DESIGN OF A REUSABLE SPACE TUG

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Abstract. *The paper deals with the conceptual design of a space tug to be used in support to Earth satellites transfer manoeuvres. Usually Earth satellites are released in a non-definitive low orbit, depending on the adopted launcher, and they need to be equipped with an adequate propulsion system able to perform the transfer to their final mission orbit.*

In order to reduce the mass at launch of the satellite system, an element pre-deployed on orbit, i.e. the space tug, can be exploited to perform the transfer manoeuvres; this allows simplifying the propulsion requirements for the satellite, with a consequent decrease of mass and volume, in favour of larger payloads. The space tug here presented is conceived to be used for the transfer of a few satellites from low to high orbits, and vice versa, if needed. To support these manoeuvres, dedicated refuelling operations are envisaged.

The paper starts from the mission statement and mission objectives and proceeds with the derivation of the requirements through the application of functional analysis and concept of *operations. This approach is applied both at System of Systems and at System level and leads to the identification of the architecture and operations at both levels. Then the paper focuses on the detailed definition of the space tug, from the subsystems' assessment up to the budgets' development, through an iterative and recursive design process. The overall mission scenario has been derived from a set of trade-off analyses that have been performed to choose the mission architecture and operations that better satisfy stakeholder expectations.*

Eventually, in the last part of the work, main conclusions are drawn on the selected mission scenario and space tug and further utilizations of this innovative system in the frame of future space exploration are discussed. Specifically, an enhanced version of the space tug that has been described in the paper could be used to support on orbit assembly of large spacecraft for distant and long exploration missions.

The Space Tug development is an activity carried on in the frame of the SAPERE project (Space Advanced Project Excellence in Research and Enterprise), supported by Italian Ministry of Research and University (MIUR), and specifically in its STRONG sub-project (Systems Technology and Research National Global Operations), and related to the theme of space exploration and access to space.

1 INTRODUCTION

A space tug is a particular kind of spacecraft used to transfer payloads from Low Earth Orbit (LEO) to higher operational orbits, increasing the payload mass. In recent years, many space agencies are showing a larger interest in space tug systems concept. This interest is reflected in most of the international roadmaps and it is due to the various applications in which this system can be exploited for. Indeed, besides the aspects related to orbital payload transfer, an important issue to be faced is related to the fact that space environment is becoming more and more crowded. In this regard, it is important to develop capabilities for on-orbit maintenance of satellites and refuelling operations, as well as retrieving or removing space debris. In addition, such a system can be exploited for on orbit assembly of large spacecraft, which indeed represent a crucial point for space exploration in the future. In this framework, the development of a new element like a Space Tug is desirable [1].

The space tug discussed in this paper has the main purpose to support satellites deployment on orbit. Usually Earth satellites are released in a non-definitive low orbit, depending on the adopted launcher. This solution involves the use of a propulsion system able to perform the transfer of the payload to their final operational orbit. The use of a reusable tug system with an adequate propulsion system to be docked to the payload would be not only a way to increase the payload mass, avoiding a dedicated propulsion system, but also an attractive solution to improve the market position of the Italian VEGA launcher. Indeed, reducing the mass in the satellite dedicated to the payload transfer, a larger amount of the mass available on a launcher, even a small launcher as VEGA, can be exploited for the payload.

Furthermore, orbital transfer is not the only objective that can be pursued using the space tug. For example, the opportunity of retrieving on Earth significant payload samples/parts by means of an operative reusable vehicle, such as for example an evolution of IXV

(Intermediate eXperimental Vehicle), PRIDE, is considered. For this purpose, a suitable rendezvous in an intermediate orbit between the space tug and the vehicle during one of its operative mission phases is to be envisioned, in order to allow the transfer of the payloads to be re-entered on Earth.

The Space Tug design, described in this paper is an activity carried on in the frame of the SAPERE project and specifically in its STRONG sub-project, related to the theme of space exploration and access to space. This project has the objectives both to improve the national space operability in terms of access to space and to increase the Italian industrial capability to realize a Space Tug.

The paper starts with the description of the methodology applied in the space tug and in the STRONG system design (section 2), before applying it to the case study (section 3). In particular, section 3 focuses on the space tug conceptual design, starting with the identification of the mission objectives and the major requirements. Then it describes the tug configuration, in terms of subsystems composing it, the mission scenario and mass budget.

2 DESIGN METHODOLOGY

As already stated, the main aim of the work is the definition of an unmanned system architecture allowing STRONG system to be able to perform payload transfer and retrieval with a space tug. The design process starts from main requirements definition, taking into account all main activities that such a system has to perform to be compliant with stakeholders' needs, regulations and other imposed constraints as, for example, the environment. Then the design proceeds with the Functional Analysis, with the purpose to define activities and all the products able to perform them, according to a System Design Methodology [2][3][4].

The Functional Analysis has been used to define both system architecture and main requirements that drive the system design itself [5]. As far as requirements are concerned, the basic tools of the Functional Analysis are used to derive specific categories of requirements, as shown in Figure 1. Top-level requirements, i.e. mission requirements, directly stem out from the mission statement and mission objectives, which address the crucial issue of this paper study. Moreover, all the actors involved in this project (defined as Stakeholder [6]) impose additional requirements and constraints. However, in order to comply with the activities proposed, the functional analysis should be inserted within a framework of other activities aimed to the definition of some top-level requirements and constraints due to the peculiarities of this application.

Requirements represent the basis of the whole system design and for this reason, they have to be derived with a rational and logical process, in order not to forget any drivers of the design that could eventually lead to an unsuccessful design. In this specific application, requirements have been subdivided into many categories as shown in Figure 1, [3][6][7].

Before writing down requirements, the first activity to perform is the definition of the main objectives of the project. As suggested in [6] they can be derived analysing the Mission Statement. Primary Mission Objectives are directly derived from the mission statement. Mission Statement and Primary Mission Objectives represent mission foundation and, for this reason, they cannot be modified during the definition process.

Simultaneously, another important aspect to be accounted is the analysis of needs and expectations of the main stakeholders. This analysis mainly consists of two steps: identifying all the actors and determining their expectations. Consequently, secondary objectives can be derived. The stakeholders can be categorized as sponsors (i.e. those associations or private who establish mission statement and fix bounds on schedule and funds availability), operators (i.e. those people in charge of controlling and maintaining the main systems analysed), end-users (i.e. those people that receive and use products and capabilities) and customers (i.e. users who pay fees to utilize a specific space mission's product) [7].

Once the main objectives of the project have been derived, the requirement derivation process can start. The typical Functional Analysis tools can be usefully employed in this process. In particular, Functional Tree, Functions/Products Matrix, Product Tree, Block Diagrams and Functional Flow Block Diagram are used in this use case. The overall process is iterative and recursive, meaning that it shall be repeated starting from the highest level to lower levels, i.e. segment level, system level, sub-system level, device level. The reference level for this preliminary study will be the system level.

The Functional Tree is one of the main tools of the Functional Analysis and it allows defining the basic functions that the system shall be able to perform. In order to split the higher level functions into lower level ones, designers ask themselves "how" that higher-level function can be performed. Complementary, as a proof, it is possible to detect the higher-level function asking "why" that lower level functions have to be accomplished by the system. Once the main functions have been derived, it is necessary to map those functions onto the elements able to perform them, thus building up the Functions/Products Matrix. Checked cells of the matrix are used to identify connections between functions and products. The Product Tree can be drawn up starting from the products of the Functions/Products Matrix. Both the Functions/Products matrix and the Product Tree help define the system architecture.

Another System Engineering tool that can be exploited is the Functional/Physical Block Diagram. This diagram depicts a graphical representation of the connections among all items at each level. This tool shows not only which equipment is connected with each other but also the direction and the type of these links (e.g. data exchange, mechanical connection …).

Functional Flow Block Diagrams (FFDBs) are a particular kind of tool that use the functions already founded in order to give further information about them [3]. Particularly, FFBDs use the system functions illustrating the time and the logical sequence of functional events. Even if this tool is able to show what has to happen in the system, it always refers to the functions and not to physical solutions. A way to show the physical solutions that can be applied to solve the Mission Statement is the Concept of Operations (ConOps). It can be said that the FFBDs are a preliminary study for the definition of the ConOps and a connection between the Functional analysis and the ConOps definition.

In the definition of the ConOps, it should be consider all aspects of operations including integration, test, and launch through disposal. In this definition phase, it has to be fixed one or more operational scenarios, describing the dynamic view of the operations and including how the system is supposed to function. Typical ConOps information are [3]: Mission Phases, Modes of Operation, Mission Timeline, Design Reference Mission (DRM) and/or Operational

Scenarios, End-to-end communication strategy and/or Command and data architecture, Operational facilities (e.g., mission control, science data center), Integrated logistic support (resupply, maintenance, and assembly) and Critical events.

At the end of the ConOps definition, one or more adaptable architectures can be considered, but only one is the optimal solution of the design. Trade-off analyses have to be conducted in order to demonstrate which one of the possible architectures is the optimal one, considering the mission statement, the stakeholders' needs and the requirements.

As already explained this process is iterative and has to be performed until the desired level of detail. At each iteration it is possible to define different type of requirements that have different influences over the entire design. A scheme of the requirement definition process is provided in Figure 1.

3 SPACE TUG CONCEPTUAL DESIGN

3.1 Mission and Stakeholders' Analysis

The first phase of a design process, according to the proposed methodology, is the definition of the main objectives and constraints that will drive the design. Two main analyses can be performed to achieve this goal: mission analysis and stakeholders' analysis.

The stakeholders' analysis starts with the stakeholders' identification. The main stakeholders in STRONG project are the following ones:

1) Sponsors: the main sponsor is the MIUR;
2) Operators: engineers from ELV (Vega), Ground Segment operators and engineers (Altec), engineers from TAS-I, CIRA and Selex ES (systems design);
3) End-users: engineers from the operators companies, researcher from university (PoliTo, PoliMi, UniPa, La Sapienza), scientists (specific experiments);
4) Customers: those people that will pay for the services offered by the final System (e.g. space agencies, Universities or private users).

Having defined the main stakeholders involved, their needs can be derived. Therefore, from the stakeholders' analysis some Secondary Mission Objectives can be listed:

- To explore new mission concepts for future space exploration (MIUR);
- To validate critical technologies enabling this operative scenario (MIUR, universities);
- To enhance the cooperation between industries and universities (MIUR);
- To enhance reusability (TAS-I);
- To interface with international space facilities (TAS-I);
- To enhance modularity in interface segments (TAS-I);
- To increase the Vega usage (ELV);
- To have standardized interfaces (ELV);
- To receive data and transmit commands from/to ground (Altec);
- To exploit existing Ground facilities (Altec).

Some consideration can be provided considering the project stakeholders and their needs. Among the many objective that can drive the design of a Space Tug, the system presented in this paper has the particular purpose to improve the national space operability in terms of access to space. This purpose is obtained providing a transportation system capable to transfer satellites platforms from LEOs (Low Earth Orbits), where the launcher release them, to higher operational orbits and back, if needed. The use of such a system can simplify the propulsion system loaded on the satellite platforms thus limiting their overall mass and volume, in favour of larger payloads. In addition, being that particular Space Tug in STRONG project, the main idea is to rely as much as possible on Italian space assets: according to this, VEGA is considered as baseline launcher. This consideration will drive the system design, imposing constraints over volumes, mass and dimensions.

Commonly, a space tug is characterized by high level of reusability, since it is designed to perform many orbital transfers and servicing operations along its operational life. For this reason, periodical refuelling operations are foreseen and a refuelling system adaptable to the mission scenario and on the many stakeholders' requirements has to be considered.

Concerning the system reusability, the space tug is conceived to perform multiple satellites delivery missions in orbit. Considering the current international space roadmaps trends and the average fuel consumption and costs of many kind of propulsion systems available on the market or in development, this particular space tug will implement electrical propulsion. An advantage of the electrical propulsion is the fact that it uses less propellant and it provides a better reliability and simplicity than chemical systems. Oppositely, it offers only low thrust propulsion and, consequently, longer transfers times, but this is not an issue considering the particular application of this work (i.e. an unmanned spacecraft).

In order to define the main requirements, another activity to be performed is the definition of the main objectives of the project. According to the typical conceptual design process in Systems Engineering, for the analysed case study and the imposed hypothesis and constraints, the following Mission Statement can be derived:

To improve the national space operability in terms of access to space by providing a transportation system capable to transfer satellites platforms from Low Earth Orbits to operational orbits and back, relying on Italian space assets.

From this statement, a Primary Mission Objective (i.e. to perform satellites taxi between LEO and the operational orbit) and a Constraint can be derived (i.e.to use Italian space assets). The idea of relying on Italian space assets, has driven the performed analyses and choices, considering the space tug definition (i.e. increasing as much as possible the payload mass deployed on orbit). In addition, considering stakeholders' analysis, VEGA launcher is considered as baseline.

Finally, the need of being adaptable with international existing projects (from stakeholders' analysis), will impose the possibility of a more extended use of the International Space Station and this possibility will be further explored (see section 3.4).

3.2 Requirements

Once the main objectives of the project have been derived, the requirement derivation process should start. Firstly, the main difference between the objective and the constraint is in the kind of requirements that they create: while the objective will create a mission requirements, the constraint is more connected with programmatic requirements.

Other requirements can be derived from the functional analysis. The typical functional analysis tools can be usefully employed in this process (Figure 1). While the top-level function has to be derived directly from the Mission Statement (i.e. to perform satellites/payloads transfer between LEO and the operational orbit), an iterative process will lead to functions of higher levels of detail, defining the activities that the devices involved have to perform. Indeed, every high level function has to be analysed and expanded at higher levels of detail, reaching iteratively the desired level. At the end of this process, the main functions are clearly defined, being able to define the main actors of STRONG system and their interfaces.

Finally, another group of requirements can be derived from the definition of the Concept of Operations (ConOps), as clearly shown in Figure 1. In this phase of the design process, because of the main requirements and iterating the methodology at each level of detail, one or more final system architectures have been determined and trade-off studies to determine which is the optimal solution has to be performed.

For simplicity, all the functional analysis and the iterations performed are here only described in theory. The main results of the Functional Analysis and the ConOps are provided in the following sections. Particularly, in section 3.3 are shown the main systems derived from the Functional Analysis showing the Functions/Products Matrix at system level. Besides, in sections 3.4 are shown the main mission scenario obtained in the analysis of the ConOps. In Figure 1, some examples of requirements are reported.

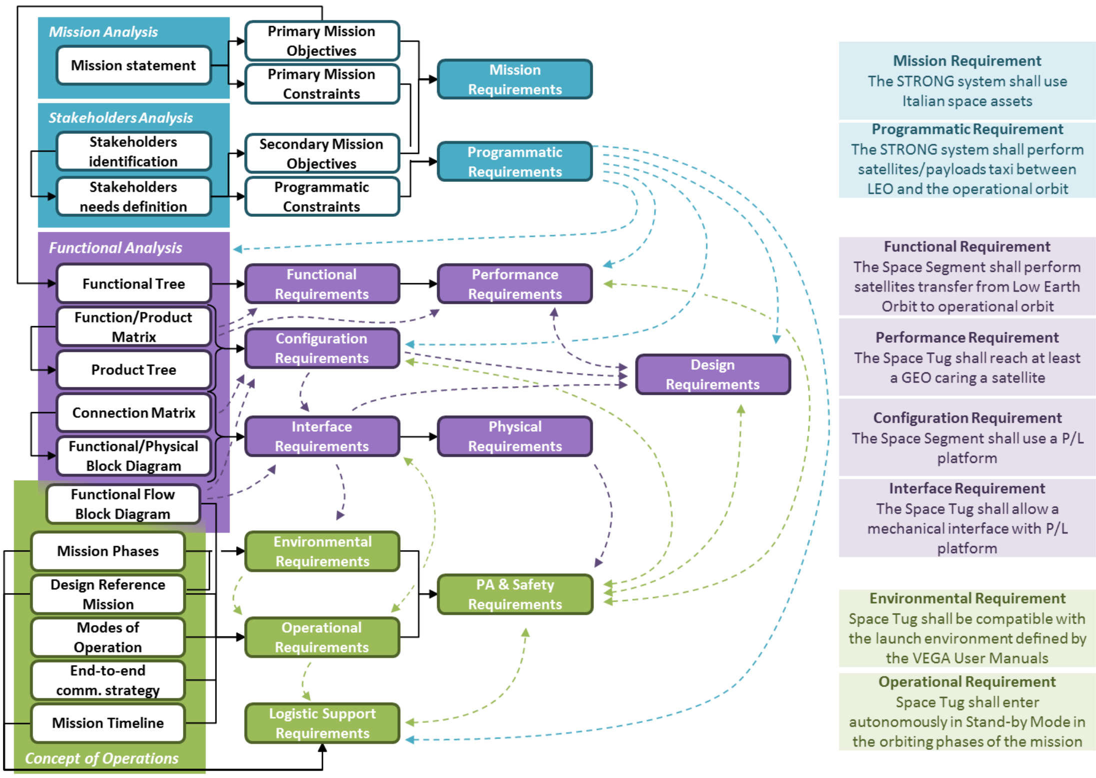

Figure 1: Requirements definition process and examples.

3.3 STRONG System of Systems Configuration

The Functional Analysis is a fundamental tool of the design process to explore new concepts and define their architectures. Indeed, through the Functional Analysis, the main functions are clearly defined, being able to define the main actors of STRONG system and their interfaces. This analysis has been performed to refine the functional requirements, to map its functions to physical components, to guarantee that all and only the required components are listed and to understand the required relationships among the new product's components [4]. According to the Functional Analysis, once the basic functions had been identified, the components to perform those functions have been selected by means of the so-called Functions/Products Matrix (Figure 2). The Functions/Products Matrix has therefore been used to map functions to physical components or systems.

The elements of the overall scenario of STRONG project to be explored include then:

- The VEGA launcher used to carry the maximum payload in LEO;
- Every launch facilities connected with the use of VEGA launcher have to be considered;
- A payload (P/L) platform, considered standard and modular and able not only to be carried in the fairing of VEGA, but also able to maximizes the payload, minimizing the primary propulsion;
- The Space Tug, central system in this study and used to transport the P/L platform between a LEO and its operational orbit, with an electric propulsion system;
- A system for on-orbit refuelling including e.g. a tank to be possibly placed at the ISS as space port or an orbital tank for periodic refuelling for the Space Tug;
- The pre-operational vehicle PRIDE (Programme for Reusable In-orbit Demonstrator for Europe), used to return on Earth samples of P/L and considered as a support for the STRONG system;
- A Mission Control Center (MCC), used to receive telemetry data, uplinks commands and exchanges voice and video contents from/to the main systems involved and ground infrastructure network;
- A Mission Support Center (MSC), used to conduct coordination and operations activities using the information collected by the MCC.

The VEGA launcher is there used to bring the highest possible payload mass in orbit supported by a Space Tug, minimizing the propulsion on the satellite. This kind of system will also give the opportunity to return on Earth significant payload samples by means of the PRIDE pre-operative reusable vehicle. In this particular case, the two systems, the Space Tug and PRIDE vehicle, would rendezvous in a defined orbit, probably the Space Tug waiting orbit, in order to retrieve the payload sample on Earth after having moved it on the PRIDE vehicle. At the end of both cases, a refuelling method is then needed to extend the Space Tug reusability.

Two main methods are then considered for refuelling. Firstly, the Space Tug may utilize the ISS as spaceport for refuelling after few services or for on-orbit maintenance operations on the Space Tug itself (option 1). This option is coherent with the objectives. Another option for the refuelling system is to use an Orbital Tank at which the Space Tug has to dock for refuelling after every mission (option 2). In both cases a refuelling is supposed after each mission performed by the Space Tug. These two configurations will be studied and analysed in the following sections. In support of the choice between the two configurations, some considerations have to be done to provide data on the preliminary figures for the mission elements that are more affected: the on-orbit refuelling system and the Space Tug.

FUNCTIONS		PRODUCTS: Ground Segm.		Launch Segm.		Space Segm.			
		MSC	*MCC*	*VEGA*	*Facilities*	*Space Tug*	*P/L platform*	*PRIDE vehicle*	*Refuelling sys.*
To reach the LEO	*To support launch*				X				
	To perform launch			X					
	To transfer requested objects on LEO			X					
	To release objects on LEO			X					
	To communicate with ground			X					
To perform satellites transfer from Low Earth Orbit to operational orbit	*To perform RvD on LEO*					X			
	To withstand RvD on LEO						X		
	To transfer the object on the operational orbit					X			
	To perform un-docking once on the operational orbit					X			
	To withstand un-docking on the operational orbit						X		
	To perform payload operation once on the operational orbit						X		
	To return on LEO from the operational orbit					X			
	To wait on LEO for the next object					X			
	To ensure communication on orbit						X		
	To maintain communication with ground					X			
To retrieve satellites from operational orbit to Low Earth Orbits	*To move from LEO to the operational orbit*					X			
	To perform RvD on the operational orbit					X			
	To withstand RvD on the operational orbit						X		
	To transfer the object on LEO					X			
	To un-dock the object once on LEO					X			
	To withstand un-docking on LEO						X		
	To wait on LEO for the next object					X			
	To re-enter in atmosphere						X		
	To maintain communication					X			
To re-enter on Earth payloads loaded on board satellites once completed their operative cycle	*To move from LEO to operational orbit*					X			
	To perform RvD on the operational orbit					X			
	To withstand RvD on the operational orbit						X		
	To transfer the object on LEO					X			
	To move on LEO							X	
	To perform RvD on LEO					X			
	To withstand RvD on LEO							X	
	To prepare the payload for the return phase							X	
	To help the preparation of the payload for the return phase						X		
	To return the payload on Earth							X	
	To perform un-docking on LEO					X			
	To withstand un-docking on LEO						X		
	To re-enter in atmosphere						X		
	To wait on LEO for the next object					X			
	To maintain communication during the transfer					X			
	To ensure communication during the re-entry							X	
To perform refuelling on orbit	*To perform RvD on LEO*					X			
	To withstand RvD on LEO								X
	To manage the power transmission								X
	To manage the power collection					X			
	To perform un-docking					X			
	To withstand un-docking								X
	To maintain LEO orbit								X
	To ensure communication								X
	To maintain the communication with ground					X			
To support mission execution	*To provide systems control*		X						
	To support mission operations	X							
	To process telemetry data		X						

Figure 2: Functions/Products Matrix obtained at system level.

The Space Tug is the core element in the proposed scenario. Its main function will be to transfer the P/L Platform into the required operational orbit: the Space Tug has to perform actively the rendezvous and docking with both the P/L platform and the refuelling system. The Space Tug will be also able to bring back from the operational orbit a payload, transfer it into a LEO and perform rendezvous with PRIDE to allow the payload retrieval (or the retrieval of some sensitive part of it) by a robotic arm (located in PRIDE vehicle). One of the main constraints in the Space Tug configuration is to be compatible in mass and volume with the VEGA capabilities (i.e. maximum diameter 2.6 m and maximum length 7.8 m). This constraint will have a significant influence on the choice and the design of the Space Tug sub-systems. To fulfil all the required functions, the Space Tug will be equipped with a number of subsystems, including Propulsion System (including propellant tanks and the refuelling system), Electrical Power Sub-system (EPS, including solar arrays and batteries), Thermal Control Sub-system (TCS), Attitude and Orbit Control Sub-system (AOCS), Communications Sub-system, Structures Sub-system, Harness Sub-system.

In the first refuelling configuration, the Space Tug has to reach the ISS, locating a pressurized propellant tank on the Space Station. Possible location able to provide the requested mechanical and functional interfaces (power and data for the monitoring) has to be defined in more advanced phases of design. During all the operations in proximity of the ISS, the ISS itself will ensure the Space Tug monitoring and control in synergy with the on-ground control stations.

In the second refuelling configuration, an Orbital Tank can be exploited. In this scenario, the Orbital Tank is supposed to stay autonomously in orbit for the time required by the Space Tug to perform a defined number of missions. The Orbital Tank is supposed to maintain the orbit and the attitude and to send information on its status. A Soyuz flight is supposed in order to maximize the fuel stored.

3.4 Concept of Operations

At the end of the Functional Analysis all the activities that the STRONG system has to perform, all the systems and sub-systems involved and their features are finally known. At this stage of the design, the system operations have to be defined.

As mentioned in the previous section, the overall reference mission scenario mainly includes the following phases, excluding logistic and preparatory phases: Space Tug deployment, Satellite platform deployment, Space Tug refuelling. Two configurations have been defined for the refuelling: to exploit the ISS (option 1) and to use an Orbital Tank (option 2).

In both refuelling configurations, the first missions set starts with the launch of the space tug, which then remains in its waiting orbit till the launch of the first satellite platform. Once the tug has docked with the satellite platform, the transfer towards the final operational orbit begins. After releasing the satellite, the tug moves to the refuelling orbit to perform the first refuelling. After refuelling operations have been completed, the second missions set can start.

In the first option, the Space Tug moves back in the parking orbit, where the satellite platform has to be launched by VEGA. This particular orbit is coincident with the Space Tug waiting orbit, in order to minimize the fuel consumption, being the change of inclination for refuelling expensive. Once the tug has docked with the satellite platform, the transfer towards the final operational orbit begins.

In this particular refuelling option, the ISS is envisaged to be exploited for the refuelling operations. In order to maximize the payload launched in the parking orbit, the reference launch orbit has a very low inclination (in this case 5.2° and at an altitude of 700 km), where then the space tug shall dock with the satellite platform. According to this, it would be convenient to have a refuelling station in an inclination orbit lower than the ISS one, which is actually a high inclination orbit (51.6°) and therefore requires a significant inclination change. However, the ISS is an already available infrastructure and it is worth to exploit it, even considering that developing, launching and maintaining a new facility would be complex and expensive.

The number of satellites deployments before a refuelling has been evaluated considering the constraints related to the employment of VEGA, which is the reference launcher. Europe's new VEGA launcher is designed to launch small payloads (300 to 2500 kg satellites) to polar and LEO [8]. Referring to its performances and considering a safety margins in order to be conservative, for the reference launch orbit 1800 kg payloads capability has been assumed having 700 km altitude and 5.2° inclination.

In the second option, the refuelling is carried out at a waiting orbit that the Space Tug is easily able to reach after every mission (i.e. at an inclination of 5.2° and an altitude of 500 km), minimizing drag effects. In this case, no change of inclination is needed for the refuelling and a lower parking orbit can be assumed for the payload launched by VEGA, maximizing their mass (i.e. at an inclination of 5.2 and an altitude of 350 km). In this particular refuelling option, being the Space Tug at launch able to carry a small amount of propellant (i.e. not particular efforts are needed for refuelling operations), a first mission for a very small P/L can be considered before the first refuelling (and Orbital Tank launch). Also in the option 1, additional propellant can be launched with the Space Tug, but considering the complexity of the scenario, this propellant will be used in other to perform a first P/L platform transfer and has to be considered filled with the maximum amount of propellant. An integration of the propellant used in the first mission will be evaluated for both the configurations at the end of this section, considering VEGA capabilities and the space tug launch mass. Also in option 2, VEGA performances have been considered to find the maximum payload capability in the scenario fixing the reference orbits. In this case, for the reference launch orbit 2100 kg, payloads capability has been assumed having 350 km altitude and 5,2° inclination.

An additional objective for the space tug can be to support the retrieval of payloads to be re-entered on Earth. The sequence of operations for this mission profile is shown in Figure 3. In this scenario there is an additional element, which is the re-entry vehicle, in charge of bring back payloads on Earth (PRIDE vehicle).

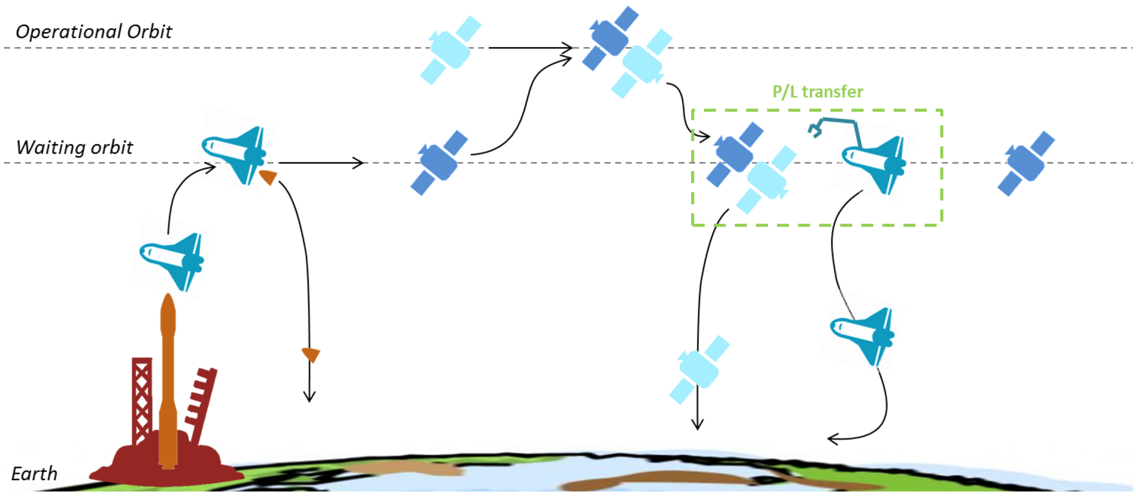

Figure 3: Design Reference Mission (DRM) for the payload retrieval scenario.

To evaluate the mission scenario a Matlab® script has been built and computations for different cases have been carried out. Indeed, it is necessary to compare the various options to select the best concept of operations [9]. The Matlab® script allowed estimating the total propellant needed to accomplish the satellites delivery missions and the transfer for refuelling to ISS or to the waiting orbit in the case of the Orbital Tank. In particular, it determines the characteristics of low thrust orbital transfer between two circular orbits. According to [11], in a low-thrust orbit transfer the total velocity change is given by:

$$\Delta V = V_0 \cos \beta_0 - \frac{V_0 \sin \beta_0}{\tan\left(\pi/_2\, \Delta i + \beta_0\right)} \tag{1}$$

Where V_0 is the initial orbit velocity, β_0 is the initial thrust vector yaw angle and Δi is the total desired inclination change. Once obtained the total required velocity change, the initial mass in the parking orbit has been computed relying on the typical Tsiolkovsky rocket equation:

$$\Delta V = v_e \ln {}^{m_0}/_{m_f} \tag{2}$$

Where v_e is the effective exhaust velocity (i.e. $v_e = I_{sp} \cdot g_0$, where I_{sp} is the specific impulse and g_0 is standard gravity), m_0 is the initial total mass, including propellant, and m_f is the final total mass.

Moreover, the transfer time has been evaluated as:

$$t_f = {}^{\Delta V}/_{f} \tag{3}$$

Where f indicates the low-thrust acceleration, which for this computation is assumed constant during the orbit transfer. The acceleration magnitude has been obtained as the ratio between thrust (constant thrust is considered) and the average between final and initial mass.

This procedure can be applied in both the refuelling options. Indeed, it has to be considered that at the end of each mission the final mass at the waiting orbit has to be the dry mass of the Space Tug, needing it a refuelling. Proceeding backwards from the final situation all the needed parameters can be evaluated.

As results of the analysis, for the option 2, the STRONG system for the entire operational scenario periodically repeats a single mission, from the payload retrieval to the refuelling. The total amount of repetitions has been set to 4 [10]. On the contrary, for option 1, after a first payload transfer and a first refuelling at the ISS, two missions are allowed before a refuelling of the Space Tug is required.

According to the scenario just described and using the developed Matlab® tool, the various phases have been analysed and the masses of propellant needed to accomplish the various transfers have been calculated for both the refuelling options. Moreover, an estimate of the transfer time has been performed, considering constant thrust acceleration.

The computations have been carried out considering the following assumption:

- parking orbit:
  - option 1) 700 km, 5.2°;
  - option 2) 350 km, 5.2°;
- waiting orbit:
  - option 1) 700 km, 5.2° (same as the parking orbit);
  - option 2) 500 km, 5.2°;
- final operational orbit (considered at GEO, i.e. the worst condition): 36000 km, 0°;
- refuelling orbit:
  - option 1) 360 km, 51.6° (ISS orbit);
  - option 2) 500 km, 5.2° (same as the waiting orbit);
- constant thrust equal to 480 mN and a I_{sp} of 2500 s;
- satellite platform mass:
  - option 1) 1800 kg;
  - option 2) 2100 kg, a first small payload of 150 kg has been considered [10];

The obtained results are summarized in Table 1 for option 1 and in Table 2 for option 2. The budgets shown have been computed considering a preliminary Space Tug dry mass equal to 600 kg in both the refuelling options. A more detailed evaluation of the Space Tug dry mass will be provided in the next section.

Phase		**Initial mass [kg]**	**Final mass [kg]**	**Propellant mass [kg]**	**Transfer Time [days]**
First missions set	*LEO-GEO 1*	3155 (3570)	2627 (2970)	528 (600)	313 (354)
	GEO-ISS 1	827 (1170)	600 (850)	227 (320)	135 (192)
Second missions set	*ISS-LEO 2*	3345	2315	1030	616
	LEO-GEO 2	4115	3427	688	408
	GEO-LEO 2	1627	1355	272	161
	LEO-GEO 3	3155	2627	527	313
	GEO-ISS 3	827	600	227	135

Table 1: Mission phases' budgets in option 1 (in brackets the case with additional propellant at launch).

	Phase	Initial mass [kg]	Final mass [kg]	Propellant mass [kg]	Transfer time [days]
First missions set	*parking LEO - waiting LEO*	914	911	3	2
	waiting LEO - parking LEO	911	908	3	2
	parking LEO - GEO	1058	874	184	109
	GEO - waiting LEO	724	600	124	73
Second missions set	*waiting LEO - parking LEO*	1322	1318	5	3
	parking LEO - GEO	3418	2824	594	352
	GEO - waiting LEO	724	600	124	73
	waiting LEO - parking LEO	1322	1318	5	3
	parking LEO - GEO	3418	2824	594	352
	GEO - waiting LEO	724	600	124	73
	waiting LEO - parking LEO	1322	1318	5	3
	parking LEO - GEO	3418	2824	594	352
	GEO - waiting LEO	724	600	124	73
	waiting LEO - parking LEO	1322	1318	5	3
	parking LEO - GEO	3418	2824	594	352
	GEO - waiting LEO	724	600	124	73

Table 2: Mission phases' budgets in option 2.

In option 1, the total propellant mass needed to accomplish the first missions set (one satellite delivery to GEO and one transfer to ISS for refuelling) is about 755 kg. In addition, the total propellant mass needed for the second missions set (Space Tug transfer to LEO, two delivery missions in GEO and a transfer to ISS for refuelling) is about 2740 kg. The tug launch mass in the first configuration is about 1350 kg (see the next section). This value is below the maximum capability of the VEGA launcher (i.e. 1800 kg in 700 km, 5.2° LEO) and therefore additional propellant can be loaded and then exploited in the following missions, as previously explained. In particular, the total additional amount that can be included is around 250 kg still being compatible with the launcher capability. In this case, the budget for the first missions set is shown in Table 1.

On the other hand, in option 2 the total propellant mass needed to accomplish a single missions set is about 723 kg, excluding the first small payload transfer, while the total propellant mass needed before an Orbital Tank change (i.e. 4 mission set) is about 2892 kg. Finally, for the first small payload transfer a propellant mass of 314 kg is required. Considering that in this particular option a complete refuelling is required after every mission performed and doubling the Space Tug tank capability for safety reasons, a total amount of propellant of 1450 kg has been considered. The launch mass of the tug in the first configuration, is about 914 kg. This value is below the maximum capability of the VEGA launcher (i.e. 2100 kg in 350 km, 5,2° LEO) and therefore additional propellant can be loaded and then exploited in the following missions or for a huger first small payload transfer. In particular, considering this difference between the launcher capabilities and the scenario required propellant mass, a first complete mission scenario can be achieved.

It is worth underlining that these concepts of operations (option 1 and option 2) represent a conservative case. Indeed, it can be necessary to deliver the satellites in orbits lower and less requiring than GEO. In this case, more than two delivery missions could be accomplished before refuelling is needed.

3.5 Main Systems Mass Breakdown

In the previous section, we have defined the systems and the sub-systems involved, taking about their features and about the Concept of Operations. In this phase, a preliminary mass budget has been performed to obtain a mass breakdown for the Space Tug, in both the refuelling configurations under study. The mass budget has been obtained taking as reference the Dawn mission spacecraft mass breakdown [12], since this is a real mission implementing electric propulsion system. Being the propulsion and the power sub-systems critical technologies, the Space Tug dry mass has been computed starting from the sizing of these sub-systems, which are the most impacting subsystems for this type of vehicle [9][10].

The propulsion sub-system can be considered as composed of two main parts: the thrusters (including the thruster and the power-conditioning unit) and the propellant (including the tanks and propellant management unit). For our analysis, its specific mass only includes the mass of the thruster and power processor (the masses of the propellant subsystem, gimbals, and other mission specifics are not included).

For the present computation, the Hall Effect Thrusters are assumed as reference and the following values are used: a specific impulse (I_{sp}) of 2500 s, a power ratio (R) of 50 mN/kW and a specific mass (SM) of 5 kg/kW. With these values, the power (P) needed to obtain the required thrust (T) has been computed ($P=T/R$) and then the mass (M) has been derived ($M=SM\cdot P$). Particularly, considering a thrust of 480 mN, the needed power amounts to about 9.6 kW and the thruster mass is about 50 kg for both the refuelling options. The propellant tanks mass is computed as the 4% of the total propellant mass to be loaded (about 2740 kg in option 1 and 1450 kg in option 2). According to this, the overall mass of the propulsion sub-system amounts to almost 160 kg (option 1) and 110 kg (option 2).

For what concerns the Electrical Power sub-system (EPS), it includes deployable solar panels for power generation and batteries for energy storage, in both the refuelling configurations. The EPS has been sized such that propulsion is constantly guaranteed both in daylight and eclipse condition. To perform the sizing of the solar arrays, the parking orbit has been taken as reference orbit, as it represents the worst case, having the longest eclipse time.

The solar arrays area has to be computed according to [6]:

$$A_{SA} = {}^{P_{SA}}/_{P_{EOL}} \tag{4}$$

In (4) the P_{SA} is the power that solar arrays must provide during daylight to power the spacecraft for the entire orbit:

$$P_{SA} = \frac{((P_e T_e) \ / \ x_e \ + (P_d T_d) \ / \ x_d \ )}{T_d} \tag{5}$$

Where P_e and P_d are the power requirements during eclipse and daylight respectively, T_e

and T_d are the length of these periods, x_e and x_d the efficiencies of the paths from the solar arrays through the batteries to the individual loads and the path directly from the arrays to the loads, respectively. The total power required to be provided by the solar arrays amounts to about 18.6 kW (option 1) or 19.2 kW (option 2), including power for batteries recharge, as well as other subsystems required power.

Another parameter to be computed is the P_{EOL} that is the array power per unit area at the end of life. It can be computed multiplying by a degradation factor the power per unit area at the beginning of life.

$$P_{EOL} = P_{BOL} L_d \tag{6}$$

$$L_d = (1 - degradation/yr)^{operative\ life} \tag{7}$$

$$P_{BOL} = \varphi_{Sun} \eta I_d \cos\theta \tag{8}$$

Where φ_{Sun} is the Sun flux, η is the conversion efficiency, I_d is the inherent degradation, which accounts for the design and assembly losses and θ is the Sun incidence angle. Considering high efficiency solar cells (i.e. triple junctions cells with 30% efficiency and 84 mg/m^2 of specific mass), the required area is about 54 m^2 (option 1) or 62 m^2 (option 2). Therefore, assuming that the blanket mass is 55% of the total array mass [7], the corresponding solar arrays mass is about 83 kg (option 1) or 95 kg (option 2).

In addition to the solar array, Li-ion secondary batteries will provide energy especially during eclipse. In addition, the batteries have to be sized. For the batteries sizing, the following equation has been used to compute the total batteries capacity:

$$C_r = {}^{P_e T_e}\!\big/_{(DOD)x_e} + self\text{-}discharge \tag{9}$$

The obtained required batteries total capacity is 10 kWh (option 1) or 9 kWh (option 2). Therefore, considering a specific energy of 175 Wh/kg, the total battery mass is about 57 kg (option 1) or 50 kg (option 2). Finally, the power control and distribution unit mass has been obtained as the 20% of the overall EPS mass: the total EPS mass is about 175 kg (option 1) or 182 kg (option 2).

Starting from the mass values obtained for the propulsion and power subsystems, the total dry mass has been computed referring to Dawn mission. The propulsion and the electrical power sub-systems constitute about the 50% of the total dry mass in Dawn spacecraft. Considering the complexity of the mission scenarios here proposed, a larger percentage has been considered since the power requirement and the quantity of propellant needed for the missions are higher. Particularly, 60% of the total dry mass has been considered. With this particular percentage, the Space Tug dry mass is about 550 kg (option 1) or 480 kg (option 2). The obtained mass breakdown is reported in Table 3, reporting also the mass fractions used for the preliminary assessment of the other sub-systems masses and derived from the Dawn ones. In addition, a system margin of 10% has been included to account for the uncertainties typical of this design phase. Accordingly, the resulting dry mass is 600 kg (option 1) or 530 kg (option 2).

Sub-system	Mass Fraction [%]	Mass [kg]	
		Option 1	*Option 2*
Propulsion (w/o tanks)	28	157	106
EPS	32	175	182
TCS	7	36	34
AOCS	5	28	24
DMS	3	16	15
Communications	3	16	15
Structures	15	83	72
Harness	7	39	34
TOTAL (w/o margin)	-	550	480
System margin	-	10%	10%
TOTAL (w margin)	-	~600	~530

Table 3: Space Tug mass breakdown.

Considering the tanks sizing, using the data obtained through the mission scenarios provided in the previous section and the Space Tug sizing, some information can be provided. In the first option, considering that a refuelling is needed after every mission, a minimum amount of 755 kg of propellant has to be stored, until a maximum of 2740 kg considering safety margins, regulations and a mission set of 2 repetitions before a refuelling of the tank has to be performed. In the second refuelling option, for the same reasons already told a minimum amount of 723 kg of propellant has to be stored, till a maximum of 2892 kg considering a mission set of 4 repetitions before a refuelling of the tank has to be performed. Considering the mission sets as described previously, a change in the refuelling tanks has to be performed after about 4.5 years and 5 years respectively. In the two considered options, the Space Tug has transferred 3600 kg and 8400 kg respectively of payload.

4 CONCLUSIONS

The interest in the development of a reusable space tug derives from several applications such a system would have, as for instance satellite servicing, debris removal or large spacecraft assembly. The space tug presented in this paper is conceived to support the transfer of satellites from LEO, where the launchers release them, to their final operational orbits. The main reason of the interest in this kind of mission scenario is in the fact that it would allow a user to reduce the need for a satellites propulsion system, in favour of larger payloads.

In order to improve the cost-effectiveness of his kind of system, the space tug has to perform many missions during its operational lifetime, requiring periodic refuelling operations. Two reference scenarios have been identified, i.e. exploiting the ISS (option 1) or an orbital fuel tank (option 2). In addition, a preliminary sizing of the space tug has been performed according to these reference scenarios and to the constraints deriving from the Space Tug conceptual design methodology. In particular, the total amount of propellant needed to accomplish the reference set of missions has been derived, supposing the use of VEGA launcher.

During the preliminary sizing performed on the STRONG system and the Space Tug, many differences have been underlined in the two identified refuelling architectures. Indeed, while the use of a dedicated orbital fuel tank leads to simplifications in the Space Tug design, not requiring high propellant storage at launch, the Space Tug design in case of the use if the ISS for refuelling has to foresee very high propellant storage capabilities. This information is connected with the differences in the mass budgets evaluation: the option 1 needs a higher mass, also considering the complexity of the mission scenario that implies a considerable change of orbit inclination. Another important difference in the two architectures is in the ratio between the payload transferred and the propellant required. Indeed, the option 2 seems more competitive, considering that is able to guarantee a higher amount of payload transferred in its final operational orbit with a single orbital fuel tank and with a lower amount of propellant required to perform these missions.

Certainty, the costs has to be studied. The development of an orbital fuel tank is the design and the creation of a very new system and an additional element permanently in orbit, increasing the STRONG system complexity and its costs. In addition, the orbital tank requires a propellant storage capability that is not compatible to VEGA dimensions, and a larger launcher has to be foreseeing (e.g. the Soyuz). On the contrary, the development and the storage of propellant at the ISS implies not only high costs, but also safety conditions to be considered and implemented, increasing the complexity of this particular architecture.

The costs and a more detailed analysis of the architectures here defined can be a future development of this work. Future works will also focus on the space tug detailed design, with specific sizing of the subsystems not yet analysed.

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