SYSML-MOTIVE: SYSML BASED SPACECRAFT POWER MANAGEMENT FUNCTION MODELING, TESTING, INTEGRATION, VERIFICATION AND EXECUTION

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ABSTRACT

Model based Systems Engineering (MBSE) is an immerging approach, which advocates a diagrammatical, verifiable and testable approach that enables communications amongst various domain experts, system of systems, hardware and software. This paper presents an MBSE approach to model the Power Management Function (PMF) of a spacecraft system in order to trade-off different strategies for optimizing power distribution of the vehicle.

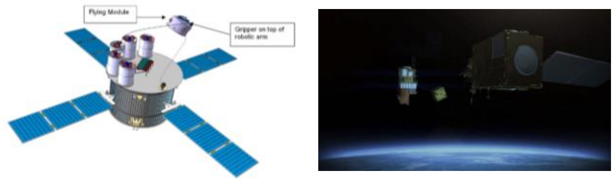

Figure 1 De-orbiter multi-mission concept [1]

The results presented in this paper are based on the outcomes of the SysML-MOTIVE[1] project for a debris removal mission scenario. This paper will present development of the PMF sub-system architectural and behavioural model to perform trade-off study between various key battery and solar cell technologies. The MBSE methodology presented here will be useful for evaluating the suitability of SysML based System Engineering approaches for future complex space missions.

1 INTRODUCTION

Modern day spacecraft Power Management Function (PMF) is an example of a complex scalable system involving system of systems and inter-connections of evolving sub-systems. The power budget for a spacecraft system plays a crucial role in phase-0/A mission analysis due to ever increasing power requirements covering payloads (PL), robotic elements, Electric propulsion, attitude and orbit control sub-system (AOCS) etc. The problems associated with designing the afore-mentioned sub-systems can be a non-trivial one. Hence, a typical phase-0/A study of a spacecraft PMF involves translating all customer and system requirements into detailed functional analysis, trade-off analysis, and concrete system architectural model i.e., with the view to carry out a detailed mission analysis of the system. To address this problem, current state of the art employs the Model-Based System Engineering (MBSE) approach as a means to alleviate this problem. Often, most MBSE approaches advocates a diagrammatical, verifiable and testable approach that enables communications amongst various domain experts, system of systems, hardware and software. This study has developed an MBSE approach to model the PMF of a spacecraft system in order to trade different strategies for optimizing power distribution of the vehicle. This study has modelled a top-level power subsystem with focus on trade-off analysis of solar cell and battery technologies for satellite servicing and debris removal mission scenarios like in [1] and Fig.1.

2 SysML-SEP: SySML BASED SYSTEMS ENGINEERING PROCESS FRAMEWORK

Across product development lifecycle, one crucial area where there has been a serious lack of a commonly acceptable language support is the conceptual stage, during which the functional architecture (and sometimes the physical architecture) is decided upon. It is well understood that this lack of support during product conceptualisation will make it extremely difficult to efficiently trace the realisation of requirements in the product.

In addition to this, lack of a formal representation for concepts hampers the ability to make important decisions at the level of systems in the product i.e. during feasibility studies. Beyond that, the lack of a clear vision of the product architecture amongst the various system stakeholders hinders team understanding and communication. Thus, this in turn increases the risk of integration issues. This paper presents one of such SysML-SEP approaches as shown in Fig.2, for supporting phase 0/A of a robotic spacecraft developments.

[1] SysML-MOTIVE study is sponsored by AIRBUS Defence and Space, Les Mureaux, France

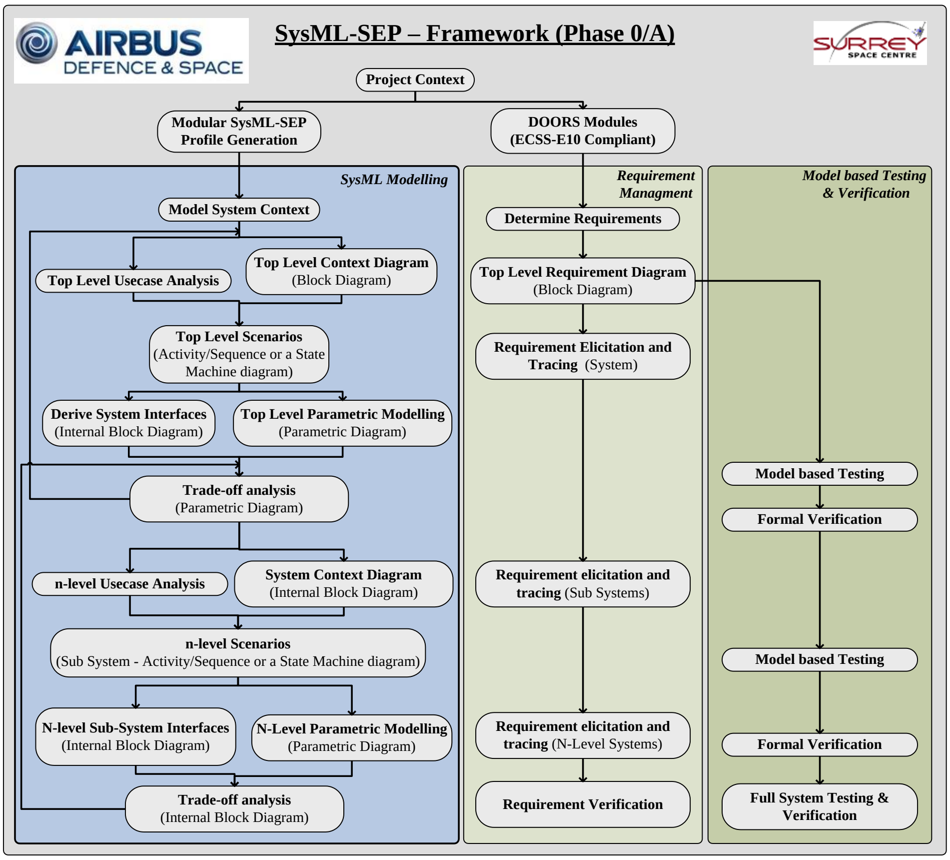

Figure 2 SysML-SEP framework

Arguably, it is these and other non-trivial challenges encountered during the conceptual phase of product and system development lifecycle that the System Modelling Language (SysML) and Model Based System Engineering (MBSE) approaches are designed to mitigate.

The proposed approach is a case of special iteratively incremental system engineering processes, which enables systems engineer to develop functional and physical systems through a tiered model development. This approach is designed in such a way that it is modular and scalable so that it could be applied for the Phase B and following stages of any typical space missions. This approach is a result of lesson learned and best practices that evolved from previous projects [2] [3] [4] [5] [6]. There are three principle activities as part of the SysML-SEP approach: SysML models, requirement management and model based testing, validation & verification that runs in parallel as shown in the Fig.2.
A first step in the SysML-SEP framework is to setup and generate a modular package structure by importing custom profile developed for this project. There are three reasons for having such modular package structure. The first reason is due to the fact that SysML is a semi-formal language and would require defining strict modeling guidelines as described by [3] and [4]. Secondly to support co-development by multiple teams at a same time and finally to aid iterative and tiered approach of product development. More detailed information about the SysML-SEP modular package structure and lifecycle is illustrated in [3] [3].

The following sections will illustrate the remaining steps of the SysML-SEP framework using a spacecraft PMF sub-system modelling as part of the SysML-MOTIVE project.

3 SysML-MOTIVE: SYSTEMS ENGINEERING OF SPACECRAFT PMF USING SysML-SEP

The orbital range of the spacecraft considered in this study is much wider than a classical LEO mission. Hence, this put extra emphasis for higher adaptability of all subsystems and especially on PMF sub-system. This is achieved by architectural, behavioural and parametric modeling of the spacecraft PMF using SysML-SEP.

3.1 Stockholder requirements and modelling system context

The top-level requirement folder within the SysML-SEP package structure contains all the requirements (stockholder requirements) which will drive the analysis and system functional modelling. Requirements are structured based on ECSS-E-ST-10-06C which is a part of the ECSS standard for technical requirements and specification. These top level stakeholder requirements usually imported from the requirement management software, like DOORS in this SysML-MOTIVE project. Typically, DOORS maintains project documents, user documents, and documentation of changes. System specification, requirement analysis and modeling are performed within Rational Rhapsody. IBM Rhapsody Gateway is being used to import and sync these stakeholder requirements to the SysML model within a top-level requirement package as mentioned in [3] and Fig.3, 4. The Rhapsody Gateway imparts the benefits of a seamless bi-directional information exchange interface with 3rd party requirements and authoring tools to extend a complete traceability solution, which allows developers to examine the upstream and downstream impact of requirements changes, in real time, at any level of iterations.

The next step in this process is to create requirement views showing important traceability links between requirements using a top-level requirement diagrams.
In parallel to the requirement diagram and management process at the top-level, another important step is to identify and create a system context diagram. This is based on the SysML block definition diagram (BDD) and the internal block definition diagrams (IBD). The top-level context diagrams help in identifying system boundaries and actors outside of the boundaries that could interact with the system under development (SUD). This diagram provides a highest-level view of a system. The purpose of the system context diagram realised through a BDD is to describe the system hierarchy and system/component classifications. One of such BDD system context diagram is shown in the Fig 5. The aim of the system context diagram realised using the IBD is to specify the interface (messages) to external systems in the form of item flows. These item flows and messages usually are abstract at this level of functional analysis. The idea is to map these abstract flows and messages to concrete message types during the next level of analysis.

3.2 Top level usecase analysis and PMF context diagram

Use case diagrams define the various scenarios that a system can have. The use cases give the basic idea of how the system would act under different conditions and which stakeholders are responsible for a particular use case. Use case analysis forms basis of system design and system behaviour. For each use case it is important to identify prime actor, trigger, preconditions, nominal scenario scenarios and any contingency scenarios if exist. It is recognized that the first level use case is a driving factor for system functional breakdown and also for development of package structure. The resulting first iteration of the context diagram of the SysML-MOTIVE PMF sub-system based on the use case analysis is shown in Fig.5(b). It shows the PMF sub-system and associated functional breakdown in the form of SysML blocks. Each block represents the basic unit of structure in SysML.

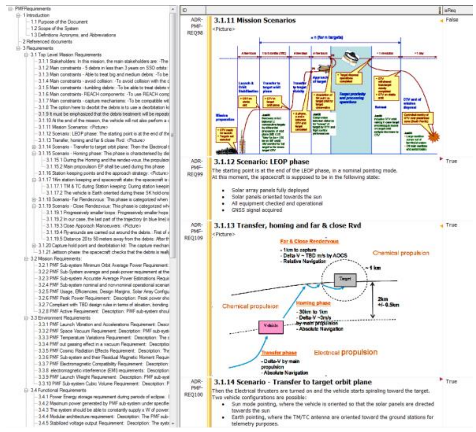

Figure 3 DOORS requirement database snapshot1

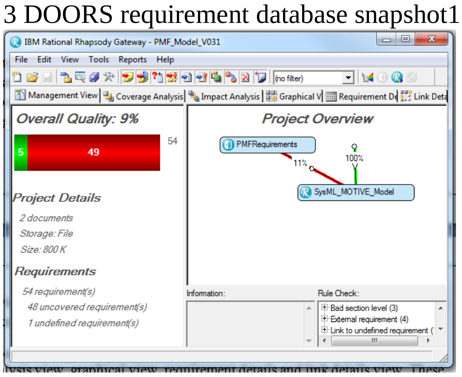

Figure 4 Management view - Rhapsody Gateway

3.3 Top level system scenarios

Top-level system scenarios are used to represent two main behavioural aspects of the SUD, one to show operational modes and secondly system scenarios. The system may have a number of operational modes perceived by stakeholders that can be modelled using a state machine diagram as shown in Fig.6 (a). The activity and sequence diagrams are used to analyses the expected usage of the system based on identified pre-condition, post condition, scenario steps during the use case analysis. Each use cases usually results in at least 5/6 top level scenarios at the first level of iteration of such complex system.

3.4 Identification of system interfaces

Data modelling and identification of interfaces are also important part of requirement analysis. Top level interfaces (e.g. messages to/from the system) should be consistent with the data model for each of the subsystems (e.g. decomposition of messages).

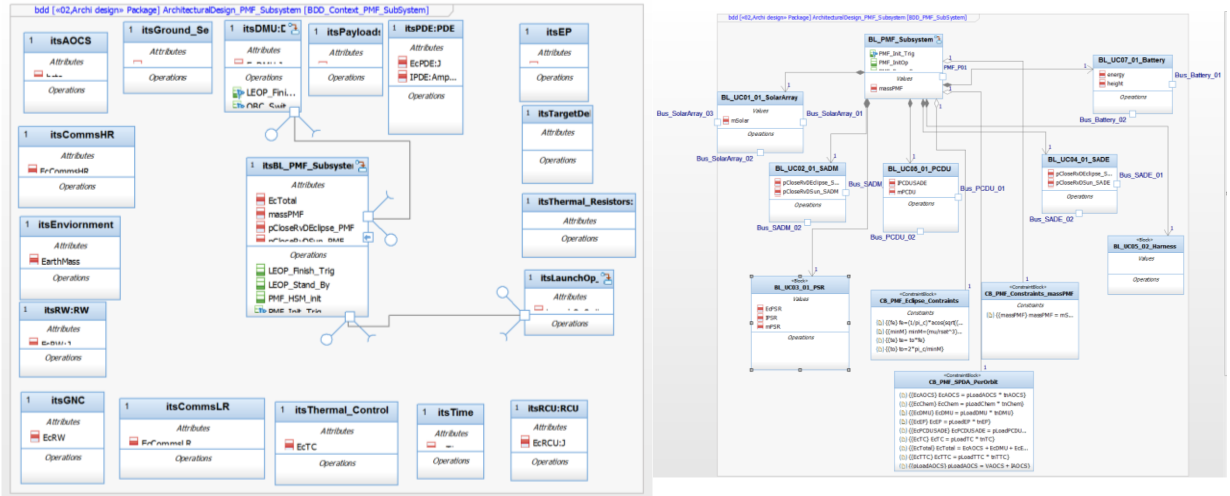

Figure 5 (a) System context diagram (b) Top level context diagram (1st iteration based on the USE Case Analysis)

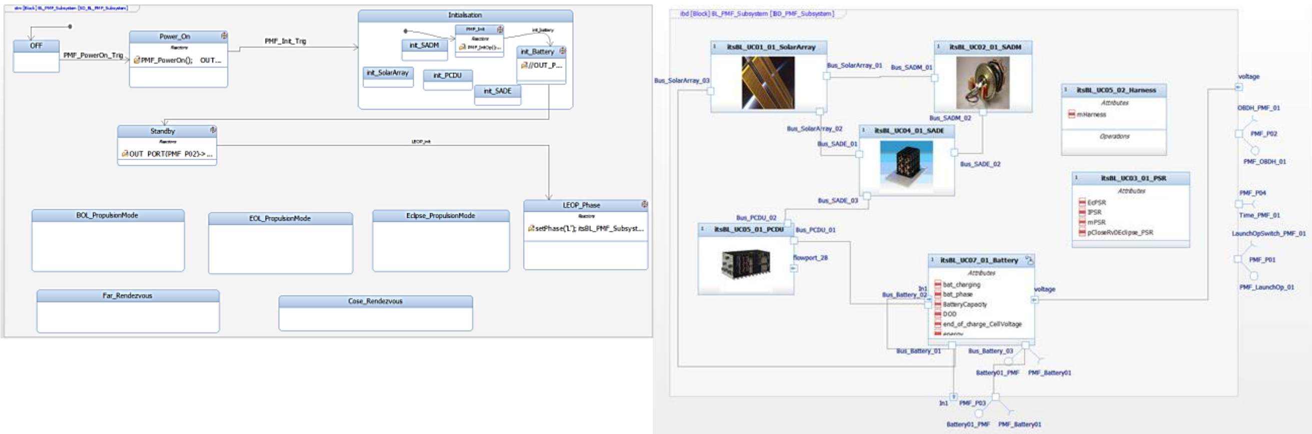

Figure 6 (a) Operational modes (b) Internal block definition diagram of the PMF sub-system

The requirement analysis starts with an identification of systems context and specification of the interfaces. This identified abstract messages and interfaces usually are between actors and SUD. Whereas the context diagram for the SUD and behaviour analysis helps in determining abstract interfaces between functional decomposition of SUD. The interfaces between parts of PMF subsystem are shown on IBD. Ports can be defined on a block or part to specify interfaces on the blocks. Whilst the port provides a structural feature, it also enables the connection to the block to be specified in more complex ways such as specifying the interface (e.g. provided and required operations) and the associated behaviour within the port. This is achieved by giving the port a type and grouping similar type of messages.

3.5 Top level parametric modeling

Understanding and analysing of requirements associated with orbital parameters, power requirement for sub-systems and data on past missions help develop a compressive model of system constraints. Mass, volume and cost are key parameters that drive the spacecraft system engineering as well. Debris removal is a complex mission. In order for this mission to be successfully stabilizing the space environment from debris and prove to be a financially viable, spacecraft needs to remove at least five large debris per year. This has direct implication on PMF sub-system architecture and behavioural design. The spacecraft altitude is directly proportional to orbit period and eclipse time. This does put extra demand on the battery discharge and the charge cycle and available average orbit power. Based on literature, the power system mass as a percentage of the satellite dry mass can range from 25% in LEO satellites to 45% in GEO satellites. Power and energy and mass budgets for the mission have been iteratively updated based on parametric analysis. These incremental changes to the power, energy and mass constraints have been incorporated in the SysML model using parametric diagrams. Parametric diagrams show mathematical relationships (such as performance constraints) among the pieces of the system being designed. The Parametric Constraint Evaluator solves equations for sets of attribute values, taking into account the constraints that are defined. In this section, one of such example for eclipse constraint on spacecraft is presented here.

3.5.1 Orbital parameters and eclipse constraints

LEO orbits are usually circular orbits at 300-9000 km altitude (>200 km to avoid large drag and <1000km to avoid van Allen belts radiation). Orbit period for such orbit is roughly between 90 to 100mins. The main orbit parameter relevant for PMF is inclination to the equatorial plane, which governs possible eclipse periods. This is explained here and captured in the SysML model through a parametric diagram as shown in Fig 7.

T_e = Eclipse period	f_e =Eclipse fraction
T_o = Orbit period	h = Altitude of satellite
R = Radius of Earth	β = Angle between the sun and orbit plane
$\mu = G*M$ = 3.986 x 10^5 km^3/s^2	G = gravitational constant
M = Mass of the more massive body (earth)	a = orbit's semi-major axis

$$\frac{T_e}{T_o} = f_E = \frac{1}{\pi}\cos^{-1}\left[\frac{\sqrt{h^2+2Rh}}{(R+h)\cos\beta}\right]$$ Unit: dimension less

Since the maximum eclipse fraction is required for this study, the angle β, is assumed to be equal to zero.

$$(f_E)_{\max} = \frac{1}{\pi}\cos^{-1}\left[\frac{\sqrt{h^2+2Rh}}{(R+h)}\right]$$

Mean motion = $\sqrt{\frac{\mu}{a^3}} = \sqrt{\frac{\mu}{r^3}}$ $km^3S^{-2} * Km^{-3} = S^{-2}$

$$T_o = 2\pi\sqrt{\frac{a^3}{\mu}} = 2\pi\sqrt{\frac{r^3}{\mu}} = S^{-1}$$

$$T_{e_{\max}} = P*(f_E)_{\max}$$

4 BATTERY SUB-SYSTEM AND TRADE-OFF STUDIES

Selection of the optimum battery for space applications results in a safe, effective, efficient, and economical power storage capability. The optimum battery also enhances launch operations, minimizes impacts to resources, supports contingency operations, and meets demand loads. For the application under consideration, a typical battery system design should consider these requirements battery capacity, depth of discharge (DOD), state of charge (SOC), mean and end of charge cell voltage, re-chargeability, numbers of cells and their configurations, weight, specific energy and transmission efficiency.

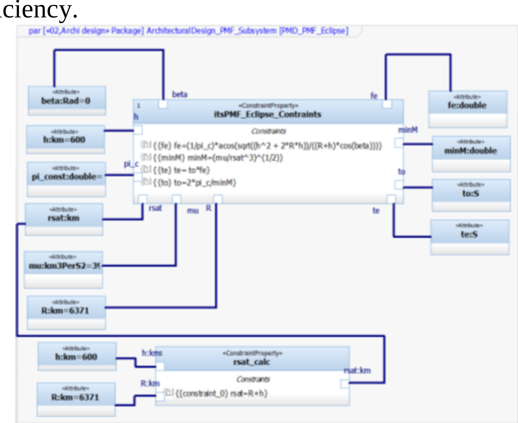

Figure 7 Constraints posed by eclipse is captured in this parametric diagram

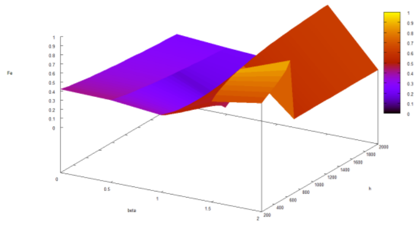
Figure 8 Plot of *beta* angle, satellite altitude *h* and eclipse fraction *Fe*

These parameters have been captured as attributes in the battery SysML block. The Fig 9 shows three different battery technologies in form of Batter1, Battery2 and Batter3 connected to the Battery block BL_UC07_01_Battery using the generalisation relationship. The generalisation relationship indicates that one of the two related blocks is considered to be a specialised form of the other and superblock is considered as generalisation of sub-block. Typically, large spacecraft runs on 50V-120V bus, which does require numbers of cells either in series and parallel arrangements. With so many cells in series, the possibility of one failing is real. One open cell would break the circuit and a shorted one would lower the overall voltage. Cell matching has always been a challenge.

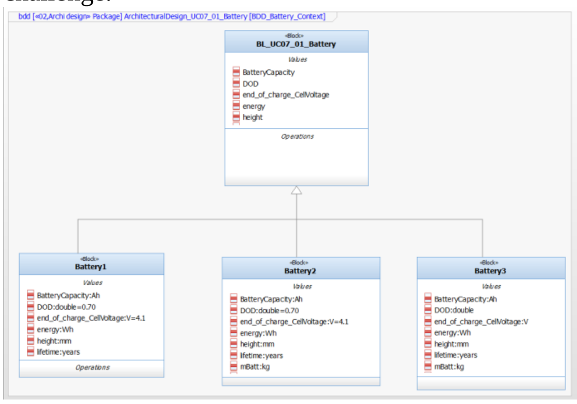

Figure 9 Context diagram of a battery sub-system

4.1 Battery modeling and Trade-off Studies

Battery sizing can be one of the more complex and important calculations in S/C system design. It is important to estimate accurate battery bank size, if the battery bank is oversized, it could pose challenge to keep it fully charged; if battery bank is sized too small, it won't be able to run your intended loads for as long as it is being planned. Trade-off analysis and results for two different battery technologies are presented in the Table 1. The mathematical model is presented below and captured using parametric diagrams Fig. 9.

- Watt Hours of electricity usage per orbit and eclipse requirements
- Estimated depth of discharge limit

$$n_{cell} = \left(\frac{V_{db} + V_{dd}}{V_d} \right) + 1$$

n_{cell} - Number of battery cells

V_{db} - Minimum battery discharge voltage (bus voltage)

V_{dd} - Bypass diode voltage over the failed cell

V_d - End of charge cell voltage

$$C_r = \frac{P_e * T_e}{(DOD) * N * n * V_{db}} \qquad A-hr$$

$$C_r = \frac{P_e T_e}{(DOD) * N * n} \qquad W-hr$$

C_r - Total S/C battery capacity

P_e - Average eclipse load (watts)

T_e - Eclipse duration (hr)

DOD - Depth of discharge (0 ≤ DoD ≤ 1)

N - Number of batteries (need at least two if want some partial redundancy)

n - Transmission efficiency between battery and load (typical value is 0.9)

$$m_{batt} = \left(\frac{C_r * V_{db}}{SpE \dfrac{(W*hr)}{kg}} \right) N$$

m_{batt} - Mass of batteries

SpE - Specific energy

5 SOLAR ARRAY SUB-SYSTEM AND TRADE-OFF STUDIES

Solar cells for space applications have to be highly efficient, capable to stand thousands of thermal cycles in orbit where the temperature, according to the mission profile may vary from -150 °C to more than 120 °C. They have to show a limited degradation during time due to cosmic radiations and ultraviolet, and they have to resist to the mechanical solicitations mainly linear accelerations and vibrations during launch and orbital manoeuvres because of these constraints the cells for space are smaller than those for terrestrial applications.

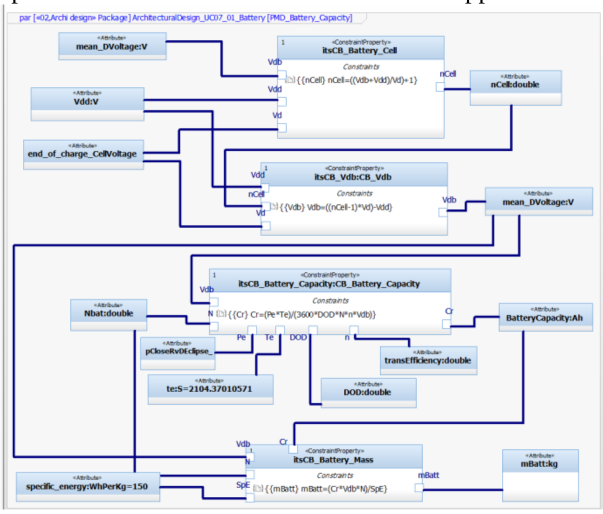

Figure 10 Parametric diagram for the battery capacity assessment

Table 1 Performance trade-off study between two types of batteries

Variables	*Units/T*	*Battery 1*	*Battery2*
BatteryCapacity	Ah	21.97155	21.97155
DOD	double	0	0
transEfficiency	double	0	0
nCell (numbers of	double	14.19512	22.28
T_e	S	2104	2104
V_{dd}	V	4	3
V_{db}	V	50	50
V_d	V	4	2
pCloseRvDEclipse _PMF	W	1184	1184
N	double	1	1
Mass of Battery	kg	103.9629	250.0133
specific_energy	WhPer	150	97
Estimated mass of	kg	16.04048	116.5244
Mass of a cell	kg	1	5

In the past silicon was the most used solar cell material and the reachable bulk efficiency was not higher than 14%. The advent of GaAs based solar cells in the last decade of the 20th century took the efficiency up to 19%, and nowadays triple and multi junction solar cells show more than 30%. Triple junction GaAs solar cells are populating more and more solar generators worldwide, while manufacturers are actively working on four to six junction cells as a way forward always increasing conversion efficiency.

The starting point for the solar array sizing is the correct identification of the power demand throughout the whole mission of the spacecraft. Such power demand may change during the satellite lifetime either because of different operational modes foreseen during the mission or, more simply, because of degradation of the electrical performances of the electrical loads. Due to solar eclipses, and possible pointing error along the orbit, an analysis of the energy demand is recalculated. In case of insufficient illumination, the on-board battery will supply the electrical power. These different power demand is modelled through various nominal and non-nominal scenarios. Comparative performance metrics (of Specific power (W/kg), power density (W/m2), specific mass (kg/m2), and specific cost ($/W).) can help study and compare different solar cell and array technologies for the mission scenarios. The ultimate purpose of this trade-off study is to maximize performance, while minimizing cost.

5.1 Solar array sizing

The solar array sizing is done for the spacecraft at end of mission life (EOL) power requirement using standard solar array sizing procedures and by considering all the factors. The solar array size computed by using following Eqs. (1.5)– (1.12).

D_{AM} - Solar cell assembly and mismatch loss factor

D_C - Calibration loss factor

D_{fb} - Solar array fabrication loss factor

D_{fi} - Solar array flight loss factor

D_{HD} - Harness and diode loss factor

D_i - Solar intensity factor

D_{MO} - Micrometeorites/orbital debris loss factor

D_{OP} - Solar array operational loss factor

D_R - Solar cell degradation factor

D_{RA} - Random loss factor

D_T - Temperature degradation factor

SCA - Solar cell assembly area

D_{UV} - UV degradation factor

D_V - Solar cell offset voltage

P_{cell} - Solar cell power

P_f - Packing factor

P_{SA} - Solar array power

T_{SA} - Temperature of the cells in orbit

t_{SA} - Test Temperature of the cell

$P_{SA}Eol$ - Solar array power at EOL

$P_{cell}Eol$ - Solar cell power at EOL

$P_{cell}Bol$ - Solar cell power at BOL

D_{tc} - Thermal cycling loss factor

$$\mathrm{D_{fi}} = \mathrm{D_{uv}} \times \mathrm{D_{MO}} \times \mathrm{D_{RA}} \times \mathrm{D_{tc}} \tag{1}$$

$$\mathrm{D_T} = 1 - 0.005(\mathrm{T_{SA}} - \mathrm{t_{SA}}) \tag{2}$$

$$Di = \left(\frac{149.6 \times 10^6}{(149.6 \times 10^6 - a)} \right)^2 \tag{3}$$

$$\mathrm{D_{op}} = \mathrm{D_i} \times \mathrm{D_T} \times \mathrm{D_v} \tag{4}$$

$$\mathrm{D_{fb}} = \mathrm{D_{AM}} \times \mathrm{D_c} \times \mathrm{D_{HD}} \tag{5}$$

$$\mathrm{P_{cell}Bol} = \mathrm{I_{Sun}} \times \eta_{\mathrm{SA}} \times \mathrm{SCA} \tag{6}$$

$$\mathrm{P_{sa}Eol} = \mathrm{P_{cell}Eol} \times \mathrm{N_{cell}} \tag{7}$$

$$\mathrm{SA\_Size} = \frac{(\mathrm{P_{sa}Eol} \times \mathrm{SCA})}{(\mathrm{P_{cell}Eol} \times \mathrm{P_f})} \tag{8}$$

The trade-off analysis results for two different scenarios and for three different solar array technologies are presented in the Table 2.3. The solar array power range chosen is up to 31 KW. Along with these two scenarios for SA sizing, the analysis is also done for mass and costs constraints for solar array trade-offs.

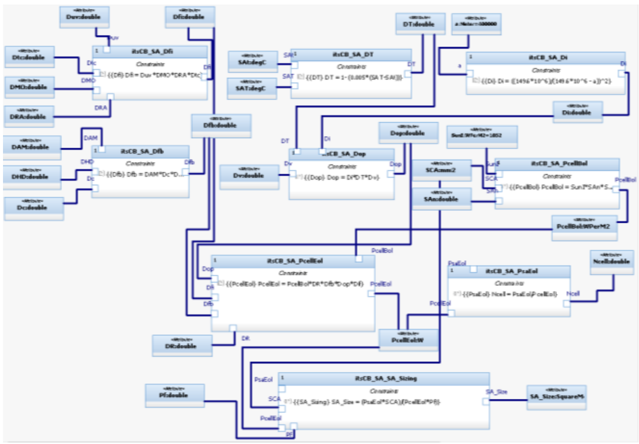

Figure 11 Parametric diagram of the solar array sizing

Table 2 Performance trade-off study between two types of Solar Array technologies – case 1 (SA1 – Thin film CIGS, SA2 – GaInP2/GaAs/Ge MJ, SA3 – Multi-junction refractive concentrator)

Variables	*Units/Type*	*SA 1*	*SA2*	*SA3*
Solar cell assembly area	m^2	0.0024	0.0024	$5.2x10^{-6}$
Solar cell efficiency	%	18.5	28	44
Solar cell power at BOL	W/m^2	0.467	0.7069	1.110
Solar cell power at EOL	W/m^2	0.2205	0.3228	0.322
Numbers of cells	-	137964.17	94234.46	94400.575
Temperature of cells in orbit	deg C	75	80	90
Temperature of cells at which cells were tested	deg C	28	28	28
Solar array size	m^2	367.90	251.29	251.734
Current largest solar array panel size	m^2	-NA-	3.92 x 2.28	3.92 x 2.28
Numbers of panel required		-NA-	28	28
Numbers of wings		-NA-	2	2
Key input parameters				
Semi-major axis (a)	400Km			
Solar Illumination	1052			
End of life power requirement (PsaEol)	30426.5			

Table 3 Performance trade-off study between two types of Solar Array technologies- case 2 (SA1 – Thin film CIGS, SA2 – GaInP2/GaAs/Ge MJ, SA3 – Multi-junction refractive concentrator)

Variables	*Units/Type*	*SA 1*	*SA2*	*SA3*
Solar cell assembly area	m^2	0.0024	0.0024	0.0024
Solar cell efficiency	%	18.5	28	44
Solar cell power at BOL	W/m^2	0.6069	0.918624	1.443
Solar cell power at EOL	W/m^2	0.2865	0.4195	0.42
Numbers of cells	-	106172.87	72519.86	72647.69
Temperature of cells in orbit	deg C	75	80	90
Temperature of cells at which cells were tested	deg C	28	28	28
Solar array size	m^2	283.127	193.386	193.727
Current largest solar array panel size	m^2	-NA-	3.92 x 2.28	3.92 x 2.28
Numbers of panel required		-NA-	22	22
Numbers of wings		-NA-	2	2
Key input parameters				
Semi-major axis (a)	400Km			
Solar Illumination	1367 W/m^2			
End of life power requirement (PsaEol)	30426.5			

5.2 Model based testing and requirement elicitation

Model based tests are added in order to ensure that the model indeed correctly captures the requirements. Behaviour scenarios developed during use case analysis phase are used as templet for definition of test cases. IBM Rhapsody MDT coverage metrics (requirements coverage and model coverage) guarantees the completeness of the model based test suite. Automatic code generation is used to generate an implementation from the model. Back-to-back testing between model and code constitute the key element for code verification. Code coverage metrics are used in order to ensure completeness of the test suite with regard to the predefined code coverage criteria. This MDT workflow provides opportunity to test system under test early in life-cycle of product development.

The end of analysis phase will result in derived requirements for the sub-systems. These derived requirements will then be starting point for the next phase of iteration.

6 CONCLUSION AND FUTURE WORK

SysML based SE can deliver the functional specifications, parametric studies, behavioural understanding, and structure definitions of any complex systems and systems of system. Using modeling and simulation methods, systems engineers can trace the stakeholder requirements and prototype the system well in advance. These strategies together can enhance communication and improve system quality of any complex project. It permits reuse of system specifications and model elements. This all confirms that SysML modeling is a perfect and valid approach in space engineering. Using it in pre-implementation phase, will identify any problems early in the project and hence will save time and money.

This project has primarily shown following key capabilities:

1. In this project, a SysML model of satellite servicing scenario for the Power management function sub-system was successfully developed. Parametric studies taking into account power consumption profile coming from mission analysis.
2. Trade-off studies providing comparison between different architecture and different technologies.
3. Test cases for nominal / non-nominal situation in order to assess robustness of the selected solution.
4. Identification and elicitation of main requirements of the power subsystem elements (solar panels, batteries, and propulsion drive units) was identified.

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