Human Service Repair Missions to GEO

Space Vehicle Design

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Abstract

The following report presents the vehicle design for the HeRMES mission, a servicing station for Geostationary satellites, expected to launch in 2030. The vehicle consists of a main station orbiting at 5000 km above the GEO stationary orbit, a crew module for transferring the astronauts, a cargo module for re-supply and a retriever satellite to fetch GEO satellites for repair. The main station is equipped with living quarters, a robotic arm, a cupola and a workshop. The crew module is based on the Orion spacecraft and the cargo ship is given by the Space X Dragon vehicle. The study of the different components and protection technologies of the vehicle shows that one of the most crucial challenges is the radiation shielding and that with the current technologies an astronaut friendly spacecraft is possible to design.

1 Introduction

Today, about 450 active satellites circle the Earth in the Geostationary Earth Orbit (GEO). These satellites are mainly used for communication and Earth observation and are essential to sustain nowadays fast-driven economy and lifestyle. However, most of these satellites are designed for a single use and life extension is rarely considered due to the lack of a platform that ensures satellite servicing. That's why repair and service missions are expected to become a potential business opportunity in the near future.
Among the different possible solutions to service GEO satellites, HeRMES group opted for a permanent main station at 5000 km above the GEO stationary orbit, allowing larger market opportunities. This station is titled Olympus and it will be the base of operations of the group. This study aims to assess the feasibility of this mission and eventually design a spacecraft that ensures successful and safe servicing and repairing operations.
This present project covers the Space Vehicle Design part of this mission and implies that all other mission aspects such as Overall Coordination, Logistics, Services and Human Aspects are covered by the cooperating groups of the HeRMES Team. All the results of those groups are used and taken as provided.
The first mission is to be conducted by the year of 2030, and the station will be capable to start operations within a year of the first launch. The station is capable of sending a retriever satellites to the GEOSATs that need to be serviced, or repaired. When arriving to the station, the robotic equipment will lock the GEOSAT in position while they are repaired by the 4-person-crew at the station.

1.1 Overall Station Requirements

The space station needs to fulfill certain specifications to fulfill the mission. These requirements where agreed upon after meeting with the cooperating groups of the HeRMES Team and are presented as follows:

- Propulsion and Station Keeping Systems
- Communication Systems
- Power Supply
- Thermal Control System
- Modular Design for Future Applications
- Minimum pressurized volume of 137 m^3
- Minimum habitable volume of 74 m^3 for a crew of 4 astronauts
- Falcon 9 Full Thrust Payload bay dimension fit
- Storage Unit
- Storage for Xenon, Hydrazine and Nitrogen Tetroxide
- Work Station with robotic Manipulator
- Emergency or Off-nominal Evacuation System

1.2 Specific Module Requirements

1.2.1 The Crew Capsule

The crew capsule is responsible for carrying the four-astronaut crew to and from the space station and providing everything needed for a safe journey. This includes life support systems, adapted radiation shielding, a docking system compatible with the space station and the necessary avionics and reentry features. The capsule must also be able to remain in space without degeneration for at least 8 months.

1.2.2 The Cargo Vehicle

The Cargo ship has the task to deliver all required up- and down masses of the mission to the station and back to Earth. It shall provide enough space for all the tools and spare parts as well as life support goods and additional mission loads in liquid, dry and gas state. It shall be able to dock to the station and be capable of staying in orbit for the time required for the mission.

1.2.3 The Retrievers

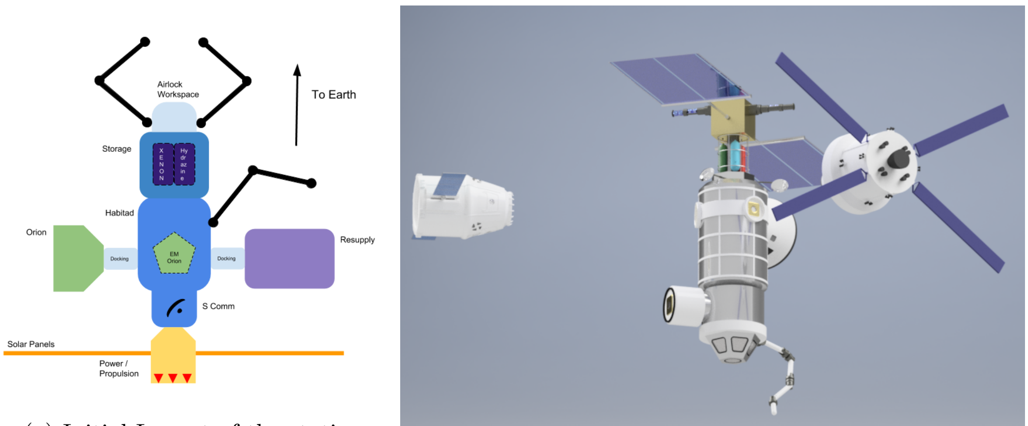
In the early stages of the mission, the retrieving satellites will be responsible for transferring client satellites to and from the space station, performing Hohmann transfers with a total ΔV requirement of 334 m/s per retrieving mission. The Satellite will also be designed to meet other service requirements in the future such as in-orbit refuelling.

2 Modules

This section presents the design and selection process of each module, the design parameters and criteria for the main modules that conform the Olympus GEO Service Station.

2.1 Olympus: The Main Station

Based on the design requirements, an initial layout of the station is proposed and is presented in Fig. 1a. After several iterations, a more specific design was reached. The final version of the station was then developed in a CAD modeling software and the result is presented in Fig. 1b. It consists of two main areas called Habitat & Command Center and Work Station. The docking is ensured and available by three ports for the crew modules, cargo ship and the emergency capsule. Olympus provides the needed specifications required for the operation of HeRMES group in GEO.



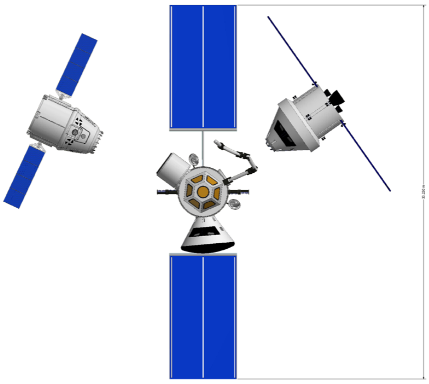
(a) Initial Layout of the station

(b) Olympus GEO Servicing & Repair Station

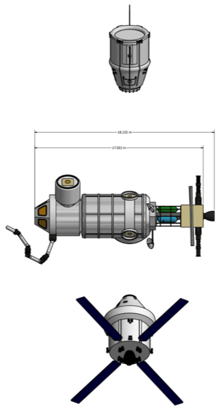
Figure 1: Schematic overview of Olympus station

2.1.1 Dimensions

The general geometry of the station in a fully deployed state can be seen in Fig. 2a and Fig. 2b. The station has a span of 33.3 m including the solar arrays, and a total length of 18 m excluding the robotic arm. The core module, the habitat and the command center was specifically designed to fit in the Falcon 9 payload bay, see Fig. 3a and Fig. 3b, as it is the chosen launcher for HeRMES group operations.

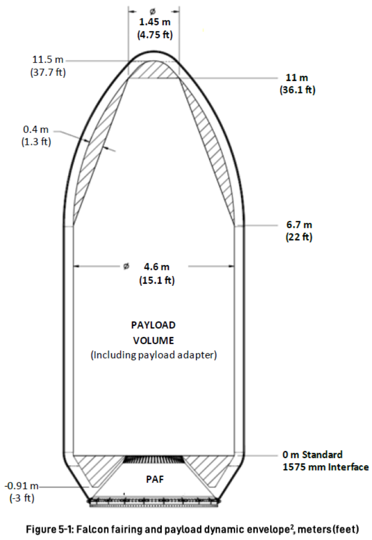


(a) Width

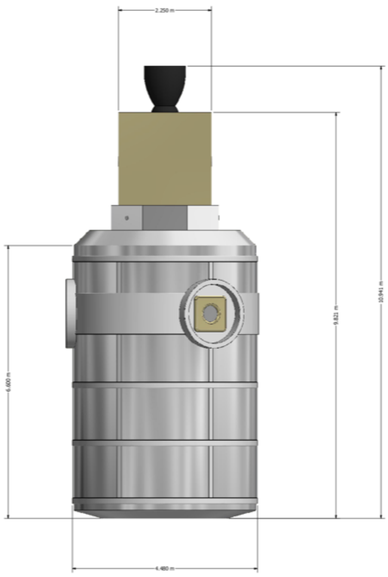


(b) Length

Figure 2: Dimensions of Olympus space station



(a) Falcon Payload Bay Specifications [1]



(b) Habitat Module General Dimensions

Figure 3: Schematic overview of Olympus interior

2.1.2 Habitat & Command Center Design

A crew of four astronauts will be living for a couple of months at Olympus station during a typical mission. Therefore, the station must contain a module that provides convenient living spaces and solutions to the vital needs of the crew.

The station's habitat contains living quarters that include personal sleeping quarters, a galley with a refrigerator and a microwave, training equipment and a table for securing meals. Life support systems include air revitalization, water recycling and waste management systems along with a toilet and hygiene compartments. This module has also three dedicated docking ports for cargo, the crew vehicle and the emergency capsule. Also, communication, and station propulsion systems are attached to this module.

The habitat is inspired by the *Zvezda* service module. Adaptation of this design is made with the help of the newest technologies available to make an efficient space environment for astronauts. This new design includes resting and spare time facilities. The new training equipment like the Swedish training wheel can provide the same effect while reducing the side effect of micro-gravity along with a very low mass.
As the time of the crew is one of the most valuable aspects of the mission, a series of computers is connected to the station systems in order to control and undertake some of the operations of the station from the ground. In addition, the crew can interact with the systems and control the station with various laptops.-The overall design of the command system is inspired by the ISS command center.

2.1.3 Work Station

The work station is used primarily as a storage unit, a robotic control center and an airlock. The concept is mainly based on the Cupola of the ISS [2] with minor variations to meet the specifics of HeRMES. The 3D model of the work station is illustrated in Fig.4, showing the deployable airlock and the robotic manipulator. The robotic manipulator is limited to 300 degree rotation angle around the station to avoid obstructing the airlock exit. The airlock contains EVA suits and can be locked from both sides of the station facilitating EVA preparation. The Cupola open view faces the working space directly, hence ensuring a clear view of the servicing area. All windows include a micrometeorite shield hatch that can be engaged while the works station is not used.

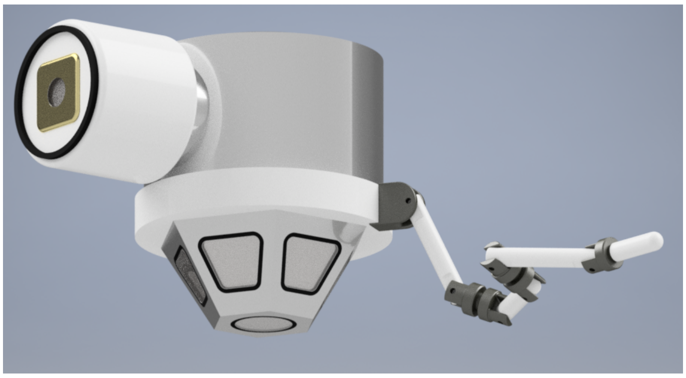


Figure 4: Work Station of the GEO Space Station

2.2 Orion: The Crew Capsule

When selecting the crew vehicle, two different capsules were considered: the SpaceX Crew Dragon and the NASA's Orion spacecraft. Both are designed to perform trans-lunar injections or even trans-martian injections in the case of the Orion. The Crew Dragon is considerably lighter than the Orions 8900 kg dry mass. However, the latter can sustain a crew for almost twice as long as the Crew Dragon without external assistance. That's why the Orion was chosen for HeRMES. Table 1 provides a quick overview on the Orion characteristics.

While closely resembling the Apollo command module, the Orion has state-of-the-art technologies such as a triple redundant flight computer, a storm shelter designed to protect against solar flares and a heat shield capable of withstanding temperatures up to 2800 °C. A schematic of the vehicle is given in Fig. 5.

Property	Value	Unit
Builder	Lockheed Martin	-
Crew Capacity	2-6	-
Dry Mass	8.9	tons
Length	3.3	m
Diameter	5.02	m
Pressurized Volume	19.56	m^3
Habitable volume	8.95	m^3
Mission Lifetime	21 days with crew and 210 days without	-

Table 1: Properties of the Orion crew capsule [3] [4]

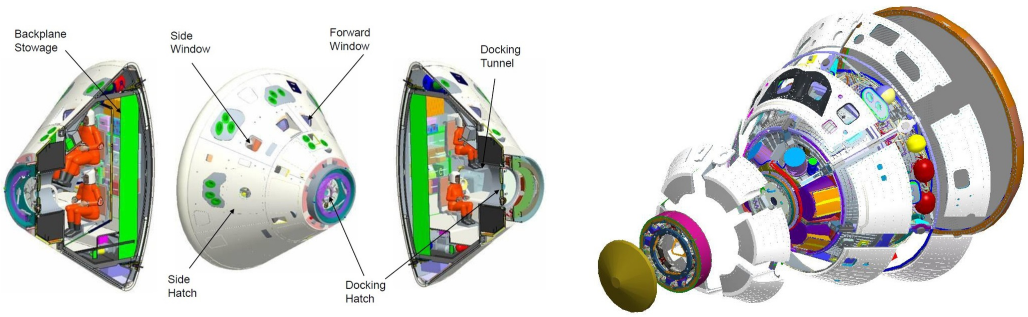


Figure 5: Orion Crew Capsule

The Orion service module is the European Service Module (ESM), presented in Fig.6. It is based on the existing ATV vehicle and can carry a total of 9200 kg of propellant, 240 kg of water, 90 kg of oxygen and 30 kg of nitrogen [5]. It can support the crew module for 21 days active space time with full tanks, and once launched to a LEO orbit it will transfer the crew to the space station.

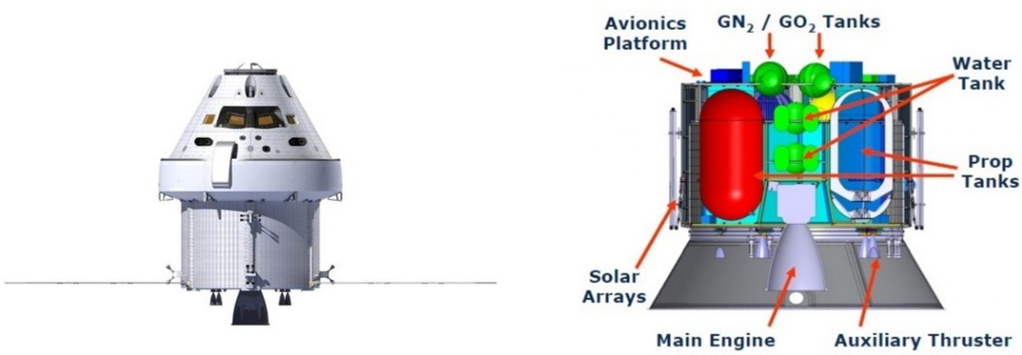


Figure 6: European Service Module

2.3 SpaceX Dragon: The Cargo Vehicle

The selection of the cargo vehicle is made by assessing current vehicles which perform cargo missions to ISS. Different cargo vehicles were considered and after preliminary research, the decision hanged between the European ATV and the Space X Dragon spacecraft. Figure 7 shows the comparison between the different spacecrafts, including the SpaceX Dragon and the ATV.

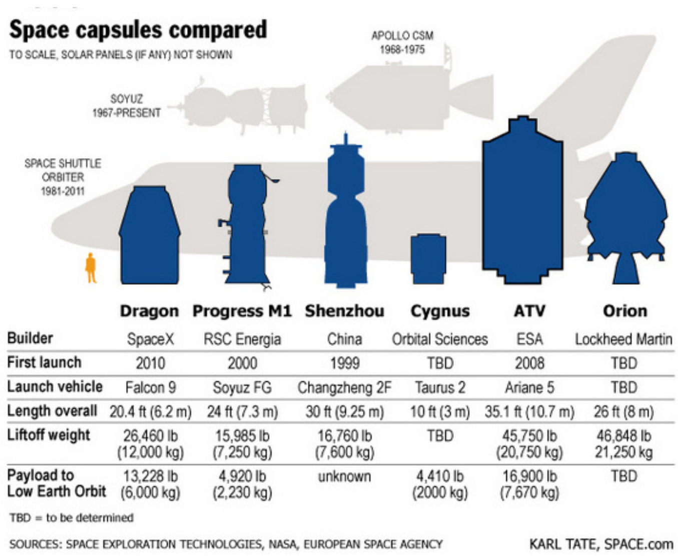


Figure 7: Comparison of different Cargo spacecrafts

The Dragon is a free-flying, reusable spacecraft developed by SpaceX under NASA's Commercial Orbital Transportation Services (COTS) program. Subsystems include propulsion, power system, thermal control, environmental control, avionics & navigation, entry descent and landing, and recovery [6]. The Dragon provides "an excellent platform for in-space technology demonstrations and scientific instrument testing. DragonLab represents an emergent capability for in-space experimentation" [6]. All in all, compared to the ATV, the Dragon had mainly three advantages:

- The lift-off Weight of the Dragon is 8 tons lighter then the one of the ATV
- It provides the possibility to bring down-mass to Earth
- It provides the capability of conducting scientific experiments on board

The Spaceship properties can be seen Table 2.

Property	Value	Unit
Builder	Space X	-
Launch Vehicle	Falcon 9	-
Lift Off Weight	12	tons
Dry Mass	4.2	tons
Launch Payload Mass	6	tons
Return Payload Mass	3	tons
Length	7.2	m
Diameter	3.7	m
Pressurized Volume	11	m^3
Un-pressurized volume	14-34	m^3
Generated Power	2000-4000	W
Mission Duration	1 week - 2 years	

Table 2: Properties of the Dragon Cargo

2.4 The Retriever

2.4.1 Design

The design of the retriever is inspired by the SMART-OLEV servicing spacecraft. It includes a retractable capturing tool, a docking system-developed by DLR-Institute of Robotics and Mechatronics [7], a state-of-the-art liquid propulsion system and an active three axis stabilization system. Future in-orbit refuelling is also enabled by using a robotic arm mounted on the front of the retriever. The retriever's main components are shown in Figure 8

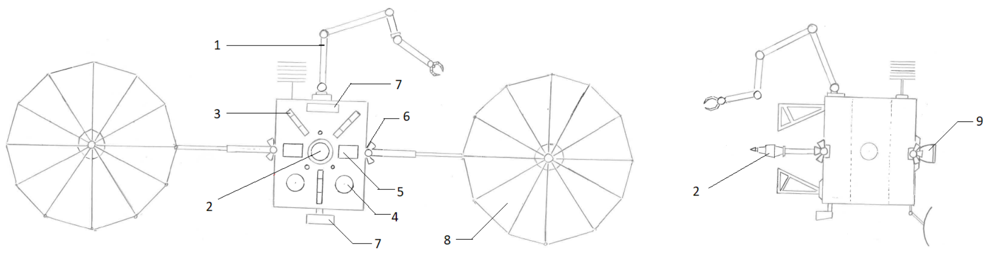


Figure 8: Retriever frontal and profile view, 1: Robotic arm, 2: Capture tool, 3. client support brackets 4. Refuel tank nozzle, 5. Cameras & Optical sensors, 6. Reaction control system, 7. Sun sensor & Star tracker, 8. Solar array, 9. Main engines

The retriever has a span of 6.5 m with extended solar arrays and a structural mass of 250 kg, including a tank capable of storing up to 500 kg of fuel intended for future in-orbit refuelling.

2.4.2 Propulsion System

The propulsion system for the retriever consists of two main engines of type ISE-100, which is currently in development by Aerojet Rocketdyne [8]. The thruster which is developed for commercial in-space applications provides 440 Newtons of thrust at a specific impulse of 300 s. The mass of each engine is only 0.74 kg [9], making them highly efficient with respect to the mass budget and suitable for in-space operations. The engines use Monomethylhydrazine (MMH) as propellant with mixed oxides of nitrogen (MON-25) as oxidizer. The low freezing point of MMH allows for a reduction in power needed for propellant thermal conditioning and is more suitable for deeper space environments. As of May 23, 2018, Aerojet Rocketdyne has successfully completed hot fire testing of the ISE-100, where the engine accumulated 75 individual tests, 774 pulses and more than 500 seconds of hot-fire time, placing the engine in the TRL3/TRL4 stage of development [8].
Apart from the main engines, the retriever has 16 active reaction control thrusters, each capable of delivering 20 N of thrust for three axis control during orbital manoeuvres and docking.
The amount of propellant needed to make the two transfers was calculated using the Tsiolkovsky rocket Eq.1 and Eq.2.

$$\Delta V = \frac{T}{\dot{m}} \ln\left(\frac{m_0}{m_f}\right) \tag{1}$$

$$\dot{m} = \frac{T}{g_0 I_{sp}} \tag{2}$$

Assuming the average wet mass of the client satellites to be 2500 kg, a total of 193 kg of fuel is spent in one retrieving mission. The fuel tanks in the retriever have a storage capacity of 250 kg, including fuel spent on reaction control.

2.4.3 Robotics and Docking System

The docking is performed by a retractable capturing tool, as seen in Fig. 9.

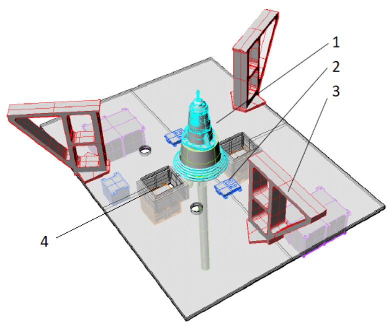


Figure 9: 1. Capturing tool, 2. Target illumination system, 3. Client support brackets, 4. Stereo cameras

The capturing tool is inserted into the client satellite apogee engine nozzle. While inserted, the capture tool locks itself onto the nozzle before the shaft is retracted until contact is made with the client support brackets. The target illumination system and the stereo cameras provide the ability to fine tune the placement of the capture tool during the insertion. The time required for insertion and capturing is estimated to be around 7 minutes [10].
The robotic arm for the retriever is inspired by the FRIEND robotic arm, as seen in Figure 10, intended for the NASA led Restore-L project [11]. The robotic arm has 7 degrees of freedom and weighs around 70 kg [12] including cables and electronics and can support a variety of modular tool kits. It has a reaching capability of 2 m, and can be stowed to fit in a 1.4 m (L) x 1m (W) x 0.65m (H) box, see Fig. 11.

Figure 10: FRIEND Robotic Arm

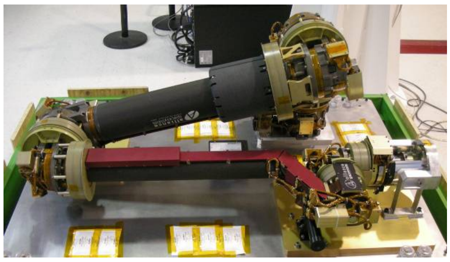


Figure 11: FRIEND Stowed Configuration

2.4.4 Power Generation

The main power supply for the retriever consists of two retractable flexible XTJ Prime solar arrays. The thickness of each cell is 80μm [13], allowing high flexibility while reducing the overall mass of the solar arrays. The solar cells have a total mass of about 3.10 kg and a area of 6.15 m^2, producing 2 kW of power to the retriever's subsystems. 1 kW of power is allocated to the avionics and thermal control system, leaving 1 kW to be divided among the docking system and the robotic arm. A 2 kW lithium ion battery will power the retriever in the event of a equinox solar eclipse or when the sun is blocked during docking. The batteries have a mass of 30 kg, and a maximum total mass of 50 kg with the solar array frames and cover glass included.

3 Olympus Subsystems

Once the main components of the vehicle presented, this sections gives further characteristics of the subsystems of the main station.

3.1 Attitude and Orbit Control System

The main attitude of the station has been chosen to maximize the power and meet the requirements for communication. The main axis points to the earth and the solar panels are kept in a North-South configuration. This allows the free rotation of the solar arrays and a maximum incidence of the solar rays. Fig. 12 illustrates the chosen attitude. However, this attitude might vary due to different external disturbances. The role of the attitude and orbit control system (AOCS) is to keep the vehicle in the desired orbit and orientation.

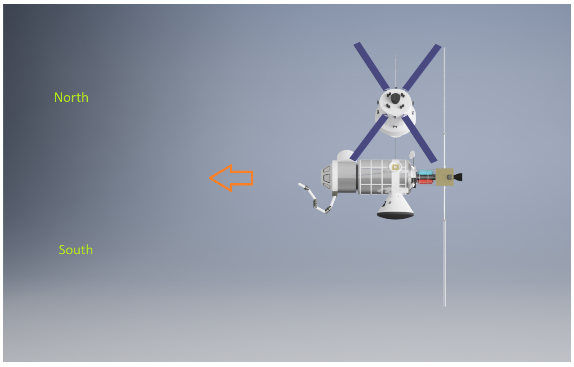


Figure 12: Attitude of station with respect to Earth

3.1.1 Attitude Control

The used approach for assessing the attitude control system requirements is shown in the Appendix [14] in more detail. The main source of torques disturbances are the on-board activities such as docking, un-docking and astronauts movement. The impact of gravity and solar radiation is of the order of 10^{-3} and was therefore neglected. The torque due to docking was estimated using the relation of $\frac{mVL}{\tau}$, where m is the mass of the docking ship, V its speed, τ the docking time and L is the length of the spacecraft. This resulted in a total impulsive torque of 500 N m. This value in mind, more importance was given to torque generation than momentum storage, so that the design favored the use of 3 control-moment gyros with double gimbals [15] instead of reaction wheels. This system is backed by the 8 Hall thrusters to desaturate (see 3.1.3).

3.1.2 Sensors

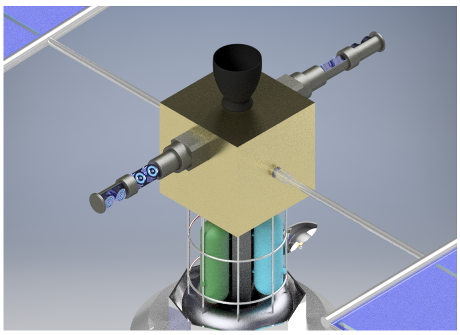
To keep track of the attitude, the station requires various sensors to point the direction of the Sun. This is ensured by multiple sun sensors [16]. Another type of sensors is also required to provide sufficient positioning information. That's why the sun sensors are complemented by a set of star trackers [16].

3.1.3 Propulsion

The selected propulsion system for orbit control is based on a modified version of the Thor Boom developed by OHB and going to be implemented for the Electra mission [17]. The boom is able to be retracted for launch and once deployed it can rotate and move to provide thrust in any necessary vector. This technology is adapted to the station needs and the number of thrusters is changed to 4 per boom in order to eventually comprise 8 hall effect thrusters for orbit control. The selected thusters to complement the boom are the BHT-1500 System from Busek Prupulsion and Systems [18]. This thruster was selected due to its wide range of operation from high thrust to high ISP. The different modes are seen in the Table below.

Power (W)	High Thrust Mode		High I_{sp} Mode	
	Thrust (mN)	Total I_{sp} (s)	Thrust (mN)	Total I_{sp} (s)
1,000	68	1,615	58	1,860
1,500	101	1,710	87	1,895
1,800	120	1,740	103	1,940
2,000	134	1,700	118	1,915
2,400	158	1,735	143	2,045
2,700	179	1,865	154	2,035

(a) BHT-1500 HET Operation Modes [18]



(b) HET Configuration [17]

Figure 13: Propulsion Systems

Also a typical hydrozine - nitrogen tetroxide engine is added to ensure redundancy. It will also provide fast orbit raising or orbit changes when needed. Both systems can be seen in Fig.13b.

3.2 Power System

The power system was designed to provide the station with the nominal required power. The estimated requirements are presented in Tab. 3.

Power (kW)	45
Maximum Bus voltage (V)	150

Table 3: Power System requirements

They are based on entries given by other groups. The primary source of energy is usually the Sun when it comes to space systems and Olympus should not be an exception. The size of the solar panels was designed based on the worst case scenario of high temperatures, off-nominal inclination and efficiency loss due to radiation and charging. The complete calculations are presented in the Appendix. Tab. 4 presents the overall design of the solar panels. The chosen solar cells are the triple junction XTJ Prime from Spectrolab [19]. They have an overall efficiency of 26.7% - 30.7%, the highest among the offered selection of this manufacturer.

Area (m)	146	Mass (kg)	72
Length (m)	15.86	Width (m)	10.37
Max Power per cell (W)	3.5	Min Power per cell (W)	2.71

Table 4: Primary Power System

The primary power system is backed by a secondary, battery based power system. The reason for that is that the orbit will traverse the earth shadow for some time, thus blocking the sun and rendering the solar panels useless. The longest eclipse lasts approximately 72 min. A set of batteries is therefore designed to provide the whole station with the required power. Since it is desired to have a high specific power, Li-ion batteries from Hy-Line manufacturer are used for the design [20]. Tab.5 gives the characteristics of the secondary power system.

Depth Of Discharge	20%	Mass (kg)	98
# in series	4	# in parallel	19
Max Power per cell (W)	3.5	Min Power per cell (W)	2.71

Table 5: Secondary Power System

Other types of batteries might be used by 2030 such LiS batteries which are a very promising candidate for future missions. They are expected to be safer, last longer and have a higher specific energy [21].

3.3 Thermal Control System

The thermal control system controls temperatures both inside and outside of the station. Without a thermal control system, the temperature of the sun facing side would soar to more than 121 °C, while the dark side-measured temperatures would be less than -157°C. A function of the thermal system is to transfer the heat generated in the station to space by using active and passive thermal control. Most of the heat produced remains at the station, including heat generated from the electronics, instruments and even heat generated due to metabolic of the crew, because of the dense shielding of the station. The total heat that thermal control system should transfer is approximately 12.6 kW which is approximately 30 percent of the total electrical power.

Due to long sun exposure, there is a demand for a positive heat balance and for materials with high emissions and low absorbance. The passive thermal control strategy consists of an appropriate mix of coefficients that will allow a positive heat rejection capacity. The condition is $\frac{\epsilon}{\alpha} > 3$, meaning that 3 times absorbed radiation is evacuated by radiated heat.

In addition, an Active Thermal Control System (ATCS) is required to transfer generated waste heat. An ATCS should perform heat collection, heat transportation and heat rejection. This can be achieved so that the heated ammonia circulates through huge radiators located on the exterior of the station, releasing heat as infrared radiation and cooling as it flows.

As for the shielding against the sun, a shield will be designed inspired by *James Webb Space Telescope* sun shield which consists of five layers. Each layer is coated with aluminum, and the sun-facing side of the two hottest layers also have a "doped-silicon" (or treated silicon) coating to reflect the sun's heat back into space. The highly-reflective aluminum surfaces also bounce the remaining energy out of the gaps at the shield layer's edges. Given an 0.08 absorbance and emittance of 0.8 for the high reflective aluminium [22]. Furthermore, adding a layer of white coating can decrease the absorbed heat. In this case, the Magnesium Oxide Aluminium Oxide paint is chosen which has an absorbance of 0.09 and emittance of 0.92. [23].

The electrical power of the station is estimated to be 42 kW from which about 12.6 kW will be waste heat generated. In addition every human approximately wastes 30 percent of the food consumed. If the average human consumes 1500 calories each day, that translates to 6.27 106 Joules per day [24], giving 7.2 W which can be negligible. The total waste heat generated is about 12.6 kW, a system of large radiators are designed with a capacity of 14 kW reduction[25], see Figure 14. The design is inspired by honey comb Aluminum panels of the ISS. The total area of the radiator is 40 m^2.

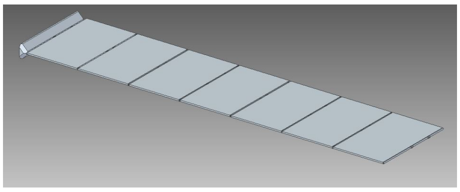


Figure 14: Radiators of the GEO Space Station

3.4 Communication

The station has to be able to communicate with both the retriever and the ground center. For the latter, the station has a very good communication access to Earth thanks to the zero-inclination of the orbit. The only limitations are thus circumstantial mainly related to the local atmospheric conditions. For the former, the situation is a bit complicated, since the retriever is constantly moving and is most of the time on a different orbit. Luckily, the retriever is always visible from the station when it is changing the orbit. This way, it is possible to establish a link with it. When the retriever is stationary with the to-be-serviced satellite, the direct communication with the retriever won't be possible and will be replaced by a ground link.

Station to Ground station link

Communications with the ground station use radio frequency. The link budget was performed and reported to the Appendix. Tab.6 gives the important characteristics of the communication system.

Antenna diameter (m)	Frequency band	Power (kW)	Mass (kg)	Link margin (dB)
0.3	Ku	1	12	7

Table 6: Characteristics of the communication system

Station to the retriever link

To communicate with the retriever it is possible to use the same communication system designed before. The maximum distance between the retriever and the station is estimated using basic trigonometry and calculated to be equal to 15 000 km, which means that the former design has sufficient power to reach the retriever, being designed for a longer distance (≈41 000 km). The use of laser communication was also investigated for this link, since laser communication offers better rate exchange for space-space communications. However, since the target is moving, it makes the laser pointing very difficult and very mass demanding.

3.5 Spacecraft Shielding

The Shielding of the Spacecraft is designed as a combination of layers of different materials in order to provide shielding against radiation and micro meteoroids. Both purposes require different materials that are combined to a total thickness of 90 mm including a Mesh Bumper, Kevlar Layer, Polyethylene layer, different Aluminum layers and a water layer, see Figure 15.

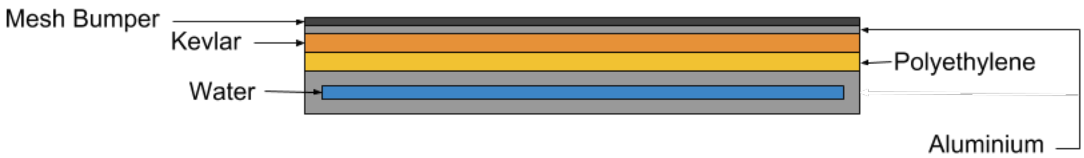


Figure 15: Sketch of the shielding layers

3.5.1 Radiation Shielding

The radiation environment hazards in GEO include energetic electrons and protons from Solar Particle Events (SPEs) and heavy ions from Galactic Cosmic Radiation (GCR). Solar cosmic rays of low solar wind particles and and highly energetic solar particle events is composed of protons, helium ions and heavy ions electrons. Exposure to space radiation may place astronauts at significant risk for acute radiation sickness (ARS), significant skin injury and numerous other biological effect [26]. A human without any shielding would get approximately 5720 rad/hour in GEO+5000 km according to SPENVIS, a tool developed by ESA [27]. Radiation exposure limits have not yet been defined for missions beyond low Earth orbit (LEO). [28]. However, for NASA astronauts, the radiation limit per months is set to 25 rad or 50 rad per year [29]. This leads to an estimated survival time in GEO of about 8 seconds without shielding. Therefore, an immense dose reduction needs to be conducted with the right shielding technology.

3.5.2 Shielding Requirements

At the operating altitude in deep space, the natural space environment poses difficult challenges for a space system design. The environment interaction with the space vehicle includes degradation of materials, thermal changes, contamination, excitation, spacecraft glow and charging and radiation damage [30]. The shielding needs to fulfill the requirement to protect the human being from the received dose equivalent that is harmful.

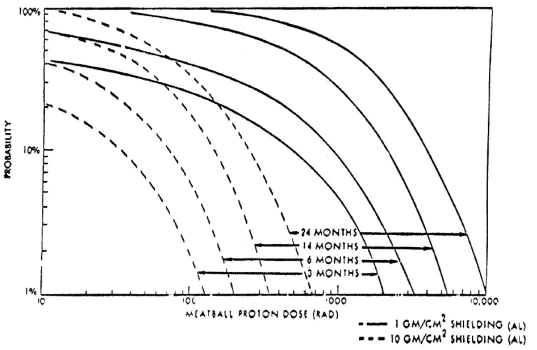
3.5.3 Shielding Material

Active Shielding With the intention of reducing weight of shielding materials, investigations of the feasibility of using active methods to shield spacecraft from hazardous space radiation have been undertaken [31]. The active concepts include use of electrostatic fields, confined magnetic fields and unconfined magnetic fields. Even though the active radiation shielding is the path forward for deep space missions and considered as a viable option, the innovative technologies are not expected to fully be discovered by the time frame of this mission. Therefore, passive shielding options are discovered.

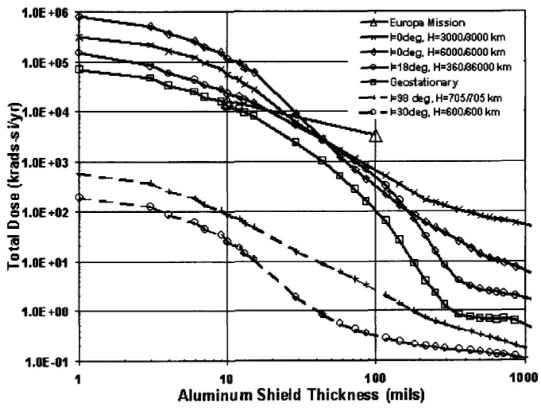
Aluminum In a case of a PSE, aluminum shielding provides the first approach for a solution. Several investigators have been suggested that the habitable areas of a deep space mission should have an aluminum shielding with an area density of $10 - 20$ g/cm^2 [32] abd NASA's cancer risk projection model shows that statistically significant improvements over aluminum shielding can be demonstrated for shields of about 10 g/cm^2 or more [33]. Assuming an aluminum shielding of 14 g/cm^2, the thickness is approximated by

$$t_{alu} = \frac{\rho_{area}}{\rho_{alu}} = \frac{14\ g/cm^2}{2.7\ g/cm^3} = 5.18\ cm \tag{3}$$

Figure 16a shows the solar flare dose probability with varying aluminum shielding thickness.



(a) Solar Flare Dose Probability [32].



(b) Total ionizing dose-depth curves for various orbits around Earth [34]

Figure 16: Solar Flare Dose Probability and Ionizing dose as a function of aluminum thickness

In Figure 16b, the higher curves on the graph are orbits that pass through more intense regions of radiation at higher altitudes, where GEO is represented by the lowest of the higher curves. However, exposure from GCR are not entirely shielded against with an aluminum shielding.The GCR dose equivalent with 15g/cm^2 of aluminum shielding during a Solar Minimum is about double the allowable annual dose for each leg of the trip to and from Mars [28]. Therefore, additional shielding materials are added to the aluminum structure to provide further protection.

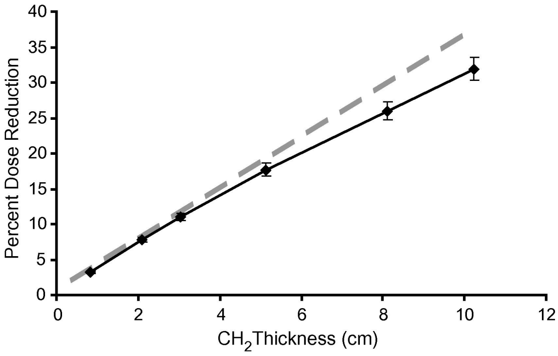
Polyethylene The Use of polyethylene as a shielding material promises optimization of cost, weight and safety while mitigating the radiation exposures from the trapped radiation and solar proton environments, as well as the GCR environment. High hydrogen content has a greater shielding effectiveness, but does not possess the quality of the required structural requirements.
NASA has chosen polyethylene CH_2 as a reference material for accelerator-based radiation testing [35]. The data presented in a study show that a 2.5g/cm^2 polyethylene target provides between 4 - 5 % reduction in dose per g/cm^2. A study as a part of the ALTCRISS project conducted the effect of polyethylene shielding in the Russian part of the ISS and found that a 5g/cm^2 polyethylene panel absorbed 10.6% of dose equivalent [36]. Figure 17a shows the percent dose reduction with increasing polyethylene thickness [35].

The thickness of the polyethylene layer is approximated to be

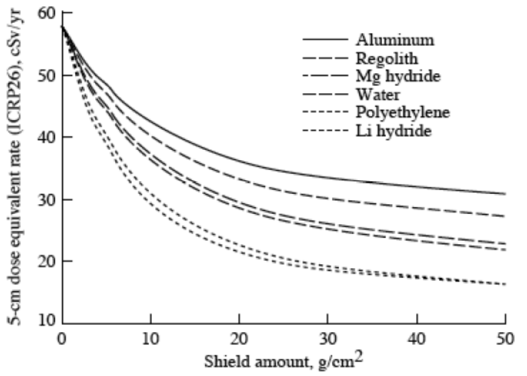
$$t_{poly} = \frac{\rho_{area}}{\rho_{poly}} = \frac{2.5\ g/cm^2}{0.97\ g/cm^3} = 2.5\ cm \tag{4}$$

It can be seen that With the thickness of 2.5 cm, a dose reduction of approximately 10 % can be achieved already. Even though the shielding effectiveness of polyethylene is better than aluminum, both materials provide some exposure reduction over thickness typically found in transportation vehicles [37], that need to be taken into account.

Figure 17b shows the comparison of the effective estimated GCR dose equivalent in space behind shields made from various materials. The better shielding characteristics with hydrogen-containing materials is visible. After 20 g/cm^2, the effectiveness diminishes [28].



(a) Percent Dose Reduction with increasing CH_2 thickness [35]



(b) Point estimates of 5 cm depth dose for GCR at Solar Minimum as a function of areal density of various materials [28]

Figure 17: Dose reduction performance for Polyethylene shielding

Water When any material used for shielding can serve a dual purpose, mission costs can be reduces significantly. Because of present water on the station as on-board resource, water is considered as an additional shielding layer option. [28] and is a goof shielding compared to Aluminum as well, see Figure 18 [38].

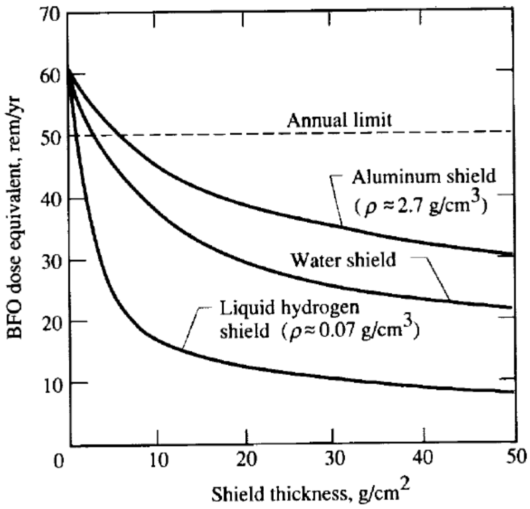


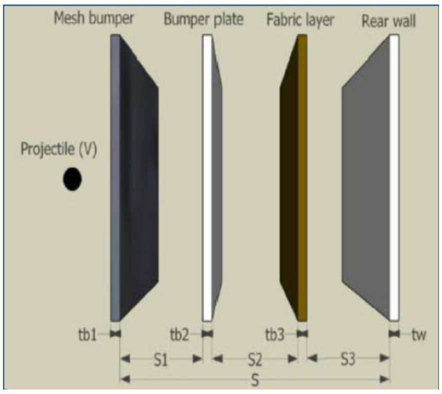
Figure 18: BFO dose equivalent as a function of shield thickness from GCR [38]

A water shield of already 0.8 cm would lower the dosage on BFO by over 17 % already to 234.5 mSv, bringing the dosage below the annual limit set by NASA [39].

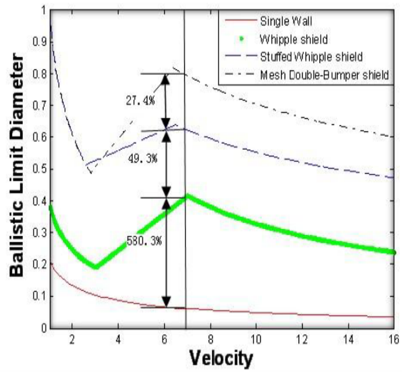
3.5.4 Micro Meteoroid Shielding

In order to protect the spacecraft from micro meteoroids and orbital debris impact, the mesh double bumper (MDB) shield is considered as a highly efficient method tested by the NASA Johnson Space Center [40]. Hyper-velocity impact testing of the shield demonstrated weight savings of 30 - 70 % compared to dual sheet aluminum whipple sheets.

The mesh double bumper consists of four distinct layers as shown in Figure 19a [41]. While the first mesh bumper (wire mesh) breaks up the projectile into smaller fragments, the bumper plate shocks them subsequently, causing the projectile fragments to release into liquid or pulverized segments. The intermediate Fabric Layer is used to stop or slow any remaining solid fragments before they contact the back plate. Through stretching and breaking the fabric fibres, energy of the debris is absorbed [40].



(a) Schematic Diagram of the Mesh Double Bumper Layer [41]



(b) Ballistic Limit Diameter of the Mesh Double Bumper compared to a Single Wall, Whipple Shield and Stuffed Whipple Shield [41]

In the condition that the area density of each shield is the same to other tequniques, the Ballistic Limit Diameter (BLC) of the Mesh-Double-Bumper is the best, depicted as the highest curve in Figure 19b.

4 Mass Budget

The Total Mass of the station is calculated to be 40 tons. A breakdown of the this mass is presented in Tab. 7.

MASS Estimations			
Part	Components	Description	Mass [kg]
Capsule	Orion	-	9380
	Service Module	ESA made	13000
	Abort Tower	-	7257
		Total	29637
Space Station	Solar Pannels	XTJ PRIME	72
	Batteries	Lithium Ion	98
	Passive Thermal Control	Reflective 0.08 absorvance emissivity 0.8	-
	Heat Exchanger	ISS based Radiators	5000
	Observing Station	Cupola Based	2000
	Communication	0.3 m Antenna System + electronics $\times 2$	24
	CPU		500
	Training Wheel		150
	Shielind and Structure	Aluminium + water + Fam + BLC	21612
	Robotics	CANADA ARM2	1400
	Escape Capsule	Orion	9380
		Total	40262
Cargo Module	Dragon Cargo		12000
Scauts		tank for refuel + arm + structure + 250 kg of fuel	500

Table 7: Mass Budget of the entire Station

5 Conclusion

One can see from the performed study that the challenges to create an independent and self-supporting environment in GEO is not an easy task, especially due to strong radiation. However, the advances in the previous years have made the goal more achievable and the vehicle team believes that a human servicing-station in GEO orbit is possible by 2030. With 3 independent, yet compatible modules, modularity was the core concept of the suggested design to ensure efficiency and durability. Indeed, with the current pace of technological development, competitiveness on the market will be ensured: One day, robotics will replace in-orbit human servicing and ironically enough, Olympus will need to have its life extended with new services and capabilities. Overall, this study suggests that the technology to create a sustainable servicing station exists and a launch can be expected by 2030 if the political and economic drive follows.

Appendices

A Power System

Tab.8 gives a breakdown of the power required on the station. These values ware estimated by comparing past missions power budgets and taking a mean of the values except the life support system power budget that was given by the Life support group.

System	Required power (kW)
Life support system	13
Thermal control	2
AOCS (CGR+sensors)	0.1
Propulsion (HET+Main thruster)	21
Command and data handling	6
Total	45

Table 8: Power consumption estimation

First, the number of battery cells needed for the secondary power system were determined. The batteries provide power to the whole station during eclipses. The worst case eclipse lasts 72 min. Assuming a DOD of 20%, the energy required from the batteries is thus estimated using Eq.5. This gives us the total Wh-index of the system. From Tab.3, the number of cells in series is estimate using Eq.6 and the number of parallel branches is found using Eq.7, where V_{nom} is the nominal voltage of the battery and C_{nom} is the nominal capacity. The DOD was chosen in function of the desired lifetime. For DOD of 20%, it is possible to ensure 40000 charge cycles [42].

$$E_{req} = \frac{P_{req} T_{eclipse}}{20\%} \tag{5}$$

$$\#InSerie = \frac{V_{max}}{V_{nom}} \tag{6}$$

$$\#Branches = \frac{E_{req}}{V_{max} C} \tag{7}$$

Voltage (V)	48
Capacity (Ah)	15.9
Watt-hour rating (Wh)	763
Mass (kg)	4.9

Table 9: Nominal Values of the HI1305PC01 battery pack

Second, a similar approach was used to size the solar panels. In addition to the electric energy delivered to the station, the power needed to recharge the batteries was also taken into account. Also several sources of efficiency losses were considered. The temperature of the solar panels is between 0 °C0 and 40 °C which will account for a decrease in the maximum produced power by 8 % for XTJ-Prime cells [19]. The efficiency of the cells decreases also with time due to radiation and charging and was estimated to a rate of 10% loss. The change in the inclination

Maximum Voltage	% Loss at Geo	Loss per °C	Real voltage
2.4	87%	-0.07	2
1.47	97%	0.009	1.43
3.52	-	-	2.88

Table 10: Accounting for efficiency losses of the solar cells

has also an impact on the maximum produced power, For that reason the maximum allowed angle is fixed to α_{max} =20°. So that in the worst case scenario, the power produced is equal to

$P_{produced} = P_{ideal}\ cos(\alpha_{max})$. The same reasoning was applied to determine the number of cells in series and the number of branches

B Communication

A link budget was made in order to find the diameter of the antenna. The transmitted power and the required SNR were fixed based on previous numbers from communication satellites[14]. Then the link margin was calculated in function of the antenna diameter and the transmission frequency. Several iterations were made to determine a frequency band that allows a reasonable antenna gain with an acceptable diameter.

Name	Value
Data rate	300 Mb/s
Bit Error rate	10^{-6}
Transmit Power P	1 kW
Frequency	20 GHz
SNR required	9 dB
End to End delay	0.12 s
Distance	41 000 km

Table 11: Parameters used for the link budget

Name	Formula	Value in dB
Antenna gain	$G = \eta \frac{4\pi A}{\lambda^2}$	18.82
System Noise temperature	$T = 150\,\mathrm{K}$	21.8
Figure of merit	$G - T$ (dB)	-2.9 dB
Boltzmann constant	$k_b = -228.6\,\mathrm{dBW/KHz}$	-228.6
Free space loss	$L_{fs} = \frac{4\pi D^2}{\lambda^2}$	174
Available SNR	$SNR_{avail} = 10log_{10}(P) + G/T - L_{fs} - k_b$	16
Link Margin	$M = SNR_{avail} - SNR_{req}$	7

Table 12: Formulas and method to make the link budget

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