The ElectroDynamic Delivery Experiment (EDDE)

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Abstract. The ElectroDynamic Delivery Experiment (EDDE) is proposed for a space demonstration. EDDE consists of an autonomous space vehicle powered by lightweight solar arrays, a bi-directional electrodynamic tether, and batteries for power leveling. The EDDE vehicle can modify its orbit repeatedly without rocket fuel, and can change all six orbital parameters by modulating and reversing the current flow in the conducting tether. The base spacecraft is connected to the service module by a 6-km-long electrodynamic tether, and is designed for 2 kW of power and a total mass of 180 kg. Tether lifetime of several years is achieved with a two-strand caduceus, with the strands connected every few meters. Tether libration is minimized by mass distribution and by active current control. The vehicle and tether system concepts are developed, the operational envelopes are examined, and potential applications are evaluated. The EDDE vehicle is about twice as fast as ion rockets for high-inclination orbital plane changes, and has much higher maximum delta-V capability. A proof-of-concept experiment is proposed for a low-cost space demonstration. This on-orbit experiment could include additional secondary payloads; for example, EDDE could place low-ΔV, free-flying inspectors into arbitrary orbits from which they could approach selected objects without concern for tether dynamics or interference.

INTRODUCTION

The Air Force has a current need for an autonomous vehicle to deliver constellations of satellites into polar, near polar, or sun-synchronous orbits at altitudes of 400-1000 kilometers, and to inspect, repair, upgrade, or refuel future satellites configured for on-orbit servicing. Collaborating constellations of the proposed Air Force TechSat 21 mini-satellites could be emplaced and serviced by an autonomous orbital service vehicle. Modular spacecraft that can be upgraded by additional plug-in modules could also use an orbital service vehicle. Both of these scenarios highlight the Air Force need for an autonomous space vehicle such as the proposed ElectroDynamic Delivery Experiment (EDDE) vehicle that could emplace, refuel, upgrade, remove, and replace satellites. For these missions, electrodynamic tethers are well suited; they provide higher thrust/power than ion rockets, much higher specific impulse than chemical or solar thermal rockets, and they are particularly effective in plane changes, which are difficult for other propulsion systems.

BACKGROUND

A conducting wire in orbit generates an emf and an induced current that produces a drag force as it moves through the Earth's magnetic field. If power is available and the current can be reversed, the resulting force is a thrust. The force available is highest in low earth orbit, and decreases rapidly with orbital radius. The in-plane force is greatest in equatorial orbit and least in polar orbit, whereas the out-of-plane force is greatest in polar orbit and least in equatorial orbit. Because electrodynamic tethers can produce forces both in the orbit plane and orthogonal to it, they can be used for on-orbit attitude stabilization of satellites. By controlling the current in the tether, all the orbital elements can be changed as desired, individually or in combination (Cosmo and Lorenzini, 1997). The in-plane forces can be used to change the orbital semi-major axis, eccentricity, line of apsides, and phase, whereas the out-of-plane forces can be used to change the inclination and ascending node.

The operation of an electrodynamic tether as both motor and generator was demonstrated by the NASA-Johnson Plasma Motor-Generator flight experiment in 1993. Tether deployment was demonstrated in several flights, such as the SEDS, and a tether retrieval system was designed for the TSS flights on the Space Shuttle. NASA Marshall is developing an electrodynamic tether to demonstrate drag to re-enter a Delta II upper stage (Johnson and Balance, 1998), and the Mir Electrodynamic Tether System (METS) is planned for a flight in 2001. With the success of these programs, the International Space Station may be equipped with electrodynamic tethers for boosting and for power storage. The result of this experience is that the technology is ready for a demonstration of a tether-powered orbital maneuvering vehicle for the Air Force satellite emplacement and servicing mission. A recent contract for the Air Force Research Laboratory by Star Technology and Research, Inc. demonstrated that a flight experiment using a secondary payload bay of the Evolved Expendable Launch Vehicle (EELV) is a feasible flight demonstration (Pearson, et al., 2000). The flight proposed is called the ElectroDynamic Delivery Experiment, or EDDE.

EDDE VEHICLE CONCEPT

The conceptual design of an operational EDDE service vehicle is shown in Figure 1. It would consist of two structures connected by a 6-km-long, meteoroid-resistant, electrically bi-directional tether hanging vertically in LEO. The high-power structure would house the solar array, batteries, and power supply; the low-power end would house the tether, deployer, diagnostic instrumentation, and sub-satellites for dispersal. Each end would contain avionics and telemetry for communications, including a GPS receiver for navigation and a computer for command and control. Each end would also include a hollow cathode electron emitter and an electron collector. The EDDE vehicle could be a testbed for demonstration of thin-film solar arrays and lightweight batteries. As a service vehicle, the low-power end could be equipped with cold-gas thrusters and video sensors for position control, and robotic manipulators to service satellites. This end, with less mass and fewer appendages, is more suitable for rendezvous.

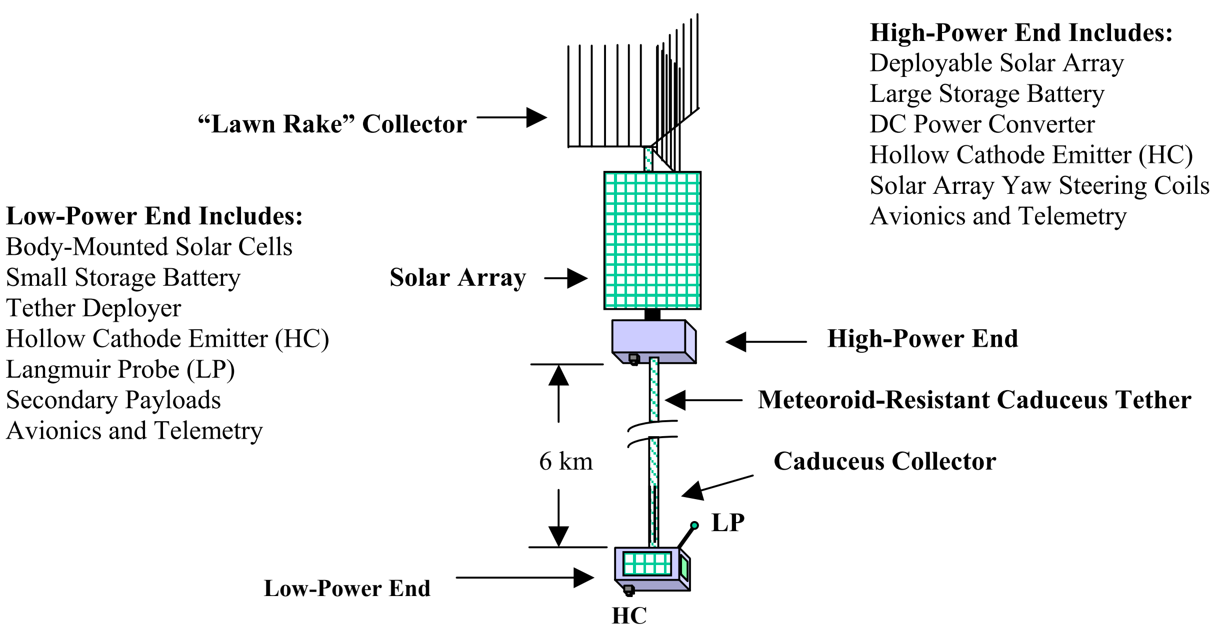

FIGURE 1. ElectroDynamic Delivery Experiment Vehicle Concept.

SYSTEM DESIGN

The EDDE vehicle designed for a demonstration flight on an EELV secondary payload bay must meet stringent mass and volume restraints. The system design consists of the tether, the collector, the power system, and the mechanical system. For pure boost missions, the power system should significantly outweigh the tether system, but for most other missions those system masses should be comparable. Operational systems might have much larger payloads for the same power plus tether system mass, but their orbit change rates would scale inversely with total system mass. A preliminary mass budget by subsystem is given below, for a total mass of 180 kg:

TABLE 1. EDDE Mass Budget.

Power System	**Mass (kg)**	**Tether System**	**Mass (kg)**	**Other Systems**	**Mass (kg)**
Solar array	20	Tether and collector	30	Structure	30
Batteries	30	Reel assembly	20	Avionics and cold gas	10
DC/DC converter	10	Hollow cathodes	10	Payload	20
TOTAL					**180**

Tether Design

There is a minimum length of several kilometers, even for the smallest system, to provide the needed emf across the length of the tether. To maintain efficiency, the tether voltage must be at least several times the sum of the collection and emission voltages. The tether must be multi-stranded, flattened, or a hollow cylinder, in order reduce the risk of failure from micrometeoroids to an acceptably low level. For EDDE, we have selected the multi-strand caduceus. For minimum tether mass, the material selected is aluminum, which has nearly twice the conductivity per unit mass of copper. The tether also must be insulated, and the coating adds appreciably to the tether mass. For bi-directional operation, we must duplicate either the hollow cathodes and the collectors, or the wire. Because the wire is the heaviest, we duplicate the hollow cathodes and collectors, and use only one tether. The hollow cathode is about shoebox size, and includes a gas supply and plumbing to provide improved ionospheric plasma contact. For missions up to at least 6 months, 10 kg for a pair of hollow cathode systems appears reasonable. Existing designs are available from the ProSEDS program and the space station application.

Collector Design

We will use electron collectors at each end, since bi-directional operation is needed for an electrodynamic system to make large orbit plane changes. The collectors are designed for about 20 m^2 per kilowatt of power. That means that multiple separated strands will be required for any reasonable system size. William Thompson, Manuel Martinez-Sanchez, Juan San Martin, and others have suggested the use of very open net-like collectors, with ligaments spaced at distances many times the width of each ligament. If the ligament separation distances are at least many times 10x their width, interference is likely to be low. We propose two multiple-element collectors. The electron collector at the high-power end uses aluminum ribs splayed radially outward from the tether, with long members hanging from them like the tines of a lawn rake. This “lawn rake” collector is very lightweight and efficient, and is ideally suited to the non-retractable tether systems. Retracting such a collector would require some careful design.

For the collector at the low-power end, a caduceus collector forms the last 400 meters of the conducting tether. A two-element caduceus is used, with each element being 30 mm wide, for a total collecting area of 48 m^2. About 8 m^2 of the collector would be coated for emittance, to keep the temperature down. This collector is located on the tether side of the low-power end, leaving that end of the vehicle free to rendezvous with other satellites.

Power systems

EDDE is designed for a peak power of 2 kW, and an average of 1 kW. The system consists of the high-voltage solar array, the four-quadrant DC-to-DC power converter, and the batteries for energy storage and leveling.

Solar Array and Power Converter

For a power output of 2 kW, the appropriate choice for the solar array appears to be conventional gallium arsenide based, multi-junction arrays. Using the data from the Hughes Spectrolab website, arrays of 2 kW have an efficiency of 22%, weigh about 1.9 kg/m^2 and have a power/area ratio of 300 W/m^2. For 2 kW power, this requires an area of 6.7 m^2 and weighs 12.7 kg. Allowing about a 50% increment for deployment and panel construction gives a total of about 20 kg. The potential mass savings of the new thin-film solar arrays are impressive, and could result in a weight reduction of several kilograms. Guha et al. (1999) report a specific power of over 2400 W/kg for lightweight amorphous silicon arrays. If these numbers can be achieved in larger scale production, the array mass would be only a kilogram. Even allowing for extra power conductors and structure, a few kilograms mass would be possible with thin-film arrays.

The DC/DC power converter provides the power to the tether needed for orbital changes, which varies with the orbital position. This requires a variable voltage and current (-500V to +1500V at currents up to 5-6 A) to move the power back and forth between the tether and the low-voltage battery as efficiently as possible. The power system must be sized for the orbital average power available, not just the instantaneous average power. This is a function of the power available during sunlight, the power available from the batteries during darkness, and the power losses from the tether electrical resistance (ohmic losses) and from the less than perfect efficiency of the electron capture and emission (contactor losses). The power system must also be able to vary the tether current by +/- 10% for tether libration control.

Batteries

Batteries are a key component of the baseline system, which is designed to operate in a bi-directional mode, where energy extracted by the tether is stored for later use. The batteries allow operation in the motor mode during eclipses. The fast charge/discharge and total number of cycles is a major design issue. We have a more severe requirement than the usual day-night LEO cycle, due to the roughly two cycles per orbit required for inclination change. Currently, the best available system for LEO is nickel-hydrogen. The nominal 60-Ampere-hour Eagle-Picher (www.spi.tech.com) Single Pressure Vessel battery has about a 60 W-hr/kg specific energy at full discharge, or about 18 W-hr/kg at our assumed 30% depth of discharge. For EDDE, we assume that a total energy storage of 1.67 kW-hr will allow the desired load leveling and regenerative storage of power, giving a battery mass of 30 kg.

Other battery types and other energy storage methods are being evaluated. Lithium ion batteries have higher power/mass ratios, but do not meet the cycle requirements for LEO applications. Flywheel energy storage could meet the high number of fast cycles, but is not nearly as mature and reliable as battery technology. We will watch developments before flight, while keeping nickel-hydrogen batteries as the baseline.

Mechanical systems

A baseline one-shot deployer compatible with the wire and collector concepts will be used for the first flight test (Carroll and Oldson, 1995); after that, knowledge gained from the system performance will guide the design of a deployer that is well suited to the mission of interest. The caduceus tether lends itself to a reel deployer, but there is a question whether a multi-strand tether with broken ligaments can be retrieved without a specially designed reel. The "yoyo-de-spin" deployment maneuver is proposed to deploy the tether. The cold-gas thrusters first spin up the entire, connected structures. When the centrifugal force reaches the required minimum, the two structures are released and extend on the tether. For the main tether reel and mechanism to deploy, a minimum tension of 0.2 N is required, and 0.5 N would provide more margin against an unwinding and possible fouling of the wire on the reel. As the tether unwinds on the reel, the two structures separate far enough to produce sufficient gravity-gradient force to complete the tether deployment. The spin rate decreases as the tether length increases, and the tether electrodynamic force halts the remaining spin so that the system has the proper orientation, with the solar array upward. The system can flip itself and re-stabilize with the solar array downward, using electrodynamic torque to produce large libration angles and the cold gas thrusters to complete the flip.

POTENTIAL APPLICATIONS

The EDDE vehicle is ideally suited to servicing polar platforms with different inclinations (Welch, 1993). It can also disperse many small satellites with mass on the order of a kilogram, like the CubeSats and their ejectors designed by The Aerospace Corporation and Stanford University. CubeSat, a 10-cm cube with a mass about 1 kg, could be used for science observations in LEO orbit through the magnetosphere when solar flares occur. Several CubeSats could be stored in EDDE and released, timed to disperse and to observe solar flare effects, particularly at the poles. EDDE could sweep out a large range of nodal positions and inclination, dispensing nanosats on command. Spherical dragsats could be dispensed into orbits with varying inclinations and altitudes, to get a better dynamic 3-D model of the neutral upper atmosphere, which is now "data poor" by comparison with the plasma. Various GPS based experiments, such as occultations for atmospheric sounding, are also candidates. EDDE could perform these tasks over a much broader range than any single launch vehicle could cover, and could do it twice as fast as ion engines. EDDE could dispense satellites, then rendezvous with them later to test close approach methods. The EDDE flight experiment could also be a testbed for several useful technologies such as thin-film solar arrays and high-voltage power handling.

VEHICLE CAPABILITIES

Tether-powered space vehicles are quite capable for orbital transfers, especially large inclination changes in near polar orbits (Johnson, et al., 1998). Table 2 shows the specific orbit transfer rates and transfer times for the EDDE vehicle from ISS to Kosmos 2150, using data from Lorenzini et al., 1997. To transfer from 51.6° to 74° inclination takes 85 days at 400 km altitude, but more than twice as long, 181 days, at 800 km altitude. The transfer from 400 to 800 km altitude is much more rapid, but it takes nearly twice as long at 74° inclination as it does at 51.6°. Nodal changes can be achieved during inclination changes, to minimize total orbit transfer times.

TABLE 2. Specific Orbit Transfer Rates and Transfer Times for the 90 kg/kW EDDE Vehicle.

Inclination	51.6°			74°	
Altitude	kg deg / kW day	kg km / kW day	Transfer Time	kg deg / kW day	kg km / kW day
400 km	18.5	↑ 11,400	← 85 days →	29.0	↑ 7,000
800 km	9.0	4.1 days ↓ 6,400	← 181 days →	13.2	7.2 days ↓ 3,000

The results also show that there is a preferred order to make the transfer. Changing the inclination first, then the altitude, takes 92 days, whereas changing altitude first takes 185 days. The nodal plane must also be matched during the transfer. Since the space station nodal plane regresses 3.17 degrees per day faster than the Kosmos 2150, there will be a 179-degree change during the trip. The maneuvering vehicle will also have its nodal plane change during the transfer, a total of about 242 degrees. Since this is greater than the Kosmos 2150, the vehicle could modify the order of changing the inclination and altitude, taking a little more time to ensure matching the nodal plane. Reaching an entire constellation of satellites for servicing, the “traveling salesman” trip problem, can be quite a complicated function of the specific orbits, and will probably be unique for each constellation.

The only real competitor to EDDE for high delta-V orbit changes is the ion engine. A preliminary comparison against current ion engine technology showed that an effective exhaust velocity of about 30 km/s for xenon is reasonable. Replacing the 60 kg of tether specific mass with 45 kg of xenon plus 15 kg supercritical tankage gives a simple comparison, although the battery mass might decrease for the ion engine. For these assumptions, the ion system will have an average thrust (for a 50% duty cycle) of 0.045 N, an average delta-V of 23 m/s/day (8100 m/s total) and an inclination change of 0.17 degrees/day. This compares to the EDDE plane change of about 0.3 degree/day, with a much higher total delta-V capability.

Control of Libration Instabilities

The tether current also excites in plane and out of plane librations of the tether (Beletsky and Levin, 1990). Without feedback control of libration, these libration modes can go unstable (Levin, 1987), as shown in Figure 2, which shows a high-speed orbit transfer starting at the ISS. The left graphs show the rapid build up of tether libration angles (in- and out-of-plane) leading to a slack tether from excess in-plane libration. Active current control will be included in the EDDE design to suppress tether libration modes and prevent these instabilities. The rotary "skip-rope" mode is more difficult to observe and control, but this mode is also being addressed.

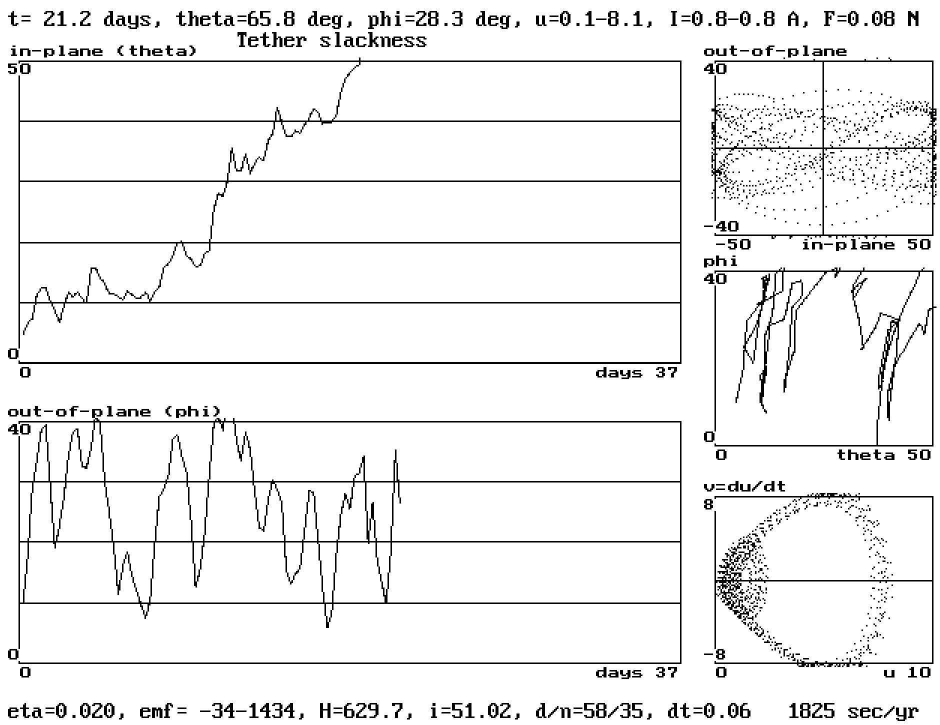

FIGURE 2. Example of Tether Libration Instability.

PROTOTYPE FLIGHT EXPERIMENT

The U.S. Air Force is currently investigating a standard Evolved Secondary Payload Adaptor (ESPA), to simplify the design of secondary payloads for the next generation of Evolved Expendable Launch Vehicles. The payloads would mount on the adaptor ring below the separation plane of the primary spacecraft, as shown in Figure 3. Six symmetrically spaced attachment points on the adaptor ring are available, with the mass margin of a specific mission often constraining the number and total mass of secondary payloads. The maximum allowable mass is currently 400 lbs per attach slot; the maximum volume is 24" x 24" x 38", with the long dimension perpendicular to the mounting ring. The cantilevered launch loads are not the optimum loading conditions, but they can be accommodated.

The proposed mass of EDDE can be accommodated in one secondary payload slot on the EELV. However, the resulting density of 500 kg/m^3 is higher than the typical 100 kg/m^3 of satellites. Our nominal one-slot design for EDDE includes a nominal payload mass of 20 kg of low-volume payloads. For EDDE to handle heavier or bulkier payloads, EDDE can be mounted in two adjacent slots on the host vehicle. Then, depending on the host's payload margin, EDDE's payload capacity might grow to 120 kg. EDDE's orbit change rate would be cut by half if its total mass doubles, but the payload mass would go up by a factor of 6, and available payload volume might increase by an even larger ratio. Adjacent payload slots could be used for the two end structures, so they could be connected during launch. On reaching orbit, the payloads could be released radially at the same time, but at different velocities, so they would start deploying from each other and also start spinning around each other. The cold gas thrusters could then be used to speed up the spin to complete the deployment using the yo-yo de-spin maneuver.

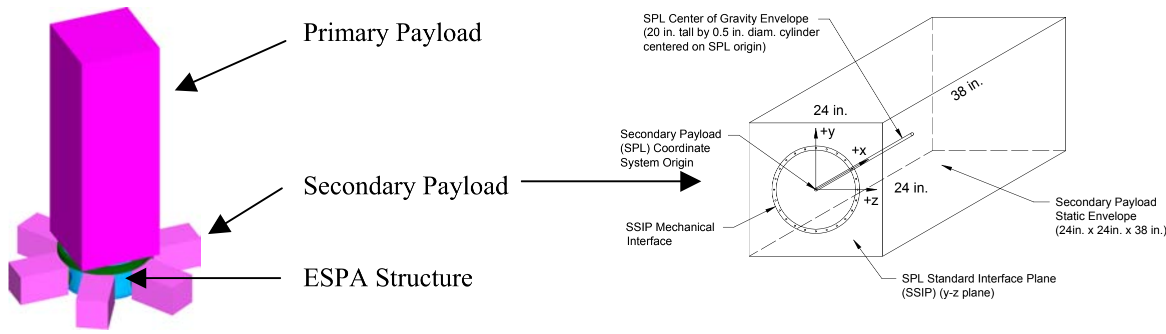

FIGURE 3. Secondary Payload Bay Configuration of Evolved Expendable Launch Vehicles.

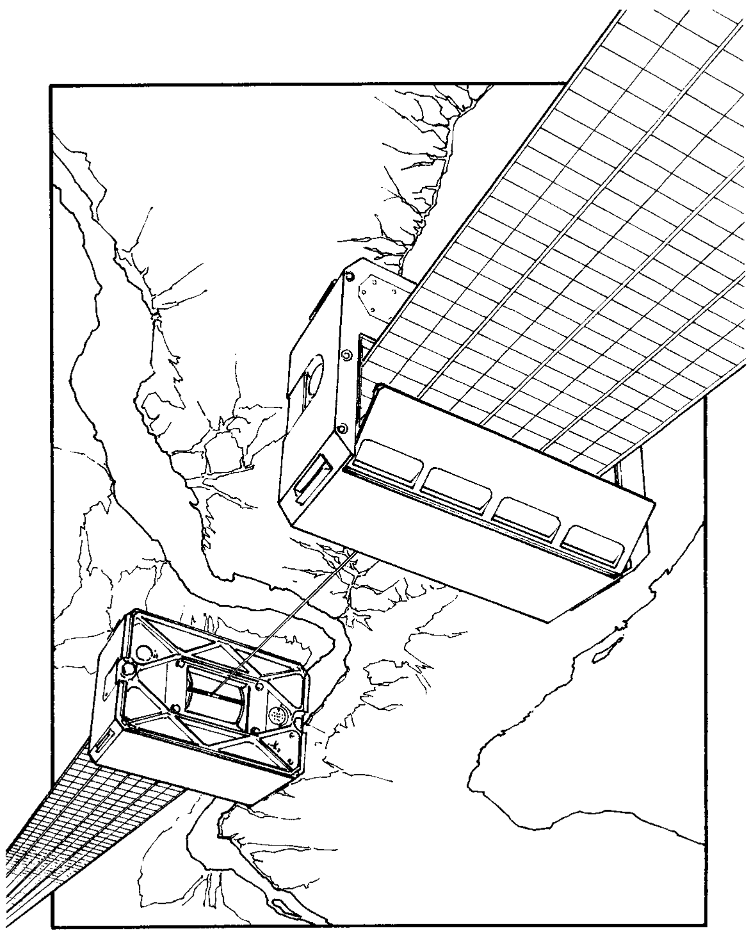

FIGURE 4. Artist's Concept of Prototype EDDE Flight Experiment.

An artist's concept of the EDDE flight experiment is shown in Figure 4. This concept was based on an earlier design that had identical end bodies, each with its own solar array. The current design clears away the appendages from the low-power end, allowing it greater freedom for rendezvous and docking with other spacecraft.

CONCLUSIONS

The concept of the tether-propelled ElectroDynamic Delivery Experiment (EDDE) is feasible. Operational envelopes were developed for potential Air Force missions to emplace and service constellations of satellites. Results show that electrodynamic tethers are highly efficient in low earth orbit, and best for altitudes near 400 km. Plane changes are easier for inclinations above 50^{o}, and altitude changes are easier for low-inclination orbits. Tether current must be modulated to suppress tether libration modes for highest performance, and we have a method for accomplishing this. Operational envelopes were developed, revealing nonlinearities in the rates of orbital transfer as functions of altitude and inclination. These nonlinearities mean that outbound and return trip times can differ significantly. An EDDE prototype vehicle concept was developed that can be used with a common spacecraft bus structure with adequate solar power. The EDDE vehicle with an electrodynamic tether is about twice as fast as an ion engine, and is also lighter than the ion engine for missions with total ΔV exceeding 7 km/s. This makes it potentially very useful in emplacing multiple nanosats in multiple inclinations or planes.

ACKNOWLEDGMENT

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