Design and analysis of nano momentum wheel for picosatellite attitude control system

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Abstract

Purpose – Recent advances in nano and picosatellite missions and future such missions require faster and accurate pointing accuracies using reaction wheels for attitude control purposes. The ability to put one or three reaction wheels on the spacecraft in the 1-20 kg range enables new classes of missions. The purpose of this paper is to present the detailed design, analysis, and construction of a miniature reaction wheel prototype. The designed pico-reaction wheel promises to fulfill the need for low cost, low mass, low power, high reliability, and high-accuracy attitude control systems for applications such as communications, remote sensing, and space science.

Design/methodology/approach – Details about the design, analysis and development of pico-reaction wheel are discussed. The development status of the system is outlined and the working prototype of the device is described and some preliminary test results are given. Requirements specifications, design and analysis and finite element analysis are covered.

Findings – A fully functional prototype has been developed and testing has been conducted that demonstrated the effectiveness of the device. The pico-reaction wheel offers a new attitude control system implementation strategy for pico and nanosatellite missions that can help to significantly reduce the spacecraft costs. The key to our success has been to design the reaction wheel from ground-up for simplicity.

Originality/value – The designed pico-reaction wheel satisfied all the constraints and requirements. Furthermore, its advantages include scalability and modularity by virtue of using commercial-off-the-shelf components. A pico-reaction wheel has been successfully designed and is now available to pico and nanosatellite builders at a cost that is consistent with low-cost research missions.

Keywords Artificial satellites, Design and development, Spacecraft, Nanotechnology

Paper type Technical paper

Nomenclature

μ = Earth's gravity constant = 3.986×10^{14} km
R = spacecraft orbit radius
F = force
C_{ps} = location of center of solar pressure
C = speed of light (3×108 m/s)
M = magnetic moment
F_s = solar constant = 1,367 W/m^2
A_s = surface area of the spacecraft
q = reflectance factor (range: 0-1)
i = angle of incidence from the sun
D = residual dipole
B = magnetic field
C_{pa} = center of the aerodynamic pressure
ρ = atmospheric density
C_d = drag coefficient (has nominal values between 2-2.5)
v = spacecraft velocity
A = surface area

1. Introduction

Use of satellites for military, scientific, and commercial purposes has been rising year after year. Faced with such a diversity of uses, it has become possible and even desirable to launch a large number of small mini-satellites in place of a few large ones to take advantage of more flexibility in scheduling and orbit deployment at a lower overall cost. However, this means that the components that go on board have to shrink in size and mass also. Furthermore, these components have to operate under a tight power budget, for the power supply that is carried aboard would necessarily be smaller as well.

If the mission of these mini-satellites were to involve precision pointing, it would need an attitude control system. Attitude control systems consist of multiple reaction wheels that spin at a fast enough rates to allow the conservation of angular momentum to generate control torques on the rotation axis, so that the satellite itself can turn about this fixed axis and orient itself in desired directions.

Ma *et al.* has presented the design and development of a prototype reaction wheel with low-power consumption for mini satellites. The reaction wheel uses high-temperature superconductor bearings for higher angular momentum generation and applications in space (Ma *et al.*, 2003, 2000; Zhang *et al.*, 2002). Christian *et al.* (2004) has developed a variable inertia momentum wheel which increases the capability and flexibility for space applications. Argondizza *et al.* (1999)

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and Carabelli *et al.* (2003) have presented the feasibility study of a reaction wheel on active magnetic bearings with a fully operational engineering model.

Reaction wheels for the attitude control on satellites are an important key technology and difficult to miniaturize due to the special nature of their function. In this paper, detailed design, analysis, and construction of a miniature reaction wheel prototype is presented. A detailed finite element analysis is performed to analyze the stress and vibration profiles will be presented. Finally, results from an experimental test run will also be discussed.

2. Design methodology

In an effort to improve performance and reduce the size and mass of the reaction wheel, as well as reduce the cost, design and analysis of a low-cost nano reaction wheel is presented here. The flywheel is in the shape of a disk with open ends so that the mass is distributed as far away from the rotation axis as possible to maximize the moment of inertia, thus producing greater torque.

2.1 Design requirements

The nano reaction wheel was designed based on the requirements presented in Table I.

2.2 Design analysis

To satisfy the design requirements for the nano reaction wheel, several design analysis constraints were performed. The most important of those constraints were the environmental disturbance torques that the satellite will encounter in the low Earth orbit and the slew requirement for satellite's pointing accuracy. These constraints are discussed in detail in the next two sections.

2.2.1 Environmental disturbance torque

To optimally design the moment of inertia for the flywheel and select the motor that produces adequate torque for controlling the pico satellite, several factors affecting the satellite attitude control are considered to determine how much overall torque the reaction wheel can generate to overcome the environmental disturbances that the satellite experiences in the orbit. Factors affecting the attitude control of the satellite come from environmental disturbance torques, mode of attitude determination and control, the reaction wheel itself, and the capacity of the DC motor in the reaction wheel. The moment of inertia for the pico satellite example considered in this paper is shown below:

- $I_x = 1.476 \times 10^{-3}\,\mathrm{kg\,m^2}$;
- $I_y = 3.177 \times 10^{-3}\,\mathrm{kg\,m^2}$; and
- $I_z = 3.129 \times 10^{-3}\,\mathrm{kg\,m^2}$.

2.2.1.1 Gravity gradient torque. Gravity gradient torques is resulting from the fact that two opposing points of the spacecraft have a finite distance in a declining potential field. Only the center of mass of the satellite experiences a static equilibrium:

$$\begin{aligned} T_g &= \frac{3\mu}{2R^3}|I_z - I_y|\sin(2\theta) \\ &= \frac{3(3.986\times10^{14}\mathrm{m^3/s^2})(4.8\times10^{-5}\,\mathrm{kg\,m^2})\sin(2^\circ)}{2(7.078\times10^6\,\mathrm{m})^3} \\ T_g &= 2.82\times10^{-12}\,\mathrm{Nm} \end{aligned} \tag{1}$$

2.2.1.2 Solar radiation torque. The pressure and the torque generated by the radiation emitted by the sun are governed by the solar constant. It is defined as the normal energy flux onto a unit area per unit time, outside of the atmosphere, at one astronomical unit (1 AU = average Earth-Sun distance):

$$\begin{aligned} T_{sp} &= \frac{F_s}{c}A_s(1+q)\cos i(c_{ps} - cg) \\ T_{sp} &= \frac{(1{,}367\,\mathrm{W/m^2})}{(3\times10^8\,\mathrm{m/s})}(0.1\times0.1\,\mathrm{m})(1+0.6)(2\times10^{-3}\,\mathrm{m})\cos(0) \\ T_{sp} &= 1.46\times10^{-10}\,\mathrm{Nm} \end{aligned} \tag{2}$$

2.2.1.3 Geo-magnetic field torque. The spacecraft's motion across the geomagnetic field induces a motional electromagnetic field in the spacecraft which in return interacts with the geomagnetic field, generating a disturbance torque. The magnetic torque imposed on the satellite can be expressed by:

$$\begin{aligned} T_m &= D\frac{2M}{R^3} = (1A\mathrm{m^2})\frac{2(7.96\times10^{15}\,\mathrm{teslam^3})}{(7.078\times10^6\,\mathrm{m})^3} \\ T_m &= 4.5\times10^{-5}\,\mathrm{Nm} \end{aligned} \tag{3}$$

2.2.1.4 Aerodynamic torque. In order to calculate the aerodynamic drag acting on the satellite one must first model the density as a function of altitude. The model for the torque due to aerodynamic effects is:

$$\begin{aligned} T_a &= 0.5[\rho C_d A V^2](c_{pa} - cg) \\ T_a &= 0.5[10^{-13}\,\mathrm{kg/m^3}](2)(0.1\,\mathrm{m})^2(7{,}504\,\mathrm{m/s})^2] \\ &\quad \times(2\times10^{-3}\,\mathrm{m}) \\ T_a &= 1.5\times10^{-14}\,\mathrm{Nm} \end{aligned} \tag{4}$$

Among all the disturbance torques calculated, the geo-magnetic field torque is the maximum torque that needs to be countered for attitude control of the satellite. Hence, this torque is considered for the disturbance torque rejection.

Table I Nano reaction wheel design requirements

Category	Design requirements
Size and mass	Maximum dimensions (volume): 8.5 cm (L) × 8.5 cm (W) × 2.5 cm (H)
	Maximum mass: not to exceed 150 g (including housing, motor, and drive electronics)
Performance	Speed: max 12,000 rpm (variable) Max voltage: either 5 or 12 V DC
	Angular momentum $= \pm 3.7 \times 10^{-5}$ Nms Slew rate = 3°/s Continuous current: ≤0.14 A Continuous torque: ≥0.00545 in-lbs Continuous power: ≤0.7 W Operating temp.: −20 to +100°F Vibration load = 10 g RMS
	DC motor with better characteristics in producing more torque with less power with all other factors remaining relatively constant
Cost	Less than US$1,000
Components	Commercially off the shelf

Total disturbance rejection torque: $T_{RW} = (T_m)$ (margin factor).

With a margin of safety factor = 1.3:

$$T_{RW} = (4.5 \times 10^{-5}\,\mathrm{Nm})(1.3) = 5.9 \times 10^{-5}\,\mathrm{Nm}$$

This is the amount of torque that the reaction wheel should produce to counter the environmental disturbance torques during orbital maneuvers.

2.2.2 Slew maneuver requirement

As a first step towards developing reaction wheel for highly agile spacecraft, the reaction wheel was designed with an average slew requirement of 3°/s or the spacecraft to be able to perform a 30° maneuver in 10 s about a particular axis. In order to complete the 30° maneuver in 10 s, there needs to be an acceleration phase (which will take 5 s) and a deceleration phase (which will take 5 s) as shown in Figure 1. Thus, the calculations are performed for 15° in 5 s. The required slew torque is given by:

$$T = \frac{2 \times \theta \times I}{t^2} \quad T = \frac{2(15^\circ)(\pi/180)(3.177 \times 10^{-3}\,\mathrm{kg\,m^2})}{(5\,\mathrm{s})^2} \tag{5}$$

$$T = 6.7 \times 10^{-5}\,\mathrm{Nm}$$

Thus, the total torque requirement will be:

$$T_{total} = T_{RW} + T = 1.26 \times 10^{-4}\,\mathrm{Nm} \tag{6}$$

2.2.3 Flywheel design

The optimum design of the flywheel for the reaction wheel must be capable of storing one-half of the total angular momentum of the entire satellite. This is necessary to allow the system to spin up from a static equilibrium into a steady-state rotation. The total angular momentum that must be stored in the reaction wheels must be equal to the angular momentum generated during the spin up of the satellite. Based on computer modeling and simulations, the best design that produces fairly high-inertia torque consists of a thick outer ring and thin inner disk. This design allows the maximum mass of the material concentrated away from the axis of rotation, making the torque generation more efficient. Figure 2 shows the cross-section of the designed flywheel. Figure 3(a) and (b) shows the actual PROE model of the flywheel that was fabricated. Actual flywheel was fabricated using brass.

Figure 1 Slew maneuver acceleration and deceleration illustration

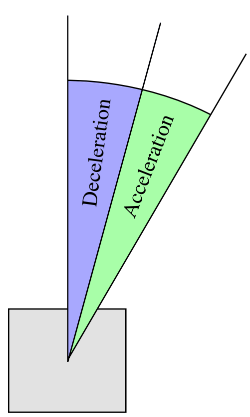
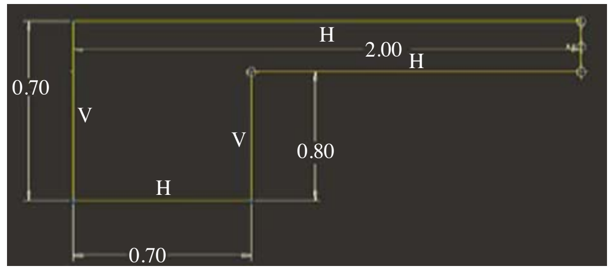


Figure 2 Cross-section of the flywheel design



2.2.4 Motor selection

To select appropriate motor that can generate required amount of torque and speed, several design criteria were considered.

2.2.4.1 Load bearing capacity. The motor load bearing capacity was chosen to be 200 percent of actual required load bearing capacity to ensure that the motor will not be overloaded and run at maximum set limit.

2.2.4.2 Motor speed. The maximum motor speed was chosen to be 200 percent of the actual required calculated speed for the same reasons stated above. These criteria will also depend on other parameters such as power rating, manufacturer rating system, etc.

2.2.4.3 Motor physical size. The selected motor should fit within the volume and the mass budget described in the design requirements.

2.2.4.4 Motor type. A coreless DC motor was selected because of its low-electromagnetic interference which affects digital data transfers and functioning of electronics.

2.2.4.5 Power matching. The selected motor will have a variable combination of operating voltage, operating current, and operating speed to give the required operating torque.

2.2.4.6 Environmental conditions. The selected motor should be capable of withstanding the operational environment, i.e. high vacuum and extreme operating temperature variations when the satellite goes from eclipse to sunlight.

To satisfy the above constraints, a coreless DC motor was chosen which had the following capabilities shown in Table II.

The chosen motor was FAULHABER coreless DC motor 1516 SR 12 V. The specification of this motor is shown in Table III.

2.2.5 Reaction wheel casing

A reaction wheel casing was designed to house the flywheel, motor and associated integrated electronics. The designed casing is lightweight, durable, and strong and easy to assembly and to repair or replace parts. Poly-ether-ether-ketone material was chosen for casing material since it is strong, light weight and NASA space qualified. Figures 4-6 show the assembled and exploded views of the casing parts including flywheel and the top cover design.

3. Reaction wheel finite element analysis

Detailed finite element analysis (vibration analysis) was performed for the flywheel using PROMECHANICA. The vibration analysis was done at the maximum flywheel speed of 10,000 rpm and under extreme launch vehicle loads

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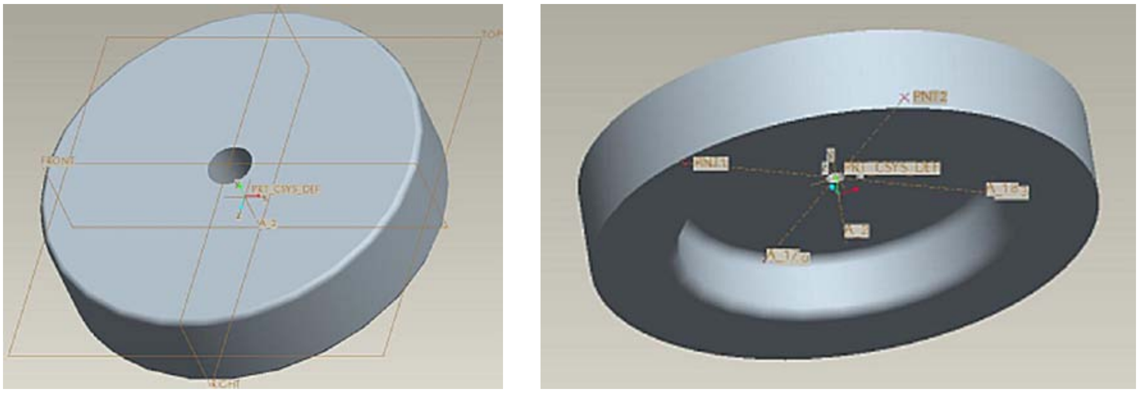
Figure 3 PROE models of the flywheel design



(a) (b)

Table II capability requirments for reaction wheel motor

Torque capability	$T_{total} = T_{RW} + T = 1.26 \times 10^{-4}$ Nm
Speed capability	Variable with a maximum speed of 10,000 rpm
Power consumption	Approximately 1 W with a maximum voltage of 12 V DC
Operating temperature	−20 to +70°C

Table III Specifications of the chosen FAULHABER coreless DC motor

	Required	Actual
Torque	0.126 mNm	0.8 mNm
Max speed	10,000 rpm	12,900 rpm
Power consumption	~1 W	1.04 W
Operating temperature	−20 to + 70°C	−30 to +85°C
Stall torque	–	1.53 mNm
Voltage	–	12 V
Mass	–	13 g

Figure 4 PROE model of assembled casing

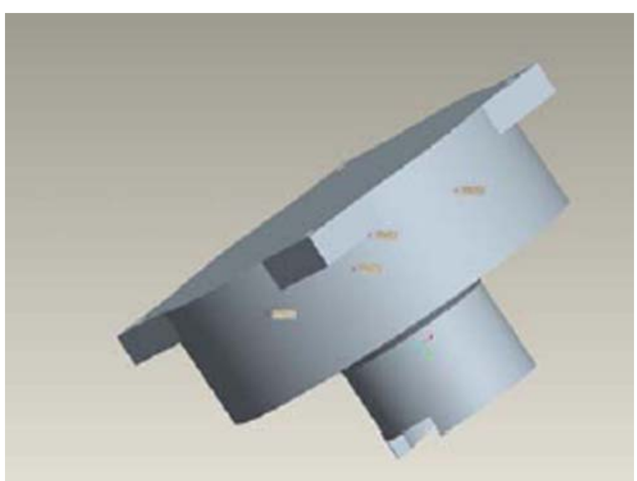


Figure 5 Exploded view

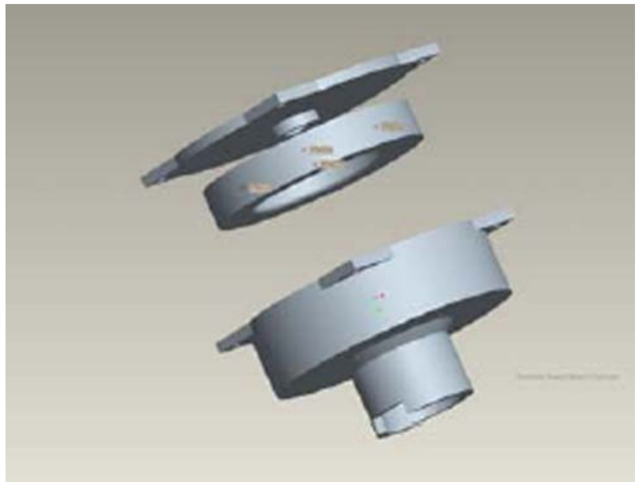
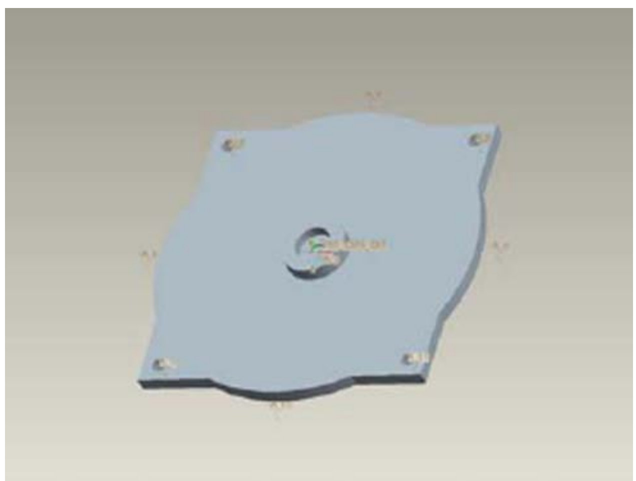


Figure 6 PROE model of the casing top cover



(10 g loading) to analyze the failure modes of the flywheel. The frequency values of the modes shown were very high compared to what the nano reaction wheel will actually experience both during launch and in the orbit. Survival of the flywheel at these extreme loading confirms the longevity of the life of the flywheel factors like wear, misalignment, etc. does not significantly contribute to the failure of the flywheel. The modal frequency values obtained through the analysis are approximately in the range of five to sixth powers than that of actual vibration experienced, i.e. the worst vibration frequency was 8.2×10^{-5} Hz while the actual value experienced is about 50 Hz. Figure 7 shows vibration analysis carried out on a cut-out section of the flywheel as a way of confirming the failure modes. The deformations are shown in Figure 8.

4. Experimental data analysis

Experiments were performed to collect data of angular acceleration and angular velocity of the flywheel with respect to time. These profiles are shown in Figure 9. A laboratory experiment was conducted to suspend the satellite with the flywheel from the roof using a piano wire to test the torque generation capability for slew requirements. The total amount of time for a 30° slew maneuver was approximately 3 s which is within the design constraints.

Figure 10 shows the final assembled PROE model of the nano reaction wheel. Figure 11(a) shows the cross-section of the reaction wheel prototype and Figure 11(b) is the actual prototype of the nano reaction wheel.

5. Summary

A detailed design analysis and construction of a nano reaction wheel has been presented. All the required design constraints have been achieved and the total cost of the reaction wheel is kept low by using only commercial off the shelf products. This reaction wheel design will be extremely useful for nano and picosatellite three axis control requiring stringent pointing requirements. With the overall cost being very low, university nanosatellite and Cubesat programs will benefit from this design. The specifications of the final product along with dimensions (Figure 12) are shown in Table IV.

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Figure 7 Stress analysis of the cut-out section of flywheel

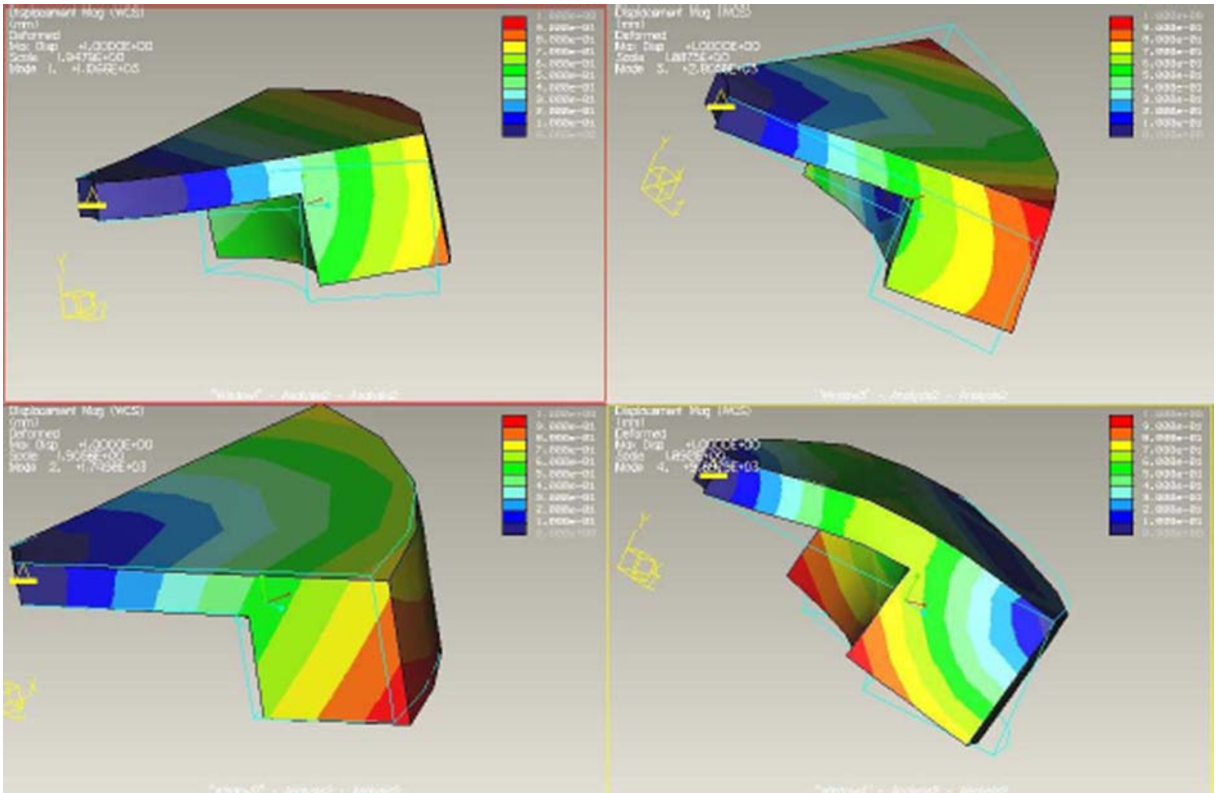

Figure 8 Flywheel deformations at different modal frequencies

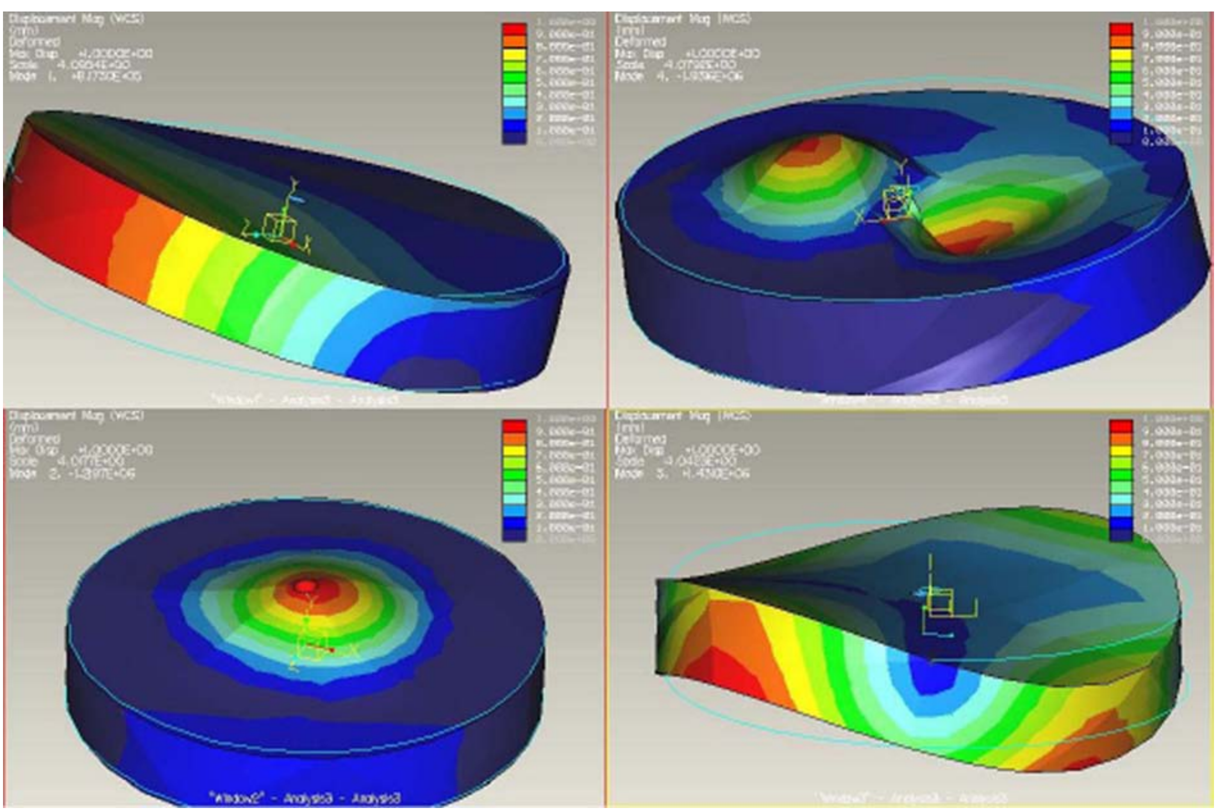

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Figure 9 Experimental data of the reaction wheel velocity and acceleration profiles

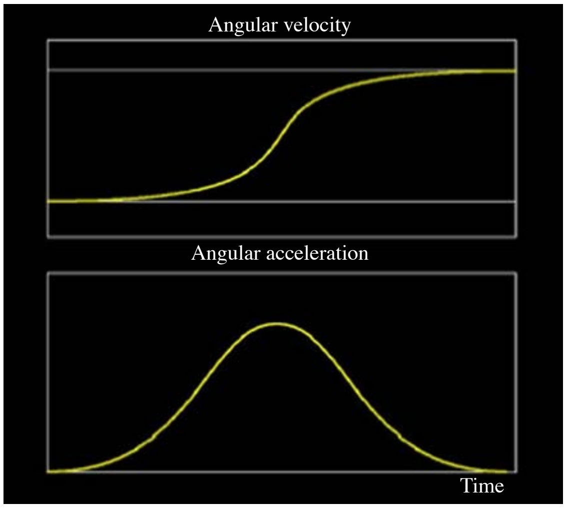
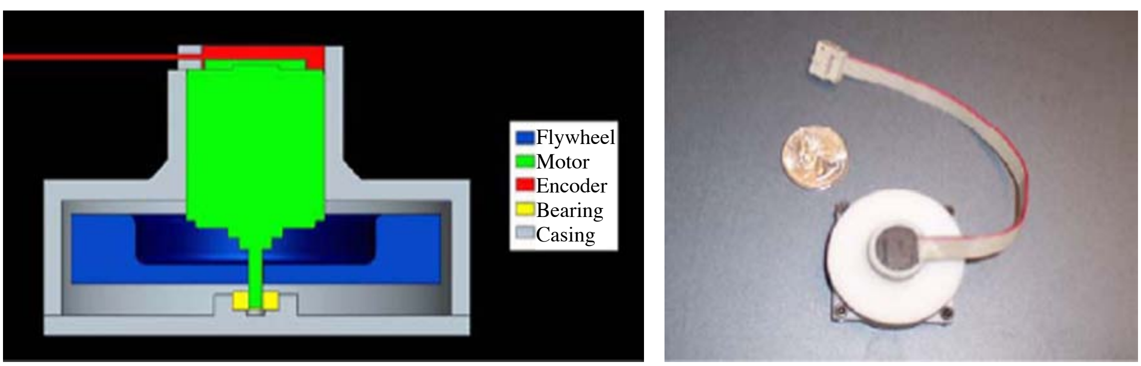


Figure 10 PROE model of the final reaction wheel prototype

(a) **(b)**

Figure 11



(a) **(b)**

Notes: (a) Cross-section of the prototype; (b) actual nano reaction wheel prototype

Figure 12 Nano reaction wheel assembly with dimensions

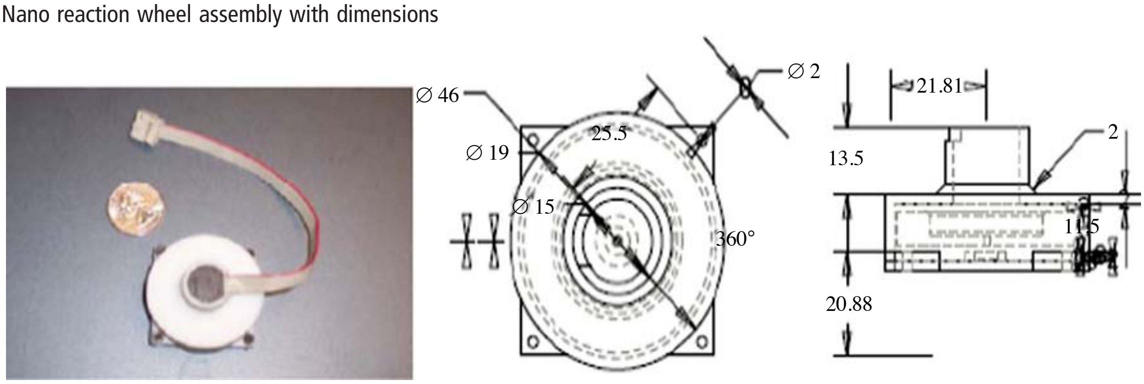


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Table IV Designed nano reaction wheel specification sheet

Parameter	Data
Mass of momentum wheel assembly	~76 g without integrated electronics ~125 g with integrated electronics
Electrical	
Voltage	12 V
Power	1.04 W
Mechanical	
Angular momentum	0.8 mNm
Maximum torque	~10°/s
Slew rate SAT	12, 900 rpm
Maximum wheel rate	13 g
Mass flywheel	~50 Hz at10, 000 rpm
Vibration load	0.8 mNm
Other parameters	
Temperature	−30 to +85°C
Control mode	Speed
Interface control	MCDC 3006S, Encoder IE2-512

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