CARETAKER
Coronal **M**ass **E**jection **A**nalysis **R**eporting to **E**arth **T**o **A**llow **K**eeping **E**verything **R**unning

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Abstract

The Earth's magnetosphere is formed as a consequence of interaction between the planet's magnetic field and the solar wind, a continuous but varying plasma stream originating from the Sun. A number of different phenomena in the solar wind have been studied over the past forty years. The CARETAKER mission aims at launching a cluster of 6 satellites on an orbit around the Sun at 0.72 AU in order to study the large scale structures coming from the Sun (particularly CMEs). For in-situ measurements, the spacecraft will contain Solar Wind Analyser and Fluxgate magnetometers. For remote-sensing, externally occulted coronagraphs will observe the corona and the interplanetary environment. Communication with the six satellites will be provided by two ground stations through the CARETAKER Network 24/7. The data from the sensors will be transmitted to the Data Processing Centre where raw data is transformed into information for the scientific community and other end users. Our space mission represents a new reference for space weather event warning as well as for premium scientific content.

1. Introduction

Space weather is characterized as "conditions on the Sun and in the solar wind, magnetosphere, ionosphere and thermosphere that can influence the performance and reliability of space-borne and ground-based technological systems and can endanger human life or health".

2. Mission Overview

2.1. Social and Service Motivation

The OECD report on geomagnetic storms, 2011 stated "The international community should improve the current geomagnetic storm warning and alert system" and "The relative absence of public awareness of space weather phenomenon contributes to the lack of national-level strategic planning amongst OECD member states". [1]

Our team developed an information-service system to provide space weather information in a timely manner as a large step towards protecting technology and human life.

2.2. Scientific Background

Coronal Mass Ejections (CMEs) are eruptions in the solar corona which carry an amount of plasma (1015 to 1016 g) into the interplanetary medium.

CMEs are observed at the Sun by white-light coronagraph. In brief, a coronagraph records photospheric radiation from the corona, scattered by electrons (Thomson scattering process). Figure 1 shows a 'Halo CME', observed by LASCO. A Halo CME is a CME that is directed towards the Earth or to the opposite direction. There are three characteristics of this structure: a high density leading loop, a dark cavity and a bright interior feature [Howard et al., 1982]. The occurrence of CMEs varies according of the solar cycle, with ≈ 3.5 per day at solar maximum and ≈ 0.2 per day at solar minimum. The average opening angle is ≈ 60° and the apparent speed ranging from 50 to 3000 km/s [Howard et al., 1985 ; Gopalswany et al., 2004]. During the propagation of CMEs into the interplanetary medium (also known as ICMEs), some missions (for example, ACE, Wind or Stereo) measure the CME internal structure.

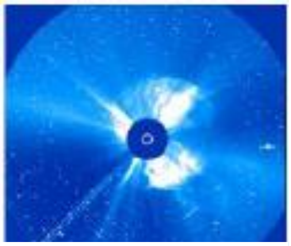

Figure 1 - Example of a Halo CME observed by the LASCO C3 Coronograph on Soho

Geomagnetic storms are mainly caused by CMEs. They may contain southward magnetic field in their leading part. Southward magnetic field can induce magnetic reconnection at the dayside of the magnetopause [Dungey, 1961] that allows transferring the energy of the ejecta in the inner regions of the Earth's magnetosphere [Akasofu, 1981] enhancing the ring current intensity.

2.3. Mission Statement

"The system consists of a near real-time information-service, based on the physical properties of large scale structures originating from the Sun."

2.4. Mission Requirements

Primary requirements:

R1 The mission shall measure the direction of a CME and its magnetic field vector orientation;

R2 The envelope of CMEs shall be determined to less than 4.3 arcmin (i.e. the angular diameter of the Earth's magnetosphere from the Sun);

R3 CMEs shall be remotely monitored from their origin between 2 and 15 solar radii in order to constrain CME models (Thernisien et al., 2009);

R4 CMEs shall be monitored from at least two positions simultaneously which are at least 45° apart;

R5 In situ and remote sensing measurements shall be performed in order to achieve a minimum of 3,5 h warning time;

R6 The 3D velocity distribution of protons and electrons and the composition of heavy ions up to 56 amu/q shall be measured with a time resolution of 1 min for the detection of the CME shock front [Richardson and Cane, 2004];

R7 The instrument suite shall measure the low-energy ion particle flux in the range of 0.26 to 20keV/q as well as the low-energy electron flux in the range of 1 eV to 5 keV;

R8 The magnetic field shall be measured with a resolution of 0.1nT and in a range between -200nT and 200nT (Burlaga, 2001);

R9 The data shall be processed to provide data products according to the SSCC Standards and shall be made available to the SSCC;

R10 The operational life time shall be at least 5 years (+ 5 years);

Secondary Requirements

R11 In order to improve CME models the number of CME observations shall be increased by continuous monitoring.

2.5. Operational concept

The mission combines results of stereo coronagraph images and in situ measurements in order to determine the trajectory and physical properties of solar CMEs. Therefore six satellites rotate on an orbit at 0.7 AU with an angular distance of 60° degree to each other. The angular distance is driven by the argument that CMEs with an angular width larger than 60° have a heliospheric impact (Gopalswamy, 2008). All spacecraft are equipped with a coronagraph (2-15 solar radii field of view, 5 min cadence), a plasma instrument and magnetometers (both 1 Hz sampling). Plasma instruments and magnetometers of all six spacecraft operate continuously whereas only two spacecraft coronagraphs operate simultaneously. Due to communication constraints the two coronagraphs with an angular distance of 120° to each other at the Earth facing site of the Sun are in imaging mode. This mission is able to detect all CMEs mentioned above. When a CME reaches the region between 2 and 15 solar radii, its plasma structure is in the coronagraphs' field of view. Coronagraphs' images and in situ data are downlinked to Earth every 15 minutes and are fully processed within the next 45 minutes.

The outcome of the image processing is the direction and the angular width of a CME (Thernisien, 2009). In case the results predict that the CME is bound to Earth, a first warning will be given to the end-user. This warning only contains the information that a possible geo-effective CME has been detected. Additionally the velocity of the CME between 2 and 15 solar radii will be determined and published to the user if images quality and modelling performance are sufficient.

The results of the in situ data processing are the magnetic field strength and its orientation as well as plasma density, velocity and temperature. Depending on its spatial extent it can be detected in situ by one or more spacecraft.

After the first warning a second one will be given based on the in-situ data, if the CME properties exceed certain thresholds at 0.7 AU. In particular the Bz component of the magnetic field (GSM coordinate) and the velocity are supposed to be important drivers of the CME's geoeffectiveness.

3. Flight segment

3.1. Spacecraft Profile (TBD)

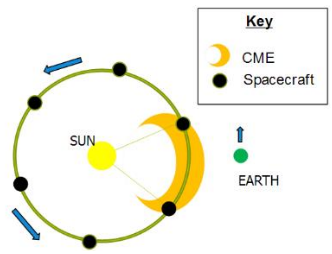

Figure 2 - Spacecraft setup

3.2. Orbit and Launch

In order to achieve the requirements for the minimum warning time, the radius of the final orbit of the satellites will have to be equal to or smaller than Venus orbital radius and separated by 60°.

To insert six satellites into a maximum orbital radius of 0.72 AU it would not be feasible to rely on only the satellite propulsion system. The vast delta-v required for this would be extremely costly. Using a Gravitational Assist Manoeuvres (GAM) with a planet, such as Venus, decreases the Δv required from the propulsion system [24].

The velocity of the spacecraft relative to the Sun can be increased or decreased by choosing the approach carefully. This can be extremely beneficial to the mission mass budget because it allows a large range of orbits to be achieved from a small range of initial trajectories. For this reason a fly-by at Venus has been designed to insert the

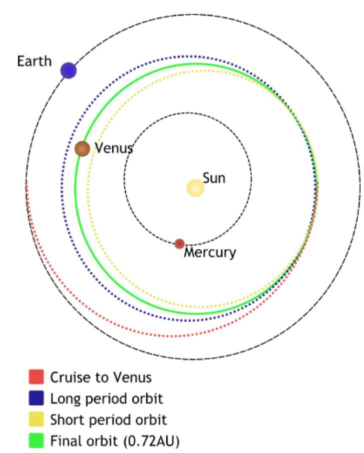

Figure 3 – Mission Profile

spacecraft into two different transfer orbits. Three of the spacecrafts will make a GAM inside of the orbit of Venus (decrease of velocity relative to Venus, S/C 1,2,3) and the remaining three on a different Venus altitude orbit (decrease of velocity relative to Venus, S/C 4,5,6). The result is an elliptical orbit with a periapsis smaller than 0.72 AU and an elliptical orbit with an apoapsis bigger than 0.72 AU, and therefore shorter/longer periods to gain the required phasing respectively separation.

Therefore a direct cruise trajectory was selected. The necessary characteristic energy (C3) is $5.57\frac{km^2}{s^2}$ to leave Earth.

After different number of orbital repetitions, circular heliocentric orbit will be accomplished by -Δv at apoapsis (S/C 1, 2 ,3) respectively +Δv at periapsis (S/C 4, 5, 6). The maximum velocity change Δv has a magnitude of 1019 m/s (without margin).

This manoeuvre with chemical propulsion (see Chapter 3.5 for comparison with electric low thrust propulsion) results in a transfer and separation time of 353 days for first S/C in final position and 1358 days until final constellation (without final commissioning and calibration). The properties of the final orbit are the same as the properties of Venus (siderial Period: 224 days and a synodic period 584 days).

Table 1 - Mission Timetable

S/C no.	Final orbit	Operational
1	13 July 2027	13 Aug. 2027
2	25 Aug. 2028	25 Sept. 2028
3	9 Oct. 2029	9 Nov. 2029
4	19 Aug. 2027	19 Sept. 2027
5	14 Dec. 2028	14 Jan. 2029
6	13 April 2030	13 May 2030

The final mission operation schedule is demonstrated in the ***Table 1* - Mission *Timetable*.**

After mission operation there is an option for decommissioning the S/Cs with a planned impact on Venus, with potential for additional scientific research.

A trade-off was completed to work out which launcher would be best suited for this mission: Ariane 5 or Soyuz launcher. If Soyuz launchers would be used the large mass budget would require at least three launchers which is not only extremely costly but also an operational nightmare. Consequently, one Ariane 5 will be used to launch CARETAKER[22][23] .

3.3. Spacecraft Design

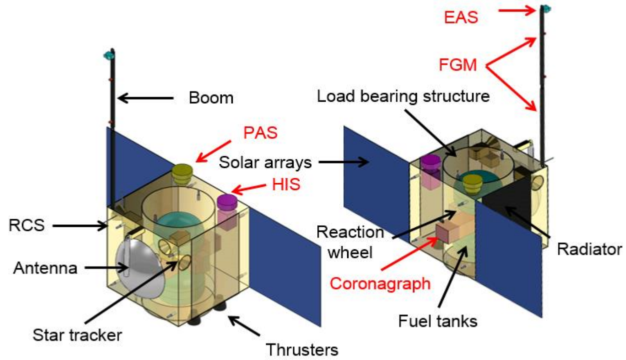

Figure 4 - Spacecraft design

3.4 Payload

3.4.1 Coronagraph

Each spacecraft will include a coronagraph. Stereoscopic images of the Sun will be obtained by pairs of coronagraphs separated by 120 degrees. In order to obtain satisfactory stereoscopic images, each coronagraph will be switched off for the period the spacecraft orbits along the Sun-Earth line.[1]

The coronagraphs used are externally occulted Lyot coronagraphs that observe from 2 R_o to 15 R_o. This choice of heliocentric distances provides the opportunity to observe the early stages of a CME.

The coronagraphs derive their heritage from the coronagraph COR2 on-board the SECCHI (Sun Earth Connection Coronal and Heliospheric Investigation) suit of instruments of the STEREO (Solar TErrestrial RElations Observatory) mission. The external occultation shields the objective lens from direct Sunlight, hence it enables a low stray light level and makes the observation in this range of heliocentric distances possible. [3]

The technical features of each coronagraph are presented in the **Table *2***

Table 2 - Coronagraph Technical Details

Mass (kg)	11
Dimensions (m^3)	1.31 x 0.246 x 0.28
Power (W)	5.5
Field Of View (FOV) (deg.)	11.4
Passband (nm)	450-750
Data rate (kbps)	16.7
Compression factor	10
Pixel size (arcsec)	15
Exposure time (sec)	<4
Images per hour	12
Photometric response (L_o/DN)	$1.3 \cdot 10^{-12}$

Where, L_o is the solar polarization brightness and DN is the measured response of the instrument in the passband of the observations.

The stereoscopic observation, combined with geometrical models (e.g. ice cream cone model, hollow croissant model) and reconstruction methods (forward modelling, inversion, and triangulation) allow us to obtain the three-dimensional structure of CMEs and extract their propagation direction and velocity. In our mission, the stereoscopic images obtained will assist us to find the propagation direction and velocity of CMEs in the region 2-15 R_o, hence acquire an estimation of whether a CME will be Earth-directed and, finally, when the CME is Earth-directed, reckon its arrival time.

3.4.2 Solar Wind Analyser

The Solar Wind Analyser (SWA) contains of 3 sensors with a shared processing unit. The main objective of the SWA is comprehensive in-situ measurements of CMS to characterize their properties and to determine their magnetic field structure.

To meet or exceed all the measurement requirements, SWA must be able to measure the three-dimensional velocity distribution functions of the major solar wind components: protons, electrons and heavy ions. **[5]** In ***Figure 2* - Spacecraft *setup***, the placement of the sensors can be observed.

- **Electron Analyser System (EAS)**

The EAS will make a high temporal resolution determination of the 3D electron velocity distributions and derive their moments (density, temperature, bulk velocity, heat flux).

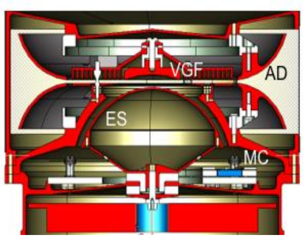

Figure 5 - Cross section EAS
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- **Proton Alpha Sensor (PAS)**

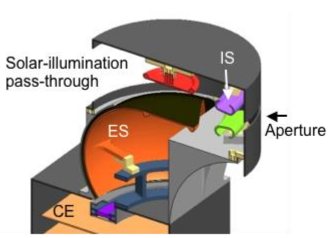

Figure 6 - PAS Cross Section
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The PAS contains a top hat electrostatic analyser that measures the 3D velocity distribution of protons and alpha particles in the energy range of 0.2 to 20 keV/q with a relative accuracy of 8% (energy), and an angular resolution of smaller 2%. Unlike the EAS, the PAS consists of only one device and therefore the field of view is narrower.

- **Heavy Ion Sensor (HIS)**

The HIS contains an electrostatic analyser module and a time of flight detector to measure the key properties of the heavy ions (up to 56 amu/q): mass, charge, energy distribution, and direction of incidence.

Table 3 - Properties of the Solar Wind Analyser (SWA)

	EAS	PAS	HIS
Field of view	4 π	-24 to 42°(Az), -22.5 to 22.5°(El)	-30 to 66°(Az) -17 to 22.5°(El)
Particle Species Energy Range	Electrons 1 eV to 5 keV	H+,He++ 0.2 to 20 keV/q	3He - Fe 0.5 to 100 keV/q(Az) 0.5 to 16 keV/q(El)
Measurement Parameters	e- flux v, e distribution 3D	3D velocity distribution	energy, charge, mass, direction 3D
Angular resolution	10°	2°	6°
Energy Resolution	12 %	8 %	6 %
Cadence	4 sec	4 sec	30 sec

3.4.3 Fluxgate Magnetometer [4]

The magnetic measures the magnetic field vector in situ. The instrument uses two triaxial fluxgate sensors which allow separation of stray field effects of the spacecraft from the ambient magnetic field. One of the sensors is mounted on a 3m boom while the other sensor is directly attached to the spacecraft.

The sampling rate is 1 Hz for normal operation. Values are averaged to one minute values.

The instrument design is based on the Fluxgate Magnetometer (MAG) of the Venus Express Mission. Similar instruments have also been flown on Rosetta Lander and the Mir Space station. **Figure 7 - Magnetometer with Sensor 1 (a), Boom (b), Controller Electronics (c), Sensor 2 (d) from Zhang et al. (2005)** shows the Magnetometer configuration of Venus Express including the boom. The range is ±262nT and the resolution is 16 pT.

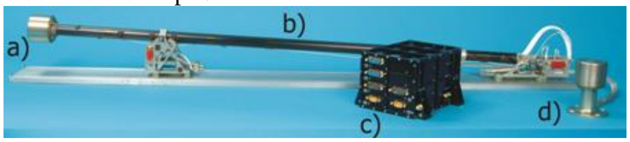

Figure 7 - Magnetometer with Sensor 1 (a), Boom (b), Controller Electronics (c), Sensor 2 (d) from Zhang et al. (2005)

3.4.4 Instrument Operation

In operational Mode, all in-situ instruments will continuously take data while only two coronagraphs are operated simultaneously. The operational coronagraphs are chosen so that the trajectory of the Earth-bound CME can be determined.

3.5 Attitude Orbit Control System and Propulsion

3.5.1 AOCS

The reaction wheels are mounted in a tetrahedral configuration; attitude control can be achieved with four wheels operating simultaneously (the nominal operational scenario) or any combination of three wheels. AOCS comprises three star trackers (one in cold redundancy) on a common mounting integrated with an inertial measurement unit (IMU), rate measurement units, Sun sensors and 4 reaction wheels. During science operations, at least two STRs will be used in combination. In the event of major system anomaly on the spacecraft and consequent loss of attitude control, dedicated shutters will protect the STR optical paths to prevent damage due to accidental Sun pointing.

- **CPU (Control Process Unit)**
- **3+1 Momentum wheels intg. electronics (values per wheel)[7]**

Table 4 - Technical Details

Model	Rockwell Collins European - RSI 4-75/60[8]
Rating	Satellites from 200kg to 1000kg
Config:	Tetrahedral[9]
Mass	3.7kg (+-6000 rpm max. and Torque = 75 mNm)
Power	<20W (Holding) 40W(Nominal)
Dimension	0.222(W) x 0.222(H) x 0.087 (D) m
Lifetime	>15 years

- **1+1 IMU [Inertial Measurement Unit with integrated FOG][10]**

Table 5 - Technical Details

Model	Astrix 200
Mass	13kg
Power	6W
Dimension	295 x 150 x 145 mm and 330 (d) x 280 (h) mm
Accuracy	<0.5 arcmin (Safe Mode Pointing Accuracy)
Heritage	CNES for Pléiades, ESA for Aeolus, Sentinel 2 et al.
Lifetime	15 years
Radiation	15 krad total dose SEP tolerant, latchup immune

- **1+1 Sun Sensor[11]**

Table 6 - Technical Details

Model	Selex S3
Tech. Level	OTS
Mass	<330g
Power	0.7W
Dimension	112 x 12 x 43 mm
Heritage	ESA GOCE, SICRAL-1B, Lisa (Future), ESA Pathfinder
Lifetime	>15 years
Radiation	300 Krad with Detector 1Mrad, SEU tolerant, Latch-up free

- **2+1 Star Trackers (values per tracker)[12]**

Table 7 - Technical Details

Model	Selex AA-STR
Mass	2.6kg
Power	5.6W (Nominal) 12.6W (Peak)
Dimension	164(L) x 156(W) x 348(H) mm
Heritage	ISO, SAX, SOHO, Cassini, XMM, Integral, SAC-C et al.
Lifetime	18 years
Radiation	GEO orbit radiation shielding

- **12 RCS Thrusters (values per item)**

Solar Wind Estimates: approximately 400km/s and 4.6uPa[13]

Antenna Torque: I = 30uNm << Thrust capability[14]

Table 8 - Technical Details

Model	4N Astrium Hydrazine BiPropellant Vernier Thrusters[15]
Mass	0.2kg
Power	0W
Dimension	142(L) x 19(r) mm

- **4 x Drive Thrusters (Values per item)[16]**

Table 9 - Technical Details

Model	22N Astrium Hydrazine BiPropellant Thrusters
Mass	0.650kg
Power	0.55W
Dimension	212(L) x 55(r) mm
Heritage	Giotto, Skynet 4B, MeteoSat et al.
Lifetime	15 Single, 70 Accumulated hours total burn

3.5.2 RCS (Reaction Control System) and Propulsion System

One mission requirement for the spacecrafts is that they must be separated by 60 degrees in their orbit around the Sun. As discussed in the orbital section, this means that the period of the transfer orbits is very important to allow for efficient insertion into the desired orbit with the correct phase.

As stated by one of the mission drivers, the orbital radius of the spacecraft must be at least 0.7 AU in order to achieve the requirements of the minimum warning time of

3.5 hours. Subsequently, the mission will have a high delta-v which means the mass of the propellant system will be high. The maximum delta-v required for one of the spacecrafts is 1019m/s.

Table 10 - Trade-off study for propulsion

Parameter\Engine	Electric	Chemical
Time to reach required final state	8 years	4 years
Power consumption	4320 W	2 W
Thrust	0.12 N	420 N
Fuel mass	32 kg	275 kg
Total wet mass of Satellite	~670 kg	~800 kg

In conclusion, the mass decrease from ion propulsion to chemical propulsion is not sufficient enough to justify the vast increase in power consumption and orbit insertion time. Chemical propulsion allows for the satellites to be inserted into the correct orbit with the required phase difference. Consequently, CARETAKER will use chemical propulsion.

3.6 Power

3.6.1 Overview

The electrical power sub-system provides, stores, distributes and controls the spacecraft. The primary power sources are considered the solar panels, while the secondary ones are the batteries. A brief explanation of the usage of these systems during sub-missions is presented in the table above.[17]

Table 11 - Power distribution per sub-mission

Sub-mission	Power required	Power supply	Components
Trajectory to Orbit	5W 780W	Solar Panel	Propulsion system All the sub-systems except the instruments
Operation without propulsion	600W 20W 20W 110W	Solar Panel	AOCS Instrumentation CPU Communication
Orbit correction	7W 600W 15W	Solar Panel	Thrusters AOCS CPU
Safe Mode	15W 2W	Battery	CPU Sun tracker
Launch	15W 10W	Battery	CPU Communication

3.6.2 Primary Power Sources

For solar cells, 28% Triple Junction GaAs Solar Cell Type: TJ Solar Cell 3G28C were considered the best options in terms of efficiency and lifetime. Considering the total amount of power needed to operate the spacecraft, the area of the solar panel is $3m^2$.

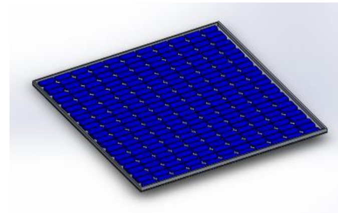

Figure 8 - Solar Cells

Panel Structural Mass	**6.4 kg**
Mass of single Cell	**0.002 kg**
Number of cells per panel	**264**
Panel Mass	**0.685 kg**
Number of Panels	**2**
Overall mass for solar Power	**14 kg**
Area	**3m²**

3.6.3 Secondary Power Sources

Batteries are used as a secondary power system, operating when the solar panels are not available. Taking into account the available space-rated batteries and the mission requirements, the option is the nickel hydrogen battery because it is very stable and is known to exceed on-orbit performance requirements for long duration missions.

During the launch, the CPU and the communication requires power, up to 30W, as stated in chapter 3.6.1

- Launch Battery Operation

The batteries will still be discharged at 6% after 2h.

- Safe Mode Battery Operation

The batteries will still be charged at 52% after 24h.

The secondary power supply will contain 12 cells of NiH_2, to provide approximately 100Wh per cell.

3.7 Thermal Control System

The thermal control system maintains the spacecraft within operational temperature limits of 20°C. Our mission uses passive cooling and MultiLayer Insulator (MLI). The solar flux density at 0.64 AU is3400 W/m^2.

The diagram below explains the heating transfer from the environment to and from the craft.

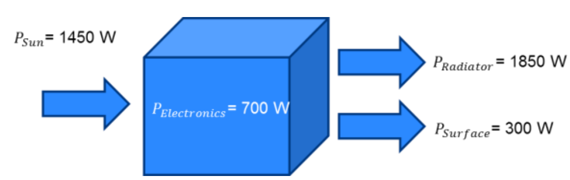

Figure 9 - Output and Input heat

Considering a surface finish of white paint Z93 with an emissivity of 0.92 and absorptivity of 0.17, the incoming heat from the Sun is 1420 W [17]. By using 10 layer MLI blankets with an effective emittance of 0.05 to cover the spacecraft, the heat radiated out of the spacecraft will be around 300 W. The remaining heat must be radiated out by radiators, and by assuming a temperature T_i of 300 K inside the spacecraft, a temperature T_j of 4.2 K outside the spacecraft and a radiative view factor F of 1, the area of the needed radiator will be $4\ m^2$. Since the Solar Orbiter is going to use a radiator with a mass of about 3.5 kg, it is safe to assume that the mass of the needed radiator will at most be at the same level [18].

3.8 Radiation Shielding

In order to account for the increased radiation exposure in the space environment, a preliminary shielding study is performed employing the online tool SPENVIS (Space Environment Information System, http://www.spenvis.oma.be). Solar protons are dominating, hence only solar particles are considered in this analysis. As an upper limit for the total ionizing dose (TID) the time period between the launch date and the end of the five-year mission operation (9 years in total) is considered at a distance from the Sun of 0.7 AU. Error! Reference source not found. shows the dose as a function of aluminium shielding thickness for nine years and for the extended mission period of 14 years in total. All electronic components shall tolerate a TID of 20 krad. The shielding will be designed such as to yield a TID exposure of 10 krad, providing a factor 2 margin [19]. This results in an estimated thickness of 9 mm for the nominal mission lifetime. Under the assumption that the spacecraft itself provides a minimum shielding of 3 mm aluminium, all critical components are shielded with an individual aluminium envelope of 6 mm thickness. This results in an estimated shielding mass of 43 kg for the spacecraft (*Table* 12). Preliminary studies of the total non-ionizing dose (TNID) on the solar panels [20] that are used in CARETAKER show an estimated loss of efficiency of less than 2% during the operational time.

4. Communication Segment

4.1. Communications Scenarios & Subsystem

We have identified two communications scenarios depending on the mission phase. This is depicted in Figure *10*.

- **Launch and Early Orbit Phase** (LEOP) and **Cruise** phases. The downlink data consists of housekeeping information and also science data. Communication [18] via 2 wide beam-width **LGA**s using X-band is considered as baseline for both up 7.145-7.190 GHz and 8.40-8.45 GHz downlink links. In the safe mode the LGA provide omni-directional coverage and Telemetry (TM) is possible only until a distance Earth – S/C of 0.8 AU using 15 m antennas.
- **Nominal and Extended phases**: Both phases are similar from the communications point of view and their length is presented in section 5. The articulated, deployable High Gain Antenna (**HGA**), 1 m dish, 1.75 deg beamwidth, will have simple pointing mechanism for use in Science Mode, see ***Figure 13 - Launch and full orbit phases***

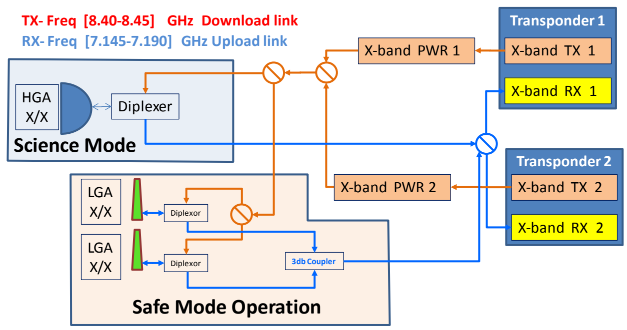

Figure 10 - Science/Safe Mode Operation

Additionally, the communications subsystem [21] will include hot redundant set of Band X/X transponders and 60W Traveling Wave Tube Amplifier (TWTA) and a RF Distribution Unit (RFDU) with diplexers, 3dB couplers and wave guides to provide the nominal communication with the Earth during different phases of the mission depicted in figure

4.2 Earth Stations

To download science and housekeeping data from all the spacecraft we propose to use 6 dedicated 15 m dishes for the CARETAKER mission. The uplink and downlink will use X Band. The geographical distribution of the Earth station in the world is shown in ***Figure 11 - Earth Station over the world***. It is foreseen to spread the stations equally over the Earth.

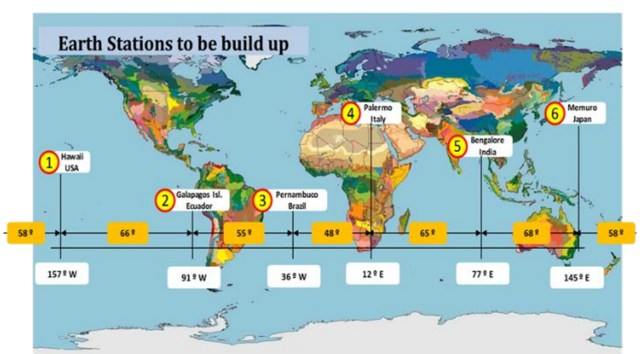

Figure 11 - Earth Station over the world

As it is shown **Figure 12 - Earth Station connection with the spacecrafts**, the science and housekeeping data is received by two antennas in parallel.

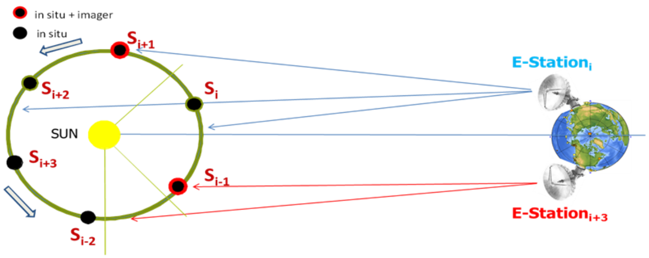

Figure 12 - Earth Station connection with the spacecrafts

4 Operation and Ground Segment

The launch will be in 25 June 2026 with a window of 3 weeks and there will be an opportunity every each 19 months.

The Trajectory Earth-Venus until the final orbit is depicted in figure A, in which data for launch, gravity assist manoeuvre (GAM), fast and low stack cruise, commissioning, final orbit for nominal and extended mission and the final decommissioning in 2040, are scheduled.

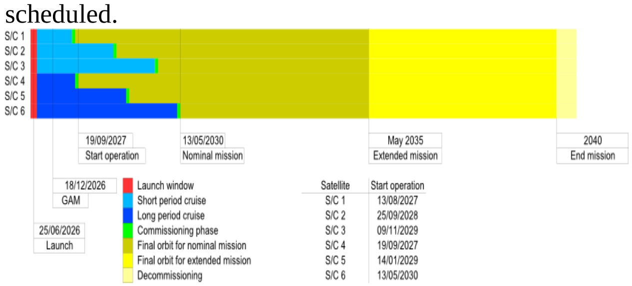

Figure 13 - Launch and full orbit phases

Mission operations concept for the mission shall minimize the costs both in the area of ground segment tools and facilities and in the sharing of manpower and expertise in the development and operations teams.

Also, it is important prior to launch, a joint approach to spacecraft system-level testing between the spacecraft manufacturer and the spacecraft operations team, and maximizing the synergy between spacecraft manufacturer and operators in the preparation of operational documentation, spacecraft user manual, operations database etc.

The Ground Segment will rely on six ground stations all around the world with two of them continuously communicating with the spacecrafts. The ground segment organization is presented below.

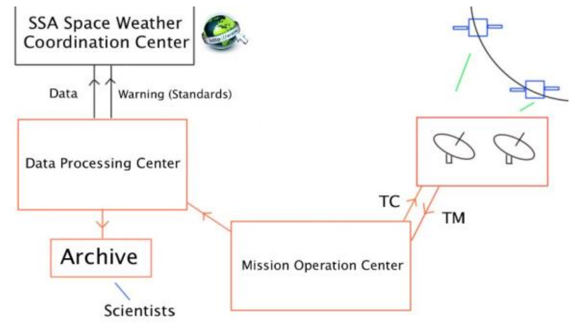

Figure 14 - Ground Segment

5 Data Processing

5.1 Mission Operation Center (Victor)

The CARETAKER Mission Operation Centre (MOC) will be in charge of all telecommand and telemetry operations of the mission. Communication with the six spacecrafts will be provided by the two ground stations through the CARETAKER Network 24/7. The whole communication schedule is given in ***Figure 15 - Data Collection from the S/C***.

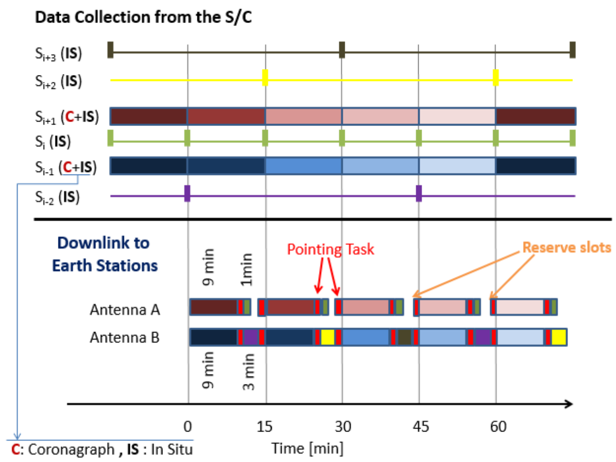

Figure 15 - Data Collection from the S/C

The Mission Operation Centre will perform 0-level data processing as well as a 30 days back-up storage of the processed data. Data will be transmitted continuously to the Data Processing Centre.

5.2 Data Processing Centre

The Data Processing Centre shall perform different tasks:

1. Calibration, Validation and Processing of the data for the early phase of the mission.
2. Provide algorithms, methods and models to a warning unit which shall be able to give standards compatible data to the SSA Space Weather Coordination Centre (SSCC). These standards require particularly a

timeliness between the measurements and the data products.
3. Perform higher level (scientists aimed) processing on the data and make it available on the Internet.
4. Archiving

5.3 Data Products

Given the extraordinary position of the spacecrafts constellation, the processed data is to become a new reference for space weather event warning as well as premium scientific content. For the In-Situ measurement, a fifteen minutes range between measurement and data product will be ensured as required by the SSCC [12]. For 3D-modelling of CMEs and Coronagraphs data, an extended time line is proposed: those data should be available within a maximum of sixty minutes after remote measurement.

Further processing will be performed to make the data fully exploitable by the scientific community in longer time delays. The whole content will be available on the Internet.

6 Budgets

6.1 Mass budget

Table 12 - Mass Budget

Subsystem	Mass / kg	Margin	Mass total / kg
Power	44.00	2.80	46.80
Payload	31.00	2.96	33.96
Communications	28.20	5.22	33.42
Onboard Data Handling / Avionics	15.00	0.75	15.75
AOCS	95.13	2.57	97.70
Thermal Control	32.00	3.20	35.20
Additional Shielding	42.67	0.00	42.67
Chemical Propulsion System (dry mass)	44.60	4.33	48.93
Harness (5%)	16.63	0.00	16.63
Structure (20% of dry mass)	69.85	0.00	69.85
TOTAL (dry, without system margin)	419.07	21.83	440.90
System Margin (20 %)	83.81		
TOTAL (dry, with margin)			524.71
Propellant			275.32
TOTAL (wet mass)			800.03

6.2 Power budget

Table 13 - Power Budget

Subsystem	Power Consumption / W	Margin	Power Consumption total / W
Power	48.0	0	48
Payload	18.5	1.8	20.3
Communications	165.0	5.3	110.3
Onboard Data Handling / Avionics	17.0	0.9	17.9
AOCS	211.2	105.6	316.8
Thermal Control	0	0	0
Chemical Propulsion System	5.0	1.0	6.0
TOTAL (without margin)	464.7	114.5	579.2
System Margin (20 %)	92.9		
TOTAL (with margin)			683.1

The power supply will be provided by the solar panels.

7 Risk and Cost Analysis

In principal, the **probability (P)** of an undesired event during the operational phase of the mission increases with the number of spacecraft. On the other hand the **impact (S)** of such an event is mostly less crucial. The risk of an event is classified by the likelihood (A (minimum)-D (High)) and the impact (1(minimal or no impact)-5(catastrophic)) on the mission. Additionally, adequate reactions on each event have been considered.

Very low

- **Launch fails**
- Blackout of a measurement device
- Failure of one coronagraph
- Failure of one magnetometer
- Failure of one SWA

Low

- Launch misses time window
- One spacecraft cannot recover from the safe mode
- Collisions during the deployment
- Not enough data rate during safe mode to ensure communication between spacecraft and ground station
- Damage of sensitive optics of a coronagraph during space flight
- If the spacecraft enters safe mode in a 40° range on the backside of the Sun, connection will not be possible in this area.
- Pixel losses in the coronagraph

Medium

- Slightly miss the trajectory for Venus gravity assist

Solutions to the risks are outlined in the presentation.

Costs

The total mission costs are estimated to be $ 1.5 billion with an uncertainty of 20%. This includes space segment (60%), ground segment & operations (30%) and launch (10%). Since the mission requires a continuous communication to six spacecraft in deep space for nine years, the ratio for ground segment & operations is unusually large. It also drives the major uncertainties of the total costs.

8 Conclusions

CARETAKER will be the first multi-spacecraft, deep space mission with continuous 360° coverage at 0.7 AU, far closer to the Sun than previous missions. The set of multi-point data from magnetic field measurement, plasma composition and remote sensing have the potential to significantly advance our understanding of heliospheric physics. The first near real time interplanetary medium data at this distance between the Sun and the Earth is a unique ambitious approach for space weather.

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Abbreviations
AOCS – Attitude Orbit Control System and Propulsion
AU – Astronomical Unit
CME – Coronal Mass Ejection
OECD – Organization for the Economic Co-operation and Development
EAS -Electron Analyser System
HGA - High Gain Antenna
HIS - Heavy Ion Sensor
LEOP - Launch and Early Orbit Phase
MAG - Fluxgate Magnetometer
PAS - Proton Analyser System
S/C - Spacecraft
SWA Solar Wind Analyser
SM – Safe Mode
SSCC – SSA Space Weather Coordination Centre
SSA – Space Situational Awareness

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