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 [4, 98-105], [3] and [2]

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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

EOL End-of-Life

GLAS Geoscience Laser Altimeter System

LiDAR Light Detection And Ranging

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

HERE ADD VIBRATION ANALYSIS

2.1.1.1 Cost and Reliability Analysis

The costs of launch vary greatly between different vehicles. Table 2.1 on page 3 lists approximate total launch costs of respective platforms. The prices are given in Fiscal Year 2000 dollars for consistency.

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 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	Operator	Price (FY00$m)
Ariane V	ESA	97.96
Soyuz	Starsem	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.*

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.

2.1.1.2 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 10 [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.

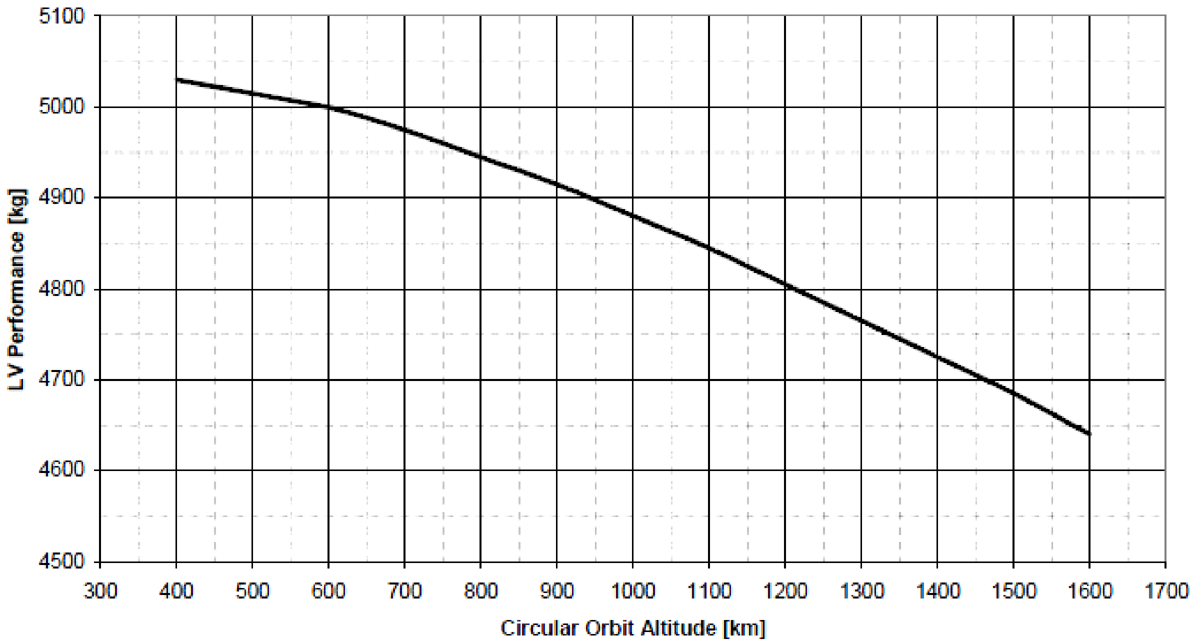

Figure 2.1: Mass performance of the Soyuz-ST for circular Orbits. *Source: [1].*

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 the same launch arises.

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 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 [5]. 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.

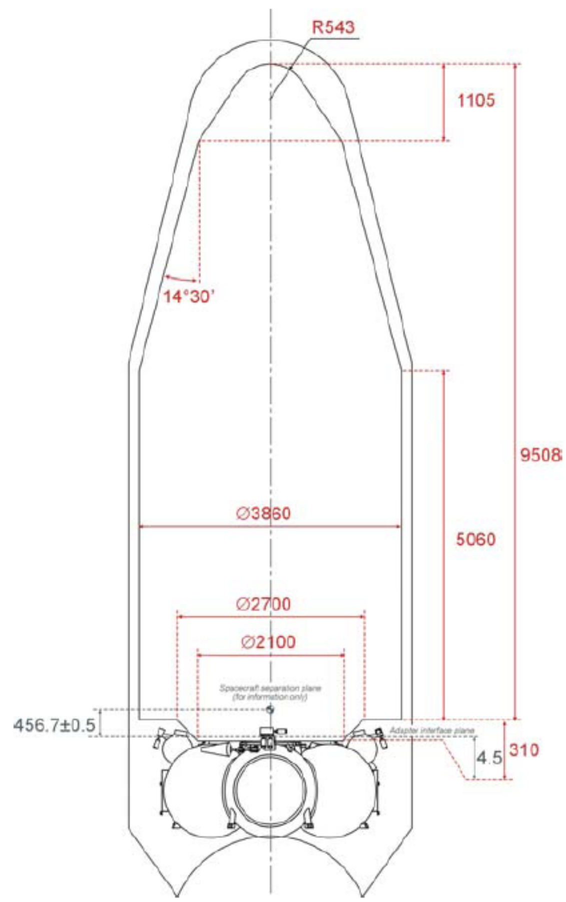

Figure 2.2: Fairing volume of the Soyuz-ST launch vehicle. *Source: [1].*

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: [5].*

2.1.3 Orbit Acquisition

Once the final stage of the launch vehicle has reached desired orbit altitude and inclination, preparations can start for proper separation maneuvers. This maneuver has to be designed in such a way as to accommodate all phase shifts required by the satellite orbits. Using the

Figure 2.3: Launch site locations and allowable inclinations for the Soyuz-ST.

2.2 Space Segment

2.2.1 Emitter Orbit

bla

2.2.2 Receiver Orbits

b lss

2.2.3 Collision Avoidance

bla

2.2.4 Stationkeeping

bla

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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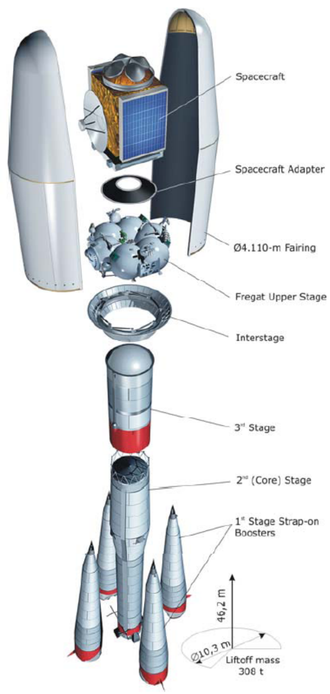
Appendix A

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		