Aerospace Science and Technology 71 (2017) 156–162

Contents lists available at ScienceDirect

Aerospace Science and Technology

www.elsevier.com/locate/aescte

Swing principle in tether-assisted return mission from an elliptical orbit

Vladimir S. Aslanov *, Alexander S. Ledkov

Samara National Research University, 34, Moscovskoe shosse, Samara 443086, Russia

ARTICLE INFO

Article history:
Received 21 March 2017
Received in revised form 5 September 2017
Accepted 5 September 2017
Available online 12 September 2017

Keywords:
Space tether
Re-entry
Deorbiting
Elliptical orbit
Control law
Swing principle

ABSTRACT

The problem of a tether-assisted payload return from an elliptical orbit is considered in this study. In contrast to the existing works devoted to this issue, the article deals with a tether length control that provides a transfer of the payload into a descent trajectory from the tether rotation mode. Application of the swing principle for the tether control is investigated. The simplified mathematical model of the space tethered system is developed. It is shown that the stable limit cycle could exist under the considered control. The approximate analytical solution for this cycle is obtained. The stability of this solution is studied by the Lyapunov's theorems. The optimal control, which provides transfer of the payload into the descent trajectory with minimum perigee radius, is found as a result of the simulation series. It is shown that the tether should occur several turns before the payload separation. For example, in the YES-2 experiment, it is demonstrated that proposed control makes it possible to perform a payload return using a tether of considerably shorter length. The main conclusion of the paper is that the proposed scheme of the payload deorbit is more effective than the classical static or dynamic tether deployment schemes.

1. Introduction

Space tethered systems is a promising direction of modern astronautics widely discussed in the scientific literature. Applying the space tethers allows a reduction in the cost of orbital operations by eliminating the use of jet fuel. The mechanics of tethered-assisted orbital maneuvers are described in detail in the works [1–3]. This paper focuses on the problem of a payload de-orbit from an elliptical orbit by a variable-length tether. In contrast to the majority of works devoted to space debris mitigation by tethered spacecraft [4–6], the considered de-orbit scheme does not imply the removal of the active spacecraft. To date, three successful experiments of a tether-assisted payload landing have been conducted: SEDS-1 in 1993, SEDS-2 in 1994, and YES2 in 2007 [7]. These experiments proved the feasibility of the proposed technology, and showed the need for further theoretical research in this field.

Several approaches to the tether control for payload deorbiting maneuver are considered in the literature. Zimmerman [3] divides tether length control laws into two groups: "static," which supposes the payload release from the tether equilibrium state, and "dynamic," which assumes the use of the tether oscillations near the equilibrium position for additional payload velocity reduction. The main advantage of dynamic control in comparison with static is the possibility of carrying out the payload deorbiting using tethers of considerably shorter length. Aslanov studied tether swing control law, which allows increasing amplitude of the tether oscillations by changing its length for the case of circular orbit [8]. The proposed control provides an effective return of a re-entry capsule even from the vertical equilibrium state. Tether deployment into an upright position can be done, for example, by the control proposed by Yu. Zabolotnov [9]. In the case of an elliptical orbit, the vertical position is not an equilibrium state. It is a known fact that for momentum exchange maneuvers, the rotational mode of the tether motion is more effective from the point of view of the payload orbit attitude increase or decrease [10,11]. The authors consider the possibility of applying this idea to the tether-assisted payload de-orbiting maneuver. The hypothesis is tested in this article, that the swing control allows a transfer of the tether system into the rotation mode.

The aims of this paper are studying the ability of the tethered system transfer into rotational mode by the swing control law; searching the control law parameters, that provides effective payload de-orbit from an elliptical orbit; and researching the features of the space tethered system motion on the elliptical orbit. The radius of perigee of the payload orbit after the tether separation is used as an optimality criterion of the tether control law.

* Corresponding author.
E-mail addresses: aslanov_vs@mail.ru (V.S. Aslanov), ledkov@inbox.ru (A.S. Ledkov).
URLs: http://aslanov.ssau.ru (V.S. Aslanov), http://www.ledkov.com (A.S. Ledkov).

http://dx.doi.org/10.1016/j.ast.2017.09.006
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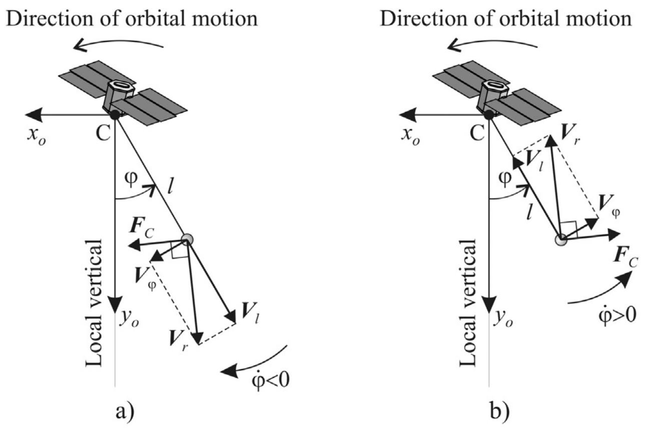

Fig. 1. The swing control stages.

2. The swing control

Considered tether assisted return mission assumes the use of a tether relative oscillation to reduce the absolute velocity of the payload. Similar to the case of a mathematical pendulum, the larger amplitude of the tether oscillations corresponds to a larger angular velocity of the bottom point. Considered swing control allows an increase in the oscillation amplitude using Coriolis force. To illustrate the idea behind this method of control, consider the motion of a satellite in a circular orbit. A weightless tether with a payload at its end is attached to the satellite through the deployment mechanism, which allows for deploying and retracting the tether. The motion of the payload relative to the satellite is considered in a non-inertial rotating orbital coordinate frame $Cx_o y_o$ (Fig. 1). The origin of this frame is located at the center of mass of the satellite. The axis Cy_o lies along the local vertical. The axis Cx_o is directed towards the orbital flight. The location of the payload can be described by the deflection angle φ and the tether length l. The relative velocity of the payload $\mathbf{V}_r$ is the sum of mutually perpendicular vectors $V_l = \dot{l}$ and $V_\varphi = \dot{\varphi} l$. The swing control can be divided into two stages: when the direction of the tether rotation is opposite to the direction of the satellite orbital motion $\dot{\varphi} < 0$ (Fig. 1a), and when they coincide $\dot{\varphi} > 0$ (Fig. 1b). At the first stage the tether length should be increased, and at the second stage it should be decreased. With this control, the Coriolis force $\mathbf{F}_C$ will increase the amplitude of the tether oscillation. The described control scheme can be specified by the law [8,12]

$$\dot{l} = -\lambda \dot{\varphi}, \tag{1}$$

where λ is a constant control coefficient.

The issue of choosing the moment of the payload separation from the tether is interesting. In the case of circular orbit and constant tether length, separation of the payload when passing through the local vertical is most effective from the viewpoint of reducing the radius of the perigee of the payload orbit. At this moment the velocity vector of the satellite $\mathbf{V}_1$ and the relative velocity of the payload $\mathbf{V}_r$ lie on one line, they are oppositely directed, and the relative velocity modulus is maximal. In this regard, at this point the payload has the lowest possible absolute velocity $\mathbf{V}_2$, which is directed along the local horizon. After separation of the tether, the payload will be at the apogee of its orbit (Fig. 2a). Significantly more complex and interesting is the case of an elliptical orbit. At the point of passage of the local vertical, the directions of the vectors do not coincide, and the modulus of the relative velocity of the payload at this point may not be minimal (Fig. 2b). Additional efforts are required to determine the optimum separation point.

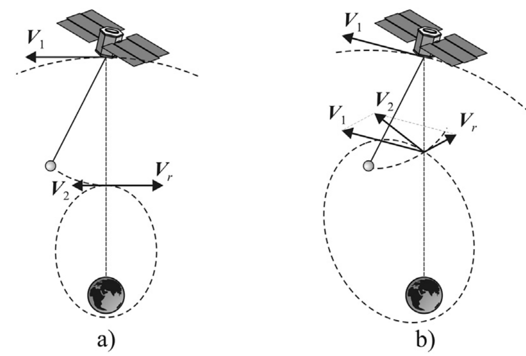

Fig. 2. The payload separation at the local vertical.

3. Mathematical models and methods

The mechanical system consists of a satellite, a tether and a re-entry capsule (Fig. 3). The satellite and the capsule are considered as material points of masses m_1 and m_2 correspondingly. It is supposed that a mass of the tether is many times less than the masses of the satellite and the capsule. The tether is deployed from the satellite. It is assumed that the center of mass of the mechanical system moves on an unperturbed Keplerian orbit. In this case plane motion of the system can be described by the equations [7,13]

$$\varphi'' = 2(\varphi' + 1)\left(\frac{e \sin\theta}{k} - \frac{l'}{l}\right) - \frac{3 \sin\varphi \cos\varphi}{k}, \tag{2}$$

$$l'' = \frac{2el' \sin\theta}{k} + l\left((\varphi' + 1)^2 + \frac{3\cos^2\varphi - 1}{k}\right) - \frac{(m_1 + m_2)T}{m_1 m_2 n^2 k^4}, \tag{3}$$

$$r = \frac{p}{k}, \tag{4}$$

$$\dot{\theta} = nk^2, \tag{5}$$

where e is the orbital eccentricity, p is orbital parameter, θ is true anomaly, φ is deflection of the tether from local vertical, l is the tether length, T is the tether tension force, r is the distance between the center of Earth and the center of mass of the system, $n = \sqrt{\mu p^{-3}}$, μ is the gravitational constant of the Earth, $k = 1 + e\cos\theta$, $(\,)'$ is the derivative with respect to θ.

For the goals set in the paper, it suffices to restrict ourselves to considering the motion in the plane of the orbit. It is known that in the absence of out-of-plane perturbations the plane motion of a space tethered system is stable [13,14]. Analysis of the YES2 experiment evidenced that in the case of a near-circular orbit the atmospheric disturbances cause out-of-plane oscillations, for which amplitude does not exceed 0.1 deg [13]. The paper [14] demonstrates that a control law developed for a plane tether deployment and retrieval gives a good result in the three-dimensional case. Thus, the consideration of the system's plane motion can be considered as the first step in constructing the tether control law for tether-assisted return missions.

In the initial moment of time $\varphi = 0$. System translates into rotation mode by the control law (1), which can be rewritten as

$$l' = -\lambda \varphi', \tag{6}$$

where λ is a constant control coefficient. Integration of Equation (6) yields

$$l = -\lambda \varphi + L_0.$$

Here L_0 is the tether length for $\varphi = 0$.