A Fission Powered Mars Telecommunications Orbiter Mission Concept

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Abstract. The Mars Program is performing ongoing studies to investigate concepts for high data rate telecommunications orbiters to support future Mars surface and orbital science missions. NASA recently completed a mission study to examine the use of a fission reactor powered telecommunications orbiter to enable high priority science at Mars and to validate key technologies in power, propulsion and telecommunications. This paper details the objectives of the mission, the key constraints that influenced the mission design, and the flight and ground system implementation. Issues that influenced the mission architecture are discussed, as well as the technology assumptions. Key mission and technology trades are listed and the decision criteria are developed. This mission concept was developed to examine the science benefit of a reactor powered telecommunications orbiter. No decision on power sources would be made until after completion of an Environmental Impact Statement.

INTRODUCTION

In accordance with NASA Headquarters direction, JPL has led an interagency mission study to examine the feasibility of a Mars Telecommunications Orbiter concept launching in 2008/2009. This mission would provide high data rate telecommunications capability to support a range of Mars Surface and orbital science investigations, and provide high resolution mapping of the surface. In addition, the mission would flight validate multiple new technologies including nuclear fission power generation for Deep Space missions, nuclear electric propulsion and optical communications.

This information was generated using the combined resources of multiple NASA centers and elements of the Department of Energy. Key NASA contributors to this concept development and cost estimate were Marshall Space Flight Center, Glenn Research Center and the Jet Propulsion Laboratory. For the Department of Energy, key contributors were DOE Headquarters, Sandia National Laboratory and Los Alamos National Laboratory.

MISSION OVERVIEW

The Mars telecommunications Orbiter (MTO) is a fission powered relay satellite that uses the high power available from a nuclear fission reactor to provide power for multiple purposes. To enable the delivery of a large payload including an advanced communications package, the mission uses Electric Propulsion (EP) for transfer to the planet and for orbit maintenance. Electric propulsion is also used to transfer between multiple preferential relay orbits, to support specialized surface mission requirements.

The spacecraft is launched to a positive C_3 (between 4 – 17 km^2/sec^2, depending on the opportunity) on a Delta IV Heavy expendable launch vehicle (Delta 4050H), which will provide between 7,000 – 9,000 kg capability. Transfer will take between 1.0 – 1.4 years, using the EP system to provide a zero C_3 approach and capture at Mars. The EP system is also used to spiral down from the initial capture orbit to the first operational orbit, which will take between 3 – 7 months. As with any EP mission, mass performance can be traded against trip time up to the limits of the launch vehicle. Total Delta-V provided by the EP system is approximately 12 km/sec.

The flight system was designed to provide a mission lifetime of greater than five years, to support at a minimum mission opportunities in 2011, 2013 and 2015. The EP system and the propellant load were sized to provide for one major orbit change between each mission opportunity. Three preferential orbits have been defined for Mars telecommunications support: areostationary, Constant Apoapsis Time Equatorial (ACE), and the Constant Apoapsis Critically Inclined (ACCI) orbits. Sufficient propellant has been assumed to provide one transition between each orbit, with additional propellant provided for orbit maintenance.

FLIGHT SYSTEM

The flight system enables high priority science investigations by providing a stable observation platform for selected remote sensing instruments and a high data rate telecommunications capability for data relay from surface and orbital science missions. The following is a summary of the key features of the flight system.

Science Payload

The science payload consists of two major subsystems: the observation systems and the telecommunications relay subsystem. The observation systems are nominally based upon existing instruments, with assumptions made to adapt to the mission. For the purposes of this study an imaging subsystem, based upon the Cassini Imaging Subsystem (ISS) was baselined, with other science payload to be determined. The primary science objectives of this payload are to continue the high resolution imaging of the surface of Mars. Imaging observations would be scheduled to coincide with low activity periods for the telecommunications relay payload, and would use the full resources of the high rate data link to provide large volumes of imaging data for analysis.

The telecommunications relay payload is based upon the desire to provide high rate data pipelines from Mars to the Earth for real time data relay from orbiting and surface assets. The general architecture of the relay capability is illustrated in Figure 1.

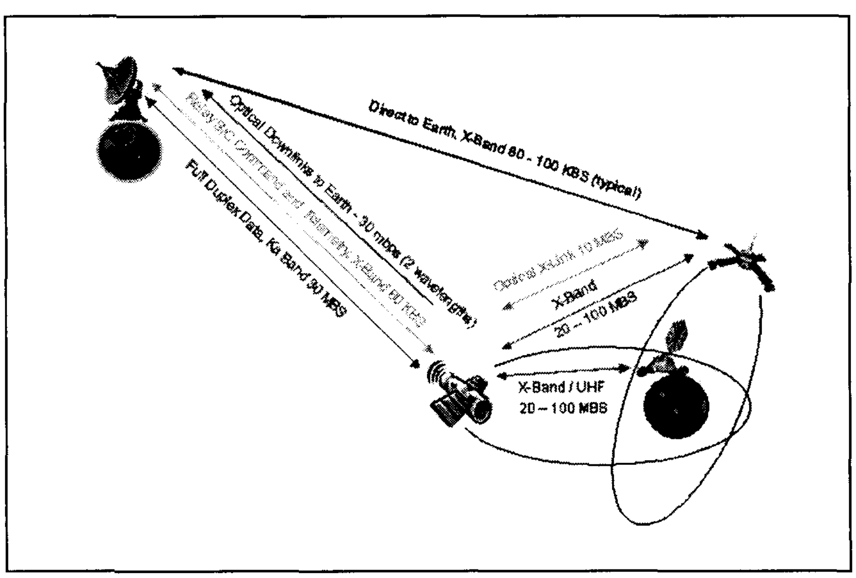

Figure 1. Mars Telecommunications Relay Architecture

The telecommunications system is designed to provide high speed deep space data communications via the Ka band link, which can provide up to 30 Mbps full duplex capability from Mars. The optical communications system is included to validate the capability, and provides an additional 30 Mbps downlink pipeline to optical receivers, to be developed and deployed at the DSN sites. X-band communications is included for relatively low data rate (approximately 60 Kbps) communications, such as engineering command and telemetry and housekeeping information.

For proximity communications to orbiting and surface assets, UHF links are provided for existing assets, as well as high speed X-band links providing 20 – 100 Mbps capability. An experimental optical crosslink capability was investigated but not baselined for this mission.

Spacecraft Bus

The spacecraft bus, which provides primary support to the payload, is of conventional design. All of the telecommunications, Command and Data Subsystem (CDS), attitude determination sensors and control electronics for the flight system are located in the primary bus. This includes the primary control electronics for the power and propulsion subsystem, which has separate control electronics for the reactor, power conversion subsystem and the ion propulsion system. Attitude control actuators, including reaction wheels and thrusters, are located as required to maximize their effectiveness.

Power and Propulsion Subsystems

The power and propulsion subsystems are the core of the new technologies for this mission. Power is generated by a fission reactor derived from the SP100 design. It is a liquid metal cooled reactor using UO_2 fuel in stainless steel clad, operating at 900 K outlet temperature, nominally producing 100 kW thermal power. A single reactor is baselined, feeding a redundant pair of Brayton cycle power conversion units. The Brayton cycle turboalternators are dual 10 kW units, operating at 900 K inlet temperature and producing approximately 20 kW electric power output, yielding a 20% energy conversion efficiency. Excess thermal power is shunted to large radiators that are configured along the extended boom, which performs two critical functions: provides structural mounting for the radiators and spacecraft subsystems, and provides separation between the reactor subsystems and the spacecraft bus.

The Brayton conversion units produce 100 VAC at 1200 Hz, which is rectified to 100 VDC for spacecraft use. Power is supplied via Power Management and Distribution (PMAD) units to the spacecraft, with Power Processing Units (PPUs) managing the power to the ion thrusters. Power to the spacecraft bus is separately conditioned at the consumer end to final requirements. Batteries are provided to provide bus power conditioning and startup/transient power, and solar arrays are provided (approximately 500W) for post launch checkout and initial startup of the reactor.

The low thrust propulsion subsystem is based upon next generation ion engines, first validated by the DS-1 mission. The ion thrusters baselined for this mission are currently in development, and can process 6.3 kW of power per thruster, and have an Isp of 4000 seconds and 68.5% efficiency. The design lifetime goal for these units is 300 – 500 kg of Xenon. Four thrusters are required with a fifth unit included as a spare. Xenon propellant is contained in a lightweight propellant tank, estimated at 2.5% of the total mass of the Xenon load. For attitude maintenance, a blowdown hydrazine propellant ACS propulsion system is provided.

Structures and Configuration

The structure of the flight system is dominated by the large radiators and the boom, which provides multiple benefits. As shown in Figure 2, the reactor is placed at one end, with the reactor shield providing radiation shielding to the rest of the flight system. The radiation shield provides a 7.5 degree half angle cone of protection to the flight system. Flight system components must be contained within that shield cone to prevent radiation damage and reflection of particles back onto the spacecraft.

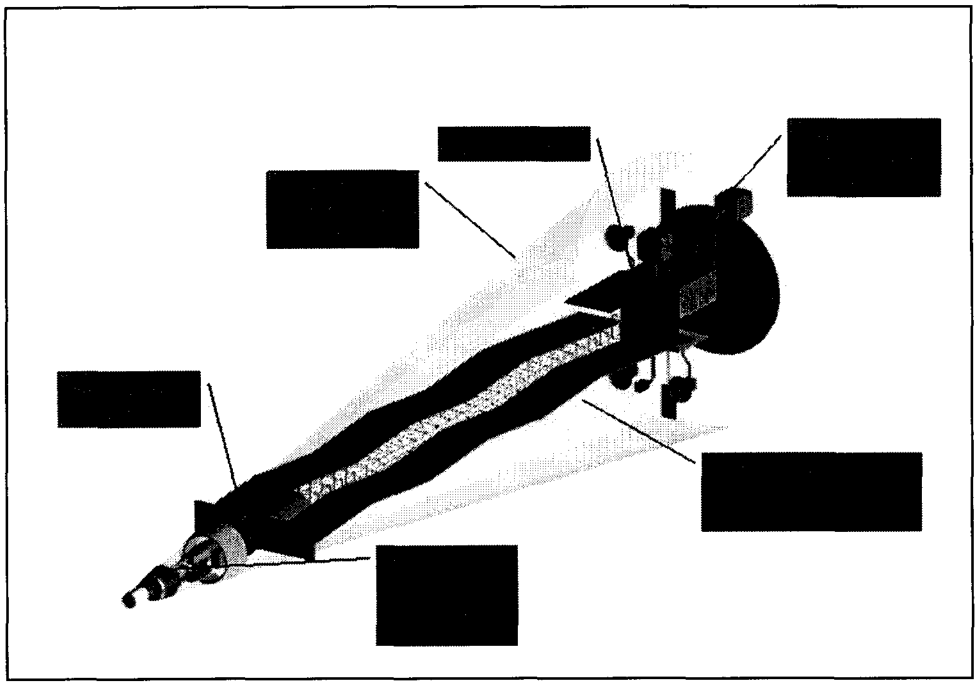

Figure 2. Flight System Configuration

The Power conversion system is located close to the reactor shield, and receives the thermal energy via the heat exchanger units. The Brayton turboalternators are set up to balance out any torques due to the rotating machinery. Excess heat from the reactor and the waste heat from the power conversion process is shunted into large radiators, which are arrayed along the 25 meter boom. The PMAD subsystem has its own dedicated shunt radiator that provides heat rejection.

The ion propulsion system is located close to the reactor, and close to the center of mass of the vehicle. The thrusters are configured to thrust perpendicular to the main axis of the boom, and are gimbaled for thrust vector control. The large Xenon tank also provides additional radiation shielding to the spacecraft bus, located at the opposite end of the boom.

The boom is a deployable structure, 25 meters long and connects the spacecraft bus with the power and propulsion subsystem. The radiators are arrayed along the sides of the boom, with cabling and power routed between the two major spacecraft elements. The boom would be deployed immediately after launch and remain deployed throughout the mission.

The spacecraft bus provides the platform for mounting the communications and observation payloads. The high gain antennas are articulated, and the observation instruments are mounted on a scan platform, to minimize the need to change the orientation of the flight system while in its operational orbit. Table 1 provides the mass estimates for the flight system configuration.

Table 1. Flight System Mass.

Element	Total Mass (kg)
Payload: avionics + telecom + S/C thermal...	**500**
Attitude Control	**66.0**
Power	**2055.0**
Propulsion	**305.3**
Structure	**850.63**
Cabling	**90.96**
Thermal	**44.60**
Propulsion module dry	**3413**
Mass contingency (30%)	1024
Spacecraft total dry with contingency	**4936**
Propellants	2213
Total Launch mass	**7150**
Initial injected mass	7458
Launch mass margin	4.1%

MISSION AND TECHNOLOGY TRADES

In assembling the mission and the flight system, multiple trade studies were performed. The baseline mission architecture is predicated on launching the system to a positive C_3, avoiding issues with operations of a reactor in Earth orbit. This placed significant constraints on the total mass of the system that could be launched, which drove many of the technology decisions. In addition, sizing of the reactor depends on many factors, including payload power requirements, propulsion power and available technologies. The current design provides the most capability within the tight mass allocation and available technology. Table 2 provides an overview of the significant trade studies and the drivers for the decisions.

Table 2. Mission and Technology Trades.

Trade Area	Chosen Option	Alternate Options	Issue
Launch	Launch to Escape	Launch to circular orbit and spiral out	Launch Approval
Reactor	Low Power, conventional technology	Advanced technology, higher power	Technology Development risk
Power Conversion	Brayton Turboalternators	Thermoelectrics	Efficiency, Mass
In Space Propulsion	NEP	Chemical	Capability, mass
Ion Propulsion	Advanced technology thrusters	NSTAR thrusters	Capability, mass, system complexity

SUMMARY

The Mars program has recognized the benefit of a telecommunications relay orbiter for Mars surface and orbital science missions. Providing a high rate data pipeline from Mars to Earth meets many scientific and programmatic goals by enabling a much greater data return. A fission reactor powered relay orbiter offers attractive benefits to the

Mars program by providing high rate data links and a stable observation platform, and provides benefits to the outer planets program by validating key technologies needed for advanced exploration missions.

The mission design seeks to balance technology development, mission architecture and programmatic risk, to provide a viable mission concept for a near term validation of the key technologies. As described, the mission could be ready for launch in the 2008 – 2009 timeframe. This would support potential applications of reactor power and NEP to outer planets missions early in the next decade.

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