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# Guidance and control for satellite in-orbit-self-assembly proximity operations

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ABSTRACT

In this paper, the autonomous in-orbit-self-assembly of multiple satellites is exploited in close proximity operations. Guidance and control algorithms are developed based on the closed-form analytical solution of relative motion equations that is completely explicit in time in a general Keplerian orbit. The guidance algorithm is based on a modified version of the inbound glideslope transfer that was used in the past for rendezvous and proximity operations of the space shuttle with other vehicles. The control algorithm is based on a discrete multipulse technique that was used to track the guidance trajectory efficiently while avoiding collisions between satellites during maneuvers. These algorithms are general, and can translate each satellite in the assembly in any direction and decelerate while approaching the desired target location in the final assembly configuration. Numerical nonlinear simulations that illustrate the performance and accuracy of the proposed algorithms are performed in a cubic formation assembly.


## 1. Introduction

Space organizations such as the European Space Agency (ESA), Japan Aerospace Exploration Agency (JAXA), and National Aeronautics and Space Administration (NASA) have identified the development of automated in-orbit-self-assembly as a key requirement for future space mission advances, including space interferometry emulating very large aperture telescopes, large inflatable and deployable antennas, solar sails and large solar power systems. Such a strategy enables the assembly of larger structures out of smaller components, which in turn overcomes the mass and volume limits associated with the fairing of many launchers. In addition, human presence for future space construction is impractical from risk and cost perspectives [15]. The interest in autonomous proximity assembly operations has increased with the recent demonstration of XSS-11, Demonstration of Autonomous Rendezvous Technology (DART), and Orbital Express. Autonomous assembly and proximity operations have also been demonstrated by the Japanese EST-VII and the Russian Progress vehicles. In addition, future missions to the ISS will require autonomous assembly and proximity operations [9,29]. This paper investigates the development of the guidance and control system for autonomous in-orbit-self-assembly of multiple satellites in close proximity.

The Clohessy–Wiltshire (CW) equations have been used extensively in the literature to describe satellite relative motion in close proximity [8]. The relative orbit coordinate distances were small compared to the reference orbit radius, so the resulting equation of motion was linearized. These equations were derived based on the two-body problem in the absence of gravitational perturbations and environmental forces (solar radiation pressure and atmospheric drag). Further attempts by Lawden [17], which were extended by Carter [6,7], included the eccentricity of the reference orbit. These attempts and the corresponding solutions were based on the Tschauner–Hempel (TH) equations [27], which use the true anomaly of the reference orbit as an independent variable instead of time. Melton [19], Broucke [4], and Yamanaka and Ankersen [30] derived the state transition matrices that reflect the effect of eccentricity implicitly in time. The current paper uses a novel formulation of the state transition matrix derived explicitly in time to analyze the relative motion between multiple satellites in close proximity more efficiently and rapidly than solving the exact nonlinear differential equations in the inertial coordinate system.

The technical difficulties of multiple autonomous spacecraft in close proximity are related to the development of robust and reliable guidance and control techniques [20]. The function of the guidance system is to provide trajectories from the actual location to the desired target location. Fuel optimal trajectories are highly desirable in order to conserve fuel, which is a key factor in the

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mission life time. Other desirable features need to be considered, such as collision avoidance, line of sight and plume impingement. Most common guidance algorithms used in the literature fall into three categories: the first generates fuel optimal trajectories using mixed-integer linear programing (MILP) to account for constraints like passive safety toward the end of the trajectory, plume impingement and even collision avoidance [3,25]; the second uses parametric programing and model predictive control (MPC), which solves for optimal control [24]; the third is called a glideslope algorithm, which is a simple but robust trajectory planning algorithm [11]. With respect to control algorithms, there are a variety of candidates based on either closed-loop linear control strategies such as the linear quadratic regulator (LQR), the LQR with artificial potential functions (LQR/APF), the $H_\infty$ approaches and the local time approximation method, or open-loop techniques such as iterative Lambert targeting, the closed form solution of Hill–Clohessy–Wiltshire (HCW) for circular orbits and the closed form solution of the TH equation for elliptical orbits to control the relative positions of multiple spacecraft [1,2,5,11–14,16,18,21–23,26,28]. This paper develops novel modified versions of the glideslope guidance transfer and employs multipulse discrete control techniques based on the analytical closed form solution of TH equations.

The literature mentioned above often takes into account fuel efficiency and complicated collision avoidance constraints for a wide range of rendezvous scenarios of two satellites in close proximity. Izzo [14] developed guidance and control techniques to assemble a set of identical elements for multi-satellite in-orbit-self-assembly. He utilized the theoretical results obtained by Gazi [10] to design the guidance system. In order to minimize the fuel consumption, he addressed the problem of decomposing the space in two different parts: curved space and flat space. For the curved space part, Gazi utilized the closed form solution of CW equations to generate the guidance path, but to avoid singularities associated with using the CW state transition matrix, he proposed a switching procedure to switch into the flat space part. Okasha [23] extended the approach by Izzo [14] for the eccentric orbit case using his proposed state transition matrix for the TH equations. In addition, he included algorithms to ensure collision avoidance during the maneuvers. The control systems used by Izzo [14] and Okasha [23] were based on continuous control techniques.

The objectives of this paper are as follows: (1) to motivate an analytical closed-form solution to the relative motion equations in the context of autonomous multi-satellite in-orbit-self-assembly close proximity operations; (2) to design collision-free autonomous guidance algorithms to assemble multiple satellites in a general orbit without dividing the space into two segments and removing singularities associated with the state transition matrix; (3) to develop a multipulse discrete control technique that can minimize the total fuel usage during the assembly maneuvers.

The analysis in the current paper is arranged as follows. Section 2 presents a brief description of the relative dynamic equations of motion for the chaser with respect to the target in a general orbit described by Gauss' variational equations. Section 3 describes how the general TH analytical solution of the linearized relative motion is obtained explicitly as a function of time in a rotating orthogonal coordinate frame fixed to the target in an arbitrary elliptic orbit. Section 4 develops the guidance and control algorithms that pertain to approaching the target autonomously with the help of a matrix formulation for a classical two-pulse rendezvous. Section 5 discusses the extensions of these algorithms for multi-satellite in-orbit-self-assembly. Section 6 examines the general applications of these algorithms through different numerical examples. The simulation results demonstrate the algorithms' brevity and accuracy. Finally, Section 7 provides conclusions and recommendations.

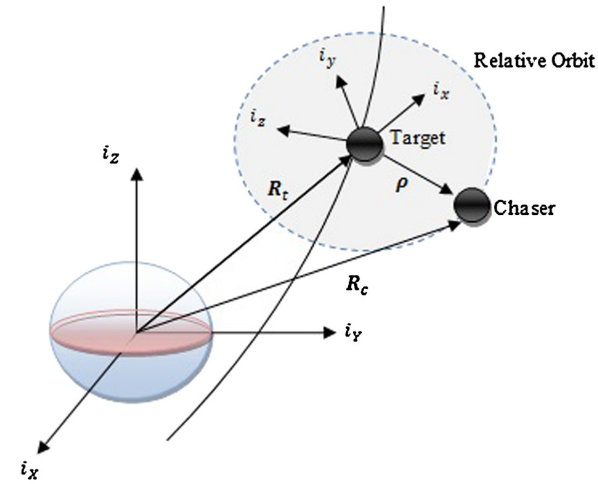


**Fig. 1.** Relative motion coordinates.

## 2. Relative motion dynamics

Relative motion is conveniently described in a Local-Vertical–Local-Horizontal (LVLH) frame that is attached to the target spacecraft and in an Earth-centered inertial (ECI) frame as shown in Fig. 1. The LVLH has an orthonormal basis $\{i_x, i_y, i_z\}$, where $i_x$ lies along the radius vector from the Earth's center to the spacecraft, $i_z$ coincides with the normal to the plane in the direction of angular momentum vector defined by the position and velocity vectors of the spacecraft, and $i_y = i_z \times i_x$. The ECI frame has an orthonormal basis $\{i_X, i_Y, i_Z\}$, where $i_X$ and $i_Y$ lie in the equatorial plane, $i_X$ coincides with the line of equinoxes, and $i_Z$ passes through the North Pole. The location of the chaser $\boldsymbol{R}_c$ is given as

$$\boldsymbol{R}_c = \boldsymbol{R}_t + \boldsymbol{\rho} \tag{1}$$

where $\boldsymbol{R}_t$ and $\boldsymbol{\rho}$ correspond to the location of the target and the position of the chaser spacecraft relative to the target, respectively. The vectors $\boldsymbol{R}_t$ and $\boldsymbol{\rho}$ can be expressed in the target LVLH reference frame as

$$\boldsymbol{R}_t = R i_x \tag{2}$$

$$\boldsymbol{\rho} = x i_x + y i_y + z i_z \tag{3}$$

where $x$, $y$, and $z$ denote the components of the relative position vector $\boldsymbol{\rho}$ along the radial, transverse, and out-of-plane directions, respectively, and $R$ is the magnitude of the target position vector $\boldsymbol{R}_t$. By use of kinematics, the most general equations modeling relative motion are given by the following expression [13,26]:

$$\ddot{\boldsymbol{\rho}} = [\mathbf{f}_c]^{\mathrm{LVLH}} - [\mathbf{f}_t]^{\mathrm{LVLH}} - 2\boldsymbol{\omega} \times \dot{\boldsymbol{\rho}} - \boldsymbol{\omega} \times (\boldsymbol{\omega} \times \boldsymbol{\rho}) - \dot{\boldsymbol{\omega}} \times \boldsymbol{\rho} \tag{4}$$

where $[\mathbf{f}_c]^{\mathrm{LVLH}}$ and $[\mathbf{f}_t]^{\mathrm{LVLH}}$ are the external acceleration forces acting on the chaser and the target, respectively, in the LVLH frame of the target vehicle. $\dot{()}$ and $\ddot{()}$ denote the first and second derivatives with respect to time. The target LVLH frame rotates with angular velocity $\boldsymbol{\omega}$, and its current orientation with respect to the inertial frame is given by the 3–1–3 direction cosine matrix comprising $\Omega$, $i$, $\omega$, and $f$, respectively (see Fig. 2) [21]. The angular velocity can also be expressed in terms of orbital elements and their rates as

$$\boldsymbol{\omega} = \begin{pmatrix} s_{\omega+f} s_i \dot{\Omega} + c_{\omega+f} di/dt \\ 0 \\ c_i \dot{\Omega} + (\dot{\omega} + \dot{f}) \end{pmatrix} \tag{5}$$

where $s_{(\cdot)} = \sin(\cdot)$ and $c_{(\cdot)} = \cos(\cdot)$. The external forces acting on the target and chaser vehicles are assumed to be given by