Human Spaceflight - Red Team: Logistics

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***Abstract*—Since the beginning of space programs, ”On-Orbit Servicing” has fascinated the space enthusiasts and engineers alike for over four decades. This paper is focused on the feasibility study from a logistics point of view of a manned mission to geostationary orbit (GEO) to perform different type of services, e.g. repair, assembly, and upgrades of satellites. The selected launchers are Ariane 6 A64, the new European launch vehicle, and Vulcan/ACES, manufactured by United Launch Alliance. Three launches are needed to bring all the payload in orbit: two Ariane 6 will bring up the space station modules, while Vulcan/ACES will be used for the crew. Five orbits are selected to perform the different steps of the mission and to optimise performance, rendezvous manoeuvres and service time. The total time required for a mission, considering four satellites to be repaired and including the launch, the chasing time, the ”on-orbit” service time to accomplish the task and re-entry, is about 24 days.**

Table I: Declaration of Variables

Variable	Meaning	Unit
ΔV	velocity	m/s
t	time	s
a	semi-major axis	km
r	orbit radius	km
$\Delta\theta$	drift angle	rad
μ	Earth standard gravitational parameter	$\mathrm{m^3s^{-2}}$
LEO	Low Earth Orbit	-
GEO	Geostationary Earth Orbit	-
CO	Chasing Orbit	-
MO	Meeting Orbit	-
MSM	Mission Service Module	-
MM	Mission Module	-
SM	Storage Module	-
LEV	Launch Reentry Vehicle	-
LSM	LEV Service Module	-
AL	Airlock	-
ISS	International Space Station	-
GTO	Geostationary Transfer Orbit	-

I. Introduction

Today, there are approximately 450 active satellites in GEO, mainly for communication and Earth observation. For now, when a satellite reaches the end of its operating time, it is considered as a ”dead satellite”. More precisely, at the end of the mission the satellite will be pushed, using the propellant that is left, to a graveyard orbit, in order to reduce the risk of a collision in GEO. The GEO satellites are mainly considered ”dead” because they don’t have any propellant left. The purpose of this project is to propose and design a new solution to these problems. And this new solution consists in a set of service/repair missions to GEO satellites. Starting from the concept of the European GEOfarm, the concept has been modified, in order to fulfil a certain kind of missions in a given time. These missions will provide extra fuel, systems updates, or repairing support. In order to work efficiently on this project, the team has been separated into different subteams. In this paper the logistics part will be discussed. Logistics is the process of planning to supply and maintain the space mission over time. For this project, the logistics groups is also in charge of the general planning of the mission. The ”general planning” consists of:

- the design of every orbit that will be used during the mission,
- the calculation of every ΔV costs for each maneuvers planned,
- the calculation of every duration of each steps of the mission,
- the creation of the timeline of the mission.

The logistic group is also in charge of other tasks like the selection of the launchers, the estimation of their costs and the inventory of every systems or objects that need to be included in the payload of the mission in order to fulfil each objectives. Finally, the group had to think about an off-design situation and found a solution to insure the safety of the mission and the crew under these conditions.

II. Mission design

A. Mission objectives

As presented in the introduction, the mission consists on a set of service/repair missions to GEO satellites. In terms of strategy, the first point that has been discussed is the average duration of one mission. With the actual space technologies, the duration can be of two orders of magnitude:

- ”Long missions”, which means a large structure in orbit, and a crew that will stay for more than a couple of months

in space. This also means a lot of support systems that will provide the resources for a human spaceflight during months (like the ISS today),
- ”Short missions”, which means a smaller platform, with a crew that will stay up to one month in space.

Regarding the complexity of the mission, but also the number of technologies involved in the mission, the team choose to go for ”short missions”. Moreover this will strongly help to reduce the total dose of radiation received by the crew during the mission.

B. Orbit selection

Now let’s focus on the selection/inventory of the orbits that will be used during the mission. First, as it’s commonly done, we may use the LEO orbit at the end of the launch, in order to proceed to test and check-out before going to a higher orbit. The LEO altitude for this mission has been set to 300 km. The other orbit that we will obviously use is GEO. For further calculations, this altitude has been set to 35 786 km. At this step, the spacecraft department has already done one sketch of what could be the shape and architecture of the spacecraft used during the mission. This shape answers to a first question: is it needed to bring the whole spacecraft to GEO altitude? The answer is obviously no, the best option is to bring to GEO only the crew, with the supplies needed to operate the mission on the satellite, and the propulsion system that will move this payload mass which will be called the ”mission module”. It also means that between each satellite, the crew will go back to a ”storage module” in order to take the supplies needed for the next satellite. So this solution involves a new orbit that can be considered as a ”parking” orbit for the storage module. Actually, this orbit will also be the place where the crew waits to phase it’s orbit with the next satellite, for this reason this orbit has been called the ”chasing” orbit. In order to minimise the waiting time of the crew and the ΔV costs, it has been found that a better solution to phase with the next satellite is to use an orbit closer to GEO than the chasing orbit. This new orbit has been called the ”drift” orbit. In conclusion, the orbits used during a nominal mission are now known:

- LEO,
- Chasing orbit,
- Drift orbit,
- GEO.

But these orbits have to be defined in terms of orbital parameters. This can be done after a design process that takes into account the influence of every aspect and constraint of the mission.

III. Orbit design

This part is dedicated to the design process of the orbits used during the mission. This design process is taken from the book ”Space Mission Engineering: The New SMAD” [1] and can be divided into three main steps:

- Establish the orbit types: this is made by dividing the mission into segments and classify each segments by its overall function. These overall functions can be grouped in categories, which leads later into categories of orbits. ”What is the purpose of this orbit? What is the main objective that needs to be achieved in this orbit?”
- Establish the orbit-related mission requirements: this can include orbital limits, individual requirements, or range of values constraining any of the orbital parameters. ”What are the different aspects that can have an influence on the orbital parameters? How are they limiting the orbital parameters?”
- Evaluate the orbit performances: for every orbit the performances reached will be evaluated, but also how much it cost to reach this performances. ”What are the final orbital parameters? Is it sufficient to complete the mission objectives? How much does it cost to fulfil the mission requirements with these selected orbits?”

A. Establish the orbit types

In the case of this project, the orbits used can be grouped into two different categories:

- Earth-referenced orbits: an operational orbit which provides the necessary coverage of the surface of the Earth or near-Earth space.
- Parking and transfer orbits:
  - Parking orbit: A temporary orbit providing a convenient location for satellite check-out, storage between activity or at end-of-life, or used to match conditions between phases.
  - Transfer orbit: An orbit used for getting from place to place.

The different stable orbits that will be used during the mission were already discussed in the section ”Orbit selection” of the Mission Design, but the method used here is also considering the transfer orbits. Moreover, every transfer orbit will be operated by doing a Hohmann transfer from one orbit to another one. Hence, let’s make a resume of every ”segment” that will be used during the flight:

- LEO segment: once ejected from the launch vehicle, the spacecraft stays briefly in LEO orbit to provide test and check-out of the spacecraft and transfer vehicle subsystems. This orbit is considered as a ”parking” orbit.
- Transfer orbit from LEO to Chasing orbit: this mission segment is a transfer orbit that moves the spacecraft from the parking orbit to the chasing orbit. To preserve propellant, the spacecraft is initially put into a drift orbit near the chasing orbit such that any errors in the transfer process can be taken out by small adjustments associated with achieving the desired location on the orbit.
- Chasing orbit: the spacecraft (at least the storage module) enters in its operational orbit in the chasing orbit where it will spend most of its time.
- Transfer orbit from Chasing to drift orbit: same strategy as for the other transfer orbits, but from chasing to drift orbit.
- Drift orbit: the purpose of this orbit is to provide an orbit that is closer to GEO in order to minimise the ΔV cost to go from one satellite to another one.

- Transfer orbit from drift to GEO orbit: same strategy than before but from drift to GEO.
- GEO orbit: once arrived in GEO by rendezvous with the selected satellite, the crew will spend some time here to operate the mission, and then when the mission is done they will go back to the drift orbit to make a new rendezvous with the next satellite, or directly to the chasing orbit if it was the last satellite.

B. Establish orbit-related mission requirements

Now that every orbit is defined, the next step is to understand what are the mission requirements that can have an influence on the orbital parameters of each orbit. This can be done for both of the orbit types defined previously.

1) Earth-referenced orbits: Earth referenced orbit will be affected by these mission requirements:

- Environment and survivability,
- Launch capability,
- Ground communication,
- Legal or political constraints.

a) Environment and survivability: One of the first constraints is the radiation. The level of radiation received depends on the altitude, and the main objective here is to avoid the Van Allen belts. Moreover, the most dangerous one is the inner belt which starts at an altitude of 700 km and ends close to 10 000 km.

The lightning conditions can also affect the orbital parameters. In fact, the sun irradiance will be critical for the design of the power system, and the altitude will define the eclipse time. So the lighting conditions will be critical in terms of energy design.

Last, the thermal conditions can be a limitation for the orbital parameters. During the design, the selected orbit needs to insure that the balance of heat received and heat that needs to be evacuated can be under control.

These three considerations will affect several orbital parameters:

- Altitude,
- Inclination,
- Right Ascension of Ascending Node (RAAN)

So in order to avoid a too high level of radiation, the orbits where the crew will stay for a long time need to be higher than 10 000 km. Moreover the power source will have to be sufficient at this altitude where the power received from the sun is lower than usual (ISS).

b) Launch Capability: Here again three constraints can affect the orbital parameters:

- Launch cost,
- On-orbit weight,
- Launch site limitations

And these aspects will affect:

- The altitude of the parking orbit (depends on the total amount of ΔV that the launcher can provide, which depends on the mass of the payload).
- The inclination of the orbit (affected by the latitude of the launching site).

This will be discussed more in details during the selection of the launchers.

c) Ground communication: First the location of the ground station can affect the orbit (the best case will be if the spacecraft is able to communicate with the ground at any time of the mission). But if the communication with the ground is not available, the position of relay satellites will also have an influence on the choice of the orbits.

The last solution is to study the data timeliness, which means to look for the time line opportunities to send data to the ground.

As before, these considerations will affect the orbital parameters:

- The altitude will modify the strength needed for the communication signal, and will also define what communication relay satellite can be used.
- The inclination will strongly be affected by the ground station available, but also by the time line of the communication opportunities.

For this kind of manned missions it seems that the communication between the crew and the ground needs to be operational at any given time of the mission. This means that the final solution for the orbit needs to take into account a constant link with ground stations.

d) Legal or political constraints: In this domain there are two main constraints:

- Launch safety restrictions,
- International allocations.

As these constraints are more complex than the ones before, a more complete explanation is needed here.

Launch safety restrictions:

The launch of any vehicle in space is under the authority of the Administration for Space Transportation (AST). This administration reviews commercial operator launch safety efforts, including:

- Procedures for safety controls for launch sites and flight corridors,
- Range safety expertise,
- Procedures for ground and flight safety,
- Range tracking and instrumentation,
- Vehicle safety systems,
- Proposed vehicle design.

International allocations:

The International Telecommunication Union (ITU) is responsible to allocate the positions of the GEO satellites, and the ITU is mandated by its Constitution to ”allocate spectrum & register frequency assignments, orbital positions & other parameters of satellites”.

So both launch safety restrictions and international allocations will affect the orbital parameters used during the mission. They will more precisely affect:

- Altitude,
- Inclination,
- Longitude in GEO.

But for now the range of these restrictions is a very complex subject. A lot of laws and treaties are acting today, and a precise restriction on the orbital parameters is very hard to find.

2) Transfer and parking orbits: Now the same work needs to be done for the transfer and parking orbits.

This orbit type can be affected by several restrictions:

- The transfer ΔV,
- transfer time,
- departure and arrival conditions,
- ground station communications,
- radiation environment.

a) Transfer ΔV: Here the main purpose is to know what are the limits of the propulsion systems enrolled in the mission. So there are two main restrictions:

- The performance of the launcher's propulsion system.
- The performance of the satellite's propulsion system.

Regarding the ΔV provided by these two systems, this will affect both the tilt and the altitude.

The selection of the launchers will be partly based on the criteria of the ΔV. When it comes to the spacecraft propulsion system, this will be designed by the vehicle design department in order to fulfil the mission requirements in terms of ΔV needed to operate a nominal mission (calculated by the logistic department later in this report).

b) Transfer time: The transfer time is constrained by two main things:

- The humans on board,
- The mission schedule.

These two criteria will affect the strategy of motion for the transfer orbit. Then the strategy of motion will define the restriction on the orbital parameters of the transfer orbit.

There are today four common ways to go from one orbit to another:

Type	Hohmann
Typical acceleration	1 to 5 g
ΔV	Hohmann delta V
Typical time	1/2 orbit period

Type	High Energy
Typical acceleration	1 to 10 g
ΔV	less than Hohmann delta V
Typical time	more than Hohmann

Type	Low Thrust Chemical
Typical acceleration	0.02 to 0.5 g
ΔV	same as Hohmann
Typical time	6 to 8 Hohmann transfer time

Type	Electrical propulsion
Typical acceleration	0.0001 to 0.001 g
ΔV	difference of orbital speeds
Typical time	120 to 240 Hohmann transfer time

The choice for the strategy of transfer will have an impact on two of the orbital parameters:

- The semi-major axis,
- The eccentricity.

The crew will not be able to be under too many g's, for this reason every transfer with a crew will be operated by a Hohmann transfer which is the best trade-off in terms of transfer time and acceleration. For other modules the transfer time is not a problem, and the electric propulsion seems to be the best choice in terms of payload mass since the propellant mass is a lot lower than for chemical propulsion.

c) Departure and arrival conditions: Obviously the altitude and the inclination of the departure and arrival orbits are the main mission requirements that define the orbital parameters. These two conditions can be matched by doing:

- A Hohmann transfer for the change in altitude,
- An instant impulse for the change in inclination.

But in fact the process of plane change can be optimised in order to reduce the overall ΔV needed. There are several solutions:

- Do the ΔV for plane inclination change at the lowest velocity (more efficient way to do it).
- Combine the ΔV for plane change with orbit raising (vector sum is less than sum of components).
- Do three burn transfers which corresponds to going to high altitude where the ΔV for plane change is low and then go back to a lower orbit.

Then there are several ways to combine the ΔV for altitude and for plane change:

- Combined manoeuvres with all plane change at apogee (at the same time that the second impulse of a Hohmann transfer).
- Combined manoeuvre with a part of the plane change at periapsis and the rest at apogee. This solution is commonly used as the most efficient in terms of ΔV cost.

So for the rest of the report, every ΔV costs for a transfer orbit will be calculated with a combined manoeuvre (with change in plane operated partially in each impulse of the Hohmann transfer).

d) Rendezvous: When the goal of a transfer is to make a rendezvous with another spacecraft (for example when the different modules meet in MO), both orbits must be in phase, i.e. both spacecrafts should not have any angle difference when the transfer is over. Any initial angle delay has to be rectified and it can be done by waiting for the lowest spacecraft to catch up the delay. It results in a waiting time that can be huge if both starting and target orbits are too close from each other.

e) Ground station communication: This problem can be solved the same way as communication for Earth-referenced orbits. So the conclusion is the same, but will also be discussed later on in this report.

f) Radiation environment: Here the threat stays the same as for the Earth-referenced orbit, the Van Allen belts. But as this part of the study is now considering parking/transfer orbits, a new aspect has to be taken into account: the speed of the spacecraft. In fact, if the spacecraft crosses the inner belt, the speed will reflect the time spent in the inner belt, and so is linked with the total dose received. Then even if the orbital parameters should be limited by this consideration, the speed could be a solution to limit this impact. This trade-off will be discussed later on in this report.

IV. Orbits performances

After the design of the orbit, a discussion was started with all the other groups of the team in order to:

- confirm the relevance of each consideration approached during the design phase,
- work on the different trade-offs that will lead to the final solution.

At the end of this group work, several problems were solved, and others where just mentioned and some values where assumed in order to get to the conclusion and find a final solution. In this part of the report, the solution will be developed and the final orbits will be presented. Their performances will also be evaluated.

A. Radiation

For this part the main discussion was with the human aspect group. After the study made by this group, the conclusion was that the main part of the dose received will be in GEO. For the logistics group this means that the time spent in GEO has to be minimised. Moreover, the overall duration of the mission has also to be minimised.

In order to reduce the overall time, as it was already mentioned previously in the "orbit selection" part, a "chasing orbit" and a "drift orbit" are considered.

Looking at the influences evaluated in the "design orbit" part, and after some discussions with other groups, the chasing orbit has been set to an altitude of 33 000 km. This altitude seems to be the best trade-off in terms of waiting time, but also in terms of dose received by the humans.

For the drift orbit the selection of the best altitude was also impacted by a new method for going from a satellite to another.

B. Earth coverage

During the whole mission the crew will be in contact with Earth, but to insure so the Earth coverage of the orbits used needs to be checked. Earth coverage refers to the part of the Earth that a spacecraft instrument or antenna can see at one instant or over an extended period. The critical criteria is to know how much percentage of the Earth the spacecraft will be able to "see" for the highest orbit, i.e. GEO. Figure 1 shows the evolution of the percentage of coverage of Earth

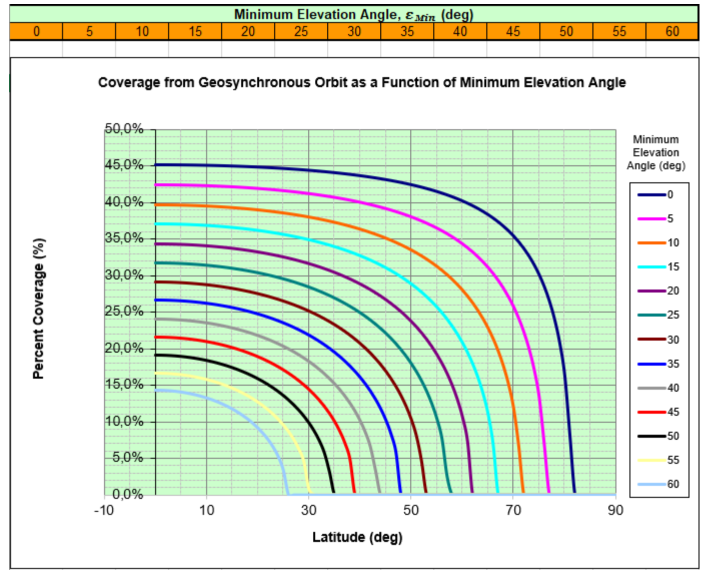


Figure 1: Evolution of Earth coverage with latitude and elevation angle, at GEO.

from a geosynchronous orbit depending on the latitude and the elevation angle.

One can see that with a latitude of 0 degrees the Earth coverage will go from 15 percent to 45 percent. Compare to other missions data these numbers seems logical and reasonable.

C. Final orbit selection

Finally, after a last discussion with the whole team five orbits were fixed:

- LEO orbit at 300 km.
- Meeting orbit at 25 000 km: this meeting orbit will be the place where the storage module and the mission module meet in order to let the crew take what is needed for the mission on the first meeting, and then take the re-entry vehicle (previously docked to the storage module) on the second meeting.
- Chasing orbit at 33 000 km.
- Drift Orbit between 34 000 and 37 000 km.
- GEO orbit at 35 786 km.

D. Time & ΔV cost

In order to perform the transfer between the selected orbits, an amount of ΔV needs to be added by the propulsion system, and the transfer time needs to be taken into account. There are two types of propulsion systems: electric and chemical. Both of them are utilised during this mission. The former has high specific impulse, I_{sp}, but low thrust is delivered by it and thus the transfer time is larger, so the electrical propulsion is used only for the unmanned vehicles which can take longer time travelling. The chemical propulsion is high thrusting, but demands larger amount of propellant, thus it is used for manned vehicles which for safety reasons need to spend the least amount of time in space.

For this mission a series of Hohmann transfer manoeuvres are applied for changing the orbit with high thrust devices (i.e. chemical propulsion). There are two types of Hohmann transfers that are used. The first is a regular transfer between two different altitude orbits (e.g. between Chasing orbit and GEO) as shown in Figure 2. The second type is an angle drift double Hohmann transfer where the target orbit is basically at same altitude as the initial orbit, but the mean anomaly is different (e.g. between satellite 1 and satellite 2 at GEO) as depicted in Figure 3.

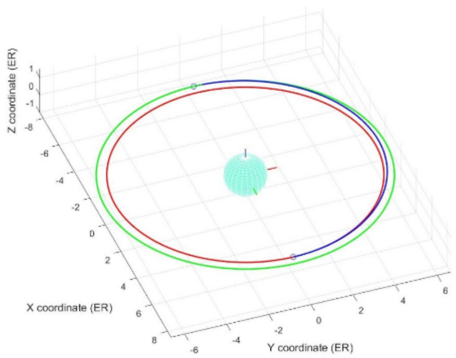


Figure 2: Hohmann Transfer between Chasing orbit (33 000 km) and GEO (35 851 km).

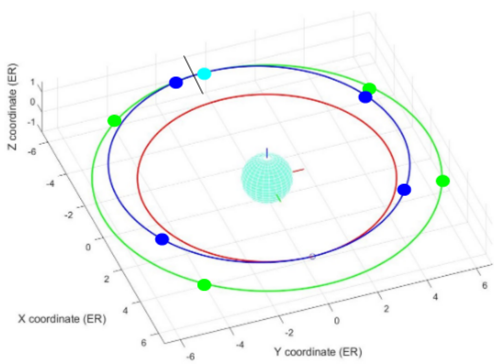


Figure 3: Hohmann Transfer between satellite 1 and satellite 2 both at GEO, but with an angle difference of 9°.

The Hohmann transfer consists of two impulse burns, one at the perigee of the initial orbit and the second one at the apogee of the transfer orbit to circularise it. Equations (1) and (2) describe the ΔV required for each burn of the manoeuvre.

$$\Delta V_1 = \sqrt{\frac{2\mu}{r_1} - \frac{2\mu}{r_1 + r_2}} - \sqrt{\frac{\mu}{r_1}} \tag{1}$$

$$\Delta V_2 = \sqrt{\frac{\mu}{r_2}} - \sqrt{\frac{2\mu}{r_2} - \frac{2\mu}{r_1 + r_2}} \tag{2}$$

Where the Earth gravitational parameter μ = 39 8601 km^3/s^2 and r_1 and r_2 are the radii of the initial and final orbit respectively. Following the same orbital mechanics it is possible to obtain the transfer time, as Equation (3) demonstrates:

$$P = \pi\sqrt{\frac{a^3}{\mu}} \tag{3}$$

Here a is the semi-major axis of the transfer orbit.

The angle drift manoeuvres consist in two burns but only at the apogees of the drift orbit. Since it is not possible to catch up with the next satellite without changing the altitude of the transfer orbit, a first burn reduces the semi-major axis of the module in order to drift, and a second burn at the apogee restores the GEO orbit when the drift is complete.

It is possible to perform the transfer without spending too much ΔV by not going too low with the transfer, but at the expense of long transfer times. So this represented a trade-off challenge when designing the transfer orbits.

The semi-major axis of the drift orbit is given by Equation 4, depending on the angle that we have to drift $\Delta\theta$ in radians and r_z the GEO altitude. The transfer time is the period of this drifting orbit given by Equation 3. At the end the values found are near one day of transfer time. The results for all the chemical propulsion manoeuvres are presented in Table II.

$$a = r_z \sqrt[3]{\left(1 - \frac{\Delta\theta}{2\pi}\right)^2} \tag{4}$$

Table II: Transfer Time and ΔV for manned vehicle manoeuvres.

Transfer	From	To	Time (h)	ΔV (m/s)	Start Alt. (km)	End Alt. (km)	Angle Diff. (°)
Launch	GTO	CO	72	1502	300	33000	-
Start	CO	Sat1	11.4	109.2	33000	35851	-
GEO	Sat1	Sat2	24.1	11.2	35851	35785	9
GEO	Sat2	Sat3	24.2	4.3	35785	35793	0
GEO	Sat3	Sat4	24.2	25.3	35793	35787	4
End	Sat4	MO	8.8	801.7	35787	25000	-
Entry	MO	Earth	6	1479	25000	100	-
		Total	170.5	3513			

Table II shows all the movements needed during the mission, from launch to re-entry, with their corresponding transfer times and ΔVs. It's possible to observe that for the angle drift Hohmann transfers, the time is around one day for every of the satellites, so only for transfer the repair mission itself would take three days (plus ~20 hours of arrival and departure to/from GEO). It is worth mentioning that the values for the altitude of the satellites are not strictly the same, as considered before, nevertheless their difference is small compared to their semi-major axis, thus it is negligible and can be treated as if they were the same for the sake of explanation.

These numbers are promising, unfortunately they're not the absolute values since the repairing time, which is four days per satellite, is still missing. Finally, it is needed to calculate waiting time. The waiting time is defined as the time the spacecraft needs to wait before doing the transfer, in order to allow the two orbiting objects to get in the correct position. Otherwise the rendezvous can't be achieved. The synchronisation or phase-correction of the orbits, in this mission, is required from CO to GEO and from GEO to MO, and can be calculated from Equation 5 with $\Delta\Phi$ the angle to drift and a_1 and a_2 the semi-axes of the both different orbits.

$$t_w = \Delta\Phi \left| \sqrt{\frac{a_1^3}{\mu}} - \sqrt{\frac{a_2^3}{\mu}} \right| \tag{5}$$

Originally, the modules that goes into GEO with the crew was supposed to meet the re-entry vehicle and the other modules in CO (so not in MO). However, the worst case waiting time between GEO and CO, when the angle difference is the worst, was calculated to be 10 days. Due to the fact that the crew is in this module in GEO, it was an unaffordable time length, it would cost too much in term of Life Support System, radiation, weight etc. Hence the MO has been design in order to optimise the phase correction in such way that the wait time doesn't exceed one day (12 hours from CO to GEO and 12 hours from GEO to MO). Indeed, the worst case wait time between GEO and MO being four days, the rendezvous is organised in MO instead of CO. The transfer between CO and MO allows us to play with the two drift speeds between GEO and CO and between GEO and MO in order to target an angle-to-drift of 20°between GEO and MO when the crew is ready to go to MO. This angle-to-drift of 20°before to do the transfer from GEO to MO is designed to be less than 12 hours but enough in the case where the crew is late compared to the schedule. Taking into account these times, the repair mission time is 20 days. With all these values the total mission time can be obtained. The values are shown in Table III.

Table III: Total Mission Time

	Transfer Time	Waiting Time	Repair Time	Total Time
Hours	170.5	2 × 12	4 × 96	578.5
Days	7.1	1	16	24.1

The electrical propulsion transfers, as already mentioned, are low thrusting and therefore take longer than those of high thrust. Nonetheless, they are more efficient in terms of propellant mass required, so it is convenient to use them when possible. Since there are several modules that will travel unmanned, time is not the most important factor when selecting the transfer orbits. ΔV requirements are a little higher than those of chemical propulsion, but this ΔV increase is exchanged by a decrease in total fuel required. Although the ΔV can be kept in a similar level if the acceleration is reduced (thrust is decreased), but that would make the transfer time even longer. Also, since the spacecraft now will move by spiralling instead of making an elliptical transfer the time is calculated differently. Equation (6) is used to obtain the transfer time with the thrusters selected by the space vehicle team.

$$\Delta V = \sqrt{\frac{\mu}{r_1}} - \sqrt{\frac{\mu}{r_2}} = A(t - t_0) \tag{6}$$

A is the acceleration of the propulsion system, which is obtained by Newton's second law, $F = MA$ (where M is the mass). Space Vehicle team selected a thrusting system that can deliver 5.5 N which gives A between 0.00020 and 0.00032 m/s^2 depending on the weight of the spacecraft. $(t - t_0)$ is the time of travel, which is of particular interest. The values for the electrical low thrusting transfers are presented in Table IV.

Table IV: Transfer Time and ΔV for unmanned vehicle manoeuvres.

Transfer	From	To	Time (days)	ΔV (m/s)	Start Alt. (km)	End Alt. (km)
Launch	Earth	LEO	70.125	-	0	300
Pos.	LEO	CO	275	4487	300	33000
Start	CO	MO	15	382.6	33000	25000
Restart	MO	CO	15	382.6	25000	33000
		Total	305.12	5252		

Here the launch is not of interest, since it is done by the rocket which is analysed in following sections. As it is expected the electrical spiralling takes a very long time, but it is affordable (by means of timing) because this movement is done only once for the whole project and can be done before the first mission starts. So the service module is launched unmanned 275 days prior to mission start, and once it reaches the target orbit, which is the chasing orbit, the crew can dock and start the mission. Then with electrical propulsion the modules that are unmanned during the repair service will go to the meeting orbit to wait for the crew. This transfer is done in 15 days, and since the repair mission time is 20 days, there is enough cushion time for the service module to be positioned. Finally, after the crew returns to the Earth the service module will return to chasing orbit to start a new mission. This transfer takes another 15 days, which means that if the next crew is launched on the 12$^{\text{th}}$ day of travel of the service module (because the launcher takes three days to deliver the crew to chasing orbit) the missions can restart every 12 days.

V. Launcher

To access space, typically $34\,\mathrm{MJ\,kg^{-1}}$ (for LEO), and $58\,\mathrm{MJ\,kg^{-1}}$ (for GEO) are required [2], which is a big challenge accounting the mass of the satellites. Hence, to access space and launch something into orbit, a specific tool is necessary: a launcher or launch vehicle.

A. Selection

To select the most suitable launcher, the first step is the selection of the mission to be performed, and all the resulting constraints. For this part the main discussion was with the Space Vehicle, Service and Human Aspects teams, in order to achieve a common decision for the overall mission, and to define the total payload mass.

According to the prescribed requirements of delivery a payload up to GEO, the selection process started with a comparison of orbital launch systems, and the first constraint established was that the launcher should have a payload capacity to GTO of at least $7.5\,\mathrm{t}$. Moreover, another constraint concerned the first date of flight. Since a lot of new technology are currently under development it seemed reasonable to select only the launch vehicles that have a scheduled first flight date by 2025 to ensure the start of the mission by 2030 [4]. The *Long March 9*, the Chinese super-heavy carrier rocket that is currently in study, has been discarded despite its promising capacities in terms of payload mass because the first flight is scheduled for 2030. So, among all the possibilities, the best two launchers developed by U.S.A. and two by Europe were selected, and the main features of each one are listed in Table V.

Table V: Main features of the selected launchers.

Vehicle	Origin	Payload to LEO - GEO
Vulcan/ACES	US (ULA)	37.4 t – 18.5 t
Vulcan/Centaur	US (ULA)	25.0 t – 15.0 t
Ariane 6 A64	EU (ArianeGroup)	21.6 t – 11.5 t
Ariane 5 ECA	EU (EADS Astrium)	21.0 t – 11.1 t

The Vulcan/ACES and Ariane 6 A64 were selected in the end after discussion with the Space Vehicle team.

B. Timeline and schedule of the mission

The mission is planned to start in 2030 with two Ariane 6 A64 rockets taking off from Kourou in French Polynesia. They will carry the MM, MSM, SM and AL to LEO. The modules will then use electric propulsion to transfer to the CO. These operations are estimated to take about 275 days.

The next step is to launch the LEV and LSM with the crew to LEO, using the Vulcan/ACES launcher. This will be done as soon the other modules has reached CO and is therefore planned to be done in 2031. The LEV, LSM and crew will use chemical propulsion to reach CO where they will dock with the other modules. A re-docking will then be performed resulting in the SM, LEV and LSM staying in CO while the MSM transports MM and AL together with the crew to GEO. The modules will then perform the different reparations by going back and forth between GEO and CO. The repair time for four satellites will be around twenty days.

Once the reparations are finished the modules will meet in the MO. The crew will transfer back to the LEV to be able to go back to Earth, and their mission is then finished. A new mission is then ready to start where a new MSM and LSM have to be launched together with the crew for every mission.

Figure 4 and 5 show all the different steps of the mission.

VI. Off-nominal case

Each group in the project was asked to think of an off-nominal case and come up with a solution for the chosen problem. Possibles issues which were discussed included environmental issues, emergency launch plans and manoeuvres, acute health problems and re-scheduling of the launch plan in case of unforeseen delays.

After some discussion, the following scenario was chosen:

> *A coronal mass ejection (CME) is detected from Earth and will reach the planet within a few days. The working astronauts are currently located in GEO and are requested to immediately return to Earth as fast as possible.*

Figure 4: On-orbit operations with MSM, MM, and AL.

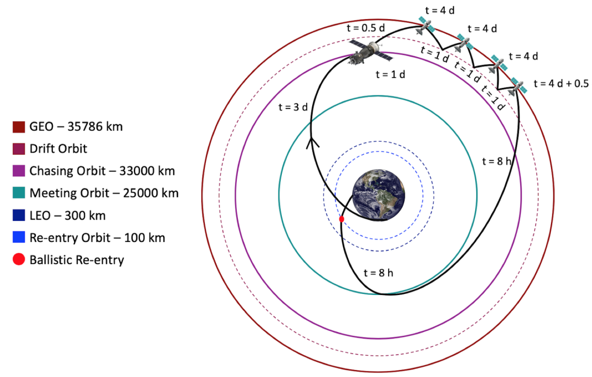
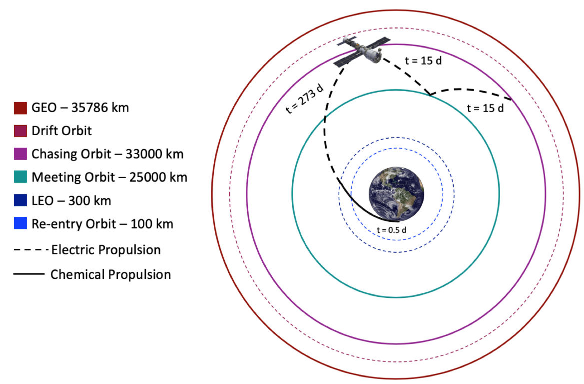


Figure 5: On-orbit operations with LEV, LSM, and SM.



CMEs are associated with both flares as well as geomagnetic storms. Flares are dangerous since they produce high energy particles and radiation that are hazardous to humans. Geomagnetic storms are caused by CMEs disturbing Earth's magnetosphere. They occur about three to five days after the CME leaves the Sun. The geomagnetic storms can cause major damage to satellites orbiting Earth, especially those in GEO. The satellites may become charged during the storm and discharge by high currents. Damaged satellites could also cause a major communications problem for the astronauts in orbit. On Earth, we are protected against these phenomena due to the magnetic field and atmosphere. The astronauts should therefore be brought back immediately in the case of a CME. [3] There are two critical factors to take into regard when planning for the solution to the scenario. The first one is time. A reasonable demand is that the astronauts should be able to return to Earth in less than a day. This demand was a trade-off between how fast the CME will reach Earth and the complexity of returning from GEO.

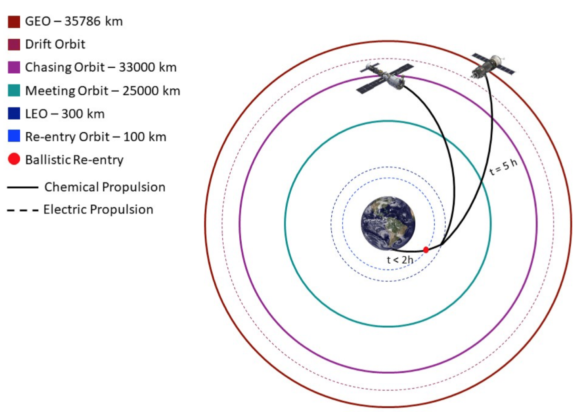
The second major factor is that the heat shield needed to pass the atmosphere is mounted on the LEV, which is located in the CO while the astronauts are in GEO. The astronauts therefore have to retrieve the LEV in some way. However, the constraint of time means it is not possible to sit and wait very long for the orbits to align properly.

The proposed solution to these issues consists of the following steps:

1) As soon as the astronauts are reached by the command to return to Earth they abort all sorts of activities. They immediately change their trajectory to reach LEO.
2) In the same time a command is sent to the LEV to undock from the rest from the modules in the CO. The LEV is then remotely commanded to meet the astronauts in LEO.
3) The astronauts make a rendezvous with the LEV and transfer themselves into it.
4) With the LEV the astronauts continue to the re-entry orbit, and make a ballistic re-entry to Earth.

The steps are shown graphically in Figure 6.

Figure 6: Off-nominal case manoeuvres.



Step one and two will according to calculations take about five hours in terms of transfer time. Step four will take less than two hours, in total adding up to less than seven hours of transfer time for the return to Earth. However, time for aborting the reparation activities and let the astronauts transfer to the LEV and make proper flight preparations will have to be added to the total return time. Regardless, the emergency return will stay well within the time limit of one day. It is also important to mention that even though the steps is a solution to the specified scenario they apply to any kind of emergency scenario which implies to abort the mission and immediately return to Earth.

From a safety point of view the proposed steps will meet the earlier set requirements. However, from a risk analysis point of view it is of course a complication that the astronauts have no way to return without first performing a rendezvous with the LEV. There are a number of things that could possibly go wrong with the auto-undocking and automatic travel through space down to LEO. It requires quite fine manoeuvres to make sure the modules will meet as planned.

The reason for not bringing the LEV along to GEO is that the heat shield is quite massive and the LEV will therefore imply a large mass increase. This mass has to be transported back and forth between GEO and the chasing orbit during the service of the satellites. That implies a lot of extra ΔV which implies more fuel is needed. In the end a trade-off must be made between the risks and increased costs. To fully evaluate this trade-off further analysis is required.

VII. Discussion

The values obtained in section IV, correspond to a set of predefined satellites, this is because the transfer parameters are dependant of their relative position. The altitude and period of the satellites are fixed, since by definition GEO satellites have a established altitude and period ($\sim$ 36 000 km and $\sim$ 24 hours respectively). This means that for an angle drift between satellites, the transfer will never be more than 24 hours since the spacecraft will go into a lower orbit to drift and wait until the target satellite reach the perigee of the transfer orbit.

In order to picture this procedure the following example is presented:

Assuming that the spacecraft is already in GEO and just finished to repair a satellite, and is ready to move to the next one, also in GEO, but with an angle difference of 180°. In the imaginary case that the spacecraft would be able to stay steady in its current position, the target satellite would catch up with the spacecraft in half period ($\sim$ 12 hours), which is the time it takes the satellite to travel the 180° left. Now, if instead of staying steady in its place the spacecraft makes a burn lowering its orbit, it will return to the same spot (perigee of the transfer orbit) after some time. This time depends on the magnitude of ΔV applied. Furthermore, if the burn is applied in a way that the period of the transfer orbit is exactly 12 hours, the spacecraft will catch the target satellite at perigee. With this explanation it would be desirable that the satellites are as far away as possible. This is, if the satellites are 359.999° away from each other it would only take the second satellite to reach the current position of the spacecraft the amount of time it takes for it to travel 0.001°, which is the minimum time. Nevertheless, the selected satellites are quite the opposite of this situation, they are rather close to each other (no more than 10° difference), the reason for this is that in order to reach the perigee of the transfer orbit the spacecraft would have to lower its altitude a considerable amount, which in turn results in an enormous quantity of ΔV needed, which is no longer feasible for satellites with high longitude differences.

Hence, the satellites selected are close to each other in order to reduce the ΔV. This doesn't affect too much the mission time since the major delimiter is the repair time. Compared to the total repair time, the transfer time is just a small fraction.

VIII. Conclusion

This paper proposed a study from a logistics point of view to realise a manned mission to GEO, in order to perform an "on-orbit" service to repair, assembly, and upgrade satellites. The space station including the Mission Service Module, the airlock, the Storage Module, and the Mission Module to be assembled in LEO orbit, and then using electrical propulsion to be transferred to a chasing orbit (33 000 km altitude). The transfer will take about 275 days. Two launches are required, and the new European rocket Ariane 6 A64 has been selected. Subsequently, the crew, the Launch Reentry Vehicle (LEV), and the Service Module for the LEV will be launched by Vulcan/ACES launch vehicle directly to the chasing orbit, and perform a docking with the assembled station. For the repair mission, four satellites has been selected to explain all the on-orbit operations. The Mission Module, the Mission Service Module, and the airlock (called the ”spacecraft”) undock from the station, and by means of chemical propulsion, they will transfer to GEO orbit. Meanwhile, the Storage Module, LEV and the Service Module for the LEV (called the ”space station”) will start a manoeuvre to transfer to the meeting orbit by means of electrical propulsion, and the journey is about 15 days. For the repair tasks, four days are required, and about one day is necessary to perform the transfer manoeuvres between two satellites. To reduce the chasing time, the ”spacecraft” will meet the rest of the ”space station” at the meeting orbit (25 000 km altitude). Chemical propulsion is used and the transfer takes about eight hours. The crew is then transferred into the LEV, to ensure a safe reentry, which will last about eight hours. Meanwhile, the Mission Service Module, the airlock, the Storage Module, and the Mission Module will come back to the chasing orbit by using electrical propulsion and as before, the journey is about 15 days. The mission can be then repeated with different satellites. The proposed solution for possible “on-orbit” repair mission with humans has also great potential for further improving concerns regarding the satellite selections, in terms of drift angle and position in orbit.

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