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Abbreviated terms

ADCS	Attitude Determination and Control System
AOP	Argument of Perigee
AOS	Acquisition Of Signal
AR	Acceptance Review
CCP	Cumulated Collision probability
CNES	Centre national d'études spatiales (French Space Agency)
COMM	Communication subsystem
DCP	Daily Collision Probability
DT	Deployment Team
ECC	Eccentricity
EOL	End of Life
EM	Engineering Model
EPS	Electrical Power System
ESA	European Space Agency
ESEO	European Students Earth Orbiter
FM	Flight Model
FRR	Flight Readiness Review
GS	Ground Station
IADC	Inter-agency space debris coordination committee
INC	Inclination
LEO	Low Earth Orbit
LOS	Loss Of Signal
LTAN	Local Time of Ascending Node
MA	Mission Analysis
MDR	Mission Definition Review
PDR	Preliminary Design Review
RAAN	Right Ascension of the Ascending Node
SC	Spacecraft
SKA	Studenckie Koło Astronautyczne (Students' Space Association of WUT)
SMA	Semi-Major Axis
SSO	Sun-Synchronous Orbit
SW	Software
TBC	To Be Continued
TBD	To Be Defined
TBP	To Be Provided
TCS	Thermal Control System
WUT	Warsaw University of Technology

1 INTRODUCTION

1.1 PURPOSE AND SCOPE

The objective of the Mission Analysis Report is to provide all of the necessary information on the mission characteristics, in particular on orbit of the satellite in sufficient detail for the planning and execution of mission operations.

This document describes the activities of the Mission Analysis team of PW-Sat2 satellite project during phase C. As during phase C launch provider has been chosen for the mission and the injection orbit is already known this study concentrate on the characteristics of the mission specific for the chosen orbit alone.

1.2 DOCUMENT STRUCTURE

The document is structured as follows:

- **Chapter 1** contains an introduction to the document
- **Chapter 2** provides the applicable mission requirements and constraints
- **Chapter 3** provides the description of the mission
- **Chapter 4** provides the description of the tools, models and assumptions applicable for the analyses in this document
- **Chapter 5** provides the analyses conducted for commissioning and operational phases
- **Chapter 6** provides the analyses conducted for de-orbit phase
- **Chapter 7** provides the results of the PW-Sat2 sail effectiveness
- **Appendix A** provides the objectives of the PW-Sat2 Mission Analysis team and the brief summary of the activities conducted in the phase B
- **Appendix B** provides the brief description of the launch opportunity selection for PW-Sat2
- **Appendix C** provides the analyses for the simulation parameters influence on the orbital lifetime.
- **Appendix D** provides complete AOS/LOS table for nominal mission duration

1.3 PROJECT DOCUMENTATION STRUCTURE

See §1.3 in [PW-Sat2-C-00.00-Overview-CDR].

1.4 REFERENCE DOCUMENTS

Following documents are referenced throughout the text but are not part of internal project documentation:

[1] Inter-Agency Space Debris Coordination Committee, „IADC Space Debris Mitigation Guidelines," 2007.

[2] ESA, Director General's Office, "Space Debris Mitigation Policy for Agency Projects," 28 03 2014. [Online]. Available: http://www.iadc-online.org/References/Docu/admin-ipol-2014-002e.pdf. [Accessed 29 11 2016].

[3] NASA, "NASA Procedural Requirements for Limiting Orbital Debris, NPR 8715.6A," 19 02 2008. [Online]. Available: http://www.iadc-online.org/References/Docu/NPR_8715.6A.pdf. [Accessed 29 11 2016].

[4] French Ministry for Education and Research, "Technical Regulations of the French Space Act (in French)," 31 05 2011. [Online]. Available: http://www.iadc-online.org/References/Docu/Technical%20Regulations%20of%20the%20French%20Space%20Act20110331.pdf. [Accessed 29 11 2016].

[5] J. Soronsen, "Terra Bella And Spaceflight Industries Sign Agreement For Falcon 9 Launch For Small Imaging Satellites," Spaceflight, 11 10 2016. [Online]. Available: http://www.spaceflight.com/terra-bella-spaceflight-industries-sign-agreement-falcon-9-launch-small-imaging-satellites/. [Accessed 20 11 2016].

[6] CNES, "STELA User Guide v3.0," 11 2015. [Online]. Available: logiciels.cnes.fr/sites/default/files/Stela-User-Manual_3.pdf. [Accessed 29 11 2016].

[7] Space Exploration Technologies Corp, Falcon 9 Launch Vehicle - Payload's User Guide, 2015.

[8] T. Grimwood, „The UCS Satellite Database v. 1.16," Cambridge: Union of Concerned Scientists, 2016.

[9] Warsaw University of Technology – Faculty of Power and Aeronautical Engineering, "TERMS OF REFERENCES in open tender for „Launching the PW-Sat2 satellite into orbit"," 27 07 2016. [Online]. Available: https://www.meil.pw.edu.pl/content/download/31255/163215/file/32-1131-2016%20SIWZ_DO%20OG%C5%81OSZENIA.doc. [Accessed 30 11 2016].

[10] NASA, „Spacecraft aerodynamic torques - Space vehicle design criteria /guidance and control/," NASA, Washington, DC, United States, Jan 01, 1971.

1.5 Document Contributors

This document and any results described were prepared solely by PW-Sat2 project team members.

2 Mission Requirements and Constraints

PW-Sat2 is a student project. Students came up with the idea of the satellite and what mission it shall serve. As there was no customer there were never strict requirements imposed on the team from above. The requirements listed below come from within the team and are results of the design getting more matured with time.

2.1 Mission Requirements

2.1.1 Mission Definition

The main mission objective of the PW-Sat2 is to **test the concept of the new design of the deorbiting sail on a relevant and appropriate orbit**. Relevant orbit in this context is understood as an orbit on which future satellites potentially using this sail design may be placed, so the orbit populated by significant number of operational satellites. Drag sail technology is a potential method for the LEO satellites to comply with the IADC *Space Debris Mitigation Guidelines* [1]

> *A spacecraft or orbital stage should be left in an orbit in which, using an accepted nominal projection for solar activity, atmospheric drag will limit the orbital lifetime after completion of operations. A study on the effect of post-mission orbital lifetime limitation on collision rate and debris population growth has been performed by the IADC. This IADC and some other studies and a number of existing national guidelines have found 25 years to be a reasonable and appropriate lifetime limit.* (paragraph 5.3.2)

Similar rules apply to ESA, NASA and CNES missions [2] [3] [4].

The secondary mission objective of the PW-Sat2 is to test the new design of the Sun sensor device. Tertiary mission objective is to test other subsystems developed in-house for this project i.e. solar array deployment system (custom hinges) and release mechanism, Electrical Power System, ADCS detumbling and Sun-pointing algorithms, and mechanical structure. Currently, anticipated mission duration before the Sail opening is set to maximum 40 days (see [PW-Sat2-C-00.00-CDR-Overview] §4 – Mission Overview). Additional activities are expected provided good link with GS after Sail deployment.

2.1.2 Primary Mission Requirements

To test the de-orbit sail device the following high-level requirements shall be met from mission analysis point of view:

- Sail shall be proved effective in a relevant environment in which it would be used in the future missions.
    - Sail shall be deployed on the orbit inside the protected region A of the IADC space debris mitigation guidelines

- Orbit altitude shall be high enough to justify the use of the additional de-orbit device – high enough for the satellites to have natural orbital lifetime longer than 25 years.
- Sail shall be deployed on the orbit similar to those which are popular for the potential spacecraft that might utilize the sail device in the future.

The analysis of the relevant orbits has been performed and is presented in §7 in Figure 7-1 and Figure 7-2.

2.1.3 SECONDARY MISSION REQUIREMENTS

There are no specific requirements to be met from the mission analysis point of view considering the secondary mission objective.

2.1.4 TERTIARY MISSION REQUIREMENTS

There are no specific requirements to be met from the mission analysis point of view considering the tertiary mission objective.

2.2 PLATFORM AND PAYLOAD DESIGN

PW-Sat2 platform is a **2U CubeSat** with **2.6 kg** mass and basic dimensions of **10x10x26 mm**. PW-Sat2 will have two deployable solar panels, the size of the side wall of the 2U CubeSat hinged along the longer wall of CubeSat.

To the following high-level requirements derived from the payload and platform design shall be met from mission analysis point of view:

- The mission shall operate in the radiation environment allowing for the COTS components to work reliably through the whole mission duration

2.3 SELECTED LAUNCHER AND ORBIT

Launch opportunity selected for the PW-Sat2 is the piggy-back launch with Falcon 9, from Vandenberg Air Force Base in California, US, with the launch date of **December 2017** (TBC) for the **SSO, circular orbit of 575 km altitude and LTAN of 10:30**.

Selected launch opportunity is provided to PW-Sat2 team by the ISL (Innovative Space Logistics B.V.) in cooperation with Spaceflight Industries. Spaceflight Industries is in charge of the all payloads launch on that Falcon 9 rocket in the program called “dedicated rideshare”. It has been announced that the “co-lead” of this launch will be multiple imaging satellites from Terra Bella company (TBC). This information has to be yet officially confirmed by the Launch Provider [5].

Reference launch date used throughout this document is **6th December 2017, 00:00:00.000.**

2.4 GROUND STATION NETWORK

PW-Sat2 primary ground station is located in Warsaw, Poland in the Electronic Faculty of the Warsaw University of Technology. Detailed description of the ground station can be found in [PW-Sat2-C-02.00-COMM-CDR]. It is planned to encourage the radio amateur society around the world and in Poland in particular to receive the telemetry form the PW-Sat2 during its mission and provide it to the PW-Sat2 Operations team, however only the primary ground station would be able to send telecommands to the satellite. Only the primary GS is considered in the coverage analysis as the worst case. Parameters of the WUT ground station as are assumed throughout the document are presented in Table 2-1.

Table 2-1 WUT Ground Station parameters as assumed in analyses

GS feature	Value
Latitude	52.2188°
Longitude	21.0107°
Altitude	114 m
Height above ground	20 m
Min. Elevation	30°

3 MISSION OVERVIEW

Total mission duration before the sail opening has been set to the maximum of 40 days. Short mission duration is considered as required due to the increasing risk of subsystems' failure, especially electronic subsystems utilizing COTS components are of major concern. After the P-POD separation a 30 minutes period of communication silence is required. Only after that the communication module is initialized and antennas are retracted. During this period the system tries to perform Detumbling. After specified period the solar panels shall be opened provided correct operation of the ADCS. Later, a nominal experiments stage begins that lasts until the de-orbit system initialization. If all subsystems will work nominally the extended mission is planned to perform further tests of the Sun Sensor experiment. It can be shortened in case of problems with power supply or any other problems as the sail opening and technology test is a primary mission objective.

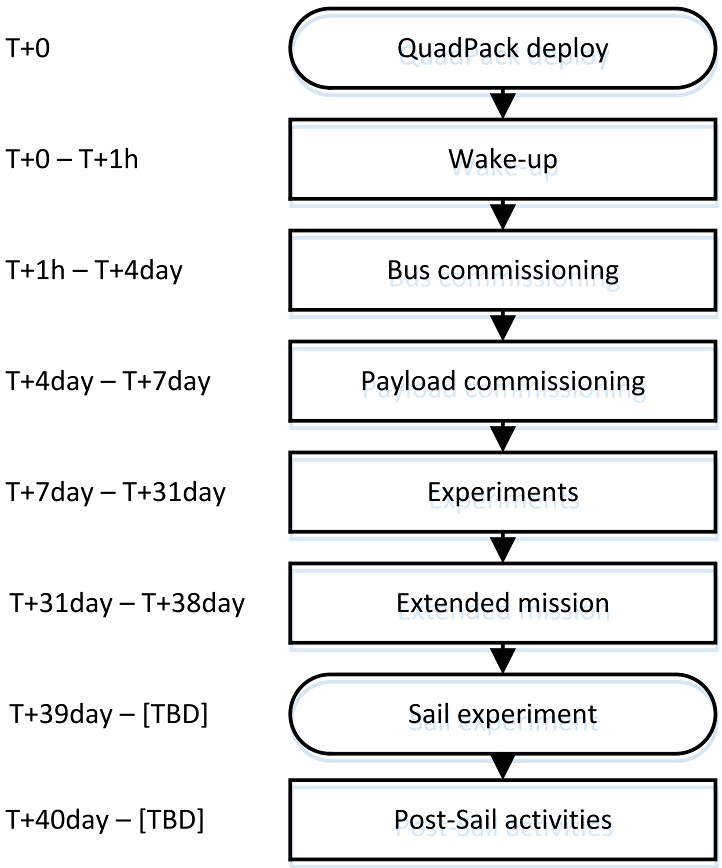

Figure 3-1 Top-level mission plan diagram (2016-11-29)

4 Analysis Tools, Models and Assumptions

4.1 Tools Used for the Analysis

Description of the software tools used for the analysis.

4.1.1 STK

STK – Systems Tool Kit is a software package developed by Analytical Graphics Inc. (AGI). It allows performing complex analyses of satellite, plane, ship and cars missions. The package is extensively used throughout the space industry worldwide. Including examples like ESA's Galileo In-Orbit Test Operational Planning project or Lunar Transfers with Four-Body Dynamics project by ESA-ESTEC.

Description from the producer website (https://www.agi.com/products/stk/):

> *Systems Tool Kit (STK) is the foundation of AGI's product line. This highly capable, free modelling environment is used by thousands of engineers, mission analysts, and software developers to model complex systems—such as aircraft, missiles, satellites and their sensors—analyse mission simulations and visualize dynamic datasets in 4D (X,Y,Z,Time).*

The basic license is available for free for non-commercial use. PW-Sat2 ground stations analysis is simple, due to only one existing ground station and very short mission duration. Overall capabilities of STK greatly exceed PW-Sat2 mission analysis needs. However, the free license has its limitations, e.g. choice of orbital propagator is limited to J2, J4, and SGP4.

4.1.2 GMAT

GMAT is a NASA tool in a development phase to become their main, versatile tool for mission analysis. It is licensed on NASA Open Source Agreement v1.3. GMAT in a current version (2016a) already includes advanced perturbations modelling which made it useful for MA orbital decay predictions. GMAT uses direct numerical integration of equations of motion and because of that it is slower than semi-analytical tools and as such less feasible for e.g. long term orbital decay simulations or Monte Carlo simulations. GMAT has been used for orbital parameters evolution analysis, before the sail opening and afterwards.

Description from the developer's website (http://gmatcentral.org/display/GW/GMAT+Wiki+Home):

> *GMAT is designed to model, optimize, and estimate spacecraft trajectories in flight regimes ranging from low Earth orbit to lunar applications, interplanetary trajectories, and other deep space missions. Analysts model space missions in GMAT by first creating resources such as spacecraft, propagators, estimators, and optimizers.*

4.1.3 STELA

Semi-analytic Tool for End of Life Analysis has been designed by CNES to support the French Space Operations Act [6]. Its interface and functionality is focused on analyzing if the given satellite breaches the EOL requirements imposed by the French Space Act. One of the possible analyses to be performed by STELA is the LEO orbit degradation analysis. Tool allows, also, performing Monte Carlo analysis for the unstable orbits running the simulation multiple times with a user-defined spread of the selected parameters. STELA is a semi-analytic tool which means that it can run much faster than tools which numerically integrate the equations. This makes STELA especially useful for long-term orbital decay analyses. PW-Sat2 MA team used STELA for orbital decay analyses described in §6.

4.1.4 DRAMA/MASTER

DRAMA and MASTER are the tools developed under ESA contract for space debris-related analyses, such as collision avoidance maneuvers estimation, impact flux analyses, de-orbiting strategy analysis etc.

Description from the ESA website:

> *MASTER (Meteoroid and Space Debris Terrestrial Environment Reference) allows assessing the debris or meteoroid flux imparted on a spacecraft on an arbitrary earth orbit. MASTER also provides the necessary computational and data reference for DRAMA and needs to be installed before DRAMA is installed.*
>
> *DRAMA (Debris Risk Assessment and Mitigation Analysis) is a comprehensive tool for the compliance analysis of a space mission with space debris mitigation standards. For a given space mission, DRAMA allows analysis of:*

- *Debris and meteoroid impact flux levels (at user-defined size regimes)*
- *Collision avoidance maneuver frequencies for a given spacecraft and a project-specific accepted risk level*
- *Re-orbit and de-orbit fuel requirements for a given initial orbit and disposal scenario*
- *Geometric cross-section computations*
- *Re-entry survival predictions for a given object of user-defined components*
- *The associated risk on ground for at the resulting impact ground swath*

MASTER annual impact flux simulations were used for the cumulative collision probability analyses performed for PW-Sat2.

4.2 Models Used in Analyses

4.2.1 PW-Sat2 Geometrical Model for Drag and SRP Simulations

Obviously, one of the most influential parameters in orbital lifetime analyses for LEO is drag area. In case of satellites with big, flat structures (as in case of PW-Sat2) the drag area is strongly dependent on the SC orientation w.r.t the velocity vector. For SCs with very low mass to area ratio on LEO orbits, like PW-Sat2 the main orientation perturbation is drag torque (see Figure 4-1). Precise analysis of drag torque is very complicated and requires detailed information on SC surface properties which are not available without comprehensive material properties research, especially during the SC design phase. Therefore, mean drag area was calculated for a set of different orientations of SC w.r.t. velocity vector and additional analyses were conducted for different values of mean area. STELA Mean Area Tool allows to build a simple 3D model of the satellite and to decide on the SC attitude w.r.t velocity vector in one of the three modes:

- Random Tumbling
- Spin (with user-defined spin axis)
- Fixed Orientation

3D model of PW-Sat2 with sail deployed is presented on Figure 4-2, Figure 4-3 and Figure 4-4.

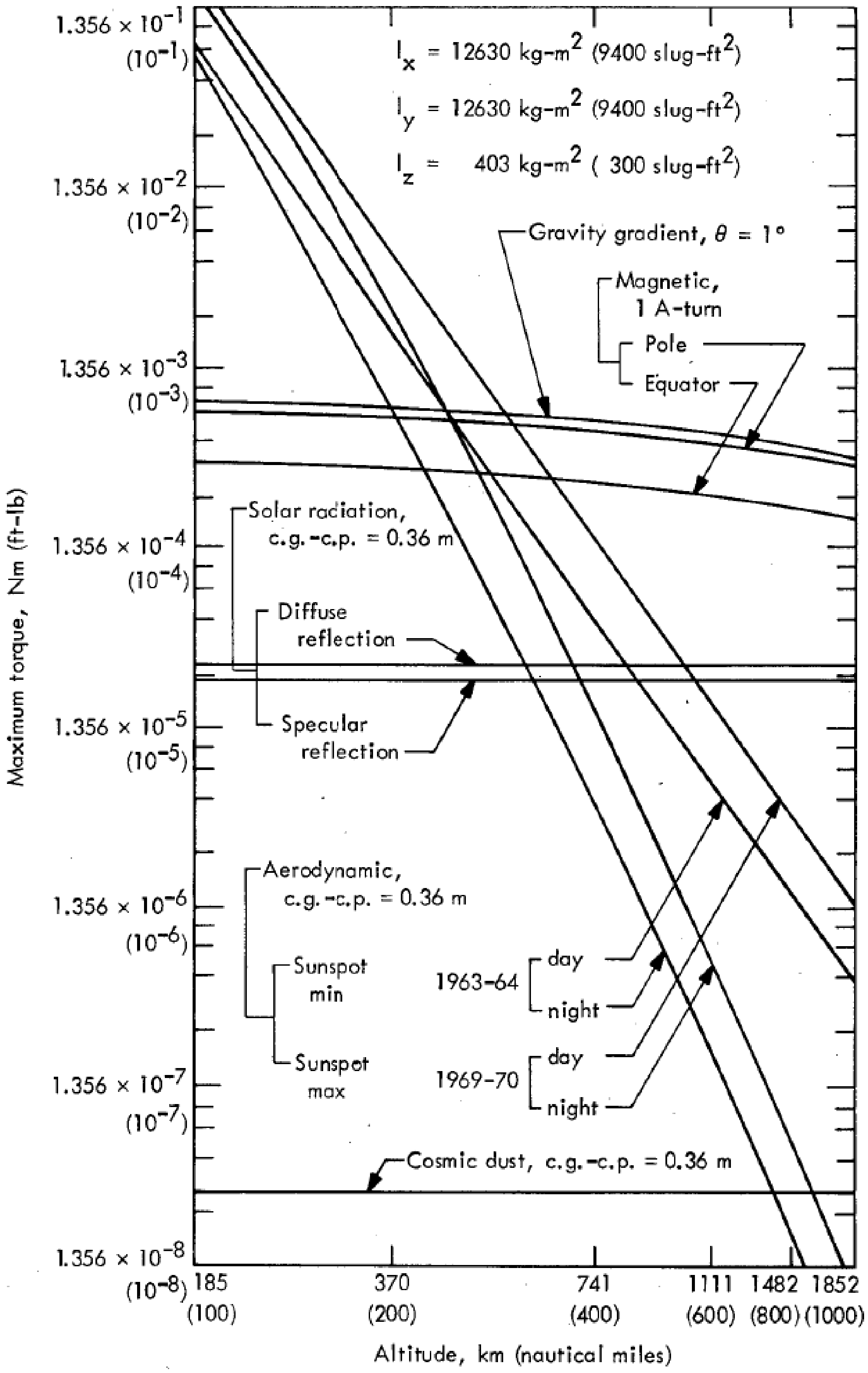

Figure 4-1 Relative magnitudes of the environmental torques on an Earth satellite [7]

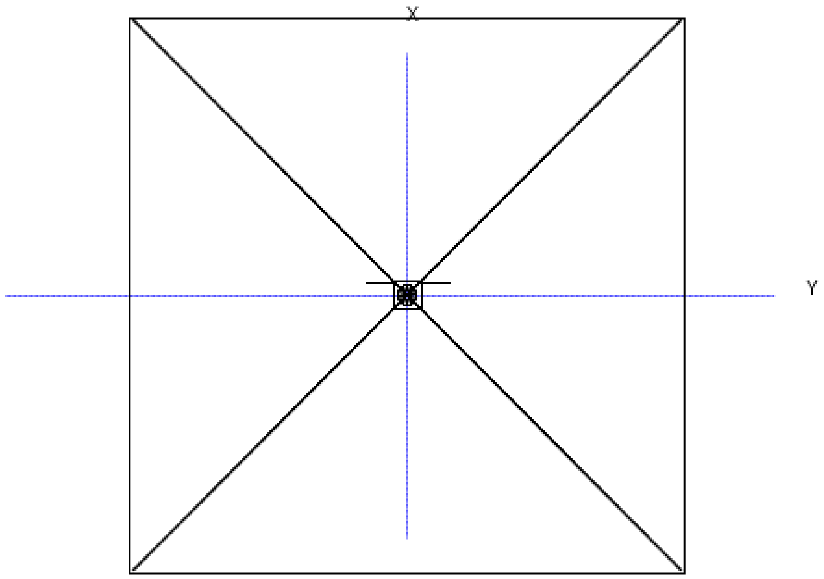

Figure 4-2 PW-Sat2 3D model created in STELA Mean Area Tool, XY-view

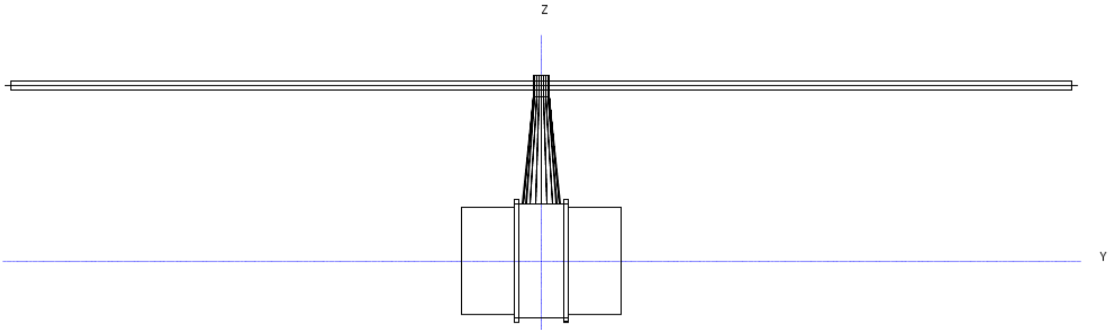

Figure 4-3 PW-Sat2 3D model created in STELA Mean Area Tool, ZY-view

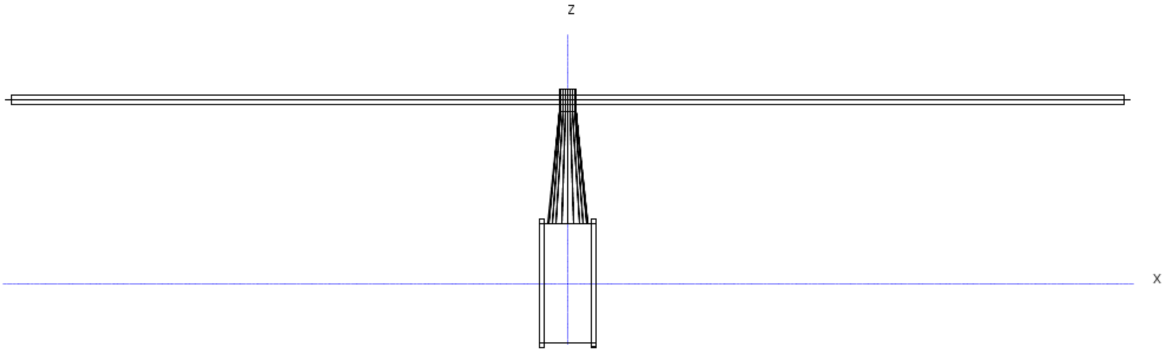

Figure 4-4 PW-Sat2 3D model created in STELA Mean Area Tool, ZX-view

Calculations were conducted for 3 configurations (CubeSat 2U, CubeSat 2U with deployed solar arrays, PW-Sat2 with deployed sail and solar arrays) for several orientations:

a) Random tumbling
b) Fixed orientation, velocity aligned with X-axis
c) Fixed orientation, velocity aligned with Y-axis
d) Fixed orientation, velocity aligned with Z-axis
e) Spin around axis at angle (5˚, 10˚, 15˚, 30˚, 45˚, 60˚, 90˚ -rotation around X-axis) to Z-axis, velocity aligned with Z-axis, (only for deployed sail configuration), see d).
f) Spin around axis at angle (2.5˚) to Z-axis, velocity aligned with X-axis.

Note that spin around axis at 45˚ to Z-axis with velocity aligned with Z-axis means that the satellite oscillate between positions in which sail is perpendicular and parallel to the velocity vector. Case f) is the oscillation of sail plane w.r.t velocity vector from 0˚ (sail plane parallel to velocity vector) to 5˚.

Spin axis orientation in case f) is showed on Figure 4-5:

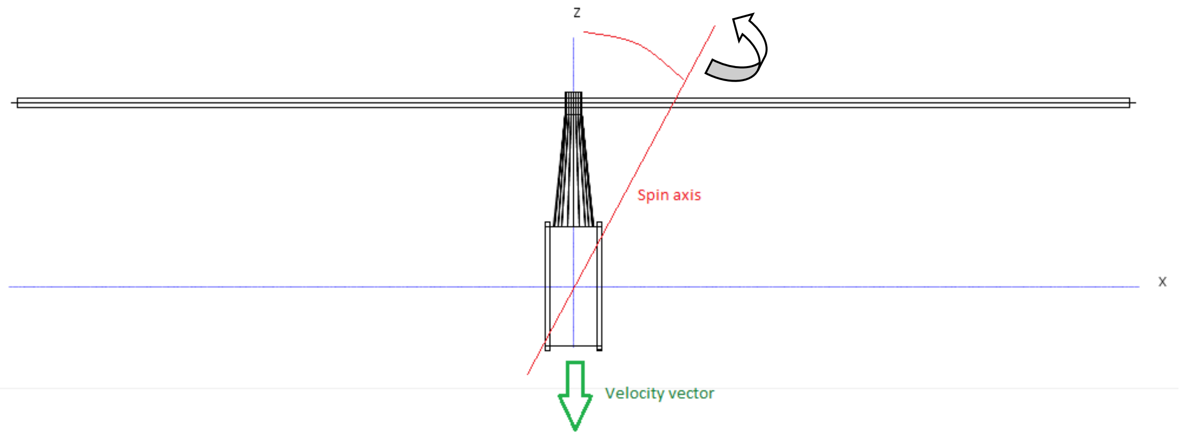

Figure 4-5 XZ-view of the 3D model with spin axis and velocity vector depicted

Results of the mean drag area are presented in Table 4-1.

Table 4-1 Mean drag area for different configurations and orientations of PW-Sat2

Mean Area [m²]	a) Random Tumbling	b) Fixed. Obs: X-axis	c) Fixed. Obs: Y-axis	d) Fixed. Obs: Z-axis	e) Rotation around axis at angle to Z-axis (obs: Z-axis)							f)
					5˚	10˚	15˚	30˚	45˚	60˚	90˚	
2U CubeSat	0.0267	0.0216	0.0216	0.0100	-	-	-	-	0.0256	-	0.0201	-
+Solar Arrays	0.0411	0.0617	0.0215	0.0102	-	-	-	-	0.0397	-	0.0201	-
+Sail	2.0150	0.0775	0.0321	4.0763	3.9708	3.8804	3.7325	3.0007	2.0256	2.0294	2.5742	0.2266

For the further analyses case d) was considered as the nominal case, and case a) (random tumbling) as worst case.

4.2.2 Orbital Perturbations

Different orbital perturbation models where used for different analyses. Orbit parameters evolution analyses used the most comprehensive perturbations model as described in §4.2.5 and ground stations contacts and eclipses analysis used only the simple J4 model §4.1.1 §4.2.4. In between are orbital lifetime analyses with the semi-analytical models which take into account most of the possible perturbations; however the short-term components are omitted in the equations for computation optimization purposes as described in [6].

4.2.3 Atmospheric Model

Atmospheric model used in orbital parameters evolution analysis in GMAT is the MSISE90 atmospheric model with constant solar flux and geomagnetic index as described in §4.2.5. Drag coefficient was constant and equal to 2.2.

As already mentioned in §4.2.2 ground station coverage and eclipses analyses in STK does not include any atmospheric modelling.

Orbital lifetime analyses in STELA use the NRLMSISE-00 atmospheric model. STELA default Solar Activity file consist of measured data of solar activity since 1957 to 2014 extended by the predictions up to 2318. Measured part of the data was analyzed to find mean values and standard deviation to determine the values for the best and worst scenario. Values of solar activity from STELA Solar Activity file are shown on Figure 4-7 and Figure 4-8.

Drag coefficient changes according to the STELA drag coefficient file Drag coefficient variation with altitude from STELA default file is presented on Figure 4-6. In Monte Carlo analyses the whole file is multiplied by a random number §4.2.4.

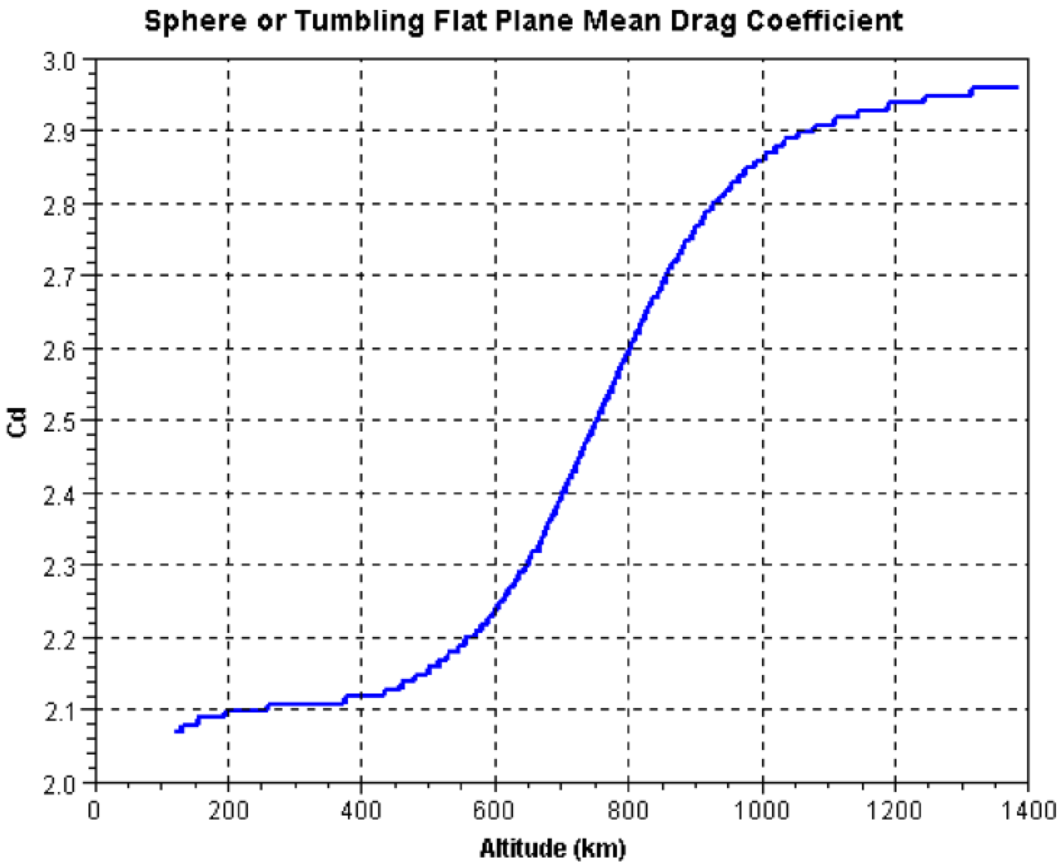

Figure 4-6 Drag coefficient variation with altitude, from STELA default file

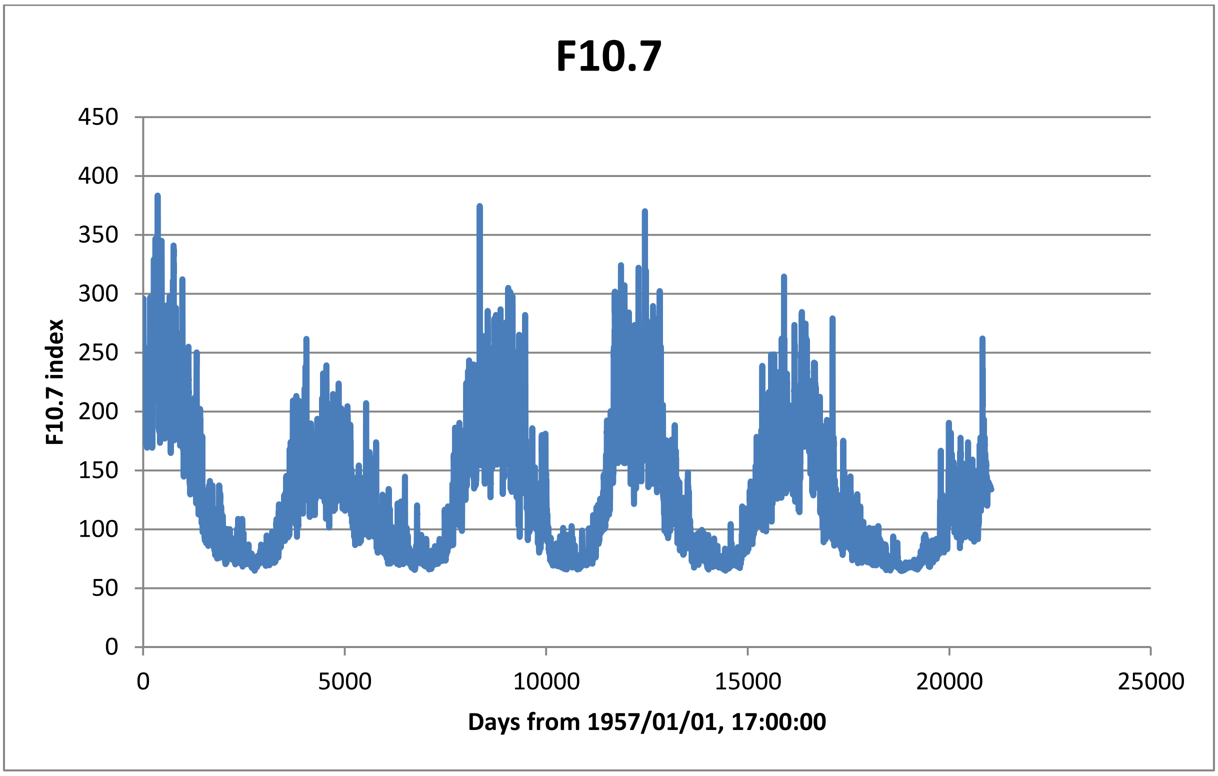

Figure 4-7 F10.7 index values between 1957/01/01 and 2014/09/24, STELA Solar Activity File

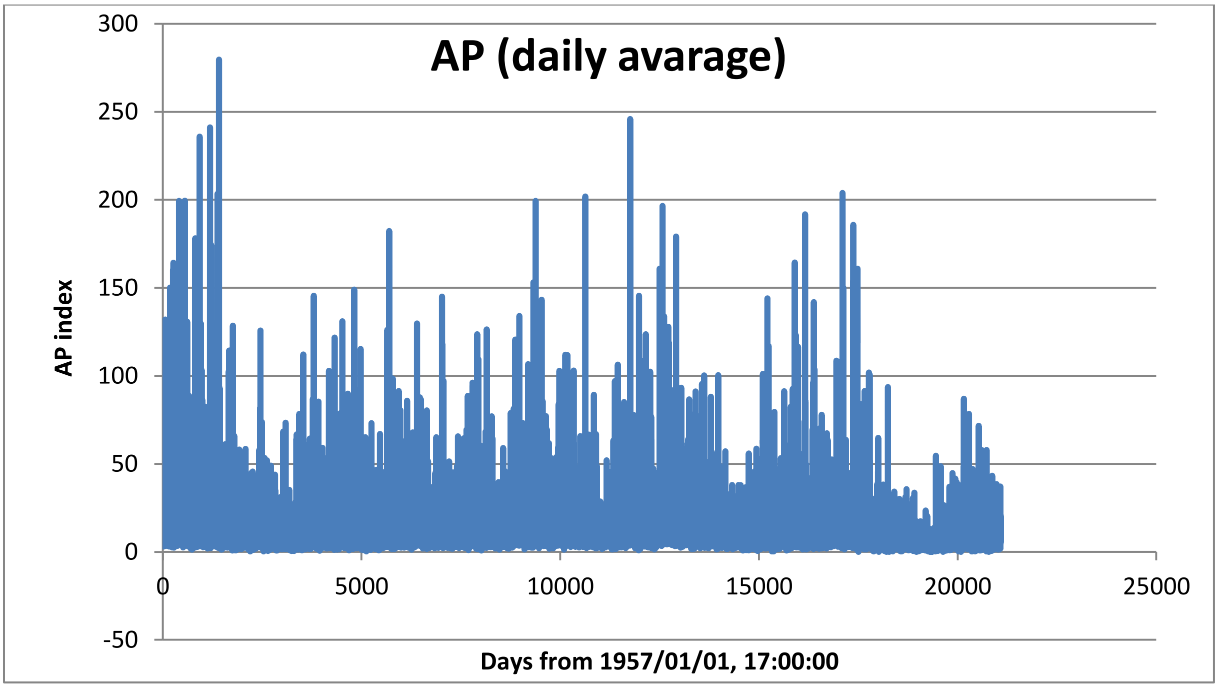

Figure 4-8 AP index values between 1957/01/01 and 2014/09/24, STELA Solar Activity File

4.2.4 STK Analyses Parameters

Without the comprehensive atmosphere drag modelling, due to limitations of the free license, it was not possible to use STK for the propagation of the orbit after the sail opening. For this reason STK has only been used for the analyses for the first 40 days of the mission. It has been used to generate the ground stations contact timetables and eclipses timetables utilizing the Access mechanics available in STK.

Table 4-2 STK analyses model parameters

STK version used	10.0	
Scenario Start date	6 Dec 2017 00:00:00.001 UTCG	
Scenario End date	16 Jan 2018 11:00:00.000 UTCG	
Propagator	J4	
Orbit definition	*Method:*	Orbit Wizard
	Type:	SSO
	Altitude:	575 km
	LTAN:	10:30
GS parameters	As provided in Table 2-1	

4.2.5 GMAT Analyses Parameters

GMAT has been used as the detailed propagator, so it is configured for precise propagation with all perturbations.

Table 4-3 GMAT analyses model parameters

GMAT version used	2016a (32-bit)	
Orbital Parameters	As described in §5.3	
SC parameters	*Dry Mass*	2.66 kg
	Coefficient of Drag	2.2
	Coefficient of Reflectivity	1.8
	Drag Area (w/o sail)	0.0267 m^2

	SRP Area (w/o sail)	0.0267 m²
	Drag Area (w/ sail)	4.0 m²
	SRP Area (w/ sail)	4.0 m²
Propagator parameters	*Integrator*	RungeKutta89
	Gravity Model	EGM-96 30x30
	Atmosphere model	MSIS90
	Solar Flux model	Constant
	Solar Flux value	150
	Geomagnetic Index (Kp)	3
	Point Masses	Sun, Luna
	SRP model	Spherical

4.2.6 STELA Analyses Parameters

STELA has been used for the generation of the ephemeris files for the MASTER simulations and for the Monte Carlo simulations for both opened sail and sail failure scenarios lifetimes.

Table 4-4 STELA analyses model parameters for ephemeris generation

STELA version used	2.6.1	
Model	GTO (statistical)	
Orbital Parameters	*As described in §5.3*	
Propagator	*Max. simulation duration*	25 years
	Integration step	21600 s (6 h)
	Atmospheric Drag quadrature Points	33
	Atmospheric Drag Recompute step	1 step
	SRP quadrature Points	11
	Sun	On
	Moon	On
	Zonal order	7

	Tesseral order	7
	Tesseral min period	5 steps
	Re-entry altitude	120 km
Space Object	*Mass*	2.66 kg
	Drag Area (w/o sail)	0.0267m^2
	Reflectivity Area (w/o sail)	0.0267m^2
	Drag Area (w/ sail)	2.0157 m^2
	Reflectivity Area (w/ sail)	2.0157 m^2
	Reflectivity Coefficient	1.5
	Drag Coefficient Type	Variable (variation with altitude)
Atmospheric Model	NRLMSISE-00	
Solar Activity Type	Variable (STELA solar activity file)	

Table 4-5 STELA model parameters for Monte Carlo analyses

STELA version used	3.0	
Model	GTO (statistical)	
Orbital Parameters	*As described in §5.3*	
Propagator	*Max. simulation duration*	100 years
	Integration step	21600 s (6 h)
	Atmospheric Drag quadrature Points	33
	Atmospheric Drag Recompute step	1 step
	SRP quadrature Points	11
	Sun	On
	Moon	On
	Zonal order	15
	Tesseral order	15
	Tesseral min period	5 steps

	Re-entry altitude	80 km
Space Object	*Mass*	2.66 kg
	Drag Area (w/o sail)	0.0267m^2
	Reflectivity Area (w/o sail)	0.0267m^2
	Drag Area (w/ sail)	2.0157 m^2
	Reflectivity Area (w/ sail)	2.0157 m^2
	Reflectivity Coefficient	1.5
	Drag Coefficient Type	Variable (variation with altitude)
Atmospheric Model	NRLMSISE-00	
Solar Activity Type	Variable (STELA solar activity file)	
Statistics	*Number of executions*	1000
	Date dispersion	UNIFORM (min = 09/01/2018, max = 15/04/2018)
	Hour dispersion	UNIFORM (min = 00:00:00:000, max = 24:00:00:000)
	Mass dispersion	GAUSSIAN (mean = 2.66 kg, deviation = 0.1 kg)
	Reflectivity area dispersion (w/o sail)	GAUSSIAN (mean = 0.0267 m^2, deviation = 0.003 m^2)
	Reflectivity area dispersion (w/ sail)	GAUSSIAN (mean = 2.0157 m^2, deviation = 0.05 m^2)
	Drag area dispersion (w/o sail)	GAUSSIAN (mean = 0.0267 m^2, deviation = 0.003 m^2)
	Drag area dispersion (w/ sail)	GAUSSIAN (mean = 2.0157 m^2, deviation = 0.05 m^2)
	Reflectivity coefficient dispersion	UNIFORM (mean = 1.5, delta = 20%)
	Drag coefficient dispersion	UNIFORM (min = 80%, max = 120%)
	Solar activity dispersion type	UNIFORM_GAUSSIAN
	Solar activity flux F10.7 dispersion	GAUSSIAN (deviation = 10%)

Solar activity AP dispersion	GAUSSIAN (deviation = 10%)
Orbital parameters dispersion correlation	None (unitary correlation matrix)
SMA standard deviation	8 km
ECC standard deviation	0.0
INC standard deviation	0.1°
RAAN standard deviation	5.0°
AOP standard deviation	0.0
Mean Anomaly standard deviation	120°

4.2.7 DRAMA/MASTER Analyses Parameters

DRAMA/MASTER has been used to calculate the annual collision probability with the space debris for a given orbit. These data were used for cumulated collision probability analyses.

Table 4-6 DRAMA/MASTER analyses model parameters

DRAMA/MASTER version used	DRAMA 2.0 / MASTER 2009	
Orbit definition	Provided from STELA-generated ephemeris file	
Functionality	1. Collision Probability Computation	
Particle Size	*Minimum size*	0.01 m
	Maximum size	100 m
Particle cloud	none	
Parameters of radar equation	default	
Spacecraft radius	*w/o sail*	0.09219 m
	(corresponding cross-section area w/o sail)	0.0267 m^2
	w/ sail	0.8 m
	(corresponding cross-section area	2.0157 m^2

	w/ sail)
Future scenario	1 – Business as usual

4.3 Assumptions for the Analyses

4.3.1 Ground Stations Analysis

As described in §2.4 the only ground stations used for PW-Sat2 is the ground station on the Electronics Faculty of Warsaw University of Technology. The geographical coordinates and altitude above the sea level are those of the rooftop of the Electronics Faculty. The minimal elevation is set to 30° which is exceptionally high for this kind of analyses; however, it is due to other buildings around blocking the visibility and due to the high noise from other electronic equipment nearby, Wi-Fi networks and antennas on the same rooftop which may make the communication on lower elevations unreliable.

4.3.2 Monte Carlo Analyses

In Monte Carlo analyses the initial date was set to 2018/01/18 which is the planned date for sail opening 40 days after the assumed launch date (2017/12/06).

For the lifetime analyses the worst case of drag area equal to 2.015 m^2 has been chosen, which corresponds to the cross-section area of the tumbling PW-Sat2 with sail §4.2.1.

4.3.3 Collision Probability Analyses

In collision probability analyses in ARES tool from DRAMA, the object is modelled as a sphere. The assumption is to input the sphere with the same cross-section area as the maximal cross-section area of the real spacecraft. So for a PW-Sat2 with the 4 m^2 sail, the mean cross-section area (when tumbling – worst case) is 2.0157 m^2, so the sphere modeming the spacecraft shall have a cross-section area of 2.0157 m^2.

5 LEOP and Operational Phase Analyses

From the mission analysis perspective, there is no distinction between the bus commissioning, payload commissioning and experimental phase for the PW-Sat2 mission. PW-Sat2 will have no propulsive capabilities; no orbital maneuvers will be performed. Throughout the whole mission one ground station will be used for the communication with the satellite.

5.1 Launch Window Characteristics

TBP - Launch window specification has not yet been provided by the Lunch Provider.

5.2 Launch Sequence of Events

TBP - Launch sequence of events specification has not yet been provided by the Lunch Provider.

As PW-Sat2 will be launch in a piggy-back opportunity, on a “dedicated rideshare” mission, details of the launch sequence of events may be still subject to change with the addition of more piggy-back payloads to the launch. Typical Falcon 9 launch sequence of events for LEO mission is presented in a Table 5-1 [7].

Table 5-1 Falcon 9 sample flight timeline - LEO mission [7]

Mission Elapsed Time	Event
T – 3 s	Engine start sequence
T + 0	Liftoff
T + 82 s	Maximum dynamic pressure (max Q)
T + 170 s	Main engine cutoff
T + 175 s	Stage separation
T + 180 s (3.0 minutes)	Second-engine start-1 (SES-1)
T + 220 s (3.7 minutes)	Fairing deploy
T + 540 s (9.0 minutes)	Second-engine cutoff-1 (SECO-1)
T + 600 s (10.0 minutes)	Spacecraft separation

5.3 Injection Errors

TBP - Injection errors for the Falcon 9 piggyback payloads have not yet been provided by the Launch Provider.

5.4 OPERATIONAL ORBIT CHARACTERISTICS AND EVOLUTION

Characteristics of the injection orbit as provided by the Launch Provider at the time of contract being signed are described in §2.3. These parameters translate into classical Keplerian parameters as presented in Table 5-2.

Table 5-2 Injection orbit keplerian parameters (TBC)

Epoch	Beginning of December 2017 (MJD1950 = 24811)
Semi-Major Axis	6953.14 km
Eccentricity	0.0
Inclination	97.6177°
Right Ascension of the Ascending Node (RAAN)	299.227°
Argument of Perigee	0°

Orbital parameters evolution analysis has been performed; the orbit has been propagated using the GMAT software. Corresponding model and assumptions has been described in §4. Results are shown in Figure 5-1.

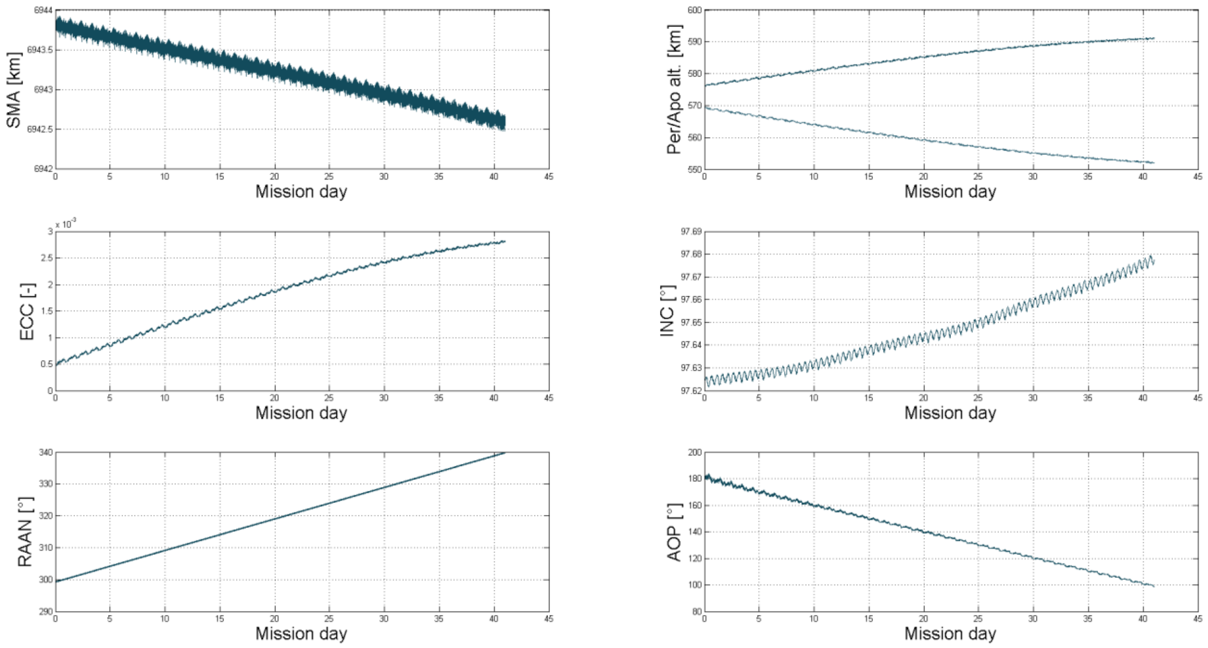

Figure 5-1 Orbital parameters evolution during mission nominal lifetime

Due to a very short nominal mission duration (40 days) orbital parameter change only slightly. With the decrease in SMA of 1.5 km, increase in inclination of 0.06° these parameters might be considered constant for the mission duration for other analyses. Variation in eccentricity is also negligible, however it strongly depends on the initial eccentricity which precise value and error margin are still unknown for PW-Sat2 injection orbit. Because of the negligible eccentricity, the AOP value does not have impact on the orbit geometry. RAAN is the only parameter which varies significantly, however the 30° raise in 30 days is the expected rate for the SSO orbit, and the simple J2 propagator would be sufficient to simulate this change.

5.5 GROUND STATIONS COVERAGE

Data for the ground stations coverage analysis has been generated using the AGI's STK software. Models and assumptions used are described in §4.

5.5.1 CONTACT DURATION PER ORBIT

Figure 5-2 presents the durations of contacts with the ground station per orbit, starting from the launch epoch.

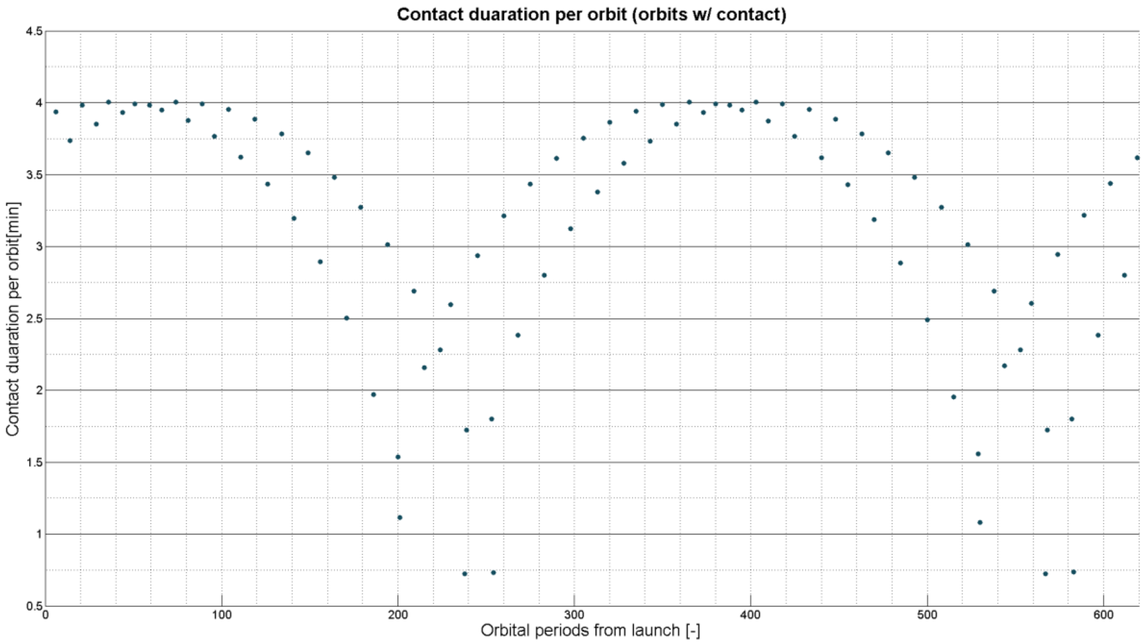

Figure 5-2 Contact duration per orbit

As presented in Table 5-3 maximal contact duration per orbit is 4.0027 min, minimal duration per orbit is 0.7238 min and mean duration is 3.1061 min.

5.5.2 CONTACT DURATION PER DAY

Figure 5-3 presents the duration of contacts with the ground station per day, starting from the launch epoch.

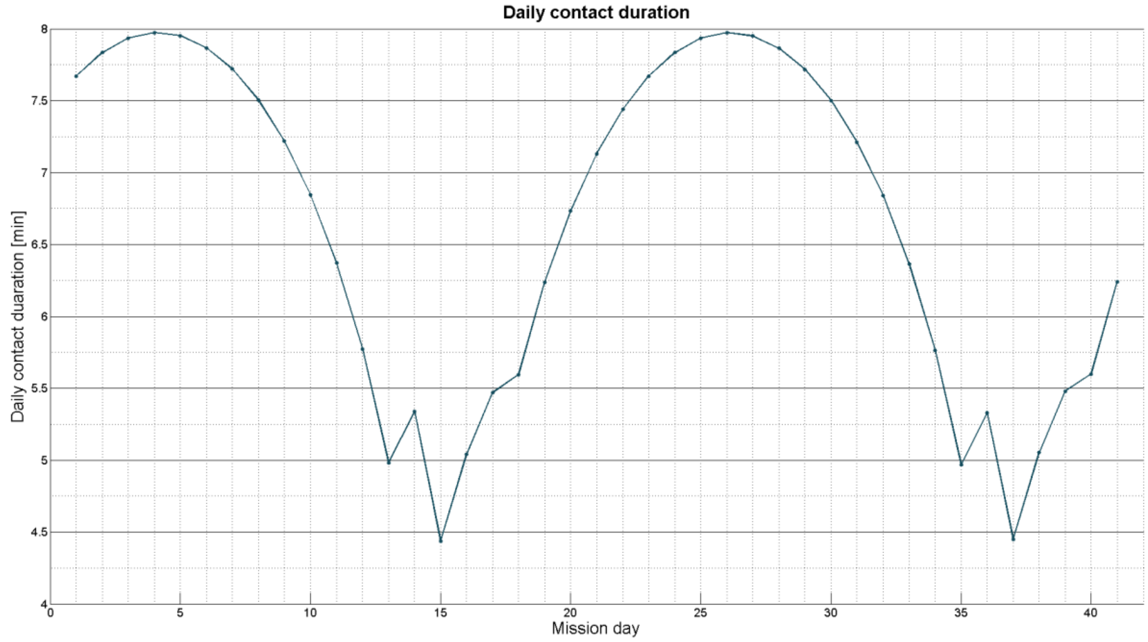

Figure 5-3 Contact duration per day

As presented in Table 5-3 maximal contact duration per day is 7.974 min, minimal duration per day is 4.4364 min and mean duration is 6.6542 min.

5.5.3 CONTACT GAP DURATION

Figure 5-4 presents the durations of the contact, starting from the launch epoch.

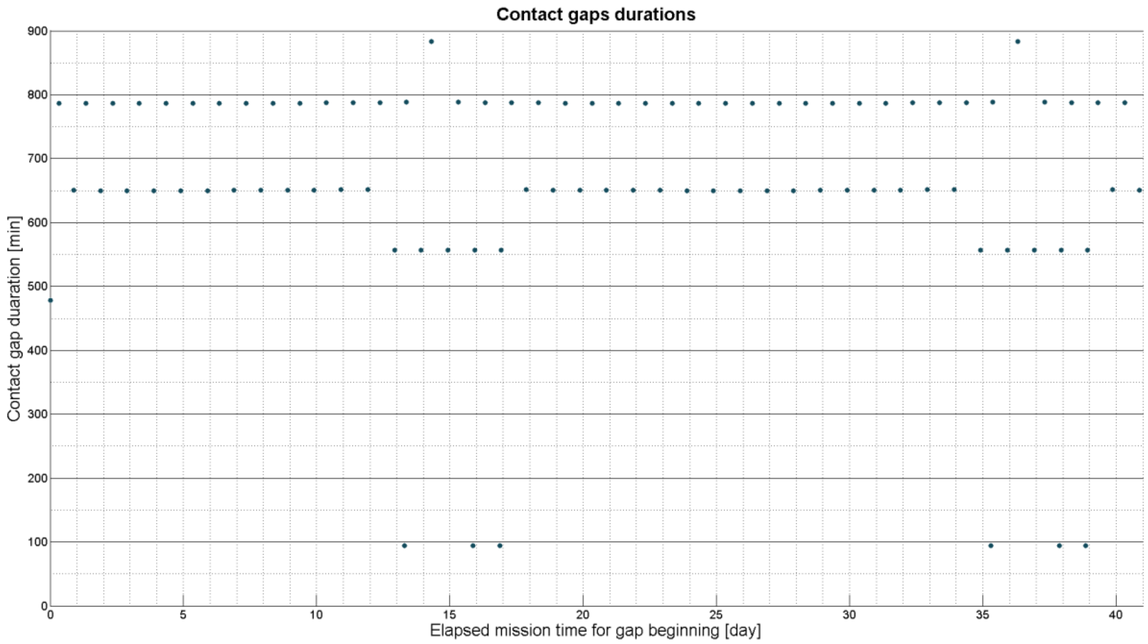

Figure 5-4 Contact gaps duration

As presented in Table 5-3 maximal gap duration is 883.1731 min, minimal duration is 93.9614 min and mean duration is 660.2618 min.

5.5.4 SUMMARY

Table 5-3 Summary of ground stations coverage analysis

Parameter		**value**	**on day**
Contact duration per orbit	*Maximal [min]*	4.0027	25
	Minimal [min]	0.7238	16
	Mean [min]	3.1061	-
Contact duration per day	*Maximal [min]*	7.974	4
	Minimal [min]	4.4364	15
	Mean [min]	6.6542	-
Gap duration	*Maximal [min]*	883.1731	15
	Minimal [min]	93.9614	14
	Mean [min]	660.2618	-

5.6 ECLIPSES DURING MISSION OPERATIONAL PHASE

Figure 5-5 presents the duration of eclipses per orbit, starting from the launch epoch.

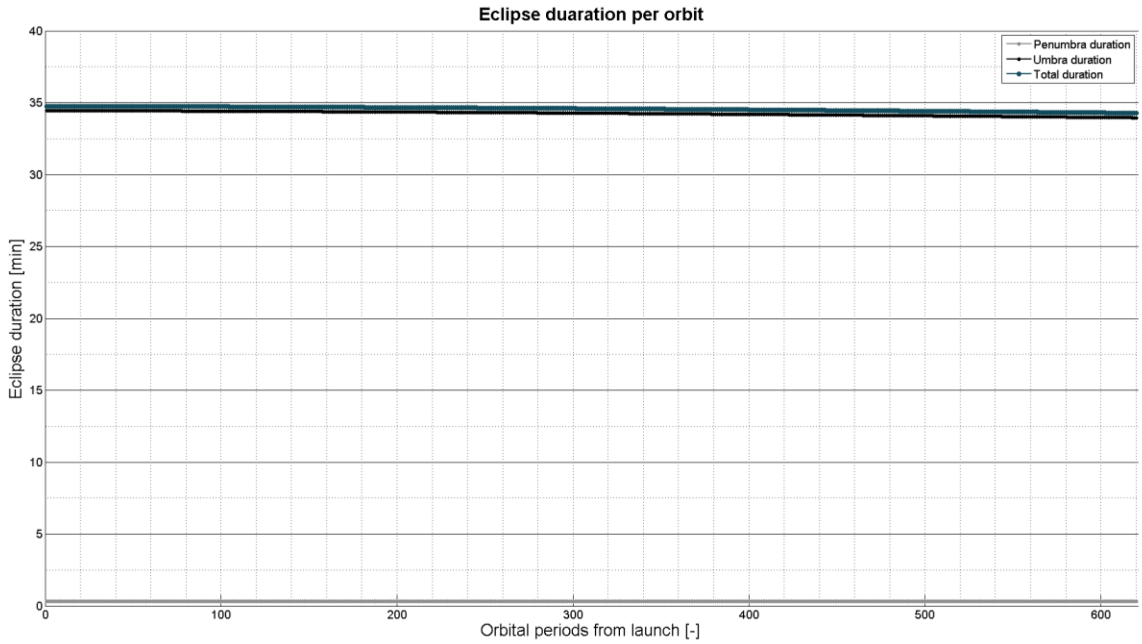

Figure 5-5 Eclipse duration per orbit

Table 5-4 presents the summary of eclipses duration during the mission nominal phase. As presented on Figure 5-5, eclipse duration is almost constant throughout the mission duration, only slightly decreasing with time.

Table 5-4 Summary of eclipse duration analysis

Parameter		**value**	**on day**
Eclipse duration per orbit	*Maximal [min]*	34.7656	1
	Minimal [min]	34.278	41
	Mean [min]	34.5658	-

5.7 Data Circulation Strategy

Strategy for the frequency of the Telemetry packets to be stored on board depends on the available memory, and on the size of one telemetry packet. Telemetry packet size is still TBD. As soon as the TM packet size is decided the optimal strategy (minimizing number of over-written packets) will be analyzed.

6 DE-ORBIT PHASE ANALYSES

6.1 ORBIT EVOLUTION WITH OPENED SAIL

Orbital parameters evolution analysis has been performed using GMAT propagator for the opened sail analogous to the one described in §5.3. Results are presented in Figure 6-1.

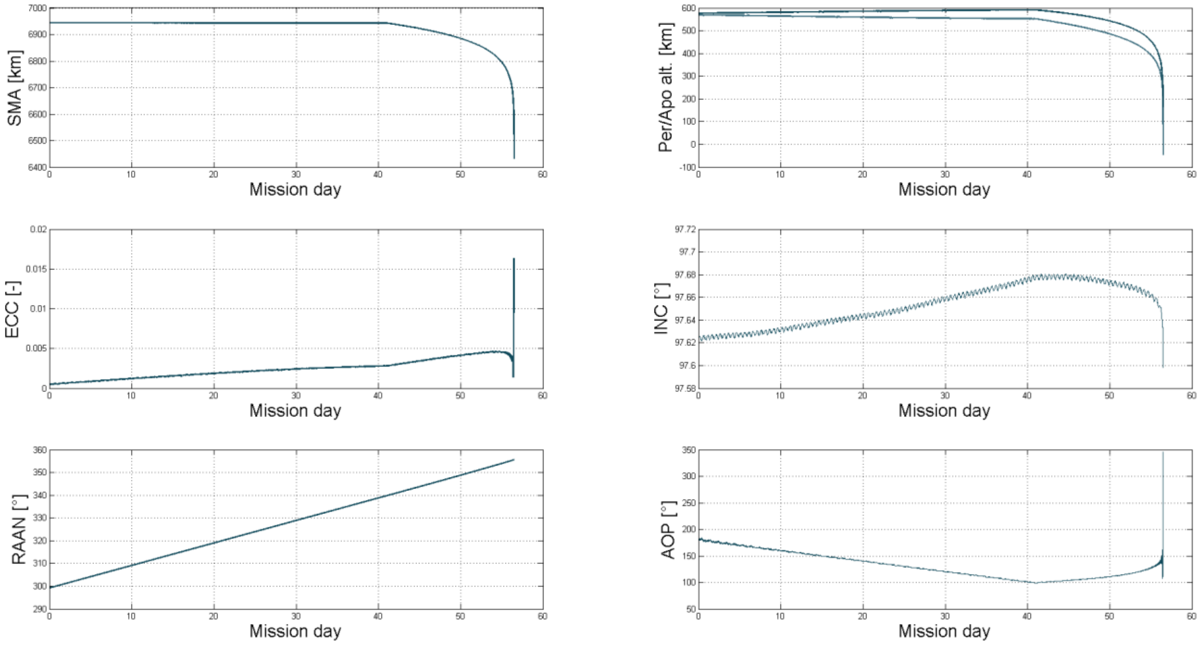

Figure 6-1 Orbital parameters evolution during the whole orbital lifetime

Comparing Figure 6-1 and Figure 5-1 it may be noted that for ECC, INC, AOP there is visible, although small, difference in the rate of change of those parameters after 40th day of mission when the sail is opened. The noticeable changes happen for those parameters closer to the r-entry date which is expected for the quickly degrading orbit. The RAAN rate of change is constant regardless of the sail opening. Obviously, there is noticeable difference in the rate of change in SMA, apogee radius and perigee radius.

Orbital evolution after the sail opening is highly dependent on the solar activity parameters which are highly unpredictable. Analysis presented here can only serve as an example of the possible orbital evolution as the results highly depend on the initial assumptions. The orbital lifetime analyses presented in §6.2 are performed using Monte Carlo method implemented in STELA software and give a better overview of the possible orbital lifetime after sail opening.

6.2 ORBITAL LIFETIME WITH SAIL MONTE CARLO ANALYSIS

Monte Carlo analyses were performed to determine the orbital lifetime of PW-Sat2 with opened sail. STELA software, models and assumptions are described in §4. Raw data from the analyses are presented in Figure 6-2. Figure 6-3 presents the distribution of the lifetime in a histogram, and Figure 6-4 presents the cumulative distribution of lifetime.

As may be seen in Figure 6-4, with the confidence level of 95%, there is a probability of 0.9 that lifetime of PW-Sat2 with opened sail will be shorter than 1.22 years. Lifetime confidence interval for probability of 0.9 is [1.16, 1.29] years.

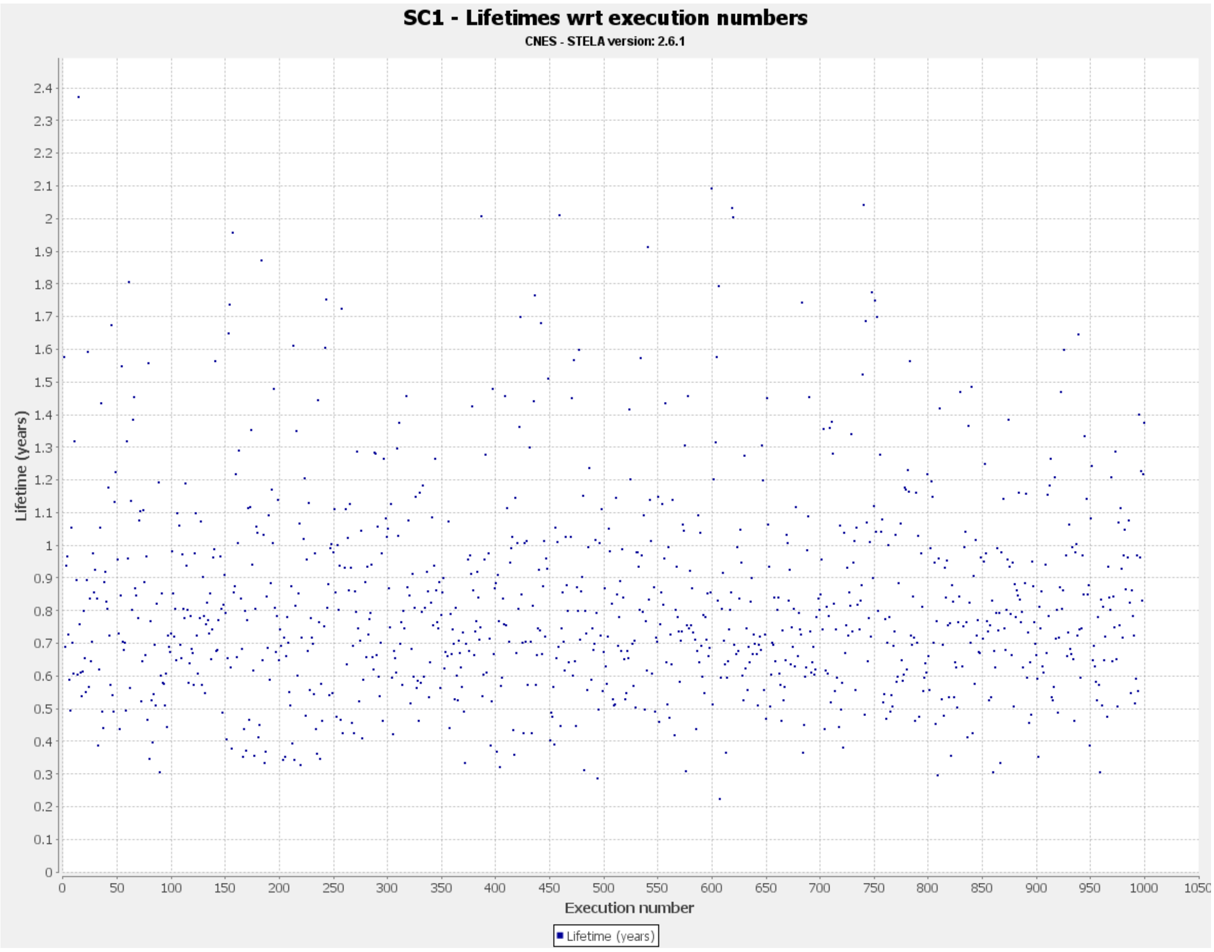

Figure 6-2 MC analysis results for opened sail - lifetime w.r.t. execution numbers

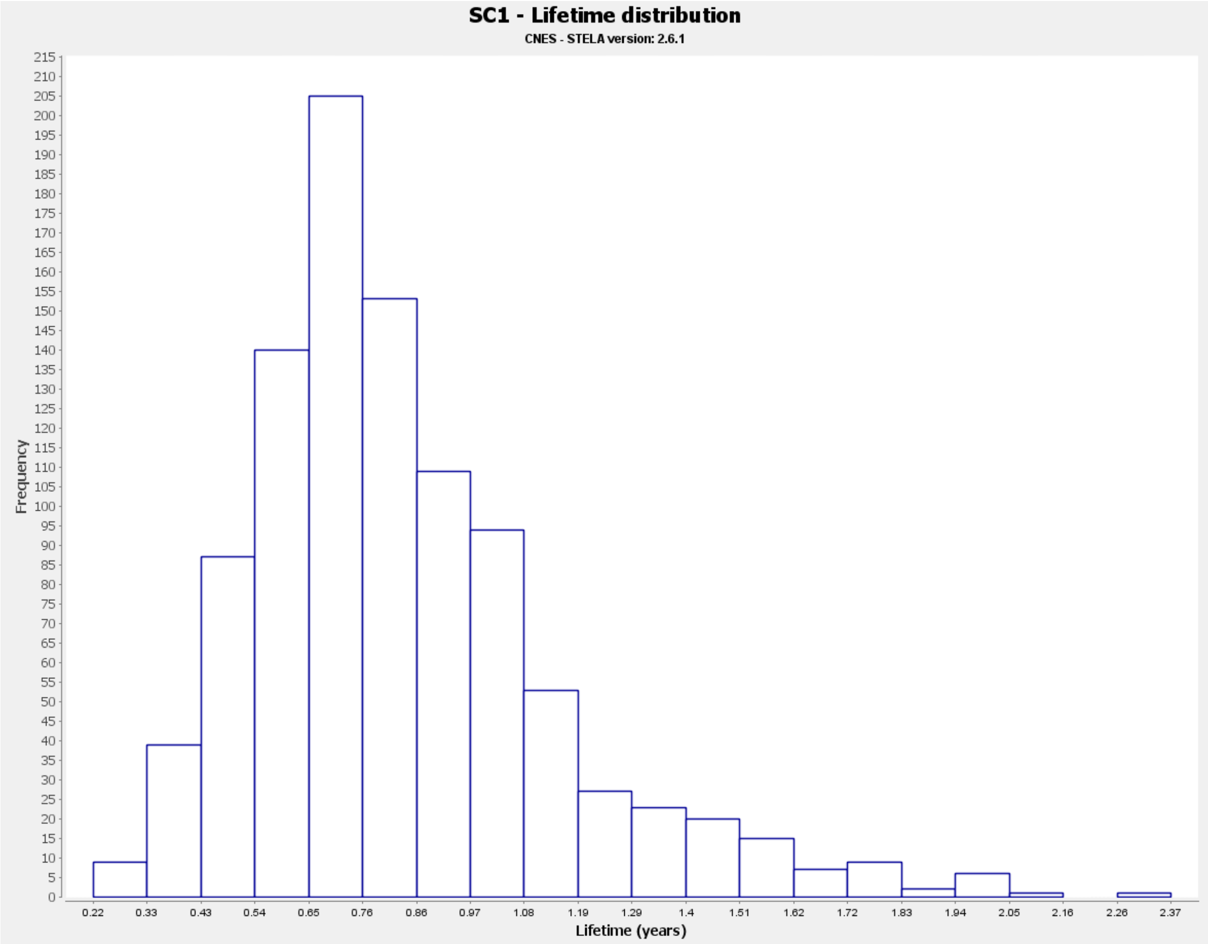

Figure 6-3 MC analysis results for opened sail - lifetime distribution

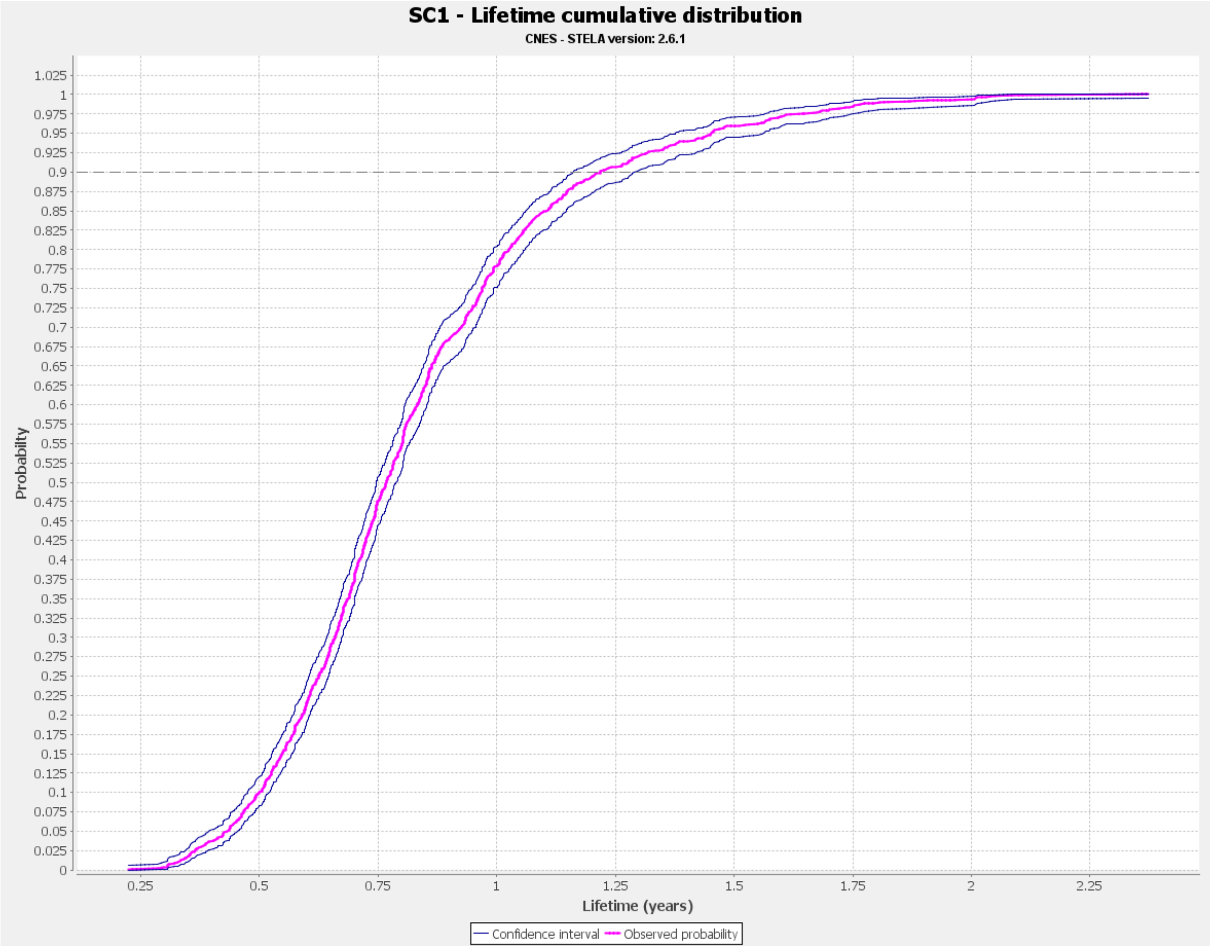

Figure 6-4 MC analysis results for opened sail - lifetime cumulative distribution

6.3 NATURAL ORBITAL DECAY IN CASE OF SAIL FAILURE

Analysis analogous to the one presented in §6.2 has been performed for the case where PW-Sat2 sail completely failed to deploy. Raw data from the analyses are presented in Figure 6-5. Figure 6-6 presents the distribution of the lifetime in a histogram, and Figure 6-7 presents the cumulative distribution of lifetime.

As may be seen in Figure 6-7, with the confidence level of 95%, there is a probability of 0.9 that lifetime of PW-Sat2 with opened sail will be shorter than 15.97 years. Lifetime confidence interval for probability of 0.9 is [15.75, 16.37] years.

In case of sail opening failure the mission will still be compliant with the IADC guidelines for space debris mitigation with the observed probability of 0.989.

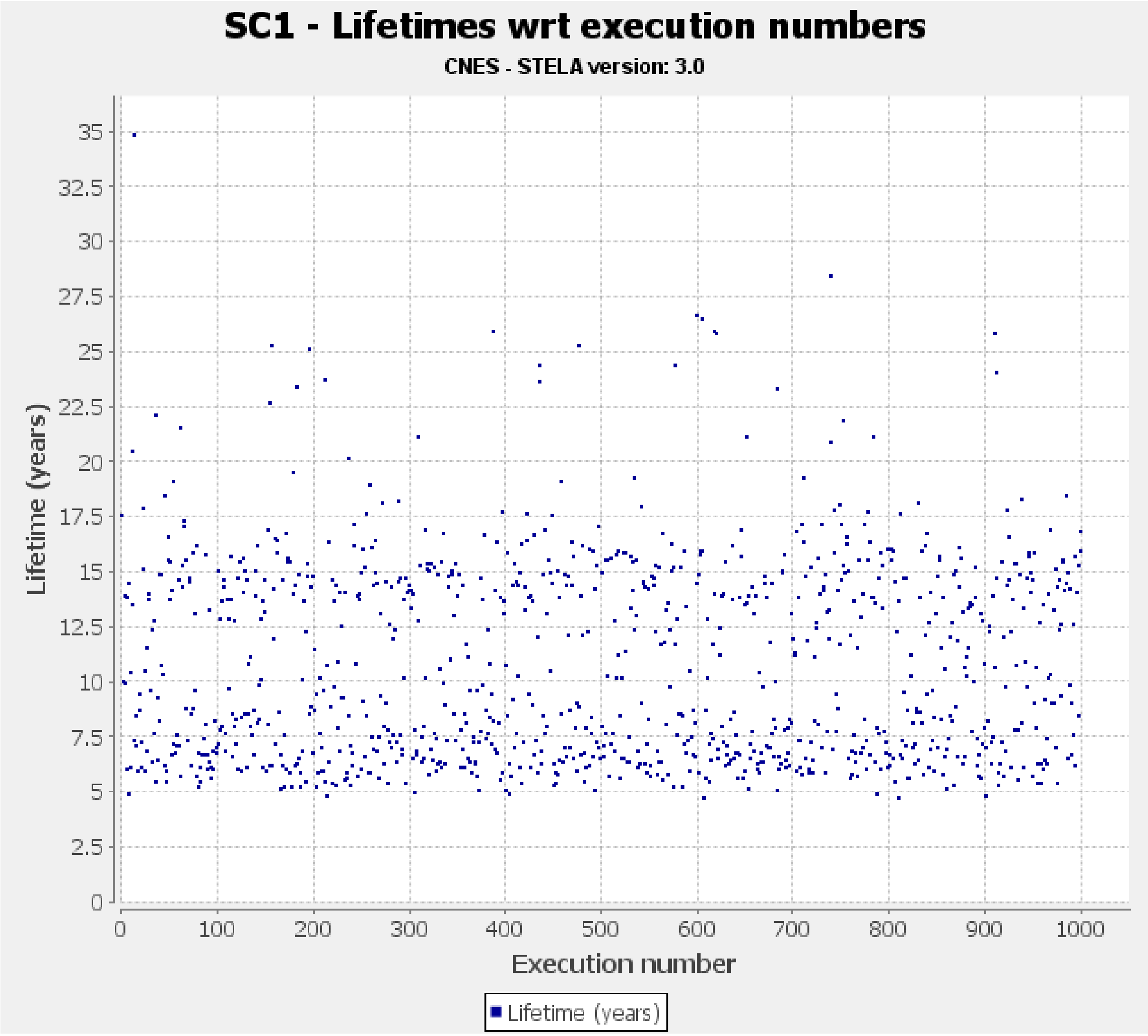

Figure 6-5 MC analysis results for no sail - lifetime w.r.t. execution numbers

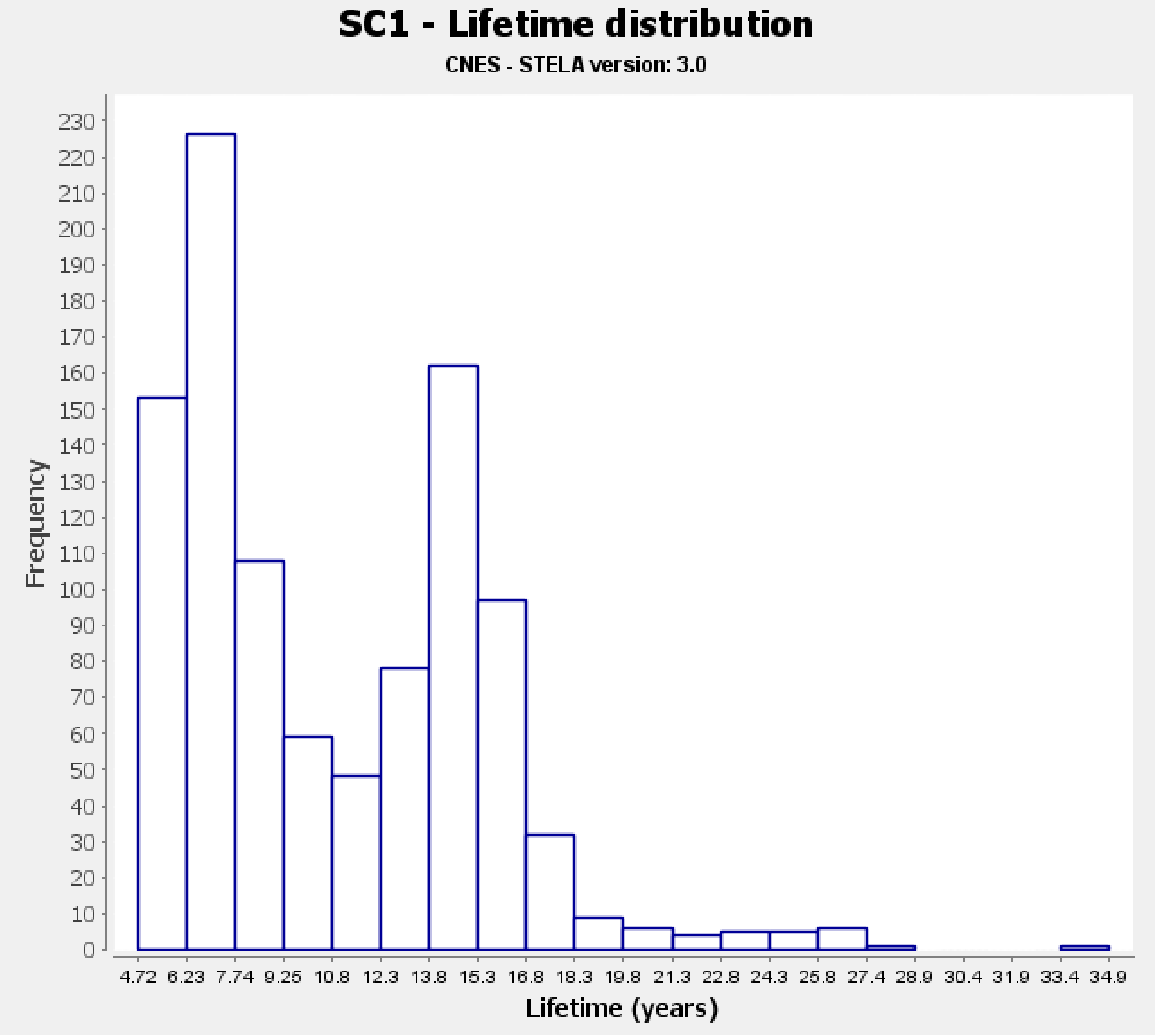

Figure 6-6 MC analysis results for no sail - lifetime distribution

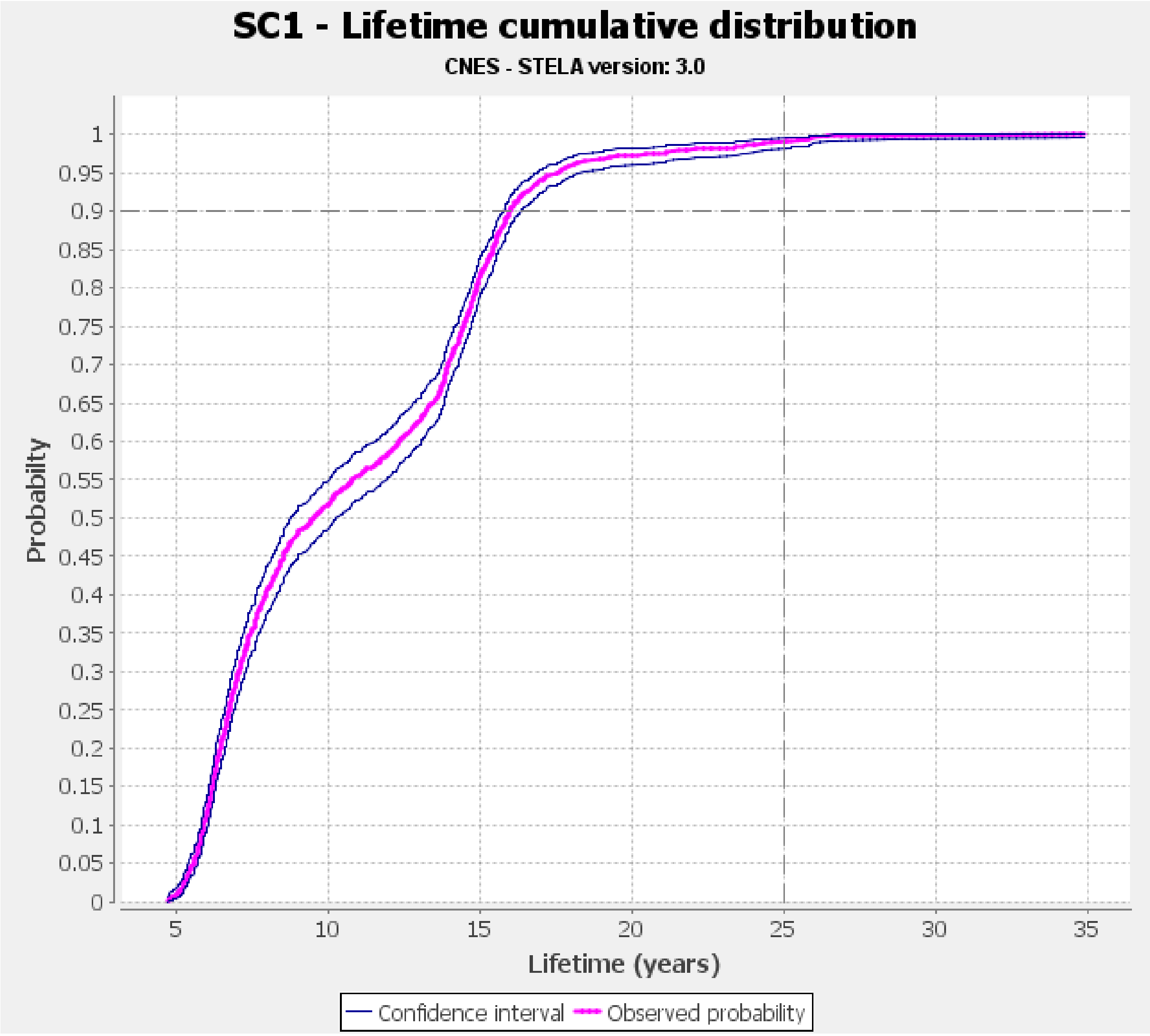

Figure 6-7 MC analysis results for no sail - lifetime cumulative distribution

6.4 COLLISION PROBABILITY ANALYSIS

The main purpose of the sail is to prevent orbital debris (satellites that completed their missions or sustained critical failure) from staying on orbit for a prolonged time, by deorbiting them. Deorbiting with sail could happen faster due to the increased area and thus decreased mass to cross-section ration of the satellite. However, increase in the satellite's cross-section area might potentially, contrary to the purpose of the sail, increase the population of space debris by increasing the on-orbit collision probability of the satellite. This phenomenon has been studied to make sure that the opening of the PW-Sat2 satellite will, in fact, decrease the probability of on-orbit collision. Debris Risk Assessment and Mitigation Analysis (DRAMA) software package has been used in this analysis, in particular Assessment of Risk Event Statistics (ARES) tool. ARES computes the annual probability of collision for a given size of the satellite on a given orbit. Generated in STELA, daily ephemerides of the satellite, from the end of mission to the disposal, with and without the sail have been taken as the inputs to the ARES analysis. Each result has been normalized to one day (divided by 365 days) resulting in Daily Collision Probability (DCP) and summed through the orbit lifetime as Cumulated Collision Probability (CCP). Results of such simulation are presented in Figure 6-8 and Table 6-1. Ephemerides used in ARES simulation were generated for the reference orbit described in Table 5-2 which was propagated until the re-entry. Rest of the simulation parameters are described in Table 4-6. It should be noted that the conservative case was analyzed for which the drag area used for ephemerides generation assumes tumbling satellite (decreasing the effective cross-section area, increasing the orbit lifetime) and for collision probability calculations, the area of a full sail aligned almost perpendicularly to the velocity vector (with small 5° oscillations around perpendicular direction) has been taken into account. Results show clearly that even though the Daily Collision Probabilities for the Open Sail scenario are one order of magnitude higher than for No-Sail scenario, the shorter lifetime on orbit does not allow the Cumulated Collision Probability to build up and the final CCP is higher for the scenario without sail. The conclusion is that even with the worst-case assumptions the sail opening shall decrease the cumulated collision probability of the satellite.

Table 6-1 ARES collision probability analysis results summary

Scenario	parameter	value
Opened sail	*Total simulation time span*	238 days
	CCP	1.382e-004
Sail failure	*Total simulation time span*	3191 days
	CCP	1.855e-04

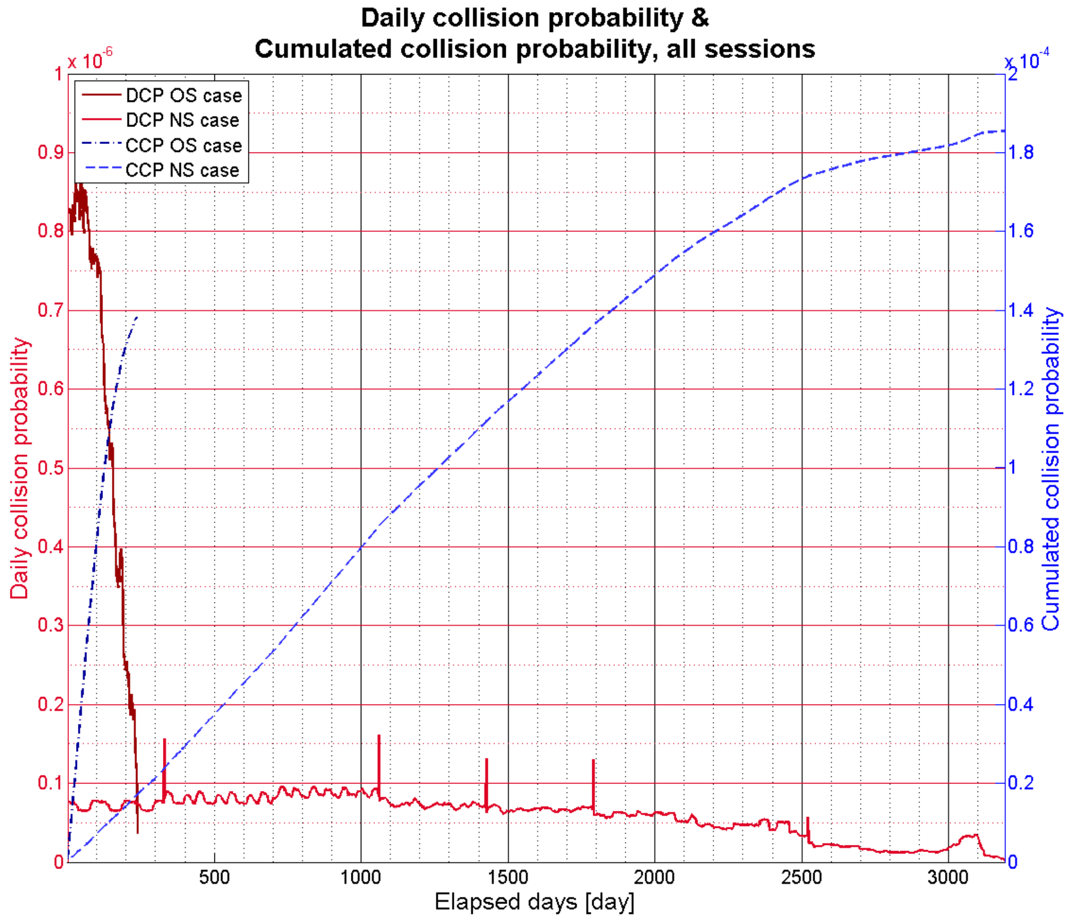

Figure 6-8 Daily collision probability (DCP) & cumulated collision probability (CCP), both scenarios, OS – Opened Sail, NS – No Sail

7 PW-SAT2 SAIL EFFECTIVENESS ANALYSIS

7.1 PW-SAT2 SAIL PERFORMANCE FOR OTHER BUSES ANALYSIS

The effectiveness of the PW-Sat2 deorbit sail has been simulated and analyzed using STELA

A study with use of STELA's statistical mode was performed in order to determine the influence of deorbit sail on the lifetime of 7 types of satellites on 6 different altitudes and 3 inclinations of orbits. For each case of satellite, a simple 3D model of spacecraft with and without deorbit sail was prepared in STELA Mean Area Computation Tool, in order to calculate active mean drag area assuming random tumbling of the spacecraft. The selected masses and respective mean areas, as well as mean ballistic coefficients are presented in Table 7-1. Assumed deorbit sail has an area of 4 m^2 and corresponds to PWSat2's deorbit mechanism.

Table 7-1 Parameters of selected spacecraft

Parameter	Values						
Type	2U	3U	6U	12U	24U	48U	SSTL-100
Mass [kg]	2.6	3.9	7.8	15	30	60	100
Drag area [m^2]	0.0267	0.0455	0.0667	0.0979	0.1571	0.2509	0.8376
Drag area w/ sail [m^2]	2.0157	2.0188	2.0210	2.0274	2.0466	2.0685	2.4553
Mean ballistic coefficient	97.61	85.85	116.8	152.6	192.0	238.3	120.0
Mean ballistic coefficient w/ sail	1.29	1.94	3.86	7.39	14.6	29.02	40.96

Representative orbital parameters were chosen based on a study of the most popular orbits for recently launched nanosatellites. The publicly available satellite database of Union of Concerned Scientists [8] was used to define 6 altitudes of orbits that are presented in Table 7-2. The vast majority of them are Sun-Synchronous orbits with very small eccentricity what may be observed in Figure 7-1 and Figure 7-2. Only few examples of satellites sent to high-eccentricity orbits still remains in space. Good example is previous PW-Sat, which was launched in 2012 on Vega rocket to 310 x 1441 km orbit and decayed 2.5 year later. There is no visible trend in case of non-SSO orbits hence; two arbitrary inclinations of 40° and 60° were added. All the orbits in simulations are assumed to be circular.

Table 7-2 Selected orbital parameters used in simulations

Parameter	Value					
Altitude [km]	500	600	650	700	750	800
SSO inclination [˚]	97.41	97.79	97.99	98.19	98.39	98.60
Non-SSO inclination [˚]	40 and 60					

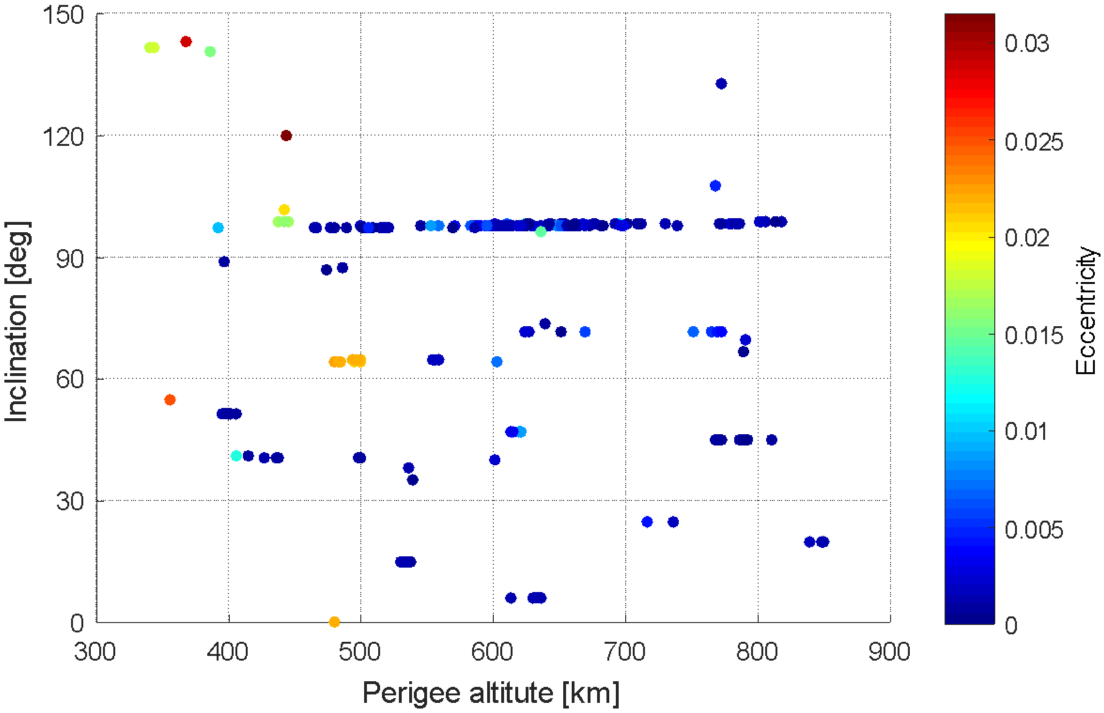

Figure 7-1 Inclination of the satellite as a function of perigee altitude for spacecraft of launch mass below 500 kg and perigee altitude below 900 km. Own work based on UCS Satellite Database [8]

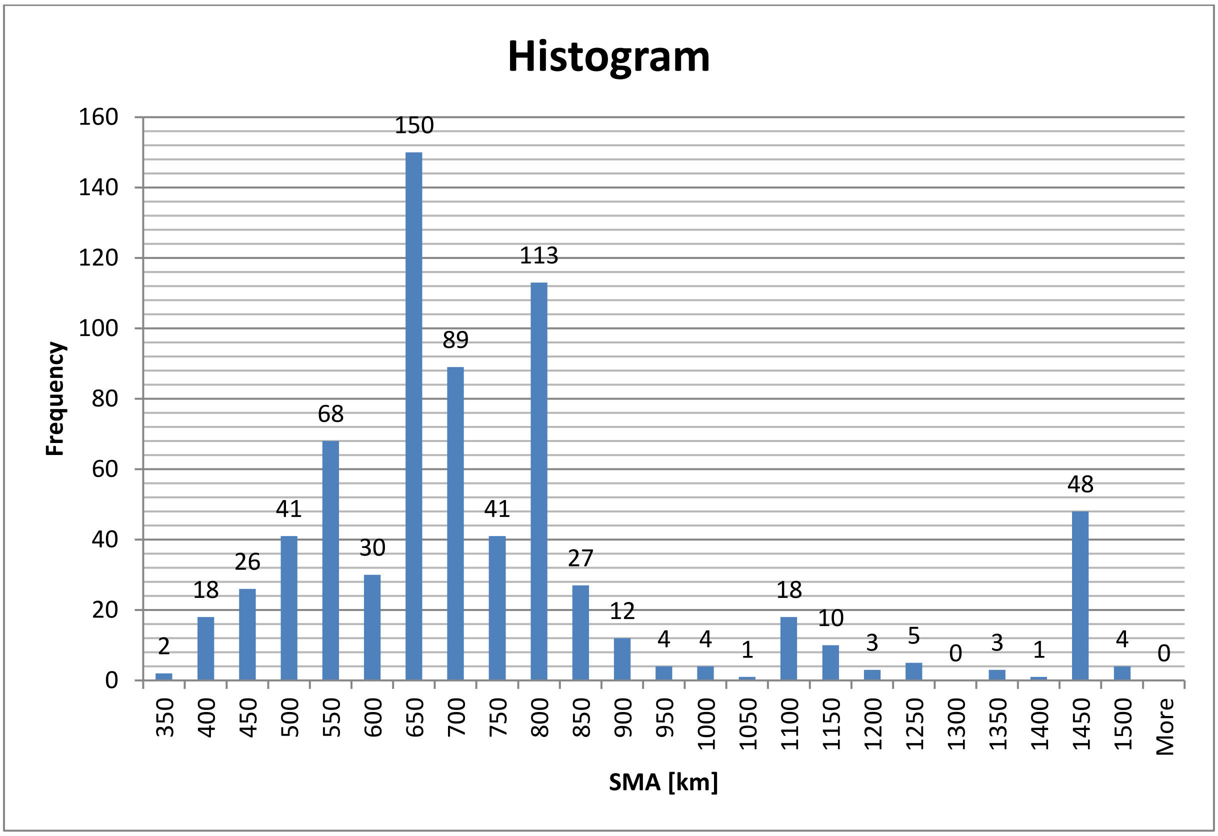

Figure 7-2 Histogram of the operational satellites on circular orbits below 1500 km [8]

Simulations in STELA were performed in statistical mode and each of them was run up to 50 times or until the probability of re-entry under 25 years was higher than 0.9. Presented in Table 3 options of STELA were used in each of 252 input files. For every input file a different simulation seed number was generated with MATLAB. Right Ascension of the Ascending Node (RAAN) was selected as main dispersed parameter in range of 180°. Moreover, three other parameters were dispersed: drag coefficient, solar flux F10.7, and solar flux AP.

Table 7-3 General and advanced options of simulation in STELA

Parameter	Value	Parameter	Value
Max no. of executions	50	Integration step	12h/24h*
Simulation duration	100/200 years*	Atmospheric drag quadrature pts.	33
Initial date	2018-10-21 21:00:00 UTC	Solar radiation quadrature pts	11
RAAN	30˚	Third bodies perturbations	Enabled
AoP	210˚	Earth gravity zonal/tesseral order	15/15
MA	190˚	Solar Tides perturbations	Enabled
Atmospheric model	NRMLSISE-00	Re-entry altitude	140 km

* In case of high orbits simulation time and integration step were longer

Output simulation files were interpreted by MATLAB script and mean value of lifetime and ballistic coefficient were calculated for each case. Usually, the dispersion of resulting lifetime was not large, but in case of high altitude orbits for satellites with high ballistic coefficient the difference between minimal and maximal predicted lifetime was in range of dozens of years.

Figure 7-3 shows approximated lifetime value calculated as a function of ballistic coefficient separately for each altitude and inclination. Drastic difference in lifetime between satellites with and without deorbit device is visible. It may be deduced that SSO orbits generally have longer lifetime than moderately inclined, perhaps because of non-spherical distribution of atmosphere around the Earth. Table 7-4 shows the same results. Left-hand half of the table ("Sail") indicates for which orbit-bus combinations PW-Sat2 sail is effective (green) and for which it is not big enough (red) as the lifetime is longer than 25 years. Grey fields show for which bus-orbit combinations sail is not necessary as the lifetimes of the satellites without sail are shorter than 25 years already. Right-hand side of the table ("No sail") shows the lifetimes of the spacecraft without the sail for comparison. Performed study shows that such a device as a deorbit sail of 4m^2 area may be very effective even for relatively massive nano- and microsatellites such as e.g. SSTL-100 bus.

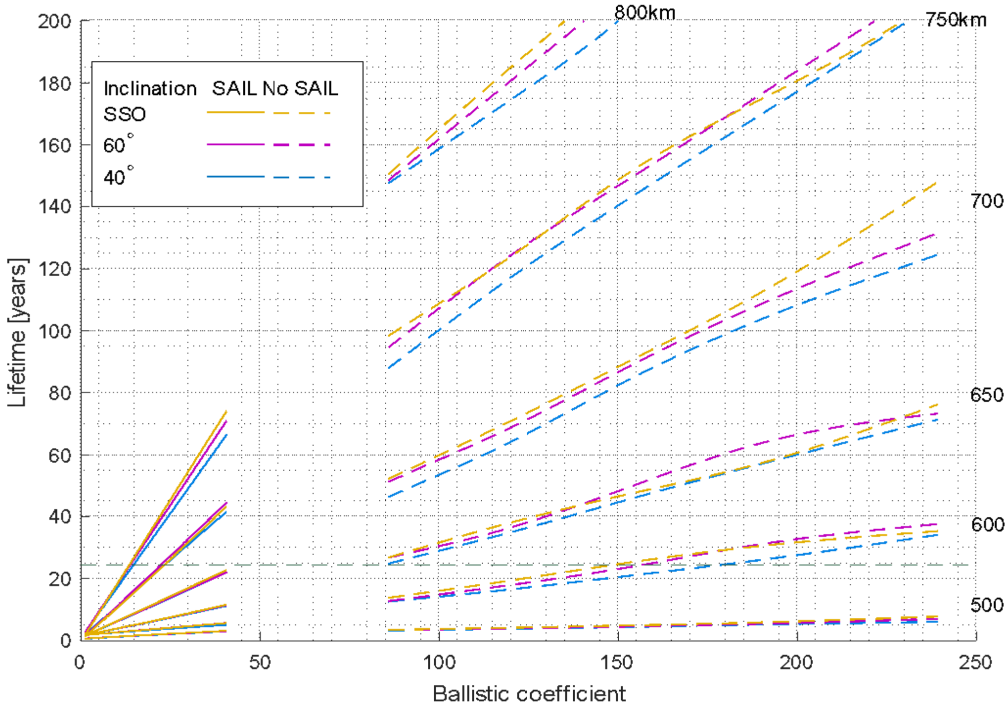

Figure 7-3 Approximated relation between orbit lifetime and ballistic coefficient of the satellite on circular orbits of various altitudes

In Figure 7-3 on the left side below BC=50 satellites with deorbit sail deployed are visible – contrary to the ones without sail on the right. Dashed horizontal line marks the 25 year time limit.

Table 7-4 Comparison of orbital lifetime [years] for selected satellite buses on selected orbits with and without sail.

	Sail							No sail						
	2U	3U	6U	12U	24U	48U	SSTL-100	2U	3U	6U	12U	24U	48U	SSTL-100
$\overline{BC}$	97.61	85.85	116.8	152.6	192.0	238.3	120.0	1.29	1.94	3.86	7.39	14.6	29.02	40.96
800 km	3.1	3.7	6.0	12.4	26.3	49.6	70.7	160.6	148.2	179.9	193.3	199.2	199.7	177.1
750 km	2.7	3.0	4.1	6.6	14.9	31.1	43.3	104.0	93.0	118.9	149.1	173.8	189.8	121.9
700 km	2.3	2.5	3.1	4.0	6.9	16.7	22.4	58.6	49.7	64.2	90.5	108.4	134.8	65.7
650 km	1.9	2.2	2.6	3.2	4.2	7.4	12.2	30.2	26.4	35.3	48.2	60.1	73.5	35.7
600 km	1.4	1.7	2.2	2.6	3.1	4.3	5.3	14.7	13.4	17.7	23.2	30.1	35.6	17.6
500 km	0.2	0.3	0.7	1.3	2.0	2.4	2.7	3.7	3.6	4.1	4.7	5.6	7.1	4.1

7.2 Drag Area Variation Analyses

An analysis of drag area influence on orbital lifetime was conducted for every orbit considered in phase B. [PW-Sat2-B-00.01-MA-PDR] Nominal parameters other than drag area are the same as in analyses of phase B. Drag area dispersion is uniform from the range of 0.22657169 - 2.01495686 m^2 (case f) – case a) from §4.2.1). Results for all considered orbits are presented in Figure 7-4 Maximal simulation duration was set to 100 years.

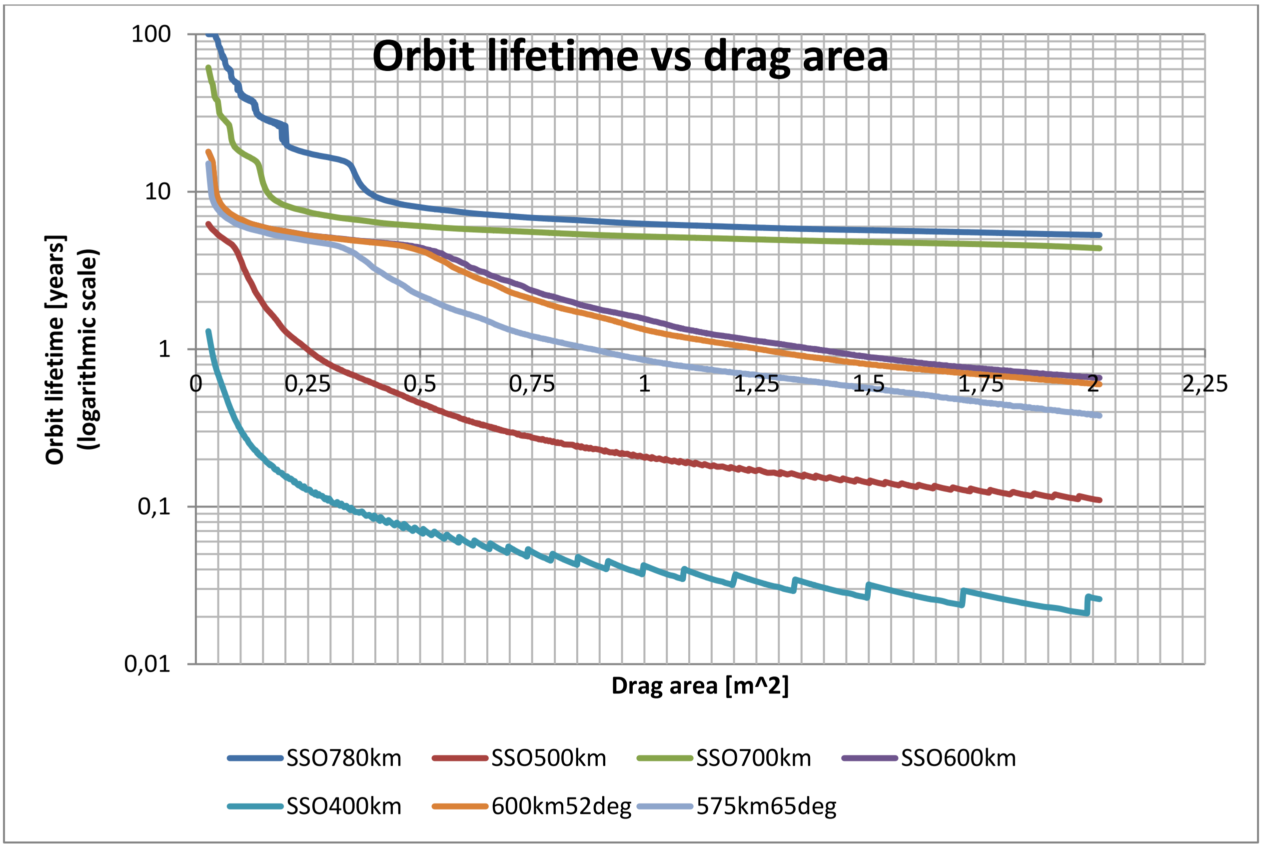

Figure 7-4 Orbit lifetime vs drag area

From the conducted analyses it can be seen that drag area together with solar and geomagnetic activity indices has the strongest influence on orbit lifetime. Orbit lifetime dependency on drag area is very complex. For drag area lower than 0.5 m^2 and orbits higher than 575 km there is a “steps” effect visible, while for the lower orbits and drag area higher than 0.75 m^2 the “saw” effect can be noticed on logarithmic plot. Explanation of these effects is beyond the scope of this analysis, however.

Appendix A REVISION OF THE PREVIOUS WORK

A.1 TEAM OBJECTIVES

During Phase A the following tasks were defined for the Mission Analysis Team:

1. Finding a way to launch the satellite into orbit

2. Mission and orbit analysis in Mission Analysis software

   2.1. Contact with software distributors

   2.2. Organization of training mission analysis software

   2.3. Mission modelling

      2.3.1. Modelling of solar panels' exposure to light

      2.3.2. Modelling of communication session with ground station

      2.3.3. Calculation of suitable time to test sun sensor

3. Implementation a of detailed mission plan

4. Preparation of the satellite operators' team (OPER)

   4.1. Radio amateur training organization

   4.2. Obtaining of radio amateur licenses

   4.3. Process mission plan to a set of telecommands

   4.4. Develop contingency plans for emergency response of individual sub-systems

   4.5. Risk analysis for satellite mission

A.2 PHASE B ACTIVITIES

As described in [PW-Sat2-B-00.01-MA-PDR] Phase B activities included:

- Launch Opportunities Selection
  - De-orbit time analyses
  - Communication sessions analyses
  - Eclipses analyses

- Acquisition of educational licenses of mission analysis software.

Appendix B LAUNCH OPPORTUNITIES SELECTION

B.1 INTRODUCTION

Initial search for the launch opportunities was conducted in a scope of phase A activities as described in [PW-Sat2-A-00.01-MA-PRR], §3. Since the time of the search preparation some of the offers became outdated as the projects AR has been postponed to no earlier than February 2016. During phase B, contact with the launch providers was maintained and updated accordingly, however without the funds secured for the launch only the list of opportunities and their prices has been maintained. In December 2015 Polish Ministry of Science and Higher Education decided to support the project with 180.000 € for the launch of the PW-Sat2. The funds have been transferred to the Ministry of Development and from there to ESA as the increase of the Polish contribution to ESA. ESA awarded the Warsaw University of Technology with the contract to organize a launch of PW-Sat2. Invitation to tender has been announced by the WUT in July 2016 for a launch service. Tender has been resolved in August and the contract has been awarded in October 2016 to Innovative Space Logistics B.V. The offered launch is the Falcon 9 launch in 4th quarter of 2017 to the SSO orbit with the altitude of 575 km and LTAN of 10:30.

B.2 CHOICE OF THE LAUNCH PROVIDER

Details of the launch service tender have been described in the ITT documents especially in the SIWZ (pol.: Specyfikacja Istotnych Warunków Zamówienia, eng.: Terms of References) document. [9].

B.3 LAUNCHER RELIABILITY DATA

Active launch vehicles reliability data have been collected for the purpose of determining the reliability threshold for the launch ITT. Data is presented in Table B-1 source of data were Wikipedia entries for each of the rocket which are very regularly updated.

Table B-1 Launch vehicle reliability data. Source: Wikipedia

rocket	family	Active /Retired	Total launches	Success	Partial failure	failure	Total failure	Reliability [%]	Total launch +1	Observed failure probability
Atlas V		Active	63	62	1	0	1	98.41	64	0.03
Delta II	Delta	Active	153	151	1	1	2	98.69	154	0.02
Delta IV	Delta	Active	32	31	1	0	1	96.88	33	0.06
Falcon 9	Falcon	Active	26	24	1	1	2	92.31	27	0.11
H-IIA	H-II	Active	30	29	0	1	1	96.67	31	0.06
H-IIB	H-II	Active	5	5	0	0	0	100.00	6	0.17
PSLV		Active	36	34	1	1	2	94.44	37	0.08
Vega		Active	6	6	0	0	0	100.00	7	0.14
Ariane V	Ariane	Active	86	82	2	2	4	95.35	87	0.06
Soyuz-FG	R-7	Active	56	56	0	0	0	100.00	57	0.02
Soyuz-2	R-7	Active	62	57	3	2	5	91.94	63	0.10
Long March 2C	Long March 2	Active	41	40	0	1	1	97.56	42	0.05
Long March 2D	Long March 2	Active	28	28	0	0	0	100.00	29	0.03
Long March 2F	Long March 2	Active	11	11	0	0	0	100.00	12	0.08
Long March 3A	Long March 3	Active	25	25	0	0	0	100.00	26	0.04
Long March 3B	Long March 3	Active	35	33	1	1	2	94.29	36	0.08
Long March 3C	Long March 3	Active	14	14	0	0	0	100.00	15	0.07
Proton		Active	365	318	0	47	47	87.12	366	0.13

Appendix C ORBIT LIFETIME VS SIMULATION PARAMETERS ANALYSIS (SINGLE-PARAMETER ANALYSES)

C.1 SINGLE-PARAMETER VARIATION

To determine the influence of every single parameter on a decay time 8 analyses were conducted for the SSO780km orbit taken as an example. The highest of the considered orbits was chosen so that the possible effects with longer periods might be visible. Nominal simulation parameters for single-parameter variation analyses are presented in a Table C-2 (for each simulation analyzed parameter varies from the nominal value by the dispersion value):

Table C-2 Single-parameter variation analyses nominal configuration

Parameter	Nominal value	Dispersion[1]
Launch date	2016/13/02, 09:00:00UTC	Uniform (2016/13/02 - 2017/13/02)
RAAN	111.738	Uniform 0˚-360˚
Mean anomaly (M)	0	Uniform 0˚-360˚
Mass	3.2 kg	Uniform 2.6-3.2kg
Coefficient of drag (Cd)	STELA Default file	Uniform +/- 20 %
Reflectivity coefficient (Cr)	1.5	Uniform +/- 20 %
Drag area	2.01495686 m2	N.A.
Reflectivity Area	2.01495686 m2	N.A.
Solar activity, F10.7 index	140	Gaussian[2]: $\sigma = 53.47; \bar{x} = 126.27$
Geomagnetic activity, AP index	15	Gaussian: $\sigma = 15.51; \bar{x} = 13.56$

C.1.1 LAUNCH DATE

Orbit lifetime variation with launch date is presented on a Figure C-1.

[1] Dispersion is applicable only in a simulation of a particular parameter influence, for other parameters there is no dispersion then.

[2] σ - standard deviation; $\bar{x}$ = mean value

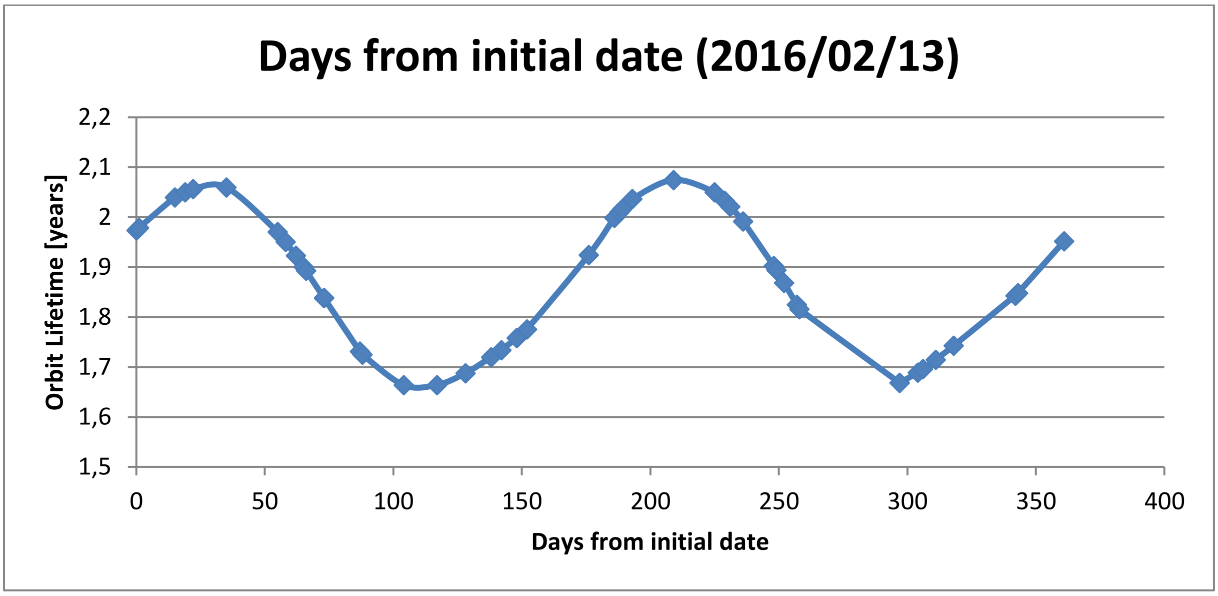

Figure C-1 Orbit lifetime vs launch date

For nominal analysis parameters and launch date dispersion between 2016/02/13 and 2017/02/13, orbit lifetime oscillates between 2.0747 and 1.6636 years, therefore the relative difference[3] is 19.81 %

[3] Relative difference $(\Delta T_{orbit})_{relative}$ of values x and y is defined as follows: $(\Delta T_{orbit})_{relative} = \frac{|x-y|}{\max(|x|,|y|)}$

C.1.2 Right Ascension of the Ascending Node

Orbit lifetime variation with RAAN is presented on a Figure C-2.

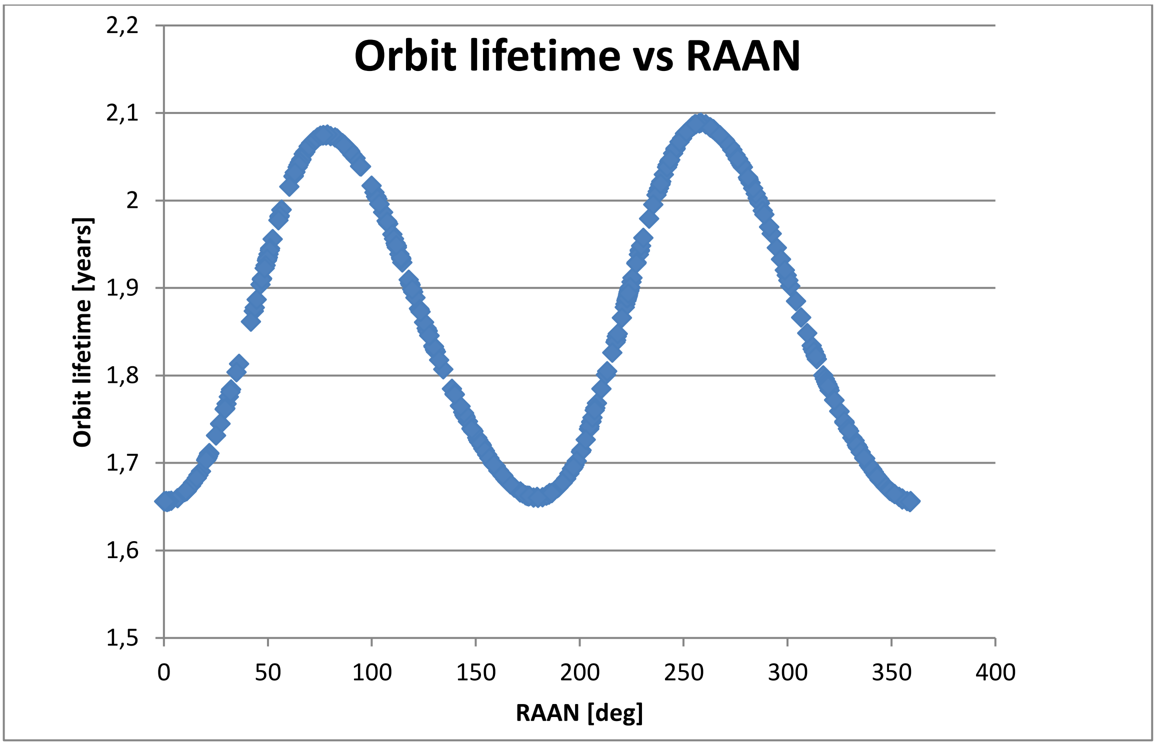

Figure C-2 Orbit lifetime vs RAAN

For nominal analysis parameters and RAAN dispersion between 0˚ and 360˚, orbit lifetime oscillates between 2.0890 and 1.6556 years, therefore the relative difference is 20.75 %

C.1.3 Mean Anomaly

Orbit lifetime variation with mean anomaly is presented on a Figure C-3.

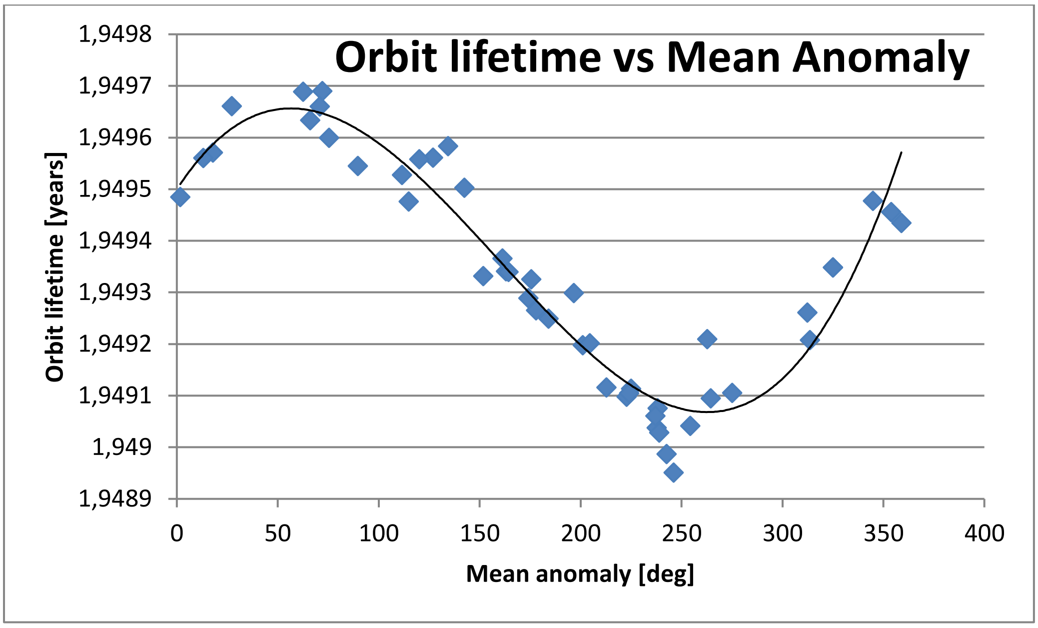

Figure C-3 Orbit lifetime vs mean anomaly

For nominal analysis parameters and mean anomaly dispersion between 0˚ and 360˚, orbit lifetime oscillates between 1.9497 and 1.99490 years, therefore the relative difference is 0.04 %

C.1.4 MASS

Orbit lifetime variation with mean anomaly is presented on a Figure C-4.

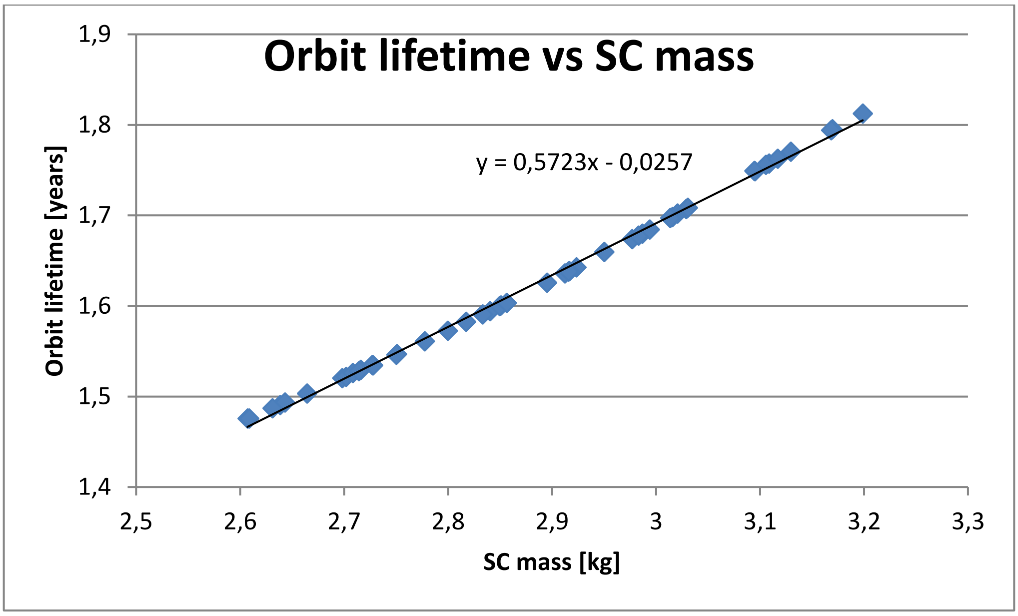

Figure C-4 Orbit lifetime vs SC mass

For nominal analysis parameters and SC mass dispersion between 2.6 and 3.2 it can be seen on Figure C-4 that orbit lifetime is linearly dependent on SC mass:

$$T_{orbit} = 0.5723 \left[\frac{years}{kg}\right] \times m_{SC} + 0.0257\ [years] \quad \textbf{(1)}$$

Where:

- T_{orbit} is orbit lifetime
- m_{SC} is spacecraft mass

For analyzed distribution range, the relative difference is 18.59 %

C.1.5 COEFFICIENT OF DRAG

Orbit lifetime variation with coefficient of drag (Cd) is presented on a Figure C-5

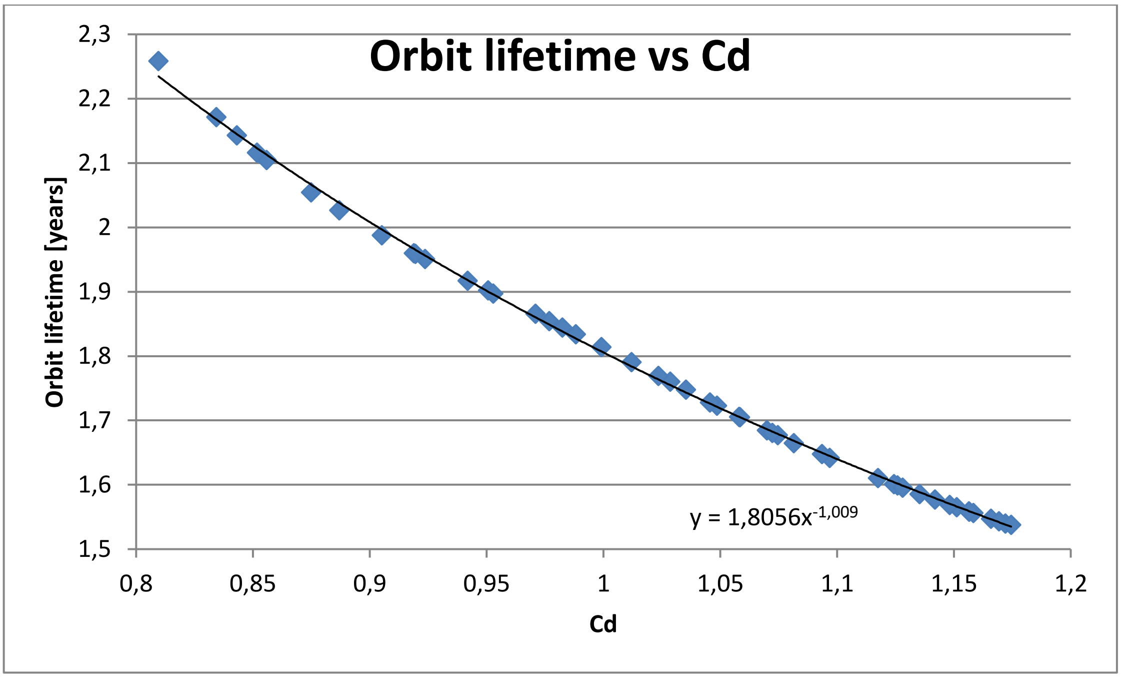

Figure C-5 Orbit lifetime vs Cd

For nominal analysis parameters and SC coefficient of drag dispersion between 0.8096 and 1.1745 it can be seen on Figure C-5 that orbit lifetime is power-function dependent on coefficient of drag:

$$T_{orbit} = 1.8056 \times C_d^{-1.009} \, [years] \quad \textbf{(2)}$$

Where:

- T_{orbit} is orbit lifetime
- C_d is coefficient of drag

For analyzed distribution range, the relative difference is 31.92 %

C.1.6 Reflectivity Coefficient

Orbit lifetime variation with mean anomaly is presented on Figure C-6:

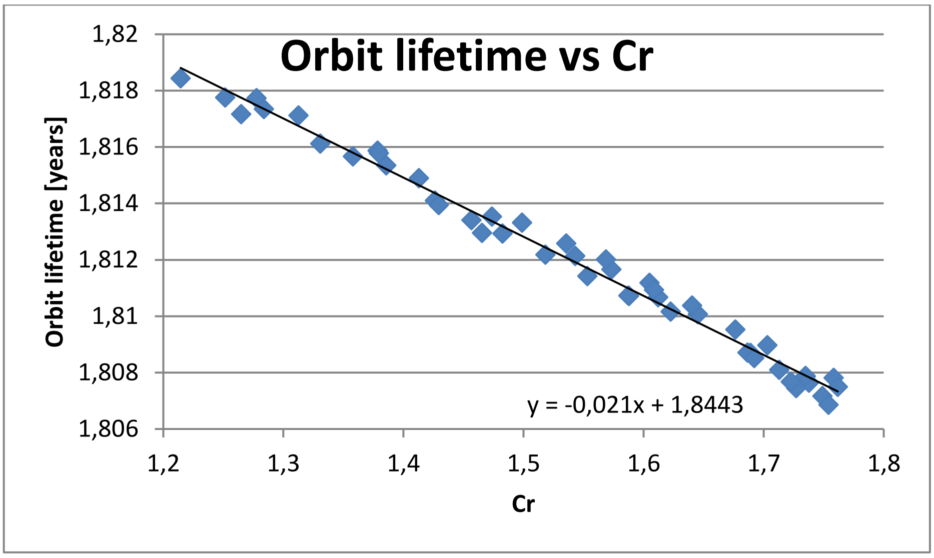

Figure C-6 Orbit lifetime vs Cr

For nominal analysis parameters and SC reflectivity coefficient dispersion between 1.2144 and 1.7617 it can be seen on Figure C-6 that orbit lifetime is linearly dependent on SC reflectivity coefficient:

$$T_{orbit} = -0.021[years] \times C_r + 1.8443\ [years] \qquad \textbf{(3)}$$

Where:

- T_{orbit} is orbit lifetime
- C_r is SC reflectivity coefficient

For analyzed distribution range, the relative difference is 0.64 %

C.1.7 Solar Activity, F10.7 Index

Orbit lifetime variation with F10.7 index is presented on Figure C-7

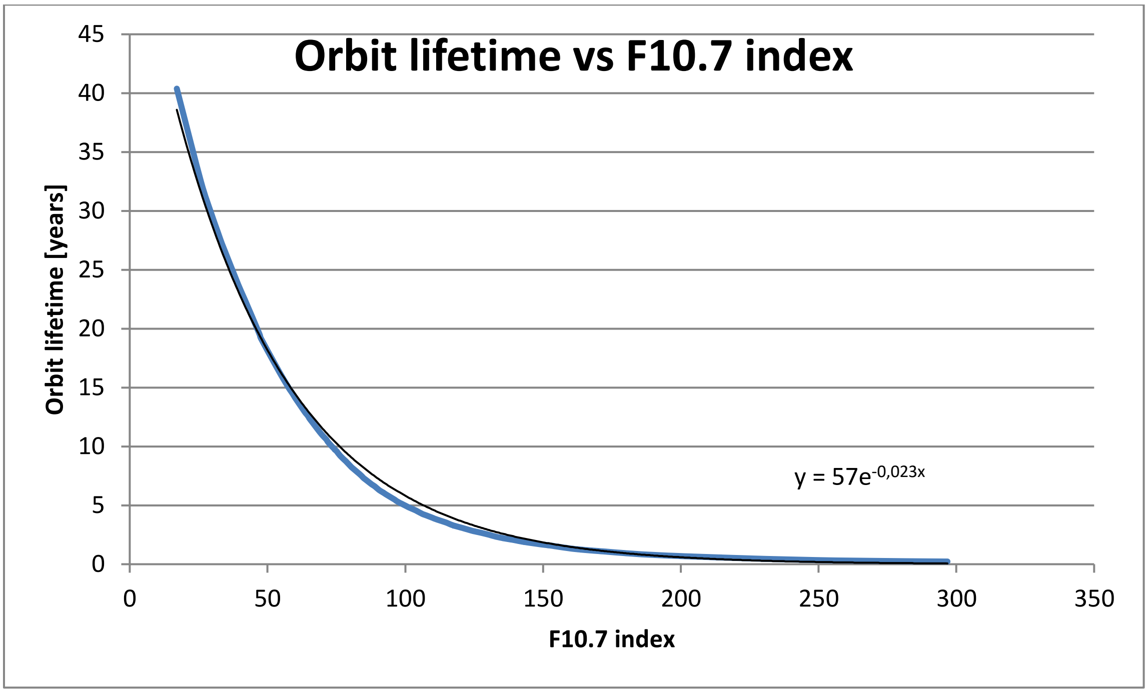

Figure C-7 Orbit lifetime variation with F10.7 index

For nominal analysis parameters and F10.7 index dispersion between 17.0793 and 296.7797 it can be seen on Figure C-7 that orbit lifetime is exponentially dependent on F10.7 index:

$$T_{orbit} = 57\times\exp(-0.023\times F_{10.7})\ [years] \quad \textbf{(4)}$$

Where:

- T_{orbit} is orbit lifetime
- $F_{10.7}$ is F10.7 index

For analyzed distribution range, the relative difference is 99.43 %

C.1.8 GEOMAGNETIC ACTIVITY, AP INDEX

Orbit lifetime variation with AP index is presented on Figure C-8

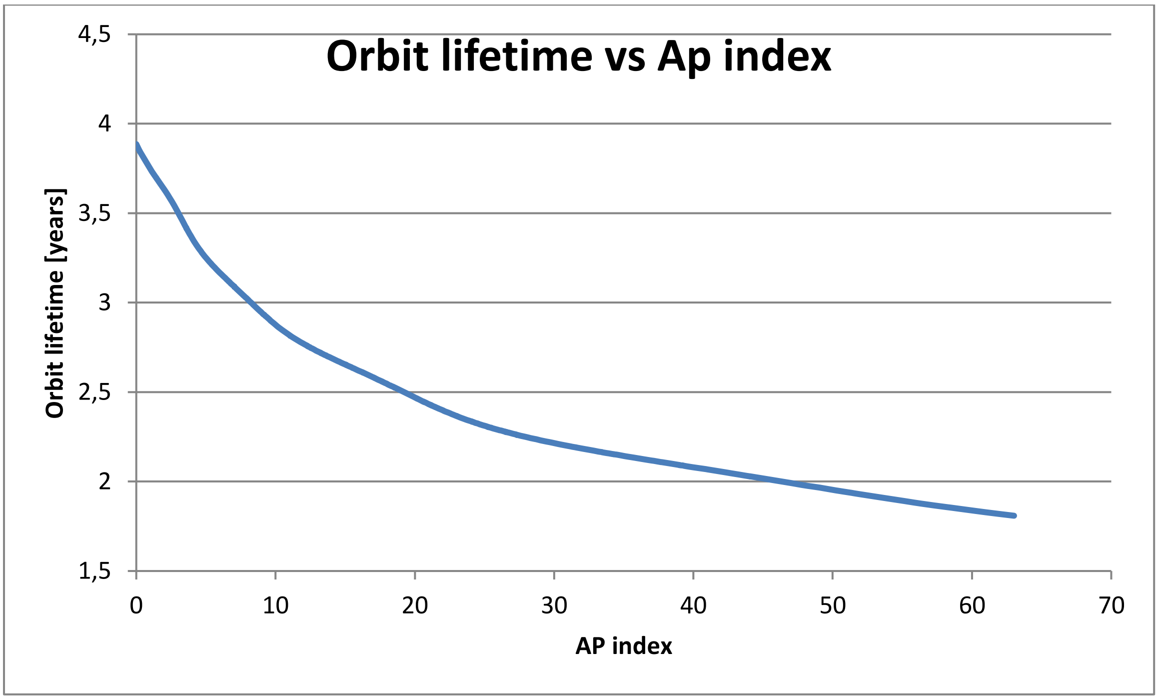

Figure C-8 Orbit lifetime variation with AP index

For nominal analysis parameters and AP index dispersion between 0 and 63.0196 it can be seen on Figure C-7 that orbit lifetime is exponentially dependent on AP index:

$$T_{orbit} = 2.2\times \exp(-0.048\times A_P) + 1.7 \quad [years] \tag{5}$$

Where:

- T_{orbit} is orbit lifetime
- A_P is AP index

For analyzed distribution range, the relative difference is 53.44 %

C.2 Summary of Single-Parameter Variation Analysis

Table C-3 summarizes the single-parameter variations results:

Table C-3 Summary of the single-parameter variation analyses

Distributed parameter	Dispersion range	Orbit lifetime dependency
Launch date	Uniform: (2016/13/02 - 2017/13/02)	Oscillation with relative difference of 19.81 %
RAAN	Uniform: 0˚-360˚	Oscillation with relative difference of 20.75 %
Mean anomaly (M)	Uniform: 0˚-360˚	Oscillation with elative difference of 0.04 %
Mass	Uniform: 2.6-3.2kg	Linear with relative difference of 18.59 %
Coefficient of drag (Cd)	Uniform: +/- 20 %	Power function, with relative difference of 31.92 %
Reflectivity coefficient (Cr)	Uniform: +/- 20 %	Linear with relative difference of 0.64 %
Solar activity, F10.7 index	Gaussian: $\sigma = 53.47$; $\bar{x} = 126.27$	Exponential with relative difference of 99.43 %
Geomagnetic activity, AP index	Gaussian: $\sigma = 15.51$; $\bar{x} = 13.56$	Exponential with relative difference of 53.44 %

Appendix D AOS/LOS Table

Table D-4 AOS/LOS Table

Contact	Start Time (Elapsed day)	Orbit	AOS Time (UTCG)	LOS Time (UTCG)	Duration (sec)
1	0.33189523	6	6 Dec 2017 07:57:55.749	6 Dec 2017 08:01:51.969	236.22
2	0.88079911	14	6 Dec 2017 21:08:21.044	6 Dec 2017 21:12:05.094	224.05
3	1.33486794	21	7 Dec 2017 08:02:12.591	7 Dec 2017 08:06:11.669	239.077
4	1.88375698	29	7 Dec 2017 21:12:36.604	7 Dec 2017 21:16:27.659	231.056
5	2.33785272	36	8 Dec 2017 08:06:30.476	8 Dec 2017 08:10:30.631	240.155
6	2.88672516	44	8 Dec 2017 21:16:53.054	8 Dec 2017 21:20:49.031	235.977
7	3.3408494	51	9 Dec 2017 08:10:49.390	9 Dec 2017 08:14:48.854	239.464
8	3.88970274	59	9 Dec 2017 21:21:10.318	9 Dec 2017 21:25:09.292	238.974
9	4.34385843	66	10 Dec 2017 08:15:09.369	10 Dec 2017 08:19:06.312	236.943
10	4.8926891	74	10 Dec 2017 21:25:28.339	10 Dec 2017 21:29:28.487	240.147
11	5.34688013	81	11 Dec 2017 08:19:30.444	11 Dec 2017 08:23:22.975	232.531
12	5.89568404	89	11 Dec 2017 21:29:47.102	11 Dec 2017 21:33:46.644	239.542
13	6.3499154	96	12 Dec 2017 08:23:52.691	12 Dec 2017 08:27:38.759	226.068
14	6.89868748	104	12 Dec 2017 21:34:06.600	12 Dec 2017 21:38:03.768	237.168
15	7.35296547	111	13 Dec 2017 08:28:16.218	13 Dec 2017 08:31:53.570	217.352
16	7.90169948	119	13 Dec 2017 21:38:26.836	13 Dec 2017 21:42:19.856	233.02
17	8.35603217	126	14 Dec 2017 08:32:41.180	14 Dec 2017 08:36:07.247	206.066
18	8.90472048	134	14 Dec 2017 21:42:47.851	14 Dec 2017 21:46:34.863	227.012
19	9.35911828	141	15 Dec 2017 08:37:07.820	15 Dec 2017 08:40:19.552	191.732
20	9.9077512	149	15 Dec 2017 21:47:09.705	15 Dec 2017 21:50:48.738	219.033
21	10.36222836	156	16 Dec 2017 08:41:36.532	16 Dec 2017 08:44:30.083	173.551
22	10.91079277	164	16 Dec 2017 21:51:32.496	16 Dec 2017 21:55:01.375	208.879

Contact	Start Time (Elapsed day)	Orbit	AOS Time (UTCG)	LOS Time (UTCG)	Duration (sec)
23	11.36537072	171	17 Dec 2017 08:46:08.031	17 Dec 2017 08:48:38.138	150.107
24	11.91384691	179	17 Dec 2017 21:55:56.374	17 Dec 2017 21:59:12.634	196.26
25	12.36856355	186	18 Dec 2017 08:50:43.892	18 Dec 2017 08:52:42.142	118.25
26	12.91691636	194	18 Dec 2017 22:00:21.575	18 Dec 2017 22:03:22.268	180.693
27	13.30555301	200	19 Dec 2017 07:19:59.781	19 Dec 2017 07:21:32.015	92.233
28	13.3718715	201	19 Dec 2017 08:55:29.699	19 Dec 2017 08:56:36.504	66.805
29	13.92000582	209	19 Dec 2017 22:04:48.504	19 Dec 2017 22:07:29.882	161.378
30	14.30831182	215	20 Dec 2017 07:23:58.142	20 Dec 2017 07:26:07.534	129.392
31	14.92312405	224	20 Dec 2017 22:09:17.919	20 Dec 2017 22:11:34.710	136.791
32	15.31113459	230	21 Dec 2017 07:28:02.029	21 Dec 2017 07:30:37.814	155.785
33	15.86045766	238	21 Dec 2017 20:39:03.543	21 Dec 2017 20:39:46.971	43.428
34	15.92629208	239	21 Dec 2017 22:13:51.637	21 Dec 2017 22:15:34.939	103.302
35	16.31399325	245	22 Dec 2017 07:32:09.018	22 Dec 2017 07:35:05.292	176.273
36	16.86309544	253	22 Dec 2017 20:42:51.447	22 Dec 2017 20:44:39.321	107.874
37	16.92960762	254	22 Dec 2017 22:18:38.099	22 Dec 2017 22:19:22.128	44.028
38	17.3168771	260	23 Dec 2017 07:36:18.182	23 Dec 2017 07:39:30.891	192.709
39	17.86590169	268	23 Dec 2017 20:46:53.907	23 Dec 2017 20:49:16.805	142.898
40	18.31978067	275	24 Dec 2017 07:40:29.051	24 Dec 2017 07:43:55.077	206.026
41	18.86876324	283	24 Dec 2017 20:51:01.145	24 Dec 2017 20:53:49.194	168.049
42	19.32270087	290	25 Dec 2017 07:44:41.357	25 Dec 2017 07:48:18.125	216.768
43	19.87165669	298	25 Dec 2017 20:55:11.139	25 Dec 2017 20:58:18.528	187.389
44	20.32563574	305	26 Dec 2017 07:48:54.929	26 Dec 2017 07:52:40.197	225.268
45	20.87457216	313	26 Dec 2017 20:59:23.036	26 Dec 2017 21:02:45.645	202.609
46	21.32858403	320	27 Dec 2017 07:53:09.662	27 Dec 2017 07:57:01.411	231.749
47	21.87750443	328	27 Dec 2017 21:03:36.384	27 Dec 2017 21:07:11.004	214.62
48	22.33154495	335	28 Dec 2017 07:57:25.484	28 Dec 2017 08:01:21.829	236.345

Contact	Start Time (Elapsed day)	Orbit	AOS Time (UTCG)	LOS Time (UTCG)	Duration (sec)
49	22.8804504	343	28 Dec 2017 21:07:50.916	28 Dec 2017 21:11:34.881	223.966
50	23.33451797	350	29 Dec 2017 08:01:42.354	29 Dec 2017 08:05:41.492	239.139
51	23.88340802	358	29 Dec 2017 21:12:06.454	29 Dec 2017 21:15:57.449	230.995
52	24.33750299	365	30 Dec 2017 08:06:00.259	30 Dec 2017 08:10:00.419	240.16
53	24.88637597	373	30 Dec 2017 21:16:22.885	30 Dec 2017 21:20:18.822	235.937
54	25.34050005	380	31 Dec 2017 08:10:19.205	31 Dec 2017 08:14:18.608	239.403
55	25.88935335	388	31 Dec 2017 21:20:40.131	31 Dec 2017 21:24:39.083	238.952
56	26.3435093	395	1 Jan 2018 08:14:39.205	1 Jan 2018 08:18:36.034	236.829
57	26.89233952	403	1 Jan 2018 21:24:58.136	1 Jan 2018 21:28:58.277	240.141
58	27.34653129	410	2 Jan 2018 08:19:00.305	2 Jan 2018 08:22:52.663	232.359
59	27.8953343	418	2 Jan 2018 21:29:16.884	2 Jan 2018 21:33:16.433	239.549
60	28.34956687	425	3 Jan 2018 08:23:22.578	3 Jan 2018 08:27:08.415	225.837
61	28.89833759	433	3 Jan 2018 21:33:36.369	3 Jan 2018 21:37:33.555	237.187
62	29.35261727	440	4 Jan 2018 08:27:46.133	4 Jan 2018 08:31:23.193	217.06
63	29.90134945	448	4 Jan 2018 21:37:56.594	4 Jan 2018 21:41:49.642	233.048
64	30.35568434	455	5 Jan 2018 08:32:11.128	5 Jan 2018 08:35:36.829	205.701
65	30.90437033	463	5 Jan 2018 21:42:17.597	5 Jan 2018 21:46:04.646	227.049
66	31.3587709	470	6 Jan 2018 08:36:37.807	6 Jan 2018 08:39:49.091	191.284
67	31.90740095	478	6 Jan 2018 21:46:39.443	6 Jan 2018 21:50:18.518	219.075
68	32.36188156	485	7 Jan 2018 08:41:06.568	7 Jan 2018 08:43:59.568	173
69	32.91044239	493	7 Jan 2018 21:51:02.223	7 Jan 2018 21:54:31.151	208.927
70	33.36502476	500	8 Jan 2018 08:45:38.140	8 Jan 2018 08:48:07.542	149.401
71	33.91349644	508	8 Jan 2018 21:55:26.093	8 Jan 2018 21:58:42.406	196.313
72	34.36821909	515	9 Jan 2018 08:50:14.130	9 Jan 2018 08:52:11.414	117.284
73	34.91656581	523	9 Jan 2018 21:59:51.287	9 Jan 2018 22:02:52.035	180.748
74	35.30519579	529	10 Jan 2018 07:19:28.917	10 Jan 2018 07:21:02.365	93.448

Contact	Start Time (Elapsed day)	Orbit	AOS Time (UTCG)	LOS Time (UTCG)	Duration (sec)
75	35.37153216	530	10 Jan 2018 08:55:00.380	10 Jan 2018 08:56:05.330	64.951
76	35.91965519	538	10 Jan 2018 22:04:18.210	10 Jan 2018 22:06:59.644	161.434
77	36.30795715	544	11 Jan 2018 07:23:27.499	11 Jan 2018 07:25:37.658	130.159
78	36.92277336	553	11 Jan 2018 22:08:47.619	11 Jan 2018 22:11:04.467	136.847
79	37.3107811	559	12 Jan 2018 07:27:31.488	12 Jan 2018 07:30:07.831	156.343
80	37.86010785	567	12 Jan 2018 20:38:33.319	12 Jan 2018 20:39:16.763	43.444
81	37.92594133	568	12 Jan 2018 22:13:21.332	12 Jan 2018 22:15:04.690	103.359
82	38.31364056	574	13 Jan 2018 07:31:38.546	13 Jan 2018 07:34:35.237	176.692
83	38.86274545	582	13 Jan 2018 20:42:21.208	13 Jan 2018 20:44:09.114	107.907
84	38.92925667	583	13 Jan 2018 22:18:07.778	13 Jan 2018 22:18:51.886	44.109
85	39.31652488	589	14 Jan 2018 07:35:47.750	14 Jan 2018 07:39:00.789	193.039
86	39.86555158	597	14 Jan 2018 20:46:23.657	14 Jan 2018 20:48:46.595	142.938
87	40.31942886	604	15 Jan 2018 07:39:58.654	15 Jan 2018 07:43:24.937	206.282
88	40.86841304	612	15 Jan 2018 20:50:30.888	15 Jan 2018 20:53:18.981	168.093
89	41.32234939	619	16 Jan 2018 07:44:10.989	16 Jan 2018 07:47:47.953	216.965