ABSTRACT

The 1990s initiated the era of “Faster, Better, Cheaper” for small spacecraft, and met with mixed success. However, even under the best of circumstances, “Faster, Better, Cheaper” faces the hurdle of high launch costs. In programs with typical launch costs of $20 - 50M or more, spacecraft capabilities are pushed to the limit in order to extract maximum value from the overall system. This drive for high performance results in complex systems, which require significant engineering to address relatively minor changes. However, new launch vehicles offer the potential to reduce launch costs to under $10M and will change the economics of small spacecraft design and production. The US Army Space and Missile Defense Command is developing a series of spacecraft and payloads with modest performance and dramatically lower cost for use as a technology testbed. The program uses a capabilities-driven approach rather than requirements-driven approach, which enables significant cost savings. The first spacecraft and payload will support the Missile Defense Agency airborne testbed by providing an exo-atmospheric IR calibration source. The bus is designed for a 50 kg class satellite with a recurring cost target of $1M. A future system will be used as a testbed for advanced sensors and will target a 150 kg class satellite with a recurring cost of $3M. The concept design for each satellite class is discussed, including anticipated capabilities.

INTRODUCTION

A new generation of low-cost, small launch vehicles is emerging, such as the SpaceX Falcon 1, Kistler Rocketplane, and Microcosm Sprite. With these vehicles, the cost for launch of small spacecraft may be reduced from $20M to $5-10M. This change, in turn, will drive the economics of spacecraft acquisition for certain missions. With the reduction in the cost of launch, no longer are users necessarily driven to extract maximum performance from their spacecraft. Obtaining optimum performance requires significantly increased resources. Pareto’s “80/20 law” reflects the common experience that achieving the final 20% of performance requires 80% of the effort. While these numbers are exemplary, significant savings can be achieved by accepting slightly lower performance. This mindset requires a revolutionary paradigm shift from the traditional approach to spacecraft development. This approach is embodied in a program Digital Fusion Solutions (DFS) has dubbed “Low-cost Innovative Microsatellite Access To Space (LIMATS).”

The need to capitalize on these changes is particularly strong in the R&D arena. Space acquisition programs have stretched from 5 years up to 15 or more years, often driven by immature technology adopted early in the program before it had sufficiently matured. In other cases, new technology (for example, solar cells or field programmable gate arrays) was adopted by a program office but did not perform in the space environment as expected. Schedules and budgets do not allow for 3-4 years and $40-50M for a space-based technology demonstration within the scope of a major acquisition. Program managers are often (wisely) unwilling to accept the risk of technology that has not been demonstrated in the space environment. Thus the hurdle from TRL 4-5 to TRL 7 is extremely difficult to overcome, sometimes referred to as the “Technology Valley of Death.” By reducing the cost and time required to conduct a space mission, LIMATS enables the transition of various technologies to acquisition programs.

Digital Fusion Solutions, is performing a related task in support of the US Army Space and Missile Defense Command (SMDC) for a payload for a relatively low-performance microsatellite. This 50-kg, 12-W microsatellite will be gravity-gradient stabilized, with pointing control within 5° and pointing knowledge within 1°. However, for most remote sensing missions, this performance is inadequate. Therefore, we are planning development of a second class of satellite, which will be a 150-kg, 3-axis stabilized platform with arc-minute class pointing.

APPROACH

There are several key approaches within LIMATS that break with traditional satellite development. These
distinctions will enable Digital Fusion Solutions to dramatically lower the cost of developing a spacecraft, and thus reduce the overall cost of the space mission. These factors include: limited duration mission, use of commercial-off-the-shelf (COTS) hardware wherever suitable, use of space heritage hardware where necessary, capability-driven rather than requirements driven design, careful use of advanced technology components, and adoption of simplified interfaces.

**Limited duration mission**

While some technology demonstrations require an extended period in orbit, many do not. However, when the mission cost is driven by high launch costs, often several payloads are combined to reduce the burden on each experiment. This approach creates a requirement for an extended duration mission because payloads have to share operations time and power. The duration requirement flows down to requirements on reliability, redundancy, radiation hardness, and so on. Furthermore, integration of multiple payloads results in a much more complex system with greater integration challenges. Moreover, incorporating multiple payloads increases the schedule risk, which directly contributes to increased cost risk.

**Use of COTS hardware wherever suitable**

High-reliability programs require high-reliability parts and design. For operational missions, this approach is appropriate and required. The radiation, thermal, and vacuum environments of space are tortuous conditions for many parts. However, for limited duration missions that can accept slightly higher risks, a dramatic reduction in cost and complexity can be achieved. Over the last decade, a variety of COTS components have been used successfully by university (and some government) programs, which were willing to accept higher risk to reduce cost. On the other hand, there may be certain applications which are either not amenable to COTS or deemed so critical to the mission that the risk associated with COTS is unacceptable. In these applications, adoption of non-development items with space heritage is appropriate. This approach is only suitable for low earth orbits, below 800-1000 km, where the radiation environment is not too severe.

**Capability driven design**

Following the traditional systems engineering process, such as described in Space Mission Analysis and Design, the designer begins with mission definition and flows down requirements to the payload, then to bus, the ground station, and so on. As the design evolves, the cycle is repeated numerous times. In this process, the mission drives the design of the entire system, and results in a system that is highly optimized for the particular mission. The corollary is that every system is unique and a significant portion of the program cost goes into non-recurring engineering. When high performance and high reliability are required, this process works very well. However, modest changes can result in significant re-engineering because of the degree of optimization. By contrast, LIMATS proposes to follow a capability-based design. Digital Fusion Solutions has been reviewing a number of sensor systems and the requirements they impose on the satellite bus in terms of power, thermal management, pointing knowledge and control, vibration, etc. By designing a system that meets the requirements for a wide variety of payloads, re-engineering is kept to a minimum. Furthermore, by including significant margin in critical areas, unanticipated excursions can be managed. If the bus cannot meet a particular requirement (such as average power), then the concept of operations (CONOPS) can be adjusted (such as by reducing duty cycle). Certainly an operational spacecraft may not be able to meet mission requirements in this manner, but R&D missions with more relaxed requirements can achieve significant cost savings with this approach.

**Judicious use of advanced technology components**

Often advanced technology components can provide significant performance gains, but at significant expense. Composite materials are an excellent example. Generally lighter and stronger than metals, composites require careful attention to modifications to ensure the design is not compromised from even minor modifications such as moving reinforcements or bolt holes. Thus, the flexibility of the design is limited. Some advanced technology components do not limit flexibility of design and provide valuable savings in weight, but each must be carefully weighed with an eye toward maintaining flexibility for diverse applications. Only in cases that are justified through “order of magnitude” level improvements will we include more complex and less flexible approaches in the trade space.

**Simplified interfaces**

Carefully optimized spacecraft will remove all undesired redundancies within the system. By undesired redundacy, we mean those functions performed by two or more systems, but not with the intention to improve performance or improve reliability. As an example, the payload and bus may both have a data processing and storage capability. The bus must have its own system to perform its needed functions, and by sizing the subsystem with additional margin, the bus may be able to handle processing for the payload as well. For some missions, that margin will not be sufficient and the payload will have to have
its own processing subsystem. In that manner, the processor for the bus does not have to handle the widely diverse processing requirements of various payload options. In traditional satellite design, the unwanted duplication represents a waste of precious mass and power and would be eliminated. In this model, the opportunity cost of the excess power and mass are made up by the decrease in design and manufacturing cost. As another example, the LIMATS bus will provide a limited thermal management capability; beyond that capability the payload will have to handle its own thermal management through a combination of hardware and CONOPS. This approach will greatly simplify the design of the bus and allow for much greater reuse.

SYSTEM CONCEPTS
LIMATS currently includes two classes of satellites: a smaller, less capable system with a total mass of approximately 50 kg and a larger, three-axis stabilized satellite of approximately 150 kg. These design points were chosen based on the capabilities they provide, but are considered to be descriptive of the class of satellite and not system specifications.

50-kg class
This satellite bus, shown in Figure 1, is being designed to fulfill the need for missions requiring relatively coarse attitude control and low average power. Examples of test payloads include batteries, solar cells, processors, and similar component technologies. The payload will attach to the bottom of the bus in the view shown, and will generally be nadir-pointing. Bus dimensions are approximately 0.5 m x 0.5 m x 0.2 m, exclusive of the deployed solar panels. Component layout is primarily to demonstrate feasibility, and does not represent final placement for center of gravity and moment of inertia location.

Concept designs for several subsystems have been examined for feasibility. Table 1 lists the subsystems and respective mass and daily average power allocations.

**Attitude.** The satellite is passively stabilized using a gravity gradient boom. Although magnetic hysteresis rods could be used to provide damping of the initial motion, torque rods will be used instead to rapidly dampen any undesired rotation due to deployment tip off. Simulation has shown small rods with magnetic moments of 5 A-m² (e.g., the ZARM/Microcosm MT-52) can reduce the angular speed by >99% (from 4°/s rotation) in less than 20 minutes using a –ıβ control law. The gravity gradient boom can then be extended to provide passive roll-pitch stabilization. In the yaw direction, a small reaction wheel is used to keep the solar panels pointed sunward, to the extent allowed by the gravity gradient boom. By combining passive and active stabilization, in the event of a temporary loss of attitude control (e.g., due to processor reset from an SEU/SEL), the satellite will not deviate significantly from its desired attitude nor end up in an unrecoverable attitude. While the gravity boom provides relatively coarse attitude control, attitude knowledge will be better than 1° through the use of a combination of 2-axis digital sun sensors and a three-axis magnetometer. Finally, a Global Positioning System sensor will provide orbital position and velocity data.

**Electrical Power.** Power generation is the most constraining subsystem in this particular design. Because of the passive stabilization, the solar panels are often in a less than optimal geometry for collecting sunlight. Preliminary estimates indicate that multi-junction solar panels will provide about 12 W orbital average power, depending on the choice of orbit. The solar cells must be mounted on a supporting panel, with a combined mass of approximately 2 kg. In order to meet the power budget, the duty cycle for the payload and communications systems may have to be adjusted.
Lithium-ion batteries will provide 15 A-hr energy storage for the 28-V electrical bus. Although 420 W-hr is larger than one might expect for a microsatellite, the design is being driven by two factors. First, the initial payload is an active infrared calibration source to support airborne IR sensor tests. The system needs to be able to radiate during its pass over the test site for two sequential passes and downlink data following the second pass. This energy needed was one input for sizing the energy storage system. Second, our goal is to use a common energy storage system between the two satellites in order to minimize the non-recurring engineering (NRE) costs. In order to support a variety of R&D tests, we want to ensure instantaneous power is not a constraint, even when average power is a constraint.

**Table 1: 50-kg Satellite Subsystem Allocations**

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Mass (kg)</th>
<th>Power (W)</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Payload</td>
<td>15.0</td>
<td>2.7</td>
<td></td>
</tr>
<tr>
<td>Attitude</td>
<td>10.0</td>
<td>1.6</td>
<td>Passively stabilized, rotation about yaw axis</td>
</tr>
<tr>
<td>Power</td>
<td>9.7</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Communication</td>
<td>1.1</td>
<td>2.3</td>
<td></td>
</tr>
<tr>
<td>Processing</td>
<td>5.0</td>
<td>7.0</td>
<td>Key driver in power budget</td>
</tr>
<tr>
<td>Thermal</td>
<td>0.4</td>
<td>0</td>
<td>Passive control feasible</td>
</tr>
<tr>
<td>Structure</td>
<td>10.1</td>
<td>0</td>
<td></td>
</tr>
<tr>
<td>Propulsion</td>
<td>0</td>
<td>0</td>
<td>None in this system</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>51.3</strong></td>
<td><strong>13.6</strong></td>
<td></td>
</tr>
</tbody>
</table>

**Communication.** The communication system is based on a store-and-forward model, operating in the S-band. The system is sized so that only one uplink/downlink is required per day, thereby reducing operations cost. The maximum data rate needed is anticipated to be 256 kbps, which can be achieved with a 5-W transmitter, such as the AeroAstro S-band radio. Moreover, a straightforward change will enable downlink speeds of up to 10 Mbps. Since the primary power requirement for the communications system is the transmitter (40 W while transmitting), a significant power savings can be achieved by reducing the amount of time spent communicating during seasons of low solar illumination. Although this may reduce the amount of time available to the mission, this is a good example of how modifying the CONOPS can yield valuable cost savings. One option currently under consideration is use of the NASA Tracking and Data Relay Satellite System (TDRSS) Multiple Access mode for frequent monitoring of the state health of the satellite during initialization. An output power of 5 W may provide a marginal link for data rates below 1000 bps and will require further study. While a TDRSS transceiver would be prohibitively expensive for this application in the past, Wallops Flight Facility (WFF) is developing the “Low Cost TDRSS Transceiver (LCT2)” with an anticipated unit cost of $50k. If the LCT2 can be produced for this cost, it will open the potential for microsatellite systems to affordably incorporate space relay services.

**Processing.** Two approaches are being considered for the processor. Because the processor is critical to mission success, a space qualified processor is currently used in the baseline design. A processor such as the Space Micro Proton 100k, in combination with an input/output board, will adequately handle the command and data handling requirements of the microsatellite in a compact package. However, a number of programs have successfully used modified COTS processors instead of processors designed for space. The unique nature of the mission opens the possibility for use of commercial parts. Important constraints include emphasis on LEO orbits and limited mission life, which both reduce the impact of the radiation environment. Finally, the research nature of the mission reduces the requirement for the system to be operable at all times. Thus the system could be shut down for short periods of time, such as when traversing the South Atlantic Anomaly or during magnetic storms. Therefore, we are exploring an alternative, parallel path of using a COTS processor that is modified to survive the space environment. At a minimum, required modifications include replacing of electrolytic capacitors with space-qualified components, soldering of components rather using chip sockets, adding potting and staking, and adding watchdog timer circuits to mitigate latch-up events. The processor may also require modification for thermal management; use of a sealed pressure vessel is also under consideration to meet this need. A sealed vessel has the advantage of simplicity as well as providing a certain amount of shielding against charged particle ionizing radiation. Obvious disadvantages include extra weight to withstand the internal pressure and risk of leakage that could jeopardize the mission.

**Thermal.** Our initial analysis indicates that passive thermal control of the satellite is feasible. Control is achieved through the use of multi-layer insulation, thermal coatings, and radiator panels. The satellite structure provides the thermal paths between the heat sources and the radiators. Figure 2 shows results of preliminary thermal modeling. The first figure shows the temperature profile of the satellite in thermal equilibrium under normal solar illumination. This is beyond the worst case because the same face is not exposed to the sun for long enough to reach equilibrium, save in the unlikely sun-synchronous,
End of the eclipse period, the internal components are well within their operating ranges. Significant analysis remains to be done, but these preliminary results are promising.

Figure 2. Thermal Analysis. (a) Equilibrium Case in Sun. (b) Transient Case at the End of Eclipse.

Structural. Simplicity is the guiding word behind the structural design. The side panels are designed to be interchangeable, while the upper and lower panels are as very similar as possible. The truss is designed with significant margin, so that minor modifications will not stress the structure. While the fine iso-grid truss used commonly in small satellites could save some mass, the truss structure used here provides robustness and some flexibility in component placement. Similarly, some weight savings could be achieved through the use of composite materials, but would result in a design that is very difficult to modify. The release mechanism has been allocated to the structure subsystem, and Planetary Systems 15" Lightband is the leading candidate for the baseline system. This highly standardized component makes interface to the launch vehicle very simple and cost effective. Furthermore, the motorized release generates much lower shock than a pyrotechnic device. The simplicity of resetting the system also results in time (and cost) savings during testing. The system is being designed with four launch vehicles in mind: the Evolved Expendable Launch Vehicle Secondary Payload Adapter (ESPA) ring, the WFF Multiple Payload Ejector (MPE), the SpaceX Falcon I, and the Sandia National Laboratory-designed Super Strypi. Designing the satellite to be compatible with both cantilevered (MPE and ESPA) and vertically supported systems has been very challenging. To date,
only static loads have been considered; no dynamics analysis has been completed. Static analysis has been done to support feasibility studies, and a rigorous design analysis has not yet been conducted. Therefore, the design is currently using a factor of safety of four below the yield stress of the materials. The initial feasibility results have been promising. For example, Figure 3 shows the vertical deflection due to the static load from the maximum acceleration of the Super Strypi, which is 12g.

Figure 3. Static Analysis of Bus Under 12g Launch Load.

In this case, the centrifugal acceleration due to the spinning vehicle has not been included. The maximum deflection, shown by the dark red area, is about 400 microns downward. The corresponding stress is well within the tolerance of aerospace grade aluminum alloy (nominally 7075-T6). The minimum deflection (dark blue) is zero, since the separation ring is fixed by the boundary conditions in the model. Clearly the load distribution needs to be adjusted to balance the deflection of the sides of bus but the bus is able to withstand the loads in this particular situation.

**Propulsion.** The decision was made to exclude a propulsion system from this satellite. In order to maintain cost in the desired range, the only propulsion options available are cold gas (e.g., N₂) and resistojet (e.g., butane or ammonia). These systems would require too much mass and volume to keep the system in the 50-kg class. By not including a propulsion system, the available altitudes are limited. The minimum ballistic coefficient, with solar panels and gravity boom normal to the orbit velocity, is about 40 kg/m², while the maximum, with the panels parallel to the velocity, is about 60 kg/m². Therefore, the minimum altitude to maintain orbit for at least 1 year during solar maximum is approximately 500 km and during solar minimum is approximately 375 km. The upper altitude limit will be based on the time required for atmospheric drag to deorbit the satellite, and should comply with 25-year deorbit guidelines. Because the orbital decay rate can vary by over an order of magnitude between the solar maximum and the solar minimum, the lifetime is much more difficult to predict. Initial analysis suggests the maximum altitude will have to be about 600-650 km.

**150-kg class**

The larger satellite reuses as many components and designs from the smaller satellite as reasonably possible. The model shown in Figure 4 bears a strong resemblance to the 50-kg satellite in Figure 1. The satellite is made up of three bays, each of which uses the same basic structure of the smaller satellite.

Figure 4. Overview of 150-kg Satellite
The top bay is the avionics bay, which houses the ADCS, power, and processing systems. The middle bay is the propulsion bay, housing the propellant tanks, thruster, and associated plumbing. The bottom bay is for the payload, and in this case holds a generic box of representative mass and thermal load.

Table 2: 150-kg Satellite Subsystem Allocations

<table>
<thead>
<tr>
<th>Subsystem</th>
<th>Mass (kg)</th>
<th>Power (W)</th>
<th>Comments</th>
</tr>
</thead>
<tbody>
<tr>
<td>Payload</td>
<td>35.0</td>
<td>20.0</td>
<td>3-axis stable</td>
</tr>
<tr>
<td>Attitude</td>
<td>12.3</td>
<td>8.5</td>
<td>Unchanged</td>
</tr>
<tr>
<td>Power</td>
<td>17.2</td>
<td>--</td>
<td>Heat pipes, heaters</td>
</tr>
<tr>
<td>Communication</td>
<td>1.1</td>
<td>2.3</td>
<td>Unchanged</td>
</tr>
<tr>
<td>Processing</td>
<td>5.0</td>
<td>7.0</td>
<td>Butane resistojet (tentative)</td>
</tr>
<tr>
<td>Thermal</td>
<td>1.0</td>
<td>5</td>
<td></td>
</tr>
<tr>
<td>Structure</td>
<td>42.3</td>
<td>0</td>
<td></td>
</tr>
<tr>
<td>Propulsion</td>
<td>28.3</td>
<td>0.3</td>
<td></td>
</tr>
<tr>
<td>Total</td>
<td>142.2</td>
<td>43.1</td>
<td></td>
</tr>
</tbody>
</table>

**Attitude.** The larger satellite will be three-axis stabilized, since we envision it will be used for testing sensors or components that require a pointing capability. The primary control actuators will be a triad of small reaction wheels, comparable to the Rockwell Collins RSI 02 wheel. Because of the difficulty in designing a wheel that is highly reliable, generates minimal disturbance, and can survive launch, we are planning to adopt a wheel which has a demonstrated space heritage. In addition, torque rods will be used to null out initial tip-off disturbances and provide a means for dumping angular momentum to prevent saturation of the wheels. Attitude sensing will be accomplished using a star tracker, an inertial measurement unit (IMU), sun sensors, and a magnetometer. Recent advances in star trackers have made this technology viable for microsatellites at a reasonable cost. Available products have their own particular strengths and weaknesses, and analysis will be left for future trade studies. The IMU will provide high frequency updates during precision maneuvers, providing rotation rate accuracy of 6 µrad/s (0.00032/°s) or better. The sun sensor will be used to maintain peak power generation while not conducting payload operations and to serve as a backup for the star tracker. Likewise, the magnetometer is used in conjunction with the torque rods and as a backup for the star sensor. These components are anticipated to provide at least arc-minute pointing knowledge and 0.1° pointing control.

**Power.** The power system will be based closely upon the system used for the smaller satellite. The same battery will be used to provide energy storage. However, much greater power generation capability will be available because the satellite will be three-axis stabilized and can point sunward whenever not conducting payload operations. The combination of increased size and increased efficiency quadruples the power generation to 45 W orbital average and increases peak power to 90 W.

**Communication.** The only difference in the communication system is that the higher bandwidth option of 10 Mbps will be the baseline configuration.

**Processing.** The same processor will be used on both satellites.

**Thermal.** In Figure 5, the results of the preliminary thermal analysis are shown, using a somewhat different approach than used previously. Rather than computing the temperature based on thermal equilibrium, a new thermal model was available (I-deas Thermal Analysis TMG). This model accounted for the thermal load of Earth IR and albedo throughout an orbit, in addition to the solar flux. The cooler image shows the results of modeling the spacecraft based on launch at 27 °C and then passing immediately into eclipse. In this case, the spacecraft solar panels are pointing sunward, even during eclipse. The results are quite similar to the 50-kg results, lending validation to the approximation used earlier. The most significant difference between the 50-kg results and these results is due to the presence of the Earth IR, which presents a small but valuable heat flux. Less than 20% of the solar flux, it does not add significantly to the heating in the presence of the sun. However, it helps to moderate the radiative cooling of the solar panels during eclipse. The other significant difference is the temperature of the coolest location. Because the satellite does not reach thermal equilibrium in the sun, the radiators are somewhat cooler at the end of one orbit, which will further moderate the temperatures experienced by the components. In both eclipse and illumination, all the components remain within their operational temperatures using only the conduction of the aluminum structure. As time allows, we will repeat the analysis of the 50-kg satellite using I-deas TMG.

**Structural.** Structural design of this satellite was much more challenging. Several iterations were required to obtain a structure that could withstand the varied launch environments, even considering only static loads. The cantilever configuration required for ESPA launch was particularly challenging.
because of the longer body on this satellite. The Von Mises stress results under 12g static load are shown in Figure 6. (The Von Mises stress represents roughly the magnitude of the stress tensor.)

Figure 5. Thermal analysis results for the 150 kg bus. (a) Transient results following passage through eclipse. (b) Transient results based on completion of orbit following eclipse.

Figure 6. Von Mises stress pattern for the 150-kg design under 12g load (scale is 0 to 111 MPa)
Because the body shape is not matched to the shape of the release mechanism, line loading increases stress in the area indicated. Nevertheless, by adding reinforcing material, the stress was limited to an acceptable level. The cantilevered orientation also places severe constraints on the Lightband separation system, particularly the bending moment. The Lightband is rated to support 15 kN-m bending moment, and the static load from ESPA (10.5g) could be as high as 4.6 kN-m. Since this exceeds the factor of safety of 4 previously established, we examined the possibility of using the 18” Lightband. By enlarging the bus by 0.5”, it could accommodate the 18” Lightband, which can support a 20 kN-m moment. This is an area for further careful study. Because this satellite is composed of vertically stacked sections of the smaller satellite, analysis of rotation of the satellite during launch on the Super Strypi is very similar to analysis of the smaller satellite. A very important difference, however, is the potential for slosh in the propellant tanks, and is described below

**Propulsion.** Selection of a propulsion system has proven to be one of the more difficult design choices. The initial design goal was for the system to provide sufficient impulse to maintain a 300-km orbit for the mission duration of 1 year during the upcoming solar maximum. In order to do so, the system must provide as much as 400 m/s ∆v or 60 kN-s impulse. Our approach of using low-cost, simple components has pushed the design toward a proven cold-gas or butane resistojet design. However, the mass required to store sufficient pressurized N₂ was excessive and the cold-gas approach eliminated. Initially a propulsion system based on a butane resistojet design was the leading candidate. This design was favored for its simplicity and safety of the propellant, but to meet the initial goal would require almost 80 kg of propellant. Therefore, two options are currently under consideration: 1) Moving to a more efficient but complex propulsion system or 2) reducing the planned capability. The first option is currently under review and includes a low thrust catalyst-bed hydrazine thruster and a (relatively) low power ammonia arcjet. While these are not likely to meet the stated goal within the mass allocation, they will provide significantly greater capability at a somewhat higher cost. The second option entails increasing the minimum altitude of the satellite during solar max from 300 km to about 425 km. The actual minimum altitude will depend on the intensity of the solar max; this value is based on drag associated with the 90th percentile of the F10.7 solar flux value. Additional capability (lifetime) can be gained by including the orbital decay time as part of the mission life, not just the time that can be maintained at the initial altitude. Using this approach, the propulsion system as modeled here includes 5 kg butane propellant, which will provide approximately 4 kN-s impulse. At this point, the design still has considerable margin relative to the allocation in Table 2 and additional propellant may be added. Nonetheless, this modest impulse will allow the satellite to start at about 425 km and remain in orbit for one year. During periods of lower solar activity, the initial altitude can be somewhat reduced. This conservative estimate also assumes the ballistic coefficient is constant at its lowest value. This value can be significantly increased during detailed design when considering actual operations, and will result in a longer lifetime. The inclusion of the propellant system introduces other considerations. The roughly 20% ullage leaves a relatively small space for slosh. For the spinning Super Strypi, the center of mass of the propellant in each tank can be expected to shift by approximately 1 cm, resulting in a total change to the moment of inertia of ~0.02 kg-m². The overall moment of inertia about the vertical axis is 4.2 kg-m², so the effects of slosh can be expected to be minimal unless the slosh frequency is resonant with the spin rate. This will have to be confirmed through further analysis. However, propellant slosh will be an important effect for the attitude control system and its impact will have to be considered in detail.

**Costs**

As noted above, cost is a key driver in the design trades. By way of comparison, the Aerospace Corp Small Satellite Cost Model was used to estimate the cost using “traditional” small satellite techniques. The model is based on data from 53 programs, covering satellites from 113 – 877 kg. The mass allocation and capabilities described were used as inputs. The SSCM predicted the satellite cost for development and first unit would be $23M, consistent with our expectations for development using the traditional model. Recurring cost from the SSCM was $12M. However, our analysis to date indicates the total cost and recurring cost will be less than half the amounts predicted by the SSCM. Recurring materials cost for the subsystems as described here is about $2M; assembly, integration, and test for the second unit are estimated to be comparable.

**CONCLUSION**

The impending low-cost small launch vehicles open a window to change the economics associated with small satellites. Systems with modest capabilities and risk can be developed for significantly less cost than those using traditional high-reliability,
requirements-driven approach. Such systems are particularly well-suited for science and technology testbeds. We have conducted preliminary studies and demonstrated that a low-cost, risk tolerant microsatellite is feasible.

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REFERENCES


