Delft University of Technology

Laser Swarm

Final Report

Design Synthesis Exercise

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Preface

This final document contains our evaluation of the Laser Swarm concept. Ten weeks have passed and we are finally ready to present our conclusions.

We did not, however, do everything by ourselves, and this is a good place to thank those who have been so kind as to give us some of their precious time.

First I want to mention professor Edoardo Charbon, who kindly explained and showed to us the workings of his Single Photon Avalanche Diode (SPAD)[1]: without his help we would be hopelessly in the dark on these fascinating devices. From this place I also want to thank professor Kourosh Khoshelham for helping us to some valuable material on remote sensing. I also want to thank Jasper Bouwmeester, who provided us with some helpful information on solar panel pricing.

And finally I thank our tutor, professor Ben Gorte, who was very generous to us with his time and has been very supportive of the entire project.

I hope you will have as much fun reading this report, as we had in making it!

Yours sincerely,

Co Florijn,
Chairman

[1] See [7, 98-105], [6] and [4]

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List of Symbols

Symbol	Description	SI units
A	accuracy	rad
A	area	m^2
a	semimajor axis	km
C_D	drag coefficient	–
D	air drag	N
e	eccentricity	–
$\dot{e}$	time rate of change of the eccentricity	s^{-1}
g	gravitational acceleration	m/s^2
I_{rt}	round-trip interactivity loss	Watt
I_{sp}	specific impulse	s
i	inclination	rad
i_R	relative inclination	rad
$\dot{i}$	time rate of change of the inclination	rad/s
J_{ii}	inertia tensor for i	m^4
J_2	second zonal coefficient	0.001082645 –
J_3	third zonal coefficient	0.000002546 –
M_C	torque about c	N/m
m	mass	kg
m_f	fuel mass	kg
m_p	payload mass	kg
m_0	dry mass	kg
N	angular seperation	rad
n	nr of energy levels	–
n	mean motion	$7.292 \cdot 10^{-5}$ rad/s
n_e	refractive index	–
P	power	Watt
$p_{p,th}$	threshold power	Watt
R	mass ratio	–
R_C	radius of the orbit of the center of mass	km
R_E	earth equitorial radius	6378 km
s	variable in the Laplace equation	–
T	temperature	K
T_i	period of i	s
V	velocity	m/s
V_a	voltage at a	V
V_B	voltage at B	V
W	weight	N
η	pump efficiency	–
θ_i	angle w.r.t. rotation axis i	rad
θ_{man}	manoeuvre angle	rad
λ	wavelength	m
μ	gravitational constant of the Earth	$398600 \cdot 10^9$ m^3/s^2
ν	true anomaly	rad
ρ	density	kg/m^3
s	stimulated emission cross-section	m^2
t	upper level lifetime	s
ϕ	phase angle	rad
ϕ_R	relative phase angle	

Ω	right ascension of the ascending node	rad
$\dot{\Omega}_{J_2}$	rate of precession	rad/s
ω	argument of perigee	rad
$\dot{\omega}$	time rate of change of the argument of perigee	rad/s

List of Acronyms

ELS Ensemble de Lancement Soyuz

EOL End-of-Life

GLAS Geoscience Laser Altimeter System

LiDAR Light Detection And Ranging

RAAN Right Ascension of the Ascending Node

SPAD Single Photon Avalanche Diode

Abstract

In February 2010 the ICESat mission ended after 7 years of measuring ice sheet mass balance, cloud and aerosol heights, as well as land topography and vegetation characteristics using a space based Light Detection And Ranging (LiDAR) system. ICESat used only one of the possible approaches for LiDAR, namely the use of a high energy laser and a large receiver telescope. The other approach is using a high frequency, low energy laser and a single photon detector. The advantage of the latter approach is that it has a much lower mass, but it is uncertain if even a single photon per pulse reaches the receiver. One possible solution could be to use a swarm of satellites around the emitter, each equipped with a single photon detector. However, the technical feasibility of this concept has not yet been proven.

This report briefly highlights different ways in which such a mission could be accomplished, and then decides what way will be the most likely to succeed.

Chapter 1

Introduction

In February 2010 the ICESat mission ended after 7 years of measuring ice sheet mass balance, cloud and aerosol heights, as well as land topography and vegetation characteristics. To do all this, ICESat had only one instrument on board: a space based LiDAR system (Geoscience Laser Altimeter System (GLAS)), allowing for an unprecedented 3D view of the Earth's surface and atmosphere. The laser lifetimes, however, were severely limited because of manufacturing errors in one of the laser components.

ICESat followed only one of the possible approaches for LiDAR, namely the use of a high energy laser and a large receiver telescope. The other approach is using a high frequency, low energy laser and a single photon detector. The advantage of the latter approach is that it has a much lower mass, but it is uncertain if even a single photon per pulse reaches the receiver. One possible solution could be the use of a swarm of satellites around the emitter, each equipped with a single photon detector. However the technical feasibility of this concept has not yet been proven.

In this mid term review some options for this concept are proposed and traded off so as to find the best fit for the mission. This tradeoff is what the bulk of this report is about. Also described are the workings and results from the simulator, as well as the project and risk management. Chapters on operation and logistics, stability and control and sustainability are included too.

Chapter 2

Launch and Astrodynamic Characteristics

This chapter explains the characteristics of the launch and space segments of the mission. Section 2.1 describes the three main characteristics of bringing the swarm into the orbit: launch vehicle, launch location and orbit insertion. Section 2.2 discusses all the aspects of the constellation and its configuration, and delves into estimations of ΔV and needed propellant. Finally, section 2.3 deals with the hazardous space environment and ways of protecting the satellites against it.

2.1 Launch Segment

The launch segment of the mission includes the selection of the launch vehicle, the launch location and the procedure of inserting the satellites into their respective orbits. The following subsections discuss all of these aspects.

2.1.1 Launch Vehicle

The costs of launch vary greatly between different vehicles. Table 2.1 on page 2 lists approximate total launch costs of respective platforms. The prices are given in Fiscal Year 2000 dollars for consistency.

Platform	Operator	Price (FY00$m)
Ariane V	ESA	97.96
Soyuz	Starsem or Arianespace	8.16 - 22.1
Vega	European and Italian SA	15.1
Falcon 1E	SPACEX	8.89
PSLV	ISRO	13.88 - 16.33
Rokot	Eurokot	9.8 - 11.4

Table 2.1: Estimated price comparison of different launch vehicles. *Source: various.*

Based on the information in the above table it is possible to single out a few platforms which will be affordable for the purpose of this feasibility study. The values represent the total launch segment costs. The Ariane V launcher is the most expensive option by far and would push the budget quite heavily, with an estimated cost of launch almost half of the total budget. However the payload capabilities of the Ariane V launcher far outweigh that of all other platforms, thus making it possible for a combined launch with other satellites, leading to shared costs. This however could jeopardize the mission in the sense that it becomes secondary priority. If that happens, the constellation would have have have higher requirements for orbit acquisition: an extra booster stage or higher onboard fuel capacity for the altitude and/or plane shift, which is not feasible. The Ariane V is therefore an unsuitable platform for this project.

The rest of the launchers can be analyzed with respect to reliability. All of the launch vehicles have been tested, with the exception of the Vega system, which is yet to make its maiden flight. The Vega is therefore not suitable for the analysis at this time. It is for this reason that the project will no longer consider this system at all. However, better data should be available in the near future which should allow for the Vega platform to be reevaluated.

The same goes for the Falcon - 1e system. The launcher is still under development. Furthermore, the predecessor of the 1e system is the Falcon - 1, which out of total of five launches only had two successful, fails to make a favorable impression.

Table 2.2 on page 3 shows some reliability statistics for the remaining 3 vehicles.

Platform	Total No. of Launches	Total Failures	Reliability	No. of Successful Launches Since Last Failure
Soyuz	1754	88	95%	57
PSLV	16	1	94%	15
Rokot	17	2	88%	6

Table 2.2: Reliability figures for several launch vehicles. *Source: various.*

The Soyuz launch vehicle presents itself as the most reliable platform, with a track record that far surpassed all other options. This launch vehicle will be the one considered for this project. In the next section, its payload capabilities are discussed.

2.1.1.1 Soyuz LV Payload Capability Analysis

There is a large selection of Soyuz vehicles available for consideration. For the purpose of this project, the newest modification - Soyuz-ST will be used. This vehicle is part of the Soyuz-2 family, which are technologically superior to the older Soyuz-U and U2 launchers. An illustration and technical parameters of the Soyuz-ST can be found in Appendix A on page 15 [1].

An estimation of mass performance for the launch vehicle can be seen in figure 2.1 on page 4.

The payload mass data provided in [1] is estimated, yet is good enough to have a reasonable idea about the maximum mass. The Soyuz-ST is able to launch roughly a maximum of 5000 kg into a 500km orbit. This is well above the design mass of the formation thus will allow for further considerations of joint launches (as long as the swarm mission is considered to be the primary payload) and thus spread launch costs.

The available volume in the Soyuz-ST Fairing can be seen in figure 2.2 on page 5.

The dimensions of the fairing are visibly too large and there is no possibility of using a different one, however that leaves a lot of possibilities for different designs of release adapters to adapt to the unique sequence of separation upon orbit injection. Again, the possibility of taking other small satellites along on

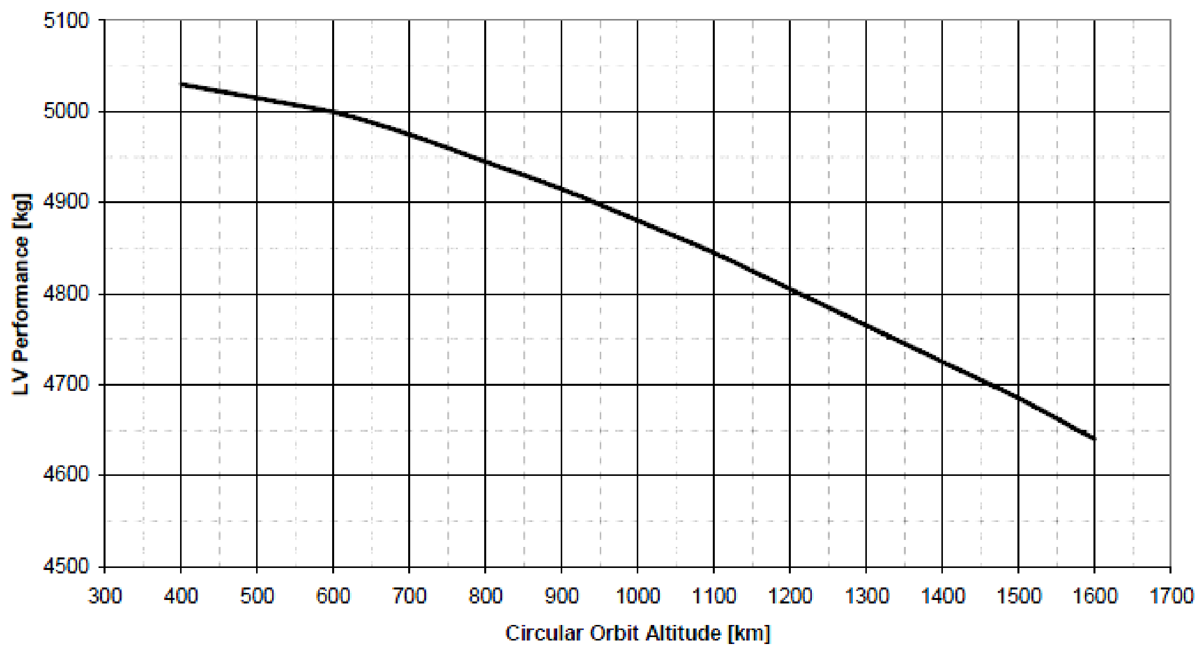

Figure 2.1: Mass performance of the Soyuz-ST for circular Orbits. *Source: [1].*

the same launch arises. The dual launch configuration shown in the previous figure is the one being considered for the launch.

The choice of the Soyuz brings forth one more advantage: the use of a unique orbit insertion booster, the Fregat. This final stage will allow for minimization of fuel on the satellite as all orbit insertion maneuvers can be done using the Fregat. The Fregat stage has been designed to handle 20 individual burns.

2.1.1.2 Vibrational Analysis

HERE ADD VIBRATION ANALYSIS

2.1.2 Launch Site

The selection of the launch site relies on several factors:

- Availability of attainable inclinations from launch.
- Compatibility with the launch vehicle.
- Accessibility and cost.
- Security and political reasons.

The first factor is crucial. It is paramount that the satellites are injected into their final inclinations at launch and do not have to perform any inclination change maneuvers, which require a substantial ΔV. With this in mind choosing a launch site closer to the equator is necessary. Launch sites at higher latitudes

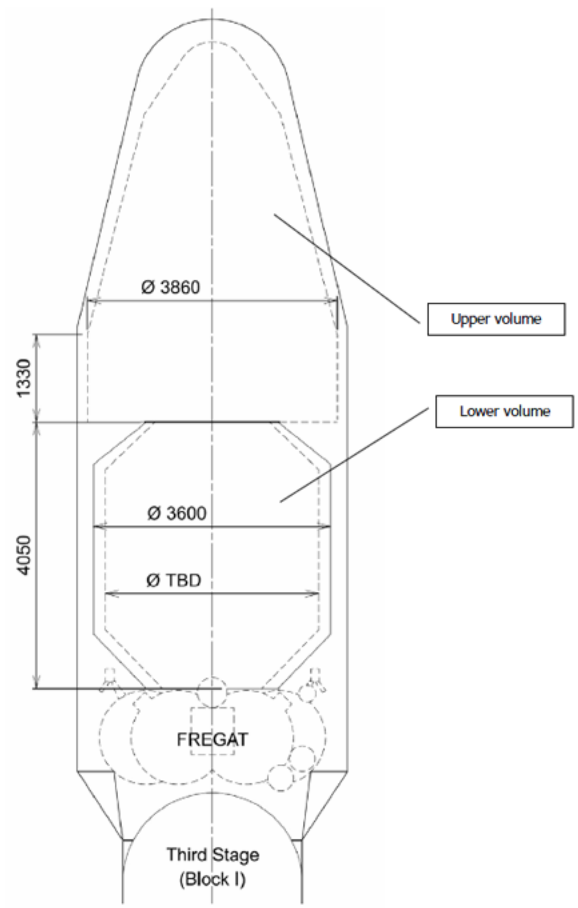

Figure 2.2: Fairing volume of the Soyuz-ST launch vehicle. Here shown in dual launch carrying configuration. *Source: [1].*

would need to sacrifice velocity and thus payload mass because of their location. Table 2.3 on page 5 shows a number of possible launch sites and their locations [9]. The list contains only the sites compatible with the Soyuz-ST launch vehicle. Furthermore, in figure 2.3 on page 6, the same sites are indicated with their respective authorized inclination ranges.

Launch Site	Operator	Latitude (deg min)	Longitude (deg min)
Baikonur LC-31/6	Russia (Starsem)	45 54 N	63 18 E
Plesetsk LC-43	Russia (Starsem)	62 48 N	40 24 E
Guiana Space Centre ELS	CNES/Arianespace	5 18 N	52 50 W

Table 2.3: Available launch sites for the Soyuz-ST. *Source: [9].*

Even though all the above sites allow inclinations of 85 degrees, it is still preferable to select a site closer to the equator in order to utilize the full effect of Earth's rotation and lower the launch costs. With this in mind, the Guiana Space Centre is selected to be the preferable location for launch.

Figure 2.3: Launch site locations and allowable inclinations for the Soyuz-ST. *Sources: [8] and [5].*

The Ensemble de Lancement Soyuz (ELS) in Kourou, Guiana is currently being finalized and should accommodate its first Soyuz launch in 2010 [2].

A typical launch profile of the Soyuz launch vehicle from Kourou is shown in figure 2.4 on page 7. The first three stages of the vehicle are used propel the payload and the Frigate booster to a circular orbit at around 200 km. After separation (stage 6 on the figure), the Fregat initiates the first orbit injection burn to bring the satellites to the appropriate orbits.

With this information it is possible to further calculate some important launch parameters:

- The inertial velocity of the launch site is given by:

$$V_L = (464.5)cosL \tag{2.1}$$

  where L is the site latitude. For the case of Kourou the inertial velocity is 462.51 m/s.

- The launch azimuth in inertial frame of reference is given by:

$$A_{Z_I} = arcsin(\frac{cosi}{cosL}) \tag{2.2}$$

  and is equal to 5.022 degrees for this launch.

- The launch azimuth corrected for the Earth's rotation, is given by:

$$A_Z = arctan(\frac{V_0 sinA_{Z_I} - V_{eq} cosL}{V_0 cosA_{Z_I}}) \tag{2.3}$$

  where V_0 is the orbital velocity reached by the launcher before separation and the first burn of Fregat upper stage (see above, around 7.784 km/s), V_{eq} is the velocity of the Earth's rotation at the equator - 464.5 m/s. Thus the corrected launch azimuth becomes 1.61 degrees.

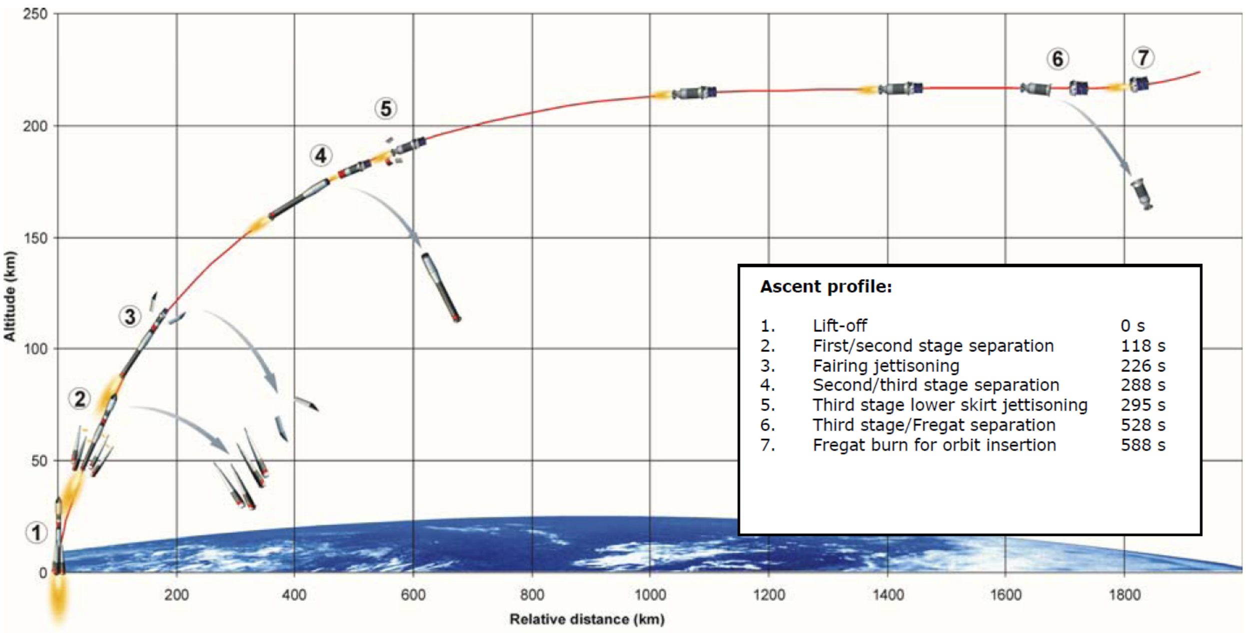

Figure 2.4: Launch profile of the Soyuz LV from Kourou. *Sources: [1].*

- The required launch velocity is calculated to be 7.76 km/s which means that due to Earth's rotation 26.8 m/s are saved.

Furthermore the region is politically stable and the launch site is under the protection from the French government and other security forces. All safety procedures are kept to the highest of standards and the spaceport is easily accessible by air or sea [1].

2.1.3 Orbit Insertion

The orbit insertion is separated into two distinct stages: primary orbits and secondary orbits. The primary orbits are located at an altitude of 500 km and contain the emitter and four initial receivers. The secondary orbits are located at a slightly different altitude of 550 (CHECK ALTITUDE!!!!) km and contain the auxiliary receivers intended for replenishment.

The configuration of the primary orbits is discussed in section 2.2.3 and an image of the formation can be seen in figure 2.6 on page 10. The release sequence and the important parameters are as follows (for satellite and orbit numbers please refer to figure 2.6):

1. The Fregat is injected in Orbit 1.
2. The ascending intersection of Orbit 1 and Orbit 2 is reached at a latitude of 85° and a longitude offset (from the ascending node of Orbit 1) of 88.91°. At this point the Fregat should be orientated in the direction of Orbit 2 and separate Rec 2. The separation ΔV that the adapter should produce is calculated using:

$$\Delta V = 2V_i sin\frac{\alpha}{2} \tag{2.4}$$

where V_i is the orbital velocity (7.612 km/s) and α is the relative inclination (2.17°, see section 2.2.3) [3]. The ΔV is calculated to be 289.63 m/s.

3. As the Fregat crosses the descending node and approaches the second plane intersection of the orbit, it does not need to change orientation (Orbit 3 intersects in the same direction on the descent phase as Orbit 2 does on the ascent phase). At the intersection Rec 1 is separated with a ΔV of 289.63 m/s.
4. After this the Fregat again aligns with the velocity vector of Orbit 1.
5. The Frigate should inject itself into a drift orbit with a negative drift rate of no less then 9.19 deg/orbit and re-injected back to Orbit 1 after traveling $90 + \Delta\phi/2 = 90.09505$ degrees. This will bring the launcher 2.3 degrees behind the final emitter position, or 0.12 degrees behind Rec 4.
6. At this point the remaining three satellites: Rec 3, Base and Rec 4 should be put into drift orbits by the attachment mechanism in order to acquire the 2.18 degree orbital separation. The exact order and timing can be designed and adjusted accordingly.

After this, the satellites can be considered to be in their orbits. The Fregat can start the burn to insert into the secondary orbits. The formation in the secondary orbit is the same as in the primary orbits minus the emitter, thus similar maneuvers have to be performed.

2.1.4 Launch Date

The launch date is dominated by development times and lifetime considerations. The satellite orbit decay is and thus lifetime is a function of atmospheric density. The density is, in turn, a function of the solar activity. The number of sun spots on the surface of the sun rises and falls every eleven years. The measurements are commonly represented in 10.7cm radio flux intensities. Figure 2.5 on page 9 presents a projection for the next 10 years.

In order to provide the longest possible lifetime for the mission it is essential that the launch is timed in such a way that the satellites are in orbit most of the time during solar minimum. For this reason a launch on March 1st 2017 is planned. This also gives enough time for development and production. For further information on the timeline, please refer to section **??**.

2.2 Space Segment

This section covers the astrodynamical characteristics of the mission. Emitter orbit is covered first, then in section 2.2.2, all receiver orbits are examined. The formation and its properties are discussed in section 2.2.3. Collision avoidance is described in section 2.2.4 and, finally, section 2.3 covers the orbital environment and whether the mission is going to be heavily affected by it.

2.2.1 Emitter Orbit

2.2.1.1 Orbital Parameters

The emitter satellite is injected into a circular orbit at an altitude of 500 km. The eccentricity is frozen at 0. Inclination is chosen to be 85 degrees, as this allows access to polar areas, which are places of interest for the mission. Furthermore, the inclination provides an inherent relative phase for the crossing orbits. The Right Ascension of the Ascending Node can be chosen arbitrary for the start of the mission, as no specific target is assumed for the mission. For consistency, in all following discussions it is assumed to be 0 degrees. The

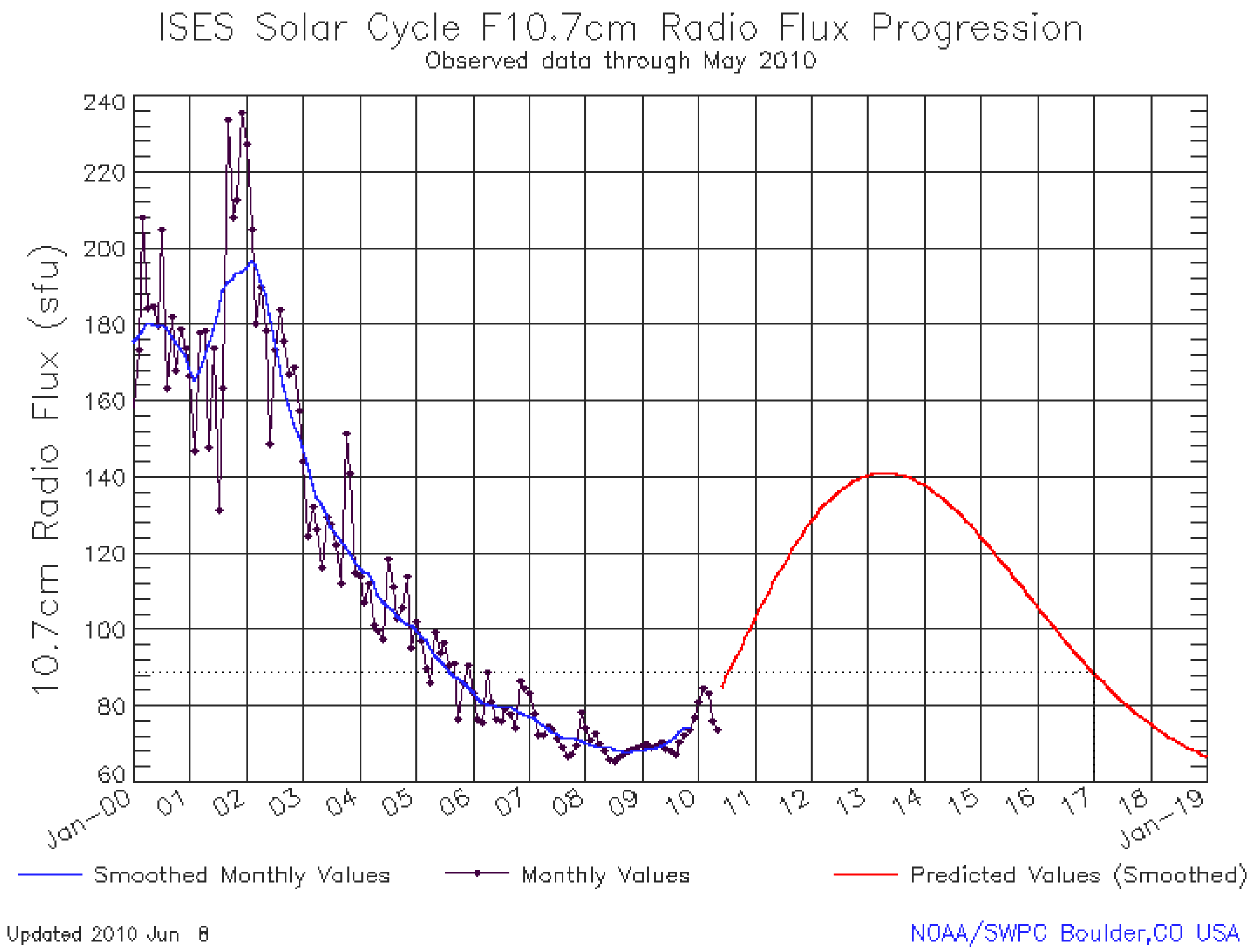

Figure 2.5: Solar activity projection up to 2019. *Source: NOAA/Space Weather Prediction Center.*

same goes for the Argument of Perigee and the True Anomaly. As the orbit is circular, these values are not really relevant. The emitter satellite acts as the reference for all the receivers.

The orbit is non-sunsynchronous and does not have a specific repeat track. This allows for larger coverage of the Earth.

Figure 2.6 on page 10 shows the emitter satellite (labeled as Base) in Orbit 1.

A few basic properties of this orbit can be derived and are shown in table 2.4 on page 10. All the values were generated with the help of [9], [3] and [8].

2.2.2 Receiver Orbits

b lss

Parameter	Symbol	Value	Unit
Altitude	h	500	[km]
Semi-major Axis	a	6871	[km]
Eccentricity	e	0	[-]
RAAN	Ω	0	[deg]
Period (mins)	P	94.6135	[mins]
Revolutions per day		15.2198	[revs/day]
Angular Velocity	n	3.805	[deg/min]
Circular Velocity	V	7.6127	[km/s]
Max. Eclipse	T_e	35.75	[mins]
Node Spacing		23.72	[deg]
Node Precession	$\dot{\Omega}$	-0.6667	[deg/day]

Table 2.4: Orbital properties of the emitter satellite.

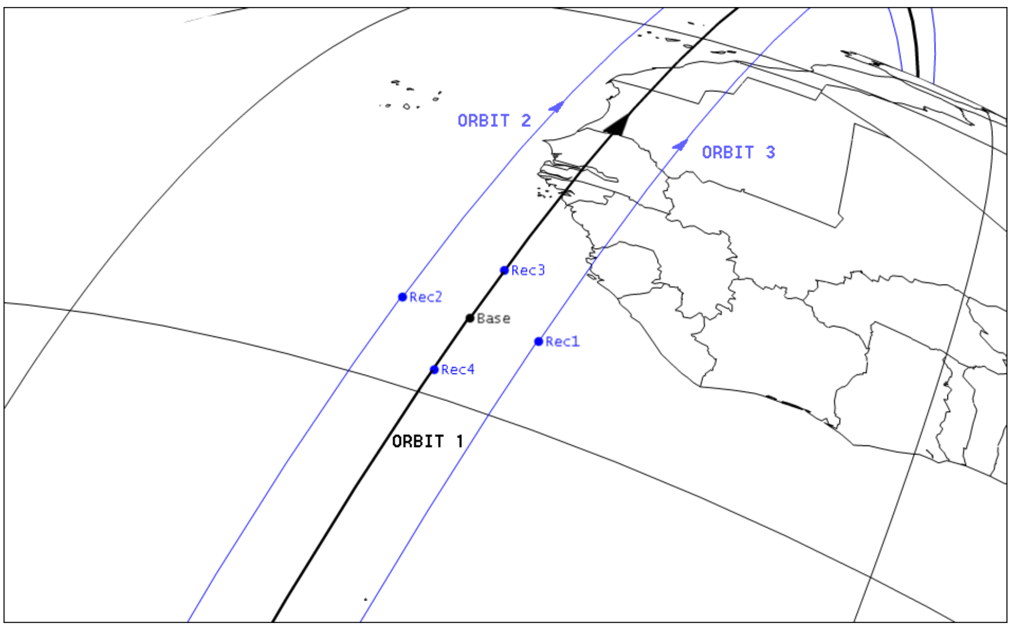

Figure 2.6: Swarm configuration as seen when the emitter (labeled here as Base) crosses its ascending node. Orbit numbers represent the different orbital planes.

2.2.3 Swarm Configuration

2.2.4 Collision Avoidance

bla

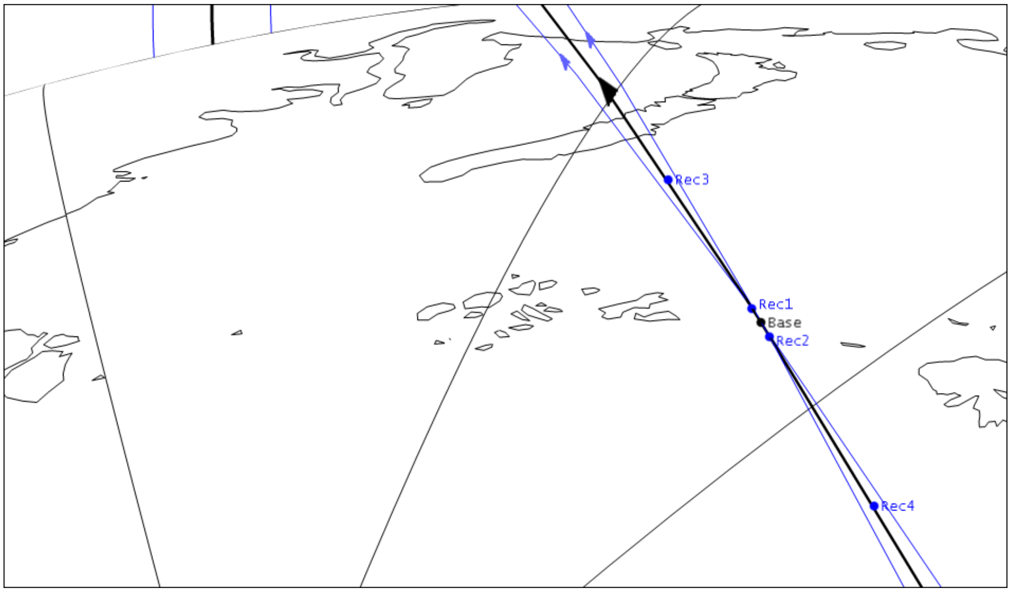

Figure 2.7: Swarm configuration as seen when the orbit planes intersect. In this figure the intersection in ascent is pictured.

2.3 Space Environment and Shielding

Bla

Chapter 3

Sustainable Development Strategy

In this chapter the Sustainable Development Strategy is discussed in the order of production (section 3.1), operations (section 3.2) and End-of-Life (EOL) (section 3.3).

3.1 Production and Logistics

The design is aimed at a swarm of mostly identical satellites. This may allow for series production which is more efficient in terms of resources than a one-of large satellite with a lot of unique components. This also implies that the number of different spare parts could be reduced. Smaller satellites could also use smaller facilities for production and testing.

Transportation can be split up into two parts: transportation to the launch site and the launch from the surface to the final orbit in space. On both occasions the system can again profit from its small size. If the satellites are not launched all together, they can "piggyback" on another satellite's launcher.

Spreading the swarm, i.e. piggybacking using different launchers, has several advantages. First of all, the emissions are lower than in case of a dedicated launcher. Also, if the first satellite fails before the launch of the rest of the swarm, the others can be repaired and thus less resources are wasted.

3.2 Operations

Once in orbit, the satellite's influence on the Earth is very limited. The only real concern is the debris it leaves behind during launch and deployment, which can be dangerous to other satellites orbiting the Earth. However, the deployment mechanism, which is responsible for most of the debris, is not included in this technical feasibility study. Later studies developing the ideas from this feasibility study should take it into account, since more satellites could mean more deployment mechanisms and hence more waste. One aspect that can be dealt with is the efficient use of resources. The swarm can be designed in such a way that if one of the satellites fails a replacement satellite can be sent, whilst any remaining satellites can be reused.

3.3 End of Life

Each satellite will be at the end of its life if it cannot perform its function anymore. It is important that after the mission is over all satellites are removed from their orbit and burn up in the atmosphere so that they do not pose any danger to other satellites. Final decommissioning of the swarm will be more complex than for a regular satellite, since every individual satellite has to be decommissioned separately.

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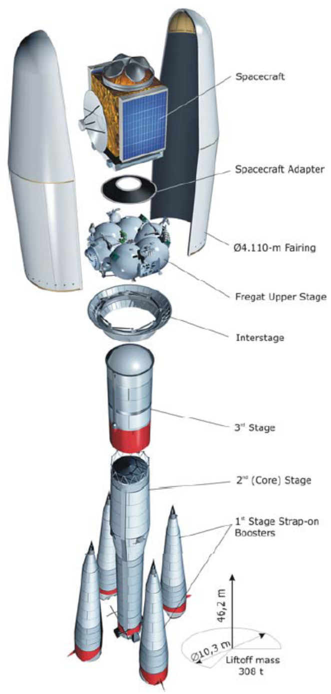
Appendix A

Soyuz-ST Factsheet



PAYLOAD FAIRINGS

Fairing	**ST**
Diameter:	4.110 m
Length:	11.433 m
Mass:	1700 kg
Structure:	Two-half-shell carbon-fiber reinforced plastic
Separation	Mechanical locks/ pneumatic jack/pushers
Interstage	
Mass:	400 kg
Structure:	aluminum skin-stringer

PAYLOAD ADAPTERS

Off-the-shelf devices:		
	1194SF	(110 kg);
	937SF	(45 kg);
	1666SF	(100 kg)

FREGAT UPPER STAGE

Size:	3.35-m diameter × 1.50-m height
Inert mass:	950 kg
Propellant:	5350-kg N_2O_4/UDMH
Subsystems:	
Structure:	Structurally stable aluminum alloy 6 spherical tanks/8 cross rods
Propulsion	S5.92
- Thrust	Two mode thrust 19.85/14.00 kN - Vac
- Isp	Two mode thrust 331/316 s - Vac
- Feed system	Pump-fed, open cycle gas generator
- Pressurization	Ghe vaporization
- Burn time / Restart	Up to 900 s / up to 20 controled or depletion burn
Attitude Control	
- pitch, yaw	Main engine translation or eight 50-N hydrazine thrusters
- roll	Four 50-N hydrazine thrusters
Avionics	Inertial 3-axis platform, on-board computer, TM & RF systems, Power
Stage separation:	gas pressure locks/pushers

.	1st STAGE (FOUR BOOSTERS)	2nd STAGE (CORE)	3rd STAGE	
Size:	2.68-m diameter × 19.60-m length	2.95-m diameter × 27.10-m length	2.66-m diameter × 6.70-m length	
Gross/Dry mass:	44 413 kg / 3 784 kg	99 765 kg / 6 545 kg	27 755 kg / 2 355 kg	
Propellant:	27 900-kg LOX 11 260-kg Kerosene	63 800-kg LOX 26 300-kg Kerosene	17 800-kg LOX 7 600 kg Kerosene	
Subsystems:				
Structure	Pressure stabilized aluminium alloy tanks with intertanks skin structure	Pressure stabilized aluminum alloy tanks with intertanks skin structure	Pressure stabilized aluminum alloy tanks with intertanks and rear skin structure	
Propulsion	RD-107A 4-chambers engine,	RD-108A 4-chambers engine,	RD-0110 4-chamber engine (Soyuz 2-1a)	RD-0124 4-chamber engine (Soyuz 2-1b)
- Thrust	838.5 kN – SL; 1021.3 kN –Vac	792.5 kN – SL; 990.2 kN –Vac	297.9 kN (Vac)	297.9 kN (Vac)
- Isp	262 s – SL; 319 s –Vac	255 s – SL; 319 s –Vac	325 s -Vac	359 s (Vac)
- Feed system	pump-fed by hydrogen peroxide (H_2O_2) gas generator	pump-fed by hydrogen peroxide (H_2O_2) gas generator	Pump-fed gas generator, generator's gas blow down through verniers	Multi-stage pump-fed close cycle gas generator
- Pressurization	Liquid nitrogen (N_2)vaporization	Liquid nitrogen (N_2) vaporization	Oxygen vaporization/generator gases	Helium vaporization
- Burn time / Restart	118 s / No – two level thrust throttling	286 s / No – one level thrust throttling	250 s / No	270 s / No
Attitude Control	Two 35-kN vernier thrusters and one aerofin	Four 35-kN vernier thrusters	Four 6-kN vernier thrusters	Each chambers gimbaling in one axis
Avionics	Input/Output units, TM, power	Input/Output units, TM, power	Centralized control system: inertial 3-axis platform, on-board computer, TM & RF system, power	
Stage separation:	Pyronuts/pushers/reaction nozzle	Pyronuts and 3rd stage engine ignition		