Orbital Debris Mitigation Costs & Implications

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1. Introduction:

Due to concerns regarding the accumulation of potentially dangerous debris in Low Earth Orbit (LEO), the United States Government has implemented procedural requirements to minimize the creation of debris by government and non-government operations in space. Several other spacefaring nations have done the same. At present, slightly different requirements have been imposed by various government agencies, and waivers or exemptions granted to allow the continued use of legacy systems. Abroad, the nations of the European Union have adopted a Code of Conduct for space operations that includes debris-mitigation measures. It is expected that in the near future these requirements will converge towards a single, broadly applicable standard, possibly with the force of law or treaty.

At present, the only requirements explicitly levied on commercial space launch operations are to refrain from intentional generation of debris and from any collisions between components of the launch system, and to passivate the launch vehicle by depleting propellants, pressurant gasses, and stored energy in batteries. Launch licenses have been revoked in the past for failure to meet these requirements, and compliance is now high. There is no statutory or regulatory requirement for the safe or timely disposal of launch vehicle hardware left in Earth orbit.

However, it is likely that commercial launch operations in the future will be expected to comply with such a requirement. Even if no statutory or explicit regulatory changes impose such a standard, government customers are likely to have internal requirements that will flow down to commercial launch providers. As it would be exceedingly difficult for a US launch provider to make a successful business case without government customers, there will be increasing pressure on all launch providers to comply with general government procedural requirements in this regard.

Two requirements in particular are likely to impose significant costs on commercial launch operators. First, upper stages which are left in Low Earth Orbit (or in any transfer orbit that intersects LEO), must be disposed of by re-entry into the Earth's atmosphere or by transfer to an orbit with a perigee of greater than 2000 km, within twenty-five years. Second, any re-entry event must pose a negligible hazard to bystanders on Earth. The latter requirement can be met by ensuring that the re-entry occurs over an unpopulated area, or by ensuring that the object breaks up so thoroughly and at sufficiently high altitude that little or no damaging debris can reach the surface.

2. Scope of Effort

The intent of this work is to assess the various technical means by which commercial launch providers can meet this requirement. Rough estimates will be made as to the cost and complexity of proposed solutions, and where possible detailed calculations will be made of the performance impact on the launch vehicle. It is expected that the performance impact will be the dominant cost of any debris-mitigation strategy. The cheapest commercial launch vehicles cost $10-20 million per flight, so even a 10% reduction in delivered payload will result in at least $1 million of potential lost revenue for flight. This is significantly higher than the expected marginal cost of debris-mitigation systems suitable for a small launch vehicle.

Proposed measures to increase the performance or reduce the cost of commercial launch vehicles to compensate for the adverse impact of debris-mitigation efforts are beyond the scope of this work. We presume that the developers of commercial launch systems already know how to build the best systems for their particular markets, and are not likely to be improved by any suggestions we would make here. If a commercial launcher uses gas generator cycle main engines, it is presumably because the developer believes that this represents the ideal cost vs. performance trade for their market. If implementation of debris-mitigation practices robs the system of 5% of its baseline performance, the result will almost certainly be a lower-performing launcher and not a crash program to develop a staged-combustion main engine to recover the lost performance.

The effort will also be largely focused on debris mitigation practices that can be implemented for individual expendable launch vehicles using existing or near-future technology. Reusable launch vehicles are not considered, nor are various proposals to collaboratively “clean up” Low Earth Orbit by removing debris.

3. Launch Vehicle Proxies

While the range of possible future commercial launch systems is essentially unbounded, this study will use four hypothetical vehicles as proxies for the likely range of near-future systems which would benefit from improved disposal techniques. These systems are based on reviews of publicly-available engineering and performance data from actual launch systems where possible, but should not be considered as representing any specific commercial launch vehicle.

3.1 System I

System I is a two-stage liquid fuel launch vehicle optimized for small satellite delivery to Low Earth Orbit (LEO). No such vehicle is currently in commercial service, but one has been marketed in the recent past. Also, several commercial suborbital providers have expressed an interest in adding expendable liquid-propellant upper stages to their systems to provide small satellite launch capability; as the disposal problem is relatively insensitive to lower-stage design, System I is also a reasonable proxy for expendable orbital stages launched from reusable suborbital vehicles.

The System I upper stage uses a pressure-fed bipropellant main engine and a monopropellant Reaction Control System (RCS). Payloads of up to 300 kg can be delivered to low orbits, and there is no capability for launching beyond LEO. This represents the high end of small/microsatellites; smaller vehicles may also see service but should face similar end-of-life disposal concerns. The very smallest microsatellite launchers may be inherently passively demisable due to their low ballistic coefficient.

3.2 System II

System II is a four-stage solid rocket system also largely devoted to launching small satellites to LEO, but with higher payload capability and a limited ability to reach higher orbits. The four-stage solid configuration is most suitable for designs with ICBM heritage, but at least one purely commercial system uses the same configuration. Some three-stage solid rocket launch vehicles would have very similar characteristics and should also be considered in this category.

As a four-stage system, the upper stage of System II consists of a relatively small solid rocket motor with a filament-wound case, along with avionics and controls. Flight control is provided by thrust vectoring of the solid rocket nozzle plus cold-gas thrusters for roll control. Payloads of nearly 1,500 kg can be delivered to LEO, or 500 kg to Geostationary Transfer Orbit (GTO)

3.3 System III

System III is an advanced two-stage system with either a liquid or solid first stage (the difference is irrelevant to EOL disposal issues) and a relatively large solid-propellant upper stage. While technologically similar to System II, System III is scaled for larger payloads and is thus no longer associated with heritage ICBM systems. At this scale and with a clean-sheet design, only two stages are required for efficient payload delivery to LEO. One such system is currently in operation, and a second is under development but has not yet been fully funded.

System III's upper stage is similar to that of System II, with a thrust-vectored solid motor and cold-gas reaction control system, but much larger in scale. Payloads of 5,000 kg can be delivered to LEO by the baseline configuration; adding a small solid kick motor would allow 1,000 kg to GTO.

3.4 System IV

System IV is the largest and most capable system considered, a roughly EELV-equivalent system designed to support ISS logistics missions, large communications satellite delivery to Geostationary Earth Orbit (GEO), and possibly commercial human spaceflight. For human spaceflight missions the disposal issues may be enveloped by the assured crew return requirement, but GEO and unmanned LEO missions would likely and even manned missions possibly require the disposal of an expended upper stage. Two such systems are currently in operation, two others are believed to be under development though details are scarce.

System IV is nominally an advanced two-stage liquid propellant system using pump-fed LOX/hydrocarbon rocket engines for both stages. At least one company is believed to be considering LOX/LH2 for this application, but the increased performance of LH2 would largely be balanced by the increased weight of bulky LH2 tanks and pumps, so the overall performance should be similar. The upper stage includes a bipropellant reaction control system and a sophisticated avionics suite capable of supporting extended maneuvers and multiple engine starts. Payload capacity is approximately 12,500 kg to LEO or 5,000 kg to GTO.

3.5. Upper Stage Data

Table 1 gives the key relevant performance parameters for the upper stages of the launch systems being considered. Where possible, data was obtained by analogy to existing vehicles with similar characteristics and missions.

Table 1 – Upper Stage Design Assumptions

System	System I	System II	System III	System IV
Dry Weight	550 kg	400 kg	1500 kg	3500 kg
Propellant Load	4000 kg	800 kg	12500 kg	45000 kg
Stage Diameter	1.25 m	1.0 m	2.5 m	3.75 m
Thrust	30 kN	30 kN	250 kN	450 kN
Specific Impulse	320 s	285 s	295 s	335 s
RCS Type	Hydrazine	Cold Gas	Cold Gas	Bipropellant

4. Commercial Missions

The two largest commercial launch markets at present are for the delivery of communications satellites to GEO and the delivery of logistics payloads to the International Space Station. Delivery of satellites to LEO is a smaller but growing market; in particular, replacement launches for Iridium and similar constellations may represent a market comparable to ISS resupply. Commercial human spaceflight may also be a substantial market in the future, but launch vehicle disposal/demisability requirements will likely be subsumed into the more critical requirement for intact recovery of the manned vehicle. If a separate upper stage is also left in orbit, requirements for that stage will be very similar to those for a cargo mission to ISS.

In order to simplify the trade space, this study will consider arbitrary LEO payload delivery missions to circular orbits from 200 km to 1500 km altitude at 51.6 degree inclination. This encompasses the lowest plausible altitudes for short-lived experimental satellites, the roughly 400 km altitude of ISS, and the 800 to 1400 km altitude of existing and proposed LEO communications satellite constellations. The 51.6 degree inclination was chosen to match ISS; the various LEO comsats are generally in mid-inclination orbits with similar disposal properties. A nominal GTO mission to a 185 x 35,786 km orbit at 27 degrees inclination will also be considered. This is typical for commercial GEO payload launches; the apogee altitude may vary but will have little effect on the disposal strategies considered. In all cases, Cape Canaveral Air Force Station will be assumed to be the launch site, though the implications of alternative launch sites will be discussed.

5. Passive Disposal

For small upper stages in low orbits, it may be possible to ensure safe and timely disposal by purely passive means. Atmospheric drag will result in any upper stage left in Low Earth Orbit undergoing reentry in a moderately predictable fashion. As a 25-year timeframe is required by commercial guidelines, several complete solar cycles will occur within the allowed disposal period and upper atmosphere variation due to solar cycle effects can be largely ignored. Assuming the upper stage is passively stabilized in an engine-forward configuration with a drag coefficient of 2.2 (both conservative assumptions), reentry time of the candidate upper stages will be as shown in Figure 1.

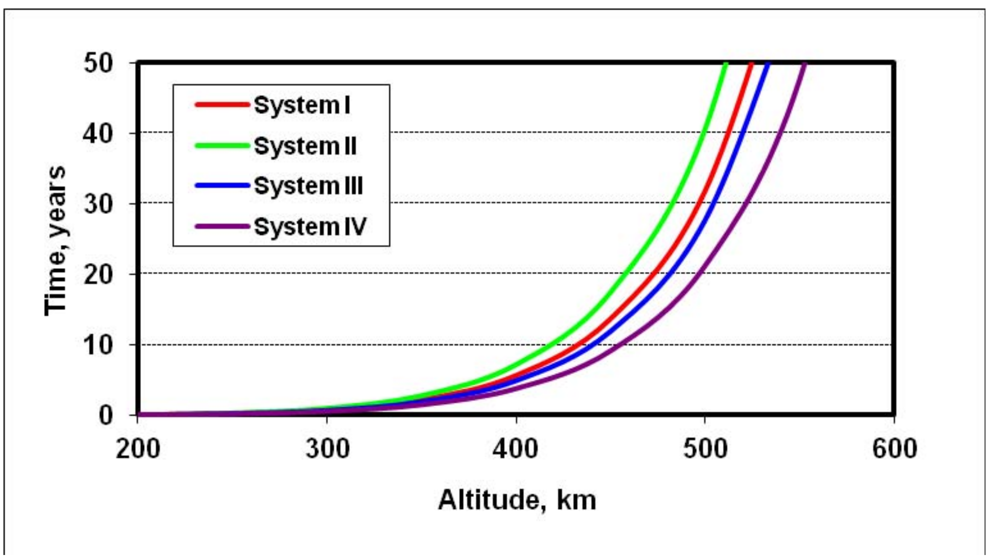

Figure 1 – Passive Disposal

Passive disposal is a viable option for launch systems delivering payloads to orbits of up to 450-500 km altitude. Unfortunately, the same atmospheric drag that facilitates passive disposal in such cases will tend to ensure that any payloads delivered will have similarly short lifetimes. This may be acceptable for short-lived experimental or scientific payloads, but these alone are unlikely to be sufficient to be a viable business model for a commercial launch vehicle. Commercial cargo delivery to ISS is another plausible mission within this altitude range, but purely passive disposal would be complicated by the requirement to avoid possible collisions with ISS itself.

Passive disposal for launches to Geostationary Transfer Orbit (GTO) is also problematic. GTO has a very low perigee, typically 185 km, which would allow for rapid orbital decay even of a highly elliptical orbit, but the perigee altitude is highly sensitive to perturbations near apogee. Such perturbations can occur due to third-body gravitational effects (primarily lunar or solar), collision-avoidance maneuvers performed by the upper stage, or propellant venting required by debris-mitigation standard practices. It is likely that some degree of active control will be required to prevent a GTO upper stage from being perturbed into an orbit stable for >25 years, and as will be shown later even a small propulsive capability near apogee for a GTO stage will allow a more reliable disposal strategy.

Even where passive reentry is likely within 25 years, safe disposal is accomplished only if no injury or damage results to uninvolved parties on the ground. As passive disposal allows no control over the upper stage reentry location, this can be assured only by minimizing the debris footprint at the ground. The Aerospace Corporation has extensively studied the type of debris recovered from upper stage and satellite events, as summarized at http://www.aerospace.org/cords/reentry-data-2/summary-of-recovered-reentry-debris/

An examination of this record shows that large and potentially quite dangerous debris often survives entry and impacts the Earth surface – see Figure 2. Virtually all large (>10 kg) debris recovered from upper stage impact events consists of stainless steel or titanium pressure vessels. These can be gas bottles used for pressurization or cold-gas RCS purposes, solid rocket motor casings, or the main propellant tanks of vehicles using pressure-fed liquid rocket upper stages. Other upper-stage components are rarely recovered intact, but small fragments are occasionally found.

Figure 2 – Examples of Upper Stage Debris after Surface Impact

It is not practical to eliminate pressure vessels from upper stages, but it may be possible to design pressure vessels such that they will not survive atmospheric entry. This would probably require using aluminum rather than titanium or stainless steel, and ensuring that any composite overwrap uses materials that will not incidentally serve as an ablative thermal protection system. Several major vendors of aerospace pressure vessels are developing such pressure vessels.

Though detailed price and performance data are not yet available, it is possible to make a reasonable estimate simply by considering the properties of the necessary materials. A designed-for-demise metallic pressure vessel would likely have a mass ~65% greater than an equivalent non-demisable system, and composite-overwrap pressure vessels would only suffer a ~25% mass increase compared to a non-demise-able composite tank. The unit cost would likely not be greatly increased, but a non-recurring cost of several million dollars would be incurred for each new type of tank or pressure vessel required.

There would not likely be any substantial design or performance impact beyond the non-recurring cost and the increased mass. As upper stage dry mass trades for payload mass at a 1:1 ratio, the performance impact is easily quantified. This impact is relatively small for vehicles using pump-fed liquid upper stage engines, modest for pressure-fed liquid upper stages, and can be quite large for solid-propellant upper stages. Table 2 gives the estimated payload penalty of a passively demisable design for the four candidate launchers.

Table 2 – Passive Disposal Impact

Launcher	**Payload Change**
System I	-45 kg
System II	-35 kg
System III	-360 kg
System IV	-175 kg

Passive disposal is not likely to be sufficient to meet demisability requirements for any commercially viable launch system due to the need to launch payloads into higher orbits. However, it may be useful as one element of a more comprehensive disposal strategy.

6. Augmented Passive Disposal

Passive disposal could in principle be extended to higher altitudes by deploying drag devices at the completion of the mission. By greatly increasing the effective cross-section of the vehicle, disposal could be accomplished from altitudes greater than 500 km while remaining within the 25-year allowable timeframe. However, even an order of magnitude increase in the effective cross section of the discarded upper stage would only increase the achievable altitude to perhaps 650 km. More importantly, it would increase the probability of a debris-generating collision by nearly an order of magnitude for the duration of the 25-year decay time. The integrated product of effective cross section and decay time, which represents the true risk of a debris incident, would be unchanged. This certainly does not meet the spirit any debris-mitigation policies, and is explicitly disallowed by some policies.

One possible exception to this would be the use of a passive electrodynamic tether for upper-stage deorbit. Electrodynamic tethers use the interaction between a long conducting wire and the geomagnetic field to produce an induced electric current and thus an electromagnetic force on the tether. The tether is maintained in a radial orientation, roughly perpendicular to the geomagnetic field at least for low-inclination orbits, by gravity gradient forces acting on the upper stage and a small counterweight at the far end of the tether. The necessary return current is carried through the space plasma.

Unlike aerodynamic drag devices, passive electrodynamic tethers have a low physical cross-section and so should not greatly increase the probability of debris-generating impacts. Also, the geomagnetic field extends to a much higher altitude than the atmosphere, such that a tether should be an effective means of disposal from any LEO altitude. Fine control of the disposal trajectory would not likely be possible, so a passively demisable structure such as described above would still be necessary.

Tethers Unlimited, Inc., has extensively promoted this concept, and estimates the mass of a disposal tether at ~2% of the dry mass of the upper stage in question. (See e.g. http://www.tethers.com/papers/TTReno00.pdf). While some proof-of-concept hardware has been built for ground testing, there has not yet been a flight demonstration of a tether deorbit system. Flight demonstrations of tether systems generally have shown poor reliability. This is still an experimental technology, with tethers frequently failing to deploy properly or being severed after deployment. If developed to a suitable degree of reliability, a combination of passive tether deorbit and demisable structure could represent a solution broadly applicable to commercial launch vehicles with low recurring cost and low performance impact.

7. Controlled Deorbit.

Where passive disposal is not sufficient, the simplest active solution is a controlled deorbit and reentry using on-board propulsion systems. This allows disposal in a matter of hours, rather than years, and it allows the reentry trajectory to be controlled such that any debris impact will occur in an unpopulated area. The latter feature eliminates the requirement that the upper stage be designed to break up completely on reentry, with its mass and cost penalties. However, it does mean that the expended upper stage will require guidance, navigation, control, and propulsion capability.

For vehicles in a circular orbit, the deorbit maneuver will be performed by waiting until the vehicle is approximately half an orbit from a suitably unpopulated disposal area, orienting the vehicle to perform a retrograde burn with the thrust vector approximately opposite to the geocentric velocity vector, and firing the appropriate thruster(s) until the perigee is reduced to approximately 50 km. At that altitude, atmospheric effects will rapidly break up and decelerate the vehicle, resulting in a debris field extending on the order of 1000 km downrange. The delta-V required for the deorbit maneuver will range from 45 m/s at 200 km altitude to 370 m/s at 1500 km altitude.

For a vehicle in GTO, the ideal disposal strategy might be to wait until apogee and perform an essentially identical maneuver, but this would entail a nearly six-hour coast period. Instead, we will impose an additional constraint that the disposal maneuver must take place within one hour of the final payload orbit insertion maneuver. With this constraint, the required delta-V is 40 m/s, and rather than a pure retrograde burn the optimal disposal burn will be at a pitch angle of roughly 135 degrees. A small yaw component may also be desired to target the disposal area.

7.1 Controlled Deorbit using Existing Propulsion Systems

The requirement for guidance, navigation, control, and propulsion for a deorbit maneuver may not necessarily be a major impact on the vehicle, given that the upper stage must have been capable of all of these functions to deliver a payload to orbit in the first place. Guidance, Navigation, and Control (GN&C) capability will need to be extended to cover a period of roughly one hour after payload separation, to allow proper targeting of the reentry area. This may require only software changes and perhaps a marginal increase in battery life. The vehicle avionics will need to provide for a modest coast period, a slew maneuver to point the vehicle appropriately for the deorbit burn, and at least yaw and pitch steering during the burn.

There will of course be increased demands on the propulsion system. At a minimum, additional propellant will be required to perform the necessary maneuvers. There may also be a need for additional Reaction Control System (RCS) thrusters. This is particularly likely for vehicles with solid-fuel upper stages using direct ascent trajectories. Such vehicles never experience sustained coast flight in their normal mission profile, and can thus meet yaw and pitch steering requirements by gimballing the main engine or motor. There is similarly no need for propellant settling burns. In such cases, the baseline RCS may be called upon only to provide roll control.

Adding additional thrusters to perform pitch and yaw steering and if necessary the deorbit burn itself, is unlikely to be a major expense. The marginal cost of the thrusters themselves would range from $100K per shipset for a cold-gas system to perhaps $500K for small bipropellant thrusters. Integration costs should be similarly small given an already-qualified propulsion system, and the marginal cost of increasing RCS tank propellant capacity should also be $100K or less. Battery capacity increase and similar changes might bring the total cost of the necessary modifications to perhaps $250K for a small, simple launch vehicle. Larger and more complex systems would of course cost more, but larger and more complex launch vehicles are more likely to already have most or all of the necessary functionality. In all cases, the mass penalty should be negligible except for the additional propellant required and its associated tankage.

For a liquid-propellant upper stage, actual propulsion for the deorbit maneuver could come from the RCS thrusters, from settling thrusters normally used prior to upper-stage main engine restarts, or from the main engine itself. Most liquid-propellant upper stages are capable of at least a single restart to support two-burn orbit insertion; adding the capability for an additional restart should have a negligible cost and mass penalty except for the need to delta-qualify the modified engine with a ground test program. Modifying the engine to support the extremely short burns associated with an efficient deorbit trajectory might be more problematic, particularly for pump-fed engines. It would also be necessary to modify the ascent trajectory to ensure that sufficient propellant remains in the tanks for a successful restart and deorbit burn, which would reduce the deliverable payload.

If these considerations preclude using the main engine for this purpose, or if the main engine is a solid-propellant motor, the RCS thrusters can be used for the deorbit maneuver themselves – they will certainly be available, as they will be needed to position the vehicle for this maneuver. The RCS thrusters, however, are likely to have a lower specific impulse than the main engine – particularly in the case of vehicles using a cold-gas RCS. Thus, propellant consumption will be increased compared to the use of the main engine.

The net payload penalty for the four candidate vehicles is shown in Figure 3, expressed as a percentage of total payload to the indicated orbit. The solid diamond symbols represent the payload penalty for GTO missions with 185 km perigee, in all cases using RCS thrusters for the deorbit maneuver. Where the main engine is used for the deorbit burn, the effect of startup transients on short-duration burns are approximated by assuming the thrust ramps to 100% of nominal over a period of 3 seconds for pump-fed engines and 1 second for pressure-fed engines, with propellant flow rate 100% of nominal throughout.

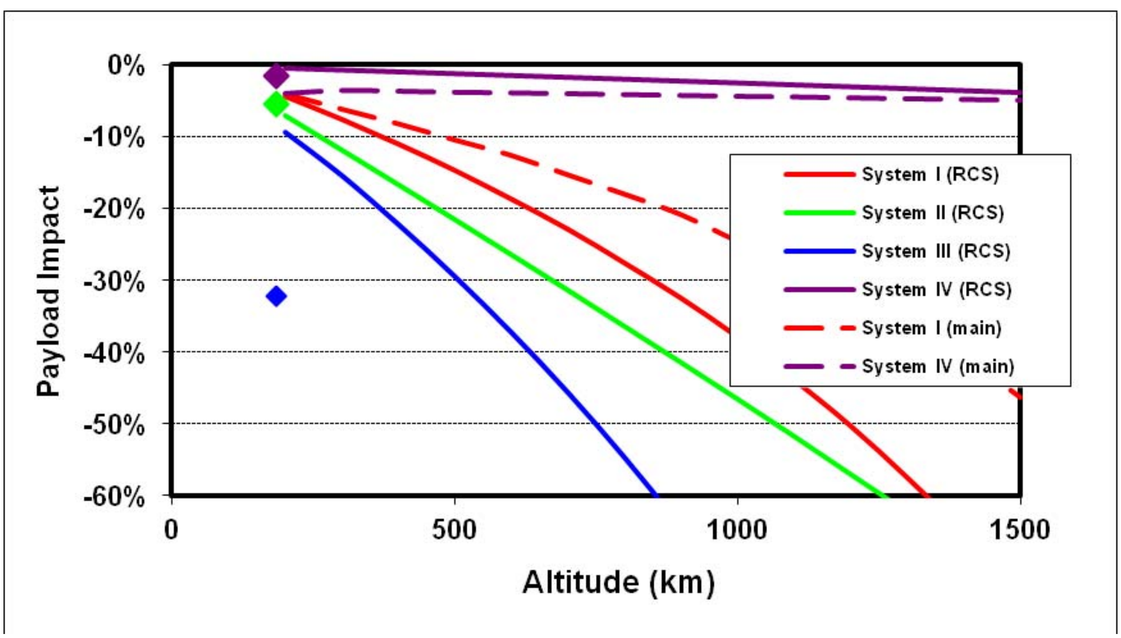

Figure 3 – Deorbit Using On-Board Propulsion

System IV, with its lightweight high-performance pump-fed upper stage and bipropellant reaction control system, can perform deorbit maneuvers efficiently even from relatively high altitudes. This is particularly true if the main engine can be used, though the burn times involved may be too short for reliable main engine startup. The lower performance of the System I upper stage, along with the relatively heavy upper stage structure, results in substantial penalties above 500 km altitude. If the main engine can be used it may be possible to retain acceptable performance with thus system. As the pressure-fed engine of System I is well suited for short burn operation, this is likely possible.

Systems II and III are not as well suited to this disposal strategy, with payload penalties reaching particularly high values for altitudes above 500 km. As purely passive disposal is adequate at lower altitudes, on-board propulsive deorbit may not be a viable option for these vehicles. This is due to the combination of non-restartable solid propellant motors for main propulsion and highly inefficient cold-gas thrusters for RCS propulsion.

7.2. Controlled Deorbit using Dedicated Propulsion

For vehicles where on-board propulsion is inadequate for efficient disposal, a dedicated deorbit propulsion system may be considered. The high performance of the System IV's bipropellant RCS suggests that bipropellant thrusters should be used in this application, but such a system is likely to be complex and expensive. Nearly the same performance can be achieved with small solid rocket motors dedicated to the deorbit mission.

Solid rocket motors are inherently inflexible at the motor level, but reasonable operational flexibility can be achieved by using a modular approach. Upper stages can be designed with brackets allowing multiple small motors of standard design (e.g., ATK's STAR-5 or STAR-6) to be added as needed for the specific mission. For missions to altitudes of 500 km or below, where passive disposal is adequate, no solid motors would be carried and only the small marginal cost and weight penalty of the empty brackets would be incurred. For missions to higher orbits, anywhere from 2 – 12 motors might be carried depending on the exact altitude. The marginal cost of small solid motors purchased in large lots is likely to be on the order of $10K each, making this a very cost-effective solution.

Again, it would be necessary to ensure that the vehicles GN&C and reaction control systems are capable of orienting the spacecraft for the deorbit burn, which may involve some hardware or software modifications. The concept of operations would be the same as for missions using existing propulsion systems to perform a deorbit maneuver.

Figure 4 shows the performance impact of deorbit using multiple small solid rocket motors. The performance of the ATK STAR-6 motor is used as the basis for these calculations, though similar systems are available from other vendors. The motors are presumed to be arbitrarily scalable and optimized for each mission altitude; in practice, a small additional penalty would likely be incurred for the use of standardized modular motors. As in the previous case, the diamond symbols represent the payload penalty for GTO missions with a 185 km perigee,

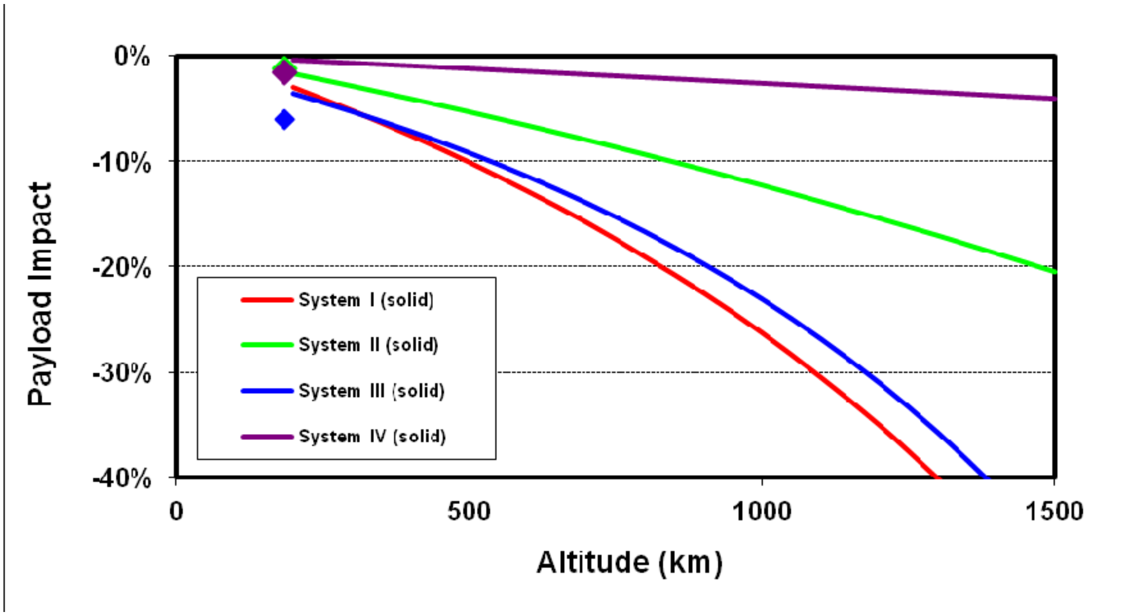

Figure 4 – Deorbit Using Dedicated Propulsion

In the case of the System I and IV, assuming the main engine is available for the deorbit maneuver, performance using dedicated solid rocket motors is roughly equivalent to that using existing on-board propulsion systems. For all other cases, the solid motor solution offers better performance and at most only slightly increased marginal costs. This increased performance is due to the superior specific impulse and propellant mass fraction of solid motors compared to liquid monopropellant or cold-gas thrusters.

The difference in relative performance between the four launch vehicles under consideration, using identical propulsion technology for upper-stage disposal, is entirely due to the relative masses of the upper stages compared to their payloads. The extremely lightweight System IV upper stage, achievable only with pump-fed liquid rocket engines, allows for a very easy deorbit. The System II uses a heavy-cased solid motor for the upper stage, but with four stages (compared to two for all of the other vehicles), that upper stage is relatively small. In essence, most of what would be the "second stage" in a two-stage configuration is jettisoned in a suborbital trajectory during ascent, leaving only a small kick stage to be disposed of. System I and III are two-stage vehicles using relatively heavy pressurized structures for the upper stage, which is an inherently difficult problem for disposal.

8. Novel Propulsion Systems

Several novel deorbit propulsion systems were considered during the course of this effort. With the exception of the possible use of electrodynamic tethers, discussed earlier, none were found to offer any advantage over passive disposal and/or controlled deorbit using small solid rocket motors. These techniques will nonetheless be summarized here.

The use of residual propellants, particularly liquid oxygen, in expended liquid-propellant upper stages seemed particularly appealing. The requirement to passivate the upper stage meant that these fluids would need to be vented to space in any event, and using them as deorbit propellants could be a substantial benefit at essentially zero marginal cost.

Unfortunately, residual propellants are quite unpredictable – if it was certain that there would be usable propellant in the system at a particular point, a maximum-performance mission would presumably plan to use that propellant. The only residuals that can be relied upon for deorbit propulsion are those which are wholly unusable for the main propulsion system – liquids trapped in dead zones in the plumbing, liquid propellants in vapor form, and pressurant gasses. It was found that the quantity of propellants thus available for deorbit purposes was not sufficient to support deorbit from altitudes greater than roughly 400 km, where passive disposal is still a suitable (and cheaper) alternative.

Advanced propulsion systems such as ion and Hall effect thrusters could greatly reduce the mass of the deorbit system through their greatly reduced propellant consumption. One particularly interesting candidate would be a helium arcjet, which could use residual pressurant gas as a propellant. However, while these would reduce the mass of the deorbit propulsion system, they would greatly increase the cost. A single arcjet, ion, or hall thruster with its associated power processing electronics will typically cost $1-2 million, and the necessary solar power system would likely be more expensive still.

Perhaps more importantly, the upper stage would be required to operate in a controlled fashion for several months after launch, requiring substantial changes not only to flight hardware and software, but to ground operations as well. And while advanced propulsion systems offer very low propellant consumption, their similarly low thrust would prevent targeting a specific reentry area and the upper stage would thus still require the full range of demisability modifications described earlier.

The possibility of propulsive disposal to a graveyard orbit of >2000 km perigee was also considered and dismissed. This would be a favorable option in some cases if advanced propulsion were used, but as noted this was already found to be uneconomical. Using chemical propulsion, disposal to a graveyard orbit almost always resulted in lower payload delivery performance than controlled reentry while requiring a more complex concept of operations.

Finally, some consideration was given to the cost impact of requiring payloads to self-deploy to higher orbits, if the launch vehicle is only capable of safe and reliable disposal from lower altitudes. This is a viable strategy that is likely to be adopted by at least some customers. However, it turns out to be technically very similar to the “deployer bus” strategy that will be discussed below, differing primarily in that the deployment system would be integrated with the payload rather than the launch vehicle. As there are likely to be many payloads for each launch vehicle, it would seem more cost-effective to design the deployment system once, as part of the launcher. Lacking the time to study both alternatives in detail, we chose to do detailed calculations for the more efficient solution. If customers are left to develop their own self-deployment capability, expect the performance impact to be similar but each customer would have to pay the full non-recurring cost of developing a custom system.

9. Separate Deployer Bus

The relatively high payload penalties seen so far for launches to altitudes of >600 km, is due to the large propulsive impulse that must be provided to target a relatively heavy spent upper stage to a controlled reentry from high altitude. The penalty could thus be greatly reduced if the upper stage mass is similarly reduced. This is incidentally accomplished by vehicles of the System II type, whose four-stage configuration results in only a very small upper stage in orbit. However, redesigning existing two-stage vehicles to a three- or four-stage configuration to exploit this effect would be prohibitively expensive.

A two-and-a-half stage configuration may be plausible. In this concept, the existing launch system would be used to place a payload in a very low orbit, perhaps 150 km circular, from which passive disposal would occur in a matter of days. A minimal liquid-propellant deployer bus would be used to deliver the payload to its operational orbit, and would then conduct a propulsive deorbit maneuver. Such deployment stages are already in common use on some small solid-propellant launch systems, e.g. Pegasus and Athena, where they are used to reduce the insertion errors associated with solid-propellant upper stages. Expanding their capabilities to allow a modest level of orbit-raising and then self-disposal would require only marginal changes. Adding such a capability to a liquid-propellant launch system would be a more substantial change, but as will be discussed below there may be synergistic advantages.

Notional deployer bus systems for the four launch vehicles in question are described in Table 3. These have been scaled by interpolation from four similar systems in use or under development elsewhere – the Hydrazine Auxiliary Propulsion System used for orbit insertion on the Pegasus launch vehicle, the Orbit Assist Module used for the same purpose on the Athena launch vehicle, an ESPA Orbital Maneuvering System being developed by AFRL for secondary-payload deployment from EELV-class vehicles, and the Fregat upper stage used on the Russian Soyuz launch vehicle. These four systems envelope the range of mass and performance required for this application. In all cases a bipropellant main engine is assumed, and the system is sized for delivery to orbits of up to 1,500 km.

Table 3 – Scaled Deployer Bus Systems

Parent Vehicle	System I	System II	System III	System IV
Dry Mass	90 kg	240 kg	415 kg	490 kg
Propellant Capacity	80 kg	330 kg	865 kg	2685 kg
Thrust	22 N	88 N	458 N	458 N
Specific Impulse	307 s	307 s	325 s	325 s
Cost, NRE	~$3M	~$6M	~$10M	~$11M
Cost, Recurring	~$1M	~$2M	~$4M	~$4M

Effect on LEO payload delivery performance is shown in Figure 5. In all cases, it is assumed that the deployer is loaded with just the propellant necessary to complete the mission with a 5% margin.

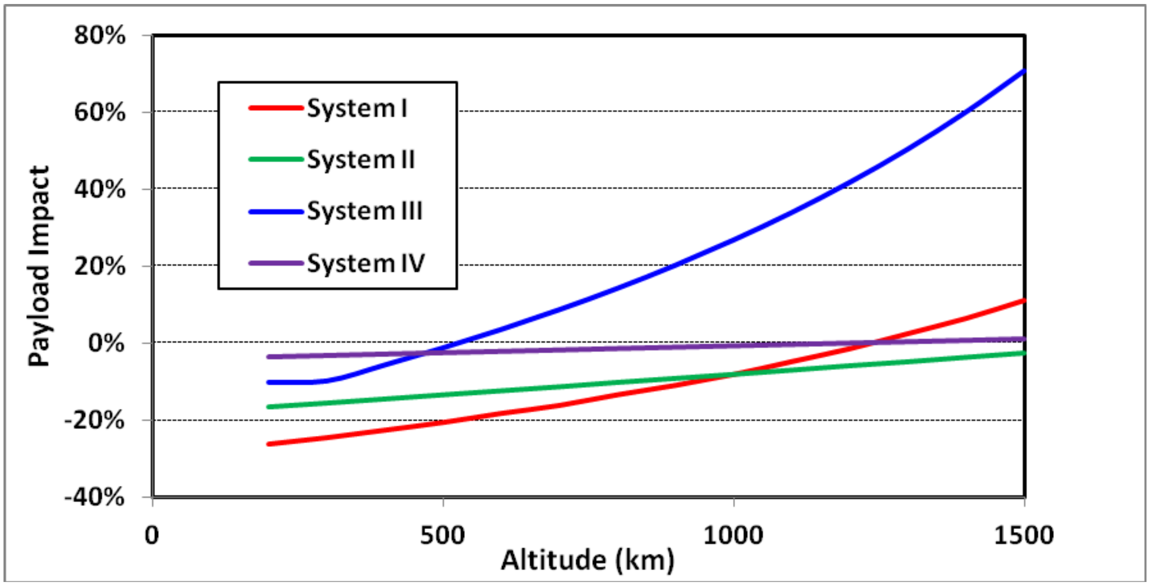

Figure 5 – Performance with Deployer Bus

Not surprisingly, payload is reduced by approximately the mass of the deployer bus for launches to very low orbits. In such cases, the mass of a nearly empty deployer bus is being carried for no real benefit. What is surprising is that for every launcher except System II, payload to higher orbits is actually increased over the baseline performance of the launch vehicle – sometimes by a very large factor. The deployer bus, in addition to being an efficient disposal mechanism, is an efficient payload delivery system.

The case for a deployer bus becomes even stronger if we consider new-design launch vehicles, or major revisions to existing systems. In the basic concept, the deployer bus requires RCS propulsion, GN&C avionics, and payload attach fittings independent of those already present on the upper stage, a heavy and expensive duplication of functionality. With a new design system, or at least a major redesign of the upper stage, it would be possible to remove most of the existing RCS, GN&C, and PAF functionality from the upper stage and use the systems on the deployer bus during normal upper stage operation.

This approach is already used with the Orbit Assist Module on the Athena series of launch vehicles. It does preclude the option of conducting low-altitude launches without using the deployer bus, which would seem to condemn the system to a 15-20% payload penalty for those missions. However, the payload mass penalty (and most of the cost of the deployer bus) would be largely recovered with the removal of redundant systems from the upper stage. There would be little or no cost or payload penalty for low orbit missions, and increased gains to higher orbits.

Several additional benefits may be obtained in specific missions. If the launch site and azimuth are such that the launch vehicle can be placed into a suborbital trajectory, additional payload could be carried to very low orbits by using the fully-fueled deployer as an ascent stage as well as an orbit transfer vehicle. This would not be feasible for the nominal 51.6 degree inclination launches from CCAFS that were studied for this mission as upper-stage debris would be likely to fall on populated areas in Europe or Africa, but may be feasible for launches to lower inclinations or from different sites (e.g. polar launches from Vandenberg or Kodiak.

A sufficiently versatile deployer bus could be used to deploy multiple small payloads to modestly different orbits before disposing of itself. Indeed, this was the primary mission of the ESPA OMS used as one of the benchmarks for this study's deployer bus designs. For missions to ISS or to future commercial space stations, a deployer bus with sufficiently precise and reliable GN&C capability might be able to deliver a payload to close enough proximity that it could be grappled with a robot arm for assisted docking, eliminating the need for a dedicated and expensive cargo transfer vehicle. If a dedicated cargo vehicle, or even a crewed capsule, is required, the deployer bus might well be able to serve as a framework on which such a vehicle could be built.

It was unfortunately not possible to examine these options further during the course of this study, but the deployer bus does seem to be an extremely promising candidate capable of doing far more than simply meeting debris-mitigation guidelines.

10. Comparison

Figures 6 through 9 show the relative impact of the proposed disposal strategies on the performance of each of the four launch systems considered in this study.

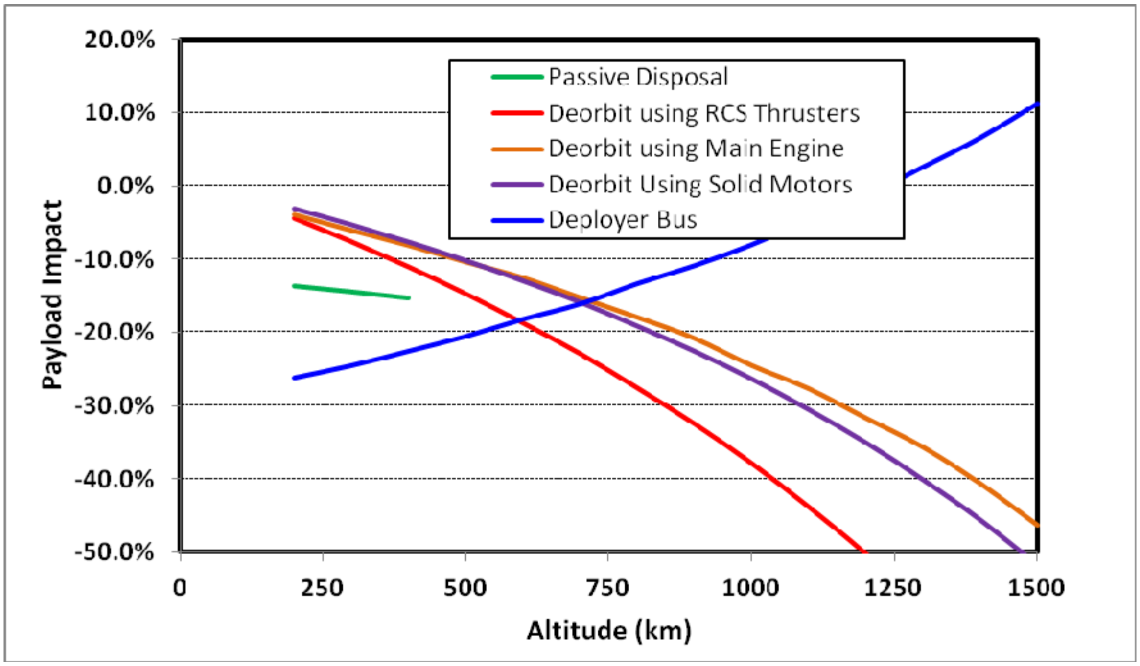

Figure 6 – System I Disposal Options

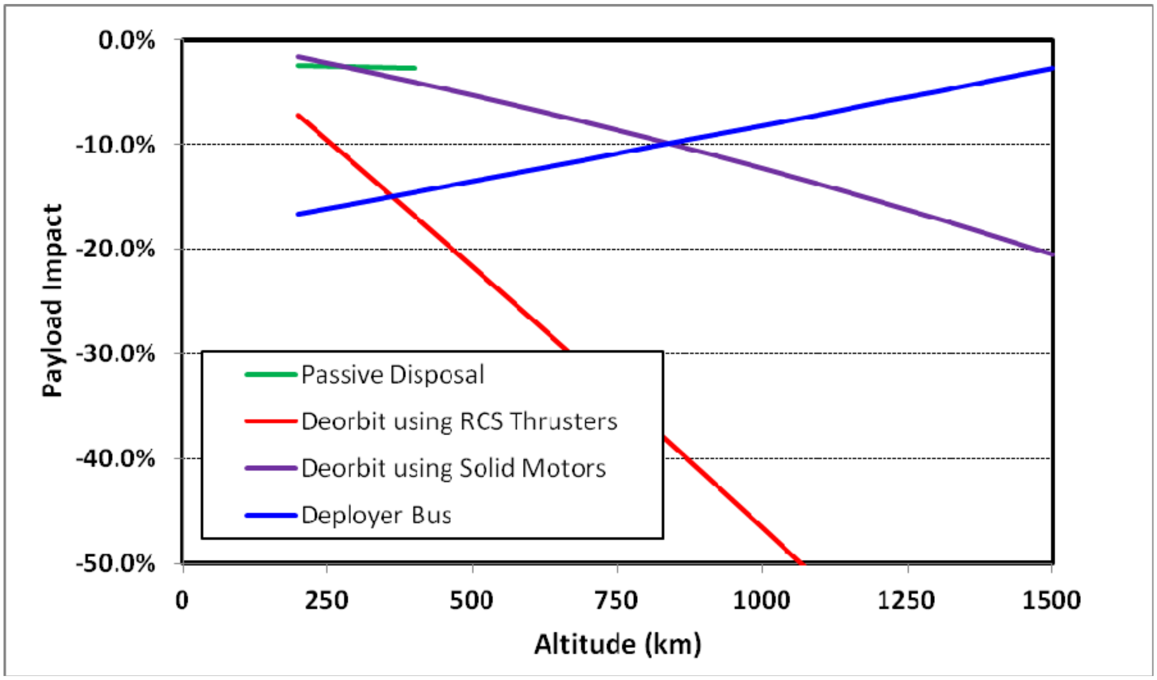

Figure 7 – System II Disposal Options

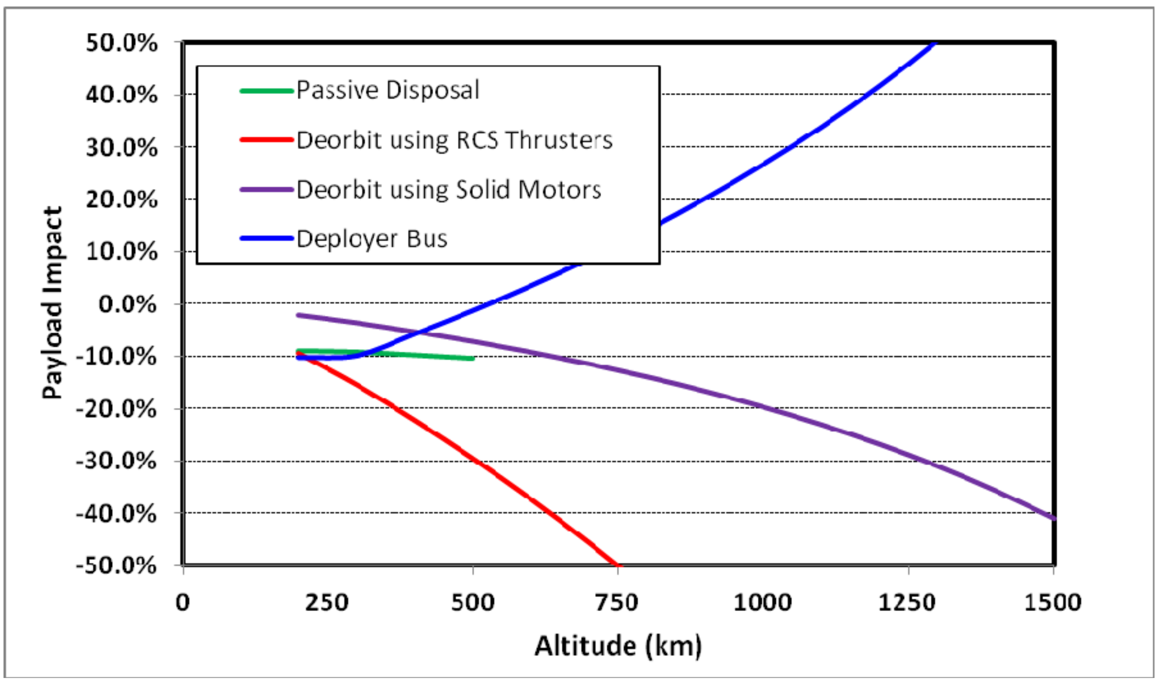

Figure 8 – System III Disposal Options

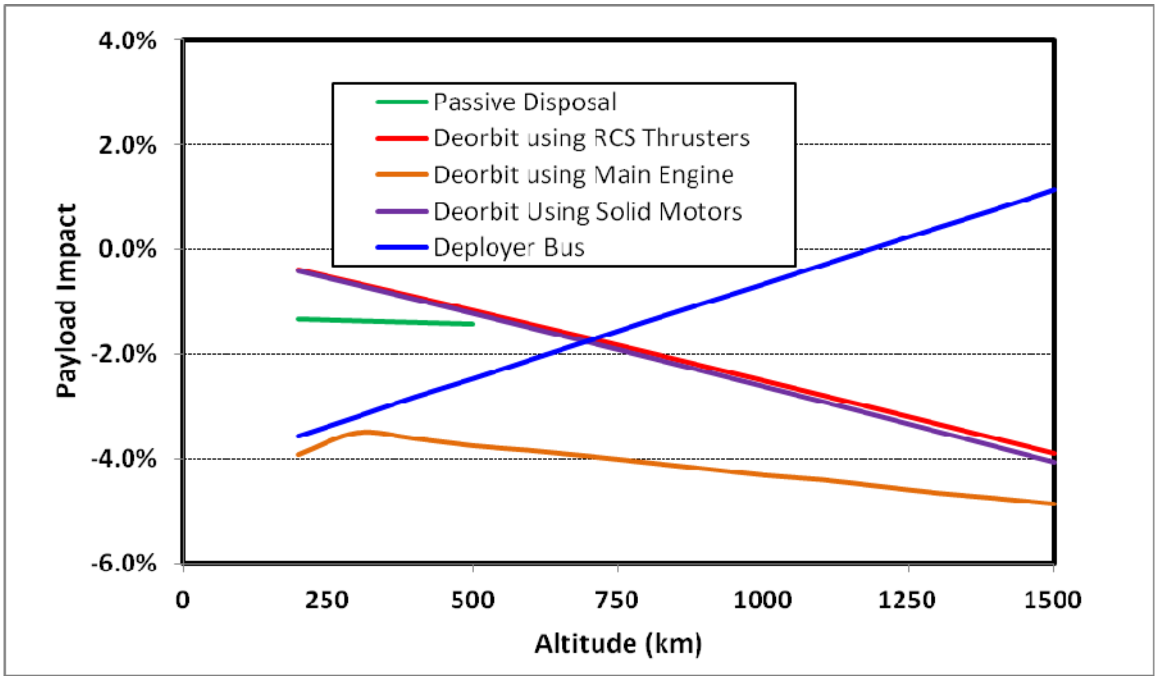

Figure 9 – System IV Disposal Options

As can be seen, no one disposal option is universally better or worse than the rest for all cases. Still, some general trends can be observed. Vehicles with lightweight, high performance face a much smaller disposal challenge than vehicles with heavier and lower-performance upper stages, e.g. solid propellant or pressure-fed liquid propellant systems. In the case of System IV, there is little payload impact no matter what system is chosen.

Passive disposal is only suitable for orbits up to 4-500 km, and is rarely the highest-performing system. Due to its simplicity and low cost, it may well be worth consideration in spite of the performance penalty.

Active disposal using some sort of on-board propulsion system generally offers the best performance up to about 750 km altitude. If a pressure-fed main engine or a bipropellant reaction control system is available on the upper stage, this will probably be the most attractive option for the disposal mission as well. Otherwise, a modular system of small solid-propellant thrusters will probably give the best performance at the lowest cost.

For orbits of 750 km to 1500 km, the deployer bus gives the best performance, and may actually represent a net improvement over the baseline launch vehicle. Not noted in the payload comparison chart are various improvements in mission flexibility allowed by this system. Also not noted is the substantial up-front cost and complexity of the system.

Finally, the disposal problem for GTO missions requires very little propulsive effort and has a similarly small impact on payload. The principle challenge is ensuring that the vehicle remains under control and capable of performing a targeted disposal maneuver an hour or so after what would otherwise be the end of mission; given that, any propulsion system other than cold-gas thrusters will provide good performance.

11. Conclusions

Commercial launch vehicles will require some modifications to hardware, software, and operational concepts to meet debris mitigation requirements as those requirements are increasingly imposed on the commercial market. These modifications need not be onerous or expensive; recurring costs of only a few hundred thousand dollars per vehicle are a reasonable expectation, as are payload penalties of no more than ten percent or so.

Passive disposal will likely be the lowest-cost option for vehicles delivering payloads to orbits no higher than 4-500 km. This will require using designed-for-demise pressure vessels and solid motor casings, and provisions for passivating any systems with stored energy, e.g. batteries and pressurized tanks. Given these provisions, the upper stage can be safely left in orbit to decay and reenter under the influence of atmospheric drag in less than twenty-five years, with negligible probability of either generating debris on orbit or causing damage on the ground.

If there is any expectation of delivering payloads to higher orbits, an active disposal capability will be required. This will require enhancing the vehicle's avionics and reaction control system to allow positive control of the vehicle for roughly one hour past normal payload separation, and will require some propulsion system more capable than a cold-gas RCS thruster to perform the deorbit maneuver. Such a propulsion system may already be present on the upper stage, e.g. a restartable main engine or a bipropellant RCS thruster set. If this is not the case, a modular system of small solid rocket motors is likely the lowest-cost option still providing good performance. The main and/or RCS propellant budget, and if appropriate the solid motor configuration, should be tailored for the specific mission to avoid excess performance penalties. These penalties are nonetheless likely to become substantial for orbits with perigee altitudes greater than 750 km.

For payload delivery to orbits above 750 km, an active deployer bus is likely the preferred solution. This would best be combined with a minimal active deorbit capability for the upper stage, allowing the active deployer to be dispensed with for missions to low orbits. It may also be worth considering the combination of an active deployer and a passively demisable upper stage, in which case the deployer bus would be required for any destination above 400 km or so. In any event, the cost of an active deployer system would be in the low millions of dollars per launch, but this cost would likely be recovered in the form of enhanced payload capability and more flexible mission operations.

For a new-build design, it would be worth considering a deployer bus tightly integrated with the launch vehicle upper stage. In this case, the deployer bus would be mandatory for all missions, but the ability to remove redundant systems from the upper stage would recover most of the associated cost and performance penalties. The deployer bus would then provide a net performance gain to any higher-orbit mission, while facilitating more flexible operations in several other respects.