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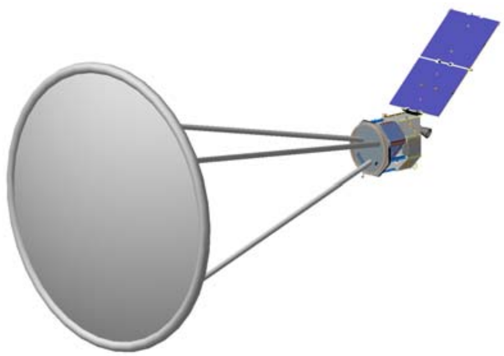
Rapid De-Orbit of LEO Space Vehicles Using Towed Rigidizable Inflatable Structure (TRIS) Technology: Concept and Feasibility Assessment



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Rapid De-orbit of LEO Spacecraft Using Towed Rigidizable Inflatable Structure (TRIS) Technology

1.0 INTRODUCTION

The amount of debris in Earth orbit is increasing at an extraordinary rate and presents a growing hazard to orbital operations. Most of the debris in low earth orbit (LEO) is man made and consists of inactive spacecraft and/or launch vehicle upper stages. International treaties and US Government requirements dictate that all space vehicles (SV's) must be de-orbited or moved to a higher orbit within a limited time (within 25 years of EOL – NASA/FCC Requirement). A separate guideline requires that medium to large spacecraft be disposed of via a controlled de-orbit into a remote portion of the Pacific Ocean. The cost and mass of a spacecraft can increase significantly to meet these post-mission disposal requirements. For example, approximately 75% of the propellant on Ball Aerospace and Technologies (BATC) NPP spacecraft is used to perform a controlled de-orbit at the end of the mission. For a small spacecraft, a propulsion system may not be feasible because of mass, volume, and cost constraints. Without a de-orbit propulsion system, it can take years for a spacecraft to re-enter the Earth's atmosphere. The rate of decay of this orbital debris/SV is dependent upon the orbit altitude, ballistic coefficient of the SV, activity of the sun (i.e. solar cycle) and variations in density of the upper atmosphere which makes de-orbit predictions difficult.

This paper describes a low cost and mass de-orbit system which uses inflatable technology that can be rigidized to increase the drag of a Low Earth Orbit (LEO) space vehicles. Other applications include aerobraking maneuvers or orbital adjustments. Some of the key technologies of the de-orbit system have been previously demonstrated on-orbit during other unrelated NASA missions. In this paper we present various technologies and concepts including:

- A brief background of the technology including Ball's expertise
- A conceptual system design for the towed rigidizable inflatable de-orbit system
- Critical issues associate with implementation
- General requirements for the system deployment and operation (Concept of Operations)
- The system requirements – size, mass, cost, technology maturity, and materials
- The system performance
- Secondary effects associated with a de-orbit system deployment

Our proposed approach uses the knowledge gained on our previous inflatable structure development programs[1], integrates the technologies into a new de-orbit application, provides a concept of operation, and adapts this technology for use on multiple space vehicles.

2.0 TECHNICAL

2.1 NASA Debris Guidelines

NASA Safety Standard 1740, “Guidelines and Assessment Procedures for Limiting Orbital Debris”, is the governing document for orbital debris generation and post-mission disposal. There are two relevant guidelines related to post-mission spacecraft disposal.

Guideline 6-1 of NSS 1740 states:

“A spacecraft or upper stage with perigee altitude below 2000 km in its final mission orbit will be disposed of by one of three methods:

- Atmospheric reentry option: Leave the structure in an orbit which, using conservative projections for solar activity, atmospheric drag will limit the lifetime to no longer that 25 years after completion of mission.
- Maneuvering to a storage orbit between LEO and GEO: maneuver to an orbit with perigee altitude above 2500 km and apogee altitude below 35,288 km (500 km below GEO altitude).
- Direct retrieval: Retrieve the structure and remove if from orbit within 10 years after completion of the mission.”

Of these, atmospheric reentry is usually the only practical option.

Guideline 7-1 of NSS 1740 states:

“Limit the risk of human casualty: If a space structure is to be disposed of by uncontrolled reentry into the Earth’s atmosphere, the total debris casualty area for components and structural fragments surviving reentry will not exceed 8 m^2. The total debris casualty area is a function of the number and size of components surviving reentry and of the average size of a standing individual.”

Small spacecraft will satisfy this guideline without controlled reentry. Based on performing analyses of the breakup and reentry of several spacecraft, this guideline will be satisfied if the spacecraft mass is less than 1000 to 1500 kg, and if the spacecraft doesn’t contain much high melting point material such as titanium, ceramics, and beryllium.

In light of the NASA guidelines, there are two applications of the device described in this paper:

- Small spacecraft (<~1500 kg) at altitudes of between 600 and 1000 km. This device would allow a significant reduction in the amount of propellant carried, or allow the propulsion system to be deleted from the design.
- Spacecraft that are sufficiently large, which require controlled reentry. This device could be used in place of a large propulsion system to perform the controlled reentry.

2.2 Background

BATC is known as one of the industry leaders for inflatable structures system design and development based upon multiple NASA contracts to design and develop these technologies. Extensive analytical work and testing has been performed in conjunction with these efforts. BATC, in conjunction with NASA has performed multiple simulations and created analysis tools for aerobraking applications on interplanetary missions.

BATC is working with NASA, L’Garde, Inc., and ILC Dover, Inc. to develop inflatable structures for space applications. An example of a conceptual aerobraking design is shown in **Figure 2.2-1**. This design uses a trailing toroidal ballute that is supported by three tensile tethers.

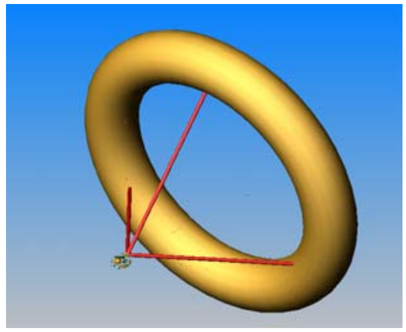

Figure 2.2-1 Aerobreaking Toroid[1]

L'Garde also has extensive experience with inflatable experiments. **Figure 2.2-2** shows a large (14 meter) inflatable parabolic dish antenna that was tested on the STS-77 mission. The mission demonstrated that large inflatable structures could be deployed and utilized in the space environment. An overlooked side effect of the parabolic dish was the rapid de-orbit of the experimental spacecraft due to the increased drag. **Figure 2.2-3** is a prototype 2.2-meter decoy balloon, which was demonstrated by L'Garde in the zero-G facility at the NASA Glenn Research Center.

Figure 2.2-2. L'Garde Inflatable Antenna

Figure 2.2-3. Inflatable Decoy

Inflatable structures made of commercially available films are lightweight, tightly packed, and occupy a very small volume when stowed. Thus they can be launched along with a SV, without affecting normal operations. In general, they are about ten times less expensive to produce than mechanical deployable structures, are more reliable, and can be accommodated in smaller spacecraft and launch vehicles. The structural films lend themselves easily to metallization, lamination, painting, and coating, if desired. Inflation of the deployable structure is performed using stored gas or with a gas generator (similar to an automotive airbag device).

Depending on the application, inflatable structures are constructed of purely inflatable/stretched membrane components, or inflatable-rigidizable components, or both. Rigidizable systems are comprised of metallized polymers or structural fabrics (e.g. Kevlar®, graphite cloth, etc.) impregnated with or coated by a resin. Upon inflation in space, they can be rigidized using the following means: 1) using spacecraft power for heat generation – thermoset resins, 2) initiating a pressurization pulse to harden the metallized polymer, or 3) through passive means using the space environment, (i.e. temperature, UV radiation, particle radiation, vacuum). Rigidized inflatable structures will maintain their shape even if internal pressure is lost. Thus the structures are inherently robust and insensitive to micrometeorite impacts.

2.3 Demonstration System CONOPS

The important technologies associated with a Towed Rigidizable Inflatable Structure (TRIS) de-orbit system can be validated during an on-orbit demonstration. It is expected that the De-orbit system will most likely be a secondary payload on an advanced technology demonstrator mission.

Figure 2.3-1 summarizes the mission phases of our flight demonstration. The mission begins with launch operations. After successful launch, the demonstration satellite separates from the upper stage and stabilizes to the flight orientation. After completion of the primary mission, the spacecraft or a ground command activates the TRIS system. The canister cover opens and the device is deployed with a current pulse (~30 Watts, 1 Amp) by either a preprogrammed command or an up-link signal. The inflatable device is then pressurized and rigidized using compressed gas or a gas generator. The SV attitude control system of the spacecraft can counteract the secondary effects associated with the device deployment. The NASA/L'Garde demonstration mission (STS-77) showed that perturbations do occur during deployment, but induced oscillations damp out quickly after complete deployment. Temperature and pressure sensors will also be used to monitor the deployment and on-board cameras could be used to monitor the deployment and operation of the de-orbit system. All telemetry is downlinked during this phase of the mission. Additional space-based and ground-based assets view the de-orbit system operation, if desired.

Onboard systems and ground-based sensors closely monitor the rate of orbital decay. These results are evaluated and compared with the predicted results. At a pre-determined point during the orbital decay, the rigidized structure could be jettisoned, enabling the spacecraft to perform a controlled re-entry using the remaining Reaction Control System (RCS) propellant (if equipped) or on a predetermined trajectory using BATC's proprietary re-entry analysis tools. Otherwise, the spacecraft enters the atmosphere via normal atmospheric decay. Due to its low weight and large cross-sectional area, the jettisoned de-orbit device quickly combusts in the Earth's atmosphere. Multiple simulations determine the exact point of device separation for controlled re-entry of the SV. Preliminary simulations indicate that the impact location can be controlled to occur at any location around the orbit by varying the TRIS release altitude. Releasing the TRIS at high altitudes produces longer flight times and distances after release, while releasing at lower altitudes produces shorter flight times and distances. Simulations show that by varying the release altitude between 120 and 150 km, the impact time can be varied by more than one orbit. This implies that the impact zone can be selected to be at any location along the groundtrack of the last orbit.

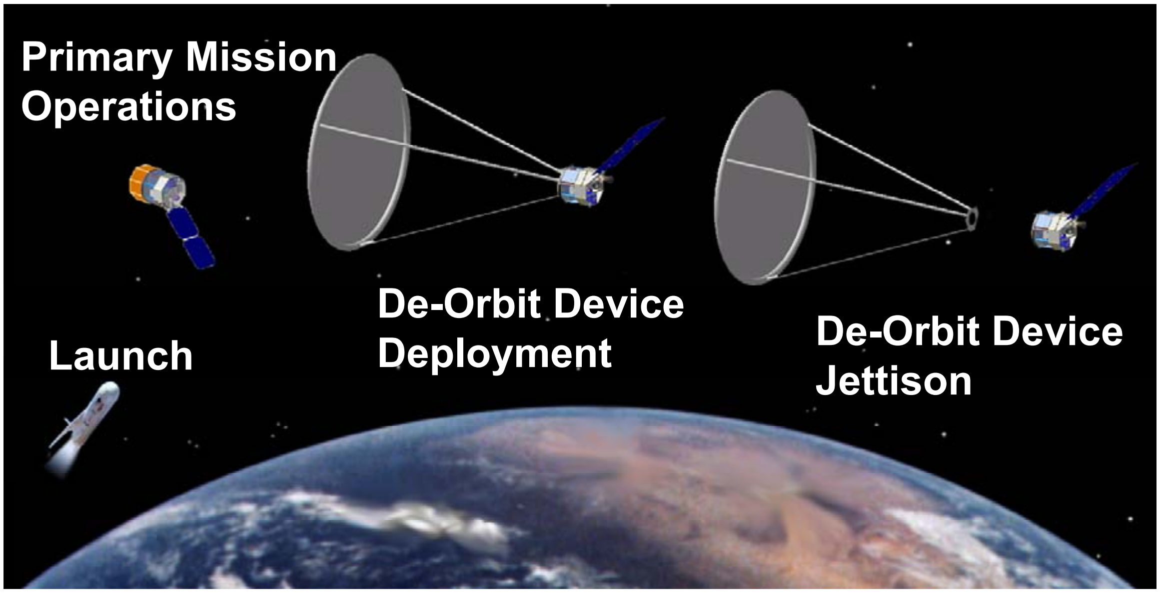

Figure 2.3-1 Mission Profile

2.4 De-Orbit Payload
Technical Discussion

The de-orbit payload concept deployed configuration is shown in **Figure 2.4-1**. In this Figure, the TRIS system is attached to a representative BATC RS-300 spacecraft. The payload is comprised of a central body with rigidized structural ring (to increase the cross sectional area) and multiple struts/columns to offset the body from the spacecraft (to improve the aerodynamic stability). The column/struts provide both tension and compression capability for the central body. The center of the ring structure contains a stretched nonstructural membrane. In a stowed condition, the TRIS system (de-orbit device, gas generator, separation system, and support electronics) is packaged into relatively small self-contained structure. Preliminary mass, volume, and cost data are shown in **Table 2.4-T1**. Costs are identified in the Table to emphasize that the TRIS system is a low cost solution for missions needing a de-orbit capability. For comparison, a typical medium sized BATC spacecraft propellant based de-orbit system can range from $500 K to $1.5 M and have a mass exceeding 435 kg *(de-orbit propellant - 240 kg, hardware - 110 kg).*

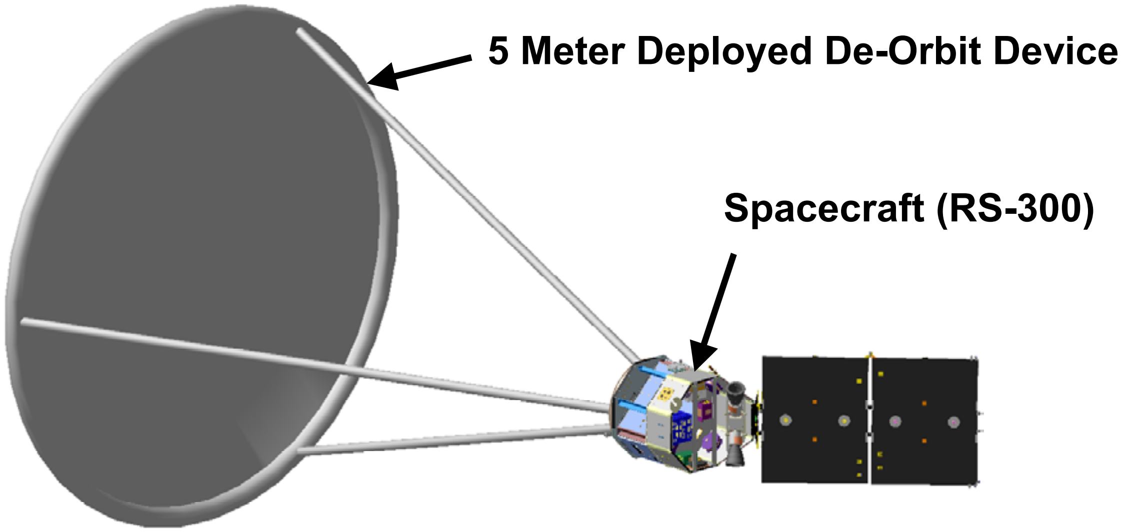

Figure 2.4-1 Payload Configuration (5 m Deployed)

	Mass (kg)	Stowed Volume (m3)	Average Box Side (m)	Estimated Cost ($ K)
5 Meter	5.33	.0026	.139	72
10 Meter	13.39	.011	.220	80
15 Meter	23.99	.024	.288	105

Table 2.4-T1 Mass, Volume, Power, and Cost Estimates

2.4.1 Technology Risks and Issues

Some of the issues and risks associated with the TRIS system include:

1) Extended Storage Requirements –Materials tend to degrade after years in orbit due to the extreme space environment (thermal cycling, vacuum, atomic oxygen, etc.). On-orbit lifetime issues of the materials and the TRIS system must be addresses.
2) Stability of the SV During Deployment – Deployment induced forces affect the spacecraft stability and orientation. Non-uniform inflation of the TRIS device drives the majority of the induced disturbances. Packaging of the TRIS device is also critical since there is a spring back (or out) of the material when the canister door is opened during deployment. The TRIS device design ensures that induced forces are minimized or damped out passively after deployment.
3) SV Attitude Control - Attitude maintenance of the SV after TRIS deployment. The attitude control system must be sized to accommodate larger solar radiation, gravity gradient, and drag torques when the TRIS is deployed.
4) Shape and Size of the TRIS Device – Aerodynamic stability is essential to maintaining proper orientation in a low drag environment. The potential exists for the wake gener-

ated by the TRIS device to affect its stability. Hypersonic shock interactions and the associated instability will be major issues affecting the performance of the TRIS system. Additional analyses will ensure that the TRIS device is aerodynamically stable in a near vacuum environment.

5) Materials Properties and Manufacturability of the Device – Material strength, thermal properties, induced packaging stresses, bonded seams and material folds in the structure will be tested and evaluated further to find the optimum solution.
6) Mission Lifetime of the SV – In some cases, the SV must be capable of operating for extended periods during disposal phase. This occurs when the SV uses the reaction control system for a precision re-entry profile. Therefore, the SV lifetime can become a critical trade parameter or system driver. For a passive (not dependant upon the SV for operation or reentry) TRIS system, the re-entry system has to be sized and controlled so as to not add significant lifetime to the core SC subsystems.

2.5 De-Orbit Payload Predicted Performance

To remove a space vehicle (SV) from a circular Low Earth Orbit (LEO) requires a significant amount of energy (Delta-V) which is usually delivered via a propulsion system. With a TRIS de-orbit device, drag is the principal mechanism to remove energy to lower the SV orbit. The time to de-orbit is a function of the solar cycle activity, and the ballistic coefficient (B*) of the SV. Using a TRIS de-orbit device allows SV designers to increase the ballistic coefficient, by orders of magnitude. As shown in **Figure 2.5-2**, most SV will meet the de-orbit requirement if their initial altitude is less than ~650 km. The TRIS de-orbit device is most useful for SV that are above ~650 km. For example, a typical SV with B* = 150 kg/m2 will have a decay time of approximately 75 years at 700 km. An appropriately sized TRIS de-orbit device will change the ballistic coefficient to 50 kg/m2 and hence the change the decay time to less than 25 years.

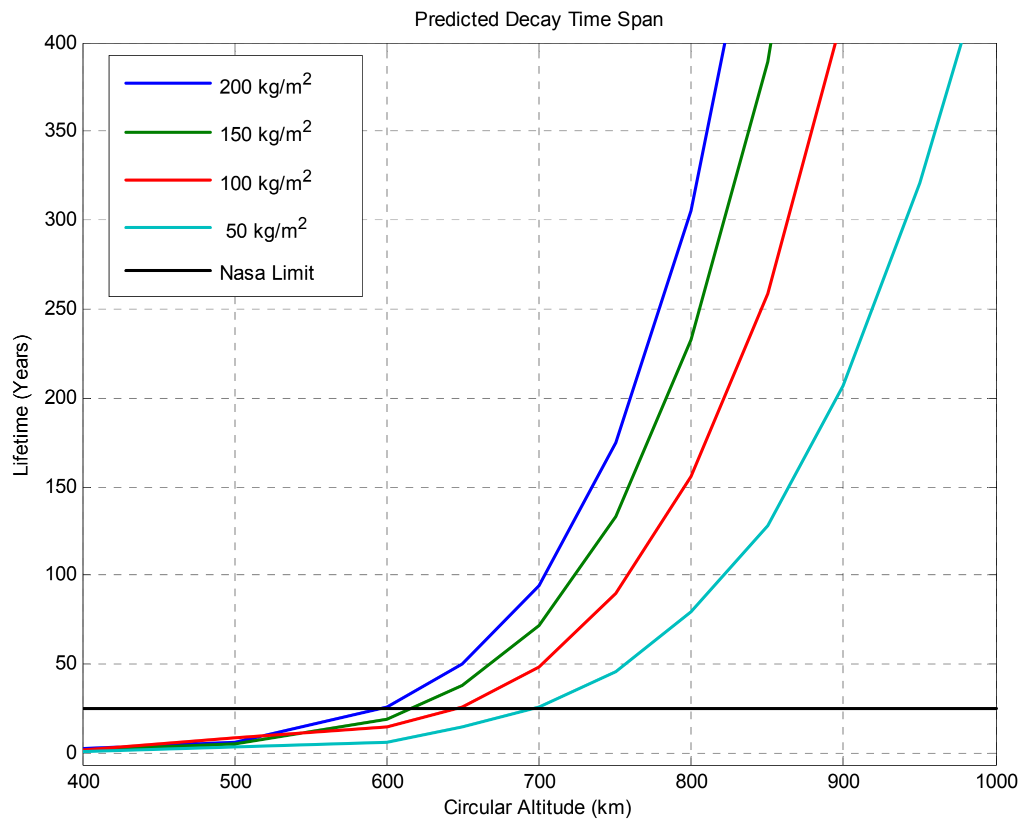

Figure 2.5-1 Orbital Decay vs. Altitude with respect to B* (kg/m2)[a]

De-orbit performance estimates have been evaluated for three de-orbit payload diameters (5, 10, and 15 m) and for various operational altitudes (500 km, 700 km, 1000 km) with circular orbits (initial condition). For these estimates, **Figures 2.5-2, -3, -4** show the relative performance of the deployed de-orbit system of a representative SV (without a propulsion system). The predicted time required for de-orbit is shown on the bottom legend of each graph. Performance estimates were calculated using STK and the nominal Schatten atmosphere model[b].

[a] Calculated using STK with Schatten atmospheric density model. Other assumption include Cd = 2.0, Inclination = 45 degrees, circular orbit.

[b] Atmosphere models are inherently difficult to asses due to short and long term cycles in the true atmospheric density and simplified atmospheric dynamics, all of which when integrated over time can erode significant accuracy from the model. Long-term predictions are shown for trending purposes and guidelines only, not for accurate numerical predictions.

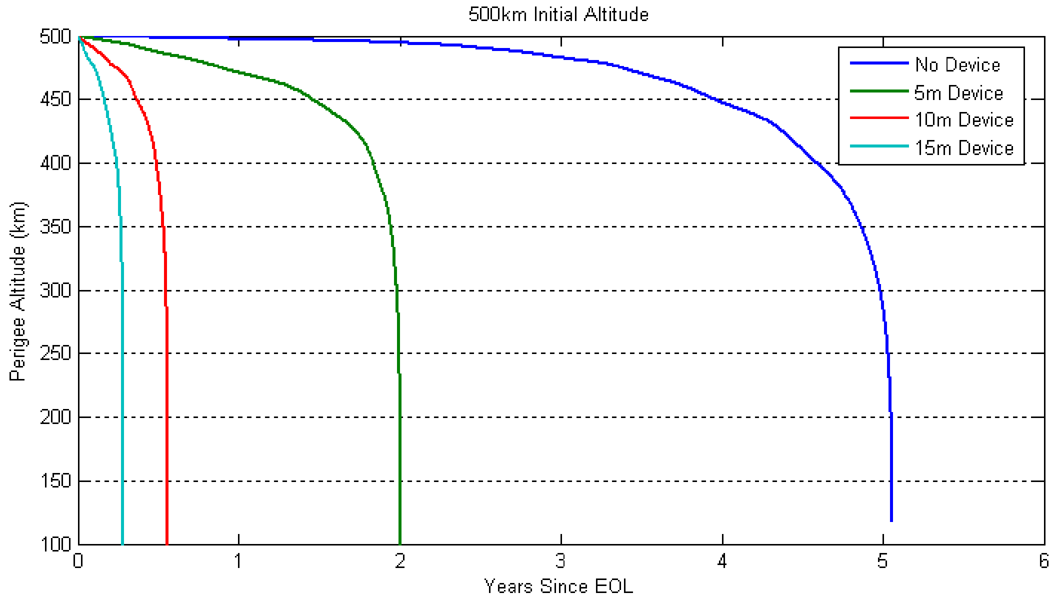

Figure 2.5-2 Orbital Decay of SV w/wo De-Orbit Payload (500 km circular altitude)

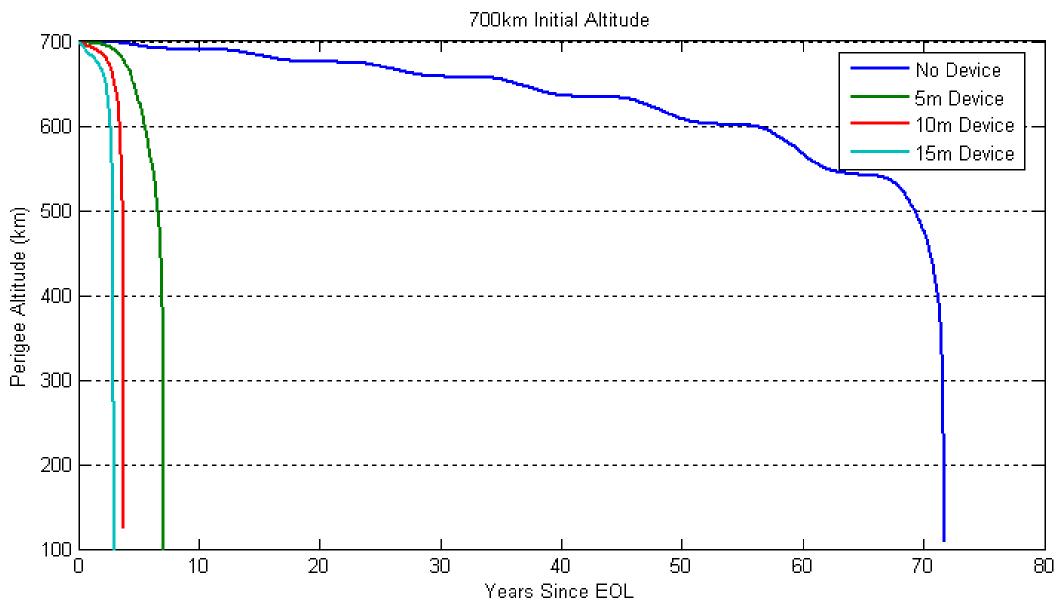

Figure 2.5-3 Orbital Decay of SV w/wo De-Orbit Payload (700 km circular altitude)

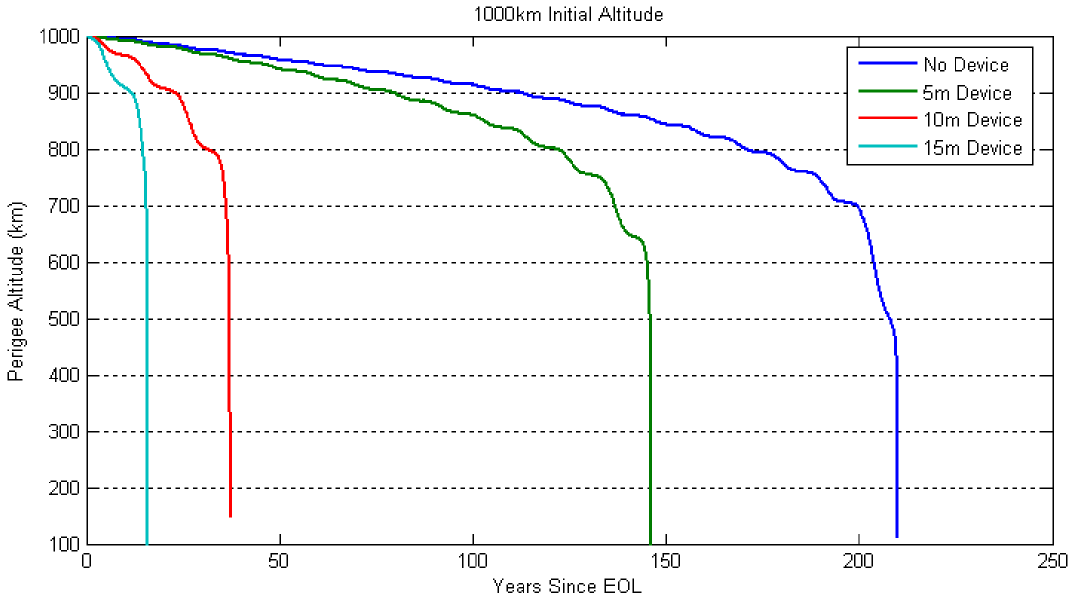

Figure 2.5-4 Orbital Decay of SV w/wo De-Orbit Payload (1000 km circular altitude)

It is important to compare the performance of the de-orbit system with respect to a typical propulsion system. Our method of comparison is to use a propulsion system to lower the perigee of the SV such that it reenters in the same time (~16 years) as the TRIS equipped SV. The required Delta-V and propellant mass are used as performance metrics. For the purposes of this comparison, a simplified analysis was performed (single propellant burn). Propellant system mass, power usage, and additional fuel margins were not included as part of this analysis. Normally, SV's with a de-orbit propulsion system perform multiple burns to reach the desired re-entry location and carry significant propellant margin. The magnitude of the Delta-V can be calculated using the following equations:

$$\Delta V = \sqrt{\frac{\mu}{r_{circ}}} - \sqrt{\mu\left(\frac{2}{r_{a,xfer}} - \frac{1}{sma_{xfer}}\right)}$$

$$sma_{xfer} = \frac{1}{2}\left(r_{p,xfer} + r_{circ}\right)$$

where μ is the gravitational constant (3.986004418×10^5 km^3/sec^2), $r_{a,xfer}$ is the radius of apogee of the elliptical transfer orbit, $r_{p,xfer}$ is the radius of perigee of the transfer orbit, sma_{xfer} is the semi-major axis of the transfer orbit, and r_{circ} is the radius of the circular orbit.

For a ~300 kg[c] SV with a cross sectional area of 2 m^2 at 700 km circular altitude, the perigee altitude must be lowered to approximately 315 km to match the re-entry time of a 15m De-orbit device. Note: For this class of SV, the propulsion system dry mass is approximately 15 kg and

[c] Note: For this class of SV, the propulsion system dry mass is approximately 15 kg and contain 31 kg of fuel.

contain 31 kg of fuel. The Delta-V requirement for such a maneuver is approximately 106 m/s. The propellant load for the SV is derived from the rocket equation

$$m_p = m_d \left(e^{\left(\frac{\Delta V}{g I_{sp}}\right)} - 1 \right)$$

where m_p is the mass of the propellant load, I_{sp} is the specific impulse of the propulsion system, g is the gravitational acceleration at sea level, and m_d is the dry mass of the SV (including propulsion system hardware such as thrusters, valves, and plumbing). For the scenario presented above, the propellant load is slightly larger than 15 kg if Hydrazine is used. **Table 2.5-T1** shows the propellant required for the notional satellite to de-orbit in the same time as a SV with a 15m diameter De-orbit device. No attempt was made to characterize the propulsion system parameters (i.e. mass, power requirements, thrust, envelope, cost). For this table, the assumed values include: coefficient of drag = 2.0, simulated atmosphere based on the nominal Schatten model, and epoch time of June 1, 2005.

Based upon the results shown in **Table 2.5-T1**, one can conclude that the TRIS system mass is very close to the mass of the propellant used. However, this conclusion would be in error since many factors were omitted from the calculation (mass, power usage, envelope, fuel margin, etc.). It may be of value to compare the results obtained using the above method with a known SV with a de-orbit capability. For this comparison the 1200 kg SV operates at ~800 km and has a 435 kg propulsion system (110 kg system hardware, 325 kg propellant (85 kg mission, 240kg de-orbit)). Using the same parameters (desired elliptical orbit, 422 km Radius of Perigee) the SV will require 107 m/s Delta-V (61 kg propellant) to meet the 16-year de-orbit time. From this relative comparison it is clear that the TRIS system has a mass advantage. However, additional analyses / trade studies are required to determine the best option for any particular mission.

Table 2.5-T1.
Propellant load for a single transfer maneuver to elliptical trajectory such that atmospheric drag de-orbits the SV in the same time (~16 years) as a 15m diameter De-orbit Device.*

Initial Altitude (km)	Radius of Perigee (km)	Delta-V (m/s)	Propellant Load (kg)
500	225	78.04	11.06
700	315	105.64	15.07
1000	422	151.24	21.80

* Note: The calculated propellant load does not include any margin. Assumes 300 kg SV, Cross Sectional Area = 2 m^2, Mass of 15 meter TRIS system = 23.99 kg

An accelerated method of de-orbiting a SV combines a propellant burn with the TRIS system in an aerobraking maneuver. The onboard station keeping propulsion system is used to lower the perigee (using the propellant load shown in Table 2.4-T1), placing the SV in an elliptical orbit. The TRIS device is inflated and used to increase the SV cross sectional area. The time to de-orbit is reduced as shown in **Table 2.5-T2**. **Figure 2.5-5** graphically shows the orbital decay time for a SV with and without a TRIS device from an initial circular altitude of 1000 km.

Table 2.5-T2. Time to De-orbit With and Without (Natural Decay) Device

Initial Altitude (km)	Time to De-orbit w/o Device	Time to De-orbit w/Device
500	103 days	1 day
700	3.0 years	21 days
1000	16.1 years	1.0 years

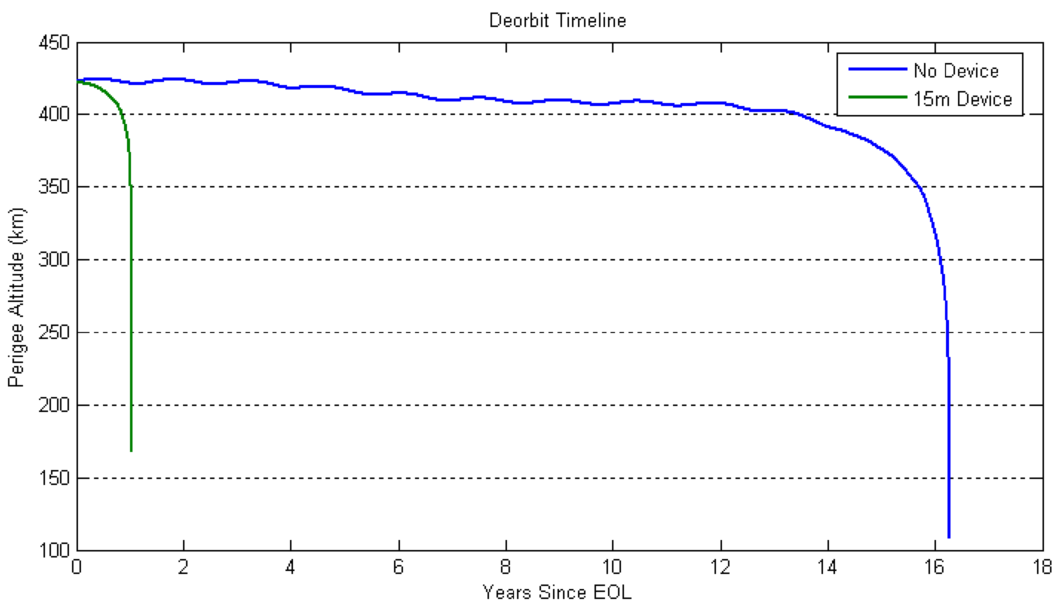

Figure 2.5-5 Orbital Decay of SV Using a Aerobraking Maneuver (15 m De-orbit System)

As the mass of the SV increases the benefits of the system become more noticeable since additional propellant is required to de-orbit the SV. In general, it appears that TRIS-equipped vehicles get the most benefit when the operational altitudes are above ~650 km, the SV mass is greater that 300 kg and with the larger diameter devices. Since each mission is unique, analyses and trade studies need to be performed to design the optimum de-orbit system.

3.0 CONCLUSION

Orbital debris will continue to be a potential threat to future missions. Up to this point, very little has been done to reduce the number of objects in Earth orbit. The TRIS de-orbit system provide a simple cost effective means to reduce the amount of debris generated on new missions. Cost and mass issues are one of the biggest drivers for implementation of this technology. Based on the performance simulations shown in the preceding sections, the TRIS De-orbit System will achieve the primary objectives:

- Low Cost ($72 to $105 K vs. $500 K to $1.5 M w/ propulsion system)
- Minimal impact to the primary payload
- Low Device Mass (5 to 24 kg vs. +46 kg (for a 300 kg SV) w/ propulsion system including propellant required for de-orbit)
- Minimal stowed volume
- Rapid de-orbit of a LEO SV w/o a propulsion system (25 year de-orbit requirement achieved for select mission parameters)
- Enhanced operational performance of the de-orbit device using the SV propulsion system
- Design builds upon previous designs and proven technologies (lower risk)
- Rigidized TRIS device maintains shape after inflation without maintaining pressure
- Capability of precision reentry profile using the TRIS device
- Optimum mission area where a TRIS system has advantages over a traditional propellant based de-orbit system (over ~650 km)

Preliminary analysis indicates that there is a straightforward risk mitigation path leading to successful implementation of the TRIS system. Our approach uses the knowledge gained on previous inflatable structure development programs, provides a concept of operation, and adapts this technology as a de-orbit device on space vehicles. BATC in conjunction with our teammates can produce an effective and inexpensive system to rapidly de-orbit a space vehicle from altitudes that would normally require a propulsion system.

References

[1] Kevin Miller, Doug Gulick, Jake Lewis, Bill Trochmen, - Ball Aerospace and Technologies Corp., Jim Stein - ILC Dover, Inc., Daniel T. Lions - JPL, Richard G. Wilmoth – NASA Langley Research Center, "Trailing Ballute Aerocapture: Concept and Feasibility Assessment", AIAA 2003-4655