Platform for Attitude Control Experiment (PACE)
An Experimental Three-Axis Stabilized CubeSat

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Abstract

Owing to their low cost, fast development time, and multi-discipline educational purpose, CubeSats have been widely advocated by universities in recent years. However, few have employed three-axis stabilization schemes due mainly to the limitation of power and mass. The PACE, Platform for Attitude Control Experiment for short, is a three-axis stabilizing CubeSat developed at the National Cheng Kung University. It is a 20x10x10 cm³ double cube satellite weight less than 2-kg. A miniature momentum wheel is employed to achieve stability alone the pitch (orbit normal) axis. Magnetic coils are used to generate control torques to stabilize roll and yaw axes. The attitude sensors used in the design include a three-axis magnetometer, a three-axis gyro, and coarse sun sensors. In addition, the development of the PACE exploits MEMS technology in the design and fabrication of the payload and sensors. MEMS temperature sensors are used as payloads for internal and external temperature monitoring. Coarse sun sensors based on MEMS technology are designed and fabricated to facilitate attitude determination and control. The paper delineates the PACE mission and subsystems including structure mechanism, thermal control, on-board computer, telemetry, as well as ground station with the emphasis on the attitude control and MEMS payload.

1. Introduction

A recent development in spacecraft technology is the paradigm shift toward lighter, smaller, cheaper spacecraft design and the employment of distributed constellation [1,2]. CubeSats, which are pico-satellites, have been widely developed by universities around the world to achieve certain scientific missions with lower cost.

Previously, a CubeSat, YAMSAT has been developed in Taiwan by a group of engineers at National Space Program Office (NSPO) and National Cheng Kung University. It is a 10x10x10 cm³ single cubic CubeSat equipped with a micro spectrometer. The launch is scheduled in September, 2004 [3,4].

Existing CubeSats do not employ three-axis stabilization due to the power, mass, and computation budget constraint. The paper describes the design of a three-axis stabilizing CubeSat, PACE (Platform for Attitude Control Experiment),...
that is currently under development at National Cheng Kung University, Taiwan.

The primary mission of the PACE is to conduct attitude control experiments. The secondary mission of the PACE is to test indigenously design and fabricate MEMS sensors in space including temperature sensors and coarse sun sensors. Figure 1 illustrates the configuration of PACE.

PACE satellite is developed by the students and staff at NCKU. The CubeSat is indeed an outcome of the web-based course “Space System Engineering” organized by a term of university professors and experts in industry, under the support of ministry of Education. During the course, a consensus was reached to develop a CubeSat by teaming students and staff in the University. This paper describes the development of PACE, with emphasis on its unique feature of three-axis stabilization and MEMS sensors. The paper is organized as follows, In Section 2, each subsystem of the PACE satellite is described. Section 3 emphasizes on the attitude control and determination of the PACE satellite. The application of MEMS sensors in space is discussed in Section 4. The conclusion is given in Section 5.

2. PACE Satellite Overview

To perform 3-axis stabilization require more power and weight, however the PACE satellite, which has to be placed in the launcher tubes P-POD [5] developed by Calpoly must comply with the CubeSat standard. Each P-POD is capable of housing 1 to 3 CubeSats. The power/mass budgets are highly restrictive. Therefore, a 20x10x10 cm³ design is selected. The characteristics of PACE reflect the constraints and the available launch opportunities. The characteristics of PACE are tabulated in Table 1. The PACE satellite is comprised of a set of subsystems to provide attitude stabilization, power system, data handling, communication, structure and mechanism system, and thermal control. Figure 2 depicts the hierarchy of PACE project.

Table 1 PACE Characteristics

<table>
<thead>
<tr>
<th>Satellite Weight</th>
<th>Less than 2-Kg</th>
</tr>
</thead>
<tbody>
<tr>
<td>Dimension</td>
<td>20x10x10 cm³</td>
</tr>
<tr>
<td>ADCS</td>
<td>3-axis stabilization</td>
</tr>
<tr>
<td>Operating Orbit</td>
<td>Near-circular orbit 600 km, inclination 98(deg) (TBD)</td>
</tr>
<tr>
<td>Life Time</td>
<td>2 months</td>
</tr>
<tr>
<td>Launch</td>
<td>Scheduled in 2005</td>
</tr>
</tbody>
</table>

Figure 1  PACE Satellite Configuration

Figure 2  PACE project hierarchy
PACE operational modes were carefully planned in order to ensure safe operation and meet the performance require. PACE satellite operates in three different modes, stand-by mode, 3-axis mode, and safe mode. The mode transition is based on two criteria, power condition, and attitude condition. Figure 3 shows the mode transitional diagram of PACE satellite.

![Operation mode transition](image)

**Figure 3  Operation mode transition**

### 2.1 Structure and Thermal Control Subsystems

The requirements of PACE SMS and TCS are defined according the mission, orbit, and the size/weight constraints. The total mass of the PACE is limited to 2-Kg. The Center of Gravity (CG) is located within 2 cm from the geometrical center. Also, the structure system is required to maintain each component within its required temperature range during all mission phases.

PACE is constructed with 7075-T6 aluminum alloy in order to avoid thermal mismatching between P-POD and PACE. Due to thermal protection requirement all printed circuit boards (PCB) are fixed by the slot design. Figure 4 illustrates the allocation of each component. To minimize power consumption, a passive thermal control through insulation is adopted in PACE. Extensive analyses are conducted to ensure that all critical units are within allowable limits. Figure 5 shows the temperature variation of the battery in the worst hot and cold condition.

The antenna deployment mechanism is assembled by a set of nickel-chromium resistor and nylon wire. The design is lightweight and simple. Moreover, the separating velocity can be adjusted to meet the requirements. After the PACE is released from the P-POD, a kill switch turns on the power. When the nylon wire is melt by nickel-chromium resistor, the antennas are deployed. Consequently, the telemetry subsystem is capable of transmitting Status of Health (SOH) by Morse code via VHF antenna. Figure 6 shows the antenna deployment mechanism design.

![Exploded view of PACE](image)

**Figure 4  Exploded view of PACE**

![Battery temperature variation](image)

**Figure 5  Temperature variation of battery**

![Antenna deployment mechanism design](image)

**Figure 6  Antenna deployment mechanism design**
2.2 Command & Data Handling

The on-board computer which is also called Command and Data Handling subsystem (C&DH) performs data processing, computation, and satellite bus maintenance. After searching for available components, we selected an 8051-class microprocessor, C8051F020, as the main computer. It is a mixed-signal microcontroller with enhanced 8051 core which is up to 25 MIPS [6]. Since the microchip integrates analog signal circuitry, it simplifies interface circuit design. A 64KB SRAM is connected with the microprocessor due to the needs to store SOH, MEMS sensor measurements, and attitude data. Figure 7 illustrates the electrical block diagram of the PACE satellite.

Considering the computational load and resource consumption from Attitude Determination and Control Subsystem (ADCS), a master-slave architecture has been employed to separate these tasks from C&DH. We selected two identical microprocessors. Master microcontroller performs C&DH operation, and slave performs ADCS. Two microcontrollers are linked by serial interface, and digital I/O.

2.3 Telemetry, Tracking & Command

The Telemetry, Tracking, and Command (TT&C) subsystem facilitates the half-duplex communications between the PACE and the ground station. Commands in the form of RF signals are received, and demodulated by MX614 modem chip, and then passed to on-board computer for command decoding. The telemetry subsystem, on the other hand, transmits the SOH, and data of the sensor back to ground station. The format of command/telemetry is compatible with the AX.25 protocol with a data rate of 1200 bps via UHF band. To facilitate the TT&C functions, a ground station has been established at NCKU for satellite tracking. The ground station is capable of transceiving UHF, VHF, and HF signal. A Yagi antenna is used to transmit and receive signal from PACE. The antenna is computer-controlled so as to track the satellite automatically. Figure 8 and 9 illustrate the antenna and transceiver at the ground station. For safety reason, we also
select VHF band to broadcast vital SOH by Morse code at constant time interval.

![Ground station antenna](image)

**Figure 8** Ground station antenna

![Ground transceivers](image)

**Figure 9** Ground transceivers

### 2.4 Electrical Power Subsystem

The Electrical Power Subsystem (EPS) is responsible for power generation, storage, regulation, and distribution. The power is provided by 20 pieces of gallium arsenide (GaAs) solar cell mounted on the surface panel. Considering the attitude of PACE in space, the average power generation is 2.53W. Solar cell efficiency is taken into account. Figure 10 depicts the simulation of power generation.

![Power generation simulation](image)

**Figure 10** Power generation simulation

The power is converted to 5V first, in order to charge three 2200mAH Lithium-Ion batteries manufactured by E-One Moli Energy Corp. Battery charger and protector are used to prevent from overcharge of the battery. The power bus provides 5V and 12 V regulated source for component onboard. The power generation, consumption, and battery status is monitored by C&DH. The latter is responsible for power management and mode switching by considering the status of power supply and storage. Figure 11 illustrates the block diagram of EPS.

![EPS block diagram](image)

**Figure 11** EPS block diagram

### 3. Attitude Determination and Control Subsystem

#### 3.1 Introduction

The attitude determination and control subsystem (ADCS) of the satellite is to
ensure attitude acquisition, moment dumping, stabilizing and pointing control. The subsystem obtains measurements from the on-board sensors and command subsystems and, in turn, provides control commands to drive actuation devices. The configuration of the PACE ADCS system includes a three-axis magnetometer, coarse sun sensor, three orthogonal magnetic coils, and a micro momentum wheel. The ADCS employs a momentum-biased attitude stabilization scheme that has been widely used in small satellites. As the rotation axis of the micro wheel is perpendicular to the orbit plane, the pitch axis stability can be achieved by controlling the wheel speed. The momentum bias provides couplings for the roll-yaw (x-z) systems. Through magnetic coil control along the pitch (y)-axis, the precession and nutation along the roll-yaw system due to environmental torques can then be stabilized. Further, magnetic coils along the roll and yaw axes, respectively, are used to facilitate momentum dumping.

The dynamic equations of motion of a rigid spacecraft are [7]:
\[
 J \frac{d\omega}{dt} = N_{\text{dist}} + N_{\text{cont}} - \omega \times J \omega - \omega \times h - N_{\text{wheel}}
\]
\[
 \frac{dq}{dt} = \frac{1}{2} \Omega q
\]
\[
 \frac{dh}{dt} = N_{\text{wheel}}
\]
where \( J \) is the moment of inertia tensor, \( \omega \) is the spacecraft angular velocity vector, \( q \) is the quaternion that is used to characterize the attitude of spacecraft, \( N_{\text{dist}} \) stands for the disturbance torque acting on the spacecraft, \( N_{\text{cont}} \) is the control torque, \( h \) is the total angular momentum of the momentum wheel, \( N_{\text{wheel}} \) is the torque on the wheel, and \( \Omega \) is a 4×4 matrix that bears the form
\[
\Omega = \begin{bmatrix}
0 & \omega_3 & -\omega_2 & \omega_1 \\
-\omega_3 & 0 & \omega_2 & \omega_1 \\
\omega_2 & -\omega_1 & 0 & \omega_3 \\
-\omega_1 & -\omega_2 & -\omega_3 & 0
\end{bmatrix}
\]
in which \( \omega_1, \omega_2, \) and \( \omega_3 \) are the components of the angular velocity vector \( \omega \). In the PACE ADCS design, the control torque \( N_{\text{wheel}} \) is modulated to stabilize the pitch axis so that an appropriate momentum is generated. The magnetic control torque \( N_{\text{cont}} \) is generated through magnetic coils. The magnetic control system is used to facilitate initial attitude acquisition, rate detumbling, and momentum dumping.

The design of ADCS must account for disturbances. The sources of disturbance are gravity gradient torque, aerodynamic torque, solar radiation torque, internal residual magnetic torque. Figure 12 depicts the simulation results of disturbance for the PACE at the designed orbit and attitude. The most significant disturbance torque is due to the aerodynamic effect which may result in a torque of \( 5.37 \times 10^{-8} N \cdot m \). The analysis is used in the sizing and current circuit design of the magnetic coils.
Table 2 lists the ADCS operating modes and the sensors/actuators being used.

<table>
<thead>
<tr>
<th>Hardware</th>
<th>Standby Mode</th>
<th>3-Axis Mode</th>
</tr>
</thead>
<tbody>
<tr>
<td>Magnetometer</td>
<td>◎</td>
<td>◎</td>
</tr>
<tr>
<td>Magnetic coils</td>
<td>◎</td>
<td>◎</td>
</tr>
<tr>
<td>Coarse sun sensor</td>
<td>◎</td>
<td>◎</td>
</tr>
<tr>
<td>Momentum Wheel</td>
<td>◎</td>
<td>◎</td>
</tr>
<tr>
<td>Gyroscope</td>
<td></td>
<td>◎</td>
</tr>
<tr>
<td>Control Law</td>
<td></td>
<td>◎</td>
</tr>
<tr>
<td>B-dot Control</td>
<td>◎</td>
<td>◎</td>
</tr>
<tr>
<td>Momentum Wheel Control</td>
<td>◎</td>
<td>◎</td>
</tr>
<tr>
<td>Momentum Dump Control</td>
<td>◎</td>
<td>◎</td>
</tr>
<tr>
<td>Pointing Control</td>
<td>◎</td>
<td>◎</td>
</tr>
</tbody>
</table>

◎: on

3.2 Standby Mode

The main purpose of standby mode is moment dumping to stabilize the angular rate of the PACE satellite and acquires the Earth-pointing attitude roughly. As depicted in Table 2, the mode employs magnetometer, coarse sun sensor, magnetic coils and B-dot control law.

The control torque from the magnetic coils $N_{\text{cont}}$ can be expressed as

$$N_{\text{cont}} = M \times B$$

where $M$ is the dipole moment of the coil which depends on the area, number of turns, and current, and the variable $B$ is the geomagnetic field vector, which is available through the on-board magnetometer. By the B-dot control law [10]:

$$M_i = -K_b \dot{B}_i$$

where $K_b$ is a parameter to be determined, and $\dot{B}_i$ is the rate change of geomagnetic field, which is obtained through differential measurement of magnetometer. When the mode is activated, the angular rates along $X_{\text{Body}}$ and $Z_{\text{Body}}$ axes are expected to approach zero and that about $Y_{\text{Body}}$ rotates at two times the orbital rate as shown in Figure 13.

3.3 Three-Axis Mode

3.3.1 Momentum Wheel

PACE ADCS design is featured with a momentum wheel, shown in Figure 14, which was packed and tested in Taiwan. The momentum wheel is capable of providing 0.0001N.m torque, and 0.01N.m-s angular momentum as in Table 3. Figure 15 shows the test of tracing angular velocity, it is
suitable for the PACE satellite.

![Momentum wheel images](image)

Figure 14. (a) Un-packaged Momentum wheel (b) Finished product.

![Angular velocity tracing result](image)

Figure 15. Angular velocity tracing result.

Table 3 Momentum Wheel Specification

<table>
<thead>
<tr>
<th>Source</th>
<th>Specification designed by PACE team and produced by CSIST</th>
</tr>
</thead>
<tbody>
<tr>
<td>Motor</td>
<td>FAULHABER AMW1628B</td>
</tr>
<tr>
<td>Input voltage</td>
<td>12V</td>
</tr>
<tr>
<td>Power consumption</td>
<td>0.498 – 0.616W</td>
</tr>
<tr>
<td>Operation angular velocity</td>
<td>2000 – 4000 rpm</td>
</tr>
<tr>
<td>Angular momentum</td>
<td>0.01 N-m-s</td>
</tr>
<tr>
<td>Torque</td>
<td>0.0001 N.m</td>
</tr>
<tr>
<td>Weight</td>
<td>300g</td>
</tr>
<tr>
<td>Operation temperature</td>
<td>-40 – 71°C</td>
</tr>
</tbody>
</table>

3.3.2 Attitude Determination

An extended Kalman filter, as shown in Figure 16, is used for attitude determination.

The filter relies on both the measurement equation and dynamic equation to predict next status and parameters of nonlinear system. Figure 17 depicts that after about 1200 sec, the extended Kalman filter can predict satellite attitude accurately.

![System block of Extended Kalman filter](image)

Figure 16. System block of Extended Kalman filter.

![Simulation of PACE attitude determination](image)

Figure 17. Simulation of PACE attitude determination.

3.3.3 Attitude Control

The proper solution towards 3-axis stabilization is enabled Y-momentum wheel to pointing control, and preventing the satellite body rotating. The momentum wheel is commanded to 2000–4000rpm with the spin axis roughly normal to the orbit normal [8,9]

\[
N_{\text{wheel}} = K_{wp} \dot{\theta} + K_{wq} \dot{\theta}
\]
where $K_{wp}$ and $K_{wd}$ are proportional and derivative gains, respectively. The pitch angle $\theta$ is the rotation from the spacecraft body coordinates to the orbital coordinates.

The momentum wheel is subject to the influence of disturbance torque in space, resulting in nutation and precession. A Y-axis coil is adopted to interact with earth magnetic field to exert magnetic torque to eliminate it. The pitch magnetic control law is given by

$$m_y = K_{y1}\dot{B}_y + K_{y2}B_x\phi$$

where $\phi$ is the roll angle and both $K_{y1}$ and $K_{y2}$ are gains.

Besides, due to the output torque in Y-axis of momentum wheel may exceed or less than design range, which may break down the stabilization of X-axis and Z-axis. Hence, X-axis and Z-axis magnetic coils are used to exert torque to enhance or reduce the momentum of wheel. This is known as Momentum Unloading. The roll-yaw magnetic control for momentum dumping is given by

$$m = K(\Delta h) \times B$$

where $\Delta h$ is the momentum to be dumped to avoid saturation.

A simulation was conducted to assess the attitude control design and three-axis stabilization properties. Figure 18 and 19 depict the simulation results of the spacecraft Euler angle and angular velocity responses. The desired goal of $\pm 5^0$ about the roll and yaw axes are achievable and the pitch control accuracy is about $\pm 2^0$. With respect to the rate response in Figure 18 and 19, the steady-state pitch rate is equal to the negation of the orbital rate as desired, shown in Figure 20.
4. MEMS Payload

MEMS technology has already developed in Taiwan’s academic research community for years. Although application to the area of photoelectric, biology MEMS and medicine are popular, application to the space technology has been little. Since PACE employs a passive thermal control scheme, it is desirable to verify the passive thermal control design with a set of newly-developed MEMS temperature sensors. The sensors are placed on several components of the satellite to monitor the temperature variations. Accordingly, the temperature can be monitored, and safety measures can be taken if necessary. In addition, the data measured will be downloaded for verifying the performance of the sensors. In addition, MEMS technology was also extended for fabrication of coarse sun sensors that are essential in PACE for attitude determination. These sensors are placed at the outside panels of PACE to determine the sun direction, based on the signals measured.

4.1 Flexible MEMS Temperature Sensor

The MEMS temperature sensor was developed based on a novel technique having platinum film deposited on a flexible skin. The resistance of platinum film varies with temperature linearly [11]. A simplified MEMS manufacturing process was employed to accomplish that platinum resistors as sensing materials are sandwiched between two polyimide layers as flexible substrates. A cross-sectional view of the sensor is shown in Figure 21(a) and Figure 21(b) depicts the finished products.

The advantage of the flexible skin with a MEMS temperature sensor were light in weight, high frequency response, low power consumption, with a high mechanical flexibility, that can be handily attached on a highly curved surface to detect tiny temperature distribution within a area. The performance of sensor is shown in Figure 22. MEMS platinum resistor temperature sensor has linear output and a sensitivity of 4.5 mV/°C at a drive current of 1 mA.

![Figure 21 (a) A cross-sectional view of the MEMS temperature sensor (b) Flex. skin sensor finished products.](image)

![Figure 22 The performance of MEMS platinum resistor temperature sensor.](image)

4.2 Self-fabricated Coarse Sun Sensor

Since the ameliorate design can work properly in space environment, the present design sensor was made by improved further process on single-crystalline silicon solar cells [12]. Further, in order to fulfill the goal of
assisting in determination of the satellite attitude. The photovoltaic performance of the solar cells was enhanced by increasing the effective absorption of the solar irradiation, which was achieved by modifying the surface texturing structure and applying anti-reflective coating (ARC). The fabrication employed isopropanol and Na (OH) admixture for texture etching, and POCl3 for n+ diffusion. PECVD (Plasma Enhanced Chemical Vapor Deposition) was employed for nitride (Si3N4) deposition which is served as ARC. Shown in figure 23(a) the electrode on the top of the sensor was made with E-beam for gold (Au) film deposition, while the electrode at the bottom was made with E-beam for aluminum deposition. Here the serves as a aluminum film not only serves as a electrode but also serves as a back-surface field (BSF) to increase photovoltaic conversion efficiency. Figure 23(b) present a view of finished products. Each sensor is 24 mm by 12.8 mm in dimension. After the process of co-firing of Au and Al electrode and firing through passivating ARC layer, the photovoltaic conversion efficiency of the fabricated cells has been improved to as high as 14.5-16% under AM1 solar illumination.

In this scientific experiment, we attempted to use the self-fabricate coarse sun sensor plaster on the four outside-walls and top-wall. By comparing the signals measured by different sensors, the PACE Sat can determine its attitude in the orbit. Figure 24 depicts the test result of using four coarse sun sensors packaged on the four side-walls, which simulates that PACE satellite rotated along the Z-axis. The test result of the self-fabricated coarse sun sensor has a consistent wave-like characteristic and can then be used to detect the direction of sunlight.

Figure 23 (a) Coarse sun sensor configuration (b) Finished products.

Figure 24 Coarse sun sensor angle performance.

5. Conclusion

The PACE CubeSat is designed to fulfill 3-axis control objective and verify MEMS technology. Due to the constraints on weight and power, a micro wheel is used to provide pitch momentum bias so as to achieve acceptable pointing accuracy. Although the power consumption of the wheel might be large, preliminary simulation analysis has indicated that a double-cube design is
capable of deliver sufficient power. Since CubeSats are smaller, MEMS sensors are highly attractive. The development of the PACE has utilized the self-fabricated flexible skin MEMS temperature sensors and silicon base coarse sun sensors to take the advantage of high sensitivity, low weight, and low power consumption. The design, fabrication and verification of the MEMS sensors that are to be used in the PACE are delineated. In the future, more in-depth analyses and tests will be conducted to achieve the goal of 3-axis stabilization of a CubeSat.

6. Acknowledgement

The authors would like to acknowledge all those students and staff at NCKU who contributed to the development of PACE satellite. Especially, we would like to thank our colleges Mr. Kevin Chiu, Allen Lee, C.L. Chiang, Warren Lin, Jordon Tsai, Y.F. Tsai, Tsung-Hsin Tsou. During the course of the PACE development, the help from NSPO, CSIST, and Prof. Jer-Nan, Juang are also acknowledged.

7. Reference