PUTEMP

Purdue University Thermodynamic Experimental Micro-gravity Platform

System Requirements Document

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1.0 Mission Overview

The useful life of large satellites is constrained by the onboard propellant. Improvements in the efficient management of the onboard propellant could result in significant extensions of the satellite's useful life. To this end PUTEMP shall design a platform for micro-gravity propellant research. Specifically the small satellite experiment will pertain to acquiring data that will be used to correlate a fluid thermal gradient model to the actual propellant level in the tank. Only a micro-gravity platform will be able to accurately simulate the propellant distribution of a satellite on orbit. The properly correlated thermal gradient model will allow for the accurate prediction of propellant levels in larger satellites.

Professor Collicott of Purdue University is actively involved in low gravity fluids experimentation and modeling for industry customers. The thermal gradient model developed by his research team is what the experimental results will be correlated against. Therefore, Professor Collicott and his team are considered the primary customer. Satisfying their experimental requirements shall drive the satellite design.

2.0 Payload

The baseline experiment consists of a simulated propellant tank (SPT) with known mass of fluid. The tank shall be pressurized to a to be determined (TBD) amount. The tank shall be constructed of aluminum of a TBD thickness based on aluminum's low cost and easy manufacturability. A TBD number of small heaters and thermistors shall be placed on the outside of the tank as a representative model of current large satellite tank designs. As for the experiment information, known amount of heat energy shall be transferred into the tank. The thermistor temperature readings shall be recorded and stored for transmission to a ground station when appropriate.

Power capabilities and experiment needs[1] shall determine the SPT size. The SPT tank shall be representative of current large satellite tank designs, having a cylindrical shape with hemispherical end-caps. Based on experimental considerations, the smallest tank size allowable is 7.5cm radius by 20cm total length. A mixture of water and propylene glycol (antifreeze) shall be the fluid used to simulate the propellant. The antifreeze shall lower the freezing temperature of the fluid. Propylene glycol shall be chosen if this is deemed necessary over glycol because it is considered less hazardous than glycol (i.e. propylene glycol has food variants). The overall size of the SPT shall be determined based on the power required for heating, launch vehicle restrictions, and temperature measurement resolution (as based on tank surface area). The resulting science collected may be slightly different for SPT's of drastically different size. The SPT shall not incur local accelerations above those deemed acceptable for experimental accuracy (sloshing requirements).

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[1] Experiment needs is in reference to the validity of thermal information collected by the tank sensors.

Our mission versatility lies in that several experiments can be conducted for any on SPT. As well, future low gravity fluid experiments on small satellites may be anticipated. See appendix for additional experimentation ideas.

3.0 Launch Vehicle

The launch vehicle shall provide a Low Earth Orbit (LEO) for the satellite as a secondary payload. The satellite (i.e. payload) shall be capable of launching on as many launch vehicles as possible. Launch vehicle versatility requires the overall dimensions of the satellite to fit inside many secondary payload compartments. The launch vehicle trajectory characteristics are a consideration in vehicle choice (discussed more in Trajectory Design). Telecommunication restrictions require an orbit altitude greater than 275km and less than 1000km. Table 3.1 is a condensed list of launch vehicles being investigated and their corresponding compartment sizes.

Launch Vehicle	Performance Capability	Max. Payload Mass (kg)	Max. Payload Height (cm)	Max. Payload Width (cm)
Ariane IV	LEO, SSO, GTO	50	44.96	44.96
Ariane V	SSO, GTO	120	75.95	59.94
Atlas II	LEO, GTO	?	?	?
Delta II	LEO, SSO, GTO	?	77.98	33.02
Delta IV	LEO, SSO, GTO	?	?	?
Titan IVB	LEO, GTO	?	?	?
Space Shuttle	LEO	68	65.02	48.26

Table 3.1: Potential Launch Vehicles
(see Appendix for expanded list)

Based on current launch vehicles providing for secondary payload accommodations the following maximum overall dimensions for the satellite have been determined:

Maximum Satellite Mass - 68 kg
Maximum Satellite Width - 33 cm
Maximum Satellite Height - 63 cm

These requirements allow the satellite to be carried aboard the Ariane 5, Delta II, and the Space Shuttle (Hitchhiker Program). The launch vehicles being investigated have established methods for launching secondary payloads. For design purposes, the maximum target satellite mass shall be 40kg. This will allow the spacecraft size to grow during the design process, if necessary, and still meet the overall target mass of 68kg.

Currently, the Ariane 5 shall be baselined as the launch vehicle. This will allow the design of the satellite to proceed with specific launch vehicle integration parameters. It does not, however, completely rule out the possibility of launching on other vehicles should a slot become available.

4.0 Trajectory Design

The satellite orbit shall be in LEO. The exact trajectory shall be determined by the launch vehicle chosen. Favorable characteristics of the satellite's trajectory are; eccentricity no greater than a TBD value, a sufficiently high inclination (ground track should pass over Purdue Ground Station), and attitude control (discussed in detail in the Attitude Control section). A sun synchronous orbit shall be favorable for both communication and power needs (sun exposure time). Some of the launch vehicles previously mentioned do accommodate sun synchronous launches. As with baselining the Ariane 5 as the launch vehicle, a sun synchronous orbit shall be specified to all satellite design to proceed with specific requirements. Adjustments could then possible be made to allow for a different LEO.

5.0 On-board Propulsion

The satellite shall require no velocity changes. Requirements on de-orbiting capabilities shall depend on government requirements and trajectory orbit altitude.

6.0 Attitude Control

Attitude control devices requiring minimal cost and complexity shall be considered for the satellite. The payload and communications require 2-axis stabilization to within TBD acceleration limits (spin coupled with oscillations for sloshing concerns). A pointing accuracy (about nadir) of a TBD amount (anticipated between 0.1° and 5°) is required to meet the payload and telecommunications requirements. Spin around the nadir axis must also be kept below TBD limits based on payload requirements

A gravity gradient control design shall be used. Once the satellite has separated from the launch vehicle it shall assume a predetermined orientation. The use of magnetic torquers or momentum wheels to initially orient the vehicle and/or control oscillations incurred by a gravity gradient design shall be investigated. Simple deployable mechanisms for gravity gradient boom shall be used.

7.0 Thermal Control

The thermal control system (TCS) shall maintain all the components of the satellite within allowable temperature limits and operating limits. See Appendix for Table 7.1 of historical temperature ranges for some satellite components.

8.0 Power

The power system shall provide a TBD amount of energy to support all satellite subsystems. For example, the SPT science to be conducted as well as communication and data transfer with ground stations will require power. Batteries shall be capable of performing for the duration of a single experiment run without exceeding the acceptable TBD depth of discharge percentage. Solar power shall be used for recharging the batteries and maintaining required power levels during battery charging. There shall be no limit for battery recharge time. The duration of the mission (< 1 year) is anticipated to be such that degradation of the solar panels will not be a major consideration for End-of-Life power. Battery usage characteristics shall be such as to allow power requirements to be met throughout the mission duration.

9.0 Command and Data Handling

The command and data capabilities shall consist of a continuous satellite health log and data acquisition from the experiment. This TBD amount of data shall be stored onboard the satellite. The satellite shall have transmission capabilities for both information logs. The satellite onboard health log will contribute to the determination of experiment run times. All data shall be transmitted to a TBD ground station as raw data to be analyzed by the scientists. The TBD electronic components, and software shall be low cost off the shelf technology. These electronics may not be radiation hardened and this issue shall be investigated.

The antenna used shall be either omni directional. Attitude control, ground station requirements, and data rates shall determine the exact specifications of the antenna. The satellite shall be half-duplex, that is it will be able to transmit and receive over TBD frequencies, at a TBD rate. The satellite shall receive signals primarily for power-up and shut down and experiment commencing and ceasing. Between these segments of the mission, the satellite shall be autonomous. Information on the satellite’s health shall be periodically transmitted to ensure the attitude characteristics are as required.

10.0 Telecommunications

The telecommunications subsystem shall be capable of transmitting the desired information. The current modulation being analyzed is GMSK (Gaussian Minimum Shift Keying), as it is being used in AMSAT (AMateur SATellite). This “packet technology” has been used in earlier missions. Other techniques, such as QPSK (Quadri-Phase Shift Keying) and BPSK (Binary Phase Shift Keying) are being analyzed. A convenient and achievable data rate is still being investigated; however the team has noticed that a 9.6 kbps transmission rate has historically been typical. The mass of the transmitter and receiver has historically been 10% of the total mass. The frequencies shall be of approximately 437 MHz for the downlink and 145 MHz for the uplink. A bit error rate of 1e-5 shall be considered acceptable.

11.0 Structures

The structure shall support the SPT payload and all subsequent subsystems. The support structure shall carry the primary TBD loads (including thermal expansion) incurred throughout the satellite’s life within a TBD factor of safety. The structure of the satellite bus shall also securely accommodate the internal components, i.e. payload and electronics. The structure shall contribute to the TBD necessary inertia properties for attitude control. The possibilities of deployable mechanisms for gravity gradient or power needs (antenna, solar arrays) may affect structural design. Electrical charge buildup issues shall be addressed with the materials chosen and methods of discharge investigated.

12.0 Risk

Risk shall be incurred in several aspect of the design. TBD factors of safety and TBD mass and volume budget limits shall potentially reduce the risk of a design overrun or rework due to potential mission hazards.

Appendix

Additional Experimentation Ideas

Thermocapillary Baseline Experiment

Thermocapillary dynamics is a poorly understood phenomena. The effects of thermocapillary dynamics on satellite propellant gauging techniques are an even more complicated problem. This experiment looks to understand this effect better by allowing the fluid to freeze completely solid inside of the spacecraft tank, then a complete heating cycle is performed to bring the frozen fluid back to the liquid state. Measuring the differences between the temperature profile of the primary experiment and this frozen experiment the thermocapillary effect can be conferred found.

Cyclical Heating Experiment

A major constraint on many space missions is power requirements. One possible way to reduce the required power may be to keep propellant tanks at a constant temperature. A modulated heater cycle is one proposed method for reducing this power. Current methods use drastic heating compensation to deal with temperature changes of the environment. By operating at a specific frequency it is believed that the power required to heat propellant tanks can be reduced.

P-V-T gauging Experiment

A current method used to determine propellant volume is based on the tank pressure. The pressure, when coupled with the temperature of the tank, can provide the volume of the gas in the tank. Knowledge of the gas volume provides knowledge of the propellant volume. Pressure gauging involves a more complex system than is currently being proposed. However, if small satellite experimental methods for gauging accurate tank volumes is proven feasible, this mission might serve as a platform for future implementation of this type of experiment.

Table 3.1 Expanded to include other relative launch vehicle information.

Launch Vehicle	Manufacturer	Auxiliary Payload Launch Program	Available Inclination (deg)	Performance Capability	Max. Payload Mass (kg)	Max. Payload Width (in)	Max. Payload Height (in)	Comments
Ariane IV	Arianespace (Europe)	Ariane Structure for Auxiliary Payloads (ASAP)	5.2-100.5	LEO, SSO, GTO	50	17.7	17.7	Established program to launch small satellites as secondary payloads - Majority of launches are SSO - Being replaced by Ariane 5
Ariane V	Arianespace (Europe)	Ariane Structure for Auxiliary Payloads 5 (ASAP5)	5.2-100.5	SSO, GTO	120	23.6	29.9	Follow-on program established by Ariane 4 - Increased capabilities over Ariane 4 ASAP - Minimal launch history
Atlas II	Lockheed Martin (US)	Unknown	28.5-55 (CCAS), 63-120 (VAB)	LEO, GTO	?	?	?	No established Auxiliary Payload program, however small satellites are occasionally flown - Minimal launches to LEO
Delta II	Boeing (US)	Unknown	28.5-60 (CCAS), 63-145 (VAB)	LEO, SSO, GTO	?	13	30.7	Provisions exist to attach auxiliary payloads to the sides of the second-stage avionics section - Majority of launches are to LEO
Delta IV	Boeing (US)	Unknown	28.5-51 (CCAS), 63-120 (VAB)	LEO, SSO, GTO, GO	?	?	?	Standardized Auxiliary Payload Program will be established, but status is currently unknown
Titan IVB	Lockheed Martin (US)	Unknown	28.5-55 (CCAS), 63.4-113 (VAB)	LEO, GTO, GO	?	?	?	One auxiliary payload has been carried
Space Shuttle	U.S.A. - NASA (US)	Hitchhiker	28.5-57	LEO	68	19	25.6	Part of a larger NASA program to carry small, non-commercial payloads to LEO - Hitchhiker provides capabilities to eject payload

Table 7.1 Historical Temperature Data for Satellite Components

Component	Lower Temperature Limit, °C	Upper Temperature Limit, °C
Antenna (TT&C)	-65	+95
Batteries (NiCd)	0	+25
Batteries (NiH_2)	-5	+20
[2]Electronics w/out Operating Operating Temp.	 -65 0	 +150 +60
Multilayer Insulation (MLI)	-160	+250
Payload ([3]SPT)	+7	+55
Power Control Unit	-20	+55
Solar Cells (Arrays)	-105	+110
Structures (alignment critical)	+18	+22
Worse Case Envelope	0	+40

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[2] Storage Temperature is constrained from 0-100°C. Ref. Microchip.com.

[3] The SPT temperature ranges shall reflect the temperature ranges of Hydrazine. The experimentation conducted will be of greater purpose if the operating limitations of the fluid are the same. A mixture of H_2O and antifreeze is the anticipated fluid inside the SPT, which is also feasible for this temperature range.