Lecture Series on NUCLEAR SPACE POWER & PROPULSION SYSTEMS -2- Nuclear Thermal Propulsion Systems (Last updated in January 2021) Eric PROUST

Eric Proust

FROM RESEARCH TO INDUSTRY

LECTURE SERIES ON

NUCLEAR SPACE POWER & PROPULSION SYSTEMS

-2- Nuclear Thermal Propulsion Systems

Eric PROUST

Commissariat à l'énergie atomique et aux énergies alternatives - www.cea.fr

Last update: January 2021

Nuclear Space Power & Propulsion in the last 2 month news

Nuclear Thermal Propulsion

NUCLEAR ENGINEERING INTERNATIONAL

Home | News | Features | Opinion | Video | Events | Jobs | Buyers' Guide | White Papers | Press R

UK Space Agency and Rolls Royce to co-operation on nuclear propulsion

14 January 2021

The UK Space Agency (UKSA) is joining forces w Rolls-Royce for a unique study into how nuclear and technologies could be used as part of space exploration.

This new research contract will see planetary scientists work together to explore the potential c nuclear power as a more plentiful source of ener capable of making possible deeper space.

Nuclear propulsion, which would involve channel fission energy to accelerate propellants, such as hydrogen, at huge speeds, has the potential to revolutionise space travel, UKSA said. By some estimates, this kind of engine could be twice as efficient as the chemical engines that currently power rockets.

Spacecraft powered by nuclear propulsion could, conceivably, make it to Mars in 3-4 months, of half the time of the fastest possible trip in a spacecraft using the current chemical propulsion.

Nuclear space power is expected to create new skilled jobs across the UK to support the burgeoning UK space economy.

"As we build back better from the pandemic, it is partnerships like this between business, industry and government that will help to create jobs and bring forward pioneering innovations that will advance UK

Forbes

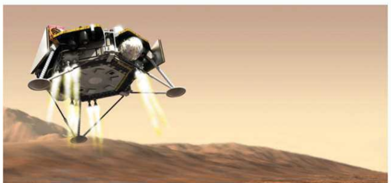
ec 21, 2020, 10:10am EST | 1,155 views

Nuclear Propulsion To Be A Key Part Of US Space Strategy

Ariel Cohen Contributor

Energy

I cover energy, security, Europe, Russia/Eurasia & the Middle East



SCIENTIFIC AMERICAN

FUTURE

SPACE

Trump Signs Directive to Bolster Nuclear Power in Space Exploration

One goal laid out in the new policy is the testing of a fission power system on the moon by the mid- to late 2020s

By Mike Wall, SPACE.com on December 21, 2020

NUCLEAR ENGINEERING INTERNATIONAL

Home | News | Features | Opinion | Video | Events | Jobs | Buyers' Guide | White Papers | Press Releases

Russia signs contract for design of nuclear space tug

18 December 2020

Nuclear Electric Propulsion

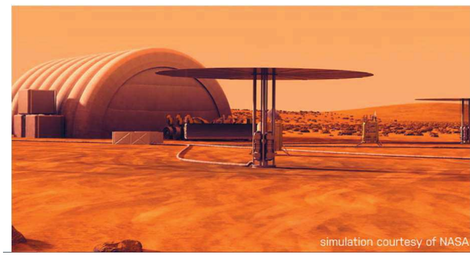
wnn world nuclear news

Energy & Environment | New Nuclear | Regulation & Safety | Nuclear Policies | Corporate | Uranium & Fuel |

Los Alamos spin-off to commercialise space reactors

04 November 2020

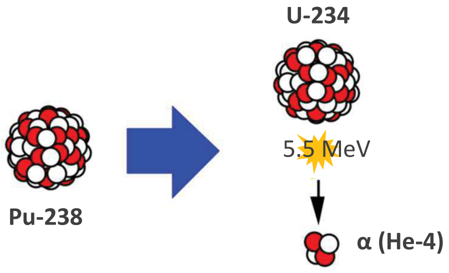
Los Alamos National Laboratory (LANL) has agreed to license Kilopower space reactor technology to New Mexico company Space Nuclear Power Corporation (SpaceNukes), which aims to commercialise the technology for use in space in the next few years.



Space Nuclear Power Reactor

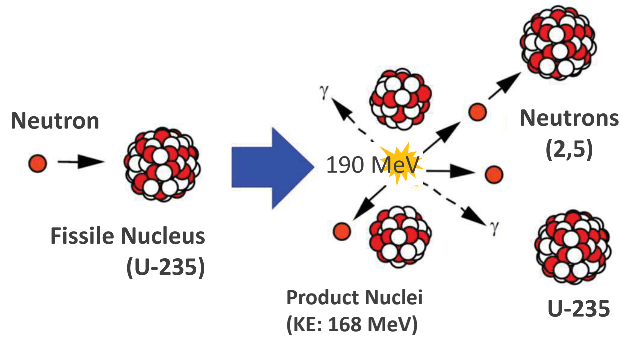
Space Nuclear Power Systems: Radioisotope or Fission-based?

Energy released by the **radioactive decay (alpha) of a radioisotope**



Applications:
- Thermal management: **RHU**
- Power generators: **RTG**, DIPS

Energy released by the **neutron-induced fission** of a **fissile nuclide**



Applications:
- **Power generation**, for supplying
  - A moon/mars base
  - Electric thrusters (**Nuclear Electric Propulsion**: **NEP**)
- **Direct propulsion** (by heating a propellant gas)
  - **Nuclear Thermal Propulsion (NTP)**
- Both combined

Today's lecture ➡

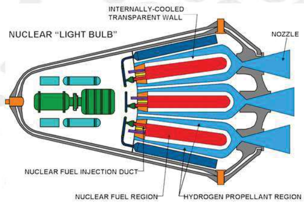
Lecture Outline (1/2)

- **Why Nuclear Thermal Propulsion?**
  - In-space propulsion principle; Nuclear Space Propulsion: Thermal or Electric; Space propulsion: some basics; Performances: NTP vs Chemical prop; NTP: enabler of manned missions to Mars?
- **The US Rover/NERVA Program (1956-1972)**
  - 27 NTP rocket reactors and 3 nuclear engines ground tested; NERVA Rocket Engine Design; NERVA nuclear fuels; Program achievements; The program legacy engine concept: the SNRE
- **The USSR NTP Program (~1960-1989)**
  - An effort comparable with the US, a full carbide fuel, a quite different design approach
- **Nuclear Fuels for NTP: Beyond Composite/Carbides Fuels, Cermet Fuels**
  - NTP nuclear fuel design issues; W-UO_2 Cermet fuel developments in the 60's; Cermet-fuel-based engine concepts: ANL 2000, ANL 200, XNR2000; Cermet vs. carbide fuels for NTP
- **The CEA-CNES MAPS Study Program (1994-1997)**
  - Study goals; MAPS engine conceptual design; Safety issues for NTP; Development and ground testing approaches; The challenges of nowadays testing nuclear rocket engines
- **Current Orientations**
  - Why waiting decades?; 2010: new goal of humans orbiting Mars by mid 30's; NTP engines for manned Mars mission; current NASA project to assess the feasibility of an LEU-based engine; Impacts of switching from HEU to LEU fuel

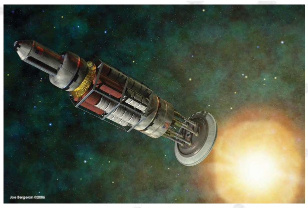
Lecture Outline (2/2)

And then, for you to choose one among 3 bonus presentations:

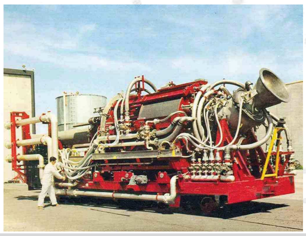
#1 ''Advanced'' Nuclear Thermal Propulsion Systems



#2 Nuclear Pulse Space Propulsion Systems

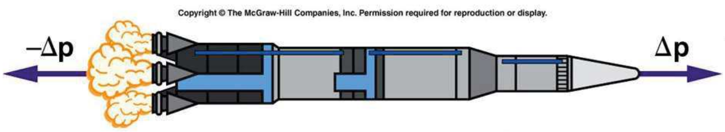


#3 Air-Breathing Nuclear Thermal Propulsion

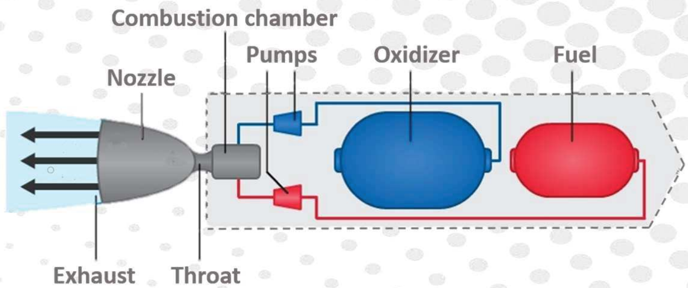


In-Space Propulsion: Basic Principle

Remember: it's Newton's 2nd Law of Motion



Liquid Chemical Rocket

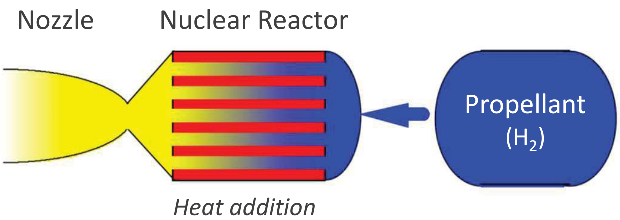


$$Thrust = \dot{m}\, v_{eject} + p_e\, A_e$$

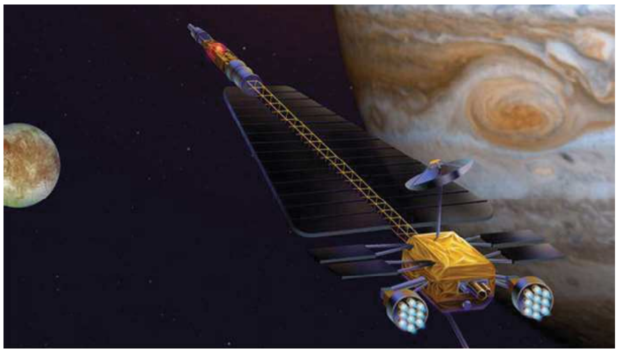
$\dot{m}$	Ejected mass flow rate
v_{eject}	Velocity of ejected gases
p_e	Ejected gases pressure at nozzle exit
A_e	Nozzle exit area

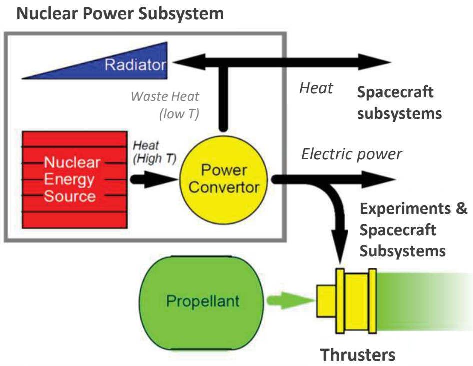
Nuclear In-Space Propulsion: Can Be Thermal (NTP), or Electric (NEP)

Nuclear Thermal Propulsion



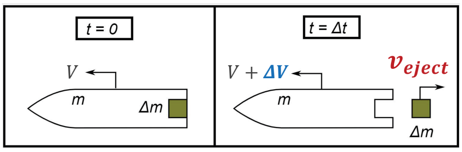
Nuclear Electric Propulsion





In-Space Propulsion: some basics

Launch mass (cost) exponentially decreases with v_{eject}



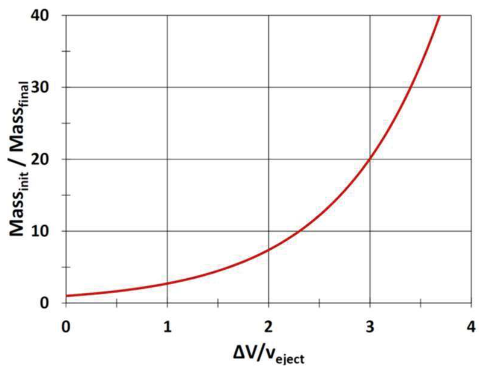
$$F_{thrust} = m\frac{dV}{dt} = v_{eject}\frac{dm}{dt}$$

$$\int_{V}^{V+\Delta V} dV = -\,v_{eject}\int_{M_{init}}^{M_{final}} \frac{1}{m}\,dm$$

$$\Delta V = v_{eject}\ln\left(\frac{M_{init}}{M_{final}}\right)$$

$$M_{init} = M_{final}\, e^{\Delta V / v_{eject}}$$

(Tsiolkowsky's "rocket equation")



For **"thermal" rocket engines** (chemical, nuclear thermal)

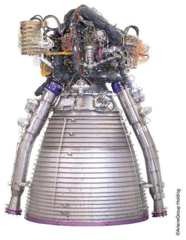
$$v_{eject} \propto \sqrt{\frac{2\kappa}{(\kappa-1)}\frac{RT}{M}}$$

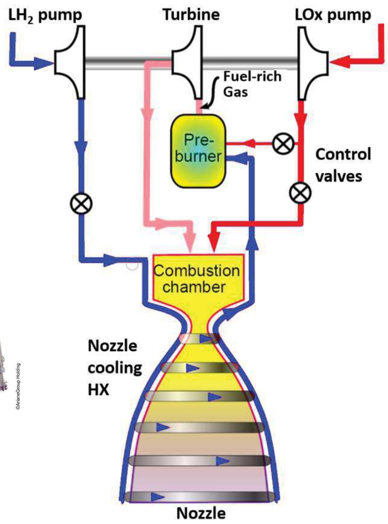
T : chamber temperature (K)
M : molecular weight
$\kappa = C_P/C_V$

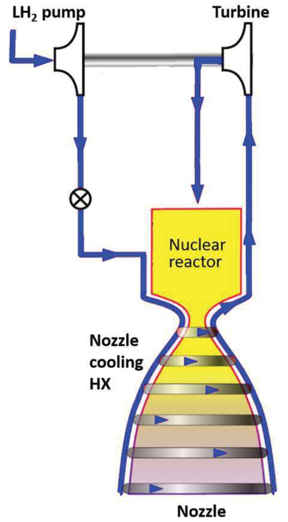
Specific Impulse: $I_{sp}\,(s) = \dfrac{v_{eject}}{g_0}$

Nuclear Thermal Propulsion: a ~Twice Higher Isp than Chemical Propulsion

$$I_{sp} \propto \sqrt{\frac{2\kappa}{(\kappa-1)}\frac{RT}{M}}$$





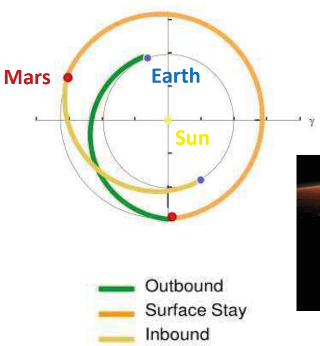


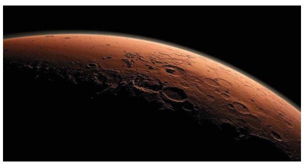
Chemical (LH_2 /LO_2) : M~13.8 g/mol, T ~3420 K ⇨ I_{sp} **~480 s**
Thrust ~ 2 000 kN; burn time ~500 s; thrust/weight ~150
''Energy-limited'' performances (energy stored in chemical bounds)

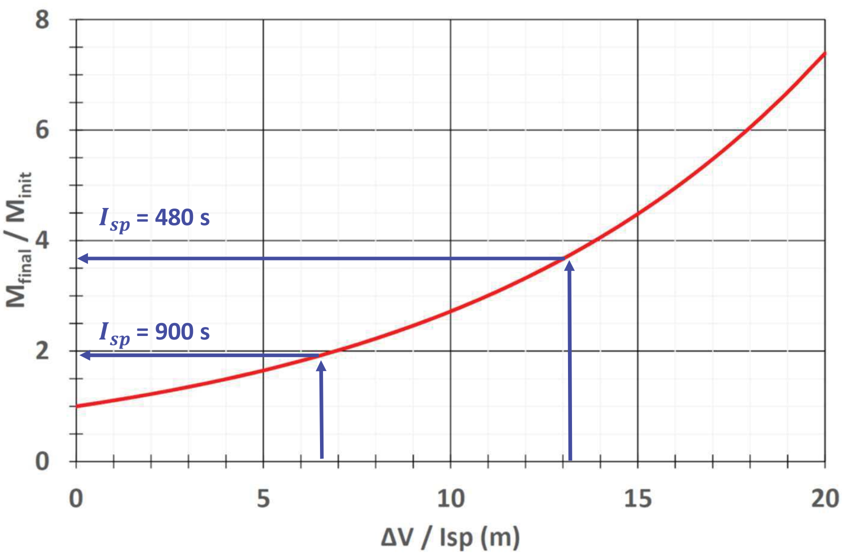
Nuclear Thermal (LH_2 propellant): M~2 g/mol, T ~2700 K ⇨ I_{sp} **~900 s**
Thrust ~50 - 1 000 kN; burn time ~1 000 s; thrust/weight ~10 - 30
Performances limited by fuel resistance to high temperature H_2

Space Propulsion: an example of the benefits of a twice higher Isp

Example of **Earth-Mars round trip**: **a twice higher I_{sp}**
⇨ **reduces mass to put in LEO (cost) by a factor ~2**
or ⇨ enables to **shorten manned round trip time** (space radiation dose!)





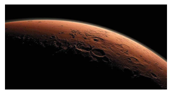


Round trip low **Earth** orbit ⇔ low **Mars** orbit:
minimum ΔV **~6 km/s**

Chemical (Thermal) propulsion:
I_{sp} **~480 s** ⇨ ΔV/I_{sp} ~13 ⇨ M_{init}/M_{final} **~3.7**
Nuclear Thermal Propulsion:
I_{sp} **~900 s** ⇨ ΔV/I_{sp} ~6.7 ⇨ M_{init}/M_{final} **~1.9**

Space Propulsion: another example of the benefits of a twice higher Isp (1/2)

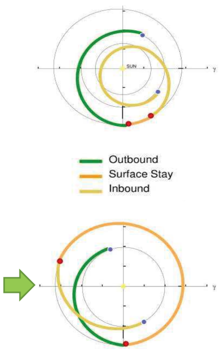
Manned Mission To Mars: ΔV budget vs. Transit Time / Radiation Dose*

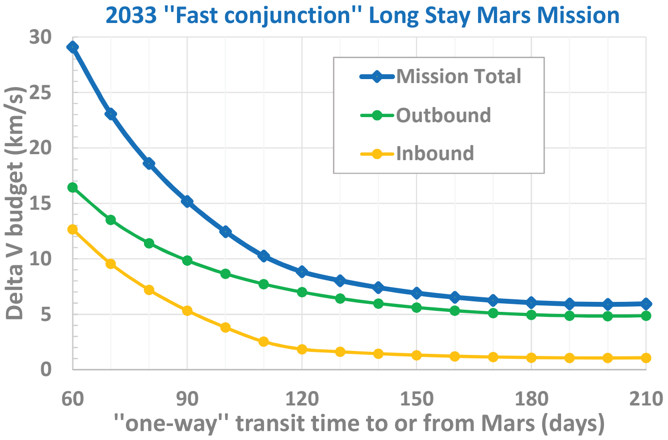


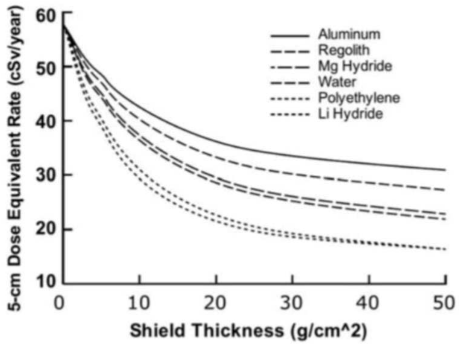
Annual Ambient Radiation Levels for the Earth, Mars and Space*

	Earth	Mars	Moon	Space
Annual Total	3 mSv	245 mSv	438 mSv	657 mSv
Daily Average	$8.2\ 10^{-3}$ mSv	0.67 mSv	1.2 mSv	1.8 mSv

Source: L. Joseph Parker, Human radiation exposure tolerance and expected exposure during colonization of the moon and Mars, 2016







(Poor) Shielding effectiveness against galactic cosmic radiation at solar minimum

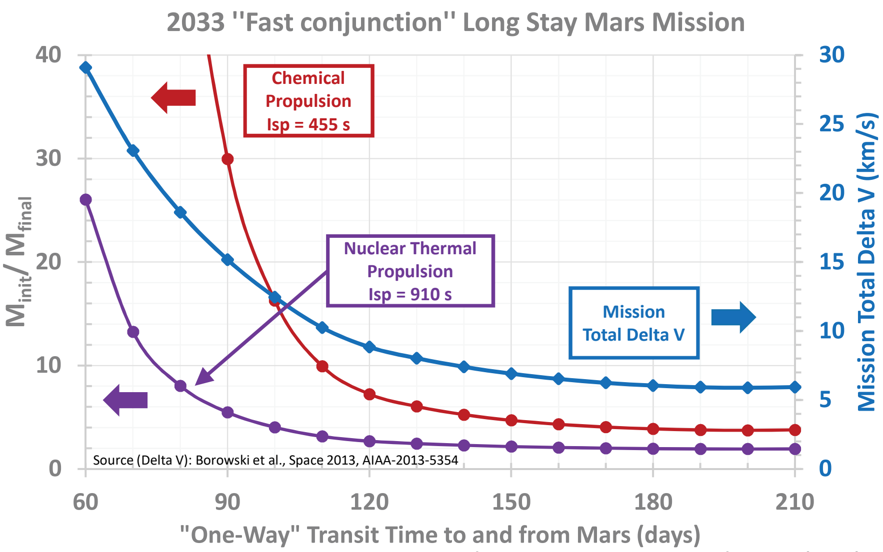
* See back-up slide

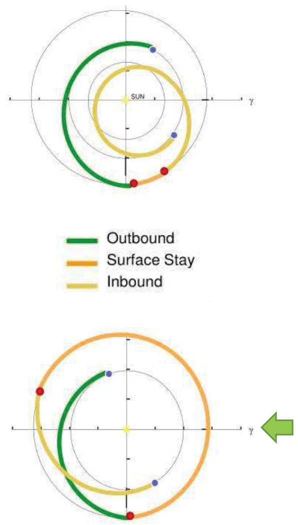
Source: based on Borowski et al., Space 2013, AIAA-2013-5354

Space Propulsion: another example of the benefits of a twice higher Isp (2/2)

Mission mass ratios vs. transit time to Mars

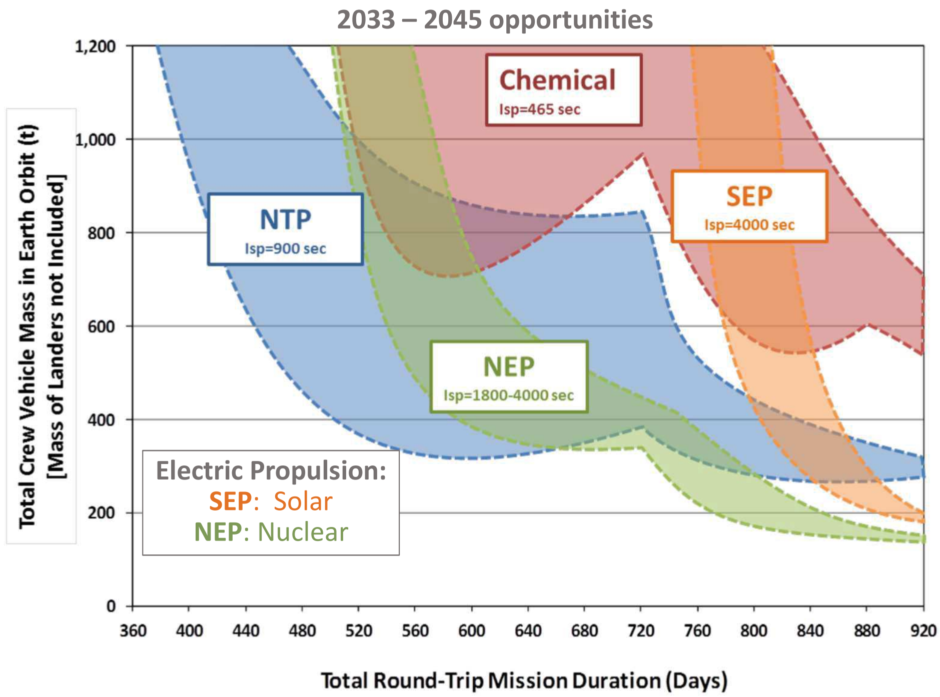
$M_{init} = M_{final}\, e^{\Delta V / g_0 Isp}$ M_{init}: initial total mass of spacecraft in LEO (= Spacecraft mass + Payload mass **+ propellant mass**)

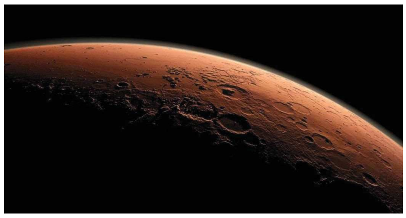




Reminder: average space radiation level ~1.85 mSv/day

Nuclear Thermal Propulsion (NTP): enabler of manned missions to Mars?





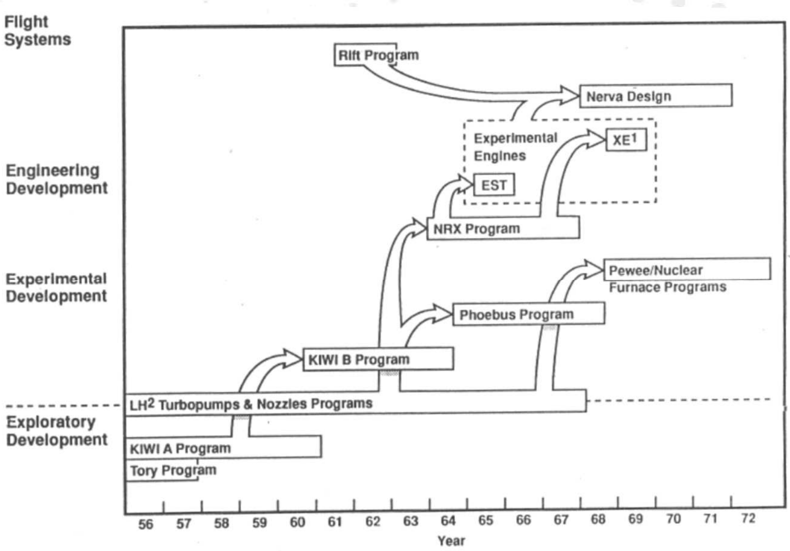
For an detailed explanation of why, although EP has a much higher Isp, NTP outperforms NEP (and SEP), wait for my next lecture on Space Fission Power and Electric Propulsion

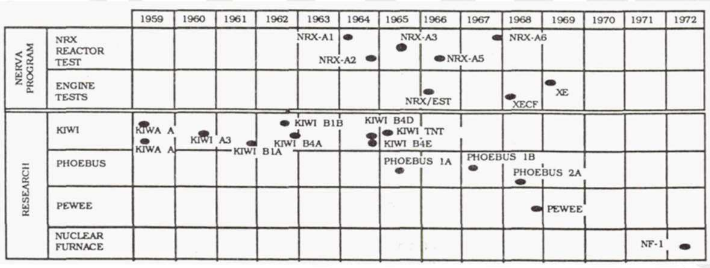
The short explanation: electric thrusters have a very (very) low thrust* and they need a power supply

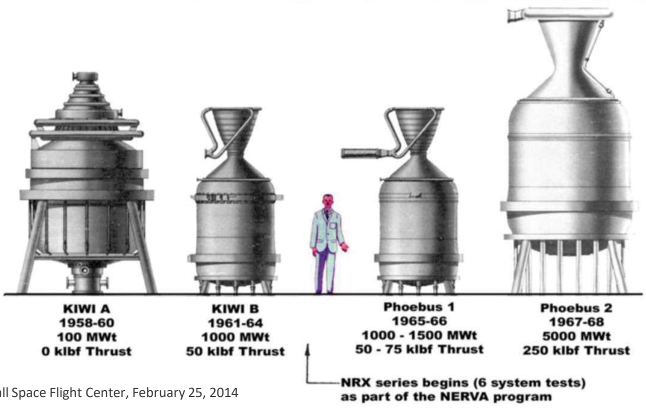
** See back-up slide*

The US Rover/NERVA program (1956-1972): 20 NTP rocket reactors designed, built and ground tested

- 1956 – 1972, Project "Rover" / NERVA (Nuclear Engine for Rocket Vehicle Application)





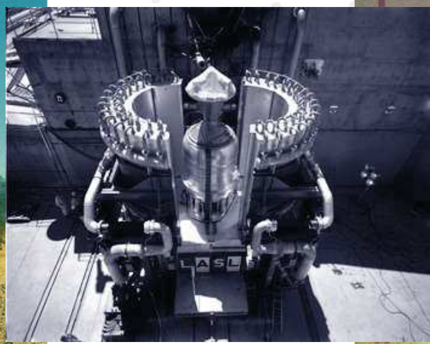


Source: Nuclear Thermal Propulsion Ground Test History – The Rover/NERVA Program; Harold P. Gerrish, NASA Marshall Space Flight Center, February 25, 2014

Phoebus-2A: the most powerful nuclear rocket reactor ever tested (1968, Rover/NERVA Program)

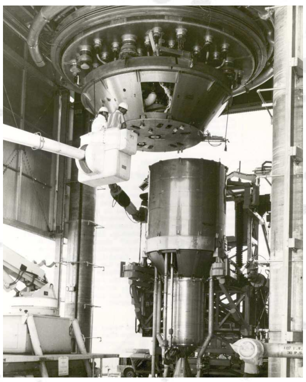
Phoebus-2A being railed to its test-stand,
at its test stand and during a high-power test

The reactor operated for ~32 minutes,
including 12 minutes at > 4 GWth power



Source: Nuclear Thermal Propulsion Ground Test History – The Rover/NERVA Program; Harold P. Gerrish, NASA Marshall Space Flight Center, February 25, 2014

NERVA XE': as close as possible to a flight engine (1969)

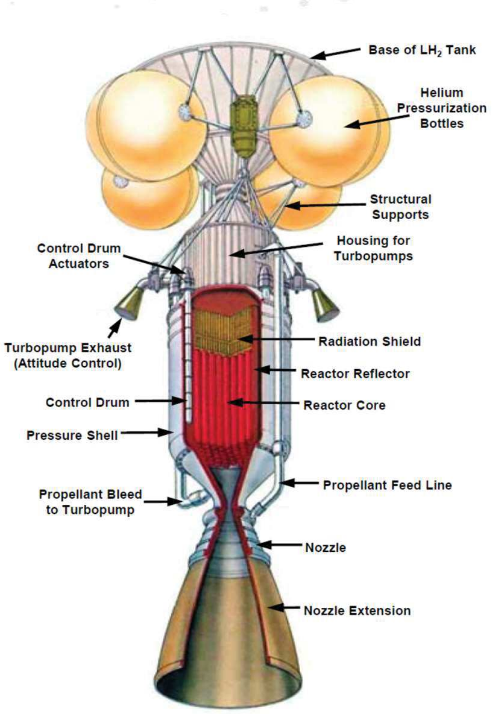


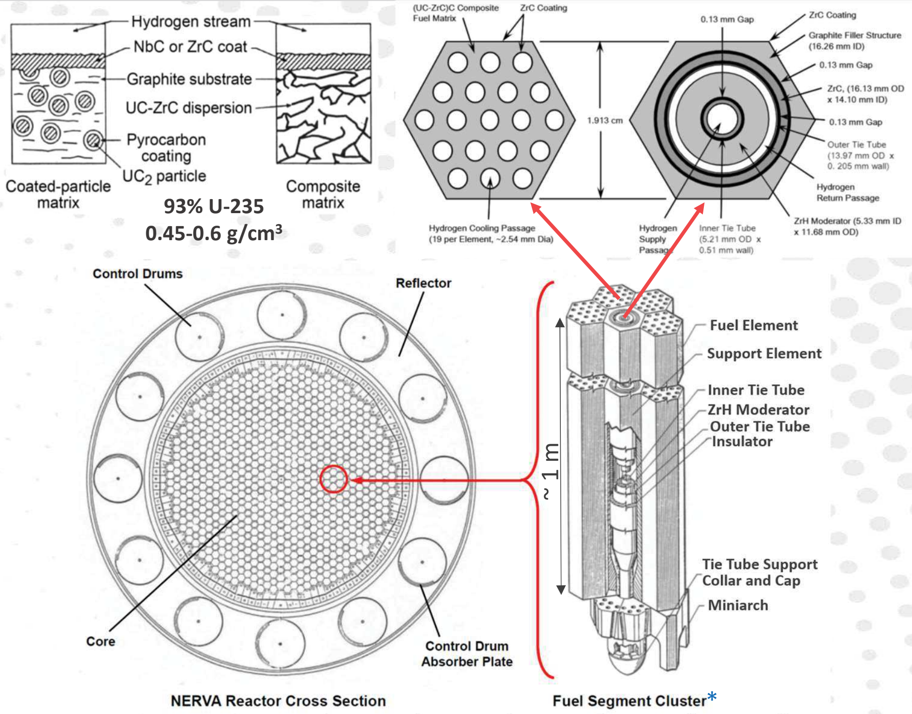
1140 MW nuclear reactor integrated in a complete mock-up of a nuclear rocket flight engine, tested in a simulated space vacuum

710 s Isp (hot bleed cycle),
2 270 K chamber temperature, 24 restarts,
28 minutes at full power / **250 kN thrust**

Source: Nuclear Thermal Propulsion Ground Test History – The Rover/NERVA Program; Harold P. Gerrish, NASA Marshall Space Flight Center, February 25, 2014

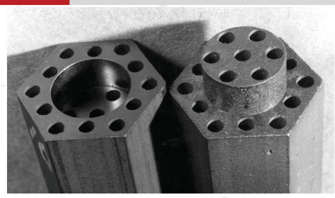
Rover/NERVA-Type Engine: Typical Design

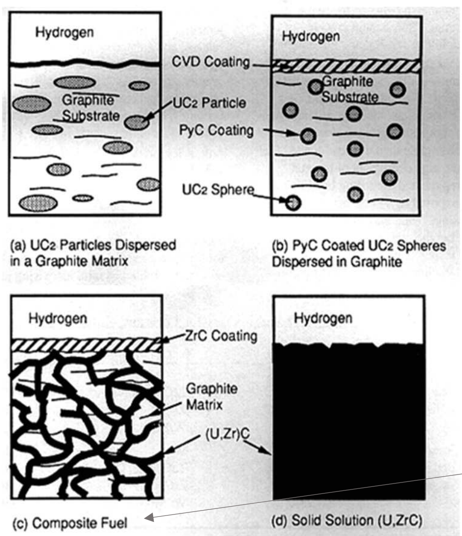




** SNRE design; the only tested NTP reactor having used ZrH moderation is Pewee*

NERVA Fuels



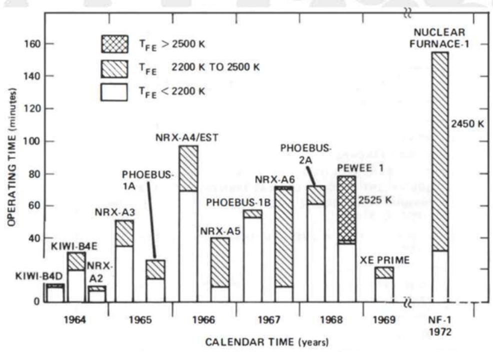


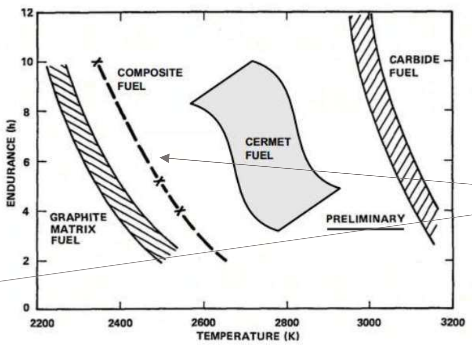
Thermal properties of nuclear thermal rocket Carbide fuel options

Fuel Option	Melting point (K)	Thermal conductivity [W/(m K)]
- UC2	2 710	18
- (U, Zr)C	3 350	30
- (U, Zr, Nb)C	3 800	50[a]
- (U, Zr, Ta)C	3 900	50[a]

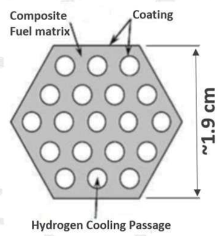
[a] 20-100 W/(m K) depending on temperatures

Operating time vs coolant exit Temperature for the full-power NERVA/Rover reactor tests



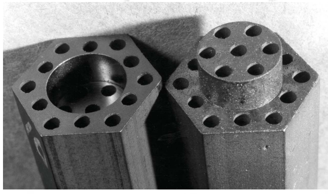


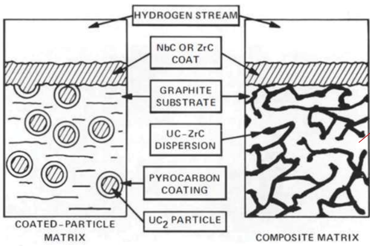
Composite fuel:
35 vol% (U, Zr)C web
15 Vol% void graphite matrix
U: 0.64 g/cm^3

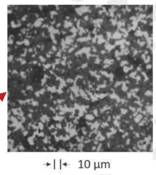


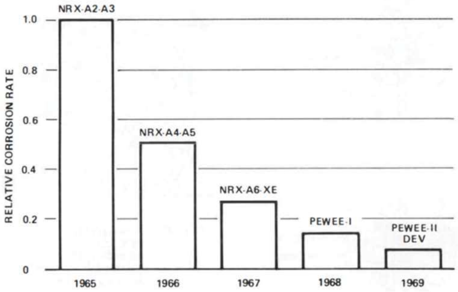
Source: Experience gained from the Space Nuclear Rocket Program (Rover), D. R. Koenig, LA-10062-H UC-33, May 1986

NERVA Fuel Corrosion Resistance

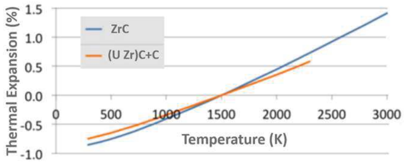


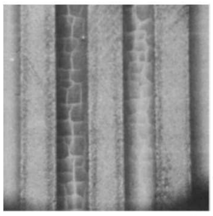




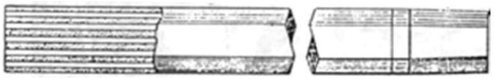


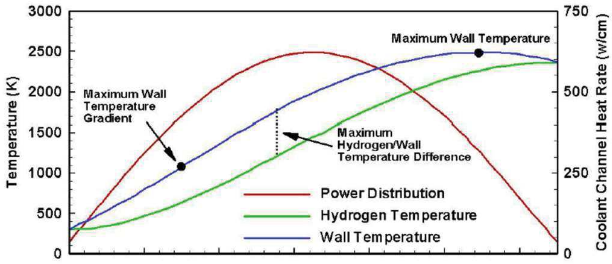
Improvements in relative corrosion rate of NERVA/Rover fuel elements

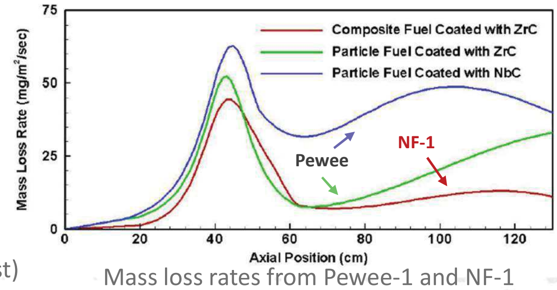




Cracks on the ZrC coating of coolant-channels (composite FE, NF-1 test)







Mass loss rates from Pewee-1 and NF-1

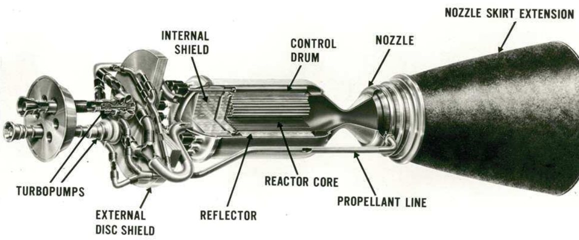
NF-1: 8-50 g mass loss per FE from ~90 mn run

Sources: Harold P. Gerrish Jr (NASA Marshall Space Flight Center), Nuclear Termal Propulsion Ground Test History, February 25, 2014;
Performance of (U, Zr)C-Graphite (Composite) and of (U,Zr)C (Carbide) Fuel Elements in the Nuclear Furnace 1 Test Reactor; LANL Report LA-5398-MS, September 1973

Achievements of the Rover/NERVA Program

Quantity	Record	Test	Year
Power	4.1 GW	Phoebus-2A	1968
Thrust	930 kN	Phoebus-2A	1968
Specific Impulse	838 s	Pewee-1	1968
Temperature (exit gas/fuel)	2 550/ 2 750 K	Pewee-1	1968
Specific Power	0.43 MW/kg	Phoebus-2A	1968
Avg. Power Density	2.34 GW/m^3	Pewee	1968
Peak Power Density	5.2 GW/m^3	Pewee	1968
Runtime	109 min	NF-1	1972
Repeatability	28 restarts	XE'	1969

Source: Experience gained from the Space Nuclear Rocket Program (Rover), D. R. Koenig, LA-10062-H UC-33, May 1986



"Demonstrated all the requirements needed for a viable lunar space transportation system as well as for human Mars exploration missions"

"Achieved a TRL ~6"

Project Rover/NERVA shut down in 1973 (Nixon): loss of interest of the public for human space flight, end of space race, growing use of low-cost unmanned robotic space probes, budget cuts due to cost of Vietnam war …

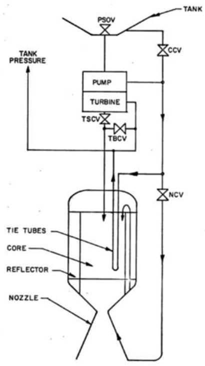
The Small Nuclear Rocket Engine Design Concept (NERVA, 1972)

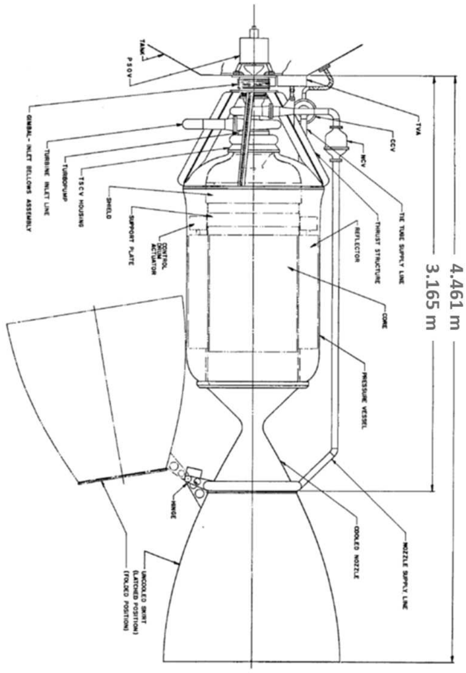
Engine System	
Thrust	**72.95 kN**
Chamber Temperature	2696 K
Chamber Pressure	3.1 MPa
Nozzle Expansion Ratio	100:1
Specific Impulse	**875 s**
Engine Diameter	0.985 m
Engine Length (approx)	4.46 m
Engine Thrust-to-Weight ratio	**3.2**

Engine Component masses (kg)	
Reactor	1901
Pressure Vessel	149
Nozzle	224
Turbomachinery & Piping	85
Gimbal	28
Engine Total	**2387**

Reactor	
Power	**367 MW**
Active Fuel Length	89 cm
Effective Core Radius	29.5 cm
Reflector Thickness	14.7 cm
Pressure Vessel Diameter	98.5 cm
Nber of Fuel Elements	564
Number of Tie Tube Elements	241
Fuel Fissile loading	0.6 g U per cm^3
Maximum Enrichment	93 (wt% U-235)
Maximum Fuel Temperature	2860 K
Margin to Fuel Melt	40 K
^{235}U mass	**59.6 kg**


Fuel / Tie Tube Element arrangement (2:1)



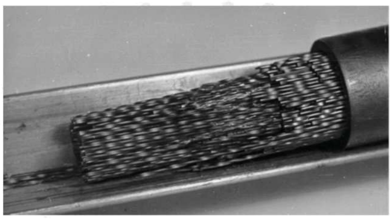


Source: Nuclear Engine Definition Study Preliminary Report, Volume 1 – Engine Description, LA-5044-MS, 1972

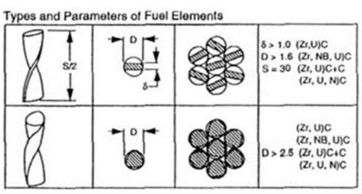
USSR NTP Program (~1960-1989)

RD-0140 (196 MW/35 kN, 910 s Isp)
Propellant: H_2 + Hexane
Core outlet T: 3000 K
Core: ϕ 0.5 x H 0.8 m^2
Engine: ϕ 1.2 x L 3.7 m^2
Engine mass ! 2 000 kg

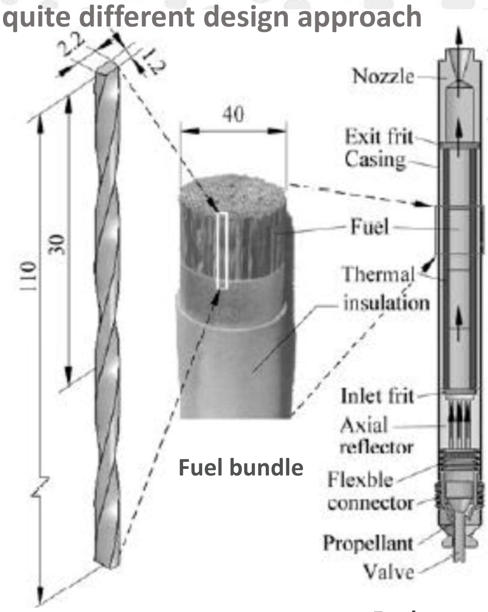
An overall effort comparable with the US NERVA program, Also carbide fuel, however a quite different design approach



Twisted ribbon fuel bundle
(fuel surface-to-volume 2.6 times higher than prismatic NERVA fuel)



Ternary carbides (U ≤ 2.5 g/cm^3)
UC-ZrC-C (< 2500K)
UC-ZrC-NbC (up to 3100K),



Fuel bundle

Twisted ribbon fuel element (mm)

Fuel assembly
(6-8 bundles stacked)


EWG-1 research reactor (ZrH matrix)

Source: Zakirov, Vadim, and Vladimir Pavshook. *Russian Nuclear Rocket Engine Design for Mars Exploration*. Rep. no. 1007-0214. N.p.: TSINGHUA SCIENCE AND TECHNOLOGY, June 2007

USSR NTP Program (~1960-1989)

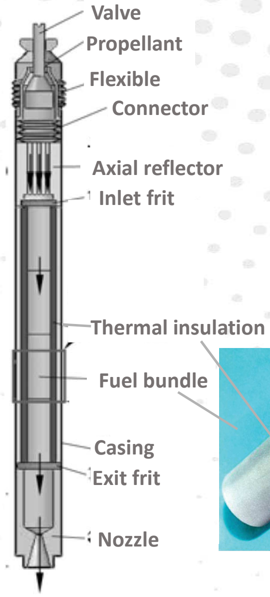
An overall effort comparable with the US NERVA program,
A quite different design approach

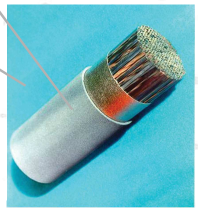
A modular heterogeneous core design

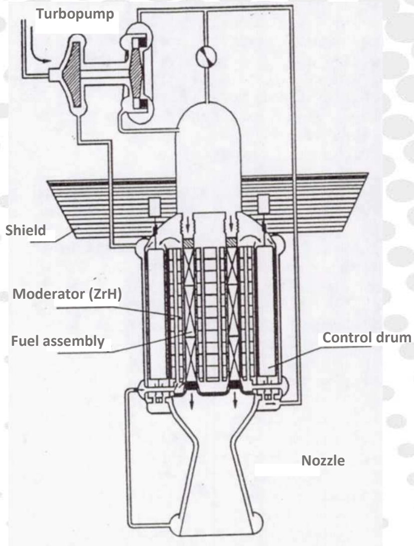
Twisted ribbon fuel:
- fuel surface-to-volume 2.6 times higher than prismatic NERVA fuel

Stacked fuel bundles:
- possibility of axial profiling / axial variation of fuel composition (UC-ZrC-C upstream, UC-ZrC-NbC downstream)

Individual fuel assemblies with high temperatures localized to fuel bundles:
- simplifies design of the rest of the core which operates at much lower temperatures (moderator, core support structures, including downstream support plate)
- enables radial and hydraulic profiling
- simplifies nuclear testing (enables H_2 irradiation loop testing of FA in research reactor: no need for whole core testing to assess nuclear performances like in NERVA)







USSR NTP Program (~1960-1989)

An overall effort comparable with the US NERVA program,
A quite different design approach

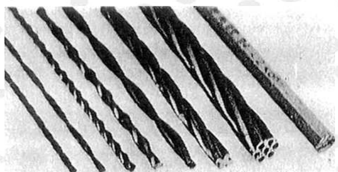
Nuclear testing summary

1550 fuel assemblies tested
Full core tests : 4 in EWG-1 reactor
2 in IRGIT Reactor
1 in RA reactor
Best performances (for different tests)
- Hydrogen exit temperature : 3100 K
- Test duration: 4000 s
- FA Power density: 3.4 GW/m^3
- Heating rate: 1000 K/s
- Number of cycles: 12
- Power cycle duration (1200 K): 6000 hrs

Test at Semipalatinsk site

Time-Temperature limits of
Non-nuclear hot H_2 Testing

Fuel type	Test temperature	Maximum test time
Uzr(CN)	2800 K (H_2 + N_2)	100 hrs
UC-ZrC-NbC	2800 K	200 hrs
	3300 K	1 hr
	3500 K	0.5 hr
UC-ZrC-TaC	3300 K	2 hrs



Nuclear test facilities (Semipalatinsk site):
- IGR reactor (5 s power pulses in hot H2 loop)
- EWG-1 reactor (230 MW, flowing H_2, multiple NTP FAs)
- IRGIT reactor (prototype NTP reactor, designed for 3000 K outlet K, tests run up to 270 MW, converted in the early 80's to the RA reactor test facility for investigating FP deposition)

Source: S.K. Bhattacharyya, An assessment of fuels for nuclear thermal propulsion, ANL/TD/TM01-22, 2001

Nuclear Fuels for Nuclear Thermal Propulsion: Beyond Carbides, Cermet Fuels

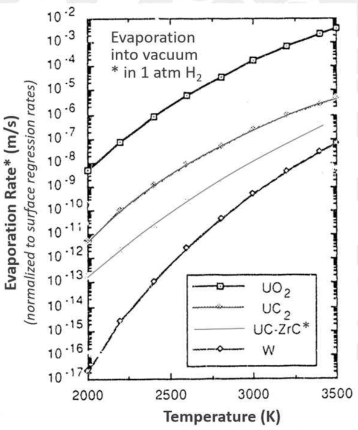
Nuclear Fuel Design Issues:

- Material evaporation,
- Melting temperature,
- Thermal conductivity,
- High temperature chemical stability,
- Corrosion/mass loss in flowing H_2,
- Fission product release,
- Uranium density,
- Reactor neutron spectrum,
- Fabrication,
- Fuel swelling,
- Thermal expansion mismatch with coating/cladding
- Thermal shock resistance,
- Mass density,
- ….,

Properties of nuclear thermal rocket fuel options

Fuel Option Matrix Material	Density (g/cm3)	U Density (g/cm3)	Melting point (K)	Surface Vaporization Rates at 2800 K (mil/hr)	Thermal conductivity [W/(m K)]
CERMET					
- UO_2 particles	10.9	10.9	3 120	6 000	3.5
- W matrix	19.6		3 695	< 0.01	170-90[a]
- W-UO_2 Cermet[*]	14.4	3.4			66-33[a]
- Mo matrix	10.2		2 890	>> 10	140-85[a]
- Mo-UO_2 Cermet	10.5				
CARBIDES					
- C matrix	2.3		3 915	10	90-40[a]
- ZrC	6.6		3 805	>> 10	20-40[a]
- TaC	14.6		4 150	0.1	
- UC_2	11.6	10.5	2 710	10	18
- (U, Zr)C[£]	5.7	0.3	3 350	2	8
- (U, Zr)C, C[$]	3.6	0.6	3 350	2	90-40[a]
- (U, Zr, Nb)C			3 800		100-20[a]
- (U, Zr, Ta)C			3 900		100-20[a]

[a] depending on temperatures Room T - High T
[*] W-UO2 Cermet: 60 v% (10m% GdO1.5-stabilized UO2)
[£] (U, Zr)C as tested in NF-1
[$] (U, Zr)C, C as tested in NF-1



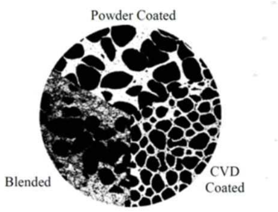
W: the only known fully stable material in flowing H_2 at T> 2500K ⇨ W-based Cermet fuel

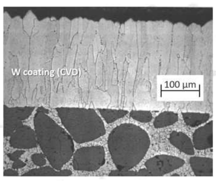
Sources incl. L. B. Lundberg & R.R. Hobbins, Nuclear Fuels For Very High Temperature Applications, 27th IECEC (1992) EGG-M--92067

W-UO_2 Cermet Fuel

1962-1968 developments GE (710 program) & ANL Nuclear Rocket Program

Material & FE development + critical experiments + NTP conceptual designs

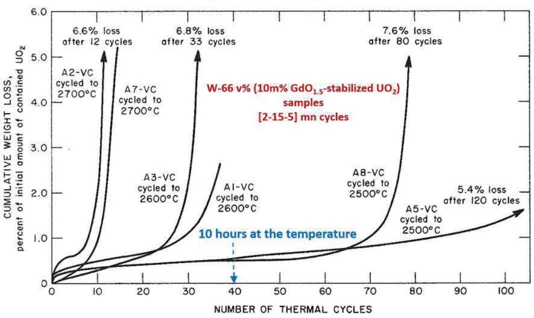


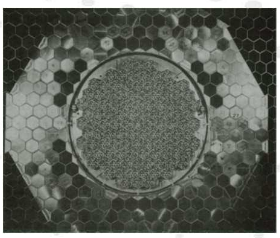


After 1 h exposure to H_2 at 2610 K

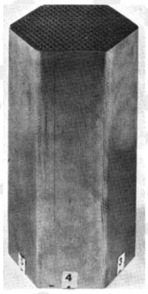
⇨ W-UO_2 Cermet: 60 v% (10m% $GdO_{1.5}$-stabilized UO_2)

Promise for high fission product retention

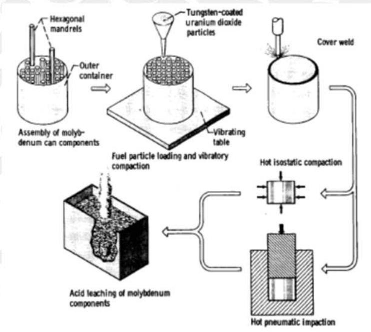


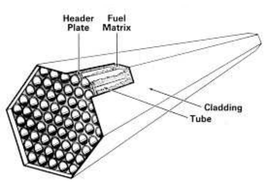


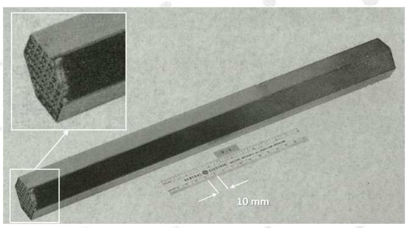
710 Critical Mock-Up



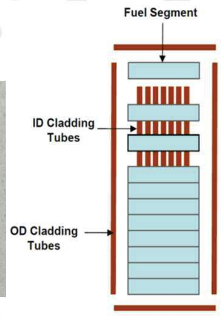
2000 MW Engine FE
331 coolant holes





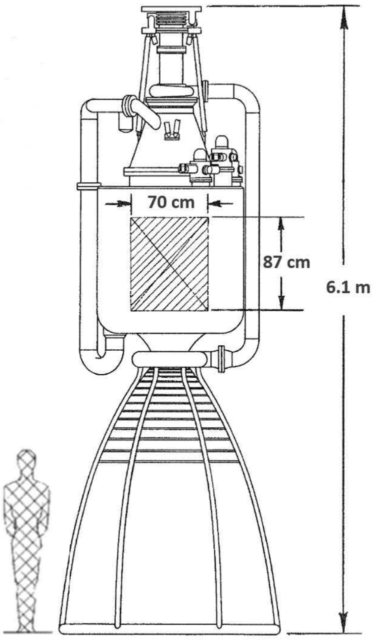


710 Program FE 37 coolant holes



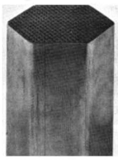
Sources incl. ANL Nuclear Rocket Program Quarterly Progress Report Fourth Quarter 1965 ANL-7150; 710 High-Temperature Gas Reactor Program Summary Report, GEMP-600, 1968

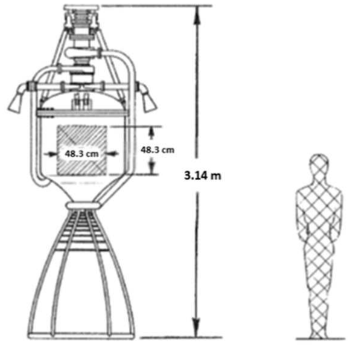
ANL Cermet-Fuel-based 2000 MW & 200 MW Engines Conceptual Designs (1965)



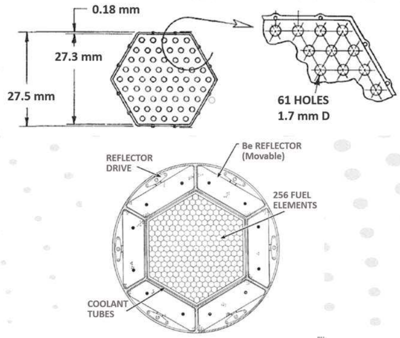
2000 MW/445 kN Engine

ENGINE		
Thrust	445	kN
Chamber H_2 Pressure	3.6	Mpa
Chamber H_2 Temperatur	2500	K
Isp	832	s
Thrust/Weight	~5	
Operating time	up to 10	hours
Restart capabilities	up to 40	
REACTOR		
Neutron spectrum	Fast	
Power	2000	MW
FUEL		
Fuel composition	$W-60\%UO_2-6\%Gd_2O_3$	
^{235}U enrichment	93%	
Fuel clad	W-25 Re	
FUEL ELEMENT		
Number	163	
Active length	87	cm
Across flats	4.75	cm
Coolant holes number	331	
diameter	1.7	mm
Peak fuel temperature	2728	K





200 MW/44.5 kN Engine



Sources incl. ANL Nuclear Rocket Program Quarterly Progress Report Fourth Quarter 1965 ANL-7150

Cermet-Fuel-Based Engine Concept Revisited: the XNR2000 (Pratt & Whitney, 1993)

A 111 kN Rocket Engine

Two-Path ''Folded H_2 Flow'' configuration (outer, then inner core)

Cermet Fuel drawing from GE 710 and ANL program results

Inner core: $W-UO_2$ (60 vol% UO_2, 93% ^{235}U, 6 mol% Gd_2O_3)

Outer core: $Mo-UO_2$ (it is ~50% lighter!)

Fast spectrum (criticality-limited) Core: 270 kg ^{235}U

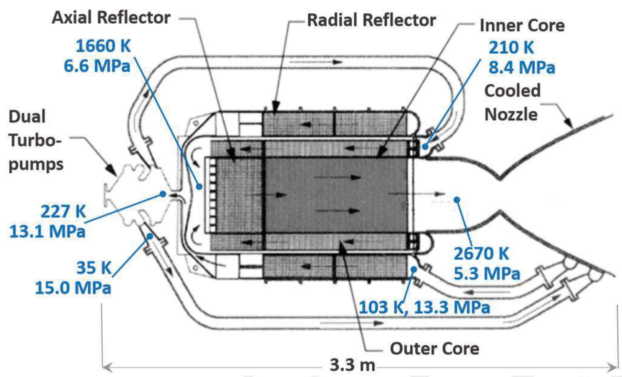
P: 510 MWth; Active core Vol: 66 dm^3; Fuel power density: 9.4 GW/m^3

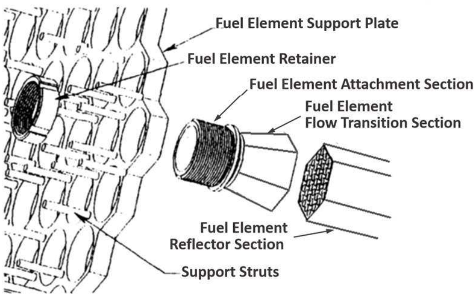
Max Outer/Inner Core Fuel T: 2010 / 2880 K,

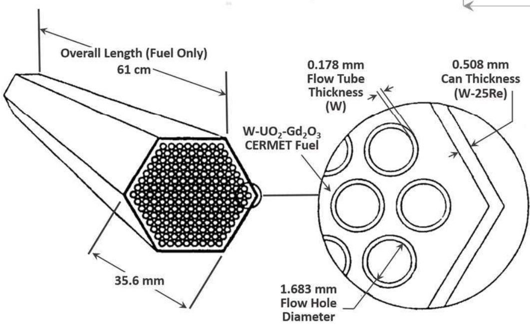
Chamber T: 2670 K, Isp: 900 s

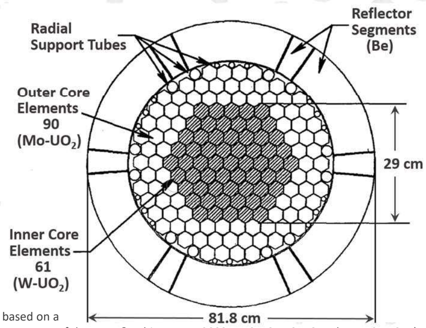
Engine Mass: 2500 kg (incl. 115 kg internal shield)

Thrust to Weight ratio: 5.3







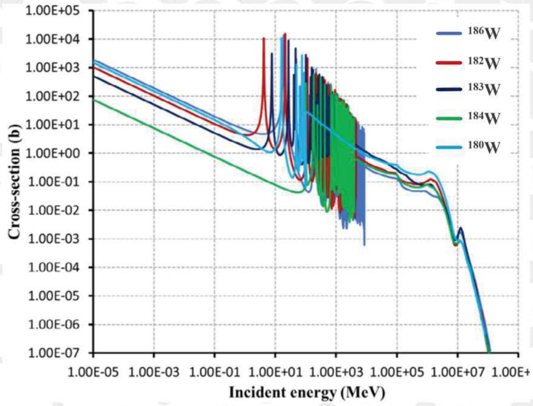


Sources incl. Stephen D. Peery et al., XNR2000 -- A Near Term Nuclear Thermal Rocket Concept, AIP Conference Proceedings 271, 1743 (1993); Randy C. Parsley, Advanced Propulsion Engine Assessment based on a Cermet Reactor, Nuclear Propulsion Technical Interchange Meeting, NASA Lewis Research Center, October 20-23, 1992 (NASA-CP- 10116, Vol. I, pp 150-216); K. O. Westerman et al., Babcock & Wilcox Assessment of the Pratt & Whitney XNR2000, NASA-CP-10116, Vol. I, pp 217-245)

CERMET vs. Graphite-Based Nuclear Fuels for Nuclear Thermal Propulsion Engines

''Historic'' CERMET Fuel (W-UO_2) Rocket Reactor Concepts:

- ☺ Better fission products retention
- ☺ Long operating life with multiple restarts and temperature cycling (thermal shock resistance, ductility, strength)
- Fast spectrum cores:
  - ☺ Simpler design (no moderator to cool, simpler core support)
  - 😐 Much more compact core (than with thermal spectrum)
  - ☺ Negligible Xenon reactivity effect(*), no hydrogen reactivity feedback (negligible reactivity worth, important for startup with cold H2)
  - ☺ Intrinsic ''neutronics spectral shift effect'' ensures reactor subcriticality in the event of a water immersion accident, idem compaction



Absorption XS of stable W isotopes

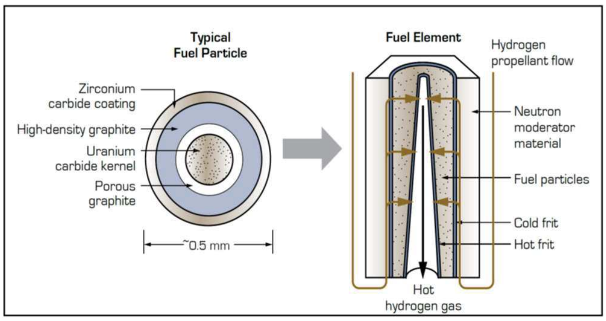
- ☹ Inherently higher ^{235}U mass (x ~3) than thermal spectrum reactors
- 😐 Much higher fuel density (offsets compactness)
- ☹ Lack of nuclear power reactor tests
- ☹ Relies on HEU (criticality-limited): a LEU CERMET fast spectrum core would be prohibitively large/massive

A (moderated) Cermet-fueled LEU core would require using W 95 w% enriched in ^{184}W, enriched Re due to the large absorption XS of natural W (30% ^{184}W) and rhenium in a thermal spectrum. UO_2 stabilizer Gd_2O_3 will have to be replaced by ThO_2

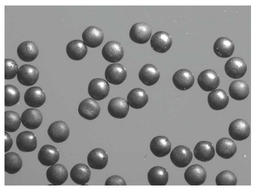
(*) *see back-up slide*

The CEA-CNES MAPS Study Program (1994-1997)

- **Goal: develop the knowledge required to decide whether to initiate an R&D program on NTP**
- **(paper) Study included**
  - Assessment of **engine performances** (mean Isp, mass, operating constraintsts, reliability, recurring cost, …)
  - **Safety evaluations**
  - **Proposal of an R&D Program**
- **Design Strategy**
  - **Rely AFAP on off the shelves or near term technologies**
  - While offering prospects for performance improvements with more advanced technlogies
- **Mission as study framework:**
  - 5 round-trip cargo missions from LEO to moon orbit
  - Launched with ARIANE V
  - Payload with its H_2 for a one-way journey launched separately



⇨ **Choice of a particle-bed reactor design concept* with Beryllium as moderator/reflector**

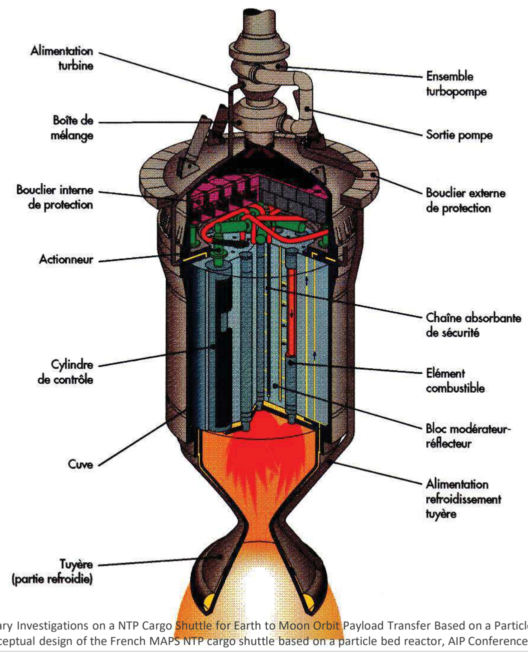


Sources: Raepsaet, X., E. Proust, et al. (1995), "Preliminary Investigations on a NTP Cargo Shuttle for Earth to Moon Orbit Payload Transfer Based on a Particle Bed Reactor," AIP Conference Proceedings No. 324, 1: 401-408; R. Lenain et al., Conceptual design of the French MAPS NTP cargo shuttle based on a particle bed reactor, AIP Conference Proceedings 361, 1169 (1996)

* *See back-up slides*

The CEA-CNES MAPS study program 1994-1996

MAPS
Moteur Atomique de Propulsion Spatiale



Mission phases

	Δt	ΔV
start-up from earth orbit	811 s	3 100 m/s
to the moon	90 h	-
injection into moon orbit	211 s	1 100 m/s
	-	-
start-up from moon orbit	64 s	1 100 m/s
to earth	90h	-
injection into earth orbit	132 s	3 100 m.s

Engine operating conditions

Thermal power / Thrust (flow rate)	300 MW / 72 kN (9 kg/s)
Propellant	H_2
Cycle	Expander
Chamber pressure/ temperature	4.3 MPa / 2 200 K
Mach at core outlet	0.7
Turbopump power / rotational speed	1.1 MW / 51 600 rpm
Nominal / average Isp (vacuum)	816 s / 786 s
Nozzle expansion	200
Engine mass	2 390 kg
Height / Diameter	3.98 m / 0.94 m
Weight to thrust ratio	30 N/kg

Reactor design point

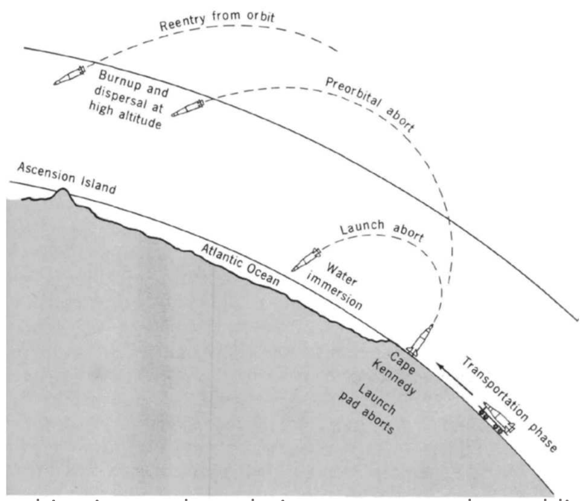
Fuel/Moderator/Reflector/Absorber	UC_2/Be/Be/B_4C
Number of fuel elements	19
- U mass (93% U-235)	19.2 kg
- Particle bed volume	24.7 litres
Active core diameter / height	60 / 70 cm
Overall diameter	94 cm
Mass of the core	967 kg
of the shielding	465 kg
Particle bed power density	12.1 MW / bed liter
Core power density	1.5 MW / core liter
Radial / axial power peaking factor	1.24 / 1.31
H_2 propellant	
- core inlet/outlet Temperature	150 / 2 200 K
- core inlet/ chamber Pressure	5.0 / 4.2 Mpa

Sources: Raepsaet, X., E. Proust, et al. (1995), "Preliminary Investigations on a NTP Cargo Shuttle for Earth to Moon Orbit Payload Transfer Based on a Particle Bed Reactor," AIP Conference Proceedings No. 324, 1: 401-408; R. Lenain et al., Conceptual design of the French MAPS NTP cargo shuttle based on a particle bed reactor, AIP Conference Proceedings 361, 1169 (1996)

Space Nuclear Reactors: Safety Principles

List of **internationally agreed-upon principles**
(UN Committee on the Peaceful Uses of Outer Space, 1992)
but no specified safety criteria or regulations so far

Source: Principles relevant to the use of Nuclear Power Sources in Outer Space. Report of the Committee on the Peaceful Uses of Outer Space, General Assembly Official Records Forty-seventh Session. Supplement No. 20(A/47/4/20)



Safety objectives and regulations are currently established on the basis of national political/legal rules: USA, Russia (Europe?)

"Space Nuclear Safety Culture" inspired from the experience learned from past "nuclear launches"

Use **only fresh Uranium as fuel**$ (reactor launched free of fission products)**; Use of Plutonium precluded**

Reactor designed to **prevent accidental criticality** whatever the emergency situation (in case of reactor **compaction** and/or **flooding** upon impact following **launch abort**, …)

First criticality and operation started **only once prescribed "sufficiently high orbit"* reached ("nuclear safe" orbit,** allowing for sufficient FP radioactive decay before reentry)

Minimize fission product release (principle ALARA)

Reactor designed **either survive accidental reentry or to disintegrate upon reentry** and disperse its residual radioactivity in the upper atmosphere (soviet strategy adopted in the latest RORSATs, Cf. 1983: Kosmos 1402)

$ 1 kg 235U ~ 2 10^{-3} Ci

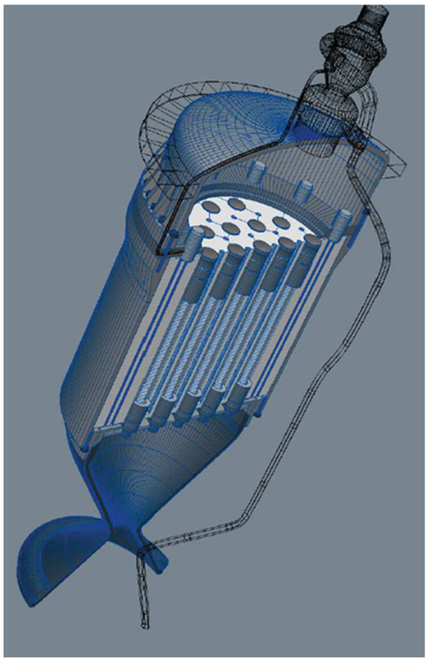
** See back-up slide*

The CEA-CNES MAPS study program 1994-1997

- Engine start-up and shut-down transients have little impact on engine performances : ~5% degradation of I_{sp}

Safety aspects

- Normal operation
  - Low fluence (3 10^{19} n/cm^2) / low burn-up (< 0,2% FIFA) / short duration of high temperature operation (< 6000 s)
  - Fission products inventory < 1 MCi, down to <1 kCi after 1 month
  - Even if 100% release of FP during ΔV for leaving LEO (over 7000 km), small radiation impact compared to natural space radiation background
- Accidents during operation
  - Low decay heat, <1 m^3 high pressure H_2 tanks sufficient (loss of turbopump)
  - Passive operation on main H_2 tank pressurization provides some residual manoeuvring capability (> 4 kN thrust at 490 s Isp) to avoind re entry
- Launch abort / Re-entry issue
  - Launch abort: subcriticallity ensured in case of flooding (B_4C chains, Gd wires), structure likely ideal to prevent criticality-leading reconfiguration
  - Operation beyond a 600 km circular orbit (11 y lifetime = < 100 Ci re entry)



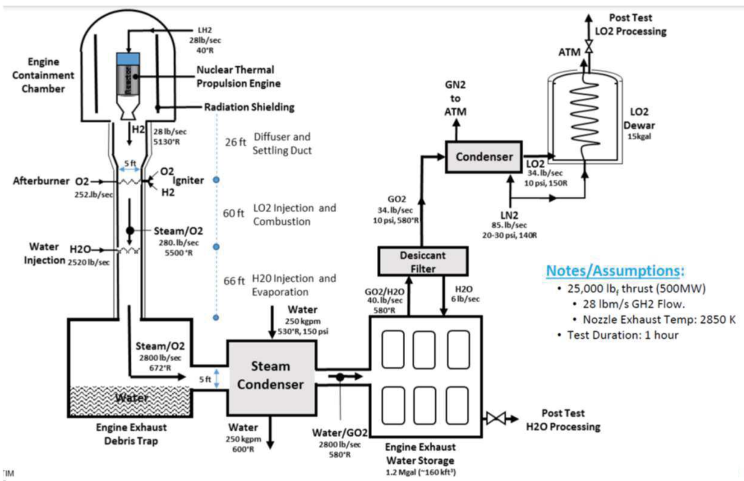
Sources: Raepsaet, X., E. Proust, et al. (1995), "Preliminary Investigations on a NTP Cargo Shuttle for Earth to Moon Orbit Payload Transfer Based on a Particle Bed Reactor," AIP Conference Proceedings No. 324, 1: 401-408;
R. Lenain et al., Conceptual design of the French MAPS NTP cargo shuttle based on a particle bed reactor, AIP Conference Proceedings 361, 1169 (1996)

The CEA-CNES MAPS Study Program 1994-1997

- Recurring cost estimate: 20 M US$\$_{1996}$

Development and ground testing approach:

- **Small thrust engines** (cluster them if need be)
  - minimize ground demonstration costs (reactor/engine tests require *exhaust capture*!) … flight demonstration
  - maximize applications (moderate and high thrusts)
- **Design aiming at minimizing RD&D costs**
  - Scalable design
  - Modular design enabling nuclear testing of individual fuel assemblies
  - Design enabling pertinent *non-nuclear* high temperature fuel and fuel element testing in stagnant and flowing H_2



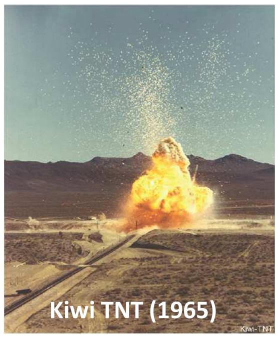
Engine Exhaust Capture Process Scheme (NASA)

Source: Low Enriched Uranium (LEU) Nuclear Thermal Propulsion: System Overview and Ground Test Strategy, David Coote (NASA/SCC), JANNAF Programmatic and Industrial Base Testing and Evaluation Technical Interchange Meeting, Nov 8, 2017

The Challenges of Nowadays Testing Nuclear Thermal Propulsion Engines

Ground testing NTR engines

⇨ To build a facility that captures engine exhaust and **guarantees containment for all credible core dispersion scenarii**



Kiwi TNT (1965)

Qualifying this facility might be harder than qualifying the NTR itself: NTR safety concerns, compounded by a high volume of combustible hydrogen, are orders-of-magnitude beyond any reactor safety approval that has been attempted in decades

NTRs combine very small temperature margins and very high power densities
Average adiabatic heat-up rates: ~500 to 2000 K/s (25 to 100 times higher than a typical PWR)
⇨If cooling is lost during powered operation, an NTR core would melt within seconds
Decay power also a significant concern; one hour after shutdown the adiabatic heat-up rate can still be >1 K/s

Regulators focused on the confidence, or lack thereof, in **system dynamics and the potential to melt fuel** (Cf. DOE regulators: Duff/Krusty)
⇨ **A circular dilemma for NTR systems**:

- complex, unknown reactor dynamics and control issues that only be solved via nuclear system testing (temperature reactivity feedbacks: thermal expansion & XSs; H_2 reactivity worth; peaking factors; coupling between H_2 pressure/flow rate and power through turbopump; requirement for very quickly reaching full power after the onset of H_2 flow)
- without in-hand solutions to these issues, there may be no ability to get approval for and successfully execute the tests

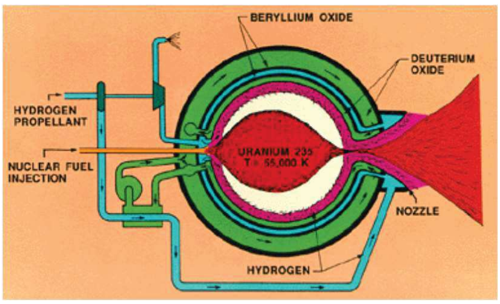
⇨ **Make the system as simple as possible in terms of system dynamics and controllability**

⇨ In-space testing instead??

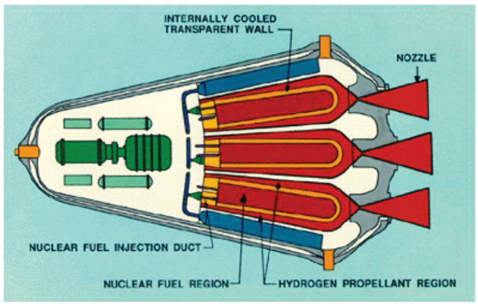
Source: Considerations Inspired from David I. Poston, Nuclear Testing and Safety Comparison of Nuclear Thermal Rocket Concepts, ANS NETS 2018; Roy Reider, KIWI-TNT ''explosion'', LANL Report LA-3351 UC-30, 1965

Nuclear Thermal Propulsion: Beyond Solid Core Concepts Towards Higher Isp

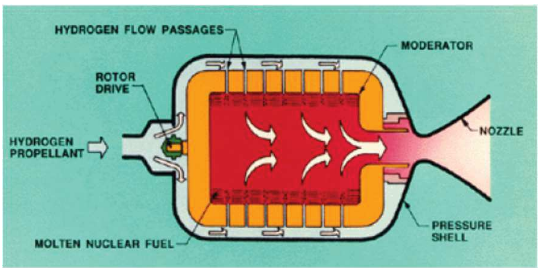
Other Nuclear Rocket Engine Concepts studied in the 60's (*see ''bonus'' slides)



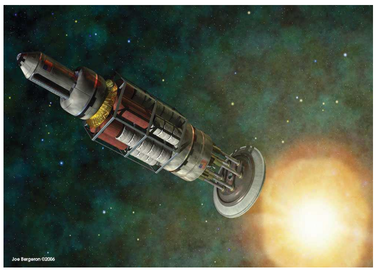
Open-Cycle Gas Core Nuclear Rocket



Closed-Cycle Gas Core Nuclear Rocket



Liquid Core Nuclear Rocket



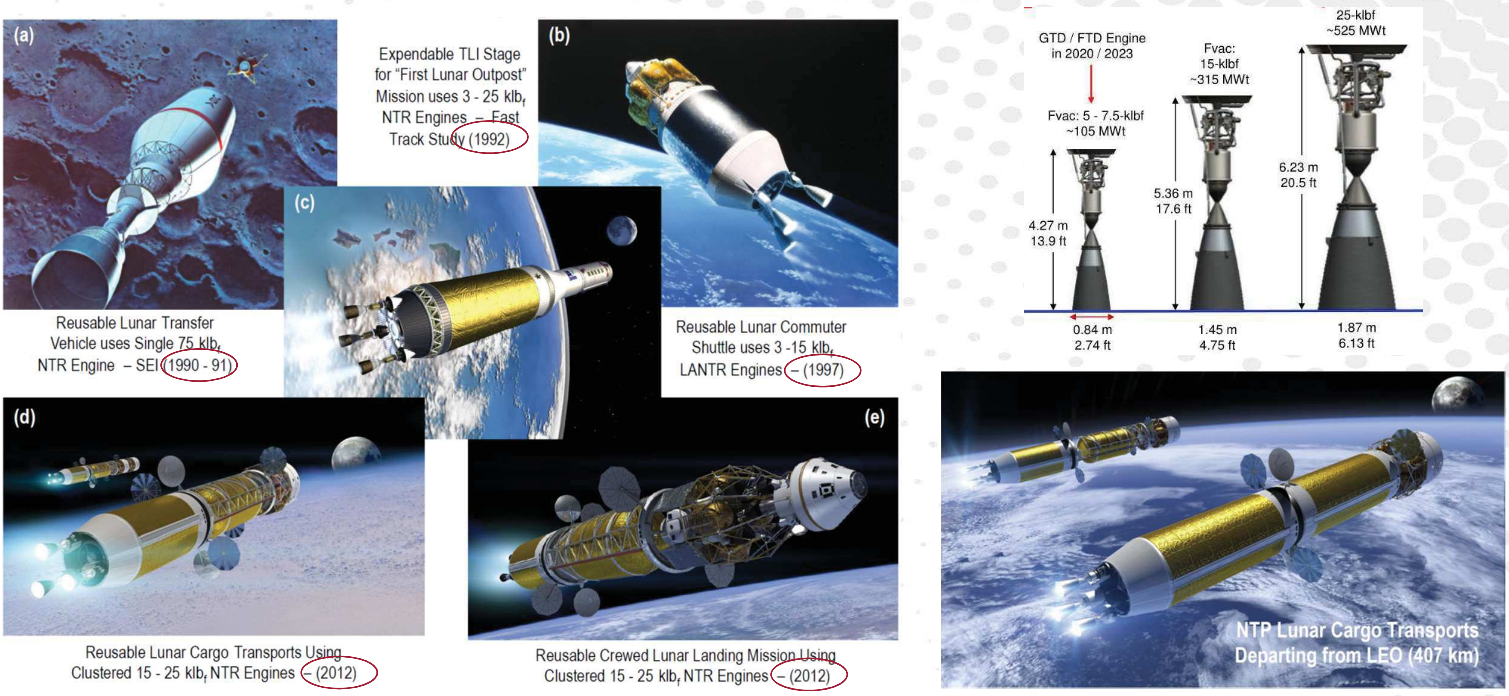
Nuclear Pulse Rockets

If NTP is so good, why hasn't it happened? (a few quotes)

A chicken and egg syndrome?

- ''It takes longer to develop an NTP system than to develop a space mission. Project managers cannot include NTP systems in mission planning until system has been developed and tested''
- "If only reactors could be developed, users would emerge to claim them''
- "NTP ready for flight tests and yet no users have come forward in ensuing decades''
- "NASA was dominated by people who built there life around chemical rockets; they didn't want to see [nukes] come in because they feared it"

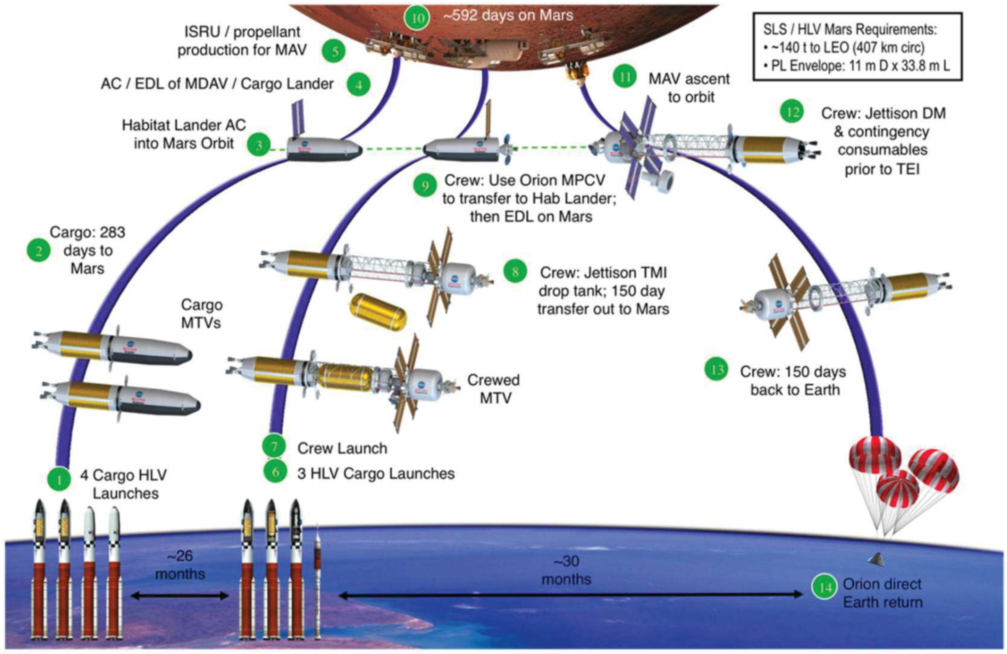
NTP: having being waiting to go (back) to the moon … and beyond, to Mars?



Source: The Nuclear Thermal Propulsion Stage (NTPS): A Key Space Asset for Human Exploration and Commercial Missions to the Moon, *Stanley K. Borowski et al.*, , NASA/TM—2014-218105, AIAA–2013–5465

Until 2010 Obama's Space Policy Directive: By the mid-2030s, send humans to orbit Mars

Long-stay manned MARS 2033 mission: NTP strategy



Source: S. K. Borowski, D. R. McCurdy and T. W. Packard, "Nuclear Thermal Propulsion (NTP): A proven growth technology for human NEO/Mars exploration missions," 2012 IEEE Aerospace Conference, Big Sky, MT, 2012, doi: 10.1109/AERO.2012.6187301.

"Pewee-like" 25 klbf (~110 kN) thrust NTP engine considered for manned Mars mission

Performance Characteristic	7,420-lbf Option	SNRE Baseline	Axial Growth Option Nominal	Axial Growth Option Enhanced	Radial Growth Option Nominal	Radial Growth Option Enhanced
Engine System						
Thrust (klb_f)	7.42	16.4	25.1	25.1	25.1	25.1
Chamber Inlet Temperature (K)	2736	2695	2790	2940	2731	2807
Chamber Pressure (psia)	1000	450	1000	1000	1000	1000
Nozzle Expansion Ratio(NAR)	300:1	100:1	300:1	300:1	300:1	300:1
Specific Impulse (s)	894	875	906	941	894	913
Engine Thrust-to-Weight	1.87	2.92	3.50	3.50	3.60	3.60
Reactor						
Active Fuel Length (cm)	89.0	89.0	132.0	132.0	89.0	89.0
Effective Core Radius (cm)	14.7	29.5	29.5	29.5	35.2	35.2
Engine Radius (cm)	43.9	49.3	49.3	49.3	55.0	55.0
Element Fuel/Tie Tube Pattern Type	Dense	SNRE	SNRE	SNRE	Sparse	Sparse
Number of Fuel Elements	260	564	564	564	864	864
Number of Tie Tube Elements	251	241	241	241	283	283
Fuel Fissile Loading (g U per cm^3)	0.60	0.60	0.25	0.25	0.45	0.45
Maximum Enrichment (wt% U-235)	93	93	93	93	93	93
Maximum Fuel Temperature (K)	2860	2860	2860	3010	2860	2930
Margin to Fuel Melt (K)	40	40	190	40	110	40
U-235 Mass (kg)	27.5	59.6	36.8	36.8	68.5	68.5

Source: *S. K. Borowski et al.*, Nuclear Thermal Propulsion (NTP): A Proven, Growth Technology for "Fast Transit" Human Missions to Mars, NASA/TM—2014-218104, AIAA 2013–5354, October 2014

Source: B. G. Schnitzler, Small Reactor Designs Suitable for Direct Nuclear Thermal Propulsion: Interim Report, INL/EXT-12-24776, January 2012

US space policy

2010 Obama's Space Policy Directive(SPD): ''By the mid-2030s, send humans to orbit Mars …''
NASA analysis reconfirms benefits of nuclear thermal propulsion
2017 Trump's SPD 1: emphasis switched to a human exploration program that would return astronauts to the surface of the moon by 2028 and lay the groundwork for a sustained presence, followed by missions to Mars
2019: timeline for the return accelerated to 2024
2020: SPD6 ''The ability to use space nuclear power and propulsion systems safely, securely, and sustainably is vital to maintaining and advancing United States dominance and strategic leadership in space''

A strong push in the US to switch from ''historic'' HEU use to LEU(*) (<20% ^{235}U)

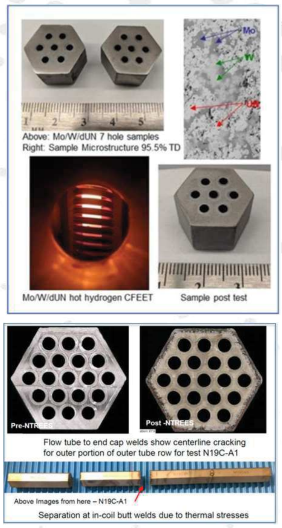
- **Non-proliferation policy motivation**: US commitment to minimize, and *to the extent possible* eliminate, the use of HEU in civilian nuclear applications
- Prospects from cheaper costs through **a commercial development effort** (Cf. SpaceX) for which LEU is probably the only option
- Additionally,
  - Avoid the costs of managing the security risks associated with HEU
  - Avoid the political risks of project cancellation due to controversy over the use of nuclear weapon-grade fuel and facilitate international partnership for space missions with non nuclear weapon states

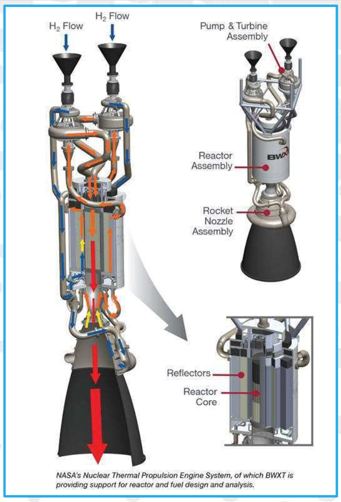
⇨ **A NASA effort** to address the key challenges related to determining the **technical feasibility and affordability of an LEU-based NTP engine**

(*) *see back-up slides*

A NASA effort to address the key challenges related to determining the **technical feasibility and affordability of an LEU-based NTP engine**

- Maturing technologies associated with fuel production, fuel element manufacturing and testing
- Developing reactor and engine conceptual designs
- Performing detailed cost analysis for developing an NTP flight system
- Pursuing multiple study paths to evaluate the cost/benefits and route to execute a NTP Flight Demonstration Project





The main focus is on the LEU Cermet fuel option (potential for better fission products retention compared with NERVA-type fuels)

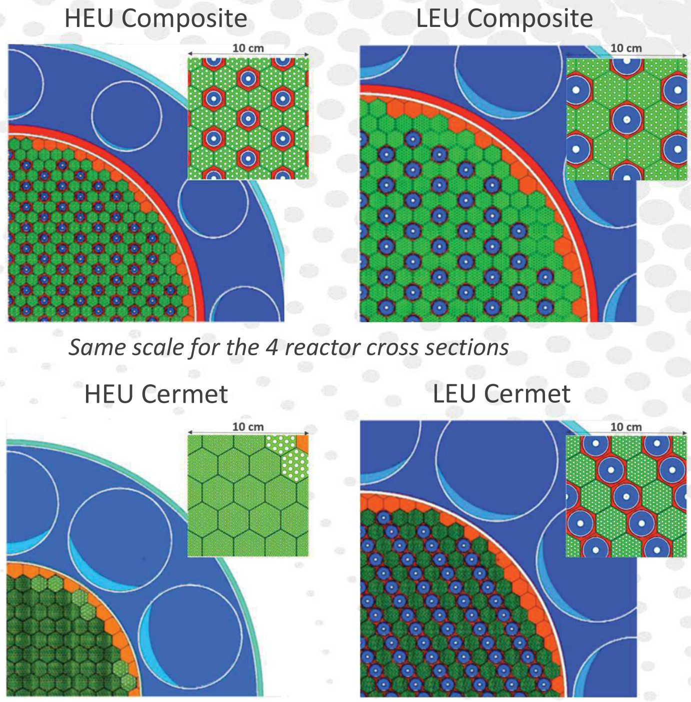
Source: Richard Ballard, Nuclear Thermal Propulsion Update, NASA Space Technology Mission Directorate, October 29, 2019

Impact of Switching from LEU to HEU for NTP: one out of many assessments (1/2)

A concept comparison study by D. Poston (LANL)

Parameter \ Concept	HEU Composite	LEU Composite	HEU Cermet	LEU Cermet
Fuel composition	(9w% U, 53.9 w%Zr) 37.1 w%C-C		W-60vol%(UO_2-6vol% Gd_2O_3)	enriched W, Gd, Mo
FE outer coating/ clad	ZrC	ZrC	W25Re	W5Re
Fuel holes coating/liner	ZrC	ZrC	W	W
Tie Tube Structure	Inc718	SS316	n/a	Mo
Peak Fuel T (K)	2800	2800	2800	2800
Mix-mean outlet T (K)	2652	2630	2698	2669
Hot-channel factor(*)	1.27	1.38	1.19	1.68
3D fuel power peaking f.	1.94	2.12	1.61	2.76
Reactor Power (MW)	540.2	538.6	548	543.1
Nber of FE/TT	564/241	396/151	211	282/247
Core Diameter (cm)	59	72.1	43	64.9
Fuel Length (cm)	121	93	64	84
Vessel Diameter (cm)	90	116.5	82	110.4
Vessel Length (cm)	206	179	164	177
U235 Mass (kg)	**43.3**	**12.9**	**233.4**	**71.3**
Reactor+Shield Mass (kg)	**2207**	**3106**	**1802**	**3544**
Thrust (kN)	111	111	111	111
Full thrust Isp at BOL (s)	885	880.8	895.2	888.8
Decay cooling Isp adj. (s)	-6.7	-8.8	-1	-8.9
Peaking change Isp adj. (s)	-5	-10	-0.1	-2.5
Estimated average Isp (s)	873.3	862	894.1	877.4
Thrust to ractor weight	**5.1**	**3.7**	**6.3**	**3.2**

(*) *assumed managed by orificing each fuel element and each individual flow channel for LEU Cermet*



Source: David I. Poston, Design Comparison of Nuclear Thermal Rocket Concepts, ANS NETS 2018

Impact of Switching from LEU to HEU for NTP: one out of many assessments (2/2)

A concept comparison study by D. Poston (LANL)

Reactor dynamics & control:

Parameter \ Concept	HEU Composite	LEU Composite	HEU Cermet	LEU Cermet
Net burnup reactivity (Δk, pcm)	-800	-1500	-100	-200
Net hydrogen worth (Δk, pcm)	3100	800	0	900
Temperature defect (Δk, pcm)	-1600	-6400	-1200	-3800
-Temp defect via expansion	-600	-500	-600	-300
-Temp defect via XSs	-1000	-5900	-600	-3500
Overall power peak/average	1.94	2.12	1.61	2.76

Creates a positive power coefficient

~- 10$ (compensation)

Change of peaking factors over lifetime as control drums are moved to compensate
⇨ may prevent use of orificing

limits heat-up rate during start-up (risk of fuel melting)

Launch accident safety: core flooding

NTRs = large empty (coolant) volume fraction
+ lesser reflector/drums worth (large diam. Cores)
+ difficulty of integrating safety rods ⇨ Gd wires

Parameter \ Concept	HEU Composite	LEU Composite	HEU Cermet	LEU Cermet
Flooded Keff (drums in)	1.4233	1.1587	0.9894	1.1291
Flooded Keff (with Gd wires)	0.9769	0.9739	n/a	0.9686
Gd wires				
- Number of wires	3876	1332	n/a	2684
- Wire diameter (mm)	1.13	1.04	n/a	0.092
- Area of perfect bundle (cm^2)	42.9	12.5	n/a	19.8
- Area of throat (cm^2)	92.5	107.9	n/a	98
- Pull/tug angle (degrees)	19.8	13.4	n/a	15.6

Achieved owing to spectral shift (W, Mo, Gd) + restricted coolant area (but higher P, ΔP)

Testability:

make the system as simple as possible in terms of system dynamics and controllability, so as to reduce challenges in the licensability of ground test facility and so that testing can be closer to a demonstration than an actual test

Source: Considerations Inspired from David I. Poston, Nuclear Testing and Safety Comparison of Nuclear Thermal Rocket Concepts, ANS NETS 2018

Main Take-Away Points (1/2)

- **NTP is a proven technology**
  - 20 reactors designed, built and tested in the Rover/NERVA program
  - In less than 5 years, four different thrust engines tested (250, 330, 1100, 55 kN) using a common fuel element design
  - NERVA XE' (as close as possible to a flight engine) ground tested in a simulated space vacuum: demonstrated 250 kN thrust, 710 s Isp, restart capability (24) and endurance (28 minutes at full power)
  - TRL~6 achieved
  - Demonstrated all the requirements needed for a viable lunar space transportation system as well as for human Mars exploration missions
- **NTP consistently identified over the last 30 years as ''preferred propulsion option'' for human Mars Missions because of better system performances than other in-space transportation alternatives**
  - Due to NTP's combination of **high thrust** (~100 kN/engine) and **high Isp** (~900 s)
  - Chemical systems have **high thrust** (~100 kN/engine) but **low Isp** (~460 s)
  - Solar Electric Propulsion systems have **very hig Isp** (~3000 s) but very (very) **low thrust** (~5-12 kN/stage)

Main Take-Away Points (2/2)

- **The robustness**/insensitivity to required mission energy (the combination of payload mass and ΔV) **offered by NTP can be used to provide flexible mission planning by trading objectives including:**
  - Enabling faster trip time for crew
  - More payload
  - Fewer super heavy-lift launch vehicle launches (the launch mass –thus cost- savings over ''all chemical'' or ''chemical+ aerobrake'' for one human mission alone can pay for NTP development/qualification effort)
  - Enabling off-nominal mission opportunities and wider injection windows
  - Enabling crew mission abort options not available from other architectures

NTP is a safe, affordable "game changing" technology for space propulsion that enables faster trip times and safeguards astronaut health

- **The biggest challenge facing NTR development: the ability to ultimately perform a successful rocket test**
  - Fuel development, safeguards and launch safety are all major challenges …
  - … but the ability to create the infrastructure (money, facilities, people) to test, get the approvals to test, and design/engineer a system that actually works is the biggest hurdle.

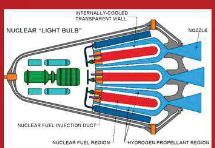
Thank you for your attention!

Any question?

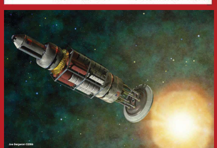
Last update: January 2021

Author: Eric PROUST email: firstname.lastname@cea.fr

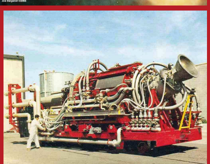
Bonus



#1 "Advanced" Nuclear Thermal Propulsion Concepts

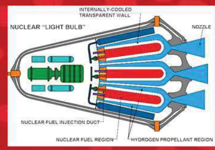


#2 Nuclear Pulse Propulsion Concepts



#3 Air-Breathing Nuclear Thermal Propulsion

Bonus # 1



cea

DE LA RECHERCHE À L'INDUSTRIE

"Advanced" Nuclear Thermal Propulsion Concepts

Commissariat à l'énergie atomique et aux énergies alternatives - www.cea.fr

---

cea

A foreword, by USN Admiral Rickover ("the father of the Nuclear Navy"), June 1953

***An academic reactor or reactor plant** almost always has the following basic characteristics:*

(1) It is simple
(2) It is small
(3) It is cheap
(4) It is light
(5) It can be built very quickly
(6) It is very flexible in purpose ('omnibus reactor')
(7) Very little development is required. It will use mostly off-the-shelf components
(8) The reactor is in the study phase. It is not being built now

*On the other hand, **a practical reactor plant** can be distinguished by the following characteristics:*

(1) It is being built now
(2) It is behind schedule
(3) It is requiring an immense amount of development on apparently trivial items. Corrosion, in particular, is a problem
(4) It is very expensive
(5) It takes a long time to build because of the engineering development problems
(6) It is large
(7) It is heavy
(8) It is complicated

The tools of the academic-reactor designer are a piece of paper and a pencil with an eraser. If a mistake is made, it can always be erased and changed.
If a practical-reactor designer errs, he wears the mistake around his neck ; it cannot be erased. Everyone can see it.

How to Significantly Exceed Past (NERVA) Performances?

Engine Performance Trade-offs:

High Specific Impulse Isp (high propellant exit velocity/temperature)
High Thrust for manned missions (high propellant flow rate)
High Thrust to Weight ratio T/W ratio (high power density)

$$I_{sp} = \frac{F_{thrust}}{\dot{m}} = \frac{1}{g_0}\sqrt{\left[\frac{2\gamma}{\gamma-1}\frac{RT}{M}\right]\left[1-\frac{p_e}{p_c}\right]^{\frac{\gamma-1}{\gamma}}}$$

- **Reduce "molecular" weight of propellant**
  - ⇨ **Dissociation of the H_2 propellant**
- **Increase Fuel Temperature**
  - ⇨ **Liquid (fuel) cores**
  - ⇨ **Gaseous (fuel) cores**
- **And what about fission fragments as the propellant** rather than slowing them down to heat the fuel that will heat the H2 propellant?
- **Switch from fission controlled chain reaction to explosive reaction (fission pulse propulsion)**
- **Switch from fission to fusion pulse propulsion**
- **Ultimately: antimatter propulsion**

"Advanced" "Futuristic" Concepts

Some of them investigated in the 60's in parallel with solid fuel NTP concepts

Hydrogen Dissociation

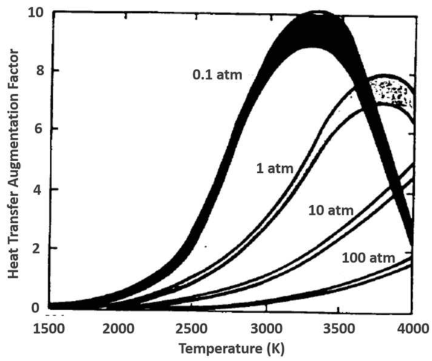
- H2 Dissociation to provide a low molecular weight propellant
- Dissociated H2 recombination in the nozzle to add thermal energy to increase T
- Both

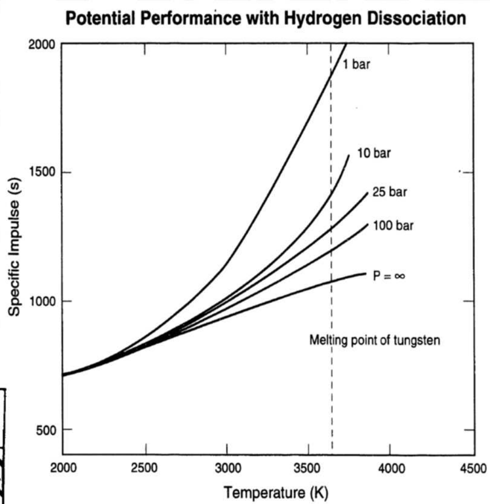
Maximum theoretical Isp with NERVA-type engine:
Isp = $\sqrt{2}$ (1.41) x NERVA Isp (~900 s) = **~1300 s**

Dissociation insignificant at realizable $T_{chamber}$ for a solid-fuel engine unless $P_{chamber}$ well below the 40 Bars required to achieve high power density in NERVA reactors

A Low-Pressure Design?
⇨Low power, low Thrust and longer burn time

A joker?: dissociation/recombination quite significantly increase convective heat transfer
⇨ **The LPNTR concept**





Source: Clayton W. Watson, Nuclear Rockets: High-Performance Propulsion for Mars, Report LA-12784-MS, UC-743, May 1994

LPNRT: The Low Pressure Radial Flow Nuclear Thermal Rocket Concept (US INEL, 1990)

Merits and challenges of low pressure concepts!

Low pressure: $P_{chamber}$ = 100 kPa (ΔP = 140 kPa)

☺ No need for turbopumps (works on tank pressure)

☺ No need for control drums: reactivity control through H_2 nozzle bypass (to be confirmed)

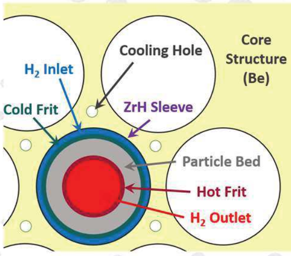
☺ High Isp owing to H_2 dissociation-recombination effects (strong increase of heat transfer); still, very high fuel T!)

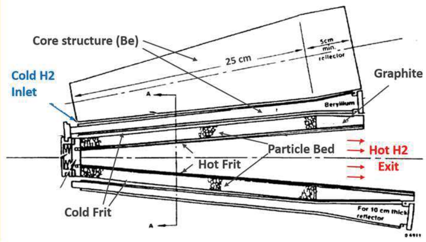
☹ Ground testing (low pressure exhaust cleanup)!

☹ Effects of dissociated H (on fuel corrosion, …)?

☹ Low thrust (requires clustering)

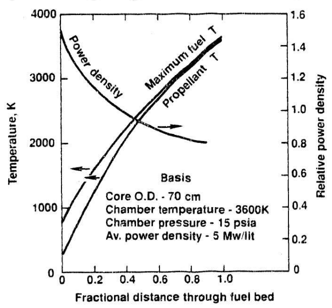
A fuel element looking like the one of Timberwind (but conical)

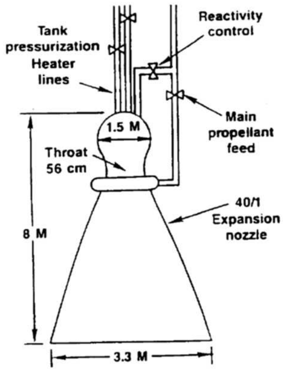


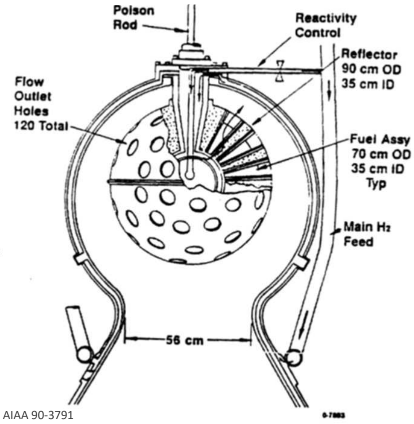


Power / Thrust:	260 MW / 48 kN
Chamber T:	3200 K
Isp:	1050 s
Engine Mass:	1840 kg (w/o shield)
T/W(*):	~6 (w/o any shield)
Fuel:	UC-ZrC (1 mm beads)
	40 kg ^{235}U
	120 conical Fas
	5 MW/l
Max Fuel T:	3636 K (!)

(*) *T/W expressed in lbf/lbm (= N/(g_0 kg))*







Sources: C. F. Leyse et al., A Preliminary Stage Configuration For A Low Pressure Nuclear Thermal Rocket (LPNTR), 1990, AIAA 90-3791
J.H. Ramsthaler, Low Pressure Nuclear Thermal Rocket (LPNTR) Concept; Nuclear thermal propulsion: a joint NASA/DOE/DOD workshop; Cleveland, OH (United States); 10-12 Jul 1990

The Liquid Annular Reactor System (LARS) Concept (US BNL, 1991)

Key features:

1. **Molten fuel contained in its own material**
2. **Molten layer stabilized by centrifugal force**
3. **Hydrogen is dissociated ⇨ high Isp**

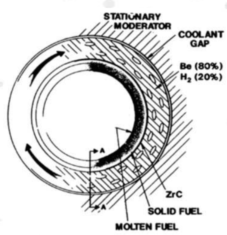
Pre-conceptual studies

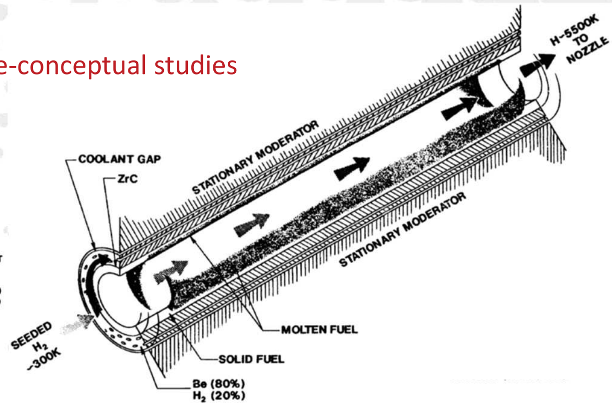
Fuel: high temperature refractory material, contains Uranium and appropriate diluents, possibly a mixture of UC_2 and ZrC

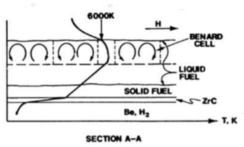
Heat transfer to H_2 by convection and radiation (H_2 seeded with micron-size W particles), 5-10 kW/cm^2
Fuel to Be can heat flow ~100 W/cm^2

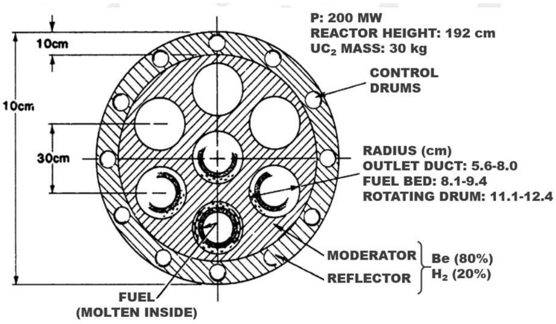
10 atm chamber pressure & 5500 K = H_2 fully dissociated
⇨ Isp ~1600- 000 s

? Evaporation loss of fuel
? Stability of molten fuel layer (acceleration, ….)
? Compatibility of molten fuel with H_2
? Nozzle cooling
? H_2 seeding









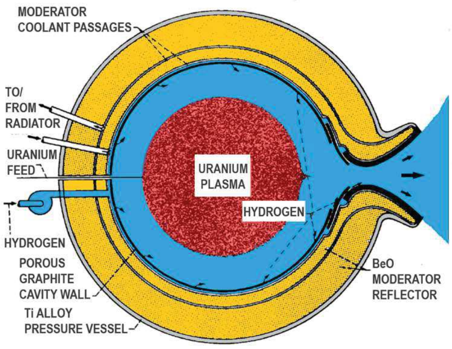
Source: James Powell et al., The Liquid Annular Reactor System (LARS) Propulsion, in NASA Lewis Research Center, Vision-21: Space Travel for the Next Millennium, 1991

Gas-Core-Based Nuclear Thermal Propulsion

Fissioning fuel in plasma-like state at peak temperatures ~50 000 K
⇨ **specific impulses: 2000 – 7000 s** (Hydrogen propellant chamber temperature up to 20 000 K **with high thrusts (100s kN)** but with **lower Thrust to Weight ratios** (~1/10) compared with solid cores

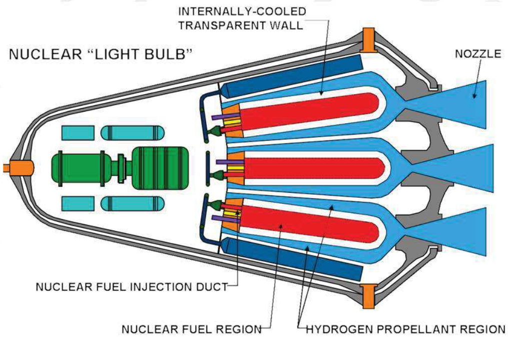
Open-Cycle Gas-Core Reactor

☹ Fission products exhausted with H_2 propellant (ground nuclear testing in particular, non only)
☹ Fissile fuel loss by entrainment with H_2 propellant (fuel plasma hydrodynamic/magnetic confinement)



Closed-Cycle Gas-Core Reactor

☹ ☹ Transparent material to confine fuel plasma
☹ Complexity of preventing transparent material from being plated with Uranium and being damaged by impinging fission fragments



The Open-Cycle Gas-Core Reactor Concept (US, mid 60's to early 70's)

Ball of Uranium plasma (peak/edge temperature ~55 000/ 26 000 K) hydrodynamically confined (vortex stabilized) by the flowing hydrogen propellant (which enters the cavity by flowing though its porous graphite wall)

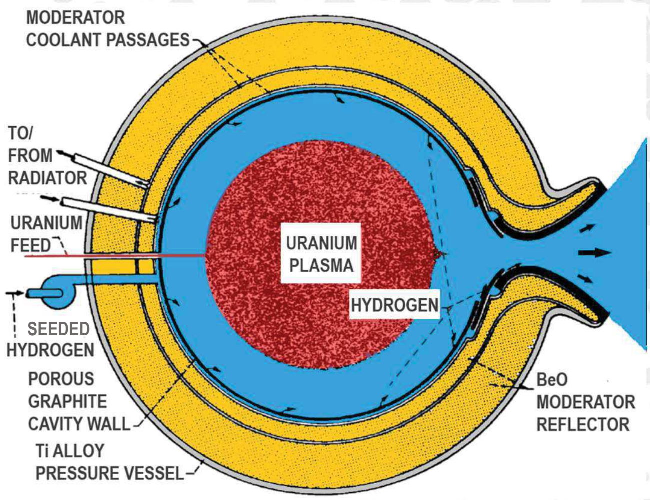
Hydrodynamically confined Uranium plasma occupies no more than ~25% of the cavity to limit the amount of uranium lost (scrapped off by and exhausted with the hydrogen propellant) to 1/100 to 1/400 of the propellant flow

High Hydrogen Pressure (~500 atm) to ensure criticality / an appropriate Uranium density (~6 10^{-3} g/cm^3) in the plasma within a cavity of reasonable diameter (~4 m); critical mass ~ 50 kg U(98% U5) with ~60 cm thick BeO moderator/reflector (high temperature hydrogen has a negative impact on reactivity due to upscattering);
Engine for 196 kN at 4400 s Isp: 6000 MWth, ~258 tons (60% due to radiator!)

Hydrogen propellant seeded with ~5w% C or W nanoparticles absorbs >99% of the heat radiated by the Uranium plasma; Isp up to 6500 s

~7% of the power (gammas and neutrons) deposited in the cavity wall and moderator/reflector; their regenerative cooling by H_2 possible up to Isp < 3000 s; beyond, need for a cooling circuit & radiator to dissipate heat. Radiator dominates engine mass at high thrust (at > ~110 kN for 5000 s Isp)

US: hydrodynamic confinement; TRL=3-4
(USSR: magnetic confinement)



$$P_{H_2} = 3.8\ 10^{-3} \frac{M_U^{1.385}\ F^{0.383}\ I_{sp}^{0.383}}{D_c^{4.54} \left(\frac{V_U}{V_c}\right)^{1.51}}$$

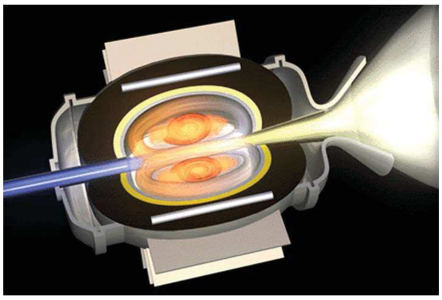
Sources: Robert G. Ragsdale, High-Specific-Impulse Gas-Core Reactors, NASA TM X-2243, March 1971; Maynard F. Taylor et al., Reactor Moderator, Pressure Vessel and Heat Rejection System of an Open-Cycle Gas-Core Nuclear Rocket Concept, NASA TM X-2772, July 1973

Open-Cycle Gas-Core Nuclear Thermal Propulsion (US & USSR, 60's to early 70's)

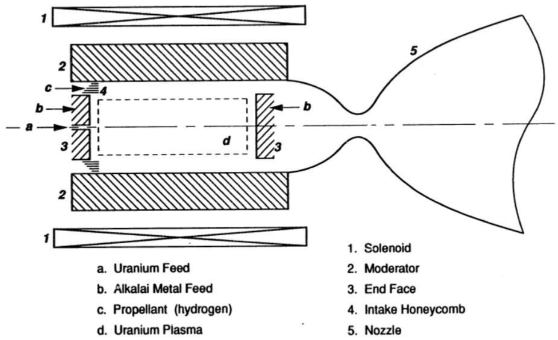
In the US, a mostly empirical experimental program on:
- Plasma stability,
- Uranium plasma emissivity,
- Hydrogen opacity,
- Gas-phase criticality (3 zero power reactors built)

Lack of computational capabilities and plasma dynamics in its infancy at that time, accurate assessment of the chaotic, complex behavior of a fluid-stabilized plasmoid was unreachable

Concept revisited in the late 90's, challenge of hydrodynamic confinement confirmed ⇨ innovative configurations?



In the USSR: hydrodynamic + magnetic confinement



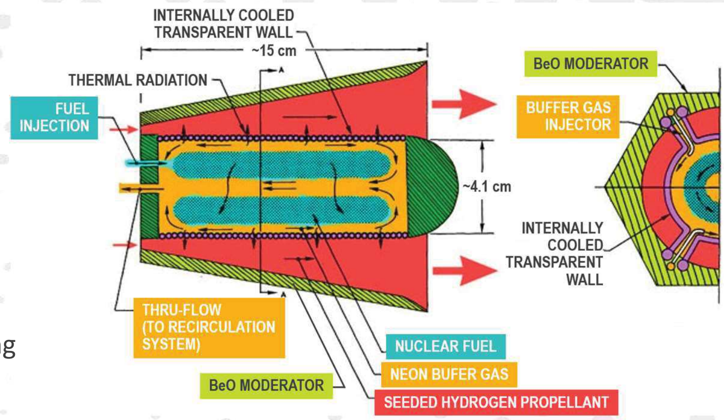
Magnetic field strengths required to stabilize undesirable flow characteristics:

Acoustic instability	2-3 Tesla
Hydrodynamic instability (Turbulence suppression)	3-4 Tesla
Longitudinal acceleration	3-5 Tesla
Rotational instability	7-10 Tesla

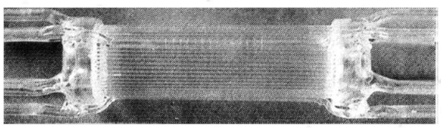
Closed-Cycle Gas-Core: The Nuclear Light Bulb Engine (US, mid 60's to early 70's)

Main features of the design concept

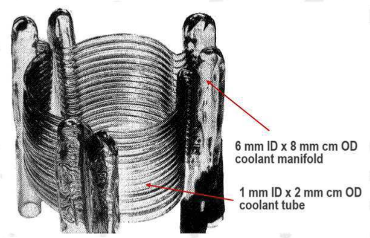
- Energy is transferred by thermal radiation from gaseous Uranium fuel (surface T > 8000 K) suspended in a neon vortex to W-nanoparticle-seeded H_2 propellant
- The vortex and propellant regions are separated by an internally-cooled transparent wall (fused silica)
- Neon buffer gas injection is aimed at avoiding diffusion of Uranium fuel towards the transparent wall (U plating prevention) and at preventing fission fragments from impinging on the wall
- Silicon-seeded Neon is injected to drive the vortex, flows laminarly to the end wall where it is removed. The neon discharged from the cavity along with any entrained fuel and fission fuels, is cooled by mixing with low T Ne, Uranium condensed to liquid form centrifugally separated from Ne and pumped back into the vortex region
- ~500 atm pressure to ensure criticallity
- Neutron moderation by BeO, Graphite reflector



Engine Unit Cavity Schematic Arrangement



Axial fused silica coolant tubes



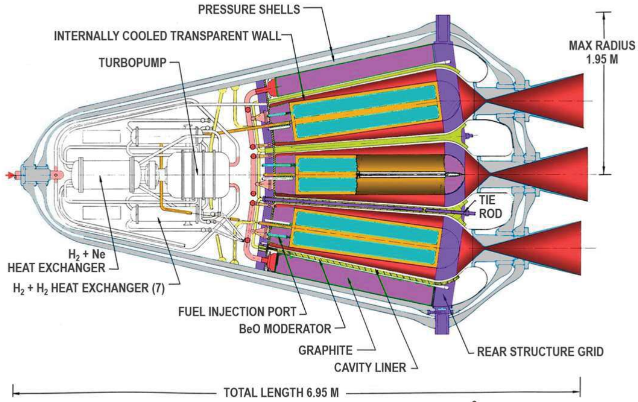
Circumferential fused silica coolant tubes

Source: C. H. McLafferty and H. E. Bauer, Studies of Specific Nuclear Light Bulb and Open-Cycle Vortex-Stabilized Gaseous Nuclear Rocket Engines, United Aircraft Corporation, NASA CR-1030, 1968

Closed-Cycle Gas-Core: The Nuclear Light Bulb Engine (US, mid 60's to early 70's)

Four principal coolant circuits:

- H_2 Propellant (~4480 MW)
- Secondary H_2 coolant (~250 MW)
- Fuel-Neon separator and recirculation (~150 MW)
- Space radiator H_2 coolant (~120 MW nominal(*))



Designed to fit with the US space shuttle (mass and bay volume)

Cavity pressure: 500 atm

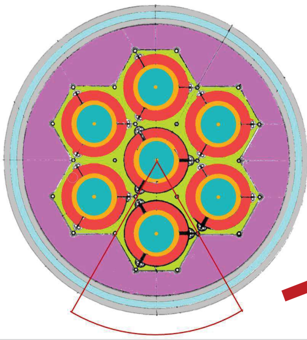
Specific Impulse: 1870 s (H_2 inlet(**)/exit T: 2260/6670 K)

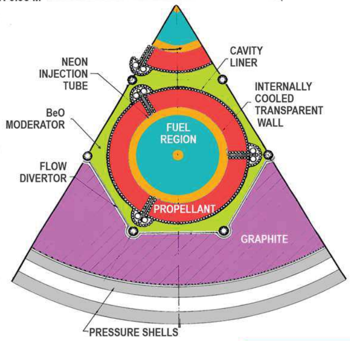
Thrust: 410 kN

Reactor Power: 4600 MW (^{233}U fuel(***))

Engine Weight: 37 tons (incl. 5.5 tons radiator)

Thrust/Weight = 1.1





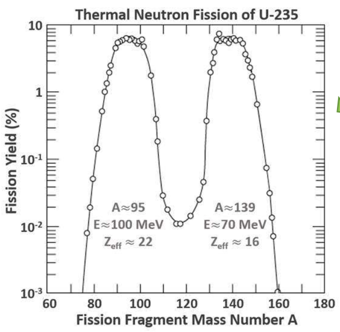
(*) also used for decay heat removal

(**) Cavity upper end-wall liner outlet propellant Temperature

(***) 1/3 critical mass of ^{235}U

Source: Richard J. Rodgers et al., Analytical Studies of Nuclear Lightbulb Engine Radiant Heat Transfer and Performance Characteristics, United Aircraft Research Laboratories report K-910900-10, September 1973

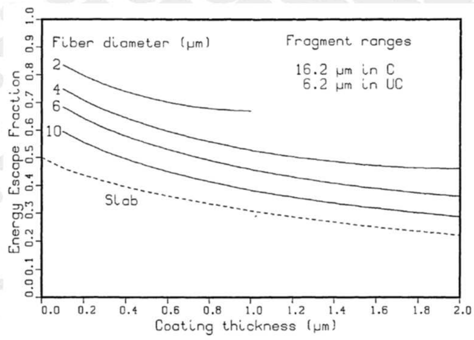
Fission Fragment Propulsion (1/2): Chapline's Concept (1988, US LLNL&INEL)

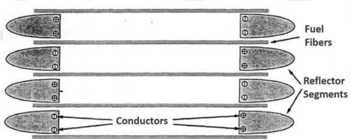


Fission fragments: 169 MeV or ~3% speed of light
(out of ~203 MeV released by U235 nucleus fission)

To recover > 50% of fission fragments energy:

- ⇨ **sub-micron thick fissile material coating (UC) on micron-scale diameter C fiber**
- ⇨ fibers arranged in **layers with $\rho.t_l$ < 1 mg/cm²**
- ⇨ **fast rotation of fiber layers** for small residence time in reactor and heat radiation outside
- ⇨ **very low fuel density ~10^{-4} g/cm³** requiring large reactor (reflector) size with **highly fissile materials** (242*Am: the best but …)
- ⇨ a few 0.1 Tesla enough to extract FFs

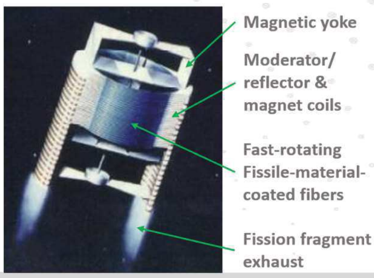


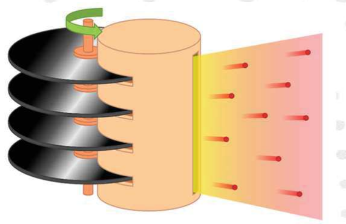


Critical Mass
for H 5 m x D 1 m core surrounded with a 3 m thick D2O reflector

^{242*}Am	0.5 kg
^{245}Cm	1.1 kg
^{239}Pu	5.6 kg
^{235}U	11. kg

?? Waste heat extraction
? Fuel burn-up
? Radioactive pollution
!! Very high Isp but…
… very low thrust





Source: George f. Chapline et al., Fission fragment rockets - a potential breakthrough, in Proc. 1988 International Reactor Physics Conference, Vol. 4

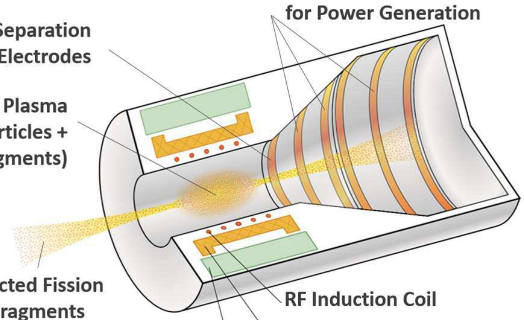
Fission Fragment Propulsion (2/2): Dusty Plasma Based Reactor Concept (2005)

Fuel nanoparticles (< 100 nm):
- high probability of fission fragment escape
- high heat radiation efficiency (large surface to volume ratio)

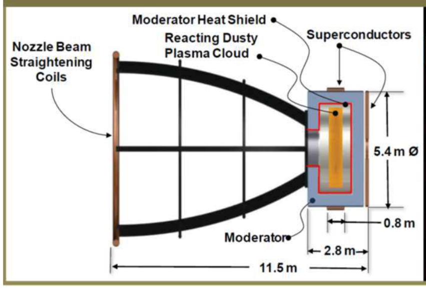
Dusty plasma cloud:
- fuel nanoparticles: E/q = 10^{-5}eV/q, 10^5 amu/e
- fission fragments: E/q = 10^3eV/q, 5 amu/e

⇨ **fuel particles electrostatically or magnetically contained within the reactor core**

⇨ fission fragments magnetically extracted for thrust and/or power generation



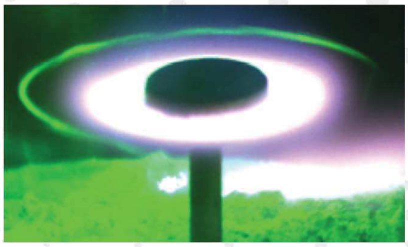
Initial Generation FFRE Design



Master Equip List Mass incl 30% MGA	
FFRE System Total, mT	113.4
Nozzle	6.4
Magnetic Mirror	28.6
Exit Field Coil	11.1
Moderator	51.2
Moderator Heat Shield	0.1
Control Drum System	0.7
Electrostatic Collector	0.3
Dust Injector	7.2
Shadow Shield	7.8

Distribution	(MW)
Total Reactor Power	*1,000*
Neutrons (30% to FFRE)	24.2
Gammas (5% to FFRE)	95.6
Other	70.2
Thermal (IR)	699
Jet Power	**111**
Performance	
Thrust	**43 N** (9.7 lbf)
Exit Velocity	**5170 km/s**
Specific Impulse	**527,000 s**
Mass Flow	**0.008 gm/s**

➡ "Afterburner" ? (neutral gas injection)



Magnetically confined dusty plasma

Sources: Rodney A. Clark and Robert B. Sheldon, Dusty Plasma Based Fission Fragment Nuclear Reactor, AIAA 2005-4460; Robert Werka et al., Final Report: Concept Assessment of a Fission Fragment Rocket Engine (FFRE) Propelled Spacecraft, 2012

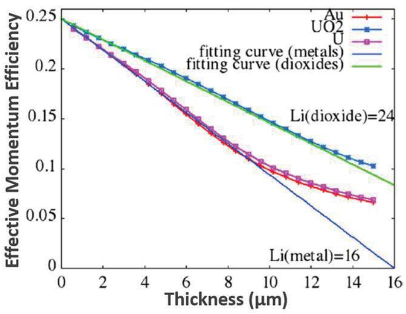
Alpha-Decay Particle Propulsion Sail: a Dead End! (A.A. Bolokine, 1982 patent)

$^4_2He^{2+}$: **5-9 MeV = ~5% speed of light**

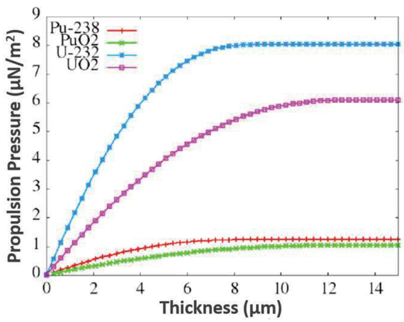
$^{232}U \xrightarrow[68.9\ y]{5.41\ MeV} {}^{228}Th \xrightarrow[1.9\ y]{5.52\ MeV} {}^{224}Ra \xrightarrow[3.6\ d]{5.78\ MeV} {}^{220}Rn \xrightarrow[55.6\ s]{6.40\ MeV} {}^{216}Po \xrightarrow[0.15\ s]{6.91\ MeV} {}^{212}Pb$

$^{212}Pb \xrightarrow{\beta^-\ 10.6\ h} {}^{212}Bi$; $^{212}Bi \xrightarrow[61\ min]{6.21\ MeV} {}^{208}Tl$; $^{212}Bi \xrightarrow{\beta^-\ 1.0\ h} {}^{212}Po$; $^{212}Po \xrightarrow[61\ min]{6.21\ MeV} {}^{208}Pb$

$^{238}Pu \xrightarrow[87.7\ y]{5.1\ MeV} {}^{234}U$

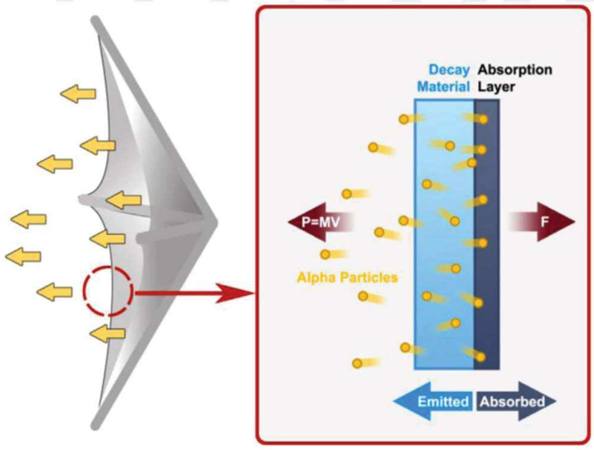


Maximum effective momentum efficiency: 25%



Propulsion pressure performance with ^{232}**U** comparable to the one of solar sails near the earth (~9 µN/m²)

^{238}Pu sail not competitive with solar sail



!!! Production (and handling) of ^{232}U !!! (^{232}U daughters ^{224}Ra, ^{220}Rn, ^{212}Bi are strong γ emitters)

Source: Wenwu Zhang et al., Revisiting alpha decay-based near-light-speed particle propulsion, Applied Radiation and Isotopes 114 (2016) 14–18

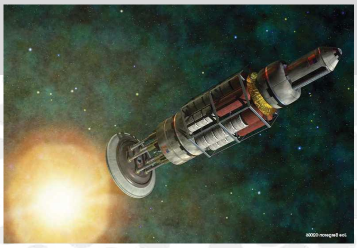
Bonus # 2

cea

DE LA RECHERCHE À L'INDUSTRIE

Nuclear Pulse Space Propulsion

Project ORION, General Atomics, 1958-1965



7-y, US$ 11 M project carried by General Atomics, funded par DARPA, USAF, NASA

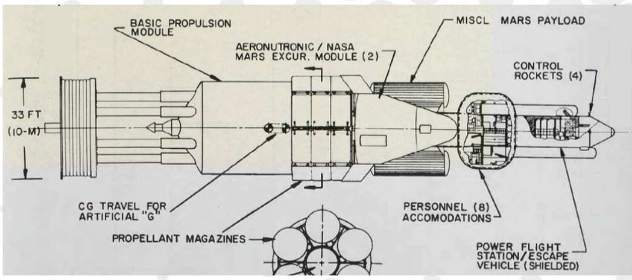
Spacecraft propelled by a series of atomic bombs explosions behind the spacecraft

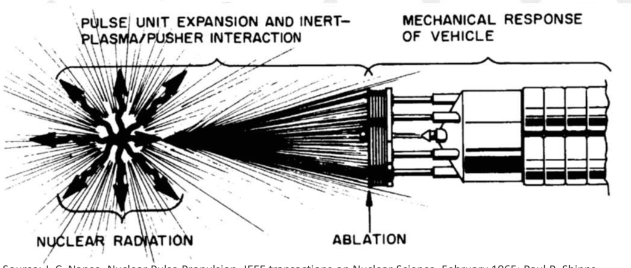
- Uranium fission has an energy density of ~7.8 10^6 MJ/kg corresponding to a maximum theoretical Isp of ~1.3 10^6 s
- Impingement velocity ~100-200 km/s (limited by pusher-plate ablation)
  Fraction of pulse unit reaching pusher-plate: 10-50%
  ⇨ **Isp ~3 000-10 000 s**

$$I_{sp} = \frac{f V_i}{g_0}$$

- The (only) way to achieve **both together high thrust and high Isp, thought feasible using technologies available at that time**
- VIPER experiment at Eniwetok island nuclear facility (20 kt nuclear device detonated at 10 m from two 1-m diameter graphite-coated steel spheres, later recovered 2 km from ground zero with their interior completely unscathed and a few tens of micros of graphite ablated)
- 6 non-nuclear tests conducted using models demonstrated stable flight
- Program stopped following the entry into force of the Partial Test Ban Treaty

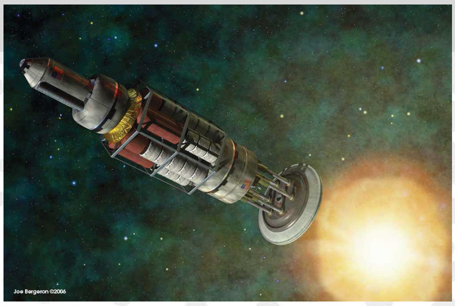
Source: G. R. Schmidt et al., Nuclear Pulse Propulsion – ORION and Beyond, AIAA 2000-3856



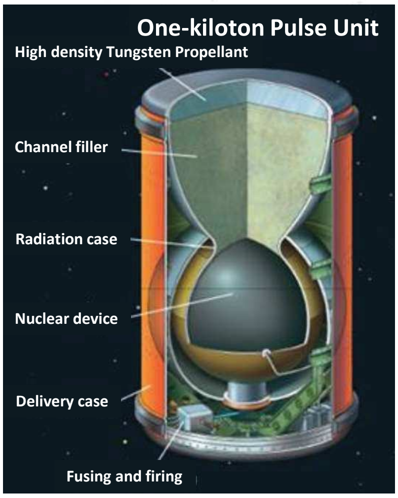


Source: J. C. Nance, Nuclear Pulse Propulsion, IEEE transactions on Nuclear Science, February 1965; Paul R. Shipps, Manned Planetary Exploration Capability Using Nuclear Pulse Propulsion (1965), The Space Congress® Proceedings. 2.

Project ORION, General Atomics, 1958-1965



Design Principle of the self-actuating nuclear pulse unit

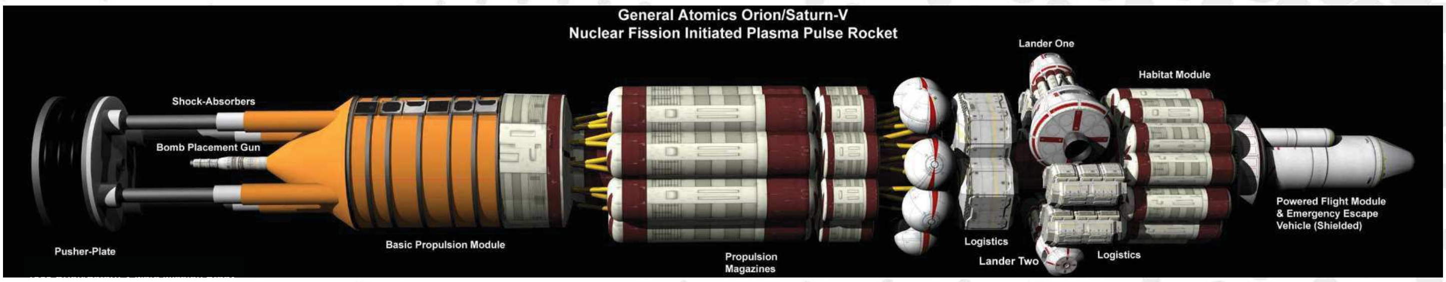


- Bomb ejected via a magnetic rail gun, passes through an aperture in the center of the pusher-plate
- Channel filler absorbs radiation and rises to high temperature
- Radiation case contains the energy released so that more energy is absorbed by the channel filler
- High pressure achieved in the heated channel filler drives a strong shock into the propellant, which vaporizes and is driven to the pusher plate
- During the few millionths of a second of the bomb expansion, chamber filler and tungsten absorb neutrons and X-rays, thus
  - reducing shielding requirements for the crew, and
  - transforming much of the bomb output into kinetic energy that can be intercepted by the pusher plate and propel the ship

Source : Nuclear Pulse Space Vehicle Study, Vol. III, Conceptual Vehicle Designs and Operational Systems, General Atomics Report GA-5009, issued Sept. 19, 1964

Project ORION, General Atomics, 1958-1965

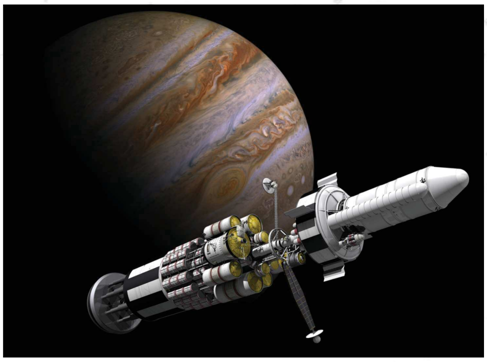
ORION vehicle for MARS Crewed mission (the motto of the time in the US: Mars by 1965, Saturn by 1970)

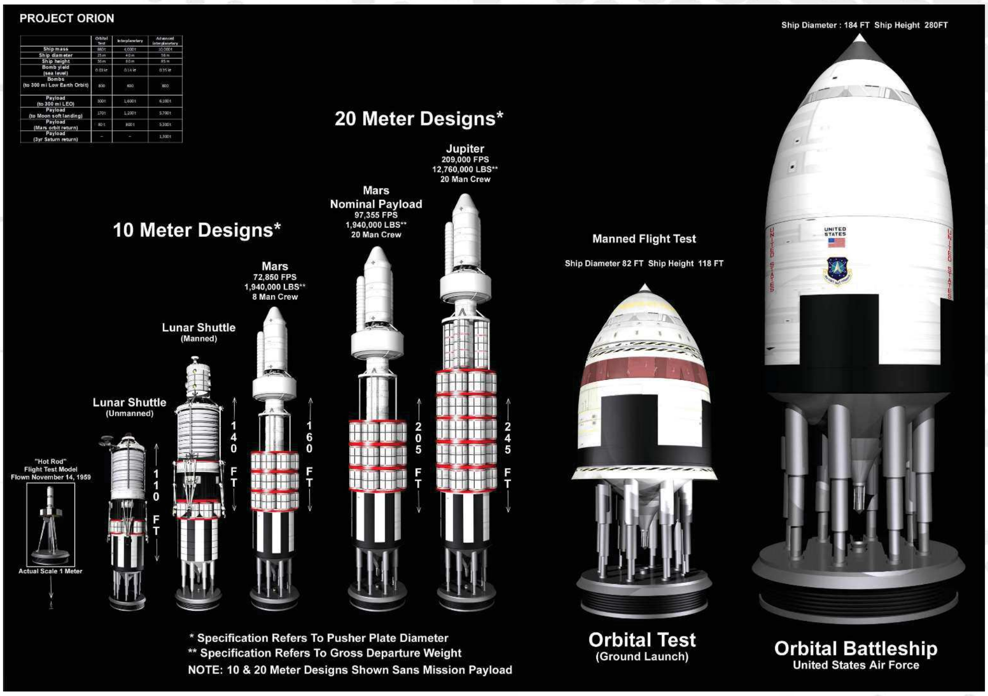


1960 Orion/Saturn-V Mars Mission Study
- Summary: General Atomics study NASA Orion/Saturn-V Interplanetary Spacecraft
- Propulsion: 15kt Nuclear fission initiated Plasma Pulse
- Braking at Mars: propulsive
- Mission Type: opposition
- Split or All-Up: all up
- Launch Year: 1965
- Crew: 20
- Mars Surface payload-metric tons: 150
- Outbound time-days: 42.5
- Mars Stay Time-Days: 40
- Return Time-Days: 42.5
- Total Mission Time-Days: 125
- Total Mass metric tons: 200
- Propulsion System Mass: 100
- Launch Vehicle Payload to LEO metric tons: 100
- Number of Launches Required to Assemble Payload in Low Earth Orbit: 2
- Launch Vehicle: Saturn V-25(S)U

- Crew Size: 20
- Length: 204 ft
- Basic Diameter: 33 ft
- Main Engine: Orion Nuclear Fission Initiated Plasma Pulse Rocket
- Propulsion Units: 11,400 Nuclear Fission Propulsion Charges
- Propulsion Unit Delivery: Electric-Rail Placement Gun.
- Impulse Charge Yield: 15 kt
- Shock Absorber System: Reciprocal, Two-Stage.
- Main Engine ISP: 2500 sec.
- Exhaust Velocity: 120,000 m/s
- Thrust: 4e8N
- Main Engine Acceleration: 4 G's.
- Main Engine Delta v: 72, 850 fps (49,670 mph)

Project ORION, General Atomics, 1958-1965





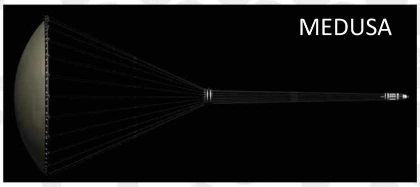
The larger the diameter of the pusher-plate, the higher the Isp

(thrust-to-weight ratio ~independent of pusher-plate diameter)

Improving Nuclear Pulse Propulsion

Nuclear Fission Pulse Propulsion: towards improved performances

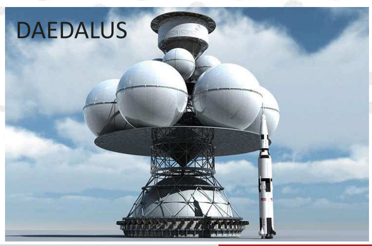
- Use smaller yield pulse units (reduces shock-absorbers mass)
- Use a puller-sail instead of a pusher-plate to capture more of the exhaust products and to improve shock absorption capacity
- Use of a magnetic field to shield the pusher-plate surface from the high energy plasma, thus reducing ablation and enabling higher Isp
- Miniaturize nuclear pulse units



Nuclear Fission Pulse Propulsion Major drawbacks / showstoppers:

- Environmental impact
- Launch safety
- Proliferation

Switch from nuclear fission to nuclear fusion pulse propulsion

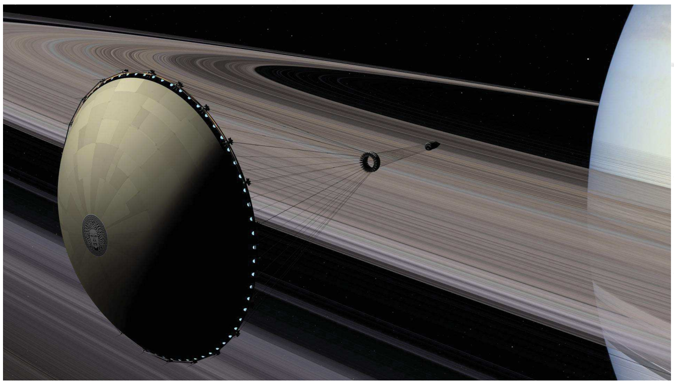


Nuclear Fission Pulse Propulsion: The MEDUSA concept (LANL, 1991)

Main advantages over ORION:

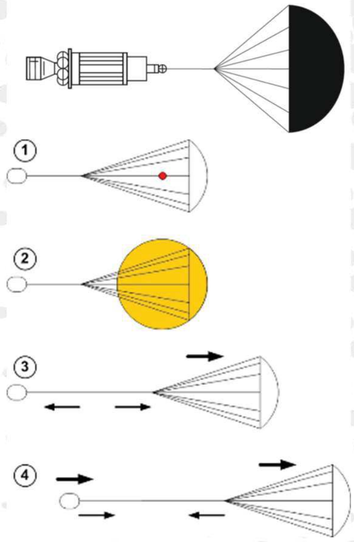
- Improved Isp (more exhaust products captured)
- improved shock absorption capacity
- Lower mass / vehicule size (was dictated by pusher-plate diameter and massive shock absorbers in the ORION concept)

Main drawback: any crew or the payload will be dragged through the radioactive detonation cloud of each pulse + deceleration (redeploy sail after years of storage)



$$I_{sp} \approx \sqrt{\frac{2\,E_b}{5\,m_b}}$$

for m_b = 25 kg / E_b= 10^5 MJ (25 tons)
I_{sp} = 4 250 s



Automated servowinches between the sail and the vehicle control the acceleration pulses

Source: Johndale C. Solem, Some New Ideas for Nuclear Explosive Spacecraft Propulsion, LA-22289-MS, UC-940, issued: October 1991

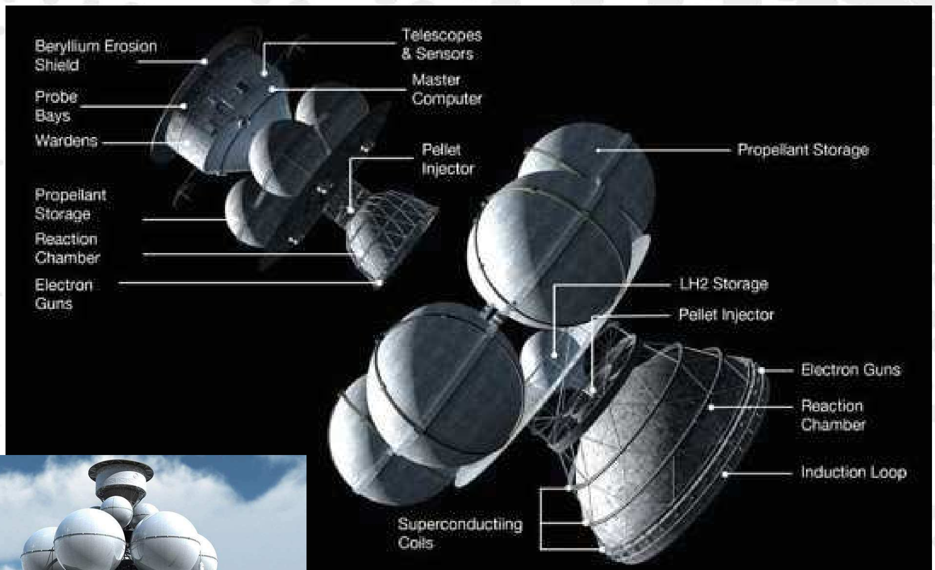
Nuclear Fusion Pulse Propulsion: Project DAEDALUS (UK study, 1973-1978)

Project Daedalus, a two-stage fusion microexplosion propulsion (ICF) spacecraft designed to send a scientific payload of 450 tons at 12% of light speed a one-way, 50-year fly-through mission to the 5.9 light-years distant Barnard's star

10^6 s Isp engines using D/^{3}He fuel
(^{3}He would have to be "mined" from Jupiter's atmosphere before the flight!)

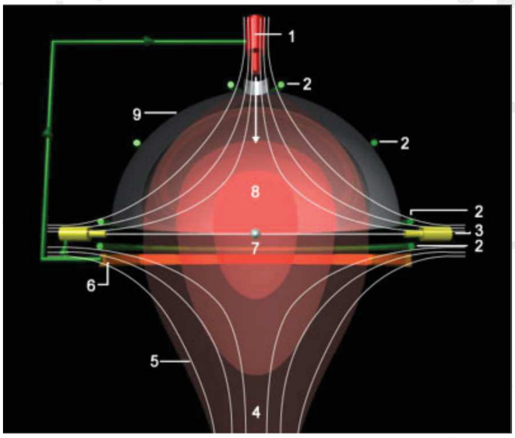
Daedalus spacecraft mass: 54 000 tons, including 50 000 tons of pellets ignited 250 times per second by inertial confinement using relativistic electron beams, the resulting plasma being directed by a magnetic nozzle

First stage fired for 2 y up to 7% c, second for 1.8 y up to 12% c



Source: Project Daedalus: Demonstrating the Engineering Feasibility of Interstellar Travel, Edited by K.F.Long and P.R. Galea, The British Interplanetary Society (2015)

Nuclear Fusion Pulse Propulsion: principle adopted for DAEDALUS (UK study, 1973-1978)

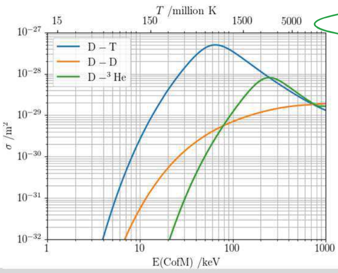


1 Pellet injection gun
2 Superconducting field coils
3 Electron beam generators
4 Plasma exhaust jet
5 Magnetic field
6 Energy extraction coils
7 Frozen fusion fuel pellet
8 ''Nuclear explosion''
9 Reaction chamber

Isp ~ 10^6 s

Fusion burn efficiency

Isp ~ 2.6 10^6 s



$D + {}^3H_e \rightarrow {}^4H_e\ (3.67\ MeV) + p\ (14.67\ MeV) \rightarrow 18.34\ MeV \quad v_{eject} = 26\,500\ km/s$

$D + T \rightarrow {}^4H_e\ (3.52\ MeV) + n\ (14.06\ MeV) \rightarrow 17.58\ MeV \quad v_{eject} = 26\,400\ km/s$

$D + D \rightarrow T\ (1.01\ MeV) + p\ (3.03\ MeV) \rightarrow 4.04\ MeV \quad v_{eject} = 13\,920\ km/s$

$D + D \rightarrow {}^3H_e\ (0.82\ MeV) + n\ (2.45\ MeV) \rightarrow 3.27\ MeV \quad v_{eject} = 12\,510\ km/s$

Nuclear Fusion Pulse Propulsion: Project DAEDALUS (UK study, 1973-1978)

DAEDALUS Nominal Mission profile and vehicle configuration

Parameter	First stage value	Second stage value
Propellant mass (tons)	46,000	4,000
Staging mass (tons)	1,690	980
Boost duration (years)	2.05	1.76
Number tanks	6	4
Propellant mass per tank (tons)	7666.6	1,000
Exhaust velocity (km/s)	1.06×10^4	0.921×10^4
Specific impulse (million s)	1.08	0.94
Stage velocity increment (km/s)	2.13×10^4 (0.071c)	1.53×10^4 (0.051c)
Thrust (N)	7.54×10^6	6.63×10^5
Pellet pulse frequency (Hz)	250	250
Pellet mass (kg)	0.00284	0.000288
Number pellets	1.6197×10^{10}	1.3888×10^{10}
Number pellets per tank	2.6995×10^9	7.5213×10^9
Pellet outer radius (cm)	1.97	0.916
Blow-off fraction	0.237	0.261
Burn-up fraction	0.175	0.133
Pellet mean density (kg/m^3)	89.1	89.1
Pellet mass flow rate (kg/s)	0.711	0.072
Driver energy (GJ)	2.7	0.4
Average debris velocity (km/s)	1.1×10^4	0.96×10^4
Neutron production rate (n/pulse)	6×10^{21}	4.5×10^{20}
Neutron production rate (n/s)	1.5×10^{24}	1.1×10^{23}
Energy release (GJ)	171.82	13.271
Q-value	64	33

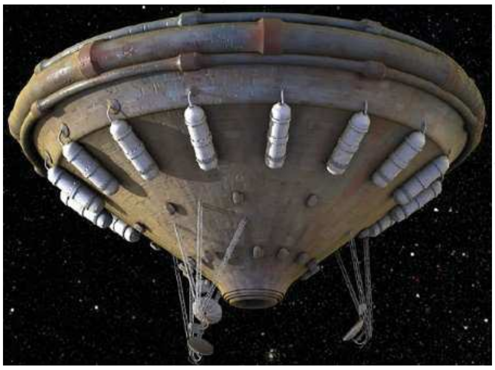
NB.: D-^{3}He fusion is not exempt of neutrons! (D-D fusion)

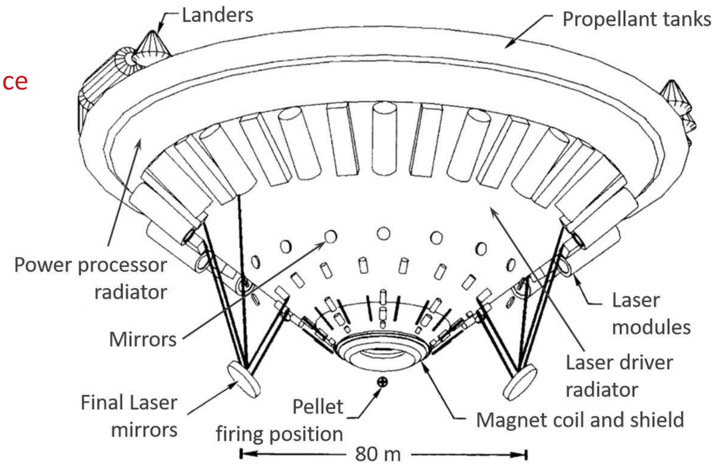
Nuclear Fusion Pulse Propulsion: The VISTA Concept using D/T Fusion (LLNL, 1988-2003)

1988-89 study was conducted to determine whether inertial confinement fusion (ICF) could be adapted for piloted space transport to Mars with sufficient increase in speed / transit time over conventional technologies
Extensive additions led to 2003 report

Relies on D-T fusion, use technologies thought to be available by mid 21st century, magnetic thrust chamber, …

6 000 tons spacecraft including 100 tons payload
Effective Isp: 15 000 s
Mars round-trip in 145 days, including 20 days on Mars surface





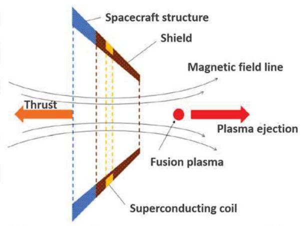
Source: C. D. Orth, VISTA – A Vehicle for Interplanetary Space Transport Application Powered by Inertial Confinement Fusion, LLNL report UCRL-TR-110500 (2003)

Nuclear Fusion Pulse Propulsion: VISTA addressing the issues associated with D/T Fusion

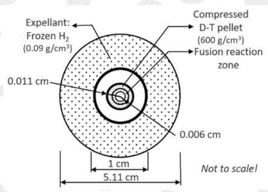
$v_{eject} = 4\,120\ km/s$ ↑ (for ^4H_e); $v_{eject} = 26\,400\ km/s$ ↑ (for n)

$$D + T \rightarrow\ ^4H_e\ (3.52\ MeV) + n\ (14.06\ MeV) \rightarrow 17.58\ MeV$$

Use a cone-surface-shaped spacecraft to minimize the neutron (+ X-rays) fraction intercepted by the spacecraft structures (2 x 6° or 3% of 360°)

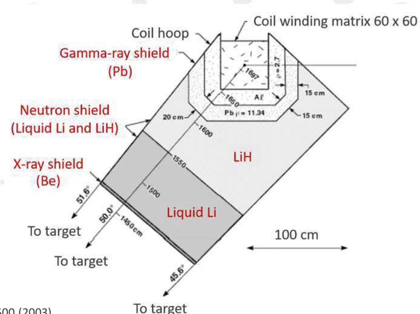


Transfer some of the neutron energy to "expellant" (frozen H_2 surrounding the DT pellet) to increase debris kinetic energy / Isp (typically 30% of neutron energy) and reduce shielding requirements (spatial shaping of expellant can further reduce neutron irradiation of spacecraft components)



Shield the superconducting coil from neutrons (and X-rays) to avoid quenching and reduce heat load to be extracted by the cryogenic system and radiate the deposited heat to space (typically, 2% of the DT fusion is deposited in the coil shield, requiring a ~500 tons shield)

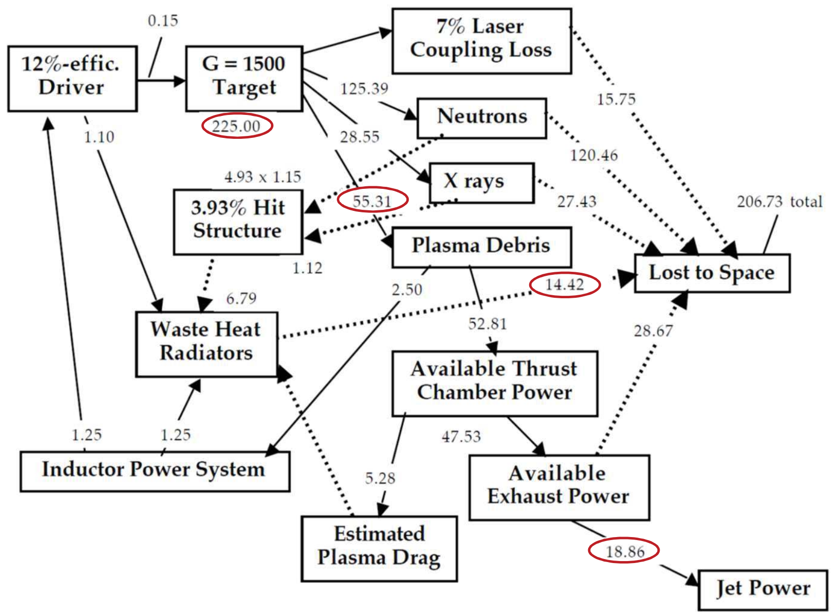
Breed Tritium onboard using the fusion neutrons (through (n, 7Li) reaction in the liquid Li coolant of the superconducting coil shield): transporting the T inventory for the mission (~2 000 kg for a trip to Mars) raises launch safety and cost issues



Source: C. D. Orth, VISTA – A Vehicle for Interplanetary Space Transport Application Powered by Inertial Confinement Fusion, LLNL report UCRL-TR-110500 (2003)

Nuclear Fusion Pulse Propulsion: The VISTA Concept using D/T Fusion (LLNL, 1988-2003)

VISTA Overall power flow (GW) for the advanced-technology mission to Mars using an Inductor power system



The jet power is ~8% of the power released by D-T fusion, and 15 times the laser driver power

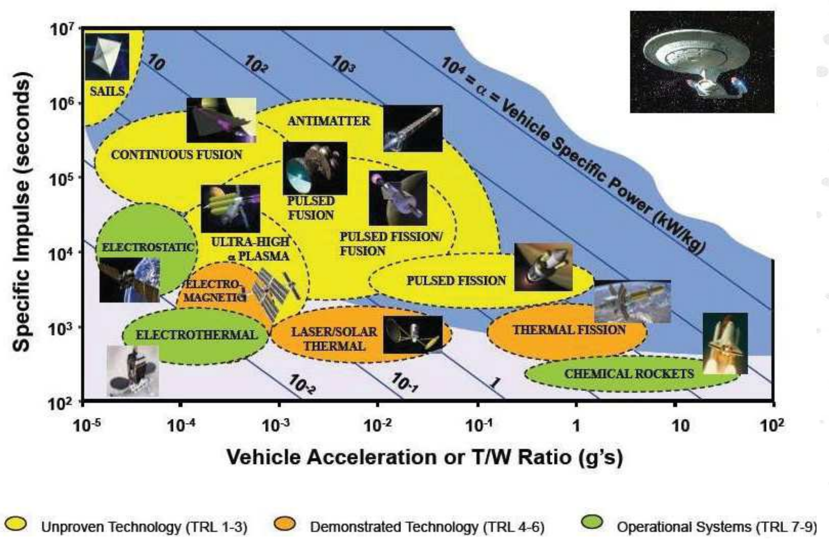
The spacecraft surface is a huge radiator (radiating 14 GW at 1 500 K requires 62 000 m^2)

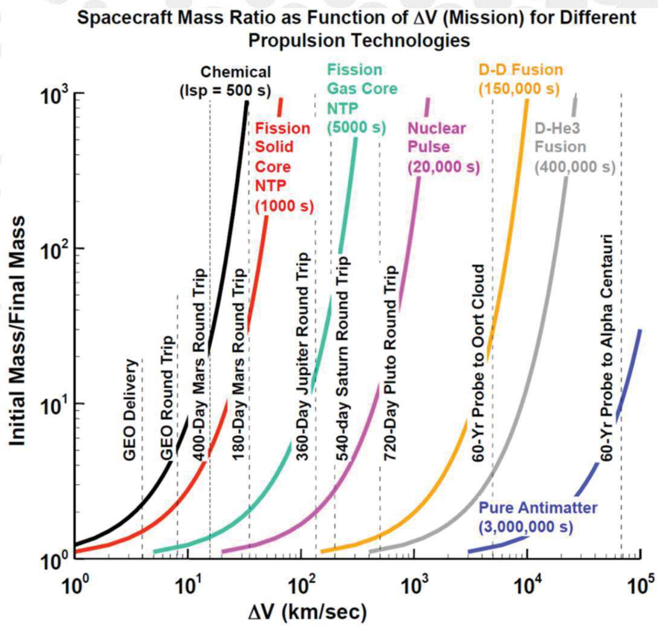
An inductor power system extract ~5% of the debris energy to power the laser drivers

A 100 kWe nuclear power reactor provides the energy for the first power pulse (~10 minutes charging time) and to the auxiliary systems

Source: C. D. Orth, VISTA – A Vehicle for Interplanetary Space Transport Application Powered by Inertial Confinement Fusion, LLNL report UCRL-TR-110500 (2003)

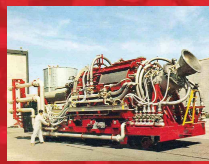
Capabilities of Candidate Propulsion Technologies





Source: George Schmidt, Nuclear Systems for Space Propulsion and Power, FISO Seminar, 8 Dec. 2010

Bonus # 3



DE LA RECHERCHE À L'INDUSTRIE

Air-Breathing Nuclear Thermal Propulsion

Russian 9M730 Burevestnik Missile: Nuclear Powered? Nuclear Ramjet Engine?

cnsnews — THE RIGHT NEWS. RIGHT NOW. | Washington | National | International | Commentary | Blog | Videos

Trump Links Explosion in Russian Arctic to Putin's New, Hyped Nuclear Cruise Missile

By Patrick Goodenough | August 13, 2019 | 4:38am EDT

A staffer at a nuclear museum in the closed city of Sarov with the first Soviet nuclear bomb. Behind that, the first Soviet thermonuclear bomb is visible. (Photo by Alexander Nemenov/AFP/Getty Images)

(CNSNews.com) – Authorities in Russia are saying little about a deadly explosion off the northern Russian coast five days ago, but President Trump on his Twitter account Monday signaled that the U.S. has linked it to a cutting-edge new cruise missile, which President Vladimir Putin has been touting.

ALL SECTIONS | SEARCH | THE DIPLOMAT — READ THE DIPLOMAT, KNOW THE ASIA-PACIFIC | SIGN IN | SUBSCRIBE

Russia Reveals 'Unstoppable' Nuclear-Powered Cruise Missile

Putin announced a new high-yield intercontinental-range cruise missile purportedly capable of penetrating any missile defense system.

By Franz-Stefan Gady
March 02, 2018

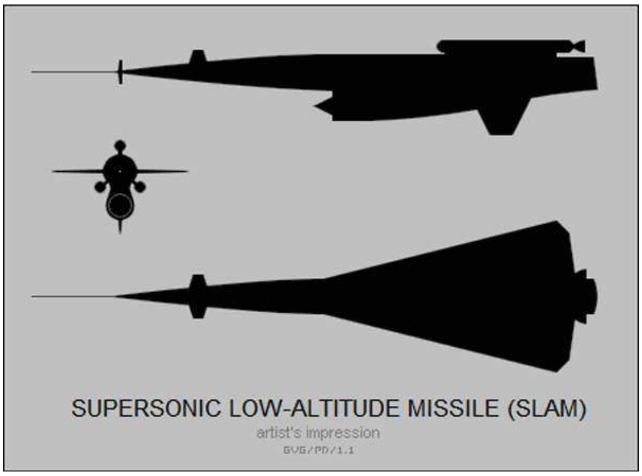
Russian President Vladimir Putin announced during his annual State of the Nation address on March 1 that the Russian defense industry has begun developing an intercontinental-range nuclear-powered cruise missile capable of penetrating any interceptor-based missile defense system.

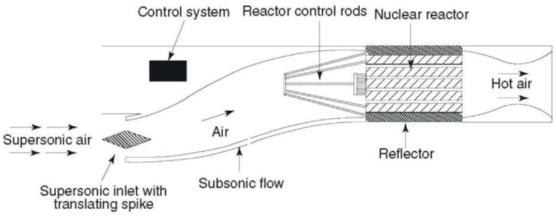
"We've started the development of new types of strategic weapons that do not use ballistic flight paths on the way to the target. This means that the missile defense systems are useless as a counter-means and just senseless," Putin said in his speech

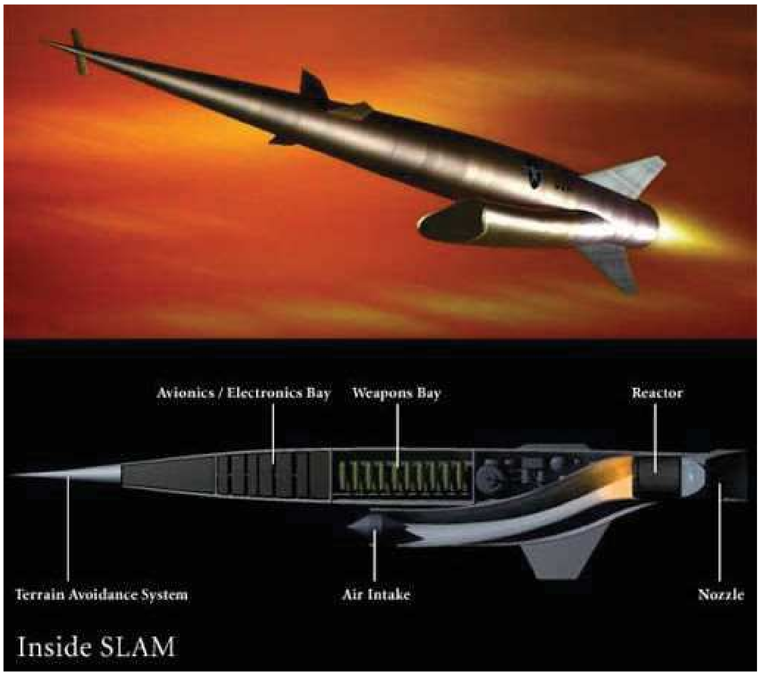
Credit: YouTube Still Shot

The 9M730 Burevestnik missile has unusual propulsion system with nuclear power unit

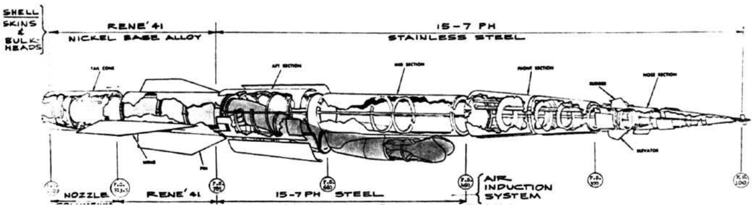
Air-breathing NTP / Nuclear Ramjet Engine : Project PLUTO / (1957-1964)

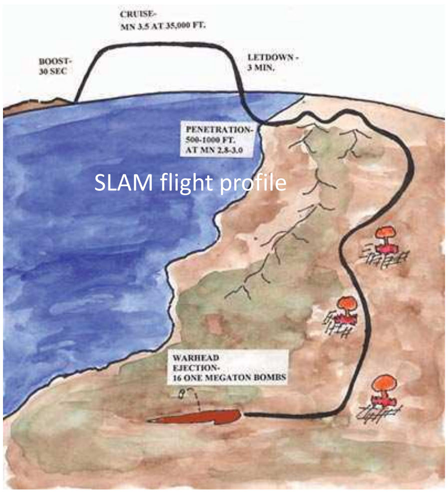




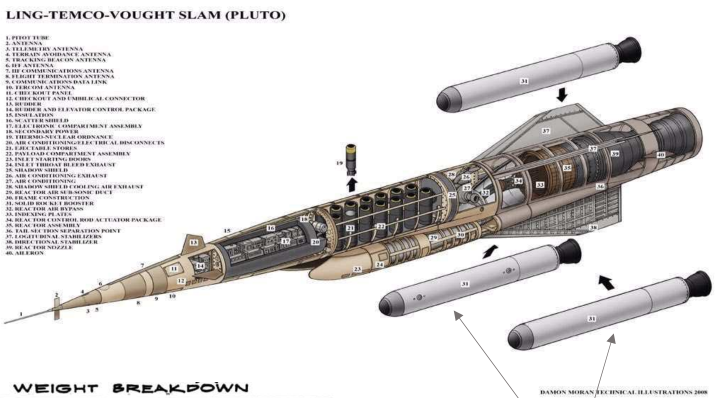


Air-breathing NTP / Nuclear Ramjet Engine : Project PLUTO / (1957-1964)





Mach 2.8
~5-10 FP hours engine lifetime



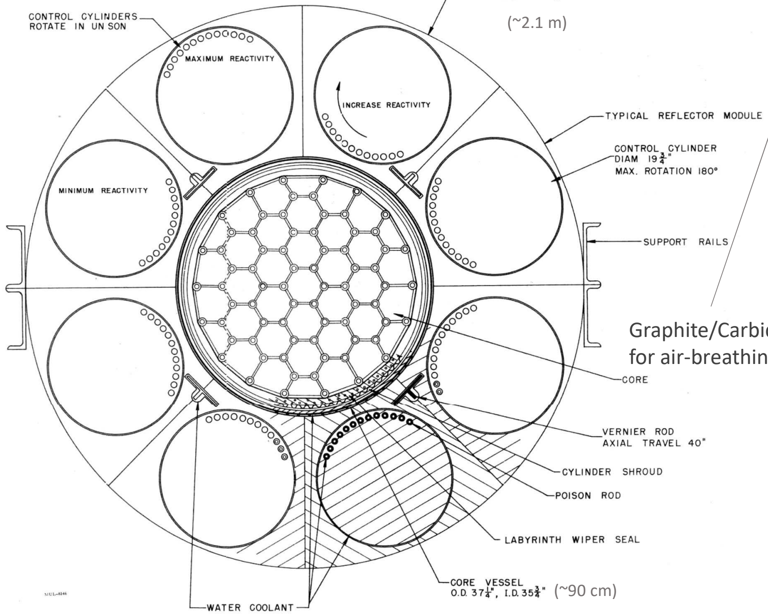
WEIGHT BREAKDOWN

COMPONENTS	WEIGHT (lbs)
SURFACES	(2710)
WINGS	1672
FIN	670
CONTROL SURFACES	368
FUSELAGE	(9195)
NOSE SECTION	491
FRONT SECTION	1071
MID SECTION	3349
AFT SECTION	3516
TAIL CONE	768
POWER PLANT	(22454)
REACTOR & SHELL	12867
AIR INDUCTION SYSTEM	4016
SHIELDING	4954
CONTROLS	617
EQUIPMENT	(6149)
WARHEAD	(8640)
FLIGHT GROSS WEIGHT	49148
BOOSTER WEIGHT	54401
LAUNCH WEIGHT	103549

Boosters needed to bring the missile to the speed needed for the ramjet engine to operate

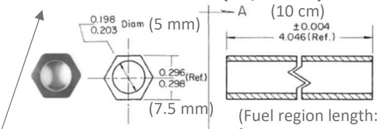
Air-breathing NTP / Nuclear Ramjet Engine : Project PLUTO / (1957-1964)

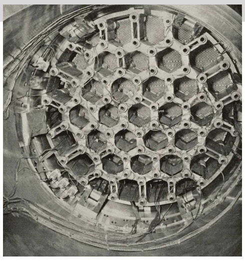
Propellant flow passages: 32% of front area



8623 fuel element passages

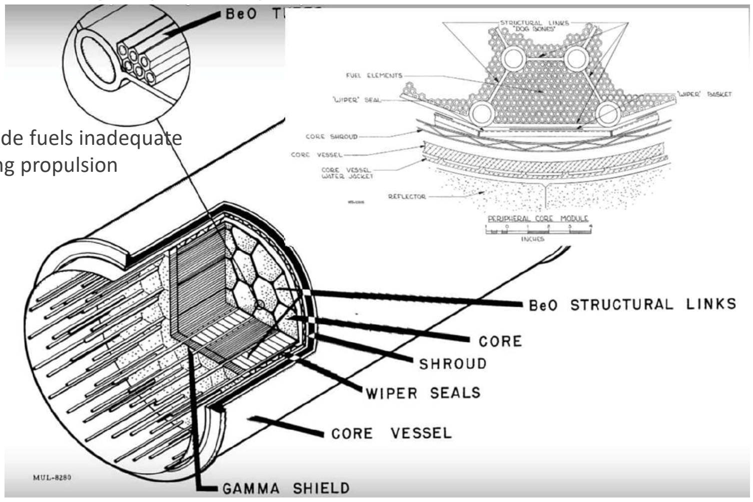
BeO-3.5-7.5Wt%UO_2 (93,2% U5) + εZrO_2





Max fuel wall T: 1230°C

Avg. fuel power density: 470 W/cm³



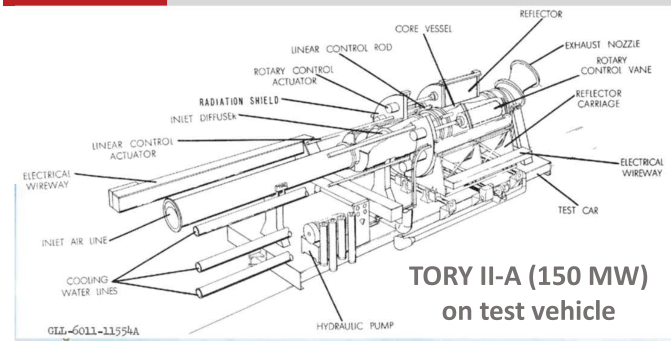
Graphite/Carbide fuels inadequate for air-breathing propulsion

Source: The PLUTO Program, UCLR-6398, 1961

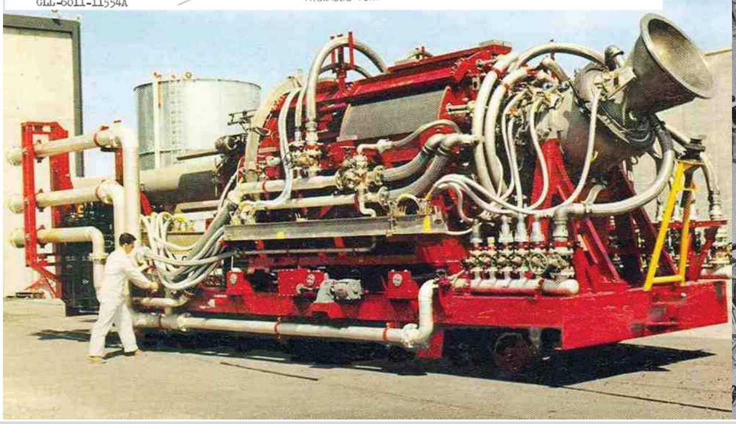
TORY IIA core and reflector (150 MW, 49 kg 93.2%U)

tested at full power in 1961

Air-breathing NTP / Nuclear Ramjet Engine : Project PLUTO / (1957-1964)



TORY II-A (150 MW) on test vehicle



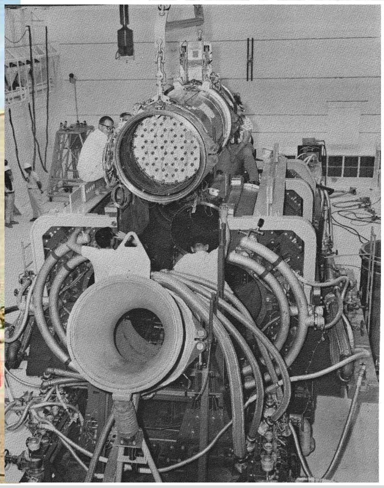
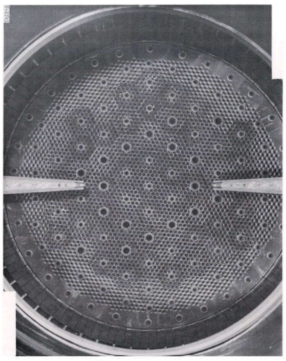


Table 4-1. Tory II-A Design and achieved operating parameters*

Test designation	Inlet air stagn. temp (°F)	Air-flow rate (pps)	Max power (MW)	Max av. fuel element temp (°F)	Av. exhaust air stagn. temp (°F)
Design op. pt.	946	634	150	2250	1763
IPT, May 14	393	114	46	2580	1840
HP-1, Sept. 28	394	445	144	2330	1560
HP-2, Oct. 5	925	650	166	2300	1810
HP-3, Oct. 6	402	432	162	2640	1745

Source: The TORY II-A Reactor Tests Final Report UCRL-7249, 1963

Air-breathing NTP / Nuclear Ramjet Engine : Project PLUTO / (1957-1964)



TORY II-C core (600 MW, 63 kg 93.2%U)

tested for 292 s at full power in 1964

21 000 fuel element passages

294 000 fuel elements

BeO-1.2-8.1Wt%UO2 (93,2% U5) + $\varepsilon Y_2O_3 + ZrO_2$

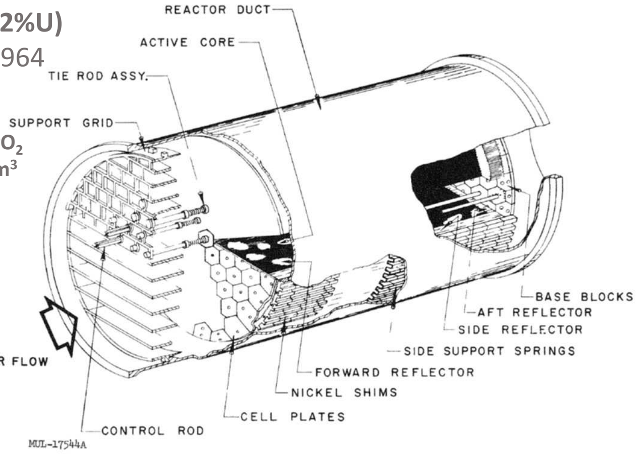
Max fuel material power density: 830 W/cm^3

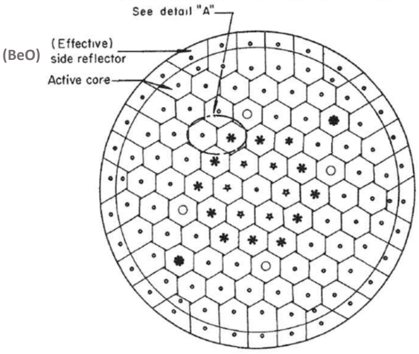
Active core: ϕ 120 x L 130 cm

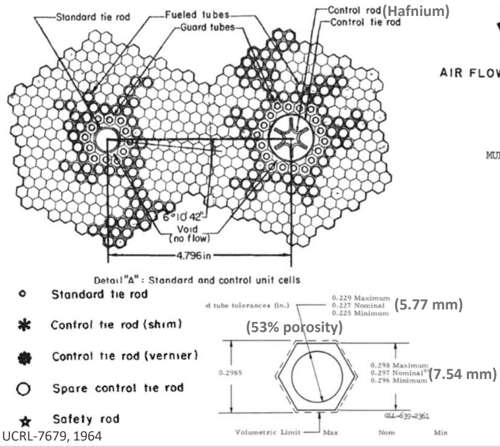
Side (BeO) reflector thickness: 7.5 cm

Overall: ϕ 145 x L 165 cm

Source: TORY II-C Data Book, UCRL-7315, 1963





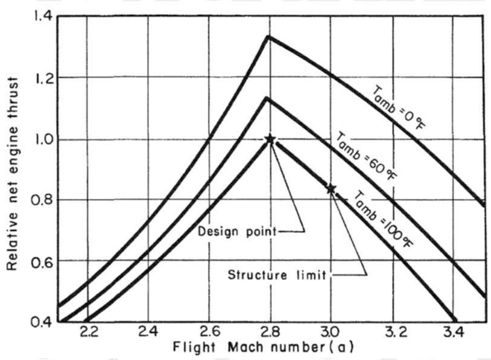


Source: Engineering Design of the TORY II-C Nuclear Ramjet Reactor UCRL-7679, 1964

Air-breathing NTP / Nuclear Ramjet Engine : Project PLUTO / (1957-1964)

TORY II-C Performance Parameters			
Flow Mach Number	2.8	3	2.8
Ambient Temperature (°C)	38	38	-45
Altitude (m)	330	330	330
Reactor Inlet Temperature (°C)	508	573	316
Reactor Inlet Pressure (MPa)	2.22	2.41	2.24
Reactor gas Power (MW)	513	512	633
Reactor Flow Rate (kg/s)	788	845	840
Net Base Thrust (kN)	178	150	273
Max Fuel Element Wall Temperature (°C)	1371	1371	1371
Max Fuel Element Thermal Stress (MPa)	121	121	150
Max Fuel Element Power Density (W/cm^3)	675	673	832
Normal Fuel Element Exit Mach No.	0.443	0.443	0.44
Reactor Pressure Drop (kPa)	676	738	655

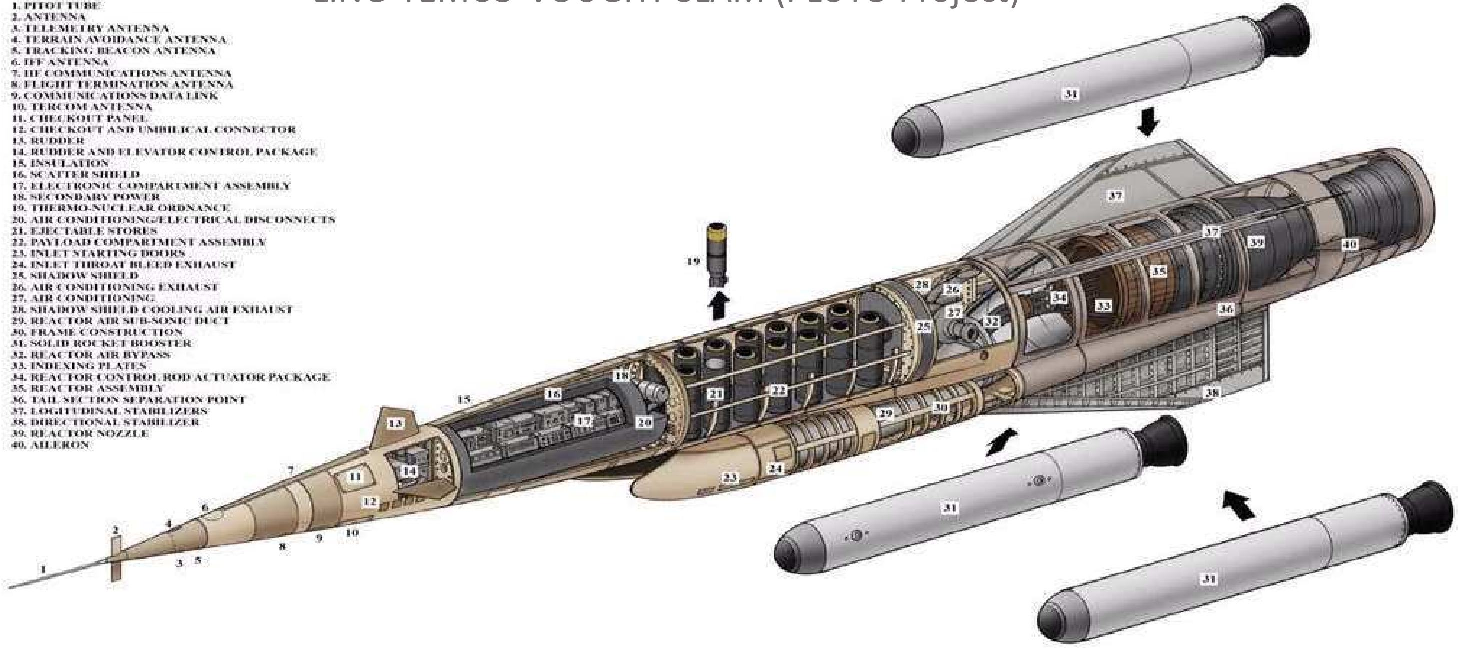
Source: TORY II-C Performance Parameters UCRL-6842-T, 1962



Flow Distribution among Structural Components	
Fuel Elements	79.80%
Unfueled BeO	1.75%
Side Reflector Unfueled BeO	1.93%
Nickel Side Support Shims	1.05%
Tie Rods (Hastelloy)	4.27%
Control Tie Rods (control rods fumlly withdrawn)	3.64%
Side Support Structure	7.43%

LING-TEMCO-VOUGHT SLAM (PLUTO Project)

1. PITOT TUBE
2. ANTENNA
3. TELEMETRY ANTENNA
4. TERRAIN AVOIDANCE ANTENNA
5. TRACKING BEACON ANTENNA
6. IFF ANTENNA
7. HF COMMUNICATIONS ANTENNA
8. FLIGHT TERMINATION ANTENNA
9. COMMUNICATIONS DATA LINK
10. TERCOM ANTENNA
11. CHECKOUT PANEL
12. CHECKOUT AND UMBILICAL CONNECTOR
13. RUDDER
14. RUDDER AND ELEVATOR CONTROL PACKAGE
15. INSULATION
16. SCATTER SHIELD
17. ELECTRONIC COMPARTMENT ASSEMBLY
18. SECONDARY POWER
19. THERMO-NUCLEAR ORDNANCE
20. AIR CONDITIONING/ELECTRICAL DISCONNECTS
21. EJECTABLE STORES
22. PAYLOAD COMPARTMENT ASSEMBLY
23. INLET STARTING DOORS
24. INLET THROAT BLEED EXHAUST
25. SHADOW SHIELD
26. AIR CONDITIONING EXHAUST
27. AIR CONDITIONING
28. SHADOW SHIELD COOLING AIR EXHAUST
29. REACTOR AIR SUB-SONIC DUCT
30. FRAME CONSTRUCTION
31. SOLID ROCKET BOOSTER
32. REACTOR AIR BYPASS
33. INDEXING PLATES
34. REACTOR CONTROL ROD ACTUATOR PACKAGE
35. REACTOR ASSEMBLY
36. TAIL SECTION SEPARATION POINT
37. LOGITUDINAL STABILIZERS
38. DIRECTIONAL STABILIZERS
39. REACTOR NOZZLE
40. AILERON



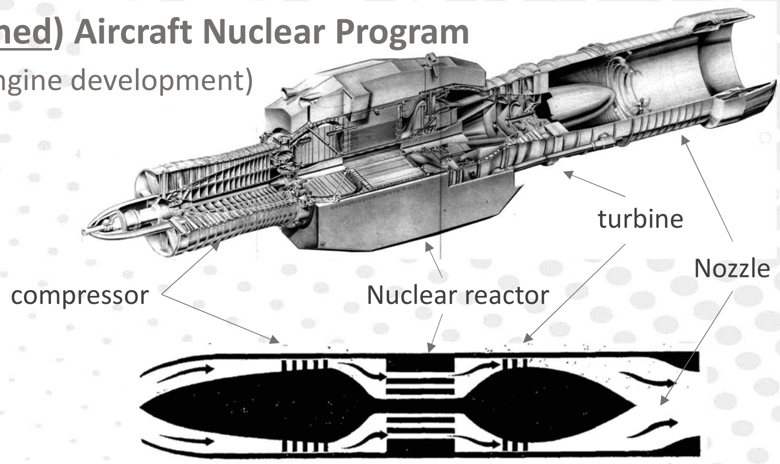
DAMON MORAN TECHNICAL ILLUSTRATIONS 2008

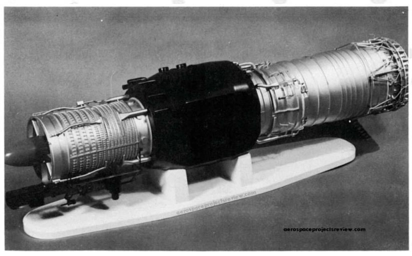
The very last development of the US (Manned) Aircraft Nuclear Program

(1946-1961, US$$_{1950's}$ 24 billions, incl. 2 billions for engine development)

Specifications:

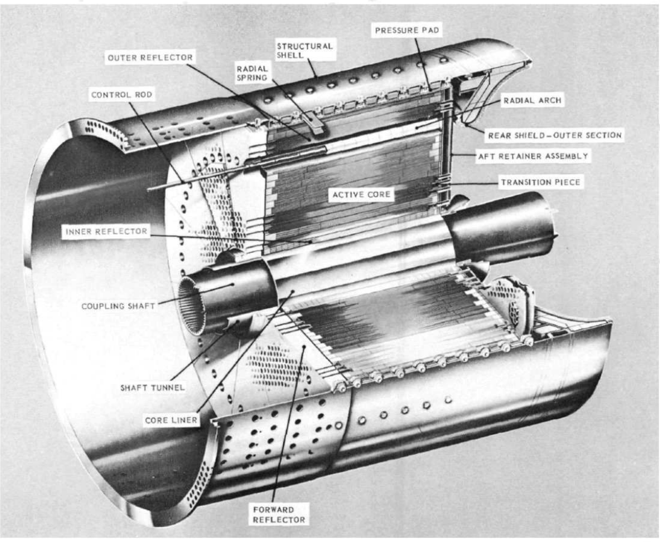
- Mach 0.8 speed at 10 000 m
- Engine life potential: 1 000 hrs
  > 36 kN thrust
- In a Convair NX-2 aircraft
  or equivalent





Source: Comprehensive Technical Report, GE Direct-Air-Cycle ANP Program, XNJ 140E Nuclear Turbojet, Section 4. Reactor, APEX-908 Part B, May 1962

Air-Breathing NTP: GE XNJ 140E-1 Nuclear Turbojet Engine (1960-1961)



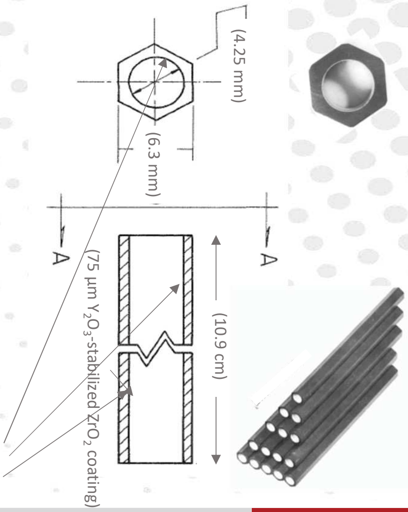
Program cancelled before nuclear testing

Reactor Design Point	
Reactor Power	**50.4 MW**
Reactor / Turbine Inlet T	306 / 949 °C
Fuel Element Peak T	**1388 °C**
Fuel Elements Airflow fraction	84%
Mach No. Fuel Inlet / Outlet	0.121 / 0.214
Inner Al_2O_3 Reflector ID / Thickness	34.3 / 4.7 cm
Active core ID / OD	**43.8 / 114.5 cm**
Outer BeO Reflector Thickness	21.3 cm
Outer BeO Reflector OD / Thickness	157.4 / 21.3 cm
Over-all Diameter w/o neutron shield	167.6 cm
LiH Neutron Shield Thickness	47.8 cm
Front Borated BeO/SS Shield Length	68.0 cm
Front Be Reflector Length	8.2 cm
Active Core Length	**76.2 cm**
Rear BeO Reflector Length	3.8 cm
Rear Borated BeO Shield Length	62.2 cm
Over-all Length	263.0 cm
93% U5 Uranium Mass	118 kg
Total Weight w/o Shield	**5635 Kg**

> 1 000 hrs reactor operating lifetime
(BeO subject to water-vapor corrosion)

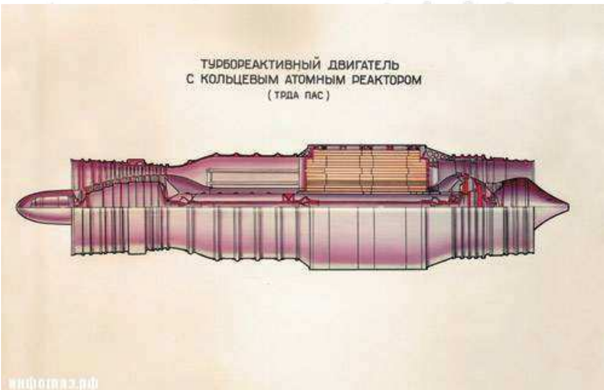
170 000 Fuel Elements
(25 000 airflow passages)
Y_2O_3-stabilized BeO + 4-10 Wt% UO_2
(UO_2: 8.5 Wt% average)
118 kg UO_2

Fuel element similar to Tory's

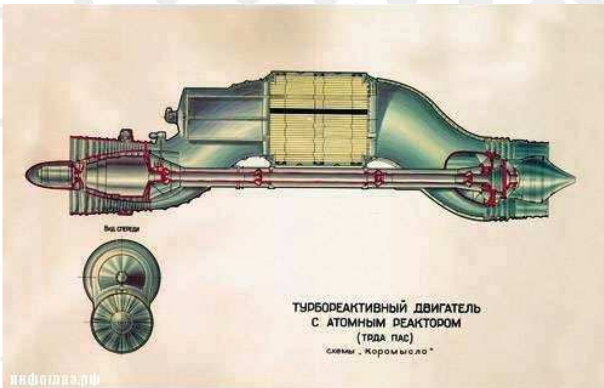


Source: Comprehensive Technical Report, GE Direct-Air-Cycle ANP Program, XNJ 140E Nuclear Turbojet, Section 4. Reactor, APEX-908 Part B, May 1962

USSR Nuclear Turboreactor Program

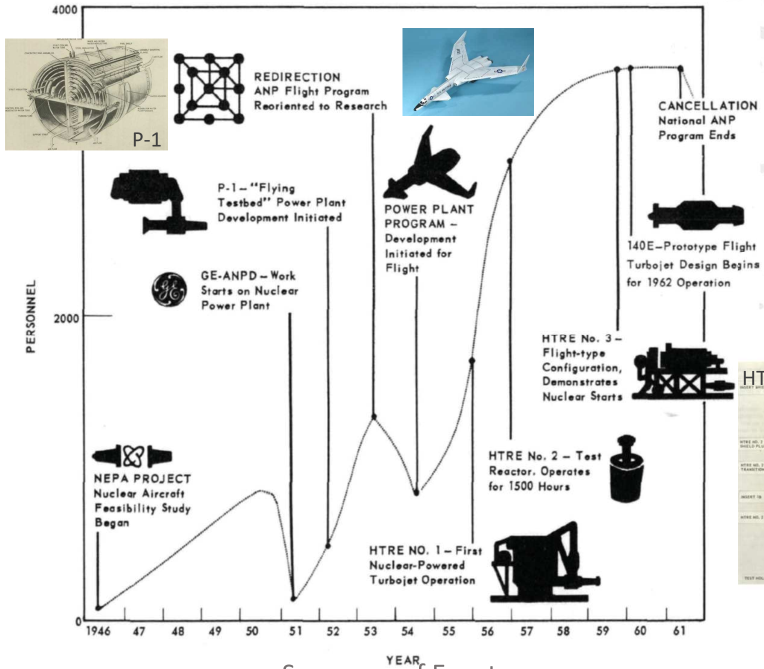


Annular Shaft-Axis-Symmetrical Nuclear Reactor

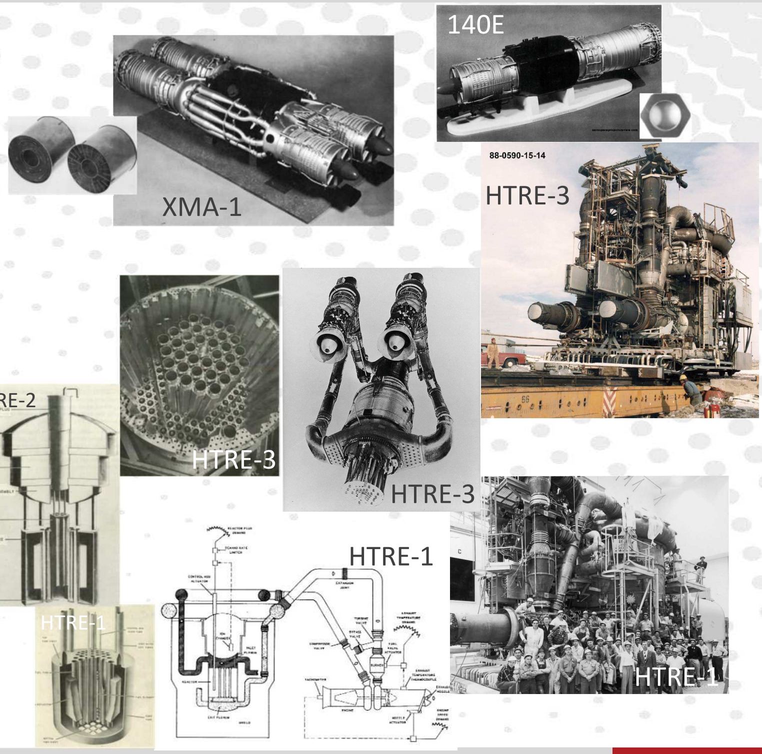


Off-Shaft-Axis Nuclear Reactor

US (Manned) Aircraft Nuclear Propulsion Program (1946-1961)

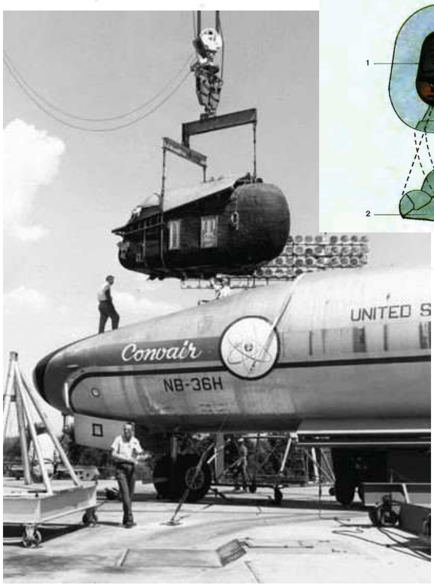


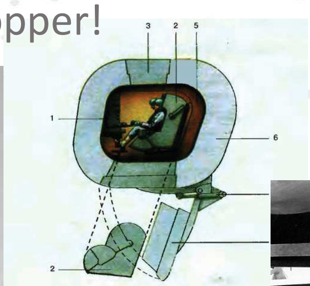
Summary of Events:
General Electric Aircraft Nuclear Propulsion Program



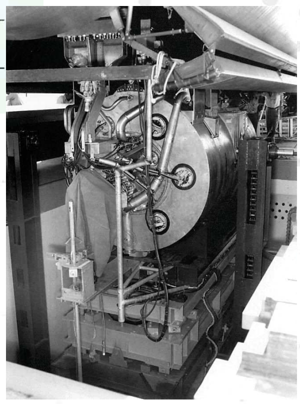
US (Manned) Aircraft Nuclear Propulsion Program (1946-1961)

The show stopper!





The shielded cockpit



Aircraft Shield Test Reactor (ASTR)

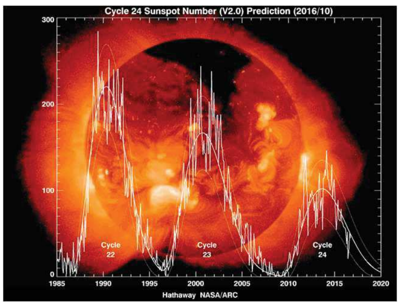
The only nuclear reactor to have flown!
(with its USSR's equivalent)

The NB-36H Nuclear Test Aircraft (NTA)

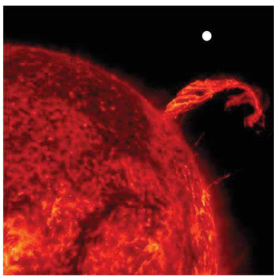
Back-up slides

- **Space Radiation Environment, Radiation levels vs Career exposure limits for NASA astronauts**
- **Radiation Shielding from Nuclear Thermal Engines**
- **The Context and Stakes of Switching from HEU to LEU Fuel for NTP**
- **Miscellaneous**
  - A Nuclear Thermal Propulsion Third Stage for the SATURN Heavy Launcher?
  - Why is NTP attractive for human missions to Mars?
  - Possible Turbopump Cycles for NTP Engines
  - Rover/NERVA Overall Program Budget
  - Properties of candidate moderators & reflectors for NTP
  - Xenon Effect in ''Thermal Spectrum'' Nuclear Rocket Engines
  - Typical Characteristics of the Nuclear Rocket Engine Startup
  - Nuclear Bi-Modal Thermal Propulsion + Payload Power Supply / Electric Propulsion

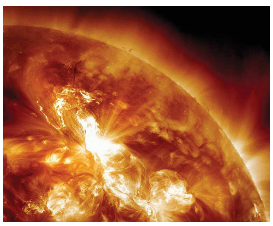
Space Radiation Environment



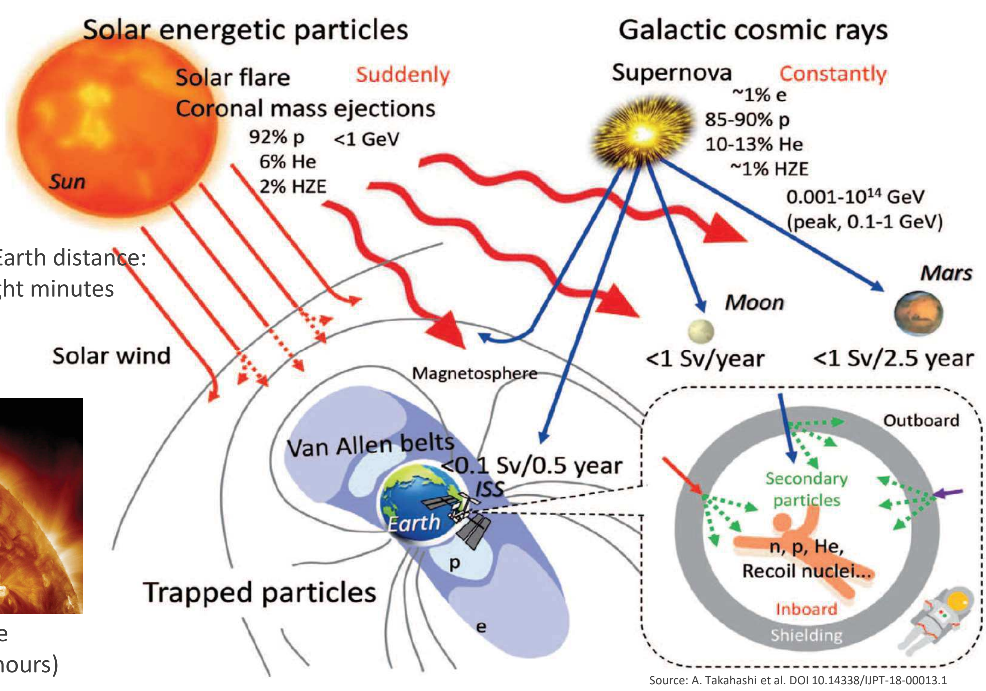
Solar activity cycle (~11 y)



Coronal mass ejection
(lasts several days)



Solar flare
(lasts several hours)



Source: A. Takahashi et al. DOI 10.14338/IJPT-18-00013.1

Radiation levels vs Career exposure limits for NASA astronauts

Annual Ambient Levels for the Earth, Mars and Space

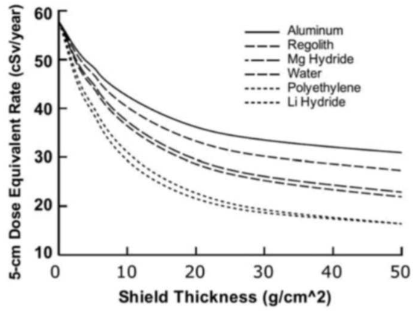
	Earth	Mars	Moon	Space
Annual Total	3 mSv	245 mSv	438 mSv	657 mSv
Daily Average	$8.2\ 10^{-3}$ mSv	0.67 mSv	1.2 mSv	1.8 mSv

Source: L. Joseph Parker, Human radiation exposure tolerance and expected exposure during colonization of the moon and Mars, 2016

Career Exposure Limits for NASA Astronauts				
Age (years)	25	35	45	55
Male	1.50 Sv	2.50 Sv	3.25 Sv	4.00 Sv
Female	1.00 Sv	1.75 Sv	2.50 Sv	3.00 Sv

The NASA astronaut career depth equivalent dose limit is based upon a maximum 3% lifetime excess risk of cancer mortality

Depth of Radiation Penetration and Exposure Limits for Astronauts and the General Public (in Sv)				
	Exposure Interval	Blood Forming Organs (5 cm depth)	Eyes (0.3 cm depth)	Skin (0.01 cm depth)
Astronauts	30 Days	0.25	1.0	1.5
	Annual	0.50	2.0	3.0
	Career	1-4	4.0	6.0
General Public	Annual	0.001	0.015	0.05

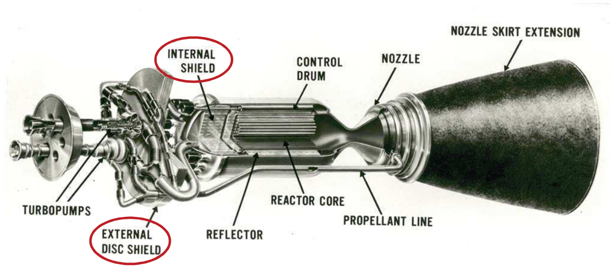


(poor) Shielding effectiveness against galactic cosmic radiation at solar minimum

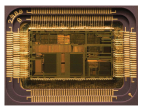
Mission Type	Radiation Dose
Space Shuttle Mission 41-C (8-day mission orbiting the Earth at 460 km)	5.59 mSv
Apollo 14 (9-day mission to the Moon)	11.4 mSv
Skylab 4 (87-day mission orbiting the Earth at 473 km)	178 mSv
ISS Mission (up to 6 months orbiting Earth at 353 km)	160 mSv
Estimated Mars mission (3 years)	1,200 mSv

Source: Space Faring – The Radiation Challenge. NASA, EP-2008-08-116-MSFC

Radiation Shielding from Nuclear Thermal Engines

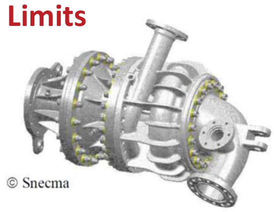


Material Dose Limits


ASIC: 10^5 Rad

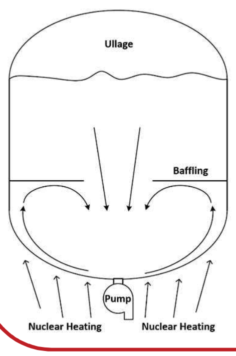

FPGA: 10^4 Rad

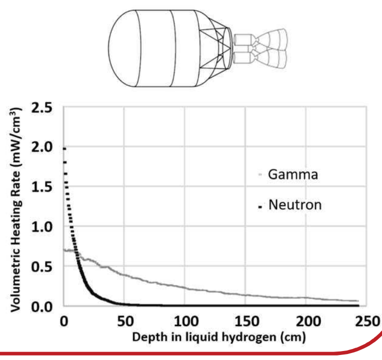

Stepper Motors: 10^9 Rad


© Snecma

LH Turbopumps: ?? Rad

Heat Deposition Limits in Cryotanks



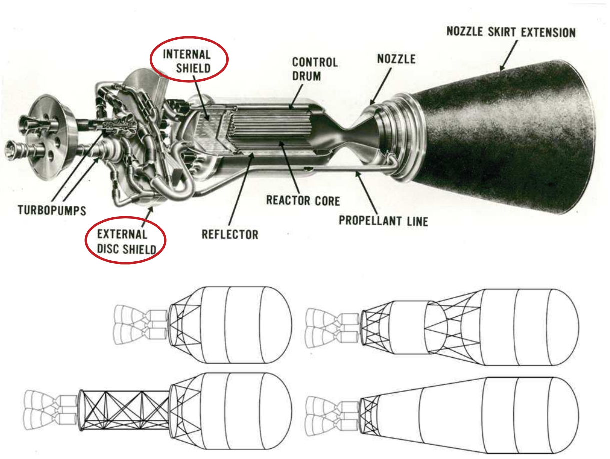


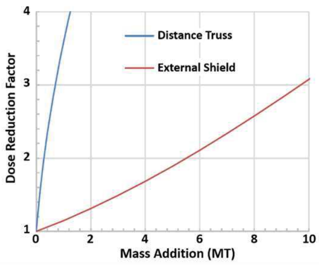
Human Dose Limits

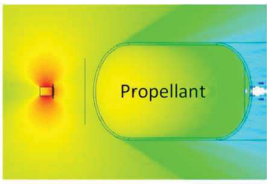
Exposure Limits for Astronauts and the General Public (in Sv)				
	Exposure Interval	Blood Forming Organs (5 cm depth)	Eyes (0.3 cm depth)	Skin (0.01 cm depth)
Astronauts	30 Days	0.25	1.0	1.5
	Annual	0.5	2.0	3.0
	Career	1-4	4.0	6.0
General Public	Annual	0.001	0.015	0.05

Sources include: Space Faring – The Radiation Challenge. NASA, EP-2008-08-116-MSFC;
B. D. Taylor et al., Cryogenic Fluid Management Technology Development for Nuclear Thermal Propulsion, AIAA 2015-3957

Radiation Shielding from Nuclear Thermal Engines

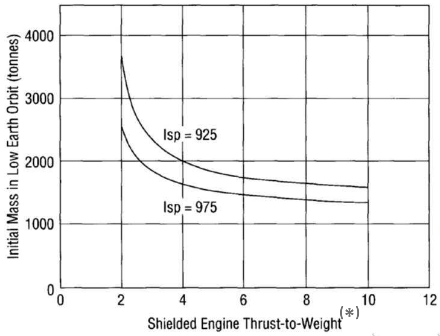






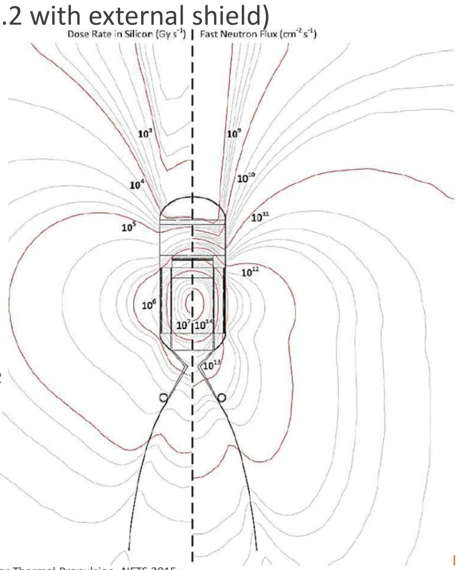
External shield mass may reach 50% of (unshielded) engine mass

Shielded engine Thrust to Weight ratio (T/W) impacts performance (NERVA-Derived 100 kN ENABLER: T/W=4.8(*) w/o ext. shield; could decrease down to 3.2 with external shield)



2016, 365 Day Mission to Mars with 30 Day Stay

(*) *T/W expressed in lbf/lbm (= N/(g_0 kg))*

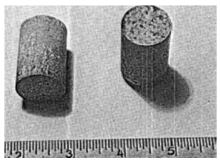


Sources of graphs include: Javis A. Caffrey, Shielding Development for Nuclear Thermal Propulsion, NETS 2015;
H. Ludewig et al., Design of Particle Bed Reactors For The Space Nuclear Thermal Propulsion Program, Progress. In Nuclear. Energy, Vol 30, 1996

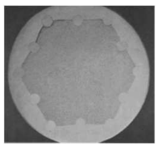
Radiation Shielding from Nuclear Thermal Rocket Engines

Neutron Shield Materials

LiH: ☺ the most effective per unit mass: H density 90% of room T water, absorption by ^{6}Li: σ_{th}=940 barns, 7.5% in Li nat
☺ neutron capture does not emit gammas: $^6Li + n \rightarrow\ ^4He\ (2.05\ MeV) +\ ^3_1T\ (2.75\ MeV)$
☺ 9 SNAP shadow shield fabricated (cast), developed for SP-100 (cold-pressed)
☹ narrow operating temperature range: [600 – 800] K and poor thermal conductivity: 4-5 W/mK
> 600 K to prevent unacceptable irradiation swelling)
< 800 K to prevent unacceptable thermally-induced dissociation/swelling
☹ chemically unstable (pyrophoric) in oxidizing atmospheres and 23% volume expansion at melting



B_4C: ☹ mass penalty >20% (90% ^{10}B) up to > 300% (Nat B) compared to ''practical LiH shield''
C density 25% that of graphite; absorption by ^{10}B: σ_{th}=3800 barns, 20% in B nat
☺ minimal production de secondary gammas by neutron capture (^{10}B(n, α1,γ))
☺ excellent thermal conductivity and chemical stability, currently fabricated in large quantities
☹ cost of ^{10}B enrichment
😐 mass reduction by combining B_4C with **Be** (neutron moderator) in a multilayer sandwich design



BATH: developed for the internal Shield of NERVA-derived engines (Al 70w%, $TiH_{1.8}$ 30w%, B_4C 5w%)

Gamma Shield Materials

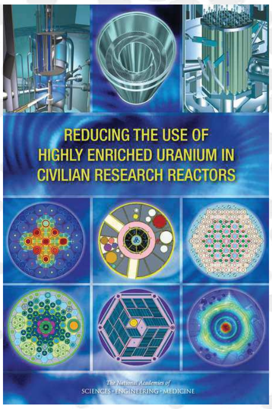
Pb: ☺ the most effective per unit mass (except U); ☺ inexpensive; ☹ 600 K melting point
W alloy: ☺ effectiveness per unit mass comparable to Pb, 30% high than Fe; ☹ cost; ☺ high strength at high temperature
W + 8% B_4C (90% ^{10}B) to reduce secondary gammas: improve mass effectiveness

Sources include: The Evaluation of Lithium Hydride for Use in a Space Nuclear Reactor Shield, KAPL, Inc. Report MDO-723-0048, December 9, 2005; Javis A. Caffrey, Shielding Development for Nuclear Thermal Propulsion, NETS 2015

The Context and Stakes of Switching from HEU to LEU Fuel for NTP (1/3)

The political non-proliferation context

Long standing commitment of The United States to eliminate (to the extent possible) the use of HEU in all civilian applications, including in the production of medical radioisotopes, because of its direct significance for potential use in nuclear weapons, acts of nuclear terrorism, or other malevolent purposes



The Reduced Enrichment for Research and Test Reactors (RERTR) program, initiated in 1978 by the US DOE: an international effort to support ''**the minimization and, *to the extent possible*, elimination of the use of HEU in civil nuclear applications** by working to convert research reactors and radioisotope production processes to the use of LEU fuel and targets throughout the world''

To reduce penalty to switch to LEU, development and qualification of:

- existing fuels with increased U density: UAl_x-Al dispersion fuel (1.7 to 2.3 gU/cm^3), U_3O_8-Al dispersion fuel (1.3 to 3.2 gU/cm^3), $UZrH_x$ alloy fuel (0.5 to 3.7 gU/cm^3)
- new fuel: U_3Si_2-Al dispersion fuel (qualified at 4.8 gU/cm^3)

Since 1978, more than 70 **civilian research reactors** have been converted from HEU to LEU (> 20% ^{235}U) and ~30 additional civil reactors that used HEU have been verified as shutdown.
Since 1980, more than 20 large (>1 MW) new research reactors have been designed to use LEU fuel

+ development of targets and processes for the production of the medical isotope Molybdenum-99 with LEU

The Context and Stakes of Switching from HEU to LEU Fuel for NTP (2/3)

Minimization and, <u>to the extent possible</u>, elimination of the use of HEU in civil nuclear applications:

In 1986, new U.S. NRC regulation, 10 CFR 50.64, which places limitations on the use of HEU in nonpower reactors: "The Commission will not issue a construction permit after March 27, 1986 for a non-power reactor where the applicant proposes to use highly enriched uranium (HEU) fuel, **unless the applicant demonstrates that the proposed reactor will have a unique purpose''** (= ''a project, program, or commercial activity which cannot reasonably be accomplished without the use of HEU fuel'')

In the US, eight civilian research and test reactors continue to use HEU since an alternative fuel has not yet been developed for their conversion.

The current U.S. policy on the use of HEU in reactor systems endorses the use within naval vessels.
There is currently no U.S. policy on the use of HEU in space nuclear reactors.

The use of HEU in highly specialized systems such as space power reactors and propulsion systems must be balanced with the potential risks associated with the proposed mission

The Context and Stakes of Switching from HEU to LEU Fuel for NTP (3/3)

The US political economic context

SpaceX: the demonstration that switching from government to private development of launchers is a successful and cost-efficient policy

A policy of public-private partnerships for space transportation and its ''return humans on lunar surface'' strategy, and encouragement of commercial space activities

The issues

For a commercial space nuclear propulsion effort, LEU is probably the only option

A commercial development effort with LEU could prove to be cheaper:

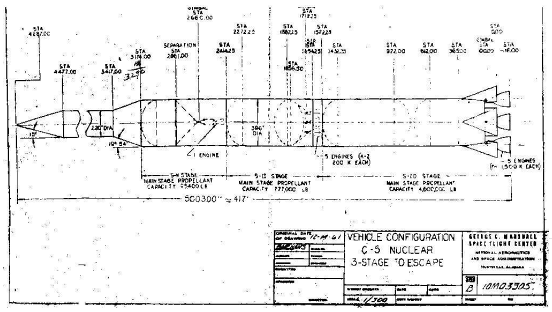
- Reduction of security risks
- Benefits of commercial effort (cf. SpaceX)

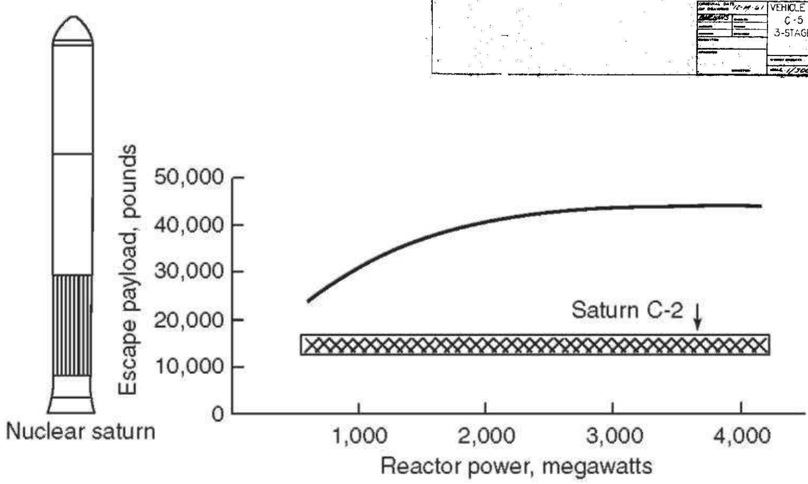
HEU: Political risk of cancelation due to controversy over the use of nuclear weapons-grade fuel

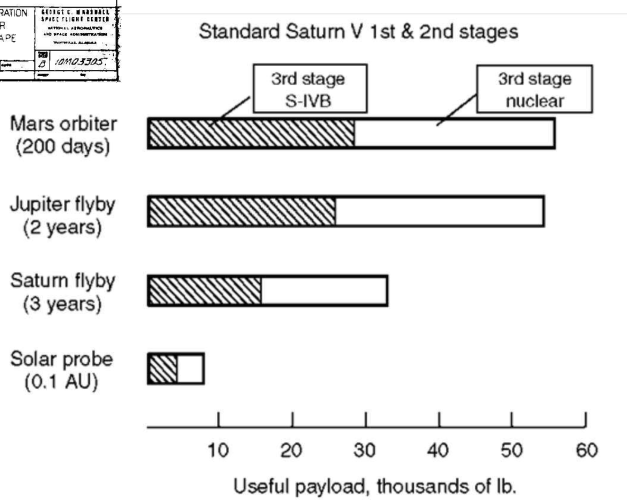
? Penalty on performances (mass) of switching from HEU to LEU?

? Can the cost increase of launching a heavier reactor be offset by the above cost reductions?

A Nuclear Thermal Propulsion Third Stage for the SATURN Heavy Launcher?



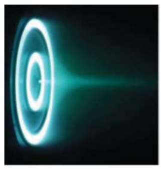


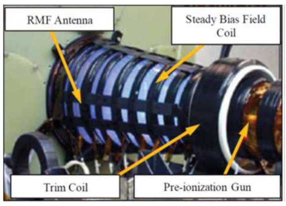


High-Power Electric Space Propulsion Engines: Estimate Performances

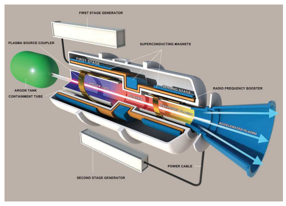
Estimated performances of high-power propulsion scaled to 200 kWe

	Concentric Channel HET (3 channels)	NASA-457M Cluster (3 thrusters)	ELF-375 (200-kW design goals)	VASIMR VX-200 (design goals)
Input Power	200 kW	200 kW (3 devices at 67-kW)	200 kW	200 kW (2 devices at 100-kW)
Specific Impulse	1300 – 5000 s	3000 s	1500 – 5000 s	5000 s
Thrust	5 – 14 N (25 – 70 mN/kW)	8.4 N (42 mN/kW)	7 – 18 N (35 – 95 mN/kW)	5 N (25 mN/kW)
Mass Flow Rate	100 – 1100 mg/s (Xe)	280 mg/s (Xe)	140 - 1200 mg/s (Xe)	130 mg/s (Ar)
Efficiency ε	45% – 64%	63%	65% – 85%	60%
Specific Mass	0.5 kg/kW (thruster) 1.4 kg/kW (thruster, PPU)	1.3 kg/kW (thruster[4]) 2.2 kg/kW (thruster, PPU)	0.25 kg/kW (thruster) 0.7 kg/kW (thruster, PPU)	1.5 kg/kW (thruster[5])
Major Thruster Dimensions	0.65-m diameter 0.10-m length	0.55-m by 1.6-m 0.15-m length	0.38-m diameter 0.5 meter length	1.5-meter diameter 3.0 meter length


Concentric channel HET



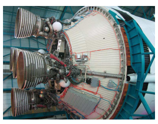
ELF device

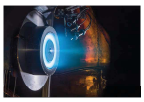

VASIMR

HET: Hall Effect Thruster, NASA-457M: Hall Effect Thruster
ELF: Electrodeless Lorentz Force (ELF) thruster
VASIMR: Variable Specific-Impulse Magnetoplasma thruster

Source: Air Force Research Laboratory High Power Electric Propulsion Technology Development, Daniel L. Brown, Brian E. Beal, James M. Haas, 2010 IEEE Aerospace Conference (2010)

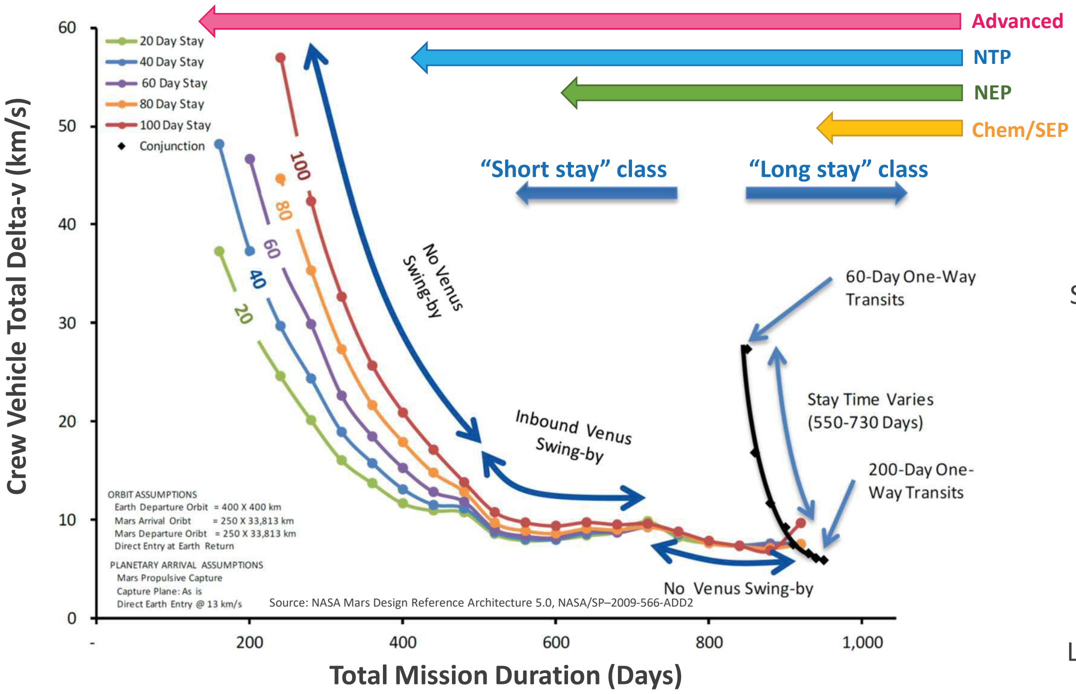
Comparing Chemical, Nuclear Thermal and Electric Propulsion

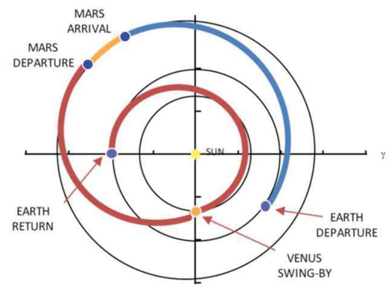




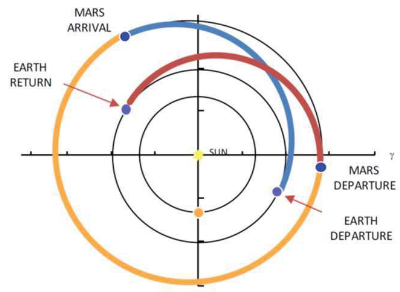
	Type of propulsion		
	Chemical (SSME)	NTP	Ion NEP
Propellant	$LH_2 + LO_2$	LH_2	Xe
I_{sp}(s)	453	800-900	6 000-8 000
Thrust (kN)	2 200	100 - 1000	0.005 - 0.05
Time of single burn (s)	480	~3600	years

Why is NTP attractive for human missions to Mars?



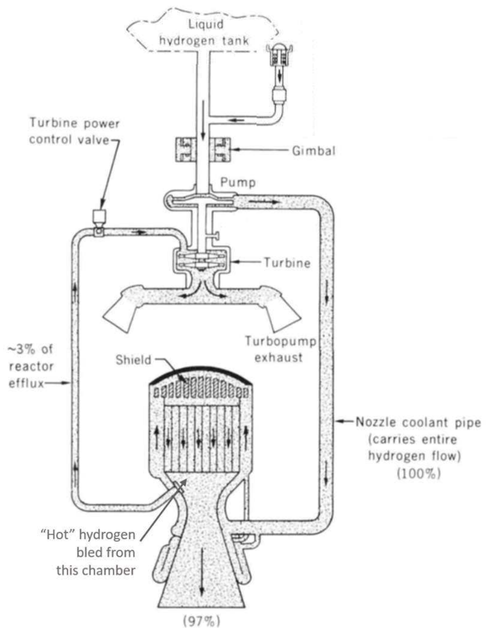


Short stay (opposition) class missions

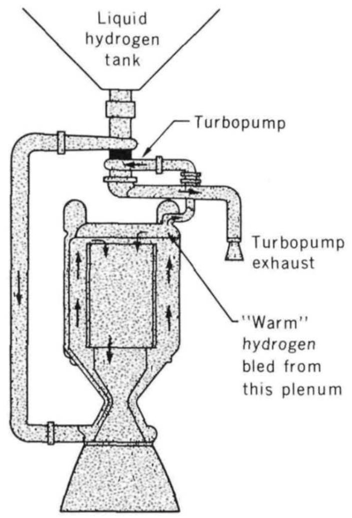


Long stay (conjunction) class missions

Possible Turbopump Cycles for Nuclear Thermal Rocket Engines

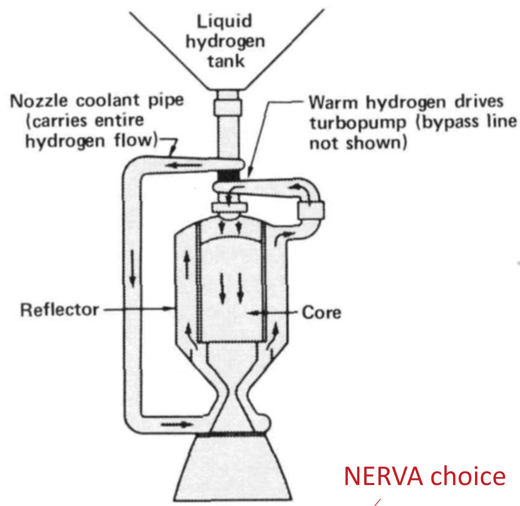


"Hot Bleed" Cycle
Small lightweight high T turbine
3% H_2 flow wasted = ~25s Isp lost



"Cold Bleed" Cycle
More massive lower T Turbine + Isp loss

Cold start bootstrapping on tank pressure

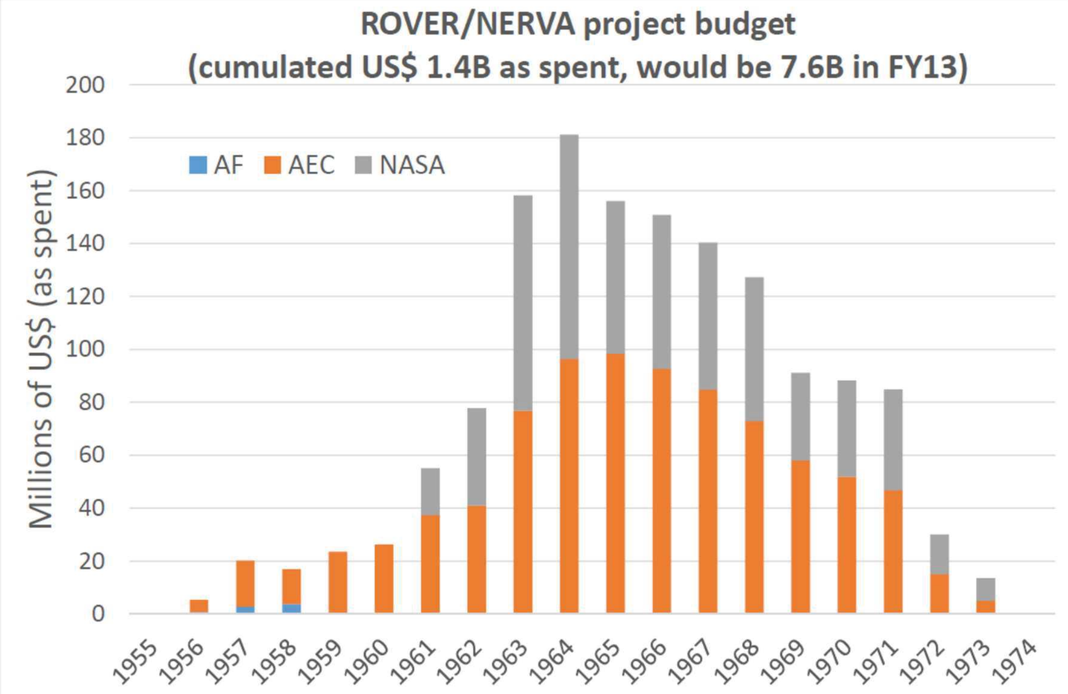


NERVA choice

"Topping" / "Expander" Cycle
100% flow more massive lower T turbine
No wasted flow/ Highest Isp

Source: William R. Corliss and Francis C. Schwenk, Nuclear Propulsion for Space, USAEC-DTI 1968; 1971 (rev.)

Rover/NERVA Overall Program Budget



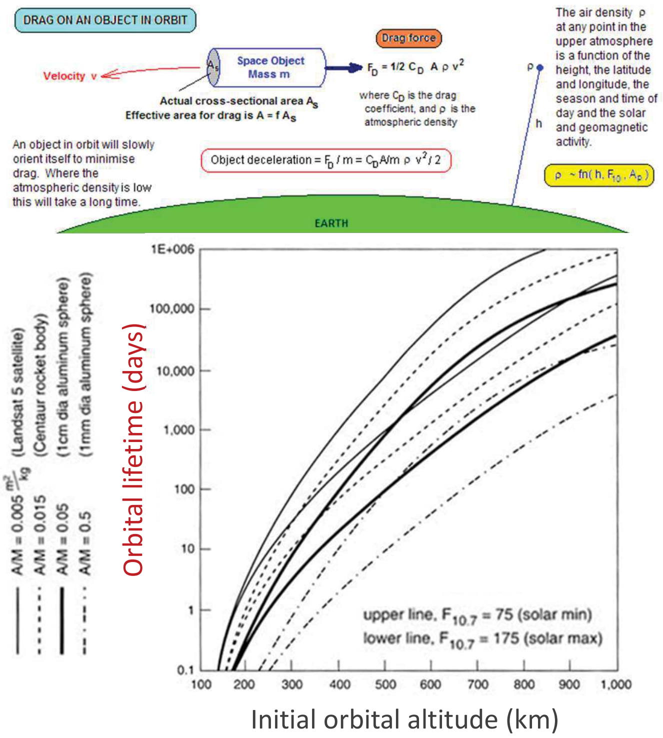
Source: Nuclear Thermal Rockets, Lecture 24, G. L. Kulcinski, March 22,2004

Properties of candidate moderators & reflectors for NTP

Properties of Moderator and Reflector candidates for Nuclear Thermal Rockets

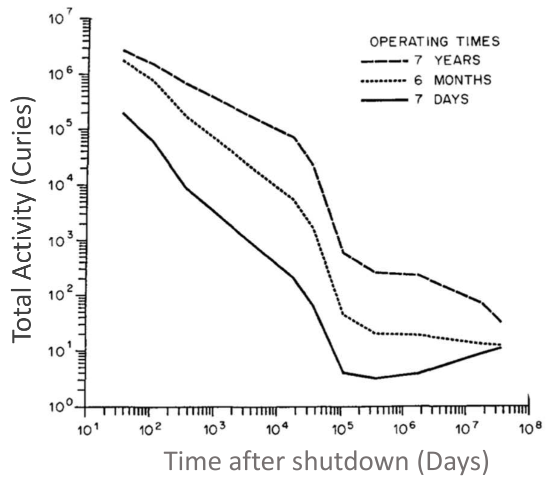
Candidates	C/C	^{7}LiH	$ZrH_{1.8}$	Be
Density (g/cm^3)	~1.98	0.77	5.65	1.85
Melting point (K)	3 923	962	1 073	1 560
Tensile strength (MPa)	~700	27.6	~800	395
Thermal expansion (10^{-6}/K)	0~1	35.2	27	11.6
Thermal conductivity ([W/(m	350	7.5	17	201
Slowing down power (cm^{-1})	0.06	3	2.9	0.16
Moderating ratio	220	127	110	138

A "Nuclear Safe Orbit"?



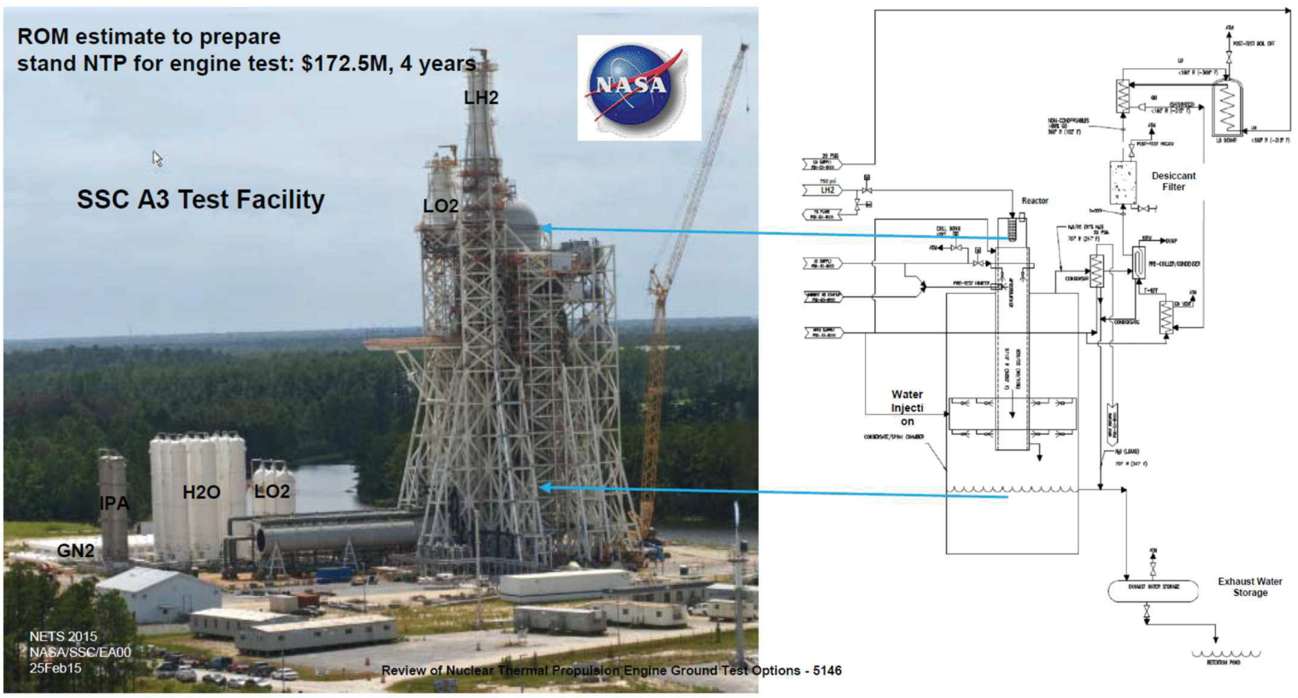
A "Nuclear Safe Orbit":
a (typically 1000 km or higher) orbit providing an unattended orbital life of sufficient lifetime (typically 10 000 y or more) so that the core's radioactive nuclide inventory will have decayed down to ''acceptable'' levels

Typical 300 kWe SNPS core activity decay



Test Facility for Nuclear Ground Testing with Exhaust Capture

''Most of the infrastructure required for ground test facility (including exhaust capture) in already in place''



Xenon Effect in "Thermal Spectrum" Nuclear Rocket Engines

Typical Conjunction Class human Mars NTP mission outline

Mission Phase	Engine state	Duration
Trans-Mars Injection 1 (TMI 1)	Full Thrust	25 min
Waiting in an Elliptical Orbit around Earth	Idle	5 hours
Trans-Mars Injection 2 (TMI 2)	Full Thrust	25 min
Transit to Mars	Idle	200 days
Martian Orbit Injection (MOI)	Full Thrust	12 min
Surface Operations on Mars	Idle	500 days
Trans-Earth Injection (TEI)	Full Thrust	9 min
Transit to Earth	Idle	200 days

$$^{135}_{52}Te \xrightarrow[19.0\ sec]{\beta^-} {}^{135}_{53}I \xrightarrow[6.57\ hr]{\beta^-} {}^{135}_{54}Xe \xrightarrow[9.10\ hr]{\beta^-} {}^{135}_{55}Cs \xrightarrow[2.3\times10^6\ yr]{\beta^-} {}^{135}_{56}Ba\ \text{(stable)}$$

6% fission yield

Thermal absorption XS: ~2.6 10^6 barns
(^{235}U fission XS: ~580 barns at 0.025 eV)

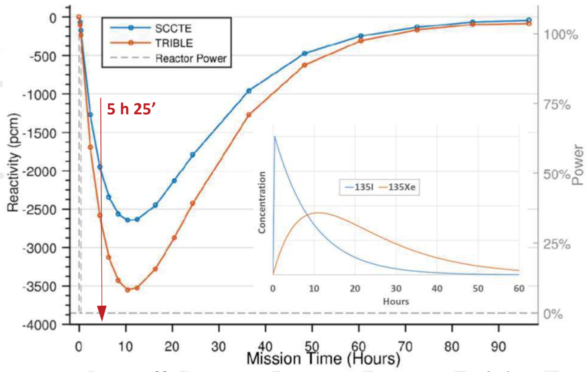
Marginal Xe build up during 25 mn burn (high neutron flux)
Xe builds up during dwell-time, driven by ^{135}I decay

^{135}I build-up during burn time: $$N_i(t) = \frac{\gamma_I \Sigma_f^{fuel} \phi}{\lambda_I}\left(1 - e^{-\lambda_i t}\right)$$

ϕ: ~5 10^{14} n.cm^{-2}.s^{-1} (E<0.65 eV)

$\left(1 - e^{-\lambda_i t}\right)$: 4% at 25 min

Xe antireactivity needs to be compensated by control drum rotation
Control drum reactivity worth is usually sufficient but …
⇨ increased radial neutron reflection which changes the radial power profile/ location of the hot channel
⇨ some loss of performances (Isp)



Xenon effect on LEU CERMET conceptual designs

Effect present in HEU NERVA engines, HEU->LEU significantly increases it (higher neutron flux)

Source of table and figure: Michael J. Eades, 135Xe in LEU Cermet Nuclear Thermal Propulsion Systems, PhD Dissertation, Ohio State University (2016)

Typical Characteristics of the Nuclear Rocket Engine Startup

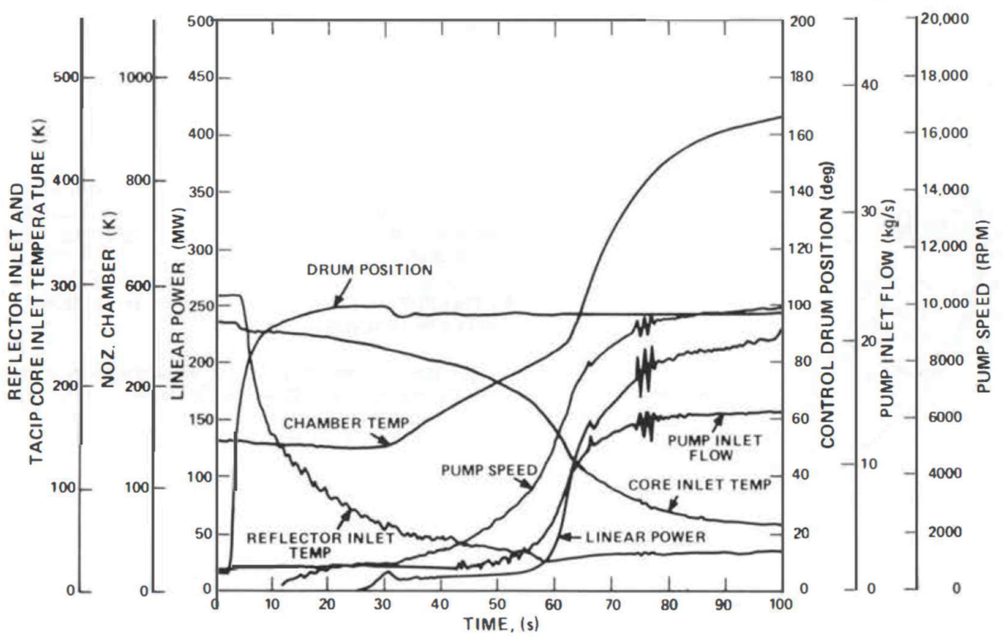
An NTR needs to come to full power very quickly after the onset of hydrogen flow, or the wasted hydrogen will significantly reduce the Isp of the system

Chill-down of the various engine components takes ~60 s.

The engine can then be turned on to full power et a rate limited by thermal stresses in the core resulting from the transient.

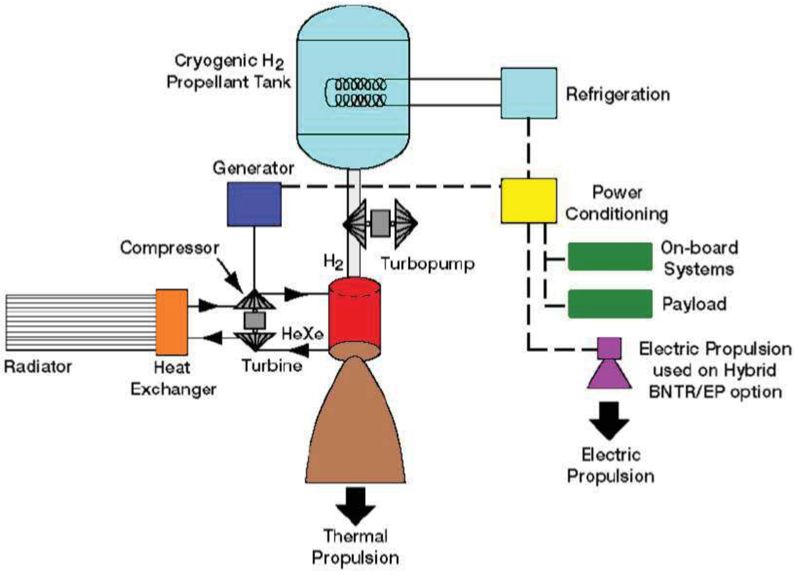
For NERVA-type engines, the rate of core temperature raise was not to exceed 83 K/s.

Temperature and H_2 reactivity feedbacks during the transient depend on the engine design concept



Source: Daniel R. Koenig, Experience Gained from the Space Nuclear Rocket Program (Rover), LANL Report LA-10062-H (1986)

Nuclear Bi-Modal Thermal Propulsion + Payload Power Supply / Electric Propulsion



- During short, high thrust propulsion phase, each BNTR produces ~340 MW_t and ~15 klb_f of thrust
- During long, power generation phase, each BNTR operates in "idle mode" producing just ~150 kW_t
- A Brayton conversion unit on each BNTR produces up to 25 kW_e to enhance stage capabilities

Lecture Series on SPACE NUCLEAR POWER & PROPULSION SYSTEMS

Coming next: -3- Space Fission Power & Electric Propulsion Systems

Author: Eric PROUST email: firstname.lastname@cea.fr