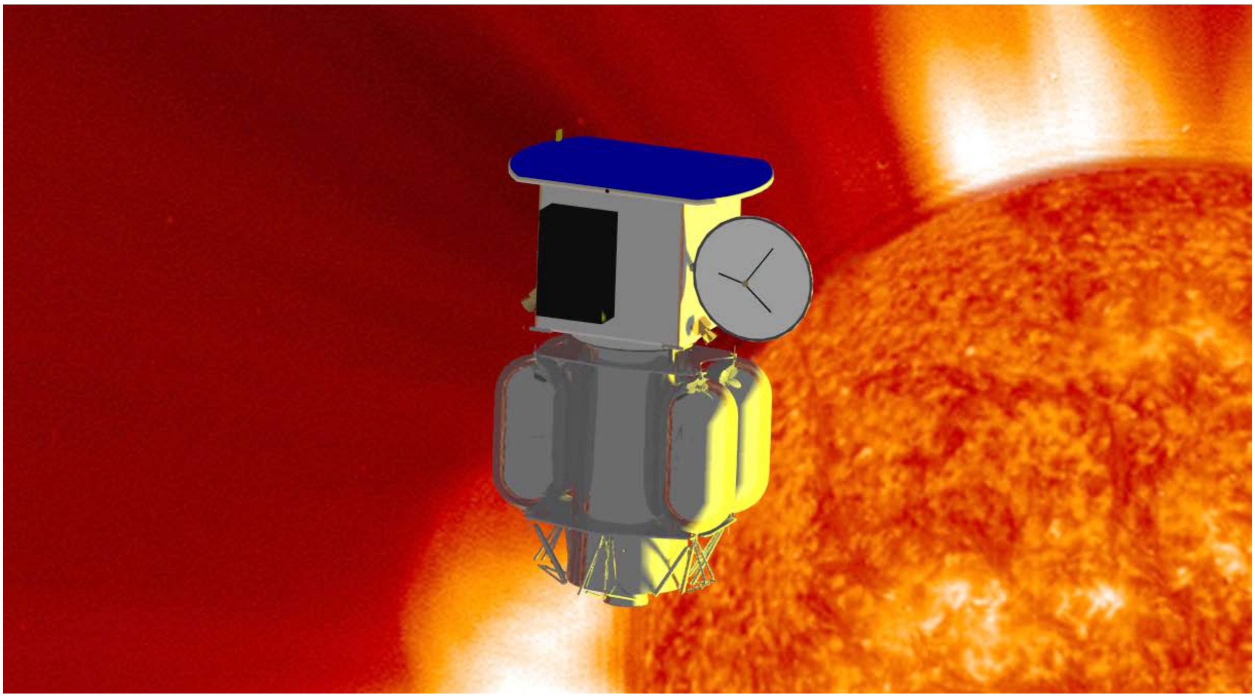
CDF Study Report

HAGRID

Heliospheric imaging for Assessment of Global and Regional Infrastructure Damage



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List of Acronyms

AOCS	Attitude and Orbital Control System
AIT	Assembly Integration and Test
AIV	Assembly Integration and Verification
APE	Absolute Pointing Error
AU	Astronomical Unit
AVM	Avionics Model
BOL	Beginning of Life
CAD	Computer Aided Design
CBE	Current Best Estimate
CCD	Charge-Coupled Device
CDMS	Command and Data Management System
CEB	Camera Electronics Box
CFRP	Carbon Fibre Reinforced Plastic
CHT	Cylindrical Hall Thruster
CME	Coronal Mass Ejection
CMS	Carbon Monoxide and Methane Spectrometer
COTS	Commercial Off The Shelf
DSM	Deep Space Manoeuvre
DSN	Deep Space Network
ECSS	European Cooperation for Space Standardization
EGSE	Electronic Ground Support Equipment
EPC	Electronic Power Conditioner
EMC	Electro Magnetic Compatibility
EOL	End of Life
ESA	European Space Agency
ESATAN	European Space Agency Thermal Analysis software
ESD	Electro Static Discharge
FDIR	Fault Detection, Isolation and Recovery
FM	Flight Model
FMECA	Failure Modes Effects and Criticality Analysis
FOV	Field of View
FPGA	Field Programmable Gate Array
FY	Financial Year
GS	Ground Station
GSE	Ground Support Equipment
HGA	High Gain Antenna
HI	Heliospheric Imager
HKTM	House Keeping Telemetry Monitoring
HOP	High Output Paraphin
ICU	Instrument Control Unit

IMU	Inertia Measurement Unit
Isp	Specific Impulse
ISRO	Indian Space Research Organisation
ITAR	International Traffic in Arms Regulations
LED	Light Emitting Diode
LEO	Low Earth Orbit
LEOP	Launch and Early Operations Phase
LGA	Low Gain Antenna
LISA Pathfinder	Laser Interferometer Space Antenna
LPF	LISA Pathfinder
Ls	Solar Longitude Difference
ML	Multi-Layer Insulation
MOC	Mission Operations Centre
MOI	Moment of Inertia
MR	Mission Requirements
OBC	On Board Computer
OCO	Orbiting Carbon Observatory
OSR	Optical Solar Reflector
PA	Product Assurance
PCDS	Power Conditioning and Distribution System
PDE	Pointing Drift Error
PFM	Proto-Flight Model
PRM	Propulsion Module
PSF	Point Spread Function
PSLV	Polar Satellite Launch Vehicle
ROI	Region of Interest
RPE	Relative Pointing Error
SCIP	Sheared Coherent Interferometric Photography
SEB	SECCHI Electronics Box
SMEI	Solar Mass Ejection Imager
SNR	Signal to Noise Ratio
SOC	Science Operations Centre
SR	System Requirements
STEREO	Solar Terrestrial Relations Observatory
STK	Satellite Tool Kit
STM	Structural Thermal Model
STR	Star Tracker
TBD	To Be Determined
TC	Tele-command
TCS	Thermal Control System
THEMIS	Thermal Emission Imaging System
TMTC	Telemetry and Tele-command
TRL	Technology Readiness Level
TT&C	Telemetry Tracking and Control
UR	User Requirement
WSB	Weak Stability Boundary

1. Executive Summary

The primary objective of the Heliospheric Imaging for Assessment of Global and Regional Infrastructure Damage (HAGRID) Concurrent Design Facility (CDF) study was to assess the feasibility of a low-cost technology pathfinder mission capable of providing accurate and timely warnings/alerts of terrestrial space weather events. These alerts would be integrated within a global response system to aid the operational space weather forecast, protecting critical human infrastructure.

The HAGRID design centred on a spacecraft carrying a Heliospheric Imager beyond the Sun-Earth line from where it could image Earth-directed space weather events. Earth-directed solar wind transients, in particular Coronal Mass Ejections, are known to impact space and ground based technological systems at Earth. Solar wind tracking is most effective when the transients can be viewed from a position perpendicular to their direction of propagation. Techniques have been developed using data from science missions such as STEREO, demonstrating that a heliospheric imager can be used for real-time predictions when near-real time data are available.

The HAGRID concept is designed around low-cost COTS components including a duplicate of a STEREO Heliospheric Imager in order to create a low-cost platform from which observations can be taken, transmitted to Earth and processed with sufficiently low latency that they can be used for genuine predictions of Earth-directed solar wind transients. For operational purposes only sub-fields of the images need be transmitted, allowing the telemetry rates to be kept low (~500 bps).

During the CDF study, it was shown that it is feasible to construct a spacecraft that could be placed into an Earth-like heliocentric orbit by a low-cost launcher. The orbital characteristics would cause the spacecraft to drift ahead (or behind) the Earth at a rate that increased the Earth-Sun-spacecraft angle at around 22.5 degrees per year, similar to STEREO. Unlike STEREO however, HAGRID would carry sufficient fuel to stop the spacecraft drift relative to the Earth. In this scenario, the spacecraft would drift out to a position some 60° ahead (or behind) the Earth and then stop relative to our planet. From this location, Earth-directed CMEs could be imaged ahead of their arrival at Earth.

The CDF study considered the feasibility of both a separate and integrated propulsion module and concluded that both designs were possible, with the advantage that a separate propulsion module would ultimately result in a smaller, more manoeuvrable spacecraft during the operational phase of this mission.

It was originally hoped to make observations during the cruise phase but this proved challenging for accurate pointing and would require a redesign of the HI door-closure mechanism so that the HI door could be closed to protect the instrument during the final thruster firing that would place it in the operational orbit. Removing the requirement to make observations during the cruise phase would solve these issues and also removed the need for a steerable high-gain antenna. It is recommended that this trade trade-off is studied further in a follow-on study.

In order for the data to be useful for operational predictions of solar wind conditions, the latency within the system was considered. A worst-case scenario was considered to be the Carrington event of 1859 during which a CME was launched that reached Earth within 17 hours. With a requirement for a warning to be issued at least six hours prior to a CME's arrival at Earth, this leaves 11 hours from the first observation for the data to be transmitted, processed and a warning issued. The latency from transmission to prediction was estimated to be of the order of 7.2 hours, well within the required time.

While the HAGRID study focussed on a mission sent ahead of the Earth at a relative Earth-Sun-Spacecraft angle of 60 degrees, the study showed that the spacecraft could be stopped at any point after being injected into a heliocentric orbit. The ideal position being a trade-off between viewing geometry, cruise time and the telemetry budget. The study also showed that the HAGRID mission could also be launched into an orbit lagging behind the Earth. Indeed, such a position would be desirable if the spacecraft was also carrying instrumentation to make in-situ measurements of any co-rotating solar wind streams ahead of their arrival at Earth (due to the solar rotation). The CDF study considered a mission carrying only a Heliospheric Imager and experience from the STEREO mission has shown that the Heliospheric Imager facing the Sun-Earth line from behind the Earth in its orbit is subject to micrometeoroid impacts that compromise the efficiency of the instrument. An additional advantage of placing a spacecraft in a heliocentric orbit ahead of the Earth is that co-rotating solar wind streams can be viewed by the imager before they arrive at Earth. Though this study concentrated on stopping the spacecraft 60 degrees ahead, a smaller angle of 40 degrees would also enable co-rotating solar wind streams to be imaged ahead of their arrival at Earth.

The HAGRID study was necessarily limited in its scope but it nonetheless demonstrated that such a low-cost operational space-weather mission was indeed eminently feasible. The 1878 kg launch mass for the composite option is compliant with the baseline launch vehicle. In taking this concept forward, consideration should be given to a cost-benefit analysis of modifying the instrument and in optimising the data telemetry and ground-station specifications.

2. Introduction

RAL Space has recently developed a Concurrent Design Facility (CDF) for mission-level pre-phase-A studies. It is located at the International Space Innovation Centre (ISIC) on the Harwell Campus in Oxfordshire, UK. When a pilot study was being sought for qualification of the CDF, an obvious candidate was a pathfinder for an operational space weather alert mission. A pathfinder mission was proposed as such a mission has more flexibility in design compared with a full operational mission. The pre-phase-A CDF study is based on re-use of the STEREO/HI instrument design and goes by the working title HAGRID (Heliospheric Imaging for Assessment of Global and Regional Infrastructure Damage).

The study took place during the interval extending from March to June 2011. Experts from RAL Space, Astrium, SSBV and the University of Southampton participated in the study - and this included scientific and instrument direction from a study team that was led by Drs Chris Davis and Chris Eyles, with participation from Dr Jackie Davies, Prof. Mike Hapgood and Prof. Richard Harrison. We summarise below the baseline for the study, key trade-offs considered, and the major conclusions drawn, together with the identification of open issues requiring future study.

2.1 Mission baseline and key requirements

HAGRID consists of a single spacecraft in an Earth-leading 1 AU heliocentric orbit. The payload consists of a single instrument, a wide-field imaging photometer based directly on the design of STEREO/HI (i.e. essentially build-to-print).

A major goal of the study was to assess the feasibility of ‘stopping’ a spacecraft in a 1 AU orbit, at a location remote from the Earth. Results from such an assessment are critical for deciding on the best approach for space weather applications, for example in deciding whether it is best to deploy a single spacecraft (stationary with respect to the Sun-Earth line) or a single spacecraft drifting in the same direction away from the Earth. In the latter concept, as the spacecraft drift away from Earth, it would only be at prime location for a limited amount of time to observe Earth-directed CMEs and CIRs.

Although in the CDF study, a single-spacecraft in an Earth-leading 1 AU solar orbit was assessed, the basic conclusions would be equally applicable to an Earth-trailing mission. A vantage point remote from the Sun-Earth line enables Earth-directed solar wind transients to be imaged and their speeds and trajectories accurately determined. A location ahead of the Earth is considered here since it also alleviates technical difficulties due to instrument pointing offsets caused by micro-meteoroid impact.

Although the intention was to re-use the basic design of STEREO/HI, it is necessary to make provision for an Instrument Control Unit (ICU) to perform the functions provided for HI by the SECCHI Electronics Box (SEB) (and all the other other solar imaging instruments on-board STEREO). Such functions include controlling the cameras and acquiring the CCD images, processing, compressing images, and providing telecommands, telemetry and power interfaces with the spacecraft. The ICU is a new development but the requirements are straightforward so it is not expected to be a major cost or schedule driver.

A further development of the STEREO/HI design that has been considered was to replace the one-shot door release system with a re-latchable mechanism based on a stepper motor - this is relevant for the option of physically stopping the spacecraft at the preferred operational location in its orbit. It permits useful observations to also be made during the spacecraft cruise phase, followed by a door close and re-open cycle that avoids possible contamination of the optics during the final thruster firings required to stop at, for example, L4.

A Rockot launcher option was selected for HAGRID, primarily on the basis of low cost, although piggyback launches, if available, could be considered. The HAGRID mission design draws significantly on experience gained from the LISA Pathfinder mission. It is suggested that direct injection from elliptical LEO to escape orbit would be used to achieve the final operational orbit. A Lunar gravity assist was not considered, this being regarded as excessively complicated and offering little advantage over simpler options in the case of a single-spacecraft configuration (unlike for STEREO where lunar fly-bys were required to place the two spacecraft, launched by a single rocket, into two very different heliospheric orbits).

The key requirements for the HAGRID mission are that:

- Cruise phase duration (from launch to start of operation) shall be less than a year;
- Routine operations shall last for at least 3.5 years;
- The required CME surface brightness sensitivity of the HAGRID imager shall be half that of STEREO/HI;
- The alert time provided shall be a minimum of 6 hours, and shall be maximised (although alert time must be tensioned against alert reliability);
- The cost of development and operation are low.

3. Mission Overview

3.1 Objectives and rationale

Space weather is the impact of the solar wind on the Earth's space and ground-based technological systems. Our Sun is a variable star undergoing an approximately eleven year activity cycle. During this cycle, vast eruptions occur that throw billions of tonnes of plasma (and its associated magnetic field) from the solar atmosphere into space at speeds of up to 2500 kms^{-1} (and potentially faster). These eruptions, known as Coronal Mass Ejections (CMEs) pose a significant risk to the efficient operation of spacecraft, aircraft and ground-based technologies including power-grids. It is therefore desirable to predict the occurrence of such eruptions and track their speed and direction in order to predict their arrival at Earth.

The UK-built Heliospheric Imagers (HIs) on-board the twin NASA STEREO spacecraft have proved revolutionary in this respect. Launched into Earth-like orbits, one drifting ahead of the Earth and the other lagging behind, the HIs are able to image the solar wind from outside the Sun-Earth line. From such vantage points it is possible to determine the radial speed and direction of CMEs by triangulating observations from both spacecraft. In addition, techniques have been developed to determine both the speed and direction of such solar wind transients by observing their propagation across the wide field of view of one HI instrument on a single spacecraft. While STEREO is a science mission, with high-resolution data downlinked to Earth via the Deep Space Network (DSN), there is also provision for a continuous low-resolution downlink of the data in near-real time via a network of smaller ground-stations receiving data on a best-efforts basis. CME tracking techniques have been applied to data from this near real-time space weather beacon to demonstrate that the STEREO HIs can be used to make genuine predictions of the arrival of space weather events at Earth.

Since their launch in 2006, the STEREO spacecraft have been continually drifting away from Earth at a rate that increases the spacecraft-Sun-Earth angle by approximately 22.5 degrees per year. At the time of writing, this angle is in excess of 110 degrees for each spacecraft and as this angle increases further, the ability of this science mission to observe Earth-directed events will decline. Without STEREO, our ability to monitor space weather conditions will be restricted to images of the solar wind made along the Sun-Earth line by the LASCO coronagraph on the SOHO mission. The LASCO instrument plays an important role in identifying Earth-directed CMEs though it is challenging to determine their speed and direction from this vantage point since they appear as a faint halo around the Sun in LASCO images. While a CMEs speed can be approximated from the expansion of the halo, it cannot be measured directly. The SOHO mission is already 17 years old and the development of future space weather monitoring missions is urgently required.

Modern society has an increasing reliance on technological systems that are at risk from space weather events (such as spacecraft, air travel and extended ground-based power grids). Consequently it is important to develop concepts for missions that help mitigate space weather effects. This study has focused on establishing a mission concept based on heliospheric imaging that satisfies the key observational and operational space weather requirements, with particular emphasis on experience gained from our intimate involvement with similar instruments on the Coriolis and STEREO spacecraft (namely SMEI and HI, respectively).

Some aspects of the STEREO/HI and SMEI observations have demonstrated aspects suitable for space weather applications. These include the following:

- Image cadence rates of order 30 minutes to 2 hours (depending on elongation range observed) have been shown to be adequate for efficient solar transient tracking.
- The brightness sensitivity and stray-light rejection levels achieved by STEREO/HI and SMEI have been sufficient for the effective identification of CMEs and CIRs (see Tables 6 and 4) in the heliosphere out to, and beyond, 1 AU.
- The spacecraft resources required for an HI-type instrument are modest, e.g. mass of 16 kg, power 12 W, (excluding data handling systems) and telemetry of 10 kbps (average) for STEREO/HI.
- Analysis techniques have been developed that determine the longitudinal directions of solar transients with respect to a single spacecraft (and therefore the Earth).
- The experience being gained in forecasting both with real-time beacon and post-event data from the STEREO spacecraft. The studies to date appear to satisfy the requirements in terms of lead times and accuracy of predicted arrival times.
- The STEREO/HI field of view encompasses elongations from 4° to 88° and this range has provided comprehensive coverage for CME tracking from the corona to Earth and beyond. For space weather monitoring purposes this range can be reduced, resulting in less stringent requirements for instrument baffling and telemetry rates. Minimising the field of view in this way will retain a capability for tracking events to the Earth and an ability to image the solar transient structure. Restricting the field of view between 10° to 70° with a (nominal) 20° extent perpendicular to the ecliptic would still enable Earth-directed events to be identified and tracked with sufficient precision.
- Reducing the size of the field of view could potentially simplify the instrument to a single camera with a significant reduction in complexity, resources and cost (although this did not come within the scope of this initial study).

On the other hand, STEREO is a science research mission. While it has informed our understanding of the best operational strategy, several aspects of that mission are not suited to space weather monitoring:

- A long-term space weather monitoring platform needs to be positioned between 20 and 90 degrees with respect to the Sun-Earth line and not drifting away. The STEREO spacecraft are now located at over 100 degrees from the Earth and are thus now far beyond the ideal location for observing Earth-directed space weather events.
- Space weather monitoring requires 24/7 contact, data processing and assessment. STEREO has occasional, scheduled contact periods with science data delivered to the PI groups some days after the event.
- The STEREO Beacon mode has shown potential for real-time monitoring, but is significantly degraded with respect to the science data due to telemetry constraints. Comparisons of event tracking (post event) show that analysis of the science data provides far more accurate forecasts. Given the relatively low telemetry rate for an HI-type instrument a dedicated real-time telemetry rate near to the science rate of STEREO is not excessive. Values of order 10 kbps (average rate) would be appropriate based on the STEREO/HI experience (Table 5).

Finally, several additional points feed into the selection of preferred mission scenario, namely:

- While geometrical modelling of plasma structures aligned along the Parker Spiral has suggested that L5 (behind the Earth in its orbit) is a better location for a spacecraft than L4 (ahead of the Earth) for detecting and tracking CIRs, experience with the STEREO spacecraft indicates the reverse. Significantly more CIRs have been detected by the HI on the STEREO Ahead spacecraft. This may be due to particle impacts affecting the efficiency of imaging from the HI on STEREO Behind where the instrument faces the ram direction of the spacecraft. Factors relating to other subtle differences in the performance of the two HI instruments cannot be excluded, however.
- Any ‘open’ instrument imaging the Sun-Earth line from a spacecraft behind the Earth in its orbit will image in the direction of the spacecraft motion. Particle impacts on STEREO/HI-B have been much more severe than anticipated, resulting in a degradation of the pointing stability of the instrument which in turn affects the clarity of long-exposure images. Steps to make a future instrument more rigid will have merit but may not remove the problem completely.
- Studies have shown that it is not necessary to place a spacecraft as far away from the Earth as the L4 or L5 points to make effective out of Sun-Earth line observations for space weather applications. A spacecraft can be ‘stopped’ with respect to the Earth, at any point along Earth’s orbit of the Sun. The CDF study stresses the feasibility of doing this. A position ahead of the Earth but closer than L4 (60 degrees ahead of Earth) might be the optimum site and provides advantages in the available telemetry, and in the time taken before final orbit insertion. However the CDF study shows that stopping as far as L4 is technically feasible.
- Geometrical factors involving the Thomson sphere and the so-called locus of enhanced visibility, for CIRs, have been considered and strongly suggest that a spacecraft ahead of the Earth, at 40° to the Sun-Earth line is ideal, i.e. 20° closer than the L4 point.
- Any long-duration space weather monitoring system requires minimal mechanisms and uses synoptic observations. We anticipate a single mechanism, for opening and closing the instrument door, that could be used only once or possibly twice as outlined in the CDF study conclusions. We do not anticipate any variation in the observation mode during the mission.

3.2 CDF Study Overview

The Heliospheric imaging for Assessment of Global and Regional Infrastructure Damage (HAGRID) concept builds on the knowledge gained from the STEREO mission and applies this to a dedicated low-cost operational space-weather mission.

The HAGRID study was carried out over six one-day sessions within the Concurrent Design Facility (CDF) which forms part of the International Space Innovation Centre (ISIC) at Harwell, Oxfordshire. Before the study commenced, consideration was given to the objectives and scope of the study.

3.2.1 Study Objectives and Scope

The budget was restricted to fifty million pounds including launch, payload and provision of ground-segment infrastructure. The budget for running the ground segment was restricted to five million pounds per year.

The objectives were set by consideration of the observed range of space-weather parameters such as speed and frequency of Earth-directed CMEs along with the end user requirements for advanced warning of space weather events. This is a technology demonstrator for an operational space weather mission and so imaging, telemetry and latency requirements are set by the operational requirements of the end user. These are likely to be significantly less than those required for a science mission like STEREO.

The aims of the study were summarised as follows;

- The HAGRID mission is a low-cost technology demonstrator mission for a future operational Space Weather Alert System.
- HAGRID will provide early warning of Space Weather Events to the end users of the system.
- The Space Weather forecasts provided to the end users will be used to inform the operational planning of systems vulnerable to space weather effects.

From the outset, the HAGRID study had a well-defined scope. The mission would provide images of Earth-directed CMEs from a position outside the Sun-Earth line which would be down-linked with a latency compatible with providing useful alerts to the end user. In order to reduce costs, the mission design would use Commercial Off The Shelf (COTS) components where possible, only low-cost launch options were considered and the instrument design would be a duplicate of the HI instrument used successfully on the STEREO mission. After consideration of the telemetry budget, an investigation of the ground-segment focussed on the latency involved in processing the data and the cost of providing the necessary ground-station coverage.

Within the time-frame of the study, various trade-offs were considered.

- Stopping the spacecraft at a fixed position from Earth versus a constant relative drift between the two (as with the STEREO spacecraft).

- A spacecraft with an integrated propulsion module versus a stand-alone spacecraft with separate propulsion module

- Use of a mono versus a bi-propellant

The following points were outside the scope of the CDF study

- Detailed definitions for issuing space weather alerts (green, amber or red)

- Requirements for modifying existing ground-stations

- The minimum data rate required to issue a warning

- While there could be advantages to power, mass and telemetry constraints in modifying the HI instrument this was not considered to be part of the initial study

3.2.2 User Requirements

The HAGRID mission is considered to be a technology pathfinder for an operational space-weather mission. The CDF study was carried out using the following user requirements;

UR01. That the mission shall commence routine operations within six years of the commencement of Phase A

UR02. For planning purposes, Phase A shall be assumed to start on 1st February 2012

UR03. The length of time between launch and the commencement of routine operations shall be less than two years

UR04. Routine operations shall last for at least 3.5 years

UR05. The cost of the programme up to and including flight commissioning, shall be less than £50 million (based on costs from FY2011/12)

UR06. The cost of the operations shall be less than £5 million per year

UR07. The HAGRID HI instrument shall have sufficient sensitivity to detect a CME twice as bright as the detection threshold of the STEREO HI instruments. **Comment:** this is an initial estimate and may have to be revisited in light of the latest results from the STEREO mission

UR08. The HAGRID Mission shall raise an amber alert to the end user when a CME has been detected which has an estimated probability greater than 1% of inducing a space weather event at Earth

UR09. The HAGRID mission shall provide estimates of the geomagnetic Kp index as a result of any predicted space weather event

UR10. The HAGRID Mission shall provide estimates of the time taken for a space weather event will reach Earth with an uncertainty of better than 20% with a 90% confidence level

UR11. The minimum alert time shall be greater than six hours (this is set by the lead time required for the airline industry to take operational decisions). **Comment:** A worst-case scenario is generally considered to be the Carrington event of 1859 during which a CME reached Earth around 17 hours after launch. For a modern-day equivalent event, a warning would need to be issued at least 6 hours ahead of the CME's arrival at Earth, meaning the data must be down-linked, analysed, and a warning issued within 11 hours. (Goal: The lead time between an alert and the arrival of a space weather event shall be maximised)

UR12. The Mission shall provide data coverage exceeding 99% and provide space weather services with a reliability exceeding 80% for the duration of the mission

UR13. There shall be sufficient spacecraft consumables, and margin in the EOL performance of the spacecraft to extend the mission duration a further 2.0 years

3.2.3 System level study requirements

The study requirements define the scope of the CDF study and are not in general applicable to the eventual mission requirements.

SR01. The scope of the study shall include the space and the ground segments. **Comment:** The scope of the Ground Segment (GS) is limited to the assessment of the latency in the delivery of the space weather data to the end users (inc. link-budget) and GS costing. The scope of the GS has been limited as the architecture of the alerting system (and its inter-relationship with other systems) is complex and not well understood

SR02. The number of spacecraft in the HAGRID mission shall be limited to one

SR03. The launch service shall insert the spacecraft in the correct initial orbit and be compatible with the development cost envelope. **Comment:** The intent of this requirement is to limit the scope of the assessment of the launcher; for example, the spacecraft could be launched as either a primary or secondary payload providing that the cost envelope is respected.

SR04. The spacecraft shall be three-axis stabilised. **Comment:** This is both a derived requirement from the payload and an architectural decision to exclude spin-stabilised attitude control with a de-spun payload module

SR05. The primary instrument shall be a wide field imaging photometer based on the STEREO/Heliospheric Imager (HI) instrument. **Comment:** other instruments may be considered to augment the quality of the space weather forecast data. The justification for selecting HI as a core instrument on this mission is the fact that HI data from STEREO has been shown to be useable for prediction of CME related space weather events.

SR06. For the drifting scenario, routine operations shall commence when the spacecraft is greater than 20° from the Sun-Earth line and shall cease when the spacecraft is more than 90° from the Sun-Earth line. **Comment:** This is derived from analysis of SEREO/HI data which shows that CMEs cannot be tracked before the S/C reaches 20° and the ability to track them degrades significantly after 90°. For the stopping scenario, an Earth-Sun-Spacecraft separation angle of 60 degrees is considered (though this will need to be refined based on the science requirements)

3.2.4 Subsystem study requirements

These are the main requirements identified within the scope of the study for the major technical domains of the HAGRID mission.

Table 1: List of Subsystem Study Requirements

Reference	Requirement
MISN.1	The mission analysis shall provide details of launcher choices and selection needed for baseline orbital analysis
MISN.2	The orbital manoeuvres shall determine a trajectory starting from spacecraft separation from the launcher to end of life operations. It shall assume injection by launch provider into a 300 km circular low earth orbit.
MISN.3	The spacecraft shall reach operational orbit within 1-1.5 years, in order to reach an observation range of 20-90° with a preferred observational position of 60° from the Earth, along its orbit.
MISN.4	The mission analysis shall determine the best method for interplanetary orbit injection.
MISN.5	The mission analysis shall specify orbital options allowing for both drifting and ‘stopping’ near L4 in the operational orbit.
MISN.6	For each orbital option, mission analysis shall specify the DeltaV needed as well as time to reach the operational orbit.
MISN.7	For each of the drifting and ‘stopping’ cases, the mission analysis shall determine an optimal orbit trajectory.
MISN.8	The mission analysis shall provide DeltaV values to the Propulsion subsystem for each orbit case.
PROP.1	The propulsion system shall select propellant capable of providing the thrust needed to get the HAGRID spacecraft from LEO to operational orbit and end-of-life manoeuvres.
PROP.2	The propulsion system shall determine the amount of propellant needed for each spacecraft configuration.
PROP.3	The propulsion system shall size propellant tanks for spacecraft selection, and select off-the-shelf tanks to contain the propellant as necessary.
PROP.4	The propulsion system shall account for support structure in the subsystem mass budget.
PROP.5	The propulsion system shall specify any auxiliary propulsion systems to be used.
PROP.6	The propulsion system shall be compatible with the AOCS subsystem.
PROP.7	The propulsions system shall calculate the amount of propellant needed for AOCS manoeuvres based on input from the AOCS subsystem.
AOCS.1	The spacecraft AOCS system shall use a three-axis stabilised architecture
AOCS.2	Upon separation of the spacecraft from the upper stage or loss of attitude, the AOCS shall stabilise the spacecraft into SAFE mode within 600 sec
AOCS.3	The initial rate of the spacecraft before commencement of attitude acquisition shall be less than 10 deg./sec. about any axes of the spacecraft and a worst case attitude.
AOCS.4	The mission timeline includes 6 apogee raising burns.
AOCS.5	The spacecraft shall slew 90 deg. before and after each apogee raising burn.
AOCS.6	The spacecraft AOCS system shall satisfy the fine pointing requirements of the payload

Reference	Requirement
AOCS.7	The spacecraft AOCS system shall provide the attitude information to the Communications subsystem for fine pointing of the HGA during ground contact periods
AOCS.8	The spacecraft shall have sufficient propellant to re-acquire SAFE mode pointing 6 times during the mission. Half of these shall be prior to the apogee raising manoeuvres and half when the spacecraft is in the operational orbit
POWR.1	The power subsystem shall provide sufficient electrical power to each subsystem during all stages of the mission in every applicable spacecraft mode
POWR. 2	The power subsystem shall provide sufficient power EOL for general operations with downlink
POWR. 3	The power subsystem shall be able to provide peak power during transfer engine firings
POWE. 4	The power subsystem shall have a secondary power source sufficient for eclipse during transfer and anomalies
COMM.1	The communications subsystem shall provide sufficient link-budget to enable downlink of data volumes recorded by the payload and the spacecraft bus.
COMM.2	The communications subsystem shall select hardware that fit within the fairing of the selected launcher.
COMM.3	The communications subsystem power requirements shall be within the capability of the power subsystem, during all orbits in mission lifetime.
COMM 4	The communications subsystem shall have a bit error rate of 10^{-5} or less
COMM.5	The communications subsystem shall downlink data suitable to predict the direction of a CME 8 hours or more before CME reaches 1 AU distance from Sun.
THER.1	The thermal subsystem shall provide a stable thermal interface to the HI instrument (the payload)
THER.2	The thermal subsystem shall keep subsystem equipment within their non-operational and operational temperature ranges as applicable through all stages of the mission.
THER.3	The thermal subsystem shall specify the location of thermal elements with respect to both the spacecraft configuration and the external environmental factors (e.g. sun-facing panel)
THER.4	The thermal subsystem analysis shall size the area needed for radiators and MLI, as well as allocations for active units (e.g. heaters/thermistors) for each spacecraft case studied.
STRU.1	The structures subsystem shall design a structure that is strong enough to withstand launch loads, while providing a casing for internal spacecraft elements and mounting fixtures for external spacecraft elements, including the HI instrument.
STRU.2	The spacecraft configuration shall allow for stowing the spacecraft in a launch configuration that fits within the selected launcher. (Rockot used as a baseline for this study).
STRU.3	The structures subsystem shall determine the placement of all subsystem elements of the spacecraft for each trade-off case in the study.
STRU.4	The structures subsystem shall be responsible for keeping the CAD model of the spacecraft current within the study, as well as confirming that mass budget estimates are reasonable as seen from the CAD model.

3.3 Programmatic constraints

Contents:

High level list of constraints, design drivers and objectives with explanation.

There are several assumptions, both implicit and explicit, that have been made in the production of this report. While these were necessary in order to provide some framework for the study, they are not necessarily definitive and may prove to be inapplicable in practice. They are noted here to provide the reader with some context for the report that follows.

We have assumed;

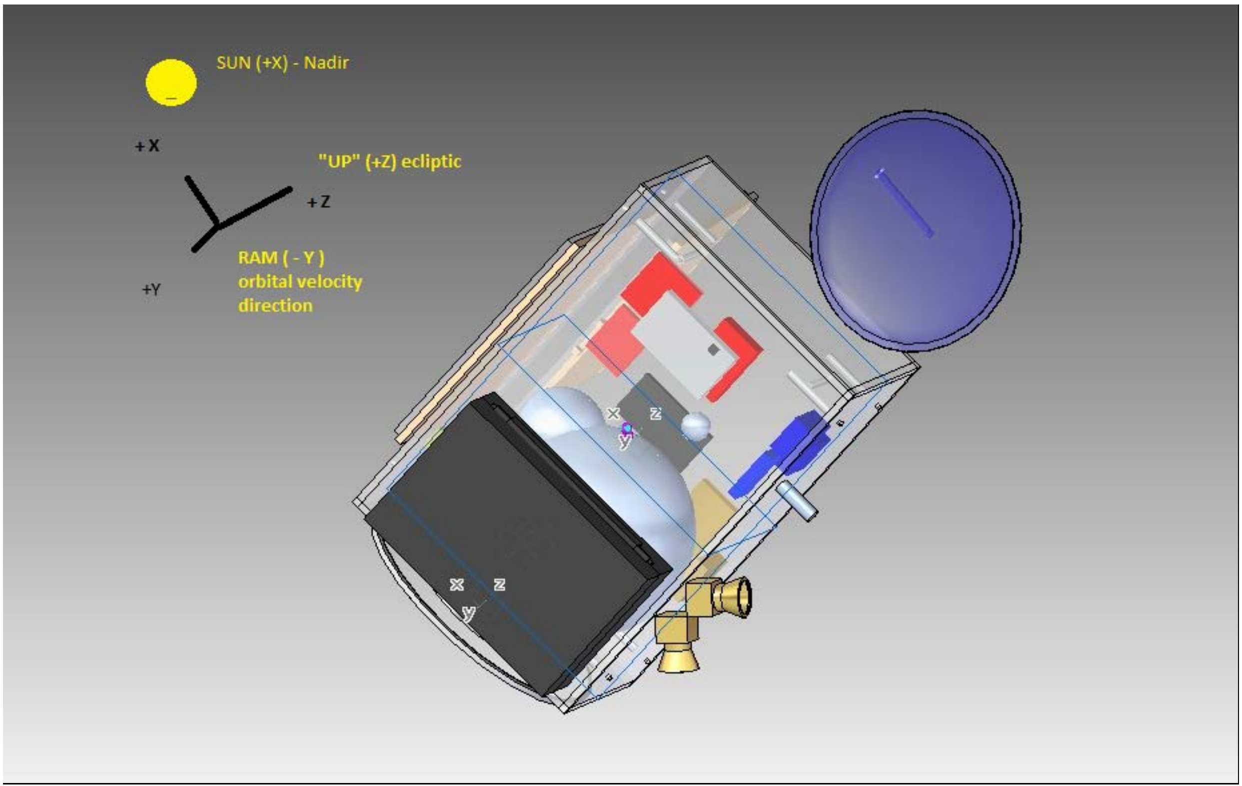
- that this study is a partnership between Astrium, SSBV,the University of Southampton and RALSpace
- that there are no constraints on mission partners
- that the launch date will be four years from the commencement of phase A
- that there are no constraints on procurement
- that there will be no ITAR issues
- that the budgetary margins used are the same as that for ESA studies
- that the technology readiness level (TRL) of subsystems is high
- a mandatory reuse of existing hardware, software and services where applicable (for example the HI instrument and existing ground-stations)
- that the project will adopt a medium policy with respect to risk, since it will be mostly based upon pre-qualified EEE components allowing a proto-flight model to be used for system qualification
- that the project will adhere to ECSS standards
- cleanliness requirements (particulate and molecular) will be the same as for the STEREO/HIs
- that the mission will adhere to planetary protection requirements
- that the mission lifetime is 5 years
- spacecraft autonomy requires the implementation of a safe-mode and data-down link
- that the mission will use a low-cost Rokot launcher

3.4 HAGRID Mission Overview

Mission Objectives	There are 5 main goals: 1. Cruise phase duration ▪ (from launch to start of operation) shall be less than a year; 2. Routine operations ▪ shall last for at least 3.5 years; 3. Instrument Science ▪ The required CME surface brightness sensitivity of the HAGRID imager shall be half that of STEREO/HI; 4. Alert ▪ The alert time provided shall be a minimum of 6 hours, and shall be maximised (although alert time must be tensioned against alert reliability); 5. Cost ▪ The cost of development and operation are low (< £50M).	
Launch	Launcher	Rockot (or similar)
Orbit	Launch injection	LEO (launcher parking orbit)
	Trajectory	Ballistic insertion into Heliospheric orbit
	Drift rate	~20°/year (similar to STEREO)
	Operational Orbit	Earth-leading (Ahead), 'Stopped' at L4 (60° from Sun-Earth Line)
Spacecraft Design	Design lifetime	5 years
	Number of spacecraft	1
	Launch mass (incl. adapter)	1950 kg
	Injected mass	1900 kg
	Payload	• Re-use, as far as possible, of existing Helispheric Imager o 2-telescope, combined field of view imaging the Sun-Earth line from 4° elongation out past the Earth • Accommodation on spacecraft to prevent stray light entering the baffles • 48 summed images per day • ~270 bps (out of 500 bps allocation baseline) • New design of Instrument Control Unit (ICU) needed • Re-latching door needed for instrument in stopping scenario o Previous 1 shot door prevents observation during the cruise phase, as door must be closed during final thruster firing to 'stop' spacecraft at L4
	Propulsion	• Monopropellant Hydrazine system • LISA Pathfinder PRM • 1538 kg (incl. 1275 kg propellant)

		• 1N thrusters (for use in operational orbit once PRM is separated) • 84 kg propellant
	AOCS	• Three-axis stabilized • 12 1N thrusters and 4 reaction wheels • 2 Star trackers, 6 sun sensors and an IMU
	Power	• 28 V bus • GaAs Triple Junction solar cells o 3.4 m^2 and 19.95 kg (composite) • Li-Ion, 7.8 kg (composite)
	Communications (Comms)	• TC, HKTM, Ranging in X-band (1 LGA LEO only, 1 HGA 1 m diameter) • 56.3 W RF power provided by 120 W EPC
	Structure	• Composite (spacecraft + LISA Pathfinder PRM) • Panel mounted solar arrays • Deployable HGA • HI instrument panel mounted with door mechanism
	Thermal	• Passive control • Thermal isolation of instrument, and operational and non-operational component temperatures
Operations	Ground Station	RAL Chilbolton with X-band upgrade
	Number of ground stations needed for coverage	3 minimum
	Mission/Science operations centre (MOC/SOC)	TBD (not within the scope of this study)

4. Space Segment Design



The following section describes the design options and choices for the spacecraft.

4.1 Mission Analysis

Mission Analysis Contents:
- Mission orbital design strategy
- Launch and initial orbit injection
- Drift and operational phase
- Drifting vs. stopping

4.1.1 Mission design strategy

Using the STEREO mission as a baseline, the ideal spacecraft-Sun-Earth angle for observing the Sun-Earth line from the Heliospheric Imager, is between 20 and 90 degrees Ahead or Behind the Earth, along its orbit. This puts the spacecraft on an elliptical orbit with a semi-major axis close to 1AU, outside the Earth's sphere of influence.

In order to exclude complex orbits (e.g. horse-shoe) which could provide low DeltaV manoeuvres with long timescales, it was decided that the HAGRID spacecraft should reach operational orbit within 1 year of launch. This reduced the orbit options to a highly elliptical escape trajectory with or without a lunar assist. As the semi-major axis is close to 1AU, the relative decrease or increase of orbital velocity when leaving the Earth's sphere of influence can be used to put the spacecraft on a drifting orbit that is ahead or behind the Earth respectively, see Figure 1.

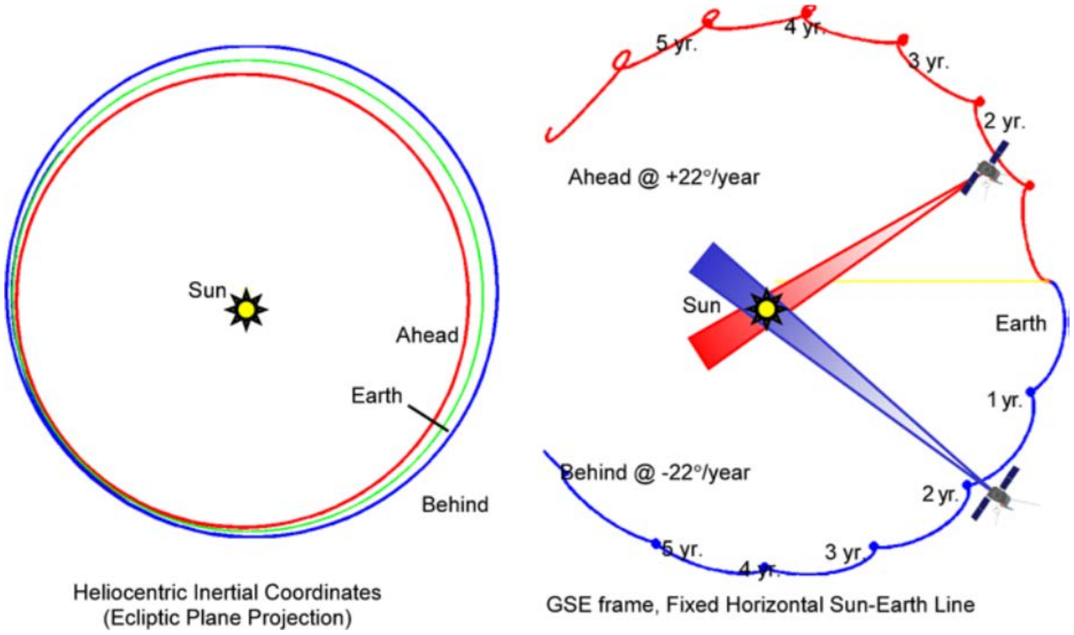


Figure 1 - A diagram of the Ahead and Behind orbits as compared with the Earth's orbit of about 1AU (left). A drawing of the spacecraft movement with respect to a fixed Sun-Earth line (right).

Once the spacecraft is in a drifting heliocentric orbit, for initial observations, it needs to drift to a point at which the spacecraft-Sun-Earth angle is at least 20 degrees. After 20 degrees, there is an option of stopping the spacecraft at a "sweet spot" of 30-60 degrees or letting the spacecraft continue to drift. The stopping case requires more propellant, which increases the launch mass of the spacecraft significantly, but provides continuous operation in an optimal location. This scenario also provides an option to continue observations if resources are available after the initial operational lifetime of the spacecraft. The drift case will allow for a lighter spacecraft, but will have a limited operational lifetime in optimal viewing angles and reduced telemetry rates at higher angles. A block diagram showing a time-line for the HAGRID mission design is presented in Figure 2

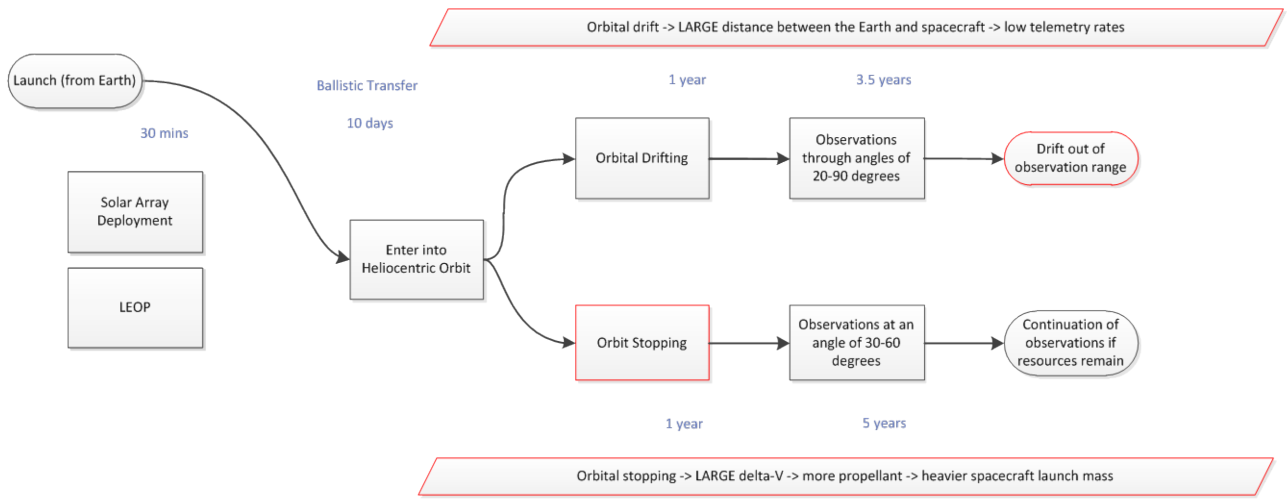


Figure 2 - Block diagram of the orbital manoeuvre options for the HAGRID mission.

The assumptions used to select the preferred orbital manoeuvres were:

- Rockot launcher
- Minimum value of 20 degrees should be reached within 1 year
- Time spent between 20-90 degrees should be at least 3.5 years

The Rockot launcher was assumed for a baseline, as a small, low cost launch option, but other similar options are listed in Table 2.

Table 2 - Low cost launch options for the HAGRID mission, including the Rockot used for analysis [International reference guide to space launch systems (2004)].

Launcher	Rockot	PSLV	Vega
Mass to orbit [kg]	1950 kg to LEO	3700 kg to LEO	1395 kg to SSO
Cost [USD]	$12-15 M	$15-17 M	€25 M
Launch Provider	Russia	India	Europe
Launch Site	Plesetsk	Satish Dhawan	CSG (Kourou)

With the assumptions defined, the following scenarios were studied:

- Launch and initial orbit injection
- Drift and operation phase
  (drift rate vs. time to operational orbit and duration in optimal 20-90 degree viewing angles)
- Drifting or stopping orbit vs. spacecraft mass

4.1.2 Launch and initial orbit injection

With the assumed Rockot launcher, the injected mass to LEO (assumed to be a 300km circular orbit) is 1950kg (including the launch adaptor). A comparison with the Lisa-Pathfinder launch condition was used, with 1900kg injected to an orbit with 200km perigee and 900 km apogee. This analysis was done outside of the CDF and used as a baseline for the drift and operational phase work completed in Session 3.

DeltaV budgets:

300km circular orbit:

- Apogee raising manoeuvres to reach apogee of circa 1.4 million km: 3270 m/s (assuming 3% loss for the apogee raising sequence)
- Dispersion correction/navigation: 30 m/s
- No maintenance manoeuvres
- No deep space manoeuvres

200km * 900km orbit:

- Apogee raising manoeuvres to reach apogee of circa 1.4 million km: 3010 m/s (assuming 3% loss for the apogee raising sequence)
- Dispersion correction/navigation: 30 m/s
- No maintenance manoeuvres
- No deep space manoeuvres

Transferring from an Earth-centred orbit to a Sun-centred orbit can be reached via a low energy Earth escape orbit. The V_∞ should lie at or below approximately 1km/sec. Several methods exist to reach such an escape condition:

1. Injection to the escape orbit (either by a launcher or by satellite propulsion system after elliptical orbit injection and apogee raising)
2. Injection to a Lunar crossing orbit (either by a launcher or by satellite propulsion system after elliptical orbit injection and apogee raising) – followed by a Lunar gravity assist
3. Injection to a high elliptical orbit (either by a launcher or by satellite propulsion system after lower elliptical orbit injection and apogee raising). This orbit will have the property that it can achieve a ballistic escape via the action of third body (ie the Sun) perturbation. It is possible to achieve V_∞ in the region of 1km/sec via such a method.

Using method 2 or 3 requires less DeltaV, or launch energy, than method 1. Method 2 is the most efficient, although the difference from method 3 is small (<50 m/s). Operationally it can be difficult to set up a Lunar gravity assist, as the spacecraft generally needs to wait in an intermediate elliptical orbit before raising apogee to cross the Moon's orbit and execute the fly-by. Planning strategies that are robust to a missed apogee raising firing can be complex and also attract a DeltaV penalty that detracts from the gain compared with method 3. Method 3 also needs to be robust to missed firings but can be less sensitive than Method 2.

Therefore method 3 is used in the following analyses. An example of such a ballistic escape trajectory can be seen in Figure 3.

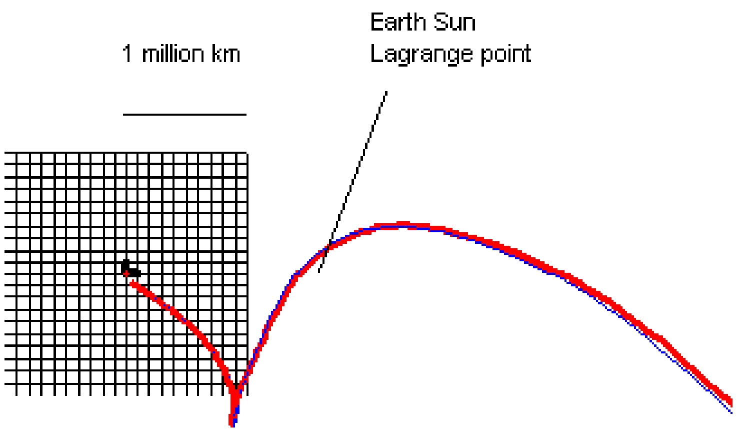


Figure 3 - Example of ballistic escape from Earth achieving a V∞ of circa 1km/sec

4.1.3 Drift and operational phase

The objective is to achieve a V∞ on departing Earth such that an increase on the spacecraft-Sun-Earth angle of 20 deg in 1 year and 90 deg in 3.5 years can be achieved without the need for deep space manoeuvres (which would add significantly to the DeltaV budget).

4.1.3.1 Initial Analysis

During the CDF session 3, an initial analysis was done looking at Sun-centred circular orbits with a semi-major axis of 0.94 to 1.02 AU, to show how far the spacecraft would drift over 4 years, using different drift rates as shown in Table 3.

Table 3 - Comparison of drift rates for a spacecraft at a radial distance of 0.94 – 1.02 AU (1.41E11 – 1.53E11 m).

r_sat [m]	period_sat [sec]	period_sat [days]	degrees/day	degrees/year	drift/year	drift 2 years	drift 3 years	drift 4 years	orbital velocity	insertion vel [m/s	Delta-V [m/s]	prop mass inj	sc mass inj
1.41E+11	2.89E+07	334.228	0.928	338.870	-21.130	-42.261	-63.391	-84.522	30679.357	30679.357	0.000		
1.42E+11	2.92E+07	337.789	0.938	342.481	-17.519	-35.038	-52.557	-70.076	30571.141	30679.357	108.216		
1.43E+11	2.95E+07	341.364	0.948	346.105	-13.895	-27.790	-41.685	-55.580	30464.061	30679.357	215.296		
1.44E+11	2.98E+07	344.951	0.958	349.742	-10.258	-20.516	-30.775	-41.033	30358.099	30679.357	321.258		
1.45E+11	3.01E+07	348.550	0.968	353.391	-6.609	-13.217	-19.826	-26.435	30253.234	30679.357	426.123		
1.46E+11	3.04E+07	352.162	0.978	357.053	-2.947	-5.893	-8.840	-11.787	30149.449	30679.357	529.908		
											750	1400	500
1.47E+11	3.07E+07	355.786	0.988	360.728	0.728	1.456	2.184	2.912	30046.725	29451.682	595.043	1370	530
1.48E+11	3.11E+07	359.423	0.998	364.415	4.415	8.830	13.245	17.660	29945.044	29451.682	493.362	1352	548
1.49E+11	3.14E+07	363.072	1.009	368.115	8.115	16.229	24.344	32.459	29844.388	29451.682	392.706	1335	565
1.50E+11	3.17E+07	366.733	1.019	371.827	11.827	23.653	35.480	47.307	29744.740	29451.682	293.058	1315	585
1.51E+11	3.20E+07	370.407	1.029	375.551	15.551	31.102	46.653	62.205	29646.084	29451.682	194.402	1296	604
1.52E+11	3.23E+07	374.092	1.039	379.288	19.288	38.576	57.864	77.152	29548.403	29451.682	96.721	1276	624
1.53E+11	3.26E+07	377.790	1.049	383.037	23.037	46.074	69.111	92.148	29451.682	29451.682	0.000	1257	643

4.1.3.2 Detailed Analysis

Further analysis was done outside of the CDF session to optimize the drift rate versus the time taken to reach operational orbit. Figure 4 - Figure 10 illustrate a number of examples, showing the evolution of solar longitude difference (Earth to satellite distance) and range to the Sun (provided by Astrium). Examples of both the ahead and behind orbits are shown, as either is possible with this escape strategy. DeltaVs are the same in each case.

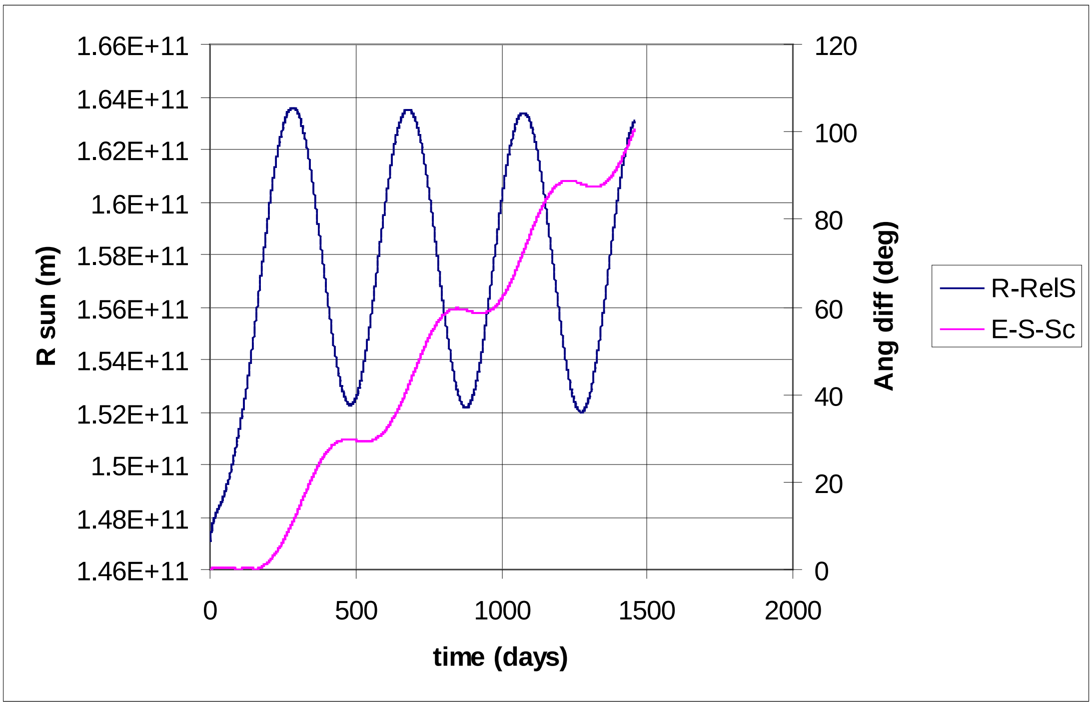


Figure 4 - Behind with high drift rate reaching Ls difference of 20 deg in 0.9 years and 90 deg in 3.8 years provided by Astrium. R-RelS is the distance (radius) from the Sun in the Sun frame (blue) and E-S-Sc is the Earth-Sun-Spacecraft angle (pink).

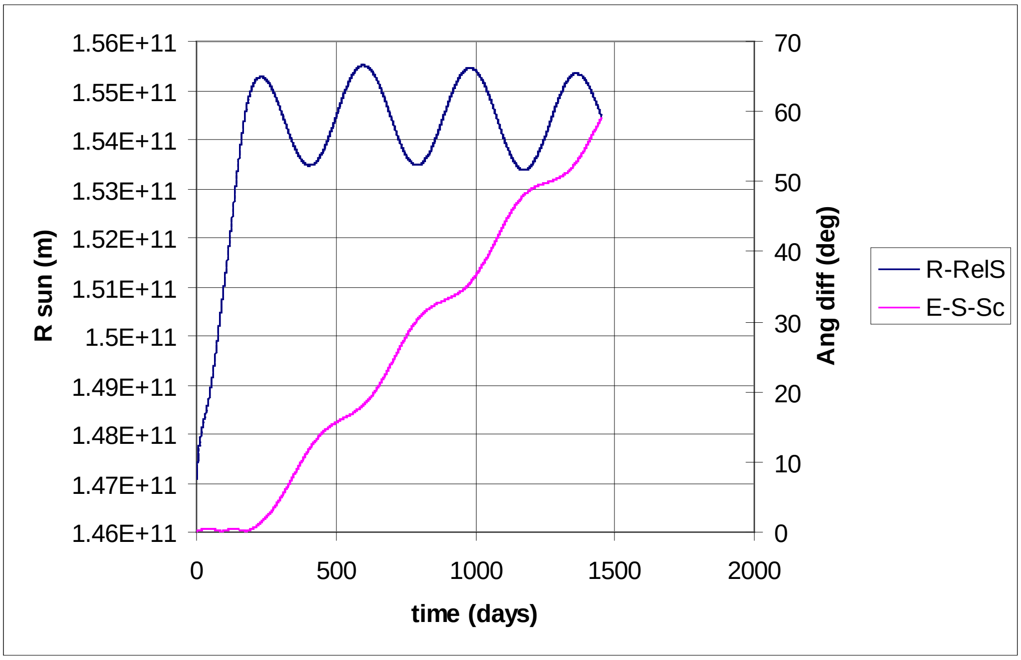


Figure 5 - Behind with low drift rate reaching Ls difference of 20 deg in 0.9 years and 90 deg in 3.7 years provided by Astrium. R-RelS is the distance (radius) from the Sun in the Sun frame (blue) and E-S-Sc is the Earth-Sun-Spacecraft angle (pink).

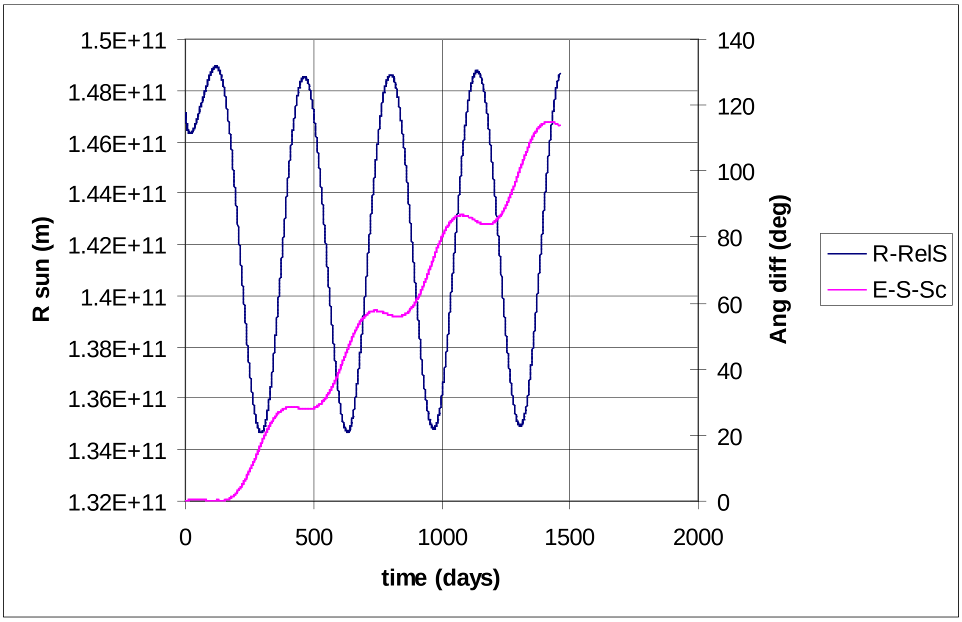


Figure 6 - Ahead with high drift rate reaching a longitude difference of 20 deg in 0.8 years and 90 deg in 3.7 years (provided by Astrium). R-RelS is the distance (radius) from the Sun in the Sun frame (blue) and E-S-Sc is the Earth-Sun-Spacecraft angle (pink).

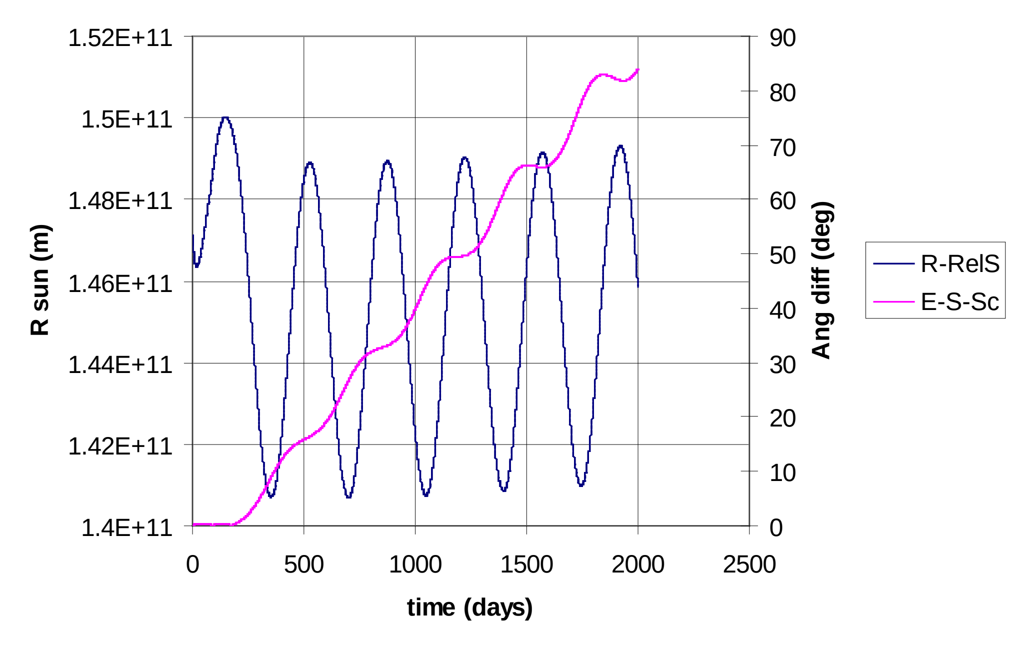


Figure 7 - Ahead with low drift rate reaching a longitude difference of 20 deg in 1.6 years and 90 deg in 5.9 years (provided by Astrium). R-RelS is the distance (radius) from the Sun in the Sun frame (blue) and E-S-Sc is the Earth-Sun-Spacecraft angle (pink).

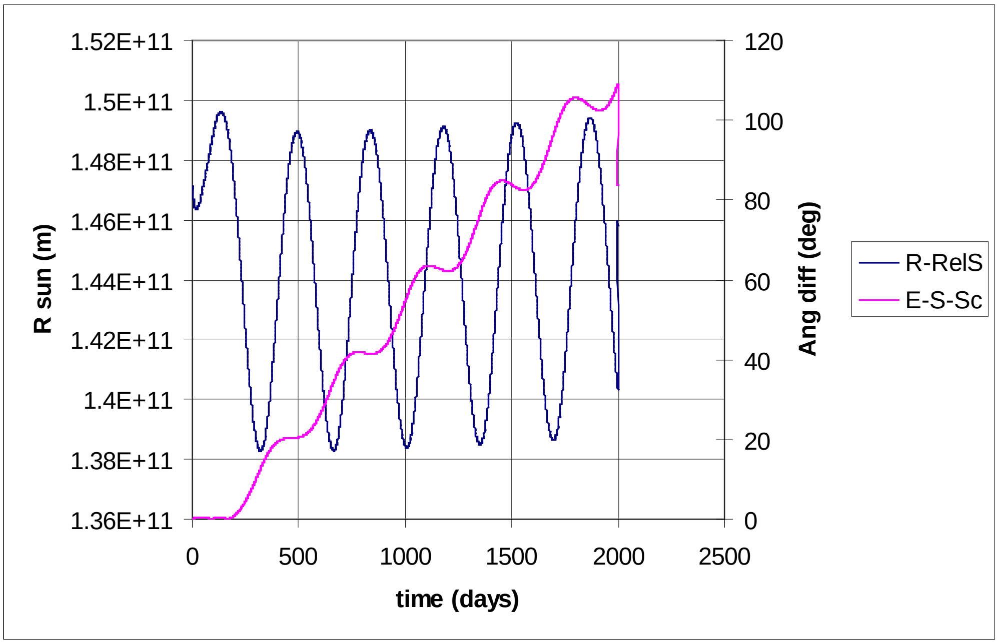


Figure 8 - Ahead with intermediate drift rate reaching a longitude difference of 20 deg in 1.05 years and 90 deg in 4.45 years (provided by Astrium). R-RelS is the distance (radius) from the Sun in the Sun frame (blue) and E-S-Sc is the Earth-Sun-Spacecraft angle (pink).

The orbital solution chosen for the HAGRID mission was the Earth ahead case, with an intermediate drift rate of about 19 degrees per year, reaching 20 degree in 1.05 years and drifting past 90 degrees in 4.45 years. The Ahead case was selected to reduce the impact of "spacecraft ramming", micrometeorites hitting the HI instrument as experienced by the STEREO-B spacecraft that affected image stability.

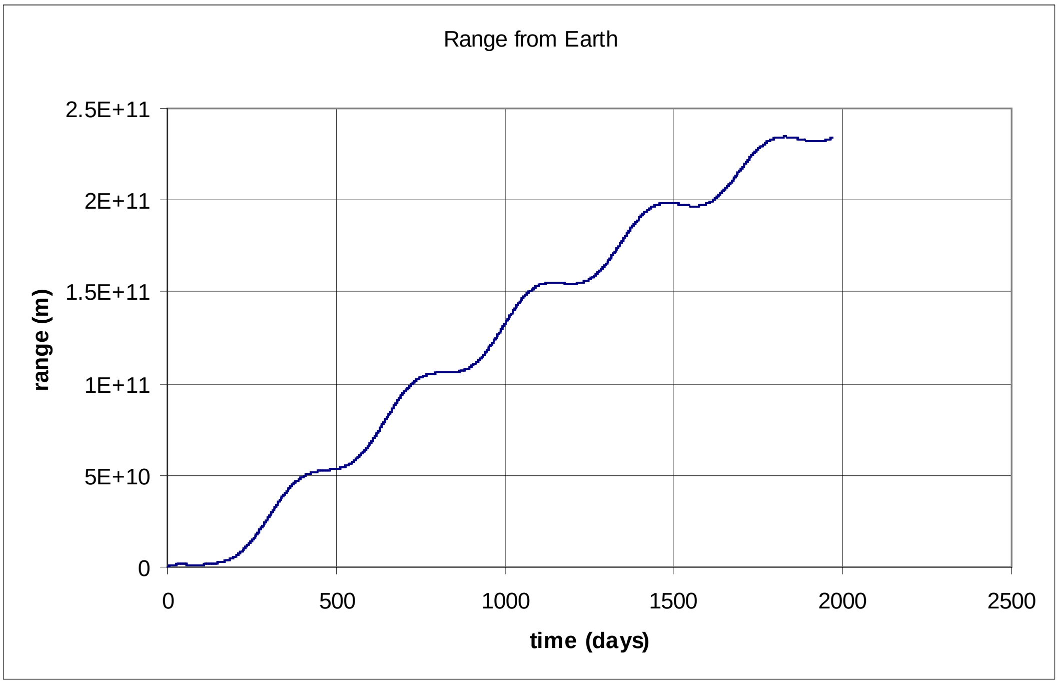


Figure 9 - Ahead distance to Earth evolution with intermediate drift rate reaching a longitude difference of 20 deg in 1.05 years and 90 deg in 4.45 years provided by Astrium. E-S-Sc is the Earth-Sun-Spacecraft angle (blue).

The last solution essentially fills the mission requirements, missing slightly the 3.5 years operational period. A slight compromise on time to start of mission would allow fulfilment of the 3.5 years.

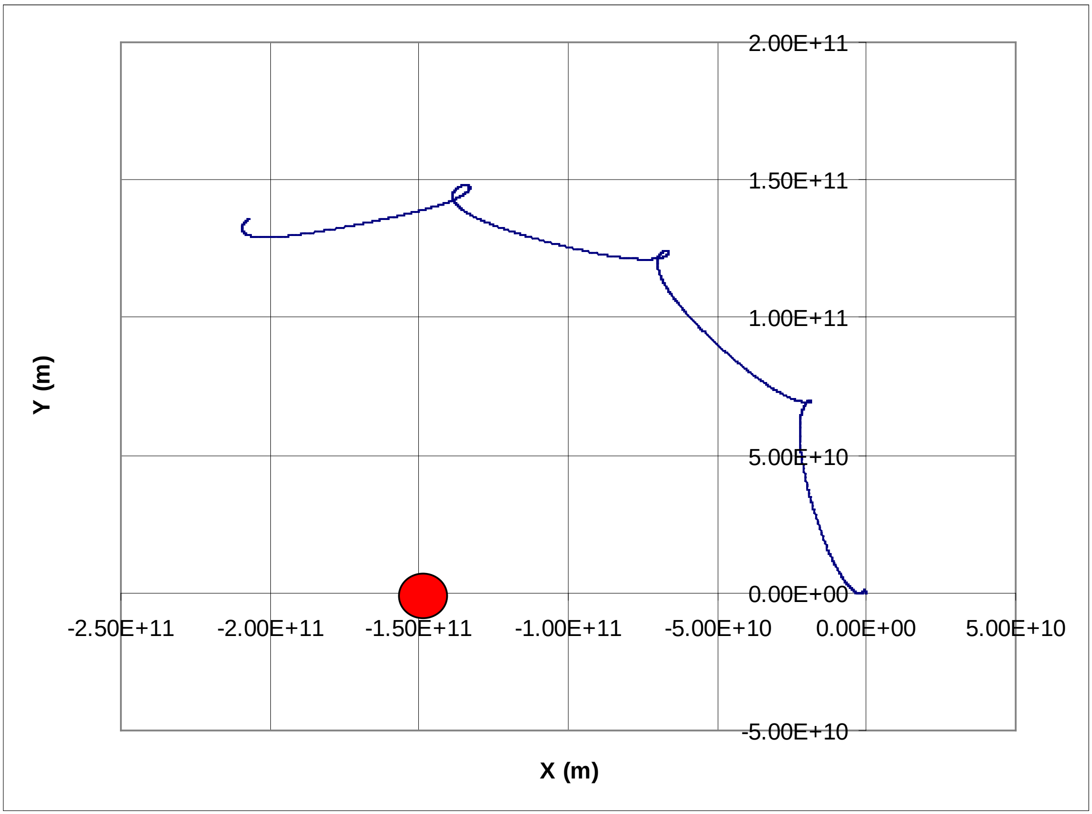


Figure 10 - Illustration of an ahead orbit seen in a fixed Sun-Earth line frame. Earth lies at 0, 0 and Sun at -1.5e11m. In the axis frame of this figure an Earth-leading orbit is anticlockwise (provided by Astrium).

Drift away orbits that fulfil the mission requirements can be found that minimise the mission DeltaV needed for a spacecraft injected to a low energy orbit. These missions avoid the need for deep space manoeuvres.

Total mission DeltaV is typically 3000 to 3300 m/s depending on the details of the injection orbit.

4.1.4 Drifting vs. Stopping

Due to the complexity of the ballistic transfer, work was done offline to determine the deltaV needed to stop the spacecraft at 20 degrees. While this is the minimum useable angle for such a mission, the values calculated differ little for scenarios in which the spacecraft is stopped at a greater angle necessary to optimise the mission geometry (40-90 degrees). The initial orbit parameters used for the drifting orbits used a weak-stability boundary (WSB) transfer. This allowed multiple drift-rates to be achieved with little impact on DeltaV. For the “stopping” case, there is actually a choice between a standard Deep Space Manoeuvre (DSM), a direct transfer, and a WSB transfer. Figure 11 shows the different “stopping” transfer methods comparing DeltaV with time to 20 degrees from the Sun-Earth line. A 560 m/s DeltaV is required for the direct method.

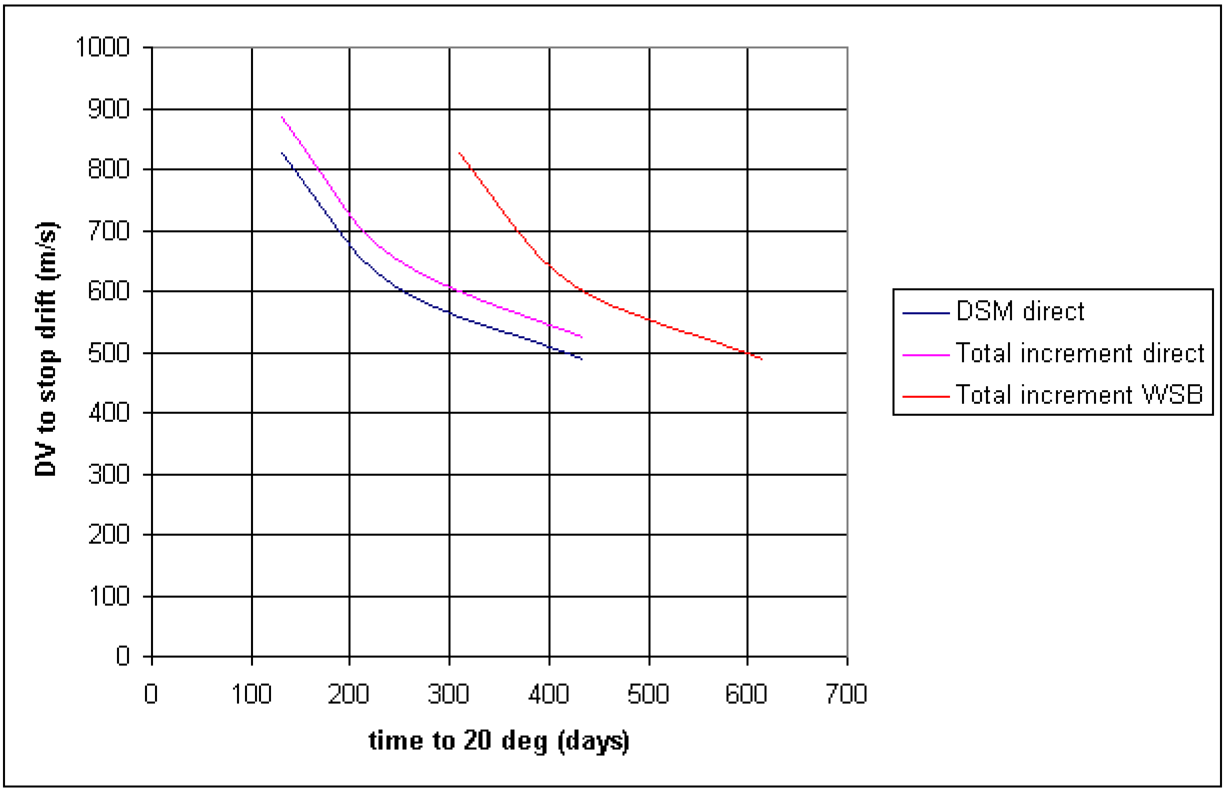


Figure 11 - Comparison of stopping orbit DeltaV and time to 20 degrees from teh Sun-Earth line (provided by Astrium).

Further analysis showed that a perigee DeltaV increment could be excluded from the “stopping” DeltaV and included in the main escape DeltaV. While the perigee DeltaV increment only reduces the DeltaV by 39.4 m/s over 1 year, it reduced the “stopping” DeltaV to 520.4 m/s.

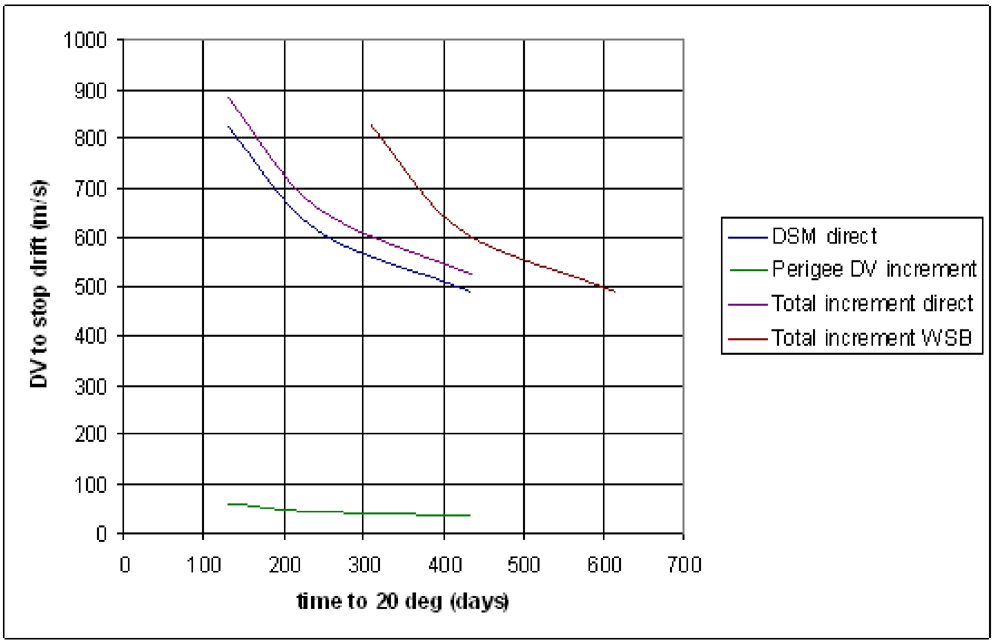


Figure 12 - Comparison of stopping orbit DeltaV and time to 20 degrees from the Sun-Earth line, including the perigee DeltaV increment, which was found to be excluded from the "Stopping" DeltaV (provided by Astrium).

4.1.5 Summary

Using the assumption of a Rockot launcher injecting 1900kg into an elliptical Low Earth Orbit (taken from LISA Pathfinder), analysis was done on the method to transfer the spacecraft from Earth orbit to a Sun orbit. The ballistic transfer method was chosen, as it allows flexibility in launch windows, while providing the needed escape velocity to put the spacecraft in operational orbit with a semi-major axis of 0.94 to 1.02 AU (for drift ahead and behind the Earth respectively)

The drift-rates possible between 0.94 and 1.02 AU give an average maximum drift of +/- 22 degrees per year. Further analysis showed that a drift-rate of 19 degrees per year was preferred to minimise the time to 20 degrees, while increasing the amount of time between 20 and 90 degrees for the drifting orbit case. This used the ahead spacecraft orbit, as that was selected to reduce AOCS corrections needed due to “spacecraft ramming” as experienced by the STEREO-B spacecraft.

The ‘drifting’ orbit case reaches the 20 degree location for operations in 1.05 years and allowed 3.5 years between 20 and 90 degrees. This is similar to the requirements for start and duration of operations. The ‘stopping’ orbit case requires 520.4 m/s of extra DeltaV. In extending the initial requirement of reaching operational orbit within 1 year to 2 years, a preferred operational observation location of 40 degrees can be reached for the stopping case.

Due to its increasing range from the Earth, the drifting spacecraft orbit will considerably increase free space losses in the link budget, resulting in a lower telemetry data rate or an increase in power requirements as the spacecraft moves towards 90 degrees. These trade-offs are considered in the Communications and Propulsion sections respectively.

4.2 Payload

Payload Contents:

- Heliospheric Imager
  - Description
  - Performance
  - Interface Requirements
- Critical Items
  - Requirement for an Instrument Control Unit (ICU)
  - Requirement for a Re-Latching Door Mechanism
  - Telemetry Data Rate Considerations
- TRL of the HI Instrument
- Open Issues and Future Work

4.2.1 Heliospheric Imager

The HAGRID study is based on the re-use, as far as possible, of the existing design (i.e. build-to-print) of the Heliospheric Imager (HI) instruments currently operating aboard the NASA STEREO mission (System Requirement SR-05).

The number of spacecraft is limited to one (SR-02) and the baseline is that the payload consists of single instrument – the HI wide-field imaging photometer (SR-05).

At the beginning of the study, the issue of whether the orbital configuration should be Earth-leading (as for STEREO-A) or Earth-trailing (as for STEREO-B) was considered.

- The Earth-leading configuration was selected for the following reasons:
- Both configurations are equally suitable for imaging and tracking of CMEs propagating along the Earth-Sun line – neither has any particular advantage in this respect
- The Earth-leading configuration allow the imaging and tracking of CIRs prior to their impact with Earth
- The Earth-leading configuration minimises the technical problem of micro-meteorite impacts on the instrument or spacecraft disturbing the pointing attitude of the cameras and causing “jitter” in the pointing of the images, as has been experienced with STEREO-B (Brown et al., 2009, Solar Physics 254, 185.)

Note that the above would need to be re-visited if the final payload includes a secondary instrument such as either a solar-disk imager (*e.g*. an EUV imager) which would clearly need to image the part of the disk which is about to rotate into view of the Earth, or in the case of an in-situ particle monitor which would detect CIRs prior to their impact with Earth.

4.2.1.1 Description

The basic design concept of the HI instrument aboard STEREO is shown in Figure 13 (Eyles et al., 2009, Solar Physics 254, 387.). The instrument is essentially a box shape, of major dimension about 800 mm. A door was used to protect the optical and baffle systems from contamination during ground operations, launch and the initial cruise phase activities. The door is a one-shot system – it is opened once during instrument commissioning and remains open thereafter. The two telescope/camera systems, HI-1 and HI-2, are housed deep within a baffle system as shown in Figure 13. The direction to the Sun is indicated – the Sun remains below the vanes of the forward baffle system. The detectors are CCDs (charge-coupled detectors), that are passively cooled by radiators facing space in the anti-Sunward direction.

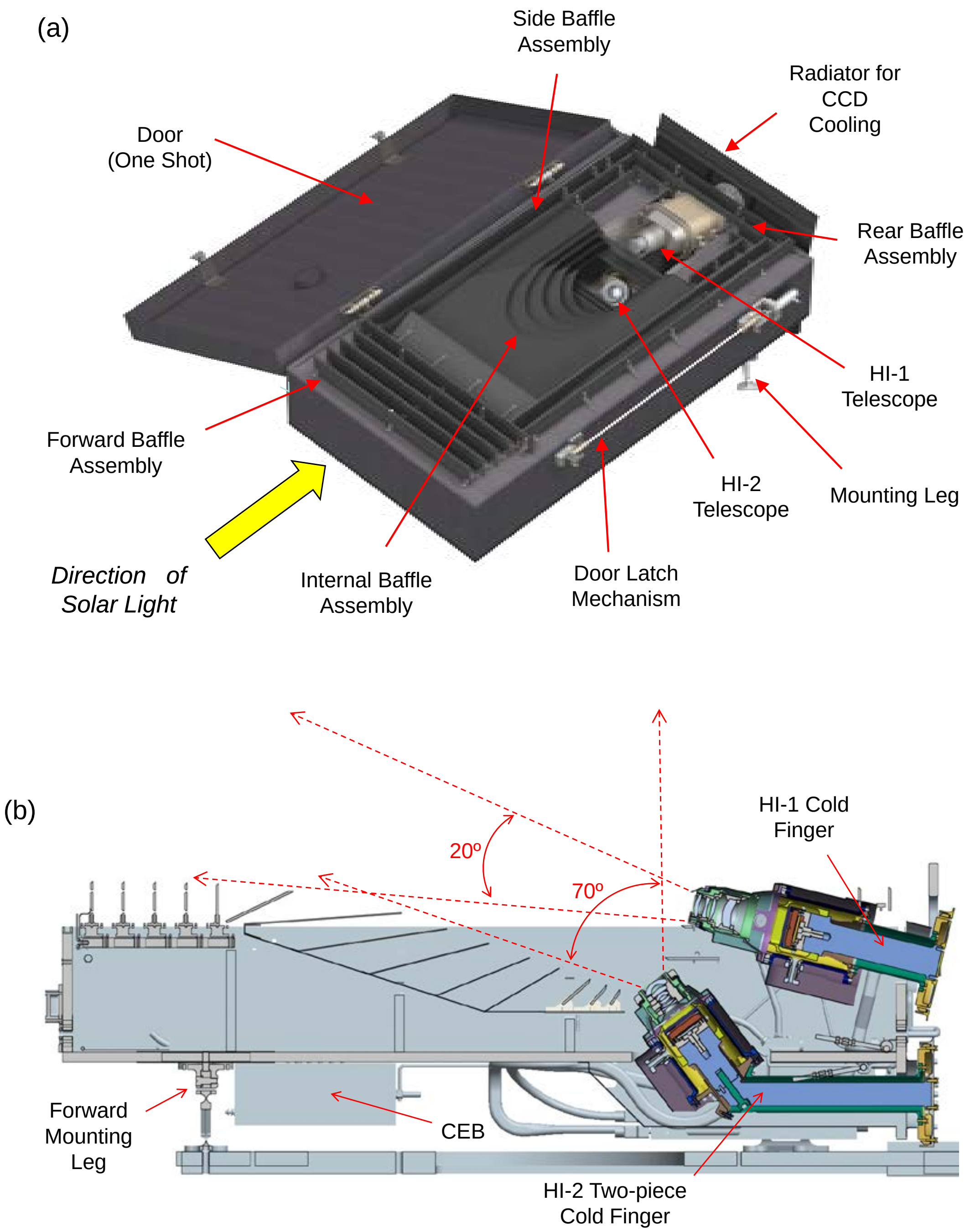


Figure 13 - (a) The Heliospheric Imager design concept. (b) A cross-sectional view through the instrument, showing the fields-of-view of the two telescopes.

The performance specifications for the STEREO HI instrument are listed in Table 4. The HI-1 and HI-2 telescope boresights are directed at angles of 13.65 and 53.35 degrees from the principal axis of the instrument, which in turn is tilted upwards by approximately 20 arcmin to ensure that the Sun is sufficiently below the forward baffles horizon. Thus, the two optical axes are nominally set to 14.0 and 53.7 degrees from the Sun, in the ecliptic plane, with fields of view of 20 and 70 degrees, respectively. This provides an overlap in solar elongation angle of about 5 degrees between the two fields of view, with complete coverage along the Sun-Earth line from 4.0 to 88.7 degrees elongation.

Table 4 - Performance specifications of the HI instruments.

	HI-1	HI-2
Direction of centre of field of view from Sun centre	14.0 degrees	53.7 degrees
Angular field-of-view	20 degrees	70 degrees
Angular range	4 – 24 degrees	18.7 – 88.7 degrees
	(15 – 90 R_o)	(70 – 330 R_o)
CCD pixel size	35 arcsec	2 arcmin
Image array (2x2 binning) *	1024 x 1024	1024 x 1024
Image bin size *	70 arcsec	4 arcmin
Spectral band-pass	630 – 730 nm	400 – 1000 nm
Exposure time *	40 s	50 s
Exposures per summed image sequence *	30	99
Summed image cadence *	40 min	2 hr
Brightness sensitivity (B_o = solar disc)	3×10^{-15} B_o	3×10^{-16} B_o
Stray-light rejection (outer edge of field)	3×10^{-13} B_o	10^{-14} B_o

* These are the values used for STEREO and will not be the same for HAGRID (see below)

Both telescopes are designed to image visible light. In the case of HI-2, the camera is designed to have as wide a spectral response as possible in order to maximise the weak coronal signal at large solar elongations. In the case of HI-1, the coronal signal is much stronger so it is possible to restrict the pass-band with a filter in order to ease the optical design of the wide-angle optics. In the case of STEREO, the HI-1 filter band-pass was chosen to approximately match that of COR-2, the outermost of the Sun-pointing coronagraphs in the SECCHI instrument suite but this pass-band will be equally suitable for HAGRID.

The HI detectors are CCDs with 2048 x 2048 pixels, where each pixel has a size of 13.5 x 13.5 µm. On STEREO they are usually binned on board to 1024 x 1024 bins, resulting in image bin angular sizes of 70 arcsec and 4 arcmin, for HI-1 and HI-2 respectively. In order to obtain sufficient statistical accuracy, long-duration exposures are required. However, the rate of cosmic ray hits would compromise such images if they were taken as single exposures. Thus, short exposures are taken and cleaned of cosmic rays on board, and a number of such exposures are then summed to produce an image to be down-linked (see Table 4). The short exposures must be summed on board and down-linked as summed images in order to keep the overall telemetry requirements reasonable.

For each telescope, Table 4 lists the nominal exposure time and number of exposures per summed image sequence, as used for STEREO. With a CCD line transfer rate of rather more than 2 ms per line and an image clear sequence of 124 μs per line, the readout time of each exposure is approximately 4.8 s, including overheads. Because of mechanical accommodation constraints, the cameras do not have shutters, so the fact that the readout time is not insignificant compared with the exposure time results in the images being smeared during the readout process. However, this image smearing due to the shutter-less operation of the cameras can be corrected fairly easily during subsequent data processing. For STEREO this is done on the ground, but for HAGRID we envisage that this would be done on board.

4.2.1.2 Performance

Table 4 gives the Brightness Sensitivity and Stray-light Rejection of the STEREO-HI instruments. These values are fully compliant with the requirements of HAGRID.

In fact, since the surface brightness sensitivity requirement for HAGRID is a factor of two lower than for STEREO-HI (User Requirement UR-07), there is scope for modifying some of the operational requirements of HAGRID, in particular with respect to image cadence:

- The HI-2 summed-image cadence should be reduced from 2 hr to 1 hr for the following reasons:
  - 2 hr represents a significant "quantum step" in terms of the availability of images for generating plots of elongation versus time (*J-maps)* and satisfying the Alert Time requirement of 6 hr (UR-11)
  - For a fast-moving CME there is significant image-motion blurring during a 2-hr image exposure
  - Reducing the total exposure time in the summed-exposure image by a factor ~ 2 will reduce the sensitivity by a factor ~ 1.4.

- On the other hand, the HI-1 summed-image cadence is relaxed from 40 min to 1 hr for HAGRID:
  - Provides images at adequate cadence for constructing *J-maps*
  - Reduces the telemetry requirements.

The net effect of the changes is neutral in terms of telemetry requirements since the total number of images per 24 hr is 48 in both cases.

4.2.1.3 Interface Requirements

The principle mechanical properties and power requirements of the HI instrument and associated Instrument Control Unit (ICU) are summarised in Table 5.

Of particular importance is that the HI Instrument shall be accommodated on the spacecraft in such a way that there are no spacecraft subsystems or other appendages above the horizontal plane defined by the uppermost edges of the forward, side and rear baffles (see Figure 13). This is so that scattered solar light from nearby objects cannot destroy the solar stray-light rejection performance of the instrument.

Table 5 - Mechanical properties and power requirements of the payload (HI and ICU).

HI Instrument	Dimensions	840 mm x 560 mm x 220 mm
	Mass	17.3 kg Includes 1 kg allowance for re-latching Door Mechanism
	Spacecraft I/F	Semi-kinematic mount using 3 Ti6Al4V flexible mounting feet. Also provides thermal isolation.
	Field-of-View I/F	**No spacecraft sub-systems or other appendages to appear above plane defined by perimeter baffles.**
Instrument Control Unit (ICU)	Dimensions	250 mm x 150 mm x 60 mm
	Mass	1.8 kg
	Spacecraft I/F	Conventional panel-mounted box
Combined Payload	Power (peak)	29 W
	Power (nominal)	25 W
	Power (standby)	18 W

The principle temperature requirements of the HI instrument and ICU are summarised in Table 6.

Of particular importance here is that the operating temperature of the camera CCD detectors must be below –70ºC in order to minimise detector dark current, hot pixels and other effects of radiation damage in the interplanetary orbit.
This requirement was fairly easily satisfied for the STEREO-HI cameras by mounting the CCDs on conducting cold-fingers which are coupled to thermal radiators facing deep-space in the anti-Sunward direction (see Figure 13), and is not seen to be a major design driver in the case of HAGRID.

Table 6 - Temperature requirements of the payload (HI and ICU).

HI Instrument	Operating temperature (max)	+30 ºC
	Operating temperature (min)	–35 ºC
	Standby temperature (max)	+60 ºC
	Standby temperature (min)	–50 ºC
	Detector operating temp (max)	–70 ºC
	Detector operating temp (min)	–100 ºC
	Detector standby temperature (max)	+100 ºC
	Detector standby temperature (min)	–120 ºC
Instrument Control Unit (ICU)	Operating temperature (max)	+50 ºC
	Operating temperature (min)	–30 ºC
	Standby temperature (max)	+60 ºC
	Standby temperature (min)	–40 ºC

The principle attitude control requirements of the HI Telescopes are summarised in Table 7.

The following points are noted:

- The requirements for Absolute Pointing Error (APE) and Pointing Drift Error (PDE) are relatively relaxed because the precise pointing direction for each HI camera image can be obtained from the background stars, i.e. the images are essentially self-calibrating
- The most demanding requirement for APE and PDE is for rotation about the y-axis. This corresponds to the *pitch* of the HI instrument, *i.e*. tilting the camera bore-sights towards or away from Sun-centre. This affects the solar stray-light rejection of the Forward Baffles. An accuracy of ± 0.*25 deg (at 3σ) is adequate to ensure that the stray-light rejection requirement is met (Halain et al., 2011, Solar Physics 271, 197*).
- The Relative Pointing Error (RPE) requirements are much more stringent since they represent instability in telescope pointing attitudes while summed camera exposures are being accumulated. Change in attitude during the accumulation period would result in blurring of star images making the attitude calibration of the images and subsequent subtraction of the F-Corona background difficult or impossible.
- The RPE requirement (3σ) is set at 36 arcsec (0.01 deg) which corresponds to 1 CCD pixel for the HI-1 camera, thereby assuring that the additional contribution to the optics PSF due to pointing instability is negligible.
- The RPE time was specified as 7200 s based on the 2 hr image cadence for HI-2 on STEREO. Subsequently, during the study it was decided to reduce the HI-2 cadence for HAGRID to 1 hr, so this value should be reviewed.

Table 7 - Attitude control requirements of the HI telescopes.

Absolute Pointing Error x (3σ)	1.00 deg
Absolute Pointing Error y (3σ)	0.25 deg
Absolute Pointing Error z (3σ)	1.00 deg
Pointing Drift Error x (3σ)	1.00 deg
Pointing Drift Error y (3σ)	0.25 deg
Pointing Drift Error z (3σ)	1.00 deg
Relative Pointing Error x (3σ)	36 arcsec
Relative Pointing Error y (3σ)	36 arcsec
Relative Pointing Error z (3σ)	36 arcsec
Relative Pointing Error time	7200 s (TBD)

4.2.2 Critical Items

4.2.2.1 Requirement for an Instrument Control Unit (ICU)

The HI instrument on STEREO has very few direct electrical interfaces with the spacecraft – only Pyro Bus Power (for operation of the HOP Actuator in order to release the HI Door) and four direct Temperature Monitors. The majority of primary power, control, data acquisition and monitoring functionality for the instrument is provided by the SECCHI Electronics Box (SEB).

An Instrument Control Unit (ICU) must be developed to replace the functionality provided by the SEB on STEREO.

Figure 14 shows the interfaces between the HAGRID HI instrument, essentially similar to STEREO-HI, and the ICU.

The ICU provides the following functionality:

- Power, Command and Telemetry interfaces with the spacecraft
- Supply spacecraft Survival Heater power to HI
- Provide switched spacecraft Primary Power to the CCD Decontamination Heaters
- Control the Door Latch Release Mechanism and the Door Open/Close Motor
- Control the Lens Barrel Operational Heaters
    - Switched spacecraft Primary Power, using a control loop closed around Lens Barrel Temperature Monitors
- Control the Calibration LEDs in the Cameras
- Provide switched spacecraft Primary Power to the Camera Electronics Box (CEB)
- Communicate with the CEB via a SpaceWire link (100 Mbps)
    - Load CCD sequencer tables
    - Initiate CCD exposures
    - Acquire the camera images
- Process the acquired images
    - Sum the individual exposures into a summed image sequence
    - Perform the shutterless correction (Eyles et al., 2009, Solar Physics 254, 387.) – this is optimally performed before pixel binning and region-of-interest selection
    - Pixel binning and region-of-interest selection
    - Data compression (square-root encoding, Rice compression)
    - Format images for telemetry downlink
- Provide Instrument Housekeeping and Health & Status monitoring
    - Temperature Monitors
    - Door Mechanism Monitors
    - Power supply rail voltage and current monitors.

Although a new development, the requirements for the ICU are not particularly demanding and should be fairly easy to meet using a modern space-qualified processor system, *e.g.* a Leon 2 or Leon 3 processor implemented in radiation tolerant FPGA.

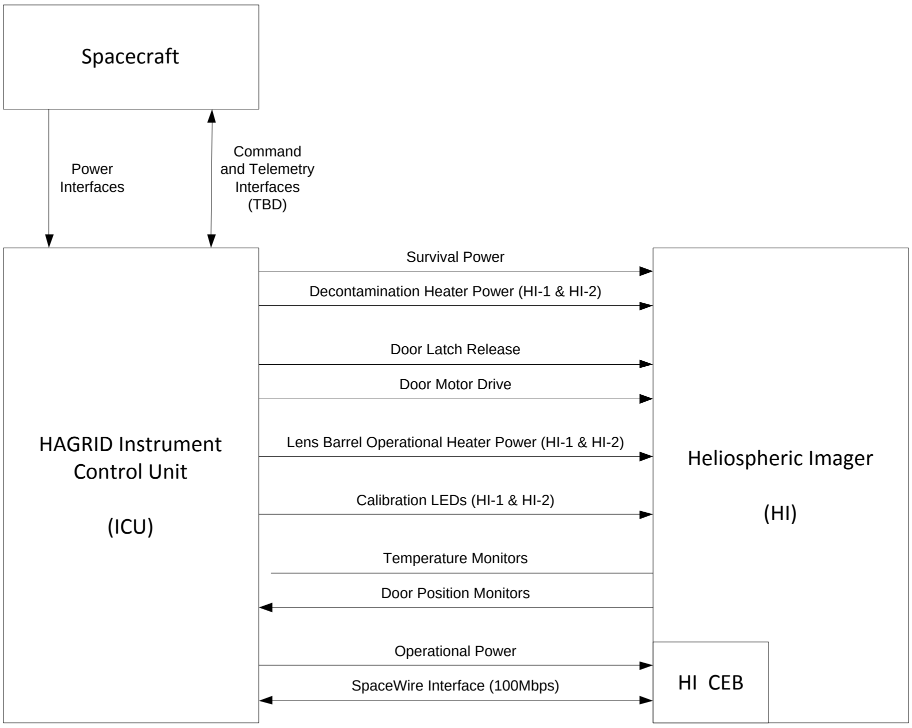


Figure 14 - Interfaces between the HAGRID HI instrument and the Instrument Control Unit.

4.2.2.2 Requirement for a Re-Latching Door Mechanism

During the study two options were considered for the orbital configuration:

- A *Drifting* configuration where the spacecraft continues to drift away from the Earth
    - Results in a relatively short time period with optimum CME viewing geometry
    - Results in a strictly limited mission lifetime
- A *Stopping* configuration where a final firing of thrusters is used to stop the spacecraft at an optimum viewing geometry (*e.g*. elongation range 40 – 60º)
    - Mission lifetime can be extended.

In the *Stopping* case it is necessary that the HI Door remains closed until after the final thrusters firing to avoid possible contamination of the optics. If a one-shot Door mechanism is used (as on STEREO-HI) this implies that no observations can be made until the end of an extended Cruise Phase.

It was therefore decided during the study to make provision for a re-latching Door Mechanism to enable the door to be opened during the Cruise Phase, and then closed and re-opened at the time of the final thrusters firing.

The baseline for the upgraded Door Mechanism for HAGRID is:

- Re-use of STEREO-HI design as far as possible, *e.g*. latch mechanism to take launch loads
- Use stepper motor instead of springs to open (and re-close) the door
- If possible, provide a HOP back-up release system (one-shot) to open the door in case of motor failure (as for STEREO SCIP cameras)
- Allowance of additional 1 kg mass budget (peak power value not affected).

4.2.2.3 Telemetry Data Rate Considerations

It became clear during the study that the instrument operational telemetry requirement is a major design driver for the mission, in terms of transmitter power, hi-gain antenna size, ground station requirements, *etc* and consequently has a major impact on the total cost.

Taking the telemetry requirements for STEREO-HI science data images as a starting point, we have:

- 48 summed images per day
- Each image is 2 x 2 CCD pixel binned, i.e. 1024 x 1024 image bins
- 32 bits per pixel
- Assumed Rice compression factor ~ 2.

This corresponds to a continuous data rate of 9.1 kbps. On the other hand the telemetry rate allocated to HI in the STEREO space weather beacon mode data is only ~ 260 bps and is used to down-link images that are more highly-binned (8 x 8 CCD pixel binning; 256 x 256 image bins) at a lower cadence (24 images per day total).

In order to make progress within the total cost (and other) constraints on the HAGRID mission, it was decided to consider 500 bps as a baseline telemetry data rate (which is similar to the total STEREO beacon mode allocation to all the SECCHI instruments).

In order to fit within this allocation, a number of ways of reducing the image telemetry have been considered:

- 4 x 4 CCD pixel binning, resulting in 512 x 512 image bins
    - o Results in 2 arcmin and 8 arcmin sky pixels for HI-1 and HI-2
    - o This is midway between the STEREO-HI science (1024 x 1024 bins) and beacon mode (256 x 256 bins) binning

- Selection of regions-of-interest within images for down-linking (see Figure 15)
    - o For HI-1 select a horizontal strip of 20º x 2.5º (512 x 64 bins), centred on the ecliptic for producing J-maps
    - o For HI-2 select a horizontal strip of 70º x 8.75º (512 x 64 bins), also centred on the ecliptic for producing J-maps
    - o For HI-1 also select a vertical strip of 5º x 20º (128 x 512 bins), perpendicular to the ecliptic to enable the latitudinal extent of a transient to be determined, thus enabling CIRs to be distinguished from CMEs.

- Dynamic range of images reduced to 16 bits, *e.g*. square-root encoding

- Optimised Rice compression factor ~ 2 – 2.5

Implementing all of the above results in a total date rate ~ 270 bps, which leaves some scope for optimising the image products within the 500 bps baseline allocation.

It should be emphasised that the assumptions made here regarding possible ways of reducing the data rate (in particular the 4 x 4 CCD pixel binning) must be verified by further work during any follow-on study of the HAGRID mission (*e.g*. Phase A study).

Vertical strip perpendicular to Ecliptic Plane in HI-1: distinguish CMEs and CIRs

Horizontal strips centred on Ecliptic Plane in HI-1 & HI-2: for tracking CMEs

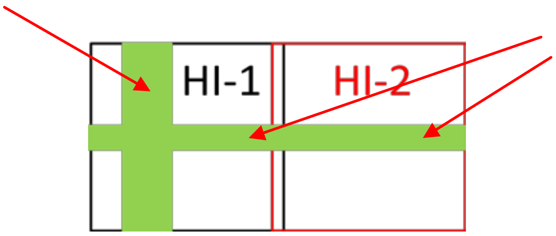


Figure 15 - Regions-of-interest (green) of the HI-1 and HI-2 camera images selected for telemetry down-link in order to reduce the data rate requirements.

4.2.3 TRL of the HI Instrument

The TRL of the HI instrument should be regarded as very high, given that the instruments have been operating aboard the NASA STEREO mission without problems since Oct 2006.

However, the following caveats must be taken into account:

- An Instrument Control Unit (ICU) must be developed to replace the functionality provided by the SECCHI Electronics Box (SEB) on STEREO for (i) spacecraft power, command and data interfaces, (ii) control and monitoring of the instrument, and (iii) on-board data processing
- The ICU will be a new development, although the requirements are not particularly demanding in terms of modern space-qualified processors
- The methods of manufacture of STEREO HI were highly labour intensive, due the extensive use of CFRP structures. These are inherently labour-intensive manufacturing processes, but furthermore required labour-intensive fitting and adjustments when integrating the piece parts into the assembled instrument
- It will be necessary to review the manufacturing processes for HI, for example the trade-offs between CFRP and traditionally-machined Aluminium structures, within the context of a “build-to-print” development philosophy.

4.2.4 Open Issues and Future Work

Although a great deal was achieved, the HAGRID study was necessarily limited in duration and depth and a number of issues relating to the payload remain open and will require further study in any future Phase A (or pre-Phase A) study of HAGRID. These include the following –

- Specifications and parameters of the down-linked camera images:
    - What is the optimum format in terms of CCD pixel binning, ROI selection and image cadence?
    - Confirm that the baseline assumption of 500 bps telemetry down-link is adequate
    - Data compression – square-root encoding, Rice compression, other techniques?

  All these can be assessed by re-processing HI science images, particularly those containing studied events, in order to assess the effects of reduced telemetry on event predictions.

- Is further on-board data processing feasible in order to reduce the down-link telemetry requirements?
    - Is it possible to process images further on board, to the extent of producing J-maps?

This can also be assessed by re-processing HI science images.

- Review the built-to-print philosophy for the HI instrument
  - How to produce a new copy of the instrument without requiring the high levels of labour inputs required for STEREO-HI?
  - Trade-offs between CFRP and traditionally-machined aluminium structures for the baffles.

- Consider possible simplification to 1 camera, e.g. 54º FOV covering solar elongation range 6º to 60º
  - Increasing the inner solar elongation to 6º (or even 8º) would ease the design and manufacturing tolerances of the Forward Baffles
  - The optics design of a 54º FOV wide-angle camera optics would be much less demanding than the 70º FOV of the current HI-2, and would result in significantly improved image quality
  - Potential savings of mass, size, power, AIT complexity, *etc*
  - But no longer build-to-print!

- Formulate detailed requirements for the ICU hardware and software (on-board data processing).

- Re-address the issue of the minimum solar elongation angle required for start of mission operations which satisfy the mission requirements.

4.3 Systems Design

The CDF study considered the spacecraft with all its subsystems and components and the virtual model of it was iterated as a complete system. A preliminary spacecraft architecture was drawn up before the study and refined with each iteration. The following section shows the final version of the architecture for the two configuration options which will be put forward for the next phase.

4.3.1 Spacecraft Architecture

Figure 16 shows the composite option for the spacecraft. The separate propulsion module is based on a similarly staged module designed for the LISA Pathfinder mission. Separating fuel and main engine this way enables the rest of the spacecraft structure to be considerably smaller as shown in the Structures section.

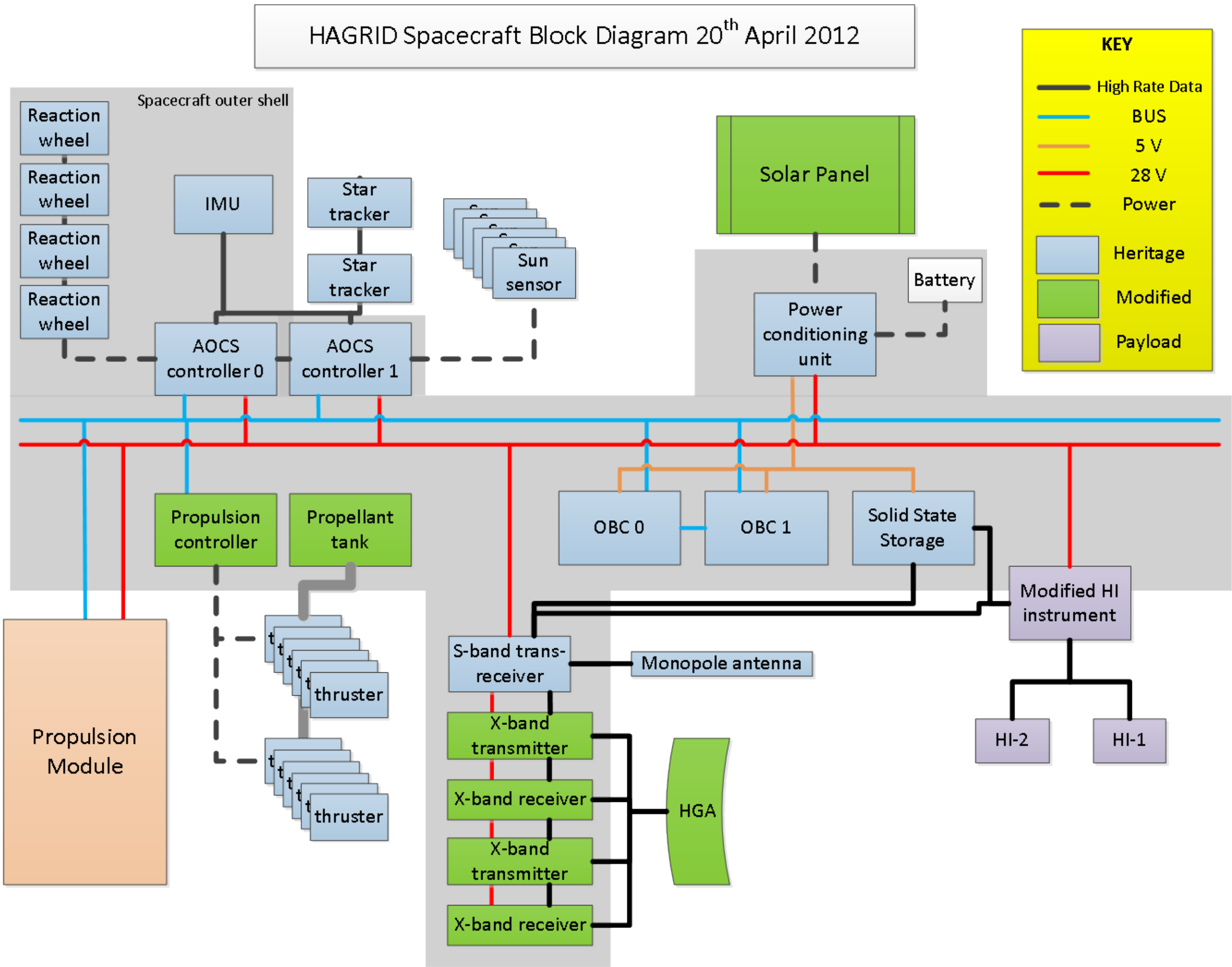


Figure 16 - Block diagram for HAGRID spacecraft showing all subsystems, composite option

Figure 17 show the integrated option of the spacecraft where fuel tanks are accommodated inside the structure of the spacecraft, and there is no stage-separation beyond Earth orbit.

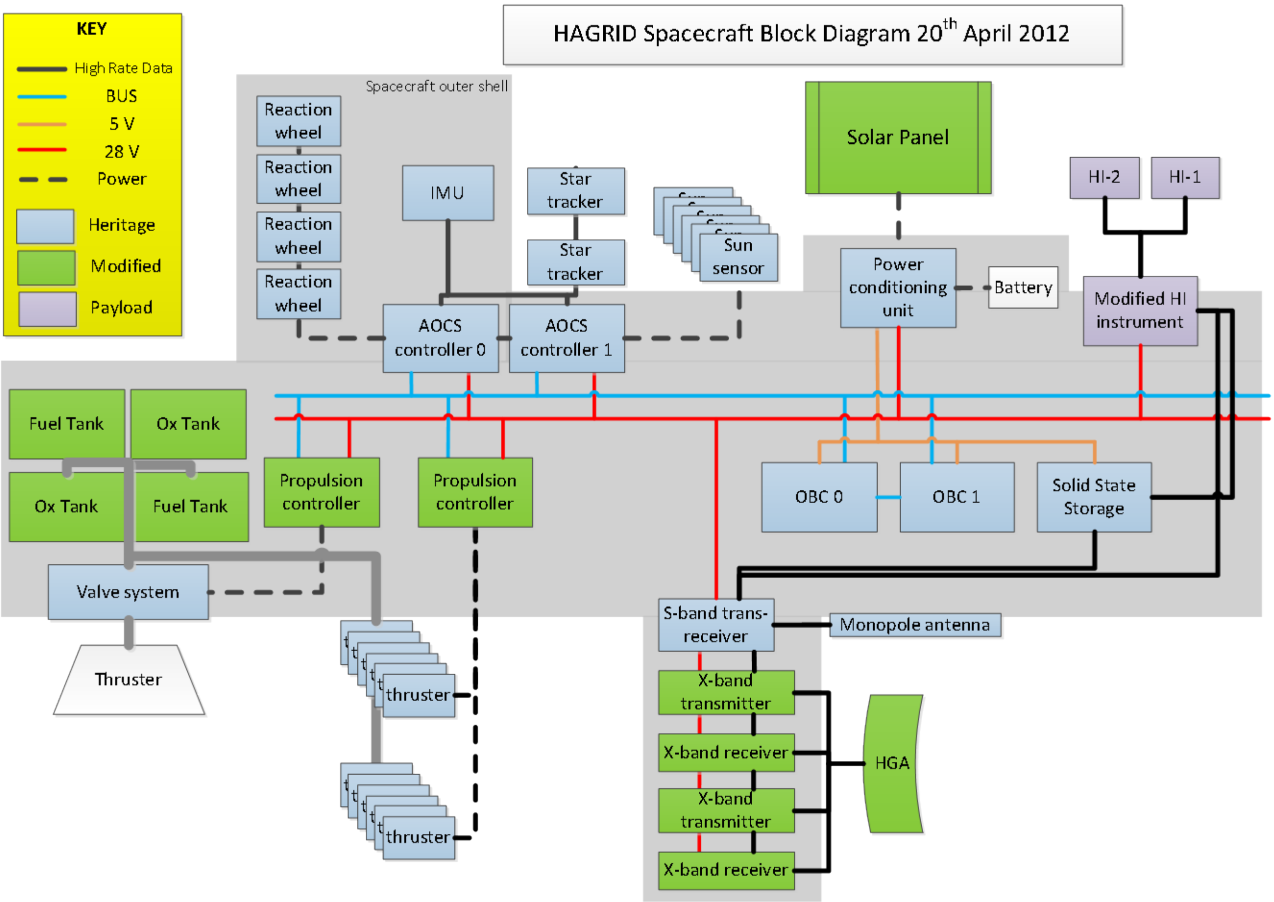


Figure 17 - Block diagram for HAGRID spacecraft showing all subsystems, integrated option

4.4 Mass Budget

For compatibility with the launch vehicle and in order to predict launch costs, an overall mass budget has been calculated throughout the study. The Rockot launcher has been selected as the baseline, which has a payload capacity of 1900 kg to an elliptical Low Earth Orbit (LEO). Other similar size vehicles were considered such as VEGA, Soyuz and PSLV and details about these launchers are presented in the Mission Analysis section.

The spacecraft mass budget was compiled in the Configuration subsystem workbook. These tables include areas for potential spacecraft components which may or may not be included in general spacecraft CDF studies. The 'Earth sensor mount info', for example, was not included in this study as the spacecraft is orbiting the Sun, and does use an Earth sensor. The detailed mass budgets for the integrated spacecraft, composite monopropellant spacecraft and composite bi-propellant spacecraft are shown in Table 12, Table 16, and Table 20 respectively.

Initial wet mass calculations took the dry mass with a system margin of 20% and then calculated the wet mass on top of that. However during the study the parametric equations have been changed to calculate the propellant mass based on the maximum injected launch mass, which is more appropriate for an early design study if the spacecraft wet mass is large enough to be a similar order of magnitude to the maximum injected mass. With the propellant Isp in Table 9, Table 13, and Table 17; the delta V budget from Mission Analysis in Table 10, Table 14, and Table 18; and assumed Maximum Injection Mass (1900 kg) the amount of initial propellant needed to perform the delta V manoeuvres was calculated in the propulsion workbook using the formula;

Mass of Initial Propellant =Injection Mass*(1-EXP(-summed_deltaV/(Isp*gravity)))

By subtracting the mass of initial propellant from the injection mass, a spacecraft mass limit was determined. This was then compared with the mass budget tabulated in the Configuration workbook.

During the study the spacecraft model diverged into two options: a spacecraft containing the propulsion module necessary to reach final orbit (the integrated option) and a spacecraft using a separate propulsion module based on an existing design (the composite option).

A comparison of the spacecraft differences between the integrated and composite configurations is presented in Table 8 while the CAD models are shown in Figure 18. The power subsystem is significantly heavier in the composite case, as the Lisa Pathfinder PRM requires more power, but the propulsion subsystem and spacecraft structure both have significantly greater masses for the integrated spacecraft.

Table 8 - Comparison of subsystem mass allocations between the integrated and composite spacecraft options.

Subsystem	Integrated [kg]	Composite [kg]
AOCS	15.1	15.1
CDMS	16.3	16.3
Comms	27.8	27.8
Payload	20.3	20.3
Power	30.3	50.4
Propulsion	113.3	22.6
Structures	201.6	94.7
Thermal	18.0	3.7
Total	464.9	263.4

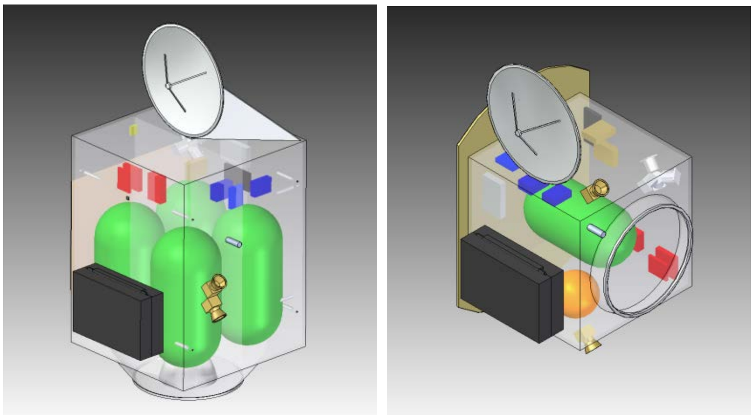


Figure 18- The integrated spacecraft (left) and composite spacecraft without the PRM (right).

4.4.1 Final Mass budget for Integrated option

Table 9 - Integrated spacecraft propulsion parameters relevant to propellant sizing.

Engine Parameters	Values	Units
Engine Type	Bi-propellant	
Fuel	MMH	
Oxidiser	MON_3	
O/F ratio	1.650	
Engine Thrust	420.000	**N**
Mass Flow Rate	0.135	**kg/s**
Specific Impulse	318.000	**s**

Table 10 - Integrated spacecraft DeltaV requirements and resulting propellant mass.

Delta-V and Propellant Items	Values	Units
Estimate Wet Mass	1900	Kg
Estimated op aocs deltaV	30	**m/s**
Estimated drift deltaV	3160.5	**m/s**
Estimated dispersion deltaV	32	**m/s**
Estimated transfer deltaV	588	**m/s**
Estimated tx aocs deltaV	80	**m/s**
Summed deltaV	3890.5	**m/s**
Initial Mass of Propellant	1354.073	**Kg**
Initial Mass of Fuel	510.971	**Kg**
Initial Mass of Oxidizer	843.102	**Kg**
Injection Dry Mass	545.927	Kg

Table 11 - Integrated spacecraft mass budget including margin.

Subsystem	Current Best Estimate	Units
AOCS	16.680	kg
CDMS	30.000	kg
Comms	36.990	kg
Payload	19.275	kg
Power	45.171	kg
Propulsion	109.528	kg
Structures	196.158	kg
Thermal	16.136	kg
Harness	23.497	kg
Nominal Dry Mass	493.436	kg
System Margin	20	%
System Margin	98.687	kg
Total Dry Mass at Launch	592.123	kg
Mass of Fuel and Oxidizer	1379.57	kg
S/C Total Wet Mass at Launch	1971.69	kg
Max S/C Total Wet Mass At Launch	1900.00	kg
Margin	-71.688	kg

Table 12 - Complete mass budget for the integrated spacecraft.

Subsystem Name	Part	Mounting Considerations	Equipment CBE Mass [kg]	Number		Curent Best Estimate [kg]	Estimation Method	Growth Margin [%]	Growth Margin [kg]	Mass Allocation [kg]	Subsystem Current Best Estimate [kg]
AOCS	Star Tracker (Heads)	non-sun face	0.142	2		0.28	Off the Shelf	5	0.014	0.298	
AOCS	Star Tracker (Processing Units)	star tracker pu mount info	1.190	2		2.38	Off the Shelf	5	0.119	2.499	
AOCS	Sun Sensor	sun sensor mount info	0.035	6		0.21	Off the Shelf	5	0.011	0.221	
AOCS	Earth Sensor	earth sensor mount info	0.000	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	Magnetometer	magnetometer mount info	0.000	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	Magnetorquer	magnetorquer mount info	0.000	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	Reaction Wheels	tetrahedral configuration	1.550	4		6.20	Off the Shelf	5	0.310	6.510	
AOCS	ACS Computer	ACS computer mount info	0.012	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	ACS IMU	ACS IMU mount info	1.800	1		1.80	Off the Shelf	5	0.090	1.890	
AOCS	Thrusters	ACS Thruster mount info	0.290	12		3.48	Off the Shelf	5	0.174	3.654	
											15.072
CDMS	OBCS(s)	OBC Mount Info	15.500	1		15.50	Off the Shelf	5	0.775	16.275	
CDMS	OBMM integrated in OBC	Not included	10.000	0		0.00	New Design	20	0.000	0.000	
											16.275
Comms	Antenna	Antenna mount info	10.000	1		10.00	New Design	20	2.000	12.000	
Comms	Amplifier	Amplifier mount info	0.800	2		1.60	Off the Shelf	5	0.080	1.680	
Comms	Transmitter	Transmitter mount info	3.500	2		7.00	Off the Shelf	5	0.350	7.350	
Comms	Pointing Mechanism	Antenna mount info	0.000	1		0.00	New Design	20	0.000	0.000	
Comms	Receiver	Receiver mount info	0.000	2		0.00	Off the Shelf	5	0.000	0.000	
Comms	EPC	Amplifier mount info	1.350	2		2.70	Off the Shelf	5	0.135	2.835	
Comms	RFDU		1.000	1		1.00	New Design	20	0.200	1.200	
Comms	Low Gain Antenna and Filter		1.150	2		2.30	New Design	20	0.460	2.760	
											27.825
Payload	Heliospheric Imager (FPI)	Instrument Mount Info	17.300	1		17.30	Off the Shelf	5	0.865	18.165	
Payload	Heliospheric Imager (DPU)	DPU Mount Info	1.800	1		1.80	New Design	20	0.360	2.160	
											20.325
Power	Solar Array (1)	Solar Array 1 mount info	5.778	1		5.78	New Design	20	1.156	6.934	
Power	Solar Array (2)	Mounting	5.000	1		5.00	New Design	20	1.000	6.000	
Power	Batteries	Batteries mount info	6.410	1		6.41	New Design	20	1.282	7.692	
Power	Power Processing Unit	PCU mount info	8.300	1		8.30	Modified Off the Shelf	10	0.830	9.130	
Power	Bus regulators	Array Bus Regulator mount info	0.550	1		0.55	Off the Shelf	5	0.028	0.578	
											30.333
Propulsion	Mono/Solid/Bi/EP (engine)	Apogee Engine mount info	6.300	1		6.30	Off the Shelf	5	0.315	6.615	
Propulsion	Mono/Solid/Bi/EP (oxidiser tanks)	N/A	34.000	1		34.00	Off the Shelf	5	1.700	35.700	
Propulsion	Mono/Solid/Bi/EP (fuel tanks)	N/A	34.000	1		34.00	Off the Shelf	5	1.700	35.700	
Propulsion	Mono/Solid/Bi/EP (pressure tanks)	N/A	18.600	1		18.60	Off the Shelf	5	0.930	19.530	
Propulsion	Pressure sys and distribution equip	N/A	8.300	1		8.30	Off the Shelf	5	0.415	8.715	
Propulsion	Pipework and Supports	N/A	5.870	1		5.87	New Design	20	1.174	7.044	
											113.304
Structures	Spacecraft Bus	Spacecraft Bus mount info	167.098	1		167.10	New Design	20	33.420	200.517	
Structures	Spacecraft Bus Margin		33.420	0		0.00	New Design	20	0.000	0.000	
Structures	Solar Array mechanism(s)	Solar Array Mech Mount Info	0.000	1		0.00	Modified Off the Shelf	10	0.000	0.000	
Structures	Deployment mechanism	Deployment Mech Mount info	1.000	0		0.00	New Design	20	0.000	0.000	
Structures	Antenna mechanism (1 TX)	TX Antenna Mech Mount Info	0.500	1		0.50	Modified Off the Shelf	10	0.050	0.550	
Structures	Antenna mechanism (2 RX)	RX Antenna Mount Info	0.500	1		0.50	Modified Off the Shelf	10	0.050	0.550	
Structures	Payload pointing mechanism (EP/aux)	Auxillary Mech Mount info	None	0		N/A	New Design	20	N/A	N/A	
											201.617
Thermal	Radiator 1		0.000	0.005	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 2		1.200	0.003	Aluminum	9.72	New Design	20	1.944	11.664	
Thermal	Radiator 3		0.000	0.025	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 4		0.000	0.03	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 5		0.000	0.008	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 6		0.000	0.01	Aluminum	0.00	New Design	20	0.000	0.000	
			area	thickness	material	mass					
Thermal	MLI					6.00	Off the Shelf	5	0.300	6.300	
Thermal	Cold Link 1		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 2		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 3		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 4		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 5		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 6		0.000		Copper	0.00	New Design	20	0.000	0.000	
											17.964
										Dry Mass (w/o harness)	442.715
										Harness %	5%
										Harness (kg)	22.136
										Nominal Dry Mass	464.851

4.4.2 Final Mass budget for Composite option

4.4.2.1 Composite monopropellant option

Table 13 - Composite monopropellant spacecraft propulsion parameters relevant to propellant sizing.

Engine Parameters	Values	Units
Engine Type	Bi-propellant	
Fuel	MMH	
Oxidiser	MON_3	
O/F ratio	0.001	
Engine Thrust	1.000	**N**
Mass Flow Rate	0.00044	**kg/s**
Specific Impulse	215.000	**s**

Table 14 - Composite monopropellant DeltaV requirements and resulting propellant mass.

Engine Parameters	Values	Units
Estimate Wet Mass	1900	kg
Estimated op aocs deltaV	0	**m/s**
Estimated drift deltaV	618	**m/s**
Estimated dispersion deltaV	0	**m/s**
Estimated transfer deltaV	0	**m/s**
Estimated tx aocs deltaV	0	**m/s**
Summed deltaV	618	**m/s**
Initial Mass of Propellant	482.571	**kg**
Initial Mass of Fuel	482.089	**kg**
Initial Mass of Oxidizer	0.482	**kg**
Injection Dry Mass	1417.429265	**kg**

Table 15 - Composite monopropellant spacecraft mass budget including margin.

Subsystem	Current Best Estimate	Units
AOCS	15.072	**kg**
CDMS	16.275	**kg**
Comms	27.825	**kg**
Payload	20.325	**kg**
Power	38.851	**kg**
Propulsion	10.463	**kg**
Structures	46.700	**kg**
Thermal	16.136	**kg**
Harness	9.582	**kg**
Nominal Dry Mass	201.229	**kg**
System Margin	20	**%**
System Margin [kg]	40.246	**kg**
Total Dry Mass at Launch	241.475	**kg**
Mass of AOCS Monopropellant	98.29	**kg**
Spacecraft Total Wet Mass at Launch	339.77	**kg**
Propulsion Module Dry mass	219.500	**kg**
System Margin	20	**%**
System Margin [kg]	43.9	**kg**
Total Dry Mass at Launch	263.4	**kg**
Mass of Fuel and Oxidizer	1275.00	**kg**
Propulsion Module Total Wet Mass at Launch	1538.4	**kg**
S/C + Propulsion Module Total Wet Mass at Launch	1878.17	**kg**
Max S/C Total Wet Mass At Launch	1900.00	**kg**
Margin	21.83	**kg**

Table 16 - Complete mass budget for the composite monopropellant spacecraft.

Subsystem Name	Part	Mounting Considerations	Equipment CBE Mass [kg]	Number		Curent Best Estimate [kg]	Estimation Method	Growth Margin [%]	Growth Margin [kg]	Mass Allocation [kg]	Subsystem Current Best Estimate [kg]
AOCS	Star Tracker (Heads)	non-sun face	0.142	2		0.28	Off the Shelf	5	0.014	0.298	
AOCS	Star Tracker (Processing Units)	star tracker pu mount info	1.190	2		2.38	Off the Shelf	5	0.119	2.499	
AOCS	Sun Sensor	sun sensor mount info	0.035	6		0.21	Off the Shelf	5	0.011	0.221	
AOCS	Earth Sensor	earth sensor mount info	0.000	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	Magnetometer	magnetometer mount info	0.000	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	Magnetorquer	magnetorquer mount info	0.000	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	Reaction Wheels	tetrahedral configuration	1.550	4		6.20	Off the Shelf	5	0.310	6.510	
AOCS	ACS Computer	ACS computer mount info	0.012	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	ACS IMU	ACS IMU mount info	1.800	1		1.80	Off the Shelf	5	0.090	1.890	
AOCS	Thrusters	ACS Thruster mount info	0.290	12		3.48	Off the Shelf	5	0.174	3.654	
											15.072
CDMS	OBCS(s)	OBC Mount Info	15.500	1		15.50	Off the Shelf	5	0.775	16.275	
CDMS	OBMM integrated in OBC	Not included	10.000	0		0.00	New Design	20	0.000	0.000	
											16.275
Comms	Antenna	Antenna mount info	10.000	1		10.00	New Design	20	2.000	12.000	
Comms	Amplifier	Amplifier mount info	0.800	2		1.60	Off the Shelf	5	0.080	1.680	
Comms	Transmitter	Transmitter mount info	3.500	2		7.00	Off the Shelf	5	0.350	7.350	
Comms	Pointing Mechanism	Antenna mount info	0.000	1		0.00	New Design	20	0.000	0.000	
Comms	Receiver	Receiver mount info	0.000	2		0.00	Off the Shelf	5	0.000	0.000	
Comms	EPC	Amplifier mount info	1.350	2		2.70	Off the Shelf	5	0.135	2.835	
Comms	RFDU		1.000	1		1.00	New Design	20	0.200	1.200	
Comms	Low Gain Antenna and Filter		1.150	2		2.30	New Design	20	0.460	2.760	
											27.825
Payload	Heliospheric Imager (FPI)	Instrument Mount Info	17.300	1		17.30	Off the Shelf	5	0.865	18.165	
Payload	Heliospheric Imager (DPU)	DPU Mount Info	1.800	1		1.80	New Design	20	0.360	2.160	
											20.325
Power	Solar Array (1)	Solar Array 1 mount info	9.694	1		9.69	Modified Off the Shelf	10	0.969	10.663	
Power	Solar Array (2)	Solar Array 2 mount info	9.000	1		9.00	Modified Off the Shelf	10	0.900	9.900	
Power	Batteries	Batteries mount info	7.800	1		7.80	Modified Off the Shelf	10	0.780	8.580	
Power	Power Processing Unit	PCU mount info	8.300	1		8.30	Modified Off the Shelf	10	0.830	9.130	
Power	Bus regulators	Array Bus Regulator mount info	0.550	1		0.55	Off the Shelf	5	0.028	0.578	
											38.851
Propulsion	Mono/Solid/Bi/EP (engine)	Hydrazine Engine	3.000	1		3.00	Off the Shelf	5	0.150	3.150	
Propulsion	Mono/Solid/Bi/EP (oxidiser tanks)	N/A	0.000	0		0.00	Off the Shelf	5	0.000	0.000	
Propulsion	Mono/Solid/Bi/EP (fuel tanks)	N/A	8.070	1		8.07	Off the Shelf	5	0.404	8.474	
Propulsion	Mono/Solid/Bi/EP (pressure tanks)	3.22	3.890	1		3.89	Off the Shelf	5	0.195	4.085	
Propulsion	Pressure sys and distribution equip	N/A	3.220	1		3.22	Off the Shelf	5	0.161	3.381	
Propulsion	Pipework and supports	N/A	2.930	1		2.93	New Design	20	0.586	3.516	
											22.605
Structures	Spacecraft Bus	Spacecraft Bus mount info	35.000	1		35.00	New Design	20	7.000	42.000	
Structures	Spacecraft Bus Margin		7.000	0		0.00	New Design	20	0.000	0.000	
Structures	Solar Array mechanism(s)	Solar Array Mech Mount Info	0.000	1		0.00	Modified Off the Shelf	10	0.000	0.000	
Structures	Deployment mechanism	Deployment Mech Mount info	3.000	1		3.00	New Design	20	0.600	3.600	
Structures	Antenna mechanism (1 TX)	TX Antenna Mech Mount Info	0.500	1		0.50	Modified Off the Shelf	10	0.050	0.550	
Structures	Antenna mechanism (2 RX)	RX Antenna Mount Info	0.500	1		0.50	Modified Off the Shelf	10	0.050	0.550	
Structures	Payload pointing mechanism (EP/aux)	Auxillary Mech Mount info	None	0		N/A	New Design	20	N/A	N/A	
											46.700
Thermal	Radiator 1		0.000	0.005	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 2		0.840	0.004	Aluminum	9.07	New Design	20	1.814	10.886	
Thermal	Radiator 3		0.000	0.025	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 4		0.000	0.03	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 5		0.000	0.008	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 6		0.000	0.01	Aluminum	0.00	New Design	20	0.000	0.000	
			area	thickness	material	mass					
Thermal	MLI					5.00	Off the Shelf	5	0.250	5.250	
Thermal	Cold Link 1		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 2		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 3		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 4		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 5		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 6		0.000		Copper	0.00	New Design	20	0.000	0.000	
											16.136
										Dry Mass (w/o harness	203.789
										Harness %	5%
										Harness (kg)	10.189
										Nominal Dry Mass	213.978

4.4.3 Composite Bi-propellant option

Table 17 - Composite bi-propellant spacecraft propulsion parameters relevant to propellant sizing.

Engine Parameters	Values	Units
Engine Type	Bi-propellant	
Fuel	MMH	
Oxidiser	MON_3	
O/F ratio	0.001	
Engine Thrust	1.000	**N**
Mass Flow Rate	0.00044	**kg/s**
Specific Impulse	215.000	**s**

Table 18 - Composite monopropellant DeltaV requirements and resulting propellant mass.

Engine Parameters	Values	Units
Estimate Wet Mass	1900	kg
Estimated op aocs deltaV	0	**m/s**
Estimated drift deltaV	618	**m/s**
Estimated dispersion deltaV	0	**m/s**
Estimated transfer deltaV	0	**m/s**
Estimated tx aocs deltaV	0	**m/s**
Summed deltaV	618	**m/s**
Initial Mass of Propellant	482.571	**kg**
Initial Mass of Fuel	482.089	**kg**
Initial Mass of Oxidizer	0.482	**kg**
Injection Dry Mass	1417.429	**kg**

Table 19 - Composite bi-propellant spacecraft mass budget including margin.

Subsystem	Current Best Estimate	Units
AOCS	15.072	**kg**
CDMS	16.275	**kg**
Comms	27.825	**kg**
Payload	20.325	**kg**
Power	38.851	**kg**
Propulsion	10.463	**kg**
Structures	46.700	**kg**
Thermal	16.136	**kg**
Harness	9.582	**kg**
Nominal Dry Mass	201.229	**kg**
System Margin	20	**%**
System Margin	40.246	**kg**
Total Dry Mass at Launch	241.475	**kg**
Mass of AOCS Fuel and Oxidizer	69.55	**kg**
Spacecraft Total Wet Mass at Launch	311.02	**kg**
Propulsion Module Dry mass	219.500	**kg**
System Margin	20	**%**
System Margin	43.9	**Kg**
Total Dry Mass at Launch	263.4	**Kg**
Mass of Fuel and Oxidizer	1275.00	**Kg**
Propulsion Module Total Wet Mass at Launch	1538	**Kg**
S/C + Propulsion Module Total Wet Mass at Launch	1849.42	**Kg**
Maximum Spacecraft Total Wet Mass At Launch	1900.00	**Kg**
Margin	50.58	**Kg**

Table 20 - Complete mass budget for the composite monopropellant spacecraft.

Subsystem Name	Part	Mounting Considerations	Equipment CBE Mass [kg]	Number		Curent Best Estimate [kg]	Estimation Method	Growth Margin [%]	Growth Margin [kg]	Mass Allocation [kg]	Subsystem Current Best Estimate [kg]
AOCS	Star Tracker (Heads)	non-sun face	0.142	2		0.28	Off the Shelf	5	0.014	0.298	
AOCS	Star Tracker (Processing Units)	star tracker pu mount info	1.190	2		2.38	Off the Shelf	5	0.119	2.499	
AOCS	Sun Sensor	sun sensor mount info	0.035	6		0.21	Off the Shelf	5	0.011	0.221	
AOCS	Earth Sensor	earth sensor mount info	0.000	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	Magnetometer	magnetometer mount info	0.000	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	Magnetorquer	magnetorquer mount info	0.000	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	Reaction Wheels	tetrahedral configuration	1.550	4		6.20	Off the Shelf	5	0.310	6.510	
AOCS	ACS Computer	ACS computer mount info	0.012	0		0.00	Off the Shelf	5	0.000	0.000	
AOCS	ACS IMU	ACS IMU mount info	1.800	1		1.80	Off the Shelf	5	0.090	1.890	
AOCS	Thrusters	ACS Thruster mount info	0.290	12		3.48	Off the Shelf	5	0.174	3.654	
											15.072
CDMS	OBCS(s)	OBC Mount Info	15.500	1		15.50	Off the Shelf	5	0.775	16.275	
CDMS	OBMM integrated in OBC	Not included	10.000	0		0.00	New Design	20	0.000	0.000	
											16.275
Comms	Antenna	Antenna mount info	10.000	1		10.00	New Design	20	2.000	12.000	
Comms	Amplifier	Amplifier mount info	0.800	2		1.60	Off the Shelf	5	0.080	1.680	
Comms	Transmitter	Transmitter mount info	3.500	2		7.00	Off the Shelf	5	0.350	7.350	
Comms	Pointing Mechanism	Antenna mount info	0.000	1		0.00	New Design	20	0.000	0.000	
Comms	Receiver	Receiver mount info	0.000	2		0.00	Off the Shelf	5	0.000	0.000	
Comms	EPC	Amplifier mount info	1.350	2		2.70	Off the Shelf	5	0.135	2.835	
Comms	RFDU		1.000	1		1.00	New Design	20	0.200	1.200	
Comms	Low Gain Antenna and Filter		1.150	2		2.30	New Design	20	0.460	2.760	
											27.825
Payload	Heliospheric Imager (FPI)	Instrument Mount Info	17.300	1		17.30	Off the Shelf	5	0.865	18.165	
Payload	Heliospheric Imager (DPU)	DPU Mount Info	1.800	1		1.80	New Design	20	0.360	2.160	
											20.325
Power	Solar Array (1)	Solar Array 1 mount info	9.694	1		9.69	Modified Off the Shelf	10	0.969	10.663	
Power	Solar Array (2)	Solar Array 2 mount info	9.000	1		9.00	Modified Off the Shelf	10	0.900	9.900	
Power	Batteries	Batteries mount info	7.800	1		7.80	Modified Off the Shelf	10	0.780	8.580	
Power	Power Processing Unit	PCU mount info	8.300	1		8.30	Modified Off the Shelf	10	0.830	9.130	
Power	Bus regulators	Array Bus Regulator mount info	0.550	1		0.55	Off the Shelf	5	0.028	0.578	
											38.851
Propulsion	Mono/Solid/Bi/EP (engine)	Apogee Engine mount info	6.300	1		6.30	Off the Shelf	5	0.315	6.615	
Propulsion	Mono/Solid/Bi/EP (oxidiser tanks)	N/A	3.580	1		3.58	Off the Shelf	5	0.179	3.759	
Propulsion	Mono/Solid/Bi/EP (fuel tanks)	N/A	3.580	1		3.58	Off the Shelf	5	0.179	3.759	
Propulsion	Mono/Solid/Bi/EP (pressure tanks)	N/A	3.610	1		3.61	Off the Shelf	5	0.181	3.791	
Propulsion	Press sys and distribution equip	N/A	6.650	1		6.65	New Design	20	1.330	7.980	
Propulsion	Pipework and supports	N/A	5.870	1		5.87	New Design	20	1.174	7.044	
											32.948
Structures	Spacecraft Bus	Spacecraft Bus mount info	35.000	1		35.00	New Design	20	7.000	42.000	
Structures	Spacecraft Bus Margin		7.000	0		0.00	New Design	20	0.000	0.000	
Structures	Solar Array mechanism(s)	Solar Array Mech Mount Info	0.000	1		0.00	Modified Off the Shelf	10	0.000	0.000	
Structures	Deployment mechanism	Deployment Mech Mount info	3.000	1		3.00	New Design	20	0.600	3.600	
Structures	Antenna mechanism (1 TX)	TX Antenna Mech Mount Info	0.500	1		0.50	Modified Off the Shelf	10	0.050	0.550	
Structures	Antenna mechanism (2 RX)	RX Antenna Mount Info	0.500	1		0.50	Modified Off the Shelf	10	0.050	0.550	
Structures	Payload pointing mechanism (EP/aux)	Auxillary Mech Mount info	None	0		N/A	New Design	20	N/A	N/A	
											46.700
Thermal	Radiator 1		0.000	0.005	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 2		0.840	0.004	Aluminum	9.07	New Design	20	1.814	10.886	
Thermal	Radiator 3		0.000	0.025	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 4		0.000	0.03	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 5		0.000	0.008	Aluminum	0.00	New Design	20	0.000	0.000	
Thermal	Radiator 6		0.000	0.01	Aluminum	0.00	New Design	20	0.000	0.000	
			area	thickness	material	mass					
Thermal	MLI					5.00	Off the Shelf	5	0.250	5.250	
Thermal	Cold Link 1		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 2		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 3		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 4		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 5		0.000		Copper	0.00	New Design	20	0.000	0.000	
Thermal	Cold Link 6		0.000		Copper	0.00	New Design	20	0.000	0.000	
											16.136
										Dry Mass (w/o harness	214.131
										Harness %	5%
										Harness (kg)	10.707
										Nominal Dry Mass	224.838

4.5 Propulsion

Propulsion Contents:
- Propulsion system requirements
- Propulsion system design
- Propulsion system selection
- Future work for the propulsion subsystem

4.5.1 Propulsion system requirements

The scope of the propulsion subsystem is to design and/or select of hardware to provide all manoeuvres between launch injection and the end of operational lifetime of 5 years for this study.

The specific tasks are;
- Selection of propellant, propellant tanks and pressurant tanks
- Provision of any upper-stage or auxiliary propulsion system needed for interplanetary orbits
- Consideration of all pipes and valves, and necessary support structure or geometrical constraints

Major constrains on the propulsion system are;
- Large DeltaV manoeuvres require in large amounts of propellant
- Size and configuration of propellant tanks due to launcher fairing size and sloshing/operational orbit MoI
- Compatibility of propellant selection with AOCS subsystem

4.5.2 Propulsion subsystem design

Initially, due to cost constraints, the mission was considered as a piggyback payload on an Ariane 5 or Soyuz-Fregat launch. However, the size and mass limits for either of these was too small to provide propellant for the interplanetary transfer. The low cost options suggested were either a PSLV or Rockot dedicated low cost launch. For designing the propulsion subsystem, it was assumed that the Rockot launch vehicle would be used and that the maximum injected mass was 1900 kg.

The specific manoeuvres between launch injection and end of operational lifetime are the apogee raising burns at perigee, injection into heliocentric orbit, drift alteration or ‘stopping’, and operational AOCS burns. With 1900 kg of launch mass, there are two main options of spacecraft configuration: an integrated spacecraft propulsion system providing all propulsion manoeuvres or a composite spacecraft propulsion system with a propulsion module to provide the transfer burns as well as a small propulsion system within the spacecraft providing the operational orbit burns.

For the large DeltaV burns needed for the integrated spacecraft case, providing propellant for the transfer and operational mission phases, a propellant with greater specific impulse was preferred. A propellant with higher specific impulse would reduce the amount of propellant needed, but limited the selection to bi-propellent propulsion systems. For the composite case, with reduced DeltaV requirements, monopropellants could be considered alongside bi-propellants.

The integrated spacecraft case was looked at first and a 400 N engine, 400N Model S400-12, was selected from Astrium to provide large burns, and the AOCS subsystem engineer chose thrusters based on operational orbit needs. The bi-propellant used for this study was monomethylhyrdrazine and mixed oxides of nitrogen with 3% nitrogen oxide (MON-3). For the composite case, the Lisa-Pathfinder Propulsion Module (PRM) was used along side each of a bi-propellant MMH/MON-3 and monopropellant hydrazine (N_2H_4) system. The properties of these propellants are shown in Table 21.

Table 21 - Characteristics of the bi-propellant combination used.

Characteristics	MMH/MON-3	Hydrazine
nominal thrust [N]	420	1
nominal specific impulse [s]	318	220
nominal mass flow rate [g/s]	135	0.44
oxidizer to fuel ratio (O/F)	1.35	N/A

Based on these characteristics, the amount of propellant could be determined from the maximum injection mass using the DeltaV values sent from Mission Analysis. For the integrated case we accounted for the ballistic transfer, drift-change, dispersion, AOCS transfer, and AOCS operational DeltaVs totalling 3918 m/s as shown in Table 22.

Table 22 - Mission analysis DeltaV estimation for the integrated spacecraft.

Mission Delta-V Estimation					
	Velocity [m/s]	CBE Delta V [m/s]	Margins [%]	Margins [m/s]	Allocated Delta-V
GTO Perigee	7978				
Ballistic Transfer Delta-V	10972	3049.40	5	152.47	3201.87
	72				
Drift Change Delta-V		520.40	5	26.02	546.42
	72				
	3658				
	4465				
Gravity Assist Margin (celetial body)	Cases...			0	0
Gravity Assist Margin (EP)	Other			GAM EP margin NA	GAM EP margin N/
EP dV margin	Other			EP margin NA	EP margin NA
Sun-Frame Velocity	30034				
Delta-V Accuracy margin	accurate trajectory including maintanence				
AOCS Margin	100				
Dispersion dV		30	100	30	60
AOCS transfer dV		40	100	40	80
AOCS operational dV		15	100	15	30
				Total Delta-V	3918

For the composite case, the Lisa-Pathfinder Propulsion Module accounted for ballistic, dispersion, and AOCS transfer DeltaV leaving only drift-change and operational AOCS DeltaVs totalling 577 m/s for the spacecraft propulsion system as shown in Table 23.

Table 23 - Mission analysis DeltaV estimation for the composite spacecraft.

Mission Delta-V Estimation					
	Velocity [m/s]	**CBE Delta V [m/s]**	**Margins [%]**	**Margins [m/s]**	**Allocated Delta-V**
GTO Perigee	7978				
Ballistic Transfer Delta-V	10972	0	5	0	0
	72				
Drift Change Delta-V		521	5	26.05	547.05
	72				
	3658				
	4465				
Gravity Assist Margin (celetial body)	Cases...			0	0
Gravity Assist Margin (EP)	Other			GAM EP margin NA	GAM EP margin NA
EP dV margin	Other			EP margin NA	EP margin NA
Sun-Frame Velocity	30034				
Delta-V Accuracy margin	accurate trajectory including maintanence				
AOCS Margin	100				
Dispersion dV		0	100	0	0
AOCS transfer dV		0	100	0	0
AOCS operational dV		15	100	15	30
				Total Delta-V	577

Combining the DeltaV from mission analysis with the propellant characteristics allowed determination of the amount of fuel the spacecraft would need to carry, excluding the PRM in the case of the composite spacecraft. The mass of the Lisa-Pathfinder PRM is 263kg with a propellant mass of 1275kg (provided by Astrium).

Table 24 - Total mass of internal propellant for the integrated and composite spacecraft cases.

Option	Delta-V [m/s]	Total Propellant [kg]
integrated spacecraft	3918	1395
composite spacecraft to L4 using Monopropellant (N2H4)	577	84
composite spacecraft to L4 using Bipropellant (MMH/MON3)	577	60

A secondary set of calculations determined the amount of propellant required considering the entire trajectory from launch to end of life. This allowed calculation of burn time and number of burns, which fed into calculations of the power required. This information was sent to the Power subsystem, and can be seen in

Table 25. The integrated transfer power is larger than the final iteration sent to the Power subsystem, as it was realized that the transfer only included dispersion DeltaV without the actual large transfer burns. Also, the stopping burn duration was calculated using 1 N thrusters to simulate the worst case, but a larger thruster would probably be selected for the stopping burn and then smaller thrusters during operational burns. This is also included in section 4.5.4 where future work on propulsion is discussed.

Table 25 - Burn characteristics which determined power values.

Parameters	Integrated S/C	Integrated S/C	Composite S/C
number of thrusters	1	1	4
amount of thrust [N]	400	400	1
manoeuvres	Transfer	Stopping	Stopping
continuous power [W]	45	45	38
maximum burn time [s]	900	900	900
total burn duration needed [s]		624	162548
number of burns	6475	0.7	1716
total power needed [W]	291375	28080	1544206

From the amount of propellant needed, the propellant fuel, oxidizer, and size of pressurant tanks could be determined. This was initially estimated by empirical formulae and honed by further analysis undertaken by engineers at Astrium who provided more detailed values from specific off-the-shelf tanks and gave figures for items like pipe and support structure that are otherwise difficult to estimate. A detailed mass breakdown for each configuration (integrated, composite bipropellant, and composite monopropellant) can be found in Table 27 to Table 30.

In order to size the subsystem appropriately, the integrated spacecraft nominal dry mass and propulsion module cases were determined assuming lower margins than are being assumed elsewhere in this study. The worst case spacecraft nominal dry mass and propulsion module cases were also calculated. For the Lisa Pathfinder PRM the dry mass was assumed to be 263 kg with a propellant mass of 1275 kg. From these values the subsystem hardware could be sized for a nominal case and tanks could be sized to accommodate larger amounts of propellant in case more propellant was required. A comparison of all the configurations is shown in Table 26.

Table 26 - Comparison of configuration cases assessed for the propulsion subsystem.

Option	Fuel [kg]	Ox [kg]	PRM mass [kg]	Spacecraft Nominal Dry Mass [kg]	20% Spacecraft Mass Margin [kg]	Launch Mass [kg]	Delta-V [m/s]
Integrated (Bi)	868	528	0	464	93	1953	3919
Composite (Mono)	84	0	1538	214	43	1879	577
Composite (Bi)	20	40	1538	225	45	1868	577

4.5.3 Propulsion system selection

The propellant needed for the integrated transfer was 1396 kg, and resulted in a propulsion subsystem dry mass of about 113 kg. This is less than half of the dry mass of the Lisa-Pathfinder PRM, and still produces a spacecraft that is over the 1900 kg launch mass allowable in the Rockot. For the monopropellant case 84 kg of propellant was required along with a 23 kg dry mass, which is comparable with the THEMIS ratio of 50 kg of propellant with a 12 kg subsystem mass.

The integrated option is the heaviest, as it provides propellant for the transfer as well as "stopping" and operations. Additionally, it impacts the AOCS subsystem moment of inertia of the spacecraft is very high due to the mass of the propellant subsystem, even when it is empty. The composite cases both reduce the size of the spacecraft significantly; however the difference between them is not very significant. Both composite spacecraft designs are under the 1900 kg maximum launch mass, demonstrating that the mission concept is preliminarily feasible. The monopropellant system, which requires more fuel but is less complex than the bi-propellant option, was chosen as the ideal configuration for this mission at the close of the HAGRID CDF study.

4.5.4 Future work for the propulsion subsystem

As mentioned previously in this section, there are areas of future work which can be completed as part of a further study or phase of this potential mission. The propulsion subsystem is the most ambiguous part, as many broad assumptions were made to enable the study to proceed. Improvements in the thruster and engine choices, as well as continuous power requirements would improve the design. The option to use sets of medium and small AOCS thrusters for drift-alteration and operational manoeuvres respectively is only one of the more detailed options that has not been considered in this study. Additionally, the difference between the two composite configurations is small for the subsystem mass budget, but the impact may be widely different in other subsystems. These two configurations, as well as a solid motor configuration, which was also not considered in this study, should be considered in more depth as the intricacies may provide a more definite conclusion.

Table 27 - Propulsion subsystem mass budget for the integrated 4 tank configuration from Astrium.

Integrated S/C 4 tank config	Item Mass [kg]	Number	Accum. Mass [kg]	Development Method	Margin [%]	Growth Margin [kg]	Estimated Mass [kg]	Total
engine	6.300	1	6.30	Off the Shelf	5	0.315	6.615	
oxidizer tank(s)	34.000	1	34.00	Off the Shelf	5	3.4	71.4	
fuel tank(s)	34.000	1	34.00	Off the Shelf	5	3.4	71.4	
pressurant tank	18.600	1	18.60	Off the Shelf	5	0.93	19.53	
Pressure and distribution equipment	8.300	1	8.30	Off the Shelf	5	0.415	8.715	
Pipework and Supports	5.870	1	5.87	New Design	20	1.174	7.044	
								113.304

Table 28 - Propulsion subsystem mass budget for the composite monopropellant configuration from Astrium.

Composite S/C Monoprop	Item Mass [kg]	Number	Accum. Mass [kg]	Development Method	Margin [%]	Growth Margin [kg]	Estimated Mass [kg]	Total
engine	3.000	1	3.00	Off the Shelf	5	0.150	3.150	
oxidizer tank(s)	0.000	0	0.00	Off the Shelf	5	0.000	0.000	
fuel tank(s)	8.070	1	8.07	Off the Shelf	5	0.404	8.474	
pressurant tank	3.890	1	3.89	Off the Shelf	5	0.195	4.085	
Pressure and distribution equipment	3.220	1	3.22	Off the Shelf	5	0.161	3.381	
Pipework and Supports	2.930	1	2.93	New Design	20	0.586	3.516	
								22.605

Table 29 - Propulsion subsystem mass budget for the composite bipropellant configuration from Astrium.

Composite S/C Biprop	Item Mass [kg]	Number	Accum. Mass [kg]	Development Method	Margin [%]	Growth Margin [kg]	Estimated Mass [kg]	Total
engine	6.300	1	6.30	Off the Shelf	5	0.315	6.615	
oxidizer tank(s)	3.580	1	3.58	Off the Shelf	5	0.179	3.759	
fuel tank(s)	3.580	1	3.58	Off the Shelf	5	0.179	3.759	
pressurant tank	3.610	1	3.61	Off the Shelf	5	0.181	3.791	
Pressure and distribution equipment	6.650	1	6.65	New Design	20	1.330	7.980	
Pipework and Supports	5.870	1	5.87	New Design	20	1.174	7.044	
								32.948

Table 30 - Summary of propulsion system analysis from Astrium.

Option	Volume per tank (litres)	Total Wet Prop System Mass (kg)	Propellant Mass (kg)	Dry Mass (kg)
Propulsion Module+Minisat (Monoprop), 1 Tank	151	110	84	26
Propulsion Module+Minisat (Biprop), 2 Tanks (1 Fuel, 1 Ox)	59	98	60	38
Integrated Spacecraft (Biprop), 4 Tanks (2 Fuel, 2 Ox)	393	1518	1395	123

[The report continues on the next page]

1.1 Attitude and Orbital Control System

Contents:

- AOCS requirements
- AOCS Design description
- AOCS Modes of operation
- Critical areas for the development of the AOCS

1.1.1 AOCS requirements

The AOCS subsystem is focused on designing and selecting appropriate hardware and configuration of hardware within the spacecraft to provide attitude and orbital control manoeuvres throughout the lifetime of the spacecraft.

AOCS.1 - The AOCS subsystem shall be able to control and repoint the spacecraft around the three axes. The spacecraft shall be able to re point each axis independently.

AOCS.2 - Whenever the spacecraft acquires an undesired angular momentum (momentum given to the spacecraft at the separation from the launcher or for any other unexpected momentum transfer), the AOCS subsystem shall be able to absorb the momentum within 600sec .

AOCS.3 - The maximum momentum that the AOCS subsystem shall be able to absorb shall be relative to an initial angular velocity of 10deg/s about each axis

AOCS.4 - The AOCS subsystem shall be able to control the firing direction 6 times.

AOCS 5 - To control the firing direction the initial attitude of the spacecraft should be considered to be 90 degrees away from the required firing direction. After the firing is completed the AOCS subsystem shall be able to let the spacecraft return to the nominal direction;

AOCS 6 - Once in the final orbit the AOCS subsystem shall be able to control the FOV (field of view) of the main payload with the required performances. The following tables summarize the pointing performances to match on the final orbit.

Table 31 – List of Absolute Pointing, Pointing Drift, and Relative Pointing Error along with Relative Pointing Error time.

Absolute Pointing Error x	1	deg
Absolute Pointing Error y	0.25	deg
Absolute Pointing Error z	1	deg
Pointing Drift Error x	1	deg
Pointing Drift Error y	0.25	deg
Pointing Drift Error z	1	deg
Relative Pointing Error x	0.01	deg
Relative Pointing Error y	0.01	deg
Relative Pointing Error z	0.01	deg
Relative Pointing Error time	7200	sec

1.1.2 AOCS Design Description

Consequently the AOCS subsystem shall be designed such that in the final orbit the spacecraft points to the SUN at any time with an accuracy of 1deg, 0.25deg and 1 deg around X,Y and z axis respectively. Within a time frame of 7200sec the spacecraft shall be stabilized within 0.01deg of accuracy.

This implies that the design is driven by two different bandwidths.

For the first (low frequency bandwidth) ranging from 0Hz to 1.39E-04 Hz the accuracy is 1deg, 0.25deg and 1deg around X,Y and Z axis respectively.

For the second ranging from 1.39E-04 Hz upward, the accuracy is 0.01deg on each axis.

Consequently the maximum angular velocity around each axis should be 1.39E-06 deg/s

1.1.3 Modes of operation

Figure 19 - Diagram of the AOCS states for the HAGRID spacecraft.Figure 19 shows a simplified diagram of the state machine of the AOCS subsystem and Figure 20 shows the AOCS subsystem architecture accounting for sensors, on-board intelligence, and actuators.

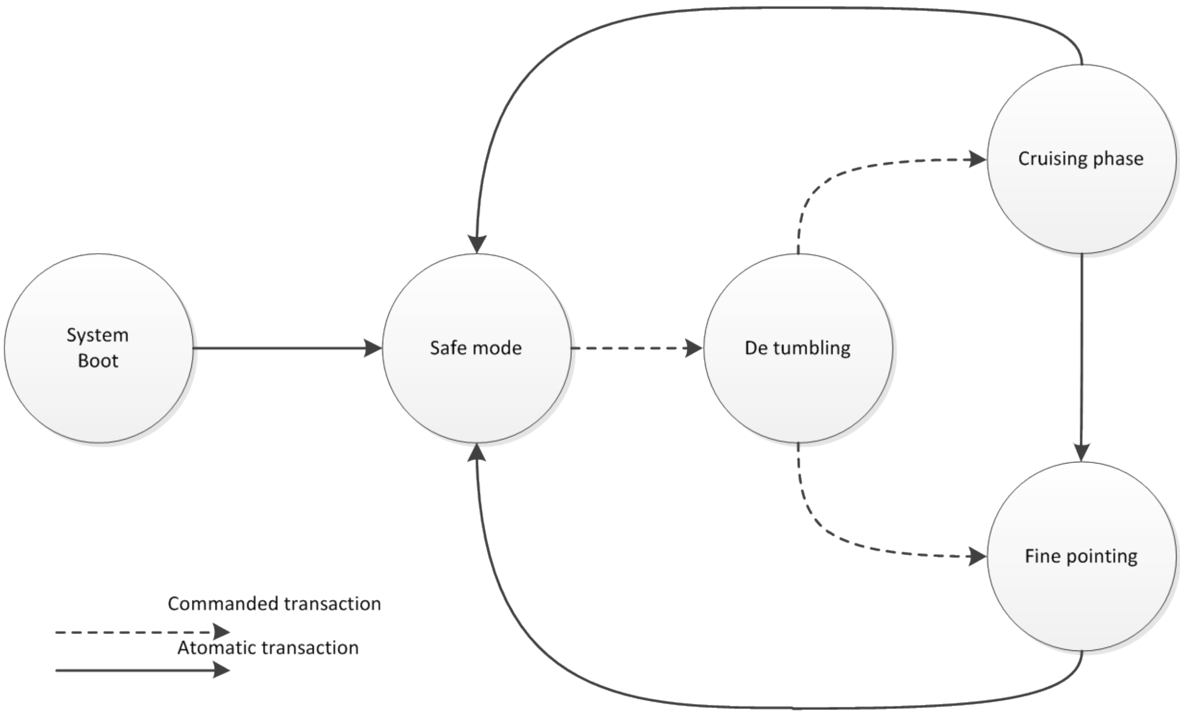


Figure 19 - Diagram of the AOCS states for the HAGRID spacecraft.

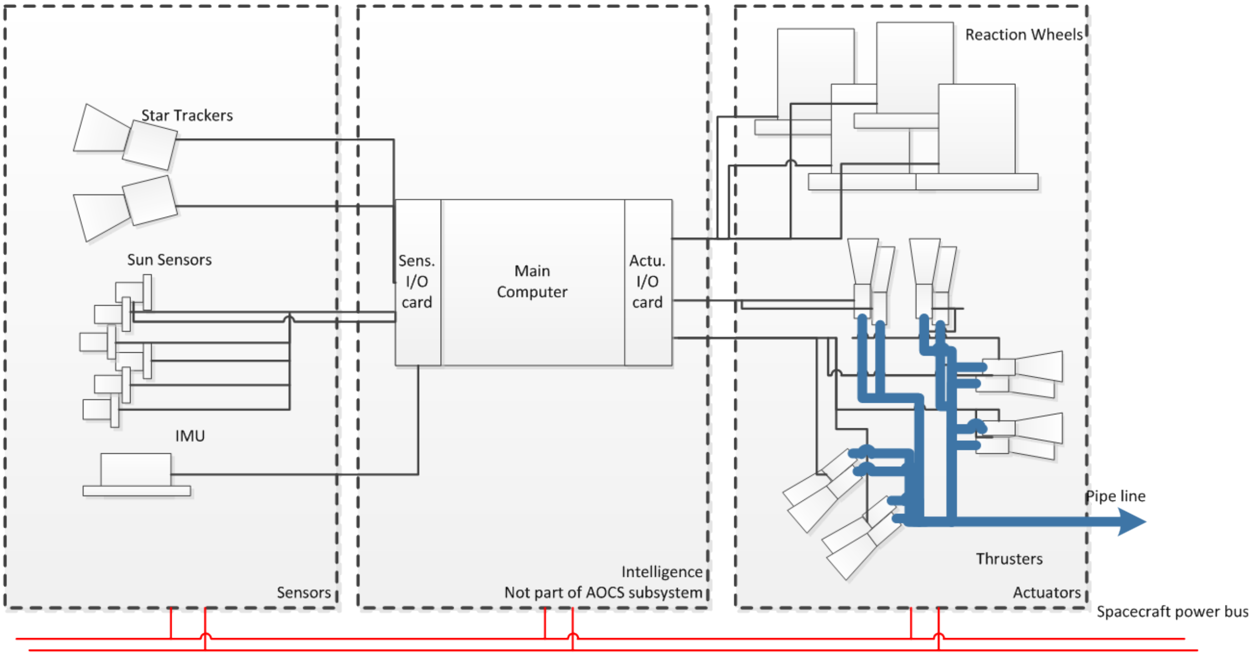


Figure 20 - Overall architecture of the AOCS subsystem.

4.6 Power

Power subsystem Contents:
- Power subsystem design strategy
- Power budget
- Critical areas for further development

4.6.1 Power subsystem design strategy

From the preliminary review of the HAGRID spacecraft and mission at the commencement of this study, a number of design requirements were discussed regarding the HAGRID power subsystem and two key phases were identified as being critical design drivers:

The heliospheric orbit required for the HI instrument requires transfer from Earth-centred to Sun-centred orbit through apogee-raising manoeuvres using the propulsion subsystem. During the transfer burn the spacecraft must rotate, using AOCS thruster firings, to align the apogee motor with the direction of burn. Throughout this period the spacecraft would be without direct solar flux and will discharge its batteries in order to supply power to the major subsystems (approx. time: 10 mins AOCS + 10-30 mins apogee burn + 10 mins AOCS) This may also coincide with Earth eclipse in LEO, which may take to up to 60 minutes in fundamental eclipse. The large power loads required by the AOCS thrusters and apogee motor during this period for the ‘first burn’ case would typically determine the size of the whole solar array.

A second issue introduced by the final orbit was the requirements for the Communications subsystem at a large (>60°) Earth-Sun angle in heliospheric orbit at EOL. A key consideration was to examine the power loads required for the communications downlink at EOL with coincident operation of the payload subsystem. In heliospheric orbit, the spacecraft receives a constant solar flux, but still has the potential to discharge its batteries during large power loads such as during the communications downlink. The large distance between the ground antenna and the spacecraft at EOL leads to large free-space losses in the link budget. As a consequence, the power consumption to maintain a satisfactory SNR is large. The communications downlink scheduling would have a key impact on the power subsystem design and requirements for secondary power while on-station.

A body-mounted solar array is desired in order to reduce single-point failure of deployment mechanisms. Initial considerations looked at a deployable solar array in addition to a body-mounted one, but further analysis of the power requirements and consideration of the structural design allowed a single, body-mounted array to be adopted. Off-the-shelf and modified-off-the-shelf components are desired to maintain flight heritage and limit the requirements for pre-flight testing.

Two power subsystem designs were developed during the HAGRID study, relating to the initial ‘integrated’ case of the spacecraft with its own propulsion subsystem and apogee motor, and the ‘composite’ case in which a stand-alone spacecraft is used in combination with the LISA Pathfinder propulsion module.

Throughout the design conservative margins at both equipment and system level were used.

4.6.2 Power Budget

For each scenario (Integrated and Composite spacecraft), four primary power load cases were identified and examined. In each case, operation of each subsystem component could be selected as operating at ‘Peak’, ‘Maximum’, ‘Nominal’, ‘Off’ or ‘Standby’. Consideration of the various power cases involved examining each component required for that phase and adopting the appropriate operating level.

BOL (Beginning-of-Life) considers the case when the spacecraft has been inserted into LEO and is engaging in LEOP. The main payload (the HI instrument) remains on standby, communications are performed by the LGA while the HGA is on standby. This case represents the minimum power loads experienced during the lifetime of the mission. Operation here is considered to be powered by the solar array and from batteries during eclipses. CDMS operates nominally while a hot redundant OBC is on standby.

Orbit Transfer represents operation of the propulsion module for apogee-raising manoeuvres while in LEO. Communications still operate on the LGA with the HGA on standby but with the EPC running at nominal levels. All AOCS components are on nominal operation, with the 1 N CHT thrusters operating at a maximum power of 15.9 W for rotation of the spacecraft prior to and during the apogee burn. The payload remains on standby at this time. The 400N apogee motor is in peak operation. Since not all AOCS thrusters fire coincidently, a duty cycle of 50% is assumed in this case to account for the thruster firing control for the approximate 10 minutes either side of the apogee burn: At this time, the thruster control cycle has not been modelled, so this assumption regarding the length and frequency of thruster operation may be considered as very conservative.

Payload Only represents the case where the spacecraft is in heliocentric orbit and the HI Payload is operating nominally. Following orbital insertion the LGA is switched off and during payload operation only, the HGA is off. The receiver is maintained at nominal power levels at all times and AOCS components operate at nominal power levels in order to maintain pointing. The batteries are charged during this time to store solar energy for the communications downlink.

Payload + Comms represents the phase of operation where downlink is occurring (approximately one-quarter of the time). The main payload remains operational and all other subsystems are as above for the ‘Payload Only’ case with the exception of the HGA which is active along with the antenna, amplifier, transmitter, receiver, EPC and other electronics which operate nominally. Operation of the downlink antenna means that it transmits once every four hours for one hour while on standby for three hours, putting a large load on the solar array and secondary power during transmission.

The power budget for the ‘Integrated’ case is shown in Table 32.

	Orbit Transfer	BOL	Payload Only	Payload + Comms
Numbers include equipment level margin but not system margin				
AOCS	142.12 W	16.75 W	41.95 W	41.95 W
CDMS	46.20 W	46.20 W	46.20 W	46.20 W
Comms	21.00 W	21.00 W	21.00 W	162.75 W
Payload	18.90 W	18.90 W	26.25 W	26.25 W
Propulsion	47.25 W	0.00 W	0.00 W	0.00 W
Power	36.52 W	36.52 W	36.52 W	36.52 W
Structures	0.00 W	0.00 W	0.00 W	0.00 W
Thermal	36.00 W	36.00 W	36.00 W	36.00 W
TOTAL	347.99 W	175.37 W	207.92 W	349.67 W
TOTAL with system margin	382.79 W	192.90 W	228.71 W	384.63 W

Table 32 - Power budget for the four primary power load cases in the integrated spacecraft scenario

For the scenario where the LISA Pathfinder propulsion module is coupled with the main HAGRID spacecraft, the associated power budget for these four power cases are presented in Table 33. For this scenario, adoption of the LPF solar array was considered, with modification to its shape and design to match the dimensions and launcher envelope of the HAGRID spacecraft.

	Orbit Transfer	BOL	Payload Only	Payload + Comms
Numbers include equipment level margin but not system margin				
AOCS	142.12 W	142.12 W	41.95 W	41.95 W
CDMS	44.10 W	44.10 W	44.10 W	44.10 W
Comms	152.25 W	152.25 W	21.00 W	162.75 W
Payload	18.90 W	18.90 W	26.25 W	26.25 W
Propulsion	189.00 W	105.00 W	0.00 W	0.00 W
Power	36.52 W	36.52 W	36.52 W	36.52 W
Structures	0.00 W	0.00 W	0.00 W	0.00 W
Thermal	36.00 W	36.00 W	36.00 W	36.00 W
TOTAL	618.89 W	534.89 W	205.82 W	347.57 W
TOTAL with system margin	680.78 W	588.38 W	226.40 W	382.32 W

Table 33 - Power budget for the four primary power load cases in the composite spacecraft scenario.

The data presented in Table 32 and Table 33 show that the orbital transfer and downlink phases of operation are the main drivers of the power subsystem, with similar total power loads active. These requirements led to the design of a power system comprising of a body-mounted solar array and secondary batteries that operate during transfer and downlink.

The power requirements are significantly greater for the LPF propulsion module case, showing power loads 300 W higher during orbital transfer than the integrated case. However, where the integrated case requires a series of motor burns to raise the apogee of the spacecraft's Earth orbit into heliospheric orbit, the LPF scenario requires a single orbital insertion burn to transfer to Solar orbit, requiring a larger load for a shorter burn phase.

AZUR SPACE Ga-AS triple-junction solar cells (TJ Solar Cell 3G30C) were selected for the solar array of the integrated spacecraft, covering an area of 1.49 m^2 of the Sun-facing body and generating 402.07 W of power. At 30%, cell efficiency for the array is high with a typical operating temperature of -150 – 110 °C. The final mass of the cell (using empirical calculations) was estimated to be 6.28 kg, with 5 kg added to account for any array fixtures/mechanisms. Details of the primary power design are presented in Table 34 in addition to the equivalent design for the 'composite' spacecraft case.

	Integrated	Composite
Cell Type	GaAs Triple Junction	
Cell Conversion Efficiency	30 %	
Cell Packing Efficiency	90 %	
Solar Cell Unit Mass	0.86 kgm^{-2}	
Cover Glass Unit Mass	1.36 kgm^{-2}	
Circuity/Adhesives Unit Mass	0.61 kgm^{-2}	
Mechanisms Unit Mass	1.39 kgm^{-2}	
Number of Arrays	1	
Mean Temperature	28 °C	
Efficiency Gradient per Degree	0.02 %/°C	
Operating Temperature	-150 - 110°C	
Mass	6.28 kg (+5 kg)	9.69 kg (+9 kg)
Solar Array Area	1.49 m^2	2.3 m^2
Generated Power	402.07 W	944.80 W
Dimension (x)	1000 mm	1000 mm
Dimension (y)	1490 mm	2300 mm
Dimension (z)	100 mm	100 mm

Table 34 - Primary power system; integrated and composite case

Secondary power generation is achieved through ABSL Lithium-Ion batteries (ABSL18650HC) using 144 cells in a configuration of 8s-18p (18 parallel strings of 8 cells each) with a cell voltage of 4V and 20% depth-of-discharge. The size of the batteries was estimated using BEAST (Battery Electrical Analysis Software Tool) for the ABSL18650HC. The total battery mass calculated was 7.11 kg, providing a theoretical capacity and theoretical EMF of 13.50 Ahr and 25.20 V respectively. Data for the battery sizing for the integrated and composite cases are presented in Table 35.

	Integrated	Composite
Battery Type	Li-Ion	
Bus Voltage	28 V	
Shunt Dump Efficiency	95 %	
Harness Efficiency	99 %	
Battery Charging Efficiency	92 %	
Battery Discharging Efficiency	90 %	

Cell Voltage	4 V	
Depth of Discharge	20 %	
Energy Density	155 Whr/kg	155 Whr/kg
No. Cells in Parallel	18	18
No. Cells in Series	8	8
Theoretical Capacity	13.50 Ahr	59.60 Ahr
Theoretical Energy	291.60 Whr	1668 Whr
Maximum EMF	25.20 V	25.20 V
Battery Mass	7.11 kg	7.8 kg

Table 35 - Secondary power system; integrated and composite case

The design of the power subsystem was completed with the selection of the Power Conditioning Unit and Array Bus Regulator. The bus voltage was selected as 28 V, centrally-regulated. The mass and volume characteristics of the integrated and composite cases for the entire power subsystem are presented in Table 36 and Table 37.

component name	ID	Number	development status	Mounting Considerations	Mass (kg)	x (mm)	y (mm)	z (mm)	shape
Power Conditioning Unit	POW-PCDU-002	1	Modified Off the Shelf	PCU mount info	8.3	67	238	158	box
Array Bus Regulator	POW-PCDU-003	1	Off the Shelf	Array Bus Regulator mount in	0.55	193	150	24	box
Solar array1	N/A	1	Modified Off the Shelf	Solar Array 1 mount info	6.28	1000	1490	100	plate
Solar array2	N/A	0	Modified Off the Shelf	Solar Array 2 mount info	5	0	0	0	N/A
Batteries	N/A	1	Modified Off the Shelf	Batteries mount info	7.11	N/A	N/A	N/A	box
				TOTAL	27.24				

Table 36 - Components for integrated case

component name	ID	Number	development status	Mounting Considerations	Mass (kg)	x (mm)	y (mm)	z (mm)	shape
Power Conditioning Unit	POW-PCDU-002	1	Modified Off the Shelf	PCU mount info	8.3	67	238	158	box
Array Bus Regulator	POW-PCDU-003	1	Off the Shelf	Array Bus Regulator mount in	0.55	193	150	24	box
Solar array1	N/A	1	Modified Off the Shelf	Solar Array 1 mount info	9.69	1000	2300	100	plate
Solar array2	N/A	0	Modified Off the Shelf	Solar Array 2 mount info	9	0	0	0	N/A
Batteries	N/A	1	Modified Off the Shelf	Batteries mount info	7.80	N/A	N/A	N/A	box
				TOTAL	35.34				

Table 37 - Components for composite case

4.6.3 Critical areas for further development

Areas which are critical for further development of the power system would include in-depth modelling of the power loads and budget during the orbital transfer phase: At present, the power profile during this phase is calculated very simply and would benefit from development of a power-time profile. This would model the exact timing of rotation of the spacecraft, thruster firings and periods of eclipse. Accurate computation of these data would result in a considerable reduction in uncertainty of the required power level required, since components like thruster firings are crudely estimated by assuming only 50% are active at any given time, whereas the number of concurrent active thrusters may be significantly less than this. Since this phase is a main driver for the primary power system design, increased analysis of this phase may yield improved results and reduce battery requirements. This would include modelling of the charge and discharge cycles and power profiles using software such as BEAST. At present, no detailed power profile for the battery-charging/downlink cycle has been modelled to estimate the effects on the secondary power generation for the communications downlink.

No consideration has yet been made of performance degradation, or analysis of temperature effects on power generation (e.g. examining the I-V power performance). This may be crucial to the design and require modifications to the solar cell area and battery design to accommodate the resulting loss of performance at EOL.

At present, the structural design for the antenna mounting characteristics have not been studied in-depth. An estimated figure of 5 kg has been placed on the mass requirements of the array to account for mounting component(s), but this estimate may be improved with further analysis of the structural requirements.

Adoption of the LPF propulsion module here has assumed subsequent re-use of the solar cell technology used on LISA Pathfinder. The use of the LPF solar panel, with the array 'cut down' to match the size of the Sun-pointing face of the HAGRID spacecraft was discussed. Further considerations should be made of the feasibility of this and of adopting the full LISA Pathfinder power subsystem, with some modifications. An analysis of modified LFP panels is show in Table 38.

Solar cell analysis using an improved LPF W/m^2 of 203W/m^2 assuming 30% cell efficiency, and draft report data

	Cell Efficiency	SA Pwr		SA Area		Power / m^2		Mass		Mass / m^2
	28%	650	W	3.8	m^2	171	W/m^2	22.6	kg	5.95
Integrated - LPF scaled array mass/power using solar array array area in report	30%	309	W	1.49	m^2	203.0	W/m^2	8.9	kg	
Composite - LPF scaled array mass/power using solar array array area in report	30%	476	W	2.3	m^2	203.0	W/m^2	13.7	kg	
Integrated - Required array area and mass using max required solar array power in report	30%	**385**	**W**	**1.9**	m^2	203.0	W/m^2	**11.3**	**kg**	
Composite - Required array area and mass using max required solar array power in report	30%	**681**	**W**	**3.4**	m^2	203.0	W/m^2	**20.0**	**kg**	

Summary	**Old Mass**	**New Mass**	**Mass Increase (kg)**
Integrated Array should be 1.9m^2 and 11.28kg (in report - 1.49m^2 and 11.28kg (6.28kg+5kg))	11.28	11.28	0.0
Composite Array should be 3.4m^2 and 19.95kg (in report - 2.3m^2 and 18.69kg (9.69kg+9kg))	18.69	19.95	1.3

Table 38 - Astrium's analysis of LFP solar panels

4.7 Communications

Communications Contents:
- Communications requirements
- TMTC uplink and downlink budgets, data rate requirement and antenna sizing
- Design description of the Communications system and architecture
- Critical areas for the development of the communications system

4.7.1 Communications requirements

The design drivers for the communication subsystem requirements were;
- To have a sufficient link-budget at EOL to be able to downlink the required data volume with sufficient latency
- That the high gain antenna fits inside the launch vehicle
- That the power supply has the capacity to serve the communications system while having a solar array small enough to fit on the spacecraft
- A bit rate error of 10^{-5} or less

The following section describes the initial design of the communication subsystem. The sizing of components was optimised using the data rate requirement from payload and housekeeping telemetry requirement from each subsystem. Initially the sizing was made for the drift scenario (maximum distance approximately 2 AU) but later on the subsystem was optimised for the spacecraft stopping at L4. The subsystem was designed using the parametric link budget equation and Astrium's link budget tool. The main trade-off between the size of the antenna and the power requirement is also presented.

4.7.2 Link Budget

The link budget was analysed using Astrium's link-budget tool as well as RAL Space's software based on parametric equation and heritage data from the STEREO mission. Multiple data rates, and ground stations were tested during the study. Finally Chilbolton was selected as a reference ground station (as described in the Ground Segment section). Data from a future X-band upgrade was used which quotes G/T to be 37 dB/K.

The payload data rate was optimised by considering the minimum data required for an accurate determination of the CME trajectory. Based on data from STEREO, Figure 20 shows the field of view of the instrument projected onto the Sun-Earth line. Overlain on the standard field of view is the reduced frame which is considered the minimum sufficient to predict the propagation of the CME.

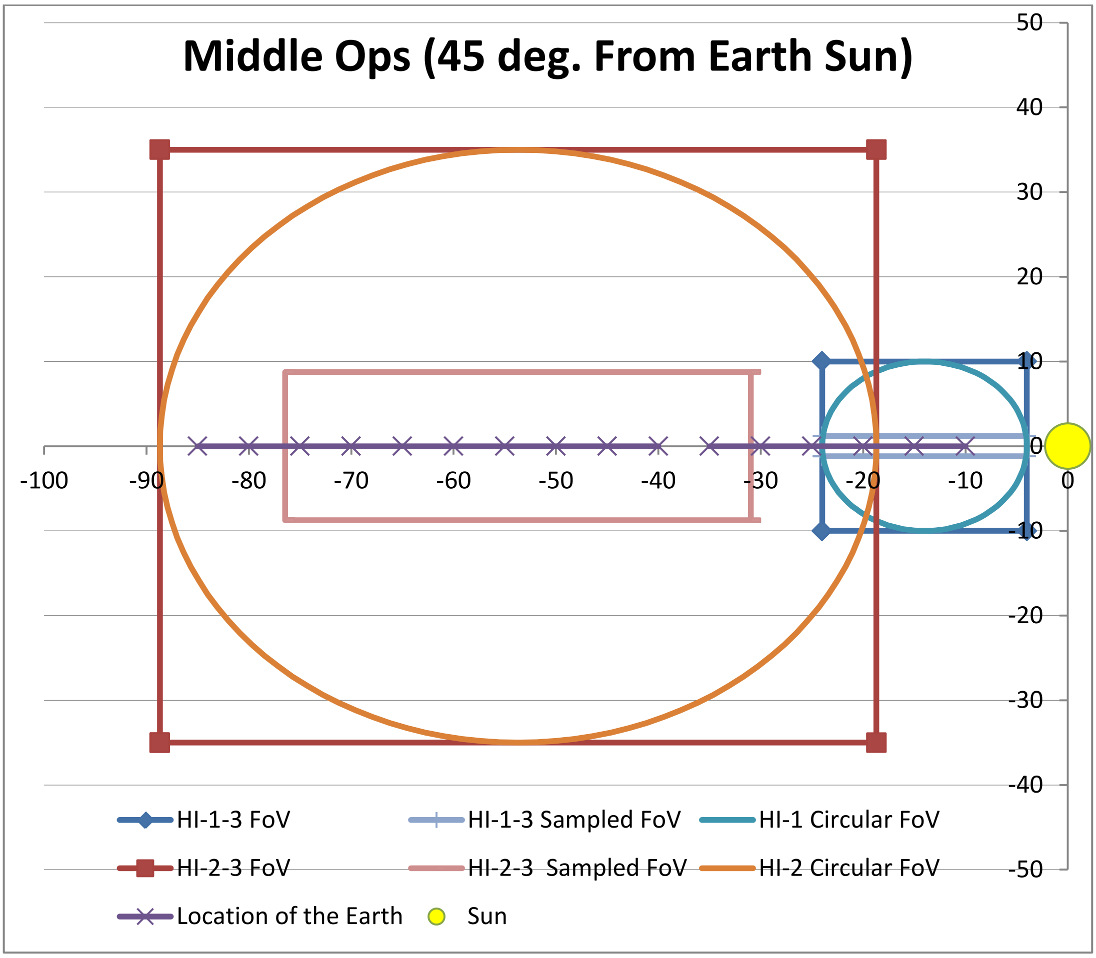


Figure 21 - HI-1 and HI-2 Field of View (FoV) with the position of the Earth ploted as a function of the spacecraft-Earth separation angle (x's). Also shown are the reduced FOV considered useable for CME tracking

Various cases were tested for a drift scenario where the spacecraft constantly moves away from Earth up to 2 AU in distance (i.e. 90 degrees separation). Figure 22 shows the worst latency case of a drifting spacecraft as well as the optimal latency case for an L4 stationary spacecraft.

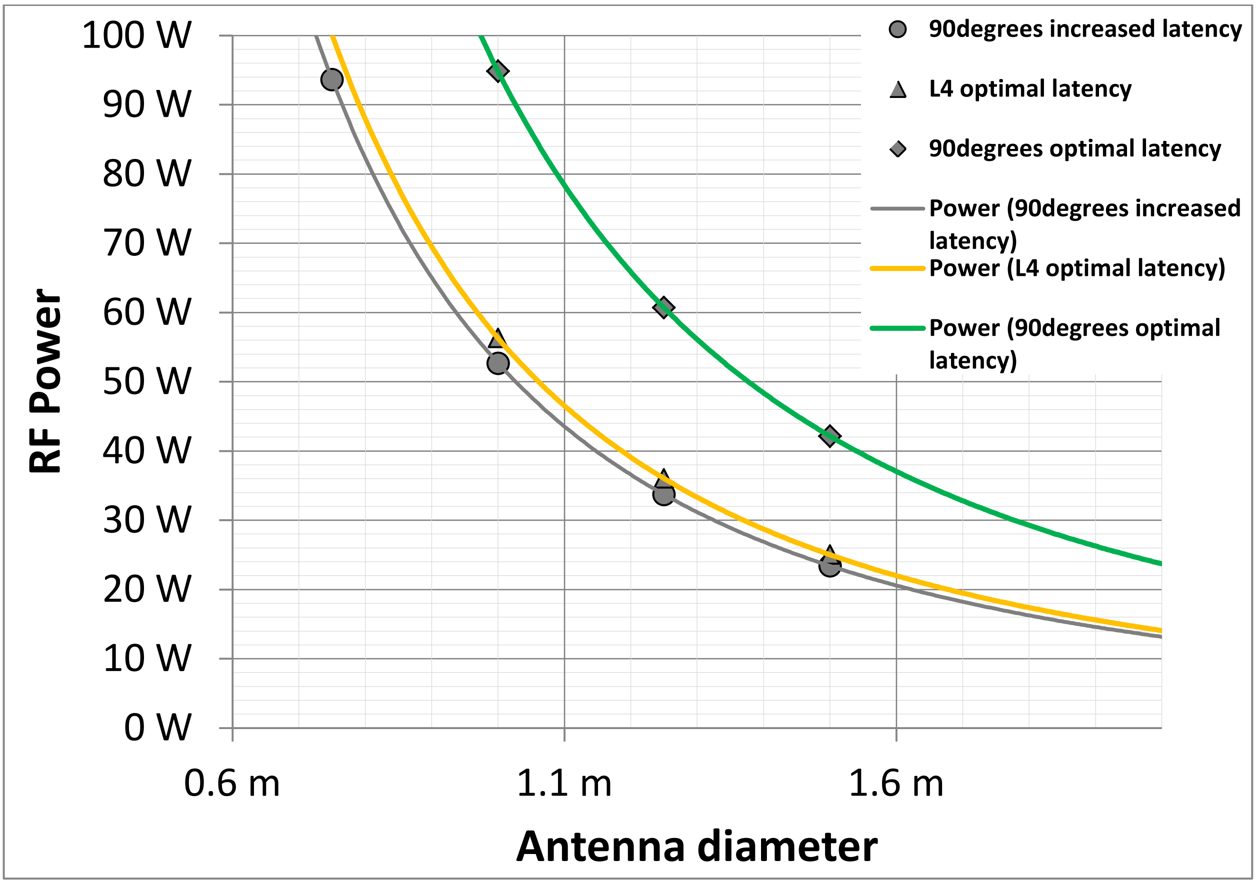


Figure 22 – RF power against antenna diameter for L4 position with the required latency (orange) as well as the furthest away point (2 AU) for the drifting scenario with required latency (green) or a constant low data rate download which would increase the latency (gray)

Finally the link budget was calculated for an L4 stationary spacecraft, with the optimised data rate and latency requirement, which would use the upgraded Chilbolton antenna. Figure 23 shows the losses in the link budget equation where the major loss is the free space loss due to the distance between the spacecraft and the ground station.

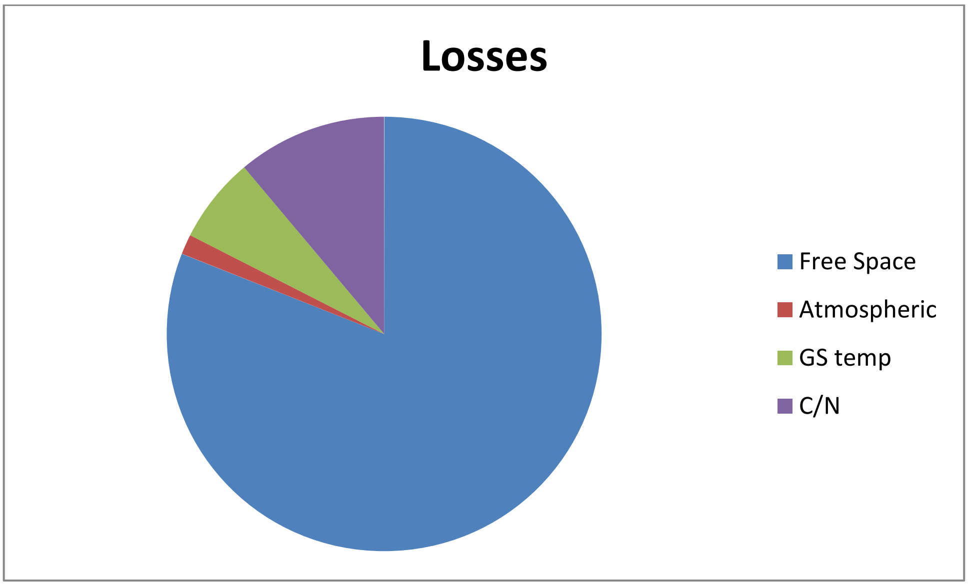


Figure 23 – Breakdown of losses in the link budget

Summary of link budget:

- 10^{-5} was used for bit error rate
- 37 dB/K ground station performance was assumed which can be achieved using an upgraded X-band receiver at RAL Space's Chilbolton facility
- The diameter of the high gain antenna dish is 1 m
- 56.3 W of RF power is needed for downlink which is provided by a 120 W EPC
- That a worst-case alert scenario with a latency of 7.2 hours can be achieved

4.7.3 Architecture

Figure 24 shows the TT&C architecture for the X-band system on-board HAGRID. All systems are double redundant and the primary and secondary system can be switched between the high gain and low gain antenna. The block diagram shows the data connections between the modules (TX red, RX blue) and waveguides (purple). The low gain antenna is only used during the commissioning phase on Earth orbit and before and after orbit transfer manoeuvres.

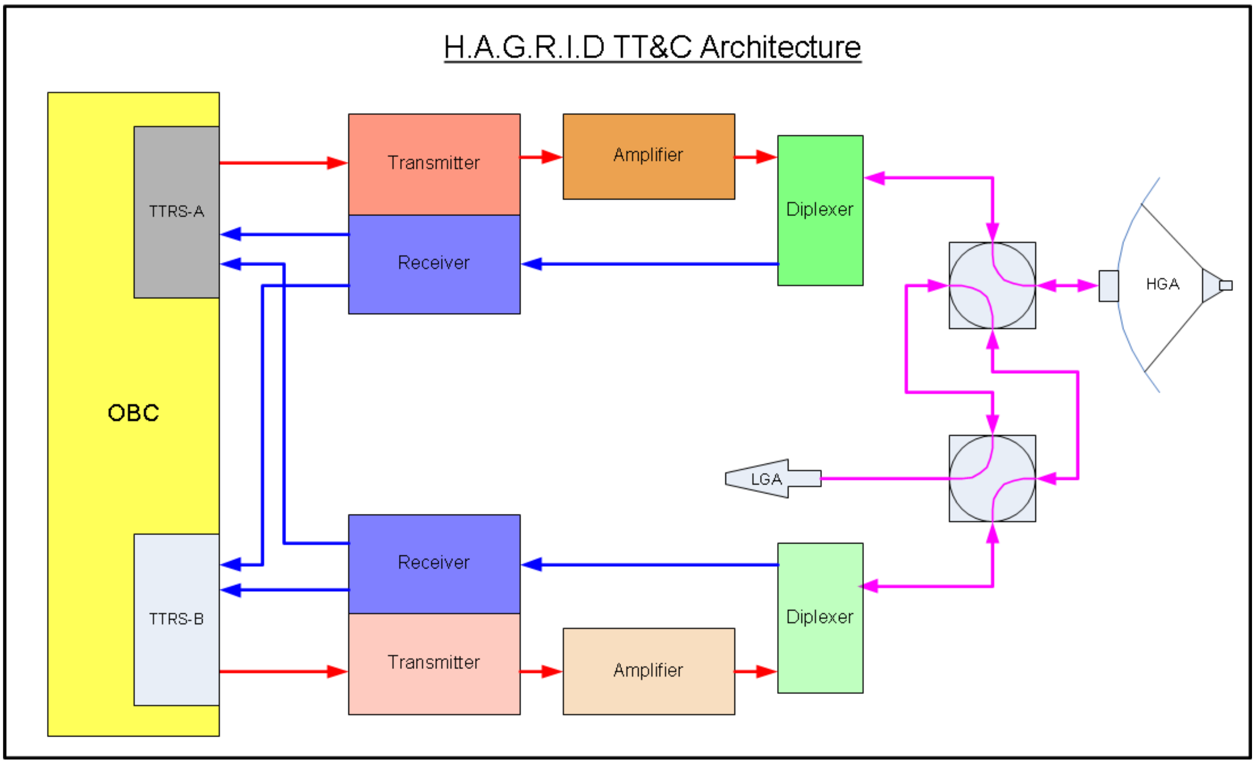


Figure 24 - TT&C architecture for HAGRID mission

The on board computer (OBC) design is not detailed in this study although size and mass were estimated using heritage data. Similarly the cost was estimated for the OBC hardware and code. Encoding and decoding of telecommands is performed within the TT&C subsystem and not by the OBC. For storage, double redundant solid-state storage is used with a size of 1 GB per disk.

4.7.4 *Areas of development*

While the link-budget calculations were reasonably detailed for a pre phase-A study only heritage data were used for the performance of the hardware components. Therefore it is suggested to determine the performance of a bespoke RF system designed for the HAGRID mission. The calculated efficiency of the hardware components could potentially decrease the required power for the communication system. Furthermore, with the bespoke system in mind, the pointing accuracy of the high gain antenna should be revisited considering the use of the antenna itself as an external reference sensor to increase precision. Regarding the compression and coding of the transmitted data it has been suggested that the use of turbo coding would improve the link budget and thus it should be studied in more depth.

4.8 Structure

Structures Contents:

- Structures requirements
- Structure design strategy
- Structural design and configuration of spacecraft
- Future subsystem work

4.8.1 Structures requirements

The main requirements for the structure of the spacecraft and configuration of internal parts are to:

- Provide structure for the payload (the HI instrument)
- Accommodate all necessary subsystems
- Accommodate propulsion system capable of delivering the spacecraft to its final orbit, or provide interface to an additional propulsion module
- Fit inside a *Rokot* or a similar size launch vehicle.
- Survive launch loads imposed by launch vehicle

Areas not considered in the HAGRID study were:

- Structural model and finite element model for launch load analysis
- Interface requirements for subsystem
- Detailed interface requirement for propulsion module option

4.8.2 Structures design strategy

HAGRID is a proof-of-concept for an economically viable space weather monitoring infrastructure, therefore one of the main aims is to reduce the mass and size of the spacecraft in such a way that it can be delivered to the right orbit by a piggy-back launch, or by using a smaller, less expensive, launch vehicle.

During the early stage of the study the use of an auxiliary payload slot on an ARIANE 5 or similar size launch vehicle had been considered. It quickly became apparent that the propulsion system required to deliver the spacecraft to its final orbit as well as the size of the high gain antenna and the HI instrument itself meant that a dedicated launch was required.

The Russian *Rockot* has been identified as a possible dedicated launch vehicle and the mass budget has therefore been deduced from the lift capability of the *Rockot*. A CAD model of the fairing was also produced to aid in checking configuration volumes, as seen in Figure 26.

When designing a spacecraft that could be launched in the Rockot launcher, two configuration options were looked at during the study. The *integrated* option carried the propulsion system within the spacecraft while the *composite* option used an off-the-self propulsion module attached to the spacecraft. This module has been developed for LISA Pathfinder by Astrium and a detailed solid model has been used for the HAGRID study.

4.8.3 Structural design of spacecraft

Due to the significant differences between the integrated and composite spacecraft configurations, the subsystem configurations had to be looked at in detail. During the study the use of an off-the-shelf propulsion module has been suggested and both models were iterated and refined. Figure 25 shows both options including the payload, the HI instrument, that is shown in black.

An empirical model was used to determine the thickness and mass of the spacecraft panels based on a representative cylinder. Comparison of these figures with the mass of the panels from the Solid Edge software package showed this to be correct.

The 4-tank propulsion system configuration was used in the integrated spacecraft to balance the moment of inertia of the spacecraft as propellant is being used. The oxidizer/fuel ratio chosen is used by Astrium to make the volume of the oxidizer tank similar to the fuel tank, and fuel rates drain propellant in relatively equal amounts from the two oxidizer tanks and two fuel tanks.

In the composite configuration, the high gain antenna uses a mechanism to fold into a stowed position for launch, while the integrated configuration has enough room for a rigidly mounted antenna.

Initially, a deployable solar array was considered. The solar arrays are mounted on a 'rear' spacecraft panel for the integrated case, and for the composite case they are mounted on the 'top' to allow power generation during launch and transfer orbit injection.

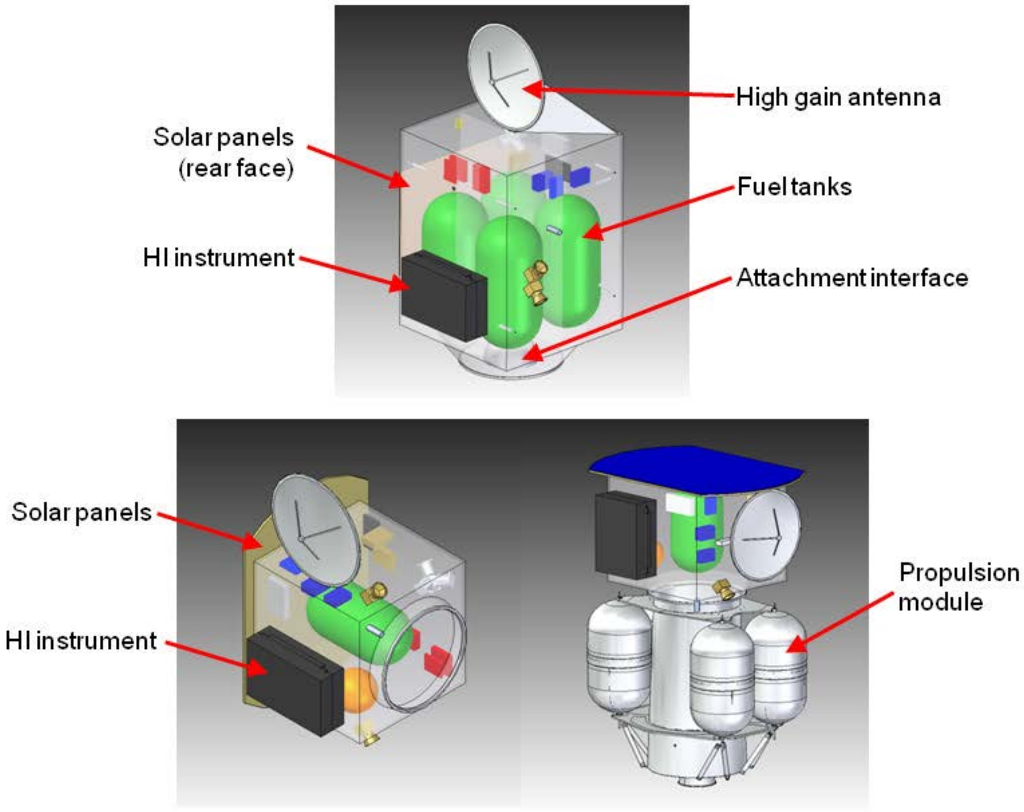


Figure 25: Overview of both integrated spacecraft (top) and composite spacecraft (bottom left) and the composite spacecraft with the propulsion module attached (bottom right)

It was found for both cases that it is possible to design a spacecraft bus capable of supporting all subsystems which fits within the envelope of the launch vehicle. Figure 26 shows the integrated option inside the Rokot launch vehicle fairing while Figure 27 shows the composite option similarly inside the same fairing.

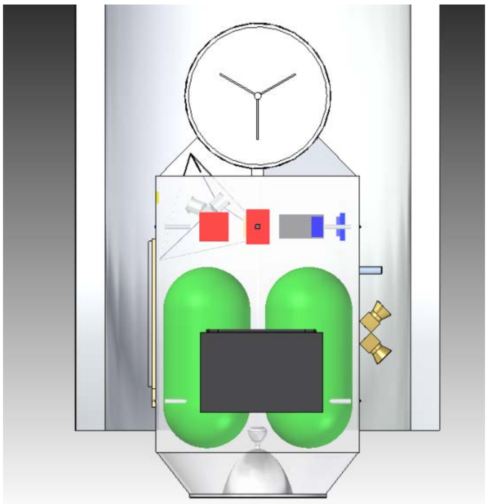


Figure 26: Integrated option within the launch vehicle envelope.
note: some small details on the inside of the faring are not shown. Using the Rokot manual it has been confirmed that the launch vehicle can accommodate the integrated configuration option

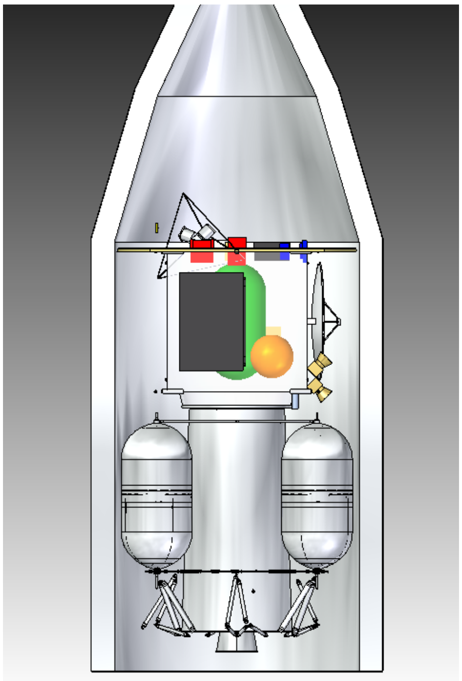


Figure 27: Composite option within the launch vehicle envelope.
note: some small details on the inside of the faring are not shown. Using the Rokot manual it has been confirmed that the launch vehicle can accommodate the composite configuration option

4.8.4 Future structures subsystem work

While the structural model and finite element model for launch-load analysis were not included in this study, they should be considered if the HAGRID mission proceeds to a Phase-A study. Specific areas, such as the high gain antenna mount, multi-shot HI door, and a detailed propulsion module interface for the composite model should be assessed.

Additionally, while the volumes of all subsystem elements were represented in the spacecraft, more work should be put into determining exact locations, and constraints. This includes incorporating elements such as propellant feed pipes, stringers, and other necessary support structures.

4.9 Thermal

Thermal Contents:

- Thermal requirements
- Thermal modelling
- Description of the thermal control system
- Thermal analysis cases
- Thermal predictions
- Critical areas for Thermal
- Thermal conclusions
- Future thermal subsystem work

4.9.1 Thermal requirements

There are two equally important requirements of the HAGRID spacecraft thermal design; one is to provide a stable thermal interface for the HI instrument (the payload) and the other is to ensure that the various components housed within the spacecraft are maintained within their operational and non-operational temperature ranges. These are summarised in Table 39 below.

Table 39 - Principal Thermal Requirements

Unit	Operational Temperature Range
HI Thermal Interface	-35°C to +45°C
Solar Array	0°C to +110°C
Spacecraft Batteries	-10°C to +15°C

Although additional requirements exist, those presented above represent the bounding cases in the thermal design. It became apparent early on in the study that the key areas would be preventing the solar arrays from over-heating and the batteries from getting too cold.

4.9.2 Thermal Modelling

A 40-node thermal model was created during the study and updated as the spacecraft design progressed. HI was modelled with 10 nodes and the remaining 30 nodes modelled the spacecraft and on-board subsystems. ESATAN images of the geometric model are shown in Figure 28.

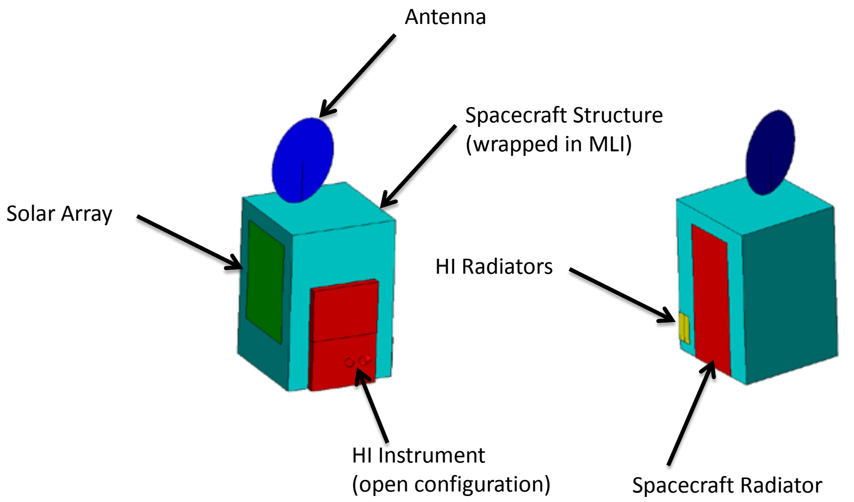


Figure 28 - ESATAN images of the Geometric Mathematical Model

4.9.3 Description of thermal control system

Each panel of the spacecraft will be wrapped in MLI to minimise the absorbed solar energy. The solar pointing face will house the body mounted solar array. The anti-sun face will house the spacecraft main radiator and the separate instrument radiators. The radiators will be painted black to maximise the heat rejection. A summary of the radiator areas is shown in Table 40.

Table 40 - Estimated Radiator Areas

Radiator	Area (m2)
Spacecraft	1
HI Camera 1	0.05
HI Camera 2	0.05

Survival heaters will be required on the various internal components to maintain them above the minimum non-operating temperatures.

The thermal design of the HI instrument is not considered in detail although it is modelled approximately in order to gauge likely power budgets and interface temperatures.

4.9.4 Thermal Analysis Cases

Two analysis cases were considered in detail in order to capture the full range of environmental conditions for the HAGRID spacecraft. The hot operational case considers the nominal operational scenario but with worst case radiator performance and warmest environmental conditions (solar constant etc). The hot non-operational case considers a safe mode case where the spacecraft pointing cannot be guaranteed and the spacecraft radiators are solar pointing.

4.9.5 Thermal Predictions

Model predictions for the two cases are presented below in Table 41.

Table 41 - TMM Predictions

	Component	Predicted Temeparture (°C)	
		Nominal Case	Hot non-Operational Case
SPACECRAFT	+Z Panel	9.2	66.1
	+X Panel	18.5	55.04
	+Y Panel	4.3	63.15
	-X Panel	-7.2	84.52
	-Y Panel	5.8	66.88
	-Z Panel	4.8	66.21
	Solar Array	97.1	-29.86
	AOCS SUbsystem	38.1	55.04
	Comms Subsystem	29.4	66.1
	Power Subsystem	-7.2	84.52
	Propulsion Subsystem	5.3	67.4
	CDMS Subsystem	20.9	84.52
	Antenna	37.2	63.24
	Instrument Structure	-29.3	-19.18
HI	HI1 Camera Baffle	-24.1	-8.49
	HI1 CCD	-65.9	0.47
	HI1 Rad	-66.8	0.67
	HI1 Optics	5.6	-8.49
	HI2 Camera Baffle	-7.9	-22.13
	HI2 CCD	-65.9	0.58
	HI2 Rad	-66.8	0.78
	HI2 Optics	7.2	-19.18
	Instrument Cover	-52.7	-49.98
	Instrument Cover MLI	-52.7	-49.98
	Camera Electronics Box	-4.1	-19.18

The model predictions indicate that the required spacecraft temperatures should be achievable with the proposed thermal control system. In addition to the analyses presented, an estimation of cold case survival powers was made. The coldest non-operational case would be in the nominal pointing scenario as this gives the radiators a view only of cold space. With no internal power dissipations, survival heaters are required on the power subsystem (batteries) to maintain the minimum temperature. Approximately 10W of additional heating is required for the spacecraft.

4.9.6 Critical Areas for Thermal

The predicted operational temperature of the solar array in the nominal case is ~100°C. When a 10°C uncertainty margin is applied this prediction matches the maximum operational temperature of the solar arrays which may be an area of concern. This was a key trade-off during the study sessions. A body mounted solar panel has several advantages over a deployable array; these include cost, risk and complexity. A deployable array however would have a lower temperature as the rear face would be able to radiate some of the absorbed heat to space. The efficiency of the solar cells depends on temperature (they are less efficient at higher temperatures) so further work would be necessary on optimising the solar panel configuration.

4.9.7 Thermal conclusions

Overall the HAGRID thermal design can meet requirements through standard thermal engineering practices such as radiators, MLI and heaters. The nature of the heliocentric orbit is that the thermal stability is high.

Several spacecraft configurations were explored in the frame of this study and the thermal design is able to be compatible with all of them. So long as an anti-sun face is available for the radiators the thermal control system should be straightforward to implement.

4.9.8 Future thermal subsystem work

As discussed, a potential issue may be an excessively high solar panel temperature and this should be further optimised in the next stage of the design. Other areas for future work should include an examination of the thermal behaviour during the transfer orbits as these may prove a more demanding case for the thermal design. Additionally, more detailed modelling of each subsystem should be included in the next design iteration.

5. Ground Segment Design

5.1 Ground Stations

Ground station Contents:

- Which ground station will be used
- What is the performance and the constraints of the ground station
- Ground stations for a future operational mission

Due to the large distance between the operational/final position of the spacecraft and Earth it was important to consider ground station performance during this early stage of the space mission design. This section describes the options considered during the study, the ground station performance characteristics used for the study and future recommendations to consider.

Considering the drift scenario, where the spacecraft is greater than 2 AU from Earth at the end of life, the ESA and NASA Deep Space Network was considered. It was shown that at least three ground stations are necessary to provide continuous communication. The ESA stations proposed during the first study session were Perth, Santiago, Cebreros and Kiruna. However it became apparent that if a similar spacecraft is to become an operational mission such in-demand ground stations cannot be used to for the long periods of time the mission would require.

Considering the operational requirement, it is desirable to enable smaller more accessible ground stations to be able to downlink data from HAGRID. A future operational mission should consider building new ground stations to be able to provide this continuous downlink. It is necessary to scale the cost of any required new ground stations to the overall mission cost.

For the HAGRID technology demonstrator, an existing ground station should be used to minimise cost and prove that a smaller ground antenna can have the required performance. Three antennas were considered, the 4.5 m and the 25 m antennas at Chilbolton UK, and the 12 m antenna at Harwell, Oxford UK. The 4.5 m antenna has already been used to provide beacon data downlink from the STEREO mission across distances up to 120 million km. The size of the HAGRID spacecraft and the baseline for the launch vehicle limits the size and RF power of the high gain antenna on board the spacecraft and the link budget calculations for the Communication subsystem clearly show that a higher performing ground station antenna is necessary.

The 12 meter antenna at Rutherford Appleton Laboratory currently operates in S-band but an upgrade plan shows that with some modification and an installation of a modern X-band system a performance of 37 dB/K can be achieve. The predicted performance values on this RAL ground station was used as the baseline of the mission. Similar performance has been demonstration by older 15 meter antennas.

Figure 29 shows a picture of the RAL antenna and Table 42 summarises the main characteristics of the antenna after the proposed upgrade.

Figure 29 – 12 m antenna dish at Rutherford Appleton Laboratory in the UK with the potential of becoming an X-band ground station

Table 42- Summary characteristics of the upgraded RAL antenna

CDF database ID	name	position (longitude, latitude)	diameter (m)	dB / K performance	band
Ground Station 17	RAL HARWELL OXFORD (S)	51.572175,-1.312695	12	37	X-band

An STK simulation was used to show downlink windows. It is recommended that at least 3 strategically placed ground stations are used for any operational mission to provide uninterrupted visibility of the satellites.

5.2 Ground Segment

Contents:

- Space Weather monitoring and alert system overview
- Space segment capability
- Ground segment requirements

5.2.1 Space Weather monitoring and alert system overview

The HAGRID spacecraft is only one part of a Space Weather monitoring and alert system that could be provided if this spacecraft is launched. In order to provide consistent, continuous monitoring and warning of space weather events, a ground segment is needed in addition to a space segment.

In the diagram in Figure 30, a depiction is shown of an event being observed, downlinked to Earth, processed by algorithms, analysed by scientists and a warning or alert being raised. While this study has shown the capability of the space segment to provided monitoring of space weather events using the STEREO instrument, the ground segment ability to initiate data analysis and raise warnings has not been looked at.

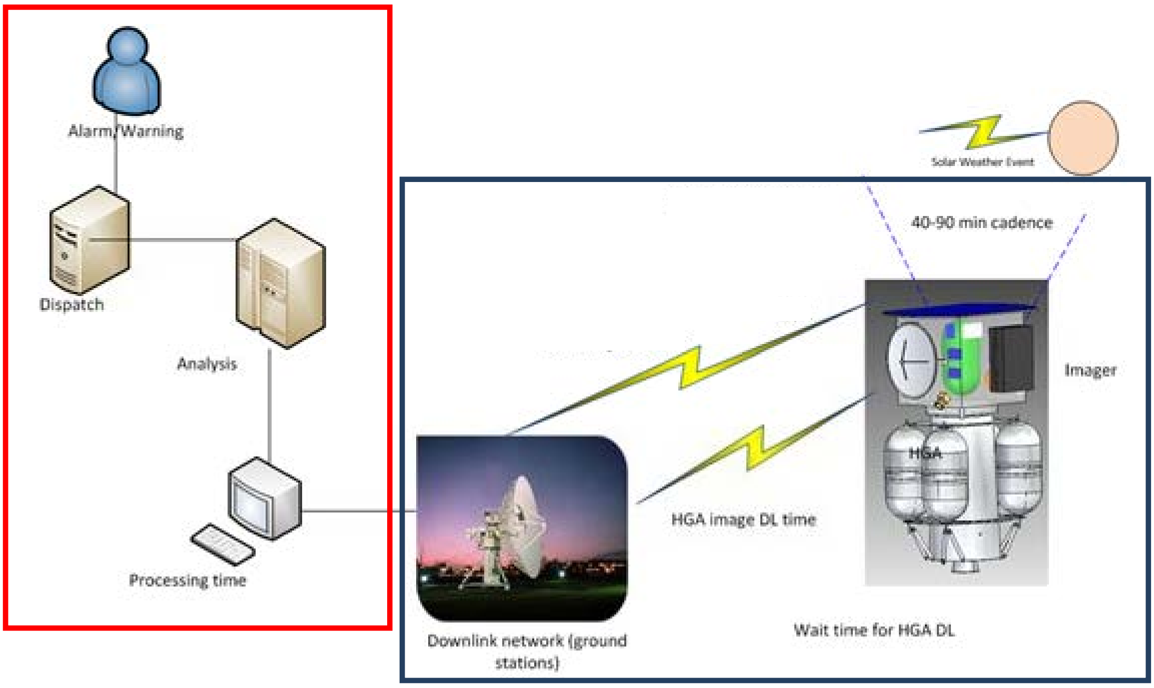


Figure 30 - Diagram of Space Weather monitoring and alert system.

5.2.2 Space Segment Capability

In this study, it was shown that a spacecraft that can be launched from a low-cost launcher, using existing space technology, and can provide images of the Sun-Earth line to observer Earth directed Coronal Mass Ejections (CMEs).

The spacecraft design is robust enough to provide multiple configurations that could put the STEREO instrument into operational orbit near L4 and provide station keeping manoeuvres. The subsystems on the spacecraft can downlink enough images for the identification and characterisation of a CME.

The elements in the blue box in Figure 30 are feasible, as shown in this pre phase-A study. The spacecraft can also provide an alert in a timely fashion (less than 8 hours). This is a reasonable amount of time considering average transients can take days to reach the Earth, while fast transients may arrive in 17 hours (UR-11). Using the cadence of the STEREO instrument of about 1 hour, these warnings can be updated to provide tracking of the transient event.

5.2.3 Ground Segment Requirements

In order for a warning to be raised, the images downlinked from the spacecraft need to be analysed. Currently, this analysis is done on an ad-hoc basis by the STEREO science team. An additional hindrance is that, due to telemetry constraints, the low gain antenna is the means for the continuous downlink of images, and these data are currently only down-linked on a voluntary basis. Those with capable antennas, that are willing, downlink the low gain antenna beacon data.

This data is often noisy and patchy, so the current capability for reacting to space weather events is weak at best. The algorithms used by the STEREO science team have been shown to predict transients when applied to downlinked images. Potentially this technology could be developed further to do the processing on board the spacecraft, enabling it to send a warning (with minimal telemetry) before images are down-linked (using high rate telemetry) or as a data handling process on the ground.

By considering the analysis currently done on STEREO, a standard monitoring process could be developed, and implemented. The elements that still need to be considered are shown in the red box in Figure 30. These include; the infrastructure needed to complete data processing, specialists required to determine a level of warning and the means by which warnings are dispatched.

The level of warning should also be considered. When is the severity of a space weather event high enough to trigger a warning? Who should receive this information? Is this a service that interested parties, such as power grid suppliers or operators could subscribe to, and then at certain warning levels be signalled to turn off critical-at-risk systems? Should governments be notified for high level warnings or alerts, and if so what actions should be put in place as a result of this information?

6. Summary and Conclusions

In considering all of the options and trade-offs, the conclusion of this study is that an operational mission making use of a single spacecraft stopping at a fixed location, 60° (or less) ahead of, or indeed behind, the Earth in its orbit is entirely feasible within the constraints of this study.

Contents:

- Scope of HAGRID CDF Study
- Final baseline design of HAGRID CDF Study
- How well are objectives of the mission achieved within the global constraints?
- What are the key issues with the implementation of the mission?
- Key recommendations
    - o Should the mission proceed to Phase-A ?
    - o Are there any fundamental changes to the initial study baseline which need to be assessed in future work?
    - o Where should development activity be focussed?

6.1 Scope of HAGRID CDF Study

The scope of the HAGRID CDF Study was to determine the feasibility of a technology demonstrator spacecraft for a future Space Weather alerting system that:

- Can provide service within 1 year of launch (2 years for optimal positioning)
- Uses a low-cost launch option
- Positions a single satellite in an 'ahead' or 'behind' orbit
- Provides accurate warning at least 10 hours in advance of a significant Earth-Impacting Space Weather event

For the science and payload areas of the mission the HAGRID study used lessons learned from STEREO/HI, that heliospheric imaging from a vantage point outside the Sun-Earth line enables imaging and tracking of Earth-directed solar transients (e.g. CMEs). It also incorporated new techniques based on heliospheric imaging out to large (> 30^{o}) elongations that demonstrate that a single-spacecraft could be used to provide accurate determination of solar transient velocity and direction. It was also demonstrated that by down-linking selected sub-fields from the images the data downlink could be reduced to a rate that can be continually broadcast by the spacecraft and received by a series of moderate sized (~14m diameter) dedicated ground-stations.

In terms of technology, the spacecraft subsystems were all designed using off-the-shelf items. While bespoke items will be needed in carrying this mission forward, this provides confidence that current space technology can be used to put the spacecraft in a preferred operational orbit.

6.2 Final baseline design of the HAGRID CDF Study

The final baseline design determined during the HAGRID CDF Study is a single spacecraft that will be deployed in the 'Ahead' orbit (being less susceptible to micro-meteoroid impacts on the instrument than a 'Behind' orbit). The 'Stopping' case orbit, putting the spacecraft ahead of the Earth at the L4 point, was chosen to allow the spacecraft more time in operational position and to reduce telemetry distances. This design uses the composite spacecraft configuration with the Lisa Pathfinder PRM to perform the transfer orbit burns. The spacecraft itself has an internal monopropellant propulsion system (to reduce complexity) that provides propellant for drift-alteration ('stopping') burns as well as operational AOCS burns. The deployable high gain antenna is on the 'top' of the spacecraft while the fixed-panel solar array is on the 'rear' sun-directed panel of the spacecraft, with radiators mounted on the 'front' space-directed spacecraft panel as shown in Figure 31. All other subsystem components fit reasonably within the spacecraft structure that is a 1.2 m cube. External parts are mounted below the baffles of the HI instrument mounted on the side of the spacecraft, and the stowed spacecraft with PRM fits in the Rockot fairing, as seen in Figure 32.

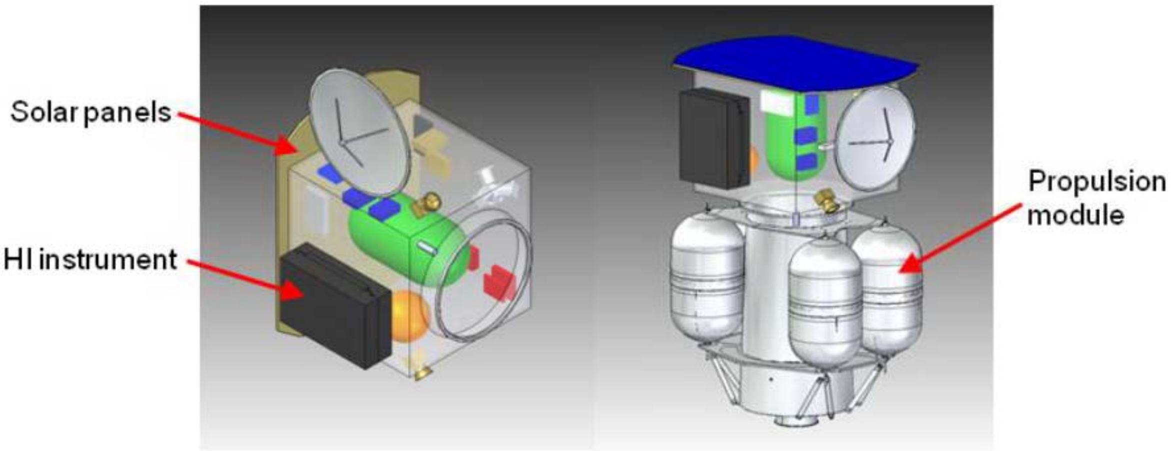


Figure 31 - Spacecraft in deployed configuration along an Earth-Ahead orbit with the solar panels pointing towards the Sun and the HI instrument looking towards the Sun-Earth line (left). The spacecraft and Lisa Pathfinder PRM in stowed configuration (right).

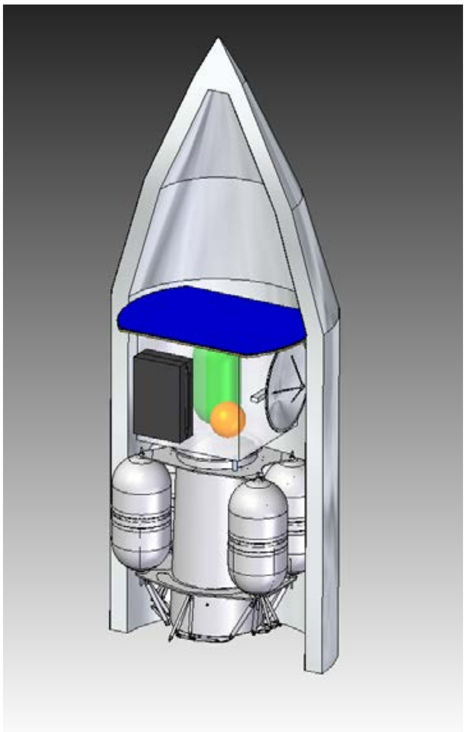


Figure 32 - Final baseline design of the HAGRID CDF Study.

6.3 What are the key issues with the implementation of the mission?

While it would be desirable to image the heliosphere during the cruise phase of such a mission, this is not possible without a significant increase in the technical specification (and therefore cost) of the spacecraft because:

- The attitude control system requirements become more demanding, because of the liquid propellant required for the stopping manoeuvre;
- A steerable high-gain antenna would be required for telemetry during the cruise;
- A re-useable door mechanism would be required to prevent potential contamination of the optics during the stopping manoeuvre.

The impacts of these requirements will be assessed in more detail in any follow-on study.

Without the ability to make accurate observations during the cruise phase, this spacecraft configuration contravenes one key requirement used to define this study; namely the requirement for useable observations within a year of launch. Given the other constraints, it was decided to change the baseline for the current study so that the requirement for observations 1 year after launch should be increased to 2 years. This enables the spacecraft to reach a position of at least 40^o ahead of the Earth in its orbit, from which is a prime position for observation of both Earth-directed CMEs and CIRs ahead of their arrival at Earth.

While this study focussed on a mission orbiting ahead of the Earth, the conclusions are equally applicable to a spacecraft ahead or behind the Earth in its orbit. Experience from the STEREO mission has shown that there are advantages to both positions. Orbiting behind the Earth enables in-situ measurements of co-rotating solar wind streams prior to their arrival at Earth and is a prime location for imaging the solar disk prior to active regions rotating to a longitude where they could potentially launch Earth-directed CMEs. However, such measurements would require additional instrumentation and are not considered within the scope of this study. The disadvantage of a behind orbit, as experienced by HI on STEREO-B, is micro-meteoroid bombardment of the HI instrument that can severely compromise the quality of imaging observations (unless the instrument is significantly redesigned leading to an increased financial budget). A spacecraft in a location ahead of the Earth does not suffer in this way, and - provided the spacecraft is not separated from Earth by too large an angle - can still be used to image CIRs ahead of their arrival at Earth.

Both integrated and separate off-the-shelf propulsion modules were considered. Both are feasible but a comprehensive assessment of the relative merits of each was outside the scope of this study. The study demonstrated that the HAGRID mission concept is feasible and generally compliant with the key technical requirements. With data down-link, processing and the issuing of a warning estimated to take 7.2 hours, a worst-case advanced warning for a Carrington-type event is shown to be 9.8 hours. However, the non-compliance of the length of the specified cruise phase is of some concern; the spacecraft would require a 2-year cruise to attain a location 40^o ahead of the Earth. For such a 'stopping' mission, although the mass budget is compliant with launcher capability, there is no excess margin.

6.4 Key recommendations

6.4.1 Should the mission proceed to Phase-A ?

In the HAGRID CDF study, a solid design for a composite spacecraft was produced which is compliant with the requirements defined in the scope of the study. There are some important tradeoffs which need to be completed which could represent a significant improvement in the performance of the mission. For example; there are some potential savings between 'stopping' the spacecraft at 60° from the Sun-Earth line (as considered in this study) and the shorter drift distance of 40° from the Sun-Earth line. This could reduce mass through a smaller HGA for communications as well as reducing the required Delta-V, reducing the amount of fuel needed.

Consideration of the use of monopropellant or bi-propellant showed that there were only minor differences in mass savings between the two composite spacecraft options compared with the additional complexity of using a bi-propellant. The components chosen are off-the-shelf parts requiring little or no redesign, reducing risk in the development of this spacecraft.

This concept uses sufficient margins that show a high level of confidence for proceeding with the design of the spacecraft. This includes 20% margin on the LISA pathfinder PRM which is already designed, as well as a 20% system level margin on top of that for all subsystems.

The HAGRID mission should proceed to Phase A, as no significant risks or engineering show-stoppers have been brought up concerning the design or TRL of the spacecraft, its payload, or the complex orbits. This study has brought confidence to the spacecraft mission, and also highlights the areas to focus Phase-A work.

6.4.2 Are there any fundamental changes to the initial study baseline which need to be assessed in future work?

Yes, areas identified as requiring further investigation in any follow-on study, e.g. a future phase-A study, include:

- Performing a detailed analysis of mass budget, especially of the spacecraft structure;
- Addressing the problem of cruise phase duration and the feasibility of performing useful mission operations during the cruise phase;
- A careful re-assessment of operational orbit requirements (including the definition of a minimum acceptable spacecraft-Sun-Earth angle for successful operation; defining an optimum image cadence; assessing whether cruise phase observations are of any merit in terms of forecasting);
- Optimising the image format for downlink (in terms of defining regions-of-interest, pixel binning) within the assumed available telemetry budget, while ensuring that the resultant image quality is adequate to meet prediction requirements;
- Analysing the impact of using a simplified HI instrument with a single camera (as on the SoloHI instrument on the Solar Orbiter mission).
- Re-assessment of the materials and fabrication techniques used for the Heliospheric Image instrument in order to decrease the labour costs required for fabrication.

6.4.3 Where should development activity be focussed?

At the conclusion of a Pre-Phase A study, there is still much work that can be done improving the technical baseline for the HAGRID CDF study. However, the spacecraft looks feasible, and, based on the list of elements needed for space weather monitoring and warning, it may be important to focus work on developing the business case for a space weather monitoring and alert mission.

Basic elements are needed for a space weather monitoring and warning mission. While some of these were within the scope of the HAGRID CDF study others were not.

- **assessed during the study**:
  - The ability to image the Sun-Earth line to observe Earth-directed Coronal Mass Ejections
  - The ability to downlink enough image data, in a relatively small amount of time, to identify and characterize CMEs.
- **NOT assessed during the study:**
  - Timely downlink and retrieval of data (Current downlink of STEREO space weather is on a voluntary basis)
  - Ground segment to handle the Alerts
    - A mechanism would need to be established to distribute the information
  - Who is responsible for classifying and raising an alert
    - What is a warning vs. an alert?
    - Who gets this information?
      - Government agencies?
      - Power grid management?
      - Air traffic control?