A STUDY FOR MARS MANNED EXPLORATION

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Overview

Over the last five decades there have been numerous studies devoted to developing, launching and conducting a manned mission to Mars by both Russian and U.S. organizations. These studies have proposed various crew sizes, mission length, propulsion systems, habitation modules, and scientific goals. As a first step towards establishing an international partnership approach to a Mars mission, the most recent Russian concepts are explored and then compared to NASA's latest Mars reference mission.

Current Concepts

This first section explores the latest Russian concepts. Data for the conceptual Mars mission were obtained or derived from Refs. (1; 2), with supporting data obtained from Refs. (3; 4). Data from these sources were used to construct the overall mission parameters, as shown in Table 1.

Payload required in LEO	500-600 mT
Total mission duration	~2 years
Crew size	6
Interplanetary engine thrust	140-170 N
Total Power (input)	15 MW (thermal) – 2.25 MW (electric)

Table 1. Mars mission parameters based on the most recent Russian concepts.

Based on the same references it is also assumed that Hall-type thrusters will be used for the orbital maneuvers, station keeping and interplanetary thrust. The two propellants commonly used in Russian (and U.S.) Hall thrusters are xenon and bismuth. Recently, bismuth has become an attractive propellant due to its high density, low cost, condensability at room temperature, low ionization potential and high atomic mass (5). Reference (1) was used as the basis for determining performance values of state of the art Hall thrusters (assuming bismuth as the propellant), with the results being shown in Table 2. The values in Table 2 are consistent with the values in Refs. (5; 6; 7; 8). Mission masses will be given for both bismuth and xenon powered thrusters.

Efficiency	60%
Discharge Power (kW)	100 (340,000 BTU/hr)
Thrust (N)	6.96-8.53 (1.57-1.92 lbf)
Specific Impulse (Isp)	1000-2000
Mass flow rate (mg/sec)	250

Table 2. Hall thruster performance values (mass flow rate based on Bismuth).

In the current Russian concepts the engines would produce 140-170 N of thrust, depending on the operating mode. This value is lower than those stated in previous Russian concepts, which varied between 300 N (Ref. (2)) and 441 N (Ref. (9)). Based on Table 2 approximately 24 thrusters (20 main plus 4 spare/redundant, in pods of 12 thrusters each) would be needed to generate the required thrust. A cluster of thrusters has the inherent advantage of redundancy in the event of an individual thruster failure. Each thruster would have an average diameter of 1100 mm (~43 inches), resulting in a total thruster area of at least 22.8 m^2 (246 ft^2) (6). The total mass of the thrusters would be on the order of **10 mT** (at 400 kg per thruster, or 4 kg/kW) (5). The (20 operating) thrusters would produce 2.0 MW (electric) discharge power. The thrusters would be powered by either nuclear reactor(s) or solar panels. Two notes of interest: 1) to date a maximum of 4 thrusters have been run as a cluster, and 2) potential concerns with clusters of Hall thrusters include oscillations and non-linear effects in the plumes (10).

The 2-year mission to Mars will spend approximately 686 days in transit. The 686 days can be broken down into powered and unpowered segments. The powered segments include spiraling out of the planetary gravity wells, reaching interplanetary trajectory velocities, braking maneuvers and spiraling into planetary orbits. The total length of the powered segments is approximately 166 days (9). This operating time, 3984 hours, is well within the tested limits of Hall thrusters. The unpowered segments consist of coasting or station keeping, and total 520 days. Thus, assuming a conservative mass flow rate of 250 mg/sec the total Bismuth propellant needs would be:

166 days x 24 hrs/day x 3600 sec/hr x 250 mg/sec-thruster x 20 thrusters x 1e-6 kg/mg = 71,712 kg (1)

Thus, before the inclusion of flight performance reserves (FPR) and station keeping, approximately **72 mT** of bismuth propellant would be needed. In 2006 it was determined that 20 mT of bismuth would cost approximately $1.5 M and require a tank volume of 2 m^3 (5). Converting the cost to 2010 dollars requires two assumptions. First, the price of bismuth increased by a factor of 3 between the fall of 2006 and the fall of 2007 (mainly because it is being used to replace lead in many applications), then dropped again as demand went down with a downturn in the global economy (see Fig. 1) (11). It will be assumed that the price of bismuth will level off at approximately twice its value of 2006. Second, a common price index of 1.077 will be assumed (12). Therefore, in 2010 dollars the 72 mT of bismuth would cost approximately **$11.6 M** and require a tank with a volume of **7.2 m^3**. Note that 7.2 m^3 (254 ft^3) is the size of a standard work cube!

If xenon was used as the propellant in place of bismuth, a total of approximately **183 mT** of propellant would be needed (due to higher mass flow rates of ~500 mg/sec), at cost of approximately **$291 M** and a tank volume of almost **37.7 m^3** (1331 ft^3) (5). Xenon is a gas at room temperature, and has a boiling point of 165 degrees Kelvin. Xenon can be a supercritical fluid between the boiling point and about 290 degrees Kelvin, depending on the pressure. The tank cooling requirements for xenon would be similar to those of liquid oxygen.

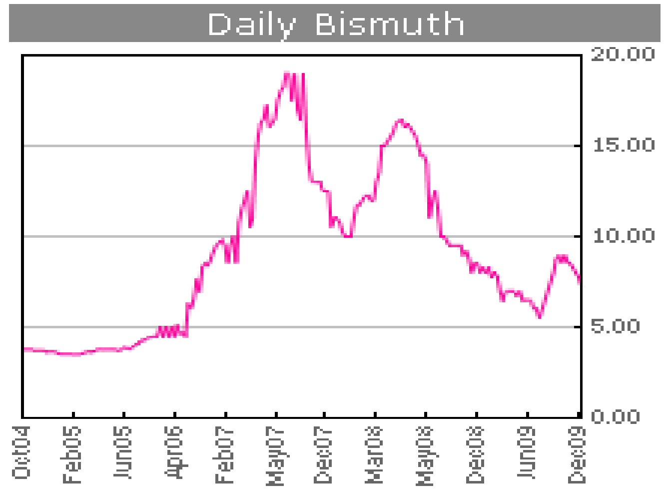

Figure 1. Historical price of bismuth in dollars/lb (Ref. 9). Note, $10/lb is approximately $22/kg.

Some of the potential issues associated with increasing the thrust and power of Hall thrusters to reach the levels shown in Tables 1 and 2 include (6; 10):

1. Retaining an azimuthally uniform magnetic field/gas distribution (effects performance/wear)
2. Increased thermal stresses (effects wear)
3. Design of enhanced propellant insulators (effects performance)
4. Plume effects for clustered thrusters (effects performance/wear)
5. Determining scaling laws for performance and geometric parameters

<u>Correlating Masses with Previous Concepts</u>

A survey was conducted of previous Russian Concepts for manned Mars missions utilizing xenon (Hall type) propulsion. Five concepts were found (9):

1. KK - Korolev -1966
2. MEK – Korolev -1969
3. Mars 1986 – NPO Energia -1986
4. Mars 1989 - NPO Energia – 1989
5. Marspost – RKK Energia – 2000

There are a number of interesting points common to the Russian concepts:

1. The average mass of xenon propellant in the 3 most recent concepts (1986, 1989 and 2000),

175 mT, correlates well with the xenon mass calculated above (see Fig. 2)
2. If one subtracts the mass of the propulsion module and propellant from the overall mission mass, then the average mass per crew member for all 5 missions is clustered around **46 mT** (see Fig. 3). Note that all the missions have similar mission durations of approximately 700 days. Also included in Fig. 3 is the average mass based on a Mars Society Australia concept (13).
3. The missions all spend approximately 1 month in the vicinity of Mars (orbit and surface)

Based on a crew of 6 (Table 1), an average mass of 46 mT per crew member, a bismuth propellant mass (including **~15 mT** for station keeping and unplanned maneuvers, along with a **10% FPR**) of **96 mT** , a propulsion structure mass of **30 mT**, and a nuclear power subsystem mass of **45 mT,** the total mass of the current mission is approximately **447 mT**. This value is somewhat below the range given in Table 1, but does not include the addition of any design margin (which will be added below). If xenon propellant were used, the corresponding mass would be **569 mT,** which is in the range shown in Table 1. A second approach to determine the total mass is to look at the basic element masses based on previous Russian concepts and other data sources:

1. Interplanetary Orbiter (or Orbital Apparatus) – **total mass of ~156 mT**
    a. Assume **120 mT** base mass by scaling up a previous Russian module designed for 4 crew members
    b. Crew each consumes (based on several U.S. and Russian references)
        i. 2.5 kg drinking water/day
        ii. 2.0 kg food/day
        iii. 0.85 kg oxygen/day (or ~600 liter oxygen/day)
        iv. 6.0 kg wash water/day
        v. 0.5 kg general consumables/day
        vi. Total consumable mass of **~30 mT** for 630 days
    c. **6 mT** - 20% consumables reserve
2. Mars Ascent-Descent vehicle – based on previous Russian designs and including provisions for a crew of 3 for 1 week on the surface and one day on orbit – **total mass of 60 mT**
3. Earth re-entry vehicle – based on a Soyuz-TMA and accounting for 6 crew - **total mass of 15 mT**
4. Nuclear power subsystem/structure – based on a specific mass of **10 kg/kW** and including structure, radiators, mounting hardware and cabling **– total mass of 45 mT**
5. Propulsion element – thruster weight (**10 mT**) and structure – **total mass of 30 mT**
6. Propellant mass **–** assuming Bismuth, with FPR **- total mass of 96 mT**
7. Total mass – **402 mT**

Figure 2. Xenon propellant mass requirements for Mars mission concepts.

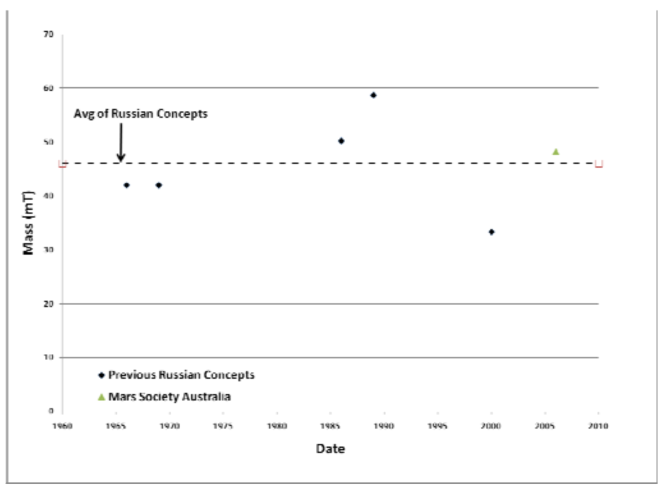

Figure 3. Mass per crew member, excluding propulsion system and propellant mass.

The two methods of estimating total system mass yield values within ~10% of one another. Applying a margin of **20%**, which is typical for the early stages of conceptual design, to the average of the two masses (**425 mT**) produces a final mass of **510 mT** (which is the range shown in Table 1).

Additional sources of mass could include:

1. Additional structure mass if solar arrays are employed for auxiliary electrical power
2. Backup cryogenic engines and propellant
3. Additional equipment (e.g., scientific, etc.)

<u>Alternate Propulsion Systems</u>

An alternative to using Hall thrusters is to use magnetoplasmadynamic (MPD) thrusters (14; 15). Hall thrusters are generally not designed to operate at greater than 10 N of thrust. MPD thrusters are used for greater thrust levels due to the difficulties (stated above) associated with the scaling of Hall thrusters. The Technology Readiness Level (TRL) of MPD thrusters is not as high (TRL=3/4) as for Hall thrusters (TRL=9), but a large amount of research is being conducted on MPDs. MPD thrusters can have very high specific impulse values, up to 10,000 sec. There are several propellants available for MPD thrusters, including the metals lithium and gallium. A survey of MPD thrusters using lithium provides an average mass flow rate of approximate 1700 mg/sec, although there is some uncertainty in the value (16). Assuming eight 20 N thrusters, Eqn. (1) can be re-written as:

166 days x 24 hrs/day x 3600 sec/hr x 1700 mg/sec-thruster x 8 thrusters x 1e-6 kg/mg = 195,057 kg (2)

Thus, the propellant mass using lithium would be about **195 mT**, not including FPR. This value is roughly the same mass as for xenon Hall thrusters. Adding **15 mT** for station keeping and a **10% FPR** to this value yields **231 mT**. At $7/kg , as of April 2009, this would equate to **~$1.6M** for a flight to Mars and back. At 535 kg/m^3 the volume would be 419 m^3. For reference, this would fit in a in a cylindrical tank that was 8.0 m (26.2 ft) in diameter and 8.3 m (27.3 ft) long. The total mission system mass using the MPD thrusters would be approximately **582 mT** before adding a **20%** margin, and **698 mT** with the margin.

Variable Specific Impulse Magnetoplasma Rocket (VASIMR) engines might also be used to power the spacecraft (17). VASIMR engines are distinguished by extremely high specific impulse values (up to 30,000 sec expected) and high exhaust velocities. In addition, the thrust and specific impulse can be optimized for different flight regimes. The advantages of VASIMR engines are that they do not contain parts subject to erosion (like Hall thrusters) and the high specific impulse can lead to shorter mission times. One proposal put forth envisions sending 2 vehicles to Mars (17; 18). The first vehicle is a cargo vehicle using a single 4 MW (electric) VASIMR engine. The cargo vehicle would take 15 months to escape the Earth's gravity well and make the transit to Mars. The second vehicle would be crewed and utilizes three 4 MW (electric) VASIMR engines and would take 120 days to escape the Earth's gravity well and reach Mars. The propellant for the VASIMR engines would be argon or hydrogen, both of which would need to be stored cryogenically. Boil-off concerns would need to be addressed for such a mission. The combined mass of the propulsion modules and propellant is **207 mT** with no FPR, and **220 mT** with a **10%** FPR**.** Applying the VASIMR engines to the current mission reduces the duration from 630 days to approximately **260 days**. The reduced mission time reduces the mass of consumables from 36 mT to approximately **13 mT** (including a **20%** reserve). However, the additional power usage increases the mass of the nuclear power subsystem, hardware and radiators to nearly **255 mT**. Thus, the total mission mass without margin would be **683 mT**, and **820 mT** including a **20% margin.**

Other concerns with VASIMR engines include:

1. Each 4 MW engine would produce on the order of 100 N, but to date the largest VASIMR engine tested, the VX-200 (18), only produces about 5 N.
2. The superconducting magnets used in the VASIMR engine are liquid cooled on Earth, and may require cooling for space applications. If the temperature of the magnets rises above a critical temperature, the efficiency of the engine drops off rapidly.
3. VASIMR engines require a large amount of input electrical power.

<u>Nuclear Power versus Solar Power</u>

The input power for the propulsion system will come from either solar cells or a nuclear reactor (19; 20). As shown in Fig. 4 (from Ref. (20)), solar cells are considered feasible for power levels up to approximately 100 kW, but not for the expected 15 MW (thermal) needed for the Mars mission.

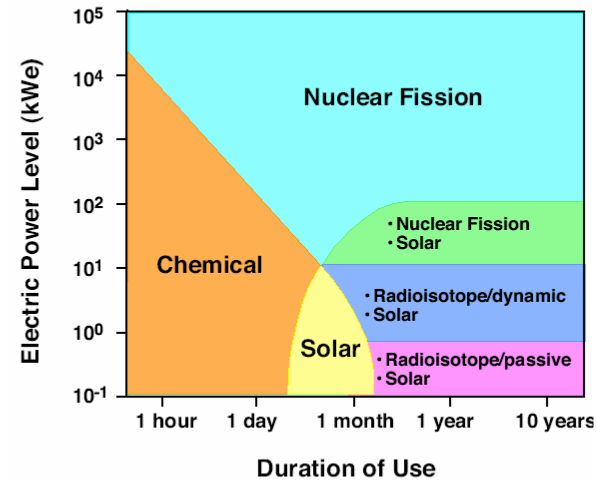

Figure 4. Power source utility regimes (Ref. (20)).

Currently, the specific mass (α = kg/kW) of multi junction solar cells is approximately **10-15 kg/kWe** implying that for a 2.25 MWe Mars mission the solar cells alone (**not** including structure) would account for **22-44 mT.** The Russians have tested thin film solar arrays between 20 and 50 microns thick, with a mass of 0.2 kg per square meter (2; 9). For the 150,000 m^2 of array needed to generate 2.25 MWe (3), this would result in a mass of **30 mT** for the thin film array. Based on the logistics and structure associated with such a large area of cells needed (the area equivalent to 2.5 football fields!), concerns associated with the solar arrays withstanding the harsh interplanetary environment, and the reduction in power with distance from the Sun, it is unlikely that solar power will be used. DARPA is working on low specific mass solar arrays, but are targeting the 20-80 kW range.

The most likely candidate for a nuclear reactor is a fission reactor operating with a Brayton or Stirling power conversion system (21; 22; 23). The largest nuclear reactors tested in space are on the order of 30 kW, although much larger reactors have been ground tested (24). While most U.S. plans envision the use of one large 4-10 MW reactor (17; 21; 22), the Russians have tended towards the use of multiple reactors in the 3-5 MW range (23). Figure 5, from Ref. (25), shows the trend towards decreasing specific mass with increasing power. The mass of the nuclear power subsystem, including a **7200 m^2** radiator, cabling, hardware and **15%** margin can be estimated using an equation from Ref. (22) which was originally developed for a lunar tug, but is based on a similar design and should provide a reasonable estimate for the current application:

$$M_{NPS}\ (kg) = 0.214\ kg/kW * P_e + [\alpha * P_e + 7200\ m^2 * 3\ kg/m^2] * 1.15 \qquad (3)$$

where M_{NPS} is the mass of the nuclear power subsystem, Pe is the required electrical (not thermal) power, and α is the specific mass. Assuming a Brayton cycle (which is more efficient at high power than a Stirling cycle) with **α=10** and a required electrical Power of **2.25 MW**, the mass of the nuclear power subsystem will be approximately **45 mT.** This value was used in the mass breakdown outlined above.

An example power budget for a Mars mission can be outlined as:

1. Nuclear reactor(s) generate a total of 15 MW (thermal)
2. A 15% efficient Brayton cycle power conversion system produces 2.25 MW (electric) of power
3. 0.25 MW (electric) is consumed by vehicle systems
4. 2.0 MW (electric) is supplied to the propulsion system
5. 12.75 MW (thermal) of heat is rejected to the environment
6. 60% efficient Hall thrusters produce 1.2 MW (electric) of propulsive power
7. The overall system efficiency is approximately (1.2 MW + 0.25 MW)/15MW = 9.7%

<u>Comparing the Russian Concepts with the NASA DRA 5.0</u>

A comparison of the Russian concepts, in terms of masses and mission length, can also be made with NASA's Human Exploration of Mars Design Reference Architecture (DRA) 5.0 (26). Table 3 contains the overall mission highlights for both mission concepts (assuming bismuth Hall thrusters for the Russian concepts), while Table 4 contains a comparison of the mission mass estimates. In Table 3 it is assumed that the heavy lift launch vehicle will carry **130 mT** to low Earth orbit (LEO). In addition, the masses for the Russian concepts are based on the **510 mT** discussed above for bismuth propellant. In Table 4 the reusable (or partially reusable) components are shaded in green.

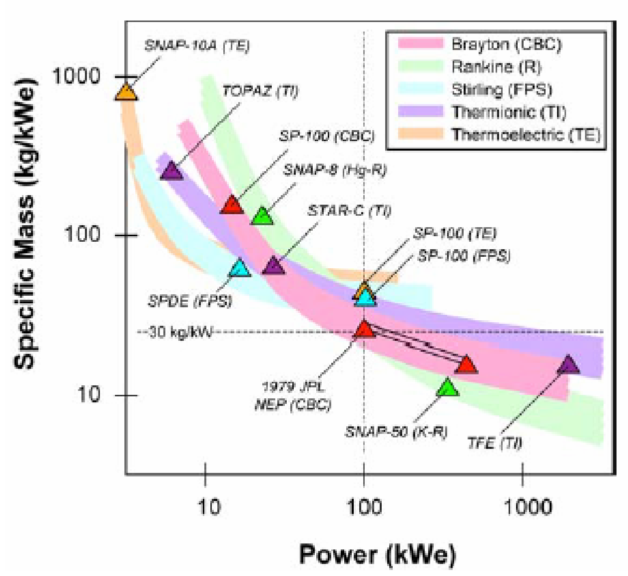

Figure 5. Specific Mass versus Electric Power (from Ref. (25)).

	Russian Concepts	DRA 5.0
Mission Length (yrs)	~2	~6
Surface Stay (months)	~0.5	~18
Propulsion Type	NEP	NTR
Isp (sec)	1000-2000 (Hall)	~900
Crew Size	6	6
Mass to LEO (mT)	~510	~848
Mass per Crew Member (mT) (w/ Interplanetary Power/Propulsion Elements)	~85	~141
Mass per Crew Member (mT) (w/o Interplanetary Power/Propulsion Elements)	~46	~44
Habitation Module Volume (m^3)	~410	~785
Number of Heavy Lift Launches	4-5	7
ISRU	No	Yes

Table 3. Overall mission parameters for the Russian concepts and DRA 5.0.

	Russian Concepts	DRA 5.0
Interplanetary Propulsion Vehicles (# of vehicles/total mT)	1/233	3/583
Mars Ascent/Descent Vehicle Including Any Aeroshell (mT)	72	107
Transit Habitation Module (mT)	187	41
Surface Habitation Module (mT)	Part of Ascent/Descent Vehicle	107
Earth Re-entry Vehicle (mT)	18	10
Total Mass (mT)	510	848

Table 4. Element masses for the Russian concepts and DRA 5.0.

Several observations can be made based on Tables 3 and 4:

1. The DRA 5.0 mission has 40% greater mass, requires 2-3 additional heavy lift launch vehicles, and will have 3 times the duration of the Russian mission. If xenon propellant (total mission mass of **683 mT**, which includes a 20% margin) is used then the DRA 5.0 mission has 20% greater mass and requires 1 additional heavy lift launch vehicle. It should be noted, however, that the DRA 5.0 surface stay (and the associated science and engineering accomplished) is an order of magnitude greater than that in the Russian mission. In addition, more than 3 of the 6 years of the DRA 5.0 mission are accounted for by two unmanned cargo vehicles sent in advance of the manned transit.
2. DRA 5.0 assumes that at least 26 mT of oxygen and 3.5 mT of water will be generated via in situ resource utilization (ISRU). The Russian concepts do not use ISRU, and no closed loop Environmental Control and Life Support System (ECLSS) was assumed.
3. The main difference in the masses is accounted for by the interplanetary propulsion vehicles. The masses of the two missions are within 5% (277 mT for the Russian concepts versus 265 mT for the DRA

5.0) if one does not include the Interplanetary Propulsion Vehicles.

4. Two candidate variables that can be adjusted to meet programmatic and budgetary constraints (i.e., the two biggest knobs) appear to be the propulsion element(s) and the number of crew. In keeping with this, one can decompose the mass per crew member (excluding the power/propulsion elements and propellant) into a fixed value plus a variable value based on the duration of the mission. Utilizing the data used to generate Fig. 3, the follow empirical equation can be derived:

   Mass/crewmember (mT) = 35.9 mT + (mission duration - days) x 14.5 kg/day * 1 mT/1000kg (4)

   where 35.9 mT is the fixed value and the second term is based on the mass of per day consumables with a 20% margin.

Summary

The Mars mission outlined in several recent Russian concept studies can be summarized as follows:

1. The mission would require around 510 mT (including a 20% margin) and the use of 4-5 heavy lift launch vehicles using bismuth propellant, or 683 mT (again including a 20% margin) and the use of 6 heavy lift launch vehicles using xenon propellant.
2. The in-space propulsion would be provided by Hall thrusters
   a. Xenon or bismuth propellant can be used, but bismuth may be preferable because of cost, density and condensability.
   b. VASIMR engines could be used in place of Hall thrusters. The higher Isp of VASIMR engines could result in significantly shorter mission times. However, several technical issues need to be resolved, including: 1) cryogenic storage of propellants, and 2) cooling of the super-conducting magnets.
   c. The in-space propulsion vehicle would be reusable
3. The electric power to the engines/thrusters would be provided by nuclear fission reactors
   a. Solar arrays would be too large and the power available decreases with distance from the Sun
   b. The reactors would probably use Brayton or Stirling cycle conversion systems
4. The mission would spend approximately one month in the vicinity of Mars, and 7-14 days on the surface of the planet
5. Total mission length is approximately 2 years
6. The composition of the concepts lends itself towards partnering

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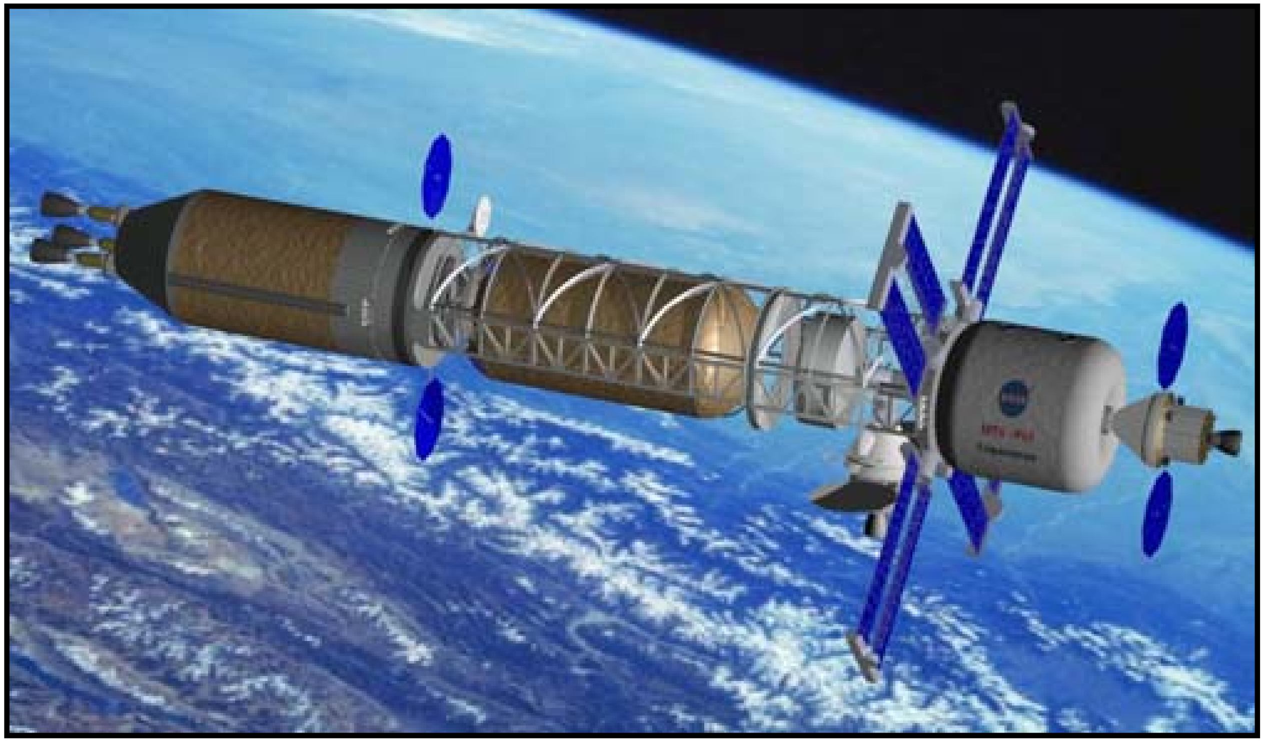
A STUDY FOR MARS MANNED EXPLORATION



D. Dorney, D. Schumacher, D. Scimemi

NASA

8 May 2012

Outline

- Previous Russian concepts
- Current Russian concepts
- Comparisons to the NASA DRA 5.0 concept
- Status and challenges of a nuclear powered mission
- Conclusions

Previous Russian Concepts

KK - 1966
15 MW Nuclear Electric
Crew of 3
630 day Mission
150 mT to LEO
Two N1 launches

MEK - 1969
15 MW Nuclear Electric
Crew of 3
630 day Mission
150 mT to LEO
Two N1 launches

Mars - 1986
15 MW Nuclear Electric
Crew of 4
716 day Mission
365 mT to LEO
Five Energia launches

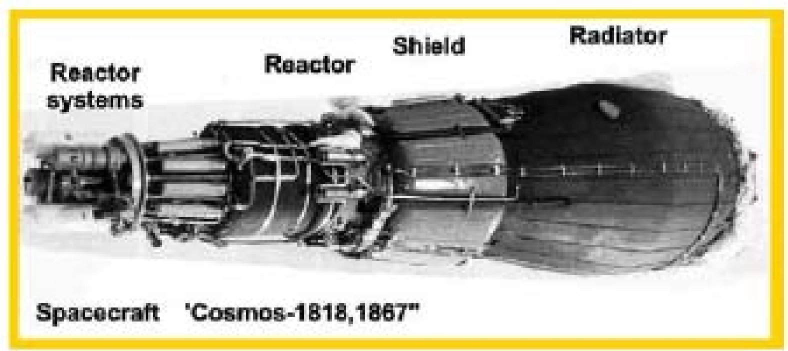
Mars - 1989
15 MW Solar Electric
Crew of 4
716 day Mission
355 mT to LEO
Five Energia launches

Mars - 1994
Bimodal Nuclear Thermal
Crew of 5
460 day Mission
800 mT to LEO
Nine Energia launches

Marspost - 2000
Solar Electric
Crew of 6
730 day Mission
400 mT to LEO
Five Energia launches

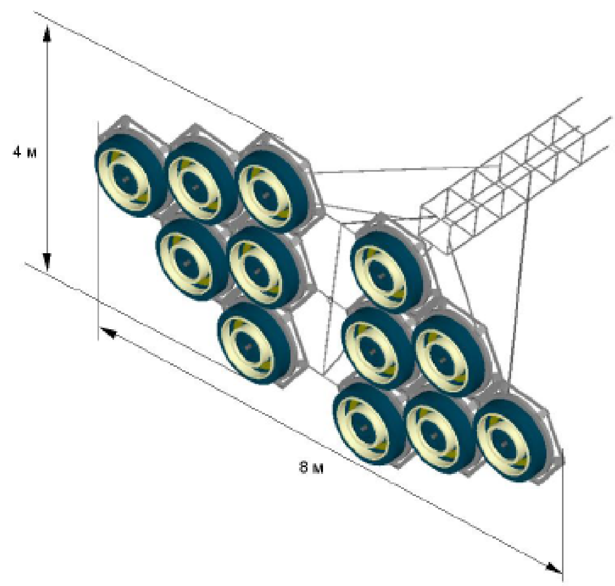
Russian Nuclear Power Configurations

- Early designs used one large reactor → 15 MW
- Contemporary designs use 2 reactors → 7.5 MW
  - Redundancy and failure tolerance
- Largest reactor flown to date generated ~10 kWe
- Used to power ion thrusters
- Designed to be reusable



Ion Thrusters

- Early mission architectures included 2-4 larger ion thrusters
- Recent mission concepts contain 10-20 smaller ion thrusters
  - 7-9 N thrusters
  - Included extra thrusters for redundancy



Design objectives of EPS with 10 main and 2 redundant SPT modules

Power	1 MW
Thrust	7 - 9 kgs
Specific impulse	1400 – 1700 s

From Koroteev

Summary of Previous Concepts

- Electric propulsion used for interplanetary transit in all but one concept
  - Both nuclear and solar electric considered
  - 15 MW of power in all cases
  - 441 N (99 lbf) of thrust
  - Xenon used for propellant
- Nuclear thermal propulsion considered for one concept
  - Total mass of mission nearly double that using electric propulsion
- Most concepts include reusable components
- All concepts but one consider 1 week surface stay

Current Russian Concepts

- There is not one comprehensive Russian concept/plan available, rather there are several concepts with similar features
  - Energia
  - Keldysh Research Center
    - A. Koroteev
    - V. Akimov, A. Gafarov
- The following analysis is based on reviewing and correlating the previous concepts, and synthesizing the new concepts

Current Russian Concepts and NASA DRA 5.0

	Russian Concepts	NASA DRA 5.0
Mission Length (yrs)	~2	~6
Surface Stay (months)	~0.5	~18
Propulsion Type	NEP	NTR
Isp (sec)	1000-2000 (Hall)	~900
Crew Size	6	6
Mass to LEO (mT)	~510 (bismuth) ~683 (xenon)	~848
Mass per Crew Member (mT) (w/ Interplanetary Power/Propulsion Elements)	~85 (bismuth) ~114 (xenon)	~141
Mass per Crew Member (mT) (w/o Interplanetary Power/Propulsion Elements)	~46	~44
Habitation Module Volume (m^3)	~410	~785
Number of Heavy Lift Launches	4-5	7
ISRU	No	Yes

Values for Russian concepts based on correlation of previous concepts and synthesizing current concepts.

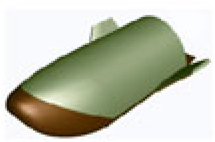
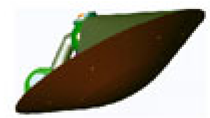
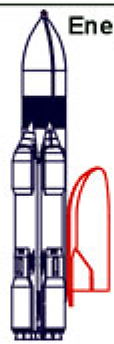
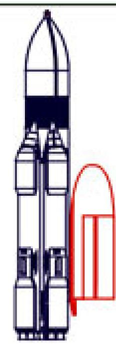
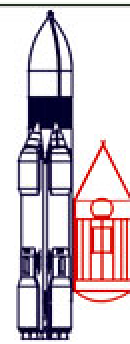
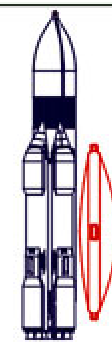
Mission Masses

	Russian Concepts	DRA 5.0
Interplanetary Propulsion Vehicles (# of vehicles/total mT)	1/233 (bismuth) 1/406 (xenon)	3/583
Mars Ascent/Descent Vehicle Including Any Aeroshell (mT)	72	107
Transit Habitation Module (mT)	187	41
Surface Habitation Module (mT)	Part of Ascent/Descent Vehicle	107
Earth Re-entry Vehicle (mT)	18	10
Total Mass (mT)	510 (bismuth) 683 (xenon)	848

Values based on correlation of previous concepts and synthesizing current concepts.
Values shaded in green denote reusable components.

Launching the Components

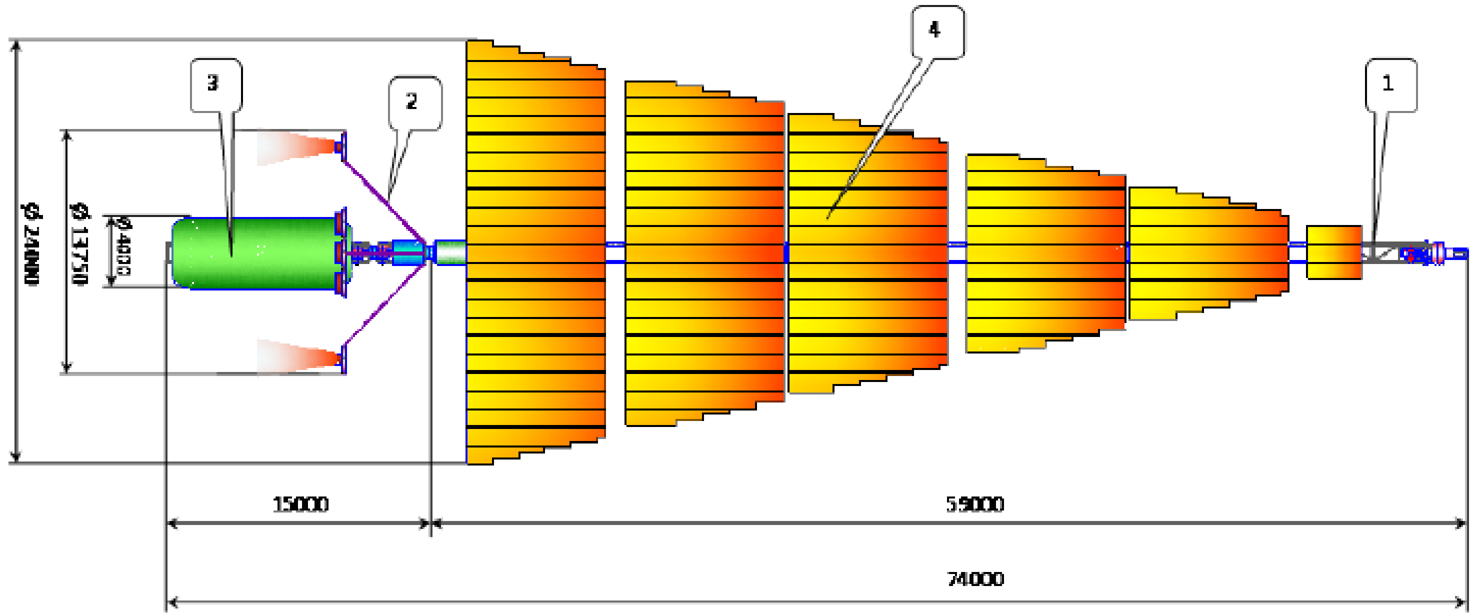
ASCENT-DESCENT COMPLEX CONFIGURATION OPTIONS (ADC)

	Primary structure	Primary structure with folds	Deployable shield	Disc
ADC				
Install LV	Energia-type LV 			
Design parameters: Mass (t)	70	70	70	70
Aspect area (m^2)	140	250	450	450
Weigt to aspect area ratio (kg/m^2)	500	280	155	155
Weak points	- Weight to aspect area ratio compearativly high	- Additional operations for aerodynamic planes orbit deployment - Complicated design - Mass increase	- Additional operations for aerodynamic shield mounting and deployment - Modified nose fairing - Complicated design - Mass increase	

From Energia

A total of 8-9 Energia Launches or 5-6 Ares V launches.
The Russians envision a the need for a 70-80 mT launch vehicle

In-Space Vehicle Configuration

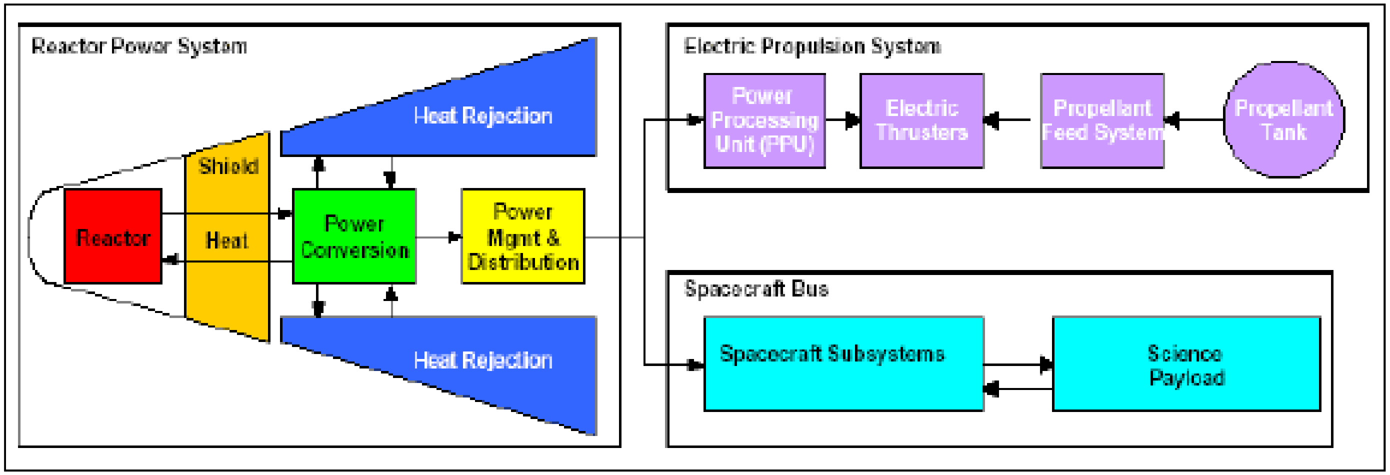


1 – *NPS* (operation mode; basic transport module);
2 – Electric Propulsion System – EPS (operation mode; basic transport module);
3 – Payload;
4 – Radiator-cooler (operation mode; basic transport module);.

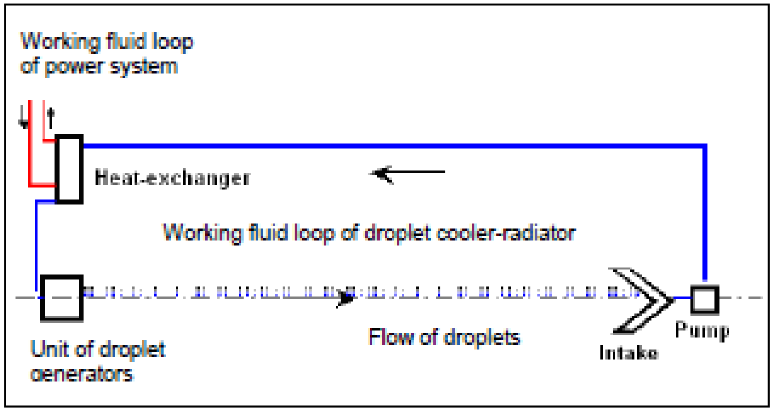
From Koroteev

The Russians intend use of liquid droplet radiators (LDRs) to alleviate need for very large radiators. LDRs were tested on Mir.

Vehicle Schematic and LDR



Vehicle schematic



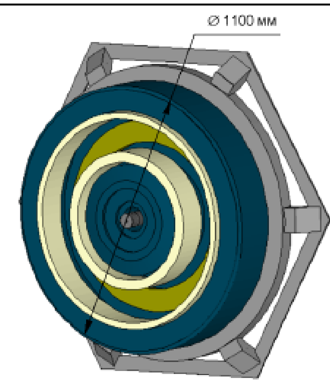
Liquid droplet radiator

From Akimov

Nuclear Electric Propulsion (NEP)

- Xenon or bismuth ion thrusters
- Xenon propellant mass values have remained consistent throughout the evolution of Mars mission designs
  - ~175 mT
- 7-9 N thrusters
- 140-170 N total thrust
- η=60%, Isp=1600 sec
- ~4000 hrs of operation

From Koroteev

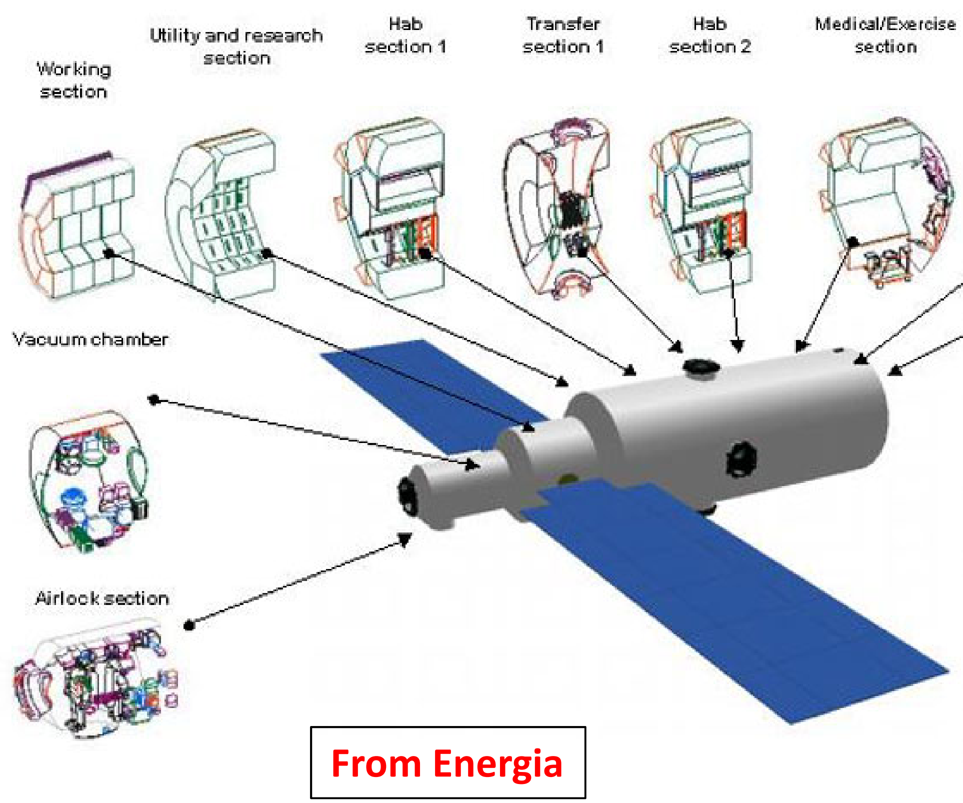


Design parameters of
single Thruster module

Power, kW	100		
Discharge voltage, B	200	250	300
Discharge current, A	490	392	327
Thrust, kgs	0,87	0,78	0,71
Xenon flow rate, mg/s	635	508	423
Specific impulse, s	1400	1565	1700
Efficiency, %	60		

Transit Habitation Module - I

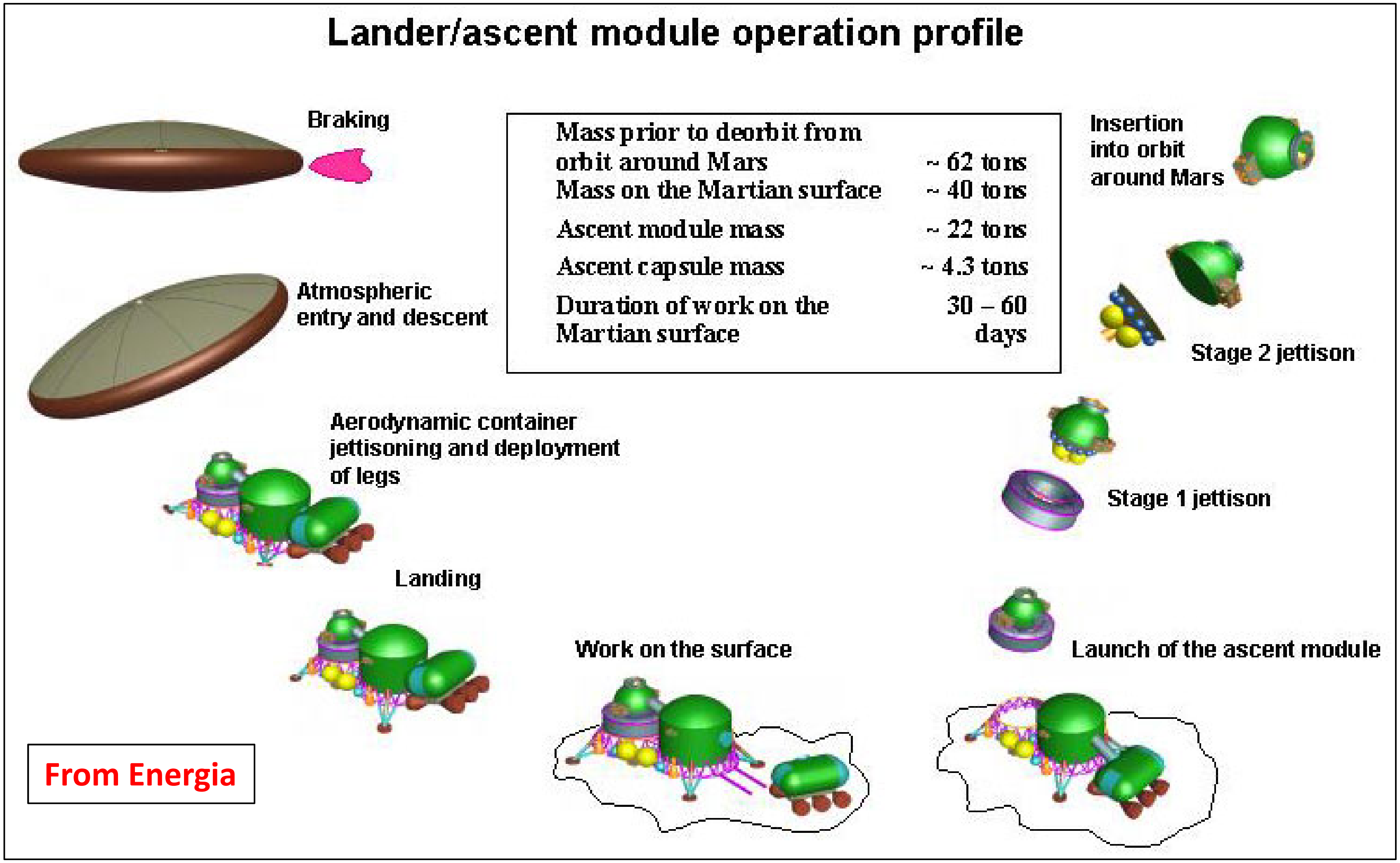
Layout of Interplanetary orbiter sections



Mass	70 tons
Total volume of pressurized sections	410 m^3
Number of pressurized sections	6
Number of the crew	Up to 6
Diameters of pressurized sections	2500, 4100, 6200
Installed capacity of the solar arrays	30 kW
Number of docking ports	Up to 6

Transit Habitation Module has remained constant over the last two mission designs. The module is reusable.

Lander/Ascent Module Configuration - I

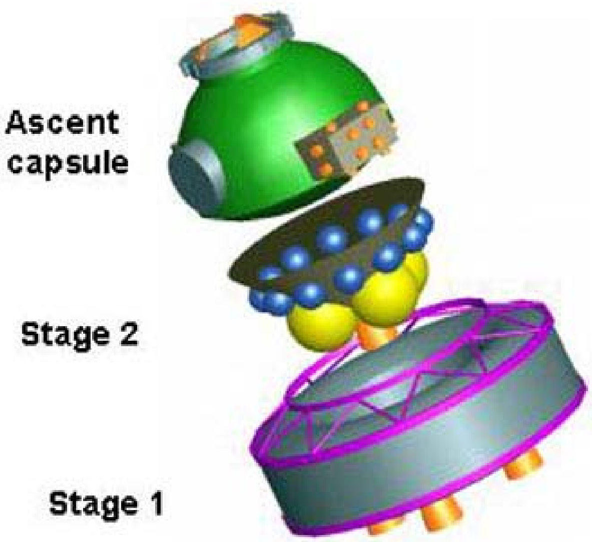


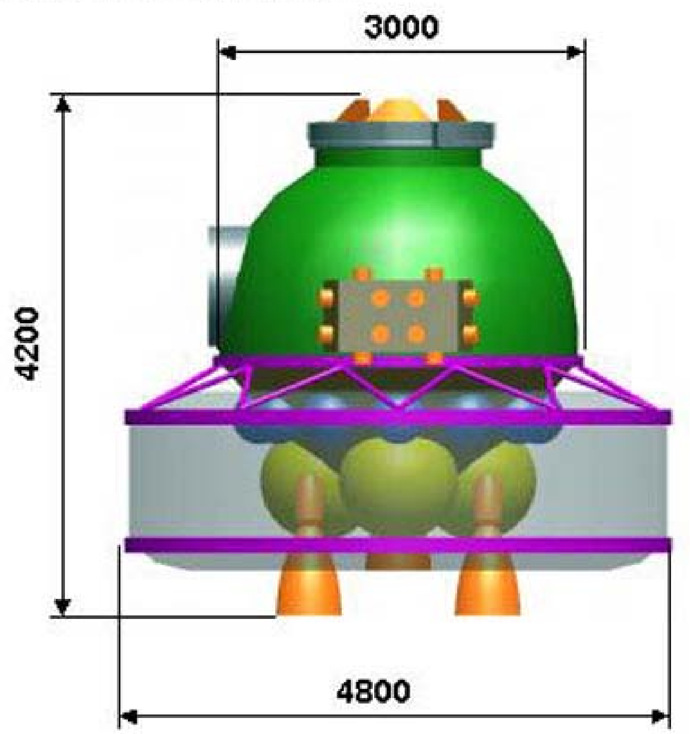
Mass prior to deorbit from orbit around Mars	~ 62 tons
Mass on the Martian surface	~ 40 tons
Ascent module mass	~ 22 tons
Ascent capsule mass	~ 4.3 tons
Duration of work on the Martian surface	30 – 60 days

Lander/Ascent Module Configuration - II

From Energia

Ascent module overall view





Ascent Module (AM) data

Total AM mass		21250 kg
Ascent Capsule (AC)		4300 kg
Stage 2		3050 kg
Stage 1		13900 kg
Structural mass:	AC	3700 kg
	Stage 2	500 kg
	Stage 1	1750 kg
Propellant mass:	BK	600 kg
	Stage 2	2550 kg
	Stage 1	12150 kg

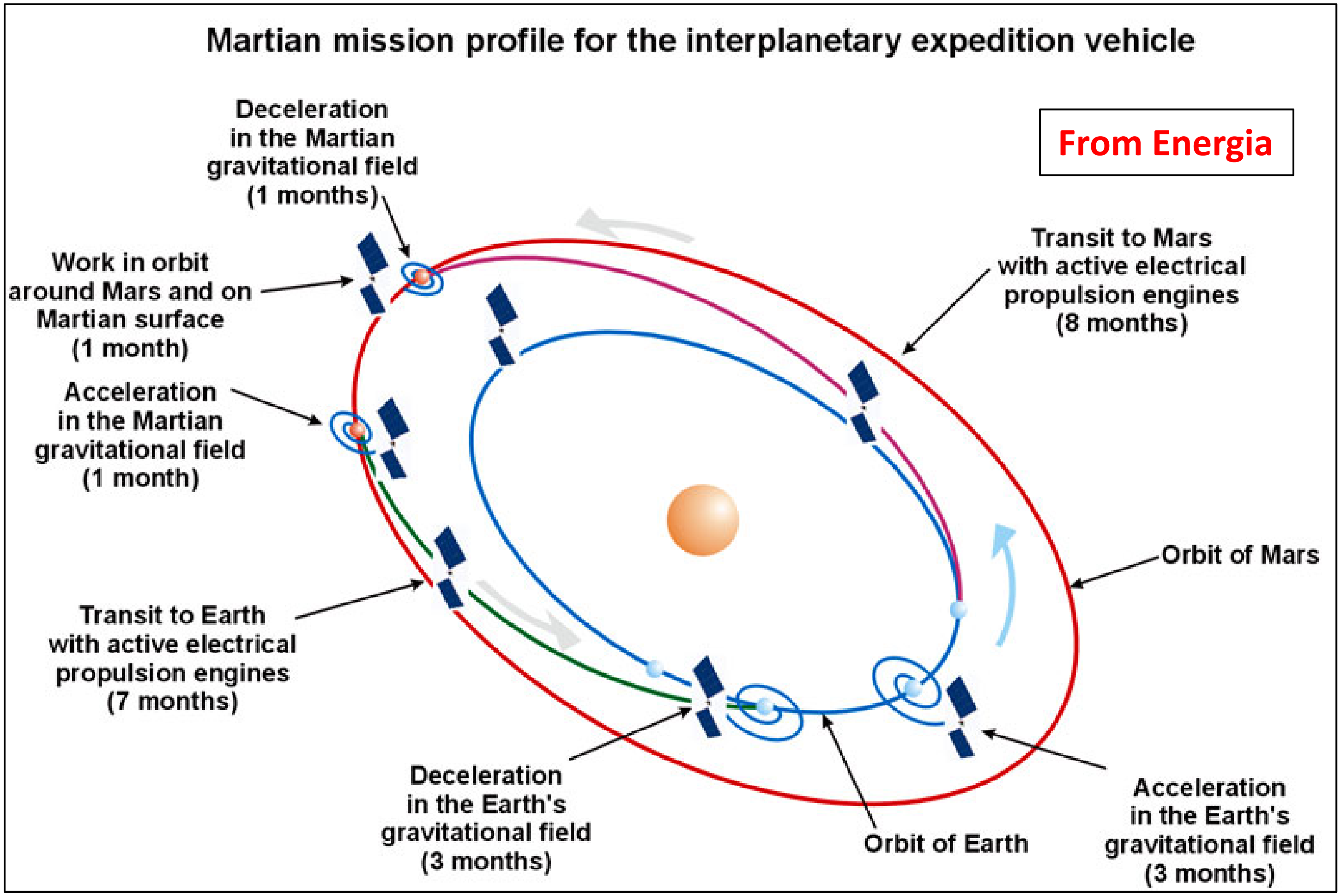
Ascent Capsule (AC) data

Command section:	Crew and equipment	430 kg
	Structure	710 kg
	Systems	1085 kg
	Reserve	230 kg
Propulsion section:	Propellant	600 kg
	Structure	200 kg
	Systems	845 kg
	Reserve	200 kg

Mass per Crew Member

- Mass per crew member has remained relatively constant across different designs
  - ~46 mT per crew member, excluding propulsion system and propellant
- Values are similar to U.S. values (~44 mT) → again, excluding the propulsion system and propellant

Mission Profile



Mission Length

- Total mission length of 716 days
  - Constant low acceleration/deceleration rather than high impulsive values → overall mission time similar to chemical and nuclear thermal propulsion concepts
- 686 days in transit, 30 days in Mars vicinity
  - 166 powered days in transit
- ~4000 hrs of ion thruster operation
- 7-21 days on Mars surface

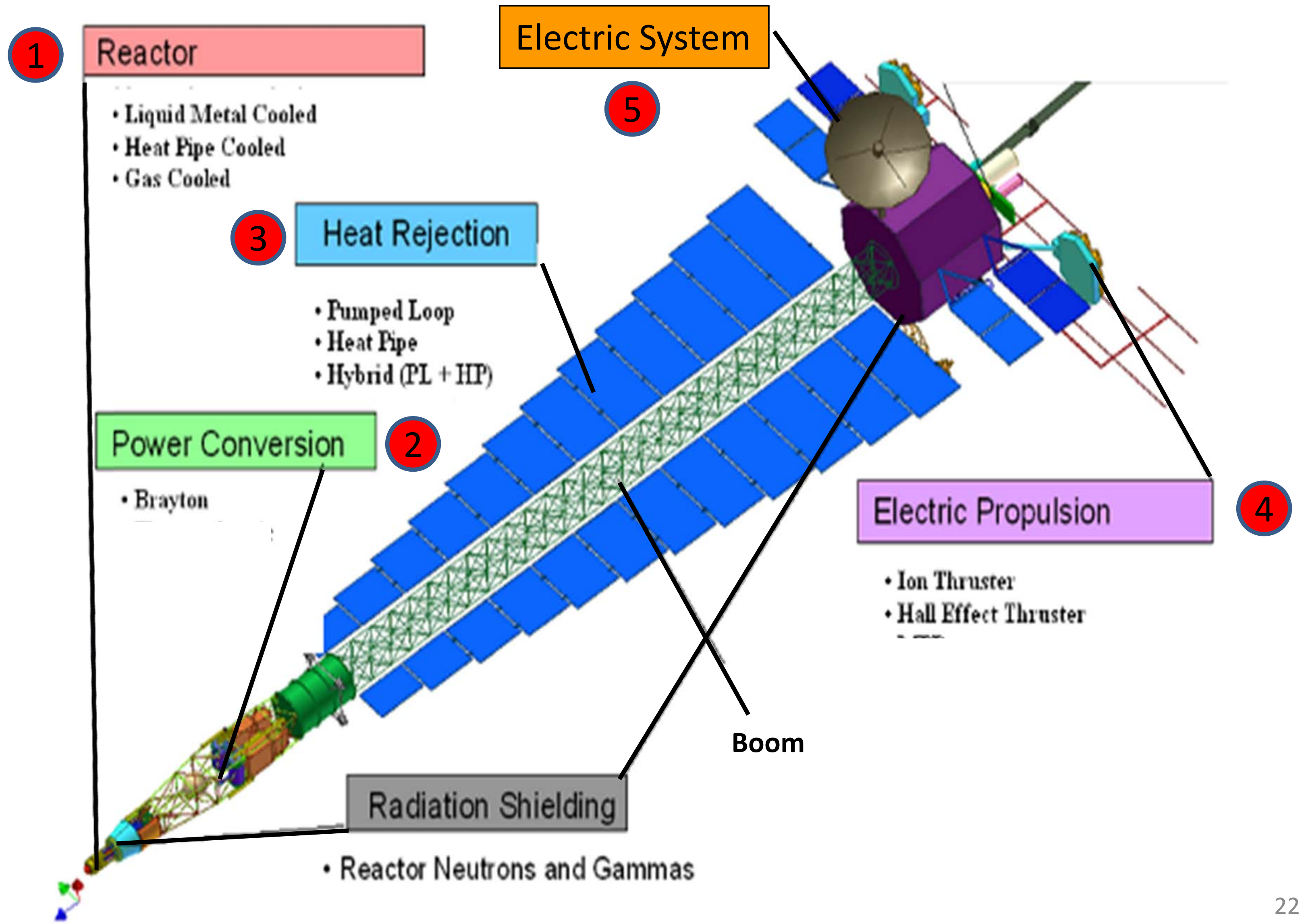
Challenges

- The greatest needs are in the following areas:
  - Heat rejection – this area may be the limiting factor in considering reactor power levels
  - Radiation shielding – silicon carbide shows great promise, but funding is inconsistent
  - Power conversion – large Brayton cycle systems have not been tested in space

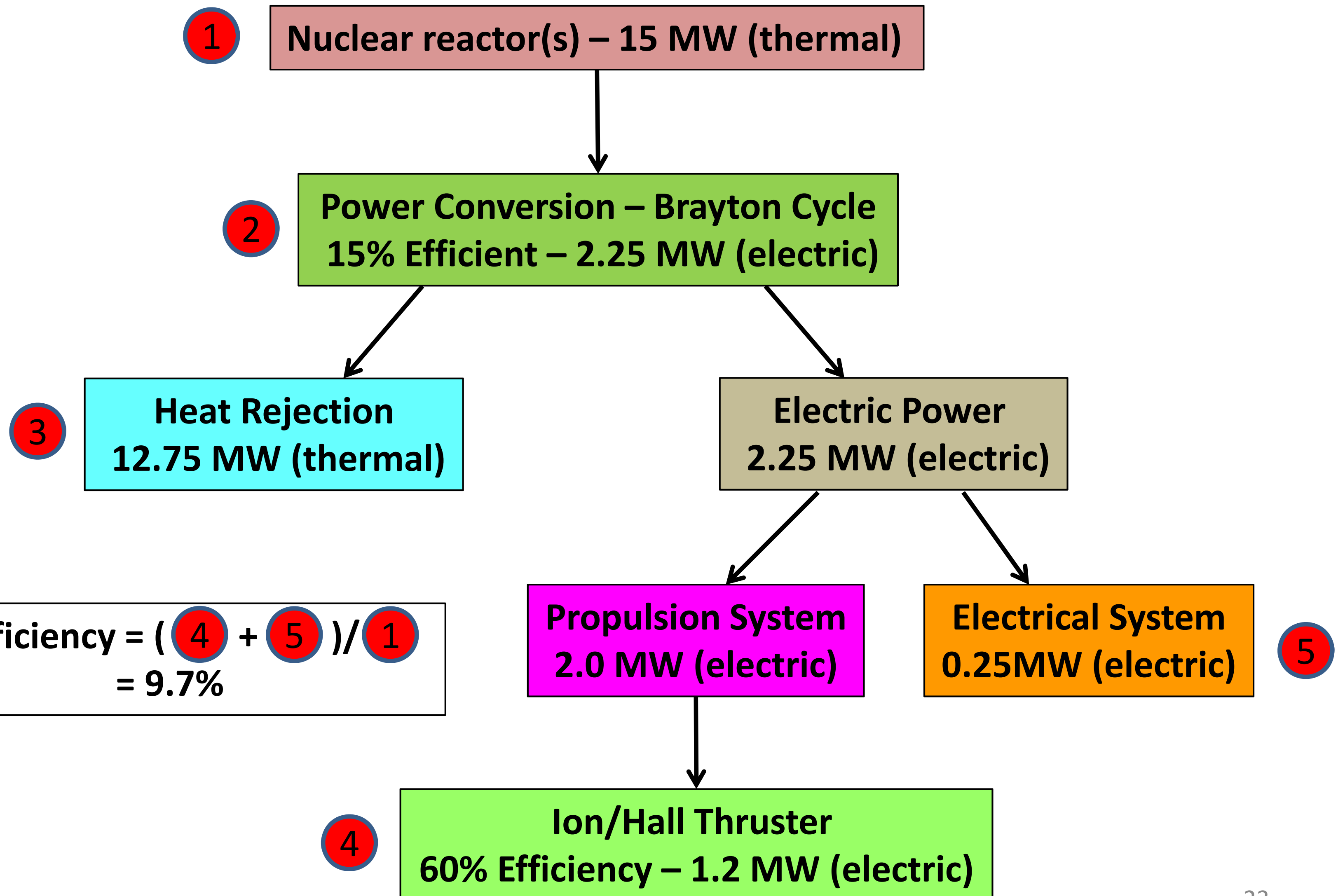
Power Conversion Cycles

- Thermoelectric → less than ~1 kWe
- Stirling cycle → 1 kW to 10 kWe
  - Difficult to scale due to limited heat exchange area
  - Tend to be heavier
- Brayton cycle - > ~10 kWe
  - Scalable – both up and down
- **Note: the largest reactor flown to date generated ~10 kWe**

NEP Systems - I



NEP Systems - II



Overall Efficiency = (4 + 5)/ 1 = 9.7%

Conclusions

- Mars mission can be accomplished with 500-600 mT
- Propulsion can be provided by Hall thrusters
  - Bismuth or xenon fuel
- Many advantages to reusable in-space components
- Power generation using fission reactors
  - Solar arrays likely too large
- Brayton power conversion cycle
- Composition of concepts ideal for partnering