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New Methods of Removing Space Debris

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Abstract

In 1957 the new era of studying outer Space by space apparatus was ushered in. During this past half century, thousands of satellites, space ships and space stations were launched. The tens of thousands of slivers from old rockets remaining in Space represent a serious danger for new satellites, space craft, space stations and space travelers. Currently, about 19,000 pieces of debris larger than 5 cm (2.0 in) are tracked. Any of them can damage a space apparatus. This work delineates the problems of the space debris (amount, distribution, growth, danger and so on), reviews the old and contemporary methods of cleaning up the debris, and evaluates the relative efficiency of the current and new methods. This paper offers new methods and installations for cleaning outer space from space debris and specific protection of important space ships and stations from large space debris (SD). Advantages of the offered method and apparatus are the following: 1. Smaller size and weight by 2 -3 times than conventional SD Collector. 2. Greater efficiency by 2 -3 times. 3. Saves fuel by some times. 4. No limits in size for SD. 5. Can easily protect a space ship and station (for example, International Space Station) from SD.

Key words: Space debris, collection space debris, cleaning space, methods of removing the space debris, removal space debris technology.

Introduction

Short Review of Debris Problem

Satellites.

The world's first artificial satellite, the Sputnik 1, was launched by the USSR in 1957. Since then, thousands of satellites have been launched into orbit around the Earth. Artificial satellites originate from more than 50 countries and have used the satellite launching capabilities of ten nations. A few hundred satellites are currently operational, whereas thousands of unused satellites and rocket fragments orbit the Earth as space debris.

About 6,600 satellites have been launched. The latest estimates are that 3,600 remain in orbit. Of those, about 1000 are operational; the rest have lived out their useful lives and are part of the space debris. Approximately 500 operational satellites are in low-Earth orbit, 50 are in medium-Earth orbit (at 20,000 km), and the rest are in geostationary orbit (at 36,000 km).

When satellites reach the end of their mission, satellite operators have the option of de-orbiting the satellite, leaving the satellite in its current orbit or moving the satellite to a graveyard orbit. As of 2002, the FCC requires all geostationary satellites to commit to moving to a graveyard orbit at the end of their operational life prior to launch.

A graveyard orbit, also called a junk orbit or disposal orbit, is a supersynchronous orbit that lies significantly above synchronous orbit, where spacecraft are intentionally placed at the end of their operational life. It is a measure performed in order to lower the probability of collisions with operational spacecraft and of the generation of additional space debris (known as the Kessler syndrome).

A graveyard orbit is used when the change in velocity required to perform a de-orbit maneuver is too high. De-orbiting a geostationary satellite requires a delta-v of about 1,500 metres per second (4,900 ft/s), whereas re-orbiting it to a graveyard orbit only requires about 11 metres per second (36 ft/s).

For satellites in geostationary orbit and geosynchronous orbits, the graveyard orbit is a few hundred kilometers above the operational orbit. The transfer to a graveyard orbit above geostationary orbit requires the same amount of fuel that a satellite needs for approximately three months of station keeping. It also requires a reliable attitude control during the transfer maneuver. While most satellite operators try to perform such a maneuver at the end of the operational life, only one-third succeed in doing so.

Space Debris.

Space debris, also known as orbital debris, space junk, and space waste, is the collection of defunct objects in orbit around Earth. This includes everything from spent rocket stages, old satellites, and fragments from disintegration, erosion, and collisions. Since orbits overlap with new spacecraft, debris may collide with operational spacecraft.

Currently, about 19,000 pieces of debris larger than 5 cm (2.0 in) are tracked, with another 300,000 pieces smaller than 1 cm below 2000 km altitude. For comparison, the International Space Station orbits in the 300–400 km range and both the 2009 collision and 2007 antisat test events occurred at between 800 and 900 km.

Most space debris is less than 1 cm (0.39 in), including dust from solid rocket motors, surface degradation products such as paint flakes, and coolant released by RORSAT nuclear-powered satellites. Impacts of these particles cause erosive damage, similar to sandblasting. Damage can be reduced with "Whipple shield", which, for example, protects some parts of the International Space Station. However, not all parts of a spacecraft may be protected in this manner, e.g. solar panels and optical devices (such as telescopes, or star trackers), and these components are subject to constant wear by debris and micrometeoroids. The flux of space debris is greater than meteoroids below 2000 km altitude for most sizes circa 2012.

Safety from debris over 10 cm (3.9 in) comes from maneuvering a spacecraft to avoid a collision. If a collision occurs, resulting fragments over 1 kg (2.2 lb.) can become an additional collision risk.

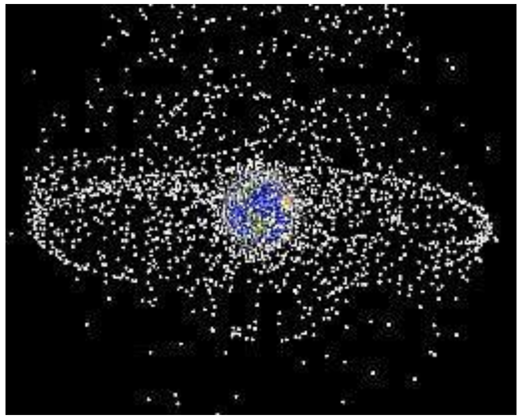
Total mass of space debris is about 5000 tons (2009).

As the chance of collision is influenced by the number of objects in space, there is a critical density where the creation of new debris is theorized to occur faster than the various natural forces remove them. Beyond this point, a runaway chain reaction may occur that pulverizes everything in orbit, including functioning satellites. Called the "Kessler syndrome", there is debate if the critical density has already been reached in certain orbital bands.

Through the 1980s, the US Air Force ran an experimental program to determine what would happen if debris collided with satellites or other debris. The study demonstrated that the process was entirely unlike the micro meteor case, and that many large chunks of debris would be created that would themselves be a collisional threat. This leads to a worrying possibility – instead of the density of debris being a measure of the number of items launched into orbit, it was that number plus any new debris caused when they collided. If the new debris did not decay from orbit before impacting another object, the number of debris items would continue to grow even if there were no new launches.

Measurement, growth mitigation and active removal of space debris are activities within the space industry today.

The USAF's conclusions about the creation of debris. Although the vast majority of debris objects by number was lightweight, like paint flecks, the majority of the *mass* was in heavier debris, about 1 kg (2.2 lb.) or heavier. This sort of mass would be enough to destroy any spacecraft on impact, creating more objects in the critical mass area. As the National Academy of Sciences put it:


Space debris

A 1-kg object impacting at 10 km/s, for example, is probably capable of catastrophically breaking up a 1,000-kg spacecraft if it strikes a high-density element in the spacecraft. In such a breakup, numerous fragments larger than 1 kg would be created.

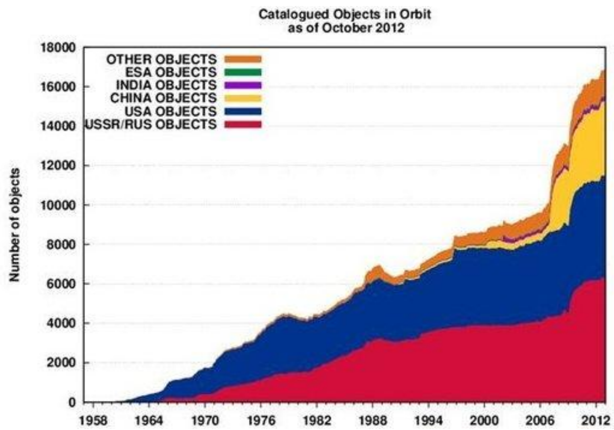
Aggressive space activities without adequate safeguards could significantly shorten the time between collisions and produce an intolerable hazard to future spacecraft. Some of the most environmentally dangerous activities in space include large constellations such as those initially proposed by the Strategic Defense Initiative in the mid-1980s, large structures such as those considered in the late-1970s for building solar power stations in Earth orbit, and anti-satellite warfare using systems tested by the USSR, the U.S., and China over the past 30 years. Such aggressive activities could set up a situation where a single satellite failure could lead to cascading failures of many satellites in a period of time much shorter than years.

Growth of Debris.

Faced with this scenario, as early as the 1980s NASA and other groups within the U.S. attempted to limit the growth of debris. One particularly effective solution was implemented by McDonnell Douglas on the Delta booster, by having the booster move away from their payload and then venting any remaining propellant in the tanks. This eliminated the pressure build-up in the tanks that had caused them to explode in the past.[31] Other countries, however, were not as quick to adopt this sort of measure, and the problem continued to grow throughout the 1980s, especially due to a large number of launches in the Soviet Union.

A new battery of studies followed as NASA, NORAD and others attempted to better understand exactly what the environment was like. Every one of these studies adjusted the number of pieces of debris in this critical mass zone upward. In 1981 when Schefter's article was published it was placed at 5,000 objects, but a new battery of detectors in the Ground-based Electro-Optical Deep Space Surveillance system quickly found new objects within its resolution. By the late 1990s it was thought that the majority of 28,000 launched objects had already decayed and about 8,500 remained in orbit. By 2005 this had been adjusted upward to 13,000 objects, and a 2006 study raised this to 19,000 as a result of an ASAT test and a satellite collision. In 2011, NASA said 22,000 different objects were being tracked.

The growth in object count as a result of these new studies has led to intense debate within the space community on the nature of the problem and earlier dire warnings. Following Kessler's 1991 derivation, and updates from 2001, the LEO environment within the 1,000 km (620 mi) altitude range should now be within the cascading region. However, only one major incident has occurred: the 2009 satellite collision between Iridium 33 and Cosmos 2251. The lack of any obvious cascading in the short term has led to a number of complaints that the original estimates overestimated the issue. Kessler has pointed out that the start of a cascade would not be obvious until the situation was well advanced, which might take years.



Growth of Space Debris.

China produced the space debris — 40 %; USA — 27, 5 %; Russia — 25, 5 %; the rest country — 7 %. (2009).

A 2006 NASA model suggested that even if no new launches took place, the environment would continue to contain the then-known population until about 2055, at which point it would increase on its own. Richard Crowther of Britain's Defense Evaluation and Research Agency stated that he believes the cascade will begin around 2015. The National Academy of Sciences, summarizing the view among professionals, noted that there was widespread agreement that two bands of LEO space, 900 to 1,000 km (620 mi) and 1,500 km (930 mi) altitudes, were already past the critical density.

In the 2009 European Air and Space Conference, University of Southampton, UK researcher, Hugh Lewis predicted that the threat from space debris would rise 50 percent in the coming decade and quadruple in the next 50 years. Currently more than 13,000 close calls are tracked weekly.

A report in 2011 by the National Research Council in the USA warned NASA that the amount of space debris orbiting the Earth was at critical level. Some computer models revealed that the amount of space debris "has reached a tipping point, with enough currently in orbit to continually collide and create even more debris, raising the risk of spacecraft failures". The report has called for international regulations to limit debris and research into disposing of the debris.

The great majority of debris consists of smaller objects, 1 cm (0.39 in) or less. The mid-2009 update to the NASA debris FAQ places the number of large debris items over 10 cm (3.9 in) at 19,000, between 1 and 10 centimeters (3.9 in) approximately 500,000, and that debris items smaller than 1 cm (0.39 in) exceeds tens of millions. In terms of mass, the vast majority of the overall weight of the debris is concentrated in larger objects, using numbers from 2000, about 1,500 objects weighing more than 100 kg (220 lb.) each account for over 98% of the 1,900 tons of debris then known in low earth orbit.

Using the figure of 8,500 known debris items from 2008, the total mass is estimated at 5,500 tons (12,100,000 lb.).

Debris in Low Earth Orbits.

At the most commonly used low earth orbits for manned missions, 400 km (250 mi) and below, residual air drag helps keep the zones clear. Collisions that occur under this altitude are less of an issue, since they result in fragment orbits having perigee at or below this altitude. The critical altitude also changes as a result of the space weather environment, which causes the upper atmosphere to expand and contract. An expansion of the atmosphere leads to an increased drag to the fragments, resulting in a shorter orbit lifetime. An expanded atmosphere for some period of time in the 1990s is one reason the orbital debris density remained lower for some time. Another was the rapid reduction in launches by Russia, which conducted the vast majority of launches during the 1970s and 80s.

Space Debris at higher orbits.

At higher altitudes, where atmospheric drag is less significant, orbital decay takes much longer. Slight atmospheric drag, lunar perturbations, and solar radiation pressure can gradually bring debris down to lower altitudes where it decays, but at very high altitudes this can take millennia. Thus while these orbits are generally less used than LEO, and the problem onset is slower as a result, the numbers progress toward the critical threshold much more quickly.

The issue is especially problematic in the valuable geostationary orbits (GEO), where satellites are often clustered over their primary ground "targets" and share the same orbital path. Orbital perturbations are significant in GEO, causing longitude drift of the spacecraft, and a precession of the orbit plane if no maneuvers are performed. Active satellites maintain their station via thrusters, but if they become inoperable they become a collision concern (as in the case of Telstar 401). There has been estimated to be one close (within 50 meters) approach per year.

Sources of space debris.

In a catalog listing known launches up to July 2009, the Union of Concerned Scientists listed 902 operational satellites. This is out of a known population of 19,000 large objects and about 30,000 objects ever launched. Thus, operational satellites represent a small minority of the population of man-made objects in space. The rest are, by definition, debris.

One particular series of satellites presents an additional concern. During the 1970s and 80s the Soviet Union launched a number of naval surveillance satellites as part of their RORSAT (Radar Ocean Reconnaissance SATellite) program. These satellites were equipped with a BES-5 nuclear reactor in order to provide enough energy to operate their radar systems. The satellites were normally boosted into a medium altitude graveyard orbit, but there were several failures that resulted in radioactive material reaching the ground (see Kosmos 954 and Kosmos 1402). Even those satellites successfully disposed of now face a debris issue of their own, with a calculated probability of 8% that one will be punctured and release its coolant over any 50-year period. The coolant self-forms into droplets up to around some centimeters in size and these represent a significant debris source of their own.

Tracking debris from the ground.

Radar and optical detectors such as lieder are the main tools used for tracking space debris. However, determining orbits to allow reliable re-acquisition is problematic. Tracking objects smaller than 10 cm (4 in) is difficult due to their small cross-section and reduced orbital stability, though debris as small as 1 cm (0.4 in) can be tracked. NASA Orbital Debris Observatory tracked space debris using a 3 m (10 ft.) liquid mirror transit telescope.

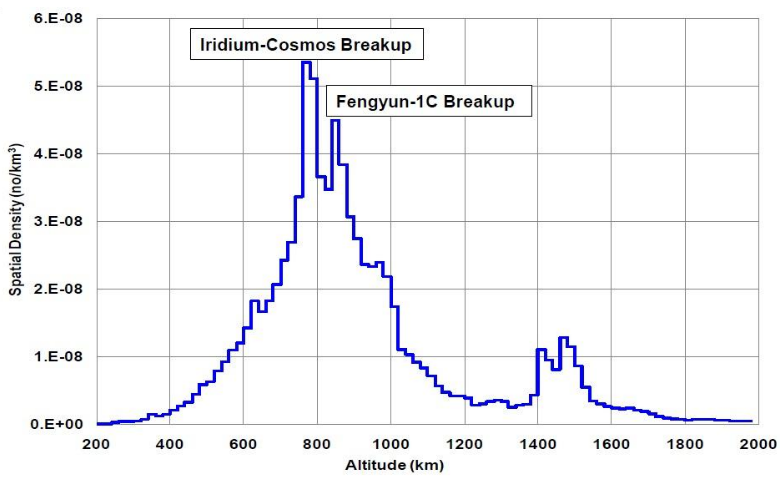
The U.S. Strategic Command maintains a catalogue containing known orbital objects. The list was initially compiled in part to prevent misinterpretation as hostile missiles. The version compiled in 2009 listed about 19,000 objects. Observation data gathered by a number of ground-based radar facilities and telescopes as well as by a space-based telescope is used to maintain this catalogue.

Nevertheless, the majority of expected debris objects remain unobserved – there are more than 600,000 objects larger than 1 cm (0.4 in) in orbit (according to the ESA Meteoroid and Space Debris Terrestrial Environment Reference, the MASTER-2005 model).

Other sources of knowledge on the actual space debris environment include measurement campaigns by the ESA Space Debris Telescope, TIRA (System), Goldstone radar, Haystack radar, the EISCAT radars, and the Cobra Dane phased array radar. The data gathered during these campaigns is used to validate models of the debris environment like ESA-MASTER. Such models are the only means of assessing the impact risk caused by space debris, as only larger objects can be regularly tracked.

Dealing with space debris.

Manmade space debris has been dropping out of orbit at an average rate of about one object per day for the past 50 years. Substantial variation in the average rate occurs as a result of the 11-year solar activity cycle, averaging closer to three objects per day at solar max due to the heating, and resultant expansion, of the Earth's atmosphere. At solar min, five and one-half years later, the rate averages about one every three days.



Spatial density of LEO space debris by altitude according to NASA report to UNOOSA of 2011.

Self-removal debris.

It is an ITU requirement that geostationary satellites be able to remove themselves to a graveyard orbit at the end of their lives. It has been demonstrated that the selected orbital areas do not sufficiently protect GEO lanes from debris, although a response has not yet been formulated.

Firm EADS Astrium is Major European space. Astrium has been developing an aerobraking sail concept for the forthcoming French Microscope satellite. Microscope is a science mission that will investigate the force of gravity and the behavior of free-falling objects in a test of what has become known as the equivalence principle.

Rocket stages or satellites that retain enough propellant can power themselves into a decaying orbit. In cases when a direct (and controlled) de-orbit would require too much propellant, a satellite can be brought to an orbit where atmospheric drag would cause it to de-orbit after some years. Such a maneuver was successfully performed with the French Spot-1 satellite, bringing its time to atmospheric re-entry down from a projected 200 years to about 15 years by lowering its perigee from 830 km (516 mi) to about 550 km (342 mi).

Several passive means of increasing the orbital decay rate of spacecraft debris have been proposed. Rather than using rockets, an electrodynamic tether could be attached to the spacecraft on launch. At the end of the spacecraft's lifetime, the tether would be rolled out to slow down the spacecraft.[140]

Although tethers of up to 30 km have been successfully deployed in orbit the technology has not yet reached maturity.[41] Other proposals include booster stages with a sail-like attachment[141] or a very-large but ultra-thin inflatable balloon envelope[142] to accomplish the same end.

External removal debris.

A well-studied solution is to use a remotely controlled vehicle to rendezvous with debris, capture it, and return to a central station. The commercially developed MDA Space Infrastructure Servicing vehicle is a refueling depot and service spacecraft for communication satellites in geosynchronous orbit, slated for launch in 2015. The SIS includes the vehicle capability to "push dead satellites into graveyard orbits." The Advanced Common Evolved Stage family of upper-stages is being explicitly designed to have the potential for high leftover propellant margins so that derelict capture/deorbit might be accomplished, as well as with in-space refueling capability that could provide the high delta-V required to deorbit even heavy objects from geosynchronous orbits.

The laser broom uses a powerful ground-based laser to ablate the front surface off of debris and thereby produce a rocket-like thrust that slows the object. With a continued application the debris will eventually decrease their altitude enough to become subject to atmospheric drag. In the late 1990s, US Air Force worked on a ground-based laser broom design under the name "Project Orion". Although a test-bed device was scheduled to launch on a 2003 Space Shuttle, numerous international agreements, forbidding the testing of powerful lasers in orbit, caused the program to be limited to using the laser as a measurement device. In the end, the Space Shuttle Columbia disaster led to the project being postponed and, as Nicholas Johnson, Chief Scientist and Program Manager for NASA's Orbital Debris Program Office, later noted, "There are lots of little gotchas in the Orion final report. There's a reason why it's been sitting on the shelf for more than a decade."

Additionally, the momentum of the photons in the laser beam could be used to impart thrust on the debris directly. Although this thrust would be tiny, it may be enough to move small debris into new orbits that do not intersect those of working satellites. NASA research from 2011 indicates that firing a laser beam at a piece of space junk could impart an impulse of 0.04 in (1.0 mm) per second. Keeping the laser on the debris for a few hours per day could alter its course by 650 ft. (200 m) per day. One of the drawbacks to these methods is the potential for material degradation. The impinging energy may break apart the debris, adding to the problem. A similar proposal replaces the laser with a beam of ions.

A number of other proposals use more novel solutions to the problem, from foamy ball of aerogel or spray of water, inflatable balloons, electrodynamics tethers, boom electroadhesion, or dedicated "interceptor satellites". On 7 January 2010, Star Inc. announced that it had won a contract from Navy/SPAWAR for a feasibility study of the application of the ElectroDynamic Debris Eliminator (EDDE). In February 2012, the Swiss Space Center at École Polytechnique Fédérale de Lausanne announced the Clean Space One project, a nanosat demonstration project for matching orbits with a defunct Swiss nanosat, capturing it, and deorbiting together.

As of 2006, the cost of launching any of these solutions is about the same as launching any spacecraft. Johnson stated that none of the existing solutions are currently cost-effective. Since that statement was made, a promising new approach has emerged. Space Sweeper with Sling-Sat (4S) is a grappling satellite mission that sequentially captures and ejects debris. The momentum from these interactions is used as a free impulse to the craft while transferring between targets. Thus far, 4S has proven to be a promising solution.

A consensus of speakers at a meeting held in Brussels on 30 October 2012, organized by the Secure World Foundation, a US think tank, and the French International Relations Institute, report that active removal of the most massive pieces of debris will be required to prevent the risks to spacecraft, crewed or not, becoming unacceptable in the foreseeable future, even without any further additions to the current inventory of dead spacecraft in LEO. However removal cost, together with legal questions surrounding the ownership rights and legal authority to remove even defunct satellites have stymied decisive national or international action to date, and as yet no firm plans exist for action to address the problem. Current space law retains ownership of all satellites with their original operators, even debris or spacecraft which are defunct or threaten currently active missions.

Description of Removing the Debris and Innovations

Air Braking. Removing Space Debris (SD) from Space is very expensive. It is more expensive than producing and launching new satellites. It requires designing a special Debris Apparatus (DA), and launching it. The craft must have enough fuel for flight, braking and connecting to the specific piece of space debris, braking (DA and SD), to deliver them (together DA and SD) to Earth atmosphere and, accelerate DA, to fly in the next debris and repeating all maneuvers (flight, braking, connection, impulse of braking, acceleration and so on) to the next piece of debris.

Author offers the new economical AB method of removing the SD (Brake-Reflector). Braking the SD by the special light parachute-reflector (mirror, space sail) which uses the space gas of an upper atmosphere and solar pressure. This method also may be used by the interplanetary space apparatus (SA) for acceleration, braking and landing of SA on planets.

The problem of cleaning up Space is shown in Fig.1.

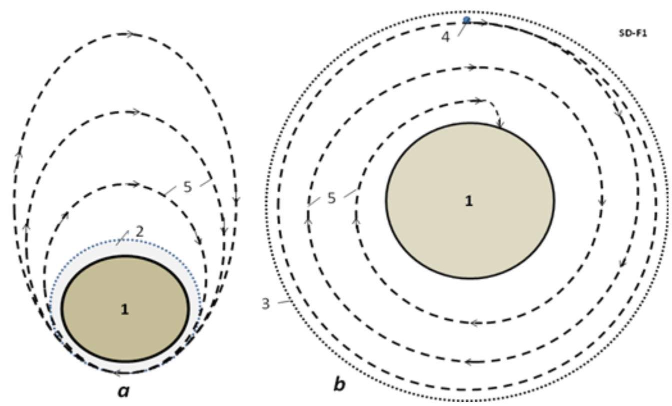


Fig.1. The Problem of returning Space Debris to the Earth. (a)- Trajectories of SD having a high apology (> 500 km). (b)- Spiral trajectory of SD having the initial circle trajectory in the Earth atmosphere (< 100 km). *Notations*: 1 – Earth, 2 – Earth atmosphere, 3 – boundary of the Earth atmosphere, 4 – SD or SA, 5 – trajectory of SD or SA.

The air brake works in the following way. Apparatus/debris has the ellipse trajectory. The Earth is located in focus of ellipse. The minimal altitude is named - perigee, the maximal altitude is named – apogee (Fig.1a). When apparatus is in the Earth's atmosphere, the air drag brakes the craft. As a result, the apogee decreases while the trajectory becomes closer to a circle such that the craft is fully in the Earth atmosphere. Here the air drag makes the trajectory in the form of a spiral. Small debris/apparatus burns in atmosphere. The large SD/SA or having the special control parachute can land the SA to Earth surface.

This method may be used if the altitude of perigee is less 350 km. If the altitude is more 350 - 450 km the SD/SA lifetime is some (tens) years (see Computation section). That may be not acceptable for humanity.

Solar braking/acceleration. In this method the SD/SA connects with a special thin film solar sail or parachute that has surfaces having a different color. For example, black-white or black-mirror. As a result of the solar pressure (solar radiation) on the sail (parachute) in left and right sides of circle orbit will be different (Fig.2) ($F_2 > F_1$). If the braking is more than acceleration, the SD/SA will decrease the perigee (Fig. 2a). If the braking is less than acceleration. The SD/SA will increase the apogee (Fig. 2b).

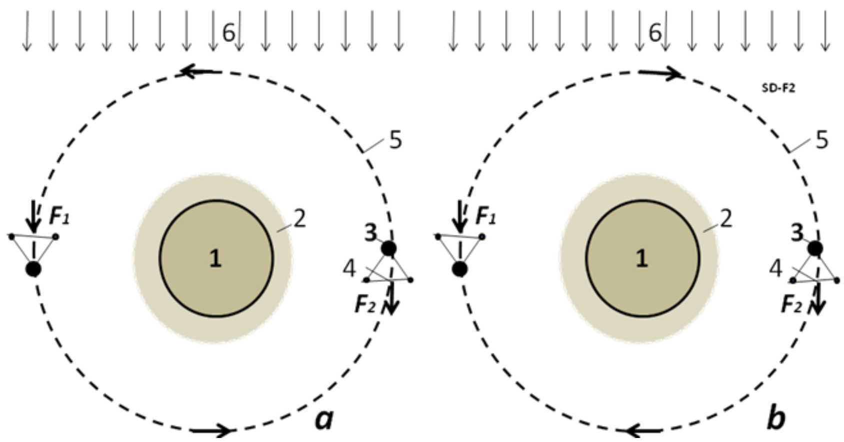


Fig.2. Braking/acceleration of the Space Debris by the parachute-reflector. (a) – braking, $F_2 > F_1$; (b) – acceleration, $F_2 > F_1$ (again direction of apparatus moving). *Notations:* 1 – Earth, 2 – Earth atmosphere, 3 – SD or SA, 4 – parachute-reflector, 5 – trajectory of the SD/SA, 6 is solar radiation, F_1 and F_2 are the light force (pressure).

Work of the offered Apparatus.

The work of the offered AB Brake-Reflector is shown in fig.3. The Debris Apparatus 1 (DA) fly up to space debris (SD) 2 and shutting by a small rocket projectile 3 (fig.3a). (That may be connected by thin cable to Debris Apparatus 1). The projectile opens a net 4 (fig.3b). The net 4 catches the space debris 2 (fig. 3c) and releases, unwind (reel off) a brake cable 5 (fig.3d). As the result the SD pass its impulse to the projectile 3 or SA 1. The SD is braked, the SA (if one connected to SD) gets the acceleration. After this the projectile disconnected from SA 1 (if it was connected to SA 1), open the Brake-Reflector 6 and brakes the space debris 3 by the Brake-Reflector 6 in the top atmosphere (fig.3e). The Space Apparatus fly to next SD for delivery of an important satellite to Space Station for repair.

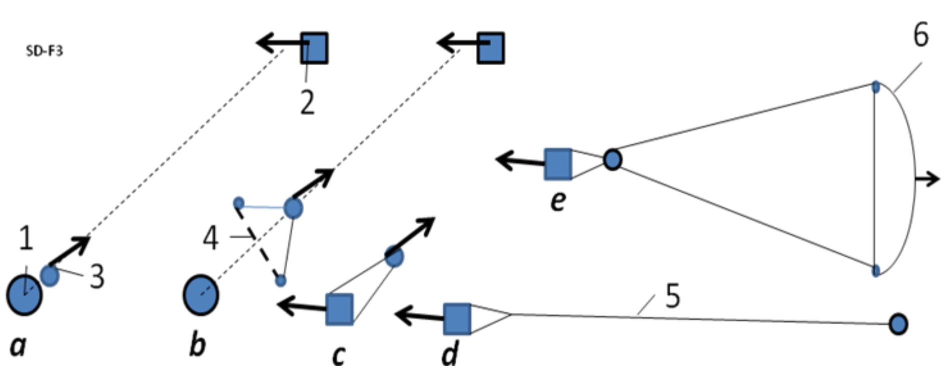


Fig. 3. AB Method. Process of catching and braking SD: (*a*) – shot of SA 1 by special projectile 3; (*b*) – open the special net 4 for catching the debris 2; (*c*) – space debris after its catching; (d) – braking SD by projectile which is used the cable 5; (e) – the (air/solar light) braking by parachute/reflector 6.
Notations: 1 – Space apparatus (SA); 2 – Space debris (SD), 3 – reactive gun, 4 – catch net, 5 – brake cable, 6 – control parachute, reflector, mirror, brake.

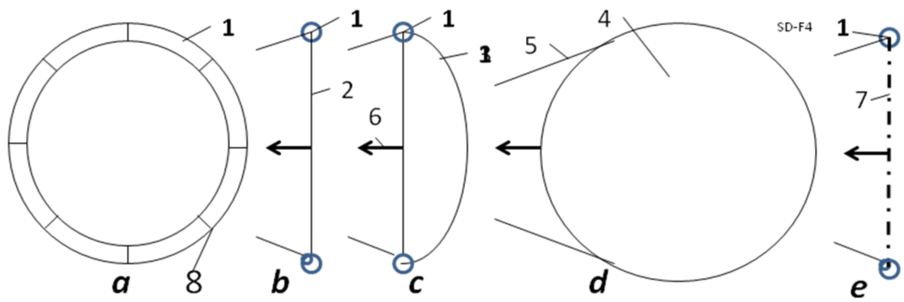


Fig.4. Some possible forms of the offered Drag-Reflect Space AB Brake (Parachute, Sail).
Notations: (***a***) – forward view of drag-Reflector, (***b, c***)- side view of drag-reflector and parachute, (***d***) – spherical drag-reflector, (***e***) – net (grid) for catching the space debris; 1 – inflatable ring (toroid), 2 – thin film (or solar sail), 3 – parachute, 4 – inflatable thin film ball, 5 – connection cables; 6 – direction of moving, 7 – light thin net (grid), 8 – partition into toroid.

Design of offered apparatus

One possible design of the AB Space Apparatus (Space Cleaner) is shown in Fig.5.

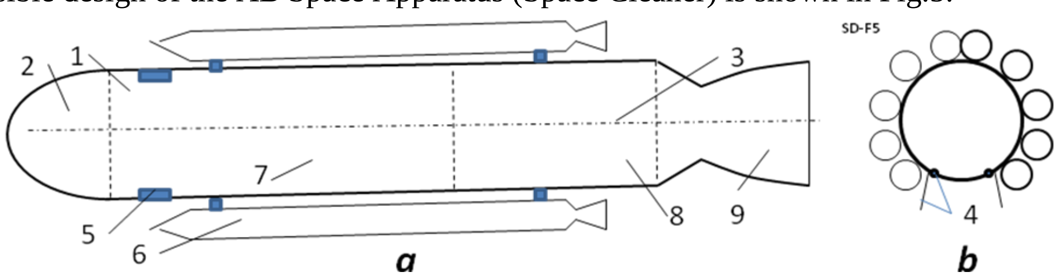


Fig.5. Possible design of the offered AB Space Apparatus (AB Space Cleaner). *Notations:* (***a***) – side view, (***b***) – forward view; 1 – offer AB Space Cleaner; 2 – head section contains: locator, TV and radio translator, radio receiver, computer, control and so on; 3 – rocket engine section; 4 – doors and artificial arms for catching the space debris; 5 – maneuver small rocket engines; 6 - projectiles for catching the big objects or pieces of the space debris (for example: satellites, last rocket stagy); 7 – storage for the small pieces of the space debris; 8 - fuel for main rocket engine; 9 – rocket nozzle.

The projectile for catching the large space objects is shown in fig.6.

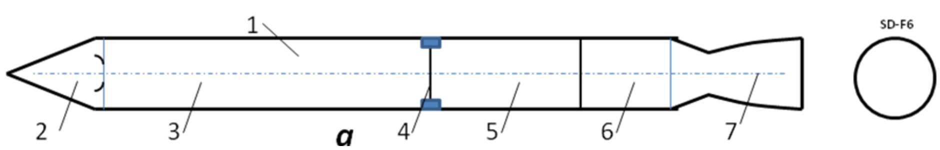


Fig.6. Projectile for catching and braking or delivery the satellite for repair to space station. *Notations*: (***a***) – side view; (***b***) – the forward view; 1 – projectile body; 2 – head section contains: locator, TV and radio translator, radio receiver, computer, control and so on; 3 – brake parachute or solar sail; 4 - maneuver small rocket engines; 5 – net section; 6 – solid fuel section of rocket engine; 7 – rocket engine.

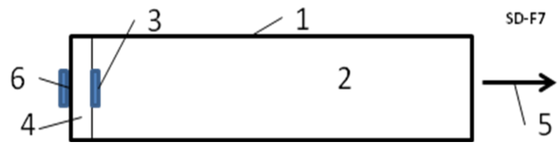


Fig.7. Cartridge of parachute for quick landing the space apparatus. *Notations:* 1 – body, 2 – brake parachute, 3 – air balloon for inflatable ring or ball, 4 – knockout charge, 5 – direction of parachute moving, 6 – fuse.

Differences and Innovations in AB Method and Apparatus.

The offered Brake-Reflector has numerous differences in method and installation from the ordinary method and usual collector of the space debris (see Project "Babushka" of Jozef Resnick and Project Vaughan Ling (fig.8):

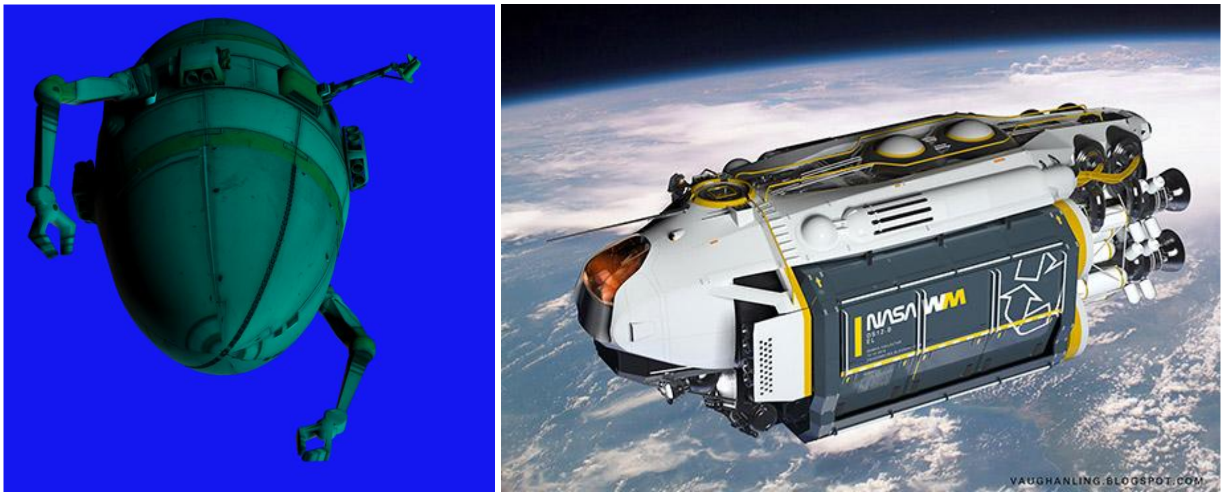


Fig.8. Project "Babushka" of Joseph Resnick (left) and Vaughan Ling (right) for collection the space debris.

Conventional Method:

In usual methods, the Apparatus has remote control, radio locator, computer, rockets, arms, storage for space debris (Fig.8.). They fly up to SD, brakes its speed to equal the SD, complex maneuvers for getting the suitable position, opens the door of the storage, catches the SD (for example the old satellite), puts into storage, closes the door, turns on the rocket engine and they fly to the Earth atmosphere (to dump the SD for burning in atmosphere) or to Space station for repair of satellite.

Offered AB Method:

In offered method the Apparatus has remote control, radio locator, computer, small rocket engine. It does not have a big storage for space debris. One can have near the SD a high speed. Apparatus shoots in SD from a guide rails or a special gun by the small special rocket (projectile) having the net and the brake-reflect parachute. The net catches the SD. The special cable uncoils and breaks the SD or passes the SD impulse to

SA and accelerate SA. The small special rocket disconnected from SA, opens the brake-reflector parachute and sends the SD into the Earth atmosphere. Space apparatus can save connection to satellite and delivery it to the Space Station or Space Ship.

No operations: braking at near SD, complex maneuvers of SA at near SD. Arms for catching SD, putting SD into storage, delivery SD to the Earth atmosphere (spending fuel). Further acceleration (spending fuel) for fly back of SA to space.

Advantages of the offered AB apparatus (AB Collector of space debris).
1. Less size and weight in 2 -3 times than conventional SD Collector (Not need in big storage for SD).
2. More efficiency in 2 -3 times (it can collect more SD).
3. Need less of fuel in some times (No maneuvers at SD, it can get the impulse from SD, SA can be far from SD and has a high speed).
4. No limits in size for SD. (All old and new space satellites have the solar panels and can NOT be eliminated by other space apparatus, fig. 9).
5. May easily protect selected space ship and station (for example, International Space Station) from SD.
6. The brake-reflector parachute makes SD easy for view and detection by locator).

Fig.9. Typical satellite has solar panels for getting energy.

Theory of AB Space Collector.

Escape velocity. The total specific orbital energy

$$\frac{v^2}{2} - \frac{GM}{r} \qquad (1)$$

does not depend on the distance, r from the center of the central body to the space vehicle in question. Therefore, the object can reach infinite r only if this quantity is nonnegative, which implies

$$v \geq \sqrt{\frac{2GM}{r}}, \qquad (2)$$

The escape velocity from the Earth's surface is about 11.2 km/s, but that is insufficient to send the body an infinite distance because of the gravitational pull of the Sun.

Formulae for free orbits. Orbits are conic sections, so, naturally, the formulas for the distance of a body for a given angle corresponds to the formula for that curve in polar coordinates, which is:

$$\mu \approx GM, \quad p = \frac{h^2}{\mu}, \quad r = \frac{p}{1 + e \cdot \cos\theta}, \tag{3}$$

where $\mu = 4\times10^{14}$ m^3/s^2 is called the gravitational parameter of the Earth, m^3/s^2; G = 6,674×10^{-11} m^3/kg·s^2 is the gravitational constant, m^3/kg· s^2; M the mass (kg) of objects 1 (planet, Earth, Sun, etc.); and h is the specific angular momentum of object 2 (debris, satellite, space ship, etc.) with respect to object 1. The parameter θ is known as the true anomaly, p is the semi-latus rectum, while e is the orbital eccentricity, all obtainable from the various forms of the six independent orbital elements.

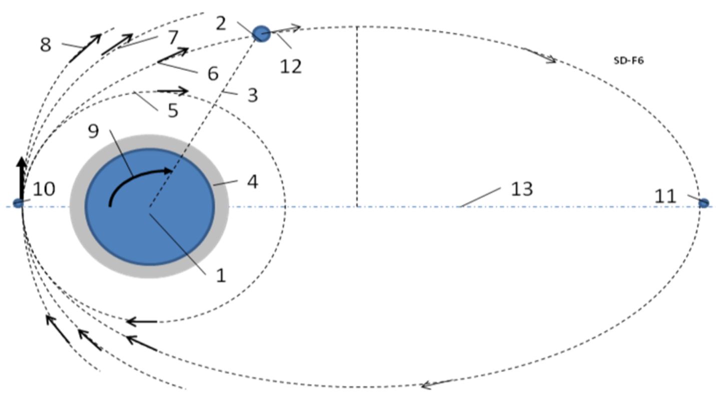


Fig.10. Trajectories of space object (satellite, space ship, space debris, so on) from value of the rocket impulse in perigee (point 10). *Notations*: 1 – planet or star (Earth, Sun); 2 – space object; 3 – radius-vector; 4 – atmosphere of Earth; 5 – circular trajectory; 6 – elliptic trajectory; 7 – parabolic trajectory; 8 – hyperbolic trajectory; 9 - angle θ; 10 – perigee; 11 – apology; 12 – speed of space object; 13 – semi-major axis "*a*".

Circular orbits. All bounded orbits where the gravity of a central body dominates are elliptical in nature. A special case of this is the circular orbit, which is an ellipse of zero eccentricity. The formula for the velocity of a body in a circular orbit at distance *r* from the center of gravity of mass *M* is

$$v = \sqrt{\frac{GM}{r}}, \tag{4}$$

where G = 6.673 84 × 10^{-11} m^3/(kg·s^2) is the gravitational constant.

To properly use this formula, the units must be consistent; for example, *M* must be in kilograms, and *r* must be in meters. The answer will be in meters per second.

The quantity *GM* is often termed the standard gravitational parameter, which has a different value for every planet or moon in the Solar System. Sun has $\mu = 1.3276\times10^{20}$ m^3/s^2.

Once the circular orbital velocity is known, the escape velocity is easily found by multiplying by the square root of 2:

$$v = \sqrt{2}\sqrt{\frac{GM}{r}} = \sqrt{\frac{2GM}{r}}\ . \qquad (5)$$

Elliptical orbits. If $0< e < 1$, then the denominator of the equation of free orbits varies with the true anomaly θ, but remains positive, never becoming zero. Therefore, the relative position vector remains bounded, having its smallest magnitude at *periapsis (perigee)* r_p which is given by:

$$r_p = \frac{p}{1+e}\ . \qquad (6)$$

The maximum value r is reached when $\theta = 180^o$. This point is called the *apoapsis (apology)*, and its radial coordinate, denoted r_a, is

$$r_a = \frac{p}{1-e}\ ,\ \ 2a = r_p + r_a\ . \qquad (7)$$

Let 2*a* be the distance measured along the apse line from periapsis *P* to apoapsis *A*, as illustrated in the equation below:

Substituting the equations above, we get:

$$a = \frac{p}{1-e^2}\ ,\ \ r = \frac{a(1-e^2)}{1+e\cos\theta}\ . \qquad (8)$$

Orbital period. Under standard assumptions the orbital period (T) of a body traveling along an elliptic orbit can be computed as:

$$T = 2\pi\sqrt{\frac{a^3}{\mu}}\ , \qquad (9)$$

where: μ is standard gravitational parameter, a is length of semi-major axis.

Velocity. Under standard assumptions the orbital speed (v) of a body traveling along an **elliptic orbit** can be computed from the Vis-viva equation as:

$$v = \sqrt{\mu\left(\frac{2}{r} - \frac{1}{a}\right)}, \qquad (10)$$

where: μ is the standard gravitational parameter, r is the distance between the orbiting, bodies, m; a is the length of the semi-major axis, m.

The velocity equation for a hyperbolic trajectory has either $1/(\pm a)$, or it is the same with the convention that in that case a is negative.

Energy. Under standard assumptions, specific orbital energy (ϵ) of elliptic orbit is negative and the orbital energy conservation equation (the Vis-viva equation) for this orbit can take the form:

$$\frac{v^2}{2} - \frac{\mu}{r} = -\frac{\mu}{2a} = \epsilon < 0 \ . \qquad (11)$$

Conclusions: For a given semi-major axis the specific orbital energy is independent of the eccentricity.

Parabolic orbits. If the eccentricity equals 1 (e = 1), then the orbit equation becomes:

$$r = \frac{h^2}{\mu}\frac{1}{1+\cos\theta}, \quad \epsilon = \frac{v^2}{2} - \frac{\mu}{r} = 0, \quad v = \sqrt{\frac{2\mu}{r}}, \qquad (12)$$

where: v is the speed of the orbiting body, m/s.

Hyperbolic orbits. If $e > 1$, the orbit formula,

$$r = \frac{h^2}{\mu}\frac{1}{1+e\cos\theta} \qquad (13)$$

describes the geometry of the hyperbolic orbit. The system consists of two symmetric curves. The orbiting body occupies one of them. The other one is its empty mathematical image. Clearly, the denominator of the equation above goes to zero when $\cos\theta = -1/e$. we denote this value of true anomaly $\theta_\infty = \cos^{-1}(-1/e)$ since the radial distance approaches infinity as the true anomaly approaches θ_∞. θ_∞ is known as the *true anomaly of the asymptote*. Observe that θ_∞ lies between 90° and 180°. From the trig identity $\sin^2\theta+\cos^2\theta = 1$ it follows that:

$$\sin\theta_\infty = (e^2-1)^{1/2}/e. \qquad (14)$$

Energy. Under standard assumptions, specific orbital energy (ϵ) of a hyperbolic trajectory is greater than zero and the orbital energy conservation equation for this kind of trajectory takes form:

$$\epsilon = \frac{v^2}{2} - \frac{\mu}{r} = \frac{\mu}{-2a}\,. \qquad (15)$$

Hyperbolic excess velocity. Under standard assumptions the body traveling along hyperbolic trajectory will attain in infinity an orbital velocity called hyperbolic excess velocity (v_∞) that can be computed as:

$$v_\infty = \sqrt{\frac{\mu}{-a}}\,, \qquad (16)$$

where: μ is standard gravitational parameter, a is the negative semi-major axis of orbit's hyperbola.

The hyperbolic excess velocity is related to the specific orbital energy or characteristic energy by

$$2\epsilon = C_3 = v_\infty^2\,. \qquad (17)$$

The rocket speed equals

$$\Delta V = -w \ln \frac{M}{M_0}\,, \qquad (18)$$

where ΔV is increasing the speed of rocket, m; w is the velocity of the combustion gas the rocket fuel (specific impulse) , m/s ; M_0 is initial rocket mass, kg; M is final rocket mass, kg.

For liquid rocket propellant has the specific $w \approx 3200 \div 3500$ m/s, for liquid hydrogen – liquid oxygen $w \approx 4630$ m/s, for the solid rocket booster $w \approx 2600 \div 3400$ m/s.

For multi-stagy rocket the final speed equals

$$\Delta V = \sum_{i=1}^{i=n} (\Delta V_i)\,, \quad \Delta V = -w_1 \ln \frac{M_1}{M_{0,1}} - w_2 \ln \frac{M_2}{M_{0,2}} - w_3 \ln \frac{M_3}{M_{0,1}} - \ldots - w_n \ln \frac{M_n}{M_{0,n}}\,, \qquad (19)$$

where i is number of rocket stagy.

If we computed the request ΔV, the need ratio $\Delta\overline{M} = M / M_0$ is

$$\Delta\overline{M} = \frac{M}{M_0} = e^{-\Delta V / w}. \quad \text{If} \quad \frac{\Delta V}{w} < 0.1 \quad \text{than} \quad \Delta\overline{M} \approx 1 - \frac{\Delta V}{w}\,. \qquad (20)$$

Computation of Removing Space Debris

1. Impulse Method

The simplest method to return the debris in Earth is installation in the future satellites and the last stage of rockets the rocket engine which will give the sufficient brake impulse for returning into the Earth atmosphere. If the space debris will not be used for further use, it will burn up as it enters the atmosphere. If it must be saved, it must be protected or must use the special AB parachute which the author detailed in [2].

If we have the speed V_1 in apology r_1 and want to reach the perigee in r_2 (Fig. 12, from point 3 to point 7) by optimal trajectory, we can get the speed V_2 in point 3:

$$V_1^2 - \frac{2\mu}{r_1} = V_2^2 - \frac{2\mu}{r_2}, \quad V_1 = V_2 \frac{r_2}{r_1}, \quad V_2 = \left(\frac{2\mu r_1}{r_2(r_1 - r_2)} \right)^{1/2}, \tag{21}$$

Hence the requested rocket speed is

$$\Delta V = V_2 - V_1 , \tag{22}$$

Unfortunately this method requires a lot of fuel and significantly decreases the useful load.

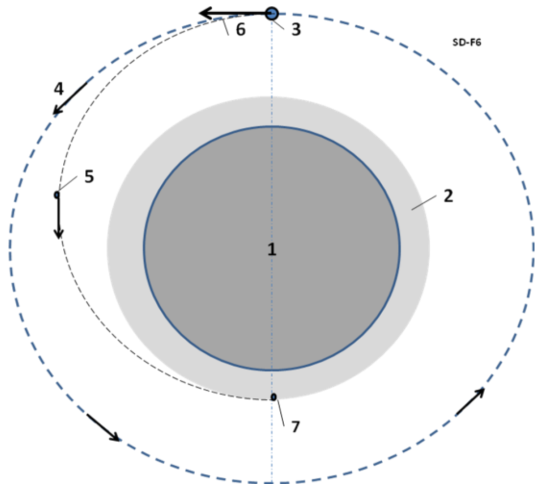


Fig. 11. Transfer space object from the circular high orbit to the low circular orbit or to Earth atmosphere. *Notations:* 1 – planet or star (Earth, Sun); 2 – atmosphere of Earth; 3 – space object; 3 – radius-vector; 4 – speed of initial orbit; 5 – speed of the transfer trajectory; 6 – rocket brake impulse.

2. Air brake parachute method

In this method we compute the lifetime of the space object (satellite, space ship, space debris, etc.) located in the top planet atmosphere and braking by the atmosphere drag.

For circulate orbit from Energy Law we have:

$$mgdH = DdL \tag{23}$$

where m is mass of the space object, kg; g is the planet (Earth) acceleration, m/s^2; H is altitude, m; D is atmosphere drag at space speed in rare atmosphere, N/m^2; L is path (way) of space object, m.

The magnitudes of (23) approximately equal:

$$D = C_D \rho(H) V^2 S , \quad \rho = a_1 e^{-(H-H_0)/b}, \quad q = \frac{m}{S}, \quad -\frac{dH}{e^{-(H-H_0)/b}} = \frac{C_D a_1 V^2}{gq} dL , \tag{24}$$

where ρ is the atmosphere density, N/m^2; V is object speed, m/s; $C_D \approx 1$ is drag coefficient; S is cross section area of object (perpendicular of flight), m^2; H_0 = 100 km = 10^5 m is the first base point of atmosphere (the 2-nd point is 300 km); $a_1 = 5.4\times10^{-7}$ kg/m^3 , b = 20640 m are the approximation Coefficients of the Earth atmosphere for base points H1 = 100 and H_2=300 km the Standard Earth Atmosphere; m is mass of space object, kg; q is mass load, kg/m^2.

Only one magnitude ρ (H) has strong depend from altitude. That way we suppose g = constant = average value in our diapason. The speed V equals the speed in perigee V_p.

Let us to integrate the last equation in (24) from H to H_0. We have:

$$L \approx \frac{bq}{C_D a_1 V_p^2}\left(e^{(H-H_0)/b} - 1\right), \quad t = \frac{L}{V}, \quad t \approx \frac{bq}{C_D a_1 V_p^3}\left(e^{(H-H_0)/b} - 1\right). \qquad (25)$$

Here t is lifetime (flight time) of the space object, s; H is perigee altitude, m.
If the orbit is ellipse, the lifetime of space object is

$$t \approx \frac{bq}{C_D a_1 V_p^3}\left(e^{(H-H_0)/b} - 1\right)\cdot\left(\frac{H_a}{H_p}\right) \quad or \quad t \approx \frac{bq}{C_D V_p^3}\left(\frac{H_a}{H_p}\right)\cdot\left(\frac{1}{\rho} - \frac{1}{\rho_0}\right) \quad , \qquad (26)$$

where ρ is the air density at the perigee altitude, kg/m^3; $\rho_0 = a_1 = 5.4\times10^{-7}$ kg/m^3 is the air density at H = 100 km = 10^5 m; H_a, H_p is apology and perigee of the initial orbit respectively.
The lifetime from equations (25), (26) may be recalculated from seconds in days or years:

$$1 \text{ day} = 8.64\times10^4 \text{ sec.}, \quad 1 \text{ year} = 3.145\times10^7 \text{ sec.} \qquad (27)$$

The second equation (26) is used if you have a standard top atmosphere or know the density of the top atmosphere.
In diapason of perigee altitude H = 200 ÷ 400 km we can use the more simple equations:

$$t \approx \frac{bq}{C_D a_1 V_p^3}\left(e^{(H-H_0)/b}\right)\cdot\left(\frac{H_a}{H_p}\right) \quad or \quad t \approx \frac{bq}{C_D V_p^3}\left(\frac{H_a}{H_p}\right)\cdot\left(\frac{1}{\rho}\right). \qquad (28)$$

Example. The first soviet satellite had the following data: m = 83.7 kg; d = 0.56 m; q = 340 kg/m^2; H_p = 288 km, H_a = 947 km. Let us substitute these data in equations (28), we receive the flight time 95 days. In reality one had 92 days. But we did not consider an air drag the 4 three-meter antennas.
If we want to delete this satellite as the space debris, we can connect it to a small parachute having diameter d = 5.6 m. The flight time decreases in 100 times and in during one day the satellite (debris) comes in the lower Earth atmosphere.

Braking by Solar Radiation

Over altitude 400 km the atmospheric pressure (density) is very small. At H = 400 ÷ 500 km it approximately equals the solar pressure $p = 4\times10^{-6}$ N/m^2. For changing the high orbit we can use only the solar radiation. For increasing the efficiency the space object must have a big light film surface (fig. 2) having the different reflectivity (color) in the different sides (for example: black – white or black – mirror). In left side of space trajectory (relativity of solar radiation) this reflector is turned one side to Sun, in right side of space trajectory this reflector is turned to Sun the other side (fig.2). The solar pressure is different and we can increase or decrease the perigee or apology of the space object. We can decrease the perigee up H = 400 km and further to use the air drag.
The author researched and offers the following equations for computation the changing orbit for right control:

$$t \approx \frac{q\mu}{2c_2\delta pV}\cdot\left(\frac{1}{r_2} - \frac{1}{r_1}\right) \quad or \quad t \approx \frac{q\mu}{2c_2c_1\delta p}\ln\left(\frac{r_1}{r_2}\right), \qquad (29)$$

where $c_2 \approx 1/6$ is coefficient activity (part of used orbit); $\delta = 0 \div 1$ is coefficient reflectivity; p = 4×10^{-6} N/m^2 is radiation (solar) pressure at the Earth orbit; V is average orbit speed, m/s; r_1 is initial radius of orbit, m; r_2 is final radius of orbit, m; $c_1 = rV$ is orbit parameter (r is average radius); $\mu = 4\times10^{14}$ m^3/s^2 is called the gravitational parameter, m^3/s^2.
Example: Let us to estimate the time needed for decreasing the orbit perigee r_1 = 700 km to r_2 = 400 km by the reflector having q = 1 kg/m^2, δ = 0.5, p = 4×10^{-6} N/m^2**.** The average orbit speed is 7.15

km/s. Equation (29) gives $t \approx 16$ years.

Deformation of Orbit by Solar Pressure

Author research shows if the Sun permanently pressures one side orbit, the altitude of orbit is deformed. This deformation may be estimated by equation:

$$\Delta H = \frac{c_2 \delta p r \sqrt{\mu}}{g q a^{3/2}} \Delta t \quad , \qquad (30)$$

where a is the semi-major orbit initial axis, m; g is average gravitation, m/s^2; t in sec; H in m.

Collision of space objects

If space object connects (in one line) to other space object by an inelastic collision or a brake cable, the summary velocity the system is:

$$V_0 = \frac{m_1 V_1 + m_2 V_2}{m_1 + m_2} \quad . \qquad (31)$$

Here m are mass of collision objects, V is their velocity, m/s.

Conclusion

Author offers new methods and installations for cleaning the outer space from space debris and individual protection the important space ship and stations from big space debris (SD). Advantages of the offered method and apparatus are following: 1. Less size and weight in 2 -3 times than conventional SD Collector. 2. More efficiency by 2 -3 times. 3. Saves fuel by a factor of some times. 4. No limits in size for SD. 5. Can easy protect selected space ship and station (for example, International Space Station) from SD.

Author also offers to enter the International Agreement: all objects will launch into space must have the installation for Removing them from orbit after active lifetime.

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Acknowledgement

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(The reader may find some articles in this field at the web pages: Third List of publication: http://www.scribd.com/doc/173336338/Publications-by-Bolonkin-30-September-2013, http://viXra.org/abs/1310.0022. Some books and articles by A. Bolonkin published in (2007-2013) are on line in http://www.scribd.com, http://www.archive.org , http://arxiv.org (45), http://viXra.org, http://intellectualarchive.com (28), http://AIAA.org (41) and in Bolonkin's WEB http://Bolonkin.narod.ru.)

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Professor Bolonkin is the author of more than 200 scientific articles and books, and has 17 inventions to his credit. His most notable books include: Non-Rocket Space Launch and Flight (Elsevier, 2006) **http://www.scribd.com/doc/203941769/** ; New Concepts, Ideas, and Innovations in Aerospace, Technology and Human Life (NOVA, 2008); Human Immortality and Electronic Civilization (Lulu, 1994); Femtotechnologies and Revolutionary Projects. Lambert Academic Publishing, Germany, 2011. 538 p.;. **http://www.scribd.com/doc/75519828/** , Life and Science. Lambert, 2011, 205 pgs

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