ESPRIT SUN SENSOR: RESULTS AND FUTURE USE

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ABSTRACT

A triple slit, digital sun sensor was designed for the ESPRIT rocket to gather sun angle data. It was modeled after the sun sensor used for the Oscar-10 satellite because of its low cost and simplistic design. Based on the results, this sun sensor has proven to be an effective instrument aboard a spin stabilized payload.

1. INTRODUCTION

Rockets and satellites require information about their relative orientation in space in order to determine attitude. The sun provides a good reference point because its position is fixed in space which makes a sun sensor a valuable instrument aboard any spacecraft. The ESPRIT rocket utilizes a sun sensor to help determine the attitude of the payload and aid in the post flight analysis of other onboard instruments.

A typical sun sensor provides the sun's angle relative to the payload's coordinate reference frame. Analog and digital forms of these sensors are available, but the latter is more desirable for several reasons. First, telemetry bandwidth is normally reduced with a digital design. This is due to the data being transmitted only when it is collected. Secondly, digital circuitry is typically more energy efficient.

The proposed sensor described in this paper is a derivative of the sun sensor design for the Oscar 10 satellite [1]. Though the basic theory of these two designs is analogous, there are many differences which make the ESPRIT sun sensor more promising. The sun sensor used for the Oscar-10 satellite uses an oscillator that varies with the angular velocity of the spacecraft and utilizes a single variable to calculate the sun angle. In the ESPRIT Rocket sun sensor design, a constant frequency is implemented. This results in the use of two variables for the sun angle calculation which is described later. Also, the circuitry is different in both designs. The ESPIRT Rocket will be rotating much faster than the Oscar-10 Satellite, so a microcontroller operating at a high frequency was utilized for the design so that precise timing could be achieved. The microcontroller was also used to transfer data, via an RS-232 driver, to the payload's PCM. Finally, the sun sensor designed here utilizes a third photodiode and vertical slit for additional data analysis. Besides these differences, the overall theory of sun angle calculation is basically the same used for the Oscar 10 satellite.

2. THEORY OF OPERATION

This digital sun sensor senses the sun by means of three photodiodes. These photodiodes provide pulses that trigger multiple counters within a microcontroller. Two of the photodiodes lie behind slits oriented 30 degrees to the spacecraft's spin axis and are used for sun elevation angle related to the payload longitudinal axis. The remaining photodiode lies behind a vertical slit and is used to determine rotation rate of the payload.

This design utilizes the payloads constant rotation. As the sun passes the face of the sun sensor, pulses are generated by the photodiodes. The sun effectively traces an arc around the spacecraft (see Fig. 1).

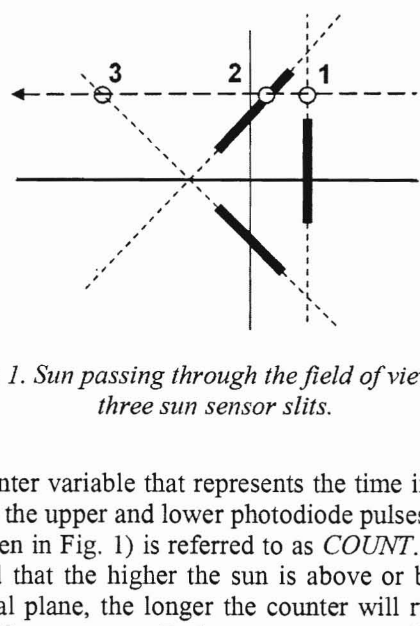

Figure 1. Sun passing through the field of view of the three sun sensor slits.

The counter variable that represents the time interval in between the upper and lower photodiode pulses (event 2 and 3 seen in Fig. 1) is referred to as *COUNT*. It should be noted that the higher the sun is above or below the equatorial plane, the longer the counter will run. If the spacecraft rotates at R degrees per second and the counter counts at f counts per second, the arc angle in degrees is as computed by,

$$arc = \frac{R}{f} \cdot COUNT \qquad (1)$$

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The variable *RCOUNT* denotes the time difference between each consecutive pulse of the photodiode behind the upper slit (event 2 seen in Fig. 1). This information will enable the calculation of the rotational velocity. If the frequency of the counter counts at f counts per second and *RCOUNT* is the counter value after one rotation, then the rotational speed of the spacecraft in degrees per second is,

$$R = \frac{360^\circ \cdot f}{RCOUNT} \tag{2}$$

There are a few other equations that need to be addressed before the sun angle can be calculated. As seen in Fig. 2, trigonometric relations can be applied by dividing the arc angle into two parts.

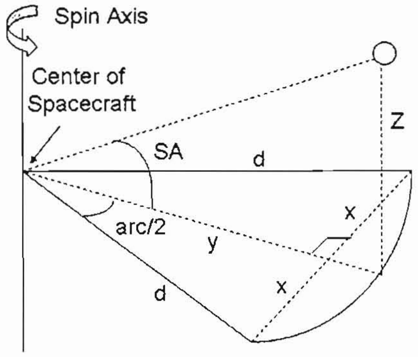

Figure 2. SA is the sun angle, d is the distance from the center of the spacecraft to the photodiodes, and x, y, and z are temporary unknowns and are used only to show the trigonometric relationships.

From Fig. 3,

$$\tan(60) = \frac{z}{x} \tag{3}$$

Equations formed from Fig. 2 yield,

$$\begin{aligned} SA &= \tan^{-1}\frac{z}{d} \\ &= \tan^{-1}\left(\frac{x}{d}\times\frac{z}{x}\right) \\ &= \tan^{-1}\left(\tan 60^\circ \times \sin\left(\frac{arc}{2}\right)\right) \end{aligned} \tag{4}$$

By combining the result from Eq. 4 and Eq. 1,

$$= \tan^{-1}\left\{\tan 60^\circ \times \sin\left[\frac{RCOUNT}{2f}\right]\right\} \tag{5}$$

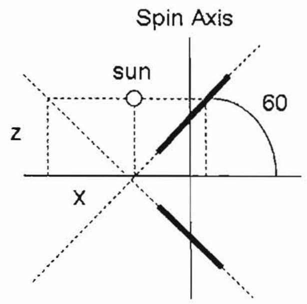

Figure 3. This is a view of the sun positioned at half the arc angle that it traces around the spacecraft.

Finally, substituting the rotational velocity of the payload R (Eq. 2) into Eq. 5 results in the sun angle seen in Eq. 6.

$$SA = \tan^{-1}\left(\tan(60^\circ)\cdot\sin\left(\frac{180^\circ \cdot COUNT}{RCOUNT}\right)\right) \tag{6}$$

3. RESULTS

The first deployment, 2nd stage motor separation, occurred at 62.19 seconds Mission Elapsed Time (MET). As seen in Fig. 4, this separation caused the most significant portion of the coning angle and coning period. Before this point there appears to be a slight amount of coning but there isn't enough data to draw conclusions on coning angle or coning rate before motor separation. Nosecone deployment occurs next at 66.17 seconds MET. There is a slight perturbation due to this deployment in the coning period, refer to Fig. 4, but not as significant as the one caused by motor separation. The final deployment in this data is the boom deployment at 69.18 seconds MET. This deployment has no significant effect on the angle data but has a quite apparent effect on the rate data. Rate falls from 6 Hz to 5.45 Hz, approximately.

Coning angle, Fig. 5, is found using a MATLAB™ script to find the maximum and minimum angles in the angle data set. The difference between consecutive maximum and minimum angles is then plotted as the coning angle. The average of this data is taken as the coning angle over flight from 69 seconds to 357 seconds MET, which is 12.89 degrees coning full angle.

Coning period, Fig. 6, is found by taking the Fast Fourier transform (FFT) of the angle data. An FFT was taken for the upleg of flight and another for the downleg. The FFT taken for upleg resulted in a coning period of .2618 seconds. The result of the FFT on downleg was .2595 seconds. The difference in the two periods is most likely caused by the time period chosen to take each FFT. For upleg the FFT was performed on data after payload deployments up to apogee. The FFT for downleg was performed on data from apogee to the point before the payload went into flat spin, which is not precisely known.

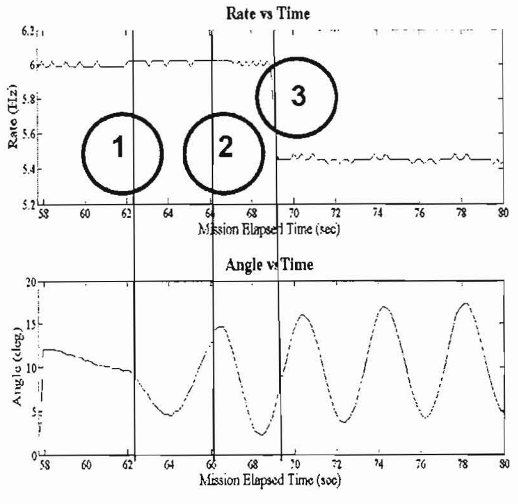

Figure 4. The top plot is of payload rotation rate and the bottom graph is payload angle relative the sun.

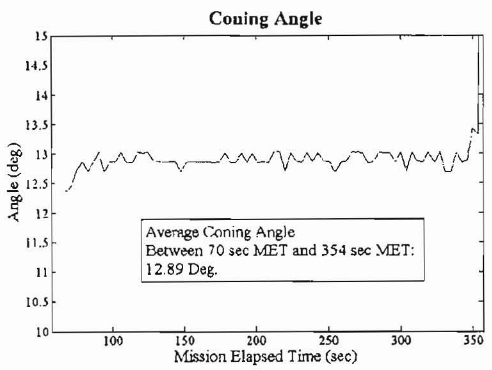

Figure 5. Coning angle from 57 seconds MET.

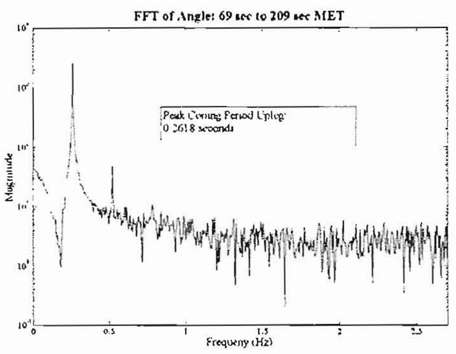

Figure 6. Fast Fourier transform (FFT) of angle data y.

4. ATTITUDE ANALYSIS

In order to determine the attitude of the spacecraft, the TRIAD method will be implemented. This method uses two sensing sources such as a magnetometer and sun-sensor to determine three orthogonal vectors defining the attitude of the spacecraft with respect to a body reference frame. Then using models such as the Earth's Magnetic field model and the equation of time which models the location of the sun, the attitude of the spacecraft is then defined with respect to an inertial frame. A Direction Cosine Matrix (DCM) is formulated which creates a matrix transformation from the body frame of the spacecraft to the inertial frame. The TRIAD method is widely used and ideal for the ESPRIT sounding rocket because it is simple to implement and still produces accurate results. The only issue when using the TRIAD method is that the accuracy relies upon the accuracy of the data. If there are associated errors within the magnetometer or the sun-sensor, they will carry through to the DCM and effect the accuracy of the attitude determination. For the ESPRIT case, a student built 3-axis magnetometer was used with a sun sensor to determine the attitude in the body frame. The Department of Defense World Magnetic Model (WMM) for 2005-2010) was used for the Earth's magnetic field. A MATLAB™ script was used for the equation of time. Both the equation of time and WMM give the attitude of the spacecraft within the geodetic reference frame.

To fully determine the attitude of the spacecraft when using a magnetometer, a second sensing source is needed. A sun-sensor is a very simplistic and effective way to accomplish this. The sun sensor gives the direction of the sun-vector with respect to the body of the satellite. Fig. 7 shows the sun sensor used in the ESPRIT payload. For attitude determination, the sun sensor gives two important pieces of information. As seen in Fig. 7, there are 3 different labeled slits corresponding to a separate sun sensor behind each slit. The slits define a plane and record data when the sun passes through each plane. Slits 1 and 3 are used in combination to determine the incident angle of the sun with respect to the body frame. This is shown in Fig. 8 where θ is defined as the angle between the sun-vector and *i-j* plane. Slit 3 gives a specific time where the *i-k* plane is parallel or coplanar to the incoming sun vector. This combination of both pieces of information gives an incident angle of the sun-vector at a specified time in flight.

Figure 7. Sun Sensor for ESPRIT Payload.

Figure 8. Sun Angle.

In order to define this sun-vector in the body frame of the satellite, the sun-vector must be broken down into i, j, and k components. First, the sun vector can be treated as a unit vector with a magnitude of 1. In addition, the sun-sensor is aligned along the i-axis and only generates data when the sun crosses the i-k plane, therefore the j-value of the sun vector will always be zero. This is shown in Fig. 8. Simple geometry is shown in Fig. 9, which defines the i and k components of the spacecraft.

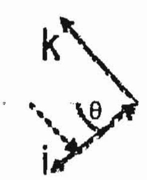

Figure 9. Geometry of Sun Angle.

By using Fig. 9, the red dashed lines show the sun-vector broken down into its i and k components in the body frame. The cosine and sine of the angle can be used to define the vector components. The sun vector is fully defined in the body frame in Eq. 7.

$$\begin{Bmatrix} i \\ j \\ k \end{Bmatrix}^{B} = \begin{Bmatrix} \cos\theta \\ 0 \\ \sin\theta \end{Bmatrix} \tag{7}$$

5. FUTURE PROSPECTS

It is possible to reuse the ESPRIT sun sensor design for future missions with only minor modifications to hardware and software. Ideally, the next sun sensor design would be packaged in a single unit which is universal for applications. The current packaging was specific only to the ESPRIT payload. The entire design, using surface mount components, could easily fit into a single, compact package mountable on most payloads.

The sun sensor design for the ESPRIT payload was adequate for the needs of the mission but could be upgraded for better performance. One way to improve the design would be investigate photodetectors that offer a faster response to changes in sun light such as charged coupled devices, or complementary metal oxide semiconductor devices. Another way to improve the design is to increase the sampling rate. Sampling rate may be increased by adding multiple sun sensor units around the payload's perimeter. To optimize the sensor resolution, each package needs to be matched to one another.

This sun sensor could also be designed for missions occurring at night. A sounding rocket launched during one of the moon's full phases could use it to find attitude in a similar fashion to that of the sun sensor.

This sensor can also be used aboard satellites, considering the theory was derived from a satellite application. An additional set of photodiodes, perpendicular to the original set, could add a second axis of operation making a two axis sun sensor. However, there is the limitation that the satellite must be spinning for the sun sensor to work.

6. CONCLUSIONS

The ESPRIT sun sensor demonstrated its effectiveness to operate on a spin stabilized sounding rocket during a daylight launch. The sensor was successful in extracting data to calculate the sun angle, the rotational velocity of the payload, and coning angle. With minor hardware and software changes, derivatives of the ESPRIT sun sensor may be made as relatively simple, cost effective alternatives to current sun sensors.

7. REFERENCES

1. James Miller, "Sensorship-A Question of Attitudes," *Amsat-UK's Oscar News*, 1994. (Amsat. 8 May 2007 http://www.amsat.org/amsat/articles/g3ruh/113.html)