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High-performance three-dimensional maneuvers control in the area of spacecraft

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ABSTRACT

Contemporary research is improving techniques to maneuvers control in the area of spacecraft. In the aspect of further development of investigations, a high-performance strategy of maneuvers control is proposed in the present research to be applicable to deal with a class of the aforementioned spacecrafts. In a word, the main subject behind the research is to realize high-performance three-dimensional orbital maneuvers to transfer the system under control from the initial orbit to the corresponding final ones, in finite burn, while the referenced trajectories are provided through an optimization approach. It is shown that two sets of actuators such as the reaction thrusters should be handled through the proposed control strategy, as long as the x-axis in connection with the y, z axes are dealt with through low and high-thrust control channels, respectively. It aims the strategy to manage the orbital parameters to be accurately varied with respect to time, in order to complete the procedure of transferring, in finite burn. A series of experiments in association with a number of benchmarks are considered to verify the performance of control strategy, tangibly. The contribution of the research is to organize an integration of the trajectory optimization and the new structure of the control scheme to deal with the orbital maneuvers, in finite burn, where the low and the high-thrust controls are synchronously compounded to present high-performance of the investigated outcomes.

KEYWORDS

High-Performance Three-Dimensional Maneuvers Control, Trajectory Optimization, Spacecraft, Orbital Parameters, Finite Burn.

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1. Introduction

Up to now, the performance of a class of spacecrafts is worthy of investigation to be improved, due to the fact that a number of state-of-the-art techniques in the area of modeling and its control focus on new insights to deal with the aforementioned complicated processes, efficiently. With this goal, the present research is organized to handle the present spacecrafts to be applicable in the process of three-dimensional orbital maneuvers. There are a number of orbital parameters including the semi-major axis, the eccentricity, the apogee radius, the inclination, the argument of perigee and finally the right ascension of ascending node that are needed to manage with respect to time through the efficient strategy. In fact, the steering angles in the form of n^{th}order polynomials should be adjusted, at each instant of time, in its accurate manner.

In the research proposed here, the high-performance three-axis control strategy is realized to track the three-axis referenced trajectories that are directly provided by the aforementioned n^{th}order polynomials, while the dynamics of the spacecrafts under control are taken into real consideration. It is provided by realizing an optimization technique such as the genetic algorithm, the simulated annealing method and so on to generate the whole of the coefficients of the same n^{th} order polynomials, instantly. Subsequently, the proposed three-axis control strategy is implemented to handle the procedure of transferring the same systems under control from each initial orbit to its final ones, accurately, in finite burn. Two sets of actuators such as the reaction thrusters need to drive through the proposed one, as long as the x-axis in connection with the y, z axes are dealt with via two separated control channels.

The backgrounds of the research are twofold. One of them is related to the space trajectory optimization and other is also related to the spacecrafts control, as well. With a focus on the first one, Curtis et al. present the fundamental information regarding the orbital mechanics that are so useful to consider in the area of orbital maneuvers, at first [1].

In one such case, Xiao-yong et al. propose an approach based on the virtual gravitational body for low-thrust trajectory optimization, where the original optimal control problem is converted into a nonlinear programming method that could be solved fast owing to the analytical nature of conic arcs [2]. As another attempt, Xiao-yong et al. present a multi-impulse extended method for low-thrust trajectory optimization. There is the multi-impulse trajectory that can be converted to the low-thrust one by applying to all the other segments [3]. Dai et al. propose the optimal trajectory generation to establish connectivity in proximity networks. The problem of designing the optimal trajectories is examined to establish its connectivity in the present network regarding initially scattered dynamic agents [4]. Tian et al. suggest real-time trajectory and attitude coordination control in the area of reusable launch vehicle in reentry phase. The pseudo spectral-based optimal feedback reentry guidance is realized using the successive real-time optimal open-loop control [5].

With a focus on the second one, Sidi presents the dynamics and control of the spacecrafts, fundamentally. Those obtained from the outcomes quote there are traditional and also efficient to deal with the spacecrafts under control [6]. Zheng et al. suggest an autonomous attitude coordinated control, while Du et al. consider the attitude synchronization control for a class of flexible spacecrafts to resolve the problem of attitude synchronization [7]-[8]. Yang et al. investigate the attitude determination and control through the quaternion based method. It is to note that the present quaternion based method is discussed to cope with attitude determination in connection with the corresponding control approach [9].Zou et al. research work is to address the adaptive fuzzy fault-tolerant attitude control, once the mass moment of inertia matrix, the external disturbances, the actuator failures and the input constraints are taken into consideration [10]. Reyhanoglu et al. explore orbital and attitude control around an asteroid, while Sun et al. focus on robust adaptive control for proximity maneuvers under dynamic coupling and uncertainty [11]-[12].

Cai et al. research work is to deal with the leader-following attitude control of multiple rigid spacecrafts, where Kuo et al. research work is to handle the attitude dynamics and control of miniature one via the pseudo-wheels[13]-[14]. Erdong et al. propose the robust decentralized attitude coordination control, where Bustan et al. work is in the robust fault-tolerant tracking control design under input saturation [15]-[16]. Lu et al. design the control strategy for the rigid attitude tracking with actuator saturation, where the adaptive sliding mode control relative to the motion of system under input constraints is given by Wu et al. [17]-[18]. Furthermore, the realization of attitude control, in a different form, is presented by Butyrin et al. [19]. Yongqiang et al. suggest time-varying sliding mode attitude tracking control strategy in rigid spacecraft [20].

Regarding the control allocation research, Johansen et al. present a survey to address this issue. With this purpose, the subject of control allocation suggests the advantage of a modular design, as long as a high-level motion control algorithm is designed and there are no detailed knowledge about the effectors and actuators [21]. Servidia research is to deal with control allocation for gimbaled-fixed thrusters, which are using a distribution law between the controller and the actuators [22].

Consideration of the above-referenced techniques indicates that each one of them tries to address the constructive approach, in its different situations. In making an effort to design the control method in the same way, the proposed strategy is investigated, where the integration of the trajectory optimization and the new structure of the control idea to deal with the orbital maneuvers is of the novelty. The accurate outcomes, which are provided through a series of experiments, finalize the strategy to be applicable in this area, while a number of benchmarks can verify the competitive results, as well.

The rest of the manuscript is organized as follows: The proposed control strategy is first presented in Section 2, while the simulation results are then given in Section 3. The investigated outcomes of the research are finally concluded in Section 4.

2. THE PROPOSED CONTROL STRATEGY

The proposed control strategy is illustrated in Fig. 1, which is designed based upon the double closed loops including the inner and the corresponding outer loops, as well. There is another loop to realize the orbital module (ORM) and the corresponding optimization module (OPM) to deal with the spacecraft under control $(SPCSYS)$, overally. It should be noted that the outer closed loops of the present strategy are organized to handle the three-axis rotational angles, while the x −axis rotation is related to the low-thrust mode and the y, z −axis rotations are related to the high-thrust mode, as well. The responsibility of the both outer closed loops are to adjust the three-axis system rotational angles (SRA_{xyz}) to track the three-axis referenced rotational angles (RRA_{xyz}). Furthermore, the responsibility of the both inner closed loops are to adjust the three-axis system angular rates (SAR_{xyz}) to deal with the three-axis angular rate errors to be zero in steady state. In this strategy, the four control approaches are designed, while two of them including the x −axis rotational angle control in the low-thrust mode (RAC_{Lx}) and the y, z −axisrotational angles control in the high-thrust mode (RAC_{Hyz}) are designed in the outer closed loops.

Hereinafter, the rest of them including the x-axis angular rate control in the low-thrust mode (ARC_{Lx}) and the y, z −axis angular rates control in the high-thrust mode (ARC_{Hyz}) are designed in the inner closed loops. It is noted that RAC_{Lx} and RAC_{Hyz} are realized based upon the proportional-integral-derivative approach, while ARC_{Lx} and ARC_{Hyz} are realized based on the proportional approach.Due to the fact that the proposed control strategy does not possess the linear relationship between the input to the control approach and its output torque, the pulse-width pulse-frequency $(PWPF)$ modulators are needed to

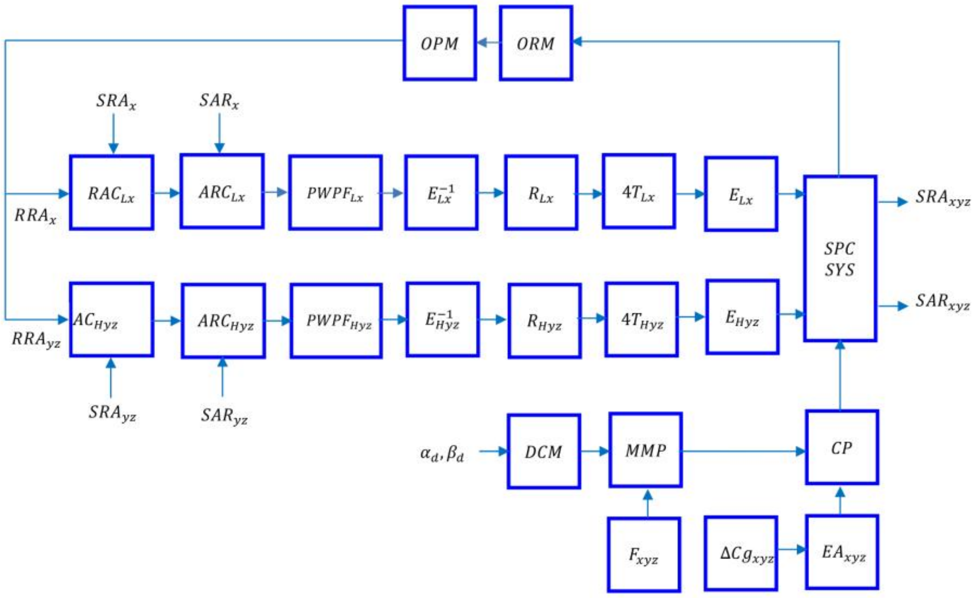

Fig. 1. The schematic diagram of the proposed control strategy

relate between the level of required torque and the width as well as the frequency of pulses, in order to shape the non-linear output of on-off thrusters into linear request output. It can be shown that there is a total of the eight actuators including the reaction thrusters, where the four low thrusters $\boldsymbol{T_{Lx}}$ including $T_{Lxi}; i = 1,2,3,4$are employed to deal with the $SPCSYS$in the $x-$axis in the low-thrust mode, while the four high thrusters $\boldsymbol{T_{Hyz}}$ including $T_{Hyzi}; i = 1,2,3,4$ are employed to deal with the same $SPCSYS$ in the $y, z-$ axis in the high-thrust mode, as notated in the proposed strategythrough$4T_{Lx}$ and $4T_{Hyz}$. Now,$\boldsymbol{E_{Lx}}$and$\boldsymbol{E_{Hyz}}$matrices in the low and the high-thrust modes are taken as

$$\begin{cases} \boldsymbol{E_{Lx}} = [-R \quad -R \quad R \quad R] \\ \boldsymbol{E_{Hyz}} = \begin{bmatrix} 0 & -R & 0 & R \\ R & 0 & -R & 0 \end{bmatrix} \end{cases} \quad (1)$$

where $R$is given as the thruster's arm. Based upon the information presented, the modules of $\boldsymbol{E_{Lx}^{-1}}$ and $\boldsymbol{E_{Hyz}^{-1}}$ are taken as the corresponding pseudo inverse of $\boldsymbol{E_{Lx}}$and$\boldsymbol{E_{Hyz}}$,respectively.Correspondingly, the three-axis control torques in the both low and high-thrust modes are to be calculated as

$$\begin{cases} \tau_{L_x} = \boldsymbol{E_{Lx}}\, \boldsymbol{T_{Lx}}^T \\ \boldsymbol{\tau_{H_{yz}}} = \boldsymbol{E_{Hyz}}\, \boldsymbol{T_{Hyz}}^T \end{cases} \quad (2)$$

In one such case, the three-axis control torques are applied to the spacecraft, presented by [6]

$$\frac{d(SAR_x(t))}{dt} = \frac{\tau_{L_x}(t)}{I_x} - \frac{I_z - I_y}{I_x} SAR_y(t) SAR_z(t)$$

$$\frac{d(SAR_y(t))}{dt} = \frac{\tau_{H_y}(t)}{I_y} - \frac{I_x - I_z}{I_y} SAR_x(t) SAR_z(t)$$

$$\frac{d(SAR_z(t))}{dt} = \frac{\tau_{H_y}(t)}{I_z} - \frac{I_y - I_x}{I_z} SAR_x(t) SAR_y(t)$$

$$\frac{d(SRA_x(t))}{dt} = SAR_x(t) + \tan(SRA_y(t)) \sin(SRA_x(t))\, SAR_y(t) + \tan(SRA_y(t)) \cos(SRA_x(t))\, SAR_z(t) \quad (3)$$

$$\frac{d(SRA_y(t))}{dt} = \cos(SRA_x(t))\, SAR_y(t) - \sin(SRA_x(t))\, SAR_z(t)$$

$$\frac{d(SRA_z(t))}{dt} = \frac{\sin(SRA_x(t))}{\cos(SRA_y(t))} SAR_y(t) + \frac{\cos(SRA_x(t))}{\cos(SRA_y(t))} SAR_z(t)$$

where I_x, I_y and I_z are taken as the $x-$axis, $y-$axis and the $z-$axismoments of inertial of the system, respectively. A three-axis relay includingR_{Lx} and R_{Hyz} should be realized to provide a sequence of $0 - T_{Lxi}$ and $0 - T_{Hyzi}$ to enable the reaction thrusters, appropriately, with ε_{Lx} and ε_{Hyz}, respectively. Now, it is to note that the propellant engine disturbances are provided in connection with the engine misalignments as notated by α_d and β_d. In considering the direction cosine matrix(DCM) of the engine misalignments, it is possible to calculate the same engine disturbances by focusing on the thrust vector$(\boldsymbol{F_{xyz}})$to be multiplied with the investigated ones through the matrix multiplications (MMP) in the three axes, while by varying the engine arm (EA_{xyz}) through the variation of the center of the mass in the same three axes (ΔCg_{xyz}), the outcomes are finally calculated, provided that the engine arm and the three-axis engine force misalignments are multiplied through the cross product module (CP).

To consider the ORM, the spacecraft position is represented along with the Kepler's equation by [1]

$$\ddot{\boldsymbol{r}}(t) = -\frac{\mu}{r^3(t)} \boldsymbol{r}(t) + \frac{\boldsymbol{F}(t)}{m(t)} \quad (4)$$

where $\boldsymbol{r}(t)$is taken as the orbital position of the spacecraft regarding the inertial reference frame, $r(t)$is taken as the scalar value of $\boldsymbol{r}(t)$, $\mu = 398600$ is taken as the constant gravitational parameter of earth and finally $m(t)$ denotes the mass of the spacecraft. Now, the thrust vector, relative to the inertial coordinate system, is taken as

$$\boldsymbol{F_{xyz}}(t) = F_{xyz}(t) \begin{bmatrix} cos(RRA_y(t)) \cos(RRA_z(t)) \\ cos(RRA_y(t)) \sin(RRA_z(t)) \\ -\sin(RRA_y(t)) \end{bmatrix} \quad (5)$$

where $F_{xyz}(t)$ is taken as the magnitude of the aforementioned thrust vector and there is

$$\begin{cases} RRA_x = 0 \\ RRA_y = -\alpha_d \\ RRA_z = \beta_d \end{cases} \quad (6)$$

The aforementioned ORM is organized in connection with a number of independent parameters, related to the Keplerian orbit including the semi-majoraxis $(a_{or}(t))$, the eccentricity $(e_{or}(t))$ and the inclination(i_{or}), respectively, and so on. The angular momentum vector is calculated through the cross product of the distance and the corresponding rate by

$$\boldsymbol{h}(t) = \boldsymbol{r}(t) \times \boldsymbol{v}(t) \quad (7)$$

where its scalar value is calculated through the dot product of the angular momentum vectors, i.e. $h^2(t) = \boldsymbol{h}(t).\boldsymbol{h}(t)$. The inclination is given as

$$i_{or}(t) = cos^{-1}\left(\frac{h_z(t)}{h(t)}\right) \quad (8)$$

Here, $h_z(t)$ indicates z-axis angular momentum. The eccentricity is presented as

$$\boldsymbol{e}(t) = \frac{1}{\mu}\left(v^2(t) - \frac{\mu}{r(t)}\right)\boldsymbol{r}(t) - \frac{1}{\mu}r(t)v_r(t)\boldsymbol{v}(t) \tag{9}$$

where $v_r(t)$ is taken as $\frac{\boldsymbol{v}(t).\boldsymbol{r}(t)}{r(t)}$. The eccentricity, in its scalar form, is given as

$$e_{or}^2(t) = \boldsymbol{e}(t).\boldsymbol{e}(t) = 1 + \frac{h^2(t)}{\mu^2}\left(v^2(t) - \frac{2\mu}{r(t)}\right)^2 \tag{10}$$

To consider the OPM, the coefficients of the referenced steering angles commands, i.e. $A_c^i, B_c^i; i = 1,2,\dots,n$ to determine $RRA_{xyz}(t)$ should be generated, which are presented through the n^{th} order polynomials, in an off-line manner, by using the genetic algorithm (GA). In a word, the main objective to realize the aforementioned algorithm is that the whole of the coefficients of the same referenced steering angles' commands can accurately be calculated to achieve the best performance of the maneuvers control with a focus on the tracking errors of the system under control. With this goal, a number of parameters including the population size, the maximum generations, the crossover points, the crossover probability and the mutation probability need to be appropriately chosen to calculate the optimum values for these coefficients of the referenced steering angles commands, as long as the GA can successfully be terminated at the limited number of theiterations. Now, the aforementioned n^{th} order polynomials are given by

$$\begin{cases} RRA_x(t) = 0 \\ RRA_y(t) = -\sum_{i=0}^{n} A_c^i t^i \\ RRA_z(t) = \sum_{i=0}^{n} B_c^i t^i \end{cases} \tag{11}$$

It should be noted that the fitness function can be organized based upon the orbital parameters to be optimized. In one such case, the present fitness function needs to be satisfied in the best value and the corresponding mean value of the coefficients of the referenced steering angles commands to be covered at the limited number of the iterations. Moreover, the behavior of the fitness function versus the variation of the mentioned coefficients of the referenced steering angles commands may synchronously be considered to indicate that the GA is able to be converged to reach the reliable information with respect to time. There are two orbital parameters including the inclination and the eccentricity to be chosen to realize the present function by the following, while the scaling factors including $SF_i; i = 1,2$ are usedto weight the effectiveness of the items to be considered

$$J(t) = SF_1(i_{or}(t) - i_{ord})^2 + SF_2(e_{or}(t) - e_{ord})^2 \tag{12}$$

where i_{ord} and e_{ord} denote the desirable values for the inclination and the eccentricity, respectively. And $SF_1 = \frac{w_1}{|i_{ori} - i_{ord}|}$ and $SF_2 = \frac{w_2}{|e_{ori} - e_{ord}|}$ are taken, where i_{ori} and e_{ori} denote the values for the inclination and the eccentricity in the initial orbit and $w_i; i = 1,2$ are taken as the constant weights.

3. THE SIMULATION RESULTS

To carry out the proposed control strategy, the orbital position and the corresponding rate are first taken by

$$\begin{cases} \boldsymbol{r}(0) = [6.580 \quad -0.933 \quad -1.332]^T\ km \\ \dot{\boldsymbol{r}}(0) = [1.840 \quad 4.270 \quad 6.098]^T\ km/s \end{cases} \tag{13}$$

In such a case, the mission operation plan in connection with a range of variations of the orbital parameters including the inclination and the eccentricity are tabulated in Table 1.

TABLE 1. THE ORBITAL TRANSFER MISSION

	The parameters	*The initial values*	*The final values*
1	$i(t)$	55.0 $deg.$	50.0 $deg.$
2	$e(t)$	0	0.30

The GA is realized to minimize the fitness function of Eq. (12), while $SF_1 = \frac{1}{5}$ and $SF_2 = \frac{1}{0.3}$ are taken. And Table 2 indicates the GA parameters.

TABLE 2. THE GA PARAMETERS.

	The parameters	*The values*
1	*Population size*	25
2	*Maximum generations*	80
3	*Crossover probability*	0.80
4	*Mutation probability*	0.01
5	*Crossover points*	2.0

The GA is successfully terminated at about 79^{th} generation, where the present fitness function is satisfied in the best value to be taken as 0.00172 and the corresponding mean value to be taken as 0.00124, whereits variation is illustrated in Fig. 2.

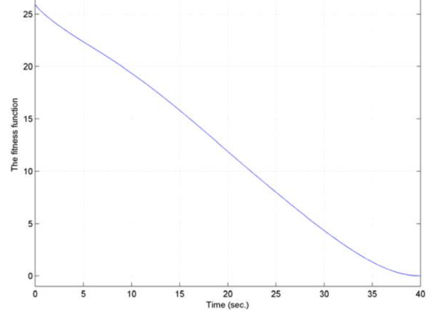

Fig. 2. The variation of the fitness function in the orbital maneuver

The optimum values for the coefficients of the referenced rotational angles, presented in Eq. (11), are completely satisfied by the following at about 1975^{th} iteration.

$$\begin{aligned} &A_c^0 = 22.0, A_c^1 = 3.315642, A_c^2 = -0.481685, A_c^3 = 0.025729, A_c^4 = -0.000623, A_c^5 = 5.525100e-6 \\ &A_c^0 = 120.0, A_c^1 = 3.311253, A_c^2 = -0.461003, A_c^3 = 0.025005, A_c^4 = -0.000603, A_c^5 = 5.000200e-6 \end{aligned} \tag{14}$$

The variations of the aforementioned orbital parameters with respect to time in association with the predefined mission are illustrated in Fig. 3-4, respectively, while the span of the three-axis thrust vector are taken as

$$\boldsymbol{F_{xyz}} = [0 - 8350 \quad 0 \quad 0]^T N \tag{15}$$

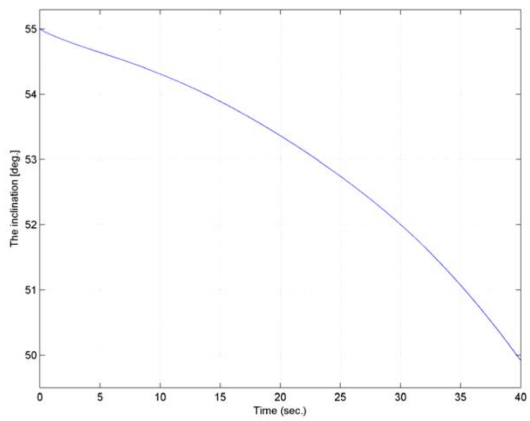

Fig. 3. **Variation of the inclination in the orbital maneuver**

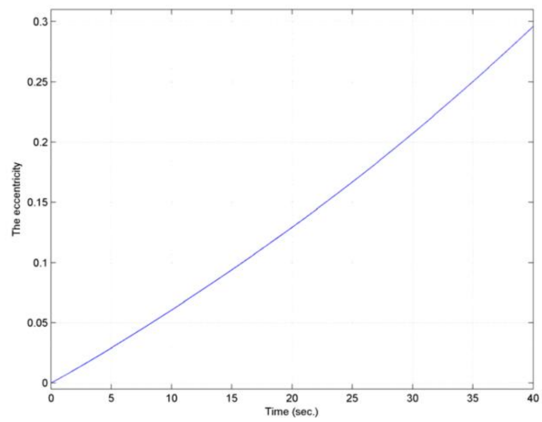

Fig. 4. **The variation of the eccentricity in the orbital maneuver**

The values of the parameters of the sub-systems, realized in the procedure of carrying out the proposed control strategy, are tabulated in Table 3. It is to note that the values of the parameters in the rows 1-4 of the present Table are provided through the Matlab optimization toolbox.

TABLE 3. THE VALUES OF THE PARAMETERS OF THE PROPOSED CONTROL STRATEGY

	The sub-systems	*The values of the parameters*
1	RAC_{Lx}	[8.86 52.42 20.0]
2	RAC_{Hyz}	$\begin{cases} [30 \quad 0 \quad 25.83] \\ [48.88 \quad 0 \quad 14.64] \end{cases}$
3	ARC_{Lx}	15
4	ARC_{Hyz}	[5000 0 6105]
5	$T_{Lxi}; i = 1,2,3,4$	600 N
6	$T_{Hyzi}; i = 1,2,3,4$	150 N
7	$\varepsilon_{Lx} = \varepsilon_{Hyz}$	0.1
8	R	0.5 m

Now, RRA_{xyz} are provided via Eq. (11) and then the three-axis tracking results of SRA_{xyz} in line with RRA_{xyz} are respectively illustrated in Figs. 5-7, while the initial values of them are taken by

$$\begin{cases} SRA_x(0)=0° \\ SRA_y(0)=-20° \\ SRA_z(0)=110° \\ RRA_x(0)=0° \\ RRA_y(0)=-22° \\ RRA_z(0)=120° \end{cases} \tag{16}$$

Finally, the tracking errors of SRA_{xyz} in line with RRA_{xyz} is illustrated in Fig. 8, as well.

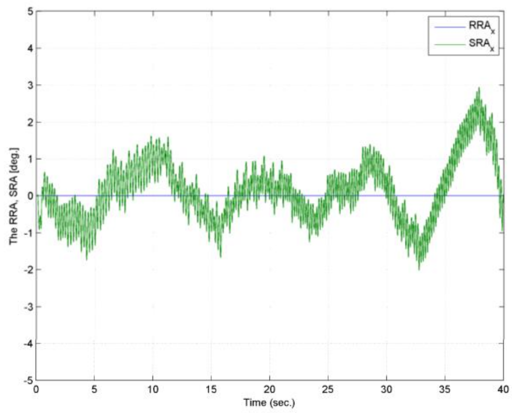

Fig. 5. **The tracking of SRA_x in line with RRA_x**

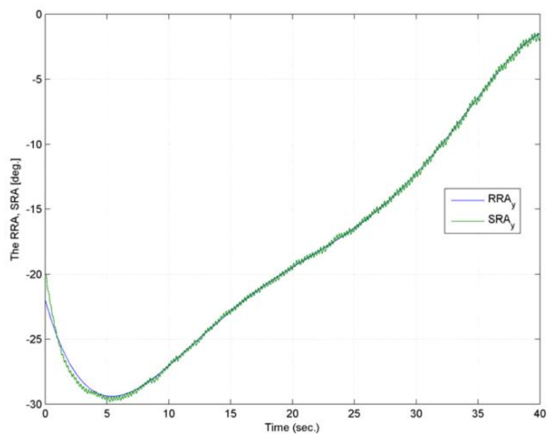

Fig. 6. **The tracking of SRA_y in line with RRA_y.**

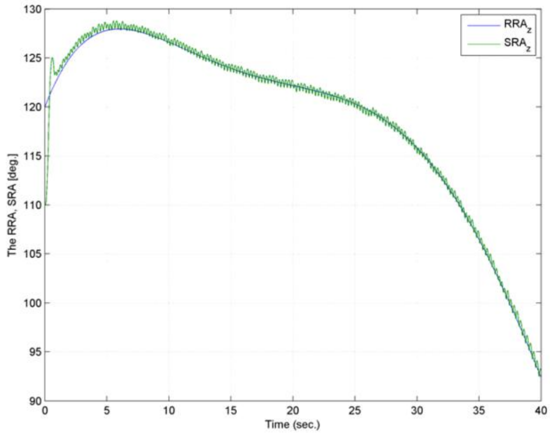

Fig. 7. **The tracking of SRA$_z$ in line with RRA$_z$**

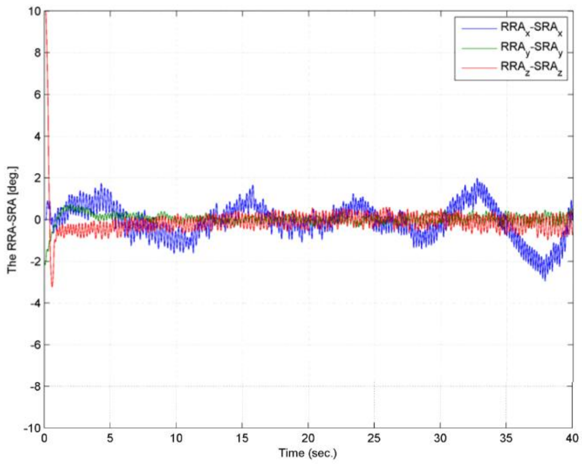

Fig. 8. **The tracking errors of SRA$_{xyz}$ in line with RRA$_{xyz}$**

A. The Verification of the Proposed Control Strategy

To verify the proposed control strategy, a number of the criteria including (1) the maximum three-axis SRA in steady state, (2) the maximum three-axisSARin steady state and finally (3) the trajectory convergence time are considered in Table 4, which are all important to evaluate the proposed strategy performances in connection with the Yang, the Reyhanoglu and the Sunbenchmarks, respectively. Regarding the Yang approach, a presentation of the system modeling, the attitude determination and its control is investigated. In one such case, the quaternion based methods in this area are considered by using the Newton's method, as long as the analytic solution to Wahba's problem is taken into account. In designing the quaternion based control system, the reduced models that realize in line with the vector component in the state space models are resulted, where some features are included. These ones are listed as the analytic solution of the linear quadratic regulator and the designed system to deal with the disturbances and the uncertainties. Regarding the Reyhanoglu approach, the dynamics formulation is considered to realize the control approach for the asteroid orbiting space system. It is to note that the control objective is based upon the nadir pointing attitude on the circular equatorial orbit, while the Lyapunov-based nonlinear feedback laws are designed to handle the rotational and its translational motion of the aforementioned space system. Regarding the Sun approach, a solution of the position tracking and its attitude synchronization is presented in rendezvous and docking. The kinematics and dynamics for the relative position and its attitude are modeled considering the dynamic coupling, the parametric uncertainties and the external disturbances. And subsequently the robust adaptive control approach is designed; where the six degrees of freedom closed-loop system is asymptotically stable. It is to note that the core of finding of these benchmarks are considered to be simulated in line with the specifications of the proposed control strategy to be appropriately compared. As a deduction matter, the results indicate that the proposed control strategy is now well behaved in line with the present benchmarks regarding the items (1) and (2), while the Yang benchmark is well behaved regarding the item (3) w. r. t. the aforementioned strategies.

TABLE 4. **THE VERIFICATION OF THE PROPOSED CONTROL STRATEGY IN CONNECTION WITH A NUMBER OF THE BENCHMARKS**

The strategy titles	***Maximum three-axis SRA in steady state (∘)***	***Maximum three-axis SAR in steady state (∘/sec.)***	***Trajectory convergence time (sec.)***
The proposed control strategy	≤ 0.50	≤ 0.05	≤ 30
The Yang benchmark [9]	≤ 0.80	–	≤ 2.50
The Reyhanoglu benchmark [11]	≤ 0.60	–	≤ 15
The Sun benchmark [12]	≤ 0.60	≤ 0.07	≤ 30

4. Conclusion

The research is focused on the improvement of the control strategy to deal with spacecrafts in the procedure of orbital maneuvers. It is realized to be implemented on the aforementioned system under control by organizing the optimization technique to provide the three-axis referenced trajectories. It aims the same spacecrafts to transfer from the initial orbit to its final ones in the process of managing the whole of orbital parameters that should accurately be varied with respect to time. It is to note that the proposed high-performance strategy of the control is designed in the present research to handle the x-axis and the y, z axes in the separated control channels via two sets of actuators. The indication of the novelty is to organize an integration of the three-axis control strategy and the trajectory optimization, as long as the desirable performance of the outcomes are verified through a series of experiments and also a number of benchmarks, notably.

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