Design of an Earth-to-Mars Tether Launch System

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Submitted by AOE All-Stars

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List of Abbreviations

ADCS: Attitude determination and control system
C&DH: Command and data handling
CMG: Control moment gyro
GEO: Geosynchronous Earth orbit
GN&C: Guidance, navigation, and control
GPS: Global positioning system
GTO: Geosynchronous transfer orbit
HEX: Hexagonal Earth-to-Mars Exchange
HTPC: High-temperature-phase-change
ISS: International Space Station
LEGO: LEO-GEO TLS
LEM: Lunar Excursion Module
LEO: Low-Earth orbit
LST: Lunar slingshot trajectory
MIPS: Millions of instructions per second
MOE: Measure of effectiveness
MTO: Mars transfer orbit
NASA: National Aeronautics and Space Administration
RFP: Request for proposals
RTG: Radioisotope thermoelectric generator
SIMPL: Six-sided Multi-catch Pinwheel
SMARTS: Stair-stepping to Mars Autonomous Rendezvous Tether System
SRM: Solid rocket motor
STINC: Stack integrated configuration
TLS: Tether launch system
TSS: Tethered Satellite System
VSD: Value system design
List of Symbols

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>a</td>
<td>Semi-major axis</td>
</tr>
<tr>
<td>$a_c$</td>
<td>Centripetal acceleration</td>
</tr>
<tr>
<td>$\alpha$</td>
<td>Angle between initial and lunar velocity</td>
</tr>
<tr>
<td>$\alpha'$</td>
<td>Angle between final and lunar velocity</td>
</tr>
<tr>
<td>$\mathbf{B}$</td>
<td>Magnetic field vector</td>
</tr>
<tr>
<td>$B$</td>
<td>Strength of Earth’s magnetic field</td>
</tr>
<tr>
<td>$B_0$</td>
<td>Strength of Earth’s magnetic field at the surface</td>
</tr>
<tr>
<td>$B_H$</td>
<td>Perpendicular strength of Earth’s magnetic field</td>
</tr>
<tr>
<td>$\beta$</td>
<td>Angle between initial and final velocity</td>
</tr>
<tr>
<td>$d$</td>
<td>Distance between Earth and Moon</td>
</tr>
<tr>
<td>$D$</td>
<td>Tether diameter</td>
</tr>
<tr>
<td>$\Delta R$</td>
<td>Altitude change</td>
</tr>
<tr>
<td>$\Delta t$</td>
<td>Change in time</td>
</tr>
<tr>
<td>$\Delta V$</td>
<td>Change in velocity</td>
</tr>
<tr>
<td>$E_o$</td>
<td>Orbital energy</td>
</tr>
<tr>
<td>$\varepsilon$</td>
<td>Emissivity of tether material</td>
</tr>
<tr>
<td>$erf$</td>
<td>Error function</td>
</tr>
<tr>
<td>$\phi$</td>
<td>Flight path angle</td>
</tr>
<tr>
<td>$F$</td>
<td>Force/thrust</td>
</tr>
<tr>
<td>$FS$</td>
<td>Factor of safety</td>
</tr>
<tr>
<td>$g$</td>
<td>Gravitational acceleration on Earth</td>
</tr>
<tr>
<td>$\eta_g$</td>
<td>Tether efficiency, including only cable losses</td>
</tr>
<tr>
<td>$h$</td>
<td>Angular velocity</td>
</tr>
<tr>
<td>$I_{sp}$</td>
<td>Specific impulse</td>
</tr>
<tr>
<td>$\mathbf{J}$</td>
<td>Current density vector</td>
</tr>
<tr>
<td>$l_2$</td>
<td>Distance from center of mass to tip mass</td>
</tr>
<tr>
<td>$\lambda$</td>
<td>Magnetic latitude</td>
</tr>
<tr>
<td>$l_E$</td>
<td>Tether length</td>
</tr>
<tr>
<td>$m$</td>
<td>Mass of the TLS</td>
</tr>
<tr>
<td>$M$</td>
<td>Mass of the tether</td>
</tr>
<tr>
<td>$\mu$</td>
<td>Gravitational constant of Earth</td>
</tr>
<tr>
<td>$m_p$</td>
<td>Payload mass</td>
</tr>
<tr>
<td>$N$</td>
<td>Number of cables in the tether</td>
</tr>
<tr>
<td>$P_N$</td>
<td>Nominal power density</td>
</tr>
<tr>
<td>$P_S$</td>
<td>Survival probability</td>
</tr>
<tr>
<td>$\Theta$</td>
<td>Inclination referenced from the magnetic north pole</td>
</tr>
<tr>
<td>$\rho$</td>
<td>Density</td>
</tr>
</tbody>
</table>
$r$ Orbital radius  
$R$ Radial distance in Earth radii  
$Re$ Radius of Earth  
$\sigma$ Stefan-Boltzmann constant  
$\sigma_c$ Electrical conductivity  
$\tau_2$ Trajectory  
$T$ Temperature of the tether  
$T_{BK}$ Background temperature  
$U$ Ultimate tensile strength  
$v$ System velocity  
$v_f$ Final velocity  
$v_i$ Initial velocity  
$V$ Lunar velocity  
$V_1$ Velocity before propulsion is activated  
$V_c$ Circular velocity  
$V_{crit}$ Critical velocity of the tether material  
$W_B$ Ohmic power density  
$W_T$ Radiated power density  
$\omega$ Angular velocity  
$X$ Cable power generation relation
Chapter 1: Introduction and Problem Definition

The mission of the Earth-to-Mars tether launch system (TLS) is to launch payload spacecraft into a Mars transfer orbit (MTO) or into an Earth escape trajectory by exchanging momentum between the TLS and the payload spacecraft. A TLS minimizes the amount of propellant needed to perform such a transfer, thereby giving a boost to the payload spacecraft and reducing the cost of sending satellites into orbit around Mars. The TLS will be capable of launching six payload spacecraft into a MTO every Earth-Mars alignment, which occurs approximately every two years.¹

1.1 History and background

1.1.1 Tethers

Tethers are a potentially simple, inexpensive way of using fundamental physical principles to propel spacecraft already in orbit. Tethers can theoretically be used for controlling satellite motion through both momentum transfer techniques and electrodynamic propulsion. Tethers are being considered for moving payloads from the space shuttle to the International Space Station (ISS) as well as for interplanetary transfer missions such as the one proposed in this report.

In 1974 Guiseppe Colombo, a renowned Italian scientist, formulated a concept which developed into the design of the Tethered Satellite System (TSS). This system introduced the use of a long tether to support a spacecraft from an orbiting platform. Colombo and one of his colleagues, Mario Grossi, approached the National Aeronautics and Space Administration (NASA) and the Italian Space Agency with their idea, and the
TSS was launched in 1992 aboard the Shuttle Transport System STS-46. The concept of long gravity-gradient stabilized tethers was confirmed upon TSS mission success.\textsuperscript{5}

Since the launch of the first TSS experiment, at least 16 other space tethers have been deployed, few of which have been successful. These failures confirm that our knowledge of tether technology is lacking. In 1992 the Italian Space Agency was scheduled to deploy a satellite tethered to the shuttle Atlantis. The spool mechanism on the tether jammed and the experiment failed. In 1996 NASA tried the same experiment a second time with a different Italian satellite. As the tether reached full deployment, the tether’s motion through the Earth’s magnetic field created 3,500 volts of electricity that then flowed through the tether. This experiment confirmed that tethers could be used to generate power in low-Earth orbit (LEO). A crack in the tether’s insulation caused an electric arc, which burned through the tether. The resulting momentum transfer sent the satellite into a higher orbit, increasing its altitude by seven times the original tether length.\textsuperscript{5}

Tether technology is still expanding today. Electrodynamic tethers are being studied to provide space propulsion without the added cost and mass of propellant. Such tethers may be used to move upper stage rockets into lower orbits as well as to keep the ISS in its orbit without refueling.\textsuperscript{5}

1.1.2 Mars mission history

Mars missions began in 1960 with the launch of the Marsnik 1 by the Soviet Union. This first mission failed when the third stage pumps were unable to produce enough thrust.\textsuperscript{22} The Soviet Union and the United States attempted several Mars flybys and orbiters throughout the rest of the 1960’s with little mission success. Some of the
problems encountered in space were communication loss, hardware deployment failure, and inadequate thrust. However, some missions did succeed, and images of Mars were sent back to Earth.

In the 1970’s, more orbiters and flybys were attempted, along with the first attempts at landers. The Viking missions (1975) by the United States were the first completely successful Mars landers. The Soviet Union landed the Mars 3 successfully, but all of its instruments failed. The Viking missions resulted in an overall view of the Martian surface. Not all Mars exploration missions were as successful. The Mars Observer, launched in 1992, was to orbit Mars for the purpose of studying the geoscience and climate of the planet. The spacecraft lost contact for unknown reasons 3 days before the scheduled orbit insertion, and communication was never re-established. None of the primary objectives of the mission were accomplished. The total cost of the Mars Observer was $980 million. Another failed mission, the Mars Polar Lander (1999), lost communication near the Martian atmosphere, and data from Mars were never received.

Overall, 35 missions to Mars have been attempted with 22 failures. These are expensive losses, especially when the satellite produces no data for the specified mission. Many Mars orbiters have been launched for the purpose of determining the magnetic field, atmosphere, surface geology, and climate on Mars. Landers have been used to characterize the surface in more detail and to search for evidence of life. Some previous missions have been specifically designed to study the effects of extended space flight on instruments and communications. Future Mars missions are currently being planned with the purpose of gathering atmospheric and geologic data of the planet. By the year 2011, a Mars sample return mission is expected.
Future Mars missions need a more reliable communications network to keep costs down and to progress forward. There is a wealth of knowledge to be learned from Mars. A satellite constellation around Mars has been proposed to provide future mission support. The constellation would be a network of navigation and communication satellites in orbit around the planet. This constellation should provide a better link between the Earth and the Mars exploration vehicles to prevent future failures.

1.2 Problem definition

The objective of this project is to design an Earth-to-Mars tether launch system. This system is one step in the process of building a satellite constellation around Mars. This section details the relevant elements involved in the design of the TLS.

1.2.1 Required disciplines

The TLS presents a complex problem that must be solved by a multi-disciplinary group of individuals. Aerospace engineers specify the necessary orbit and design the attitude control and determination systems. Both aerospace and mechanical engineers design the spacecraft structures and propulsion system, as well as handle thermal issues. Electrical engineers define requirements for the power system, including solar arrays, batteries, and power regulation and distribution. The TLS needs to communicate with the ground, and possibly with other payload spacecraft. Designing the needed communications architecture requires radio frequency engineers and ground station specialists. Computer and software engineers design both the computer hardware and the software that controls the spacecraft. In addition, many subsystems require software engineers to write computer code for their specific operational algorithms. Integration and test engineers of all disciplines ensure that all subsystems work together properly and
will survive the mission lifetime. Systems engineers interface with all areas of the project, while managers and subsystem leads oversee the project and track costs.

1.2.2 Scope

The transfer of a payload spacecraft from Earth-to-Mars is an endeavor encompassing different stages and operations. Defining the project scope is important to put the size of the project into perspective. The following is a list of all elements included in the scope.

- Design all subsystems and operations that are part of the TLS
- Select TLS launch vehicle
- Define ground station locations and requirements
- Design TLS/payload spacecraft rendezvous method and mechanical interface
- Define all orbits

Some elements not included in the scope are:

- Design payload spacecraft
- Design launch vehicle
- Design ground station

Based on the scope defined in this section, Figure 1 is a visual representation of the mission architecture.
1.2.3 Political and societal entities

The design, production, and use of the TLS affects several political and societal groups. The National Aeronautics and Space Administration and its subcontractors are involved in many aspects of the TLS project, such as material requirements and launch facilities. Other nations possessing the capability to launch interplanetary and high orbit missions may be interested in the tether launch system. Public safety may be threatened by specific subsystem components that contain harmful materials.

1.2.4 Needs, alterables, and constraints

and mission geometry and analysis. The program management and mission operations teams produce a mission schedule and a cost model for the TLS.
Table 1 lists the needs, alterables, and constraints relevant to the TLS design. The only need for this project is to launch payload spacecraft from Earth to Mars. To accomplish this need, all required subsystems are designed and integrated into one TLS design. The subsystems and operations are alterable within the limits set forth by the list of constraints. The subsystems to be designed include the attitude determination and control system (ADCS), the guidance, navigation and control system (GN&C), propulsion, the communications system, the command and data handling system (C&DH), the power system, the thermal and environmental system, and the structures and mechanisms. Other alterables include the launch vehicle selection, astrodynamics, and mission geometry and analysis. The program management and mission operations teams produce a mission schedule and a cost model for the TLS.

Table 1: TLS needs, alterables, and constraints

<table>
<thead>
<tr>
<th>Category</th>
<th>Element</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Needs:</strong></td>
<td>Launch payload spacecraft from Earth to Mars orbit</td>
</tr>
<tr>
<td><strong>Alterables:</strong></td>
<td>Tether/payload spacecraft rendezvous method and interface</td>
</tr>
<tr>
<td></td>
<td>Launch vehicle</td>
</tr>
<tr>
<td></td>
<td>Orbit design</td>
</tr>
<tr>
<td></td>
<td>Propulsion method throughout mission</td>
</tr>
<tr>
<td></td>
<td>All subsystem level design</td>
</tr>
<tr>
<td></td>
<td>Material selection</td>
</tr>
<tr>
<td><strong>Constraints:</strong></td>
<td>Under 3-g accelerations imposed on payload spacecraft</td>
</tr>
<tr>
<td></td>
<td>Rendezvous with payload spacecraft in LEO</td>
</tr>
<tr>
<td></td>
<td>Must accommodate payload spacecraft shape and size</td>
</tr>
<tr>
<td></td>
<td>Must not violate international law</td>
</tr>
<tr>
<td></td>
<td>Launch six 200 kg payload spacecraft every Earth-Mars alignment</td>
</tr>
<tr>
<td></td>
<td>Lifetime of 30 years</td>
</tr>
<tr>
<td></td>
<td>Launch TLS by 2012</td>
</tr>
<tr>
<td></td>
<td>Use momentum exchange transfer method</td>
</tr>
</tbody>
</table>
1.2.5 Relevant elements

The TLS operates in orbit around the Earth. Therefore, it must be launched and inserted into a stand-by orbit. Launch vehicle selection, launch site, and initial velocity change (\(\Delta V\)) are selected based on the chosen stand-by orbit.

A challenging element of this problem is the rendezvous between the TLS and the payload spacecraft to be launched. The accuracy and response characteristics of the ADCS and of the propulsion system must be sufficient to provide the fine control needed by either a remote or computer-controlled rendezvous system. The structure of the TLS must withstand forces or shocks during rendezvous.

Momentum transfer is used to propel the payload spacecraft into a MTO. Clean separation and precise alignment are necessary to accomplish the momentum transfer launch. The clean separation of the payload spacecraft and the TLS falls under the structures and mechanisms subsystem. The ADCS and the propulsion subsystem align the tether configuration.

The TLS is designed for multiple launches; therefore, it must re-boost itself into its stand-by orbit after each launch. This re-insertion requires a propulsion subsystem that provides the appropriate \(\Delta V\). The ADCS adjusts the attitude of the TLS so that the propulsive force is directed correctly.

The TLS’s required lifetime is thirty years. Failure contingencies and a method of repairing critically damaged sections should be available to achieve this required longevity. A provision for the launch of replacement TLS platforms is one possible solution.
Upon TLS mission completion, an end-of-life contingency procedure must prevent it from adding to the debris already in LEO. One contingency is to degrade the orbit of the TLS so that it eventually burns up in the atmosphere. A propulsive force is required to initiate this orbit change, thereby involving both the propulsion and ADCS subsystems.

All of the relevant elements depicted in this section require the power, communications, and GN&C subsystems. All of the electronics require power to operate. Communication with the ground is vital for successful long term, multiple launch operation. The attitude and orbit determination algorithms of the TLS require up-to-date information on the payload spacecraft’s position and velocity during rendezvous, provided by the GN&C subsystem.

1.3 Summary

Chapter 1 gives an introduction to the history of tethers and Mars missions. Information needed to solve the TLS problem is also introduced. The problem definition identifies the scope, boundaries, and relevant elements of the problem. The following chapters outline in detail the process of producing a preliminary design for a TLS. A value system is established to evaluate possible solutions idealized during system synthesis. These solutions are then analyzed to reach one optimal solution for a TLS design.
Chapter 2: Value System Design

The value system design (VSD) is used to evaluate potential TLS design concepts. The VSD puts all mission objectives into a hierarchy beginning with the top-level objective: optimize the TLS design. Maximizing the performance objectives and minimizing the cost objectives optimizes the TLS design. The performance and cost objectives are described in detail in this chapter. Sub-levels then lead to measures of effectiveness (MOEs), which are quantified and used to analyze alternative designs.

2.1 Objectives

Below the top-level objective are two second-level mission objectives: maximize performance and minimize cost. Each subsystem achieves these objectives in different ways. Table 2 depicts the interactions between subsystems and mission objectives.

Figure 2 illustrates the entire objective hierarchy including all MOEs.

Table 2: List of objectives and their relevant subsystems

<table>
<thead>
<tr>
<th>Objective Description</th>
<th>Relevant Subsystem(s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Maximize Performance</td>
<td></td>
</tr>
<tr>
<td>Maximize power efficiency</td>
<td>Power</td>
</tr>
<tr>
<td>Maximize power output</td>
<td>Power</td>
</tr>
<tr>
<td>Minimize power consumption</td>
<td>All except structures</td>
</tr>
<tr>
<td>Maximize computer compatibility</td>
<td>C&amp;DH</td>
</tr>
<tr>
<td>Maximize thrust</td>
<td>Propulsion, power</td>
</tr>
<tr>
<td>Minimize position error</td>
<td>Guidance and navigation</td>
</tr>
<tr>
<td>Minimize pointing error</td>
<td>ADCS, mechanisms</td>
</tr>
<tr>
<td>Maximize available control torque</td>
<td>ADCS</td>
</tr>
<tr>
<td>Maximize material effectiveness</td>
<td>Structures</td>
</tr>
<tr>
<td>Maximize launch vehicle performance</td>
<td>Launch vehicle selection</td>
</tr>
<tr>
<td>Minimize escape $\Delta V$</td>
<td>Mission analysis, astrodynamics</td>
</tr>
<tr>
<td>Minimize station-keeping $\Delta V$</td>
<td>Mission analysis and geometry</td>
</tr>
<tr>
<td>Maximize thermal efficiency</td>
<td>Structures, environment</td>
</tr>
<tr>
<td>Minimize radiation exposure</td>
<td>Electronics, structures, environment</td>
</tr>
<tr>
<td>Minimize mass</td>
<td>All</td>
</tr>
<tr>
<td>Minimize production cost</td>
<td>All</td>
</tr>
<tr>
<td>Minimize launch cost</td>
<td>All</td>
</tr>
<tr>
<td>Minimize operations cost</td>
<td>All</td>
</tr>
</tbody>
</table>
2.1.1 Performance objectives

The objective hierarchy is further divided into sub-levels of performance and cost that are associated with MOEs. For example, a MOE for the ADCS is pointing error of the TLS. This MOE should be minimized to achieve greater precision in attitude maneuvers and allow for successful rendezvous. Available thrust is a MOE of the performance of the propulsion system. Communications system performance is measured in terms of bandwidth, in Hertz, and transmission distance, in kilometers. Computer performance is measured by the number of operations per second that the computer is able to perform, in millions of instructions per second (MIPS).18

Several performance MOEs are comprised of separate quantities. The performance of the spacecraft material is measured by a material effectiveness factor, X. This factor depends on several characteristics of the material including thermal conductivity, thermal expansion, fracture toughness, and ductility. Other characteristics that determine the material’s effectiveness are strength, stiffness, toughness, and density. The objective to minimize power consumption depends on all internal components of the TLS such as the computer, ADCS components, and the propulsion system. The computer and communications system performance also depends upon factors, which affect the entire TLS, such as the radiation dose. Electronics are radiation-sensitive and may be easily disrupted by radiation that the TLS receives in its environment. Minimizing the number of system failures per year due to radiation maximizes performance.18

2.1.2 Cost objectives

The cost of the TLS is divided into three sections: production, launch, and operation. The goal is to minimize all three of these cost categories. Building the TLS
involves material costs, fabrication and assembly costs, and costs of the individual components of the design. Launch vehicle cost is directly dependent upon the mass and size of the TLS as well as the desired initial orbit. Operational cost depends upon the personnel and functional requirements of the ground station. Additionally, there is a cost associated with any required servicing of the TLS.

2.2 Summary

The VSD is used to evaluate the possible solutions that are developed in the next chapter, system synthesis. The objective hierarchy serves as the basis for weighting MOEs and evaluating alternative solutions. The goal is to find a solution that best reflects the values decided upon in the VSD.
Chapter 3: System Synthesis

The system synthesis chapter reviews alternative elements and conceptual designs for the TLS. A list of system elements is created in a Chinese menu, which is then used to develop system alternatives. For each system element, several component level alternatives are described, and their advantages and disadvantages are discussed. Following the separation into system elements, four alternative TLS configurations are formed using different components for each system element.

3.1 System elements

3.1.1 Operation

The TLS is required to rendezvous with the payload spacecraft in LEO.\(^1\) A Hohmann transfer can be used to place the TLS into a higher altitude before the momentum exchange.

A Hohmann transfer changes the payload spacecraft’s orbit from a LEO to a higher energy orbit. In a Hohmann transfer, two \(\Delta V\) impulses are applied to the spacecraft: the first to insert the spacecraft into an elliptical transfer orbit, and the second to insert the spacecraft into the higher energy orbit. A momentum exchange tether can provide a \(\Delta V\) using a slingshot launch or using an extendable tether to launch the payload spacecraft into a MTO. The “strawman” architecture consists of a single tether that boosts the spacecraft into a higher orbit. Instead of boosting directly into a MTO, the payload spacecraft is placed into a highly elliptical Earth orbit. From the highly elliptical Earth orbit the payload spacecraft performs a burn to achieve a MTO.\(^1\)
A multiple TLS constellation using a stair-stepping architecture propels the payload spacecraft into a MTO. The $\Delta V$ impulse required by the individual TLSs is decreased, and therefore the momentum that each TLS transfers to the payload spacecraft is reduced. Thus, the $\Delta V$ impulse required for the TLS to return to its stand-by orbit decreases. The stair-stepping architecture minimizes the power, propellant, and reinsertion time the TLS requires. The smaller reinsertion time allows the TLS to be used to launch more spacecraft more often. The versatility of a TLS is increased when using stair-stepping architecture. The separate TLSs can perform orbit-raising or lowering maneuvers for any spacecraft with the appropriate docking mechanism. An optimized set of rendezvous and momentum transfer launches could be determined for ease in launching payload spacecraft to other planets. The inherent redundancy in using multiple TLSs allows for the failure of a single TLS without rendering the entire constellation useless. With both software and orbit adjustments, the remaining TLS could still perform its momentum transfers.

Transfer orbit options increase when using a multiple TLS constellation. The LEO constraint requires that one TLS rendezvous with the payload spacecraft in LEO. Although the beginning of the process is restrained to LEO, the other TLSs in the constellation can be placed at any altitude or in any orbit. An example of such a system has a TLS in LEO and one in geosynchronous Earth orbit (GEO). The LEO TLS rendezvous with the payload spacecraft and then executes a momentum exchange launch that propels the payload spacecraft into a geosynchronous transfer orbit (GTO). The payload spacecraft rendezvous with the GEO TLS. The GEO TLS then performs another momentum transfer launch. The second launch places the payload spacecraft either
directly into a MTO or into a highly elliptical Earth orbit from which it performs a burn to enter into a MTO.

A lunar slingshot trajectory (LST) could be used by any transfer orbit option. Velocity is gained by using the Moon as a gravitational slingshot. The payload spacecraft is inserted into a hyperbolic orbit around the Moon where it experiences a velocity change due to a momentum exchange with the Moon.

### 3.1.2 Attitude control

All spacecraft require attitude control actuators for reorientation and pointing during a mission. Three commonly used techniques are spin stabilization, gravity-gradient stabilization, and three-axis control. Choosing attitude control actuators involves listing all possibilities and examining their advantages and disadvantages. Some actuators are designed to move large payloads quickly, whereas others are designed to be accurate. Table 3 lists four attitude control actuator possibilities for the TLS and their respective performance characteristics. From this table, actuators may be chosen and sized for different variations of the TLS.\(^ {18} \)

<table>
<thead>
<tr>
<th>Actuator</th>
<th>Torque range, N·m</th>
<th>Weight, kg</th>
<th>Power output, W</th>
</tr>
</thead>
<tbody>
<tr>
<td>Reaction wheels</td>
<td>0.01 – 1</td>
<td>2 – 20</td>
<td>10 – 110</td>
</tr>
<tr>
<td>Control moment gyro</td>
<td>25 – 500</td>
<td>Greater than 10</td>
<td>90 – 150</td>
</tr>
<tr>
<td>Thrusters</td>
<td>0.5 – 9000 N</td>
<td>Variable</td>
<td>Variable</td>
</tr>
<tr>
<td></td>
<td>multiplied by</td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>moment arm</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Magnetic torquers</td>
<td>(4.5 \times 10^{-5} – 0.18)</td>
<td>0.4 – 50</td>
<td>0.6 – 16</td>
</tr>
</tbody>
</table>

Reaction wheels and control moment gyros (CMG) are momentum storage devices. Reaction wheels are flywheels on spin motors. They can spin clockwise or counter-clockwise and provide control along one axis of a spacecraft. Therefore, three
reaction wheels are required for complete attitude control. Reaction wheels are sized according to their angular momentum capacity, which limits their capability for moving larger spacecraft. A CMG is a gimbaled flywheel. The wheel spins at a constant rate, and when the spacecraft requires attitude adjustment, the gimbal rotates the wheel so that the angular momentum vector created by the spinning wheel shifts directions and turns the spacecraft. These devices create large torque values about all three orthogonal axes of a spacecraft. They are capable of moving large spacecraft quickly. However, they require a complex control algorithm and momentum exchange, and they tend to be expensive and heavy.

Thrusters produce torque by expelling mass from the spacecraft. Thrusters are often used to avoid problems inherent in using momentum storage devices, such as saturation and frequent momentum dumping. They vary in mass and cost and produce a wide range of thrust. Different types of thrusters are discussed in the propulsion section later in this chapter.

Magnetic torquers use magnetic coils or bars to generate magnetic dipole moments. These devices can de-saturate momentum-exchange systems, and they compensate for residual magnetic fields around the spacecraft. One major advantage of these actuators is a lack of moving parts. However, they are not effective in high orbits.18

3.1.3 Power generation

Spacecraft can generate power using solar power, nuclear power techniques, fuel cells, or an electrodynamic tether. Solar power generation involves using photovoltaic cells to convert solar energy into electricity. Nuclear power techniques include using radioisotope thermoelectric generators (RTGs) or nuclear fission reactors. Fuel cells use
a chemical reaction to produce power. These are used to generate power on the Space Shuttle.\textsuperscript{18}

Solar power generation is a reliable, relatively inexpensive option that can be used in LEO. However, solar cells only convert 15-20\% of the sun’s energy into electricity. Solar cells are strung together and may be either mounted on the spacecraft bus, or mounted in large quantities to external structures or solar panels. Solar panels require large surface areas to be effective and are susceptible to mechanical problems during deployment and orbital debris impact.

Nuclear power techniques involve harnessing heat generated by a nuclear process and converting it into electricity. RTGs use heat produced by the natural decay of radioactive isotopes, whereas nuclear reactors use heat from nuclear fission. Both options are efficient in terms of power production from their fuel sources, and are limited only by fuel availability. Nuclear power techniques are expensive and present safety and political concerns. Launch failure of a spacecraft carrying radioactive materials could result in radiation exposure. Safe disposal of nuclear materials left in orbit at end-of-life must be determined.\textsuperscript{18}

Fuel cells use chemical reactions to generate power and have been used extensively on manned space missions. Fuel cells are highly efficient (50-80\%) and produce large amounts of power. They are low mass and relatively long life, but their mass increases with lifetime due to additional fuel requirements. While fuel cells are efficient, they are typically used as a secondary power source.\textsuperscript{18}

Electrodynamic tether power generation moves a conductive tether through a magnetic field to produce a current in the tether. The amount of power the tether can
produce is directly proportional to the length of the tether. Electrodynamic tethers have
the ability to produce power while in eclipse, which is a significant advantage over solar
panels.\textsuperscript{18}

\subsection*{3.1.4 Energy storage}

Spacecraft that rely on solar power store the energy generated in the sun for use
during eclipse. Energy is stored using batteries, flywheels, or thermal storage techniques.
Batteries are devices that store electricity using a chemical reaction. This chemical
reaction is replenished by reversing the storage process and by adding electricity to the
batteries. This process allows for batteries to be recharged. A flywheel is spun up by a
motor to store energy in kinetic form. The flywheel is then spun down, and the motor
becomes a generator.\textsuperscript{16} Thermal storage techniques use high-temperature-phase-change
(HTPC) materials to store energy. Heat energy is added to a canister of HTPC material,
and the material changes from solid phase to liquid phase. Heat energy is released when
the material returns to solid phase.\textsuperscript{11}

Batteries have been used extensively as energy storage devices in space
applications. They are inexpensive, simple, and versatile. Batteries are categorized by
their reactant chemicals, which create electricity. By changing the chemicals used in the
battery, the performance characteristics are tailored to specific design needs. Batteries
have various lifetimes, some of which reach around 20 years.\textsuperscript{18}

The flywheel energy storage method’s lifetime theoretically provides a longer
lifetime than batteries. Long lifetimes of flywheels have not been proven in space
applications because they are a relatively new technology. However, if a simple and
reliable method of spinning and maintaining is used, the lifetime of the flywheel may
increase. The spinning of flywheels causes torques that must be accounted for and or used by the ADCS.\textsuperscript{16}

Thermal energy storage is a method that is unproven in space applications, but one that could provide lifetimes longer than batteries. The HTPC materials can provide a simple and inexpensive way of storing energy if they are insulated properly. However, if not insulated well, the HTPC materials lose heat and become inefficient. Therefore, the canister that houses the HTPC materials must be capable of maintaining the temperature of its contents under all conditions. Heat energy must be transferable through the canister walls only when intended. Insulation materials and transfer method technology is being developed in order to make thermal energy storage more efficient.\textsuperscript{11}

### 3.1.5 Rendezvous and capture

There are several options that can be used to achieve a spacecraft rendezvous in orbit. In general, a solution is either autonomous or ground controlled. Autonomous rendezvous is advantageous because it does not require ground station support. For an operator on the ground to control the payload spacecraft during the rendezvous, all burns must be completed at specific times when both spacecraft are commandable. An autonomous rendezvous system needs only starting position and time information from the ground. Combining global positioning system (GPS), radar, and cross-link communications enables two payload spacecraft to intercept each other without ground control. The GPS is used to obtain position vectors for both spacecraft. Software algorithms calculate the appropriate burn time and position vector using known orbital positions of the payload spacecraft. As the payload spacecraft nears the rendezvous
point, radar and cross-link communications update range and range rate data calculated from the GPS through braking and docking.\textsuperscript{18}

Four capture methods are illustrated below in Figure 3. These interfaces mechanically connect the payload spacecraft to the TLS once positioning is complete. Any capture method and interface should not impose more than 3-g’s on the payload spacecraft.

![Capture methods](image)

**Figure 3: Capture methods**

The net capture method uses a flexible-cable net, similar to a fisherman’s net. The payload spacecraft enters the net and the TLS’s control. The TLS maneuvers the net to place the payload spacecraft into position for the upcoming launch to Mars. The main advantage of this system is its simplicity; the net captures the payload. The net capture
method can also be automated. The main disadvantage of this system is the possibility of the payload spacecraft becoming tangled in the net, which may cause enough collateral damage to the payload spacecraft to terminate its use.

The clamp capture method uses a “claw” to clamp around the payload spacecraft. This method requires a sophisticated set of robotic fingers. The claw clamp attaches to a specially designed, structurally reinforced receiver plate on the payload spacecraft, creating a solid connection. The clamp capture method is difficult to automate and requires accurate pointing and mechanism control.

The magnet capture method is used either for capture alone or to align the clamp in the clamp capture method. An electromagnet has a powerful magnetic field. This field does not extend past one-and-a-half times the diameter of the magnet. The payload spacecraft must be close to the magnet for full functionality. The TLS applies power to the electromagnet to capture the payload spacecraft and cuts power to the electromagnet to release the payload spacecraft. The magnet capture method requires a specific location on the payload spacecraft where it must be captured so that C&DH subsystem components both inside the TLS and inside the payload spacecraft are not damaged. This method can be automated.

The cone/inverse cone capture method is similar to the method used by the command module to capture the Lunar Excursion Module (LEM) in the Apollo program. The cone on the payload spacecraft slides into the inverse cone on the TLS. The simplicity of this capture method allows for automation. The cone corrects itself as it travels into the inverse cone for a perfect fit, and the cone locks into the inverse cone.
The cone/inverse cone capture method has a high probability for success due to self-correction.

### 3.1.6 Propulsion

Rocket propulsion is used on spacecraft for any type of desired velocity change. Two factors, thrust, $F$, and specific impulse, $I_{sp}$, are measures of a propulsion system’s performance. Thrust is the amount of force applied to the rocket and specific impulse is the measure of the energy content of the propellants and how efficiently that energy is converted into thrust.¹⁸ Four types of propulsion systems are chemical, electrical, cold gas, and propellantless momentum transfer. Properties of the first three of these systems are listed in Table 4.

<table>
<thead>
<tr>
<th>Propulsion System</th>
<th>Orbit Insertion</th>
<th>Orbit Maintenance</th>
<th>Attitude Control</th>
<th>Thrust Range, N</th>
<th>Typical $I_{sp}$, sec</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cold gas</td>
<td>No</td>
<td>Yes</td>
<td>Yes</td>
<td>0.05 – 200</td>
<td>50 - 75</td>
</tr>
<tr>
<td>Solid</td>
<td>-</td>
<td>-</td>
<td></td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Solid propellant</td>
<td>Yes</td>
<td>No</td>
<td>No</td>
<td>50 - 5 × 10⁶</td>
<td>280 - 300</td>
</tr>
<tr>
<td>Hybrid</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>225 - 3.5 × 10⁵</td>
<td>225</td>
</tr>
<tr>
<td>Liquid</td>
<td>-</td>
<td>-</td>
<td></td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Monopropellant</td>
<td>No</td>
<td>Yes</td>
<td>Yes</td>
<td>0.05 - 0.5</td>
<td>150 - 225</td>
</tr>
<tr>
<td>Bipropellant</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>5 - 5 × 10⁶</td>
<td>300 - 425</td>
</tr>
<tr>
<td>Dual mode</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>3 – 200</td>
<td>300</td>
</tr>
<tr>
<td>Electric</td>
<td>-</td>
<td>-</td>
<td></td>
<td>-</td>
<td>-</td>
</tr>
<tr>
<td>Ion engine</td>
<td>No</td>
<td>Yes</td>
<td>No</td>
<td>5 × 10⁻⁶ – 0.5</td>
<td>2,000 - 6,000</td>
</tr>
<tr>
<td>Electromagnetic</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>25 – 200</td>
<td>2000</td>
</tr>
<tr>
<td>Electodynamic</td>
<td>Yes</td>
<td>Yes</td>
<td>Yes</td>
<td>5 × 10⁻⁶ - 20</td>
<td>-</td>
</tr>
</tbody>
</table>

Chemical propulsion systems include solid propellant motors and liquid propellant engines, and are the most widely used propulsion systems in space applications. Solid rocket motors (SRM) burn solid chemicals for propulsion and have a high thrust value. However, this thrust is not adjustable, so SRMs are only used for launch or for orbit insertions that require large $\Delta V$s. Solid rocket motors are simple,
reliable, and inexpensive systems. Another form of solid propellant motors is a hybrid rocket system, which stores propellant in different forms and is a safe and environmentally clean system. They also have the ability to throttle and restart. Hybrid rockets have a high performance, but they are more massive than SRMs.\textsuperscript{18}

Liquid propellant engines include monopropellant, bipropellant, and dual mode systems. Monopropellant engines are used most effectively for attitude control and orbit maintenance; their thrust output is relatively low. Monopropellant systems are inexpensive and simple but cannot provide large $\Delta V$s. Bipropellant engines burn two types of fuel to achieve high performance. These systems are complicated and dangerous but can be used to perform virtually any orbit maneuver. Dual mode systems integrate bipropellant and monopropellant systems to achieve high specific impulses for large $\Delta V$s, as well as precise burns for orbit maneuvering. Dual mode engines are toxic and dangerous.\textsuperscript{18}

Electric propulsion systems use electric power to accelerate a working fluid and produce thrust. Ion engines use an electric field to accelerate charged particles and have a high specific impulse. Electromagnetic thrusters accelerate plasma through a magnetic field and have a high specific impulse. Both ion engines and electromagnetic thrusters are power limited.\textsuperscript{18} Electrodynamically propulsion works by sending current along a tether. This current interacts with the Earth's magnetic field to generate thrust.\textsuperscript{10}

Cold gas propulsion is used if hot gases and liquid and solid systems present a safety threat. Cold gas systems have low performance, so they are only used for orbit maintenance and attitude control. These systems are massive for their level of performance, but are inexpensive, simple, and reliable.\textsuperscript{10}
Momentum exchange tether propulsion is a propellantless method of changing a spacecraft’s orbit. Momentum transfer between a payload spacecraft and a TLS changes both velocities. Momentum is transferred to the payload spacecraft to achieve a higher energy orbit, and momentum is transferred from the payload spacecraft to decrease the energy of its orbit. The momentum exchange is achieved through two methods: the exploitation of the difference in the gravity potentials and spinning the system.

3.2 System alternatives

3.2.1 Stack configuration

The concept for this design involves integrating the payload spacecraft into a stack configuration, or STack INtegrated Configuration (STINC). The STINC consists of stacks of multiple payload spacecraft that fit into a host spacecraft as illustrated in Figure 4.

![Figure 4: Illustration of STINC alternative](image)

The host spacecraft integrates with the chosen launch vehicle, supplies sufficient power and instrumentation for rendezvous with the TLS and stack separation, and carries the stacks through all phases of travel to Mars. The STINC uses the cone/inverse cone
docking interface to be captured by the TLS. The rendezvous and launch procedure for this design consists of the following steps:

1. The payload spacecraft are launched in stack configuration to a parking orbit.
2. STINC executes a burn to intercept the TLS in its stand-by orbit.
3. On-board radar and GPS guide STINC and the TLS together autonomously.
4. With the tether retracted, the TLS captures STINC using a cone/inverse cone docking mechanism.
5. Following capture, the TLS executes a Hohmann transfer burn to raise the orbit to launch altitude.
6. The launch orbit is of sufficient altitude so that after release of STINC, the perigee of the TLS orbit is at the altitude of its original stand-by orbit.
7. Upon reaching launch altitude STINC is reeled out on the tether and released into a LST.
8. STINC travels around the moon for an additional boost into a MTO.
9. Upon arriving in Mars orbit, STINC executes an orbit insertion burn, and the individual payload spacecraft separate from STINC and begin their mission.
10. After the release of STINC, the TLS executes a single burn at the perigee of the new orbit to place it back into the circular stand-by orbit.

STINC saves time, energy, and complexity in all stages of the TLS mission. Each time a momentum exchange transfer occurs, the TLS must return to its stand-by orbit to rendezvous with another spacecraft. Depending on the design and mass of the TLS, this orbit-raising maneuver could take a long time. Catching multiple payload spacecraft simultaneously significantly reduces mission execution time. Launching the payload
spacecraft in STINC reduces the number of required rendezvous, but it results in a greater
decrease in the TLS’s orbit after separation. Integrating the payload spacecraft in a stack
decreases the complexity of the mission.

The TLS for this alternative must provide power over its lifetime, have attitude
control actuators for reorientation and pointing control, and have propulsion to execute its
major orbit maneuvers. The TLS uses batteries for energy storage and a radioisotope
thermoelectric generator for power generation. Reaction wheels and magnetic torquer
bars provide attitude control. Bipropellant propulsion is used for orbit maneuvering. The
TLS is capable of capturing and launching any spacecraft with the proper interface
including different sized stacks, single payload spacecraft, and possibly other spacecraft
outside the Earth-Mars alignment.

The STINC requires separate power, propulsion, and attitude control systems.
Power is generated on STINC using solar arrays, and is stored in batteries. Bipropellant
propulsion is used for STINC’s orbital maneuvers. Four orthogonally mounted thrusters
control STINC’s attitude during rendezvous and orbit insertion. The TLS and STINC use
GPS, radar, and cross-link communication for orbit determination. These systems are
particularly important during autonomous rendezvous and capture.

3.2.2 Pinwheel configuration

The pinwheel design, or S1x-sided Multi-catch PinwheelL (SIMPL), is a
hexagonal shaped TLS with one retractable tether on each of its six faces, as shown in
Figure 5.
Figure 5: Illustration of SIMPL alternative

Each tether launches one payload spacecraft every Earth-Mars alignment, which reduces tether use over the SIMPL lifetime. The rendezvous and launch procedure of this design consists of the following steps:

1. All six of the payload spacecraft are launched into LEO.
2. Each payload spacecraft performs a burn to intercept SIMPL in its stand-by orbit.
3. On-board radar and GPS guide the payload spacecraft to SIMPL.
4. With the tether retracted, SIMPL captures the payload spacecraft using a cone/inverse cone docking mechanism.
5. Following each individual capture, SIMPL rotates so that the next payload spacecraft can rendezvous with the opposing side from the previous payload spacecraft.
6. After all captures, SIMPL deploys two opposing tethers for electrodynamic propulsion to place itself into the desired launch orbit.

7. When SIMPL reaches the launch orbit, the tethers are reeled in and SIMPL waits for the Earth-Mars alignment.

8. SIMPL spins prior to payload spacecraft release.

9. Two opposing tethers deploy and SIMPL releases the first payload spacecraft into a LST.

10. Its propulsion system returns SIMPL to the launch orbit.

11. The opposing tethered spacecraft is released in the next cycle.

12. Steps 9 through 11 are repeated for the remaining payload spacecraft.

13. Each payload spacecraft travels around the moon for an additional boost into a MTO.

14. After all payload spacecraft are released, SIMPL stops spinning and uses its propulsion system to place itself back in its stand-by orbit for the next set of rendezvous.

With current technology, the Earth-Mars alignment is approximately two weeks long. Having all six payload spacecraft captured and ready to launch prior to the Earth-Mars alignment is important. There are two propulsion systems on SIMPL. The primary propulsion system is an electrodynamic propulsion system. Two tethers are always deployed on SIMPL to keep the center of mass at the system’s center. The secondary form of propulsion is a monopropellant system for quick orbit maneuvers. Control moment gyros are used to spin the configuration and for attitude control. These CMGs
require a large amount of power, so this design uses RTGs for power generation and the HTPC materials for energy storage.

### 3.2.3 Hexagon configuration

The hexagonal TLS configuration, HEX (Hexagonal Earth-to-Mars eXchange), contains a single, retractable tether and is capable of holding up to six payload spacecraft simultaneously. The six payload spacecraft are attached to a hexagonal body at the end of the tether as shown in Figure 6. This configuration allows for the collection of all payload spacecraft before Earth-Mars alignment so that this brief launch window can be used solely for launching the payload spacecraft. Any errors that occur during rendezvous and capture are fixed during the stand-by time and do not waste the time in the launch window.

![Figure 6: Illustration of HEX alternative](image)

The following steps define the procedure that the HEX uses for rendezvous and launch of the payload spacecraft.

1. HEX is in a low-Earth stand-by orbit with the tether retracted.
2. The six payload spacecraft are launched into LEO prior to Earth-Mars alignment.
3. HEX extends its tether and rendezvous autonomously with one payload spacecraft.

4. The payload spacecraft docks to an empty side of the hexagonal body at the end of the extended tether.

5. Docking is accomplished using a cone/inverse cone mechanism.

6. Steps 3 through 5 are repeated for all the remaining payload spacecraft prior to Earth-Mars alignment.

7. At the beginning of Earth-Mars alignment, HEX performs a burn to place itself into higher altitude, circular orbit (the launch orbit).

8. With the tether fully extended, the system spins to prepare for release.

9. One payload spacecraft is released from the hexagon and is sent into a MTO.

10. When the payload spacecraft reaches the end of the MTO, the payload spacecraft uses the on-board propulsion system to insert itself into Mars orbit.

11. The momentum loss of the HEX caused by the release of the payload places the HEX into a lower altitude orbit than the launch orbit.

12. The tether is retracted and HEX performs a burn to place itself back into the launch orbit.

13. Steps 8 through 12 are repeated until all the remaining payload spacecraft are released.

14. After the last release, HEX is sent back into a lower altitude orbit and a small burn is performed to get the HEX back into the standby orbit.

The HEX alternative has many benefits. Docking onto the extended tether allows for flexibility in the payload spacecraft initial orbit insertion. Extended tether docking
requires GN&C and ADCS on the hexagon. The cone/inverse cone docking mechanism is beneficial because it is tolerant of small positioning and pointing errors. By launching one payload at a time, the momentum loss of the TLS after each launch is reduced due to the extra mass at the end of the tether.

Reaction wheels and magnetic torquers control the attitude of the HEX for pointing accuracy during rendezvous and orbit maintenance. To extend the HEX to a higher orbit, it is zenith facing. Turning the HEX to this orientation occurs slowly, so the design has no slew rate restriction, and the use of reaction wheels is possible. The HEX alternative uses an autonomous rendezvous and a bipropellant propulsion system.

3.2.4 LEO-GEO TLS constellation configuration

The LEO-GEO TLS (LEGO) with a lunar slingshot consists of two tether platforms: one stationed in LEO and the other stationed in GEO. The tether platforms have retractable tethers. The rendezvous is performed with the tether fully retracted. There are a number of rendezvous methods, docking mechanisms, power systems, and propulsion systems that could be used that would not change the overall concept of this solution. The arrangement and number of payload spacecraft launched does not affect this solution. This type of TLS involves two rendezvous and carefully planned astrodynamics. The rendezvous and launch procedure for this design consists of the following steps:

1. Payload spacecraft are launched into LEO configured in a stack formation.
2. The LEO TLS rendezvous with the stack.
3. Attitude control actuators on both the LEO TLS and the stack apply a torque to the system to achieve a pre-determined angular velocity.
4. The LEO TLS reels out the tether.

5. The stack is released by a mechanism at the end of the tether, and is inserted into a GTO.

6. The propulsion subsystem of the LEO TLS applies the required force to re-insert the platform into its stand-by orbit.

7. The propulsion subsystem of the stack applies the required force to insert the stack into GEO.

8. The GEO TLS rendezvous with the stack.

9. Attitude control actuators on both the GEO TLS and the stack apply a torque to the system to achieve a pre-determined angular velocity.

10. The GEO TLS reels out the tether.

11. The stack is released by a mechanism at the end of the tether, and is inserted into a LST.

12. The propulsion subsystem of the GEO TLS applies the required force to re-insert the platform into its stand-by orbit.

13. The stack performs the appropriate burns to intercept Mars and insert into an 800 km circular orbit.

Momentum transfer is achieved through a slingshot launch of the payload spacecraft. In a slingshot launch, the TLS/payload spacecraft system is rotated, and the payload spacecraft is released so that it achieves a $\Delta V$ in the appropriate direction. Extra $\Delta V$ is also attained through the use of an extended retractable tether by exploiting the difference in the gravity potentials of the TLS and the payload spacecraft.
Figure 7 is a diagram this type of TLS system. The solid lines represent the orbits of the TLSs. The dashed lines represent the transfer trajectories of the payload spacecraft after each rendezvous. The small circles at the intersection of the orbits represent the momentum launch points.

Figure 7: Illustration of LEO-GEO TLS constellation alternative

A derivation of this option consists of multiple TLSs operating independently in LEO. These TLSs are capable of performing multiple capture and toss operations simultaneously. The TLSs are located in various LEOs, giving greater flexibility to the initial orbital insertion of the payload spacecraft. Each of the TLSs acts as the LEO TLS described in steps 1 through 4 above. The remaining procedure consists of the following steps:
5. Each TLS captures one payload spacecraft and performs a burn to place both itself and the payload spacecraft into a higher orbit while extending the tether.

6. After reaching the desired altitude, the TLS launches the payload spacecraft into a higher Earth orbit.

7. The payload spacecraft performs a burn to place it into an MTO while the TLS re-orientates itself and waits for the next rendezvous.

Figure 8 shows one TLS in a circular standby orbit waiting to capture a payload spacecraft while another TLS simultaneously launches a payload spacecraft from LEO.

**Figure 8: Illustration of LEO TLS constellation alternative**

### 3.3 Summary

The system synthesis chapter develops four possible alternatives for the TLS. Each alternative uses a different combination of system elements generated using a Chinese menu approach. The alternatives presented in this chapter represent preliminary conceptual designs. In the following chapters, each alternative is evaluated, and one optimal solution is chosen for further analysis and optimization.
Chapter 4: System Analysis

The system analysis chapter describes the methods used to rank the system alternatives from the previous chapter. A set of new high-level objectives is created to identify which alternatives best satisfy implicit requirements of the Request for Proposals (RFP) as interpreted by the team. Each alternative undergoes qualitative analysis through which its advantages and disadvantages are fully defined. A quantitative ranking of each alternative based on the new objectives leads to a better understanding of the needs of the team.

4.1 New objectives

Three new high-level objectives, used to rank the four alternatives arrived at in Chapter 3, include maximizing creativity, minimizing operational complexity, and maximizing the probability of practical mission completion. Creativity in both design and operation gives the TLS solution signature quality. While a creative design is high priority, a complex TLS that requires difficult operational algorithms should be avoided. The preferred TLS is at a medium between simple but dull and creative but complicated. The lifetime of the TLS, from concept to end-of-life, should also function on a practical budget, using realistic manufacturing and operating techniques.

Table 5 lists the four system alternatives and how they measure up to these three objectives. A rating of 3 signifies the most creative, least complex, most realistic alternative.
### Table 5: High-level ranking of system alternatives

<table>
<thead>
<tr>
<th>Alternative</th>
<th>Creativity</th>
<th>Complexity</th>
<th>Realistic</th>
<th>TOTAL</th>
</tr>
</thead>
<tbody>
<tr>
<td>STINC</td>
<td>3</td>
<td>3</td>
<td>2</td>
<td>8</td>
</tr>
<tr>
<td>SIMPL</td>
<td>1</td>
<td>1</td>
<td>3</td>
<td>5</td>
</tr>
<tr>
<td>HEX</td>
<td>3</td>
<td>2</td>
<td>3</td>
<td>8</td>
</tr>
<tr>
<td>LEGO</td>
<td>3</td>
<td>2</td>
<td>2</td>
<td>7</td>
</tr>
</tbody>
</table>

### 4.2 Qualitative analysis

This section includes tables listing advantages and disadvantages for each system alternative. These tables illustrate the conceptual strengths and weaknesses of each alternative. From these tables, desired characteristics of each alternative are extracted to form one optimal alternative, which is discussed in the next chapter. Tables 6 through 9 are shown below.

#### Table 6: Qualitative analysis of STINC

<table>
<thead>
<tr>
<th>Advantages</th>
<th>Disadvantages</th>
</tr>
</thead>
<tbody>
<tr>
<td>Multiple payload spacecraft catch/release</td>
<td>STINC design + TLS design</td>
</tr>
<tr>
<td>Reduced number of rendezvous</td>
<td>Need new STINC every 3 payload spacecraft</td>
</tr>
<tr>
<td>Payload spacecraft needs less propulsion</td>
<td>Greater momentum exchange</td>
</tr>
<tr>
<td>Room to hold tether on STINC</td>
<td></td>
</tr>
</tbody>
</table>

#### Table 7: Qualitative analysis of SIMPL

<table>
<thead>
<tr>
<th>Advantages</th>
<th>Disadvantages</th>
</tr>
</thead>
<tbody>
<tr>
<td>Multiple payload spacecraft catch/release</td>
<td>Complex attitude control algorithms</td>
</tr>
<tr>
<td>Room for back-up tethers</td>
<td>Increased number of rendezvous</td>
</tr>
<tr>
<td>Less momentum exchange</td>
<td></td>
</tr>
</tbody>
</table>

#### Table 8: Qualitative analysis of HEX

<table>
<thead>
<tr>
<th>Advantages</th>
<th>Disadvantages</th>
</tr>
</thead>
<tbody>
<tr>
<td>Multiple payload spacecraft catch</td>
<td>Single tether – no room for back-up tethers</td>
</tr>
<tr>
<td>Spinning momentum exchange (larger ΔV)</td>
<td>Spinning control</td>
</tr>
<tr>
<td>Extended tether catch</td>
<td></td>
</tr>
<tr>
<td>Advantages</td>
<td>Disadvantages</td>
</tr>
<tr>
<td>------------------------------------------------</td>
<td>--------------------------------------</td>
</tr>
<tr>
<td>Multiple payload spacecraft catch/release</td>
<td>Multiple TLS expenses</td>
</tr>
<tr>
<td>Faster recovery</td>
<td>Limited subsystems in GEO</td>
</tr>
<tr>
<td>Less momentum exchange on each TLS</td>
<td></td>
</tr>
</tbody>
</table>

Through examination of these tables and the high-level ranking system, the best characteristics of each alternative are chosen for the final concept, and new elements are added to better the concepts contained in the alternatives. The host spacecraft concept from STINC is a fundamental piece of system architecture that adds creativity and simplicity to the TLS. Using the tether for power generation and electrodynamic propulsion adds an element of hardware redundancy and creativity to the TLS design. Rendezvous with the tether extended and performing tether release operations, which may potentially be executed by all four alternatives, enables the TLS to provide the payload spacecraft with large $\Delta V$s. The lunar slingshot orbital geometry, which also may be used by any of the alternatives, increases the transfer capabilities of the TLS, and adds creativity to the mission. Placing the TLS into a highly elliptical orbit extends the lifetime of the power system.

### 4.3 Summary

Through examination of both the objective ranking and qualitative analysis of the system alternatives, it is concluded that each possesses its own distinguishing characteristics. These characteristics are extracted from their corresponding alternative and optimized in the following chapter. These characteristics will ultimately comprise the final TLS system.
Chapter 5: System Design Optimization

This chapter discusses the ways in which the favored elements of each system alternative are optimized to arrive at the final TLS. The previous chapter led to the extraction of various system elements from the alternatives of Chapter 3, and a conceptual TLS formed. Chapter 5 analyzes and attempts to optimize each of these system elements to decide if the TLS concept is feasible, practical, and optimal.

5.1 Stack element

The purpose of launching the payload spacecraft in a stack is to minimize the number of rendezvous required during the mission cycle. The rendezvous of the payload spacecraft with the TLS is one of the most complex aspects of the mission. By integrating some number of individual payload spacecraft into a single spacecraft, less rendezvous and capture operations occur. The number of rendezvous decreases as the number of payload spacecraft integrated in each stack increases.

The stacks are integrated into a host spacecraft, as in the STINC alternative, which integrates with the launch vehicle. The STINC supplies power and instrumentation for rendezvous with the TLS and separation of the stack, and includes the structural interface required for capture by the TLS. This element is advantageous because it involves all systems needed in LEO. If GPS, radar, and cross-link communications are used for autonomous rendezvous and capture, the STINC can carry these systems. The Mars orbiting payload spacecraft save mass and money on systems that will only work in LEO.
Three disadvantages of the stack concept include added design, fabrication, and mission mass. The STINC must be designed and its subsystems and interfaces with both the TLS and payload spacecraft defined, adding to the overall complexity of the mission. The mass of STINC and its multiple payload spacecraft add to the mass that the TLS must transfer. Therefore, higher stresses are induced on the TLS’s tether. In addition, performing a momentum transfer on such a massive spacecraft decreases the orbit of the TLS significantly after release.

5.2 Tether power generation element

Tether power generation adds an element of efficiency to the TLS system. This concept allows for overlapping subsystems, which decreases the total amount of subsystem hardware inside the TLS. The tether generates power while moving through the Earth’s magnetic field. This section describes some important factors relating to tether power generation while describing how this type of power system is modeled.

The density of the Earth’s magnetic field is an important design parameter for tether power generation. The magnetic field density is related to altitude, solar illumination and sun spot activity. The strength of the magnetic field has the greatest impact on the amount of power the tether can generate. Variations in magnetic field density and daily changes in sunspot activity cause the produced tether voltages to fluctuate between 1500 and 4500 volts. The amount of power generated is affected by orbital speed: voltage increases with increase in tether velocity through the magnetic field.\(^\text{13}\)

The following equations outline the relations pertaining to tether power generation. The radiated power density is given by:
\[ W_T = \varepsilon \sigma (T^4 - T_{BK}^4) \]  
\[ W_B = \sigma_c B_H^2 V_C^2 \]  
where \( \varepsilon \) is the emissivity of the tether material, \( \sigma \) is the Stefan-Boltzmann constant, \( T \) is the temperature of the tether, and \( T_{BK} \) is the background temperature. The Ohmic power density is given by:

\[ \eta_g = 1 - \left( \frac{4\pi W_T^2}{W_B} \right)^{\frac{3}{2}} \frac{1}{X^{\frac{3}{2}}} \]

The tether power efficiency, including only cable losses, is given by:

\[ D = \left( \frac{2}{\pi} \right)^{\frac{3}{2}} \frac{X^{\frac{3}{2}}}{(W_T W_B)^{\frac{3}{2}}} \]

The cable specific mass for all \( N \) cables is given by:

\[ \frac{M}{P_N} = \frac{\rho}{(4\pi)^{\frac{3}{2}} (W_T W_B)^{\frac{3}{2}}} \]

where \( \rho \) is the cable material density.
Table 10 illustrates sample calculations for a power-generating tether. This example assumes one cable, 100 km in length, constructed of aluminum. A circular orbit is assumed to be 200 km above the Earth’s surface. These calculations show that to create 100 kW of power, a tether diameter of 5.74 mm is required. The relevant variable names and descriptions and the values used for this calculation are displayed. The table shows, where applicable, which subsystem most significantly influences the value used. The relevant subsystems choose the values of these parameters and use them in this model.

Table 10: Tether power sample calculation

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
<th>Units</th>
<th>Description</th>
<th>Defined by:</th>
</tr>
</thead>
<tbody>
<tr>
<td>σ</td>
<td>5.67E-08</td>
<td>W/m²K⁴</td>
<td>Stefan-Boltzmann constant</td>
<td>Constant</td>
</tr>
<tr>
<td>T_BK</td>
<td>250</td>
<td>K</td>
<td>Background temperature</td>
<td>Constant</td>
</tr>
<tr>
<td>ε</td>
<td>0.4</td>
<td>-</td>
<td>Emissivity of tether</td>
<td>Thermal</td>
</tr>
<tr>
<td>T</td>
<td>273</td>
<td>K</td>
<td>Temperature of tether</td>
<td>Thermal</td>
</tr>
<tr>
<td>ρ</td>
<td>2.70E+03</td>
<td>kg/m³</td>
<td>Cable density</td>
<td>Structures</td>
</tr>
<tr>
<td>P_N</td>
<td>100</td>
<td>kW</td>
<td>Nominal power</td>
<td>Power</td>
</tr>
<tr>
<td>l_E</td>
<td>20</td>
<td>km</td>
<td>Tether length</td>
<td>Orbit</td>
</tr>
<tr>
<td>N</td>
<td>1</td>
<td>-</td>
<td>Number of cables</td>
<td>Structures</td>
</tr>
<tr>
<td>σ_c</td>
<td>3.50E+07</td>
<td>1/Ωm</td>
<td>Cable conductivity</td>
<td>Structures</td>
</tr>
<tr>
<td>V_c</td>
<td>7784</td>
<td>m/s</td>
<td>Orbital velocity (200km altitude)</td>
<td>Orbit</td>
</tr>
<tr>
<td>B_H</td>
<td>2.60E-05</td>
<td>Gauss</td>
<td>Magnetic field strength</td>
<td>Orbit</td>
</tr>
<tr>
<td>W_T</td>
<td>37.39</td>
<td>W</td>
<td>Radiated power density</td>
<td>Output</td>
</tr>
<tr>
<td>W_B</td>
<td>1434</td>
<td>kW</td>
<td>Ohmic power density</td>
<td>Output</td>
</tr>
<tr>
<td>X</td>
<td>5</td>
<td>kW/km</td>
<td>Tether parameter</td>
<td>Output</td>
</tr>
<tr>
<td>D</td>
<td>5.74</td>
<td>mm</td>
<td>Tether diameter</td>
<td>Output</td>
</tr>
<tr>
<td>η_g</td>
<td>0.897</td>
<td>-</td>
<td>Efficiency (includes only wire losses)</td>
<td>Output</td>
</tr>
<tr>
<td>M/P_N</td>
<td>1.397</td>
<td>kg/kW</td>
<td>Cable specific mass</td>
<td>Output</td>
</tr>
</tbody>
</table>

In addition to the amount of power a tether can produce, the tether’s survivability from orbital debris impact is considered. Equation 8 gives the probability of $n$ tether
cables surviving out of $N$ total cables if the survival probability of a single tether is given as $P_s$.

$$P_s(N) = \sum_{k=0}^{N} \binom{N}{k} p_s^k [1 - p_s]^{N-k}$$  \hspace{1cm} (8)$$

Intuitively, a short cable has a greater survivability rate than a long one since it presents less of a target for orbital debris. For the same reason, implementing spacing in between the cables is desirable. In addition, cable redundancy is advantageous.\textsuperscript{13}

Many design parameters must be considered before a tether power system is implemented. The tether material must be electrically conductive to generate power. This constrains the tether length if low mass is an objective since electrically conductive materials are generally high density. In addition, the mission requirements dictate the orbit of the TLS, and thus the magnetic field density. Therefore, other mission elements must be decided upon before a tether power system is chosen as a feasible, practical system element.

### 5.3 Electrodynamic propulsion element

Electrodynamic propulsion uses the Earth’s magnetic field to apply a $\Delta V$ to the TLS. While dragging the TLS through the magnetic field, electricity is sent through the tether. The space plasma in the magnetosphere completes the electric circuit, much the same way lightning completes its electric circuit. Ampere’s Force Law governs the resulting force on the TLS:

$$F = J \times B$$  \hspace{1cm} (9)$$

In this equation, $F$ is the resulting force vector, $J$ is the current density vector, and $B$ is the magnetic field vector at the altitude of the TLS. The component of $B$ that results in a force in the direction of motion is given by:
\[ B = B_0 \sin \Theta \left( \frac{R_e}{R} \right)^3 \]  

(10)

where \( B_0 \) is the magnetic field strength at the surface of the Earth, \( 3.5 \times 10^{-5} \) Tesla, \( R_e \) is the radius of the Earth, \( R \) is the mean radius of the TLS, and \( \Theta \) is the inclination of the orbit \( \pm 11^\circ \) to account for the difference between the geographic north pole and the magnetic north pole.\(^{13}\)

The tether naturally points towards the Earth through gravity gradient stabilization, and the magnetic field generates a force in the direction of motion or opposing the direction of motion, depending on whether the current is applied up or down the tether. By varying the amount of current in the tether, it is possible to spin the TLS. The propulsive force is related to the \( \Delta V \) applied to the TLS through Newton’s second Law:

\[ \Delta V = \frac{F \Delta t}{m} \]  

(11)

The \( \Delta V \) is then related to a change in altitude, \( \Delta R \), using the following equation, which is derived from the equation for the velocity of a circular orbit:

\[ \Delta R = \frac{\mu}{(V_1 - \Delta V)^2} - R \]  

(12)

In this equation, \( \mu \) is the gravitational constant of the Earth, \( 398,601 \text{ m}^3/\text{s}^2 \), and \( V_1 \) is the velocity before the propulsion system is activated.\(^{2}\)

For a 20,000 kg TLS in a 400 km circular orbit with 20 degrees of inclination, a 20 km long tether supplied with 1 A of current and 1500 V of electricity for propulsion, the above equations determine the increase in altitude per day. Equation 10 determines the average magnetic field value that is encountered by the TLS to be approximately \( 9.97 \times 10^{-6} \) Tesla. Equation 9 then determines that the force applied by the Earth’s field
is 0.02 N. This force acting over the entire day results in a decrease of $8.6 \times 10^{-5}$ km/s and an increase of 1.5 km of altitude using Equations 11 and 12 respectively. By diverting 1500 W of power to the tether system the TLS get a practically free altitude boost. The $\Delta R$ is directly proportional to the amount of current in the tether, so if 2 A of current is applied a 3.0 km per day increase results.

### 5.4 Extended tether rendezvous element

Rendezvous of a payload spacecraft with a retracted tether is ideal only if a small $\Delta V$ is needed. However, a system required to send a massive payload spacecraft on an LST requires large $\Delta V$s. Large $\Delta V$s require extended tether retrieval. Extended tether captures are more difficult than retracted tether captures, but they are not impossible. The margin of error in predictions of the payload and tether tip positions and velocities is required to be very low. The payload spacecraft is in an orbit properly phased so the position is accurate at the rendezvous time. Some type of navigation system, such as GPS, is placed on the payload spacecraft as well as the end of the tether so that rendezvous occurs at the right time and at the right place. The tether’s grappling device grabs the payload spacecraft at the appropriate time and makes a strong connection in 5-15 seconds. 7

Sending a small payload from a circular LEO to a lunar transfer orbit requires a $\Delta V$ of approximately 3 km/s. To accomplish this $\Delta V$, the retracted tether on the TLS rendezvous with the payload spacecraft. The tether then extends and uses electrodynamic propulsion to spin the tether until the tether tip velocity is 3 km/s. Once this desired tip velocity is achieved, the payload is released into a LST. 9
This example seems simple, but it contains errors. A circular orbit is unreasonable if large $\Delta V$s are required since the momentum transfer can be strong enough to send the TLS back into the Earth’s upper atmosphere. Also if electrodynamic propulsion is used to spin the TLS, its tether must be conductive, and conductive materials are heavy. The tether mass dependence on the tether tip velocity makes large $\Delta V$s impractical with current materials. The critical velocity, $V_{\text{crit}}$ is the maximum velocity at which a hoop made of the tether material can be safely spun and is given by:

$$V_{\text{crit}} = \sqrt{\frac{2U}{\rho FS}}$$

(13)\(^4\)

where $U$ is the ultimate strength of the tether, $FS$ is a factor of safety, and $\rho$ is the density of the tether material. Spectra is a high strength, low-density composite tether material, and it has a critical velocity of approximately 1.66 km/s for a factor of safety of 3. The highest possible critical velocity is desired to minimize the mass of the tether. For the above example, even if Spectra is used, the mass of the tether must be 100 times the payload spacecraft mass to accomplish the 3.1 km/s $\Delta V$ desired.\(^9\)

If the ratio of $\Delta V / V_{\text{crit}}$ is reduced to 1, the tether mass reduces to reasonable levels. This $\Delta V$ reduction is accomplished by splitting the $\Delta V$ into two parts. The TLS orbits in an elliptical orbit, which helps prevent re-entry and allows two $\Delta V$s. At perigee, the end of the extended tether captures the payload spacecraft, which is in a lower altitude LEO. Since the payload spacecraft is in a lower altitude LEO, the capture gives approximately a 1.5km/s $\Delta V$ to the payload. When the tether returns to perigee, the TLS releases the tether giving it an additional 1.5 km/s $\Delta V$. Extended tether rendezvous
allows for smaller individual $\Delta V$s, reduces the orbit requirements on the payload spacecraft, and reduces the mass requirements on the tether.\(^9\)

### 5.5 Tether release element

Tether dynamics are altered to supply $\Delta V$s of varying magnitudes. Releasing a payload spacecraft from the end of a vertically extended tether transfers momentum due to the gravity gradient and boosts the payload with minimal $\Delta V$. Larger $\Delta V$s are achieved by lengthening the tether. “Pumping” a tether by varying the length in resonance with tether libration causes the tether to swing or spin. Winching the tether inward quickly transfers momentum to the payload spacecraft. Figure 9 shows the effect of altitude on various types of tether deployment and release.\(^4\)

![Example diagram showing effects of tether releases](image)

**Figure 9: Effects of various types of releases**\(^4\)

A simple approximation for the $\Delta V$ that can be produced from the release of a payload spacecraft is given by the equation:

$$\Delta V = l_z \omega + l_z \sqrt{\frac{\mu}{a^3}}$$  \hspace{1cm} (14)
where $l_2$ is the distance from the center of mass of the combined system to the payload spacecraft, $\omega$ is the angular velocity of the system induced by spinning about the center of mass, and $a$ is the semi-major axis of the system’s orbit. Spinning the system contributes an extra velocity element, which is shown as the first element of Equation 14. This element contributes approximately 95% of the total $\Delta V$ to the payload spacecraft.

### 5.6 Lunar slingshot element

The purpose of a LST is to provide a velocity boost to the payload spacecraft without the expenditure of propellant or the loss of momentum from the TLS. A reduction in the $\Delta V$ required by the payload spacecraft’s propulsion system allows for an increase in the length of the launch window or capability. For a given propellant-based propulsion system, a reduction in the required change in velocity reduces the mass of the propellant needed. A reduction in the mass of a spacecraft reduces launch costs, as well as propellant expenditures required for other maneuvers.

A LST is a gravity-assist trajectory using the Moon as the contributing celestial body. A momentum and energy transfer occurs between the spacecraft and the contributing celestial body by means of the gravitational attraction of the two bodies. A significant change in the velocity of the spacecraft occurs as a result. Because the celestial body is more massive than the payload spacecraft, the effect on its velocity is negligible.\(^{11}\)

The $\Delta V$ given to the payload spacecraft is calculated using Equation 15.\(^{11}\)

$$\Delta V = v_f - v_i = \frac{V \sin \beta + v_i \sin(\alpha - \beta)}{\sin \alpha'} - v_i$$

The symbols $v_f$ and $v_i$ represent the velocity relative to the Earth before and after the gravity-assist maneuver. The angles $\alpha$ and $\alpha'$ are the angles between the velocity of the
Moon, \( V \), and \( v_i \) and \( v_f \). The angle between the initial and final velocities relative to the Moon is represented as the symbol \( \beta \). A geometric definition of the angles is demonstrated graphically in Figure 10. The dotted lines represent the initial and final velocities of the payload spacecraft relative to the Moon.

![Figure 10: Vector illustration of lunar slingshot maneuver](image)

The National Aeronautics and Space Administration has previously used gravity-assist maneuvers. The Microwave Anisotropy Probe mission employed a lunar gravity-assist maneuver while traveling to the second Langrangian point.\(^3\) The Voyager 2 probe used Jupiter, Saturn, and Uranus as contributing celestial bodies to reach the outer solar system.\(^{15}\) The quantitative benefit of these maneuvers is observed in the Cassini mission. Cassini achieved a 7 km/s velocity boost in the first Venus flyby, and a 5.5 km/s boost due to its Earth flyby.\(^{11}\)

### 5.7 Power system element

A satellite’s orbit affects any solar power system. The time spent in sunlight is used for power generation, and stored energy is used in eclipse. The lifetime of batteries
depends on their depth of discharge during eclipse. The depth of discharge depends on the amount of energy used and the duration of their use. If the amount of energy used is held constant, minimizing the duration of discharge maximizes the lifetime of the batteries. Therefore, in order to maximize the lifetime of the power system, an orbit is chosen to minimize time in eclipse. To maximize the lifetime of the power system, minimizing the charge/discharge cycles is also important. An elliptical orbit achieves both of these objectives.

The period of the orbit depends on the semi-major axis of the orbit. For a circular orbit, the semi-major axis is the altitude of the orbit plus the radius of the Earth. For an elliptical orbit, the semi-major axis of the orbit is the average of the radius at apogee and the radius at perigee. An elliptical orbit has a larger semi-major axis and, therefore, a larger period than a circular orbit at the elliptical orbit’s perigee altitude. An elliptical orbit allows the spacecraft to perform its duties at a lower orbit without sacrificing semi-major axis size. The elliptical orbit’s larger period reduces the number of eclipse cycles the spacecraft has in its lifetime, and thus, it has a larger lifetime than the circular orbit at the perigee altitude.

In a circular orbit, the TLS moves with relatively constant velocity. In an elliptical orbit the TLS’s velocity changes throughout the orbit. At elliptical perigee, the spacecraft moves faster than the circular speed at that same altitude. If the relative position of the earth, sun, and orbital plane is held constant, the perigee of the orbit can be in constant eclipse. Because it moves very fast at perigee, the TLS decreases its time spent in eclipse, lowering the batteries’ depth of discharge and increasing the power system’s lifetime.
5.8 System optimization

An optimization code determines the optimal characteristics of the TLS. The mathematical optimization problem is defined in a MatLab® “m-file” contained in Appendix 1. This file is linked with VisualDOC v2.17 optimization software to solve the given problem. For a given tether material, assuming a single 200 kg payload spacecraft launch, the optimization program maximizes the $\Delta V$. This $\Delta V$ is defined as the difference between the circular velocity of the center of mass of the TLS/payload spacecraft with the tether extended and the velocity of the payload spacecraft at release. The design variables used in this optimization code include the mass of the TLS, the length of the tether, the orbital radius of the center of mass, the angular velocity about the center of mass, and the diameter of the tether.

The optimization problem has several constraints that limit the design space. The RFP supplies a 3-g acceleration limit on the payload spacecraft.\textsuperscript{1} The maximum stress in the tether must not exceed the ultimate strength for a given tether material. The total volume of the tether must not exceed the volume of a 4 m diameter $\times$ 4 m long cylinder.

Multiple trials using the optimization program inputting different initial conditions uncovered several facts. The majority of the $\Delta V$ gained is due to the spinning of the TLS, not the altitude change. In all cases, the 3-g acceleration limit and the ultimate stress limit are the active constraints. These active constraints show that the angular velocity and tether diameter are the most influential design variables. To appreciably increase the $\Delta V$, the angular velocity must increase, therefore increasing the stress on the tether. Increasing the cross-sectional area of the tether, which is dependent on the tether’s diameter, decreases the stress on the tether. Therefore, to obtain the
largest amount of $\Delta v$ possible, the tether strength and diameter must be large, and the spacecraft must be rotated quickly.

5.9 Summary

The system elements have been evaluated for feasibility, practicality, and have been optimized for application in the final TLS system. Some of the system elements are incapable of working simultaneously or alternately. The final TLS system will incorporate as many of these system elements as is reasonable. The last chapter selects the most beneficial system elements and presents the TLS system chosen for further analysis and design.
Chapter 6: Final Decision

The system optimization chapter describes the system elements chosen for the final design. Following the system optimization step in the design phase, the chosen elements are integrated together to form the final design concept. This chapter identifies the specific elements of the final design and how they work together. The final decision chapter also discusses several mission operations concepts such as momentum launch procedures, TLS maintenance, and alternative mission capabilities.

6.1 The SMART System

Through the development of desired system elements, analysis of those elements, and optimization of element combinations, the Stair-stepping to Mars Autonomous Rendezvous Tether System (the SMART System) is conceptualized. The following subsections describe the elements of the SMART System and how they interact.

6.1.1 Configuration

The SMART System is composed of two TLSs: one positioned in LEO and one positioned in MEO. The payload spacecraft must be picked up in LEO so a TLS is necessary in LEO. However, a single TLS in LEO is unable to supply the required \( \Delta V \) for hyperbolic trajectories within realistic design constraints. The momentum exchange needed for a LEO to LST transfer degrades the TLS’s orbit significantly, and requires an excessive tether length.

The SMART System sends payload spacecraft from the LEO TLS to the MEO TLS instead of sending them from LEO directly into a LST. This reduces the amount of momentum exchange imposed on the LEO TLS and reduces the spin angular velocity
required of both TLSs. This stair-stepping method of LEO-MEO-LST-MTO places fewer requirements on the each of the TLSs compared to the LEO-LST-MTO transfer. The payload spacecraft launch from LEO to MEO is accomplished before the Earth-Mars alignment to maximize the available launch window. However, the SMART System requires design and production of two TLSs and two rendezvous.

6.1.1.1 Payload spacecraft

The SMART System TLSs integrate with stacked payload spacecraft. Each payload spacecraft is equipped with identical cone-inverse cone rendezvous mechanisms. Multiple payload spacecraft are stacked together in a launch vehicle like Legos™ and launched into high LEO. The first stack launched rendezvous with the SMART System. The next stack rendezvous with the payload stack already attached to the SMART System’s LEO TLS. Figure 11 is an illustration of the multiple stack rendezvous.

![Payload spacecraft stack captured by extended tether](image)

The rendezvous mechanisms between each payload spacecraft are identical. Therefore, both TLSs are capable of collecting multiple stacks of payload spacecraft.
6.1.1.2 Rendezvous and release

The two TLSs in the SMART System perform the same rendezvous and release maneuvers. Rendezvous is autonomous, using cross-link communications on board the payload spacecraft stacks and each TLS. Both TLSs rendezvous with the payload spacecraft using an extended tether to capture the stacks in a lower orbit. The extended tether rendezvous decreases the initial orbital insertion requirement on the payload spacecraft stacks. A spinning release generates the necessary $\Delta V$ for payload spacecraft transfers from both TLSs.

The spinning motion of the TLS/payload spacecraft configuration adds an extra $\Delta V$ to the total $\Delta V$ that the momentum transfer gives the payload spacecraft stack. Equation 14 from Chapter 5 includes this added $\Delta V$ to calculate the total $\Delta V$ gained from the momentum transfer maneuver. The $\Delta V$ gained from spinning the configuration contributes approximately 95% of the total $\Delta V$ that the payload spacecraft receives through momentum transfer.

6.1.2 Orbits

The SMART System TLSs are launched into LEO and MEO stand-by orbits. The first TLS is in an elliptical orbit with the perigee in LEO. The estimated perigee altitude of the stand-by orbit is between 400 km and 1000 km. The second TLS is placed in a circular orbit in MEO, with an altitude of approximately 10,000 km, to increase the possible number of payload spacecraft stack rendezvous attempts. The following is a list of the steps involved during the payload spacecraft stack transfer to Mars. Figure 12 is an illustration of these steps.
1. The payload spacecraft stack launches into a circular orbit slightly lower than the perigee altitude of the LEO TLS.

2. The LEO TLS uses an extended tether autonomous rendezvous to capture the payload spacecraft at the perigee of its stand-by orbit.

3. The LEO TLS/payload spacecraft stack configuration spins during multiple cycles of its new orbit.

4. The payload spacecraft stack is launched from the LEO TLS at the configuration’s orbit perigee.

5. The LEO TLS re-boosts to its original stand-by orbit using on-board propulsion.

6. The MEO TLS uses an extended tether autonomous rendezvous to capture the payload spacecraft stack.

7. The MEO TLS/payload spacecraft stack configuration spins during multiple cycles of its new orbit.

8. When the MEO TLS/payload spacecraft stack combination is in the correct phase with the Moon, the payload spacecraft is released into a LST.

9. The MEO TLS re-boosts to its original stand-by orbit using on-board propulsion.

10. The payload spacecraft stack separates following launch from MEO.

11. Each payload spacecraft receives a velocity boost as a result of the gravitational influence of the Moon.

12. The payload spacecraft each perform burns after the lunar gravity-assist maneuver is completed to achieve a MTO.
13. After approximately 9 months, the payload spacecraft reach Mars, and burns are performed to enter into a highly elliptical orbit with a perigee of approximately 800 km.

14. Aero-braking maneuvers can be performed to lower the apogee of the payload spacecraft’s orbit until a circular orbit is attained.

Both TLS stand-by orbits degrade upon capture of payload spacecraft stacks. The LEO TLS captures the payload spacecraft at perigee. The combined (launch) orbit of the LEO TLS and payload spacecraft stack is a lower energy elliptical orbit compared to the original LEO TLS stand-by orbit. The energy decreases because the lower energy circular orbit of the payload spacecraft combines with the higher energy elliptical orbit of the LEO TLS. Upon capture of the payload spacecraft stack, the MEO TLS drops to a new launch orbit.

Loss of momentum after release causes both TLS orbits to degrade. Each TLS must then re-boost into their respective stand-by orbits to prepare for the next rendezvous cycle. The LEO TLS captures and releases the stack at launch orbit perigee, so the altitude loss occurs at apogee, which is high enough to avoid degrading into the Earth’s atmosphere.
6.1.3 Power system

To generate power, both TLSs use solar panels to capture the energy from the sun and turn it into electricity, thus providing a constant source of power to the SMART System. In addition to solar panels, the LEO TLS uses a conductive tether to augment its power generation. The MEO TLS is at an altitude where the magnetic field of the Earth is too weak for tether power generation. Placing the LEO TLS in an elliptical orbit extends the lifetime of its batteries. An elliptical orbit has a longer period than a circular orbit, which extends battery lifetime.

To store the energy produced by the solar panels and the tether, the SMART System uses a combination of batteries and flywheels. Batteries are used to provide power to all TLS components during eclipse. Flywheels are used for peak power
operations, such as winching the tethers or deploying the solar panels. Flywheels are coupled and used as reaction wheels for attitude control.

6.1.4 Tether

Tethers used in space applications are susceptible to failure due to orbital debris. Therefore, it is important to design the tether so that it survives orbital debris impacts. The SMART System tether is modeled after the Hoytether™ design to increase the survival probability of the TLS over mission lifetime. The Hoytether™ is a multi-line, braided tether. The lines of the tether that support the tension load placed on the tether are tied together so that if one line breaks its load is transferred to the surrounding lines. Figure 13 is a picture of the Hoytether™ construction.8

![Hoytether™ multi-line configuration](image)

Figure 13: Illustration of Hoytether™ multi-line configuration

The vertical lines of the tether in Figure 13 bear the axial loads imposed on the tether. When the tether is impacted by orbital debris and one of the load-bearing lines breaks, the load is distributed to the interconnecting lines around the break. This type of tether construction extends the tether’s lifetime past that of single-line tethers. The
survivability of the Hoytether\textsuperscript{TM} compared with a single-line tether is shown in Figure 14. As Figure 14 shows, using the Hoytether\textsuperscript{TM} design increases the lifetime of the tether.

![Figure 14: Plot of Hoytether\textsuperscript{TM} survival probability compared to single-line tethers\textsuperscript{8}](image)

Based on approximate calculations, tether length and TLS orbits are determined with the objective of minimizing the amount of its own propellant expended prior to the LST. The amount of $\Delta V$ that the tether produces is maximized while keeping tether lengths and orbital altitudes reasonable. A circular MEO orbit of 10,000 km and circular LEO orbits ranging in altitude from 400 km to 2,000 km are examined. Tether lengths varying from 20 km to 100 km are also examined.

The amount of $\Delta V$ produced by the tether is determined by Equation 14:

$$\Delta V = l_2 \omega + l_2 \sqrt{\frac{\mu}{a^3}}$$  \hspace{1cm} (14)

The angular velocity of the system is constrained by the 3-g acceleration limit on the payload spacecraft. The maximum possible angular velocity is determined from Equation 16:

$$a_c = \omega^2 l_2$$  \hspace{1cm} (16)
where $a_c$ is the centripetal acceleration (limited to 3-g’s), and $l_2$ is the distance from the center of mass of the system to the payload spacecraft stack (the moment arm). The escape $\Delta V$ from MEO is calculated based on a parabolic trajectory by Equation 17:

$$\Delta V = \sqrt{\frac{2\mu}{r}} - \sqrt{\frac{\mu}{r}}$$
(17)

where $r$ is the circular radius of the MEO orbit ($r = 10,000$ km).

These calculations are performed for several combinations of tether length and orbit. The amount of payload spacecraft $\Delta V$ is determined by subtracting the $\Delta V$ given by the tether from the required $\Delta V$ for the specific transfer. The results are calculated using a LEO TLS orbit altitude equal to 1,000 km and a MEO TLS orbit altitude equal to 10,000 km and are shown in Table 11. Based on these approximate calculations, the optimal orbit for the LEO TLS is at a perigee altitude of 1,000 km and the optimal tether length is between 80 km and 100 km. Though approximate, since the actual orbits are elliptical, these calculations give a rough idea of orbits and tether length.
Table 11: Results of calculations performed for preliminary determination of orbits and tether lengths

<table>
<thead>
<tr>
<th>Tether Length (km)</th>
<th>$l_2$ (km)</th>
<th>$\omega$ (rad/sec)</th>
<th>$\Delta V$ from tether (km/sec)</th>
<th>$\Delta V$ needed for Hohmann transfer to MEO (km/sec)</th>
<th>$\Delta V$ from payload spacecraft needed to complete Hohmann transfer to MEO (km/sec)</th>
</tr>
</thead>
<tbody>
<tr>
<td>20</td>
<td>19.05</td>
<td>0.039</td>
<td>0.77</td>
<td>1.28</td>
<td>0.51</td>
</tr>
<tr>
<td>50</td>
<td>47.62</td>
<td>0.025</td>
<td>1.23</td>
<td>1.28</td>
<td>0.05</td>
</tr>
<tr>
<td>80</td>
<td>76.19</td>
<td>0.020</td>
<td>1.57</td>
<td>1.28</td>
<td>-0.29</td>
</tr>
<tr>
<td>100</td>
<td>95.24</td>
<td>0.018</td>
<td>1.77</td>
<td>1.28</td>
<td>-0.49</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Tether Length (km)</th>
<th>$l_2$ (km)</th>
<th>$\omega$ (rad/sec)</th>
<th>$\Delta V$ from tether (km/sec)</th>
<th>Lowest $\Delta V$ needed to escape (km/sec)</th>
<th>$\Delta V$ from payload spacecraft needed to complete escape (km/sec)</th>
</tr>
</thead>
<tbody>
<tr>
<td>20</td>
<td>19.05</td>
<td>0.039</td>
<td>0.75</td>
<td>2.04</td>
<td>1.29</td>
</tr>
<tr>
<td>50</td>
<td>47.62</td>
<td>0.025</td>
<td>1.20</td>
<td>2.04</td>
<td>0.85</td>
</tr>
<tr>
<td>80</td>
<td>76.19</td>
<td>0.020</td>
<td>1.52</td>
<td>2.04</td>
<td>0.52</td>
</tr>
<tr>
<td>100</td>
<td>95.24</td>
<td>0.018</td>
<td>1.70</td>
<td>2.04</td>
<td>0.34</td>
</tr>
</tbody>
</table>

The total $\Delta V$ needed from the payload spacecraft for both the LEO-MEO transfer and escape transfer is shown in Table 12. A 100 km tether produces enough $\Delta V$ so that the payload spacecraft does not use any of its own propulsion until it nears the Moon.

Table 12: Total $\Delta V$ required from payload spacecraft for all orbit transfers within the Earth's sphere of influence

<table>
<thead>
<tr>
<th>Tether Length (km)</th>
<th>Total $\Delta V$ needed from payload spacecraft (km/sec)</th>
</tr>
</thead>
<tbody>
<tr>
<td>20</td>
<td>1.80</td>
</tr>
<tr>
<td>50</td>
<td>0.90</td>
</tr>
<tr>
<td>80</td>
<td>0.23</td>
</tr>
<tr>
<td>100</td>
<td>-0.15</td>
</tr>
</tbody>
</table>

6.1.5 Maintenance

The operational lifetime of the TLS is 30 years over which some components are expected to fail. For mission success, these components must be either replaced or
repaired during the SMART System’s operational lifetime. One maintenance option is to send a manned mission to the LEO TLS in which astronauts can either repair or replace faulty components. Another option is to send a new TLS to MEO and to de-orbit the original MEO TLS. Each of these options is expensive and, therefore, must be included in the mission budget.

6.1.6 Alternate mission possibilities

Design and production of two TLSs is expensive. If the TLSs only launch six payload spacecraft each Earth-Mars alignment, the relativistic cost is impractical. For this reason, other missions must be considered for the interval between Earth-Mars alignments. One option for this down time is to rendezvous with other spacecraft and transfer power to them through the tether on the LEO TLS. Another option is to use the TLSs to boost orbits of government and commercial satellites. A final option is to use the MEO TLS for transfers outside of the Earth’s sphere of influence. These options can offset the cost of the entire SMART System.

The SMART System’s conductive tether allows the LEO TLS to generate power. The SMART System is designed so that its solar cells are used for primary power generation. Since the power generated from the tether is supplemental to this primary power supply, it is possible to rendezvous with another spacecraft and “give” them power. Because the power generated flows away from the Earth, the LEO TLS must be in a lower orbit than the receiving spacecraft.

The TLSs are capable of providing orbit boosts to any satellite with the correct docking mechanism. The TLSs can spin at varying rates to comply with individual satellite constraints. The LEO TLS can boost government or commercial satellites to an
orbit between LEO and MEO. If the satellite requires an additional boost into an orbit beyond the MEO TLS, the satellite may use that TLS to boost into the desired orbit. This option can also be used in reverse to lower satellite orbits. A GEO satellite that needs to return to LEO de-orbits to the MEO TLS, and can then be launched into a LEO orbit from that TLS.

The final option involves launching spacecraft to points in or out of the solar system. The SMART System is designed to launch spacecraft to Mars on a LST. Instead of continuing on a hyperbolic path around the Moon, the spacecraft initiates a burn and places itself into orbit around the Moon. The SMART System can also launch spacecraft into an escape trajectory during any Earth-planet alignment. Escape trajectories are not limited to planet missions. The SMART System can boost spacecraft to asteroids, comets, Langrangian points, or other locations in the solar system and beyond.

6.2 Summary

The final decision chapter introduces the system elements chosen and integrated into a complete design. These elements are optimized to work together and satisfy the mission objectives. The final chapter summarizes the design process, and it describes future plans for achieving a final optimal design that solves the problem and fulfills the objectives.
Chapter 7: Remaining Work and Conclusions

The final product of the engineering design process is the SMART System. Upon completion of the conceptual design, it is necessary to perform additional iterations of the design process to attain a detailed design of the SMART System. This detailed design includes all information required to manufacture, maintain, and operate the SMART System.

7.1 Conclusions

The value system design requires more detailed subsystem specifications and therefore is useful in later iterations of the design process. The three major factors in determining the conceptual design are creativity, operational complexity, and practicality. These high-level objectives differentiate between conceptual design alternatives. Through analysis using these high-level objectives, no single design alternative dominates. Rather than choosing one alternative, the beneficial aspects of each of the concepts are chosen to form the final conceptual design. Conclusions are drawn throughout the design process leading to the creation of the SMART System.

Several decisions involving the SMART System account for the high-level objectives. An important objective is to give the payload spacecraft as much $\Delta V$ possible to minimize the amount of propellant used. The SMART System therefore uses two TLSs, one in LEO and one in MEO. This configuration allows for the use of a stair-stepping architecture, which transfers the payload spacecraft from LEO to MEO, through a lunar fly-by and into a MTO. Another important objective is to minimize the number of rendezvous and re-boost operations performed by the TLS. The payload spacecraft are
integrated in a stack configuration, which allows for fewer rendezvous and less complexity.

7.2 Summary

Following the seven steps of the engineering design process leads to the development of the SMART System. First the problem and the scope are defined to determine the boundaries of the conceptual design. The possible solutions for the relevant elements are used to generate alternative TLS designs. High-level objectives are used to evaluate the different TLS options. System analysis shows that the different options are closely rated in value. Therefore the specific favorable aspects of the alternatives are analyzed and optimized where possible. From the analysis of these concepts, a TLS design is created. This design is named the Stair-stepping to Mars Autonomous Rendezvous Tether System.

7.3 Remaining work

The next iteration of the design process is developing a detailed design. Information needed to maintain, manufacture, and operate the SMART System is specified. This iteration includes the design of all the subsystems, a cost analysis of the SMART System, and the creation of a mission timeline.

The primary and secondary structures of each TLS must be designed. All necessary mechanisms need to be selected and sized. The structure needs to withstand the forces and moments applied to the spacecraft. Mechanisms to be defined include the docking mechanism, the tether winch motor, and the solar panel deployment mechanism.
Lifetime and power analyses of the mechanisms are performed. A thermal analysis of the structures and mechanisms is performed.

The design of the ADCS involves the determination of the actuators, sensors, and algorithms. The sizing of the ADCS actuators requires knowledge of the environmental disturbance moments and the desired slew rates of the spacecraft. The pointing accuracy requirements determine the characteristics of the attitude sensors chosen. The control algorithm created is influenced by the number of sensors, number of actuators, and the desired pointing accuracy.

Sizing the power system involves first determining the power requirements of the spacecraft’s components. The power required by the all the spacecraft components determines the size of the solar panels. To define the size of the energy storage system the orbits and the eclipse power requirements are defined.

The purpose of the SMART System is to provide a cost-effective means of supplying and replenishing constellation of communication/navigation satellites. A analysis comparing the cost of construction and operation of the SMART System to traditional interplanetary transport methods is created.

A mission timeline is created during the next iteration of the design process. The mission timeline includes operational procedures of the SMART System and the payload spacecraft. The timing and order of the mission procedures defines the elements of the mission timeline.
References


http://map.gsfc.nasa.gov


http://www.tethers.com/papers/JPC00MMOSTTT.pdf


Appendix 1: Optimization code

Program 1: Optimization function and constraints

```matlab
function [resp]=tether3(dvar)

rho=1560;  %density of Zylon kg/m^3
yu=5.8*10^9;  %ultimate strength of Zylon N/m^2
M1=200;  %mass of the payload spacecraft in kg

M2=dvar(1);  %TLS Mass kg
T=dvar(2);  %Tether Length m
R=dvar(3);  %Orbital Radius m
w=dvar(4);  %angular velocity rad/s
D=dvar(5);  %diameter of the tether m

% dV is the difference between the circular orbital velocity of
% the COM and the velocity of the payload

M3=((pi*D^2)/4)*T*rho  %mass of tether
CM = (M1*T+M3*(T/2))/(M1+M2+M3)  %center of mass measured from payload
Force=(M1+M3*((T-CM)/T))*w^2*(T-CM)  %centripital force

dV=w*(T-CM)+(T-CM)*sqrt(3.986*10^14/(R^3))^0.5  %objective function

g1=w^2*(T-CM)-3*9.8;  %acceleration constraint

g2=T/100000-1;  % tether length constraint

g3=R/20000*10^3-1;  %orbital radius constraint

g4=Force/((pi*D^2)/4)/yu-1;  %ultimate strength constraint

g5=((pi*D^2)/4)*T/(4*pi^2)-1;  %volume constraint

g6=(M2+M3)/25000-1;  %launch mass constraint

resp(1)=dV;
resp(2)=g1;
resp(3)=g2;
resp(4)=g3;
resp(5)=g4;
resp(6)=g5;
resp(7)=g6;
```