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# Geostationary satellites autonomous closed loop station keeping

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ABSTRACT

This paper deals with a novel approach to geostationary satellite station keeping based on the use of a closed loop control law without recourse to previously computed reference trajectories. The closed loop control implementation requires the knowledge of the spacecraft position and velocity with respect to Earth in real time. Numerical results are presented to illustrate this technique. Simulation results showing the feasibility of station keeping on a spacecraft equipped with electric thrusters are also presented.


## 1. Introduction

A satellite in geostationary orbit is subjected to various forces which tend to move it from its assigned orbital position. Station keeping is therefore required and implemented by on-board thrusters. The north/south station keeping (NSSK) maneuvers serve to control the oscillations in latitude due to the luni-solar perturbations on the inclination of the satellite's orbit, the east/west station keeping (EWSK) maneuvers are performed to control the satellite longitude evolution, which is due to the drift caused by the perturbing terms of the Earth potential and to the nonzero eccentricity because of the solar radiation pressure effects.

The classical technique of station keeping uses orbit prediction techniques to determine secular or long term variations in the orbit evolution, as well as linearized equations to compute the correction maneuvers and the time of their implementations. Shrivastava [1] provides a survey of station-keeping methods and the perturbation environment in geosynchronous orbit. Chao and Baker [2] discuss propagation and station keeping of GEO spacecraft. Kamel et al. [3] describe a station-keeping method for a particular thruster configuration in which both xenon plasma and bipropellant thrusters are used and inclination and eccentricity are controlled by a single maneuver. Emma and Pernicka [4] propose a three-phase algorithm that couples longitude control with eccentricity control.

Due to the increasing number of spacecraft within the geostationary orbit, which often involves the collocation of several satellites in the same orbital window [5–7], the control of geostationary satellites requires more precise techniques as well as increased autonomy. This means on one hand that, as the dead-band limits for the station keeping became narrower, seasonal and short terms effects of satellite perturbations should be accounted for and on the other hand more control functions are to be implemented in real time in the spacecraft itself. Following this line of thought, we can find several recent works. Thus, it can be found that Ref. [8] takes into account the actual orbit using a high precision orbit propagator, as well as curve fitting techniques to find out the target orbit in the maneuver planning with the impulse effects computed by linearizing equations. Ref. [9] combines an accurate

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numerical integration to propagate the orbit with an iterative process for the calculation of the maneuvers.

In Ref. [10] an innovative strategy is presented based on formation keeping with a fictitious satellite whose ephemeris is predetermined. The authors indicate that this strategy can significantly improve the long term prediction accuracy without ground support. The strategy utilizes onboard GPS measurements.

The use of Global Positioning System (GPS) receivers to provide orbit determination data for spacecraft has been verified and refined over the last years. As GPS receiver technology has matured, researchers have been attempting to expand the applicability of GPS based navigation to high Earth orbit, and even to the Geostationary (GEO) altitude. The Navigator GPS receiver [11,12] was developed from the ground up to operate in LEO, GEO and beyond. Advanced signal acquisition capabilities provide increased sensitivity and drastically reduced latency for satellite acquisition. When coupled with orbit determination software, this GPS receiver is able to do real-time position estimation to within 10 m RMS and velocity estimation within 0.2 cm/s RMS. This level of precision opens the way for new techniques for geostationary satellites station keeping able to provide a higher level of autonomy.

The autonomy of Earth oriented communications and navigation satellites is mainly motivated by operational and economical reasons. The activities are targeted towards increasing operational availability and safety, while reducing operational cost [13,14].

The purpose of this work is to present a real time closed loop control to be autonomously implemented for geostationary satellites station keeping. This closed loop control implementation, requires the knowledge of the actual spacecraft position and velocity in real time with respect to Earth.

The paper is organized as follows. In Section 2 the closed loop orbit control of a geostationary satellite dynamics is formulated. The performance evaluation of the inclination, eccentricity and semi-major axis control is presented in Section 3, while in Section 4 the modifications for longitude control are analyzed and results of station keeping control are presented. Finally, in Section 5 a summary and conclusions are presented.

## 2. Closed loop orbit control

In present station keeping, the orbit elements are maintained only in an average sense and an orbit propagator is needed to predict where the satellite will be at any future time. The prediction is limited in time.

In contrast, a controlled orbit is one which maintains continuously all of the elements of the orbit and an orbit propagator is not needed to determine future positions. Furthermore, it is the *relative to Earth position* that is controlled and not the orbital parameters. The spacecraft position in Earth centered fixed (ECF) coordinates is in fact the fundamental quantity of interest for geostationary satellite applications.

A closed orbit control system becomes similar to the attitude control system in that commands are autonomously computed and executed on board and, except for monitoring performance and equipment, has little interaction with the ground. In a controlled orbit, there is no need to know the orbit perturbations that the satellite will need to overcome. The normal control system will absorb the small disturbance torques involved.

The equations of motion of the geosynchronous spacecraft as stated in the Earth centered fixed frame are given by,

$$\ddot{\vec{r}} + 2\vec{\Omega} \times \dot{\vec{r}} + \vec{\Omega} \times (\vec{\Omega} \times \vec{r}) = -\frac{\mu \vec{r}}{r^3} + \vec{f} + \vec{a}_c \tag{1}$$

where $\vec{r}$ is the spacecraft position vector, $\vec{\Omega}$ the Earth rotation vector, $\mu$ the Earth's gravitational constant, $\vec{f}$ the vector of perturbation forces acting on the geostationary satellite, due mainly to the Earth nonsphericity, the sun and moon gravitational effects and the solar radiation and $\vec{a}$ is the acceleration control vector to be defined such that the satellite remains within specified limits in latitude and longitude during its active life.

In Ref. [15] was defined a feedback control law in the equatorial plane, function of the spacecraft position and velocity with respect to the central body, for transfer to a *circular orbit*. This planar control law is here presented in Appendix A for completeness.

This control law is now extended for the general three dimensional case with the acceleration control vector now defined as,

$$\vec{a}_c = -c_1 \vec{v} + c_1 \frac{h_d}{r} \vec{1}_T \tag{2}$$

where $\vec{v}$ and $\vec{1}_T$ are, respectively, the spacecraft velocity vector with respect to Earth and the unit tangential vector. The tangential vector $\vec{1}_T$ is normal to the radius vector $\vec{1}_r$ and parallel to the *desired orbit plane* (normal to $\vec{1}_{nd}$) as shown in Fig. 1. This control acceleration reduces to the planar guidance law (see in Appendix A (A3) and (A4)) when both $\vec{v}$ and $\vec{1}_T$ belong to the equatorial plane.

The control law has two components: The first term is proportional with gain $c_1$ to velocity with respect to the central body. The second term is inversely proportional to radius vector amplitude $r$, has gain $c_1 h_d$ and is applied

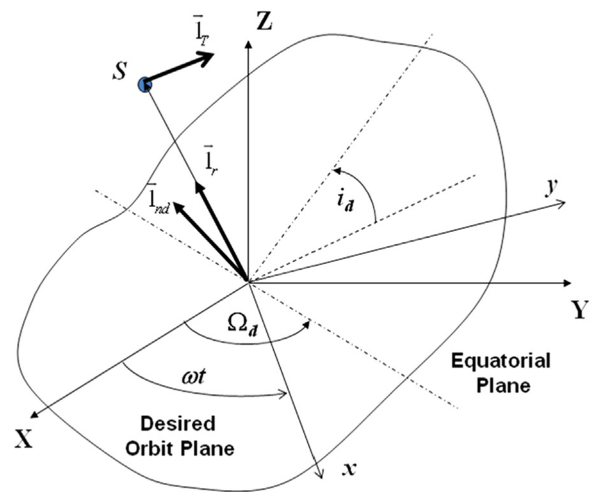


**Fig. 1.** Desired orbit plane and tangential vector.