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Space Debris Mitigation and Space Systems Design

Dietrich Rex
Technical University of Braunschweig, Germany
Institute for Flight Mechanics and Spaceflight Technology

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- old upper stages launched before the passivation measures still do explode
- Delta and Ariane launches represent only a fraction of all launches
- much fragmentational catalogued debris is being generated by satellite break-ups in addition to rocket upper stages.

From this simple statistical consideration we can learn that future mitigation measures must be much more comprehensive to become effective. Because of the stepwise implementation of such measures in the real world, i.e. for all the existing rocket types and newcomers, it will take a long time, probably decades to result in a sensible reduction of debris. The transition phase to cleaner spaceflight is not only long but also will require a lot of engineering effort in detail. So it is even more urgent to get it started soon.

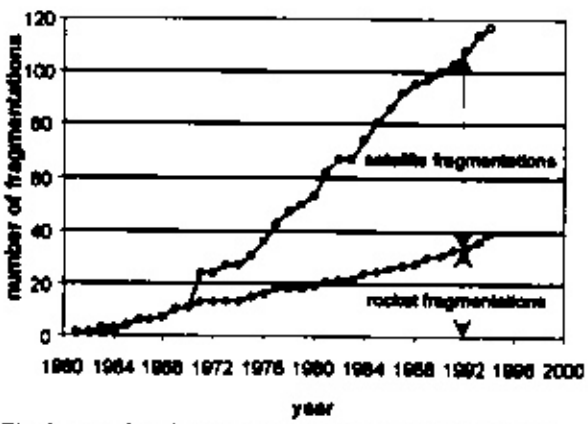

Fig.1 cumulated no. of fragmentations (1) satellite break-ups (2) rocket bodies break-ups

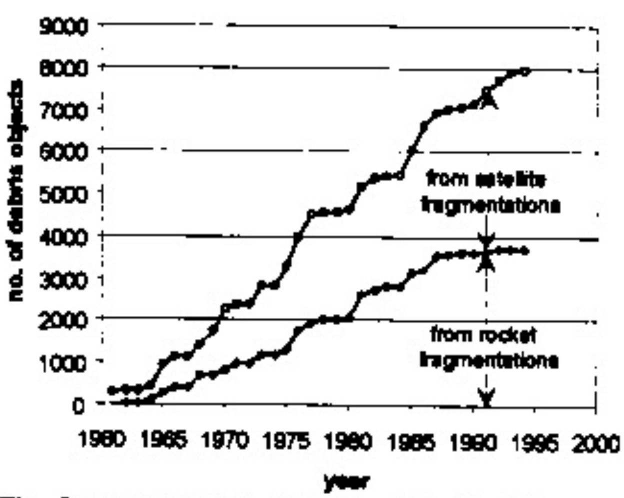

Fig. 2 cumulated no. of fragmentational debris objects catalogued

Technological Problems of Fuel Venting

The venting of liquid oxygen and liquid hydrogen from cryogenic rocket upper stages is by no means a simple procedure. First, the opening of the venting valves on command and especially their tightness before command has to be secured reliably and an extra command line has to be implemented. Usually, as e.g. in the Ariane 4 upper stage, the O_2- and H_2-tanks have a common bulkhead, which requires overpressure always from its concave side while overpressure from the other side will easily break it. Such fracture of the bulkhead, of course, would result in a desastrous explosion and fragmentation of the structure. So, during the venting process, the differential pressure has to be carefully watched, and if it develops towards the wrong direction, the venting in one of the tanks has to be stopped until the proper pressure gradient is re-established. The implementation of such rather delicate technology is not easily got accepted into a system, the market value of which is extremely dependent on its proven launch reliability. The most elegant way of removing residual propellants from the tanks is the depletion burn also used for lowering the orbit. So far, this has only been accomplished in some Japanese launches. The residual fuel and oxidizer is fed, without turbopump action, into the combustion chamber, the ignition is restarted. The low pressure burn produes thrust which, properly directed, can be used for de-orbiting of the stage. Of course, it requires full attitude control over the stage until the end of the depletion burn.

DE-ORBITING OF LARGE SPACE OBJECTS

Types of Satellites and Rocket Stages to be De-orbited

In the beginning of the technical discussions on de-orbiting, there was a very general and idealistic way of thinking: All rocket upper stages after payload delivery and all satellites at the end of their active life should be removed from orbit immediately, so that they reach the Earth's surface on one of the big oceans. Of course, this would effectively limit the accumulation of big orbital objects to a low level and at the same time would eliminate any possible risk posed by re-entering objects to inhabited areas. A closer examination of the orbital accumulation process and its influence on the future on orbit collision threat revealed, that the collision probability is mainly due to big objects with very long orbital lifetimes. It was proven, that the orbital density could sufficiently be limited by only reducing the remaining orbital life time of those long-living objects at the end of their active life (or after payload delivery in the case of rocket upper stages). Of course this is no longer a remedy to the impact threat on inhabited areas, because the present situation of randomly distributed re-entry points is not changed. Although no major damage to the ground by re-entering objects has occured since the beginning of spaceflight, there were repeated world-wide concerns on the occasion of massive objects' re-entries. It should be considered that for very massive objects – whether on long living orbits or not – an end-of-life manoeuvre should be performed with direct impact on a predetermined uncritical location on the Earth's surface.

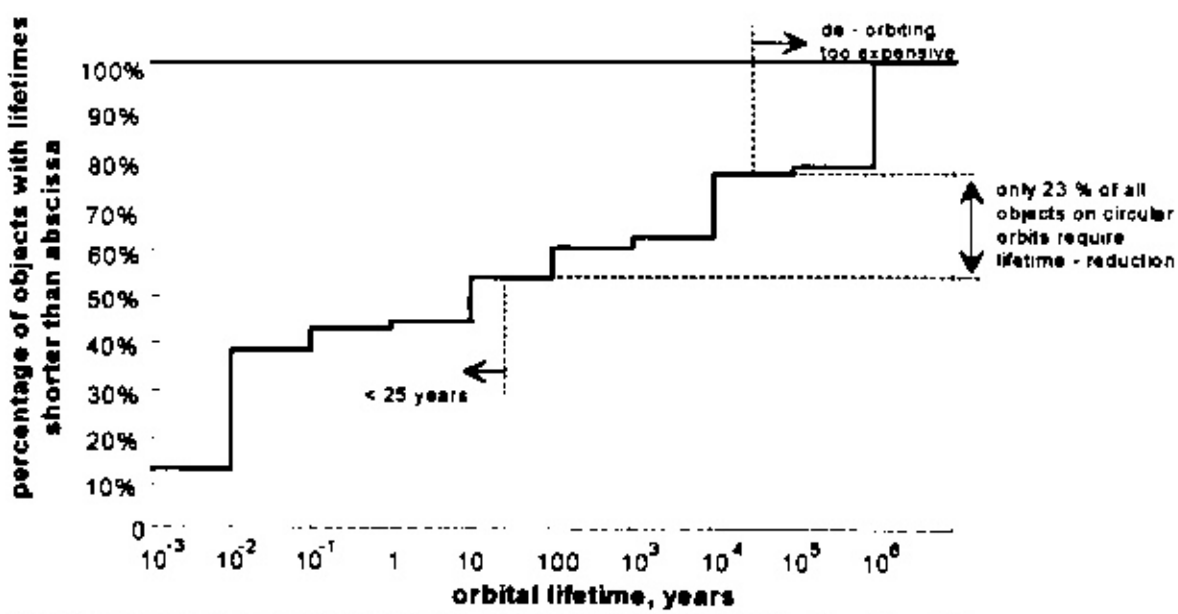

Fig.3. Lifetime distribution of launched objects in the past. near circular orbits with $\varepsilon < 0.1$

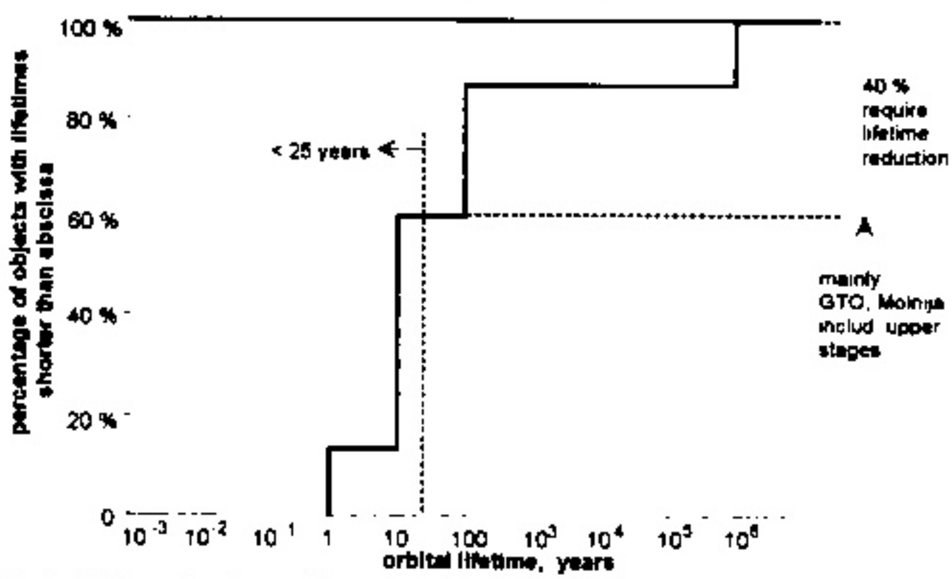

Fig.4 Lifetime distribution of launched objects in the past. Highly elliptic orbits with $\varepsilon > 0.1$

For the purpose of spatial density limitation, it can be discussed which remaining life time after the active life of satellites and rocket upper stages might be acceptable. NASA has made the proposal to accept remaining lifetimes shorter than 25 years.
After many calculations of future scenarios with this value and other values, our group at the Technical University of Braunschweig supports the value of 25 years for the residual lifetime and also recommends it to ESA and DARA. By limiting the remaining lifetime to 25 years it can be shown, that the future collision risk does not reach high levels and that the cascading effect is avoided, while also the cost burden for spaceflight as a whole is very limited.
The lifetime limitation to 25 years also implies, that de-orbit manoeuvres need not be performed for all objects which have an orbital lifetime shorter than 25 years at the beginning of their mission. What percentage of all objects will need to perform an end-of-life manoeuvre? Of course this depends on the future altitude distribution of satellites and upper stages according to mission needs. If we assume that there will be no marked change in the altitude and eccentricity distribution in future launches compared to past launches, then we can use the orbital lifetime distribution of past launches to assess the frequency of de-orbit manoeuvres necessary in the future. This has been done in Fig. 3 and Fig. 4. The launches of recent years have been statistically evaluated using the RAE-Tables. The percentage of launched objects with lifetimes shorter than the abscissa value is plotted for near circular orbits ($\varepsilon < 0.1$) in Fig. 3, for highly elliptical orbits ($\varepsilon > 0.1$) in Fig. 4.

De-orbiting of Objects in Near Circular Orbits

We first consider the situation on near circular orbits in Fig. 3. Objects with orbital lifetimes shorter than 25 years make up for about 50 % of all trackable objects. The remaining 50 % with larger lifetimes would be candidates for de-orbiting or lifetime reduction. However, for very long initial orbital lifetimes, i.e. very high near circular orbits, any propulsive de-orbit or lifetime reduction

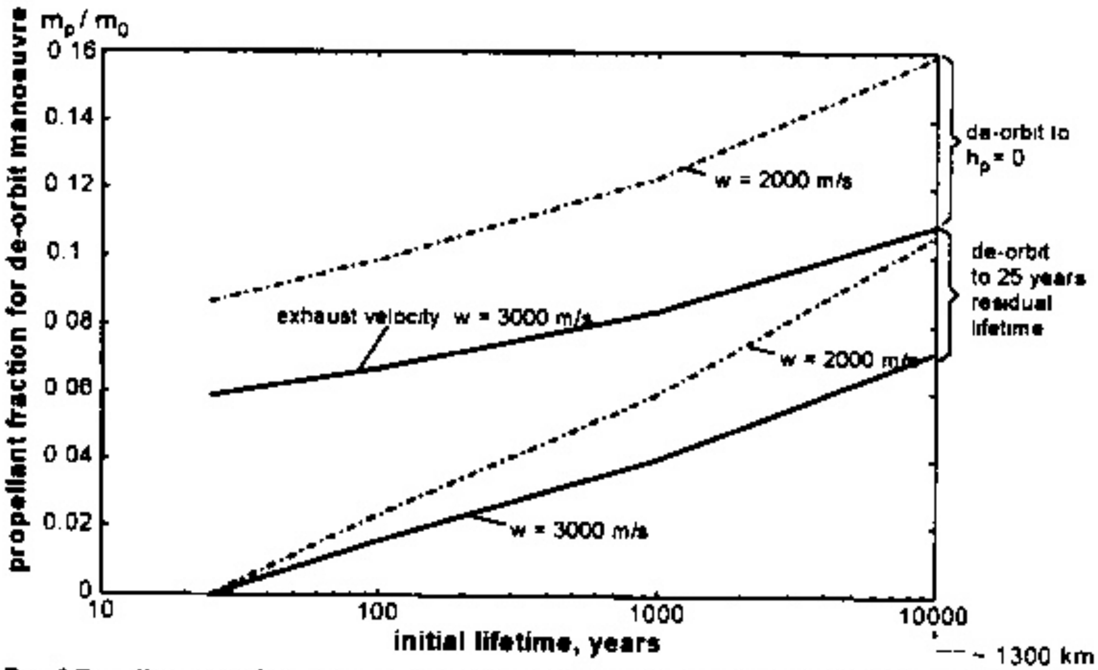

Fig. 5 Propellant mass fraction required for de-orbit or lifetime reduction from near circular orbits. m/A = 100 kg/m²

manoeuvre will be too costly to be seriously considered. A transfer to higher storage orbits above the presently used LEO altitude band would be cheaper. We do not recommend this practice, though, because it would tend to become a burden to altitudes which often have to be traversed by and other missions and which might turn out to be needed for new mission types in the future.
Rather than relying on such a pseudo-solution, one should search for other means of lowering the orbit: i.e. electric propulsion, electrodynamic tethers etc.

The propellant fraction which is needed for a propulsive orbit lowering manoeuvre starting from orbits between 25 years and 10.000 years initial lifetime is plotted in Fig. 5. Two sets of curves show the requirements for lowering to elliptical orbits of 25 years lifetime and to elliptical orbits with perigee altitude zero, i.e. direct re-entry at a pedetermined location on the Earth's surface. Practical thruster exhaust velocities will be in the band 2000 ... 3000 m/s (specific impulse ~ 200 ... 300 s) indicated in the graph. Certainly propellant mass fractions for the de-orbit manoeuvre beyond 10 % are unacceptable. That would mean, that such a propulsive manoeuvre for orbits with initial lifetimes longer than 10.000 years can hardly be proposed. So, propulsive de-orbiting or lifetime reduction is only feasible for lifetimes of the initial near circular orbit between 25 years and 10.000 years. Fig. 3 shows, that only about 23 % of all launched objects fall into that intervall. Propulsive de-orbiting is a debris mitigation measure for only a small fraction of all launched satellites and upper stages! Nearly all earth-observation satellites on circular sun-synchronous orbits are at the upper edge of this 23 % class. It will become a major decision, whether satellites of this type will require de-orbiting (lifetime reduction) or not. For the sake fo the protection of these satellites it would of course be desirable.

<u>De-orbiting of Objects on Highly Elliptical Orbits</u>
We now come back to the objects shown in Fig. 4 which have been launched on highly elliptical orbits with eccentricities larger than 0.1. About 40% of these have lifetimes longer than 25 years (Fig. 4) and so would require de-orbiting. The majority of these are rocket stages on GTO and Molnija orbits. For Molnija-Type missions the de-orbit Δv is small if applied at the apogee and would not pose a big problem. Rocket stages on GTO, however, have a special problem. Shortly after perigee passage (near to the first equator crossing after the launch) they are seperated from the satellite and then they are abandoned, i.e. there is no longer attitude control or any other control over the stage. They are powered by batteries, and these are designed to survive only until a few minutes after satellite separation. The best orbital position for a manoeuvre to reduce the lifetime (if such manoeuvre is to be performed because of lifetime longer than 25 years) is at the apogee, however, that position is reached after more than 5 hours, long after the end of battery capacity. The de-orbit manoeuvre can be performed earlier on the way from perigee to apogee, but it is less efficient then. This trade-off between the time of the manoeuvre and the perigee-lowering effect has been analyzed also by Eichler et.al.[1]. The geometry is shown in Fig. 6. The Δv imposed by the de-orbit thrust is given at the angle φ from the perigee which is

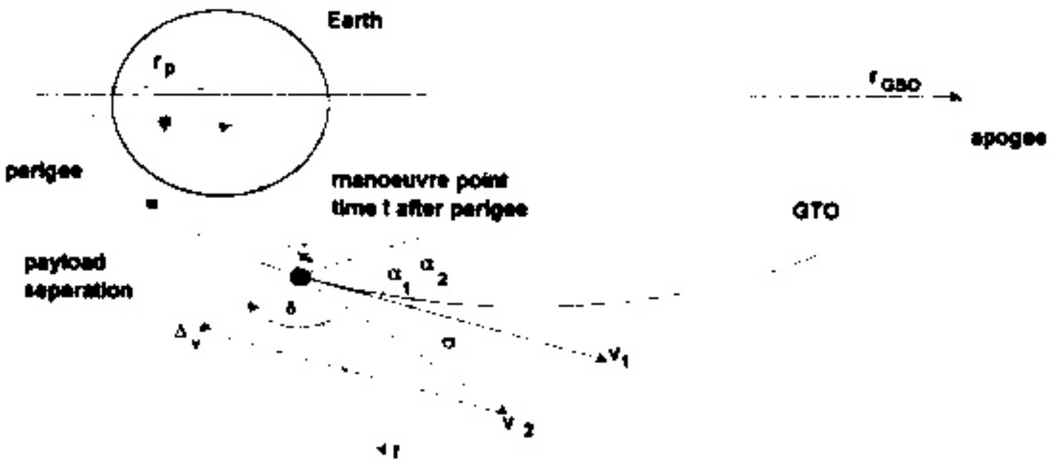

Fig. 6 Geometry for de-orbit thrust Δv shortly after perigee passage

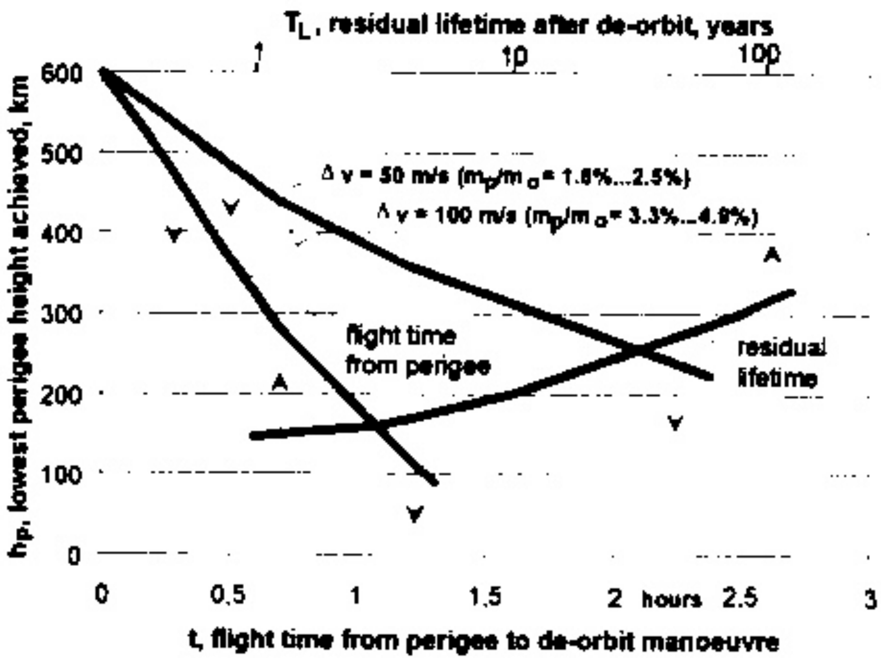

Fig. 7. De-orbiting from GTO with h_p= 600 km. Optimal lowest perigee altitude and corresponding residual orbital lifetime achieved by a thrust manoeuvre at time t after perigee passage. Initial orbit: $h_a = h_{GSO} = 42164$ km, $h_p = 600$ km.

given in the orbital plane at an angle δ from the velocity vector, producing the velocity increment Δv. The best manoeuvre would be with $\varphi = 180^0$ and $\delta = 180^0$ as is well known. That manoeuvre would be performed at 5.2 hours (half period of ellipse) after perigee passage. Because of the battery time constraint, the effectiveness of earlier manoeuvres to lower the perigee is analyzed. For each manoeuvre point φ, the optimal direction angle δ for the Δv has to be determined. The equations which determine the perigee altitude reached for each φ and δ are:

preset: $h_p = 600$ km → $r_p = 6971$ km

$r_a = r_{GSO} = 42164$ km

parametric variation: φ, δ, Δv

calculation:

subscript (1) before the manoeuvre

subscript (2) after the manoeuvre

semimajor axis $a_1 = \frac{1}{2}(r_{a,1} + r_{p,1})$

Orbital angular momentum: $h_2 = r v_2 \cos(\alpha_1 + \sigma)$

parameter $p_1 = a_1 (1 - \varepsilon_1^2)$

eccentricity $\varepsilon_1 = (r_{a,1} - r_{p,1})/(r_{a,1} + r_{p,1})$

$$r = \frac{p_1}{1 + \varepsilon_1 \cos\varphi}$$

$$\cos\alpha_1 = \frac{\sqrt{\mu p_1}}{(r v_1)}$$

$$v_1 = \sqrt{\mu(2/r - 1/a_1)}$$

$$v_2 = \sqrt{v_1^2 + (\Delta v)^2 - 2 v_1 \Delta v \cos(180 - \delta)}$$

$$a_2 = \frac{\mu r}{2\mu - r v_2^2}$$

$$\sin \sigma = \Delta v \bullet \sin(\delta) / v_2$$

$$p_2 = h_2^2 / \mu$$

$$\varepsilon_2 = \sqrt{1 - p_2 / a_2}$$

$$r_{p,2} = a_2 (1 - \varepsilon_2)$$

$$r_{a,2} = a_2 (1 + \varepsilon_2)$$

In addition, the Kepler equation determines the flight time from perigee to the manoeuvre point.For each manoeuvre point, the optimal angle δ is determined, i.e. the angle δ for which the lowest perigee is obtained. The results for the lowest perigee and the resulting residual orbital lifetime after the manoeuvre are plotted in Fig. 7. If the manoeuvre is done 1 hour after perigee with 50 m/s, then the resulting new perigee would come down to 385 km (instead of 600 km initially), but the lifetime would still be far above 100 years. If however, the same manoeuvre is performed 2.2 hours after perigee (larger batteries) then the new perigee altitude is 240 km corresponding to 25 years residual lifetime. If batteries are limited to a shorter time after perigee, say 0.8 h, then one would have to spend Δv = 100 m/s to reach the same perigee altitude and the same residual lifetime of 25 years.

Slight variations of the apogee altitude resulting from these manoeuvres have no sensible influence on the lifetime.

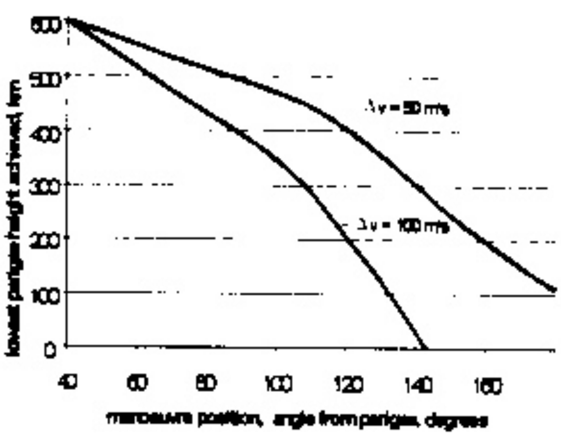

Fig. 8. De-orbiting from FTO with h_p = 600 km. Lowest perigee altitude by a thrust at revolution angle from perigee

From Fig. 8, where the lowest perigee altitudes are plotted over the manoeuvre point angle φ, one can see, that the angles corresponding to the three examples given above are in the interval φ = 90 ... 115°. The optimal thrust angle δ between the velocity vector and the thrust vector is in these cases close to 130° (not visible in the figure).The trade-off between battery mass and propellant mass can only be analyzed on the basis of the exact values of specific impulse of the thruster and the mass/watt-hour of the batteries. Also the altitude control over the time t has to be taken into account. However the results indicate, that an upper stage of 1200 kg empty mass can be de-orbited to 25 years residual lifetime with propellant mass ranging from 20 kg to 61 kg with w = 2000 ... 3000 m/s applied 0.8 hours to 2.2 hours after perigee (i.e. shortly after payload separation) if the manoeuvre is properly optimized. So de-orbiting of upper stages from GTO with perigee altitude of around 600 km seems to be feasible with e.g. small solid propellant thrusters.

CONCLUSIONS

The further planing of mitigation measures has to be considered in technical detail. That will require the co-operation with system design engineers. De-orbiting to the ground (i.e. over the oceans) should be considered for very big objects in order to avoid impact hazard to populated areas. Objects on long living orbits will require lifetime reduction to 25 years. Such manoeuvres should be designed seperately for high circular orbits, for sun synchronous orbits, for Molnija-type orbits and for GTO with high perigees. Lifetime reduction to 25 years from GTO with high perigee is feasible if properly optimized. Altogether, only about 25 % of all launched payloads and rocket upper stages will require a lifetime reduction manoeuvre. New technologies, as electrical propulsion and electrodynamic tethers should be analysed for lifetime reduction of objects on circular orbits higher than about 1300 km, which with conventional propulsion might need more than 10 % of their mass for this manoeuvre.

Reference

[1] Peter Eichler, Robert Reynolds, Jingchang Zhang, Anette Bade, a.a. Jackson, Nicholus L. Johnson, and Roger Mc Namara. Post Mission Disposal Options for upper stages
SPIE, Optical Science, engineering and Instrumenation SD97 Symposium, July-August 1997, paper No. 3116-17.