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Moon/Sun interference analysis, identification and suppression on Dual Cone Earth Sensor

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ABSTRACT

Focused on the problem of the Sun and Moon interference to Dual Cone scanning infrared Earth Sensor (DCES) which is installed on the near-Earth satellite, this paper puts forward the corresponding solutions. First of all, according to the working principle of the sensor, the paper analyzed the interference mechanism of the Sun and Moon in detail and carried out a simulation. By detecting and discriminating the pulse produced by the Sun, the Moon and the Earth, a novel method was presented and new software was developed to suppress the interference. The method was realized without additional sensors or other devices, and without remote control on the ground, resulted in low-cost and high autonomy of the satellite. Furthermore, the method was verified through an electrical signal testing as well as an optical signal source testing. Results showed that the Moon and Sun interference was identified and suppressed without corrupting the horizon crossing.

1. Introduction

Infrared Earth sensor (IRES) is used as the main attitude sensor for the near-Earth satellite [1], which determines the satellite attitude deviation by the vector offset of the datum shaft relative to the satellite-Earth. The output signal of the sensor is applied by the Attitude-Orbit Control System (AOCS) to make the satellite antenna, meteorological payloads, scientific loads or other remote measuring loads toward the Earth, or guide a direction for the satellite in space.

The operating mode of IRES can be divided into single cone scanning, swinging scanning, spin scanning, etc. DCES is designed on the basis of the flight proved single cone infrared Earth sensor (CES). A CES has an optical head which, when spun by a motor, sweeps the sensor's field of view (FOV) in a cone shaped pattern. By improving the optical head, the scan path of each FOV can describe different mission-specific cones enabling one DCES to scan the Earth twice, in different locations, with each turn of the motor. Thus, a DCES provides the same information as two concentric single CES at the lower cost, weight and power.

The basic principle of CES and DCES is that, when the FOV crosses the Earth horizon, a pyroelectric detector senses the change in target infrared radiation and produces an electric signal. Three horizon crossing signals and a fixed reference signal produced at each turn provides attitude information of a satellite. For a near-Earth satellite, the infrared radiation intensity of the Sun and Moon is approximately equal to the Earth. When the Sun or Moon is in the scan path and near the Earth disk, an interference pulse is produced besides the normal Earth wave, which causes calculated error of attitude [2,3]. During the on-orbit flight of Haiyang-1 (HY-1), there were three times of all-attitudes capturing caused by the disturbance of the Sun [4].

Therefore, it is of great significance to analyze the mechanism of interference of IRES by the Sun and the Moon. Based on the mechanism, a method to inhibit the interference should be proposed to improve the reliability of IRES. At present, there are four exiting method to suppress the Sun/Moon interference [2,4]: (1) Add the additional Sun probe, which the FOV is concentric with IRES, but bigger. After detecting the Sun, the output of IRES is prohibited to suppress the interference. (2) Design a special optical head to make it difficult for interference light to enter. (3) By means of software prediction. According to the satellite orbit parameters, the ephemeris of the Sun/Moon, the moment of the Sun/Moon interfering IRES can be predicted before the interference. The corresponding interfered output of IRES can be prohibited and be restarted after the expected interference. (4) Send telemetry command through the ground satellite control center (GSCC). The output of IRES can be prohibited by GSCC before the interference and be restarted after the interference. It is obvious that (1) and (2) are by the way of the hardware, (3) and (4) by way of the software,

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but (1) increases the additional costs, (2) increases the difficulty of design, and (3) results in the reduction of attitude measuring precision of the sensor, (4) makes the satellite lacks of autonomy. Therefore, on the basis of these methods, it is very important to put forward a method of low cost, high reliability and full autonomy which can suppress the interference.

2. Attitude determination method of DCES

Before the analysis of the mechanism of interference of the Sun/Moon, the working principle of the DCES must be discussed. Based on the previous introduction of DCES and single CES, the DCES can be considered as two concentric single CES. Thus the working principle of single CES was discussed as follows.

There are four parts consisted in a CES: the optical system, infrared detector, scanning system and signal processing electronics (SPE). The optical system redirects the collimated energy from the Earth or other warm body toward the objective lens of the system. As the optical system is rotated by the scanning system, a cone shaped scan path is produced. The scanning cone in the CES is shown in Fig. 1.

The Earth's infrared radiation energy is focused through an optical filter and onto an infrared detector. The filter limits the spectral response of the detector to the CO_2 absorption band (14–16 μm wavelengths). As the FOV scans from space to Earth and back to space repetitively, the detector generates electrical signals. The signals are sent to the SPE after gained and preconditioned [3]. Two pulses are generated corresponding the transition of infrared radiation. Then these pulses are processed with a reference pulse, which was generated in the optical system once per cycle, to determine the satellite attitude.

The attitude measurement geometry is shown in Fig. 2. The axis of optical system of CES is parallel to the satellite axis.

Assumed that CES is installed along the satellite's pitch axis Y, the angle between the sight line and the axis of CES is β. The sight line is rotated by the motor to generate a conical FOV. The direction is counter-clockwise according to right-hand rule. The attitude reference pulse generator is mounted on the YOZ coordinate plane.

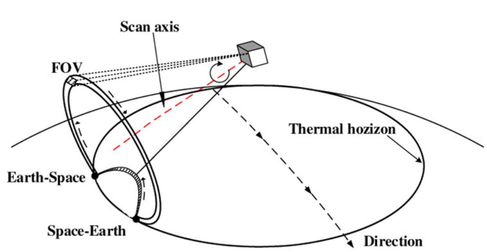

Fig. 1. Operation principle of CES.

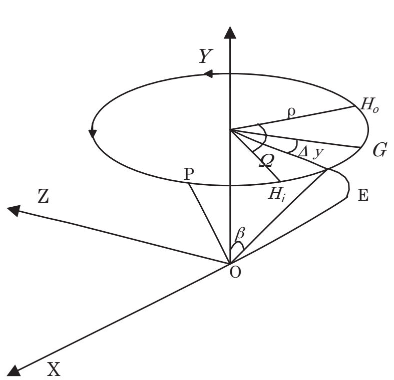

Fig. 2. The attitude determination geometry of CES.

Table 1
Angular diameter of the Sun, Moon and Earth on different orbits.

Orbit	Angular diameter (°)		
	Sun	Moon	Earth
LEO (height = 1000 km)	0.5331	0.5180	81.6747
MEO (height = 2000 km)	0.5331	0.5180	74.5460
GEO (height = 35786 km)	0.5331	0.5711	17.1854

When the sight line crosses the point G on the YOZ plane at the moment t_G, a reference pulse (REF.P) is generated. E is the space vector pointing to the Earth, ρ is view angle radius of the Earth disk. The sight line P scans from space to the Earth across point H_i at the moment t_i, a hardware circuit output the Space/Earth pulse (S/E.P). While that scans from the Earth to space across point H_o at the moment t_o, the circuit outputs the Earth/Space pulse (E/S.P). CES obtains the scan period T based on the periodical attitude reference pulse. The Earth chord width is Ω. It is equal to the angle that the sight line rotates from H_i to H_o around Y-axis. The space-Earth angle width is W_i from the space-Earth cross point H_i to the reference point G. The Earth-space angle width is W_o from the reference point G to the Earth-space cross point H_o. The attitude deviation of the pitch axis Y is Δy. A photoelectric encoder outputs 10,800 grating clock pulses in each scanning circle based on the reference signal, i.e., each pulse is equivalent to 2′. There are the equations between these variables:

$$W_i = \frac{t_G - t_i}{2} \times 360^\circ = N_{Gi} \times 2' \tag{1}$$

$$\Omega = \frac{t_o - t_i}{2} \times 360^\circ = N_{oi} \times 2' \tag{2}$$

$$\Delta y = W_i - \frac{\Omega}{2} \tag{3}$$

where N_{Gi} and N_{oi} are the numbers of the grating clock pulses between two moments. It can be clearly seen that the key of attitude determination is the chord width Ω and space-Earth angle width W_i.

When the Sun is in the CES' scan track, it will enter the FOV and stimulate the corresponding infrared detector with its tremendous energy to generate a pulse signal. After being processed, the pulse signal will be mixed with S/E.P or E/S.P produced by the sensor scanning the infrared Earth, resulting in attitude calculation error. The radiation energy of a full Moon equals to that of the Earth, which also produces misleading pulse signal to make attitude calculation error. Therefore, it is necessary to identify and suppress the interference of the Sun and the Moon in the Earth signal and an elaborate analysis of interference mechanism is the key to solve the problem.

3. Mechanism analysis of the Sun and the Moon interference

According to the working principle of CES and DCES, the output signal of sensor is related to the scanning time on the celestial bodies and the radiation intensity of the corresponding planet.

Firstly, the scanning time on the celestial bodies is related to the angular diameter on the celestial bodies and the scanning speed of the FOV. The scanning speed is constant on DCES, therefore, the differences among celestial bodies are depended on the angular diameters of them respectively.

The diameter of the Sun is 1.392×10^6 km, that of the Moon is 3.475×10^3 km and that of the Earth is 1.27×10^4 km, radius is 6.371×10^3 km. For a CES mounted on a near-Earth satellite, the view angular diameters of the Sun, the Moon and the Earth are shown in Table 1. The position relationship of the Sun, the Moon,