Advanced Space Propulsion (Master Class)

(at the Institute of Space Systems, Stuttgart)

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Content

- Introduction/ Motivation
  - Some definitions and thoughts
  - Propulsion problem
  - Electric Propulsion
- EP systems at IRS (omitted: Thermal Arcjets):
  - PPT
  - Self-field Magnetoplasmadynamic and Applied-field Magnetoplasmadynamic Devices
  - TIHTUS
  - IEC (only if time is sufficient)
  - RAM EP (only if time is sufficient)
- Propulsion for missions in the solar system and beyond
- Summary

- (add slides: Verification + validation Tools (plasma diagnostics + modelling tools)

Space Travel – a Definition

○ Generally all anthropogenic activities in space:

- Launch (to access space) ✓
- Operation of diverse objects in vicinity to Earth ✓
- Missions near Earth, but ouside its gravitation regime ✓/✗
- Travel within our solar system ✓/✗
- Interstellar flight ✗

Space Travel – a Resumé

○ Balance:
- Space travel mostly in regime near Earth.
- Most missions are un-manned.

○ Examples:
- Ballistic launchers, satellites, probes
- manned launchers, space stations, moon landers

○ Propulsion characteristics:
- Launch with chemical high thrust systems, energy limited → limited effective exhaust velocities c_e.
- Either impulsive orbit maneuvres applying low c_e or un-/pulsed low thrust spiralling applying high c_e systems.

Quelle: http://commons.wikimedia.org

Propulsion Problem

Δv based on trajectory analysis (e.g. spiraling, continuous transfers, …)

Mission:

Δv

(velocity increment)

$$\Delta v = \int_{\Delta t_b} a_c dt$$

$$\Delta v = c_e \log\left(\frac{m_0}{m_1}\right)$$

- Space flight missions defined via *propulsion requirement* typically via assignment of velocity increment Δv – based on orbit mechanics.
- But: Consideration of *propulsion capabilty* as Δv (as well). Determined by propulsion system. Available acceleration a_c impacts (continuous) „burn time“, effective exhaust velocity c_e impacts required propellant mass.
- **Propulsion problem: Potentially long lasting transfers.**

(Further) Consideration of the propulsion problem

How can we overcome this arduousness in propulsion?

A transfer is characterized by a velocity increment Δv

$$\Delta v = \int_{\Delta t_b} a_c \, dt$$

Kinematics:
Relating a_c and “burn” time

$$\Delta v = c_e \log\left(\frac{m_0}{m_1}\right)$$

Tsiolkovsky’s equation:
Relating c_e and dry mass

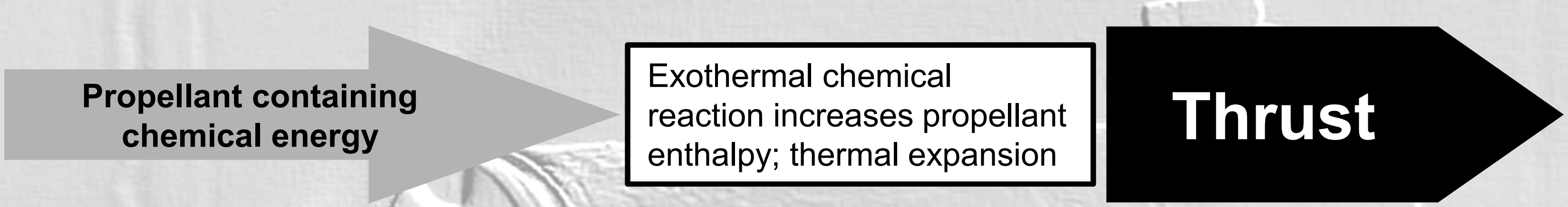
The link of a_c and c_e is the mass specific thrust power.

$$\frac{1}{2} a_c c_e \leq \alpha_{\mathrm{lim}}$$

Mass specific thrust power often limited (concerning power supply). Often desire of enhancement.

→ advanced propulsion with increased α_{lim}, enhancement of EP thruster power

Power Increase for Electric Propulsion

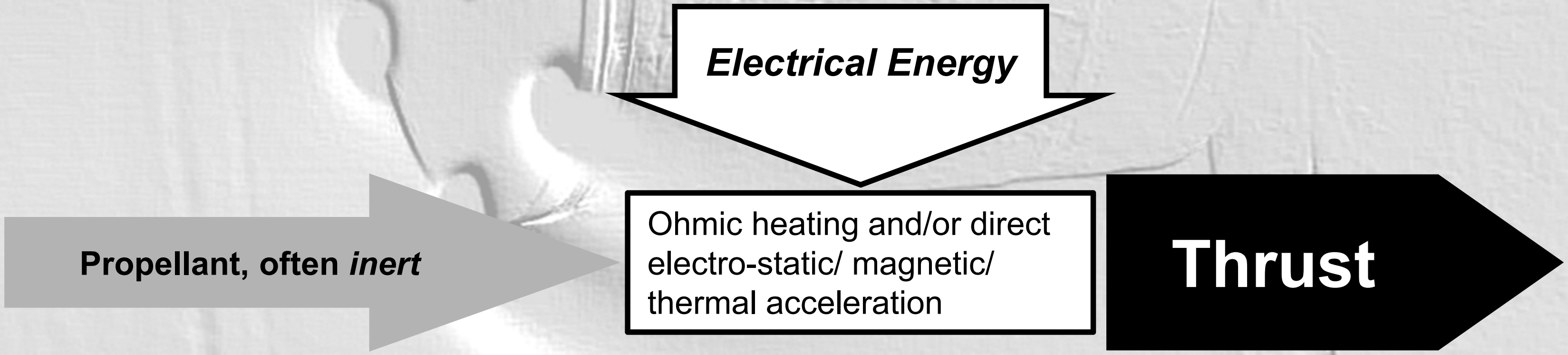


Propellant containing chemical energy

Exothermal chemical reaction increases propellant enthalpy; thermal expansion

Thrust

Schematic of typical chemical thruster



Electrical Energy

Propellant, often *inert*

Ohmic heating and/or direct electro-static/ magnetic/ thermal acceleration

Thrust

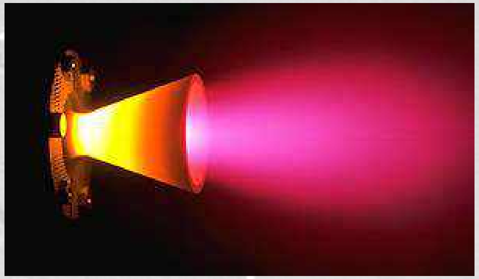
Schematic of electric thruster

Types of Electric Propulsion (EP)

Electrothermal Thrusters (Arcjets)

- Devices up to 100 kW were tested (TRL 6)
- High thrust levels with exhaust velocities up to 20 km/s
- Light atomic gases as propellant (He, H_2; alternative NH_3)
- Thrust efficiency depends on propellant 20-50%

IRS HiPARC



Steady state Electromagnetic Thrusters (SF-/AF-MPDT)

- Scalable up to MW power range; lab devices up to 1 MW (TRL 4)
- Average thrust levels with ce between 10 and 70 km/s
- Various propellants can be used (H_2, He, Ar, Kr, Xe, Li)
- Comparable high thrust efficiency 20-60%

IRS AF-MPD

Electrostatic Thrusters (HET, GIT)

- Tested in space environment (TRL 9)
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- HET: average thrust levels with exhaust velocities of 20-50 km/s and thrust efficiency of 40-70%

Advanced Thrusters:

- RF inductive and Helicon
- Hybrid thrusters (TiHTUS, VASIMR)
- IEC, FRC, etc.

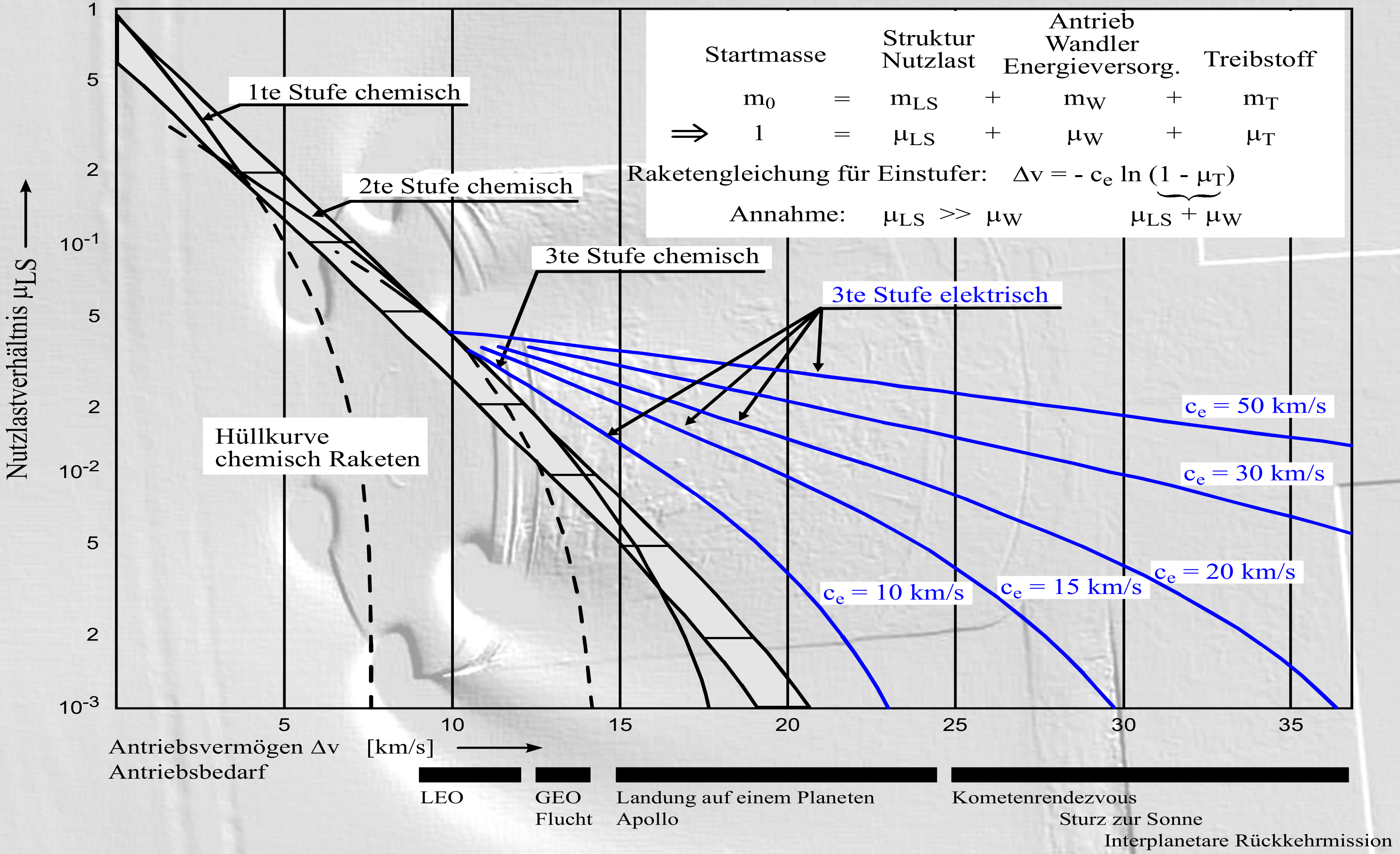
Characteristics of electrical Propulsion

- Extended propellant flexibility
  - Depending on concept used (e.g. Ion-Thruster versus. AF-MPD…)
  - Depending on environmental aspects (e.g. wastes from manned spacecraft, In Situ Resources utilization, …)
- …and extended/ flexible thrust control
  - Via electrical parameter, propellant adjustment, etc...
- EP can achieve signficantly higher c_e as campared with chemical propulson. Leads to significant **savings of propellant mass** or **higher Δv** if the same amount of propellant was used. (next slide).

→ *EP is, therefore, very beneficial.*

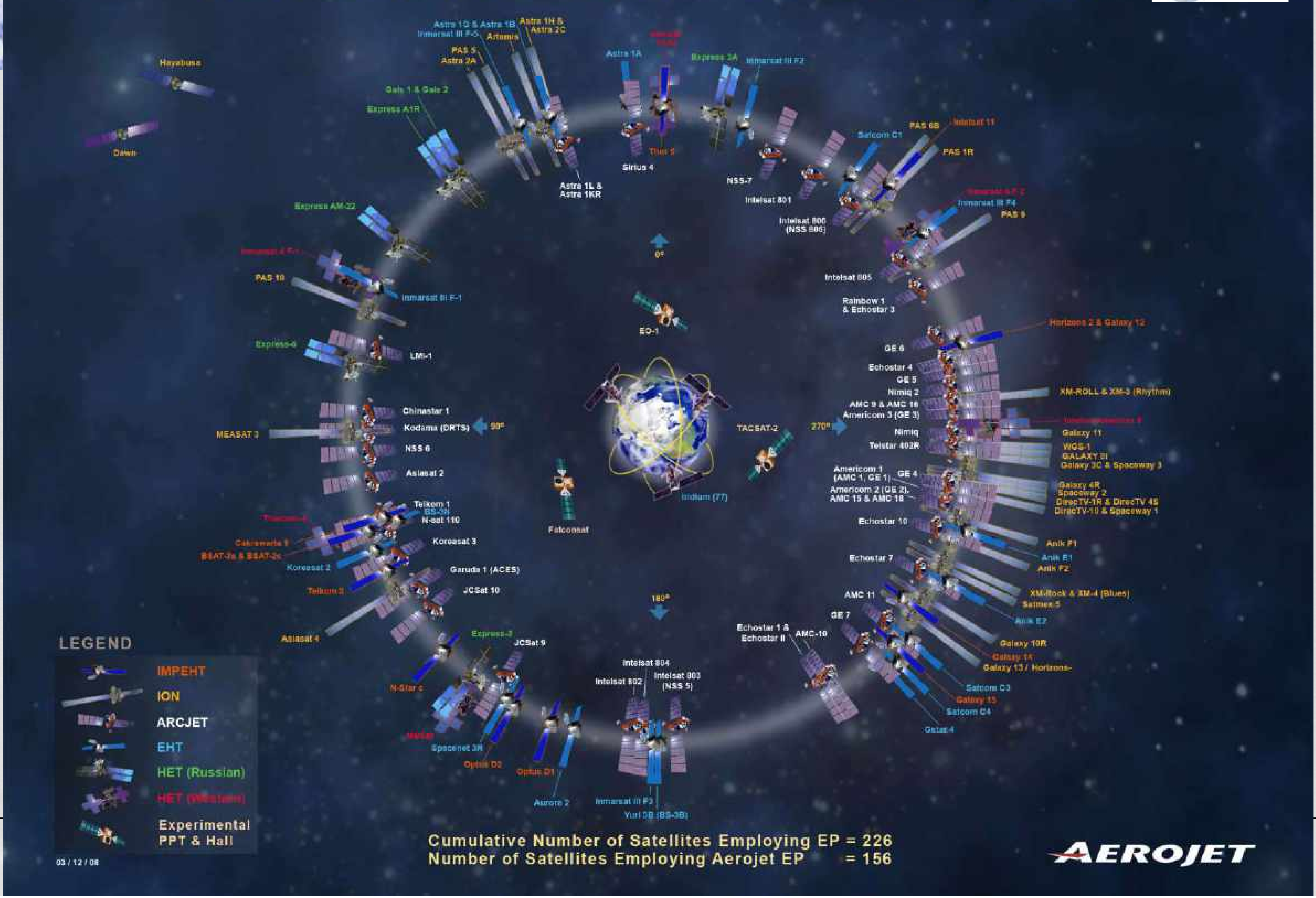
- The low thrust level limits EP to operations in space.

Electric Propulsion for interplanetary missions and orbit transfers



P/L ratio depending on Dv using chemical and electrical upper stage

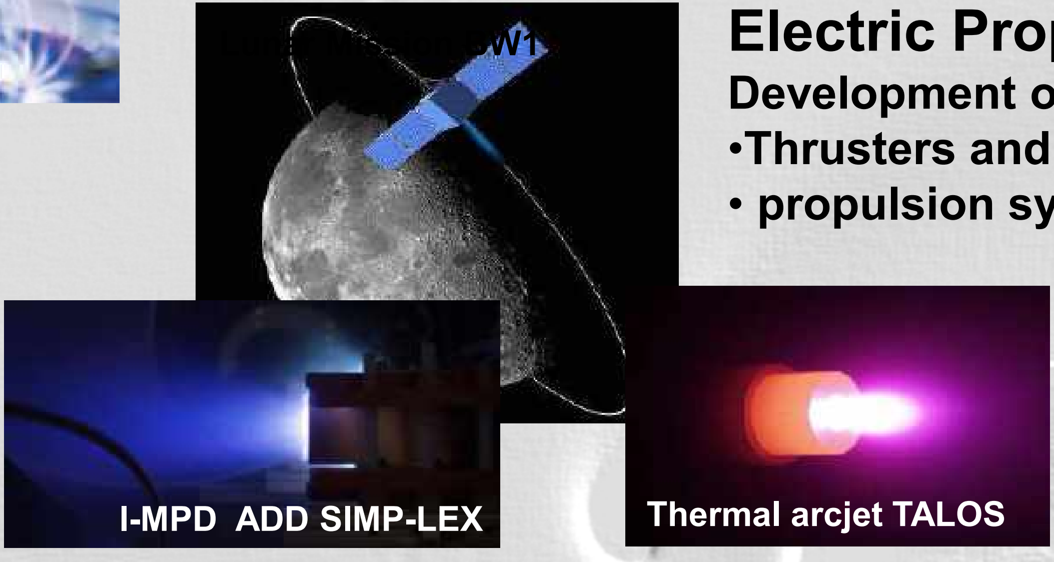
Operational Satellites with Electric Propulsion

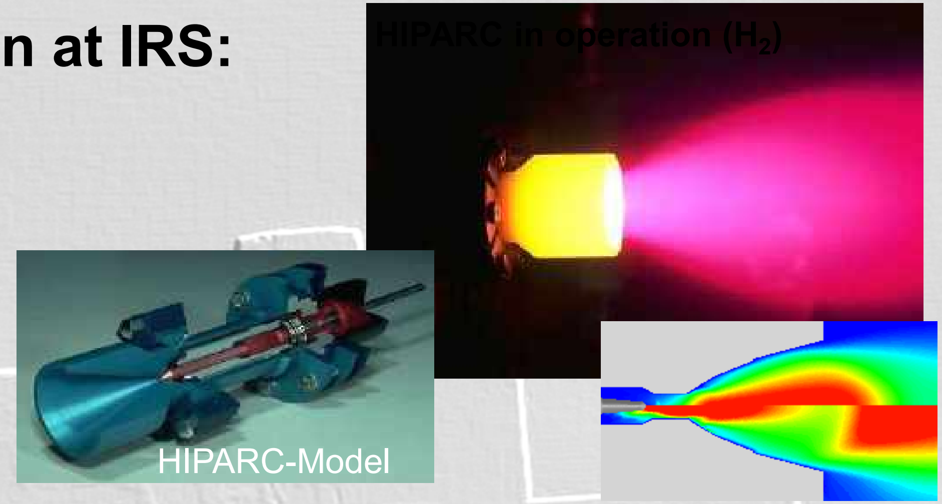


Cumulative Number of Satellites Employing EP = 226
Number of Satellites Employing Aerojet EP = 156

Electric Propulsion at IRS:

Development of
- **Thrusters and**
- **propulsion systems**



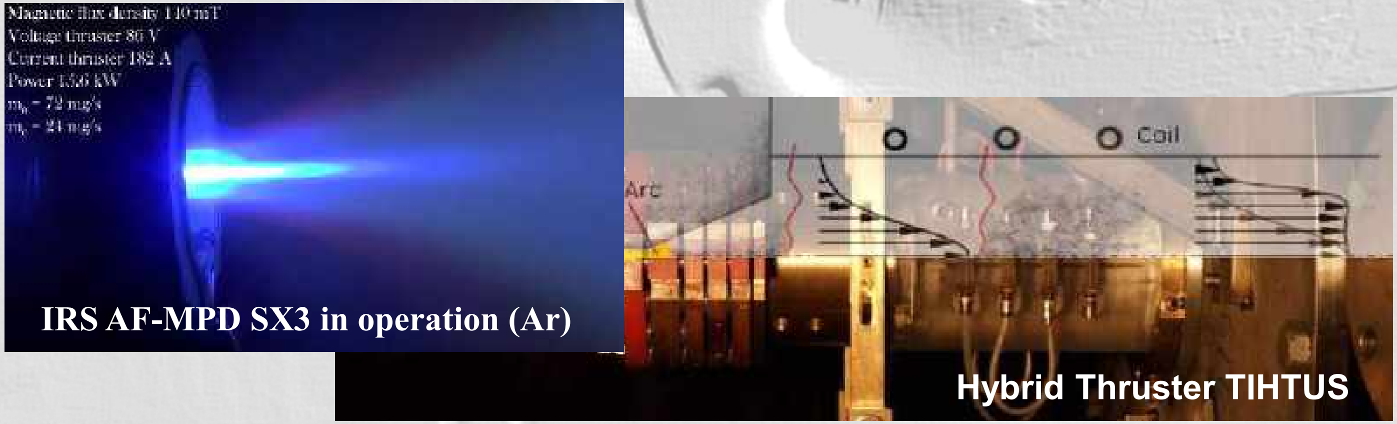


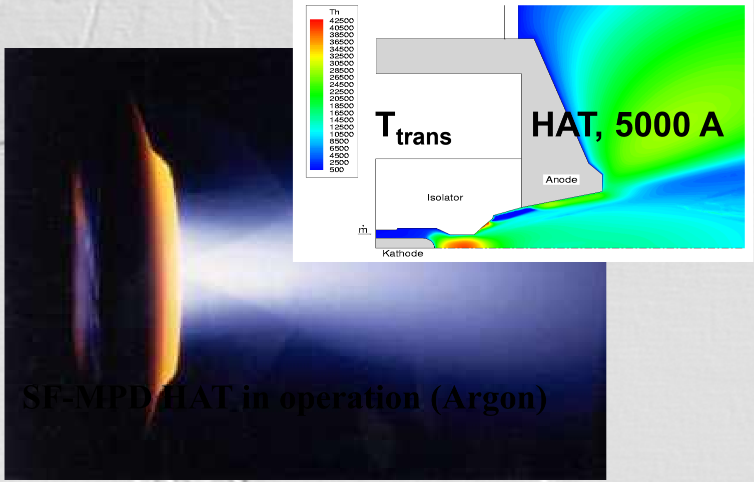
Low power EP (50W up to some 10kW) (Satellites and Exploration)

- Thermal arcjet thrusters
- PPT (iMPD)
- Applied-field MPD thrusters

High power EP (50 kW up to MW) (Transport of large payloads)

- thermal arcjet thrusters
- Self-field MPD thrusters
- Hybrid thruster TIHTUS





Electric Propulsion at IRS:

Development of
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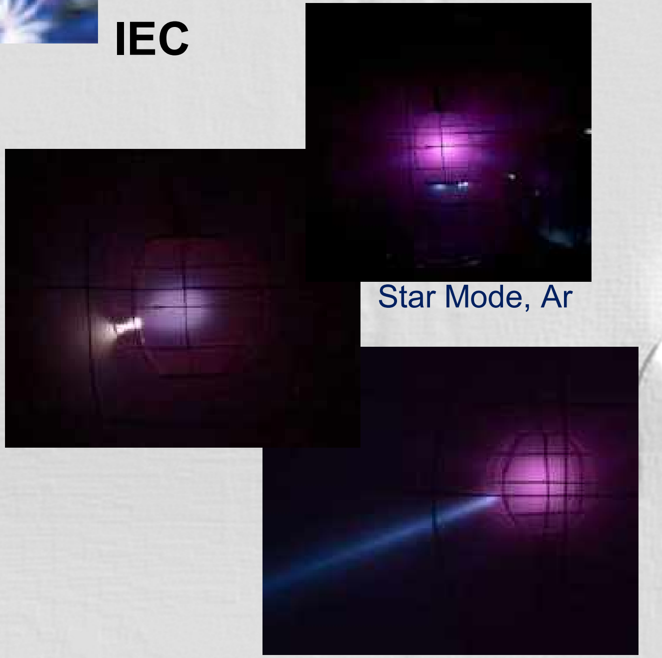
IEC


Star Mode, Ar

“Tight“ Jet Mode, He

Secondary electric propulsion

- Electrolyzer
- 1 N catalytic thruster
- Green propellant

→ Small satellites

ABEP

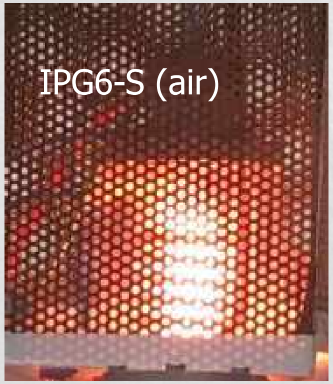
- Intake verification
- Intake design
- Inductive thruster

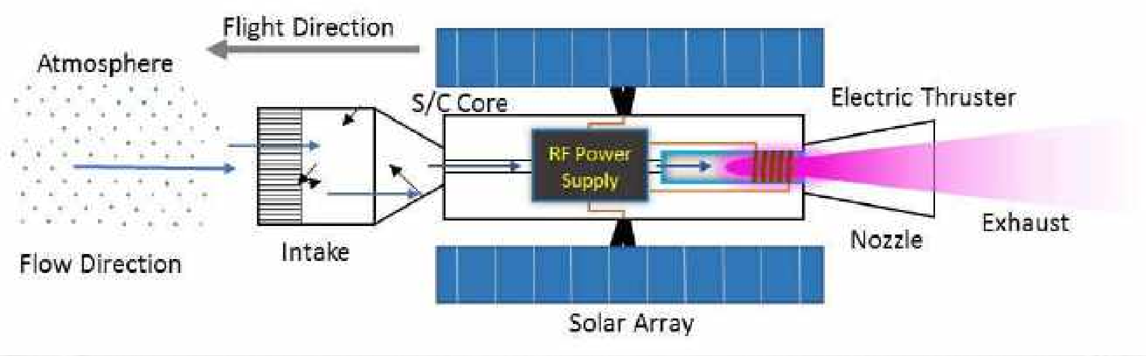
Mini PPT

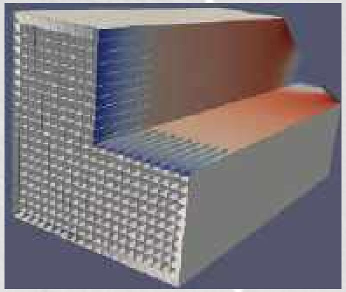
- PETRUS
- Thermal PPT
- Reliable, robust, cheap, …

→ CubeSat application

IPG6-S

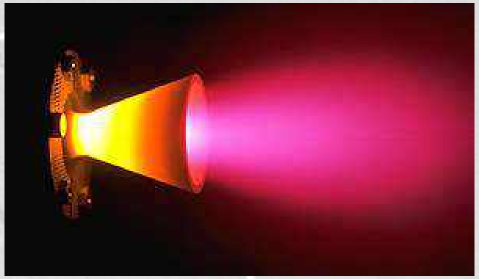





DSMC simulation of adapted intake geometry

Types of Electric Propulsion (EP)

IRS HiPARC



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IRS AF-MPD

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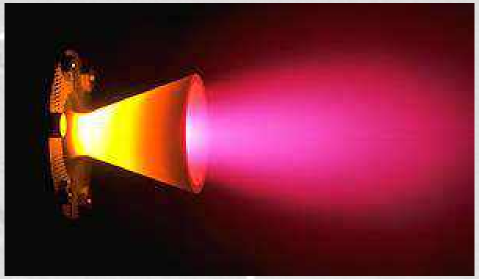

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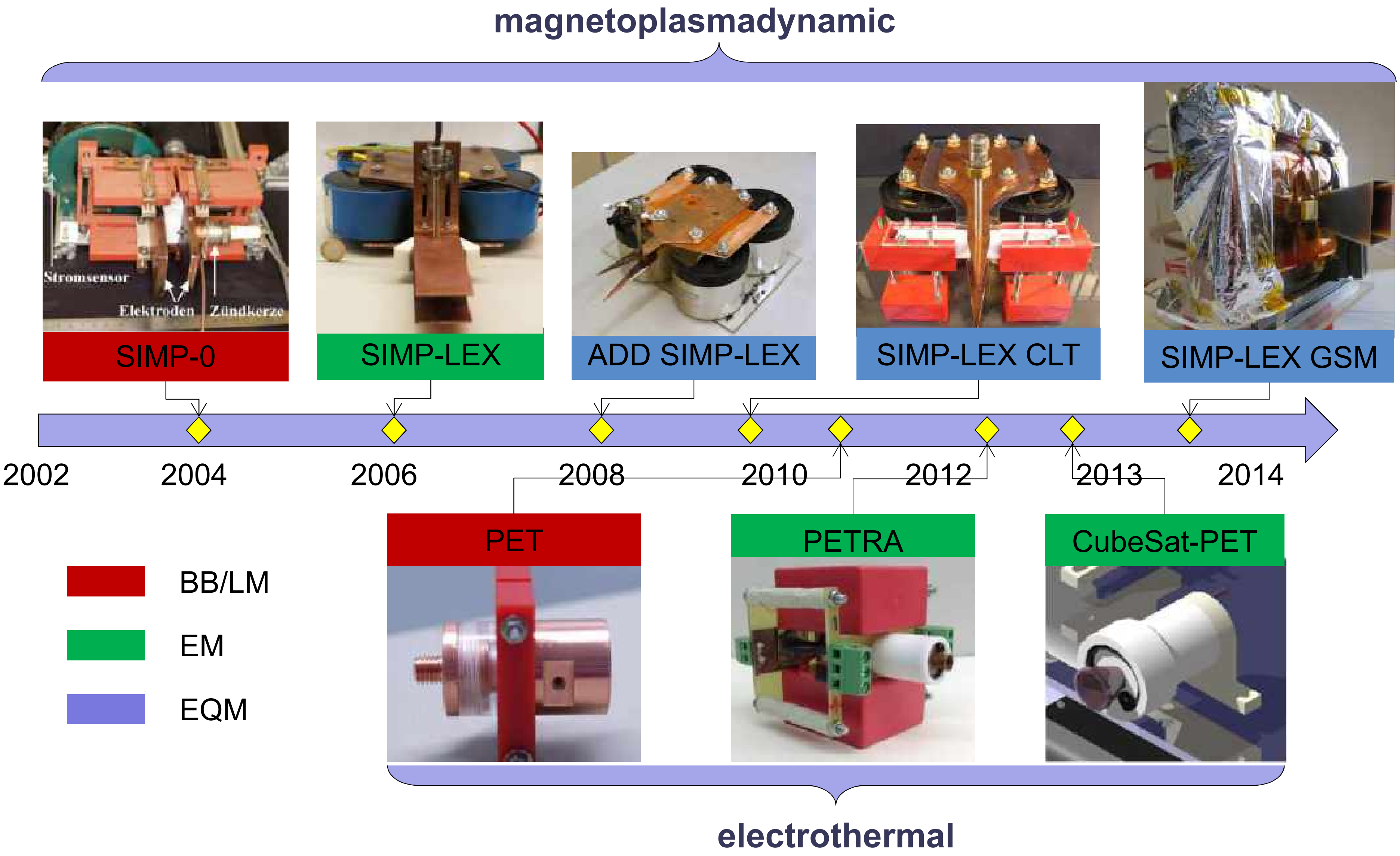
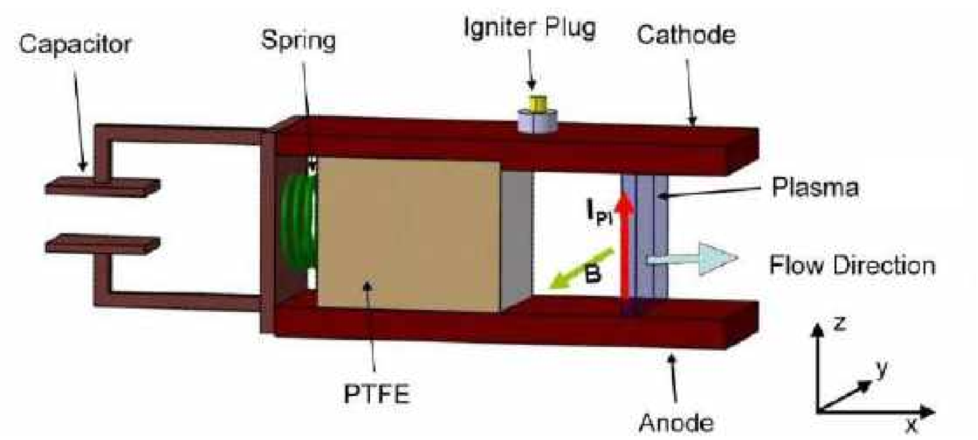
PPT Mile Stones at IRS (Flanking Projects for CAPE)



🟥 BB/LM
🟩 EM
🟦 EQM

Function of iMPD and PET

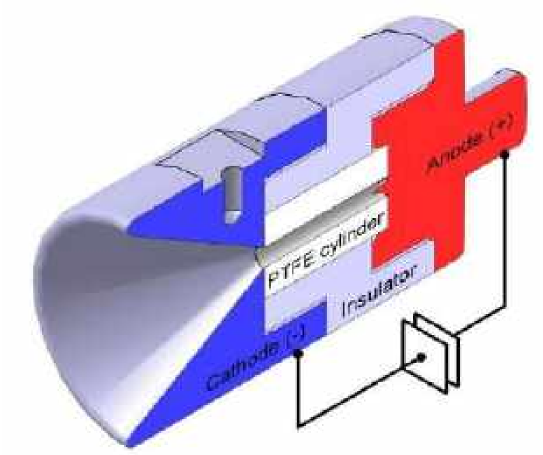
iMPD



- Discharge ablates PTFE
- MHD-based acceleration

⇨ **Thrust: µN - mN**

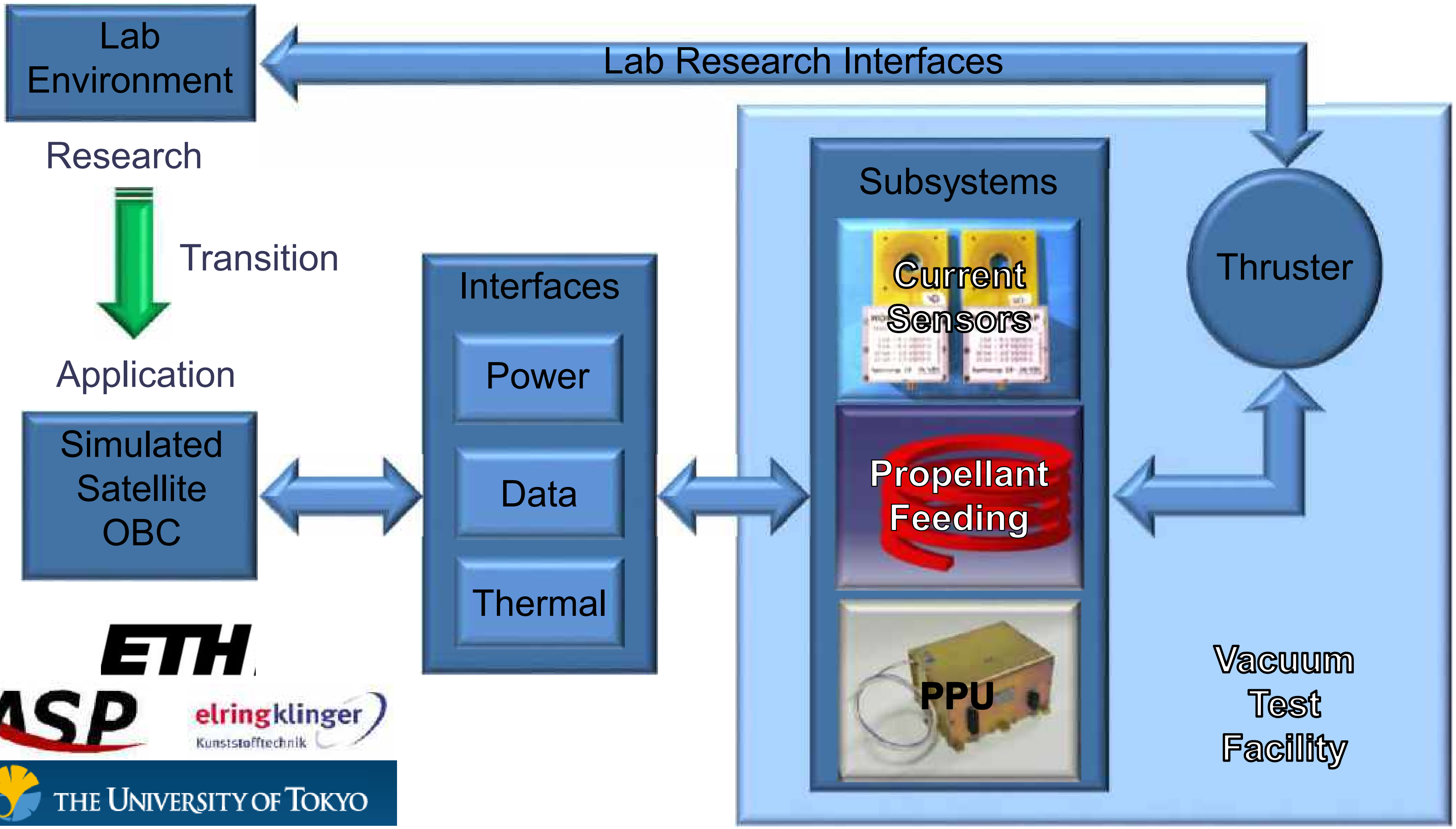
PET



- Discharge ablates PTFE
- Gas dynamic acceleration

⇨ **Thrust: nN - µN**

Thruster System Design Strategy



ETH

ASP

elringklinger
Kunststofftechnik

THE UNIVERSITY OF TOKYO

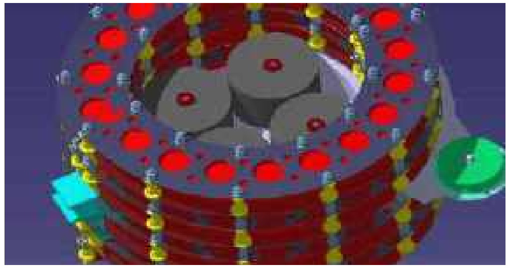
Low Power EP: ADD SIMP-LEX

- Pulsed Plasma Thruster (PPT)
- Propellant: PTFE
- Improved Electrode Design
- High Overall Efficiency
- Increased Lifetime
- Flexible Power Consumption

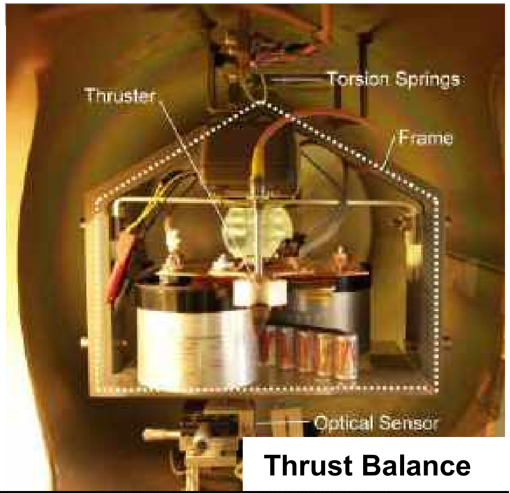
- **Thruster Characteristics:**

Capacity:	**80 µF**
Bank Energy:	**< 68 J**
Pulse Frequency:	**< 2 Hz**
Impulse Bit:	**>1.5 mNs/pulse**
Exhaust Velocity :	**max. 30 km/s**
Thrust Efficiency:	**max. 31 %**

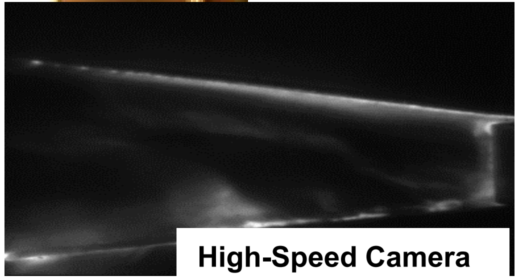
ADD SIMP-LEX for Satellite Missions



Propellant Feeding System
Dupont Award 2007



Thrust Balance



High-Speed Camera

- Main Propulsion System for Lunar Mission BW1
- Secondary: Attitude and Orbit Control
- Cluster of Thrusters for required Δv of 5km/s
- Award-Winning Propellant Feed Concept

- Investigation of ADD SIMP-LEX at IRS and UoT:
  - Thrust Balance + Mass Balance
  - Electrostatic Probes
  - Inductive Magnetic Field Probes
  - High-Speed Camera
  - Voltage and Current Monitors

- Further Research Topics (Cooperation with UoT):
  - Further Lifetime Investigation
  - Plasma Diagnostics
  - **Succesful System Approach (with ASP GmbH)**

ADD SIMP-LEX GSM (TRL = 7)

Test of total system in DLR STG Facility (incl. PPU)

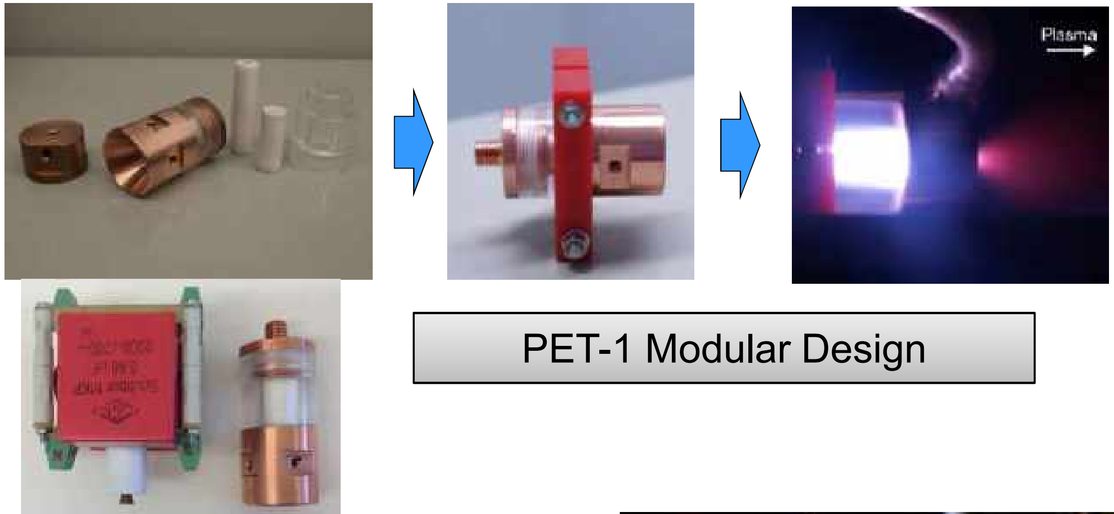
Parameter	Unit	Value
Max. Capacitor Dry Mass	kg	< 6
Max. Pulse Energy	J	67.6
Electrode gap (HxW)	mm	21 x 20
Propellant	-	Solid PTFE
Mechanical Feeding Type	-	Side-fed
Total Capacitance	µF	80 (4 x 20)
Capacitor Type	-	Wound Mica Foil
Capacitor Voltage	V	1300
Igniter Insulation Material	-	Al_2O_3
Igniter Voltage Capability	kV	Up to 20
iMPD Power Consumption	W	< 85
Impulse per Pulse	µNs	1.5
Specific Impulse	s	< 2650
Mass per Pulse	µg	60
Thrust Efficiency	%	~ 30
Demonstrated Pulse Life	-	2 Mio.
Demonstrated Total Impulse	Ns	3000

Scaling – PETRA Thruster TRL 4-5

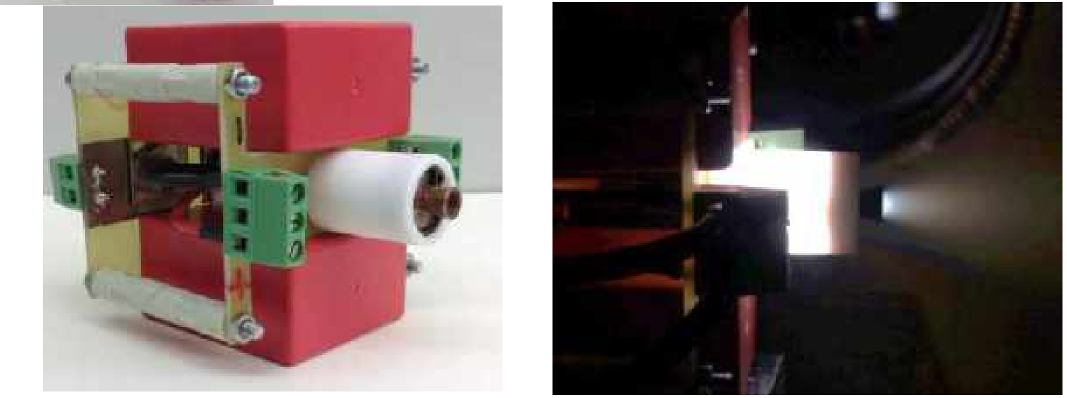
IRS PET-Line Measured Performance

	PET-1	PETRA
Mass Thruster	494 g	16 g!
Max. Pulss-Energy	3 J	2,7 J
Capacitor voltage	< 3 kV	
Spezific Impulse	< 140 s	
Thrust efficiency	2 %	
System Input Power	4 W	3.5 W
Thrust range	1-80 µN	TBD
Propellant mass	5 g	4 g
Design Life	50000 Pulses	
Total Impulse	< 7 Ns	

Very low system mass!
(Kondensator ESA/SCC 3006/022)



PET-1 Modular Design



PETRA Prototype of MikroThruster

iMPD & PPT - Facilities at IRS

Test Heritage for PPTs at IRS:

- Development Testing (Miniaturization, Compatibility, Thrust Modulation, etc.), Thrust pendulum (> 1μNs)
- Functional Verification Testing (Thruster, System Hardware, Coupling etc.)
- Performance Characterization (Thruster and Subsystem Parameters, Magnetic field, (Fast) Cameras)
- Confidence Life Testing (Full Scale and Accelerated)


Inductive Probe vs 10 cent

Tank 16:	Tank 19:	Tank 17:	Tank 18:
Performance	**Miniaturization**	**Intrusive probes**	**Conf. Life Time**
Ø 0.5 m, L: 1.6 m	Ø 0.3 m, L: 0.5 m	Ø 0.3 m, L: 1.0 m	Ø 0.4 m, L: 0.6 m
p ~ 10^{-5} hPa	p ~ 10^{-6} hPa	p ~ 10^{-6} hPa	p ~ 10^{-6} hPa

Motivation for (steady state) MPD-Thrusters

Why MPD Thrusters?

- ⇨ High thrust density and thrust efficiency
- ⇨ High specifc impulse
- ⇨ Throttability
- ⇨ Propellant flexibility

ISRU e.g. Mars Atmosphere !

20 – 50 %
20 – 50 km/s
kW – MW

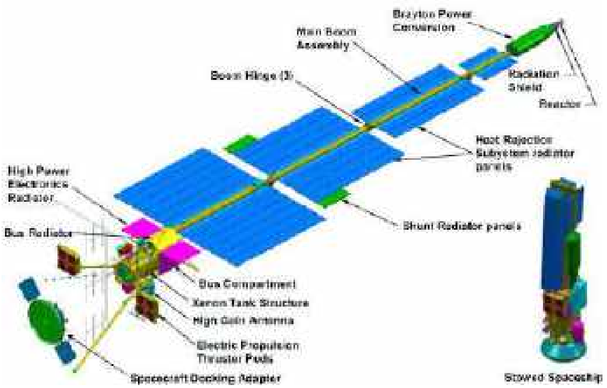
Pulsed
Quasi steady state
Steady state

H_2, N_2, NH_3,
He, Ne, Ar, Kr, Xe,
Li, Na, K.

Scenarios for high thrust missions:

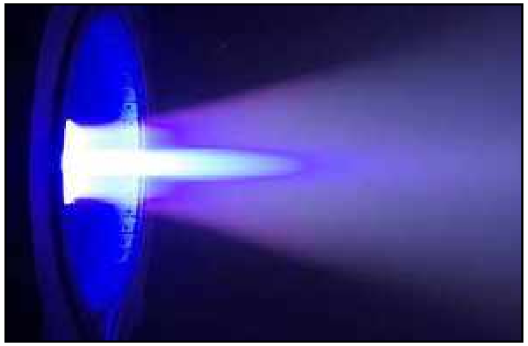
- ⇨ Manned / heavy cargo missions to Mars (100 N and 3000 s)
- ⇨ Manned / heavy cargo missions > 1,5 AU
- ⇨ Un-manned scientific missions (e.g. Kuiper-Belt Objects)

Concept in general: as **SEP** < 1 AU and as **NEP** > 1 AU



Prometheus nuclear electric Deep Space Vehicle, JIMO Mission Module, NASA.

SF MPDT DT6 / IRS



AF MPDT SX3 / IRS

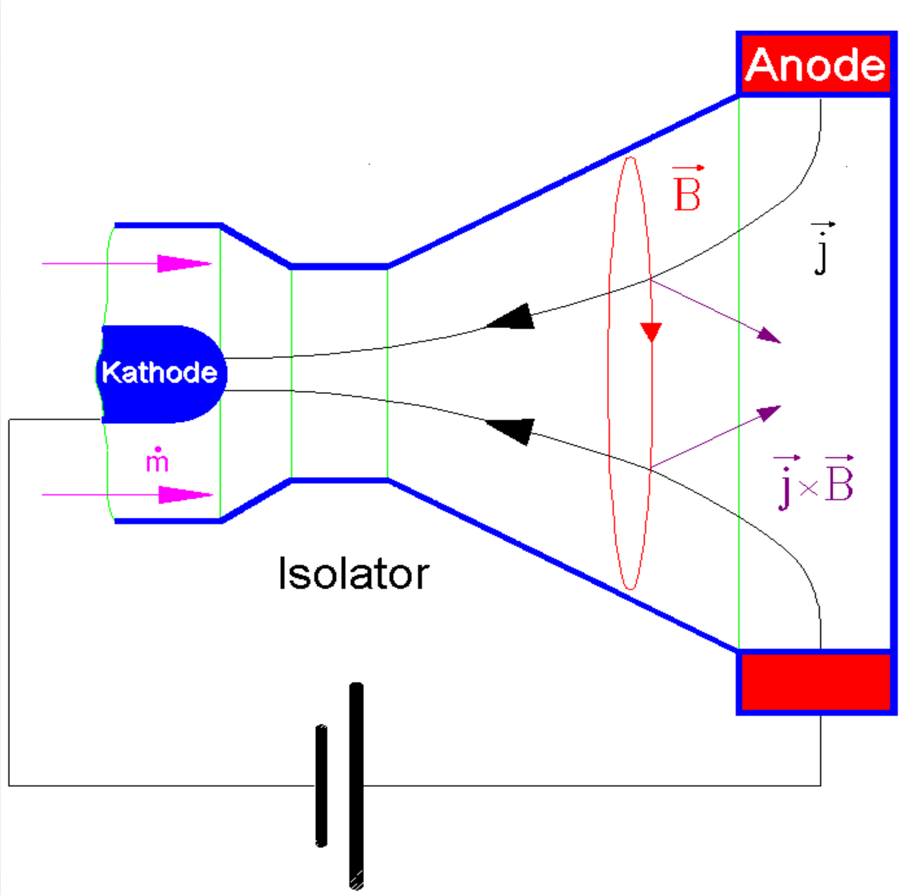
Self-field MPD Thruster: Thrust Generation

through:

- ➢ Thermal expansion
- ➢ Magnetic forces

$$F = C_1 \dot{m} \sqrt{\frac{T_0}{M}} + \frac{\mu_0}{4\pi}\left(\frac{3}{4} + \ln\frac{r_a}{r_c}\right) I^2$$

thermal magnetic



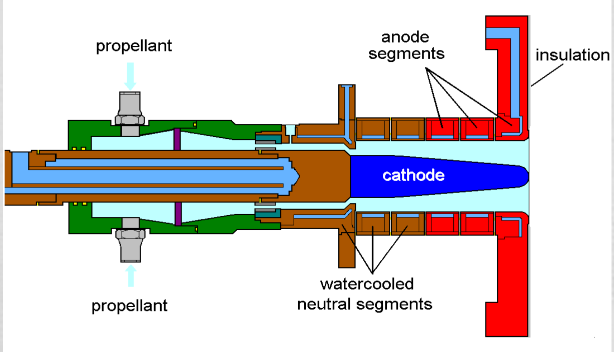
For high magnetic thrust:

- ➢ Propellant should be fully ionized
- ➢ Magnetic forces ~ I²
  - → high current levels necessary
  - → low voltage levels are desired

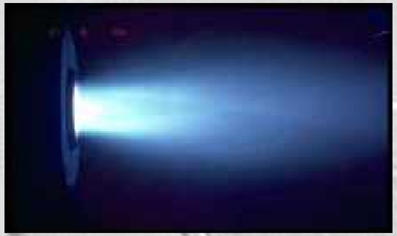
For high thermal thrust:

- ➢ Light weight propellant
- ➢ Nozzle geometry

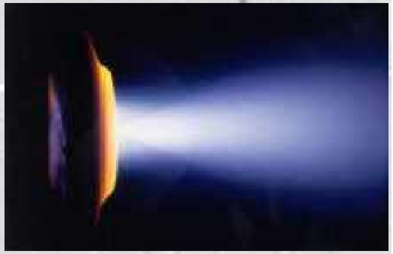
Steady State self-field (SF) MPD thrusters

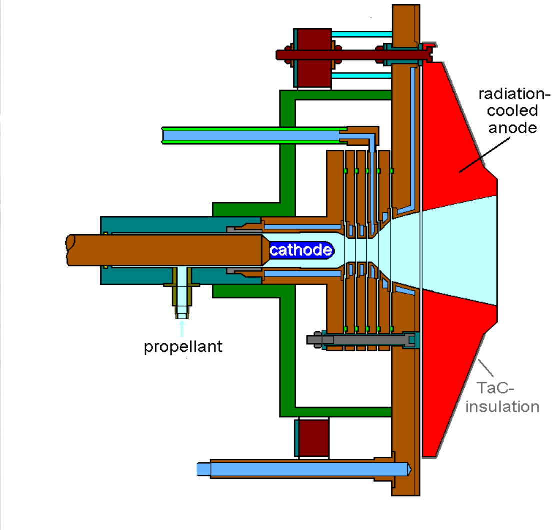


ZT

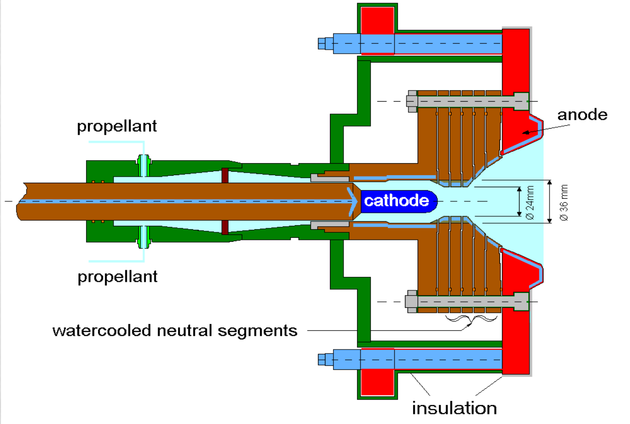


HAT



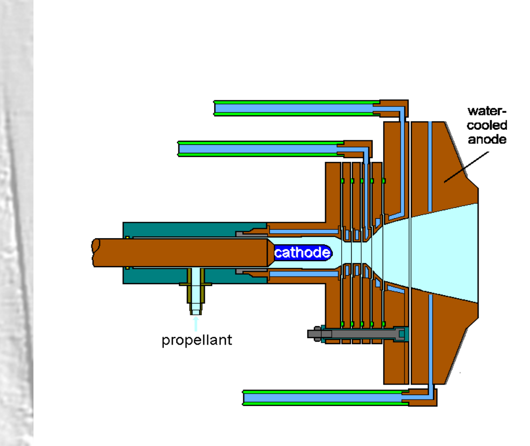


Investigation of **cylindrical thrusters** (ZT) and **nozzle type** thusters (with water-cooled (DT, CAT) and radiation-cooled (HAT) anode),
Power up to 550 kW, current up to 15 kA, thrust up to 27 N and efficiency up to 27% with argon as propellant.

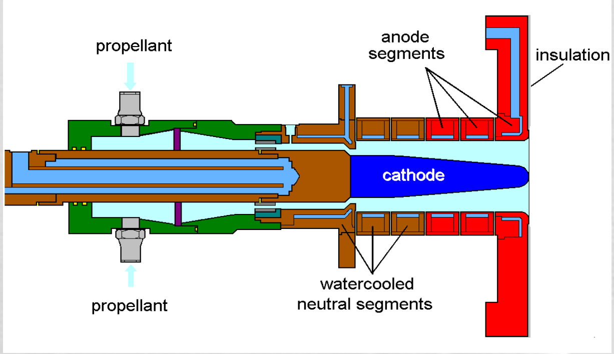


DT

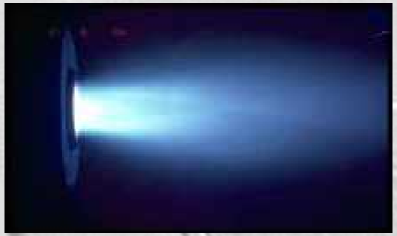
CAT

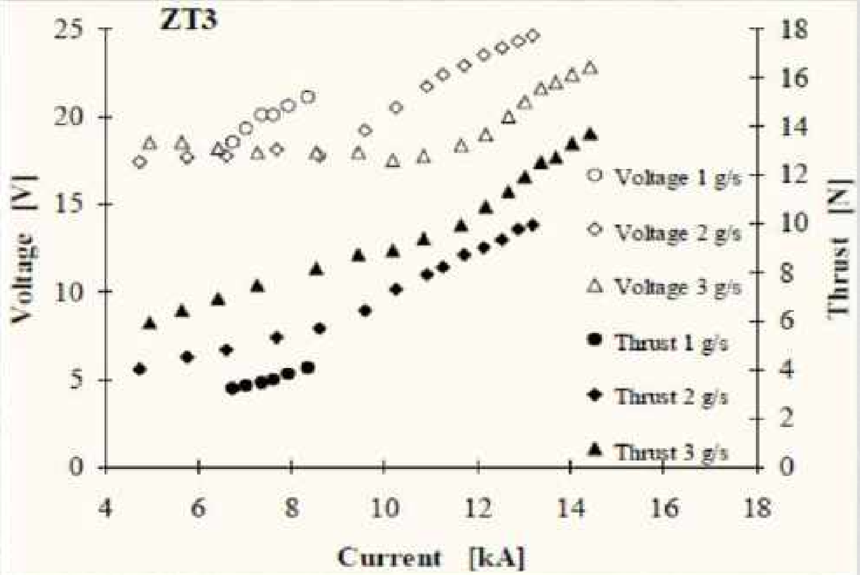


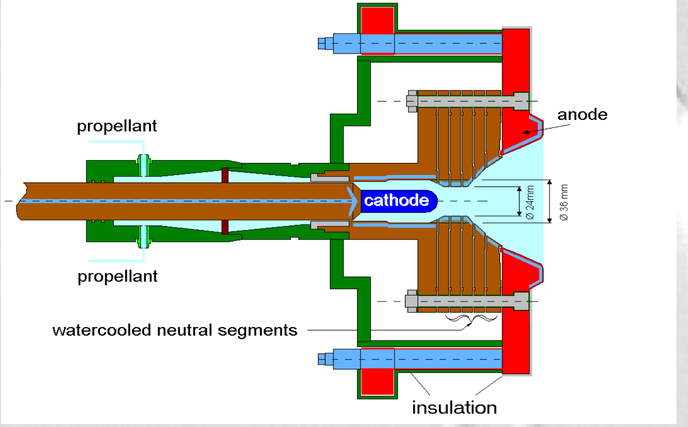
Steady State SF MPD thrusters



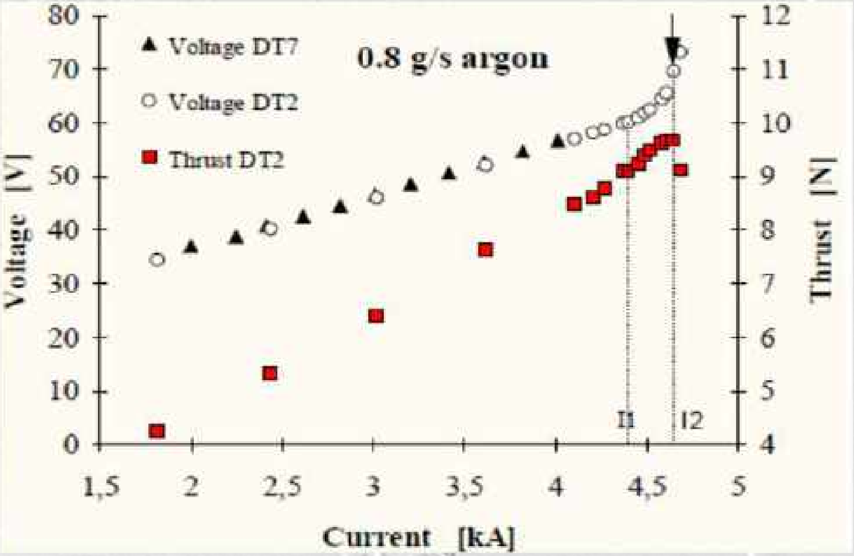
ZT



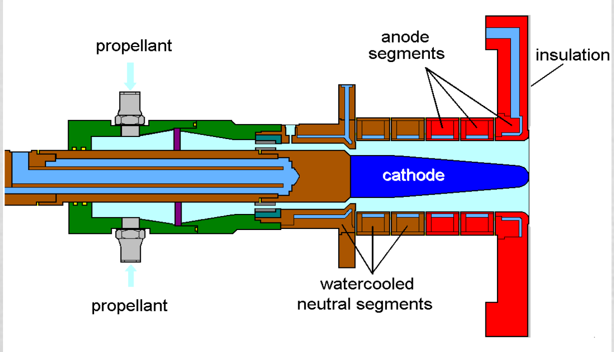




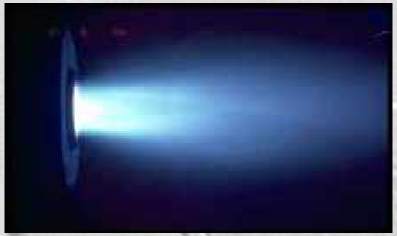
DT

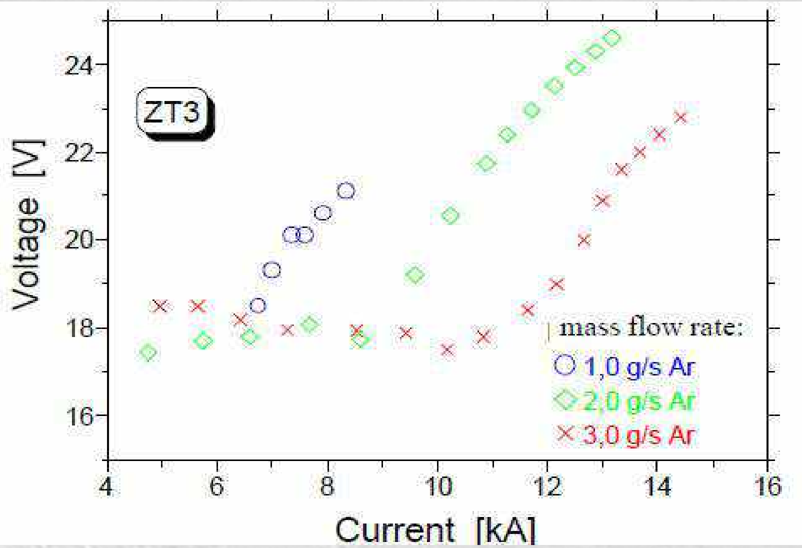


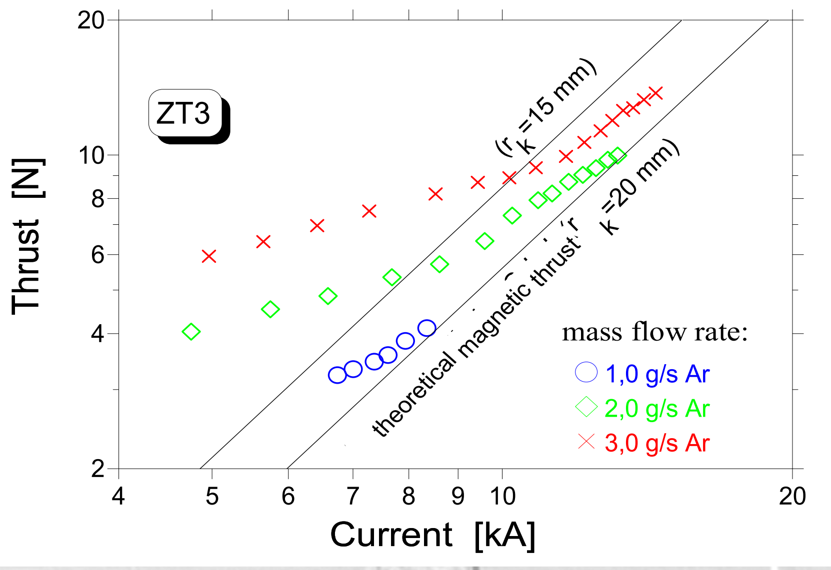
Steasy State self-field MPD thrusters



ZT





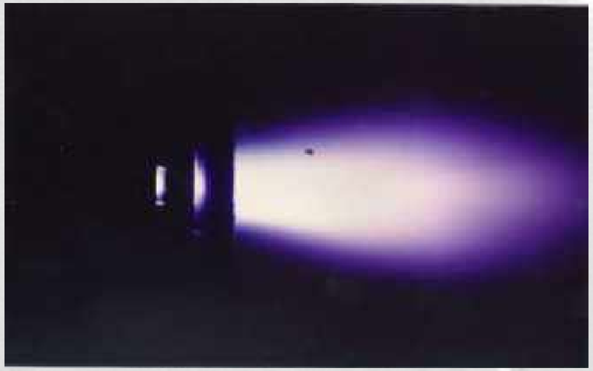


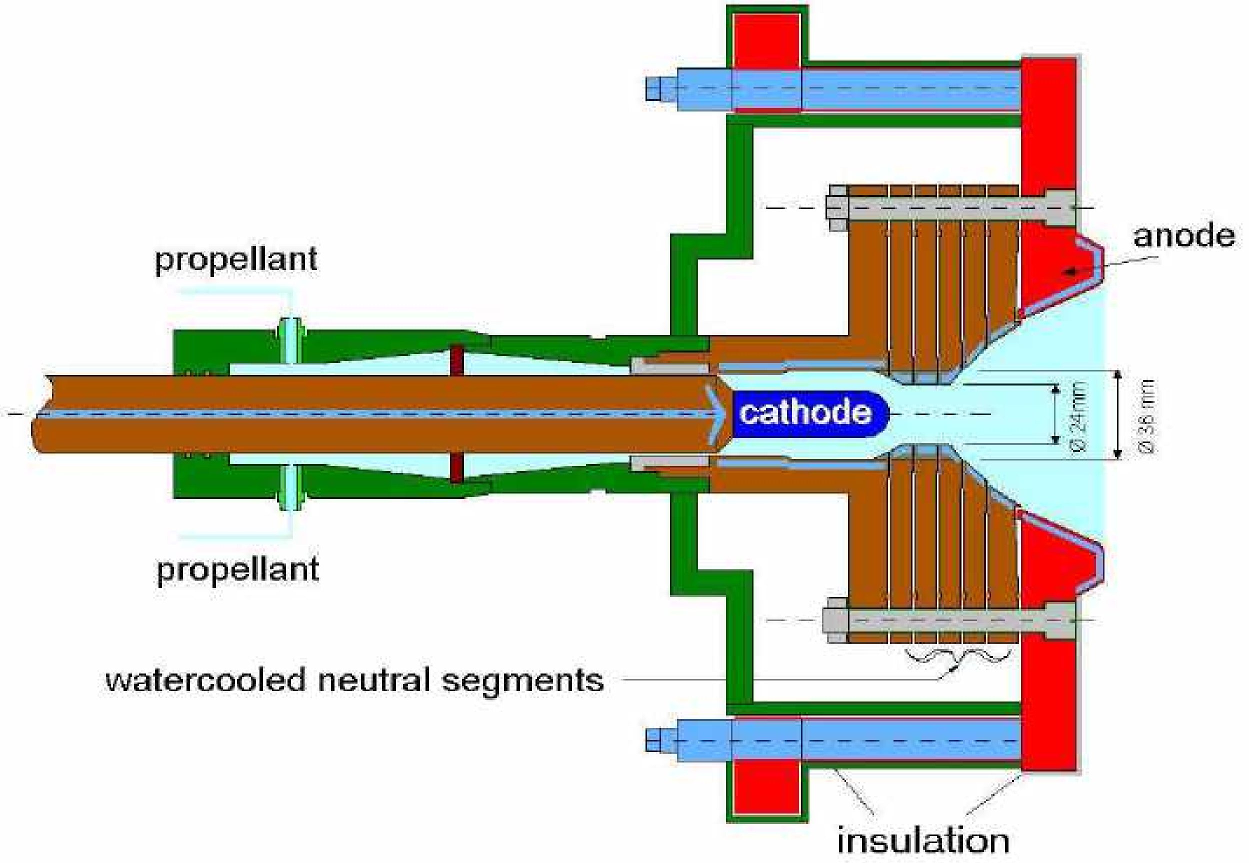
- Thrust values up to 15 N
- No instabilities up to 15 kA at 0.8 g/s (Argon)
- Electrical power up to 350 kW

SF-MPD Nozzle-Type Thrusters (DT) – at IRS

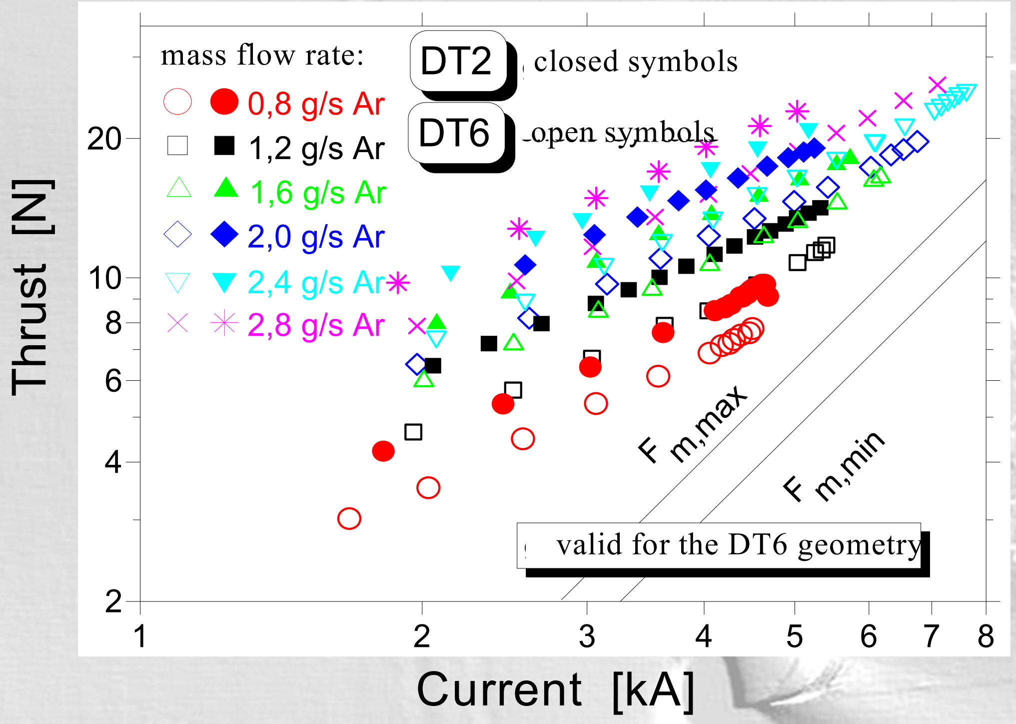
Laboratory devices:

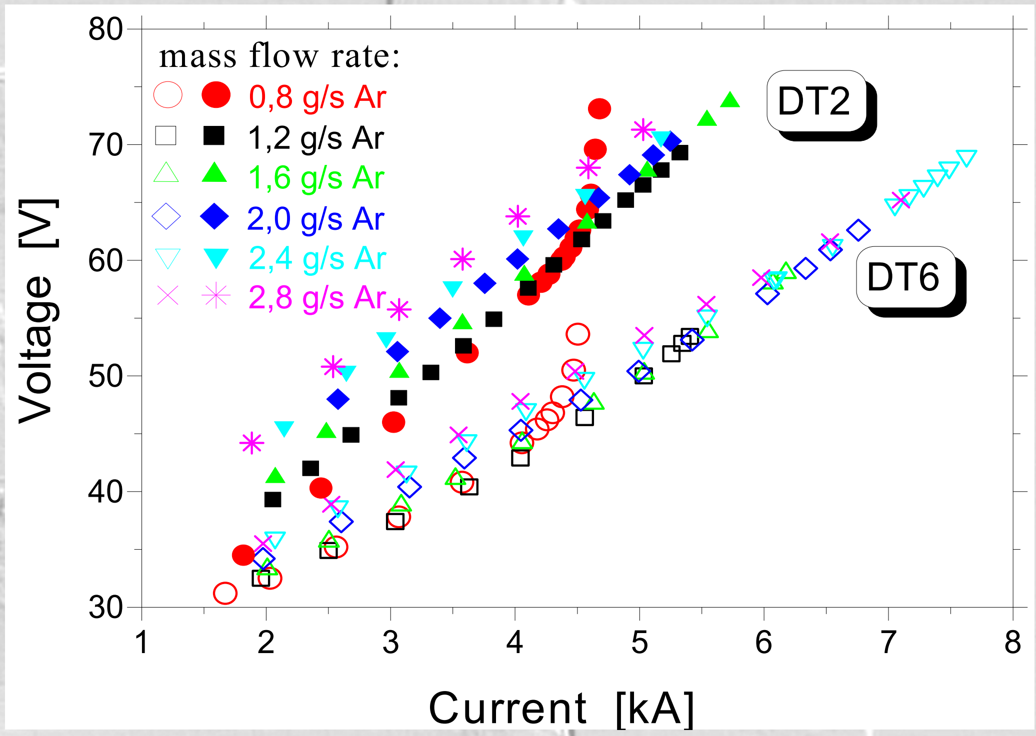
- Anode and segments water-cooled and made from Copper
- Power level:
  several 100 kW up to 1 MW
- Designs differ in cathode and nozzle throat diameter
- Operated with
  Argon, Nitrogen
  and Hydrogen





SF-MPD Nozzle-Type Thrusters (DT) – Performance Data

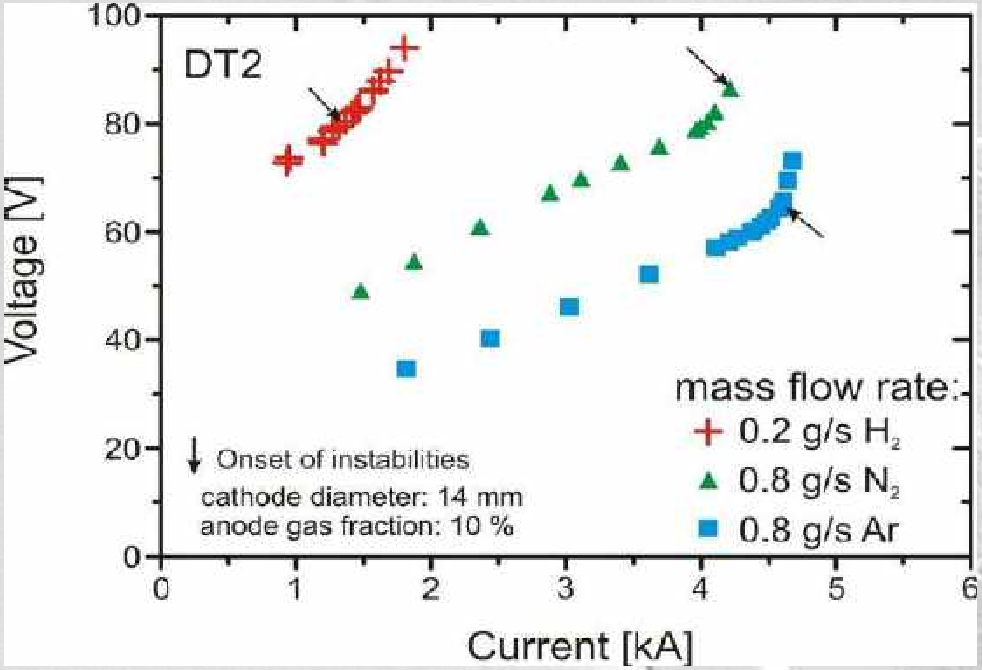


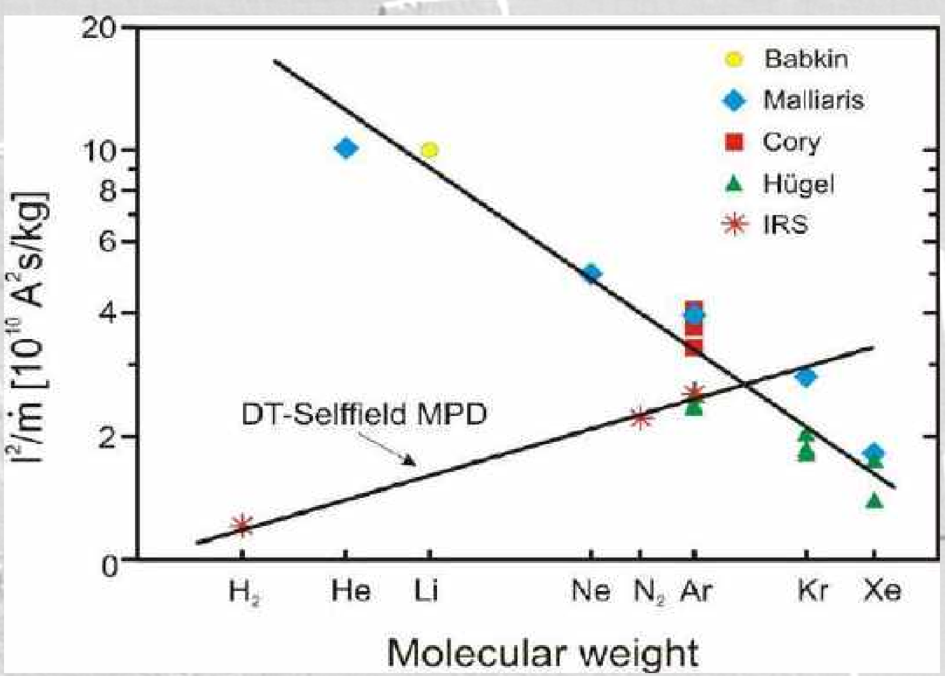


Operated up to 25 N at 8000 A with Argon

Plasma instabilities limitation at ca. 15 km/s with Ar, H_2 and N_2

SF-MPD Nozzle-Type Thrusters (DT) – Plasma Instabilities

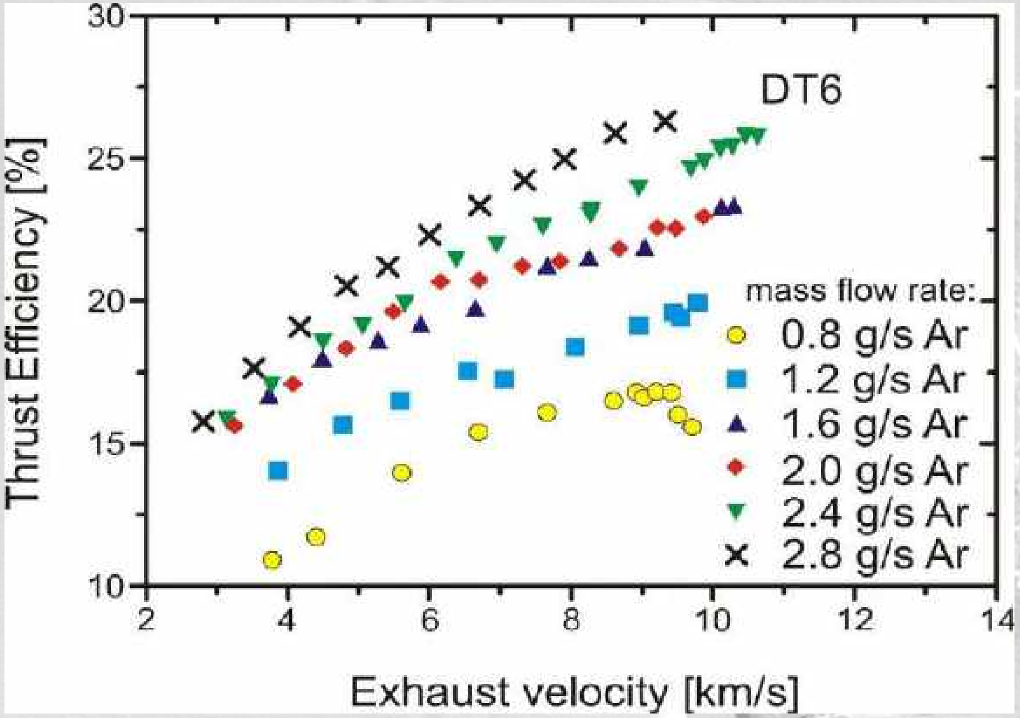




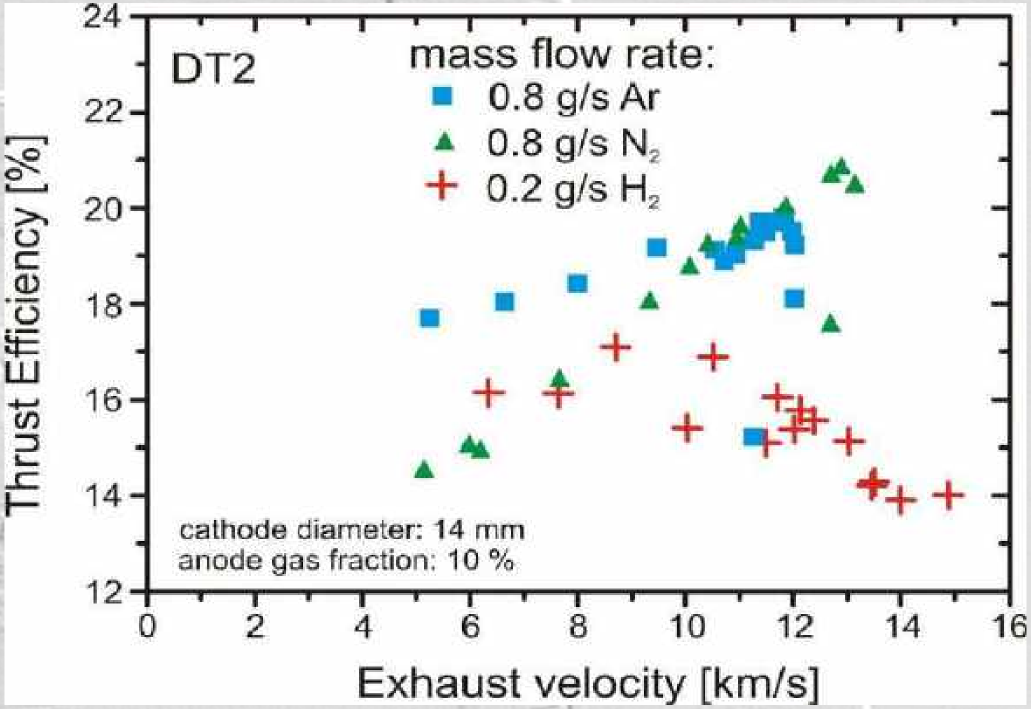
Scaling of Nozzle Equipped steady state SF MPD (Schmidt)

- Hydrogen data are significantly lower in power (I_{max}=1,35 kA)
- Argon data as shown here

SF-MPD Nozzle-Type Thrusters (DT) – Performance Data

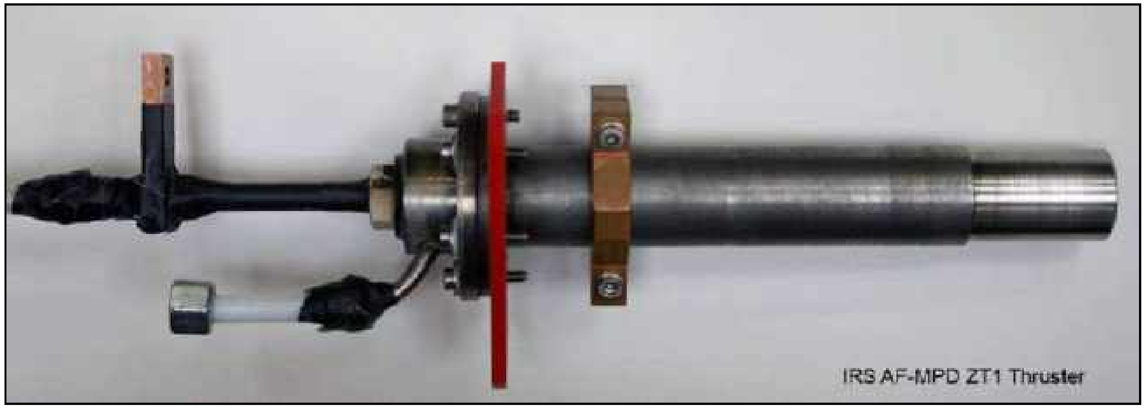


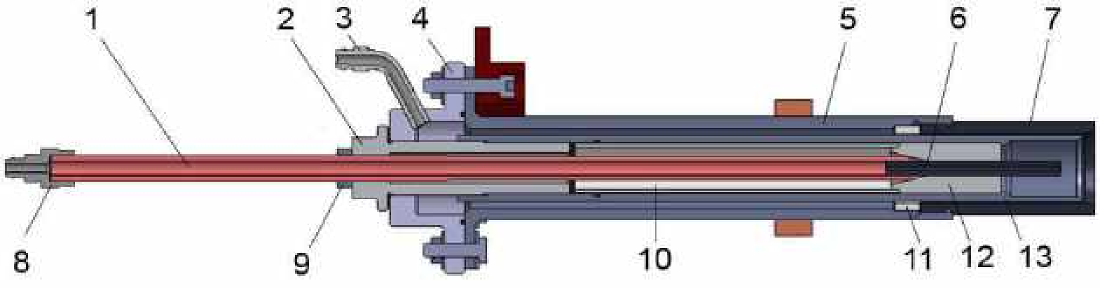
DT6: thrust efficiency vs. exhaust velocity



DT2: thrust efficiency vs. exhaust velocity

IRS 10 kW Steady State AF-MPD ZT1 Thruster





1. **Cathode liner (Cu)**
2. **Insulator (Peek)**
3. **Anode gas connector**
4. **Back flange (SS)**
5. **Anode liner (TZM)**
6. **Hollow cathode (WT20)**
7. **Anode (WL10)**
8. **Cathode gas connector**
9. **Fixing**
10. **Insulation tube (ceramic)**
11. **Insulation injector (BN)**
12. **Cathode centering (BN)**
13. **Neutral liner (TZM)**

- Ignition of thruster achieved at applied-fields up to ~ 0.1 T
- Maximum operational period: 70 s
- Thrust measurement: stationary conditions of coil cooling needed

→ <u>Thrust can be measured at applied-field flux up to ~ 0.1 T</u>

<u>Current characteristic of AF-MPD ZT1 thruster:</u>

- Argon as propellant
- Propellant injection at anode and through cathode
  → two mass flow rates (mass flow rate ratio variation)

- Passive cooling

▪ Power:	~ 6 kW	
▪ Mass flow rate:	~ 7 mg/s (anode + cathode)	
▪ Applied-field:	up to 0.1 T	(0.35 T)
▪ Thrust:	up to 80 mN	(140 mN)
▪ Specific impulse:	up to 1000 s	(1800 s)
▪ Thrust efficiency:	up to 8 %	(18 %)

<u>Extrapolated characteristics of AF-MPD ZT1 thruster at 0.6 T:</u>

▪ Power:	12 kW
▪ Mass flow rate:	6/1 mg/s (anode/cathode)
▪ Applied-field:	0.6 T
▪ Thrust:	up to 250 mN
▪ Specific impulse:	up to 3600 s
▪ Thrust efficiency:	up to 38 %

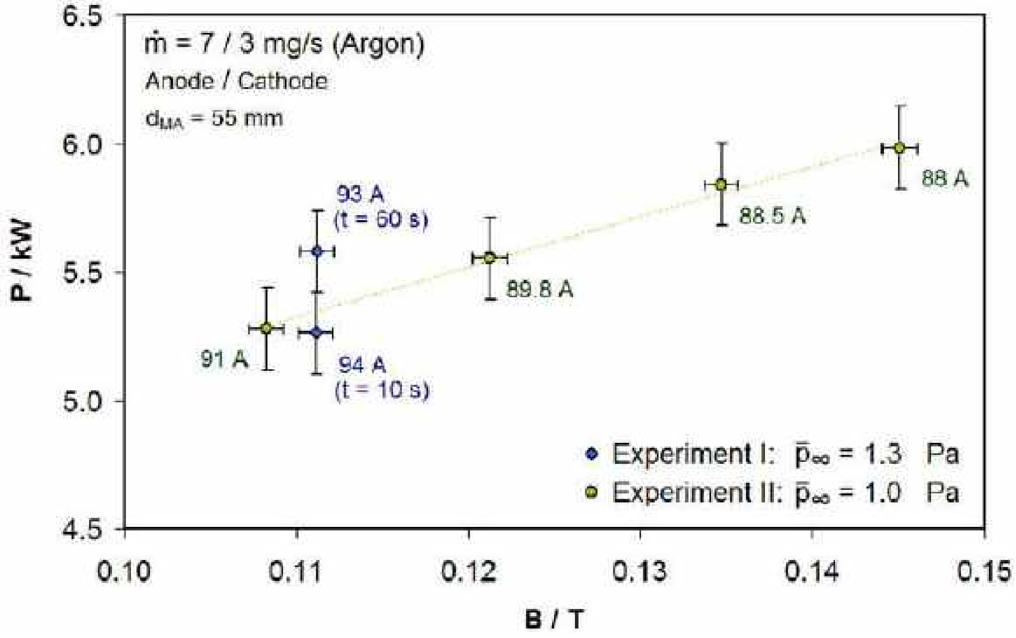
(Comparable with DLR's X16 thruster!)

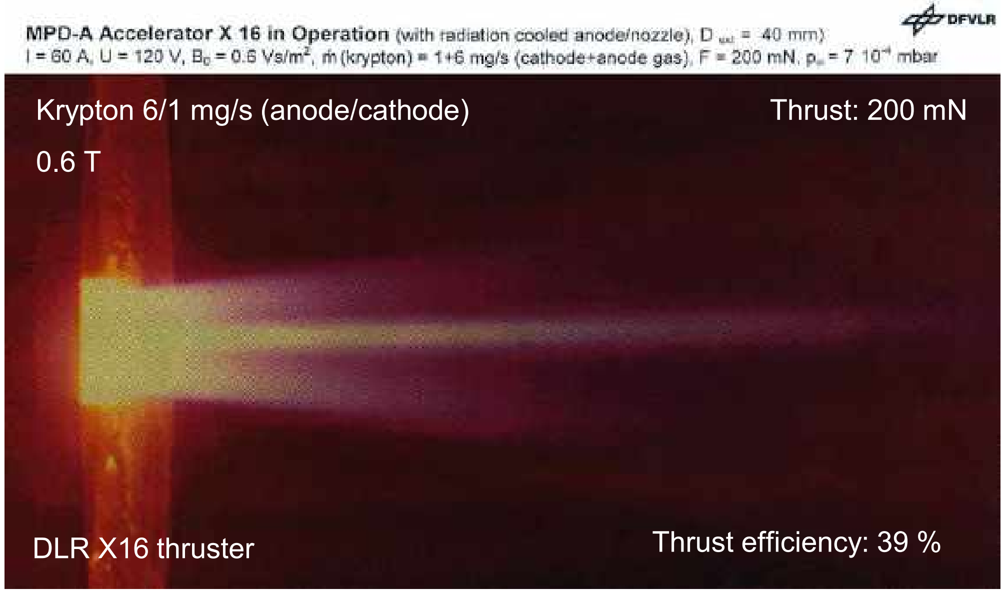
Experimental Results of AF-MPD ZT1 Thruster

Initial operation of AF-MPD ZT1 Thruster

Results:

- Discharge voltage and power are proportional to B
  → **Agreement with other thrusters and respective models**
- Stable steady state operation
- Non stationary conditions of anode temperature
- Plume similarity compared to DLR’s X16 thruster





MPD-A Accelerator X 16 in Operation (with radiation cooled anode/nozzle), D_{anl} = 40 mm)
$I = 60$ A, $U = 120$ V, $B_0 = 0.6$ Vs/m², $\dot{m}$ (krypton) = 1+6 mg/s (cathode+anode gas), $F = 200$ mN, $p_\infty = 7 \cdot 10^{-4}$ mbar

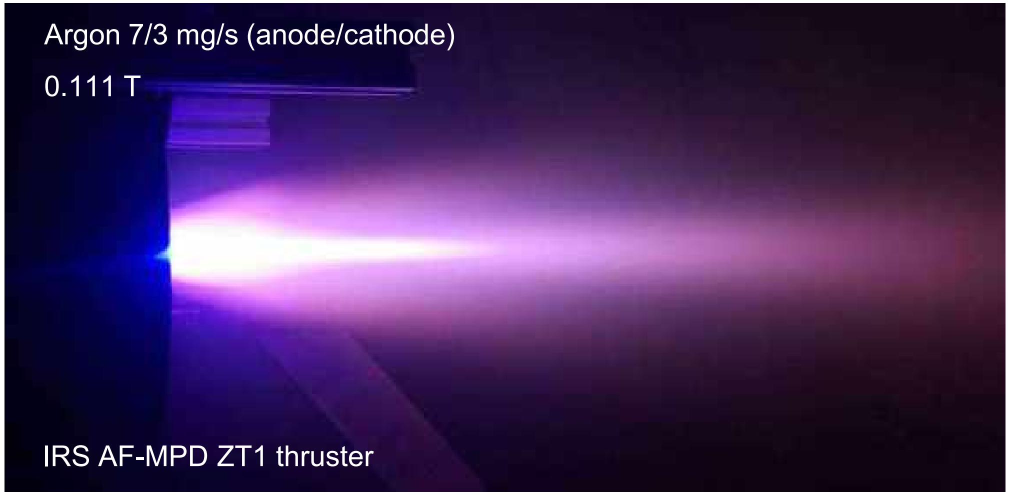
Krypton 6/1 mg/s (anode/cathode)

Thrust: 200 mN

0.6 T

DLR X16 thruster

Thrust efficiency: 39 %



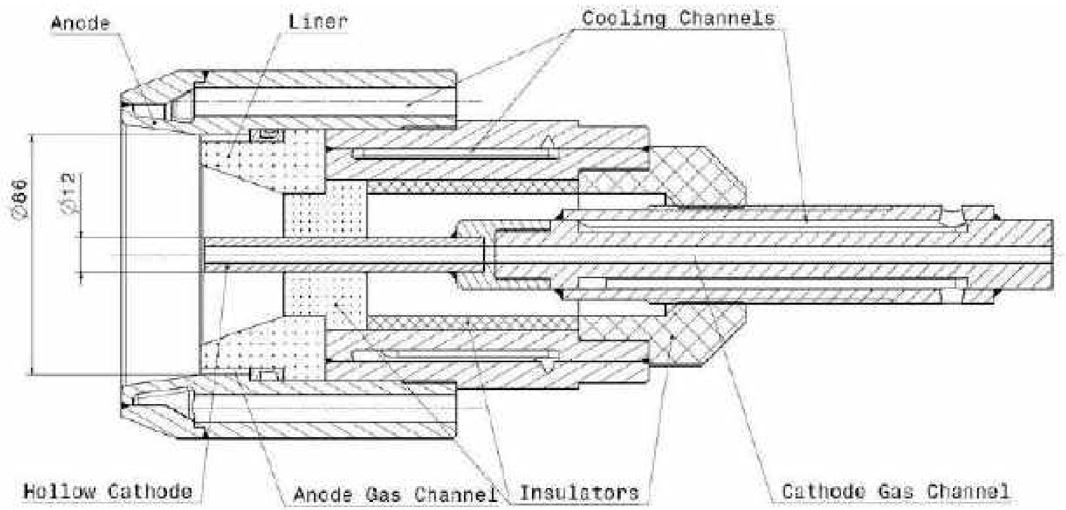
Argon 7/3 mg/s (anode/cathode)

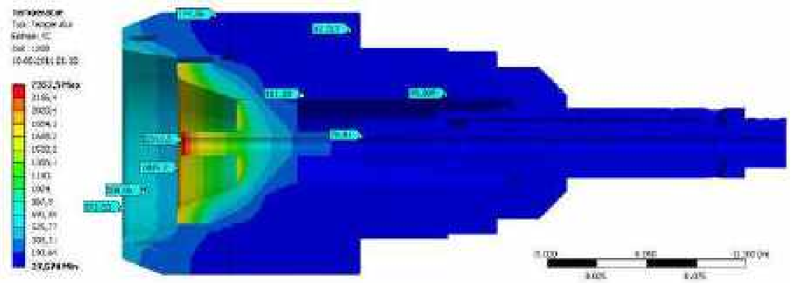
0.111 T

IRS AF-MPD ZT1 thruster

IRS 100 kW Steady State AF-MPD SX3 Thruster

Design of AF-MPDT SX3:





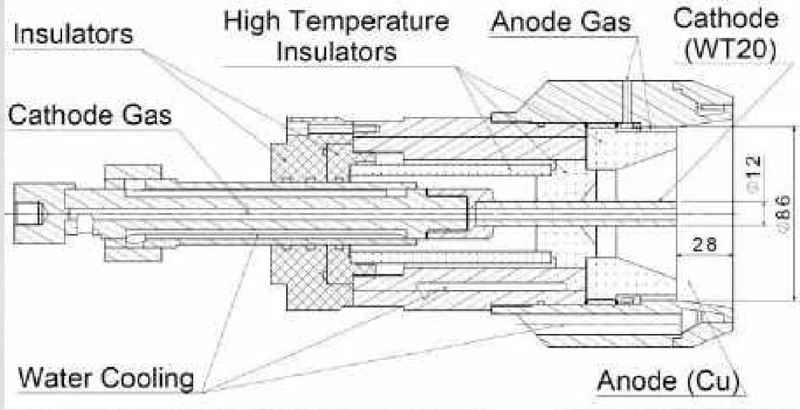
Expected Performance:

Discharge power:	~ 100 kW
Discharge current:	~ 250 - 1500 A
Applied-field:	~ 0.1 - 0.6 T
Thrust:	~ 2.5 - 3 N
Specific impulse:	~ 3000 s
Efficiency:	~ 35 %

Specifications:

- Cold anode (HP water cooled)
- Modularity: anode (cooper) / cathode (hollow cathode, WT20)
- Passively cooled anode for future activities considered
- Propellant injection: at anode and through hollow cathode
- Changeable anode gas injector
- Variable applied-field geometry (new coil)
- Possible additional coil segments considered (increase of B)

Steady State AF-MPD SX3 Thruster (100 kW class)



- Cost efficient laboratory model
- Applied field up to 400 mT, arc current up to 1kA
- Anode + cathode gas injection (argon)

Motivation for (steady state) MPD-Thrusters

Why MPD Thrusters?

- ⇨ High thrust density and thrust efficiency
- ⇨ High specifc impulse
- ⇨ Throttability
- ⇨ Propellant flexibility

ISRU e.g. Mars Atmosphere !

20 – 50 %
20 – 50 km/s
kW – MW

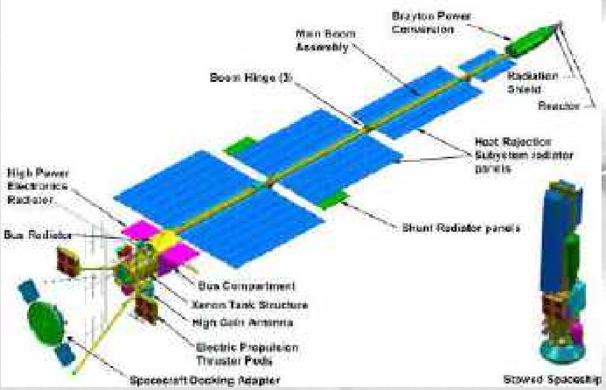
Pulsed
Quasi steady state
Steady state

H_2, N_2, NH_3,
He, Ne, Ar, Kr, Xe,
Li, Na, K.

Scenarios for high thrust missions:

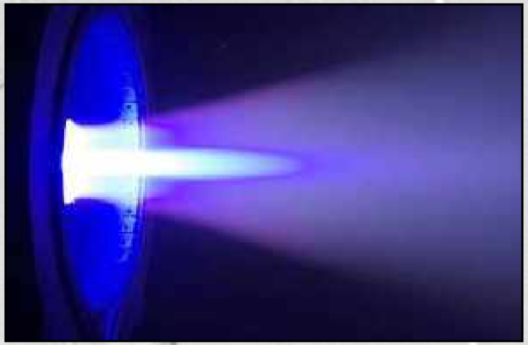
- ⇨ Manned / heavy cargo missions to Mars (100 N and 3000 s)
- ⇨ Manned / heavy cargo missions > 1,5 AU
- ⇨ Un-manned scientific missions (e.g. Kuiper-Belt Objects)

Concept in general: as **SEP** < 1 AU and as **NEP** > 1 AU



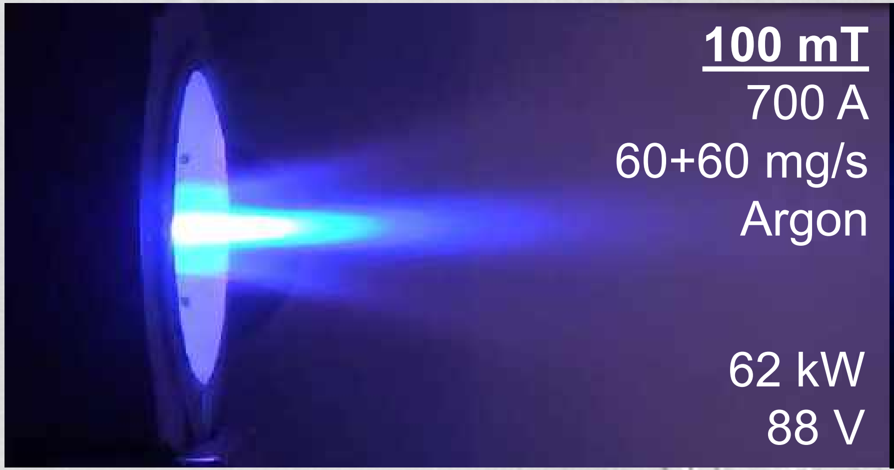
Prometheus nuclear electric Deep Space Vehicle, JIMO Mission Module, NASA.

SF MPDT DT6 / IRS



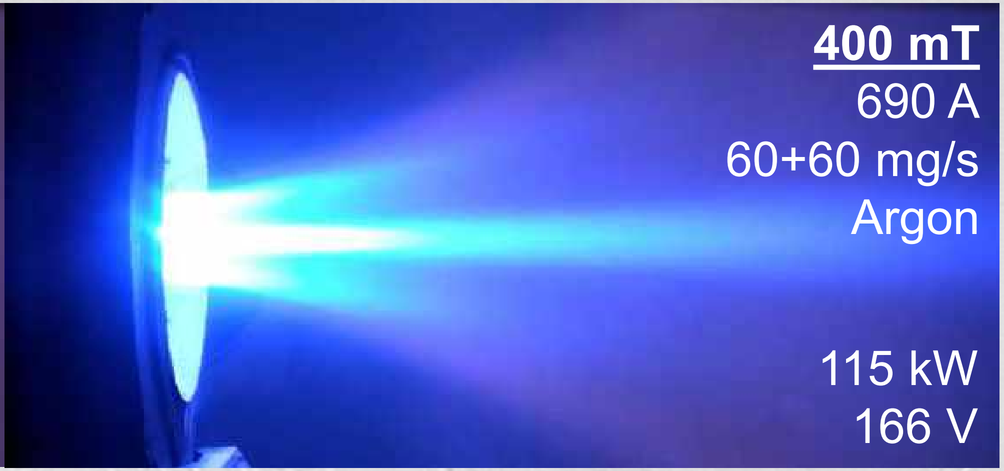
AF MPDT SX3 / IRS

100 kW Class Applied-Field MPD SX3 Thruster: Performance



100 mT
700 A
60+60 mg/s
Argon

62 kW
88 V



400 mT
690 A
60+60 mg/s
Argon

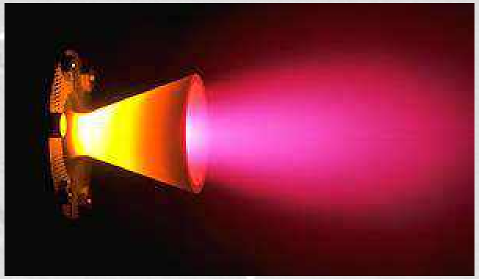
115 kW
166 V

Types of Electric Propulsion (EP)

Electrothermal Thrusters (Arcjets)

- Devices up to 100 kW were tested (TRL 6)
- High thrust levels with exhaust velocities up to 20 km/s
- Light atomic gases as propellant (He, H_2; alternative NH_3)
- Thrust efficiency depends on propellant 20-50%

IRS HiPARC



Steady state Electromagnetic Thrusters (SF-/AF-MPDT)

- Scalable up to MW power range; lab devices up to 1 MW (TRL 4)
- Average thrust levels with ce between 10 and 70 km/s
- Various propellants can be used (H_2, He, Ar, Kr, Xe, Li)
- Comparable high thrust efficiency 20-60%

IRS AF-MPD

Electrostatic Thrusters (HET, GIT)

- Tested in space environment (TRL 9)
- Heavy atomic gases as propellant (Xe)
- GIT: Relative low thrust levels with exhaust velocities of 70-90 km/s and thrust efficiency 60-80%
- HET: average thrust levels with exhaust velocities of 20-50 km/s and thrust efficiency of 40-70%

Advanced Thrusters:

- RF inductive and Helicon
- Hybrid thrusters (TiHTUS, VASIMR)
- IEC, FRC, etc.

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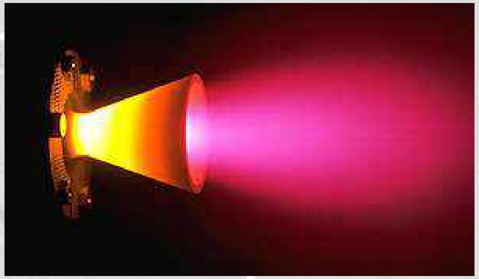

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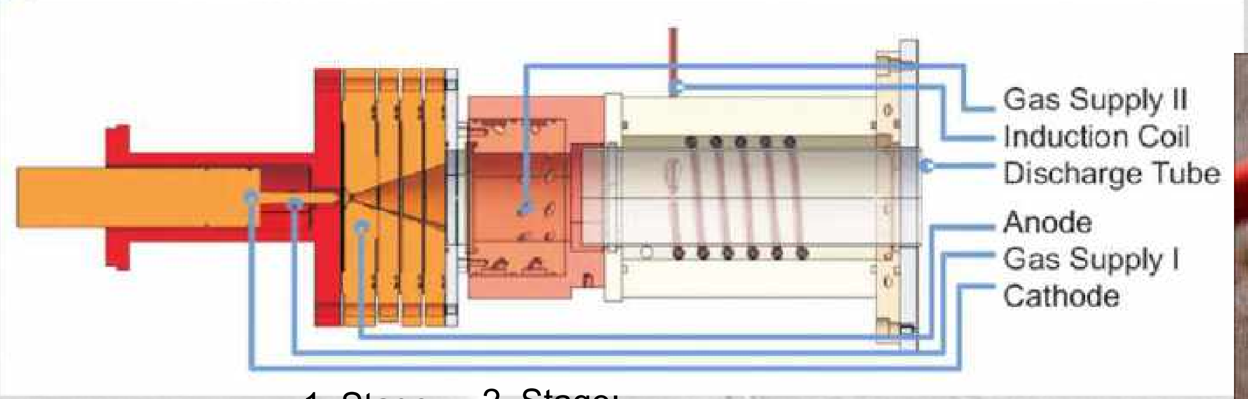
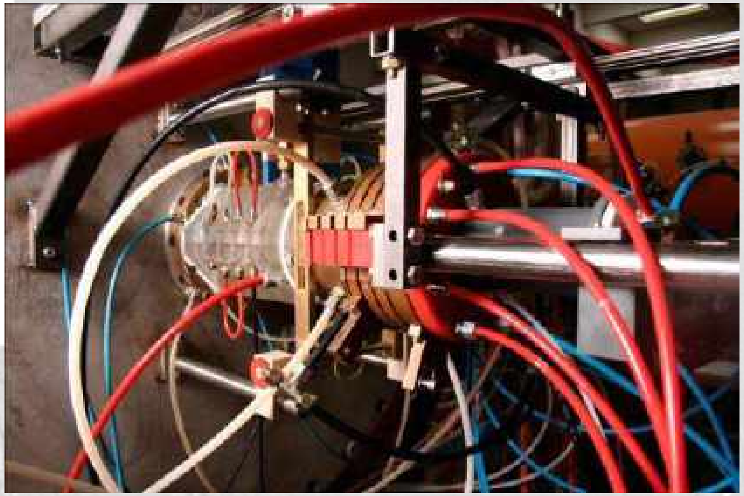
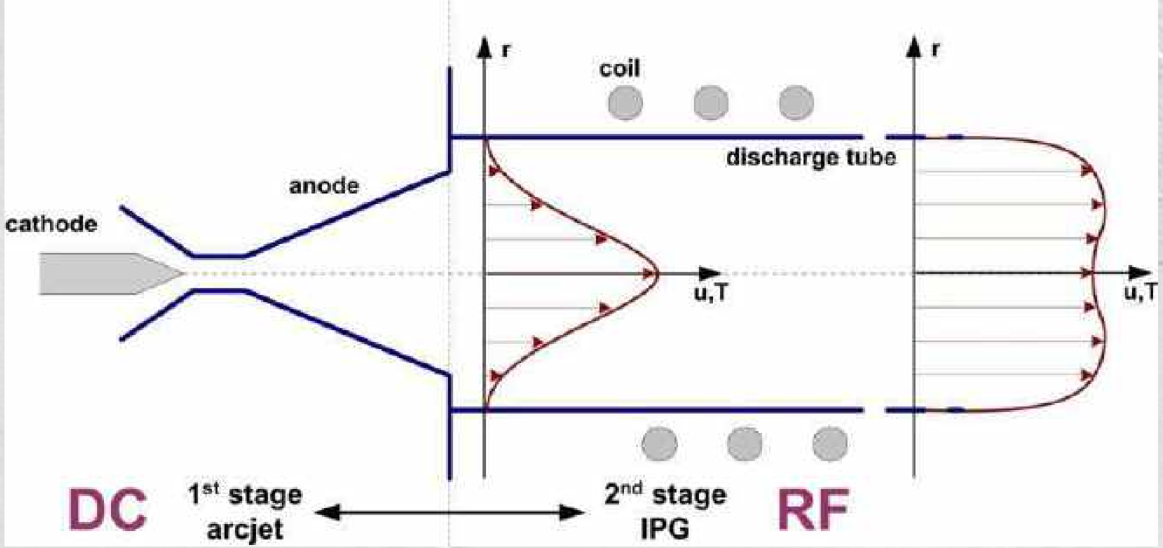
TIHTUS



1. Stage:
Thermal arcjet
(100 kW)

2. Stage:
Inductively heated
(180 kW, 0.5-1.5 MHz)





Working Principle

- Superposed T- and v- profiles (from arcjet and IPG)
  → homogenized profile at outlet
- effective energy coupling
  → higher exhaust velocities

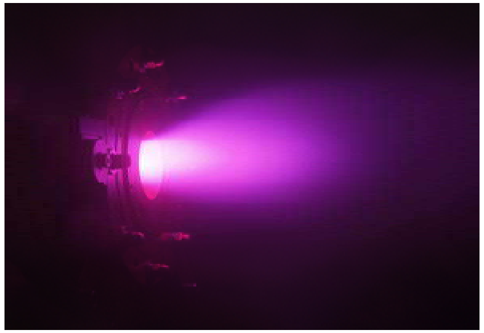
Applications

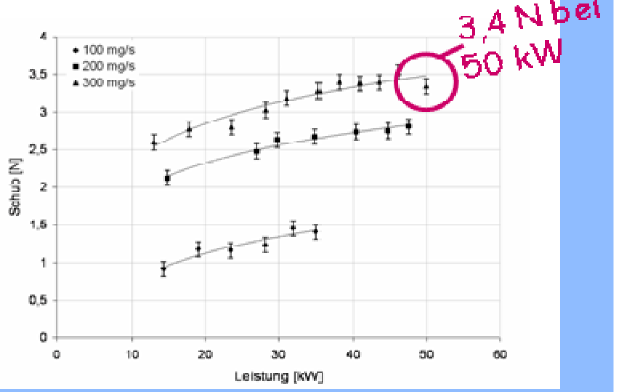
- High power propulsion also mitigating waste mass flow rates (2nd stage) e.g. manned missions to Mars
- Plasma technology

Two-Stage Thruster TIHTUS

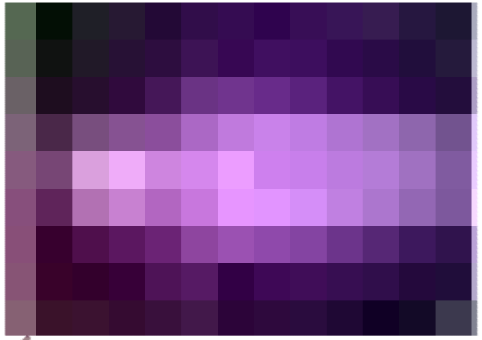
thermal arcjet:
Hot Core, Cold Gas Layer
HIPARC-Series: **100 kW**
>20.000 m/s, >3 N, 29%

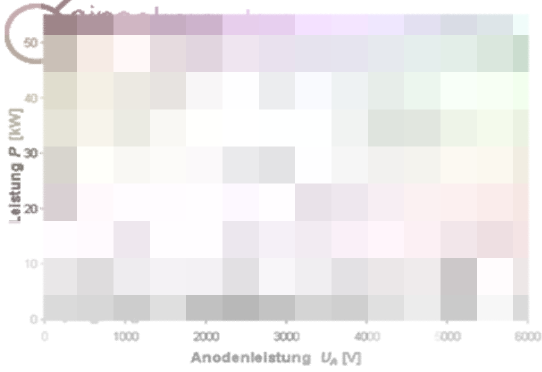
inductively heatedPlasma generator:
Due to skin effect: Major part of power at near-wall position (Skin-Effekt) IPG3 **180 kW-RF**, Operated at 640 kHz

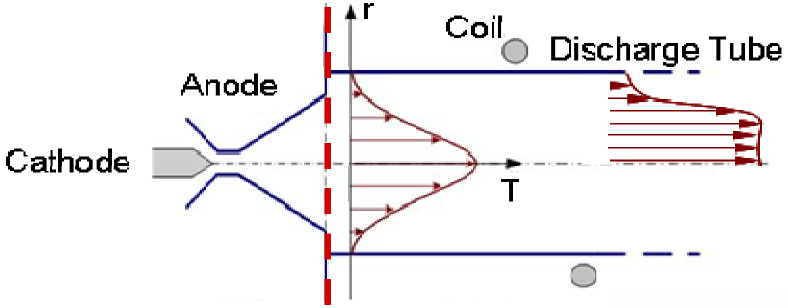


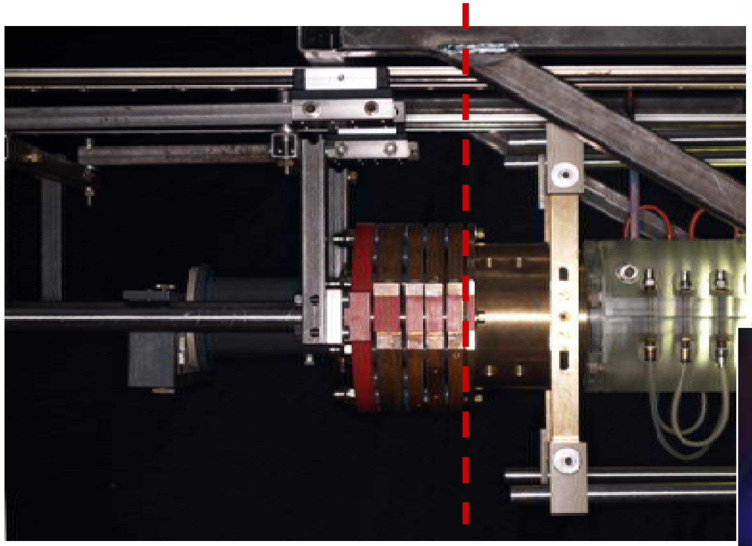


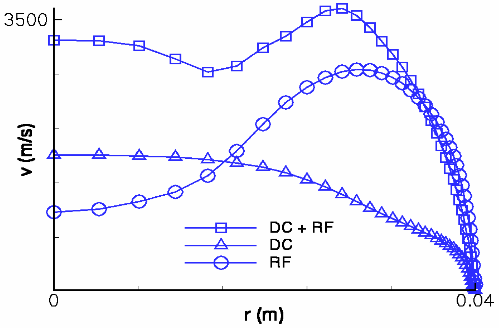
3,4 N bei 50 kW

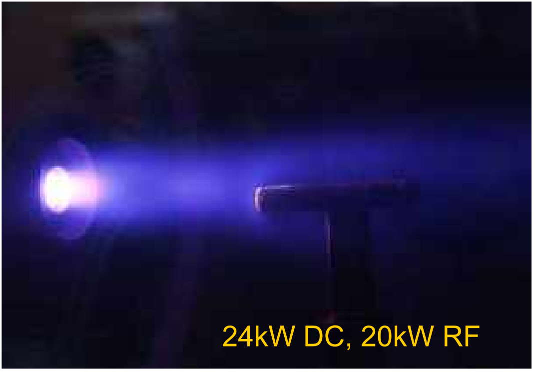












24kW DC, 20kW RF

Inertial Electrostatic Confinement

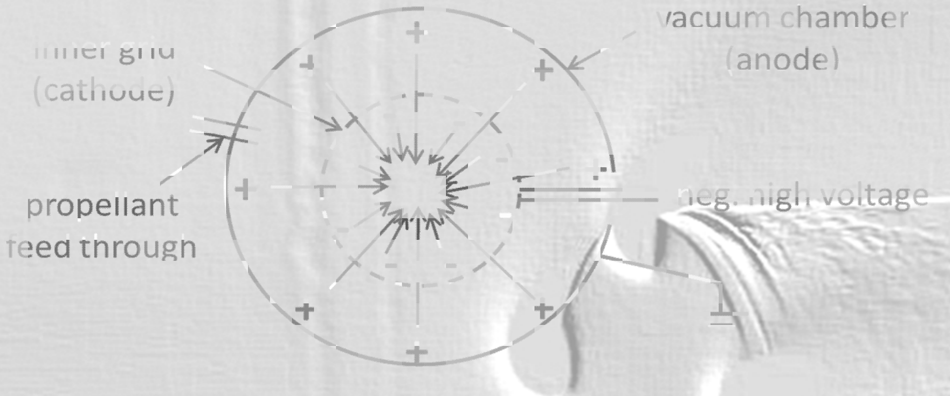


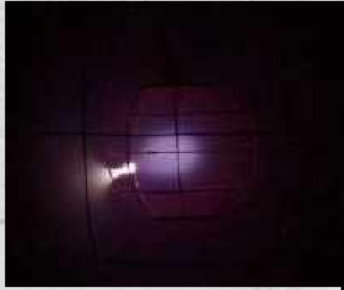
Figure : Scheme of a Farnthworth – Hirsch Fusion Reactor

Principle

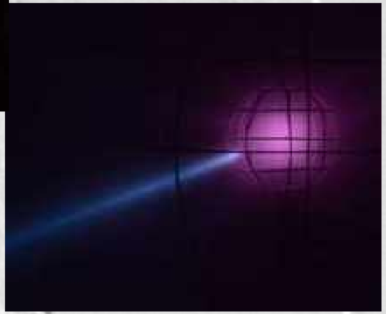
- Grounded chamber flooded with propellant, high negative voltage applied to inner grid
- Glow discharge betw. electrodes generates ions
- Ions accelerated into grid center, collisions may occur resulting in fusion processes
- If ions leave the center due to the kinetic energy, they will be accelerated into the center again
- Extraction of plasma beam from center by allowing for a through in potential surface of cathode

Applications

- Ion thruster with high I_{sp}
- Reference case for PICLAS code
- Simulation of natural plasmas with high energetic radiation (cooperation with **Center for Astrophysics, Space Physics and Engineering Research**, Baylor University)
- Projects with Advanced Concepts Team of **ESA, Gradel and EADS Ottobrunn** (ESA Ariadna study, NEAT)



Star Mode, Ar



“Tight“ Jet Mode, He

Inertial Electrostatic Confinement (IEC): Plasma extraction modes

IEC Configuration:

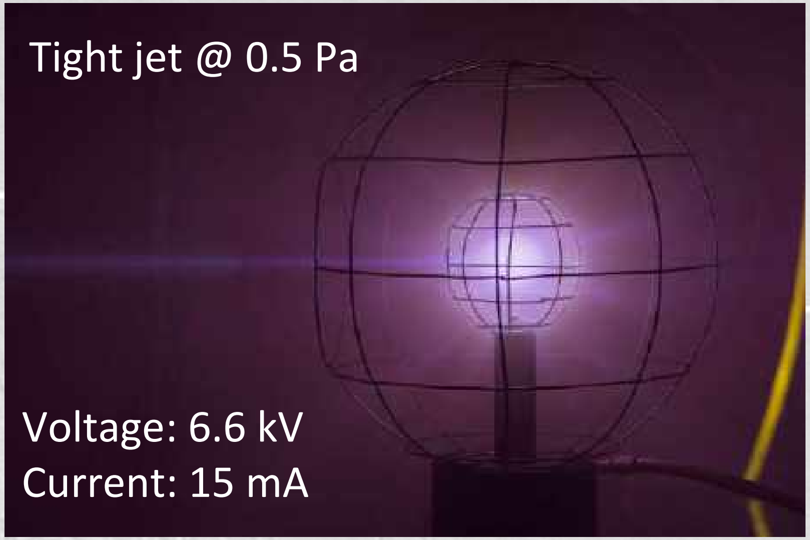
- Length: 8 wires, width: 5 wires
- $D_{cathode}$: 5 cm, D_{anode} : 15 cm

Tightjet Mode:

1. High energy electron beam (*EB*)
2. Compacte plasma jet
3. Low currents (1 ~ 50 mA)

Sprayjet Mode:

1. Diffuse ion jet
2. Higher currents (> 50 mA)
3. High radiation from core region



Tight jet @ 0.5 Pa

Voltage: 6.6 kV
Current: 15 mA

Spray jet @ 0.5 Pa

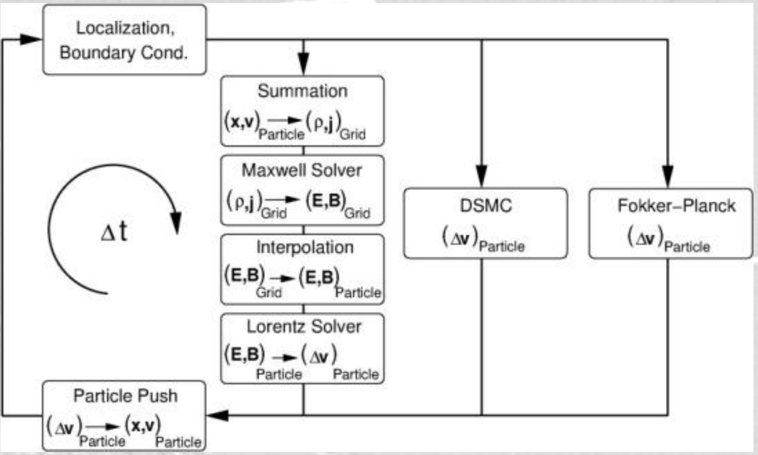
Voltage: ~2.4 kV
Current: ~100 mA

IEC DEVICE FOR SPACE APPLICATIONS

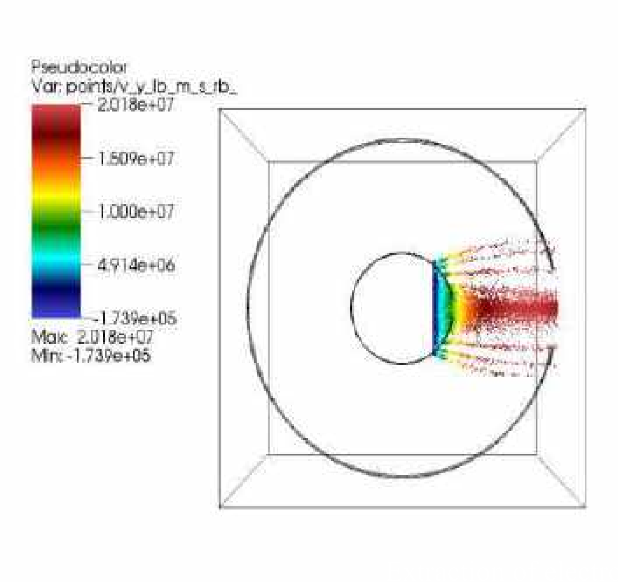
ESA-ARIADNA Study: Kinetic Modelling of Extraction Mechanisms

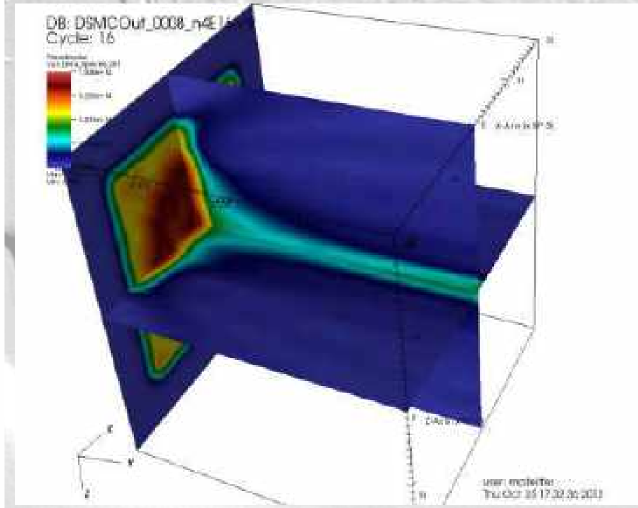
Objectives:

- Investigation of Phenomena leading to jet using particle code
- PICLas: PIC, DSMC and Fokker-Planck

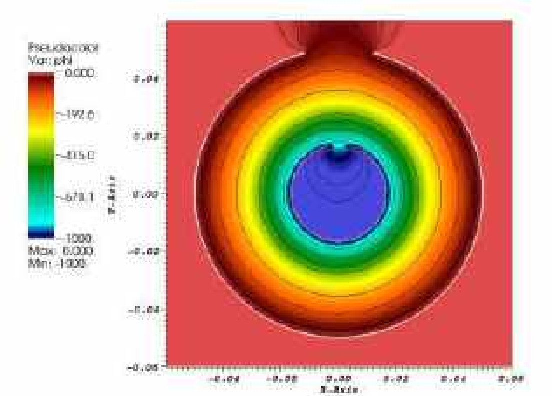


PICLas simulation framework





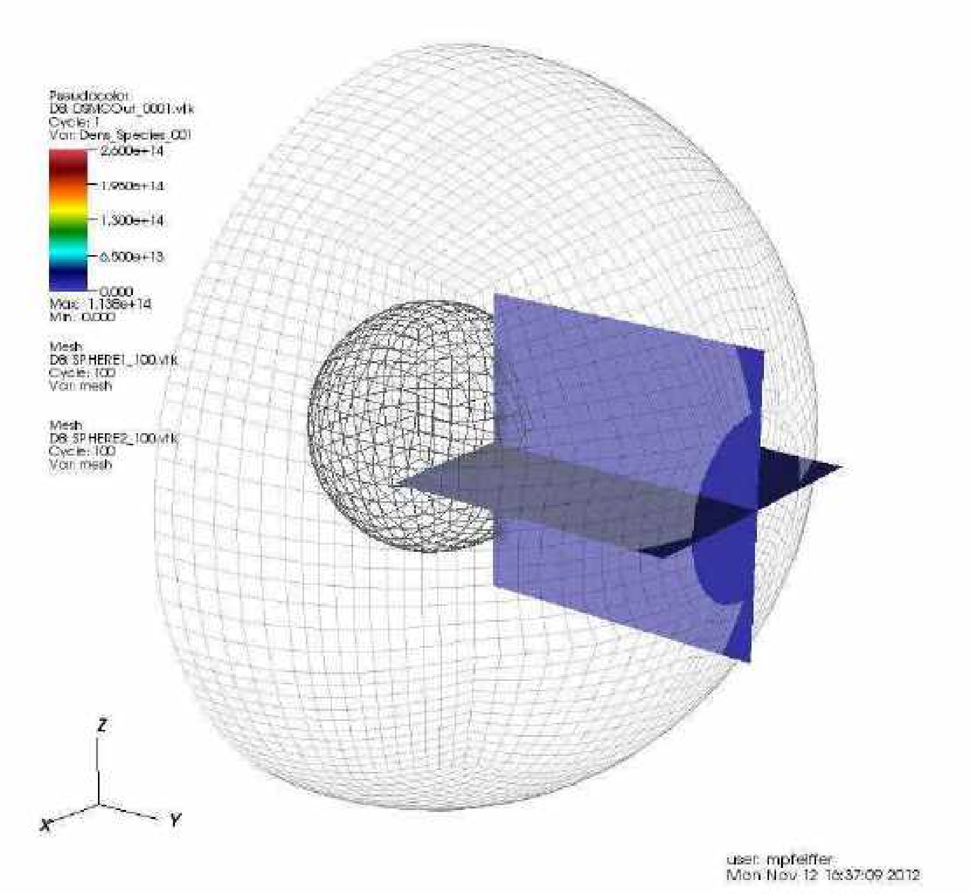
ns into a jet



Vizualization of equipotential lines between the grids with OpenFoam (no particles included) used as Boundary Conditions

Simulation Results of the IEC Jet Extraction

Simulation Results with diffe-rent densities, only PIC



- Study of electron motion with different electron densities

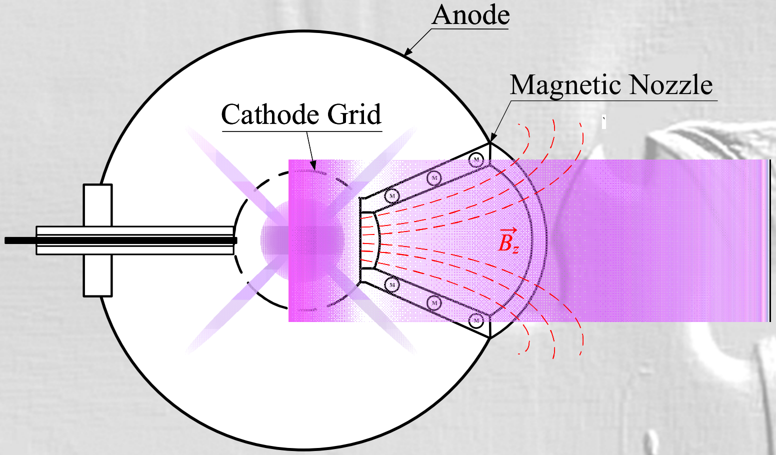
  Simultaneous study of electron and ion motion not possible due to significantly different motion time scales

- Small densities: electrons move exactly on the electric field lines of the IEC grid
- High densities: self fields of the particles increase → self force increases, loss of the jet characteristic

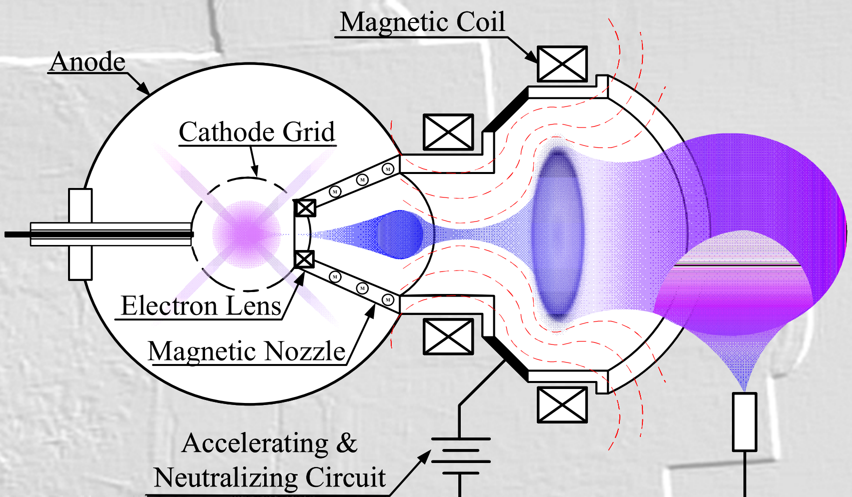
Video:

- Temporal evolvement of $n_{elec} = 4*10^{16} m^{-3}$
- Simulation time $5.5*10^{-9}$s
- Formed electron jet is clearly visible

IEC: Potential Thruster Concepts

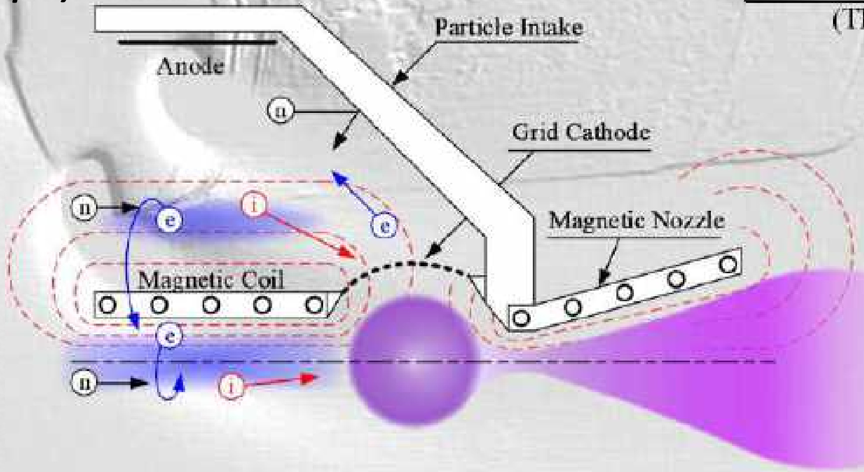


IEC Ion Thruster (concept)

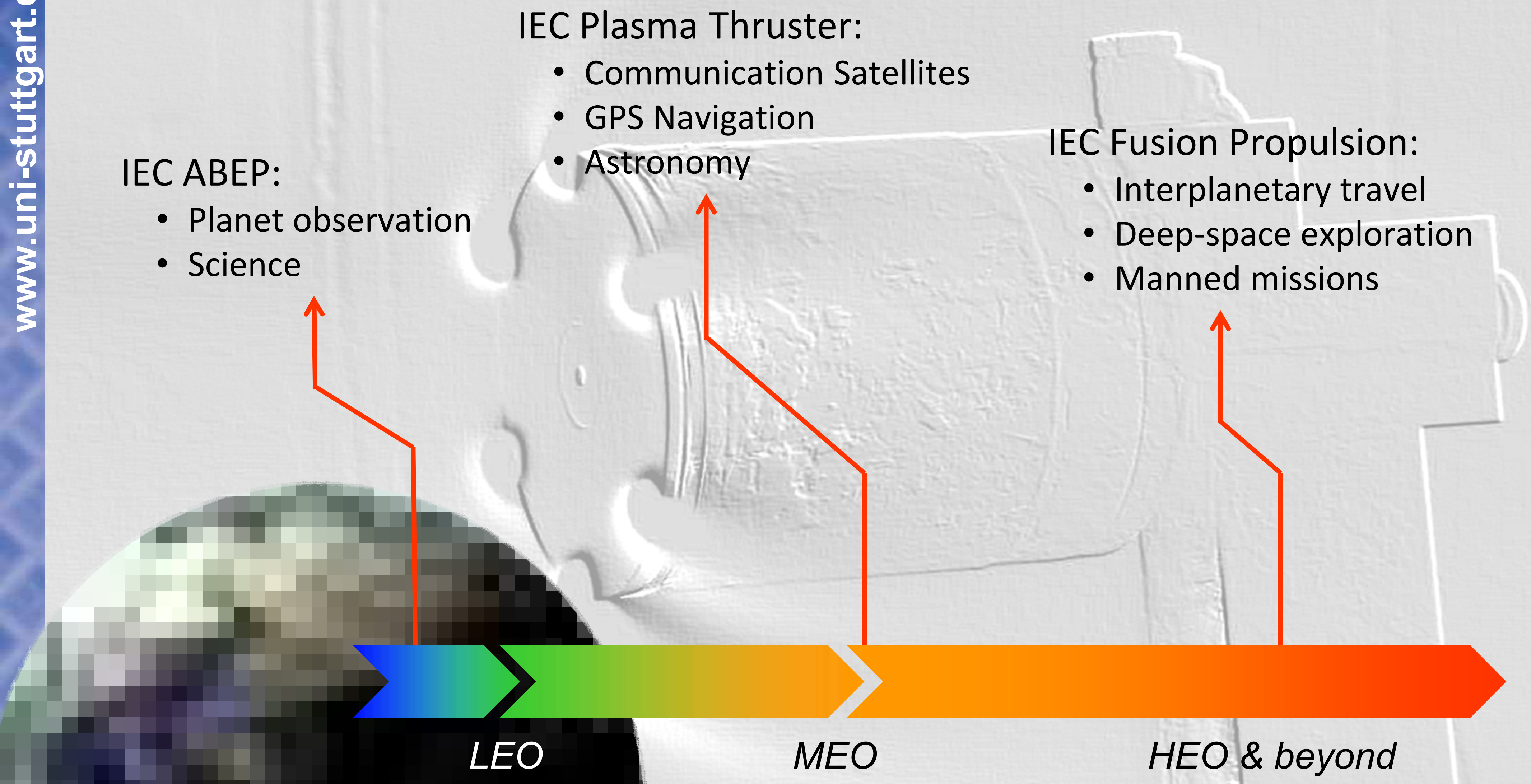


IEC HET concept

IEC ABEP concept



IEC: Application Vision (only space related)



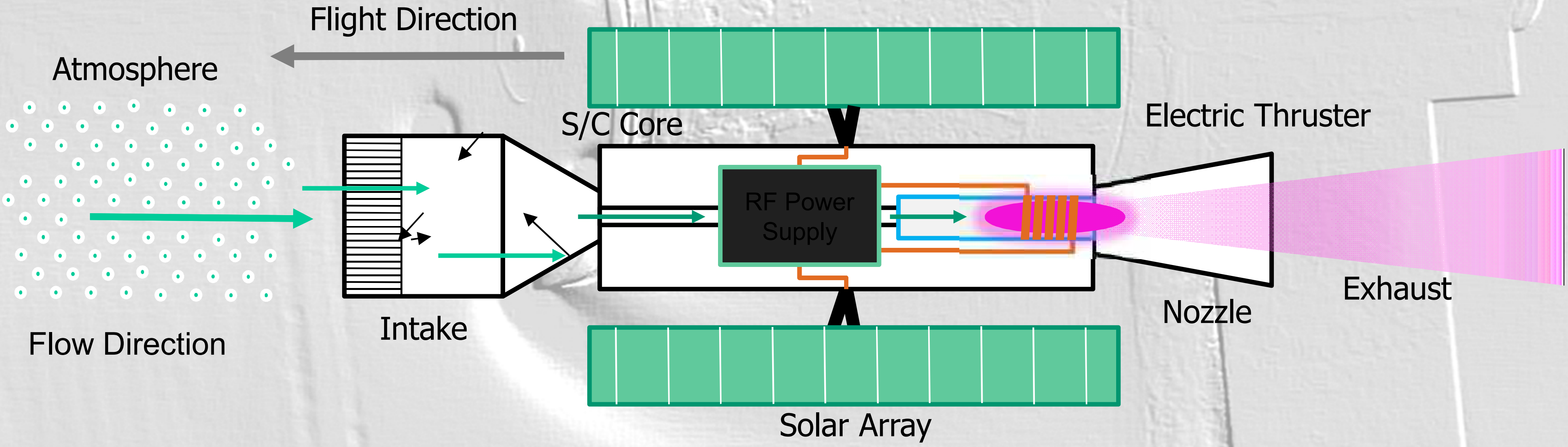
IEC Plasma Thruster:
- Communication Satellites
- GPS Navigation
- Astronomy

IEC ABEP:
- Planet observation
- Science

IEC Fusion Propulsion:
- Interplanetary travel
- Deep-space exploration
- Manned missions

Atmosphere Breathing Electric Propulsion (ABEP)

- Uses residual atmosphere as source for propellant;
- Mass collector;
- Collected mass brought to thruster → Thrust Generation to compensate drag.
- Enables VLEO missions with significant life time.

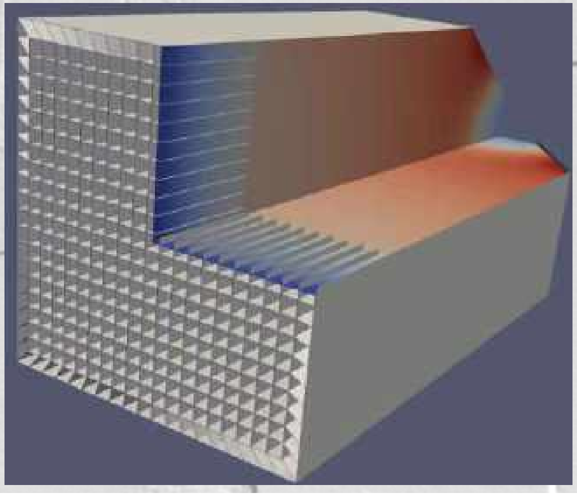


IRS: Development of Intake and Thruster (IPT)

- IPT:
  - Derived from IPG6-S;
  - Passively cooled;
  - Optimized for ABEP-related mass flow;
  - Optimized for power between 0.5 and 5.5 kW.

IPG6-S operating with N_2

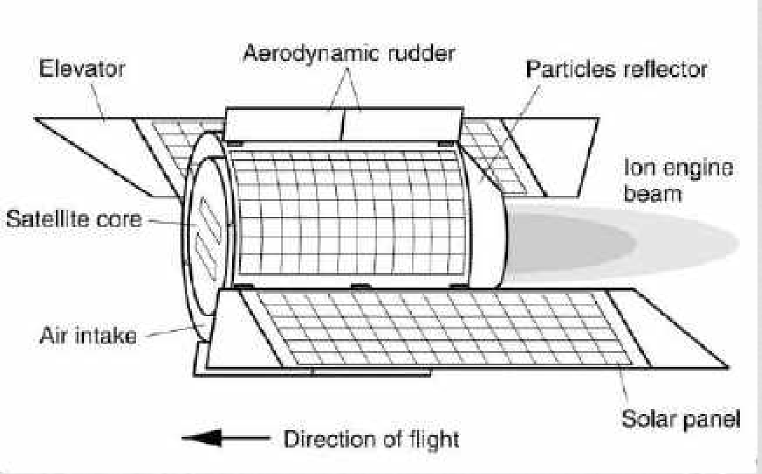
- Collector/ Intake:
  - Based on verified DSMC in-house code;
  - Balance model developed;
  - Principle of molecular trap;
  - Optimized for IPT.



DSMC simulation of adapted intake geometry

Atmosphere-Breathing Electric Propulsion / RAM-EP

- ➢ Extend S/C lifetime in LEO, access VLEO range (100-160 km);
- ➢ Characterization of low orbit planet´s properties (atmosphere, magnetic and gravitational field (e.g. GOCE), etc.).



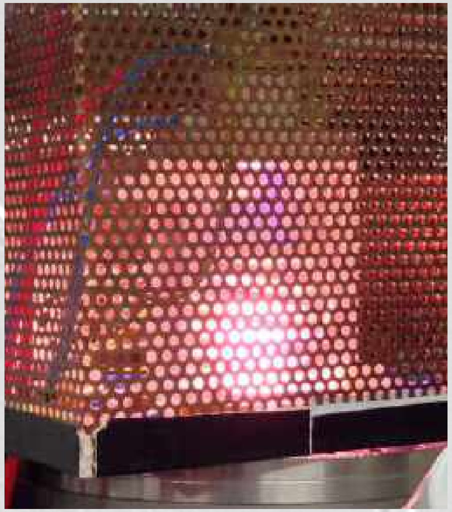
- ➢ Propulsion System → Compensates Drag → Propellant: atmospheric gas
- ➢ Intake collects gas particles → Feeds Thruster

- ❖ Altitude Range: > 120 km (heating effects) to 250 km (competitive against conventional EP);
- ❖ Dominant elements collected in VLEO and LEO: N_2 and O;
- ❖ Erosion problems due to O might arise.

Atmosphere-Breathing Electric Propulsion / RAM-EP

IPG6-S - Inductively heated Plasma Generator:

- Electrodeless and gridless → no limited lifetime due to erosion;
- Size (d=40 mm discharge channel) and power levels (max input in op. 3.5 kW) suitable to small S/C.



- Operated with Air (N_2) and O_2 → $\dot{m}=0.245\text{-}120$ mg/s → collection eff. $\eta_c=\dot{m}_{thr}/\dot{m}_{inlet}$ → 40% experimentally achieved

Plasma Enthalpy measured through Calorimeter → Thrust estimated considering full conversion of Thermal into Kinetic energy

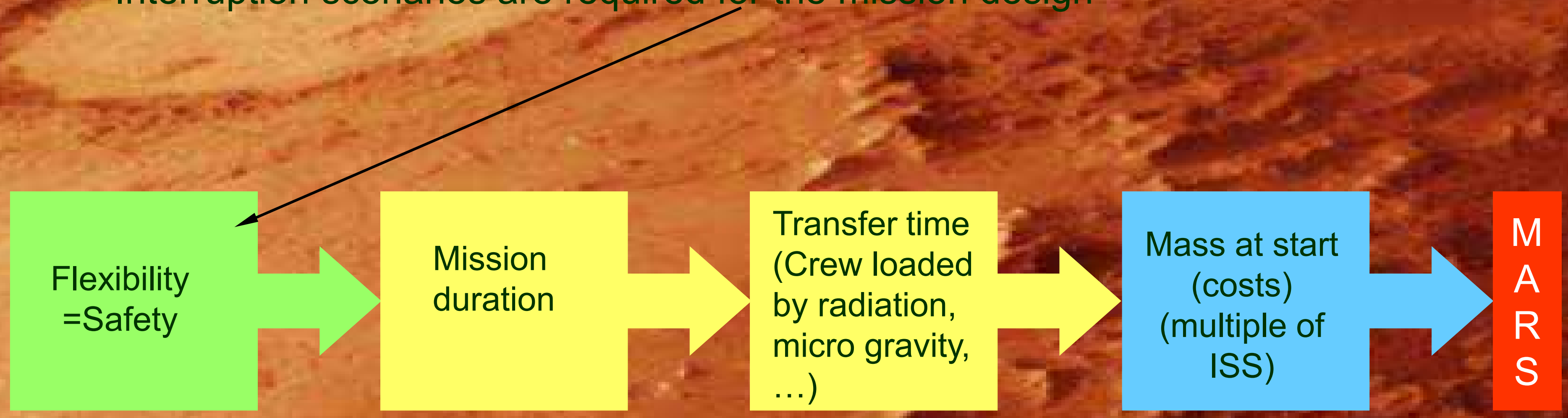
Drag in Free Molecular Flow for $A_f=1\ m^2$
→T/D about 1 over all the altitude range
→η_c from 0.35 to 0.9 and
→V=0.55, 0.85 and 1.00 kV

✧ DSMC for different designs of Air Intake being currently developed.

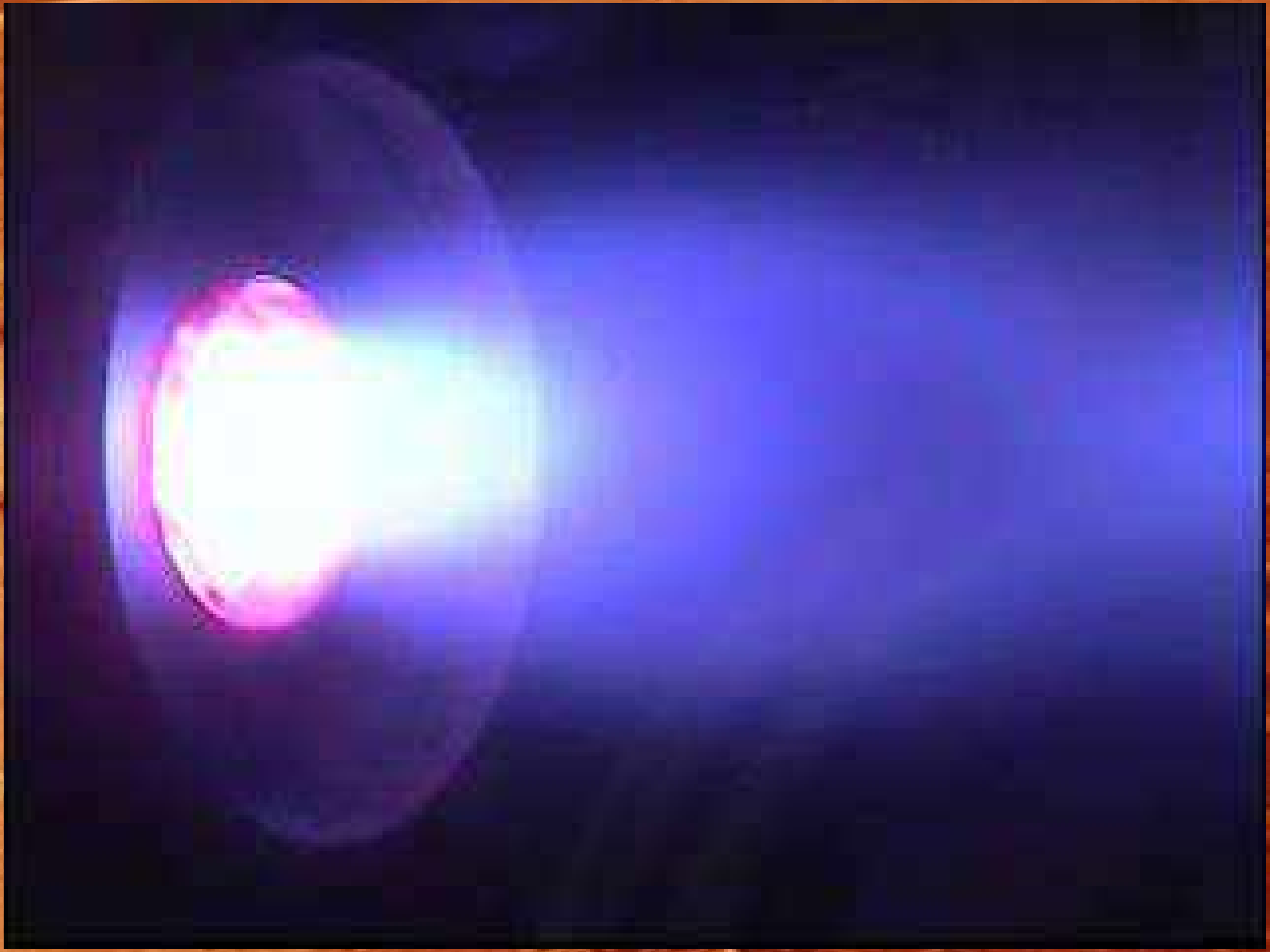
Manned Mission to Mars – Scenario

20. Juli 1969: Neil Armstrong, Edwin Aldrin und Michael Collins on the moon

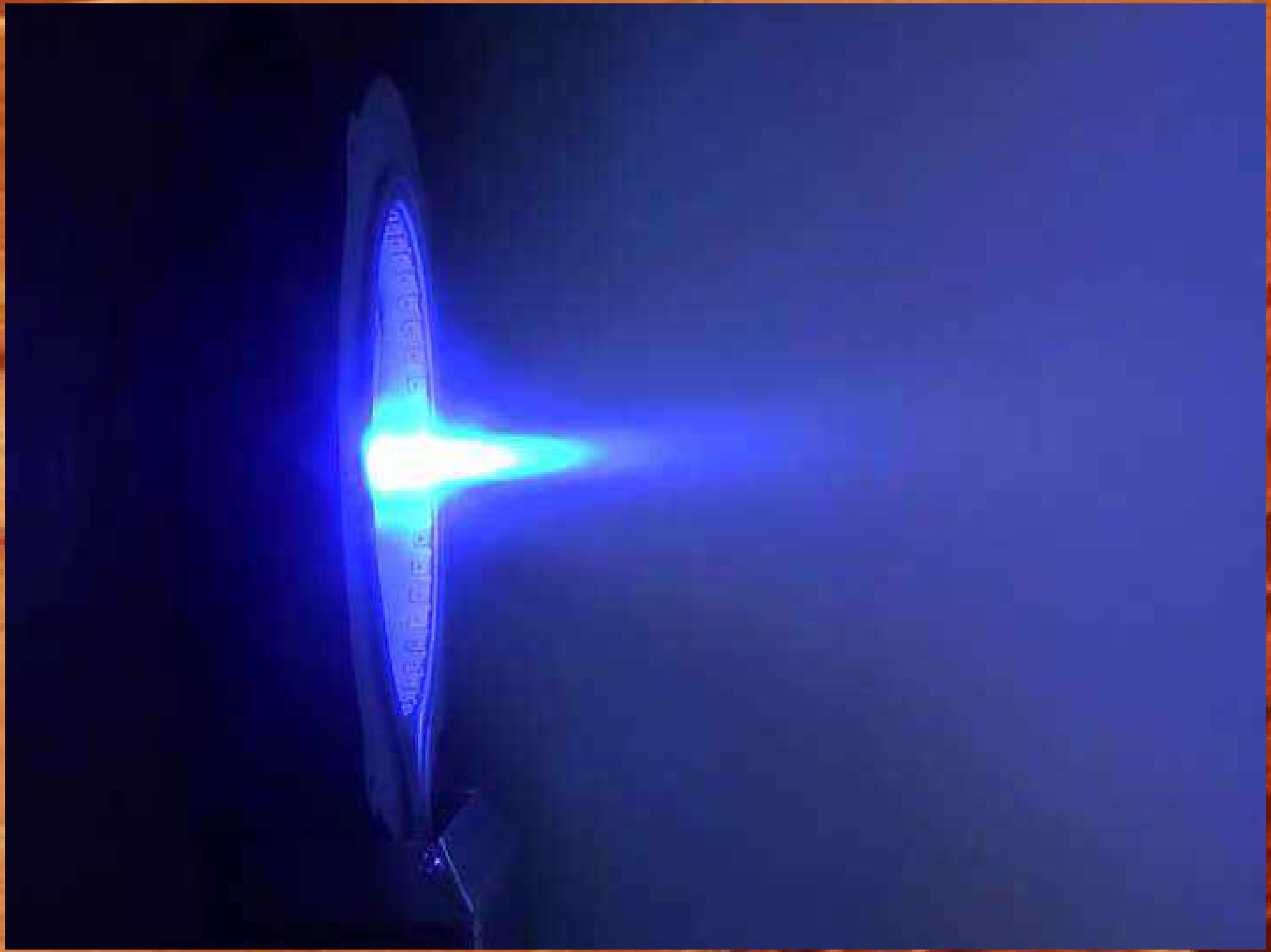
- Manned mission to Mars realistic
- main aspect: to return astronauts safely to Earth
- Kritical Phases of mission have to be assessed in detail
- Interruption scenarios are required for the mission design



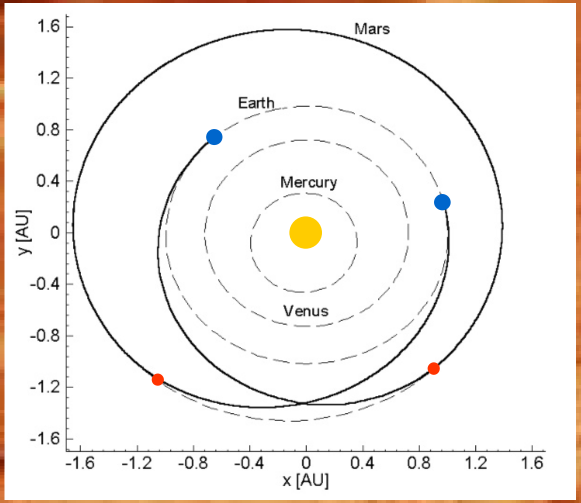
Manned Mission to Mars – TIHTUS

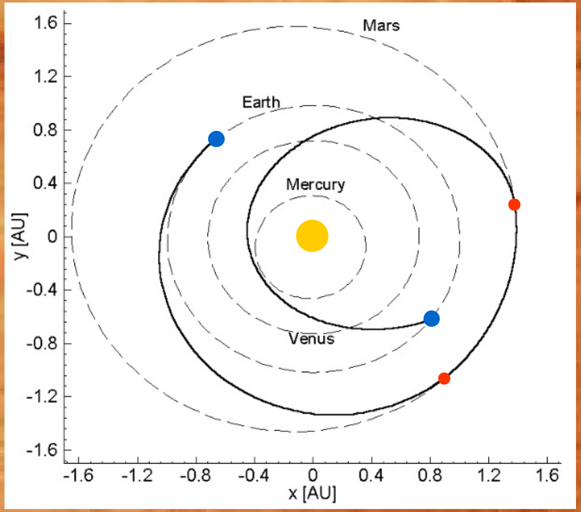


Manned Mission to Mars – AF MPD SX3



Manned Mission to Mars – Scenario





Start windows every 26 months

Constellation of minimal energy offers useful „long-stay“-option. Durationat/on Mars: 1,5 years and short transfer times
Total duration: 3 years

Shortest total mission duration: 1,5-2 years, period at Mars: 30-60 days,
higher amount of energy needed

Referenz Missions:

NASA DRM (Design Reference Mission, USA): surface base

ISTC-Konzept (Intl. Science and Technology Center, RUS): orbital base

ESA DC (Design-Case, EUR)

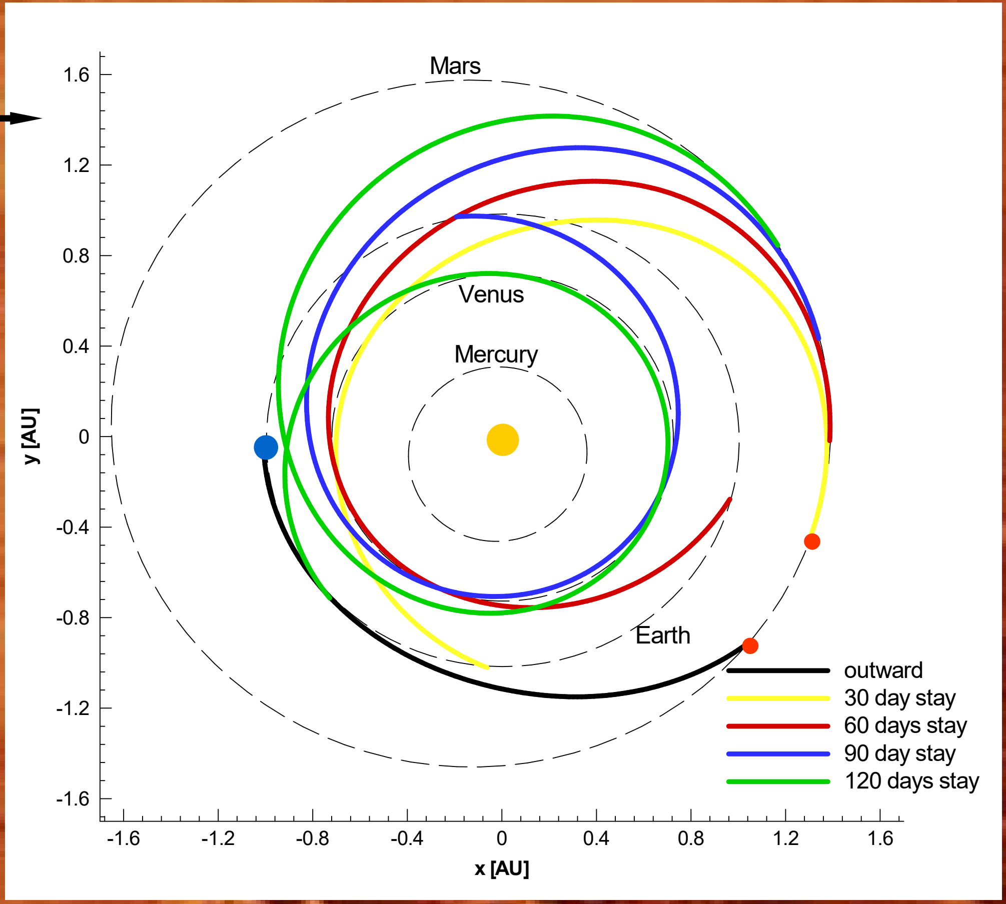
Mars Direct

Manned Mission to Mars – Scenario

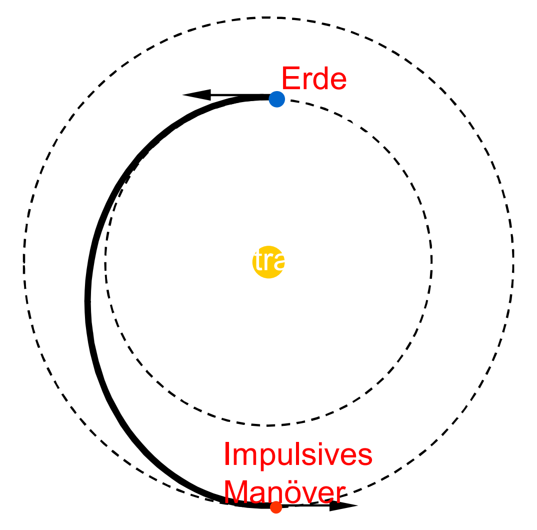
Propulsion sytems:

impulsive thruster

continuous thruster



→ Shorter stays possible with moderate masses and moderate flght times



30-day

100 N
3000 s
η_{th}=44

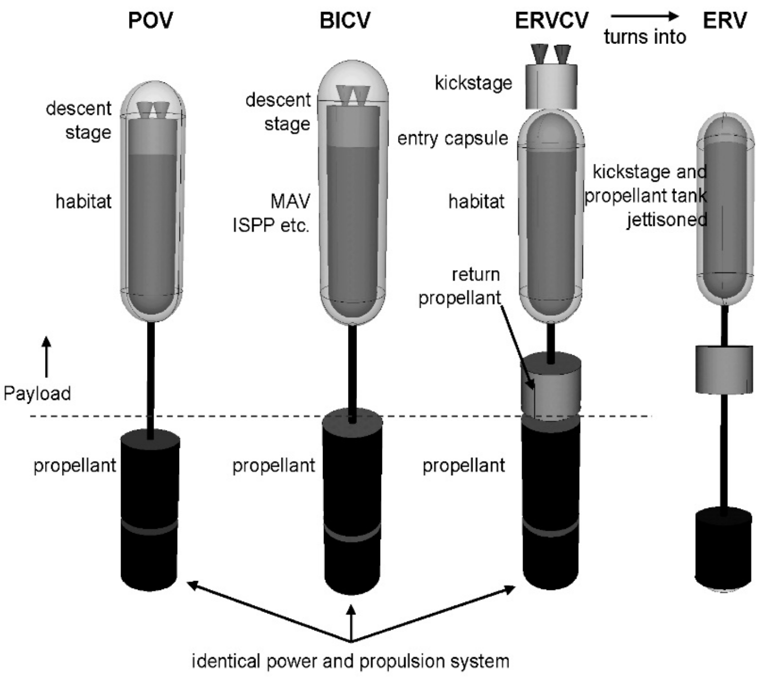
Techn
- Eff
vel
achieved
- Thrust: Clustering

Manned Mission to Mars – Scenario

Splitted Mission:
6 crew members

3 manned missions
in intervals of 2 years
each.

Use of similar designs referring to scenario, spacecraft and sub systems

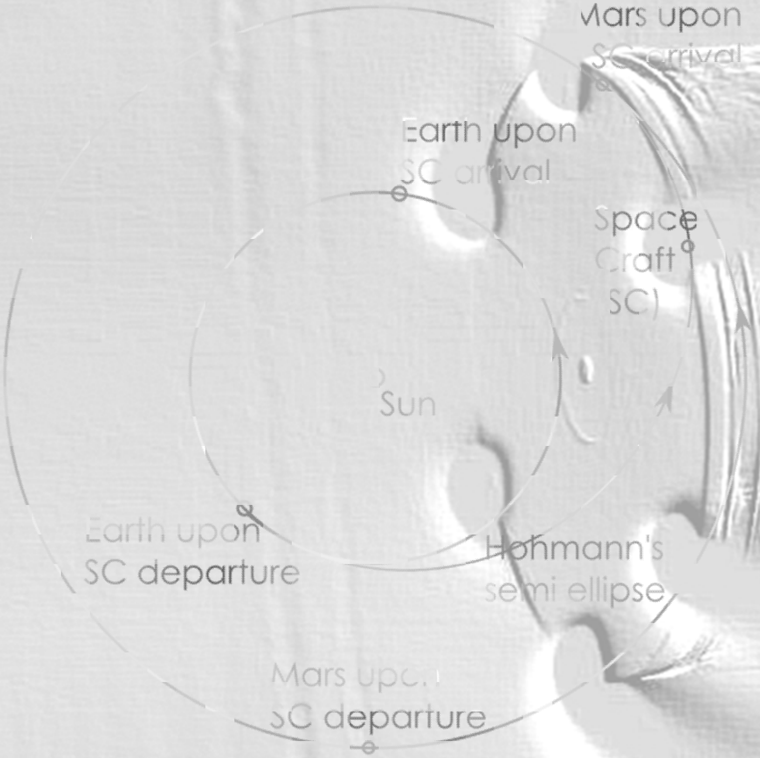


1st crew leaves in 2033

From orbit at Earth:
400 km circular with 23° inclination
3 spacecraft
1 returning spacecraft

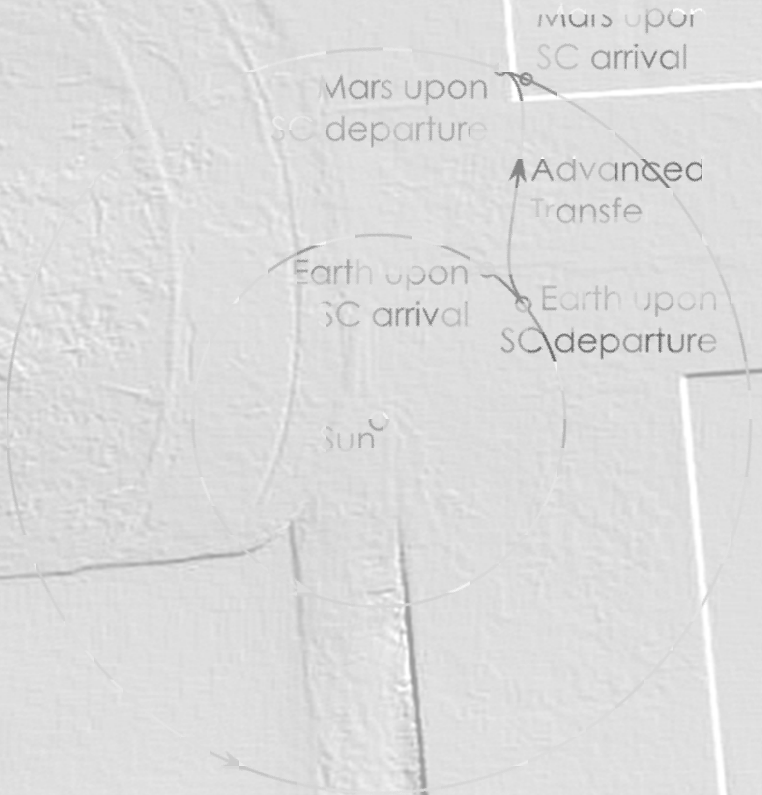
- Piloted outward vehicle, POV, lands on surface
- Base-infrastructure cargo vehicle, BICV, science P/L, habitat, propellant for return, low energy trajectory, long transfer
- Earth return vehicle transporter, ERVT, transports fully tanked return vehicle to elliptic park orbit at Mars
- Earth return vehicle, ERV

Motivation for High Power Thruster for interplanetary Travel



What we are currently doing…

…and what we ought to do.



(Further) Consideration of the propulsion problem

How can we overcome this arduousness in propulsion?

A transfer is characterized by a velocity increment *Δv*

$$\Delta v = \int a_c dt \qquad \Delta v = c_e \log\left(\frac{m_0}{{}_1}\right)$$

Rel

ion:

mass

Increase of α_{lim} or assess the currently available theoretical base of physics

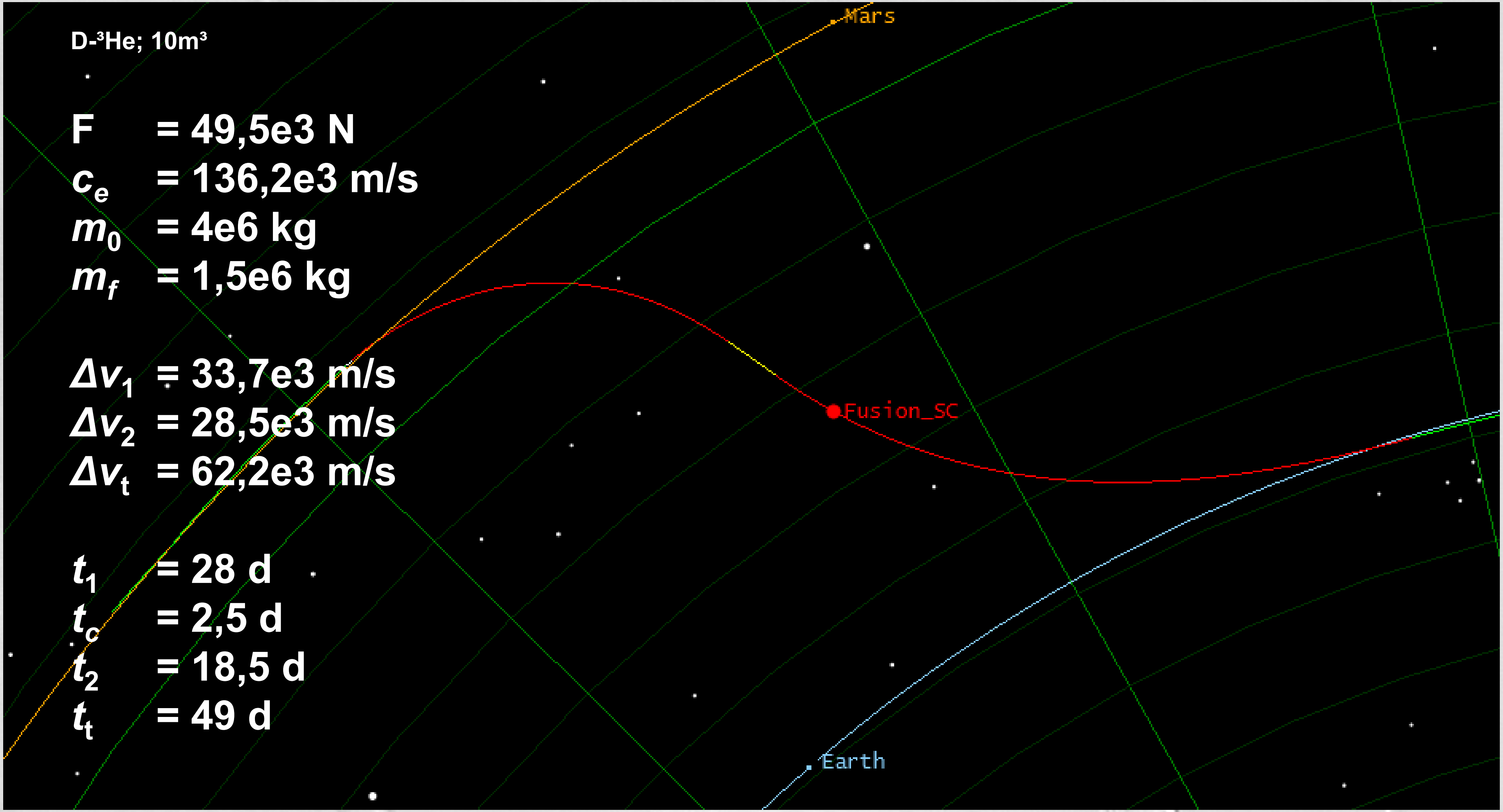
The link of a_c and c_e is the mass specific thrust power.

$$\frac{1}{2} a_c c_e \leq \alpha_{lim}$$

Mass specific thrust power often limited (concerning power supply). Often desire of enhancement.

→ *advanced propulsion with increased α_{lim}, enhancement of EP thruster power*

Fusions Propulsion 1: Mars



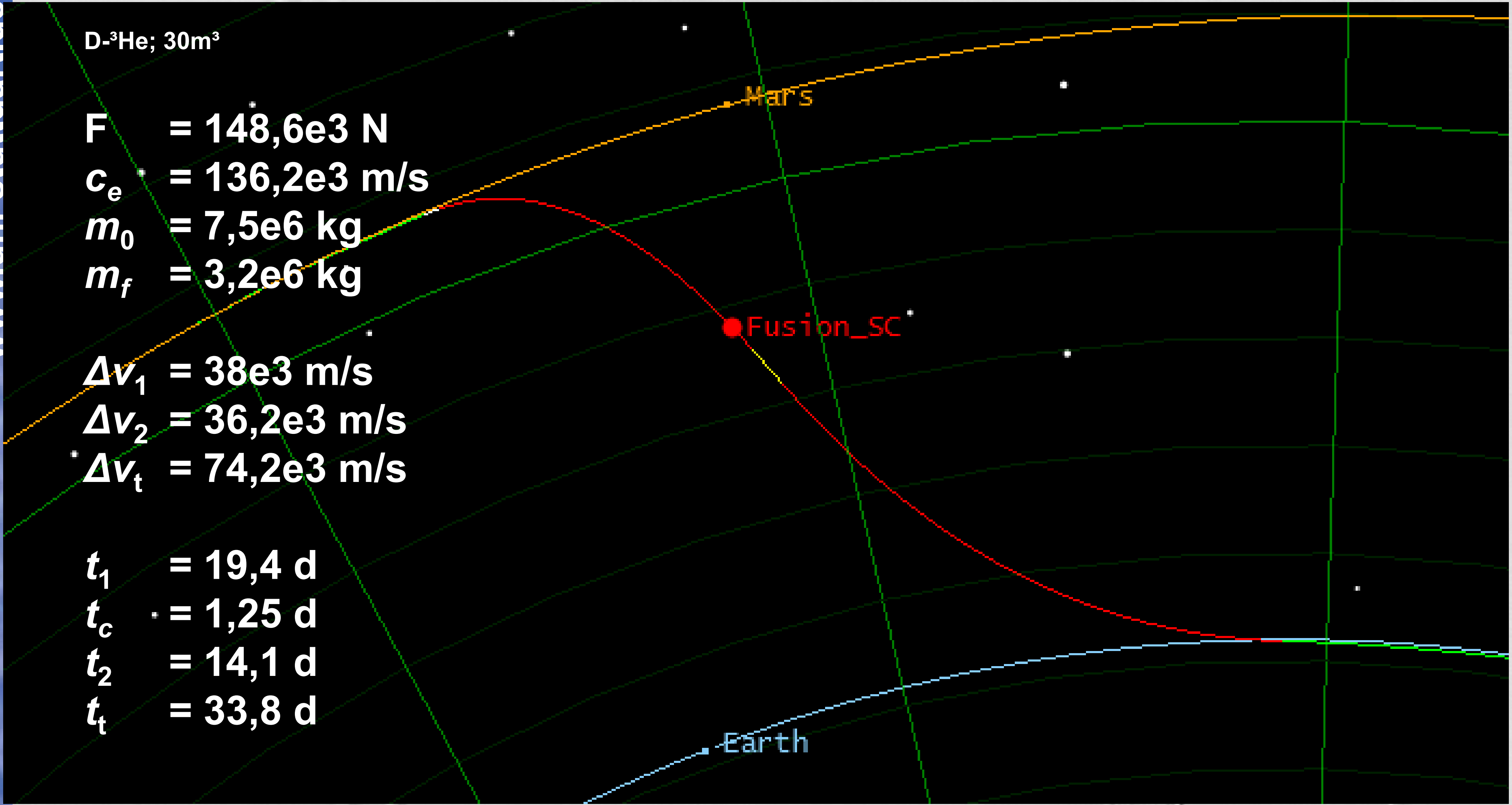
D-³He; 10m³

$F = 49{,}5e3\ N$
$c_e = 136{,}2e3\ m/s$
$m_0 = 4e6\ kg$
$m_f = 1{,}5e6\ kg$

$\Delta v_1 = 33{,}7e3\ m/s$
$\Delta v_2 = 28{,}5e3\ m/s$
$\Delta v_t = 62{,}2e3\ m/s$

$t_1 = 28\ d$
$t_c = 2{,}5\ d$
$t_2 = 18{,}5\ d$
$t_t = 49\ d$

Fusion Propulsion 2: Mars



D-^{3}He; 30m^3

$F = 148{,}6\text{e}3$ N
$c_e = 136{,}2\text{e}3$ m/s
$m_0 = 7{,}5\text{e}6$ kg
$m_f = 3{,}2\text{e}6$ kg

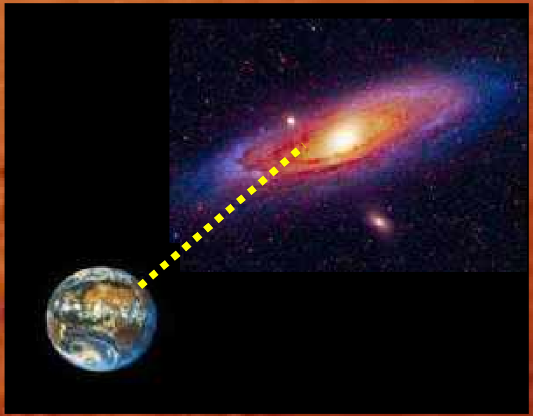
$\Delta v_1 = 38\text{e}3$ m/s
$\Delta v_2 = 36{,}2\text{e}3$ m/s
$\Delta v_t = 74{,}2\text{e}3$ m/s

$t_1 = 19{,}4$ d
$t_c = 1{,}25$ d
$t_2 = 14{,}1$ d
$t_t = 33{,}8$ d

WARP - Propulsion?

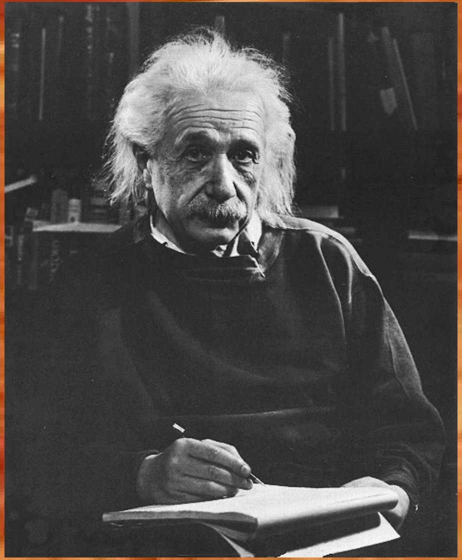
Andromeda is at a distance of
2,7 Millionen light years.

Light travels along
9 460 000 000 000 000 m during a year – one light year!



From / To	Distance	Duration with current propulsion	Light
Stuttgart-Rust	200km	2,5 hours (car)	1 ms
Earth-Mars	75 million km	> 6 months	4 Min 10 s
Erde - Proxima Centauri	4,24 light years	91500 years (50000 km/h)	4,24 years
Erde- Andromeda	2,7 million light years	58 300 000 000 years	2,7 million years

(How) does WARP-propulsion work?

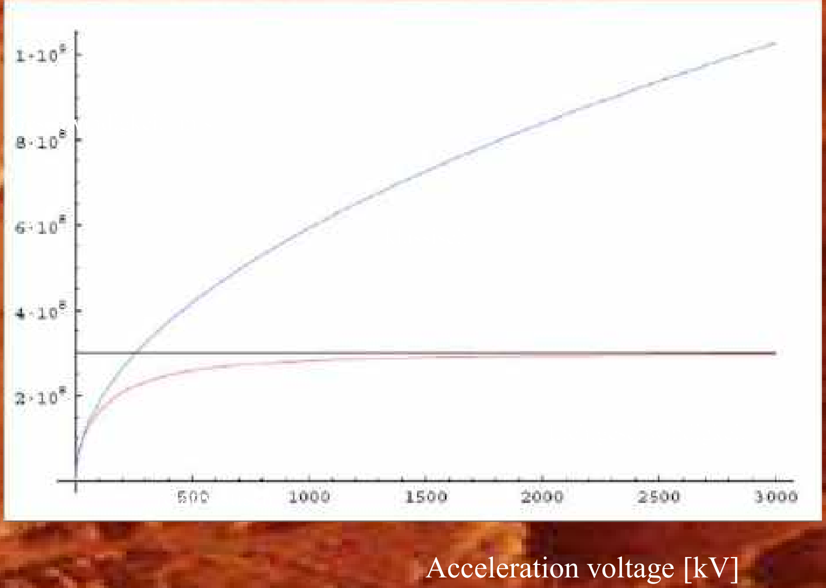


Special Relativity Theory

Significant impact to physics

New (relativistisc) correlations between
momentum ↔ velocity and energy ↔ momentum introduced.

Key result: Materia particles cannot be accelerated to speed of light (and above)



Example:
Acceleration of electrons in electric field

$$E = mc^2$$

$$m = \frac{m_0}{\sqrt{1 - v^2/c^2}}$$

(How) does WARP-propulsion work?

Was does General Relativity Theory tell us?

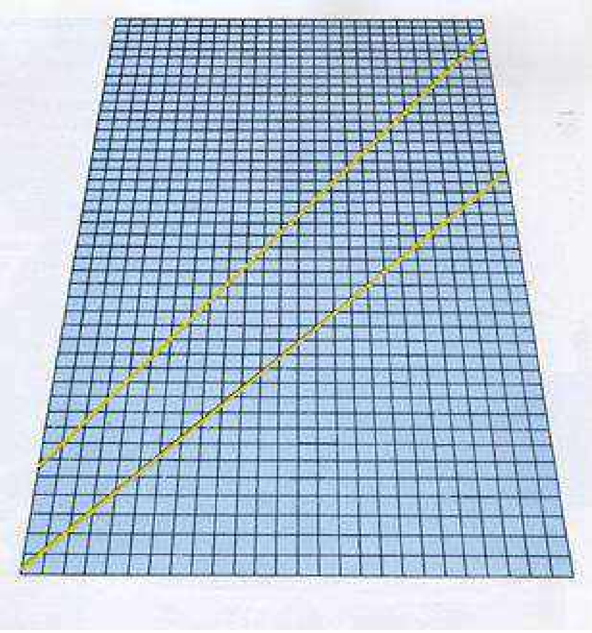
More detailed statement: Nothing can be faster than light on a local reference

What does LOCAL mean?

Introduction of a 4-dimensional space time
(3 space coordinates, 1 time coordinate)

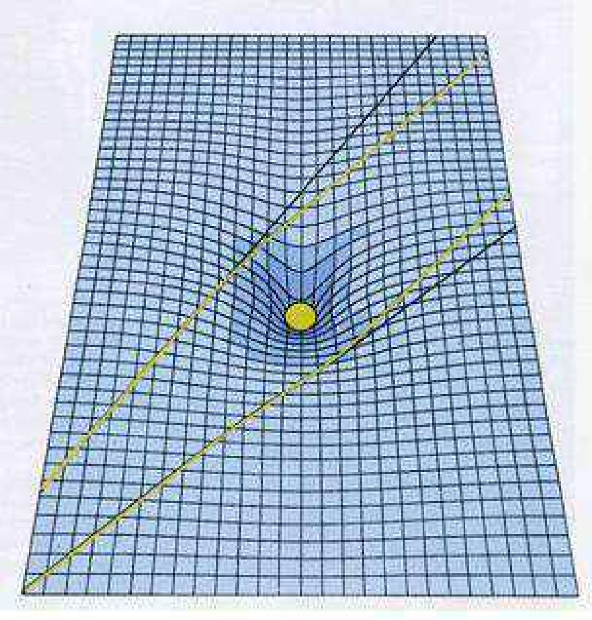
Metric describes mathematically the lengths in space time (coordinate system)

Space is flat w/o mass (present) (Euclidean)
→ Euclidean metric sufficiently describes space time
(global coordinate system necessary)



Note: For better understanding here neither 4D nor 3D, but 2D spacial

(How) does WARP-propulsion work?



Warping of space time with mass present
→ Metric now depending on local coordinates.
Only in near zone (local zone) approximately Euclidean (local coordinate system possible)

Special relativity theory only valid in inertial system
= flat space times with Euclidean metric

Conclusion:
Two objects at places that cannot be reperesented by a single and common local coordinate system can move relative to each other with speeds greater than speed of light

No contradiction to the Special Relativity Theory

Note: For better understanding here neither 4D nor 3D, but 2D spacial

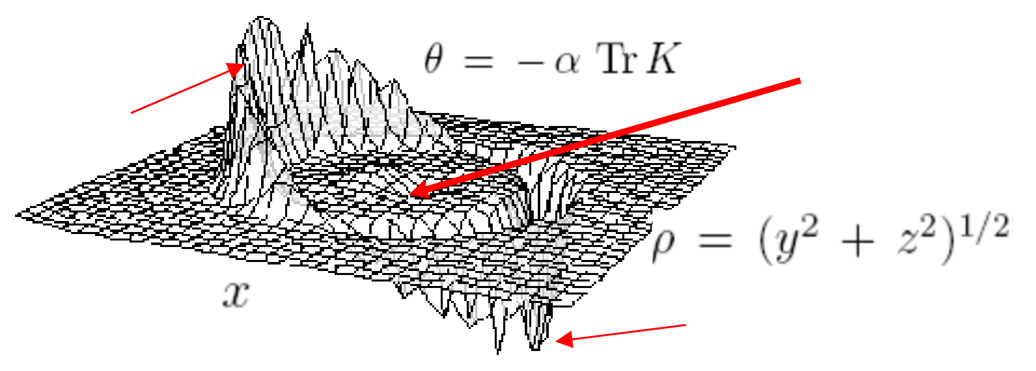
(How) does WARP-propulsion work?

Construction (Generation) of space time distortion to achive faster than light movement ?

Theoretical assessment by M. Alcubierre (Class. Quantum Grav. 11-5, L73-L77 (1994)
and Ch. van den Broeck (Class. Quantum Grav. 16 3973-3979 (2005)

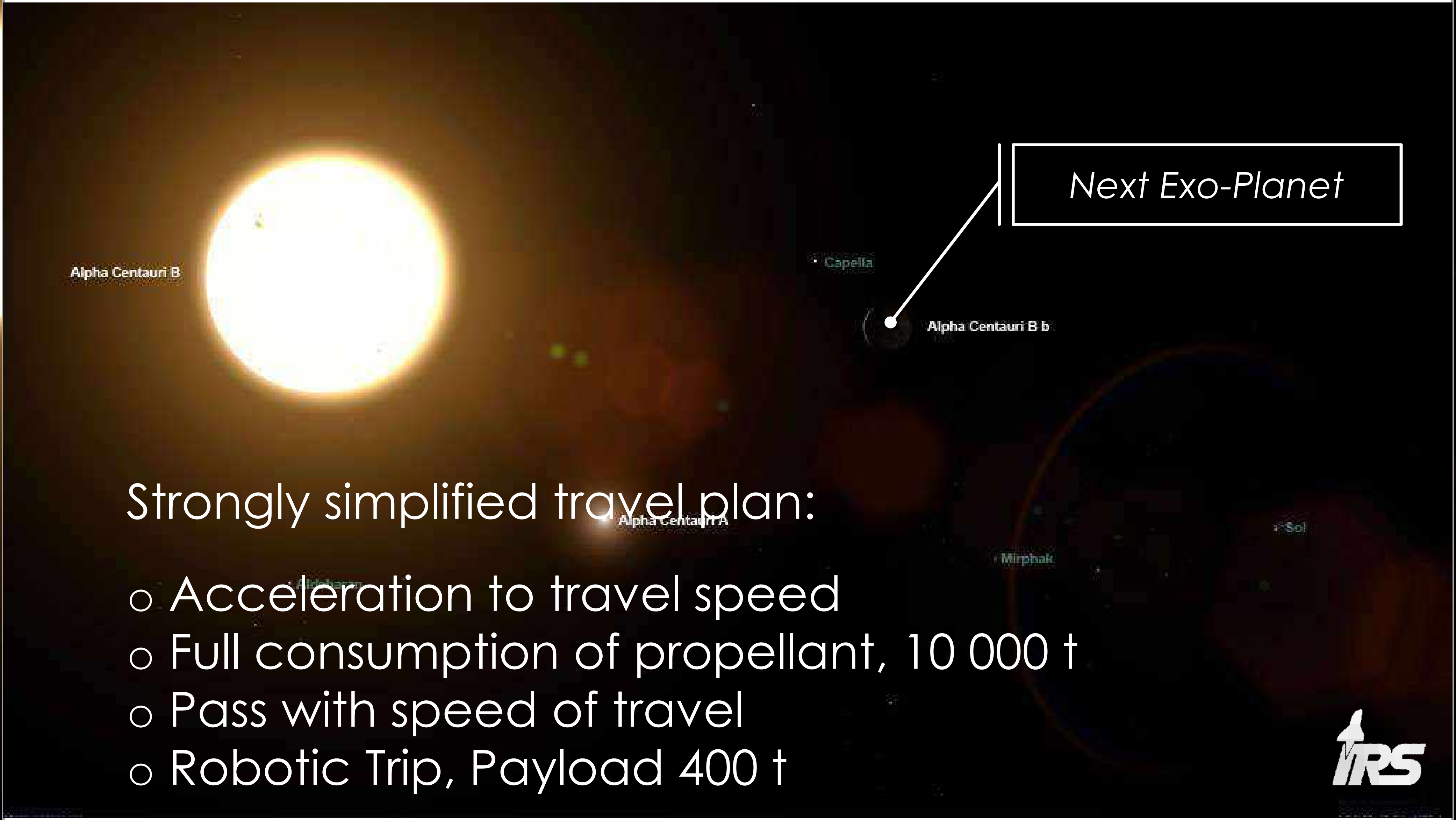
Modification of space time

- Generation of compression of the space time in front of spacecraft
- Generation of expansion of the space time behind spacecraft



But:
Generation of each of the kown metrics achieving this requires masses with negative energy density → doubtful

Centauri Star System



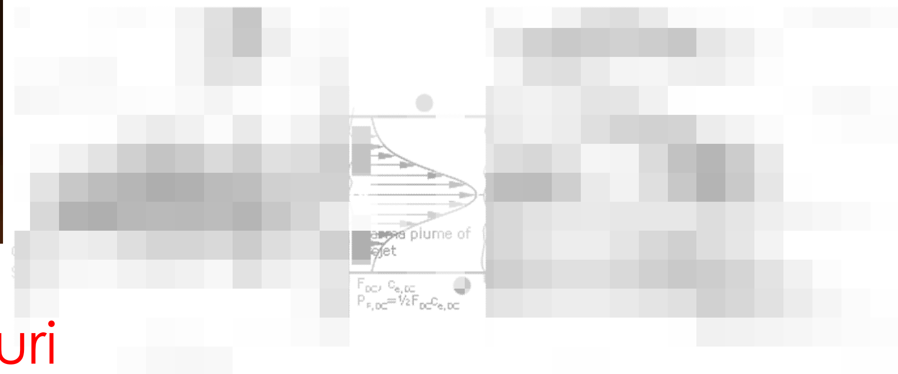
Next Exo-Planet

Strongly simplified travel plan:

- Acceleration to travel speed
- Full consumption of propellant, 10 000 t
- Pass with speed of travel
- Robotic Trip, Payload 400 t

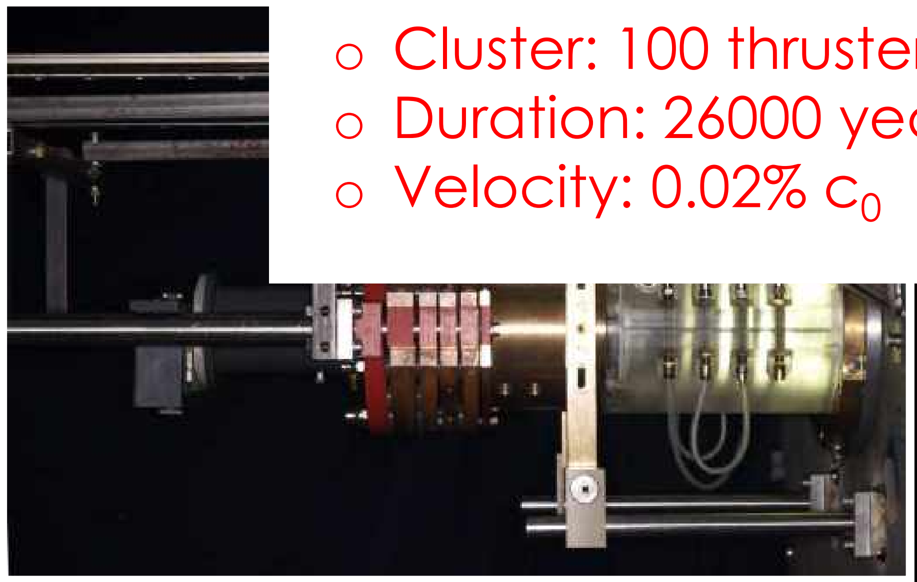
Quelle: Adrian Mann, BISBOS

Hybrid Thruster TIHTUS



To Proxima Centauri

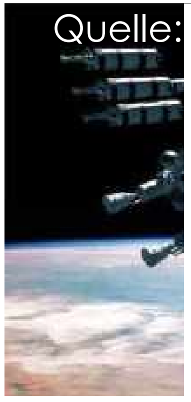
- Cluster: 100 thrusters
- Duration: 26000 years
- Velocity: 0.02% c_0



Source: IRS

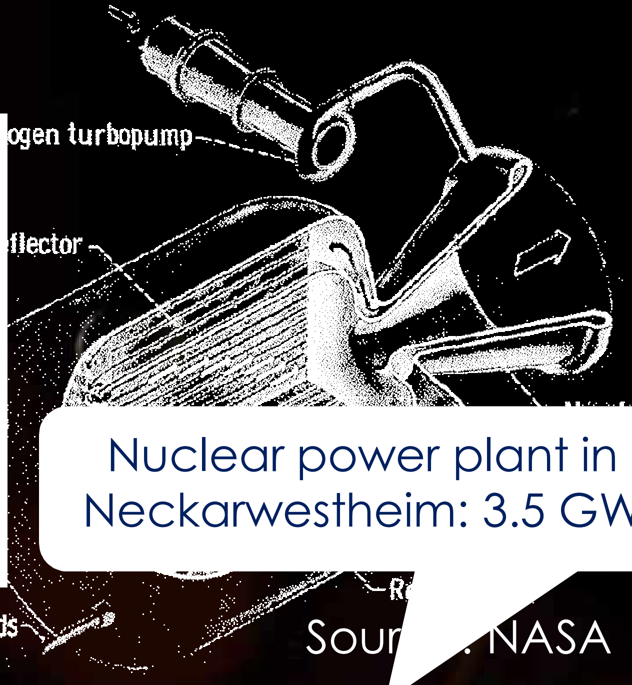
Power	50 kW
Propellant	Hydrogen
Thrust	3.5 N
xhaust velocity	15000 m/s
Remarks	Diverse combinations feasible
Development status	Laboratory model

Fission Thruster NERVA


Quelle:

To Proxima Centauri

- Cluster: 1thruster
- Duration: 48000 years
- Velocity: 0.01% c_0



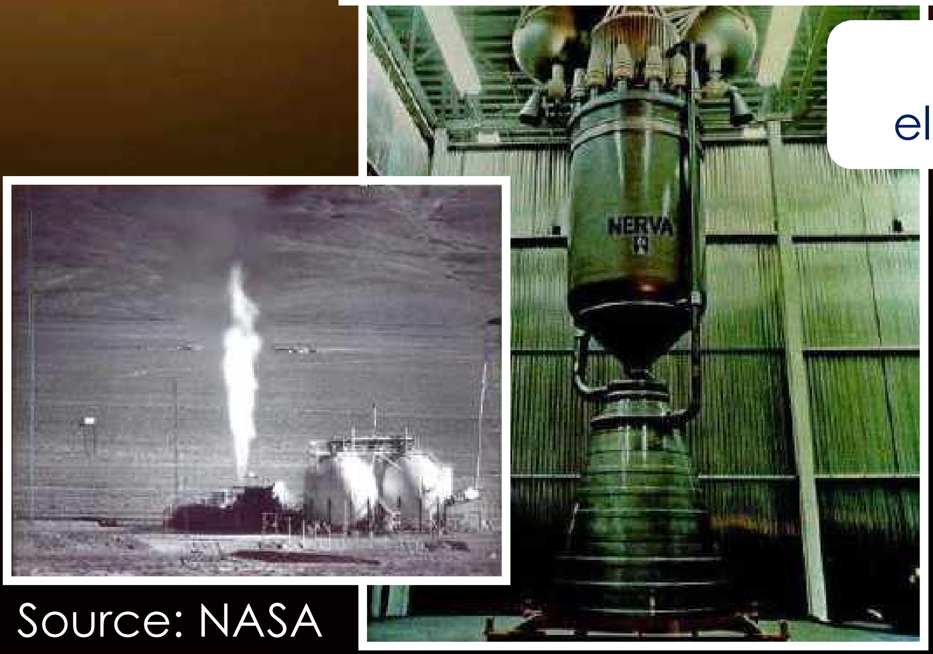
Sour : NASA

Nuclear power plant in Neckarwestheim: 3.5 GW

29 big elephants

28800 km/h

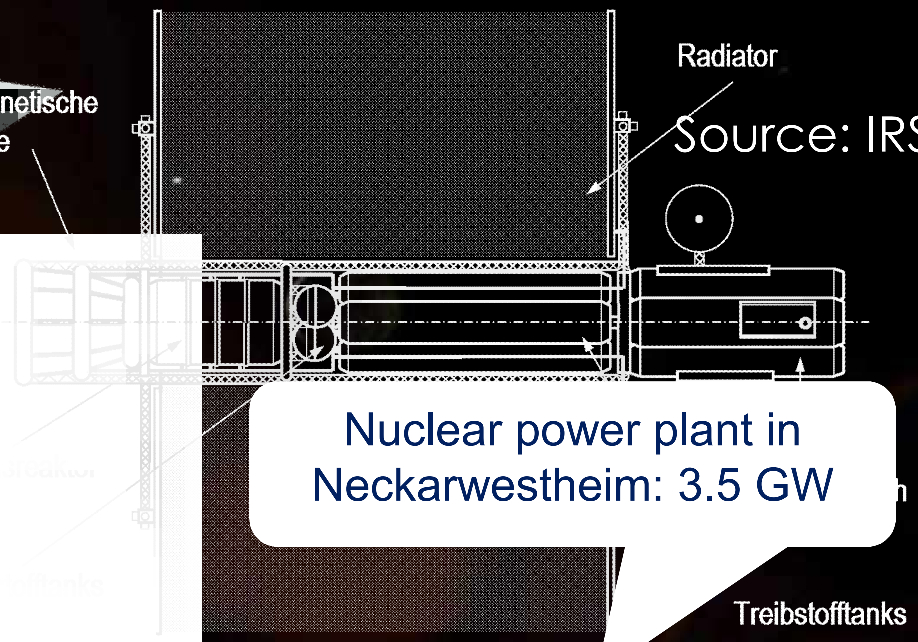
Power	Up to 4.1 GW
Propellant	Hydr
Thrust	1135 kN
Exhaust velocity	8000 m/s
Remarks	Nuclear thermal
Development status	Prototype; Development + Tests



Source: NASA

Fusion Thruster Concept at IRS


Source: NASA



Source: IRS

To Proxima Centauri

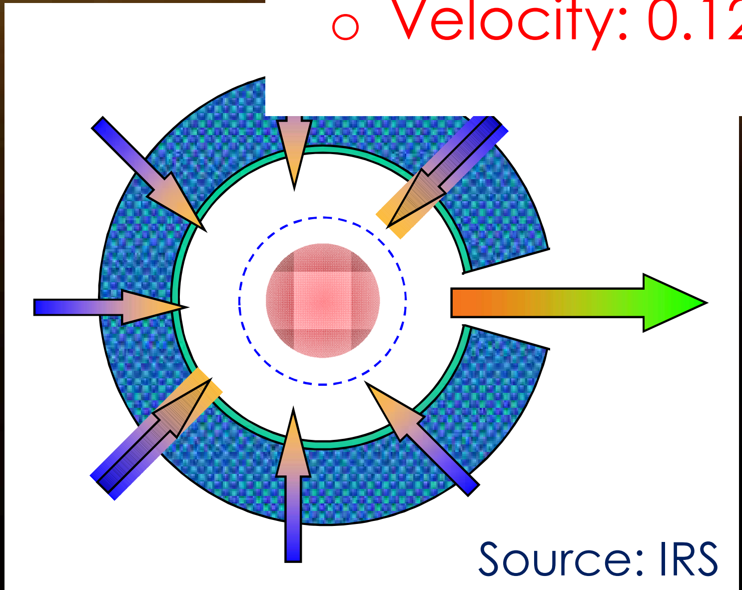
- Cluster: 1 thruster
- Duration: 3800 years
- Velocity: 0.12% c_0

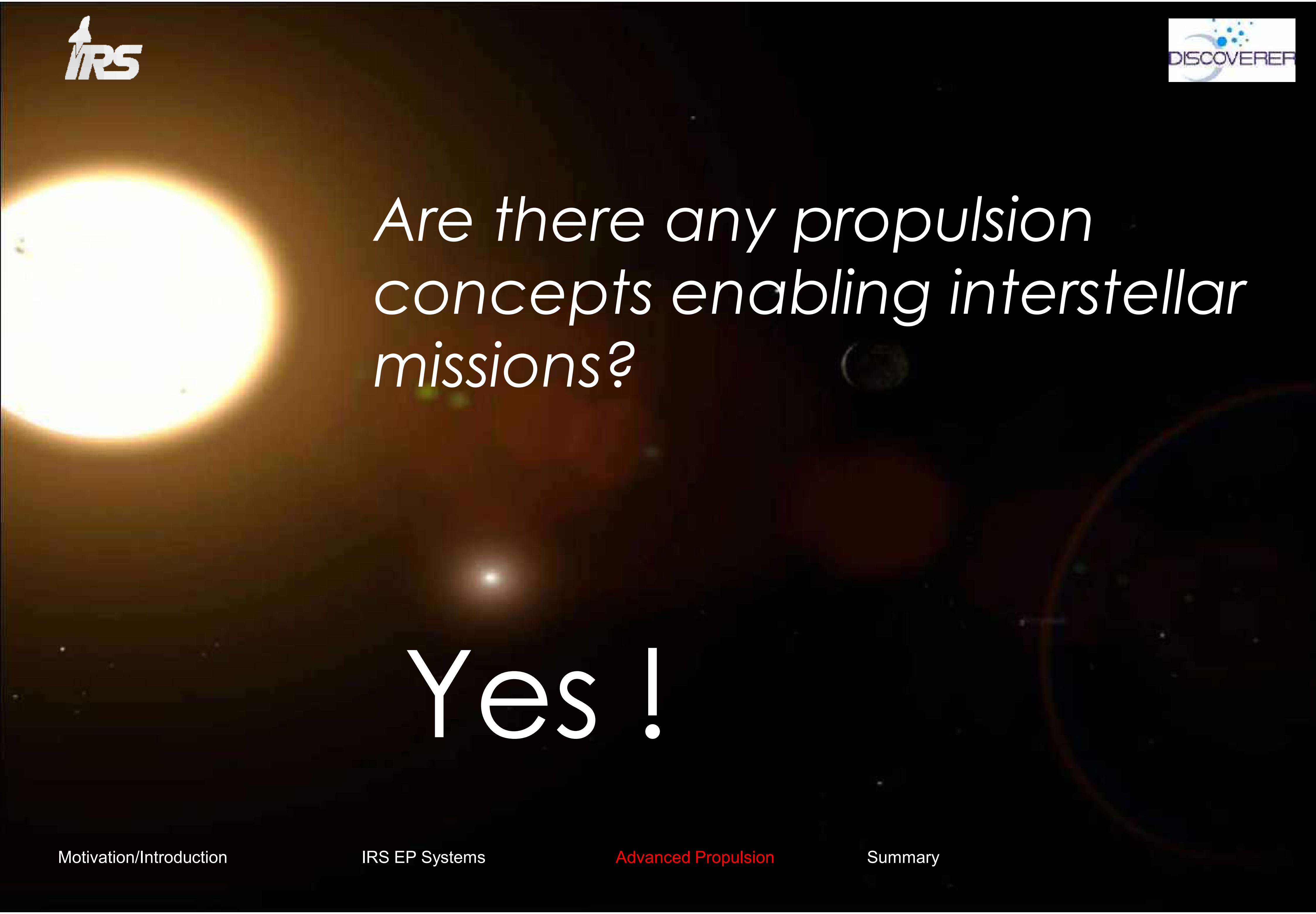
Nuclear power plant in Neckarwestheim: 3.5 GW

1 big elephant

378000 km/h

Power	from 2 GW
Propellant	Hydrogen
Thrust	from 3[illegible] kN
Exhaust velocity	105000 m/s
remarks	Nuclear thermal
Development status	converging concept studies

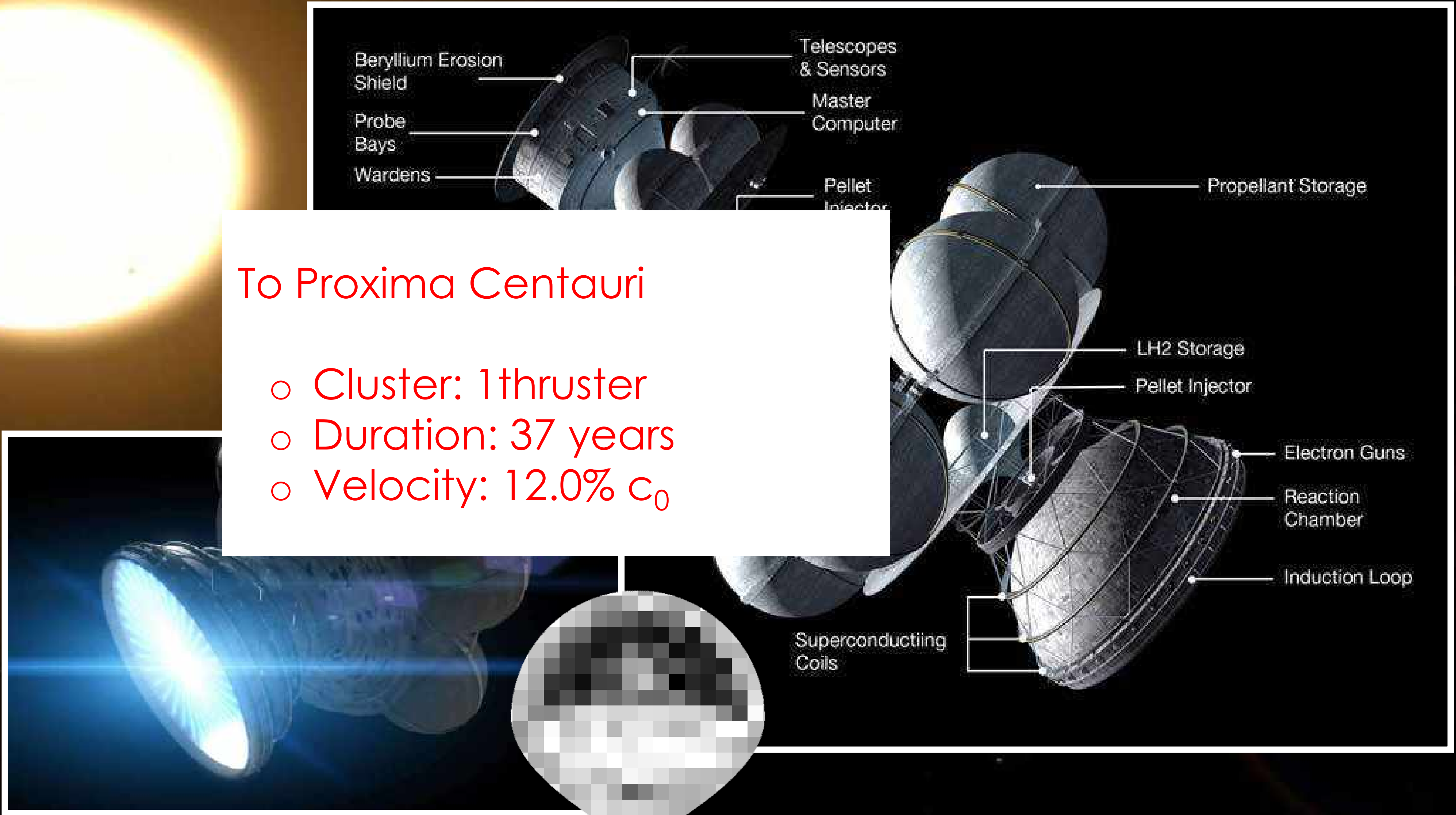

Source: IRS



Are there any propulsion concepts enabling interstellar missions?

Yes !

Project DAEDALUS (BIS) (Inertial Fusion)

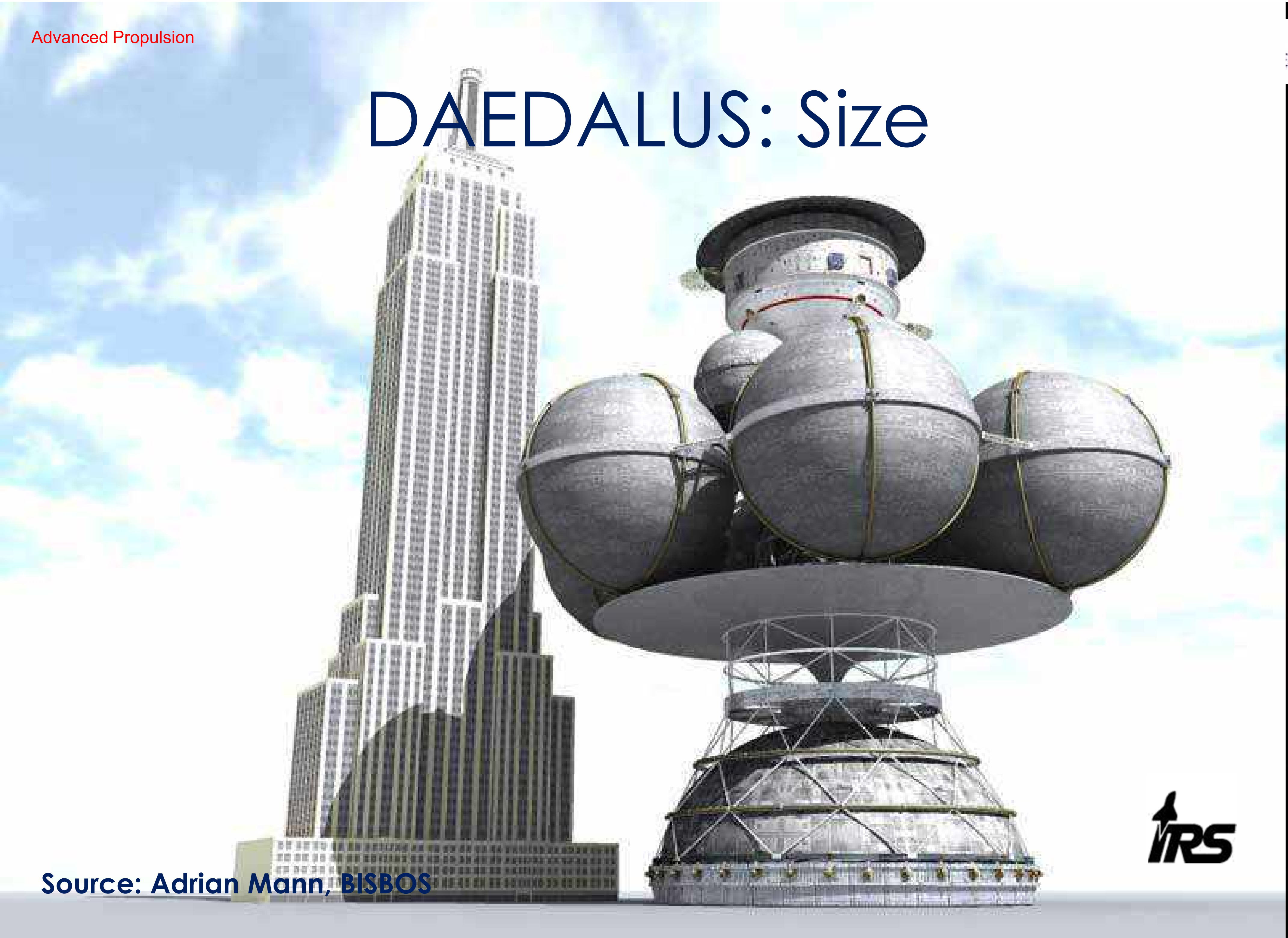


To Proxima Centauri

- Cluster: 1thruster
- Duration: 37 years
- Velocity: 12.0% c_0

Source: Adrian Mann, BISBOS

DAEDALUS: Size



Source: Adrian Mann, BISBOS

Hypothetical Field-based Propulsion


BURKHARD HEIM (1925 - 2001)

Attempt of a unified field theory

Burkhard Heim, German Physicist, taught by W. Heisenberg and C.F. von Weizsäcker

4 fundamental forces
- strong Interaction
- weak Interaction
- Electromagnetism
- Gravitation

General Relativity Theory: Gravitation (Mass) leads to non-Euclidean 4-dim. space time
Gravitation force explained by distortion of space time
→ metric theory of gravitation

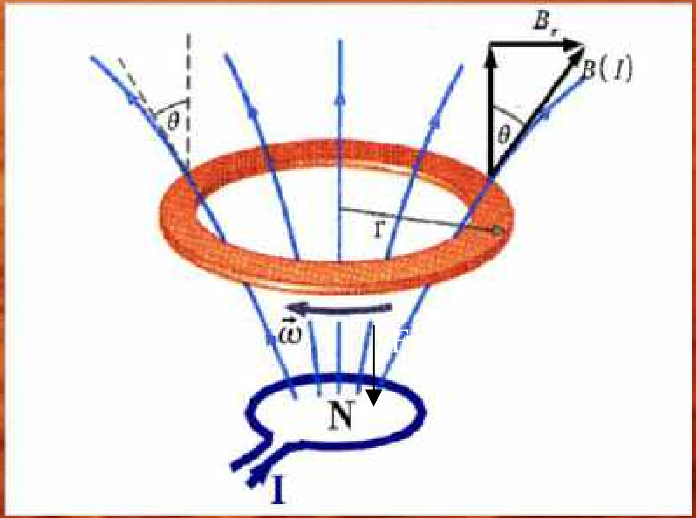
Heim‘s Theory: All forces distort a 6-dimen. „space time“
→ metric theory of all forces;
postulates 2 addtional forces

Scientific correctness of theory and relevance have to be assessed

Hypothetical Field-based Propulsion

Theoretical prediction of a Gravito-Photometric force
→ fifth force, conversion of
electromagnetic field energy in
kinetic energy

proposed experimental set-up:
Measurement of Gravito-Photometric
force on a rotiating torus
in strong magnetic fields ~ 50 Tesla



Application to interplanetary trip to Mars Dröscher & Häuser, AIAA 2003-4990

Magnetic flux:	50 Tesla	Mass of spacecraft:	30000 kg
Rotation rate:	83 Hz (Torus)	Flight duration:	3,7 days
Force:	2700 N		

Mind:
Theory is at the limits of nowadays understanding in physics. Review is not yet performed.
Consider this as considerations that may be worth to look at.

Summary I

- EP Systems and advanced EP systems at IRS introduced
  - Application (e.g. science) and business scenarios
  - Significant thrust density required for manned mission to Mars (or beyond)
- Advanced Propulsion
  - Typical aiming at
    - Advanced propellent acceleration and/or
    - Significant flexibility (e.g. TIHTUS wrt to propellant flexibility of 2nd stage) and/or
    - Increase of mass specific power (and energy) density to solve the α_{lim}*c_e problem and/or
    - New domains in physics that, however, have to be assessed in a critical way

This project has received funding from the European Union's Horizon 2020 research and innovation programme under agreement No 737183

Summary II

- Team networks with
    - Space agencies (NASA- Space Act Agreement, JAXA- Diverse MoUs, ESA- contracts, DLR- contracts)
    - Funding Agencies (such as e.g. EU, DFG)
    - Firms (in particular European)
    - Research institutions (worldwide, e.g. RIAME + Kurtschatov- Cooperation Contract, University of Tokyo, Baylor University, University of Madrid, University of Capetown)
    - ISU
- Network extension (currently ongoing)
    - BIT (e.g. with Prof. Liu as Visiting Professor at IRS)
    - BUAA (e.g. with Prof. Tang as Visiting Professor at IRS, MoU currently in prep.)
    - NCKU, Taiwan
- Instrumentalization of further cooperation
    - Student exchange: Internships, PhDs, scholars
    - Seminar on Project and Strategic Project Management with UPC

THANK YOU

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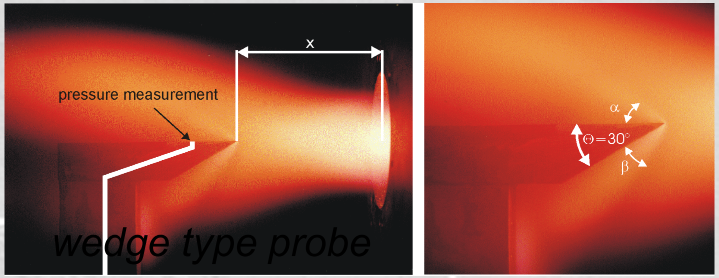
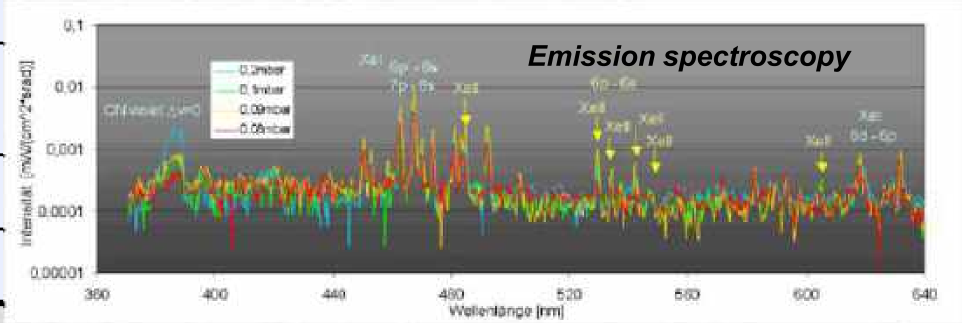
Plasma Diagnostics

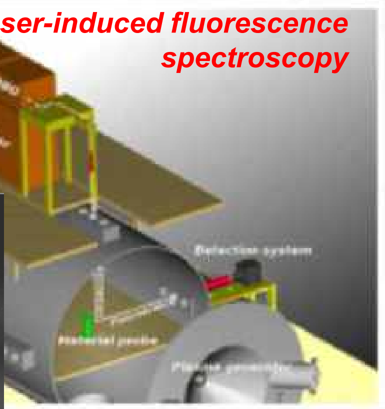
Probe-type	Value measured
Heat Flux Probe	**heat flux**
Pitot Probe	**total pressure**
Mass Spectrometer Probe	**plasma composition**
Wedge Type and Conical Probe	**static pressure, Mach number**
Enthalpy Probe	**enthalpy**
Electrostatic Probes	T_e, T_i, v, n_e, …



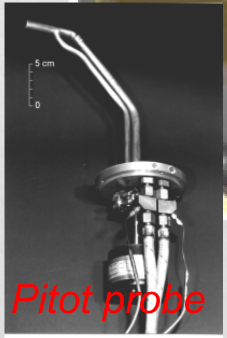
wedge type probe



Emission spectroscopy



Laser-induced fluorescence spectroscopy



Pitot probe

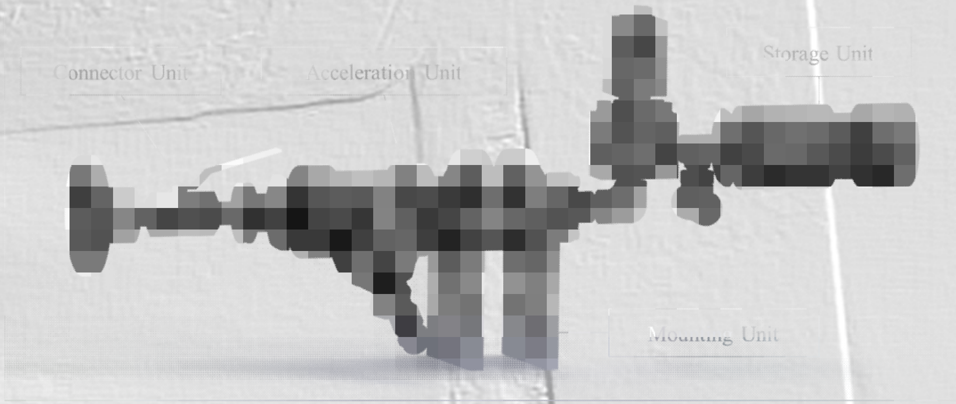
Method	Measured quantity
Emission Spectroscopy (EMS)	T_{ex} , T_{rot} , T_{vib} , T_e , n_e , (n_{Plasma} ?)
Laser-Induced Fluorescence (LIF)	T_{rot} , (T_{vib}), T_e , n_e , n_{Plasma} , v_{Plasma}
Thompson Scattering	n_e , T_e
Fabry-Perot Interferometry (FPI)	T_{Trans} , v_{Plasma}
Laser Absorption Spectroscopy (LAS)	n_{Plasma} , T_{Trans} , v_{Plasma}

Under Development…New Diagnostics and Facilities

- Measurement techniques:
  - Refurbishment of electrostatic probe set-up (in use)
  - Multi-pressure port probe (set in operation)
  - Mechanical material sample holder with cooling (first preliminary tests)
  - Mach-Zender-Interferometer
  - Compact LAS-system (B/L assessment)
- Facilities:
  - Compact Light Gas Gun (<50 cm, readily developed, being manufactured); later in combination with ICP


TPS material probe


electrostatic probe, detector head

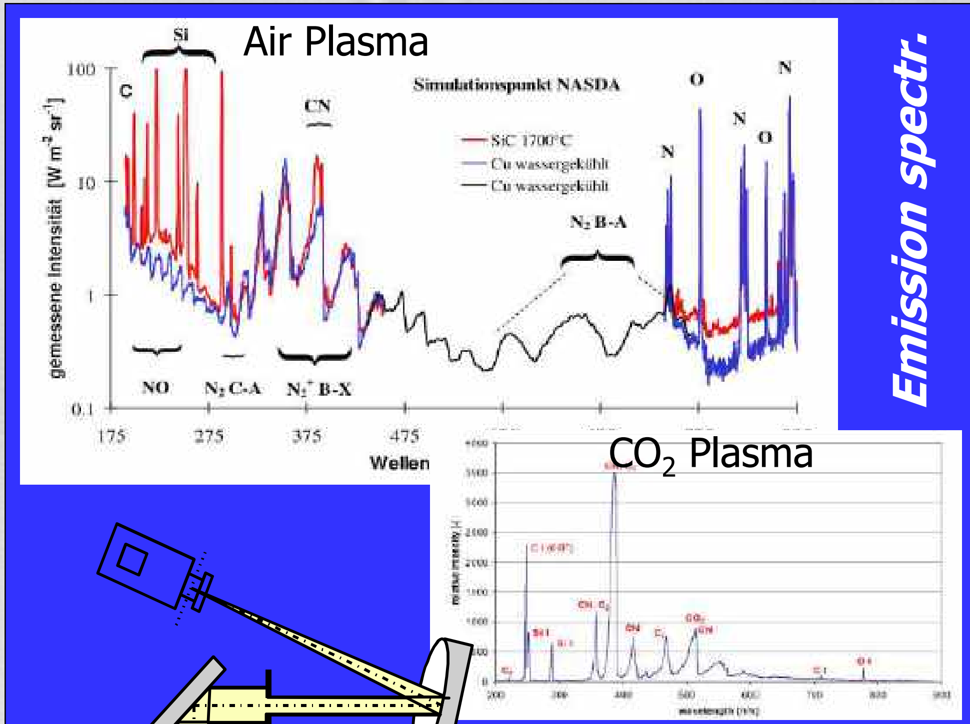


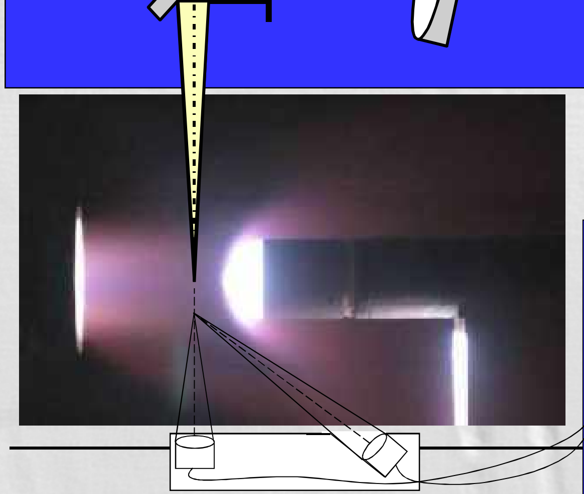
Optical Diagnostics

Emission Spectroscopy, LIF, DLAS, FPI

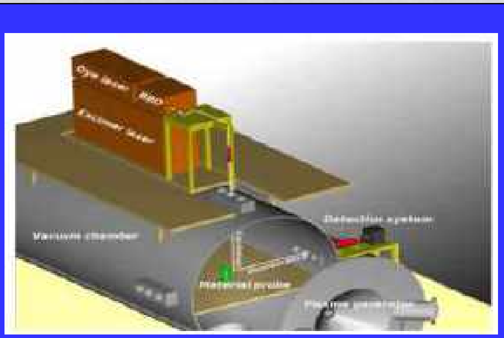
Verification

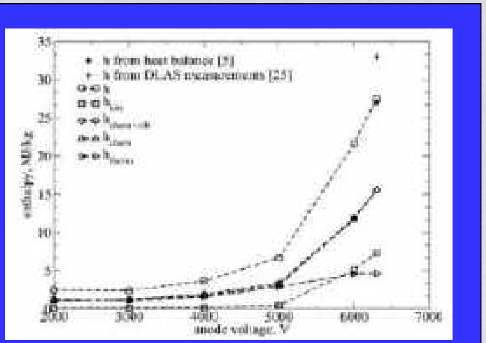
Emission spectr.

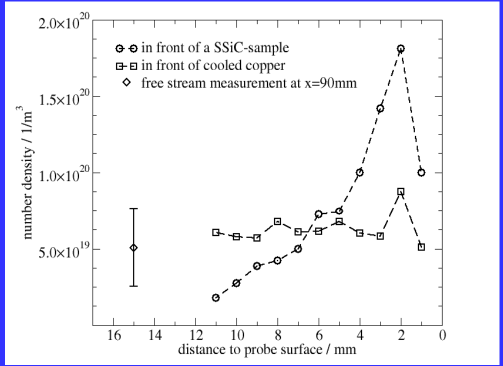


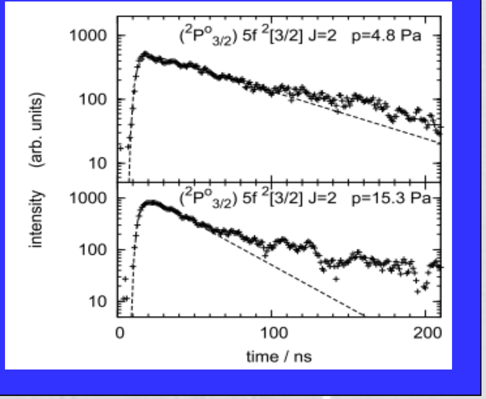


Laser-induced fluorescence





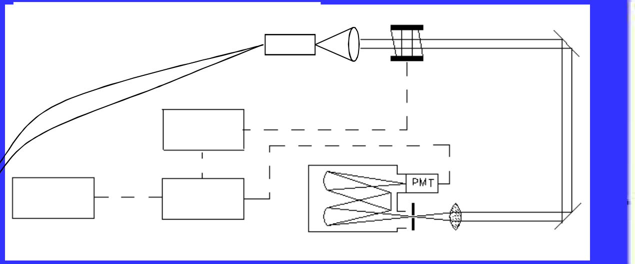


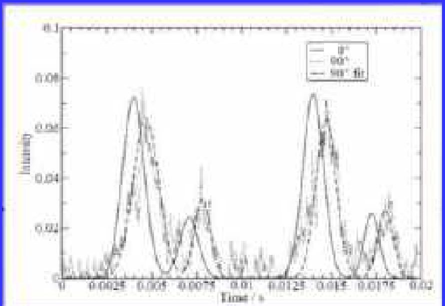


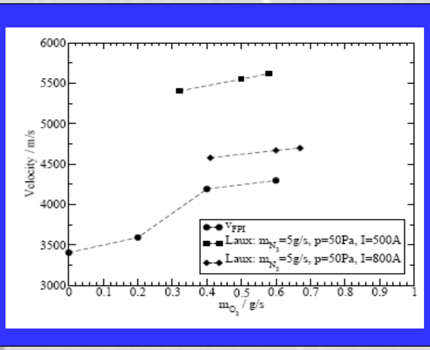
Optical Diagnostics at IRS means

Mobile non-intrusive diagnostic setups

→ species temperatures number densities velocities







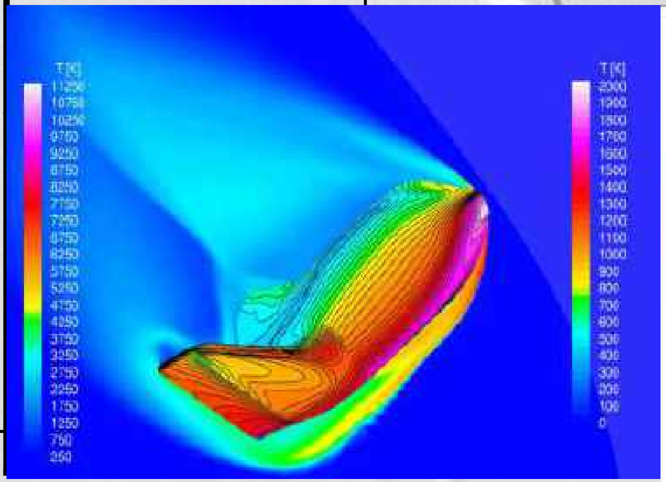
Fabry-Perot-Interf.

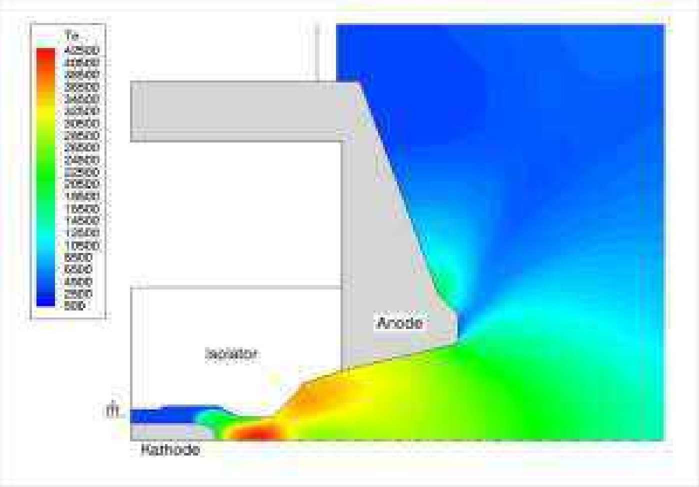
Overview of available codes at IRS

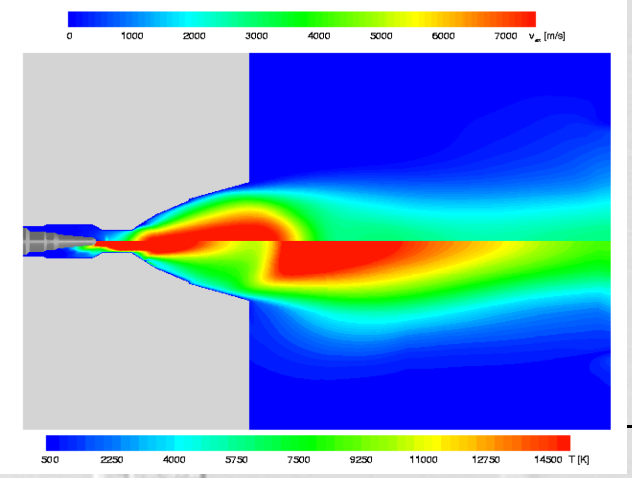
URANUS	SAMSA	SINA / ARCHE	PICLas	LasVegas
Navier-Stokes			Particle Method / Boltzmann Equation	
continuum flow, thermal and chemical non-equilibrium			rarefied *plasmas*, strong non-equil.	Rarefied *gases*, strong non-equil.
re-entry	MPD (self and applied field)	TLT, IPG, PWK	PPT, Ion thruster,…	Re-entry
2D rotational symm. / 3D	2D rotational symmetric	3D (rotational symmetric)	3D	2D rotational symmetric
fully implicit	explicit			
fully coupled		loosely, iteratively coupled		single
structured multiblock grids	unstructured, adaptive grids	structured multiblock grids	Unstructured grids	unstructured, adaptive grids
Air, CO_2	Argon	Air, N_2, H_2, CO_2	any	any
PARADE/HERTA gas-radiation coupling		**HERTA** gas-radiation coupling		
Gas kinetic gas-surface interaction model with catalytic reaction schemes. CVCV mult. temperature gas-phase model		changeable chemical modules		Gas kinetic gas-surface interaction model with catalytic reaction schemes.

Numerical / theoretical modeling: Available codes at IRS

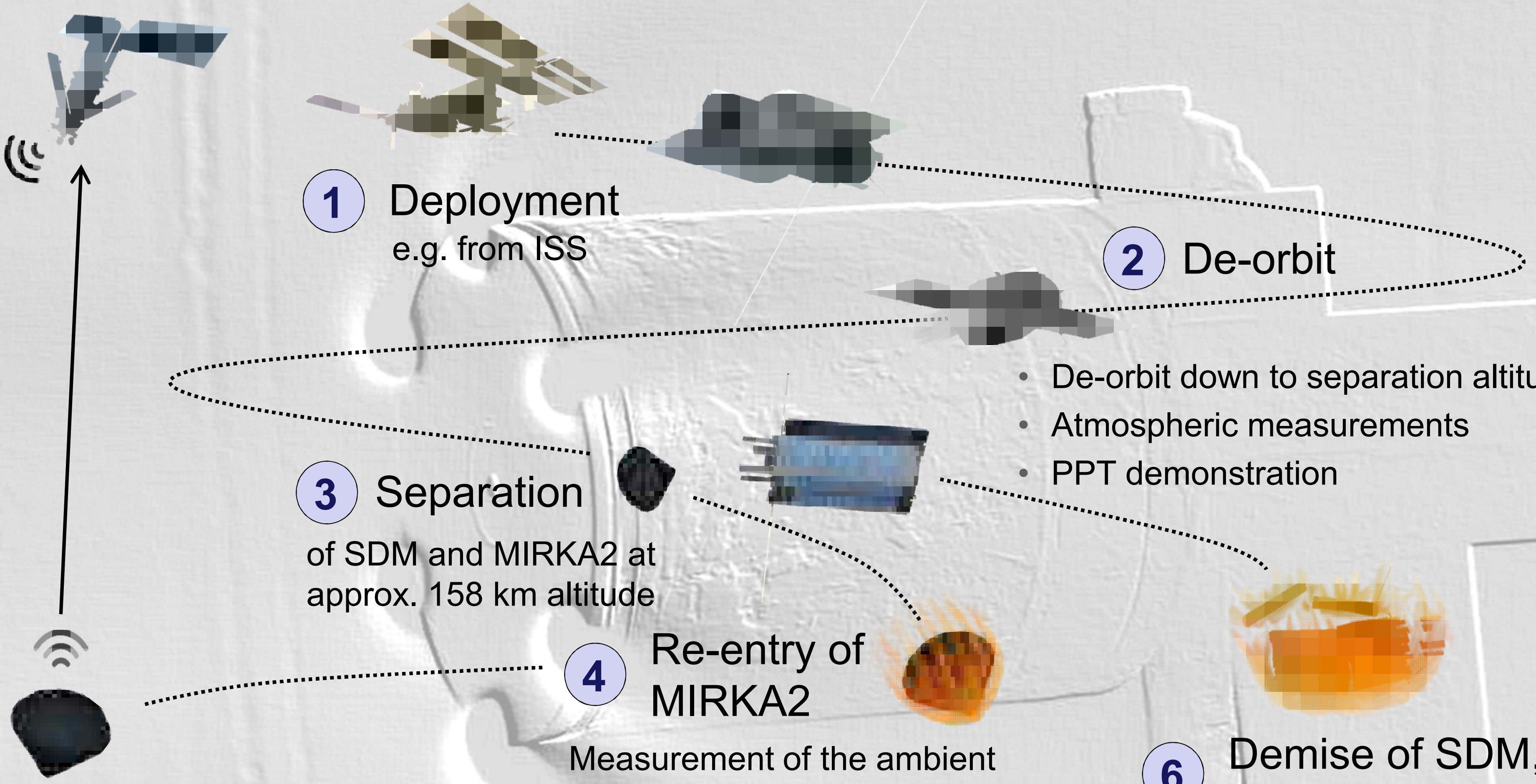
URANUS	**SAMSA**	**SINA / ARCHE**	**PICLas**	**LasVegas**
Navier-Stokes			Particle Method / Boltzmann Equation	
continuum flow, thermal and chemical non-equilibrium			rarefied *plasmas*, strong non-equil.	Rarefied *gases*, strong non-equil.
re-entry	MPD (self and applied field)	TLT, IPG, PWK	PPT, Ion thruster,…	Re-entry
2D rotational symm. / 3D	2D rotational symmetric	3D (rotational symmetric)	3D	2D rotational symmetric
fully implicit	explicit			
fully coupled		loosely, iteratively coupled		single
structured multiblock grids	unstructured, adaptive grids	structured multiblock grids	Unstructured grids	unstructured, adaptive grids
Air	Argon	Air, N_2, H_2	any	any







CubeSat Atmospheric Probe for Education (CAPE)



① **Deployment**

e.g. from ISS

② **De-orbit**

- De-orbit down to separation altitude
- Atmospheric measurements
- PPT demonstration

③ **Separation**

of SDM and MIRKA2 at approx. 158 km altitude

④ **Re-entry of MIRKA2**

Measurement of the ambient conditions and the re-entry performance of the capsule

⑤ **Transfer of re-entry data via satellite**

Transmission of data through Iridium communication system

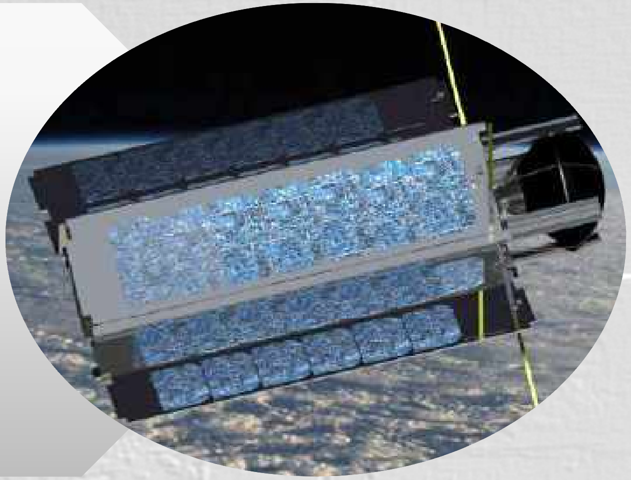
⑥ **Demise of SDM, impact of MIRKA2**

Potential for monitoring of break-up and demise

CAPE Vehicle Configuration

Service and Deorbit Module SDM *(3 CubeSat units):*

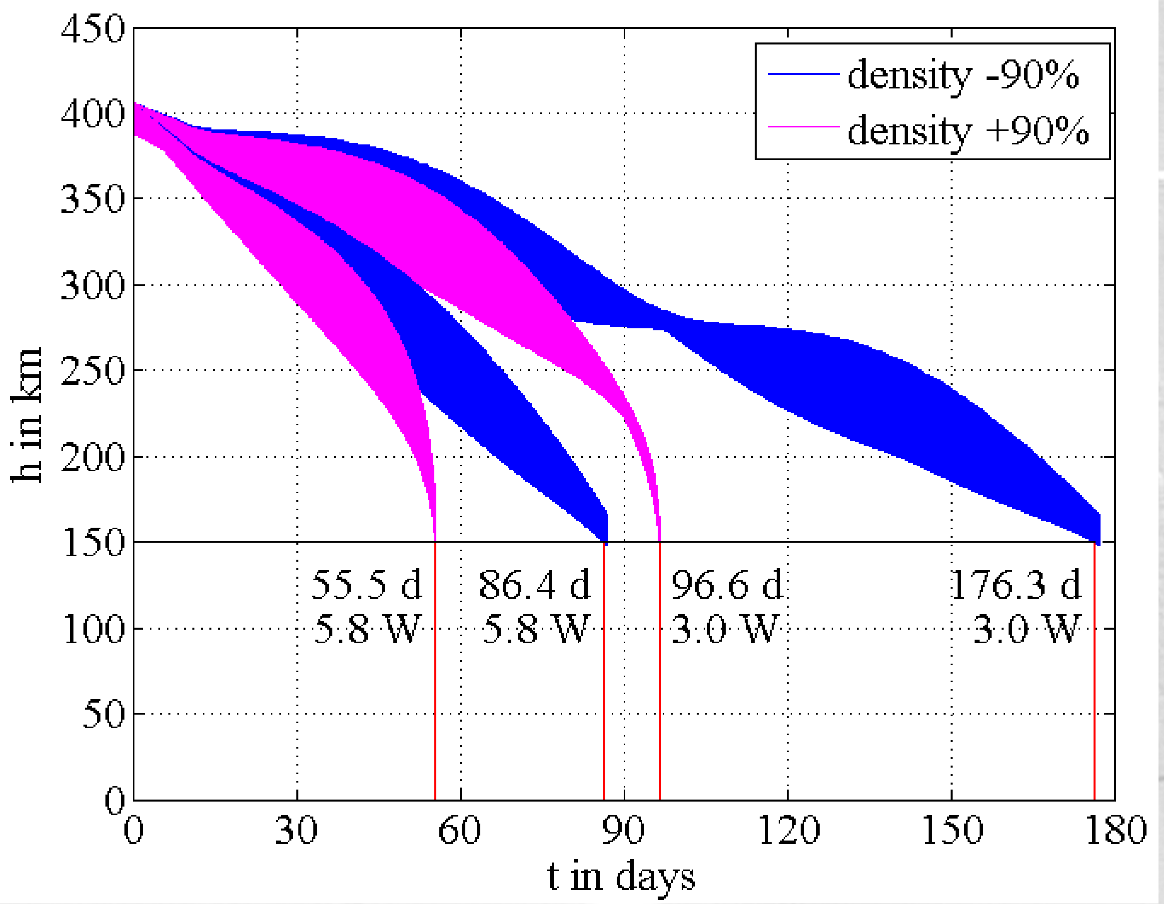
- Performs deorbit manoeuvre using PPT
- Demises upon re-entry
- Scientific payload: e.g. FIPEX, dust sensors
- Potential standard carrier for future CubeSat science missions



MIRKA2 re-entry capsule *(1 CubeSat unit):*

- Uses ZURAM-based ablative TPS
- Scientific payloads: e.g. Thermocouples, radiometer, pressure transducers, etc.
- Potential standard for flight qualification of heat shield materials

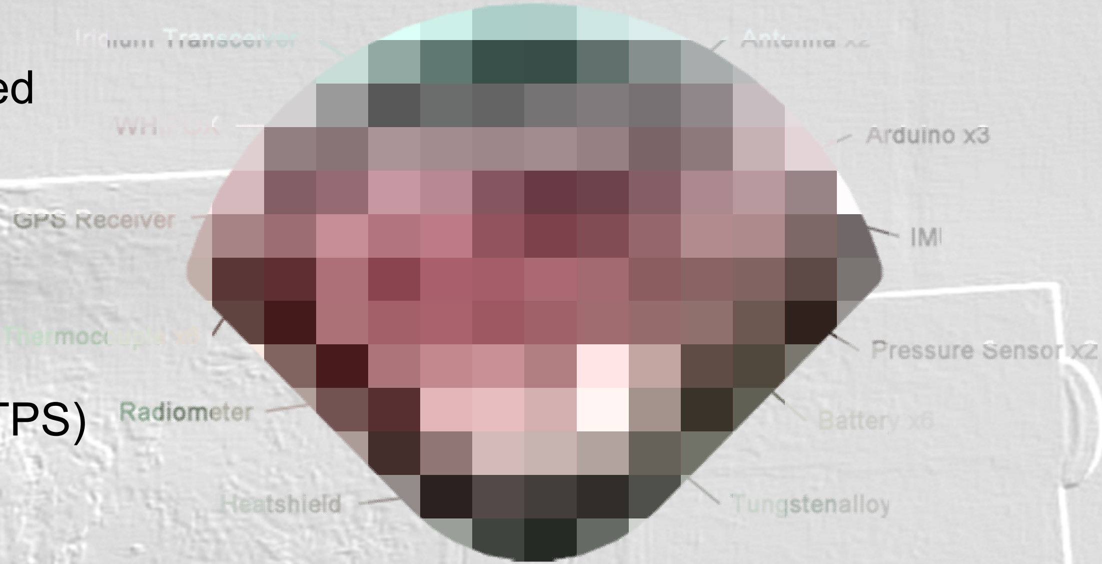
Trajectory Analysis



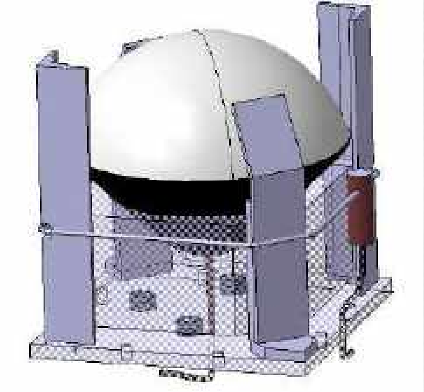
- Mission analysis performed with IRS in-house code REENT
- Starting orbit: ISS
  - 400 km altitude
  - 51.56° inclination
- Downward spiralling manoeuvre
- Propulsion system: Pulsed Plasma Thruster (PPT):
  - 93 µN at 5.8 W (continuous)
  - 48 µN at 3.0 W (continuous operation with extra contingency for payloads)
- De-orbit:
  - Min.: 56 days
  - Max.: 176 days

Technology Demonstration: Micro Return Capsule

- CubeSat-compatible size
- Small size allows for ground-based testing of entire vehicle, e.g. in plasma wind tunnel
- Basic measurements of:
  - flight behaviour
  - re-entry environment
  - thermal protection system (TPS) performance
- Transmission of data via Iridium network after black-out phase




Air, PWK3-ZURAM, 3 MW/m^2



- Separation via LOTUS (Low Orbit Technical Unit Separator) spring-powered ejection mechanism
- High-performance ablator materials ZURAM considered as candidates for TPS

  → Flight qualification

Scientific Application: Atmospheric Characterisation

Two optional on-board scientific experiments are under investigation. The gradual downward spiralling manoeuvre provides ideal conditions for a spatial characterisation of the lower thermosphere.

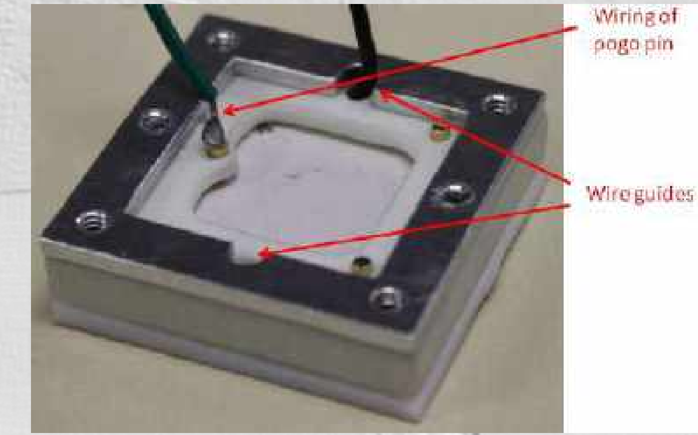
FIPEX

- In-situ measurement of atomic oxygen concentrations in lower thermosphere
- Successfully flown on ISS

Piezo Dust Detectors

- In-situ measurement of impact velocity and incidence angle of µm- to mm-sized dust particles
- Relies on piezoelectric effect

(DLR)



Potential Applications of CAPE

Technological / economical

- Cost-effective potential standard for
  - Flight qualification of thermal protection system materials
  - Nanosatellite servicing
- Conceivably controlled capture and deorbit of space debris and meteoroids using SDM

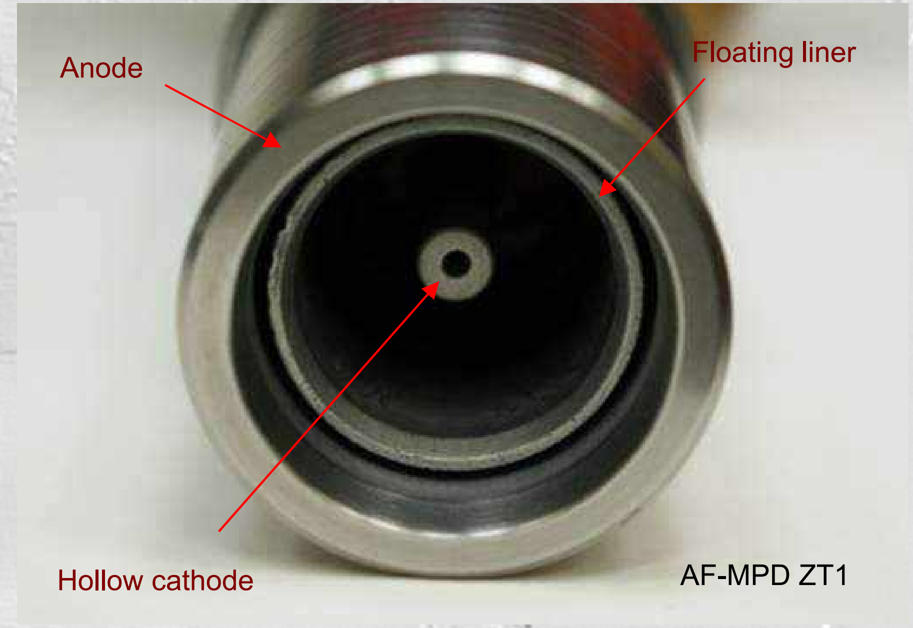
Scientific

- Standardised platform for spatially and temporally resolved atmospheric measurements in general.
- Quantitative data of contamination of atmospheric gas sensor measurements through electric thruster exhaust.
- Micro return capsule data provides valuable reference scenario with direct application to ground testing facilities.
- Potential for tracking of atmospheric break-up and demise of SDM

Experimental Results of AF-MPD ZT1 Thruster

Single channel hollow cathode:

- Total operation time in steady state mode: 3550 s
- Number of ignitions: >12
- Number of test runs: 6



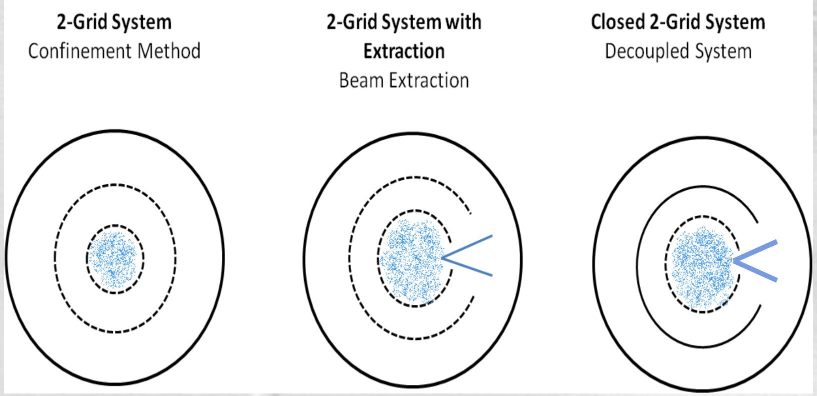
Future activities:

- Development of Laboratory thruster SX1
- New applied-field coil: up to 0.6 T
- Increase of efficiency can be expected
  → Better performance
- Possible development of engineering model (Improved passively cooled ZT1 thruster)
  - Redesign of insulation parts (PEEK -> ceramics)
  - Improvement of sealing and propellant injection

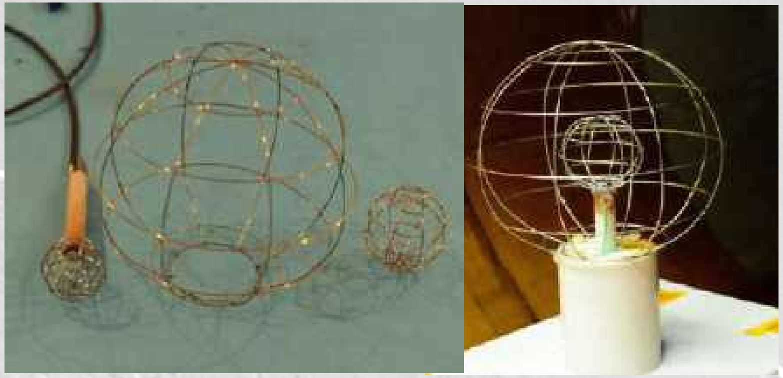
IEC-Setup at IRS

Discharge characterization …

- Variation of propellant (understanding e.g. wrt impact to discharge mode):
  - Argon: easy to ionize, easy to procure
  - Helium: high first ionization, low molar mass
  - Nitrogen: molecular, 80% in air → alternative propellants
- Electrode size variation → discharge power:
  - Interaction with electrode distance and surface and grid transparancy



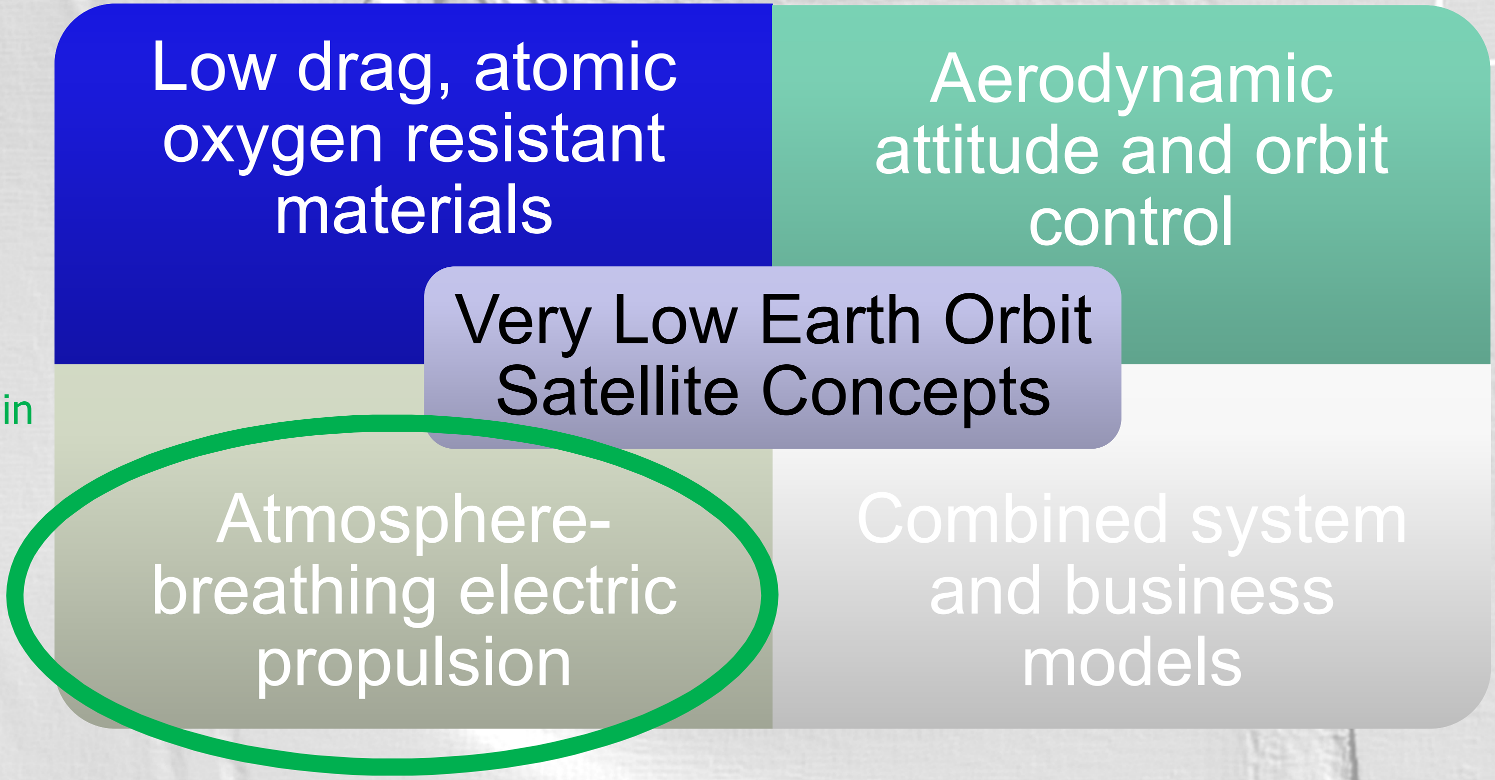
Modular grid configurations: C1/C2



Grid configurations

Grid Configuration	Anode Diameter	Kathode Diameter
C1	30 cm	10 cm
C2a	15 cm	5 cm
C2b	15 cm	3.5 cm

This project has received funding from the European Union's Horizon 2020 research and innovation programme under agreement No 737183



Low drag, atomic oxygen resistant materials

Aerodynamic attitude and orbit control

Very Low Earth Orbit Satellite Concepts

IRS Main Task

Atmosphere-breathing electric propulsion

Combined system and business models