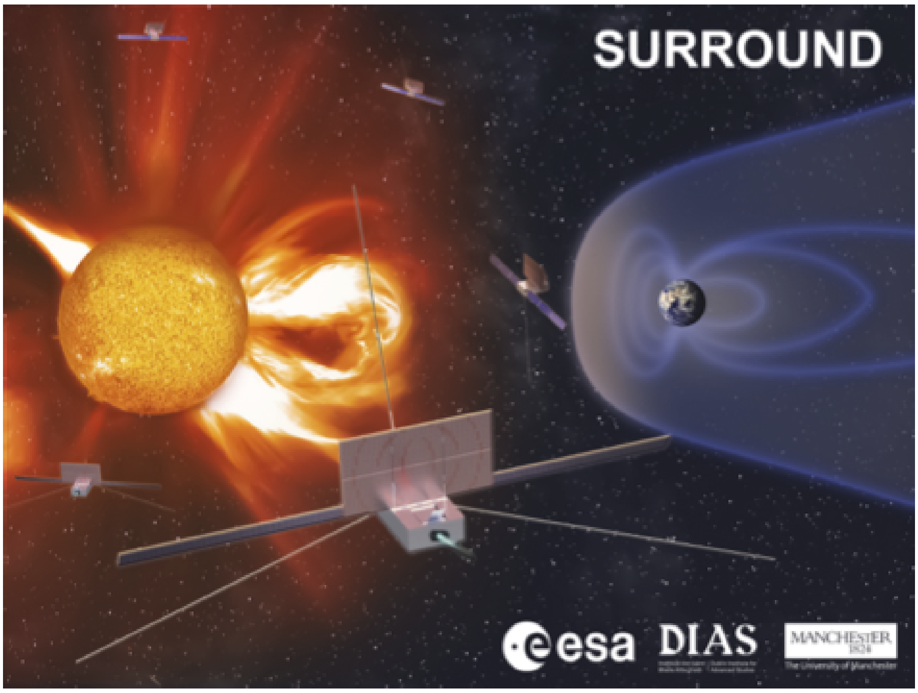

SURROUND

A Constellation of Nanosatellites around the Sun

AO 2-1827/22/NL/GLC/ov

Executive Summary

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The Sun regularly releases considerable amounts of energy resulting in coronal mass ejections (CMEs) and accelerated particles, which can have a variety of adverse space weather (SW) effects at Earth and in the near-Earth environment. A useful means of tracking SW activity is via solar radio bursts (SRBs) associated with CMEs and solar energetic particle events (SEPs); CME-driven shocks can be tracked via Type II SRBs, while energetic electrons escaping into the heliosphere can be tracked via Type III SRBs. Currently, there are no operational means to monitor and track SRBs throughout the inner heliosphere. Here, we propose a constellation of CubeSats called SURROUND to observe SRBs in order to track CME and SEPs for SW monitoring (Fig. 1). This concept would complement the mission goals of ESA Solar Orbiter/RPW, NASA Parker Solar Probe/FIELDS and SunRISE, among other missions.

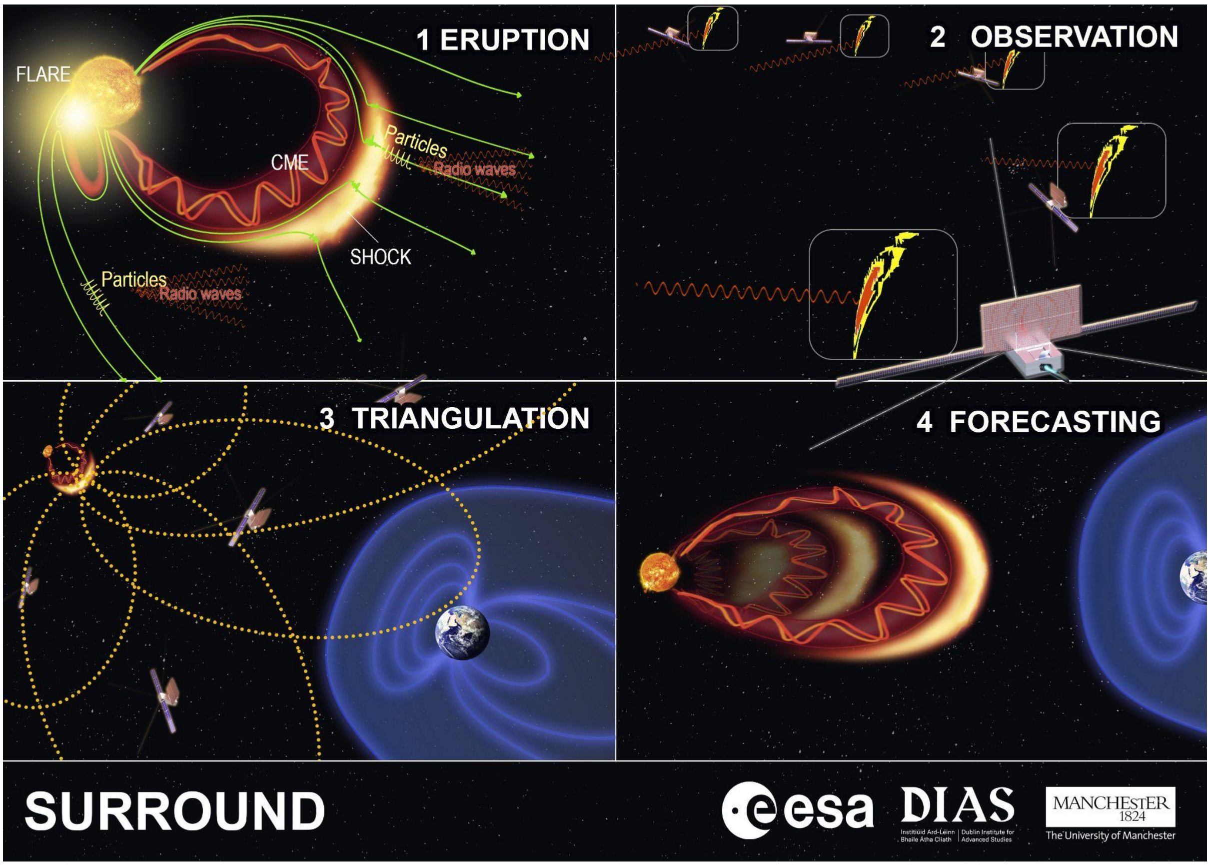


Figure 1: Illustrated summary of SURROUND mission.

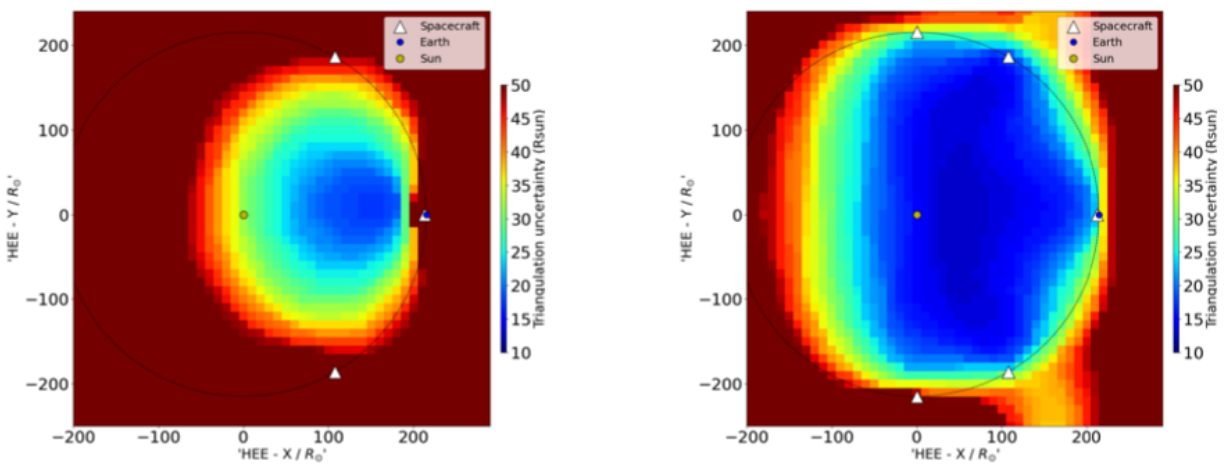
Mission Requirements

Through this work, nine use cases have been established and analysed to define operational requirements for a SW monitoring constellation and are summarised in Table 1. A variety of multilateration and direction finding techniques were considered including time-of-arrival (TOA), time-difference-of-arrival (TDOA), and goniopolarimetry (GP).

Use Case	Description
UC1	Detect and track Type-II radio bursts - fast CMEs shocks accelerated electron beams
UC2	Detect and track Type-III radio bursts - flare accelerated electron beams
UC3	Localise in 3D Type-II radio bursts
UC4	Localise in 3D Type-III radio bursts
UC5	Nowcast CMEs associated with Type-IIs
UC6	Nowcast SEP electrons associated with Type-IIIs
UC7	Forecast arrival of CME associated with Type-II at Earth[1] (propagation time scale ~12 hours)
UC8	Forecast arrival of SEP electrons associated with Type-III at Earth[1] (propagation time scale ~10 minutes)
UC9	Forecast the potential arrival of SEP protons associated with Type-III (propagation time scale ~2 hours)
UC10	Scientific Analysis

Table 1: Summary of the SURROUND mission Use Cases

The TOA multilateration technique is the simplest to implement but relies on knowledge of the source signal (i.e., time of emission). By taking the difference between the time of arrival of the signal and assuming a constant propagation velocity, the distance from the source can be calculated. As the number of satellites increases, the uncertainty in location of the source decreases. However in a SW monitoring scenario, little-to-no instantaneous knowledge of the source is available during observations. For the purposes of SURROUND, the TOA method is used as a “ground truth” to test more robust methods of multilateration and compare the errors in source location. Our findings implement a geometrical and Bayesian statistical TOA algorithm to perform this analysis on various satellite configurations (3-, 4-, and 5- satellites positioned around the Sun) and time cadences (10--60s, in 10s increments) in line with the mission goals of SURROUND (see Fig. 2). The analysis shows that the lowest time cadence (10s) with a 5-spacecraft setup yields the smallest uncertainties. Using lower time cadences with a smaller number of satellites also provides acceptable results for monitoring SRBs, allowing many descope options.



(a) 3 Spacecraft SC1-SC2-SC3 (b) 5 Spacecraft SC1-SC2-SC3-*a*-*b*

Figure 2. Results from the Bayesian TOA algorithm for (a) 3 spacecraft and (b) 5 spacecraft configurations. The colour scale represents the localisation accuracy with areas in dark red exceeding a chosen threshold of 50 solar radii. For a given cadence, increasing the number of spacecraft increases the area where accurate localisation is possible.

TDOA and GP were considered most suitable for the SURROUND mission as they do not require knowledge of the time of source emission. TDOA is more complex as it relies on the difference between the times of arrival of the signal at each receiver (or satellite) to localise the source. GP functions by determining the propagation vector of detected electromagnetic waves emitted from a target using three carefully positioned and configured antennas (or two antennas where one acts as a spinning dipole). The antenna measures the polarisation and associated flux of the emitted electromagnetic waves (i.e. radio source) which are used to reconstruct the resulting wave and back propagate to the source (within a known uncertainty). Using GP with multiple spacecraft detecting radio signals from the same source, should result in each wave vector being back propagated to the same point, providing the location of the radio source. Both TDOA and GP techniques can be implemented using the same number of antennas and configuration to localise the radio source location.

To meet the defined operational requirements, the proposed instrument should be able to detect radio waves with spectral flux greater than $10 \times 10^{-22}\, Wm^{2}/Hz$, in a frequency range of $0.25 - 25\, MHz$. The data should be at a cadence of at least 10 seconds and with communications to ground at least once every hour.

Mission Analysis

To achieve the desired localisation accuracy, a minimum of three spacecraft are required. The possible locations considered for these spacecraft are (in order of preference): the Lagrange points L1, L4, L5, an Earth-leading orbit, an Earth-trailing orbit, and an out-of-ecliptic orbit. Mission analysis considering orbit insertion and maintenance requirements of these spacecraft identified the out-of-ecliptic option to be highly challenging to obtain with current technology and therefore rejected this as an option. The other proposed positions investigated were found to be theoretically achievable using existing electric propulsion systems suitable for nanosatellites, in the case that a suitable Earth escape trajectory could be provided by a launch vehicle or space tug.

Orbit selection is primarily driven by ease of insertion and maintenance, as well as the constraint to avoid entering the solar exclusion zone (SEZ). Spacecraft within this region will not be contactable from Earth due to the interference of the Sun. The halo orbit at L1 is selected to avoid entering the SEZ and having an orbit period on the order of 1 year, with a recommendation for orbit maintenance manoeuvres every 6 months (Fig 3). The Earth-leading and Earth-trailing spacecraft will enter the SEZ as they pass behind the Sun. As such the rate of separation for these is selected to avoid entering the SEZ until mission end, approximately 10 years after launch (Fig 4). The halo orbits around L4 and L5 are unconstrained and sample orbits are selected to calculate order of magnitude insertion requirements. These will vary significantly depending on the transfer time, with longer transfers having lower escape and orbit insertion requirements. An example trajectory to L5 is shown in Fig 5**.**

For the orbits and trajectories selected, the change in velocity (ΔV) requirements are summarised in Table 2**.** For the L1, Earth-leading and Earth-trailing orbits are established as on the order of $100 - 500\, m/s$ per spacecraft. The L4 and L5 spacecraft would have a significantly higher ΔV requirement, on the order of $1000 - 1500\, m/s$, for an insertion time on the order of 15 - 23 months.

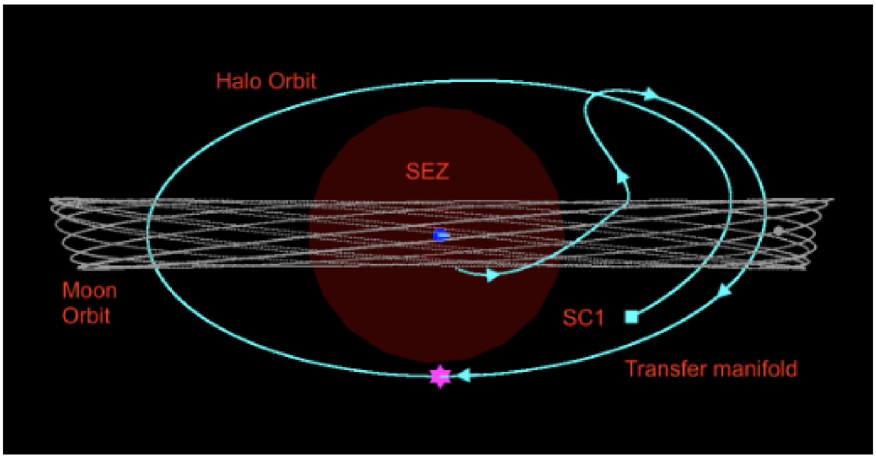


Figure 3: Transfer manifold trajectory and first halo orbit around L1 as seen from the -X synodic axis, i.e. from the Earth-Moon barycentre to the Sun. The spacecraft paths lie outside the solar exclusion zone. Patching point between the transfer and halo segments is shown with the magenta marker. Objects with respect to Earth in synodic reference frame.

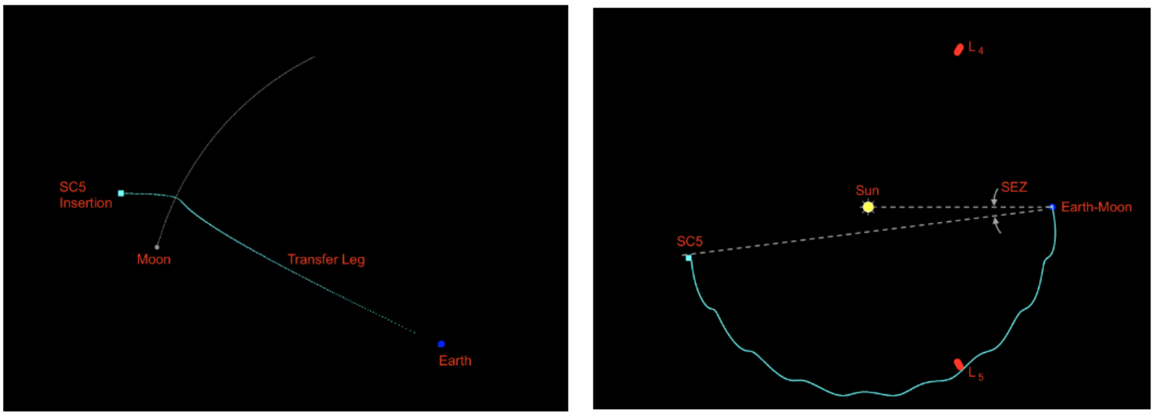


Figure 4: (left) Lunar transfer leg for Earth-trailing spacecraft. (right) Earth-trailing spacecraft trajectory during 10 year drift with mission ending prior to spacecraft entering the solar exclusion zone. L4 and L5 points shown for reference. Objects in synodic heliocentric reference frame.

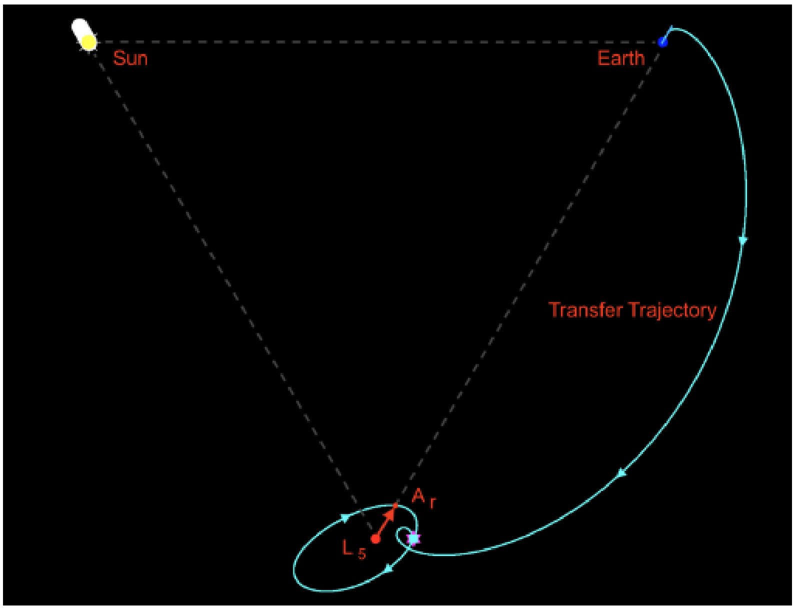


Figure 5: Example of a 15-month direct transfer trajectory to L5. Final planar periodic orbit after arrival manoeuvre are also shown. Objects with respect to L5 in a synodic reference frame.

Spacecraft name	Reference	Transfer time [month]	Transfer $\lvert\Delta V\rvert$ [$\mathrm{m\,s^{-1}}$]	Maintenance $\lvert\Delta V\rvert$ [$\mathrm{m\,s^{-1}\,yr^{-1}}$]	Total $\lvert\Delta V\rvert$ [$\mathrm{m\,s^{-1}}$]
SC1	L1	4	30	30	330
SC2	L4	15 – 23	1500 – 900	0	1500 – 900
SC3	L5	15 – 23	1500 – 900	0	1500 – 900
SC4	Ahead	0	200	20	220
SC5	Behind	5	30	20	50

Table 2: Estimated transfer and maintenance requirements for the SURROUND Constellation. Results should be interpreted as order of magnitude estimates.

The mission analysis performed also considered the variation in the Sun-spacecraft-Earth angle for each spacecraft throughout its insertion and mission life (Fig 6). This highlighted key considerations for the communications systems to maintain reliable communication with Earth throughout the mission. In particular, the Earth-leading and Earth-trailing spacecraft will be required to have a steerable antenna, or flexible spacecraft orientation to maintain suitable communications links to Earth as the orbits progress.

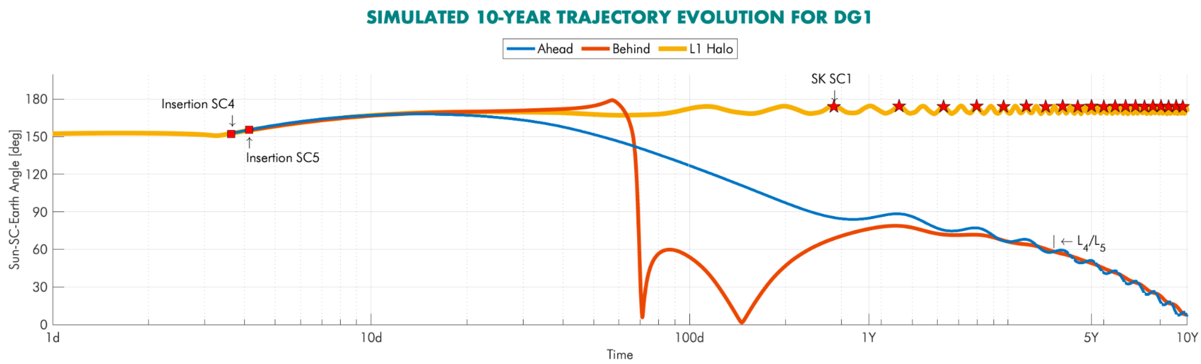


Figure 6: Simulated Sun-spacecraft-Earth angle evolution throughout spacecraft trajectories and mission life. Time from trans-lunar injection to the end of the mission.

System Design

Initial system analysis identified the communications architecture to be the most challenging subsystem element for the proposed mission. Considering the regularity of downlink required for some of the proposed operational mission use cases, it was deemed desirable to avoid reliance on the ESTRACK and/or Deep Space Network ground stations. Using deployable reflectarray antennas (for example previously flown on the MarCO CubeSat mission) it was found to be feasible to achieve the data link required to return all payload data from the L1 spacecraft using a $0.2m^2$ antenna with $4W$ transmit power and a commercial-level $13m$ ground station (see Fig 7). The L4, L5, Earth-leading, and Earth-trailing options, on the other hand, experience lower data rates due to their distance from Earth (1 to 2 au), though all the payload data can be returned to Earth using higher power transmitters ($15W$) and larger ground station antennas (e.g., $70m$ DSN). Operationally, however, precise data retrieval from these spacecraft for tracking and localisation of specific phenomena, triggered by detection from the L1 spacecraft, will be considered using smaller commercial ground stations.

Electric propulsion systems were identified following a tradeoff study as a suitable option to perform the required orbit transfers and maintenance manoeuvres. Available Hall effect thruster and gridded ion engine options can achieve these requirements, whilst further systems in development have the potential to improve performance. Additional key technology drivers for the mission, where development beyond the current state-of-the-art is expected to be necessary, are explored, and electromagnetic compatibility, attitude control/momentum management, and long-term survivability for nanosatellites in deep-space are highlighted as most critical.

Based on these studies, a nominal spacecraft system concept (see Fig 8) has been established, noting that the differing requirements of the spacecraft may lead to minor differences in design. Significant design drivers for each spacecraft are identified and it is noted that the final selection of options for the mission must be made considering a variety of factors, including budget, launch epoch and availability, and reliability of critical technologies.

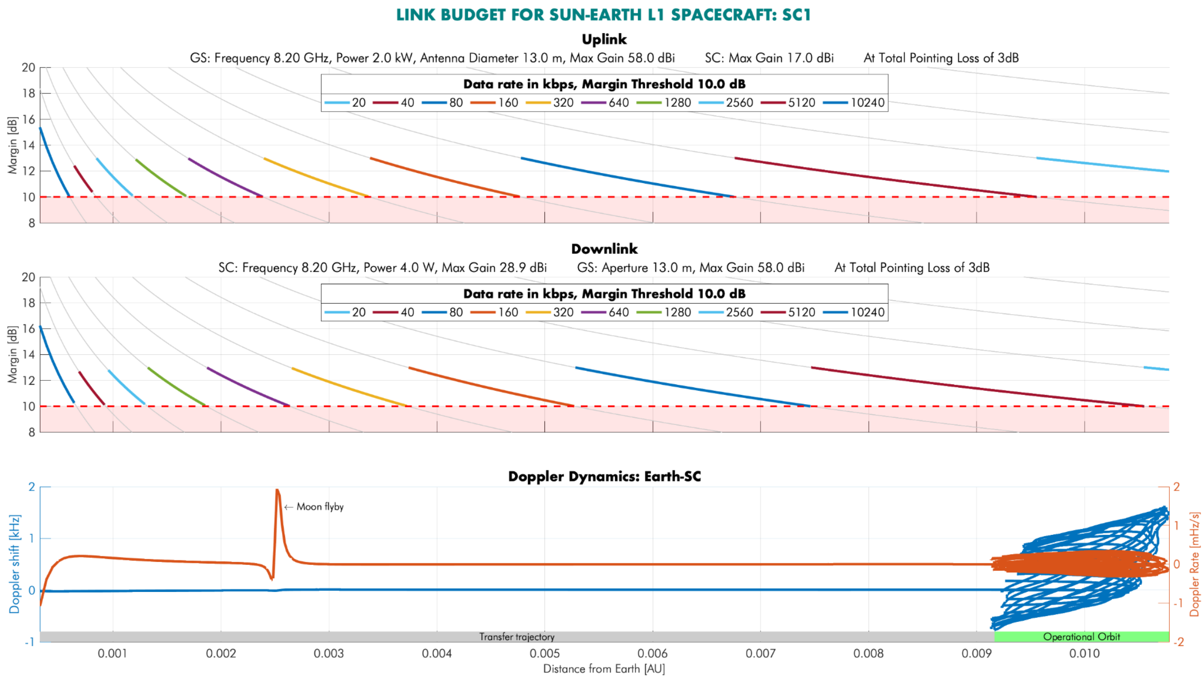


Figure 7: Link budget analysis for the L1 spacecraft demonstrating sufficient data rates to downlink all payload data to Earth on a regular schedule using 13m commercial ground station options.

Five spacecraft positioned at L1, L4, L5, Earth-leading, and Earth-trailing are proposed as an ideal solution to meet the defined operational requirements and provide the most precise localisation of space weather events; a descope option of three spacecraft at L1, L4 and L5 is the second highest priority operationally.

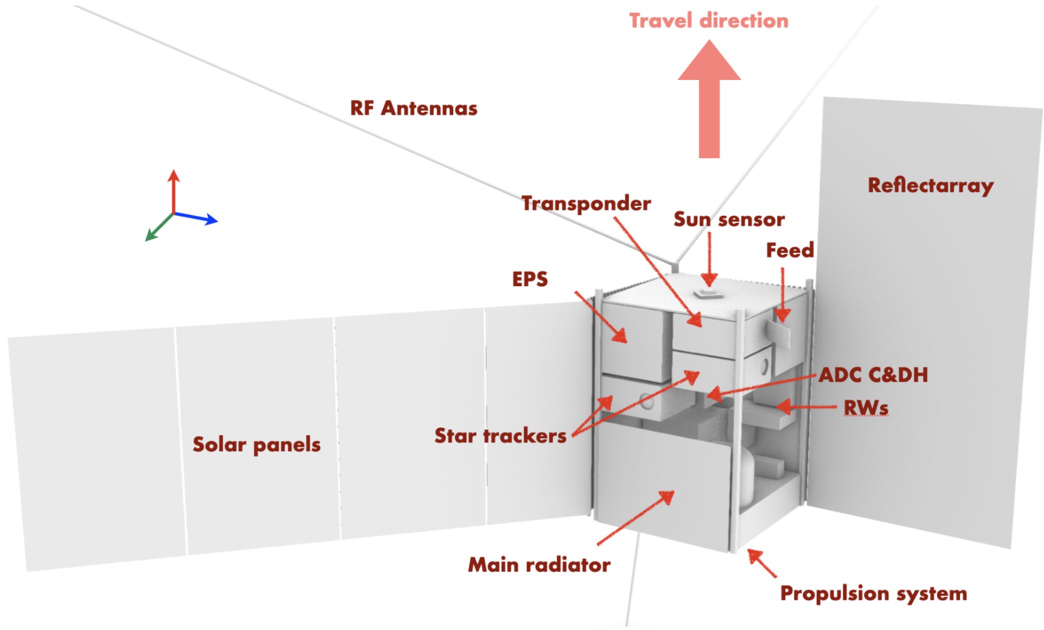


Figure 8: Preliminary concept for a SURROUND spacecraft with propulsion system, reflectarray antenna for communications, solar arrays, and RF payload antennas.