FUZZY LOGIC CONTROLLER FOR SMALL SATELLITES NAVIGATION

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ABSTRACT: *The development of a navigator for small satellites (SSN) is in progress with a joint effort of Space Engineering and ASI in the frame of a technological development program dedicated to SME[1]. The navigator aims at operating satellites in orbit with a minimum ground support and very good performances, by the adoption of innovative technologies, such as attitude observation by GPS, attitude state estimation by Kalman Filter and fuzzy logic for attitude control.*

The SSN is very attractive in space applications where analytical non-linear models prevent an easy synthesis of classical controllers, and where the volume of parameters affecting the plant behaviour is very high. The navigator was verified through HW-in-the-loop simulations and the following features emerged:

- *three-axes control with control performances compatible with Earth observation missions with optical payloads*
- *autonomous on-board management, and non-nominal pointing capacity without ground planning, permitting to acquire images without scheduling in advance*
- *independence from ground commands in selecting operational modes*
- *autonomous wheels desaturation*
- *autonomous system reconfiguration in response to unexpected events, such as sensors or actuators failures*
- *capability to perform both large re-pointing in low times and accurately maintain attitude*
- *robustness against external and internal disturbances and large variations of platform parameters.*

The SSN development is planned in two one-year phases: the first phase, already completed, was dedicated to the development of the engineering model of the navigator device, of the EGSE, and of the pointing platform. The next phase will be dedicated to the development of an electrical model of the on-board navigator, hosting the state observation and control functions, based upon components available on the market. This activity will include the porting of the application software from the simulator to the specific environment of the spatial processor.

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1 INTRODUCTION

The recent development of soft computing theoretical background and the increasing number of successful applications of fuzzy control to non-linear problems, suggested the idea to apply such techniques to space problems. The fuzzy attitude controller presents very interesting characteristics of autonomy, modularity and robustness. Autonomy benefits from the introduction on-board of relevant decision capability; modularity derives from the possibility to load specific sets of rules for different operational procedures; robustness consists of the capability of fuzzy logic to cope with the uncertainty.

The core of fuzzy controller is an inference engine, which processes the input dynamic state on the basis of logical rules, to produce output control actions. The transformation of input from numerical to linguistic information, and of output from qualitative to numerical commands, is performed by membership functions designed according to the fuzzy theory [1], [2], [3].

Space Engineering acquired a wide Fuzzy Logic expertise by designing the algorithms of a fuzzy processor for small satellites attitude control and by extensive simulation campaigns for several types of missions and platforms [5], [6], [7].

This expertise was the base for the development of a high technological autonomous navigation device, easily to be customized and ready for integration in small satellites.

In addition to the fuzzy controller, the navigator device is based on GPS interferometry for attitude observation, and Extended Kalman Filter for attitude state estimation. GPS interferometry is widely recognized for the precision measurement of the vector distance between pairs of antennas. Earlier investigations on the application of GPS interferometry to the problem of attitude determination were made by [9], [13], and [14]. Short baselines, typically of the order of the meter, are involved.

It is worth to note that the features of the navigator presented in this paper well fit the actual trend of scientific and Remote Sensing missions that is to allow easy, fast and cheap access in orbit to a large community of users, ranging from scientific laboratories to Earth observers community. Further, there is the tendency to distribute the payloads in several satellites flying in close proximity, in a *virtual platform* configuration, or in small satellite constellations. This poses tight requirements for autonomous and coordinated orbit estimation and control that can be only satisfied by innovative attitude and orbit observation and control methods, similar to those presented in this paper.

Finally, the decision of the European Community to develop the GNSS Galileo reinforces the proposed approach of a navigation device based upon the existing GPS, and compatible with future Galileo system.

2 MISSIONS DEFINITION

Two missions, both belonging to the class of small satellites (<150 kg), have been chosen in order to validate the SSN:

- Earth pointing mission for Earth observation or telecommunication.
- Inertial pointing mission for scientific applications (i.e. survey of the sky).

Both missions are selected to operate in a 6:00 A.M. sun-synchronous orbit; this is a quite common low orbit, particularly suitable for small missions for its good geometry and sun illumination. These two missions are representative of a good number of the inertial pointing and nadir pointing missions and therefore represent a valid test scenario for the SSN.

2.1 INERTIAL POINTING MISSION

The reference inertial pointing mission is defined as a scientific mission for the observation of the sky performed with an optical telescope or another instrument for X-ray or Gamma-ray observation. The essential attitude characteristics for this type of mission are:

- to keep fixed the instrument axis for a long period (in the order of hours or days) with a good stability.
- to re-point the satellite towards a different target within the permitted zone of the Sky within a short time.

The following table 1 reports some relevant orbital parameters. The eclipses have a very limited duration and occur only around the winter solstice.

Height	689 km
Eccentricity	0
Inclination	98.1°
Ascending Node Local Time	6:00 A.M.
Orbital Period	98 minutes
Ground Track Repetitivity	5 days
Ground Track Longitude Shift (one orbit)	-24.74°
RAAN rotation (one day)	+0.986°

Table 1 – Inertial pointing mission orbital parameters

In concerning the attitude features, the required pointing control accuracy (2 σ) is assumed of 0.5°, which is quite poor for optical instruments, but not for other scientific instruments. The attitude pointing knowledge is more stringent, is assumed 0.2° (2 σ). The main drivers for the attitude control system design are:

- attitude stability during periods of 300 s, $1*10^{-2}$ °.
- capability to re-point the satellite body of ±45° around the roll axis and ±180° around the pitch axis.
- slew rate: 20° to be performed in 5 minutes (including the tranquillisation period).

2.2 EARTH POINTING MISSION

The Earth Pointing Mission covers any kind of application, both communication and Earth observation or scientific. This mission requires an accurate Earth nadir pointing; moreover an off-nadir pointing mode is required to link the satellite to different ground stations, or to point an on-board instrument to a given site. This off-nadir mode requires an accurate pointing accuracy, to guarantee proper link efficiency with a narrow on-board antenna-beam field of view. Moreover, note that the capability to track the station during the pass is an important input for the reaction wheel sizing.

The orbit selected is a Sun-Synchronous Orbit synchronised to the Earth rotation, allowing revisiting periodically the same location at the same local time. The orbit parameters are reported in table 2. The nominal orbit height (570 km) allows a good compromise between access duration, launch cost and link budget constraints. The value has being optimised considering a long propagation time and using a High Precision Orbit Propagator (HPOP) with no perturbations except for gravity perturbation (30x30 harmonics) with the goal to keep unchanged the ground tracks.

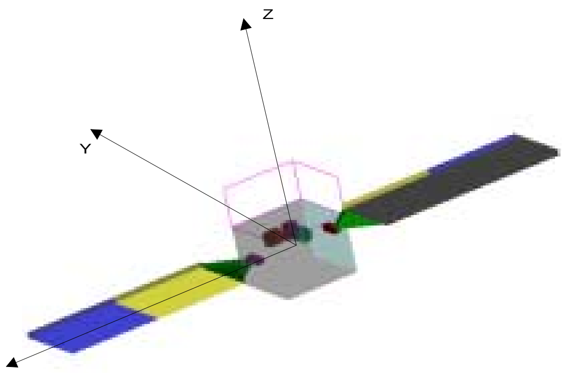
Height (km)	570
Eccentricity	0.001
Inclination	97.676
Ascending Node Local Time	6:00 A.M.
Orbital Period (min)	96
Ground Track Repetetivity	1 day
Ground Track Longitude Shift (one orbit)	24°
RAAN rotation (one day)	0.986°

Table 2 – Inertial pointing mission orbital parameters

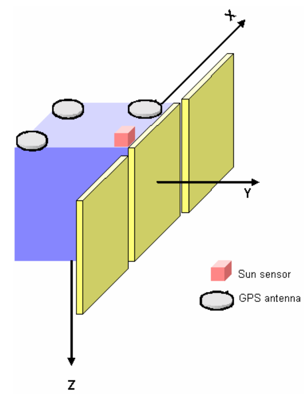
The attitude features derive from the ground-stations visibility constraints. The satellite body is controlled around roll and pitch axes at the same time, to point a given site during a pass. For that, it is assumed a maximum off-nadir rotation angle of ±30° around roll, and of ±60° around pitch. The maximum attitude control error (2 σ) is assumed of 0.5°. In concerning the slew rate, a manoeuvre of 120° is required to last less than 8 minutes (tracking of the ground station)

3 PLATFORM DEFINITION

The satellite structure is simply a parallelepiped and the platform is customised according to the mission type in order to accomplish the different mission objectives (see Figure 1). Satellite mass and volume were chosen without reference to a particular mission, although typical values for space projects were adopted (see Table 3).



"Inertial Pointing"



"Earth Pointing"

Figure 1 - Satellite structure in "Inertial-Pointing" and "Earth-Pointing" missions

Payload weight	40 kg
Platform weight	80 kg
On-board Power	150 W
Solar Panels (two)	2*0.4 m^2
Total satellite mass	120 kg
Satellite Dimensions	0.6*0.6*0.8 m
Propulsion System	N/A
Orbital State	GPS receiver
Attitude Actuators	3 Reaction Wheels, 3 Torque Rods along the inertial principal axes
Attitude Sensors	3 Sun sensors, GPS observer, Magnetometer, Laser Gyros

Table 3 – Satellite Parameters

The "Inertial Pointing" Satellite structure includes two solar panels folded in three sections during the launch configuration. Once deployed, they have the longitudinal axis in the direction of the satellite X-axis. The solar panel can be rotated around their longitudinal axis allowing the maximisation of the solar panels illumination. In concerning the solar panel dimensions, their sizing has been done considering the required on-board power equal to 150 W.
Note that there is a certain degree of redundancy in the attitude sensors configuration, due to the fact that the SSN is designed as a modular device able to receive different kind of attitude measurements. Simulations were performed setting in idle mode different sensors to demonstrate the capability of SSN to work with different sensors and to react at possible failure in presence of a minimal sensor configuration. With regards to the attitude sensor mounting, it is assumed that:

- the platform mounts three Sun Sensors with a field of view of ±60°*60°. The boresight direction of the three sensor is -Y axis, -Z axis, +Z axis.

- The GPS antennas are mounted with the boresight along the satellite +Y direction. The field of view of the antennas is conical with an aperture of 55° (half angle); this limitation is due to the need of reducing the multipath.

The “Earth Pointing” satellite structure differs from the inertial one for the following points:

- The solar panels are fixed. Since the rotation of the platform around roll axis is of maximum 30° for short periods (max. 10 min, corresponding to a pass over a ground station), the solar panels misalignment with respect to the Sun is considered acceptable.
- The satellite is now equipped with a fixed telecommunication antenna having its axis coincident with the satellite +Z axis.

With regards to the attitude sensor mounting, it is assumed that:

- a Sun Sensor (field of view ±60°*60°) is mounted with its axis along the +Y satellite axis
- the GPS antennas are mounted on -Z face

Concerning the GPS antennas, it must be mentioned that, in both missions, the great satellite manoeuvrability could lead in some orbit arcs and in some particular attitude to a poor visibility of the GPS satellites. This condition represents a severe test for the SSN, since it implies the Kalman filter has to estimate, for short period, the attitude without using GPS observer measurements.

The following Table 4 reports the features of the AOCS HW components.

Sensor/actuator	System feature	Value	Unit of Measure
Sun sensor	Random error (STD)	0.1	°
	FOV (half-cone)	60×60	°
Magnetometer	Random error (STD)	20	nT
	Bias	5	nT
Gyro	Scale Factor	0.001	-
	Max measurable rate	0.1	rad/s
	Random error (STD)	2227	$°/\text{hour}^2$
	Bias	1	°/hour
Torque Rod	Max current	0.12	A
Reaction Wheel	Max torque	0.06	Nm
	Max speed	1500	rpm
	Speed @ Max torque	400	rpm
	Inertia	7.11e-3	Kg m^2

Table 4 – AOCS system features

4 S/S SPECIFICATIONS

The navigator is composed by several high-level subsystems, as such: *GPS observer*, *Extended Kalman Filter* for attitude estimation, *Operative Modes Management*, *Fuzzy Controller*, *Data Handling*, which processes the commands and telemetry data to interface the GPS receivers and EGSE. This has the capability to handle real-time and time-tagged commands. The above subsystems interface several auxiliary modules needed for simulation purposes that model the satellite dynamics, sensors, actuators, external environment and EGSE.

The functional model of the overall simulator is shown in the following Figure 2.

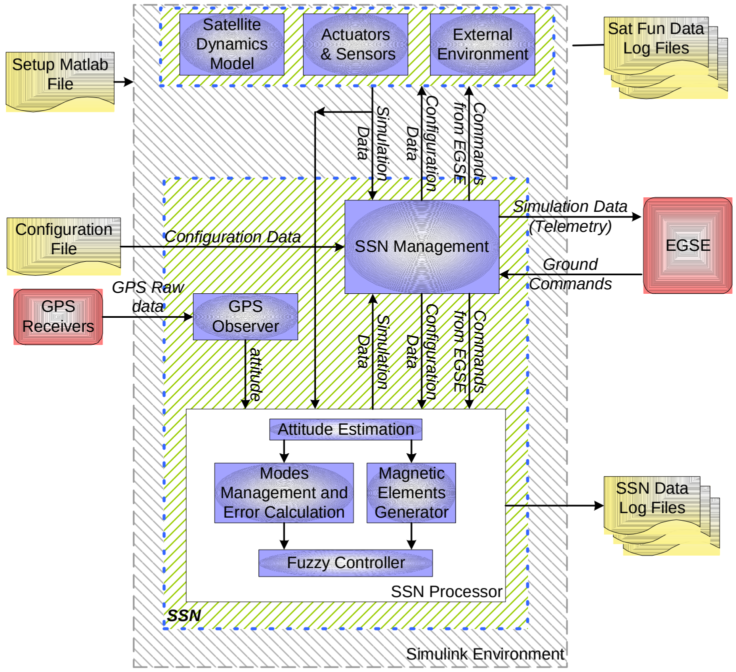


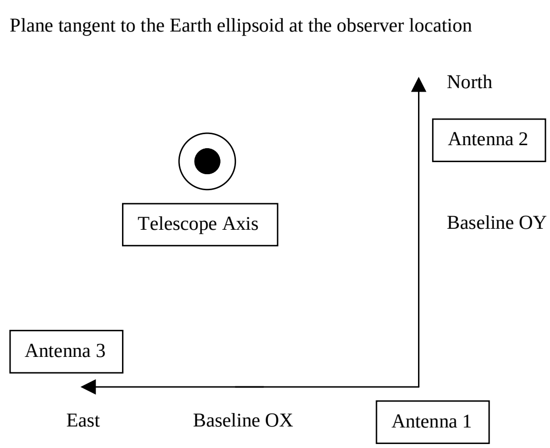
Figure 2 - Simulator functional model

4.1 GPS OBSERVER

GPS observer provides the attitude observation through the ambiguity resolution of an 'interferogram'; the interferogram is built using three antennas phase measurements. The algorithm yields an estimate of the body-fixed angles, of the two independent baselines and of the angle in-between. The estimate is made epoch-wise, i.e. regardless the value the parameters had at previous epochs. Assuming a short baseline of 0.3 meters, the rms. (root mean square) repeatability at 1 Hz is 0.3° for the horizontal angle and, almost double for the vertical angle.

The GPS observer was realised by using three standard commercial, single frequency GPS receivers providing both L1 code and carrier-phase measurements (NovAtel GPS-3151R) are connected to the three GPS antennas. The GPS antennas form two baselines that, in the “home” position of the telescope, are oriented in the topocentric reference system (see Figure 3).

GPS antennas mounted on the telescope (front view)



Baselines orientation

Figure 3 - GPS Observer

4.2 EXTENDED KALMAN FILTER

Extended Kalman Filter (EKF) provides the attitude state estimation from data fusion of GPS sensor and other attitude sensors (magnetometer, sun sensors, gyros). The filter can operate also in presence of lack of observations in case of eclipse, GPS outage and sensors failure. A model of attitude dynamic is included in the state prediction to improve the estimation accuracy.

Moreover, the EKF is supported by a coarse-attitude estimator realised with the q-method useful to recover from eventual filter divergence.

4.3 OPERATIVE MODES MANAGEMENT

It controls autonomously the transition between the operational modes (acquisition, normal, safe, wheels desaturation), based on the attitude and orbital states and on the system configuration and health. This module was realized using StateFlow, a Simulink toolbox which is a graphical design and development tool for simulating complex reactive systems based on finite state machine theory.

4.4 FUZZY CONTROLLER

The controller has been realized by means of a Multi Input Multi Output (MIMO) Mamdani Fuzzy Controller with a knowledge base composed by 53 logic rules. A total of 41 Membership Functions (MF) have been defined to cover the entire universe of discourse.

The fuzzy controller aims at controlling the satellite attitude and providing for the satellite three-axis stabilization. The three axes attitude control is performed by three single axis fuzzy controllers adopting the same algorithm but differing for using three different sets of MF. The fuzzy controller is able to act in every operative mode by means of different logic rules weights and particularly it provides for the satellite attitude control in Acquisition, Normal and Safe operative mode.

The satellite control is obtained autonomously by the fuzzy controller generating commands to the actuators (reaction wheels and torque rods). The main features of such control are:

- simplicity in spacecraft control systems design and development
  - reduced tuning time
  - easy introduction of different control operations: attitude control, trajectory control, reconfiguration
  - safety enhancement by autonomous reconfiguration
  - easy and quick development of fuzzy models
- increased robustness for automatic control reconfiguration
- reduction in development and production cost for flight control systems
- autonomous on-board control features: reduction on ground operations costs

5 SIMULATIONS

5.1 ENVIRONMENT

The navigator performances were verified through HW-in-the-loop simulations. The simulation environment (see Figure 4) is based on Simulink (Mathworks) and LabView (National Instrument), running in two different PC in a client-server configuration, hosting the on-board navigator SW and auxiliary functions needed for testing. A mechanical CAD (Dads/Plant) models the satellite kinematics and dynamics (main body, antenna, solar panels, wheels). This environment interfaces the HW that includes the GPS receivers and antennas, an optical 8” telescope used as steering platform and mechanical test equipment. As already mentioned, the HW equipment are all standard commercial items, and the SW runs within Windows environment.

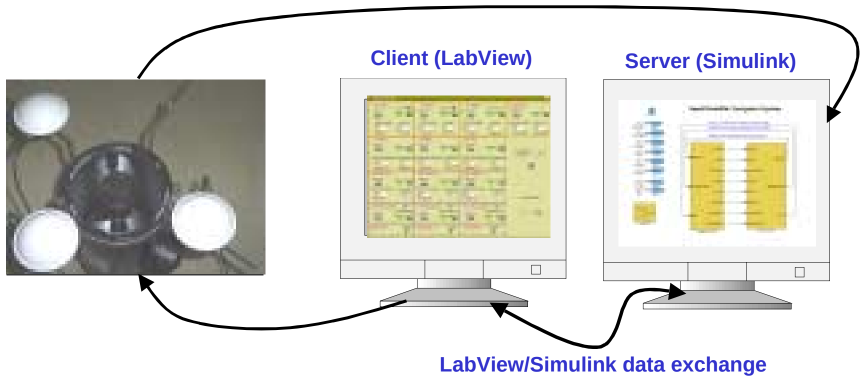


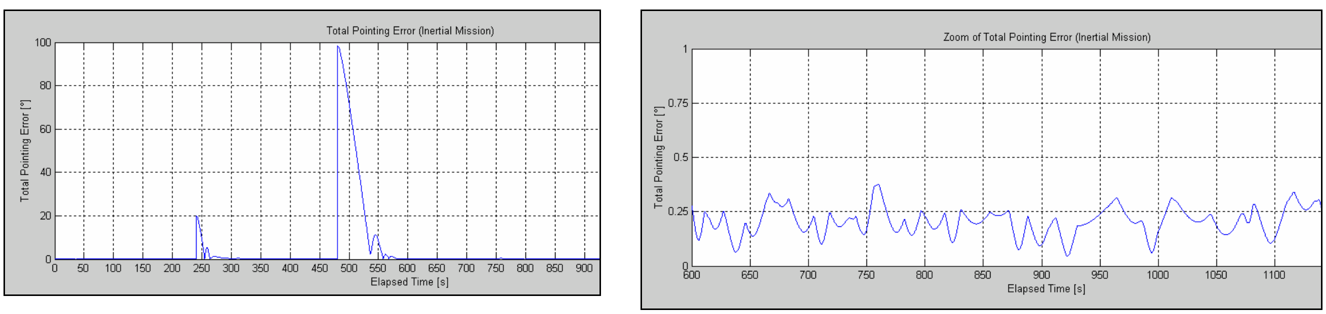
Figure 4 - SSN Physical Lay-out

5.2 RESULTS

The results hereafter presented show the behaviour of the navigator in case of inertial and earth pointing attitude maintenance, attitude acquisition, wheels desaturation and single solar panel deployment.

5.2.1 Inertial Pointing Mission

This simulation shows the SSN performances when the required reference attitude is inertial and a re-pointing maneuver is performed. The plot of total pointing error is reported in Figure 5. It shows two peaks corresponding to the requested maneuvers, the first one of 20° at 240 s (due to a roll rotation) and the second of about 100° at 480 s (due to roll and yaw rotations). Moreover, it shows that the time necessary for the execution of the first maneuver is less than one minute. The zoom of total pointing error after the second maneuver (right plot in Figure 5), reveals an attitude error of about 0.2°-0.25°.



Total Pointing Error — Total Pointing Error Zoom

Figure 5 - Total Pointing Error (Inertial Mission)

5.2.2 Earth Pointing Mission

In case of Earth-pointing mission, with off-nadir pointing, the navigator autonomously performs the satellite control propagating on-board the orbital elements and computing the pointing angles. The SSN minimizes the total pointing error and the satellite angular rate using only the reaction wheels, autonomously managing the transition between the two reference attitudes (Nominal Earth-pointing and Off-Nadir). Moreover it estimates and controls the attitude also during eclipse periods.

The following Figure 6 shows the off-nadir manoeuvre angles (upper plot), the total pointing error (intermediate plot) and its zoom (lower plot) obtained during this simulation. The manoeuvre autonomously starts about 3 minutes before the beginning of target visibility and it takes about 120 s (intermediate plot) to lower the error from 70° up to a value lower than 0.5°.

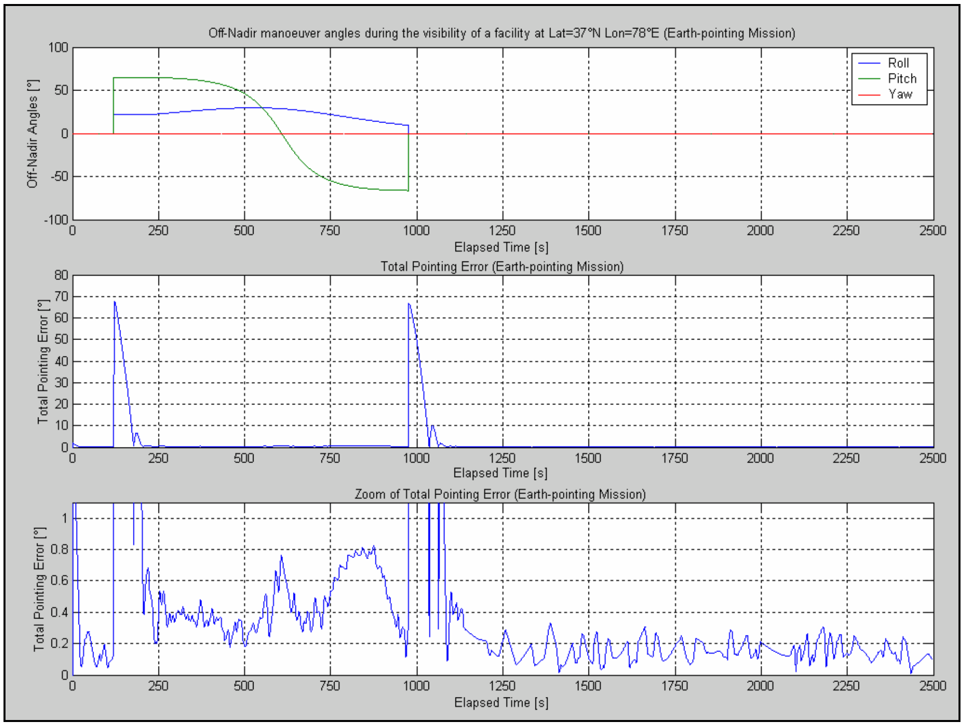


Figure 6 - Off-Nadir manoeuvre angles and Total pointing error (Earth-pointing Mission)

During the off-nadir manoeuvre the total pointing error value is about 0.4° with the exception of two little peaks (≅0.8°). This performance can be considered very satisfactory also considering the following remarks:

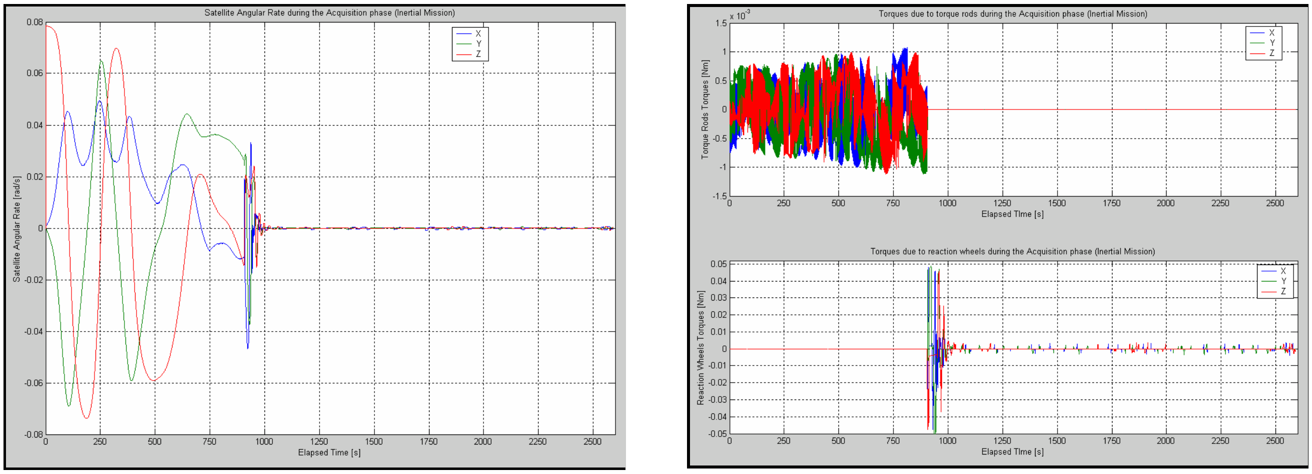
- The requested attitude manoeuvre is very demanding not only for range maximum variation but especially for attitude slew rate. In general targeting a site more closed to the ground track requires a smaller manoeuvre. In fact, in nominal Earth pointing the total pointing error is below 0.2° (see lower plot from 1200 s to the simulation end).
- This two-axis manoeuvre is required only for Earth-pointing communication missions with very small antenna beamwidth. Note that for SAR Earth-pointing mission the re-pointing angle around pitch, the more demanding manoeuvre, is not required.
- In absence of attitude manoeuvre, the attitude error STD is about 0.1° and attitude error drift is absent.
- In this simulation the GPS observer was set off. The addition of this sensor enhances the performance.

5.2.3 Acquisition Phase

This is a critical phase in a spacecraft mission following the satellite separation from the launcher, characterized by high satellite angular momentum and high angular rates. The SSN task is the reduction of the satellite angular rate components until to reach pre-defined values (despin). In the nominal sequence of events the SSN starts in Idle mode and then it goes in Acquisition mode. The exit from Acquisition mode to Normal mode occurs when the satellite angular rates are below a given threshold. In Acquisition mode, the navigator reduces the satellite angular rates using only the magneto-torques.

Acquisition mode is also autonomously activated when an increase of satellite angular rate (tumbling) caused by a momentary loss of the satellite attitude is detected.

Figure 1 shows, on the left, the plot of satellite angular rate components. It is possible to see that the initial angular rate component around Z_{body} of 0.0785 rad/s is lowered below a given threshold in less than 900 s by using only the torque rods (see right picture). At this moment, the operative mode changes in Normal mode and the fuzzy controller starts to use the reaction wheels (at about 800 s of elapsed time) to acquire the reference attitude.



Satellite Angular Rates | Torque rods (top) and reaction wheels (bottom) torques

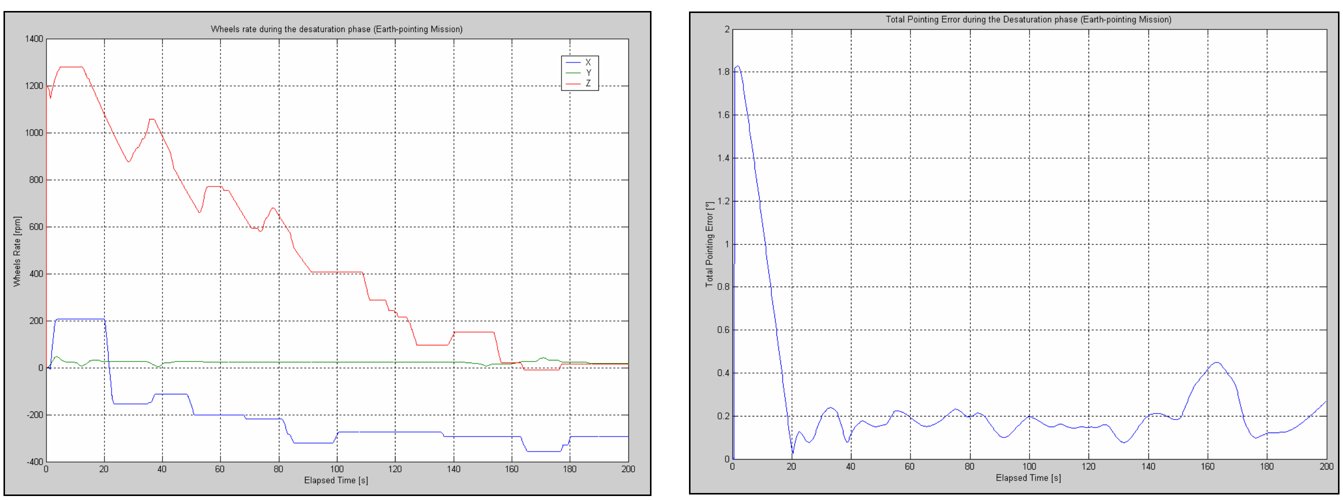
Figure 7 - Attitude acquisition after separation

After about 17 minutes the satellite acquisition and stabilization is completed with a mean pointing error of about 0.25°.

5.2.4 Wheels Desaturation

The SSN performs wheels desaturation by means of torque rods. Note that the SSN desaturation logic differs from that of the classical attitude controller. Indeed, the fuzzy controller tends to maintain autonomously the wheels angular rate low as much as possible without waiting that they exceed a predefined threshold and exploiting every favorable conditions of the Earth Magnetic Field $\underline{B}$. Anyway, since the desaturation implies a disturbance on the satellite attitude caused by the activation of the torque rods, this operation can be inhibited by ground command (desaturation flag).

Figure 8 shows the reduction of the angular rate of the wheel along Z-axis from the initial value of 1200 rpm to 0 rpm in about 200 seconds. This reduction has been obtained by using the torque rod along the Y-axis. The total pointing error (on the right of Figure 8), after the initial transient, varies around 0.2°.



Wheels rate | Total pointing error

Figure 8 - SSN performances during the desaturation phase (Earth-pointing Mission)

5.2.5 Robustness Test: single panel deployment

The purpose of this simulation is to verify the SSN performances when, in the deployment phase, only one solar panel successfully reply to the deployment command (see Figure 9). The implications are:

- high asymmetry in disturbance torque generation transmitted from the panel to the satellite main body
- variation of the inertia tensor

The opening transient was done very short (5 s) to have a further critical condition for the fuzzy controller. The springer actuator and the dumper have the following stiffness: K_{damper}=0.05, $K_{springer}$=0.1.

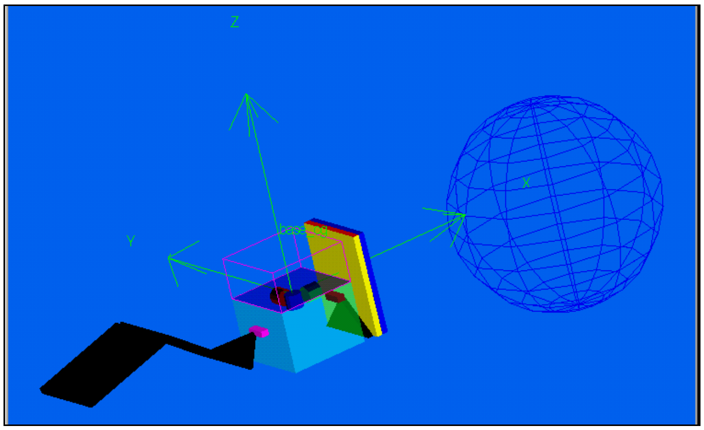
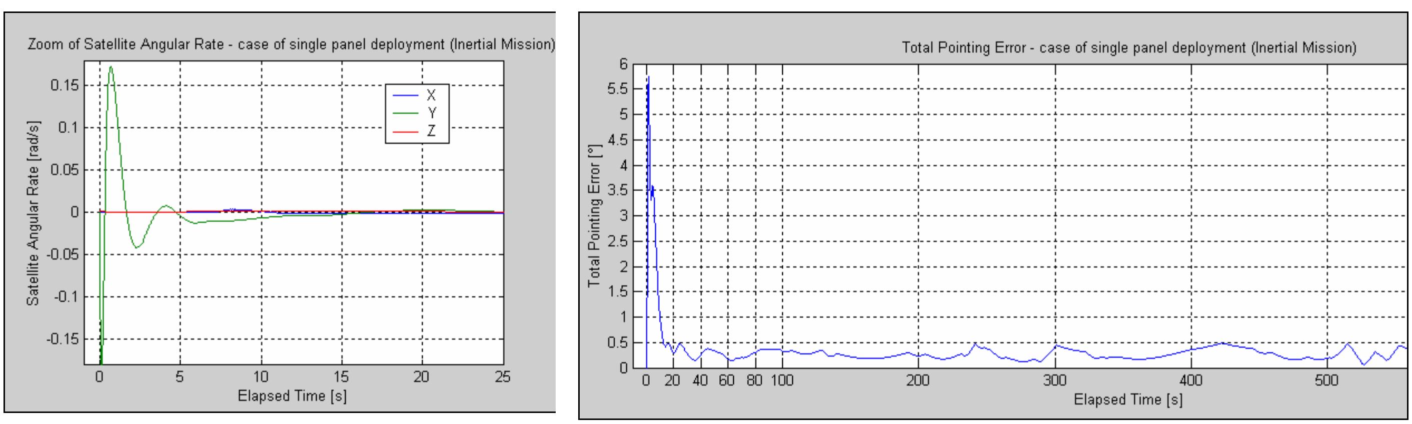


Figure 9 - DADS\Plant Satellite Model in case of single panel deployment

Figure 10 reports the plots of the components of satellite angular rate and the total pointing error. The panel deployment occurs along X-axis resulting in an increased angular rate towards pitch (satellite Y-axis). This initial high angular rate is rapidly lowered to 0 rad/s (reference value since this test was made simulating an Inertial pointing mission) and the satellite stabilization is reached and maintained until the simulation end.

In concerning the total pointing error, after the initial transient (the first 20 seconds of the simulation, where the total pointing error reach the value of 6°) the pointing error is maintained lower than 0.5°, with a knowledge error lower than 0.25°, confirming the capability of the fuzzy controller to be able to control the satellite even in very critical conditions.



Wheels rate | Total pointing error

Figure 10 - Satellite angular rate error and total pointing error in case of single panel deployment

6 CONCLUSIONS

The usage of a high technological navigator device based on fuzzy technique, GPS measurements and Kalman filter has been demonstrated feasible and convenient for operating satellites in orbit with a minimum ground support and very good performances. The availability in a near future of space missions willing to use this device will open the way to the validation by a real flight opportunity.

In order to be ready for such opportunities a second phase dedicated to the development of the SSN unit device has been planned. This processor will host the navigator state observation and control functions, using components having a space qualified version available on the market.

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SOMMAIRE/*SUMMARY*   Data Processing