PuTEMP
Purdue University Thermodynamic Experimental Microgravity Platform

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Presentation Outline

- Mission Statement and Objectives
- Satellite Description
- Concept of Operations
- Design Requirements
- Orbit Selection
- Launch Vehicle Integration
- Spacecraft as a System
- Spacecraft Subsystems
- Conclusions
PuTEMP’s Mission

The useful life of a large satellite is constrained by the onboard propellant. Improvements in efficient management of the onboard propellant could result in significant extensions of the satellite’s useful life. To this end our group intends to design a platform for micro-gravity propellant research. Specifically the satellite experiment will pertain to accurate onboard propellant measurement using a thermal gradient model.

-=PuTEMP Design Team=-

Mission Objectives

• Primary Mission Objective
  – Provide experimental data on current techniques for Thermal Propellant Gauging, with the goal of increasing the accuracy of the technique.

• Secondary Mission Objectives
  – Launch Purdue University into the small satellite spotlight
  – Provide another satellite to AMSAT after primary mission is complete
Satellite Description

- **Dimensions [cm]**
  - Maximum 33 x 33 x 63
  - **PuTemp** 30.127 x 30.127 x 60

- **Mass [kg]**
  - Maximum Allow. 68
  - **PuTemp** 45

Concept of Operations

- **Four phase Concept of Operations**
  - **Launch Preparation**
    - Minimal ground handling required
    - System checkout and battery charge
  - **Spacecraft Deployment**
    - S/C must be inert (no radio transmissions) during launch
    - Requires autonomous operation
    - G.G. Boom deployed autonomously to achieve necessary attitude
  - **On-Orbit Operations**
    - Cyclic operation of S/C defined by experimental run
    - Lifetime of <1000 cycles and 1 year
  - **End-of-Life**
    - S/C used as AmSAT relay
Major Design Requirements

- Customer requirements drive satellite design
- Payload Customer
  - Primary design driver
  - Successful collection of test data required by Customer
  - Constraints on payload environment
- Launch Provider
  - Dimension and mass limits placed on S/C
    - 33x33x63cm, 68kg
  - Structural load requirements
- Purdue University
  - Low-cost
  - Design within the capabilities of Purdue University facilities

Payload, SPT
  - Sloshing < deg/sec
  - Spinning < deg/sec
  - Knowledge of S/C motion during Exp.
  - 45 Watts for 15 min.
  - Power
  - Gravity Gradient
  - Sun-Sensors
  - Magnetic Torquers
  - Structural Layout
  - Launch Vehicle
  - Sun Synchronous Orbit
  - ADCS
Major Design Requirements

Payload requirements are presented below

<table>
<thead>
<tr>
<th>Requirement</th>
<th>Specification</th>
<th>X</th>
<th>Additional Information</th>
</tr>
</thead>
<tbody>
<tr>
<td>SPT shall be 7.5 cm radius by 25 cm height</td>
<td>Accurate tank size</td>
<td></td>
<td>Accurate tank size</td>
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<tr>
<td>Shall have the ability to raise tank temp. 10deg</td>
<td>Power capability</td>
<td></td>
<td>Battery sized for experiment run</td>
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<tr>
<td></td>
<td>Number of</td>
<td></td>
<td>Number acceptable for heat distribution and sensing</td>
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<td></td>
<td>Heaters/Thermisters</td>
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<td>Shall have the ability to store data from</td>
<td>CPU capabilities</td>
<td></td>
<td>CPU memory storage of 4 Meg</td>
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<tr>
<td>Shall have the ability to retrieve data from</td>
<td>Communication with</td>
<td></td>
<td>6am-6pm sun-synchronous orbit, two passes over ground</td>
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<td>tank experiment</td>
<td>ground station</td>
<td></td>
<td>station a day</td>
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<td>Oscillations of tank shall be under 17.99 deg/</td>
<td>Oscillations within</td>
<td></td>
<td>motion of s/c controllable within criteria with magnetorquers.</td>
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<tr>
<td>sec and spin below 50.42 rad/sec</td>
<td>control criteria</td>
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<td>Shall have the ability to determine if s/c,</td>
<td>measurement of</td>
<td></td>
<td></td>
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<td>and tank, are oscillating greater than 5 deg/</td>
<td>Sun sensors detect</td>
<td></td>
<td></td>
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<tr>
<td>sec</td>
<td>oscillations as</td>
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<td>small as 5 deg/sec.</td>
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<td>Magnetorquers create</td>
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<td>large enough dipole</td>
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<tr>
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</tr>
<tr>
<td></td>
<td>magnetorquers</td>
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</table>

Orbit Design

- Based on the power needs, the orbit needs to have a maximum solar exposure time.
- The following criteria were used in the selection of the orbit (single satellite):
  - Minimum eclipse time (<50% of orbital period)
  - Communications with Zucro (Purdue University, \(i \geq 40^\circ\) if LEO)
  - Perturbation effects (lifetime effects)
  - Accessibility to specialized orbits
- Since there is no active propulsion system and the limitations as a secondary payload, GEO, Molniya, Lagrangian libration points, and lunar crossing orbits can be ruled out.
• Candidate orbits:
  – Polar; \( i = 90° \)
  – Sun-Synchronous; \( i = i(alt) \)

• Polar orbits
  – \( J2 \) geopotential effects cause RAAN to change at a rate dependent on the semimajor axis eccentricity.
  – Subsolar point only in orbit plane during certain times of the year.
  – Worst case gives an eclipse time of 50% of the orbital period.

• Sun-Synchronous
  – Nodal precession rate is 0.986°/day.
  – Inclination is a function of the chosen altitude.
  – The RAAN can be set such that the orbit normal always points towards the sun.
  – Eccentricity is nearly zero.

• Altitude trade

<table>
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<tr>
<th>Characteristic</th>
<th>Allowed Range (km)</th>
<th>Comments</th>
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<tr>
<td>Launch Capability</td>
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<tr>
<td>Radiation</td>
<td>&lt;= 800</td>
<td>Below Van Allen Belt</td>
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<tr>
<td>Communications</td>
<td>&gt; 600</td>
<td>In general higher is better.</td>
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</table>
Orbit Design

- Chosen orbit:
  - Sun-Synchronous orbit with an altitude of 800 km, a period of 100.87 minutes, and an inclination of 98.6°. The RAAN is chosen to give a dusk-dawn orbit, which will set our launch window depending on launch site.
  - At this altitude the atmospheric drag will be the most influential perturbing force. This will give an orbital lifetime of 14.5 years.
  - There will be two good passes (max. elevation > 60°) per day with a LOS of about 15 minutes. These passes will be roughly 6 am and 6 pm.

Launch Vehicle Integration

- Launch vehicle chosen based on intended orbit and past history of launching secondary payloads
- S/C dimensions and weight chosen to allow integration into several different vehicles (Ariane 5, Delta II, Space Shuttle Hitchhiker)
  - 33x33x63cm, 68kg
- Ariane 5 chosen as primary launch vehicle based on most stringent structural limits imposed on S/C

<table>
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<th>Steady State and Dynamic Frequency (Hz)</th>
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<th>Lateral</th>
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<td>-6.0 to +6.0</td>
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<td></td>
<td>&gt;90</td>
<td>&gt;45</td>
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Spacecraft as a System

Internal Layout

- Antenna: Salmon (top view)
- Batteries: White Boxes
- Bus: Brown (not in view)
- CPU: Purple Box
- Gravity Gradient Boom Box: Red Tail Box
- Load Bearing Frame: Grey (Aluminum 7075-T6)
- Magnetic Torquers: Blue Rods
- Modem: Yellow
- Receivers: Dark Teal (top view)
- Side Panels: Orange (top view) (Aluminum 7075-T6)
- Solar Panels: Blue (top view)
- SPT: Green
- Sun Sensors: Black Boxes
- Transmitters: Green Boxes (top view)
- Attachment Plate for Launch Vehicle: Orange Plate (bottom of satellite)

Subsystem Mass Budget and Margins

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Target Mass (kg)</th>
<th>Actual Mass (kg)</th>
<th>Maximum Mass (kg)</th>
<th>Mass (kg)</th>
<th>% of Total</th>
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</table>
Spacecraft as a System

Mass Moments of Inertia
- $I_{xx} = 102.7 \text{ kg/m}^2$
- $I_{yy} = 102.8 \text{ kg/m}^2$
- $I_{zz} = 0.9 \text{ kg/m}^2$

Payload Subsystem

Payload Components
- 1 Simulated Propellant Tank
  - 15.24 cm x 30.48 cm (6in x 12 in)
  - 25% Fill Fraction
- Payload Electronics
  - 22 Thermistors
    - Multiples of 8 with current Payload interface design
  - 6 A/D converters
  - 6 Comparison Units
Payload Subsystem

- **Power**
  - 63 Watts Maximum
  - .5 Watts Minimum
- **ADCS**
  - Oscillation Frequency $< 17.99$ deg/sec
  - Spin Rate $< 50.42$ deg/sec
- **Structures**
  - Must maintain thermal isolation of SPT

Payload Subsystem Requirements

- **Thermal**
  - Requires a spacecraft equilibrium
  - 10 degree C. rejection
- **C&DH**
  - Maximum of 8 Megabytes of storage
1. Specify an experiment duration

2. What is the power required to heat by 10 degrees C?

3. Repeat for several experiment durations

- Sensor Positions
The AD&C Subsystem shall guarantee a spacecraft attitude within the ranges of operation dictated by the Payload.
- Oscillation Frequency $< 17.99$ deg/sec
- Spin Rate $< 50.42$ deg/sec

The AD&C Subsystem shall provide a Nadir Pointing Spacecraft for a useful line of sight for TT&C.

The AD&C Subsystem shall be as power economical as possible due to small satellite limited power capability and acquisition.

Control Strategy
- 4-meter Gravity Gradient boom with a 4-kg tip mass
- Two Magnetic Torquers
  1) Aligned with the Z axis (pointing Nadir) of the S/C
     - $6(Am^2)$ linear Dipole Moment
  2) Aligned with the Y axis of the S/C
     - $5(Am^2)$ linear Dipole Moment
Attitude Determination & Control

- Attitude Determination Strategy
  - 1 Sun Sensor on each of the faces that will be in contact with the sun
    - 2 Orthogonal Axis Sensors
    - Accuracy: 0.5 deg
  - 1 Magnetometer
    - 2 Orthogonal Axis Sensor
    - Placed in the Tip Mass of the Gravity Gradient Boom

Predicted Performance
- The S/C is predicted to operate well within the constraints imposed by the payload.
- The Gravity Gradient Stabilization Provides a stable but oscillatory Spacecraft
- The Magnetic Torquers successfully damp out the oscillations inherent in the Gravity Gradient Stabilization and sets the Spacecraft in an attitude favorable for the payload.
• Spacecraft Oscillations

Without Magnetic Torquers

With Magnetic Torquers

• Trade Studies
  - Trade Studies to Size the Gravity Gradient boom were carried on with the following things in mind:
    - Increase of the Moments of Inertia with respect to the X and Y axis of the Spacecraft
    - Stiffness of the Spacecraft towards the Magnetic Torquers
    - Minimum Gravity Gradient Boom-Tip Mass Configuration that provides a torque at least one order of magnitude greater than the largest Disturbance Torque
    - Magnetic Torquers with a linear Dipole Moment enough to carry on any maneuver desired (e.g. oscillations damping, emergency maneuvers)
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C&DH Requirements

- PuTEMP requirements for the C&DH are:
  - 1) low power consumption (most components below 5W)
  - 2) simple setup and usage (alternatively, small packaging, as allotted for by the satellite requirement)
  - 3) storage of the data until downlink (data amount will be on the order of 4 MB, storage provided must be greater than this)
  - 4) space-hardened

C&DH Subsystem components

- In order to meet requirements, investigation of “off-the-shelf” space-hardened hardware was made
- Found at SpaceQuest®, a company that specializes in small satellites
- Primary components:
  - Flight Computer
  - Memory Board
- These components are all space-hardened
- Flight computer has a “memory-check” default that periodically checks and corrects each time it is read, in order to correct inherent bit errors induced by some radiation
C&DH Flight Computer

- Static RAM: 512K
- Mass: 150 grams
- Area: 140 mm x 165 mm
- Power consumption: 10 mW
- Operating voltage: 3.3V
- Operating Temperature: -10°C to 60°C
- Data storage capabilities

FCV-53

C&DH Memory Board

- Static RAM: 8MB, though can go up to 16 MB
- Mass: 150 grams
- Area: 140 mm x 165 mm
- Power consumption: 10 mW
- Operating voltage: 3.3V
- Operating Temperature: -10°C to 60°C
- Directly mountable to the Flight Computer

FMB-53
Region of highest flux

Low flux

This is for an 800-km case of the Canadian Space Telescope

Software will be programmed once exact specifications are made for:

- Satellite reorientation
- Data sampling with regard to the sensors
- A/D conversion
- Data storage
- Data transmission to transmitter
- Data reception from receiver
C&DH Trade Studies

- Trade study made with the following assumption:
  - 10 minutes of visibility
  - Data rate of 9600 bps
  - Sampling at least at twice the frequency of the experiment in order to satisfy the requirements set forth by payload (note that this sampling was not a predominant factor for the trade study)
  - A total of approximately 4 MB (plus or minus of 1 MB) of data was required, including satellite health information
  - Hence the choice of the 8 MB data storage

PuTEMP requirements for the TT&C are:
- 1) low power consumption (the only component allowed to exceed 5 Watts is the antenna);
- 2) simple usage and failsafe redundancy;
- 3) be able to perform to a data rate of at least 9600 bits per second (bps);
- 4) use of existing communication protocol for simplicity (preferred protocol is existing AX.25 protocol);
- 5) in the completion of the link budget, have a satisfactory time margin with which to communicate to Earth.
TT&C System

Ant. 1

UHF

XTR

VHF

RCVR

GMSK

Modem

12V

Ant. 2

Data Collection
Device

A/D

Converter

Flight Computer and
Data storage

To

payload

To

AACS

TT&C Choosing data rate

Data rate, KBps
### TT&C Link Budget

<table>
<thead>
<tr>
<th>Item</th>
<th>Symbol</th>
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<th>Uplink</th>
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<tr>
<td>Margin</td>
<td>dB</td>
<td>12</td>
<td>22</td>
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</tbody>
</table>

### TT&C Importance

- The importance of TT&C is often underrated
- Various uses:
  - Commands
  - Attitude reorientation
  - Health information
  - Data relay

Courtesy of:
http://www.dfd.dlr.de/stations/oberpfaffenhofen.html
In order to meet requirements, investigation of “off-the-shelf” space-hardened hardware was made

Found at SpaceQuest®, a company that specializes in small satellites

Primary components:
- GMSK Modem
- VHF Receiver
- UHF Transmitter
- UHF Receiver
- Patch Antenna

These components are all space-hardened

**TT&C Modem**

- BER: $10^{-5}$
- Fixed channel: 2400 to 9600 bps
- Mass: 60 grams
- Area: 77 mm x 70 mm
- Power consumption: 130 mW
- Operating Temperature: -20°C to 70°C
TT&C UHF Transmitter

- Frequency range: 400-450 MHz
- Frequency stability: ±5 ppm
- Mass: 300 grams
- Volume: 94 mm x 72 mm x 28 mm
- Power consumption: 77 mW
- Operating Temperature: -10°C to 60°C

TT&C VHF Receiver

- Front end noise figure: < 1dB
- Frequency stability: ±5 ppm
- Mass: 198 grams
- Volume: 83 mm x 72 mm x 28 mm
- Power consumption: 77 mW
- Operating Temperature: -10°C to 60°C
TT&C Patch Antenna

- 150 mm x 70 mm x 30 mm
- Linear polarization
- 2 antennas, one for transmit, one for receive
- Omnidirectional

EPS Overview

- Based on payload needs, the EPS does not require the capability to recharge the batteries every orbit
- Basic EPS design is Direct-Energy-Transfer (DET)
- A DET system using shunt regulators provides the following advantages:
  - Efficient – Only power not needed by the S/C is dissipated
  - Simple/Reliable – Shunt regulators are self-controlled
  - Low Cost – Shunt regulators are inexpensive devices
- Bus voltage will be quasi-regulated with charge voltage being fixed and discharge voltage fluctuating based on battery DOD.
EPS Overview

- Block diagram showing major EPS subsystem components

```
Power Generation and Collection – Secondary Power

Charge and Discharge Control – Voltage/Temperature Sensing

Energy Storage – Primary Power

Power Conversion and Regulation – Quasi-Regulated

Power Distribution

Subsystem Loads
```

- Below is a schematic of the basic EPS design:

```
Secondary Power - SA

Regulation and Distribution

Primary Power - Batteries
```
Operating Modes

- All CD&H components must operate during all operating modes so S/C remains in contact with ground station

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Component</th>
<th>Required Power (Watts)</th>
<th>Margin</th>
<th>Experimental Mode</th>
<th>Recharge Mode</th>
<th>Transmission Mode</th>
<th>Reorientation Mode</th>
<th>Safe Mode</th>
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<td>Sensors</td>
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<td>1.4</td>
<td>x</td>
<td>(4)</td>
<td>(4)</td>
<td>(4)</td>
<td></td>
</tr>
<tr>
<td>Attitude</td>
<td>Magnetometer</td>
<td>0.14</td>
<td>1.4</td>
<td>x</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>Magnetic Torquers</td>
<td>1.54</td>
<td>1.4</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>Sun Sensor</td>
<td>0.49</td>
<td>1.2</td>
<td></td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Total Power (Watts): 83.76  13.38  13.38  15.54  11.08
Time to Charge (Hrs): 11.80  11.80  12.62  10.93

EPS Primary Power

- Secondary power was sized to provide all power required during peak power operation (Experimental Mode) and to provide sufficient voltage for all S/C loads
- Sanyo Cadnica NiCd battery technology was chosen for secondary power:
  - Flight tested in numerous small satellites
  - Commercially available, the Cadnica batteries are inexpensive
  - Cycle life (<1000) is not a concern for PuTEMP allowing a DOD of 60%
- Thermal control of batteries is critical performance
Overall SA efficiency was not considered a primary design driver based on EPS requirements.

Although silicon cells are less efficient and more susceptible to radiation damage, they are roughly 45% lighter and considerably more cost-effective.

K4702 Silicon Solar Cells from Spectrolab were chosen.

<table>
<thead>
<tr>
<th></th>
<th>Series Cells</th>
<th>Parallel Panels</th>
<th>SOL Total Voltage (V)</th>
<th>SOL Total Power (W)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Spectrolab GaAs Triple-Junction</td>
<td>7</td>
<td>7</td>
<td>15.22</td>
<td>43.01</td>
</tr>
<tr>
<td>Emcore GaAs Triple-Junction</td>
<td>6</td>
<td>8</td>
<td>14.71</td>
<td>51.62</td>
</tr>
<tr>
<td>Spectrolab Silicon</td>
<td>26</td>
<td>2</td>
<td>14.69</td>
<td>23.53</td>
</tr>
</tbody>
</table>

Voltage-Temperature Cutoff (VTCO) used as charge control:
- Discontinues battery charge when temperature or voltage limit is reached
- Provides high-level charging without compromising battery life and reliability

Charge control consists of three main components:
- Charge Control Unit (CCU) – Determines when battery is fully charged and regulates charge
- Voltage sensor – Relays battery voltage to CCU
- Temperature sensor – Relays battery temperature to CCU
Thermal Environment

- Heat Sources
  - Sun
    - 1370 W/m^2
  - Earth
    - 250 W/m^2
  - Electronics
    - 5 Watts on Average

- Payload
  - During Experiment and Cool-off
    - 45 Watts
  - At Equilibrium
    - 0 Watts

Predicted Spacecraft Thermal Performance

- Maximum Temperature ~ 35 degrees C
- Minimum Temperature ~ 25 degrees C
- Average Spacecraft Temperature ~ 25 degrees C

Based on Equivalent Sphere Analysis
Thermal Components

- Solar arrays on exterior
- MLI on tank
- Thermistors on batteries
  - Part of spacecraft health system

Structural Design

- Mass < 68 kg (Actual Mass = 45 kg)
- Launch Vehicle Structural Requirements – ‘design to’ Ariane 5
  - Axial Frequency >90 Hz
  - Lateral Frequency >45 Hz
  - Dimensions versatility
    - Compatible with Delta, Ariane IV, Ariane 5
- Layout Design Requirements
  - Payload (ADCS)
  - $I_{zz}$ (nadir pointing) – Minimum MOI
  - $I_{xx} \approx I_{yy} > I_{zz}$
• Frame Design and Material Selection
  • Initial SMAD Calculations
  • Other small satellite comparison (Palamede, Cubesat)
    • Greater length gives lower frequency
  • Materials Selection, Aluminum 7075.T6
    • Manufacturable at Purdue (requirement)
    • Composites evaluated – Quasi Isotropic

• Frame Evolution

<table>
<thead>
<tr>
<th>Structural Component</th>
<th>Size [cm]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Shelf</td>
<td>30x30x1</td>
</tr>
<tr>
<td>Angle Iron Shelf Support</td>
<td>30x1x1</td>
</tr>
<tr>
<td>T-Bar Cross Support</td>
<td>0.75x1.0 (web)</td>
</tr>
<tr>
<td>Side Panels (orange)</td>
<td>30x0.127x60</td>
</tr>
<tr>
<td>Frame</td>
<td>0.635 Thickness 5.08 Web</td>
</tr>
</tbody>
</table>
Structural Design

• Load Bearing Structure Sizing
  – Frequency Analysis & Static Loading (Launch)
    Ironcad Drawings → Ansys Analysis

• Assumptions and Discrepancies from Ironcad to Ansys
  – Solid volume frame (Aluminum 7075.T6)
  – Safety Factors of 2.0 on Limit Loads
  – Panels not modeled, too many elements
    ➢ Mass of entire structure is slightly larger than Ansys output
    ➢ Accounted for in Mass and Inertia Calculations
  – Frequency Analysis
    ➢ Internal Masses – Solid blocks of material imported from Ironcad
      with corresponding material mass density assigned
  – Static Loading Analysis
    ➢ TT&C components modeled as ‘lumped mass’
    ➢ Distributive forces modeled (over area/nodes)
    ➢ Components attached to bottom surface not included, bottom surface constrained by launch vehicle
Structural Design

- Frequency Analysis
  - Launch Vehicle Requirement:
    - Axial >90 Hz
    - Lateral >45 Hz
  - Including Internal Masses
  - Does not take into account mass of panels, 0.64 kg each

<table>
<thead>
<tr>
<th>Mode</th>
<th>Frequency [Hz]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Axial</td>
<td></td>
</tr>
<tr>
<td>1</td>
<td>92.3</td>
</tr>
<tr>
<td>2</td>
<td>106.1</td>
</tr>
<tr>
<td>3</td>
<td>106.8</td>
</tr>
<tr>
<td>4</td>
<td>115.0</td>
</tr>
<tr>
<td>5</td>
<td>119.2</td>
</tr>
<tr>
<td>Lateral</td>
<td></td>
</tr>
<tr>
<td>1</td>
<td>59</td>
</tr>
<tr>
<td>2</td>
<td>61</td>
</tr>
</tbody>
</table>

- Static Loading Analysis
  - Launch Vehicle Requirement: Limit Load Factor, Lateral = 5.5 g
    - Axial = 7.5 g
      - F.S. = 2.0

<table>
<thead>
<tr>
<th>Density (kg/m²)</th>
<th>Young's Modulus (MPa)</th>
<th>Ultimate Tensile Stress (MPa)</th>
<th>Yield Compressive Stress (MPa)</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>2800</td>
<td>71</td>
<td>460</td>
<td>380</td>
<td></td>
</tr>
</tbody>
</table>

Material Property Margin

<table>
<thead>
<tr>
<th>Stress [MPa]</th>
<th>Stress Compres./Tens.</th>
<th>Material Property Margin Compression / Tension</th>
</tr>
</thead>
<tbody>
<tr>
<td>X</td>
<td>-12.55 / 10.57</td>
<td>-0.967 / -0.977</td>
</tr>
<tr>
<td>Y</td>
<td>-12.38 / 12.06</td>
<td>-0.967 / -0.973</td>
</tr>
<tr>
<td>Z</td>
<td>-0.73 / 0.13</td>
<td>-0.974 / -0.98</td>
</tr>
</tbody>
</table>

Shear Stress [MPa]

<table>
<thead>
<tr>
<th>Shear Stress [MPa]</th>
<th>Stress Compres./Tens.</th>
<th>Material Property Margin Compression / Tension</th>
</tr>
</thead>
<tbody>
<tr>
<td>XY</td>
<td>-3.91 / 6.85</td>
<td>-0.990 / -0.985</td>
</tr>
<tr>
<td>YZ</td>
<td>-3.18 / 2.43</td>
<td>-0.992 / -0.995</td>
</tr>
<tr>
<td>XZ</td>
<td>-2.31 / 2.40</td>
<td>-0.994 / -0.994</td>
</tr>
</tbody>
</table>

Maximum Deflection [cm] 0.0069

All weight of SPT assumed carried by mid-shelf.
Structural Design

- Other Subsystem Requirements for Structural Design
  - ADCS: $I_{xx} = 102.7 \text{ kgm}^2$, $I_{yy} = 102.8 \text{ kgm}^2$, $I_{zz} = 0.9 \text{ kgm}^2$
  - C&DH: Computer components near batteries and tank, minimize data error
  - Power: Batteries near SPT, main power draw
  - TT&C: Communication components nadir pointing

Summary of Spacecraft

- All requirements have been met by the S/C design
- Preliminary cost estimation has not been performed