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Low Cost Earth Attitude Sensor

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ABSTRACT

A patent-pending, low-cost, moderate performance, Earth Attitude Sensor for LEO satellites is described in this paper. The paper deals with the system concepts, the technology adopted and the simulation results.The sensor comprises three or four narrow field of view mini telescopes pointed towards the Earth edge to detect and measure the variation of the off-nadir angle of the Earth-to-black sky transition using thermopile detectors suitably placed in the foci of the optical min telescopes. The system's innovation consists in the opto-mechanical configuration adopted that is sturdy and has no moving parts being , thus, inherently reliable. In addition, with a view to reducing production costs, the sensor does without hi-rel and is instead mainly based on COTS parts suitably chosen. Besides it is flexible and can be adapted to perform attitude measurement onboard spacecraft flying in orbits other than LEO with a minimum of modifications to the basic design. At present the sensor is under development by IMT and OptoService.

1. INTRODUCTION

The emerging small satellite market stimulates the development of equipments and devices tailored to the missions which are feasible and cost-effective to be implemented with these craft. Their attractiveness is focused about the creation of space distributed systems (swarms, constellations and formations) and the 'faster, cheaper, better' paradigm which, notwithstanding some mishaps, still dominates the minisatellite scenario.
The small satellite market is presently characterized by the trend of offering high performance platforms at relatively low costs: full three-axis stabilization and attitude control is commonplace among the nano and microsats above 30 kg used for scientific, remote sensing and even communication applications. Due to the small mass and tight cost restraints, the sensors and actuators offered by the commercial space market for larger spacecraft (Table 1) cannot be used and new solutions are being developed to meet the challenge.
Missions using Earth-pointed small satellites would obviously benefit from the availability of medium-high performance, low-cost (order of few ten K$) IR Earth sensors. Such devices, today, do not simply exist. Accordingly, our group decided to initiate the development of such device and is facing the challenge of meeting a fair compromise between performance and cost.

Table 1 Sample Earth sensors for Space

item	Sens. #1	Sens. #2	Sens. #3	Sens.#4
Det.type	?	pyroelectric	thermopile	Thermopile
type	Conic. scan	Mech. Scan	static	static
Coll..optics	refractive	Refr.active	refractive	refractive
material	?	?	Germ.lens	Germ. Lens
orbit	LEO	LEO	LEO/MEO	LEO
Det. type	?	Dual array	Array	Dual detect.
f.o.v. (+-)	17° x 33°	5.5°	16° x 10°	20° x 15°
Meas.error	+-0.06°	+-0.06°	< 0.03°	0.2°
N° telesc..	1	2	3	3
Mass (kg)	3.5	1.4	4.5	1.2
DC power	7.5 W	0.8 W (?)	9 W	< 5 W

The intended low cost , medium performance, IR sensor would support the Attitude Control System of nano, micro and minisatellites in Earth-pointed missions, be they for remote sensing, science or communication use.

2. SENSOR CONFIGURATION CONCEPT

A brief description of the sensor principles is provided in the following to illustrate the main optical, mechanical and electrical features.
The prospective IR sensor application leads to particular choice of components and configuration: the compactness and stiffness of the electro-optical design and the thermo-mechanical environment drive the choice of the opto-mechanical configuration. As a result the design is based on a reflective approach for the telescopes – as opposed to the refractive approach adopted in many Earth sensors – while thermopiles were chosen as detecting elements. The telescope is made by a numerically milled and polished aluminium alloy block, to assure mechanical integrity, accuracy, optical grade quality of the reflecting surfaces, stability and ease of mutual alignment with the detector set in the focal point.
The Earth-emitted infrared radiation is collected by the telescope and sent to the detector assembly. In one version of the design the assembly consists of two detectors whose beam boresights look towards different directions (Fig. 1): one points at the Earth body and constitutes the reference signal, while the second one points towards the Earth disk edge.

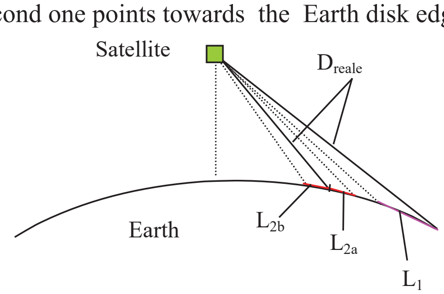

Fig. 1 Detectors 'beamlets' boresights

Accordingly, the received thermal energy changes with the variation of the off-nadir angle due to spacecraft mispointing. The distance between the two beamlets boresights is defined by the optical design and the mechanical characteristic of the receiver.
In a preferred implementation the Earth Attitude Sensor has four such 'channels' at 90° to each other, though a configuration with three 'channels' at 120° is also possible and in fact is adopted by many vendors. However the configuration with four channels has superior reliability and error-reduction performance than a three-channel version and may be thus preferable. Besides, the four channel configuration would have superior initial Earth acquisition capabilities. Fig. 2 shows a conceptual opto-mechanical embodiment of the Sensor system.

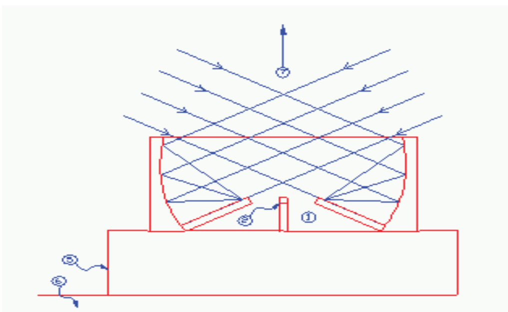

Fig. 2 Preliminary System Mechanical Arrangement

The electronic signal processing circuitry is housed in small PCBs contained in a small aluminium box below the optical heads .
The thermal design of the system is tailored to the mission requirements and depends heavily from the placement of the sensor on the spacecraft body.

3. DESIGN PHILOSOPHY

In order to correctly dimensioning the system, it could be separated in two connected parts: the first concerns the optical section regarding the configuration of the telescope (aperture, performance, Equivalent Focal Length, etc.) and a second part regarding the electrical processing of the data received.
A separate, but not least important, treatment is given to the assessment of system errors due to the operating scenario variables :

- Earth temperature
- Earth oblateness;
- Orbit parameters;
- Physical characteristics of the Earth' area 'seen' by the sensor' beamlets';
- Satellite ambient temperature
- Mechanical assembly error
- Components sensibility limits

The work on the system errors is ongoing and will be the subject of a future paper. Instead, a brief description of the simulation approach and of the achievable performance is given in the following.

The first step is to configure the optical section: two sensors of receiving square sensitive area of L side are positioned in the sensor' focal plane.
The total FOV of the receiver is ±1°, while the aperture of the optical telescope could be set around ϕ obtaining an F# of 0.5.
The power impinging on the sensors is connected to the portion of the illuminated area: on the focal plane there is an 'image' of a portion of the Earth disk with a fixed magnification factor. The two sensors are positioned in different positions: while the first one is illuminated by approximately one half of its receiving area, the second one is completely illuminated by the Earth emission. The first detector measures the angle with which the Earth 's edge is seen by the spacecraft through a continuous comparison with the received signal provided by the second detector. The expression of the energy density emitted by Earth is:

$$E(\lambda) := \frac{2 \cdot \pi \cdot h_P \cdot c^2}{\lambda^5 \cdot \left(e^{\frac{h_P \cdot c}{\lambda \cdot k_B \cdot T}} - 1 \right)} \qquad (1)$$

where:
h_P is the Planck constant; k_B is the Boltzman constant; λ is the wavelength; T is the temperature of the Earth; c is the speed of light.
Considering the upper and the lower limits of the temperature variation, T_{terra} was computed to span between 288K and 218K related to the temperature variations due to different weather conditions.
In Fig. 3 the curves describing the energy densities emitted by the Earth in these conditions, are depicted.

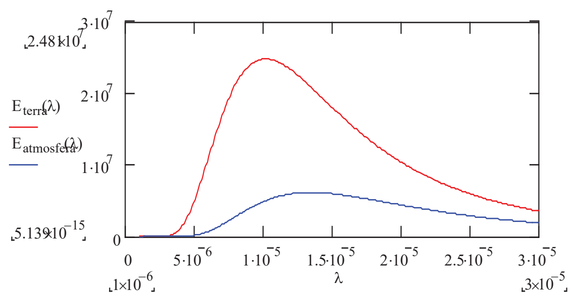

Fig. 3. Black Body Earth Emission

The geometrical variations (Earth radii, FOV angles, Earth-Satellite distance, temperature-dependent sensor receiver dimensions,..) impact on the received signal: they are not considered in this first analysis but are taken into account in the evaluation of the system errors. For the purpose of the *simulation*, the Earth's emission could be considered lambertian with an emission coefficient of ε =0.8. The transmission of the optics is connected both to the optical design of the telescope and, partly, to the mirror' surface coating and finish. For thermal and optical reasons the best performance, within the spectral range considered, is achieved with gold plating which has an infrared emissivity of 0.03. For safety margin, a very

conservative τ=0.8 for the optical system's transparency was considered.
The dimensioning of the single receiver takes into account simultaneously the variation of illumination due to different Earth weather condition and the minimum received signal detectable that could put a strong limit to the performance.
The chosen operative wavelength band, 8÷14 μm, conforms to the most widely utilised IR detectors. Within the range of temperature, the Earth emission could be approximated by a perfect black body at different temperatures.
The energy captured by the optics must be related to the Earth emitted area, the solid angle of the collecting optics, the receiver area, the geometrical configuration of the satellite with respect to the Earth.
To find the power impinging on the detector over the spectral range considered, the following expression should be integrated.

$$P_{receiver}(\lambda) := \frac{E(\lambda)\cdot A_{receiver}}{\pi}\cdot\frac{A_{optics}}{d_{real}^{2}}\cdot\varepsilon\cdot\tau \qquad (2)$$

In Eq. 2 the value of the projected emitting area is considered into he expression of $A_{receiver}$.
The resulting values are related to opposite conditions of the Earth's weather, within the chosen bandwidth, to the Earth-to-Satellite geometry, and considering the particular choice of receiver.
A system simulation is in progress with the evaluation of errors due to external factors (e.g. operational orbital and thermal parameters) and uncontrolled variations of the measurement conditions (Earth weather/radiance, Earth seasons etc.). Updated results will be presented in a future Workshop.

4. DESIGN PRINCIPLES

In what follow a brief description of the main features of the sensor' design (patent pending) is given.

4.1 Opto-Mechanical Design

The optical configuration of the single channel is constituted by a reflective telescope obtained by precision machining an aluminium block according to a defined profile, and then treated (gold plated) to reflect the Earth's radiance within the 8÷14 μm band.
The optical characteristics and performances of the telescope are connected to the sensibility of the receivers and to the Field Of View (FOV) that is chosen to enhance the resolution power of the system.
In a preferred version of the telescope the reflecting surface is a section of an offset fed paraboloid, though for more advanced versions using linear arrays in the focal plane, double curvature paraboloid segments could be used as well.
When used in LEO, the reflective telescope boresight points at a quite high angle with respect to the spacecraft nadir, therefore care must be taken in positioning the detector(s) in the focal plane to avoid optical blocking effects, multiple reflections and construction misalignments. Besides, the detector must be oriented in such a way that it sees the low emissivity paraboloidal reflector body and not the high temperature, high emittance, nearby objects that are part of the spacecraft body. This points to choose optics with rather low F/D ratios. The detector, which may have a significant size with respect to the reflector diameter, would have to be positioned to ease the handling and the interfacing with the electronic circuitry. The geometry sketched in Fig. 2 is a fair compromise between contrasting requirements.
The mechanical design of the whole sensor takes into account a typical environment for micro and minisatellites launched with modern rockets as secondary payloads. The use of aeronautical-grade aluminium alloy was determined to be sufficient. The thermal design should consider the worst operational environment, represented by an Earth-looking spacecraft in a down-dusk heliosynchronous orbit (spacecraft typically quite hot) and flying at rather low altitudes (less than 450 km) where the disturbing effects of the external torques put more demanding performance from the attitude control system and, therefore, on the attitude measurement sensors. As a baseline approach, while the inner reflecting surfaces of the telescopes are gold-plated, the external surfaces can be either polished and left uncoated or else milled and then coated with OSR. The box containing the PCBs for signal processing can be nearly thermally isolated from the spacecraft if needed, to reduce the spacecraft-to-sensor heat flow.

4.2 Electrical Design

The electrical design of the receiver is an essential part of the sensor and defines the feasibility and quality of the measurement. The sensitivity of the method adopted for the handling of the signals must be very high and this stresses the importance of the precision and stability of the measurement.
The two detectors in the focal plane of each telescope serve different functions as outlined in Section 3. The first detectors looks to an area completely contained into the Earth disk and is used as a reference, while the second detector points at the borderline of the Earth and is responsible for the detection of the variation of the Earth's edge angle w.r.t. the spacecraft theoretical Nadir. The measurement is based on the comparison between the signals from the two detectors, thus the signal change induced by temperature variations of the viewed area on the Earth can be compensated. The instrument resolution is determined considering the signal variation induced by the minimum angle displacement to be measured in case of minimum power radiated by the Earth (e.g. a cold viewed area) with respect to the maximum signal to be acquired (e.g. an hot viewed area).
From the preliminary evaluation of performance, it turns out that the dynamic range of the received signal is quite high with respect to the minimum detectable

signal of the detectors commonly found on the professional market.

This narrows considerably the detectors' choice, that were chosen to be thermopiles for bandwidth and cost reasons.

The precision of the measurement is degraded by two main sources of error: temperature variations of the mirror and of the detectors, and electrical noise and offset of the acquisition electronics. The error induced by the mirror temperature variations is minimized using high reflectivity surfaces. The error caused by detectors' temperature variations is minimized using a thermocouple coupled to the detector junction that allows to determine precisely the actual detector temperature and, accordingly, the resulting compensation factor.

The error induced by the AC noise relevant to the electronic detection chain can be demonstrated to be negligible w.r.t. the minimum signal to be acquired, due to the fact that the system bandwidth is relatively small (few tens of hertz). The predominant error is related to the DC offset. In particular in the preamplifier stage, which contribution to the system performance is critical, any ultra-low offset amplifier' error is significant and cannot be neglected. This imposes to use an 'offset cancellation loop' in order to null the offset contribution before each measurement.

A preliminary block diagram of the sensor' receiver chain is shown in Fig. 4

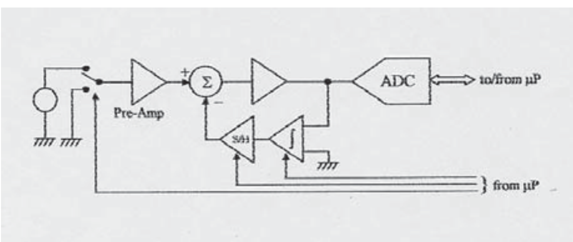

Fig. 4 Simplified block diagram of the sensor channel processing electronics

The electrical circuit' implementation makes large use of industrial-grade, extended temperature range, electrical parts. Since the complete Earth sensor would include three or, better, four such circuits, a mild level of circuital integration is foreseen. The further processing of the three, or four, digital outputs, the timing, calibration and gain setting functions will be performed by a microprocessor supervised by the spacecraft On Board Computer.

4.3 Reliability considerations

For an Earth-pointing mission the availability of the IR Sensor is instrumental to meet the pointing performance objectives: the attitude sensor reliability is as important as the technical performance. The proposed design, being static and thus without moving parts, and based on reflective optics (absence of high costs germanium lenses) is inherently reliable. The thermopile detectors are also reliable and uncritical.

The processing electronics is largely decriticized through the adoption of an offset cancellation loops, enabling using industrial extended temperature range parts. The sensor sensitivity to the temperature range is reduced and means are available to further thermally decouple the sensor from the spacecraft body if required.

A four arms sensor version offers a better redundancy and can offer, even with two failures, a form of graceful degradation which can be further exploited at software level.

The use of industrial grade parts is generally compatible, according to the IMT's experience achieved through several experimental tests, with a LEO space environment with lifetime durations of 2 to 3 years: performance which is more than acceptable for nano and microsatellites implementing many scientific and application missions characterized by modest budgets. For mini-satellites, aiming at longer lifetimes in LEO or even in slightly elliptical orbits, an upgrade of the electrical parts quality may be required, without, however, significantly changing the basic sensor' design.

5. PLANS FOR IMPROVEMENTS

At present the design team led by IMT is considering alternate solutions for the detector assembly, to achieve a greater flexibility for satellite missions characterized by greater operational orbital parameters changes, and even for satellites injected in slightly elliptical orbits. The solution is untrivial due to geometrical constraints, detectors' sizes, considerations on aberrations and sensitivity, and the scarcity of devices having the wanted characteristics.

The target performance figures of this IR attitude sensor are summarized in Table 2 for both the configuration under development (dual detector) and for a planned near-term improved version (linear array).

Table 2 Target sensor performance figures

Item	*Target value*
Intended orbits	LEO /MEO
Detector technology	Thermopile
Sensor type	Static
Detector type	Dual detector / linear array
Collecting optics	reflective
Fov	2° x 2° (dual detector) or 2° x 16° (linear array)
Measurement error	< 0.2°
N° of telescopes	4 as a baseline; 3 is a possible alternative
Mass	< 0.8 kg
DC power	< 4 W

6. CONCLUSIONS

The Earth Attitude sensor being developed by IMT and OptoService is a promising complement to existing low

cost attitude measurement sensors intended for use on board nano, micro, mini and – in general - small satellites flown in circular or even mild ellipticity LEOs.
The key features of this IR Earth sensor, all contributing to lower the expected production cost of the device, are: a reflective approach; the use of industrial-grade thermopile detectors; absence of mechanical parts in motion; use of COTS parts for the microprocessor-controlled processing electronics.
It is hoped to be able to report the progress of this Project in one of the coming Conferences dealing with small satellite missions and relevant technology developments.

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