The Florida Space Institute's Photon Satellite Bus

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Abstract: Photon is a small satellite for researching atmospheric effects on Earth-to-space laser propagation and the mitigation of those effects. Florida Space Institute engineers and scientists are designing the science mission and payload, while FSI students are responsible for the satellite bus. An agreement with NASA for a free launch aboard the Space Shuttle's Hitchhiker Ejection System has led to a unique, bottom-up design approach. Science goals and stringent safety requirements have created interesting challenges for the structure, thermal, attitude control, and power subsystems. This paper emphasizes these challenges in presenting the design of the Photon satellite bus.

1. Introduction

The Florida Space Institute (FSI), headquartered at the University of Central Florida, is currently designing a small scientific satellite called Photon. Its primary objectives are to validate FSI scientists' math models of the atmospheric propagation of lasers and to provide a proof-of-concept for techniques to mitigate the effects of atmospheric turbulence on laser transmissions [1]. The secondary objective is to provide a target for other experiments in optical communications, ladar, DIAL, geodesy, etc.

Photon's primary payload is a trihedral retroreflector (retro) which will reflect a laser beam transmitted from the Innovative Science and Technology Experimental Facility (ISTEF) back to FSI's coherent array receivers. Secondary payloads include a gimbaled laser beacon, a wide-angle laser beacon, and a photodetector.

This paper focuses on the Photon satellite bus, which has been designed by student researchers and three senior design teams under the guidance of FSI advisors.

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2. Design Objectives and Approach

Like any other engineering project, the basic design objective is to achieve the mission goals by developing the most efficient engineering design. However, Photon is a unique project due to its unusual design approach. Instead of a typical top-down design where satellite bus requirements are explicitly defined at the outset, Photon has been a bottom-up design. The design began with only a statement of the science objectives and a Space Act agreement with NASA, which will allow Photon to launch aboard the Space Shuttle's Hitchhiker Ejection System (HES) at no cost to FSI. This launch arrangement, however, imposes strict operational, physical, and safety constraints on the satellite design (Table 2.1) [2]. As a result, the science team did not refine and quantify the science objectives. Instead, they selected a retroreflector, a laser diode, and a photodetector as the three basic payloads and asked the bus design team to determine what was feasible within the constraints of the HES. Additionally, FSI's project goals were to minimize complexity and cost while maintaining high reliability. The requirements listed in Table 2.1 have evolved based on the capabilities of the bus as constrained by the HES.

Table 2.1 – Requirements and Constraints

Payload	Maximize retroreflector size Maximize laser beacon brightness
Pointing	±5° nadir pointing Yaw control during active mode
Communication	2 way at elevation >20
Telemetry	Commands - 7K bps Science data - 80K bps active mode (2MB / 5min) Health - 36 hour history, 64 ch, 8 bits, 1 sample/min
Orbit	Depends on shuttle mission – 352 km, 28.5° or 407 km, 51.6°
Lifetime	1 year
HES constraints	Max 150 lbs (68 kg) Max 19" dia, 20.5" height Max c.m. 10.5" axial, 0.5" radial Marmon ring launch adapter NASA safety requirements

The drivers of the Photon satellite bus have been to maximize the science return and minimize complexity. To these ends, the design has undergone several iterations to arrive at the current baseline.

3. Preliminary Design

Because Photon is FSI's first satellite design, considerable background research and many feasibility studies were required to produce a reasonable design point from which to begin. The following assumptions and preliminary design decisions were based on this research.

The retroreflector was assumed to have a triangular cross section with 8 inch sides, giving a cross sectional area of 27 in^2. It was also assumed that the retroreflector mirrors would be composed of fused silica because only one vendor was known. Pointing requirements were estimated to facilitate custom design of the reflection pattern of the retro. The payload was to point toward the ground station with a 1 degree accuracy and align the roll axis with the velocity vector within 1 degree.

To keep the design simple, some preliminary configuration decisions were made. Photon has no propulsion system to maintain its orbit; the Earth's atmosphere will cause the satellite to reenter after from eight months to two years, depending on the starting orbit. Thus, the design lifetime of one year was chosen.

Photon uses body-mounted solar panels rather than deployable arrays to further reduce complexity. An octagonal bus configuration was chosen over circular, rectangular, and hexagonal designs to maximize the surface area available for solar cells while maintaining flat panels for ease of manufacture. This configuration also maximized the internal volume available to the payload and components.

An attitude control design study was performed to determine which control method would best suit this mission. Spin stabilization, 3-axis control, momentum biased, gravity gradient, gravity gradient with gimbaled payload, and combinations of these methods were analyzed before deciding that 3-axis control would provide the best possible science return. However, extensive trade-off studies of the satellite configuration eventually showed that the gravity gradient approach better achieved the mission goals. During a ground station pass, 3-axis control does not offer a significant improvement over nadir-pointing gravity gradient stabilization with a 5° accuracy (Figure 2.1). By accepting this slight reduction in the science return, a much higher reliability and lower cost were achieved.

The following sections discuss the design of the subsystems as they matured from these preliminary configuration decisions.

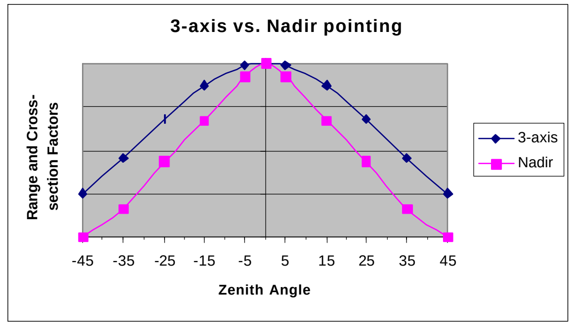


Figure 2.1 – Attitude Control Trade Study

3. Structural Design and Analysis

Support of the satellite's payload, subsystems, and the HES Marmon clamp interface are provided by the Photon bus structure. To survive ground handling, launch, and mission environments, the strength, stiffness, and thermal properties of the structure must be carefully selected. The design approach used for the Photon structure subsystem is based on that described in Spacecraft Structures and Mechanisms [3]. The mission, payload, orbit, and launch vehicle requirements are first defined. Configuration design follows, including an iteration of subsystem concepts, structural architecture, and component layouts. At the end of each iteration, the system is analyzed to verify compliance with the requirements.

3.1 Requirements

Maximizing the retroreflector cross section and withstanding the expected loads are the primary structural requirements for Photon. The launch loading conditions are the most severe and therefore drive the design. Because any payload carried on the Space Shuttle must survive both launch and a possible emergency landing without compromising crew safety, NASA requirements are very challenging. The satellite must demonstrate positive margins of safety for limit loads of 11 g's and 84 rad/s^2 in all directions simultaneously, with a 1.4 factor of safety. The minimum allowable natural frequency is 35 Hz, but a natural frequency above 50 Hz is strongly encouraged by greatly reducing analysis and testing requirements.

Restrictions on envelope, mass, and center of mass limit the configuration of the satellite (see Table 2.1). To facilitate attitude stability, the pitch axis moment of inertia must be greater than the roll axis, which must be greater than the yaw axis moment of inertia. The satellite must also provide the NASA specified Marmon clamp interface ring.

3.2 Retroreflector Optimization

The major science requirement is to maximize the allowable cross section of the retroreflector. Based on the initial assumptions of a triangular cross section and the fused silica construction, the largest possible retro had 12 inch sides and a 62.4 in^2 cross sectional area. Above these dimensions, the mirror plates became too massive to maintain the required center of mass for the satellite.

A breakthrough came from discussions with Composite Optics Inc. at the 1998 AIAA/USU Conference on Small Satellites. By using a composite mirror support structure instead of fused silica mirror plates, a retro of 1/10 the weight was feasible. The composite retro easily solved the center of mass problem, and instead the limiting factor was the HES envelope constraint. By modifying the geometry of the retro so that the cross section was hexagonal instead of triangular, the allowable cross sectional area could be increased to 118.5 in^2, almost double the area of the fused silica retro. The optimized retro occupies most of module 3 with the other payload instruments and communications antennas arrayed at its perimeter (Figure 3.3).

3.3 Configuration Design

The HES-imposed requirements are achieved through an iterative configuration design. After specifying the initial configuration, mass properties analyses and finite element analyses are used to modify the location of components and the dimensions of structural members.

To facilitate assembly and access, Photon uses a modular design with three modules defined by component mounting plates. Attitude determination and control (ADCS) and power system components make up the first module; command and data handling (C&DH) and communications components are in the second; and module three contains the payload (Figures 3.1-3.4).

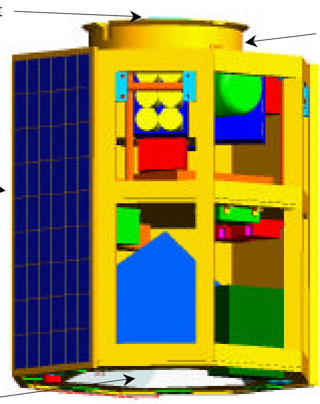


Figure 3.1 – Stowed configuration, 3 panels removed

The primary structure consists of a framework of stringers and cross pieces, the three component plates, the Marmon ring launch adapter, and 8 aluminum honeycomb solar panels. Secondary structures include the gravity gradient boom support, the retroreflector support, and the component mounting brackets. The structural components are bolted together; the specific bolt patterns and bracket designs will be addressed in the detailed design phase.

Originally, composites and several aluminum alloys were considered for the primary structural elements. The material selection was basically a tradeoff between mass and cost. Because the lifetime of the satellite will be primarily limited by aerodynamic drag, a higher mass will keep Photon on orbit longer. Increasing the mass does not increase the HES launch cost, so the 68 kg limit is the target mass of the satellite. After preparing a systems level mass budget (Table 3.1), it was apparent that exceeding the maximum mass requirement would not be a concern. Therefore, a composite primary structure would not be necessary.
Aluminum 6061-T6 is the alloy chosen for the structure. Its properties are sufficient to handle the Photon's loading conditions, and it is readily available.

Table 3.1 - Mass Budget

Photon Mass Budget		
Subsystem	*Allowable Mass (kg)*	*Design Mass (kg)*
Payload	9	7.14
Command and Data Handling	3	2.68
Power	18	17.56
Attitude Determination and Control	5	4.81
Communications	4	3.14
Thermal	1	0.85
Structure and Mechanisms	29	27.29
Satellite Total	68	63.47

3.4 Component Layout

Several criteria were established for designing the layout of components. Fields of view of the retroreflector, sun sensors, and antennas were kept clear. Component placement had to satisfy the HES c.g. requirements. Similar components were located near one another to reduce cabling mass and system noise. The batteries, some of the most massive components were located as close to the launch vehicle interface as possible. Working with the thermal design team, components were placed to maintain both operating and non-operating temperature limits.

Each component is mounted either directly to the frame or to a component plate, which is mounted to the frame. The solar panels are attached to the frame by brackets at the top and bottom to maximize the surface area available for solar cells. Component locations are as shown in Figures 3.1-3.4.

Mass properties of the satellite were calculated using the IDEAS Masterseries solid modeling task. Based on iterative calculations, the component layout was adjusted until the satellite complied with the center of mass and moment of inertia requirements. The final axial center of mass component was 8.60", and the radial component was 0.16", both well within HES requirements.

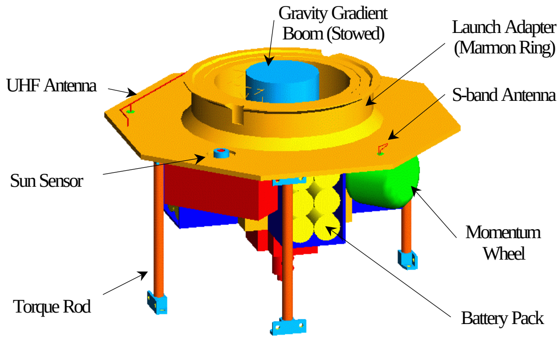


Figure 3.2 – Module 1

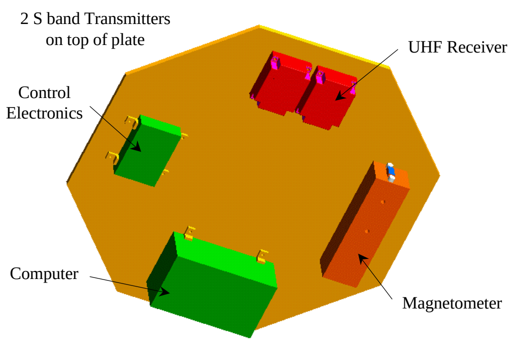


Figure 3.3 – Module 2

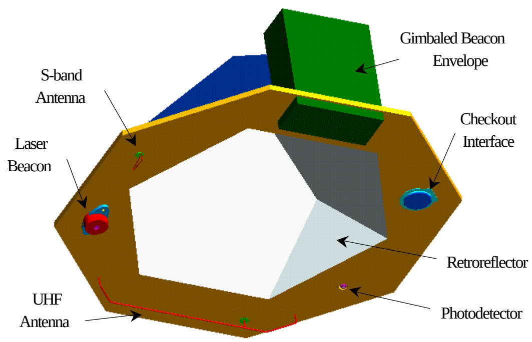


Figure 3.4 – Module 3

3.5 Static and Dynamic Analyses

Stress and vibration analyses were performed using the IDEAS Masterseries finite element analysis task. Originally, a simple model of the primary structure was created which could be improved in detail as the design developed. To give more flexibility in the creation and modification of the finite element model, the FEM was not automatically generated from the solid model. Instead, thin shell elements were used to simulate the component plates while beam elements were used for the structural frame (Figure 3.5). Solar panels were not included at this time. Fasteners holding the structure together were represented by rigid elements. Restraints were initially placed at the interface between the launch adapter and the module 1 plate to avoid modeling the Marmon ring and thus minimize the analysis run time .

Preliminary analyses indicated that the structure had more than sufficient strength, but a low stiffness. Because the satellite is clamped to the launch vehicle at one end, the first mode of the structure was bending as a cantilevered beam. To increase the stiffness, different cross sections and thicknesses for the frame components were systematically tested. Circular tubes, rectangular tubes, and a solid 135° angle section, which fits into the corners of the octagon, were analyzed at thicknesses of 0.25” to 1”. Although the rectangular tube achieved a slightly higher stiffness to mass ratio than the 135° angle section, the angle section was chosen for ease of assembly. A 0.5” thickness was necessary to achieve a natural frequency of close to 80 Hz for the primary structure.

Next, the components and the Marmon ring were included in the model. Point masses located at the component’s c.m. represented the different components. They were connected to the component plates by rigid elements where the component interfaces are located. The Marmon ring was modeled with solid elements because its thickness was greater than could be accurately modeled with thin shells. The solid element mesh was tied directly to the thin shell mesh of component plate 1. New boundary conditions were created at the interface between the Marmon ring and the HES clamp. After these additions, the natural frequency of the satellite decreased to 62.8 Hz, which is above the desired minimum (Figure 3.6).

Because the fasteners are currently simulated by rigid elements, the actual stiffness will decrease when the fasteners are modeled accurately. At the same time, the honeycomb solar panels, which have not been designed in detail yet, will increase the overall stiffness. If additional stiffeners are required after these features are included in the FEM, diagonal supports will be added to the frame and radial ribs will be added to component plate 1.

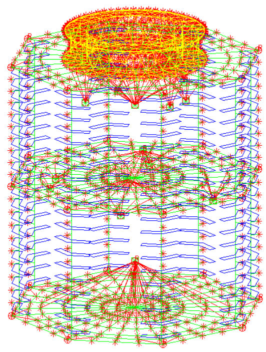


Figure 3.5 – Finite Element Model

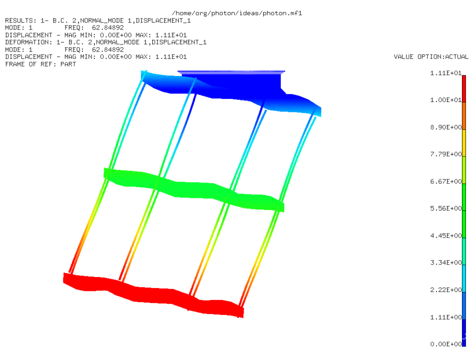


Figure 3.6 – First Mode Shape (exaggerated deflection)

3.6 Structure Summary

The conceptual design of the structure is near completion. Launch loads and vibration analyses verify that the satellite complies with NASA launch requirements, but thermal loads analysis is awaiting completion of the thermal design.

In the next phase, the individual fasteners and brackets will be designed, the finite element model will be expanded to include fastener and component interface details, and specific analyses of critical components will be conducted.

4. Thermal Control

The purpose of the thermal control subsystem is to maintain all components within their temperature limits. To keep the system as simple as possible, the objective of the Photon thermal design is to use only passive thermal control. This can be achieved by designing the layout of components, using thermal coatings, and specifying component interfaces. As an added precaution, the ADCS will be capable of slowly spinning the satellite during cruise mode to improve the temperature distribution.

The first step of the thermal design is to determine both the operating and non-operating temperature ranges of the major components. These are given in Table 4.1. Next, the satellite configuration is designed to locate components based on their thermal requirements. Finally, the satellite is analyzed using SINDA/3D and TRASYS to confirm that the components remain within their temperature limits. If the results do not indicate compliance with the temperature requirements, control techniques are applied as necessary. These steps are then iterated with increasing detail as the design progresses.

Table 4.1 – Component Temperature Ranges

Component	Non-Operating (°C)	Operating (°C)
Retroreflector	-50/+80	-30/+60
Laser Diode	-55/+80	0/+35
Collimator	-20/+70	-20/+70
Antenna	-170/+90	-170/+90
Receiver	-20/+70	0/+35
Transmitter	-40/+80	-30/+60
Battery	-10/+25	0/+25
Shunt assembly	-50/+80	-40/+40
Solar panel	-100/+85	-100/+85
Sun sensor	-30/+55	-30/+50
Magnetometer	-50/+90	-40/+80
Momentum Wheel	-40/+80	-10/+60

4.1 Configuration Design

The thermal team works closely with the structure team to design the satellite's configuration. Because the faces of the octagonal bus are covered by the solar panels, only the Earth and anti-Earth faces of the satellite are available for use as radiators. However, the Earth facing side will constantly receive Earth flux and periodically receive albedo. The top of the satellite will therefore be the best radiator.

Components that produce large amounts of heat are provided good conduction paths to the satellite's radiating surfaces. Those that must be kept warmer are placed next to those that emit more heat. Components that need to be kept cooler are placed close to the radiating surfaces. The available mounting surface and c.g. restrictions necessitate tradeoffs between thermal and structural solutions. Figures 3.1-3.4 illustrate the locations of the various components.

4.2 Analysis

The Photon satellite is modeled in SINDA/3D for thermal analysis. Similar to the finite element structural model, the level of detail in the thermal model has evolved with the design (Figure 4.1). To minimize computer time without adversely affecting the results, the model was simplified from the original 4000 nodes to only 900. Average interface thermal resistance values of 105 W/m^2K are used until the design matures to that level of detail.

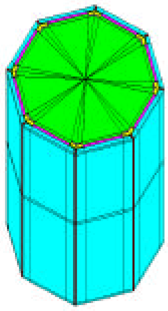
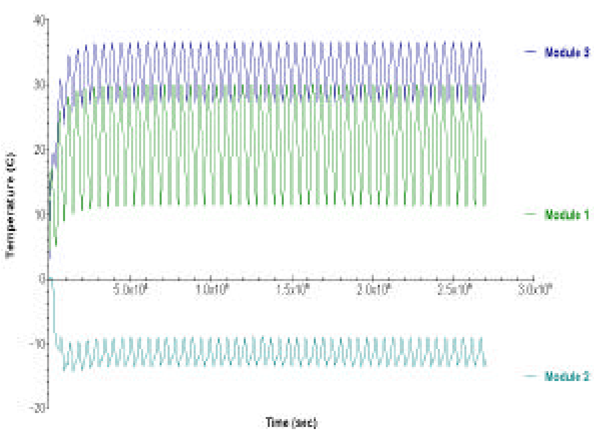


Figure 4.1 – SINDA/3D Thermal Model

SINDA and TRASYS are used to perform the thermal analysis. In the first iteration, the primary structure was analyzed without internal heat dissipation, and the resulting temperature distributions were used to help design the component layout. Then two heat sources were added within modules one and two to represent the average expected heat generation of the internal components. The analysis was run for a duration of 50 orbits with an initial temperature of 0°C. After five orbits, the model exhibited periodic behavior, describing the temperature ranges experienced by the component plates, frame, and the solar panels. Based on these results, temperature ranges were approximated for the three modules (Figure 4.2).



Module 1: +10 to +30°C
Module 2: -15 to -7°C
Module 3: +25 to +40°C

Figure 4.2 – Module Temperature Ranges

4.3 Thermal Control

As expected, results indicated that simply arranging the layout of the components would not meet each temperature requirement. The batteries may become too hot, and the electronics components in module 2 may become too cold. To verify these estimates, the components and their associated heat dissipation must be included in the model. To control the component temperatures, thermal paints and coatings will applied to the structure and to the components. External coatings will also be used to improve the properties of the radiating surfaces.

A significant problem for LEO satellites is oxygen interaction with external thermal paints and coatings. The atomic oxygen can severely degrade many of the typical coatings used, so care must be taken to select resilient coating materials. A few of the materials identified for potential use on Photon are quartz mirrors, white paints Z93 and YB71, and black paint Z306 [4]. Specific locations and coating thicknesses will be defined in the detailed design.

Another concern is the temperature range experienced by the solar panels and the gradient between the hot side and cold side. The panels exhibit a range of -45°C to +70°C for the hottest panel and –38°C to –27°C for the coldest panel. Such thermal cycling can degrade the performance of the solar cells. Several control options are being investigated to reduce this problem. Rotating the satellite is a common solution, however, any rotation would have to cease during the pass over the ground station. This is feasible using magnetic torquers, but a more passive solution is desired. Another possibility is to design a continuous, conducting substrate that the solar cells are mounted to. This would allow the heat absorbed by the Sun-facing panels to distribute around the satellite to the space-facing panels. The technical feasibility of this option is being investigated.

4.4 Thermal Summary

A more detailed model that includes all of the major components and their heat dissipation is currently under design. The interfaces between components must also be better defined to improve the accuracy of the model. The control techniques discussed above will be simulated to determine the best thermal design for Photon.

5. Attitude Determination and Control

To achieve the Photon science mission, the attitude control system must point towards nadir within 5^0 during each pass. A pass begins when the satellite enters the horizon in view of the ground station and ends when it crosses the other horizon out of view of the ground station. These passes, which occur a maximum of three consecutive orbits per day, constitute the active control mode of the satellite. For the remainder of the orbits, the satellite will be controlled in a cruise mode. The third control mode is the acquisition mode. This mode will be implemented upon ejection from the Hitchhiker canister and in the event of a temporary loss of attitude knowledge. Acquisition mode will allow the Sun sensors to reacquire the Sun, and thereby regain knowledge of the satellite's attitude.

5.1 Requirements and Major Functions

The ADCS requirements are that the satellite point within $\pm 5^0$ to nadir and that the x axis be aligned with the velocity vector during the active mode. The major functions that the ADCS subsystem must perform are payload yaw orientation, satellite stabilization in response to external disturbances, and re-orientation in case the tip mass of the boom instead of the payload points to nadir. As previously indicated, the primary stabilization technique is the gravity gradient method. The main control actuators used for orientation are magnetic torque rods.

5.2 ADCS Configuration

Figure 5.1 shows the ADCS block diagram. The attitude determination and control of the satellite will be achieved using a position-plus-rate feedback control system for the pitch, roll and yaw axes. Although the primary stabilization is passive, control is required for initial orientation, payload orientation, and damping. The attitude determination system will provide the necessary information to verify correct pointing as well as to update the control system. Two-axis medium Sun sensors and a three-axis magnetometer are the sensors used for Photon. Sensor data is processed in the on-board computer to determine the attitude and is also relayed to the ground station for further analysis.

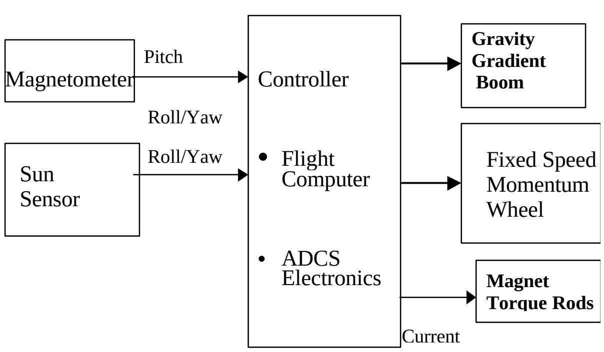


Figure 5.1 – ADCS Control Block Diagram

A six meter gravity gradient boom with a 3.3 kg tip mass is the primary means of stabilization. The restoring torque generated by the boom is greater than twice the external disturbances, which are mainly a result of aerodynamic drag (2 x 8.63e-06 Nm). A constant speed momentum wheel is fixed on the pitch axis to provide additional stability. Magnetic torque rods provide attitude and damping control; four are mounted in the z direction (along nadir), two in the x direction and two in the y direction.

5.3 Performance Analysis

Classical laws based on the measurement of the magnetic field rate of change in the satellite reference frame is used as control method since we are using magnetic torque rods as our primary control system.
Figure 5.2 shows the simulation result with the magnetic controller. Figure 5.2 shows that the errors converge to their steady-state values within two to three orbits which is quite acceptable for the Photon mission.

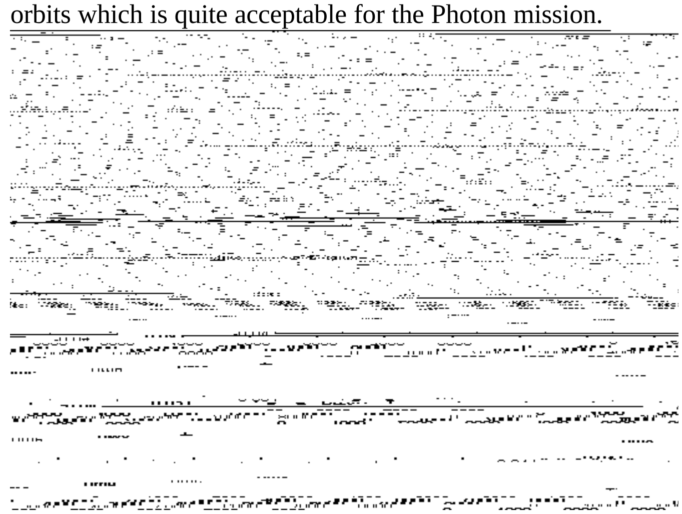


Figure 5.2 – Attitude Error for Yaw, Pitch, and Roll

5.4 ADCS Summary

A position-plus-rate feedback control system will enable Photon to point within 5^0 of nadir and align itself along the velocity vector during active mode. Attitude knowledge will be obtained using a magnetometer and sun sensor. Stabilization will be achieved using a gravity gradient boom and a constant rate momentum wheel. Magnetic torque rods will control the orientation of the satellite and provide damping.

6. Command and Data Handling

The Command & Data Handling (C&DH) subsystem is required to handle science data from the payload, and to identify, verify, and distribute commands to all other subsystems. The microprocessor and associated hardware must collect, format, multiplex, store, and download telemetry. The processor will need to do perform orbital calculations, attitude determination and control calculations, health monitoring functions, and decode and issue commands.

The processor selected for this satellite is the Intel 80C186. This processor has been used on many small satellites similar to Photon, such as CATSAT, TechSAT-1, UOSAT, FASat, KitSAT and PoSAT. The

80C186 has a good reputation due to its reliability and its flexibility with different software packages and power control features.
The primary C&DH requirements are: command (up-link) processing with 6.7 Kbps capacity, telemetry (down-link) processing with 80 Kbps capacity, and health monitoring (Table 2.1). In addition, the system must satisfy miscellaneous requirements like ADCS calculations, mission clock, and watchdog timers.

6.1 Software and Hardware Overview

The software objectives are to provide an efficient fault-tolerant design, to provide complete functionality for all spacecraft devices and to operate with a high margin of processing capability. In order to accomplish these objectives, the software design should have low level boot code located in ROM and high level boot and diagnostics located in EEPROM with an event driven scheduler loop allowing simple but accurate sequence processing (Figure 6.1).

The hardware objectives are to provide an efficient fault tolerant design, to provide ample processing power and storage capacity, to minimize power consumption and heat output, and to provide simple interfacing with external devices.

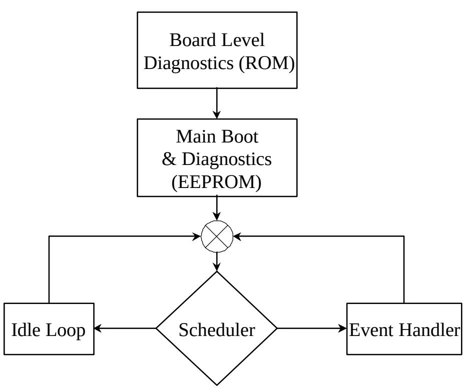


Figure 6.1 – Software Block Diagram

7. Communications

The communications subsystem utilizes much of the technology presented at the Small Satellite Conferences in prior years. A UHF receiver provided by SpaceQuest, an S-Band transmitter manufactured by Space Innovations Limited and a series of custom made Boeing antennae will serve the unique telemetry and commanding requirements of the Photon spacecraft.

The Communications system provides a hemispherical coverage for earth viewing and anti-earth viewing capable of simultaneous commanding and telemetry. Due to the unique nature of this BMDO funded project, special military frequency bands have been applied for use at Patrick Air force Base located at Cape Canaveral. The separate frequencies for commanding and telemetry coupled with the lower interference levels of the military bands will provide the science team with unparalleled data quality.

The Communications architecture will process output data rates of 80 kbps and transmit them to the Earth during an active pass. It will also pass incoming data rates to the Command and Data Handling subsystem at rates of 7 kbps (Figure 7.1, 7.2).

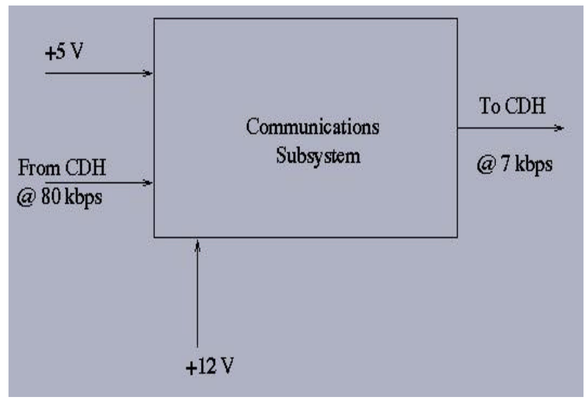


Figure 7.1 – Communication System Architecture

7.1 UHF Uplink Receiver

A SpaceQuest contracted UHF command receiver with corresponding redundancy is capable of receiving digital commands with high reliability at any time and in any orientation. The two receivers shall be continuously powered on to eliminate switching that might present a possible single point failure. In addition to having no switches in the receive path, the communications system is built with redundant serial outputs to the Command and Data Handling subsystem (C&DH) to assure high fidelity. A high C/N ratio has been designed to meet stringent bit error rates on the uplink.

A command rate of 2 MB/5 min. will more than satisfy the limited needs of this relatively autonomous experiment. The associated C&DH subsystem has the capability of receiving asynchronous serial digital data at 7 kbps with a storage capability of over 2 MB.

The requested frequency for the command uplink is 450 MHz frequency. A 30-foot Langley Space Center sponsored ground station will transmit data to the orbiting spacecraft from Cape Canaveral Airforce Station. A link analysis has been completed and is as follows for the uplink:

18 dB EIRP
2 dB losses

2 dB noise figure
0 dB reception gain
80 kHz bandwidth

On the spacecraft side, the Space Quest RX 100 series receiver requires only 100mW of power. The dimensions are also a figure of merit in a small spacecraft such as Photon. The entire enclosure shall be but 50 X 65 X 25 mm and weigh just 200 g. The receiver can handle up to 14 KBPS using a GMSK modulation scheme.

7.2 UHF Antennae
Two low profile whip antennae are mounted on either end of the spacecraft providing a hemispherical radiation pattern. The arched shape antennae circumvent the retroreflector allowing the science team to maximize the payload, while staying within the HES height restriction.

7.3 S-Band Downlink Transmitter
The Space Innovations Limited S-Band SIL STX-90 series transmitter is capable of transmitting real-time science data at a rate of 80 Kbps during active mode. The system will also transmit 36 hrs of stored engineering data (1.1 MB) during the 5 min. pass. The telemetry subsystem will handle input serial digital data asynchronous from the C&DH system.

Utilizing a 2.4 GHz frequency range the S-Band Transmitter will be capable of an 80 kHz bandwidth spread. Although only transmitting roughly .2 Watts the frequency bandwidth will enable high data rates for dowlinking the payload data during one active pass. A 32 dB reception gain has been designed. Currently the SIL ground station is under consideration for use in the downlink. The power consumption is reasonable for a low power transmitter requiring just 4 watts of input power.

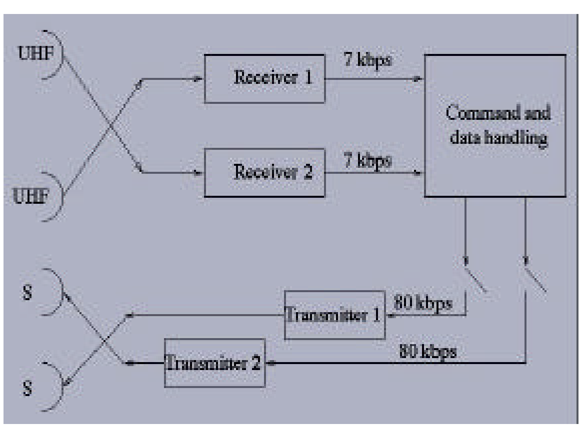


Figure 7.2 – Communications Block Diagram

A mass of 1.2 kg and dimensions of 180x95x45 mm make this the ideal transmitter for the Photon small satellite.

7.4 S-Band Antennae
Two quarter-wave bent monopole antennas will provide good low angle radiation as well as a low profile. Unlike other options such as patch antennas, which achieve maximum radiation when in the zenith position, these antennae will allow transmission during most of the ten-minute pass. The resulting system shall be capable of providing telemetry while in line of sight of the Photon earth station with the satellite in any orientation. Linear polarization will be utilized despite a 3 dB loss due to the nature of the bent monopole. The signal to noise ratios however are still sufficient to meet the Photon telemetry requirements. (Figure 7.3)

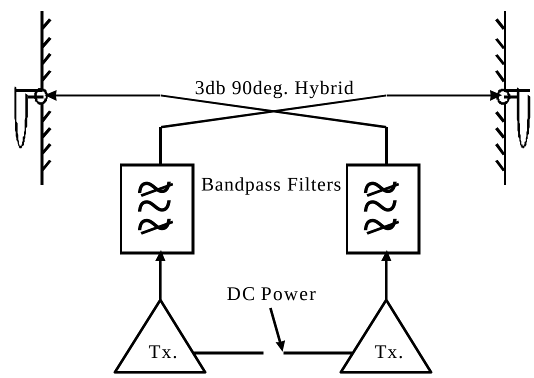


Figure 7.3 – Downlink Antennae

7.5 Communications Summary
System architecture diagrams have been finalized and the first revision of the printed circuit boards is complete. Work on the acquisition of requested frequencies is still in progress.

A link budget has been completed detailing the flux density and carrier to noise levels for both the uplink and downlink.

Utilizing an UHF/S-Band link is a unique approach to avoid the all too common interference in the VHF regime. In addition, the S-Band system permits a higher data rate to suit the needs of the Photon spacecraft. The antennae are also custom designed to meet mission constraints.

8. Electrical Power Subsystem

The Electrical Power Subsystem team has designed the Photon satellite with the ability to provide a continuous source of electrical power to all required loads for the duration of the mission. This electrical design incorporates the capabilities of power generation, storage and distribution for utilization by each subsystem within the spacecraft.

The Photon mission constraints and requirements that are major drivers in the design of the power subsystem and are as follows:

- Mission Type: LEO
- Mission Life: 1 year
- Area of satellite must be less than or equal to that of a NASA Hitchhiker
  (d = 19 in. h = 18.5 in.)
- Payload requires a peak power of 94.8 W for 10 minutes a pass, three passes per day

8.1 Power Requirements

Payload power requirements are well suited for a small satellite design. The residual operating power of the spacecraft is 6.16 W. An analysis of eclipse timings during worst case illumination reveals a 10.32 W minimum requirement for each orbit in order to sustain residual conditions. In order to store energy for the peak loads, however, 10.32 W is not an adequate power supply. Fortunately, these peak loads will be active only ten minutes of an active pass. The payload beacon requires 20 Watts and the gimbal system 50 watts.

8.2 Power Source

Solar Power Corporations Silicon Array will provide the spacecraft with a minimum of 20.6 watts per orbit under worst case illumination conditions. The area of a photon solar panel will be .185 meters by .444 meters. The proposed Silicon array features 1 circuit of 42 cells in series per panel.

These cells will be bonded to the aluminum honeycomb panel, which is bolted to the frame of the spacecraft. Typically a Kapton sheet will be utilized to provide insulation between the solar cells and the aluminum honeycomb structure. The attachment substrates accounts for up to 50% of the total panel mass weighing 114 grams per panel. A CV 2568 adhesive is used to bond this honeycomb structure to the spacecraft skin. The total panel will have a mass of approximately 234 grams.

The array will utilize coverslides to provide hermetic sealing allow the cell to receive sunlight while reject heat. A textured coverslide is used for body-mounted solar cells that do not actively point towards the sun. It reflects incident solar energy back onto the solar cell, improving overall efficiency.

One type of coverglass commonly utilized is a cerium doped material often on the order of .015 cm in thickness. The Silicon array made by Solar Power Corporation employs a similar material, a ceria doped borosilicate glass combined with an Anti-Reflecting coating, to assure maximum incidence. While only .015 cm in thickness the Corning AR213 coverglass has a mass of 18.1 grams per panel. The material is fastened to the solar array with a space qualified substance know at DC93-500 weighing only 3.76 grams per panel. This coverglass has a good thermal coefficient match to solar base materials and is resistant to UV darkening.

8.3 Energy storage

For Photon, there will be 16 eclipse periods per day with less than 60 minutes to charge the battery system between them. For short duration missions, it may be possible to drive the batteries at a Depth of Discharge higher then the recommended 30%. The design has rated the possible DoD at 50%, saving on storage capacity required and related factors.

The nickel cadmium batteries chosen for the Photon spacecraft are space qualified, very reliable, and can be repeatedly recharged for thousands of cycles without concern. To provide the nominal voltage for the satellite bus, these batteries are arranged in series. The specifications for the batteries are given in Table 8.1.

Table 8.1 - Battery Specifications

Brand: Sanyo
Model: Cadnica series. Kr5000
Voltage: 1.2 each
maH: 5600 maH
Length: 61.5 mm
Dia: 34 mm
Mass: 166 grams
Quick charge: 1200 (ma) in 7-8hr.
Depth of Discharge: 30% (With possible DoD of 50%)

For Photon, it was estimated that 24 batteries (1.2V each, 28.8V total) are required for the satellite to perform efficiently for the satellite lifetime of one year. These batteries provide the satellite with 161.28 W-hrs of energy at a total nominal voltage of 28 volts.

The power design team is currently investigating the possible use of the certified MightySat battery storage design. Should this method be implemented, NASA will require only limited safety verification. Cost savings on the validation and testing of such a design would be significant. Until that agreement is reached the NiCd system of 24 cells connected in series will meet the mission storage requirements.

8.4 Power Distribution

The general approach for the spacecraft's overall distribution system will be a decentralized, unregulated bus (Figure 8.1). Power converters will be installed at the various load interfaces to assure the required voltages are assigned.

In addition to the previously described blocking diodes on the solar array the distribution network will also carry the appropriate fuses needed to prevent short-circuits, and isolate possible fault locations. A series of fault detection devices will monitor the health and status of their assigned loads.

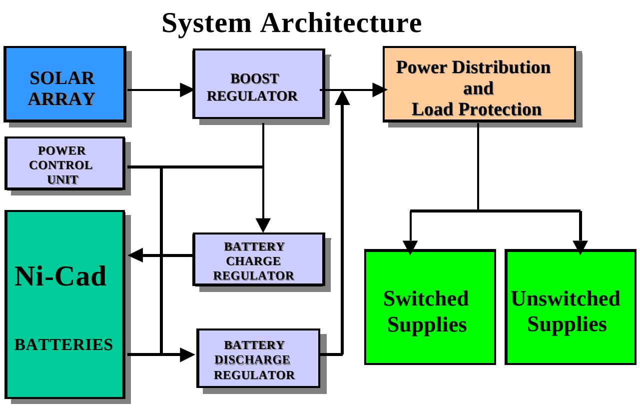


Figure 8.1 – Power Distribution Architecture

8.5 Power Regulation and Control

Each subsystem component requires a different operating voltage. Power converters will connect loads susceptible to noise or requiring voltage conversion to the distribution system. These converters isolate the load from the noise on the bus, and regulate the power provided to the load against disturbances. The typical power required by these two systems is estimated to be approximately 25% of the total power, 5% of which is lost in the transfer of power through the cables and the other 20% is power required to regulate each system.

The Photon spacecraft will use a peak power tracking system to regulate the power generated by the Solar Arrays. This is a non-dissipate method that extracts only the required energy from the Solar Arrays. Hence, if the batteries are fully charged and the energy consumption on board is minimal, the buck boost regulator will alter the voltage operating point of the solar array source to actively generate less energy. This pulse-width-modulated regulator utilizes power transistors in a switching mode rather than in an analog mode as in conventional shunting systems. With the higher efficiency also comes the added cost of increased complexity. A buck-boost regulator design tracks the input and output voltages and maintains a constant ratio by controlling the array's characteristics. When the energy demanded exceeds the peak power point the buck boost regulator allows the operating point to swing to its maximum level. Additionally, if an active mode were to occur during a sun cycle the buck-boost regulator would be able to track the increase power requirements allowing the circuit to “boost” the voltage point on the array's I-V curve. The buck-boost regulator is an ideal choice for the low earth orbiting, load variant Photon spacecraft due to its drastic reduction in thermal shunting requirements and its the dynamic production capability.

8.6 Power Summary

The components selected for the spacecraft have long flight histories. Solar Power Corporations' silicon, body mounted solar panels will supply sufficient power to both the Sanyo Ni-Cad batteries and subsystem loads. The AMSAT contracted buck boost power regulator will assure that the energy generated and stored meets the needs of the spacecraft at any given time.

9. Conclusion

Due to an agreement with NASA for a free HES launch, the Photon satellite bus design has had a unique, bottom-up approach. Instead of baseline science requirements, the design team was given broad science goals and a request to maximize bus and payload capabilities. Because this is FSI's first spacecraft design and the bus is designed by students, extensive feasibility studies were required. After preliminary systems design, design and analyses of the subsystems were performed. Optimization of the retroreflector and selection of the attitude determination and control system proved especially challenging.

With final design work and a few detailed analyses nearing completion, the conceptual design of the Photon satellite bus is coming to a close.

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