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Relative attitude and position estimation for a tumbling spacecraft

Lijun Zhang [a,*,1], Shifeng Zhang [a,2], Huabo Yang [a,3], Hong Cai [a,4], Shan Qian [b,5]

[a] College of Aerospace Science and Engineering, National University of Defense Technology, Changsha, 410073, China
[b] The State Key Laboratory of Astronautic Dynamics, China Xi'an Satellite Control Center, Xi'an, 710043, China

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ABSTRACT

This paper presents a novel relative attitude and position estimation approach for a tumbling spacecraft. It is assumed that the tumbling chief spacecraft is in failure or out of control and there is no a priori rotation rate information. The Euler's rotational dynamics is used to propagate the chief angular velocity, and the unknown inertia parameter circumstance is also considered. The integrated sensor suit comprises a rate-integrating gyro and a vision-based navigation system on the deputy spacecraft. Two relative quaternions that map the chief Local Vertical Local Horizontal (LVLH) frame to the deputy and chief body frames are involved to construct the line-of-sight observations. Therefore, the assumption that the chief body frame coincides with its LVLH frame in the traditional algorithm can be released. The general relative equations of motion for eccentric orbits are used to describe the positional dynamics. An extended Kalman filter is derived to estimate the relative quaternions, relative position and velocity, deputy gyro bias, as well as chief angular velocity and inertia ratios. Simulation results verify the validity and feasibility of the proposed algorithm.

1. Introduction

With the development of the space technology, there has been a growing interest in autonomously servicing spacecrafts on orbit to perform tasks such as spacecraft rescue and repairing, refueling, and debris removal. Several space programs have been completed or are being developed as technology demonstration of these tasks, such as JAXA's ETS-7 [7], NASA's DART program [4], DARPA's Orbital Express program [15], and SUMO program [3]. Because of the cruel space environment and the limitation of the lifetime, a spacecraft may be in failure or out of control due to some malfunctions. Therefore, the capture for a tumbling spacecraft has received detailed attentions in recent years [1,10,14,17].

For a tumbling spacecraft, precise relative attitude and position estimation are the prerequisites for the success of capture. Motivated by the Hubble robotic servicing mission, a nonlinear algorithm has been developed to estimate the rotation rate and attitude for a tumbling spacecraft [14]. However, only the relative rotational motion is considered, and the properties of the sensor and process noises as well as the inertia parameter uncertainties are not taken into consideration. By using the laser-vision system, Aghili and Parsa [1] have proposed an adaptive Kalman filter to estimate and predict the relative motion and orientation of a tumbling target spacecraft. Presently, vision-based navigation (VISNAV) systems, which have the advantage of being entirely passive, are usually used to determine the relative attitude and position within the range of a few hundred meters [8,18]. In [8], Kim et al. have designed an extended Kalman filter (EKF) to estimate the relative attitude and position between the chief and deputy spacecrafts using the line-of-sight (LOS) observations coupled with gyro measurements from each spacecraft. Using the similar relative translational and rotational modeling approach, a relative attitude and position estimation approach for a tumbling spacecraft based on VISNAV system is presented here.

The VISNAV system consists of an optical sensor combined with specific light sources (beacons) to achieve a selective vision. In general, the known beacon locations are defined in the chief body frame, whereas the relative position vector between the chief and deputy spacecrafts is expressed in its Local Vertical Local Horizontal (LVLH) frame. In [8,18], it was implicitly assumed (but not clearly stated) that the absolute position and attitude of the chief spacecraft were known, and only the relative quantities needed to be estimated. Therefore, a simplified assumption that the chief body frame coincides with its LVLH frame was made to construct the LOS observations for convenience. Unfortunately, this assump-

* Corresponding author. Tel.: +86 15274928448; fax: +86 073184512301.
E-mail addresses: alijun_007@163.com, alijun@nudt.edu.cn (L. Zhang), zhang_shifeng@hotmail.com (S. Zhang), yhang0731@mailme.cn (H. Yang), hcai@nudt.edu.cn (H. Cai), shanqian_123@163.com (S. Qian).
1 Doctoral Student.
2 Professor.
3 Associate Professor.
4 Professor.
5 Aerospace Engineer.

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Nomenclature

VISNAV vision-based navigation
LOS line-of-sight
EKF extended Kalman filter
ECI Earth-Centered-Inertial (I frame)
LVLH Local-Vertical–Local-Horizontal (H frame)
$\boldsymbol{\rho} \equiv (x, y, z)$ relative position vector
$\dot{\boldsymbol{\rho}} \equiv (\dot{x}, \dot{y}, \dot{z})$ relative velocity vector
$(r_c, \dot{r}_c)$ radius and radial rate of the chief spacecraft, respectively
$(\theta, \dot{\theta})$ true anomaly and rate of the chief spacecraft, respectively
p semilatus rectum of the chief spacecraft
$\boldsymbol{\varpi} \equiv (\varpi_x, \varpi_y, \varpi_z)$ acceleration disturbance vector
$(\boldsymbol{q}_{d/H}, \boldsymbol{q}_{c/H})$ relative quaternions from the H frame to the d frame and to the c frame, respectively
$(\boldsymbol{A}_H^d, \boldsymbol{A}_H^c)$ attitude matrices from the H frame to the d frame and to the c frame, respectively
$(\boldsymbol{\omega}_{d/I}^d, \boldsymbol{\omega}_{c/I}^c)$ inertial angular velocities of the deputy and chief spacecrafts, respectively
$(\boldsymbol{q}_d, \boldsymbol{q}_c)$ inertial attitude quaternions of the deputy and chief body frames, respectively
$\boldsymbol{\omega}_{H/I}^H$ inertial angular velocity of the chief H frame
$\boldsymbol{q}_H$ inertial attitude quaternion of the chief H frame
$\boldsymbol{\mathcal{X}}_i \equiv (X_i, Y_i, Z_i)$ coordinates of the ith beacon expressed in the c frame
$\boldsymbol{\mathcal{X}}_i' \equiv (X_i', Y_i', Z_i')$ coordinates of the ith beacon expressed in the H frame,
$\boldsymbol{J}_c$ inertia matrix of the chief spacecraft
$(J_{cxx}, J_{cyy}, J_{czz})$ three principal moments of inertia of the chief spacecraft
$\boldsymbol{l} \equiv (l_x, l_y, l_z)$ inertia ratio vector of the chief spacecraft
$(\omega_{cx}, \omega_{cy}, \omega_{cz})$ three components of the chief inertial angular velocity
$\boldsymbol{d}_c \equiv (d_{cx}, d_{cy}, d_{cz})$ disturbance torque vector acting on the chief spacecraft
$\boldsymbol{d}_c'$ redefined disturbance torque vector

tion is not valid under all situations, particular for a tumbling chief spacecraft. To circumvent this problem, two relative quaternions that map the chief LVLH frame to the deputy and chief body frames are involved to construct the line-of-sight observations in this paper. These two relative quaternions are additionally estimated, so that the attitudes of the chief spacecraft need not be known *a priori*. It is assumed that the tumbling chief spacecraft may be in failure or out of control and there is no a priori rotation rate information. The Euler's rotational dynamics is used to propagate the chief angular velocity, and a common rate-integrating gyro is employed to measure the deputy angular velocity. The inertia ratios are introduced to solve the unknown chief inertia parameter circumstance. The general relative equations of motion for eccentric orbits are used to describe the positional dynamics. An extended Kalman filter is derived to estimate the relative quaternions, relative position and velocity, deputy gyro bias, as well as chief angular velocity and inertia ratios.

This paper is outlined as follows. In Section 2, various reference frames used in this paper are summarized and a review of the relative equations of motion for eccentric orbits is provided. Two relative quaternions that map the chief LVLH frame to the deputy and chief body frames are defined, and the corresponding relative quaternion kinematics equations are given. A stringent VISNAV measurement model for the LOS observations is derived using these two quaternions. In Section 3, the angular velocity measurement models for the chief and deputy spacecrafts are derived. Section 4 derives the implementation equations for the extended Kalman filter. Section 5 includes the simulation results, followed by conclusions in the last section.

2. Overview

In this section, an overview of the general relative equations of motion for eccentric orbits is shown. Two relative quaternions that map the chief LVLH frame to the chief and deputy body frames are defined, and the corresponding relative quaternion kinematics equations are derived. The stringent measurement equations for the VISNAV sensor are derived using these two relative quaternions. Thus, the assumption that both the chief body and LVLH frames are the same can be released.

2.1. Reference frames

(1) Earth-Centered-Inertial (ECI) frame (I frame): The frame has its origin at the center of the Earth and is non-rotating with respect to the stars (except for precession of equinoxes). The z axis points in the direction of the North pole, the x axis points in the direction of the Earth's vernal equinox direction, and the y axis completes the right-handed system.

(2) Local-Vertical–Local-Horizontal (LVLH) frame (H frame): The LVLH frame is centered at the chief spacecraft body, the x axis is directed from the spacecraft radially outward and often labeled as the R-bar, z axis is normal to the chief orbital plane, and y axis is defined as the cross-product of the other two axes.

(3) Body frame: This frame is fixed onto the spacecraft body and rotates with it. Body frames fixed to the two spacecrafts are designated as chief (c frame) and deputy (d frame), respectively.

2.2. Relative orbital motion equations

In this section, the relative equations of motion using Cartesian coordinates in the rotating LVLH frame are summarized. The relative orbit position vector $\boldsymbol{\rho}$ is expressed in the chief LVLH frame components as $\boldsymbol{\rho} = [x, y, z]^T$. If the relative orbit coordinates are small compared to the chief orbit radius, the general relative equations of motion for eccentric orbits are given by [8]

$$\begin{aligned}
&\ddot{x} - x\dot{\theta}^2\left(1 + 2\frac{r_c}{p}\right) - 2\dot{\theta}\left(\dot{y} - y\frac{\dot{r}_c}{r_c}\right) = \varpi_x \\
&\ddot{y} + 2\dot{\theta}\left(\dot{x} - x\frac{\dot{r}_c}{r_c}\right) - y\dot{\theta}^2\left(1 - \frac{r_c}{p}\right) = \varpi_y \\
&\ddot{z} + z\dot{\theta}^2\frac{r_c}{p} = \varpi_z
\end{aligned} \tag{1}$$

where p, r_c and $\dot{\theta}$ are the semilatus rectum, orbit radius and true anomaly rate of the chief spacecraft, respectively. The acceleration disturbance vector $\boldsymbol{\varpi} \equiv [\varpi_x \varpi_y \varpi_z]^T$ is modeled as zero-mean Gaussian white-noise process with

$$E\left\{\boldsymbol{\varpi}(t)\boldsymbol{\varpi}^T(\tau)\right\} = \sigma_{\varpi}^2 \delta(t-\tau)\boldsymbol{I}_{3\times 3} \tag{2}$$

where $E\{\cdot\}$ denotes the expectation, the superscript T denotes the transpose, $\delta(t-\tau)$ is the Dirac delta function and $\boldsymbol{I}_{3\times 3}$ is a 3×3 identity matrix.