ExoDaedalus

Sailing through the solar wind in search of our origins

by

Kent Barbey

Instructor:	Bernard Foing
Teaching Assistants:	Belén Yu Irureta-Goyena
	Elisabeth Rachith
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Logo : the design was partly made from stock images and DALL-E 2, a creative machine learning tool developed by OpenAI.

EPFL

ExoDaedalus mission concept : solar wind sample return, Sun study and solar sail magnetic storm monitoring. K. Barbey, [1]EPFL 1024 Ecublens, kent.barbey@epfl.ch.

Introduction: In the early 2000's, missions to the Sun-Earth libration points L_1 and L_2 took a turn thanks to the implementation of dynamical systems theory and machine learning techniques to trajectory design. The *Genesis*[1] mission was the first one to make use of these methods with the goal of returning solar wind samples back on Earth. Unfortunately, the return capsule crashed violently in the Utah desert upon re-entry, severely contaminating the samples. Concomitantly, predicting magnetic storms due to solar flares became of interest as more and more artificial satellites orbited the Earth. In fact, a *Geostorm*[2] mission design concept was written by the JPL. These types of mission have the objective to warn Earth before a magnetic storm hits it and prepare the protection of the satellites' systems adequately. The first libration point of the Sun-Earth system is optimal in this case as it provides enough warning time, can communicate with Earth and monitor the Sun at all times. The downside being that station-keeping at this point requires a lot of fuel. Solar sails were as a result studied. The recent development of this kind of propulsion notably with *Ikaros* and *L'Garde*'s TRL 6 10'000m^2 sail[3] puts this kind of mission concept back on the table.

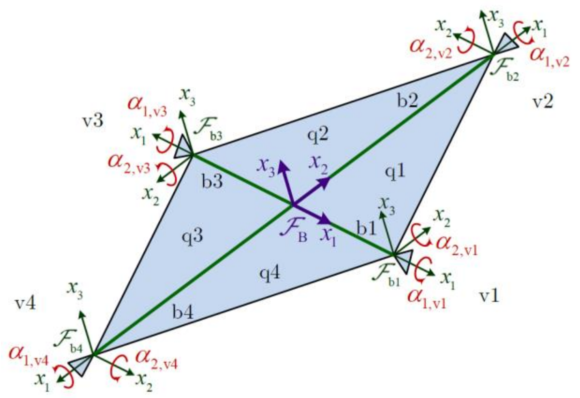
Objectives: The goal of *ExoDaedalus* is to efficiently combine the two concepts described above :

1. Return new solar wind sample to the Earth in the same fashion as *Genesis* with the *Exodus* module.
2. Provide a first solar monitoring mission using a solar sail with the *Daedalus* module.

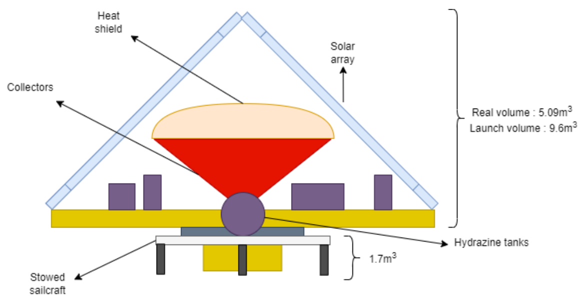
to successfully accomplish the three following objectives :

a) Return intact solar wind samples to Earth.
b) Study the solar wind in a never studied regime for a complete solar cycle(11 years).
c) Demonstrate the Geostorm mission design concept with a solar sail.

Mission scenario: The optimal mission scenario spans on 11 years with a launch by a Vega-C in Kourou, GF. The first part of the mission focuses on sample return. The S/C is inserted into a L_1 halo orbit, collects solar wind samples for 30 months and returns them back to Earth thanks to the sample return capsule. The remaining S/C is sent back to L_1, deploys its sail and inserts itself into a so-called Sub-L_1 orbit in order to study the Sun with its scientific instruments and provide solar storm monitoring. Station-keeping is provided with the tipping vanes[4] of the solar as shown on the figure below.



Spacecraft: *ExoDaedalus* is composed of two modules : *Exodus* and *Daedalus*. The former is in charge of collecting and returning the solar wind samples. The latter provides all other life and scientific systems of the S/C, having therefore the objectives to demonstrate the sail technology, study and monitor the Sun. Below a diagram of the full spacecraft in its launch configuration :



References:

[1] Martin Lo et al. (1998) *Genesis Mission Design.*
[2] Chen-Wan L. et al (2004) *Solar sail warning mission design*.
[3] David Lichodziejewski et al. (2003) *Bringing an effective solar sail to TRL 6*.
[4] Soroosh Hassanpoor et al (2018) *Linear structural dynamics and Tip-Vane attitude control for square solar sails.*

Introduction

Motivation

When imagining sample return missions, one usually thinks about going to a planet, moon or comet/asteroid of our solar system, grab a pile of dust and come back. But gathering samples from our star is possible too. It is even necessary in order to precisely measure the different abundances of elements we find in its solar wind. This wind, not having mixed with the core of the star, is a witness of our solar system's origins. Solar wind sample return is therefore an answer to the so-called *solar nebula hypothesis*. This type of mission has already been done in the early 2000's with the *Genesis* but it unfortunately failed when the sample return capsule didn't deploy its parachute upon reentry and crashed into the desert, altering severely the quality of the samples.

This mission went to a very particular point of the Sun-Earth system : the L_1 libration point which is very interesting for heliophysics missions thanks to its practically continuous communication window and sun exposition. It is therefore interesting to combine this sample return mission with a solar observation one. This was originally the extended mission of *Genesis*[1] but it was discarded during mission design[47]. It is actually really easy to do, trajectory-wise and fuel-wise, thanks to the better understanding of dynamical channels in our solar system.

Finally, this mission aims at testing a solar sail. The L_1 vicinity is also interesting for geostorm warning missions. These missions aim at informing Earth of the apparition of a geomagnetic storm susceptible to damage satellites in orbit. For that, the positioning of a spacecraft at a 0.02 A.U. distance of the Earth on a sub-L_1 orbit would be perfect in terms of warning time as it is shown on **Fig.** 1. Station-keeping at this point in however difficult because it requires at lot of propellant. A solar sail is therefore very adapted.

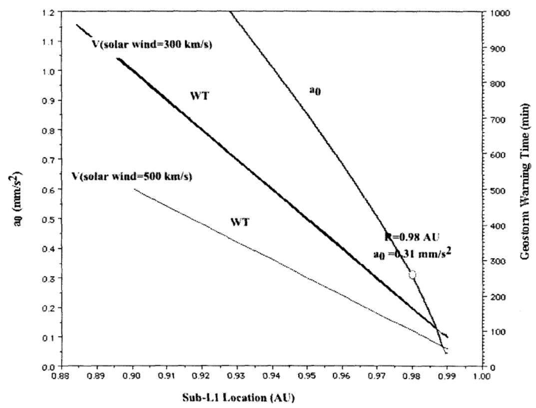


Figure 1: Warning time for sun geostorms in function of distance to the Sun and solar wind speed.

On the name and visual design

The name of the mission comes from the concatenation of *Exodus*, the next book after *Genesis* in the bible and *Daedalus*, father of *Icarus* and legendarily associated to the invention of sails.

[1] It was supposed to take place at a distant retrograde orbit instead of the Sub-L_1 used here.

Contents

1
Mission definition

1.1. Mission type

The *ExoDaedalus* spacecraft is a Sun sample return and Sun observation probe consisting of two main modules : *Exodus* and *Daedalus*. The first one consists of a set of solar wind collectors and an Earth sample return capsule(SRC). The second one is a sun observation probe propelled by a solar sail and hydrazine thrusters. The goal of the mission is to complete and improve the task the *Genesis* mission partially failed to accomplish in the early 2000's : return solar wind samples to Earth[43]. Its composition is telling of our star system's formation. This mission is however not only a scientific one but also a technological demonstrator. Indeed, after the return capsule separation, *Daedalus* puts itself into a never tested sub-L_1(SL1) orbit and acts as a long-term sun observation probe in an unexplored solar wind energy regime[47] to provide scientific measurements and solar storm monitoring. This last part is inspired from the *Geostorm mission* NASA's JPL designed twenty years ago[63]. Station-keeping for this last part is controlled by a solar sail. The first part of this mission is therefore greatly inspired to the GENESIS mission(see [36]). Some modifications have been made in order to accustom the science and technological objectives.

1.2. Launch date

The launch window depends only the sample collection phase of the mission namely getting to the wanted halo orbit around the first Sun-Earth libration point(L1). Therefore the launch windows are flexible and are of the same length as for the original *Genesis* mission. Indeed, there is a cumulative 6-month window per year and there is only one small deterministic Δv needed for insertion into the halo orbits : $6 - 36[\frac{m}{s}]$ depending on the launch date[3][36]. As a consequence, the A,B,C and D phases contribute the most to constraining the launch date which is determined to be NET mid-2029.

1.3. Mission duration

The mission duration is divided into five phases and depends strongly on the launch date as optimal trajectory design for libration points missions are very timing-specific since they arise from dynamical systems theory modelling. The mission is also divided into two sub-missions. *Exodus*, from launch to sample return, could last 4 years. After separation, *Daedalus* is station-kept on a sub-L_1 orbit for sun observation and its mission duration can last until its instruments/propulsion system fail and therefore depends only on the durability and health of *Daedalus*, the remaining spacecraft. The sun observation part shall last a complete solar activity cycle *i.e.* 11 years. In total, the mission duration is expected to be 4 years for the sample return and 7 more years for the sun observation probe so 11 years.

1. Launch phase

The launch phase lasts a month. The spacecraft is inserted into a transfer trajectory to halo orbit around the Sun-Earth L_1 point. Follows three months of transfer. Then the spacecraft performs its only deterministic maneuver for Lissajous orbit insertion(LOI).

2. Collection and science phase
Solar wind sample collection using *Exodus* and solar wind study using *Daedalus'* instruments can begin directly and shall last between 2 and 3 years depending on the science requirements evolution such as the solar wind sample concentration needed and on the final Lissajous orbit period obtained. As a reference, *Genesis'* collection lasted 29.3 months for 5.86 months period orbit[36].

3.Return phase
Return from L_1 shall last 3-6 months depending on the trajectory design selected.

4a. Recovery phase
Sample capsule recovery phase starts as soon as the reentry targeting maneuvres 1 month prior to reentry are performed. The divert maneuvre before reentry separates both spacecrafts.

4b. Sub-L_1 transfer phase
Daedalus is then sent to a sub-L_1(SL1) orbit thanks to its solar sail after returning to L_1 through the channel linking L_1 and L_2.

5. Second science observation phase
Daedalus performs its sun observation program from SL1 for at least the next 7 years.

1.4. Mission objective
The mission objectives can be separated in two groups.

1.4.1. Scientific objectives
Primary goal : Return safely samples from the solar wind to constrain the models on our understanding of the solar system's formation. In particular, give an answer to the solar nebula hypothesis.

Secondary goal Perform a long-term monitoring of our sun in order to achieve two objectives : study the solar wind in a never probed and free from Earth's magnetic field region(0.01-0.05 [AU] from Earth towards the Sun) and prove the feasibility of geomagnetic storm warning missions.

1.4.2. Technological objectives
Practical application of using a solar sail for L_1 to SL1 orbit transfer and station-keeping.

1.5. Mission class
ESA : Under ESA's classification[12] and based on the previous *Genesis* and *IKAROS* missions, *ExoDaedalus* falls into the M-class with a budget of 350 million euros taking into account a 38.58% cumulative inflation in the 2000-2020 time span[29].

NASA : Under NASA's risk classification[45] and based on the previous *Genesis* and *IKAROS* missions, *ExoDaedalus* falls into the B/C-class. Failure of the first part of the mission results in a loss of some key national science objective. This justifies the B-class. Failure of the secondary part of the mission is increased due to still novel technology(Solar sail) and complex trajectory design. This justifies the C class.

1.6. Redundancy
Redundancy is necessary to ensure the mission's success. For each sub-mission, here are the key components that need it :

Exodus(SRC)
- Collectors : Five sets of solar wind collectors are deployed. The gold collector is not implemented in this mission as it was recovered intact from *Genesis*.
- Batteries : Two cell batteries are used for avionics and electronics for descent and reentry phases.

- Accelerometers : Learning from *Genesis'* story[43], accelerometers are designed in such a way that the misconception of one doesn't affect parachutes deployment.

Daedalus

- Communications : Three antennas are deployed. Two for low-gain communications and tracking with Earth and one medium gain for high-rate scientific data transmission.
- Propulsion : For translational and attitude maneuvers, as for *GENESIS*, eight 1 [N] thrusters are used for small Δv maneuvers and attitude corrections and four 22 [N] for large maneuvers. The solar sail cannot be duplicated and in case of its failure, these engines will take relay until exhaustion of their fuel reserves. This mitigates the risk of total failure of the second part of the mission.
- Attitude knowledge is ensured thanks to a star camera which is backed-up by two axis Sun sensors and two spinning Sun sensors[36]. This is a key component as sample collection can be done in a certain pointing internal with respect to the Sun.

1.7. Other mission constraints

The biggest mission constraint concerns the trajectory design. Indeed, complex computations are needed. Hopefully, many different suitable and attainable trajectories can be computed thanks to the better understanding of libration points mission design development in the last two decades. However, some difficult constraints are put on the trajectory :

- SRC recovery : Recovery of the SRC is done by helicopters anchoring the chute of the capsule as it is descending as the solar wind collectors are fragile and require special care. This asks for day return trajectory as for the *GENESIS* mission.
- Computation of low-energy return trajectories
- SL1 orbits are still fresh ideas although better understood.

Another heavy constraint comes from the size of the sail needed to be constructed in order to achieve the characteristic velocity needed and therefore the weight of *Daedalus* is to be minimised. Indeed, for a square sail with certain characteristics, a 122 $[m]$ side length is needed here. This point is developed in **Sec.** 9 and 12.

1.8. Heritage

This mission takes a significant amount of heritage from previous missions in order to reduce cost, risk and development. In addition, several COTS parts are added for the thrusters, antennas, EPS and batteries.

Genesis The SRC, its collectors and software are inherited from the *GENESIS* mission. This mission was already inspired from the *Stardust* and *Mars Surveyor 98* missions. Telecommunications are performed thanks to the same four low-gain antennas(LGA) and one medium-gain S-band(MGA) antenna. The bus of the spacecraft is designed on the same geometry in order to accommodate the SRC more easily. The hydrazine thrusters for large maneuvers and attitude control have the same manufacturer, thrust and main characteristics but are the newer versions which are essentially lighter and have slightly better performances.

Solar Orbiter The solar study suite is taken from the *Solar orbiter mission*. Namely, the *Solar Wind Analyser*(SWA) suite and the fluxgate vector magnetometer are inherited from this mission. The control and data handling for science instruments is inherited from *Solar Orbiter* too.

LISA Pathfinder Solar array cell technology is inherited from the *LISA Pathfinder mission*.

L'Garde The solar sail architecture is inspired if not completely inherited from the *Sunjammer* design developed by L'Garde[19]. The 10'000m^2 sail was at a Technology Readiness Level(TRL) of 6. R& D is therefore needed in order to reach a minimum 8.

2

Project organisation

2.1. Phase A

This phase consists in doing the preliminary analysis meaning finishing the feasibility studies, trade-offs and preliminary design. In order to minimise risks and costs regarding the mission class, this part shall last a year. R&D for the solar sail is mostly imputable to this duration. Trajectory design, though complex, is costless in time. This is to compare to the 2 months allocated to the *Genesis* mission.

2.2. Phase B

This phase comprises the design definition, component tests and all system and subsystem level specifications. This part shall last 1.5 years.

2.3. Phase C

This part aims at a detailed definition of every component of the mission, all subsystem and system tests and the final specifications. *GENESIS'* took 1.5 years for this part so *ExoDaedalus'* C phase shall last 2 years. Again, some consequent margin is taken in order to ensure the development of the solar sail. The final industrial and scientific contractors are selected. Project management for *Exodus*(SRC), telecommunications(*Deep Space Network*) and SRC recovery are given to *NASA*. Project management for *Daedalus*, science instruments, sun monitoring and launch are given to *ESA*. Solar sail development is given to *L'Garde*'s.

2.4. Phase D

The production, assembly, qualification and acceptance tests are performed as soon as the test readiness review and flight readiness review are accepted. The launch windows being flexible, no pressure is put on a tight schedule as it could be the case for interplanetary missions. This phase could last up to 2 years due to last difficulties that can be encountered during production and assembly.

Fig. 2.1 shows the timeline of the project. Phase E is the launch which can be postponed but cannot be done earlier than phase D's assumed end : mid-2029.

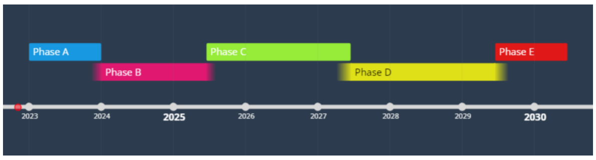


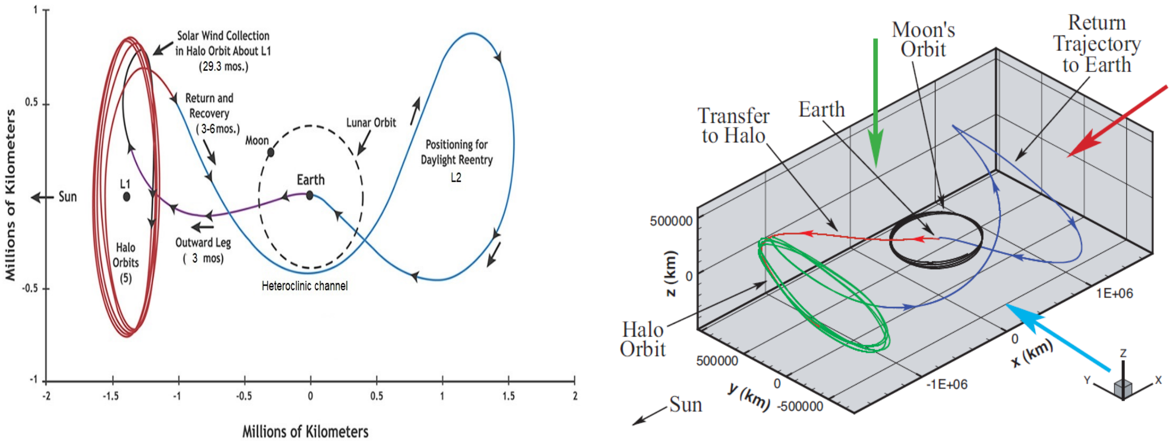
Figure 2.1: Timeline and project organisation

3
Mission design

A complex part of the mission design is taken by the trajectory design of the spacecraft which requires the application of dynamical systems theory. The trajectory design can only be done through numerical computations and requires a very precise timing procedure. The computations have to be done again months prior to the launch to ensure the most up to date parameters in the trajectory model simulation. A general outline but still rigorous trajectory design could be created thanks to numerous examples and computations done for very similar mission scenarios[9][14][20][23][37][38][40][60][62][63].

3.1. Trajectory and orbital design

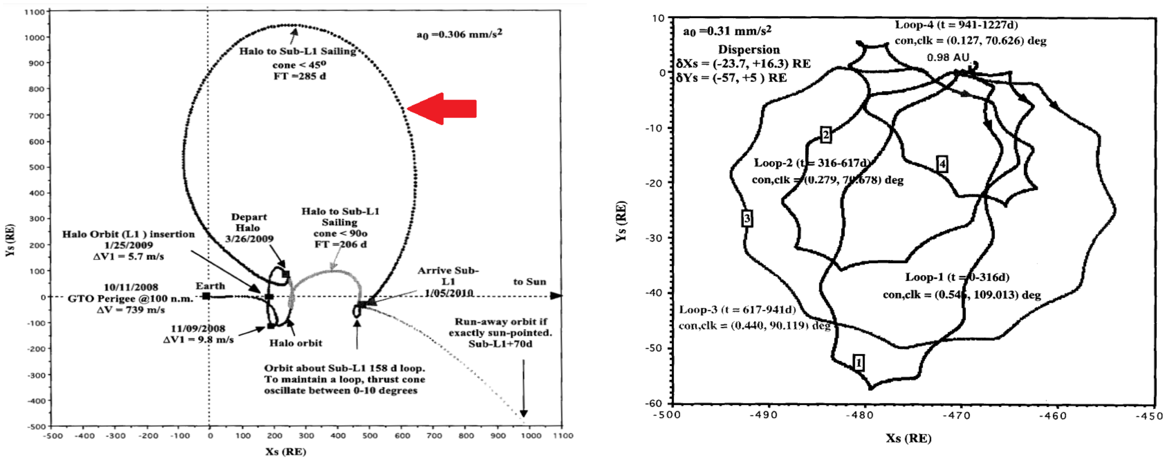
The trajectory design of *ExoDaedalus* is separated into three phases. The first sample collection part which shall be similar to the initial *Genesis* mission as illustrated on **Fig.** 3.1a. The Ax, Ay, Az amplitudes for the final selected Lissajous orbit around L_1 are, in km : $(25000, 800000, 300000)$ for a period of 5.86 months[14].



(a) (b)

Figure 3.1: (a) *ExoDaedalus* flight plan for the first part[36]. (b) *ExoDaedalus* flight plan for the first part in 3D from [40].

After launch, the S/C uses a single deterministic maneuver of $\Delta v = 6 - 36[\frac{m}{s}]$ for LOI insertion on a stable manifold ensuring its Lissajous orbit. After 5 loops around the L_1 point with a total of 2-4 small correction maneuvers per period of the $1\ [\frac{m}{s}]$ order[36], the spacecraft begins a free-return by letting itself drift on the unstable Sun-Earth manifold leading to an insertion into the heteroclinic $L_1 - L_2$ channel initiating a back-and-forth between these two points. After reaching L_2 and an entry target maneuver, a deboost maneuver is performed and the SRC ejects in order to initiate the capsule's reentry into Earth's atmosphere during the day. Daedalus performs a reboost in order to come back to a L_1 transfer trajectory. The solar sail is deployed. **Fig.** 3.2a shows the two scenarios possible for SL1 transfer. The first one is preferred with a $< 45°$ cone(see [63] for details) for its simplicity and telecommunications stability. After SL1 arrival, station-keeping is performed thanks to the solar sail in a fixed angle configuration. The final orbit is at a distance of 0.98 A.U. from the sun. This reduces the number of correction maneuvers necessary but therefore asks for a larger correction of $\Delta v = 5 - 10[\frac{m}{s}]$ every 300 days with the hydrazine thrusters. This is shown on **Fig.** 3.2b. This configuration can be kept for a maximum of 15-20 years after getting to SL1 which is largely sufficient for the mission's objectives. The orbit about SL1 is at a distance of 0.02 AU from Earth on the Sun-Earth axis with maximum dispersion around SL1 on the X and Y axes of $\delta X = (-150'000, 100'000)$km and $\delta Y = (-360'000, 30'000)$km [63].



(a) (b)

Figure 3.2: (a) *Daedalus* flight plan for SL1 transfer[36]. (b) *Daedalus* flight plan for station-keeping about SL1[63].

3.2. Eclipse, orbital period and coverage

Thanks to its unique configuration, the eclipse time at L_1 with such orbital configurations described above is negligible. This is notably necessary for geostorm warning in the second part of the mission . The orbital period for sample collection is 5.86 months. The one for the sun observation part of the mission around the SL1 point is of the order of 300 days.

3.3. Maneuvers and Delta-v budget

Tab. 3.1 shows the Δv budget for the mission[36][63]. A final Δv of 600 $[\frac{m}{s}]$ is selected.

Maneuver	Δv budget [m/s]
First leg	
LOI	6-36
L_1 Station-keeping	30
Return leg	
Station-keeping	45
ACS	75
Entry target	5
Deboost	20
Backup	100
Second leg	
Reboost	20
Sub-L_1 transfer	0
Station-keeping	140
Total	573-612

Table 3.1: Δv budget.

3.4. SL1 orbit insertion's particular case

This insertion and station-keeping requires a minimal 0.31 $[\frac{mm}{s^2}]$ characteristic acceleration from the sail if it is a perfect one[63]. The transfer orbit selected doesn't put any constraint on the tilt angle of the sail and therefore increases time from transfer but mitigates unknown sail performances for over 45° tilt angle w.r.t the Sun[63].

3.5. Launch vehicle selected, fairing

The launch vehicle selection is constrained by two factors : the fairing size and the characteristic energy needed for LOI insertion. The latter is $C_3 = -0.6$ $[\frac{km^2}{s^2}]$ and can be achieved using any medium-sized launcher. However, due to the presence of a solar sail and particularly the booms of the sail, the fairing size must be adapted. *Genesis'* used a Delta-II 7326 rocket equipped with a 2.89m fairing diameter and a length of 8.49m. It can launch 1.89t in SSO. This mission is an international collaboration between agencies(NASA, ESA) considering the heritage of the spacecraft. It is however primarily managed by ESA and an European launch vehicle is preferred. Moreover, the Delta II is no longer active. A Japanese vehicle such as the one that launched *IKAROS* (H-IIa) is not considered because its characteristics, although fulfilling the requirements of the mission, are an overinvestment. The VEGA-C launch vehicle, which performed its first successful launch in 2022, is selected thanks to its reasonable performances (2.3t in SSO) and appropriate fairing size(9.35m in length and 3.32m in diameter)[49]. Moreover, *LISA Pathfinder* was launched with a regular VEGA vehicle at L_1 with a launch mass much superior (1.9t)[34].

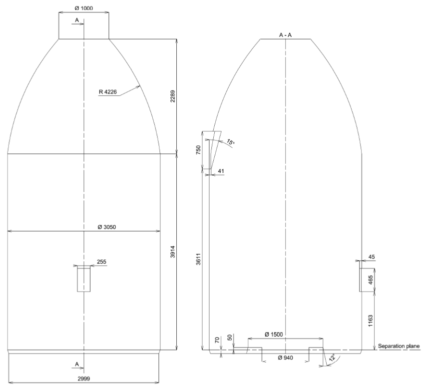


Figure 3.3: Volume available for payload with *VEGA Vampire 937* adapter.

The volume available is 30 $[m^3]$ in the VEGA-C fairing. *ExoDaedalus*'s total volume is estimated to be 10 m^3.

3.6. Launch site

The mission being primarily European and the launch vehicle the VEGA-C, the launch site selected is the *Centre spatial guyanais* in Kourou, Guyane française.

3.7. LV capability

The launcher only needs to eject the payload into a small $C_3 = -0.6[\frac{km^2}{s^2}]$ [36] which the VEGA-C is clearly capable of achieving for a payload mass of less than 1t in a single launch configuration [59] as it did for *LISA*[34]. **Fig.** 3.4 shows the capabilities of the VEGA-C vehicle in function of altitude for Sun-synchronous orbits(SSO).

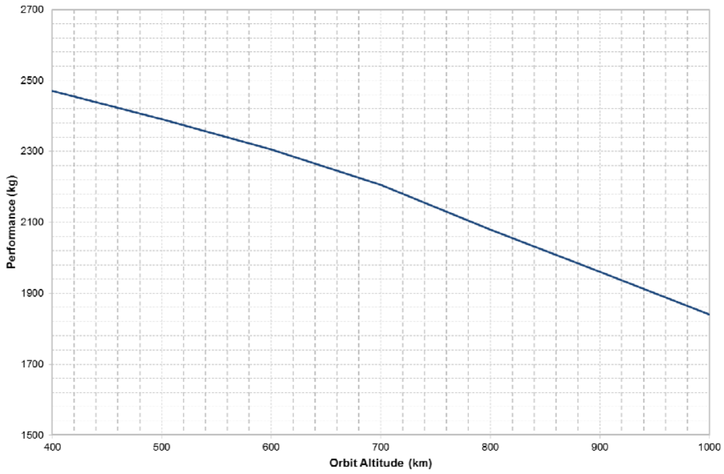


Figure 3.4: VEGA-C performances in function of altitude for SSO[59].

3.8. Mission operations scenario at mission level

Here are the operations at mission level :

1. Launch
   - VEGA-C launch vehicle inserts the payload into a L_1 halo orbit transfer.
   - $C_3 = -0.6 \; [\frac{km^2}{s^2}]$
2. LOI : Lissajous orbit insertion of the S/C with a $\Delta v = 6 - 36[\frac{m}{s}]$.
3. Science : sample collecting phase begin for 5 loops. Solar wind study using *Daedalus* instruments begin.
   - Orbit : northern class I halo orbit with 800'000×300'000 [km] in YZ plane.
   - Loops : period of 5.86 months.
   - Station-keeping on the stable manifold : 2-4 maneuvers per period thanks to the hydrazine thrusters.
   - $\Delta v = 30[\frac{m}{s}]$.
4. Return : free return thanks to the unstable manifold configuration via the $L_1 - L_2$ heteroclinic channel.
   - Station-keeping during free return : $\Delta v = 45[\frac{m}{s}]$.
   - Attitude control system(ACS) : $\Delta v = 75[\frac{m}{s}]$.
   - Backup orbit contingency : $\Delta v = 100[\frac{m}{s}]$.
5. Divert maneuver : Deboost and separation of *Exodus* and *Daedalus*. Reboost after separation to put *Exodus* back on the heteroclinic channel. $\Delta v = 40[\frac{m}{s}]$. Remaining $\Delta v = 250[\frac{m}{s}]$ goes up to $\Delta v = 397[\frac{m}{s}]$ thanks to SRC ejection.
6. SRC reentry : aiming at a day reentry.
7. Capsule recovery by helicopters. *Exodus* is completed.
8. Reboost and back to L_1 for *Daedalus* via the heteroclinic channel.
9. Solar sail deployment during the transfer to L_1. Diagnostic tests to assert the sail good functioning.
10. Sub-L_1 transfer : characteristic acceleration of the sail for transfer is $a = 0.31 \; [\frac{mm}{s^2}]$. Hydrazine thrusters in backup.
11. Science : Sun-observation phase
    - Sun science : MAG and SWA.
    - Station-keeping maneuvers thanks to the sail. Hydrazine thrusters corrections every SL1 loop.
    - Needed for the 7 years of the mission in case of sail failure : $\Delta v = 140[\frac{m}{s}]$. Expected remaining $\Delta v = 397[\frac{m}{s}]$.

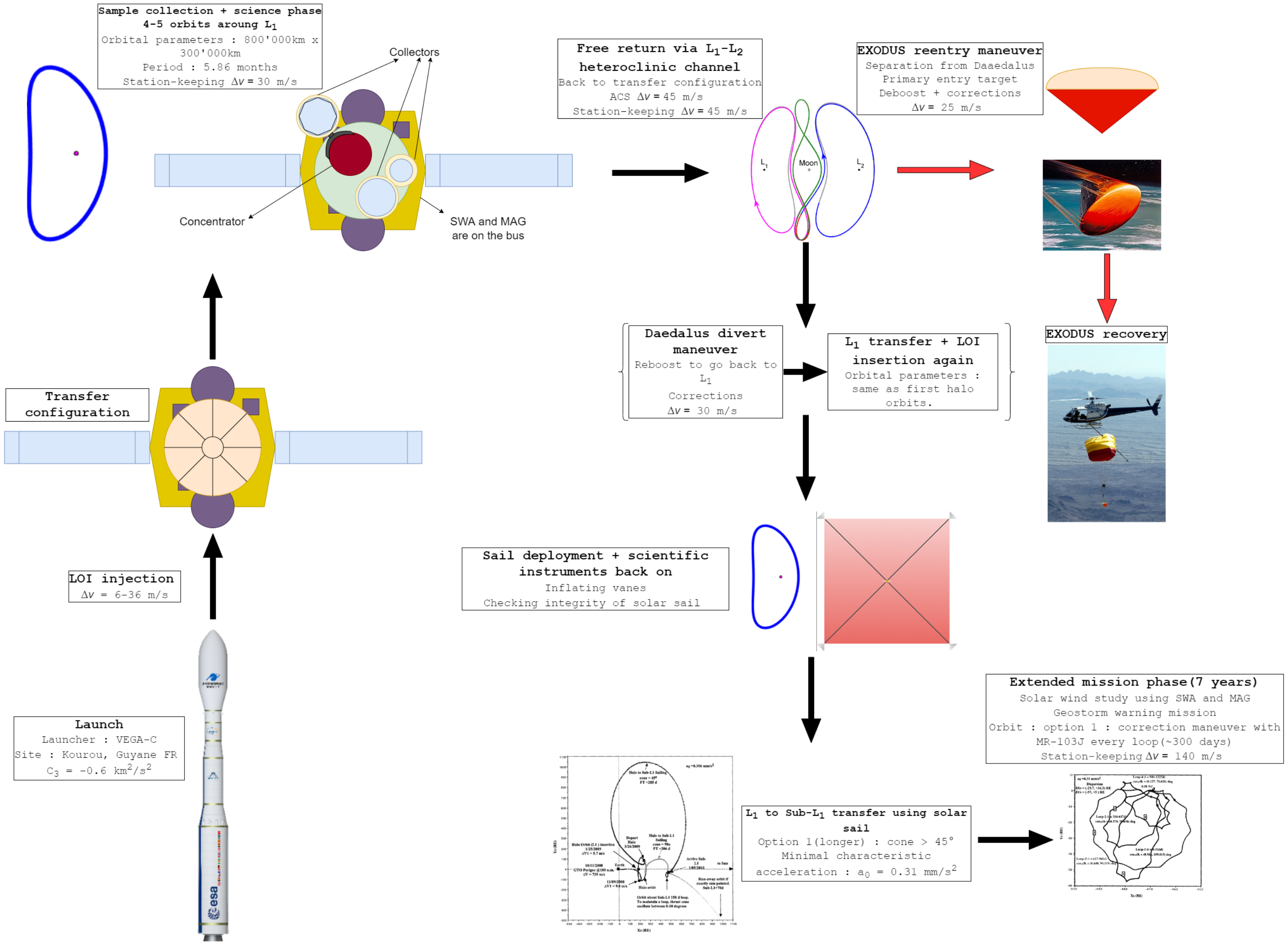


Figure 3.5: Mission operations at mission level - CONOPS

4

Space environment

The mission being substantially long, proper protection from radiations has to be applied to the space-craft.

4.1. Total radiation dose

The S/C will be impacted by three main types of radiation sources : the solar wind, solar energetic particles(SEP) and background radiation such as galaxy cosmic radiation(GCR). According to [53], missions to libration points shall use the same requirements as missions around Earth's orbit for solar wind absorption dose. A conservative analysis using the *SPENVIS* software, with models as required in [53], was performed and its results shown on **Tab.** 4.1 for the whole length of the mission e.g. 11 years. More details for radiation estimation of missions at the L_1 point can be found in [31] where a thorough investigation has been made for the upcoming *ATHENA* mission.

Total mission dose		
Target material	Si	
Configuration	Finite Al slab shields	
Al absorber thickness		**Total dose [$10^4 \cdot rad$]**
$[mm]$	$[g \cdot cm^{-2}]$	
0.05	0.014	24.52
0.1	0.027	14.44
0.5	0.135	3.77
0.8	0.216	2.43
1	0.27	1.96

Table 4.1: *SHIELDOSE 2.0* total radiation dose modelling for an eleven years mission in near-Earth interplanetary space.

According to the values obtained and considering the total dose the EPS' and batteries' manufacturer recommends not exceeding[27][28] : 30 [krad], a shielding of at least 1mm of aluminium is chosen for the sensitive parts of the S/C.

4.2. Planetary environment

Radiations : The S/C will be subject to continuous proton fluxes but also to very energetic single particles from GCR or solar flares that represent short-lived hazards near the L_1 point. If the latter were to impact the probe and cause a trigger event, it would enter into a safe mode in order to prevent complete loss : non-essential instruments such as the scientific suite are to be switched off, attitude control will be the highest priority as thermal balance and solar array orientation are essential to the S/C's minimal functions.

Micrometeoroids : Thanks to the *Genesis* and *LDEF* missions, characterisation of the micrometeoroids impact distribution on spacecrafts near the L_1 point has been done. Theoretically, spacecrafts near the L_1 point are more subject to so-called *beta meteoroids* : small particles whose gravitational attraction to the Sun is significantly offset by the pressure of sunlight. However, these eventually accounted only to 10% of the impacts on *Genesis* according to [39]. Therefore, the measurements obtained from (*Love et al, 2006*) are used for our analysis. The cumulative size-frequency distribution of craters is plotted in **Fig.** 4.1. This indicates a value of 0.08 [cm^{-2}year^{-1}] impact above $10\,\mu$m. Assuming a mean crater's diameter of $100\,\mu$m and a total surface area of *ExoDaedalus* of the order of magnitude of $150'000$ [m^2], the total number of impacts on the S/C shall be of the 10^9 order in the 11 years time span of the mission. If we assume that each impact is on a different part of the S/C, then the total solar sail area lost shall be about 150 [m^2] so 1% of the total area. This is taken into account in the sail's sizing.

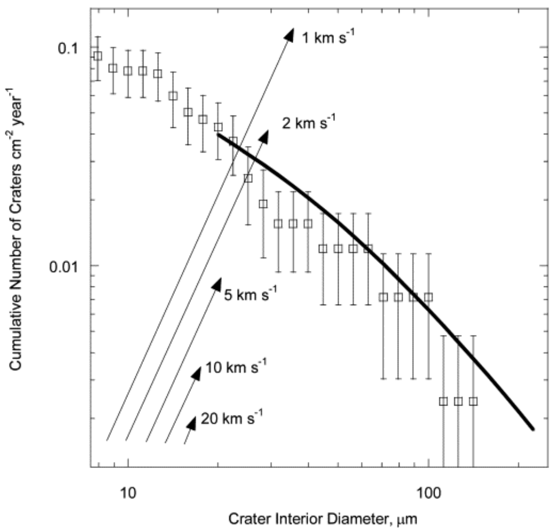


Figure 4.1: Cumulative size-frequency distribution of impact craters[39].

Impacts on sensitive parts of the S/C are more of an issue. In that case, the S/C shall enter into a safe mode ensuring correct attitude in priority. As a last resort in the second phase of the mission, the solar sail can be ejected in order to simplify attitude control which is prioritised over the solar sail demonstration objective. As mentioned, the S/C shall be able to perform its secondary science objectives of sun monitoring for at least a complete solar cycle with its remaining Δv by returning to an L_1 halo orbit.

5
System Engineering

5.1. Mass budget

The total mass budget comprises the *Exodus* module whose weight is similar to that of the *GENESIS* mission, the hydrazine fuel and the *Daedalus* module. The total mass of the spacecraft is estimated at 630.07kg. The *Mass equipment list*(MEL) is listed on **Tab.** 5.1.

Item	Quantity[.]	Estimated mass[kg]	Contingency[%]	Maximal mass[kg]
Propulsion		80.32	9.24	87.74
MR-106L	4	0.59	2	0.60
MR-103J	8	0.37	2	0.38
PEPT-590	2	7.5	2	7.65
L'Garde Solar sail	1	50	10	55
Miscellaneous	-	10	20	12
Science		27.13	5	28.49
SWA	1	23.6	5	24.78
MAG	1	3.53	5	3.71
Telecommunications		0.71	2	0.72
Low-gain Antenna	4	0.1	2	0.102
Medium-gain Antenna	1	0.31	2	0.31
EPS		14.04	4.7	14.7
IBEOS 14V Li-ion battery	5	0.38	2	0.39
IBEOS 28V EPS	1	0.14	10	0.15
Solar arrays	2	6	5	6.3
Structure		329.6	5.13	346.53
Equipment deck	1	98.6	5	103.53
Return capsule	1	205	5	215.25
Solar sail structure	1	9	10	9.9
Mechanisms	-	17	5	17.85
ADCS				10.5
Star tracker	2			
Sun sensor	2	10	5	10.5
Spinning sun sensor	2			
Command and Data handling		10	5	10.5
Thermal		15.9	5	16.7
Dry mass total		487.07	5.91	515.88
Propellant	-	143	4.6	150
Total		630.07	5.68	665.88

Table 5.1: Mass Equipment List (MEL).

The contingencies are selected under the following criteria : 2% for COTS items, 5% for inherited items, 10% for inherited items that require R&D and/or adjustments and 20% for miscellaneous. The total mass budget is highly constrained by the solar sail size and not by the launcher's capabilities. This puts a rather small contingency on the total mass budget of about 5.7%. Thankfully, most of the elements except the sail have already flown and therefore the total mass constraint is easier to achieve.

5.2. System functional and hardware database

Fig. 5.1 shows the disposition of the subsystems in the S/C and the logical links between them.

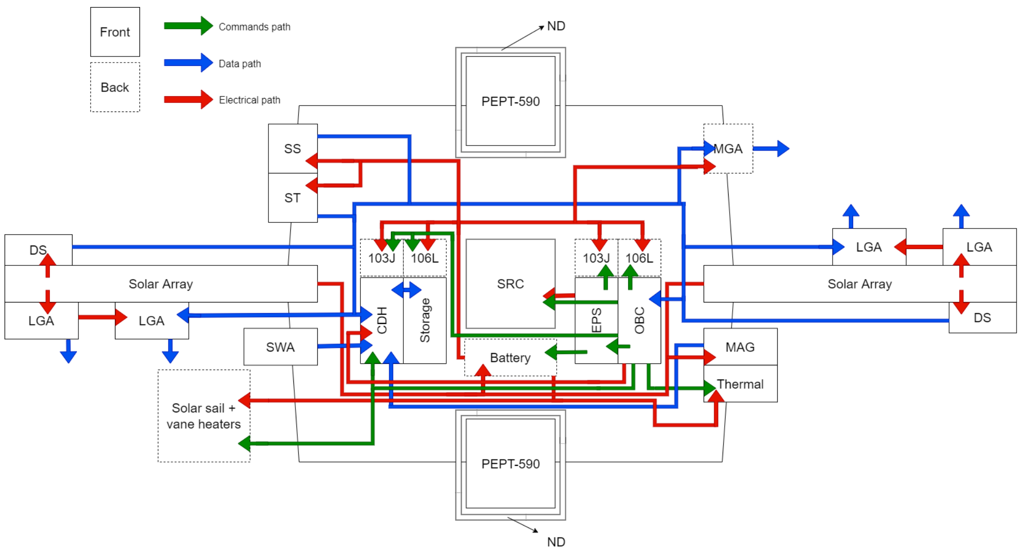


Figure 5.1: Main systems disposition diagram

5.3. Mission operations scenario at system level

Fig. 5.2 shows the *CONOPS* at system level.

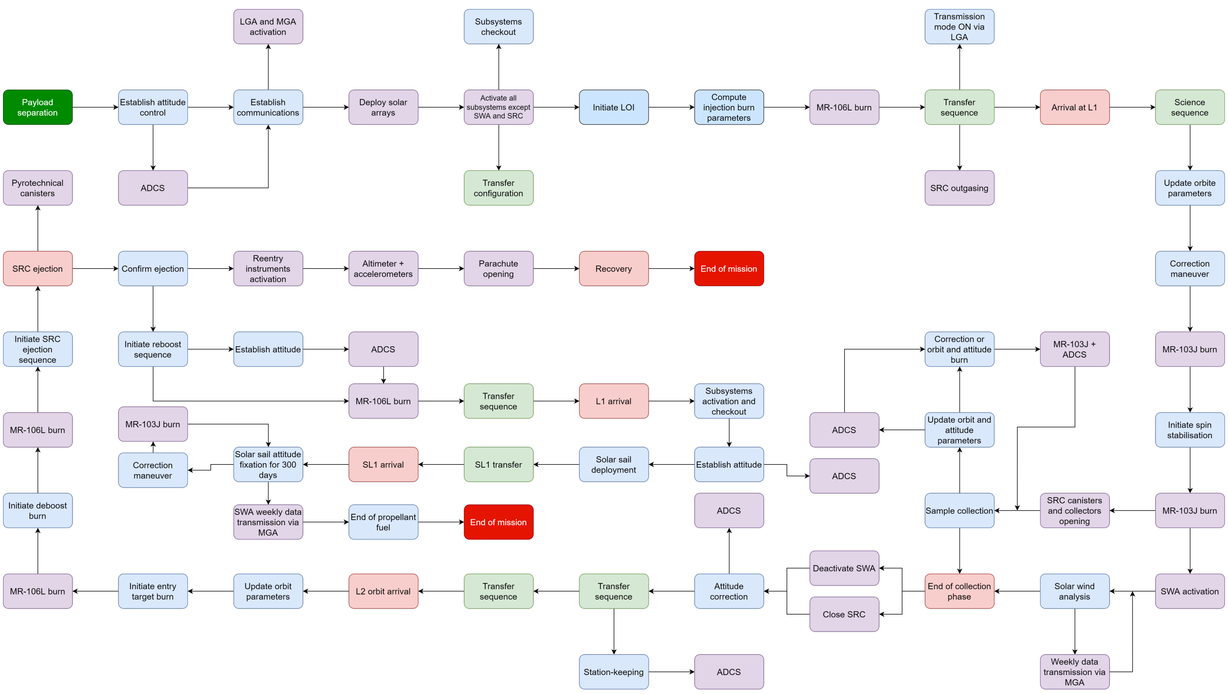


Figure 5.2: Mission operations scenario at system level : CONOPS

Sail deployment is done using inflatable carbon-fiber booms as designed by *L'Garde* in [5].

5.4. High-level requirements

Mission

- The spacecraft must collect at least 0.4 mg of solar sample with all its collectors and return it to Earth with no damage.
- The spacecraft must eject the SRC and come back to the L_1 point then SL1 point to perform solar observation and solar magnetic storm monitoring for a complete solar cycle using its solar sail as a secondary objective.
- In case of failure, the spacecraft must perform the solar observation and monitoring without solar sail.
- The S/C must have a lifespan of 11 years. If not possible, redundancy must be added.

Launch vehicle

- The launch vehicle must inject the S/C into a $C_3 = -0.6$ km^2/s^2 halo orbit about the first Sun-Earth libration point.
- The launch vehicle must integrate the S/C in its fairing.

Environment

- The S/C must be able to resist an annual radiation dose of 10^4 rad.
- The S/C must be able to resist the impact of a cumulative distribution of 0.08 cm^{-2}yr^{-1} micrometeoroids of more than $10\,\mu$m.

Payload

- The scientific instruments must function at all collection periods
- The scientific instruments must provide precise enough data of the solar activity.

Telecommunications

- The telecom subsystem must transmit at a rate of at least 18.25 kbit/s.
- The telecom subsystem must be able to transmit all scientific data once a week
- The telecom subsystem must be able to receive commands at all times.

EPS

- The EPS must provide power to all instruments during the whole length of the mission despite the ageing of the batteries and arrays.
- The power must be adequately distributed among all subsystems.
- The batteries must be able to take on the solar arrays during a short period of time before regaining sun exposition.

Structure

- The structure must accommodate the addition of a solar sail.
- The structure must resist the environment, the launch and all other mechanical and thermal constraints during the whole mission.

C& DH

- The control and data handling subsystem must share appropriately data resources between subsystems.
- The control and data handling subsystem must process fast enough the commands and data it receives.

ADCS

- The ADCS must provide accurate(0.1°) pointing of the scientific instruments and MGA.
- The ADCS must ensure pointing stability of the S/C.
- The ADCS must provide accurate attitude control of the S/C during the whole mission inclluding correct sail orientation using the tipping vanes.

Propulsion

- The *MR-103J* must provide enough thrust and reliability for small translational maneuvers and attitude corrections.
- The *MR-106l* must provide enough thrust and reliability for large translational maneuvers.
- The solar sail, although a secondary objective, must provide a characteristic acceleration of 0.31 m/s^2.

Thermal

- The thermal subsystem must ensure the different subsystems of the S/C to be within their operating temperatures.

5.5. Redundancy scheme

Most of the flying elements have strong heritage whether with past scientific missions or with commercial components. Many essential elements such as the attitude sensors, the processors and the thrusters are duplicated. All computations have contingencies and computations for sail sizing and hydrazine propellant have been done with a conservative 670kg launch weight when the actual mass of the S/C will aim at 630 kg. The most fragile component of the system is the solar sail. The S/C is designed to eject the sail in case of failure and can still accomplish its observation and geostorm monitoring mission without it thanks to its monopropellant reserves. But this would be from the L_1 point instead of SL1 which cuts the warning time for geostorm by two. It doesn't affect the solar wind observation and studying though.

6

Science and instruments

The scientific goals of this mission are divided into two parts : sample collection and long-term solar wind observation from L_1 neighbourhood.

6.1. Definition of science objectives

Objective 1 In order to confirm or not the solar nebula hypothesis, solar wind sample has to be collected and returned back to Earth. Indeed, the solar wind contains material from the original nebula since it didn't mix with the Sun's core[36]. The abundances of each element of the wind have to be measured to constraint the solar system's formation models and therefore an extensive sample of all elements has to be harvested. The *Genesis* mission collected 0.4mg of ions. The objective is to collect as least the same amount.

Objective 2 The solar wind has never been long-term studied in the energy regime between 0.95-0.99 A.U.. This is the occasion to do so. The objective is to cover a complete solar cycle of 11 years.

6.2. Measurements objectives

Sample collection : The measurement objective for the sample collection is >0.4mg of collected ions and a high-purity non-contaminated sample.

Solar wind observation(SWO) : The final goal is to better understand and model the complex behaviour of the solar wind and its consequences. The objectives for solar observation are[46] :

1. Study the sun's magnetic field with a precision ranging from far above to well below the proton gyroscale[25].
2. Measure the full 3D distribution function of solar wind electrons in the few eVs to 5 keV interval.
3. Measure the full 3D distribution function of heavy ions in the bulk solar wind.
4. Measure the full 3D distribution function of solar wind protons and alpha particle in the 200 eV/q to 20 keV/e interval.

6.3. Definition of science instruments

The collecting instruments are composed of three plates of 270 silicon elements coated with different substances in order to retrieve the different ions from the Sun[7]. A concentrator helps getting a better S/N[36]. The Sun-observation instruments consist of the *SWA* based on a top hat electrostatic analyser concept frequently used on plasma studying missions[46] :

- Fluxgate Vector Magnetometer(MAG) : in charge of objective SWO 1.
- Electron Analyser Unit(EAS) : in charge of objective SWO 2.
- Proton and Alpha particle Sensor(PAS) : in charge of objective SWO 3.
- Heavy-Ion Sensor(HIS) : in charge of objective SWO 4.

6.4. Definition of pointing requirements

Bases on [36] and [25], the solar collectors need a tilt angle between 30 and 60° around solar wind arrival direction in order to collect enough particles during the 22 months but not burn the collectors. The other scientific instruments have the following pointing capabilities[46] :

- EAS : 360° azimuth acceptance angle around the solar wind arrival direction.
- PAS : -24° to 42° Field of View(FoV) around the solar wind arrival direction.
- HIS : -33° to 66° FoV around the solar wind arrival direction.

6.5. Data volume and data rate

The scientific data is produced by the MAG and SWA instruments. According to [51], data rates for MAG and SWA are on average 1.25 kbps and 14 kbps respectively. Assuming continuous functioning of these instruments, the data volume produced per day is 1.32 Gbits so 164.70 Mo for a total of 661.17 Go for the whole 11 years mission. The volume produced isn't prohibitive and the instruments could even remain switched on continuously.

6.6. Strategy for data collection

The data of interest for solar wind measurement is outside Earth's magnetic field. Therefore the magnetometer is responsible for detecting whether data collection can be done. It is always activated. Redundancy is provided by the ion/electron monitors in case of the magnetometer's failure. Indeed, the solar wind's detectable particles should rise significantly as the spacecraft leaves the Earth's magnetic field. **Fig.** 6.1 shows the data strategy concept of operations.

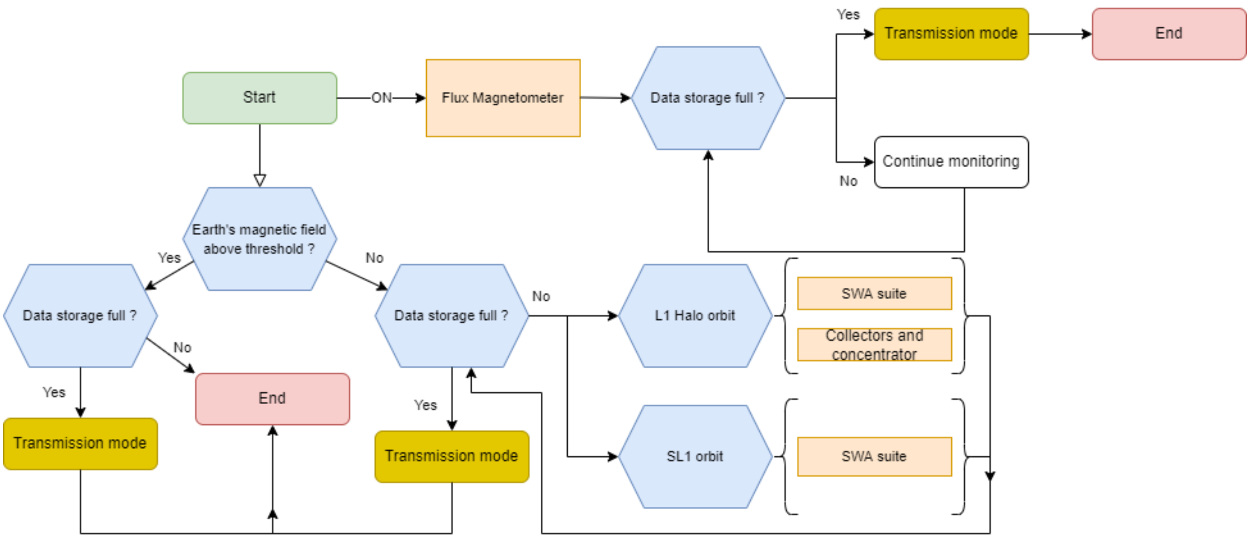


Figure 6.1: Data strategy collection CONOPS

Geostorm warning : The strategy for geostorm warning is the following. The scientific instruments will be in charge of detecting automatically geostorm worthy events such as solar flares. If such an event were to be detected, it would trigger an alarm and send it back to Earth. This strategy is already in use in multiple high energy telescopes such as *INTEGRAL*.

7

Telecommunications

Telecommunications with the S/C shall use the new programmed BWG 34m antennas of the *DSN* facilities in Canberra(Australia), Goldstone(US) and Madrid(Spain).

7.1. Upload/download frequency and data rate

The 70m antennas will probably be terminated by the time the mission launches. Therefore the 34m antennas will be used in the S-band for uplink and downlink transmissions. Tracking of the SRC during reentry is performed using a UHF receiver on ground and transmitter in the capsule. Characteristics of the telecommunications can be found below in **Tab.** 7.1.

Telecommunications	Uplink	Downlink
Frequency	2115 MHz	2295 MHz
Maximum data rate	256 kbit/s	1.6 Mbit/s

Table 7.1: Telecommunications characteristics[11].

7.2. Modulation and protocol

The modulation and protocol chosen are BPSK turbo with $r = 1/2$ and a constraint length of $K = 7$. This method is robust and is used on most similar spacecrafts in near-Earth space.

7.3. Ground stations

Considering the telecommunications and SRC heritage of the S/C, the *DSN* is the logical choice. The LGA and MGA will both use the 34m antennas in the S-band.

7.4. Telecommunications hardware and antenna design

(a)

MGA specifications	
Antenna type	Helix
Dimensions	10cm 12cm
Power	25W
Mass	110g
Polarisation	RHCP or LHCP
Gain	9 dBi
Temperature	-150°C - +95°C
HPBW	72°
Frequency	2000-2300 MHz

Table 7.2: MGA specifications

Figure 7.1: (a) Helical MGA from *AntDevCo*[2]. (b) MGA specifications.

Four patch LGA S-band antennas and a single helical MGA S-band antenna are chosen for telemetry, control and scientific data transmission. The characteristics of the MGA can be found in **Fig.** 7.1a. It is the same as the one used on *Genesis*. The transponder used is the *MST-400* from *General dynamics* because it is reliable, compact, affordable and fits with all telecommunications systems chosen in this mission. It uses BPSK modulation with $r = 1/2$ and $K = 7$ convolutional encoding. All details can be found in the data sheet in [44].

7.5. Link budget

The link budget is detailed on **Tab.** 7.2.

Medium gain antenna - S-band downlink budget		
Parameter	**Value**	**Unit**
Frequency - ν	2295	[MHz]
Wavelength - λ	0.131	[m]
Distance - R	0.01	[UA]
Spacecraft		
Antenna type	Helical 12cm $\times$ 10cm	-
Transmitted power - P_t	13.98	[dBW]
Gain - G_t	9	[dBi]
Half power beamwidth - HPBW	72	[°]
Effective radiated power - EIRP	22.98	[dBW]
Downlink path		
Free space loss - L_s	223.16	[dB]
Transmitting pointing loss - L_{tp}	1	[dB]
Atmospheric attenuation - L_a	2	[dB]
Receiving pointing loss - L_{rp}	0.1	[dB]
Polarisation loss - L_p	2.2	[dB]
Isotropic signal at ground	-205.48	[dBW]
Ground station		
Antenna diameter	34	[m]
Gain - G_r	56.74	[dB]
Receiver circuit loss - L_r	0.5	[dB]
Noise temperature - T_s	27.4	[dB/K]
Signal-to-Noise - S/N_0	79.36	[dBHz]
Link performances		
Modulation	BPSK	-
Coding	1/2 turbo	-
Bit rate	18.65	[kbps]
Bandwidth	1865000	[Hz]
Bit error rate - BER	10^{-5}	[s^{-1}]
Implementation loss	2	[dB]
Achieved energy per bit to noise spectral density - $E_s/N_{0,a}$	6.75	[dB]
Required energy per bit to noise spectral density - $E_s/N_{0,r}$	-2	[dB]
Link margin	8.75	[dB]

Table 7.2: Link budget analysis for downlink path.

8 Electrical Power

8.1. Power budget

The power budget is shown on **Tab.**8.1. The total power corresponds to the case where all elements are switched on at the same time. This is never the case especially considering that the thrusters are not used all at the same time. Solar arrays and batteries sizing is however based on this worst-case scenario.

Hardware or instrument	Contingency[%]	Allocated average power[W]
Propulsion		
8 x MR-103J	10	128.88
4 x MR-106L	10	144.4
Science		
MAG	10	25
SWA-EAS	10	31
SWA-HIS	10	
SWA-PAS	10	
Telecommunications		
2 x Low-gain antenna	10	10
1 x Medium-gain antenna	10	25
Command and data Handling	10	30
ADCS	10	10
Thermal	10	45
Total	10	449.28

Table 8.1: Estimated power budget

8.2. Sizing solar arrays and batteries

The batteries used on *Genesis* were Ni-H ones and had an energy density of $80Wh/kg$ and a capacity of 16 $A \cdot h$. The modern Li-ion batteries such as the ones used on a similar mission : *LISA Pathfinder*'s batteries had a density of 260 Wh/kg. Moreover, Li-ion batteries won't interfere with the magnetometer(MAG) as it is not ferromagnetic like Ni. As for the solar arrays, many advances have been made since then and a reduction in their size and weight can be made. The *Genesis* mission had a set of two standar silicon solar arrays for a total surface area of 2 square meters and 202W of power generated(EoL)[17]. The technology chosen here is the same as the one used on *LISA*[34] : triple junction GaAs solar cells (conversion efficiency $\eta = 0.28$) deployed into two solar arrays, one on each side of the S/C after launch. Under theses assumptions and assuming a solar constant of $S = 1370$ [W/m^2] at 0.99 A.U., the total surface needed for 450W (EoL) $+ 30$% $= 580$W is 1.5 [m^2]. We take 2[m^2] as a margin so as to not change the original geometry of the S/C.

Accompanying the solar arrays are a set of five *Ibeos 14V SmallSat batteries* with each 45 [Wh] justifying the need for five of them. Indeed, the Each have a mass of 345g and the set will be lodged opposite to Sun in order to avoid overheating. They have a maximal discharge rate of 6.5 [A] and can therefore deliver together 546W at 16.8V maximum. They can theoretically power the S/C for 30min if no power is produced by the solar arrays. Assuming that the S/C will systematically be exposed to the Sun, these batteries are sufficient for this mission. More details on their specifications can be found in [28].

A representation of the configurations the batteries can take is shown on **Fig.** 8.1. Five batteries are used so four of them are coupled into two double 90 Wh batteries and one in a single 45 Wh configuration.

(a)

Figure 8.1: (a) 45Wh single configuration. (b) 90Wh double configuration.

8.3. EPS architecture and hardware

The EPS architecture is based on the *Genesis* and *Stardust* missions which were taken from *NASA*'s *Small spacecraft technology initiative*. The EPS chosen is the *Ibeos 28V SmallSat EPS*. The EPS block diagram is shown on **Fig.** 8.2. This EPS offers regulated 3.3V, 5V, 10V and unregulated 28V buses with a 33.6V and 7A maximum voltage and current charge. It can handle 250W of maximum power. During large propulsion maneuvers with the 22N thrusters, unnecessary instruments will be reduced to minimal power consumption. More details on the EPS specifications can be found in [27].

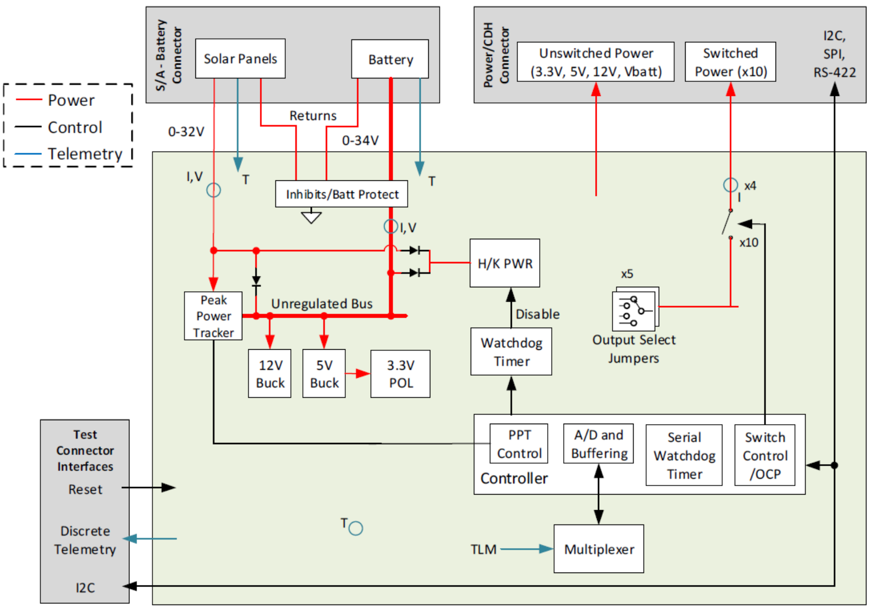


Figure 8.2: *Ibeos SmallSat 28V* EPS block diagram.

9

Structure and configuration

9.1. Main structure design and launch adapter

The S/C is composed of two main structures : a bus containing all instruments and propulsion and a sample return capsule containing the solar wind collectors.

Bus - *Daedalus* : The bus' platform is octogonal with dimensions 1.7m of diameter and 0.5m in height. It is composed of a truss of graphite-polycyanate composite tubes with titanium end-fittings like [17]. On top of the bus are disposed the instruments and the SRC. The instruments are covered with Al thermal and radiation shielding. An illustration on **Fig.** 9.1 of the old *Genesis* S/C shows the geometry adopted. Below the platform is attached the solar sail and its carbon-fibre inflatable structure. Between the platform and the sail payload are the batteries. Two solar arrays are disposed on each side of the bus. The total length of the platform with solar panels deployed is 7.9m.

SRC - *Exodus* : The SRC diameter is 1.5m in order to maximise the sample collection area[17]. It is stacked on the bus.

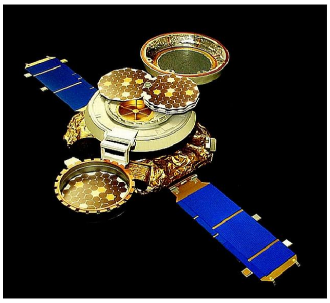


Figure 9.1: *Genesis* artist view[17]

Figure 9.2: VEGA-C single launch adapter *Vampire 937*[59]

The launcher adapter chosen considering CoG and mass is the *Vampire 937mm* for the VEGA-C. Capabilities are shown on **Fig.** 9.3.

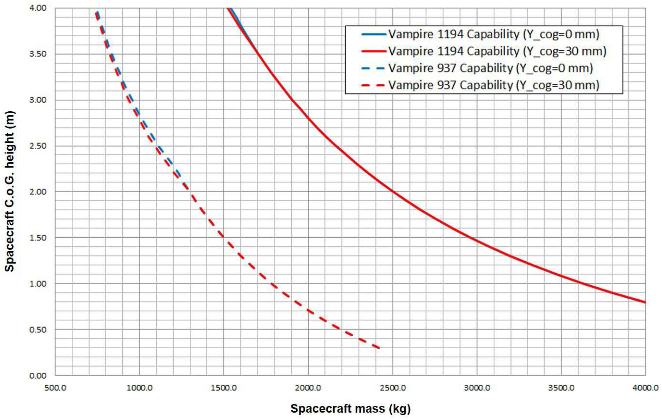


Figure 9.3: *Vampire 937*'s capabilities[59]

9.2. Main hardware boxes and elements configurations

The total volume occupied is 5.06 m^3 and an effective launch volume occupied in the fairing of 9.6 m^3 assuming the volume generated by the folding of the solar panels into a half-cone revolution shape. The available fairing volume is about 35 m^3 for the VEGA-C with the mentioned adaptor. **Fig.** 9.4, 9.5, 9.6 and 9.7 show respectively the launch, transfer, collection and sailing configurations of the S/C.

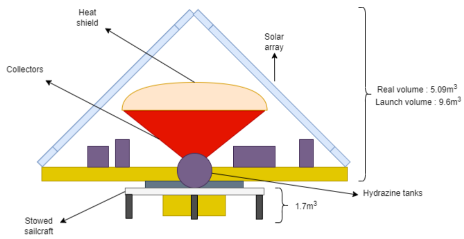


Figure 9.4: Launch configuration.

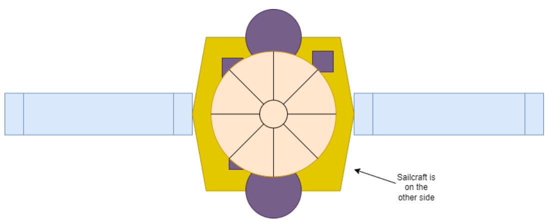


Figure 9.5: Transfer configuration.

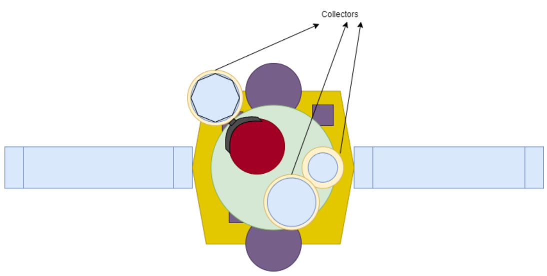


Figure 9.6: Solar wind sample configuration

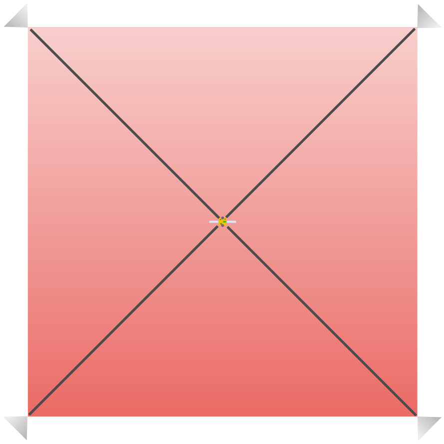


Figure 9.7: SL1 configuration

The general arrangement of the most important elements w.r.t their volume and masses is shown on the four precedent pictures. The arrangement of the instruments is already described in **Sec.** 5.2.

9.3. Mass properties

As mentioned, the bus of the S/C is made of a truss of graphite-polycyanate composite tubes with titanium end-fittings. The truss supports a composite honeycomb-reinforced deck. The SRC heatshield is a carbon-carbon one.

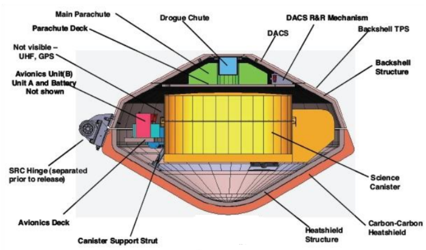


Figure 9.8: *Genesis'* SRC[17].

The density of the S/C, considering the total volume and a conservative maximum mass of 670kg is 132 kg/m^3. The density and area of the sail required is extrapolated from **Fig.**9.9 for the 1 A.U. baseline sail. The sail density needs to be 3-4 g/m^2 Mylar. The booms are made out of 62g/m carbon-fibre and are inflated for sail deployment.

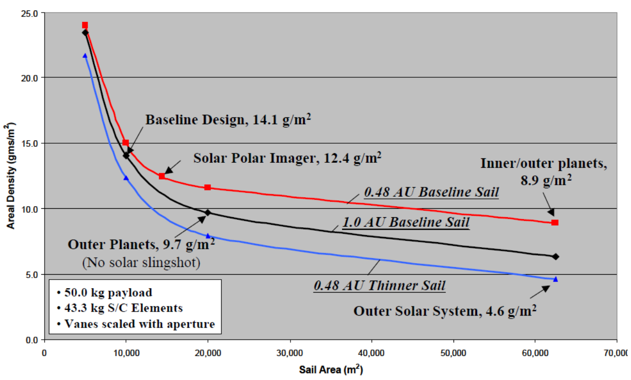


Figure 9.9: Solar sail scalability according to [64].

10

C & DH

10.1. Data storage

The MAG instrument has a data rate of 1.25 kbps so it produces a volume of 108 Mbits of data per day[51].

The SWA suite has a data rate of 14 kbps so it produces a volume of 1.21 Gbit of data per day[51].

Standard numbers for health and status telemetry is 2.4 kbps so it produces a volume of 207 Mbits of data per day[54].

Command has a data rate of 1 kbps so a volume of 86.4 Mbit per day[54].

This is a total of 436 Mbit/day or 65 Mo/day. Transmission shall be possible at all times at the acquisition rate so an internal data storage of 16 Gbit or 2 Go (Solid State Mass Memory preferably) is selected as a lower bound.

10.2. Data bus

ExoDaedalus' data bus is centralised data bus architecture as it is usually done for space applications. It revolves around an ESA *ESC-32S* radiation-tolerant *SPARC V7* processsor which acts as the central processing unit of the onboard computer(OBC). The MAG and the SWA suite have their own secondary processor, respectively a *LEON 3FT RTAX 2000* and a *LEON 2FT Atmel AT697F*. These were the same that equipped *Solar orbiter* for these instruments. Onboard communications is based on a *MIL-STD-1553B* bus and on a SpaceWire network hosted by the SSMM which manages all science data[50]. The non-science data is managed by a remote interface unit which links the OBC and specialist electronics(propulsion, pressures gauges, thermal etc...).

10.3. Data budget

During scientific data collection, under worst-case scenario, the amount of data collected represents 65 Mo/day. With a 2 Go SSMM, the S/C can hold for a month without transmitting data. Scientific data transmission is therefore scheduled once a week. At full download speed (1.6 Mbit/s), data shall be sent in about 38 min. A window of 6 hours/week is therefore scheduled. As for geostorm warning, the detection of a solar flare shall trigger an alarm and send the event directly to Earth.

10.4. Protocols and encoding

The ESA SpaceWire protocol is used which is based on the IEEE-1355 one. It supports highly fault-tolerant systems which is a reasonable choice considering that this mission is a sun observation one[55].

11

ADCS

11.1. Stabilisation method

The method retained for both phases of the mission is the spin stabilisation of the S/C. For the first phase of the mission when the sail is still not deployed, the S/C shall spin at a rate of 1 revolution every 37.5s. It is always sun-oriented except for maneuvers[17].

The attitude of the S/C is given by a set of three types of sensors for redundancy.

- Two digital sun sensors (DSS) : *SSOC-D60* with a FoV of $\pm$ 60° and an accuracy of 0.3° which are mainly used for solar array pointing and safe mode recovery.
- Two star trackers (ST) : *ST400* from *Berlin Space Technologies GmbH*. They provide high accuracy with an attitude knowledge error of 5" w.r.t. the stars and are reasonably light with 235g each.
- Two spinning sun sensors (SSS) : *Redwire fine spinning sun sensor* which has heritage from *BepiColombo, IKAROS* and other missions.

For the second part of the mission, as the sail deploys, different attitude control systems exist and the final one will depend on the development of the solar sail and its implementation into the S/C structure. For simplicity, a tipping vanes sailcraft shall be prioritised as it is less complex and consumes less propellant. The only significant torque produced on the S/C at L_1 vicinity is due to the solar radiation pressure. The effect is important when the sail is deployed but it is also the very principle of the solar sail.

Using a 3-axis stabilised sail with four tipping vanes located at each end of the booms allows precise control over the attitude of the S/C as described in [22]. It is also a propellant-free method of attitude stabilisation and doesn't affect the requirement of constant solar sail angle for station-keeping. **Fig.** 11.1 shows this concept. These vanes use a power estimated around 20W[19]. In summary, the S/C will be spin-stabilised for its first phase and 3-axis stabilised with tipping vanes for its second phase.

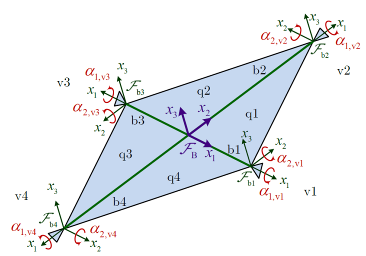


Figure 11.1: Tipping vanes concepts[22].

11.2. Pointing control and stability

The cone and clock angles of the solar sail are set to $(0.127, 70.626)$[63] which satisfies telecommunications, scientific and thermal requirements in our case. The attitude changes are done using the 1N thrusters.

11.3. Pointing Delta-v budget

Counting station-keeping in, the total Δv needed for pointing is 290 m/s. Without, it is 100 m/s.

12
Propulsion

This mission used two different propulsion systems : monopropellant hydrazine thrusters and a solar sail. The hydrazine thrusters are the primary ones and can handle the whole length of the mission in case of solar sail failure. The need to be able to reignite due to large number of maneuvers pushes in the direction of using monopropellant hydrazine thrusters.

12.1. Choice of thrusters

Monopropellant hydrazine thrusters : Two sets of blowdown monopropellant with helium pressurant systems are selected : a set of eight new generation *MR-103J*[1] 1N monopropellant hydrazine thrusters from *Aerojet Rocketdyne* and a set of four *MR-106* 22N from the same manufacturer. The first set is mainly used for small translational and attitude maneuvers and the second one for large translational maneuvers. Minimum impulse bits are respectively 0.0133 N.s and 0.015 N.s.

Solar sail : A single spin-deployed square sail with four diagonal carbon fiber booms is selected. The sail doesn't exist off-the-shelf and has to be designed and produced. It is primarily made of a $3-4$ [$\frac{g}{m^2}$] Mylar composite[19]. The booms are made with a 62 [$\frac{g}{m}$] carbon fiber material[5]. They are each of a length of 172.5m. Of course these elements can change in the course of the mission definition and development phases. The mission requirements require the sail to have a total area of 14'889 [m^2] corresponding to a $122m \times 122m$ square sail. The reflectivity coefficient of the sail is $\eta = 0.85$. The maximum thrust produced at 1 A.U. is therefore 0.12N. Computation were done using the sail's characteristic acceleration formula at 1 A.U.[54]. Its design is based on the TRL 6 *L'Garde* one.

(a)

(b)

Figure 12.1: (a) *MR-103J* (b) *MR-106L*[1].

12.2. Propulsion system block diagram

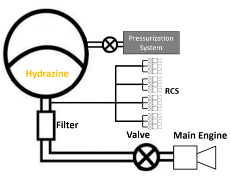


Figure 12.2: Simplified hydrazine propulsion system block diagram for a single tank[15]

12.3. Propellant budget

The S/C weight is slightly lower than the original *Genesis* mission. However, the propellant budget is computed using the old S/C weight for margin(670kg). The same original amount of hydrazine is used for the expected maneuvers : 143kg. There are two high performance *PEPT-590*[48] fuel tanks pressurised with helium gas[36] than can each host 75kg of propellant. This provides a Δv of 600 [$\frac{m}{s}$] with a margin of 30% in the case of solar sail successful deployment and usage and 22% in the case of solar sail failure thanks to the ejection of the SRC. The I_{sp} for both engines were taken from [1] in order to compute the wet mass from the Δv budget computed above(using Tsiolkovsky's rocket equation). A contingency of 7kg is applied for a total propellant budget of 150kg.

13 Thermal

13.1. Thermal balance analysis

The Sun heat flux at a distances 1 A.U., 0.99 A.U. and 0.98 A.U. is respectively 1326 [W/m^2], 1353 [W/m^2] and 1381 [W/m^2]. This is the only heat flux on the S/C at that distance. The equilibrium temperature of the S/C is given by :

$$T = \left(\frac{\alpha S A_n}{\epsilon \sigma A_t}\right)^{\frac{1}{4}}, \tag{13.1}$$

where α is the absorptance, ϵ is the emissivity, S is the solar flux at a certain distance, A_n is the projected surface, A_t is the total surface and σ is the Stefan-Boltzmann constant. The S/C will have an angle w.r.t. the perpendicular plane to the ecliptic comprised between [30°,60°]. During the sail phase, the angle will be close to 20°. The following values were computed for the different areas at the different flight phases :

- A_t = 8.51 m^2 for collection phase.
- A_t = 14900 m^2 for sail phase.
- $A_{30°}$ = 3.68 m^2.
- $A_{60°}$ = 2.13 m^2.
- $A_{20°}$ = 14095 m^2 for the whole S/C with the sail.
- $A_{20°}$ = 4.00 m^2 for the remaining bus at the center.

The $\frac{\alpha}{\epsilon}$ ratio is therefore important in order to have a good thermal balance of the S/C. Aluminium FEP with absorptance $\alpha = 0.16$ and emissivity $\epsilon = 0.47$ is used on the deck. The solar sail is made of Mylar which has coefficients $\epsilon = 0.044$. This gives equilibrium temperatures of :

- T_{30} = -0.91°C
- T_{60} = -21.45°C
- T_{20} = 3.73°C

This is a design assuming that the whole surface of the S/C is coated with aluminium FEP. In order to tweak these values, which are already acceptable, different coatings can be applied on the sensitive surfaces.

13.2. Thermal design

The S/C shall operates its components between their operating temperatures cited in **Tab.** 13.1. This requires a smart management and placement of all subsystems on the main deck. The battery as an example shall be placed behind the sun exposed surface. At this distance from the Sun and due to constant exposition to it, the problem is more overheating than low temperatures. Therefore, appropriate materials such as aluminium shall be used as shielding. Thermometers will be placed at relevant positions on the S/C. Several different cooling systems shall be implemented such as louvers, heat pipes, thermal coating, paint etc... The scientific instruments will profit from the experience gained from *Solar orbiter*. The overall geometry is well known thanks to the experience from *Genesis*. During the final phase, the sail, made of Mylar serves as a very good heat insulator. A special coating can be applied on the non-exposed face to radiate the exceeding energy.

13.3. Thermal requirements

Item	Temperature range [°C]	Remark
Propulsion		
MR103J MR-106L	+21	Integrated heaters
PEPT-590	+4 - +60	-
Science		
MAG	-100 - +45	-
SWA	Max 190	Thermal coating added
Electronics and ADCS		
EPS	-40 - +105	-
Batteries	+10 - +45	-
ST	-20 - +40	-
SSS	-45 - +85	-
DSS	-	No information
Transponder	-34 - +71	-
OBC	-55 - +125	-

Table 13.1: Thermal requirements for every component.

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A

End-of-life disposal of the spacecraft

S/C orbiting the libration points will likely increase thanks to the attractivity of these points for astrophysics missions. It is however interesting to ask ourselves the question on how to end the life of such S/C after their mission is done. Although not much harm is done by letting them enter a heliocentric orbit or drift on the heteroclinic channel between L_1 and L_2, it is a matter of principle now, considering the recent increase in "garbage" around our dear Earth, to ask ourselves these questions in every mission design although it has rarely been done before. Many studies have been done for S/C EoL disposal and one in particular is very suitable for the type of mission described here : disposal through solar radiation pressure. The idea is to orientate the S/C in a certain way in order to whether make it reenter Earth's atmosphere, crash the moon or injection into a graveyard orbit. The method retained here, thanks to the solar sail and its huge area, is the moon crashing as described in [52]. Indeed, a graveyard orbit would just add to the above problem and reentering Earth's atmosphere is out of the way as it will add kinetic energy to our atmosphere.