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POSSIBLE SCENARIOS FOR MARS MANNED EXPLORATION

Daniel J. DorneyNASA George C. Marshall Space Flight Center, MSFC, AL, USA
daniel.j.dorney@nasa.gov**Daniel M. Schumacher**NASA George C. Marshall Space Flight Center, MSFC, AL, USA
daniel.m.schumacher@nasa.gov

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 current Mars reference mission.

I. 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
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.

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 m² (246 ft²) (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).

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).

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:

$$166 \text{ days} \times 24 \text{ hrs/day} \times 3600 \text{ sec/hr} \times 250 \text{ mg/sec-thruster} \times 20 \text{ thrusters} \times 1\text{e-}6 \text{ kg/mg} = 71,712 \text{ kg} \quad [1]$$

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 m³ (5). Converting the cost to 2011 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 1.116 will be assumed (12). Therefore, in 2011 dollars the 72 mT of bismuth would cost approximately **\$12.0 M** and require a tank with a volume of **7.2 m³**. Note that 7.2 m³ (254 ft³) 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 **\$400 M** and a tank volume of almost **37.7 m³** (1331 ft³) (5). The price of xenon has increased because supply has not been able to keep up with increased demand. 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. Note that xenon propellant thrusters have been flown on many spacecraft, while the use of bismuth has been demonstrated in laboratory experiments.

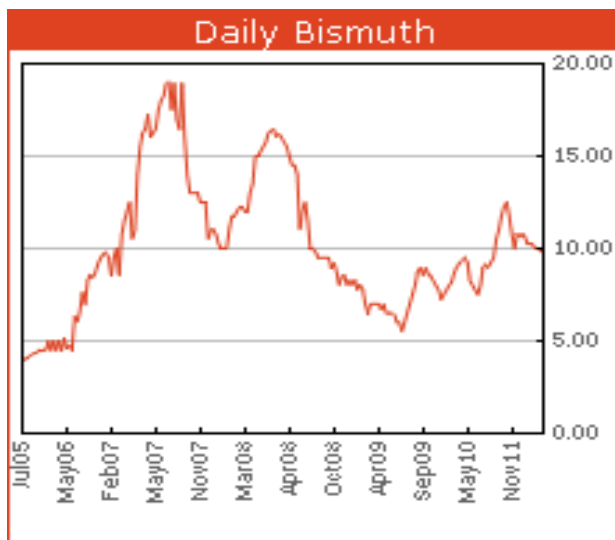


Figure 1. Historical price of bismuth in dollars/lb (Ref. (11)). Note, \$10/lb is approximately \$22/kg.

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

II. CORRELATING MASSES W/PREVIOUS DATA

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 other 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**

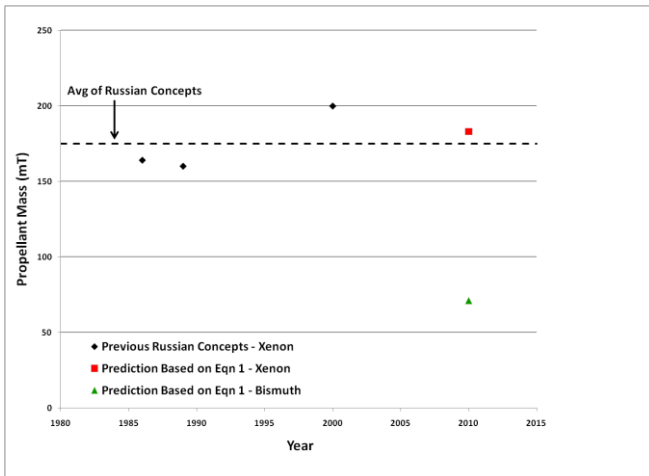


Figure 2. Xenon propellant mass requirements for Mars mission concepts.

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.)

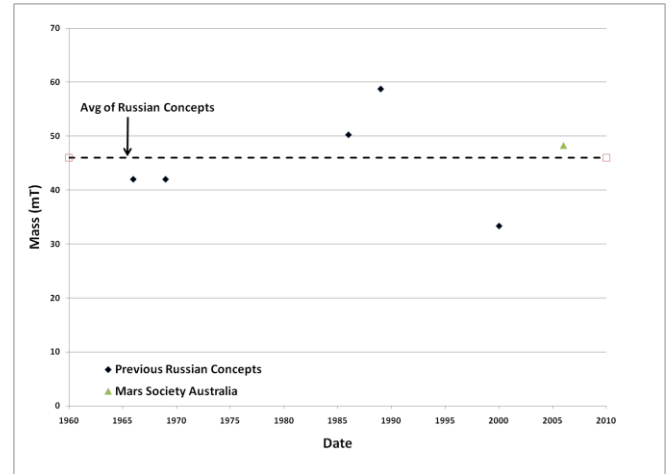


Figure 3. Mass per crew member, excluding propulsion system and propellant mass.

III. ALTERNATE PROPULSION SYSTEMS

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:

$$166 \text{ days} \times 24 \text{ hrs/day} \times 3600 \text{ sec/hr} \times 1700 \text{ mg/sec-thruster} \times 8 \text{ thrusters} \times 1\text{e-}6 \text{ kg/mg} = 195,057 \text{ kg} \quad [2]$$

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 \$270/kg (pure), as of August 2012, this would equate to **~\$62.5 M** for a flight to Mars and back. At 535 kg/m³ the volume would be 419 m³. For reference, this would fit 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), 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.

IV. NUCLEAR POWER VERSUS SOLAR POWER

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.

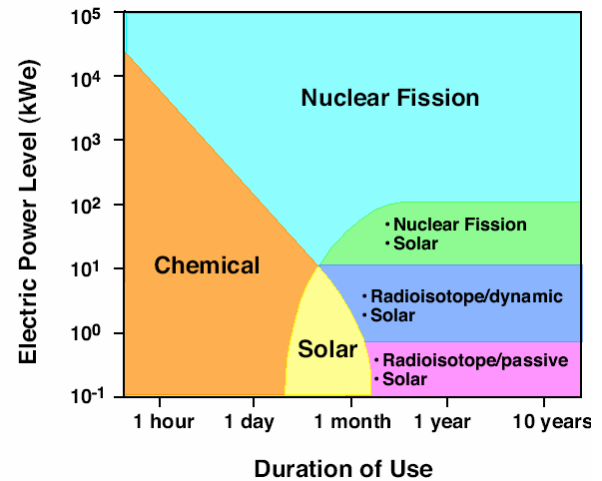


Figure 4. Power source utility regimes (Ref. (20)).

Currently, the specific mass ($\alpha = \text{kg/kW}$) 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 m² 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 use of solar arrays with direct drive may enable high-power (300 kW) electric propulsion applications (21).

The most likely candidate for a nuclear reactor is a fission reactor operating with a Brayton or Stirling power conversion system (22; 23; 24). The largest nuclear reactors tested in space are on the order of 30 kW, although much larger reactors have been ground tested (25). While most U.S. plans envision the use of one large 4-10 MW reactor (17; 22; 23), the Russians have tended towards the use of multiple reactors in the 3-5 MW range (24). Figure 5, from Ref. (26), shows the trend towards decreasing specific mass with increasing power. The mass of the nuclear power subsystem, including a **7200 m²** radiator, cabling, hardware and **15%** margin can be estimated using an equation from Ref. (23) 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:

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

where M_{NPS} is the mass of the nuclear power subsystem, P_e 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 $\alpha=10$ 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 $(1.2 \text{ MW} + 0.25 \text{ MW})/15\text{MW} = 9.7\%$

V. COMPARISON OF RUSSIAN CONCEPTS AND NASA DRA 5.0

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 (27). 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 **105 mT** (Block 1A of the Space Launch System) 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.

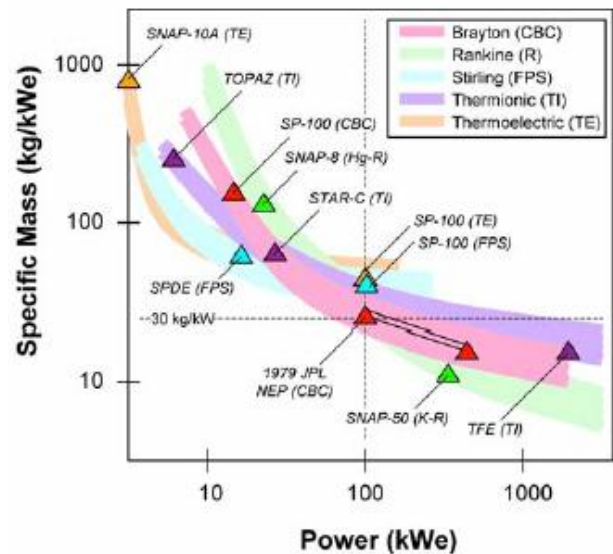


Figure 5. Specific Mass versus Electric Power (from Ref. (26)).

	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	5-6	8-9
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 2-4 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 1-2 additional heavy lift launch vehicles. 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:

$$\text{Mass/crewmember (mT)} = 35.9 \text{ mT} + (\text{mission duration} - \text{days}) \times 14.5 \text{ kg/day} * 1 \text{ mT}/1000\text{kg} \quad (4)$$

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.

VI. 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 5-6 heavy lift launch vehicles using bismuth propellant, or 683 mT (again including a 20% margin) and the use of 6-7 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

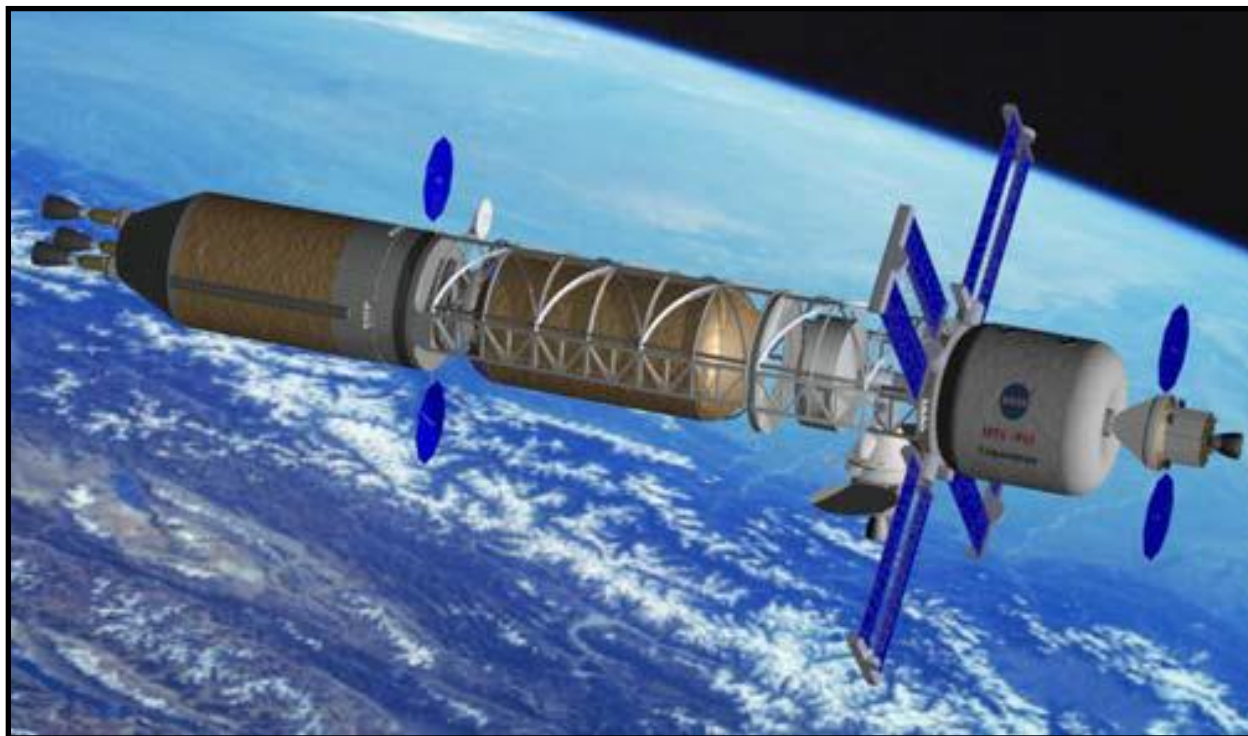
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POSSIBLE SCENARIOS FOR MARS MANNED EXPLORATION

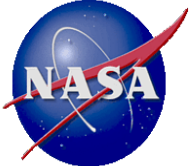


D. Dorney, D. Schumacher
NASA/MSFC
5 October 2012

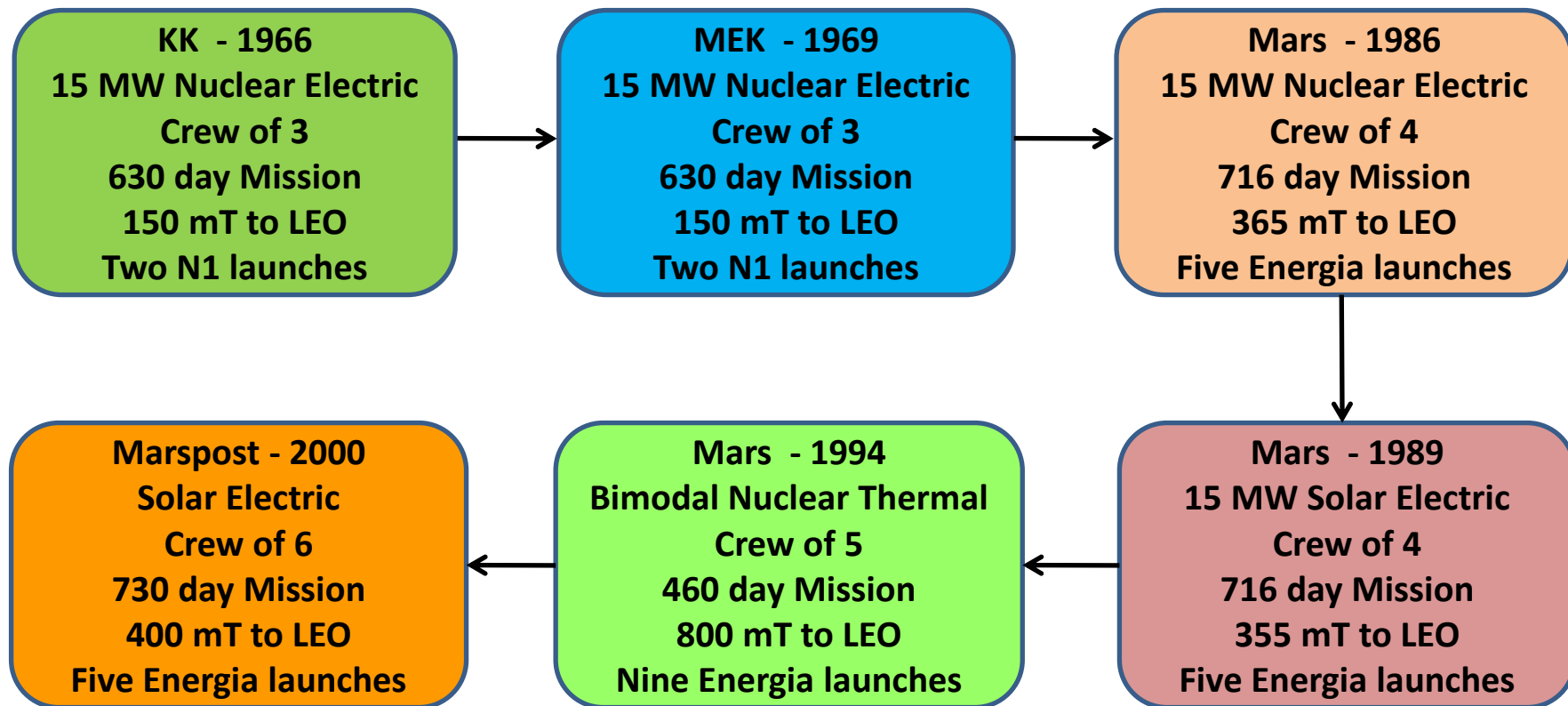


Outline

- Previous Russian concepts
- Current Russian concepts
- Comparisons to the NASA DRA 5.0 concept
- Conclusions



Previous Russian Concepts

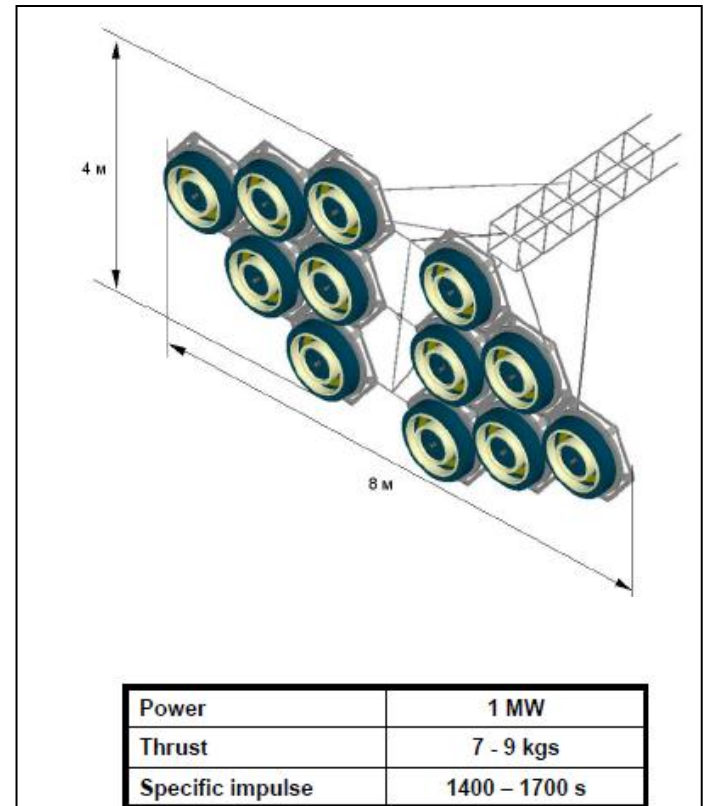


Assumes Energia launch vehicle capable of lifting
70-80 mT to LEO

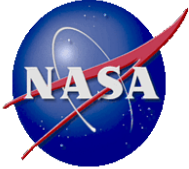


Ion Thrusters

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



From Koroteev



Summary of Previous Concepts

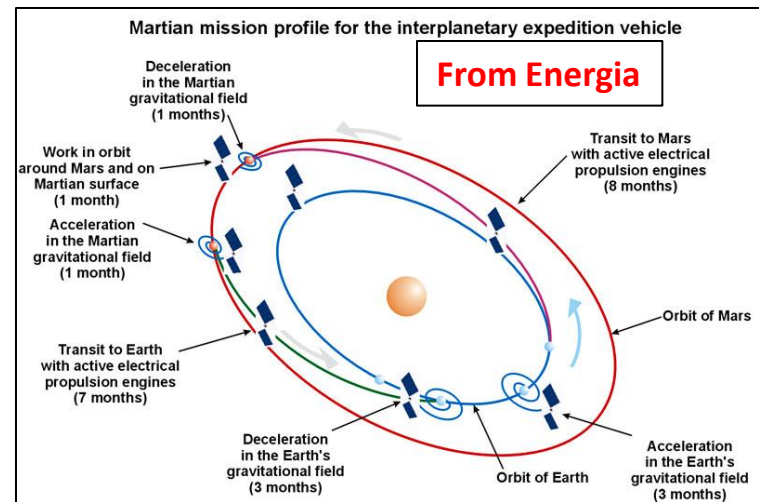
- Electric propulsion used for interplanetary transit in all but one concept
 - Both nuclear and solar electric considered
 - 15 MW of power in all cases
 - 441 N (99 lbf) of thrust
 - Xenon propellant
- Nuclear thermal propulsion considered for one concept
 - Total mass of mission nearly double that 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



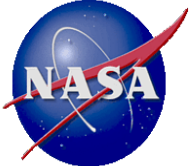
- The following analysis is based on correlating the previous concepts and synthesizing the current concepts



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	5-6	8-9
ISRU	No	Yes

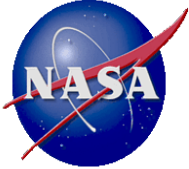
Values for Russian concepts based on correlation of previous concepts and synthesizing current concepts.



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 Aspect area (m ²) 140 Weight to aspect area ratio (kg/m ²) 500	70 250 280	70 450 155	70 450 155
Weak points	<ul style="list-style-type: none"> - Weight to aspect area ratio comparatively high 	<ul style="list-style-type: none"> - Additional operations for aerodynamic planes orbit deployment - Complicated design - Mass increase 	<ul style="list-style-type: none"> - Additional operations for aerodynamic shield mounting and deployment - Modified nose fairing - Complicated design - Mass increase 	<div style="border: 1px solid black; padding: 5px; text-align: center; color: red; font-weight: bold;">From Energia</div>

A total of 8-9 Energia Launches or 5-6 SLS Block 1A launches.