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Optimal satellite formation reconfiguration using co-evolutionary particle swarm optimization in deep space

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ABSTRACT

This paper proposes a method that plans energy-optimal trajectories for multi-satellite formation reconfiguration in deep space environment. A novel co-evolutionary particle swarm optimization algorithm is stated to solve the nonlinear programming problem, so that the computational complexity of calculating the gradient information could be avoided. One swarm represents one satellite, and through communication with other swarms during the evolution, collisions between satellites can be avoided. In addition, a dynamic depth first search algorithm is proposed to solve the redundant search problem of a co-evolutionary particle swarm optimization method, with which the computation time can be shorten a lot. In order to make the actual trajectories optimal and collision-free with disturbance, a re-planning strategy is deduced for formation reconfiguration maneuver.

1. Introduction

The concept of satellite formation has gained growing interest in recent years because of the potential advantages of formation flying missions over the use of a single large satellite. Formation reconfiguration aims to plan optimal translational trajectories, with which satellites are repositioned from their current states to desired final states based on a performance index in a given time interval [1]. In ddition, the collision avoidance and control input limits should also be taken into consideration during the optimization.

The formation reconfiguration literature can be categorized as planetary orbital environment (POE) missions, where satellites are subject to orbital dynamics and environmental disturbances, and deep space missions (gravity-free environment). For POE missions, most algorithms work off-line because the reconfiguration can be completed after several orbital periods. Autonomous formation flying is important in deep space missions, and thus the reconfiguration algorithm for deep space missions should be simple enough to run onboard and plan trajectories quickly. However, the collision avoidance constraints usually result in a non-convex feasible solution space. Also, the reconfiguration problem with collision avoidance constraints is NP-complete [1], which makes the problem difficult to solve. With these problems, the aforementioned methods suffer because as the number of satellites or collocation points increases, the computational complexity increases greatly.

Many formation reconfiguration algorithms have been proposed in the reconfiguration literature. Mixed integer linear programming (MILP) method is a popular method, where the system model and constraint formulations are simplified to a linear form [2]. The MILP method can find a global optimum; however, the branch and bound algorithm is required to solve this problem, which dramatically increases the computation time when the number of satellites or computation steps is increased. Singh and Hadaegh [3] used polynomials of a variable order in time

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to parameterize the trajectories. A similar method was also used in [4], the trajectories were first discretized in time using a cubic spline, and then a feasible MILP method issued to calculate the variables at discretized points.

The pseudospectral method is a newly developed class of methods for solving optimal control problems. In the pseudospectral method, the state and control vectors are discretized at specified points using a structure of global orthogonal polynomials, which makes the optimal control problem easy to solve with high accuracy. The method has been used in nonlinear satellite trajectory optimization problems. Aoude and How [5] proposed a two-stage path planning approach to reconfiguration maneuver, where the rapidly exploring random tree (RRT) method was used first to find a feasible initial solution, and then nonlinear programming(NLP) techniques were used to solve the optimal control problem discretized by the Gauss pseudospectral method (GPM).Huntington and Rao [6] directly transformed the tetrahedral formation reconfiguration problem into NLP using GPM, but collision avoidance was not considered. Wu et al. [7] used the Legendre pseudospectral method (LPM) to design fuel-optimal trajectories for satellite reconfiguration in near-earth orbit with an exact nonlinear relative satellite dynamic model.

Intelligent Optimization Method is also an efficient way to address the reconfiguration problem. Huang et al. [8] used the particle swarm optimization method to solve this problem which separated by discretized by LPM. Sun et al. [9] proposed a closed-loop brain storm optimization algorithm for satellite reconfiguration in near-earth orbit using two-impulse control. Wang and Zheng [10] presented a hierarchical evolutionary trajectory planning method, which contained two level planners to solve the formation reconfiguration problem.

Many other approaches have also been used in formation reconfiguration. Guibout and Scheeres [11] used a semi-analytic approach to solve the two-point boundary-value problem. Sultan et al. [12] developed a gradient-based algorithm that analytically solve the problem. Massari and Bernelli-Zazzera [13] transformed the problem into a NLP problem with a parallel multiple-shooting method. Gong et al. [14] proved that the Beginning–Ending method is the energy optimal method of all bi-impulse methods for a short time reconfiguration. Garcia-Taberner et al. [15] proposed a FEFF (finite elements for formation flight) method to deal with the nominal reconfiguration maneuvers.

Up to now, all the relative literatures just focus on the trajectory generation before satellites maneuvering for reconfiguration. Most methods calculate the reconfiguration problem just focus on collision avoidance at the discrete points, thus the fitted trajectories may not be the real trajectories, so the satellites have the possibility to collide with each other. Even with the collision-free trajectories, satellites still could collide with each other which is caused by external disturbance when the satellites are controlled separately. To solve this problem, Campbell [16] and Slater et al. [17] calculated the collision probability for maneuver considering relative position, relative velocity and uncertainty. Collision could be avoided with this method, but it makes satellites deviate from the given trajectories, thus the index consumption would be more than the desired one.

In this study, a novel method is developed to plan energy-optimal trajectories of reconfiguration maneuvers for multi-satellite formations in a deep space environment real-time, where the satellites use continuous low-thrust control inputs. The satellites are modeled as points of constant mass. Normally, the maneuver time is short, and the propulsion systems used for maneuvering are efficient; thus, the mass of each satellite is assumed to be constant during reconfiguration.

This paper is organized as follows. Section 2 describes the problem statement. Section 3 describes how to discretize the reconfiguration problem using LPM. Section 4 employs a modified particle swarm optimization (PSO) method to solve the nonlinear programming problem, wherein the modified maximum velocity and the boundary constraints are also mentioned. Section 5 proposes a novel co-evolutionary particle swarm optimization (CPSO) algorithm to solve the reconfiguration problem. Section 6 presents three numerical examples to illustrate the performance of this new algorithm. Section 7 concludes the paper.

2. Problem formulation

2.1. Problem statement

For deep space formation reconfiguration missions, the satellite dynamics can be reduced to a double integrator form [18]. Scharf et al. [19] proved that this model can be accurate to 1 cm for up to 4.5 h for an earth-trailing orbit under some assumptions. Miller and Campbell [20] proved that this model can be used for 2 h near the L2 point. In this paper, M satellites are assumed to take synchronous maneuvers within the same time interval $[0, T]$. The system dynamics can be stated as follows:

$$\dot{\boldsymbol{X}}_l(t) = \boldsymbol{A}\boldsymbol{X}_l(t) + \boldsymbol{B}\boldsymbol{U}_l(t), l = 1, 2, ..., M, t \in [0, T] \tag{1}$$

where

$$\boldsymbol{X}_l = [x_l, y_l, z_l, \dot{x}_l, \dot{y}_l, \dot{z}_l]^T, \boldsymbol{U} = [u_{x_l}, u_{y_l}, u_{z_l}]^T$$
$$\boldsymbol{A} = \begin{bmatrix} \boldsymbol{0}_{3\times3}, & \boldsymbol{I}_{3\times3} \\ \boldsymbol{0}_{3\times3}, & \boldsymbol{0}_{3\times3} \end{bmatrix}, \boldsymbol{B} = \begin{bmatrix} \boldsymbol{0}_{3\times3} \\ \boldsymbol{I}_{3\times3} \end{bmatrix}$$

$\boldsymbol{X}_l(t)$ and $\boldsymbol{U}_l(t)$ are the state and control vectors of the lth satellite at time t respectively.

The control inputs are assumed to be continuous low-thrust forces confined within specified limits:

$$-u_{\max} \le u_{l,d}(t) \le u_{\max}, \quad l = 1, 2, ..., M, \quad d = 1, 2, 3 \tag{2}$$

The initial and final states are constrained with the following conditions:

$$\begin{aligned} \boldsymbol{X}_{l0} &= \boldsymbol{X}_{lS} \\ \boldsymbol{X}_{lT} &= \boldsymbol{X}_{lF} \end{aligned} \tag{3}$$

where $\boldsymbol{X}_{lS}$ and $\boldsymbol{X}_{lF}$ are the initial and final state vectors of the lth satellite.