MECA0025 - Sattelite Engineering

Space propulsion devices

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Design of turbomachines and propulsors ($\dot{p}$)
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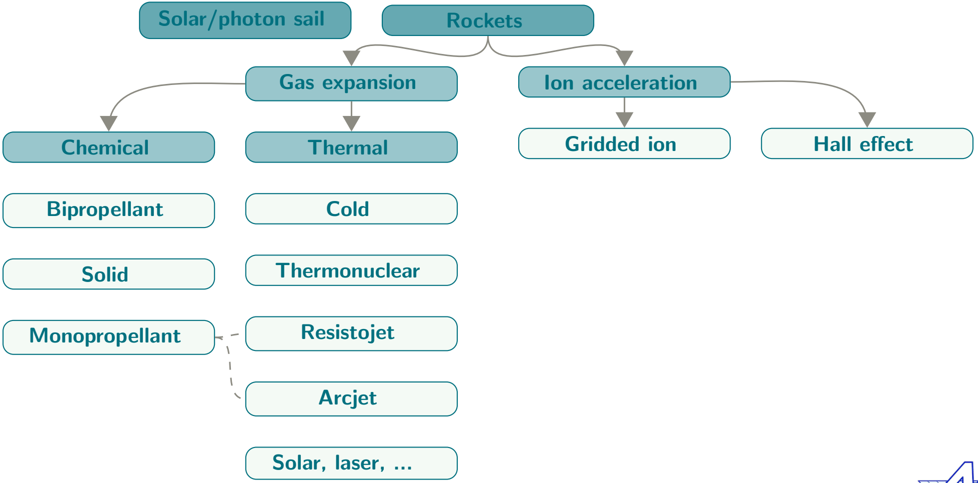
Outline


Solar/photon sail
Rockets
Gas expansion
Ion acceleration
Chemical
Thermal
Gridded ion
Hall effect
Bipropellant
Cold
Solid
Thermonuclear
Monopropellant
Resistojet
Arcjet
Solar, laser, ...

- thrust = reaction to acceleration of the propellant

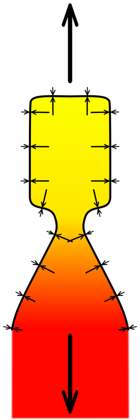
$$\mathcal{T} = \dot{m}_e c_e \ [N]$$

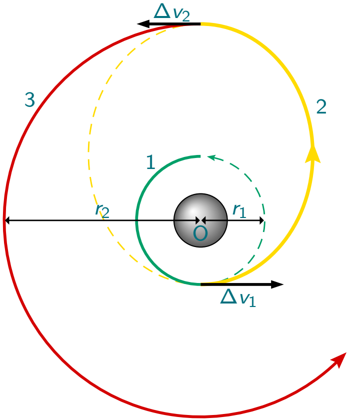
- effective exhaust velocity

$$c_e = \frac{\mathcal{T}}{\dot{m}} \ [m/s]$$

- specific impulse

$$\mathcal{I}_{sp} = \frac{\mathcal{T}}{\dot{m}g} = \frac{c_e}{g} \ [s]$$





- Hohmann transfer: single use *perigee* and *apogee* motors

$$\Delta v_1 = \sqrt{\frac{GM}{r_1}}\left(\sqrt{\frac{2r_2}{r_1+r_2}}-1\right)$$

$$\Delta v_2 = \sqrt{\frac{GM}{r_2}}\left(1-\sqrt{\frac{2r_1}{r_1+r_2}}\right)$$

  extremely short (ideally = instantaneous) duration "kick" $\rightarrow$ very high thrust

- Launch: overcome gravity

$$\Delta v = \int \frac{g(y)\,dy}{v}$$

  short duration $\rightarrow$ high thrust

- deep space missions: very long duration $\rightarrow$ low th… specific impulse

Mission	Δv	$\mathcal{T}$	duration	
Launch to low orbit	$\geq 9500\ m/s$	200 tons	8 min	continuous
Low to high orbit (kick motors)	$\approx 4200\ m/s$	few tons	seconds	continuous
High orbit to Mars	$\approx 3400\ m/s$	-	-	continuous
Escaping solar system	$+ = 8500\ m/s$	-	-	continuous
Control and positioning	$\approx 20 \ldots 400\ m/s$	mN to 10 N	-	pulsed

payload m_l, engine m_m and propellant mass m_p

$$m(t) = m_l + m_m + m_p(t) \quad \Rightarrow \quad \frac{dm}{dt} = \frac{dm_p}{dt} = -\dot{m}_e$$

Tsiolkowski equation

$$m\frac{dv}{dt} = g\mathcal{I}_{sp}\dot{m}_e = -g\mathcal{I}_{sp}\frac{dm_p}{dt} = -g\mathcal{I}_{sp}\frac{dm}{dt} \quad \Rightarrow \quad m(t) = m(0)\ e^{-\frac{\Delta v}{\mathcal{I}_{sp}g}}$$

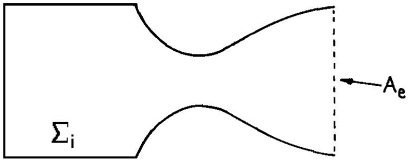
Propellant mass for given Δv

$$m_{p,\Delta v} = m(0) - m(t) = m(0)\left(1 - e^{-\frac{\Delta v}{g\mathcal{I}_{sp}}}\right)$$

high $\mathcal{I}_{sp} \rightarrow$ higher payload

- which $\Delta v \rightarrow$ integrated thrust $\mathcal{T}\Delta t$
- over short or long time span $\rightarrow$ low or high $\mathcal{T}$
- how much propellant weight can we afford $\rightarrow$ specific impulse $\mathcal{I}_{sp}$
- single burn *or* multiple burns
- variation of thrust required $\rightarrow$ can we pulse the thruster
- dry weight of the motor and its auxiliaries (*e.g.* reservoir, power generator)

Outline
Gas expansion thrusters



Thrust

$$\mathcal{T} = \dot{m}_e v_e + (p_e - p_a) A_e$$

Separate contributions of nozzle and gas/conditions

$$c_e = \frac{\mathcal{T}}{\dot{m}_e} = \underbrace{\frac{\mathcal{T}}{p^\circ A_t}}_{\mathcal{C}_\mathcal{T}} \underbrace{\frac{p^\circ A_t}{\dot{m}_e}}_{\mathcal{C}^*}$$

- thrust coefficient $\mathcal{C}_\mathcal{T}$: performance of nozzle;
- characteristic velocity $\mathcal{C}$: propellant properties and feed conditions.

In attached regime of the de Laval nozzle, *i.e.*

- underexpanded
- overexpanded with oblique shocks outside nozzle

we find

$$\mathcal{T} = \dot{m}_e v_e + (p_e - p_a)\, A_e \quad \Rightarrow \quad \mathcal{C}_\mathcal{T} = \frac{\mathcal{T}}{p^\circ A_t} = \left(\left(1 + \gamma M_e^2\right) \frac{p_e}{p^\circ} - \frac{p_a}{p^\circ} \right) \frac{A_e}{A_t}$$

$\mathcal{C}_\mathcal{T}$ depends on

- area ratio A_e/A_t
- pressure ratio $NPR = p^\circ/p_a$
- heat capacity ratio γ

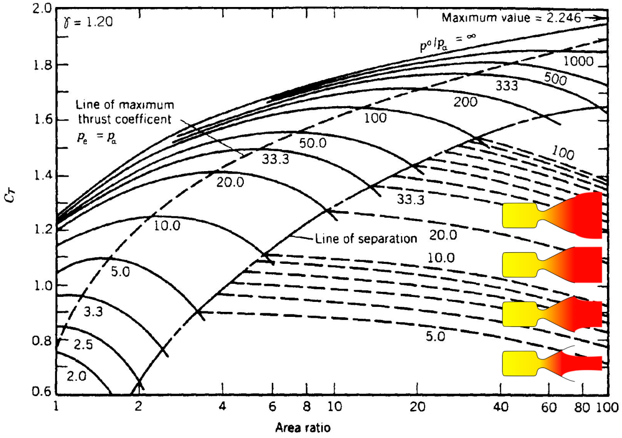
In separated regime (= heavily overexpanded), the nozzle is "shortened" to the location of separation/shock

C_T depends on

- altitude, in particular nozzle pressure ratio p°/p_a
- nozzle geometry, in particular exit to throat area ratio A_e/A_t

Observations and consequences for launchers/space thrusters

- for each altitude/pressure ratio optimal area ratio → area ratio is compromise for launchers
- maximum thrust in vacuum, increases with area ratio → maximal area ratio for space thrusters
- separation is observed up to a certain altitude/pressure ratio → launchers: high feed pressure p° to reduce separation



$$\mathcal{C}^* = \frac{p^\circ A_t}{\dot{m}_e} = \frac{p^\circ A_t}{\rho^* a^* A_t} = \frac{1}{\gamma}\,\frac{p^\circ}{p^*}\,\sqrt{\frac{T^*}{T^\circ}}\,\sqrt{\gamma \mathcal{R} T^\circ} = f(\gamma)\sqrt{\frac{\gamma \mathcal{R}^* T^\circ}{\mathcal{M}}}$$

Characteristic velocity depends on

- chamber/combustion temperature T°
- molar mass $\mathcal{M}$

Hence

- lighter molecules have higher c_e / $\mathcal{I}_{sp}$ for same T°
- combustion rockets: T° determined by reaction $\rightarrow \mathcal{C}^*$ is material property

Assuming constant $\mathcal{C}_\mathcal{T}$, p° and T°, the molar mass $\mathcal{M}$ impacts

- specific impulse $\rightarrow$ favor light gases

$$\mathcal{I}_{sp} = \frac{\mathcal{C}_\mathcal{T}\mathcal{C}^*}{g} \sim \frac{1}{\sqrt{\mathcal{M}}}$$

- thrust to power $\rightarrow$ favor heavy gases

$$\frac{\mathcal{T}}{\mathcal{P}} \sim \frac{\dot{m}_e c_e}{\dot{m}_e c_e^2/2} \sim \sqrt{\mathcal{M}}$$

- thrust to area/size $\rightarrow$ more or less independent

$$\frac{\mathcal{T}}{A} \sim \frac{\dot{m}_e v_e}{\dot{m}_e/\rho_e v_e} \sim \rho_e v_e^2 \sim \frac{\mathcal{M}}{\sqrt{\mathcal{M}}^2}$$

Specific impulse variations

- $\mathcal{C}_\mathcal{T}$ increases with pressure ratio, *i.e.* with altitude and feed pressure p°
- $\mathcal{C}_\mathcal{T}$ maximal and independent of feed pressure in vacuum
- characteristic velocity independent of p°
- mass flow per unit area $\sim p^\circ$

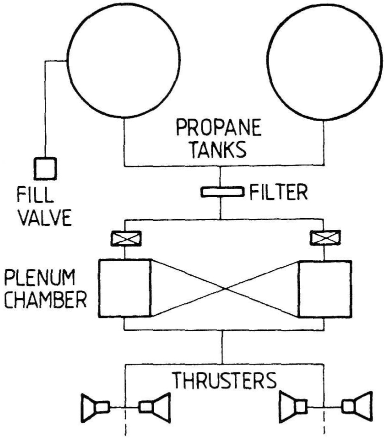
Launchers

- high feed pressure to maximise thrust coefficient and mass flow
- area ratio is chosen via compromise over altitudes

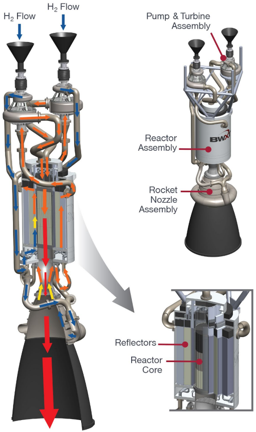
Space thrusters

- $\mathcal{I}_{sp}$ independent of feed pressure p°
- very high area ratios to maximise $\mathcal{C}_\mathcal{T}$
- feed pressure determined by size / engine weight considerations

- non-reacting gas: N_2, Ar, Fr, C_3H_8
- temperature controlled high pressure reservoir
- low specific impulse $\sim 50s$
- thrust levels $\sim 20\ mN$
- pulsed for modulation of average thrust
- precise control of position



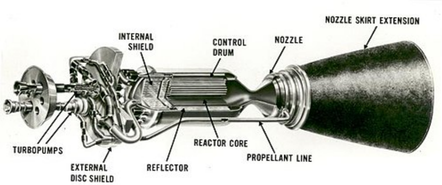
- pressurized gas heated by nuclear reactor
- very high specific impulse $\mathcal{I}_{sp} = 500\ s \ldots 900\ s$
  - low mass gases such as $H_2 \rightarrow$ high $\mathcal{R}$
  - temperature not determined by combustion
- high thrust $\mathcal{T} \approx 100\ kN$
- online thrust control
- currently investigated concept for space exploration, orbit insertion, ...



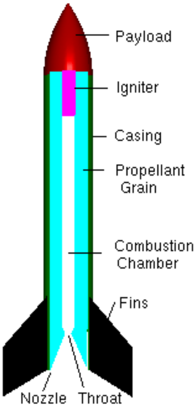
NASA's Nuclear Thermal Propulsion Engine System, of which BWXT is providing support for reactor and fuel design and analysis.

Nuclear Engine for Rocket Vehicle Application (NERVA)

- research engine at NASA
- vacuum thrust $\mathcal{T} = 246kN$
- Chamber pressure $p^{\circ} = 3.861\ MPa$
- vacuum $\mathcal{I}_{sp} = 710 \ldots 841\ s$ (SLS vs vacuum)
- dry weight: 18 tonnes
- thermal power: $\mathcal{P} = 1.1\ MW$



- grain: paste of premixed oxidiser and fuel
- pyrotechnic start to single step burn
- thrust variation **a priori** determined by grain shape
- thermo-acoustic instabilities



Solid Propellant Rocket

Chemical rockets: Solid propellant rocket- grain shape

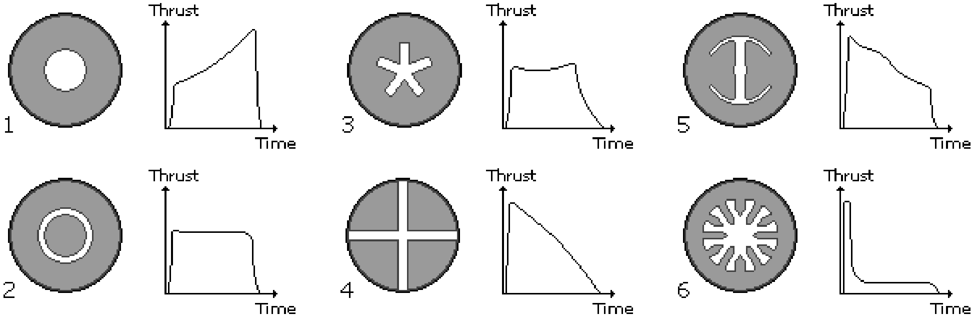
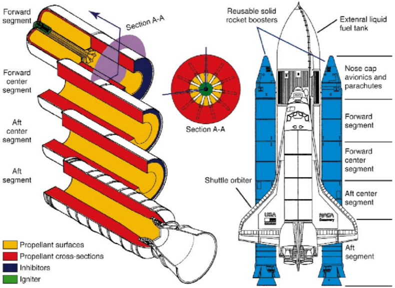


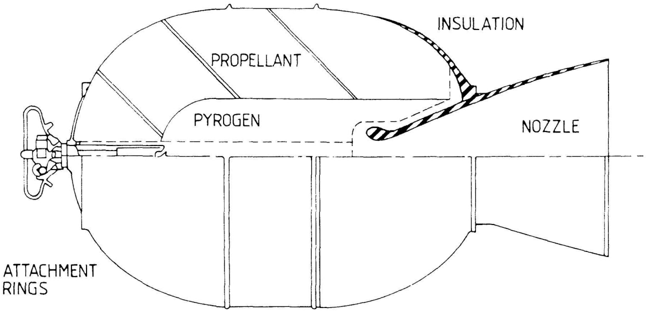
Figure 1.14

- propellant mass $m_p = 500$ *tonne*
- empty mass $m_m = 91$ *tonne*
- $\mathcal{T} = 15\ MN$
- $\mathcal{I}_{sp} = 242\ s$
- reusable

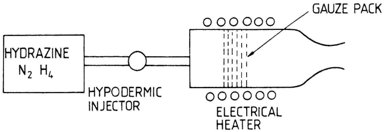


Intelsat V

- $\Delta v = 2000 m/s$
- payload $m_l \approx 1000\ kg$
- propellant mass $m_p \approx 900\ kg$
- engine mass $m_m \approx 1000\ kg$
- $\mathcal{T} = 70\ kN$ during 40 s
- $\mathcal{I}_{sp} \approx 280s$



- operating principle
  - main propellant hydrazine N_2H_4
  - pressurized reservoir
  - pulsed expansion over regulation valve
  - decomposition over heated catalyst bed
  - decomposition products N_2, H_2 and NH_3
- can be combined with thermal heating (arcjet/resistojet)
- $\mathcal{T} > 10N$ modulated during operation by pulsing
- $\mathcal{I}_{sp} \approx 200\ s$
- attitude control and station keeping (geostationary)

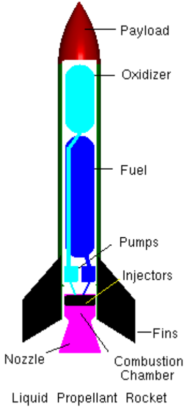


- $\mathcal{T} = 1\ N$
- $\mathcal{I}_{sp} = 210\ s$
- $\dot{m}_e = 0.44 g/s$
- Burn time = 50 hours
- length = 17 cm
- $A_e/A_t = 80$
- applications: small sattelites and deep space probes
    - attitude and orbit control
    - station keeping

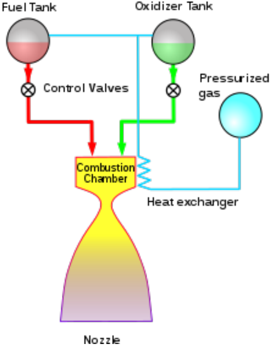
Chemical rockets: monopropellant - Astrium hydrazine

- $\mathcal{T} = 400N$
- $\mathcal{I}_{sp} = 220s$
- Burn time : 30 minutes
- Length : 32 cm
- attitude control Ariane V

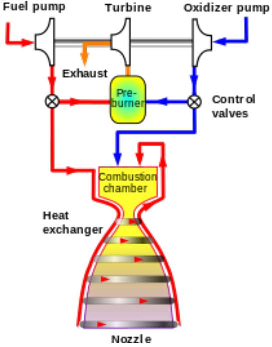
- combustion of pressurized fuel and oxidiser
- variants
  - pressure fed
  - pump fed
- $\mathcal{I}_{sp} = 300 \dots 400s$
- applications
  - launch (pump fed)
  - kick engines (pressure fed)
  - orbit and attitude control (pressure fed)
- pogo instabilities
- complex starting procedure



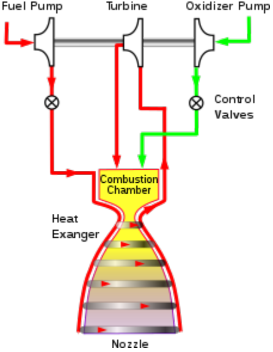
Pressure fed



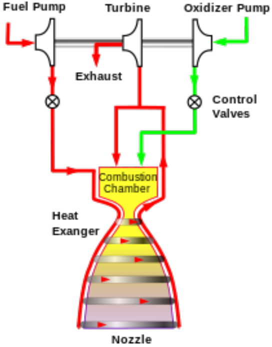
Pump fed



Gas generator cycle



Expander cycle



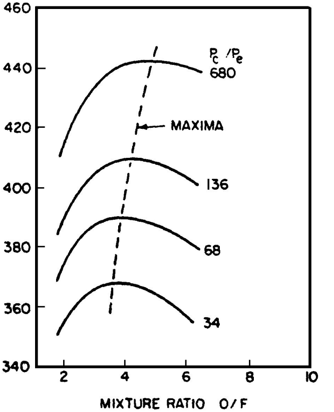
Expander cycle with bleed

Mixture ratio

- oxidizer to fuel ratio

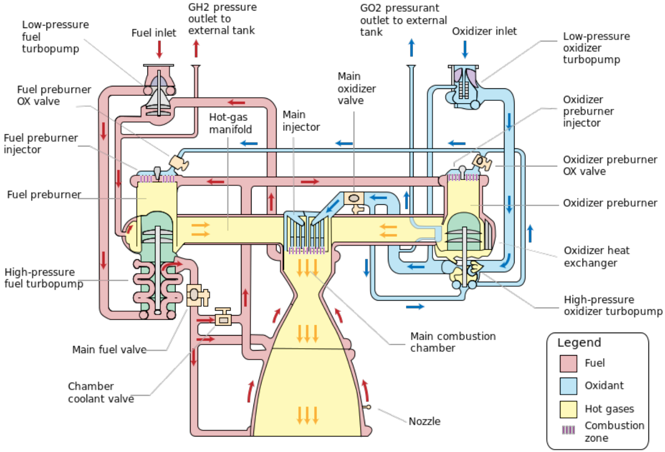
$$MR = \frac{\dot{m}_o}{\dot{m}_f}$$

- optimal MR compromises
  - high combustion temperature
  - low average molecular weight of combustion product
- (almost?) never stoechiometric



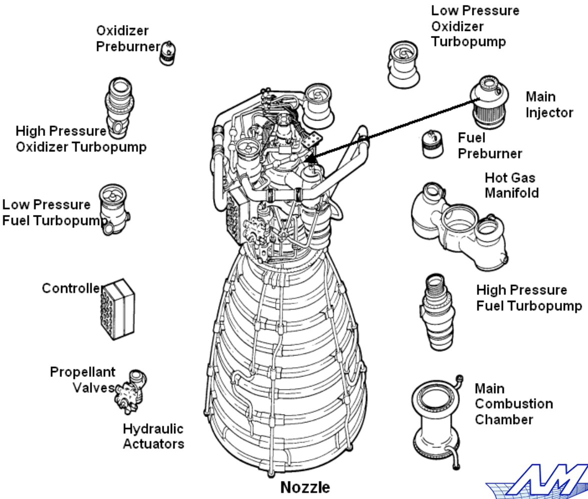
Specific impulse in function of MR for given p^0/p_e

Chemical rockets: Bipropellant liquid rocket- space shuttle main engine



Chemical rockets: Bipropellant liquid rocket- space shuttle main engine

Space shuttle main engine	
Thrust (kN) vacuum	2090
Sea level	1700
Specific impulse(s) vacuum	455
Sea level	363
Mixture ratio	6:1
(cf. 8:1 stochiometric	$2H_2 + O_2 \rightarrow 2H_2O$)
Chamber pressure (bar)	207
Nozzle area ratio	77
Flow rates (kg/s) Engine	468
Gas generator	248
Pump discharge pressure (bar)	
LOX	309
LH2	426
Length (m)	4.24
Nozzle exit diameter (m)	2.39
Burn time(s)	480
Mass (kg)	3022



Chemical rockets: Bipropellant liquid rocket- Astrium S400

- pressure fed
- $\mathcal{T} = 400\ N$
- $\mathcal{I}_{sp} = 318\ s$
- propellants: MMH / N_2O_4
- apogee orbit injection (geostationary)
- orbit manoeuvers (deep space probes: Venus express, Artemis)

Chemical rockets: Bipropellant liquid rocket- Astrium S10

- pressure fed
- $\mathcal{T} = 10N$
- $\mathcal{I}_{sp} = 291s$
- propellants: MMH / N_2O_4
- nozzle expansion ratio : 150
- attitude/orbit control (large satellites: Arabsat)
- attitude/orbit control (deep space probes: Venus Express)

Electric field generated by particle charge density ρ_q

$$\nabla \cdot \mathbf{E} = \frac{\rho_q}{\epsilon_0} = \frac{\sum_i n_i q_i}{\epsilon_0}$$

with n_i number density and q_i charge for particle i (electrons / ions)

Electric field and potential

$$\mathbf{E} = -\nabla V$$

Lorentz force on particle with charge q in electric $\mathbf{E}$ and magnetic field $\mathbf{B}$

$$m\frac{d\boldsymbol{v}}{dt} = q\left(\mathbf{E} + \boldsymbol{v} \times \mathbf{B}\right)$$

Linear acceleration subject to electric field $\mathbf{E}$

$$m\frac{\partial \boldsymbol{v}}{\partial t} = q\mathbf{E} = -q\nabla V \quad \Rightarrow \quad m\boldsymbol{v} \cdot \frac{\partial \boldsymbol{v}}{\partial t} = -q\boldsymbol{v} \cdot \nabla V \quad \Rightarrow \quad m\Delta\frac{v^2}{2} = -q\Delta V$$

Particle energy expressed in eV

Larmor precession: helicoidal motion subject to magnetic field

$$\frac{\partial v_{p,\parallel}}{\partial t} = 0 \qquad \Rightarrow v_{p,\parallel}(t) = v_{p,\parallel}(0)$$

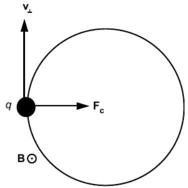
$$\frac{\partial \boldsymbol{v}_{p,\perp}}{\partial t} = \frac{q_p}{m_p}\boldsymbol{v}_{p,\perp} \times \mathbf{B} \qquad \Rightarrow \boldsymbol{v}_{p,\perp}(t) = e^{i\omega_\lambda t}\boldsymbol{v}_{p,\perp}(0)$$

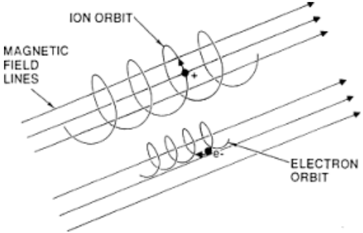
with (Larmor) frequency and radius

- frequency $\omega_\lambda = \frac{|q_p|B}{m_p}$
- radius $r_\lambda = \frac{|v_\perp|}{\omega_\lambda}$

Drift velocity: if $\mathbf{E}\perp\mathbf{B}$ $\rightarrow$ steady state velocity in equilibrium with Lorentz force

$$\boldsymbol{v}_{p,d} = \frac{\mathbf{E}\times\mathbf{B}}{B^2}$$





General principles

- ionise propellant gas
- accelerate heavy ions by electrostatic field
- thrust = reaction force
- thrust determined by ion flux/charge density, which is determined by
  - maximum potential difference of E-field
  - charge saturation (external E-field = E-field due to charge density)

Applications : requiring very high specific impulse

- orbital insertion
- deorbitalisation (demise)
- station keeping
- deep space missions

Suppose same particle charge q, thruster potential ΔV

Effective ejection speed/specific impulse

$$v_e \leq \sqrt{\frac{2q\Delta V}{m_p}} \sim \frac{1}{\sqrt{m_p}}$$

Thrust to power

$$\frac{\mathcal{T}}{\mathcal{P}} \sim \frac{\dot{m}v_e}{\dot{m}v_e^2/2} \sim \frac{1}{\mathcal{I}_{sp}} \sim \sqrt{m_p}$$

Power determines generator mass $\rightarrow$ favor "lower" $\mathcal{I}_{sp}$ and therefore "heavy" gases (Xenon, Krypton, Iodine)

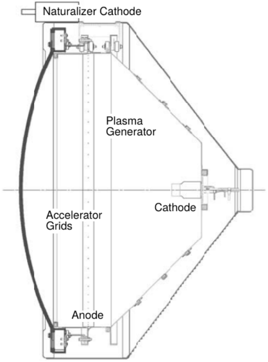
Thrust to area: suppose charge density saturated / fixed

$$\frac{\mathcal{T}}{A} = \frac{\dot{m}_e}{A} v_e \sim m_p v_e^2 \sim Cte$$

not impacted by particle mass

Principle: electrostatic acceleration of ions

- upstream generation of plasma stream
- *screen grid* removes electrons from plasma stream
- ΔV between *screen* and *accelerator* grids $\rightarrow$ **E** $\rightarrow$ ion acceleration
- thrust limited by
  - maximum potential difference ΔV
  - ion charge density saturation (Child-Langmuir)
- naturaliser cathode: electron flux neutralises ion flux outside to ensure thruster charge neutrality



Characteristics

- $\mathcal{I}_{sp} \approx 2000 \ldots 10000\ s$ ($v_e \approx 20 \ldots 100\ km/s$!)
- $\mathcal{T} \sim 10mN \ldots 1N$

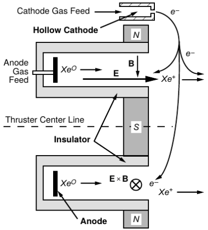
- $\mathcal{T} = 150\ mN$
- $\mathcal{I}_{sp} = 4000\ s$
- $\mathcal{P} = 4kW$
- propellant : Xenon
- beam voltage : $\Delta V = 1200\ V$
- run time $\geq$ 20000 h
- thruster mass : 154 kg
- applications
  - Station keeping
  - orbit transfer
  - deep space missions

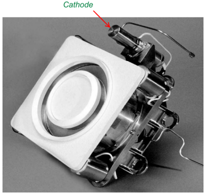
Principle:

- external radial magnetic field B_r between annular poles
- electrons "feel" B_r while ions don't
- Larmor precession and drift confine electrons to bounce between poles and rotate fast in annular space
  - electron concentration $\rightarrow$ axial electric field E_a
  - collision w/ neutrals $\rightarrow$ ionisation
- ions accelerated by axial electric field E_a
- no ion charge saturation due to presence of electrons $\rightarrow$ higher flux density $\rightarrow$ compact system
- axial migration of electrons to anode not fully understood
- electrons recombine outside with ions (thruster charge neutrality)

Characteristics

- $\mathcal{I}_{sp} \approx 1000 \dots 8000\ s$ ($v_e \approx 10\ km/s \dots 80\ km/s$)
- $\mathcal{T} \sim 40mN \dots 5N$





Tentative performances (tbc ?)

- Discharge Power: 1 $kW \ldots 2\ kW$
- efficiency $\sim 0.4 \ldots 0.5 \rightarrow$ consumed power 2.5 $kW \ldots 5\ kW$
- thruster Mass: 6.3 kg
- $\mathcal{T} = 70 \ldots 180\ mN$
- $\mathcal{I}_{sp} = 1600 \ldots 1860\ s$