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OUTLINE OF A CONCEPTUAL SPACE FUEL STATION (SFS)

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ABSTRACT

It is no doubt that the satellite's lifetime, strictly related to the available amount of fuel at launch, has been one of the main constraint limiting operational capabilities since the first satellite-based commercial services. In this frame, the possibility of on-orbit refueling operation has opened an alternative approach to develop more lifetime efficient satellites, reducing also the costs for space segments. In this study, the general market analysis of a satellite, which is named as a space fuel station (SFS) and will serve in orbits with inclinations of 94, 82, 52 and around 5 degrees, as well as some conceptual design level studies of its subsystems are outlined. Outputs of this study shows that characteristics and capability of an SFS depends on orbital locations of target satellites, types of fuel it can offer to its customer and refueling operation strategy.

ABBREVIATIONS

EDRS : European Data Relay System
CMG : Control Momentum Gyro
GEO : Geostationary Orbit
GNC : Guidance, Navigation and Control
HEO : Highly Elliptical Orbit
IMU : Inertial Measurement Unit
LEO : Low Earth Orbit
LOX : Liquid Oxygen
MEO : Medium Earth Orbit
OBDH : On-Board Data Handling
PMD : Propellant Management Device
RCS : Reaction Control System
RNDZ : Rendezvous
SFS : Space Fuel Station
SSO : Sun-synchronous Orbit
TM : Thermal Mechanical
TM/TC : Telemetry / Telecommand
TCS : Thermal Control Subsystem
XFC : Xenon Flow Controller
XFS : Xenon Feed System

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INTRODUCTION

It is clearly said that the space sector is evolving toward new space system with one idea; reducing the cost of access to space. Launch cost of one kilogram of payload for low-Earth orbit (LEO) is nearly 10,5k$ while geostationary orbit (GEO) is around 25k$ [Jonas, 2018]. One of the most important parameters affecting the launch cost is the total mass of the satellite(s). Obviously, the total mass of a satellite strongly depends on its propulsion subsystem where the amount of fuel loaded into the satellite is the main driver. Therefore, it is an undeniable fact that if the fuel supply service to the satellite can be provided in space, it will provide various benefits from the satellite manufacturer to its operator.

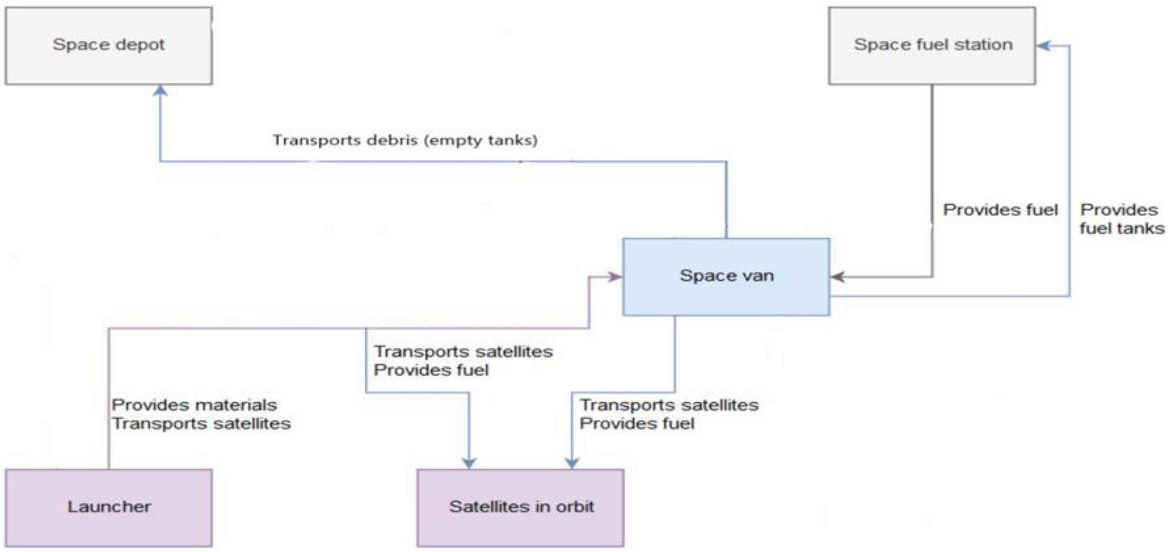

Figure 1: LEO Paradigm

The LEO paradigm falls within this tendency providing in-orbit services. It is composed of 3 reusable and multi-missions' subsystems which can work all-together. After the resupply of the SFS, the Space Van brings the empty tank to the imaginary Space Depot which is a part of LEO paradigm, where tanks and boxes are stored before deorbit. The imaginary Space Van is lighter than the satellite. Space Van is an autonomous and reusable transport system in LEO orbit. It can perform in a limited altitude range between 500 km and 1200 km. The Space Van is present in, where the SFS cannot reach to satellites, from 0° to 98° inclination.

Conceptual SFS developed within the scope of this study is a LEO platform where tanks are stored during several months/years to ensure the main mission of refueling space systems.

Cryogenic propellants—gasses chilled to subfreezing temperatures and condensed to form highly combustible liquids— provide high-energy propulsion solutions critical to future, long-term human exploration missions beyond LEO. The challenge is to develop a means of storing and transferring these propellants in space for long-duration missions, and preventing temperature fluctuations that contribute to fuel losses due to boil-off—vaporization of a liquid due to heating [NASA, 2013].

For the customer satellites having chemical propulsion, conceptual SFS have different available fuels which are liquid Methane (CH_4), Liquid Oxygen (LOX) in cryogenic state and Hydrazine (N_2H_4) whereas it has Xenon (Xe) in sufficient amount for the customer satellites needed electrical propulsion. Also, Helium (He) is needed as a pressurizer for LOX, LCH4 and hydrazine for SFS as they cannot be self-pressurized. Nitrogen has been selected for pressurizer gas for Xe as the large tank volume is needed according to the first estimations. In this study, it has been assumed that SFS has a total of 20 tons of chemical fuel and 2 tons of electrical fuel.

In frame of this study, several space fuel stations have been considered. In this case, 4 different orbital planes have been selected. These are Sun-synchronous orbit (SSO) (94°), an 82° orbit, typical for LEO telecom constellations, an 52° orbit where the ISS is and a near-equatorial orbit (<5°) to feed the GEO constellations before injection. Inclination selections have been made as a result of the market analysis. To reduce the order of magnitude of perturbative forces like the atmospheric drag and to minimize the overall probability of debris impact, altitudes of the SFS for each orbital plane has been adjusted as 800 km in circular orbit.

MARKET ANALYSIS

The market analysis conducted for the conceptual SFS allows to gain a better understanding of the market and its evolution over the next few years. This analysis is a starting point to estimate the role and number of conceptual SFS within the LEO Paradigm (whose different actors will also be customers) and in a possible standalone business if coming satellites have the capacity to rendezvous (RNDZ). In addition, such an analysis will allow to determine the number of resupply launches per year to maintain a certain baseline in propellant refueling rate. To do this analysis, several sources and files have been used such as UCS (Union of Concerned Scientist) Satellite Database and launch reports.

Operational LEO & GEO satellites

The actual number of satellites per inclinations is shown in Figure 2. For the costumers using these orbits, the refueling has been planned to be mainly Xe or hydrazine. According to the Figure 2, it can be inferred that main market is 0^0 (GEO), 52^0 and 98^0 (SSO).

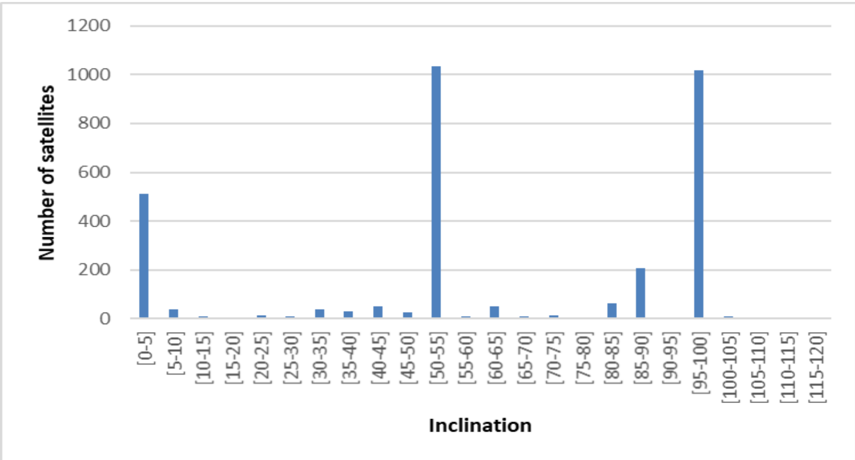

Figure 2 : Number of active LEO & GEO satellites

It is strictly recommended that refueling service should be provided on SSO because many satellites are serving in this location. From the Figure 3, regarding the altitudes on SSO, huge part of the satellites are between altitudes of 500 km – 600 km. In that point, there is going to be a need for a particular satellite, which is part of LEO paradigm and called Space Van [Koç and Ozer, 2020] and during this study, to ensure the connection between different altitudes because conceptual SFS has been assumed to serve in altitude of 800 km so as to reduce drag effect and make longer SFS's mission life.

For the 52° inclination, it can be found two different families of satellites: LEO and MEO (Medium Earth Orbit) families. Since the market considerations are based on LEO, MEO are out of the scope of this study. Here human space flights with the current ISS (International Space Station) can be potential customers. In addition, with the development of satellite constellations, the distribution which can be seen in Figure 3 will be highly modified in the following ten years. Indeed, SpaceX is planning to deploy 1600 satellites in 32 orbital planes of 53° of inclination. In that point, more than even one SFS can be located at this location next years.

In 82° inclination, satellites are mostly at 600-700 km altitude. With the deployment of constellations from OneWeb or SpaceX in polar orbits, additional satellites will appear around 1200 km and this location can be highly interested for several SFS with the contributions coming from Space Van, by bringing the satellites in polar orbits to SFSs.

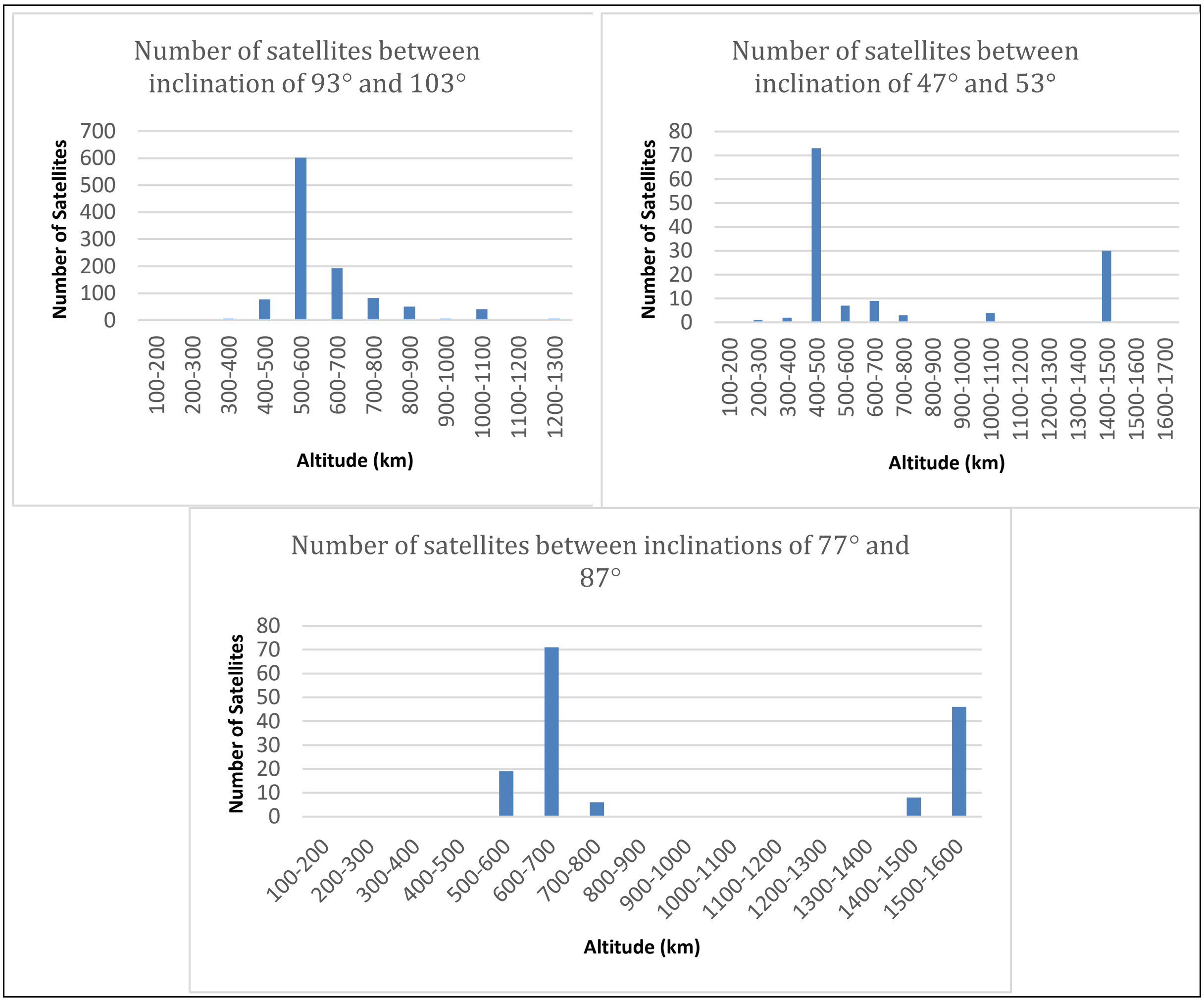

Figure 3 : Repartition per altitudes

For having better understanding of GEO market Figure 4 has been given below. According to the data obtained from UCS database, the total number of operating satellites are 3372, which also includes launches through 31/12/2020. As it can be seen from Figure 4, GEO satellites, which have different kinds of missions like telecommunication, navigation and meteorological, have the third bigger volume of total satellites orbiting around the Earth. When taking the new technologies developed based on satellite platforms into the account, it can be easily said that the need related conceptual SFS in this location is going to be mainly Xe for all electrical GEO satellites next terms. That's why, it is recommended that one SFS is supposed to serve in inclination of 0^0. Considering that the increasing competition among launcher manufacturers will gradually decrease the prices of launcher systems, it may be possible to see an increasing trend in the GEO market.

Consequently, the advantage with using the SFS is that customer will able to launch satellite with more payload without taking account the fuel mass at the launch as it will be provided by the directly SFS or Space Van as intermediary if a direct RNDZ is not possible.

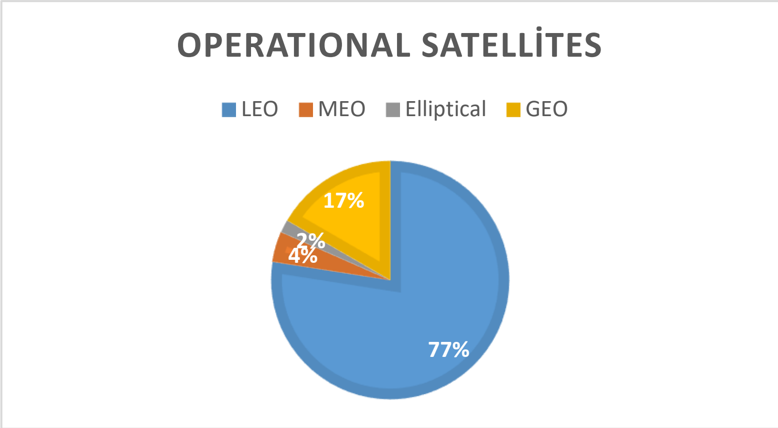

Figure 4 : Percentage distributions of operational satellites

MISSION ANALYSIS

Overall interactions of conceptual SFS with other space platforms is shown in Figure 5. Fuel tanks will be transported to space by launching rockets and then shipped to the SFS. The SFS will transmit these fuels to customer satellites.

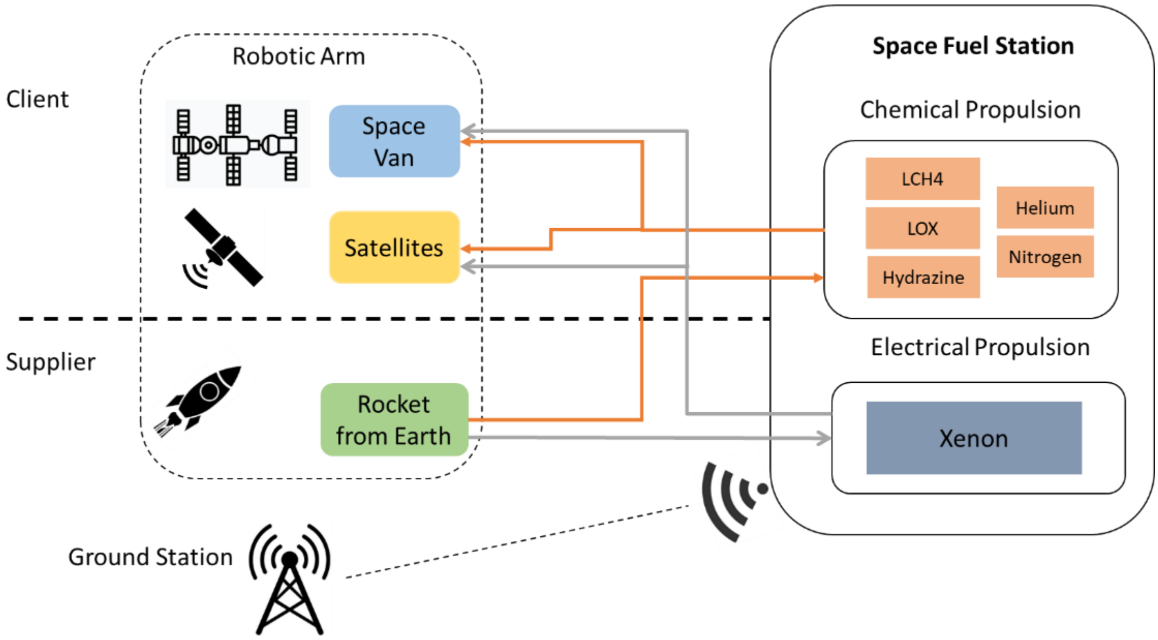

Figure 5 : Operations and interactions of SFS

Launch Scenario

SFS mission has been planned in terms of launch with A64 launcher. This launcher has a huge fairing, suitable for the large SFS solar arrays and body. According to the Figure 6, A64 approximately is able to carry 14 tons in SSO, 16 tons at 82°, 18.7 tons at 52°, 20.6 tons at 0° for an 800km circular orbit. As long as a full SFS is to heavy for a single launch, it will be launched with a minimum of propellants to ensure some refueling and its own orbit-keeping and additional launches will resupply it. Altitude of 800 km is the location that the separation with the launcher happens. Then, the mission really starts at the end of the commissioning phase where the SFS is fully operational and ready to refuel satellites.

	Altitude (km)												
	300	400	500	600	700	800	900	1000	1100	1200	1300	1400	1500
Inclination (°)	Performance (kg)												
5	21650	21500	21300	21100	20850	20600	20350	20150	19950	19700	19450	19200	19000
10	21550	21400	21200	21000	20750	20550	20300	20100	19850	19600	19400	19150	18950
15	21400	21250	21100	20900	20650	20450	20200	20000	19750	19500	19300	19050	18800
20	21250	21100	20900	20700	20500	20300	20050	19850	19600	19350	19150	18900	18650
25	21050	20900	20700	20500	20300	20100	19850	19650	19400	19200	18950	18700	18500
30	20800	20650	20450	20300	20100	19850	19650	19400	19200	18950	18700	18500	18250
35	20550	20400	20200	20000	19800	19600	19400	19150	18950	18700	18450	18250	18000
40	20250	20100	19900	19700	19500	19300	19100	18850	18650	18400	18200	17950	17750
45	19900	19750	19550	19400	19200	19000	18750	18550	18350	18100	17900	17650	17450
50	19550	19400	19200	19050	18850	18650	18400	18200	18000	17750	17550	17300	17100
55	19200	19000	18850	18650	18450	18250	18050	17850	17650	17400	17200	17000	16750
60	18800	18650	18450	18250	18050	17850	17650	17450	17250	17000	16850	16600	16400
65	18400	18200	18050	17850	17650	17450	17250	17050	16850	16650	16450	16250	16050
70	17950	17800	17600	17450	17250	17050	16850	16650	16450	16250	16050	15850	15650
75	17500	17350	17150	17000	16800	16600	16400	16250	16050	15850	15650	15450	15250
80	17050	16900	16700	16550	16350	16150	16000	15800	15600	15450	15250	15050	14850
85	16600	16450	16250	16050	15900	15700	15550	15350	15200	15050	14850	14650	14400

Figure 6 : Preliminary Ariane 64 performance assessments for LEO orbits between 5° and 85°. (Capabilities)

RNDZ with Space Van

In this study, it is assumed that SFS is the target and does not perform any maneuver to RNDZ with any vehicle. In fact, Space Van which is the chaser and have the ability to phase and approach the station. Once at 200 meters to the station after phasing and approach, the docking phase starts. The chaser aims at the RNDZ target of the station performing relative navigation during the final translation of the RNDZ. The docking system shall be able to allow both docking and berthing. With the completion of the docking, the fuel space station becomes the master and the interface of the system transfers power, TM/TC data, fuel/oxidizer and gas to the vehicle. If there is a problem about SFS AOCS or OBDH system, which affects the docking process negatively, Space Van shall go to the parking orbit and wait until the problem is solved. SFS tries to fix the problem by using redundancy and control center command. If the problem is identified, isolated and recovered, Space Van approaches to the target again. If not, SFS is lost. As a result of the mission lost, procedures defining de-orbit strategy of the SFS shall be implemented.

Refueling Operations

The refueling is ensured between the two satellites thanks to the dedicated subsystem which will be discussed through the docking system. In addition, dedicated valves and lines in both satellites allow the transfer.

Resupply of the SFS

The Space Van provides the propellant resupply for the SFS. In fact, on each orbital plane on which a fuel station is deployed, a Space Van is dedicated to resupply operations. It includes orbit transfer capability along with autonomous RNDZ and docking hardware and a standard docking interface with fluid connections to allow fluid transfer to the SFS. General and rough procedure can be described as following;

- Ariane 64 launched from Kourou (France Guyana) brings a cluster of 3 to 5, 4 tons of tanks, depending upon the inclination.
- The Space Van dedicated to the resupply, dock with one tank and RNDZ with the SFS
- It docks with the SFS using the main universal docking port;
- The resupply is done as for a refueling through the docking port

- After the resupply of the SFS, the tank and the Space Van undock, and the Space Van brings the empty tank to the imaginary Space Depot, which is a part of LEO paradigm, where tanks and boxes are stored before deorbit.
- It repeats the procedure until the last tank.

Undocking Phase

Once the refueling/resupply is completed, the vehicle can be undocked and enters a distancing maneuver after separation with the SFS.

Waiting Phase

During the waiting phase, when the SFS is not involved in a RNDZ / refueling or resupply phase, it will be in idle mode, keeping the tanks pressurized and at the right temperature balance using the power provided by the solar panels and the batteries.

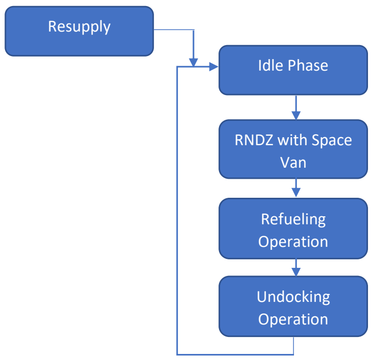

Figure 7 : Operation Cycle

Operation process of SFS has been summarized in Figure 7.

SFS SUBSYSTEMS

Platform equipment belonging to SFS are indicated in Figure 8. In this part of the study, information about definitions, functions and equipment of the subsystems with solution suggestions have been presented.

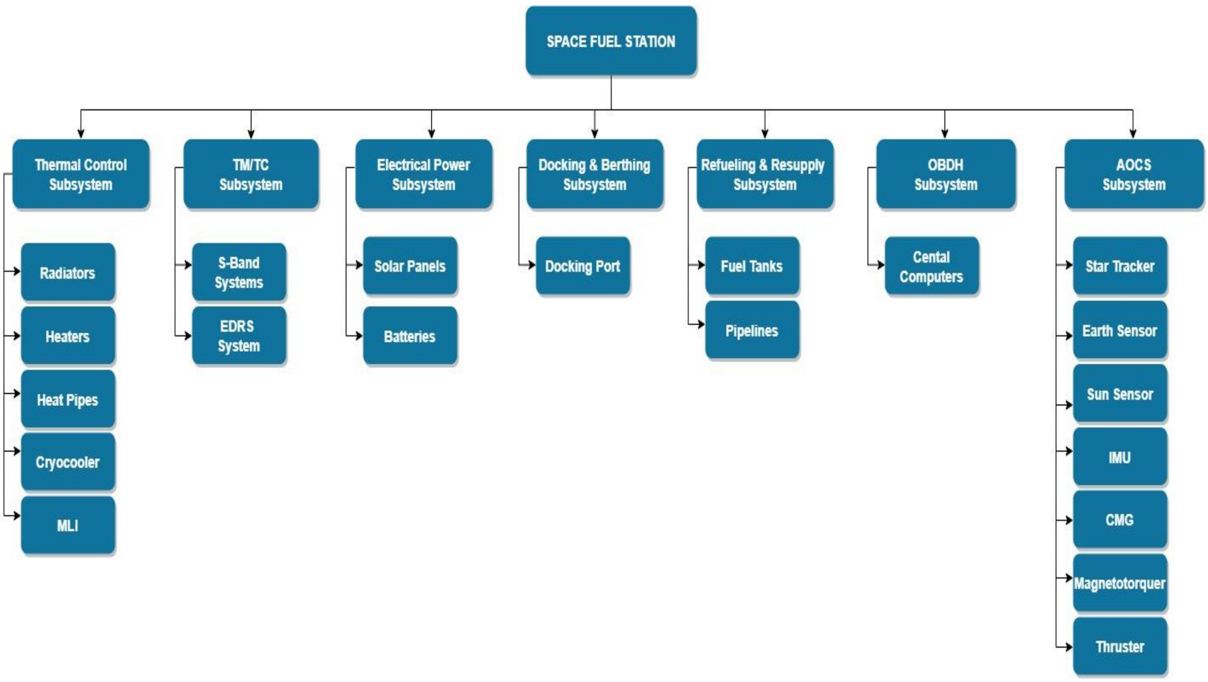

Figure 8 : SFS subsystems and equipments

Thermal Control Subsystem (TCS)

Thanks to the thermal control subsystem, it will be ensured that the equipment belonging to SFS will operate between their operational temperatures. Each equipment has different thermal requirements. Some of them can allow for wide temperature range while others require more strict temperature ranges, inducing several cooling and heating designs for each subsystem. In addition, LCH4 must be stored around 91 – 112 K and LOX at 54 – 90 K [Haynes, 2016]. SFS faces different fluxes coming from space environment. The main source of external heat load is the Sun. The Earth is also a significant source of radiant heat as the fuel station is in LEO. There are also internal heat sources, such as electronics and power generation equipment. For LCH4/LOX systems, the other main heat source is the temperature difference between two propellants. The North and South faces of the SFS are the coldest and most available ones. Then, they are ideal to place radiators with optical solar reflector covers. The other faces are considered at equilibrium, being covered by MLI blanket, inducing neutral temperatures. The highest dissipation comes from the cryocoolers used for active thermal control of the cryogenic tanks. There are also dissipations due to the electrical consumption of the avionics and batteries.

Taking everything into consideration, the final technical solution is to cover all the faces with MLI to isolate them, to paint in white the Sun facing side in order to maximize its emission, to use low conductive material to isolate nodes one from another, to use heat pipes to transport the heat from hot areas to cold ones or to dissipation systems, to install radiators on the face opposed to the Sun and to use heaters to warm-up the system during eclipse. In addition, the docking system is coated with MLI to reduce its temperature. Lastly, for the cryogenic tanks (LCH4 and LOX) thermal management is the most challenging condition. The solution for heat rejection is using an active cryocooler to pump heat out of the propellants. Although almost all cryogenic fluid cooling in space to-date has been of the passive variety, there has been significant research into active cooling techniques due to their promise for zero boil-off operations. It has been advised that the other tanks are enclosed in MLI to minimize Solar and Earth radiation heating. In addition, the main cover of the SFS acts as a sun shield, reducing system heating, while allowing residual heat to radiate to deep space.

Telemetry & Telecommand Subsystem

The SFS will have the opportunity to communicate uninterruptedly with the ground stations via TM/TC (Telemetry & Telecommand) subsystem. In this study, two different situations are approached for the communication concept. The first case covers the nominal flight (Idle phase) of the spacecraft, while the second consists of the docking and refueling/resupply phases. In the first case, the S-Band is being used while in the second case the Ka-Band is used. The antenna which will be used in this study for S-Band has been selected as S-Band patch antenna. The antenna is used for the first case (idle phase) of the spacecraft in order to monitor the data which belongs to several spacecraft's subsystems. However, this kind of antenna offers lower gain. In order to cope with this case, number of amplifiers might be increased. As mentioned, Ka-Band is only used for the docking and refueling/resupply phases. At these moments, control center needs to observe and monitor the process. Hence, it is needed high amount of data rate.

EDRS (European Data Relay System) [ESA, 2009] also plays an important role in uninterrupted communication in case of no visibility of SFS. It has been decided that EDRS will be used for the docking phase. The system provides data relay between low orbiting satellites and the EDRS nodes over optical links, with the information sent down to Europe in near-real time. EDRS laser beams are capable of higher accuracy and capacity than radio – up to a record-breaking 1.8 Gbit/s of user data. The terminal is also fitted with a Ka-Band radio transmitter to deliver the data to the ground station. Like the laser element, it is a two-way link (TM/TC).

Electrical Power Subsystem

The power that SFS will need will be provided by solar panels and batteries. The SFS has different functioning modes and thus different needs in terms of power. To have a better sizing related to EPS, worst condition, which is all redundant equipment are active, has to be taken into the consideration. Hence, SFS orbiting in equatorial plane will need more energy requirement because of having cryocooler systems. On the other hand, SFSs orbiting in other planes relatively will need less energy. To sum up, power budget for SFS in equatorial plane is calculated around 6 – 6.5 kW, assuming that equipment such as STR, Earth Sensor, Sun Sensor, IMU, CMG, Magnetorquer, EDRS, S Band System, OBDH Computers, SADM, Thrusters, Cryocooler System, Valves and Heaters are all operational with the margin of 5%. With the same approach, power budget for each SFSs in other planes has been predicted around 2.5 – 2.8 kW. Note that there is no need for cryocooler systems for SFSs orbiting other planes to pump heat out from the cryogenic fuel tanks.

The SFS will encounter eclipses during the mission. The duration of those eclipses is not constant and they are approximately in this preliminary design by a third of the orbital period. For the batteries, the lithium-ion technologies have been selected. This technology offers a significant volumetric and energy density advantage over Ni-Cd (Nickel Cadmium) and Ni-H2 (Nickel Hydrogen) battery types. In addition, the Li-Ion has a higher recharge efficiency, requires a less complex and less costly thermal control system and has a low self-discharge rate (around 5% per year). The main drawback of this technology is the need of equalization between each cell with a voltage monitoring in order to keep the battery in good health.

Solar panels of the SFS have a high importance due to the high power required for propellant management. For the computation, the impact of the thermal and radiative environment on the electrical properties of the solar cells need to be taken into the account. The sizing needs to be done by considering the worst case scenario, which is the end of life of the cells. It has been decided that the solar arrays of SFS will use triple junction gallium arsenide cells which are today the cell with the best efficiency (up to 30%).

On-Board Data Handling Subsystem (OBDH)

The OBDH subsystem of SFS must carry and store data from all the subsystems. OBDH subsystem includes on-board computers and their major components, like microprocessors and support components, meeting very stringent requirements in terms of radiation tolerance, reliability, availability, and safety. Therefore, it includes key avionics building blocks such as platform mass memories, remote terminal units, on-board buses and data networks and on-board and space to ground data communication protocols including protocol security aspects.

Within the scope of this study, it was found appropriate to follow the methodology presented in Figure 9 while designing the OBDH subsystem.

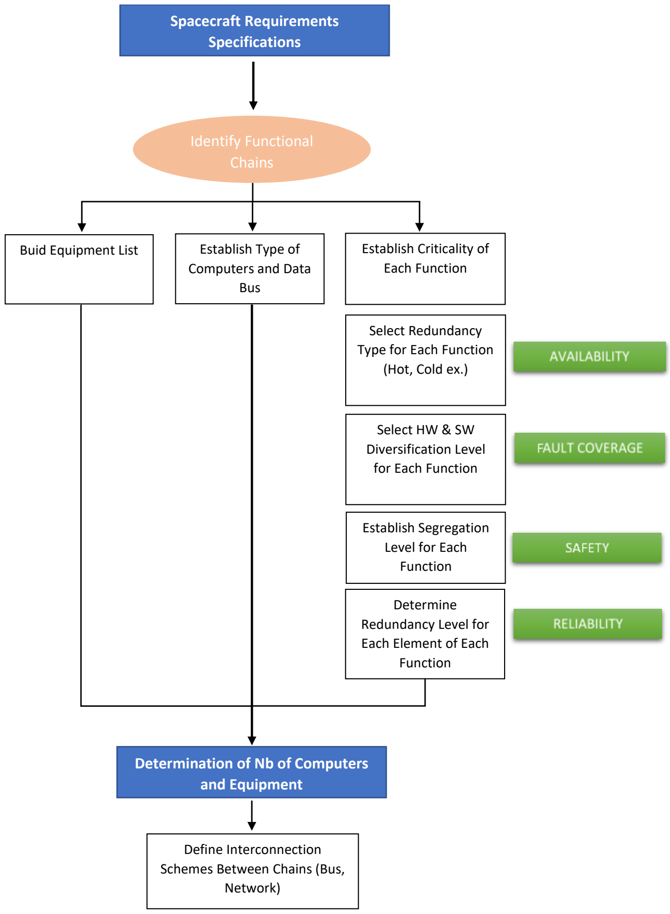

Figure 9 : OBDH design process [Koç and Bayraktar, 2020]

So, the mission phases and functions list are indicated in Figure 10.

Mission Phases	
M1	Launch
M2	Waiting Period
M3	Rendezvous
M4	Docking
M5	Refueling
M6	Resupplying
M7	Undocking
M8	Witing Period

Function	
F1	Power Control
F2	Propulsion
F3	AOCS
F4	TM/TC
F5	Refueling / Resupply System
F6	TCS
F7	Mission Management

Figure 10 : Mission phases and Function List

Then, criticality levels description and criticality levels of subsystems are shown according to the mission phases specified in the below Figure 11 and Figure 12, respectively.

Criticality Levels	
1	Function not mandatory
2	Function used in a degraded mode
3	Tolerable outage minutes
4	Tolerable outage seconds
5	Tolerable outage miliseconds

Figure 11 : Criticality levels

Phase	EPS	Propulsion	AOCS	TM/TC	Refueling/ Resupply	TCS	Mission Management
Launch	3	1	1	3	1	3	3
Waiting Period	3	2	2	2	2	3	3
Rendezvous	4	1	4	4	2	3	3
Docking	5	1	5	4	2	3	3
Refueling	5	1	5	3	5	5	3
Resupplying	5	1	5	3	5	5	3
Undocking	5	1	5	4	2	3	3

Figure 12: Criticality levels of each subsystem

After determination of mission phases and criticality levels of subsystems, RAMS (Reliability, Availability, Maintainability, Safety) analysis was performed. According to the results of this analysis, the redundancy numbers of the critical equipment were determined. The list of redundant equipment can be seen in Figure 13.

Equipment List	Nominal Number	Number of Redundant Equipment	Total Number of Equipment
CMG	3	1	4
Magnetorquer	41/7	10/2	51/9
IMU	3	1	4
S Band	2	2	4
EDRS	1	1	2
Thruster	2	2	4
Star Tracker	1	1	2
Earh Sensor	2	2	4
Sun Sensor	3	3	6
Computer	3	1	4

Figure 13: List of redundant equipment

Note that the refueling/resupply subsystem is not redundant as tanks are expected to be reliable. However, all the valves, pumps or any other actuators and sensors of this subsystem are doubled.

Docking & Berthing Subsystem

In the LEO paradigm, there is a need to develop a common docking system in order to allow interactions between the different actors and customers. For the docking/berthing system, there are two way. The first solution is using a robotic arm and the second solution is using universal docking system. The advantages and disadvantages of these two methods are given in the Table 1 below.

Table 1: Comparing Robotic Arm vs Docking Port

Solution	Advantages	Disadvantages
Robotic Arm	- Each single robotic operation are well-mastered - Independent from the other actors of the LEO paradigm	- Need of experience in robotic operations - Costs of development - Multiple dependent robotic operations to achieve
Docking Port	- Simplicity as all is done through a docking port - Innovative docking system already studied	- Dependent to the LEO paradigm actors - New docking port

The docking port solution seems to be easier than the robotic arm as it minimizes the robotic operations, for this reason the docking port solution was preferred. The universal docking port for SFS will be designed, and the study assumes that satellite at the customer position also have this system. The docking system unit was also designed to transfer data and power although no sizing activities have been performed in those fields because of the unknown power consumption and data rate. The UDS mechanism consists of male/female part and fuel lines connection. The UDS requirements is indicated in Table 2.

Table 2: Requirements of UDS

ID	Requirement
REQ-UDS-1	The UDS shall allow berthing and mating between the two spacecrafts.
REQ-UDS-2	The UDS shall allow to provide and receive electrical and data supply.
REQ-UDS-3	The UDS shall allow the refueling of Xenon/LOX/LCH4/Hydrazine.
REQ-UDS-4	The UDS shall resist to the on-orbit environment (Temperature, high energic particles, micro meteroites).
REQ-UDS-5	The UDS shall have a minimum on-orbit life of 15 years
REQ-UDS-6	The UDS shall be able to provide a minimum docking/undocking cycles of 2000.

Refueling & Resupply System

Refueling in space is not just about opening the cap, connecting the fuel nozzle to fuel the tank, and sending the satellite back on its orbit. The fuel valves of the satellites in space today are not designed to be accessed after launched and are locked down to prevent hazardous fuel from escaping during launch. The operations are more complex, and several techniques exist which are robotic arm and universal docking port. In addition, pumping fluid in the microgravity environment gives extra challenges as without gravity to settle the fluid in the bottom of a tank, like on Earth, tank plumbing and pumps must be more specialized to correctly operate in the microgravity environment of space. These extra difficulties will be addressed in the propellant management part.

Refueling/Resupply Techniques: The first refueling technique is using a robotic arm. The robotic arm is investigated by NASA Robotic Refueling Mission (RRM) and the phases are below:

- The phase of cut wire operations begin the removal of the wire keeping a cap screwed on over the fuel valve. To cut it, one robotic arm is using the wire cutter tool.
- The multifunction tool is used with an adapter to remove the cap and store it on the phase of removing cap. Then, it is used again to cut the wire securing the safety cap. The safety cap tool is used to remove the safety cap allowing to access the fuel valve.
- On the phase of refueling, the valve is accessible and the nozzle tool can be connected to the fuel valve. A sequence of command is sent to the SFS that transfer the correct propellant from tanks via the fluid transfer system, into the nozzle tool and through attached fuel valve. When the fuel transfer is complete, the nozzle tool disconnects from the valve.
- The same sequence would happen in the case of a resupply. The robotic arms would be used to remove its own caps and to ensure its own refueling from a tank sent by the Space Van to a customer.

The other refueling solution is using a universal docking port. The universal docking port is between the vehicle to refuel and the SFS and the phases are below:

- During the docking phase, the SFS is the target and do not perform any maneuver to rendezvous with any vehicle. The Space Van is which are the chasers and have the ability to phase and approach the station. Once close enough to the station, the docking phase starts. The chaser aims to rendezvous target of the station to perform relative navigation during the final translation of a rendezvous. The docking system shall be able to allow both docking and berthing.
- After completion of the docking, the fuel space station becomes the master and the interface of the system transfers power, TM/TC data to the ground, fuel/oxidizer and/or gas to the vehicle via the docking port. The choice of the propellant/gas is transmitted from the space vehicle to the SFS and the refuel operations start. The feed lines are leak checked so that to ensure a proper propellant/gas transfer. In the case of cryogenic refueling, fuel then oxidizer is transferred as separate transfer for each reduces the hazard in case of a leak.
- The resupply is ensured by the Space Van as explained in mission analysis part. The transfer is done via the docking port too.
- When the refueling/resupply sequence is completed, the vehicle can be undocked and enters a distancing maneuver phase after separation with SFS.

The second solution seems to be easier than the robotic arm solution since the universal docking port minimizes robotic operations. In this study, the refueling/resupply phase and docking system are taking in the account, the universal docking port solution was selected.

Propellant Management: There are two kind of refueling stations in orbit. On the non-equatorial orbits, it should be able to resupply Xenon for electrical propulsion and hydrazine for RCS (Reaction Control System) systems while in the equatorial one there is an additional need to refuel cryogenic propellants for interplanetary travels. The management of propellants under cryogenic conditions will be done with PMD (Propellant Management Device) [Wollen; Merino;

Schuster and Newton, 2010] and the details are being studied by the authors. The example of PMD can be seen on the below Figure 14.

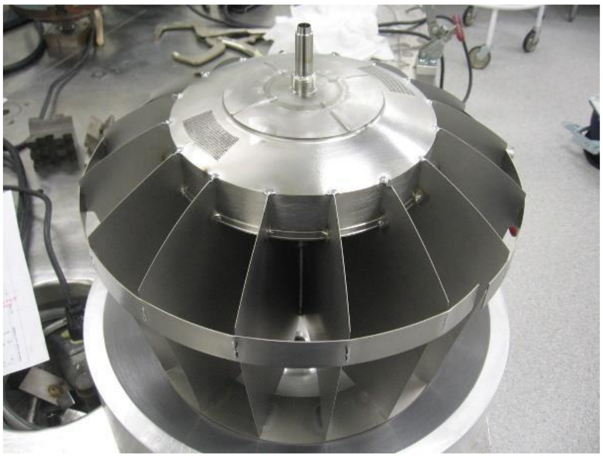

Figure 14: PMD Example [NuSpace, 2021]

i. Cryogenic Tank Handling System

The first key task of the fuel station is to store and handle the fluids on orbit and in particularly cryogenic propellants. Customers that rendezvous and dock to refill with the fuel station tanks will also require very high liquid fill fractions due to the cost of launching and storing propellant in LEO. However, the transfer of vapor-free propellant in LEO is a difficult task because of the uncertainty of the location of the liquid/vapor (L/V) interface within the propellant tank. [Meyer; Taylor; Ginty and Melis, 2014] For cryogenic tanks, spacecrafts can rely on PMDs, most often made in titanium. Their purposes are to separate the liquid and vapor phases within a propellant tank, and to transfer vapor-free liquid from the tank to a transfer line despite the variable gravitational and thermal environment of LEO. They rely on surface tension forces naturally produced by liquids to control the phase separation, and thus are best suited in low Bond number environments, where Bond number is defined as:

$$\boldsymbol{B_0 = \frac{\rho g L^2}{\gamma}} \tag{1}$$

where ρ is the density of the fluid, g is the acceleration due to gravity, L is a characteristic length scale of the system, and γ is the surface tension of the fluid. The screen channel and the vane device are the two existing PMDs. [Darr and Hartwig, 2014] They are shaped differently but both use surface tension mechanism to move liquid. The screen channel relies on the surface tension forces of the liquid in a fine pore size screen to form a barrier to vapor. Vapor cannot enter the screen until it overcomes the pressure force created by surface tension. Multiple channels are arranged throughout the tank to maximize contact with the bulk liquid. Individual channels can be tested by laying them horizontal in normal gravity. Vanes use thin pieces of rigid material to form a shape that is preferentially wet with liquid in zero-g. The vane technique has the advantage that bubbles in the retained liquid do not destroy its performance. It is also frequently lighter and lower cost. It has the disadvantage that it cannot not be tested in normal gravity.

These two techniques can be combined using vanes to retain liquid over the outlet and screen as a final barrier to vapor ingestion. However, the storage duration lasting several years, propellant transfer will require relatively high flow rates, and the fuel station will be subject to large accelerations and vibrations in any direction. The channel configuration can handle higher flow rates than vanes while also resisting vapor ingestion due to these random accelerations. If a choice has to be made between the two solutions, the channel will be selected. [Darr and Hartwig, 2014] The screen channel PMDs is about choosing the type of pressurant gas. The two ways to pressurize a cryogenic propellant tank are autogenous and non-condensable. Autogenous pressurization uses the vapor (condensable gas) of the liquid as the pressurant gas, whereas non-condensable pressurization uses a pressurant gas that does not condense into the liquid, such as helium. The type of pressurant gas will affect the performance of the PMD and therefore affect the design of the pressurization system of a fuel station. The cryogenic tank architecture can be seen on the below Figure 15.

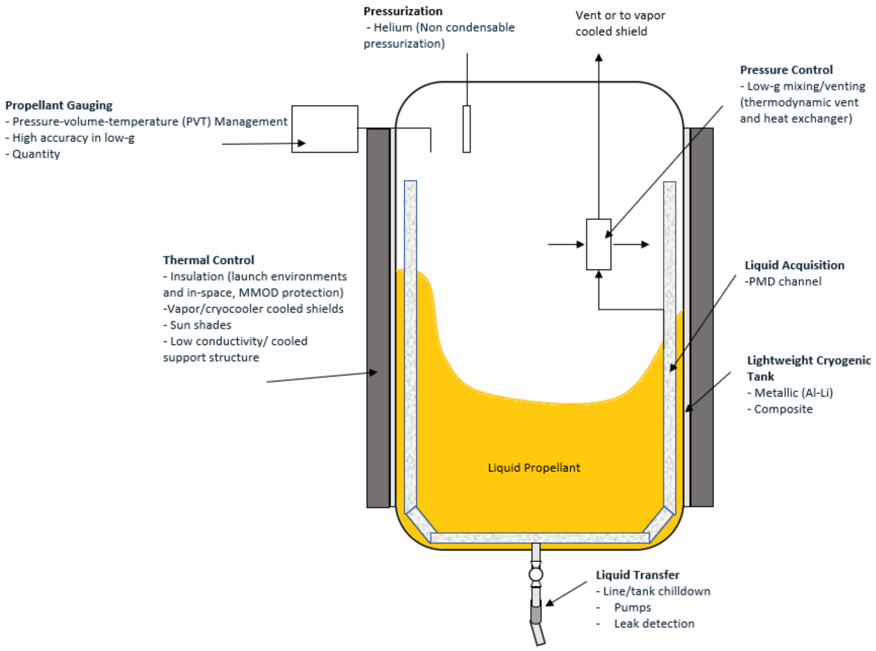

Figure 15: Cryogenic Tank Architecture

ii. Helium Management

For the SFS, helium is needed as a pressurant for LOX, LCH4 and Hydrazine as they cannot be self-pressurized. During a refueling, the tanks quickly empty and the propellant is unable to vaporize fast enough to keep up the pressure. However, helium shall only be managed as it is not transferred to any customers. This is due to the fact that spacecrafts that would need helium to pressurize the tanks can be launched with this tank full as helium is light.

However, if a possible improvement is to transfer Helium to these vehicles, the technique is known, and transfers are conducted with the use of a thermal mechanical (TM) pump using the unique property of superfluid helium to move in the direction of warmest temperature to move the fluid. In this case, it should be decided that for the resupply of helium to the fuel station, the gas is launched as normal boiling point fluid and converted to superfluid helium by venting down through a porous plug. After the conversion, helium in fluid is available and can be transferred.

For the Space Fuel Station’s own needs, gaseous helium is needed for pressurization and it is stored in this state as well. It could have been stored in liquid state as the density is bigger than the gaseous state, but it requires a temperature of 4K driving then a stringent requirement in energy needs.

iii. Hydrazine Management

Hydrazine is used by Space Vans for RCS systems and by some customers' satellites for their propulsion. It is stored in liquid phase at ambient pressure and temperature in the tank of the SFS.

To avoid the problems of phase separation, a membrane can separate the liquid from the pressurant gas. Then the liquid can be transferred by pressurizing the tank without worrying about ingesting vapor. However, drawbacks of this system include life of the membrane, weight and an inability to deal with vapor evolved from the bulk liquid. High-pressure helium is used as the pressurant. A compressor is used to lower pressure in the receiver tank by transferring helium back into high-pressure storage bottles of the Space Fuel Station.

iv. Xenon Management

In the typical electric propulsion system, the xenon fuel is stored at temperature above the critical one (around 30°C) at pressure of 150 bar guaranteeing the supercritical state where xenon's liquid and gaseous phases coexist. There is no need of pressurization as when it is needed by thrusters, it goes to a depressurization system allowing 3 bar and a gaseous phase at its exit. The fluid is controlled by a Xenon Feed System (XFS) and the power by a Propulsion Power Unit (PPU). [Bucci; Capannollo and Cavenago, 2015]

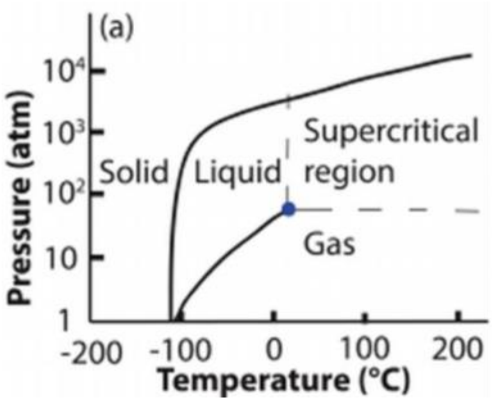

Figure 16: Xenon Phase Diagram [Lynch-Klarup; Mondloch; Raymer; Arrestier; Gerome and Benabid, 2012]

This process is not adapted to Xenon transfer in the same state from one tank to another. Nowadays, no technique allowing this kind of transfer exists. However, the fuel transfer can be done exploiting the pressure difference between customer and fuel space tanks. The easier solution is a simple free expansion where the mass flows is from the station to the serviced vessel until the pressure reaches the equilibrium. The limitation of this strategy is the impossibility to transfer all the load of the station. In order to overcome this problem, it is possible to use a pressurant gas (helium) to avoid the decrease of pressure in the fuel station tank, or an active machine able to keep a given pressure difference between the reservoirs. The former solution seems the most attractive, since it allows the transfer of the whole mass of fuel, it requires less power and it is more reliable.

If this solution is put into practice, there is a need to separate Xenon in its supercritical form and its pressurant to avoid mixing with each other. Then, the Xenon will be stored in a tank equipped with elastomer membrane able to inflate (pressurization) and deflate (depressurization) tanks to a pressurant. This membrane should be compatible with Xenon. As Xenon is highly stable (xenon only reacts with Oxygen and Fluorine as they contain a strong electronegative element which is high in energy barrier that can move the electrons inside xenon elements), this point should be observed. In addition, there is a need to quantify the number of cycles and the fatigue of the membrane as well as qualifying the elastomer regarding its contamination to the Xenon. As the tank has an important volume so that nitrogen was selected for the pressurant.

Finally, some extra-elements need to be designed. A Xenon Flow Controller (XFC), composed of a flow restrictor valve controlled by a chip connected to a mass flow sensing device, is needed for the transfer as it allows the mass flow rate to be regulated. It is very similar to XFS in the electric propulsion systems but as it is working with different mass flow rates, an adaptation is necessary for the present application. It will be located on the end-effectors of the system near the docking system. In order to control the temperature and the pressure, some sensors should be provided in the system while some heaters and a robust thermal control are necessary to keep the correct xenon conditions.

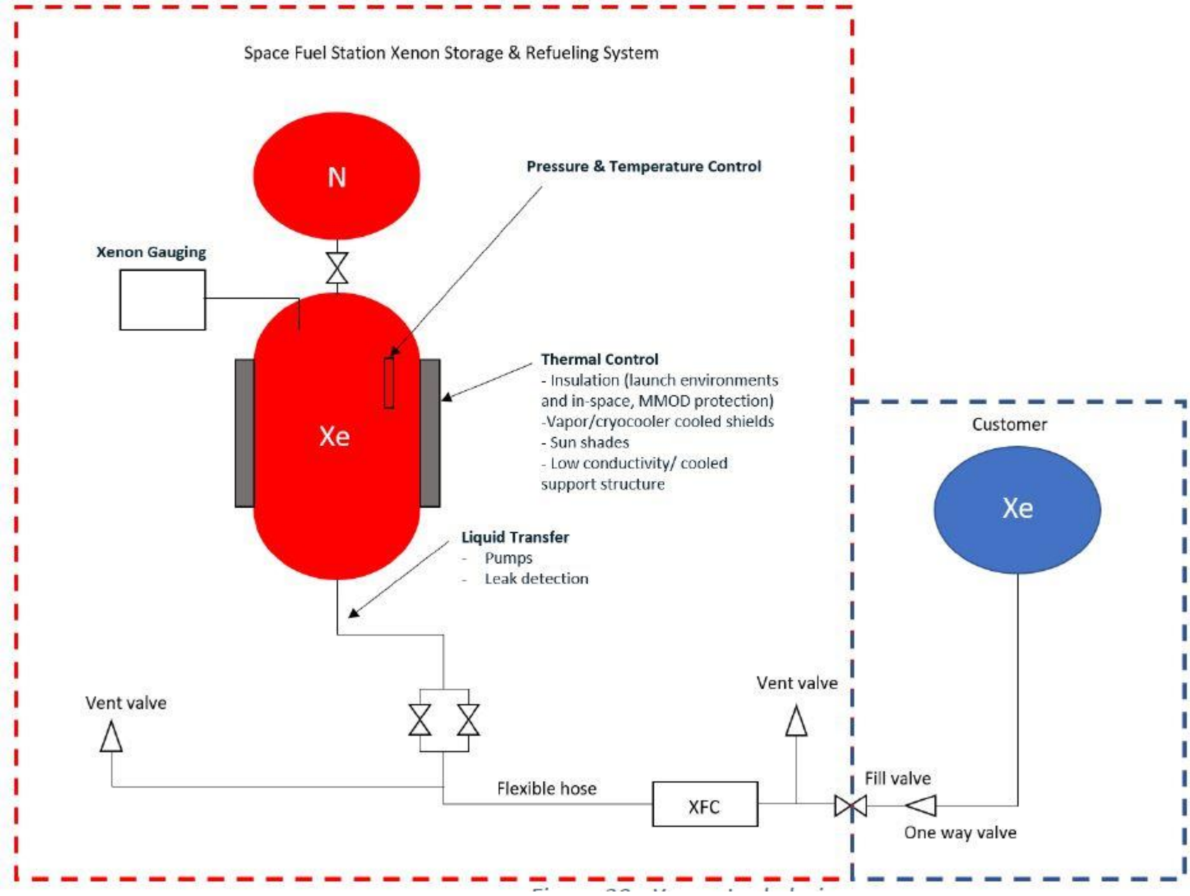

Figure 17: Xenon Tank Design

The system architecture of xenon tank design has one main tank and several valves in order to guarantee a certain level of redundancy. Different lines are connected to switches and then to the final interface valve. The Space Fuel Station can be connected to the feeding line by means of a coupler at the end of a flexible hose.

Refueling/Resupply System Architecture: In the below figures, the main architecture of the two different fuel stations are depicted to highlight the transfer process between the station and the customer and the resupply process.

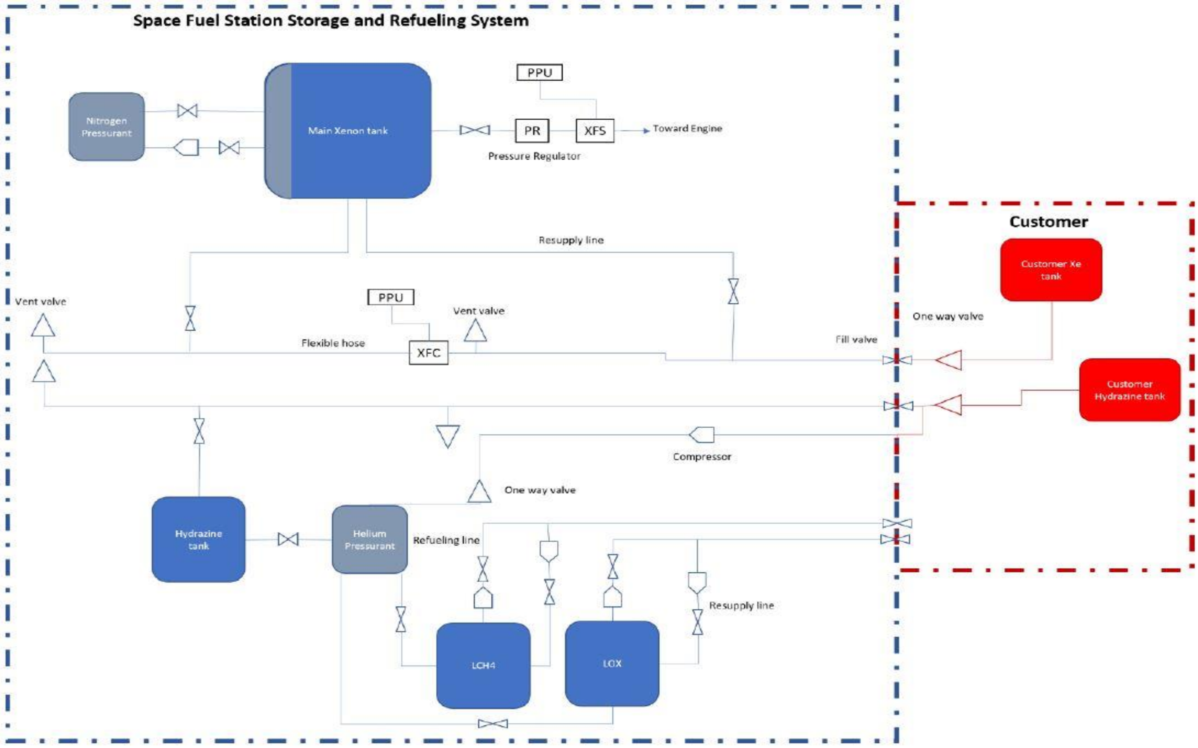

Figure 18: Refueling/Resupply Architecture in the equatorial orbit

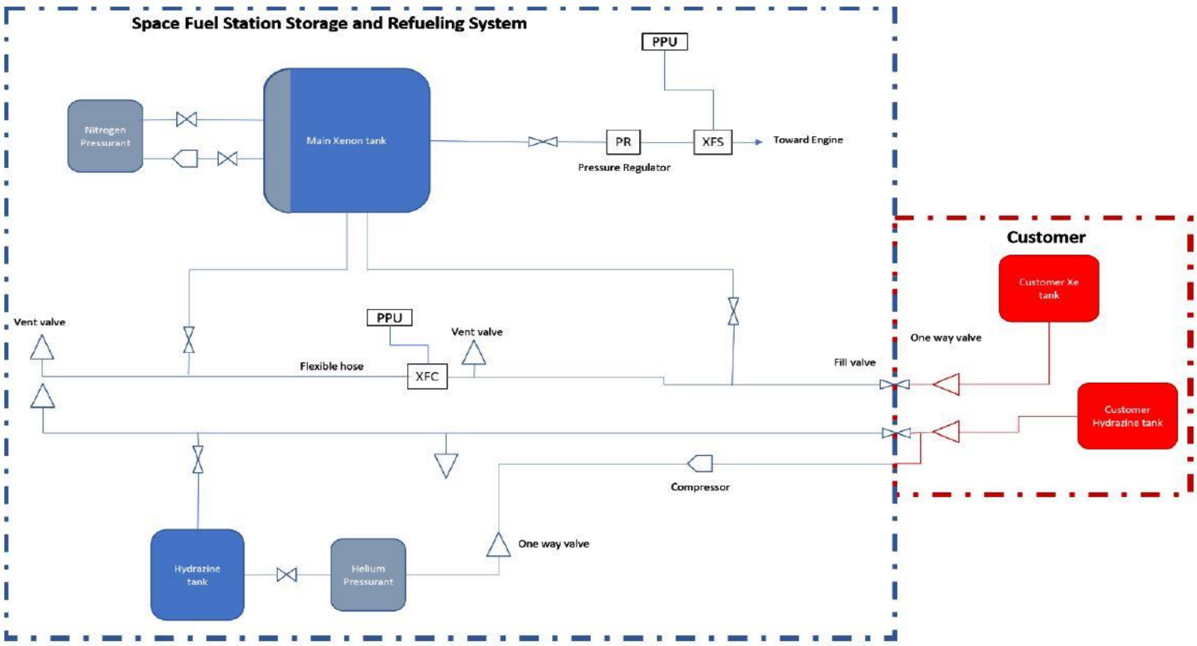

Figure 19: Refueling/resupply Architecture in the other orbits

Each tank has two parallels feed lines going to the docking port and two others coming from the docking port to the tanks allowing the resupply. An isolation valve is located at the end of each pipe and is doubled in derivation. This disposition makes it possible to avoid single point failure as the system is two failures resistant. In addition, the tanks have their own architecture with flux, pressure regulators, their power supply. As was mentioned previously, all this architecture is regulated in terms of temperature in order to remain either in the supercritical, gaseous, liquid or cryogenic states. This specific layout is not displayed in following schemes displaying the general architectures of the refueling/resupply system.

Attitude and Orbit Control System (AOCS)

AOCS is responsible for managing the orientation and the position of the SFS, which is a critical part of the SFS, because the it must hold a precise attitude. AOCS equipments have been selected by taking heritage concept into considerations. It must be reliable to prevent collision and perform orbit maneuvers to align the thrust vector phases, but also during the refueling and resupply activities.

To select compatible actuators for AOCS, external and internal disturbances must be defined well. The following Figure 20 describe main external disturbances with their approximate magnitude.

Disturbance	Source	Magnitude (Nm)
Solar Pressure	Sun	10^{-5}
Drag	Earth	10^{-6}
Gravity Gradient	Earth	10^{-4}
Magnetic Field	Earth	10^{-4}

Figure 20: Main external disturbances [Fortescue; Swinerd and Stark, 2011]

In addition to external forces, the SFS is also perturbed by internal forces, which need to be taken into the account. It is clear that the SFS will need large solar panels to produce enough energy for its cryogenic coolers. Thus, it can be expected a strong and complex flexible mode behavior. Another perturbation force is liquid sloshing due to the large propellant tanks that will compose of most of the SFS. Finally, it can be mentioned about other minor forces such as outgassing, leaks and thermal pressure on the radiators, all of which can be ignored. To determine convenient thruster, drag force to be handled with must be predicted.

The drag can be estimated roughly from the Eq.2;

$$||\vec{\boldsymbol{F}}_d||=0.5\rho \boldsymbol{V}^2 A \boldsymbol{c_d} \tag{2}$$

ρ : Density of the atmosphere at 800 km

V: Velocity of the SFS at 800 km

A : Cross section area of SFS

c_d: Drag coefficient

It is predicted that drag force will be in order of nearly 10^{-4} N. To cope with these forces, it is recommended that electrical thrusters might be used not only for station keeping maneuvers but also de-orbit method. In this study to exchange momentum and have correct attitude CMG (Control Momentum Gyro) is recommended because it has compact design, reasonable torque capacity and high performance as well as it is more agile when compared to traditional system like reaction wheels. However, it is costly and complex. Also, CMG has finite momentum capacity and can be saturated. In that point there is a need of actuators which are able to create a torque to desaturate the CMG. In this study, it was found appropriate to use magnetorquer instead of reaction control thruster. Because they need fuel, which is not reasonable in terms of cost. On the other hand, magnetorquer uses the Earth's magnetic field to produce a torque and requires only electricity.

In order to have attitude information and control the SFS, some sensors need to be defined. These sensors provide position, velocity and acceleration of the SFS. The measurement of the acceleration is essential for a spacecraft. To measure this value, the SFS will use IMU (Inertial Measurement Unit), composed of a set of accelerometers and gyro. Internally, there are (at least) three sensors of each type, one for each axis. A fourth can be included to increase the reliability of the system.

The SFS angular position is critical during the whole mission for several reasons. During docking and refueling/resupply, the SFS must point and hold to allow another customer satellite to dock without collision issues and to allow a proper refueling/resupply. This situation requires a precise angular position. Many sensors can be used to find the angular position, such as Earth and Sun sensors, or star tracker and gyroscopes. The main requirement is to design a robust system (reliability), while keeping the system simple and cheap. The most common design is to use a hybrid system with both gyroscope and star tracker sensors. Indeed, the gyroscope is very efficient to measure quick dynamics and the star tracker is more stable in time. The hybridization of these two sensors using Kalman filter will result on a robust hybrid system of measurement.

For safety, Earth and Sun sensors have also been advised in this study. Even though they are not operational during the whole orbit (eclipses), they are cheap and reliable, and will be used for safe modes.

Consequently, selected sensors and actuators have been given in the Figure 21.

	Sensors / Actuator	Nominal Number
Orbit Keeping	Electrical Thruster	2
Sensors	Star Tracker	1
	Sun Sensor	3
	Earh Sensor	2
	IMU	3
Actoators	CMG	4
	Magnetorquer	41/7

Figure 21: Selected sensors and actuators

CONCLUSION

Consequently, in this study, conceptual SFS, which is located in LEO, provides refueling to satellites in LEO and GEO-duty satellites, allowing the satellites to have a longer mission period and gain fuel mass in launch. Orbit selections were made according to the regions where satellites are concentrated with market analysis. In the technical analysis part of the SFS, the cycle of technical refueling is explained with various subsystem definitions. Paradigm operation plan and business case are out of scope of this study. However, various researches and studies can be done on these two subjects in order to contribute to the study in the future. It is envisaged that this study will shed light on national space programs of Turkey which will be conducted in future terms.

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