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Fault detection and isolation of satellite gyroscopes using relative positions in formation flying

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ABSTRACT

A fault detection and isolation method for satellite rate gyros is proposed based on using the satellite-to-satellite measurements such as relative position beside orbit parameters of the primary satellite. By finding a constant of motion, it is shown that the dynamic states in a relative motion are restricted in such a way that the angular velocity vector of primary satellite lies on a quadratic surface. This constant of motion is then used to detect the gyroscope faults and estimate the corresponding scale factor or bias values of the rate gyros of the primary satellite. The proposed algorithm works even in time variant fault situations as well, and does not impose any additional subsystems to formation flying satellites. Monte-Carlo simulations are used to ensure that the algorithm retains its performance in the presence of uncertainties. In presence of only measurement noise, the isolation process performs well by selecting a proper threshold. However, the isolation performance degrades as the scale factor approaches unity or bias approaches zero. Finally, the effect of orbital perturbations on isolation process is investigated by including the effect of zonal harmonics as well as drag and without loss of generality, it is shown that the perturbation effects are negligible.

1. Introduction

For the sake of high reliability and safety, spacecraft should tolerate the faults of their subsystems and components. Thus, fault detection and isolation (FDI) and consequently fault recovery algorithms are a part of mission management system on-board or off-board the spacecraft. However, modern space missions require the capability of handling faults with minimum ground support [1]. In a survey by Tafazoli [2] 156 on-orbit failures has been identified from 1980 to 2005 of which 40% were catastrophic. Attitude and orbit control subsystem (AOCS) caused more mission failures than any other subsystem (32% of the whole) and gyroscopes are the reason of most AOCS failures (17%).

FDI methods traditionally can be summarized in three major categories [3]; hardware redundancy based, signal processing based, and plausibility test. Hardware redundancy based FDI is the simplest and the most expensive solution. The high reliability and direct fault isolation are the most mentioned advantages of this method [4]. Nonetheless, there are cases as BeppoSAX or ERS2 that the spacecraft lost primary as well as spare gyroscopes over a period of 5 years [2]. The two other methods are more cost-effective than hardware redundancy. However, their main drawback is the need of high speed on-board computers that can run the fault diagnosis algorithms on-line. Nevertheless, some other objectives or constraints such as robustness, reactive detection, quick isolation, and limited onboard resources (CPU and memory) should be taken into account in the selection of the FDI strategy [1].

Many solutions for FDI problem of satellite gyros has been suggested in the literature. Beside gyroscopes, attitude sensors such as star trackers [5], sun sensors and Earth sensors [6–8] or redundant gyroscopes [9] can lead to FDI solutions. Most of these studies utilize different linear and nonlinear filtering approaches. Nonetheless, other approaches such as using conservation of angular momentum are also examined for gyroscope fault detection [10].

Great advantages of formation-flying (FF) have made it suitable for many space missions of NASA, Department of Defense, ESA and other space agencies [11]. Reducing the costs and increasing the flexibility of space missions are the most important advantages of using multiple satellites. FF missions can accomplish goals that are impossible or very difficult by a monolithic satellite [12]; missions such as PRISMA [13], TanDEM-X [14], and TerraSAR-X [15].

High-precision requirements in FF control strategies makes FDI more important in this kind of missions. FF satellites can use conventional FDI algorithms with/without utilizing their relative information. Actuator fault estimation in FF has been investigated

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by various methods. The concept of hierarchical architecture using a cooperative scheme is investigated in [16]. A dynamic neural network-based method using relative attitudes is presented in [17]. A hierarchical methodology using neural network-based scheme is investigated in [18]. Actuator FDI in a network of unmanned vehicles for different architectures is presented in [19]. Fault tolerant control in FF has been investigated in different approaches. Lee et al. have studied the use of GPS in estimating the relative positioning [20]. The use of RADAR sensor for measuring relative position, azimuth and elevation angle is investigated by Ilyas et al. [21]. Thanapalan et al. studied a redundancy based approach [22].

There exist many different approaches in the relative navigation (RN) of FF satellites. The RN can be done by the use of global positioning system (GPS) [23,24] for near Earth satellites or GPS-like technologies [25] for deep space missions. Satellite-to-satellite tracking (SST) methods [26] can be used for RN as well. SST can be attained using different kinds of measurements; range [27,28], range rate [29], line-of-sight vectors [30,31], and combinations of them [32–34]. Prior research also considers the dynamical behavior of satellites in FF including perturbations [35,36].

This paper deals with a novel FDI method that is based on relative equations of motion. In this proof-of-concept study, it is supposed that the relative position of the secondary satellite is measured in the primary satellite body frame. The first and second derivatives of relative position has been computed by a finite difference method of fourth order. A constant of motion is found which is independent of the absolute dynamical states of the secondary satellite. This constant of motion is used as the residual to be utilized for fault detection of the primary satellite gyroscopes. Moreover, some analytical formulas are found using this constant of motion for fault isolation and identification purposes.

The organization of this paper proceeds as follows. First, the constant of motion is derived that relates the rotational motion of primary satellite to its absolute translational motion and the relative dynamics. After that, a sensitivity analysis on the basic equation is presented. Next, fault determination process and the effect of thresholds on the detection of slight faults are analyzed. Next, fault isolation process and the proposed algorithm is described. Then, simulation results based on Monte-Carlo method for two dynamic scenarios and different faults are presented. Finally, the effect of perturbations on fault isolation for scale factors and biases are obtained.

2. Constant of motion

Consider two satellites (primary and secondary) flying in two different trajectories around the Earth (Fig. 1). The relative acceleration of the secondary satellite with respect to the primary satellite frame can be stated as [37]

$$\mathbf{a}_S^P = \mathbf{a}_S^O - \mathbf{a}_P^O - \dot{\boldsymbol{\omega}}^{PE} \times \mathbf{r}_{SP} - 2\boldsymbol{\omega}^{PE} \times \mathbf{v}_S^P - \boldsymbol{\omega}^{PE} \times \left(\boldsymbol{\omega}^{PE} \times \mathbf{r}_{SP}\right) \tag{1}$$

where $\mathbf{a}_S^O$ and $\mathbf{a}_P^O$ are the secondary and the primary satellites accelerations in an inertial coordinate system, respectively. They can be replaced by their universal gravity formulation $(-\mu\mathbf{r}/r^3)$ plus perturbation terms. The $\mathbf{r}_{SP}$ is the position vector of the secondary satellite relative to the primary and can be defined and measured in the primary body coordinate system. $\mathbf{v}_S^P$ is the time derivative of $\mathbf{r}_{SP}$ with respect to primary satellite frame. $\boldsymbol{\omega}^{PE}$ and $\dot{\boldsymbol{\omega}}^{PE}$ are the angular velocity and acceleration of the primary satellite body with respect to inertial coordinate system, respectively. By defining $\mathbf{f}(t)$, Eq. (1) can be simplified as follows:

$$\dot{\boldsymbol{\omega}}^{PE} \times \mathbf{r}_{SP} + 2\boldsymbol{\omega}^{PE} \times \mathbf{v}_S^P + \boldsymbol{\omega}^{PE} \times \left(\boldsymbol{\omega}^{PE} \times \mathbf{r}_{SP}\right) + \mathbf{f}(t) = 0 \tag{2}$$

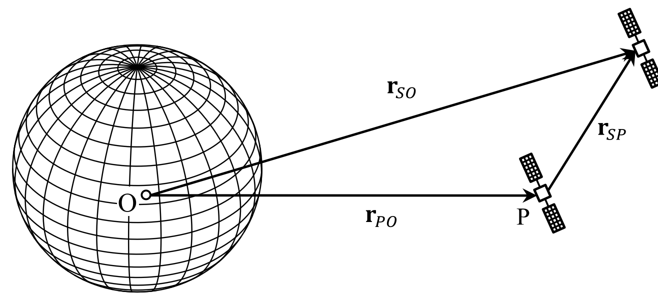

Fig. 1. Schematic of two orbiting satellites and their relative position.

Table 1
Definitions of constant of motion parameters.

Parameter	Definition
A	$-(r_{SP_y}^2 + r_{SP_z}^2) = r_{SP_x}^2 - r_{SP}^2$
B	$-(r_{SP_x}^2 + r_{SP_z}^2) = r_{SP_y}^2 - r_{SP}^2$
C	$-(r_{SP_x}^2 + r_{SP_z}^2) = r_{SP_z}^2 - r_{SP}^2$
D	$r_{SP_y} r_{SP_x}$
E	$r_{SP_z} r_{SP_y}$
F	$r_{SP_z} r_{SP_x}$
G	$r_{SP_z} v_{S_y}^P - r_{SP_y} v_{S_z}^P$
H	$r_{SP_x} v_{S_z}^P - r_{SP_z} v_{S_x}^P$
J	$r_{SP_y} v_{S_x}^P - r_{SP_x} v_{S_y}^P$
K	$\mathbf{r}_{SP}^{\mathrm{T}}\mathbf{f} = r_{SP_x} f_x + r_{SP_y} f_y + r_{SP_z} f_z$

where

$$\mathbf{f}(t) = \mathbf{a}_S^P + \frac{\mu}{|\mathbf{r}_{SP} + \mathbf{r}_{PO}|^3}(\mathbf{r}_{SP} + \mathbf{r}_{PO}) - \frac{\mu}{r_{PO}^3}\mathbf{r}_{PO} + \mathbf{f}_p(\mathbf{r}_{PO}, \mathbf{r}_{SP})$$

The perturbation term, $\mathbf{f}_p(\mathbf{r}_{PO}, \mathbf{r}_{SP})$, is a function of $\mathbf{r}_{PO}$ and $\mathbf{r}_{SP}$ that includes the effect of conservative perturbation accelerations. Effect of non-conservative perturbations are ignored here. Section 7 studies the effect of any ignored terms (including conservative and non-conservative perturbations) in the function of $\mathbf{f}(t)$. The superscript of $\boldsymbol{\omega}^{PE}$ is removed for simplicity i.e. $\boldsymbol{\omega}^{PE} \equiv \boldsymbol{\omega} = \left[\omega_x\ \omega_y\ \omega_z\right]^T$. Multiplying Eq. (2) by $\mathbf{r}_{SP}^T$ and writing the equation as a function of angular acceleration elements, the following constant of motion is obtainable:

$$\Psi = A\omega_x^2 + B\omega_y^2 + C\omega_z^2 + 2D\omega_x\omega_y + 2E\omega_y\omega_z + 2F\omega_z\omega_x + 2G\omega_x + 2H\omega_y + 2J\omega_z + K \tag{3}$$

Eq. (3) is a quadratic surface in terms of ω_x, ω_y and ω_z. Parameters A to J are defined in Table 1 and are functions of the relative position and velocity which can be measured or computed. Primary satellite absolute dynamic states are collected in K parameter.

Eq. (3) can be expressed in matrix form as:

$$\Psi = W^T \begin{bmatrix} \mathcal{G} & \boldsymbol{\beta} \\ \boldsymbol{\beta}^T & K \end{bmatrix} \qquad W = \boldsymbol{\omega}^T \mathcal{G}\boldsymbol{\omega} + 2\boldsymbol{\beta}^T\boldsymbol{\omega} + K \tag{4}$$

in which

$$W = \begin{bmatrix} \boldsymbol{\omega} \\ 1 \end{bmatrix} \qquad \mathcal{G} = \begin{bmatrix} A & D & F \\ D & B & E \\ F & E & C \end{bmatrix} \qquad \boldsymbol{\beta} = \begin{bmatrix} G \\ H \\ J \end{bmatrix}$$

The value of the scalar Ψ should be zero. Let us introduce the measured values by adding an accent mark Tilde ($\sim$). However, if the measured values of angular velocity and relative positions are used ($\widetilde{\boldsymbol{\omega}} = \boldsymbol{\omega} + \boldsymbol{\nu}$), this function may have nonzero values due to