KEPLER - 42 Day Mars Mission
Preliminary Design Specification

KEPLER / GEOMETRIC MARS MISSION DRAFT

Created by:	Samuel Reid, CSA STCC Member, NCAGE L0FH6 Director	**Date**	3/15/2023	**Contact:**	
Checked by:	John Brandenburg, Kepler Aerospace	**Date**	3/15/2023	**Contact:**	
Approved by:	Brent Nelson, Kepler Aerospace	**Date**	3/15/2023	**Contact:**	

Canadian Space Agency Agence spatiale canadienne

Ext. Checked by	TBD, CSA	**Date**	03/15/2023	**Contact:**	First.last@canada.ca
Ext. Approved by	TBD, NASA	**Date**	03/15/2023	**Contact:**	First.m.last@nasa.gov

KEPLER AEROSPACE / GEOMETRIC PARTNERS

Geometric Energy Corporation

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List of Acronyms

1U, 2U, 3U 1, 2 and 3 Unit Kepler/Geometric Payload sizes, respectively
ICD Interface Control Document
ECSS European Cooperation for Space Standardization
PPOD Poly Picosatellite Orbital Deployer
PCBS PC/104 Printed Circuit Board Standard
COMMS Communication System
OBSW On-board Software
MSS Mission Support Software
ACRR Adjacent Channel Rejection Ratio
ABF Apply Before Flight
RBF Remove Before Flight
PDR Preliminary Design Review
CDR Critical Design Review
EM/EQM Engineering Model / Engineering Qualification Model
FM Flight Model
EMC Electro-Magnetic Compatibility
ESD Electro-Static Discharge
PCB Printed Circuit Board
VHF Very High Frequency
UHF Ultra High Frequency
TBC To be Confirmed
TBD To be Determined

SS Structural Subsystem
ADCS Attitude Determination and Control
EPS Electrical Power System
OBC On-board Computer
OBH On-board Data Handling
TTC Telemetry, Tracking and Command
TMC Thermal Management and Control
CPU Computer Processing Unit
OS Operating System
ECC Error Correcting Code
TMR Triple Modular Redundancy
FRAM Ferroelectric RAM

Reference Documents

1. Kepler/Geometric Payload Design Specification (CDS), Revision 13.
https://static1.squarespace.com/static
2. Poly Picosatellite Orbital Deployer MK. III Rev. E User Guide.
https://static1.squarespace.com/static/5418c831e4b0fa4ecac1bacd/t/5806854d6b8f5
b8eb57b83bd/1476822350599/P-POD_MkIIIRevE_UserGuide_CP-PPODUG-1.0-1_Rev1.pdf
3. PC/104 Printed Circuit Board Specification (PCBS), Version 2.6.
https://pc104.org/wp-content/uploads/2015/02/PC104_Spec_v2_6.pdf
4. Kepler Aerospace Starter Payload User Guide, Version 1.
https://Kepler Aerospace.com/wp-content/
5. European Cooperation for Space Standardization, Active Standards.
http://ecss.nl/standards/ecss-standards-on-line/active-standards/
6. NASA Technical Standards.
https://standards.nasa.gov/nasa-technical-standards
a. NASA General Environmental Verification Standard GSFC-STD-7000
https://standards.nasa.gov/standard/gsfc/gsfc-std-7000
b. NASA-STD-5001, Structural Design and Test Factors of Safety for Space Flight Hardware
https://standards.nasa.gov/standard/nasa/nasa-std-5001

7. NASA Planetary Protection Categories
https://planetaryprotection.nasa.gov/categories/
a. NASA NPD 8020.7G, Biological Contamination Control for Outbound and Inbound Planetary Spacecraft
https://nodis3.gsfc.nasa.gov/displayDir.cfm?t=NPD&c=8020&s=7G
b. NASA NPR 8020.12, Planetary Protection Provisions for Robotic Extraterrestrial Missions
https://standards.nasa.gov/standard/nasadir/npr-802012
8. UK Government Guide to Space Object Licensing
https://www.gov.uk/guidance/apply-for-a-license-under-the-outer-space-act-1986#ukregisters-
of-space-objects
9. OFCOM Introduction to Satellite Regulation
https://www.ofcom.org.uk/__data/assets/pdf_file/0025/107557/Satellite-regulation-tea
ch-in-event.pdf

1. Introduction

Kepler Aerospace and Geometric Energy/Medical/Labs/Space Corporation are initiated in the design, manufacture, test and operation of the first commercial Mars mission to launch in Q4 2026 from a 3-Stage Rocket Kepler Spaceplane/Spacecraft which would reach Mars in 42 days using the Kepler Drive™ electromagnetic propulsion system. KEPLER-42 shall be launched in Q4 2026 with Kepler Aerospace spearheading a Canada-U.S. Mars Mission with Geometric Energy Corporation as well as NASA and CSA personnel involved as well as partners to Kepler Aerospace including but not limited to the U.S. Department of the Navy, the U.S. Department of Defense, the Defense Advanced Research Projects Agency (DARPA), General Dynamics, Northrup Grumman, Lockheed Martin, Boeing, and with the NORAD HQ Deputy Commander invited to advise on behalf of NATO CAGE L0FH6 / U.S. DLA CAGE L0FH6. KEPLER-42 has approximately 3,000kg of payload space for its partners to use to send to Mars.

The following payloads have been selected:

● Small-satellite compatible roving system capable of achieving a top speed of 1 km/h.
● Imaging system capable of 360 degree HD photo and video for 3D mapping, virtual
reality experiences and gamification.
● Sensor suite with 9 degree of freedom inertial measurement units, sun and temperature sensors.
● Surplus processing power and memory for computational applications.

The goals for the KEPLER-42 mission are as follows:

1. To land on the Martian surface in a secure and controlled manner.

2. To demonstrate full-system and payload functionality and move 5m or out of the KEPLER lander's shadow, whichever is closer.

3. To semi-autonomously move 500m across the Martian surface under its own weight and power whilst logging data from inertial and imaging sensors.

2. System Requirements

The following requirements are made in addition to the Kepler/Geometric Payload Design Specification (CDS) Revision 13. In the case of any discrepancies between the two documents, this document takes authority. The Kepler/Geometric Payload described in this document shall integrate with a deployer of the Poly Picosatellite Orbital Deployer (PPOD) standard in accordance with the Poly Picosatellite Orbital Deployer MK. III Rev. E User Guide. All electrical and mechanical hardware within the Kepler/Geometric Payload described in this document adhere to the PC/104 PCB standard (PCBS) for mounting. The Kepler/Geometric Payload must satisfy the requirements of the Kepler Aerospace (lander) and SpaceX (launch vehicle) Interface Control Documents (ICD). The Kepler/Geometric Payload must satisfy the requirements for commercial from U.S. sites (AFSPCMAN 91-710, AFOSH-STD 48-9 and EWR 127-1). The

Kepler/Geometric Payload must be licensed to operate in the UK with the UK Space Agency under the Outer Space Act 1986. The Kepler/Geometric Payload must also complete filing and registration process with OFCOM for ITU licensing. The Kepler/Geometric Payload must comply with the NASA Planetary Protection provisions in accordance with Planetary Protection Mission Category Definitions.

2.1. Project Management

Every failure shall be recorded and reported in accordance with the failure reporting provisions of the project. Each and every failure shall be reported to the project leader (Brent Nelson) and test and integration engineer lead (John Brandenburg) in writing within 24 hours. Any major changes and issues encountered during the development of the Kepler/Geometric Payload will be reported to Kepler Aerospace by the project leader.

2.2. Standards Infringements / Conditions Requiring a Waiver

CDS 3.4.4: All deployables such as booms, antennas, and solar panels shall wait to deploy a minimum of 30 minutes after the Kepler/Geometric Payload's deployment switch(es) are activated from P-POD ejection.

2.3. Coordinate System

The coordinate system used for this project is the one recommended by the CDS, which is also relative to the P-POD (See Figure 1).
Figure 1: Coordinate system used for the Kepler/Geometric Payload (Source: CDS).

2.4. Materials

2.4.1. The payload is expected to use materials that correspond with NASA-STD-6016.
2.4.2. Hazardous materials inside the Kepler/Geometric Payload must satisfy requirements of
AFSPCMAN 91-710, Volume 3 (from CDS).

2.4.3. Total Mass Loss (TML) of Kepler/Geometric Payloads materials must be equal to or under 1.0%.
2.4.4. Collected Volatile Condensable Material (CVCM) must be equal to or under 0.1%.

2.5. Mechanical Requirements

2.5.1. Kepler/Geometric Payload main structure and rails must be made of Aluminium 7075,6061, 5005 and or 5052.
2.5.2. The center of gravity of a 2U Kepler/Geometric Payload shall be located within 4.5 cm from its geometric center on the Z-axis.
2.5.3. Deployables have to be constrained by the Kepler/Geometric Payload, not the P-POD.

2.6. Structural Subsystem

2.6.1. As the Kepler/Geometric Payload can only be accessed through the front door after integration into the deployer, the service interface on the Kepler/Geometric Payload must be located on the front side (+Z face). This is to allow for the team to perform activities such as health checks and charging. The number and type of connections can be selected at the designers discretion providing it is in accordance with the CDS.
2.6.2. Deployment switches shall be of the non-latching type, for both electrical and mechanical switch types.
2.6.3. The Kepler/Geometric Payload rails and standoffs, which contact the deployer rails, pusher plate, door, and/or adjacent Kepler/Geometric Payload standoffs, shall be constructed of a material that cannot cold-weld to any adjacent materials.

2.7. Attitude Determination and Control (ADCS)

The Kepler/Geometric Payload in question shall be operated on the Martian surface and is therefore not free flying and as such does not require de-tumbling or pointing capabilities.

2.8. Electrical Power System (EPS)

2.8.1. The Kepler/Geometric Payload shall provide sufficient power at the required voltages to
meet the power requirements of all satellite subsystems in all modes of operation using solar panels and batteries as required.
2.8.2. The Kepler/Geometric Payload shall be powered OFF during the entire launch and delivery sequence and until it is deployed from the deployment system onto the Martian surface.
2.8.3. The Kepler/Geometric Payload must be able to provide instant power to any roving modules between deployment and touch-down on the Martian surface which are used to ensure controlled and stable touch-down.
2.8.4. The Kepler/Geometric Payload shall be able to fully operate without battery charging, inspection or functional testing for a period of up to 6 months.
2.8.5. Batteries require a MSDS and additional testing for air freight and NASA safety certification. Testing procedures and requirements are detailed in JSC - 20793 Rev.B Crewed Space Vehicle Battery Safety Requirements. TBC by Kepler Aerospace.

2.9. On-board Computer (OBC) and Data Handling (OBH)

2.9.1. The Kepler/Geometric Payload shall collect and log telemetry data every one to five minutes for the entire duration of the mission during normal operation,
where telemetry is defined as the following set of parameters: time, status, raw battery voltage, raw battery current, 3V3 bus current, 5V bus current, OBC temperature, EPS temperature, battery temperature, COMMS temperature. The data packet format shall be defined during CDR.
2.9.2. The telemetry data shall be stored in the OBC until they are successfully downlinked.
2.9.3. Any computer clock used on the Kepler/Geometric Payload and ground segment shall exclusively use Coordinated Universal Time (UTC) for reference.
2.9.4. The OBC shall have a real time clock with an accuracy of 1 second during operation. Relative times should be counted / stored according to the epoch ‘01.01.2000 00:00:00’ UTC.

2.9.5. The OBSW and MSS shall protect itself against infinite loops, computational errors and possible lock ups.
2.9.6. All incoming commands, data and messages shall be checked for consistency. Any illegal input shall be rejected by the OBSW and MSS.
2.9.7. The OBSW shall only contain code that is intended for use on the Martian surface.
2.9.8. The OBSW and MSS shall not be allowed to override safety or redundancy features in hardware, such as the deployment switch. This is not applicable during service.
2.9.9. Restart procedure: The Kepler/Geometric Payload will be programed to reboot regularly during the mission. The time betweens reboots shall be determined at CDR.
2.9.10. A bit-flip mitigation policy must be established for the Kepler/Geometric Payload and integrated into overall system architecture. For example, watchdogs timers shall be used on the Kepler/Geometric Payload main computer to prevent malfunctions induced by bit flips. A Copy of Kepler/Geometric Payload program will be saved in rad-hard memory chips etc.

2.10. Telemetry, Tracking and Command (TTC)

2.10.1. 4G-LTE shall be used for primary downlink via the KEPLER landers or Other rovers, with a data rate of at least 115.2 kbps.
2.10.2. VHF shall be used for secondary downlink, with a data rate of at least 9.6 kbps.
2.10.3. BPSK or QPSK downlinks shall be used for VHF downlink because of their spectral efficiency.
2.10.4. All Kepler/Geometric Payloads shall have and make use of its unique satellite ID in the downstream telemetry.
2.10.5. 4G-LTE shall be used for primary uplink, with a data rate of at least 9.6 kbps.
2.10.6. UHF shall be used for secondary uplink, with a data rate of at least 9.6 kbps.
2.10.7. The UHF receiver shall have an ACRR of at least 100 dB

2.10.8. The Kepler/Geometric Payload shall have the ability to receive a transmitter shutdown command at all times after the Kepler/Geometric Payload's deployment switches have been activated after deployer ejection.
2.10.9. Once a transmitter shutdown command is received and executed by the Kepler/Geometric Payload, a positive command from the ground shall be required to re-enable the transmitter.
2.10.10. The Kepler/Geometric Payload provider shall have access to a ground station which has the capability and permission to send telecommands through an uplink to control its satellite and to upload and execute timed Instrument Command Files. The format of these commands is TBD at CDR.
2.10.11. The Kepler/Geometric Payload shall transmit the current telemetry log data and its unique satellite ID through a beacon at least once every 5 minutes. Exact number to be determined by Power Management Strategy.
2.10.12. The Kepler/Geometric Payload secondary radio system shall use the AX.25 Protocol (UI Frames). The data type during downlink shall be specified in the Secondary Station Identifier (SSID) in the destination address field of the AX.25 frame. Payload data shall be indicated using 0b1234 and telemetry data with 0b4321.

2.11. Thermal Management and Control (TMC)

2.11.1. The Kepler/Geometric Payload shall maintain all its electronic components within its operating temperature range while in operation and within survival temperature range at all other times after deployment.
2.11.2. A Thermal Management Strategy must be developed for the Kepler/Geometric Payload of the Kepler/Geometric Payload, which must include a thermal model for orbital transfer and Martian operations.
2.11.3. The Kepler/Geometric Payload shall survive within the temperature range of -20 degrees centigrade to +120 degrees centigrade from the time of launch until its deployment from the deployment system as defined by the Kepler Aerospace Starter Payload User Guide. These values are TBC by Kepler Aerospace before CDR.
2.11.4. The Kepler/Geometric Payload must be able to regulate its temperature within operational parameters passively or actively.

2.12. Roving

2.12.1. The activation and deployment of the roving system must occur during the free fall phase after PPOD deployment and before the satellite reaches the Martian surface.
2.12.2. There must be a deployment strategy in place which prevents premature deployment of the roving system while the Kepler/Geometric Payload is still inside the PPOD.
2.12.3. The deployment spring of the PPOD might be changed to reduce the deployment speed of the satellite onto the Martian surface.
2.12.4. The Kepler/Geometric Payload must be able to provide passive or active mechanical stabilisation to ensure a controlled and stable touch-down on the Martian surface after PPOD deployment.
2.12.5. The Kepler/Geometric Payload must be able to move along the Martian surface under its own weight and power.
2.12.6. The Kepler/Geometric Payload must be able to move a distance of at least 500 metres.
2.12.7. The Kepler/Geometric Payload must be able to move along the Martian surface unsettling as little Martian regolith as possible, which are a risk to solar panels, electronics and optics.
2.12.8. The Kepler/Geometric Payload must be able to remain in a static upright position when idle, without power and following any kind of failure.

2.13. General

2.13.1. The Kepler/Geometric Payload shall be designed to have an lifetime on the Martian surface of at least 14 days.
2.13.2. The Kepler/Geometric Payload shall not use any material that has the potential to degrade in an ambient environment during storage, which could be as long as approximately 1 year.
2.13.3. All electronic assemblies and electronic circuit boards should be conformally coated.
2.13.4. The Kepler/Geometric Payloads shall have a dedicated case for transport and storage.
2.13.5. All ABF items, including tags and/or labels, shall not protrude past the

dimensional limits of the Kepler/Geometric Payload extended volumes when fully inserted.
2.13.6. All RBF items shall be identified by a bright red label of at least four square centimetres in area containing the words “**REMOVE BEFORE FLIGHT”** or **“REMOVE BEFORE LAUNCH”** and the name of the satellite printed in large white capital letters.
2.13.7. The Kepler/Geometric Payload name shall be printed, engraved or otherwise marked on the Kepler/Geometric Payload and visible through the access hatch in the door of the deployer.
2.13.8. The build strategy e.g. Engineering Qualification Model - Flight Model (EQM-FM) approach shall be defined by a risk analysis.

2.14. Planetary Protection and Contamination Control

The Kepler/Geometric Payload must comply with the NASA Planetary Protection provisions NPD 8020.7G and NPR 8020.12 as a Class II mission in accordance with the Planetary Protection Mission Category Definitions.

3. Qualification and Acceptance Testing Requirements

The qualification and acceptance testing requirements and resulting environmental testing plan shall be based on the SpaceX 3-Stage Rocket Launch Spaceplane (Launch Provider) requirements. The Launch Provider’s environmental testing requirements supersede environmental testing requirements from any other sources. Testing procedures are described in NASA General Environmental Verification Standard GSFC-STD-7000.

3.1. Test Condition Tolerances

Tolerable uncertainties on environmental testing are available page 1-10 of GSFC-STD-7000.

3.2. Visual Inspection

Verification of the dimensions of the Kepler/Geometric Payload will be done following the Kepler/Geometric Payload Acceptance Checklist from the CDS.

3.3. Acceleration (Quasi-Static)

Figure 2: 3-Stage Rocket Launch Spaceplane payload design load factor for lighter payloads, less than 4000 lb.

● Limit level: As shown in figure 2 above.
● Qualification: Limit levels x 1.25.
● Proto qualification: Limit levels x 1.25.
● Acceptance: Limit levels x 1.0.

3.4. Resonance

The Kepler/Geometric Payload shall not have a fundamental frequency below 100 Hz in a hard-mounted configuration.

● Qualification: Limit levels x 1.25.
● Proto qualification: Limit levels x 1.25.
● Acceptance: Limit levels x 1.0.

3.5. Sinusoidal Vibration

The maximum predicted axial and lateral sinusoidal vibration environments are shown in Figure 3 and Figure 4 respectively for the 3-Stage Rocket Launch Spaceplane. These environments represent the vibration levels at the top of the payload attach fitting for Q = 20 through Q = 50, and envelope all stages of flight.
Figure 3: 3-Stage Rocket Launch Spaceplane maximum axial equivalent sine environment.

Figure 4: 3-Stage Rocket Launch Spaceplane maximum lateral equivalent sine environment.

● Limit level: As shown in figure 3 and figure 4 above.
● Qualification: Limit levels x 1.25, two octave/minute sweep rate.

● Proto qualification: Limit levels x 1.25, two octave/minute sweep rate.
● Acceptance: Limit levels x 1.0, four octave/minute sweep rate.

3.6. Acoustic

During flight, the payload will be subjected to a varying acoustic environment. Levels are highest near liftoff and during transonic flight, due to aerodynamic excitation. The acoustic environment is shown by both full and third-octave curves.
Note: random vibration testing may be substituted for acoustic testing for spacecraft less than 182 kg (400 lb) if customer analysis shows that the random vibration test is more severe than the acoustic test at all locations on the spacecraft.

Figure 5: 3-Stage Rocket Launch Spaceplane/Spacecraft maximum predicted acoustic environment (P95/50), 60% fill-factor, 131.4
dB OASPL (full octave).
Table 1: 3-Stage Rocket Launch Spaceplane maximum predicted acoustic environment (P95/50), 60% fill-factor, 131.4
dB OASPL (full octave).

Frequency (Hz) Acoustic Limit Levels (P95/50), 60% Fill- Factor, (Full Octave)

31.5 122.4
63 124.7
125 126.1
250 125.2
500 120.1
1000 114.4
2000 110.8
4000 107.8
8000 104.8
OASPL (dB) 131.4
● Limit level: As shown in figure 5 and table 1 above.
● Qualification: Limit levels + 6 dB, 2 minutes duration.

● Proto qualification: Limit levels + 3 dB, 2 minutes duration.
● Acceptance: Limit levels, 1 minute duration.

3.7. Shock Loads

Four events during flight result in loads that are characterized as shock loads:
1. Release of the launch vehicle hold-down at liftoff.
2. Stage separation.
3. Fairing deployment.
4. Spacecraft separation.

Table 2 shows typical payload adapter-induced shock at the spacecraft separation plane for 937 mm or 1194 mm (36.89 in. or 47.01 in.) clampband separation systems. Please note the actual flight shock levels produced by the payload adapter will be mission-unique.
Table 2: Payload adapter-induced shock at the spacecraft separation plan.

Frequency (Hz)	SRS (g)
100	30
1000	1,000
10000	1,000

● Qualification: Limit levels + 3 dB at the launch vehicle-to-spacecraft interface, three shocks in all three axes; or three activations of all significant shock-producing events in a flight-like configuration.
● Proto qualification: Limit levels + 3 dB at the launch vehicle-to-spacecraft interface, two shocks in all three axes; or two activations of all significant shock producing events in a flight-like configuration.
● Acceptance: Limit levels at the launch vehicle-to-spacecraft interface, one shock in all three axes; or one activation of all significant shock-producing events in a flight-like configuration. Note: shock tests performed based on “Limit levels.” Levels may be applied to all three axes simultaneously, or to each axis individually

3.8. Thermal Vacuum

SpaceX recommends thermal vacuum testing be conducted and may request insight into relevant testing performed.

- Ref T°: 10° / + 120 °
- Qualification: 10°C extension of maximum and minimum ref temperatures.
- Acceptance: 5°C extension of maximum and minimum ref temperatures.

3.9. Thermal Cycling

SpaceX recommends thermal cycle testing be conducted and may request insight into relevant testing performed.

- Ref T°: 10° / + 120 °
- Qualification: 25°C extension of maximum and minimum ref temperatures.
- Proto-flight: 20°C extension of maximum and minimum ref temperatures.

3.10. Venting

SpaceX recommends venting analyses be conducted and may request insight into relevant analyses performed. The payload fairing internal pressure will decay at a rate no higher than 0.35 psi/sec (2.4 kPa/sec) from liftoff through immediately prior to fairing separation, except for a brief period during the transonic spike. The transonic spike will have a time-averaged decay rate that is no higher than 0.65 psi/sec (4.5 kPa/sec), for no more than 5 seconds.

Figure 6: 3-Stage Rocket Launch Spaceplane fairing internal pressure and decay rate (representative).

3.11. Electro-Magnetic Compatibility (EMC)

The payload is expected to inhibit wireless transmission until deployed. Narrow band radiation emissions shall be maintained within the levels shown in figure 7 while in an active state near the landing vehicle.

Figure 7: Narrowband Radiated E-field Limits.

The payload is expected to maintain broadband radiation emissions within the levels shown in figure 8 while in an active state within close proximity to the landing vehicle.

Figure 8: Broadband Radiation Limits.

The SpaceX standard service includes an electromagnetic compatibility assessment SpaceX recommends electromagnetic interference/compatibility testing be conducted for RF-sensitive payloads and may request insight into relevant testing performed.

3.12. Electro-Static Discharge (ESD)

To be determined at CDR.

3.13. Contamination Control

The payload is expected to be maintained in clean assembly areas meeting ISO 8 cleanliness specification as a minimum during ground processing and payload integration according to IAW ISO 14644-1. The payload is expected to be handled only with clean gloves after testing in preparation for and during ground processing and payload integration. The payload surfaces are expected to be maintained visibly clean under black light inspection, per the IEST-STD-CC1246D level 300A as a minimum during ground processing and payload integration.

3.14. Planetary Protection

To be determined at CDR.

4. Preliminary Design

4.1. Payload 1 - Imaging System

The payload partners and roving module require sensors capable of capturing images and videos, horizontally and outward from the Kepler/Geometric Payload.

In addition to providing mission critical visual guidance for moving across the Martian surface, this setup could be used for producing 3D surface maps, VR experiences and gamification. The minimum sensors requirements are as follows:

- No. of sensors and their placement - 1+, for 360 degree coverage.
- Focus - fixed.
- Lens type - as required.
- Shutter - TBD.
- Resolution - at least 8MP total.
- Video frame rates - TBD e.g. 1080p 30fps, 720p 60fps or VGA, 90fps.
- Field of view - as required.
- Filter - as required.
- Image quality controls - as required.
- Operational Temperature - as per requirements and environmental testing standards.

The case for CCD vs CMOS should be based on power, speed, reliability, and availability. It can be shown that whilst CCD's can typically retrieve data at higher speeds, CMOS architectures typically consume less power and have a more compact form factor and therefore lower mass. Research also shows that CMOS and CCD architectures both suffer similarly from ionising radiation and displacement damage, but the key advantage of the CMOS is that there is no degradation of charge transfer efficiency. Further research online also shows that the vast majority of sensors easily and cheaply available are of the CMOS architecture. Commercially available camera modules with supporting electronics, frame and lens come in a variety of interfaces (CSI, SPI and I2C and Analog), with a wide range of resolutions. Many of these modules exist due to the hobbyist electronic market which caters mostly to the users of Arduino and Raspberry Pi. The CSI interface is a proprietary interface typically used to link cameras to Raspberry Pi modules using Linux and as such immediately limits its possibilities. The use of I2C, SPI or analog communication is however universal and provides the greatest flexibility when considering design and interface with other subsystems. The use of I2C and SPI also uses less pins on the bus, as multiple addresses are used on the same lines. This, when run to the PC/104 connector means that connected device could potentially access the cameras. There are multiple designs which could provide 360 degree coverage. The first using a single sensor which rotates using a motor. The second using 4 sensors (which can be placed on external faces or close together on a PC/104 board) each have a field of view which overlap within a reasonable distance. The third option utilises a single sensor which images using a fish-eye lens or hemispherical mirror. Option 1 and option 3 utilise a single sensor and custom optics. A single sensor can be considered a high-risk as it provides mission critical service for the rover, as well as a payload service. The main source of risk in these cases would be radiation damage, which will not be possible to test due to the limited project duration. Additionally, the requirement for custom lenses can add additional risk when considering development time as well as unknown behaviour during environmental testing. The sensor that meets all requirements with sufficient resolution is the Arducam OV2640 2MP image sensor. The module includes an image buffer and communicates with using I2C and SPI. This prefabricated module allows for the plastic components to be easily removed and replaced with space grade materials. Additionally, this repackaging could be used to enhance the systems mechanical, thermal properties and help mitigate radiation dose. The Arducam OV2640 2MP image sensor has been flown by EST West Virginia on the STF-1 Satellite. Launch was on December 16, 2018 and an image transmitted taken by the camera in orbit was published on January 3, 2019.

Figure 9: Arducam OV2640 with a custom CNC machined aluminium mount.

High temperatures might soften the lens adhesive and make the lenses move. Tests will be done to estimate the max survival temperature of the lens system. The OV2640 sensor has a storage temperature of -40° to 95°C. Tests will be done to estimate its maximal survival temperature.

4.2. Payload 2 - Inertial Measurement Payload

The payload partners require the integration of inertial measurements units capable of measuring linear acceleration, angular acceleration and magnetic field strength, each in 3 axes. The particular interest is in the acceleration forces, as local measurements of gravity, seismic activity and even impact activity could be measured. However, these kinds of sensors could also be used for navigation and positioning. There are many COTS 9 degree of freedom sensors available, however the Analog Devices ADXL345 and InvenSense MPU-9250 provide the greatest overall performance and have been selected for their merits by many other small satellite developers even though neither sensor has flight heritage. The MPU-9250 has a higher accuracy, lower power consumption, lower cost and smaller footprint than the ADXL345. The MPU-9250 also features a smaller design, with less dies. Both solutions use SPI or I2C for communication and require very few additional components to function. Due to all of these factors and to minimise risk, it will be desirable to implement at least between 4 and 8 sensors to maximise chance of retrieving reliable data. The MPU-9250 also incorporates a digital temperature sensor, which can be used to assess board temperature.

Figure 10 : Sparkfun breakout board with InvenSense MPU-9250 (Courtesy TDK).

4.3. Payload 3 - Passive Payload (TBC)

Potential promotional partners such as the Planetary Society may require the Kepler/Geometric Payload to physically show its members names, carry a passive storage device with data. As passive features, the additional complexity is minimal. The additional mass and volume of a microSDcard and holder (~5g) should be considered within the respective budgets.

4.4. Payload 4 - Kepler Aerospace Roving System

The use of a roving system is required to achieve the mission goals. An estimate for the systems required mass, power, volume, cost and feasibility should be generated for critical analysis during CDR. The fundamental roving principle was determined by Kepler Aerospace prior to PDR and shall based on a quadruped design which shall be composed of 4 main parts: structure (limbs and chassis), actuation (motors), sensing (motor encoders) and control (OBC or dedicated MCU). The following configurations are proposed:

4.5. Structural Subsystem (SS)

A COTS solution is desirable due to the low lead time, low cost and low risk of failure. Multiple COTS chassis exist which have flown on dozens of missions in LEO. Additionally, a standardised COTS chassis allows for board inter-compatibility across manufacturers due to the PC/104 standard, as well as external components, such as antennas and solar panels, making the design and interface process more practical. Optimisation with respect to structural mass is not considered for this project due to the additional time required for custom structure design, manufacture and testing. From a mechanical perspective the chassis shall not experience any forces outside of the typical operating range defined by launch and environmental testing. However, as an aluminium structure the chassis acts as a heatsink within the Kepler/Geometric Payload and due to the temperature extremes on the Martian surface, the chassis is expected to be important in determining the systems power and thermal budget. The two manufacturers of COTS chassis that meet these requirements are Innovative Solutions in Space and Pumpkin Space as shown by figures XX and XX respectively. However the immediate problem with the Pumpkin chassis is lack of access to external faces which are required for the imaging system and roving module. The top and bottom plates for the Pumpkin chassis follow a similar design methodology using sheet metal, however do offer large aperture and payload adapter versions for flexibility.

Figure 16a (left): 2U Kepler/Geometric Payload structures from ISIS Space. Figure 16b (right): 2U Kepler/Geometric Payload structures from Pumpkin Space.

4.6. Attitude Determination and Control (ADCS)

The Kepler/Geometric Payload in question shall be operated on the Martian surface and is therefore not free flying and as such does not require de-tumbling or pointing capabilities.

4.7. Electrical Power System (EPS)

4.7.1. Solar Panels

An equivalent of at least 1U of solar panels shall be placed on each +X, -X, +Y, -Y and -Z face so that at least 1U of solar panels are continuously illuminated. Each solar cell shall have an efficiency of at least 30%. It is expected that each solar panel incorporates a coarse sensor and temperature sensor. A COTS solution is desirable due to the low lead time, low cost and low risk of failure. Multiple COTS panels exist which have flown on dozens of missions in LEO e.g. ISISpace, GOMSpace, DHV technology.

4.7.2. Batteries

The Kepler/Geometric Payload shall have at least 20 W/hr of stored energy on-board to ensure near continuous operation. A power management system shall also be incorporated to manage the power inputs, outputs and general health. The system shall also incorporate a heater so that thermal gradients can be minimised. A COTS solution is desirable due to the low lead time, low cost and low risk of failure. A few COTS batteries with power management exist, however minimal information on flight usage is given. The only system found to satisfy these requirements thus far is the GOMSpace P31U.

4.7.3. Power Budget

A basic power model has been created to estimate the power input and output over the
duration of the mission. The model, shown in table 3, assumes the Sun moves across the sky at a constant speed with minimum possible illumination of sides.

Table 3: Simple Power Budget.

Subsystem	**Max. Power (W)**	**Min. Power (W)**	**Av. Power (W)**	**Use Fraction Daily**	**Daily Energy Used, Worst, (J)**	**Daily Energy Used, Best (J)**	**Daily Energy Used, Average (J)**
Roving motors (Motion)	9.2	1	5.1	0.20	158976	17280	88128
Roving motors (Idle)	0.92	0	0.46	0.80	63590	0	31795
OBC	1	0.2	0.6	1.00	86400	17280	51840
Imaging	0.14	0.03	0.085	0.20	2419	518	1469

Primary Comms. 0.4 0.4 0.4 0.20 6912 6912 6912
Secondary Comms. 2 2 2 0.05 8640 8640 8640
Battery Heater 3 3 3 0.00 0 0 0
Total Energy Per Day (Whr) 91 14 52
Total Energy Needed (Whr) 1271 197 734
Battery Capacity (Whr) 20
Single Solar Panel Area (m^2) 0.01
Solar Intensity (W.m^-2) 1420
Time Illuminated (Days) 14
Solar Panel Efficiency 0.3
Total Energy Generated (W/hr) 707

4.8. Connectors

4.8.1. High reliability connectors and cables will be used to reduce risk of environmental testing failures. Example of such connectors are Harwin's Gecko line.

4.9. On-board Computer (OBC) Hardware Configuration

4.9.1. While there are many different configurations of hardware that have flight heritage, the choice of options for KEPLER-42 is limited. Tight scheduling constraints preclude the design of a custom OBC while tight budget and weight constraints preclude the use of more advanced units such as ESA's Leon 4.
4.9.2. The overwhelming majority of Kepler/Geometric Payloads have been in low-earth orbit where there is significant Van Allen shielding from cosmic radiation. Combined with short lifespans, this has limited the requirements on rad-hard components. Typical Kepler/Geometric Payload projects have had budgets in the 10s of thousands of euros, significantly below that required for high end equipment.
4.9.3. The 3 days travel time and 14 days roving time on the Martian surface means that the rad tolerances required are high with 1000s of single-event-effects (SEEs) per day, orders of magnitude higher than expected in low earth orbit. Handling this is the primary design challenge for the computational hardware/software synergy.

4.9.4. In light of the budgetary, scheduling and mass constraints, four strategies are proposed for the on-board computer.
4.9.4.1. The use of a single pre-built radiation hardened or tolerant on-board computer. The RH-OBC-1 from Vorago and Hercules from Gauss Team are the only viable solutions. The individual components on the Vorago board have significant flight heritage but the board itself has never flown. The mass and cost of the unit means that only a single OBC can fly. This single point of failure is a risk but will reduce the development time and reduce the *unknown unknowns* of using a novel system. This is the lowest power/weight solution but potentially the most expensive.
The Hercules board is currently undergoing environmental testing and has no flight heritage or price.
4.9.4.2. Use of a swarm approach. For this approach, many individual cheap computers are designed, each with independent memory and access to flight critical components as well as the ability to reset companion units. This will significantly increase the computational complexity as well the power consumption but reduce the cost and
present an exciting new approach to design. A minimum of 6-8 independent Raspberry Pi Zeroes could provide the base for this with every sensor/component linked to at least 2 OBCs. Alternatively, 3 ISIS on-board computers could be used.
4.9.4.3. Use of a combination approach. A cheaper professional OBC such as the ISIS on-board computer would be used for the flight critical components. This unit is not rad hardened but has significant flight heritage. A smaller *swarm* of Raspberry Pi Zeroes could be used for the less critical components such as the cameras.
4.9.4.4. Use of a *modular* system. While design of a whole new OBC is impossible, interfacing a rad hard CPU with an existing modular motherboard is achievable. Pumpkin, Inc sell the MBM unit, a motherboard with attachable processor daughter boards. A custom processor daughter board would be designed in-house.
4.9.5. An assessment of the options suggests the following comparison:

Name Single Swarm Combination Modular
Cost High Low Medium Medium
Development Time Low Medium High Medium

Weight Low Medium High Low
Power Consumption Low High Medium Low
Risk Medium Unknown High Medium
Libraries Available Medium High Medium Medium
Processing Power Low Medium High Medium
Additional Electronics Low High Medium Medium
Flight Heritage High Low Medium Medium-High
4.9.6. In light of the much reduced weight and complexity, options 1 and 4, appear to present the most viable solutions.
4.9.7. Upon reaching out to Vorago it was discovered that the unit has not received environmental testing.
4.9.8. This means option 4, the rad hard processor addition to a pumpkin motherboard, presents the preferred approach.
4.9.9. There is one clear leader for processor choice, the VA10820, a rad hard version of the ARM Cortex-M0. This provides sufficient processing power while still having low power consumption and many reliability features.
4.9.10. Use of the VA10820 will present a number of challenges as it lacks the support for many higher level concepts such as memory management. This will allow for greater control of the unit but increase development times.

4.10. On-board Computer (OBC) and Data Handling (OBH)

4.10.1. The OBC will handle data transfer to KEPLER over a 4G modem as well as to earth via a UHF/VHF connection.
4.10.2. The OBC will handle monitoring of all sensors, including the 9 degree of freedom IMU sensor suite payload.
4.10.3. The OBC will handle recording and reporting of telemetry data - time, status, raw battery voltage, raw battery current, 3V3 bus current, 5V bus current, OBC temperature, EPS temperature, battery temperature, COMMS temperature.
4.10.4. The OBC will handle control of the motors on the rover unit.
4.10.5. The OBC will handle monitoring and reporting of the on-board cameras.
4.10.6. The OBC will handle the roving module.
4.10.7. The OBC will handle data storage and verification during down periods.
4.10.8. There are a number of options for OS choice for Kepler/Geometric Payloads including FreeRTOS and Kubos. FreeRTOS is lower level and more widely

used while Kubos is specifically designed for space applications with some inbuilt error correction. The choice of OS will be dominated by the choice of configuration and choice of components.

4.10.9. The OBC will be capable of booting in the 5-6 seconds between activation and touch down. This rules out the use of high level operating systems.

4.10.10. For rad hardened processors it is likely that a very light real-time operating system will be the most that can be supported. In practice, systems such as the Vorago VA10820 do not require an operating system with the developer taking control of memory management.

4.10.11. The OBC will work with four categories of memory:

4.10.11.1. Permanent, non volatile *bootloader memory* . This is the base program that is loaded on reset. In the case of the VA10820, this is 128kB.

4.10.11.2. Temporary, on-board memory. This is the memory used by the processor during operation and can be wiped on reset. In the case of the VA10820, this is 132kB.

4.10.11.3. High reliability *storage memory* . This is the memory used to store mission history and telemetry data. This can be wiped before launch but must then be persistent over CPU resets. This allows for the processor to launch aware of state and history. This will be integrated over SPI and must include error correction using TMR or ECC. The reliability of the system will depend on this and must be at least 100MB in size.

4.10.11.4. Non-critical *payload data* . This is memory that stores images from the on-board cameras and similar components ready to shipping to earth. This will not include any process critical data or commands. This must be equal to at least three times the Fsize of the planned data volume returned via KEPLER.

4.10.12. The OBC will use a range of error handling and redundancy features for common error sources

4.10.12.1. Jump - Use of NOP slides and function token passing.

4.10.12.2. I/O registers - Use of multiple reads, parity bits, hash codes and cyclic updating.

4.10.12.3. Critical data - Use of function, function parameter, test, branch, register and instruction triple modular redundancy (TMR). The OBC will include a *memory scrubber*.

4.10.13. The use of significant error handling will mean that the effective storage and clock rate for the microprocessor will equal to less than one third of the rated specification.
4.10.14. The OBC will have a watchdog system as well as brown out monitoring.
4.10.15. The OBC shall use ground reporting and packet protocols such as Ball Aerospace COSMOS.

4.11. Telemetry, Tracking and Command (TTC)

4.11.1. The ability to capture images and video is highly dependent on the radio communications systems once on the Martian surface. Both images and videos can be captured and indirectly returned to Earth if the Kepler Aerospace high-bandwidth 4G-LTE network is used. In this case, communications with Earth is dependent on the link with the Kepler Aerospace KEPLER lander and Other Rovers. The standard package allows 115.2 kbit/s for downlink and 9.6 kbit/s for uplink.
4.11.2. If this mode of communications fails, the Kepler/Geometric Payload should become reliant on a secondary radio system for direct Mars-Earth communications using VHF downlink and UHF uplink. In this scenario it is likely that only basic telemetry and images can returned to Earth with downlink rates between 1.2 kbit/s and 9.6 kbit/s for up to 8 hours a day. The VHF/UHF secondary connection will be constructed from standard COTS systems with significant flight heritage.
4.11.3. Radiation hardened 4G-LTE systems are not currently available. At least 4 EC25 LTE modems from Quectel will be used with the maximum of redundancy. PCI-e mounted versions of these are available for testing while the engineering and flight models will use directly surface mounted versions. Telit and u-Blox are two mainstream manufacturers and modems in the E910 or Toby R2 series would be strong alternative modems.
4.11.4. The speed of the LTE connection will be limited by the processing speed of the OBC and throttled by the KEPLER lander. The OBC will be capable of adapting its data rate to the available downlink and uprate bandwidths.
4.11.5. LTE antennas have little space history but are widely used in mobile comms. Quectel as well as other modem manufacturers provide detailed

documentation on antenna design and a great deal of flexibility is available in the form factor. SMD antennas are available from suppliers including Taoglas.

4.12. Thermal Management and Control (TMC)

A simple thermal balance is shown in table 4 below. In this situation the heat received directly from the Sun, albedo contribution, planetary radiation contribution and heat radiation into space are considered. In this situation the equilibrium temperature of the Kepler/Geometric Payload is expected to reach between 81 and 89 degrees celsius during Martian day. This calculation assumes the entire spacecraft surface area is coated with GaAs solar panels or a material with similar IR emissivity and absorbance properties. During the Martian night the spacecraft is not expected to survive, as the equilibrium temperature of -100 degrees celsius greatly exceeds the minimum operating temperature of every component in every system.

Table 4: Simple Thermal Budget.

DAY HOT Intensity (W.m^-2)

Est. Projected Area (m^2)

Heating power (W)

Solar 1420 0.03 42.6

Martian IR 1335 0.01 13.4

Martian Albedo 140 0.01 1.4

Internal 13 1.00 13.0

Constants

Stefan-boltzmann 5.67E-08 (W.m^-2.K^-4)

Surface Area 0.08 (m^2)

Surface Coating

Solar Panels

(Si) White Paint Black Paint

Solar Panels

(GaAs)

IR emissivity 0.82 0.90 0.90 0.80

Solar absorbance 0.75 0.15 0.90 0.88

Temperature (K) 351.8 296.7 354.7 362.1

Temperature (C) 78.8 23.7 81.7 89.1

DAY COLD Intensity (W.m^-2)

Est. Projected Area (m^2)
Heating power (W)
Solar 1280 0.03 38.4
Martian IR 1114 0.01 11.1
Martian Albedo 130 0.01 1.3
Internal 13.0
Constants
Stefan-boltzmann 5.67E-08 (W.m^-2.K^-4)
Surface Area 0.08 (m^2)
Surface Coating
Solar Panels
(Si) White Paint Black Paint
Solar Panels
(GaAs)
IR emissivity 0.82 0.90 0.90 0.80
Solar absorbance 0.75 0.15 0.90 0.88
Temperature (K) 343.7 290.3 346.4 353.8
Temperature (C) 70.7 17.3 73.4 80.8
NIGHT Intensity (W.m^-2)
Est. Projected Area (m^2)
Heating power (W)
Solar 0 0.03 0
Martian IR 5 0.01 0.05
Martian Albedo 0 0.01 0
Internal 3.2
Constants
Stefan-boltzmann 5.67E-08 (W.m^-2.K^-4)
Surface Area 0.08 (m^2)
Surface Coating
Solar Panels
(Si) White Paint Black Paint
Solar Panels (GaAs)
IR emissivity 0.82 0.90 0.90 0.80
Solar absorbance 0.75 0.15 0.90 0.88
Temperature (K) 171.8 167.9 167.9 172.9

Temperature (C) -101.2 -105.1 -105.1 -100.1

4.13. Preliminary Mass, Volume and Engineering Model Budget

A budget has been created to estimate basic Kepler/Geometric Payload parameters. The previous design sections have been used to estimate values to create this first iteration. The Kepler/Geometric Payload shall fit within a ??U form factor, have a mass between ?? g and ?? g and shall cost between ?? and ?? to produce.

Table 5: Basic Mass, Volume and Cost Budget

Sub-System

Manufacturer No.

Required Unit Price (€) Total Price (€)

Unit Mass (g)

Unit Volume (U)

2U STRUCTURE ISISpace 2 €3,780 €7,560 198 -

ON-BOARD COMPUTER

TBD 2 €5,280 €10,560 85 0.15

VHF/UHF

TRANSCEIVER

ISISpace

2 €11,880 €23,760

85 0.15

VHF/UHF ANTENNA

ISISpace 85 -

EPS + BATTERIES GOMSpace 2 €7,920 €15,840 200 0.30

SOLAR PANELS Various 5 €4,800 €24,000 350 -

ROVING MODULE Internal 1 €27,600 €27,600 199 0.50

MIXED PAYLOAD

MODULE Internal 1 €18,000 €18,000 198 0.50

TOTALS €127,320 1400 1.60

MARGIN 5% €133,686 1470 1.68

MARGIN 7.5% €136,869 1505 1.72

MARGIN 10% €140,052 1540 1.76

4.14. General Configuration

The basic configuration for KEPLER-42 is shown in figure 17 below. The positioning of all modules, except the rover module shall be determined at CDR when an advanced power and thermal budget becomes available. The modules with the highest mass have currently been shifted in Z+ to aid stability, whilst taking requirement 2.5.2 into consideration. The battery and EPS module shall have a heater built in, which is also why it is positioned close to the OBC incase temperature control is required.
Figure 17: Simple cross-section schematic of KEPLER-42.
The imaging payload has been moved to the extreme in Z- to help avoid issues with dust coated lenses. The first issue with this is that the imaging system will be unable able to see the Martian surface approximately 10 cm immediately in front of the Kepler/Geometric Payload, even with a 130 degree field of view. The second issue that may occur is due to the distances between the OBC and Image, Sensing and 4G-LTE module which communicate using I2C and SPI. This shall be tested during CDR. This is not expected to be an issue for the VHF/UHF transceiver and VHF/UHF antenna as this uses a dedicated 50 ohm harness. Protective panels shall be incorporated to help prevent ingress of dust into the Kepler/Geometric Payload and protect the roving system from getting stuck on small rocks or damaged. A simple system level schematic, shown in figure 18 highlights the power and data pathways for the Kepler/Geometric Payload.
Figure 18: Simple system schematic of KEPLER-42.

4.15. Mission Architecture and Communications Budget

A preliminary mission architecture is presented in figure 19 below. A more detailed and accurate version shall be produced at CDR with broken out by specific operation into hours and minutes. A preliminary communications budget between 200 mbit/day and 400 mbit/day is expected to be required for operation.
Figure19: Preliminary mission architecture for KEPLER-42.

4.16. Build Strategy

Build strategy is a key part of every Kepler/Geometric Payload project as both in-house and COTS components are prone to failures during environmental testing and long lead times. Strategies such as building two flight models in parallel vs. single engineering model and single flight model consecutively shall be assessed. An extensive bill of materials with transport, handling and cleaning procedures shall also be produced at CDR.

4.17. Project Timeline

The project phases have been outlined in table 6 below. Each phase is linked to Kepler Aerospace and payload partner payment schemes.
Table 6: Project Timeline

Phase Duration Description
Phase 1 27/12/2023 - 27/03/2024 PDR
Phase 2 27/10/2024 - 27/01/2025 CDR
Phase 3 27/11/2024 - 27/02/2025 EM
Phase 4 27/03/2024 - 27/04/2025 EM Test
Phase 5 27/12/2024 – 27/05/2025 FM
Phase 6 27/09/2025 - 27/01/2026 FM Test
Phase 7 27/02/2026 - 27/06/2026 Margin
Phase 8 27/06/2026 - 1/12/2026 Delivery and Integration
Phase 9 10/12/2026 – 11/01/2027 Launch Window
Mars Intercept (Landing) 2/20/2027 +/-

5. References

To be populated at completion of PDR.
References currently contained within text in hyperlinks.
Appendix A: 3-Stage Rocket Launch Spaceplane Temperature and Cleanliness Environment
Phase Control
System

Approx. Duration
Temp. °C (°F) Humidity Cleanliness (Class)
Flow Rate (cfm)
Spacecraft processing Payload processing facility heating, ventilation and air conditioning
3 weeks 21 ± 3 (70 ± 5) 50% ± 15% 100,000
(Class 8)
N/A
Propellant conditioning
Facility heating, ventilation and air conditioning (HVAC)
3 days 21 ± 3 (70 ± 5) 50% ± 15% 100,000
(Class 8)
N/A
Spacecraft propellant loading Facility heating, ventilation and air conditioning (HVAC)
Mission-Unique
21 ± 3 (70 ± 5) 50% ± 15% 100,000
(Class 8)
N/A
Transport to hangar (CCAFS only)
Transport trailer unit
<2 hrs 21 ± 3 (70 ± 5) 0%-60% 10,000
(Class 7)
(supply air cleanliness)
1,000
Encapsulated in hangar
Ducted supply from hangar facility HVAC
1 week 21± 3(70±5) 50% ± 15% 10,000
(Class 7)
(supply air cleanliness)
1,000
Encapsulated roll-out to pad
None
30-60 min
N/A N/A 10,000

(Class 7)
Encapsulated
on pad (vertical or horizontal)
Pad air conditioning
1 day
VAFB:
Selectable 15 to 35 (59 to 95) CCAFS:
Selectable 16 to 30 (61 to 86)
Selectable: 0% to 65% 10,000
(Class 7)
(supply air cleanliness)
1,500

Appendix B: Temperature on Mars throughout a Martian day

Figure B-1 : Surface temperature at the Taurus Littrow heat flow site determined from thermocouple measurements. Vertical bars are estimates of the error limits. Source Night time surface temperature have been measured around 106 (±2)°K. The data shows rapid temperature changes at sunset and sunrise. At sunrise, the ground temperature is around 300 (±20)°K. At mid-day, the temperature is 380 (±5)°K.

Appendix C: Spacecraft Charging Analysis

Find other sources/papers, the theory for surface charging on Mars described below is based on a single paper from the Solar System Exploration Division, NASA Goddard Space Flight Center and the Johns Hopkins Applied Physics Laboratory. Mars being outside of the Van Allen belt, the Kepler/Geometric Payload might only be exposed to plasmas trapped in the Van Allen during transportation to Mars. The Kepler/Geometric Payload will be grounded to KEPLER during Mars transfer and the spacecraft charging mitigation belongs to Kepler Aerospace during this phase of the mission. Details TBC by Kepler Aerospace. Figure C-1: Van Allen Belts, LEO, MEO, HEO and GEO are represented. Spacecraft charging is a problem for satellites orbiting in those orbits as they are surrounded by a charged plasma. Source: Crosslink Summer 2003 After deployment, the Kepler/Geometric Payload will be in contact with Mars surface. Because Mars is not internally conductive, there is no “ground”. It is therefore possible for Mars surface to have different surface potentials at different localisations (See Figure C-2). Surface charging on Mars surface is induced by the solar wind plasma. It is made of photoelectrons, plasma electrons, plasma ions and surface-emitted secondary electrons. When the solar UV photons hit Mars surface, they release photoelectrons (surface electrons) with a current of approximately 4 uA/m2. Those photoelectrons develop a surface potential of a few volts (around +3V). At large solar zenith angles (SZA), the influx of solar wind plasma is reduced, less ions particles are hitting Mars surface. However, the electron plasma generated on the day site is mobile and as full access to Mars surface. Therefore, the polarity of the surface is negative at places with large solar angles. At the Terminator, the surface has a potential of - 40V. At the anti-solar point the surface polarity can be as low as -200V.

Figure C-2: Original caption “Martian driving currents and surface potentials including photo and plasma sheath” Source: *Rover wheel charging on the Martian surface, NASA Goddard*

The solar wind is also creating potential variations across Mars surface with the topographic features (See Figure C3). If the solar wind can’t access some parts of Mars surface because of rocks, craters, hills or mountains, some shades will be visible. Because the solar wind ions are not flowing to the shady parts and the electron cloud created on illuminated surfaces next to it are mobile, the surface potential in shades is negative.

Figure C-3: Original caption: “Illustration of solar wind deflection into a polar crater” Source:
Rover wheel charging on the Martian surface, NASA Goddard

If the Kepler/Geometric Payload is moving on Mars surface, the moving parts touching the surface will induce a tribological charging. It is an effect similar as walking on a carpet. Figure C4 and figure C5 show results of modelling done by Jackson et al. It shows that:

● Potential stays around 0 V on the dayside

● Important variations of potentials happens when wheels are rotating in the shades (nightside and craters).

● Surface charging in shades is roughly proportionally increased by the wheel rotation speed.

● Small grains create more charging than large grains (up to x20)

● Particles sticking to the wheel decrease the magnitude of the charging (x4)
Figure C-4 : Modelling results from Jackson (1/2) Source: *Rover wheel charging on the Martian surface, NASA Goddard*
Figure C-5 : Modelling results from Jackson (2/2) Source: *Rover wheel charging on the Martian surface, NASA Goddard*
Design recommendation from paper: metals wheels connected to chassis are better than electrically insulated rubber wheels.
General recommendations: due to the changing potential of Mars's surface (between +3V and -40 V), it is necessary to electrically insulate the electronics components of the Kepler/Geometric Payload from the ground (and the plasma? Faraday cage?). If driving in the shade is a requirement, the rover wheels need to be connected to the chassis and outer surfaces of the Kepler/Geometric Payload. The connection to the high surface area of the Kepler/Geometric Payload will allow for a faster return to the ambient plasma potential than if the wheel is electrically insulated.

Appendix D: UV Radiation levels on Mars

To be populated.

Appendix E: Radiation Environment on Mars

Radiation can create non-destructive soft errors (Single Event Upsets or SEU) with bitflips or transient pulses in logic or support circuitry but also hard errors potentially destructive (Single Event Latchup or SEL). A power reset is necessary to remove SEL and they can be detected because of components having high operating currents above specifications.
Radiation can therefore alter the measurements and operations of chips as well as physically destroy them; typically kRads are necessary as shown by table E1.
Table E-1 : Original Caption “Summary of TID results for PIC24 and dsPIC33 devices”
Source : Radiation Test Results for Common Kepler/Geometric Payload Microcontrollers and Microprocessors
Shielding greatly decreases the Total Irradiation Dose (TID). For example, an unprotected chip will get something around 113.5 kRad/year in GEO and 1 mm of aluminium can bring it down to 2 kRad (Source). However, it is impossible to completely protect the chips from all radiations as some particles have very high energies.

Figure E-1 : Left - Results of simulations on the effect of Aluminium Shielding on different types of radiations, Source . Right: Effect of Aluminium Shielding measured at Louvain-La-Neuve, Source Mars' radiation environment has only two main components as it is situated outside of the Van Allen belts: Galaxy Cosmic Rays (CGR) and Solar Particle Events (SPE). Three radiation detectors were fitted on the Martian Reconnaissance Orbiter (LRO). The detectors were shielded with 1 mm of Aluminium to reduce noise. Figure E2 shows an average radiation of 20-30 mRad/day. The peaks on the graph around Aug-2010 / Sep-2010 are due to solar eruptions and are nearly 5 times higher than the usual radiation dose (up to 95mRad/day).

Figure E-2 : Radiation measurements from the LRO Source: LRO's website

Further analysis of CraTER's data shows that radiations dose rate are extremely variable. In a year of low solar activity like 2016 (Figure E3), 20-30 mRad/day are constantly measured. Nonetheless, some years have much stronger solar activities. 2012 is an example with 5 peaks above 1 rad/day and even two above 100 rad/day (Figure E4). Those peaks are also the highest amount of radiations recorded during LRO measurements (2009 - 2018). Solar activity is difficult to predict and a solar eruption can happen anytime during the mission. From CraTER's data it is possible to formulate minimum radiation requirements for the satellite. A worst case scenario would be that there is a constant solar eruption during the entire mission. In this scenario, the satellite will be exposed to 1.4 kRad. (100 Rad/day during 14 days). A best case scenario, is that the sun has a low activity during the entire mission. The satellite will then be exposed to 280 mRad. (20 mRad/day during 14 days).

Figure E4 - Radiation measured on LRO during the year 2016.Data from CraTER website.

Figure E3 - Radiation measured on LRO during the year 2012, peaks of radiation happen during solar storms. Data from CraTER's website.

Energy distribution of radiation on Mars

Definitions:

LET: *"Linear Energy Transfer. The rate of energy deposit from a slowing energetic particle with distance travelled in matter, the energy being imparted to the material. Normally used to describe the ionization track caused by passage of an ion. LET is material-dependent and is also a function of particle energy. For ions of concern in space radiation effects, it increases with decreasing energy (it also* *increases at high energies, beyond the minimum ionizing energy). LET allows different ions to be considered together by simply representing the ion environment as the summation of the fluxes of all ions as functions of their LETs. This simplifies single-event upset calculation. The rate of energy loss of a particle, which also includes emitted secondary radiations, is the stopping power. " Source: ECSS standard E-ST-10-04-C*

Cross-section: *"The number of events per unit fluence. Expressed in units of cm 2/device or cm 2/bit. In the event of the device being tilted at an angle Θ, the fluence must be corrected by multiplying the fluence by Cos Θ." Source: ECSS standards.*

With the data collected by the CRaTER detector, it is possible to estimate the number of ions that will be hitting the Kepler/Geometric Payload on Mars surface. Figure E5 shows the LET of particles measured by the 6 detectors of CraTER.

Figure E-5 - Radiation measured on LRO for a month. Data from CraTER's website .

A later objective will be to compare this estimation with the tests done by the JPL to estimate cross-sections of typical Kepler/Geometric Payload electronics (Source). First, the LET unit is converted to the one used in the paper of the JPL. To go from KeV/um to MeV cm^2/mg, it is necessary to divide by the density of the materials of the detectors (Si density = 2320 kg.m^-3 or 2320mg.cm^-3) and convert KeV to MeV and cm to um. MeV cm^2/mg is a unit used to compare ion beams. The energy dissipated in a substrate is roughly proportional to the density of the substrate. By normalising the energy of the ion beam by the detector density, it makes comparison of ion beams easier. KeV/um is the unit of the CraTER data as the detectors directly measure the amount of energy transferred from a charged particle to the detector.

Figure E-6: Data from CraTER with LET converted in MeV cm^2/mg and Flux integrated on a half sphere (Data from detector 1&2 collected between 2014-01-14 and 2015-01-14)

Figure E6 shows that each cm^2 of the Kepler/Geometric Payload on Mars Surface will be hit:

- Every week by a particle with a LET of 1 MeV cm^2/mg
- Everyday by a particle with a LET of 0.4 MeV cm^2/mg
- Every hour by a particle with a LET of 0.1 MeV cm^2/mg
- Every minute by a particle with a LET of 0.02 MeV cm^2/mg
- Every second by a particle with a LET of 0.0025 MeV cm^2/mg

TBC: Comparison of the above energy distribution of particles with the radiation tests done by the JPL (Source) to estimate the number of SEL that the Kepler/Geometric Payload will encounter on the Martian surface.

Conclusion

Mars being way outside of the Van Allen belt, the radiation environment is only made of GRB and SEP. Levels of GRB and SEP measured on ISS are similar to the one measured on LRO. For this mission, it can be assumed that the radiation dose will be of 20 - 30mRad/day during normal sun activity and around 100 mRad/day during solar eruptions. From this number, a worst-case Total Irradiation Dose (TID) would be of 1.4 kRad. It is a low level of radiation and it is unlikely to create failures electronic components as COTS can usually withstand at least 5 kRads. Also, radiation levels on Mars are lower than in LEO. It means that COTS Kepler/Geometric Payload components that are capable of surviving months in LEO should be able to perform well on Mars. However, those mRads/day of radiation are made of GRB. Those are impossible to stop with shielding due to their high energy and are powerful enough to generate Single Event Effects and affect data storage or calculations with bitflips. It is therefore necessary for the mission to have an OBC with rad-hard components or a radiation tolerant architecture.

Appendix F: Radiation Levels Outside the ISS

It is interesting for this project to characterize radiation in the LEO environment, as most COTS Kepler/Geometric Payload components have operated in this environment.

Figure F-1: Summary of space environment hazards for typical orbits Source: Crosslink Summer 2003 4 main components of radiations were measured on the outside of the ISS for 442 days with the R3DR2 detector. It is possible to identify the origins of the radiations because different radiation sources emit types of particles (electrons, nuclei, protons) at various energy levels with little overlaps. Dachev and al. give the following definition for the types of particles detected by the R3DR2 detector (Source):

"Galactic cosmic rays (GCRs) are a significant radiation component in the near-Earth and free-space environment consisting of 98% nuclei and 2% electrons. These energetic charged particles originate from sources beyond the solar system and convey energy ranging from several keV/nucleon up to 1012 MeV/nucleon."

“High-energy charged particles are trapped by the Earth’s magnetic field and form two distinct belts of toroidal shapes surrounding the Earth: the Inner Radiation Belt (IRB) and the Outer Radiation Belt (ORB).”
“The IRB is situated at an altitude from 0.2 to 2.0 Earth radii at the geomagnetic equator and consists of electrons, with energies of up to 10 MeV, and protons with energies of up to 700 MeV.”
“The ORB is located in the altitudinal range from 3.4 to 10 Earth radii. The ORB energetic population is electrons with energies of a few MeV.“
“Solar flares and coronal mass ejections, caused by sporadic eruptions in the chromosphere and corona of the Sun, produce high fluxes of charged solar energetic particles (SEP) with energies up to several GeV.”

Table F1 : Summary of radiation levels measured on the ISS (2014 - 2016) (Source)

Radiation Type Radiation Levels [mRad/day]

Galactic Cosmic Ray Particles (GRB) 7.16

Protons in the South Atlantic Anomaly Region of the inner radiation belt (IRB) 56.7

Relativistic electrons and/or Bremsstrahlung in the outer radiation belt (ORB): 27.8

Solar Energetic Particle (SEP) events: 9

Total **92.5**