Application of an Electrospray Thruster in a Nanosatellite
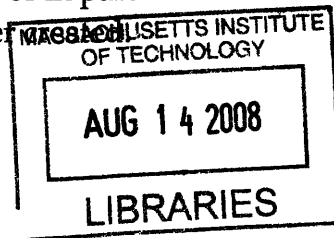
by

Courtney N. Greene

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Application of an Electrospray Thruster in a Nanosatellite

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Abstract

Use of microelectromechanical systems (MEMS) could lead to the mass production of small (<10kg) satellites that are highly reliable and low-cost. To satisfy the satellites propulsive requirements a thruster will need to have a high specific impulse yet still be small enough to not contribute significant weight to the system. An electrospray thruster, a MEMS-based electric propulsion (EP), can provide the necessary thrust requirements. After calculating the Δv budget and attitude control needs, it was determined that 0.474 mN of thrust was needed for this momentum-biased vehicle. The lifetime includes maneuvering the satellite to the correct orbital position and in-orbit operation. An array of 69x69 needles on the emitter die will provide this thrust plus a small margin. A constellation of 180 satellites will give complete global coverage. The cost to fabricate and launch the system will be approximately $100 million.

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Introduction

Satellites are an integral part of our lives. We use them for a variety of needs including navigation, observation, and communications. Many of these satellites operate in GEO. Although operating in GEO means fewer satellites are needed for total global coverage, GEO does have its disadvantages. One of these is that it is costly – over $100 million – to place a satellite into GEO. Moreover, the vehicle is costly to produce itself, from $100s of millions to over a billion dollars. Failure of such a spacecraft is catastrophic.

Advances in MEMS and nanotechnology could change how satellites are built and where they are operated. All satellites require a basic set of subsystems to do useful work in orbit.[1] Common satellite subsystems are communications, guidance/navigation, power, thermal, attitude determination and control, the bus, payload, tracking, telemetry and command, and propulsion. If MEMS technology can provide smaller components that yield the same performance, then satellites and their subsystems could eventually be mass-produced, greatly reducing costs.

For a spacecraft that uses chemical propulsion (CP), a large part of its overall mass is due to the amount of propellant it must carry to perform various in-orbit functions. In fact, the ratio of propellant to payload mass rises exponentially as the ratio of Δv to exhaust velocity increases, and so orbiting payloads are typically 1% of total lift-off mass.[2] Here is where electric propulsion (EP) can be most beneficial. EP systems are typically more efficient than CP and have significantly higher specific impulse (3X higher or more than CP). Electrospray propulsion, a MEMS-type thruster technology, has a specific impulse of 3000 seconds.

Electrospray propulsion, however, like other forms of EP has the disadvantage of only being able to provide small amounts of thrust. A small satellite, like the one used

here, only needs a small amount of thrust. The goal of this thesis is to design a thruster that will satisfy the satellite's in-orbit propulsion needs.

Thruster design is contingent upon many things. The amount of thrust needed over the lifetime of the satellite, the Δv budget, and attitude control. All of these entities are ultimately influenced by the mission of the satellite, its subsystems, and its overall mass. Even the constellation design affects thruster design because altitude and position of satellites in the constellation influences the Δv budget.

This thesis will discuss the various satellite subsystems, their function, the design of the constellation, and most importantly, how this all influences the design of the thruster.

Sizing and Power

Electrospray propulsion, like most other forms of electric propulsion, is power limited. It was therefore necessary to keep the size and power requirements on the low side. I looked at other nanosatellites, like Surrey's SNAP-1, to get a realistic idea for power needs and size of components.[3] I also kept in mind that as the satellite's size decreases, its power requirement decreases faster than its surface area. Looking at other nanosatellites, I choose a starting mass of 6 kg and a power consumption of 6 W. As I choose actual components, the size and power requirement numbers were revised.

To get the volume, dimension, and area values I use the following equations where *M* is the dry mass of the satellite:

$$\text{Volume: } V = 0.01M \text{ [m}^3\text{]} \quad (1)$$

$$\text{Linear Dimension: } s = 0.25M^{1/3} \text{ [m]} \quad (2)$$

$$\text{Body area: } A_b = s^2 \text{ [m}^2\text{]} \quad (3)$$

$$\text{Moment of Inertia: } I = 0.01M^{5/3} \text{ [kg*m}^2\text{]} \quad (4)$$

Satellite Subsystems

The typical satellite includes the following subsystems: mechanical structure, thermal control, tracking, telemetry, and command, attitude and orbit control, guidance, communications, payload, power supply, and propulsion. Below I will discuss the functions of each, choice of components and sizing. The goals were to keep the satellite below 10kg, keep the dimensions relatively small, and choose as many commercial-off-the-shelf (COTS) parts as possible.

Mechanical structure:

The structure of the satellite is important for several reasons. The structure holds everything together and provides a place to attach components – both internally and externally. The material used needs to be strong enough to survive launch and durable enough to survive the harsh environment of space. Because this nanosatellite has a short lifespan – approximately one year to maneuver it to its correct orbital position plus the amount of time it is able to stay in space and maintain its station – and is low weight, the goal is to use a material that won't significantly to the mass of the satellite. Aluminum-6061(Al 6061-T6) is a commonly used material for the structure. The Al 6061-T6 has high strength vs. weight, it's easy to machine, and ductile.

To size the amount of Al 6061-T6 needed, we first need to size the satellite. Using

$$\text{Linear Dimension: } s = 0.25M^{1/3} \text{ [m]} \quad (5)$$

with M equal to 8.2 kg, V is 0.082 m^3 and s equal to 504 mm. Due to the solar panel, the height is 300 mm and the width is 285 mm.

Thermal control:

For small satellites, there is a significant amount of integration of electronics on circuit boards. Thus these circuit boards have to endure high heat loads. Since nanosatellites are used almost exclusively in LEO, the components are subjected to a

large amount of thermal cycles. We need a way to provide efficient and effective thermal control for the satellite. The thermal subsystem needs to meet several requirements. One is to provide passive control so as to not use the satellite's resources. Another is to be lightweight to minimize the amount of added weight. Most importantly, we need a material that that is capable of absorbing all radiation and reflecting none of it. It is also desirable for the material to be able to withstand the harsh environment of the atmosphere, have good electrical isolation, be abrasion tolerant for manual assembly by the astronauts, and not contaminate the spacecraft.[4] A group out of the Materials Science and Engineering Department and Rensselaer Polytechnic Institute has shown that low-density, vertically aligned carbon nanotube (VA-CNT) trays are the blackest material ever made.

The VA-CNTs have an emissivity, ε, of approximately one, and an index of refraction, n, somewhere between 1.01 and 1.10. They are flexible, strong (Young's modulus > 1 TPa), and high thermal conductivity, $k \approx 6000$ Wm^{-1}K^{-1}.
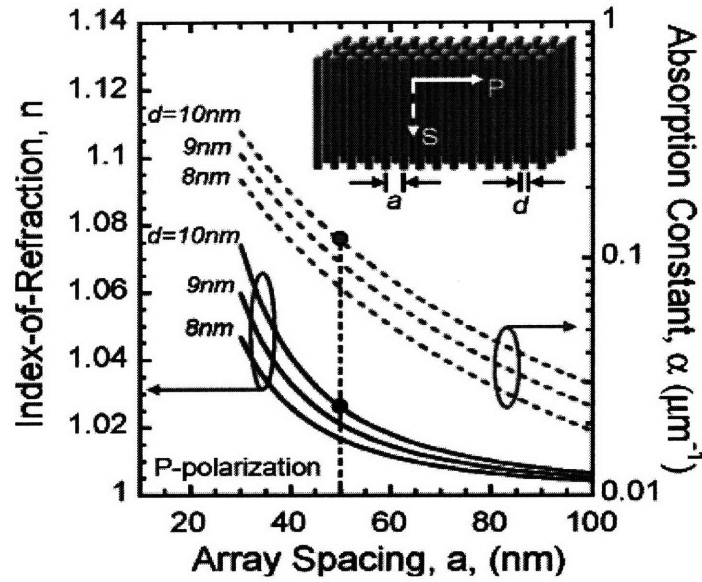
Figure 1: Effective index of refraction and the absorption constant plotted as a function of nanotube spacing. A schematic of the nanotube array and the s and p light polarizations is shown in the inset. For this example as p polarization, a = 50 nm, and d = 10 nm, we obtain $n^{p}_{effective}$ = 1.026 and $\alpha^{p}_{effective}$ = 0.12 μm^{-1}. This index value is very close to unity and could lead to an extremely low optical reflection of R^{p} = 0.02%.[5]

The question is whether a forest of CNTs would work. To show that it would, we need to prove that a significant heat gradient doesn't occur within the satellite within the orbital period. In other words, the characteristic time, τ, is much less than the period, P. Recall the heat equation is given below:

$$\rho c_p \frac{\partial T}{\partial t} - k \frac{\partial^2 T}{\partial x^2} = \dot{q} \qquad (6)$$

Where ρ: density [kg*m^{-3}]
c_p: specific heat (constant pressure) [J*kg^{-1}*K^{-1}]
k: thermal conductivity [Wm^{-1}K^{-1}]
T: temperature [K]

t: time [sec]
x: heat transfer path [m]
$\dot{q}$: rate of heat flow [W]

The thermal diffusivity, α, is defined as:

$$\alpha = \frac{k}{\rho c}\qquad(7)$$

The characteristic time is defined as:

$$\tau = \frac{l^2}{\alpha}\qquad(8)$$

In Equation 8, l is the characteristic length. Measured in terms of length, it is the ratio of an object's volume to its surface area. With a volume in this case of .04309 m^3 and a surface area of .7604 m^2, l = .057 m.

If the structure of the satellite is aluminum 6061, then ρ is 2702 kg*m^{-3}, c_p is 1090 J*kg^{-1}*K^{-1}, and k is 225 W*m^{-1}*K^{-1}, yielding an α of 76E-6 m^{2}*s^{-1}. Plug these values into Equation 8 to get τ equal to 42.75 sec. The period at 400 km is 5553.58 sec. Clearly 42.75 sec « 5553.58 sec. This shows we can assume the heat gradients within the satellite are negligible. At a height of 40 µm, and a density of 0.02 kg*m^{-3} for the CNTs, the volume of the forest is about 6 cm^3 and the mass it adds to the satellite is about 0.12 grams.

For the other surfaces, we'll use several layers of radiation shielding.

Tracking, telemetry, and command:

Tracking, telemetry, and command (TT&C) are essential for monitoring and controlling the satellite. The tracking part determines the position and of the spacecraft. The telemetry part gathers information and on how well the various subsystems of the satellite are functioning. Lastly, the command component receives, verifies, and executes remote control commands from Earth ground stations.[6] As in SNAP-1, each

subsystem will have an 8-bit CAN micro-controller, manufactured by Siemens, which will provide telemetry and telecommand
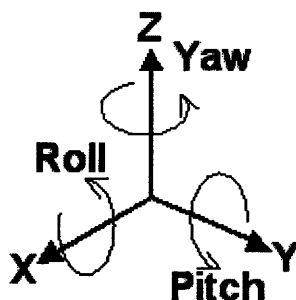
Attitude, orbit control and guidance:

Figure 2: Definition of yaw, roll, and pitch.[7]

Satellites that are 3-axis stabilized can be more stable and accurate, depending on their sensors and actuators. Traditional inertial measurement units, magnetometers, momentum wheels and sensors are too large and consume too much power for a nanosatellite. MEMS technology has allowed the successful miniaturization of these components such that they can be used on a satellite weighing only a few kilograms with dimensions of a few hundred millimeters.

A spacecraft determines its attitude using sensors and controls it using actuators. Gyroscopes and accelerometers are the inertial sensors that most spacecraft use to in their guidance, navigation, and control (GN&C) subsystem. Gyroscopes (gyros) measure the speed or angle of rotation from an inertial reference. Combined with an external sensor such as a sun sensor, gyros can give precise attitude sensing. Accelerometers measure the acceleration of an object. Gyros and accelerometers are typically combined in an inertial measurement unit. MEMS technology has allowed the

miniaturization of the IMU so that it can be integrated into a satellite weighing only a few kilograms.

The MIMU integrates three MEMS gyros and three accelerometers.[14] The gyros and accelerometers are arranged perpendicular to each other to measure the angular velocity and acceleration in three axes. The gyro is a micro-machined, vibrating quartz tuning fork sensing element. Using the Coriolis Effect, a rotational motion about the tuning fork's input axis produces a DC voltage proportional to the rate of rotation. The accelerometer's sensing element is a micro-machined silicon mass suspended by multiple beams from a silicon frame. The motion of the suspended mass measures the acceleration as it affects the resistance of piezoresistors in the beams. The MIMU design has high shock resistance, durability, and built-in damping. The dimensions are approximately 74x50x70mm.
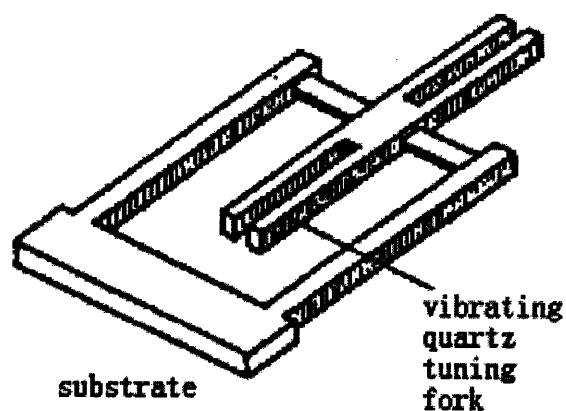
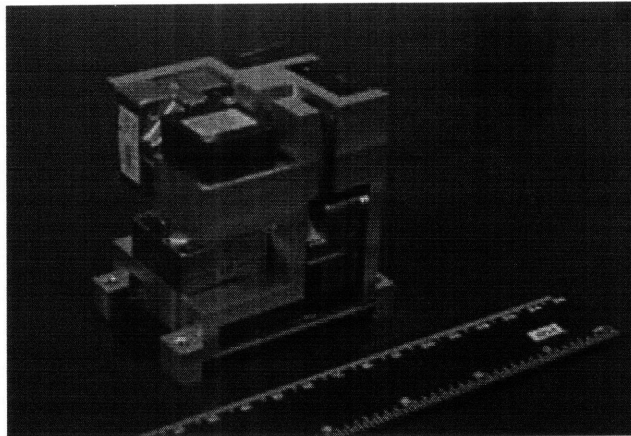Figure 3: Vibrating quartz tuning fork element in gyroscope.[8]

Figure 4: MIMU prototype.[8]

A micro magnetometer will be used to measure magnetic fields. Successfully tested on another nanosatellite, SNAP-1, this micro-magnetometer, manufactured by Billingsley, has a range of ±60 μTesla and a resolution of ± 60 nT. The mass of this device is 117 grams, its dimensions are 32x32x83mm and it only consumes 150 mW.[9]
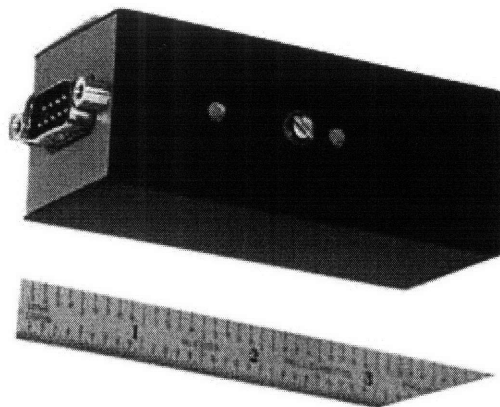

Figure 5: Miniature Magnetometer[9]

The last sensing device this nanosatellite will use is a sun sensor. AeroAstro makes a low-cost, two-axis, medium resolution sun sensor. The device has several years of successful flight history. The sensor has a diameter of about 35mm, weighs only 36 grams and consumes no power. The device provides attitude determination in two axes with an accuracy of ± 1° with a sensor inner field-of view of 60°.
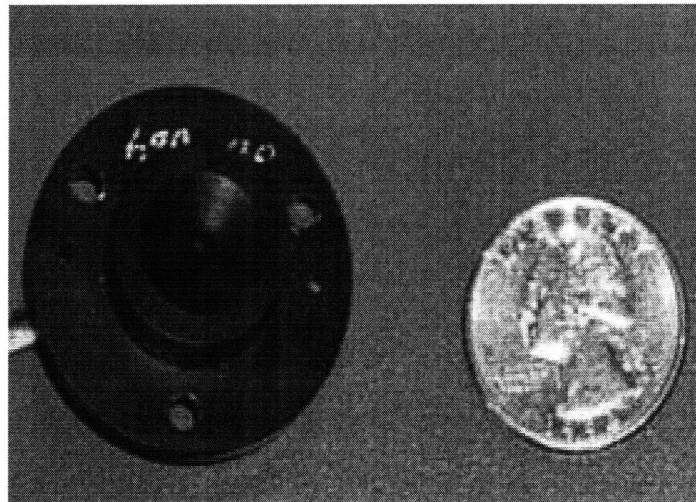

Figure 6: Two-axis Sun Sensor.[10]

The actuators used to control the satellite's attitude will be a magnetic torquer, a momentum wheel, and the thruster. Magnetic torquers, or magnetorquers, use the Earth's magnetic field, B, and electrical current through the torquer to create a magnetic dipole, D, that results in torque on the vehicle:[11]

$$D = \frac{T}{B} \qquad (9)$$

Using the worst-case disturbance torque T_D of 5.11 E -5 Nm, and assuming the worst-case Earth field, B, to be 4.5 E -5 tesla, D is found to equal 1.14 Am². Surrey produces a magnetorquer rod that weighs 510 grams and has a magnetic moment of 5.4 Am² which

has enough capacity to provide a sufficient margin. Power consumption is approximately 200mW. Three rods are needed for 3 axis stabilization.
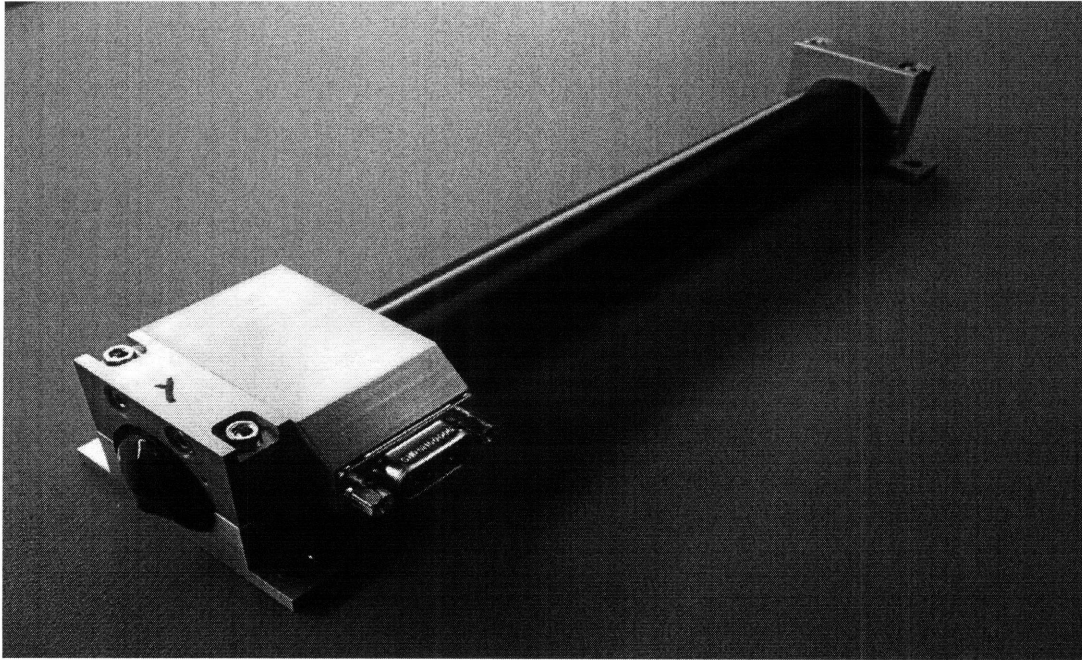
Figure 7: Magnetorquer Rod[12]

Momentum wheels are reaction wheels that have a nominal spin rate above zero to provide nearly constant angular momentum.[11] Both yaw and roll accuracy depend on the wheel's momentum and external torque.[11] For this satellite, I am using momentum bias system. A momentum bias vehicle typically only uses one wheel with its spin axis mounted along the pitch axis.[11] Before sizing the momentum wheel, it was first necessary to calculate how large the momentum storage capacity in the wheel needs to be. To find this, use the following equation:

$$h = \frac{TP}{4\theta_a} \qquad (10)$$

where h is the angular momentum [N*m*sec], T the worst-case disturbance torque [N*m], P the orbital period [sec], and θ_a the allowable motion [degrees]. Plugging in the values for these variables, assuming a 5 degree yaw accuracy, yields an angular momentum value of 0.814 N*m*s. There are small momentum wheels that weigh approximately 80 grams that would satisfy this requirement.

The thruster will be used to dump extra momentum from the momentum wheels and control the attitude. The satellite will only use the thruster momentum dumping. Assume the satellite only uses the thruster for momentum dumping for 1 second each wheel once a day. Then the total pulse is:

$$\text{Total Pulses} = 1 \text{ pulse} * 1 \text{ wheel} * 365 \text{ days/yr} * 1 \text{ yr} \quad (11)$$

$$= 365 \text{ pulses}$$

It was also necessary to size the force level for slewing this momentum-biased vehicle. Using Equation 12

$$F = \frac{h\omega}{Ld} \quad (12)$$

where h is the 0.814 N*m*s value calculated before, ω is the slew rate of 0.005 deg/sec, L is the lever arm, 0.300 m, and d is the thruster duty cycle, here 0.50. Plugging these values into Equation 12 gives a value of 0.474 mN for F.

The next step is to size the force level, F, for momentum dumping.

$$F = \frac{h}{Lt} \quad (13)$$

Using .814 N*m*s for h, 0.300 m for the lever arm L, and 6000 sec for burn time t, we find F is 0.452 mN. The thruster pulse life, or total pulse, assumes the vehicle torques the momentum wheel once a week to either speed it up or slow it down to help with attitude control and dumps momentum once a day:

$$\text{Total Pulses} = 1 \text{ pulse (speed or slow)} * 1 \text{ axis} * 52/\text{yr} * 1\text{yr} \quad (14)$$

$$+ 1 \text{ pulse} * 1 \text{ wheel} * 365 \text{ days/yr} * 1 \text{ yr}$$

$$= 52 + 365 = 417 \text{ pulses}$$

Total impulse, I, is then

$$I = 52 \text{ pulses} * 600 \text{ sec/pulse} * .000474 \text{ N} \quad (15)$$

$$+ 365 \text{ pulses} * 6000 \text{ sec/pulse} * .000452 \text{ N}$$

$$= 1005 \text{ N*s}$$

Now we can calculate the amount of propellant, M_p, needed for attitude control:

$$M_p = \frac{I}{g I_{sp}} \quad (16)$$

Assuming I is 1005 N*s, g is 9.8 m/s², and the specific impulse, I_{sp}, is 3000 s, then M_p is found to equal 34 g. (See Design of Thruster section below for more on attitude control)

An appropriate guidance system would allow satellite to navigate autonomously in orbit. The Global Positioning System (GPS), funded and supported by the U. S. Department of Defense (DOD), allows fairly accurate position determination for small spacecraft in lower orbits. The SGR-05, manufactured by Surrey, is a GPS receiver designed for smaller satellites. It has dimensions of only 70x45x10mm and consumes 0.5-0.8W @ 5V. Its applications include time-stamping, on-board clock synchronization, position and orbit determination. The receiver has an accuracy of <15m. The SGR-05 has already been successfully flight tested in the SNAP-1 nanosatellite.
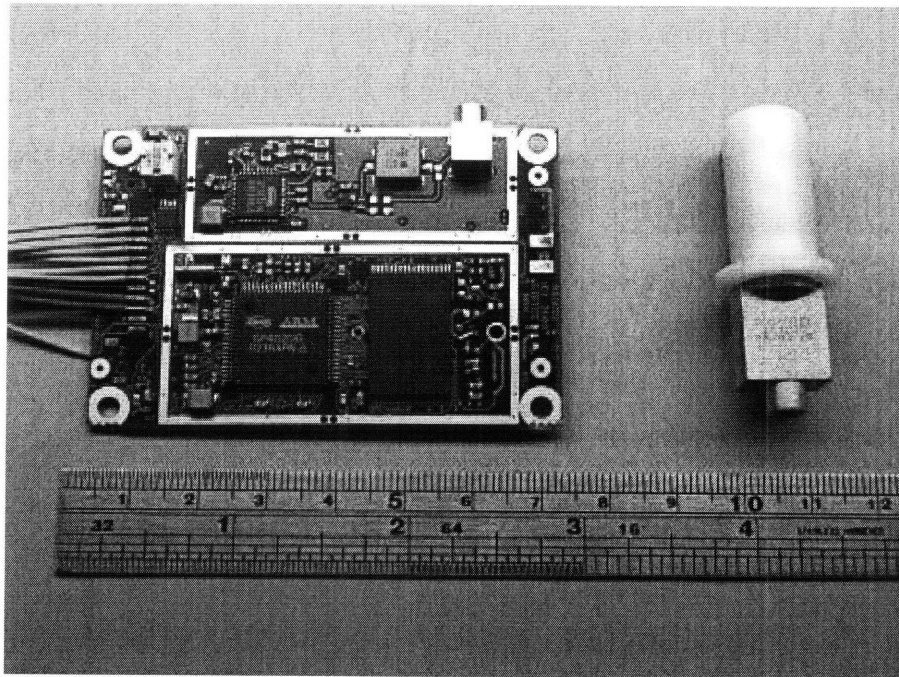
Figure 8: SGR-05 GPS receiver and antenna[13]

Lastly, there needs to be an on-board computer (OBC) to control and monitor the subsystems. A model like the one used on SNAP-1 is most desirable. This OBC has a 220 MHz processor, weighs approximately 300 grams and consumes about 1.5W of power.

Communications:

The communications subsystem enables the satellite to communicate with the ground and other spacecraft. This nanosatellite will have a VHF receiver for uplinks, a S-band transmitter for downlinks, UHF intersatellite links, and both a S-band and VHF antenna. The VHF receiver consumes about 750 mW and the S-band transmitter has a transmit power between 150 and 400 mW.

Payload:

The payload on a communications satellite can be almost anything. Here, we'll use an Earth-imaging camera. There are several small, low power, CMOS-fabricated options. Kodak makes a 1280 by 1024 array that consumes about 190 mW. At the small size and low-power consumption, it is possible to include more than one.

Propulsion:

The propulsion subsystem on this satellite will need to provide orbit maintenance and attitude control. The design of the propulsion subsystem will be discussed in more detail later.

Power supply:

Before sizing the solar array and battery, I needed to decide on the various power needs. This means I needed to know the power requirements for the various components/subsystems.

The electric power supply has two components, the primary and secondary source. The primary source, or solar panel, converts solar energy into electric energy (through a converter). The secondary source, batteries, provides energy when the primary source is unavailable, such as during an eclipse or when the solar panels are in shadow. Because this satellite is so small, it was decided to use a body-mounted solar panel. There were several steps involved in sizing the solar panel.

The first step in sizing the solar array was to determine the requirements and constraints such as: average power required during daylight and eclipse, orbit altitude and eclipse duration, and design lifetime. The average power required is 8W, the orbit altitude is 400 km, the eclipse duration was estimated to be about 40 minutes, and the desirable lifetime is 1 year.

The next step was to calculate the amount of power that must be produced by

the solar array, P_{sa}. Using the following variables:

P_e: eclipse power, 8W

P_d: daylight power, 8W

T: total period, 92.4 min

T_e: time of eclipse, 36 min

T_d: amount of daylight, 56.4 min

X_e, X_d: the efficiency of the paths from the solar
arrays through the batteries to
individual loads

To find P_{sa}, I used Equation (17):

$$P_{sa} = \frac{\frac{P_e T_e}{X_e}}{T_d} + \frac{\frac{P_d T_d}{X_d}}{T_d} \qquad (17)$$

Assuming direct energy transfer, then X_e is 0.65 and X_d is 0.85. Plugging in the

values for X_e, X_d, T_d, T_e, P_e, and P_d into Equation (17), it was found P_{sa} equals 17.3 W. I

choose gallium arsenide (GaAs) for the type of solar panel since it's more efficient than

silicon. GaAs has a efficiency of 26 %. Using the 0.26 value, I was able to calculate P_o, the

estimated power output with the Sun normal to the surface of the cells. Assuming solar

illumination intensity is 1367 W/m², then P_o can be found using the following:

$$P_o = 0.26 * 1367 \text{ W/m}^2 \qquad (18)$$
$$= 355.42 \text{ W/m}^2$$

Next was to determine the beginning-of-life (BOL) power production capability,

P_{BOL}. Assuming an inherent degradation, I_d, of 0.77, and a worst case sun incidence

angle of 23.5 degrees,

$$P_{BOL} = P_o I_d \cos \theta \qquad (19)$$

~ 18 ~

$$= 250.98 \text{ W/m}^2$$

I also had to determine the end-of-life (EOL) power production capability of the solar array, P_{EOL}. Let L_d be the lifetime degradation of the solar panel material. For GaAs solar panel with a life expectancy of 1 year, L_d is .9725. P_{EOL} can be found using the following equation:

$$P_{EOL} = P_{BOL} * L_d \qquad (20)$$

$$= 244.07 \text{ W/m}^2$$

Finally, we're ready to estimate the area of the solar array, A_{sa}, and the mass of the solar array, m_{sa}. Using the following equations:

$$A_{sa} = \frac{P_{sa}}{P_{EOL}} \qquad (21)$$

$$m_{sa} = 0.04 * P_{sa} \qquad (22)$$

it was found A_{sa} is .071m², or each side of the solar panel is about 266 mm (I used 285 mm for a little extra surface area) and m_{sa} is 692 g.

The batteries chosen are nickel-cadmium (Ni-Cd). They are commonly used in LEO satellites due to their robustness versus cycle numbers.[6] Ni-Cd batteries can provide high currents at a constant voltage of 1.2 V. The batteries have an average depth-of-discharge (DOD) of 20%. Let P_e and T_e have the same values as above. Let N, the number of batteries, equal 6 (redundant), and n, the transmission efficiency between the battery and the load be 0.90. Then the battery capacity, C_r, is found using the following:

$$C_r = \frac{P_e T_e}{(DOD) N n} \qquad (23)$$

Once the value for C_r is determined, then the mass of the battery supply is found by dividing C_r by the specific energy density of NiCd. Letting the specific energy density of NiCd be 35 W*hr/kg, then C_r is 4.4 W*hr and the mass of the NiCd batteries is 126 g.

Design of a Satellite Constellation

The closer a satellite is to the Earth (lower altitude), the smaller its coverage area. It is therefore necessary to use multiple satellites to achieve global coverage. The composite of all satellites in the system is called a constellation.

There are several factors to consider when designing a constellation of satellites. These factors are: altitude, minimum elevation angle, total number of satellites, number of orbital planes, between plane phasing, constellation pattern, eccentricity, collision avoidance, and end-of-life (EOL) strategy.

The choice of altitude is influenced by several factors such as satellite lifetime, atmospheric drag, orbital decay rate, signal delay time, and launch capabilities. The range of altitudes I looked at were 300 km to 400 km. For circular orbits the change in the semi-major axis (radius) per revolution is found using the following equation:

$$\Delta a_{rev} = -\frac{2\pi C_D A a^2 \rho C_D}{m} \qquad (24)$$

where C_D is the coefficient of drag, A is the satellite's area that is normal to the flow, m is the satellite's mass, a is the semi-major axis, and ρ the atmospheric density.

An estimate of the satellite's lifetime, L, can be computed from

$$L \approx -\frac{H}{\Delta a_{rev}} \qquad (25)$$

where H is the atmospheric density scale height. The values A of 0.0812 m², C_D of 2, and m of 8.2 kg are the same for all altitudes. The table below gives the values for altitude, atmospheric density, and scale height that I used.

Table 1: Atmospheric Data

Altitude (km)	Atmospheric Density Mean (kg/m³)	Scale Height (km)
300	1.87E-11	50.3
350	6.66E-12	54.8
400	2.62E-12	58.2

Plugging the values from Table 1 into Equations (24) and (25), I found the values listed in the Table 2 below:

Table 2: Comparison of Lifetime vs. Altitude

Altitude (km)	Lifetime, L (rev)
300	442.4
350	1333.3
400	3546.6

To put the L values in terms of days, I divided the number of hours in a day by the satellite's period at that altitude to get the number of revolutions per day, and then divided the L value by the number of revolutions per day, to get the lifetime of the satellite in days. In other words, for an altitude of 300 km and an orbital period, T, of 1.51 hours:

$$\frac{24\ hours}{1.51\ hours} = 15.9\ revolutions/day \qquad (26)$$

$$\frac{442.4\ revolutions}{15.9\ revolutions.day} = 27.8\ days \qquad (27)$$

Doing similar iterations for 350 km with T of 1.53 hours and 400 km with T of 1.54 hours, I found lifetimes of 84.9 and 227.4 days, respectively. Clearly, there is a significant increase in satellite lifetime at 400 km, so that is the altitude I chose.

Another factor influencing altitude was drag. Drag is the resistance a body feels as it moves through a fluid. A vehicle in LEO, like this nanosatellite, will experience drag as moves through Earth's thin upper atmosphere.[14] In time the action of drag on a satellite will cause it to spiral back into Earth's atmosphere. The deterioration of a satellite's orbit due to drag is called decay. Drag is calculated using Equation (28):

$$F_D = .5 \rho A v^2 C_D \qquad (28)$$

where ρ is atmospheric density, A is the area of the body normal to the flow, v is the velocity, and C_D is the coefficient due to drag. The area and drag coefficient are the same for all altitudes, $0.0812m^2$ and 2.0, respectively. Table 1 lists the density data for the various altitudes. The velocity values are 7726 m/s (at 300 km), 7697 m/s (at 350 km), and 7669 m/s (400 km). Plug these values into Equation (28) to find that the F_D is 9.06E-5 N (at 300 km), 3.20E-5 N (at 350 km), and 1.25E-5 N (at 400 km). Drag is less at 400 km; this will also let the satellite have a longer in-orbit functional lifetime.

The orbit decay rate (average) in years ranges from 0.15 years for 300 km to 0.6 years for 400 km. Since the orbit will decay four times slower at 400 km, the constellation will last longer. This was another deciding factor for choosing 400 km.

The differences in delay time for the altitudes were negligible. To calculate the transmission delay, t_d, use Equation (29):

$$t_d = \frac{h}{c} \qquad (29)$$

where h is the altitude [m] an d c is the speed of light, 3.0 m/s. At 300 km, t_d is 1 msec, or 2 msec roundtrip; at 350 km roundtrip delay time is 2.4 msec; and at 400 km it's 2.6

msec. For LEO, the satellite can be delivered to any of the three altitudes, so that did not factor in either. I chose 400 km because the satellite would have a longer lifetime.

The next steps in a satellite constellation are to calculate how many satellites are needed for complete global coverage, the number of orbital planes, and the number of satellites per plane. Some basic parameters needed to determine are shown in Figure 9 below.
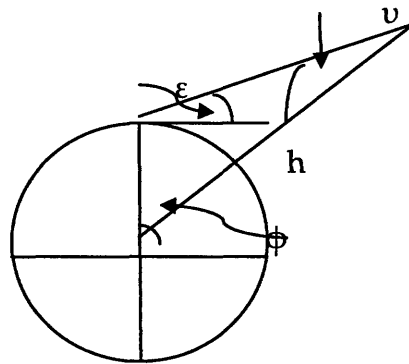

Figure 9: Geometric Relations for a Satellite System

In Figure 9, ε is the elevation angle, υ the nadir angle, and ϕ is the Earth central angle. A minimum value for ε, ε_{min}, determines the limits of the service area of the satellite. A user or Earth station located below ε_{min} will not easily be seen by the satellite. I chose a value of $10°$ for ε_{min} because it both increases the number of places on Earth that can be seen by the satellite and decreases the number of satellites needed for complete coverage. Using Equation (30)

$$\alpha = \cos^{-1}\left(\frac{R_e}{r}\cos\varepsilon_{min}\right) - \varepsilon_{min} \quad (30)$$

where R_e is the radius of the Earth, 6378 km, r is R_e plus the altitude of the satellite, 6778 km, and ε_{min} equal to 10°, α is determined to equal 12°. This α value then gets used to find both the minimum number of orbital planes, Ω, and the required number of satellites per orbit, n.

$$\Omega = \left\lceil \frac{2\pi}{3\alpha} \right\rceil \qquad (31)$$

$$n = \left\lceil \frac{2\pi}{\alpha\sqrt{3}} \right\rceil \qquad (32)$$

Plugging in 12° for α gives Ω equal to 10 and n equal to 18. Because the satellite has such a short lifetime, placing extras in-orbit is not necessary. If redundancy is desired, extra satellites can be placed either in the plane or near-by parking orbit. At this altitude, the satellite has a velocity of 7669 m/s and an orbital period 1.54 hr.

The angular difference between satellites in one plane is just $2\pi/n$ or 20°. This constellation is using polar, circular orbits.

There are several launch vehicles designed specifically to place payloads into LEO. I chose Orbital Sciences' Pegasus XL to launch this satellite constellation. The Pegasus XL has a capacity of approximately 340 kg when placing a payload at 400 km. Assuming some space between the satellites, we can probably fit at least 20 satellites per vehicle. At that rate it would take about 9 vehicles to place all 180 satellites in orbit.

Design of Thruster

The primary challenge of this thesis was to design the thruster. An electrospray thruster is desirable because it has the potential to provide precise thrust impulses over a wide range of Isps. It is also very efficient, with $\eta \geq 0.85$. Each needle can produce on the order of $0.1\mu N$ of thrust. Therefore it is necessary to use an array of needles to

achieve a useful amount of thrust, in the milli-Newton range for a nanosatellite. Before sizing the thruster, it was necessary to determine the thruster needs, calculate the amount of propellant needed, calculate the amount of thrust needed for the life of the satellite, and determine an arrangement of the needles on the thruster to maximize the thrust per area.

The first question to answer was what functions will the thruster need to perform. As mentioned in the propulsion subsystem section, this thruster will need to maneuver the satellite into the correct position in orbit, overcome drag, and help provide attitude control.

The second step was to do a Δv budget. Each time a satellite changes orbit or undergoes re-phasing, it uses energy. The Δv budget accounts for the energy required to perform these various maneuvers. To calculate the Δv budget, I first had to determine some basic data. There are launch vehicles specifically designed to deliver payloads to LEO, so initial conditions and the mission orbit will be the same, an altitude of 400 km and a 90° inclination. The aim is for the satellite to last 1 year. The 1 year would include the time to maneuver the satellite into correct position and actual in-orbit functional life. Orbit maintenance requirements will include altitude maintenance and overcoming drag and other external perturbations. Below 800 km, solar radiation is negligible. The orbit maneuver will be re-phasing.

To calculate altitude maintenance, I used the following equation. Equation 33 gives the change in velocity per revolution

$$\Delta v_{rev} = \frac{\pi C_d A \rho a v}{m} \quad (33)$$

where C_D the coefficient of drag, A the satellite's cross-sectional area, ρ the density, a the semi-major axis (or radius here), v the velocity, and m the satellite's mass. Assume C_d is 2, A is 0.0812 m², ρ is 2.62E-12 kg/m³, a is 6778100 m, v is 7669 m/s, and m is 8.2 kg.

Using these values, Δv is found to equal 0.00847 m/s per revolution. As calculated earlier, L~3546.6 rev. For the lifetime of the satellite $L * \Delta v_{rev}$ gives a Δv of 30.1 m/s.

The next step was to calculate the Δv required to re-phase the satellite. The satellites in the same orbital plane will be separated from one another by 20°. For a positive phase change, transfer the satellite to a lower (faster) orbit, let the satellite drift in this orbit for some time, and then return it to the original orbit so as to insert it at the new phase.[15] Using the following equations Δv can be found:

$$\frac{\Delta V}{v_1} = \frac{\Delta R}{R_1} \quad (34)$$

Here ΔV is the velocity change required to re-phase the satellite, v_1 is the velocity the satellite had in its original orbit, R_1 is the original altitude, and ΔR is the magnitude of the altitude change to the drift orbit. Assuming the drift orbit is at an altitude of 375 km, v_1 is 7669 m/s, and ΔR is 25 km, then ΔV is 28.3 m/s.

The total Δv budget is 30.1 + 28.3 = 58.4 m/s.

There is an amount of propellant needed to satisfy the Δv requirements. Recall the rocket equation

$$\Delta v = c \ln\left(\frac{m_o}{m_f}\right) \quad (35)$$

where m_o is the spacecraft dry mass and m_f is the sum of the spacecraft dry mass and propellant. Re-write Equation 35 to get

$$m_f = m_o e^{(\Delta v/c)} \quad (36)$$

where m_o is 8.2 kg, Δv is 58.4 m/s, and c is 30000 m/s. Plugging these numbers into Equation 36 gives a value of 16 g for the propellant.

For EOL de-orbit, the thruster will be turned off and the Earth's gravity allowed to pull the satellite back into its atmosphere. This satellite is small enough that it should burn up in the Earth's atmosphere.

The next step was to look at attitude control. The first step is to calculate the various disturbance torques. The disturbance torques affect actuator size and momentum storage requirements.[11]

One type of disturbance torque comes from the gravity-gradient. The formula to calculate it is

$$T_g = \frac{3\mu}{2R^3} abs(I_z - I_y)\sin(2\theta) \qquad (37)$$

with R = 6778100 m, I_z = 0.215 kg*m^2, I_y =0.210 kg*m^2, and θ = 1° (normal mode). Plug in these values to find T_g is 3.35E-10 Nm (normal mode).

The next type of disturbance torque, T_{sp}, is from solar radiation.

$$F = \frac{F_s}{c}A_s(1 + q)\cos i \qquad (38)$$

$$T_{sp} = F(c_{ps} - cg) \qquad (39)$$

where F_s is the solar constant, 1367 W/m^2; c the speed of light, 3.0E8 m/s; A_s the surface area of the satellite, 0.7758 m^2, cg the center of gravity, (for a rectangle of length 504 mm, and 300 mm tall, cg is located inside at the center; c_{ps} is the center of solar pressure, located at the center of the solar panel; q the reflectance factor, estimate 0.6; (c_{ps} – cg) is 0.252 m, and i the angle of incidence of the sun, 0. Substituting in these values it is found that F is 5.66E-6 N, and T_{sp} is 1.43E-6 Nm.

The Earth's magnetic field also produces a disturbance torque on the satellite. This magnetic torque, T_m, is determined by

$$T_m = DB \qquad (40)$$

$$\text{where } B \sim \frac{2M}{R^3} \quad (41)$$

with D is the residual dipole of the vehicle, assume small, uncompensated vehicle, 1 A^*m^2; M is the magnetic moment of the Earth, 7.96E15 tesla$^*m^3$; R the radius from Earth to the satellite, 6778100 m; B the Earth's magnetic field. It was determined that B is 5.11E-5 tesla and T_m is 5.11e-5 Nm.

Aerodynamic effects produce a disturbance torque, T_a. The aerodynamic torque is influenced by orbit altitude and the spacecraft's geometry.

$$T_a = F(c_{pa} - cg) = FL \quad (42)$$

where $F = 0.5[\rho C_d A v^2]$; c_{pa} the center of aerodynamic pressure; L offset, 142.5 mm. F is 6.71E-5N and T_a is 9.57E-6 Nm. The worst-case disturbance torque is due to the magnetic field.

The next step is to use the disturbance torque values to find the thruster requirements and propellant needs for attitude control. First I looked at thruster force level sizing for external disturbances.

$$F = \frac{T_D}{L} \quad (43)$$

here F is the thruster force, T_D is the worst-case disturbance torque, and L the thruster's moment arm. Let T_D equal T_m, and L be 0.300 m. Plugging these values into Equation (43) gives an F value of 0.179 mN, which is rather small. This means that slewing rates, not disturbances, will more likely determine thruster size.

Under the attitude control section, it was determined that the mass of propellant needed for attitude control is 34 g. Add this to the value from Δv and we get a nominal

propellant need of 50 g. If we add in a margin of 15%, then the total propellant needed during the satellite's lifetime is about 58 g.

To size the thruster, I used the worst-case or largest force level found in the attitude control section. The thruster force needed for this momentum-biased vehicle is 0.474 mN or 474 μN. Each emitter needle yields 0.1 μN/needle so it is necessary to have 4740 needles to provide nominal thrust. If we use a square array of 69 x 69 needles, then there will be 4761 μN of thrust provided, which provides some margin. To maximize thrust per area, we'll place the needles 100 μm apart. This leads to a length of 6.8 mm across. If we have 3 mm of space between the edge of the emitter and the closest row of needles makes the whole emitter die 12.8 mm by 12.8 mm for a total area of 163.84 mm^2.

The force per area is given by the following equation

$$\frac{P}{A} = \frac{I_e \sqrt{(2V/(q/m))}}{d^2} \qquad (44)$$

where I_e is the current from the individual needle, 1 μA/tip, V is 1000V, (q/m) is 5E5 C/kg, and d is the distance between the individual emiters, 100 μm. Plugging in the previous values gives a thrust per area of about 19 N/m^2. Input power is then found using

$$\frac{1 \, \mu A}{needle} * 4761 \, needles * 1000V = 4.7 \, W \qquad (45)$$

The last step is to size the power processing unit (PPU). The PPU converts the supplied electrical power into the form required by the thruster and its various components. The mass of the PPU is a function of the average power they can handle. TTP then is just 476.1 μN divided by 4.7 W, or 101.3 μN/W. Now the mass of the PPU, M_{PPU}, is found using the following equation

$$M_{PPU} = \frac{T\alpha}{TTP} \qquad (46)$$

In Equation (46), α is the specific mass, 30 g/W, and T and TTP are values already given above. Plugging the above numbers into Equation 46 gives a value of 141 g for the mass of the PPU.

Cost Estimate

There are several factors affecting cost, such as the cost of individual components, the ground support segment, integration, assembly and testing, and launch and orbital support. Here I'll focus on the cost to build the satellite and the cost of placing it in orbit. There are three basic methods for estimating cost; I'll use analogy-based estimating. In this method, the cost of a similar item is used, with adjustments made for size and complexity. This is the most efficient method for this satellite because it is based on COTS and other nanosatellites. Other nanosatellites, like Surrey's SNAP-1 that used many of the same components, cost approximately $1 million dollars. Assuming advances in technology since SNAP-1's launch, we'll estimate the cost per satellite to be approximately $500,000. With 180 satellites needed for the constellation, it would cost approximately $90 million to build the whole system. Economies of scale and use of MEMS technology would probably lower the cost even more.

Actually launching the satellite and placing it in its final orbit is a major factor in the cost. Using a launch vehicle like the Delta II, it would cost approximately $10000/kg to place a spacecraft into LEO. With each nanosatellite weighing 8.2 kg, and a total of 180 satellites, that is a total mass of 1476 kg. It would cost approximately $15 million dollars to place all of the satellites into orbit.

The total cost to build all 180 satellites and placing them into orbit is about $105 million.

Conclusion

To use of smaller satellites (mass <10kg) to satisfy remote sensing needs will require a small, efficient propulsion system that will satisfy the various propulsive requirements. An electrospray thruster with an emitter array of 69 x 69 needles will provide more than the necessary 0.474 mN thrust. The use of EP will help keep the mass of the satellite down, allowing more satellites per launch vehicle. Because the satellites are so small, we can fit multiple satellites per launch vehicle. This would enable us to build up the constellation more quickly. MEMS technology will enable satellites to eventually be mass produced like other electronic devices. A constellation of 180 satellites is necessary for complete global coverage at an altitude of 400 km. The use of COTS will help keep the cost of an individual satellite less than million dollars.

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