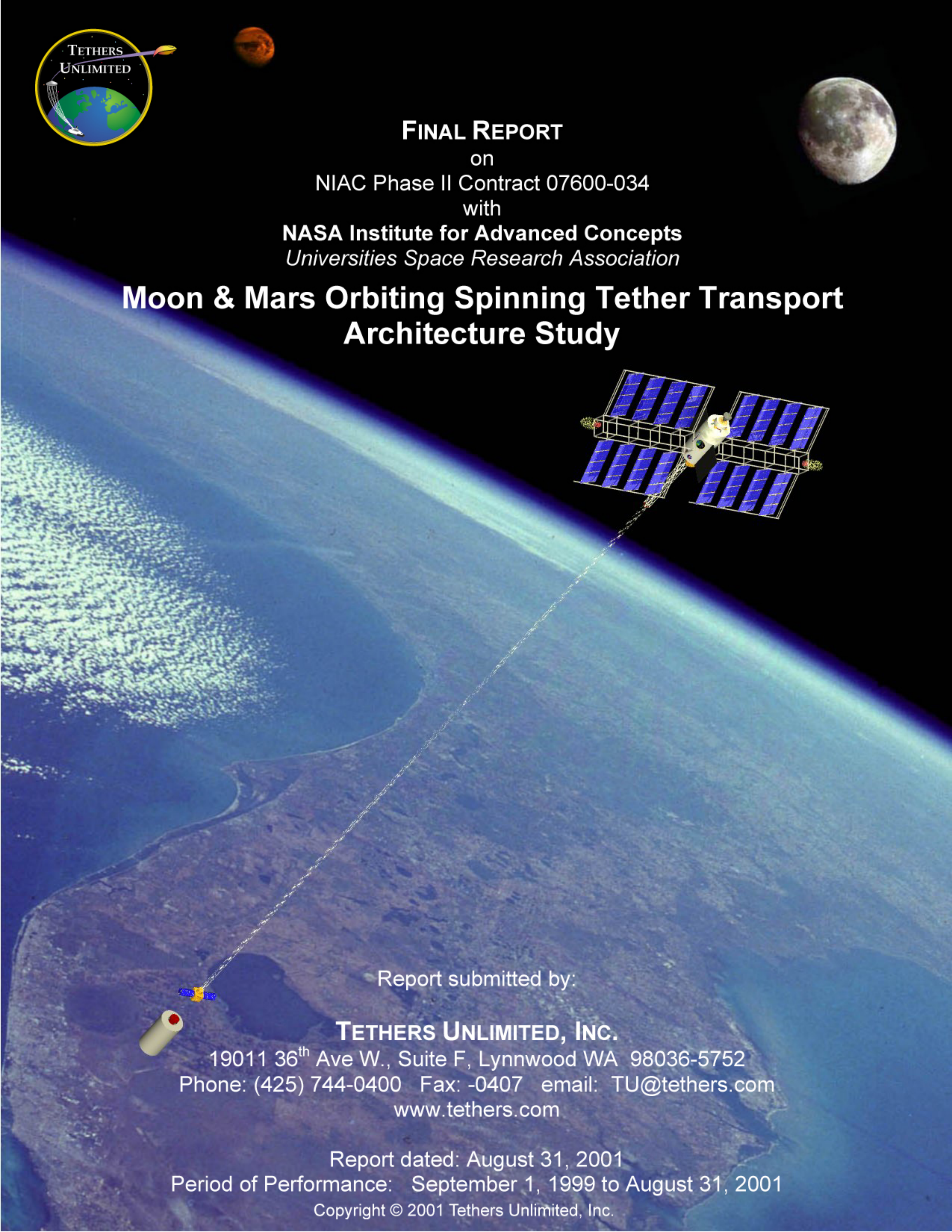

Final Report

on

NIAC Phase II Contract 07600-034

with

NASA Institute for Advanced Concepts

Universities Space Research Association

Moon & Mars Orbiting Spinning Tether Transport Architecture Study

Report submitted by:

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I. INTRODUCTION

Momentum-Exchange/Electrodynamic-Reboost (MXER) Tethers can provide a fully reusable, zero-propellant infrastructure for in-space transportation that will reduce by an order of magnitude or more the costs of delivering payloads to geostationary orbit, the Moon, Mars, and other destinations. This Phase II NIAC research program has continued the development of a tether-based architecture for in-space propulsion to service transportation needs in the Earth-Moon-Mars system. This tether architecture will utilize momentum-exchange techniques and electrodynamic tether propulsion to transport multiple payloads *with little or no propellant consumption*. The tether transport architecture is designed to be deployed incrementally, with each component able to perform a useful revenue-generating mission to help fund the deployment of the rest of the system. The Phase II effort has focused on the design of the first component of this architecture, a Tether Boost Facility optimized for transferring payloads from low Earth orbit (LEO) to geostationary transfer orbit (GTO). The resultant system concept uses a modular design that enables a single launch to deploy a fully-operational tether boost facility which can later be augmented to increase its payload capacity. The first component of the tether boost facility will be able to toss 2,500 kg payloads from a low-LEO initial orbit to GTO. This same facility will also be capable of boosting 1,000 kg payloads to lunar transfer orbit (LTO) or to escape via a lunar swingby. Additional launches of essentially identical modules can increase the payload capacity of the Tether Boost Facility to enable it to boost larger satellites and, eventually, manned spacecraft. This Tether Boost Facility can, in turn, be used to deploy components of additional tether facilities at the Moon and Mars, providing an infrastructure for frequent, low-cost transport between the Earth, the Moon, and Mars.

II. Background

Space tethers can accomplish propellantless propulsion through two mechanisms, through momentum-exchange between two space objects, and through electrodynamic interactions with a planetary magnetic field.

A. Momentum-Exchange Tethers

In a momentum-exchange tether system, a long, thin, high-strength cable is deployed in orbit and set into rotation around a central body. If the tether facility is placed in an elliptical orbit and its rotation is timed so that the tether is oriented vertically below the central body and swinging backwards when the facility reaches perigee, then a grapple assembly located at the tether tip can rendezvous with and capture a payload moving in a lower orbit, as illustrated in Figure 1. Half a rotation later, the tether can release the payload, tossing it into a higher energy orbit. This concept is termed a momentum-exchange tether because when the tether picks up and tosses the payload, it transfers some of its orbital energy and momentum to the payload, resulting in a drop in the tether facility's apogee.

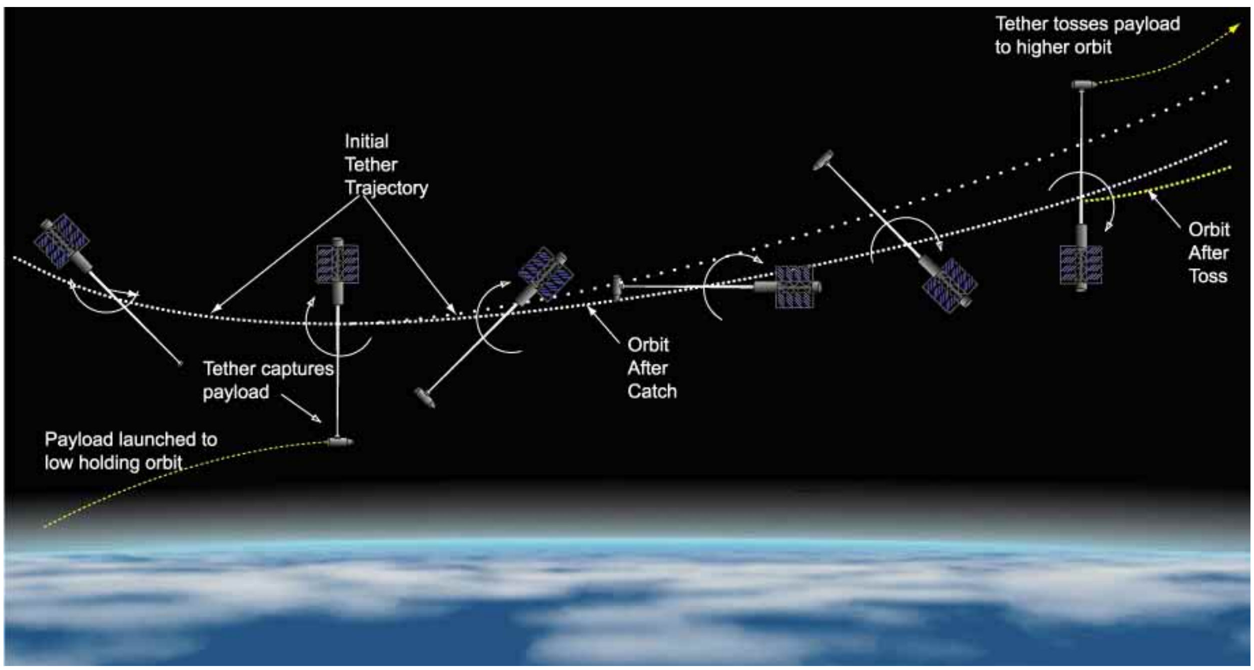


Figure 1. Concept of operation of a momentum-exchange tether facility. Orbits are depicted conceptually from the perspective of an observer on the Earth.

B. Electrodynamic Reboost

In order for the tether facility to boost multiple payloads, it must have the capability to restore its orbital energy and momentum after each payload transfer operation. If the tether facility has a power supply, and a portion of the tether contains conducting wire, then the power supply can drive current along the tether so as to generate thrust through electrodynamic interactions with the Earth's magnetic field, as illustrated in Figure 2. By properly controlling the tether current during an orbit, the tether facility can reboost itself to its original orbit. The tether facility essentially serves as a large "orbital energy battery," allowing solar energy to be converted to orbital energy gradually over a long period of time and then rapidly transferred to the payload.

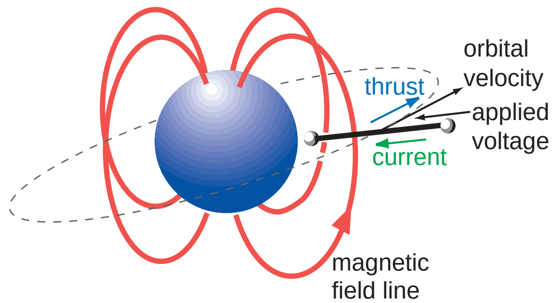


Figure 2. Electrodynamic tether thrust generation.

C. Key Advantages

A tether transportation system has several advantages compared to conventional and other advanced in-space propulsion systems:

1. (Near) Zero Propellant Usage

Chief among these advantages is the ability to eliminate the need for propellant expenditure to perform payload transfers. Of course, some propellant expenditure will be needed for trajectory corrections and rendezvous maneuvering, but with proper system design these requirements will be very small, a few tens of meters per second. The ability to cut several thousands of meters per second from the ΔV needed to deliver a payload to its destination can enable customers to utilize much smaller launch vehicles than would be required with a rocket-only system, greatly reducing total launch costs. For example, launching a 5 metric ton satellite into GEO, would require a Delta IVM+ (4,2) launch vehicle using an all-chemical propulsion system, at a cost exceeding $90M. Using a tether facility, the payload could instead be launched into LEO using a much smaller Dnepr 1 (RS-20) launch vehicle, at $1/7^{th}$ the cost of the Delta launch.

2. Short Transfer Times

A momentum-exchange tether system provides its ΔV to the payload in an essentially impulsive manner. Thus the transfer times in a tether system are very short, comparable to rocket-based systems. This can be compared with electric propulsion schemes, which offer low propellant usage, but invariably require long transfer times due to their low thrust levels. The short transfer times offered by a momentum-exchange tether system can play an important role in minimizing the lost-revenue time that a commercial satellite venture would have to accept while it waits for its satellite to reach its operational orbit and begin generating revenue if it were to use a low-thrust, high-Isp electric propulsion upper stage.

3. Reusable Infrastructure

Once deployed, a tether boost facility could transfer many, many payloads before requiring replacement. Thus the recurring costs for payload transport could be reduced to the cost of operations. A tether transportation system thus would be somewhat analogous to a terrestrial railroad or public-transit system, and might achieve comparable cost reductions for transporting many payloads.

4. Fully Testable System

Another important but often overlooked advantage of a tether transportation system is that the components that perform the actual payload transfer operations can be fully tested *in space operations* before being used for critical payloads. In conventional rocket systems, engine components and other key elements can be tested on the ground, and many individual units can be flown to provide reliability statistics, but to date only the Shuttle has re-used rocket engines, with significant maintenance after each flight. In a tether transportation system, the tether facility could be tested many times with “dummy” payloads – or, better yet, with low inherent-value payloads such as water or fuel – to build confidence for use on high value or manned payloads. In addition, "using" a tether does not damage or "wear it out", as long as the loads placed on the tether do not approach the yield point of the tether material. This means that the tether used in the operational system is the same tether in nearly the same condition in

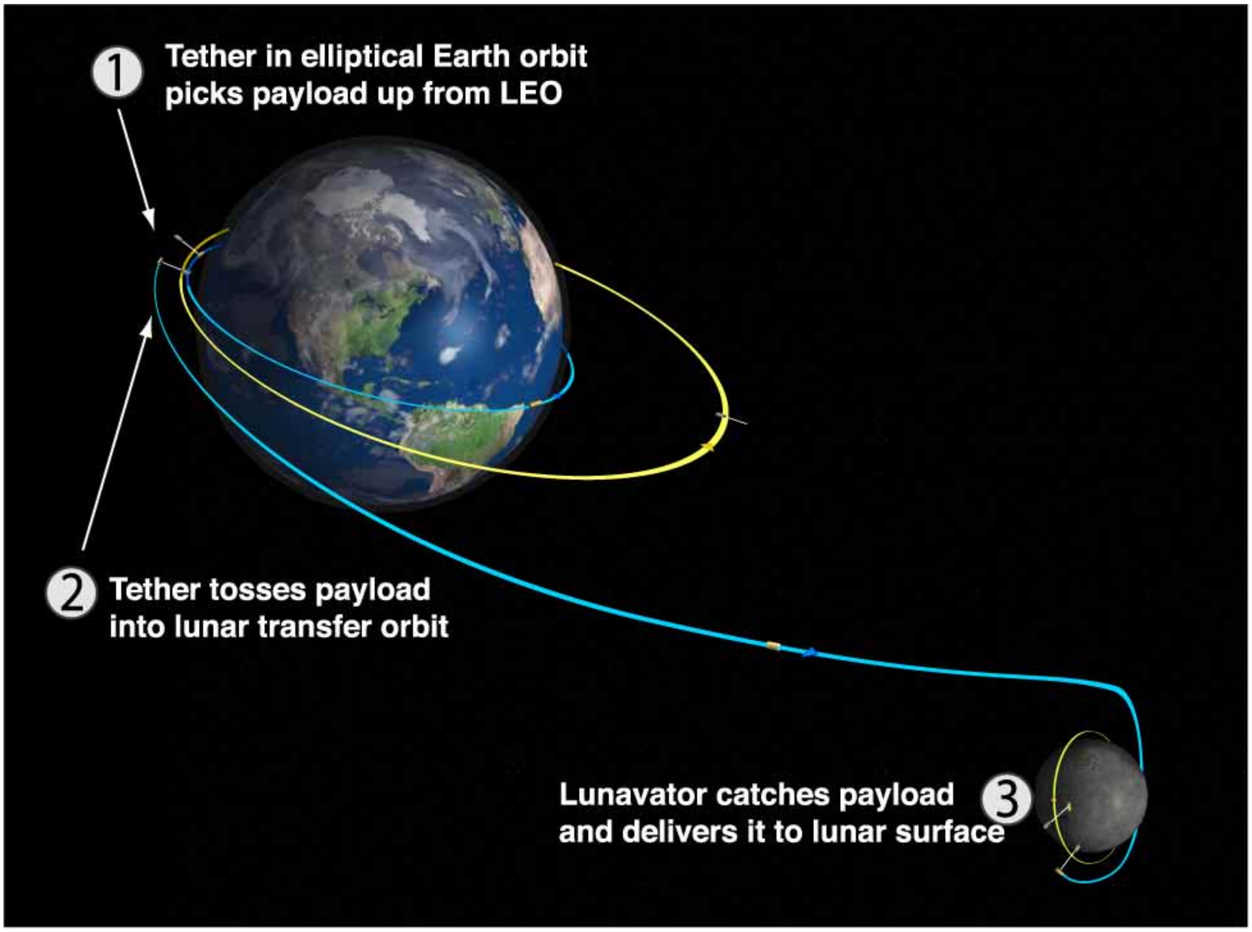


Figure 3. The Cislunar Tether Transport System concept.

which it underwent strength and reliability testing with the "dummy" payloads.

D. Summary of Phase I Results

In the Phase I effort, we investigated the feasibility of designing tether transport architectures for travel between LEO, the Moon, and Mars. We developed a design for a Cislunar Tether Transport System that uses one tether in elliptical, equatorial Earth orbit and one tether in low lunar orbit to provide round-trip travel between LEO and the surface of the Moon. This design, illustrated in Figure 3, includes considerations for finite tether facility masses, the complicated Earth-Moon orbital geometry, and the behavior of orbits in the non-ideal gravitational potentials of the Earth and Moon. We found that, using currently available tether materials, such a system would require a total mass of less than 28 times the mass of the payloads it can handle. Because a rocket-based system would require a propellant mass of at least 16 times the payload mass to perform the same job, the fully-reusable tether system would be competitive from a mass perspective after only two trips, and would provide large cost savings for frequent round-trip travel. Using numerical simulation tools with detailed models of orbital mechanics and tether dynamics, we verified the feasibility of using this tether system to transport payloads from LEO to the surface of the moon.

We also developed a design for a tether system capable of providing rapid round-trip transport between Earth and Mars. The "Mars-Earth Rapid Interplanetary Tether Transport (MERITT) System" would be composed of one rotating tether in highly elliptical Earth orbit and one tether in highly elliptical Mars orbit, and could provide short (3-5 month) transfer times in both directions while eliminating the need for transfer propellant.

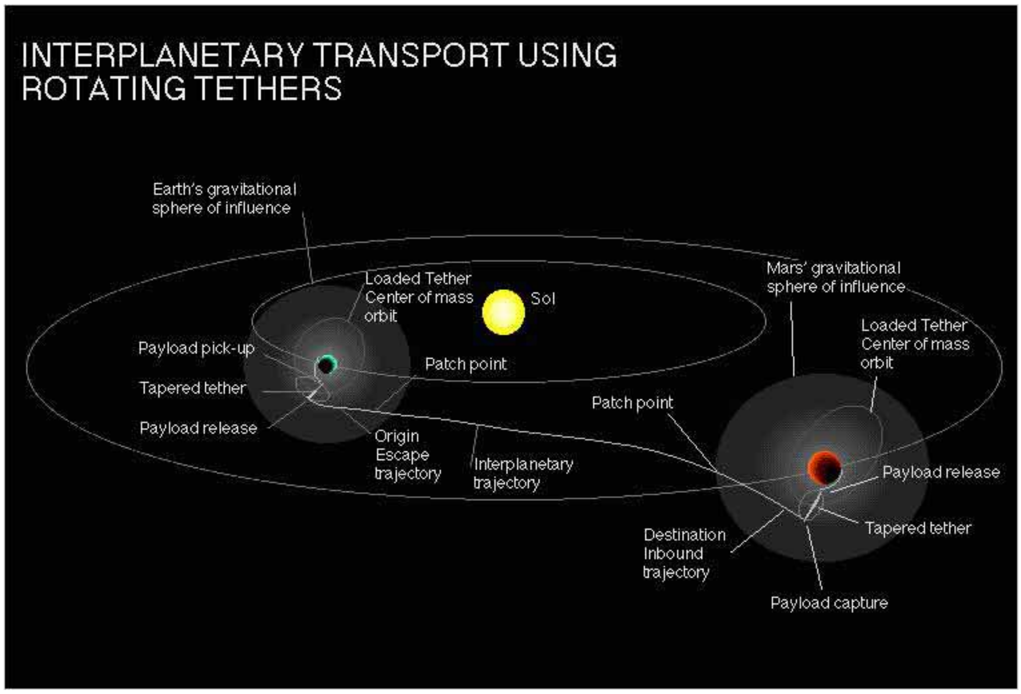


Figure 4. The Mars-Earth Rapid Interplanetary Tether Transport (MERITT) System Concept.

In addition, we developed a concept for using electrodynamic tether propulsion to restore the orbits of the Earth-orbit tether facilities after each payload boost operation. This design will enable the tether facility to repeatedly boost payloads without requiring propellant expenditure or return traffic. With this innovation, the Cislunar and MERITT systems can be developed incrementally, with the first component capable of boosting multiple payloads to GTO, lunar transfer orbits, and Mars injection so that it can earn revenue to fund the deployment of tether facilities at the Moon and Mars.

III. Phase II Technical Objectives

In the Phase II effort, we sought to build upon the conceptual transportation architectures designed in the Phase I effort by developing a more detailed understanding of the technology, systems, and methods required to implement these transportation systems based upon space tethers. The technical objectives of the Phase II effort were to:

- Design a Momentum-Exchange/Electrodynamic-Reboost Tether Boost Facility.
- Develop concepts and techniques for tether/payload rendezvous & capture.
- Combine and improve the designs of the Cislunar, MERITT, and LEO-GEO system architectures, and explore applications of rotating tethers to other NASA missions.
- Design an affordable first-step technology demonstration mission.

IV. Summary of Results of the Phase II Effort

During the course of the Phase II effort, we developed a long-term plan for designing and deploying a tether transportation system to service traffic between LEO, GEO, the Moon, and Mars, starting with near-term, low-cost technology demonstration experiments and progressing to operational systems. We then collaborated with the Boeing Company to develop a system-level design for an initial operational Tether Boost Facility that would boost commercial satellites from low-LEO to GTO, and could also boost scientific payloads to the Moon. As a part of this design effort, we evaluated the technology readiness level (TRL) of the required component technologies. Based upon this TRL evaluation, we identified several key technology needs, including systems and methods for payload-tether rendezvous, and systems and methods for electrodynamic reboost of the tether facility. We then developed concepts for satisfying these technology needs, and performed proof-of-concept demonstrations using numerical simulation. We also investigated the Earth-Mars tether transport systems further, evaluating their potential for minimizing transfer times. Finally, we developed a conceptual design for a small, low-cost momentum-exchange/electrodynamic-propulsion tether experiment that would demonstrate many of the needed technologies and techniques.

Organization of this Report

The results of each of the tasks pursued in this Phase II effort are detailed in separate documents included in this report as appendices. In the following paragraphs, we summarize the most important results of these tasks, and give references to the appropriate appendices. **Appendix A: "Tether Boost Facilities for In-Space Transportation"** contains a presentation that summarizes the results of the Phase II effort.

A. Roadmap for Development of a Earth-Moon-Mars Tether Transport System

Throughout the technical tasks pursued in the Phase II effort, we have sought to map out a plan for addressing the technology needs for a Tether Transport System and then deploying the system in a manner that can be pursued by a commercial venture. In the Phase II efforts, we have examined the technology readiness level (TRL) of the components and techniques needed for tether boost facilities. This TRL survey identified several technical challenges that must be met to enable tether transport systems to be fielded, including development of rapid automated rendezvous and capture (AR&C) capabilities, techniques for building and controlling the tether facilities, and power system technologies able to drive electrodynamic tethers at high power and voltage levels.

In order for a Tether Transport System to be built successfully, the system architecture must be designed so that its development and deployment is commensurate with a viable business plan. Because the development of a tether transport architecture for transporting frequent traffic between the Earth, the Moon, and Mars will require a significant total capital investment by government and private entities, we have sought to design a system architecture that can be propellantless reboost propulsion needed for MXER tether systems. The rendezvous and capture technologies needed for tether transportation systems can be demonstrated in separate, low-cost experiments, beginning with the "High Altitude Tether – Grapple Rendezvous and Secure Pickup" (HAT-GRASP) experiment. The HAT-GRASP experiment would demonstrate rendezvous and capture between a tether hanging from a high altitude balloon and a small payload launched into a ballistic trajectory by a suborbital rocket. Because the payload would be "coasting" in a 1 g gravity field, the rendezvous situation in this experiment would closely match the rendezous scenario in an orbital tether system, but can be done at a relatively low cost because it does not require a launch into orbit. This technology demonstration would feed into the "Microsatellite Tethered Orbit-Raising QUalification Experiment" (μTORQUE), which would be designed to fly as a secondary payload on a GEO satellite launch, and would use electrodynamic drag propulsion to spin up a tether and toss a small satellite to a lunar transfer.

Combining these technologies for electrodynamic propulsion and tethered rendezvous and capture would then enable the deployment of a Tether Boost Facility designed to boost commecial satellites to GTO and toss scientific payloads to lunar transfer orbits. This facility would be constructed with a modular design, so that its capabilities could be increased to enable it to serve as a space-based "second stage" for Earth-toOrbit launch, and to serve as a transportation hub for Earth-Moon-Mars traffic.

Further details on the incremental development roadmap for building a Tether Transport System are given in **Appendix B, "Commercial Development of a Tether Transport System."**

B. Design and Simulation of a Tether Boost Facility for LEO⇒GTO Payload Transport

The primary focus of the Phase II project was a collaborative effort between Tethers Unlimited, Inc. and The Boeing Company to develop a design for the first component of a tether transport architecture, a LEO⇒GTO Tether Boost Facility. This facility will combine momentum-exchange tether techniques with electrodynamic tether propulsion to provide a reusable infrastructure capable of repeatedly boosting payloads from low Earth orbit to geostationary transfer orbit without requiring propellant expenditure.

The design effort began by evaluating potential objectives and missions for this system concept, and developed a Systems Requirement Document to guide the rest of the design study. The **Systems Requirement Document** for the LEO⇒GTO Tether Boost Facility is presented in **Appendix F**.

The system design has progressed through several iterations, beginning with a facility sized to handle 5-ton payloads, and then moving to a facility sized to handle 2,500 kg payloads initially but designed modularly so that its capacity can be increased in an incremental fashion. The preliminary design is summarized in **Appendix E: "Tether Boost Facility Design Study Interim Report"**, and the final design is discussed in detail in **Appendix F: "Tether Boost Facility Design Study Final Report"**.

deployed in a modular, incremental fashion, in which each component can generate revenue to fund the development of the rest of the system, much as the first railroads were developed. Because the largest current commercial market for in-space transportation is the delivery of communications satellites to GEO, the initial Tether Boost Facility will be designed primarily to service traffic of satellites from LEO to GTO. This LEO⇒GTO Tether Boost Facility, however, will also be capable of transporting different payloads to other destinations, including Lunar Transfer Orbit (LTO). Once the initial facility has been deployed and proven in operation, the system capacity could then be built up incrementally by adding more modules. Then, additional tether facilities deployed to handle Earth-to-Orbit Assist, LEO⇔Lunar Surface round-trip travel, and deployment of manned Mars bases.

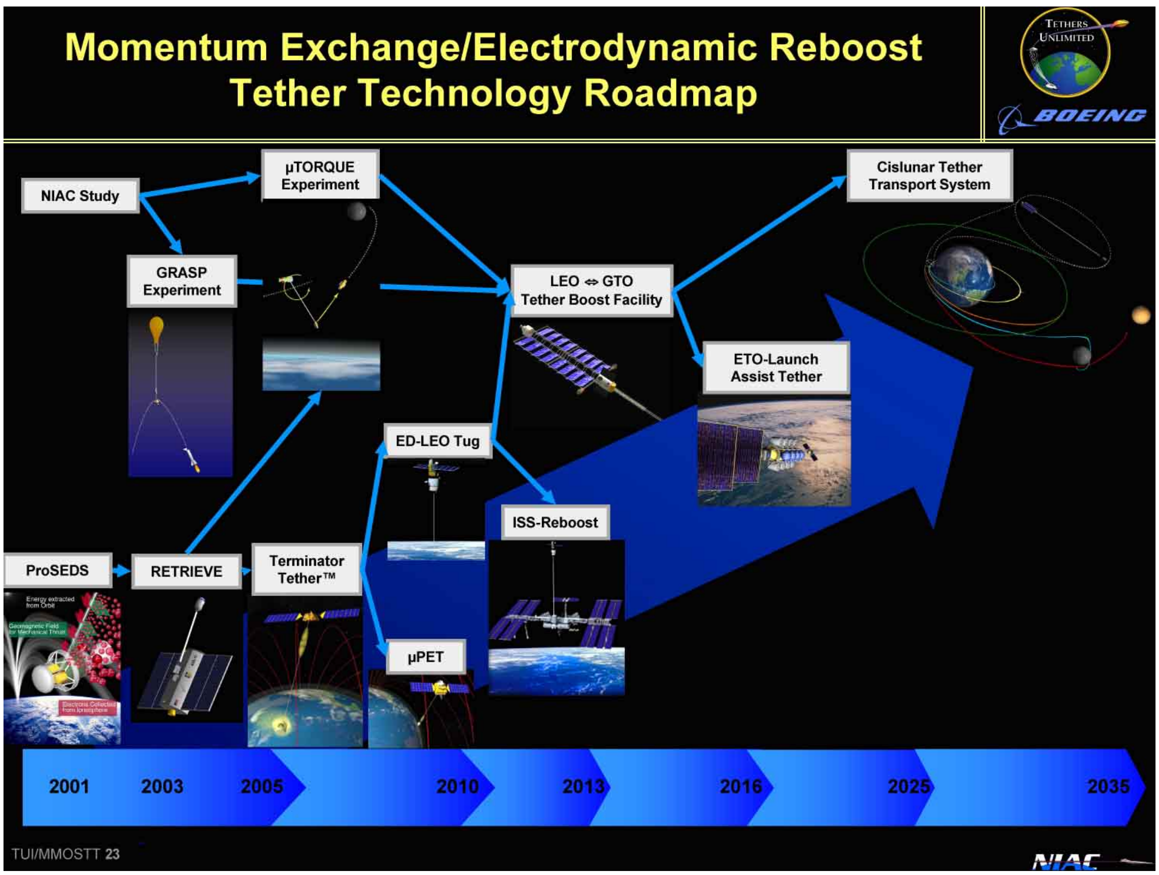


Figure 5. MXER Tether Technology Development Roadmap.

Figure 5 illustrates the proposed incremental development path for the Tether Transport System. The development will begin with several low-cost technology development and demonstration experiments. The first experiment is the ProSEDS mission, a NASA/MSFC experiment currently scheduled to fly in mid-2002 to demonstrate electrodynamic drag propulsion using a bare-wire tether. The proposed RETRIEVE experiment will demonstrate a very small (~3.5 kg) electrodynamic tether system that uses current feedback to control the tether dynamics while it deorbits a microsatellite. These two experiments will develop the electrodynamic propulsion technologies needed to first bring to the commercial market small operational electrodynamic tether systems for spacecraft propulsion and deorbit, and then to field larger tether propulsion systems such as for satellite orbital transfer and reboost of the International Space Station. These electrodynamic tether technologies will also provide the

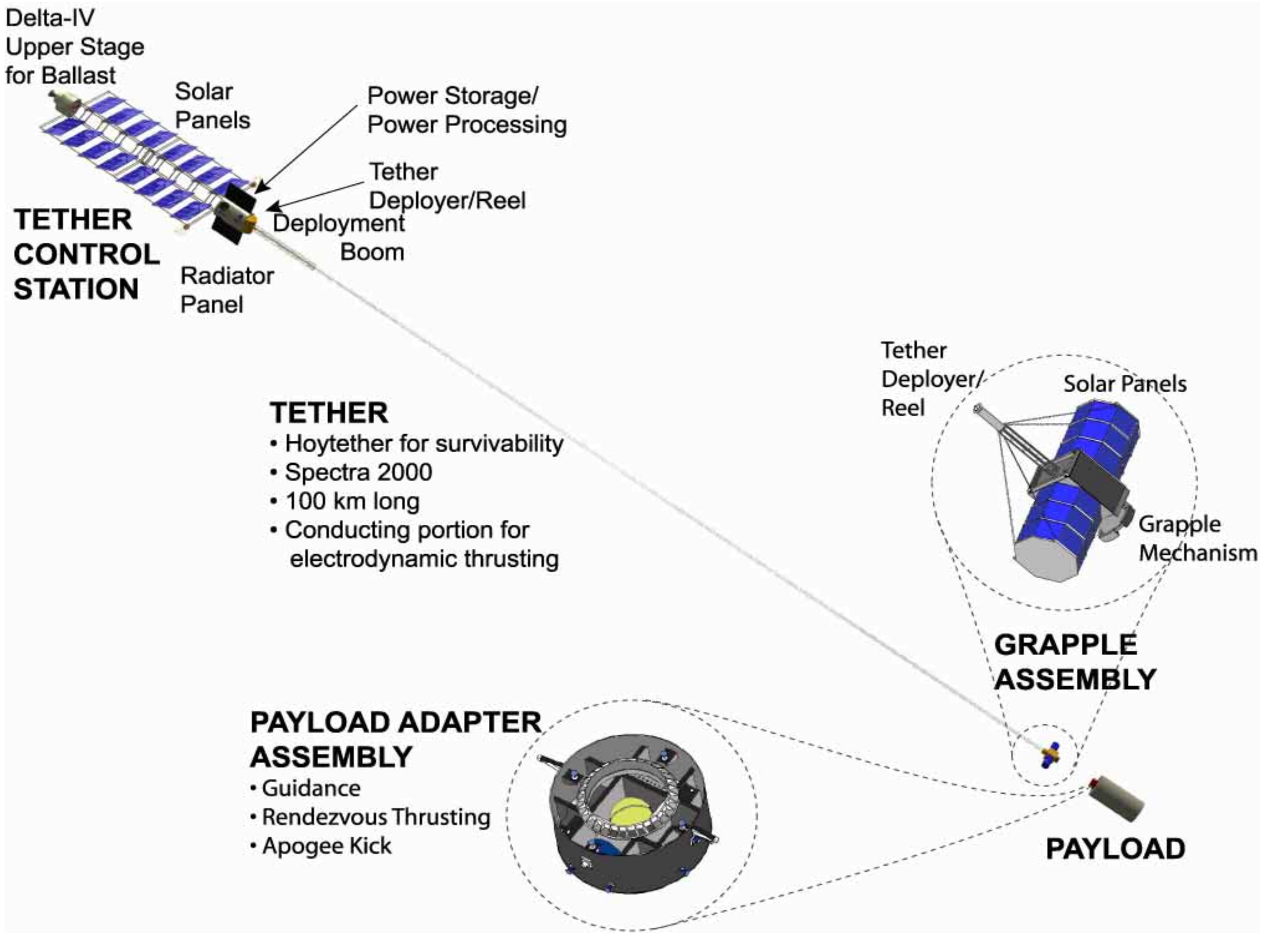


Figure 6. The Tether Boost Facility Design concept.

1. 5,000 kg Payload Tether Boost Facility

The general trend for GEO communications satellites has been for the satellites to become larger and larger with time. Our market projections indicate that a potential "sweet spot" for system payload to GTO would be around 5,000 kg. A tether system capable of delivering this sized payload to GTO could serve approximately 80% of the market projected in 2010. Consequently, the initial design effort focused on designing a tether system capable of boosting 5,000 kg payloads from 300 km circular LEO to GTO. Because the tether system is a highly reusable infrastructure, one key to achieving minimum transportation costs will be to maximize its throughput capacity. By estimating the potential market in 2010, we concluded that a reasonable throughput for which to aim would be one payload per month. Thus the system was designed to reboost its orbit within 30 days after each payload boost operation. The initial system design for this LEO⇒GTO Tether Boost Facility is described in **Appendix E: "Tether Boost Facility Design Study Interim Report"**.

The potential launch cost savings that a Tether Boost Facility could provide to a customer can be illustrated by considering a mission to place a 5,000 kg payload in GTO. To do so using conventional rocket systems would require a launch vehicle comparable to a Delta IVM+ (4,2), costing upwards of $90M. With a Tether Boost Facility capable of picking the payload up from LEO and tossing it to GTO, the customer could instead use a much smaller launch vehicle such as a Dnepr 1, with a launch cost of approximately $13M. Even when the operational costs of the

Tether Boost Facility are added to this figure, this quick comparison indicates that a tether transport system could reduce the launch costs to the customer by 50%-80%.

One of the results of the initial tether boost facility design was that a tether facility capable of tossing 5,000 kg to GTO would require a total on-orbit mass of approximately 50 metric tons. Currently, a launch vehicle capable of placing 50 metric tons in orbit does not exist. Should the needs of NASA's HEDS program or other government-led initiatives result in the development of the proposed "Magnum" launch rocket, it may become possible to deploy such a tether facility in one launch. In the absence of such a beefy rocket, however, a tether boost facility will either have to be sized for a smaller vehicle, and thus sized for a smaller payload, or will require multiple launches and on-orbit assembly.

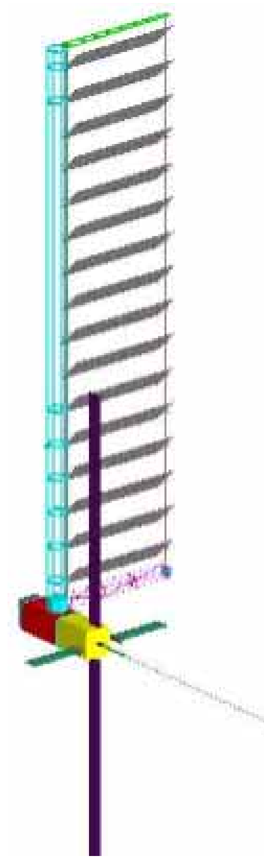
In order for the development of a Tether Boost Facility to be affordable for a commercial venture, it will be vital for the facility to be capable of performing a useful, revenue-generating service after the first launch. Consequently, our design effort evolved the system concept into one that would be capable of being launched on a single large launch vehicle expected to be in service in 2010. This facility would initially have a smaller payload capacity, but would be designed in a modular fashion so that its capacity can be increased to service 5,000 kg and larger payloads:

2. <u>2,500 kg Payload Modular Tether Boost Facility</u>

Appendix C, "Design and Simulation of a Tether Boost Facility for LEO⇒GTO Payload Transport" presents the concept design for a modular Tether Boost Facility capable of boosting 2,500 kg payloads from LEO to GTO. Using analytical methods, we developed designs for the orbital mechanics and system sizing of the tether facility. The orbital designs were chosen so that the payload and tether orbits are synchronous, so that the tether will have multiple opportunities to capture a payload with minimal maneuvering requirements. These designs account for orbital perturbations due to Earth oblateness. The tether facility power system is sized to enable a throughput of one payload every 30 days. The entire tether facility is sized to enable an operational capability to be deployed with a single Delta-IV-H launch. The system is designed in a modular fashion so that its capacity can be increased with additional launches. The tether facility can also boost 1000 kg payloads to lunar transfer orbits, and will serve as the first building block of an Earth-Moon-Mars Tether Transportation Architecture.

Appendix F. : "Tether Boost Facility Design Study Final Report" presents the details of the subcontract effort by Boeing to define a system-level design for this initial operational tether facility, and Table 1 presents a summary of the design. The tether facility is sized at 19,891 kg to be launched into LEO on a single Delta IV-H rocket, and will retain the Delta IV's upper stage rocket as ballast mass, giving it a total operational mass of 23,358 kg.

Table 1. 2,500 kg to GTO Tether Boost Facility Design Summary:



- Control Station mass = 13,267 kg (includes 21% mass margin)
- Operational mass = 23,358 kg, no margin Control Station w/Payload Adapter Fixture
- GLOW = 19,891 kg with 15% margin, no PAF

Power System:

- *Scarlet*-like concentrator PV arrays, 563 square meters
- Standard, state-of-the-art PV array drive motors
- State-of-the-art power management and distribution except for electrodynamic tether subsystem
- Lithium-ion battery power storage system
- 5,410 kg (includes 14% mass growth margin)

Communication Subsystem

- Downlink communication with ground station(s) and communication with Grapple Assembly and PAA (via Tether Facility Network)
- State-of-the-art, COTS hardware (antennae/transceivers)
- Dual redundancy
- 4.2 kg (includes 16% mass growth margin)

C&DH

- State-of-the-art, COTS hardware
- Dual redundancy
- 29 kg (includes 13% mass growth margin)

ADCS/GN&C

- 2 Control Moment Gyros (no redundancy), each assumed half size of a Skylab CMG
- 2 sun sensors
- 2 inertial navigation unit
- GPS antennae (3)/tranceivers (2)
- 213.8 kg (includes 6% mass margin)

Electrodynamic Tether Subsystem

- Sized for 80 km conductive tether, total length 100 km, 300,000 W, 40 μN/W thrust efficiency
- Control Subsystem with 1m diameter, 1.5m long reel, motor, tether guides, power conversion, FEACs (field emitter array cathodes)
- 1,933 kg (includes 36% mass margin)

C. Tether Facility Reboost

A key factor in the economic competitiveness of a Tether Boost Facility will be the frequency with which the facility can boost payloads. The throughput capacity of a tether facility will be determined largely by the time required to restore the facility's orbit after each payload boost operation. In this work we have investigated using electrodynamic tether propulsion to reboost the orbit of the tether without requiring propellant consumption.

As with the system design, the investigation of electrodynamic reboost has gone through several iterations. In **Appendix K, Tether Reboost Study**, we present results of our initial analytical and numerical investigations of the time required to reboost the orbit of the 5,000 kg payload facility using electrodynamic tether propulsion. We used these results to guide the design of the tether facility described in the previous section. The results of the tether system design were then fed back into the simulation effort. In the latter part of **Appendix C, "Design and Simulation of a Tether Boost Facility for LEO⇒GTO Payload Transport"** we present more recent simulation results that use more advanced methods for optimizing the orbital reboost. These latest results indicate that a Tether Boost Facility sized for boosting 2,500 kg payloads to GTO once per month will require a solar power system of approximately 100 kW.

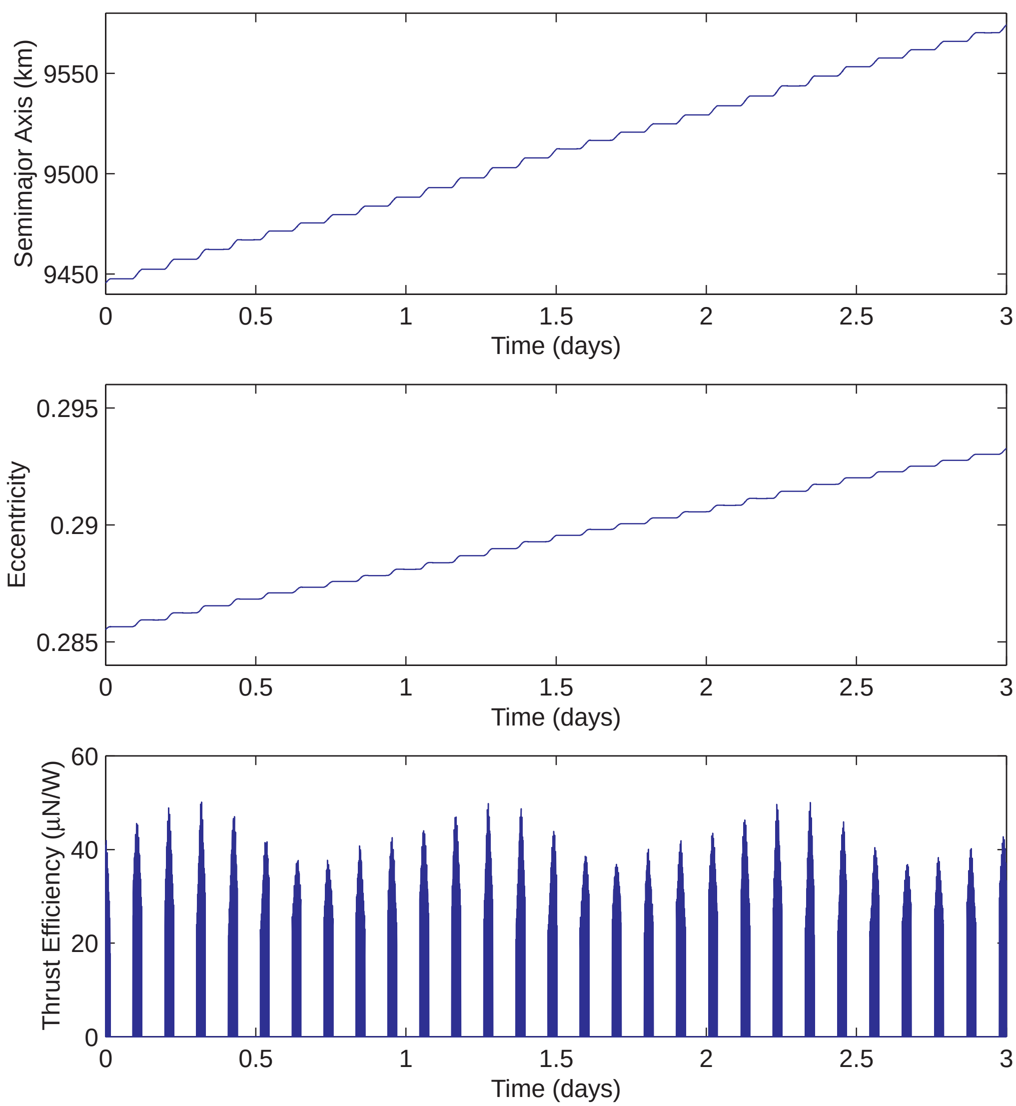


Figure 7. Semimajor axis, eccentricity, and thrust efficiency during the first three days of the reboost operation.

D. Development and Simulation of Tether Rendezvous Methods

Of the technology needs identified in the TRL evaluation, the most significant challenge is to enable payloads to successfully and reliably rendezvous with the grapple on a rotating tether. To begin addressing this challenge, we used a numerical simulation that includes models for orbital mechanics and tether dynamics to study the rendezvous between a payload in orbit and a rotating tether facility. In a tether-payload rendezvous, the relative motion between the tether tip and payload is primarily along the local vertical direction. The relative acceleration is constant, so, from the perspective of the payload, the tether tip descends to the payload, halts instantaneously, then accelerates away. We have developed a method for using tether deployment to increase the length of time that the payload and grapple are near each other. This method is illustrated in Figure 8. As shown in Figure 9, numerical simulations indicated that this tether deployment maneuver can extend the "instantaneous" rendezvous to a window of tens of seconds, without need for propellant usage. We also studied the effects of the payload capture on the tether tension. The simulations indicated that for an ideal rendezvous, tension wave behavior will cause tension excursions roughly double that of the steady-state loads. If the rendezvous is not ideal, that is, if the tether must be deployed for several seconds while the payload and tether tip vehicle maneuver to achieve a docking, the resultant tension spikes can further increase the peak tether loads. Additional tether deployment maneuvers can help to ameliorate the peak tension excursions and damp the longitudinal oscillations.

A more detailed discussion of the study of Tether Rendezvous Methods is presented in **Appendix G**.

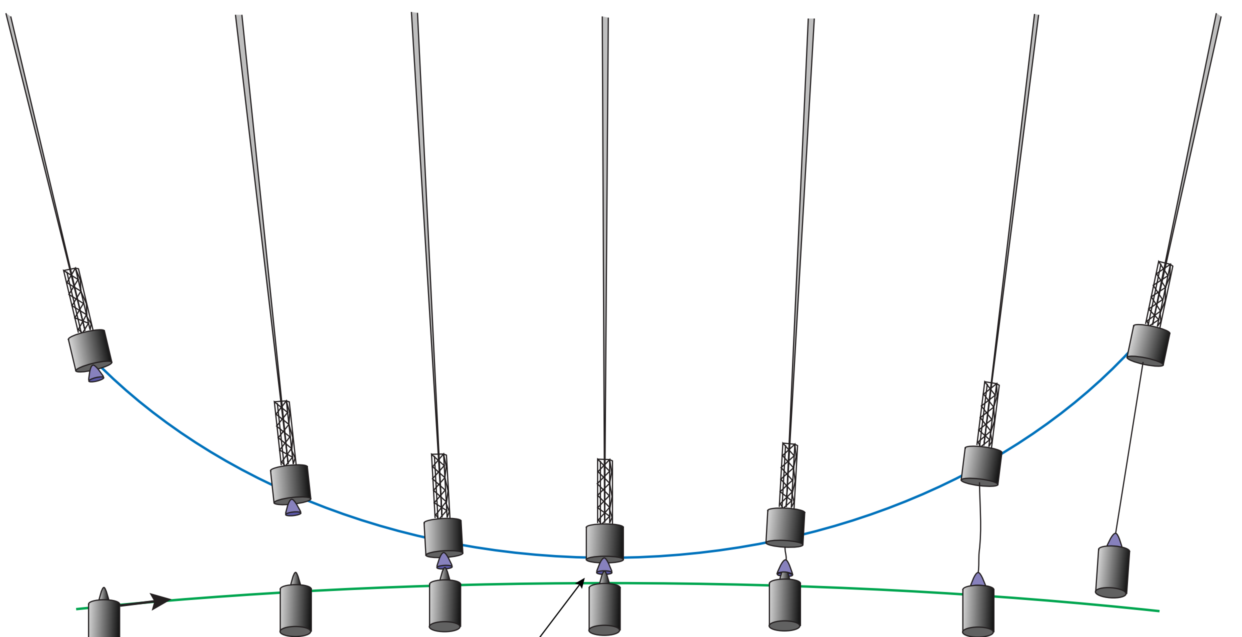


Figure 8. Schematic of rendezvous method where the Payload Capture Vehicle drops a tethered grapple into free fall.

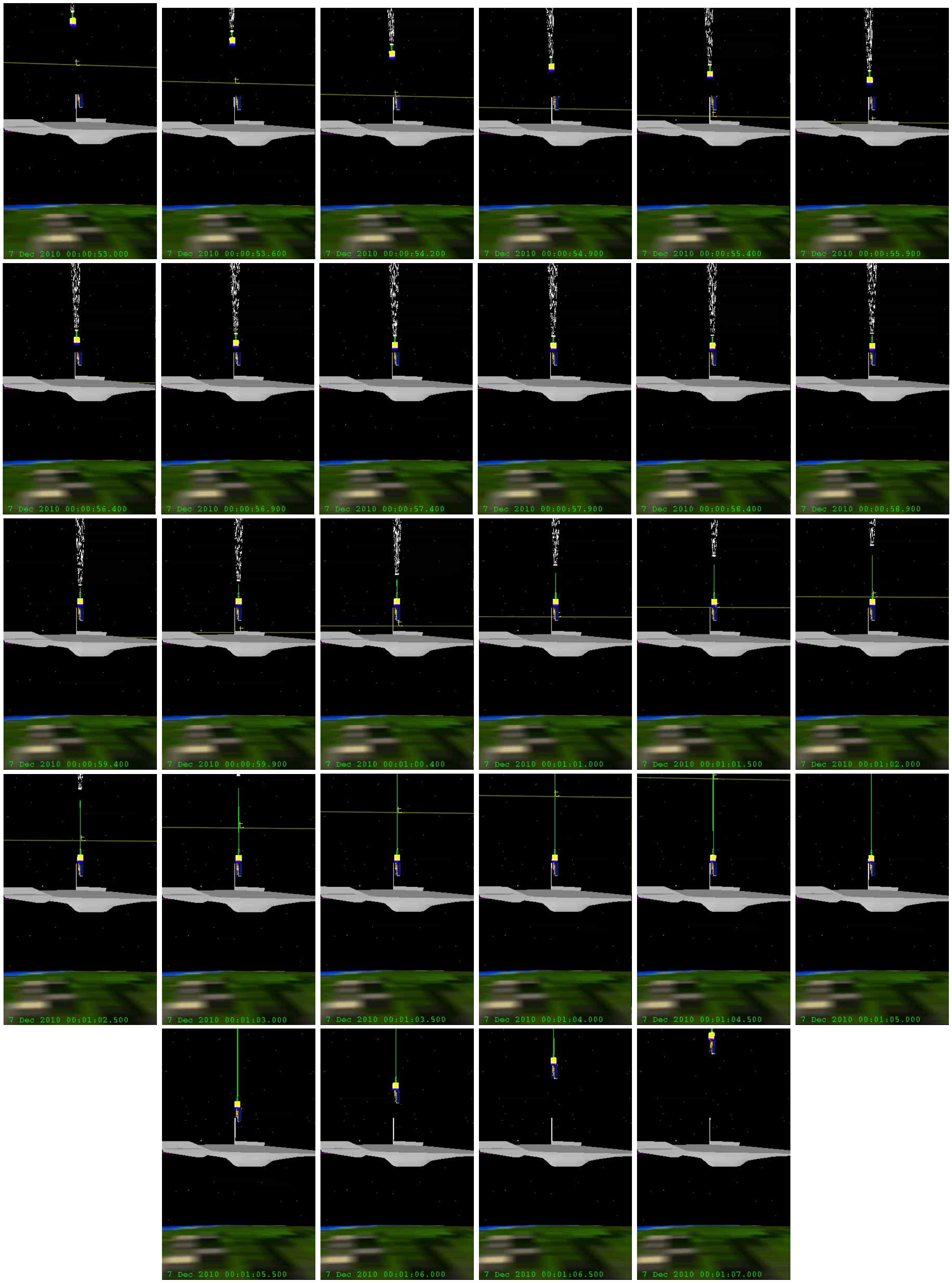


Figure 9. Simulation of a rendezvous between a reusable launch vehicle and the grapple end of a tether boost facility, using the tether deployment method of extending the rendezvous time. The simulation is shown at half-second intervals**.** (pictures courtesy Boeing)

E. Tether Systems for Interplanetary Transport

Momentum-exchange tether systems may also enable rapid propellantless transport of payloads between Earth, Mars, and other planets. As a part of this Phase II effort, we investigated concepts for using rotating tethers in elliptical orbits around the Earth and Mars to provide a means of tossing payloads between the planets. One concept for such a system is the "Mars-Earth Rapid Interplanetary Tether Transport (MERITT)" System, discussed in **Appendix P, "Interplanetary Tether Transport Overview"** and **Appendix R, "Mars-Earth Rapid Interplanetary Tether Transport (MERITT) Architecture."** In the MERITT architecture, a tether facility in a highly elliptical orbit around one planet would pick up payloads when it is near periapsis and, when it returns to periapsis, toss them at a velocity sufficient to give the payload a substantial hyperbolic excess velocity. At the destination planet, a second tether system would catch the payloads and release them a short time later into a low orbit or a suborbital trajectory, as illustrated in Figure 10.

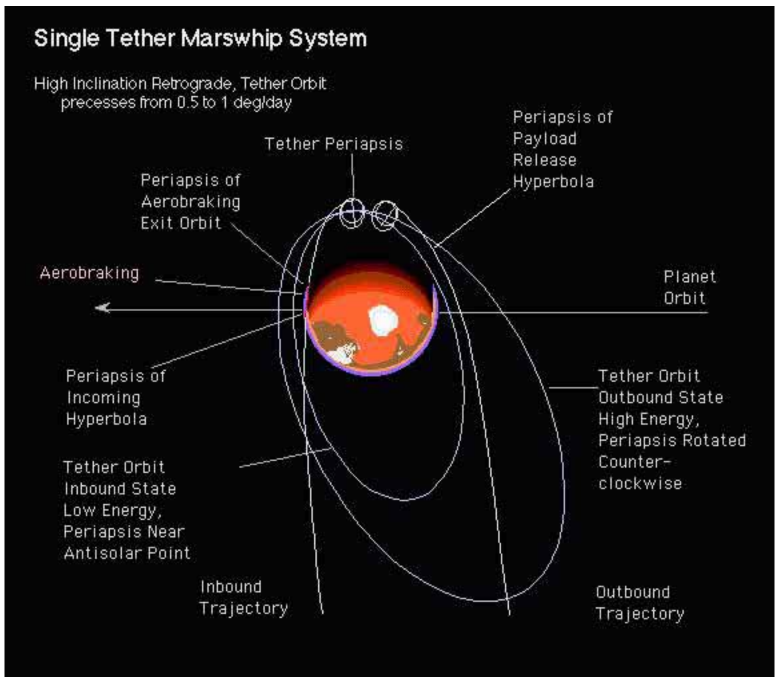


Figure 10. Design of the orbital architecture of the Mars tether facility in the MERITT system.

The system works in both directions and is reusable. Kinetic energy lost by the throwing tethers can be restored either by catching incoming payloads, by propellantless tether propulsion methods, and/or high specific impulse propulsion systems. We investigated launch window lengths and transfer times that a MERITT system could achieve. Figure 11 summarizes the results of the launch window analysis. As shown in Figure 12, tethers with tip velocities of 3 km per second can send payloads to Mars in as little as 70 days if aerobraking is used at Mars to dissipate excess relative velocity and the orbital phasing is favorable. Tether-to-tether transfers without aerobraking may be accomplished in about 110 to 160 days.

We also investigated a concept for using momentum exchange tethers to enable missions to two outer planets to be accomplished by a pair of spacecraft launched by a single rocket. In this concept, detailed in **Appendix Q: "Application of Synergistic Multipayload Assistance with Rotating Tethers (SMART) Concept to Outer Planet Exploration**," a tether would be deployed between two spacecraft, and the system would then be spun up as it approaches the first target planet. When the tethered spacecraft reaches periapse, the tether would release the spacecraft, leaving one payload in orbit around the planet and tossing the other satellite towards its destination planet. This method would provide a significant enhancement to the "gravitational slingshot" delta-V boost that the second spacecraft could obtain from its flyby of the first planet.

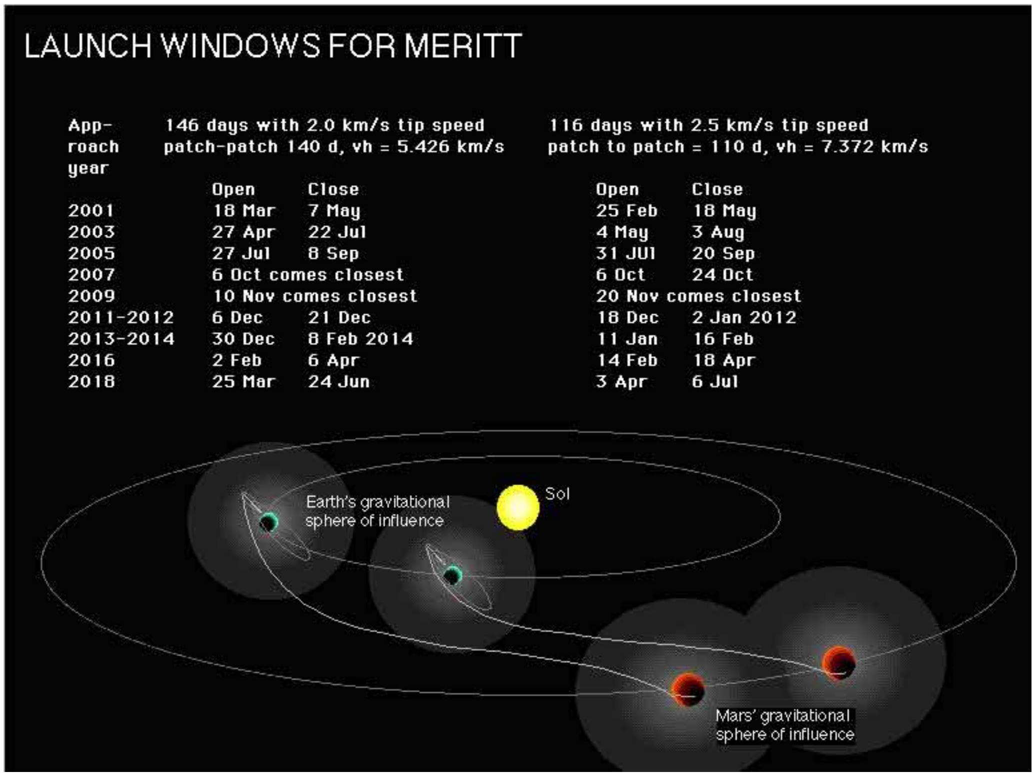


Figure 11. Launch window analysis for the MERITT Architecture.

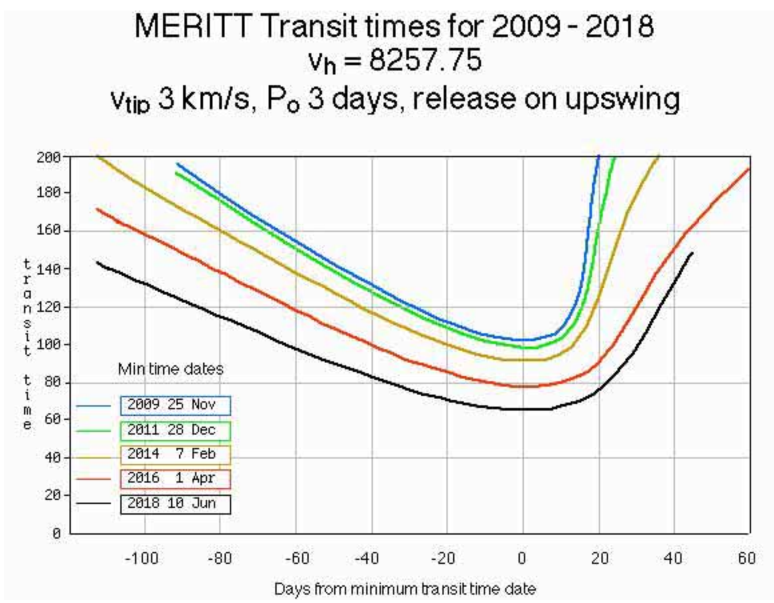


Figure 12. Transit times for the MERITT System.

F. μTORQUE:Low-Cost Momentum-Exchange/Electrodynamic-Propulsion Demonstration

In order to begin addressing the key technical challenges in MXER tether systems, we have developed a concept design for a very small momentum-exchange/electrodynamic-propulsion tether system capable of boosting a microsatellite by a ΔV of 0.4 km/s. This "Microsatellite Tethered Orbit Raising QUalification Experiment" (μTORQUE) system is sized to fly, along with its microsatellite payload, as a secondary payload on an upper stage rocket such as the SeaLaunch Block DM 3rd Stage.

The μTORQUE concept is illustrated in Figure 13. The μTORQUE tether system and a microsatellite payload would be integrated onto a rocket upper stage prior to launch. After the stage releases its primary payload into GTO (1), the μTORQUE system would deploy the microsatellite from the stage at the end of a high-strength conducting tether (2). The system would then use electrodynamic-drag thrusting during several successive perigee passes (3), to spin-up the tether system. This would effectively convert some of the upper stage's orbital energy into system rotational energy. Because the system utilizes electrodynamic drag to perform the spin-up of the system, it will not require the mass and complexity of a dedicated solar power supply; the system can also power its own avionics utilizing the power generated by the tether. When the tether tip velocity reaches 0.4 km/s, the μTORQUE system could then release the payload during a perigee pass (4), injecting the payload into a minimum-energy lunar transfer trajectory (5). With a 0.4 km/s ΔV capability, the μTORQUE tether system could also be useful for missions such as deploying microsatellites into high-LEO and MEO orbits as secondary payloads on launches of larger satellites into LEO.

A μTORQUE experiment sized to fly as a secondary payload with a 100 kg total mass allocation would mass approximately 20 kg, and could boost a payload massing 80 kg from GTO to a lunar transfer.

The μTORQUE experiment concept is discussed in more detail in **Appendix L**, and an operational tether facility concept designed for boosting microsatellites from LEO to GTO and lunar transfers is discussed in **Appendices N** and **O.**

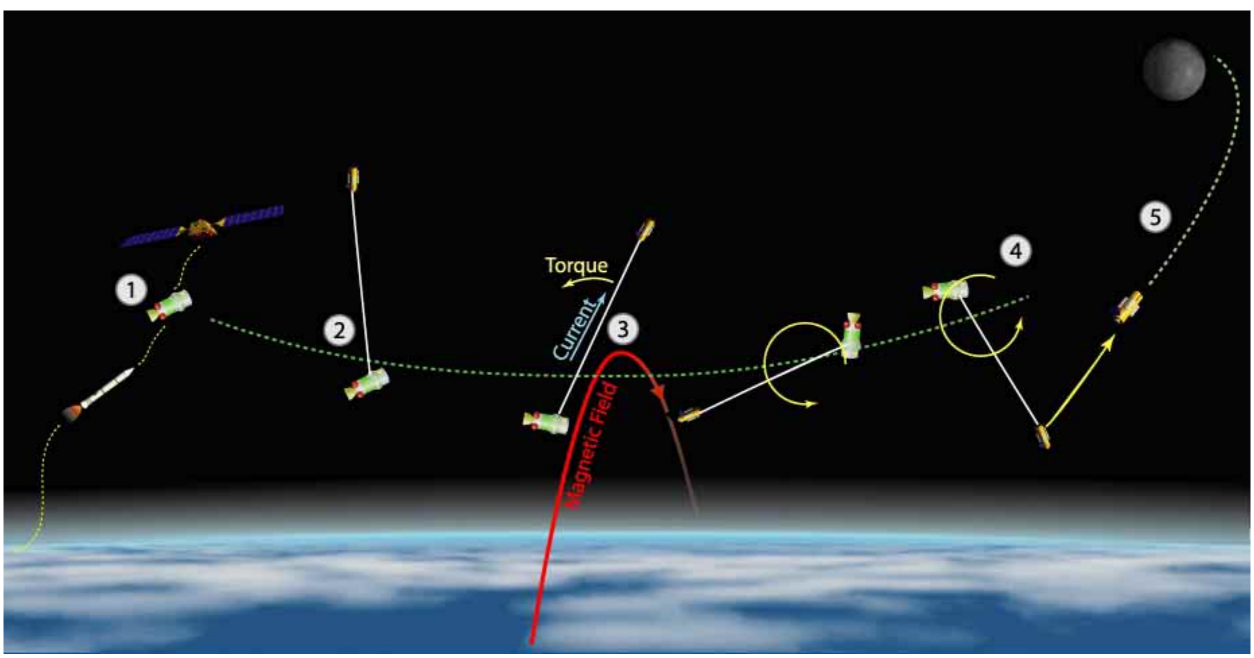


Figure 13. **The "Microsatellite Tethered Orbit-Raising Qualification Experiment (μTORQUE)" concept.**

V. Publications

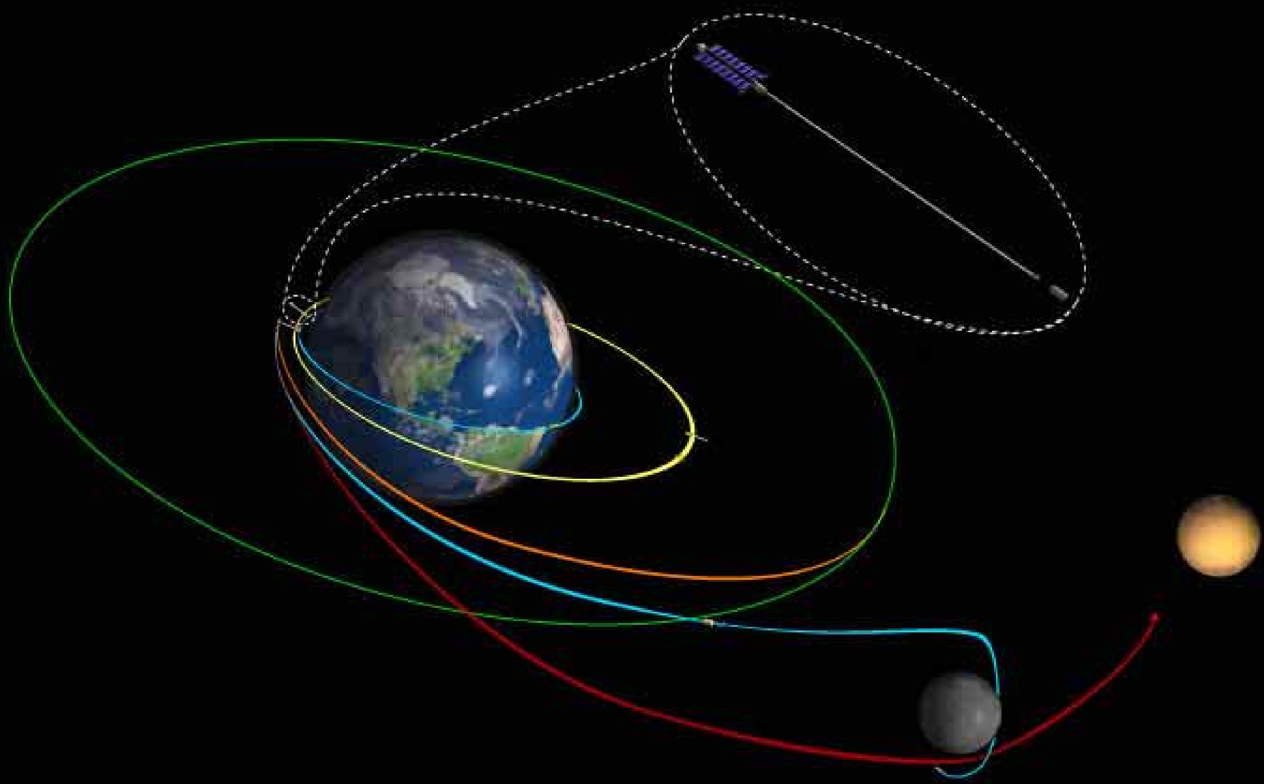
The Phase I and II efforts resulted in a total of 11 publications and technical conference papers. These publications are listed below:

1. Hoyt, R.P., Uphoff, C.W., "Cislunar Tether Transport System," *Journal of Spacecraft*, Vol. 37, No. 2, pp. 177-186, March-April 2000.
2. Hoyt, R.P., Uphoff, C.W., "Cislunar Tether Transport System", AIAA Paper 99-2690, *35th Joint Propulsion Conference*, June 1999.
3. Nordley, G.D., and R.L. Forward, "Mars-Earth Rapid Interplanetary Tether Transport System: I Initial Feasibility Analysis", *Journal of Propulsion and Power* (17) 3 May-June 2001, pp. 499-507.
4. Hoyt, R.P., Forward, R.L., Nordley, G.D., Uphoff, C.W., "Rapid Interplanetary Tether Transport", IAF Paper 99-A.5.10 *50th International Astronautical Congress*, Oct 1999.
5. Hoyt, R.P., "Design and Simulation of a Tether Boost Facility for LEO to GTO Transport," AIAA Paper 2000-3866, 36th Joint Propulsion Conference, Huntsville, AL, 17-19 July 2000.
6. Hoyt, R.P., "Commercial Development of a Tether Transport System," AIAA Paper 2000-3842, 36th Joint Propulsion Conference, Huntsville, AL, 17-19 July 2000.
7. Hoyt, R.P., "Tether Systems for Satellite Deployment and Disposal", IAF Paper 00-S.6.04, *51st International Astronautical Congress*, 2-6 Oct 2000, Rio de Janiero, Brazil.
8. Hoyt, R.P., "The Cislunar Tether Transport System Architecture", Paper presented at the 2nd *Lunar Development Conference*, Las Vegas, NV, July 20, 2000.
9. Hoyt, R.P., "Momentum-Exchange/Electrodynamic-Reboost Tether Facility for Deployment of Microsatellites to GEO and the Moon", Paper presented at the 2001 Space Technologies and Applications Forum, Albuquerque, NM.
10. Hoyt, R.P., "The μTORQUE Momentum-Exchange Tether Experiment," paper submitted to the 2002 Space Technologies and Applications Forum, Albuquerque, NM.
11. Nordley, G.D., and R.L. Forward, "Interplanetary Tether Transport Overview," Special Presentation at the *AAS/AIAA Space Flight Mechanics Meeting*, Santa Barbara, CA, 11-15 February 2001.

VI. CONCLUSIONS

The Phase I and II NIAC-funded efforts evaluated the feasibility of using rotating space tethers to serve as the backbone of a reusable in-space transportation infrastructure. We began by developing concept designs for tether systems for LEO-to-GTO, LEO⇔Lunar, and Earth⇔Mars transport, and used numerical and analytical tools to demonstrate that these systems can be designed to account for the complex orbital dynamics in the Earth-Moon system. We then developed a realistic system-level design of a tether boost facility, based upon present-day and near-term technologies, and evaluated the components and technologies required for this system in terms of technology readiness. The two most important key technology needs identified by this study were the rendezvous and capture to enable a tether to reliably pick up a payload, and the high-power electrodynamic tether propulsion systems needed to provide propellantless reboost of the tether facility in between payload transport operations. We investigated these two technology needs further, developing concept designs for methods to make these challenges solvable, and demonstrated their feasibility using detailed numerical simulations. To continue developing the technologies needed for Momentum-Exchange/Electrodynamic-Reboost tether systems in an affordable manner, we developed a concept design for a small initial tether boost demonstration experiment that could fly as a secondary payload on a GEO satellite launch, and could boost a microsatellite to a lunar transfer. This experiment will serve as the first step in an incremental technology development plan, in which an Earth-Moon-Mars tether transportation system could be deployed in stages, and each stage could perform useful transportation missions to generate revenue to fund the development and deployment of the rest of the system.

Tether Boost Facilities for In-Space Transportation



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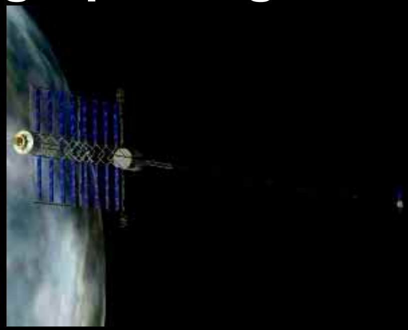
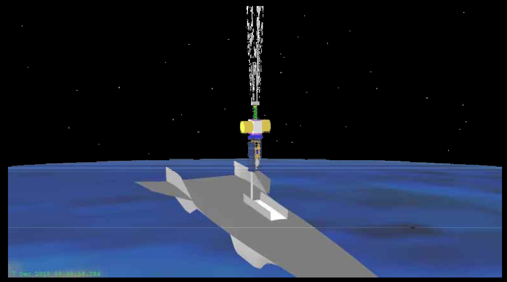
NIAC Funded Tether Research

- **Moon & Mars Orbiting Spinning Tether Transport (MMOSTT)**



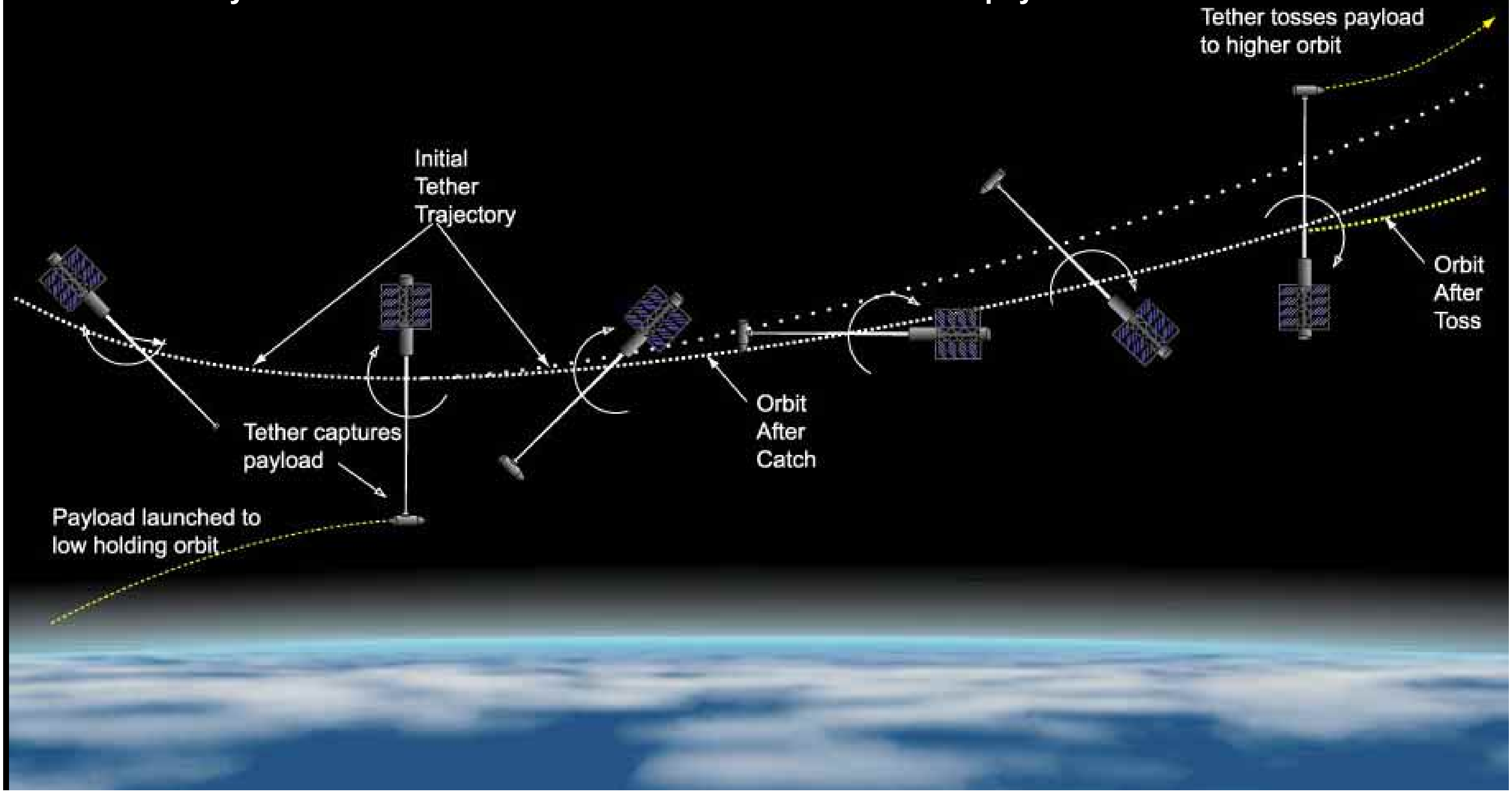
- **Hypersonic Airplane Space Tether Orbital Launch (HASTOL)**



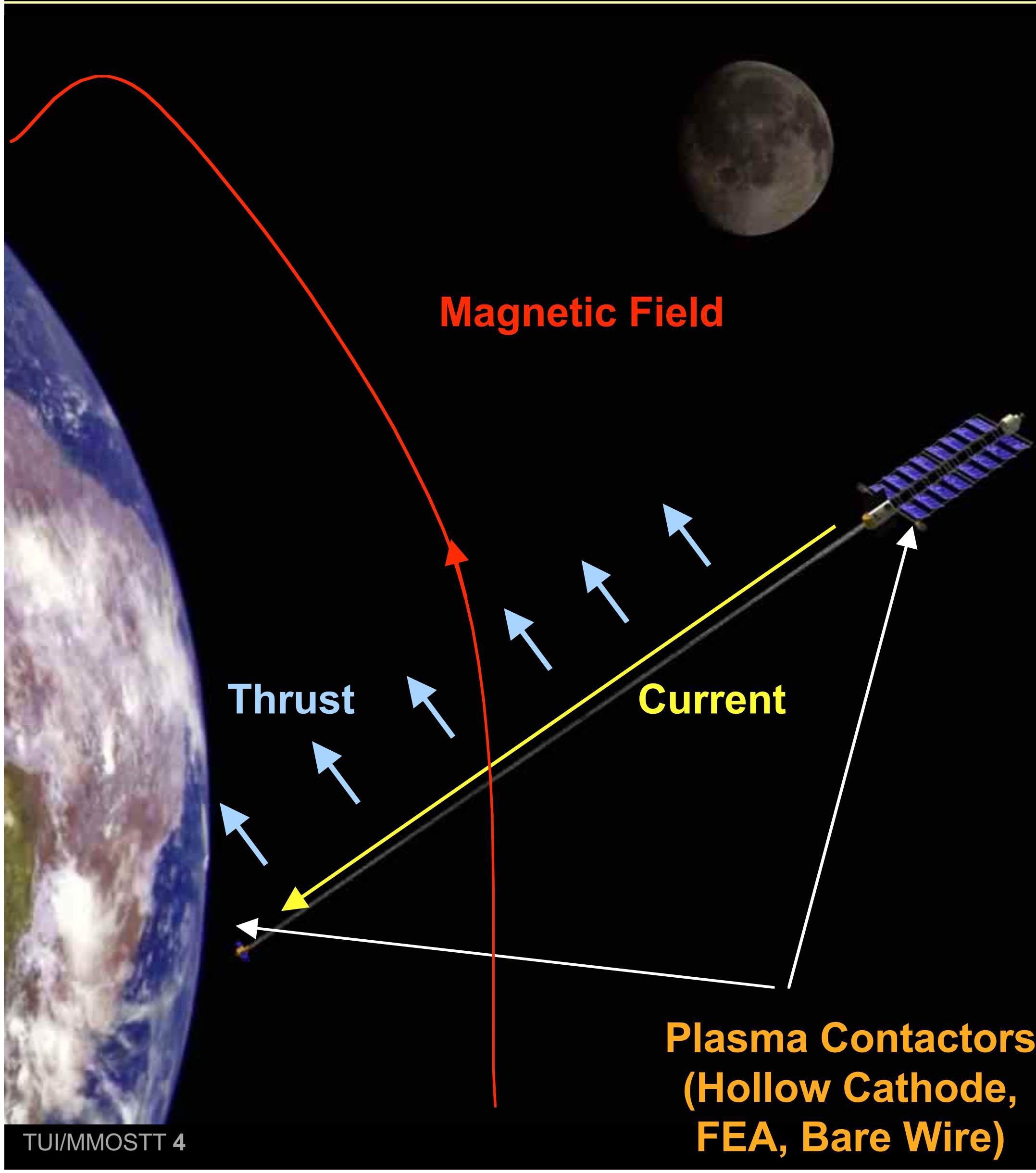
- **Objectives:**
  - **Perform Technical & Economic Analysis of Tether Transport Systems**
  - **Identify Technology Needs**
  - **Develop Conceptual Design Solutions**
  - **Prepare for Technology Development Efforts and Flight Experiments to Demonstrate Tether Transport Technology**

Momentum-Exchange Tether Boost Facility

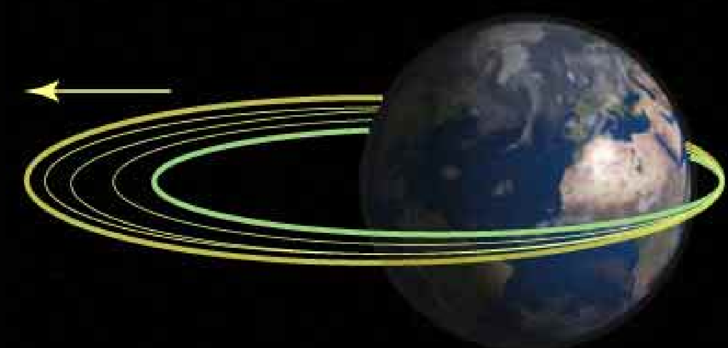
- High-strength tether rotates around orbiting control station
- Tether picks payload up from lower orbit and tosses payload into higher orbit
- Tether facility gives some of its orbital momentum & energy to payload
- Tether facility orbit must be restored to enable it to toss additional payloads



Electrodynamic Reboost



- Power supply drives current along tether
- Plasma contactors exchange current with ionosphere
- Plasma waves close current "loop"
- Current "pushes" against geomagnetic field via JxB Force



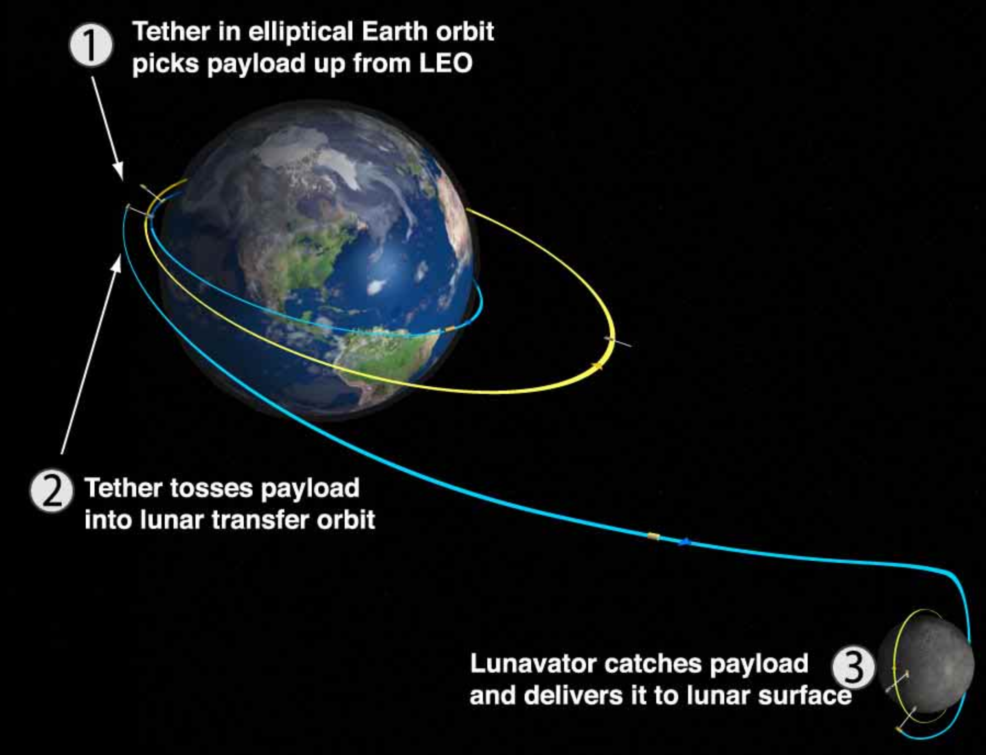
Momentum-Exchange/Electrodynamic-Reboost Tethers:

Summary of Advantages

- Tether Boost Facilities Can Provide a Fully-Reusable In-Space Propulsion Architecture
  - LEO ⇔ MEO/GTO
  - LEO ⇔ Lunar Surface
  - LEO ⇔ Mars
  - ETO Launch, in combination with Hypersonic Airplane/RLV
- Momentum Exchange + Electrodynamic Tether Can Enable Propellantless Propulsion Beyond LEO
- Rapid Transfer Times
  - 5 days to Moon
  - 90-130 days to Mars
- Operational Tether System Can Be Tested Before Use With High-Value Payloads
- Reusable Infrastructure + Low Consumables
  ⇒ Lower Cost

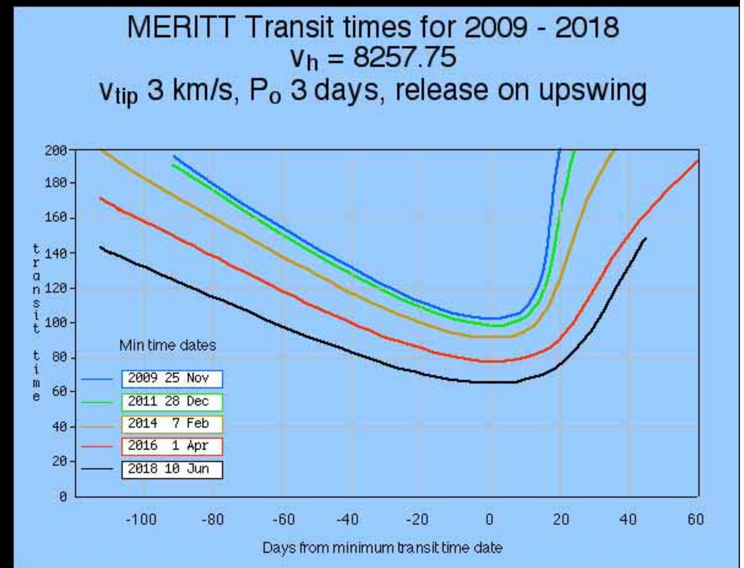
Cislunar Tether Transport System

- Developed Orbital Architecture for Round Trip LEO⇔Lunar Surface Transport
- Whole System Launch Mass = 30x Payload Mass
  - LEO Tether Boost Facility Mass = 13x Payload Mass, Lunar Tether Facility = 17x Payload
- 13 Payloads/Year
- Incremental Commercial Development Path

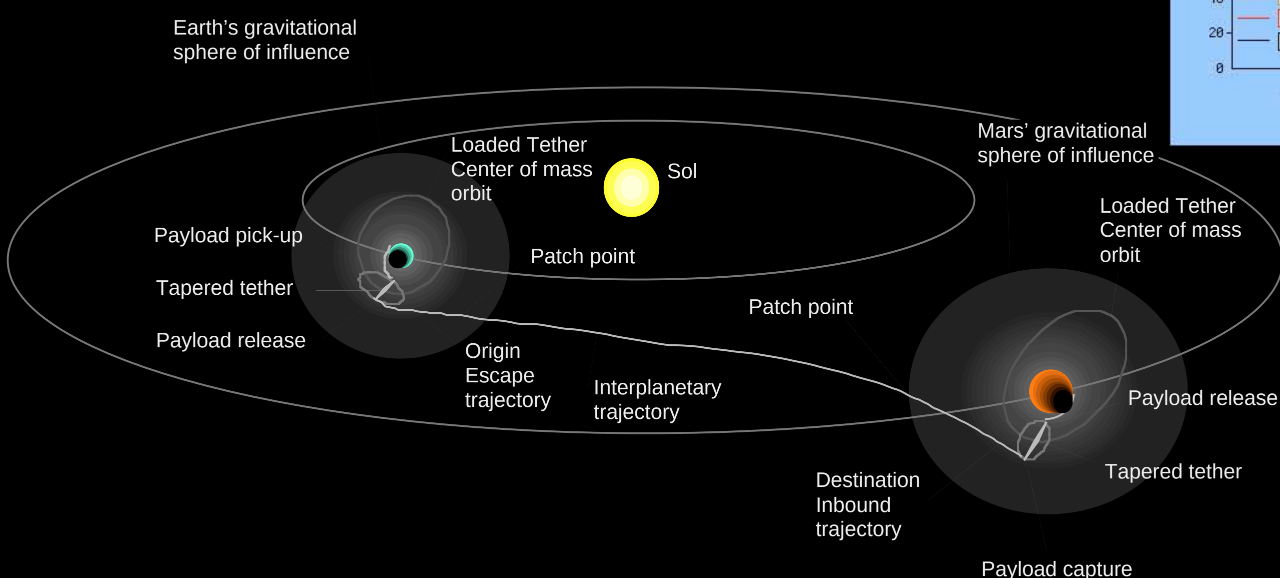


Rapid Earth-Mars Transport

- **Reusable Architecture for Round Trip Earth to Mars Transport**
- **Rapid Transfer Times (90-130 days)**



INTERPLANETARY TRANSPORT USING ROTATING TETHERS

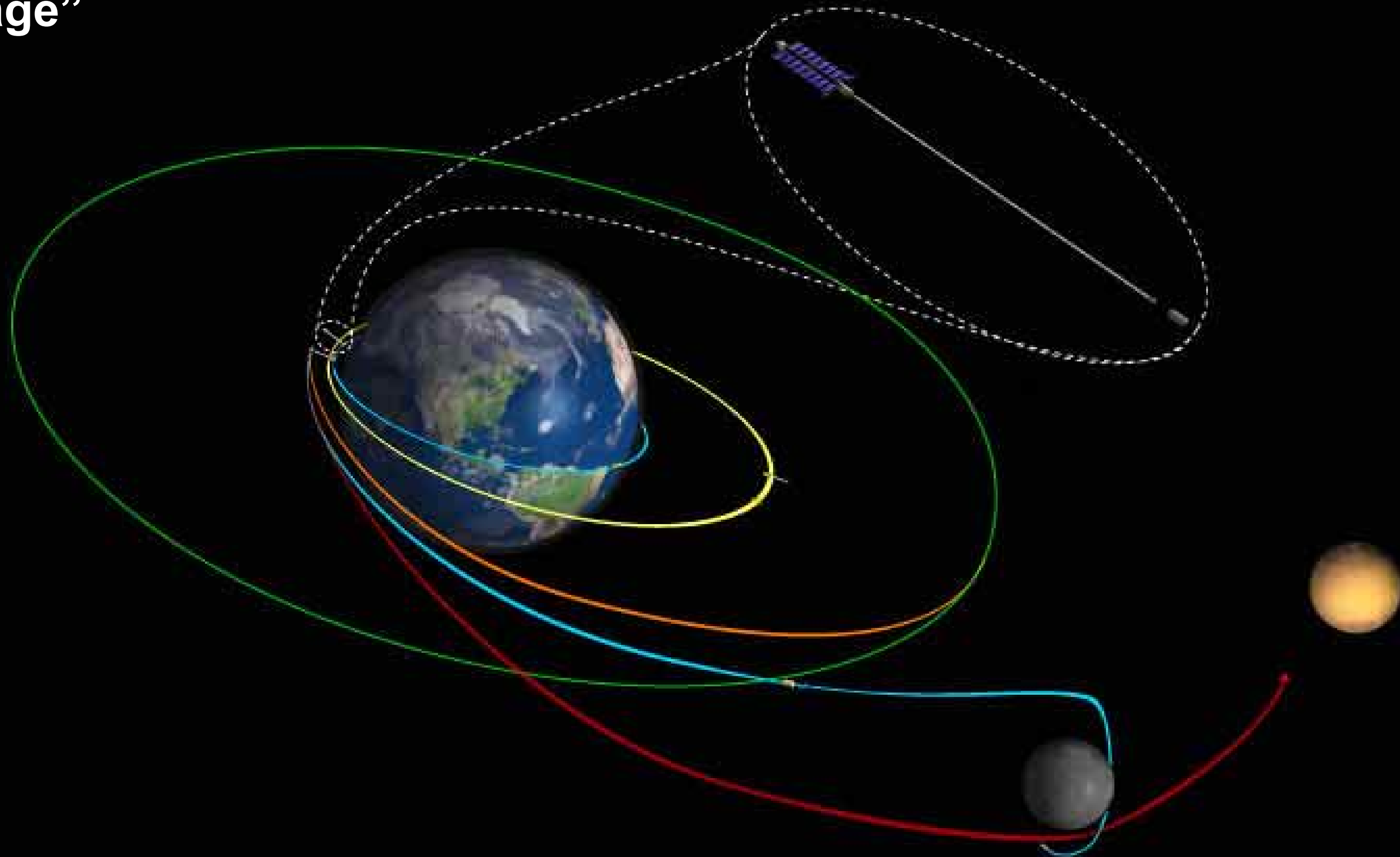


MXER Tethers Included in NASA's IISTP Process

- NIAC Funded MMOSTT and HASTOL efforts have resulted in Momentum-Exchange/Electrodynamic Reboost Tethers being considered in NASA's In-Space Integrated Space Transportation Planning Process
- TUI & NASA/MSFC developed concept designs for Tether Boost Facilities for 4 classes of missions
  - Microsatellite
  - 1 mt Payloads
  - 5 mt Payloads
  - 10 mt Payloads
- IISTP Process evaluated these designs in trade studies for several different scientific missions
- "High-Risk/High Payoff"
- MXER Tethers scored well for several classes of missions
  - High Performance metric

Tether Architecture for LEO-GTO-LTO-Mars Transport

- Tether facility serves as transport hub for multiple destinations
- Tether serves as a zero-propellant, reusable, high-Isp, high thrust "Third Stage"

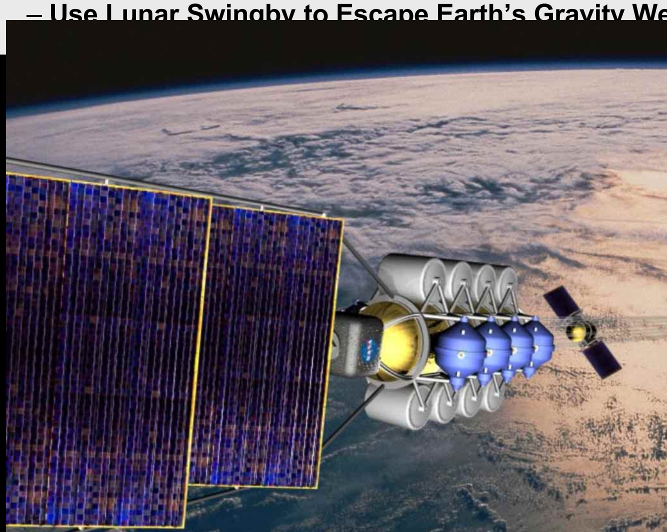


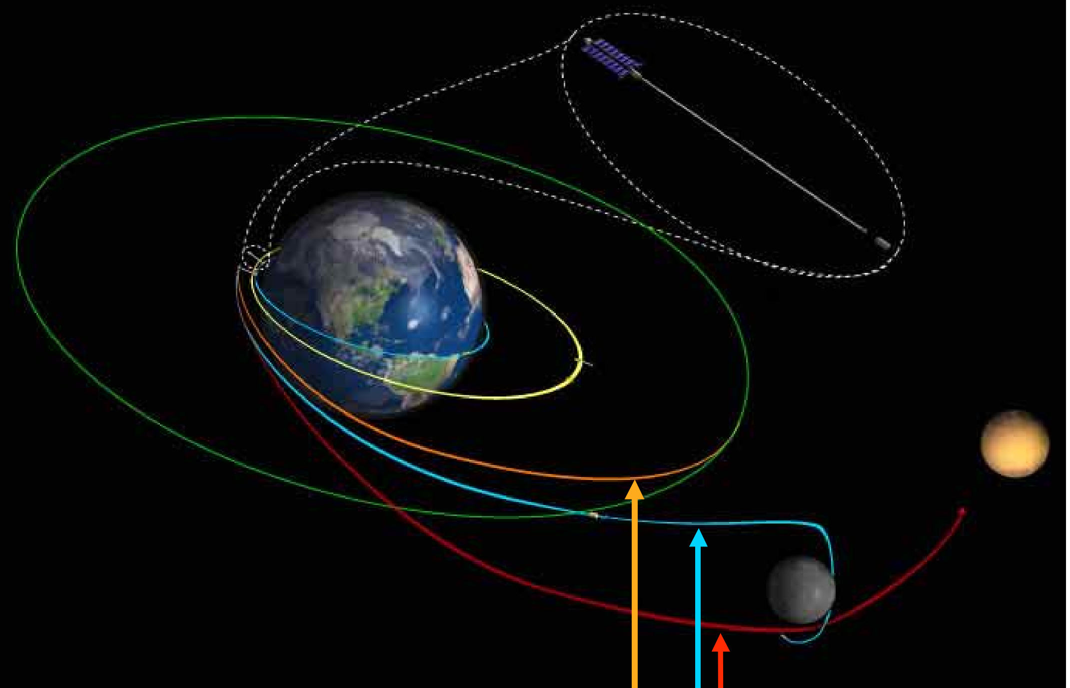
5mt Payload Tether Boost Facility for In-Space Transportation Architecture

- Reusable In-Space Transportation Infrastructure
- Payload Launched to 325 km LEO
- Tether Boosts Payload to Elliptical Orbit
- Tether Uses Electrodynamic Thrust to Reboost

Analysis of Other Propulsion Technologies with MX Tether Assist:

- Delta-II-Class LV Launches 5,000 kg Spacecraft
- Tether Boosts Spacecraft to C_3 = -1.9 km^2/s^2
- High-Thrust Propulsion Systems:
  - Do Injection Burn at Perigee (570 km, 10.62 km/s)
- Low-Thrust Propulsion Systems:
  - Use Lunar Swingby to Escape Earth's Gravity Well





Tether System Point Design:

- Boost 10,000 kg to GTO
- Boost 5,000 kg Vehicle to :
  - Highly Elliptical Orbit (C_3=-1.9)
  - Lunar Transfer Trajectory
  - Escape Via Lunar Swingby
- Tether Facility Launch Mass: 63 mt
  - Deploy using 3 Delta-IV-H LV's
  - Retain Delta Upper Stages for Ballast
  - 200 kW EOL Power Supply for 1 Month Reboost

Net Payoff: Reduced Launch Costs

To launch 5,000 kg to GTO:

- **Using Rockets: Delta IVM+(4,2) or SeaLaunch**
  - **~ $90M**
- **Using Rocket to LEO, Tether Boost to GTO:**
  - **Delta II 7920 (~$45M) or Dnepr 1 (~$13M)**

➢ **1/2 to 1/7 the launch cost**

LEO⇨GTO Boost Facility

- Initial Facility Sized to Boost 2500 kg Payloads to GTO
- First Operational Capability Can Be Launched on 1 Delta IV-H
- Modular Design Enables Capability to be Increased
- Top Level Mission Requirements:

Requirement	Value
Payload Mass	2500 kg at IOC, can grow to follow market
Pickup orbit	300 km equatorial
Release orbit	GTO
Release insertion error	< Delta IV/Ariane 5
Payload environment	< Delta IV/Ariane 5
Turnaround time	30 days
Mission life	10 years +
Collision avoidance	100% of tracked spacecraft
Operational orbit lifetime	15 days
Payload pickup reliability	99%

Mass Properties Breakdown

Control Station Mass: 10,967 kg

Tether Mass: 8,274 kg

Grapple Mass: 650 kg

GLOW: 19,891 kg

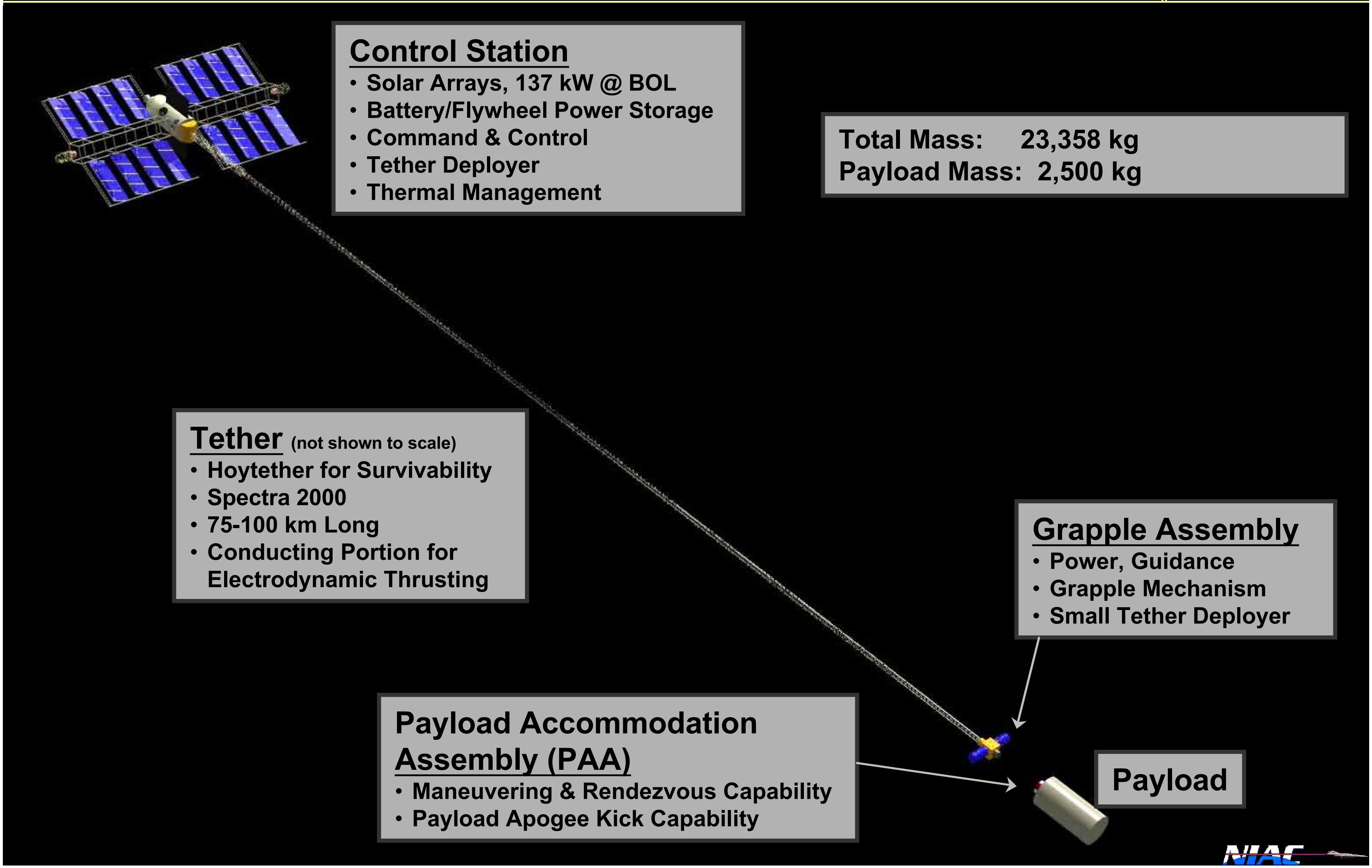
– 15% margin w/in Delta IV-H payload capacity

Expended Upper Stage 3,467 kg

On-Orbit Mass: 23,358 kg

	Qty	Redundancy	Mass Contingency	Unit mass (kg)	Mass with no margin (kg)	Mass with Contingency (kg)	Mass Margin (kg)
LEO Control Station					**10967**	**13267**	**2300**
Thermal Control Subsys	**1**		**15%**		**1104.5**	**1270.1**	**165.7**
Cabling/Harnesses			33%		**749.6**	**997.0**	**247.4**
Structure			25%		**2721.1**	**3401.3**	**680.3**
Electr.Pwr.					**4736.7**	**5409.6**	**673.0**
PV array panels	1	1	13%	1782.9	1782.9	2014.6	
Power Storage	1	1	15%	2860.5	2860.5	3289.5	
PV array drive motors	8	2	13%	3.0	48.0	54.2	
PMAD	1	2	13%	22.7	45.4	51.3	
Downlink Comm Subsys					**1.8**	**2.1**	**0.2**
Downlink Transceiver	1	2	13%	0.7	1.4	1.56	
Downlink antennae	2	1	13%	0.2	0.5	0.51	
TFS Net Comm Subsys					**1.8**	**2.1**	**0.2**
Comm. antennae	2	1	13%	0.2	0.5	0.51	
Transceiver	1	2	13%	0.7	1.4	1.6	
C&DH					**26.0**	**29.4**	**3.4**
Computer	1	2	13%	13.0	26.0	29.4	
TT&C					**6.9**	**7.8**	**0.9**
transponder	1	2	13%	3.5	6.9	7.8	
ADCS					**200.9**	**213.8**	**12.9**
ED Tether Power Subsys					**417.4**	**603.4**	**186.0**
Plasma Contactor (FEAC)	1	2	25%	45.4	90.8	113.5	
PMAD/PCUt	1	2	50%	163.3	326.6	489.9	
Docking & I/C Subsys					**0.5**	**0.54**	**0.04**
Beacon	1	1	8%	0.5	0.5	0.54	
Tether Deploy & Control					**1000.0**	**1330.0**	**330.00**
Tether reeling assembly	1	1	33%	1000.0	1000.0	1330.0	

Tether Boost Facility



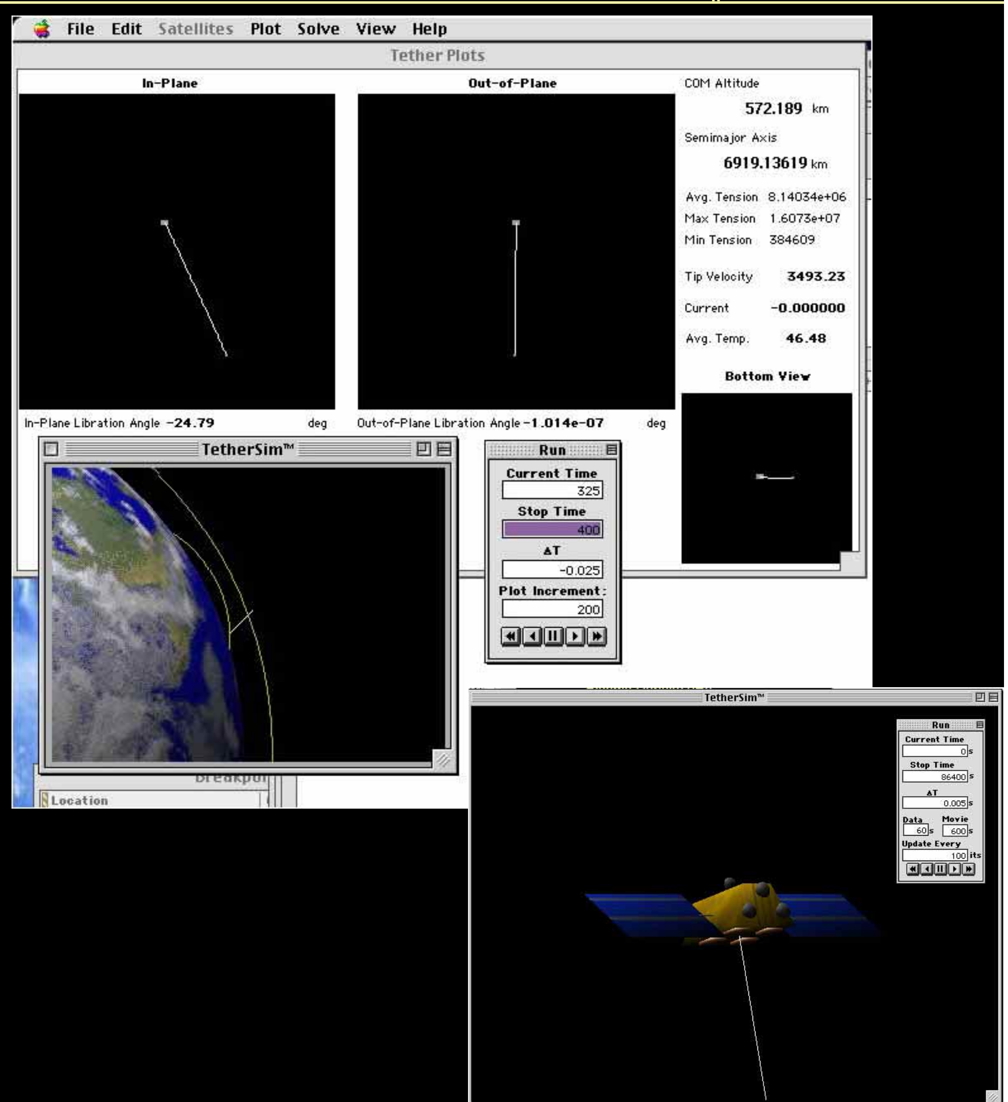
NIAC Efforts Have Developed Improved Tether Analysis Tools

Tether System Design:

- Tapered tether design
  - Spectra 2000
- Orbital mechanics considerations to determine facility mass required

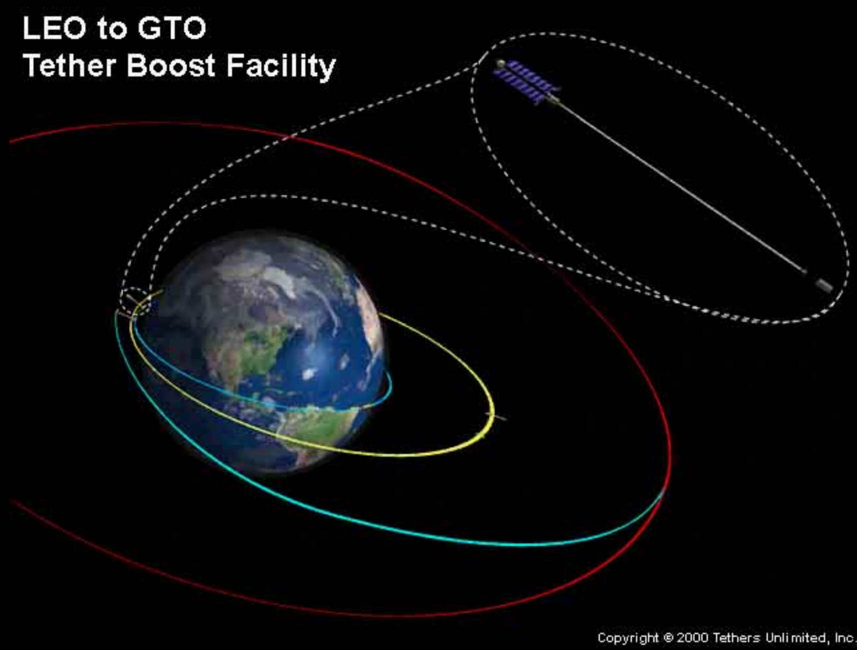
Tether operation: TetherSim™

- **Numerical Models for:**
  - Orbital mechanics
  - Tether dynamics
  - Electrodynamics
  - Hollow Cathode & FEACs
  - Geomagnetic Field (IGRF)
  - Plasma Density (IRI)
  - Neutral Density (MSIS '90)
  - Thermal and aero drag models
  - Endmass Dynamics
  - Payload Capture/Release
- **Interface to MatLab/Satellite Tool Kit**



LEO⇨GTO Boost Facility

- TetherSim™ Numerical Simulation (10x real speed)
  - Tether Dynamics, Orbital Mechanics

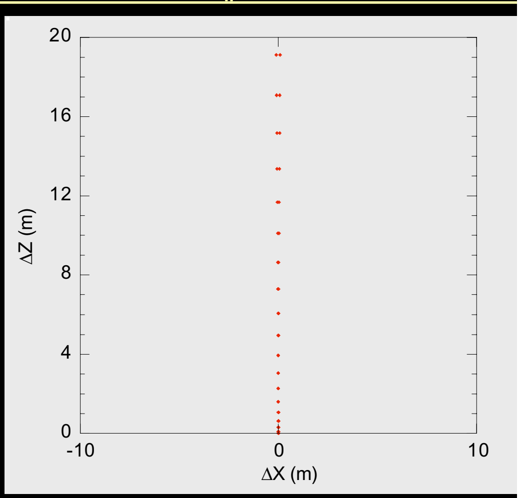


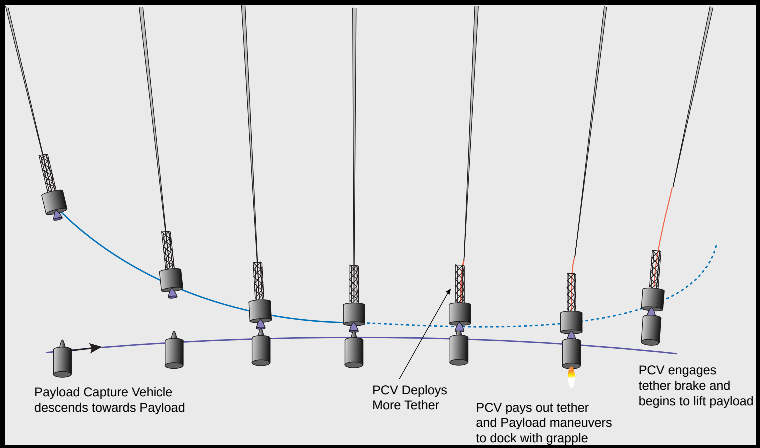
Technology Readiness Level

- Boeing & TUI Performed TRL Analysis of MXER Tether Technologies
- Many necessary components are already at high TRL
- TRL Analysis Indicates Areas for Future Work to Address:
  - Power management subsystem
  - Thermal control subsystem
    - Higher power than previously flown systems
  - Electrodynamic Propulsion Subsystem
    - Plasma contactors
    - Dynamics control
  - Automated Rendezvous & Capture technologies
    - Prediction & Guidance
    - Grapple Assembly & Payload Adapter
      - Some work ongoing in HASTOL Ph II effort
  - Flight Control Software
  - Traffic Control/Collision Avoidance

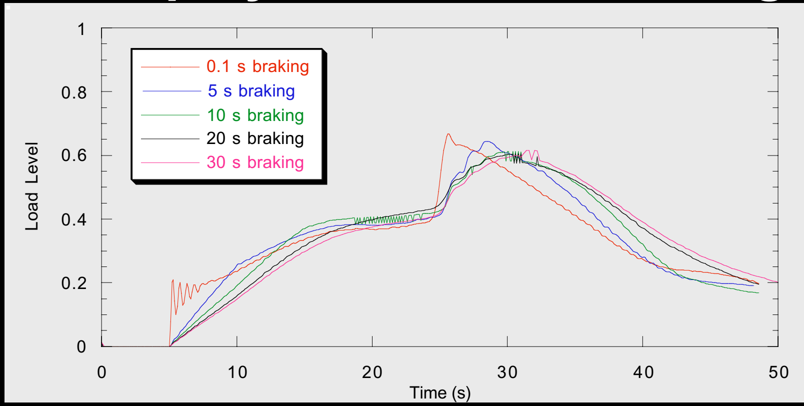
Rendezvous

- Rapid AR&C Capability Needed
- Relative Motion is Mostly in Local Vertical
- Tether Deployment Can Extend Rendezvous Window



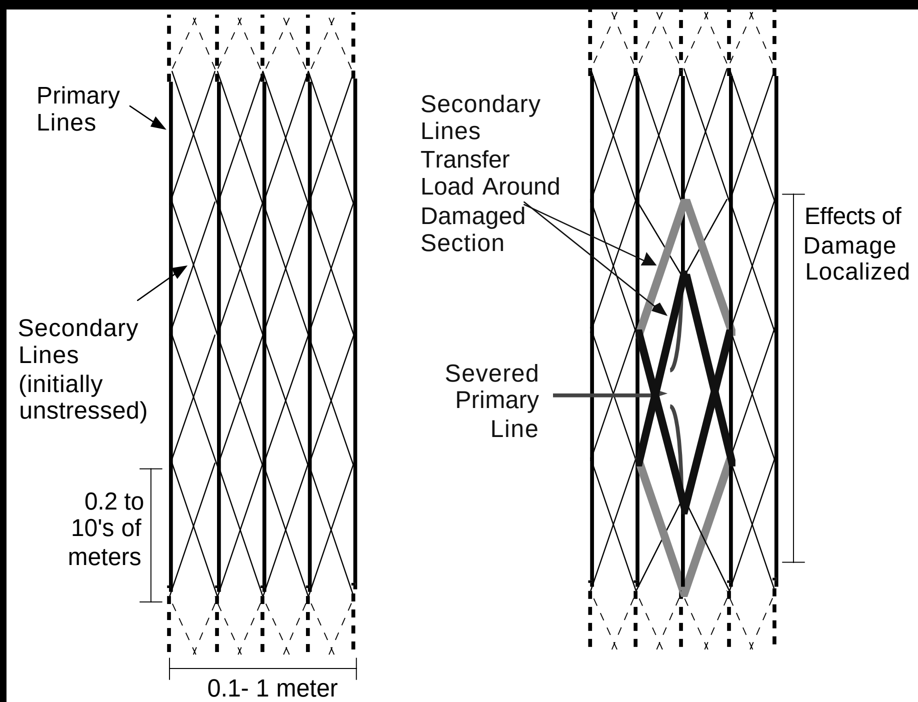


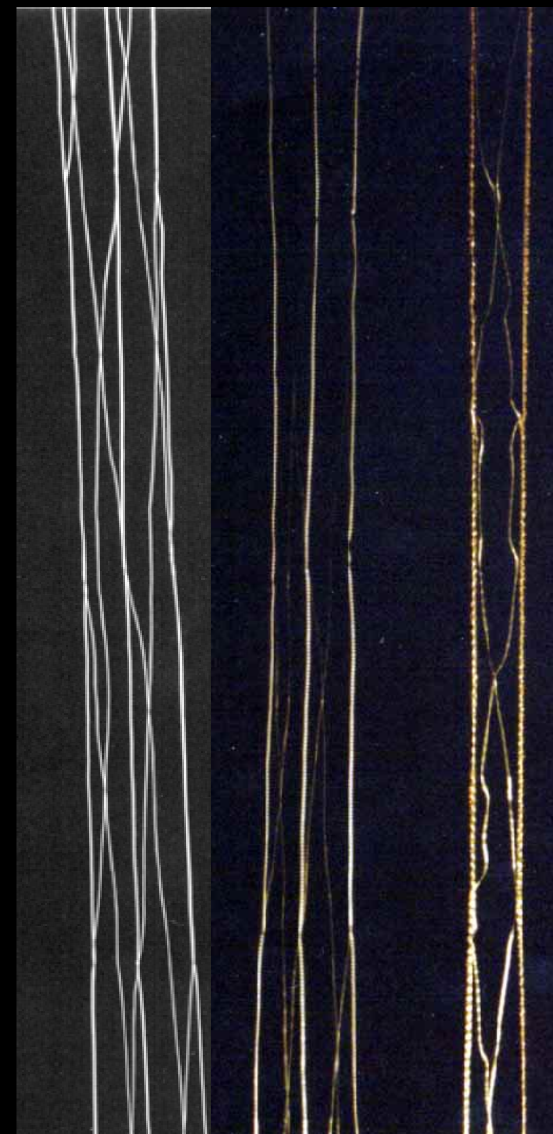
- Additional Tether Deployment Under Braking Can Reduce Shock Loads



Space Debris-Survivable Tether

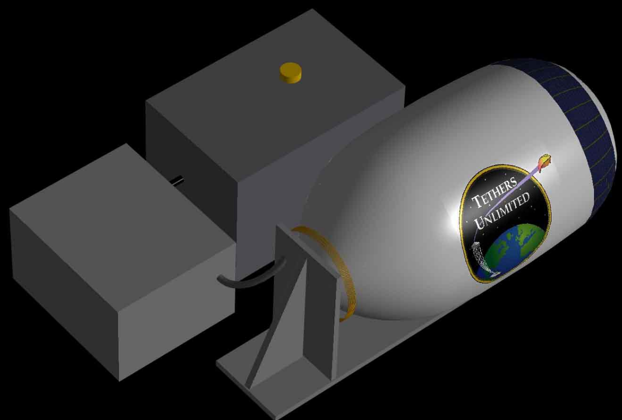
- **Micrometeoroids & Space Debris Will Damage Tethers**
- **Solution approach: spread tether material out in an open net structure with multiple redundant load/current paths**





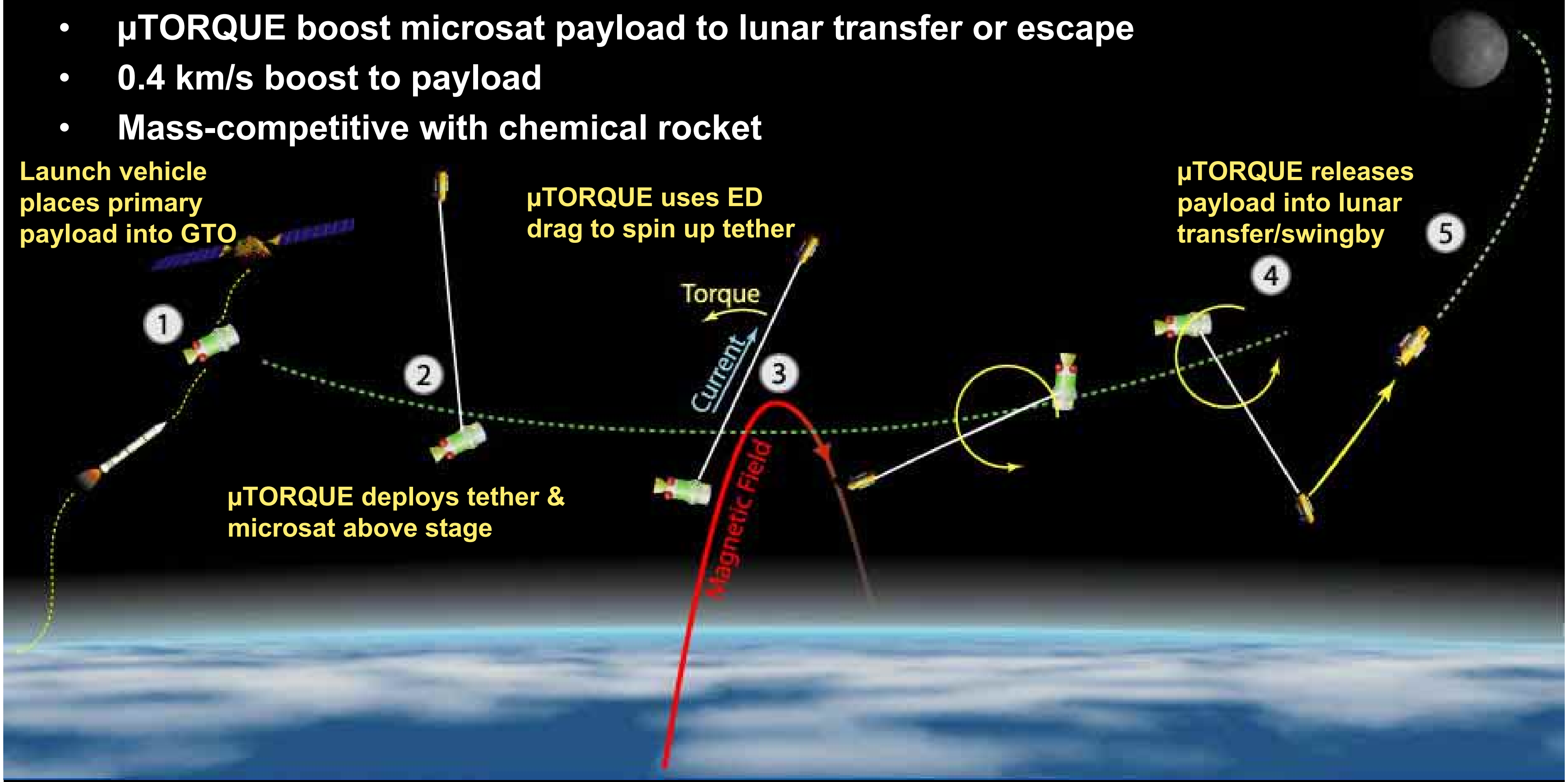
Proposed RETRIEVE Tether Experiment

- Candidate Secondary Experiment for XSS-11
- $800K in Initial Development funds from AFRL
- Small ED tether system deorbits μSat at end of mission
  - Activated only after primary mission completed
- Mass: 3.5 kg
- Demonstrate
  - Controlled orbital maneuvering with ED tether
  - Long life tether
  - Stabilization of tether dynamics



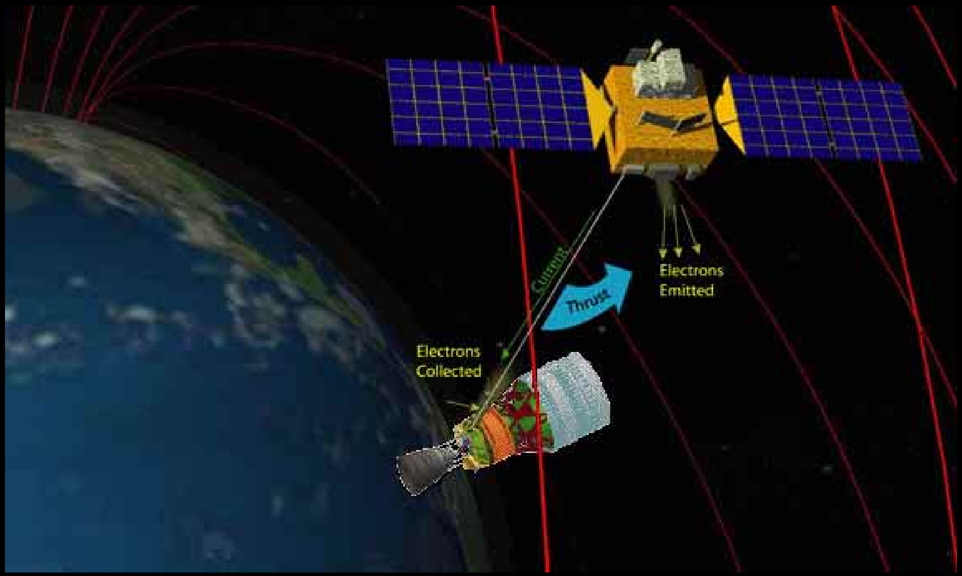
µTORQUE: MX Tether to Boost µSat to Lunar Transfer or Escape

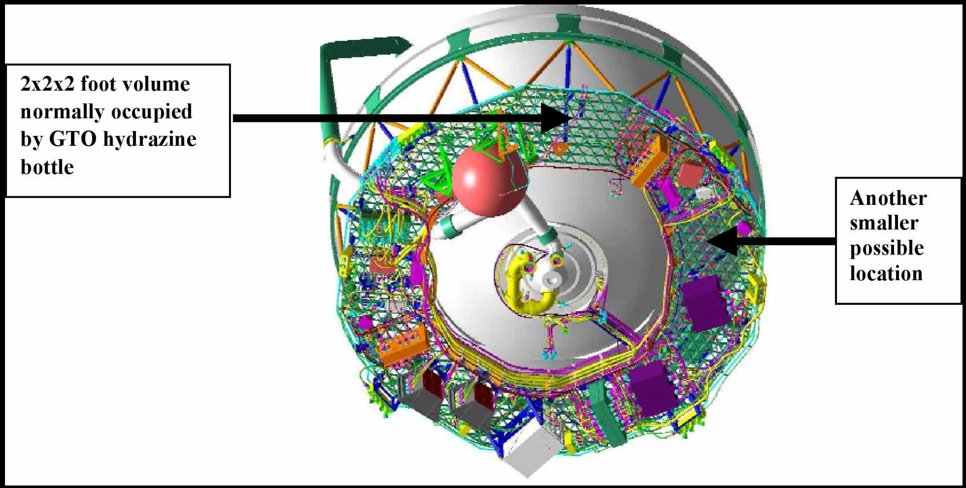
- Microsatellite Tethered Orbit Raising QUalification Experiment
- Build Upon RETRIEVE to Create Low-Cost Demo of MXER tether technology
- Secondary payload on GEO Sat launch
- µTORQUE boost microsat payload to lunar transfer or escape
- 0.4 km/s boost to payload
- Mass-competitive with chemical rocket



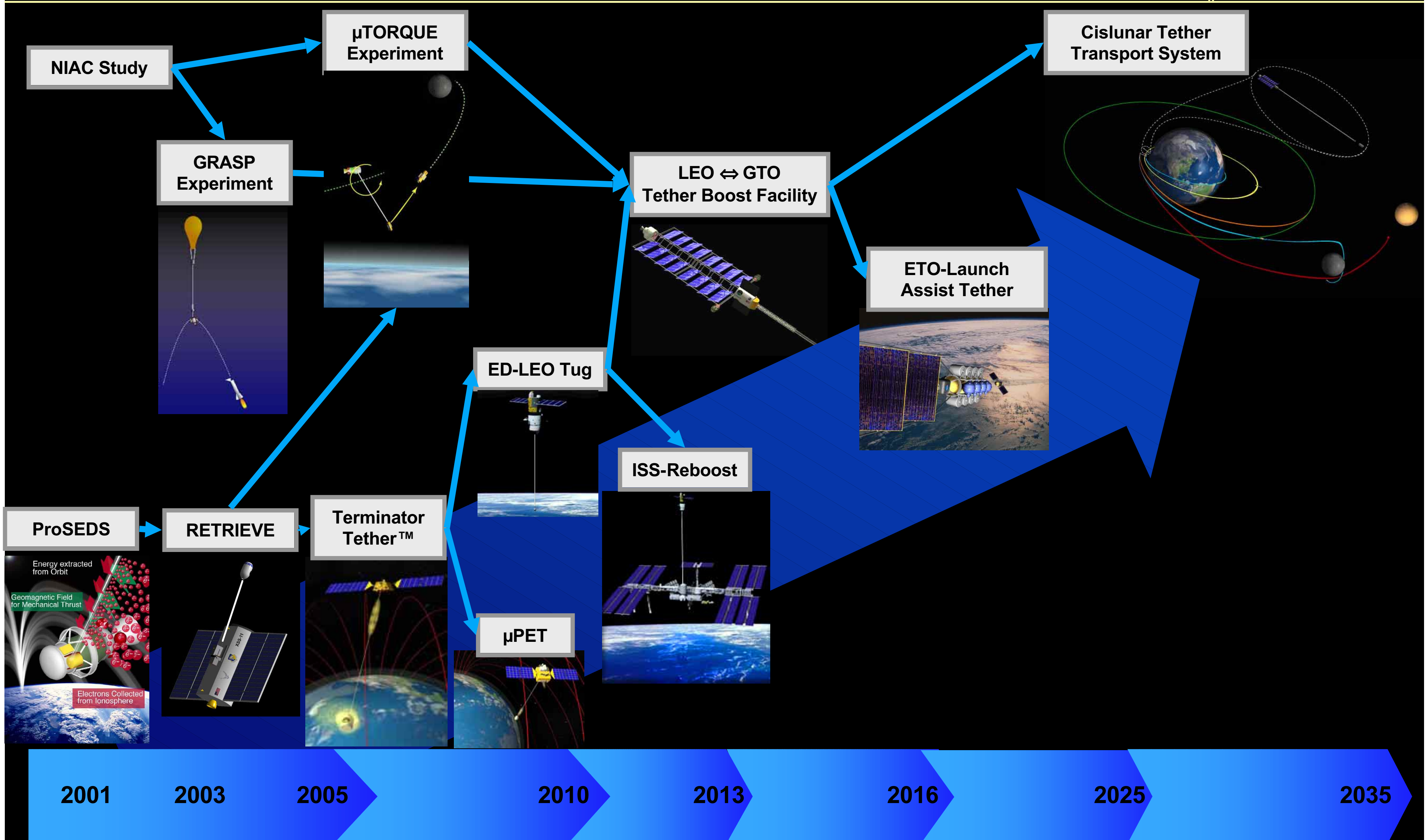
µTORQUE on Delta IV

- Delta-IV Secondary Payload
- ~100 kg weight allocation
- Boost ~80kg microsat from LEO to low-MEO





Momentum Exchange/Electrodynamic Reboost Tether Technology Roadmap



Opportunities for NASA Technology Development

- Expand AR&C Capabilities for Rapid Capture
- High Power & High Voltage Space Systems
- Electrodynamic Tether Physics
- Debris & Traffic Control Issues
- Conduct Low-Cost Flight Demo of Momentum-Exchange Tether Boost

Modest NASA Investment in Technology Development Will Enable Near-Term Space Flight Demonstration

Contributors

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- Boeing/Huntsville - Mike Bangham, Beth Fleming, John Blumer, Ben Donohue, Ronnie Lajoie, Lee Huffman
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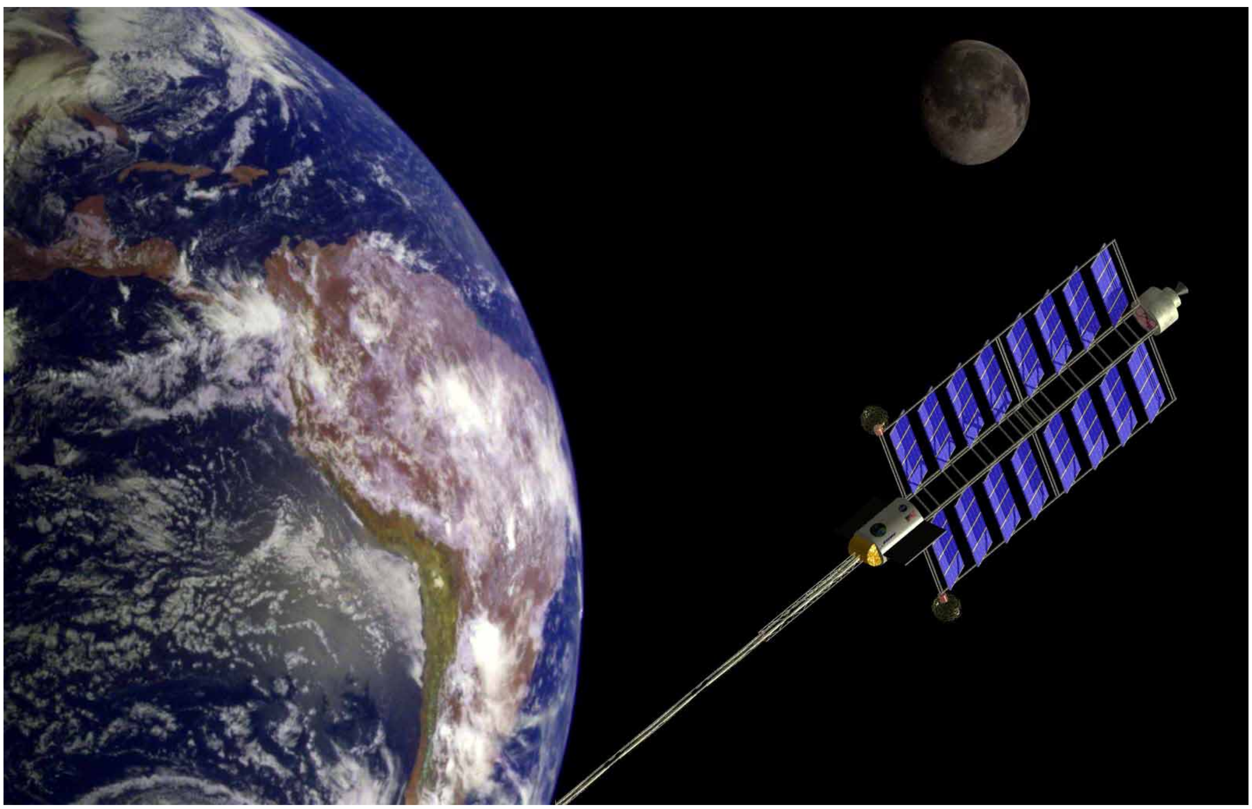

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COMMERCIAL DEVELOPMENT OF A TETHER TRANSPORT SYSTEM

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Abstract

Momentum-Exchange/Electrodynamic-Reboost tether facilities can form the infrastructure for a fully-reusable low-cost in-space transportation architecture. Several technical challenges must be met to enable tether transport systems to be fielded, including development of rapid AR&C capabilities and techniques for building and controlling the tether facilities. A tether transport system to carry frequent traffic between Earth, the Moon, and Mars can be developed in a modular, incremental fashion, in which each component can generate revenue to fund the development of the rest of the system, much as the first railroads were developed. The initial Tether Boost Facility would be sized for launch on a single large rocket vehicle, and would be designed to immediately service traffic to GEO. The capacity of this facility could then be built incrementally, and additional tether facilities deployed to handle Earth-to-Orbit Assist, LEO⇔Lunar Surface round-trip travel, and deployment of manned Mars bases.

Introduction

By providing a fully reusable, zero-propellant infrastructure for in-space transportation, momentum-exchange tethers have the potential to reduce the costs of delivering payloads to GEO, the Moon, Mars, and other destinations by an order of magnitude or more. Under funding from NASA's Institute for Advanced Concepts (NIAC), Tethers Unlimited, Inc. and the Boeing Company are developing an architecture for a tether transportation system. This system will utilize momentum-exchange techniques and electrodynamic tether propulsion to transport multiple payloads *with little or no propellant consumption*. The tether transportation architecture is designed to be built incrementally, with each component able to perform a useful revenue-generating mission to help fund the deployment of the rest of the system. The first component of the system will be a Tether Boost Facility that will transfer satellites and other payloads from low Earth orbit (LEO) to geostationary transfer orbit (GTO). This same facility will also be capable of boosting payloads to lunar transfer orbit (LTO). Later components will increase the payload capacity of the Tether Boost Facility and enable frequent round-trip travel to the surface of the Moon[1,2] and to Mars.[3] In this paper we discuss an architecture and incremental development plan for an Earth-Moon-Mars Tether Transportation System.

Background

Momentum-Exchange Tethers

In a momentum-exchange tether system, a long, thin, high-strength cable is deployed in orbit and set into rotation around a massive central body. If the tether facility is placed in an elliptical orbit and its rotation is timed so that the tether will be oriented vertically be low the central body and swinging backwards when the facility reaches perigee, then a grapple assembly located at the tether tip can rendezvous with and acquire a pay

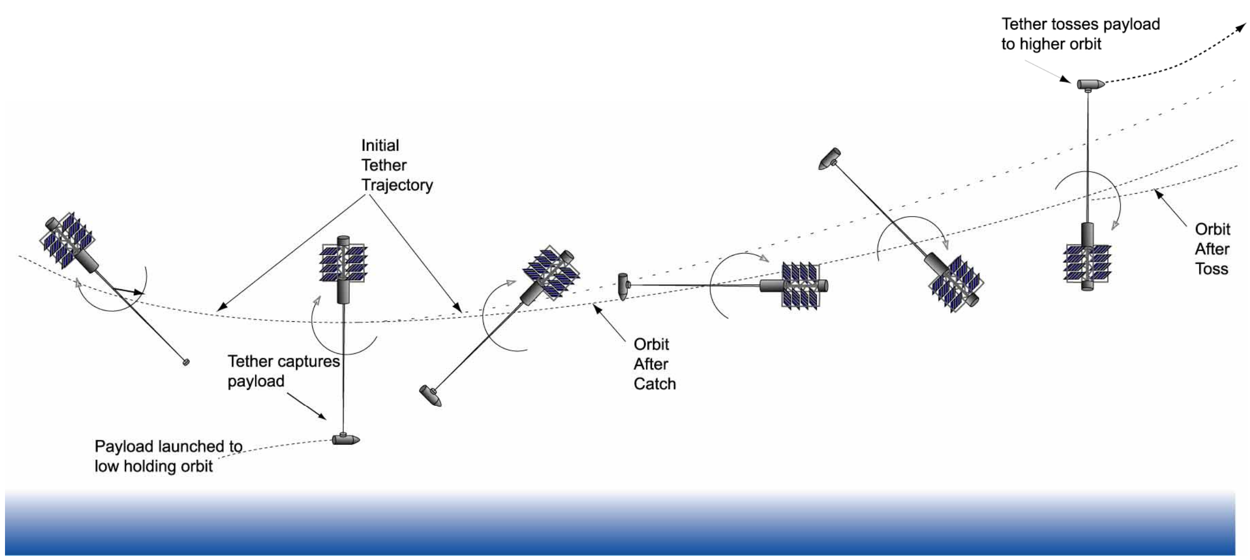

Figure 1. Momentum Exchange Tether catching and tossing payload.

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load moving in a lower orbit, as illustrated in Figure 1. Half a rotation later, the tether can release the payload, tossing it into a higher energy orbit. This concept is termed a momentum-exchange tether because when the tether picks up and throws the payload, it transfers some of its orbital energy and momentum to the pay load. The tether facility's orbit can be restored later by reboosting with propellantless electrodynamic tether propulsion or with high specific impulse electric pro pulsion; alternatively, the tether's orbit can be restored by using it to de-boost return traffic payloads.

Key Advantages

A tether transportation system has several advantages compared to conventional and other advanced in-space propulsion systems:

- *(Near) Zero Propellant Usage*

Chief among these advantages is the ability to elimi nate the need for propellant expenditure to perform pay load transfers. Of course, some propellant expenditure will be needed for trajectory corrections and rendezvous maneuvering, but these requirements will be very small, a few tens of meters per second. The ability to shave several thousands of meters per second from the ΔV needed to deliver a payload to its destination can enable customers to utilize much smaller launch vehi cles than would be required with a rocket-only system, greatly reducing total launch costs. In a later section we will discuss the potential cost advantage for a GEO-bound payload.

- *Short Transfer Times*

A momentum-exchange tether system provides its ΔV to the payload in an essentially impulsive manner. Thus the transfer times in a tether system are very short, comparable to rocket-based systems. Although orbit transfer systems based on electric propulsion schemes can offer low propellant usage, they invariably require long transfer times due to their low thrust lev els. The short transfer times offered by a momentum-exchange tether system can play an important role in minimizing the lost-revenue time that a commercial satellite venture would have to accept while it waits for its satellite to reach its operational orbit and begin gen erating revenue.

- *Reusable Infrastructure*

Once deployed, a tether boost facility could transfer many, many payloads before requiring replacement. Thus the recurring costs for payload transport could be reduced to the cost of operations. A tether transporta tion system thus would be somewhat analogous to a terrestrial railroad or public-transit system, and might achieve comparable cost reductions for transporting many payloads.

- *Fully Testable System*

Another important but often overlooked advantage of a tether transportation system is that the components that perform the actual payload transfer operations can be fully tested *in space operations* before being used for critical payloads. In conventional rocket systems, engine components and other key elements can be tested on the ground, and many individual units can be flown to provide reliability statistics, but to-date only the Shuttle has re-used rocket engines (with significant maintenance after each flight!). In a tether transporta tion system, the tether facility could be tested many times with "dummy" payloads – or, better yet, with low inherent value payloads such as water or fuel – to build confidence for use on high value or manned pay loads.

Key Limitations

For a fair analysis, we should also point out several limitations of tether transport systems relative to other technologies:

- *Limits on Payload Inclination*

One potential limitation to the competitiveness of a tether transportation system is the fact that a tether boost facility can only deliver payloads to trajectories with nearly the same inclination as that of the tether facility. The operation of a tether boost facility will be least complicated if it operates in an equatorial orbit, because orbital perturbations will be minimized and electrodynamic reboost will be most efficient there. An equatorial tether orbit is excellent for GEO satellite traffic and delivering payloads to the Moon or inter planetary trajectories, but less advantageous for deploy ing LEO or MEO constellations such as a GPS system, which would typically use a moderate or high inclina tion orbit. To draw an analogy to a terrestrial transpor tation system, a tether transport system would be like a railway system, which services cities with a train stop very efficiently, but may require additional transport methods to deliver materials to outlying towns.

- *Payload Scheduling*

Another issue for a tether transport system is the relative inflexibility for scheduling payload transfers. This is not so much an inherent issue for tethers but rather arises from the nature of the orbital mechanics that a tether system utilizes. Just as a rocket-based system must launch during a short "window" in order to deliver a payload to the right orbit, a tether and payload will have a window to perform the rendezvous, grappling, and toss which will send the payload to the correct destination; for the tether, this window will usually be significantly tighter.

- *Rendezvous Requirements*

For a tether transport system to achieve its full potential, it must provide the capability for a payload to rendezvous with a rotating tether. This will require very high accuracy in propagating the tether trajectory and maneuvering the payload to be in just the right place at just the right time, with just the right velocity. Consequently, a tether transport architecture must include components that will provide the payload with guidance and maneuvering capabilities in excess of what would be required of it in a conventional system. These components will be an additional expense, and until full round-trip traffic is established, will likely represent a significant recurring cost in the system.

Prior Work on Tether Transport Architectures

Several prior research efforts have investigated conceptual designs for momentum-exchange tether systems. In 1991, Carroll proposed a tether transport facility that could pick payloads up from suborbital trajectories and provide them with a total ΔV of approximately 2.3 km/s.[4]

Soon thereafter, Forward[5] proposed combining this system with a second tether in elliptical Earth orbit and a third tether in orbit around the Moon to create a system for round-trip travel between suborbital Earth trajectories and the lunar surface. In 1997, Hoyt[6] developed a preliminary design for this "LEO to Lunar Surface Tether Transport System."

In 1998, Bangham, Lorenzini, and Vestal developed a conceptual design for a two-tether system for boosting payloads from LEO to GEO.[7] Their design proposed the use of high specific impulse electric thrusters to restore the orbit of the tether facilities after each payload boost operation. Even with the propellant mass requirements for reboost, they found that this system could be highly economically advantageous compared chemical rockets for GEO satellite deployment.

Under a Phase I NIAC effort, Hoyt and Uphoff[1] refined the LEO⇒Lunar system design to account for the full three-dimensional orbital mechanics of the Earth-Moon system, proposing a "Cislunar Tether Transportation System" illustrated in Figure 10. This architecture would use one tether in elliptical, equatorial Earth orbit to toss payloads to minimum-energy lunar transfer orbits, where a second tether, called a "Lunavator™" would catch them and deliver them to the lunar surface. The total mass of the tether system, could be as small as 27 times the mass of the payloads it could transport.

The same NIAC effort also resulted in a preliminary design by Forward and Nordley[3] for a "Mars-Earth Rapid Interplanetary Tether Transport (MERITT)" system capable of transporting payloads on rapid trajectories between Earth and Mars.

Momentum-exchange tethers may also provide a means for reducing the cost of Earth-to-Orbit (ETO) launches. This architecture would use a hypersonic airplane or other reusable launch vehicle to carry a payload up to 100 km altitude at Mach 10-12, and handing it off to a large tether facility in LEO which would then pull it into orbit or toss it to either GTO or escape.[8,9]

Building a Tether Transport System

If a tether-based transportation architecture is to be developed in part or in whole by a commercial venture, the deployment of the system must follow a path that is commensurate with a viable business plan. An Earth-Moon-Mars Tether Transportation System will require at least three tether facilities, one in Earth orbit, a second in lunar orbit, and a third in Martian orbit. Each of these will require a significant investment in technology development, system fabrication, and facility launch. To keep the capital investments small enough for a business plan to close, the system architecture must be designed in a manner in which the first components can immediately serve useful functions to generate revenue to fund the development of the rest of the system. This would be quite analogous to the development of the cross-continental railroads, where each extension of the rail line was used to generate revenue to help build the rest of the line.

In this document we will attempt to lay out a roadmap for developing a full Tether Transportation System, beginning by discussing the technology development needed to prepare for the deployment of tether boost facilities, and then describing a possible sequence for building a tether transport system to service commercial transport markets.

First Steps: Technology Development and Demonstration

We have conducted an evaluation of the Technology Readiness Levels (TRL's) of the components and technologies required for the tether facilities and other subsystems of a tether transportation system. Many of the required technologies, such as communications & control, solar power systems, thermal control, power storage, and plasma contactors are already at relatively advanced readiness levels, or are expected to be brought to high levels within the next few years by ongoing NASA and commercial programs. Several key technologies, however, are unique to momentum-exchange and electrodynamic tether systems, and will require investment in technology development and risk reduction demonstrations in order to enable the commercial development of tether transportation systems. These technologies are:

- *Tether Rendezvous & Grappling*

As mentioned previously, the rendezvous and grappling maneuver is currently the "tall technology tent pole" for momentum-exchange tether systems. For a payload to successfully grapple with a rotating tether, the system must first obtain a very accurate prediction for the position and velocity of the tether tip grapple assembly at the appropriate pick-up time. The payload must then maneuver into an orbit properly phased so that it will be at that position at the pick-up time. When the tether grapple and the payload do come into proximity, the payload must then maneuver to meet up with the grapple and a secure, high-strength connection must be made between the payload and grapple within a relatively short period of time – typically 5-15 seconds. While this is a much shorter time period than has been demonstrated in space to date, other systems have demonstrated rendezvous and capture on equivalent or even shorter timescales. One example would be the landing of jets on an aircraft carrier; this maneuver occurs with high relative velocities, unpredictable relative accelerations, and small physical windows for successful capture of the aircraft's hook by the arresting rope, yet it is performed successfully many times every day. A second example would be the mid-air capture of film canisters dropped by surveilance satellites. This system again had short (~2 seconds) rendezvous windows and high relative velocities, yet this maneuver was performed many times with a 100% success rate.

- *Tether Dynamics Control and Stabilization*

The dynamics of flexible tethers in orbit are complex, and system that utilize electrodynamic propulsion must be controlled to avoid problems with dynamical instabilities. TUI has already developed a simple method for stabilizing the dynamics of the Terminator Tether™, an electrodynamic tether drag system.[10] A momentum-exchange/electrodynamic-reboost tether facility, however, will require a more complex dynamics control system to maintain optimum performance of the tether thrusting and ensure that tether dynamics do not adversely impact the rendezvous and capture maneuvers.

- *High-Strength Survivable Tether with Integrated Electrodynamic Tether*

TUI has already demonstrated fabrication of multi-kilometer lengths of conducting multiline tethers and nonconducting tethers made of high-strength fibers such as Spectra 2000. However, a tether boost facility will require a very high strength-to-weight micrometeoroid survivable tether structure that has both high-strength fibers and conducting elements for electrodynamic thrusting. Furthermore, the electrodynamic component of this tether must be designed to reliably operate at many kilovolts of potential relative to the tether facility and ambient plasma.

- *High-Power, High-Voltage Systems*

In order to perform electrodynamic thrusting on a Tether Boost Facility that has a tether length of many 10's of kilometers, the power system on the facility's control system must be capable of processing many kilowatts of powers and converting them to voltages on the order of 20 kV, while ensuring that no electrical arcing can occur to threaten the integrity of the tether or other systems.

- *Tether Orbit Propagation & Collision Avoidance*

A Tether Boost Facility will be a very large object moving through altitudes where there are many existing satellites and space debris objects. Although a survivable space tether structure such as the Hoytether™ can enable the tether to withstand degradation by impacts with small pieces of space debris, the tether system will still have to deal with large objects that may get in its way. One of the significant issues for this is developing accurate and fast methods for propagating the orbit and dynamical behavior of a tethered system so that the tether system controllers can reliably predict close-encounter events and command avoidance maneuvers. Working to our advantage, however, is the fact that a momentum-exchange/electrodynamic-reboost tether facility will have significant ΔV capabilities using its electrodynamic thrusting. Thus if close encounters can be predicted with sufficient advanced notice, the tether facility can avoid these encounters.

Suggested Technology Development Efforts:

In order to address the technology needs listed above, there are several development efforts that could significantly advance the technology readiness levels of appropriate solutions with relatively low investment requirements.

Grapple Mechanism Development

The payload-tether rendezvous is the most significant challenge for a momentum-exchange tether system. There are, however, several grappling concepts that could make this problem more tractable. One concept, originally suggested by Tillotson and recently improved by Sorenson,[11] is illustrated in Figure 2. In this concept, the tether grapple assembly at the end of the tether would open a net structure, providing a very large target area for the payload. The task for the payload would then be to intersect this net and secure itself to the net. To minimize chances of the net damaging the payload, rather than intersecting the net, the payload might instead maneuver to come within a short distance of the net and shoot a tethered "harpoon" into the net. The payload would then ride the net for half a revolution of the tether. To release itself from the net, the payload would retract the barbs on its harpoon, thereby injecting itself into its transfer orbit.

GRASP Demo

Some of the methods for achieving the rendezvous between the payload and the rotating Tether Boost Facility could be demonstrated in a low-cost ground experiment that would utilize existing Automated Ren-

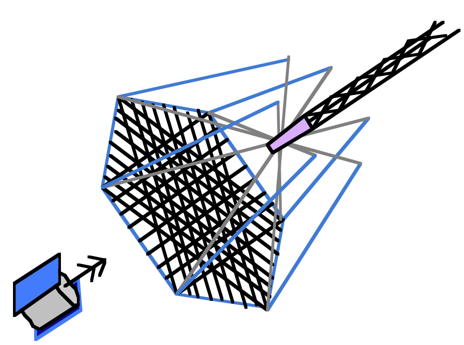

Figure 2. Sorensen's "Net and Grapple" concept for facilitating payload-tether rendezvous.

dezvous and Capture (AR&C) laboratory test facilities. Several organizations, including the Lawrence Livermore National Laboratory (LLNL) and NASA/ MSFC have AR&C air-table test facilities. Figure 3 shows the LLNL facility. This facility could be useful for tests such as a "Grapple Rendezvous and Secure Pickup" (GRASP) experiment to demonstrate that a payload could perform the required terminal rendezvous maneuvers and securely dock with a tethered grapple within the short (<10sec) time frames available in a momentum-exchange tether system.

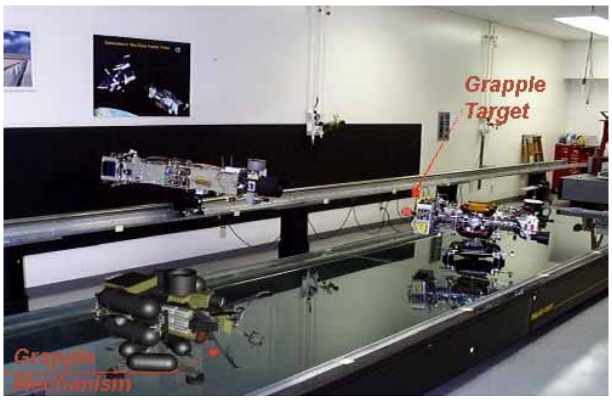

Figure 3. The LLNL microsatellite AR&C test facility, suitable for the GRASP experiment.

<u>High-Altitude Tether (HAT) GRASP</u>

The HAT-GRASP experiment, illustrated in Figure 4, would be a low-cost, real-world demonstration of the automated rendezvous and capture (AR&C) capabilities necessary for an operational Tether Boost Facility. A balloon would carry a tether deployer up to a high altitude. The deployer would drop a grapple mechanism down below the balloon at the end of a high-strength tether. A small sub-orbital rocket would then launch a small microsatellite-like rendezvous vehicle. The rocket would release the microsatellite, which would coast up to the balloon on a ballistic trajectory; the free-fall trajectory of the rendezvous vehicle would match the relative motion between a payload and the tip of a Tether Boost Facility. The microsat would then acquire and maneuver to rendezvous with the grapple. The dynamic disturbances induced by winds and other effects could give a good approximation for propagation/modeling errors in an orbital system. This test would also provide a test of tether dynamics experienced upon payload capture. If the microsatellite missed the grapple, it could deploy a parachute and return to Earth safely, and the experiment repeated at a later time.

<u>TORQUE™ Demonstration Experiment</u>

In order to begin demonstrating tether transportation techniques and retiring the risks associated with the momentum-exchange/electrodynamic-reboost tether technologies, we suggest the performance of a mission

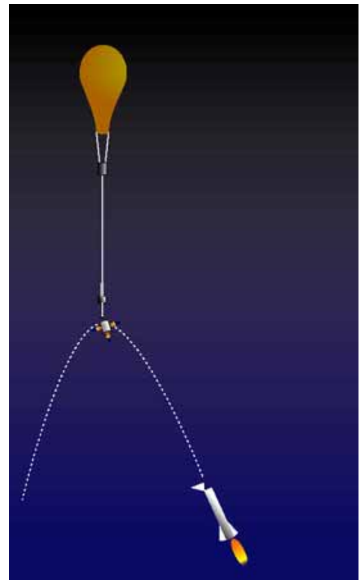

Figure 4. The HAT-GRASP Experiment Concept.

such as the "Tether Orbit-Raising Qualification Experiment" (TORQUE™). The objectives of the TORQUE™ mission would be to demonstrate:

- Rendezvous & Capture of a payload.
- Controlled electrodynamic spin-up of a tethered system.
- Controlled and accurate toss of a small payload. For example, the mission might toss a microsatellite to GTO or lunar transfer orbit.
- Controlled re-boost of the tether facility.

The TORQUE™ mission could be launched into low-LEO on a small expendable launch vehicle such as a Taurus. The TORQUE™ vehicle would first deploy its

Figure 5. Artist's concept of the TORQUE™ mission.

tether in a gravity-gradient stabilized orientation and use electrodynamic tether propulsion to both boost its orbit and torque the orbit down to the Earth's equator. The TORQUE™ vehicle would then use ED propulsion to spin-up the tether. A small launch vehicles such as a Pegasus would then be used to place a 150 kg microsatellite into a low circular LEO orbit. The microsatellite would maneuver to rendezvous with the spinning TORQUE™ tether, and the TORQUE™ system would catch and then toss the payload, injecting it into a GTO trajectory. The TORQUE™ tether system would have a total mass of approximately 1500 kg. It might be possible to design the mission hardware so that after the TORQUE™ experiment concludes its technology demonstration missions, it would then enter operational service, performing useful, revenue-generating operations such as sending service & refueling microsatellites to GTO as well as boosting lunar/interplanetary microsatellites into pre-escape trajectories.

First Operational System:
LEO⇒GTO/LTO Tether Boost Facility

Because the launch costs for deploying components of a Tether Transportation system will be a significant driver in the overall development costs, it will be imperative to the economic viability of the tether transportation architecture that every component placed into orbit be capable of generating revenue very soon after deployment. Although our ultimate goal is to develop a tether transport system capable of providing low-cost travel to the Moon and Mars, we have chosen to focus our initial development efforts on designing a Tether Boost Facility optimized for servicing traffic to geostationary orbit because lunar, Mars, and even LEO traffic volumes are currently speculative or highly uncertain, whereas GEO satellite deployment is a relatively well-understood and growing market.

The LEO⇒GTO Tether Boost Facility will boost payloads from low-LEO to geostationary transfer orbits (GTO). In sizing the facility design, we have sought to balance two somewhat competing drivers: first, the desire to be able to have a fully-operational, revenue-generating tether boost facility that can be deployed in a single launch on a rocket expected to be available in the 2010 timeframe, and second, the desire for the tether facility to be capable of gaining as large as possible a market share of the projected GEO traffic. Recent projections of GEO traffic, shown in Figure 6, indicate that the general trend for GEO payloads is to become more and more massive. Over the projected timeframe, payloads in the range of 4-6 metric tons are expected to account for roughly 80% of the commercial market. Consequently, it would be highly desirable to design the Tether Boost Facility to handle payloads on the order of 5,000 kg. On the other hand, a tether facility designed to toss payloads to GTO must mass roughly 9 times the mass of the payloads it can handle (due primarily to tether sizing, orbital mechanics, and conservation-of-momentum considerations). If the tether facility is to provide an operational capability after one launch, the tether facility must fit within the payload capacity of an available launch vehicle. In the 2010 timeframe, the largest payload-to-LEO anticipated is that of the Delta-IV-Heavy rocket, which will be able to place 20,500 kg into LEO.

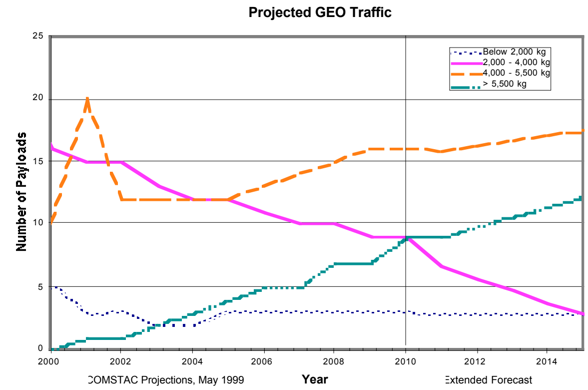

Figure 6. GEO traffic projections for 2000-20015.

Consequently, we have chosen to follow a modular development approach in which the initial Tether Boost Facility launched will be sized to fit on a Delta-IV-H. This facility will be capable of boosting 2,500 kg payloads to GTO as well as 1,000 kg payloads to lunar transfer orbit (LTO). This facility could potentially service approximately one-quarter of the ~400 payloads expected to be launched to GEO in the next 40 years. The facility hardware is designed in a modular fashion, so that after the initial facility has proven its capability and reliability, a second set of essentially identical hardware could be launched and combined with the first set to create a Tether Boost Facility capable of tossing 5,000 kg to GTO and 2,000 kg to LTO. Additional modules can increase the system capacity further.

To obtain a first-order estimate of the potential cost savings of the Tether Boost Facility, consider a mission to boost a 5 metric ton class payload into GTO. To do so using currently-available rocket launch systems would require a vehicle such as a Delta IVM+ (4,2), a Proton M, or a SeaLaunch Zenit 3SL. Depending

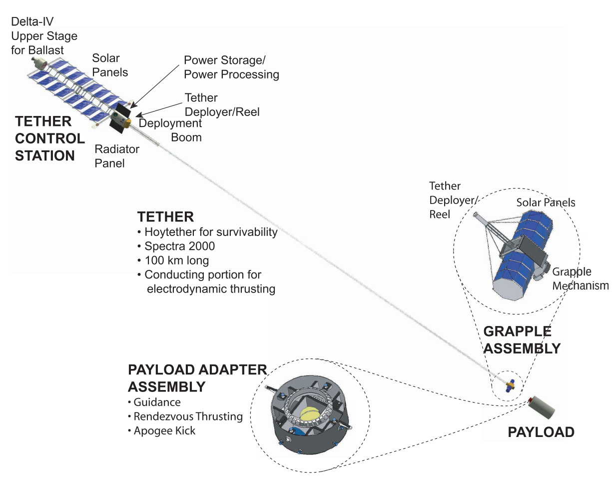

Figure 7. The LEO⇒GTO Tether Boost Facility

upon the launch service chosen and other business factors, current costs for this launch will be approximately $90M. If, however, a Tether Boost Facility is available that is capable of boosting the 5 metric ton payload from a LEO holding orbit to GTO, the customer could use a smaller launch vehicle, such as a Delta-II 7920, with an estimated launch cost of $45M, or a vehicle comparable to the Dnepr 1 (RS-20), with an estimated sticker price of $13M. While exact comparisons at this level are difficult due to differing payload capacities of each vehicle and the dependence of launch pricing upon other business factors, these estimates indicate that a reusable Tether Boost Facility could enable commercial and governmental customers to reduce their launch costs by 50% to 85%. Thus there is a significant opportunity for tether transportation systems to offer large cost savings in the LEO⇒GTO market. The key to the commercial viability of the tether facility, then, will be in designing the system architecture so that the operating costs and the cost of amortizing the investment in development and deployment are low enough that the LEO⇒GTO boost service can be offered at a price that will capture a large share of the market while sustaining the business.

The design of this LEO⇒GTO Tether Boost Facility is discussed in more detail in an accompanying paper.[12] The facility is designed to boost one 2,500 kg payload to GTO once every month. Although the facility design is optimized for boosting 2,500 kg payloads to GTO, it can also boost different-sized payloads to different orbits; the payload capacity depends upon the total ΔV to be given to the payload.

As a result, in addition to boosting payloads to GTO and LTO, this Tether Boost Facility could also serve as a component of a transportation architecture for delivering payloads to other orbits and other destinations. For example, the initial (2,500 kg to GTO) Facility could boost 5,000 kg payloads to the 20,335 km altitude used by the GPS system. As a component in the transportation system for Mars-bound payloads, the facility could be used to inject a 5,000 kg spacecraft into a highly elliptical equatorial orbit. At the apogee of this holding orbit, the payload could then perform a small ΔV maneuver to torque its orbit to the proper inclination for a Mars trajectory, then perform its Trans-Mars-Injection burn at perigee. The tether facility thus could reduce the ΔV requirements for a Mars mission by over 2 km/s.

Lunar/Mars Boost Facility

By adding more modular components to the LEO⇒GTO boost facility, we can build up its capacity to create a heavy-lift facility designed to boost 20-25 metric ton payloads to Lunar Transfer Orbits and to Mars transfer trajectories. This facility would provide a low-cost capability for transporting large quantities of cargo such as food, fuel, and construction supplies to

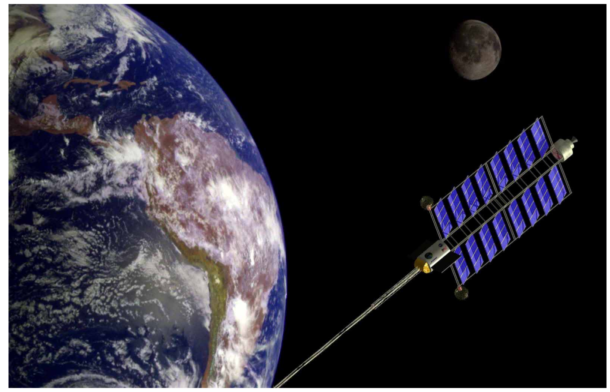

Figure 8. LEO⇒ Lunar/Mars Tether Boost Facility

facilitate the deployment of manned lunar and mars bases.

Cislunar Tether Transport System

This heavy-lift Boost Facility could then be used to deploy a second tether facility in polar lunar orbit. This facility, called a "Lunavator™" would be capable of catching payloads sent from Earth on minimum-energy transfer trajectories and delivering them to the surface of the Earth. The Lunavator facility could also be built incrementally. The first system would be sized to catch payloads from minimum-energy lunar transfers and drop them into low lunar orbit (LLO) or suborbital

Figure 9. Lunavator™ orbit before and after catching a payload sent from Earth.

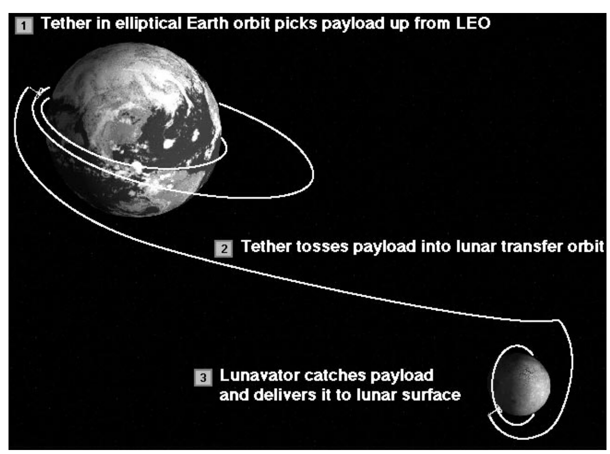

Figure 10. The Cislunar Tether Transport System.

trajectories, and to pick-up return payloads from LLO and send them down to LEO. The Lunavator mass could be built up using lunar resources, until it is capa ble of catching payloads sent from Earth and depositing them directly on the lunar surface, with zero velocity relative to the surface.

The deployment of a tether in lunar orbit would enable the tether system to begin servicing round-trip traffic, creating a "Cislunar Tether Transport System", illustrated in Figure 10, that could deliver payloads from LEO to the surface of the Moon with little or no propellant expenditure.[1]

Earth-To-Orbit Assist Tether System

In parallel with the development of the Cislunar Tether Transport System, a large, long tether facility could be built, again in an incremental fashion, to serve as a second stage in an Earth-to-Orbit Launch system. Currently, several designs exist for hypersonic airplanes and other reusable launch vehicles that can economically carry large payloads up to the upper atmosphere at speeds in the range of Mach 10-15. For example, the DF-9 hypersonic airplane designed by Boeing could carry 15 metric tons to 100 km at Mach 10, and a smaller variant of the Gryphon™ system proposed by Andrews Space Technology could deliver 15 metric tons to 150 km at Mach 15 (~5 km/s inertial on the equator). Their payload capabilities to orbital altitudes and velocities, however, are typically very small.

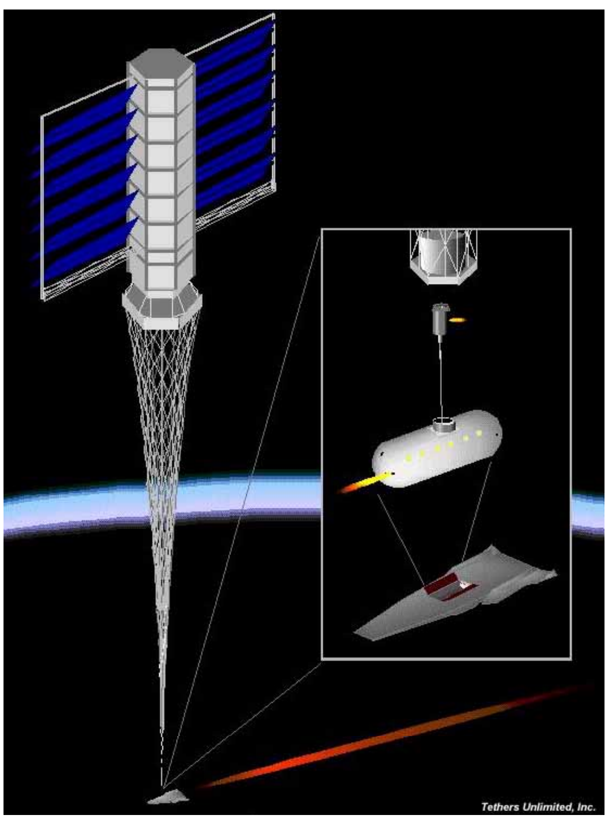

Figure 11. Illustration of the HASTOL architecture.

Rather than using rockets to get the payloads all the way into orbit, a large Tether Boost Facility could be used to pick the payloads up from the reusable launch and either pull them into orbit or toss them to escape. A joint effort by Tethers Unlimited, Inc. and the Boeing Company's Reusable Space Systems is investigating a concept for a "Hypersonic-Airplane Space Tether Orbital Launch (HASTOL)" system, illustrated in Figure 11.

Earth-Mars Round-Trip Architecture

Once a manned presence on Mars has been estab lished, an additional tether facility could be deployed in Mars orbit to catch payloads sent from Earth and toss return payloads back to Earth, as illustrated in Figure 12.

Summary

Momentum-Exchange/Electrodynamic-Reboost tether facilities can form the infrastructure for a fully-reusable low-cost in-space transportation architecture. Several technical challenges must be met to enable tether transport systems to be fielded, including development of rapid AR&C capabilities and techniques for building and controlling the tether facilities. A tether transport system to carry frequent traffic between Earth, the Moon, and Mars can be developed in a modular, incremental fashion, in which each component can generate revenue to fund the development of the rest of the system, much as the first railroads were developed. The initial Tether Boost Facility would be sized for launch on a single large rocket vehicle, and would be designed to immediately service traffic to GEO. The capacity of this facility could then be built incrementally, and additional tether facilities deployed to handle Earth-to-Orbit Assist, LEO⇔Lunar Surface round-trip travel, and deployment of manned Mars bases.

Acknowledgments

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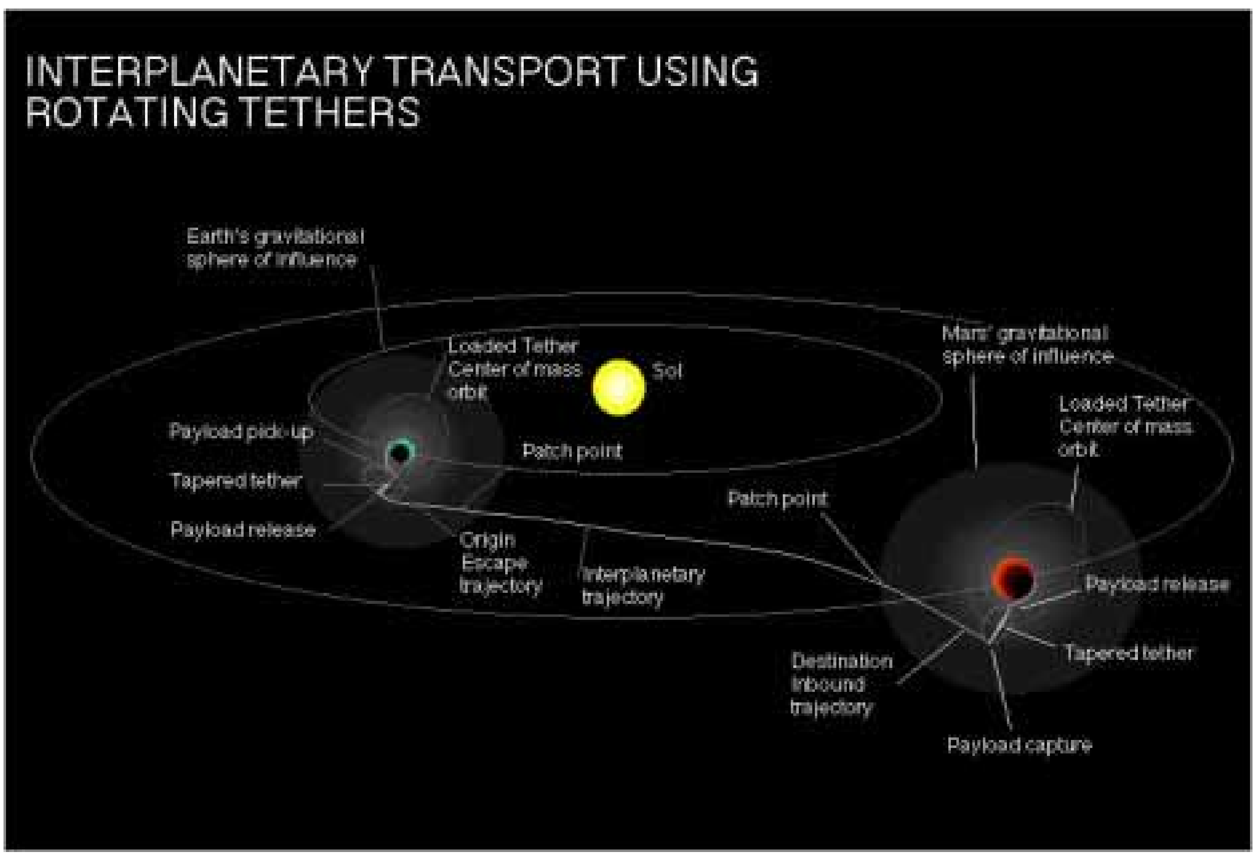
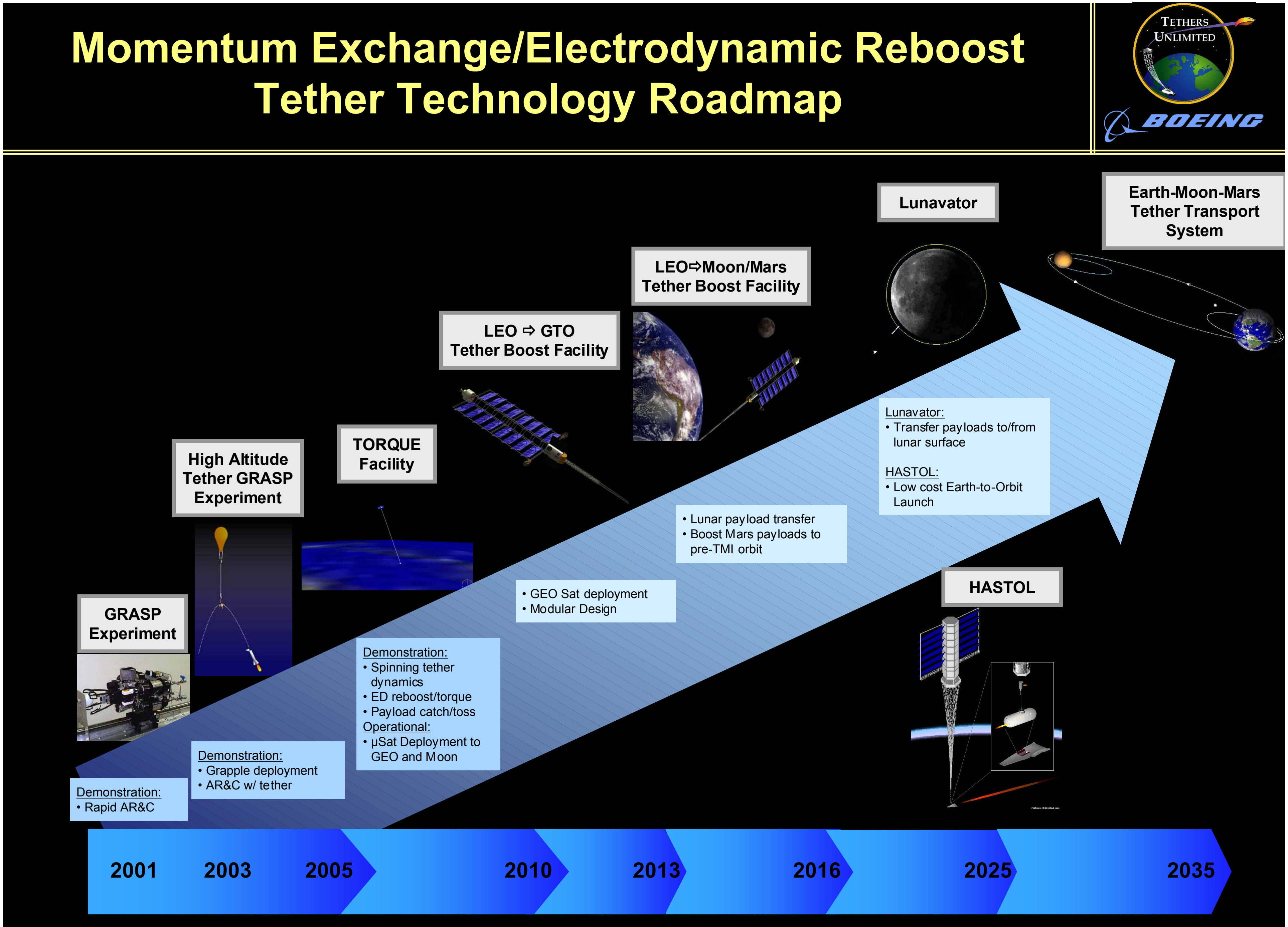


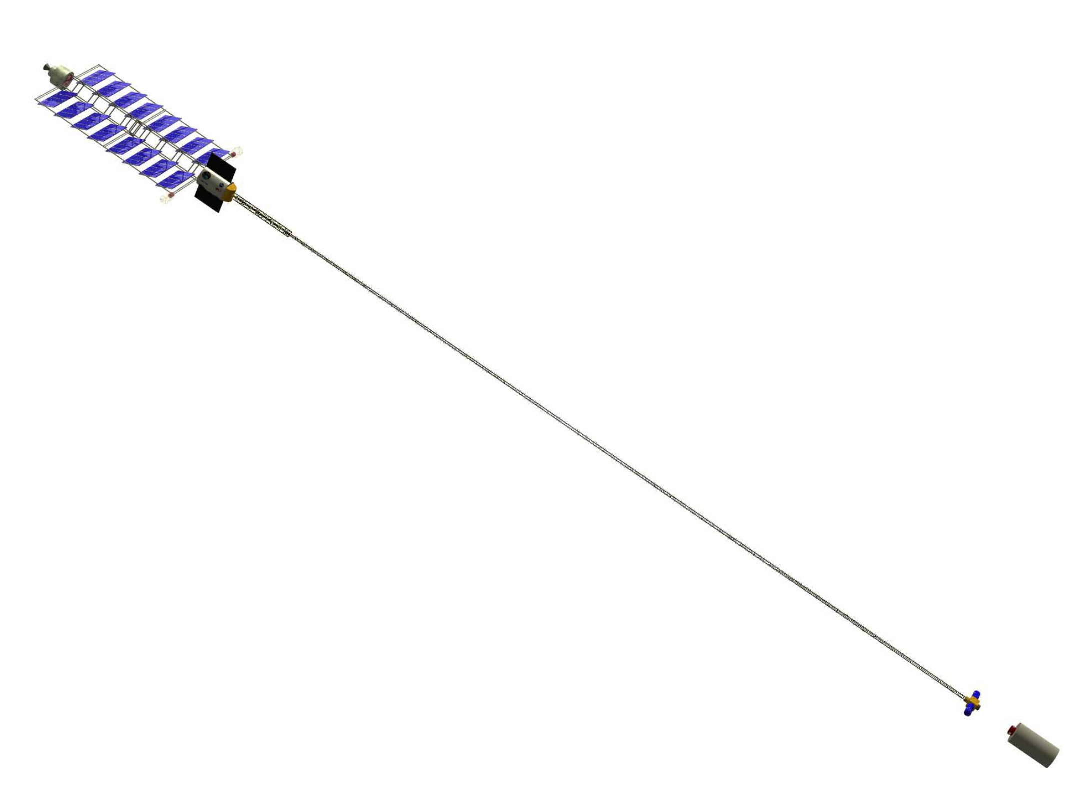
Figure 12. The MERITT architecture concept.


Momentum Exchange/Electrodynamic Reboost
Tether Technology Roadmap
TETHERS UNLIMITED
BOEING
GRASP Experiment
Demonstration:
• Rapid AR&C
High Altitude Tether GRASP Experiment
Demonstration:
• Grapple deployment
• AR&C w/ tether
TORQUE Facility
Demonstration:
• Spinning tether dynamics
• ED reboost/torque
• Payload catch/toss
Operational:
• μSat Deployment to GEO and Moon
LEO ⇨ GTO Tether Boost Facility
• GEO Sat deployment
• Modular Design
LEO⇨Moon/Mars Tether Boost Facility
• Lunar payload transfer
• Boost Mars payloads to pre-TMI orbit
Lunavator
Lunavator:
• Transfer payloads to/from lunar surface
HASTOL:
• Low cost Earth-to-Orbit Launch
HASTOL
Earth-Moon-Mars Tether Transport System
2001
2003
2005
2010
2013
2016
2025
2035

AIAA 2000-3866

DESIGN AND SIMULATION OF A TETHER BOOST FACILITY FOR LEO⇒GTO TRANSPORT

R. Hoyt
Tethers Unlimited, Inc., Seattle, WA



36th AIAA/ASME/SAE/ASEE
Joint Propulsion Conference & Exhibit
17-19 July 2000
Huntsville, AL

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DESIGN AND SIMULATION OF A TETHER BOOST FACILITY FOR LEO ⇒ GTO TRANSPORT

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www.tethers.com

Abstract

The LEO⇒GTO Tether Boost Facility will combine momentum-exchange tether techniques with electrodynamic tether propulsion to provide a reusable infrastructure capable of repeatedly boosting payloads from low Earth orbit to geostationary transfer orbit without requiring propellant expenditure. Designs for the orbital mechanics and system sizing of a tether facility capable of boosting 2,500 kg payloads from LEO to GTO once every 30 days are presented. The entire tether facility is sized to enable an operational capability to be deployed with a single Delta-IV-H launch. The system is designed in a modular fashion so that its capacity can be increased with additional launches. The tether facility can also boost 1000 kg payloads to lunar transfer orbits, and will serve as the first building block of an Earth-Moon-Mars Tether Transportation Architecture. The tether facility will utilize electrodynamic tether propulsion to restore its orbit after each payload boost operation. Using numerical modeling of tether dynamics, orbital mechanics, electrodynamics, and other relevant physics, we validate the orbital design of the system and investigate methods for performing electrodynamic reboost of the station.

Introduction

Under funding from NASA's Institute for Advanced Concepts (NIAC), Tethers Unlimited, Inc. and the Boeing Company are developing a modular architecture for a tether transportation system. This system will utilize momentum-exchange techniques and electrodynamic tether propulsion to transport multiple payloads *with little or no propellant consumption*. The tether transportation system will be built incrementally. The first component of the system will be a Tether Boost Facility that will transfer satellites and other payloads from low Earth orbit (LEO) to geostationary transfer orbit (GTO). This same facility will also be capable of boosting payloads to lunar transfer orbit (LTO). Later components will increase the payload capacity of the Tether Boost Facility and enable frequent round-trip travel to the surface of the Moon[1,2] and to Mars.[3] In this paper we present results of the development of a conceptual design for the first component of the tether transportation architecture, the LEO⇒GTO Tether Boost Facility, and discuss simulations used to investigate the operation of the tether system.

Background

Momentum-Exchange Tethers

In a momentum-exchange tether system, a long, thin, high-strength cable is deployed in orbit and set into rotation around a massive central body. If the tether facility is placed in an elliptical orbit and its rotation is timed so that the tether will be oriented vertically be low the central body and swinging backwards when the facility reaches perigee, then a grapple assembly located at the tether tip can rendezvous with and acquire a pay load moving in a lower orbit, as illustrated in Figure 1. Half a rotation later, the tether can release the payload, tossing it into a higher energy orbit. This concept is termed a momentum-exchange tether because when the

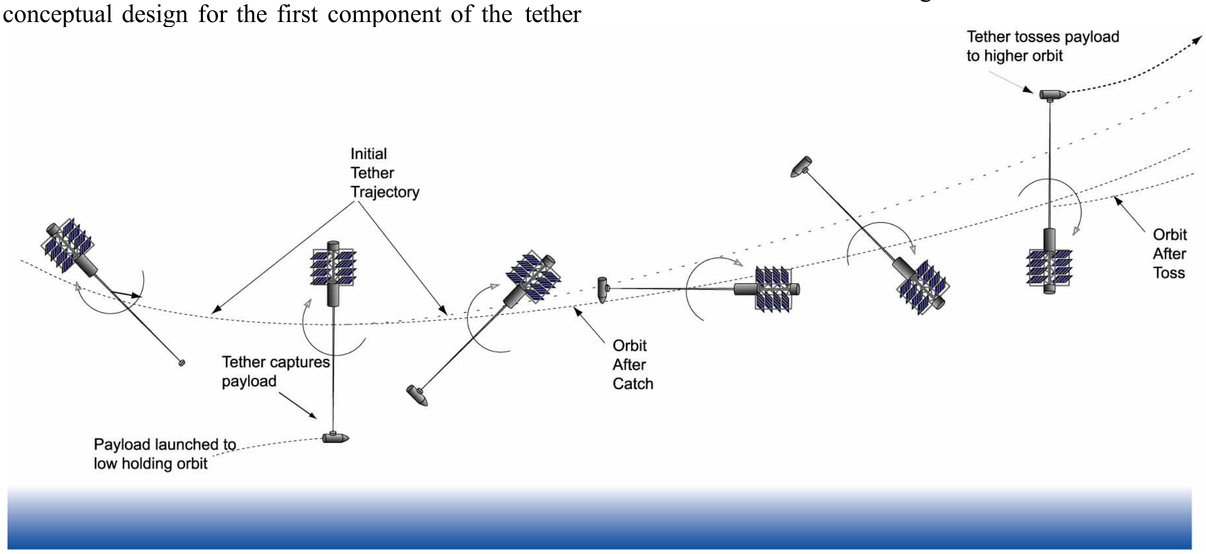

Figure 1. Momentum Exchange Tether catching and tossing payload.

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tether picks up and tosses the payload, it transfers some of its orbital energy and momentum to the payload. The tether facility's orbit can be restored later by re boosting with propellantless electrodynamic tether pro pulsion or with high specific impulse electric propul sion; alternatively, the tether's orbit can be restored by using it to de-boost return traffic payloads.

Prior Work

Several prior research efforts have investigated con ceptual designs for momentum-exchange tether sys tems. In 1991, Carroll proposed a tether transport facil ity that could pick payloads up from suborbital trajecto ries and provide them with a total ΔV of approximately 2.3 km/s.[4] Carroll's design, however, assumed that the tether would be placed in a circular LEO orbit. In order for this facility and tether to remain above the atmos phere after a payload boost operation, the central facility had to mass 50-100 times the payload mass. This large mass would require a very large launch cost to set up the tether facility, which would likely hinder the eco nomic viability of the concept.

In 1997, Hoyt[5] investigated a concept proposed earlier by Forward[6] for a tether system for transporting payloads from LEO to the surface of the Moon. This design used two tethers in Earth orbits to minimize the total tether mass required for the system. Hoyt proposed placing the tethers in elliptical orbits and performing all catch and toss operations at or near perigee. Doing so minimized the drop in the tether's perigee, enabling a tether facility to boost a payload and still stay above the atmosphere with facility masses as low as 5-10 times the payload mass.

In 1998, Bangham, Lorenzini, and Vestal developed a conceptual design for a two-tether system for boosting payloads from LEO to GEO.[7] The tether transport system was proposed to stage the ΔV operations using two tether facilities in elliptical orbits so as to minimize the required tether mass. Their design proposed the use of high specific impulse electric thrusters to restore the orbit of the tether facilities after each payload boost operation. Even with the propellant mass requirements for reboost, they found that this system could be highly economically advantageous compared chemical rockets for GEO satellite deployment.

In a Phase I NIAC effort in 1999, Hoyt and Uphoff studied the orbital mechanics of multi-tether systems for transporting payloads between LEO and the surface of the Moon and found that orbital perturbations caused by Earth oblateness and other effects would make scheduling transfers in a staged system difficult or impossible.[1] Consequently, they concluded that tether systems for transporting payloads from LEO to GTO or LTO should use one tether facility in Earth orbit to provide all of the ΔV. Further study revealed that although a single-tether system requires a much larger total tether mass than a staged two-tether system, the total system mass for a one-tether system, including the mass required for the control station and grapple assemblies, is the same or less than a multi-tether system.[2]

LEO⇒GTO Tether Boost Facility Design

Design for Incremental Development

The ultimate goal of this research effort is to develop a fully reusable in-space transportation infrastructure capable of providing frequent rapid round-trip transport between Earth, the Moon, and Mars. The technical development of such a transportation architecture must, however, follow a path that is commensurate with a viable business plan, in which early components can serve useful functions to generate revenue to fund the development of the rest of the system. Accordingly, as the first step in the deployment of this architecture, this effort has designed an initial tether transportation capability that will provide a cost-competitive transportation service for a significant and well-understood market, namely that of delivering payloads to GEO. The first component deployed will generate revenue by boosting commercial satellites and other payloads to GTO, as well as sending small payloads to the Moon. This revenue will be invested in the deployment of additional modules to increase the system capacity enable large payloads to be sent to either GEO or the Moon. Later, similar tether facilities will be deployed in orbit around the Moon and Mars, enabling round-trip transport between LEO, the lunar surface, and Mars orbit with zero transfer propellant requirements.

System Requirements

Payload Mass:

The mission of the LEO⇒GTO Tether Boost Facil ity will be to pick 2,500 kg payloads up from low-LEO orbits and inject them into transfer orbits to GEO alti tudes. To do so, the Tether Boost Facility will provide the payload with a total ΔV of 2.4 km/s.

Expandability:

The 2,500 kg payload size was chosen primarily so that a fully operational tether facility can be launched on a single large launch vehicle. The likely "sweet spot" for the GTO market in 2010, however, is ex pected to be closer to 5,000 kg. Consequently, this effort has sought to design the Tether Boost Facility to be expandable so that a second launch of nearly identi cal equipment will enable it to handle larger payloads and larger ΔV's.

Payload Design Impacts:

The Tether Boost Station architecture must minimize the design impacts upon payloads. Consequently, the system is designed to expose the payload to dynamic loads that are no larger than those it would experience in a conventional launch vehicle such as an Ariane or Delta rocket. In order to enable the payload to be boosted by the tether facility, a payload accommodation adapter (PAA) will be fitted to the payload's standard mounting fixtures. The PAA will provide the rendezvous maneuvering and docking capabilities to the pay load, and may also provide the apogee kick ΔV.

Safety Factor:
To provide ample margin for error and degradation of the tether over time, the tether structure is sized to pro vide a safety factor of 2 for the largest loads expected in the system. The largest loads will be due to transient oscillations immediately after the payload capture. These loads are predicted using numerical modeling with TetherSim™. Computed with respect to the nominal loads, the safety factor is roughly 3.5.

Throughput:
Because one of the primary advantages of momen tum-exchange tethers is their reusability, to maximize the cost-competitiveness of the system it will be de signed to boost payloads as frequently as once every 30 days.

Momentum-Exchange/Electrodynamic-Reboost Facility Concept
In order for the tether facility to boost one payload per month, the tether must restore its orbital energy after each payload boost operation. Previous efforts have proposed using ion thrusters or other electric pro pulsion to accomplish this reboost;[4,7] electric thrusters, however, require propellant expenditure and thus would incur launch mass costs and resupply operations costs which would limit the competitiveness of the tether system.

If the tether facility operates at least partly within LEO, it can instead utilize electrodynamic tether propulsion to perform reboost of its orbit. This concept, called the "High-strength Electrodynamic Force Tether" (HEFT) Facility (also referred to as a "Momentum-Exchange/Electrodynamic-Reboost (MXER) Tether Facility),[8] is illustrated in Figure 2. The Tether Boost Facility will include a control station housing a power supply, ballast mass, plasma contactor, and tether deployer, which would extend a long, tapered, high-strength tether. A small grapple vehicle would reside at the tip of the tether to facilitate rendezvous and capture of the payloads. The tether would include a conducting core, and a second plasma contactor would be placed near the tether tip. By using the power supply to drive current along the tether, the HEFT Facility could generate electrodynamic forces on the tether. By properly varying the direction of the current as the tether rotates and orbits the Earth, the facility can use these electrodynamic forces to generate either a net torque on the system to change its rotation rate, or a net thrust on the system to boost its orbit. The HEFT Facility thus could repeatedly boost payloads from LEO to GTO, and in between each payload boost operation it would

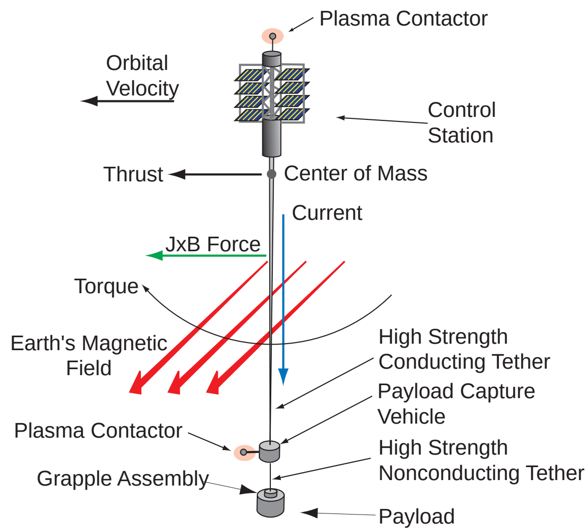

Figure 2. Schematic of the HEFT Facility concept.

use propellantless electrodynamic propulsion to restore its orbital energy.

Orbital Design
To boost a payload from LEO to GTO, the tether facility performs a catch and release maneuver to provide the payload with two ΔV impulses of approximately 1.2 km/s each. To enable the tether to perform two "separate" ΔV operations on the payload, the facility is placed into a highly elliptical orbit with its perigee in LEO. The tether facility's initial orbit is chosen so that when the tether is near perigee, its center of mass is moving approximately 1.2 km/s faster than the payload in circular LEO. It can then catch the payload, hold it for half a rotation, and then release it at the top of the tether's rotation. This injects the payload into the high-energy transfer trajectory.

Table 1 shows the orbital design for the LEO⇒GTO Tether Boost Facility. To minimize the mass of the tether, it is tapered along its length to maintain a constant load level; Figure 3 illustrates this tapering.

The orbital parameters and system masses shown in Table 1 are chosen so that the payload's orbit and the facility's initial orbit are harmonic. For this design the resonance is 41:20. This enables the tether facility to have multiple opportunities to capture the payload. If the payload and tether do not succeed in achieving docking during the first rendezvous attempt, they will wait for 2.6 days, adjusting the tether spin and correcting any trajectory errors, and then a second rendezvous will be possible without any significant maneuvering. The resonance design shown in Table 1 accounts for regressions of both orbits due to the Earth's non-ideal gravitational potential, up to the J4 term.

Table 1. System Orbital Design for LEO⇒GTO Boost

System Masses		
Tether mass	8,274	kg
CS Active Mass	11,514	kg
CS Ballast Mass	3490	kg
Grapple mass	650	kg
Total Facility Mass	**23,928**	kg
Total Launch Mass	**20,438**	kg
Payload Mass	**2,500**	kg

Tether Characteristics		
Tether Length	100,000	m
Tether mass ratio	3.31	
Tether tip velocity at catch	1,267	m/s
Tether tip velocity at toss	1,147	m/s
Tether angular rate	0.015514	rad/s
Gravity at Control Station	0.64	g
Gravity at payload	1.81	g
Rendezvous acceleration	2.00	g

		Pre-Catch		Joined System	Post-Toss	
Positions & Velocities		**Payload**	**Tether**	**Post-catch**	**Tether**	**Payload**
resonance ratio		41	20		1	4.1
perigee altitude	km	325	407	399	391	473
apogee altitude	km	325	8445	7199	6105	35786
perigee radius	km	6703	6785	6777	6769	6851
apogee radius	km	6703	14823	13578	12483	42164
perigee velocity	m/s	7711	8978	8858	8738	10005
apogee velocity	m/s	7711	4109	4421	4739	1626
CM dist. From Station	m		18356	26080	18356	
CM dist. To Grapple	m		81644	73920	81644	
ΔV to Reboost	m/s				240	
ΔV to Correct Apogee	m/s					0
ΔV to Correct Precess.	m/s					0
ΔV To Circularize	m/s					1449
Basic Orbital Parameters						
semi-major axis	km	6703	10804	10177	9626	24508
eccentricity		0.0	0.372	0.334	0.297	0.720
inclination	rad	0	0	0	0	0
semi-latus rectum	km	6703	9309	9041	8778	11787
sp. mech. energy	m2/s2	-2.97E+07	-1.84E+07	-1.96E+07	-2.07E+07	-8.13E+06
vis-viva energy	m2/s2	-5.95E+07	-3.69E+07	-3.92E+07	-4.14E+07	-1.63E+07
period	sec	5462	11176	10218	9399	38183
period	min	91.0	186.3	170.3	156.7	636.4
station rotation period	sec		405.0	405.0	405.0	
rotation ratio			27.6	25.2	23.2	

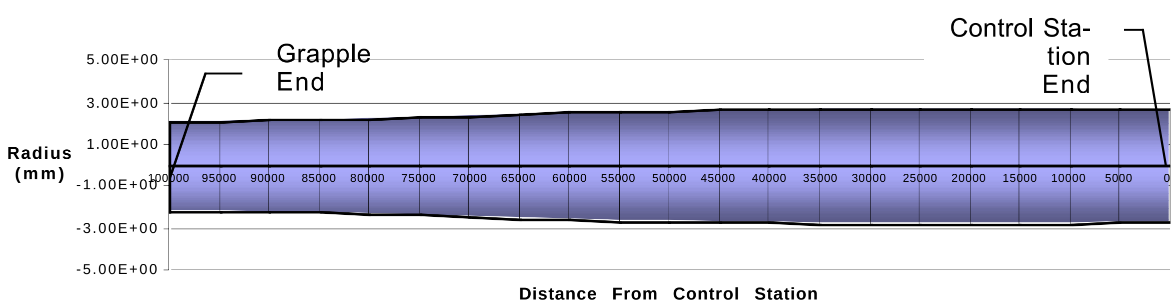

Figure 3. Taper of the tether cross-section (tether will actually be composed of multiple smaller lines).

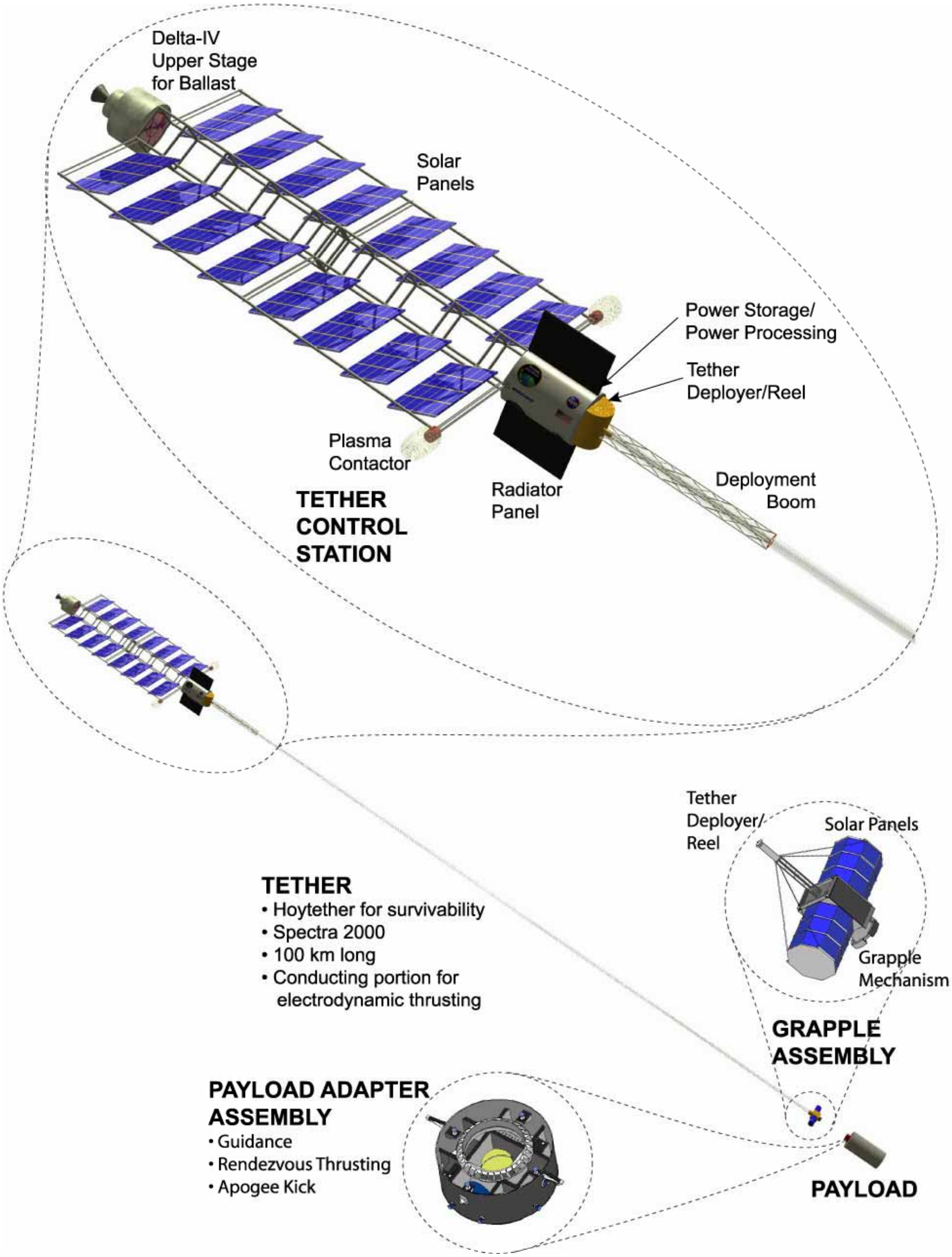

Figure 4. System Design for a Tether Boost Facility.

Figure 5. Tether Boost Facility with two modules, capable of tossing 5000 kg to GTO and 2000 kg to LTO. (Tether length not to scale)

System Design

Figure 4 illustrates the system concept design for the Tether Boost Facility. The Tether Boost Facility is composed of a Control Station, a tapered high-strength tether, and a Grapple Assembly. In addition, a Payload Accommodation Assembly (PAA) will be attached to the payload to provide maneuvering and guidance for rendezvous. For LEO⇒GTO traffic, this PAA will be an expendable unit incurring recurring costs.

To meet the requirement for operational capability with a single launch, the tether facility is sized to be deployed with a single launch of a Delta-IV-H or comparable vehicle. As Figure 4 shows, the 3490 kg Delta upper stage will be retained for use as ballast mass. The control station includes an array of solar panels which swivel to track the sun as the tether facility rotates. In this design, we have chosen to place the control station at the end of the tether, rather than at the center of mass of the facility. This choice was made for several reasons: because it minimizes the dynamical complexity, because it requires only one tether deployer, and because the center of mass of the system shifts when the payload is captured and released.

Electrodynamic Tether:

The tether in this system is composed of Spectra 2000® fibers braided into the Hoytether™ structure.[9] The nominal length of the tether is 100 km. Along the 80 km of the tether closest to the Control Station, a total of 500 kg of insulated aluminum wire is woven into the structure, providing a current path for electrodynamic thrusting.

Power System Sizing:

In order for the tether facility to reboost its orbit within 30 days, the facility will require a solar power generation capability of 100 kW. Because the facility will pass through the radiation belts frequently, its solar power system will utilize a concentrator-type solar panel design, such as the Scarlet design, with 150 mil Aluminum backside and 100 mil glass cover slides to shield the arrays from the belt particles. In order for the solar array to produce the desired power levels after 10 years of operation, they system will be deployed with 137 kW of initial power generation capability. Using Scarlet-type panel technology, this solar array would mass approximately 1,370 kg. The tether facility will collect this solar power during the roughly 80% of its orbit that it is in the sunlight, and store it in a battery system. Then, during perigee pass, it will drive the electrodynamic tether at an average power level of 300 kW (modulated as to be described later). In order to provide a maximum battery depth-of-discharge of 30%, the control station will have a battery system with 5,700 A•hr of capacity (120 V power system). Using advanced Li ion batteries, this will require approximately 4,600 kg of batteries. The control system will also require the capability to transform the 120 V battery voltage up to the 20+kV needed to drive tether currents on the order of 15 A.

Payload Capacity vs. Tip Velocity

The boost facility described herein is optimized for tossing 2.5 metric ton payloads to GTO. The same facility, however, can also service traffic to other orbits by changing its rotation rate and initial orbit. Because the stress in the tether increases exponentially with the rotation rate, the payload capacity drops as the tip velocity increases. Figure 6 shows the payload mass capacity versus the total ΔV that the tether facility could impart to the payload in a catch-toss operation. The boost facility could toss 1000 kg into a minimal-energy lunar transfer orbit, or toss 500 kg into an escape trajectory.

System Modularity

The Tether Boost Facility concept has been designed to enable it to be grown incrementally. After the initial facility, capable of tossing 2,500 kg to GTO and 1000 kg to LTO, has been deployed and tested, a second module of nearly identical hardware can be launched and combined in a parallel fashion with the first module, as illustrated in Figure 5. This will increase the system's capacity to 1,000 kg to LTO and 5,000 kg to GTO. The parallel construction will provide redundancy to the system, reducing the need for redundancy within each module. Cross-linking between the two parallel tethers could be added to increase their redundancy. Additional modules can be launched to increase the system capacity further.

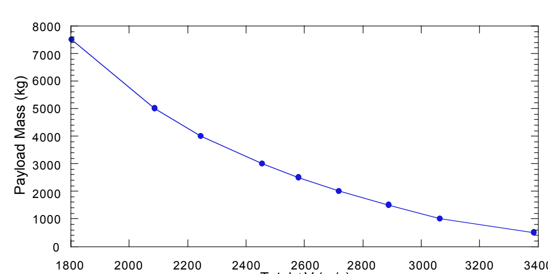

Figure 6. Payload capacity for the facility design given in Table 1 at different tip velocities.

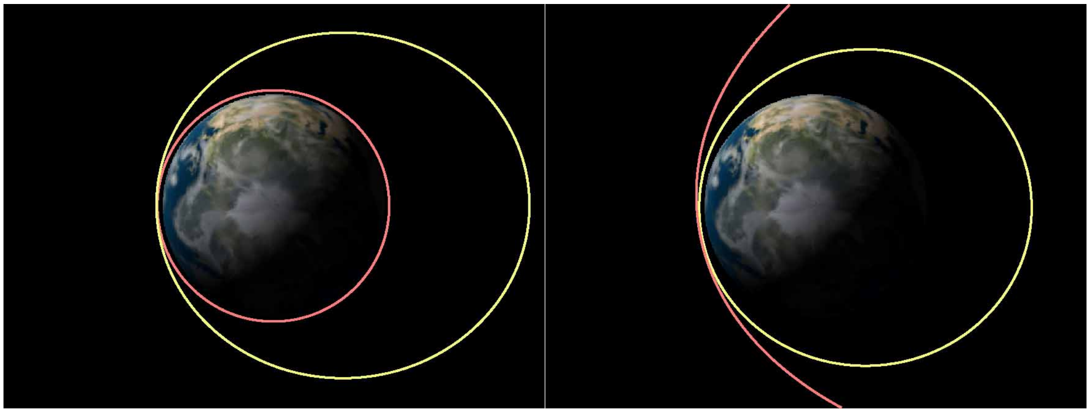

Figure 7. LEFT: Tether Boost Facility initial orbit (yellow ellipse) and payload initial orbit (red circle). RIGHT: Tether Facility orbit after payload boost (inner yellow ellipse) and Payload GTO (red outer ellipse).

Simulation of Electrodynamic Reboost

As the Tether Boost Facility catches and tosses a payload into GTO, its orbit drops, as illustrated in Figure 7. The apogee drops 2340 km, and the perigee drops 16 km. To restore the orbit, the tether system must increase the facility's orbital energy by 54 GJ, and it will do so by performing electrodynamic thrusting while the tether is within the dense portion of the ionosphere near the perigee of its orbit. Because the tether is rotating, the direction of the current must be alternated as the tether rotates to produce a net thrust on the facility. Using the TetherSim™ program, we have modeled reboost of a rotating tether system to investigate the efficiency of the reboost, and to develop methods for controlling the electrodynamic thrust to achieve the desired final orbit.

Method:

To study the performance of electrodynamic reboost of the tether facility, TetherSim™ was used to simulate reboosting of the orbit of the Tether Boost Facility described in Table 1 over a period of two days. TetherSim™ is a numerical simulation tool that includes models for tether dynamics, orbital mechanics, electrodynamics, thermal behavior, geopotential, geomagnetic field, ionospheric density variations, neutral gas density variations, and other relevant physics.

In the simulations, thrusting was performed when the tether facility's altitude was under 2000 km. The electrodynamic tether system had hollow-cathode plasma contactors at both ends of the conducting tether, so that it could carry current in both directions. The thrusting was performed at a maximum power of 450 kW. The Control Station contained a 150 kW solar power supply, a 8500 A•hr (120 V) battery system. Peak tether current levels were limited to 20 A, with typical currents varying between 15 and 20 A. In addition, thrusting was performed only when the tether was within $\pi/4$ of vertical.

Results

Reboost Simulations

Figure 8 shows the orbit semimajor axis, and Figure 9 shows the orbit eccentricity during the two days of boosting simulated. The semimajor axis increases at 52 km/day. Note that if the electrodynamic boost system adds energy to the orbit at a constant rate, the rate of semimajor axis increase will accelerate due to the inverse relation between orbital energy and semimajor axis. The eccentricity increases at 0.0034/day. Note that the eccentricity change rate will also vary during reboost. Figure 10 shows the apogee altitude increase.

Thrust Efficiency:

The thrust efficiency is shown in Figure 12. The graph shows that the thrust efficiency varies cyclically during each day; this variation is due to the fact that the Earth, and its magnetic field, are rotating inside the facility's orbit, and thus the angle between the geomagnetic field's axis and the orbit plane varies once per day. In addition, not readily apparent on this timescale, the thrust efficiency varies with altitude and with the angle of the tether relative to local vertical. Over this one day period, the average thrust efficiency is 40 μN/W (thrust efficiency calculated using the power input to the electrodynamic tether).

Reboost Time:

Since the rate of semimajor axis increase varies during the reboost operation, the best way to estimate the time needed to reboost the orbit is to assume that the rate at which the orbital energy of the system is increased is relatively constant during the reboost period. To reboost the orbit from 391x6105 km to 407x8445 km, the electrodynamic system must restore 54 GJ of energy to the tether facility's orbit. In the 2-day simulation, the electrodynamic thrusting restored the facility's orbital energy at a rate of 2.7 GJ/day, as illustrated in Figure 11.

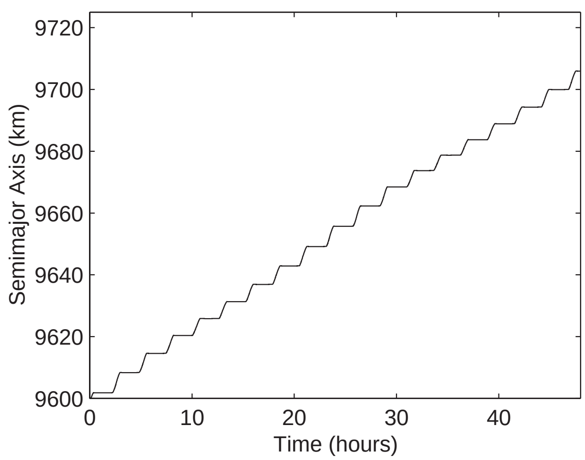

Figure 8. Semimajor axis during the first two days of the reboost operation.

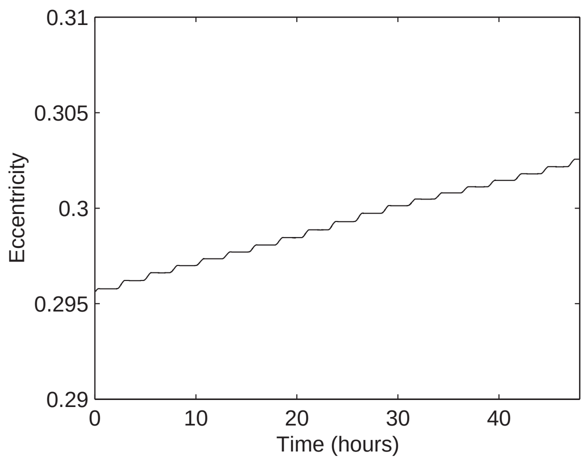

Figure 9. Orbit eccentricity.

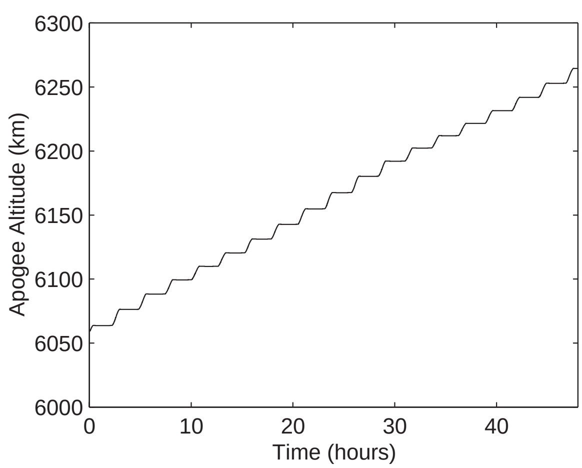

Figure 10. Apogee altitude

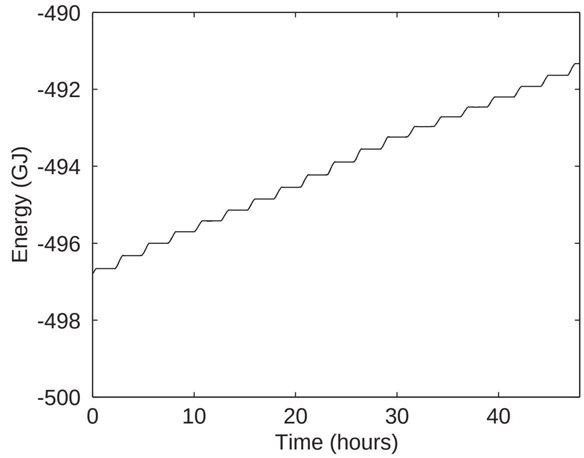

Figure 11. Orbital Energy.

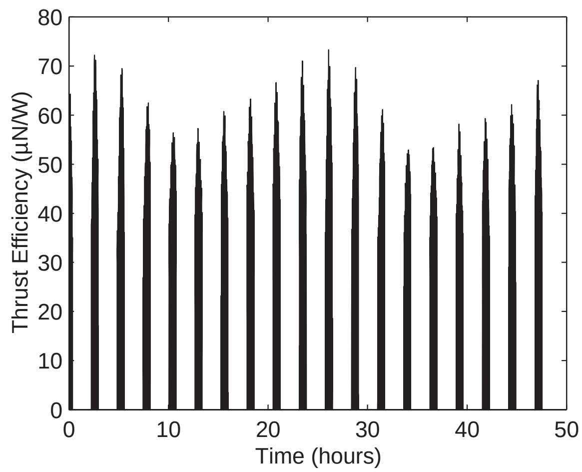

Figure 12. Thrust efficiency.

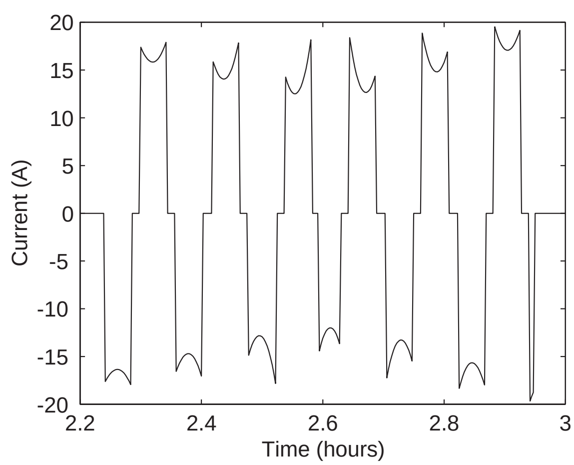

Figure 13. Current driven through the electrodynamic tether during a perigee pass.

Energy System:

The tether current during one of the perigee passes is shown in Figure 13. The charge level of the energy storage system (batteries or flywheels) over the two days is shown in Figure 14. With the solar power sup ply generating 150 kW during the portions of the orbit that the tether facility is illuminated, and processed through the batteries at an efficiency of 88%, the sys tem maintains its energy balance and the depth of charge does not exceed 20%.

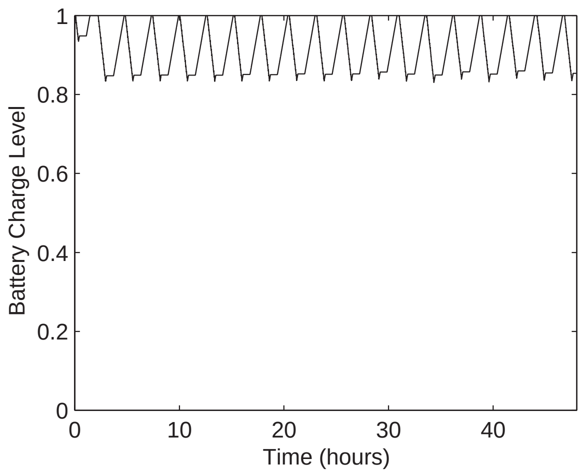

Figure 14. Battery charge level.

Analysis:

The simulated system, which had 150 kW of solar panel power and thrusted at 450 kW during perigee passes, would reboost the orbit energy within approxi mately 20 days. To achieve the 30 day reboost desired for the LEO⇒GTO Tether Boost Facility, we thus need a lower solar panel power of approximately 100 kW. Thrusting would be performed at 300 kW during peri gee passes, and tether current levels would be roughly 15 A.

Summary

We have presented an orbital design and system-concept level definition for a tether facility capable of boosting 2,500 kg payloads from LEO to GTO once every 30 days. The entire tether facility is sized to enable an operational capability to be deployed with a single Delta-IV-H launch. The system is designed in a modular fashion so that its capacity can be increased with additional launches. The tether facility can also boost 1000 kg payloads to lunar transfer orbits, and will serve as the first building block of an Earth-Moon-Mars Tether Transportation Architecture. The tether facility will utilize electrodynamic tether propulsion to restore its orbit after each payload boost operation. Using numerical modeling we have validated the orbital design of the system and investigated methods for performing electrodynamic reboost of the station.

Acknowledgments

This research was supported by Phase I and Phase II grants from NASA's Institute for Advanced Concepts. The author wishes to acknowledge important contributions by Robert Forward of TUI and Michal Bangham, John Grant, Brian Tillotson, Beth Fleming, John Blumer, Ben Donahue, Bill Klus, and Harvey Willenberg of the Boeing Company, as well as valuable discussions with Kirk Sorensen of NASA/MSFC.

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THE CISLUNAR TETHER TRANSPORT SYSTEM ARCHITECTURE

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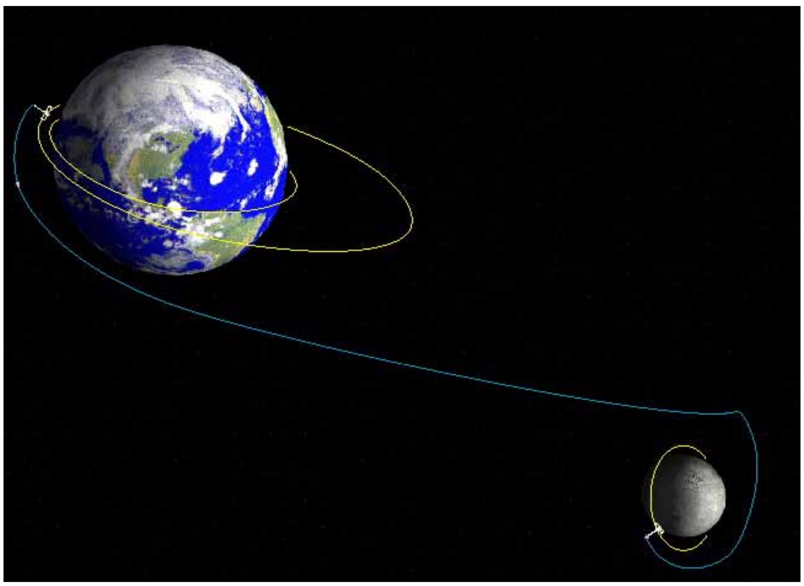

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THE CISLUNAR TETHER TRANSPORT SYSTEM ARCHITECTURE

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Abstract

We describe a space systems architecture for repeatedly transporting payloads between low Earth orbit and the surface of the moon without significant use of propellant. This architecture consists of one rotating momentum-exchange tether in elliptical, equatorial Earth orbit and a second rotating momentum-exchange tether in a circular low lunar orbit. The Earth-orbit tether picks up a payload from a circular low Earth orbit and tosses it into a minimal-energy lunar transfer orbit. When the payload arrives at the Moon, the lunar tether catches it and deposits it on the surface of the Moon. Simultaneously, the lunar tether picks up a lunar payload to be sent down to the Earth orbit tether. By transporting equal masses to and from the Moon, the orbital energy and momentum of the system can be conserved, eliminating the need for transfer propellant. The Earth-orbit tether can also be used to send payloads to the Moon without return traffic if electrodynamic tether propulsion is used to restore its orbit in between payload boost operations. Using currently available high-strength tether materials, this system can be built with a total mass of less than 37 times the mass of the payloads it can transport. Using numerical simulations that incorporate the full three-dimensional orbital mechanics and tether dynamics, we have verified the feasibility of this system architecture and developed scenarios for transferring a payload from a low Earth orbit to the surface of the Moon that require less than 25 m/s of thrust for trajectory targeting corrections.

Introduction

Under funding from NASA's Institute for Advanced Concepts, Tethers Unlimited, Inc. has investigated the feasibility of using momentum-exchange tether techniques and electrodynamic tether propulsion to create a modular architecture for transporting payloads from low Earth orbit (LEO) to the surface of the Moon, and back, *with little or no propellant consumption*.[1,2] A "Cislunar Tether Transport System" would be composed of one rotating momentum exchange/electrodynamic reboost tether in elliptical, equatorial Earth orbit and a momentum-exchange rotating tether facility in a low circular polar lunar orbit. This architecture can repeatedly exchanging payloads between LEO and the surface of the Moon, with the only propellant requirements being for trajectory corrections and rendezvous maneuvering.

In 1991, Forward[3] showed that such a system is theoretically possible from an energetics standpoint. A later study by Hoyt and Forward[4] developed a first-order design for such a system. These previous studies, however, utilized a number of simplifying assumptions regarding orbital and tether mechanics in the Earth-Moon system, including assumptions of coplanar orbits, ideal gravitational potentials, and infinite facility ballast masses. In this paper, we summarize work done to develop an architecture for such a system that takes into account the full complexities of orbital mechanics in the Earth-Moon system. We then present a system concept for a Tether Boost Facility designed to boost 1000 kg payloads to the Moon.

The basic concept of the Cislunar Tether Transport System is to use a rotating tether in Earth orbit to pick payloads up from LEO orbits and toss them to the Moon, where a rotating tether in lunar orbit, called a "Lunavator™", could catch them and deliver them to the lunar surface. As the Lunavator™ delivers payloads to the Moon's surface, it can also pick up return payloads, such as water or aluminum processed from lunar resources, and send them down to LEO. By balancing the flow of mass to and from the Moon, the orbital momentum and energy of the system can be conserved, eliminating the need to expend large quantities of propellant to move the payloads back and forth. This system is illustrated in Figure 1.

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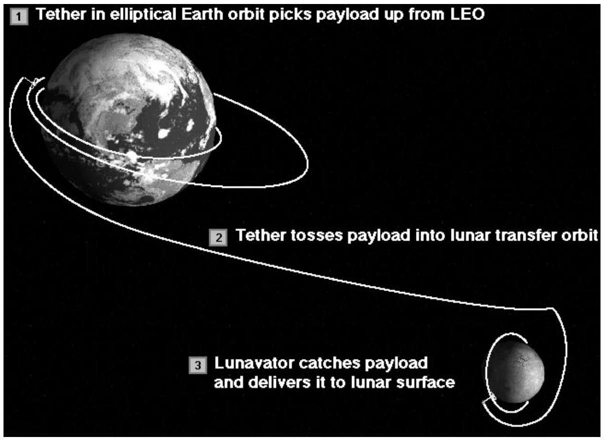

Figure 1. Conceptual illustration of the Cislunar Tether Transport System.

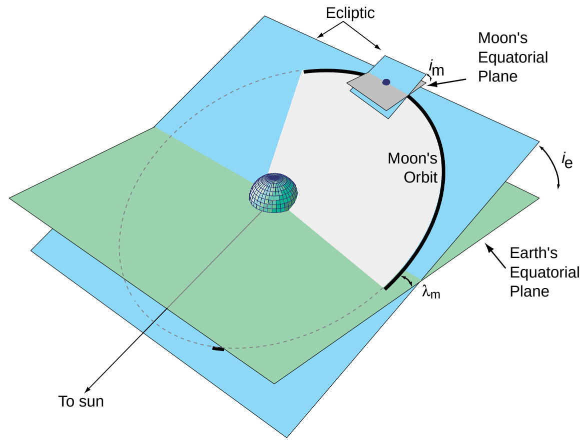

Figure 2. Schematic illustrating the geometry of the Earth-Moon system.

Orbital Mechanics of the Earth-Moon System

Orbital mechanics in cislunar space are made quite complex by the different and varying orientations of the ecliptic plane, the Earth's equatorial plane, the Moon's orbital plane, and the Moon's equatorial plane. Figure 2 attempts to illustrate these different planes. The inclination of the Earth's equatorial plane (the "obliquity of the ecliptic"), is approximately 23.45°, but varies due to tidal forces exerted by the Sun and Moon. The angle i_m between the Moon's equatorial plane and a plane through the Moon's center that is parallel to the ecliptic plane is constant, about 1.58°. The inclination of the Moon's orbit relative to the ecliptic plane is also constant, about λ_m = 5.15°.[5] The line of nodes of the Moon's orbit regresses slowly, revolving once every 18.6 years. As a result, the inclination of the Moon's orbit relative to the Earth's equator varies between 18.3-28.6 degrees. The Moon's orbit also has a slight eccentricity, approximately e_m = 0.0549.

Tether Orbits

After considering many different options, including the three-tether systems proposed previously and various combinations of elliptical and circular orbits, we have determined that the optimum configuration for the Cislunar Tether system is to utilize one tether in an elliptical, equatorial Earth orbit and one tether in a circular, polar lunar orbit, as illustrated in Figure 1. This two-tether system will require the lowest total system mass, minimize the system complexity and provide the most frequent transfer opportunities. The Earth-orbit tether will pick payloads up from equatorial low-LEO orbits and toss them towards one of the two points where the Moon crosses the Earth's equatorial plane. The toss is timed so that the payload reaches its apogee ahead of the Moon. The Moon approaches the payload from behind, and its gravity causes the payloads velocity to slow and then reverse, pulling it into a hyperbolic polar lunar trajectory. As the payload approaches the Moon, it will need to perform a small ΔV maneuver to set it up into the proper approach trajectory; the size of this maneuver will vary depending upon the inclination of the Moon's orbit plane and launch dispersions, but under most conditions it will only require about 25 m/s of ΔV.

In the following sections, we will first develop a design for a tether facility for boosting payloads from low-LEO orbits to lunar transfer orbits (LTO). We will then develop a design for a Lunavator™ capable of catching the payloads and delivering them to the surface of the Moon. We will then discuss the numerical simulations used to verify the feasibility of this system architecture.

Design for Incremental Development

This effort has sought to design the Cislunar Tether Transport System so that it can be developed and deployed in an incremental, modular fashion. The first components deployed will generate revenue by transporting materials to the Moon to facilitate lunar base development, and this revenue will be invested in the deployment of additional modules to increase the system capacity and eventually enable round trip transport between LEO and the lunar surface.

Although the system will realize its full potential when it is capable of transporting payloads both to and from the Moon, and thus can use the orbital energy of inbound payloads to boost outbound payloads, it is possible for the first component of the system, the Earth-orbit Tether Boost Facility, to repeatedly boost payloads into lunar transfer trajectories *without propellant expenditure or return traffic needed*. The key to achieving this is the combination of momentum-exchange tether techniques with electrodynamic tether propulsion.

HEFT Tether Boost Facility

This concept, the "High-strength Electro-dynamic Force Tether" (HEFT) Facility,[6] is illustrated in Figure 3. The HEFT Facility would include a central facility housing a power supply, ballast mass, plasma contactor, and tether deployer, which would extend a long, tapered, high-strength tether. A small grapple vehicle would reside at the tip of the tether to facilitate rendezvous and capture of the payloads. The tether would include a conducting core, and a second plasma contactor would be placed near the tether tip. By using the power supply to drive current along the tether, the HEFT Facility could generate electrodynamic forces on the tether. By properly varying the direction of the current as the tether rotates and orbits the Earth, the facility can use these electrodynamic forces to generate either a net torque on the system to change its rotation rate, or a net thrust on the system to boost its orbit. The HEFT Facility thus could repeatedly boost payloads from LEO to the Moon, and in between each payload boost operation it would use propellantless electro-dynamic propulsion to restore its orbital energy.

Design of a Tether Boost Facility for Lunar Transfer Injection

The first stage of the Cislunar Tether Transport System will be a Tether Boost Facility in elliptical, equatorial Earth orbit. The mission of this facility is to pick up a payload from low-Earth orbit and inject it into a near-minimum energy lunar transfer orbit. The desired lunar transfer trajectories have a C_3 of approximately –1.9 $(km/s)^2$. A payload originating in a circular orbit at 350 km altitude has an initial velocity of 7.7 km/s and a C_3 of –60 $(km/s)^2$. To impulsively inject the payload into the lunar transfer orbit would require a ΔV of approximately 3.1 km/s.

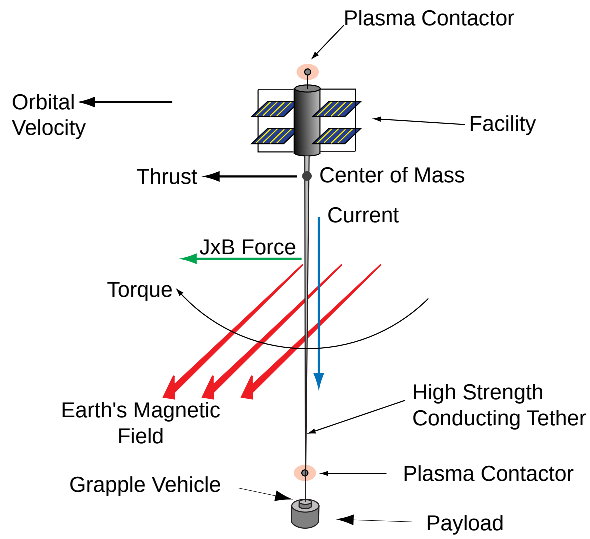

Figure 3. Schematic of the HEFT Facility design.

Orbital Design

In the Cislunar Tether Transport System, the transfer of payloads between a low-LEO and lunar transfer orbits is performed by a single rotating tether facility. This facility performs a catch and release maneuver to provide the payload with two boosts of approximately 1.5 km/s each. To enable the tether to perform two "separate" ΔV operations on the payload, the facility is placed into a highly elliptical orbit with its perigee in LEO. When the tether is near perigee, its center of mass is moving approximately 1.5 km/s faster than the payload in circular LEO. The tether rotation is arranged such that when the facility is at perigee, the tether is swinging vertically below the facility so that it can catch a payload moving more slowly than the facility. After it catches the payload, it holds the payload for half a rotation and then releases it at the top of the tether's rotation, injecting the payload into the high-energy transfer trajectory.

Table 1 shows the orbital design for the LEO⇒LTO Tether Boost Facility. To minimize the mass of the tether, it is tapered along its length to maintain a constant load level; Figure 4 illustrates this tapering.

Table 1. System Orbital Design for LEO⇒LTO Boost

System Masses		
Tether mass	8,274	kg
CS Active Mass	11,514	kg
CS Ballast Mass	3490	kg
Grapple mass	650	kg
Total Facility Mass	**23,928**	kg
Total Launch Mass	**20,438**	kg
Payload Mass	**1,000**	kg

Tether Characteristics		
Tether Length	100	km
Tether mass ratio	8.27	
Tether tip velocity at catch	1,555	m/s
Tether tip velocity at toss	1,493	m/s
Tether angular rate	0.01905	rad/s
Gravity at Control Station	0.80	g
Gravity at payload	2.90	g
Rendezvous acceleration	3.02	g

		Pre-Catch		Joined System	Post-Toss	
Positions & Velocities		**Payload**	**Tether**	**Post-catch**	**Tether**	**Payload**
perigee altitude	km	300	382	378	375	457
apogee altitude	km	300	11935	11018	10172	406515
perigee radius	km	6678	6760	6757	6753	6835
apogee radius	km	6678	18313	17397	16550	412893
perigee velocity	m/s	7726	9281	9219	9156	10712
apogee velocity	m/s	7726	3426	3580	3736	177
CM dist. From Station	m		18356	21632	18356	
CM dist. To Grapple	m		81644	78368	81644	
ΔV to Reboost	m/s				125	
Basic Orbital Parameters						
semi-major axis	km	6678	12537	12077	11652	209864
eccentricity		0.0	0.461	0.441	0.420	0.967
inclination	rad	0	0	0	0	0
semi-latus rectum	km	6678	9875	9733	9592	13447
sp. mech. energy	m2/s2	-2.98E+07	-1.59E+07	-1.65E+07	-1.71E+07	-9.50E+05
vis-viva energy	m2/s2	-5.97E+07	-3.18E+07	-3.30E+07	-3.42E+07	-1.90E+06
period	sec	5431	13970	13208	12517	956793
period	min	90.5	232.8	220.1	208.6	15946.5
station rotation period	sec		329.8	329.8	329.8	
rotation ratio			42.4	40.0	37.9	

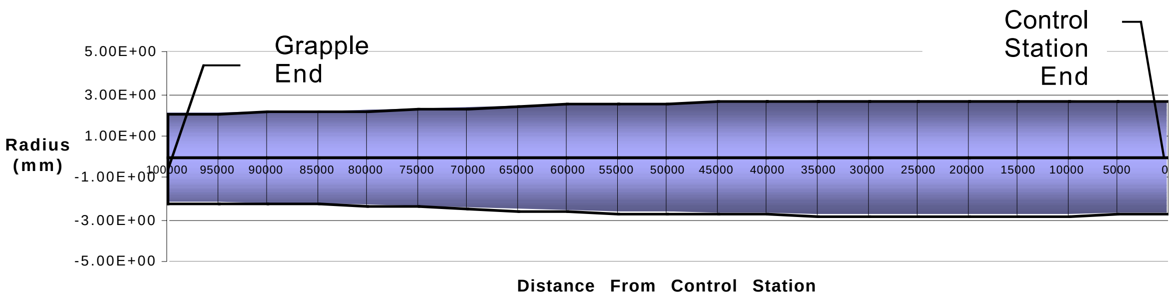

Figure 4. Taper of the tether cross-section (tether will actually be composed of multiple smaller lines).

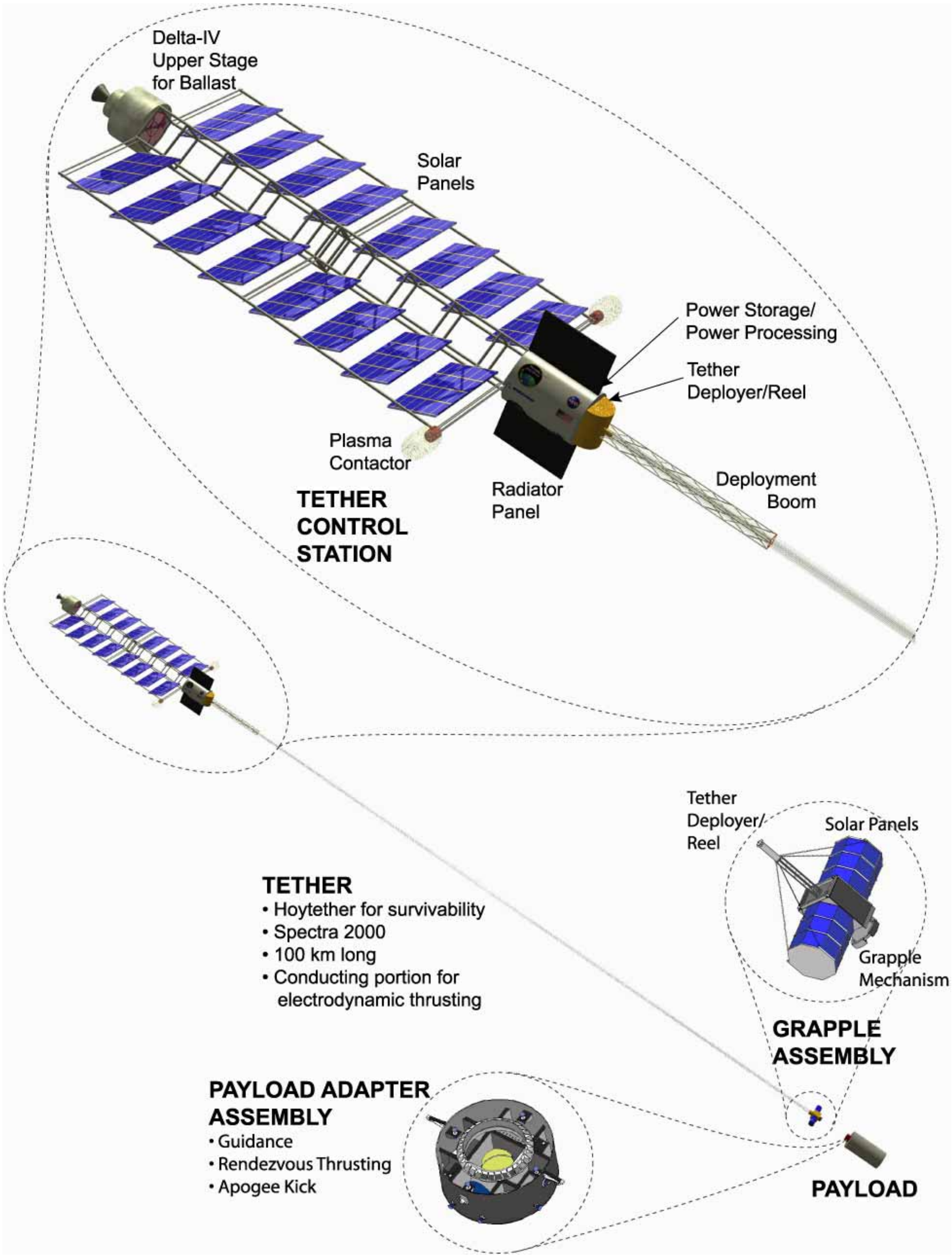

Figure 5. System Design for a Tether Boost Facility.

System Design

Figure 5 illustrates the system concept design for the Tether Boost Facility. The Tether Boost Facility is composed of a Control Station, a tapered high-strength tether, and a Grapple Assembly. In addition, a Payload Accommodation Assembly (PAA) will be attached to the payload to provide maneuvering and guidance for rendezvous. This PAA will initially be an expendable unit incurring recurring costs, but once round-trip traffic is established the PAA's could be re-fueled and reused for return payloads.

The tether facility is sized to be deployed with a single launch of a Delta-IV-H or comparable vehicle. Note that the system mass given in Table 1 is not the minimum possible system mass; a lighter system mass could be designed for a system optimized for boosting payloads to LTO. This system mass was chosen to utilize the full capability of the Delta-IV-H vehicle, and to optimize the system for boosting larger satellites to geostationary transfer orbits. As Figure 5 shows, the 3490 kg Delta upper stage will be retained for use as ballast mass. The control station includes an array of solar panels which swivel to track the sun as the tether facility rotates. In this design, we have chosen to place the control station at the end of the tether, rather than at the center of mass of the facility. This choice was made for several reasons: because it minimizes the dynamical complexity, because it requires only one tether deployer, and because the center of mass of the system shifts when the payload is captured and released.

Electrodynamic Reboost of the Tether Orbit

After boosting the payload, the tether facility will be left in a lower energy elliptical orbit. To restore the orbit, the tether system must increase the perigee velocity by 125 m/s, and increase the facility's orbital energy by 29 GJ. Because the tether is rotating, the direction of the current must be alternated as the tether rotates to produce a net thrust on the facility. Using a simulation of tether dynamics and electrodynamics, we have modeled reboost of a rotating tether system and found that the electrodynamic thrusting efficiency is approximately 33 μN/W, averaged over the perigee thrust period (shown in Figure 6). The tether facility will be able to collect solar power over approximately 80% of its orbital period. To reboost the orbit within 30 days, the facility will need a solar panel able to collect approximately 50 kW, and the tether facility will expend the collected energy at a rate of 200 kW during the perigee passes.

Dealing with Apsidal Precession

In order to deliver the payload to the Moon, the tether facility in equatorial Earth orbit must toss the payload out to a point near where the Moon will cross the Earth's equatorial plane. Thus the tether's perigee must be lined up on the opposite side of the Earth from that point. The oblateness of the Earth, however, will cause the line of apsides of the tether facility's elliptical orbit to precess. In the Cislunar Tether Transport System, we can deal with this issue in three ways.

First, we can use propellantless electrodynamic tether propulsion to change or oppose the oblateness-induced precession, either by raising/lowering the orbit or by generating thrust perpendicular to the facility's velocity.

Second, we can utilize tether reeling maneuvers to counteract the apsidal precession.[7] By reeling the tether in and out a small percentage of its total length once per orbit, the tether facility can exchange angular momentum between its rotation and its orbit, resulting in precession or regression of the line of apsides. With proper phasing and amplitude, tether reeling can hold the tether's orbit fixed so that it can send payloads to the

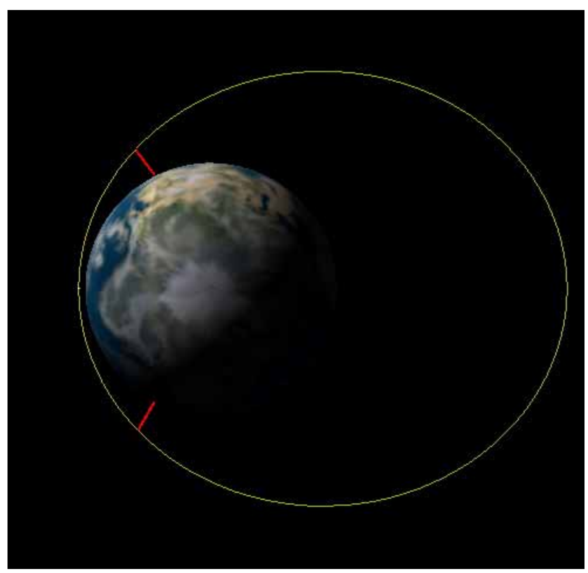

Figure 6. The HEFT Boost Facility's initial orbit. The red lines indicate the bounds of the perigee portion of the orbit where electrodynamic thrusting is effective.

Moon once per month.[8]

A third method is to choose the tether orbits such that their precession rates are nearly harmonic with the Moon's orbital rate, so that the line of apsides lines up with the Moon's nodes once every several months.

LEO⇒GTO Payload Transfer

The same Tether Boost Facility can, by changing its initial orbit and rotation rate, boost 2,500 kg payloads from a 308 km circular orbit to geostationary transfer orbit. To perform this LEO⇒GTO boost operation once per month, the system must have a 150 kW solar power array, and expend the collected energy at a rate of 450 kW during perigee passes. The Control Station shown in Figure 5 is sized with a 200 kW (beginning of life) solar array.

System Modularity

The Tether Boost Facility concept has been designed to enable it to be grown incrementally. After the initial facility, capable of tossing 1000 kg to LTO and 2,500 kg to GTO, has been deployed and tested, a second module of nearly identical hardware can be launched and combined in a parallel fashion with the first module, as illustrated in Figure 7. This will increase the system's capacity to 2,000 kg to LTO and 5,000 kg to GTO. The parallel construction will provide redundancy to the system, reducing the need for redundancy within each module. Cross-linking between the two parallel tethers could be added to increase their redundancy. Additional modules can be launched to increase the system capacity further.

Design of a Lunavator™ Compatible with Minimal-Energy Lunar Transfers

The second stage of the Cislunar Tether Transport System is a lunar-orbit tether facility that catches the payloads sent by the Earth-orbit tether and deposits them on the Moon with zero velocity relative to the surface.

Background: Moravec's Lunar Skyhook

In 1978, Moravec[9] proposed that it would be possible to construct a tether rotating around the Moon that would periodically touch down on the lunar surface. Moravec's "Lunar Skyhook" would have a massive central facility with two tether arms, each with a length equal to the facility's orbital altitude. It would rotate in the same direction as its orbit with a tether tip velocity equal to the orbital velocity of the tether's center-of-mass so that the tether tips would periodically touch down on the Moon with zero velocity relative to the surface (to visualize this, imagine the tether as a spoke on a giant bicycle wheel rolling around the Moon).

As it rotates and orbits around the Moon, the tether will capture payloads from Earth as they reach perilune and then set them down on the surface of the Moon. Once round-trip traffic is established, the tether could simultaneously pick up payloads to be returned to Earth, and later toss them down to LEO.

Lunavator™ Design

In order to minimize the ΔV requirements placed upon the Earth-orbit portion of the Cislunar Tether Transport System and thereby permit the use of a single Earth-orbit tether with a reasonable mass, we have developed a method for a single lunar-orbit tether to capture a payload from a minimal-energy lunar transfer orbit and deposit it on the tether surface with zero velocity relative to the surface.

Moon-Relative Energy of a Minimum-Energy LTO

A payload that starts out in LEO and is injected into an elliptical, equatorial Earth-orbit with an apogee that just reaches the Moon's orbital radius will have a C_3 relative to the Moon of approximately 0.72 km^2/s^2. For a lunar transfer trajectory with a closest-approach altitude of several hundred kilometers, the payload will have a velocity of approximately 2.3 km/s at perilune. As a result, it would be moving too slowly to rendezvous with the upper tip of

Figure 7. Tether Boost Facility with two modules, capable of tossing 2000 kg to LTO. (Tether length not to scale)

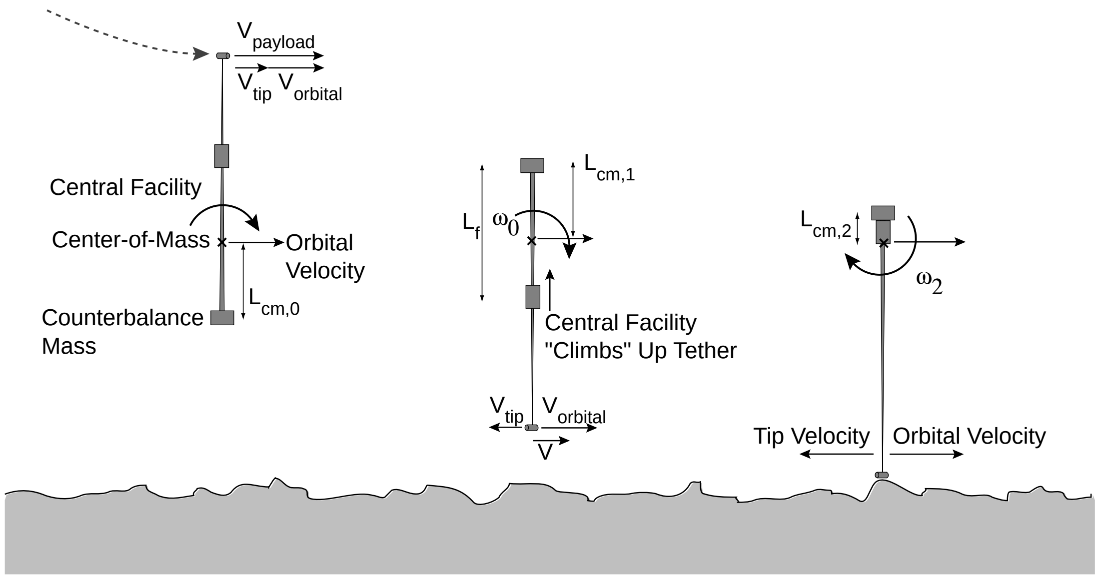

Figure 8. Method for a lunar tether to capture a payload from a minimal-energy LTO and deposit it on the Moon with zero velocity relative to the surface.

Moravec lunar Skyhook, which would have a tip velocity of 2.9 km/s at the top of its rotation. Consequently, the design of the lunar tether system must be modified to permit a tether orbiting the Moon at approximately 1.5 km/s to catch a payload to at perilune when the payload's velocity is approximately 2.3 km/s, then increase both the tether length and the angular velocity so that the payload can be set down on the surface of the Moon with zero velocity relative to the surface. Simply reeling the tether in or out from a central facility will not suffice, because reeling out the tether will cause the rotation rate to decrease due to conservation of angular momentum.

A method that can enable the tether to catch a payload and then increase the tether rotation rate while lowering the payload is illustrated in Figure 8. The "Lunavator™" tether system is composed of a long tether, a counterbalance mass at one end, and a central facility that has the capability to climb up or down the tether. Initially, the facility would locate itself near the center of the tether, and the system would rotate slowly around the center-of-mass of the system, which would be located roughly halfway between the facility and the counterbalance mass. The facility could then capture an inbound payload at its perilune. The facility would then use energy from solar cells or other power supply to climb up the tether towards the counterbalance mass. The center-of-mass of the system will remain at the same altitude, but the distance from the tether tip to the center-of-mass will increase, and conservation of angular momentum will cause the angular velocity of the system to increase as the facility mass moves closer to the center-of-mass.

Lunavator™ Design

Using analyses of the orbital mechanics of the system, we have found the following first-order design for a Lunavator™ capable of catching payloads from minimal-energy lunar transfer orbits and depositing them on the surface of the Moon:

Payload Trajectory:
- mass M_p = 1000 kg
- perigee altitude h_p = 328.23 km
- Moon-relative energy $C_{3,M}$ = 0.719 km^2/s^2

Lunavator™:
- tether length L = 200 km
- counterbalance mass M_c = 6,000 kg
- facility mass M_f = 6,000 kg
- tether mass M_t = 4,706 kg
- Total Mass M = 16,706 kg
  = 16.7 x payload mass
- Orbit Before Catch:
  central facility position L_f = 155 km
  tether tip velocity $V_{t,0}$ = 0.748 km/s
  rotation rate ω_0 = 0.00566 rad/s
  circular orbit altitude $h_{p,0}$ = 170.5 km
- Orbit After Catch:
  perigee altitude $h_{p,0}$ = 178 km,

apogee altitude $h_{a,0} = 411.8$ km
eccentricity $e_0 = 0.0575$

After catching the payload, the central facility climbs up the tether to the counterbalance mass, changing the rotation rate to:

- adjusted rotation rate $\omega_0 = 0.00929$ rad/s
- adjusted tip velocity $V_{t,2} = 1.645$ km/s

Payload Delivery:

- drop-off altitude $h = 1$ km (top of a lunar mountain)
- velocity w.r.t. surface $v = 0$ m/s

Lunavator™ Orbit: Polar vs. Equatorial

In order to provide the most consistent transfer scenarios, it is desirable to place the Lunavator™ into either a polar or equatorial lunar orbit. Each choice has relative advantages and drawbacks, but both are viable options.

Equatorial Lunar Orbit

The primary advantage of an equatorial orbit for the Lunavator™ is that equatorial lunar orbits are relatively stable. An equatorial Lunavator™, however, would only be able to service traffic to bases on the lunar equator. Because the lunar equatorial plane is tilted with respect to the Earth's equatorial plane, a payload boosted by the Earth-orbit tether facility will require a ΔV maneuver to bend its trajectory into the lunar equatorial plane. For most transfer opportunities, this correction can be accomplished by a small rocket thrust on the order of 25 m/s.

Polar Lunar Orbit

A polar orbit would be preferable for the Lunavator™ for several reasons. First, direct transfers to polar lunar trajectories are possible with little or no propellant expenditure required. Second, because a polar lunar orbit will remain oriented in the same direction while the Moon rotates inside of it, a polar Lunavator™ could service traffic to any point on the surface of the Moon, including the potentially ice-rich lunar poles. Low polar lunar orbits, however, are unstable. The odd-harmonics of the Moon's potential cause a circular, low polar orbit to become eccentric. Eventually, the eccentricity becomes large enough that the perilune is at or below the lunar surface. For the 178 km circular orbit, the rate of eccentricity growth is approximately 0.00088 per day.

Fortunately, the techniques of orbital modification using tether reeling, proposed by Martínez-Sánchez and Gavit[7] and by Landis[10]

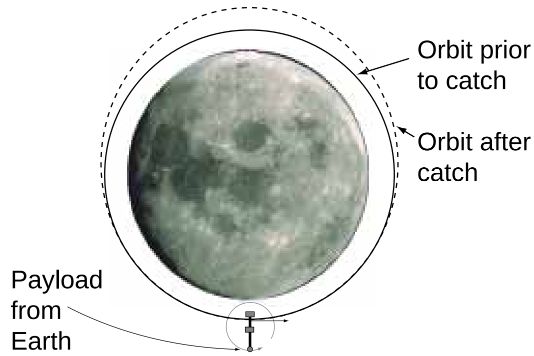

Figure 9. Lunavator™ orbits before and after payload capture.

may provide a means of stabilizing the orbit of the Lunavator™ without requiring expenditure of propellant. Tether reeling can add or remove energy from a tether's orbit by working against the non-linearity of a gravitational field. The basic concept of orbital modification using tether reeling is illustrated in Figure 10. When a tether is near the apoapsis of its orbit, the tidal forces on the tether are low. When it is near periapsis, the tidal forces on the tether are high. If it is desired to reduce the eccentricity of the tether's orbit, then the tether can be reeled in when it is near apoapsis, under low tension, and then allowed to unreel under higher tension when it is at periapsis. Since the tidal forces that cause the tether tension are, to first order, proportional to the inverse radial distance cubed, more energy is dissipated as the tether is unreeled at periapsis than is restored to the tether's orbit when it is reeled back in at apoapsis. Thus, energy is removed from the orbit. Conversely, energy can be added to the orbit by reeling in at periapsis and reeling out at apoapsis. Although energy is removed (or added) to the orbit by the reeling maneuvers, the orbital angular momentum of the orbit does not change. Thus the eccentricity of the orbit can be changed.

The theories developed in references 7 and 10 assumed that the tether is hanging (rotating once per orbit). Because the Lunavator™ will be rotating several times per orbit, we have extended the theory to apply to rapidly rotating tethers.[8] Using a tether reeling scheme in which the tether is reeled in and out once per orbit as shown in Figure 10, we find that a reeling rate of 1 m/s will reduce the eccentricity of the Lunavator™'s orbit by 0.0011 per day, which should be more than enough to counteract the effects of lunar perturbations to the tether's orbit. Thus tether reeling may provide a means of stabilizing the orbit of a polar Lunavator™ without requiring propellant expenditure. This tether reeling, however, would add additional complexity to the system.

Cislunar System Simulations

Tether System Modeling

In order to verify the design of the orbital dynamics of the Cislunar Tether Transport System, we have developed a numerical simulation called "TetherSim" that includes:

- The 3D orbital mechanics of the tethers and payloads in the Earth-Moon system, including the effects of Earth oblateness, using Runge-Kutta integration of Cowell's method.
- Modeling of the dynamical behavior of the tethers, using a bead-and-spring model similar to that developed by Kim and Vadali.[11]
- Modeling of the electrodynamic interaction of the Earth-orbit tether with the ionosphere.

Using this simulation tool, we have developed a scenario for transferring a payload from a circular low-LEO orbit to the surface of the Moon using the tether system designs outlined above. We have found that for an average transfer scenario, mid-course trajectory corrections of approximately 25 m/s are necessary to target the payload into the desired polar lunar trajectory to enable rendezvous with the Lunavator™. A simulation of a transfer from LEO to the surface of the Moon can be viewed at *www.tethers.com*.

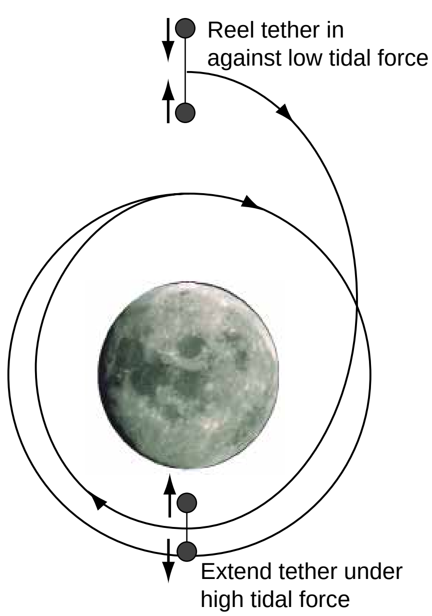

Figure 10. Schematic of tether reeling maneuver to reduce orbital eccentricity.

Targeting the Lunar Transfer

In addition to the modeling conducted with TetherSim™, we have also conducted a study of the Earth-Moon transfer to verify that the payload can be targeted to arrive at the Moon in the proper plane to rendezvous with the Lunavator™. This study was performed with the MAESTRO code,[12] which includes the effects of luni-solar perturbations as well as the oblateness of the Earth. In this work we studied targeting to both equatorial and polar lunar trajectories.

We have found that by varying the energy of the translunar trajectory and adjusting the argument of perigee, it is possible to target the payload to rendezvous with a polar orbit Lunavator™ with a wide range of ascending node positions of the Lunavator™ orbit. Our simulations indicate that the viable nodal positions ranges at least ±10° from the normal to the Earth-Moon line.

Comparison to Rocket Transport

Travelling from LEO to the surface of the Moon and back requires a total ΔV of more than 10 km/s. To perform this mission using storable chemical rockets, which have an exhaust velocity of roughly 3.5 km/s, the standard rocket equation requires that a rocket system consume a propellant mass equal to 16 times the mass of the payload for each mission. The Cislunar Tether Transport System would require an on-orbit mass of less than 37 times the payload mass, but it would be able to transport many payloads. In practice, the tether system will require some propellant for trajectory corrections and rendezvous maneuvers, but the total ΔV for these maneuvers will likely be less than 100 m/s. Thus a simple comparison of rocket propellant mass to tether system mass indicates that the fully reusable tether transport system could provide significant launch mass savings after only a few round trips. Although the development and deployment costs associated with a tether system would present a larger up-front expense than an existing rocket-based system, for frequent, high-volume round trip traffic to the Moon, a tether system could achieve large reductions in transportation costs by eliminating the need to launch large quantities of propellant into Earth orbit.

Summary

Our analyses have concluded that the optimum architecture for a tether system designed to transfer payloads between LEO and the lunar surface will utilize one tether facility in an elliptical, equatorial Earth orbit and one tether in low lunar orbit. We have developed a system concept design for a 100 km long Earth-orbit Tether Boost Facility capable of picking 1,000 kg payloads up from LEO and injecting them into a minimal-energy lunar transfer orbit. This system will also boost 2,500 kg payloads to GTO. The payload capacity of the system can be built incrementally by deploying additional tether modules. After boosting a payload, the facility can use electrodynamic propulsion to reboost its orbit, enabling the system to repeatedly send payloads to the Moon without requiring propellant or return traffic. When the payload reaches the Moon, it will be caught and transferred to the surface by a 200 km long lunar tether. Using two different numerical simulations, we have tested the feasibility of this design and developed scenarios for transferring payloads from a low-LEO orbit to the surface of the Moon, with only 25 m/s of ΔV needed for small trajectory corrections. Thus, it appears feasible to construct a Cislunar Tether Transport System that can greatly reduce the cost of round-trip travel between LEO and the surface of the Moon by minimizing the need for propellant expenditure.

Acknowledgments

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Tether Boost Facility Design Study
Interim Report
July 21, 2000

The Boeing Company

Subcontract MMOSTT-01

to Tethers Unlimited, Inc. Contract No. NIAC-07600-034

Moon Mars Orbiting Spinning Tether Transport (MMOSTT)

Architecture Study

Interim Report Outline

- Configuration Description
  - LEO Facility System Architecture
  - Baseline Concept
  - Alternate (De-Spun) Concept
  - Subsystem Descriptions

- Mass Properties for Baseline Concept

- Configuration Refinements and Issues for Further Consideration

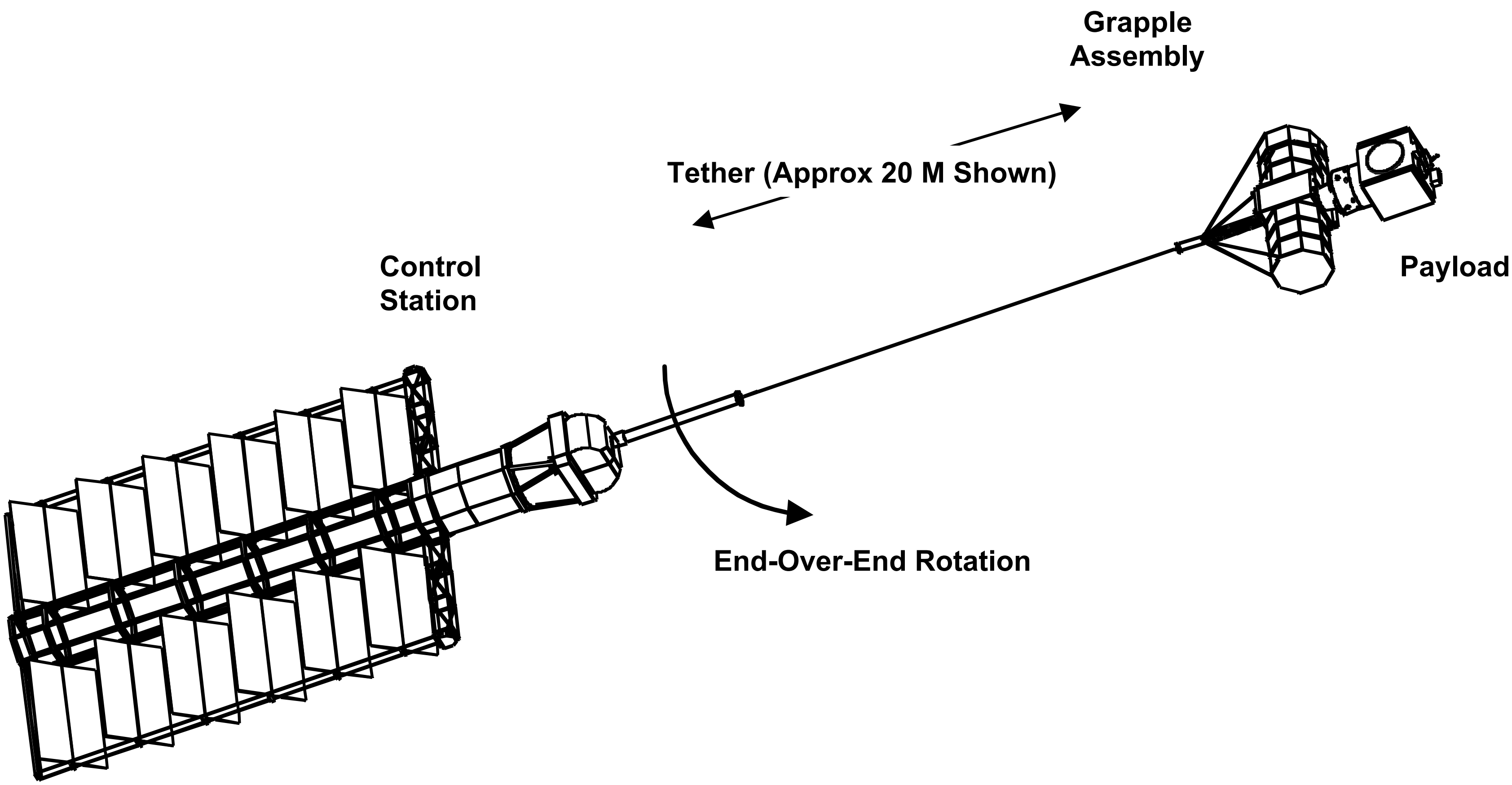
- Appendices
  A. Required Power for Electrodynamic Reboost
  B. Reel Subsystem
  C. Tether Boost Facility System Requirements Document

Tether Boost Facility Configuration Description

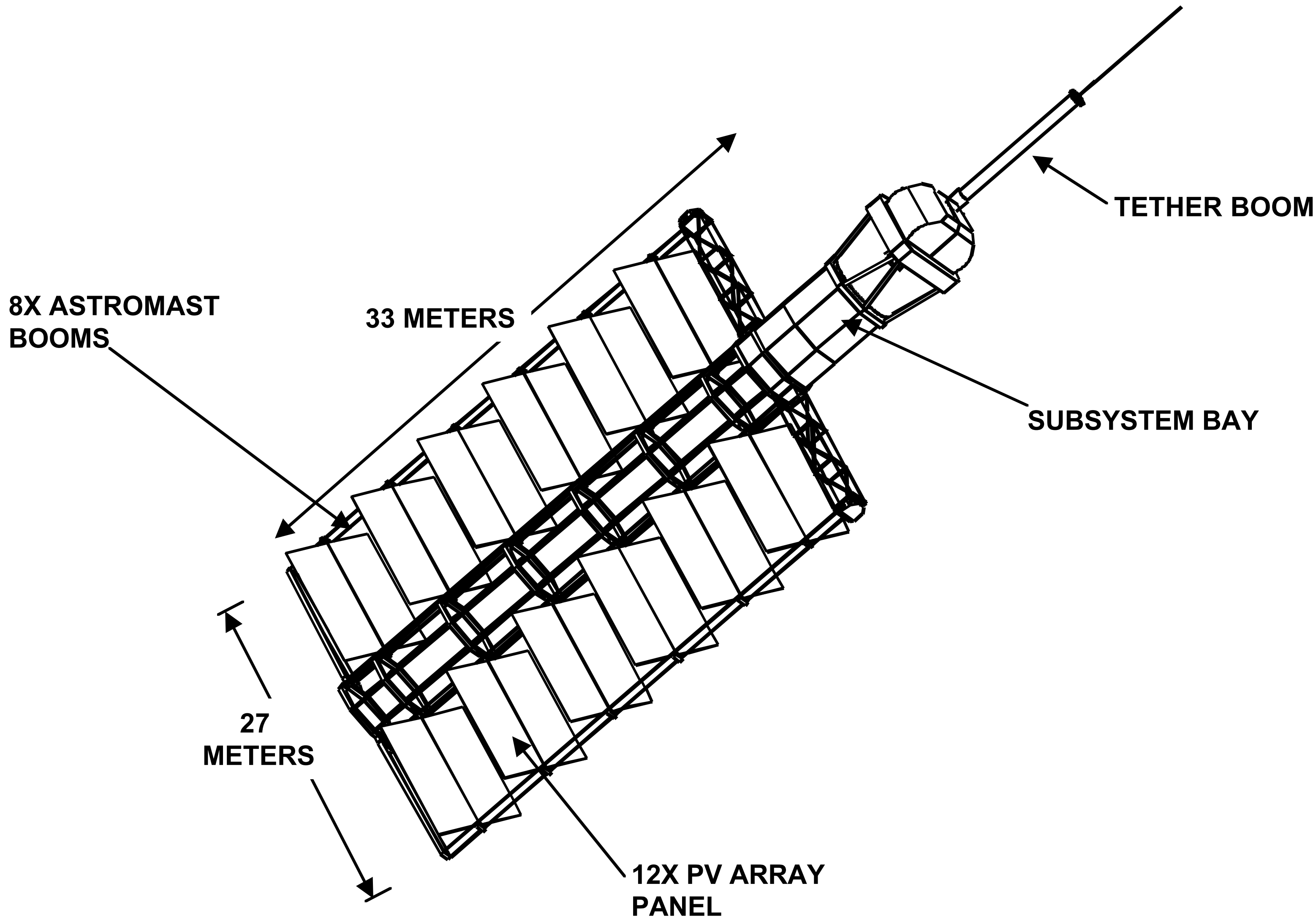
Mission Definition

- Payload Capacity: 5,000 kg
- Throughput: 1 payload every 30 days
- Boost Capability: 300 km circular -> GTO

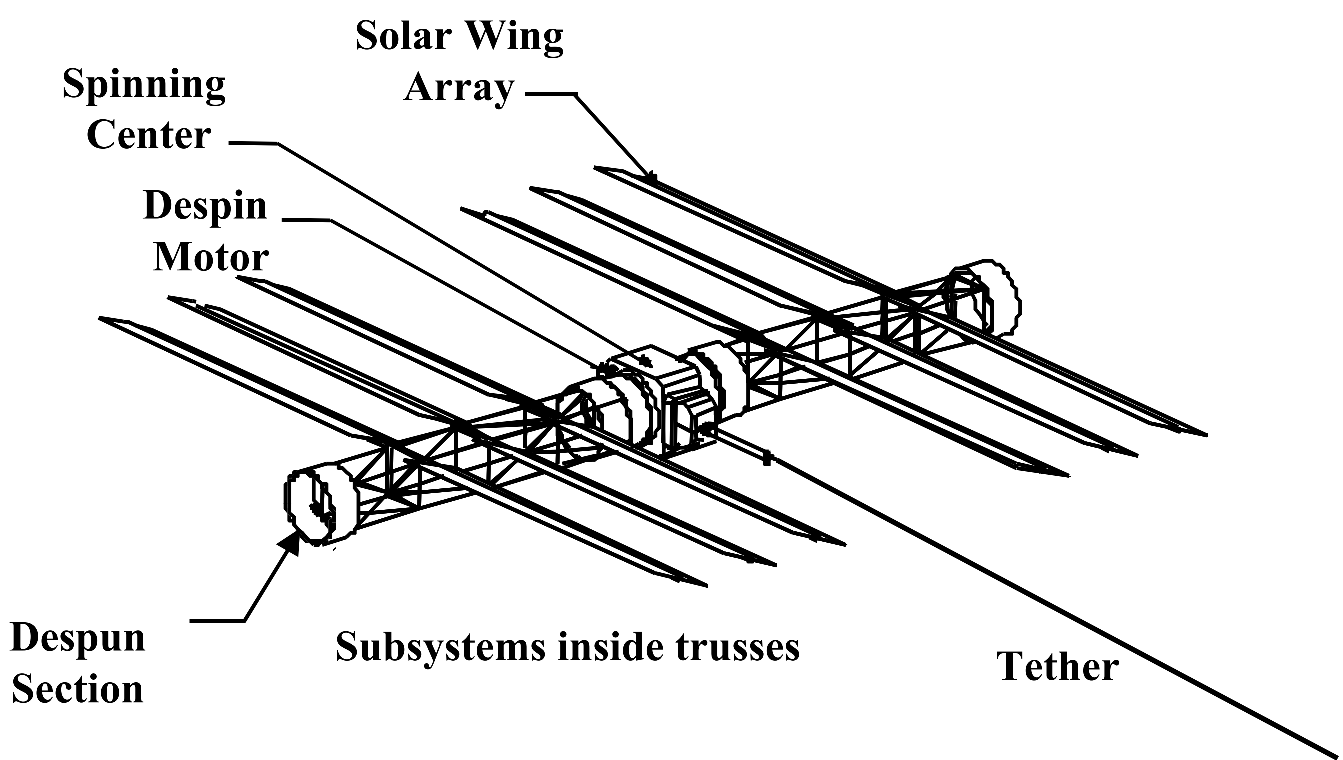
Baseline Configuration



Baseline Control Station Details



Despun Configuration

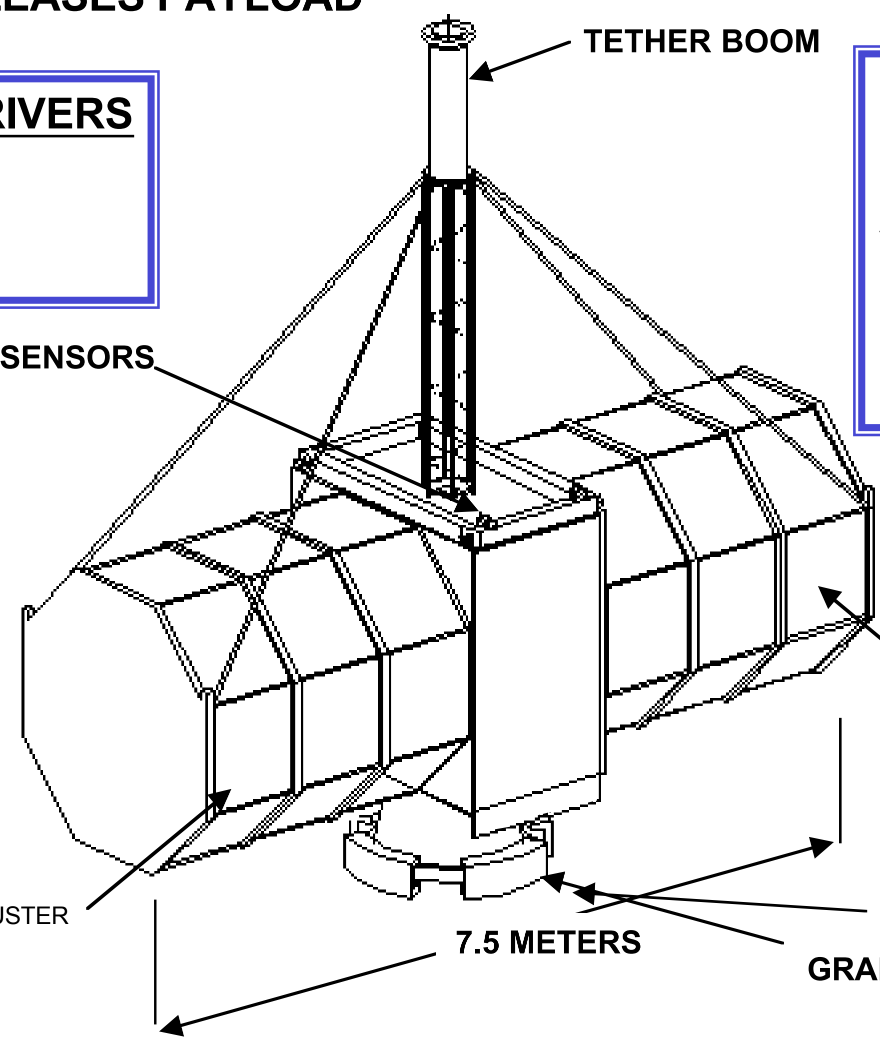


Baseline Grapple Assembly

CAPTURES AND RELEASES PAYLOAD

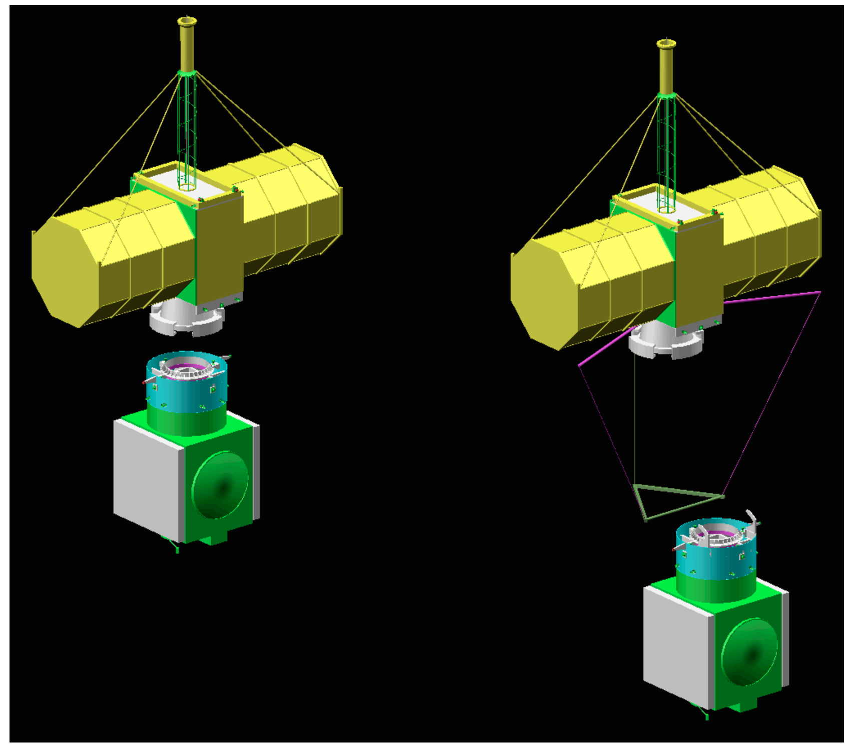
CONFIGURATION DRIVERS
- Capture Options
- System Rotation
- Loads

SIZING PARAMETERS
- 1380 Watts
- 15.61 Square Meter Pv Arrays
- 1.25 Meter Dia Capture Ring
- 50 Kg Batteries (Ni-H_2)
- TBD Fuel

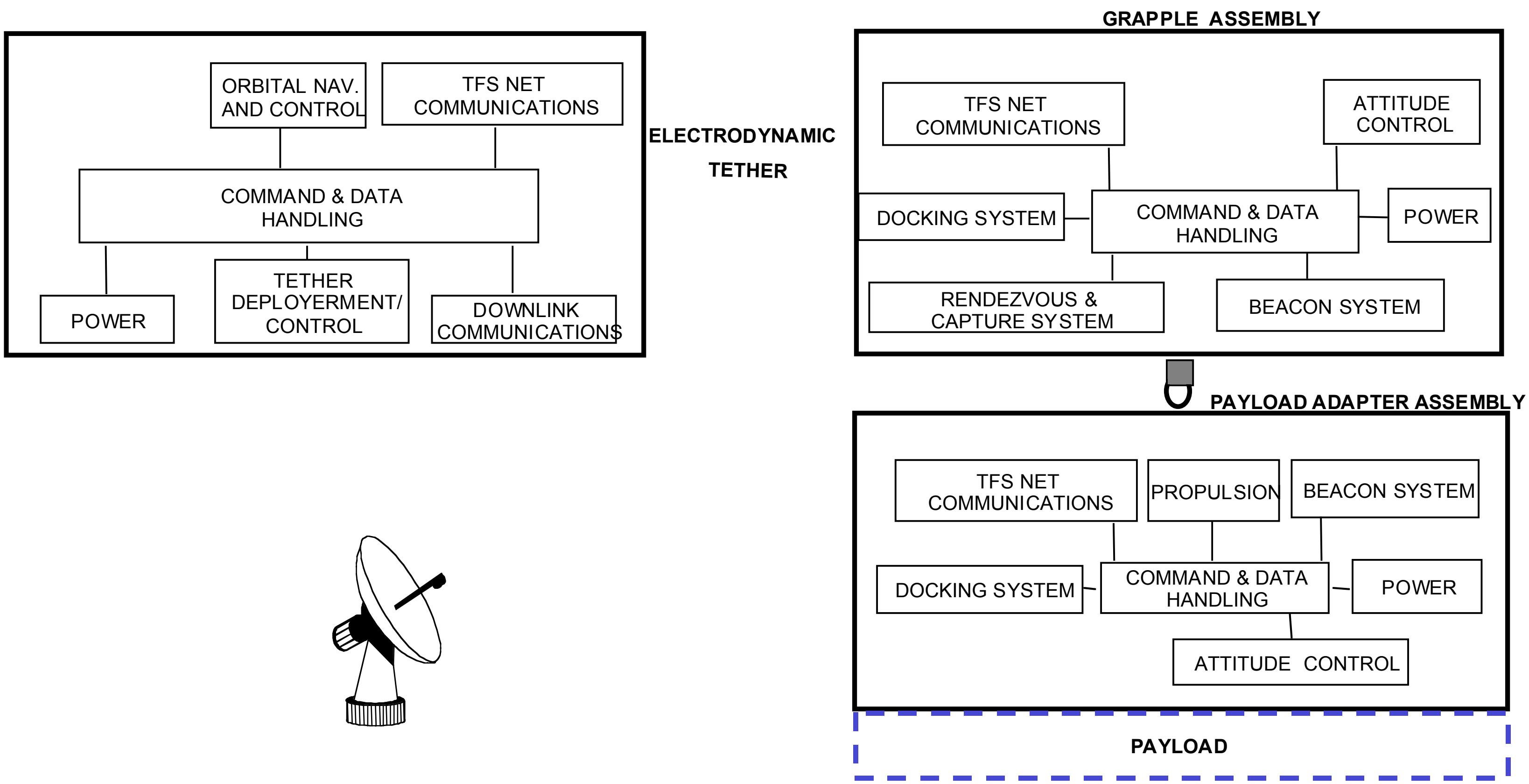


Payload Capture Concepts

- Several capture concepts examined
  - Two version of mechanical capture
    - Direct mechanical latching
    - Trapeze deployment and reel in
  - Next page shows Electromagnetic grapple approach that use electromagnetic to capture and mechanical latches to secure payload before system is loaded



LEO Facility System Architecture



Control Station Subsystems

Communications Subsystem (CS)
- Comm net Antenna, Transmitter, Receiver
- Downlink Antenna, Tx, Rcvr

Attitude and Location Determination /Control Subsystem (ALDCS)
- GPS Antenna, Receiver
- Attitude Stabilization Software

Propulsion Subsystem (PS)
- Thruster control electronics
- Ion Thrusters
- Solar electric main propul unit

Mechanical Subsystem (MS)
- Facility Structures
- Erosion Protection
- Micrometeoroid Protection

Thermal Control Subsystem (TCS)
- Temp Sensors
- Heaters
- MLI, Radiators

Electrical Power Subsystem (EPS)
- Solar Collectors/Drive Motors
- Sun Angle Sensor
- Pwr Mgt Unit
  - EPS Control Processor
  - Battery Charger
  - Power Regulator
  - Power Distributor
- Batteries
- Electrodynamic Tether
- Software

Cmd & Data Handling Subsystem
- Computer
- TLM Mux/Demux
- Software
  - Operating System
  - Applications
    - Orbital Mechanics
    - Equipment Control

Networks Subsystem (NS)
- Power Cables
- Data Cables

Retrieval,Deployment and Spin Control Subsystem (RDSCS)
- Winch, traction drive motors and controller
- Tether
- Tether cutter
- Tether deploy speed sensor
- Tether deployed length

sensor
- Tether fully deployed sensor
- Tether tension sensor
- Tether impact detector
- Separated tether detector
- Tether departure angle sensor
- Tether position sensor
- Spin maintenance sensors
- Spin maintenance software

Payload Adapter Assembly Subsystems

Payload Adapter Assembly (PLAA)

- CDHS
- Propulsion Subsystem
- ALDCS
- Passive Reflectors
- Docking Adapter
- Comm Subsystem
- Beacon Homing Subsystem
- Electrical Power Subsystem
- Thermal Control Subsystem
- Mechanical Subsystem
- Networks Subsystem

Grapple Assembly Subsystems

- Comm Subsystem
- ALDCS
- Propulsion Subsystem
- Mechanical Subsystem
- Thermal Control Subsystem
- Electrical Power Subsystem
- Cmd and Data Handling Subsystem
- Networks Subsystem
- Proximity Sensors
- P/L Capture/Release Device
    - Docking assembly
- Final Lock-on & Capture Navigation Subsystem
- Beacon Homing System

Potential Reeling Functions

Reeling out at high speed during rendezvous/capture phase, then reeling back in more slowly

"Pumping" the tether to modify the orbit after reboost (increase eccentricity and turn excess perigee altitude into more apogee altitude)

After Facility deployment, use reeling to spin up Facility and enter operational stage

Reeling to allow long time storage of tether/grapple and Facility repair/servicing

At current design stage, more concerned with reeling in/pumping impacts on tether deployment and control system on Control Station design

- Expect Winch to have to overcome high forces to begin reeling tether once it's fully deployed, resulting in high electrical power requirements similar to those of the electrodynamic tether reboost power requirements
- Expect tether dynamics induced by pumping to be harder to damp/control than those that exist when deploying tether slowly or reeling in slowly

Mass Properties

LEO Control Station Weights

	Qty	Redundancy	mass (kg)	total mass (kg)	peak pwr (w)	aver pwr (W)	total aver pwr (W)	Source
LEO Control Station				**28464.8**			#NAME?	includes Pwr estimate for ED tether reboost
Station dry wt. Subtotal				**28464.8**			#NAME?	
Thermal Control Subsys	1	1	3063.20	3063.20		#NAME?	#NAME?	
Cabling/Harnesses				**1945.6**				SMAD, 10% powered sys.wt.
Structure				**7062.7**				10% supported weight to account for rotational acceleration loads + SMAD:20% supported wt, 10% fastener allowance, 5% uncertainty
Electr.Pwr.				**14993.0**			**3200**	
PV array panels	1	1	3011.5	3011.5			0	Cells, cover glass, backing
Supporting structure	1	1	451.7	451.7				
Power converter	1	1						
Power Storage	1	1	11,196	11,196			0	John Blumer
PV array drive motors	12	2	12.00	288.00		250	3000	Alcatel Space Industries similarity, 1997 LEO-GEO pwr estim. Was 500 W
PMAD unit	1	2	22.68	45.36		200	200	*LEO-GEO used .025*pwr req'd for all subsystems
Downlink Comm Subsys				**3.4**			**3.43**	
Downlink Transceiver	1	3	0.69	2.07		3.43	3.43	COTS average estimate
Downlink antennae	2	3	0.23	1.36				SOTV
TFS Net Comm Subsys				**3.4**			**3.43**	
Comm. antennae	2	3	0.23	1.36				SOTV
Transceiver	1	3	0.69	2.07		3.43	3.43	COTS average estimate
C&DH				**60.0**			**75**	
Computer	1	3	20.00	60.00		75	75	SOTV, total computer (box, cables, I/O)
TT&C				**24.0**			**53.7**	
transponder	1	3	8.00	24.0		17.9	53.7	COTS average: L3, Motorola, Cincinnati Electronics
ADCS				**309.5**			**124.9**	
sun sensor	1	3	0.26	0.78		1.5	1.5	SOTV
CMG	1	4	69.79	279.14				Similarity, ISS CMG's
Inertial Navigation Unit	1	3	8.39	25.17		40	120.00	SOTV-Honeywell
GPS antennae	3	3	0.25	2.29				SOTV
GPS transceiver	1	3	0.69	2.07		3.43	3.43	COTS average estimate
ED Tether Subsystem				**6631.4**			**200.0**	
Plasma Contactor	2	2		0.00				
Power converter	1	2	3293.0	6586.00				similarity to HRF Power Supply's Pwr Converter (JSFC),120V-to-20V conversion
PMAD unit	1	2	22.68	45.36		200	200	*LEO-GEO used .025*pwr req'd for all subsystems
(Optional) separate tether for ED reboost	1	2		0.00				
(Optional) separate tether's reeling assembly	1	2		0.00				
Tether Deployment/Control		1		**1000.0**			**#NAME?**	
Thrust Pwr for Reboost							717,682	Brian Tillotson
eeling assembly (motor, etc)	1	1	1000.00	1000.00		#NAME?	#NAME?	1997 LEO-GEO/used TSS similarity

Grapple Assembly Weights

	Qty	Redundancy	mass	total mass	peak pwr	aver pwr	total aver pwr	Source
Grapple Assembly				**584.1**			**327**	
Electrical Power Subsystem				**177.9**			75	
PV arrays		1	134.0	134.0			0	1997 LEO-GEO figure
Pwr Storage		1.5	21.2	0.0			0	1997 LEO-GEO figure
PMAD unit	1	2	22.7	22.7		75	75	SOTV
TFS Comm Subsys				**3.0**			**50**	
Beacons	1	2	1.5	3.00		50	50	1997 LEO-GEO figure
C&DH				**7.5**			**75**	SOTV
Computer Chassis	1	1	5.00	5.00				
Processor	1	2	1.00	2.00		75	75	
I/O cards	2	2	1.50	5.99				
ADCS				**19.7**			0	
GPS antennae	3	2	0.25	1.52				SOTV
GPS transceiver	1	2	0.69	1.38		3.43	3.43	COTS average estimate
Inertial Navigation Unit	1	2	8.39	16.78		40	80.00	SOTV-Honeywell
Docking seeker	1	2	2.3			50	50	1997 LEO-GEO figure
Thermal Subsys	1	1	7.0	**7.0**			0	1997 LEO-GEO figure
Grapple Subsystem				**200.0**			**127**	1997 LEO-GEO figure
Sensor Processing Electronics Assem	1	2		0.00		80	80	1997 LEO-GEO figure
CCd array	1	2		0.00		16	16	1997 LEO-GEO figure
Laser Diode illuminator	1	2		0.00		3	3	1997 LEO-GEO figure
rapple/Release Mechanism	1	2	200.0	400.00				1997 LEO-GEO figure
Grapple/Release Mechanism actuators				0.00				1997 LEO-GEO figure-rolled up into Mechanisms wt. Figure
Docking sensors	4	2		0.00		1	4	1997 LEO-GEO figure-rolled up into Mechanisms wt. Figure
Docking actuators	4	2		0.00		6	24	1997 LEO-GEO figure-rolled up into Mechanisms wt. Figure
Cabling/Harnesses				**40.8**				10% wt. Of subsystems using elec.pwr
Structure				**148.0**				same estimation method as used for Control Station Structural mass estimate

Payload Adapter Assembly Weights

	Qty	Redunda ncy	mass	total mass	peak pw	aver pwr	total aver pwr	Source
Payload Adapter Assembly, wet				547.0			1365.43	
Fuel Load			54.0				0	1997 LEO-GEO figure
PAA dry wt. Subtotal				493.0			1365.43	
Tankage/Propulsion subsys				**29.0**			**72**	1997 LEO-GEO figures
Valve drivers	8	1				5	40	
Pressure sensors	8	1				2	16	
Temperature sensors	8	1				2	16	
thrusters			7.3					
RCS tanks		1	15.0				0	
Plumbing & valve allowance		1	6.8				0	
Electrical Power Subsystem				**120.9**			**75**	1997 LEO-GEO figures
Batteries			81.66					
PMAD	1	1	22.7			75	75	
other control avionics	1	1	16.6			75	75	
TFS Net Comm Subsys				**13.3**			**60**	
Comm. antennae	2	1	0.2	0.5				SOTV
Comm. Avionics	1	1		11.4		50	50	1997 LEO-GEO figures
Beacon	1	1	1.5	1.5		10	10	1997 LEO-GEO figures
C&DH				**20.0**			**75**	SOTV
Computer	1	2	20.0	40.0		75	75	
ADCS				**25.1**			**83.43**	
GPS antennae	3	2	0.25	1.52				SOTV
GPS transceiver	1	2	0.69	1.38		3.43	3.43	COTS average estimate
Inertial Navigation Unit	1	2	8.39	16.78		40	80.00	SOTV-Honeywell
Docking Seeker	1	2	2.72	5.4				Estimate based on Ithaco CES
~~*Reaction Wheels*~~	~~3~~	~~2~~	~~10.5~~	~~62.9~~		~~5~~	~~15~~	1997 LEO-GEO figures, 3 Honeywell
Cooperative Grapple mech	1	1		**20.0**				1997 LEO-GEO figures
Thermal Subsys				**5.0**			**1000**	1997 LEO-GEO figures
Tank line heaters	4	1				150	600	
Heater mats	2	1				100	200	
Catalytic heaters	8	1				25	200	
Cabling/Harnesses				**20.8**				
Structure				**263.8**				
Primary				250.0				1997 LEO-GEO, 5% PL mass
Secondary				13.8				1997 LEO-GEO, 6% supported mass

Issues for Further Consideration and Refinement

- **Mass Properties Refinement**
- **Closed Form Analyses of Facility Definition, Operation, and Dynamics**
- **Boost Facilities Growth for Larger Mass Payloads**
- **Utility After 10 Year Lifetime**
- **System Element Interface Definition**

Tether Boost Facility
Required Power for Electrodynamic Reboost

Electrical Propulsion Energy

¥ Re-Boost of tether drives energy requirement

¥ Scenario

D 10:1 payload:tether mass => 50,000 kg tether facility

D 2460 m/s payload Δv => 246 m/s reboost Δv

D Post-boost perigee speed = 9250 m/s

D Kinetic energy = ? 5x10 ^{4} (9496 ^{2} - 9250 ^{2}) = 115,288 MJ

D Reboost time = 30 days = 2,592,000 sec

D Average reboost mechanical power = 44,478 W

D Fraction of time reboosting = 20%

D Active reboost mechanical power = 222,392 W

D Efficiency = 25 uN/W @ 9250 m/s = 23% (electric->kinetic)

¥ RobÕs number for thrust-while spin, 100% of the time < 2200 km alt

D Active electric reboost power input = 962 kW

Electrical Propulsion Voltage

- ¥ **Speed & length drive voltage requirement**
- ¥ **Scenario**

D V = v **L**x**B** = 9500 x 10,000 m x 0.7E-4 T = 6650 Volts

just to prevent current back-flow

D Also, P = IV = I (v **L**x**B** + R)

where I v **L**x**B** = mechanical power and IR = resistance loss

¥ I v **L**x**B** = 222 kW (avg) to provide propulsive power

¥ I = 222kW/(v **L**x**B**) ~ 33.4 Amps at 10 km tether length

¥ V = 6650 / 0.23 = 28,800 V to overcome resistive loss

Reel Subsystem

Potential Reeling Functions

Reeling out at high speed during rendezvous/capture phase, then reeling back in more slowly

"Pumping" the tether to modify the orbit after reboost (increase eccentricity and turn excess perigee altitude into more apogee altitude)

After Facility deployment, use reeling to spin up Facility and enter operational stage

Reeling to allow long time storage of tether/grapple and Facility repair/servicing

At current design stage, more concerned with reeling in/pumping impacts on tether deployment and control system on Control Station design

- Expect Winch to have to overcome high forces to begin reeling tether once it's fully deployed, resulting in high electrical power requirements similar to those of the electrodynamic tether reboost power req'mnts (flywheel/motor solution proposed, others?)
- Expect tether dynamics induced by pumping to be harder to damp/control than those that exist when deploying tether slowly or reeling in slowly

Derived Tether Deployer Requirements

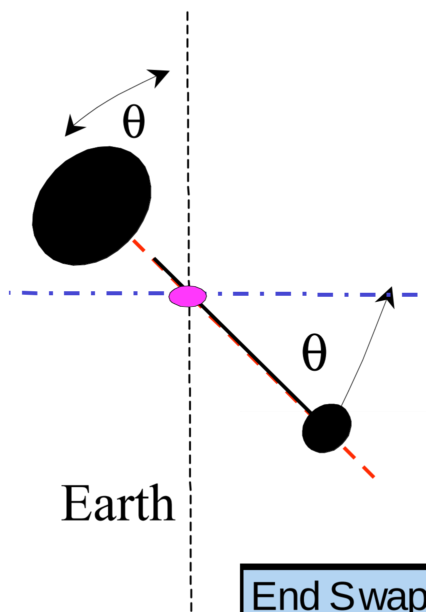
Tether kept at constant tension when pulling off reel/running through boom guidepost

Braking ability required (space experiments' lessons learned)

desire controllability of breaking application to reduce waste heat generation/tether wear and have smooth tether deployment

after some length of tether has been reeled in, the brake must resist torsional force wanting to unreel tether

Preliminary Reeling Analysis

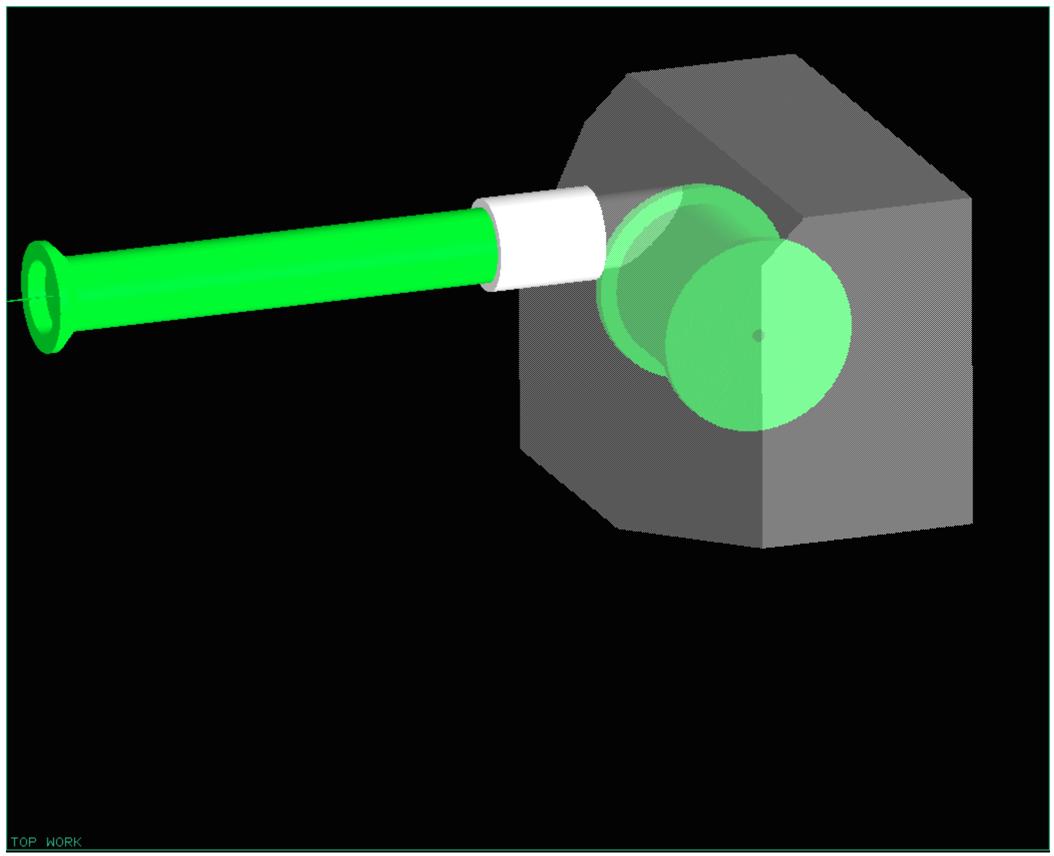


Assumptions

- tether = massless rod
- tether completely deployed
- gravitational forces normal to tether axis neglected

	Optimum Case #1	Less Optimum Case #1	Reboost Case, 4/17/00
End Swapping, Maximum Tension (N)	524,627	319,697	325,814
Despun Station, Maximum Tension (N)	7,655	7,943	9,524
Gravity Gradient Only, Max Tension (N)	2,152	2,154	2,218
For reel rate = 1 m/s, estimate electrical motor efficiency= **0.9**			
estimate electrical power to mechanical work efficiency= **0.4**			
End Swapping Reeling Power ("in") Estimate (kW)	1,457	888	905
Despun Station Reeling Power ("in") Estimate (kW)	21	22	26
Gravity Gradient Only, Reeling Power in (kW)	6	6	6

Reel Subsystem Concept



Tether Boost Facility
System Requirements Document

5 May, 2000

Moon & Mars Orbiting Spinning Tether Transport (MMOSTT)
LEO Tether Boost Facility
System Requirements Document

Prepared for Tethers Unlimited, Inc.

by

Boeing Space & Communications Group

Table of Contents

1 Scope

The MMOSTT system is an in-space transportation system that incorporates an Earth-orbiting facility with a spinning tether as a primary element. This document describes the top-level system requirements for a tether boost facility in low Earth orbit. The requirements defined here are for the objective system, that is, for a facility that is part of an operational full-scale transportation system that boosts payloads from low Earth orbit to higher orbits or Earth escape. Requirements for sub-scale or demonstration systems may be described at a future date in other documents. Requirements for tether facilities in higher Earth orbits or in orbits about other planetary bodies may be described at a future date in other documents.

The tether facility consists of the tether, a control station, and a grapple assembly. The control station controls the tether and tether dynamics. The grapple assembly captures and releases payloads.

A payload is any object that will be accelerated or decelerated toward a destination in space. The transportation system includes a payload accommodation assembly to provide the interface between the payload and the grapple assembly on the tether facility.

This document describes requirements including a nominal payload mass and a nominal release orbit. It is envisioned that the tether facility will be able to grapple payloads with greater or lesser mass than nominal and release them after imparting less or more delta-v, respectively, than nominal. However, no requirements are imposed on the delta-v that must be imparted to non-nominal payloads, except that the system can deliver a smaller payload to the same release orbit as a nominal payload. The tether facility is envisioned to accommodate modular assembly and growth, so its capacity may be expanded to handle larger payloads if a need arises.

2 References

This section lists documents to which the text of this document refers.

Ariane 5 User's Guide

Delta IV Payload Planners Guide, October 1999

Cislunar Tether Transport System, Phase One Final Report, May 30, 1999, Tethers Unlimited, Inc., NIAC Contract 07600-011.

Cislunar Tether Transport System, Phase Two Proposal, May 28, 1999, in response to NIAC CP99-01.

3 System Requirements

3.1 System Performance

3.1.1 Payload Mass

The nominal payload mass shall be 5000 kg.

3.1.2 Nominal Pickup Orbit

The system shall pick up nominal payloads from an equatorial circular orbit at 300 km altitude above the Earth.

3.1.3 Nominal Release Orbit

The system shall release nominal payloads into Geosynchronous Transfer Orbit (GTO).

The change in velocity (delta-v) from the nominal pickup orbit to GTO is 2460 m/s.

3.1.4 Release Orbit Insertion Error

The release orbit insertion error shall be no greater than the release orbit insertion error of Ariane 5 or Delta 4, whichever is smaller, in each of the seven orbital elements.

3.1.5 Payload Interfaces & Accommodation

Payload interfaces and accommodations shall be compatible with any payload designed to be compatible with Ariane 5 or Delta 4.

3.1.6 Payload Environment

The payload environment shall be compatible with any payload designed to be compatible with Ariane 5 or Delta 4.

3.1.7 Turnaround Time

The maximum system turnaround time between deliveries of nominal payloads shall be 30 days.

3.2 System Durability, Reliability, Maintainability, and Safety

3.2.1 Design Life

The design life of the system shall be 10 years with 99% confidence.

This requirement does not mean that no system elements or components will be expended, repaired, or replaced in 10 years. For example, the Payload Accommodation Assembly (PAA) for GTO payloads may be expendable.

3.2.1.1 Self-entanglement

The system shall avoid entanglement of the tether with any element of the system.

3.2.1.1.1 Tether Dynamics

The system shall measure and control tether dynamics.

3.2.1.2 Debris Tolerance

The system shall survive orbital debris impacts for 10 years with greater than 99% confidence.

3.2.2 Maximum Lifetime

Consumables and expendables shall not constrain the lifetime of the system.

The system will not preclude resupply or addition of consumables and expendables, even after the design lifetime has been exceeded. The system may continue to operate beyond its design lifetime with degraded performance. It should not become inoperable simply because the system was not designed to allow some consumable or expendable item, e.g. propellant, to be restocked.

3.2.3 Repairability

Orbiting elements of the system shall be repairable on orbit.

3.2.4 Evolvability

The system shall be growable and evolvable on orbit to deliver a payload ten times more massive than the nominal payload to the nominal release orbit.

3.2.5 Disposal

The system shall provide for safe disposal of all system elements.

3.2.6 Fail Operational

The system shall operate after any one credible component failure.

3.2.7 Two-Failure Safety

The system shall be safe after any two credible component failures.

3.2.8 Collision Avoidance

3.2.8.1 Tracked Debris

Orbiting elements of the system shall avoid collision with tracked debris that would diminish the system's ability to perform its mission.

The system may be designed to shield against or otherwise survive some debris impacts, thereby making avoidance unnecessary in some cases.

3.2.8.2 Tracked Satellites

Orbiting elements of the system shall avoid collision with tracked satellites.

3.2.8.3 Manned Spacecraft

Orbiting elements of the system shall avoid collision with human-occupied spacecraft.

This is a safety requirement, and therefore must be satisfied after any two credible component failures.

3.2.9 Operational Orbital Lifetime

The minimum operational orbital lifetime of orbiting elements shall be 15 days.

Orbital lifetime is the time from loss of control until an orbiting asset becomes so deeply snared by the atmosphere that re-entry becomes unavoidable. Operational orbital lifetime refers to the orbital lifetime of an asset that has reached its operational orbit. The orbital lifetime may be lower while the asset is in an assembly orbit or deployment orbit.

3.2.10 Payload Pickup Reliability

The system shall pick up a nominal payload from a nominal pickup orbit with better than 99% reliability.

The sum of Error (99.5%, grapple position, attitude, and velocity vector) plus Error (99.5%, PAA position, attitude, and velocity vector) will be smaller than the operating envelope of the grapple mechanism.

This is equivalent to saying the system must control state vectors of elements well enough that when the system reports it has placed the payload within the grapple envelope, then the operator can have 99% confidence that the payload is actually within the grapple envelope.

3.3 Communications, Control, Sensing, & Telemetry

3.3.1 Communicate Mission Readiness

The system shall assess and communicate its health and status to the operator in sufficient detail to enable a determination of mission readiness prior to payload launch.

3.3.2 Communicate Status

Each system element shall communicate its position using a common time reference, its health, and its status.

3.3.3 Automated Mission Profile

The system shall provide a capability to automatically produce mission profiles.

3.4 Deployment

3.4.1 Launch Vehicle

Orbiting elements of the system shall be capable of being launched on an existing launch vehicle.

Existing means that the selected launch vehicle is credibly expected to be operational when the tether system is ready for deployment.

3.4.2 First Launch Capability

The first launch segment shall provide some operational capability.

Some operational capability means a non-zero delta-v is added to a non-zero payload mass.

4 Ground Rules & Assumptions

This section defines ground rules and assumptions to be used in concept definition and assessment. It also provides guidance for element-level requirements.

4.1 Safety Factor

The structural safety factor shall be two times the highest expected stress.

Where the maximum stress is not known, the structure shall be designed for three times the nominal steady-state stress.

4.2 Economic Analysis Window

The economic analysis window shall be 10 years from the 1^{st} launch after full-up system authority to proceed.

4.3 Power Source

The primary power source for long-term orbiting elements shall be solar. On-board energy storage is permitted.

4.4 Primary Propulsion

Electrodynamic propulsion shall be the primary mode of propulsion for the LEO Tether Facility.

4.5 PAA Role in Rendezvous & Grapple

The PAA shall cooperate in meeting the grapple.

This could include position and speed control, or just attitude control, or just transmission of navigation data, or (conceivably) no action.

4.6 Tether Control Station Reeling Rate

If the Tether Control Station reels in the tether, the reeling rate shall not be required to exceed 2 m/s.

4.7 Tether Control Station Thermal Control

The Tether Control Station shall use passive thermal control.

5 Terminology

GEO – Geostationary Earth Orbit

Grapple Assembly - End mass that captures, releases, and interfaces with payloads.

Ground Station - Provides system control interface.

GTO – Geostationary Transfer Orbit

LEO – Low Earth Orbit

LLO - Low Lunar Orbit

LMO - Low Mars Orbit

MMOSTT – Moon Mars Orbiting Spinning Tether Transport

PAA – Payload Accommodation Assembly

Payload - Any useful object that will be accelerated or decelerated toward a new trajectory in space.

Payload Accommodation Assembly - System to provide the interface between the payload and the Grapple Assembly.

Tether - Flexible connector between major elements of system

Tether Control Station - Facility that controls the tether and tether system dynamics. Contains all of the control hardware and subsystems required. Located in LEO.

Tether Boost Facility Design Study

Final Report

Subcontract No. MMOSTT-01

Submitted to

Tethers Unlimited, Inc.
19011 36th Ave W., Suite F
Lynnwood, WA 98036-5752

by

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5301 Bolsa Avenue
Huntington Beach, California 92647

April 30, 2001

Tether Boost Facility Design Study
Final Report

Table of Contents

1 Summary and Introduction

The Boeing Company is pleased to submit this Tether Boost Facility Final Report to Tethers Unlimited, Inc., Lynnwood, Washington, in compliance with Subcontract Number MMOSTT-01, in support of TUI's prime contract from Universities Space Research Association (USRA), sponsored by the NASA Institute for Advanced Concepts, entitled Moon and Mars Orbiting Spinning Tether Transport (MMOSTT) study. The report presents results generated during the contract performance period from December 1, 1999 through April 30, 2001.

Architectures were defined by TUI during Phase I that used momentum exchange tethers to boost payloads from LEO to the Moon and Mars. The Phase II program has focused on defining an initial capability to boost payloads from LEO to GTO. Preliminary mission requirements were furnished by TUI to Boeing at the beginning of the contract.

Boeing derived preliminary system requirements for a low Earth orbit (LEO) tether facility, capable of boosting a 2.5 MT payload from LEO to GTO. The Tether Boost Facility System Requirements Document (SRD) is included as an Appendix to this report.

A LEO tether boost facility was then designed, based on the SRD, for a near-term system to boost payloads from LEO to GTO. Although the facility is intended for use in LEO, system modifications can be implemented for use in other orbits. Subsystem concepts and mass allocations have been identified. Although the system will be designed for unmanned spacecraft, but will consider the incorporation of provisions to allow future modification to accommodate manned transportation.

Technology readiness was evaluated for all system elements and subsystems of the MMOSTT architecture. Critical technology issues were identified, and technology demonstrations to address those issues have been defined. A top-level program development plan was developed, which shows an Initial Operational Capability is achievable by 2014 by conducting critical technology demonstrations within the next two to three years

Boeing, with TUI as a team member and subcontractor, is also a USRA contractor for the Hypersonic Airplane Space Tether Orbital Launch (HASTOL) study, which investigates an architecture to deliver a payload from the earth's surface to orbit. HASTOL and MMOSTT are synergistic, and investment in near-term follow-on technology developments will benefit the maturity and risk mitigation of both architecture concept.

Acknowledgements

The Boeing Program Manager for this subcontract, as well as other tether applications programs, is Mr. John Grant, Boeing Phantom Works, Huntington Beach, California, e-mail john.e.grant@boeing.com, telephone (714) 372-5391. Contracts Manager is Ms. Kimberly Harris, Boeing Phantom Works, Huntington Beach, California, e-mail Kimberly.Harris@West.Boeing.com, telephone (714) 372-2681.

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2 Tether Boost Facility System Requirements

2.1 Overview of Mission Requirements

The MMOSTT system is intended to become part of a commercially viable enterprise to transfer payloads between various pairs of orbits. Requirements for the MMOSTT system are focused on making the system commercially viable. Commercial viability means the system must meet customer needs and must be financially, politically, legally, and technically feasible. Customer needs drive mission requirements for payload mass, destination orbit, release orbit precision, and reliability, as well as system requirements for payload interfaces and payload environment. We chose to focus on customers whose need is to deliver commercial comsats to GTO. Initial requirements for GTO delivery missions were defined in "Tether Boost Facility Mission Requirements Specification" by Dr. Rob Hoyt on November 17, 1999. Mission requirements were refined and the initial system requirements were defined during a meeting of the MMOSTT team and NASA MSFC personnel in March 2000. The system requirements were further refined during a MMOSTT technical interchange meeting in May 2000.

Table 1 summarizes the current top-level mission requirements for the tether boost facility. For the near future, comsat GTO packages will have about 5000 kg mass. Comsat launch customers usually wish to avoid dependence on a single launch vendor, so we require MMOSTT to easily accommodate satellites designed to fly on other launch vehicles, Delta 4 and Ariane 5. That is, a payload designed to fly on either of those launchers should be able to fly on MMOSTT with no modification and no loss of capability. This appears as two mission requirements: one that the orbit insertion error for the release payload should be no greater than the insertion error for other common launch vehicles, and another that the payload's environment, e.g. acceleration levels, should not be more stressing than the environment aboard those vehicles. (Note that responsibility for the release orbit error may be shared among the tether boost facility and the payload accommodation assembly. Allocating error budget to these elements will be accomplished in later work.) Customer need drives the requirement for payload pickup reliability of at least 99%.

Financial feasibility drives requirements for turnaround time and mission lifetime. A financially successful project must earn enough revenue to recover startup costs and operating costs plus a healthy annualized return on investment. The mission requirement for 30-day turnaround time matches the rate of GTO launches in the addressable market. This permits MMOSTT to earn revenues as fast as the market will bear. The ten-year minimum life requirement means the system will operate long enough to recover startup costs and earn a profit. The unconstrained maximum lifetime and the requirement for growth to larger payloads make the system likely to have an extended life as a cash cow, responding to changing customer needs and earning revenues after startup costs are fully paid off.

Political and legal feasibility drive the mission requirement for avoiding collisions with other spacecraft. U.S. and international organizations would take political or legal action to block the launch of a system that would pose a hazard to other spacecraft or to people on Earth.

Requirement Name	Value
Payload Mass	5000 kg at IOC, can grow to follow market
Pickup orbit	300 km equatorial
Release orbit	GTO
Release insertion error	< Delta IV/Ariane 5
Payload environment	< Delta IV/Ariane 5
Turnaround time	30 days
Mission life	10 years +
Collision avoidance	100% of tracked spacecraft
Operational orbit lifetime	15 days
Payload pickup reliability	99%

Table 1.1. Top-Level Mission Requirements.

Technical feasibility drives mission requirements for nominal pickup orbit and operational orbital lifetime. The 300 km circular pickup orbit is high enough that drag will not greatly complicate the rendezvous calculations, nor is the payload in danger of de-orbiting quickly if the tether misses one or two grapple attempts, but it is low enough to be relatively inexpensive for launch vehicles to reach. The 15-day operational orbital lifetime for the tether gives a reasonable amount of time for operators to debug and correct problems if a failure makes the system temporarily unable to reboost itself.

2.2 Overview of System Requirements for Tether Boost Facility

System requirements are derived from mission requirements. At the top level, many system requirements for the MMOSTT boost facility are identical to mission requirements. Not all are, however. Table 2 summarizes the current top-level system requirements. Some requirements, e.g. payload mass, are key design drivers and had a strong influence on Phase Two work. Some others, such as control of atomic oxygen erosion, are less critical as design drivers but are included in our technology development plans. Still other requirements, e.g. payload release orbit error, were not believed to be key design drivers or technical feasibility challenges and therefore were not addressed in Phase Two work. Table 2 summarizes the impact of each requirement on the current design and how each will be addressed in the follow-on technology validation program. Below we describe system requirements that were not described earlier as mission requirements. These are marked with asterisks in Table 2.

Payload interfaces must be compatible with satellites designed to fly on Delta 4 and Ariane 5. This completes the compatibility criterion that allows customers to avoid dependence on a single launch vendor. Not only must the mission be compatible with payloads for those vehicles, but the interfaces to the payload must also be compatible. (Interfacing to the payload will be the primary function of the payload accommodation assembly, but some payload interfaces might touch the tether boost facility.)

The requirement for a ten-plus year design life means a single tether boost facility will operate for at least ten years. An alternative we considered was to deploy multiple short-lived facilities over the ten-year required mission life; but the high cost of hardware procurement and launch makes a long-life facility preferable. The requirement that maximum lifetime should not be limited by consumables and expendables means the system must permit any consumables or expendables to be resupplied in orbit. It led to a design goal of avoiding all consumables and expendables. Our current design satisfies that goal, though further analysis is needed to see whether the facility can maintain adequate attitude control without thrusters. Like the lifetime requirements, the requirements for evolvability and one-failure operation also support the financial goal of letting the system earn revenues for as long as possible. Disposal and two-failure safety requirements address political and legal feasibility.

Automatic production of mission profiles increases startup costs, but is likely to be a good investment in the long run due to reduced operating costs. The requirement for launch on existing vehicles also supports financial feasibility by avoiding the cost of developing a new launcher. The requirement for some capability after the first launch allows the system to begin earning revenues sooner, which improves the financial feasibility. Our payload requirement of 2500 kg to GTO for the first-launch system does not mean the system can handle only payloads of 2500 kg or less. A system of that size can handle the more common and lucrative 5000 kg payloads, though it cannot transfer one all the way to GTO from a 300 km circular orbit every 30 days. Some additional propulsion would be needed until the second launch increases GTO capacity to 5000 kg.

Technical feasibility drives the requirements for communication functions. Communication of readiness and status is necessary to ensure both that the system executes missions properly and that faults can be detected, diagnosed, and corrected by operators on the ground.

The Moon & Mars Orbiting Spinning Tether Transport (MMOSTT) LEO Tether Boost Facility ystem Requirements Document is included in this report as Appendix L-1.

Requirement Name	Value	Phase Two Design Impact	Technology Development Impact
Payload Mass	5000 kg at IOC	Facility mass, tether mass	N/A
Pickup orbit	300 km equatorial	Facility orbit, atomic oxygen, orbital decay rate	Erosion control
Release orbit	GTO	Facility mass, tether mass	Demonstrate release speed
Release insertion error	< Delta IV/Ariane 5	N/A	Demonstrate
Payload interfaces*	Match Delta & Ariane	N/A	N/A
Payload environment	< Delta IV/Ariane 5	PAA, tether length	N/A
Turnaround time	30 days	Power system sizing	Demonstrate high power ED thrust
Design life	10 years minimum	PV array life, battery depth of discharge, Hoytether, modularity	Erosion control, impact survivability under full tension
Maximum lifetime*	Not constrained by consumables and expendables	Minimal use of consumables; on-orbit restocking where needed	
Repairability*	All elements repairable on orbit	N/A	ISS experience will increase repair TRL
Evolvability*	10 x payload mass growth beyond IOC	Modularity	
Disposal*	Safe disposal of all system elements	N/A	
Fail Operational*	Operate after any one credible failure	Redundancy levels	
Two-Failure Safety*	Safe after any two credible failures	Redundancy levels	
Collision avoidance	100% of tracked spacecraft	N/A	Demonstrate timely avoidance maneuvers
Operational orbit lifetime	15 days	Orbit constraints	Verify ED thrust can rise from high-drag orbit
Payload pickup reliability	99%	N/A	Demonstrate
Communicate mission readiness*			
Communicate status*			
Produce automated mission profile*			
Launch on existing launch vehicle*			
Some capability after first launch package*	2500 kg to GTO		

Table 2.2. Top-Level System Requirements for Tether Boost Facility

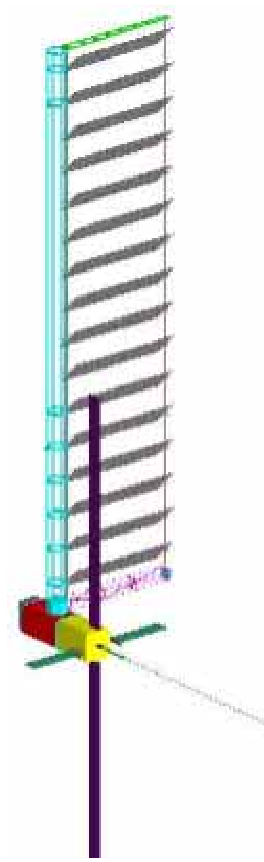
3 Boost Facility

Section 3.2 summarizes the current design features of the first launch of a partial capability Control Station. The Control Station (along with grapple and tether) is within mass budgets for launch on a Delta-IV Heavy Launch Vehicle, allowing for an initial operational capability with a single launch mission. Primary features of the system are use of PV concentrator solar arrays, Lithium-Ion batteries, CMGs for attitude maneuver and control, GPS/INS for guidance and navigation input, and a tether subsystem that will support a 100 Km, 300 kW electrodynamic tether.

A detailed description of the tether boost facility is inlcuded in this report as Appendix L-2.

3.1 First Facility Launched

- Control Station mass = 13,267 kg (includes 21% mass margin)
- Operational mass = 23,358 kg, no margin CS w/PAF
- GLOW = 19,891 kg with 15% margin, no PAF



3.2 Features

EPS

- *Scarlet*-like concentrator PV arrays, 563 square meters
- Standard, state-of-the-art PV array drive motors
- State-of-the-art power management and distribution except for electrodynamic tether subsystem
- Lithium-ion battery power storage system
- 5,410 kg (includes 14% mass growth margin)

Communication Subsystem

- Downlink communication with ground station(s) and communication with Grapple Assembly and PAA (via Tether Facility Network)
- State-of-the-art, COTS hardware (antennae/transceivers)
- Dual redundancy
- 4.2 kg (includes 16% mass growth margin)

C&DH

- State-of-the-art, COTS hardware
- Dual redundancy
- 29 kg (includes 13% mass growth margin)

ADCS/GN&C

- 2 Control Moment Gyros (no redundancy), each assumed half size of a Skylab CMG
- 2 sun sensors
- 2 inertial navigation unit
- GPS antennae (3)/tranceivers (2)

- 213.8 kg (includes 6% mass margin)

Electrodynamic Tether Subsystem

- Sized for 80 km conductive tether, total length 100 km, 300,000 W, 40 μN/W thrust efficiency
- Control Subsystem with 1m diameter, 1.5m long reel, motor, tether guides, power conversion, FEACs (field emitter array cathodes)
- 1,933 kg (includes 36% mass margin)

4 Technology Readiness Assessment

4.1 Assessment

Technology readiness of each MMOSTT subsystem was assessed using NASA's TRL scale, as shown in Table 1. TRL scores were developed for today's technology and for two future dates: 2005, the earliest feasible year in which MMOSTT could move to full-scale engineering development, and 2010, a more likely date for beginning full-scale development following a more reasonably paced technology development program. Tables 2 through 5 show the scores for the subsystems of each MMOSTT element (control station, tether, grapple assembly, and payload accommodation assembly). Where TRL is predicted to increase over the 2001-2010 interval, the table includes a reference to the planned or ongoing activity that will raise the technology level. Table 6 shows TRL scores for some challenges that arise from integrating the elements into a single system.

	Technology Readiness Levels (TRL)
9	Actual system "flight proven" through successful mission operations.
8	Actual System completed and "flight qualified" through test and demonstration.
7	System prototype demonstration in an operational environment.
6	System/subsystem model or prototype demonstration in a relevant environment.
5	Component and/or breadboard validation in a relevant environment.
4	Component and/or breadboard validation in a laboratory environment.
3	Analytical and experimental critical function and/or characteristic proof-of-concept.
2	Technology concept and/or application formulated.
1	Basic principles observed and reported.

Table 4.1. Definitions of NASA Technology Readiness Levels

Many subsystems or components of MMOSTT are easily within today's technology. Communications, computing hardware, most structures, and many of the sensors will require little or no improvement. TRLs for these subsystems range from 6 to 8. TRL 6 means the technology is ready to be implemented in a component for the objective system. TRL 8 means a working product exists that could be plugged directly into the objective system with no development effort.

Some subsystems of MMOSTT will face requirements that are similar in quality but much greater in quantity than today's systems. Examples are electric power management, power conversion, and passive thermal control. Each of these must handle many hundreds of kilowatts of power - an order of magnitude increase above current technology. TRLs for these subsystems are typically 4 or 5. The power conversion system for the electrodynamic tether faces the additional challenge of working with unusually high voltage as well as high power.

The flight control subsystems of various MMOSTT elements face qualitatively different requirements than previous flight control systems. Previous spacecraft have not had to control their attitude or position while attached to a long, flexible tether that exerts considerable force and is as massive as the spacecraft. Orbital mechanics codes must contend with a spacecraft whose mass is so widely distributed that orbital speed varies appreciably as the system rotates. The magnitude and direction of electrodynamic thrust vary as the local magnetic field and plasma density change, so trajectory planning requires more flexibility than systems that use well-defined rocket firings. All of these challenges can be solved, but they have not been solved and tested in flight, yet the TRL for flight control is 3.

Another area with qualitatively new requirements is erosion protection for the tether itself. The preferred tether structural material is Spectra 2000. Tests at MSFC suggest that Spectra is susceptible to rapid erosion by atomic oxygen. (and degradation by UV?) Coatings protect other spacecraft surfaces from atomic oxygen and UV, but unlike those surfaces, the tether will stretch and bend. It is unknown whether any extant coatings will adequately adhere to Spectra through many stretching cycles, and if so, how they might affect the overall strength to mass ratio of the tether system. Erosion protection for the high-voltage insulation around an ED tether faces similar issues and is similarly undefined. Erosion protection for both the tether structure and the current-carrying component get a TRL of 2.

Two MMOSTT subsystems are defined only at the functional level with no design yet defined to implement the functions. These are the grapple mechanism and the collision avoidance subsystem. Both currently rate a TRL of 2. The grapple concept will be defined by the on-going HASTOL Phase Two contract. The HASTOL team is currently developing a rendezvous and capture simulation that will define requirements for a grapple system. Once grapple requirements are known, the HASTOL team will define a grapple concept, bringing the grapple TRL to 3. The collision avoidance subsystem covers ground elements as well as flight elements, but the key challenge is a flight issue: maneuvering the tether system to avoid a predicted collision or close approach. Current spacecraft have only one reasonable approach to collision avoidance: change the spacecraft's trajectory to provide a safe miss distance. In most cases, a tether can use at least four approaches: change trajectory, change rotation rate or phase, change tether attitude (e.g. change libration angle or phase), or change tether shape (e.g. induce a bending motion different from the bend due to ED thrust.) Beyond defining these options, no development has yet occurred in collision avoidance technology for tethers.

CONTROL STATION	**TRL 2001**	**TRL 2005**	**TRL 2010**	**Comments**
Communications Subsystem (CS)	7	7	7	
- Comm net Antenna, Transmitter, Receiver	7	7	7	
- Downlink Antenna, Tx, Rcvr	7	7	7	
Attitude and Location Determination /Control Subsystem (ALDCS)	**5**	**5**	**5**	
- GPS Antenna, Receiver	7	7	7	
- Attitude Stabilization H/W (w/ CMGs)	6	6	6	ISS CMGs are w/in an order of magnitude of right size
Software	5	5	5	TSS **probably** used relevant algorithms in flight - check that
Electrodyn. Tether Subsystem	**4**	**5**	**6**	
- Power Convertor (high voltage/high power), Power Controller	5	5	6	High power: ISS, high voltage: electric propulsion = Today's TRL. Space Based Radar = 2010 TRL.
-FEAC Plasma Contactor	4	5	6	FEAC under development at U Michigan and JPL. TRLs for 2005 and 2010 assume those programs proceed.
- Tether Dynamics Control System	4	7	8	ProSEDS/Terminator Tether/mPET/MIR for 2005 TRL; MIR for TRL 2010
Mechanical Subsystem (MS)	**7**	**7**	**7**	
- Facility Structures	7	7	7	No unusual structural challenges
- Erosion Protection	7	7	7	ISS system
- Micrometeoroid Protection	7	7	7	ISS system
Thermal Control Subsystem (TCS)	**7**	**7**	**7**	
- Temp Sensors	7	7	7	
- Heaters	7	7	7	
- Radiators, Passive Heat Transfer Path	4	TBD	TBD	Much higher power than has been done with passive systems, and we're assuming pretty high performance
Electrical Power Subsystem (EPS)				
- Solar Collectors/Drive Motors	6	6	6	SCARLET arrays = Today's TRL, but we'll run them at non-zero g which imposes some challenges
- Sun Angle Sensor	7	7	7	

Table 4.2. TRL assessments for Control Station subsystems

- Pwr Mgt Unit	5	6	6	10x more power than anything that's been done. NRA 8-30 will reach TRL 6 by 2005.
- EPS Control Processor				
- Battery Charger				
- Power Regulator				
- Power Distributor				
- Batteries	6	7	8	Rechargeable lithium batteries have been flown
- Software	5	6	6	
Cmd and Data Handling	**6**	**6**	**6**	**some autonomy w/ ground assist**
- Computer				
- TLM Mux/Demux				
- Software				
- Operating System				
Flight Control Software	**3**	**3**	**3**	**Hoyt's models include simple control**
- Orbital Mechanics	3	3	3	distributed mass and ED thrust complicate this
- Equipment Control	3	3	3	non-standard equipment
Networks Subsystem (NS)	**8**	**8**	**8**	
- Power Cables				
- Data Cables				
Retrieval,Deployment and Spin Control				**need Hoyt's input on these items - most at very low TRL**
- Winch, winch motor, traction drive, motors and controller	6	6	6	TSS = Today's TRL
- Tether cutter	7			
- Tether deploy speed sensor				
- Tether deployed length sensor				
- Tether fully deployed sensor				
- Tether tension sensor				
- Tether impact detector				
- Separated tether detector				
- Tether departure angle sensor				
- Tether position sensor				

Table 4.2 (cont'd). TRL assessments for Control Station subsystems

	TRL 2001	TRL 2005	TRL 2010	Comments
GRAPPLE ASSEMBLY	2	3	3	**Concept undefined. HASTOL aims for TRL 3.**
Communications Subsystem (CS)				
- Comm net Antenna, Transmitter, Receiver	7	8		
Attitude and Location Determination /Control Subsystem (ALDCS)				
- GPS Antenna, Receiver				
- Attitude Stabilization				tether changes the situation from previous attitude control problems
Software				
Mechanical Subsystem (MS)				
- Facility Structures				
- Erosion Protection				
- Micrometeoroid Protection				
Thermal Control Subsystem (TCS)	7	7	7	
- Temp Sensors				
- Heaters				
- MLI, Radiators				
Electrical Power Subsystem (EPS)				
- Solar Collectors	5	7		Current effort on quad-junction cells = 2005 TRL
- Sun Angle Sensor	8			
- Pwr Mgt Unit				
- EPS Control Processor				
- Battery Charger				
- Power Regulator				
- Power Distributor				
- Batteries	7	7	8	
- Software				
Cmd and Data Handling	6	6	6	
- Computer				
- TLM Mux/Demux				
- Software				
- Operating System				
Flight Control Software	**2**	**2**	**2**	
- Orbital Mechanics	2	3		tether complicates situation
- Equipment Control	2	3		equipment undefined

Table 4.3. TRL assessments for Grapple Assembly subsystems

Networks Subsystem (NS)	8	8	8	
- Power Cables				
- Data Cables				
Proximity Sensing	2	3	3	requirements unknown
LIDAR	4	7		STS demos = 2005 TRL
Radar	7			
P/L Capture/Release Device	2	3		HASTOL Phase II = 2005 TRL
Differential GPS (Beacon)	4	7		ProSEDS success = 2005 TRL

Table 4.3 (cont'd). TRL assessments for Grapple Assembly subsystems

	TRL 2001	**TRL 2005**	**TRL 2010**	**Comments**
PAYLOAD ACCOMMODATION ASSEMBLY	2	3	3	**same as grapple: low TRL, concept TBD**
Communications Subsystem (CS)	7			
- Comm net Antenna, Transmitter, Receiver	7	8	8	
Attitude and Location Determination /Control Subsystem (ALDCS)	**2**	**3**		
- GPS Antenna, Receiver				
- Attitude Stabilization				
Software				
Mechanical Subsystem (MS)				
- Facility Structures				
- Erosion Protection				
- Micrometeoroid Protection				
Thermal Control Subsystem (TCS)	7	7	7	
- Temp Sensors				
- Heaters				
- MLI, Radiators				
Electrical Power Subsystem (EPS)				
- Sun Angle Sensor	8	8	8	
- Pwr Mgt Unit				
- EPS Control Processor				
- Power Regulator				
- Power Distributor				
- Batteries	8	8	8	
- Software				

Table 4.4. TRL assessments for Payload Accommodation Assembly subsystems

Cmd and Data Handling	6	6	6	
- Computer				
- TLM Mux/Demux				
- Software				
- Operating System				
Flight Control Software	**2**	**3**	**3**	**HASTOL Phase II = 2005 TRL**
- Orbital Mechanics				straightforward
- Equipment Control				equipment undefined
Networks Subsystem (NS)	8	8	8	
- Power Cables				
- Data Cables				
Proximity Sensing				requirements unknown
LIDAR	4	7	7	STS demos = 2005 TRL
Radar	7	7	7	
Passive Reflectors				requirements unknown
Docking Adaptor	2	3	3	HASTOL Phase II = 2005 TRL
Differential GPS (Beacon)	4	7	7	ProSEDS success = 2005 TRL

Table 4.4 (cont'd). TRL assessments for Payload Accommodation Assembly subsystems

TETHER	**TRL 2001**	**TRL 2005**	**TRL 2010**	**Comments**
Mechanical Subsystem (MS)				
- Survivable Tether Structure	4	5	5	Terminator Tether = 2005 TRL
- Tether Material(s)	7	7	7	TiPs/SEDS = Today's TRL
- Erosion Protection (AO **& UV**)	2?	2?	2?	MSFC tests show serious issue of long-term Spectra survival. No coating identified yet.*
Electrodynamic Tether				
- Bare Wire Anode	5	6	6	ProSEDS = 2005 TRL
- High Voltage Insulation	4?	4?	4?	incl. debris impact countermeasures
- Erosion Protection (AO **& UV**)	2?	2?	2?	Do we know about erosion of high-voltage insulating material?

*Possible solution is to use higher altitude for payload pickup, thereby reducing AO exposure

Table 4.5. TRL assessments for Tether subsystems

INTEGRATION	TRL 2001	TRL 2005	TRL 2010	Comments
Software - Algorithms	3	4		HASTOL Phase II = 2005 TRL
Collision Avoidance	2			Avoidance incl. change orbit, change rotation, change tether attitude, change tether shape. Only the first has been done, and that wasn't with ED thrust or tether reeling.
System	2			

Table 4.6. TRL assessments for MMOSTT Integration technologies

4.2 Technology Needs

Some of the technologies needed for MMOSTT are likely to be developed for other space applications. High-power electrical systems and heat rejection systems are the subject of a new technology project sponsored by the Air Force Research Lab. Development of Field Emitter Array Cathodes, our preferred choice for plasma contact technology, is underway at JPL. It's unclear whether that effort will succeed and whether it will be demonstrated in flight.

Areas where technology development must be planned specifically for a tether system include:

Rendezvous and capture. Simulation work in the HASTOL program will raise this technology to TRL 3. Hardware tests and demonstrations are needed for higher levels. A thorough flight test would include a variety of sensors, beacons, and thrust modes. This variety would allow several different R&C techniques to be raised to TRL 5 or 6 and would provide data by which to compare their performance.

Flight control. This technology is especially needed for electrodynamic propulsion. An ED flight control system must contend with variations in the geomagnetic field and in plasma density, factors that make ED thrust much less predictable and less controllable than conventional propulsion. The ED thrust direction depends on the tether's orientation and curvature. Like conventional propulsion, ED thrust can modify the system's trajectory, change its rotation rate or phase, or influence its attitude. In addition, ED thrust will change the tether's shape - an issue not addressed at all by previous flight control methods.

To bring tether flight control to TRL 6 requires flight demonstration. The demonstration should use ED thrust and/or tether pumping to perform controlled changes in all seven orbital elements; in rotation rate and phase; in libration amplitude and phase; and in tether shape (fundamental mode amplitude and phase, at a minimum). It should demonstrate these in an orbit that is eccentric enough to take the tether from a strong thrust region to a zero-thrust region (e.g. 2000 km) in every orbit.

Collision avoidance. Collision avoidance is a demanding application of flight control. All the flight control capabilities listed above may be used. In addition, collision avoidance requires a timely flow of information from a satellite tracking system to a decision making system and then to the flight control system. Raising tether collision avoidance to TRL 6 could be accomplished without performing collision avoidance against real threats. Rather, the satellite tracking system would report realistic simulated threats to the decision making system. The decision system prioritizes each threat, chooses an avoidance strategy, and sends appropriate commands to the flight control system of a real tether in flight. The tether's motion is then monitored to assess whether it would have avoided each threat, had the threat been real.

Erosion protection. A ground-based program is needed to identify and characterize coatings that adhere well to Spectra or to insulation while being stretched, bent, heated, cooled, and exposed to atomic oxygen and ultraviolet. This ground-based effort can reach TRL 4. When promising materials are identified, they would be tested on a pallet attached to Space Station or Shuttle. The pallet would provide bending and stretching forces and temperature control. This experiment would bring erosion protection to TRL 5. A subscale tether with samples returned to Earth for analysis would be needed to reach TRL 6. Such an experiment could be attached to Space Station, Shuttle, or some unmanned re-entry system.

High-voltage, high power conversion. Components of this technology may be separately developed and demonstrated before an integrated, operational high-voltage, high power system is demonstrated. One is thermal control. Heat rejection of several hundreds of kW may be developed by Air Force programs or may be purchased from Russia. A tether-oriented program might do a separate short-duration demo of high-voltage, high power conversion itself. This would run for only a few seconds at a time, demonstrating that the requisite voltage and power can be handled safely. The short run time would prevent heat from building up catastrophically. It would use stored energy, so PV arrays need not provide hundreds of kW. (The energy storage system would have to provide hundreds of kW for a few seconds.) Such a demo done in space could bring the conversion technology by itself to TRL 6. However, an integrated, continuously running system for power conversion would only reach TRL 5 (component validation in a relevant environment) from that demo and the Air Force thermal demo. To reach TRL 6 would require flight demonstration of an integrated, continuously running system.

High-power plasma contact with FEAC. A technology demonstration program for FEACs must address two issues: high-power performance and lifetime. We can define a demo that addresses both issues economically. For high-power performance, the issue is how well the large FEAC array contacts the ionosphere. That can be measured with a run time of a few seconds. A brief test like this could use stored energy and could end quickly enough that heat buildup is not catastrophic. (Obviously this test would be synergistic with the brief power conversion demo outlined above.) To measure lifetime, the same hardware could run for months or years at low power with all the current going through a single FEAC rather than the large, high-power array. This test would use PV arrays for power and would provide modest cooling. The combination of the high-power test and the lifetime test would bring high-power FEAC technology to TRL 6.

5 Near Term Follow-On Programs

We assessed technology development needed to bring the maturity of a tether-based launch architecture to a level comparable to other advanced reusable launch architectures presently being considered for next generation access to space. The technology assessment discussed in Section 4 has been used to define near-term technology development and demonstration programs to bring maturity levels up to at least TRL 6 for all tether architecture systems and subsystems.

Key technologies identified in Section 4 that require near-term focus are listed here in rough order of priority:

- Rendezvous and capture
- Electrodynamic tether operation, propulsion, and flight control
- High-voltage, high power conversion.
- Erosion protection.
- Collision avoidance.
- High-power plasma contact with FEAC
- Guidance algorithms
- Flight mechanics software
- Cooperative sensors
- Autonomous operation
- Limited duration capture window
- Overall System Architecture and Integration
- Ground and flight operations

These key technologies can be combined into the three following near-term technology demonstration programs:

- Ground-Based Rendezvous and Capture Demo
    - Do detailed simulation and analysis of rendezvous and capture.
    - Design an operational grapple
    - Do detailed design of demo hardware

- Sub-Scale Electrodynamic Tether Dynamics Experiments
    - Fly as secondary payload
    - 4-5 km long ED tether; assess tether dynamics, survivability

- Sub-scale tether system to capture and toss payloads in orbit
    - Four phase program:
        - Design
        - Fabrication and ground testing
        - Flight experiment execution at increasing levels of performance
            - 1st, tether is in circular orbit just a little higher than payload and hanging in gravity gradient mode during the rendezvous and capture. Tether and payload rendezvous and capture at low

relative speed. The tether then uses thrust to begin rotating and throw payload.

- 2nd, tether is in a higher elliptic orbit and rotating slowly. It rendezvous with and captures the payload at moderate relative speed. It then rotates a few times and tosses the payload.
- 3rd, rendezvous and capture at maximum rotation rate and high relative speed.

- Enter limited operation for paying customers.

These near-term technology development programs are discussed in subsequent sections of this report, along with rough order of magnitude (ROM) costs for each. Information from this report will be provided to the team working on the NIAC funded HASTOL Phase 2 contract. We anticipate that the HASTOL team will extend and refine these concepts and costs over the next three to five months.

Figure 5.1 presents an overall program development roadmap that captures all past and ongoing tether-related programs, recommended near-term technology development programs, and an optimistic but achievable full scale development and production schedule that leads to an Initial Operational Capability in the second decade of the 21st century.

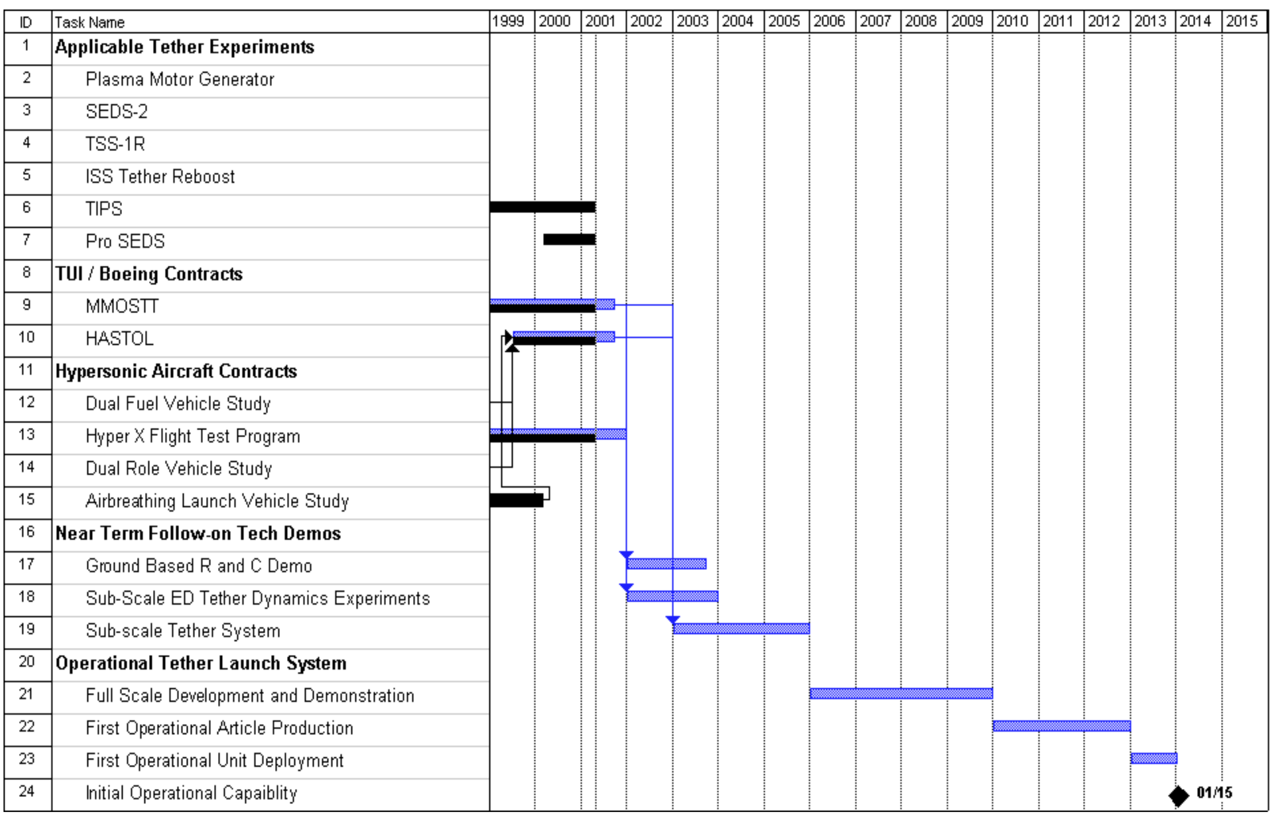


Figure 5.1. Near-term technology demonstrations will enable tether system operational capability in 2014.

5.1 Ground Based Rendezvous and Capture Demonstration

Rendezvous and capture is a technology that needs to be investigated to validate a critical element of the MMOSTT operational concept. Traditional space vehicle rendezvous and capture (R&C) scenarios take place over span times of several minutes to several hours. In the case of a momentum exchange tether architecture, the rendezvous and mating of the tether tip and the payload needs to take place in minutes. There are mere seconds of time during which the tether tip and payload are in close enough proximity to rendezvous and successfully complete a docking operation. This is about two orders of magnitude less time than traditional space docking scenarios.

At first glance, rendezvous and dock of two orbiting space assets in this short amount of time seems unthinkable. However, one needs to recognize that the two assets have very predictable paths, and if designed properly with the appropriate sensor, tracking, communication, and perhaps propulsion subsystems, the two assets can cooperate to accomplish the maneuver. To our knowledge, this scenario timeline has not been investigated, nor have any systems been designed to accomplish rapid rendezvous and dock.

Background. Boeing and TUI are presently studying space tether and payload rendezvous and capture as part of the NIAC funded Hypersonic Airplane Space Tether Orbital Launch (HASTOL) program. HASTOL is a reusable launch architecture that delivers a payload from the earth surface to a sub-orbital altitude, using a hypersonic aircraft, at which point it meets the tip of a rotating space tether. An assembly at the tether tip grapples the payload, and the tether continues to rotate, boosting the payload to orbit. Although the MMOSTT and HASTOL architectures are different, their rendezvous and capture scenarios have many similar characteristics.

The objective of the HASTOL R&C task is to develop and utilize a digital simulation to determine the preliminary R&C requirements for sensor, propulsion, and communication subsystems for each architecture element, which include the grapple, the tether, the delivery aircraft, and the payload. Ideally, the aircraft will be guided, or flown, to the rendezvous point, using it's own navigation algorithms and tether tip and grapple position information that will be downlinked to the aircraft. However, trade studies will be performed to determine if aircraft maneuvering is the most effective way to successfully bring the payload to the tether tip, such as active maneuvering of the grapple. Once the preliminary R&C requirements have been established, a preliminary grapple concept can be developed and the required subsystems can be identified.

At the end of the HASTOL contract, we expect to have preliminary rendezvous and capture requirements defined and preliminary concepts for the grapple, its subsystems, and any subsystems that need to be incorporated to other HASTOL system elements for successful payload and grapple rendezvous and capture. The HASTOL simulation model, grapple, and other system element requirements and subsystems can then be adapted to the MMOSTT architecture, thus serving as a point of departure for more detailed analyses and configuration design in follow-on programs.

Near Term Ground Demonstration Plan. Figure 5.2 is a suggested plan for a ground demonstration of the MMOSTT tether grapple and payload R&C scenario. A 21 month program is envisioned, to be executed in two phases:

- Phase 1
  - Operational rendezvous and capture requirements definition
  - Laboratory R&C demo requirements definition
  - Ground test planning
  - Ground test hardware design
  - Fabrication plan
- Phase 2
  - Ground test hardware fabrication
  - Test facility preparation, including all fixtures and lab modifications
  - Test article and lab hardware integration and check-out
  - Lab demonstration and data collection
  - Post-test data analyses

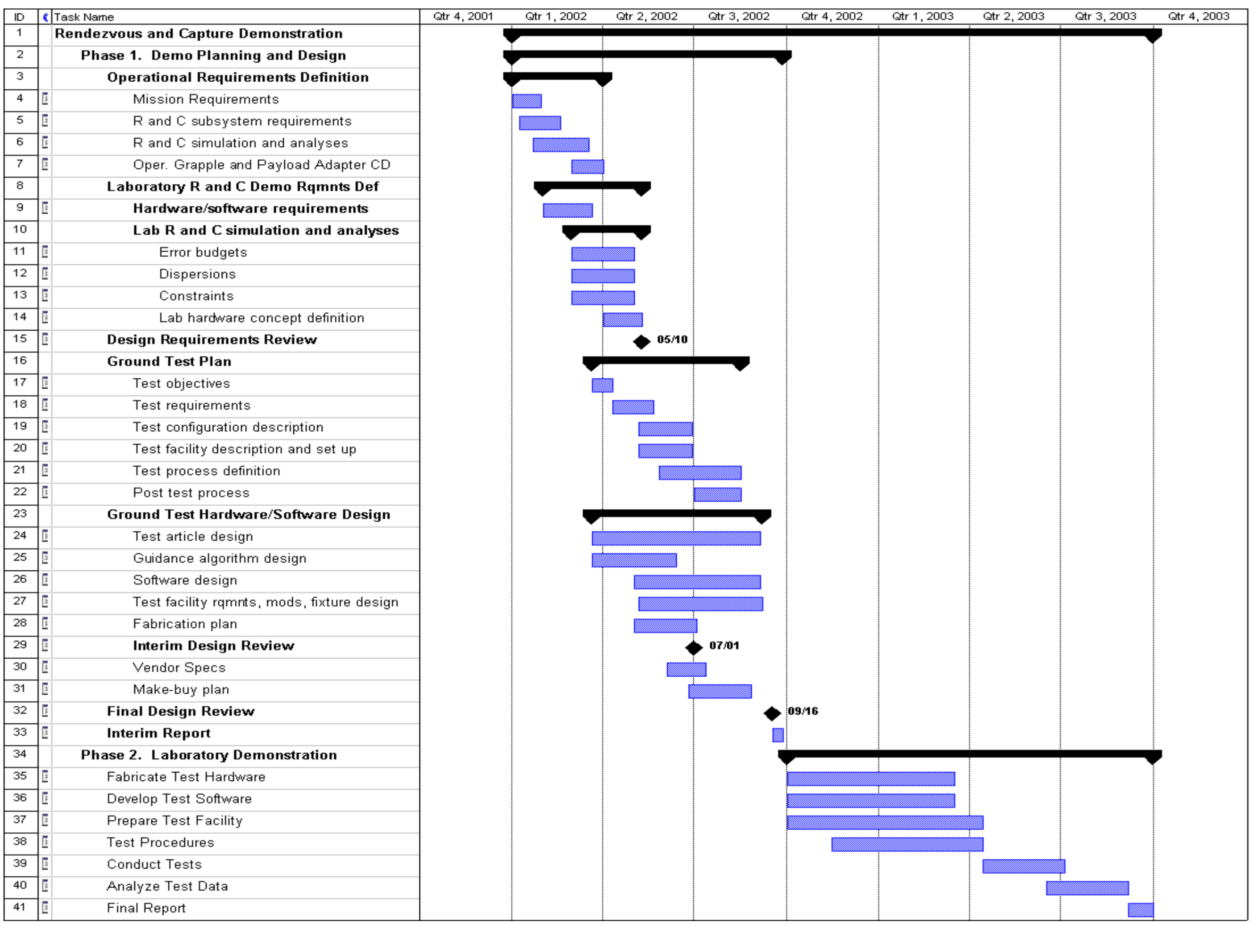


Figure 5.2. Tether grapple and payload rendezvous can be demonstrated in 21 months.

Cost of this two-phase program is estimated to be as little as $3M to $4M. Further details and cost estimate refinements are required. Furthermore, test facility requirements need to be established in order to properly estimate range facilitization, test fixture, and special equipment costs. Other considerations are the following:

- Demonstration complexity, i.e., scope of work
- Validity of demonstration – relative grapple and payload motion in the lab compared to real life scenario
- Use of government furnished equipment and facilities

On-Going Related Efforts. Regarding the use of government facilities, Boeing and NASA Marshall Space Flight Center (MSFC) currently have a Space Act Agreement in place to conduct ground testing of key hardware and software technologies for autonomous vehicle operations during on-orbit proximity operations. Testing is being conducted in the MSFC Flight Robotics Laboratory in Huntsville Alabama. Under this agreement, efforts were initiated during 2000 and will continue through 2001.

Boeing’s responsibilities include the following:

a. Develop and supply test bed software for autonomous vehicle operations, including proximity operations.

b. Develop and supply test bed hardware for autonomous vehicle operations, including proximity operations. Boeing-supplied hardware will include all cabling needed to interface with MSFC Flight Robotics Lab systems.

c. Design fabricate and install on the Dynamic Overhead Target Simulator (DOTS) a target model simulator to be used in proximity operations testing in the MSFC Flight Robotics Laboratory.

d. Supply personnel to integrate the test system, perform testing, and perform post-test de-integration.

e. Create detailed test plans based on design reference missions, and review test plans with Flight Robotics Laboratory personnel.

MSFC’s responsibilities include the following:

a. Review Boeing test plans and hardware/software interfaces with the Flight Robotics Laboratory.

b. Support test hardware/software integration and de-integration in the Flight Robotics Laboratory. Provide test stand for supporting AVO sensor package optical bench. Assist with alignment and setup of optical bench and target simulator.

c. Support Hardware-In-Loop testing of Boeing-supplied autonomous vehicle operations systems in the Flight Robotics Laboratory facility. This test support will include operating the Flight Robotics Laboratory Dynamic Overhead Target Simulator and Solar Simulator systems during testing in the Flight Robotics Lab facility. Testing will include at least three hardware-in-loop test scenarios. Test support will include facility support for operations of all required Flight Robotics Lab hardware and software systems. MSFC will supply video equipment as required to record test operations.

d. Provide assistance in the integration of far-field rendezvous software, MATRIX-x models, and C source code associated with MSFC-developed AR&C study "on-board computer (OBC)" products into the SOTV-SE MATRIX-x software simulation.

e. Provide consulting support for the Boeing Orbital Express proposals. Tele-robotics lab leads shall review proposed design approaches, and recommend any changes or system architecture improvements required for Specific design cases.

Boeing and MSFC are cooperating with the following responsibilities as necessary to accomplish the purpose of the agreement:

a. Review and define hardware and software interface requirements for interfaces between Boeing-supplied test systems and MSFC lab facilities.

b. Review and define detailed test plans and procedures to maximize test utility and information return.

c. Review and adjust detailed test procedures as required to ensure that test activities will not create a risk of test hardware/facility damage or personnel injury.

d. Integrate and checkout all test hardware in the test facility.

e. Execute planned test scenarios. Repeat tests if required due to anomalous results or test interruptions. Revise test plans as required based on observed system performance.

Progress made during the year 2000 is depicted in Figure 5.3, culminating with a sensor demonstration in December 2000. Boeing and MSFC jointly prepared test plans and developed Boeing/MSFC hardware and software interfaces, using MSFC supplied Fight Robotics Laboratory hardware and software design and interface documentation. Boeing supplied test hardware and software were integrated into the Flight Robotics Laboratory. Hardware-in-the-loop testing of Boeing-supplied autonomous vehicle operations systems was conducted in the Flight Robotics Laboratory; test data was consolidated and evaluated.

During 2001, test data review is continuing, and test reports will be prepared. Future autonomous vehicle operations development activity and testing will be considered as planned.

Although the testing to date has been primarily oriented to more traditional rendezvous and docking of space systems, we have the opportunity to leverage the testing and possibly some of the hardware that is being tested. We are also gaining an immense amount of familiarity with the Flight Robotics Laboratory and with the MSFC facility personnel. Once we have completed the definition of the tether grapple rendezvous and capture requirements and developed preliminary R&C concepts, we will be in a better position to propose testing that could be done as follow-on to the testing currently planned.

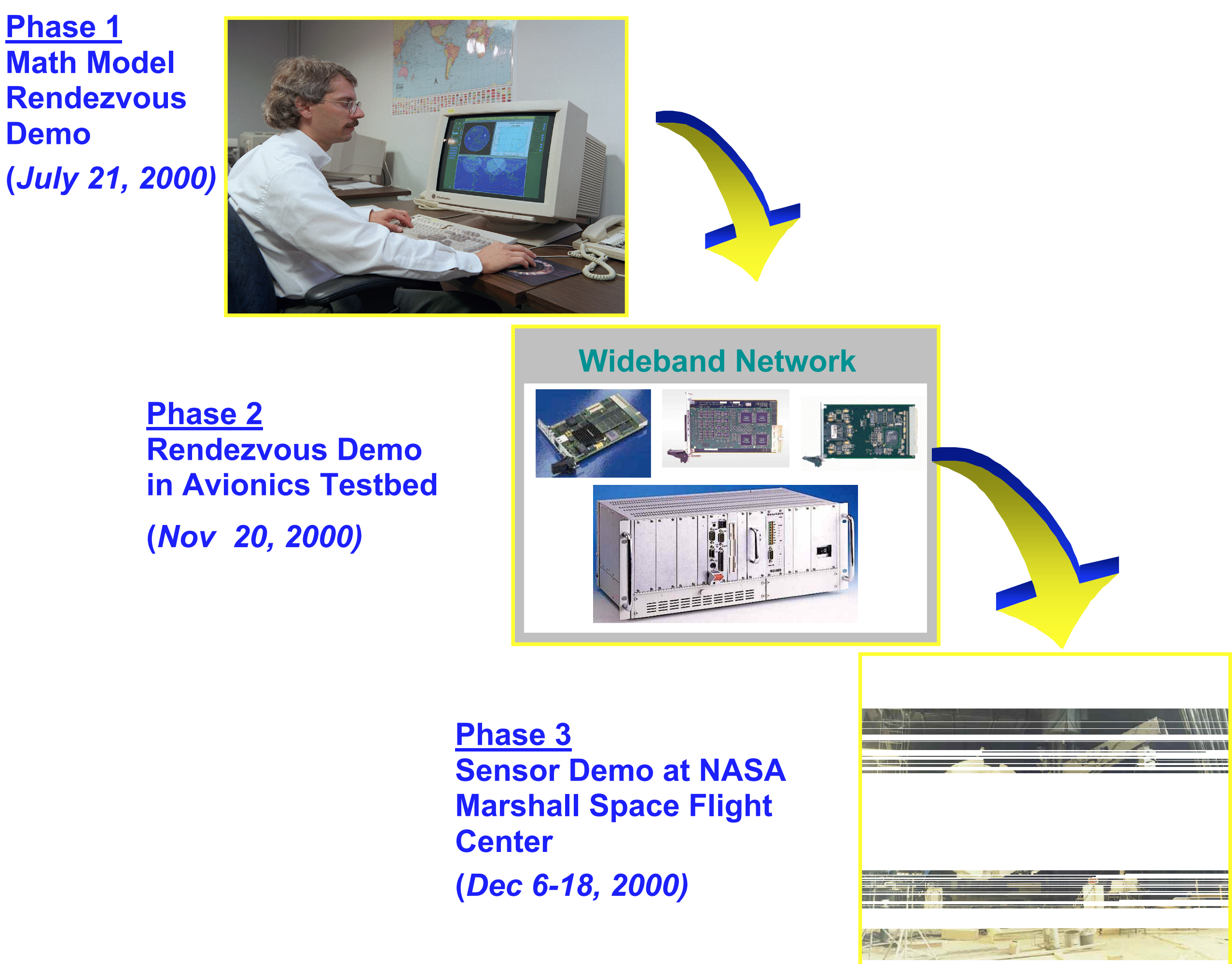


Figure 5.3. Boeing and MSFC are jointly investigating advanced autonomous vehicle rendezvous and docking concepts and sensors.

5.2 Sub-Scale Electrodynamic Tether Propulsion Flight Experiment

While substantial data exists that strongly suggests ED tether propulsion will work as predicted, to date a flight experiment has not been conducted to verify this. An ED flight experiment should verify not only the basic concept but also associated factors and modes of operation as well (e.g., station keeping and altitude raising) to bring most elements of an ED tether transport system to TRL 6. The objectives of an ED tether flight experiment should include:

- Verify models of high-current contact between FEACs and the ionosphere.
- Validate high-power, high-voltage electrical systems.
- Measure FEAC lifetime.
- Validate systems and procedures for ED tether reboost and deboost.
- Validate systems and procedures to maintain tether tension and configuration within acceptable limits in all flight regimes.
- Verify capability for a full range of collision avoidance maneuvers.

Optional tasks such as orbit phase adjustment may also be verified. Also ED tether issues, such as off-angle thrust, out-of-plane libration, and the shifting of the vehicle center of mass will be verified as to the predicted magnitude of effects.

An ED tether flight experiment would need an ED tether of at least 4-5 km in length and enough power to adequately demonstrate the mission parameters described previously. Power requirements would depend on size of the flight experiment, but a kilowatt or more of power would be required to adequately perform the required flight tests. An operational life of a few days may be adequate to meet the basic requirements, but a longer life to demonstrate or verify some tether concepts and concerns is preferred. Tether life in the predicted LEO environment of atomic oxygen and ultraviolet light has been a recent concern. A long term flight experiment could address basic tether material questions as well as ED tether propulsion; however, this approach is not considered a requirement for the ED tether flight experiment.

Several flight design configurations are possible to demonstrate ED tether propulsion. The most obvious option, and probably the best technical launch option, is to use an ED tether to launch a small satellite. This would require a dedicated launch vehicle, thus the most costly option when the launch vehicle cost is added in (e.g., $20M and up). Even using a secondary PAF (Payload Adapter Fitting), the ED tether portion of cost for the launch vehicle can easily exceed $10 million (possibly considerably more).

Alternative lower cost or near zero cost launch options were explored, such a riding in the avionics shelf of an expendable vehicle, such as a Delta II or IV, or using one of the many Space Shuttle options. *Table 5-1* lists some possible carrier/deployment options.

The two best options appear to be the Delta IV avionics shelf and a Shuttle carrier such as the Hitchhiker. Hitchhiker at first glance is the better option, it can provide adequate power for an ED tether at up to 1500 watts and additional costs like avionics are not required. However, operational usage would be limited to a few days at best and the integration and safety costs may be considerable. Experiments like TSS experienced considerable integration costs resulting from NASA Space Shuttle Program imposed safety requirements.

Table 5-1. Possible Carrier/Deployment Options for ED Flight Experiment

	Power	Data	Volume	Weight
Delta II	Limited, est. < 2 hrs	need CPU, no uplink	in work-small	<100 lbs
Delta IV	Limited, est. < 5 hrs	need CPU, no uplink	2x2x2 ft	<200 lbs
Shuttle HitchHiker	1500 W	Downlink & I/O	5 cu.ft.	200
Shuttle HitcHiker Jr.	100 W	PGSC only	5 cu.ft.	200
Shuttle Gas Can	None	None	5 cu.ft.	200
Shuttle SL Pallet/MPESS	> 200 W	SpaceLab CPU	large	>200
Shuttle SpaceHab ICC	> 200 W	SpaceHab CPU	large	>200
Shuttle RMS Arm	>100 W	Need CPU, Serial port	large	TBD

ProSEDS used the Delta II avionics shelf, however the Delta II shelf can support only limited volume and weight, and assuming a ProSEDS type deployer, no excess weight or volume is left for avionics and other systems needed for Electrodynamic propulsion. The Delta IV avionics shelf can support about 2-3 times the weight and volume as Delta II. A microsat, with a deployer or reel on the Delta IV side, could be deployed. Adequate power is nominally available in the upper stage batteries after the upper stage mission is complete for a couple of days of minor tasks, or enough to deploy a microsat via a tether deployer/reel. The microsat would have to provide the power, via solar arrays, to power the ED tether for propulsion and minor avionics overhead (small controller and S-Band transceiver, power equipment, and CMG). The advantage of such a system is that mission life could be measured in months or even years. Also ProSEDs proved the concept is feasible, as well as qualifying a Delta avionics shelf mounted deployer (albeit for Delta II, not Delta IV). In this example, a microsat, instead of a dumb-end mass, is on the other end of the tether, as shown in Figure 5.4.

The avionics shelf shown in Figure 5.5 is a typical LEO configuration for a Delta IV, with a couple of positions unused. The engine and nozzle, are not shown, attaches in the center. The empty locations on the shelf are normally filled with an extra battery and a hydrazine bottle for extended length mission (e.g., GTO, GEO). Thus it might be possible to have these locations open for both a ProSEDS-type deployer and a microsat.

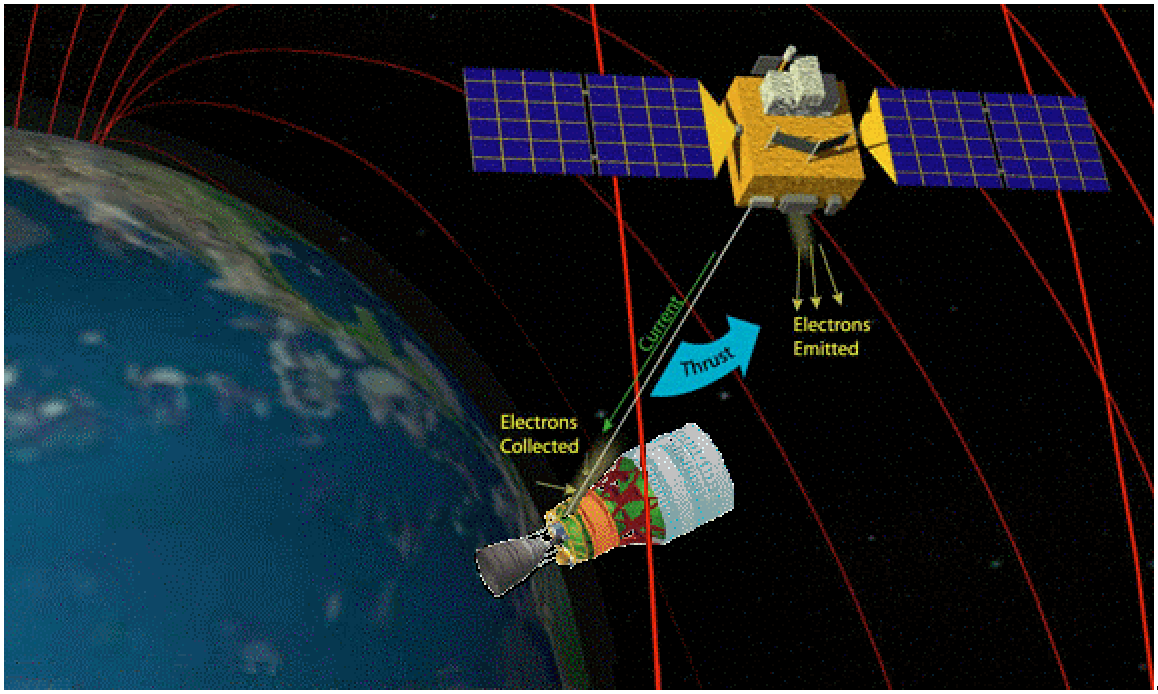


Figure 5.4. ED tether thrust experiment utilizing a microsat.

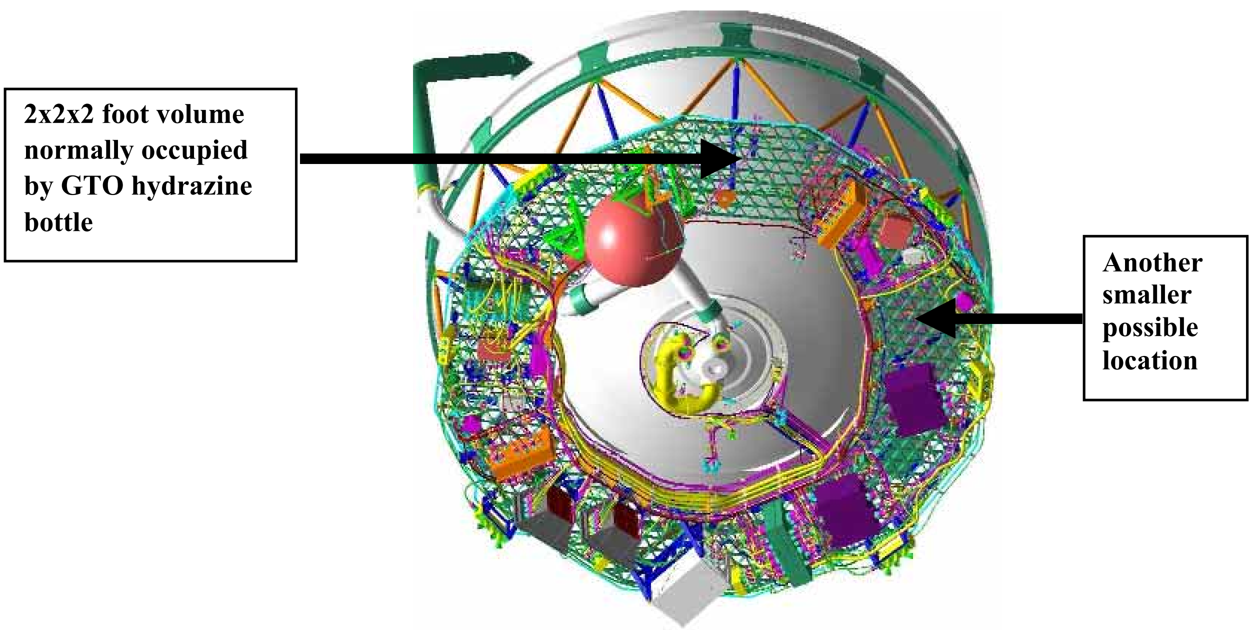


Figure 5.5. Experiment locations on Delta IV avionics shelf.

5.3 Sub-Scale Electrodynamic Tether Boost Facility Demonstration

A sub-scale flight experiment is needed to demonstrate the MMOSTT unique concepts. Demonstrations would include spin-up of the tether boost facility, deploy of the ED tether during spin-up, release of a payload, capture of a different payload (e.g., microsat), and reboost of the tether boost facility.

5.3.1 Concept

A maximum mission cost of $50M, not including launch vehicle costs, was assumed. The depleted launch vehicle upper stage would be used for the Control Station. In additional to providing the needed mass, the upper stage would provide also the avionics and attitude control system for the Control Station, thus reducing costs. The launch vehicle payload would consist of the tether, the tether deployer, a maneuverable Grapple Assembly (with a simple, deployable-only payload attached), solar arrays, and a second, maneuverable, deployable/retrievable payload (e.g., a microsat). Having a low-cost, deployable-only payload attached to the grapple at launch allows the experiment to verify the payload release mechanism and release precision even if the second payload cannot be captured due to a malfunction.

The first part of the mission would consist of launching into LEO, deploying the microsat, firing the upper stage to place the Control Station in the required orbit, spinning-up the Control Station while deploying the tether, stabilizing this demonstration tether boost facility, and then releasing the simple payload. The second part would consist of the microsat adjusting its orbit for rendezvous with the rotating tether boost facility, approaching the tether Grapple Assembly, being captured by the maneuverable grapple mechanism, and then being released to a higher orbit, one-half rotation later. The third part of the mission would consist of demonstrating various orbit maneuvers of the tether boost facility, evaluating tether survivability and motion predictability over some desired duration, and eventually ending the mission with a tether-powered deorbit.

5.3.2 Design

To further reduce cost, the maneuverable Grapple Assembly and target payload could be a previously-designed microsat, such as an XSS-10 (see graphic right). An XSS-10 or similar vehicle is capable of rapid movement and is already space-qualified.

Two such units would be used, with additional grapple fixtures added on the unit destined to be the Grapple Assembly, and the mating half on the vehicle used as the target payload. If a Delta II upper (second) stage is used, with a dry weight of 950 kg (see Figure 5.6), a 10-kilometer ED tether would require a power system of 5-10 kW. Assuming a 10 kW power system, the mass of the solar arrays, batteries, and additional power controller would be about 100 kg. Approximately 100-150 kg is estimated for the Grapple Assembly and target payload. Depending on the final length of the tether, 100-400 kg is estimated for tether and deployer. Miscellaneous systems, such as thermal and telemetry, will probably add another 100 kg.

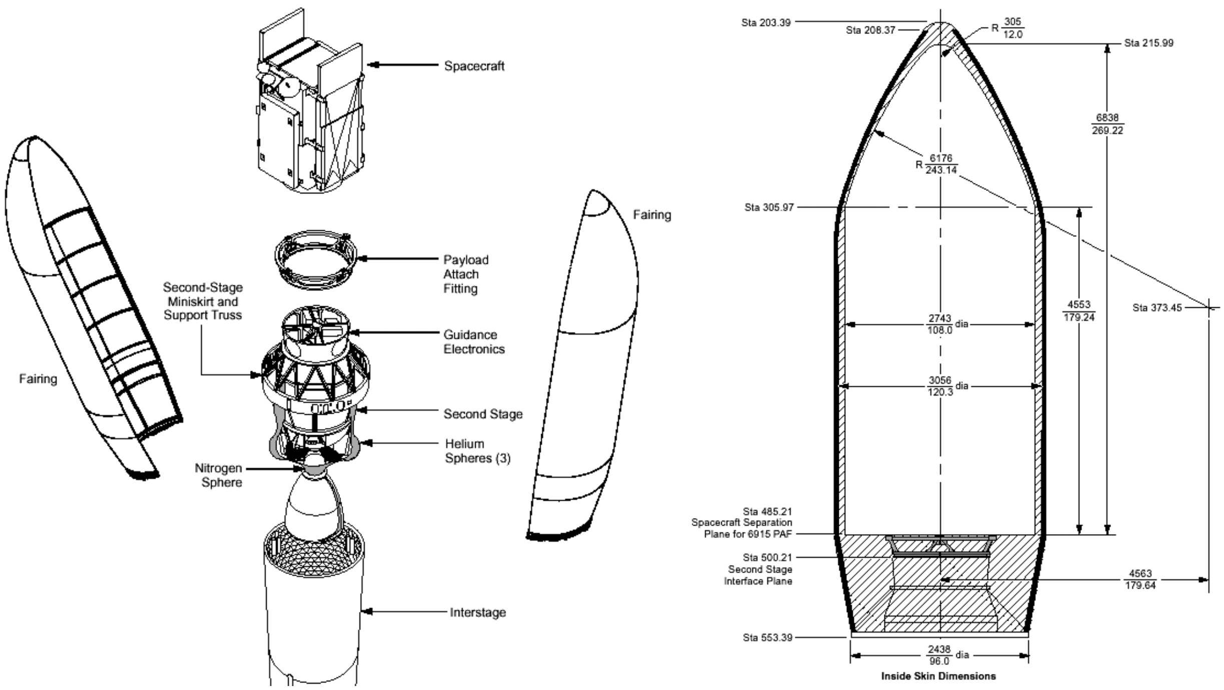


Figure 5.6. Delta II upper (second) stage and payload compartment.

Cost and integration time should be dramatically reduced because few items are of a new design, other than the tether and possibly the rendezvous and capture devices. The launch vehicle at approximately $50M, would be over half the system cost, assuming the vehicle is not shared. Cost could be reduced by utilizing the excess launch weight and going to the large 3×9-meter fairing; thus allowing rapid packaging and simpler and larger system designs. For example, simple solar array design that only folds once or twice, rather than a more complex fanfold design. Flying as a secondary payload using a PAF, could easily reduce the launch vehicle cost by 50-90%. The Control Station could still make use of the upper stage for additional mass. However, packaging could be considerably more constrained with the smaller volume and weight provided by flying as a secondary payload. Also, issues with the orbit the primary payload will be deployed in may complicate the orbit desired for the sub-scale demonstrator and cause mission work arounds.

5.3.3 Fabrication, Assembly, and Ground Testing

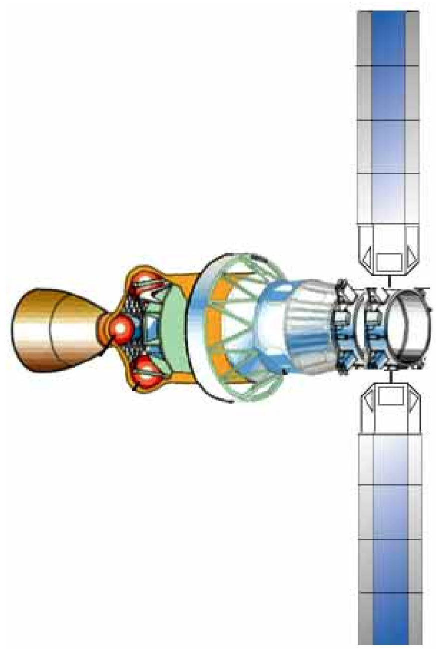
Due to a large portion of the system being previously qualified space hardware, a large cost savings is projected in the fabrication, assembly and test arena. The majority of the high cost systems, avionics and attitude control, could be reused from the Delta upper stage, as well as reusing the XSS-10 or similar type microsat for the space maneuvering part of experiment. Note: additional attitude and control systems, as well as other avionics will be needed. Additionally, if a long term mission is desired, the short term upper stage systems will need to be supplemented.

- 3-phase flight experiment
- Possible limited IOC

5.4 Experiment Major Steps

The three-phase mission profile for the sub-scale ED Tether Boost Facility demonstration flight has been further divided into 16 major steps, each with a number of possible minor steps. The first phase of the mission will be accomplished in Steps 1 through 8; the second phase, in Steps 9 through 13; and the third phase, in Steps 14 through 16.

1. Achieve on-orbit starting configuration (min 300 km, circular)
    - Phase orbit if necessary to obtain desired ground track coverage
    - Deploy solar arrays on Control Station (see right)
    - Start experiment subsystems on Control Station
    - If non-recoverable mission failure occurs, deorbit upper stage using onboard propellant



2. Deploy Microsat Target Vehicle
    - Checkout pre-deploy Microsat Target Vehicle systems
    - Deploy Microsat Target Vehicle
    - Checkout remaining Microsat Target Vehicle systems
    - If non-recoverable mission failure occurs, deorbit upper stage using onboard propellant
3. Perform ΔV burns to raise orbit of Control Station by the length of the tether (10km)
    - Burn to raise apogee 10 km
    - One-half orbit later, burn to raise perigee 10 km (i.e., circularize orbit)
    - If non-recoverable mission failure occurs, deorbit upper stage using onboard propellant
    - Coast in circular orbit until desired new perigee point is reached

4. Perform ΔV burn to raise apogee of Control Station to slightly elliptical orbit
   - Burn to raise apogee (500-1000km, limited by amount of propellant in upper stage)
   - Apogee also limited by mass of demo Tether Boost Facility hardware (see Figure 5.7)
   - Possibly use only half of propellant for orbit raising and save rest for abort deorbit
   - Burn to trim orbit

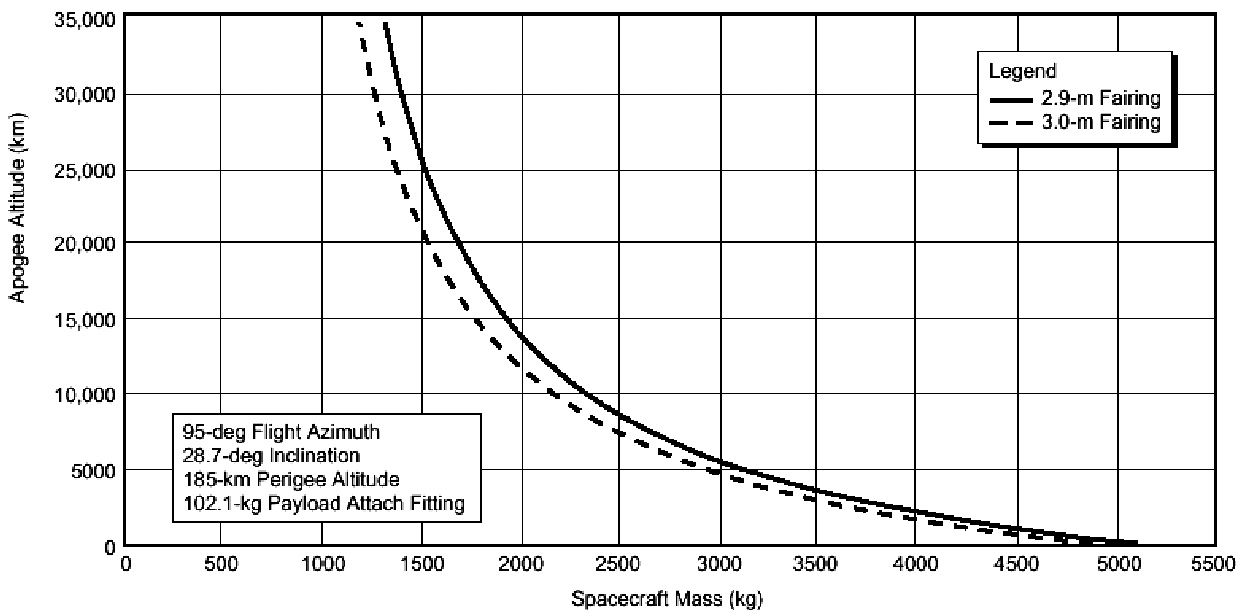


Figure 5.7. Delta II elliptical orbit payload delivery capability.

5. Spin-up Control Station
   - Perform star alignment calibration
   - Maneuver to start attitude
   - Spin-up the Control Station using onboard thrusters

6. Deploy Tether and Grapple Assembly with simple payload attached (see Figure 5.8)
   - Deploy Tether and Grapple Assembly
   - Trim Control Station attitude rate
   - Apply current to tether as necessary to maintain or increase rotation rate
   - Repeat above steps until tether fully deployed

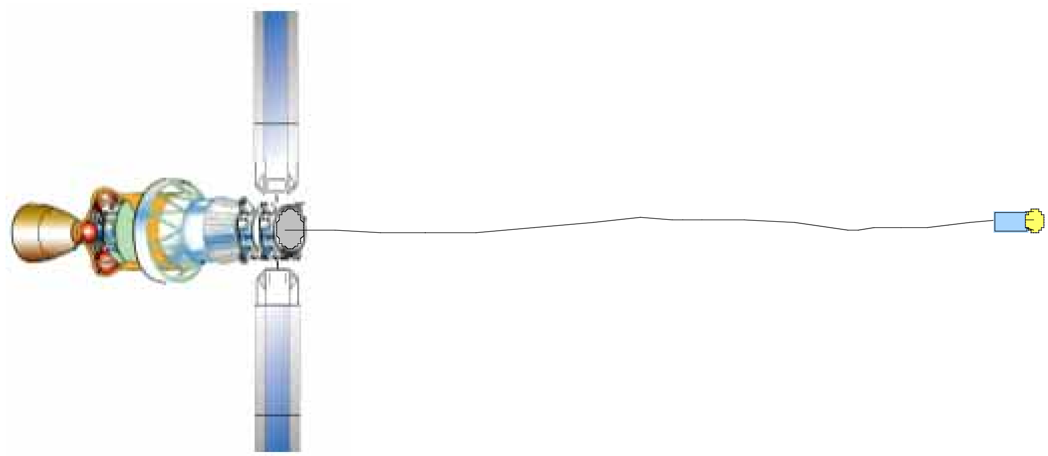
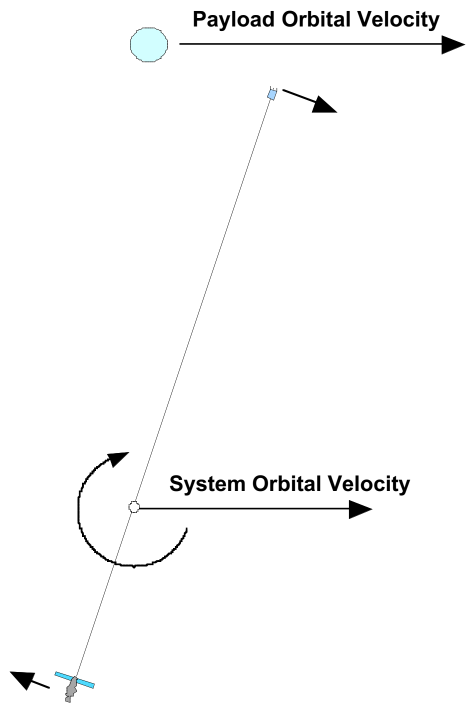


Figure 5.8. Deployment of Grapple Assembly with Simple Payload.

7. Spin-up Tether Boost Facility (if not at desired rotation rate)
   - Apply current to tether as necessary to increase rotation rate to desired value
   - Stabilize the system as necessary in preparation for payload release



8. Release and Track Simple Payload
   - For active Simple Payload, activate radar beacon and/or GPS data transmitter
   - Release Simple Payload at release point (highest point of rotation of Tether Boost Facility, see graphic to right)
   - For passive Simple Payload, deploy radar enhancement ballute
   - Track Simple Payload to verify accurate orbit placement

9. Trim Orbit of Tether Boost Facility using ED tether propulsion (while spinning)
   - Calculate initial intercept scenario
   - Trim Tether Boost Facility orbit (to capture orbit) using propulsion/drag cycles
   - Trim Tether Boost Facility attitude rate
   - Calculate final intercept scenario

10. Perform ΔV burn to move Microsat Target Vehicle to rendezvous start position
    - Move Microsat Target Vehicle to rendezvous run starting point
    - Perform station-keeping relative to Tether Boost Facility orbit

11. Perform ΔV burn to direct Microsat Target Vehicle towards Tether Boost Facility
    - Burn motor on Microsat Target Vehicle to achieve rendezvous with Grapple Assembly
    - Prepare Microsat Target Vehicle for capture/grapple maneuvers
12. Capture Microsat Target Vehicle with Grapple Assembly
    - Deploy Grapple Mechanism and prepare for intercept and capture
    - Maneuver Grapple Mechanism to capture Microsat Target Vehicle
    - Capture Microsat Target Vehicle at lowest point of rotation of Tether Boost Facility (see Figure 5.9)
    - Trim Grapple Assembly attitude maneuver rate
    - Trim Control Station attitude maneuver rate

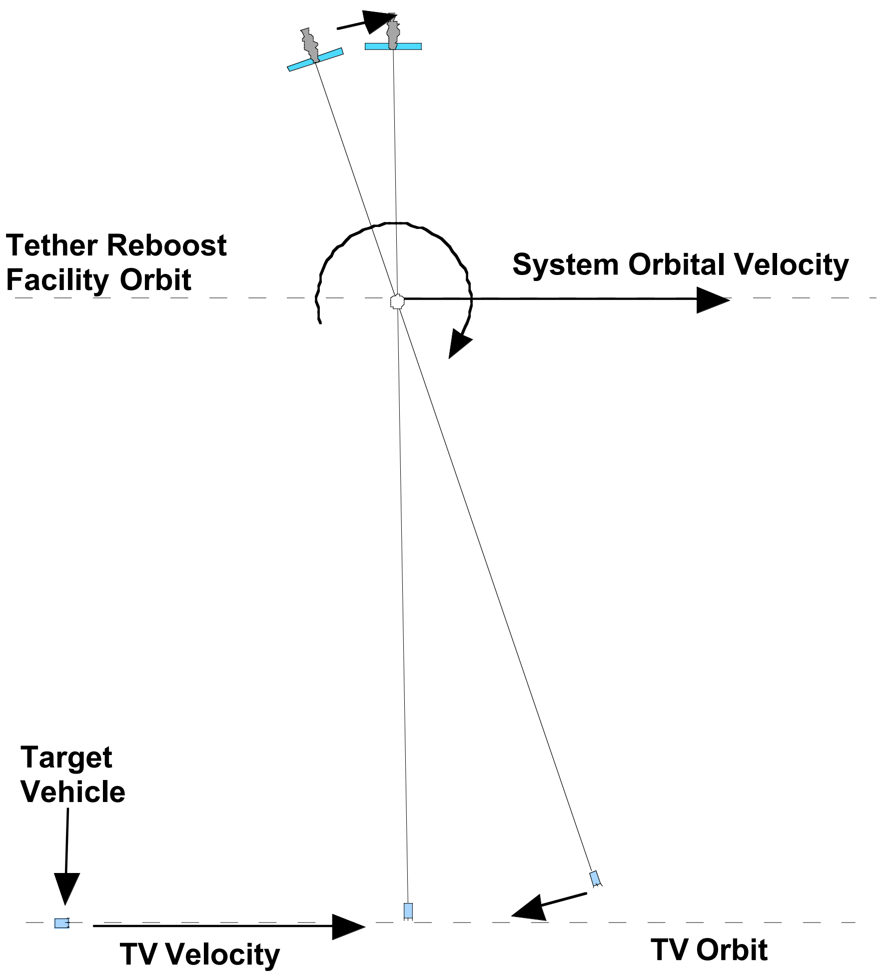


Figure 5.9. Capture of Microsat Target Vehicle Payload.

13. Toss Microsat Target Vehicle into Higher Orbit
    - Prepare Microsat Target Vehicle for toss release
    - Release Microsat Target Vehicle one-half rotation after capture
    - Track Microsat Target Vehicle and verified insertion into desired new orbit

14. Adjust Tether Boost Facility orbit using ED tether propulsion
    - Apply ED tether propulsion to reboost Tether Boost Facility
    - Apply ED tether propulsion to raise or lower Tether Boost Facility orbit
    - Apply ED tether propulsion to change plane of Tether Boost Facility orbit
    - Trim Tether Boost Facility attitude maneuver rate

15. Perform other Mission Objectives (as budget permits)
    - Monitor interaction of ED tether with Earth space environment
    - Monitor durability of ED tether
    - Monitor durability of Control Station

16. Deorbit Tether Boost Facility using ED tether propulsion
    - Apply ED tether propulsion to slow/zero Tether Boost Facility rotation rate
    - Apply ED tether propulsion to lower Tether Boost Facility orbit
    - Apply ED tether propulsion (as possible) to avoid orbital traffic
    - Apply ED tether propulsion to prepare Tether Boost Facility for deorbit
    - Apply ED tether propulsion to safely deorbit Tether Boost Facility

APPENDIX L- I

Tether Boost Facility
System Requirements Document

May 5, 2000

Moon & Mars Orbiting Spinning Tether Transport (MMOSTT)
LEO Tether Boost Facility
System Requirements Document

Prepared for Tethers Unlimited, Inc.

By

Boeing Space & Communications Group

Table of Contents

1 Scope

The MMOSTT system is an in-space transportation system that incorporates an Earth-orbiting facility with a spinning tether as a primary element. This document describes the top-level system requirements for a tether boost facility in low Earth orbit. The requirements defined here are for the objective system, that is, for a facility that is part of an operational full-scale transportation system that boosts payloads from low Earth orbit to higher orbits or Earth escape. Requirements for sub-scale or demonstration systems may be described at a future date in other documents. Requirements for tether facilities in higher Earth orbits or in orbits about other planetary bodies may be described at a future date in other documents.

The tether facility consists of the tether, a control station, and a grapple assembly. The control station controls the tether and tether dynamics. The grapple assembly captures and releases payloads.

A payload is any object that will be accelerated or decelerated toward a destination in space. The transportation system includes a payload accommodation assembly to provide the interface between the payload and the grapple assembly on the tether facility.

This document describes requirements including a nominal payload mass and a nominal release orbit. It is envisioned that the tether facility will be able to grapple payloads with greater or lesser mass than nominal and release them after imparting less or more delta-v, respectively, than nominal. However, no requirements are imposed on the delta-v that must be imparted to non-nominal payloads, except that the system can deliver a smaller payload to the same release orbit as a nominal payload. The tether facility is envisioned to accommodate modular assembly and growth, so its capacity may be expanded to handle larger payloads if a need arises.

2 References

This section lists documents to which the text of this document refers.

Ariane 5 User's Guide

Delta IV Payload Planners Guide, October 1999

Cislunar Tether Transport System, Phase One Final Report, May 30, 1999, Tethers Unlimited, Inc., NIAC Contract 07600-011.

Cislunar Tether Transport System, Phase Two Proposal, May 28, 1999, in response to NIAC CP99-01.

3 System Requirements

3.1 System Performance

3.1.1 Payload Mass

The nominal payload mass shall be 5000 kg.

3.1.2 Nominal Pickup Orbit

The system shall pick up nominal payloads from an equatorial circular orbit at 300 km altitude above the Earth.

3.1.3 Nominal Release Orbit

The system shall release nominal payloads into Geosynchronous Transfer Orbit (GTO). The change in velocity (delta-v) from the nominal pickup orbit to GTO is 2460 m/s.

3.1.4 Release Orbit Insertion Error

The release orbit insertion error shall be no greater than the release orbit insertion error of Ariane 5 or Delta 4, whichever is smaller, in each of the seven orbital elements.

3.1.5 Payload Interfaces & Accommodation

Payload interfaces and accommodations shall be compatible with any payload designed to be compatible with Ariane 5 or Delta 4.

3.1.6 Payload Environment

The payload environment shall be compatible with any payload designed to be compatible with Ariane 5 or Delta 4.

3.1.7 Turnaround Time

The maximum system turnaround time between deliveries of nominal payloads shall be 30 days.

3.2 System Durability, Reliability, Maintainability, and Safety

3.2.1 Design Life

The design life of the system shall be 10 years with 99% confidence.
This requirement does not mean that no system elements or components will be expended, repaired, or replaced in 10 years. For example, the Payload Accommodation Assembly (PAA) for GTO payloads may be expendable.

3.2.1.1 Self-entanglement

The system shall avoid entanglement of the tether with any element of the system.

3.2.1.1.1 Tether Dynamics

The system shall measure and control tether dynamics.

3.2.1.2 Debris Tolerance

The system shall survive orbital debris impacts for 10 years with greater than 99% confidence.

3.2.2 Maximum Lifetime

Consumables and expendables shall not constrain the lifetime of the system.
The system will not preclude resupply or addition of consumables and expendables, even after the design lifetime has been exceeded. The system may continue to operate beyond its design lifetime with degraded performance. It should not become inoperable simply because the system was not designed to allow some consumable or expendable item, e.g. propellant, to be restocked.

3.2.3 Repairability

Orbiting elements of the system shall be repairable on orbit.

3.2.4 Evolvability

The system shall be growable and evolvable on orbit to deliver a payload ten times more massive than the nominal payload to the nominal release orbit.

3.2.5 Disposal

The system shall provide for safe disposal of all system elements.

3.2.6 Fail Operational

The system shall operate after any one credible component failure.

3.2.7 Two-Failure Safety

The system shall be safe after any two credible component failures.

3.2.8 Collision Avoidance

3.2.8.1 Tracked Debris

Orbiting elements of the system shall avoid collision with tracked debris that would diminish the system's ability to perform its mission.
The system may be designed to shield against or otherwise survive some debris impacts, thereby making avoidance unnecessary in some cases.

3.2.8.2 Tracked Satellites

Orbiting elements of the system shall avoid collision with tracked satellites.

3.2.8.3 Manned Spacecraft

Orbiting elements of the system shall avoid collision with human-occupied spacecraft. This is a safety requirement, and therefore must be satisfied after any two credible component failures.

3.2.9 Operational Orbital Lifetime

The minimum operational orbital lifetime of orbiting elements shall be 15 days.
Orbital lifetime is the time from loss of control until an orbiting asset becomes so deeply snared by the atmosphere that re-entry becomes unavoidable. Operational orbital lifetime refers to the orbital lifetime of an asset that has reached its operational orbit. The orbital lifetime may be lower while the asset is in an assembly orbit or deployment orbit.

3.2.10 Payload Pickup Reliability

The system shall pick up a nominal payload from a nominal pickup orbit with better than 99% reliability.
The sum of Error (99.5%, grapple position, attitude, and velocity vector) plus Error (99.5%, PAA position, attitude, and velocity vector) will be smaller than the operating envelope of the grapple mechanism.
This is equivalent to saying the system must control state vectors of elements well enough that when the system reports it has placed the payload within the grapple envelope, then the operator can have 99% confidence that the payload is actually within the grapple envelope.

3.3 Communications, Control, Sensing, & Telemetry

3.3.1 Communicate Mission Readiness

The system shall assess and communicate its health and status to the operator in sufficient detail to enable a determination of mission readiness prior to payload launch.

3.3.2 Communicate Status

Each system element shall communicate its position using a common time reference, its health, and its status.

3.3.3 Automated Mission Profile

The system shall provide a capability to automatically produce mission profiles.

3.4 Deployment

3.4.1 Launch Vehicle

Orbiting elements of the system shall be capable of being launched on an existing launch vehicle.
Existing means that the selected launch vehicle is credibly expected to be operational when the tether system is ready for deployment.

3.4.2 First Launch Capability

The first launch segment shall provide some operational capability.
Some operational capability means a non-zero delta-v is added to a non-zero payload mass.

4 Ground Rules & Assumptions

This section defines ground rules and assumptions to be used in concept definition and assessment. It also provides guidance for element-level requirements.

4.1 Safety Factor

The structural safety factor shall be two times the highest expected stress.
Where the maximum stress is not known, the structure shall be designed for three times the nominal steady-state stress.

4.2 Economic Analysis Window

The economic analysis window shall be 10 years from the 1st launch after full-up system authority to proceed.

4.3 Power Source

The primary power source for long-term orbiting elements shall be solar. On-board energy storage is permitted.

4.4 Primary Propulsion

Electrodynamic propulsion shall be the primary mode of propulsion for the LEO Tether Facility.

4.5 PAA Role in Rendezvous & Grapple

The PAA shall cooperate in meeting the grapple.
This could include position and speed control, or just attitude control, or just transmission of navigation data, or (conceivably) no action.

4.6 Tether Control Station Reeling Rate

If the Tether Control Station reels in the tether, the reeling rate shall not be required to exceed 2 m/s.

4.7 Tether Control Station Thermal Control

The Tether Control Station shall use passive thermal control.

5 Terminology

GEO – Geostationary Earth Orbit

Grapple Assembly - End mass that captures, releases, and interfaces with payloads.

Ground Station - Provides system control interface.

GTO – Geostationary Transfer Orbit

LEO – Low Earth Orbit

LLO - Low Lunar Orbit

LMO - Low Mars Orbit

MMOSTT – Moon Mars Orbiting Spinning Tether Transport

PAA – Payload Accommodation Assembly

Payload - Any useful object that will be accelerated or decelerated toward a new trajectory in space.

Payload Accommodation Assembly - System to provide the interface between the payload and the Grapple Assembly.

Tether - Flexible connector between major elements of system

Tether Control Station - Facility that controls the tether and tether system dynamics. Contains all of the control hardware and subsystems required. Located in LEO.

APPENDIX L-2

Tether Boost Facility Concept Description

May 1, 2001

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LIST OF FIGURES

ACRONYMS AND ABBREVIATIONS

ADCS	Attitude Determination and Control Subsystem
C&DH	Command and Data Handling
CCHP	Constant Conductance Heat Pipe
CMG	Control Moment Gyro
Comm	Communications
CS	Control Station
ED	Electrodynamic
EPS	Electrical Power System
FEAC	Field Emitter Array Cathode
GEO	Geosynchronous Earth Orbit
GN&C	Guidance, Navigation, and Control
GN_2	Gaseous Nitrogen
GPS	Global Positioning Satellite
LEO	Low Earth Orbit
MMOSTT	Moon-Mars Orbiting Spinning Tether Transport
NIAC	NASA Institute for Advanced Concepts
PAA	Payload Adapter Assembly
TUI	Tethers Unlimited, Inc.
VCHP	Variable Conductance Heat-Pipe

1 SCOPE

1.1 Purpose

The Moon-Mars Orbiting Spinning Tether Transport (MMOSTT) project is a design study funded by the NASA Institute for Advanced Concepts (NIAC) and led by Principal Investigator Dr. Rob Hoyt of Tethers Unlimited, Inc. The purpose of the study is to define a space transportation system that uses a spinning tether facility in orbit about the Earth to boost payloads from Low Earth Orbit (LEO) to higher orbits or to Earth escape. The tether facility consists of the tether, a control station, and a grapple assembly. The control station controls the tether and tether dynamics. The grapple assembly captures and releases payloads. Other major components of the MMOSTT objective system are the ground station and the payload accommodation assembly. The latter provides the interface between the payload and the grapple assembly.

1.2 Scope of Text

This document describes results of engineering design and analysis conducted by the Boeing Company, a subcontractor to Tethers Unlimited, Inc. (TUI), through Phase 2 of the MMOSTT contract. The scope of this work covers design and analysis of the Tether Control Station (CS). Section 2 of this document describes key assumptions and design drivers for the overall CS. Section 3 describes each major subsystem of the CS. For each major subsystem, the text describes key requirements and assumptions, new issues and requirements identified during Phase 2 of the project, trade studies and design recommendations, and recommended follow-on studies. Section 4 is a summary of the current design.

2 SYSTEM-LEVEL DESIGN DRIVERS

2.1 Launch Mass and On-Orbit Assembly

A key requirement for the MMOSTT system is that only current launch vehicles can be used to deploy the tether facility. To meet the requirement for launch on an existing launch vehicle, with Delta IV-Heavy assumed to be the most powerful launcher available, each launch package must have gross lift-off weight (GLOW) less than 23, 382 kg. Preliminary design studies showed that deployment of the objective system, i.e. one capable of delivering 5000 kg payloads to GTO, would require two launches of the Delta IV-Heavy. The tether facility is required to provide some operational capability after the first launch. The facility will have full capability after two launches.

2.1.1 First Facility Requirements

Requirements for the objective system are described in the MMOSTT LEO Tether Boost Facility System Requirements Document. Except for the mass of the payload to be boosted, the first-launch system has the same requirements as the objective system. These include payload pickup and release, 30-day turnaround, 10-year design life, and collision avoidance. The objective system is required to transport payloads as massive as 5000 kg. The first-launch system must transport payloads up to 2500 kg. In addition, the first facility must accommodate attachment of more modules that arrive on subsequent launches.

2.1.2 Facility Add-On Requirements

The purpose of Control Station expansion is to increase the station's mass and power. Increased mass enables it to boost heavier payloads and impart larger ΔV's to the P/L, without having its orbit drop so low as to risk Control Station re-entry after the momentum transfer to the payload. Increased power enables the more-massive tether facility to re-boost itself in 30 days after transferring momentum to a heavier payload. An additional unit added to the first-launch Control Station is called a module. The first module (i.e. the one delivered on the second launch) must enable the tether facility to handle the full operational payload, 5000 kg.

Each module added to the facility must perform unmanned rendezvous and attachment to the Control Station. It must work cooperatively with the Control Station to execute required MMOSTT functions. We have not studied add-on modules in any detail. Issues to be studied in future work include:

- Should the facility de-spin during module attachment, remain spinning at full speed, or spin at some intermediate speed?
- Should the module attach via hard docking, or should it attach via some length of tether?
- What functions should the module provide in addition to inertia and power? Additional plasma contact, for example?

Though we do not have answers to these questions, our design of the first facility includes a large open area at the end of the CS to accommodate docking of a module.

2.2 Zero or Low Consumables

We set a design goal of zero consumables for the MMOSTT tether facility. If met, this goal avoids the need for resupply operations and thereby reduces risk and operations cost. The key design impact of this goal is the use of electrodynamic propulsion for reboost of the facility and CMGs for orientation control of the facility.

Electrodynamic (ED) propulsion imposes a number of challenges on the design and on technology development. ED thrust varies in magnitude and direction according to variations in the geomagnetic field and the ionospheric plasma density. This drives the flight control system to use new methods. ED thrust also requires quite high voltage - tens of thousands of volts. This is beyond current practice for high-power space electrical systems.

2.3 High Power For Thrust

To achieve 30-day reboost using electric propulsion requires quite high power levels, primarily near perigee. This means the CS must provide hundreds of kilowatts of average power from solar arrays and much higher peak power from batteries for thrust during near-perigee passes in the Earth's shadow. The size of the solar arrays and the mass of the batteries dominate the CS design.

3 CONTROL STATION DESIGN

3.1 ADCS/GN&C

3.1.1 Design Requirements, Drivers, and Assumptions

Attitude determination and control system requirements are driven mainly by a combination of spin maneuver requirements and solar panel pointing requirements. The following are preliminary requirements based on the current top-level sizing done to date.

- Attitude Knowledge: +/- .05 deg / axis
- Attitude Hold Capability (non-maneuvering): +/- 1.0 deg / axis
- Attitude Maneuver Axis Accuracy: (spin axis): +/- 1.0 deg (non-spin axis): +/- 1.0 deg
- Attitude Rate Accuracy (non-maneuvering): +/- 0.1 deg/sec/axis
- Attitude Rate Accuracy (maneuvering): +/- 0.1 deg/sec/axis
- Position Knowledge: (GPS Receive)
- Velocity Knowledge: (GPS Receive)
- Orbit Positioning Accuracy: Inclination, Ascending Node, Arg. of Latitude, etc.: TBD

Attitude maneuver capability requirements for spinning the control station while not using expendable propellant will require the use of control moment gyros. The following discussion explains the design drivers for using CMGs and some possible concepts for desaturation that also do not use consumables.

Reaction wheels provide control torques by changing wheel speed, ie. converting stored momentum into torque (Tj = d (hj)/dt). Control moment gyros (CMGs) provide control torques by changing the direction of the wheel momentum vector (gyroscopic torque). Both reaction wheels and CMGs can have large stored momentum but the maximum reaction wheel torque available is limited by power considerations, as shown in Figure 3-1.

An example: RW Power (@ 6000 rpm) = 1 ft -lb @ 1300 watts or 3 ft-lbs @ 3900 watts

Reaction wheels require large power to produce torques due to changing momentum by accelerating/decelerating a spinning disc about a fixed axis. However, CMGs (see Figure 3-2) can produce very large torques by changing momentum vector direction (using fairly small input torques) of a large spinning disc. CMG discs usually take long periods for spin-up since the spin motors are not sized nearly as large as a reaction wheel motor.

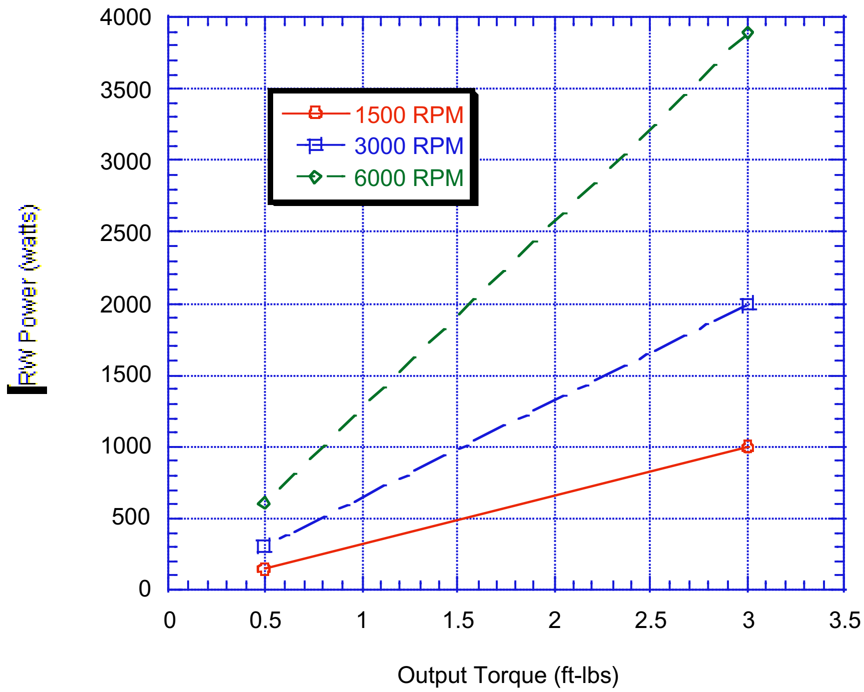


Figure 3-1. Reaction Wheel Power vs. Output Torque

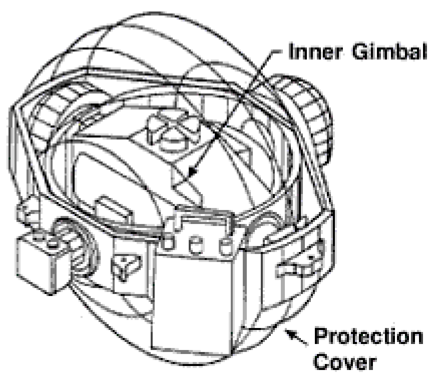


Figure 3-2. Control Moment Gyro for the International Space Station

3.1.2 New Issues and Requirements Identified

Typically, double-gimbal CMGs are used for control torques from 3 to 300 ft-lbs and single-gimbal CMGs are used for applications requiring more than 300 ft-lbs. Skylab was designed to use two double-gimbal CMGs, each having a capability of +/-160 ft-lbs (a third unit was installed as a backup). Skylab mass was 76295 kg and design power with all solar panels working was 11 kW average for the entire workshop system. Both reaction wheels and CMGs usually require desaturation if a large amount of maneuvers are required or if system biases cause long duration torques in one direction. Another source of control torques (eg. RCS, or a secondary set of RWs or CMGs) will be needed to remedy a situation like this.

For MMOSTT, the control station mass is roughly 1/5 the mass of Skylab, however the moment of inertia of MMOSTT due to a large amount of solar panels may be as much as half that of Skylab. A calculation of inertias for an early configuration of the MMOSTT control station gives: Ixx = 218,000 kg-m^2, Iyy = 444,000 kg-m^2, Izz = 647,000 kg-m^2. With expansion modules the inertia might reach fairly large values, thus a first approximation for CMG sizing is roughly half that of Skylab. Once the MMOSTT control station is brought up to maneuver rate about the spin axis, the station itself will have a large momentum vector which will allow attitude control about the two non-spin axes with fairly small torques. This scaled-up version of gyroscopic control will need to be studied in detail in the next study phase.

To solve the problems of desaturation without use of consumables (i.e., an RCS system) will require a combination of attitude management and secondary torques. Attitude management techniques may be used to cancel bias torques that are attitude dependent by rotating the vehicle (180 deg) to produce disturbance torques in the opposite direction. Other techniques for non-maneuvering vehicles that are used are balancing two disturbance torques (e.g., aerodynamic and gravity-gradient) however, for MMOSTT, the large spin maneuver will complicate the use of balancing disturbance torques. Another possible desaturation technique would be to use a combination of reaction wheels and energy storage transfer techniques with the power system. Two reaction wheels per axis spinning in opposite directions would be used to add desaturation torque when necessary and each reaction wheel would be chosen depending on required torque direction and whether energy would be added or captured in the required reaction wheel.

Some other possible secondary torque sources are magnetic torquers, aerodynamics, and tether attachment torques (would require tether thrust-line offset with respect to control station cg). Magnetic torquers might be very massive but they are used in many satellites for nutation control, and if sized large enough could be used. Aerodynamic torques would only be present during the perigee part of the orbit and might not be large enough to help. The current mass properties report includes the CMGs for attitude control but does not reflect the secondary systems necessary for desaturation. Techniques for desaturation will need to be studied in detail in the next study phase.

3.1.3 Trades and Recommendations

Current and future developments in GNC sensors are expected to drive the size and mass of sensor units to extremely small values, thus the major driver in GNC sizing is expected to be the CMGs, RWs, etc. For instance, star trackers used on the Clementine vehicle were small and weighed only 3 lbs each including sun shade cover. Current small GPS/IMU systems weigh less than 10 lb.

The above discussion can be summarized as:

- CMGs baselined for attitude control actuation: more efficient than reaction wheels.
- CMG Sizing based on mass & inertia ratios with respect to ISS and Skylab heritage, MMOSTT required size is less than required for ISS and Skylab.
- Monitoring technology developments in reaction wheel storage capabilities.
- Current and future developments in guidance and navigation sensors are expected to drive size and mass of sensor units to extremely small values.

3.1.4 Recommended Follow-on studies

- Detailed ADCS modeling and sizing
- Detailed modeling of CMG control during spin maneuver
- Detailed Study of Desaturation Techniques

3.2 Command and Data Handling (C&DH)

3.2.1 Design Requirements, Drivers, and Assumptions

Due to 10-year mission design requirement a highly reliable system is required. Depending on grade of equipment utilized a dual, triple, or quad redundant system will be utilized. Performance estimates of 20 MIPS (Million Instruction Per Second), place the processing requirement at the high end for today's space flight controller, but easy achievable by 2005-2010 time frame. Even multiplying today's MIPS estimate by five shall not pose a design issue as several 100 MIPS space flight controllers are presently being developed and should be available by 2005.

3.2.2 New Issues and Requirements Identified

The control station computer shall process data from and /or command the following systems. It shall receive commands and process data via the telemetry system. Data from the attitude sensors shall be utilized to command attitude and control devices, such a control mass gyros or thrusters. Through the PMAD (Power Management And Distribution) system the flight computer shall control power to various subsystems and tether systems, such as ED Tether power system and tether deployer/reel. It shall also control various other subsystems, such as the thermal control system. Also, each flight computer shall maintain its own internal fault detection system and an external fault system to determine sensor failure and failure of companion devices, such as the other redundant flight controllers.

3.2.3 System Level Design

Figure 3-3 shows the command and data flow, with the Control Station Flight computer controlling all data flow and commands between the ground and other flight components, such as the Grapple Assembly Computer. The Grapple assembly computer will have telemetry and sensor contact with the PAA computer during the rendezvous activities. The PAA computer, via host/carrier vehicle will have some contact with the ground prior to and possibly during rendezvous.

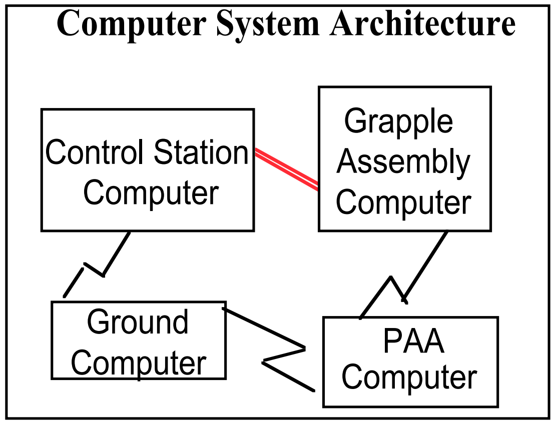


Figure 3-3. Control Station Computer System Architecture

3.2.4 Recommended Follow-on studies

No follow-on studies are presently recommended as the technology required is an upgraded version of existing technology and should be available prior to the MMOSTT implementation date.

3.3 Tether Deployment/Control

3.3.1 Design Requirements, Drivers, and Assumptions

- Design Driver 1: Tether length and diameter (100 km long, think diameter tapers from 5.6mm to 4 mm)
- Design Driver 2: Launch vehicle Dynamic Payload Envelope (assumed Delta IV-H, 5 m fairing)

This phase concentrated on tether reel sizing. Assumption:

- Current tether reel has a drum of diameter = 1m, and width = 1.5m.
- The diameter of the drum plus wound tether = 68.3 inches

3.3.2 New Issues and Requirements Identified

- Method of electrically grounding the reel assembly and rest of Control Station while high power is being applied to the electrodynamic tether during orbit raising.

3.3.3 Trades and Recommendations

- Trades w/ Reel sizing program
- Show some test run results and convergence to 1 meter diameter
- Recommendation: reel drum diameter of 1 meter with the motor housed within the ID of the reel drum.

3.3.4 Recommended Follow-on studies

- More detail in tether reel mechanism design.
- Know that a tether guiding mechanism or system is needed. Proposed design solutions include (1) a fishing reel-type reel design with a structural guide that shuttles back and forth as the tether is wound onto or off of the spool and (2) a system of pulleys guiding the tether at a specified tension off of the spool.
- How much electrical power will it need for drum to turn and being deploying tether (with copper or aluminum wire wound into last 20 km, will tether want to stick or jam in tether guides?)

3.4 Electrical Power System (EPS)

3.4.1 Design Requirements, Drivers, and Assumptions

Major design drivers are the high power requirements of the ED Tether and reel/deployer mechanism. An assumption is made that both the ED Tether thrust mode and reel/deployer will not be utilized simultaneously. Power estimates for the Tether reel vary dramatically depending on the size of the tether and rate of reel-in. While energy required for initial reel start-up can exceed that of the ED Tether, it is assumed that the reel on a per orbit average, will be equal to or less than that required of the ED Tether. Thus the power system is designed for nominal worse case ED Tether usage. Present design assumes ED Tether power of approximately 300 kW.

3.4.2 New Issues and Requirements Identified

Do to high power requirements of ED Tether and reel, several issues were identified. High power space DC/DC converters do not presently exist. Although converters in the 100kW class and at high voltage are on the drawing boards for such space applications as space based radar. Also the high voltage required for the ED Tether creates other issues in space. Interaction with space plasma at 20kV is a significant design driver. The potential for discharge and corona effects will drive insulation requirements, thus increasing system weight.

Other design drivers are thermal control. In order to drive down weight and cost, the batteries selected are being utilized at a high depth of discharge. In order to get both 10 years of life and a high depth of discharge, an active thermal control system is required for the batteries. Optimal life is reached at approximately 25degrees C. Also due to the high power of the system, items like the DC/DC converter can create than 10 –20 kW of waste heat in a small area. Thus an efficient thermal system is required for the power systems. Controlling the temperature of the DC/DC converter and power switching electronics is also critical to reliability. Reliability drops dramatically as temperature goes up.

3.4.3 Trades and Recommendations

As mentioned previously, the battery life and reliability of the electronic components are directly effected by their temperature. Design trades like increasing performance, and thus size, of the thermal systems vs. decreased the number of batteries and redundant electronic units, thus decreasing their size and weight, need to be further researched and traded.

Depending on duty cycle and whether the tether is ON during Earth eclipse can significantly drive power system requirements, particularly in regards to power storage requirements. Figure 3-4 shows the relationship between the power system weight for the batteries and solar arrays, for a 300 kW ED Tether, vs. the percentage of the ED Tether Thrust that is ran during Earth Eclipse. Note: Performance factors such as whether the ED Tether is more efficient during certain parts of the orbit were not taken in to account for graph below, thus actual savings may not be as large as indicated.

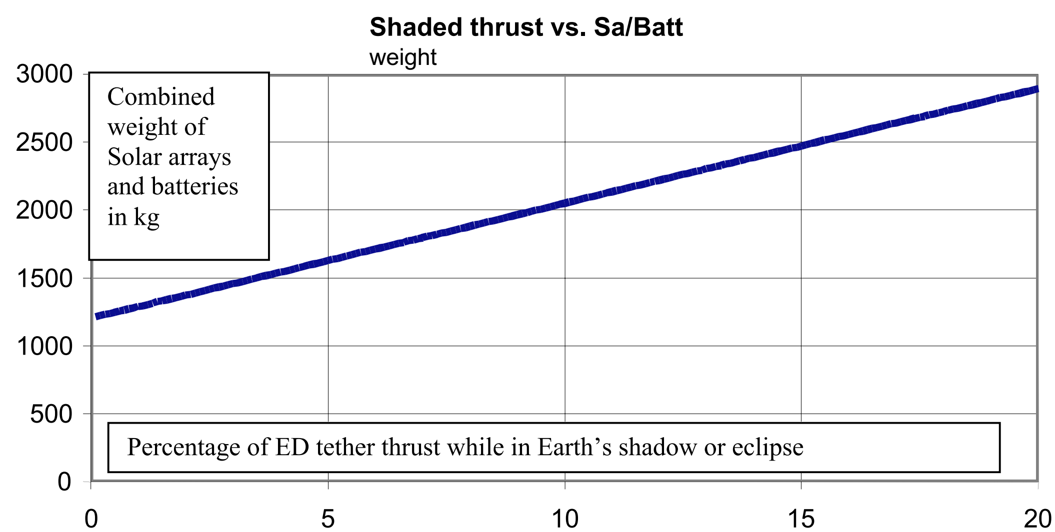


Figure 3-4. Solar Array & Battery Weight vs. Percent Thrusting in Shade

Other trades conducted include the choice of power storage system. The new lithium-ion batteries are the clear choice in regards to batteries, the traditional choice for space craft energy storage. However, alternatives energy storage options were also researched. Flywheels, while an immature option, have the potential to displace lithium-ion batteries. They also could be used for alternate functions, such as replacing flywheels for attitude and control. By placing two counter rotating flywheels and slowing down or speeding up just one of the flywheels, you can also use the energy storage flywheels to supplement the attitude and control flywheels (or control mass gyros). The charts below shows items tracked and approximately how they relate.

PARAMETER \ TECHNOLOGY	FLYWHEEL	NiH_2	NiCd	NiMH	Li ION	Turbine
MATURITY (YEAR)	2002	1980	1970	1995	2001	TBD
FLIGHT HISTORY	NONE	LOTS	LOTS	NONE	Near term	NONE
MASS (Kg/kW)	5-10	53.2	374	53.2	7-9	TBD
VOLUME (M^3/kW)	0.028 M^3	0.08 M^3	0.14 M^3	0.04M^3	0..04 M^3	TBD
DEPTH OF DISCHARGE	60%	30%	8%	20%	30%	TBD
COST (00 $'s) / 40 AH	TBD	$300,000	$300,000	N/A	$350,000	TBD
SYSTEM ENG. DENSITY	72 WH/Kg	14 WH/Kg	2 WH/Kg	14 WH/Kg	45 WH/Kg	TBD
ROBUSTNESS	>10 YEARS	>10 YEARS	10 YEARS ?	TBD	>10 years	TBD
SCALE ECONOMIES	HIGH	MED.	MED.	HIGH	HIGH	TBD

Good potential (FLYWHEEL)

Starting point (Li ION)

NiH_2 - Nickel Hydrogen
NiCd - Nickel Cadmium
NiMH - Nickel Metal Hydride
Li ION - Lithium Ion

Figure 3-5. Energy Storage Options

3.4.4 Recommended Follow-on studies

Items like the energy storage trade study and thermal transfer should be periodically (every one-two years) revisited to determine if a technology breakthrough has allowed one technology to displace the other as the system of choice.

Also methods of converting the considerable waste heat generated by the high power electrical to useful energy should be explored. A quick look was performed to determine if a Sterling Cycle Engine or similar device could be used to convert waste heat back to useful energy. Preliminary results indicate the delta-T is borderline for efficient energy conversion. This is further covered in the thermal control section.

3.5 Thermal Control

3.5.1 Design Requirements, Drivers, and Assumptions

Many of the subsystems of the control station will only require passive thermal control using the normal techniques of material selection, coatings, and thermal insulation. Removing waste heat from the batteries and power conversion unit will be the main driver in thermal system sizing. This waste heat is expected to be on the order of 35-40 kW minimum and will be in a fairly compact area. The support area for the batteries and power conversion unit must be limited as much as possible to limit overhead mass and limit thermal capacitance of the support structure for a passive system or cold plate for an active system. Using current and near-term technologies, to remove this amount of heat from a limited area will require a fluid loop interfaced to a set of heat exchangers. The heat exchangers would then be interfaced to a set of radiators as shown in Figure 3-6. The desired battery temperature and range is 25°C ± 5°C. The power system design requires that the battery temperature range be limited in order to achieve desired battery life.

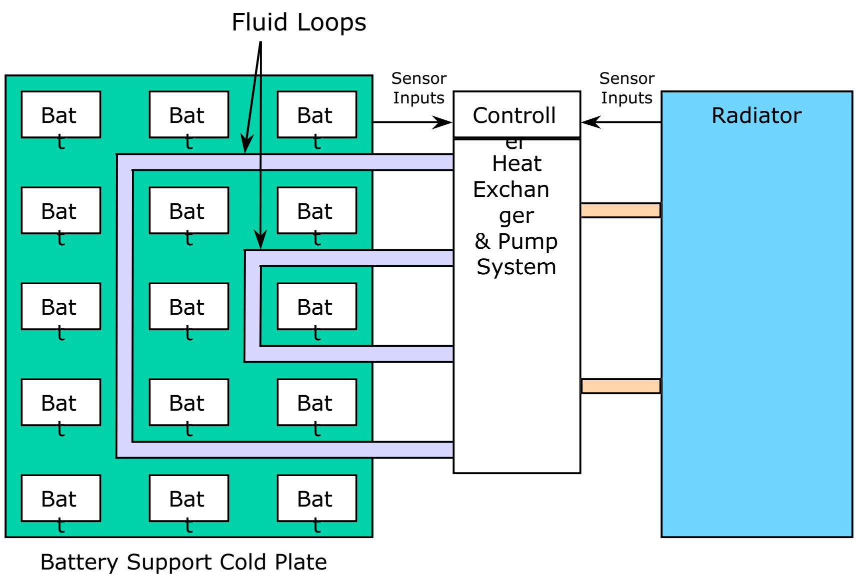


Figure 3-6. Thermal Control System Schematic

The 10 year life requirement will require an extremely reliable heat exchanger/pump system and the cost and complexity would be similar to the fluid loop systems used on ISS. A similar heat removal loop will be needed for the power conversion unit as shown in Figure 3-7. The desired power conversion unit temperature and range is 80°C ± 5°C. The power system design requires that the power conversion unit temperature range be limited in order to achieve desired reliability.

The two heat exchanger systems can be designed to work on the same fluid loop with heat removal occurring in the battery units first as shown in Figure 3-8. A trade study will need to be conducted in the next design phase to see if this is feasible. The design driver for a combined system sizing is the amount of heat that must be removed from the battery stage and the amount fluid flow required to meet the inlet temperature to the power conversion unit stage.

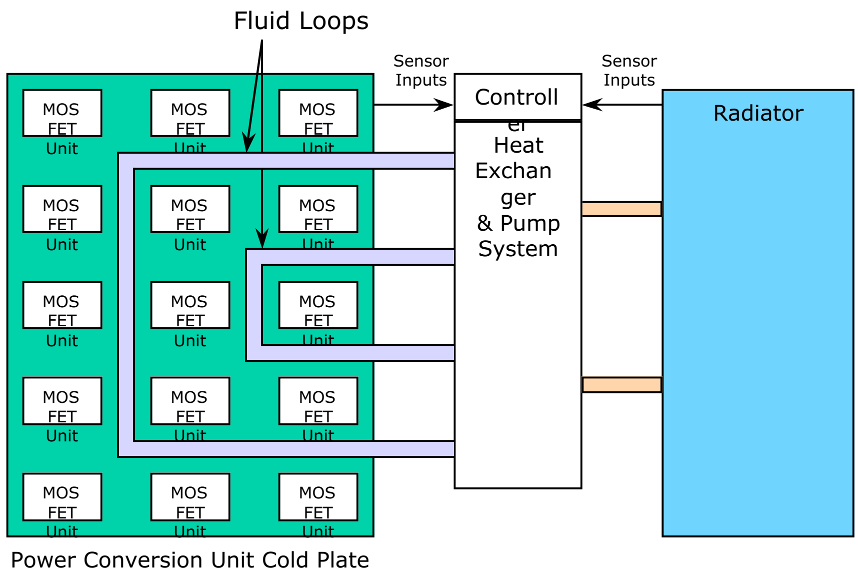


Figure 3-7. Control Station Heat Removal Loop

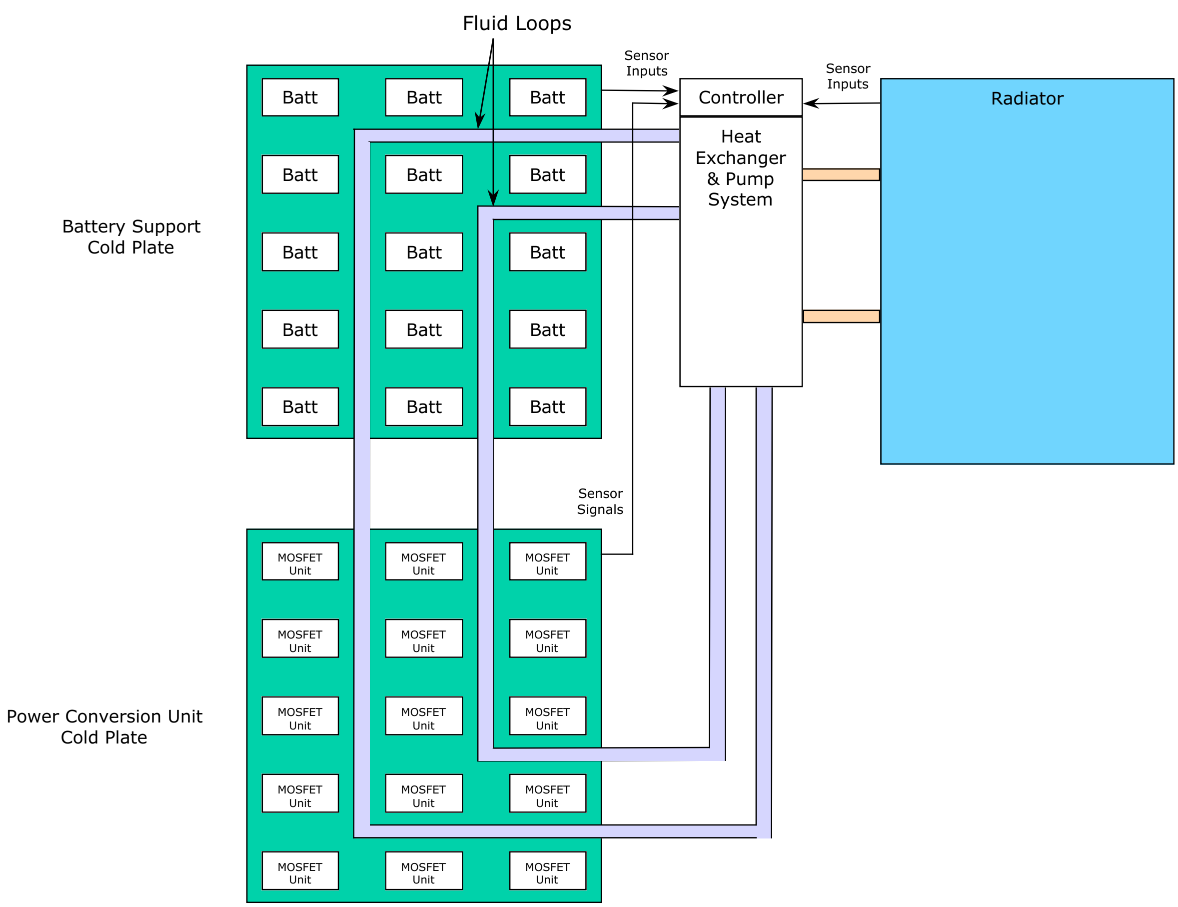


Figure 3-8. Dual Heat Exchanger Concept.

If heat pipe technologies continue to advance, they may help in minimizing the system complexity and overhead associated with the concentrated heat removal problem. The limiting factor associated with using heat-pipes is that they depend on heat transfer across an area whereas an active fluid loop enhances the heat transfer rate by using mass transfer within the fluid loop. For a dense packed piece of equipment such as the control station battery units and power conversion units this can be problematic since enough surface area for heat transfer is hard to obtain. An example of a constant conductance heat pipe (CCHP) setup is shown in Figure 3-9 along with the illustration (Figure 3-10) of the simplicity obtained in the system. However, the tradeoff for using heat-pipes are that temperature range control is passive and controlling to a tight temperature range may not be obtainable. A mix of constant conductance and variable conductance heat-pipes (VCHP) is better than using CCHPs alone.

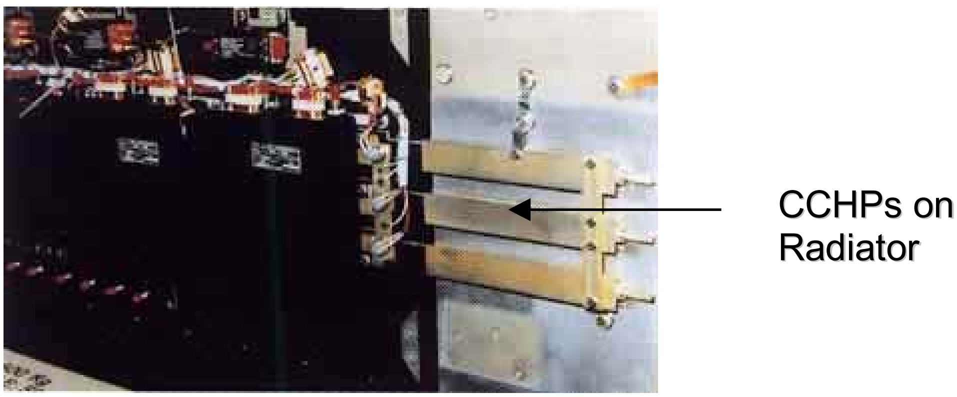


Figure 3-9. Constant Conductance Heat Pipes on radiator.

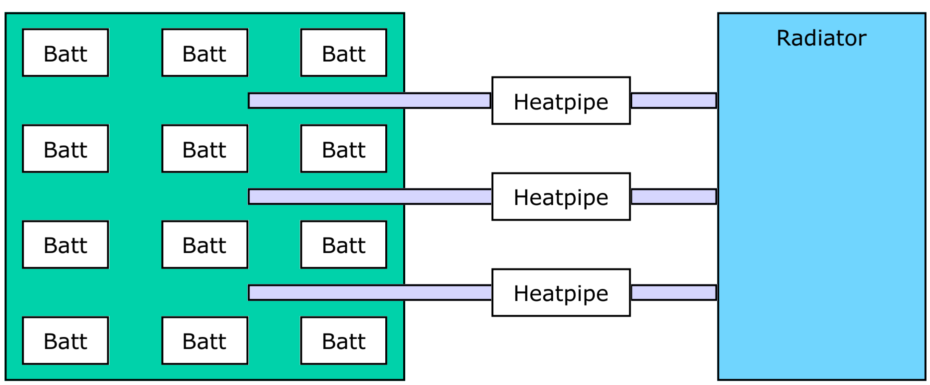


Figure 3-10. Constant Conductance Heat Pipe design.

One possible approach would be to use a combined system. Using heat-pipes to remove as much heat as possible assisted by an active system to achieve tight temperature might significantly reduce the size of the active part of the system. This option should be studied further.

Another design under consideration is using the waste heat to drive a Stirling engine/ generator combination which would reduce the amount of waste heat to be removed by radiators and would convert a large fraction of the waste heat into useful work to drive other subsystems. Stirling engines are currently being used in solar energy conversion projects and we will be monitoring this technology to see if the reliability and life of the Stirling engine would be adequate.

3.5.2 New Issues and Requirements Identified

Removal of 35-40 Kw of concentrated heat has not been designed and demonstrated in an on-orbit thermal control system. The International Space Station used an active system with heat exchangers but is designed for much less waste heat removal and much less waste heat density. Removal of concentrated waste heat without incurring large mass penalties will require innovative techniques.

3.5.3 Trades and Recommendations

- Heat exchangers vs. Heat-pipes vs. combination of both
- Mix of CCHP and VCHP configuration trades
- Stirling engine usage and sizing

3.5.4 Recommended Follow-on studies

- Detailed Study of Sizing and Performance Trade-offs of Heat exchangers vs. Heat-pipes vs. combination of both
- Detailed Study of Sizing and Performance of Mix of CCHP and VCHP configurations
- Detailed Study of Stirling engine usage and sizing for conversion of waste heat

3.6 Structure/Configuration/Mass Properties

3.6.1 Design Requirements, Drivers, and Assumptions

The Control Station structure supports subsystem components, and protects these components from the launch environment and on-orbit loading environment from Facility rotation. The Control Station must be configured to accommodate future on-orbit expansion (mass addition) and will be composed of aluminum and composite structures. Equipment bays will have aluminum isogrid shelves for mounting avionics and routing cables. The deployable solar array supports will also have cable trays. Main design drivers are as follows:

- Current Facility expansion concept (mass growth) — Drives placement of arrays, radiators, and FEACs
- Size of reel, CMG's — Sizes equipment bays
- Total PV array area — Drives launch configuration, and determines bulk of structure required to support gimbaled arrays and transmit on-orbit loading

- Waste heat from power conversion for ED thrust — Drives thermal control design/mass which impacts station configuration and structural design of ED tether bay

The Control Station was assumed to follow the 41:20 Resonance Design mission profile provided by TUI and was assumed to be launched by a Delta IV-H on a due east launch to a 600 km circular orbit, 28° inclination.

3.6.2 Mass Properties Statement

The mass properties, summarized in Table 3-1, show an operational Facility (Control Station, tether, Grapple) that can be launched with a single launch of a Delta IV-Heavy is feasible. Mass margins between 25-30% at this level of conceptual design are desirable. Note the two overall mass margins labeled "Bottom-up Calculated Margin," and "Available Margin," in Table 3-1. The mass properties were estimated using a bottom-up approach, starting from the component level of each subsystem. Mass margins were placed upon each subsystem component; the sum of these component-level margins expressed as a percentage of estimated Control Station mass is the "Bottom-up Calculated Margin." The "Available Margin" is the allowable mass growth for a single launch on a Delta IV-H. The lower "Bottom-up Margin" indicates the estimated Control Station mass may be low, but the fact that the "Available Margin" is higher indicates that there is reasonable margin for mass growth. Table 3-2 shows a more detailed mass properties statement.

3.6.3 New Issues and Requirements Identified

As design development continues, the launch configuration is growing in importance. A quick look at launch configurations showed that the current concept looks promising, but still is not geometrically sized to fit within the Delta IV-H dynamic P/L envelope. This was not unexpected, with the level of subsystem definition at this phase. The only other new configuration requirements involve PV array and FEAC packaging requirements. The FEAC panels must be sealed prior to flight, with a minimum requirement of a GN2 environment. This leads to derived requirements for FEAC packaging: a vented housing with a removable cover to expose the FEAC panels to space. FEAC packaging is expected to be a beefed up box with a mechanized cover release and panel deployment system. The packaging requirements were taken into account for the FEAC mass estimate.

Table 3-1. MMOSTT Control Station Mass Properties Summary

	41:20 Resonance Design
Due east launch, 350 km circular orbit at 28-deg inclination	
Delta IV-H Useful Load	23,768
Delta IV Payload Adapter Fitting mass	386
Delta IV-H Max Payload mass (GLOW)	23,382
Grapple Mass (assumed a constant)	650
Tether Mass (assumed a constant)	8,274
Maximum Control Station Mass (Delta IV-H Max Payload Mass – Grapple Mass – Tether Mass)	14,458
Estimated Control Station Mass (kg)	
No Margin	10,967
Bottom-up Calculated Margin	2,300
Avg. Margin	21%
Mass with Bottom-up Margin	13,267
Available Margin Available Margin = 1-(Maximum – Estimated Control Station Mass)	32%
Estimated GLOW vs. Delta IV-H Payload GLOW	
Estimated GLOW (No margin C. Station) *3.6.3.1.1.1.1 GLOW = Control Station + Tether + Grapple mass*	19,891
Margin	3,491
Avg. Margin	15%
On-Orbit Mass	
Expended Upper Stage Mass	3,467
On-Orbit Mass (No margin C. Station) On-Orbit Mass = Control Station + Tether + Grapple + Expended Upper Stage Mass	23,358

Table 3-2. MMOSTT Mass Properties Breakdown

	Qty	Redun-dancy	Mass Contin-gency	Unit mass (kg)	Mass with no margin (kg)	Mass with Contin-gency (kg)	Mass Margin (kg)
LEO Control Station					**10967**	**13267**	**2300**
Thermal Control Subsys	**1**		**15%**		**1104.5**	**1270.1**	**165.7**
Cabling/Harnesses			33%		**749.6**	**997.0**	**247.4**
Structure			25%		**2721.1**	**3401.3**	**680.3**
Electr.Pwr.					**4736.7**	**5409.6**	**673.0**
PV array panels	1	1	13%	1782.9	1782.9	2014.6	
Power Storage	1	1	15%	2860.5	2860.5	3289.5	
PV array drive motors	8	2	13%	3.0	48.0	54.2	
PMAD	1	2	13%	22.7	45.4	51.3	
Downlink Comm Subsys					**1.8**	**2.1**	**0.2**
Downlink Transceiver	1	2	13%	0.7	1.4	1.56	
Downlink antennae	2	1	13%	0.2	0.5	0.51	
TFS Net Comm Subsys					**1.8**	**2.1**	**0.2**
Comm. antennae	2	1	13%	0.2	0.5	0.51	
Transceiver	1	2	13%	0.7	1.4	1.6	
C&DH					**26.0**	**29.4**	**3.4**
Computer	1	2	13%	13.0	26.0	29.4	
TT&C					**6.9**	**7.8**	**0.9**
transponder	1	2	13%	3.5	6.9	7.8	
ADCS					**200.9**	**213.8**	**12.9**
sun sensor	1	2	1%	0.3	0.5	0.53	
CMG	2	1	5%	90.7	181.4	190.5	
Inertial Navigation Unit	1	2	21%	8.4	16.8	20.3	
GPS antennae	3	1	13%	0.3	0.8	0.9	
GPS transceiver	1	2	13%	0.7	1.4	1.6	
ED Tether Power Subsys					**417.4**	**603.4**	**186.0**
Plasma Contactor (FEAC)	1	2	25%	45.4	90.8	113.5	
PMAD/PCUt	1	2	50%	163.3	326.6	489.9	
(Optional) separate tether for ED reboost	1	2				0.0	
(Optional) separate tether's reeling assembly	1	2				0.0	
Docking & I/C Subsys					**0.5**	**0.54**	**0.04**
Beacon	1	1	8%	0.5	0.5	0.54	
Tether Deploy & Control					**1000.0**	**1330.0**	**330.00**
Tether reeling assembly (motor, etc)	1	1	33%	1000.0	1000.0	1330.0	

3.6.4 Trades and Recommendations

The usual parametric trades conducted at subsystem design levels were not conducted for the configuration trades, except for reel sizing which drove the tether bay size. The final Phase II configuration resulted from iterating the configuration from phase I and seeing if each iteration met the criteria shown in Table 3-3. Figure 3-11 shows the first configuration that resulted after the first few subsystem sizing iterations (the “guppy” configuration). This first configuration did not place deployable PV arrays, radiators, and FEACs in a manner that met Configuration Guidelines #5 (array shading). Configurations 2 and 3, which generally follow the Configuration Guidelines, were drafted and are shown in Figure 3-12 and Figure 3-13, respectfully. The main concern with configuration 2 is the occasional array shading. Though Configuration 3 has no array shading, rotating such a large collection of panels may not prove to be advantageous over Configuration 2. Rotating the panel collections will require motors and stiff structural supports. Array gimbaling component lifetimes, reliability, power requirements, and extra mass are the parameters which need to be defined. Mass properties reported above do not reflect Configuration 3. A closer view of Configuration 2 with component labels is in Section 4.0.

Table 3-3. Configuration Guidelines

1.	Tether reel realistically sized, not prohibiting single launch of operational system
2.	Reel Bay sized to accommodate Tether reel
3.	Reel and Equipment Bays provide ample mounting surfaces for antennae/sensors/bracketry
4.	PV array panels deploy and placement does not prohibit future Station sections from being added
5.	PV arrays able to be sun-pointing for a considerable part of an orbit
6.	Radiators are body-mounted to Reel Bay and are deployable
7.	Radiators and FEACs face away from PV arrays
8.	Radiators and FEAC placement do not prohibit future Station sections from being added
9.	Tether should not wrap itself around Control Station

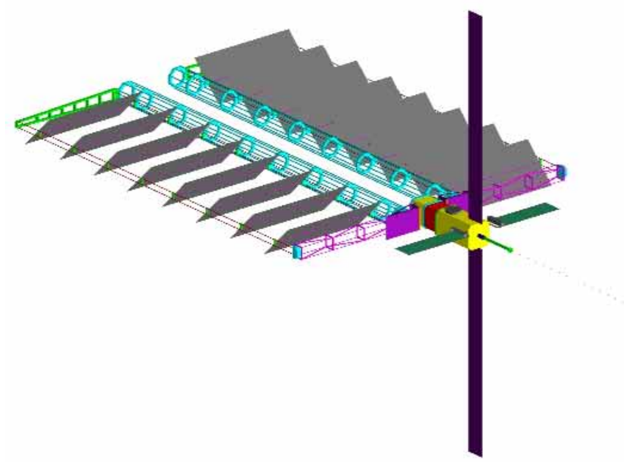


Figure 3-11. Control Station Configuration 1

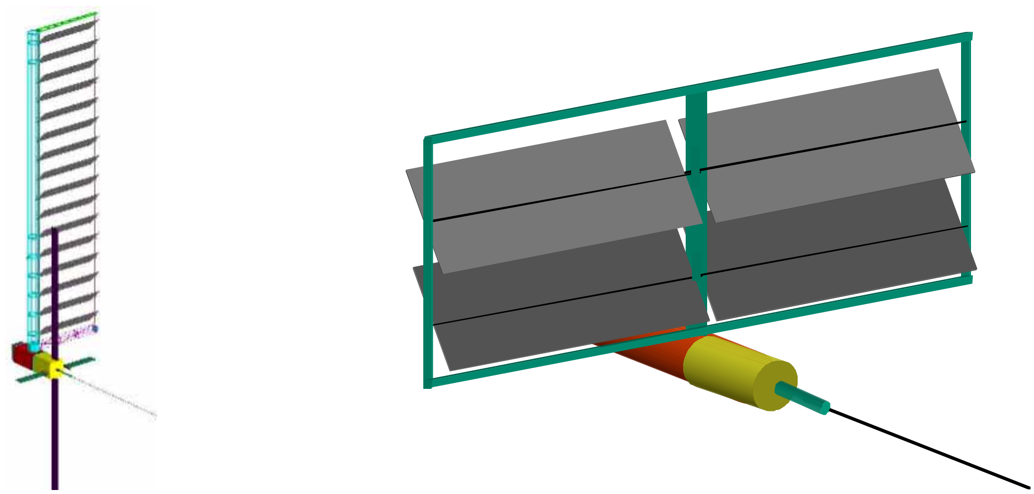


Figure 3-12. Control Station Configuration 2

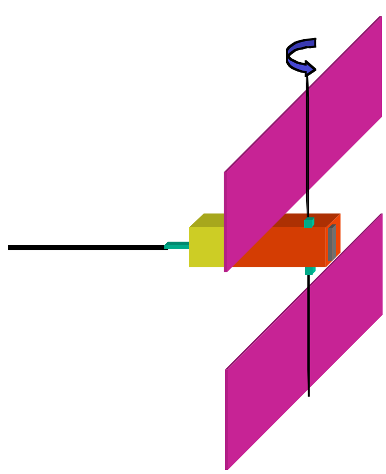


Figure 3-13. Control Station Configuration 3

3.6.5 Recommended Follow-on studies

More detailed trades within the tether deployment, attitude control, and thermal control subsystems are recommended to drive the structural/configuration in future studies. Future configurations should trade configurations 2 and 3 above, making sure to characterize the issues of array shading, array gimbaling power requirements, and Control Station evolvability. Future design iterations should also include launch configuration updates as a volumetric feasibility check against the "launch an operational system with a single launch" requirement. Packaging trades for folding, stowing, and deploying the PV arrays, FEACs, and radiators should also be performed. FEAC technology development with respect to manufacturability into deployable panels should continue to be monitored.

3.7 Facility Evolvability

3.7.1 Design Requirements, Drivers, and Assumptions

Facility evolvability from the standpoint of Control Station mass growth began to be addressed during Phase II with the concept of "Expansion Modules." No studies on "docking with a rotating Station" vs. "spinning down the Station for module docking" have been performed to date. The current concept for Expansion modules assumes that Expansion Modules will be added as the Control Station rotates at some nonzero rotation rate, and that this docking is performed as an autonomous, unmanned mission. This is a simplifying assumption, allowing certain complicated issues to be addressed in future studies. For example, the issue of exactly how to despin the Control Station has numerous design solutions that would need to be traded.

Error! Reference source not found. shows a possible evolved version of the Control Station Configuration 3 design. Additional mass modules are appended to the end of the control station. This mass modules can include spent rocket stages and other space debris, as long as any hazardous chemicals are first purged. In this manner, the center of mass of the tether facility is moved closer to the solar arrays, thus reducing the acceleration and stress on the solar arrays. Meanwhile, solar arrays grow along the axis of solar array rotation. While this ensures that the new solar arrays do not shadow the original set, it does create additional bending moments on the main solar array axis of rotation. All the more reason to move the center of mass of the tether facility closer to the solar arrays prior to adding additional power modules.

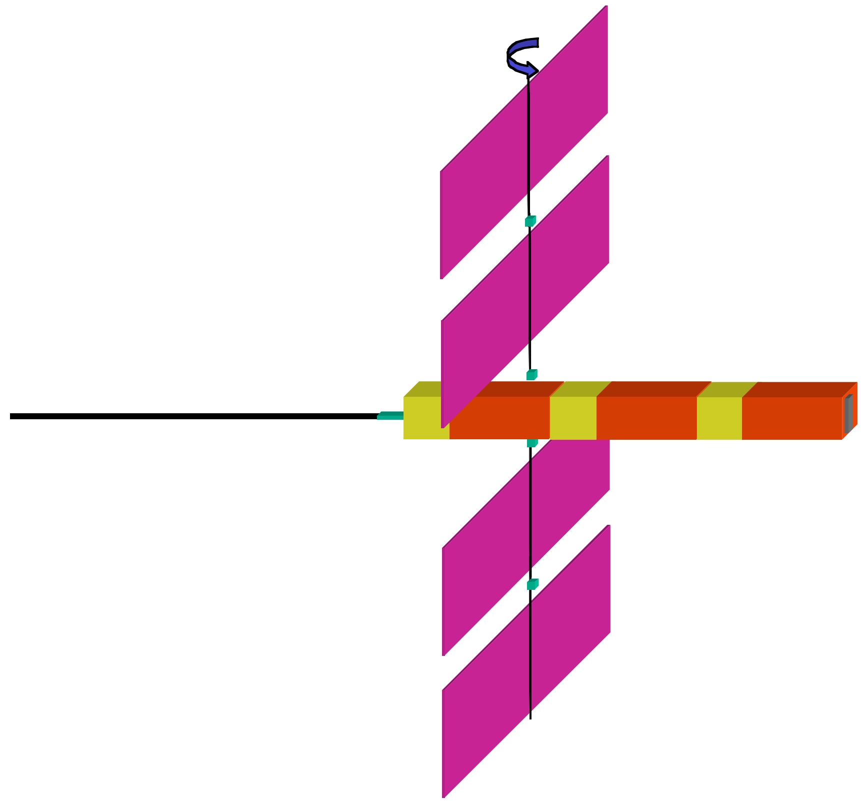


Figure 3-14. Evolved Control Station with additional mass modules and solar arrays.

An autonomous docking subsystem will require GN&C, ADCS, and a docking mechanism. In order for the Control Station to stay within the payload weight limits of an existing launch vehicle, it is suggested that the Control Station have the passive half of a docking assembly and that the Modules have the active half with the docking mechanisms. The docking mechanisms on the Modules require power, resulting in an EPS with power storage and distribution. The Modules' docking functions, and EPS all require C&DH. It is also assumed that and ground tracking of and ground communication with expansion Modules during pre-docking maneuvers is desired, requiring Modules to have an on-board ground communication subsystem. The potential difference between pre-docked Modules and the Control Station will more than likely cause some arcing. Therefore, an additional assumption is that Modules will need to be designed with a method of preventing arcing during docking procedures. A cold gas ACS is recommended if future studies show that docking maneuvers will need to be performed in close proximity to instrumentation or other delicate surfaces (PV arrays, for example), though a need for a cold gas ACS was not identified in this phase.

Thus far, an Expansion Module consists of components common to every satellite bus and an autonomous docking subsystem. More mass can be added to each Expansion Module as ballast, if so desired. Some other concepts for Control Station evolvability were brainstormed in an earlier technical interchange meeting (May 2000). Two examples of these brainstormed concepts are additional on-board tethers to increase the payload mass the station could toss and additional solar arrays and power storage with "plug-n-play" connections to link them into the existing EPS system (to increase power generation capability).

In summary, as a minimum, the following are the required on-board subsystems for a Module Assembly to initiate Control Station evolution:

- Autonomous Docking Assembly
- GN&C
- ADCS
- EPS (storage and distribution)
- C&DH
- Arc suppression

3.7.2 Recommended Follow-on studies

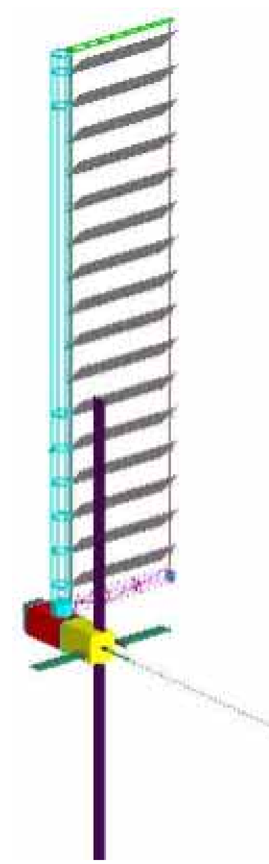
Evolvability Trade studies to answer the following questions are recommended for future study:

- Adding more tethers to the operating Facility vs. launching the first operational system with a tether already sized to accommodate ten times the 2500 kg baseline payload.
- At what point of mass growth will the Control Station require more power generating capability? How much power can we give the Station with a single Module launch?
- At what point of mass growth will the Control Station require more attitude control capability? How much more? Can this capability be increased with a single Module launch?

4 CONTROL STATION PHASE II DESIGN SUMMARY

Section 4.2 summarizes the current design features of the first launch of a partial capability Control Station. The Control Station (along with grapple and tether) is within mass budgets for launch on a Delta-IV Heavy Launch Vehicle. Primary features of the system are use of PV concentrator solar arrays, Lithium-Ion batteries, CMGs for attitude maneuver and control, GPS/INS for guidance and navigation input, and a tether subsystem that will support a 100 Km, 300 kW electrodynamic tether.

4.1 First Facility Launched



- Control Station mass = 13,267 kg (includes 21% mass margin)
- Operational mass = 23,358 kg, no margin CS w/PAF
- GLOW = 19,891 kg with 15% margin, no PAF

4.2 Features

EPS

- *Scarlet*-like concentrator PV arrays, 563 square meters
- Standard, state-of-the-art PV array drive motors
- State-of-the-art power management and distribution except for electrodynamic tether subsystem
- Lithium-ion battery power storage system
- 5,410 kg (includes 14%mass growth margin)

Communication Subsystem

- Downlink communication with ground station(s) and communication with Grapple Assembly and PAA (via Tether Facility Network)
- State-of-the-art, COTS hardware (antennae/transceivers)
- Dual redundancy
- 4.2 kg (includes 16% mass growth margin)

C&DH

- State-of-the-art, COTS hardware
- Dual redundancy
- 29 kg (includes 13% mass margin)

ADCS/GN&C

- 2 Control Moment Gyros (no redundancy), each assumed half size of a Skylab CMG
- 2 sun sensors
- 2 inertial navigation unit
- GPS antennae (3)/tranceivers (2)
- 213.8 kg (includes 6% mass margin)

Electrodynamic Tether Subsystem

- Sized for 80 km conductive tether, total length 100 km, 300,000 W, 40 μN/W thrust efficiency
- Control Subsystem with 1m diameter, 1.5m long reel, motor, tether guides, power conversion, FEACs
- 1,933 kg (includes 36% mass margin)

TETHER RENDEZVOUS METHODS

Rob Hoyt
Tethers Unlimited, Inc.

Abstract

Using a numerical simulation that includes models for orbital mechanics and tether dynamics, we have studied the dynamics of rendezvous between a payload in orbit and a rotating tether facility. In a tether-payload rendezvous, the relative motion between the tether tip and payload is primarily along the local vertical direction. The relative acceleration is constant, so, from the perspective of the payload, the tether tip descends to the payload, halts instantaneously, then accelerates away. The simulations indicated that tether deployment maneuvers can extend this "instantaneous" rendezvous to a window of tens of seconds, without need for propellant usage. We also studied the effects of the payload capture on the tether tension. The simulations indicated that for an ideal rendezvous, tension wave behavior will cause tension excursions roughly double that of the steady-state loads. If the rendezvous is not ideal, that is, if the tether must be deployed for several seconds while the payload and tether tip vehicle maneuver to achieve a docking, the resultant tension spikes can further increase the peak tether loads. Additional tether deployment maneuvers can help to ameliorate the peak tension excursions and damp the longitudinal oscillations.

Introduction

Rotating momentum-exchange tethers hold great potential for reducing the costs of in-space transportation by eliminating the need for transfer propellant for many missions. One of the primary technical challenges that must be accomplished if momentum-exchange tethers are to achieve their potential is the need to enable a payload to rendezvous with a grapple mechanism at the tip of the rotating tether. This rendezvous and capture maneuver is significantly more challenging than a standard orbital rendezvous, such as that between the Space Shuttle and the International Space Station, because whereas the Shuttle can take many orbits to gradually match its position and velocity with the ISS, a tether and payload must achieve rendezvous at a specific location, velocity, and time. The rendezvous windows available in a rotating tether system will be very short, requiring that the system must be able to predict and control the tether location to a very high degree of accuracy, and must be able to guide the payload to the desired location with the right terminal velocity. Once the payload and tethered grapple have come into proximity, they must then be able to maneuver and complete a secure docking within a very short window of time. In order to make this rendezvous and capture maneuver more feasible, in this document we investigate the possibility of using tether deployment maneuvers to extend the rendezvous window.

Baseline Tether-Payload Rendezvous

To illustrate the challenge of the rendezvous maneuver between the payload and tether, we have calculated the relative positions and velocities of a payload and tether tip grapple during a rendezvous using the TetherSim simulation.[i] In these simulations, we modeled a 100 km long tether, rotating with a tip velocity of 1 km/s, picking up a payload from a LEO orbit. Figure 1 shows the relative vertical and horizontal separations of the payload and grapple during an ideal rendezvous, where the trajectory of the payload has been specified so that it meets with the grapple at just the right position and velocity. The figure shows that the relative motion is predominantly along the local vertical direction; this vertical motion is illustrated more clearly in Figure 2, which plots the relative separations at 0.1 second intervals. In this rotating tether system, the tether tip experiences an acceleration level of approximately 1 gee, so one way to visualize this relative motion is to picture oneself reaching out from a fire escape and having a friend toss a ball up vertically so that it just reaches one's hand. There is just a brief second or so when the ball is close enough and moving slowly enough that you can catch it. Nonetheless, this "1-gee rendezvous and capture" is quite feasible for a low-tech human, and thus it should be feasible for an advanced autonomous rendezvous and capture technology.

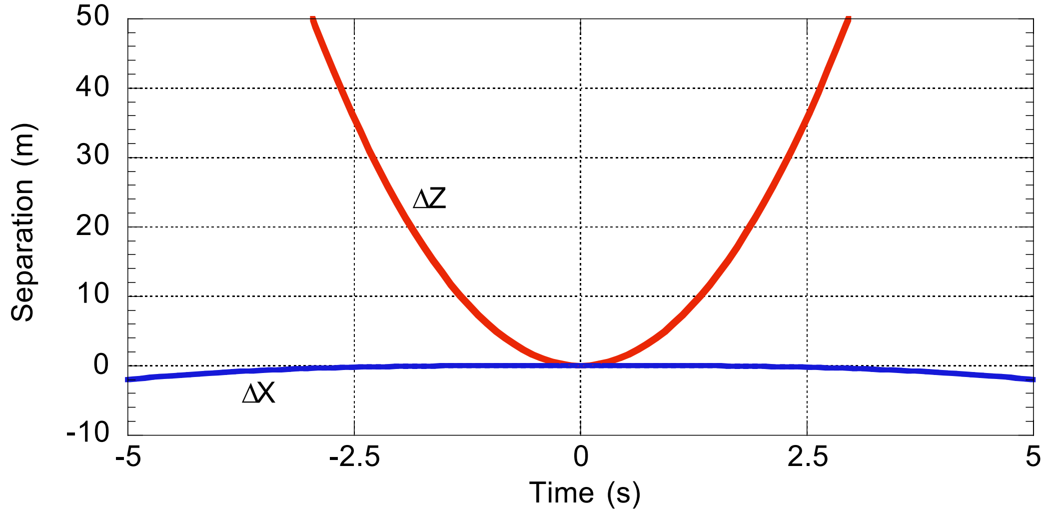


Figure 1. Relative separation between tether tip grapple and payload for a 100 km tether rotating at a tip velocity of 1 km/s.

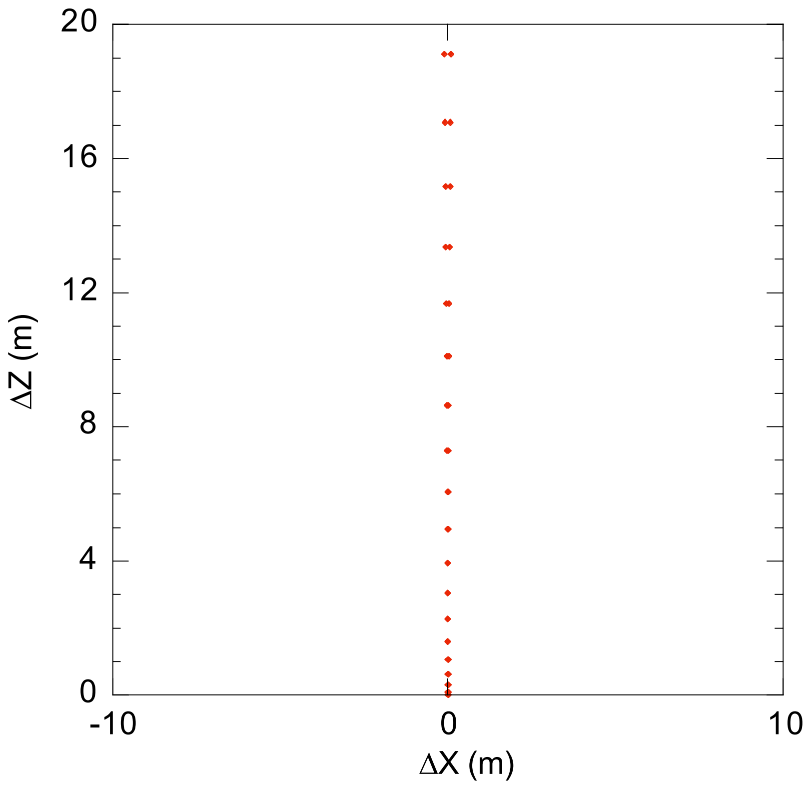


Figure 2. Relative vertical and horizontal separation of grapple and payload, plotted every 0.1 second.

Tether Deployment to Extend Rendezvous Window

Although the rendezvous shown in Figures 1 & 2 was an "ideal" case where the payload and grapple met perfectly, in real life, the payload and grapple will likely meet with errors in their position and or velocities. Consequently, it will be necessary to enable the system to deal with reasonable errors and still achieve rendezvous. Although automated systems may be capable of matching the position and velocity within the several-second window in the baseline rendezvous scenario, a tether transportation system may be made more technically feasible and salable to potential customers if the rendezvous window can be extended to a longer period.

While constant thrusting by either the payload vehicle or the tether tip vehicle could be used to extend the rendezvous window, the high acceleration levels between the payload in free-fall and the rotating tether tip would require high thrust levels from the rockets used and would present large total ΔV requirements on the system. Since the primary objective of the tether transport system is to minimize the need for propellant usage for in-space propulsion it is highly desirable to minimize the propellant requirements of the rendezvous method.

<u>Tethered Grapple Deployment</u>

One method for extending the rendezvous window that does not require the use of propellant is a tether deployment maneuver illustrated in Figure 3. In this method, the payload would be guided to rendezvous with a Payload Capture Vehicle (PCV) at the tether tip. When the payload and PCV reach their closest point of proximity, the PCV releases a tethered grapple fixture. This grapple tether will be deployed at the minimum deployment tension possible. This will place the grapple fixture in a nearly free-fall trajectory which will closely match the payload's free-fall trajectory until the grapple tether is fully deployed. The grapple fixture might contain some maneuvering capability; because it would be relatively light, a small thruster system could provide it with the ability to maneuver quickly to intercept the payload. With this deployment maneuver, it will be possible to extend the deployment window to several tens of seconds, depending upon the length of the tether and the tether tip acceleration level. An approximate rule-of-thumb for the resulting rendezvous window duration is

$$\Delta t \approx \sqrt{\frac{2l}{a}}, \qquad (1)$$

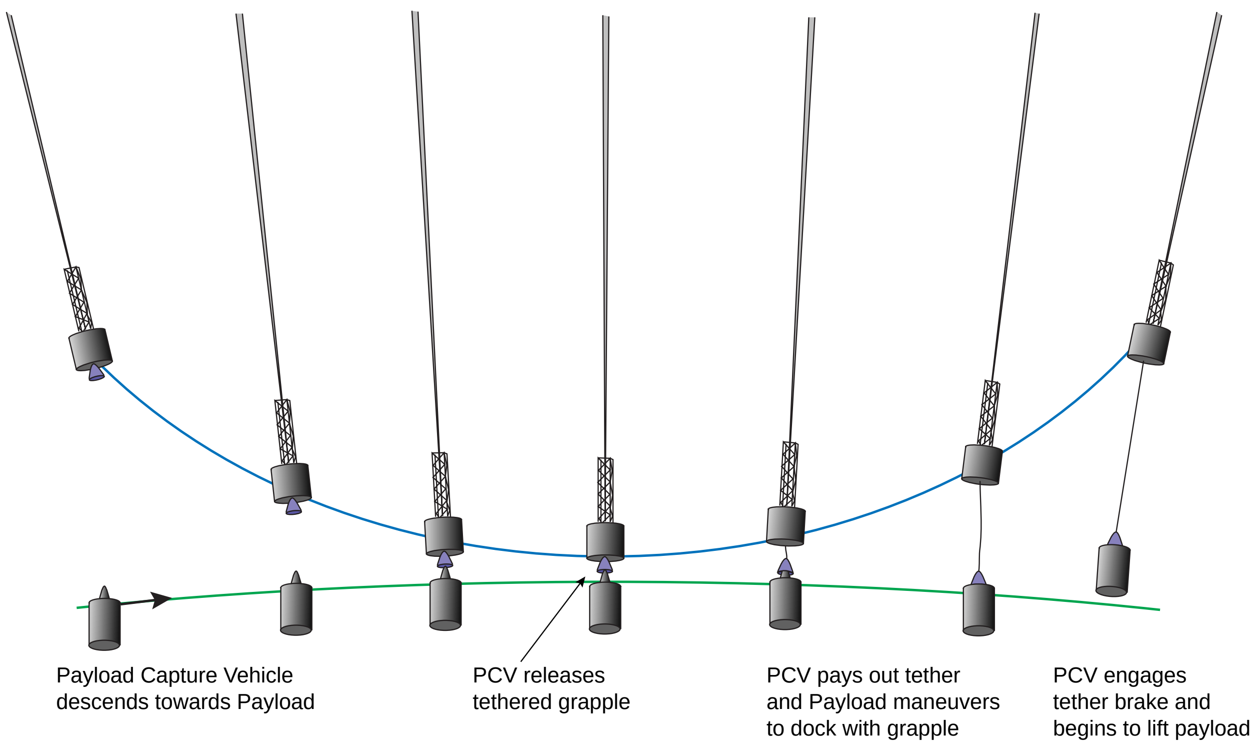


Figure 3. Schematic of rendezvous method where the Payload Capture Vehicle drops a tethered grapple into free fall.

where l is the length of the grapple tether and a is the tether tip acceleration.

Deployment of Additional Tether by PCV

Alternatively, instead of the PCV releasing the grapple on an additional tether, the PCV could release itself into free-fall by deploying a length of the main tether, as illustrated in Figure 4. The disadvantage of this second method would be that the heavier PCV would be less maneuverable than a smaller grapple fixture.

However, this second method could have an advantage in that it might be able to achieve a trajectory closer to "free-fall" than the previous method. When a tether is pulled off a SEDS deployer without braking, the deployment tension varies with the deployment speed roughly as:[ii]

$$T = \left[T_0 + I\rho \dot{L}^2 A_{rel}^{-E}\right] \quad (2)$$

where the first term in the brackets is a static (or minimum) tension and the second term is a tension dependent upon the square of the deployment velocity that is due to the inertia of the tether being pulled off of the deployer spool. Here, A_{rel} = 1-AL/L_{end}, L_{end} = total tether length, A = tether wind solidity (=0.942 for the SEDS winding), E is an area exponent = 0.8, I is an inertia multiplier = 4.1, and ρ is the linear density of the tether.

Equation (2) describes the tension on the *tether*. The force experienced by the deployer, however, can be significantly lower, because the deployer will not experience the force due to the inertia of the tether. As a result, a tethered grapple deployed from a PCV, where the deployer remains on the PCV, will experience a larger tether tension, and thus a larger acceleration relative to the payload, than a PCV that deploys more of the main tether.

Simulation Results

Using TetherSim, we modeled a rendezvous maneuver where the PCV begins with 1 km of tether stored on a spool. When it reaches closest proximity to the payload, it releases a brake on the deployer and deploys tether at low tension, dropping the PCV into free fall with the payload. Figure 5 shows the

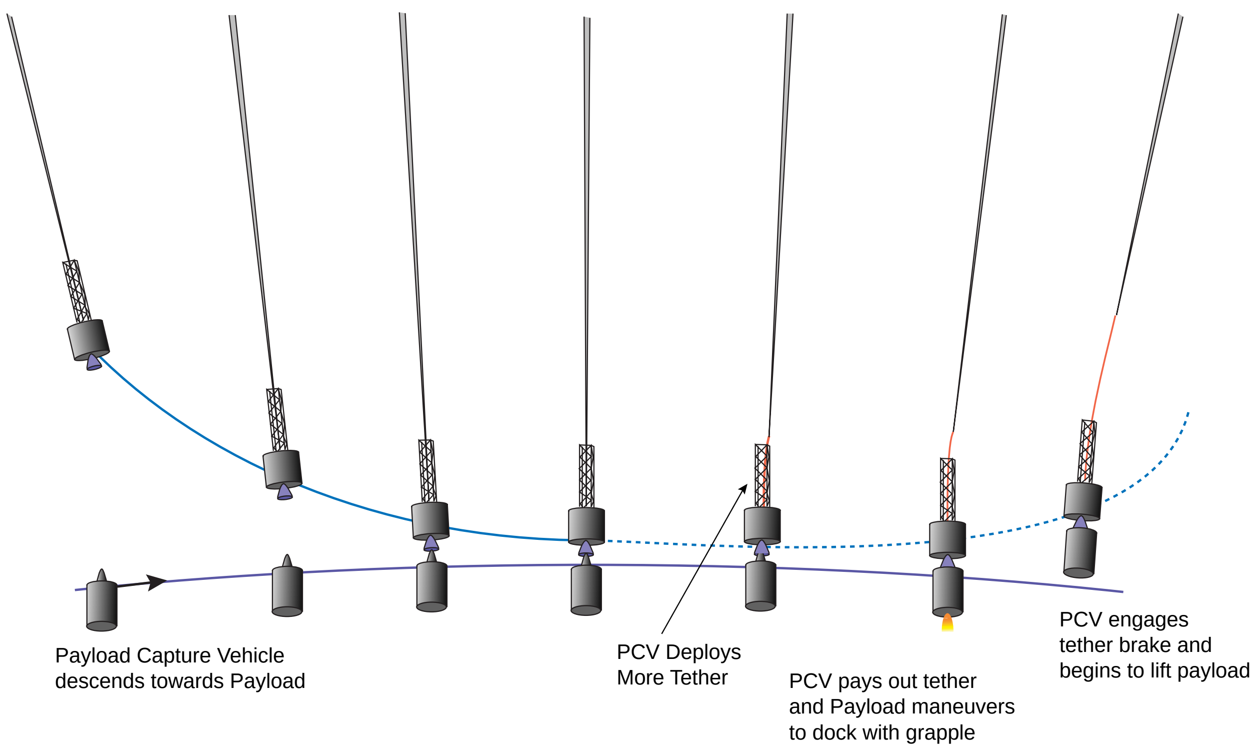


Figure 4. Schematic of rendezvous method where the Payload Capture Vehicle deploys tether to follow a free fall trajectory.

separation between the payload and PCV along the local vertical and orbital velocity directions. Comparing this graph with the same graph for the rendezvous without tether deployement shown in demonstrates that this deployment maneuver can extend the rendezvous windown by about 12 seconds; after 12 seconds the 1 km of tether runs out and the PCV is accelerated quickly away from the payload. In this simulation, the payload and PCV rendezvoused at T=0 with zero positional error, but a slight velocity error. This error, along with the slight force on the PCV due to friction of the tether leaving the deployer caused the PCV and payload to drift apart slightly during the 12 second window. The relative motion of the payload and PCV is illustrated more clearly in Figure 6, which plots the relative separation at intervals of 0.1 seconds. Like Figure 2, this plot shows that the relative motion is primarily along the local vertical. The payload could, however, perform small thrusting adjustments to counter this drift and achieve docking. Figure 7 shows the relative velocity of the payload and PCV in the seconds before the rendezvous.

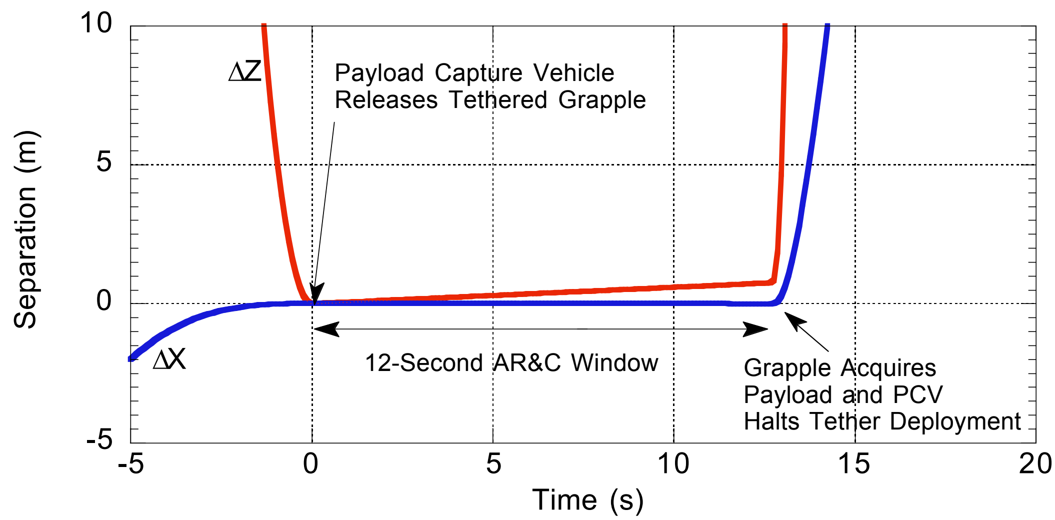


Figure 5. Separation between payload and tethered grapple, with deployment of 1 km of tether to extend rendezvous window to 12 seconds. (Z axis is local vertical axis)

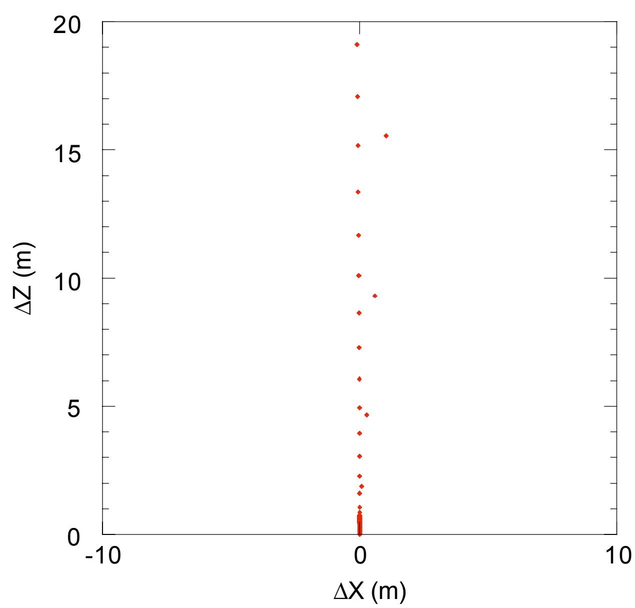


Figure 6. Relative separation between grapple and payload with tether-deployment for rendezvous window extension. Plots are every 0.1 seconds, and there are many points concentrated near 0,0.

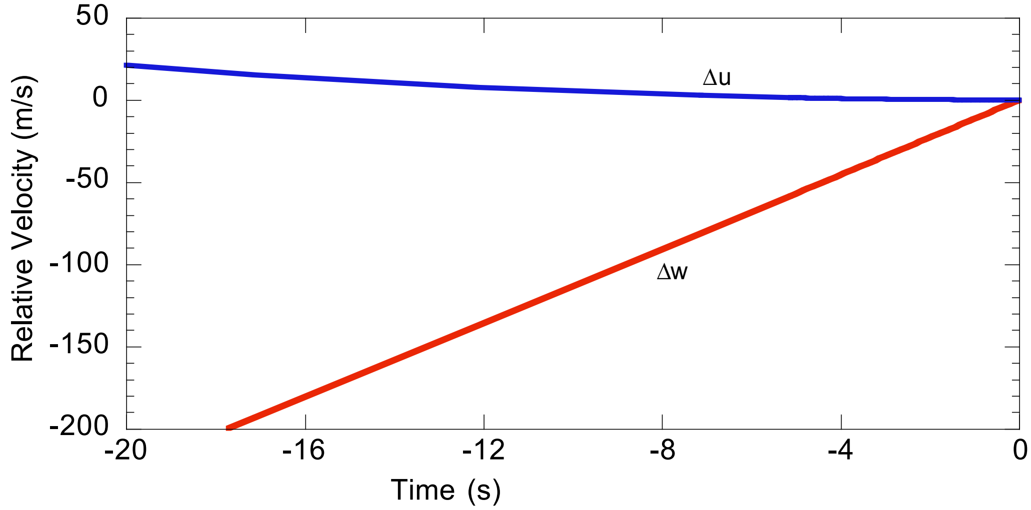


Figure 7. Relative velocity between tether tip and payload during approach. (Δw is velocity along local vertical)

Tether Tension Behavior

When the tether captures a payload, the tether system must handle a rapid transition from a state of low tether tension to a state of high tether tension. The sudden loading of the tether causes a longitudinal tension wave to travel up the tether. When this wave reaches the facility end of the tether it is reflected back, and the wave will propagate up and down the tether until it is damped either by passive dissipation in the tether structure or by active damping by the tether system. Of particular concern is the possibility that a reflected tension wave could superimpose upon itself and overstress a segment of tether, leading to tether failure. To examine this issue, we conducted a number of simulations of tether-payload rendezvous, varying the capture time and the methods of tether deployment to affect the tether tension behavior.

<u>Perfect Catch:</u>

The first case simulated was a "perfect catch", where the payload and tether tip meet at the appointed time with zero error in position and velocity, and the Payload Capture Vehicle acquires the payload immediately upon rendezvous. The tension behavior of the tether for this case is shown in Figure 8, where T=0 is the time of rendezvous. In this simulation, the tether was designed with a safety factor of 3.33 for the "static" load the tether will experience when loaded with the payload. Thus, under steady-state conditions, the load level will be approximately 0.3. The figure shows, however, that tension transients due to the payload capture cause the load level at the tether tip to exceed that level. Close inspection of the traces near T=0 reveals that the tension wave takes approximately 10 seconds to travel the 100km length of the tether. This wave then reflects back, reaching the tether tip at just after T=20 seconds, and causes the tip load level to reach 0.5. Figure 9 shows the longer-term evolution of the tether tension. The tension waves decay due to frictional damping and dispersion, reaching a steady-state level of approximately 0.3.

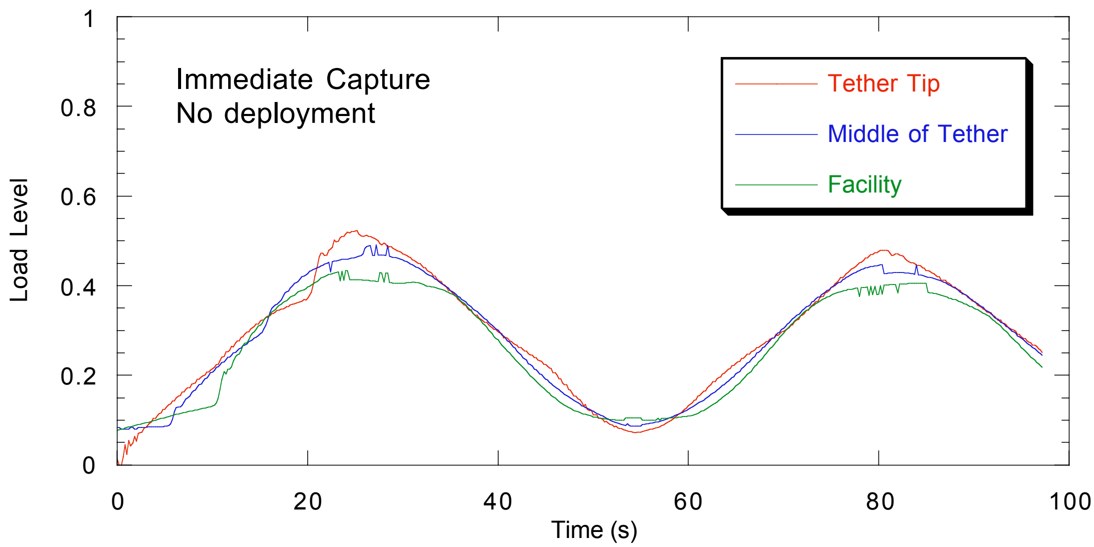


Figure 8. Load level (tension/break tension) of the tether after immediate payload capture in an ideal rendezvous.

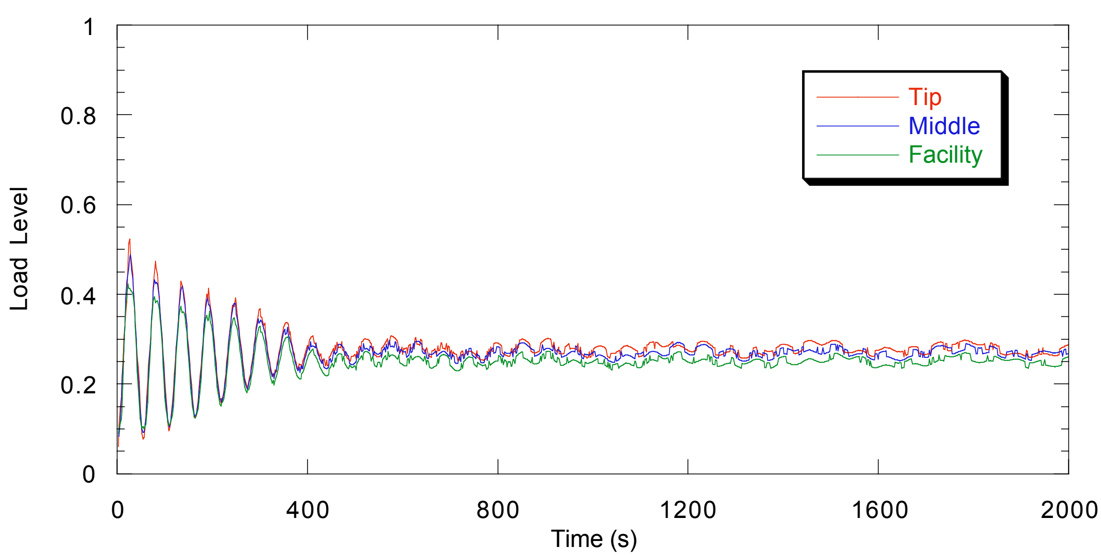


Figure 9. Tether load level after an ideal rendezvous.

Delayed Catch

In real-world missions, the payload and tether will likely have positional and/or velocity errors at the time of rendezvous. The system can utilize the tether deployment method described above to extend the rendezvous window for a number of seconds to allow time for the payload and PCV to maneuver to achieve a secure docking. When the PCV pays out tether to extend the rendezvous window, however, the centrifugal acceleration of the PCV relative to the tether's center of mass causes it to build up a velocity along the direction of the tether. In the 1 km/s, 100 km long tether system studied here, the tip accelerationlevel is approximately 1 gee. Thus, after a 5-second deployment maneuver, the PCV will be deploying tether at a rate of almost 50 m/s. When the PCV captures the payload, it must stop the tether deployment. This will cause an additional tension spike on the tether.

Figure 10 shows plots of the load level of the tether tip in simulations where the deployment is brought to a halt in varying lengths of time, from an immediate halt (0.1 s braking) to a gradual braking

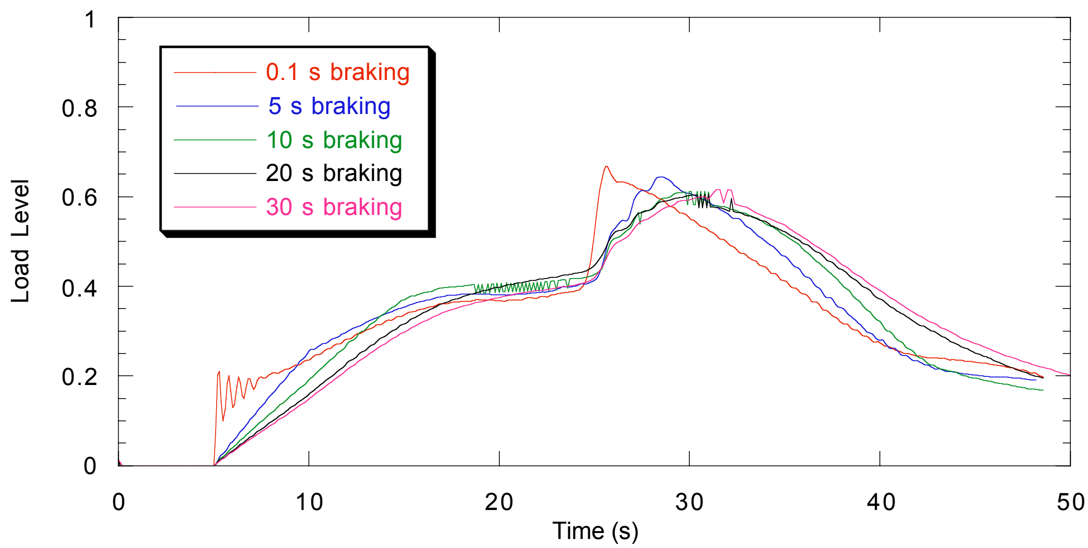


Figure 10. Load level on the segment of tether nearest the payload after a payload capture. In these simulations, the PCV pays out tether for 5 seconds to facilitate rendezvous, and then slows the tether deployment using braking.

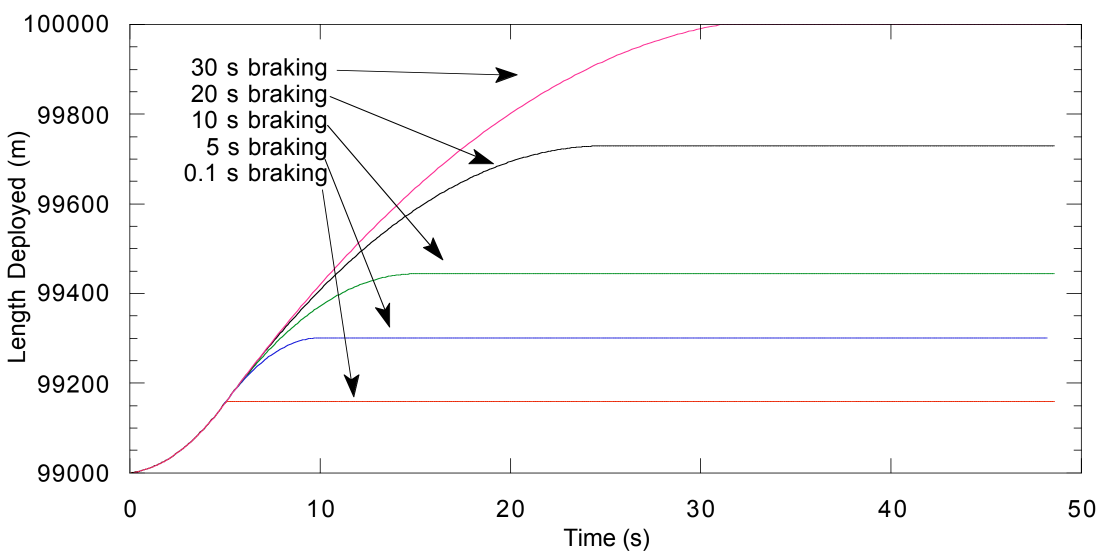


Figure 11. Total length of tether deployed during a payload capture maneuver. In these simulations, the PCV pays out tether for 5 seconds to facilitate rendezvous, and then slows the tether deployment using braking.

over half a minute. Figure 14 shows the length of tether deployed in these various cases. The load level trace for the 0.1 s braking case shows that halting the deployment instantaneously (which would likely be unrealistic in practice) causes a sharp tension spike to 0.2 of the tether's load capacity. The tension then builds as the tether accelerates the payload along. At T=25 seconds, the tension spike reflected from the facility end of the tether returns to the tether tip, causing the tension to exceed 0.6 of the tether's capacity. Halting the deployment more gradually eliminates this initial tension spike, allowing the tether tension to build more gradually. However, the tether tension still peaks at almost 0.6 of the tether's capacity.

Figure 12 and Figure 13 show the tension level at the tip, middle, and facility end of the tether for cases where the deployment is halted immediately and braked over 10 seconds, respectively. In the 10 second braking case, rebound of the tether causes the tether to go slack at about T=60 s. This is undesirable because it could permit fouling of the tether on the PCV or Payload.

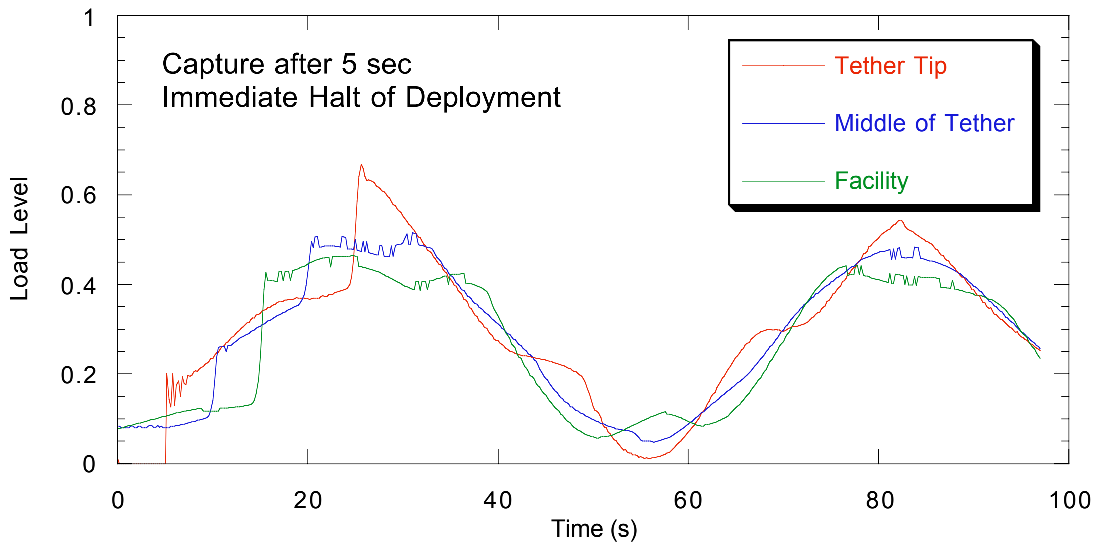


Figure 12. Load level of the tether when payload is captured after a 5-second tether deployment maneuver, and tether deployment is halted immediately after payload is captured.

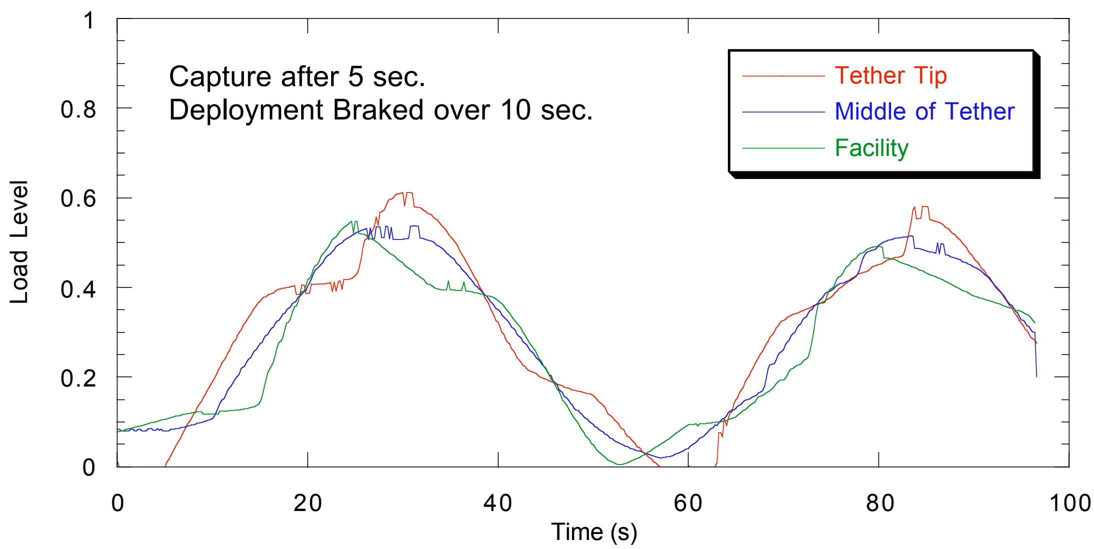


Figure 13. Load level of the tether when payload is captured after a 5-second tether deployment maneuver, and tether deployment is slowed to a halt over 10 seconds.

Using Deployment At Controlled Tension To Dampen Reflected Tension Wave

Although at first consideration it might seem preferable to deploy all of the stored tether during the initial braking, inspection of Figure 10 and Figure 11 reveals that there is little difference in the maximum tension if the tether deployment is braked over 5 seconds (using 150 m of tether) or if the tether deployment is braked over 30 seconds (using 850 m of tether). It may, therefore, be most advantageous to brake the initial tether deployment relatively quickly, and then use the tether remaining on the deployer to reduce the tension excursion experienced when the reflected tension wave returns to the tether tip. The idea here is to allow the tether to deploy under lower tension when the reflected wave reaches the tether tip. Figure 14 shows the total length of tether deployed for such a maneuver. From T=0 to T=5, the tether is deployed at minimum tension to extend the rendezvous window. The deployment is then brought to a halt in 10 seconds. Ten seconds later, when the first tension wave returns tot he tether tip, the PCV again pays out tether, this time at a controlled tension. Figure 15 shows the tip load level for simulations with and without this additional deployment maneuver. Comparison of the traces indicates that the additional deployment succeeds in reducing the peak tension from 0.6 of the tether's capacity to just over 0.5 of capacity; if the PCV had more tether on the spool, it could reduce the maximum tension even further. Note also that this additional deployment actually prevents the tether from going slack at T=60. Figure 16 shows the tether load level at the tip, middle, and facility end of the tether during this rendezvous maneuver.

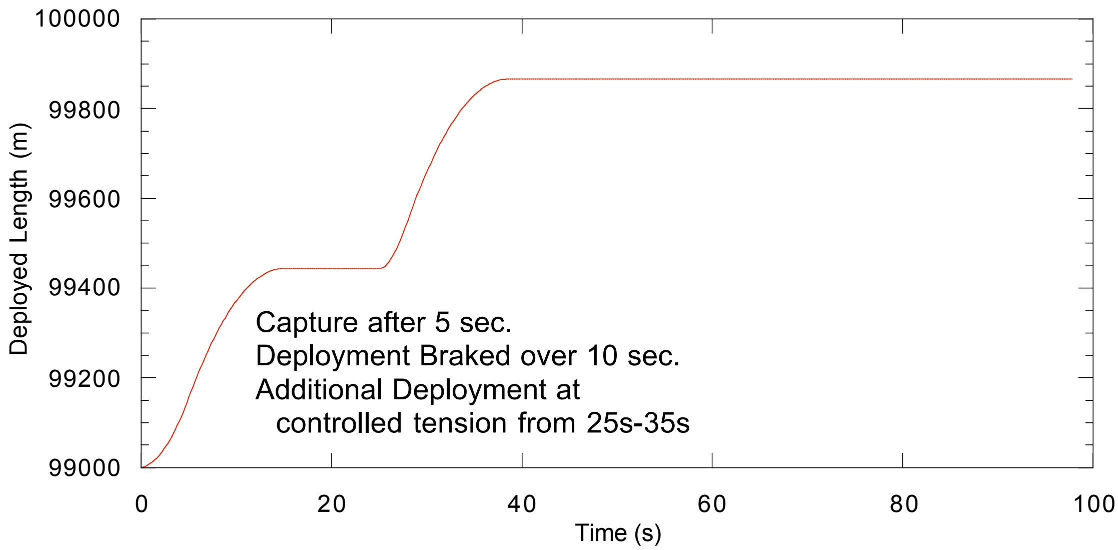


Figure 14. Total length of tether deployed during rendezvous maneuver in which the PCV deploys the tether for 5 seconds to facilitate rendezvous, then captures the payload and slows the deployment over a period of 10 seconds, and then later allows the tether to deploy under controlled tension to dampen the tether tension wave.

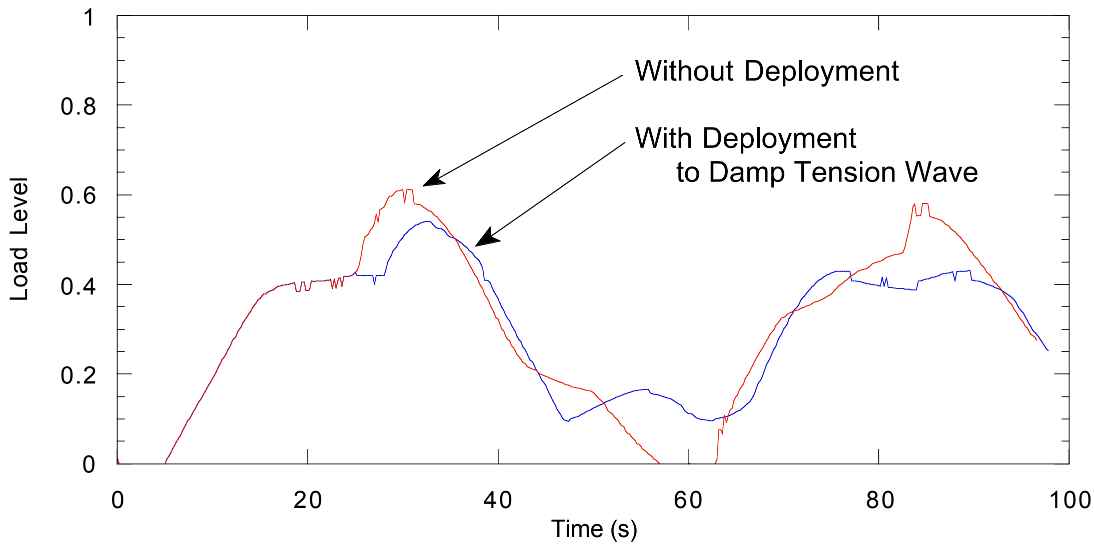


Figure 15. Comparison of load level at tether tip in rendezvous maneuvers with and without additional deployment to damp tension waves.

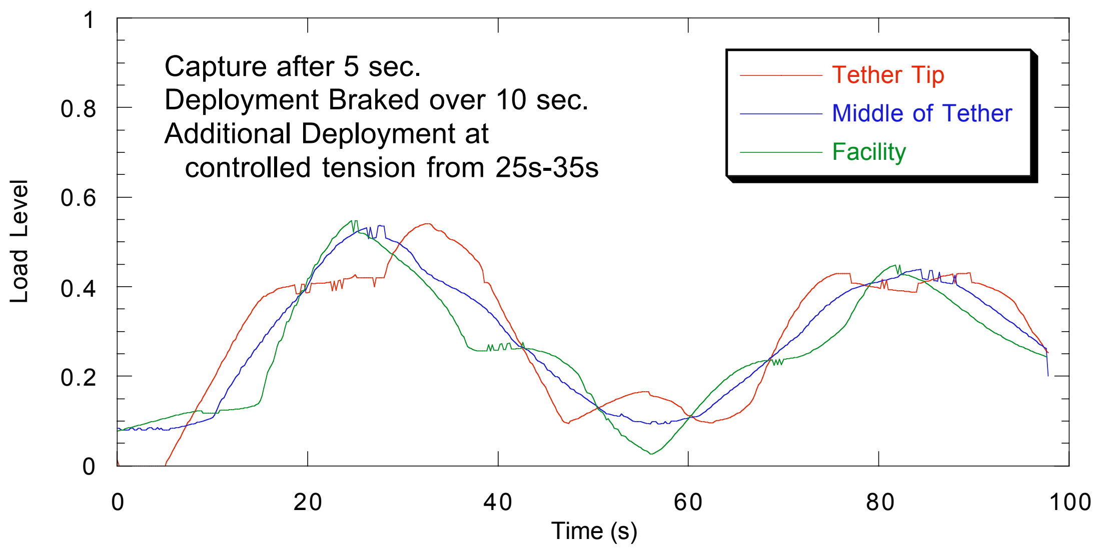


Figure 16. Tether load level during rendezvous maneuver in which the PCV deploys the tether for 5 seconds to facilitate rendezvous, then captures the payload and slows the deployment over a period of 10 seconds, and then later allows the tether to deploy under controlled tension to dampen the tether tension wave. (Compare with Figure 13)

Tension Behavior Due to Payload Release

When a rotating tether releases a payload, the sudden change in loading also causes tension waves to travel up and down the tether. These longitudinal oscillations, however, reduce the tether loading, so they do not cause the tension to exceed the desired limits. Figure 17 shows the tether loading at tip, middle, and facility end of the tether after a payload release. The tip loading immediately drops to low levels, but the tension levels at the facility end and the middle oscillate. These oscillations eventually damp out due to frictional forces in the tether structure. The oscillations could also be dampened more quickly through controlled reeling and deployment of the tether.

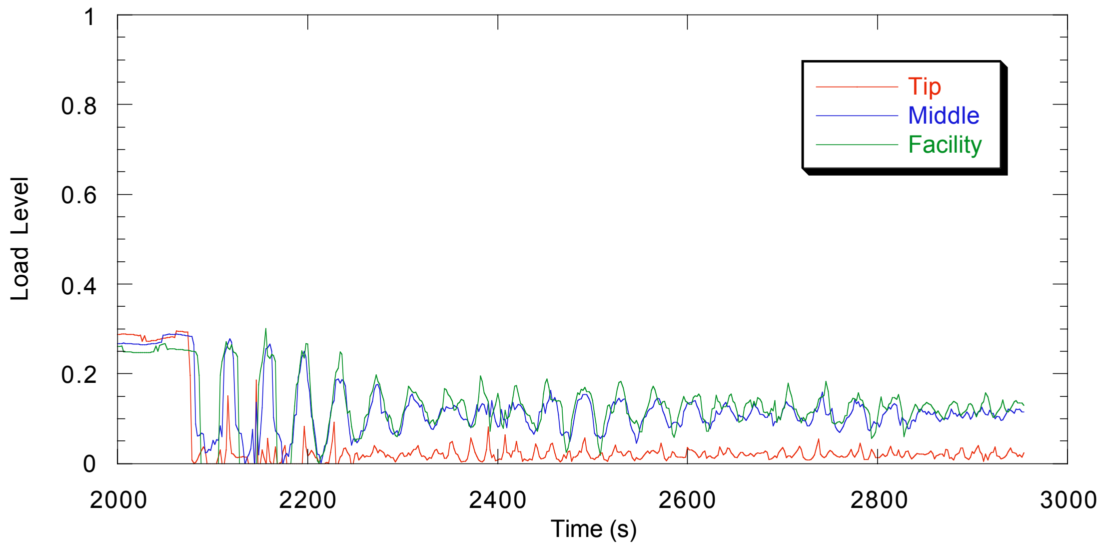


Figure 17. Tether loading at tip, middle, and facility after the payload is released at T=2075.

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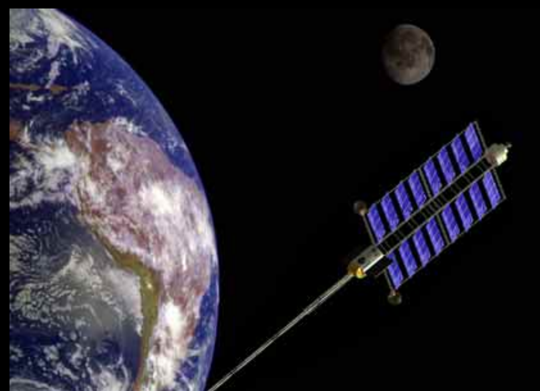
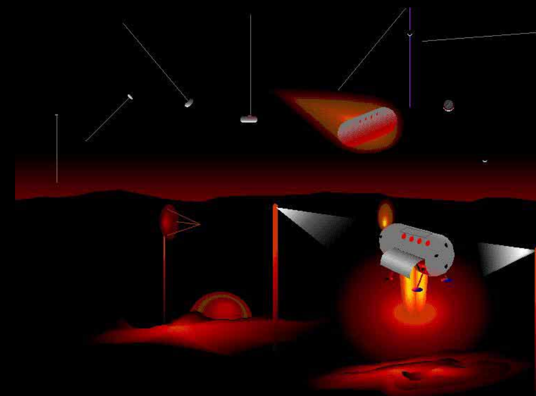
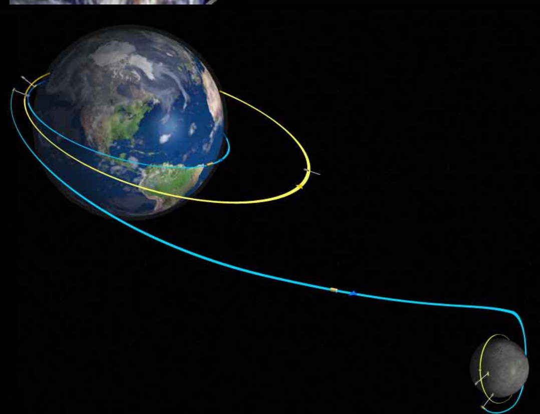
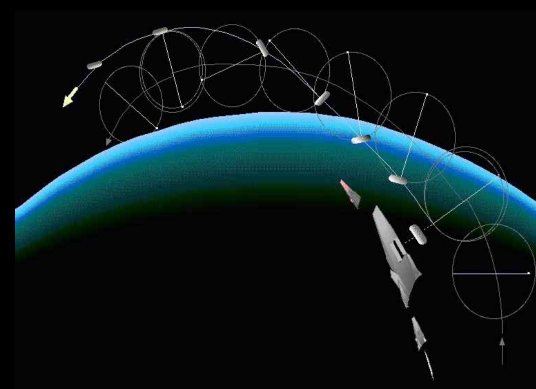
[i]. Hoyt, R.P., "Cislunar System Dynamics Verification Through Simulation", Appendix C in *Cislunar Tether Transport System*, Tethers Unlimited, Inc. Final Report on NIAC Contract 07600-011.

[ii]. Lorenzini, E.C., Mowery, D.K., Rupp, C.C., "SEDS-II Deployment Control Law and Mission Design", *Proceedings of the Fourth International Conference on Tethers In Space*, Vol II, p. 669.

Tether Boost Facilities for In-Space Transportation

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Ongoing Tether Work Under NIAC Funding

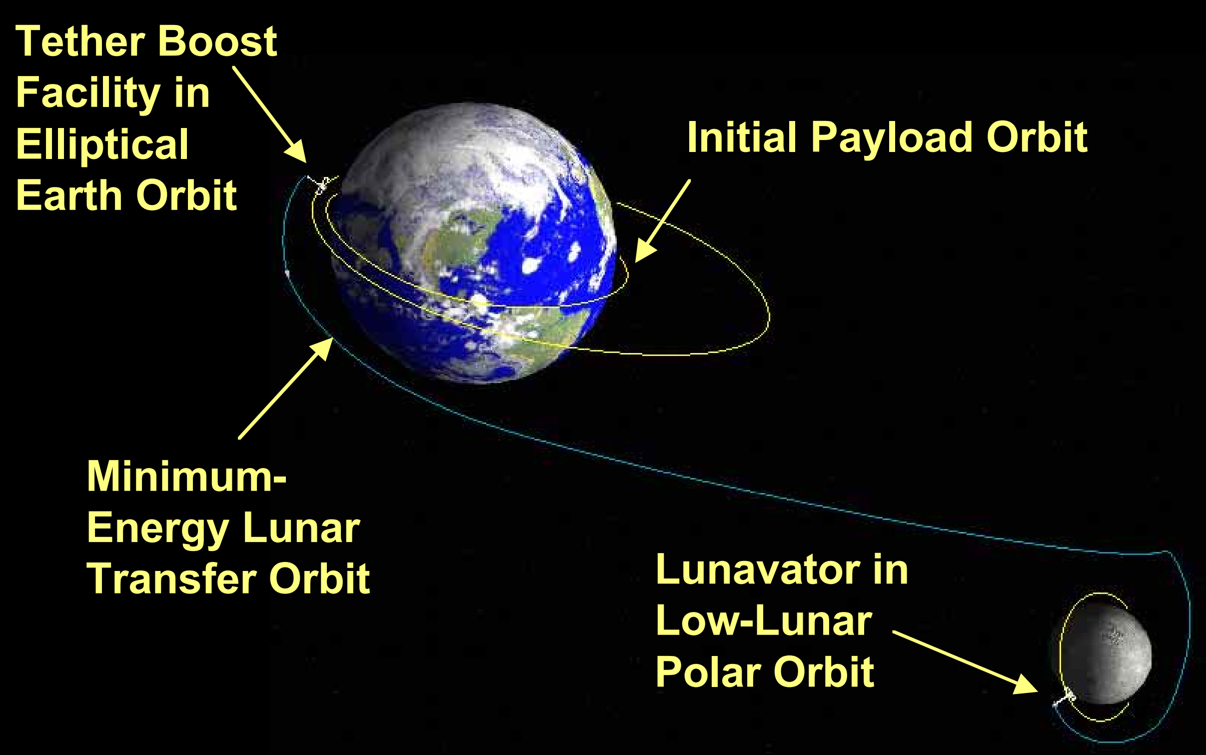
- Objectives:
  - Perform Technical & Economic Analysis of Tether Transport Systems
  - Identify Technology Needs
  - Develop Conceptual Design Solutions
  - Prepare for Flight Experiments to Demonstrate Tether Transport Technology
- Moon & Mars Orbiting Spinning Tether Transport (MMOSTT)
  - TUI Prime, Boeing/RSS sub
  - Develop Design for a 2.4 km/s ΔV LEO⇨GTO Tether Boost Facility
  - Develop & Simulate Methods for Tether-Payload Rendezvous
  - Identify Near-Term Commercial and Scientific Applications
  - Investigate Cislunar, Mars, & other Tether Transport Architectures
- Hypersonic Airplane Space Tether Orbital Launch (HASTOL)
  - Boeing/RSS prime, TUI sub
  - Design Launch Architecture Combining a ~7 km/s ΔV Tether Boost Facility with a Mach 10-12 Hypersonic Airplane

Summary of Advantages

- **Tether Boost Facilities Can Provide a Fully-Reusable In-Space Propulsion Architecture**
  - LEO ⇔ MEO/GTO
  - LEO ⇔ Lunar Surface
  - LEO ⇔ Mars
  - ETO Launch, in combination with Hypersonic Airplane/RLV
- **Momentum Exchange + Electrodynamic Tether Can Enable Propellantless Propulsion <u>Beyond</u> LEO**
- **Rapid Transfer Times**
  - 5 days to Moon
  - 90 days to Mars
- **Reusable Infrastructure + Low Consumables**
  ↳ <u>Lower Cost</u>

Cislunar Tether Transport System

- Developed Orbital Architecture for Round Trip LEO⇔Lunar Surface Transport
- Whole System Mass < 27x Payload Mass
  - LEO Tether Boost Facility Mass = 10x Payload Mass, Lunar Tether Facility = 17x Payload
- 13 Payloads/Year
- Incremental Commercial Development Path

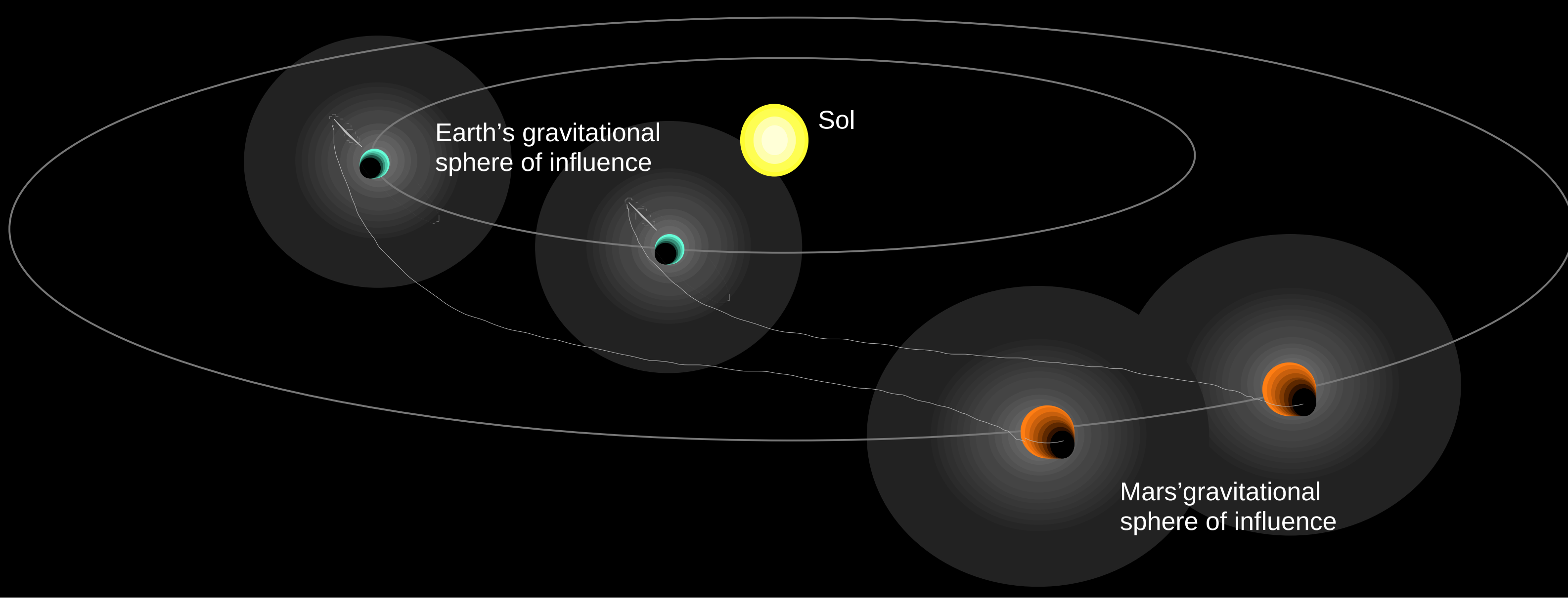


Rapid Earth-Mars Transport

- Reusable Architecture for Round Trip Earth ⇔ Mars Transport
- Rapid Transfer Times (90-130 days)
- Extended Launch Windows

Approach Year	146 Day Transfer 2.0 km/s tether tip speed			116 Day Transfer 2.5 km/s tether tip speed		
	Open	Close	Window (days)	Open	Close	Window (days)
2001	03/18/01	05/07/01	50	02/25/01	05/18/01	82
2003	04/27/03	07/22/03	86	05/04/03	08/03/03	91
2005	07/27/05	09/08/05	43	07/31/05	09/20/05	51
2007	6 Oct	comes closest	-	10/06/07	10/24/07	18
2009	10 Nov	comes closest	-	20 Nov	comes closest	-
2011	12/06/11	12/21/11	15	12/18/11	01/02/12	15
2013	12/30/13	02/08/14	40	01/11/13	02/16/13	36
2016	02/02/16	04/06/16	64	02/14/16	04/18/16	64
2018	03/25/18	06/24/18	91	04/03/18	07/06/18	94

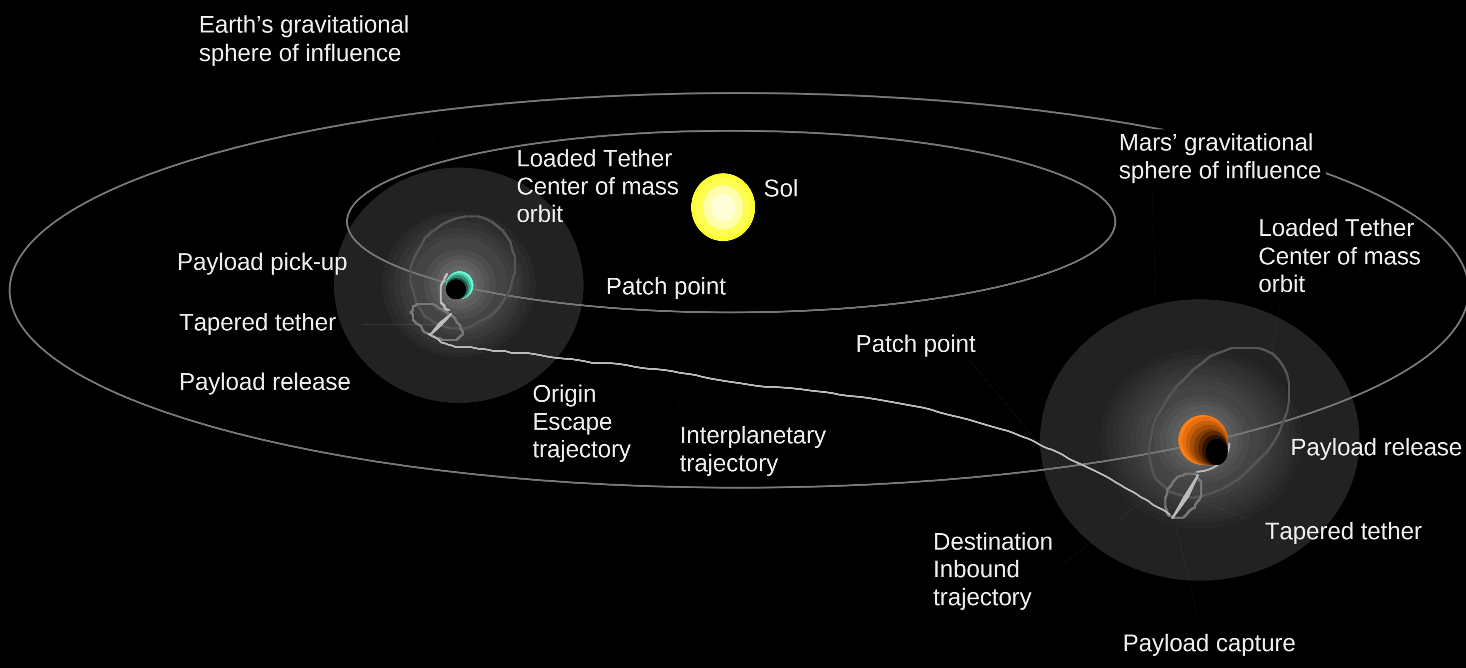
- Currently Evaluating Architectures
  - All Tether
  - Tether/Chemical



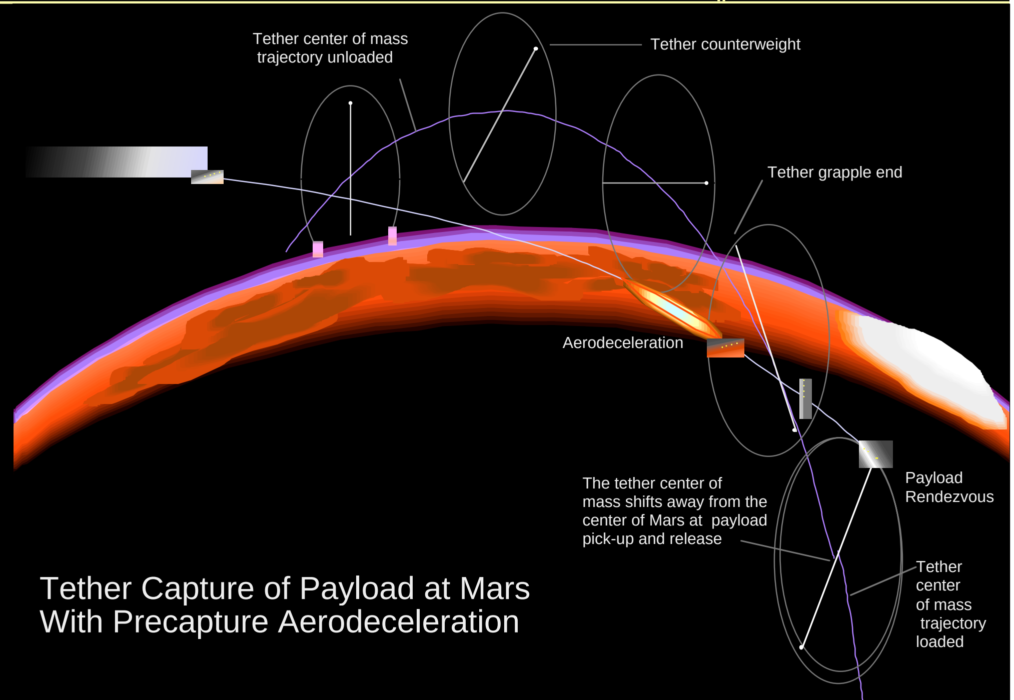
Rapid Earth-Mars Transport

- **Reusable Architecture for Round Trip Earth ⇔ Mars Transport**
- **Rapid Transfer Times (90-130 days)**

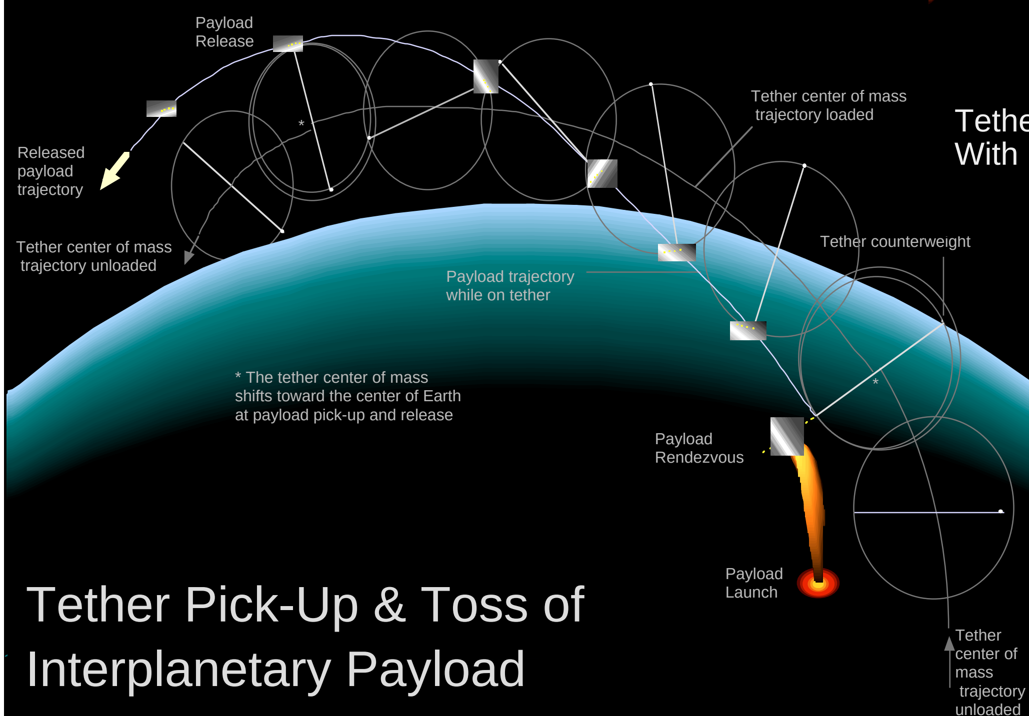
INTERPLANETARY TRANSPORT USING
ROTATING TETHERS



Earth⇨Mars Transport



Tether Capture of Payload at Mars With Precapture Aerodeceleration



Tether Pick-Up & Toss of Interplanetary Payload

Earth⇨Mars Launch Windows

- MMOSTT Architecture Provides Large Launch Windows
- Varies With Launch Opportunity
  - Mars Orbit Has High Eccentricity
- Launch Window
  - Earth-Mars Transit Time = 135 days
  - Tether Boost Capability ΔV=4 km/s
    - 2008 - 11 days
    - 2010 - 37 days
    - 2012 - 60 days
    - 2014 - 72 days
    - 2017 - 57 days
    - 2019 - 12 days
- Launch Window Larger With Faster (ΔV=5 km/s) Tether
- Launch Window Larger With Longer Transit Time

Earth⇨Mars Architectures

- **Currently Evaluating System Architectures**
  - Transfer opportunity frequency
  - System flexibility
  - System complexity
  - Modularity & Compatibility with LEO⇨GEO & Cislunar Architectures
- **Combined Tether/Chemical System Probably Most Flexible**
  - Use LEO⇨GEO Tether Boost Facility to raise Mars payload into high elliptical, equatorial orbit
  - Use Chemical/NTR/Other high-thrust propulsion to do Trans-Mars-Injection

Incremental Development Path

1. TORQUE™ Experiment
   - Demonstrate Momentum-Exchange & Electrodynamic Reboost
   - Experiment Becomes Operational Facility for Microsat Deployment
2. LEO ⇔ GTO Tether Boost Facility
   - Initial Capability: 2,500 kg to GTO once per month
   - Modular Design: add additional components ⇨ 5,000 kg, 7,500 kg…
3. LEO ⇔ Lunar Tether Transport System
   - LEO ⇔ GTO Facility Can also Send Payloads to Moon
   - Add Lunavator to Enable Round-Trip Transport to Lunar Surface
4. LEO ⇔ Mars Tether Transport
   - Tether Boost Facility Places Mars Payloads in Highly Elliptical Orbit
   - Use Rocket for Trans-Mars Injection & Mars Capture
   - Deploy Tether at Mars to Enable Round-Trip Transport Without Rockets

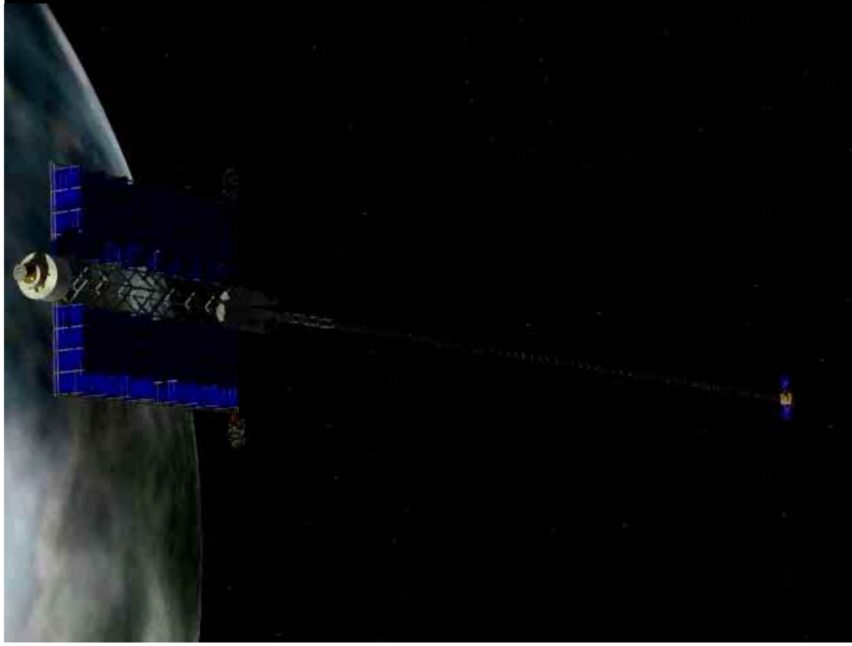
- Each Stage Generates Revenue to Fund Development of Later Stages

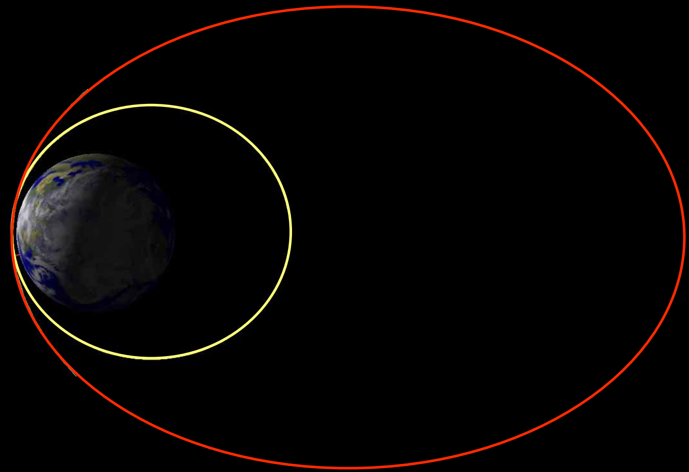
LEO⇨GTO Tether Boost Facility

- Designed to Boost 2,500 kg payloads from LEO to GTO - Total ΔV = 2.4 km/s
- Operational Capability Can be Placed in LEO with One Delta-IV-H Launch

Tether Mass:	8,275 kg
Grapple Assembly:	650 kg
Control Station Mass:	11,500 kg
Total Launch Mass:	20,500 kg
+ Delta-IV Upper Stage for Ballast:	3,490 kg

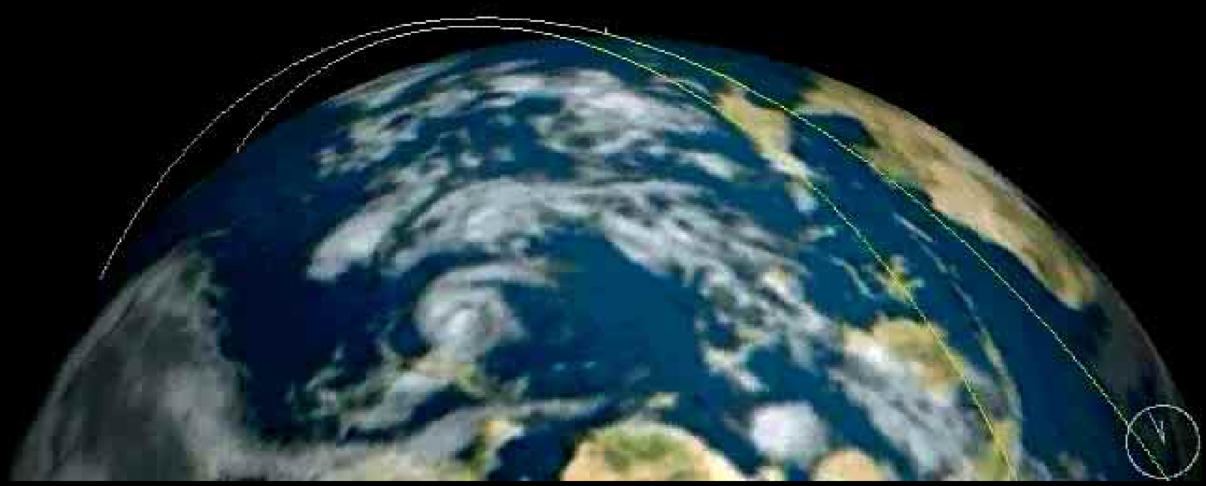
- Facility Can also Toss 500 kg payloads to Lunar Transfer Orbit
- Uses Electrodynamic Reboost to Enable Facility to Boost 1 Payload Per Month





LEO⇨GTO Boost Facility

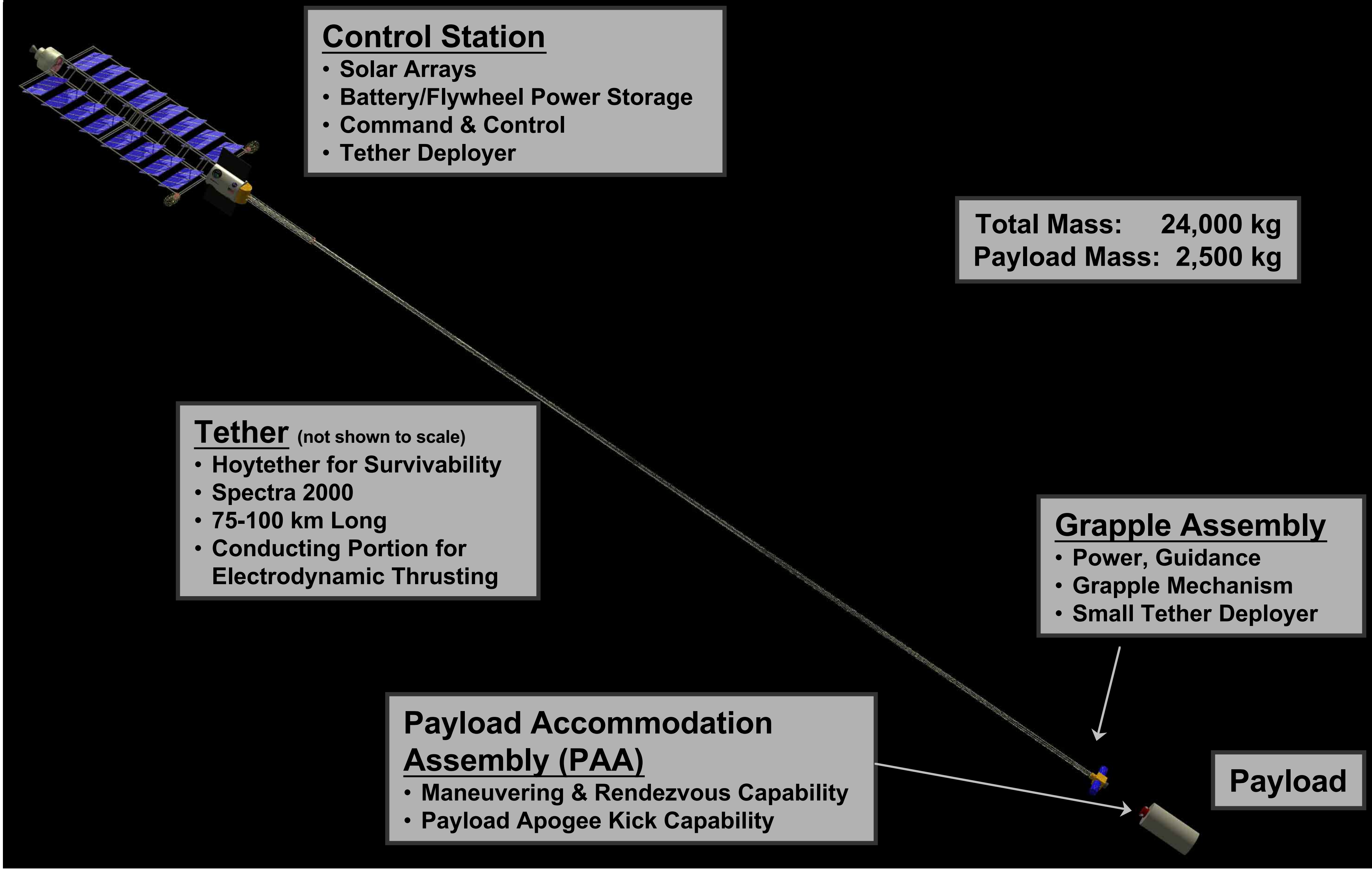
- **TetherSim™ Numerical Simulation (10x real speed)**
  - **Tether Dynamics, Orbital Mechanics**



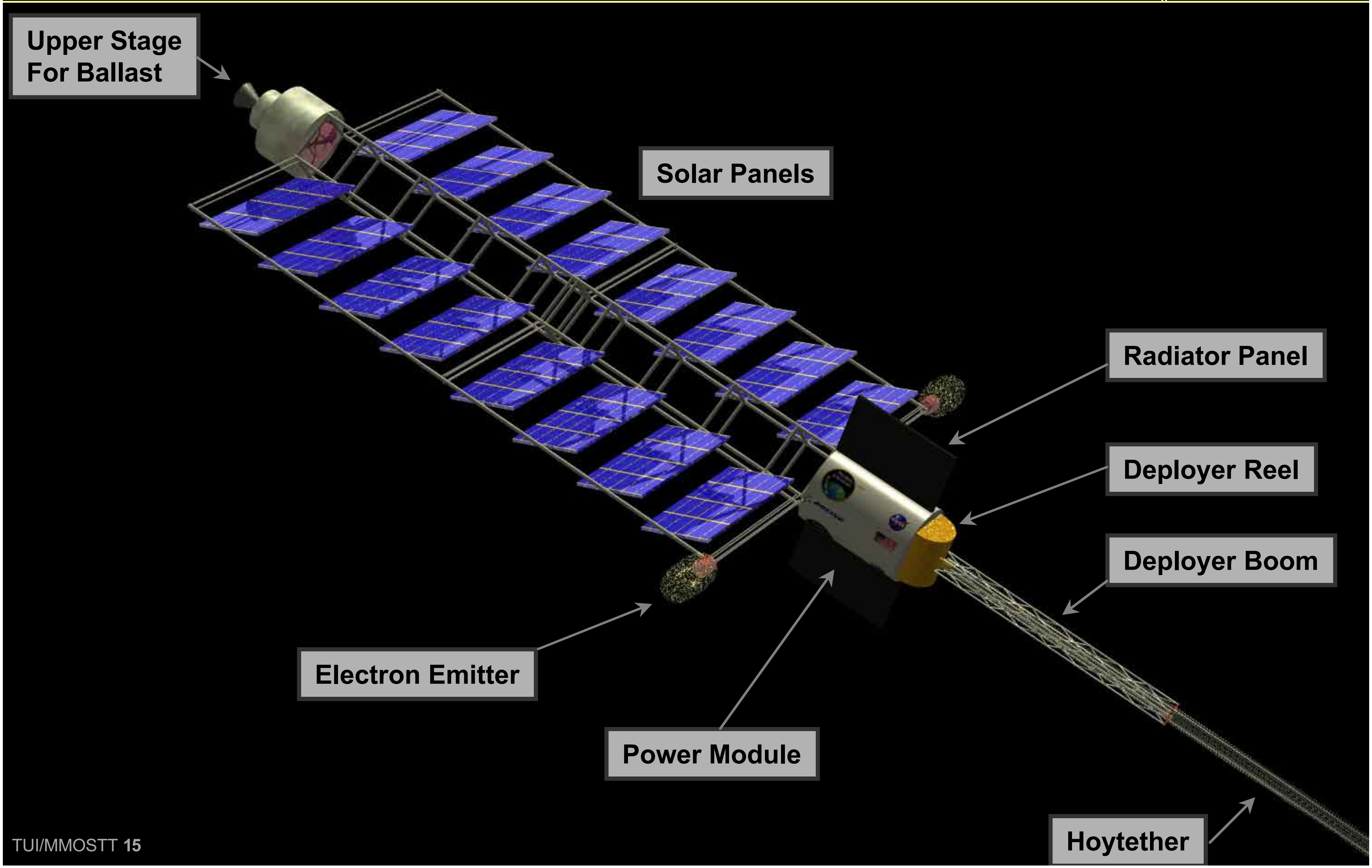
LEO⇨GTO Boost Facility System Definition Task

- **TUI & Boeing have developed System Requirements Document for Tether Boost Facility**
- **System Concept Definition**
  - **Identify key technologies**
  - **Mass and power budgets**
- **Technology Readiness Level Evaluation**

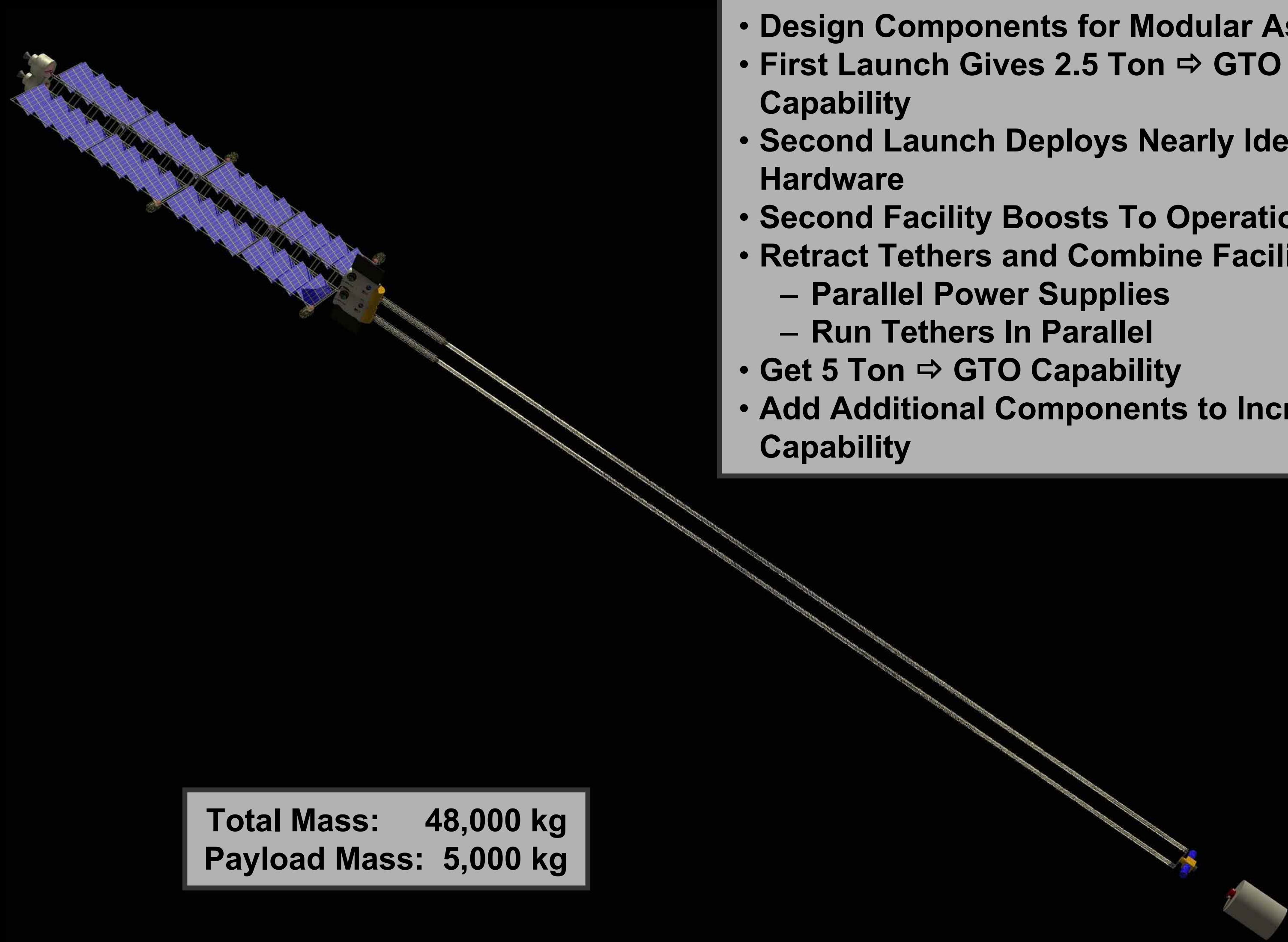
Tether Boost Facility



Control Station



Modular Design



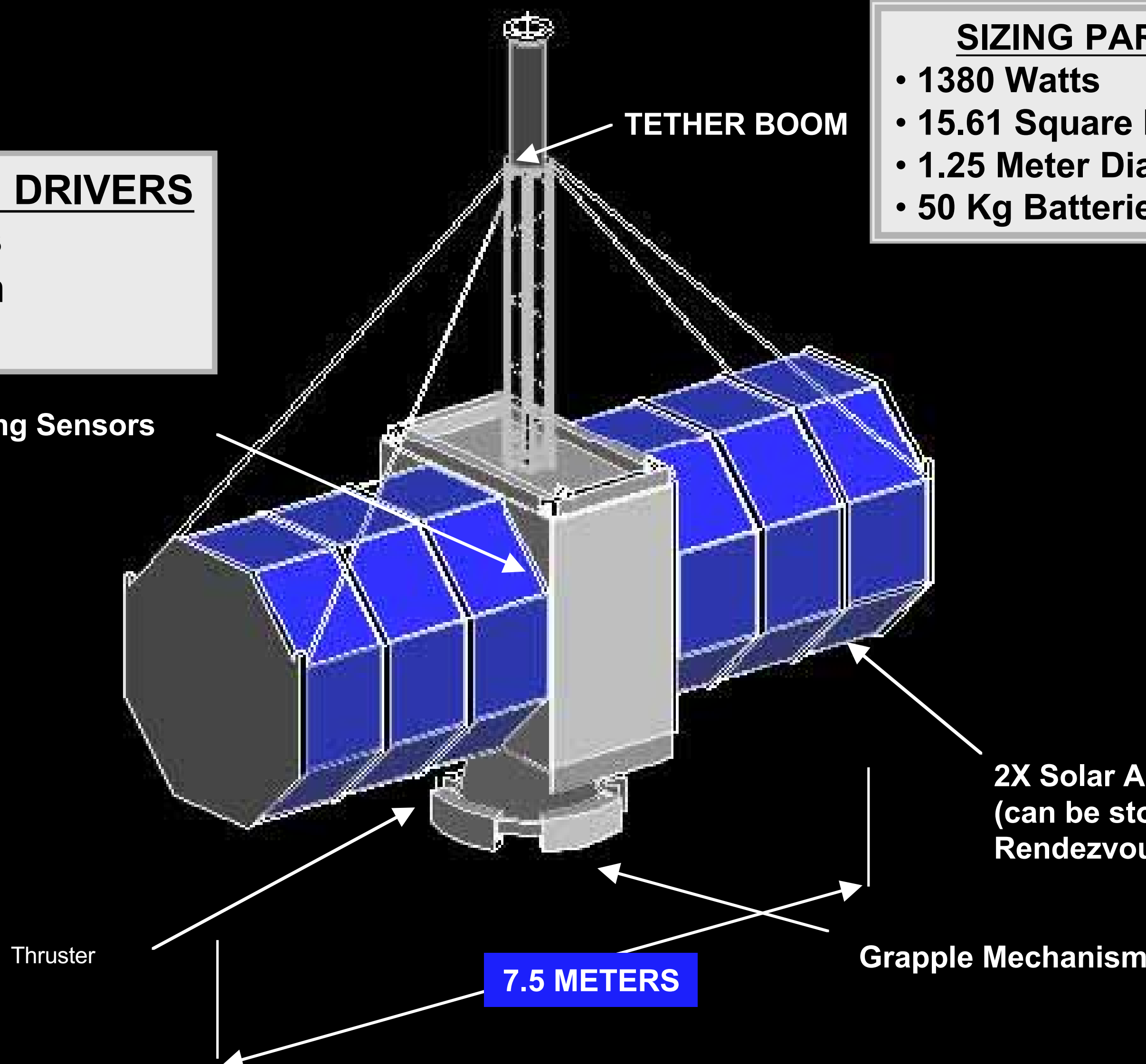
- Design Components for Modular Assembly
- First Launch Gives 2.5 Ton ⇨ GTO Operational Capability
- Second Launch Deploys Nearly Identical Facility Hardware
- Second Facility Boosts To Operational Orbit
- Retract Tethers and Combine Facilities On-Orbit
  - Parallel Power Supplies
  - Run Tethers In Parallel
- Get 5 Ton ⇨ GTO Capability
- Add Additional Components to Increase Payload Capability

Total Mass: 48,000 kg
Payload Mass: 5,000 kg

Grapple Assembly

CONFIGURATION DRIVERS
- Capture Options
- System Rotation
- Loads

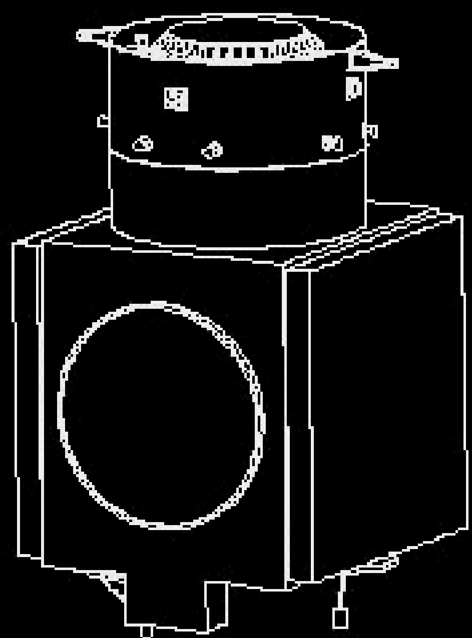
SIZING PARAMETERS
- 1380 Watts
- 15.61 Square Meter Arrays
- 1.25 Meter Dia Capture Ring
- 50 Kg Batteries (Ni-H_2)



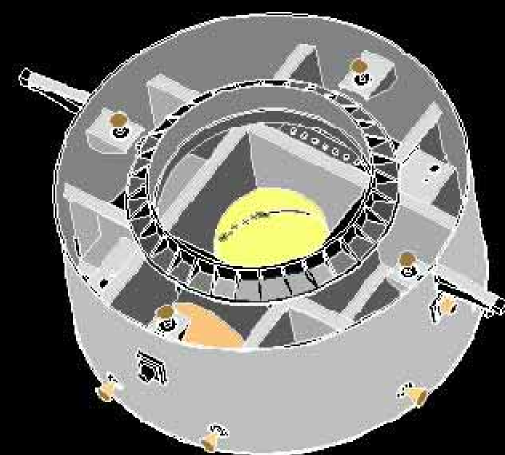
Payload Accommodation Assembly

Configuration Drivers

- **Mimic Conventional Upper Stage Interfaces To Payload And Booster**
- **Track Grapple And Make Rendezvous Corrections**
- **Provide Circularization ΔV**



Payload with PAA



- **1.9 m Dia x 1 M Long**
- **12 Thrusters, 0.7 M Dia Fuel Tank**
- **2 Primary Batteries**
- **Communications & Guidance Systems**
- **3 Reaction Wheels**

Baseline Orbital Design

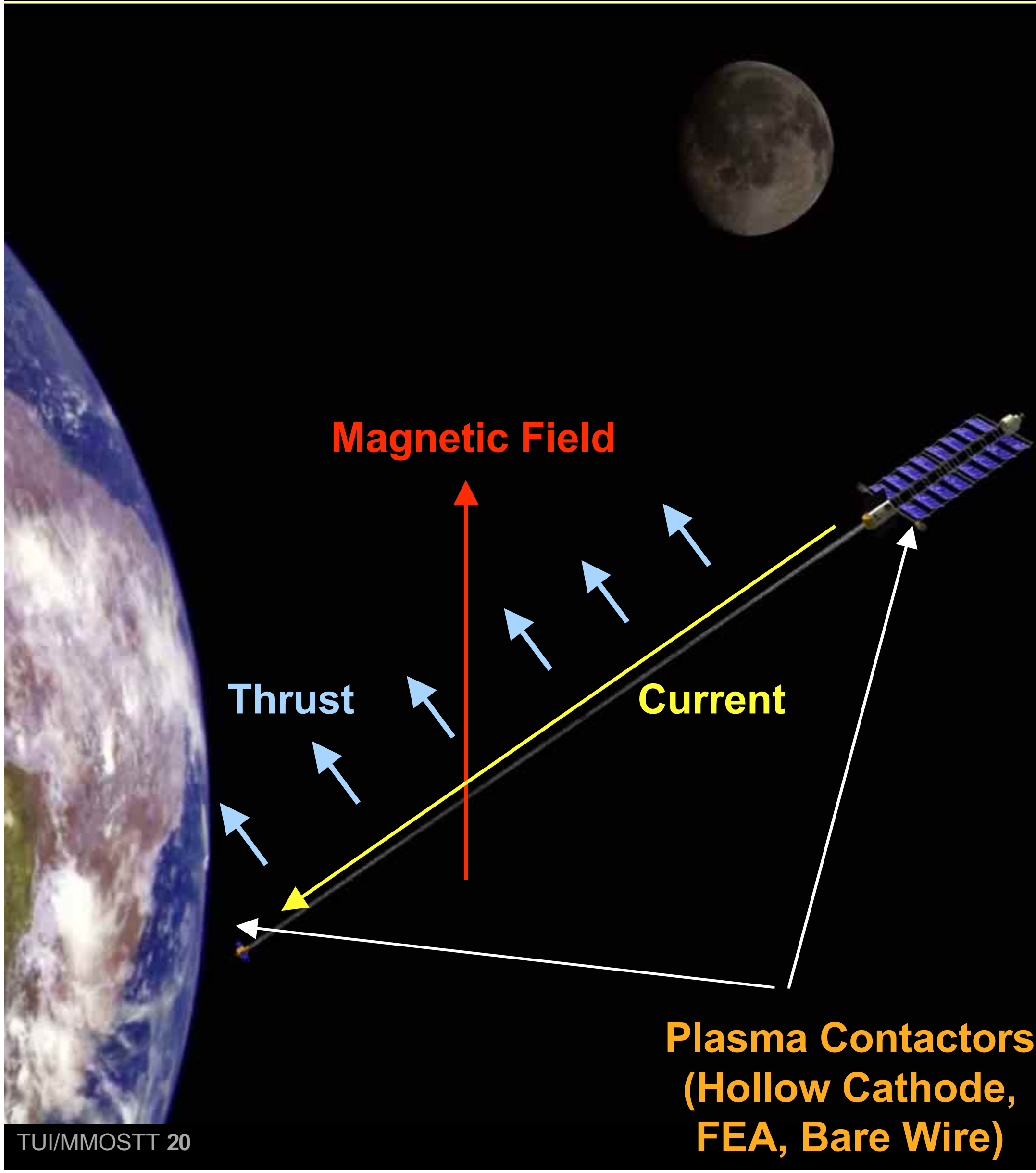
- **Payload and Tether in Synchronous Orbits**
  - **Periodic Rendezvous Opportunities**
- **Design Accounts for:**
  - **Precessions Due to Earth Oblateness**
  - **Tether Taper**
  - **Center of Mass Shifts**

System Masses		
Tether mass	8,274	kg
CS Active Mass	11,507	kg
CS Ballast Mass	3490	kg
Grapple mass	650	kg
Total Facility Mass	**23,922**	kg
Total Launch Mass	**20,432**	kg
Payload Mass	**2,500**	kg

Tether Characteristics		
Tether Length	100,000	m
Tether mass ratio	3.31	
Tether tip velocity at catch	1,267	m/s
Tether tip velocity at toss	1,147	m/s
Tether angular rate	0.015515	rad/s
Gravity at Control Station	0.64	g
Gravity at payload	1.81	g
Rendezvous acceleration	2.00	g

		Pre-Catch		Joined System	Post-Toss	
Positions & Velocities		**Payload**	**Tether**	**Post-catch**	**Tether**	**Payload**
resonance ratio		41	20		1	4.1
perigee altitude	km	325	407	399	391	473
apogee altitude	km	325	8445	7199	6104	35783
perigee radius	km	6703	6785	6777	6769	6851
apogee radius	km	6703	14823	13577	12482	42161
perigee velocity	m/s	7711	8978	8858	8738	10005
apogee velocity	m/s	7711	4109	4422	4739	1626
CM dist. From Station	m		18362	26087	18362	
CM dist. To Grapple	m		81638	73913	81638	
ΔV to Reboost	m/s				240	
ΔV to Correct Apogee	m/s					0
ΔV to Correct Precess.	m/s					0
ΔV To Circularize	m/s					1449
Basic Orbital Parameters						
semi-major axis	km	6703	10804	10177	9626	24506
eccentricity		0.0	0.372	0.334	0.297	0.720
inclination	rad	0	0	0	0	0
semi-latus rectum	km	6703	9309	9041	8778	11787
sp. mech. energy	m2/s2	-2.97E+07	-1.84E+07	-1.96E+07	-2.07E+07	-8.13E+06
vis-viva energy	m2/s2	-5.95E+07	-3.69E+07	-3.92E+07	-4.14E+07	-1.63E+07
period	sec	5462	11176	10218	9399	38178
period	min	91.0	186.3	170.3	156.6	636.3
station rotation period	sec		405.0	405.0	405.0	
rotation ratio			27.6	25.2	23.2	

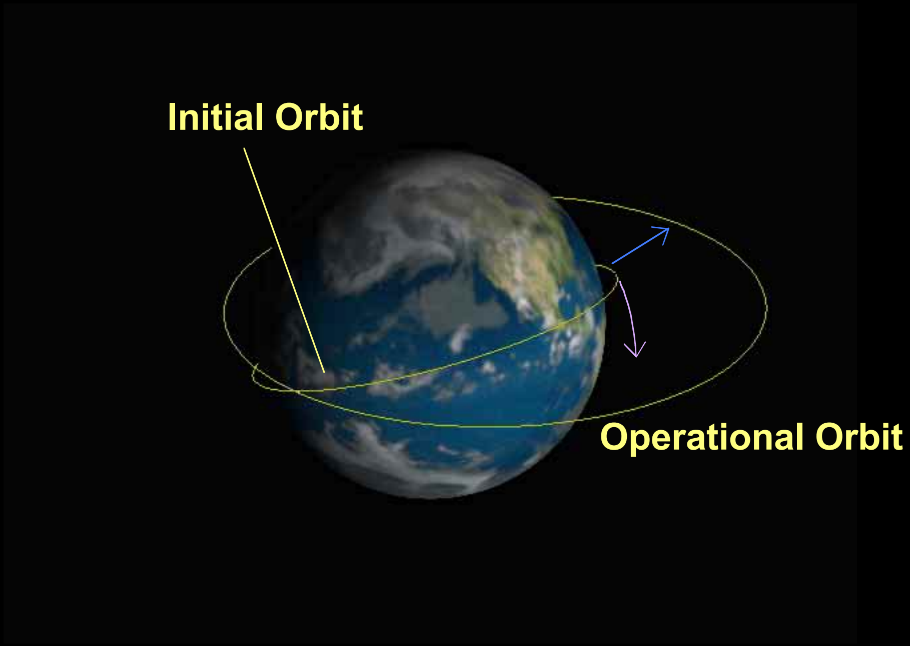
Electrodynamic Thrusting



- Drive current along tether
- Plasma contactors exchange current w/ ionosphere
- Plasma waves close current "loop"
- Current "pushes" against geomagnetic field via JxB Force
- Current: 10-12 A
- Voltage: ~20 kV

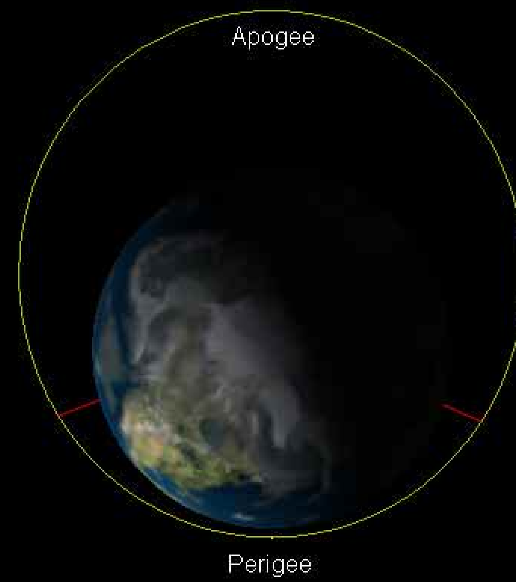
Tether Facility Deployment

- Launch Tether Facility on Delta-IV-H (20,500 kg to LEO)
  - Retain 3490 kg Upper Stage for Ballast
- 250 km, 20° Initial Orbit
- Assemble Facility On-Orbit
- Deploy Tether Upwards
- Use Electrodynamic Thrust to:
  - Torque Orbit to Equatorial Plane
  - Boost Apogee
  - Spin Up Tether
- ~8 Months to Operational Orbit

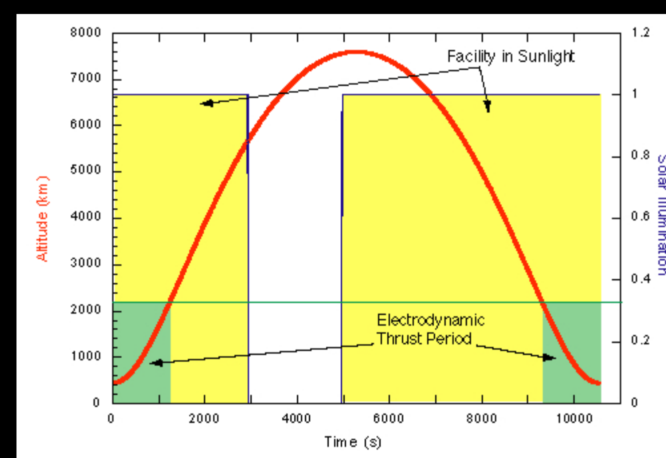


Tether Facility Reboost

- **Use Electrodynamic Propulsion Near Perigee to Reboost Orbit**
  - **Collect electrons from ionosphere at one end of tether & emit electrons at other end of tether**
  - **Use power from batteries to push curent along tether**
  - **Current interacts with geomagnetic field to give JxB force**
  - **Vary current to generate net thrust**



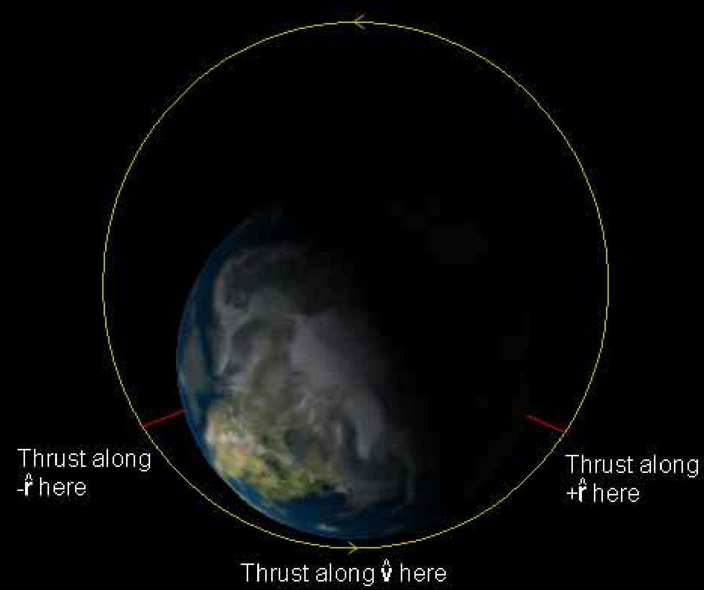
- **To achieve Reboost in 30 days:**
  - **Solar Panel Power: 100 kW**
  - **Power To ED Tether: 300 kW**
  - **Currents: ~15 A**

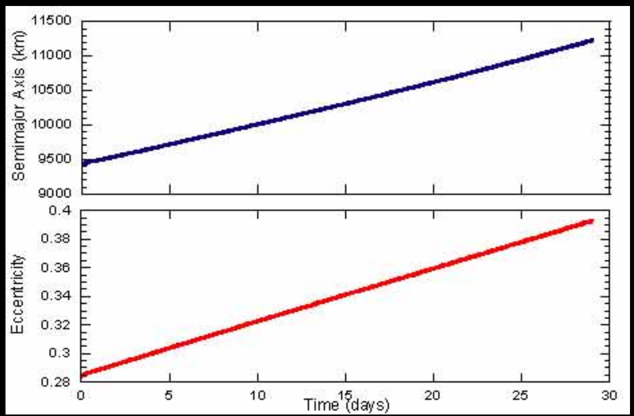


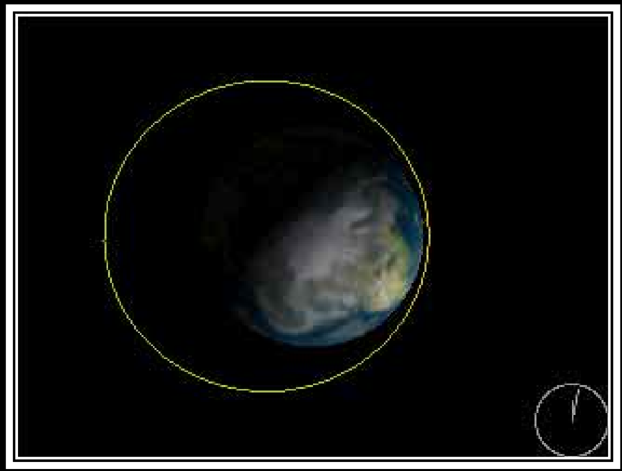
- **Issues:**
  - **High Power, High Voltage (20 kV)**

Reboost Tuning

- Electrodynamic thrusting possible below 2000-2200 km
- Must control thrusting to achieve desired final orbit
  - Otherwise perigee raised too much
- Tether is rotating, so thrust direction varies
- Vary average thrust direction during perigee pass to boost apogee and keep perigee down







Rendezvous

- Rapid Automated Rendezvous & Capture Needed
- Major Technology "Tentpole"
- Must Accomplish:
  - In advance, place payload on trajectory that will osculate with tether tip trajectory
    - Payload and grapple will be in proximity with zero relative velocity for a brief time
  - Achieve rendezvous & docking within very short time frame
  - Minimize dynamic disturbance to tether system

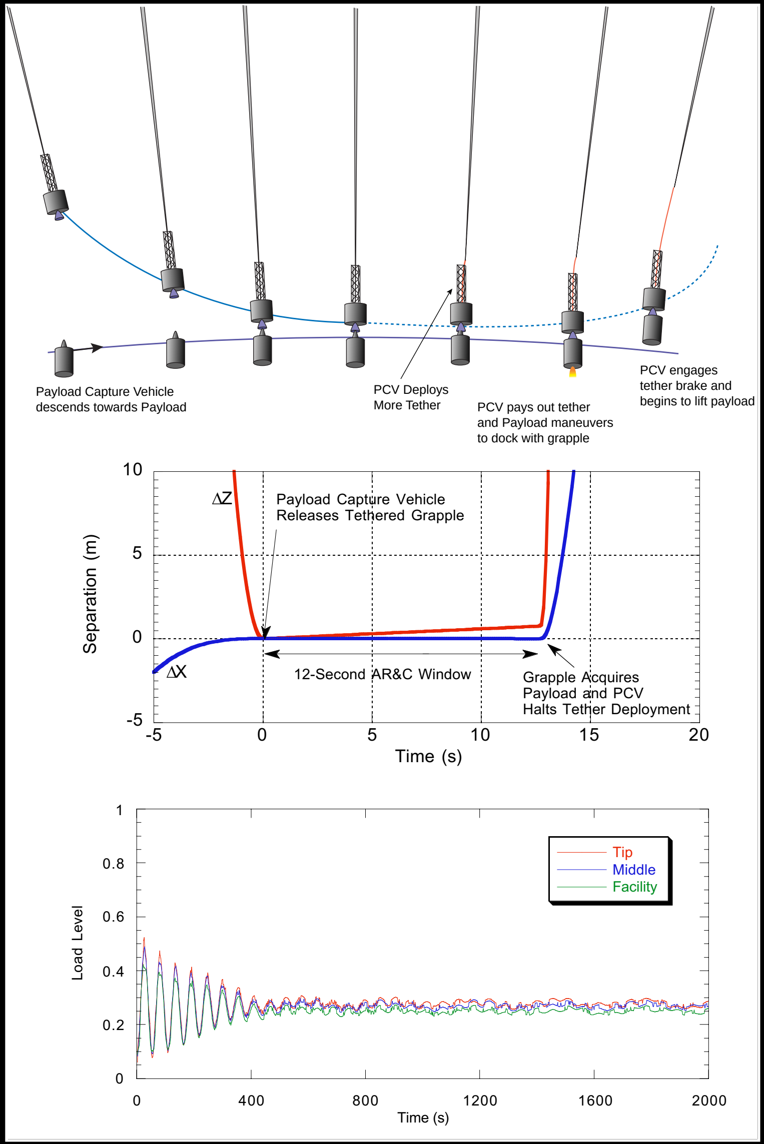
Rendezvous

Rendezvous Method: Preparation

- Propagate tether orbit to obtain future tip position & velocity
- Propagate a “virtual payload” backwards in time
- Real payload performs standard, slow rendezvous with “virtual payload”
- During approach, payload performs corrections to account for propagator errors

Rendezvous: Payload Acquisition

- Rapid Automatic Rendezvous & Capture (AR&C) is a Key Requirement
  - Payload is in free-fall orbit
  - Tether tip under 1-2 gees centrifugal acceleration
  - Relative speed zero only momentarily
  - 1 s @ 1 gee => 5 m & 10 m/s
- TUI Has Developed Methods for Extending Rendezvous Window
  - Grapple Assembly has small tether deployer
  - At conjunction of payload and tether tip, grapple assembly deploys tether at low tension
    - 1 km tether gives 10s @ 2 gees
  - Grapple and payload “float” in free fall together for 5-10 seconds
  - Payload maneuvers to dock with grapple
  - Grapple applies brake to tether gradually to minimize tether tension excursion

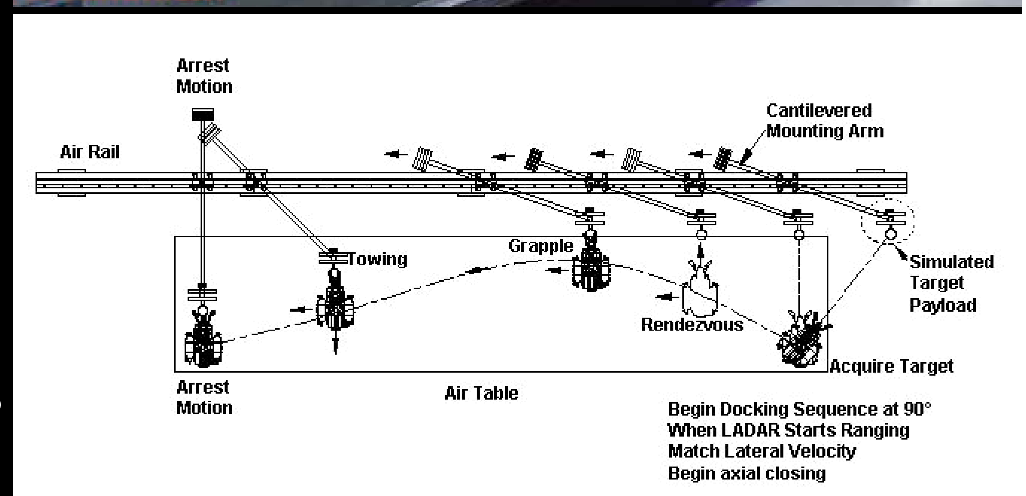


Development Issues

- Automated Rendezvous & Capture
  - Time Window for Capture < 10 Seconds
  - High Accuracy Requirements
- Electrodynamic Tether Operation
  - High Power & Voltage Issues
  - Control of Tether Dynamics
- Traffic Control/Collision Avoidance
- Economic Analysis/Business Plan
  - Technology Risk Reduction Requirements
  - Incremental Commercial Development Path
  - Customer Acceptance

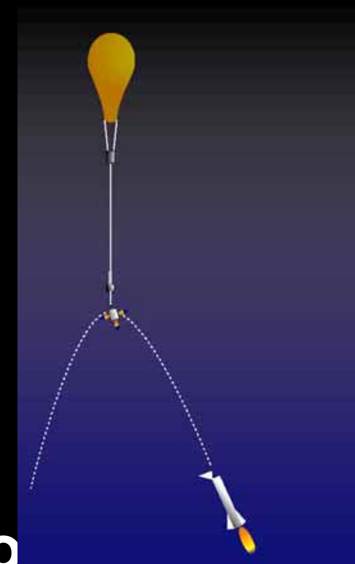
Validate Rapid Tether-Payload AR&C With Demonstration at LLNL Facility

- **TUI & LLNL Propose Rapid Grapple Rendezvous And Secure Pickup (GRASP) Demo**
- **LLNL Has In Operation:**
  - **Air Rail and Air Table**
  - **Cold Gas Jet Stabilized and Propelled Microsat Test Vehicle on Air Ball on Air Puck (5DOF)**
  - **Automatic Grapple Mechanism**
  - **Fully Autonomous Acquisition, Tracking, Rendezvous and Capture Sensors and Software**
- **LLNL Has Demonstrated AR&C of Stationary Target in ~40 s**
- **TUI/LLNL Wish to Demonstrate AR&C of Moving Target in <10 s**

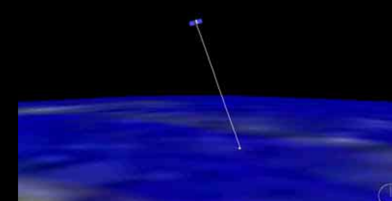


Potential Flight Experiments

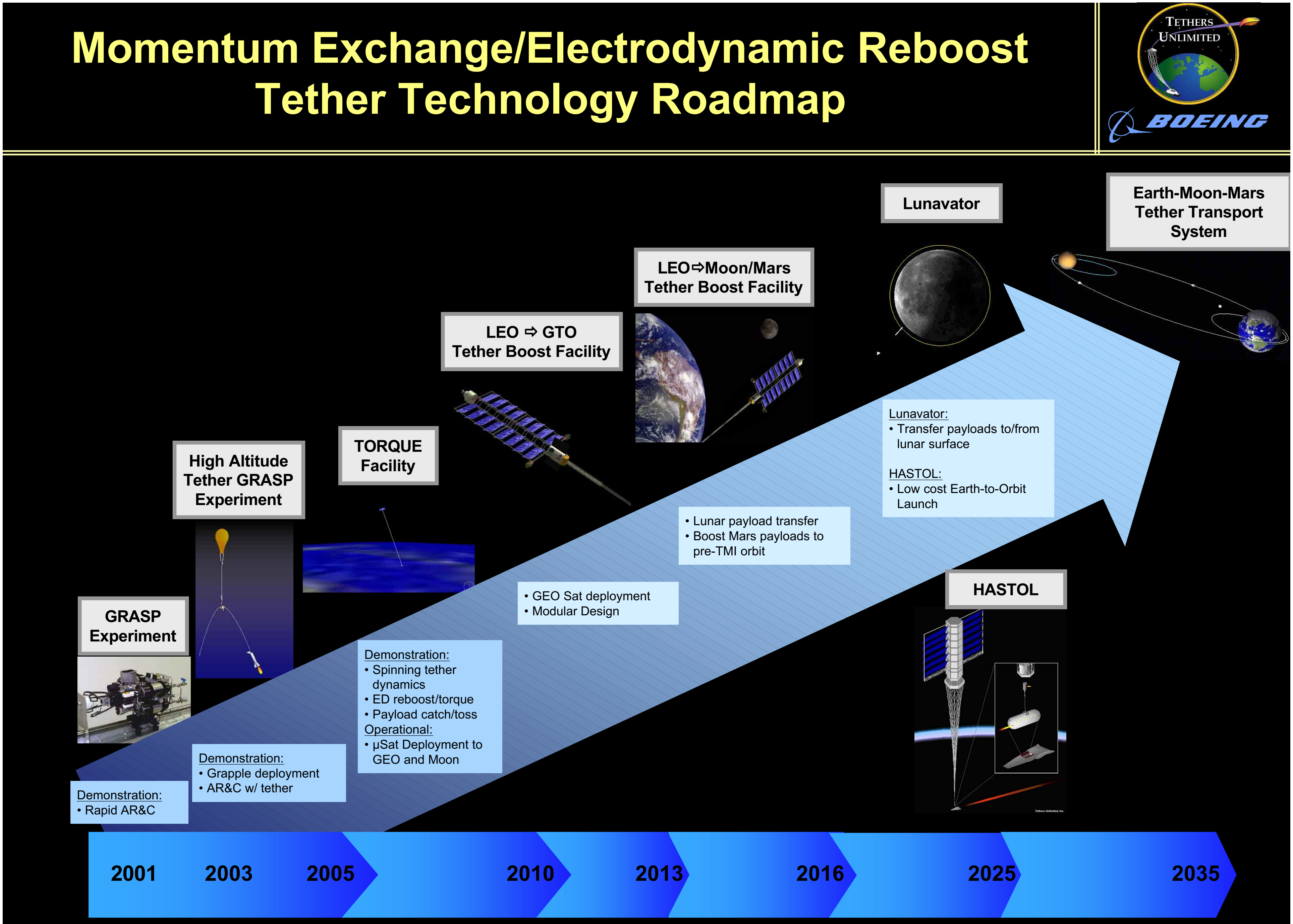
- **High-Altitude Tether (HAT)-GRASP**
  - Deploy Tether Below High-Altitude Balloon
  - Launch Payload On Small Sounding Rocket
  - Payload Maneuvers + Rendezvous with Tether



- **TORQUE - Tether Orbit Raising Qualificatio[illegible]nt(s)**
  - Deploy Hanging Tether
  - AR&C w/ Hanging Tether
  - Electrodynamic Spin-Up of Tether
  - Controlled Toss of Payload
  - Electrodynamic Reboost of Facility
  - Repeated Boosting of Commercial & Scientific μSats



Momentum Exchange/Electrodynamic Reboost Tether Technology Roadmap



Opportunities for NASA Technology Development

- Expand AR&C Capabilities for Rapid Capture (GRASP)
- High Power & High Voltage Space Systems
- Electrodynamic Tether Physics
- Debris & Traffic Control Issues
- Include Tether Options in HEDS & Other Mission Architecture Studies

Modest NASA Investment in Technology Development Will Enable Near-Term Space Flight Demonstration

Plans for Second Year of Study

- Costing/Economic Analysis
- Technology Maturity Assessment
  - Focus Technology Development Plans
- System Design for:
  - TORQUE Technology Demonstration
  - Boost Station sized for µSat payloads
- Architectures for using tethers in a Mars transportation system
- Evaluate modular construction approaches
- Tether dynamics and rendezvous studies

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- Chauncey Uphoff

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Rapid Interplanetary Tether Transport Systems

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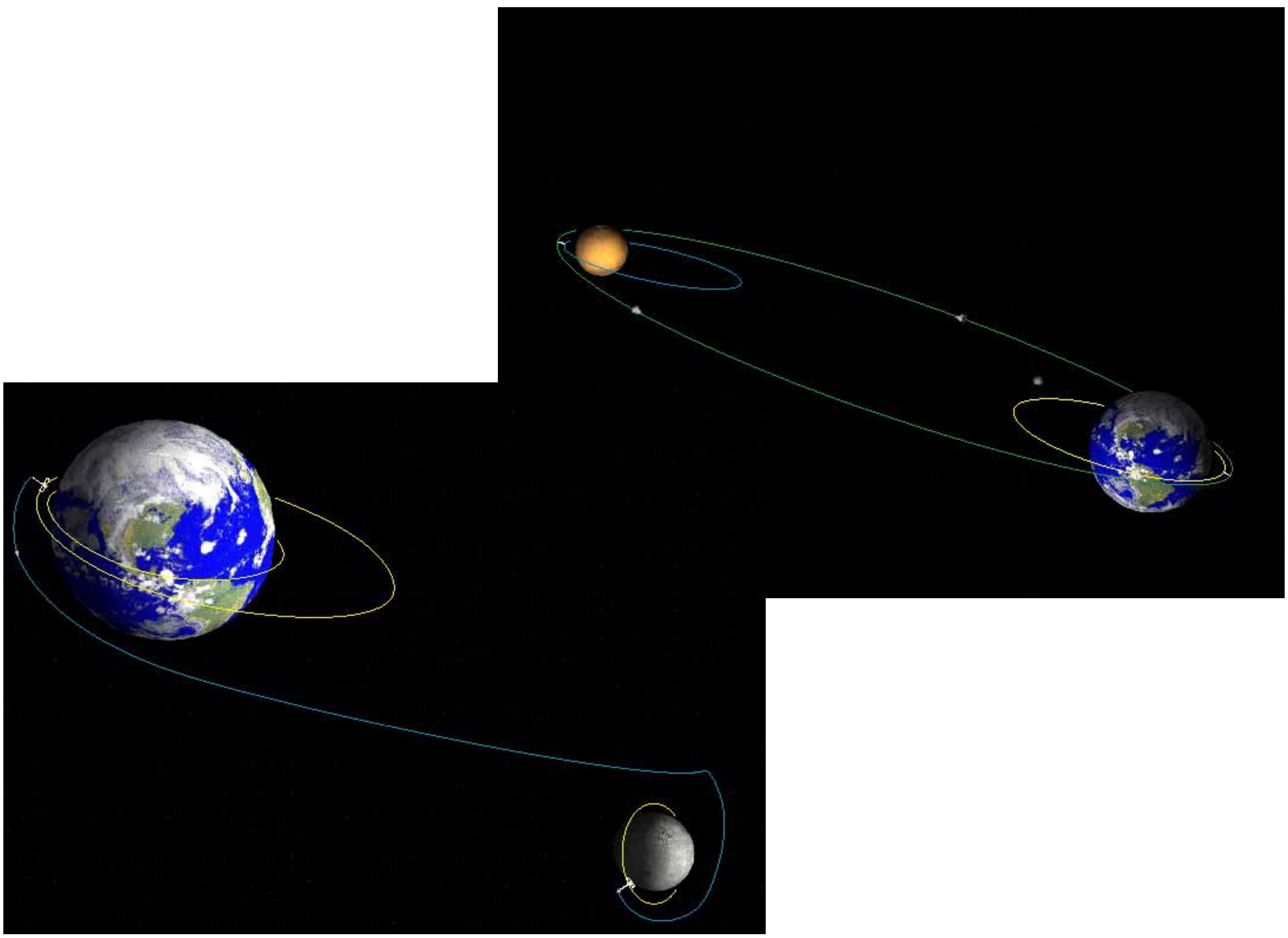

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RAPID INTERPLANETARY TETHER TRANSPORT SYSTEMS

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Abstract

Routine transport to and from Luna, Mars, and the other moons and planets in the solar system demands an efficient, rapid, low-cost transportation system. We have invented an innovative interplanetary transport architecture to meet that need. It consists of two rotating tethers in elliptical orbits, one around Earth and the other around the destination moon or planet. These two tethers, made of commercially available polymers, suffice to move payloads back and forth without the use of propellant except for midcourse corrections. For airless bodies, like Luna or Mercury, the payloads can be delivered to the surface of the body. We will describe two such architectures in detail, a Cislunar Tether Transport system and a Mars-Earth Rapid Interplanetary Tether Transport (MERITT) system. The Cislunar Tether Transport scenario takes into account the full complexities of the orbital mechanics of the Earth-Moon system, including non-spherical gravitational potentials, inclined orbit dynamics, and luni-solar perturbations. We also describe a design for the first stage of the system, a "rotating electrodynamic force tether" that combines the technology of electrodynamic tethers with the principles of rotating momentum-transfer tethers to enable multiple payloads to be boosted from LEO to higher orbits with no propellant needed. In the MERITT system, a payload capsule in LEO is picked up by the Earth orbiting tether as the tether nears perigee and is tossed a half-rotation later, slightly after perigee. The velocity increment given the payload deep in the gravity well of Earth is sufficient to send the payload on an escape trajectory to Mars, where it is caught by the Mars tether and placed in low Martian orbit. The mass of each tether system, using commercially available polymers and reasonable safety factors, including the central facility and ballast mass, can be as little as 15 times the mass of the payload being handled. Tethers with tip velocities of 2.5 km per second can send payloads to Mars in as little as 90 days if aerobraking is used dissipate some of the high relative velocity on the Mars end. Tether-to-tether transfers without aerobraking take 130 to 160 days.

Nomenclature & Units

- a semimajor axis, m
- C_3 orbital energy, $\equiv V^2 - 2\mu/r$, km^2/s^2
- d density, kg/m^3
- e ellipse eccentricity
- E orbital energy, J
- F safety factor
- h specific angular momentum, m^2/s
- i orbit inclination, degrees
- J_2 2nd geopotential coefficient
- L tether arm length, m
- l distance from facility to system's center of mass.
- M mass, kg
- N orbital resonance parameter
- p orbit semiparameter, $= a(1-e^2)$, m
- r radius, m
- R_e Earth radius, m
- r_p perigee radius, m
- T tensile strength, Pa
- V velocity, m/s
- V_C characteristic velocity, m/s
- λ argument of tether perigee w.r.t. Earth-Moon line
- μ_e Earth's gravitational parameter $= GM_e$, m^3/s^2
- μ_m Moon's gravitational parameter $= GM_m$, m^3/s^2
- ω angular velocity, radians/s
- θ true anomaly
- $\dot{\omega}$ Apsidal precession/regression rate, rad/s
- $\dot{\Omega}$ Nodal regression rate, radians/s

subscripts:

$\blacksquare_a$	apoapse	$\blacksquare_p$	periapse
$\blacksquare_c$	critical	$\blacksquare_m$	moon
$\blacksquare_f$	facility	$\blacksquare_g$	grapple
$\blacksquare_P$	payload	$\blacksquare_t$	tether

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Introduction

The possibility of using rotating "momentum-exchange" tethers to pick up payloads from one orbit and toss them into another orbit has been discussed conceptually numerous times over the past several decades.[1,2,3,4] In this paper, we investigate the design of specific tether system architectures for two important missions: first, transport between low Earth orbit (LEO) and the surface of the Moon, and second, transport of payloads between LEO and low Mars orbit.

The Cislunar Tether Transport System

A "Cislunar Tether Transport System" composed of one rotating momentum-exchange tether in elliptical, equatorial Earth orbit and a second rotating tether facility in a low lunar orbit can provide a means for repeatedly exchanging payloads between low Earth orbit (LEO) and the surface of the Moon, *with little or no propellant expenditure required*. In 1991, Forward[5] showed that such a system is theoretically possible from an energetics standpoint. A later study by Hoyt and Forward[6] developed a first-order design for such a system. These previous studies, however, utilized a number of simplifying assumptions regarding orbital and tether mechanics in the Earth-Moon system, including assumptions of coplanar orbits, ideal gravitational potentials, and infinite facility ballast masses. The purpose of this paper is to remove these assumptions and develop an architecture for such a system that takes into account the complexities of orbital mechanics in the Earth-Moon system.

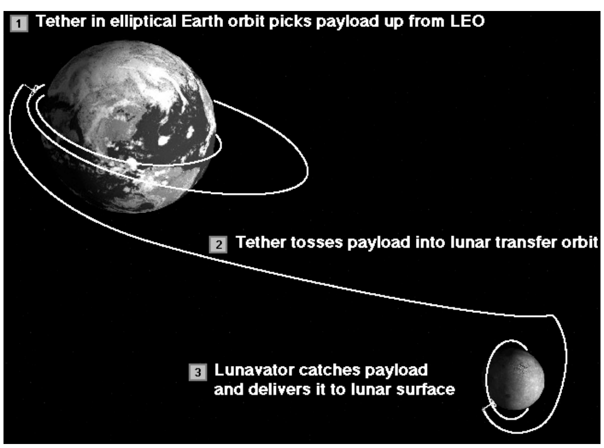

Figure 1. Conceptual illustration of the Cislunar Tether Transport System.

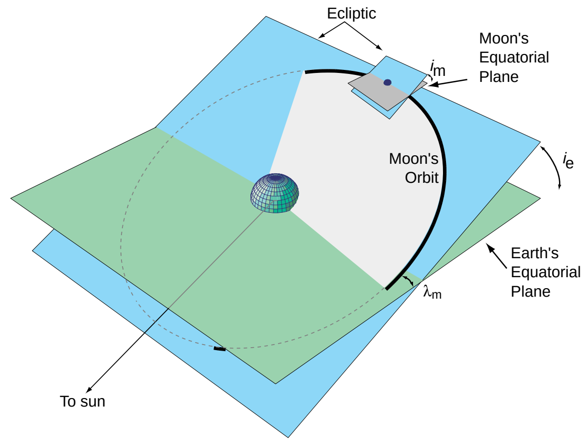

Figure 2. Schematic illustrating the geometry of the Earth-Moon system.

The basic concept of the Cislunar Tether Transport System is to use a rotating tether in Earth orbit to pick payloads up from LEO orbits and toss them to the Moon, where a rotating tether in lunar orbit, called a "Lunavator™", could catch them and deliver them to the lunar surface. As the Lunavator™ delivers payloads to the Moon's surface, it can also pick up return payloads, such as water or aluminum processed from lunar resources, and send them down to LEO. By balancing the flow of mass to and from the Moon, the orbital momentum and energy of the system can be conserved, eliminating the need to expend large quantities of propellant to move the payloads back and forth. This system is illustrated in Figure 1.

Orbital Mechanics of the Earth-Moon System

Orbital mechanics in cislunar space are made quite complex by the different and varying orientations of the ecliptic plane, the Earth's equatorial plane, the Moon's orbital plane, and the Moon's equatorial plane. Figure 2 attempts to illustrate these different planes. The inclination of the Earth's equatorial plane (the "obliquity of the ecliptic"), is approximately 23.45°, but varies due to tidal forces exerted by the Sun and Moon. The angle i_m between the Moon's equatorial plane and a plane through the Moon's center that is parallel to the ecliptic plane is constant, about 1.58°. The inclination of the Moon's orbit relative to the ecliptic plane is also constant, about λ_m = 5.15°.[7] The line of nodes of the Moon's orbit regresses slowly, revolving once every 18.6 years. As a result, the inclination of the Moon's orbit relative to the Earth's equator varies between

18.3-28.6 degrees. The Moon's orbit also has a slight eccentricity, approximately $e_m = 0.0549$.

Tether Orbits

After considering many different options, including the three-tether systems proposed previously and various combinations of elliptical and circular orbits, we have determined that the optimum configuration for the Cislunar Tether system is to utilize one tether in an elliptical, equatorial Earth orbit and one tether in a polar, circular lunar orbit, as illustrated in Figure 1. This two-tether system will require the lowest total system mass, minimize the system complexity and provide the most frequent transfer opportunities. The Earth-orbit tether will pick payloads up from equatorial low-LEO orbits and throw them towards one of the two points where the Moon crosses the Earth's equatorial plane. As the payload approaches the Moon, it will need to perform a small ΔV maneuver to set it up into the proper approach trajectory; the size of this maneuver will vary depending upon the inclination of the Moon's orbit plane and launch dispersions, but under most conditions it will only require about 25 m/s of ΔV.

In the following sections, we will first develop a design for a tether facility for boosting payloads from low-LEO orbits to lunar transfer orbits (LTO). We will then develop a design for a "Lunavator™" capable of catching the payloads and delivering them to the surface of the Moon. We will then discuss the numerical simulations used to verify the feasibility of this system architecture.

Design of a Tether Boost Facility for Lunar Transfer Injection

The first stage of the Cislunar Tether Transport System will be a tether boost facility in elliptical Earth orbit capable of picking payloads up from low-LEO orbits and tossing them to the Moon. In order to determine an optimum configuration for this facility, we must balance the need to minimize the required masses of the tethers and facilities with the need to make the orbital dynamics of the system as manageable as possible.

The mission of the Earth-orbit portion of the Cislunar Tether Transport System is to pick up a payload from low-Earth orbit and inject it into a near-minimum energy lunar transfer orbit. The desired lunar transfer trajectories have a C_3 of approximately –1.9 (km/s)^2. A payload originating in a circular orbit at 350 km altitude has an initial velocity of 7.7 km/s and a C_3 of –60 (km/s)^2. To impulsively inject the payload into a trajectory with a C_3 of –1.9 would require a ΔV of approximately 3.1 km/s.

Design Considerations

Tether System Staging

From an operational standpoint, the most convenient design for the Earth-orbit portion of a Cislunar Tether Transport System would be to start with a single tether facility in a circular low-Earth-orbit, with the tether retracted. The facility would rendezvous with the payload, deploy the payload at the end of the tether, and then use propellantless electrodynamic tether propulsion to spin up the tether until the tip speed reached 3.1 km/s and the tether could inject the payload into a LTO. However, because the tether transfers some of its orbital momentum and energy to the payload when it boosts it, a tether facility in circular orbit would require a very large ballast mass so that its orbit would not drop into the upper atmosphere after it boosts a payload. Furthermore, the strong dependence of the required tether mass on the tether tip speed will likely make this approach impractical with current material technologies. The required mass for a tapered tether depends upon the tip mass and the ratio of the tip velocity to the tether material's critical velocity according to the relation derived by Moravec:[8]

$$M_t = M_p \sqrt{\pi} \frac{\Delta V}{V_C} e^{\frac{\Delta V^2}{V_C^2}} erf\left\{\frac{\Delta V}{V_C}\right\}, \qquad (1)$$

where erf() is the error function. The critical velocity of a tether material depends upon the tensile strength, the material density, and the design safety factor according to:

$$V_C = \sqrt{\frac{2T}{Fd}}. \qquad (2)$$

The exponential dependence of the tether mass on the *square* of the velocity ratio results in a very rapid increase in tether mass with this ratio.

Currently, the best commercially-available tether material is Spectra® 2000, a form of highly oriented polyethlene manufactured by AlliedSignal. High-quality specimens of Spectra® 2000 have a room temperature tensile strength of 4 GPa, and a density of 0.97 g/cc. With a safety factor of 3, the material's critical velocity is 1.66 km/s. Using Eqn. (1), an optimally-tapered Spectra® tether capable of sustaining a tip velocity of 3.1 km/s would require a mass of over 100 times the payload mass. While this might be technically feasible for very small payloads, such a large tether mass probably would not be economically competitive with rocket technologies. In the future, very high strength materials such as "buckytube" yarns may become available with tensile strengths that will make a 3 km/s tether feasible; however, we will show that a different approach to the system architecture can utilize currently available materials to perform the mission with reasonable mass requirements.

The tether mass is reduced to reasonable levels if the $\Delta V/V_c$ ratio can be reduced to levels near unity or lower. In the Cislunar system, we can do this by placing the Earth-orbit tether into an elliptical orbit and arranging its rotation so that, at perigee, the tether tip can rendezvous with and capture the payload, imparting a 1.6 km/s ΔV to the payload. Then, when the tether returns to perigee, it can toss the payload ahead of it, giving it an additional 1.5 km/s ΔV. By breaking the 3.1 km/s ΔV up into two smaller boost operations with $\Delta V/V_c < 1$, we can reduce the required tether mass considerably. The drawback to this method is that it requires a challenging rendezvous between the payload and the tether tip; nonetheless, the mass advantages will likely outweigh that added risk.

Behavior of Elliptical Earth Orbits

One of the major challenges to designing a workable tether transportation system using elliptical orbits is motion of the orbit due to the oblateness of the Earth. The Earth's oblateness will cause the plane of an orbit to regress relative to the Earth's spin axis at a rate equal to:[9]

$$\dot{\Omega} = -\frac{3}{2} J_2 \frac{R_e^2}{p^2} \bar{n} \cos(i) \qquad (3)$$

And the line of apsides (ie. the longitude of the perigee) to precess or regress relative to the orbit's nodes at a rate equal to:

$$\dot{\omega} = \frac{3}{4} J_2 \frac{R_e^2}{p^2} \bar{n} \left(5\cos^2 i - 1\right) \qquad (4)$$

In equations (3) and (4), $\bar{n}$ is the "mean mean motion" of the orbit, defined as

$$\bar{n} = \sqrt{\frac{\mu_e}{a^3}}\left[1 - \frac{3}{4} J_2 \frac{R_e^2}{p^2} \sqrt{1-e^2} \left(1 - 3\cos^2 i\right)\right]. \qquad (5)$$

For an equatorial orbit, the nodes are undefined, but we can calculate the rate of apsidal precession relative to inertial space as the sum $\dot{\Omega} + \dot{\omega}$ of the nodal and apsidal rates given by Eqs. (3) and (4).

In order to make the orbital mechanics of the Cislunar Tether Transport System manageable, we place two constraints on our system design:

- First, the orbits of the tether facility will be equatorial, so that *i*=0 and the nodal regression given by Eq. (3) will not be an issue.
- Second, the tether system will throw the payload into a lunar transfer trajectory that is in the equatorial plane. This means that it can perform transfer operations when the Moon is crossing either the ascending or descending node of its orbit.

Nonetheless, we still have the problem of precession of the line of apsides of an orbit. If the tether orbits are circular, this is not an issue, but it is an issue for systems that use elliptical orbits. In an elliptical orbit system we wish to perform all catch and throw operations at or near perigee. As illustrated in Figure 3, for the payload to reach the Moon's radius at the time when the Moon crosses the Earth's equatorial plane, the payload must be injected into an orbit that has a line of apsides at some small angle λ from the line through the Moon's nodes. If the orbit

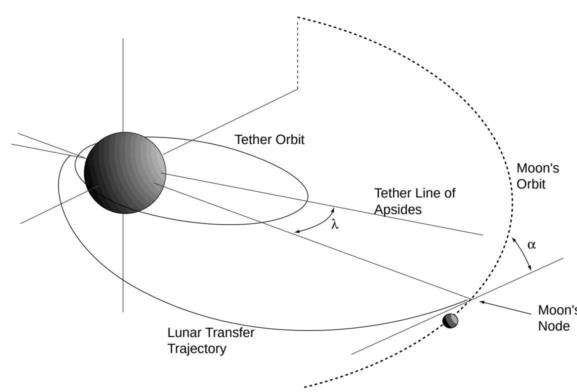

Figure 3. Geometry of the tether orbit and the Moon's orbit.

experiences apsidal precession, the angle λ will have the proper value only periodically. Consequently, in our designs we will seek to choose the orbital parameters such that the apsidal precession of the orbit will have a convenient resonance with the Moon's orbit.

Elliptical-Orbit Tether Boost Facility

In the Cislunar Tether Transport System, the transfer of payloads between a low-LEO and lunar transfer orbits is performed by a single rotating tether facility. This facility performs a catch and release maneuver to provide the payload with two boosts of approximately 1.5 km/s each. To enable the tether to perform two "separate" ΔV operations on the payload, the facility is placed into a highly elliptical orbit with its perigee in LEO. First, the tether rotation is arranged such that when the facility is at perigee, the tether is swinging vertically below the facility so that it can catch a payload moving more slowly than the facility. After it catches the payload, it waits for one orbit and adjusts its rotation slightly (by reeling the tether in or out) so that when it returns to perigee, the tether is swinging above the facility and it can release the payload into a trajectory moving faster than the facility.

HEFT Tether Boost Facility

In order to enable the Earth-orbit tether facility to boost materials to the Moon before a lunar base has been established and begins sending return payloads back to LEO, we propose to combine the principle of rotating momentum-exchange tethers with the techniques of electrodynamic tether propulsion to create a facility capable of reboosting its orbit after each payload transfer without requiring return traffic or propellant expenditure. This concept, the "High-strength Electrodynamic Force Tether" (HEFT) Facility,[10] is illustrated in Figure 4. The HEFT Facility would include a central facility housing a power supply, ballast mass, plasma contactor, and tether deployer, which would extend a long, tapered, high-strength tether. A small grapple vehicle would reside at the tip of the tether to facilitate rendezvous and capture of the payloads. The tether would include a conducting core, and a second plasma contactor would be placed near the tether tip. By using the power supply to drive current along the tether, the HEFT Facility could generate electrodynamic forces on the tether. By properly varying the direction of the current as the tether rotates and orbits the Earth, the facility can use these electrodynamic forces to generate either a net torque on the system to increase its rotation rate, or a net thrust on the system to boost its orbit. The HEFT Facility thus could repeatedly boost payloads from LEO to the Moon, using propellantless electrodynamic propulsion to restore its orbit in between each payload boost operation.

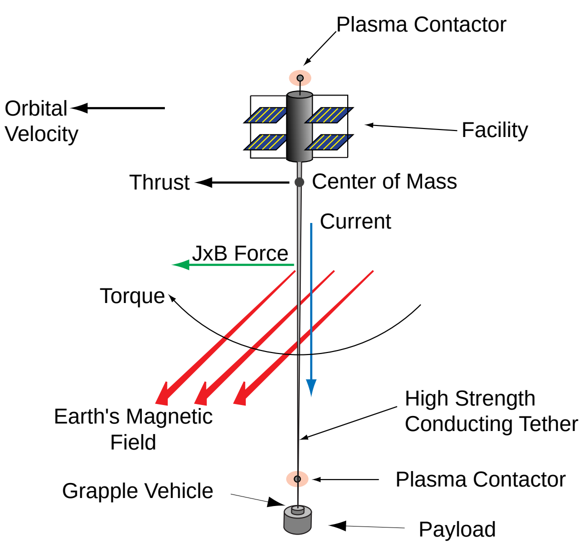

Figure 4. Schematic of the HEFT Facility design.

Tether Design

In order to design the tether boost facility, we must determine the tether length, rotation rate, and orbit characteristics that will permit the tether to rendezvous with the payload and throw it into the desired lunar transfer trajectory.

In the baseline design, the payload begins in a circular Initial Payload Orbit (IPO) with a velocity of

$$V_{p,0} = \sqrt{\frac{\mu_e}{r_{IPO}}}. \tag{6}$$

The facility is placed into an elliptical orbit with a perigee above the payload's orbit, with the difference between the facility's initial perigee and the payload orbital radius equal to the distance from the tether tip to the center of mass of the facility and tether:

$$r_{p,0} = r_{IP0} + (L - l_{cm,unloaded}), \tag{7}$$

where $l_{cm,unloaded}$ is the distance from the facility to the center of mass of the system before the payload arrives (this distance must be calculated numerically for a tapered tether).

The tether tip velocity is equal to the difference between the payload velocity and the facility's perigee velocity:

$$V_{t,0} = V_{p,0} - V_{IPO}. \tag{8}$$

In order to ensure that a payload will not be "lost" if it is not caught by the tether on its first opportunity, we choose the semimajor axis of the facility's orbit such that its orbital period will be some rational multiple *N* of the payload's orbital period:

$$P_{f,0} = NP_{IPO} \quad \Rightarrow \quad a_{f,0} = N^{2/3}\ r_{IPO} \tag{9}$$

For example, if N=5/2, this condition means that every two orbits the facility will have an opportunity to rendezvous with the payload, because in the time the facility completes two orbits, the payload will have completed exactly five orbits.

An additional consideration in the design of the system are the masses of the facility and tether. A significant facility mass is required to provide "ballast mass." This ballast mass serves as a "battery" for storing the orbital momentum and energy that the tether transfers to and from payloads. If all catch and throw operations are performed at perigee, the momentum exchange results primarily in a drop in the facility's apogee. A certain minimum facility mass is necessary to keep the post catch and throw orbit above the Earth's upper atmosphere. Some of the "ballast mass" will be provided by the mass of the tether deployer and winch, the facility power supply and power processing hardware, and the mass of the tether itself. If additional mass is required, it could be provided by available material in LEO, such as spent upper stage rockets and shuttle external tanks.

The tether mass required will depend upon the maximum tip velocity and the choices of tether material and design safety factor, as described by Eq. 1. For a tapered tether, the tether's center-of-mass will be closer to the facility end of the tether. This can be an important factor when the tether mass is significant compared to the payload and facility masses. In the calculations below, we have used a model of a tether tapered in a stepwise manner to calculate tether masses and the tether center-of-mass.

By conservation of momentum, the perigee velocity of the center of mass of the tether and payload after rendezvous is:

$$V_{p,1} = \frac{V_{p,0}(M_f + M_t) + V_{IPO}M_P}{(M_f + M_t) + M_P}. \tag{10}$$

When the tether catches the payload, the center-of-mass of the tether system shifts downward slightly as the payload mass is added at the bottom of the tether:

$$r_{p,1} = \frac{r_{p,0}(M_f + M_t) + V_{IPO}M_P}{(M_f + M_t) + M_P} \tag{11}$$

In addition, when the tether catches the payload, the angular velocity of the tether does not change, but because the center-of-mass shifts closer to the tip of the tether when the tether catches the payload, the tether tip velocity decreases. The new tether tip velocity can be calculated as

$$V_t^{'} = V_t \frac{\left(L - l_{cm,loaded}\right)}{\left(L - l_{cm,unloaded}\right)} \tag{12}$$

At this point, it would be possible to specify the initial payload orbit, the payload/facility mass ratio, the facility/payload period ratio, and the desired LTO C_3, and derive a system of equations from which one particular tether length and one tether tip velocity can be calculated that determine an "exact" system where the tether tip velocity need not be adjusted to provide the desired C_3 of the payload lunar trajectory. However, the resulting system design is rather restrictive, working optimally for only one particular value of the facility and tether masses, and results in rather short tether lengths that will require very high tip acceleration levels. Fortunately, we can provide an additional flexibility to the system design by allowing the tether facility to adjust the tip velocity slightly by reeling the tether in or out a few percent. If, after catching the payload, the facility reels the tether in by an amount ΔL, the tip velocity will increase due to conservation of angular momentum:

$$V_t^{''} = \frac{V_t^{'}\left(L - l_{cm,loaded}\right)}{\left(L - l_{cm,loaded}\right) - \Delta L} \tag{13}$$

Then, when the facility returns to perigee, it can throw the payload into a lunar transfer trajectory with perigee characteristics:

$$r_{p,LTO} = r_{p,1} + \left(L - l_{cm,loaded}\right) - \Delta L$$
$$V_{p,LTO} = V_{p,1} + V_t' \qquad (14)$$

Using the equations above, standard Keplerian orbital equations, and equations describing the shift in the system's center-of-mass as the payload is caught and released, we have calculated a design for a single-tether system capable of picking up payloads from a circular LEO orbit and throwing them to a minimal-energy lunar trajectory. During its initial period of operation, while a lunar facility is under construction and no return traffic exists, the tether system will use electrodynamic tether propulsion to reboost itself after throwing each payload. Once a lunar facility exists and return traffic can be used to conserve the facility's orbital momentum, the orbit of the tether will be modified slightly to permit round trip traffic. The system parameters are listed below.

Table 1:

Initial System Design: Outbound Traffic Only

<u>Payload:</u>
- mass M_p = 2500 kg
- altitude h_{IPO} = 308 km
- velocity V_{IPO} = 7.72 km/s

<u>Tether Facility:</u>
- tether length L = 80 km
- tether mass M_t = 15,000 kg
  (Spectra® 2000 fiber, safety factor of 3.5)
- tether center-of-mass $L_{t,com}$ = 17.6 km
  (from facility)
- central facility mass M_f = 11,000 kg
- grapple mass M_g = 250 kg
  (10% of payload mass)
- total system mass M = 26,250 kg
  = **10.5 x payload mass**
- facility power Pwr = 11 kW avg
- initial tip velocity: $V_{t,0}$ = 1530 m/s
- <u>Pre-Catch Orbit:</u>
  - perigee altitude $h_{p,0}$ = 378 km,
  - apogee altitude $h_{a,0}$ = 11,498 km
  - eccentricity e_0 = 0.451
  - period P_0 = $5/2P_{IPO}$
  (rendezvous opportunity every 7.55 hrs)
- <u>Post-Catch Orbit:</u>
  - perigee altitude $h_{p,1}$ = 371 km,
  - apogee altitude $h_{a,1}$ = 9687 km

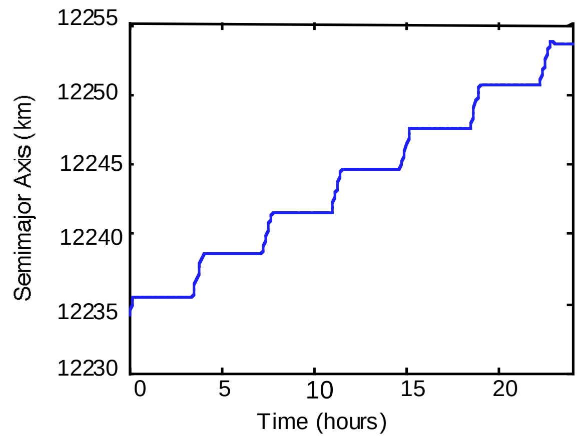

Figure 5. Electrodynamic propulsion reboost of the tether's orbit after the tether has boosted a payload into LTO.

  - eccentricity e_1 = 0.408

After catching the payload, the facility reels in 2950 m of tether, increasing the tip velocity to 1607 m/s,

- <u>Post-Throw Orbit:</u>
  - perigee altitude $h_{p,2}$ = 365 km,
  - apogee altitude $h_{a,2}$ = 7941 km
  - eccentricity e_2 = 0.36

<u>Lunar Transfer Trajectory:</u>
- perigee altitude $h_{p,lto}$ = 438.7 km
- perigee velocity $V_{p,lto}$ = 10.73 km/s
- trajectory energy C_3 = -1.9 km^2/s^2

Note that for a particular system design, the tether and facility mass will scale roughly linearly with the payload mass, so an equivalent system designed for sending 250 kg payloads to the Moon could be constructed with a tether mass of 1,500 kg and a facility mass of 1,100 kg. Note also that the tether mass is not dependent upon the tether length, so longer tethers can be used to provide lower tip acceleration levels with no mass penalty.

Electrodynamic Reboost of the Tether Orbit

After boosting the payload, the tether facility will be left in a lower energy elliptical orbit with a semimajor axis that is approximately 1780 km less than its original orbit. Once a lunar base and a lunar tether facility have been established and begin to send return traffic down to LEO, the tether facility can restore its orbit by catching and de-boosting these return payloads. In the period before a lunar base is established, however, the tether facility will use electrodynamic propulsion to reboost its apogee by driving current through the tether when the tether is near perigee. Because the tether is rotating, the direction of the current must be alternated as the tether rotates to produce a net thrust on the facility. Using a simulation of tether dynamics and electrodynamics, we have modeled reboost of a rotating tether system. Figure 5 shows the reboost of the tether's orbit over one day, assuming that the tether facility has a power supply of 11 kW and is able to store up power during most of its orbit and expend it at a rate of 75 kW during the portion of the orbit when the tether is below 2000 km altitude. In one day, the facility can restore roughly 20 km to its orbit's semimajor axis; in roughly 85 days it could restore its orbit and be prepared to boost another payload to the Moon. More rapid reboost could be accomplished with a larger power supply.

Dealing with Apsidal Precession

As noted earlier, the oblateness of the Earth will cause the line of apsides of the tether facility's elliptical orbit to precess. In the Cislunar Tether Transport System, we can deal with this issue in two ways. First, we can utilize tether reeling maneuvers to counteract the apsidal precession.[11] By simply reeling the tether in and out slightly once per orbit, the tether facility can exchange angular momentum between its rotation and its orbit, resulting in precession or regression of the line of apsides. With proper phasing and amplitude, tether reeling can hold the tether's orbit fixed so that it can send payloads to the Moon once per month.[12]

A second method is to choose the tether orbits such that their precession rates are nearly harmonic with the Moon's orbital rate, so that the line of apsides lines up with the Moon's nodes once every several months. Furthermore, we can use propellantless electrodynamic tether propulsion to "fine-tune" the precession rate, either by raising/lowering the orbit or by generating thrust perpendicular to the facility's velocity.

In the design given above, the mass and initial orbit of the tether facility was chosen such that after throwing a payload to the Moon, the tether enters a lower energy elliptical orbit which will precess at a rate of 2.28 degrees per day. The initial, high-energy orbit has a slower precession rate of approximately 1.58 degrees per day. These orbits were chosen so that in the 95.6 days it takes the Moon to orbit 3.5 times around the Earth, the tether facility can reboost itself from its low-energy orbit to its high-energy orbit using propellantless electrodynamic propulsion, and, by properly varying the reboost rate, the apsidal precession can be adjusted so that the line of apsides will rotate exactly 180°, lining the tether orbit up properly to boost another payload to the Moon.

System Design for Round-Trip Traffic

Once a lunar base is established and begins to send payloads back down to LEO, the orbit of the tether system can be modified slightly to enable frequent opportunities for round-trip travel. First, the facility's orbit will be raised so that its high-energy orbit has a semimajor axis of 12577.572 km, and an eccentricity of 0.41515. The tether will then pick up a payload from a circular, 450 km orbit and toss it to the Moon so that it will reach the Moon as the Moon crosses its ascending node. The facility will then drop to a lower energy orbit. At approximately the same time, the return payload will be released by the lunar tether and begin its trajectory down to LEO. When the return payload reaches LEO, the Earth-orbit tether facility will catch it at perigee, carry it for one orbit, and then place it into the 450 km initial payload orbit. Upon dropping the return payload, the facility will place itself back into the high-energy orbit. The perigee of this orbit will precess at a rate such that after 4.5 lunar months (123 days) it will have rotated 180°, and the system will be ready to perform another payload exchange, this time as the Moon crosses its descending node. If more frequent round-trip traffic is desired, tether reeling could again be used to hold the orientation of the tether's orbit fixed, providing transfer opportunities once per sidereal month.

Design of a Lunavator™ Compatible with Minimal-Energy Lunar Transfers

The second stage of the Cislunar Tether Transport System is a lunar-orbit tether facility that catches the payloads sent by the Earth-orbit tether and deposits them on the Moon with zero velocity relative to the surface.

Background: Moravec's Lunar Skyhook

In 1978, Moravec[8] proposed that it would be possible to construct a tether rotating around the Moon that would periodically touch down on the lunar surface. Moravec's "Skyhook" would have a massive central facility with two tether arms, each with a length equal to the facility's orbital altitude. It would rotate in the same direction as its orbit with a tether tip velocity equal to the orbital velocity of the tether's center-of-mass so that the tether tips would periodically touch down on the Moon with zero velocity relative to the surface (to visualize this, imagine the tether as a spoke on a giant bicycle wheel rolling around the Moon).

As it rotates and orbits around the Moon, the tether could capture payloads from Earth as they passed perilune and then set them down on the surface of the Moon. Simultaneously, the tether could pick up payloads to be returned to Earth, and later throw them down to LEO.

Moravec found that the mass of the tether would be minimized if the tether had an arm length equal to one-sixth of the diameter of the Moon, rotating such that each of the two arms touched down on the surface of the Moon three times per orbit. Using data for the best material available in 1978, Kevlar, which has a density of 1.44 g/cc and a tensile strength of 2.8 GPa, Moravec found that a two-arm Skyhook with a design safety factor of F=2 would have to mass approximately 13 times the payload mass. Each arm of Moravec's tether would be 580 km long, for a total length of 1160 km, and the tether center-of-mass would orbit the Moon every 2.78 hours in a circular orbit with radius of 2,320 km. At that radius, the orbital velocity is 1.45 km/s, and so Moravec's Skyhook would rotate with a tip velocity of 1.45 km/s.

Using Moravec's minimal-mass solution, however, requires not only a very long tether but also requires that the payload have a very high velocity relative to the Moon at its perilune. Because the lunar tether in Moravec's design has an orbital velocity of 1.45 km/s and the tether tips have a velocity of 1.45 km/s relative to the center-of-mass, the payload's perilune velocity would need to be 2.9 km/s in order to match up with the tether tip at the top of their rotation. In order to achieve this high perilune velocity, the outbound lunar transfer trajectory would have to be a high-energy hyperbolic trajectory. This presented several drawbacks, the most significant being that if the lunar tether failed to capture the payload at perilune, it would continue on and leave Earth orbit on a hyperbolic trajectory. Moreover, as Hoyt and Forward[6] found, a high lunar trajectory energy would also place larger ΔV demands on the Earth-orbit tethers, requiring two tethers in Earth orbit to keep the system mass reasonable.

Lunavator™ Design

In order to minimize the ΔV requirements placed upon the Earth-orbit portion of the Cislunar Tether Transport System and thereby permit the use of a single Earth-orbit tether with a reasonable mass, we have developed a method for a single lunar-orbit tether to capture a payload from a minimal-energy lunar transfer orbit and deposit it on the tether surface with zero velocity relative to the surface.

Moon-Relative Energy of a Minimum-Energy LTO

A payload that starts out in LEO and is

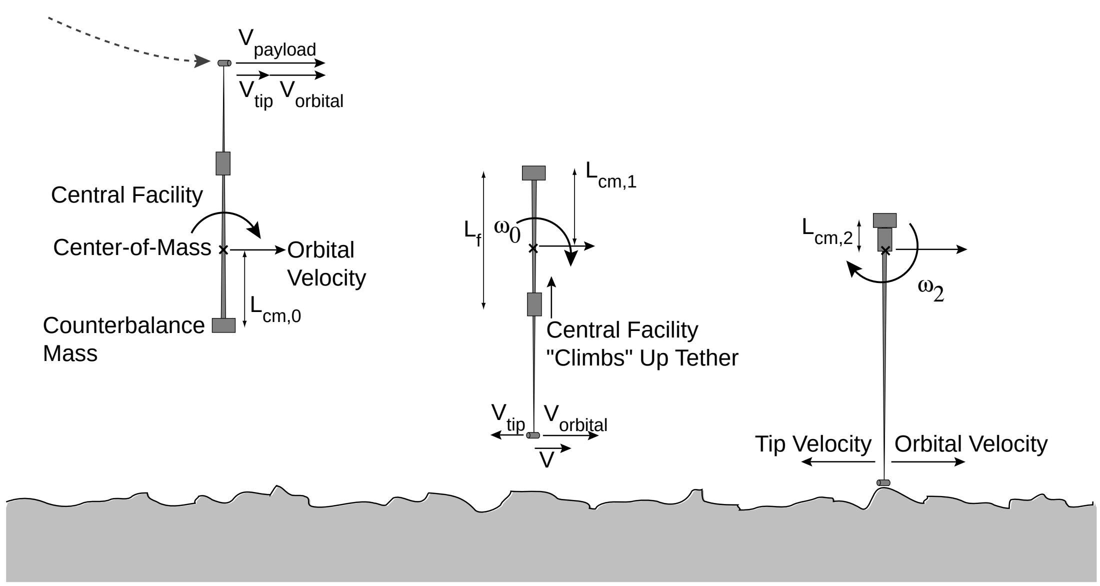

Figure 6. Method for a lunar tether to capture a payload from a minimal-energy LTO and deposit it on the Moon with zero velocity relative to the surface.

injected into an elliptical, equatorial Earth-orbit with an apogee that just reaches the Moon's orbital radius will have a C_3 relative to the Moon of approximately 0.72 km^2/s^2. For a lunar transfer trajectory with a closest-approach altitude of several hundred kilometers, the payload will have a velocity of approximately 2.3 km/s at perilune. As a result, it would be moving too slowly to rendezvous with the upper tip of Moravec lunar Skyhook, which will have a tip velocity of 2.9 km/s at the top of its rotation. Consequently, the design of the lunar tether system must be modified to permit a tether orbiting the Moon at approximately 1.5 km/s to catch a payload to at perilune when the payload's velocity is approximately 2.3 km/s, then increase both the tether length and the angular velocity so that the payload can be set down on the surface of the Moon with zero velocity relative to the surface. Simply reeling the tether in or out from a central facility will not suffice, because reeling out the tether will cause the rotation rate to decrease due to conservation of angular momentum.

A method that can enable the tether to catch a payload and then increase the tether rotation rate while lowering the payload is illustrated in Figure 6. The "Lunavator™" tether system is composed of a long tether, a counterbalance mass at one end, and a central facility that has the capability to climb up or down the tether. Initially, the facility would locate itself near the center of the tether, and the system would rotate slowly around the center-of-mass of the system, which would be located roughly halfway between the facility and the counterbalance mass. The facility could then capture an inbound payload at its perilune. The facility would then use energy from solar cells or other power supply to climb up the tether towards the counterbalance mass. The center-of-mass of the system will remain at the same altitude, but the distance from the tether tip to the center-of-mass will increase, and conservation of angular momentum will cause the angular velocity of the system to increase as the facility mass moves closer to the center-of-mass.

Analysis

A first-order design for the Lunavator™ can be obtained by calculating the shift in the system's center-of-mass as the central facility changes its position along the tether. We begin by specifying the payload mass, the counterbalance mass, the facility mass, and the tether length. The required tether mass cannot be calculated simply by using Moravec's tapered tether mass equation, because that equation was derived for a free-space tether. The Lunavator™ must support not only the forces due to centripetal acceleration of the payload and tether masses, but also the tidal forces due to the Moon's gravity. The equations for the tether mass with gravity-gradient forces included are not analytically integrable, so the tether mass must be calculated numerically.

Prior to capture of the payload, the distance from the counterbalance mass to the center-of-mass of the tether system is

$$L_{cm,0} = \frac{M_f L_f + M_t L_{cm,t}}{M_c + M_f + M_t}, \tag{15}$$

where L_f is the distance from the counterbalance to the facility and $L_{cm,t}$ is the distance from the counterbalance to the center-of-mass of the tether. $L_{cm,t}$ must be calculated numerically for a tapered tether.

If the Lunavator™ is initially in a circular orbit with radius a_0, it will have a center-of-mass velocity of

$$v_{cm,0} = \sqrt{\frac{\mu_m}{a_0}}. \tag{16}$$

At the top of the tether swing, it can capture a payload from a perilune radius of

$$r_p = a_0 + (L_t - L_{cm,0}). \tag{17}$$

A payload sent from Earth on a near-minimum energy transfer will have a $C_{3,m}$ of approximately 0.72 km^2/s^2. Its perilune velocity will thus be

$$v_p = \sqrt{\frac{2\mu_m}{a_0 + (L_t - L_{cm,0})} + C_{3,m}}. \tag{18}$$

In order for the tether tip's total velocity to match the payload velocity at rendezvous, the velocity of the tether tip relative to the center of mass must be

$$v_{t,0} = v_p - v_{cm,0}, \tag{19}$$

and the angular velocity of the tether system will be

$$\omega_{t,0} = \frac{v_{t,0}}{L_t - L_{cm,0}}. \tag{20}$$

When the tether captures the payload, the center of mass of the new system, including the payload, is at perigee of a new, slightly elliptical orbit, as illustrated in Figure 7 (it was in a circular orbit and caught a payload going faster than the center-of-mass). The perigee radius and velocity of the center–of-mass are

$$v_{p,1} = \frac{v_{cm,0}(M_c + M_f + M_t) + v_p M_p}{M_c + M_f + M_t + M_p}, \quad (21)$$

$$r_{p,1} = \frac{a_0(M_c + M_f + M_t) + r_p M_p}{M_c + M_f + M_t + M_p}, \quad (22)$$

and the new distance from the counterbalance mass to the system's center-of-mass of the system changes to

$$L_{cm,1} = \frac{M_f L_f + M_t L_{cm,t} + M_p L_t}{M_c + M_f + M_t + M_p}. \quad (23)$$

To increase the rotation rate of the tether system and increase the distance from the system's center of mass to the tether tip, the facility climbs up the tether to the counterbalance mass, reducing the distance from the counterbalance to the center-of-mass to

$$L_{cm,2} = \frac{M_t L_{cm,t} + M_p L_t}{M_c + M_f + M_t + M_p}. \quad (24)$$

By conservation of angular momentum, the angular velocity will increase to a new value of

$$\omega_2 = \omega_0 \frac{\begin{bmatrix} L_{cm,1}M_c + (L_f - L_{cm,1})M_f + \\ (L_{cm,t} - L_{cm,1})M_t + (L_t - L_{cm,1})M_p \end{bmatrix}}{\begin{bmatrix} L_{cm,2}M_f + (L_{cm,t} - L_{cm,2})M_t \\ + (L_t - L_{cm,2})M_p \end{bmatrix}} \quad (25)$$

and the payload will then have a velocity relative to the center-of-mass of

$$v_{t,2} = \omega_2 (L_t - L_{cm,2}). \quad (26)$$

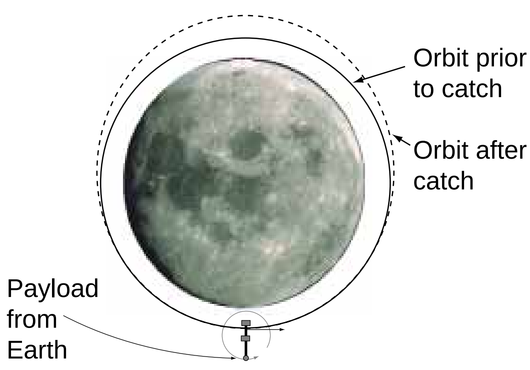

Figure 7. Lunavator™ orbits before and after payload capture.

If the initial orbit parameters, tether lengths, and facility and tether masses are chosen properly, then $v_{t,2}$ can be made equal to the perigee velocity of the tether system and the distance from the center of mass to the payload can be made equal to the perigee altitude. When the tether returns to its perigee it can then deposit the payload on the surface of the Moon and simultaneously pick up a payload to be thrown back to Earth.

Lunavator™ Design

Using the equations given above, we have found the following first-order design for a Lunavator™ capable of catching payloads from minimal-energy lunar transfer orbits and depositing them on the surface of the Moon:

Table 2: Baseline Lunavator™ Design

<u>Payload Trajectory:</u>
- mass M_p = 2500 kg
- perigee altitude h_p = 328.23 km
- Moon-relative energy $C_{3,M}$ = 0.719 km^2/s^2

<u>Lunavator™:</u>
- tether length L = 200 km
- counterbalance mass M_c = 15,000 kg
- facility mass M_f = 15,000 kg
- tether mass M_t = 11,765 kg
- Total Mass M = 41,765 kg

= 16.7 x payload mass

- <u>Orbit Before Catch:</u>
  central facility position L_f = 155 km
  tether tip velocity $V_{t,0}$ = 0.748 km/s
  rotation rate ω_0 = 0.00566 rad/s
  circular orbit altitude $h_{p,0}$ = 170.5 km
- <u>Orbit After Catch</u>:
  perigee altitude $h_{p,0}$ = 178 km,
  apogee altitude $h_{a,0}$ = 411.8 km
  eccentricity e_0 = 0.0575

After catching the payload, the central facility climbs up the tether to the counterbalance mass, changing the rotation rate to:
- adjusted rotation rate ω_0 = 0.00929rad/s
- adjusted tip velocity $V_{t,2}$ = 1.645 km/s

<u>Payload Delivery:</u>
- drop-off altitude h = 1 km
  (top of a lunar mountain)
- velocity w.r.t. surface v = 0 m/s

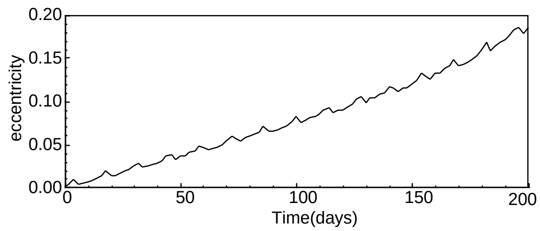

Figure 8. Evolution of the eccentricity of an initially circular 178 km polar lunar orbit, without tether reeling.

Lunavator™ Orbit: Polar vs. Equatorial

In order to provide the most consistent transfer scenarios, it is desirable to place the Lunavator™ into either a polar or equatorial lunar orbit. Each choice has relative advantages and drawbacks, but both are viable options.

Equatorial Lunar Orbit

The primary advantage of an equatorial orbit for the Lunavator™ is that equatorial lunar orbits are relatively stable. An equatorial Lunavator™, however, would only be able to service traffic to bases on the lunar equator. Because the lunar equatorial plane is tilted with respect to the Earth's equatorial plane, a payload boosted by the Earth-orbit tether facility will require a ΔV maneuver to bend its trajectory into the lunar equatorial plane. This ΔV can be provided either using a small rocket thrust or a lunar "slingshot" maneuver. These options will be discussed in more detail in a following section.

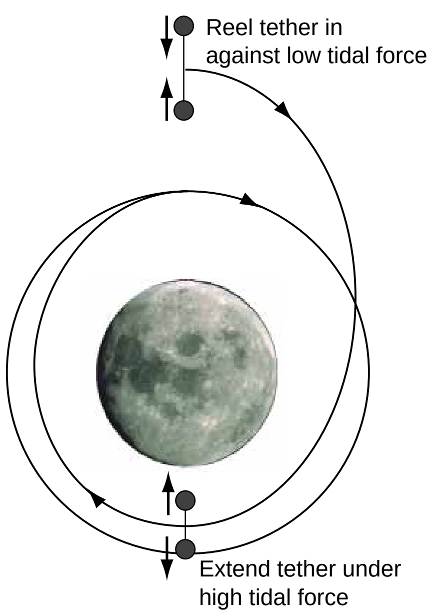

Figure 9. Schematic of tether reeling maneuver to reduce orbital eccentricity.

Polar Lunar Orbit

A polar orbit would be preferable for the Lunavator™ for several reasons. First, direct transfers to polar lunar trajectories are possible with little or no propellant expenditure required. Second, because a polar lunar orbit will remain oriented in the same direction while the Moon rotates inside of it, a polar Lunavator™ could service traffic to any point on the surface of the Moon, including the potentially ice-rich lunar poles. Polar lunar orbits, however, are unstable. The odd-harmonics of the Moon's potential cause a circular, low polar orbit to become eccentric, as illustrated in Figure 8. Eventually, the eccentricity becomes large enough that the perilune is at or below the lunar surface. For the 178 km circular orbit, the rate of eccentricity growth is approximately 0.00088 per day.

Fortunately, the techniques of orbital modification using tether reeling, proposed by Martínez-Sánchez and Gavit[11] and by Landis[13] may provide a means of stabilizing the orbit of the Lunavator™ without requiring expenditure of propellant. Tether reeling can add or remove energy from a tether's orbit by working against the non-linearity of a gravitational field. The basic concept of orbital modification using tether reeling is illustrated in Figure 9. When a tether is near the apoapsis of its orbit, the tidal forces on the tether are low. When it is near periapsis, the tidal forces on the tether are high. If it is desired to reduce the eccentricity of the tether's orbit, then the tether can be reeled in when it is near apoapsis, under low tension, and then allowed to unreel under higher tension when it is at periapsis. Since the tidal forces that cause the tether tension are, to first order, proportional to the inverse radial distance cubed, more energy is dissipated as the tether is unreeled at periapsis than is restored to the tether's orbit when it is reeled back in at apoapsis. Thus, energy is removed from the orbit. Conversely, energy can be added to the orbit by reeling in at periapsis and reeling out at apoapsis. Although energy is removed (or added) to the orbit by the reeling maneuvers, the orbital angular momentum of the orbit does not change. Thus the eccentricity of the orbit can be changed.

The theories developed in references 11 and 13 assumed that the tether is hanging (rotating once per orbit). Because the Lunavator™ will be rotating several times per orbit, we have extended the theory to apply to rapidly rotating tethers.[12] Using a tether reeling scheme in which the tether is reeled in and out once per orbit as shown in Figure 9, we find that a reeling rate of 1 m/s will reduce the eccentricity of the Lunavator™'s orbit by 0.0011 per day, which should be more than enough to counteract the effects of lunar perturbations to the tether's orbit. Thus tether reeling may provide a means of stabilizing the orbit of a polar Lunavator™ without requiring propellant expenditure. This tether reeling, however, would add additional complexity to the system.

Cislunar System Simulations

Tether System Modeling

In order to verify the design of the orbital dynamics of the Cislunar Tether Transport System, we have developed a numerical simulation called "TetherSim" that includes:

- The 3D orbital mechanics of the tethers and payloads in the Earth-Moon system, including the effects of Earth oblateness, using Runge-Kutta integration of Cowell's method.
- Modeling of the dynamical behavior of the tethers, using a bead-and-spring model similar to that developed by Kim and Vadali.[14]
- Modeling of the electrodynamic interaction of the Earth-orbit tether with the ionosphere.

Using this simulation tool, we have developed a scenario for transferring a payload from a circular low-LEO orbit to the surface of the Moon using the tether system designs outlined above. We have found that for an average transfer scenario, mid-course trajectory corrections of approximately 25 m/s are necessary to target the payload into the desired polar lunar trajectory to

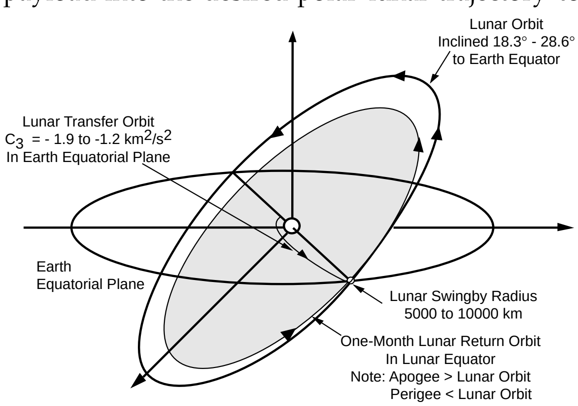

Figure 10. Schematic of one-month "resonance-hop" transfer to place payload in lunar equator without using propellant.

enable rendezvous with the Lunavator™. A simulation of a transfer from LEO to the surface of the Moon can be viewed at *www.tethers.com*.

Targeting the Lunar Transfer

In addition to the modeling conducted with TetherSim, we have also conducted a study of the Earth-Moon transfer to verify that the payload can be targeted to arrive at the Moon in the proper plane to rendezvous with the Lunavator™. This study was performed with the MAESTRO code,[15] which includes the effects of luni-solar perturbations as well as the oblateness of the Earth. In this work we studied targeting to both equatorial and polar lunar trajectories.

Transfer to Equatorial Lunar Trajectories

Transfer of a payload from an equatorial Earth trajectory to an equatorial lunar trajectory can be achieved without propellant expenditure, but this requires use of a one-month "resonance hop" transfer, as illustrated in Figure 10. In a resonance hop maneuver, the payload is sent on a trajectory that passes the Moon in such a way that the lunar gravitational field slingshots the payload's orbit into a one-month Earth orbit that returns to the Moon in the lunar equatorial plane. Using MAESTRO, we have developed a lunar transfer scenario that achieves this maneuver.

In order to avoid the one-month transfer time, we can instead use a small impulsive thrust as the payload crosses the lunar equator to bend its trajectory into the equatorial plane. A patched-conic analysis of such a transfer predicts that such a maneuver would require 98 to 135 m/s of ΔV. However, our numerical simulations of the transfer revealed that under most conditions, luni-solar perturbations of the payload's trajectory will perform much of the needed bending for us, and the velocity impulse needed to place the payload in a lunar equatorial trajectory is only about 25 m/s. Figure 11 shows the time-history of a transfer of a payload from the Earth-orbit tether boost facility to the Moon, projected onto the Earth's equatorial plane.

Figure 12 shows this same transfer, projected onto the lunar equatorial plane in a Moon centered, rotating frame, with the x-axis pointing at the Earth. The motion of the payload relative to the lunar equator can be observed in Figure 13, which shows the trajectory projected onto the lunar x-z plane. The payload crosses the lunar equator approximately 10 hours before its closest

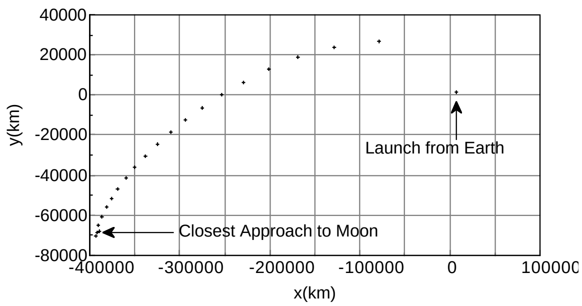

Figure 11. Transfer of payload to lunar equatorial trajectory, projected onto the True Earth Equator.

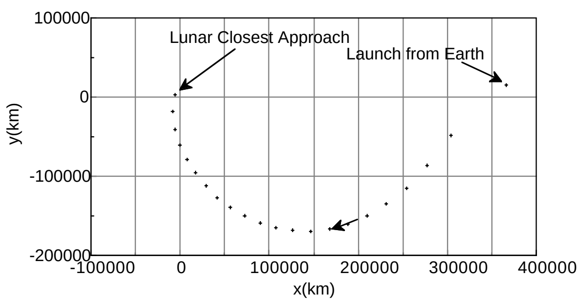

Figure 12. Projection of payload transfer onto Lunar Equatorial Plane (Moon centered frame).

approach to the Moon. Figure 14, which plots the Moon-relative velocity of the payload, shows that the payload's velocity at the time of lunar equatorial crossing is about 925 m/s. However, a plot of the declination of the payload's velocity with respect to the lunar equator, shown in Figure 15, reveals that that the declination of the Moon-relative velocity vector is only a few degrees, much less than the 18°-29° value predicted by a simple zero-patched conic analysis; the Moon's (or Sun's) gravity has bent the velocity vector closer to the lunar orbit plane.

At the time when the payload's trajectory crosses the lunar equator, the declination of the incoming velocity vector is only 1.52°. This dynamical situation permits us to bend the approach trajectory into the lunar equator with a very small amount of impulse supplied by the spacecraft propulsion system. In the case shown here, the amount of ΔV required is only 24.5 m/s, applied about 10 hours before closest approach to the Moon, as the spacecraft crosses the lunar equator.

Transfer to Polar Lunar Trajectories

Figure 16 shows a payload transfer targeted to a polar lunar trajectory with an ascending node (with respect to the lunar prime meridian) of –100.95°. This particular trajectory is a Type II

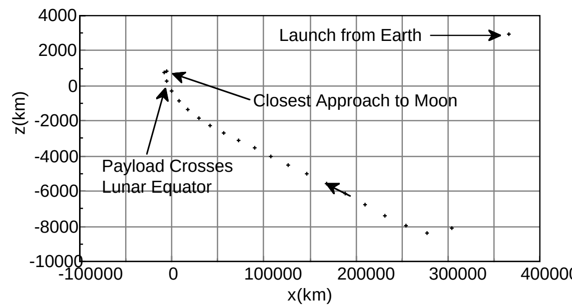

Figure 13. Projection of payload transfer onto Lunar x-z plane (Moon centered frame).

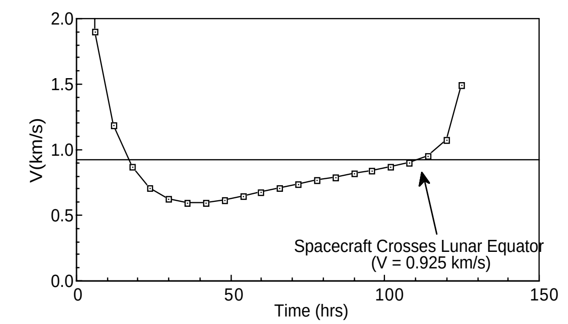

Figure 14. Moon-relative velocity of spacecraft.

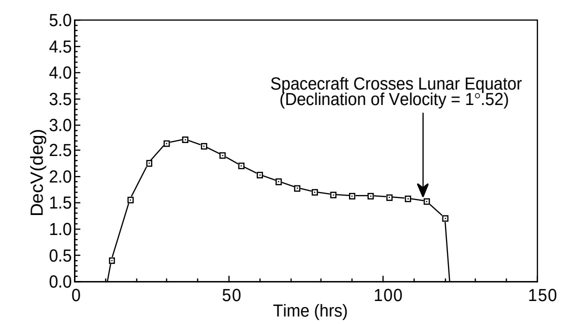

Figure 15. Declination of Moon-relative velocity vector with respect to Lunar Equator.

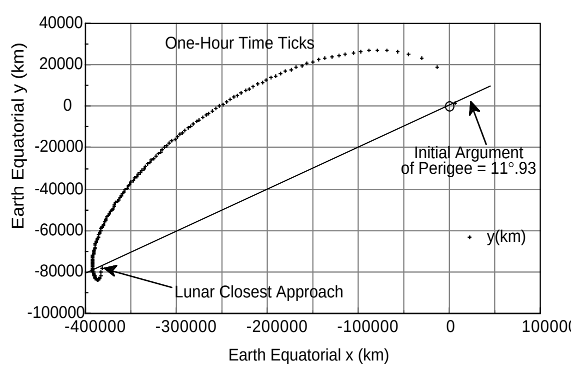

Figure 16. Time history of an Earth-Moon transfer targeted to a polar lunar trajectory.

transfer, with a central angle on the initial orbit of greater than 180°. Similar transfers can be achieved with Type I trajectories (central angle of less than 180°). Essentially, these transfers are achieved by injecting the payload into an orbit that just reaches the Moon's orbit near the point where the Moon will cross the Earth's equatorial plane. When the payload reaches its apogee, it is moving only a few hundred meters per second. As the payload slowly drifts towards its apogee, the Moon approaches, moving at just over 1 km/s. The Moon then "captures" the payload, pulling it into a trajectory that is just barely hyperbolic relative to the Moon.

We have found that by varying the energy of the translunar trajectory and adjusting the argument of perigee, it is possible to target the payload to rendezvous with a polar orbit Lunavator™ with a wide range of ascending node positions of the Lunavator™ orbit. Our simulations indicate that the viable nodal positions ranges at least ±10° from the normal to the Earth-Moon line.

Comparison to Rocket Transport

Travelling from LEO to the surface of the Moon and back requires a total ΔV of more than 10 km/s. To perform this mission using storable chemical rockets, which have an exhaust velocity of roughly 3.5 km/s, the standard rocket equation requires that a rocket system consume a propellant mass equal to 16 times the mass of the payload for each mission. The Cislunar Tether Transport System would require an on-orbit mass of less than 28 times the payload mass, but it would be able to transport many payloads. In practice, the tether system will require some propellant for trajectory corrections and rendezvous maneuvers, but the total ΔV for these maneuvers will likely be less than 100 m/s. Thus a simple comparison of rocket propellant mass to tether system mass indicates that the fully reusable tether transport system could provide significant launch mass savings after only a few round trips. Although the development and deployment costs associated with a tether system would present a larger up-front expense than a rocket based system, for frequent, high-volume round trip traffic to the Moon, a tether system could achieve large reductions in transportation costs by eliminating the need to launch large quantities of propellant into Earth orbit.

Mars-Earth Tether Transport System Architecture

In earlier work,[6] we developed a preliminary version of a LEO-Lunar Tether Transport System in which the Earth-orbit tethers were designed to throw the payload to the Moon on a fast trajectory. This provided short transit times and enabled a rendezvous with a standard Moravec Lunar Skyhook, but, as discussed earlier, it presented a problem in that if the payload failed to rendezvous with the tip of the Lunar Skyhook, the payload would leave the Earth-Moon system on a hyperbolic trajectory. This raised the question of how far a tether in a highly elliptical Earth orbit could throw a payload. A simple energetics-based calculation indicated that the answer was "All the way to Mars." The Mars-Earth Rapid Interplanetary Tether Transport (MERITT) System is the result. In the following sections, we develop a detailed design for a MERITT system architecture.

MERITT System Description

The MERITT system consists of two rapidly rotating tethers in highly elliptical orbits: EarthWhip around Earth and MarsWhip around Mars. A payload capsule is launched from Earth into a low orbit or suborbital trajectory. The payload is picked up by a grapple system on the EarthWhip tether as the tether nears perigee and the tether arm nears the lowest part of its swing. It is tossed later when the tether is still near perigee and the arm is near the highest point of its swing. The payload thus gains both velocity and potential energy at the expense of the tether system, and its resulting velocity is sufficient to send it on a high-speed trajectory to Mars with no onboard propulsion needed except for midcourse guidance.

At Mars, the incoming payload is caught in the vicinity of periapsis by the grapple end of the MarsWhip tether near the highest part of its rotation and greatest velocity with respect to Mars. The payload is released later when the tether is near periapsis and the grapple end is near the lowest part of its swing at a velocity and altitude which will cause the released payload to enter the Martian atmosphere. The system works in both directions.

The MERITT system can give shorter trip times with aerobraking at Mars because the incoming payload velocity is not limited by the

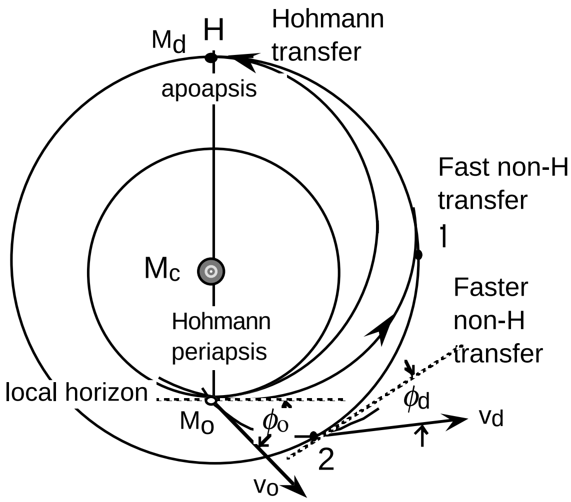

Figure 17. General Orbit Transfer Trajectories.

maximum tether tip velocity and thus payloads can use faster interplanetary trajectories.

In the following subsections we illustrate the general outlines of the system and define the terms used. This initial "feasibility" analysis has not dealt with the many problems of interplanetary phasing and trades. These issues will be addressed in future papers as time and funding allow.

Interplanetary Transfer Orbits

As shown in Figure 17, in the frame of reference of the Sun, acting as the central mass of the whole system, a payload leaves the origin planet, on a conic trajectory with a velocity v_o and flight path angle ϕ_o and crosses the orbit of the destination planet with a velocity v_d and flight path angle ϕ_d. Departure from the origin planet is timed so that the payload arrives at the orbit of the destination body when the destination body is at that point in its orbit. Many possible trajectories satisfy these conditions, creating a trade between trip time and initial velocity.

The classic Hohmann transfer ellipse (H) is a bounding condition with the least initial velocity and longest trip time. The Hohmann transfer is tangential to both the departure and destination orbits and the transfer orbits. The direction of the velocity vector is the same in both orbits at these "transfer" points and only differs in magnitude. A ΔV change in payload velocity (usually supplied by onboard propulsion) is required at these points for the payload to switch from one trajectory to another.

Faster non-Hohmann transfers may be tangential at origin, destination, or neither. They may be elliptical or hyperbolic. For a given injection velocity above the Hohmann minimum constraint, the minimum-time transfer orbit is generally non-tangential at both ends. An extensive discussion of the general orbit transfer problem may be found in Bate, Mueller and White.[16]

For reasons discussed below, using tethers in an elliptical orbit with a fixed tip velocity to propel payloads results in an injection velocity constrained to the vector sum of a hyperbolic excess velocity of the released payload and the orbital velocity of the origin planet. When a

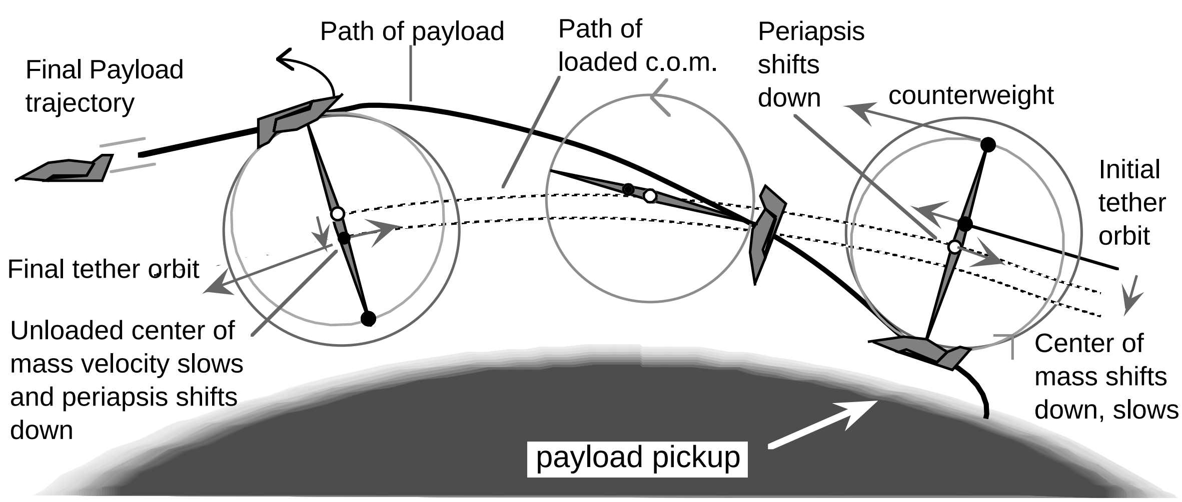

Figure 18. General geometry of tether pickup and throw orbital injection.

tether only is used to receive the payload, a similar constraint exists on the destination end; the incoming trajectory is a hyperbola and the periapsis velocity of the hyperbolic orbit must not exceed what the tether can handle. This periapsis velocity is determined by the vector sum of the orbital velocity of the destination planet, that of the intersecting payload orbit at the intersection, and the fall through the gravitational field of the destination planet.

When passage through the atmosphere of the destination planet (aerobraking) is used to remove some of the incoming velocity, the constraint becomes an engineering issue of how much velocity can be lost in the atmospheric passage. Experience with the Apollo mission returns (circa 12 km/s) and the Mars Pathfinder landing indicates that with proper design, much more velocity can be dissipated than is required to assist tether capture.

Real passages through space take place in three dimensions. To the first order, however, transfer orbits are constrained to a plane incorporating the Sun, the origin planet at launch and the destination planet at arrival. The injection vector must occur in this plane, or close enough to it that on-board payload propulsion can compensate for any differences. This analysis considers only coplanar trajectories, but, as discussed later, this is not a great handicap.

As the payload moves out from the influence of the mass of the origin planet, its trajectory becomes more and more influenced by the mass of the Sun, until the origin planet mass can be essentially neglected. Likewise, inbound payloads become more and more influenced by the destination planet mass until the mass of the Sun may be neglected. For first order Keplerian analysis it is customary to treat the change of influence as if it occurred at a single point, called the patch point. At this point, a coordinate transformation is made.

Payload Pickup and Injection

Figure 18 shows the general geometry of a tether picking up a payload from a suborbital trajectory at a point just outside the atmosphere of the origin planet and injecting it into an interplanetary transit trajectory. The payload is picked up, swung around the tether's center of mass along the circle as it moves along its orbit, and is released from the tip of the tether near the top of the circle. In the process, the tether center of mass loses both altitude and velocity, representing the loss of energy by the tether to the payload. This energy loss may be made up later by propulsion at the tether center and/or in the reverse process of catching incoming payloads.

Around the time of pick-up, the trajectory of the payload must be of equal velocity and should be very nearly tangential (no radial motion) to the circle of motion of the tether tip in the tether frame of reference. This tangential condition increases the time for a docking maneuver to be consummated. It is easy to see how this condition may be satisfied by rendezvous at the mutual apsides of the tether orbit and the payload pickup orbit, but other, more complex trajectories work as well. It is not a requirement, however, that the tether plane of rotation, the tether orbit, and the payload pickup orbit be coplanar. The mutual velocity vector at pick-up is essentially a straight line, and an infinite number of curves may be tangent to that line. The tether rendezvous acts as a kind of patch point, as the plane of the tether's rotation becomes dominant. The practical effect of this is to allow considerable leeway in rendezvous conditions. It also means that the kind of two dimensional analysis presented here has a wide range of validity.

Capturing of an incoming payload is essentially the time reversal of the outgoing scenario; the best place to add hyperbolic excess velocity is also the best place to subtract it. If the tether orbital period is an integral multiple of the rotation period following release of a payload, the tip will be pointed at the zenith at periapsis and the capture will be the mirror image of the release.

Capturing a payload after a pass through the destination body's atmosphere is more complex than a periapsis capture, but involves the same principle: matching the flight path angle of the payload exiting trajectory to the tether flight path angle at the moment of capture and the velocity to the vector sum of the tether velocity and tip velocity. Aerodynamic lift and energy management during the passage through the atmosphere provide propellant-free opportunities to accomplish this.

There is a trade in aerobraking capture between momentum gain by the capturing tether and mission redundancy. To make up for momentum loss from outgoing payloads, the tether would like to capture incoming payloads at similar velocities. That, however, involves hyperbolic trajectories in which, if the payload is not captured, it is lost in space. Also, in the early operations before extensive ballast mass is accumulated, care must be taken that the tether itself is not accelerated to hyperbolic velocities as a result of the momentum exchange.

Payload Release

The release orbit is tangential to the tether circle in the tether frame of reference by definition, but it is not necessarily tangential to the trajectory in the frame of reference of the origin planet. The injection velocity vector is simply the vector sum of the motion of the tether tip and the tether center, displaced to the location of the tether tip. Note in the third part of Figure 18 that this does not generally lie along the radius to the tether center of mass. For maximum velocity, if one picks up the payload at tether periapsis, one must wait for the tether to swing the payload around to a point where its tip velocity vector is near parallel to the tether center of mass orbital velocity vector. By this time, the tether has moved significantly beyond periapsis, and there will be a significant flight path angle, which both orbits will share at the instant of release. Large variations from this scenario will result in significant velocity losses, but velocity management in this manner could prove useful. If, on the other hand, maximum velocity transfer and minimum tether orbit periapsis rotation is desired, the payload can be retained and the tether arm length or period adjusted to release the payload in a purely azimuthal direction at the next periapsis.

Rendezvous of Grapple with Payload

The seemingly difficult problem of achieving rendezvous of the tether tip and payload is nearly identical to a similar problem solved daily by human beings at circuses around the world. The grapple mechanism on the end of a rotating tether is typically subjected to a centrifugal acceleration of one gee by the rotation of the tether. Although the grapple velocity vector direction is changing rapidly, its speed is constant and chosen to be the same speed as the payload, which is moving at nearly constant velocity in its separate free fall suborbital trajectory. The timing of the positions of the tether tip and the payload needs to be such that they are close to the same place (within a few meters) at close to the same time (within a few seconds), so their relative spacing and velocities are such that the grapple can compensate for any differences. This situation is nearly identical to the problem of two trapeze artists timing the swings of their separate trapeze bars so that that the "catcher," being supported in the 1 gee gravity field of the Earth by his bar, meets up with and grasps the "payload" after she has let go of her bar and is in a "free fall" trajectory accelerating with respect to the "catcher" at one gee. They time their swings, of course, so that they meet near the instant when both are at near zero relative velocity. The tether grapple system will have the advantages over the human grapple system of GPS guidance, radar Doppler and proximity sensors, onboard divert thrusters, electronic synapses and metallic grapples, which should insure that its catching performance is comparable to or better than the demonstrated human performance.

An essential first step in the development of the MERITT system would be the construction and flight test of a rotating tether-grapple system in LEO, having it demonstrate that it can accurately toss a dummy payload into a carefully selected orbit such that n orbits later the two meet again under conditions that will allow the grapple to catch the payload once again.

The Automated Rendezvous and Capture (AR&C) Project Office at Marshal Space Flight Center (MFSC) has been briefed on the AR&C requirements for the capture of a payload by a grapple vehicle at the end of a tether with a one-gee acceleration tip environment. MSFC has been working AR&C for over six years and has a great deal of experience in this area. It is their opinion [14] that their present Shuttle-tested [STS-87 & STS-95] Video Guidance Sensor (VGS) hardware, and Guidance, Global Positioning System (GPS) Relative Navigation, and Guidance, Navigation and Control (GN&C) software, should, with sufficient funding, be able to be modified for this tether application.

Tether Considerations

For a tether transport system to be economically advantageous, it must be capable of

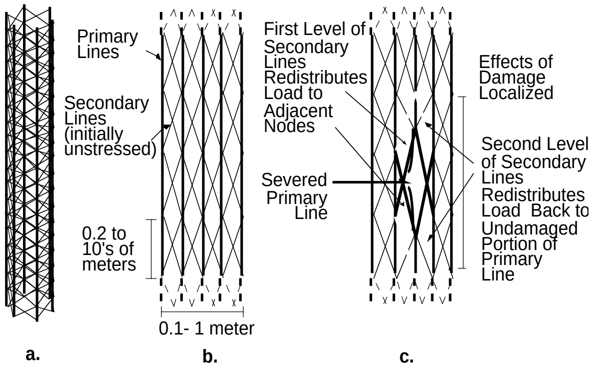

Figure 19. The Hoytether™ design and its response to a cut line.

handling frequent traffic for many years despite degradation due to impacts by meteorites and space debris. Fortunately, a survivable tether design exists, called the Hoytether™, which can balance the requirements of low weight and long life.[17] As shown in Figure 19, the Hoytether™ is an open net structure where the primary load bearing lines are interlinked by redundant secondary lines. The secondary lines are designed to be slack initially, so that the structure will not collapse under load. If a primary line breaks, however, the secondary lines become engaged and take up the load.

Note that four secondary line segments replace each cut primary line segment, so that their cross-sectional area need only be 0.25 of the primary line area to carry the same load. Typically, however, the secondary lines are chosen to have a cross-sectional area of 0.4 to 0.5 of the primary line area, so as to better cope with multiple primary and secondary line cuts in the same region of the tether.

This redundant linkage enables the structure to redistribute loads around primary segments that fail due to meteorite strikes or material failure. Consequently, the Hoytether™ structure can be loaded at high stress levels, yet retain a high margin of safety.[18]

Tether Mass Ratios

Assuming that the grapple on the end of the tether masses 20% of the payload mass, we can use Eqn. (1) to calculate the mass ratio of a one arm Spectra® 2000 Hoytether™ to the payload it is handling, assuming various different safety factors and various different tether tip velocities, to be:

Table 3:
Ratio of Spectra™ 2000 Tether Material Mass to Payload Mass (Grapple Mass 20% of Payload Mass)

	Tether Material Safety Factor (F)			
Tip Speed V_T	1.75	2.0	2.4	3.0
1.5 km/s	2.2	2.5	3.4	4.9
2.0 km/s	3.7	4.7	6.4	10.0
2.5 km/s	8.0	11.0	17.0	30.0

From this table we can see that by using Spectra™ 2000, we can achieve tether tip velocities of 2.0 km/s with reasonable tether mass ratios (<10) and good safety factors. Higher tip velocities than 2.0 km/s are achievable using higher mass ratios, lower safety factors, and stronger materials.

Tether Survivability

There are many objects in Earth space, ranging from micrometeorites to operational spacecraft with 10 meter wide solar electric arrays. We can design interconnected multiple strand open net Hoytether™ structures that can reliably (>99.9%) survive in space for decades despite impacts by objects up to 30 cm (1 foot) or so in size.

Objects larger than 30 cm will impact all the strands at one time, cutting the tether. These large objects could include operational spacecraft, which would also be damaged by the impact. Objects larger than 30 cm are all known and tracked by the U.S. Space Command. There are about 6000 such objects in low and medium Earth orbit, of which an estimated 600 will be operational spacecraft in the 2005 time frame.

Depending upon the choice of the EarthWhip orbit, calculations show that there is a small (<1%) but finite chance of the EarthWhip tether striking one of the 600 operational spacecraft. It will therefore be incumbent on the tether system fabricators and operators to produce EarthWhip tether systems that maintain an accurate inventory of the known large objects and control the tether system center of mass orbital altitude and phase, the tether rotation rate and phase, and the tether libration and vibration amplitudes and phases, to insure that the tether system components do not penetrate a volume of "protected space" around these orbiting objects.

MERITT Modeling

Calculations of the MERITT system performance were performed using the mathematical modeling software package "TK Solver" which allows the user to type in the relevant equations and get results without having to solve the model algebraically or structure it as a procedure, as long as the number of independent relationships equals the number of variables. This is very useful in a complex system when one may wish to constrain various variables for which it would be difficult, if not impossible, to solve and to perform numerical experiments to investigate the behavior of the system.

Two versions of a tether based interplanetary transfer system are being worked on, one for tether-only transfers and the other incorporating an aerobraking pass at the destination body to aid in capture and rotation of the line of apsides. It should be emphasized that the results presented here are very preliminary and much remains to be done with the software. Because of the ongoing work and the growing number of variables and lines of code, we will not try to go through this line by line here. Questions concerning the code should be referred to Gerald Nordley at the above address.

The general architecture of the models is sequential. A payload is picked up from a trajectory at the origin planet, and added to a rotating tether in a highly elliptical orbit around the origin planet. The pickup is accomplished by matching the position and velocity of the grapple end of the unloaded rotating tether to payload position and velocity.

This addition of the payload mass to one end of the tether shifts the center of mass of the tether toward the payload. The tether used in these examples is modeled as a rigid line with two arms, a grapple, a counterweight and a central mass. The tether is assumed to be designed for a payload with a given mass and a "safety factor" of two, as described in Hoyt and Forward[18] and to be dynamically symmetrical with a payload of that mass attached.

The mass distribution in the arms of the tether was determined by dividing the tether into ten segments, each massive enough to support the mass outward from its center; this was not needed for the loaded symmetric tether cases presented here, but will be useful in dealing with asymmetric counterweighted tethers. The total mass of each tether arm was determined from Eqn. (1). The continuously tapered mass defined by Eqn. (1) was found to differ by only a few percent from the summed segment mass of the 10 segment tether model used in the analysis, and the segment masses were adjusted accordingly until the summed mass fit the equation. The small size of this adjustment, incidentally, can be taken as independent confirmation of Eqn. (1).

We ended up designing many candidates for the EarthWhip and MarsWhip tethers, from some with very large central station masses that were almost unaffected by the pickup or toss of a payload, to those that were so light that the toss of an outgoing payload caused their orbits to shift enough that the tether tip hit the planetary atmospheres, or the catch of an incoming payload sent the tether (and payload) into an escape trajectory from the planet. After many trials, we found some examples of tethers that were massive enough that they could toss and catch payloads without shifting into undesirable orbits, but didn't mass too much more than the payloads they could handle. The tethers are assumed to be made of Spectra™ 2000 material braided into a Hoytube™ structure with a safety factor of 2. The tether design consists of a large central station with a solar array power supply, winches, and control systems, plus any ballast mass needed to bring the mass of the total system up to the desired final mass value. From the tether central station is extended two similar tethers, with a taper and mass determined by Eqn. (1) according to the loaded tip velocity desired. At the end of the tethers are grapples that each mass 20% of the payloads to be handled. To simplify this initial analysis, we assumed that one grapple is holding a dummy payload with a mass equal to the active payload, so that after the grapple on the active arm captures a payload, the tether system is symmetrically balanced. Later, more complex, analyses will probably determine that a one arm tether system will do the job equally well and cost less.

Shift in Tether Center of Mass

The shift of the center of mass of the tether system when a payload was attached or released was determined by adding the moments of the unloaded tether about the loaded center of symmetry and dividing by the unloaded mass.

Figure 20 illustrates the four general circumstances of tether operations: origin pickup, origin release, destination capture and destination release. The shift of the center of mass of the tether system when a payload was attached or released was determined by adding the moments of the unloaded tether about the loaded center of symmetry and dividing by the unloaded mass. Figure 20 illustrates the four general circumstances of tether operations; origin pickup, origin release, destination capture and destination release. It turns out that the dynamics of an ideal rigid tether system with a given payload can be fairly well modeled by simply accounting for the change in the position and motion of the tether's center of mass as the payload is caught and released.

When the payload is caught, the center of mass shifts toward the payload and the tether assumes a symmetrical state. The velocity of the tip around the loaded center of mass is simply its velocity around the unloaded center of mass minus the velocity of the point which became the new center of mass about the old center of mass. The change in the tether orbital vector is fully described by the sum of the vector of the old center of mass and the vector at the time of capture or release of the point that becomes the new center of mass relative to the old center of mass. Since the tether loses altitude with both the catch and the throw, its initial altitude must be high enough so that it does not enter the atmosphere after it throws the payload.

Once the payload is released, its velocity and position are converted to Keplerian orbital elements which are propagated to the outgoing patch point. At this point, they are converted back to position and velocity, and transformed to the Sun frame of reference.

The velocity of insertion into the orbit in the Sun's frame of reference is essentially the vector sum of the hyperbolic excess velocity with respect to the origin planet and the

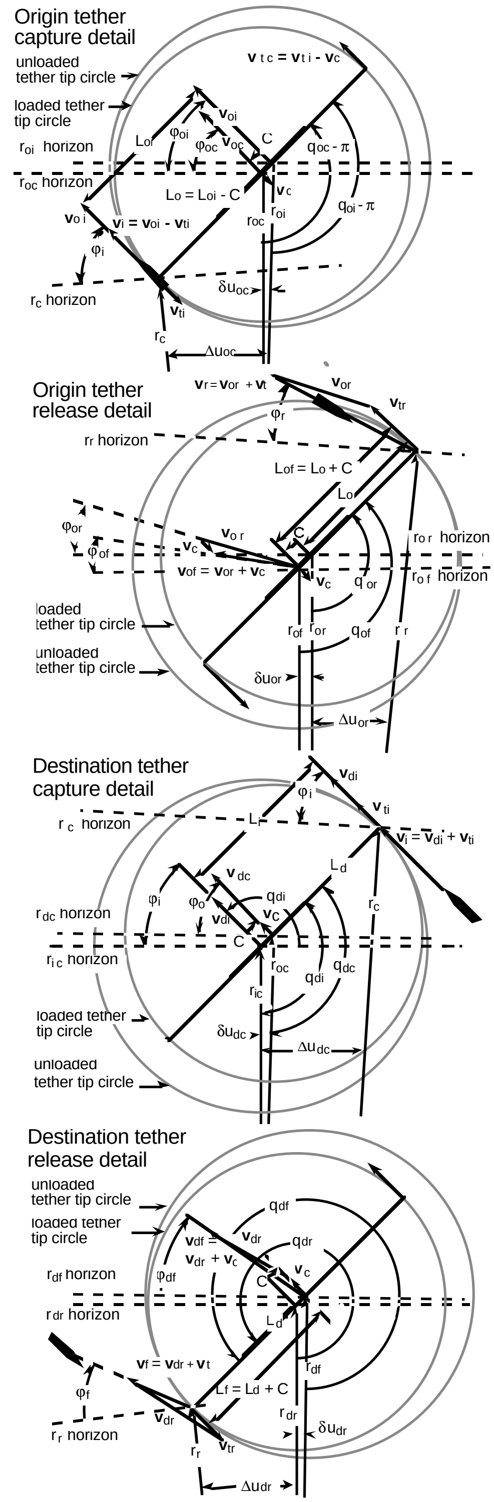

Figure 20. Tether Capture/Release Operations.

origin planet's orbital velocity about the Sun. This vector is done in polar coordinates, and the angle portion of this vector in the origin planet frame is, at this point, a free choice. For now, an estimate or "guess" of this quantity is made. The resulting vector is then converted into Sun frame orbital elements and propagated to the patch point near the orbit of the destination planet. There, it is transformed into the destination planet coordinates.

Tether-Only Incoming Payload Capture

For the tether-only capture scenario, the velocity and radius of the tip of the tether orbiting the destination mass are calculated and iteratively matched to the velocity of the payload on an orbit approaching the destination planet, as shown in Figure 21.

The distance of the patch point and the relative velocity there provide the energy of the orbit. The radius and velocity of the tether tip provide another pair of numbers and this is sufficient to define an approach orbit when they match. There are a large number of free parameters in this situation with respect to the tether orbit which can be varied to produce a capture. There is a good news/bad news aspect to this. The difficulty is that the problem is not self- defined and to make the model work, some arbitrary choices must be made. The good news is that this means there is a fair amount of operational flexibility in the problem and various criteria can be favored and trades made.

In this work, we have generally tried to select near-resonant tether orbits that might be "tied" to geopotential features so that they precess at the local solar rate and thus maintain their apsidal orientation with respect to the planet-Sun line. The Russian Molniya communications satellites about Earth and the Mars Global Surveyor spacecraft use such orbits.

The Sun-referenced arguments of periapsis, ω, in the figures are technically not constants, but can be treated as such for short spans of time when apsidal precession nearly cancels the angular rate of the planet's orbit about the Sun .

The fastest transfer times are generally associated with the fastest usable periapsis velocities. These are found when the tether is at

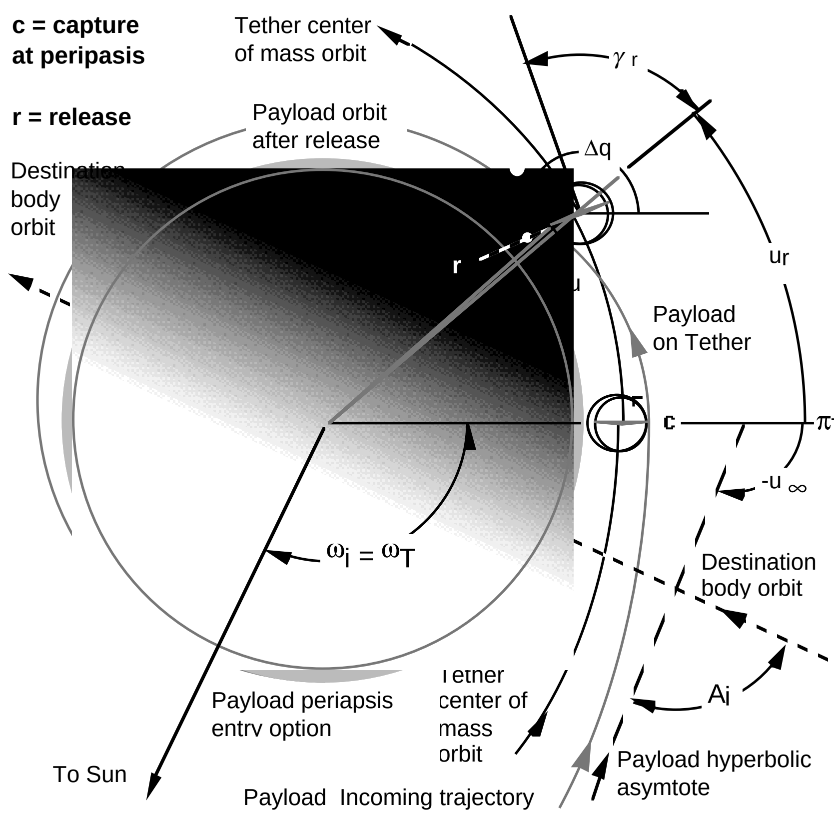

Figure 21. Tether-Only Capture Scenario.

periapsis and its tip at the zenith of its swing. In one approach to this model, these tether conditions are used to set the periapsis velocity and radius of the incoming orbit. This, in turn, defines the relative velocity at the patch point, and the origin planet injection angle can be iterated to produce a Sun frame orbit that produces that relative velocity at the destination planet patch point.

Aerobraking Payload Capture

In the case of using aerobraking in the planetary atmosphere, the injection angle can be optimized for minimum transfer time. As shown in Figure 22, the radius at which the atmosphere of the destination planet is dense enough to sustain an aerodynamic trajectory is used to define the periapsis of the approach orbit; there is no velocity limit.

In a similar manner, the tether tip at an estimated capture position and velocity, together with the radius at which the outgoing payload resumes a ballistic trajectory define an exit orbit which results in tether capture. The difference in the periapsis velocity of this orbit and the periapsis velocity of the inbound trajectory is the velocity that must be dissipated during the aerodynamic maneuver. For Mars bound trajectories, this aerobraking ΔV is on the order of 5 km/s, as compared to direct descent ΔV's of 9 km to 15 km/s. Also, payloads meant to be released into suborbital trajectories already carry heat shields, though designed for lower initial velocities.

After the tether tip and the incoming payload are iteratively matched in time, position and velocity, the center of mass orbit of the loaded tether is propagated to the release point. This is another free choice, and the position of the tether arm at release determines both the resulting payload and tether orbit. In this preliminary study, care was taken to ensure that the released payload did enter the planet's atmosphere, the tether tip did not, and that the tether was not boosted into an escape orbit.

Initial Planet Whip Analysis

We first carried out analyses of a number of MERITT missions using a wide range of assumptions for the tether tip speed and whether or not aerobraking was used. The trip times for the various scenarios are shown in Table 4. As can be seen from Table 4, the system has significant

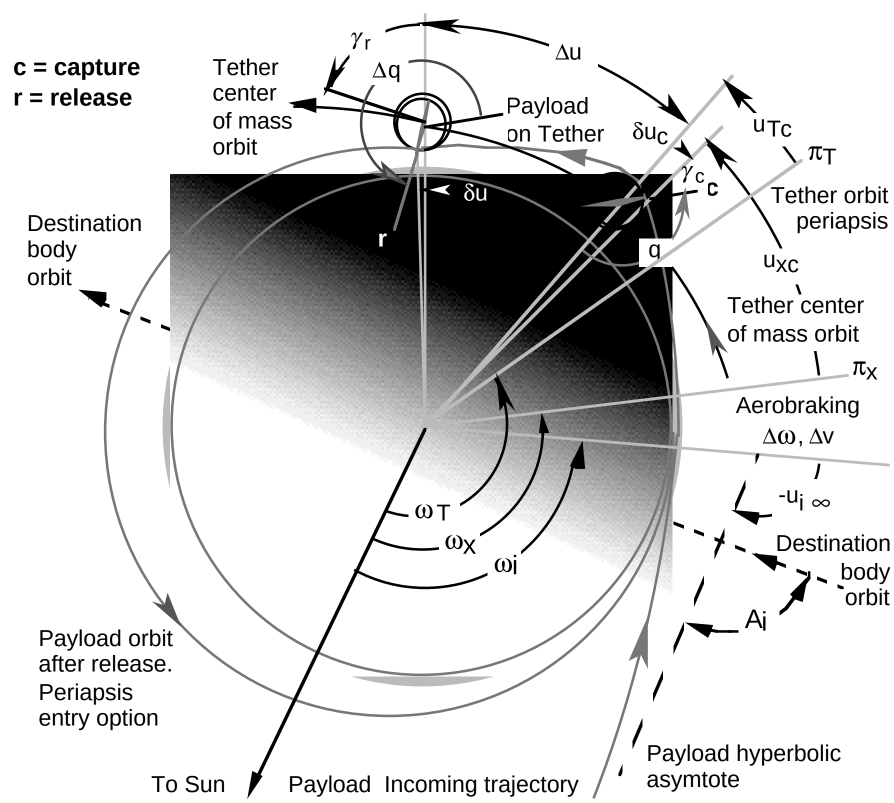

Figure 22. Aerobraking Tether Capture.

growth potential. If more massive tethers are used, or stronger materials become available, the tether tip speeds can be increased, cutting the transit time even further. The transit times in Table 4 give the number of days from payload pickup at one planet until payload reentry at the other planet, and include tether "hang time" and coast of the payload between the patch points and the planets. Faster transit times can be made with higher energy initial orbits for the payload and the tether. With a 2.5 km/s tip speed on the PlanetWhip tethers and using aerobraking at Mars (see Figure 22), the Earth orbit-Mars orbit transit time can be made about 94 days.

PlanetWhip Analysis

The periapsis of the tether orbit is pushed counterclockwise for where a tether-only capture would occur by the angular distance needed for aerobraking and the periapsis rotations caused by capturing and releasing the payload at non-zero true anomalies. If the periapsis is shifted enough, the tether may be able to inject a payload on a return trajectory without waiting for many months, or using substantial amounts of propellant to produce the needed alignment.

Detailed MERITT Example

There are a large number of variables in the MERITT system concept, and many of those variables can be freely chosen at the start of the system design. We have carried out dozens of complete round-trip scenarios under various different assumptions, such as: aerobraking before tether catch versus direct tether-to-tether catch; sub-, circular, and elliptical initial and final payload orbits; 1.5, 2.0, 2.5 and higher tether tip velocities; large, small and minimum tether central facility masses; etc. We will present here just one of the many possible MERITT scenarios using finite mass EarthWhip and MarsWhip tethers, but do it in extensive detail so the reader can understand where the broad assumptions are, while at the same time appreciating the accuracy of the simulations between the broad assumptions. In most cases, the matches between the payload trajectories and the tether tip trajectories are accurate to 3 and 4 decimal places.

Table 4.
Potential MERITT Interplanetary Transfer Times

Tip Speed (km/s)	System Mass Ratio	Transfer direction From->To	Tether-only (days)	Aero-braking (days)
1.5	15x	Earth->Mars	188	162
		Mars->Earth	187	168
2.0	15x	Earth->Mars	155	116
		Mars->Earth	155	137
2.5	30x	Earth->Mars	133	94
		Mars->Earth	142	126

Figure 23 is a diagram showing how a single tether toss and catch system would work on either the Earth or Mars end of the MERITT system, for a finite mass PlanetWhip tether. The incoming payload brushes the upper atmosphere of the planet, slows a little using aerobraking, and is caught by a rotating tether in a low energy elliptical orbit. After the payload is caught, the center of mass of the tether shifts and the effective length of the tether from center of mass to the payload catching tip is shortened, which is the reason for the two different radii circles for the rotating tether in the diagram. The orbit of the tether center of mass changes from a low energy elliptical orbit to a higher energy elliptical orbit with its periapsis shifted with respect to the initial orbit. The tether orbit would thus oscillate between two states: 1) a low energy state wherein it would be prepared to absorb the energy from an incoming payload without becoming hyperbolic and 2) a high energy state for tossing an outgoing payload.

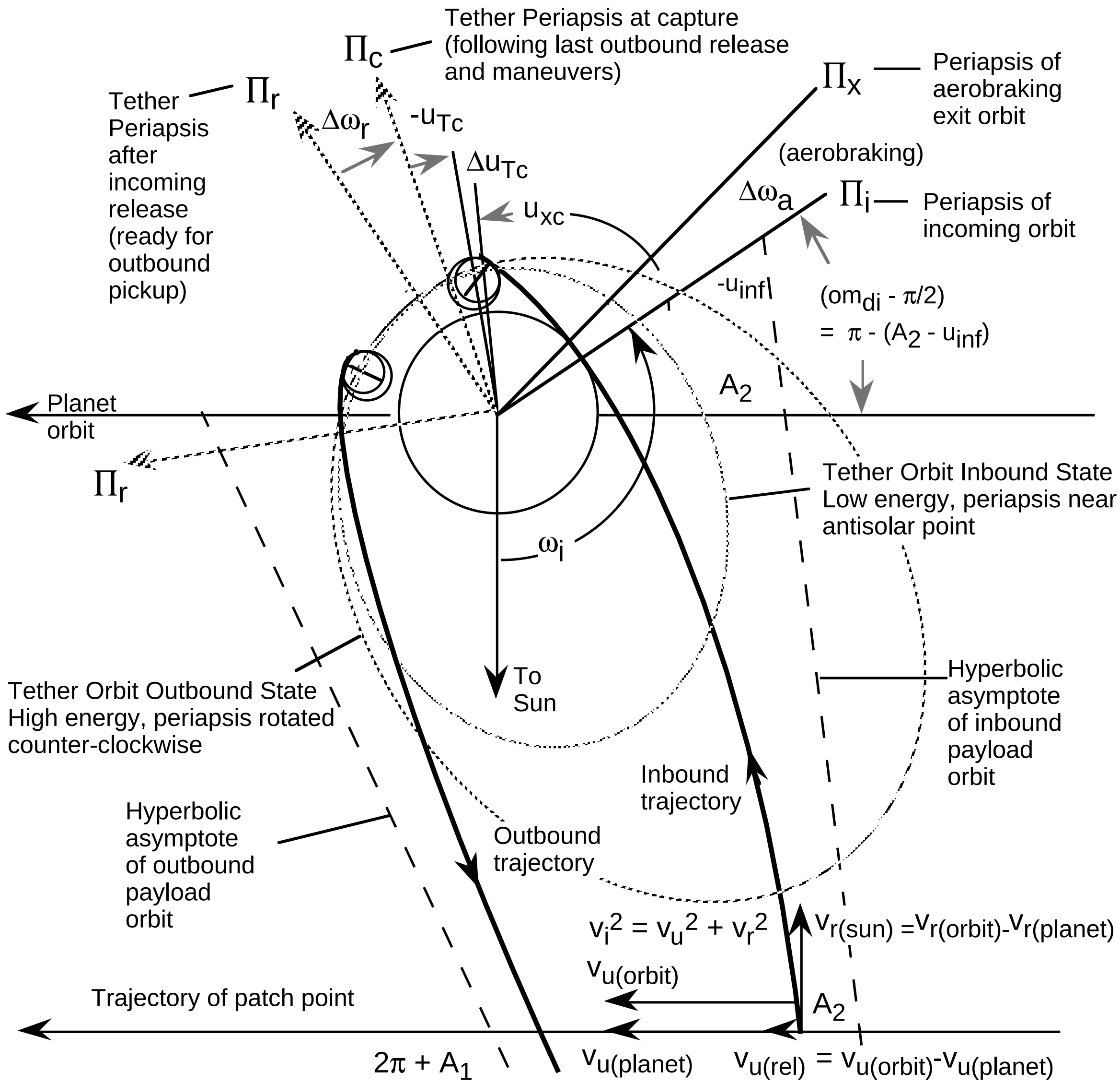

Figure 23. "Planet"Whip showing catch and toss states using aerobraking.

The scenario we will describe uses EarthWhip and MarsWhip tethers of near minimum mass made of Spectra™ 2000 with a tip speed of 2.0 km/s. Because they have small total masses, the toss and catch operations significantly affect the tether rotation speed, center of mass, and orbital parameters, all of which are taken into account in the simulation. The payload is assumed to be initially launched from Earth into a suborbital trajectory to demonstrate to the reader that the MERITT system has the capability to supply all of the energy and momentum needed to move the payload from the upper atmosphere of the Earth to the upper atmosphere of Mars and back again. We don't have ask the payload to climb to nearly Earth escape before the MERITT system takes over.

In practice, it would probably be wise to have the payload start off in an initial low circular orbit. The energy needed to put the payload into a low circular orbit is not that much greater than the energy needed to put the payload into a suborbital trajectory with an apogee just outside the Earth's atmosphere. The circular orbit option also has the advantage that there would be plenty of time to adjust the payload orbit to remove launch errors before the arrival of the EarthWhip tether.

In the example scenario, the payload, in its suborbital trajectory, is picked up by the EarthWhip tether and tossed from Earth to Mars. At Mars it is caught by the MarsWhip tether without the use of aerobraking, and put into a trajectory that enters the Martian atmosphere at low velocity. Since this scenario does not use aerobraking, the return scenario is just the reverse of the outgoing scenario.

Payload Mass

We have chosen a canonical mass for the payload of 1000 kg. If a larger payload mass is desired, the masses of the tethers scale proportionately. The scenario assumes that the payload is passive during the catch and throw operations. In practice, it might make sense for the payload to have some divert rocket propulsion capability to assist the grapple during the catch operations. In any case, the payload will need some divert rocket propulsion capability to be used at the midpoint of the transfer trajectory to correct for injection errors.

Tether Mass

Both the EarthWhip and MarsWhip tethers were assumed to consist of a robotic central station, two similar tethers, two grapples at the ends of the two tethers, and, to make the analysis simpler, one grapple would be holding a dummy payload so that when the active payload is caught, the tether would be symmetrically balanced.

The tether central station would consist of a solar electric power supply, tether winches, and command and control electronics. There may be no need to use center of mass rocket propulsion for ordinary tether operations. Both tethers can be adequately controlled in both their rotational parameters and center-of-mass orbital parameters by "gravity-gradient" propulsion forces and torques generated by changing the tether length at appropriate times in the tether orbit. [11,13,]

The EarthWhip tether would also have a small conductive portion of the tether that would use electrodynamic tether propulsion,[18] where electrical current pumped through the tether pushes against the magnetic field of the Earth to add or subtract both energy and angular momentum from the EarthWhip orbital dynamics, thus ultimately maintaining the total energy and angular momentum of the entire MERITT system against losses without the use of propellant.

The grapple mechanisms are assumed in this scenario to mass 20% of the mass of the payload, or 200 kg for a 1000 kg payload. It is expected, however, that the grapple mass will not grow proportionately as the payload mass increases to the many tens of tons needed for crewed Mars missions.

In the scenario presented here, it is assumed that the grapples remain at the ends of the tethers during the rendezvous procedure. In practice, the grapples will contain their own tether winches powered by storage batteries, plus some form of propulsion.

As the time for capture approaches, the grapple, under centrifugal repulsion from the rotation of the tether, will release its tether winches, activate its propulsion system, and fly ahead to the rendezvous point. It will then reel in tether as needed to counteract planetary gravity forces in order to "hover" along the rendezvous trajectory, while the divert thrusters match velocities with the approaching payload. In this manner, the rendezvous interval can be stretched to many tens of seconds.

If needed, the rendezvous interval can be extended past the time when the tip of the tether passes through the rendezvous point by having the grapple let out tether again, while using the divert thrusters to complete the payload capture. The grapple batteries can be recharged between missions by the grapple winch motor/dynamos, by allowing the grapple winches to reel out while the central winches are being reeled in using the central station power supply. The grapple rocket propellant will have to be resupplied either by bringing up "refueling" payloads or extracting residual fuel from payloads about to be deorbited into a planetary atmosphere.

For this scenario, we assumed that, when loaded with a payload, the EarthWhip and MarsWhip tethers were rotating with a tether tip speed of V_T = 2,000 m/s. The length of each tether arm was chosen as L=400 km in order to keep the acceleration on the payload, $G=V_T^2/L$, near one gee. We also assumed that the total mass of the Whips are 15,000 kg for a 1000 kg payload (16,000 kg total). This mass includes the central station, both tethers, the grapples at the ends of the tethers, and the dummy payload mass. This is about the minimum tether mass needed in order for the tether center-of-mass orbits to remain stable before and after a catch of a payload with a velocity difference of 2000 m/s.

The tether material was assumed to be Spectra™ 2000 with an ultimate tensile strength of U=4.0 GPa, a density d=970 kg/m^3, and an ultimate tip velocity for an untapered tether of $V_U=(2U/d)^{1/2}$ =2872 m/s. The tether safety factor was initially chosen at F=2.0, which results in a engineering characteristic velocity for the tether of $V_C = (2U/2d)^{1/2}$ = 2031 m/s.

Using V_C and V_T in Eqn. (1), we find that the mass ratio of one arm of a tapered Spectra™ 2000 tether is 3.841 times the mass at the tip of the tether. Since the mass at the end of the tether consists of the 1000 kg payload and the 200 kg grapple, the minimum total mass of one tether arm is 4609 kg, or about 4.6 times the mass of the 1000 kg payload. The amount of taper is significant, but not large. The total cross-sectional area of the tether at the tip, where it is holding onto the payload, is 6 mm^2 or 2.8 mm in diameter, while the area at the base, near the station, is 17.3 mm^2 or 4.7 mm in diameter. This total cross-sectional area will be divided up by the Hoytether™ design into a large number of finer cables.

Eqn. (1), however, applies to a rotating tether far from a massive body. Since the EarthWhip and MarsWhip tethers are under the most stress near periapsis, when they are closest to their respective planets, we need to take into account the small additional stress induced by the gravity gradient forces of the planets, which raises the mass to about 4750 kg for a 1000 kg payload. We will round this up to 4800 kg for the tether material alone, corresponding to a free-space safety factor of 2.04, so that the total mass of the tether plus grapple is an even 5000 kg. With each tether arm massing 5000 kg including grapple, one arm holding a dummy payload of 1000 kg, and a total mass of 15,000 kg, the mass of the central station comes out at 4000 kg, which is a reasonable mass for its functions.

There are a large number of tether parameter variations that would work equally well, including shorter tethers with higher gee loads on the payloads, and more massive tethers with higher safety factors. All of these parameters will improve as stronger materials become commercially available, but the important thing to keep in mind is that the numbers used for the tethers assume the use of Spectra™ 2000, a commercial material sold in tonnage quantities as fishing nets, fishing line (SpiderWire), and kite line (LaserPro). We don't need to invoke magic materials to go to Mars using tethers.

Tether Rotational Parameters

When the EarthWhip or MarsWhip tethers are holding onto a payload, they are symmetrically balanced. The center-of-mass of the tether is at the center-of-mass of the tether central station. The effective arm length from the tether center-of-mass to the payload is 400,000 m, the tip speed is exactly 2000 m/s and the rotation period is P = 1256.64 s = 20.94 min = 0.3491 hr.

When the Whips are not holding onto a payload, then the center-of-mass of the Whip shifts 26,667 m toward the dummy mass tether arm, and the effective length of the active tether arm becomes 426,667 m, while the effective tip velocity at the end of this longer arm becomes 2,133 m/s. (Since there is no longer a payload on this arm, the higher tip velocity can easily be handled by the tether material.) The rotational period in this state is the same, 1256.64 s.

Payload Trajectory Parameters

The Earth-launched payload trajectory chosen for this example scenario is a suborbital trajectory with an apogee altitude of 203,333 m (6581.333 km radius) and a apogee velocity of 7,568 m/s. The circular orbit velocity for that radius is 7,782 m/s.

EarthWhip Before Payload Pickup

The EarthWhip starts out in an unloaded state with an effective length for its active arm of 426,667 m from the center-of-rotation, a tip velocity of 2,133 m/s and a rotational period of 1256.64 s. The center-of-mass of the EarthWhip is in a highly elliptical orbit with an apogee of 33,588 km (almost out to geosynchronous orbit), an eccentricity of 0.655, an orbital period of exactly 8 hours, a perigee radius of 7008 km (630 km altitude), and a perigee velocity of 9,701 m/s. The tether rotational phase is adjusted so that the active tether arm is pointing straight down at perigee, with the tether tip velocity opposing the center-of-mass velocity. The tip of the tether is thus at an altitude of 630 km-426.7 km = 203.3 km and a velocity with respect to the Earth of 9,701 m/s - 2,133 m/s = 7,568 m/s, which matches the payload altitude and velocity.

EarthWhip After Payload Pickup

After picking up the payload, the loaded EarthWhip tether is now symmetrically balanced. Since the added payload had both energy and momentum appropriate to its position on the rotating tether, the EarthWhip rotation angular rate does not change and the period of rotation remains at 1257 s. The center of mass of the loaded EarthWhip, however, has shifted to the center of the tether central station, so the effective length of the loaded tether arm is now at its design length of 400,000 km and tip velocity of 2,000 m/s. With the addition of the payload, however, the orbit of the tether center-of-mass has dropped 26.7 km to a perigee of 6981.3 km, while the perigee velocity has slowed to 9,568 m/s. The apogee of the new orbit is 28,182 km and the eccentricity is 0.603, indicating that this new orbit is less eccentric than the initial orbit due to the payload mass being added near perigee. The period is 23,197 s or 6.44 hours.

Payload Toss

The catch and toss operation at the Earth could have been arranged as shown in Figure 23, so that the payload catch was on one side of the perigee and the payload toss was on the other side of the perigee, a half-rotation of the tether later (10.5 minutes). To simplify the mathematics for this initial analysis, however, we assumed that the catch occurred right at the perigee, and that the tether holds onto the payload for a full orbit. The ratio of the tether center-of-mass orbital period of 23,197 s is very close to 18.5 times the tether rotational period of 1256.64 s, and by adjusting the length of the tether during the orbit, the phase of the tether rotation can be adjusted so that the tether arm holding the payload is passing through the zenith just as the tether center-of-mass reaches its perigee. The payload is thus tossed at an altitude of 603 km + 400 km =1003 km (7381 km radius), at a toss velocity equal to the tether center-of-mass perigee velocity plus the tether rotational velocity or 9,568 m/s + 2,000 m/s = 11,568 m/s. In the combined catch and toss maneuver, the payload has been given a total velocity increment of twice the tether tip velocity or Δv=4,000 m/s.

EarthWhip After Payload Toss

After tossing the payload, the EarthWhip tether is back to its original mass. It has given the payload a significant fraction of its energy and momentum. At this point in the analysis, it is important to insure that no portion of the tether will intersect the upper atmosphere and cause the EarthWhip to deorbit. We have selected the minimum total mass for the EarthWhip at 15,000 kg to insure that doesn't happen. The new orbit for the EarthWhip tether has a perigee of its center of mass of 6955 km (577 km altitude), apogee of 24,170 km, eccentricity of 0.552, and a period of 5.37 hours. With the new perigee at 577 km altitude, even if the tether rotational phase is not controlled, the tip of the active arm of the tether, which is at 426.67 km from the center-of-mass of the tether, does not get below 150 km from the surface of the Earth where it might experience atmospheric drag. In practice, the phase of the tether rotation will be adjusted so that at each perigee passage, the tether arms are roughly tangent to the surface of the Earth so that all parts of the tether are well above 500 km altitude, where the air drag and traffic concerns are much reduced.

With its new orbital parameters, the EarthWhip tether is in its "low energy" state. There are two options then possible. One option is to keep the EarthWhip in its low energy elliptical orbit to await the arrival of an incoming payload from Mars. The EarthWhip will then go through the reverse of the process that it used to send the payload from Earth on its way to Mars. In the process of capturing the incoming Mars payload, slowing it down, and depositing it gently into the Earth's atmosphere, the EarthWhip will gain energy which will put it back into the "high energy" elliptical orbit it started out in. If, however, it is desired to send another payload out from Earth before there is an incoming payload from Mars, then the solar electric power supply on the tether central station can be used to generate electrical power. This electrical power can then be used to restore the EarthWhip to its high energy elliptical orbit using either electrodynamic tether propulsion[18] or gravity-gradient propulsion. [11,13]

Payload Escape Trajectory

The velocity gain of $\Delta v \approx 4{,}000$ m/s given the payload deep in the gravity well of Earth results in a hyperbolic excess velocity of 5,081 m/s. The payload moves rapidly away from Earth and in 3.3 days reaches the "patch point" on the boundary of the Earth's "sphere of influence," where the gravity attraction of the Earth on the payload becomes equal to the gravity attraction of the Sun on the payload. An accurate calculation of the payload trajectory would involve including the gravity field of both the Sun and the Earth (and the Moon) all along the payload trajectory. For this simplified first-order analysis, however, we have made the assumption that we can adequately model the situation by just using the Earth gravity field when the payload is near the Earth and only the Solar gravity field when we are far from the Earth, and that we can switch coordinate frames from an Earth-centered frame to a Sun-centered frame at the "patch point" on the Earth's "sphere of influence."

Payload Interplanetary Trajectory

When this transition is made at the patch point, we find that the payload is on a Solar orbit with an eccentricity of 0.25, a periapsis of 144 Gm and an apoapsis of 240 Gm. It is injected into that orbit at a radius of 151.3 Gm and a velocity of 32,600 m/s. (The velocity of Earth around the Sun is 29,784 m/s.) It then coasts from the Earth sphere-of-influence patch point to the Mars sphere-of-influence patch point, arriving at the Mars patch point at a radius of 226.6 Gm from the Sun and a velocity with respect to the Sun of 22,100 m/s. (The velocity of Mars in its orbit is 24,129 m/s.) The elapsed time from the Earth patch point to the Mars patch point is 148.9 days.

Payload Infall Toward Mars

At the patch point, the analysis switches to a Mars frame of reference. The payload starts its infall toward Mars at a distance of 1.297 Gm from Mars and a velocity of 4,643 m/s. It is on a hyperbolic trajectory with a periapsis radius of 4451 km (altitude above Mars of 1053 km) and a periapsis velocity of 6,370 m/s. The radius of Mars is 3398 km and because of the lower gravity, the atmosphere extends out 200 km to 3598 km. The infall time is 3.02 days.

MarsWhip Before Payload Catch

The MarsWhip tether is waiting for the arrival of the incoming high velocity payload in its "low energy" orbital state. The active tether arm is 426,667 m long and the tip speed is 2,133 m/s. The center-of-mass of the unbalanced tether is in an orbit with a periapsis radius of 4025 km (627 km altitude), periapsis velocity of 4,236 m/s, apoapsis of 21,707 km, eccentricity of 0.687, and a period close to 0.5 sol. (A "sol" is a Martian day of 88,775 s, about 39.6 minutes longer than an Earth day of 86,400 s. The sidereal sol is 88,643 s.) The orbit and rotation rate of the MarsWhip tether is adjusted so that the active arm of the MarsWhip is passing through the zenith just as the center-of-mass is passing through the perigee point. The grapple at the end of the active arm is thus at 4024.67+426.67 = 4,451.3 km, moving at 4,236 m/s + 2,133 m/s = 6,370 m/s, the same radius and velocity as that of the payload, ready for the catch.

MarsWhip After Payload Catch

After catching the payload, the MarsWhip tether is now in a balanced configuration. The effective arm length is 400,000 m and the tether tip speed is 2,000 m/s. In the process of catching the incoming payload, the periapsis of the center-of-mass of the tether has shifted upward 26,667 m to 4,051 km and the periapsis velocity has increased to 4,370 m/s, while the apoapsis has risen to 37,920 km, and the eccentricity to 0.807. The period is 1.04 sol.

Payload Release and Deorbit

The payload is kept for one orbit, while the phase of the tether rotation is adjusted so that when the tether center-of-mass reaches periapsis, the active tether arm holding the payload is approaching the nadir orientation. If it were kept all the way to nadir, the payload would reach a minimum altitude of about 250 km (3648 km radius) at a velocity with respect to the Martian surface of 4370 m/s - 2000 m/s = 2370 m/s. At 359.5 degrees (almost straight down), this condition is achieved to four significant figures. The payload is then moving at a flight path angle with respect to the local horizon of 0.048 radians and enters the atmosphere at a velocity of 2,442 km/s.

MarsWhip after Deorbit of Payload

After tossing the payload, the MarsWhip tether is back to its original mass. The process of catching the high energy incoming payload, and slowing it down for a gentle reentry into the Martian atmosphere, has given the MarsWhip a significant increase in its energy and momentum. At this point in the analysis, it is important to check that the MarsWhip started out with enough total mass so that it will not be driven into an escape orbit from Mars.

The final orbit for the tether is found to have a periapsis radius of 4078 km (676 km altitude so that the tether tip never goes below 253 km altitude), a periapsis velocity of 4,503 m/s, an apoapsis radius of 115,036 km, an eccentricity of 0.931, and a period of 6.65 sol. The tether remains within the gravity influence of Mars and is in its high energy state, ready to pick up a payload launched in a suborbital trajectory out of the Martian atmosphere, and toss it back to Earth.

Elapsed Time

The total elapsed transit time, from capture of the payload at Earth to release of the payload at Mars, is 157.9 days. This minimal mass PlanetWhip scenario is almost as fast as more massive PlanetWhip tethers since, although the smaller mass tethers cannot use extremely high or low eccentricity orbits without hitting the atmosphere or being thrown to escape, the time spent hanging on the tether during those longer orbit counts as well and the longer unbalanced grapple arm of the lightweight tether lets it grab a payload from a higher energy tether orbit.

Summary

We have developed tether system architectures for Earth-Luna and Earth-Mars payload transport. Our analyses have concluded that the optimum architecture for a tether system designed to transfer payloads between LEO and the lunar surface will utilize one tether facility in an elliptical, equatorial Earth orbit and one tether in low lunar orbit. We have developed a preliminary design for a 80 km long Earth-orbit tether boost facility capable of picking payloads up from LEO and injecting them into a minimal-energy lunar transfer orbit. Using currently available tether materials, this facility would require a mass 10.5 times the mass of the payloads it can handle. After boosting a payload, the facility can use electrodynamic propulsion to reboost its orbit, enabling the system to repeatedly send payloads to the Moon without requiring propellant or return traffic. When the payload reaches the Moon, it will be caught and transferred to the surface by a 200 km long lunar tether. This tether facility will have the capability to reposition a significant portion of its "ballast" mass along the length of the tether, enabling it to catch the payload from a low-energy transfer trajectory and then "spin-up" so that it can deliver the payload to the Moon with zero velocity relative to the surface. This lunar tether facility would require a total mass of less than 17 times the payload mass. Both equatorial and polar lunar orbits are feasible for the Lunavator™. Using two different numerical simulations, we have tested the feasibility of this design and developed scenarios for transferring payloads from a low-LEO orbit to the surface of the Moon, with only 25 m/s of ΔV needed for small trajectory corrections. Thus, it appears feasible to construct a Cislunar Tether Transport System with a total on-orbit mass requirement of less than 28 times the mass of the payloads it can handle, and this system could greatly reduce the cost of round-trip travel between LEO and the surface of the Moon by minimizing the need for propellant expenditure.

Using similar analytical techniques, we have shown that two rapidly spinning tethers in highly elliptical orbits about Earth and Mars can be combined to form a similar system that provides rapid interplanetary transport from a suborbital trajectory above the Earth's atmosphere to a suborbital trajectory above the Martian atmosphere and back.

Acknowledgments

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Tether Systems for Satellite Deployment and Disposal

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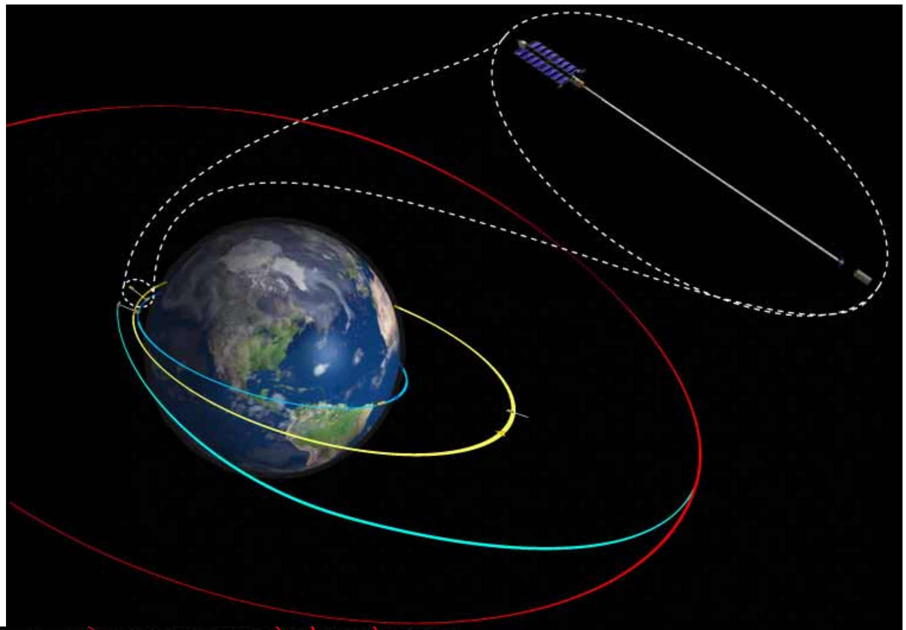

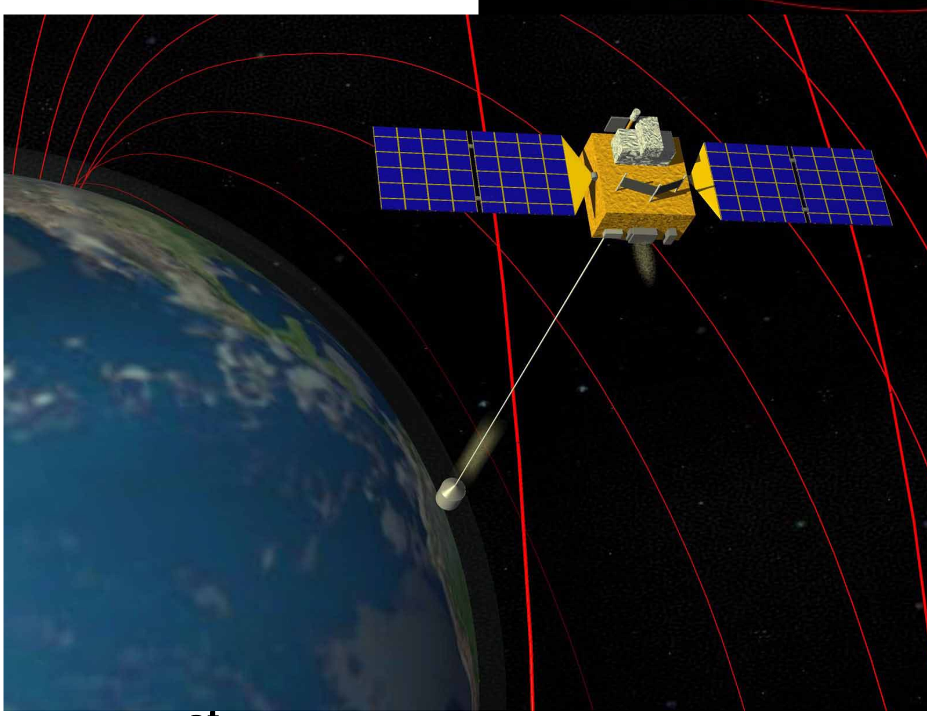

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TETHER SYSTEMS FOR SATELLITE DEPLOYMENT AND DISPOSAL

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Abstract

Space tether systems have strong potential for providing significant reductions in the cost of propulsion for a number of important applications, including spacecraft deployment, post-mission spacecraft disposal, and satellite orbital maintenance. Tether systems can provide propulsion to space systems both through electrodynamic interactions with the Earth's magnetic field and through momentum-exchange interactions between two objects in orbit. This paper summarizes recent work by Tethers Unlimited, Inc. to develop a product line of tether-based technologies to service markets for LEO microsatellite propulsion, LEO satellite disposal, and deployment of spacecraft to geostationary orbits.

Introduction

Tether systems can provide propellantless in-space propulsion for a wide range of space missions. A space tether is a long thin cable constructed of either high-strength fibers or conducting wires that is extended between two or more objects in space. Tethers can accomplish propulsion missions through two different mechanisms. First, current flowing in conducting tethers can generate forces through electrodynamic interactions with the Earth's magnetic field. Second, tethers can provide a mechanical link between two objects in space, enabling orbital momentum and energy to be exchanged between the objects. Both mechanisms can accomplish significant ΔV operations without consumption of propellant. By minimizing the need for propellant to be carried into orbit, tether systems can greatly reduce the total propulsion costs for many missions.

Tethers Unlimited, Inc. (TUI) is currently developing a line of tether products to provide cost-effective propulsion capabilities for applications including end-of-mission LEO satellite deorbit, microsatellite orbit-raising and stationkeeping, and deployment of large payloads to geostationary and lunar transfer orbits.

Small Electrodynamic Tether Devices

The first two products in development at TUI will utilize electrodynamic tether techniques to provide low-mass and low-cost propulsion for satellites operating in LEO. The principle of electrodynamic tethers are illustrated in Figure 1. In an electrodynamic tether system, a long tether constructed of conducting wire is extended from a spacecraft. Gravity-gradient forces will tend to orient this long flexible structure along the local vertical direction. The orbital motion of the tethered system across the Earth's magnetic field generates a voltage along the tether. If the tether system provides a mechanism for electrically contacting the ionospheric space plasma at both ends of the tether, the induced voltage can drive a current up the tether, as illustrated in Figure 1 a. This current interacts with the magnetic field to generate a **J**x**B** force on the tether which opposes the motion of the tether system. This force is thus a "drag" force which drains orbital energy from the system, lowering the tethered system's orbit. If, however, the tether system applies a voltage down the tether sufficient to overcome the induced voltage, it can drive a current down the tether, resulting in a **J**x**B** force on the tether that raises the orbit of the tether system.

The first product in development, the Terminator Tether™, is a small electrodynamic tether drag device intended to enable the users of LEO space to mitigate the growth of space debris by providing them a low-cost and reliable means of ensuring that their satellites are removed from orbit after they has completed their missions. The second device, the Microsatellite Propellantless Electrodynamic Tether (μPET™) Propulsion System, will provide an essentially infinite ΔV propulsion capability for orbit raising, orbital modification, and long-term stationkeeping for small satellites operating in LEO. The μPET™ Propulsion System builds upon the Terminator Tether™ technologies by adding the capability to process input power from the satellite to drive the electrodynamic tether in a thrust mode, rather than a drag mode.

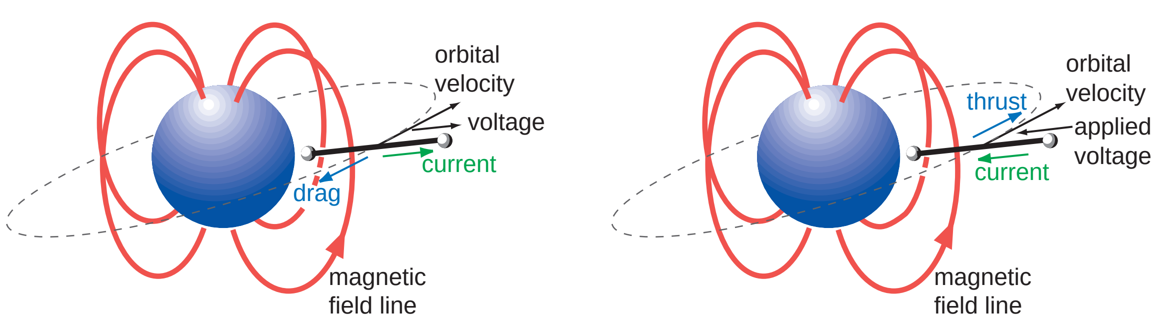

Figure 1. a) Electrodynamic tether drag mode. b) Electrodynamic tether thrust mode.

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Terminator Tether™ Deorbit Device

TUI's first tether product is the Terminator Tether™, a small electrodynamic tether drag device designed to provide a cost-effective method for autonomously deorbiting low Earth orbit (LEO) spacecraft to mitigate the growth of orbital debris.[1] The Terminator Tether™ is a small, lightweight, low-cost device that will be attached to satellites and upper stages before launch. The device contains a conducting tether, a tether deployer, an electron emitter, and electronics to monitor the host spacecraft and control the deployment and operation of the tether. During the operational period of the host spacecraft, the tether will be stored in the deployer and the Terminator Tether™ electronics will be dormant, waking up periodically to check the status of the host spacecraft. When the device receives an activation command, or when it determines that the host spacecraft is defunct, the Terminator Tether™ will activate springs in the deployer to kick the device down and away from the spacecraft, deploying the tether.

A schematic of the device is shown in Figure 2. Once deployed, the motion of the conducting tether through the Earth's magnetic field will generate a voltage along the length of the tether; in a direct orbit, the top of the tether will be charged positively relative to the ambient ionospheric plasma. Most of the tether length will be left uninsulated, so that the bare wires can efficiently collect electrons from the ionosphere.[2] These electrons will flow down the tether to the Terminator Tether™ endmass, where the electron emitter will expel them

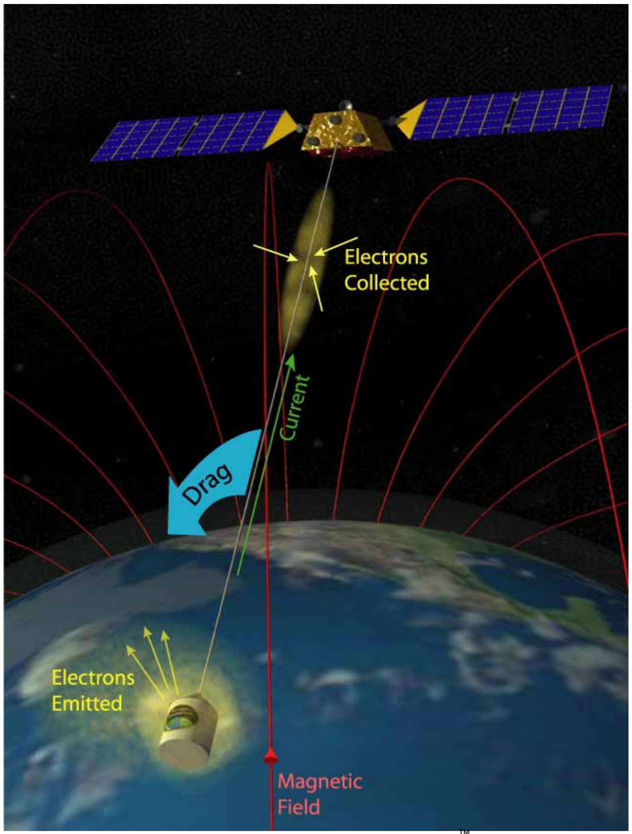

Figure 2. The Terminator Tether™.

back in to the ionosphere. Thus a current will flow up the tether, and the current "loop" will be closed by plasma waves in the ionosphere.[3,4] This current will then interact with the Earth's magnetic field to generate a Lorentz **J**x**B** force on the tether. This force will oppose the orbital motion of the tether. Through its mechanical connection to the host spacecraft, the tether will thus drain the orbital energy of the spacecraft, lowering its orbit until it disintegrates in the upper atmosphere.

The Terminator Tether™ Satellite Deorbit System will be composed of several subsystems: a conducting, survivable tether, a tether deployment system, a device for emitting electron current, and an electronic control system called the Tether Control Unit (TCU).

Tether

In order to electrically insulate the host spacecraft from the tether, a short section of the tether nearest the spacecraft will be constructed of high-strength, nonconducting yarns. The rest of the tether will be a survivable Hoytether™ structure constructed of thin aluminum or copper wires, shown in Figure 3. The tether design will vary depending upon the mass and orbit of the host spacecraft, but for a typical LEO constellation satellite massing 1500 kg, the tether will be 5 km long and mass approximately 15 kg (1% of the host spacecraft mass). In the Hoytether™ design, the wires are knitted together in an open-net structure that provides redundant paths to carry the mechanical load and current. This design will enable the tether to provide a very high probability of surviving the orbital debris environment for the period of several weeks or months required to deorbit the spacecraft.[5]

Tether Control Unit

The TCU is part of the endmass that is deployed below the host spacecraft at the end of the tether. The TCU carries the responsibilities of monitoring the host spacecraft during the spacecraft's operational phase (the dormant phase for the Terminator Tether™), activating the deployment system when it is time to deorbit the host, monitoring and controlling the tether dynamics to optimize the descent rate, and responding to ground control signals to perform avoidance maneuvers.

Electron Emitter

The electron emitter will be a Field Emission Array Cathode (FEAC) device, also known as a Spindt Cathode.[6] This device will be designed to emit up to 1 A of electron current. Electron emission is achieved by applying a gate voltage of approximately 75-100 V between an array of millions of microscopic needle points and a gate electrode. The emitted current can be

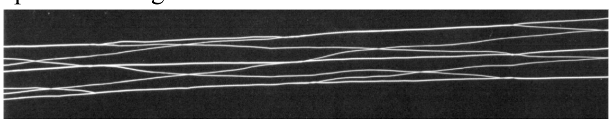

Figure 3. Photo of a 20-cm length of the conducting Tri-Line Hoytape made of 30 gauge aluminum wires knitted together with 22-Tex P.T.F.E. thread.

controlled very precisely by varying this gate voltage.

To enable the FEAC devices to operate reliably in the LEO environment, the emitter tips must be ruggedized to survive bombardment by the atomic oxygen and other constituents found in LEO. TUI is currently collaborating with NASA/MSFC, JPL, the University of Michigan, SRI, LRI, and NRL to develop and test FEAC devices coated with carbide materials intended to provide the necessary ruggedization.

Tether Deployer

The prototype deployer for the Terminator Tether™ is shown in Figure 4. In this figure in which the tether is wound onto a spool. The TCU electronics and other components are contained inside this spool. The Terminator Tether™ system will be housed inside the host satellite, with the bottom surface (with the RF antenna) positioned flush with the satellite's bottom or side surface. The TCU, electron emitter, and batteries are contained in a cylindrical housing that slides inside of the deployer spool, so that during the dormant phase the electronics will be shielded from radiation by the several-cm of the wound aluminum wire tether. When the TCU activates the deployment sequence, it triggers an ejection mechanism which propells the entire Terminator Tether™ unit (except for the mounting bracket and tether anchor) down and away from the host spacecraft at a velocity of several meters per second.

Device Mass and Sizing

TUI is currently building a prototype of the Terminator Tether™ that is sized to provide deorbit capability for a 2000-3000 kg LEO spacecraft. A mass breakdown for this prototype is given below.

Tether mass:	10.0	kg
Shroud:	1.9	kg
Spool Assembly:	4.9	kg
Ejection Mechanism	5.0	kg
Electron Emitter:	1.2	kg
TCU Electronics:	3.7	kg
Tether Anchor	0.06	kg
Total Tether system mass:	26.76	kg

Deorbit Performance

Using the TetherSim™ numerical simulation tool, we have studied the potential performance of the Terminator Tether™ for deorbit of satellites from various LEO orbits.[7] Figure 5 shows the time required to decrease the altitude of a satellite to 250 km from a range of initial orbital altitudes and inclinations. A Terminator Tether™ massing approximately 2% of the mass of the host spacecraft could deorbit an upper stage from a 400 km, 50° orbit within about two weeks, or a communications satellite from a 850 km, 50° orbit within about three months. Figure 6 shows data for satellite altitude, tether current, and tether libration angle from a simulation of deorbit of a small satellite from a 370 km, 51° inclination initial orbit.

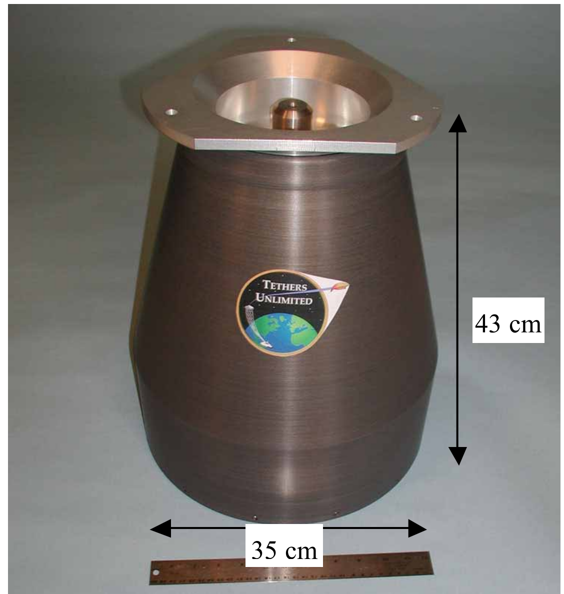

Figure 4. Photo of the Terminator Tether™ deployer prototype.

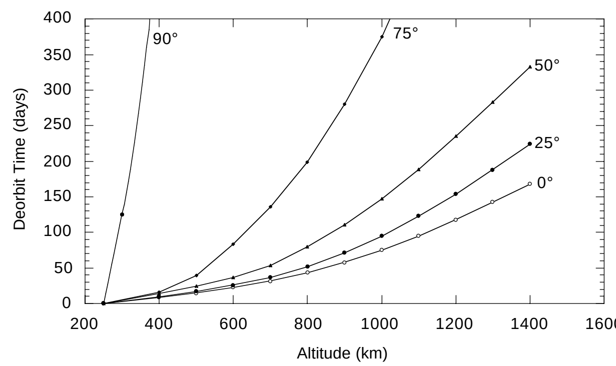

Figure 5. Time for a Terminator Tether™ with a 7.5 km, 15 kg aluminum tether and 15 kg endmass to decrease the perigee altitude of a 1500 kg spacecraft to 250 km. Note that deorbit time can be decreased by using longer or more massive tethers.

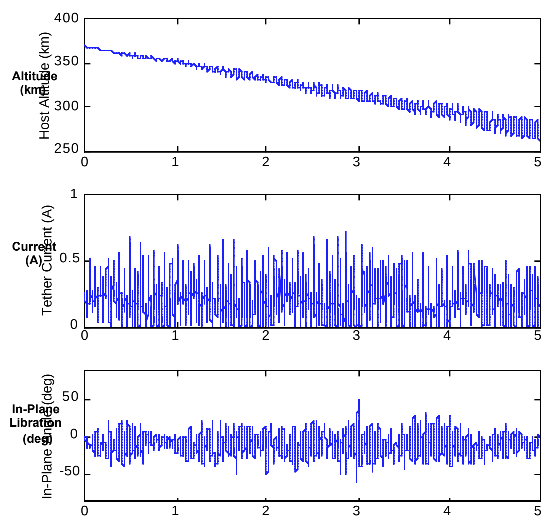

Figure 6. TetherSim™ results for altitude, tether current, and tether libration for deorbit of a microsatellite from a 370 km, 51° orbit.

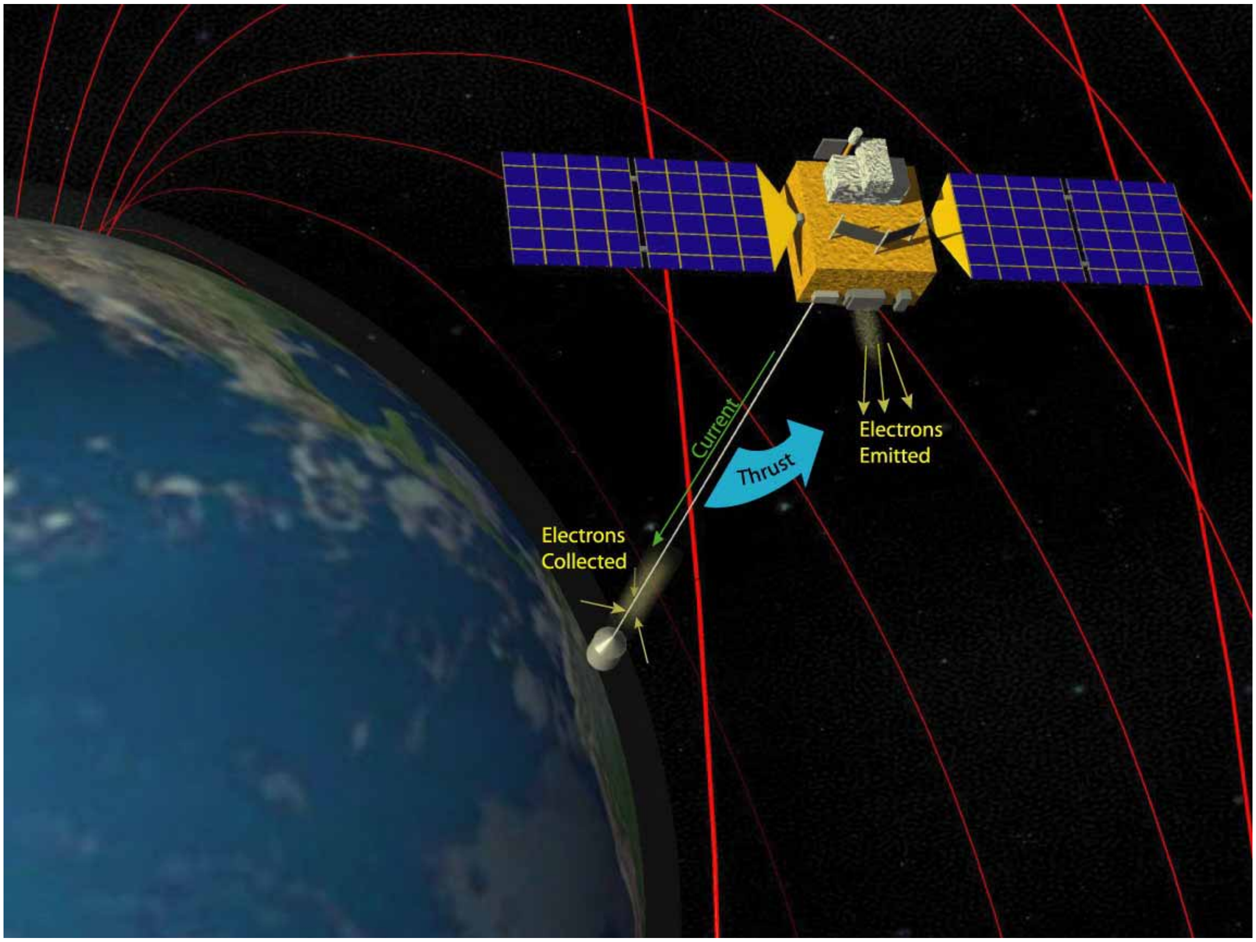

Figure 7. Schematic of the μPET Propulsion System boosting a small satellite.

μPET Propulsion System

Currently, economic, programatic, and technology issues are driving NASA, DoD and other organizations to reduce their dependence on single, large satellites and instead develop systems of several or many small, inexpensive satellites. Many of the potential applications of these small satellites will require the satellites to change orbits frequently and rapidly or to hold at altitude for long periods of time. Because many of these satellites will operate in low-LEO orbits, stationkeeping propulsion to counteract atmospheric drag will impose large total ΔV requirements on the satellite's propulsion system. These microsatellites, however, will be very power- and weight- limited, so there is a need for small propulsion systems able to provide both rapid orbit transfer capability and high specific impulse operation.

Electrodynamic tether propulsion can provide both rapid orbit transfer and effectively infinite specific impulse operation. A small, lightweight, and inexpensive tether system can provide propulsion capabilities for orbital transfer, inclination changes, and stationkeeping. Because it uses electrodynamic forces to provide thrust to the satellite, it will not consume propellant, and thus it can enable small satellites to stationkeep indefinitely, even in low LEO orbits where aerodynamic drag would otherwise impose prohibitive ΔV requirements. TUI has recently begun the development of a "Microsatellite Propellantless Electrodynamic Tether (μPET) Propulsion System". The μPET Propulsion System concept is illustrated in Figure 7.

Performance

Numerical simulation of the μPET system indicate that its thrust efficiency is competitive with other small propulsion systems such as micro-Hall and micro-Ion thrusters. Because electrodynamic thrusting utilizes interactions with the geomagnetic field, the thrust efficiency of the μPET varies with inclination and altitude. In an equatorial orbit at 370 km altitude, the thrust efficiency is approximately 69 μN/W. At 51° inclination, it is approximately 50 μN/W, while at 70° it drops to 31 μN/W. Figure 8 shows TetherSim™ simulation results for operation of a μPET on a microsatellite a 370 km, 51° orbit at several input power levels. These simulations indicate that a small electrodynamic tether system can provide significant propulsive capability for small satellites with very low power requirements.

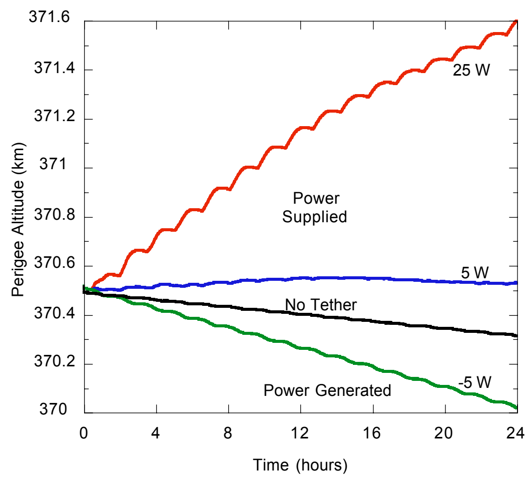

Figure 8. Evolution of perigee altitude over 24 hours for a 125 kg microsatellite in a 51° direct orbit with a μPET™ propulsion system using a 1 kg, 1 km long tether, in Boost Mode (25 W), Stationkeeping Mode (5W), and Deboost Mode (0 W). Power was supplied to the tether system only when the satellite was in sunlight.

Tether Boost Facilities for In-Space Propulsion

By taking advantage of tethers' abilities to transfer orbital momentum and energy from one space object to another, tether systems can also be useful for propulsion applications beyond LEO. A tether transportation system may be able to provide a fully reusable, zero-propellant infrastructure for in-space transportation, and thus may have the potential to reduce the costs of delivering payloads to GEO, the Moon, Mars, and other destinations by an order of magnitude or more. Under funding from NASA's Institute for Advanced Concepts (NIAC), Tethers Unlimited, Inc. and the Boeing Company are developing an architecture for such a tether transportation system. This system will utilize momentum-exchange techniques and electrodynamic tether propulsion to transport multiple payloads *with little or no propellant consumption*. The tether transportation architecture is designed to be built incrementally, with each component able to perform a useful revenue-generating mission to help fund the deployment of the rest of the system. The first component of the system will be a Tether Boost Facility that will transfer satellites and other payloads from low Earth orbit (LEO) to geostationary transfer orbit (GTO). This same facility will also be capable of boosting payloads to lunar transfer orbit (LTO). Later components will increase the payload capacity of the Tether Boost Facility and enable frequent round-trip travel to the surface of the Moon[8,9] and to Mars.[10] In this paper we discuss an architecture and incremental development plan for an Earth-Moon-Mars Tether Transportation System.

Background: Momentum-Exchange Tethers

In a momentum-exchange tether system, a long, thin, high-strength cable is deployed in orbit and set into rotation around a massive central body. If the tether facility is placed in an elliptical orbit and its rotation is timed so that the tether will be oriented vertically below the central body and swinging backwards when the facility reaches perigee, then a grapple assembly located at the tether tip can rendezvous with and acquire a payload moving in a lower orbit, as illustrated in Figure 9. Half a rotation later, the tether can release the payload, tossing it into a higher energy orbit. This concept is termed a momentum-exchange tether because when the tether picks up and throws the payload, it transfers some of its orbital energy and momentum to the payload. The tether facility's orbit can be restored later by reboosting with propellantless electrodynamic tether

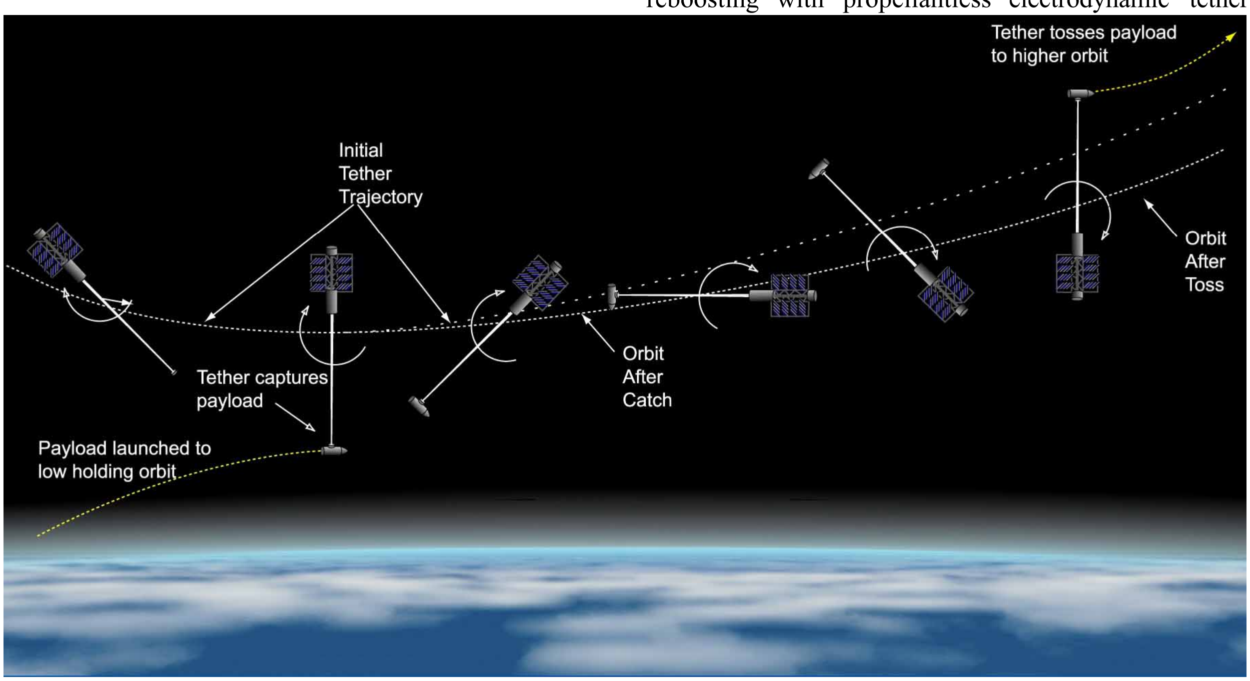

Figure 9. Momentum Exchange Tether catching and tossing payload.

propulsion or with high specific impulse electric pro pulsion; alternatively, the tether's orbit can be restored by using it to de-boost return traffic payloads.

Key Advantages

A tether transportation system has several advantages compared to conventional and other advanced in-space propulsion systems:

- *(Near) Zero Propellant Usage*

Chief among these advantages is the ability to elimi nate the need for propellant expenditure to perform pay load transfers. Of course, some propellant expenditure will be needed for trajectory corrections and rendezvous maneuvering, but these requirements will be very small, a few tens of meters per second. The ability to shave several thousands of meters per second from the ΔV needed to deliver a payload to its destination can enable customers to utilize much smaller launch vehi cles than would be required with a rocket-only system, greatly reducing total launch costs.

- *Short Transfer Times*

A momentum-exchange tether system provides its ΔV to the payload in an essentially impulsive manner. Thus the transfer times in a tether system are very short, comparable to rocket-based systems.

- *Reusable Infrastructure*

Once deployed, a tether boost facility could transfer many, many payloads before requiring replacement. Thus the recurring costs for payload transport could be reduced to the cost of operations.

- *Fully Testable System*

Another important but often overlooked advantage of a tether transportation system is that the components that perform the actual payload transfer operations can be fully tested *in space operations* before being used for critical payloads. A tether facility could be tested many times with "dummy" payloads – or, better yet, with low inherent value payloads such as water or fuel – to build confidence for use on high value or manned payloads.

Prior Work on Tether Transport Architectures

Several prior research efforts have investigated conceptual designs for momentum-exchange tether systems. In 1991, Carroll proposed a tether transport facility that could pick payloads up from suborbital trajectories and provide them with a total ΔV of approximately 2.3 km/s.[11]

Soon thereafter, Forward[12] proposed combining this system with a second tether in elliptical Earth orbit and a third tether in orbit around the Moon to create a system for round-trip travel between suborbital Earth trajectories and the lunar surface. In 1997, Hoyt[13] developed a preliminary design for this "LEO to Lunar Surface Tether Transport System."

In 1998, Bangham, Lorenzini, and Vestal developed a conceptual design for a two-tether system for boosting payloads from LEO to GEO.[14] Their design proposed the use of high specific impulse electric thrusters to restore the orbit of the tether facilities after each payload boost operation. Even with the propellant mass requirements for reboost, they found that this system could be highly economically advantageous compared chemical rockets for GEO satellite deployment.

Under a Phase I NIAC effort, Hoyt and Uphoff[8] refined the LEO⇒Lunar system design to account for the full three-dimensional orbital mechanics of the Earth-Moon system, proposing a "Cislunar Tether Transportation System" illustrated in Figure 12. This architecture would use one tether in elliptical, equatorial Earth orbit to toss payloads to minimum-energy lunar transfer orbits, where a second tether, called a "Lunavator™" would catch them and deliver them to the lunar surface. The total mass of the tether system, could be as small as 27 times the mass of the payloads it could transport.

The same NIAC effort also resulted in a preliminary design by Forward and Nordley[10] for a "Mars-Earth Rapid Interplanetary Tether Transport (MERITT)" system capable of transporting payloads on rapid trajectories between Earth and Mars.

Momentum-exchange tethers may also provide a means for reducing the cost of Earth-to-Orbit (ETO) launches. This architecture would use a hypersonic airplane or other reusable launch vehicle to carry a payload up to 100 km altitude at Mach 10-12, and handing it off to a large tether facility in LEO which would then pull it into orbit or toss it to either GTO or escape.[15,16]

First Operational System:
LEO⇒GTO/LTO Tether Boost Facility

Because the launch costs for deploying components of a Tether Transportation system will be a significant driver in the overall development costs, it will be im perative to the economic viability of the tether transpor tation architecture that every component placed into orbit be capable of generating revenue very soon after deployment. Although our ultimate goal is to develop a tether transport system capable of providing low-cost travel to the Moon and Mars, we have chosen to focus our initial development efforts on designing a Tether Boost Facility optimized for servicing traffic to geosta tionary orbit because lunar, Mars, and even LEO traffic volumes are currently speculative or highly uncertain, whereas GEO satellite deployment is a relatively well-understood and growing market.

The LEO⇒GTO Tether Boost Facility will boost payloads from low-LEO to geostationary transfer orbits (GTO). In sizing the facility design, we have sought to balance two somewhat competing drivers: first, the desire to be able to have a fully-operational, revenue-generating tether boost facility that can be deployed in a single launch on a rocket expected to be available in the 2010 timeframe, and second, the desire for the tether facility to be capable of gaining as large as possible a market share of the projected GEO traffic. Recent projections of GEO traffic, shown in Figure 11, indicate that the general trend for GEO payloads is to become more and more massive. Over the timeframe covered by the projections, payloads in the range of 4-6 metric tons are expected to account for roughly 80% of the commercial market. Consequently, it would be highly desirable to design the Tether Boost Facility to handle payloads on the order of 5,000 kg. On the other hand, a tether facility designed to toss payloads to GTO must mass roughly 9 times the mass of the payloads it can handle (due primarily to tether sizing, orbital mechanics, and conservation-of-momentum considerations). If the tether facility is to provide an operational capability

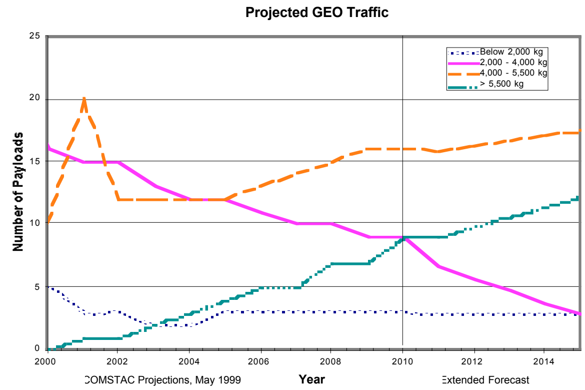

Figure 11. GEO traffic projections for 2000-20015.

after one launch, the tether facility must fit within the payload capacity of an available launch vehicle. In the 2010 timeframe, the largest payload-to-LEO anticipated is that of the Delta-IV-Heavy rocket, which will be able to place 20,500 kg into LEO.

Consequently, we have chosen to follow a modular development approach in which the initial Tether Boost

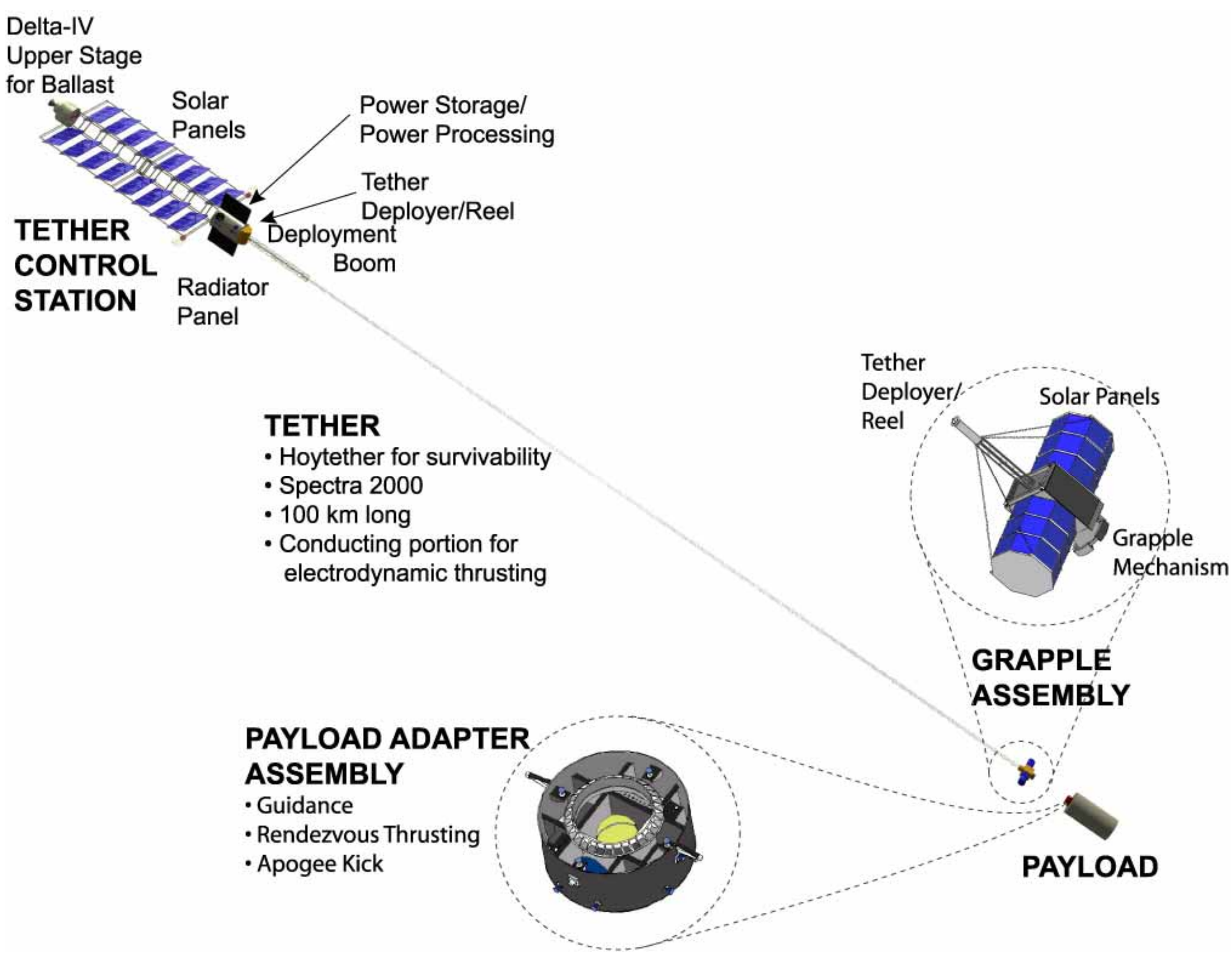

Figure 10. The LEO⇒GTO Tether Boost Facility

Facility launched will be sized to fit on a Delta-IV-H. This facility will be capable of boosting 2,500 kg payloads to GTO as well as 1,000 kg payloads to lunar transfer orbit (LTO). This facility could potentially service approximately one-quarter of the ~400 payloads expected to be launched to GEO in the next 40 years. The facility hardware is designed in a modular fashion, so that after the initial facility has proven its capability and reliability, a second set of essentially identical hardware could be launched and combined with the first set to create a Tether Boost Facility capable of tossing 5,000 kg to GTO and 2,000 kg to LTO. Additional modules can increase the system capacity further.

To obtain a first-order estimate of the potential cost savings of the Tether Boost Facility, consider a mission to boost a 5 metric ton class payload into GTO. To do so using currently-available rocket launch systems would require a vehicle such as a Delta IVM+ (4,2), a Proton M, or a SeaLaunch Zenit 3SL. Depending upon the launch service chosen and other business factors, current costs for this launch will be approximately $90M. If, however, a Tether Boost Facility is available that is capable of boosting the 5 metric ton payload from a LEO holding orbit to GTO, the customer could use a smaller launch vehicle, such as a Delta-II 7920, with an estimated launch cost of $45M, or a vehicle comparable to the Dnepr 1 (RS-20), with an estimated sticker price of $13M. While exact comparisons at this level are difficult due to differing payload capacities of each vehicle and the dependence of launch pricing upon other business factors, these estimates indicate that a reusable Tether Boost Facility could enable commercial and governmental customers to reduce their launch costs by 50% to 85%.

The design of this LEO⇒GTO Tether Boost Facility is discussed in more detail in a previous paper.[17] The facility is designed to boost one 2,500 kg payload to GTO once every month. Although the facility design is optimized for boosting 2,500 kg payloads to GTO, it can also boost different-sized payloads to different orbits; the payload capacity depends upon the total ΔV to be given to the payload.

As a result, in addition to boosting payloads to GTO and LTO, this Tether Boost Facility could also serve as a component of a transportation architecture for delivering payloads to other orbits and other destinations. For example, the initial (2,500 kg to GTO) Facility could boost 5,000 kg payloads to the 20,335 km altitude used by the GPS system. As a component in the transportation system for Mars-bound payloads, the facility could be used to inject a 5,000 kg spacecraft into a highly elliptical equatorial orbit. At the apogee of this holding orbit, the payload could then perform a small ΔV maneuver to torque its orbit to the proper inclination for a Mars trajectory, then perform its Trans-Mars-Injection burn at perigee. The tether facility thus could reduce the ΔV requirements for a Mars mission by over 2 km/s.

Cislunar Tether Transport System

This heavy-lift Boost Facility could then be used to deploy a second tether facility in polar lunar orbit. This facility, called a "Lunavator™" would be capable of catching payloads sent from Earth on minimum-energy transfer trajectories and delivering them to the surface of the Earth. The Lunavator facility could also be built incrementally. The first system would be sized to catch payloads from minimum-energy lunar transfers and drop them into low lunar orbit (LLO) or suborbital trajectories, and to pick-up return payloads from LLO and send them down to LEO. The Lunavator mass could be built up using lunar resources, until it is capable of catching payloads sent from Earth and depositing them directly on the lunar surface, with zero velocity relative to the surface.

The deployment of a tether in lunar orbit would enable the tether system to begin servicing round-trip traffic, creating a "Cislunar Tether Transport System", illustrated in Figure 12, that could deliver payloads from LEO to the surface of the Moon with little or no propellant expenditure.[8]

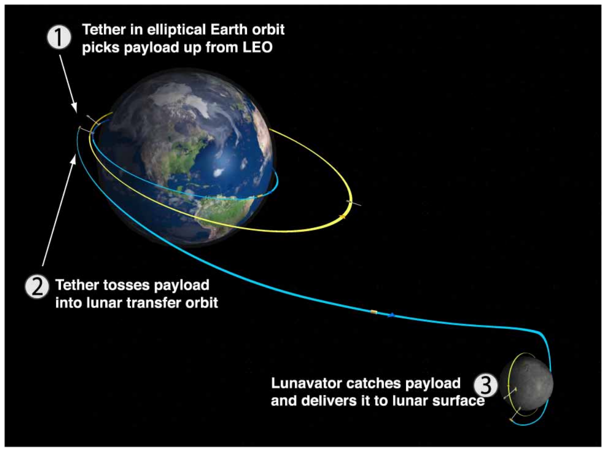

Figure 12. The Cislunar Tether Transport System.

Summary

Tether systems have strong potential for providing low-cost propulsion capabilities for a number of applications. Tethers Unlimited, Inc. is currently developing several small electrodynamic tether propulsion products, including the Terminator Tether™ for satellite deorbit and the μPET™ Propulsion System for microsatellite propulsion. These products will provide cost-effective propulsion for satellite orbit raising, station-keeping, and end-of-life deorbit for LEO spacecraft. TUI is also developing designs for momentum-exchange tether systems capable of transporting many payloads from LEO to GTO and beyond with minimal propellant requirements.

Acknowledgments

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Appendix K

TETHER FACILITY REBOOST

Rob Hoyt
Tethers Unlimited, Inc.

Introduction

A key factor in the economic viability of a Tether Boost Facility will be the frequency with which the facility can boost payloads. The throughput capacity of a tether facility will be determined largely by the time required to restore the facility's orbit after each payload boost operation. In this document we present analytical methods for calculating the changes in a tether facility's orbit due to a boost operation, and use numerical simulation to estimate the time required to reboost the facility orbit using electro-dynamic tether propulsion.

Changes In Momentum-Exchange Tether Facility Orbit:

When the facility catches and tosses a payload, it imparts a fraction of its orbital energy and momentum to the payload, and thus its orbit is lowered. In this section we present a brief summary of an analytical method for calculating the facility orbit changes. This method assumes that the orbits are Keplerian.

Useful equations for calculating the semimajor axis and eccentricity of an orbit from the velocity and radius at perigee or apogee are:

Semimajor Axis:
$$a = \left[\frac{2}{r} - \frac{V^2}{\mu_e}\right]^{-1} \tag{1}$$

Eccentricity:
$$e = \pm\left(1 - \frac{r}{a}\right) \tag{2}$$
, where sign is (+) for values calculated at perigee, and (–) for apogee

When the tether boost facility and payload rendezvous, the tether facility has a center of mass radius $r_{facility,0}$ and velocity $V_{facility,0}$, and the payload has radius and velocity $r_{payload,0}$ & $V_{payload,0}$.. In calculating the facility position and velocity, one must account for the tether. The Tether Boost Facility will use a tapered tether to minimize the tether's mass; the location of the tether's center-of-mass (COM) relative to the facility is computed numerically.

<u>Facility+Payload Orbit After Payload Acquisition:</u>

After catching the payload, the new system COM radius and velocity are:

$$r_{COM,1} = \frac{r_{facility,0}\left(M_{facility} + M_{tether}\right) + r_{payload,0}M_{payload}}{M_{facility} + M_{tether} + M_{payload}} \tag{3}$$

$$V_{COM,1} = \frac{V_{facility,0}\left(M_{facility} + M_{tether}\right) + V_{payload,0}M_{payload}}{M_{facility} + M_{tether} + M_{payload}}, \tag{4}$$

and we use Eqns. (1) & (2) to calculate the new semimajor axis a_1 and eccentricity e_1.

<u>Tether System Tip Velocity</u>

When the facility catches the payload, the rotational inertia of the system is conserved, and thus the angular velocity ω remains constant. However, the COM of the system shifts closer to the payload end of the tether when the mass of the payload is added to the system. Thus the tip velocity of the system decreases. The new velocity of the tether tip relative to the system's COM can be estimated as:

$$V_{tip,1} = \omega L_1, \tag{5}$$

where L_1 is the new distance from the tether tip to the system's center-of-rotation (=COM).

Facility Orbit After Payload Toss:

After tossing the payload, the new facility radius and velocity are:

$$r_{facility,2} = \frac{r_{COM,1}\left(M_{facility} + M_{tether} + M_{payload}\right) - r_{payload,2}M_{payload}}{M_{facility} + M_{tether}} \tag{6}$$

$$V_{facility,2} = \frac{V_{COM,1}\left(M_{facility} + M_{tether} + M_{payload}\right) - V_{payload,2}M_{payload}}{M_{facility} + M_{tether}}, \tag{7}$$

and we again use Eqns. (1) & (2) to calculate the new semimajor axis a_2 and eccentricity e_2.

In order for the tether boost facility to prepare itself to boost another payload, it must restore its orbital parameters to the original values of a and e.

Simulation Method

The simulation was performed using the TetherSim numerical code. This run utilized the IGRF magnetic field model and a heuristic plasma density model based upon IRI data for equatorial orbits. The model was run for 3 days of simulation time.

System Design

The Tether Boost Facility system design used in this simulation is detailed in Table 1. The orbital parameters of this system were chosen to make the facility's pre-catch orbit and the payload's initial orbit resonant with a 36:17 ratio, so that they have a rendezvous opportunity approximately once every two days. This choice of resonance enables the facility's control station mass to be relatively small, just over 4 times the payload mass, and the facility's total mass is under 7 times the payload mass. Before payload catch, the facility is in a 385x8938 km elliptical orbit, and after payload toss it drops into a 370x5732 km orbit. The geometry of the facility and payload GTO orbits immediately after payload toss, and the desired final facility orbit and the payload GEO orbit are shown in Figure 1.

Reboost Thrust Operation

In this simulation, the tether facility included an electrodynamic tether system in which a 500 kg aluminum conductor was included in 60 km of the tether length. The electrodynamic tether system had

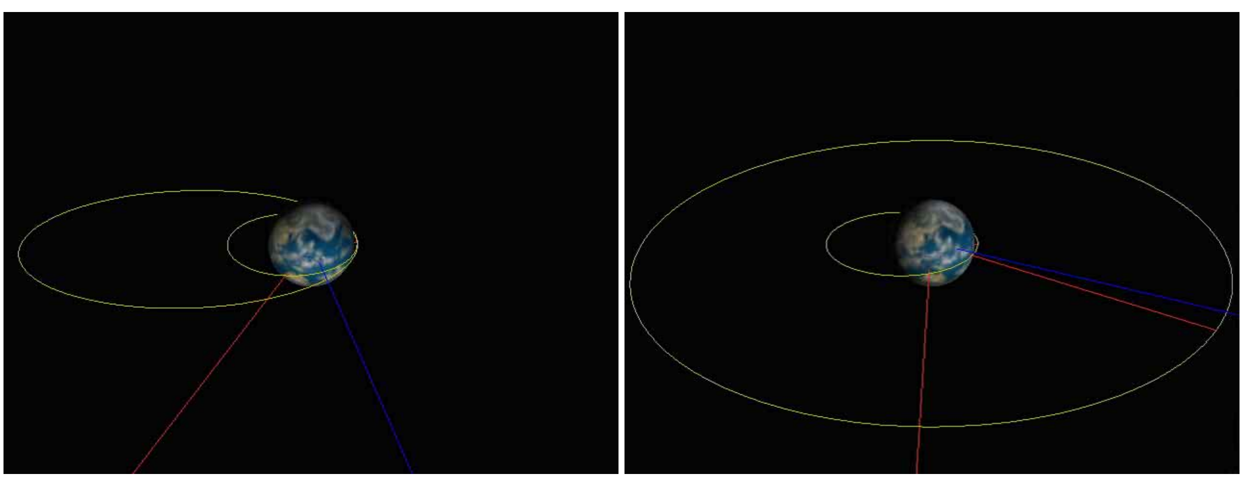

Figure 1. LEFT: Orbital geometry of tether system immediately after payload toss. The inner yellow ellipse is the facility's post-toss orbit, and the outer ellipse is the payload's geo-transfer orbit. The blue line is the vector to the sun, and the red line is the vector to the moon. **RIGHT:** Desired orbital geometry of the tether system at the end of reboost. The outer yellow circle is the payload's geostationary orbit, and the inner ellipse is the facility's high-energy orbit.

hollow-cathode plasma contactors at both ends of the conducting tether, so that it could carry current in both directions. Thrusting was performed when the tether was under 2000 km of altitude. Peak current levels were limited to 30 A, with typical currents varying between 25 and 30 A. In addition, thrusting was performed only when the tether was within $\pi/4$ of vertical.

Table 1: System Design #4:

LEO ⇒ GTO Tether Boost Facility With 1/48 hr Payload Rendezvous Opportunities

System Masses		
Tether mass	14,510	kg
Control station mass	20,210	kg
Grapple mass	250	kg
Total Facility Mass	***34,720.2***	kg
Payload Mass	***5,000***	kg

Tether Characteristics		
Tether Length	75,000	m
Tether mass ratio	2.902	
Tether tip velocity at catch	1,331	m/s
Tether tip velocity at throw	1,164	m/s
Tether angular rate	0.022	rad/s
Gravity at Control Station	1.105	g
Gravity at payload	2.621	g

Positions & Velocities		***Pre-Catch*** ***Payload***	***Pre-Catch*** ***Tether CM***	***Joined System Post-catch CM***	***Post-Toss*** ***Tether CM***	***Post-Toss*** ***Payload***
resonance ratio		36	17		1	4.2
perigee altitude	km	325	385	378	370	431
apogee altitude	km	325	8938	7192	5729	35786
perigee radius	km	6703	6763	6756	6748	6809
apogee radius	km	6703	15316	13570	12107	42164
perigee velocity	m/s	7711	9042	8876	8709	10040
apogee velocity	m/s	7711	3993	4419	4855	1621
CM dist. From Station	m		14698	22241	14698	
CM dist. To Grapple	m		60302	52759	60302	
ΔV to Reboost	m/s				333	
ΔV to Correct Apogee	m/s					0
ΔV to Correct Precess.	m/s					0.00
ΔV To Circularize	m/s					1453
Basic Orbital Parameters						
semi-major axis	km	6703	11040	10163	9428	24486
eccentricity		0.0	0.387	0.335	0.284	0.722
inclination	rad	0	0	0	0	0
semi-latus rectum	km	6703	9383	9021	8666	11724
sp. mech. energy	m^2/s^2	-2.97E+07	-1.81E+07	-1.96E+07	-2.11E+07	-8.14E+06
vis-viva energy	m^2/s^2	-5.95E+07	-3.61E+07	-3.92E+07	-4.23E+07	-1.63E+07
period	sec	5462	11544	10196	9110	38133
period	min	91.0	192.4	169.9	151.8	635.5
station rotation period	sec		284.7	284.7	284.7	
rotation ratio			40.6	35.8	32.0	

Reboost Parameters		
Solar Power	225	kW
Max. Power During Thrusting	1000	kW
Energy Storage System Efficiency	90	%
Energy Storage System Capacity	600	kW/hr

Results

Figure 2 shows the orbit semimajor axis and eccentricity during the three days of boosting. It also shows the thrust efficiency calculated during periods of thrusting. Figure 3 shows the altitudes of the orbit perigee and apogee, as well as the perigee velocity.

Orbit Boost Rates:

The semimajor axis increases at 42.9 km/day. Note that if the electrodynamic boost system adds energy to the orbit at a constant rate, the rate of semimajor axis increase will accelerate due to the inverse relation between orbital energy and semimajor axis.

The eccentricity increases at 0.0026/day. Note that the eccentricity change rate will also vary during

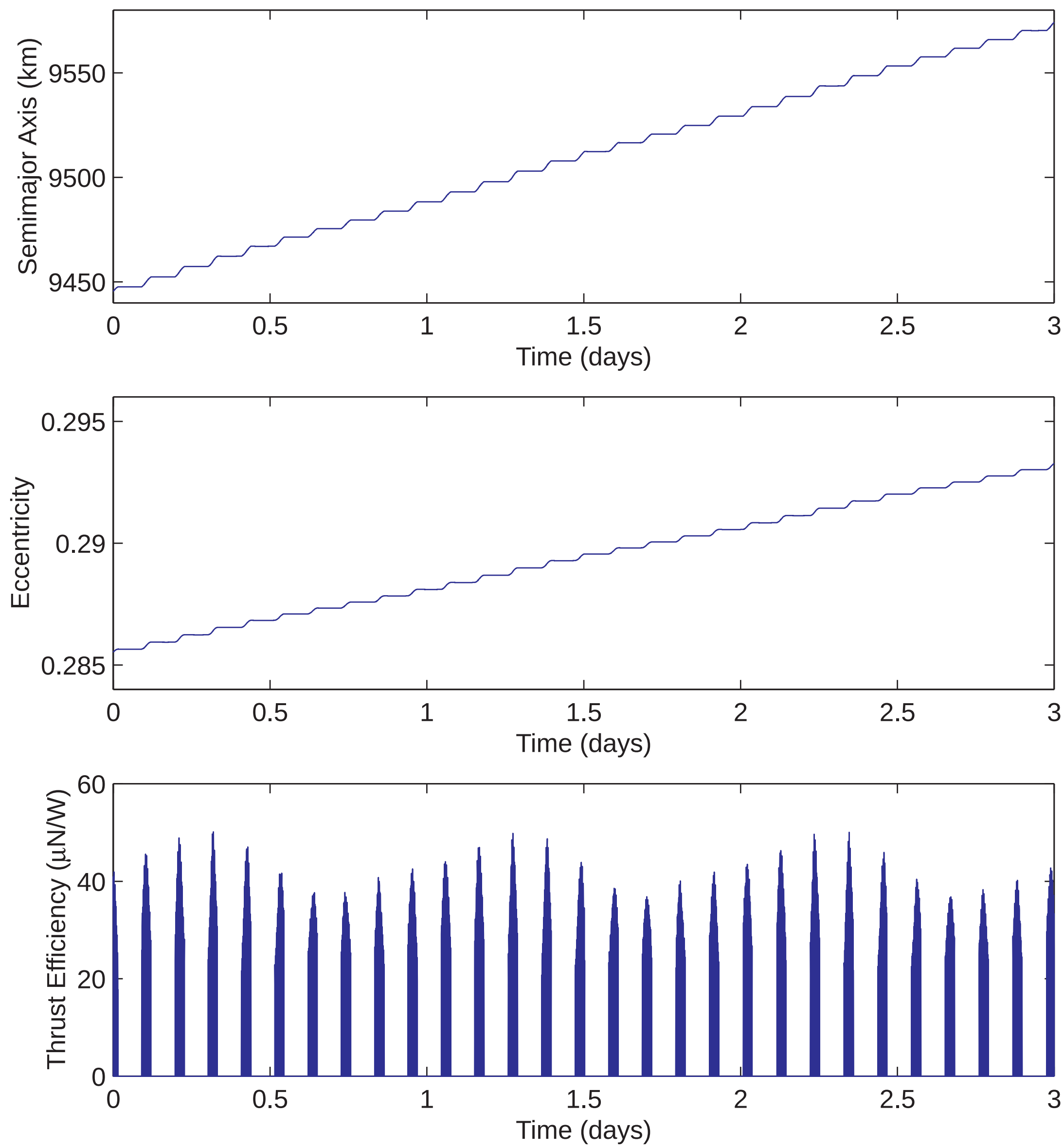

Figure 2. Semimajor axis, eccentricity, and thrust efficiency during the first three days of the reboost operation.

reboost.

Thrust Efficiency:

The thrust efficiency is shown in the bottom graph of Figure 2. The graph shows that the thrust efficiency varies cyclically during each day; this variation is due to the fact that the Earth, and its magnetic field, are rotating inside the facility's orbit, and thus the angle between the geomagnetic field's axis and the orbit plane varies once per day. In addition, not apparent on this timescale, the thrust efficiency varies with altitude and with the angle of the tether relative to local vertical. Over this three day period, the average thrust efficiency is 33.5 μN/W (thrust efficiency calculated using the power input to the electrodynamic tether).

Reboost Time: Since the rate of semimajor axis increase varies during the reboost operation, the best way to estimate the time needed to reboost the orbit is to assume that the rate at which the orbital energy of the system is increased is relatively constant during the reboost period. To reboost the orbit from 370x5732 km to 385x8938 km, the electrodynamic system must restore 100 GJ of energy to the tether facility's orbit. In the 3-day simulation, the electrodynamic thrusting restored the facility's orbital energy

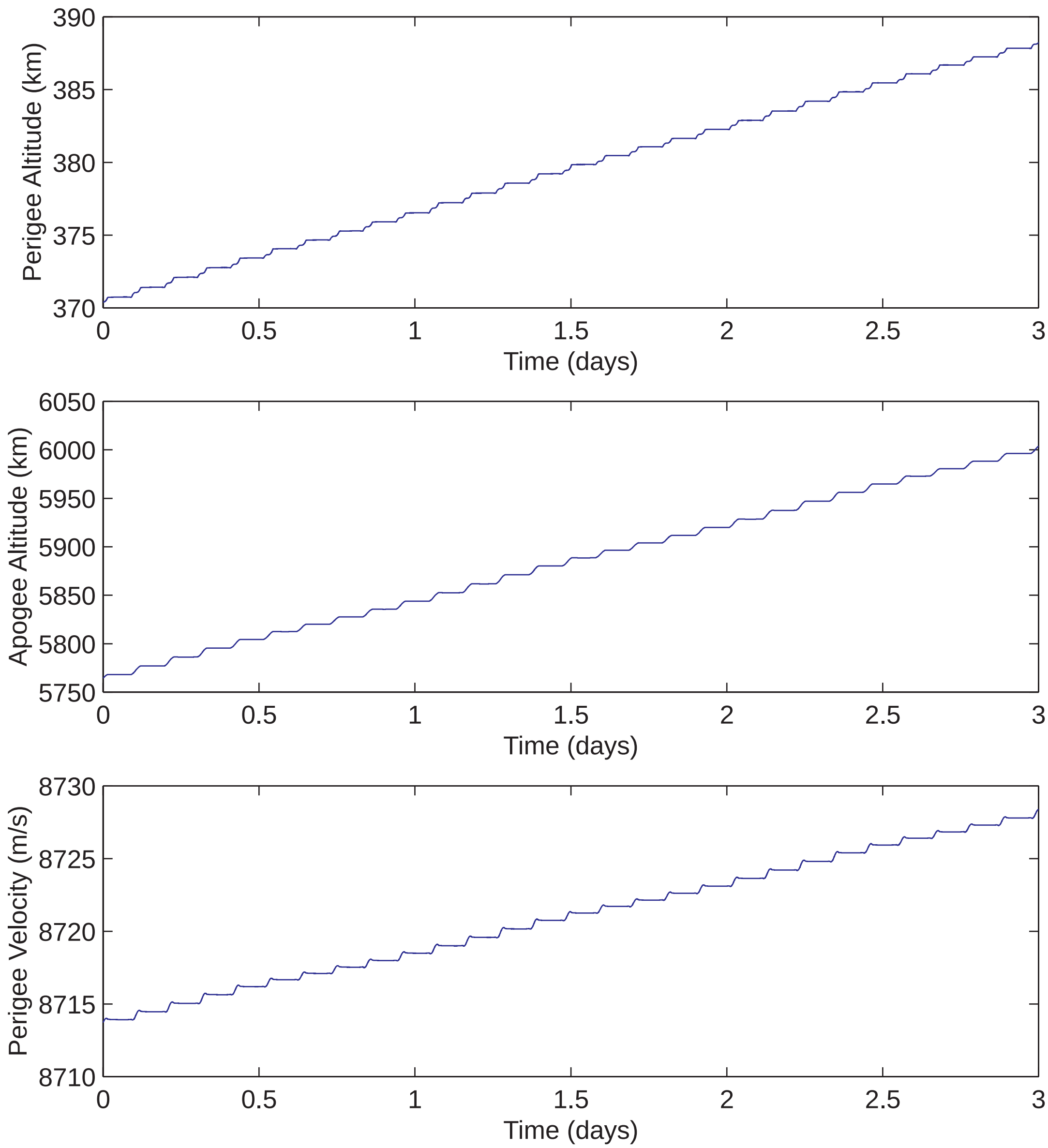

Figure 3. Perigee and apogee altitude, and perigee velocity during the first three days of the reboost operation.

at a rate of 3.3 GJ/day. Thus the system will reboost the orbit in approximately one month.

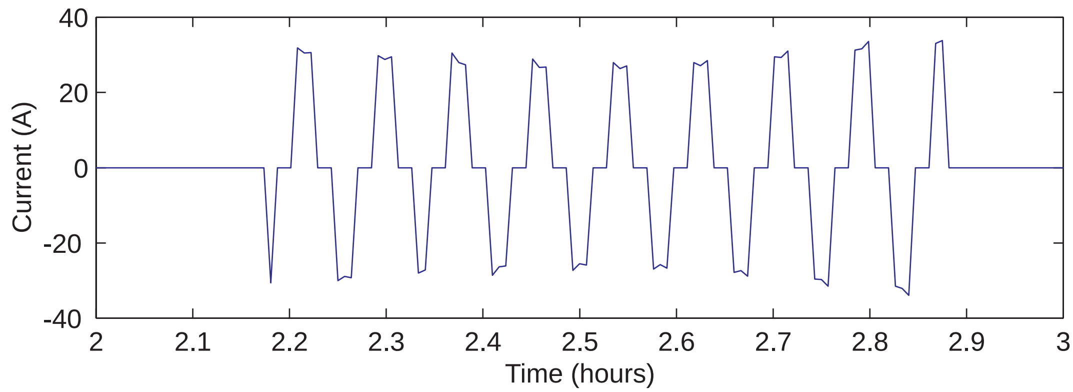

Figure 4. Current driven through the electrodynamic tether during the second perigee pass.

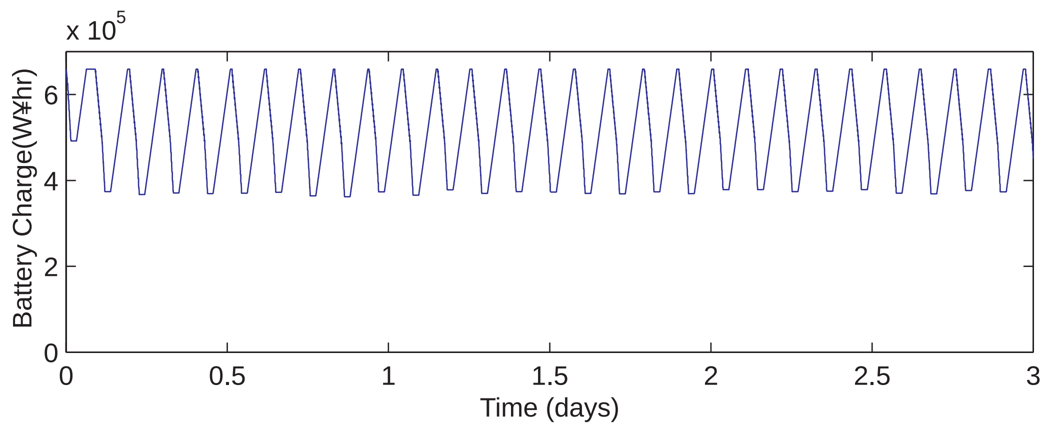

Figure 5. Charge level of the energy storage system. The solar panels collected 225kW during the portion of the orbit that the facility was illuminated by the sun. Energy storage system efficiency was assumed to be 90%.

<u>Energy System:</u>

The charge level of the energy storage system (batteries or flywheels) over the three days is shown in Figure 5. With the solar power supply generating 225 kW during the portions of the orbit that the tether facility is illuminated, and stored at an efficiency of 90%, the system maintains its energy balance and the depth of charge never drops below 50%.

Achieving the Desired Final Orbit:

In order to maximize the payload throughput capacity of a tether boost facility, it is desirable to perform electrodynamic thrusting whenever the magnetic and plasma conditions allow it. The tether boost facility will be located in or near the equatorial plane. At these low inclinations, the ionospheric plasma has a "bulge" which significantly enhances the plasma density compared to inclined orbits. As a result, it will likely be possible to perform significant thrusting out to 2000 altitude. Figure 6 illustrates the geometry of the tether facility's orbit, and the portion of the orbit in which thrusting can be performed.

Figures 2-5 show results for reboosting operations in which the tether current is controlled to achieve a net average thrust that is perpendicular to the vector between the facility's center of mass and the center of the Earth. This method maximizes the efficiency of adding both energy and momentum to the tether facility's orbit. Because thrusting is performed during periods when the tether facility is well away from perigee, however, this thrusting method results in significant boosting of the perigee altitude. Figure 3 shows that the perigee altitude is raised above the desired final value of 385 km in less than 3 days. Consequently, it will be necessary to modify the reboost method in order to achieve the desired final orbit shape.

The orbit eccentricity is plotted as a function of the semimajor axis in Figure 7. With no adjustment to the reboost program, the eccentricity increases roughly as $e \propto 6e\text{-}8\ a$. If this relationship holds during the entire reboost maneuver, by the time the semimajor axis has been restored to 11,040 km, the eccentricity will be approximately 0.3807, and the perigee altitude will be approximately 74 km too high.

There are several methods that could be used to enhance the eccentricity of the orbit while still achieving the same reboost time. First, the tether facility could perform thrusting at a higher level near perigee and at a lower level when it is away from perigee. Second, the tether facility could perform tether reeling operations to increase the orbit eccentricity. Third, the tether facility could vary the direction of the net thrust during its passage through the ionosphere. In the following paragraphs we will describe

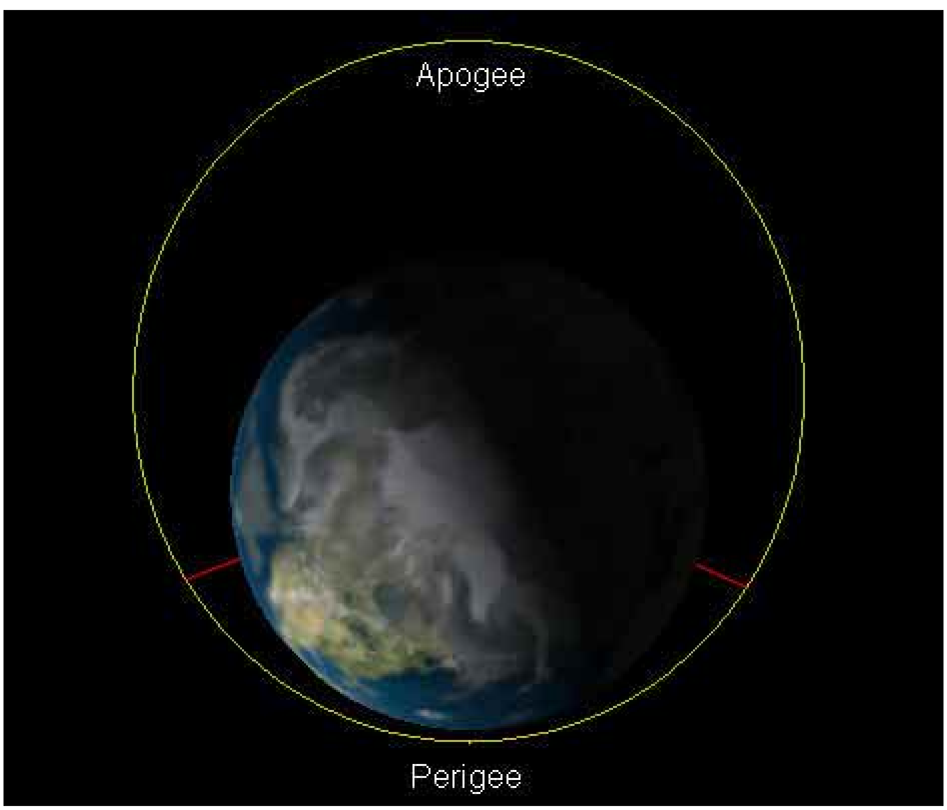

Figure 6. Geometry of the tether facility's orbit after it has boosted a payload. The red lines indicate the limits of the portion of the orbit during which the facility altitude is below 2000 km and electrodynamic thrusting is likely to be effective.

these methods in more detail.

Variation of Thrust Power:

This method is relatively obvious, and easiest to perform in theory. Thrusting while the tether is near perigee boosts the apogee but not the perigee. However, this method would require the system to operate at higher thrust levels at perigee, which is undesirable considering the already high-power level of 1 MW that has been baselined.

Tether Reeling:

A second method of adjusting the orbit shape would be to perform tether reeling maneuvers to add energy to the orbit without adding orbital momentum, thus increasing the orbit eccentricity.[1] To boost the orbit eccentricity, the tether system would reel in the tether while it is near perigee and the gravity gradient forces are high, and allow it to deploy when it is at apogee and the gravity gradient forces are low.

In order to achieve the desired final eccentricity of 0.387, the tether reeling must add eccentricity to the orbit at a rate of approximately 2.2e-4 per day. If the distance from the tether facility's control station to the center of mass of the tether is L, and the control station reels the tether in and out according to a program $\Delta L(t)$, the rate of eccentricity change is approximated by:

$$\frac{d}{dt}e = \frac{3\sqrt{\mu}}{a^{7/2}}\left[\frac{m_{12}}{m}\right]\left\{2\cos(\omega_{orb}t)\sin(\omega_T t)\cos(\omega_T t) - \sin(\omega_{orb}t)\left(1-\frac{3}{2}\sin^2(\omega_T t)\right)\right\}\left[L+\Delta L(t)\right]^2. \quad (1)$$

where $\mu = GM_e$, a is the orbit semimajor axis, ω_t is the tether rotation rate, ω_{orb} is the orbital rate, m is the total mass of the tether system, and m_{12} is the reduced mass of the system. Integrating this equation over an orbit, we find that if the tether control station reels the tether in and out sinusoidally once per orbit, with an amplitude of 1 km, the rate of eccentricity change is approximately 2.2e-4. The peak reeling rate would be approximately 0.7 m/s. The tether tension at the control station end of the tether is approximately 157,000 N. The power required to reel the tether in during perigee thus would be 110 kW. Although this power could be recovered when the tether is unreeled at apogee, this scheme would still require the tether facility to process a higher level of power during the perigee passage.

Thrust Vector Variation

The third method takes advantage of the fact that the tether is rotating, and thus the direction of the electrodynamic thrust varies relative to the direction of motion. Rather than always thrusting when the tether is near a local vertical orientation, the tether boost facility can instead vary its current to apply thrust along the –**r** direction on its inbound trajectory, along the velocity vector when the system is near perigee, and along the +**r** direction on its outbound trajectory, as illustrated in Figure 8. This adds energy to the orbit without adding as much orbital momentum. The green plot in Figure 7 shows the increase of eccentricity with the semimajor axis using this thrusting program. The variation of eccentricity with the semimajor axis is increased to $e \propto 6.7\text{e-}8\ a$, which is more than enough to achieve the desired final orbit eccentricity. Because this method does not require additional power capability nor tether reeling capability, it is the preferred method for tuning the rates of eccentricity and semimajor axis boosting to achieve the desired final orbit.

1. Hoyt, R. P., "Maintenance Of Rotating Tether Orbits Using Tether Reeling", Appendix F in *Cislunar Tether Transport System*, Tethers Unlimited, Inc. Final Report on NASA Institute for Advanced Concepts Contract NIAC-07600-011.

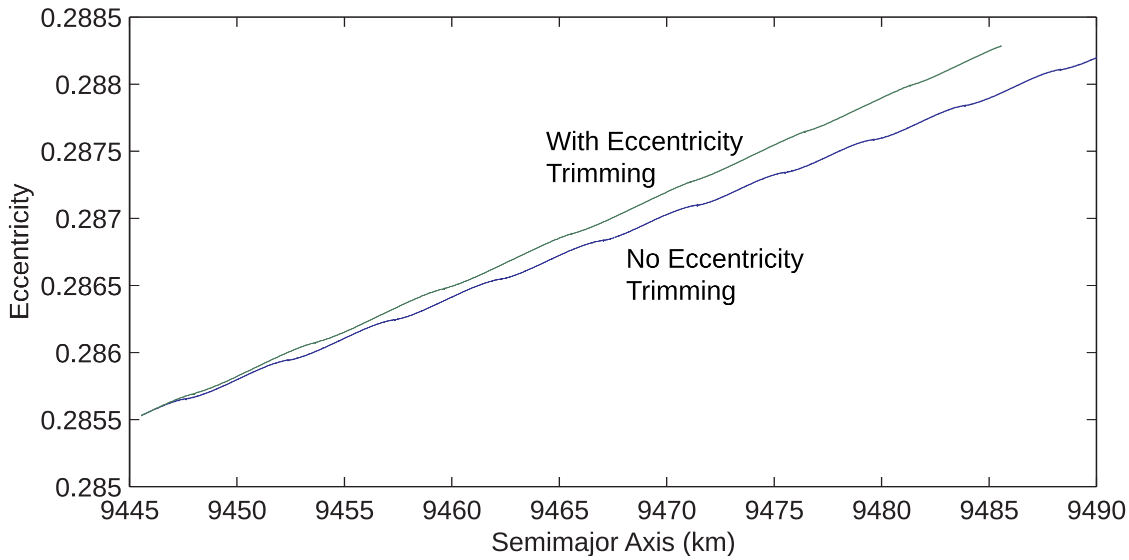

Figure 7. Plots of eccentricity versus semimajor axis with and without the eccentricity trimming. Without the eccentricity enhancement, the eccentricity increases as e ∝ 6e-8 a, and with the enhancement it increases as e ∝ 6.7e-8 a.

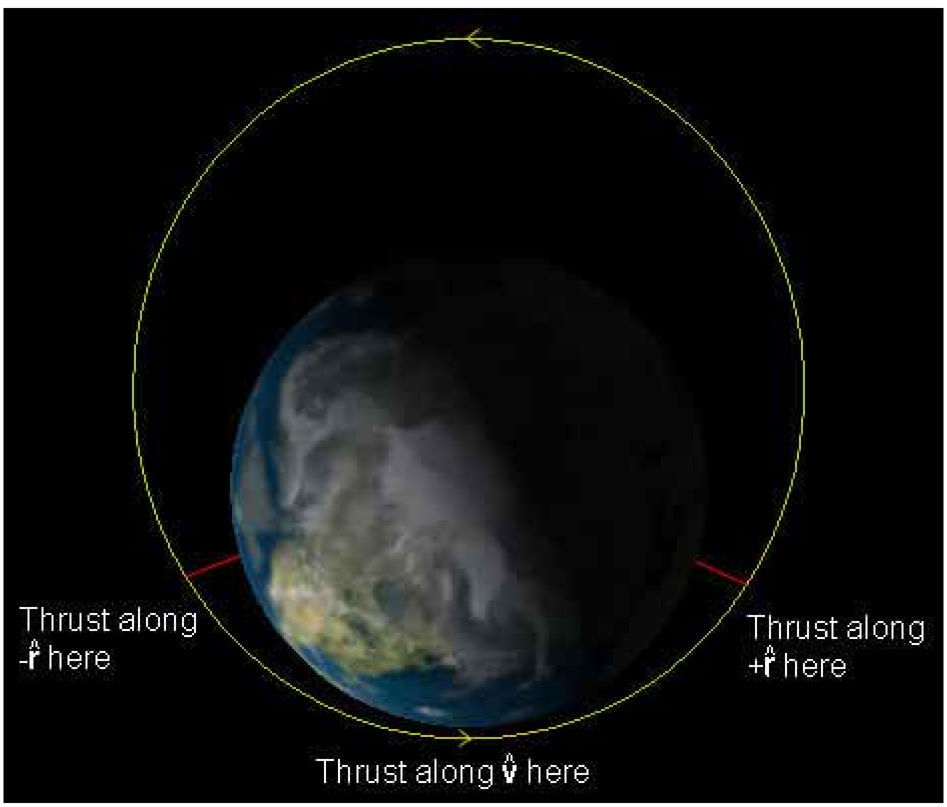

Figure 8. Method for adjusting orbit boosting to achieve the desired orbital shape.

Appendix L

Paper Submitted to the 2002 Space Technologies and Applications International Forum

The µTORQUE Momentum-Exchange Tether Experiment

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Abstract. Long, high-strength tethers can provide a mechanism for transferring orbital momentum and energy from one space object to another without the consumption of propellant. By providing a highly-reusable transportation architecture, systems built upon such "momentum-exchange" tethers may be able to achieve significant cost reductions for a number of in-space propulsion missions. Before such systems could be placed into operation, however, a number of technical challenges must be met, including flight demonstration of high-strength, highly survivable tethers, demonstration of the ability to control the dynamics of a rotating tether system, and the ability for a tether system to rendezvous with, capture, and then toss a payload. In this paper, we discuss a concept design for a small momentum exchange tether experiment that is intended to serve as the first step in demonstrating these key technologies. The "Microsatellite Tethered Orbit Raising QUalification Experiment" (µTORQUE) will be designed to fly as a secondary payload on an upper stage of a rocket used to deliver a satellite to GEO. The µTORQUE experiment will remain on the upper stage left in a GTO trajectory. After the primary satellite has been deployed into GEO, the µTORQUE experiment will deploy a microsatellite at the end of a 20 km long tether. Utilizing tether reeling and/or electrodynamic propulsion, the µTORQUE system will set the tether in rotation around the upper stage, accelerating the rotation until the tip velocity is approximately 400 m/s. The experiment will then release the microsatellite when the system is at its perigee, tossing the payload into a near-minimum-energy transfer to the Moon. The microsatellite can then utilize a Belbruno weak-boundary trajectory to transfer into a lunar orbit using only a few m/s of delta-V. Preliminary analyses indicate that the tether system could be mass-competitive with a chemical propellant system for the same mission.

INTRODUCTION

Momentum-Exchange/Electrodynamic-Reboost (MXER) tethers have strong potential for providing a reusable in-space propulsion capability that can dramatically reduce the cost of many space missions (Hoyt, 2000b; 2000c; Sorensen 2001). In order for these concepts to progress towards operational service, however, flight experiments must be carried out to develop and demonstrate the key technologies needed for these systems. In this paper, we will first briefly review the concepts of momentum exchange and electrodynamic tether propulsion, describe two previous in-space demonstrations of momentum exchange, and then discuss the key technologies required for MXER systems. We will then discuss a concept for a small, low-cost flight experiment intended to perform risk reduction demonstration of several of these key technology needs.

Background: Momentum-Exchange Tethers

In a momentum-exchange tether system, a long, thin, high-strength cable is deployed in orbit and set into rotation around a central body. If the tether facility is placed in an elliptical orbit and its rotation is timed so that the tether is oriented vertically below the central body and swinging backwards when the facility reaches perigee, then a grapple assembly located at the tether tip can rendezvous with and capture a payload moving in a lower orbit, as illustrated in Figure 1. Half a rotation later, the tether can release the payload, tossing it into a higher energy orbit. This concept is termed a momentum-exchange tether because when the tether picks up and tosses the payload, it transfers some of its orbital energy and momentum to the payload, resulting in a drop in the tether facility's apogee.

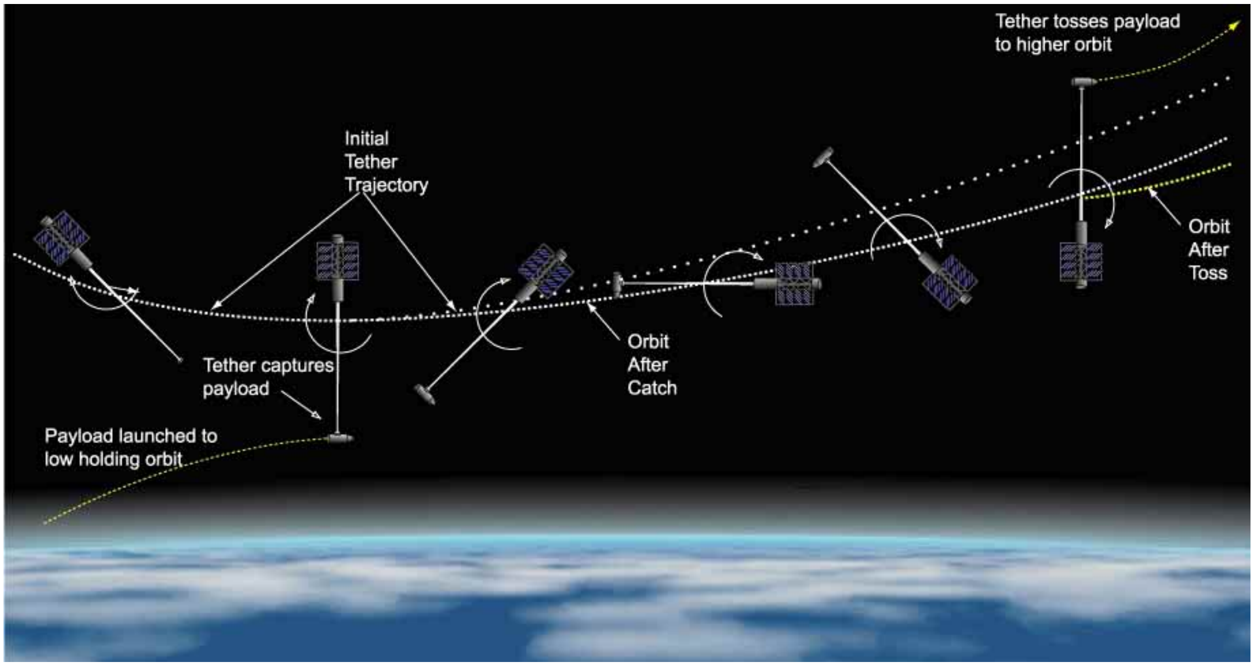

Figure 1. Concept of operation of a momentum-exchange tether facility. Orbits are depicted conceptually from the perspective of an observer on the Earth.

Electrodynamic Reboost

In order for the tether facility to boost multiple payloads, it must have the capability to restore its orbital energy and momentum after each payload transfer operation. If the tether facility has a power supply, and a portion of the tether contains conducting wire, then the power supply can drive current along the tether so as to generate thrust through electrodynamic interactions with the Earth's magnetic field. By properly controlling the tether current during an orbit, the tether facility can reboost itself to its original orbit (Hoyt, 2000a; 2001). The tether facility essentially serves as a large "orbital energy battery," allowing solar energy to be converted to orbital energy gradually over a long period of time and then rapidly transferred to the payload.

Key Advantages

A tether transportation system has several advantages compared to conventional and other advanced in-space propulsion systems:

1) *(Near) Zero Propellant Usage* Chief among these advantages is the ability to eliminate the need for propellant expenditure to perform payload transfers. Of course, some propellant expenditure will be needed for trajectory corrections and rendezvous maneuvering, but these requirements will be very small, a few tens of meters per second. The ability to cut several thousands of meters per second from the ΔV needed to deliver a payload to its destination can enable customers to utilize much smaller launch vehicles than would be required with a rocket-only system, greatly reducing total launch costs. For example, launching a 5 metric ton satellite into GEO, would require a Delta IVM+ (4,2) launch vehicle using an all-chemical propulsion system, at a cost exceeding $90M. Using a tether facility, the payload could instead be launched into LEO using a much smaller Dnepr 1 (RS-20) launch vehicle, at 1/7th the cost of the Delta launch.

(2) Short Transfer Times A momentum-exchange tether system provides its ΔV to the payload in an essentially impulsive manner. Thus the transfer times in a tether system are very short, comparable to rocket-based systems. This can be compared with electric propulsion schemes, which offer low propellant usage, but invariably require long transfer times due to their low thrust levels. The short transfer times offered by a momentum-exchange tether

system can play an important role in minimizing the lost-revenue time that a commercial satellite venture would have to accept while it waits for its satellite to reach its operational orbit and begin generating revenue.

3) Reusable Infrastructure Once deployed, a tether boost facility could transfer many, many payloads before requiring replacement. Thus the recurring costs for payload transport could be reduced to the cost of operations. A tether transportation system thus would be somewhat analogous to a terrestrial railroad or public-transit system, and might achieve comparable cost reductions for transporting many payloads.

4) *Fully Testable System* Another important but often overlooked advantage of a tether transportation system is that the components that perform the actual payload transfer operations can be fully tested *in space operations* before being used for critical payloads. In conventional rocket systems, engine components and other key elements can be tested on the ground, and many individual units can be flown to provide reliability statistics, but to date only the Shuttle has re-used rocket engines, with significant maintenance after each flight. In a tether transportation system, the tether facility could be tested many times with “dummy” payloads – or, better yet, with low inherent-value payloads such as water or fuel – to build confidence for use on high value or manned payloads. In addition, "using" a tether does not damage or "wear it out", as long as the loads placed on the tether do not approach the yield point of the tether material. This means that the tether used in the operational system is the same tether in nearly the same condition in which it underwent strength and reliability testing with the "dummy" payloads.

Previous Demonstration Missions

The use of space tethers to transfer orbital momentum and energy from one spacecraft to another has been demonstrated in a rudimentary fashion at least twice in the past, once intentionally, the other as a serendipitous outcome of a premature mission termination. In the SEDS-1 mission, a small payload was deployed below a Delta-II upper stage at the end of a 20 km long Spectra tether. The physical connection of the payload to the upper stage by the tether forced the payload to orbit the Earth with the same velocity as the upper stage. At the payload’s location, 20 km closer to the Earth, however, this velocity was less than that required for the payload to remain in orbit. After completion of the deployment, the tether was released from the upper stage. This dropped the payload into a suborbital trajectory that re-entered the Earth’s atmosphere half an orbit later (Smith, 1995). In the Tethered Satellite System Reflight Experiment carried out on the Shuttle orbiter in 1996, a satellite was deployed upwards from the Shuttle at the end of a 20 km conducting tether. Unfortunately, a flaw in the tether’s insulation allowed an arc to jump from the tether to the deployment boom, causing the tether to burn and separate near the Shuttle. Although this ended that experiment prematurely, it did unintentionally demonstrate momentum exchange, because after the tether was cut, the satellite was injected into an orbit with an apogee approximately 140 km higher than the Shuttle’s orbit.

Technology Needs

Although momentum-exchange/electrodynamic reboost tethers have strong potential for achieving significant cost reductions for a wide range of space missions, and many of the core technologies are available or at a high technology readiness level, as a system-level propulsion technology MXER tethers are currently at a relatively low TRL level. Several key challenges must be met before MXER tethers can be considered for operational use. NASA's 2000 H/READS Strategic Research and Technology Road Map for Space Transportation identified the following four technology elements key to the success of MXER tethers:

1. Highly accurate prediction and control of the tether dynamics associated with catching and tossing a payload
2. Integrated high-strength electrodynamic tethers, and improved modeling and control algorithms for electro-dynamic thrusting.
3. Efficient orbital propagators able to accurately model all of the perturbative effects on rotating space tethers, as well as methods for obtaining highly accurate orbital knowledge of tethers and their payloads.
4. Low mass, inexpensive, and reliable methods for catching payloads.

The previous flight demonstrations of momentum exchange did not address any of these issues. Consequently, in order for MXER tether concepts to advance towards operational capability, further flight demonstrations must be carried out to develop and prove these key technologies.

THE μTORQUE EXPERIMENT

In order to begin addressing these key technical challenges, we propose to develop a very small momentum-exchange tether system capable of boosting a microsatellite by a ΔV of 0.4 km/s. This "Microsatellite Tethered Orbit Raising QUalification Experiment" (μTORQUE) system will be sized to fly, along with its microsatellite payload, as a secondary payload on an upper stage rocket such as the SeaLaunch Block DM 3rd Stage. The primary goal of the μTORQUE system will be the development and low-cost demonstration of key technologies for MXER tether architectures.

The μTORQUE concept is illustrated in **Figure 2**. The μTORQUE tether system and a microsatellite payload would be integrated onto a rocket upper stage prior to launch. After the stage releases its primary payload into GTO (1), the μTORQUE system would deploy the microsatellite from the stage at the end of a high-strength conducting tether (2). The system would then use electrodynamic-drag thrusting during several successive perigee passes (3), to spin-up the tether system. This would effectively convert some of the upper stage's orbital energy into system rotational energy. Because the system utilizes electrodynamic drag to perform the spin-up of the system, it will not require the mass and complexity of a dedicated solar power supply; the system can also power its own avionics utilizing the power generated by the tether. When the tether tip velocity reaches 0.4 km/s, the μTORQUE system could then release the payload during a perigee pass (4), injecting the payload into a minimum-energy lunar transfer trajectory (5). With a 0.4 km/s ΔV capability, the μTORQUE tether system could also be useful for missions such as deploying microsatellites into high-LEO and MEO orbits as secondary payloads on launches of larger satellites into LEO.

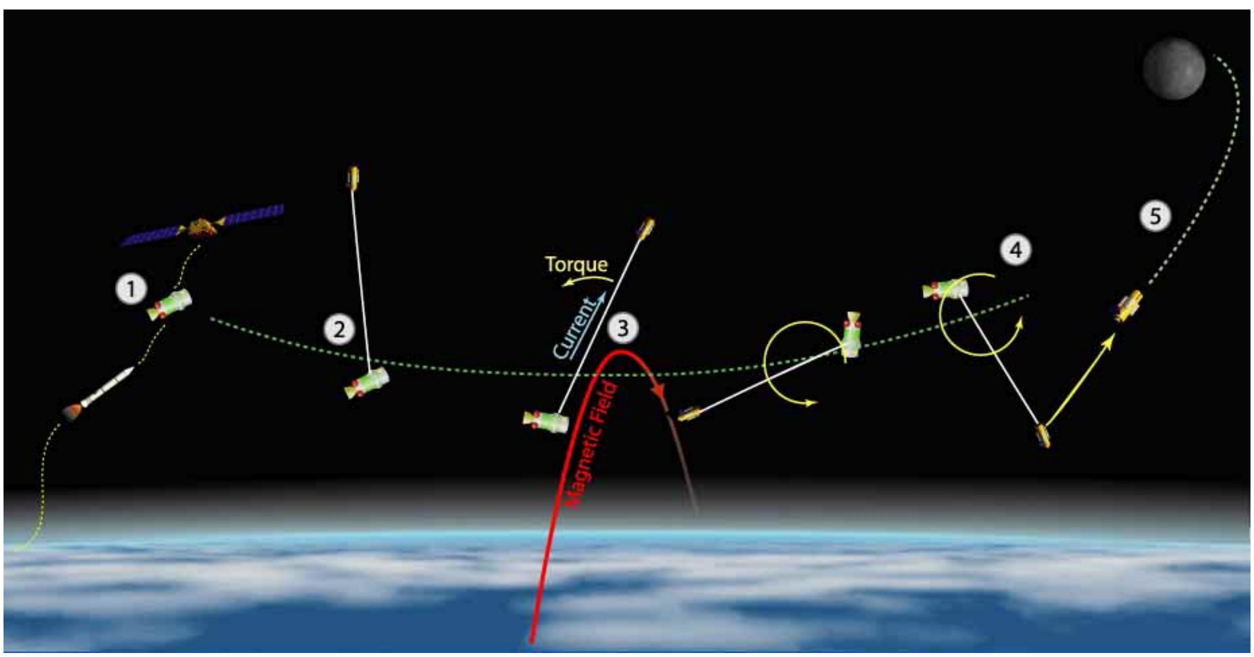

Figure 2. The "Microsatellite Tethered Orbit-Raising Qualification Experiment (μTORQUE)" concept.

System Concept

The μTORQUE system would be composed of:

- A 20 km long high-strength tether; this tether would have an integrated conductor, enabling it to carry currents for generation of electrodynamic forces. The tether would utilize an interconnected, multiline structure to ensure that it survives the orbital debris environment for the duration of the mission.
- A small, simple deployment system. The microsatellite would be ejected away from the upper stage, pulling the tether out of a deployer canister.
- An avionics package, to control deployment and operation of the tether, and to sense and control tether dynamics.
- An electron emission system, such as a hollow-cathode plasma contactor or a Field Emission Array Cathode, located on the upper stage side of the tether.
- A commandable release mechanism, enabling the tether system to release the microsatellite payload into its transfer orbit.

Preliminary System Sizing

The mass of a tapered rotating tether with tip speed V_t depends upon ratio of the tip speed to the tether material's characteristic tip speed: $V_c = \sqrt{2T/Fd}$, where T is the tensile strength of the material, F is the design safety factor, and d is the material density. For Spectra 2000, the best fiber presently available in quantity, T = 4 GPa, d = 0.97 g/cc, and thus V_c = 1.8 km/s for a safety factor of F = 2.5. In an unpublished paper, Moravec found that the mass of a tapered tether depends upon the tip mass (payload) and the tip velocity according to (Moravec 1978):

$$M_T = M_p \sqrt{\pi} \frac{\Delta V}{V_c} e^{\frac{\Delta V^2}{V_c^2}} erf\left\{\frac{\Delta V}{V_c}\right\}$$, where erf(x) is the error function of x.

Using a stepwise-tapered approximation of this ideal tapering, we have developed a preliminary design for a 0.4 km/s momentum-exchange tether system sized to boost an 80 kg microsatellite from a GTO trajectory to a minimum-energy lunar transfer orbit (LTO), using a 20 km long tether:

Tether Mass: (kg)	10.5	(2 kg of which is conductor)
Avionics & Emitter Mass: (kg)	5.5	
Deployer Mass (kg)	3.5	Max acceleration on payload: 0.83 gees
Ejection Mechanism Mass (kg)	0.5	
Total (kg)	20.0	

The initial experimental version of the system will likely include significant diagnostics for performance and dynamics verification, which will add to the system mass, but an operational version of this system with a mass on the order of 20 kg should be feasible. This (estimated) tether system mass is approximately 25% of the 80 kg microsatellite mass.

The primary objective of the proposed µTORQUE effort would be to develop a small, low-cost method for demonstrating many of the key technologies required for larger MXER tether facilities for boosting communications satellites to GTO and scientific payloads to the Moon. Using a Belbruno Weak-Boundary Transfer technique, the system may be capable of placing small payloads into lunar orbits without the need for a capture burn (Belbruno, 2000). Alternatively, it could place microsatellites into lunar-flyby-to-escape trajectories. The µTORQUE system as defined above will provide a testbed to demonstrate technologies for meeting the first three key technologies listed above (dynamics modeling, high-strength conducting tethers, and orbital propagation and sensing capabilities).

If the initial payload toss demonstration is successful, the µTORQUE system can then be augmented for a second flight demonstration to validate the fourth key technology, rendezvous and capture capability. In this second test, a grapple mechanism would be integrated at the tip of the tether. The µTORQUE experiment could then fly as a secondary payload on an upper stage that is placed into a LEO trajectory. The tether system could then be used to catch and toss a microsatellite payload, providing it with 800 km/s of total ΔV.

In addition, the μTORQUE effort will result in a small propulsion system that could be competitive with chemical propulsion for missions such as boosting secondary payloads from GTO drop-off orbits to lunar transfer or to other high-energy trajectories. A chemical-rocket stage sized to boost a microsatellite from GTO to LTO would require a propellant mass of approximately 20% of the microsatellite mass. When the necessary avionics and thruster hardware are included, a chemical-based system would likely have a mass penalty of approximately 25%, roughly equal to the (estimated) tether system mass penalty. A rocket system, however, could boost only one microsatellite. The μTORQUE system could be configured to deploy multiple payloads with zero or minimal additional mass requirements.

CONCLUSIONS

Momentum-Exchange/Electrodynamic-Reboost tether systems have strong potential for reducing the cost of in-space transportation, but several key technology challenges must be addressed before they can enter operational service. Given the large expense of conducting space demonstrations, and the relatively small amount of funding available for the development of advanced space propulsion technologies, we have sought to design a very small, affordable experiment that can achieve a significant advance in technology demonstration and risk reduction while performing a technically and scientifically significant propulsion mission. The μTORQUE concept can be flown as a secondary payload on a GEO satellite launch, enabling it to be conducted with relatively low launch costs. With a 100 kg total secondary payload mass allocation, the tether system can be sized to boost approximately 80 kg into a lunar transfer trajectory, and thus could deliver a significant science microsatellite to the Moon.

ACKNOWLEDGMENTS

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Tether Transport System Dynamics Verification Through Simulation

Robert P. Hoyt
Tethers Unlimited, Inc.

Abstract

In order to validate the orbital mechanics and tether dynamics of various tether transport system architectures, we have developed a numerical simulation of the system that includes models for the full 3D orbital mechanics in the Earth-Moon system, tether dynamics, tether electrodynamics, and other relevant physics. Using this code, we have designed and simulated scenarios for transferring payloads from LEO to GEO, from LEO to the lunar surface, and from suborbital trajectories into Earth orbit.

Introduction

The operation of the tether facilities utilized in the tether transport systems such as the Cislunar Tether Transport System and the Mars-Earth Rapid Interplanetary Tether Transport (MERITT) system involve many different interrelated phenomena, including orbital dynamics, tether librations and oscillations, interactions with the ionospheric plasma, day/night variations of the ionospheric density, solar and ohmic heating of the tether, magnetic vector variations around an orbit, and the behavior of electron emission devices. In order to enable accurate analyses of the performance and behavior of the this and other tether system, we have developed a numerical simulation of electrodynamic tethers called "TetherSim" that includes models for all of the aforementioned physical phenomena. The TetherSim code is implemented in C++, and runs on BSD Unix and MacOS platforms.

The simulation is capable of simultaneously simulating multiple tethers of different types, satellites, payloads, and other vehicles. In the following sections we summarize the physics models used in the TetherSim program.

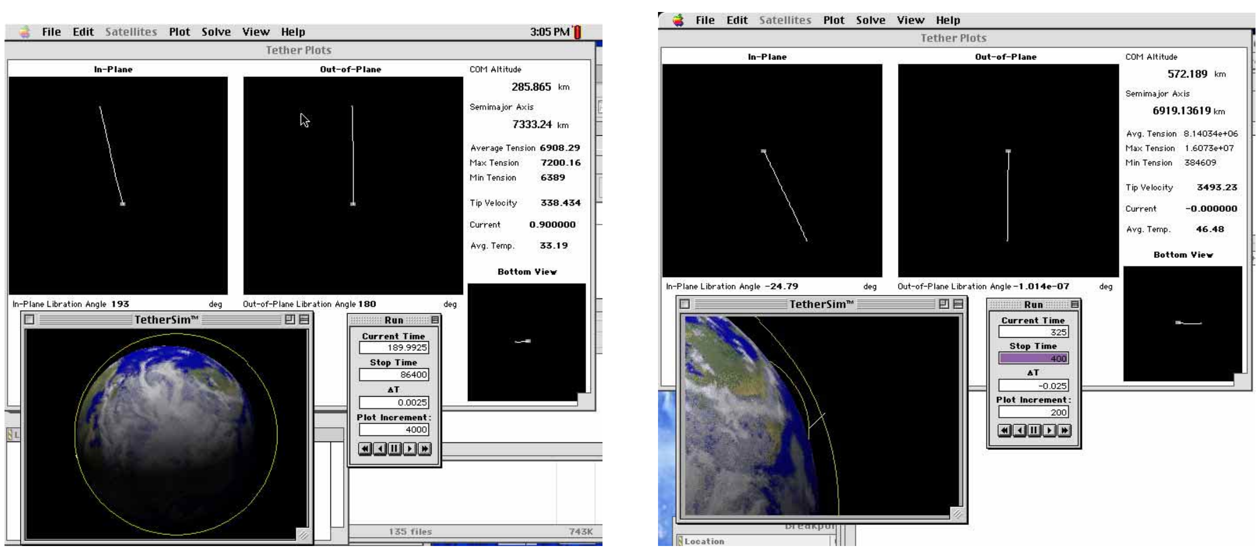

Figure 1. Screen captures of the TetherSim program simulating orbital reboosting of a 25 km HEFT Tether Facility (left) and simulating rendezvous between a 600 km long orbiting tether and a hypersonic airplane (right).

Tether Dynamics

TetherSim™ can utilize two different algorithms for propagating the dynamics of the tether, one a Runge-Kutta-based explicit algorithm, the other an implicit finite-element based algorithm. In both of the algorithms, the continuous tether mass is approximated as a series of point masses linked by massless springs. In the explicit algorithm, the forces on each of the point masses are calculated and summed, and Runge-Kutta integration is used to advance their positions over a timestep. Explicit propagation of tether dynamics, however, requires the use of extremely small timesteps. Stability of an explicit simulation scheme requires that the integration timestep be maintained smaller than the time it takes for longitudinal and transverse waves to propagate along the length of a segment of the tether. The equations of tether oscillations with small deflections predict that these two modes will travel with different velocities; the transverse velocity V_t depends upon the tension T on the tether, and the longitudinal wave velocity V_l depends upon the tether extensional stiffness E:

$$V_t = \sqrt{\frac{T}{\rho}} \qquad V_l = \sqrt{\frac{E}{\rho}}$$

where ρ is the linear density of the tether. To illustrate the challenge this poses, consider a very small tether system that utilizes a 2 km long metal tether massing 1 kg. The linear density of the tether is 0.5 g/m, the nominal gravity-gradient tension is approximately 0.2 N, and the tether stiffness approximately 5000 N/m. The transverse wave propagation speed will be roughly 20 m/s, and the longitudinal wave propagation speed approximately 3162m/s. If the tether is modeled as twenty 100 m segments, the timestep must be lower than 0.03 s to maintain numerical stability. In practice, the true stability limit on the timestep is even smaller, as low as 0.003s, depending upon implementation. Since each tether "node" has six degrees of freedom, and some complex simulations may require calculation of electrodynamic and aerodynamic forces at each segment, these small timesteps mean that an explicit propagator can require a large amount of computational power, and detailed simulations may run very slowly.

To address this challenge, TetherSim™ also can utilize an implicit cable dynamics propagation algorithm based upon finite element simulation methods.[1] This implicit method can use much larger timesteps and remain numerically stable.

Because the temperature of the tether can fluctuate significantly due to solar heating and ohmic dissipation, the simulation uses a temperature-dependent model for the stress-strain behavior of the aluminum tether. The model also assumes that the tether has no torsional or flexural rigidity.

Orbital Dynamics Model

The code calculates the orbital motion of the satellite, endmass, and tether elements using a 4th order Runge-Kutte algorithm to explicitly integrate the equations of motion according to Cowell's method.[2] The program uses an 8th-order spherical harmonic model of the geopotential and a 1st order model for the lunar gravity. When a satellite enters the Moon's sphere of influence, the trajectory is updated using the lunar potential as the primary body and a 1st order model of the geopotential as a perturbing force.

1. Hoyt, R.P., "A Stable Implicit Propagator for Space Tether Dynamics," Appendix C in *Stabilization of Electrodynamic Space Tethers,* TUI final report on NASA Contract NAS8-01013.
2. Battin, R.H., *An Introduction to the Mathematics and Methods of Astrodynamics*, AIAA, 1987, p. 447.

Geomagnetic Field Model

The TetherSim code has two options for calculation of the Earth's magnetic field: a simple tilted dipole model for fast calculations, and the International Geomagnetic Reference Field (IGRF 1995).

Simple Model:

The Earth's magnetic field is modeled as a a magnetic dipole with the magnetic axis of the dipole tilted off from the spin axis by φ=11.5°, as illustrated in Figure 2. In this model, we have ignored the 436 km offset of the dipole center from the Earth's geometric center.

The magnetic field vector is given by

$$\mathbf{B} = \frac{B_E\ R_E^3}{r^3}\begin{bmatrix} 3xz/r^2 \\ 3yz/r^2 \\ 3z^2/r^2 - 1 \end{bmatrix},$$

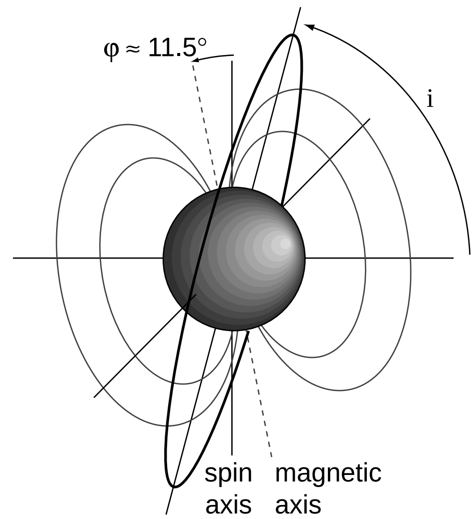

Figure 2. Tilted-dipole approximation to the geomagnetic field.

where $B_E = 31\ \mu$T is the dipole moment of the Earth, R_E is the Earth's mean radius, and x, y, and z are cartesian coordinates expressed in a reference frame that has been rotated so that the z axis is aligned with the magnetic axis.

The geomagnetic field rotates with the Earth as it spins, so in calculations of **vxB** induced voltages experienced by the tether as it orbits the Earth, the local velocity of the geomagnetic field is subtracted from the tether's velocity before the cross product is calculated.

IGRF Model:

For more detailed calculations, the code also can utilize the IGRF model.[3] For use in TetherSim, the FORTRAN code available from the GSFC server has been translated into C.

Ionospheric Plasma Density Model

The density of the ionospheric plasma is computed using data on electron density for average solar conditions provided by Enrico Lorenzini of the Smithsonian Astrophysical Observatory.[4] The electron density is computed by determining if the tether is in sunlight or shade, and then interpolating the density on the appropriate curve shown in Figure 3.

3. Barton, C.E., International Geomagnetic Reference Field: The Seventh Generation, *J. Geom. Geoelectr.* 49, 123-148, 1997.

4. Lorenzini, E., email 1/9/98.

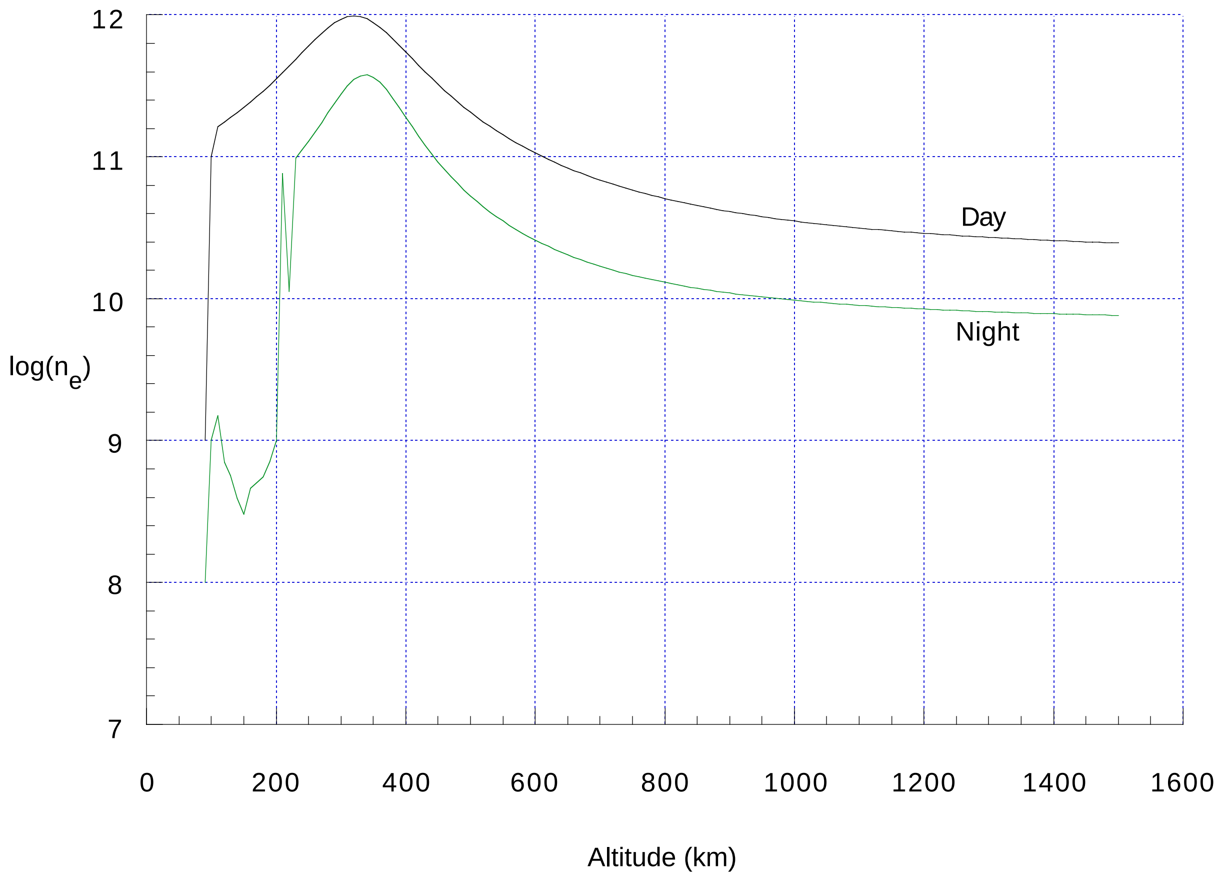

Figure 3. Average ionospheric plasma electron density as a function of altitude for sunlit and eclipse conditions.

Atmospheric Drag Model

For calculating atmospheric drag and heating, TetherSim can use one of two methods: a fast heuristic model of the neutral density, and a more detailed, but slower model based upon the MSISE90 Neutral Atmospheric Empirical Model.

Heuristic Model:

At low altitudes, neutral particle drag on the tether may become a significant effect. The code thus calculates the neutral particle drag on the satellite, endmass, and tether elements according to

$$F_{drag} = \frac{1}{2}\rho C_D V_{rel}^2 A$$

where $C_D \approx 2.2$ is the coefficient of drag for a cylindrical tether in free-molecular flow, V_{rel}^2 is the relative velocity between the tether and the atmosphere (assumed to rotate with the Earth), A is the cross-sectional area the tether presents to the wind, and ρ is the neutral density, calculated according to the heuristic formula developed by Carroll:[5]

$$\rho = \frac{1.47\times10^{-17} \quad T_{ex}(300 - T_{ex})}{1 + \dfrac{2.9(h-200)}{T_{ex}}} \qquad\qquad h > 200 km,$$

5. Carroll, J.A., "Aerodynamic Drag", p 160 in *Tethers In Space Handbook, 3rd Edition*, Cosmo and Lorenzini, editors, Smithsonian Astrophysical Observatory, 1997.

where h is the altitude and T_{ex} is the average exospheric temperature, 1100K.

MSISE 90 Model:

For more detailed simulations, such as the simulation of the aerodynamic drag and heating on the tethers in the Hypersonic Airplane Space Tether Orbital Launch (HASTOL) architecture, the code can utilize an aero drag and heating model developed by Stuart Bowman and Professor Mark Lewis of the University of Maryland, which uses the MSISE 90 model to calculate the atmospheric density.

Tether Reeling and Deployment

In order to study the dynamics of tether deployment and reeling maneuvers, the code has been extended to include the capability to model these behaviors. The tether can be deployed/reeled by either endmass. Currently, the code has models for:

- Free deployment (with deployment tension depending upon deployment rate)
- Deployment at a controlled tension
- Deployment at a controlled rate or rate program
- Tether retraction at controlled tension
- Tether retraction at controlled rate or rate program

These models have been used to model the deployment of a Terminator Tether™ and spin-up of the TORQUE experiment.

Endmass Dynamics

TetherSim™ can model the attitude dynamics of spacecraft attached to the tether. The attitude dynamics propagation algorithm is based upon standard quaternion methods.

APPENDIX N

Paper Presented at the 2001 Space Technologies and Applications International Forum

Momentum-Exchange/Electrodynamic-Reboost Tether Facility for Deployment of Microsatellites to GEO and the Moon

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Abstract. The LEO⇒GTO Tether Boost Facility will combine momentum-exchange tether techniques with electrodynamic tether propulsion to provide a reusable infrastructure capable of repeatedly boosting payloads from low Earth orbit to geostationary transfer orbit without requiring propellant expenditure. Designs for the orbital mechanics and system sizing of a tether facility capable of boosting 2,500 kg payloads from LEO to GTO once every 30 days are presented. The entire tether facility is sized to enable an operational capability to be deployed with a single Delta-IV-H launch. The system is designed in a modular fashion so that its capacity can be increased with additional launches. The tether facility can also boost 1000 kg payloads to lunar transfer orbits, and will serve as the first building block of an Earth-Moon-Mars Tether Transportation Architecture. The tether facility will utilize electrodynamic tether propulsion to restore its orbit after each payload boost operation. Using numerical modeling of tether dynamics, orbital mechanics, electrodynamics, and other relevant physics, we validate the orbital design of the system and investigate methods for performing electrodynamic reboost of the station.

INTRODUCTION

Under funding from NASA's Institute for Advanced Concepts (NIAC), Tethers Unlimited, Inc. is investigating the use of rotating tether facilities to provide a reusable infrastructure for in-space transportation. These systems will utilize momentum-exchange techniques and electrodynamic tether propulsion to transport multiple payloads *with little or no propellant consumption*. The ultimate goal of this work is to develop a tether transportation system able to provide frequent round-trip transport between Low Earth Orbit (LEO), geostationary orbit (GEO), the Moon (Hoyt 1999, 2000), and eventually Mars (Forward 1999). This ambitious tether transport system, however, will have to be built incrementally, beginning with small, simple tether transport facilities and adding additional components to build the system capacity. In this paper we develop a preliminary design for a small tether facility that could begin to demonstrate the technologies and techniques needed for tether transportation systems by transferring multiple microsatellites from LEO to geostationary transfer orbit (GTO) or lunar transfer orbit (LTO).

Background: Momentum-Exchange Tethers

In a momentum-exchange tether system, a long, thin, high-strength cable is deployed in orbit and set into rotation around a central body. If the tether facility is placed in an elliptical orbit and its rotation is timed so that the tether will be oriented vertically below the central body and swinging backwards when the facility reaches perigee, then a grapple assembly located at the tether tip can rendezvous with and acquire a payload moving in a lower orbit. Half a rotation later, the tether can release the payload, tossing it into a higher energy orbit. This concept is termed a momentum-exchange tether because when the tether picks up and tosses the payload, it transfers some of its orbital energy and momentum to the payload. The tether facility's orbit can be restored later by reboosting with propellantless electrodynamic tether propulsion or with high specific impulse electric propulsion; alternatively, the tether's orbit can be restored by using it to de-boost return traffic payloads.

Prior Work

Several prior research efforts have investigated conceptual designs for momentum-exchange tether systems. In 1991, Carroll (1991) proposed a tether transport facility that could pick payloads up from suborbital trajectories and provide them with a total ΔV of approximately 2.3 km/s. Carroll's design, however, assumed that the tether would be placed in a circular LEO orbit. In order for this facility and tether to remain above the atmosphere after a payload boost operation, the central facility had to mass 50-100 times the payload mass. This large mass would require a very large launch cost to set up the tether facility, which would likely hinder the economic viability of the concept.

Hoyt (1997) investigated a concept proposed earlier by Forward (1991) for a tether system for transporting payloads from LEO to the surface of the Moon. This design used two tethers in Earth orbits to minimize the total tether mass required for the system. Hoyt proposed placing the tethers in elliptical orbits and performing all catch and toss operations at or near perigee. Doing so minimized the drop in the tether's perigee, enabling a tether facility to boost a payload and still stay above the atmosphere with facility masses as low as 5-10 times the payload mass.

Bangham, Lorenzini, and Vestal (1998) developed a conceptual design for a two-tether system for boosting payloads from LEO to GEO. The tether transport system was proposed to stage the ΔV operations using two tether facilities in elliptical orbits so as to minimize the required tether mass. Their design proposed the use of high specific impulse electric thrusters to restore the orbit of the tether facilities after each payload boost operation. Even with the propellant mass requirements for reboost, they found that this system could be highly economically advantageous compared chemical rockets for GEO satellite deployment.

In a Phase I NIAC effort in 1999, Hoyt and Uphoff studied the orbital mechanics of multi-tether systems for transporting payloads between LEO and the surface of the Moon and found that orbital perturbations caused by Earth oblateness and other effects would make scheduling transfers in a staged system difficult or impossible (Hoyt 1999). Consequently, they concluded that tether systems for transporting payloads from LEO to GTO or LTO should use one tether facility in Earth orbit to provide all of the ΔV. Further study revealed that although a single-tether system requires a much larger total tether mass than a staged two-tether system, the total system mass for a one-tether system, including the mass required for the control station and grapple assemblies, is the same or less than a multi-tether system because the total ballast mass required in a single-tether system is lower (Hoyt 1999).

In a follow-on Phase II effort funded by NIAC, Hoyt (2000) developed a design for a tether facility optimized for boosting commercial communications satellites to geostationary transfer orbit. This paper extends that work by investigating the scaling of the system to handle small microsatellite-class payloads.

µSAT TETHER BOOST FACILITY DESIGN

The ultimate goal of the NIAC-funded research effort is to develop an architecture for a fully reusable in-space transportation infrastructure capable of providing frequent rapid round-trip transport between Earth, the Moon, and Mars. The technical development of such a transportation architecture must, however, follow a path that is commensurate with a viable business plan, in which early components can serve useful functions to generate revenue to fund the development of the rest of the system. The deployment of a tether boost facility requires the launching of a tether and control station which, together, mass roughly 10 times the mass of the payload. For deploying one or a few spacecraft, a tether facility thus would not be economically competitive with conventional rocket systems. For applications where a large number of spacecraft must be deployed, however, a tether boost facility can become highly advantageous because it eliminates the need to launch transfer propellant for each spacecraft. One potential application of a tether boost facility is in the deployment of microsatellites. A small tether facility could provide a low-cost means for deploying swarms of microsatellites or delivering multiple small satellites to GEO to service and refuel communications and observation satellites. The same tether boost facility could also be capable of delivering numerous microsatellites to lunar orbit. In the following sections we describe a concept for a tether boost facility designed to be deployed into LEO using an Athena-II class launch vehicle.

System Requirements

Payload Mass: The baseline mission of a µSat Tether Boost Facility will be to pick 200 kg microsatellites up from low-LEO orbits and inject them into transfer orbits to GEO altitudes. To do so, the Tether Boost Facility will provide each microsatellite with a total ΔV of 2.4 km/s. This same facility will also be capable of boosting approximately 100 kg payloads to lunar transfer orbits.

Safety Factor: To provide ample margin for error and degradation of the tether over time, the tether structure is sized to provide a safety factor of 2 for the largest loads expected in the system. The largest loads will be due to transient oscillations immediately after the payload capture. These loads are predicted using numerical modeling with TetherSim™. Computed with respect to the nominal loads, the safety factor is chosen to be 3.5 over most of the length of the tether. To provide additional safety during rendezvous and capture dynamics, the safety factor is increased to 4.0 for the 10 km portion closest to the grapple.

Throughput: Because one of the primary advantages of momentum-exchange tethers is their reusability, to maximize the cost-competitiveness of the system it will be designed to boost microsatellites as frequently as once every 30 days.

Momentum-Exchange/Electrodynamic-Reboost Facility Concept

In order for the tether facility to boost one payload per month, the tether must restore its orbital energy after each payload boost operation. If the tether facility operates at least partly within LEO, it can instead utilize electro-dynamic tether propulsion to perform reboost of its orbit. This concept, called the "High-strength Electrodynamic Force Tether" (HEFT) Facility (also referred to as a "Momentum-Exchange/Electrodynamic-Reboost (MXER) Tether Facility), is illustrated in Figure 1 (Forward and Hoyt 1997). The Tether Boost Facility will include a control station housing a power supply, ballast mass, plasma contactor, and tether deployer, which would extend a long, tapered, high-strength tether. A small grapple vehicle will reside at the tip of the tether to facilitate rendezvous and capture of the payloads. The tether will include a conducting core, and a second plasma contactor would be placed near the tether tip. By using the power supply to drive current along the tether, the HEFT Facility will generate electrodynamic forces on the tether. By properly varying the direction of the current as the tether rotates and orbits the Earth, the facility can use these electrodynamic forces to generate either a net torque on the system to change its rotation rate, or a net thrust on the system to boost its orbit. The HEFT Facility thus can repeatedly boost payloads from LEO to GTO, and in between each payload boost operation it will use propellantless electrodynamic propulsion to restore its orbital energy.

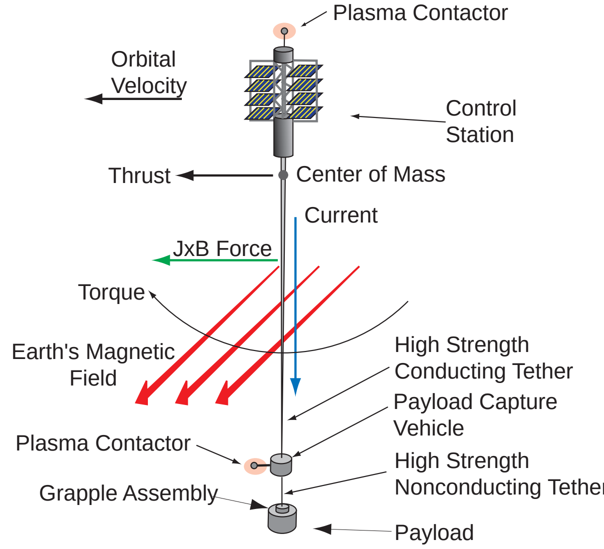

Figure 1. Schematic of the HEFT Facility concept.

Orbital Design

To boost a microsatellite from LEO to GTO, the tether facility performs a catch and release maneuver to provide the microsatellite with two ΔV impulses of approximately 1.2 km/s each. To enable the tether to perform two "separate" ΔV operations on the payload, the facility is placed into a highly elliptical orbit with its perigee in LEO. The tether facility's initial orbit is chosen so that when the tether is near perigee, its center of mass is moving approximately 1.2 km/s faster than the payload in circular LEO. It can then catch the payload, hold it for half a rotation, and then release it at the top of the tether's rotation. This injects the payload into the high-energy transfer trajectory.

Table 1 shows the orbital design for the μSat Tether Boost Facility. The orbital parameters and system masses shown in Table 1 are chosen so that the payload's orbit and the facility's initial orbit are harmonic. For this design the resonance is 41:20. This enables the tether facility to have multiple opportunities to capture the payload. If the payload and tether do not succeed in achieving docking during the first rendezvous attempt, they will wait for 2.6 days, adjusting the tether spin and correcting any trajectory errors, and then a second rendezvous will be possible without any significant maneuvering. The resonance design shown in Table 1 accounts for regressions of both orbits due to the Earth's non-ideal gravitational potential, up to the J4 term.

Table 1. System Orbital Design for LEO⇒GTO Boost

System Masses		
Tether mass	743	kg
CS Active Mass	747	kg
CS Ballast Mass	371	kg
Grapple mass	50	kg
Total Facility Mass	**1,911**	kg
Total Launch Mass	**1,540**	kg
Payload Mass	**200**	kg

Tether Characteristics		
Tether Length	100	km
Tether mass ratio	3.71	
Tether tip velocity at catch	1,268	m/s
Tether tip velocity at toss	1,148	m/s
Tether angular rate	0.015803	rad/s
Gravity at Control Station	0.70	g
Gravity at payload	1.85	g
Rendezvous acceleration	2.04	g

Positions & Velocities		**Pre-Catch Payload**	**Pre-Catch Tether**	**Joined System Post-catch**	**Post-Toss Tether**	**Post-Toss Payload**
perigee altitude	km	325	405	398	390	470
apogee altitude	km	325	8446	7199	6103	35786
perigee radius	km	6703	6783	6776	6768	6848
apogee radius	km	6703	14824	13577	12481	42164
perigee velocity	m/s	7711	8979	8859	8739	10007
apogee velocity	m/s	7711	4109	4421	4739	1625
CM dist. From Station	m		19765	27365	19765	
CM dist. To Grapple	m		80235	72635	80235	
ΔV to Reboost	m/s				240	

System Design

Figure 2 illustrates the system concept design for the Tether Boost Facility. The Tether Boost Facility is composed of a Control Station, a tapered high-strength tether, and a Grapple Assembly. In addition, a Payload Accommodation Assembly (PAA) will be attached to the payload to provide maneuvering and guidance for rendezvous. For LEO⇒GTO traffic, this PAA will be an expendable unit incurring recurring costs.

To meet the requirement for operational capability with a single launch, the tether facility is sized to be deployed with a single launch of an Athena-II or comparable vehicle. As Figure 1 shows, the 371 kg Athena Orbit Adjust Module will be retained for use as ballast mass. The control station includes an array of solar panels which swivel to track the sun as the tether facility rotates. In this design, we have chosen to place the control station at the end of the tether, rather than at the center of mass of the facility. This choice was made for several reasons: because it minimizes the dynamical complexity, because it requires only one tether deployer, and because the center of mass of the system shifts when the payload is captured and released.

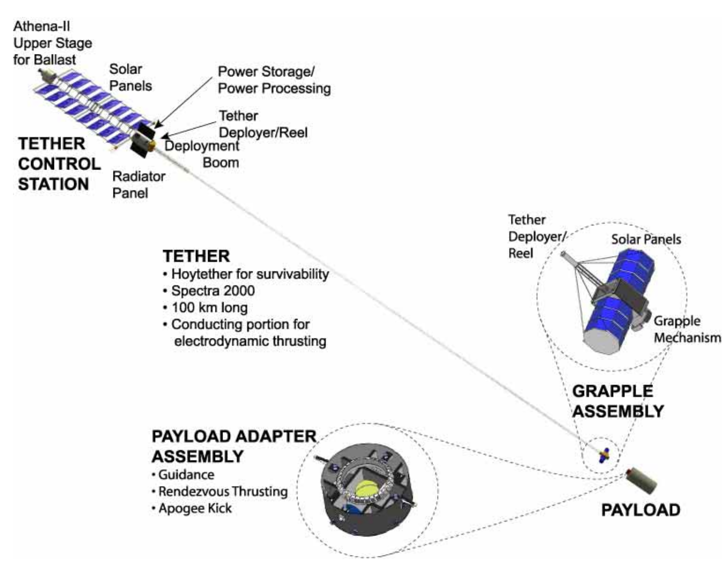

Figure 2. The μSat Tether Boost Facility

Electrodynamic Tether: The tether in this system is composed of Spectra 2000® fibers braided into the Hoytether™ structure (Forward 1995). The nominal length of the tether is 100 km. Along the 50 km of the tether closest to the Control Station, a total of 80 kg of insulated aluminum wire is woven into the structure, providing a current path for electrodynamic thrusting.

Power System Sizing: In order for the tether facility to reboost its orbit within 30 days, the facility will require a solar power generation capability of 5.5 kW. Because the facility will pass through the radiation belts frequently, its solar power system will utilize a concentrator-type solar panel design, such as the Scarlet design, with 150 mil Aluminum backside and 100 mil glass cover slides to shield the arrays from the belt particles. In order for the solar array to produce the desired power levels after 10 years of operation, they system will be deployed with 7.5 kW of initial power generation capability. Using Scarlet-type panel technology, this solar array would mass approximately 75 kg. The tether facility will collect this solar power during the roughly 80% of its orbit that it is in the sunlight, and store it in a battery system. Then, during perigee pass, it will drive the electrodynamic tether at an average power level of 300 kW (modulated as to be described later). In order to provide a maximum battery depth-of-discharge of 30%, the control station will have a battery system with 315 A•hr of capacity (120 V power system). Using advanced Li ion batteries, this will require approximately 255 kg of batteries.

Payload Capacity vs. Tip Velocity

The boost facility described herein is optimized for tossing 200 kg payloads to GTO. The same facility, however, can also service traffic to other orbits by changing its rotation rate and initial orbit. Because the stress in the tether increases exponentially with the rotation rate, the payload capacity drops as the tip velocity increases. This same boost facility could toss 100 kg-class microsatellites into a minimal-energy lunar transfer orbit, or toss 50 kg microsatellites into an escape trajectory. Furthermore, by taking advantage of the weak-stability-boundary lunar transfer methods invented by Belbruno (2000), the tether system could potentially toss the microsatellites directly into elliptical lunar orbits.

ELECTRODYNAMIC REBOOST TO RESTORE FACILITY ORBIT

As the Tether Boost Facility catches and tosses a payload into GTO or LTO, its orbit drops. The after boosting a payload to GTO, the apogee drops 2343 km, and the perigee drops 15 km. To restore the orbit, the tether system must increase the facility's orbital energy by 4.3 GJ, and it will do so by performing electrodynamic thrusting while the tether is within the dense portion of the ionosphere near the perigee of its orbit. Because the tether is rotating, the direction of the current must be alternated as the tether rotates to produce a net thrust on the facility. Reboost of a rotating tether system has been investigated with the TetherSim™ program in a previous paper (Hoyt 2000).

CONCLUSIONS

We have presented an orbital design and system-concept level definition for a tether facility capable of boosting 200 kg microsatellites from LEO to GTO once every 30 days. The entire tether facility is sized to enable an operational capability to be deployed with a single Athena-II launch. The tether facility can also boost 100 kg payloads to lunar transfer orbits, The tether facility will utilize electrodynamic tether propulsion to restore its orbit after each payload boost operation.

ACKNOWLEDGMENTS

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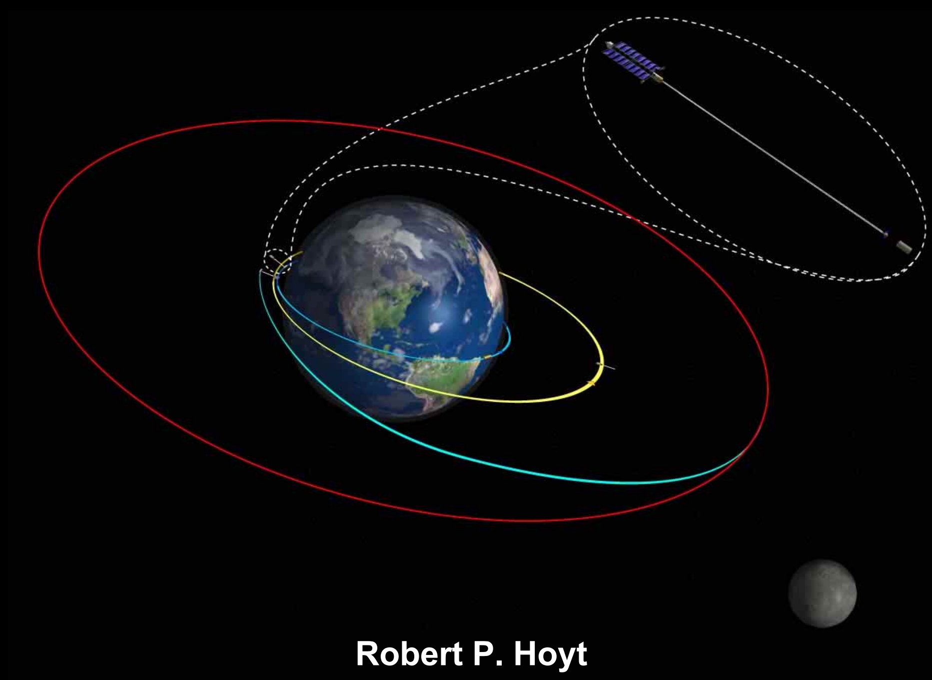
Hoyt, R.P., "Design and Simulation of a Tether Boost Facility for LEO⇒GTO Transport", AIAA Paper 2000-3866, 36th Joint Propulsion Conference, Huntsville AL, July 17-19 2000.

µSatellite Tether Boost Facility



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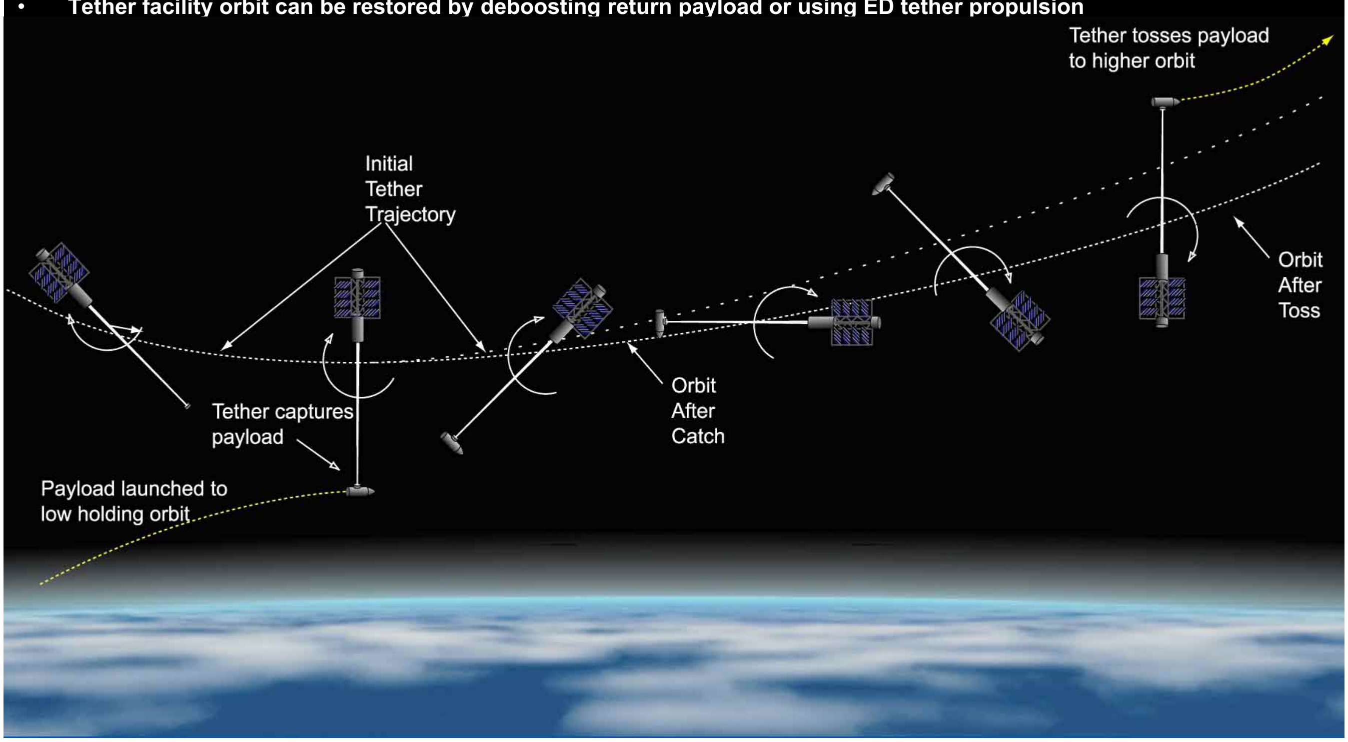
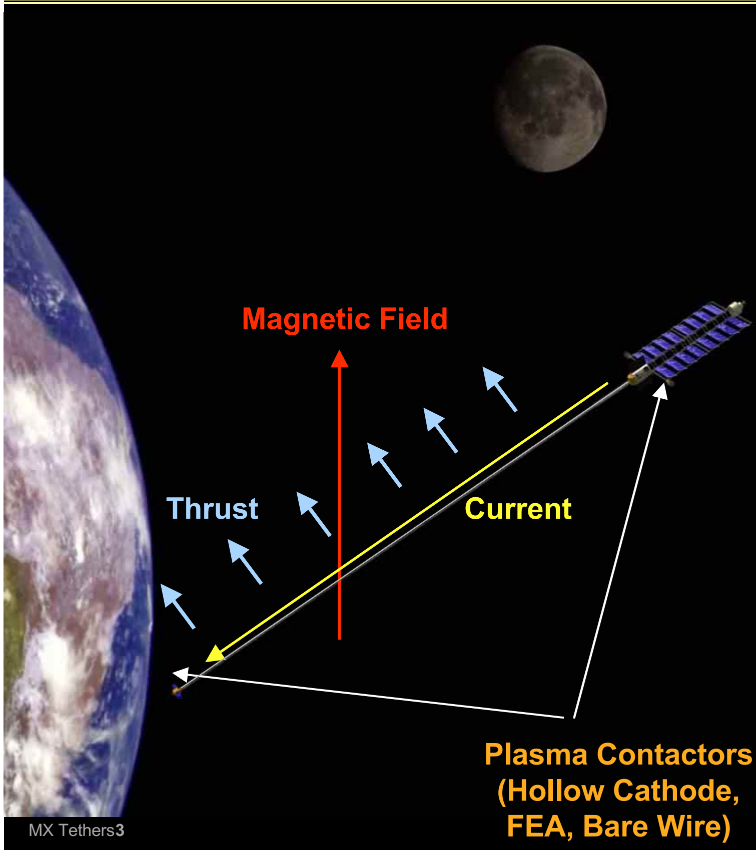
Momentum-Exchange Tether Boost Facility

- **High-strength tether rotates around orbiting control station**
- **Tether picks payload up from lower orbit and tosses payload into higher orbit**
- **Tether facility gives some of its orbital momentum & energy to payload**
- **Tether facility orbit can be restored by deboosting return payload or using ED tether propulsion**



Electrodynamic Thrusting to Reboost Orbit



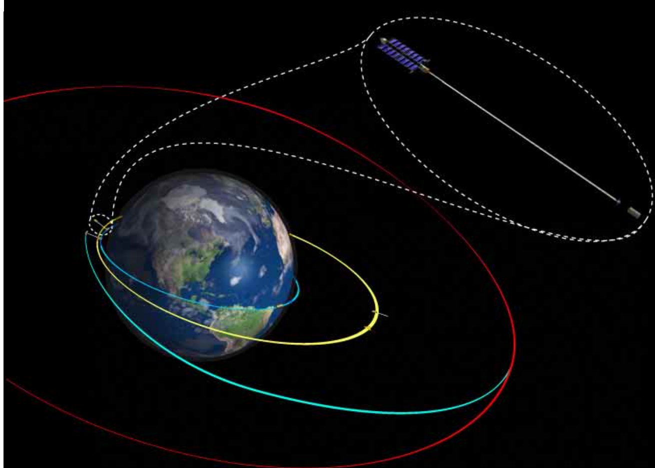
- Drive current along tether
- Plasma contactors exchange current w/ ionosphere
- Plasma waves close current "loop"
- Current "pushes" against geomagnetic field via JxB Force
- Vary current as tether rotates to achieve desired net thrust

Microsatellite Boost Facility

- DARPA, DoD/AF, NASA, & Commercial organizations developing concepts for "fleets" of small spacecraft
  - Servicing/Refueling of on-orbit assets
  - Distributed sensing architectures
  - Distributed communications architectures
- Momentum-Exchange/Electrodynamic Reboost Tether Boost Facility
  - good match for multiple µSat market
    - Small payloads -> reasonable facility investment
    - Numerous payloads -> rapid amortization of investment
    - Flexibility in transfer orbit

LEO⇒GTO μSat Tether Boost Facility

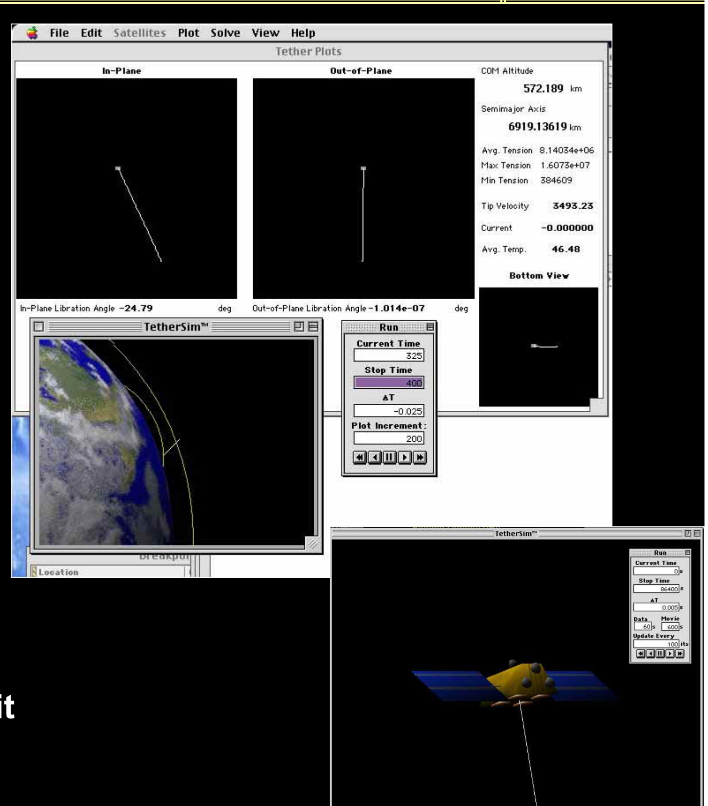
- Designed to Boost 200 kg payloads from LEO to GTO - Total ΔV = 2.4 km/s
- First Operational Capability Can be Placed in LEO with One Athena-II Launch
- Uses Electrodynamic Reboost to Enable Facility to Boost 1 Payload Per Month
- Facility Can also Toss 100 kg payloads to Lunar Transfer Orbit, 50 kg to escape



Analysis Methods

Tether System Design:
- Tapered tether design
  - Spectra 2000
- Orbital mechanics considerations to determine facility mass required

Tether operation: TetherSim™
- Numerical Models for:
  - Orbital mechanics
  - Tether dynamics
  - Geomagnetic Field (IGRF)
  - Plasma Density (IRI)
  - Neutral Density (MSIS '90)
  - Thermal and aero drag models
  - Endmass Dynamics
- Interface to MatLab/Satellite Tool Kit



µSat Tether Facility Design

Mass Ratios:

- Control Station 747 kg
- Upper Stage (Ballast) 371 kg
- Grapple 50 kg
- Tether 743 kg
- TOTAL: = 9 x payload

Tether Length: 100 km

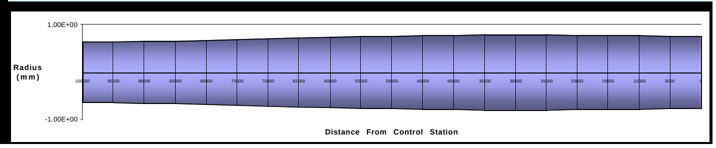
Orbit:

- 405x8446 km ->390x6103 km

System Masses		
Tether mass	743	kg
CS Active Mass	747	kg
CS Ballast Mass	371	kg
Grapple mass	50	kg
Total Facility Mass	**1,911**	kg
Total Launch Mass	**1,540**	kg
Payload Mass	**200**	kg

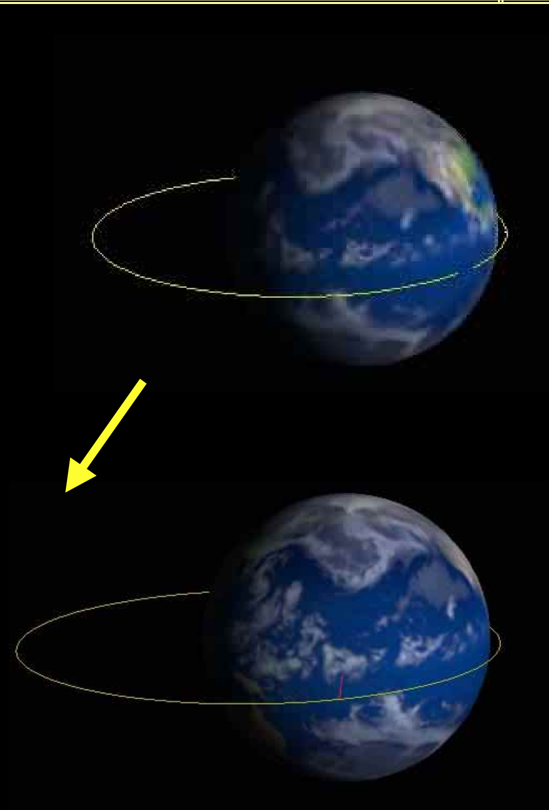
Tether Characteristics		
Tether Length	100	km
Tether mass ratio	3.71	
Tether tip velocity at catch	1,268	m/s
Tether tip velocity at toss	1,148	m/s
Tether angular rate	0.015803	rad/s
Gravity at Control Station	0.70	g
Gravity at payload	1.85	g
Rendezvous acceleration	2.04	g

		Pre-Catch		Joined System	Post-Toss	
Positions & Velocities		**Payload**	**Tether**	**Post-catch**	**Tether**	**Payload**
perigee altitude	km	325	405	398	390	470
apogee altitude	km	325	8446	7199	6103	35786
perigee radius	km	6703	6783	6776	6768	6848
apogee radius	km	6703	14824	13577	12481	42164
perigee velocity	m/s	7711	8979	8859	8739	10007
apogee velocity	m/s	7711	4109	4421	4739	1625
CM dist. From Station	m		19765	27365	19765	
CM dist. To Grapple	m		80235	72635	80235	
ΔV to Reboost	m/s				240	
ΔV to Correct Apogee	m/s					0
ΔV to Correct Precess.	m/s					0
ΔV To Circularize	m/s					1449
Basic Orbital Parameters						
semi-major axis	km	6703	10804	10176	9625	24506
eccentricity		0.0	0.372	0.334	0.297	0.721
inclination	rad	0	0	0	0	0
semi-latus rectum	km	6703	9308	9040	8777	11783
sp. mech. energy	m2/s2	-2.97E+07	-1.84E+07	-1.96E+07	-2.07E+07	-8.13E+06
vis-viva energy	m2/s2	-5.95E+07	-3.69E+07	-3.92E+07	-4.14E+07	-1.63E+07
period	sec	5462	11176	10217	9397	38179
period	min	91.0	186.3	170.3	156.6	636.3
station rotation period	sec		397.6	397.6	397.6	
rotation ratio			28.1	25.7	23.6	

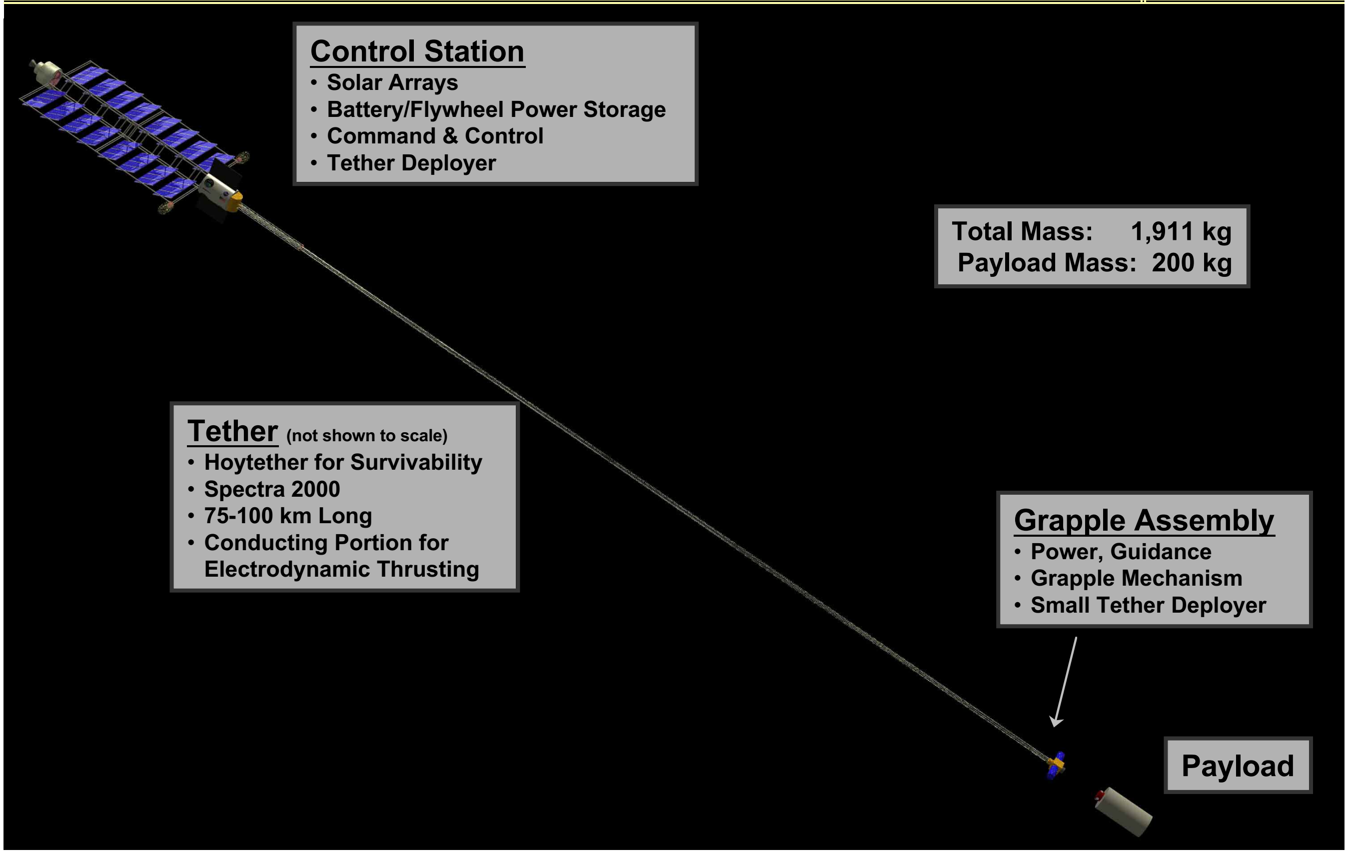


Tether Facility Reboost

- Tether facility transfers momentum & energy to payload
  - tether orbit apogee drops ~2300 km
- Tether must restore 4 GJ of orbital energy
- To reboost within 30 days, must add energy to orbit at 1.6 kW
- Requires EOL Panel Power of 7.5 kW

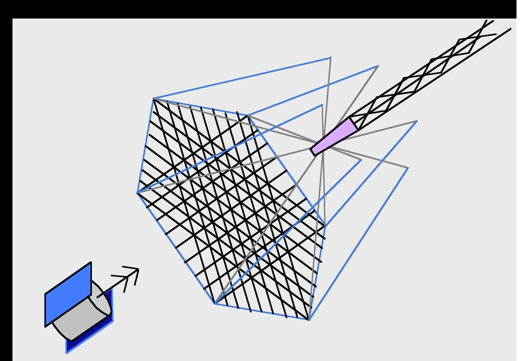


Tether Boost Facility



"Net & Harpoon" Grapple Method

- For a small tether facility, grapple mechanism must be small and lightweight
  - precludes highly capable grapple assembly
- Favored grapple method is "harpoon & net"
- Payload maneuvers to proximity with net
- Payload "shoots" tethered grapple into net
- Payload releases from net by retracting barbs on grapple

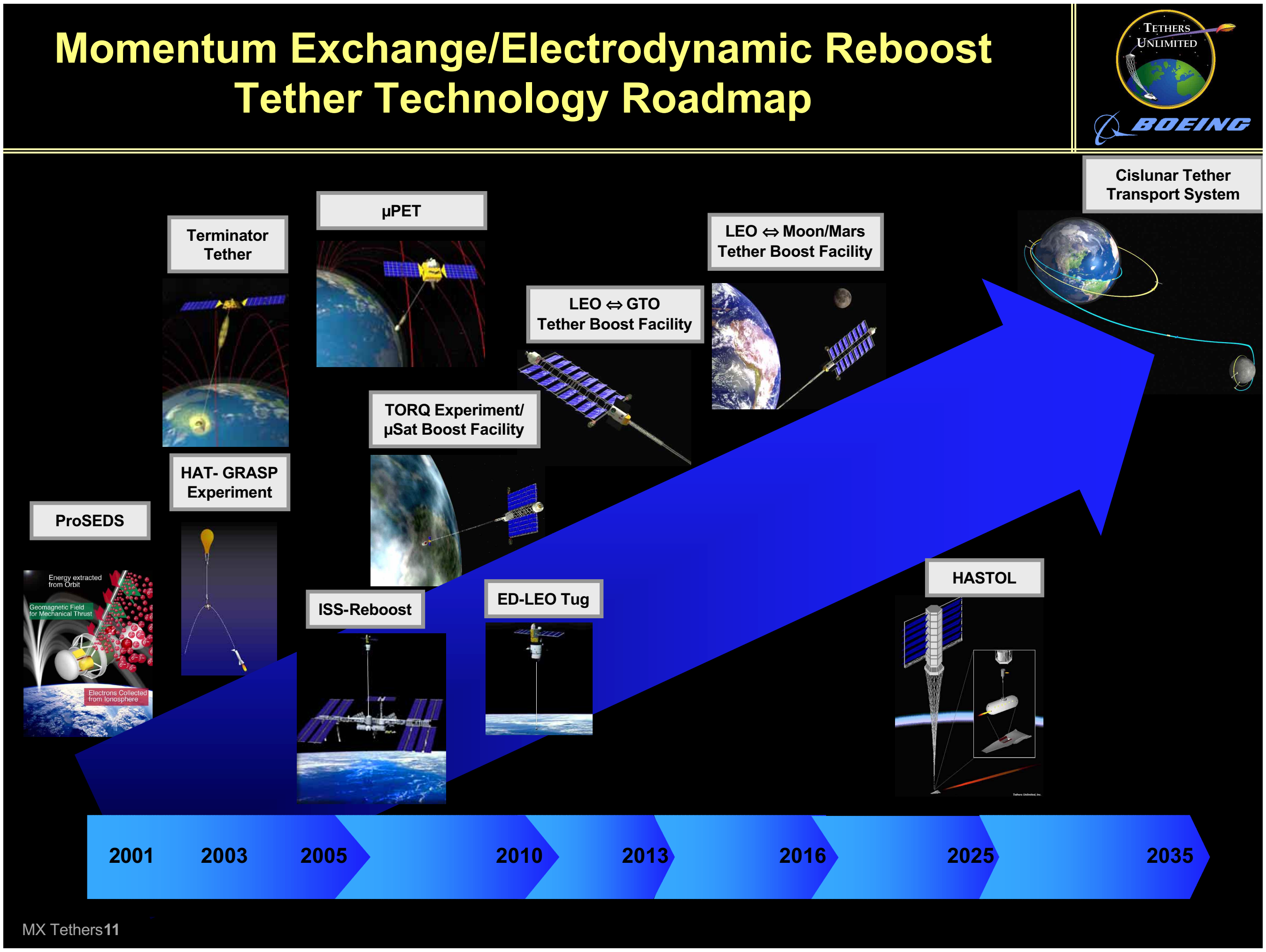


Advantages:

- Large capture envelope
- Minimizes need for fast maneuvering by payload
- Minimizes dynamic loads on payload

Issues:

- Load distribution on net
- Assuring secure grappling


Momentum Exchange/Electrodynamic Reboost
Tether Technology Roadmap
TETHERS UNLIMITED
BOEING
Cislunar Tether Transport System
µPET
Terminator Tether
LEO ⇔ Moon/Mars Tether Boost Facility
LEO ⇔ GTO Tether Boost Facility
TORQ Experiment/ µSat Boost Facility
HAT- GRASP Experiment
ProSEDS
Energy extracted from Orbit
Geomagnetic Field for Mechanical Thrust
Electrons Collected from Ionosphere
ISS-Reboost
ED-LEO Tug
HASTOL
2001
2003
2005
2010
2013
2016
2025
2035
MX Tethers 11

INTERPLANETARY TETHER TRANSPORT OVERVIEW

Gerald D. Nordley* and Robert L. Forward**

Abstract

Travel to and from planets may be accomplished by a system of rotating tethers in elliptical orbits about each planet. A payload is picked up near periapsis and tossed later still near periapsis, at a velocity sufficient to give the payload a substantial hyperbolic excess velocity. At the destination planet, it may be caught near periapsis and released a short time later in a bound trajectory. The system works in both directions and is reusable. Kinetic energy lost by the throwing tethers can be restored either by catching incoming payloads, by propellantless tether propulsion methods, and/or high specific impulse propulsion systems. In preliminary studies with simplified assumptions, tethers with tip velocities of 3 km per second may send payloads to Mars in as little as 70 days if aerobraking is used at Mars to dissipate excess relative velocity and the orbital phasing is favorable. Tether-to-tether transfers without aerobraking may be accomplished in about 110 to 160 days. A rotating free space tether with a payload at each end approaching a planet may be split at periapsis, leaving one payload bound and the other getting an additional velocity boost. Tether systems using commercially available tether materials at reasonable safety factors can be as little as 15 times the mass of the payload being handled, however mass ratios on the order of 100 provide more tether orbit stability.

INTRODUCTION

The idea of using rotating tethers to pick up and toss payloads has been in the tether literature for decades [1-7]. In 1991, Forward [8] combined a number of rotating tether concepts published by others [2,6,7] to show that three rotating tethers would suffice to move payloads from a suborbital trajectory just above the Earth's atmosphere to the surface of the Moon and back again, without any use of rockets except to get out of the Earth's atmosphere. The three tethers consisted of a rotating tether in a nearly circular Low Earth Orbit (LEO), a rotating tether in a highly Elliptical Earth Orbit (EEO), and a rotating "Lunavator" tether cartwheeling around the Moon in a circular orbit whose altitude is equal to the tether arm length, resulting in the tip of the tether touching down on the lunar surface. This concept has since been examined in detail by Hoyt and Forward [9-12].

In thinking about ways to improve the performance of the system, Forward realized that much of the gain in the three-tether system came from the EEO tether, since its center-of-mass velocity at perigee was quite high, and when the tether tip rotational velocity was added, the toss velocity was hyperbolic with respect to the Earth-Moon system and could be used for interplanetary transport.

Forward enlisted the aid of his co-author, who extended the back-of-the-envelope calculations with more detailed calculations, including the post separation orbit of the tether. The Mars-Earth Rapid Interplanetary Tether Transport (MERITT) System was the result [13]. Extending the principle of high periapsis velocity in EEO to a hyperbolic, with the negative delta v to the post separation tether led to the idea of a rotating tether in a hyperbolic orbit leaving one payload bound and the other getting a velocity boost. This could apply, for instance, to a Jupiter-Pluto mission.

Background material from [13] is reproduced in part below for overview purposes, but the we have incoporated some new data and concepts included in pending archival papers.

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NOMENCLATURE

Symbol Meaning

A payload approach or departure angle from planetary orbital path
a_c centripetal acceleration
d density of tether material
F factor by which tether material is derated for safety
L length of rotating tether arm from center-of-mass to tip
M_T mass of tether material from center-of-mass to tip
M_P mass of payload and grapple at tip of tether
q rotation angle of tether from vertical
U ultimate strength of tether material (force per unit area at failure)
u true anomaly of payload
u_T true anomaly of tether center of mass
u_C true anomaly of tether/payload system at payload capture
u_r true anomaly of tether/payload system at payload release
u_∞ true anomaly of hyperbolic orbit at infinity (hyperbolic asymptote)
u_r true anomaly of tether/payload system at payload release
v_C characteristic (maximum) tip velocity of derated tether material
v_d velocity at destination of interplanetary trajectory
v_o velocity at origin of interplanetary trajectory
v_t tip velocity of rotating tether
v_U maximum tip velocity of tether material at ultimate strength
Δv velocity change
Δq change in rotation angle of tether
$\Delta\omega$ change in payload orbit argument of periapsis
δu Earth-centered arc from u_r to tether tip
γ_C tether-payload system flight path angle from zenith at payload capture
γ_r tether-payload system flight path angle from zenith at payload release
φ_d flight path elevation angle of interplanetary trajectory at destination
φ_o flight path elevation angle at interplanetary trajectory at origin
Π periapsis of payload orbit
ω argument of periapsis of payload orbit from Sun-planet line
ω_T argument of periapsis of tether orbit from Sun-planet line

SYSTEM DESCRIPTION

The system consists of two rapidly rotating tethers in highly elliptical orbits, generically called "planetwhips" [13], or by the name of the planet they orbit; thus "MarsWhip." The rotating systems would consist of long (hundreds of kilometers), thin (a few square centimeters in cross sectional area) cables of high strength material with a grapple at one end and a counterweight at the other. A service module would be attached to the tether and include a solar electric power supply, tether winches, command and control electronics, and propulsion systems. This might be positioned near the center of mass, for ballast, or at the end opposite the grapple as part of the counterweight.

A payload is launched from low orbit or suborbital trajectory. The payload is picked up by a grapple system on the EarthWhip tether as the tether nears perigee and the tether arm nears the lowest part of its swing. It is released from the tether later when the tether is still near perigee and the arm is near the highest point of its swing. The payload thus gains both velocity and potential energy at the expense of the tether system.

The incoming payload is caught in the vicinity of periapsis by the grapple end of the tether near the highest part of its rotation and greatest velocity with respect to the destination planet. The payload is released later when the tether is near periapsis and the grapple end is near the lowest part of its swing at a velocity and altitude which will cause the released payload to enter orbit around the destination, or enter an atmosphere for surface descent. The system can be designed to work in both directions. Aerobraking gives shorter trip times because the incoming payload velocity change is not limited by the maximum tether tip velocity at the destination planet.

Energy and momentum lost by the tethers can be replenished over times by highly efficient propulsion systems which use electrical energy. The system thus acts a propulsive energy-momentum bank which delivers the propellant efficiency of advanced propulsion systems with the dynamic advantages of impulsive deep-gravity-well maneuvers.

In the following subsections we describe the system in detail, discuss the modeling of the system and present some preliminary results for different tip speeds.

Payload Pickup and Release

Fig. 1 shows the general geometry of a tether picking up a payload from a suborbital trajectory at a point just outside the atmosphere of the origin planet and injecting it into an interplanetary transit trajectory. The payload is picked up, swung around the tether's center of mass in a circle over angle q as the tether/payload system moves along its orbit . The payload is released from the tip of the tether near the top of the circle. In the process, the tether center of mass loses both altitude and velocity twice, representing the transfer of energy by the tether to the payload.

Around the time of pick-up, the trajectory of the payload must be of equal speed and should be very nearly tangential (no radial motion) to the circle of motion of the tether tip in the tether frame of reference. It is easy to see how this condition may be satisfied by rendezvous at the mutual apsides of the tether orbit and the payload pickup orbit, but other, more complex trajectories work as well.

It is not a requirement, however, that the tether plane of rotation, the tether orbit, and the payload pickup orbit be coplanar. The mutual velocity vector at pick-up is essentially a straight line, and an infinite number of curves may be tangent to that line. The practical effect of this is to allow considerable leeway in rendezvous conditions. It also means that the general conclusions reached from the kind of two-dimensional analysis presented here should hold for more complicated geometries.

The release orbit is tangential to the tether tip circle in the tether frame of reference by definition, but it is not necessarily tangential to the trajectory in the frame of reference of the origin planet. By this time, the tether has moved beyond periapsis through angle Δu and there will be a significant flight path angle.

Large variations from this scenario will result in significant velocity losses, but velocity management in this manner could also prove useful. If, on the other hand, maximum velocity transfer and minimum tether orbit periapsis rotation is desired, the payload can be retained and the tether arm length or period adjusted to release the payload in a purely azimuthal direction at the next periapsis.

Following payload release, tethers could also provide artificial gravity for crewed missions. The payload tossed by the EarthWhip and caught by the MarsWhip would consist of two capsules connected by a tether and put into slow rotation during the toss operation. After the toss, a solar electric powered winch on one of the payload capsules would change the length of the tether to attain any desired artificial gravity level during the transit time interval.

Since the payload can be caught by the tether grapple at either capsule end, and the capsule velocity can add or subtract from the MarsWhip tether tip velocity.

Interplanetary Transfer Orbits

The system achieves its rapid transfer times because its interplanetary trajectories result from high, essentially fixed velocity increments and are not subject to the mass/Δv trade that pushes other propulsion systems toward minimum Δv trajectories with low flight path angles at injection. For fast tip velocities, interplanetary injection flight path angels do not need to be small.

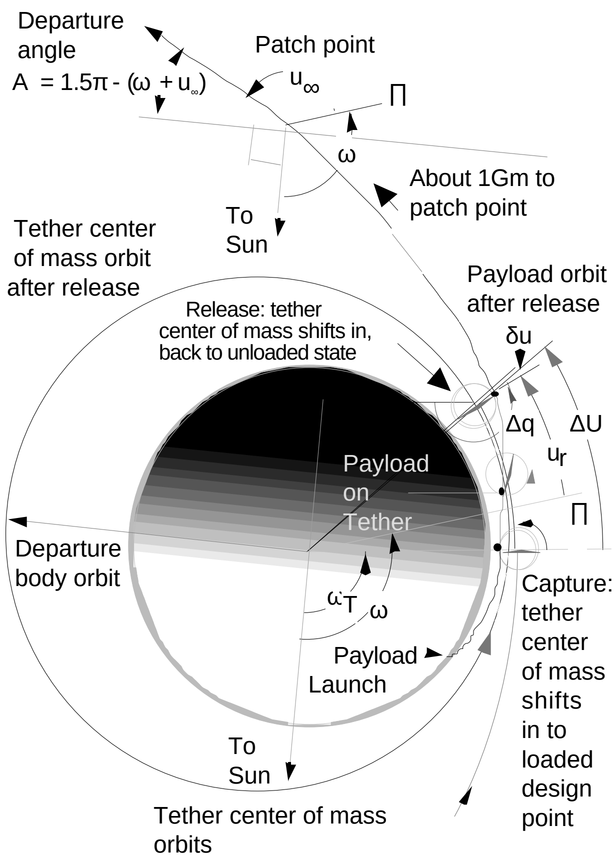

Figure 1. Payload pickup and release to an interplanetary trajectory

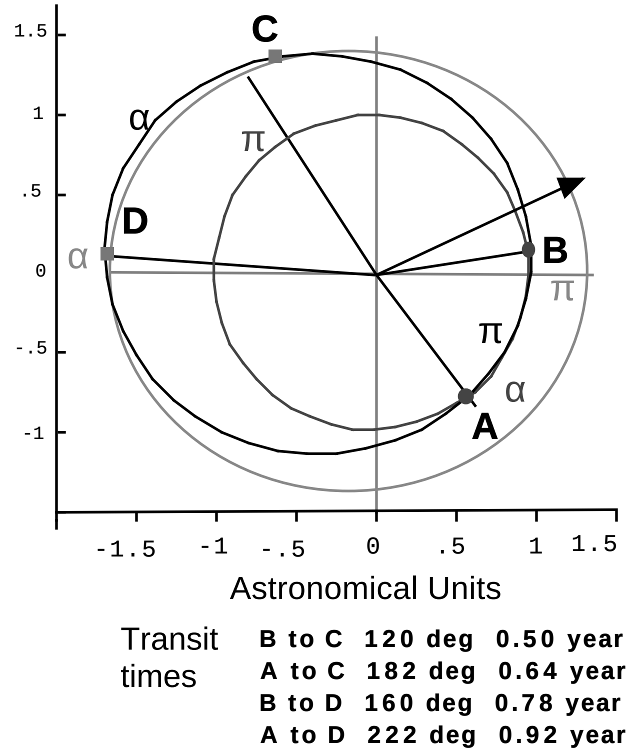

Figure 2. Payload interplanetary trajectories

Fig. 2 shows an example of a Mars Earth Interplanetary Tether Transport (MERITT) system transfer orbit, approximating a launch opportunity from B to C on Sep 12 2005 with a 2 km/s tether. Several orbit crossing rendezvous are possible, however the fastest trip times are generally found in the B-C trajectory case. An extensive discussion of the general orbit transfer problem may be found in Bate, Mueller and White [14]

Capture and Release at Destination

Capturing of an incoming payload with a tether (Fig. 3) is essentially the time reversal of the outgoing scenario; the best place to add hyperbolic excess velocity is also the best place to subtract it. If the tether orbital period is an integral multiple of the rotation period following release of a payload, the tip will be pointed at the zenith at periapsis and the capture will be the mirror image.

Capture after a pass through the destination body's atmosphere (Fig. 4) is more complex than a periapsis capture, but involves the same principle: matching the flight path angle of the payload exiting trajectory to the tether flight path angle at the moment of capture and the velocity to the vector sum of the tether velocity and tip velocity. Aerodynamic lift and energy management during the passage through the atmosphere provide propellant-free opportunities to do this.

After capture, the payload swings around the tether and is released into a trajectory that either orbits the destination planet or intersects its atmosphere so that the payload can land. The tether center of mass shifts outward and its velocity increases in this process, leaving the tether orbit in a higher energy state. It is, in some ways, as if the incoming payload had "bounced" off the tether.

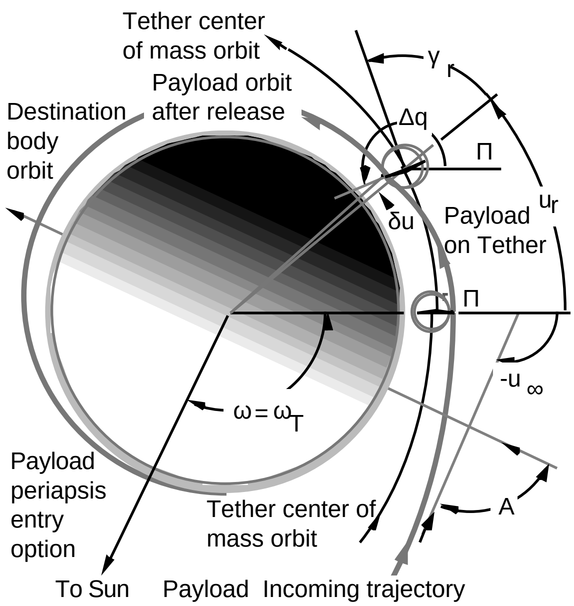

Figure 4. Tether-only payload capture.

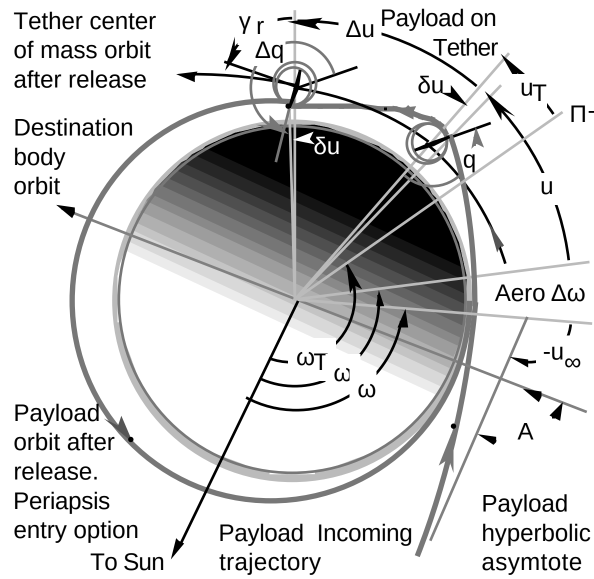

Figure 5. Aeroassisted Tether Capture

TETHER ENGINEERING

For a tether transport system to be economically advantageous, it must be capable of handling frequent traffic for many years despite degradation due to impacts by meteorites and space debris. Fortunately, a survivable tether design exists, called the Hoytether™, which can balance the requirements of low weight and long life [15,16]. As shown in Fig. 5, the Hoytether™ is an open net structure where the primary load bearing lines are interlinked by redundant secondary lines. The secondary lines are designed to be slack initially, so that the structure will not collapse under load. If a primary line breaks, however, the secondary lines become engaged and take up the load.

Note that four secondary line segments replace each cut primary line segment, so that their cross-sectional area need only be 0.25 of the primary line area to carry the same load. Typically, however, the secondary lines are chosen to have a cross-sectional area of 0.4 to 0.5 of the primary line area, so as to better cope with multiple primary and secondary line cuts in the same region of the tether. This redundant linkage enables the structure to redistribute loads around primary segments that fail due to meteorite strikes or material failure. Consequently, the Hoytether™ structure can be loaded at high stress levels, yet retain a high margin of safety [9].

Tether Mass Ratio

The mass of a rapidly spinning tether is determined primarily by the tip speed of the tether, not the tether length or the tether tip acceleration. In a rotating tether system, where the tether mass itself is part of the mass being rotated, adding mass to a tether to increase its strength also increases the load, thus limiting the tip motion to a given velocity level, not acceleration level. A short, fat tether will have the same tip velocity v_t as a long, skinny tether of the same mass. The acceleration felt by the payload at the tip of the tether will vary as the tether length L with:

$$a_c = v_t^2/L. \qquad (1)$$

Tether material has a "characteristic velocity" which depends on the ultimate strength and density of the material [2,9]:

$$v_u = (2U/d)^{1/2} \qquad (2)$$

For safety, this velocity can be reduced by dividing the ultimate strength by an engineering safety factor, so that the characteristic velocity for the derated material is:

$$v_c = (2(U/F)/d)^{1/2} \quad (3)$$

The engineering safety factor F to be used in different applications is discussed in detail by Hoyt[9] and is typically between 1.75 and 3.0.

The basic equation for the ratio of the mass M_T of one arm of a spinning tether to the mass M_P of the payload plus grapple on the end of the tether arm is [2,9]:

$$M_T/M_P = \pi^{1/2}(v_t/v_c) \exp[(v_t/v_c)^2] \operatorname{erf}(v_t/v_c) \qquad (4)$$

Where the error function $\operatorname{erf}(v_t/v_c) \approx 1$ for $v_t/v_c>1$

The material presently used for space tethers is a polyethylene polymer called Spectra™, which is commercially available in tonnage quantities as fishing net line. Although slightly stronger materials exist, and should be used when they become commercially available, we do not need them to make the MERITT system feasible. Spectra™ 2000 has an ultimate tensile strength of 4.0 GPa, a density of 970 kg/m^3, and an ultimate (F=1) characteristic velocity of = 2.9 m/s. Assuming that the grapple on the end of the tether masses 20% of the payload mass, we can use Equation (4) to calculate the mass ratio of a Spectra™ tether from its center-of-mass to the payload for various different safety factors and tether tip velocities, as in Table 1.

Table 1. Ratio of Spectra™ 2000 Tether Material Mass to Payload Mass
(Grapple Mass assumed to be 20% of Payload Mass)

	Tether Material Safety Factor (F)			
Tip Speed	1.75	2.0	2.4	3.0
2.0 km/s	3.7	4.7	6.4	10.0
2.5 km/s	8.0	11.0	17.0	30.0

From this table we can see that by using Spectra™ 2000, we can achieve tether tip velocities of 2.0 km/s with reasonable tether mass ratios (<10) and good safety factors. Higher tip velocities than 2.0 km/s are achievable using higher mass ratios, lower safety factors, and stronger materials.

Tether Orbital Energy Control

Following payload release, the tether system will be left in a less energetic orbit. To throw another payload, the lost orbital energy should be made up.

One of the major advantages of the MERITT system over rocket methods for getting to Mars is that once two-way traffic is established, the system can, in principle, be self-powered, with incoming payload capsules restoring energy and angular momentum lost by the tethers when throwing outgoing payloads. A payload thrown to Mars from a tether on Earth typically arrives with much more velocity than the tether can handle at feasible tip velocities, and trajectories have to use aerobraking or be deliberately deoptimized to allow capture.

But there is a trade in aerobraking capture between momentum gain by the capturing tether and mission redundancy. To make up for momentum loss from outgoing payloads, the tether would like to capture incoming payloads at similar velocities. That, however, involves hyperbolic trajectories in which, if the payload is not captured, it is lost in space. Also, in the early operations before extensive ballast mass is accumulated, care must be taken that the tether itself is not accelerated to hyperbolic velocities as a result of the momentum exchange.

Tethers orbiting planets with magnetic fields can also have a small conductive portion of the tether that would use electrodynamic tether propulsion[9,13], where electrical current pumped through the tether pushes against the magnetic field to add or subtract both energy and angular momentum, thus ultimately controlling the total energy and angular momentum of the orbiting tether system. This can be used to make up energy losses in a MERITT-type system, or perform orbital maneuvers about Jupiter.

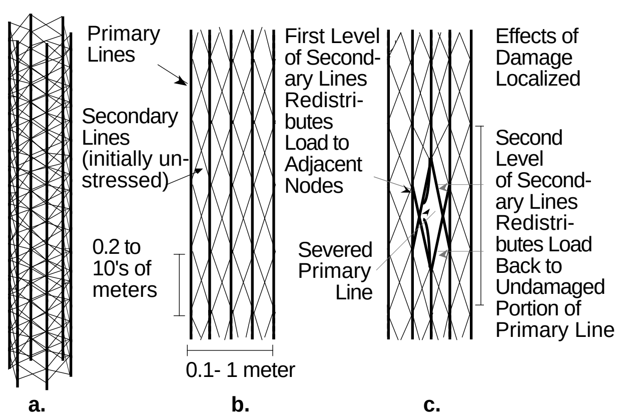

Figure 5. Hoytether damage resistance

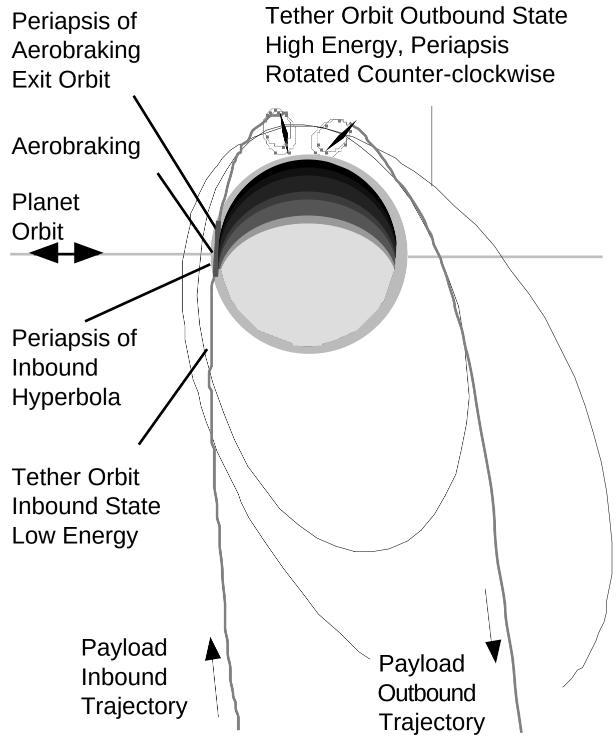

Figure 6. Tether concept for inbound and outbound payloads at Earth or Mars

MarsWhip tether energy management can be aided by including a solar arrays to power a tether winch to periodically change the tether length at the proper point in the MarsWhip elliptical trajectory [17,18], making the orbit more or less elliptical for the same angular momentum. This power could also run an fuel-efficient electric propulsion systems using in-situ resources from Mars or its moons.

We have not modeled long-term orbit perturbations of the planetwhip tethers, however, the changes made to the tether orbits from frequent payload capture and release would dwarf any perturbation effects, and minor adjustments in release angles and timing could be used to counter or enhance such effects as needed.

The large number of free parameters in this system produce a "good news/bad news aspect to analysis. The difficulty is that the problem is not self-defined and to make the model work, a number of arbitrary choices must be made. The good news is that this means there is a fair amount of operational flexibility in the problem and various criteria can be favored and trades made.

Fig. 6 shows how a single tether toss and catch system might work on either the Earth or Mars end of the MERITT system.

Rendezvous of Grapple with Payload

The tether rendezvous problem resembles one solved daily by trapeze acrobats, where one is caught in mid air by a colleague hanging from a trapeze bar. The catcher meets up with and grasps the "payload" after she has let go of her bar and is in a "free fall" trajectory accelerating with respect to the "catcher" at one gee. They time their swings, of course, so that they meet near the instant when both are at near zero relative velocity.

In tether rendezvous, the grapple velocity vector is arranged to match, as closely as possible, that of the payload at the time of rendezvous. Though their accelerations are different, the large radius of curvature of both trajectories makes the differential vertical motion near the time the curves are small (ideally, zero). The grapple mechanism on the end of a rotating tether is subjected to a centrifugal acceleration by the rotation of the tether, but may change its radial position by reeling itself in or out and change its lateral position with thrusters. As the time for capture approaches, the grapple could reel out and use its propulsion to fly ahead to the rendezvous point. As the payload comes along, it can reel tether in and out to match the curvature of the payload trajectory and adjust its position with thrusters to compensate for any lateral errors. In this manner, the rendezvous interval can be stretched to many tens of seconds.

The grapple batteries can be recharged from a solar array or from grapple winch motor/dynamos, by allowing the grapple winches to reel out while the central winches are being reeled in using the central station power supply.

In addition to having more time to perform its task, the tether grapple system will have many advantages over its human analog: GPS guidance, radar Doppler and proximity sensors, onboard divert thrusters, and the speed of electronic "synapses."

An essential first step in the development of the MERITT system would be the construction and flight test of a rotating tether-grapple system in LEO, having it demonstrate that it can accurately toss a dummy payload into a carefully selected orbit such that, n orbits later, the two meet again under conditions that will allow the grapple to catch the payload once again. Preliminary discussions were held by Dr. Forward with staff at the Automated Rendezvous and Capture (AR&C) Project Office at Marshal Space Flight Center (MFSC) and it appears that the present Shuttle-tested [STS-87 & STS-95] Video Guidance Sensor (VGS) hardware, and Guidance, Global Positioning System (GPS) Relative Navigation, and Guidance, Navigation and Control (GN&C) software could be modified for tether operations.

Tether System Construction

The EarthWhip tether can be built up incrementally, first serving to send small science payloads to Mars, while at the same time accumulating central facility mass by keeping upper stages and other unwanted masses. The Hoytether™ design also lends itself to incremental construction, not only in length but in thickness and taper, so that a 10, 20 or even 100 ton tether can be built out of a large number of 1 to 5 ton deploy-only canisters each containing a 10-20 km long section of tether.

Preliminary analysis also shows that a minimal mass MarsWhip can be tossed to Mars by a similar mass EarthWhip tether, arriving at Mars 180 days after toss. The MarsWhip could halt itself by use of an aerobraking module. Alternatively, it could employ the Landis [19] tether assisted planetary orbital capture procedure, where prior to close approach to Mars, the tether is deployed so that one end is ahead of and much closer to Mars than the other, pulling that end of the tether into a different trajectory than the other end. If properly done, the tether system gains rotational energy and angular momentum from the non-linear gravity-whip interaction, at the expense of its center-of-mass orbital energy and angular momentum, and thus ends up rotating around its center-of-mass, with the center-of-mass in a highly elliptical capture orbit around Mars. Once in the capture orbit, the MarsWhip tether can use tether pumping [17,18] to change the rotation rate of the tether and the ellipticity of its orbit to the desired values. After the MarsWhip is ready to receive incoming payloads, it can then be built up incrementally by catching modules.

MERITT SYSTEM SIMULATION MODEL

Calculations of the MERITT system performance were performed on Macintosh™ personal computers using the mathematical modeling software package TK Solver™ which allows the user to type in the relevant equations and get results without having to solve the model algebraically or structure it as a procedure, as long as the number of independent relationships equals the number of variables. This is very useful in a complex system when one may wish to constrain various variables for which it would be difficult, if not impossible, to solve and to perform numerical experiments to investigate the behavior of the system.

These initial models were intended to provide a quick, top-level look at the performance potential of the system and contained simplifying assumptions for speed and generality, including: coplanar keplerian orbits about point masses, Earth and Mars in circular orbits with a radius equal to their semimajor axes, and rigid tethers of constant length.

Two versions of a tether based interplanetary transfer system were modeled, one for tether-only transfers and the other incorporating an aerobraking pass at the destination body to aid in capture and rotation of the line of apsides. The general architecture of the models is sequential. A payload is added to a rotating tether in a highly elliptical orbit around the origin planet, released from the tether on an interplanetary trajectory, captured at the destination planet by another tether and released to a trajectory that allows descent to the target planet.

Tether Model

The tether is modeled as a rigid line with two arms, a grapple, a counterweight and a central mass. The tether is assumed to be designed for a payload with a given mass and a "safety factor" of two, as described in Hoyt and Forward [9] and to be dynamically symmetrical with a payload of that mass attached.

The mass distribution in the arms of the tether was determined by dividing the tether into ten segments, each massive enough to support the mass outward from its center; this was not needed for the loaded symmetric tether cases presented here, but will be useful in dealing with asymmetric counterweighted tethers. The total mass of each tether arm was determined from equation (4). The continuously tapered mass defined by equation (4) was found to differ by only a few percent from the summed segment mass of the 10 segment tether model used in the analysis, and the segment masses were adjusted accordingly until the summed mass fit the equation. The small size of this adjustment, incidentally, can be taken as independent confirmation of equation (4).

Shift in Tether Center of Mass with Payload Pickup and Release

It turns out that the dynamics of an ideal rigid tether system with a given payload can be fairly well modeled by simply accounting for the change in the position and motion of the tether's center of mass as the payload is caught and released. The position and velocity of the grapple end of the unloaded rotating tether is matched to the payload position and velocity of the payload as shown in the lower right part of Fig. 1.

When the payload is caught, the center of mass shifts toward the payload and the tether assumes its "design" state, with maximum tension at the center of rotation. The amount of the shift is determined by adding the moments of the unloaded tether about the loaded center of symmetry and dividing by the unloaded mass. The tip speed around this new "loaded" center-of-mass is simply its speed around the unloaded center of mass minus the speed of the point which became the new center of mass about the old center of mass, which is the angular rate times the shift distance.

This speed and the rotation angle at capture define a velocity vector which is transformed to the planetary frame and added to the velocity vector of the old center-of-mass to give the new velocity vector of the loaded tether system. The center-of-mass displacement and the rotation angle also provide a position vector in the frame of the old center-of-mass. This is transformed to the planetary frame of reference and added to the radius vector of the old center-of-mass to get the radius vector to the new center of mass in the planetary frame. These two vectors constitute the orbital state vector for the combined payload/tether system.

This state vector is converted to orbital elements which are propagated until the payload is released. At release, the center-of-mass shifts away from the grapple end of the tether by the same amount and the model calculates the final tether orbit , essentially by reversing the above procedure.

Since, in the outgoing case, the tether loses altitude with both the catch and the throw, its initial altitude must be high enough so that it does not enter the atmosphere after it throws the payload. This was done by defining the periapsis of the initial tether center-of-mass orbit as the sum of the planet's radius, the height of the sensible atmosphere (taken as 140 km for both Earth and Mars), the length of the unloaded tether arm and two center of mass shifts.

Released Payload Trajectory

The injection velocity vector in the planetary frame is simply the vector sum of the motion of the tether tip as a function of its rotation angle and that of the tether center-of-mass, displaced to the location of the tether tip. This velocity and position are converted to Keplerian orbital elements

Real passages through space take place in three dimensions. To the first order, however, transfer orbits are constrained to a plane incorporating the Sun, the origin planet at launch and the destination planet at arrival. The injection vector must occur in this plane, or close enough to it that on-board payload propulsion can compensate for any differences. This analysis considered only coplanar trajectories, but given the foregoing flexibility, this is not a great handicap.

As the payload moves out from the influence of the mass of the origin planet, its trajectory becomes more and more influenced by the mass of the Sun, until the origin planet's mass can be essentially neglected. Likewise, inbound payloads become more and more influenced by the destination planet mass until the mass of the Sun may be neglected. For first-order Keplerian analysis it is customary to treat the change of influence as if it occurred at a single point, called the patch point. For this model, the locus of patch points about a planet is approximated as a circle, the radius of which is equal to the distance from the planet away from the Sun (inner planet) or toward the Sun (outer planet) at which the combination of solar gravity, planetary gravity and solar frame centrifugal acceleration result in zero radial force. We call this the patch radius.

The outbound trajectory is propagated to this distance, the orbital elements are converted to a state vector, the vector is transformed to solar inertial coordinates, and the solar frame orbital elements are generated.

The angle the state vector at the patch point makes with a vector normal to the radius vector of the Sun to the planet's orbit, A, is a free choice at this point. For now, an estimate or “guess” of this quantity is made. The resulting vector is then converted into Sun frame orbital elements and propagated to the patch point near the orbit of the destination planet. The solar radius of the patch point is estimated by dividing the patch radius by the sine of the flight path angle (φ_d in Fig. 5) of the solar frame trajectory at the planet's semimajor axis and subtracting that from the semimajor axis. There, it is transformed into the destination planet coordinates.

When a tether only is used to receive the payload (Fig 3.), a constraint exists on the destination end; the incoming trajectory is a hyperbola and the periapsis velocity of the hyperbolic orbit must not exceed the maximum tip velocity of the capture tether. This periapsis velocity is determined by the vector sum of the capture tether's orbital motion and the tip velocity as a function of the tether rotation angle, "q". This defines the hyperbolic excess velocity of the incoming payload. The argument of periapsis of the destination tether center-of-mass can be found from the true anomaly of the incoming orbit at the patch point (essentially u_∞) and the angle that transfer orbit makes with the Sun-planet radius transformed to the planet's frame of reference.

The inbound hyperbolic excess velocity is also given by the vector sum of the orbital velocity of the destination planet, and that of the intersecting payload orbit at the patch point. The angle of injection at the Earth end, "A" in Fig. 1, is iterated until a match exists.

When passage through the atmosphere of the destination planet (aerobraking) is used to remove some of the incoming velocity, the constraint becomes an engineering issue of how much velocity can be lost. Experience with the Apollo mission returns (circa 12 km/s) and the Mars Pathfinder landing indicates that, with proper design, much more velocity can be dissipated than is required to assist tether capture, and minimum time transfers can be used. In this case, the injection angle "A" is iterated until a minimum time is found.

Once "A" is determined, it can be used to define the argument of periapsis of the departure orbit with respect to the Sun-planet radius. This, with the tether rotation rate, the time of release, and the initial tether orbit are used to define the argument of periapsis of the initial tether orbit.

Payload Capture and Release at Destination

After the payload is caught, the center of mass of the tether shifts and the effective length of the tether from center of mass to the payload catching tip is shortened, which is the reason for the two different radii circles for the rotating tether in the diagram. The orbit of the tether center of mass changes from a low energy elliptical orbit to a higher energy elliptical orbit with its periapsis shifted with respect to the initial orbit. The tether orbit would thus oscillate between two states: 1) a low-energy state wherein it would be prepared to absorb the energy from an incoming payload without becoming hyperbolic and 2) a high-energy state for tossing an outgoing payload.

In capture, the estimated tether tip capture position and velocity, together with the radius at which the outgoing payload resumes a ballistic trajectory, define a post aerobraking orbit which results in tether capture. The difference in the periapsis velocity of this orbit and the periapsis velocity of the inbound trajectory is the velocity that must be dissipated during the aerodynamic maneuver. The angle traversed is a free parameter that depends on choices for deceleration limits and the aerodynamic capabilities of the atmosphere transiting payload. For Mars bound trajectories, this aerobraking Δv is on the order of 5 km/s, as compared to direct descent Δv's of 9 km to 15 km/s. Also, payloads meant to be released into suborbital trajectories already carry heat shields, though designed for lower initial velocities.

As shown in Fig. 4, the radius at which the atmosphere of the destination planet is dense enough to sustain an aerodynamic trajectory is used to define the periapsis of the approach orbit. Note that the capture of a payload exiting the atmosphere at the high end of the tether is necessarily off the vertical line, and so rotates the periapsis of the tether/payload system center-of-mass.

After the tether tip and the incoming payload are iteratively matched in time, position and velocity, the center of mass orbit of the loaded tether is propagated to the release point. This is another free choice, and the position of the tether arm at release determines both the resulting payload and tether orbit. In this preliminary study, care was taken to ensure that the released payload did enter the planet's atmosphere, the tether tip did not, and that the tether was not boosted into an escape orbit.

Tether Transport Point Cases

We ended up designing many candidates for the EarthWhip and MarsWhip tethers, from some with very large central station masses that were almost unaffected by the pickup or toss of a payload, to those that were so light that the toss of an outgoing payload caused their orbits to shift enough that the tether tip hit the planetary atmospheres, or the catch of an incoming payload sent the tether (and payload) into an escape trajectory from the planet. After many trials, we found some examples of tethers that were massive enough that they could toss and catch payloads without shifting into undesirable orbits, but didn't mass too much more than the payloads they could handle.

We then looked at a number of MERITT missions using a wide range of assumptions for the tether tip speed and whether or not aerobraking was used. The trip times for the various scenarios are shown in Table 3. As can be seen from Table 3, the system has significant growth potential. If more massive tethers are used, or stronger materials become available, the tether tip speeds can be increased, cutting the transit time even further. The transit times in Table 3 give the number of days from payload pickup at one planet until payload reentry at the other planet, and include tether "hang time" and coast of the payload between the patch points and the planets. Faster transit times can be made with higher energy initial orbits for the payload and the tether.

Table 3. Potential MERITT Interplanetary Transfer Times*

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Aerobraking	Earth to Mars	Mars to Earth	Earth to Mars	Mars to Earth
tip velocity (m/s)	2500	2500	3000	3000
payload vel. at capture (m/s)	7979	2015	7474	1510
payload vel. at release (m/s)	12979	7015	12920	7364
hyperbolic excess velocity	7732	5437	8258	5967
patch to patch time (days)	84.94	118.74	80.74	110.59
periapsis velocity (m/s)	13591	14934	14175	15418
aerobraking Δv (m/s)	6643	1798	7028	1819
velocity at capture (m/s)	4305	10207	4517	10211
total trip time (days)	90.64	124.92	85.95**	114.29

Non Aerobraking	Earth to Mars	Mars to Earth	Earth to Mars	Mars to Earth
tip velocity (m/s)	2500	2500	3000	3000
payload vel. at capture (m/s)	7979	2015	7474	1510
payload vel. at release (m/s)	12979	7015	12919	7359
hyperbolic excess velocity	7724	5437	8249	5964
patch to patch time (days)	123.26	125.20	114.08	115.73
periapsis velocity (m/s)	7020	12749	7525	13254
velocity at capture	4520	10249	4525	10254
total trip time (days)	130.70	132.04	118.57	119.95

*** planets are assumed to be at their average radius from the sun**

**≈ 70 Days, favorable opposition

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Rough launch windows for the next twenty years were found for four cases using the hyperbolic excess velocities, vh, given by the above model, assuming a fixed patch to patch transit time, Δt, and generating transfer ellipses for every tenth month. The Δv needed to enter such a transfer ellipse was compared to vh and the window was assumed to exist where the Δv was less than than vh. By successively shrinking Δt until a window no longer existed, a rough minimum transit time of 64 days and total trip time of 70 days was found for 9 Jun 2018.

Synergistic Multipayload Assistance by Rotating Tether

The concept of the rotating tether in an elliptical orbit can be extended to a rotating tether in a hyperbolic orbit. If the payloads are separated near periapsis at the right velocity, one payload will be captured and the other given an additional booset to its destination, as illustrated in figure 7.

An object on an efficient elliptical transfer orbit from Earth's orbit to Jupiter's is moving roughly 5.6 km/s slower than Jupiter. Taking this as the hyperbolic excess velocity of an orbit with a periapsis rp the periapsis velocity can be obtained by solving the equation for hyperbolic excess velocity (Bate) for vp:

(5) v_{inf} = sqrt(vp^2 - 2*mu/rp)

(6) vp^2 = v_{inf}^2 + 2*mu/rp

where mu is the gravitational parameter of the planet. If the period of rotation of the tether is arranged so that it is posigrade and vertical at periapsis when the tether separates, the tip velocity of the tether will be subtracted from the lower payload and added to the upper one. As can be seen by inspection of equation (1), addition of a small number to a number that is squared has a disproportionate effect. In this case, the outgoing payload acquires a v_{inf} of

(7) v_{inf}^2 = (vp+vtip)^2 - 2*mu/rp

The velocity of the lower payload is reduced. If vp^2 is close to -2*mu/rp, then v_{inf} becomes imaginary, which is to say that the energy of the orbit has become negative with respect to infinity. The lower payload is thus captured. Again, by inspection, one sees that one can produce this state of affairs by making rp small enough. One should note that the payloads are not released precisely at rp, but are lower or higher due to the length of the tether. Unless the tether length is a significant fraction of the Jovian radius, this can be ignored for first order work. In keeping with the 2001 theme of this conference, Gerald Nordley studied a combined Jupiter/Pluto mission with a personal computer model based on the above and got the following results.

Table 1: Dual Payload Tether Jupiter/Pluto Performance

tip vel. m/s	Jupiter P/L orbit per. days	Pluto transfer injection vel km/s	Pluto transfer coast time years
400.	1473.82	21.22	10.24
500.	227.23	21.73	9.72
600.	101.55	22.21	9.28
700.	60.36	22.67	8.90
800.	41.09	23.11	8.58
900.	30.28	23.53	8.30
1000.	23.51	23.94	8.04

*Based on near-minimum encounter velocity of 5.63 km/s, a perijove of 2 R_J, Pluto at 31 AU at time of encounter, a zero flight path angle leaving Jupiter, and no encounters with monoliths

At 2 R_J, the velocity of the incoming tethered system reaches about 42.5 km/s. With a 1 km/s increase in this velocity for the upper "Pluto" payload, the payload reaches a hyperbolic excess velocity with respect to Jupiter of 10.9 km/s, almost double what it had as it entered Jupiter's gravity field. If, for simplicity, this velocity is added in parallel to Jupiter's orbital velocity, one gets a periapsis velocity for the Pluto transfer orbit of 23.94 km/s. Using the time of flight equations in Bate for periapsis to a given radius, one gets the coast times in the final column.
Strictly speaking, this transfer time is for the approximate distance from the sun of Pluto in some 14 years; no effort was made to work out the phasing or calculate an actual launch windows. Initial comparisons with a rough model of an equivalent rocket mission yielded roughly comparable masses, however improvements in tether material strength and synergistic use of the tether in an electromagnetic mode for Jupiter orbit propulsion may make this an attractive option.

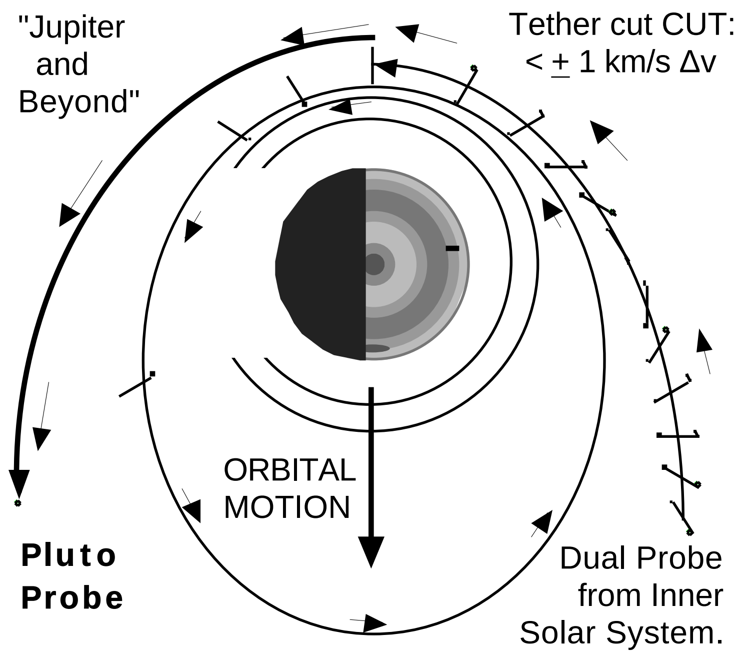

Figure 7. Hyperbolic Dual Payload Tether Concept: Jupiter and Beyond.

CONCLUSIONS

We have shown that two ideal rigid tethers rotating tethers in highly elliptical orbits about Earth and Mars can provide rapid interplanetary transport from a suborbital trajectory above the Earth's atmosphere to a suborbital trajectory above the Martian atmosphere and back. Real tether materials have both elasticity and damping. The Hoytether™ structure then adds its own damping and a non-linear elasticity and strength response as the secondary strands come into play after sufficient elongation. Then, depending upon the placement of intermediate masses along the tether, the long tether structure would have libration, pendulum, and skip-rope modes, plus longitudinal, transverse, and torsional vibrational modes. Additional analysis is needed to study the excitation of those modes, ways to minimize the excitation, and how the existence of high amplitude oscillations of those modes could affect the accuracy of the catch-and-throw operations. Synergistic use of a rotating tether between parts of a payload for artificial gravity for additional velocity increments in encounters with other planets looks potentially attractive but needs further study. Use of reels and grappling fixtures already developed for MERITT would reduce the technological risk.

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APPENDIX Q

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APPLICATION OF SYNERGISTIC MULIPAYLOAD ASSISTANCE WITH ROTATING TETHERS (SMART) CONCEPT TO OUTER PLANET EXPLORATION. Gerald David Nordley[1], Robert L. Forward[2] and Robert P. Hoyt[2], [1]Consultant <GDNordley@aol.com>, [2]Tethers Unlimited, Inc., 19011 36th Ave. W, Suite F, Lynnwood, WA 98036 <TU@tethers.com> [www.tethers.com].

Introduction: We propose an innovative approach to outer planet exploration using the Synergistic Multipayload Assistance with Rotating Tethers (SMART) concept invented by Gerald David Nordley. The basic concept can be implemented in many different ways to accomplish many different types of planetary missions, especially missions to the outer planets.

SMART Concept: A pair of spacecraft are connected by a tether, set to rotating about their common center of mass, and injected into a hyperbolic orbit around a massive planet. The tether is caused to separate as the combined system approaches the periapsis of the hyperbolic orbit. The spacecraft which is rotating "backward" relative to the hyperbolic flight path receives a velocity decrease at separation which causes it to go into a lower energy orbit, typically an elliptical capture orbit about the planet. The spacecraft that is rotating "forward " with respect to the hyperbolic flight path gets a velocity increase which will cause it to exit the gravitational field of the planet with a higher velocity than it entered. In effect, each payload acts as the reaction mass for the other. Because the velocity increments obtained from the tip speed of the tether take place deep in the gravity well of the massive planetary body, they are "amplified" by the high periapsis velocity to produce significant changes in the final trajectories of the two separated bodies.

Combined Pluto/Europa Mission: Figure 1 illustrates the use of the SMART concept to carry out both the Pluto Flyby mission and the Europa Orbiter/Lander mission with one launch. The combined system arriving from Earth is moving 5.6 km/s slower than Jupiter. The velocity of the incoming system reaches 42.5 km/s at a periapsis of two Jupiter radii. With the tether giving the Pluto payload a 1 km/s increase in this velocity, the Pluto payload reaches a hyperbolic excess velocity with respect to Jupiter of 10.9 km/s, almost double what it had as it entered Jupiter's gravity field. Adding this velocity to Jupiter's orbital velocity gives a velocity of 23.9 km/s for the Pluto injection velocity. For the Europa payload, a ΔV of only 400 m/s from the tether is sufficient to provide capture into an elliptical orbit about Jupiter. Table I shows the results on the parameters of the two missions of using different tether tip speeds.

Table I - Post-Periapsis Payload Trajectories

Tether Tip Velocity (km/s)	Capture Orbit (days)	Pluto Inj. Velocity (km/s)	Pluto Trip Time (years)	Tether Mass Ratio
0.0	∞	18.7	11.6	-
0.4	1473	21.2	10.2	0.05
0.6	102	22.2	9.3	0.12
0.8	41	23.1	8.6	0.22
1.0	24	23.9	8.0	0.36

The tether mass ratio was determined using the well-known tether mass ratio formulas for fail-safe interconnected multistrand tethers [1] and assuming a 50% improvement in tether strength in the coming years.

Other Outer Planet Applications: The SMART concept can be used in many other ways than a dual mission. Either the Pluto mission or the Europa mission, or any other mission to any other planet could use the dead mass of the Earth escape injection stage as reaction mass. Any mission putting a payload into orbit around Jupiter could retain a conductive portion of the tether and use it to obtain both power and propulsion. With the tether available to provide the instantaneous thrust at periapsis, any mission could be redesigned to use efficient electric propulsion and completely eliminate the need for chemical propulsion.

Comparison With Rocket Assist: The mass of the tether necessary to obtain the necessary ΔV at periapsis in the SMART concept is typically comparable to the mass of the storable propellant and tanks needed to obtain a comparable ΔV. Detailed analyses will be required to determine the exact mass comparison numbers for each mission example. Rockets have more flight heritage, but one would think that the reliability and accuracy of a tether system that imparts all if its exactly known mechanical energy at a single point in time by the action of a simple mechanical separation system would be better than the release of an uncertain amount of chemical energy over a long burn time initiated by a complex ignition sequence.

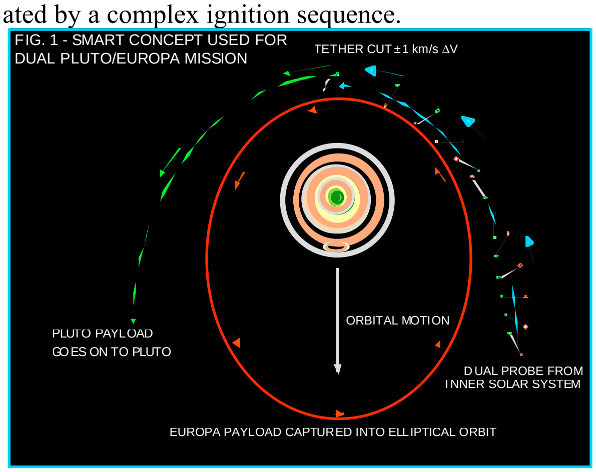

FIG. 1 - SMART CONCEPT USED FOR DUAL PLUTO/EUROPA MISSION

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Mars-Earth Rapid Interplanetary Tether Transport (MERITT) Architecture

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Abstract. Routine travel to and from Mars demands an efficient, rapid, low cost means of two-way transportation. To answer this need, we have invented an architecture consisting of two spinning tether systems in highly elliptical orbits about each planet. At Earth, a payload is picked up near periapsis and tossed a half-rotation later, still near periapsis, at a velocity sufficient to send the payload on a high-speed trajectory to Mars. At Mars, it is caught near periapsis and is released a short time later on a suborbital reentry trajectory. The system works in both directions and is reusable. Energy and momentum lost by the throwing tethers can be restored either by catching incoming payloads or by propellantless tether propulsion methods. Tethers with tip velocities of 2.5 km per second can send payloads to Mars in as little as 90 days if aeroslowing is used at Mars. Tether-to-tether transfers without aeroslowing may be accomplished in about 130 to 160 days. Tether systems using commercially available tether materials at reasonable safety factors can be as little as 15 times the mass of the payload being handled

INTRODUCTION

The idea of using orbiting spinning tethers to pick up and toss payloads has been in the tether literature for decades (Cosmo and Lorenzini, 1997). Forward (1991) and Hoyt (1997) have combined a number of spinning tether concepts to show that such tethers could move payloads from a suborbital trajectory just above the Earth's atmosphere to the surface of the Moon and back again, without any use of rockets except to get out of the Earth's atmosphere. Forward and Nordley (1999) then showed in their paper "Mars-Earth Rapid Interplanetary Tether Transport (MERITT) System", that orbiting spinning tethers would also provide two-way transport between Earth and Mars. Work on the Earth-Moon and Earth-Mars systems is continuing under a $500,000 grant from the NASA Institute for Advanced Concepts, and the presentation at the meeting will cover the latest results from that study.

MERITT ARCHITECTURE DESCRIPTION

The Mars-Earth Rapid Interplanetary Tether Transport (MERITT) Architecture consists of two essentially identical rapidly spinning tether facilities in highly elliptical orbits: EarthWhip around Earth and MarsWhip around Mars. A payload capsule is launched from Earth into a low orbit or suborbital trajectory. The payload is picked up by a grapple system on the EarthWhip tether as the tether nears perigee and the tether arm nears the lowest part of its swing. It is tossed later when the tether is still near perigee and the arm is near the highest point of its swing. The payload thus gains both velocity and potential energy at the expense of the tether system, and its resulting velocity is sufficient to send it on a high-speed trajectory to Mars with no onboard propulsion needed except for midcourse guidance. At Mars, the incoming payload is caught in the vicinity of periapsis by the grapple end of the MarsWhip tether near the highest part of its rotation and greatest velocity with respect to Mars. The payload is released later when the tether is near periapsis and the grapple end is near the lowest part of its swing at a velocity and altitude which will cause the released payload to enter the Martian atmosphere. The system works in both directions. The MERITT system can give shorter trip times with aerobraking at Mars because the incoming payload velocity is not limited by the maximum tether tip velocity and thus payloads can use faster interplanetary trajectories.

DETAILED MERITT EXAMPLE

There are a large number of variables in the MERITT system concept, and many of those variables can be freely chosen at the start of the system design. We have carried out dozens of complete round-trip scenarios under various different assumptions, such as: aerobraking before tether catch versus direct tether-to-tether catch; sub-, circular, and elliptical initial and final payload orbits; 1.5, 2.0, 2.5 and higher tether tip velocities; large, small and minimum tether central facility masses; etc. We will present here just one of the many possible MERITT scenarios using finite mass EarthWhip and MarsWhip tethers, but do it in extensive detail so the reader can understand where the broad assumptions are, while at the same time appreciating the accuracy of the simulations between the broad assumptions. In most cases, the matches between the payload trajectories and the tether tip trajectories are accurate to 3 and 4 decimal places.

The scenario we will describe uses EarthWhip and MarsWhip tethers of near minimum mass made of Spectra™ 2000 with a tip speed of 2.0 km/s. Because they have small total masses, the toss and catch operations significantly affect the tether rotation speed, center of mass, and orbital parameters, all of which are taken into account in the simulation. The payload is assumed to be initially launched from Earth into a suborbital trajectory to demonstrate to the reader that the MERITT system has the capability to supply all of the energy and momentum needed to move the payload from the upper atmosphere of the Earth to the upper atmosphere of Mars and back again. We don't have ask the payload to climb to nearly Earth escape before the MERITT system takes over. In practice, it would probably be wise to have the payload start off in an initial low circular orbit. The energy needed to put the payload into a low circular orbit is not that much greater than the energy needed to put the payload into a suborbital trajectory with an apogee just outside the Earth's atmosphere. The circular orbit option also has the advantage that there would be plenty of time to adjust the payload orbit to remove launch errors before the arrival of the EarthWhip tether. In the example scenario, the payload, in its suborbital trajectory, is picked up by the EarthWhip tether and tossed from Earth to Mars. At Mars it is caught by the MarsWhip tether without the use of aerobraking, and put into a trajectory that enters the Martian atmosphere at low velocity. Since this scenario does not use aerobraking, the return scenario is just the reverse of the outgoing scenario.

Payload Mass

We have chosen a canonical mass for the payload of 1000 kg. If a larger payload mass is desired, the masses of the tethers scale proportionately. The scenario assumes that the payload is passive during the catch and throw operations. In practice, it might make sense for the payload to have some divert rocket propulsion capability to assist the grapple during the catch operations. In any case, the payload will need some divert rocket propulsion capability to be used at the midpoint of the transfer trajectory to correct for injection errors.

Tether Mass

Both the EarthWhip and MarsWhip tethers were assumed to consist of a robotic central station, two similar tethers, two grapples at the ends of the two tethers, and, to make the analysis simpler, one grapple would be holding a dummy payload so that when the active payload is caught, the tether would be symmetrically balanced. The tether central station would consist of a solar electric power supply, tether winches, and command and control electronics. There may be no need to use center of mass rocket propulsion for ordinary tether operations. Both tethers can be adequately controlled in both their rotational parameters and center-of-mass orbital parameters by "gravity-gradient" propulsion forces and torques generated by changing the tether length at appropriate times in the tether orbit. The EarthWhip tether would also have a small conductive portion of the tether that would use electrodynamic tether propulsion, where electrical current pumped through the tether pushes against the magnetic field of the Earth to add or subtract both energy and angular momentum from the EarthWhip orbital dynamics, thus ultimately maintaining the total energy and angular momentum of the entire MERITT system against losses without the use of propellant.

The grapple mechanisms are assumed in this scenario to mass 20% of the mass of the payload, or 200 kg for a 1000 kg payload. It is expected, however, that the grapple mass will not grow proportionately as the payload mass increases to the many tens of tons needed for crewed Mars missions. In the scenario presented here, it is assumed that the grapples remain at the ends of the tethers during the rendezvous procedure. In practice, the grapples will contain their own tether winches powered by storage batteries, plus some form of propulsion. As the time for capture approaches, the grapple, under centrifugal repulsion from the rotation of the tether, will release its tether winches, activate its propulsion system, and fly ahead to the rendezvous point. It will then reel in tether as needed to counteract planetary gravity forces in order to "hover" along the rendezvous trajectory, while the divert thrusters match velocities with the approaching payload. In this manner, the rendezvous interval can be stretched to many tens of seconds. If needed, the rendezvous interval can be extended past the time when the tip of the tether passes through the rendezvous point by having the grapple let out tether again, while using the divert thrusters to complete the payload capture. The grapple batteries can be recharged between missions by the grapple winch motor/dynamos, by allowing the grapple winches to reel out while the central winches are being reeled in using the central station power supply. The grapple rocket propellant will have to be resupplied either by bringing up "refueling" payloads or extracting residual fuel from payloads about to be deorbited into a planetary atmosphere.

For this scenario, we assumed that, when loaded with a payload, the EarthWhip and MarsWhip tethers were spinning with a tether tip speed of 2,000 m/s. The length of each tether arm was chosen as 400 km in order to keep the acceleration on the payload near one gee. We also assumed that the total mass of the Whips would be 15,000 kg for a 1000 kg payload (16,000 kg total). This mass includes the central station, both tethers, the grapples at the ends of the tethers, and the dummy payload mass. This is about the minimum tether mass needed in order for the tether center-of-mass orbits to remain stable before and after a catch of a payload with a velocity difference of 2000 m/s.

The tether material was assumed to be Spectra™ 2000 with an ultimate tensile strength of 4.0 GPa, a specific density of 0.97, and an ultimate tip velocity for an untapered tether of 2872 m/s. The tether safety factor was initially chosen at 2.0, which results in a engineering characteristic velocity for the tether of 2031 m/s. The mass ratio of one arm of a tapered Spectra™ 2000 tether was calculated to be 3.84 times the mass at the tip of the tether. Since the mass at the end of the tether consists of the 1000 kg payload and the 200 kg grapple, the minimum total mass of one tether arm is 4609 kg, or about 4.6 times the mass of the 1000 kg payload. The amount of taper is significant, but not large. The diameter of the tether at the tip, where it is holding onto the payload, is 2.8 mm, while at the base, near the station, it is 4.7 mm in diameter.

Since the EarthWhip and MarsWhip tethers are under the most stress near periapsis, when they are closest to their respective planets, we need to take into account the small additional stress induced by the gravity gradient forces of the planets, which raises the mass to about 4750 kg for a 1000 kg payload. We will round this up to 4800 kg for the tether material alone, corresponding to a free-space safety factor of 2.04, so that the total mass of the tether plus grapple is an even 5000 kg. With each tether arm massing 5000 kg including grapple, one arm holding a dummy payload of 1000 kg, and a total mass of 15,000 kg, the mass of the central station comes out at 4000 kg, which is a reasonable mass for its functions. There are a large number of tether parameter variations that would work equally well, including shorter tethers with higher gee loads on the payloads, and more massive tethers with higher safety factors. All of these parameters will improve as stronger materials become commercially available, but the important thing to keep in mind is that the numbers used for the tethers assume the use of Spectra™ 2000, a commercial material sold in tonnage quantities as fishing nets, fishing line (SpiderWire), and kite line (LaserPro). We don't need to invoke magic materials to go to Mars using tethers.

Tether Rotational Parameters

When the Whips are holding a payload, they are symmetrically balanced. The center-of-mass of the tether is at the center-of-mass of the tether central station. The effective arm length from the tether center-of-mass to the payload is 400,000 m, the tip speed is exactly 2000 m/s and the rotation period is 1256.64 s = 20.94 min = 0.3491 hr. When the Whips are not holding a payload, then the center-of-mass of the Whip shifts 26,667 m toward the dummy mass tether arm, and the effective length of the active tether arm becomes 426,667 m, while the effective tip velocity at the end of this longer arm becomes 2,133 m/s. (Since there is no longer a payload on this arm, the higher tip velocity can easily be handled by the tether material.) The rotational period in this state is the same, 1256.64 s.

Payload Initial Trajectory Parameters

The Earth-launched payload trajectory chosen for this example scenario is a suborbital trajectory with an apogee altitude of 203.333 km (6581.333 km radius) and a apogee velocity of 7,568 m/s. The circular orbit velocity for that radius is 7,782 m/s.

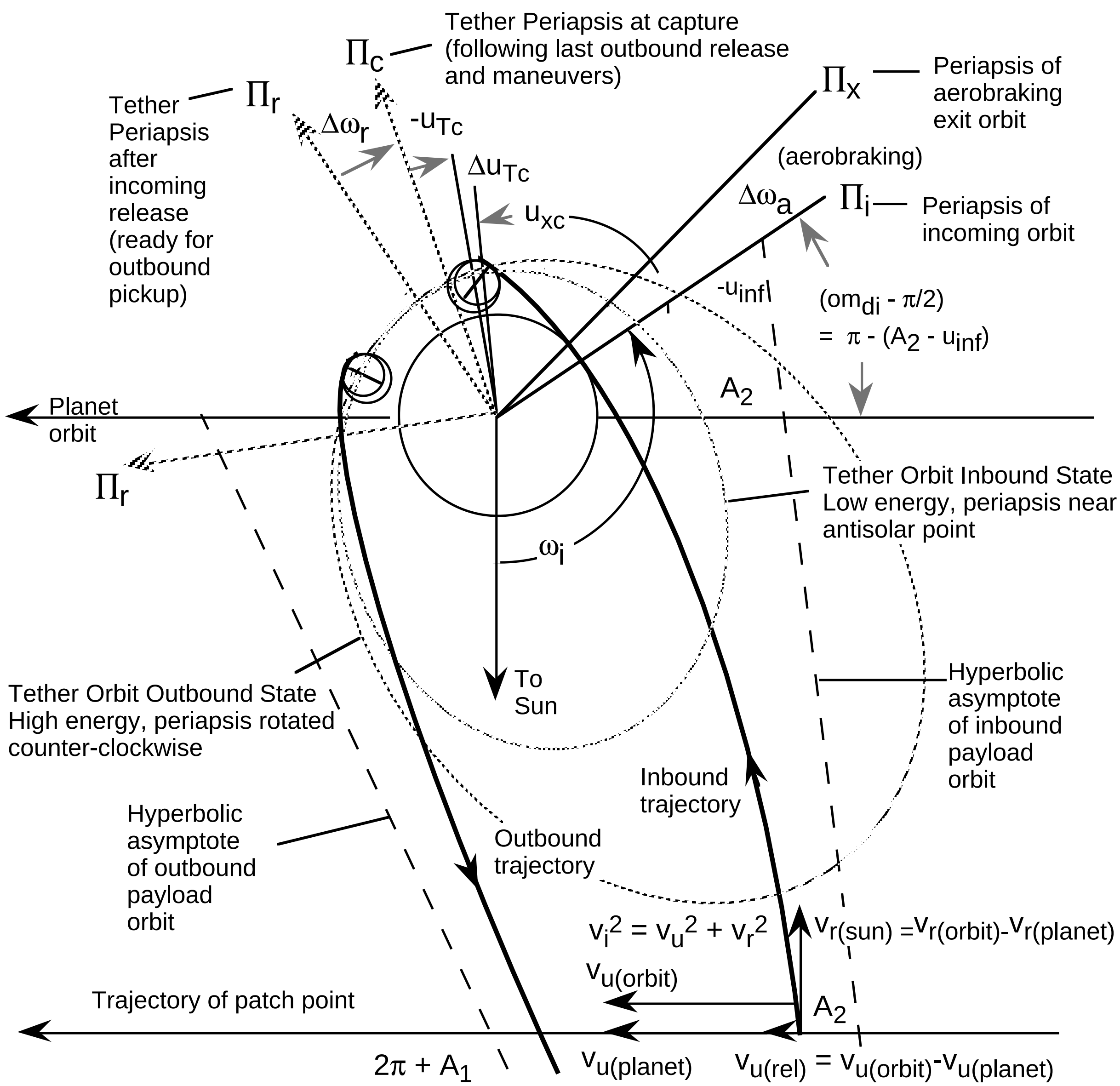

FIGURE 1. EarthWhip or MarsWhip Showing Catch and Toss States of Orbit

EarthWhip

The EarthWhip starts out in an unloaded state with an effective length for its active arm of 426,667 m from the center-of-rotation, a tip velocity of 2,133 m/s and a rotational period of 1256.64 s. As shown in Figure 1, the center-of-mass of the EarthWhip is in a highly elliptical orbit with an apogee of 33,588 km (almost out to geosynchronous orbit), an eccentricity of 0.655, an orbital period of exactly 8 hours, a perigee radius of 7008 km (630 km altitude), and a perigee velocity of 9,701 m/s. The tether rotational phase is adjusted so that the active tether arm is pointing straight down at perigee, with the tether tip velocity opposing the center-of-mass velocity. The tip of the tether is thus at an altitude of 630 km-426.7 km = 203.3 km and a velocity with respect to the Earth of 9,701 m/s - 2,133 m/s = 7,568 m/s, which matches the payload altitude and velocity. After picking up the payload, the loaded EarthWhip tether is symmetrically balanced. Since the added payload had both energy and momentum appropriate to its position on the spinning tether, the EarthWhip rotation angular rate does not change and the period of rotation remains at 1257 s. The center of mass of the loaded EarthWhip, however, has shifted to the center of the tether central station, so the effective length of the loaded tether arm is now at its design length of 400,000 m and tip velocity of 2,000 m/s. With the addition of the payload, however, the orbit of the tether center-of-mass has dropped 26.7 km to a perigee of 6981.3 km, while the perigee velocity has slowed to 9,568 m/s. The apogee of the new orbit is 28,182 km and the eccentricity is 0.603, indicating that this new orbit is less eccentric than the initial orbit due to the payload mass being added near perigee. The period is 23,197 s or 6.44 hours.

Payload Toss

The catch and toss operation at the Earth could have been arranged as shown in Figure 1, so that the payload catch was on one side of the perigee and the payload toss was on the other side of the perigee, a half-rotation of the tether later (10.5 minutes). To simplify the mathematics for this initial analysis, however, we assumed that the catch occurred right at the perigee, and that the tether holds onto the payload for a full orbit. The ratio of the tether center-of-mass orbital period of 23,197 s is very close to 18.5 times the tether rotational period of 1256.64 s, and by adjusting the length of the tether during the orbit, the phase of the tether rotation can be adjusted so that the tether arm holding the payload is passing through the zenith just as the tether center-of-mass reaches its perigee. The payload is thus tossed at an altitude of 603 km + 400 km =1003 km (7381 km radius), at a toss velocity equal to the tether center-of-mass perigee velocity plus the tether rotational velocity or 9,568 m/s + 2,000 m/s = 11,568 m/s. In the combined catch and toss maneuver, the payload has been given a total velocity increment of twice the tether tip velocity or Δv=4,000 m/s.

EarthWhip After Payload Toss

After tossing the payload, the EarthWhip tether is back to its original mass. It has given the payload a significant fraction of its energy and momentum. At this point in the analysis, it is important to insure that no portion of the tether will intersect the upper atmosphere and cause the EarthWhip to deorbit. We have selected the minimum total mass for the EarthWhip at 15,000 kg to insure that doesn't happen. The new orbit for the EarthWhip tether has a perigee of its center of mass of 6955 km (577 km altitude), apogee of 24,170 km, eccentricity of 0.552, and a period of 5.37 hours. With the new perigee at 577 km altitude, even if the tether rotational phase is not controlled, the tip of the active arm of the tether, which is at 426.67 km from the center-of-mass of the tether, does not get below 150 km from the surface of the Earth where it might experience atmospheric drag. In practice, the phase of the tether rotation will be adjusted so that at each perigee passage, the tether arms are roughly tangent to the surface of the Earth so that all parts of the tether are well above 500 km altitude, where the air drag and traffic concerns are much reduced.

With its new orbital parameters, the EarthWhip tether is in its "low energy" state. There are two options then possible. One option is to keep the EarthWhip in its low energy elliptical orbit to await the arrival of an incoming payload from Mars. The EarthWhip will then go through the reverse of the process that it used to send the payload from Earth on its way to Mars. In the process of capturing the incoming Mars payload, slowing it down, and depositing it gently into the Earth's atmosphere, the EarthWhip will gain energy which will put it back into the "high energy" elliptical orbit it started out in. If, however, it is desired to send another payload out from Earth before there is an incoming payload from Mars, then the solar electric power supply on the tether central station can be used to generate electrical power. This electrical power can then be used to restore the EarthWhip to its high energy elliptical orbit using either electrodynamic tether forces to push against the magnetic field of the Earth or tether length "pumping" to "push" against the gravity gradient field of Earth or Mars.

Payload Trajectory

The velocity gain of $\Delta v \approx$4,000 m/s given the payload deep in the gravity well of Earth results in a hyperbolic excess velocity of 5,081 m/s. The payload moves rapidly away from Earth and in 3.3 days reaches the "patch point" on the boundary of the Earth's "sphere of influence," where the gravity attraction of the Earth on the payload becomes equal to the gravity attraction of the Sun on the payload. An accurate calculation of the payload trajectory would involve including the gravity field of both the Sun and the Earth (and the Moon) all along the payload trajectory. For this simplified first-order analysis, however, we have made the assumption that we can adequately model the situation by just using the Earth gravity field when the payload is near the Earth and only the Solar gravity field when we are far from the Earth, and that we can switch from an Earth-centered frame to a Sun-centered frame at the "patch point" on the Earth's "sphere of influence."

When this transition is made at the patch point, we find that the payload is on a Solar orbit with an eccentricity of 0.25, a periapsis of 144 Gm and an apoapsis of 240 Gm. It is injected into that orbit at a radius of 151.3 Gm and a velocity of 32,600 m/s. (The velocity of Earth around the Sun is 29,784 m/s.) It then coasts from the Earth sphere-of-influence patch point to the Mars sphere-of-influence patch point, arriving at the Mars patch point at a radius of 226.6 Gm from the Sun and a velocity with respect to the Sun of 22,100 m/s. (The velocity of Mars in its orbit is 24,129 m/s.) The elapsed time from the Earth patch point to the Mars patch point is 148.9 days.

At the patch point, the analysis switches to a Mars frame of reference. The payload starts its infall toward Mars at a distance of 1.297 Gm from Mars and a velocity of 4,643 m/s. It is on a hyperbolic trajectory with a periapsis radius of 4451 km (altitude above Mars of 1053 km) and a periapsis velocity of 6,370 m/s. The radius of Mars is 3398 km and because of the lower gravity, the atmosphere extends out 200 km to 3598 km. The infall time is 3.02 days.

MarsWhip

The MarsWhip tether is waiting for the arrival of the incoming high velocity payload in its "low energy" orbital state. The active tether arm is 426,667 m long and the tip speed is 2,133 m/s. The center-of-mass of the unbalanced tether is in an orbit with a periapsis radius of 4025 km (627 km altitude), periapsis velocity of 4,236 m/s, apoapsis of 21,707 km, eccentricity of 0.687, and a period close to 0.5 sol. (A "sol" is a Martian day of 88,775 s, about 39.6 minutes longer than an Earth day of 86,400 s. The sidereal sol is 88,643 s.) The orbit and rotation rate of the MarsWhip tether is adjusted so that the active arm of the MarsWhip is passing through the zenith just as the center-of-mass is passing through the perigee point. The grapple at the end of the active arm is thus at 4024.67+426.67 = 4,451.3 km, moving at 4,236 m/s + 2,133 m/s = 6,370 m/s, the same radius and velocity as that of the payload, ready for the catch. After catching the payload, the MarsWhip tether is now in a balanced configuration. The effective arm length is 400,000 m and the tether tip speed is 2,000 m/s. In the process of catching the incoming payload, the periapsis of the center-of-mass of the tether has shifted upward 26,667 m to 4,051 km and the periapsis velocity has increased to 4,370 m/s, while the apoapsis has risen to 37,920 km, and eccentricity to 0.807. The period is 1.04 sol.

Payload Release and Deorbit

The payload is kept for one orbit, while the phase of the tether rotation is adjusted so that when the tether center-of-mass reaches periapsis, the active tether arm holding the payload is approaching the nadir orientation. If it were kept all the way to nadir, the payload would reach a minimum altitude of about 250 km (3648 km radius) at a velocity with respect to the Martian surface of 4370 m/s - 2000 m/s = 2370 m/s. At 359.5 degrees (almost straight down), this condition is achieved to four significant figures. The payload is then moving at a flight path angle with respect to the local horizon of 0.048 radians and enters the atmosphere at a velocity of 2,442 km/s.

MarsWhip after Deorbit of Payload

After tossing the payload, the MarsWhip tether is back to its original mass. The process of catching the high energy incoming payload, and slowing it down for a gentle reentry into the Martian atmosphere, has given the MarsWhip a significant increase in its energy and momentum. At this point in the analysis, it is important to check that the MarsWhip started out with enough total mass so that it will not be driven into an escape orbit from Mars. The final orbit for the tether is found to have a periapsis radius of 4078 km (676 km altitude so that the tether tip never goes below 253 km altitude), a periapsis velocity of 4,503 m/s, an apoapsis radius of 115,036 km, an eccentricity of 0.931, and a period of 6.65 sol. The tether remains within the gravity influence of Mars and is in its high energy state, ready to pick up a payload launched in a suborbital trajectory out of the Martian atmosphere, and toss it back to Earth.

Elapsed Time

The total elapsed transit time for this particular scenario, from capture of the payload at Earth to release of the payload at Mars, is 157.9 days. This minimal mass PlanetWhip scenario is almost as fast as more massive PlanetWhip tethers since, although the smaller mass tethers cannot use extremely high or low eccentricity orbits without hitting the atmosphere or being thrown to escape, the time spent hanging on the tether during those longer orbit counts as well and the longer unbalanced grapple arm of the lightweight tether lets it grab a payload from a higher energy tether orbit.

SUMMARY OF MULTIPLE MERITT ANALYSES

We carried out analyses of a number of MERITT missions using a wide range of assumptions for the tether tip speed and whether or not aerobraking was used. The trip times for the various scenarios are shown in Table 1. As can be seen from Table 1, the system has significant growth potential. If more massive tethers are used, or stronger materials become available, the tether tip speeds can be increased, cutting the transit time even further. The transit times in Table 1 give the number of days from payload pickup at one planet until payload reentry at the other planet, and include tether "hang time" and coast of the payload between the patch points and the planets. Faster transit times can be made with higher energy initial orbits for the payload and the tether. With a 2.5 km/s tip speed on the PlanetWhip tethers and using aerobraking at Mars the Earth orbit-Mars orbit transit time can be made about 94 days.

TABLE 3. MERITT Interplanetary Transfer Times.

Tether Tip Speed (km/s)	System to Payload Mass Ratio	Transfer Direction From -> To	Tether-to-Tether (days)	Aeroslowing (days)
2.0	15x	Earth -> Mars	155	116
		Mars -> Earth	155	137
2.5	30x	Earth -> Mars	133	94
		Mars -> Earth	142	126

CONCLUSIONS

We have shown that two rapidly spinning tether systems in highly elliptical orbits about Earth and Mars, can be combined into a tether transport architecture that provides rapid interplanetary transport from a suborbital trajectory above the Earth's atmosphere to a suborbital trajectory above the Martian atmosphere and back again.

ACKNOWLEDGMENTS

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