Acta Astronautica 204 (2023) 840–853

Contents lists available at ScienceDirect

Acta Astronautica

journal homepage: www.elsevier.com/locate/actaastro

Space debris removal – Review of technologies and techniques. Flexible or virtual connection between space debris and service spacecraft

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ARTICLE INFO

Keywords:
Space debris
Service spacecraft
Graveyard orbit
Space debris removal

ABSTRACT

The term "space debris" refers to spacecraft not removed from orbit at the end of their service life, upper stages in the geostationary orbit region, as well as to the fragments of spacecraft and upper stages formed as a result of deliberate or accidental collision of spacecraft and upper stages with each other or with natural space debris.

The problem of removing space debris from outer space is a global problem. Many countries are realizing projects on space debris cataloging, various technical means are being studied for space debris removal into graveyard orbits with parameters agreed upon by the international community. Various countries conducting space exploration have adopted special standards and guidelines for preventing the space debris formation [1,2,3,4].

The report [5] emphasizes that contamination of the near-Earth space is growing steadily. The probability of spacecraft collisions with each other is significantly increasing. This can lead to the functioning spacecraft characteristics degradation and even to complete loss of performance.

The article presents a brief overview of the methods and techniques that can be used to space debris removal into disposal orbits using flexible or virtual connection between space debris and service spacecraft.

1. Introduction

Various countries conducting space exploration have adopted special standards and guidelines for preventing the space debris formation [1-4]. The report [5] emphasizes that contamination of the near-Earth space is growing steadily.

In 1979 Lubos Perek presented the report "Outer Space Activities versus Outer Space", which for the first time considered a set of measures to prevent the space debris (SD) formation, including the spacecraft (SC) removal from geostationary orbit (GEO) at the end of life [6].

Over the period of space activities, the SD amount in the near-Earth orbits has increased to the point where the SD mass in the Earth's upper atmosphere (at altitudes above 200 km) is 1% of the mass of the upper atmosphere itself [7–10].

The main SD sources in the near-Earth space (NES) are:

- Spontaneous and/or intentional SC destruction in orbit, which leads to the long-term NES pollution. 40% of the total SDs number are the result of the destruction of large SD in near-Earth orbits [11]. For example, on April 12, 2019, the destruction of the Centaurus upper stage was recorded with the formation of at least 20 fragments, which were recorded and traced [12];
- SD released intentionally during the operation of the upper stages of launch vehicles and SC.

SD fragments resulting from SD–SC or SD - SD collisions are expected to become a significant source of SD, since in recent years there has been a trend towards an increase in the frequency of such episodes.

High SD velocities around the Earth pose a severe threat to SC that are being launched into operating orbits and to SC that are already operating there, including astronauts. According to experts, the risk of a catastrophic collision between the Space Shuttle type spaceship and SD was roughly rated as 1 to 300. The same risk for the Hubble Space Telescope was rated as 1 to 185 due to the fact that the Telescope's operating orbit is more populated with SD. The ISS performs a collision avoidance maneuver when the probability of collision is 1–10,000 [13].

According to the Kessler syndrome, proposed by US scientist D.J. Kessler in 1978 [14], the SD number with poorly predictable masses and velocities increases exponentially over time. This, in turn, increases the cost of protection against SD for SC under development, and significantly increases the risk of the operating SC destruction. One of the

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https://doi.org/10.1016/j.actaastro.2022.09.027
Received 22 August 2022; Received in revised form 7 September 2022; Accepted 13 September 2022
Available online 24 September 2022

List of abbreviations

ADR – Active Debris Removal
AFRL – Air Force Research Laboratory
CHARGE – Cooperative High Altitude Rocket Gun Experiment
EADS – European Aeronautic Defence and Space Company
E.T. PACK – Electrodynamic Tether Technology for Passive Consumable-less Deorbit Kit
EDDE – ElectroDynamic Delivery Express/Electro-Dynamic Debris Eliminator
ESA – European Space Agency
ENVISAT – Environmental Satellite
FELDs – Flexible Electromagnetic Leash Docking system
GEO – Geostationary Orbit
GOLD – Gossamer Orbit Lowering Device
GTO – GEO Transfer Orbit
IKAROS – Interplanetary Kite-craft Accelerated by Radiation Of the Sun
ISIS – Innovative Solutions In Space
JAXA – Japan Aerospace Exploration Agency
LEGEND – LEO-to-GEO Environment Debris Model
LEO – Low Earth Orbit
LODR – Laser Orbital Debris Removal
NASA – National Aeronautics and Space Administration
NES – Near-Earth Space
OEDIPUS – Observations of Electric-Field Distribution in the Ionospheric Plasma
PACMAN – Position and Attitude Control with MAgnetic Navigation
PMD – Post Mission Disposal
ROGER – Robotic Geostationary orbit Restorer
SC – Spacecraft
SD – Space Debris
SDMR – Space Debris Micro Remover
SEDS – Small Expendable Deployer System
SSTL – Surrey Satellite Technology Ltd.
STAR – Space Tether Automatic Retrieval
STARS – Space Tethered Autonomous Robots Satellite
TED – Tethered Electromagnetic Docking
TPE – Tethered Payload Experiment
TSNR – Tethered Space Net Robot
TSS – Tethered Satellite System
UK – United Kingdom of Great Britain and Northern Ireland
US/USA – United States of America
USRA – Universities Space Research Association

consequences of this is that the SD proliferation will make space activities and SC use in certain orbital ranges impractical for Mankind for many generations [15].

SC fragments are expected to become a significant source of SD, since in recent years there has been a trend towards an increase in the frequency of such episodes. The first collisions between SC and SD were recorded in 1983, a grain of sand about 0.2 mm in diameter caused formation of depression of about 0.4–0.7 mm in diameter in the window of the Space Shuttle. The number of collision is steadily growing year by year [16].

In March 2006, the Express-AM11 SC collided in GEO with SD, which resulted in depressurization of the liquid line of the thermal control system. The SC received a significant dynamic impulse for rotation, lost its orientation in space and started uncontrolled motion. Its further use for the intended purpose turned out to be impossible [17]. The SC was transferred from the control point in GEO into the disposal orbit in April 2006 [18].

On June 17, 2017, a commercial telecommunication satellite AMC-9 (GE-12) (USA), launched by the Proton launch vehicle from the Baikonur launch site on June 6, 2003, collided with a SC in geostationary orbit (GEO), as a result of which the SC lost its operability. Its fragments are still observed in the plane of the SC orbit, with some deviations in parameters. And these are just a few examples.

The problem of man-made pollution of the NES is especially acute for the region of low orbits and the vicinity of GEO, since at the altitudes of up to 600 km several years are needed for SD to enter the atmosphere of Earth, while at the altitudes above 1000 km some centuries are needed for this [19].

Currently, about 40 SC, whose age has exceeded 20 years, are operating in GEO. The total number of geostationary SC, taking into account the disposal orbits, already exceeds a thousand. Among them, only 35.6% are functioning [19]. According to estimates, only about a half of geostationary SC that have reached the end of their active life are transferred to higher disposal orbits [20]. Only 6% of the catalogued orbit population related to the operational SC, while 28% can be attributed to the decommissioned SC, spent upper stages, and mission objects (launch adapters, etc.). The remainder of about 66% is originating from more than 200 in-orbit fragmentations, which have been recorded since 1961 [21].

In [22] it is emphasized that the immediate implementation of measures to prevent the formation of SD is necessary to preserve the space environment for future generations. Thus, development of systems for the SD collection and disposal is an urgent task.

In [5], it is indicated that it is necessary to take immediate measures to prevent the SD formation in order to preserve the space environment for future generations.

A significant part of the projects under development is aimed at searching for SD larger than 10 cm and their de-orbiting. The SD of smaller sizes is currently studied much worse, including due to the difficulties of observing such objects from the Earth's surface and from space [23].

There are a lot of articles about space debris determination, space debris evolution, capturing, and mitigation strategies, space debris collisions and impact were published during last years [24–33].

Application of active SD removal systems can significantly change the pollution of the NES with man-made SD. The LEGEND model can be used to assess the SD number in the NES (Fig. 1). Various publications provide estimates for NES contamination with SD, taking into account different scenarios of SD active removal (Fig. 2) [34–37].

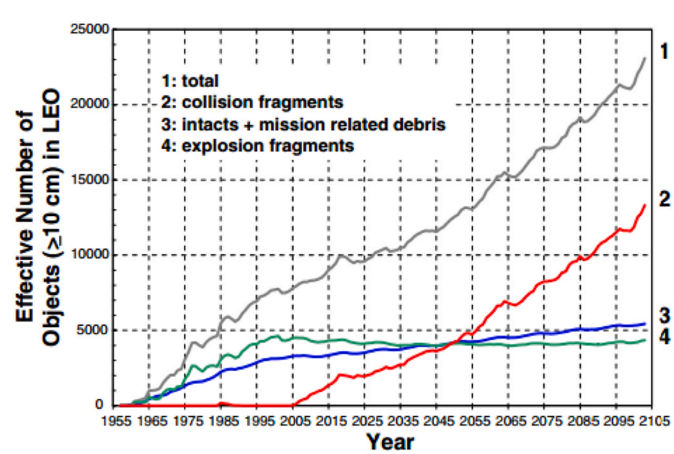

Fig. 1. Estimation of SD Number in LEO Using the LEGEND Model [35] (with kindly permission Elsevier and Copyright Clearance Center's, order number 5374631040119).

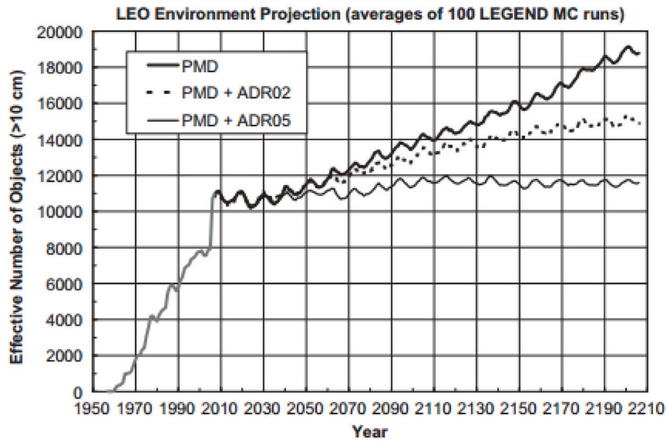

Fig. 2. Comparison of Three Different Scenarios (from top to bottom: Post-mission Disposal (PMD) only, PMD and ADR of Two Objects per Year, and PMD and ADR of Five Objects per Year) [36] (with kindly permission Elsevier and Copyright Clearance Center's, order number 5374631323130).

NASA's Beakup Standard Model is used to estimate the probability of SC destruction into fragments. This model describes a probabilistic model for the distribution of a number of fragments formed as a result of the SC destruction, velocity of SD fragments, as well as their area-to-mass ratio [38].

The formation of a large number of SC fragments can potentially cause significant damage to the functioning SC due to high velocities and, accordingly, high destruction energy. The given data testify to the relevance of the problem of active SD removal due to the seriousness of the threat to safe space activities [39–41]. Below, we will consider various examples of projects for SD removal from NES using a flexible connection between the service SC and SD, as well as via the formation of a virtual assembly between the service SC and SD. The practical implementation of such technologies will significantly reduce the risks for space flights in the future.

2. RemoveDEBRIS mission

Various systems and methods of SD transportation are tested in this project. Let us consider this mission.

The Surrey Space Centre of Surrey University (Guildford, UK) jointly with SSTL (Guildford, UK), Airbus (Leiden, Netherlands), Ariane Group (formerly Airbus Safran Launchers) (Issy-les-Moulineaux, France), Swiss Center for electronics and Microtechnology (Neuchâtel, Switzerland), Inria (Chesnay-Rocquencourt, France), Innovative Solutions In Space (Delft, Netherlands), and Stellenbosch University (Stellenbosch, Western Cape, South Africa) developed a SC to demonstrate various technologies for SD removal, including experiments with a net and a harpoon and testing navigation and guidance by video image.

The DebrisSat1 SC deploys a ball with a diameter of 1 m and a weight of 2 kg, designed to simulate SD, and the RemoveDEBRIS SC catches this ball with the 5-m net, maneuvers and tries to enter the Earth's atmosphere. The high-strength fiber net is deployed with concentric weights and a central cover, dragging the net. Motors and winches integrated within the weights are used to close the net after successful SD capture. The net deployment and closure will be achieved via redundant mechanisms. The net will be released to capture the SD at a distance of 6 m from the SC. Another DebrisSat2 SC maneuvers to check the image received by optical cameras and data provided by lidar installed on board the RemoveDEBRIS SC. A harpoon tied to the end of the rope will be released over the plate on the RemoveDEBRIS manipulator. After the completion of all experiments, the RemoveDEBRIS SC will deploy a large sail (Fig. 3) [42–44].

The RemoveDEBRIS SC (Fig. 4) is built on the basis of the SSTLX50 platform. The SC weight is 100 kg (40 kg payload), its dimensions are 65 × 65 × 72 cm. SC was launched by the Falcon 9 FT launch vehicle on April 2, 2018 to reach International Space Station. On June 20, 2018, the RemoveDEBRIS SC was launched into space from the International Space Station [42,43].

3. Solar sail

The solar sail is a promising SC concept allowing maneuvers without

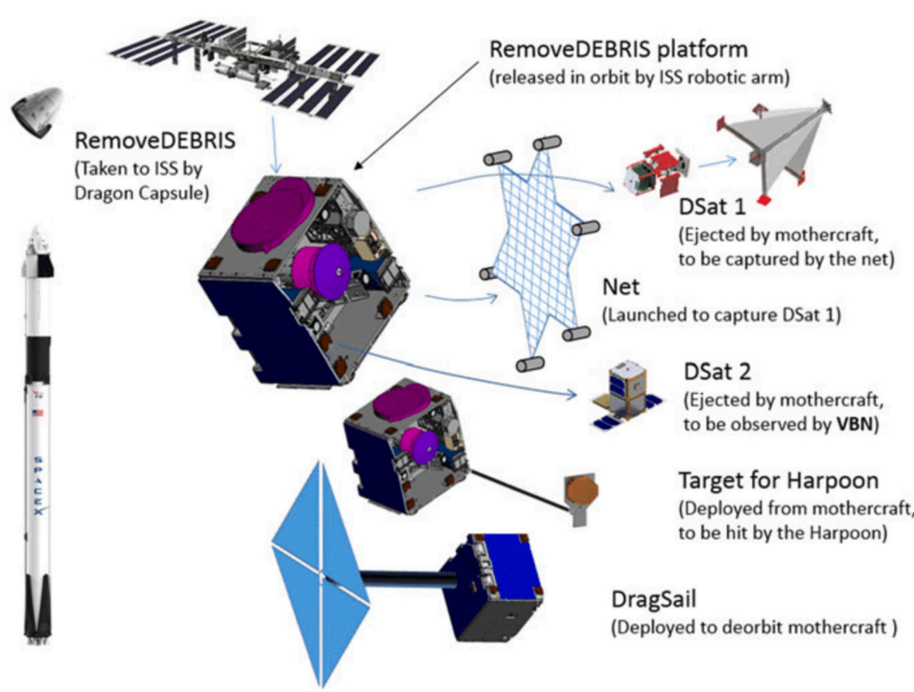

Fig. 3. RemoveDEBRIS Mission Scenario [45] (with kindly permission Cambridge University Press and Copyright Clearance Center's, order number 5374640313191).

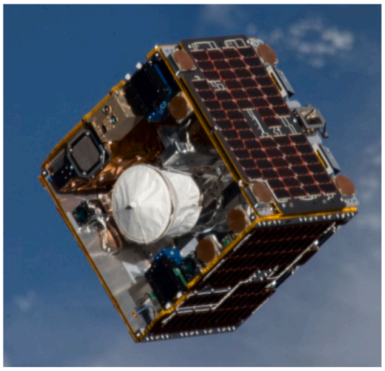

Fig. 4. RemoveDEBRIS SC [42] (public domain).

the propellant consumption.

An example of such a system is the solar sail project developed by ATK Space Systems of Goleta (Goleta, California, USA) and L'Garde Inc. (Tustin, California, USA). Ground tests of a 20-m sail were carried out in 2004. The sail is supported by a coilable booms. There are extended via remote control from a central stowage container, and is made of an aluminized, plastic-membrane material (CP-1). An another 20-m solar sail, developed by L'Garde Inc. (Tustin, California, USA) was fully deployed during testing at NASA Glenn Research Center's Plum Brook facility (Sandusky, Ohio, USA). The sail is supported by a inflatable booms that become rigid in the space. The system is deployed via remote control from a central stowage container [46,47].

Similar experiments are being carried out at the Cranfield University (Bedford, UK). Icarus-1 was the first drag sail as a demo payload for the TechDemoSat-1 mission, successfully tested in May 2019 (Fig. 5). Icarus-3 was a smaller, simplified version of Icarus-1 for SSTL's Carbonite-1 microsatellite mission. The sail was successfully deployed in November 2018 [48].

3.1. Interplanetary kite-craft accelerated by radiation of the sun

The first experimental SC weighing 315 kg (dimensions 1.6 × 1 m) with an IKAROS solar sail (Fig. 6) developed by JAXA (Chōfu, Tokyo, Japan) was launched into the departure trajectory to Venus as a secondary payload with the Akatsuki SC in May 2010. The 200 m^2 sail was successfully deployed on June 3, 2010. Successfully functioning thin-film solar arrays were additionally located on the sail. The SC motion control in terms of velocity and direction depending on the magnitude and direction of solar radiation was studied as part of the program [49,50].

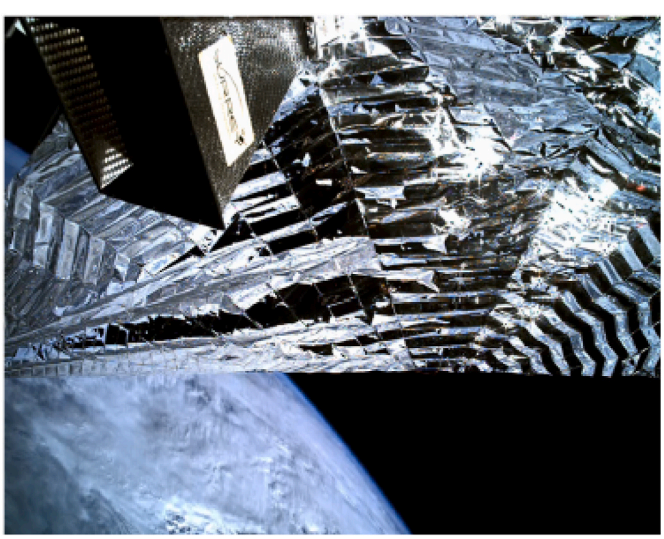

Fig. 5. Sail Deployment on TechDemoSat-1 SC [48] (with kindly permission Elsevier and Copyright Clearance Center's, order number 5374640781987).

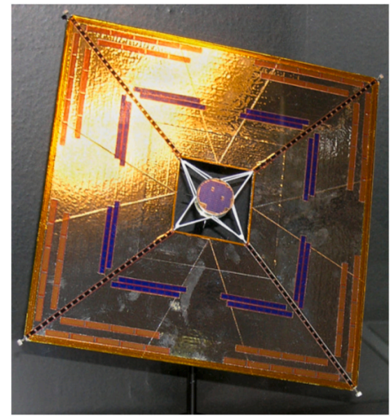

Fig. 6. IKAROS Project [51] (CC BY-SA 3.0 license).

3.2. NanoSail-2 nanosatellite

A 10 m^2 polymer sail was deployed in LEO from the NASA NanoSail-D2 nanosatellite (Fig. 7) on January 21, 2011 [52,53]. A 3U SC (CubSat) with a mass of 4 kg was developed by the NASA Marshall Space Flight Center (Huntsville, Alabama, USA) and NASA Ames Research Center (Moffett Field, California, USA).

3.3. LightSail-1/LightSail-2/LightSail-3

The LightSail-1 (Fig. 8) involved the development of a 3U SC within the frames of the project of The Planetary Society (Pasadena, California, USA). The goal of the project is to demonstrate solar sail technology in terms of deployment and controlled flight. It is planned to equip the SC with four triangular sails arranged in the shape of a rhombus with an area of 32 m^2. The orbit altitude for solar sail deployment is 800 km [54].

LightSail-1 SC has 10 solar panels assembled into 4 deployable arrays

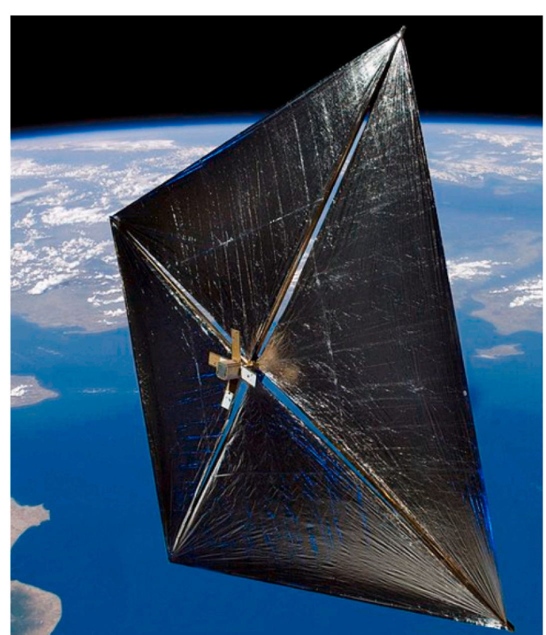

Fig. 7. NanoSail–D2Nanospacecraft [52] (public domain).

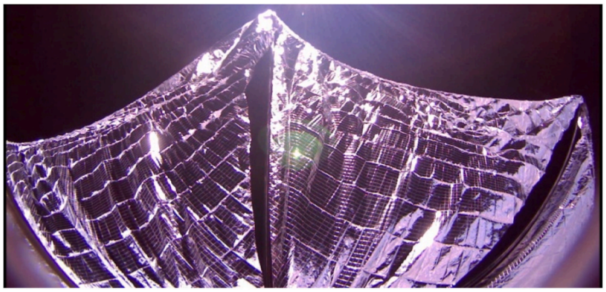

Fig. 8. LightSail-1 Concept [55] (CC BY-SA 3.0 license).

and 1 end panel; two megapixel cameras at the ends of two solar arrays, 4 solar sensors at the ends of the solar array wings, six ultra-sensitive accelerometers, three single-axis gyroscopes, reaction wheel, and a battery [54].

LightSail-2 SC (Fig. 9) developed by Stellar Exploration Inc. (San Luis Obispo, California, USA) is a 3U CubeSat. The platform is supplied by CalPoly (San Luis Obispo, California, USA). The SC weight is about 4.5 kg and its dimensions are 340 × 100 × 100 mm. The solar sail module is a SC payload. The SC is additionally equipped with video cameras, sensors and a control system to control the solar sail deployment [54].

It is intended to demonstrate the possibility of using a deployed solar sail as an early warning station for geomagnetic storms as part of the LightSail-3 project [54].

4. Inflatable or deployable structures

4.1. iDod project

ISIS (Delft, Netherlands) proposed a SC deorbiting concept based on a deployable structure (iDod), which is a thin membrane supported by spacers. The tightly packed structure unfolds under the action of gas pressure. The positive aspects of this structure are the absence of pyrotechnical devices, pressure vessels and valves, ease of integration with the SC, and minimal weight [56].

4.2. Gossamer orbit lowering device project

The GOLD project (Fig. 10) was considered by the Global Aerospace Corporation (Mississauga, Ontario, Canada) [57]. The project involved the deployment of a 100 m spherical structure. It was assumed that the structure would retain its performance even though it could be pierced by micrometeorites in several places. Gas leaks should be compensated by the pressurization system. This structure was supposed to be installed on board SC before their launch.

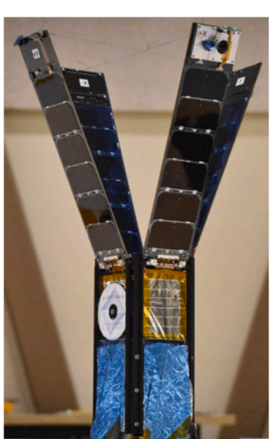

Fig. 9. LightSail-2 SC [55] (CC BY-SA 3.0 license).

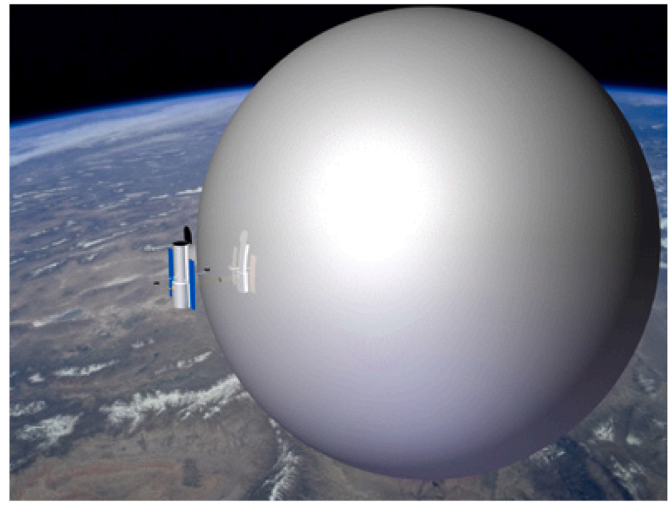

Fig. 10. GOLD Project [57] (with kindly permission Kelly T. Nock, Global Aerospace Corporation).

5. Foam application for space debris gripping

In [58], it was proposed to use expanding foam in the active space debris removal system. The foam provides increase in both the SC area-to-mass ratio and aerodynamic drag, and, accordingly, accelerates the SD entry into the Earth's atmosphere. The system does not require additional docking devices with the object to be removed; however, the entry into the atmosphere is uncontrolled. The system can be located both on a special service SC, and it can be built into the SD itself.

Polymeric foams were chosen as the main option because they are characterized by a fairly high coefficient of expansion, flexibility and relatively easy production. A model of foam behavior during expansion was built. The foam ejection nozzle is considered as the basic solution and its dimensions roughly correspond to the available static mixers.

Preliminary design of service SC with a launch weight of about 5 tons, equipped with an electric propulsion unit based on the PPS-5000 SPT (power: 5 kW; specific impulse: 3000 s; efficiency: above 50%; thrust: 200 mN) for interorbital flights from one SD to another, has been developed.

According to the space mission analysis, such systems can reentry up to 3 tons of SD per year.

6. Tether systems

A large number of different concepts of tether systems have been developed for carrying out transport operations without mass consumption (E.T. PACK [59], etc.). Some of them have passed the modeling stage [60–63] and ground experimental testing of the main units and assemblies [64]. Several space experiments were carried out with the deployment of tether systems in real space flight conditions (TSS-1, -1R; SEDS-1, -2; TPE; CHANGE-1, -2; OEDIPUS-A, –C; ProSEDS, etc.) [65–71].

A small part of the tether system projects related to SD removal into disposal orbits is presented below.

6.1. Mechanical tether system

The idea of a swinging/rotating/asynchronous tether is to connect SD to be removed from NES and the service SC by tether with subsequent twisting/swinging of this assembly in the orbital plane. Twisting/swinging is carried out by the service SC remote control. In the desired phase of rotation/rocking, the SD separates from the tether and enters a new orbit. To increase the efficiency of such energy exchange, the elastic properties of the tether can be used. The rotating/swinging tether makes it possible to reduce the cost of the propellant for an interorbital flight compared to a purely rocket maneuver. In the limit, for a circular initial orbit, when the service SC mass is significantly greater than the SD mass, the required costs of the propellant are halved. The SD can dock with the service SC using a robotic arm. The flexible, long, retractable tether system combines the maneuverability of a tug with the ability to transfer additional momentum through a tether system (Fig. 11) [72–74].

After the SC/SD capture in GEO, the tether system can unfold along the local horizon (Fig. 12) to a distance of 300 km, for example. In such a system it is possible to excite a pendulum oscillation around the vertical plane relative to the center of mass of the "service SC-SD" system, which will lead to the appearance of excess orbital momentum, which can be spent for descending the SD from the orbit or transferring it to higher orbits [75].

The advantage of the tether system is in the significantly lower propellant consumption for the implementation of maneuvers, however, it is necessary to take into account the tether system mass, as well as to evaluate the forces arising in the tether. According to the estimates, a gradient of 1.6×10^{-6} g/km arises along the tether in GEO (a 1500-kg SD causes a tension force in the 300-km tether at a level of 7 N [72]).

One of the possible configurations of the tether system includes a SC located in GEO and two long vertical tethers with end-masses (Fig. 13) [76].

It is possible to use the effect of gravitational stabilization, in which the entire system at any time remains in a vertical position relative to the local horizon if the system has sufficiently long tethers and massive end-masses. Tether deployment and winding back leads to a shift in the center of gravity of the system relative to the working orbit, and, consequently, to the removal of the SC from the working orbit.

A SC in GEO or any other sufficiently massive SC currently in orbit above the GEO (for example, a SC that was transferred earlier to the disposal orbit) can serve as one of the end-masses in the tether system. It is possible to transfer new SC into GEO in this configuration.

Promising projects with a tether system can use the idea of SD gradual collection into a single object using a tether system, and then its gradual transfer to a disposal orbit.

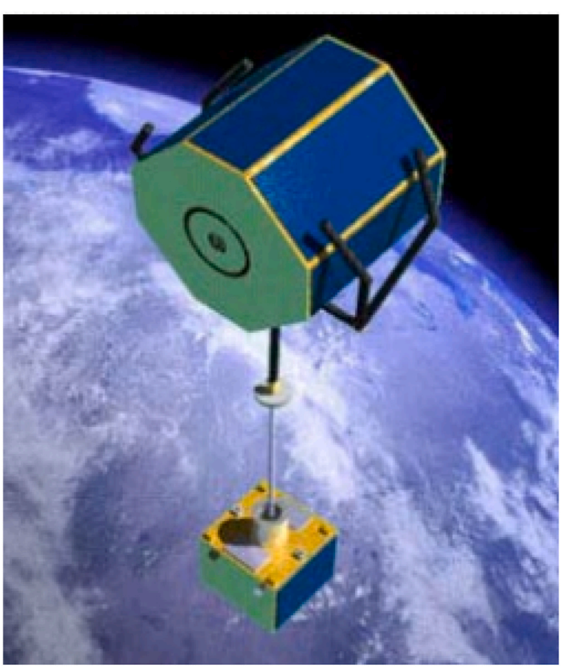

Fig. 11. Service SC Concept with Tether and Grip Mechanism [74] (with kindly permission Elsevier and Copyright Clearance Center's, order number 5374650044119).

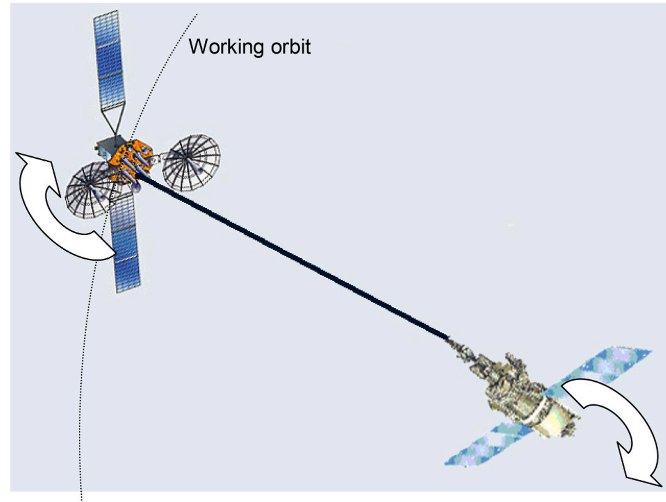

Fig. 12. Pendulum oscillations in a horizontal tether system.

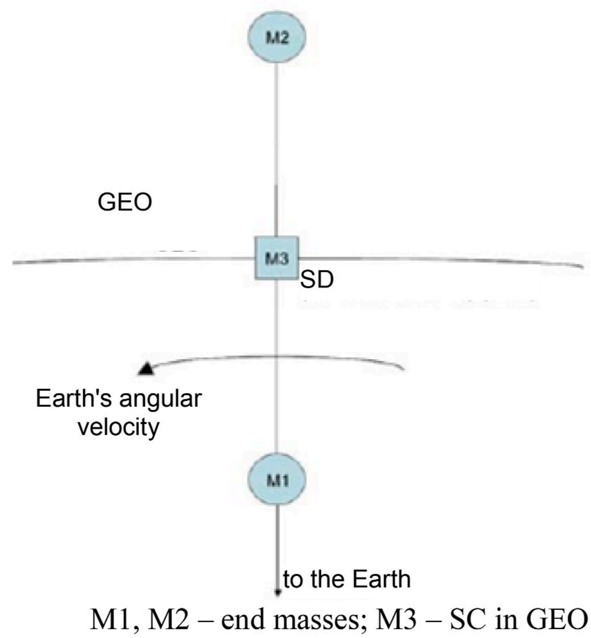

Fig. 13. Configuration of tether system with end-masses.

The required tether length is about 300 km. In this regard, questions arise related to the tether deployment, and to ensuring the tether system stability in the presence of disturbances associated, for example, with the angular movement of end-mass and SD, and disturbances arising during SD docking to end-masses.

Tether deployment to a given length can technically be implemented in the following ways [77]:

– with the help of impulse repulsion – passive deployment;
– with the help of impulse repulsion and subsequent control of the tether tension force – active deployment;
– with the help of continuous thrust of thrusters installed on one or two SC of the assembly - active deployment;
– by controlling the combination of reactive forces and elastic forces of tether tension.

In the first two options, the deployment is carried out due to the internal forces of the tether system.

The required terminal conditions at the end of the separation area can be ensured by Ref. [77]:

– selection of initial conditions of motion during pushing and passive tether system deployment;
– selection of initial conditions of motion and parameters of the program for controlling the tether tension force in the process of space tether system deployment;
– tether power sampling and changing the moment of inertia of the angular motion;
– thruster using.

In the first three options, the transfer of the space tether system to a stable mode of motion is carried out due to the internal forces of the tether system.

Various combinations of tether deployment control options and options for achieving the required terminal conditions determine different ways of transferring the space tether system to stable motion modes. There are a large number of publications on the dynamics of tether systems deployment and winding back, including space experiments [78,79].

Target capture mechanisms at the end of the tether system can be made in the form of harpoons [80–82], manipulators and capture units [83,84], nets [85,86], soft grips, etc.

Soft grips can adapt to uneven surfaces and grab and hold objects of various shapes, sizes and materials. Grips are usually made of compatible materials such as elastomers that distribute forces evenly, which can also be beneficial from a safety standpoint. However, such designs have relatively low grip forces; to increase this force, the design often uses the built-in electrostatic and adhesive technologies [87].

Such structures include the use of controllable dry adhesives, which allow you to capture SD of any configuration. Controllable dry adhesives are of interest in gripping and manipulating in space due to their ability to attach and detach in a controlled manner to and from any smooth, clean surface, including flat and curved surfaces. This capability greatly expands the number and types of potential target docking sites. The controllable dry adhesives are also inexpensive, space-suitable (perform well in vacuum, under extreme temperatures, and radiation), and can attach to the target surface and detach from it with minimal force [88–90].

The SD capture by the net can, to a certain degree of assumption, be considered a universal technique for SD collecting, since it is possible to capture SD of almost any configuration and size, including those that perform uncontrolled rotation at high velocities. Capture can be carried out from a distance that is safe for the service SC carrying the net due to the passive damping of the SD angular rotation and the presence of a flexible connection between the service SC and SD.

The shape (flat, conical, pyramidal, etc.) and the size of the net, the number of end-masses in the net corners depend on the SD to be captured, as well as on the dynamics of the entire structure. For example, for the Envisat spacecraft, a square-shaped flat net with 4 end-masses (bullet masses) was chosen. Net dimensions should be selected based on the SD capture strategy (whether the solar panels will be wrapped by the net or the SD body only) [91].

A large number of net dynamics analyses have been carried out [91–95].

6.1.1. Tethered space net robot project

The TSNR project (Fig. 14) is designed for SD removal by a net at the end of the tether system [96]. Mini-SC (maneuverable units) placed at the corners of the net (instead of bullet masses) can be controlled after the net is launched towards the target to maintain the net in the working configuration necessary to increase the probability of the target successful capture.

After SD capturing, the mini-SC converge to one position and the net closes to the minimum area. Extensive research has been carried out on the dynamics of the deployment of such a system, the control of the net closure, both in the case of ideal contact of the target with the center of the net, and in the case of deviations from the ideal position.

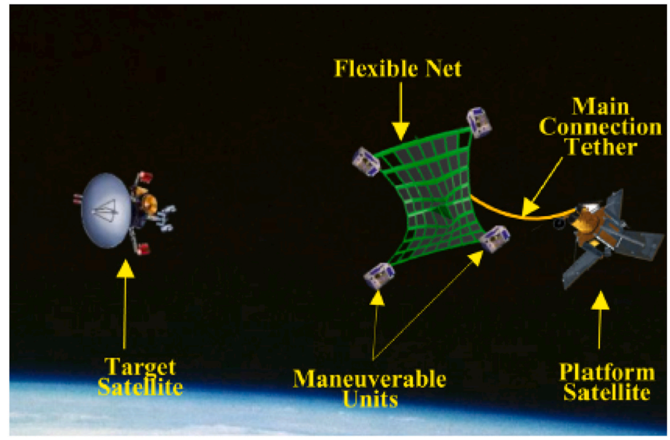

Fig. 14. TSNR Project [96] (with kindly permission Elsevier and Copyright Clearance Center's, order number 5374650257599).

Similar projects are proposed in Refs. [97–101]. Differences in the projects are related to the net type, tether lengths, maneuverable unit types, etc.

6.1.2. Robotic geostationary orbit restorer by ASTRIUM

The ROGER concept was proposed by EADS ASTRIUM (Paris, France). The 3500-kg service SC has the ability to inspect the client SC, stabilize and move it to other orbits using a capture system in the form of a net thrown out to a distance that excludes a collision between the service SC and client SC. First of all, the reusable system was intended for the removal of non-cooperative SC into disposal orbits. It was supposed to develop a service SC based on the SC platform developed by EADS ASTRIUM (Paris, France). There may be minor differences in the service SC designs for different missions. The service SC is equipped with a propulsion system (fuel mass is about 2700 kg, with the possibility of increasing up to 3200 kg), which consists of an apogee engine with a thrust of 400 N and two propulsion systems comprising 10 thrusters each with a thrust of 10 N. Solar arrays serve as a source of electrical energy. The SC power consumption is 300 W. The concept involves the use of up to twenty nets. The target capture unit mass is 9 kg; the net has four additional end-masses (bullet masses). The possibility of safe SD capture in the cases of different SC dynamics, shapes and dimensions is to be proven [102–105].

6.1.3. e.Deorbit project

The project is being developed by ESA (Paris, France) [106] as a part of the Clean Space initiative program aimed at removing SD from NES, as well as at the in-orbit SC repair/recharge/refuel [107]. The expansion of the range of possible SC missions was due to the lack of interest among customers in a SC operating exclusively with SD. The launch of a 1600-kg SC into a polar orbit with the altitude height of 800–1000 km by the Vega launch vehicle is scheduled for 2025. The SD or non-cooperative SC capture will be carried out either by a net or a robotic manipulator with subsequent coupling.

6.2. Electrodynamic tether

The electrodynamic tether (Fig. 15) is a lightweight and relatively inexpensive device that uses the electrodynamic drag generated by a conductive tether for SD removal from its working orbits. Tether has a length of 5–10 km; a part of the power generated by the tether can be used to control its movement. Thus, the tether can be independent of the

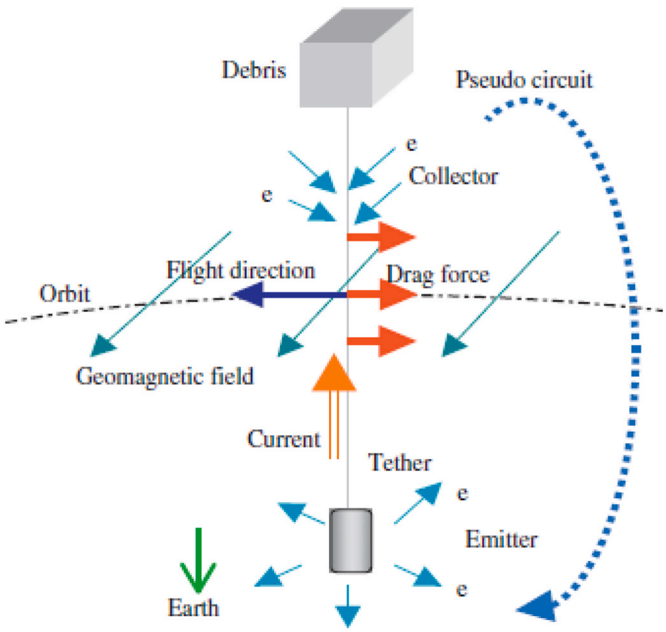

Fig. 15. Electrodynamic Tether Lay-Out [74] (with kindly permission Elsevier and Copyright Clearance Center's, order number 5374641309744).

power supply systems of the service SC – the electrodynamic tether carrier during the deorbit. Since the uncontrolled electrodynamic tether is dynamically unstable, the feedback control circuits are used.

To develop electrodynamic tether, a large number of experiments were carried out: FELDs [108–111], STAR [112], PACMAN [113,114], TED-Sat, etc.

6.2.1. Space debris micro remover project

JAXA (Chōfu, Tokyo, Japan) developed the SDMR project (Fig. 16). A microsatellite equipped with an electrodynamic tether was considered as a service SC, which makes it possible to decrease the SC orbit altitude without mass consumption or to service SC in orbit, or to place the SD capture unit at the end of the electrodynamic tether [73,74,115].

The small SC has compact dimensions (700 × 700 × 600 mm) and low mass (1400 kg, 25 kg of fuel) and can be launched as a secondary payload by the H-IIA launch vehicle. The rendezvous system is based on GPS sensors, star trackers and video cameras. Eight small thrusters with a thrust of 1 N are used for interorbital maneuvering. The SC is equipped with a robotic arm for target capture and an electrodynamic tether for SD removal. Solar arrays (average power: 100 W) are mounted on the side surfaces of the SC. The SC has 3-axis stabilization.

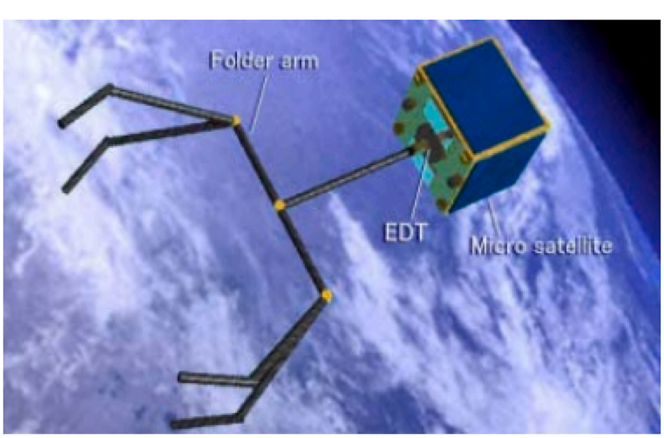

Fig. 16. SDMR Project with Capture Unit at the End of Tether [74] (with kindly permission Elsevier and Copyright Clearance Center's, order number 5374641458053).

6.2.2. Terminator Tether ™

Terminator Tether™ is designed to remove SC and upper stages of launch vehicles from LEO (Fig. 17). The system includes a tether system, a tether deployment system, a device for generating electric current, and an electronic tether control unit. To isolate SC electrically from the conductive tether, a part of the tether closest to the service SC will be non-conductive, made of high-strength threads. The rest of the tether will be made of thin aluminum filaments. Typical tether with the length of about 5 km and mass of 15 kg will provide for the removal of SC of 1500 kg in mass from LEO [116].

Mathematical models of electrodynamic tether deployment and SD removal are developed for Terminator Tether™. SEDS deployment data was used for the non-conductive part of the tether.

It is shown that a tether with the mass of 1% of that of the service SC can remove the Iridium NX SC from the orbit with 850 km altitude and inclination of 50° in 3 months, while the Sky Bridge SC can be removed by it from the orbit with 1475 km altitude and an inclination of 55° in 1.2 years. A tether system with the mass of 2% of the mass of a service SC can de-orbit an upper stage from the orbit with altitude of 400 km and inclination of 50° in less than two weeks, a SC from MEO with 850 km altitude and inclination of 50° - in less than three months, or a SC from HEO with 1400 km altitude and inclination of 50° - in less than a year. The constructed model showed that the electrodynamic tether can remove SC from the orbit with an inclination of up to 75°, while the force of interaction of the tether with the Earth geomagnetic field is quite intense [116].

6.2.3. ElectroDynamic delivery express project

The EDDE project was developed in 1999 by STAR Technology and Research (Mount Pleasant, South Carolina, USA), AFRL (Dayton, Ohio, USA), and NASA Marshall Space Flight Center (Huntsville, Alabama, USA) as a part of the SBIR project [117].

An autonomous SC is being developed that delivers several small SC with short service life to LEO from other working orbits within several months without fuel consumption. The system uses solar energy to run

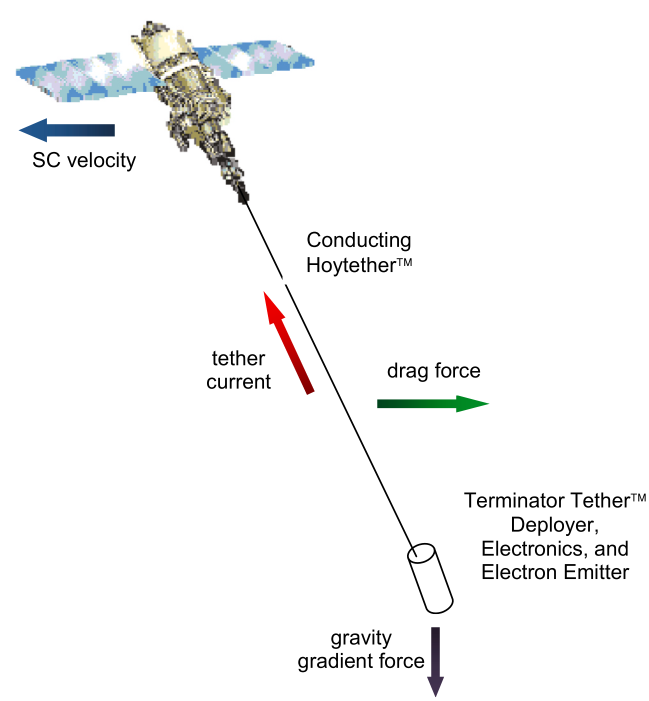

Fig. 17. –Terminator Tether™Concept.

electrical current through tether with a cross-sectional area of 1–5 mm^2 and a length of 4–20 km. A tether made of reinforced aluminum foil is oriented normally to the local Earth geomagnetic field and is electrically closed through the ionosphere. The system can rotate making 8 turns per one revolution around the Earth to improve stability and operational flexibility. A change in current depending on the orbit used and on the frequency of rotation imposes forces and moments on the system that allow changing the orbit parameters, increasing the maneuverability of the assembly, allowing it to move, including across the tether/geomagnetic field. The feedback control algorithms have been developed. An assessment was made of the dynamics of the system, tether tension, accelerations, etc. [118–122].

The main SC design features have been developed, which are optimized for rapid orbit transfers under the action of electrodynamic forces. It has been shown that the electrodynamic forces generated in the tether can work well over a wide range of velocities/orbits, but in LEO the aerodynamic drag can be equal to or greater than the generated thrust. Various EDDE configurations have been considered [117,122]. The diagram (Fig. 18) shows different variants of the system. The first one has 36 kg, and can be launched by the Delta launch vehicle [121].

A modified concept of the EDDE was considered, in which the SD was captured by a net of about 100 m^2 located at the end of the EDDE.

According to calculations, such a design is able to capture a rotating 1.4-ton SD if the rotation velocity does not exceed 2 rpm or a 8-ton SD if it the rotation velocity does not exceed 0.5 rpm [117]. It is shown that the most productive is the use of EDDE to remove SD with the mass of more than 100 kg from LEO, which accounts for about 98% of all SD in LEO, and which are the most dangerous in terms of the occurrence of Kessler's syndrome [14,122].

However, at the altitudes above 600–1200 km (depending on the eleven-year cycle of solar activity), the low plasma density limits the collection of electrons and, consequently, the thrust of the system [121]. The use of EDDE in GEO seems to be problematic due to the extremely low strength of the Earth's magnetic field at such altitudes, and the need in the propellant consumption to generate plasma when the electrical circuit is closed outside the Earth's ionosphere.

The concept of application of electrodynamic tether of an aluminum tape 0.05 mm thick and 3 cm wide (the mass of the tether was 40 kg, 60 kg, 80 kg depending on the length) with a length of 10 km, 15 km, 20 km was also considered in Ref. [123] for SD removing. It is emphasized that the dynamic behavior of the tether system (deployment, folding, stabilization) is an independent complex technical problem, some aspects of which are considered, but not fully resolved, and, moreover, not tested in practice. Taking into account the duration of transportation, it is necessary to take into account the variability of the forces acting on the tether system (for example, solar activity).

The recent transition to the use of small SC for various missions has led to the development of EDDE lightweight versions, for example, a 60-kg EDDE with the length of 8 km [117], which can be launched into orbit as a secondary payload.

6.2.4. *Space tethered autonomous robots satellite II nano-satellite*

Nano-satellite STARS-II was built by Kagawa University (Takamatsu, Kagawa, Japan) to test an electrodynamics tether in LEO and demonstrate possible technology for SD de-orbiting. STARS-II SC was launched by the H-IIA launch vehicle as a secondary payload on 27 February 2014 into the 350-km working orbit [124,125].

For the experiments, the SC splits into two parts, connected by a 300-m tether; during the experiments the tether deployment was video-recorded and the tether was used to deorbit the SC. The SC consisted of a 5-kg basic vehicle, with dimensions of 160 × 160 × 253 mm, and a 4-kg vehicle at the end of the tether measuring 160 × 160 × 158 mm. The electrodynamic tether was made of ultra-thin wires of stainless steel and aluminum.

The experiment was only partially successful, and tether deployment could not be confirmed [126].

The same project of Electrodynamic tether mounted on a 3U CubeSat was presented in Ref. [127].

7. Electromagnetic sail

The concept of an electromagnetic sail is based on the interaction of the magnetic field of a large superconducting ring with the magnetic field of ionized solar wind particles. The magnetic field generated by the current in the superconducting loop stiffens the loop, which takes the form of a ring. Charged particles in a magnetic field are deflected, imparting momentum to the loop. Under the influence of the solar wind, an electromagnetic sail is theoretically capable of producing accelerations of the order of 0.01 m/s^2 at a heliocentric distance of 1 AU, and the thrust vector can be directed tangentially, which can be used to perform maneuvers [56].

Additional force acting on the loop arises due to interaction with the Earth's magnetosphere. With a stable position of the loop, the radial component of the force is directed towards the center of the Earth, axially towards the nearest magnetic pole [56].

The main difficulties in the practical use of an electromagnetic sail are the need to deploy a superconducting loop with a diameter of tens of kilometers, as well as the high energy costs for activating the sail, even without taking into account the cost of cooling the ring to cryogenic temperatures.

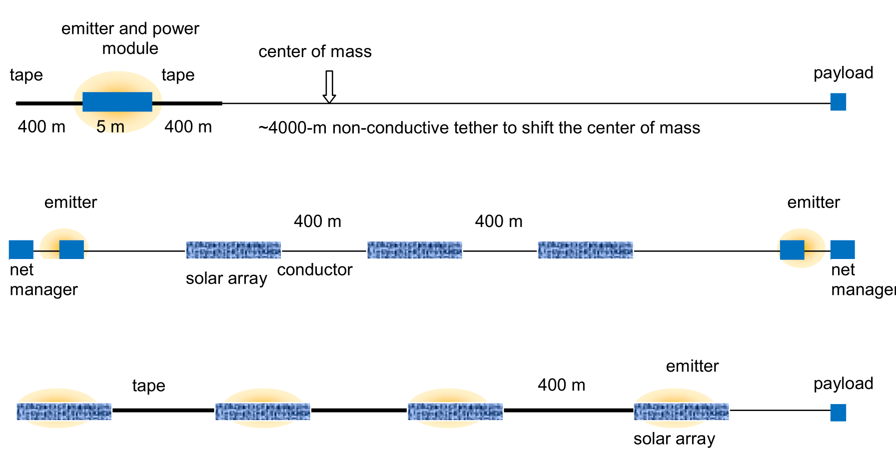

Fig. 18. Edde versions lay-out.

In view of the above, the use of an electromagnetic sail for the task of SD removal from the GEO vicinity in the foreseeable future seems to be extremely problematic.

8. Solar concentrator

Initially, solar concentrators were proposed to change the orbits of asteroids to prevent them from colliding with the Earth [128]. Several phenomena that contribute to orbital deviation have been studied: sublimation of the asteroid surface with the simultaneous creation of a directed jet of sublimated material (gas); creation of traction due to solar pressure; use of the effect of the appearance of a weak reactive impulse due to thermal radiation of the heating/cooling surface of the asteroid (the Yarkovsky effect).

Later, it was proposed to use the same technology for removing SD with an area-to-mass ratio in the range of 0.0001–1 m^2/kg for LEO, GEO, and GTO [129].

The method proposes to use a solar concentrator to focus a small beam of light with a high power density on the SD surface, thereby causing sublimation of the SD material surface layers with the creation of a reverse flow of the sprayed material, which in turn creates a small thrust that changes the SD trajectory (Fig. 19).

It is shown that a SC even with a small area-to-mass ratio (up to 0.01) can be removed from an 800-km-orbit to a 200-km orbit in several hundred days of continuous operation of solar concentrators.

9. Laser usage

9.1. Laser in the earth orbit

The laser application for SD removal was proposed in 1991 for the first time [130].It was proposed to use laser radiation for transporting large-sized SD into LEO for their subsequent burn up in the Earth's atmosphere, for transporting SC from GEO into disposal orbits, or for destroying small (about 10 cm) SD in LEO [131–136].

The SD removal system proposed in Ref. [75] includes a solar power plant with a capacity of 30 kW and a neodymium laser with a cryogenic cooling system. The accumulation of energy for the laser occurs within 10 h. Due to the scattering of the laser beam, it is necessary to use an optical system to focus it. Propulsion unit includes electric thruster.

The advantage of using such a system is that it can be operated at a great distance from SD, which is determined by the power of the available energy source. However, the use of laser radiation for removing SD over 20-cm is limited due to the smallness of the generated pulse [133].

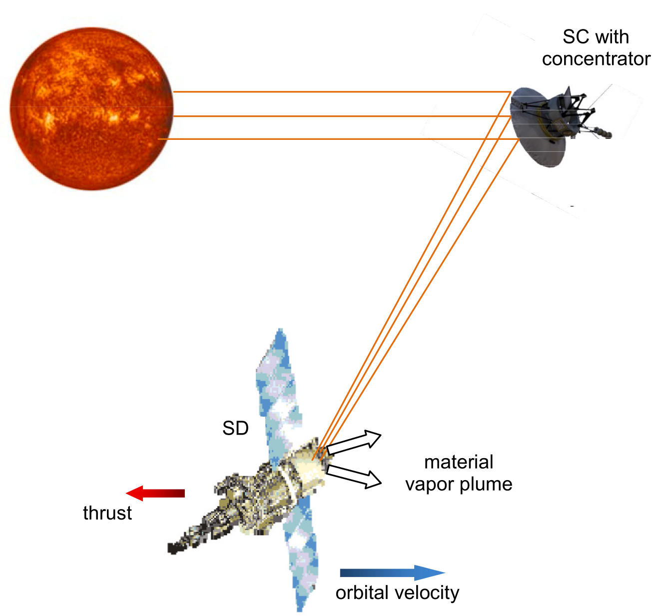

Fig. 19. Solar concentrator application for SD removal.

9.2. Ground laser

Several concepts of a ground-based laser have been proposed, differing in the technical characteristics of the lasers used (Fig. 20) [48].

NASA's Ames Research Center (Moffett Field, California, USA), Stanford University (Stanford, California, USA), USRA (Washington, DC, USA) suggested using a 5–10 kW laser, which should be connected to 1.5–m telescope [137]. The telescope is equipped with adaptive optics capable of accurately aiming the beam at SD. The light pressure of the laser should change the SD orbit.

A similar concept of using a pulsed LODR laser was proposed in Refs. [138,139].It was shown that when using a laser (5 Hz, 125 kJ, 1 μm) from an orbit (similar to ENVISAT), a 1000-kg SD can be diverted by impacting the SD for 4 min every 10 days in 3.7 years [138].

The concept of using a 30-kW laser facility was proposed in Refs. [140,141] for cleaning NES from 1 to 20-cm SD with a total number of about 200 thousand. A ground-based iodine laser (1.3 μm) or neodymium laser (1.06 μm/530 nm) with pulse duration of 5 ns using a 4–6 m mirror is capable of cleaning NES up to altitudes of 800 km in 2 years according to calculations.

Common to all concepts is that the use of lasers is considered in relation to small SD.

10. «Ion beam shepherd » method

The method of moving space objects with an ion beam was first proposed for transport operations in the Earth-Moon system [142]. The authors of this publication considered the concept of a powerful ion injector installed on the surface of the Moon, which directs a powerful high-energy ion beam to a SC located in a selenocentric orbit. The search for the optimal parameters of the ion source, collimation and scattering of the ion beam at large distances are indicated as the main problems in the implementation of the method.

A similar principle of action in relation to SD (Fig. 21) was later proposed by Japanese [143] and European [144,145] specialists.

The ion beam is generated and accelerated up to 30 km/s and over by an additional service SC propulsion unit (ion gun/ion injector) directed towards the SD, placed at a distance of several overall dimensions of the

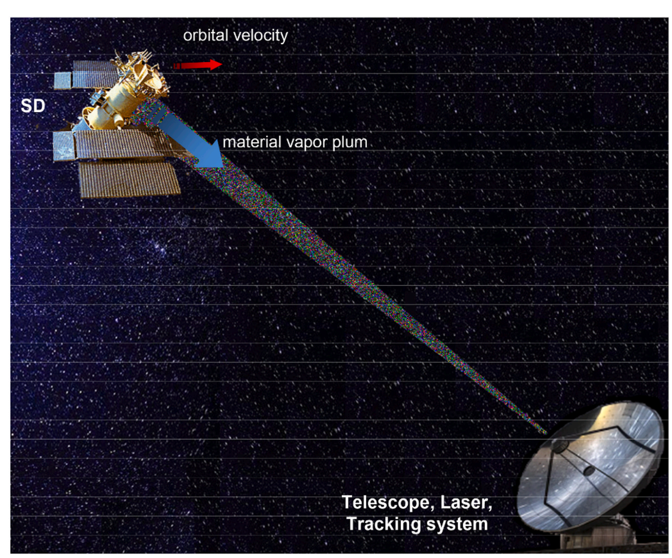

Fig. 20. Ground-based laser concept.

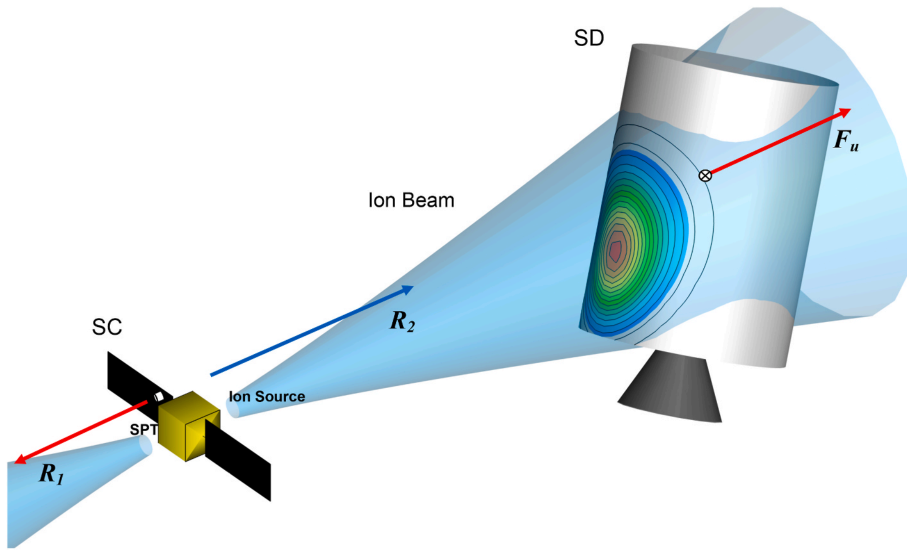

Fig. 21. The ion beam shepherd method.

SD on-board the service SC. A beam of accelerated quasi-neutral plasma hits the SD surface, acting on it with a force, the value of which depends on the ion beam divergence angle, the SD surface sputtering, the deviation of the beam axis from the SD center of mass, and other factors.

The description of the mathematical model of the interaction between the ion beam and SD is given in Refs. [144,146,147], the mass model of the system is presented in Ref. [144].

In [148–151] the force and erosion effects of an ion beam on man-made SD were simulated. The problems of service SC control during SD transportation are presented in Refs. [152–155].

The effectiveness of method for SD removal from GEO is shown. According to calculations, the mass of xenon required for the sequential removal of ten SD objects by a service SC is 298.735 kg, the period of electric propulsion system operation is 324.211 days, and the period of ion source operation is 48.71 days [154].

The problematic issues of ion sources development are described in Refs. [156,157].

11. Conclusion

The world community has developed guidelines for the use of near-Earth space which emphasize the need to take immediate measures to reduce its pollution already at the design stage of space systems, and to minimize or eliminate the generation of space debris during the implementation of missions or after their completion [158,159]. The projects presented in the article are aimed at solving this problem.

Earth-oriented tethers are permanently located in working orbits. Maneuvers are carried out by the exchange of impulses between the end-masses. The system can be built using the available materials. It is required to ensure the possibility of docking with the spacecraft/space debris in the working orbit. The launch mass is less compared to the mass of the tug. The fuel consumption of the base spacecraft equipped with a tether for maintaining the orbit and taking off at the end of service life is reduced. Space debris in working orbits can be used to reduce the initial mass of the system if they are used as one of the end-masses. High risk of tether collision with other space objects is assessed. Maintaining the stability of the tether system is not required.

The rotating tether system provides significant fuel savings compared to using tugs/booster, but increases the maneuver time by 50% compared to the same, due to restrictions on the tension in the tether [160]. Orbit maintenance and dynamic maneuvers are challenging. Materials for the tether system with a large margin of safety exist, but their significant degradation is possible under near-Earth space conditions, and there is also a high probability of collision with other spacecraft/space debris. It is shown that the probability of damage to a 0.5-mm tether with the length of 600 km by micrometeorites or 0.15-mm space debris within 5 years of service life is about 2%.

A disadvantage of tether concepts is the difficulty in managing the flexible tether assembly. An insignificant decrease in the mass of the propellant required to transfer spacecraft into a disposal orbit cannot compensate for a significant increase in technical/technological risk, as well as an increase in the mass of the service spacecraft/end-masses due to the tether mass and its deployment system.

Some researchers, based on the results of the analysis, talk about the economic benefits of using tether systems for space debris removal. In their opinion, the gain can be dozens of times compared to systems built on the basis of chemical or electric thrusters [122].

The magnetic sail has a relatively low cost, but high technical complexity and great uncertainty in fundamental questions of the physics of the interaction of the solar wind with the magnetic structure.

A solar sail is practically a “perpetual” thruster, but with a large time spent for transport operations, which can only be acceptable at the end of the service life. The system can potentially provide an increase in the spacecraft service life in geostationary orbit, but currently it has a high cost when used in an autonomous mode and low reliability. The manufacturability of the concept has not been proven to date.

The concept of space debris removal using an ion beam makes it possible to transport space debris without docking with it, and, therefore, there is no need to develop docking/capture mechanisms that would be difficult to reuse for several consecutive operations of the space debris deorbiting, develop complex algorithms for assessing the state of the space debris surface and its dynamics before docking. It becomes possible to transport non-cooperative space debris (uncontrolled motion of space debris: disordered rotation, often at high velocity, oscillations with large amplitude, etc.) with not exactly established geometry, dynamics, mass and inertial characteristics.

None of the described methods is being implemented in full to date, but this does not mean that they cannot be implemented in practice in the future or that they are not suitable for solving the problem of active space debris removal.

Modern spacecrafts at the end of the service life are removed from working orbits to disposal orbits using their propulsion system. This method of “cleaning” the near-Earth space has a number of significant drawbacks. There is a high technical risk due to the fact that the removal operation is performed at the end of the service life, and most critical failures of the main spacecraft service systems occur at this time (failure of onboard systems makes it impossible to perform the removal operation). Removal requires propellant, the amount of which is quite difficult to predict even for the fulfillment of the target task due to the rather long service life of modern spacecraft (about 10–15 years). The spacecraft service life decreases, as a part of the propellant sufficient for station-keeping during several months is to be spent for removal, and this is the time during which the spacecraft could perform the purpose task. However, the spacecraft removal from working orbits by their propulsion system is the main, cheapest and the most reliable way to put a spacecraft into a disposal orbit.

The vastness of the proposed methods and technologies sets a difficult task before the developers of space technology to choose the most promising and feasible method in each specific case. The described methods should be considered for the space debris already available in the near-Earth space and for the emergency situations that make it impossible to transfer spacecraft into a disposal orbit using its own propulsion system.

Declaration of competing interest

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

Acknowledgments

This work was supported by the RSF grant No. 22-19-00329 (https://rscf.ru/project/22-19-00329/).

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