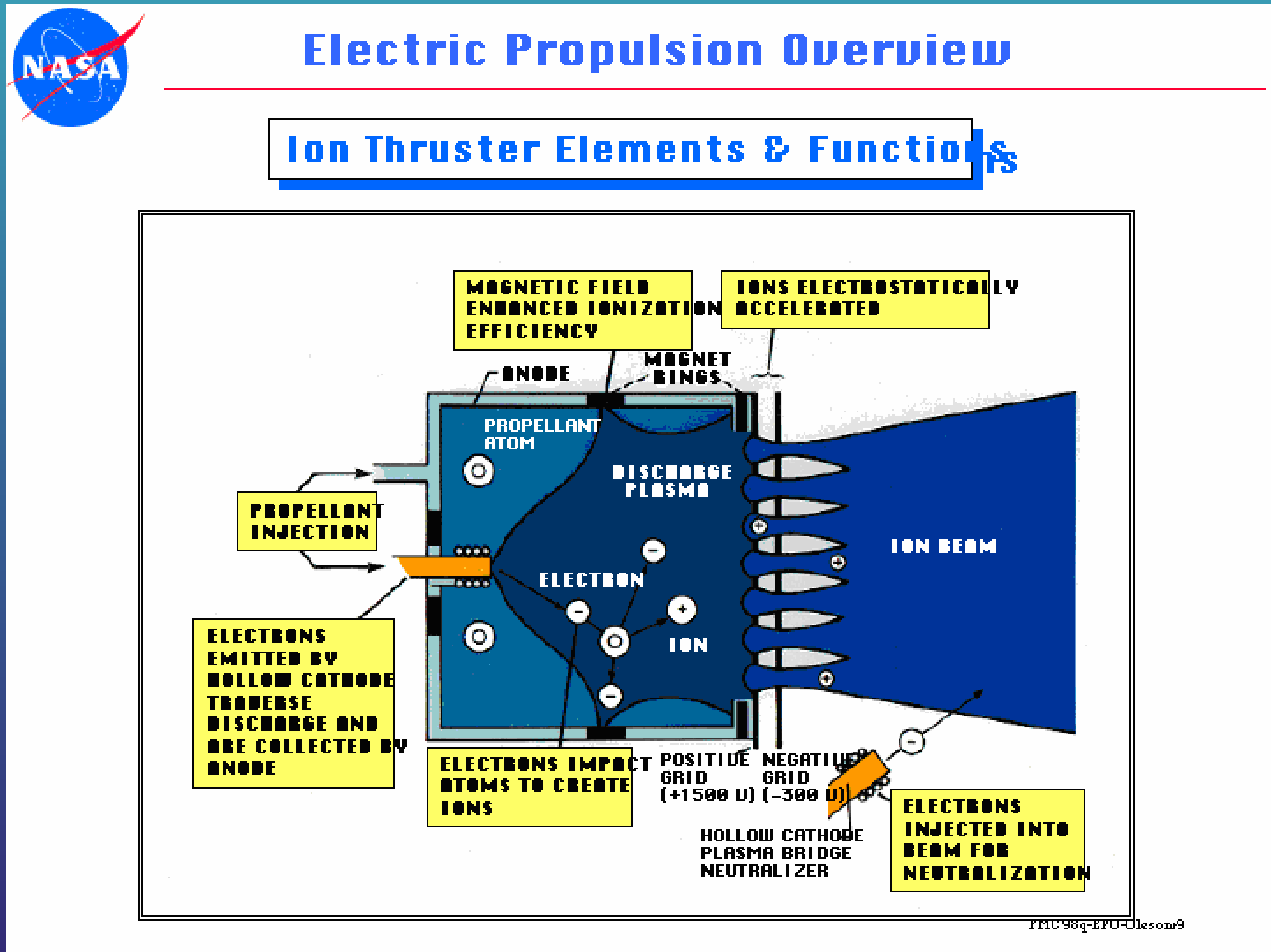
generation of thrust F by self – magnetic field of discharge current j

Space Propulsion



200-kilowatt MPD thruster, NASA - Glenn

Space Propulsion



Space Propulsion

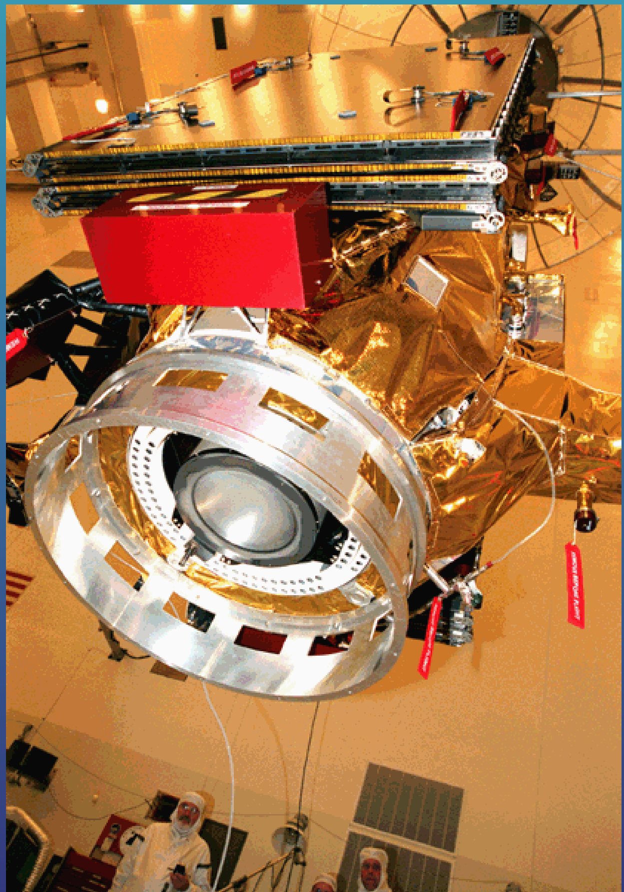
DS1 was the first spacecraft to use ion propulsion as the primary propulsion system. It is one of the 12 advanced technologies that was validated by DS1 during flight.

- **NSTAR ion thruster specs**

- grid diameter 30 cm
- input power 0.5 – 2.3 kW
- thrust 19 – 92 mN
- I_{sp} 1.9 – 3.1x10^4 m/s
- thruster mass 8.2 kg
- PPU mass 14.8 kg

- **DS1 – mission**
- propellant gas Xe
- propellant mass 82 kg
- operation time 16 246 h
- propellant used 72 kg
- mission period 1998 - 2001

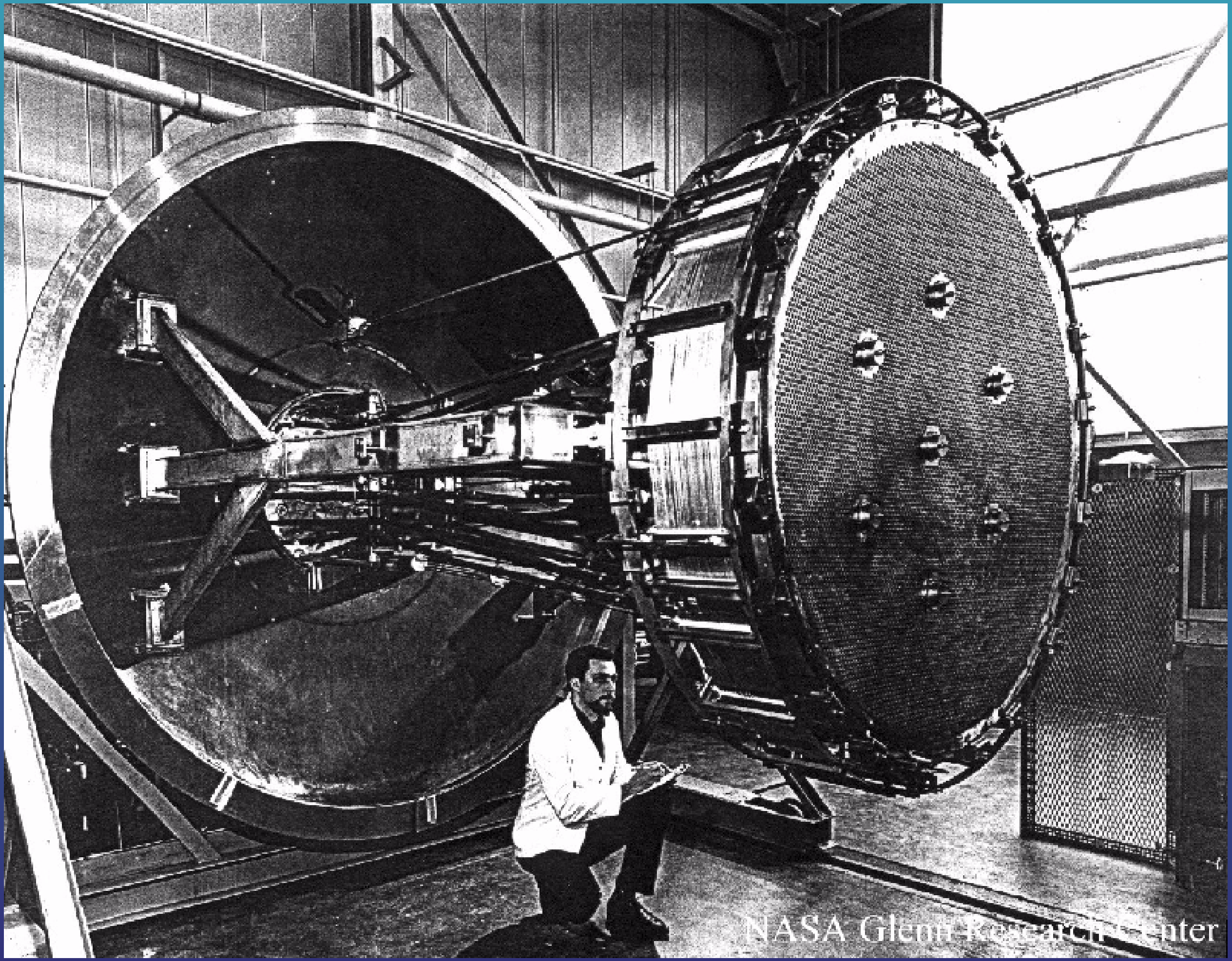
Space Propulsion



DS1 spacecraft
NSTAR ion engine from
HUGHES

Comet BORELLY, imaged by DS1
Actual size ~ 10 x 3 x 3 km

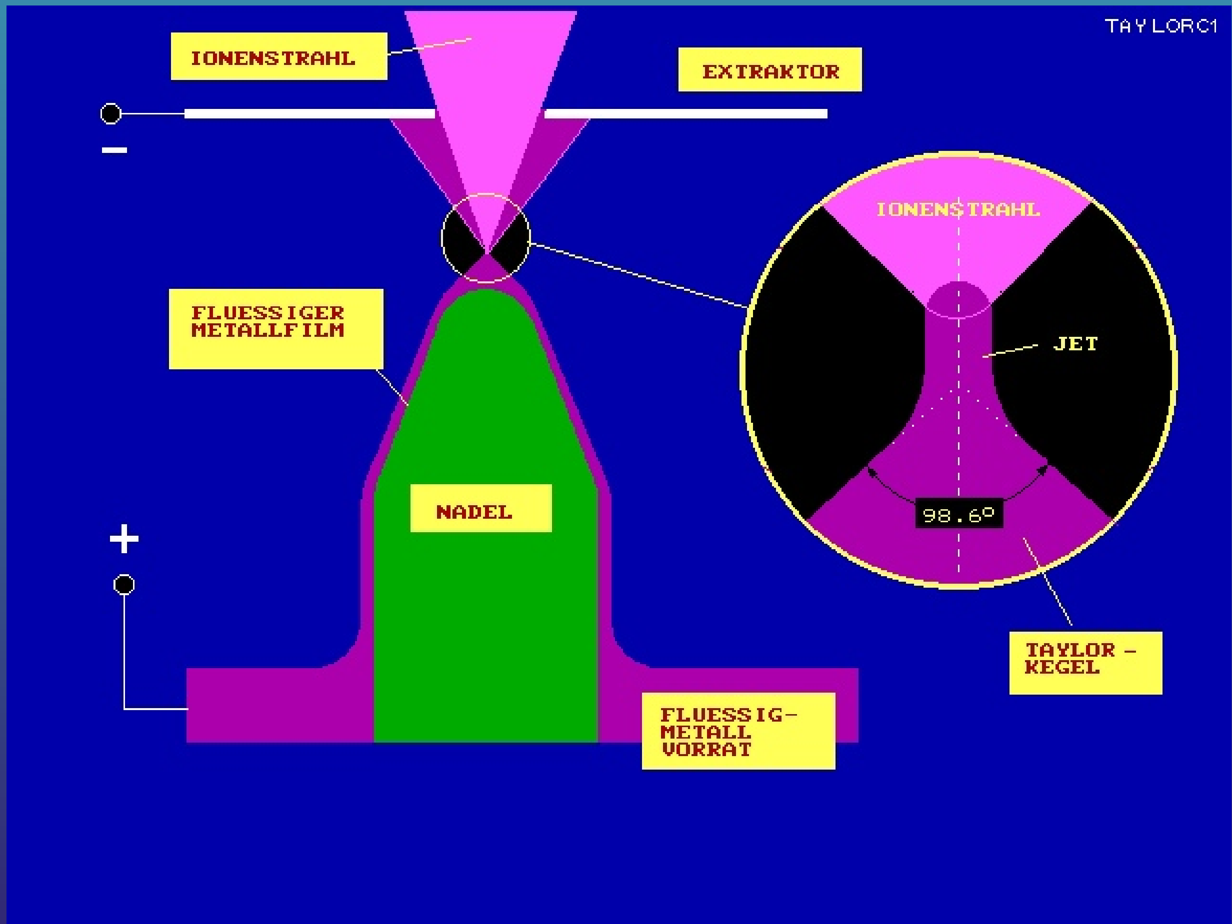
Space Propulsion



largest ion thruster ever tested (NASA GLENN Res. Ctr.); diameter 1.5 m; around 1963

Space Propulsion

Field Emission Electric Propulsion (FEEP)



Space Propulsion

mechanism of field evaporation

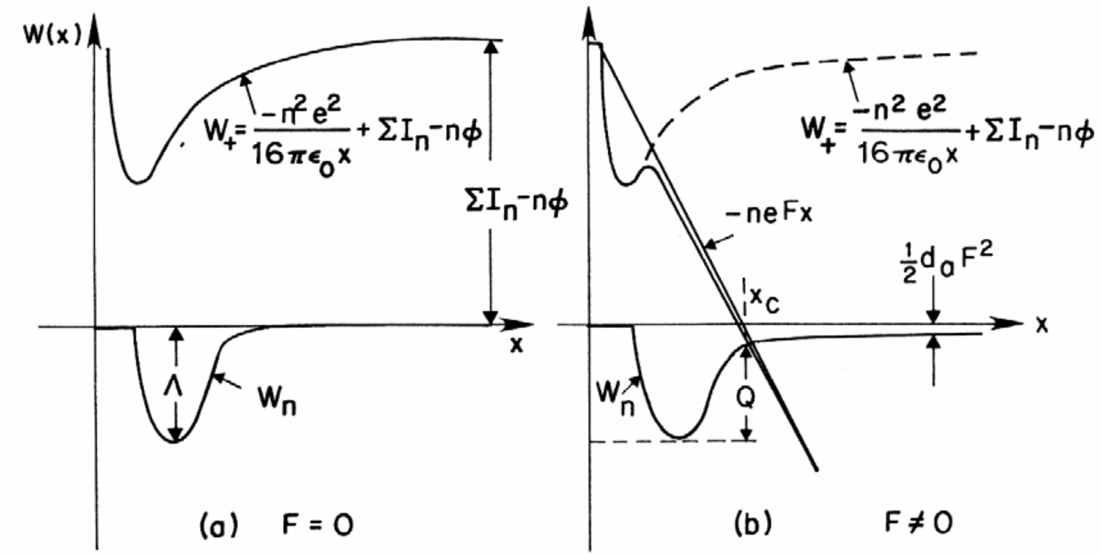
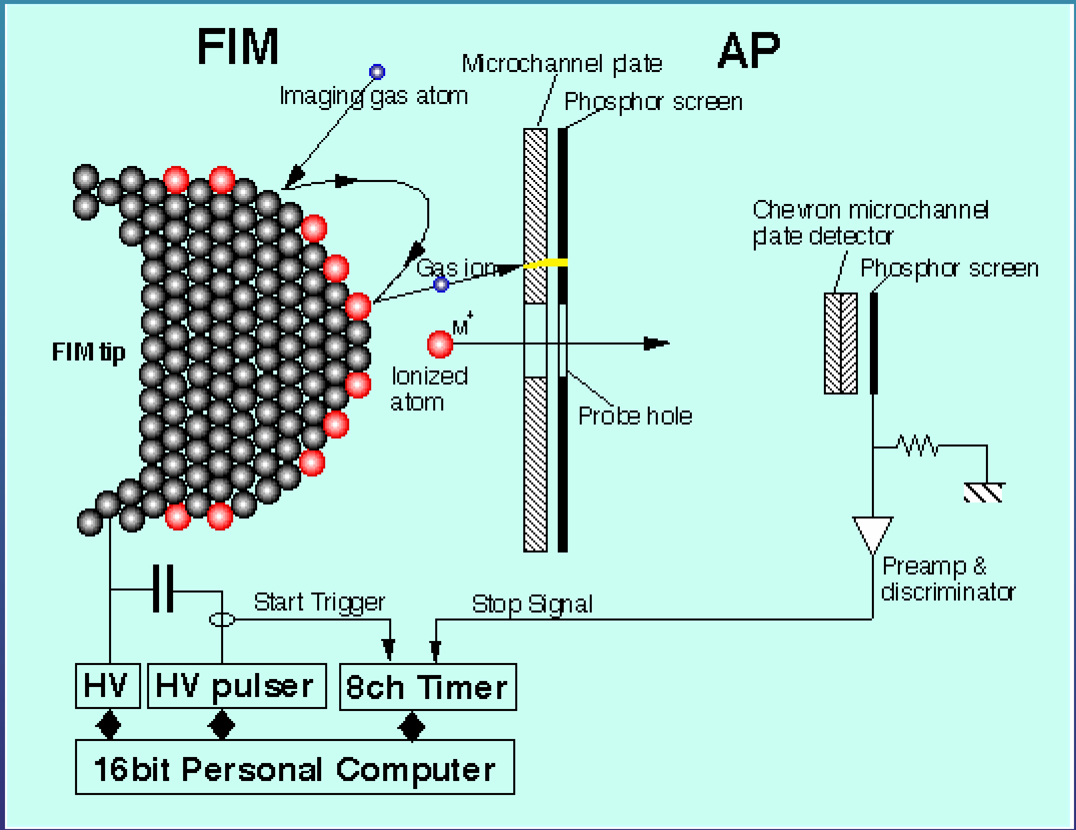
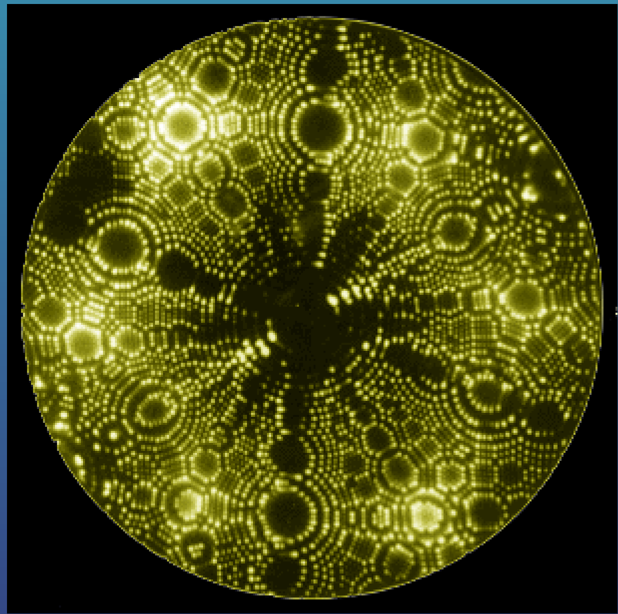


Fig. A2 Potential energy variation with distance from metal surface for
(a) W_n – atom in zero applied field (F = 0)
W_+^n – ionised atom following electron tunnelling into the metal (F = 0)
(b) As (a), but in uniform applied field ($F \neq 0$)

Space Propulsion

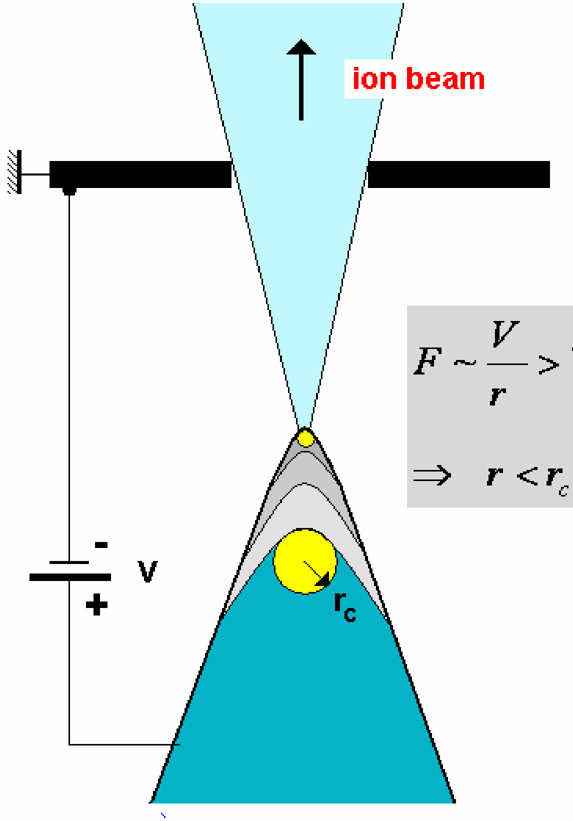
field evaporation from solids





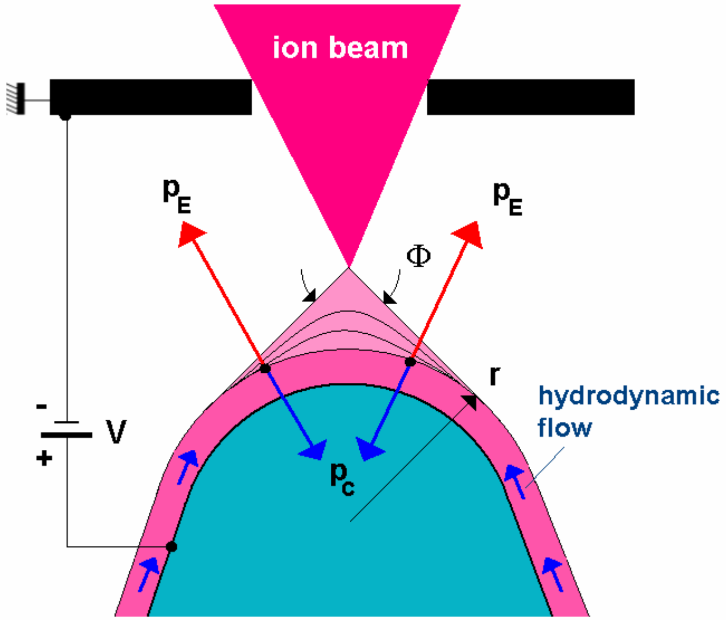
- tip radius < 100 nm
- tip voltage > 8 kV

Space Propulsion



$$F \sim \frac{V}{r} > 10 \quad V/nm$$

$$\Rightarrow \quad r < r_c$$



$$p_c = \frac{2\gamma}{r}$$ capillary pressure

$$p_E = -\frac{\varepsilon_0}{2} F_0^2 \sim -\left(\frac{V}{r}\right)^2$$ electric strain

$$if \quad p_E > p_c$$

$$\Rightarrow \quad \Phi = 98.6^0$$

TAYLOR cone

solid under strong field:
ion emission extinguishes when

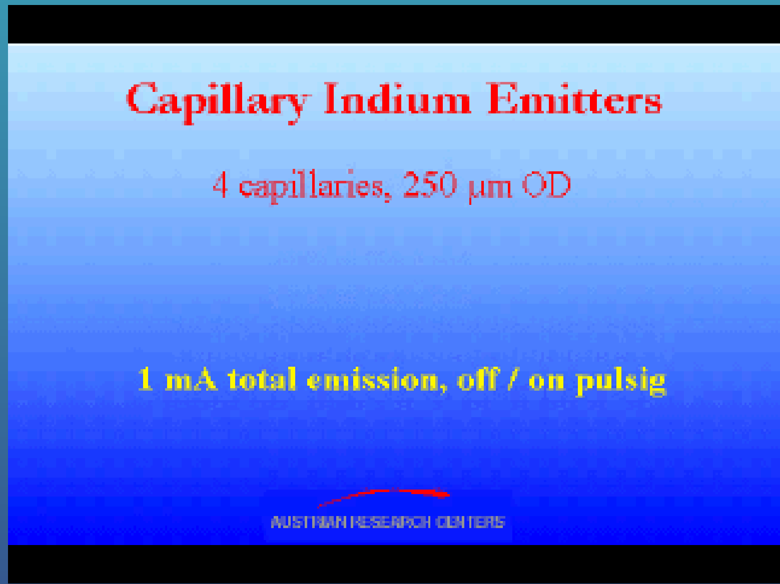
$r > r_c$

liquid under strong field:
continuous ion emission for

$$V > k\sqrt{2\gamma . r}$$

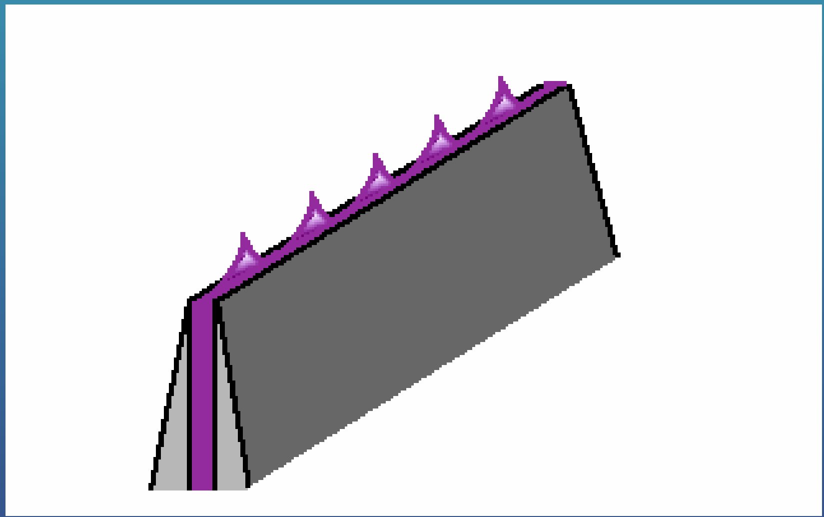
Space Propulsion

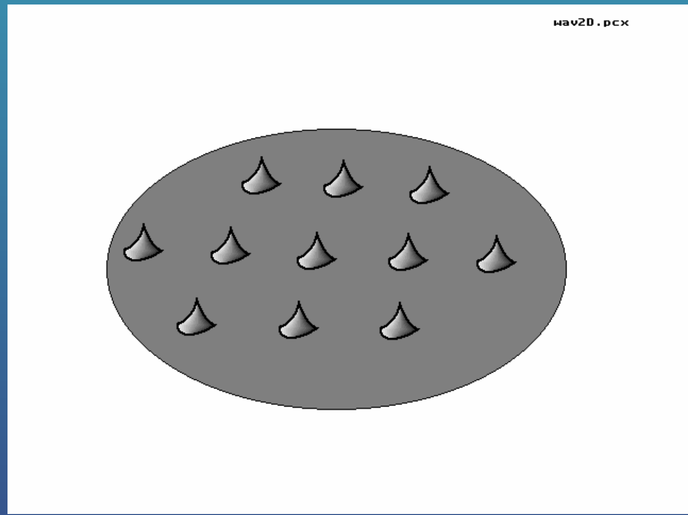
Liquid indium is contained in a capillary tube. Tube voltage is slowly increased. At low voltages, In is lifted from the tube, forming a rounded equilibrium surface. Above a critical voltage (~ 5 kV) the liquid surface jumps into a TAYLOR – cone shape and emission immediately starts. If voltage is further increased, emission intensity increases.



4 indium – filled capillaries are fed from the same HV – power supply. PS voltage is pulsed from 0 to ~ 8 kV. TAYLOR – cones are formed on all 4 capillaries and pulsed emission can be observed.

Space Propulsion



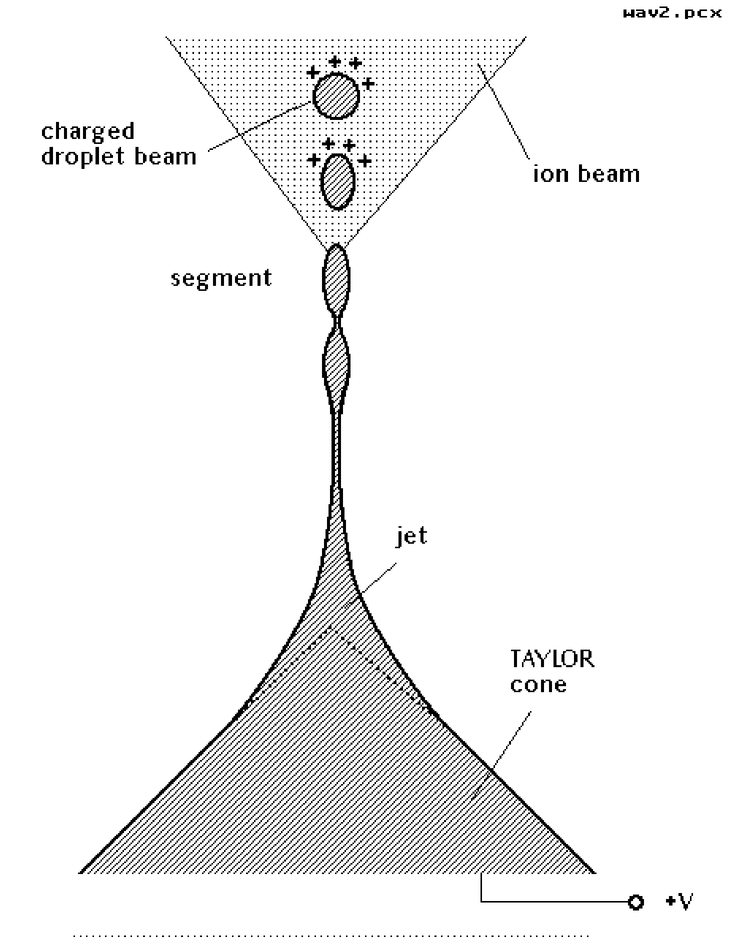


1 - dimensional

2 - dimensional

array of TAYLOR cones

Space Propulsion



- **low current**: quiescent jet with weak capillary oscillations; ion emission only
- **threshold current**: oscillations grow → separation of charged droplets from jet apex
- **high currents**: increasing droplet emission

droplets

- low contribution to thrust
- high contribution to mass loss

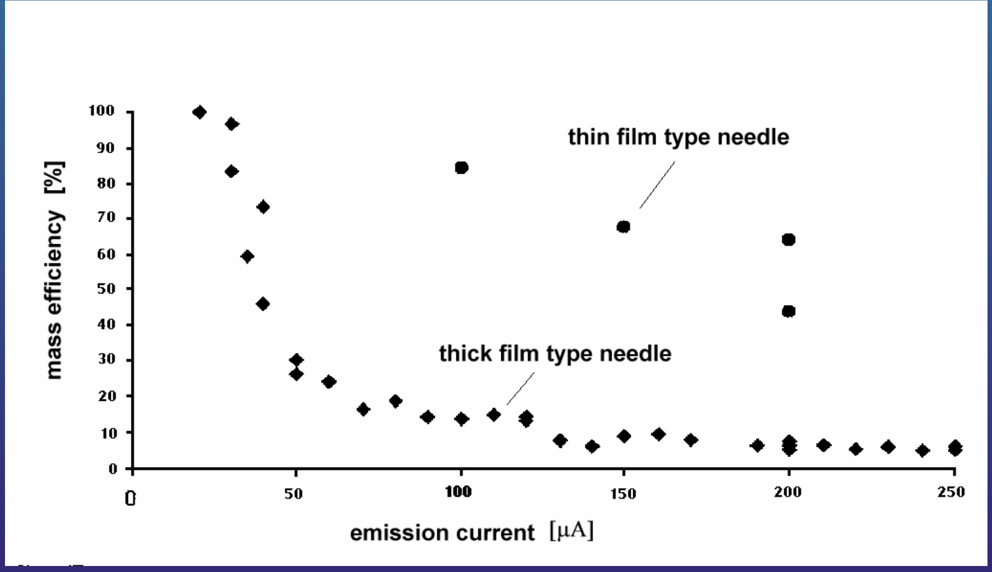
- emission of neutrals not allowed in the field evaporation process
- vapor pressure can be extremely low at melting point ($< 10^{-16}$ mbar for In, $< 10^{-6}$ for Cs)

→ mass loss only occurs via thrusting ions and nonthrusting droplets; neutral emission not supported by field evaporation

Space Propulsion

mass efficiency is that fraction of total mass loss which contributes to thrust, i.e. the ionic fraction

$$\eta = \frac{\dot{m}_i}{\dot{m}_{tot}} = \frac{\dot{m}_i}{\dot{m}_i + \dot{m}_{dr}} = \frac{1}{1 + \frac{\dot{m}_i}{\dot{m}_{dr}}}$$



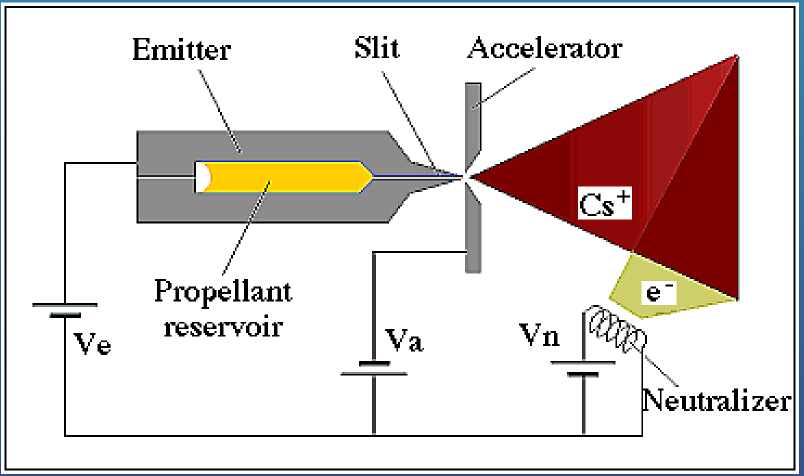
Space Propulsion

Comparative parameters for candidate FEEP propellant materials

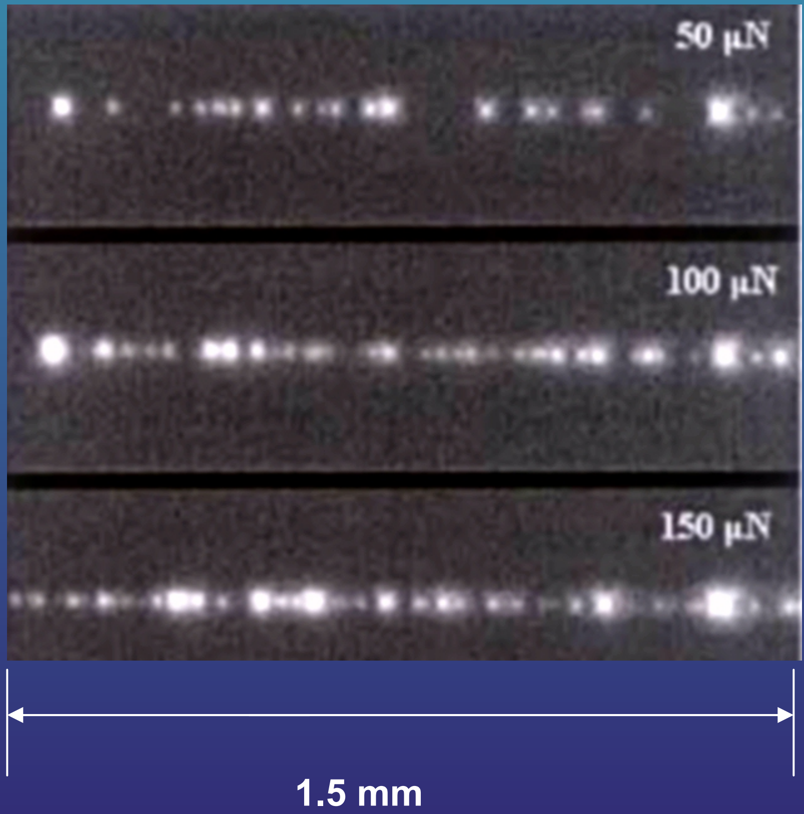
	In	Cs	Ga
mass number [amu]	114.8	132.9	69.7
density [g/cm^3]	7.31	1.9	5.91
melting point [^oC]	156.6	28.4	29.8
heat of vaporization [J/g]	2024	611	3984
vapor pressure at 150 ^oC [mbar]	$<10^{-16}$	0.15	$\approx 10^{-20}$
vapor pressure at melting point [mbar]	$\approx 10^{-16}$	$2x10^{-6}$	$\approx 10^{-24}$
surface tension at m.p.[J/m^2]	0.556	0.70	0.718
work function [eV]	4.12	2.14	4.2
1^{st} ionisation energy [eV]	5.78	3.89	5.99

Space Propulsion

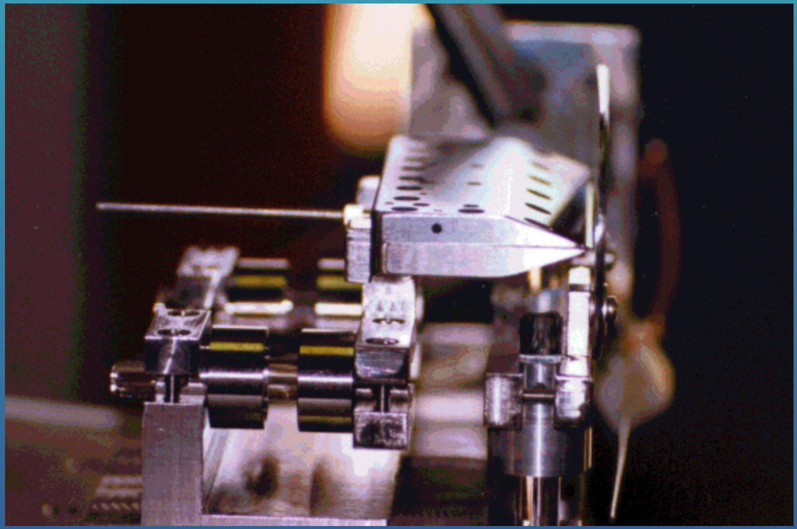
CENTROSPAZIO / ALTA Cesium slit emitter



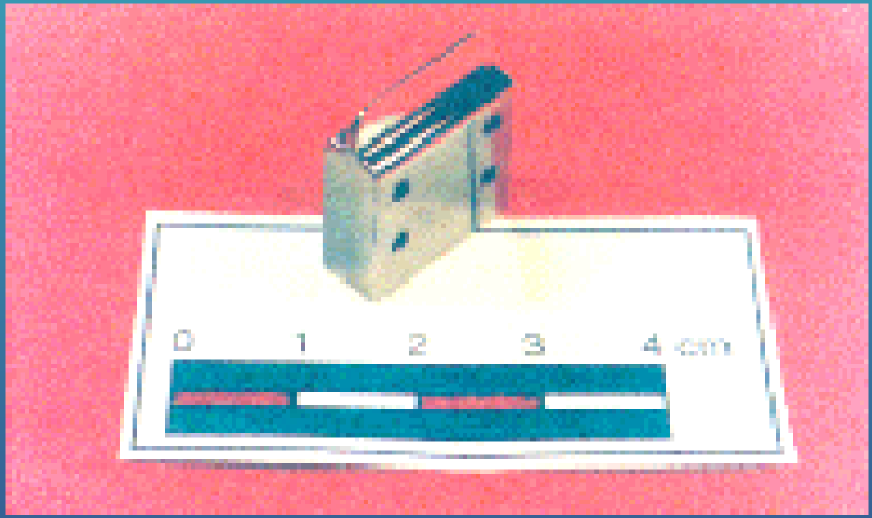
- slit length: 0.5 – 7 cm
- current: <1 mA/cm
- thrust: < 100 μN/cm



Space Propulsion



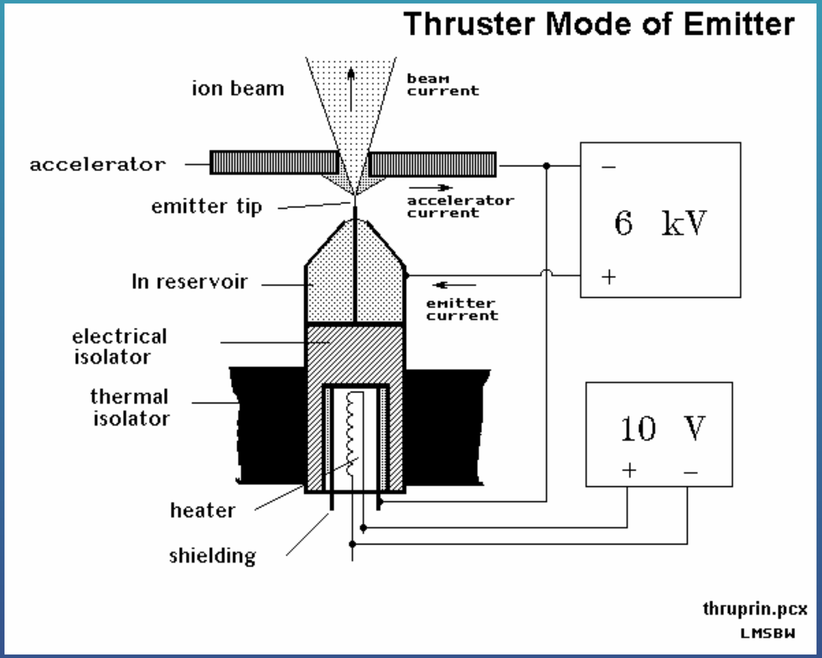
3 cm Cs – slit emitter on thrust stand



slit and reservoir section of 2 cm emitter

- **space experience:** none
- **contracted missions:** MYCROSCOPE (on halt), SMART II ?

Space Propulsion

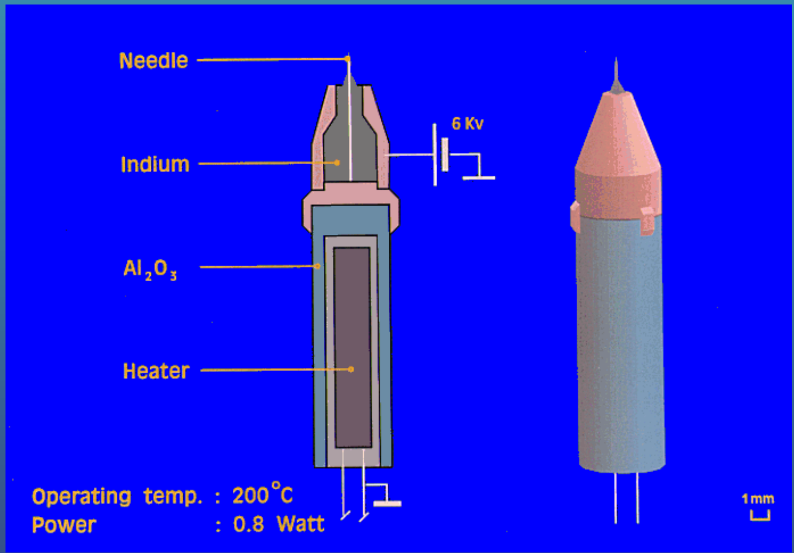


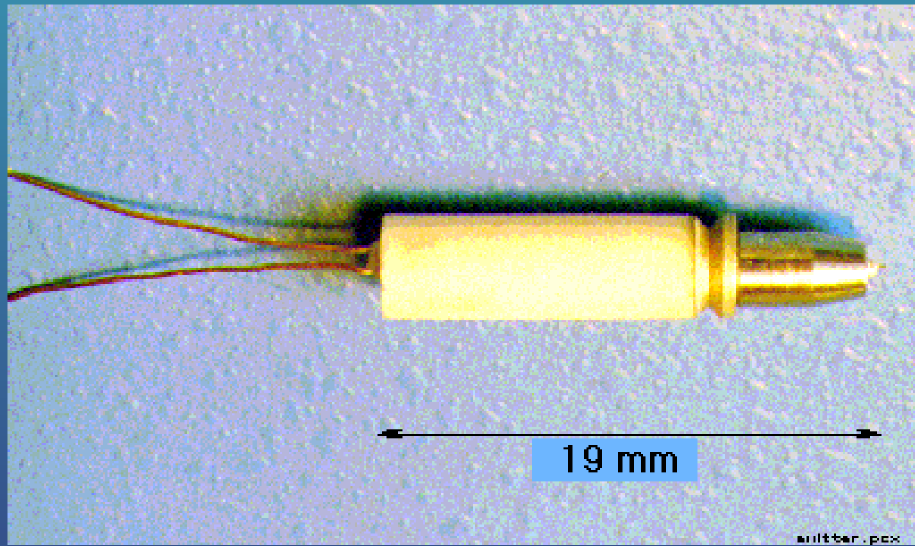
In emitter elements are combined to an emitter **module**, containing

- 1 emitter element
- beam forming electrodes (extractor, focus,…)
- thermal and electrical insulation
- electrical contacts
- mechanical structure and mounting

Space Propulsion

ARCS Seibersdorf In needle emitter





Indium single needle FEEP emitter element

- ion current: 1 – >500 μA
- thrust: 0.1 - 50 μN
- specific impulse 0.8 – 1.2x10^5 m/s
- indium capacity: ~ 0.3 g

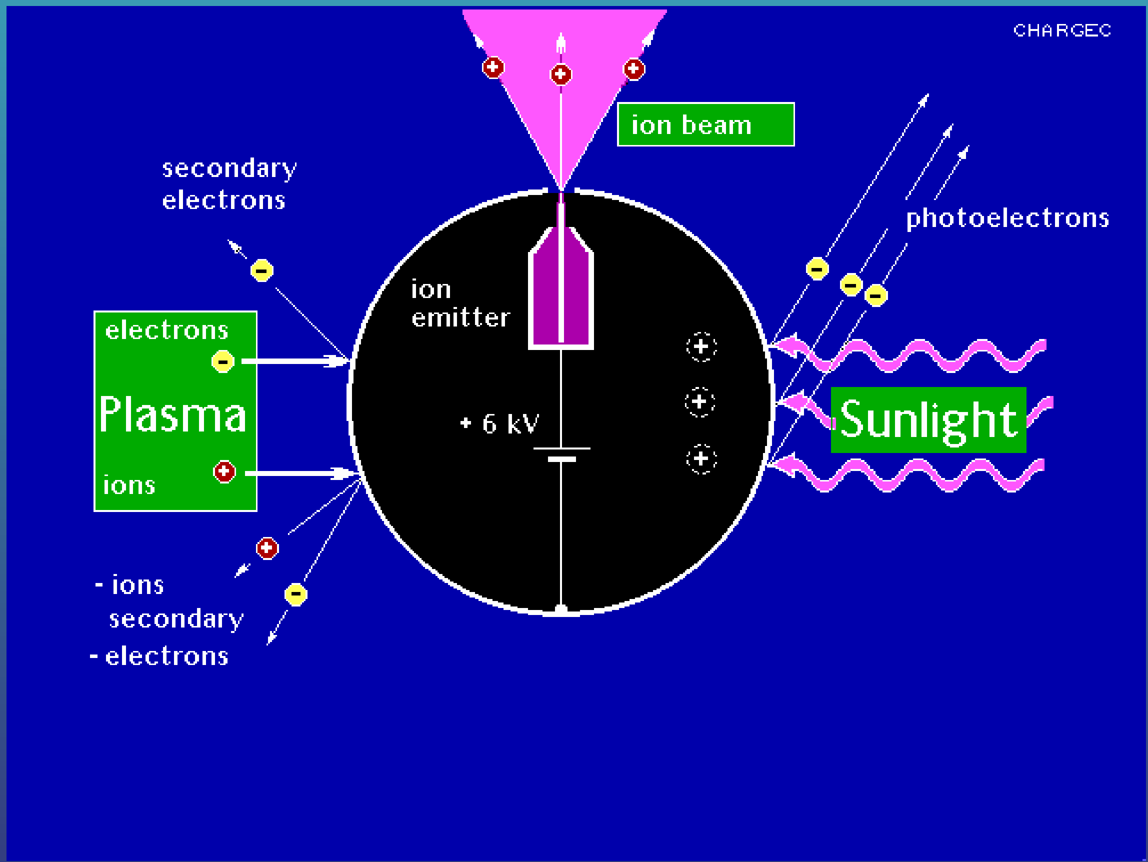
Space Propulsion

SEIBERSDORF ion emitter instruments in space

Name of instrument	purpose	S/C	status	No. of LMIS	logged op. time in space
LOGION	first test of In-LMIS in space	MIR	launch 1991, completed	3	24
MIGMAS-A	ion microscope	MIR	in orbit 1991-1994,	1	120
EFD-IE	potential control	GEOTAIL	in orbit since 1992, operational until mission end 1999	8	500 h
RON	potential control	INTERBALL	launch July 1995	4	10 h at 10 µA
IEI	potential control	EQUATOR-S	launch 1998, S/C lost after successful 1st firing	8	24 h at 10 µA
ASPOC *	potential control	CLUSTER	launch 1996	32	launch failure
ASPOC II	potential control	CLUSTER	launch 2000	32	4 x 10000h at 10µA
COSIMA	TOF-SIMS	ROSETTA	launch Jan. 2003	2	1 year
ASPOC	potential control	DOUBLE STAR	launch 2003/2004	2	1 year

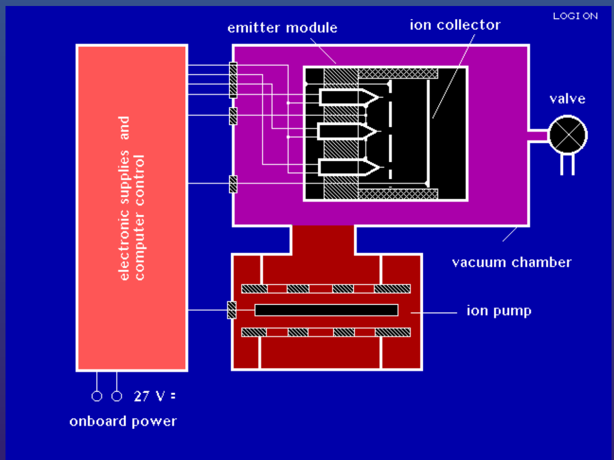
* ARIANE - V launch failure; recovered ion emitters still were functional !

Space Propulsion



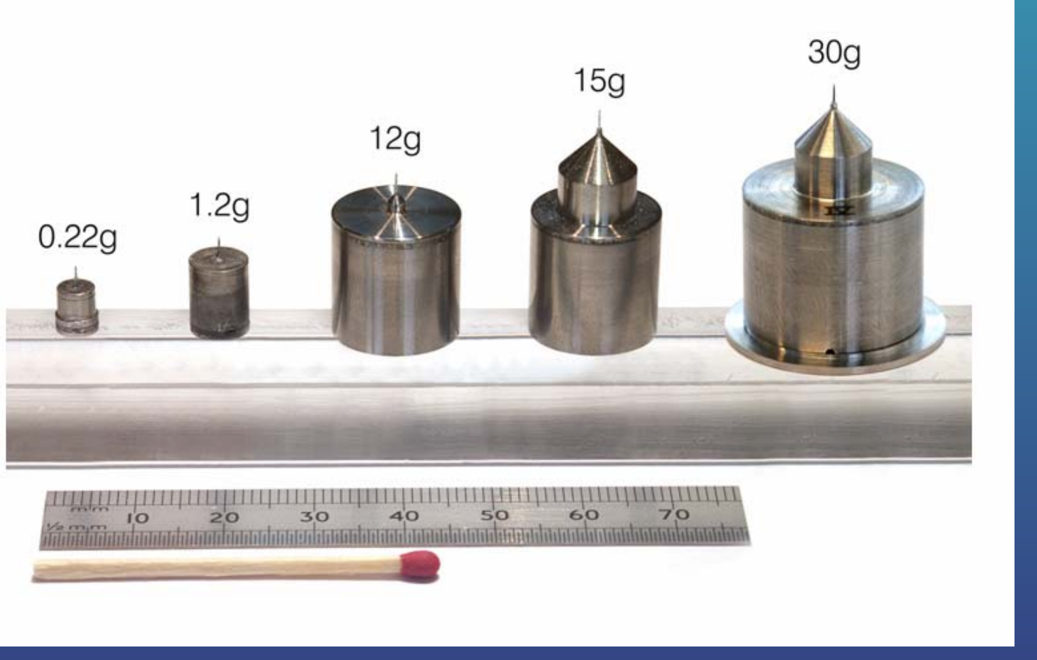
charge balance of a S/C in environmental plasma

Space Propulsion



Space Propulsion

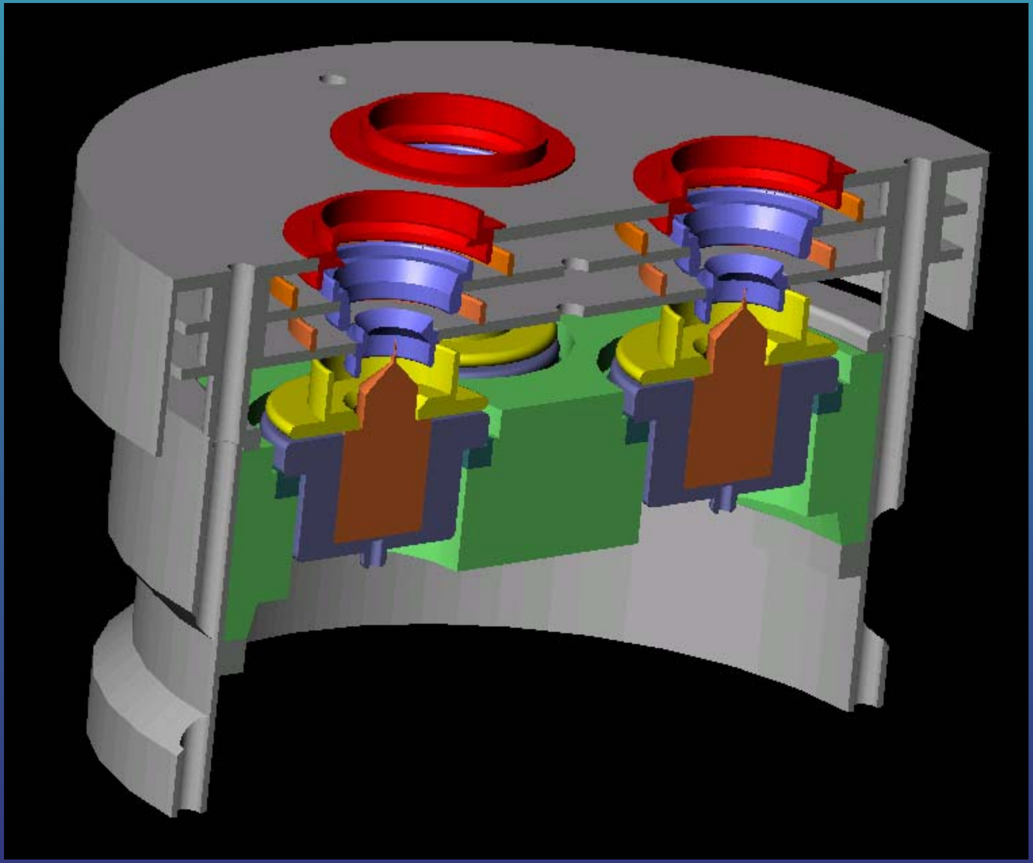
Different versions of ARCS In ion emitter



for applications as ion thruster, propellant volume has to be increased due to increased emission current and lifetime requirements

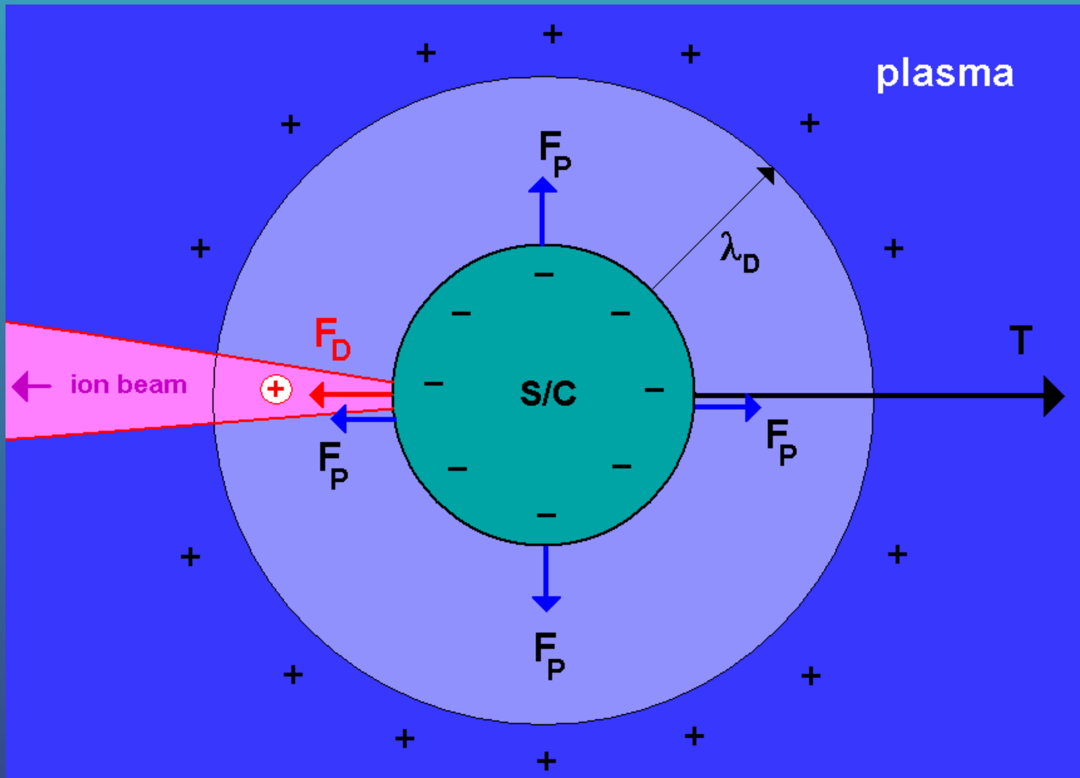
Space Propulsion

ARCS focused In FEEP cluster



At increased thrust (ion current) requirements mass efficiency of single emitter decreases; emitters are **clustered** to reduce current load on single emitter and thus increase mass efficiency. **Focusing** is introduced to lower divergence of external beam and thus reduce contamination of S/C by condensible metallic propellant

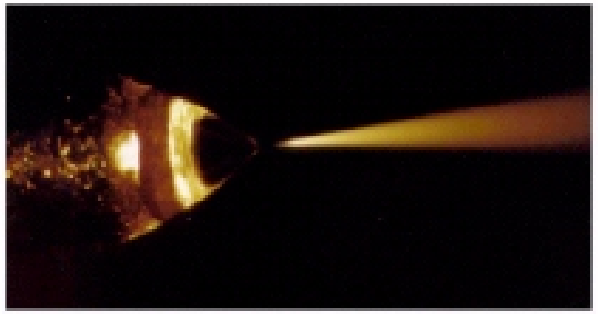
Space Propulsion

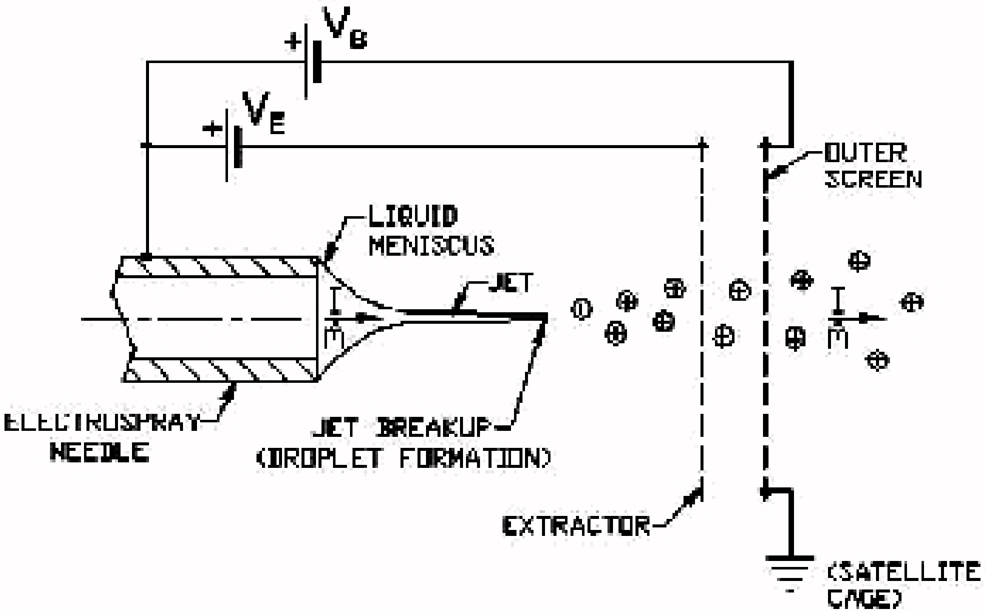


forces on ion – emitting S/C, schematic

due to emission of **positive** ions from thruster, S/C charges **negatively** w/r to environmental plasma; COULOMB forces between plasma and S/C are ~ symmetric, causing no acceleration to S/C; COULOMB forces between negative S/C and emitted positive ions give rise to **decelerating force F_D**, directed **oppositely to thrust T** of emitted ion beam. → electric thrusters always work in conjunction with electron emitters (**neutralisers**) compensating S/C charge.

Space Propulsion





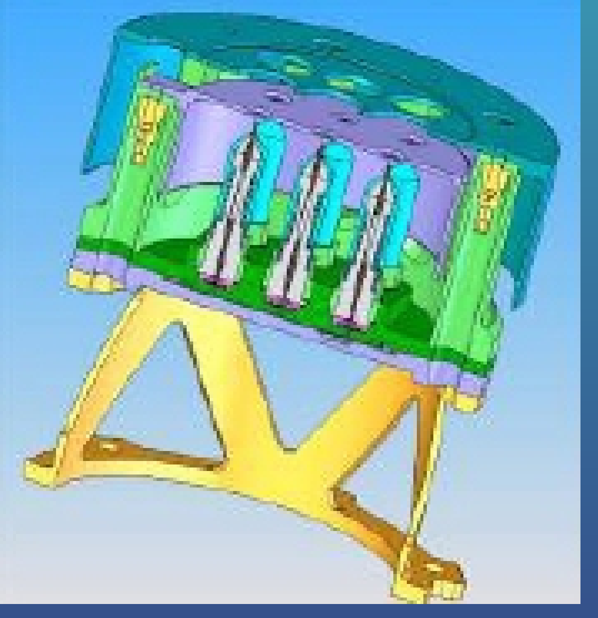
Colloid thruster

Also: electrospray ionizer; ionic liquids are deformed into a jet by strong electric fields; charged droplets separate from the jet apex and are accelerated in the field

Space Propulsion

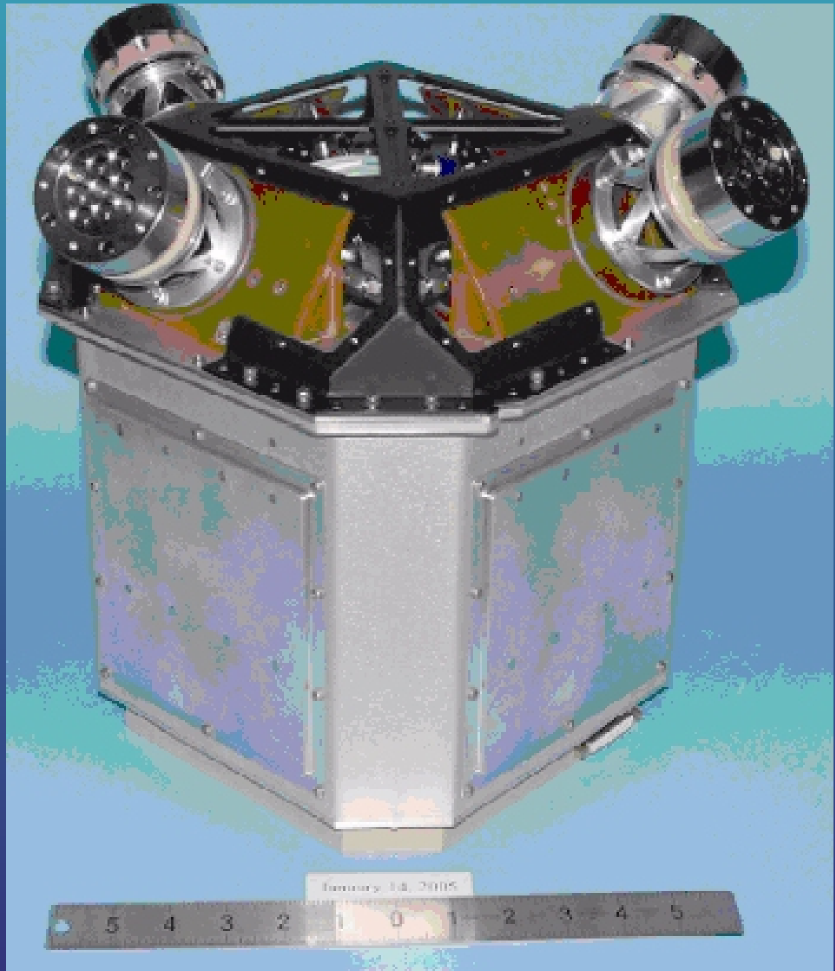
	unit	colloid	FEEP
Specific impulse	[m/s]	$5x10^3 – 1.5x10^4$	$(0.8 – 1)x10^5$
Specific beam power	[W/N]	$\sim 8x10^3$	$\sim 9x10^4$
Charge/mass ratio	[Cb/kg]	< 60 TBP (tributylphosphate) $\sim 11x10^3$ NaI / formamide	$8.7x10^5$ In^+

Space Propulsion



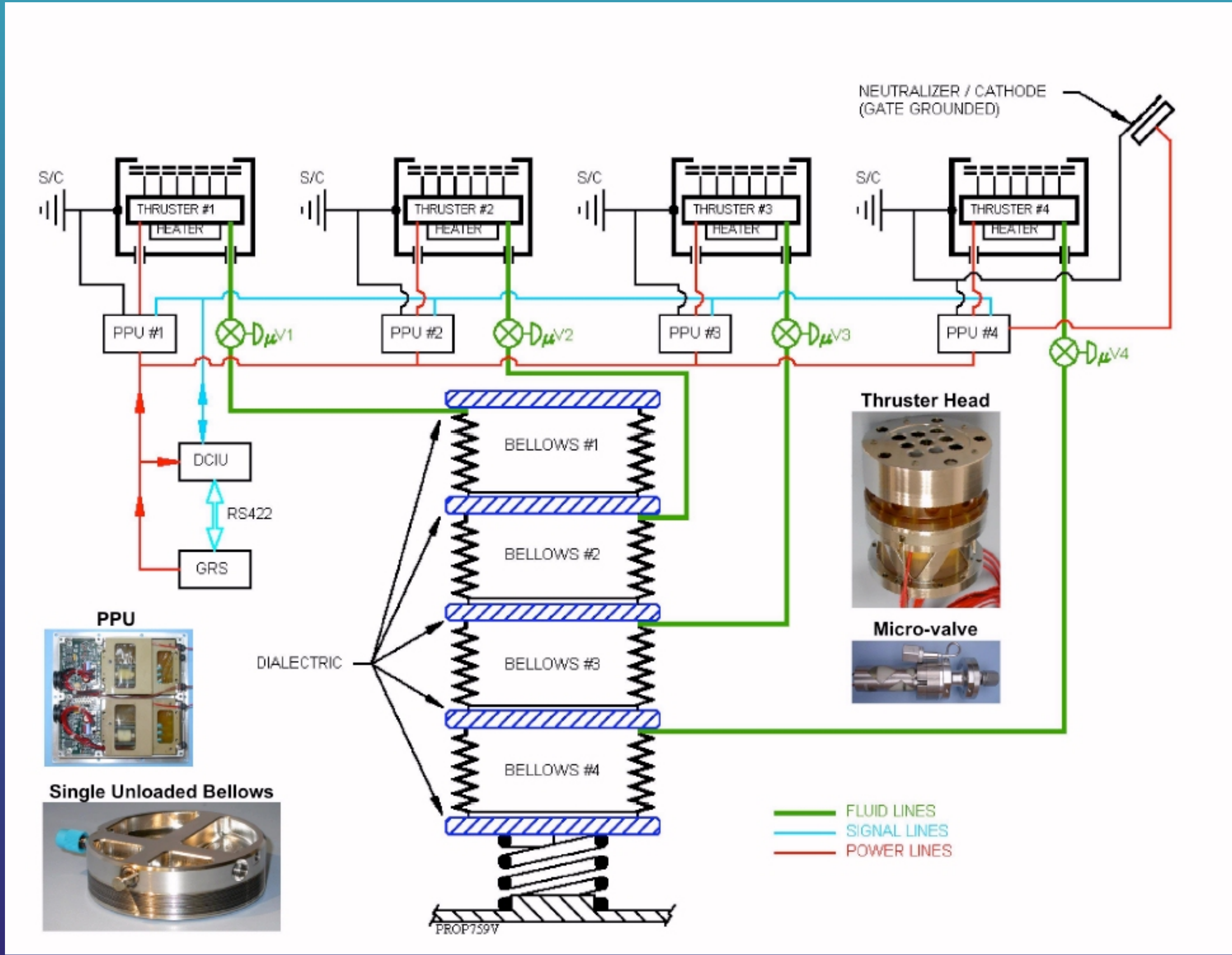
9 – emitter colloid thruster from BUSEK / USA

Space Propulsion



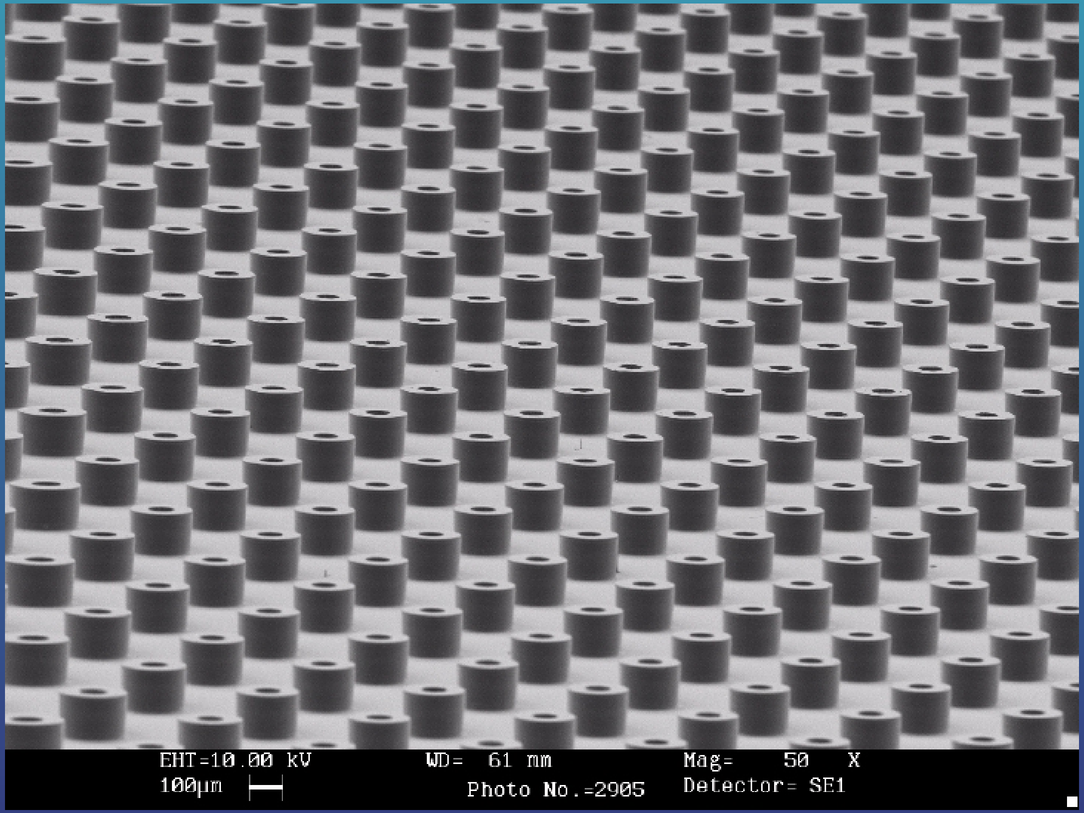
colloid thruster „pod“ serving 2 S/C axes,
flight prototype

Space Propulsion



schematic of 4 –colloid thruster pod for SMART II

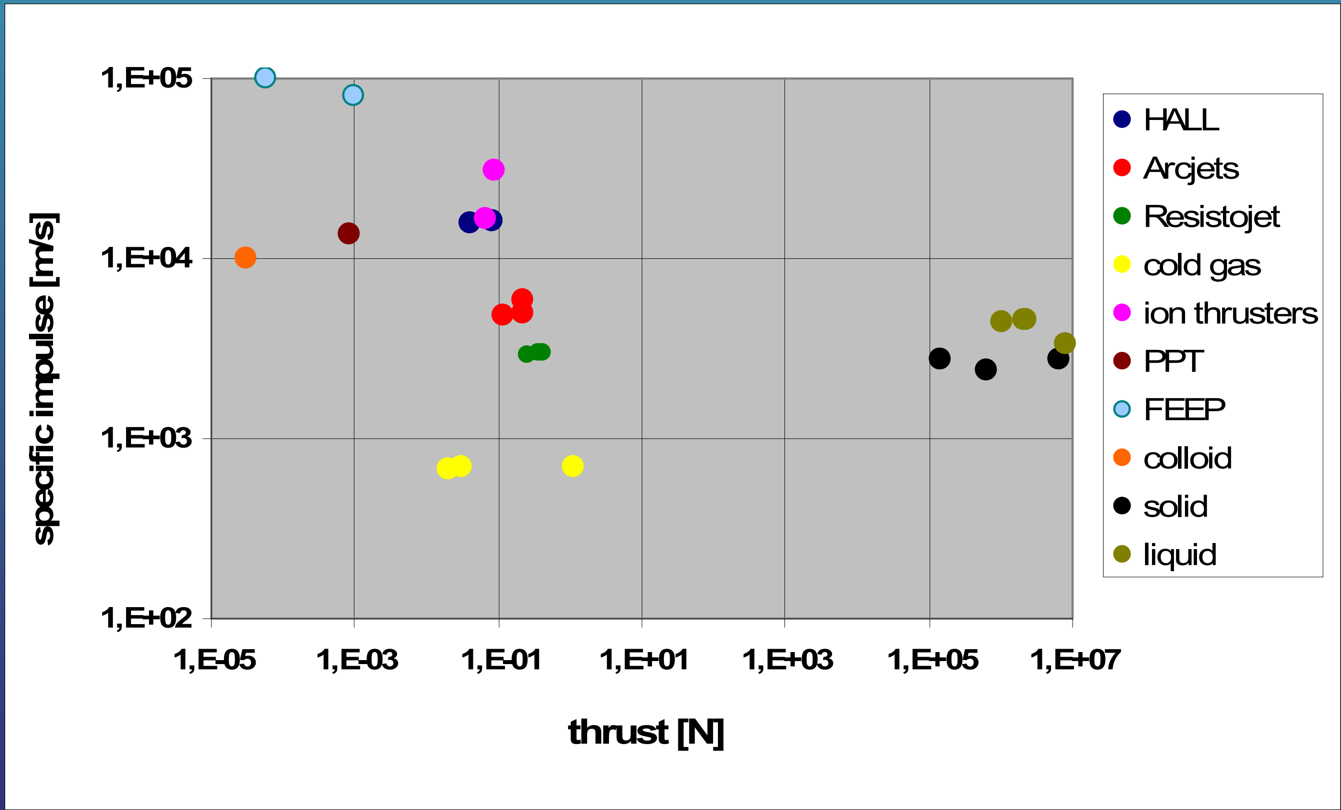
Space Propulsion



experimental miniaturized capillary array for colloid thruster lithographically machined in silicon

Space Propulsion

Thrust and specific impulse of thruster types



Space Propulsion

Type	Advantages	Disadvantages	Current and probable future applications	Status
Resistojets	Rel. Simple, compact, lightweight; simple power cond., very high PCU eff., low op. Voltages; rel high thrust levels, low P/T; can use variety of propellants, high reliability	Lowest Isp; limited by thermal prop. Of materials, high current power transmission; heater erosion problems	GEO (and lower orbit) stationkeeping, repositioning, attitude control, orbit insertion, deorbiting	>200 flown on 75 US& Russian S/C; off-the – shelf hardware
arcjets	Low / mod. Complexity; mod. Power cond. & PCU eff., low/mod P/T, rel high T	Low / mod. Isp, low/mod. Thruster eff., high power transmission, operates with life – limit. Arc discharges	GEO (and lower orbit) stationkeeping, repositioning; orbit raising LEO-> GEO,	>26 flown on > 7 US, Russian & Japanese S/C
Electron bombardment ion	High Isp, mod./high thruster efficiency, long dev. History database	Mod./high complexity, complex PCU, mod. PCU eff'y, low T/A, low T, mod./high P/T, long firing & qual. Time, large volume, reliability problems	GEO (and lower orbit) stationkeeping, repositioning, orbit raising LEO-> GEO, primary prop. for planetary/sol.syst. exploration	>6 flown on >4 US, Russian & Japanese S/C
RF ion	High Isp, mod./high thruster eff'y	High complexity, complex PCU, low/mod. PCU eff., high op. Volt.,low T/A, mod/high P/T, long firing & qual. Time, large volume, reliability problems	GEO (and lower orbit) stationkeeping, repositioning, orbit raising LEO-> GEO. primary prop for planetary/sol.syst. exploration	1 flown on 1 experim. ESA S/C
Field emission (FEEP)	Very high Isp, mod. Complex, compact, mod. Thruster eff'y, small impulse bits allow precise orbit adjustment;	High op. volt., low T/A, low T, high P/T	GEO (and lower orbit) stationkeeping, solar pressure & drag compens., ultraprecise AOC **Fundamental physics**	None flown as thruster, several flown as S/C charge control, under dev. In Europe
Hall effect (HET)	Mod. Isp, values near optimum for variety of app's; mod P/T, mod. operat. Volt, mod./high effic'y, robust, demonstr. Flight reliability	Mod. complex, mod. T/A; low/mod T, add. S/C integration issues	GEO (and lower orbit) stationkeeping, repositioning, orbit raising LEO-> GE O, primary prop for planetary/sol.syst. exploration	>98 flown on >23 Russinan S/C and several US; potential to become "workhorse" vor variety of space app's.
Pulsed plasma (PPT)	Rel. Simple, compact, mod. Isp, low power, Teflon solid propellant, small impulse bit allows precise orbit adjustment	Low T, low T – effic'y, high P/T; Teflon reaction products condensible, need pulse forming network, low PCU eff'y,	GEO (and lower orbit) stationkeeping; sol pressure & drag comp'n; attitude control, fine AOC	>24 flown on > 9US, Russian, Jap. Cin. S/C
Magneto – plasmadynamic (MPDT)	Low3/mod. complexity, mod/high Isp, high T/A, high T, pulsed + stdy state operation	Low/mod T eff'y, mod/gigh P/T, high current power transmission, very low operat. Lifetimes, life limiting arc discharge	orbit raising LEO-> GE O, primary prop for planetary/sol.syst. exploration	>5 flown on >4 Jap.&Russ. Experim. S/C; development continuing

Space Propulsion

typical properties of electrical thrusters

Thruster	Propellant	Input power [W]	Isp [s]	Thrust [mN]	Thrust efficiency [%]	Operat. Lifetime [h]	Manufacturer	status	Total impulse [kNs]	Thruster mass [kg]	PCU mass [kg]	Thruster size [cm]
MR502A Resistojet	Hydrazine	610 – 885	299	360-500		>370	Primex	Several flown	525	0,871		19,8 L x8,8 dia
HPEHT Resistojet	Hydrazine		295	220-490	85		TRW	Several flown	160			
MR-509 Arcjet	Hydrazine	1500	502	209-249	>31[3]	>1050 (1hr on 1/2hr off)	Primex	Several flown	>557	1,47	4,1	24 x 13 x 9
ATOS Arcjet	Ammonia	748 [1]	480	114	36	1010 (1hr on, 1hr off)	IRS (D)	Qual flight 97/98	>400	0,480	2,5	
ESEX Arcjet	Ammonia	26.000 [2]	810	1800 – 2000	27	1500 (400 restarts goal)	Primex	Qual. Flight 98			48,5	
UK10-T5 Ion engine	Xe	278-636 [2]	3200	10 – 25	55-64	10700 (est.)	Matra / DRA / Culham	Completed Qual. Flight				10 (grid dia)
ETS-VIIES Ion engine	Xe	730	3000	20	40		NASDA / MELCO / Toshiba	Flown experimentally				12 (grid dia)

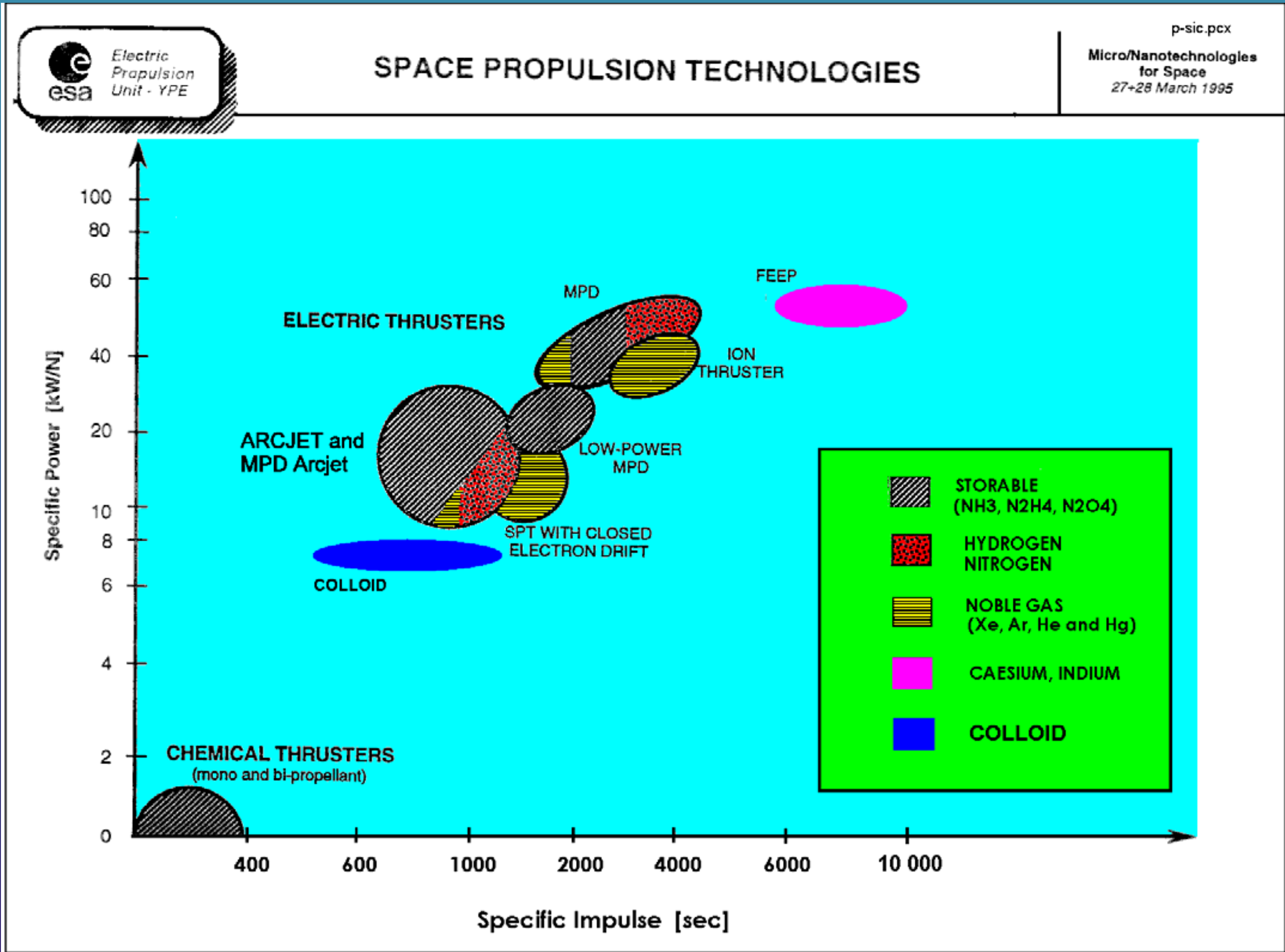
cont'd

Space Propulsion

Thruster	Propellant	Input power [W]	Isp [s]	Thrust [mN]	Thrust efficiency [%]	Operat. Lifetime [h]	Manufacturer	status	Total impulse [kNs]	Thruster mass [kg]	PCU mass [kg]	Thruster size [cm]
NSTAR 30cm ion engine	Xe	2300 – 2500	3310	92	65	>10000 (est.)	NASA / Hughes	Flown DS1		7	12	30 (grid dia)
XIPS25 Ion engine	Xe	1400	2800	63,5	66	>4350. 3850 restarts	Hughes	Quanlified 98			11 sp. Mass 7,9g/W	25 (grid dia)
RIT10 RF ion engine	Xe	585	3100	15	38		DASA	Flown Experim.			9 sp. Mass 15,5g/W	10 (grid dia)
SPT100 HET	Xe	1350[2]	1600	83	45	>7424 (50 min on, 30 min off)	Fakel (Ru)	Several flown	>2000	3,5	8	15 x 22 x 12,5
T-100SPT HET	Xe	1350	1630	83	50	>8000, 3000 restarts (est.)	NIITP (Ru)	Ground tested in RHETT 1		3	10 (est.)	23 x 10 x 13
D-55 TAL HET	Xe	1350, 1600	1600	80	60	>5000 (est.)	TsNIIMASH (Ru)	Exper. Flight 97				
LES8/9 PPT	Solid teflon	25, 30	836	0,3 [5]	6,8 – 9	$>10^7$ pulses 28,5 µg/p	Fairchild – Hillier / MIT	Qual flight 1997	7340	7,1 with fuel		6,8 x 2,7 x 2,2
EPEX Pulsed MPD arcjet	Hydrazine	430	600 (peak)	23 [5]	15		ISAS (Jap)	flown		40,5		

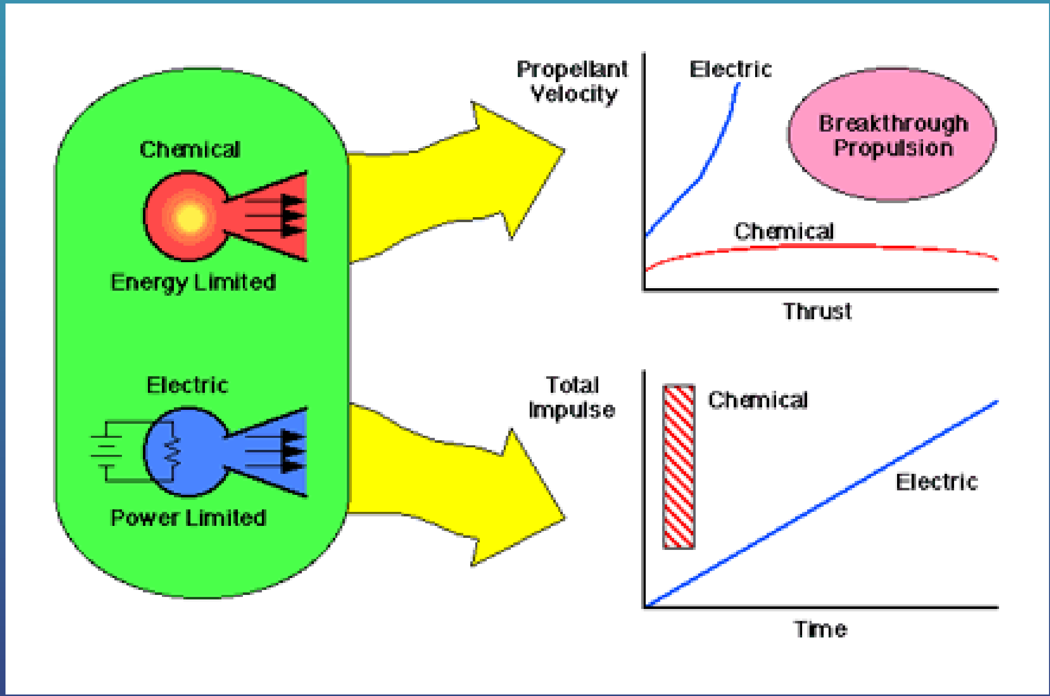
Space Propulsion

specific power vs. specific impulse of electric thrusters



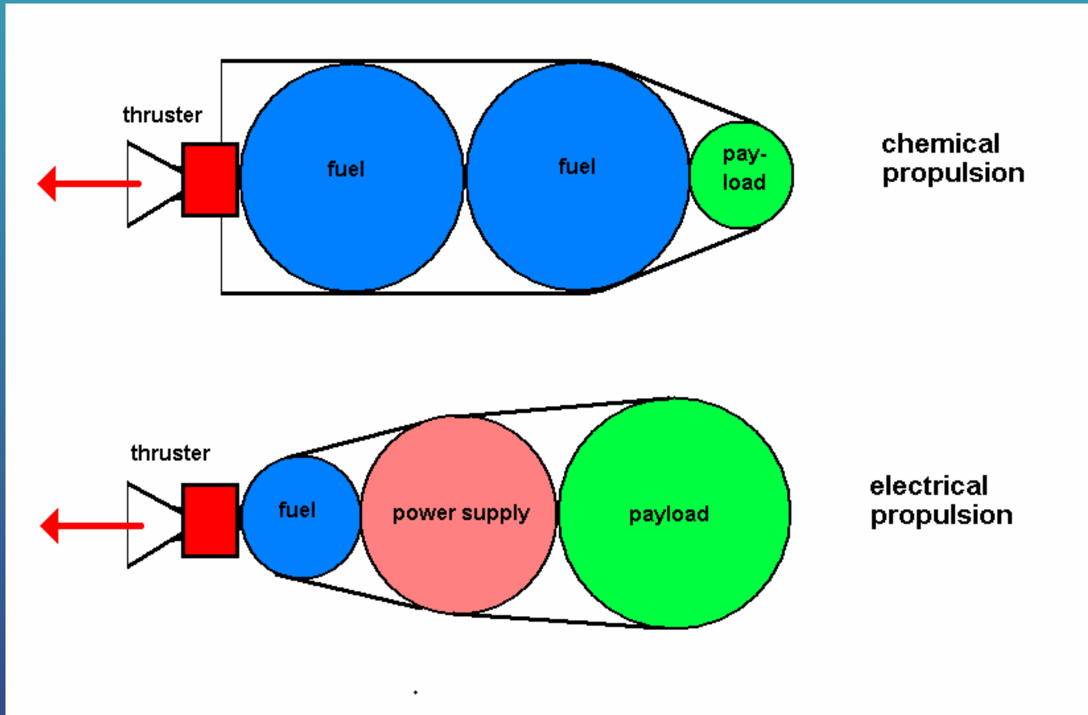
Space Propulsion

intrinsic differences between chemical and electrical propulsion



- CP is limited in total available energy at liftoff
- EP is limited by electrical power available at any moment in flight
- propellant velocity in CP is independent on thrust
- propellant velocity in EP increases with thrust
- total mission impulse in CP is delivered at liftoff
- total mission impulse in EP is accumulated during mission

Space Propulsion



fundamental difference between chemical and electrical propulsion:
EP has to carry ist own power supply

$m_{PPU} = P_{PPU} / \alpha_m$

α_m … mass specific power

Concepts in electric propulsion

thrust

$$T = \dot{m} V_e = \frac{m_p V_e}{\tau} \quad [N]$$

impulse generated per unit of mass flow

jet power

$$P_j = \frac{\dot{m} V_e^2}{2} = \frac{m_p V_e^2}{2\tau} = \frac{T . I_{sp}}{2} \quad [W]$$

kinetic energy in exhaust, generated by thruster in time unit

mass – specific power

$$\alpha_m \equiv \frac{P_j}{m_{PT}} \quad [W/kg]$$

electric power generated per kg of power supply mass

Thr **first two** relations are purely mechanical relationships and therefore hold for all thruster types, in particular **for all electrical thruster types**; dependence of fundamental parameters $\mathbf{V_e}$**, T,** $\mathbf{P_j}$ on thruster properties **depend on thruster type.**

Space Propulsion

Rocket equation for electrical thrusters

what is the payload which can be transported by an electrical rocket, which has to **carry its own power supply**, in a mission with velocity increment ΔV ?

$$\frac{m_f}{m_i} = \frac{m_{PT} + m_l}{m_i}$$

m_{PT} … power system mass
m_L … payload mass

$$\frac{m_L}{m_i} = \frac{m_f}{m_i} - \frac{m_{PT}}{m_i} = \exp\left(\frac{-\Delta V}{V_e}\right) - \frac{m_{PT}}{m_p}\frac{m_p}{m_i}$$

Tsiolkovsky for m_f/m_i

$$\alpha_m \tau = \frac{P_j \tau}{m_{PT}} = \frac{m_p V_e^2}{2 m_{PT}}$$

m_{PT}/m_P

$$\frac{m_L}{m_i} = \exp\left(\frac{-\Delta V}{V_e}\right) - \frac{V_e^2}{2\alpha_m \tau}\frac{m_p}{m_i}$$

$$\frac{m_p}{m_i} = \frac{m_i - m_f}{m_i} = 1 - \frac{m_f}{m_i} = 1 - \exp\left(\frac{-\Delta V}{V_e}\right)$$

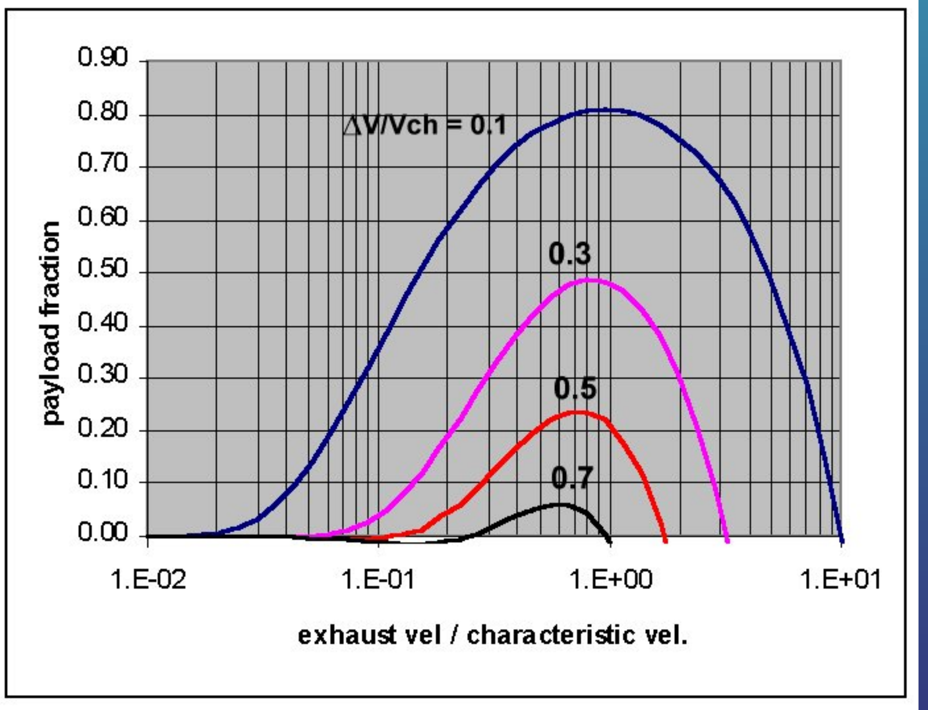
$$\boxed{\frac{m_L}{m_i} = \exp\left(\frac{-\Delta V}{V_e}\right) - \frac{V_e^2}{2\alpha_m \tau}\left[1 - \exp\left(\frac{-\Delta V}{V_e}\right)\right]}$$

$$v_{ch} = \sqrt{2\alpha_m \tau}$$

“characteristic” velocity for mission
τ …engine burn time

Space Propulsion

$$\frac{m_L}{m_i} = \exp\left(\frac{-\Delta V}{V_e}\right) - \frac{V_e^2}{v_{ch}^2}\left[1 - \exp\left(\frac{-\Delta V}{V_e}\right)\right]$$



$m_L/m_i \rightarrow 0$

for $\Delta V/v_{ch} \rightarrow 0.81$, where $V_e/v_{ch} \rightarrow 0.5$

payload fraction vanishes for $\Delta V/V_{ch} > 0.81$

payload fraction m_L/m_i of a high exhaust velocity rocket in dependence on total velocity gain and exhaust velocity, both normalized to characteristic velocity

Space Propulsion

Example: *what is approximate engine burn time to reach moon orbit (384.000 km) from LEO (300 km altitude) for a ~ 500 kg S/C with negligible payload (SMART I) ?*

SPT100 Hall engine, ~ 1kW jet power; $\alpha_m \approx$ 100 W/kg for SPT 100 Hall thruster

$$v_{ch} = \sqrt{2\alpha_m \tau}$$

at given PPU properties, v_{ch} is determined by burn time τ

however: since whole space probe, including solar panels, PPU, thrusterand structure has to be brought to final orbit, we can include total S/C mass (367 kg) into an effective $\alpha*$

$\alpha* =$ 1000 W / 367 kg ~ 2.7 W/kg

Space Propulsion

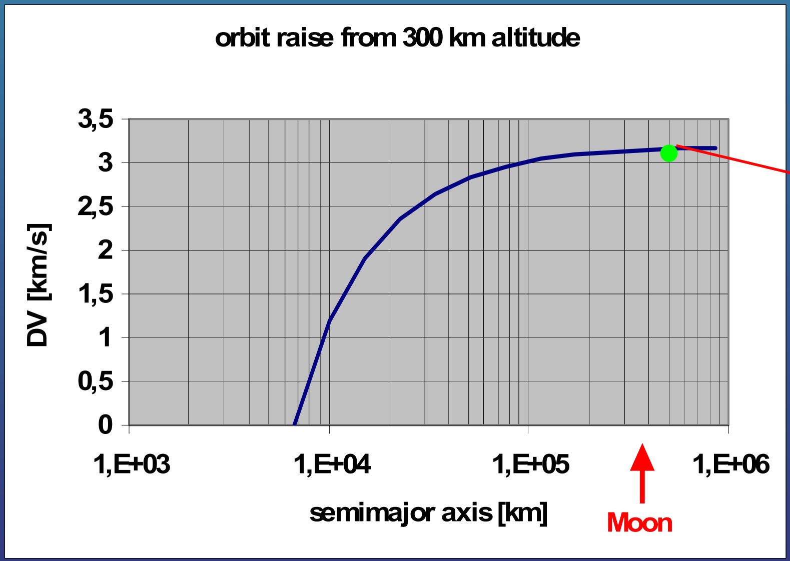
what ΔV is necessary to reach moon orbit ?

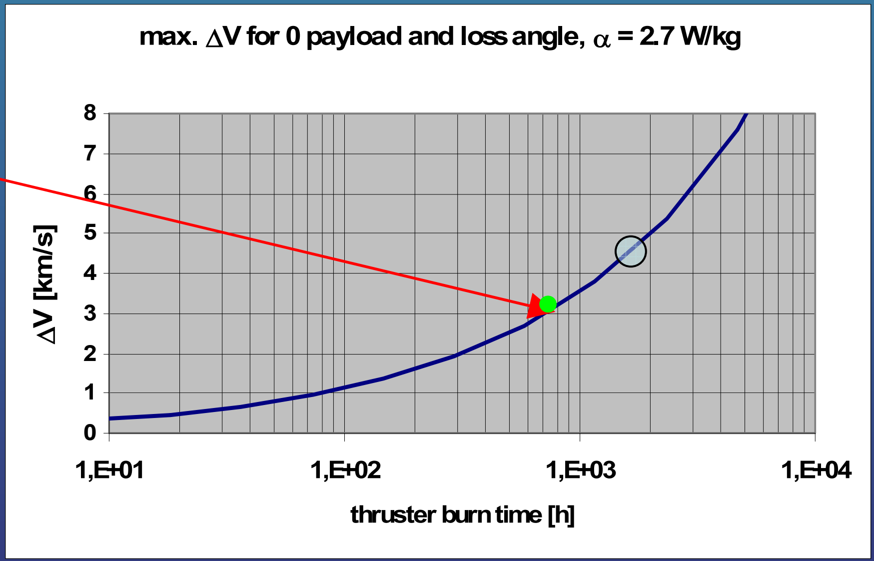
$$\Delta V = \sqrt{\frac{2\mu}{r_0} - \frac{\mu}{a}} - \sqrt{\frac{\mu}{r_0}}$$

what maximum ΔV can be obtained from ion engine at negligible payload ?

$$\Delta V < 0.8 * v_{ch} \qquad v_{ch} = \sqrt{2\alpha_m \tau}$$

$$\Delta V_{km/s} < 0.112 * \sqrt{\tau_h}$$



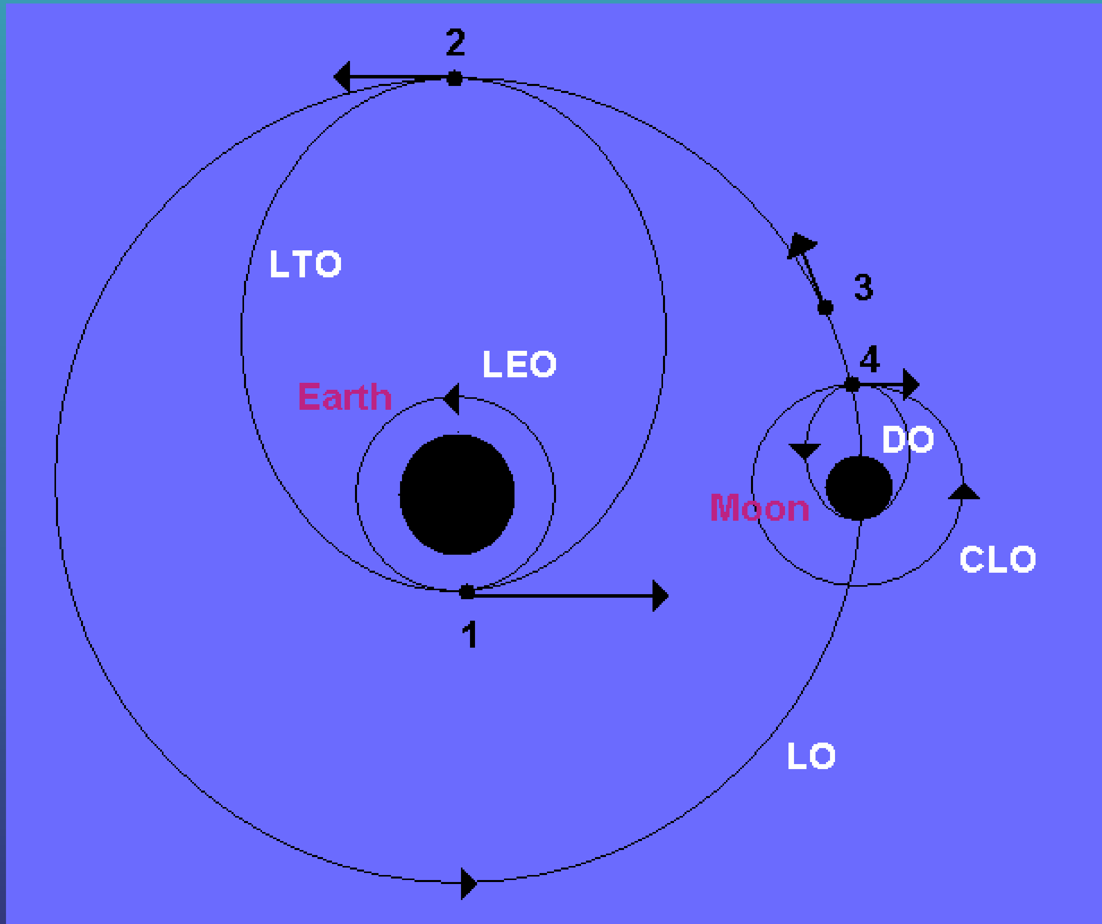


engine burntime, provided loss angle is always ~ 0: $\tau \sim 800$ h

total burntime reported for SMART 1: $\tau \sim 5000$ h

?

Space Propulsion



approximation of SMART 1 orbit by conventional conics

Space Propulsion

reasons for discrepancy:

- calculation is for lunar tranfer orbit to reach 384000 km apogee
- SMART 1 mission contains lunar orbit circularisation and deceleration for lunar impact
- payload is not exactly 0
- in SMART 1 unknown loss angles have to be considered, although average firing pulse length was only ~ 58 h; no info on thruster reorientation cycles
- orbits not elliptical
- no info available on additional maneuvres

Electric Propulsion System operations history	
Number of Pulses	844
Total number of hours fired (h)	4958.3
Xenon at launch (kg)	82.5
Remaining Xenon (g)	280
Remaining useable Xenon (g)	~ 60

Space Propulsion

estimate ΔV for lunar transfer orbit circularisation

last approx. ellipse of LTO:

$r_p \sim 6387+300 = 6687$ km
$r_a \sim r_M \sim 384000$ km
$a_{LTO} \sim (r_M + r_p)/2 \sim 195000$ km

$$V_a = \sqrt{\frac{2\mu}{r_a} - \frac{\mu}{a_{LTO}}} = 0.188 \ \ [km/s]$$

lunar velocity

$$V_a = \sqrt{\frac{\mu}{r_M}} = 1.017 \ \ [km/s]$$

ΔV for LTO circularisation:

$\Delta V \sim 0.829$ km/s

Space Propulsion

estimate ΔV for circumlunar orbit insertion at 1000 km lunar altitude

lunar mass : $m_M \sim 7.34E22$ kg
lunar radius : $R_M \sim 1738$ km
gravity constant : $G \sim 6.67E\text{-}20$ [$km^3\ s^{-2}\ kg^{-1}$]

$\mu_M = G.m_M \sim 4{,}90E{+}03\ km^3 s^{-2}$

Insert into circumlunar ellipse ~ infinite apoapsis and periapsis of desired circular orbit (r_p = 2738 km); periapsis velocity of this insertion orbit is:

$$V_p = \sqrt{\frac{2\mu_M}{r_p} - \frac{\mu_M}{\infty}} = 1.892\ \ [km/s]$$

circular orbit velocity is:

$$V_C = \sqrt{\frac{\mu_M}{r_p}} = 1.338\ \ [km/s]$$

ΔV_C for circular orbit insertion from „infinity“ is:

ΔV_C = 1.892-1.338 = 0.554 km/s

Space Propulsion

estimate ΔV for descent orbit insertio to lunar surface impact

circular orbit velocity (r_a = 2738 km) is:

$$V_C = \sqrt{\frac{\mu_M}{r_a}} = 1.338 \ [km/s]$$

semimajor axis of decent orbit with apoapsis r_a = 2738 km and periapsis r_p = r_L =1738 km is:

$$a = (r_a + r_p)/2 = 2238 \ [km]$$

apoapsis velocity of descent ellipse is:

$$V_a = \sqrt{\frac{2\mu_M}{r_a} - \frac{\mu_M}{a}} = 1.179 \ [km/s]$$

ΔV for descent orbit insertion is:

ΔV_d = 1.338 – 1.179 = 0.159 km/s

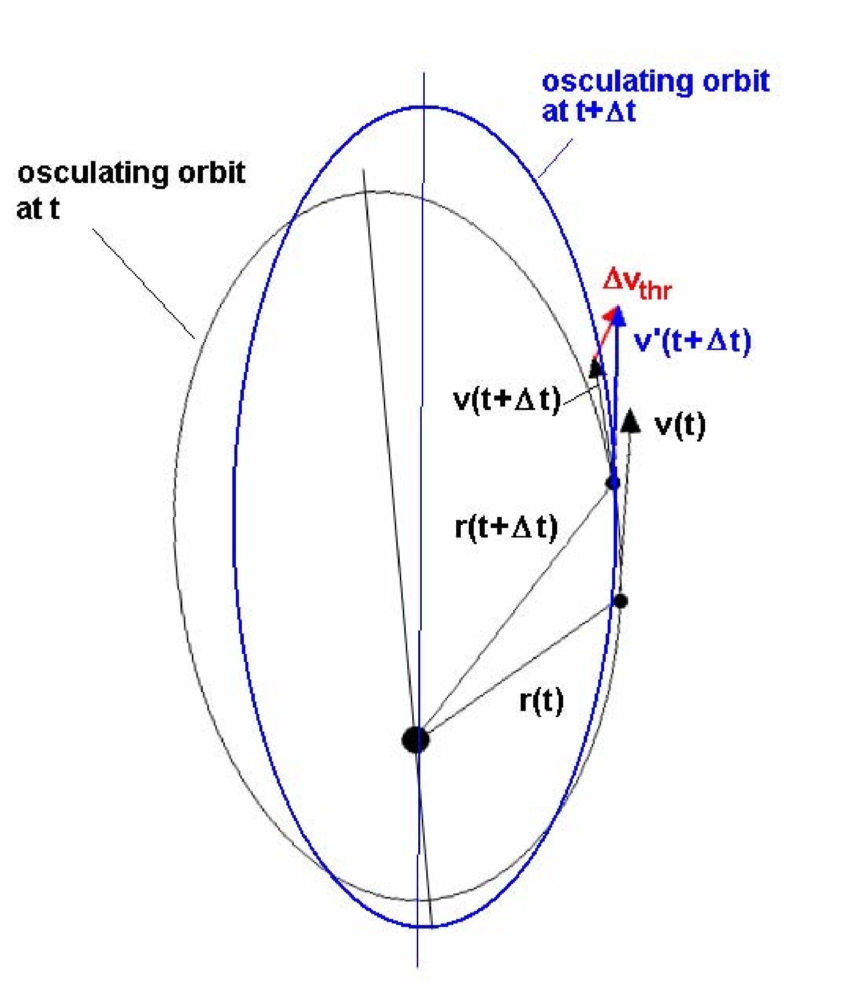
total ΔV for LTO insertion, LTO circularisation, circular lunar orbit insertion and descent orbit insertion is:

Δ_{tot} = 3.2+ 0.829 + 0.554 + 0.159 =
= **4.742** km/s

total engine burn time from LEO to lunar descent orbit:

1900 h

Space Propulsion



numerical integration of continu-ous - burn orbits

osculating orbit at time t is completely determined by r(t) and v(t)

during time increment Δt r and v change due to progress on original osculating orbit;
v changes additionally by Δv due to momentum delivered by thruster during Δt

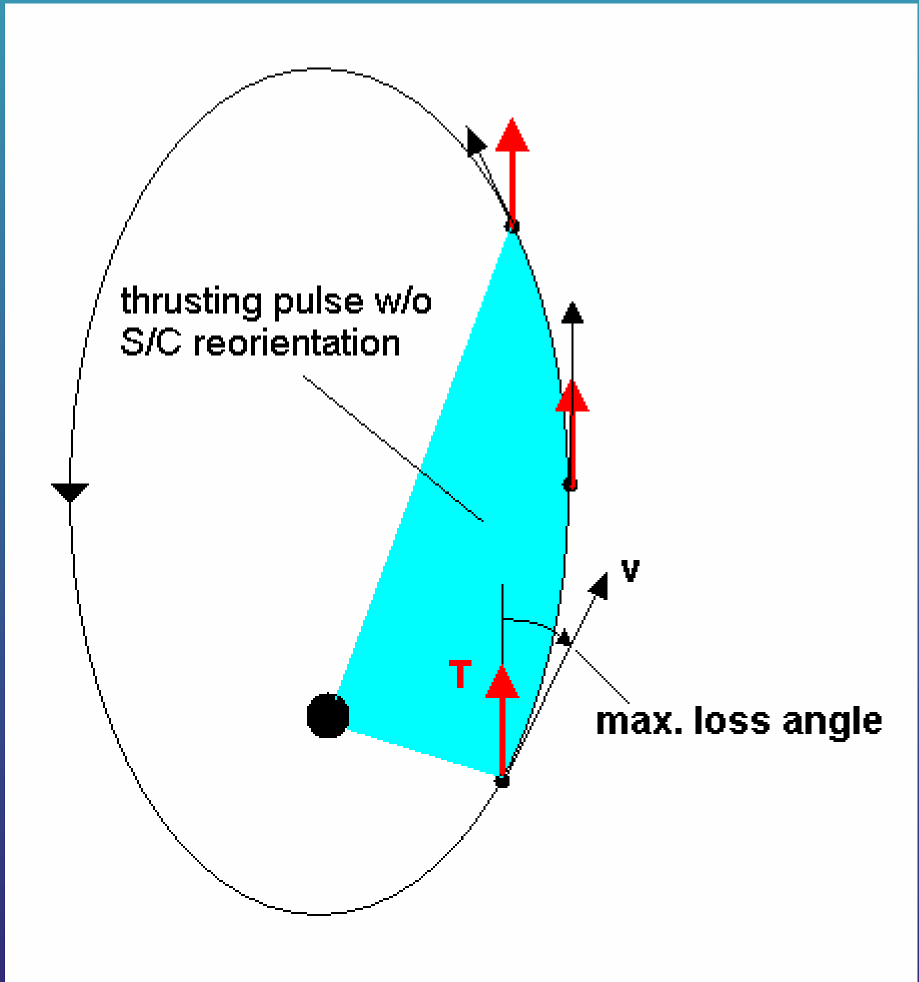
$$\Delta p \cong m_{S/C}\left(t + \Delta t/2\right).\Delta v = T.\Delta t$$

$$\Delta v_{thr} = \frac{T.\Delta t}{m_{S/C}\left(t + \Delta t/2\right)}$$

$$\Delta m = \frac{\Delta p}{I_{sp}} = \frac{T}{I_{sp}}\Delta t$$

vectorial addition of v(t+Δt) and Δv_{thr} gives new velocity v' at r(t+Δt) and new osculatory orbital elements

Space Propulsion



without reorientation, ion engine can burn only during part of orbit („thrusting pulse“)

Space Propulsion

Assume a mission requiring a given thrust level during mission time τ. How can initial mass be minimized, provided we have thrusters with different specific impulse to choose from?

chemical thrusters:

$$m_{tot} = m_p + m_L = \frac{T.\tau}{I_{sp}} + m_L = \frac{p_{tot}}{I_{sp}} + m_L$$

lowest total mass for highest possible I_{sp}

electrical thrusters:

$$m_{tot} = m_p + m_{PPU} + m_L$$

$$m_p = (T.\tau)/I_{sp}$$

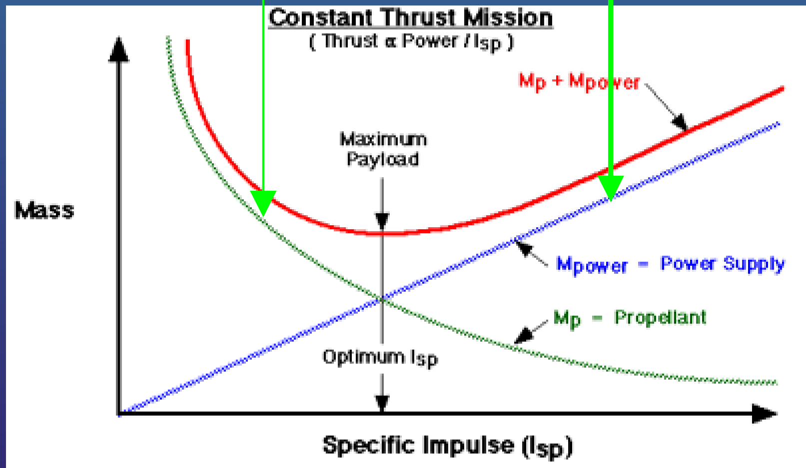
propellant mass decreasing with increasing I_{sp}

$$m_{PPU} = P_j/\alpha_m = \frac{1}{2\alpha_m} T.I_{sp}$$

PPU mass increasing with I_{sp}

α_m = (max. power output of PPU)/ (mass of PPU)

$P_j = T.I_{sp}/2$



There must be optimum specific impulse, for which total initial mass is minimum. This usually favours choice of particular thruster type

Space Propulsion

minimize total initial mass when electric thruster is main engine

a: total velocity increment ΔV and thrust T are given

$$m_i = m_f e^{\Delta V / I_{sp}} = \left(m_{PPU} + m_L\right) e^{\Delta V / I_{sp}} = \frac{T}{2\alpha_{PPU}} I_{sp} e^{\Delta V / I_{sp}} + m_L e^{\Delta V / I_{sp}}$$

minimize w/r to I_{sp} !

$$\left(I_{sp}\right)_{opt} = \frac{\Delta V}{2}\left(1 + \sqrt{1 + \frac{8\alpha_{PPU}.m_L}{T.\Delta V}}\right)$$

---

small payload, large ΔV approximation → $\left(I_{sp}\right)_{opt} \cong \Delta V$

$$m_P = m_i - m_f = m_i\left(1 - e^{-\Delta V / I_{sp}}\right)$$

I_{sp} → ΔV → $m_P \cong m_i\left(1 - \frac{1}{e}\right) \cong 0.632 m_i$

Space Propulsion

Example

Let us assume, that a small satellite of **100 kg** mass has to be brought as the payload of a S/C into a higher orbit and that this higher orbit requires a velocity increment of **$\Delta V \cong 4$ km/s**. Let us further assume that the thruster can deliver **1 N** thrust and that the electric power supply system (Iincluding solar sails) has a mass specific power of ca. **$3x10^{-2}$ kW/kg**. *What is the optimum specific impulse of the thruster system for that task, minimizing total* starting mass?

The parameters are:	m_L = 100 kg	$\Delta V = 4\ x10^3$ m/s
	T = 0.1 N	α_{PPU} = 30 W/kg

In $$\left(I_{sp}\right)_{opt} = \frac{\Delta V}{2}\left(1+\sqrt{1+\frac{8\alpha_{PPU}.m_L}{T.\Delta V}}\right)$$ $(8m_L/\alpha_{PPU}T.\Delta V) \sim 60$ and the optimum specific impulse is ***$(I_{sp})_{opt} \cong \Delta V \cong 1.76x10^4$ m/s***

***Note** that for most thruster types the specific impulse is practically fixed and cannot be chosen freely. The optimum value of I_{sp} therefore gives a recommendation for the choice of thruster type for that particular mission.*

Space Propulsion

b: total impulse p_{tot} and thrust T are known, minimize total mass of thruster system (typical for attitude control thrusters, since T and τ are given)

$$m_{\tan k} + m_{thru} = f_T . m_p$$

f_T … tankage factor

$$m_{tot} = m_p + (m_{\tan k} + m_{thru}) + m_{PPU}$$

$$m_{tot} = m_P (1 + f_T) + m_{PPU} = \frac{p_{tot}}{I_{sp}}(1 + f_T) + \frac{T}{2\alpha_{PPU}} I_{sp}$$

total thruster system mass at begin of mission

$$\left(I_{sp}\right)_{opt} = \sqrt{\frac{2 p_{tot} \alpha_{PPU} (1 + f_t)}{T}} = \sqrt{2\tau\alpha_{PPU}(1 + f_t)}$$

$$\left(m_{tot}\right)_{\min} = \sqrt{2T . p_{tot}(1 + f_t) / \alpha_{PPU}} =$$

$$= T\sqrt{2\tau(1 + f_t) / \alpha_{PPU}}$$

total mass is minimized for optimum I_{sp}

other parameters can be minimized: e.g. total resource usage, propellant mass, at different values of optimum specific impulse

Space Propulsion

Example: an electric thruster system is to be flown as an ACS; what is specific impulse minimizing total thruster system mass?

Mission parameters:

p_{tot} = 4000 N/s T = 50 μN f_T ~ 10 α_{PPU} ~ 10 W/kg

$$\left(I_{sp}\right)_{opt} = \sqrt{\frac{2 p_{tot}(1+f_t)\alpha_{PPU}}{T}}$$

→ I_{sp} = 1.33x10^5 m/s

Space Propulsion

c: total impulse and thrust are known
minimize total S/C resource usage of thruster system

(particularly applicable when thrusters are part of AOCS of space probe)

$$M = M_{dr} + M_p + M_{PPU} = M_{dr} + \frac{p_{tot}}{I_{sp}} + \frac{T.I_{sp}}{2\alpha_{PPU}}$$

mass of thruster system (= dry mass + propellant)

$$P = P_{sb} + P_j = P_{sb} + \frac{T}{2} I_{sp}$$

power consumption of thrusting system (standby power + + jet power)

$$r_{tot} = \frac{M}{M_{sat}} + \frac{P}{P_{sat}}$$

measure for relative use of S/C resources by thruster system

$$\left(I_{sp}\right)_{opt} = \sqrt{\frac{2p_{tot}/T}{1/\alpha_{PPU} + M_{S/C}/P_{S/C}}} = \sqrt{\frac{2\tau_m}{1/\alpha_{PPU} + M_{S/C}/P_{S/C}}}$$

optimum I_{sp} minimizing resource usage r_{tot}

Space Propulsion

Example

An electric thruster system is used as the ACS of a satellite; what is the specific impulse minimizing total S/C resource usage?

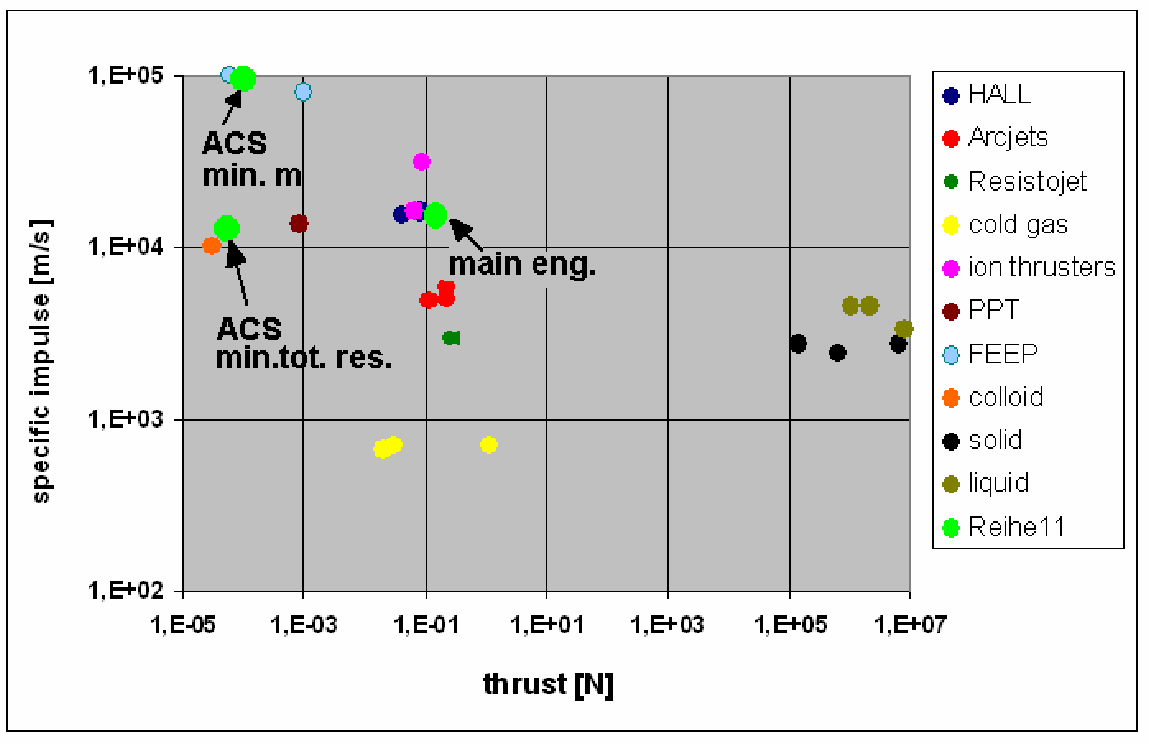
Parameters:

$M_{S/C}$ = 100 kg $P_{S/C}$ = 500 W α_{PPU} ~ 10W/kg mission time ~ 2 y

$$\left(I_{sp}\right)_{opt} = \sqrt{\frac{2\tau_m}{1/\alpha_{PPU} + M_{S/C}/P_{S/C}}}$$

→ I_{sp} ~ 1.45×10^4 m/s

Space Propulsion



Optimum thruster types:

- ACS with minimum mass: FEEP, PPT
- ACS with minimum resource usage: colloid, PPT, FEEP
- main engine: ion thruster, HALL

but:

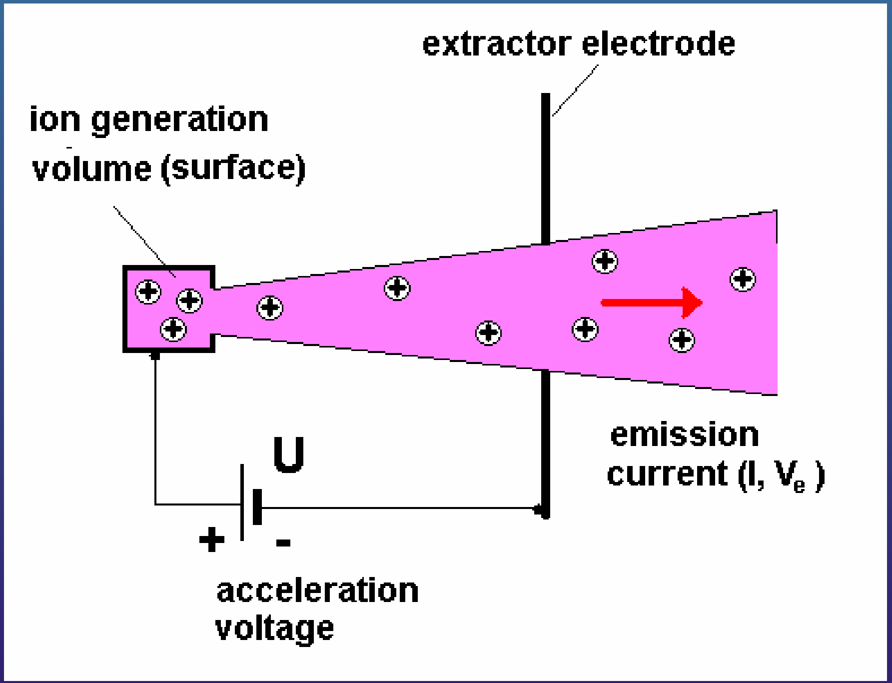
other parameters also play a role: tankage factor, α_{PPU}, mission time, …
and development is going on!

Space Propulsion

Performance variables of ideal electrostatic thrusters

electrostatic thruster: ions are accelerated by electrostatic fields only

ideal el.-st. thruster:
- all ions emitted have the same mass / charge ratio
- all ions are accelerated by the same accelerating voltage
- all ions are emitted into the same direction



Full power from power supply goes into beam

Jet power P_j is equal to electrical power

Space Propulsion

express kinetic and dynamic beam parameters by I, U and particle properties !

emission current:

$$I = \frac{dN}{dt} e_o . n_c$$ [A]

energy conservation

$$\frac{mV_e^2}{2} = qU = n_c e_0 U$$

exhaust velocity of ion

$$V_e = \sqrt{\frac{2qU}{m}} = \sqrt{\frac{2n_c e_0 U}{m_0 M}} \cong 1.3891x10^4 \sqrt{\frac{n_c U}{M}}$$ [m/s]

momentum of exhausted particle

$$p = m.V_e = \sqrt{2qUm} \cong 2.3067x10^{-23} \sqrt{n_c UM}$$ [N.s]

mechanical thrust definition

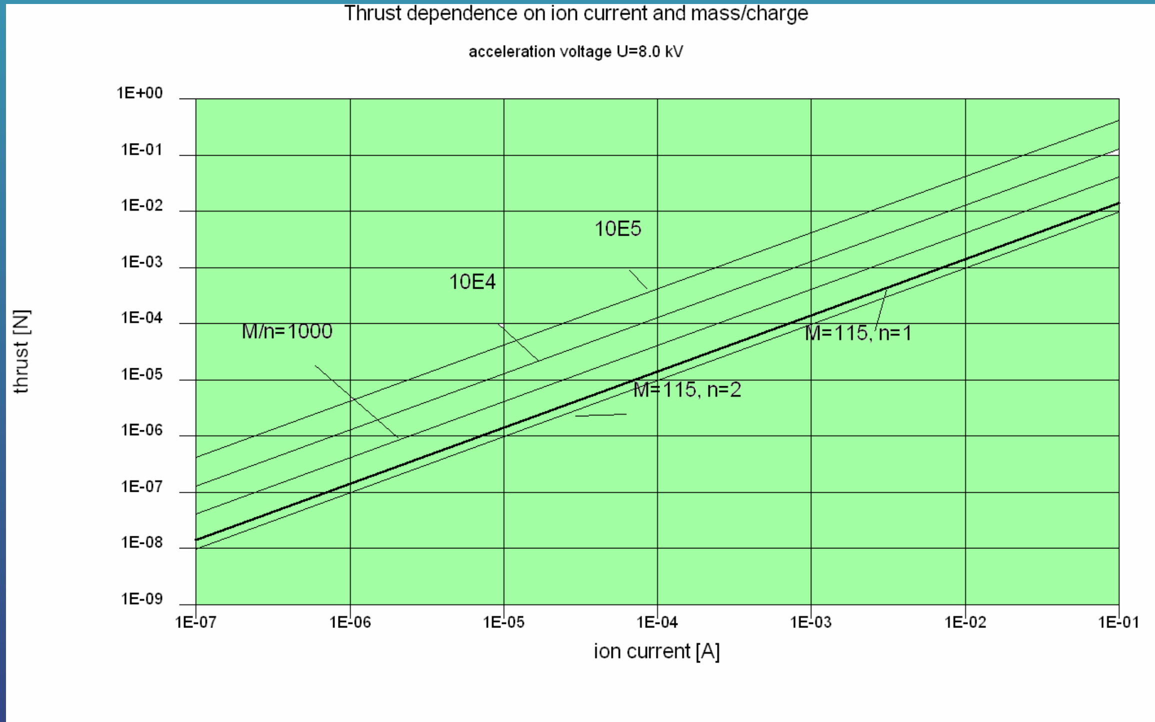
$$T = \frac{dN}{dt} p = \frac{I}{e_0 n_c} p$$

thrust of exhaust jet with current I

$$T = I.\sqrt{\frac{2mU}{e_o n_c}} \cong 1.4397x10^{-4} I \sqrt{\frac{MU}{n_c}}$$

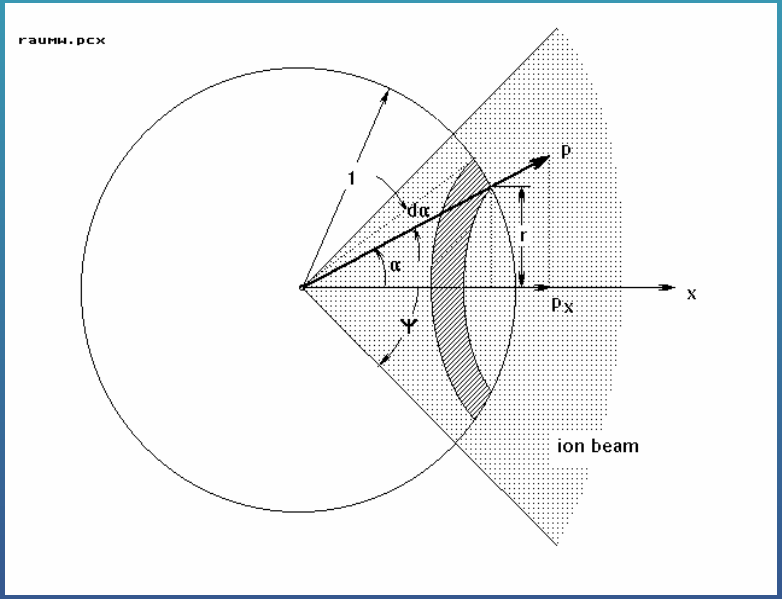
- **e_0** fundamental charge
- **n_c** number of fund. charges
- **$q = n_c . e_0$** charge of exhaust particle
- I emission current
- U acceleration voltage
- **m** mass of exhaust particle

Space Propulsion

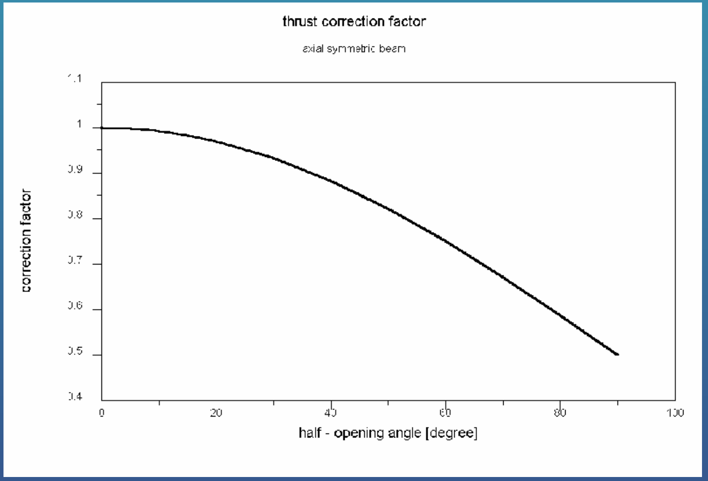


for the same ion current, thrust increases with particle mass/charge ratio

Space Propulsion



ion beam of finite current must have finite divergence angle (LIOUVILLE)



for beams with rotational symmetry and pointlike source

$$T = \iint_{\Omega} \frac{dT_x}{d\omega} d\omega = \iint_{\Omega} \frac{d^2 N}{dt.d\omega}(p.\cos\alpha) d\omega$$

when angular current density is independent on direction:

$$T = p.\frac{dN}{\Omega.dt}.\frac{\pi}{2}(1-\cos 2\psi) = \frac{dN}{dt}.p.\frac{1-\cos 2\psi}{4(1-\cos\psi)}$$

thrust of rotationally symm. beam with homogeneous current density distribution and beam opening half – angle Ψ

$$T \cong 1.4397 x 10^{-4} I \sqrt{\frac{MU}{n_c}}.\frac{1-\cos 2\psi}{4(1-\cos\psi)}$$

Space Propulsion

mass flow

$$\overset{\bullet}{m}_i = m.\frac{dN}{dt} = \frac{m_0 M}{n_c.e_o}.I = 1.0364x10^{-8}\frac{M}{n_c}I.\ [kg/s]$$

mass consumption:

$$\Delta m_i = \frac{m_0 M}{n_c.e_o}.\int_0^t I.dt\ \ [kg] = \frac{3.731x10^{-8}.M}{n_c}\left(\int_0^t I.dt\right)_{\mu Ah}\ \ [g]$$

1 µAh ≡ 4.283 µg
1 mg ≡ 233.47 µAh

for Indium (M =114.81)

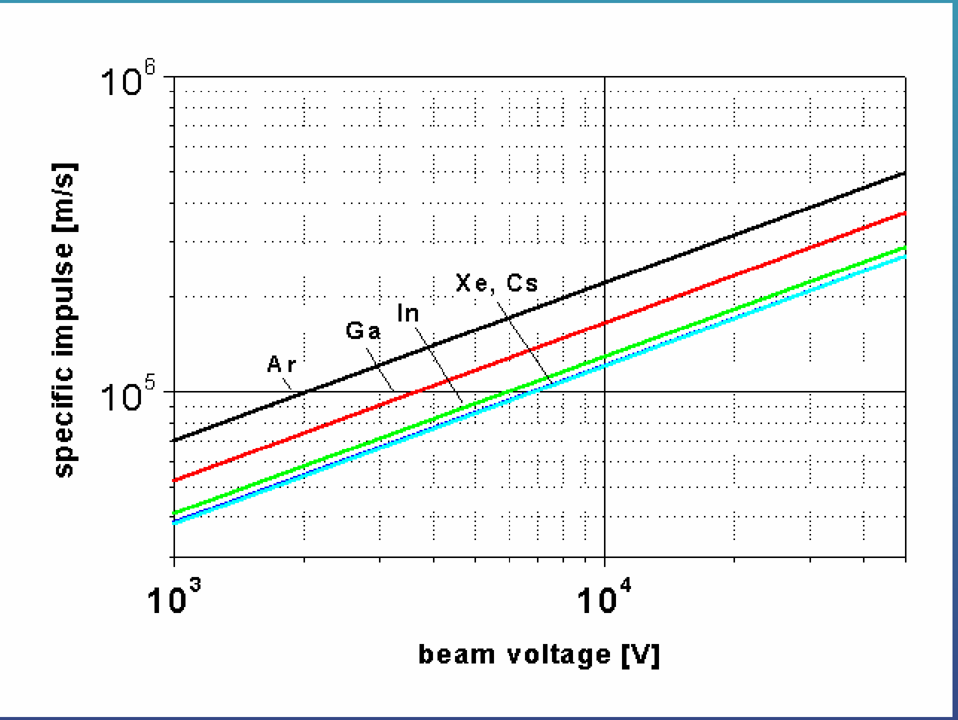
total impulse:

$$\Delta p = \int_0^t T.dt = 1.4397x10^{-4}\sqrt{\frac{M}{n_c}}.\int_0^t \sqrt{U}.Idt\ \ [N.s]$$

specific impulse:

$$I_{sp} = V_e = 1.3891x10^4\sqrt{\frac{n_c U}{M}}\ \ [m/s]$$

Space Propulsion

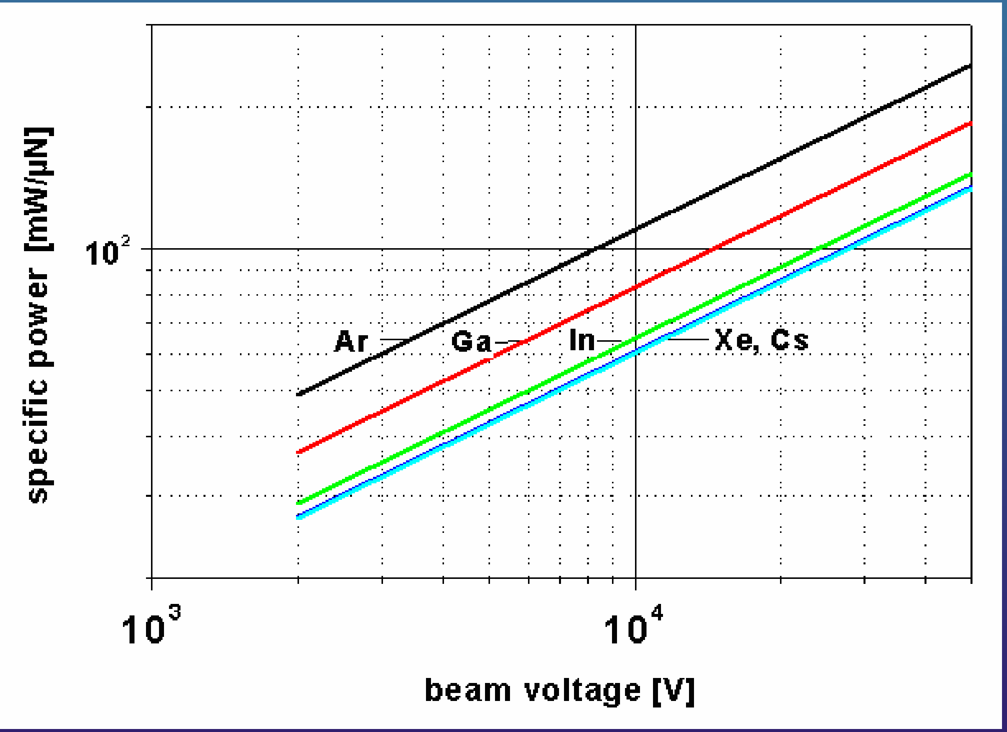


specific impulse of gaseous and metallic propellants used in ion thrusters; singly charged ions considered only

Space Propulsion

jet power: $P_j = U.I \;\; [W]$

specific power: $P_{sp} = \frac{V_e}{2} \cong 6.9458x10^3 \sqrt{\frac{n_c U}{M}} \quad [m/s], \;\; [W/N]$



Space Propulsion

Comparison of specific powers for FEEP and colloid thrusters

$$P_{sp} \cong 6.9458x10^{3}\sqrt{\frac{n_c U}{m}} \sim \sqrt{\frac{q}{m}}$$

$$\left(\frac{m}{q}\right)_{FEEP} = \frac{M.m_0}{e} \cong \frac{115*1.6x10^{-27}}{1.6x10^{-19}} \cong 1x10^{-6} \quad [kg/Cb]$$

for singly charged indium ions

$$\left(\frac{m}{q}\right)_{coll} \cong 1.7x10^{-2} \quad [kg/Cb]$$

data reported for BUSEK thruster

Specific power therefore is higher by a factor of ~ $\sqrt{1.7x10^{-2}/10^{-6}} \sim 130$ for the FEEP than it is for the colloid thrusters. The same factor obviously applies for the specific impulse. The FEEP, compared to the colloid thruster therefore will have a much lower mass consumption, but a much higher power consumption. As a consequence, the FEEP is to be preferred when mass is a limiting factor, the colloid thruster has advantages when power is limited.

Applications for ultrahigh precision ion thrusters

functional requirements:

- Ultrahigh precision pointing (< arcsec)
- Ultrahigh precision relative positioning (nm) of S/C clusters
- compensation of external forces and moments
  (air drag, solar pressure, etc.) to generate “dragfree” platforms onboard the S/C

Space Propulsion

Fundamental Physics Space missions

- Special and General Relativity
- Ultraprecise interferometry telescopes
- Exploration of Earth’s gravity field

Space Propulsion

test of Einstein's "Weak Equivalence Principle-WEP"

In a freely falling system all masses fall equally fast; hence gravitational acceleration has no local dynamical effects.

gravitational mass = inertial mass

Strong EP: *The outcome of any local experiment, whether gravitational or not, in a laboratory moving in an inertial frame of reference is independent of the velocity of the laboratory, or its location in spacetime*

$$F_g = m_g . g$$

$$\frac{d^2x}{dt^2} = \frac{F}{m_i}$$

L. Eötvös (1905): $\delta m/m < 10^{-8}$

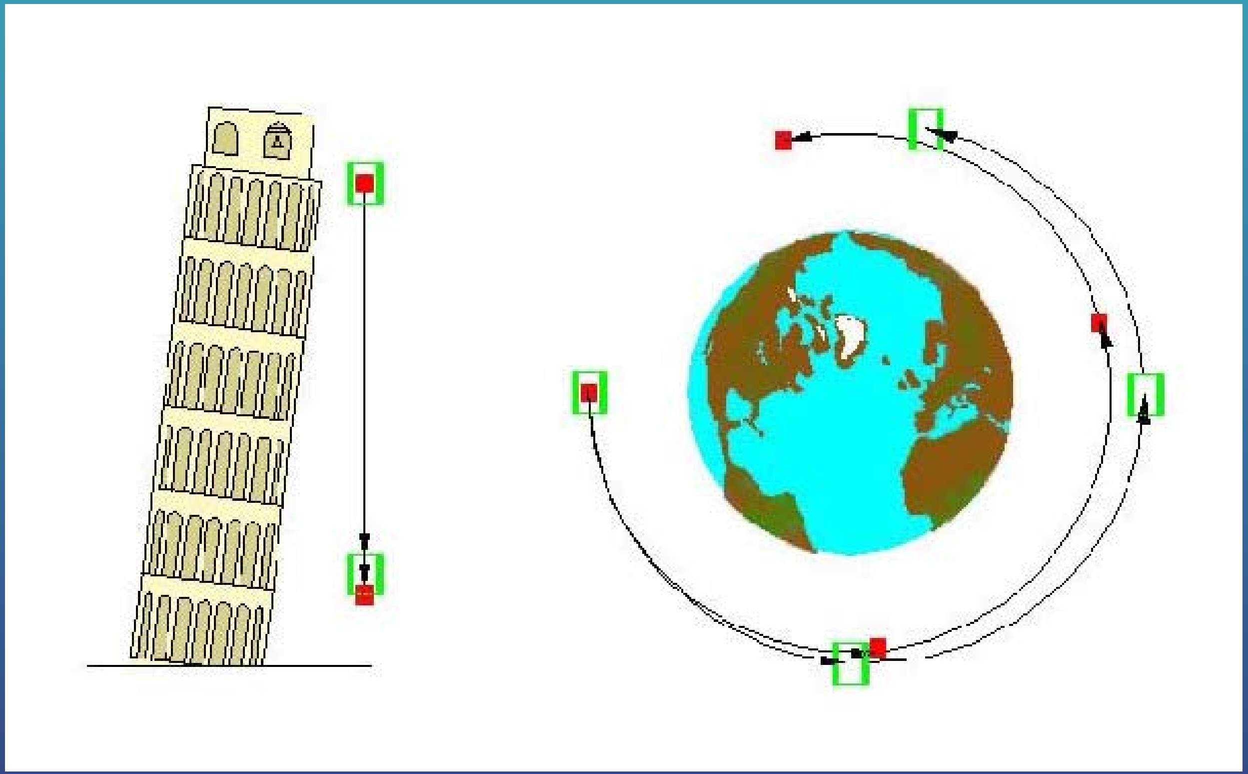
present status (1990-99): $\delta m/m < 10^{-13}$

Space Propulsion

Space missions for test of WE

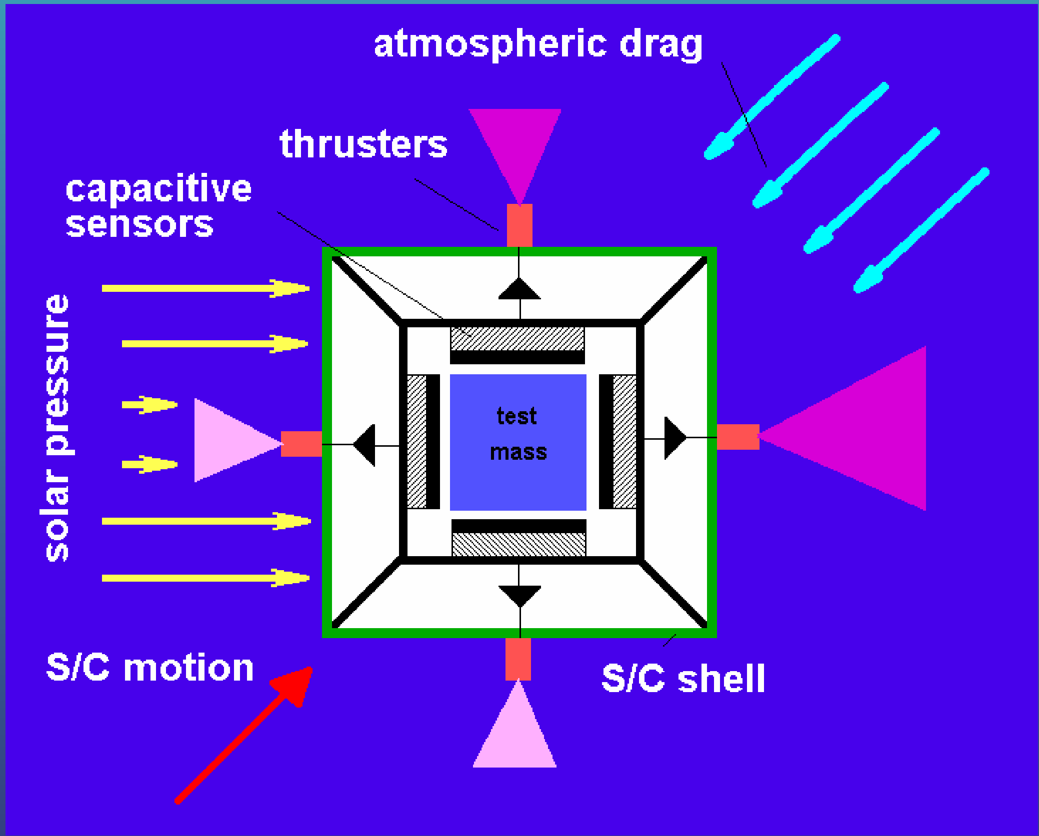
Acronym	name	agents	accuracy	status
STEP	**S**pace **T**est of **E**quivalence **P**rinciple	NASA ESA	10^{-18}	advanced study
MICROSCOPE	**MICRO S**atellite à traînée **C**ompensée pour l'**O**bservation du **P**rincipe d'**E**quivalence	CNES ESA	10^{-15}	decided; liftoff 2007?
GG	**G**alileo **G**alilei	ASI, U. Pisa	10^{-17}	study

Space Propulsion



Assume: m_g/m_i of **red material** is larger than that of green material; then on ground red material would **fall faster** than green material; in orbit around earth, red material will be displaced w/r to green material towards earth (dominance of gravity over centrifugal force) and therefore attain higher speed. Relative displacement between bodies will be **periodic** with orbital frequency (maximum displacement at $\delta m/m = 10^{-18}$ is $< 10^{-11}$ m). Multiple orbits reduce measurement noise. Both „test masses" however must be allowed to follow **purely gravitational trajectories** without external forces.

Space Propulsion



Dragfree control: external forces would force S/C onto nongravitational trajectory; capacitive sensors measure distances to free - floating **test mass**; if set distance is exceeded, appropriate **thrusters** are fired to keep distance between S/C shell and test mass constant. Therefore, S/C also will follow purely gravitational trajectory

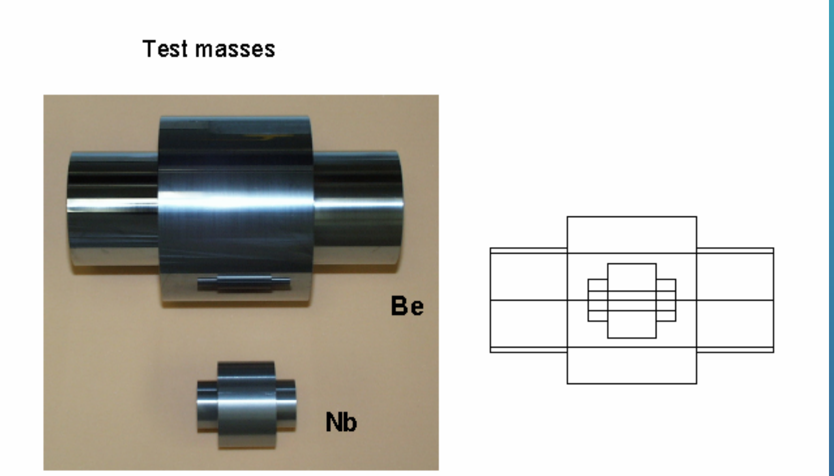
Space Propulsion

- mission requirements imply residual accelerations on S/C to be **< 10^{-11} g**
- for a 300 kg S/C the thrusters therefore must be accurate and stable to within $\delta T < 300 \times 10^{-11} \times 9.81 \sim$ **3×10^{-2} μN**
- presently, these requirements can be approximately fulfilled only by **FEEP** and **Colloid** thrusters

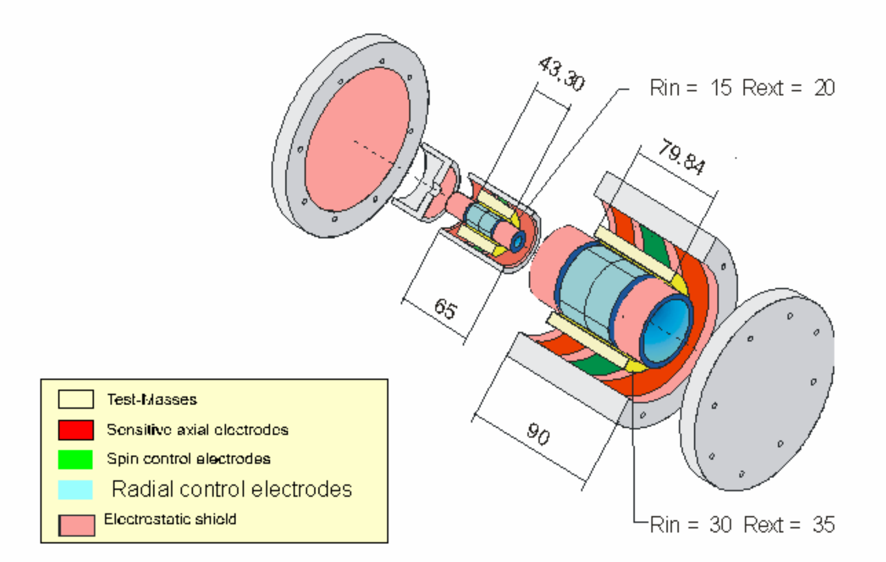
Conditions on GG spacecraft

Dimensions of S/C	R=50cm x1.3 m high
Radiation pressure	4 μN/m^2
Atmospheric drag	65 μN @ 520 km altitude
Differential displacement	
due to EP violation at 1 /10^{17} level	6.3×10^{-13} m: @ $\nu_{orb}=1.75\times10^{-4}$ Hz
air drag	3.9×10^{-13} m @ p_{orb} @1×10^{-9} torr
solar radiation	3.9×10^{-15} m @ $\nu_{spin}=2$ Hz

Space Propulsion



Inner and outer test masses of one STEP-Differetial-Accelerometer. The test masses are arranged into one another (sketch on the right side). STEP contains four Differential Accelerometers.

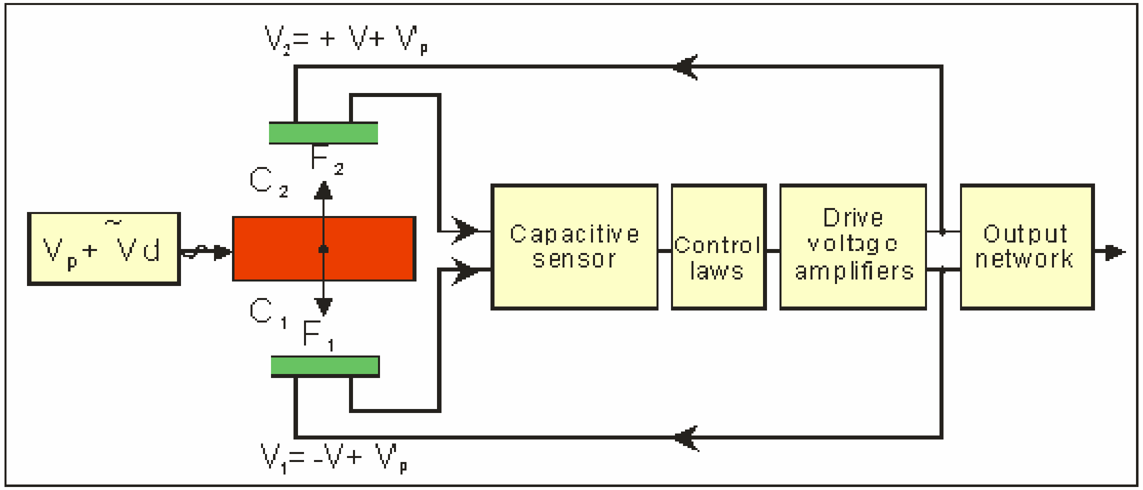


- WE is possibly violated by different coupling of nuclear forces in materials with different proton / neutron ratios
- candidate material pairs therefore consist of a high – Z and a low – Z element, e.g. Nb / Be (STEP) or Pt-Rh/ Ti (MICROSCOPE)

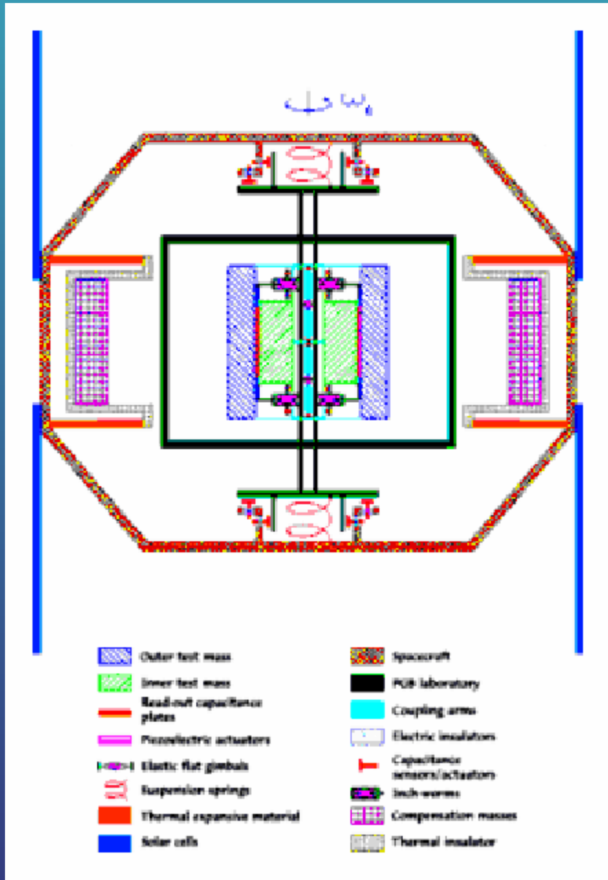
Space Propulsion

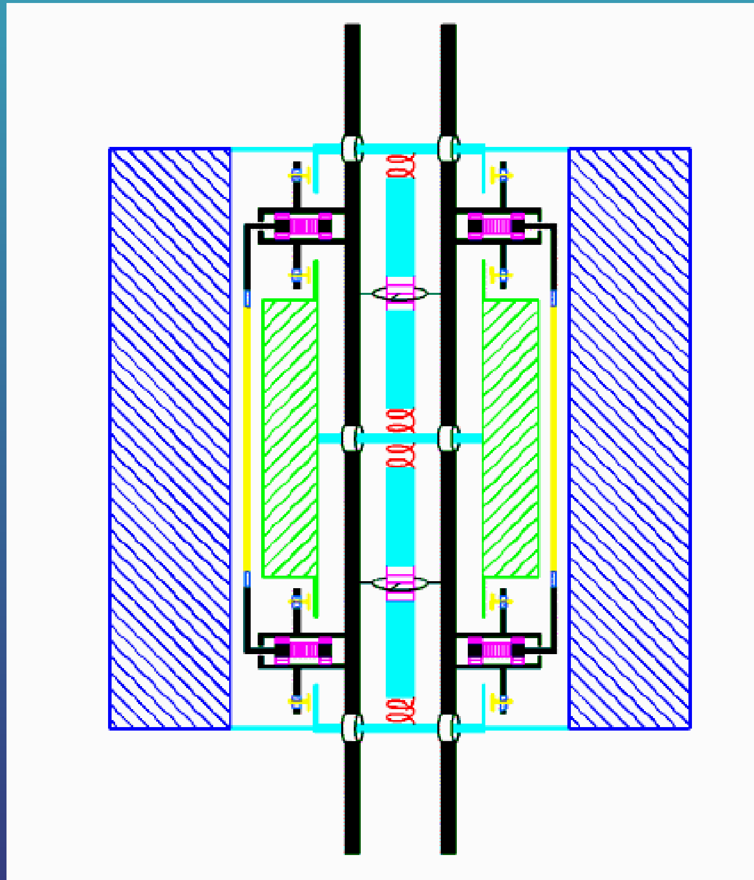
Electrostatic Accelerometer Principle | Capacitive Acceleration Sensors

- Proof-mass: motionless with respect to the cage
- Position detection: capacitive sensors with high resolution
- Actuators: Electrostatic levitation
- Measurement: from restoring voltage => tri-axial acceleration of SC



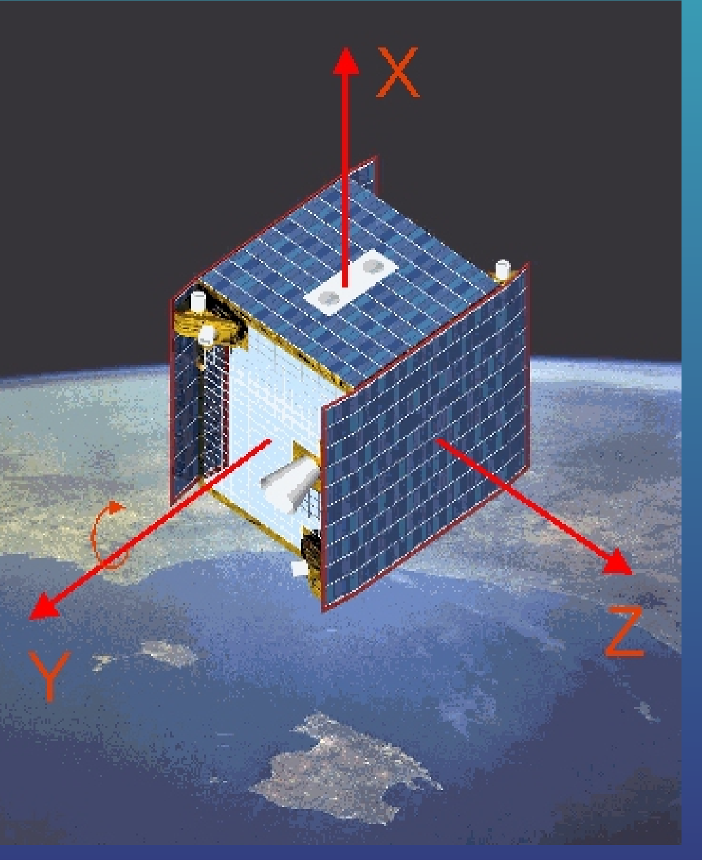
Space Propulsion





arrangement of test masses (right) on GG satellite (left)

Space Propulsion



MICROSCOPE S/C

FEEP thrusters contracted by ESA to CENTROSPZIO / ALTA (Pisa); project on halt

Space Propulsion

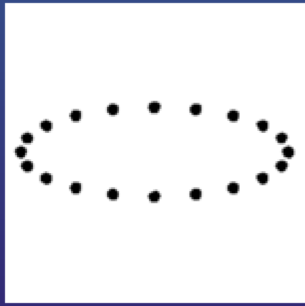
Detection of low frequency gravity waves

gravitational waves are radiated by objects whose motion involves acceleration, provided that the motion is not perfectly spherically symmetric (like a spinning, expanding or contracting sphere) or cylindrically symmetric (like a spinning disk).

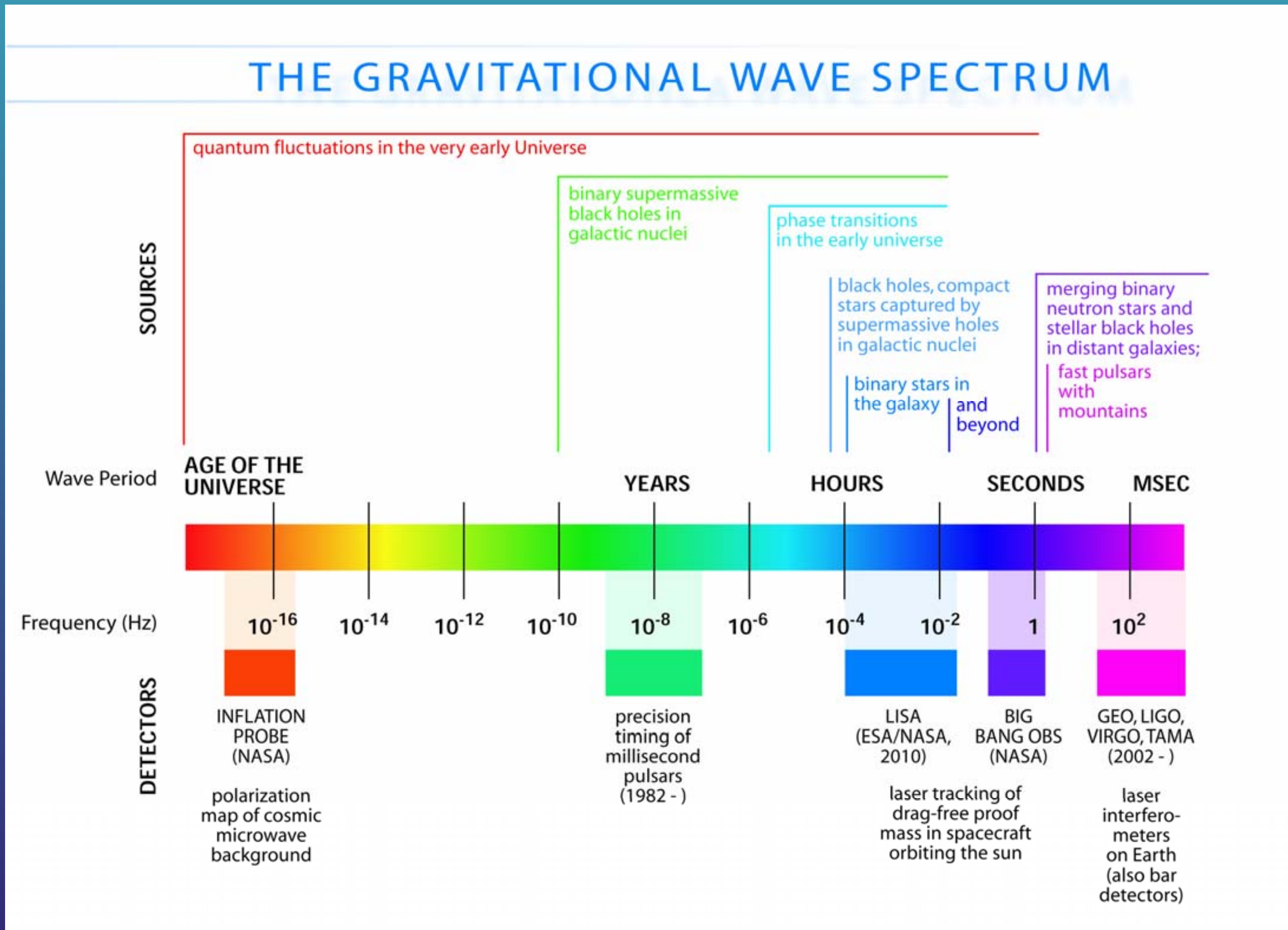
sources of gravitational waves

- two objects orbiting each other in a quasi-Keplerian planar orbit (e.g. binary star systems)
- a spinning non-axisymmetric planetoid — say with a large bump or dimple on the equator
- a supernova ***will*** radiate except in the unlikely event that it is perfectly symmetric

locally, passing gravitational waves change curvature of Spacetime in the frequency with which they are emitted; distance between objects will change periodically:



Space Propulsion

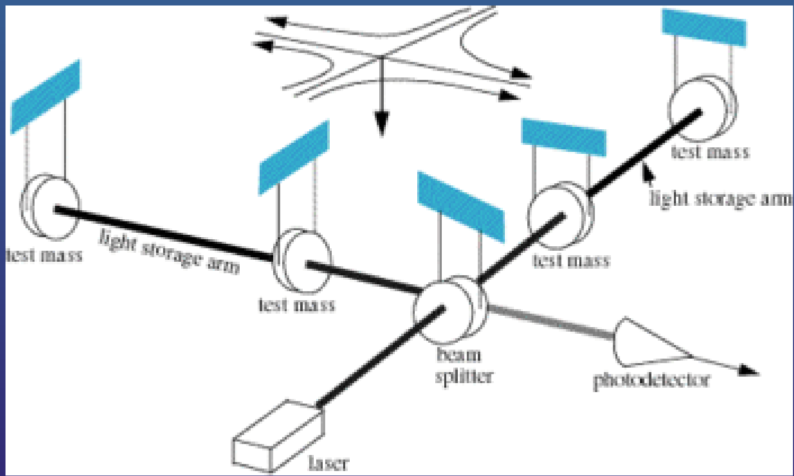


Space Propulsion

- power radiated away from a binary system of 2 masses M_1, M_2, orbiting each other at distance R: $$P = -\frac{32}{\pi}\frac{G^4}{c^5}\frac{(M_1 M_2)^2(M_1+M_2)}{R^5}$$
- power radiated by Sun / Earth – system: **313 [W]**
- expected distortion of space from largest cosmic sources at earth orbit: **10^{-18} [%]**

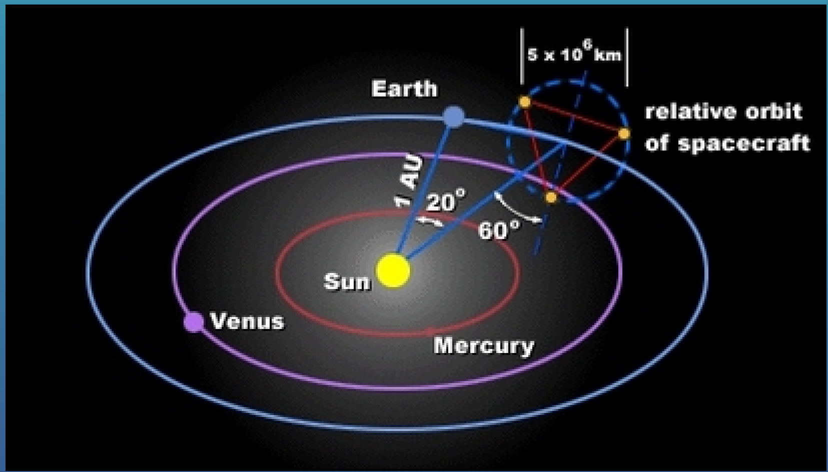
Ground – based detectors (VIRGO, LIGO) are Michelson interferometers of **2 km** armlength; expected wave amplitude: < **10^{-17} m**
ultimate sensitivity: < **$5x10^{-22}$ m**

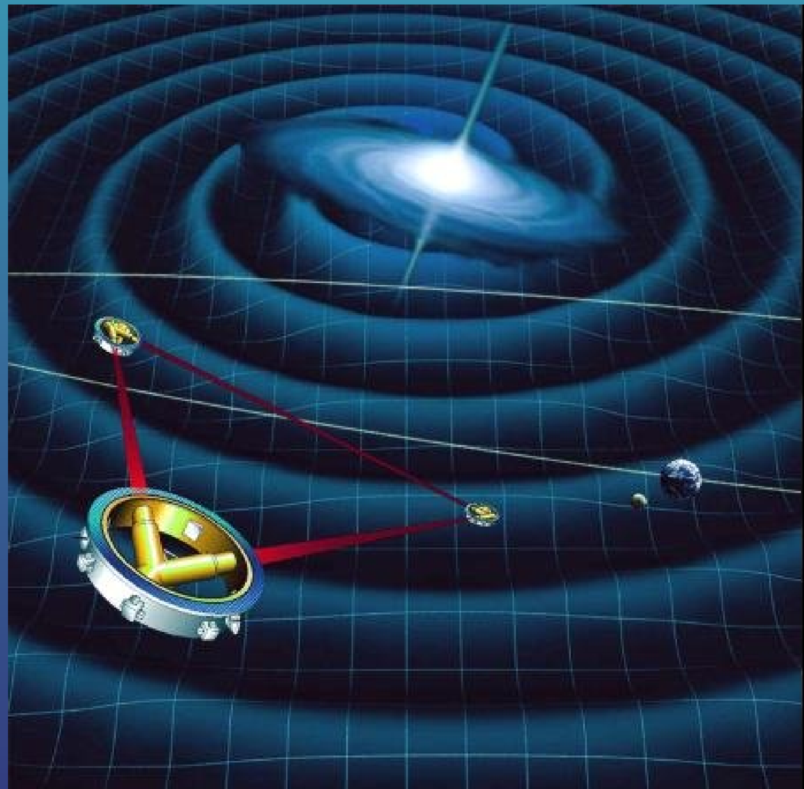
All ground based detectors are limited by seismic noise at low frequencies (< 1 Hz);
→ **Low frequency** waves (as expected for most double – star systems) can only be measured In **space – based systems**



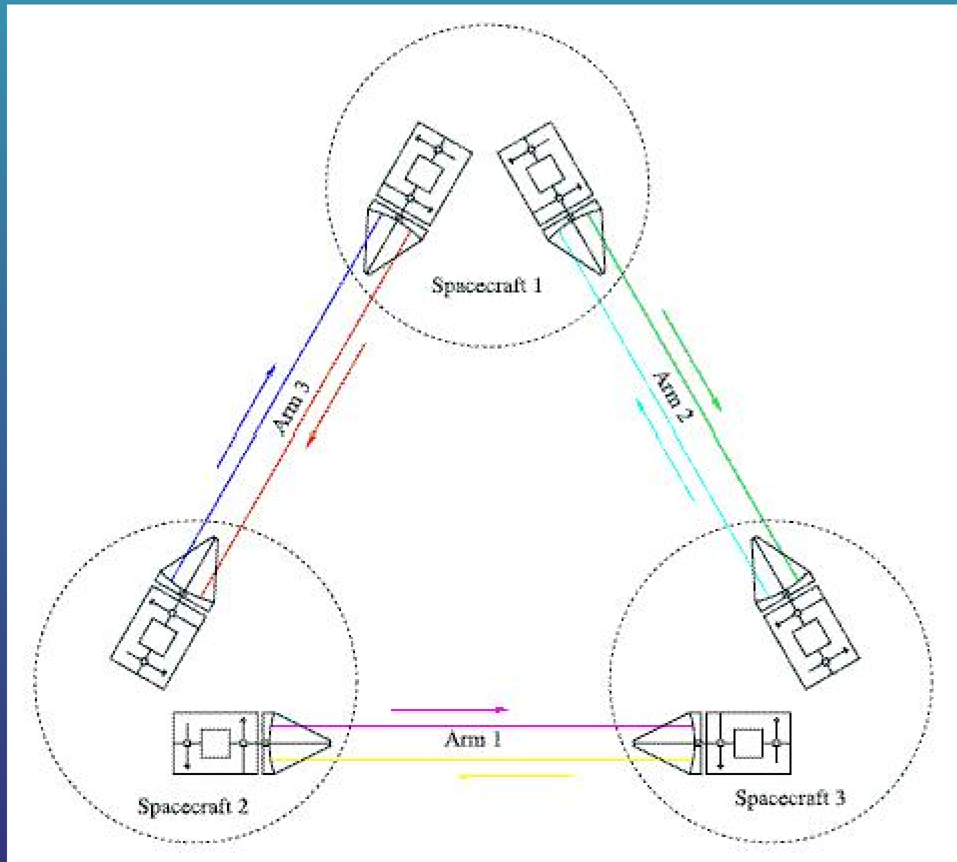
Space Propulsion

LISA (Laser Interferometer Space Antenna)





Space Propulsion



Space Propulsion

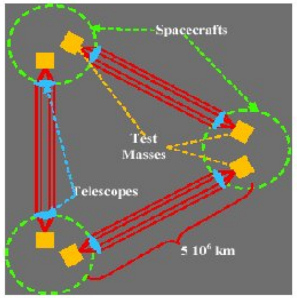
Satellite launch mass	Approximately 460 kg each (fully fuelled) including the science payload, ion drive, propellants and spacecraft adapter.
Science Payload mass	288 kg
Dimensions	Main spacecraft diameter 1.8 m, height 0.48 m
Total power	284 W

planned launch date:	2015
mission duration:	5 years

Space Propulsion

LISA mission summary

- Galactic sources are predicted to emit gravitational waves in frequency range 0.1mHz to 0.1 Hz with strain of ~ 10^{-21}
- The 3 LISA spacecraft form an equilateral triangle of side 5 million km. The Inter-satellite path length measured using interferometry techniques
- **=> LISA must detect path length changes of 50 pm**
- Acceleration noise inside the frequency measurement band is indistinguishable from a gravitational wave disturbance.
- **=>disturbance reduction mechanism must shield the proof mass from the outside environment in such a way that only gravity waves will cause measurable displacements.**



FEEP requirements for LISA satellite

- **Micro thrusters compensate for secular forces on the satellite.**
- **For solar radiation pressure at 4.5 .10-6 Nm-2, then continuous effective thrust will be in the region of 10µN (for 2 m2 projected area spacecraft).**
- **Additional torque is required to compensate for the offset between solar radiation centre of pressure and c.g.**
  - For 50mm offset, 2m2 projected area, torque is 0.45 µ Nm
- **Resolution/noise requirement for the thrust is around 0.1µN.**
- **Total velocity increment ~ 220 Nsec for 6 months operations**
- **Thrust level 0-20 micro newtons**
  - Step size 0.1-1 microNewton
  - Noise specification
- **4 clusters of 4 thrusters, providing functional redundancy**
  - No single failure to cause "loss" of mission

Positional noise	< 50 pm
Solar pressure:	~ 4.5 µN/m^2
Thrust level	~ 0 – 20 µN
Thrust step	0.1 – 1 µN
Thrust noise	< 0.1 µN/Hz$^{1/2}$
Total impulse	~ 2200 N.s (1/2 y)

$m_p = I_{tot}/I_{sp}$ = 2200/8x10^4 kg ~ 27.5g

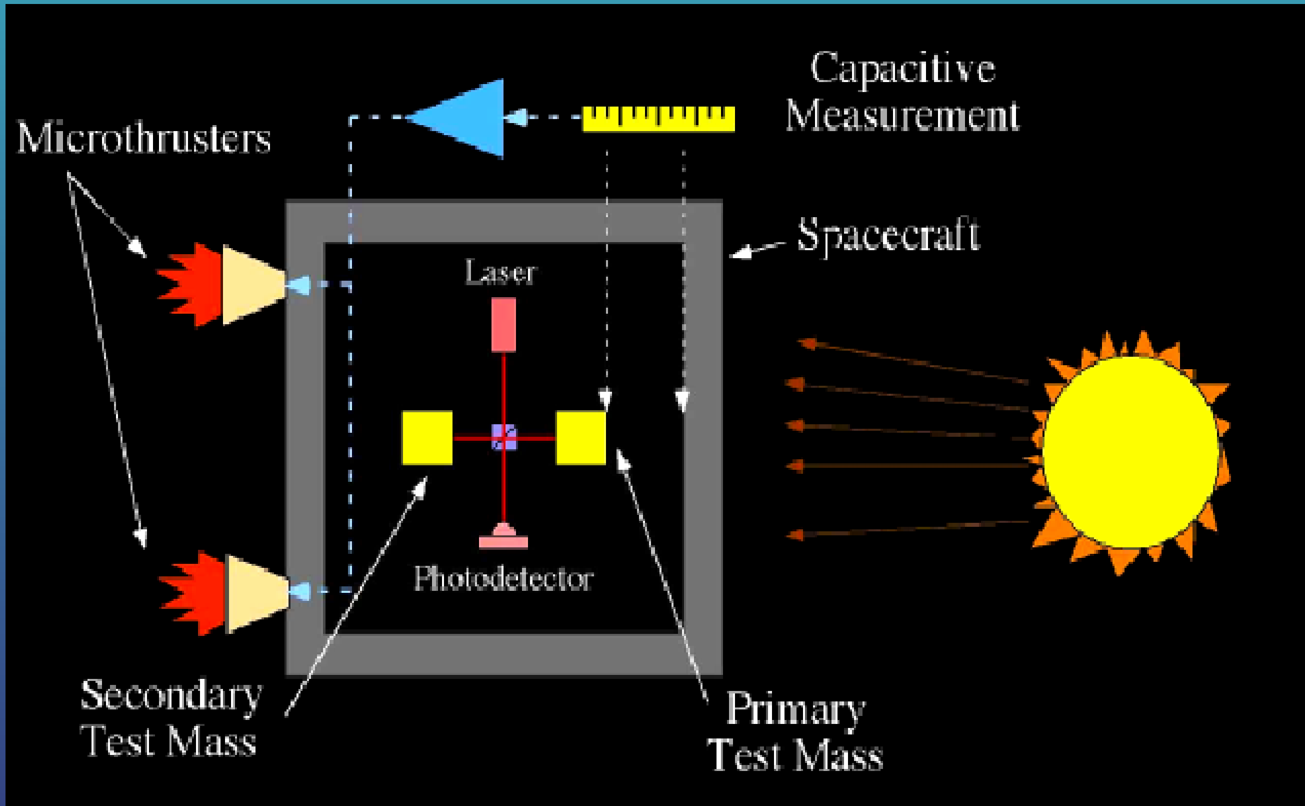
Space Propulsion

LISA Pathfinder (LPF): ESA / NASA mission to test key technologies for LISA

Importance of LPF to LISA

- LISA technologies being flight-tested:
  - Complete disturbance reduction system
  - Two-axis drag-free operation
  - Some aspects of interferometry
- A complete test of a LISA-like disturbance reduction system in a flight environment
  - How LISA-like are the equipment, the operational modes, the environment, etc.
  - Fidelity of the test (level of integration)
- Validation of performance models that can be used to extrapolate about a factor of 10 to LISA performance
- Opportunity for NASA and ESA to work closely together on LISA technologies prior to the main LISA mission

Space Propulsion



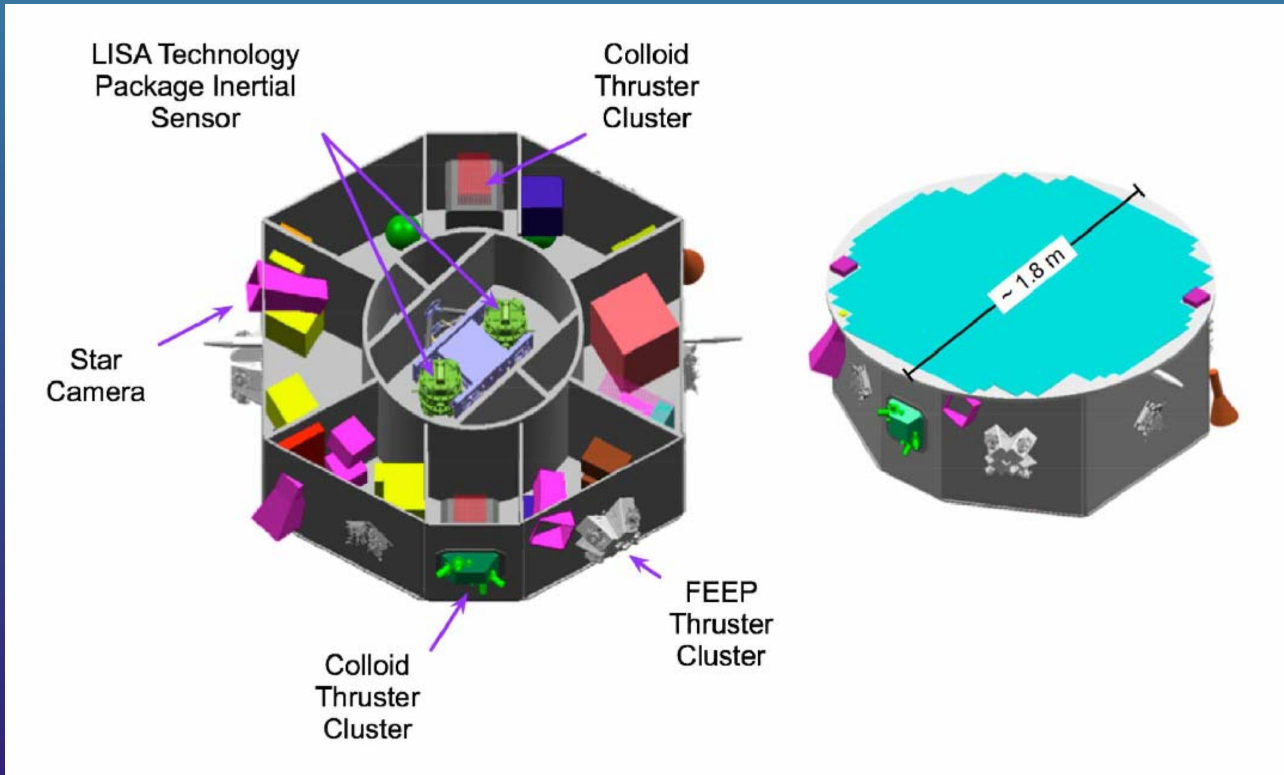
Dragfree operation of LPF: distance S/C – test mass 1 is capacitively sensed and fed back to microthrusters

Distance measurement between test masses 1 and 2 (Science operation): laser , beamsplitter, testmasses 1 &2 and photodetector from Michelson Interferometer with measurement accuracy ~ 1 pm

Space Propulsion

Experiments onboard LPF

LISA Technology Package (LTS):	laser, interferometer, FEEP thrusters (ESA)
Disturbance Reduction System (DRS):	colloid thrusters (NASA)



Space Propulsion

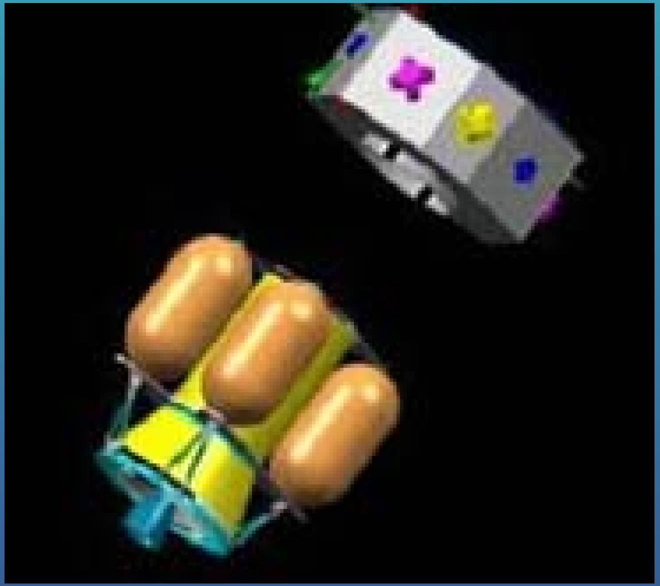
Missionsdaten und technische Parameter von LISA Pathfinder:

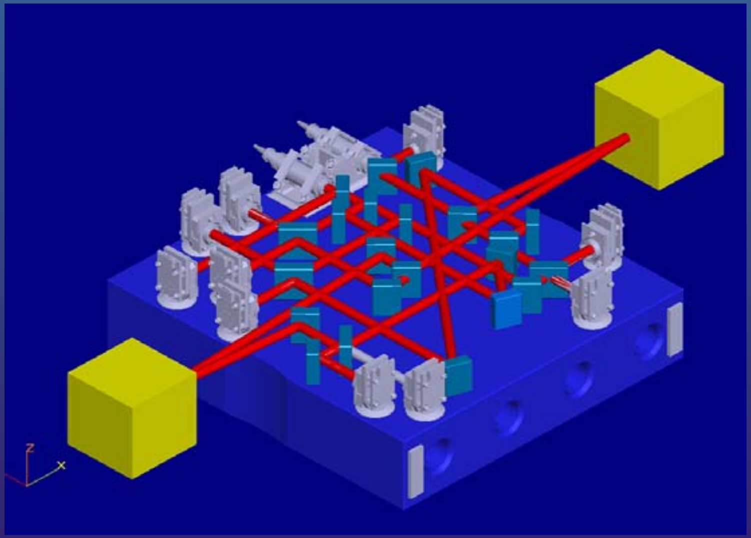
Start:	März 2010 vom Startplatz Plesetsk (Russland) oder Kourou (Franz. Guayana)
Trägerrakete:	Eurockot oder Vega
Orbit:	Halobahn um den Lagrange-Punkt L1 von Sonne/Erde, Abstand von der Erde etwa 1,5 Mio. km
Nominelle Missionsdauer:	12 Monate, davon 6 Monate im „Drag-free“ Betrieb
Masse der Sonde:	475 kg Nutzlastmodul/1900 kg Startmasse
Äußere Abmessungen der Sonde (Nutzlastmodul):	2,1 m x 1,0 m
Masse des LTP:	125 kg
Abmessungen des LTP:	64 cm x 38 cm x 38 cm
Elektrische Leistungsaufnahme:	typ. 150 W
Telemetrierate der Sonde:	1,7 kbit/s (X-Band)

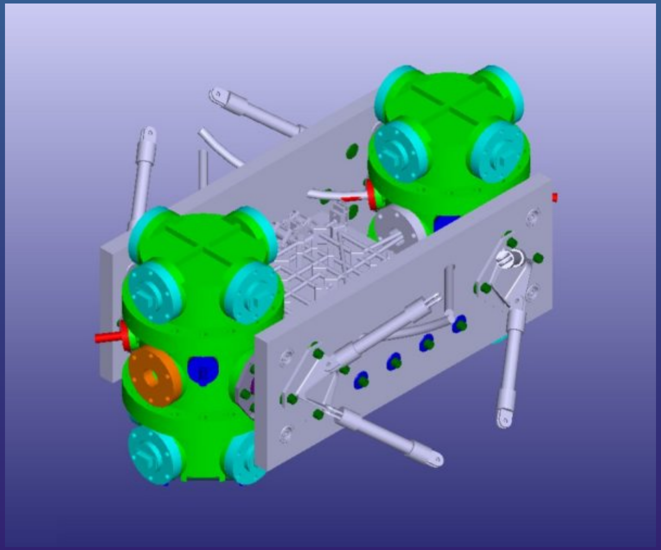
Space Propulsion

LISA Pathfinder - Facts and Figures

Mass of Science spacecraft	480 kg
Propulsion module including fuel	1420 kg
Total Launch mass	1900 kg
Dimensions	2.9 m high by 2.1m dia
Launch date	Mid-2008



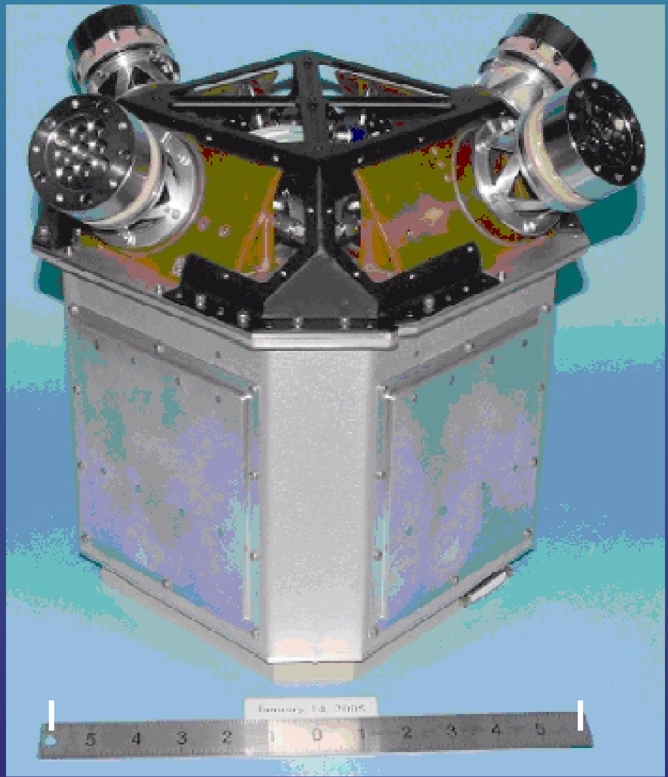




Space Propulsion

DRS

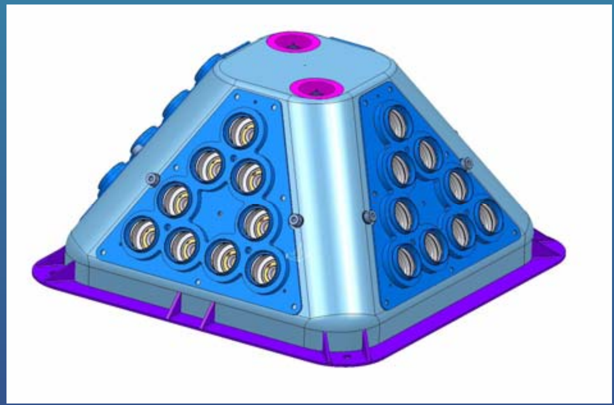
COLLOID, including reservoir and PS
BUSEK/Mass.



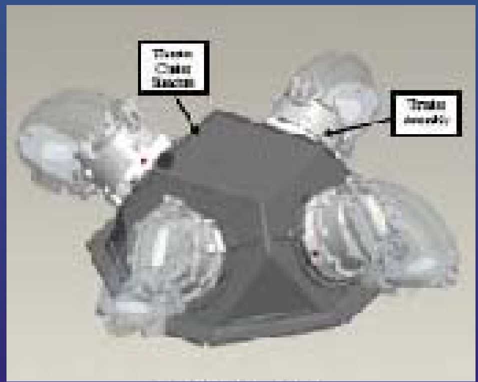
25 cm

LTP

FEEPs, including reservoir,
without PS; two candidates as suppliers:



ARCS
In – needle cluster



CENTROSPAZIO/
ALTA
Cs slit emitter

Space Propulsion

LPF specifications (2005)

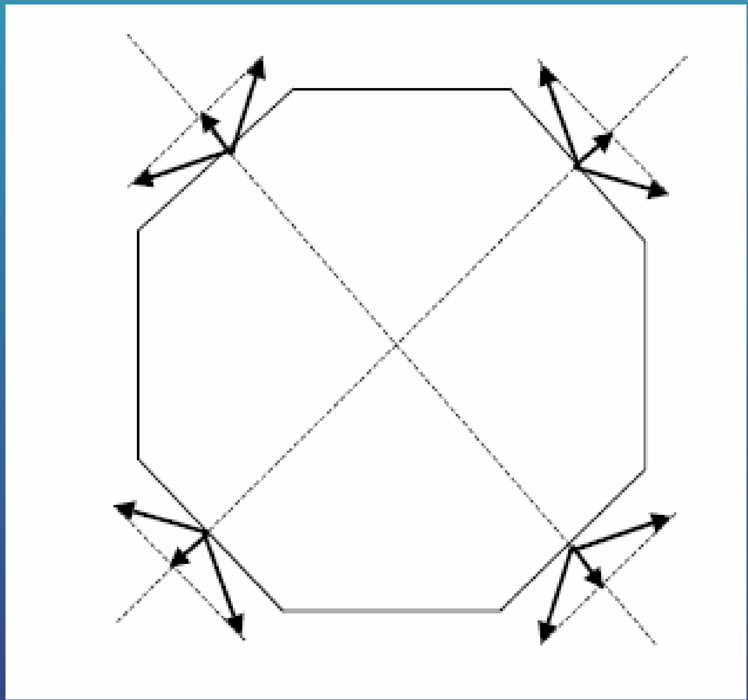
parameter	spec	comment	Section in ref. 1
Thrust profile	Tbd		3.1.1.
Thrust update rate	> 10 Hz		3.1.2.
Thrust stabilisation loop	~100 Hz		3.1.2.
Operational lifetime	510 days	after 3.5 y ground storage and testing	3.1.3.
Ground storage	no preventive maintenance requ'd		3.1.6.
Operating temperature range	-20 to +60 ^{0}C		3.1.7.
Minimum thrust	0.3 μN (0.1 μN)	in brackets: targeted	3.3.1.
Maximum thrust	> 75 μN (> 100 μN)	in brackets: targeted	3.3.1.
Thrust resolution	< 0.3 μN (< 0.1 μN)	@ < 150 μN	3.3.2.
	< 1 μN	@ > 150 μN	
Thrust noise	less than envelope		3.3.3.
STD (thrust) over 72 h	< 5.8×10^{-6} N		3.3.3.

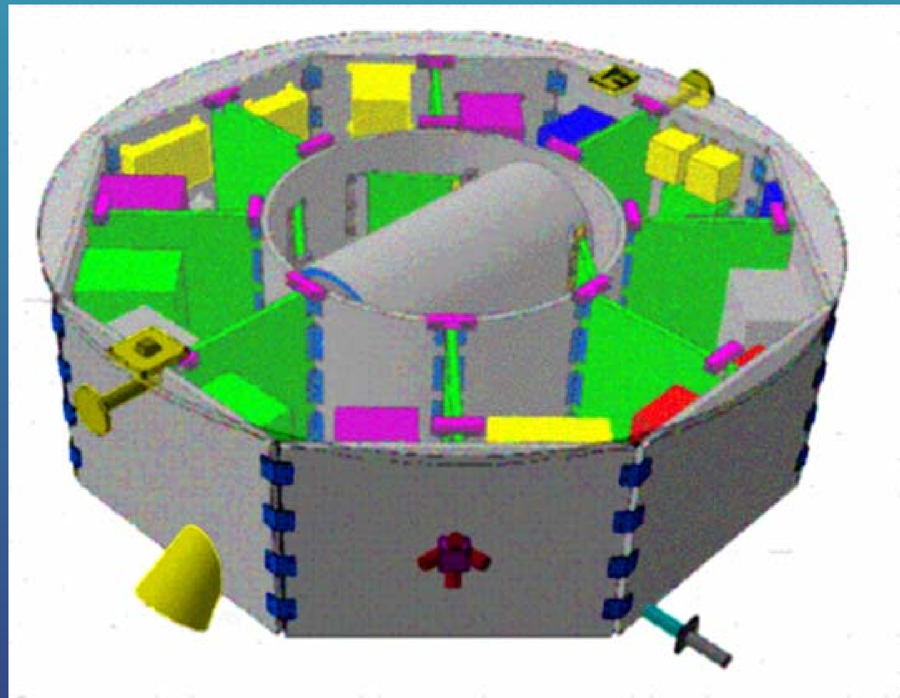
Space Propulsion

LPF specs, cont‘d

Response time (thrust)	< 100 ms (95%)	for step change < $\pm$ 30 μN	3.3.4., 3.3.5.
Thrust accuracy	< 2 μN	over full thrust range; deviation of actual from commanded thrust	3.3.6.
Thrust local linearity d(deliv’d)/d(commanded)	< 0.5% (3σ)	over full thrust range	3.3.7.
Thrust repeatability	< 0.5 μN $\pm$ 0.5% (3σ)	over full thrust range	3.3.8.
Specific impulse	> 4000 s	over full thrust range	3.3.9.
Total impulse	> 2460 N.s		3.3.10.
Leak force	< 5×10^{-8} N (3σ)	at 0 commanded thrust	3.3.11.
Thrust vector accuracy	< $5^o \pm 0.5^o$	throughout lifetime	3.3.12.
Beam divergence	< 50^o	for 99% of beam current	3.3.13.
Model predictions for specific impulse and thrust	+ 1% (3σ)	deviation of prediction from acceptance test	3.3.14.
Heater power	< 12W	for 4 thrusters	3.6.6.
Heatup time of reservoir	< 60 min		3.4.3.
Mass of FEEP cluster	< 4.5 kg	for 4 – thrusters - cluster	3.8.1.1.

Space Propulsion

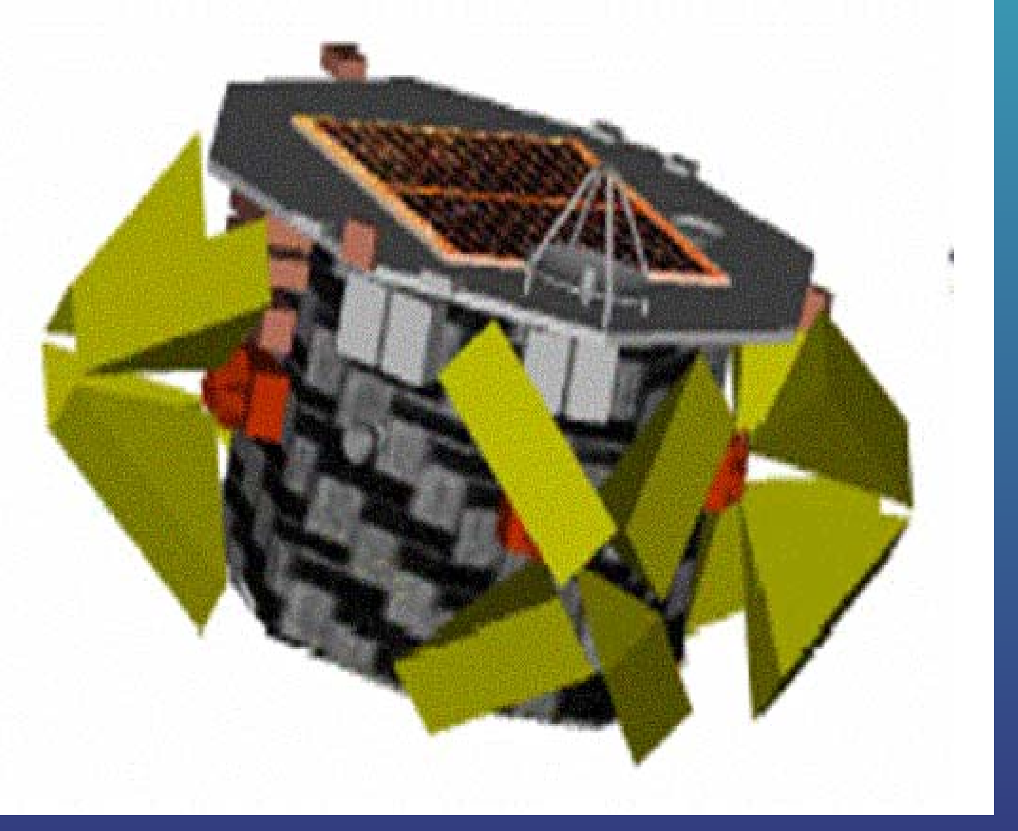




arrangement of the 4 clusters of 4 thrusters, (SMART-2, CASA-Study)

FEEP Thruster Arrangement, SMART-2

Space Propulsion



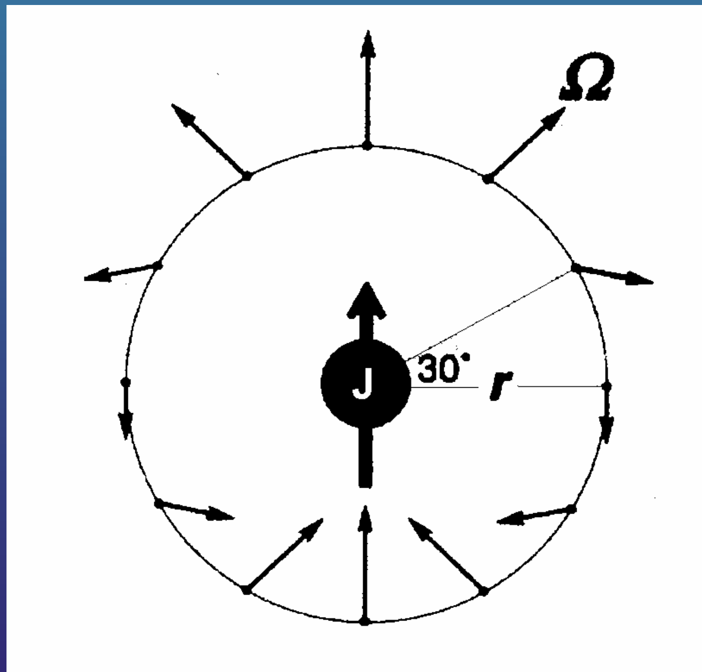
FEEP Thruster Arrangement (FEEP thrusters shown red; ion beam width shown yellow); ion beam must not hit S/C parts

Test of Relativistic Frame Dragging (HYPER)

- to test General Relativity by mapping for the first time the spatial (latitudinal) structure (magnitude and sign) of the gravitomagnetic (frame-dragging or Lense-Thirring) effect of the Earth with about 3-5 % precision
- · to independently from Quantum Electrodynamics theories (QED) determine the fine structure constant by measuring the ratio of Planck's constant to the atomic mass one to two orders of magnitude more precise than present knowledge
- · to investigate various distinct sources of matter-wave decoherence as required for an upper bound of quantum gravity effects

Space Propulsion

In the approximation for slowly rotating objects, general relativity (GR) predicts a dipole distortion of the gravitational field near a rotating body <cite>ref123</cite> which is formally analogous to the magnetic field in electrodynamic and sometimes is called the "gravitomagnetic field"

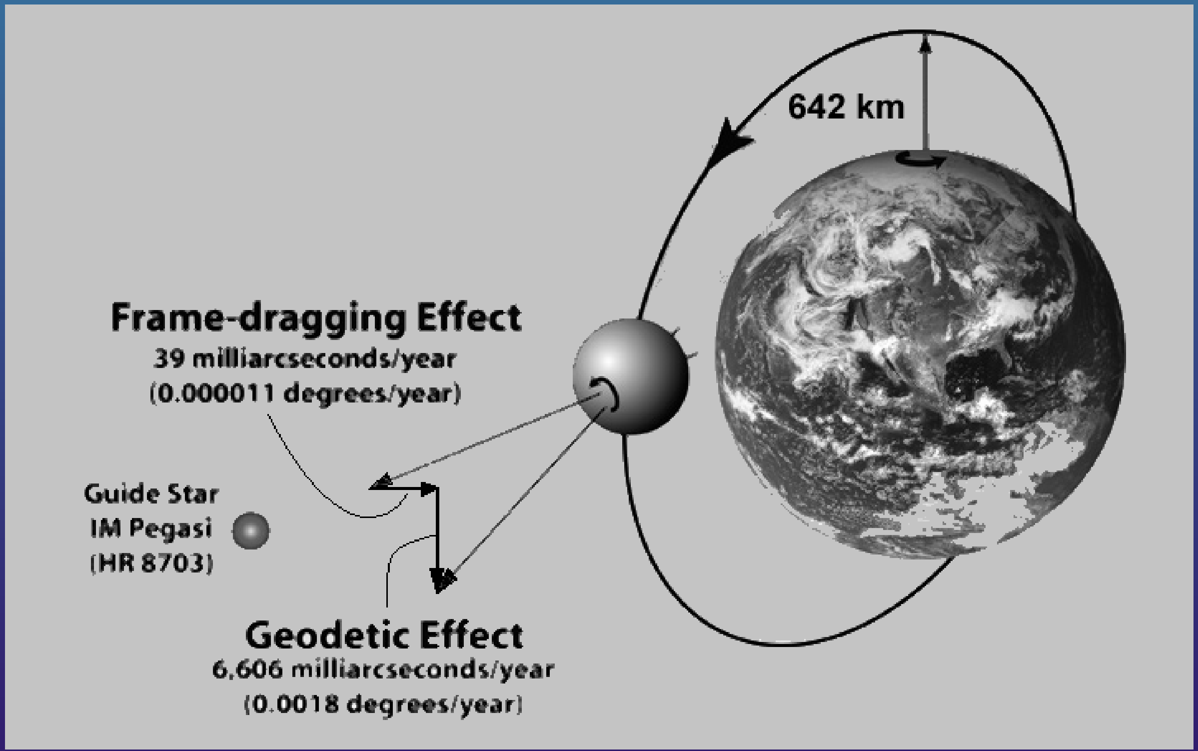


Space Propulsion

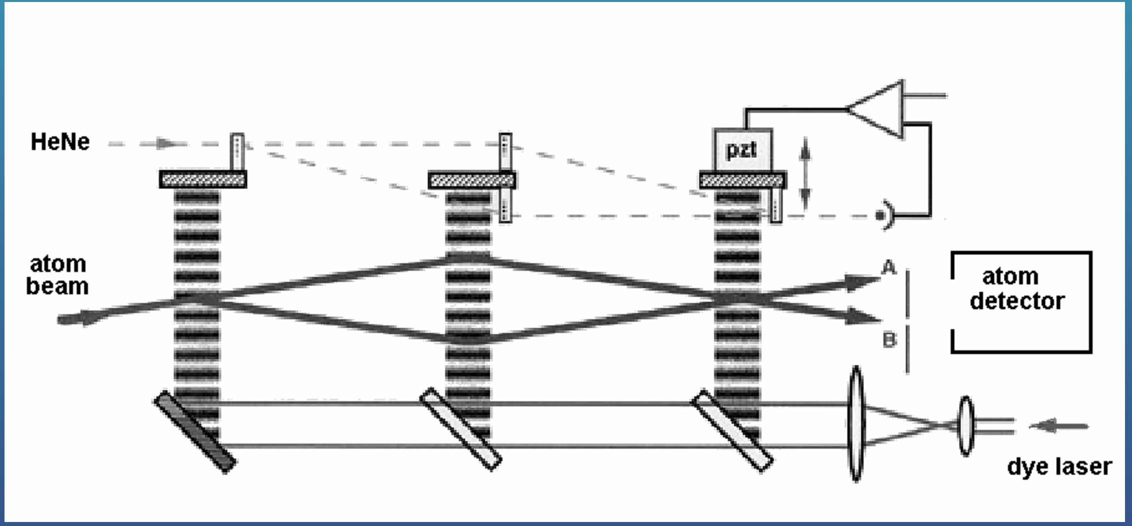
➢ HYPER aims for a further increase of precision of the gyroscope system which would allow to measure the latitude - dependency of gravitomagnetism and thus test in all details the predictions of GR. This is only possible by

➢ changing from physical gyroscopes (quartz spheres in GP - B) to novel quantum interference gyroscopes employing cold atom lasers

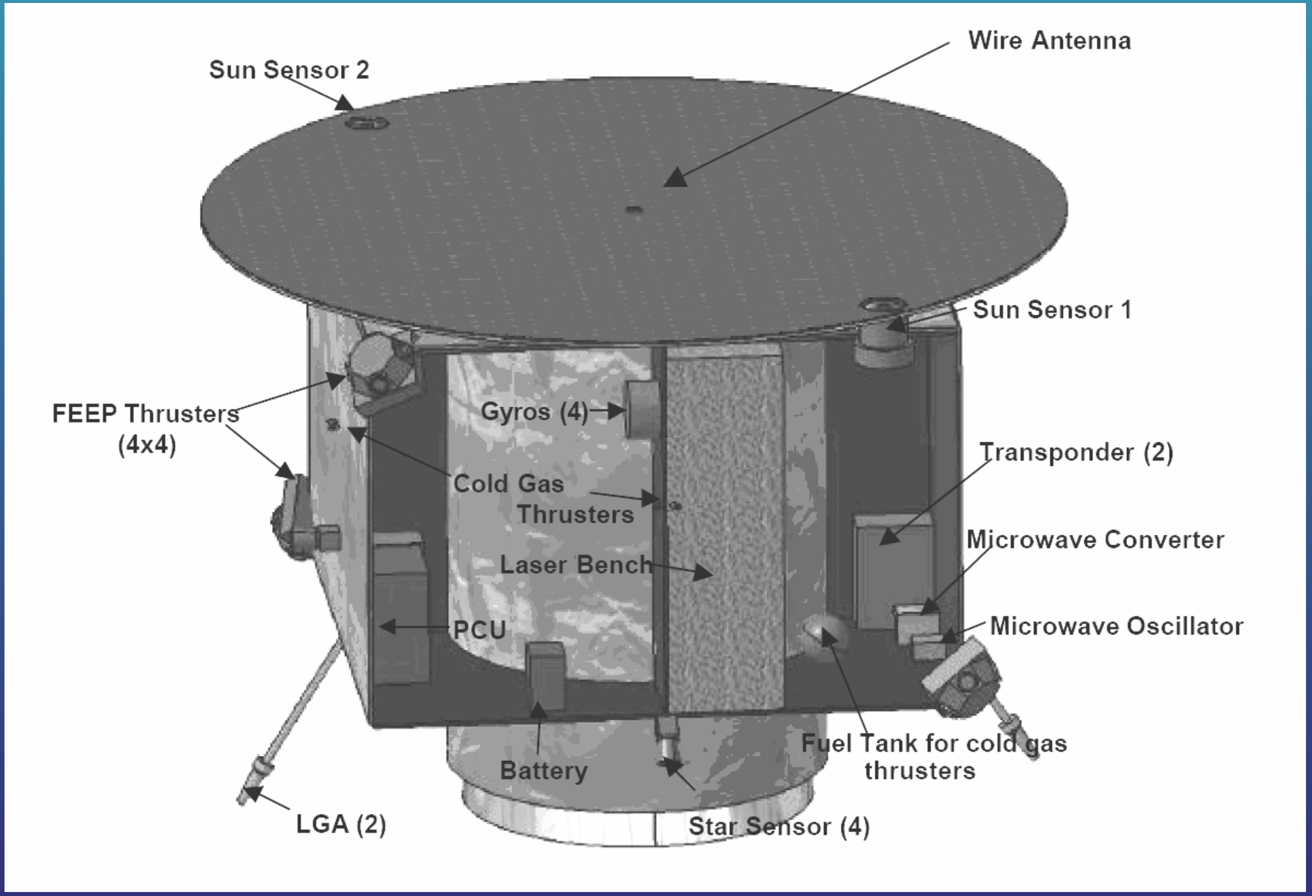
➢ reducing residual accelerations of the dragfree platform which is aligned to the guide star by using **ultraprecise FEEP thrusters** as dragfree actuators.



Space Propulsion

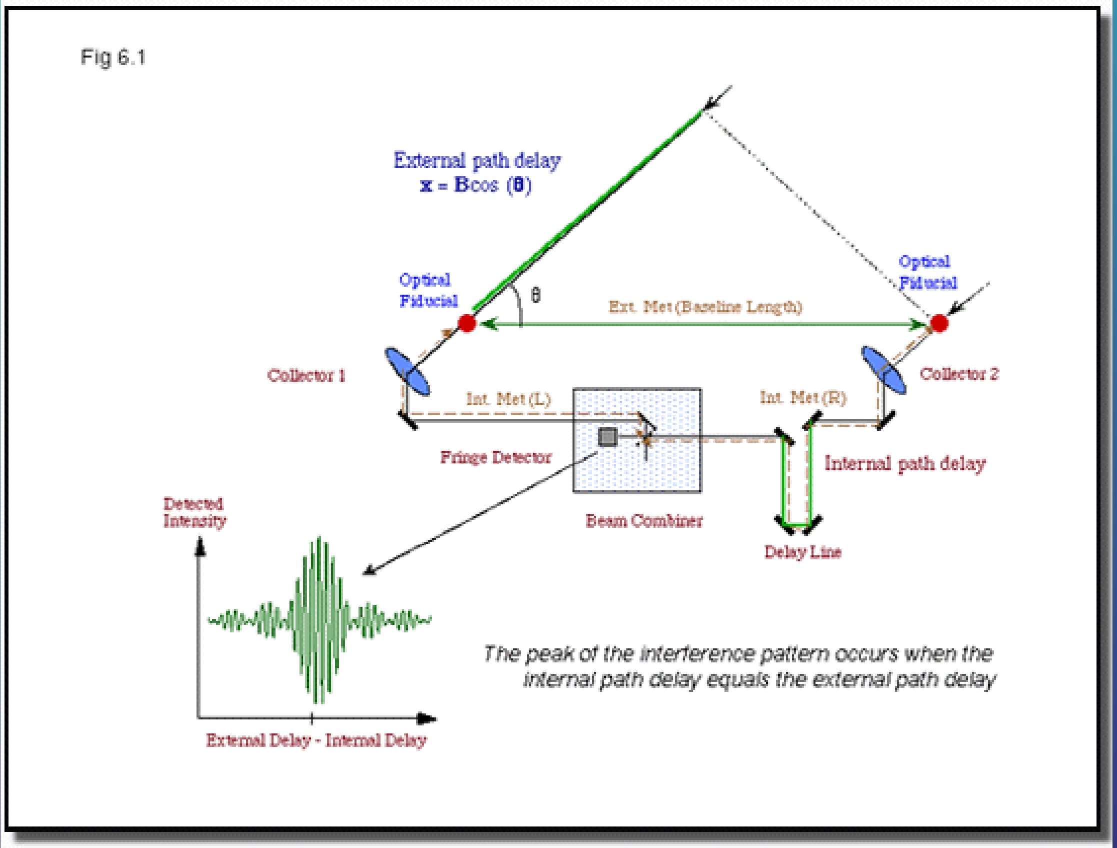


Space Propulsion



Space Propulsion

Interference telescopes (VLB)



Space Propulsion

DARWIN mission

- **DARWIN mission is to survey stars using a space based interferometer.**
  - Flying satellites in formation will enable long baseline interferometry to be performed.
- **Formation must have ability**
  - to maintain precision formation
  - to select different inertial targets (=> slew spacecraft)
  - to change resolution and FoV (=> expand/contract)
  - to resolve image in several directions (=> rotate around line of sight)
- **Position control for DARWIN is executed in three stages:**
  - Coarse mode: RF Metrology system, relative position control and knowledge at ~ 1cm
  - Medium mode: Laser Metrology system, relative position control and knowledge at ~ 100 microns
  - Fine Mode: Optical Path Difference control (OPD) within payload, relative position control and knowledge to better than 10^{-8} m

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Objective

Finding Earth-like planets is Darwin's main objective, the most likely places for life to develop - at least as we know it! Darwin will survey 1000 of the closest stars, looking for small, rocky planets.

Space Propulsion

Micropropulsion requirements - Darwin

- **Derived requirements : Darwin tests**
  - Micro thrusters compensate for forces arising from forcing one satellite to follow a non-Keplerian orbit relative to the other. This force depends on separation and on selected orbit.
  - Thrusters also compensate for solar radiation pressure
  - Thrusters provide lateral and radial accelerations and torques for formation manoeuvres
  - Deployment based on 1 m/sec initial velocity, separation held within 30km, 24 hrs to deploy. Deployment does not require precisely controlled thrusters
- **Required total thrust in micro-Newtons (200kg spacecraft, linear sum):**

Orbit	precision flying	resize	rotate		slew		deploy
	250m		rim	hub	rim	hub	both
earth-sun L1	9	63	138	9	189	9	1590
earth-sun L2	9	63	138	9	189	9	1590
HETO/HELO	9	63	138	9	189	9	1589
WSB	9	63	139	9	190	9	1615
GEO	541	595	670	9	721	9	33489
HEO apogee	14	68	143	9	194	9	1888

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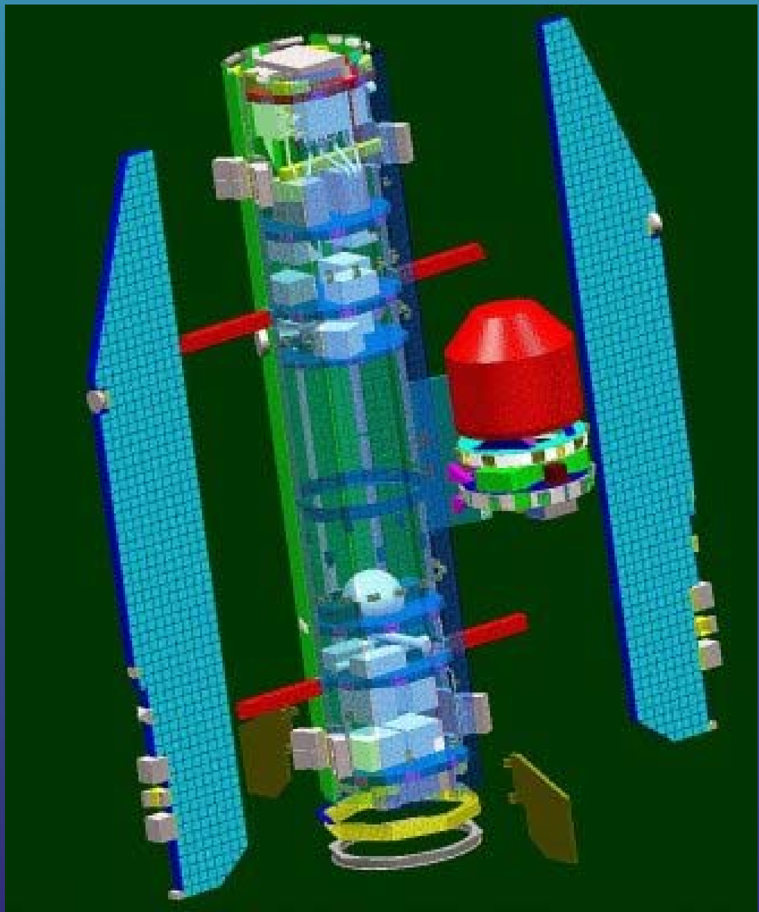
GOCE

(Gravity field and steady state Ocean circulation Explorer)

- GOCE shall determine the 3rd rate of change of gravity between pairs of accelerometers with a desired sensitivity of ~ 4 milliEötvös (mE) in each axis.
- To determine the geoid (i.e. equipotential surface for a hypothetical ocean at rest) to 1 cm accuracy.
- To accomplish both of the above at length/spatial scales down to 100 km.

Space Propulsion



1 gal = 1 [cm/s2]

1 E = 1×10^{-3} cm/s^2 / 10km = 10^{-9} (cm/s^2)/cm = 10^{-9} [s^2]
= 10^{-9} [gal/cm]