Space Mission Design Report
Remote Orbital Utility for Terrain Exploration

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ROUTE-M

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List of Acronyms

Acronym	**Full Form**
ADCS	Attitude Determination and Control System
BER	Bit Error Rate
C3	Characteristic Energy
CBE	Current Best Estimate
CDR	Critical Design Review
CMOS	Complementary Metal-Oxide Semiconductor
CONOPS	Concept of Operations
CRISM	Compact Reconnaissance Imaging Spectrometer for Mars
CSA	Canadian Space Agency
CTX	Context Camera
Δv	Change in Velocity (Delta-V)
DSN	Deep Space Network

EMC	Electromagnetic Compatibility
FRR	Flight Readiness Review
GEVS	General Environmental Verification Standard
Gb	Gigabit
GB	Gigabyte
GHz	Gigahertz
GSD	Ground Sampling Distance
GSFC	Goddard Space Flight Center
HiRISE	High Resolution Imaging Science Experiment
I&T	Integration and Test
IMU	Inertial Measurement Unit
Isp	Specific Impulse
JPEG2000	Joint Photographic Experts Group 2000 Image Compression Standard

JPL	Jet Propulsion Laboratory
Ka-band	Ka Frequency Band (26–40 GHz)
kbps	Kilobits per second
MLI	Multi-Layer Insulation
MOI	Mars Orbit Insertion
MRD	Mission Requirements Document
MRO	Mars Reconnaissance Orbiter
NX	Siemens NX Thermal Modeling Software
ORR	Operational Readiness Review
PDR	Preliminary Design Review
PSNR	Peak Signal-to-Noise Ratio
QA	Quality Assurance
RF	Radio Frequency

ROUTE-M	Remote Orbital Utility for Terrain Exploration
SHARAD	Shallow Radar
SRR	System Requirements Review
SSO	Sun-Synchronous Orbit
STK	Systems Tool Kit
STM	Science Traceability Matrix
TCM	Trajectory Correction Maneuver
TMI	Trans-Mars Injection
TRL	Technology Readiness Level
TVAC	Thermal Vacuum
ULA	United Launch Alliance
v_∞	Hyperbolic Excess Velocity
WBS	Work Breakdown Structure

1.0 Project Description

Mars is believed to have once had a thicker atmosphere and abundant surface water, potentially creating conditions suitable for life. Over time, atmospheric loss caused the planet to cool and dry, resulting in the arid environment we see today. Studying Mars' climate, geology, and global dust storms helps scientists understand how planetary climates evolve and provides insight into long-term processes such as desertification, which is also becoming a major problem on Earth due to climate change. Understanding these changes can also improve our ability to identify habitable environments on planets beyond our solar system.

High-resolution mapping of Mars is essential for advancing both scientific research and future exploration. Detailed surface maps would allow scientists to study geological features, identify evidence of past water activity, and select safe landing sites for future missions. While previous Mars orbiters such as Mars Reconnaissance Orbiter (MRO) have provided valuable datasets, current capabilities are limited to either high-resolution imaging over small targeted regions (e.g., HiRISE) or global coverage at lower spatial resolution (e.g., CTX Camera). There is currently no mission optimized to provide sustained global coverage at ~1 meter spatial resolution with high revisit frequency. The proposed ROUTE-M mission aims to address this limitation by enabling high-resolution, global-scale mapping of the Martian surface with improved temporal revisit and coverage efficiency. This approach supports both scientific investigations and future robotic and human exploration of Mars.

The ROUTE-M mission involves several stakeholders whose interests align with advancing scientific knowledge and supporting Mars exploration. Primarily, our stakeholders include space agencies such as the National Aeronautics and Space Administration (NASA), the European Space Agency (ESA), and other international partners responsible for funding, mission planning and scientific oversight. Specifically, these organisations rely on high-quality planetary datasets to support scientific investigations and guide long-term exploration strategies. Additionally, the planetary science community is an important stakeholder, including researchers studying planetary geology, atmospheric processes and astrobiology. High-resolution orbital imagery enables scientists to analyse geological formations, investigate evidence of past water activity and better understand the climatic and atmospheric evolution of Mars. Future robotic and human exploration missions are also key stakeholders.

ROUTE-M is a Mars orbital mapping mission designed to produce a high resolution global map of the Martian surface to support autonomous rover navigation and high confidence orbital operations. The mission targets a spatial resolution of approximately 1 meter/pixel and global surface coverage to enable both scientific analysis and operational planning.

The mission duration is planned for 9 years including cruise, orbit insertion and systematic surface mapping.

1.1 Motivation for the Project

High-resolution orbital imaging has fundamentally advanced the understanding of the Martian surface morphology, hazard identification, and terrain classification. Missions such as the MRO demonstrated the scientific and operational value of sub-meter surface imagery through instruments like HiRISE. However, prior missions have primarily focused on:

1. Targeted Science Campaigns
2. Landing Site Characterization
3. Regional investigations

There remains no prior mission that has been optimized for continuous, high-resolution global mapping to support autonomous navigation and mission operations. Existing datasets lack the combination of global coverage, high spatial resolution, and temporal revisit required for next-generation Mars exploration systems.

As future Mars missions transition towards increased autonomy, and globally consistent high definition terrain models become critical for:

1. Autonomous surface path planning
2. Hazard avoidance and landing site certification
3. Descent and landing risk assessment
4. Orbital maneuver assurance and safety

ROUTE-M is designed to address this operational gap.

1.2 Mission Concept and Architecture Overview

The ROUTE-M mission consists of a dual-orbiter architecture operating in low Mars orbit at an altitude of approximately 300 km. The spacecraft is designed to acquire high-resolution optical imagery of the Martian surface, enabling global mapping at approximately 1 meter spatial resolution.

Imaging is performed using a high-resolution optical instrument operating in the visible and near infrared spectrum during orbital passes. Forward off nadir pointing is utilized to improve surface coverage efficiency and enable enhanced terrain mapping.

Structural design is a key driver of imaging performance as structural stability directly affects payload alignment and pointing accuracy. The spacecraft must maintain structural integrity and minimize vibration induced disturbances throughout launch and orbital operations to ensure high-quality image acquisition.

The mission is constrained by data handling and communication limitations, requiring efficient onboard data processing, storage, and transmission strategies to maximize science return within available downlink opportunities.

1.3 Concept of Operations

1.3.1. Launch and Earth Parking Orbit

The mission begins with launch from Earth, followed by insertion into a ~300 km circular parking orbit. This phase enables initial system checkouts, including power, communication, and avionics verification. The parking orbit also provides flexibility in timing the trans-Mars injection (TMI) burn. Once systems are confirmed nominal, the spacecraft performs the TMI maneuver to begin the interplanetary transfer.

1.3.2. Interplanetary Transfer

The spacecraft travel to Mars using a Hohmann transfer trajectory with an approximate time of flight of 259 days. During cruise, periodic trajectory correction maneuvers (TCMs) are performed to maintain the desired path. The ADCS ensures proper orientation for power generation, thermal control, and communication, while routine health monitoring is conducted throughout the phase.

1.3.3. Mars Orbit Insertion (MOI)

As the spacecraft approach Mars, final navigation updates and attitude adjustments are performed. The spacecraft separate if required and execute a Mars Orbit Insertion (MOI) burn to enter Martian orbit. This critical maneuver transitions the spacecraft from a heliocentric trajectory into a captured orbit, typically initialised as elliptical for further adjustment.

1.3.4. Orbital Deployment and Configuration

Post-insertion, orbit adjustment maneuvers place the spacecraft into their operational orbits. One spacecraft is inserted into a near-polar orbit (~93°) for global coverage, while the other operates at ~45° inclination for complementary observations. Payloads are deployed and calibrated in preparation for science operations.

1.3.5. Science Mapping Operations

In the primary science phase, both spacecraft acquire high-resolution imagery of the Martian surface for geological analysis and global mapping. The dual-spacecraft configuration improves coverage, revisit time, and data quality. Operations are largely autonomous with scheduled observation windows and onboard data storage.

1.3.6. Data Handling and Transmission

Collected data are processed and compressed onboard before transmission to Earth via the Deep Space Network (DSN). Communication sessions are scheduled based on orbital geometry and ground station availability. Telemetry and command operations support continuous spacecraft monitoring and mission operations.

1.4 Modes of Operation

1. Launch & Early Operations - After launch, the spacecraft deploys solar arrays, establishes communication with Earth, and performs system health checks before beginning the transfer to Mars.

2. Cruise/Interplanetary transfer - During the Earth-Mars Hohmann transfer, the spacecraft operates in a low-power cruise configuration while performing navigation updates and trajectory correction maneuvers.

3. MOI - Upon arrival at Mars, the propulsion system performs a Mars Orbit Insertion burn to capture the spacecraft into the planned ~300 km mapping orbit.

4. Orbit Commissioning - After orbit insertion, instruments and subsystems are calibrated and the spacecraft are positioned into their respective near-polar and mid-inclination mapping orbits.

5. Science Operations - The imaging payload captures high-resolution images of the Martian surface during daylight passes with up to 15° off-nadir pointing to maximize mapping coverage.

6. Data Downlink - Captured images are stored, compressed onboard, and transmitted to Earth using Ka-band communications while telemetry and commands are handled through X-band links.

1.5 Proposed Flight Path

The interplanetary transfer trajectory is illustrated in Figure 1 showing the Earth to Mars Hohmann transfer used to deliver the spacecraft to its target orbit. The trajectory begins at Earth departure where the spacecraft is injected onto a heliocentric transfer orbit by the launch vehicle. During the transfer phase, the spacecraft follows an energy-efficient elliptical path that intersects Mars' orbit at the arrival point. A separation event is incorporated along the trajectory to deploy the two spacecraft, after which small inclination adjustment maneuvers (Δv) are applied to place each orbiter into distinct orbital planes. Upon arrival at Mars, the spacecraft performs a Mars Orbit Insertion maneuver to transition from the heliocentric trajectory into a bound Martian orbit. This transfer architecture minimizes propellant usage while enabling the dual-orbiter configuration required for increased surface coverage.

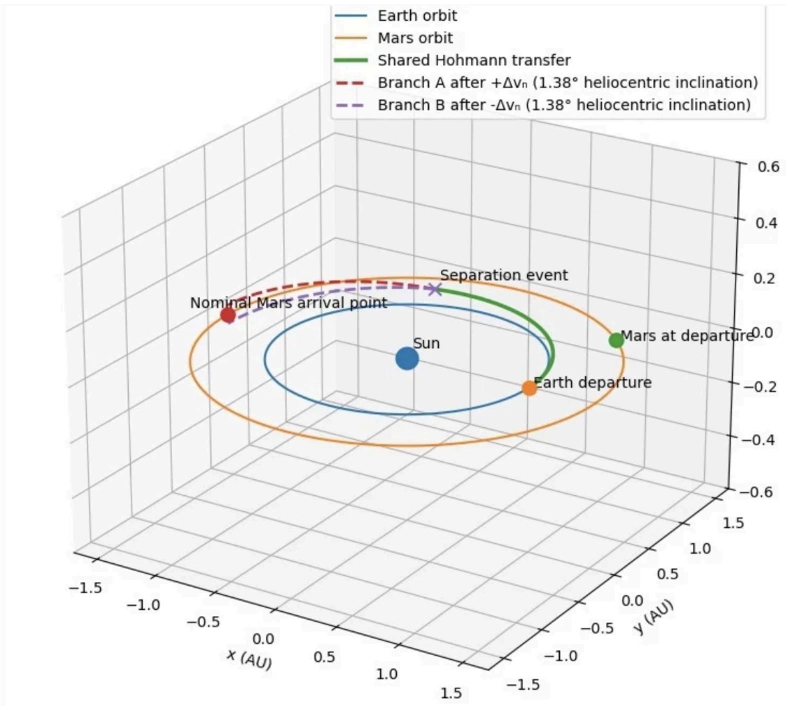


Figure 1: Proposed Flight Path

2.0 Subsystem descriptions and correlation

This section describes the major spacecraft subsystems within the ROUTE-M mission architecture, including payload, structural, communications, command and data handling, power, thermal, and propulsion. Each subsystem is presented in terms of its functional role, key performance drivers, and interactions with other subsystems.

The performance of the overall mission is strongly dependent on the integration of these subsystems as limitations or constraints in one subsystem directly impact the capabilities of others. This interconnected architecture drives the decomposition of mission-level requirements into traceable and verifiable subsystem requirements.

As a result, subsystem design must be approached from a system-level perspective to ensure balanced performance across all mission constraints.

2.1 Payload Subsystem

The payload subsystem is responsible for acquiring high-resolution imagery of the Martian surface to support the mission objective of global mapping. The payload operates in the visible and near infrared spectral range and captures multispectral image data across three different channels, enabling accurate colour reconstruction, surface feature identification, and scientific analysis.

The subsystem operates during orbital passes and supports off-nadir imaging to improve surface coverage efficiency and allow for terrain modelling using the parallax effect. Payload performance is driven by key parameters including spatial resolution, imaging rate, spectral coverage, and geolocation accuracy.

The payload subsystem interfaces with multiple spacecraft subsystems, such as the Attitude Determination and Control System (ADCS) subsystem for pointing and stabilization, the onboard data handling system for image processing and storage, and the communication subsystem for data transmission back to Earth. It is also dependent on the thermal subsystem to maintain operating temperatures and conditions within allowable limits.

The performance of the payload subsystem is highly sensitive to spacecraft disturbances, including pointing jitter and structural vibrations, which can degrade image quality, reduce effective resolution, and impact geolocation accuracy.

As a result, payload performance imposes strict requirements on spacecraft stability, pointing accuracy, and data-handling capability.

2.2 Structural Subsystem

The structural subsystem is responsible for maintaining mechanical stability and alignment of all spacecraft components throughout launch and on-orbit operations. Its performance is a key driver of payload imaging quality as structural stability directly affects pointing accuracy and optical alignment.

The subsystem must withstand launch loads, dynamic disturbances, and on-orbit perturbations while maintaining structural integrity and minimizing deformation. Structural rigidity is critical to limiting vibration-induced disturbances that can degrade image resolution and geolocation accuracy.

The subsystem also supports spacecraft packaging constraints, including accommodation within the launch vehicle payload fairing and integration of all major subsystems. It provides mechanical interfaces for integration of the payload, ADCS, communications, propulsion, and thermal subsystems.

Structural performance is closely coupled with thermal behaviour, as temperature variations can induce expansion or contraction that affects alignment. As a result, the structure must be designed to minimize thermoelastic distortions and maintain stable payload geometry across all operating conditions.

2.3 Communications Subsystem

The communications subsystem serves as the primary data link between the ROUTE-M orbiters and the Deep Space Network (DSN) on Earth. It is responsible for the bidirectional transfer of information, including the downlink of processed science imagery and the uplink of mission-critical commands. To ensure mission reliability across the distances of the interplanetary environment, the subsystem uses a dual-band radio frequency architecture. This approach provides a dedicated high-bandwidth channel for science data return and a separate link for spacecraft telemetry, health monitoring, orbiter status, command uplink, and emergency commanding.

The subsystem interfaces directly with the Attitude Determination and Control System (ADCS) to enable precise pointing of high-gain antennas, which is critical for maintaining link performance at interplanetary distances. The subsystem also interfaces with the Power Subsystem to support the high electrical loads required for deep-space transmission. Lastly, the subsystem also relies on the Thermal Subsystem to manage heat dissipation during high-power communication windows.

The performance of the communications subsystem is primarily governed by the orbital geometry between Earth and Mars. The varying distance and signal-path loss dictate the available data throughput, making the efficiency of the antenna gain and transmitter power the critical drivers for the mission's science return capacity.

2.4 Command and Data Handling Subsystem

The Command and Data Handling (C&DH) subsystem enables the primary processing and data management infrastructure for the spacecraft. It is responsible for executing the flight software, distributing ground commands to all subsystems, and monitoring the overall health and status of the vehicle.

This subsystem manages the flow of information between the payload and the rest of the spacecraft, receiving high-rate digital data from the imaging instruments and preparing it for storage and eventual transmission. It serves as the main interface for the Payload Subsystem, facilitating the transfer of imagery to the onboard storage media. It manages the Solid State Recorder (SSR), which functions as a buffer to hold science data between scheduled communication passes. The subsystem performs onboard data processing tasks, including format conversion and data compression, to optimize the use of the available downlink bandwidth.

C&DH performance is driven by the high data generation rate of the payload and limited communication windows requiring efficient data management, prioritization, and storage strategies.

The reliability of the C&DH subsystem is critical for mission longevity, as data integrity and command execution must be maintained throughout the mission lifetime. The subsystem must also ensure protection of onboard memory from the radiation environment in Mars orbit.

2.5 Attitude Determination and Control Subsystem

The Attitude Determination and Control System (ADCS) is responsible for maintaining spacecraft orientation to support both imaging and communication operations. Its performance is critical to mission success as precise pointing and stability are required to achieve the desired image resolution and maintain reliable communication links with Earth.

The subsystem must support accurate off-nadir pointing up to ±15 degrees to enable efficient surface coverage and terrain mapping. In addition, it must provide stable pointing during imaging operations and ensure accurate antenna alignment during communication windows.

ADCS performance is driven by strict requirements on pointing accuracy and stability, as even small disturbances can degrade image quality and reduce geolocation accuracy. The subsystem must also operate effectively in the presence of external disturbances, including solar radiation pressure and momentum buildup over time.

Given the long mission duration, the ADCS must maintain reliable performance throughout all operational phases and support safe mode functionality to ensure spacecraft stability under off-nominal conditions.

The subsystem design leverages heritage approaches from previous planetary missions requiring high-precision pointing.

2.6 Electrical Power and Thermal Subsystem

The Electrical Power and Thermal subsystems are responsible for enabling continuous and reliable spacecraft operation in Mars orbit. The electrical subsystem generates, stores, and distributes power to all spacecraft systems, including payload imaging, onboard processing, communication, and attitude control.

During sunlit portions of the orbit, solar arrays generate power to support spacecraft operations and recharge onboard batteries. During eclipse periods, when solar power is unavailable, stored battery energy must support all critical spacecraft functions. This requires careful power management, including operating in a reduced-power mode during eclipse, where non-essential subsystems are powered down or placed in standby. Maintaining the payload in a low-power standby mode during eclipse provides beneficial internal heat generation, reducing heater demand and improving thermal stability.

The thermal subsystem maintains all spacecraft components within allowable temperature limits under both sunlit and eclipse conditions. During sunlit operations, excess heat generated by onboard systems must be rejected. During an eclipse, insulation and heaters are used to maintain survival temperatures. Thermal control is achieved using a combination of passive controls (e.g., multi-layer insulation) and active controls (e.g., heaters and radiators).

The electrical power and thermal subsystems are tightly coupled as power availability directly affects thermal control capability, particularly during eclipse conditions. Their coordinated operation is critical to ensuring reliable spacecraft performance across all mission phases.

2.7 Propulsion Subsystem

The propulsion subsystem is responsible for enabling trajectory correction, Mars Orbit Insertion (MOI), and long-term orbit maintenance. It is critical for achieving and maintaining the required 300 km mapping orbit throughout the mission lifetime.

The subsystem must support orbit insertion upon arrival at Mars, perform trajectory correction maneuvers during interplanetary transfer, and maintain orbital altitude by compensating for perturbations over time. Its performance is constrained by mission ΔV requirements, spacecraft mass limitations, and long-duration operational demands.

The Mars Orbit Insertion maneuver represents the highest-risk phase of the mission, requiring precise execution to ensure successful capture into the target orbit. Failure during this phase would result in complete mission loss, making propulsion reliability a critical design driver.

Over the mission lifetime, the propulsion subsystem must also support periodic orbit maintenance and momentum management while minimizing propellant consumption to ensure sustained operation. As a result, propulsion system performance directly impacts mission duration, orbital stability, and overall mission success.

The subsystem design leverages heritage propulsion technologies to reduce technical risk and ensure reliable performance during critical mission phases.

3.0 Requirements Derivation and Formulation

These requirements are formulated to ensure traceability from mission objectives to subsystem-level performance, enabling verification through analysis, testing, and inspection. They define the key performance goals of the mission, including high-resolution surface mapping, global coverage, and associated operational constraints. The following science traceability matrix was created to ensure all objectives are met and all requirements are being traced back to a science objective:

Table 1: Science Traceability Matrix

Science Goals		Science Objectives		Scientific Measurement Requirements		Instrument Performance Requirement	Projected Instrument Performance	Top level Mission Requirements
Unique ID	Description	Unique ID	Description	Observable	Physical Parameters			
Goal 1	Develop a high-resolution map of the	1.1	Create a topographic map that allows to	Calibrated surface imagery in visible bands	Camera to be tilted at 15 degrees	pixel pitch must be within the manufacturable	Projected pixel pitch must be between the	The mission shall acquire stereo benefit using parallax to recover terrain

	martian surface to support Autonomous Navigation, Future landing site selection and geological science analysis		identify elevation changes			space imaging range	typical CMOS range of 5-15 um	height
		1.2	Acquire high resolution imagery of the martian surface		Ground Sampling Distance of 1 Meter per pixel		focal length must be greater than or equal to 300000m*pixel pitch	The mission shall acquire imagery of the Martian surface with a GSD of 1m/pixel within the visible band
		1.3	Provide georeferenced map tiles aligned to a mars projection system so products are usable by other missions	Georeferenced map tiles	Geolocation accuracy smaller than or equal to 150m	The attitude knowledge error should be smaller than 150/altitude	geolocation of between 50-150 m depending on chosen start tracker	
Goal 2	Demonstrate on orbit image compression and reliable Mars to Earth downlink to return high-resolution maps of the	2.1	Demonstrate onboard image preprocessing that preserves mapping utility	Raw Image bit depth and processed image bit depth	14 Bit images to 8 Bit images	The instrument must be able to process 14bit images to 8bit images		The Spacecraft must implement onboard preprocessing and compression of acquired imagery that includes 14-bit to 8-bit image conversion and 8bit to JPEG2000 with 10x compression

	Martian surface within a defined daily data budget	2.2	Demonstrate onboard compression achieving a compression ratio of 10 while meeting PSNR of at least 35dB	Compressed file size and image quality metric processing	8 Bit images to JPEG2000 with 10x compression above 35dB PSNR Rating	Compression ratio of 10:1		The spacecraft shall implement onboard image preprocessing and compression of acquired imagery, including 14-bit to 8-bit radiometric conversion followed by JPEG2000 compression at a minimum 10:1 ratio while achieving a PSNR rating of at least 35 dB.
						PSNR must be greater than or equal to 35dB		
		2.3	Achieve a sustained average delivered data volume of 3.5GB/Day	Delivered data volume to Earth on average	Above or equal to 3.5GB/day of transmission	The Orbiter must downlink at least 3.5GB/day with atleast 99% successful delivery		The mission shall achieve above or greater than 3.5GB/day of data transmission with 99% successful file delivery of downlinked image products
Goal 3	Deliver the spacecraft from Earth to a stable 300km circular Mars mapping orbit using a defined interplanetary transfer	3.1	Execute an interplanetary Earth-Mars transfer trajectory that achieves Mars arrival within the planned mission timeline.	Transfer trajectory performance	Time of Flight ≈ 259 days Earth departure hyperbolic excess velocity $v_{\infty,E} \approx 2.945$ km/s $C3 \approx 8.67$ km²/s²	Spacecraft propulsion system must achieve $\Delta v_{TMI} \approx 3.58$ km/s from a 350 km circular Earth parking orbit.	Launch vehicle must support $C3 \geq 8.7$ km²/s².	The mission shall execute a Hohmann Earth–Mars transfer with a time of flight of approximately 259 days and departure energy consistent with $C3 \approx 8.7$ km²/s².

	trajectory while operating within the allocated launch energy and propulsion budget constraints.	3.2	Insert the spacecraft into a stable, low Mars circular mapping orbit suitable for sustained imaging operations.	Achieved Mars orbital parameters	Target circular orbit altitude: 300 km Orbit radius: 3696 km Mars arrival hyperbolic excess velocity v∞,M≈2.649 km/s Required MOI ΔThe propulsionkm/s	Propulsion system must provide Δv_MOI ≈ 2.09 km/s to achieve direct circular capture at 300 km altitude.	Orbital stability sufficient for sustained 7-year mapping operations.	The spacecraft shall achieve and maintain a stable 300 km circular Mars orbit to enable global surface mapping.
		3.3	Operate within defined propulsion systems limitations and Δv budget constraints.	Total Δv budget consumption	Total impulsive Δv (TMI + MOI) ≤ 5.671 km/s (excluding correction margins) Propellant mass consistent with selected Isp and dry mass assumptions	Propulsion system shall meet total Δv requirement within allocated propellant mass fraction.	Mass ratio consistent with rocket equation and propulsion selection.	The total mission Δv shall not exceed 5.671 km/s excluding mid-course correction margins.

The following mission requirements table presents the derived requirements for the payload and structural subsystems.

Table 2: Mission Requirements for Payload and Structural system

Mission Requirements (Structural and Payload)	
Requirements Identifier	**Description**
MS-PAYL-001	The mission shall acquire imagery of the Martian surface with a ground sampling distance of ≤ 1 m/pixel from an average orbital altitude of 300 km.
MS-PAYL-002	The mission shall acquire imagery in the visible spectral range (400–900 nm) to support mapping, colour reconstruction, and surface feature identification
MS-PAYL-003	The mission shall support forward off-nadir imaging up to 15°.
MS-PAYL-004	The mission shall produce georeferenced map products with positional accuracy sufficient to support autonomous navigation and scientific analysis.
MS-PAYL-005	The mission shall achieve at least 90% global surface coverage over the mission lifetime.
MS-PAYL-006	The mission shall ensure that science data generation does not exceed onboard storage capacity or available downlink capability.
MS-PAYL-007	The mission shall maintain payload operating temperatures within defined thermal limits during all operational phases.
MS-STRUC-001	The structure shall survive launch, ascent and separation environments without structural failure or loss of mission function
MS-STRUC-002	The structure shall maintain payload alignment and structural stability to preserve imaging performance requirements.
MS-STRUC-003	The structure shall package the spacecraft within the launch vehicle fairing envelope.
MS-STRUC-004	The structure shall support the dual spacecraft launch architecture
MS-STRUC-005	The structure shall accommodate both stowed and deployed configurations required for launch and on-orbit operations.

3.1 Payload Requirements Derivation

The following list of requirements has been derived for the Payload Subsystem:

Table 3: Payload Requirements

Payload Requirements				
Requirements Identifier	**Description**	**Derived From**	**Verification Method**	**Notes**
PAYL-001	The payload shall achieve a ground sampling distance of ≤ 1 m/pixel from an orbital altitude of 300 km.	Mission objective, variance derived from the current accuracy levels of HiRise	Analysis (optical modeling, geometry) + Inspection (design parameters)	Mission Objective
PAYL-002	The payload shall support forward off-nadir imaging up to 15° with a pointing accuracy of ±1°.	Derived from Off Nadir Camera Calculations	Analysis (coverage modeling, STK) + Test (ADCS pointing validation)	ADCS Requirement
PAYL-003	The payload shall operate within the visible spectral range of 400–900 nm.	Allows for imaging in the Visible and the VIR Bands allowing for full colour reconstruction	Analysis (mission simulation) + Test (operational validation)	Science Objective
PAYL-004	The payload shall support an image acquisition rate sufficient to achieve the required global coverage over the mission lifetime.	Derived from data rate calculations that require at least 363 Images during peak image acquisition window	Analysis (mission simulation) + Test (operational validation)	Operations Requirement
PAYL-006	The payload shall generate data volumes compatible with onboard storage and communication subsystem capabilities.	Derived from Image Downlink Compression Calculations	Analysis (compression modeling) + Test (data processing validation)	Data Budget

PAYL-007	The payload shall generate science data compatible with available daily downlink capacity.	Derived from communications system architecture and available downlink rates	Analysis (data budget) + Test (end-to-end data flow validation)	Data Budget
PAYL-008	The payload shall maintain imaging performance under spacecraft pointing jitter less than or equal to 5 micro rads.	Jitter Estimation Calculations 5.2.11	Analysis (jitter-to-ground error model) + Test (ADCS stability testing)	ADCS Requirement
PAYL-009	The payload shall achieve geolocation accuracy of ≤ 150 m without reliance on ground control points.	Derived from Science Objective 1.3 in the Science Traceability Matrix	Analysis (error propagation) + Test (navigation validation)	Mission Requirement
PAYL-010	The payload shall operate within thermal limits of -10 degrees celsius to 40 degrees celsius during nominal operations	Derived from Thermal Calculations within hot and cold cases	Analysis (thermal modeling) + Test (TVAC testing)	Thermal Requirement
PAYL-011	The payload shall operate within thermal limits of -20 degrees Celsius to 50 degrees Celsius during eclipse	Derived from Thermal Calculations within hot and cold cases	Analysis (thermal worst-case modeling) + Test (TVAC testing)	Thermal Requirement
PAYL-012	The payload shall not perform imaging operations during eclipse periods.	Operations Requirements	Analysis (mission timeline) + Test (mode validation)	Operations Constraint
PAYL-013	The payload shall acquire imagery in at least three distinct spectral bands within the visible spectrum (400–900 nm) to enable colour	Allows for full colour image recreation, Functional Requirement	Inspection (instrument design) + Analysis (band selection validation)	Science Objective

	reconstruction and surface feature discrimination.			

3.2 Structural Requirements

The following list of requirements have been derived for the Structural Subsystem:

Table 4: Structural Requirements

Structural Requirements				
Requirements Identifier	**Description**	**Derived From**	**Verification Method**	**Notes**
STRUC-001	The structural subsystem shall maintain payload boresight alignment within tolerances required to achieve a ground sampling distance of ≤ 1 m/pixel.	Mission Objective	Analysis (structural deformation modeling) + Test (alignment verification)	PAYL-001
STRUC-002	The structural subsystem shall limit structural-induced pointing disturbances to ≤ 5 µrad.	Derived from Jitter Calculations	Analysis (vibration modeling) + Test (modal and vibration testing)	PAYL-008
STRUC-003	The structural subsystem shall withstand launch loads and vibration environments with a minimum factor of safety ≥ 1.25.	Operational Requirement. Need to withstand all vibrations to ensure no cracks form during launch and operations	Analysis (FEA) + Test (vibration, acoustic testing)	Launch Requirements
STRUC-004	The structural subsystem shall accommodate both spacecraft within the launch configuration prior to separation.	Mission Requirement. A single rocket is being used for launch and thus the	Analysis (load distribution) + Test (separation testing)	System Requirement

		structure should house both orbiters		
STRUC-005	The structural subsystem shall be compatible with a 4 m-class launch vehicle fairing.	Operational Requirement. Need to ensure that both orbiters fit within the same fairing	Inspection (CAD verification)	Launch Constraint
STRUC-006	The structural subsystem shall provide mechanical interfaces to support integration of payload, ADCS, communications, propulsion, and power subsystems.	The structural subsystem should be able to house all other subsystems and allow for cross-subsystem communication/interaction	Inspection (interface control documents)	System Requirement
STRUC-007	The structural subsystem shall maintain structural integrity under thermal environments experienced during all mission phases.	Functional Requirement. Structural subsystems should maintain integrity during rapid heat fluctuations	Analysis (thermal stress modeling) + Test (thermal cycling)	Thermal Requirement
STRUC-008	The structural subsystem shall support payload mounting and alignment for off-nadir pointing up to 15°.	Mission Requirement. Subsystem should ensure that the payload can point at 15 degrees off nadir while also	Inspection (mechanical layout) + Test (alignment verification)	PAYL-002

		accommodating for gyroscopic movements		
STRUC-009	The structural subsystem shall support both stowed and deployed configurations required for launch and on-orbit operations.	Mission Requirement. A single rocket is being used for launch and thus the structure should house both orbiters. Stowed systems allow for more efficient payload storage.	Analysis (deployment mechanisms) + Test (deployment validation)	System Requirement

3.3 Communications Requirements

Table 5: Communications Requirements

Communications Requirements				
Requirement Identifier	**Description**	**Derived From**	**Verification**	**Method Notes**
COM-001	The communications subsystem shall support high-rate science data downlink and low-rate command and telemetry communication with Earth.	STM Obj 2.3; Mission Ops Concept	Inspection	Review of RF block diagram, diplexer isolation, and frequency allocation.

COM-002	The communications subsystem shall achieve a minimum science data downlink rate of ≥1 Mbps at maximum Earth–Mars distance and ≥6 Mbps at minimum distance.	Link Budget Analysis; Earth-Mars Geometry Model	Analysis	Simulation of link performance using inverse-square scaling and E_b/N_0 threshold with DSN ground stations.
COM-003	The communications subsystem shall support a total daily science data return of ≥6 GB to ≥40 GB per orbiter.	Section 2.7 Plot; Data Volume Analysis	Analysis	Modeling based on 7-hour daily DSN contact per orbiter.
COM-004	The communications subsystem shall maintain pointing accuracy sufficient to ensure reliable link performance during data transmission.	Ka-band Beamwidth Calculation; Pointing Analysis	Test / Analysis	Verification of antenna beamwidth against ACS pointing jitter and gimbal performance.
COM-005	The communications system shall achieve a Bit Error Rate (BER) $\leq 1\times10^{-6}$ using forward error correction (FEC) coding.	CCSDS Standards; STM Obj 2.3	Analysis	Modeling of LDPC/Turbo coding performance under deep-space noise conditions.
COM-006	The communications subsystem shall ensure link closure at maximum Earth–Mars distance with appropriate link margin.	Link Budget Analysis	Test	RF power verification during subsystem integration testing.

COM-007	The subsystem shall provide coherent two-way Doppler and ranging tracking capability for navigation support.	STM Obj 3.1 & 3.2	Analysis	Verification of transponder turnaround ratios and ranging accuracy.
COM-008	The communications subsystem shall operate within a temperature range of −10°C to +40°C during nominal conditions.	Thermal Control Requirements	Test	Thermal vacuum (TVAC) testing of RF components under operational conditions.
COM-009	The subsystem shall be compatible with NASA Deep Space Network (DSN) 34 m and 70 m ground stations.	DSN Interface Specifications	Inspection	Interface compliance verification with NASA telecommunications standards.

3.4 Command and Data Handling Requirements

Table 6: Command and Data Handling Requirements

Command and Data Handling Requirements				
Requirement Identifier	**Description**	**Derived From**	**Verification**	**Method Notes**
CDH-001	The subsystem shall provide sufficient onboard storage capacity to buffer peak science data generation during communication gaps.	Data Volume Analysis; DSN Contact Constraints	Analysis	Storage sizing based on peak data generation (37.8 GB/day) and buffering requirements with margin.

CDH-002	The subsystem shall support storage of at least three days of peak science data generation without data loss.	Mission Operations Concept; Data Volume Analysis	Analysis	Verification using worst-case data accumulation during DSN blackout periods.
CDH-003	The subsystem shall acquire, store, and route science and housekeeping data between payload, onboard storage, and communication subsystems.	System Architecture Definition	Inspection	Review of onboard data handling architecture and interface control documents.
CDH-004	The subsystem shall support data throughput greater than or equal to the peak payload data generation rate.	Payload Data Rate Requirements	Analysis	End-to-end data flow simulation from payload to transmitter.
CDH-005	The subsystem shall minimize data loss and corruption through appropriate error detection and correction mechanisms.	Science Data Integrity Requirements	Test / Analysis	End-to-end validation using error detection and correction schemes.
CDH-006	The subsystem shall support continuous science data acquisition and storage during communication blackout periods.	DSN Contact Constraints	Analysis	Simulation of data accumulation and buffering during no-contact intervals.

3.5 ADCS Requirements

Table 7. ADCS Subsystem Requirements

Attitude Determination and Control Systems (Subsystem-Level) Requirements				
Requirements Identifier	**Description**	**Derived From**	**Verification Method**	**Notes**
ADCS-001	The ADCS shall maintain spacecraft pointing knowledge and control accuracy sufficient to support payload imaging and communication requirements.	SR-001	Analysis (error budget) + Test	Imaging performance driver
ADCS-002	The ADCS shall limit pointing stability (jitter) to ≤ 5 µrad during imaging operations.	SR-004	Analysis (jitter) + Test	Prevents image blur
ADCS-003	The ADCS shall support off-nadir pointing up to ±15° to enable increased surface coverage.	SR-002	Analysis (geometry) + Test	Coverage requirement
ADCS-004	The ADCS shall reorient the spacecraft between imaging targets within a time consistent with mission mapping requirements.	SR-002, SR-004	Analysis + Simulation	Improves mapping efficiency
ADCS-005	The ADCS shall maintain antenna pointing within ±0.1° toward Earth.	SR-003	Analysis (link budget) + Test	Communication reliability

Attitude Determination and Control Systems (Subsystem-Level) Requirements				
ADCS-006	The ADCS shall provide continuous attitude knowledge with accuracy sufficient to support payload geolocation and communication requirements.	SR-005	Inspection + Test	Continuous operation
ADCS-007	The ADCS shall include momentum management capability to prevent actuator saturation.	SR-005	Analysis + Test	Prevents saturation
ADCS-008	The ADCS shall prevent actuator saturation during nominal operations.	SR-005	Analysis + Simulation	Long-term stability
ADCS-009	The ADCS shall include redundancy for critical attitude sensors.	SR-005	Inspection + Test	Reliability
ADCS-010	The ADCS shall support a safe mode that maintains spacecraft attitude stability and ensures survivability under off-nominal conditions.	SR-005	Analysis + Test	Survival Mode
ADCS-011	The ADCS shall maintain operational capability for the full mission duration.	SR-005	Analysis + Inspection	Lifetime Requirement

3.6 Electrical Power Requirements

Table 8. Electrical Power Subsystem Requirements

Electrical Power Subsystem					
Type of Requirement	**Unique ID**	**Description**	**Source**	**WBS Deliverable**	**Verification Strategy**
Performance	EPS-001	The electrical power subsystem shall generate sufficient power at end-of-life to support all spacecraft operations during sunlit conditions.	Appendix 7.2.7 – Solar Array Sizing Calculations	Power Subsystem	Analysis
Performance	EPS-002	The electrical power subsystem shall maintain sufficient battery state-of-charge at the end of eclipse to ensure continued operation and prevent deep discharge.	Appendix 7.2.6 – Battery Sizing Calculations	Power Subsystem	Analysis
Functional	EPS-003	The electrical power subsystem shall support a reduced-power operational mode during eclipse periods, including payload standby operation.	Appendix 7.2.5 – Eclipse and Sunlit Time Calculations	Power Subsystem	Analysis
Functional	EPS-004	The electrical power subsystem shall regulate and distribute electrical power to all spacecraft subsystems throughout all mission phases.	Appendix 7.2.6 – Battery Sizing Calculations	Power Subsystem	Analysis

3.7 Thermal Requirements

Table 9. Thermal Subsystem Requirements

Thermal Subsystem					
Type of Requirement	**Unique ID**	**Description**	**Source**	**WBS Deliverable**	**Verification Strategy**
Performance	THERM-001	The thermal subsystem shall maintain spacecraft bus and avionics temperatures between 0°C and +40°C during nominal operations	NASA Spacecraft Thermal Design Standard Practice (GSFC-STD-1000)	Thermal Control	Thermal Analysis + TVAC Testing
Performance	THERM-002	The spacecraft shall survive temperatures between −20°C and +50°C	NASA Spacecraft Thermal Design Standard Practice (GSFC-STD-1000)	Thermal Control	TVAC Test
Performance	THERM-003	The thermal subsystem shall maintain payload temperatures between −10°C and +40°C during sunlit operations	NASA Spacecraft Thermal Design Standard Practice (GSFC-STD-1000)	Thermal Control	Thermal Analysis + TVAC Testing
Performance	THERM-004	The payload shall survive temperatures between −20°C and +50°C during eclipse	NASA Spacecraft Thermal Design Standard Practice (GSFC-STD-1000)	Thermal Control	Thermal Analysis + TVAC Testing
Functional	THERM-005	The thermal subsystem shall provide sufficient active heating during eclipse conditions to maintain spacecraft components within allowable temperature limits.	Appendix 7.2.8 – Heater Sizing Calculations	Thermal Control	Thermal Model/Simulation (NX)

Performance	THERM-006	The thermal subsystem shall maintain survival temperatures during eclipse using active heating with a sufficient margin	Appendix 7.2.8 – Heater Sizing Calculations	Thermal Control	Thermal Analysis
Performance	THERM-007	The thermal subsystem shall reject internal heat dissipation generated during peak sunlit operations while maintaining all components within allowable thermal limits.	Appendix 7.2.9 – Radiator Sizing Calculations	Thermal Control	Thermal Model/Simulation (NX)
Operational	THERM-08	Payload imaging operations shall occur only during sunlit orbital segments with sufficient solar illumination to satisfy image quality requirements.	Appendix 7.2.5 – Eclipse and Sunlit Time Calculations	Thermal Control	Analysis

3.8 Propulsion Subsystem Requirements (SRD)

Table 10: Propulsion Subsystem Requirements

Propulsion Subsystem Requirements			
Req. ID	**Subsystem Requirement Description**	**Traceability (Parent)**	**Verification Method**

PROP-001	The propulsion subsystem shall provide total ΔV sufficient to support trajectory correction, Mars Orbit Insertion, and orbit maintenance throughout the mission.	CON-001, PERF-003, PERF-005	Analysis
PROP-005	The propulsion subsystem shall support trajectory correction maneuvers during interplanetary transfer.	MRD 4.3 (Risk)	Inspection
PROP-006	The propulsion subsystem shall support orbit maintenance maneuvers to maintain the required orbital altitude and configuration.	MRD 3.3	Analysis / Simulation
PROP-007	The subsystem shall maintain propellant tank temperatures between 10◦C and 35◦C during all mission phases to prevent freezing or over-pressurization	CON-010, MRD 5.0	Analysis / TVAC Test
PROP-008	The propulsion subsystem shall support multiple engine restarts to enable mission operations across all phases.	MRD 3.3, 3.5	Functional Test
PROP-009	The propulsion subsystem shall include redundancy for critical components to ensure mission success in the event of partial system failure.	MRD 1.3.1 (4)	Documentation Review

4.0 Subsystem Design and Justification

This section presents the selected subsystem designs for the ROUTE-M mission and demonstrates how each design satisfies the corresponding subsystem requirements. For each subsystem, the selected architecture is described and justified using mission constraints, performance requirements, and supporting analyses. Detailed calculations are provided in the appendix where applicable.

Overall System Concept: Spacecraft Bus Architecture

To satisfy the stringent scientific criteria for the ROUTE-M mission without exceeding the Discovery Program budgetary restriction of $750 million USD, the spacecraft design employs a heritage bus that is heavily based on the Mars Reconnaissance Orbiter (MRO). A brand-new deep-space design entails excessive research and development (R&D) costs and schedule risks. The use of a proven technology at Technology Readiness Level (TRL) 9 enables economies of scale in the manufacturing process to construct two identical orbiters under the budgetary constraints of the concept. The Mars Reconnaissance Orbiter (MRO) heritage design was chosen in particular since it is physically and sub-systematically consistent with the needs of ROUTE-M. Firstly, ROUTE-M will need to transmit data at high bandwidths to receive high-resolution compressed images.

The heritage bus offers the appropriate infrastructure that is able to accommodate extensive, efficient solar panel arrays and a large High Gain Antenna (HGA) to enable constant communications with Earth. Secondly, the requirements of the propulsion system for the spacecraft call for a propulsion system strong enough to withstand the MOI process and provide momentum desaturation for 7 years. Finally, there is an oversize tankage volume on the heritage platform to accommodate the bipropellant requirement of 669 kg of each orbiter (plus a 25% contingency for the MOI).

4.1 Payload Design

To satisfy the requirement for high-resolution global mapping of the Martian surface, the ROUTE-M mission employs a pushbroom optical imaging system derived from HiRISE heritage. The payload is designed to achieve a ground sampling distance (GSD) of approximately 1 m/pixel from a nominal orbital altitude of 300 km.

The instrument utilizes a pushbroom imaging architecture in which a linear detector array continuously captures image lines as the spacecraft moves along its ground track. This approach enables extremely high-resolution imaging without requiring large two-dimensional focal plane arrays while also improving signal-to-noise ratio through longer integration times per pixel. The detector is designed to achieve an effective imaging capability equivalent to approximately 40,000 pixels in the along-track direction and 20,000 pixels across-track through spacecraft motion and time-delay integration. Note that each pixel represents 3 different channels (Sensors). This configuration enables wide-swath coverage while maintaining sub-meter spatial resolution.

The optical system operates within the visible spectral range (approximately 400–900 nm), supporting multi-band imaging for terrain classification, surface composition analysis, and colour reconstruction. The mapping performance is governed by the relationship between orbital altitude, focal length, and detector pixel size, where the ground sampling distance is given by $GSD = (H \times p) / f$. For a 300 km orbit, achieving a 1 m/pixel resolution requires a focal length on the order of several meters, which is consistent with scaled HiRISE-derived optics. Detailed derivations and parameter selections are provided in Appendix 7.2.

The pushbroom imaging geometry is strongly dependent on spacecraft motion and pointing stability. As the spacecraft traverses its orbit, each detector element traces a continuous ground line forming a two-dimensional image over time. This requires precise synchronization between spacecraft velocity and detector readout to avoid geometric distortion. Additionally, the payload supports off-nadir pointing up to ±15°, which increases surface coverage and enables more efficient global mapping. The geometric implications of off-nadir pointing are illustrated in Figure 1, where the shift in observed ground location due to viewing angle is shown.

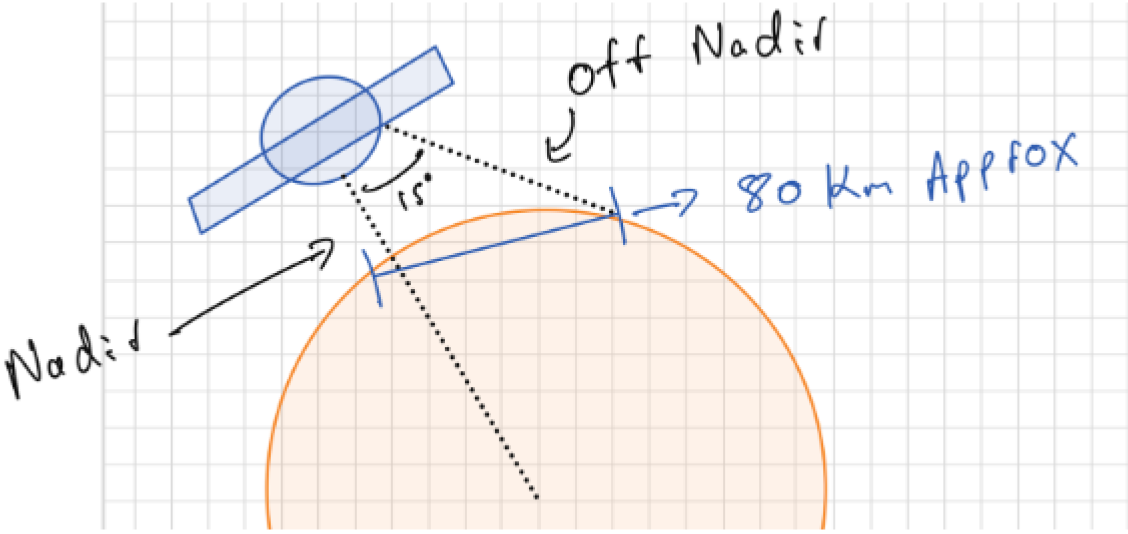


Figure 2: Off Nadir Pointing Ground Offset visual

The allowable pointing jitter is derived from image smear constraints using small-angle approximations. The ground displacement due to pointing jitter can be approximated as $\Delta x = R\theta$, where R is the slant range, and θ is the pointing jitter. For an orbital altitude of 300 km and an off-nadir angle of 15°, the slant range is approximately 310 km. Assuming a pointing jitter of 5 μrad, the resulting ground displacement is approximately 1.55 m. This value is on the order of the target GSD, indicating that a jitter requirement of ≤ 5 μrad maintains acceptable imaging performance while remaining consistent with achievable ADCS capability. This analysis, along with supporting calculations, is provided in Appendix 7.2.11.

As shown in Figure 1, off-nadir imaging introduces a horizontal ground offset and perspective distortion, which increases the effective swath width and improves coverage efficiency. However, this also results in non-uniform ground projection and potential geometric distortion, which must be corrected during image processing. Despite these effects, off-nadir imaging significantly enhances mapping efficiency and enables multi-angle observations for terrain characterization.

Due to the large data volumes generated by a high-resolution pushbroom sensor, onboard data reduction is required. Raw imagery is initially captured at high bit depth (14 bits per pixel) to preserve radiometric fidelity and is subsequently compressed to lower bit depth (8 bits per pixel). The image is then further compressed using lossy compression techniques such as JPEG2000. This reduces data volume by approximately an order of magnitude while maintaining sufficient quality for mapping applications. The data rate and compression strategy are further detailed in Appendix 7.2.2 and 7.2.3.

Overall, the payload design integrates high-resolution optics, pushbroom imaging geometry, and efficient onboard data handling to achieve global sub-meter mapping of Mars. The design is tightly coupled with ADCS, communication, and data handling subsystems, ensuring that all performance requirements are met while maintaining feasibility within mission constraints.

4.2 Structural Subsystem

The structural subsystem is designed to provide a mechanically stable and thermally robust platform that supports all spacecraft subsystems while maintaining precise alignment of the high-resolution imaging payload throughout all mission phases. Structural performance is a primary driver for mission success since even small mechanical disturbances or misalignments directly degrade image quality and reduce geolocation accuracy. As a result, the structural design is tightly coupled with both payload and ADCS performance requirements.

A key driver of the structural configuration is the launch vehicle constraint imposed by the Atlas V 401. As shown in Figure 2, the spacecraft must be packaged within a 4-meter class fairing, which defines a maximum usable diameter of approximately 4.2 meters and a maximum height of approximately 12.2 meters. These volumetric constraints directly influence the spacecraft layout, requiring a compact and efficient structural configuration that can accommodate all subsystems within the available volume. In addition, the total allowable launch mass is constrained by Atlas V 401 performance for the required Mars transfer energy rather than by its low Earth orbit payload capacity. Although the spacecraft are initially placed into an Earth parking orbit prior to trans Mars injection, the governing constraint is the mass that can be delivered to the required interplanetary trajectory.

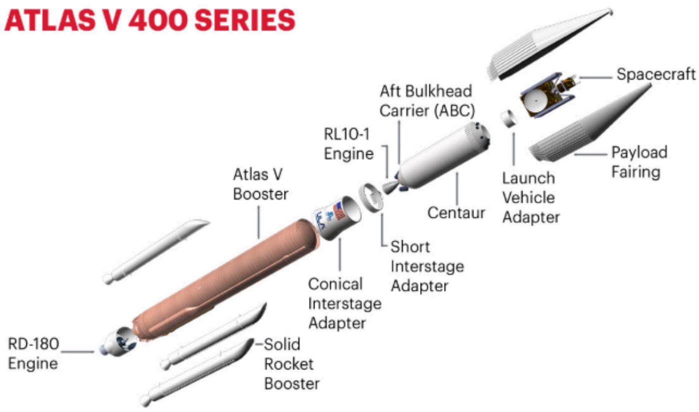


Figure 3: Atlas V 401 Rocket Configuration [2]

The dual spacecraft architecture further increases structural complexity since both spacecraft must be integrated within the fairing in a stacked or side-by-side configuration while maintaining structural integrity during launch. Load paths should be carefully designed to distribute forces through the primary structure and avoid localized stress concentrations. The structural configuration must also support reliable separation mechanisms to ensure safe deployment of both spacecraft once in orbit.

On orbit structural performance is primarily driven by the need to maintain pointing stability during imaging operations. Structural vibrations and flexibility contribute directly to pointing jitter, which must be limited to meet the imaging requirement of approximately 1 meter per pixel resolution. The allowable structural contribution to jitter is constrained within the overall ADCS error budget ensuring that total system jitter remains below 5 micro-radians. This requirement drives the selection of structural materials and geometry to maximize stiffness while minimizing mass.

Thermal effects also play a significant role in structural design. During orbit the spacecraft experiences repeated transitions between sunlit and eclipse conditions which induce thermal cycling. Differential thermal expansion between structural components and payload elements can lead to misalignment of the optical system. To mitigate this the structure needs to be designed using materials with low coefficients of thermal expansion and symmetric layouts that minimize differential deformation. Thermal isolation between subsystems should also be considered to reduce localized temperature gradients and maintain overall structural stability.

The structural subsystem should support both stowed and operational configurations. During launch all components should remain securely constrained within the fairing envelope and withstand high load conditions. Once in orbit the structure should support deployment of components such as solar arrays and communication antennas without inducing disturbances that could affect spacecraft stability or payload alignment.

Interfaces between subsystems should be designed to ensure compatibility and minimize the transmission of disturbances. Structural mounting points should be reinforced to handle both static and dynamic loads while maintaining alignment requirements. The structure should also accommodate harness routing thermal control elements and shielding without degrading mechanical performance.

The design should be constrained by the baseline Atlas V 401 fairing to ensure efficient packaging and subsystem integration. These launch vehicle constraints should be treated as primary design drivers while still allowing flexibility for future mission configurations.

Overall the structural design should provide a high stiffness thermally stable and efficiently packaged platform that supports precision imaging operations and ensures survivability throughout the mission lifetime.

4.3 Communications Subsystem

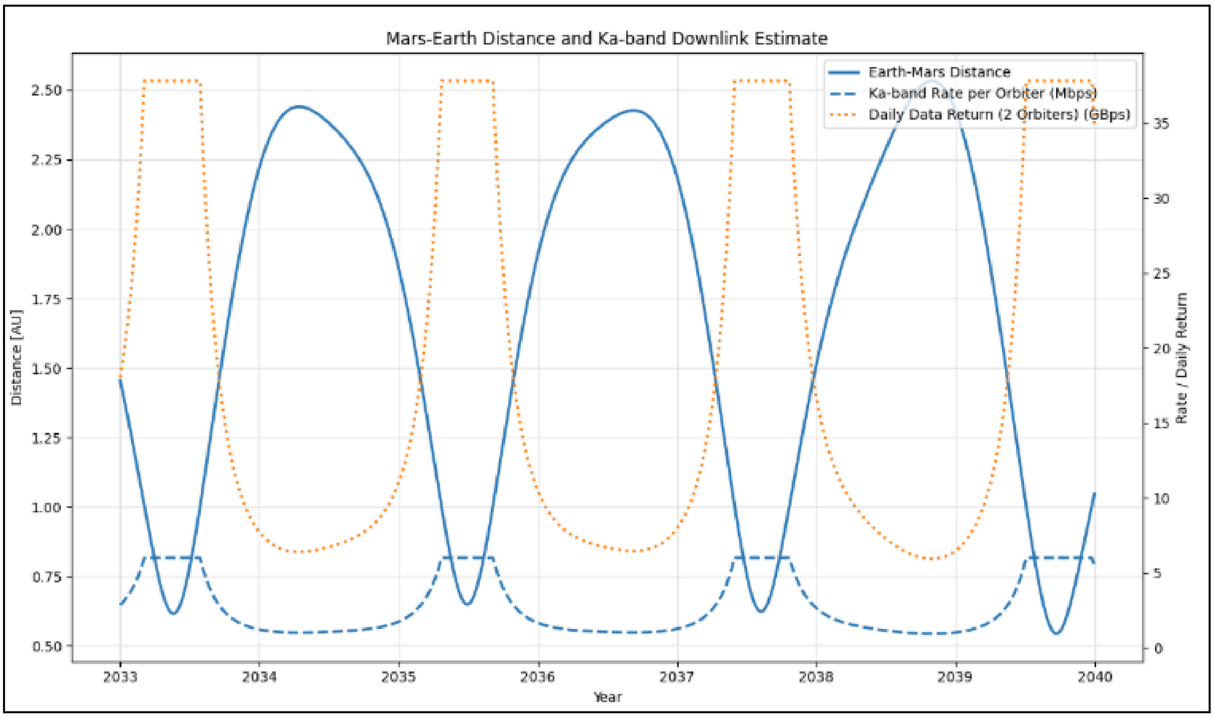


Figure 4: Downlink Rates / Distance to Mars

The communications subsystem design is driven by the need to support reliable science data return over the full range of Earth-Mars distances. Link performance was estimated using an inverse-square relationship, resulting in an expected Ka-band downlink data rate of approximately 1-6 Mbps across the mission lifetime. These values were rounded from simulation results to avoid over-specification while maintaining consistency with system-level performance.

Daily data return capability was determined based on achievable DSN contact time, resulting in an estimated data return range of 6-35 GB/day (corresponding to 0.6-3.2 Mbps average data rate). These values inform onboard storage sizing and operational planning during communication gaps.

To meet these performance requirements, the subsystem utilizes a Ka-band (32 GHz) high-rate downlink for science data transmission and an X-band (8.4 GHz) link for uplink and low-rate telemetry/command. This separation ensures reliable command capability while enabling high-throughput science data return.

A high-gain antenna (HGA) is incorporated to close the deep-space link budget. The selected design includes an antenna diameter of approximately 3.0 m, providing sufficient gain at Ka-band frequencies to achieve the required data rates at maximum Earth-Mars distance. The antenna is mounted on a two-axis gimbal with a pointing accuracy of approximately ±0.1°, enabling precise Earth-pointing necessary for narrow Ka-band beamwidth operation.

Additional design considerations include maintaining acceptable bit error rates (BER) under deep-space channel conditions and ensuring compatibility with established Ka-band communication standards and operational constraints.

4.4 Attitude Determination and Control Subsystem

The Attitude Determination and Control System (ADCS) should provide precise spacecraft orientation control to support high-resolution imaging and reliable communication throughout all mission phases. The performance of the ADCS is critical since pointing accuracy and stability directly affect image quality and geolocation performance. In addition, the ADCS should ensure accurate antenna alignment during communication windows to enable efficient transmission of scientific data to Earth.

To satisfy the imaging and coverage requirements, the spacecraft adopts a three-axis stabilized architecture, which enables continuous and precise attitude control. This configuration allows both nadir pointing and off-nadir pointing up to ±15° which is required to increase surface coverage and improve mapping efficiency. The system should also support rapid reorientation between imaging targets, which can be approximated using rotational kinematics where the required angular acceleration is given by $T = I*a$. This relationship drives actuator sizing and ensures that the spacecraft can achieve the required slewing performance within operational constraints.

Attitude determination should be achieved using a combination of onboard sensors that provide both absolute and rate-based measurements. Star trackers should serve as the primary attitude determination sensors providing high accuracy orientation knowledge during imaging operations typically on the order of arcseconds. Inertial measurement units should provide angular rate data for short term attitude propagation while sun sensors should provide coarse attitude information for safe mode operations. This combination ensures continuous and reliable attitude knowledge under all mission conditions.

Attitude control should be implemented using reaction wheels and thrusters. Reaction wheels should provide fine pointing control and maintain stability during imaging operations while thrusters should be used for large angle maneuvers and momentum dumping. The torque generated by reaction wheels follows T = H where H is angular momentum, allowing continuous control of spacecraft attitude. Momentum buildup due to external disturbances such as solar radiation pressure must be periodically removed using thrusters to prevent saturation.

The ADCS design should meet strict pointing and stability requirements driven by the payload. Image smear due to pointing jitter can be approximated using the relation delta x = R*theta where R is the slant range and theta is the pointing error. For an orbital altitude of approximately 300 km and an off nadir angle of 15° the slant range is approximately 310 km. With a jitter requirement of 5 µrad the resulting ground displacement is approximately 310,000 * 5E-6 = ~ 1.55m. This value is on the order of the 1 m/pixel ground resolution indicating that the selected jitter requirement maintains acceptable imaging performance while remaining achievable.

Communication requirements also impose constraints on attitude control. High-frequency Ka-band communication requires accurate antenna pointing due to narrow beamwidth. The pointing error must remain within approximately ±0.1° to maintain reliable link performance which further constrains ADCS accuracy and stability.

External disturbances continuously affect spacecraft orientation and must be actively compensated. These include solar radiation pressure gravity gradient effects and accumulated momentum within the control system. Disturbance torques can be approximated and must be counteracted by control actuators to maintain stable pointing over time.

Reliability is also a key requirement due to the long duration of the mission. The ADCS should include redundancy for critical components and support safe mode operation to maintain spacecraft stability under off nominal conditions. This ensures that the spacecraft can recover from faults and continue mission operations.

Overall the ADCS design should provide high accuracy attitude knowledge, stable pointing performance and reliable long term operation while supporting both imaging and communication requirements.

4.5 Command and Data Handling (C&DH)

The Command and Data Handling (C&DH) subsystem should manage the acquisition, storage processing and transmission of all spacecraft data including both science and housekeeping data. The subsystem is designed to handle the large data volumes generated by the high resolution pushbroom imaging payload while ensuring reliable data integrity and availability during communication windows.

The payload generates high resolution imagery using a pushbroom architecture with an effective resolution on the order of 40,000 pixels across track and continuous acquisition along track. The raw data volume can be approximated as Data = $N \times b$ where N represents the number of pixels and b represents the bit depth. For raw imagery captured at 14 bits per pixel this results in extremely large data volumes that exceed onboard storage and communication capabilities if left unprocessed.

To address this the subsystem should implement a multi stage data reduction pipeline. Raw imagery should first undergo bit depth reduction from 14 bits per pixel to 8 bits per pixel which reduces data volume while maintaining sufficient radiometric fidelity for mapping applications. This is followed by lossy compression using JPEG2000 with an approximate compression ratio of 10 to 1. The combined effect of bit depth reduction and compression reduces total data volume by more than an order of magnitude enabling efficient storage and transmission.

The subsystem should include solid state storage sized to support multiple days of imaging operations without downlink. Storage requirements are driven by the daily data generation rate and limited communication windows with Earth. This buffering capability ensures that science data is preserved during periods without contact.

Data flow should be managed through a centralized onboard computer which performs data formatting compression and routing between subsystems. Error detection and correction mechanisms should be implemented to ensure data integrity during both storage and transmission.

The data handling requirements are driven by the need to store and manage large volumes of science data under limited communication availability. Based on peak data generation rates of approximately 35 GB per day, the subsystem must support multi day buffering to account for communication gaps, DSN scheduling constraints and potential off nominal operations.

To meet this requirement the onboard storage capacity should be increased to approximately 1 TB. This provides the ability to store over 25 days of peak science data, significantly exceeding the minimum buffering requirement and introducing margin for data retransmission, redundancy and operational flexibility. This margin is critical in deep space missions where communication opportunities are limited and data loss cannot be tolerated.

The storage system should support continuous read and write operations at rates compatible with payload data generation while ensuring data integrity through error detection and correction mechanisms. Overall the increased storage capacity ensures that all generated data can be reliably processed, stored and transmitted without loss even during extended communication blackout periods.

Overall the C&DH subsystem should provide high capacity reliable data management that enables continuous high resolution imaging while remaining compatible with communication and storage constraints.

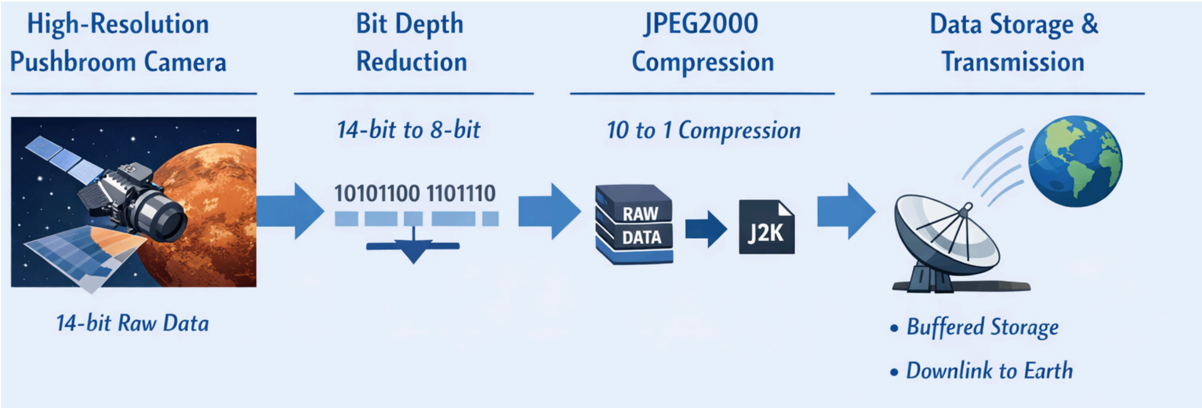


Figure 5: C&DH Pipeline

4.6 Propulsion Subsystem

The propulsion subsystem should provide all required velocity changes for Mars orbit insertion trajectory correction maneuvers and long term orbit maintenance while ensuring reliable operation throughout the mission lifetime. Following separation from the launch vehicle the Atlas V Centaur upper stage performs the Trans Mars Injection maneuver therefore the onboard propulsion system is primarily responsible for Mars Orbit Insertion (MOI) and subsequent in orbit operations.

The total mission velocity requirement can be divided into two components. The launch vehicle provides approximately $\Delta v\_TMI \approx 3.58$ km/s while the spacecraft propulsion system must deliver an additional $\Delta v\_SC \approx 1.449$ km/s to support Mars Orbit Insertion trajectory correction maneuvers and long term momentum management. This results in a combined mission requirement of approximately $\Delta v\_total \approx 5.671$ km/s which defines the overall propulsion performance requirement

The propulsion system should use a bipropellant architecture to meet these high energy requirements due to its higher specific impulse and flight heritage in planetary missions. Assuming a representative specific impulse Isp ≈ 320 s and a spacecraft dry mass of approximately 912 kg the required propellant mass can be estimated using the rocket equation. A baseline propellant mass of approximately 535 kg is required for nominal operations. However given the critical nature of Mars Orbit Insertion and the risk associated with incomplete capture an additional contingency margin of 25% should be applied resulting in a total propellant allocation of approximately 669 kg per spacecraft.

The propulsion hardware including tanks, valves feed lines and thrusters should have an estimated dry mass of approximately 85 kg. The system should include redundant valves and fault tolerant control to mitigate the risk of propulsion failure during critical maneuvers particularly during MOI which represents the highest risk phase of the mission. Autonomous stabilization capability should also be implemented to ensure spacecraft attitude control during and after major burns.

Following successful orbit insertion the propulsion subsystem should support periodic maneuvers to maintain the spacecraft orbit at approximately 300 km altitude (±20 km) over the 7 year operational lifetime. These maneuvers primarily compensate for atmospheric drag and support momentum desaturation in coordination with the ADCS. Since the two spacecraft operate in different orbital planes constellation phasing is not required, simplifying long term propulsion operations.

Thermal considerations play a critical role in propulsion system design. Hydrazine based propellants have freezing points near 1 to 2 degrees Celsius requiring active thermal control during eclipse periods. The propulsion subsystem should therefore be integrated with the spacecraft thermal control system with dedicated heater power allocated from the overall survival heating budget to maintain propellant temperatures within operational limits. This creates additional load on the electrical power subsystem during eclipse conditions and must be accounted for in power sizing.

The propulsion system is further constrained by overall spacecraft mass limits and launch vehicle capabilities. The total wet mass per spacecraft is approximately 1581 kg including propellant which ensures compatibility with the Atlas V 401 payload envelope and performance constraints. Mass growth margins of approximately 20% should be included in hardware estimates to ensure that the system remains within allowable limits as the design matures.

To reduce development risk and maintain schedule the propulsion subsystem should rely on heritage components with demonstrated flight performance such as systems derived from the Mars Reconnaissance Orbiter. All propulsion components should achieve a minimum Technology Readiness Level of 6 prior to Preliminary Design Review ensuring that the system is sufficiently mature for integration and testing.

Overall the propulsion subsystem should provide reliable high performance maneuver capability while ensuring robustness during critical mission phases particularly Mars Orbit Insertion and maintaining long term orbital stability throughout the mission lifetime.

4.7 Electrical Power and Thermal Subsystem

The Electrical Power Subsystem (EPS) should generate, store, regulate and distribute electrical power to all spacecraft subsystems throughout all mission phases while maintaining stable thermal operating conditions. The subsystem must support both high power operational modes during imaging and communication as well as reduced power conditions during eclipse. Power system performance is critical to mission success since insufficient or unstable power can directly impact payload operation thermal control and communication capability.

Electrical power generation should be achieved using deployable solar arrays which convert incident solar radiation into electrical energy. The power generated can be approximated using $P = \eta \times A \times S$ where η represents solar cell efficiency A represents array area and S represents solar irradiance at Mars orbit. Due to the increased distance from the Sun the available solar flux is reduced compared to Earth which requires larger array areas to meet mission power demands. The system should be sized to meet power requirements under end of life conditions accounting for degradation of solar cells due to radiation exposure, thermal cycling and contamination effects.

During sunlit operations the solar arrays should supply power to all active subsystems including the payload ADCS communication system and onboard computer while simultaneously charging the onboard batteries. Excess electrical energy should also be used to support thermal control systems such as heaters where required to maintain subsystem temperatures within allowable limits. The power distribution system should regulate voltage levels and ensure stable delivery of power across all subsystems to prevent performance degradation or system faults.

During eclipse periods the spacecraft should rely entirely on stored energy within rechargeable batteries. The required battery capacity is determined by the total power demand and eclipse duration which can be approximated using $E = P \times t$ where E is required energy P is power consumption and t is eclipse time. In addition to supporting electrical loads the battery must also supply power to thermal control systems including survival heaters which prevent critical components such as batteries propellant lines and payload instruments from dropping below minimum allowable temperatures. This creates a worst case power scenario during eclipse where both electrical and thermal loads must be sustained simultaneously.

A reduced power operational mode should be implemented during eclipse conditions where non critical subsystems such as the imaging payload are placed into standby. This allows available power to be prioritized toward essential systems including ADCS stabilization thermal control and command and data handling ensuring spacecraft survival during power limited conditions. Thermal management during eclipse is particularly critical as the absence of solar input leads to rapid heat loss requiring active heating to maintain thermal balance.

The EPS should include a power control and distribution unit responsible for managing power flow between generation storage and loads. This unit should provide voltage regulation switching protection and fault isolation to ensure reliable operation. It should also support controlled power allocation to thermal subsystems including heater circuits which are cycled based on temperature feedback to maintain stable operating conditions.

Thermal effects significantly influence EPS performance particularly for solar arrays and batteries. Solar array efficiency decreases with increasing temperature while battery capacity and lifetime are strongly dependent on operating temperature. The EPS should therefore be closely integrated with the thermal subsystem to ensure that components operate within optimal temperature ranges. This includes the use of thermal insulation radiators and heaters to control temperature fluctuations caused by orbital day night cycling.

Overall the EPS design should provide reliable power generation, efficient energy storage and stable power distribution across all mission phases while simultaneously supporting thermal control requirements. The subsystem should be capable of maintaining both electrical and thermal stability under varying environmental conditions ensuring long term mission performance and survivability.

5.0 Risk Analysis

Table 11. Risk Analysis

Risk Analysis					
ID	**Risk Description**	**Category**	**Likelihood**	**Impact**	**Mitigation Strategy**
R1	Launch vehicle failure	Launch	Low	Very High	Selection of a high-reliability launch

					provider and use of flight-proven systems
R2	Spacecraft separation or deployment anomaly	Integration / Launch	Low-Medium	Very High	Use of heritage separation mechanisms, extensive ground testing, and deployment verification sensors
R3	Failure to achieve Mars orbit due to MOI or navigation error	Navigation / Propulsion	Low	Very High	Inclusion of ΔV margins and trajectory correction maneuvers
R4	Propulsion system failure or performance degradation	Propulsion	Low-Medium	Very High	Redundant components, conservative design margins, and subsystem-level testing
R5	ADCS performance degradation affecting pointing accuracy and stability	ADCS	Medium	High	High-precision sensors, momentum management strategies, and redundancy
R6	Payload performance degradation (optical system or calibration)	Payload	Medium	High	Thermal control and periodic calibration procedures

R7	Communication link degradation or loss	Communications	Medium	High	High-gain antenna design, redundancy, and sufficient link margin
R8	Power system degradation (solar arrays or batteries)	Power	Medium	High	Oversizing of power system and degradation margin inclusion
R9	Onboard data handling or software failure	OBDH	Medium	High	Fault detection, correction mechanisms, and system redundancy
R10	Environmental effects (radiation, thermal cycling, contamination)	Environment	Medium	High	Radiation shielding, thermal control design, and protective measures
R11	Integration and testing deficiencies leading to in-flight anomalies	Systems Engineering	Medium	High	Early subsystem integration and comprehensive verification and validation.

The mission risks were identified and evaluated across all phases of the mission lifecycle, including launch, deployment, cruise, Mars orbit insertion, and long-duration orbital operations. To ensure clarity and avoid redundancy, risks were consolidated into system-level categories while preserving the critical failure modes associated with each subsystem.

Risks classified with very high impact correspond to single-point failures that would result in complete mission loss. These include launch vehicle failure, spacecraft separation anomalies, and failure to achieve Mars orbit. Although these risks are assigned low to medium likelihood due to the use of heritage systems and established mission design practices, their consequences remain critical and therefore dominate the overall risk posture of the mission.

Subsystem-level risks, such as propulsion performance degradation, ADCS instability, communication loss, and power system degradation, are assigned medium likelihood and high impact. These risks reflect the challenges associated with long-duration operations in Mars orbit, where environmental disturbances and component degradation can affect system performance over time. However, their impact can be mitigated through redundancy, conservative design margins, and control strategies.

Environmental and onboard system risks are grouped to reflect their continuous influence on spacecraft performance, particularly in the context of radiation exposure, thermal cycling, and potential contamination. These effects contribute to gradual degradation rather than immediate failure, which justifies their classification as medium likelihood and high impact.

Finally, programmatic risks are included to account for uncertainties in cost and schedule that are inherent to early-stage mission design. These risks are mitigated through the inclusion of contingency reserves and schedule buffers, consistent with standard aerospace project management practices.

6.0 Budget Breakdowns

6.1 Cost Breakdown

Table 12. Cost Breakdown Structure

Subsystem / Required By	Cost (USD$M)	Additional Notes
Available Budget	-750	Mission constraint
Project Management and Systems Engineering	55	~7-10% of total mission cost based on typical NASA mission allocation
Spacecraft Bus - Orbiter 1	115	Analogy-based estimate using SMEX-class spacecraft scaling (mass-based CER)
Spacecraft Bus - Orbiter 2	100	~13% reduction using multi-unit production discount
Propulsion Subsystem x2	60	Based on electric propulsion + MOI + cruise corrections; scaled from typical subsystem % allocation
Communications (X/Ka + Antennas) ×2	45	Based on ~4-15% subsystem allocation range
Payload: HiRISE-derived Imager ×2	120	$40M (2006) → $63.7M (2026) using inflation factor 1.593; reduced via heritage + duplication savings
Onboard Data Handling + Compression ×2	25	Based on C&DH typical allocation (~4-10%)
Integration & Test (I&T) + QA	60	Includes system-level verification; consistent with integration-heavy mission phases
Ground Segment + Data Processing Pipeline	20	Reduced scope assumption with automated processing
Mission Operations Segment (cruise + 7-year mapping)	30	Lifecycle cost distributed over mission duration (automated ops assumption)

Contingency / Margin Reserve (~20%)	120	System-level margin per standard practice
Remaining Budget	0	

The mission cost estimate was developed by considering all major phases of the mission, including spacecraft development, launch, operations, and data handling. To keep the estimate clear and manageable, costs were grouped into high-level categories such as payload, spacecraft bus, propulsion, launch services, ground segment, and program management.

The largest portion of the budget is typically driven by spacecraft development and payload design, as these involve specialized components, testing, and integration. While these costs are relatively well understood based on similar heritage missions, they still include some uncertainty due to design complexity and performance requirements. Conservative estimates and margins are therefore included to account for potential overruns.

Launch costs are treated as a fixed, high-impact component of the budget. Although launch services are generally well-characterized, price variations can occur depending on provider, vehicle selection, and scheduling. As a result, a margin is included to account for possible cost fluctuations.

Operational costs, including mission control, data processing, and communication infrastructure, are distributed over the mission lifetime. These are assigned moderate uncertainty, as they depend on mission duration, system performance, and potential anomalies that may require additional resources.

To account for broader economic factors, an overall contingency and inflation margin is applied to the total mission cost. This ensures that the budget remains realistic over time and can accommodate unexpected increases in cost due to market conditions or delays.

6.2 Power Breakdown

Table 13: Power Breakdown

Hot Operational Power Budget (Sunlit)		
Subsystem / Required By	**Cost (W)**	**Additional Notes**
Payload (Stereo Imager + LiDAR)	600	Payload operates only during the sunlit portion of the orbit
Communications (TX + modem + data handling overhead)	250	Downlink and communications are active during operations
Avionics	250	Core spacecraft electronics
ADCS (sensors, controls)	200	N/A
Power subsystems (PDU, converters, losses)	100	Power conditioning and distribution losses

Margin	100	N/A
Total Available Power Budget	**1500**	Constraint (set by hot operational internal dissipation used for radiator sizing)
Cold Survival Power Budget (Eclipse)		
Subsystem / Required By	**Cost (W)**	**Additional Notes**
Cold Survival Internal Power (P_int)	300	First-order conservative assumption (payload OFF), with internal power being consumed by only essential subsystems
Survival Heaters (P_Heater)	200	Installed heater wattage capacity, and is assumed ON through the entire duration of the eclipse (conservative)
Total Eclipse Power Draw	**500**	Used to determine battery sizing and battery energy required per eclipse

6.3 Communications Breakdown

Table 14: Communications Breakdown

Subsystem / Required By	Cost (KB per sol)	Additional Notes
Total Available	3500000	
Payload (High-res imaging)	-3125000	Calculated as 40 images (40000 x 20000 pixels) at 8-bit depth with 10x compression.
Spacecraft Health (Telemetry)	-51200	Housekeeping data (battery, thermal, status) sampled at 1 Hz (50MB total per day)
ADCS (Navigation/Tracking)	-25600	High precision orbital data needed for 50-150m geolocation accuracy requirement
Command Overhead (Error Correction)	-156250	Space for Turbo Coding and packet headers to ensure >99% successful file delivery
Total Remaining	141950	3.36GB - Under the 3.5GB limit with 4% margin remaining.

6.4 Mass Breakdown

Table 15: Mass Breakdown

Subsystem / Required By	Cost (Kg)	Additional Notes
Structures and Mechanism	180	

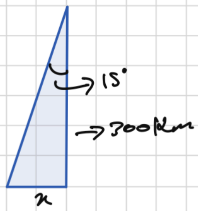
Thermal Control	45	Radiators, Heaters, MLI Blanket
Power	160	
ADCS	70	
Avionics	55	
Communications	60	
Propulsion Hardware	85	
Harness and Cabling	35	
Payload: HiRISE -derived imager	70	
Dry Mass (CBE)	760	
Dry Mass with Growth (20%)	152	
Dry Mass with Growth	912	
Propellant (MOI + TCM + Phasing +Margins)	520	
Propellant Margin (10%)	52	
Wet Mass (per spacecraft)	1,484	
Total Mass	**2968**	Accounts for both Spacecrafts

7.0 Appendix

7.2 Hand Calculations

7.2.1 Off Nadir Angle Calculations

Off-Nadir Angle Calculations



$x = (\tan(15))(300000)$

$x = 80384$ metres → ground offset

$$GSD_{15°} = \frac{GSD_{Nadir}}{\cos(15°)} \Rightarrow \frac{1}{\cos(15°)}$$

$\Rightarrow$ 1.04 m/pixel

within ± 25cm

→ Allows for ground terrain mapping
→ good balance between off-Nadir offset and Resolution

7.2.2 Camera Resolution Calculations

Camera Resolution Calculations

Assuming

40 pics/day

land area = 145 million km^2

Area per day Required

$\hookrightarrow \frac{145\,E6\ km^2}{7 \times 365} \Rightarrow 0.057E6\ km^2 \Rightarrow 57E3\ km^2$

40 image/day

Per image coverage : $\frac{57000}{40} \Rightarrow 1425\ km^2$

∵ We have 2 orbiters

$= 712.5\ km^2/\text{image}$

∵ 3 GB/day downlink

+ 40 images

1 image = $3/40$ = 75 MB/image

Assuming Compression = 10x

file pre compression = 750 MB

∵ 8 bit → 1 byte/pixel

1 byte = 8 bits

Resolution $\longrightarrow [750\,E6]^{1/2} \longrightarrow 27386$ px

Roughly 28K x 28K Resolution
at 14 bit downsized to 8bit
and compressed 10 times

7 years 3GB/day

7.2.3 Image Downlink Compression Calculations

Total time

7 x 365 ⇒ 2555 days

∵ 3GB/day ↪ 7665 GB

⇒ 7.665 TB /day /orbiter

∵ 2 x orbiter → 15.33 TB over 7 years

Compression Rate = Attained / possible

⇒ 145/15.33

⇒ 9.46 X Compression

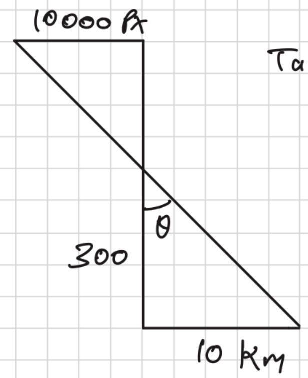
∵ Onboard ≥ 9.46x Compression to JPEG 2000 Lossy

7.2.4 Half Angle Estimations

half Angle estimations

Assuming

20 000 px
x
40 000 px



$$\tan\theta = \frac{10}{300}$$

$$\tan\theta = 0.0\overline{3}$$

$$\theta = 1.91^\circ$$

Vertical half angle

Similarly, Horizontal half angle:

$$\tan\phi = \frac{20}{300}$$

$$\phi = 3.82^\circ$$

7.2.5 Eclipse and Sunlit Time calculations

Eclipse Time (Cold Survival Case):

Orbit radius:

r = R_mars + h

r = 3390000 + 300 000 = 3690000 m

Orbital Period:

T = 2πsqrt(r^3/μ) = 2πsqrt((3.69*10^6)^3/(4.28*19^13))

T = 6806.4 sec = 113.42 mins

Eclipse Fraction:

f_E = (1/π)(cos^-1(sqrt(1-(R/r)^2))) = (1/π)(cos^-1(sqrt(1-((3.390*10^6)/(3.690*10^6))^2)))

f_E = 0.37076

Eclipse time:

T_E = (f_E)*(T) = (0.37076)*(113.42)

T_E = 2523.2 seconds **= 42.05 mins**

Sunlight Time (Hot operational Case):

T_s = (T - T_E) = 113.42 - 42.05 = **71.37 mins**

7.2.6 Battery Sizing Calculations

Battery sizing:

Eclipse time:

T_E = 42 .05 min = 0.701 hr

Cold-case internal power dissipation (first-order conservative assumption):

*** **this is the minimal internal electrical power consumed and dissipated by only essential satellite subsystems**

P_int = 300 W

Total power drawn during eclipse:

P_eclispe = P_int + P_heater = 300 + 200 W = 500 W

Battery Energy required per eclipse:

E_eclipse = P_eclipse * T_E = 500 *0.701 = **350.5 Wh**

Required end-of-life battery capacity (80% SOC constraint):

***__Battery is sized such that no more than 20% of its capacity is used during eclipse__

E_EOL = E_eclipse/0.20 = 350.5/0.20 = **1752.5 Wh**

Battery Degradation (9-Year Mission Lifetime):

Degradation = 30% = 0.30 (For our long-duration spacecraft mission, we assume a 30% capacity degradation margin for the preliminary design)

E_BOL = E_EOL/(1 - Degradation) = 1752.5/(1 - 0.30) = **2503.6 Wh** (Beginning-of-Life required battery capacity)

7.2.7 Solar Array Sizing Calculations

Solar Panel Sizing:

Sunlight time:

T_S = T - T_E = 113.42 - 42.05

T_S = 71.37 mins = 1.18 hr

Energy used during Sunlit Operations:

E_sunlit = P_sunlit * T_S = 1500 * 1.18

E_sunlit = 1784.3 Wh

Total energy required per orbit:

E_total = E_sunlit + E_eclipse = 1784.3 + 350.5

E_total = 2134.8 Wh

Required Solar Array Power:

P_array = E_total/T_S = 2134.8/1.1895

P_array = 1794 W = P_array = **1.8 kW**

Solar Array Area Estimation:

P_density = 200 W/m^2 (Assuming a first-order solar array output)

A = P_array/P_density = 1704/200

A = **9 m^2**

Solar Panel Degradation:

Power required at end of mission (P_EOL) = 1794 W, Mission life (t) = 9 years, r = 2.5% = 0.025 (yearly degradation rate assumption for a satellite)

P_BOL = P_EOL/(1 - r)^t = 1794/(1-0.025)^9

P_BOL = **2250 W** (initial solar power required)

7.2.8 Heater Sizing Calculations

Heater Sizing:

Assuming the heater is ON for the entire eclipse (conservative assumption for PDR-level sizing)

T_on = T_e

D = (T_e/T)*100 = (T_on/T)*100 = (42.05/113.42)*100 = 37%

Battery zone:

2 heaters * 25 W = 50 W

Propulsion:

6 heaters * 10 W = 60 W

Avionics:

4 heaters * 15 W = 60 W

Payload:

3 heaters * 10 W = 30 W

Total Heater Power Wattage (Survival Heaters):

P_heater = 200 W

Orbit-average Heater Power:

P_avg = P_wattage * D = 200 W * 0.37 = **74 W**

7.2.9 Radiator Sizing Calculations

Radiator sizing:

Internal Heat load (power dissipated by electronics, avionics, etc):

Q_hot = 1500 W

Radiator heat rejection rate (Assuming a first-pass radiator heat rejection capability):

Q_rad = 300 W/m^2

Compute radiator area:

A = Q_hot/q_rad = 1500/300 = 5.0 m^2

2 radiator panels with area of 2.5 m^2 each

7.2.10 Thermal Control Mass Budget Calculations:

<u>Radiator Mass:</u>
Total Radiator size (Area): 5.0 m^2 (2.5 m^2 radiator X 2)
Radiator areal density: 6kg/m^2 (first-pass assumption)
Total radiator Mass:
M_rad = (A_rad)*(p_rad) = (5.0)*(6)
M_rad = **30 kg**

<u>Heater Mass:</u>
Total Installed Heater power wattage: 200 W
Heater mass per-watt: 0.01kg/W (first-pass assumption)
Total Heater Mass:
M_ht = (P_heater)*(p_ht) = (200)*(0.01)
M_ht = **2.0 kg**

<u>MLI blanket mass:</u>
MLI areal density: 0.3 kg /m^3 (first-pass assumption)
MLI- covered area: 10 m^2 (first-pass assumption)
MLI mass = (A)*(p_MLI) = (10)*(0.3)
MLI mass = **3 kg**

Total_Mass = 45 Kg

7.2.11 Rocket Equation:

$$M_{prop} = M_{dry}\left(e^{\frac{\Delta V}{I_{SP}\, g_0}} - 1\right)$$

- M_{prop} = Required Baseline Propellant mass
- M_{dry} = Spacecraft dry mass (912 kg)
- ΔV = Spacecraft Maneaver velocity change (1449 m/s)
- I_{sp} = Bipropellant specific impulse (320 seconds)
- g_0 = Standard gravity (9.81 m/s²)

$$M_{prop} = 912\left(e^{\frac{1449}{320 \times 9.81}} - 1\right)$$

$$M_{prop} = 912\left(e^{0.4616} - 1\right) \approx 534.9 \text{ kg}$$

7.2.12 2-stage problem:

$$a = \frac{Vf - Vi}{t}$$

$$20 = \frac{Vf - 200}{10}$$

$$200 = Vf - 200$$

$$Vf = 400 \text{ m/s}$$

$$\Delta y_1 = Vit + \frac{1}{2}at^2$$

$$\Delta y_1 = 200(10) + \frac{1}{2}(20)(10^2)$$

$$\Delta y_1 = 2000 + 1000$$

$$\Delta y_1 = 3000 \text{ m}$$

$$\Delta y_2 = Vit + \frac{1}{2}at^2$$

$$\Delta y_2 = 400(15) + \frac{1}{2}(-9.8)(15^2)$$

$\Delta y_2 = 6000 - 1102.5$

$\Delta y_2 = 4897.5\ m$

$\Delta Y_{Total} = \Delta y_1 + \Delta y_2$

$\Delta Y_{Total} = 3000\ m + 4897.5\ m$

$\Delta Y_{total} = 7897.5\ m$

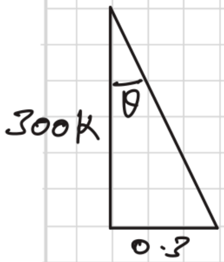
7.2.13 Jitter Estimations

Jitter estimations

∵ 1m / pixel

good estimation

↪ Jitter ≤ $\boxed{\text{0.3m on surface}}$



$$\theta = \tan^{-1}\left(\frac{0.3}{300000}\right)$$

$$\theta \Rightarrow \underline{\underline{1\,\mu\text{ rads}}}$$

Not reasonable for cost

Reduce to 5μ rads

↪ 1.5 meter

Avoid using Motion compensation

7.3 Table of Contributions

Name	**Contribution**

Sarthak Sahai	- 1.0 Project Description
	- 1.1 Motivation for the Project
	- 1.2 Proposed System Architecture
	- 2.0 Subsystem descriptions and correlation
	- 2.1 Payload Subsystem
	- 2.2 Structural Subsystem
	- Data Compression Strategy (14-bit to 8-bit conversion and JPEG2000 implementation)
	- Onboard Storage Sizing and Data Handling Architecture (1 TB capacity justification)
	- 3.0 Requirements Derivation and Formulation
	- Table 1: Mission Requirements for Payload and Structural system
	- 3.1 Payload Requirements Derivation
	- Table 2: Payload Requirements
	- 3.2 Structural Requirements
	- Table 3: Structural Requirements
	- 4.1 Payload Subsystem
	- 4.2 Structural Subsystem
	- 4.5 Command and Data Handling
	- Structural Design Integration with Atlas V 401 Constraints
	- Mission Operations Visualization and Supporting Figures
	- Appendix:
	    - 7.2.1
	    - 7.2.2
	    - 7.2.3
	    - 7.2.4
	    - 7.2.11

Prajakta Lal	- 1.3 Concept of Operations
	- 1.4 Modes of Operation
	- 2.5 Attitude Determination and Control Systems Subsystem
	    - 2.5.1 Introduction
	    - 2.5.2 Background
	    - 2.5.3 Supporting Evidence and Justification
	- 3.5 ADCS Requirements
	- Table 6. Mission/System Level Requirements

	- Table 7. ADCS subsystem Requirements - Risk Analysis - Cost Budget analysis
Hishaam Hassan	- 2.7 Propulsion Sub-system - 2.7.1 Purpose of the Propulsion Sub-system - 2.7.2 Subsystem Context within the Mission - 2.7.3 Proposed Design Summary - 2.7.4 Mission Architecture and Trajectory - 2.7.5 Design Specifications and Constraints - 2.7.6 Operational Heritage and Reliability - 2.7.7 Identification of Relevant Mission and System Requirements - 2.7.8 Mars Orbit Insertion (MOI) Criticality - 2.7.9 Thermal and Structural Integration - 2.7.10 TRL and Heritage - Table 11 - 4.0 Subsystem Design and Justification - 4.6 Propulsion Subsystem - 7.2.11 Rocket Equation - 7.2.12 2-stage problem
Sufyan Parvez	- 2.6 Electrical Power and Thermal Subsystem - 3.6 Electrical Power Requirements (Table. 9) - 3.7 Thermal Requirements (Table. 10) - 7.2.5 Eclipse and Sunlit Time calculations - 7.2.6 Battery Sizing Calculations - 7.2.7 Solar Array Sizing Calculations - 7.2.8 Heater Sizing Calculations - 7.2.9 Radiator Sizing Calculations - 7.2.10 Thermal Control Mass Budget Calculations
Meryl Almeida	- 2.3 Communications Subsystem - 2.4 Command and Data Handling Subsystem - 3.3 Communications Requirements - 3.4 Command and Data Handling Requirements - 4.3 Communications Subsystem design

8.0 Citations

[1] "Atlas V 401 Rocket Details," Space launchers - atlas, http://www.braeunig.us/space/specs/atlas.htm#:~:text=Construction%20is%20an%20aluminum%2Dlithium%20skin/stringer%20and%20frame.&text=The%20payload%20fairing%20consists%20of,long%2C%20and%204.2%20m%20diameter.&text=The%20Atlas%20V%20launch%20system,engines%20(1%20or%202).&text=Burn%20sequence:%20SRBs%20are%20ignited,next%20two%20at%20100%20s (accessed Mar. 20, 2026).

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