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MARTIAN MOONS EXPLORATION (MMX) MISSION

SYSTEM DESCRIPTION DOCUMENT

REVISION AND HISTORY PAGE

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1. INTRODUCTION

1.1. Purpose and scope of this document

This document (MMX-SysDD) describes a summary of Martian Moons Exploration (MMX) system design and is addressed to all persons concerned to MMX mission.

1.2. Reference documents

See Section 3.

1.3. Notation

[TBD-Sys] [TBC-Sys]

The results of previous study for MMX or other projects are described as reference information for systems and the PI instrument. They will be established by system design in Phase A or beyond.

[TBD-Sys/PI] [TBC-Sys/PI]

The results of previous study for MMX or other projects are described as reference information for systems and the PI instrument. They will be established through coordination between the PI instrument and systems in Phase A or beyond.

[TBD-Doc] [TBC-Doc]

Information that can be described through coordination with design standards and other documents. (Phase A or beyond)

[TBD-Plan] [TBC-Plan]

Information determined after government approval of the project plan.

2. SYSTEM DESCRIPTION

2.1. Specification (outline)

Table 2.1-1 shows an overview of spacecraft system specifications (proposed) derived from system requirements and investigations of the system configuration.

Table 2.1-1 System Outline Draft Specification

Item	Specification
Propulsion system configuration	Outgoing: chemical propulsion Return: chemical propulsion
Spacecraft configuration	Composed of the following three modules ・Propulsion Module: Responsible for outgoing navigation (chemical propulsion) ・Exploration Module: Equipped with equipment necessary for Mars satellite exploration ・Return Module: Responsible for return navigation (chemical propulsion)
Launch	Rocket: H3-24L Launch date: Summer 2024
Orbit	Direct insertion to Mars transfer orbit — Mars satellite revolution orbit (pseudo-orbit) — Transition to Mars orbit (optional) — Mars escape orbit — Earth return trajectory
Total mission lifetime	5+ years
Target mass	3,500 kg
Power consumption	Approx. 900 W
Orbital control	ΔV: Approx. 5 km/s (chemical propulsion)
Communications	At approx. 2.7 AU from Earth, X-band of at least 32 kbps (TBC-Sys), Ka-band of at least 128 kbps (TBC-Sys) downlink (Ka-band used for mission data in the vicinity of Mars)
Data processing	Data bus method: SpaceWire network
Power supply	A thin-film, lightweight solar cell paddle capable of meeting power demands during the cruise phase and during the Mars satellite phase. Permits shade during launch, orbital insertion, and orbit.
Propulsion system thrust	Max. continuous ΔV: 20 m/s+ (for canceling orbital motion during descent) Interval between long thrusts: < 10 min Min. impulse: < 200 mN·s
Attitude determination and control	Attitude determination accuracy, pointing accuracy, and stability required during Mars orbital insertion/on-orbit/landing are respectively 0.01°, 0.1°, and 0.01°/s (with respect to inertial space) (TBD-Sys/PI)

2.2. Configuration

2.2.1. General

The MMX is composed of the Exploration Module, Return Module, and Propulsion Module. The Exploration Module carries scientific equipment on its instrument interface plane.

Figure 2.2.1-1 shows an example system block diagram of MMX. The MMX system consists of the following subsystems; data handling, communication, electrical power, attitude and orbit control, chemical propulsion, thermal control, structure, and mission payload such as sampling system, Earth return capsule, and science instruments. Figure 2.2.1-2 shows example block diagrams of the chemical propulsion subsystem for the propulsion module and the return module.

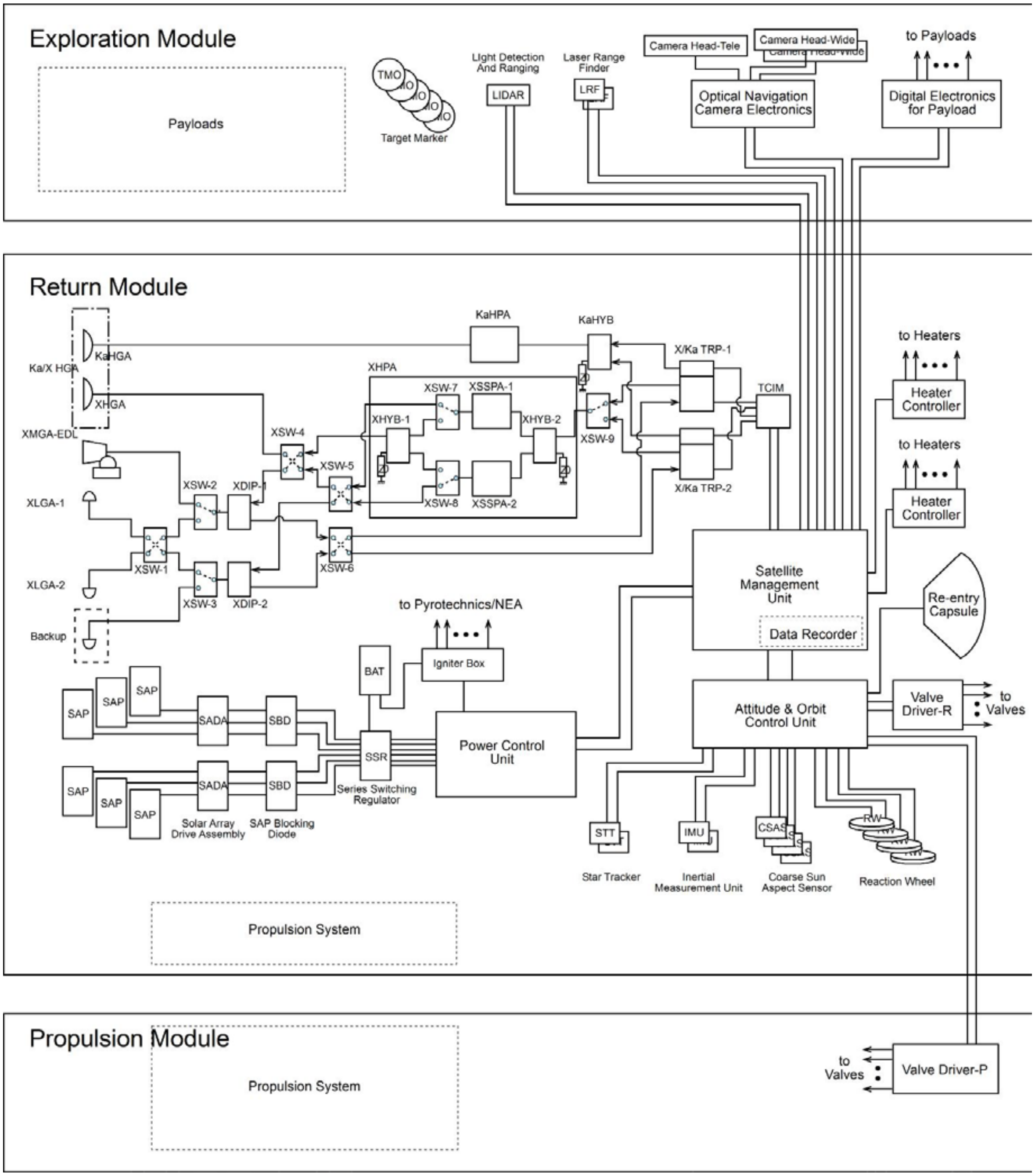
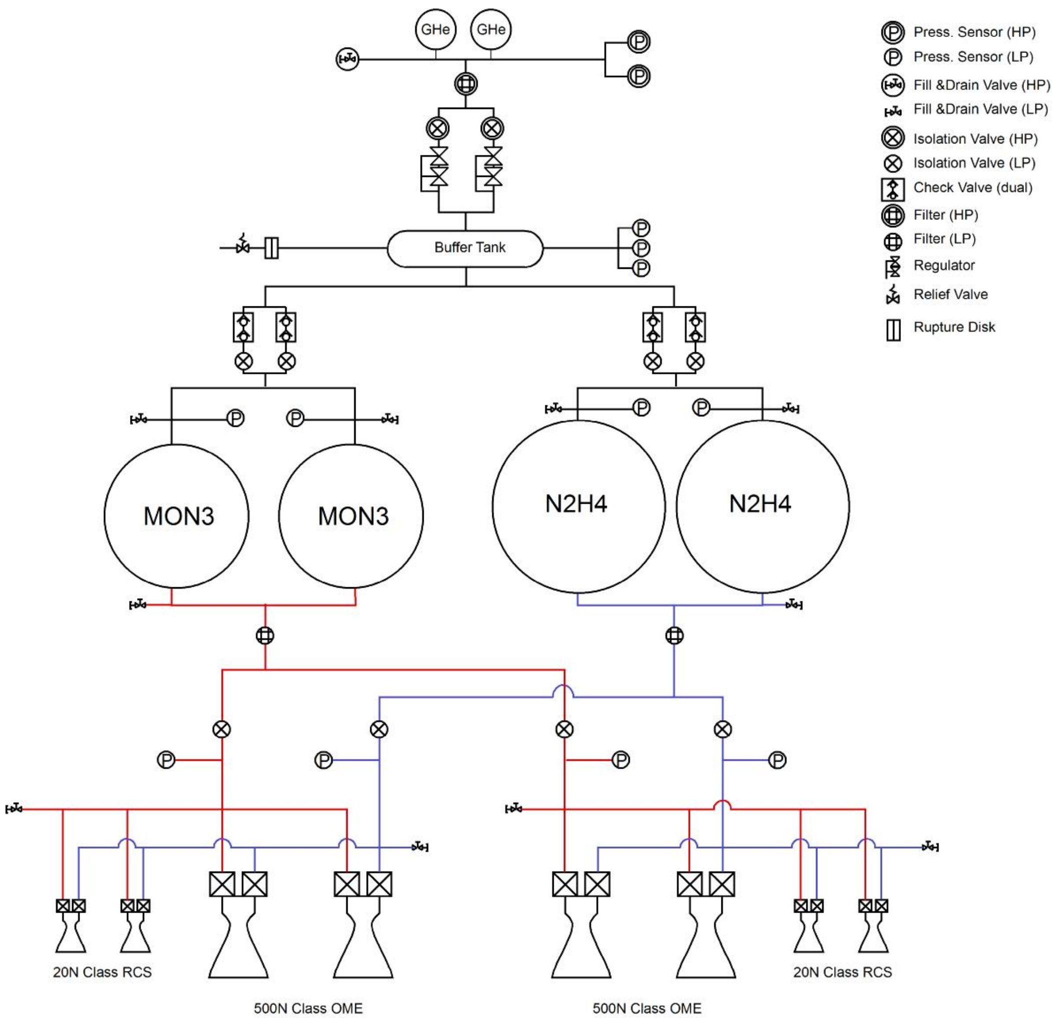
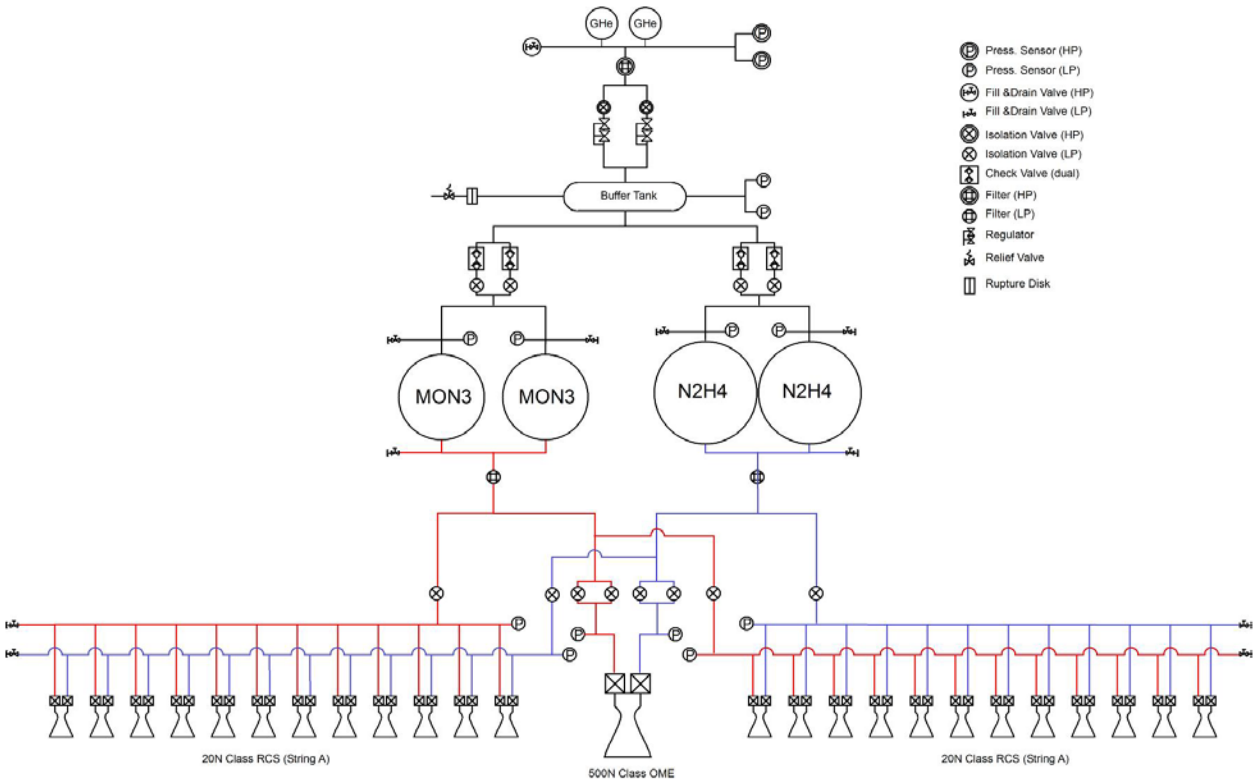


Figure 2.2.1-1 Proposed system block diagram



(a) Propulsion module



(b) Return module

Figure 2.2.1-2 Proposed block diagrams of chemical propulsion subsystem

Figure 2.2.1-3 shows an example spacecraft configuration. The spacecraft is composed of Propulsion Module, Exploration Module, and Return Module, and takes the shown form during propulsion. The following sections present details of each module.

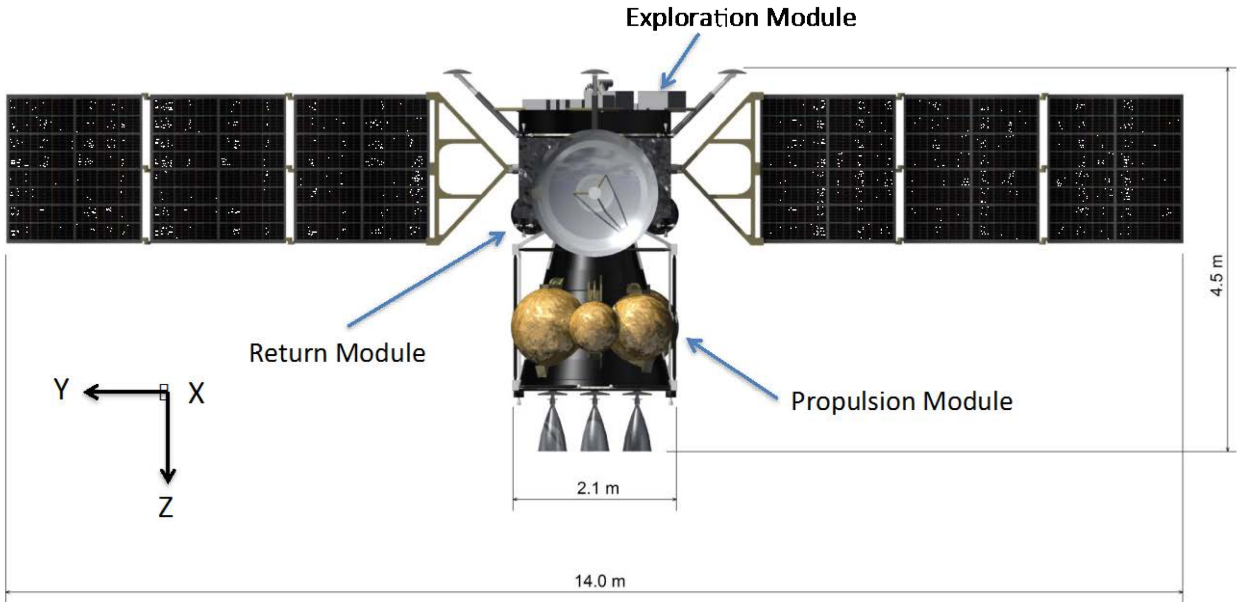


Figure 2.2.1-3 Proposed Spacecraft Configuration

2.2.2. Propulsion Module

To deliver the return and exploration modules to Mars as quickly as possible, the spacecraft features a large chemical propulsion system. The Propulsion Module separates after arrival at Mars. The Propulsion Module is mainly composed of thrusters and a tank, and the orbital maneuvering engine (OME) has 500 N thrusters arranged in the Z-plane.

2.2.3. Exploration Module

The Exploration Module is primarily composed of most (or all) of scientific observation instruments, a navigational guidance system, landing gear, and a sample collection mechanism. Following exploration, it is discarded in Mars orbit after separation when the Return Module leaves Mars orbit (TBD-sys). Collected samples will be loaded into a sample conveyance mechanism on the Return Module via manipulators.

2.2.4. Return Module

The Return Module contains a chemical propulsion system necessary for the return voyage and communication, data processing, power supply, and thermal control systems used throughout the mission. It also contains attitude control systems needed until return and a return capsule for bringing collected samples back to Earth.

2.3. MMX co-ordinate system

The MMX co-ordinate system is defined in Figure 2.3-1.

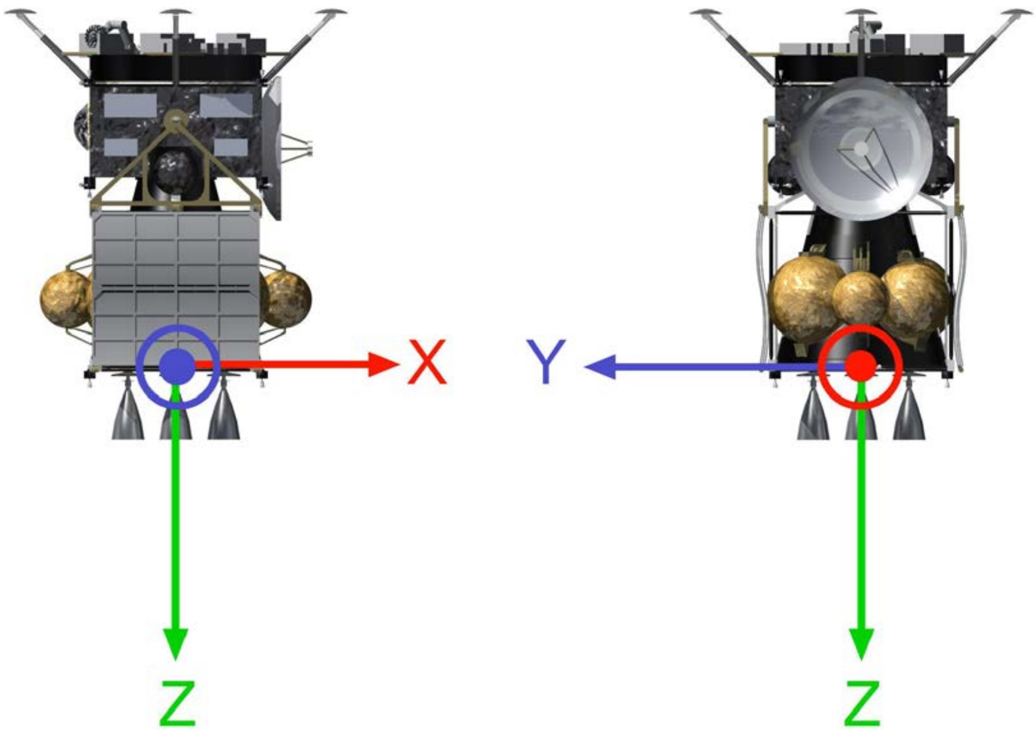


Figure 2.3-1 MMX Co-ordinate System

2.4. Mission phases

2.4.1. General overview

The MMX Mission Scenario will consist of 9 different phases:

- Launch Phase / Early Orbit Phase (see 2.4.2.)
- Mars Transfer Phase (see 2.4.3.)
- Mars Orbit Insertion Phase (see 2.4.4.)
- Phobos Proximity Operation Phase (see 2.4.5.)
- Deimos Rendezvous/Flyby Phase (see 2.4.6.)
- Mars Orbit Escape Phase (see 2.4.7.)
- Earth Transfer Phase (see 2.4.8.)
- Capsule Re-entry Phase (see 2.4.9.)

2.4.2. Launch and early orbit phase

The Launch Phase lasts from 4 h prior to launch until complete mechanical separation of the spacecraft from the upper stage of the launch vehicle.

The Early Orbit Phase extends from spacecraft separation from the launcher upper stage until completion of the initial spacecraft check-out. It includes solar array deployment, initial three-axis attitude acquisition, injection error correction maneuvering, trajectory bias removal maneuvering, and the first commissioning of the spacecraft.

2.4.3. Cruising to Mars

2.4.3.1. General

The Mars Transfer Phase lasts from the end of the Early Orbit Phase until around 1 month prior to MOI. Regular health checks of the PI instruments will be planned.

The interplanetary cruise will be a direct transfer orbit from Earth to Mars. No gravity assist is planned during this phase. The spacecraft will be guided toward the MOI targeting point around Mars by several trajectory correction maneuvers (TCMs) and precise orbit determination including ranging, two-way Doppler, and delta-differential one-way ranging (DDOR).

The duration of the cruise phase will be approximately 11 months according to current candidate profiles. The nominal arrival time will be around July 2025.

2.4.3.2. Attitude manoeuvres and profiles

Attitude maneuvers during the voyage will generally be in the sun-oriented +Z-plane, but this will not necessarily be the case during special scientific observations, etc. Stationary attitude will be set via zero-momentum 3-axis attitude maneuvers using a reaction wheel (RW), and unloading via a reaction control system (RCS) as necessary. Attitude maneuvers are planned to be performed using RW and RCS, but slow spins will be employed as necessary (TBC-Sys/PI).

2.4.4. Mars orbit insertion

MOI consists of three orbit control maneuvers: MOI1, MOI2, and MOI3. In MOI1, the spacecraft will be inserted into an elliptic orbit around Mars with a periapsis altitude of 500 km (TBC-Sys) and an apoapsis of around 40 Mars radii (TBC-Sys). At the apoapsis, MOI2 will be executed to change the orbit inclination to around zero, and to raise the periapsis up to the Phobos orbit. Then MOI3 at the periapsis will reduce the apoapsis to the Phobos orbit radius, finally inserting the spacecraft into the Phobos revolution orbit. Actually, MOI3 will be divided into several maneuvers. During several MOI3 maneuvers, Propulsion Module will be jettisoned after the propellant is exhausted, and the remaining delta-V can be performed by RCSs in Return Module. The module(s) will be jettisoned in a sufficiently stable Mars orbit in order to avoid crashing on Mars to satisfy planetary protection requirements. Detailed sequence is TBD-Sys. The total duration of MOI1–3 will be approximately 2 weeks (TBC-Sys). Commissioning of some PI instruments will be performed after completion of MOI.

2.4.5. Proximity operation around Phobos

2.4.5.1. Phasing

A phasing operation follows MOI to absorb the initial phase, and insertion maneuvers for Phobos rendezvous and high-altitude quasi-satellite orbit (QSO; see Section 2.4.5.3.) are performed. In phasing operations, an altitude difference between Phobos orbit and the Exploration Module orbit is established to absorb the phase difference by changing the orbital period.

2.4.5.2. Attitude profile

During mission phases, the observation (–Z) face of the Exploration Module is generally fixed at an attitude oriented toward the satellite. However, at times such as QSO and descent to the satellite surface, when orienting the observation face toward the satellite does not allow directing the communications antenna toward Earth, an attitude that enables communication with the Earth during visible periods will be prioritized. Three-axis attitude control performance using an RW is described below.

The attitude control performance is designed to achieve the following parameters around Mars:

- Attitude determination accuracy: 0.01 deg (3σ)
- Attitude pointing accuracy: 0.1 deg (3σ)
- Attitude stability: 0.01 deg/s (3σ)

2.4.5.3. Quasi-satellite orbit

<u>High Altitude Orbit:</u>

Operations to maintain a QSO of approximately 50 × 100 km will be performed to allow high-altitude observations of Phobos. The orbital plane of the Exploration Module is nearly the same as that of Phobos (2D-QSO). A QSO can be tailored to the orbital period of Phobos, so operations maintaining a constant attitude of the Exploration Module with respect to Phobos in inertial space are therefore assumed. Orbital phases will be determined in consideration of preserving power and communications. To improve navigational precision in low-altitude regions during high-altitude 2D-QSO, acceleration models will be updated with estimates of the gravitational field and ephemeris of Phobos that are factors contributing to model error.

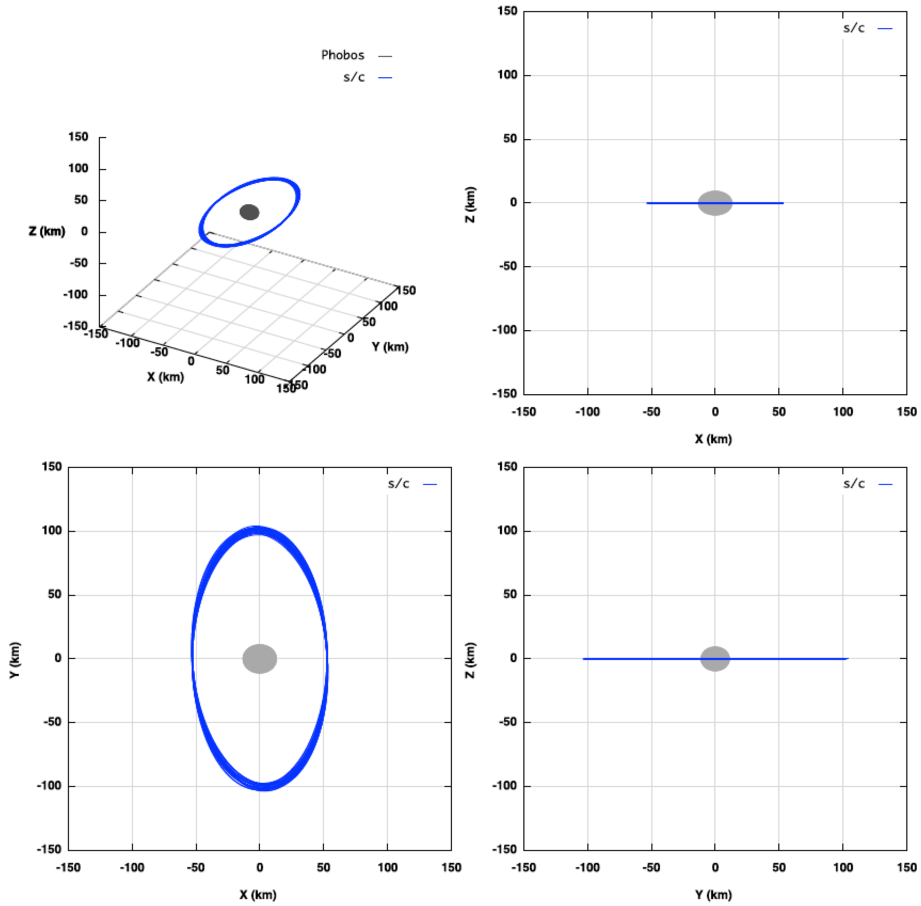


Figure 2.4.5.3-1 Example of High-altitude 2D-QSO (50x100km)

The mission orbit is designed to achieve the following parameters around Mars:

- QSO size: 50 × 100 km
- Inclination: Approx. 0° (with respect to Phobos)
- Duration: 60 days (TBC-Sys/PI)

<u>Low Altitude Orbit:</u>

Operations to maintain a QSO of approximately 24 × 34 km will be performed to allow low-altitude observations of Phobos. A 2D-QSO will generally be used as the standard orbit. Gravitational effects of Phobos cannot be ignored, so maintaining a QSO matching the orbital period of Phobos is difficult. Operations for orbits around Phobos with a given periodicity with respect to inertial space are therefore assumed (this periodicity will vary with orbital altitude above Phobos). To improve navigational precision during descent and landing, physical models will be updated in consideration of the gravitational field and ephemeris of Phobos.

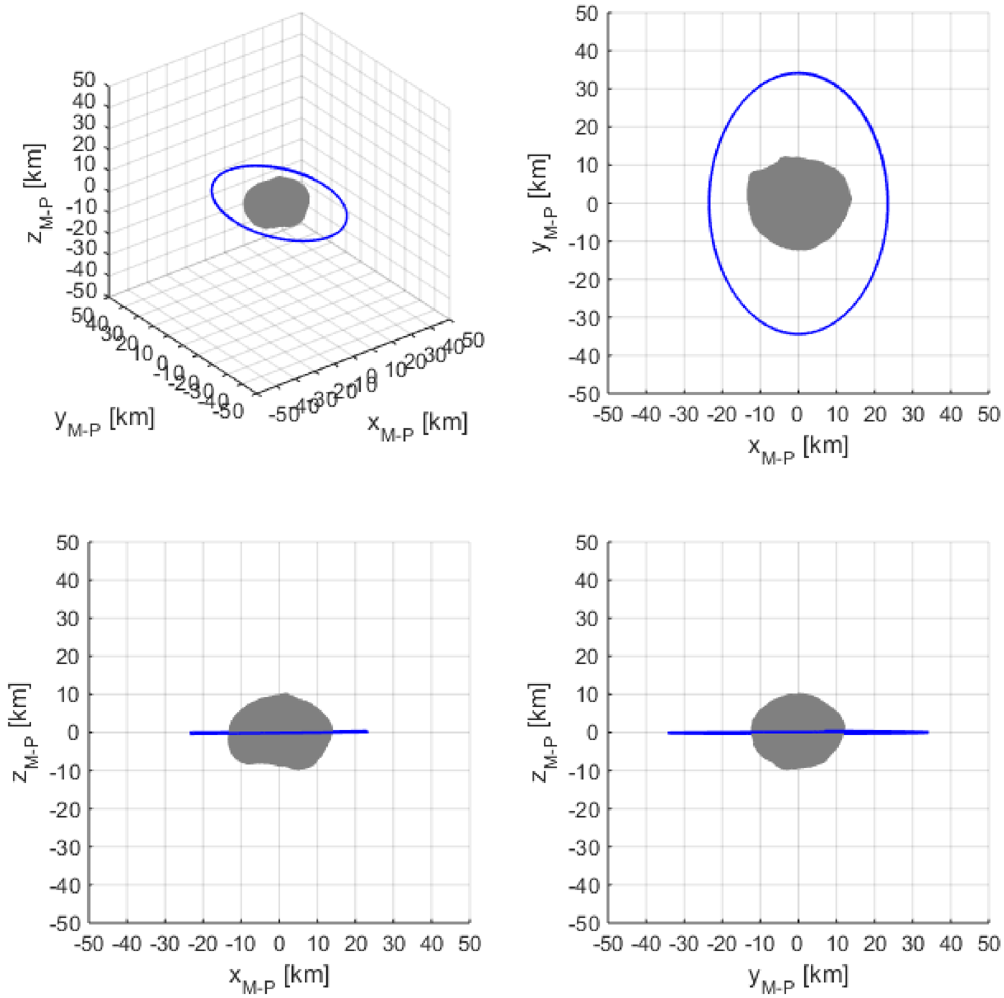


Figure 2.4.5.3-2 Example of Low-altitude 2D-QSO (24x34km)

The mission orbit is designed to achieve the following parameters around Mars:

- QSO size: 24x 34 km
- Inclination: Approx. 0° (with respect to Phobos)
- Duration: Min: 30day (TBC-Sys/PI)

<u>Inclined Orbit:</u>

High-latitude scientific observations of Phobos will be performed by adding an inclined orbit with respect to Phobos. The stability of the QSO varies with the relationship between the relative orbital inclination angle and the relative orbital altitude, so an orbital insertion that provides a stable pseudo-orbit is presumed.

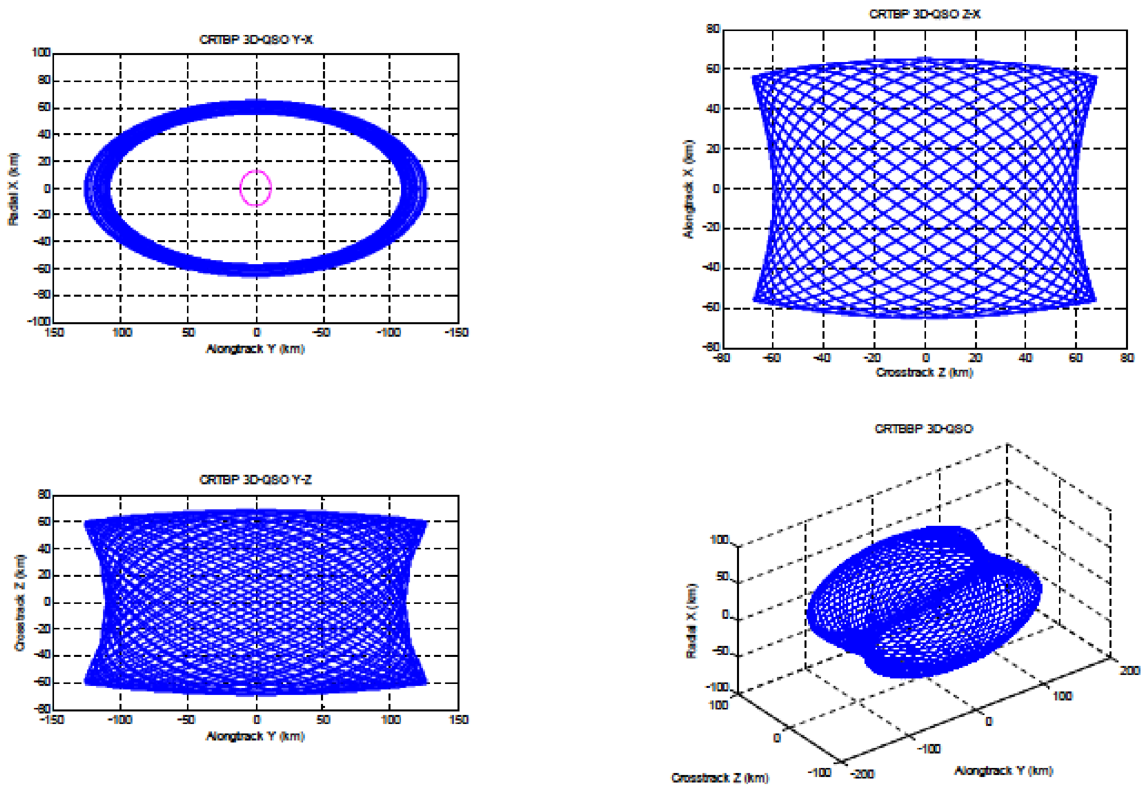


Figure 2.4.5.3-3 Example of High-altitude 3D-QSO (52 × 130 km)

The mission orbit is designed to achieve the following parameters around Mars:

- QSO size: 52 × 130 km
- Inclination: Approx. 47° (max. inclination with respect to Phobos)
- Duration: 30 days (TBC-Sys/PI)

2.4.5.4. Descent trajectory

An example of the descent operation onto the surface of Phobos can be broken down into the following phases.

Table 2.4.5.4-1 Descent Operation Sequence

Category	Phase	Operation Time	Altitude (TBD)	Lateral Velocity	Vertical Velocity
VO-3.1	Initial ΔV for Descent	1min	20km	10~20m/s	~10 m/s
VO-3.2	Ballistic Trajectory-1	60min	20km→15km	10~20m/s	~10 m/s
VO-3.3	TCM	1min	10km(TBD)	10~20m/s	~10 m/s
VO-3.4	Ballistic Trajectory-2	30min	15km→1km	10~20m/s	~10 m/s
VO-3.5	Stop-ΔV	1min	1km	0cm/s	0cm/s
VO-3.6	Lateral Position & Velocity Correction	90min	1km	10~100cm/s	0cm/s
VO-3.7	Vertical Descent	20min	1km→50m	0cm/s	~1 m/s
VO-3.8	TM Separation	1min	50m→40m	0cm/s	~1 m/s
VO-3.9	6 DOF Control	20min	40m→10m	few cm/s	few cm/s
VO-3.10	Hovering	5min	10m	0cm/s	0 cm/s
VO-3.11	Free Fall	5min	10m→1m	0cm/s	~50 cm/s

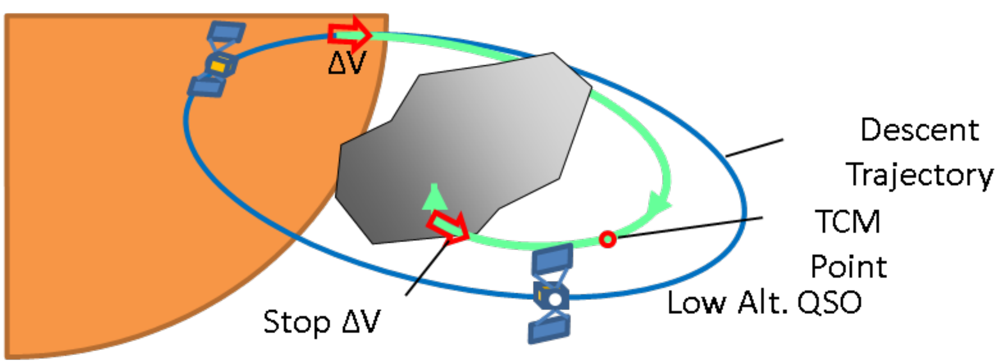


Figure 2.4.5.4-1 Descent Operations

Initially, the spacecraft is assumed to be in a 20-km-altitude quasi-satellite orbit around Phobos, and performs a ΔV to start descending. After a while, it performs a TCM and continues to descend. During ballistic descent, the spacecraft takes images of Phobos for navigation. The images may be downlinked to a ground station for on-ground navigation with a propagation delay, or ideally processed on board for real-time navigation. The spacecraft then performs a stop ΔV to eliminate its velocity with respect to Phobos. The spacecraft may have a large residual velocity after the ΔV due to RCS errors, and the position error can be very large (a few kilometers at worst) if the spacecraft waits for commands from the ground. Onboard optical navigation is therefore required. In the vertical descent phase, the spacecraft descends at 1 m/s at most using an optical navigation camera and Light Detection and Ranging (LIDAR) and performing obstacle detection and avoidance. Target markers (TMs) are released and placed on the surface of Phobos as landmarks for optical navigation. A laser rangefinder (LRF) is used to measure the distance to the surface and local terrain slope at very low altitudes. By using the TM and LRF, 6-degree-of-freedom control is performed to navigate and control the spacecraft to a point just above the target landing site. After a few minutes of hovering, the spacecraft stops using RCS and free falls onto the surface.

2.4.5.5. Landing and sampling operations

Landing operations are operations for landing on the surface of a Mars satellite and using sampling equipment to collect samples. Landing operations involve free-fall descent from 20 m (TBC-Sys), landing at a velocity of 0.5 m/s (TBC-Sys) vertically and 0.1 m/s (TBC-Sys) horizontally with respect to the satellite surface, and stabilization of the position and attitude of the Exploration Module. Sampling operations are performed after landing and stabilization, and involve operation of sampling devices to collect sample specimens. Specimen collection will be controlled to maintain the position and attitude of the Exploration Module. Scientific observations are also planned to be performed during specimen collection.

Table 2.4.5.5-1 lists primary specifications for landing and sampling operations.

In the initial plans, sampling operations will be completed during Phobos daytime, so the duration on the surface of Phobos is taken to be 2.5 h. The initial and final 30 min of that time will be dedicated to confirming the status of the Exploration Module and sampling equipment, so approximately 1.5 h will be available for actual sampling equipment operation.

Table 2.4.5.5-1 Primary specifications for landing and sampling operations

Item	Value (all numeric values TBC-Sys)	Notes
Landing position stabilization tolerance	Within 5 m from contact point	Separate from landing point precision, this refers to tolerance for movement from the initial contact point, due to sliding, bouncing, etc.
Landing position attitude tolerance	Module *z*-axis within ±5° perpendicular to surface at the landing point. Landing point attitude determination precision: ±1° (Phobos-fixed coordinates)	Landing should be at a site where the surface inclination angle is known beforehand. In case of large deviance in attitude from presumed inclination angles (calculated surface position is above or below the ground, etc.), effects on sampling equipment and the danger of toppling during sample collection should be considered. Attitude determination is performed by measuring the direction of gravity using accelerometers, etc.
Final landing velocity	0.5 m/s	Determined from planned free-fall descent from 20 m
Residual horizontal velocity	0.1 m/s	This refers to residual descent velocity that cannot be compensated for through autonomous image-based navigation.

Time after landing	2.5 h (Sampling operation time, 1.5 h)	This may be changed upon creation of a Mars mission phase calendar. Parameters include distance to Earth (propagation delay) and angle to Earth and Sun
Solar angle	TBD-Sys	Fig. TBD-Doc Reference Mission Calendar
Earth angle	TBD-Sys	Fig. TBD-Doc Reference Mission Calendar
Surface temperature	150 K (night)–325 K (day): 0° Lat. 110 K (night)–330 K (day): 30° Lat.	Estimated values from analysis
Landing point inclination	±10°	
Local irregularity at landing point	30 cm (TBC-Sys)	This refers to local height differences due to rocks, indentations, etc., but no observational data is available. Provisionally determined.
Surface conditions	Deep regolith	Surface is estimated to be deep regolith from past observations. From groove cross-sectional observations of the Phobos surface, the regolith layer is estimated to extend more than 100 m.
Area permitting landing	Lat. ±30° of the Mars satellite	TBD-Sys

2.4.5.6. Ascent trajectory

Following landing operations, an ascent ΔV will be performed for escape from Phobos. Then, after observations from a given altitude, a ΔV of 10–20 m/s (TBC-Sys) with respect to the surface of Phobos will be performed to transition the spacecraft to a pseudo-orbit. To ensure that the amount of ΔV for orbit transition will be sufficient to prevent collision with Phobos without correction for several hours (TBD-Sys), calculations will be performed beforehand on the ground, not autonomously. Several hours (TBC-Sys) after ascent, a correctional ΔV will be performed as required.

2.4.6. Deimos multi-flyby and rendezvous

2.4.6.1. Deimos flyby and rendezvous trajectories

After completion of Phobos observation and sampling, several flybys around Deimos or rendezvous with Deimos will be planned. Actual decisions will depend on remaining fuel. In the case of a rendezvous, a Hohman transfer will insert the spacecraft into Deimos-revolution orbit. In the case of a flyby, the apoapsis will be raised up to or beyond the Deimos orbit so that the spacecraft intersects the Deimos orbit, and orbital resonance will be achieved between the spacecraft and Deimos. The spacecraft will encounter Deimos several times in the flyby orbit. The duration of Deimos flybys and rendezvous will be TBD-Sys/PI.

2.4.6.2. Attitude profile and observation

Observations of Deimos during flybys are also listed as part of the mission parameters. The relative velocity of Deimos at the time of flyby will be extremely high (approximately 1 km/s), so attitude maneuvers will be performed in accordance with the orbital motion to allow observation and tracking of a point on the surface of Deimos. Assuming a 3:1 orbit with respect to the orbital period of Deimos, the relative speed will be 1150 m/s (TBC-Sys/PI). For that case, Figure 2.4.6.2-1 shows the geometric relation between the closest approach distance and maneuver rate for tracking a point on the surface, and Figure 2.4.6.2-2 shows the quantitative relation.

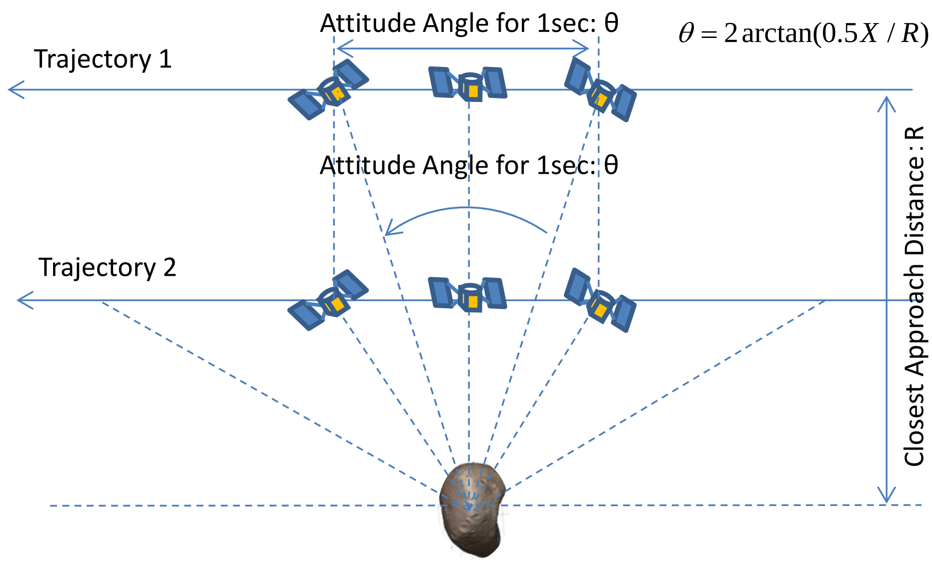


Figure 2.4.6.2-1: Schematic of Deimos fly-by

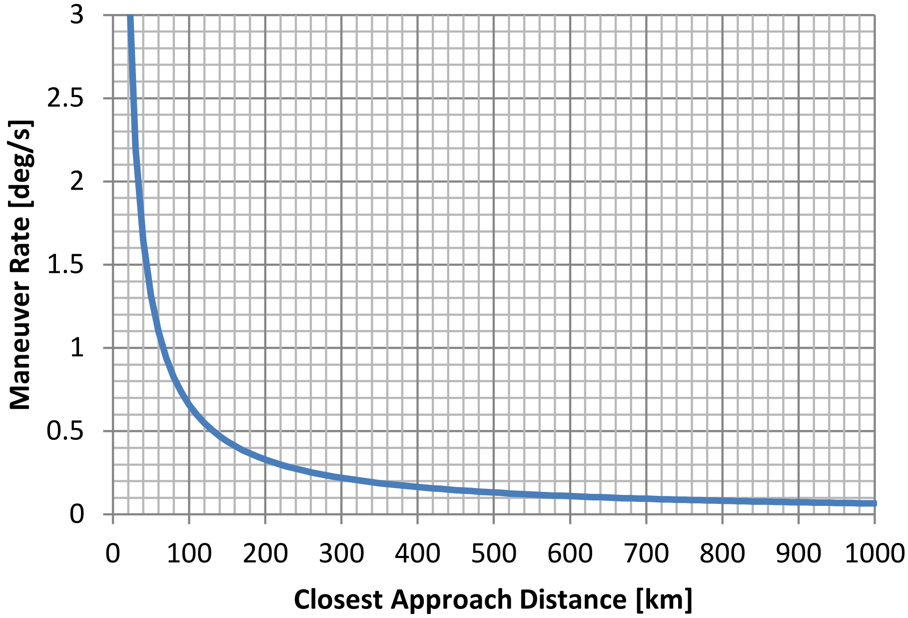


Figure 2.4.6.2-2: Maneuver Rate and Relative Distance

2.4.7. Mars orbit escape

Mars orbit escape (MOE) consists of three orbit control maneuvers: MOE1, MOE2, and MOE3. Prior to MOE maneuvers, the spacecraft is around Deimos orbit. In MOE1, the apoapsis will be raised up to around 40 Mars radii (TBC-Sys). At apoapsis, the subsequent MOE2 maneuver will change the orbit inclination in preparation for escape, lowering the periapsis altitude to 500 km (TBC-Sys). Then the MOE3 burn at periapsis will finally insert the spacecraft into an interplanetary trajectory, escaping Mars. The total duration of MOE1–3 will be approximately 2 weeks (TBC-Sys). The exploration module may be jettisoned after final scientific observation and before MOE1; it depends on the module configuration. The module(s) will be jettisoned in a sufficiently stable Mars orbit in order to avoid crashing on Mars to satisfy planetary protection requirements. The detailed timing of jettison will be TBD-Sys.

2.4.8. Cruising to Earth

2.4.8.1. General

The Earth Transfer Phase lasts from the end of MOE until around 1 month prior to capsule re-entry to Earth.

The interplanetary cruise will be a direct transfer orbit from Mars to Earth. No gravity assist is planned during this phase. The spacecraft will be guided toward the capsule separation targeting point around Earth by several TCMs and precise orbit determination, including ranging, two-way Doppler, and DDOR.

The duration of the cruise phase will be approximately 11 months, according to candidate profiles.

The nominal arrival time will be around June 2029.

2.4.8.2. Attitude maneuvers and profiles

Attitude maneuvers and profiles are as described in Section 2.4.3.2.

2.4.9. Capsule re-entry to Earth

Near Earth, the spacecraft will target the re-entry interface point. Separation of the capsule from the spacecraft will be performed several hours prior to Earth atmosphere re-entry. The capsule will re-enter the Earth atmosphere, descend using a parachute, land on the ground, and be recovered immediately. After capsule separation, the spacecraft will de-orbit from the trajectory targeting the interface point using the chemical propulsion system to escape from Earth gravity to interplanetary space.

2.5. Communication links

2.5.1. General communication configuration

The communication configuration, requirements, and features are described in the following sections.

2.5.2. MMX to ground communication

(1) Operations

Spacecraft system telemetry, command, and ranging are performed by the following ground stations.

- JAXA stations: USC34m, UDSC64m, New Usuda Station
- Overseas stations: TBD-Sys

(2) Communications

- Transmission rates in the X band will realize an uplink of at least 1 kbps and downlink of 32 kbps (TBD-Sys) at a distance of 2.7 AU from Earth
- Transmission rates in the Ka band will realize a downlink of at least 128 kbps (TBD-Sys) at a distance of 2.7 AU from Earth. Note that the Ka band is primarily used for transmission of mission data from the vicinity of Mars. No Ka-band uplink is planned; designs should allow for minimal telecommand operation via X band only.
- Simultaneous X- and Ka-band communications will be accommodated.

(3) Ranging

- In all phases from first visual to Mars arrival (2.7 AU) and return to Earth, X-band ranging shall be possible.
- Ranging is performed by switching the spacecraft from communications mode to measurement mode while in the visual path, then performing ranging.
- Tone production features for DDOR shall be provided.

3. DOCUMENTS

3.1. Applicable documents

N/A

3.2. Reference documents

N/A

4. ACRONYMS

DDOR	Doppler and delta delta-differential one-way ranging
DOF	Degree of freedom
JAXA	Japan Aerospace Exploration Agency
LIDAR	Light detection and ranging
LRF	Laser Ranging Finder
MMX	Martian Moons Exploration
MOE	Mars Orbit Escape
MOI	Mars orbit insertion
OME	Orbital maneuvering engine
PI	Principal investigator
QSO	Quasi-satellite orbit
RCS	Reaction control system
RW	Reaction wheel
TCM	Trajectory correction maneuver
TCS	Thermal control sub-system
TM	Target marker
UDSC	Usuda Space Center
USC	Uchinoura Space Center