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IODINE PROPULSION ADVANTAGES FOR LOW COST MISSION APPLICATIONS AND THE IODINE SATELLITE (ISAT) TECHNOLOGY DEMONSTRATION

John W. Dankanich
NASA MSFC, USA, john.dankanich@nasa.gov

Daniel M. Schumacher
NASA MSFC, USA, daniel.schumacher@nasa.gov

The NASA Marshall Space Flight Center Science and Technology Office is continuously exploring technology options to increase performance or reduce cost and risk to future NASA missions including science and exploration. Electric propulsion is a prevalent technology known to reduce mission costs by reduction in launch costs and spacecraft mass through increased post launch propulsion performance. The exploration of alternative propellants for electric propulsion continues to be of interest to the community. Iodine testing has demonstrated comparable performance to xenon. However, iodine has a higher storage density resulting in higher ΔV capability for volume constrained systems. Iodine's unique properties also allow for unpressurized storage yet sublimation with minimal power requirements to produce required gas flow rates. These characteristics make iodine an ideal propellant for secondary spacecraft. A range of mission have been evaluated with a focus on low-cost applications. Results highlight the potential for significant cost reduction over state of the art. Based on the potential, NASA has been developing the iodine Satellite for a near-term iodine Hall propulsion technology demonstration. Mission applications and progress of the iodine Satellite project are presented.

I. INTRODUCTION

The NASA Marshall Space Flight Center (MSFC) Science and Technology Office is continuously assessing technology options to enhance or enable future missions. Based on potential return on investment, technologies are recommended for NASA investment. Electric propulsion has already demonstrated potential to significantly reduce mission life cycle costs, such as Boeing's all-electric satellites[1] greatly reducing launch costs and NASA's Dawn mission leveraging electric propulsion to enable the first spacecraft to visit two distinct interplanetary destinations[2] through more than 12km/s of post launch ΔV. NASA also has interest in small satellites for enabling fast and affordable systems to meet NASA's needs as demonstrated through the successful flight of the Fast Affordable Science and Technology Satellite (FASTSat). Currently, small spacecraft have very limited on-board propulsion capability and therefore limited mission potential for higher value science orbits while maintaining the benefits of secondary launches.

Hall thrusters, operated on xenon, have significant flight heritage and excellent performance at moderate power levels (1-5kW). However, small satellites with severe volume constraints and safety restrictions have limited potential for high pressure xenon systems. These challenges are not unique to NASA, and alternatives have been explored.

Under the support of both the United Sates Air Force (USAF) and NASA Small Business Innovation Research (SBIR) program, Busek has been assessing high density propellant alternatives, assessed performance of the BHT-200 flight thruster operated on iodine and are maturing a 600W iodine Hall thruster system. NASA, both at MSFC and the Glenn Research Center (GRC) have performed mission studies to quantify mission advantages of the iodine system. Study results are in agreement with the opportunity for significant cost reductions to high priority NASA missions, and the potential to enable high manoeuvrability with small spacecraft. Based on the results and early investments, NASA MSFC and GRC have partnered with the USAF and Busek to perform an iodine Hall thruster technology demonstration mission through the iodine satellite (iSat) project.

II. HIGH LEVEL IODINE ADVANTAGES

Busek has been leading the exploration of alternative Hall thruster propellants.[3] Tests and analyses were completed for magnesium, zinc, iodine and bismuth as potential high density propellants. All of the options assessed were successful in demonstrating promise for increased ΔV per unit volume. However, system level considerations must be included for mission advantages or disadvantages. Storage density, storage pressure and power required to produce the required gas flow rates must be included in the mission assessment. The power required for vaporization limits the applicability of magnesium, zinc or bismuth for small satellite power limited applications. Propellant characteristics for selected alternatives are shown in table 1.

Table 1: State-of-the-art (SOA) Xenon and alternative propellant properties.

Propellant	Storage Density	Boiling Point, °C	Melting Point, °C	Vapor Pressure @ 20°C
Xe (SOA)	1.6 g/cm^3	-108.1 °C	-111.8 °C	Supercritical (>15MPa)
Iodine	4.9 g/cm^3	184.3 °C	113.7 °C	40 Pa (0.0004 atm)
Bismuth	9.8 g/cm^3	1,564 °C	271.4 °C	Solid
Magnesium	1.74 g/cm^3	1,091 °C	650 °C	Solid

For volume constrained and power limited spacecraft, the high storage density and vapor pressure make iodine a superior propellant alternative. Iodine stores as a high density diatomic solid, I_2, and sublimates at room temperature. Solid I_2 has a density of 4.9-kg/l. At typical storage conditions, e.g. 14-MPa and 50°C, the stored density of Xe is 1.6-kg/l. At the same conditions, Kr stores at 0.5-kg/l. The density of I_2 is compared to that of Xe, Kr and other electric rocket propellants in Figure 1.

In addition to the clear benefit of higher storage densities, is the advantage of lower storage pressure. While the unpressurized launch potential may in itself be enabling for secondary spacecraft, the low maximum operating pressures ~1psi, also has inherent advantages. Rather than advanced composite propellant tanks typically required for high pressure xenon, and inefficient packing spherical shapes, iodine enables the use of standard manufacturing techniques or can even enable the use of additive manufacturing of the propellant tank to nearly any desired shape for conformal placement and optimization of tank volume.

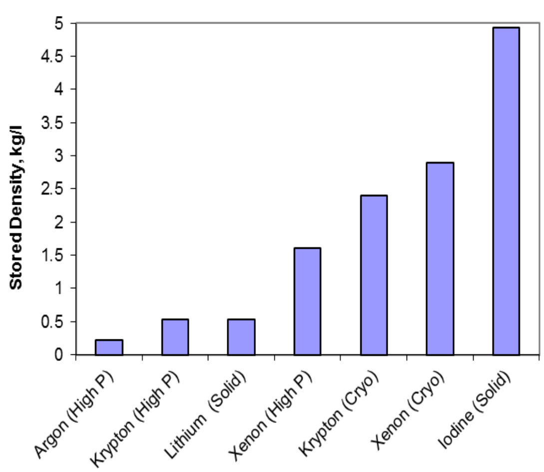

Figure 1: Stored Density of Electric Rocket Propellants in kg/l (high pressure at 14-MPa, 50°C).[3]

.II Iodine Hall System Development Plans

Iodine Hall system development remains prior to the implementation of flight systems. Busek is leading the development of the thruster, cathode and power processing unit (PPU) for iodine application. NASA MSFC has been leading the development of flight operation iodine feed systems and a comprehensive materials testing program for any iodine system regardless of the thruster type. NASA GRC has also been supporting cathode risk reduction activities, plume modelling and component level testing.

Under NASA's Space Technology Mission Directorate, the Game Changing Development (GCD) program is funding the maturation of a 600W system including the thruster, cathode and PPU. The 600W system is expected to reach maturity of Technology Readiness Level (TRL) 6 in 2016 with the exception of a life test. For the 200W system, developed under the iSat project discussed later, the Air Force SBIR commercialization program and NASA are qualifying and developing a flight system for launch in 2017. The SBIR program is providing additional risk reduction support for cathodes through Colorado State University (CSU) and Busek.

III. LOW COST MISSION APPLICATIONS

Given the potential for the iodine Hall technology, several mission studies have been completed to quantify full mission life cycle benefits and identify areas of concerns and/or interface challenges. In addition to iodine based missions for commercial, military and exploration applications, concepts have been developed for near-term mission demonstrations leading to the iodine satellite concept.[4] As expected, the benefits primarily result from the two key attributes of iodine, the increased propellant density and the low operating pressures. Primary concerns are limited to material compatibility, feed system deposition/clogging and potential science and/or system degradation due to exposure to the iodine plume.

III.I Microsatellites (Mass < 100kg)

The small satellite market continues to see strong growth. Some studies have predicted between 300 and 500 nano and microsatellite launches per year from now through 2020.[5] This potential market has led to growth in mission opportunities and in subsystem capability improvement. Small satellites have demonstrated rapid technology infusion opportunities typically limited by risk adversity on higher class missions. Cubesats are already considered and have been selected for science and exploration missions beyond technology demonstrations.

One of the critical limitations that still exists for small satellite exploration is the lack of primary propulsion. The challenges are due to limited resources available including power, mass and volume in addition to the funding and schedule requirements of standard propulsion options. Secondary spacecraft often have prohibitions for pressurized systems, hazardous propellants, stored energy, etc. Several cold and warm gas thrusters are mature and low-risk, but offer low specific impulse; limiting missions to 10s of m/s ΔV. Milli-Newton hydrazine thrusters have been developed and tested, but require special handling and may drive up costs and limit secondary payload options. Small solid motors are also options, but likely require additional liquid propulsion, which increases system complexity and introduces its own constraints for handling and secondary payload accommodations. Electric propulsion systems appear to offer potential for either precision attitude control or high ΔV capability. Pulsed plasma thrusters are the most mature, but vacuum arc thrusters and various alternatives are low risk to mature but only offer limited total impulse. Miniature ion propulsion has been explored and offers high ΔV potential, but requires high pressure gas. Microfluidic Electrospray Propulsion (MEP) technologies have great potential, but still pose development risk and lifetime limitations. Given secondary payload requirements and constrained spacecraft resources, there are limited options available for both low-risk near-term infusion and high ΔV primary propulsion for small spacecraft.

As the original motivation for exploring iodine, even excluding concept maturity and operational constraints, small spacecraft are still volume constrained; especially as the spacecraft is pushed down to the smaller form factors (i.e. 6U and 12U spacecraft). The critical figure of merit moves towards propellant ISP-Density to determine the ΔV capability of the spacecraft in a given volume. While figure 1 highlighted the storage density advantage of iodine over alternative electric propulsion options, figure 2 illustrates the ISP – Density of a range of propulsion options and resulting volume limited ΔV capability assuming the propellant is limited to a 1U volume allocation of a 6U spacecraft. Note that results in figure 2 do not account for propulsion system mass fraction or supporting subsystem requirements, but highlights the potential merit of iodine based propulsion due to the combined benefits of high specific impulse and high storage density.

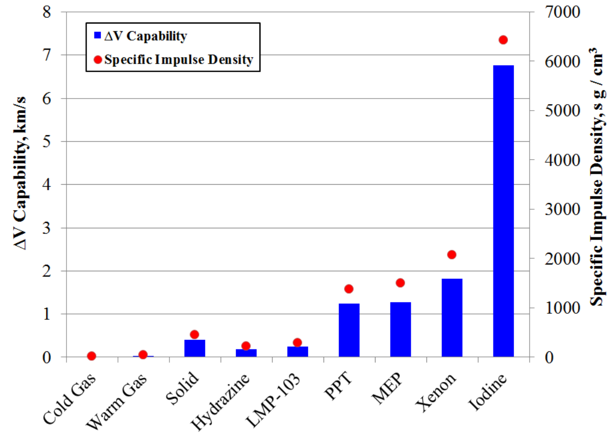

Figure 2: I_{SP}-Density and 1U ΔV capability for a 6U Spacecraft.

<u>Geocentric Applications</u>

As previously discussed, the small satellite market projections indicate on the order of 2000 spacecraft to be launched in the next 5 years. All markets including the Department of Defense, NASA and industry have significant interest in small satellite applications due to the rapid low cost deployment potential. Both military and civil space plans include deployment of large constellations of SmallSats in LEO for communciation and imaging capabilities.[6] The community continues to investigate responsive capabilities for SmallSats, either through a responsive launch capability or a responsive space deployed maneuverability capability. While the appeal of very low cost spacecraft is apparent, the cost of targeted orbits is prohibitive. The capability for high ΔV maneuvers on SmallSats may enable low cost constellation deployment, orbit tranfer to higher value orbits and provide station keeping for either formation flying or drag makeup. These capabilities enable geocentric applications including low-cost access to high value orbits, presistant coverage constellations, and required de-orbit capability.

Both 6U and 12U iodine spacecraft have been studied in great detail for LEO applications.[4] As an example, preliminary mass estimates for a 12U iodine satellite allows a 2kg payload capacity and 5kg of iodine within the standard mass limitations. At 250V discharge voltage, the 12U system would be capable of approximately 4km/s of ΔV using the BHT-200 operated on iodine. The ΔV capability can be used for more than 20,000km of altitude change or inclination change of 30 degrees in LEO or 80 degrees at GEO. A combination of propulsive manuevers and nodal regression and can easily accommodate a node change of a full 180 degrees for complete constellation deployment from a single launch. The consellations shown in figure 3 are achievable from deloyment of iodine vehicles into a single starting orbit. Additionally, scaling up to 50kg or 100kg class vehicle can allow for even greater ΔV to capture missions including ISS deployment plane change to either sun-snychrounous or equitorial orbit. The potential also exists for LEO to GEO transits if radiation exposure and transfer times are tolerable.

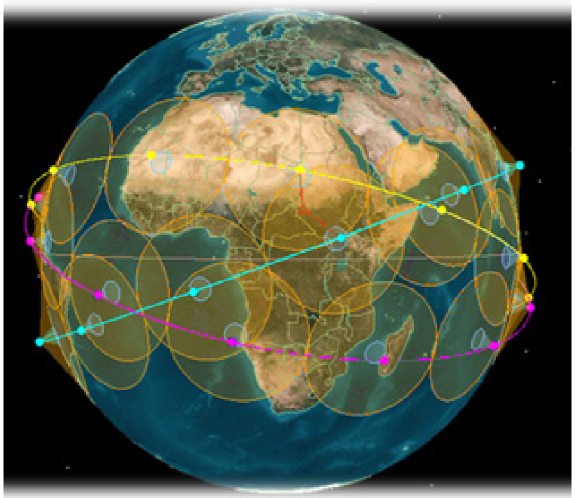

Figure 3: Example constellations achievable from a single starting orbit with iodine propelled SmallSats.

Interplanetary Applications

While far fewer opportunities, exist NASA has outlined planetary science and human exploration objectives to a diverse set of interplanetary targets. The community continues to stress the desire to increase the cadence of mission opportunities to accelerate the exploration of asteroids, comets, Mars, Mars' Moons, outer planets, etc. It is infeasible for NASA to meet its objectives with mission costs >>$100M each in addition to the high launch costs also >>$100M. NASA has recently announced the opportunity to leverage the first SLS Engineering Model flight (EM-1) with secondary payload opportunities. These opportunities are for 6U spacecraft launched into an escape trajectory; potentially eventually allowing for 12U volume systems. However, SLS secondary payloads value can be greatly increased with post-launch ΔV. The NEA Scout mission was selected under the Advanced Exploration Systems (AES) Program to develop a 6U solar sail spacecraft to enable an asteroid reconnaissance flyby mission. Like solar sails, iodine Hall systems have a niche to provide high ΔV in volume constrained spacecraft for asteroid flyby and orbiter mission. In fact, NASA announced the selection of a lunar orbiter from NASA Goddard Space Flight Center that will leverage iodine propulsion to transition from the EM-1 orbit into a stable lunar orbit.

With an intent to maximize science payoff, NASA MSFC has also assessed a 12U interplanetary iodine Satellite (iiSat) for interplanetary exploration at very low cost. A 12U iiSat with science optimized payloads was able to perform over 2.5km/s of post launch ΔV. This was sufficient not only to achieve a stable lunar orbit, but to rendezvous with a large number of near Earth asteroids, including the target 2000 SG344 in less than three years; of high interest as a potential crew rendezvous destination or an Asteroid Redirect mission.

III.II Small Spacecraft (Mass < 300kg)

NASA is interested in the potential to meet planetary science and exploration objectives at a cost that would enable the high cadence progress towards science goals and closure of strategic knowledge gaps. To that end, both the GRC COMPASS team and the MSFC Advanced Concepts Office assessed the viability of iodine based small spacecraft to deliver high value science at low cost. The goal of the studies was to determine the efficacy of broad applicability of the iodine Hall technology to enable a new class of lower cost missions while still meeting the threshold of Discovery class science.[7] The studies included Principal Investigator (PI) led mission concept designs, preliminary spacecraft design and life cycle cost estimates.

Near-Earth Asteroid Orbiter

While a 12U iiSat may be able to deliver a 2kg payload to a near Earth asteroid, Discovery class science often requires payloads on the order of the mass and volume of a 12U spacecraft. So within the capabilities of a 12U system, the science data is compromised. A Near-Earth Asteroid mission concept study was performed with the motivation to determine what cost reductions are possible through the use of the iodine technology without sacrificing science integrity relative to a Discovery class mission.

The GRC COMPASS team developed a concept that was able to leverage a common geosynchronous transfer orbit (GTO) deployment of the spacecraft for greater than $100M in launch costs over a dedicated interplanetary launch. An EELV Secondary Payload Adaptor (ESPA) Grande, can deploy spacecraft to GTO with mass allocations up to 300kg with a cost just under $10M including integration.[8] The COMPASS concept is shown in figure 4.

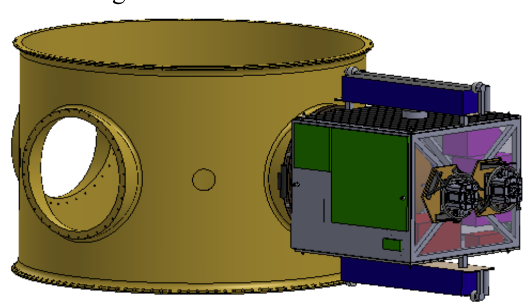

Figure 4: GRC COMPASS asteroid explorer concept vehicle stowed on the ESPA Grande.

The study required the use of two 600W BHT-600 iodine thrusters and a spacecraft power of 1500W at End of Life. The mission required a slow spiral from the starting GTO including seven months to escape. After escape, the spacecraft provides additional ΔV for the interplanetary transfer to the target of interest. The baseline target was 1999 RQ36, but a large range of viable asteroid targets exist within the performance of the same spacecraft. The system carries 13 kg of science and provides a total mission ΔV of 9.6km/s. The science payload, defined by a principal investigator from The Johns Hopkins University Applied Physics Laboratory, is representative of Discovery class science and includes visible and IR imaging, and terrain and gravity field mapping capabilities. The driving constraint for the spacecraft design was the volume limitation. The spacecraft required a wet mass of 287kg including 131kg of iodine. An equivalent xenon system was unable to fit with the volume constraints of the ESPA. Overall, the full life cycle cost of the mission, including launch costs, 3 years of mission operations and 5 months of science operations, but not including the science instrument costs is estimated to be $130M. For perspective, the life cycle cost is less than just the launch cost of a comparable mission without the use of the iodine Hall system.

Near-Earth Asteroid Orbiter

A similar mission study was performed through the MSFC Advanced Concepts Office with mission design support from GRC. The MSFC study focused on achieving Discovery class science at the moon. In order to achieve drastic cost savings, the launch was constrained to a GTO deployment as a secondary spacecraft and then leverages the iodine electric propulsion system for the spiral phases from GTO to Low-Lunar Orbit (LLO). The trajectory is very similar to that previously demonstrated by the European Space Agency's (ESA) first Small Missions for Advanced Research in Technology (SMART-1) spacecraft. Unlike SMART-1, the iodine system can launch as a secondary payload and fit within the physical constraints available to the ESPA Grande. SMART-1 had 367kg wet mass including 84kg of xenon, approximately 20% more than allowable within the ESPA Grande volume constraints.

The science objectives of the Low-Cost Discovery Lunar Orbiter required the spacecraft to perform very low altitude polar flybys. The spacecraft initially spirals down to a 100km circular orbit. After completing an initial science phase of 90 days, the spacecraft is then lowered into a 100km x 15km orbit with the perilune over the pole. The 15km perilune is maintained for 60 days before transitioning to a higher risk and reduced perilune orbit. The electric propulsion system has sufficient control authority to maintain a perilune error on the order of 1km with apolune station keeping. Note the perilune error approaches 2.5km, but that error occurs while the spacecraft is at apolune. Figure 5 illustrates the perilune error potential over the orbit for a 12 hour period during the 100km x 15km science phase. The orbit is relatively unstable at the low altitudes, and there are regions with high mountains relative to the flyby altitude. Finally, the propulsion system can lower the spacecraft to a 100km x 5km orbit with approximately 20 minute station keeping maneuvers required every 12 hours. A 50x50 lunar gravity model was used to support the low flyby trajectory analyses. This is a high risk phase of the mission, where the spacecraft would impact the lunar surface without station-keeping for 24-30 hours. Additionally, over a few regions of the moon, the perilune must be raised to avoid surface contact. It is the low thrust propulsion that enable the orbit required to achieve the desired quality of science.

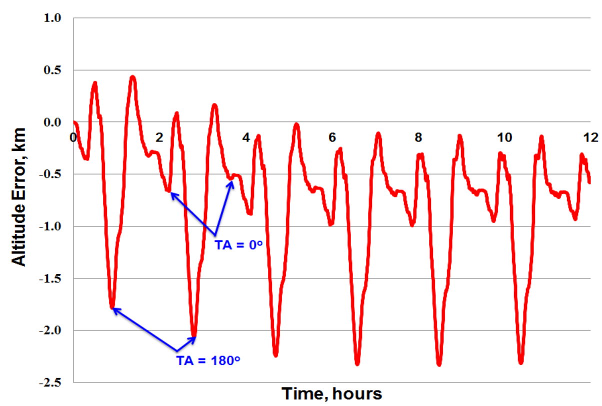

Figure 5: 100km x 15km perilune errors.

This low-cost lunar orbiter has similar attributes to the near-earth asteroid orbiter. In fact, there is potential for a common bus to provide with similar power and propulsion systems. The spacecraft is based on using two BHT-600 iodine thrusters for primary propulsion. The power system is sized to provide 1.7kW at beginning of life, sufficient to maintain full power to the thrusters at end-of-life including radiation degradation, 10% margin on the thrusters and 30% power margin for the remaining subsystems. The avionics are strategically placed in the spacecraft to increase the effective radiation shielding. The spacecraft uses the heritage Lunar Atmosphere and Dust Environment Explorere (LADEE) communication system in S-band. Similar to the asteroid orbiter, the total life cycle cost of the mission, including launch costs, operations and the instrument package is estimated at $149M. The spacecraft required 87kg of iodine to meet the mission requirements. The concept design is shown in figure 6.

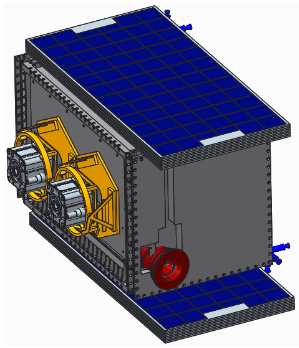

Figure 6: MSFC ACO concept vehicle.

III.III Orbital Transfer Vehicles

An alternative is to move the propulsion system to a transfer vehicle that can deploy the low cost SmallSats. This is not a new concept. Spaceflight Services is now offering a SHERPA hosted payload system with integrated in-space propulsion starting in both LEO and GTO. The SHERPA can be used to deploy SmallSats in higher value orbits. The post launch ΔV capability limited by the performance of the on-board propulsion system, notably the specific impulse and packaging efficiency. ESPA based OTVs using xenon based systems are capable of delivering payloads from GTO to GEO and even out to Mars. A single system sent to Mars could deliver several SmallSats or a large number of CubeSats to the moons of Mars or into a constellation for communication or observations at Mars.[9] Figure 7 illustrates an ESPA based OTV for geocentric transfers and figure 8 if additional xenon is need for transfers to Mars. Based on the density improvement with iodine, sufficient propellant can be placed entirely within the ESPA ring for transits to Mars, allowing for two additional slots for secondary payloads. With the same thruster costs and identical power processing unit, the change of propellant choice alone would result in minimum revenue increase potential of $10M, and that would be for the LEO slots. Higher potential revenue gains are expected for payloads beyond LEO.

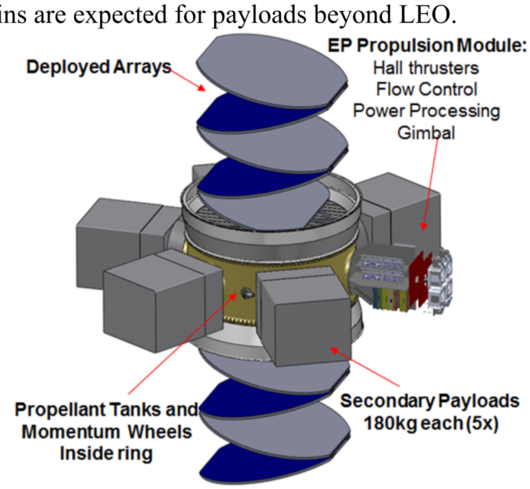

Figure 7: MSFC ACO concept vehicle.

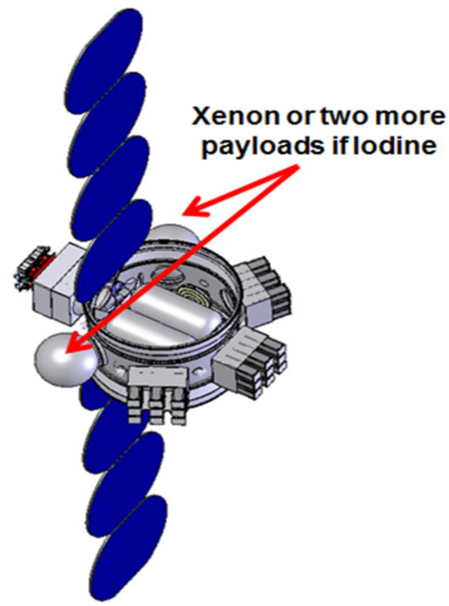

Figure 8: MSFC ACO concept vehicle.

IV. THE IODINE SATELLITE DEMONSTRATION

The technology is expected to yield significant benefits for geocentric, planetary science and human exploration through high ΔV SmallSat applications. However, during the Discovery mission studies, the iodine Hall technology was considered too immature for a Class C or higher mission. Several unknowns remain regarding the flight application of iodine, including potential degradation of the solar array performance, emissivity and passive thermal control systems and potential science instrument degradation from the iodine plume environment. To address these concerns and decrease infusion risk, NASA investigated and started developing a technology demonstration mission; iSat.

IV.I Mission Objectives

The top-level mission objectives are focused on validating the efficacy of iodine for future missions. The mission will validate in-space performance and characterize the environments. The level 1 requirement for the mission is: "*The iodine satellite shall demonstrate on-orbit operation of a 200W iodine hall thruster based satellite system no larger than 12U in low Earth orbit*". In addition to the top-level requirement, there are several success criteria. To achieve full success, the mission must demonstrate a cumulative thruster operation of more than 80 hours, including individual maneuver requirements of more than 10 minutes at full power and 15 minutes at de-rated performance. The mission must also demonstrate propulsion altitude change greater than 250km and a propulsive node change; minimum success is no less than 100m/s of ΔV. The spacecraft must also discern

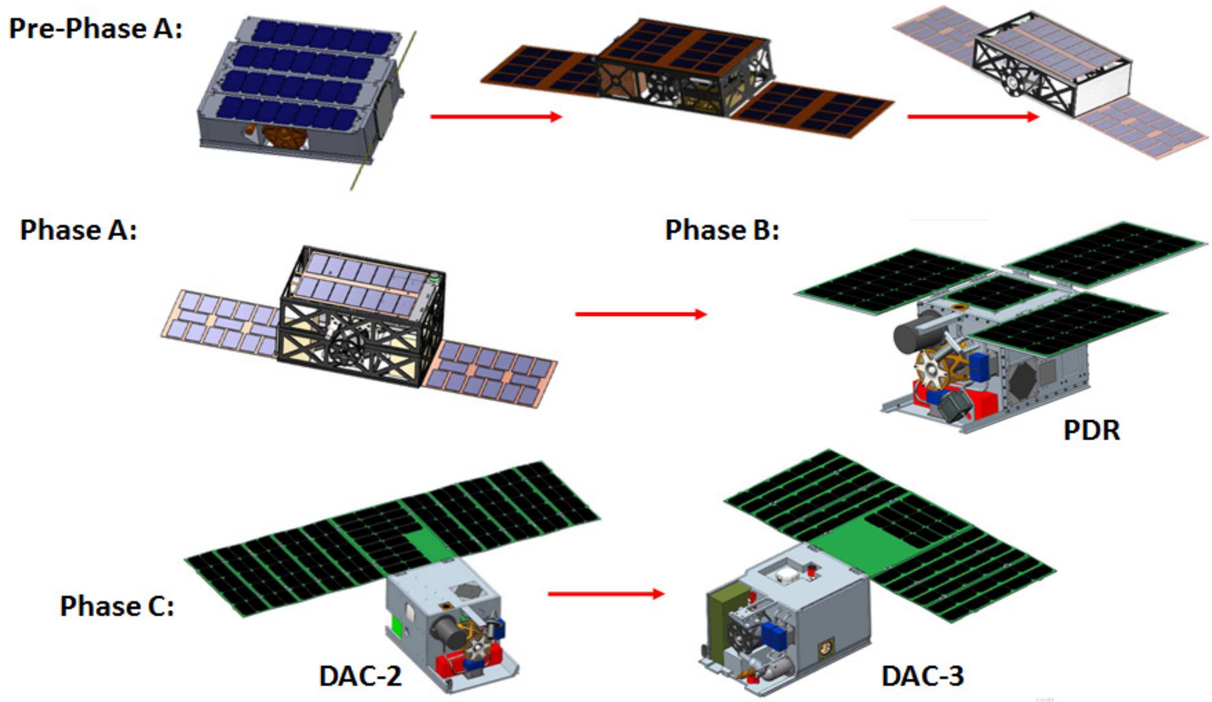

Figure 9: State-of-the-art Xenon and alternative propellant properties.

the average thrust within 5% uncertainty and specific impulse within 10% for full success. The system will include instrumentation to assess the iodine environment, track solar array performance degradation and be capable of taking images. Last, the spacecraft must lower its orbit such that it will de-orbit within 90 days of end of mission. The project will demonstrate twice the operational life of the highly successful TacSat-2 demonstration of the same thruster using xenon using a spacecraft less than 10% the mass.[10]

IV.II Mission Overview and Development Status

The first detailed concept design of a low-cost iodine Hall demonstration mission was completed by the COMPASS team at NASA GRC in 2012. The concept was a 6U spacecraft with a modified BHT-200 thruster and repackaged compact power processing unit (PPU). The initial concept was power starved, included aggressive assumptions and required significant engineering development work, but it showed the viability of a low cost and high value approach to flight demonstrate the iodine Hall technology. Multiple iterations on concept designs followed the first look. As fidelity was increased, additional power was added, followed by deploying the thruster to assist with heat rejection. The battery grew in volume resulting in bus growth to a 12U. Payloads were added and some of them later removed to accommodate system growth. The project completed the third design and analysis cycle (DAC-3) and is now progressing towards the Critical Design Review (CDR) design. The spacecraft concept evolution is illustrated in figure 9.

IV.III Concept of Operations

The mission concept of operations (CONOPS) is partially dependent on the orbit into which the iSat spacecraft is deployed. The assumption is that the spacecraft will be deployed into a sun-synchronous 600km circular orbit. The altitude limit is set by no fault tolerance to orbital debris requirement of a natural deorbit in less than 25 years. The spacecraft power and thermal systems are designed to accommodate the full range of altitude and inclinations without any specific node requirements (e.g. noon-midnight vs. 6AM-6PM). The vehicle will have significant battery charge available for deployment, tip-off correction and attitude tracking; all performed prior to ground intervention. An initial check-out period of two weeks is planned. The spacecraft will charge the power system while in sunlight, using momentum wheels and magnetic torque rods to rotate the vehicle to the required attitude and operating the thruster to perform maneuvers when appropriate. After check-out, the spacecraft will transition to a more automated mode with pre-programmed sequences. The spacecraft will lower its altitude from a 600km circular orbit to a 300km circular orbit, perform a plane change, complete any final operational maneuvers and then continue to lower only perigee until achieving < 90day deorbit. The CONOPS are illustrated in Figure 10.

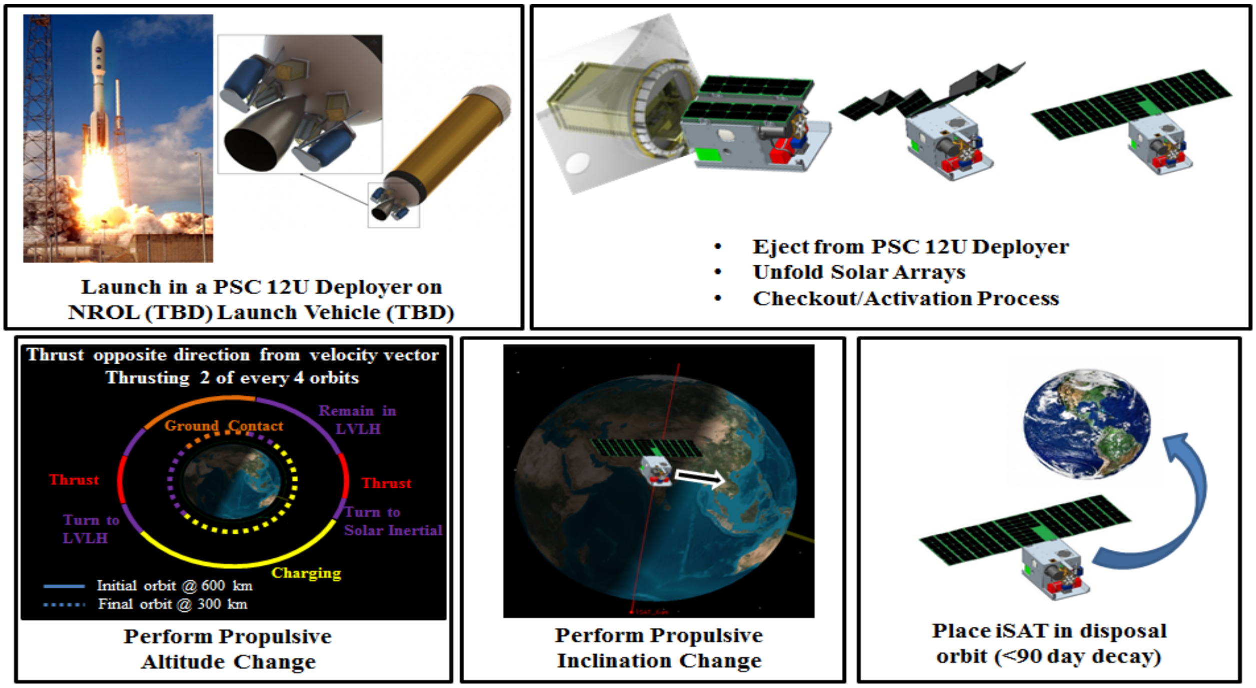

Figure 10: High level concept of operations.

IV.IV Flight System Overview

The mission is built around the demonstration of the Busek 200W iodine propulsion system. The propulsion system is a major driver for power, thermal management and attitude control. The iSat project has been working with engineering model hardware early in the project for risk mitigation, with significant testing planned prior to flight system integration. The thruster will undergo qualification testing with a separate flight unit, while the PPU will be a protoflight development.

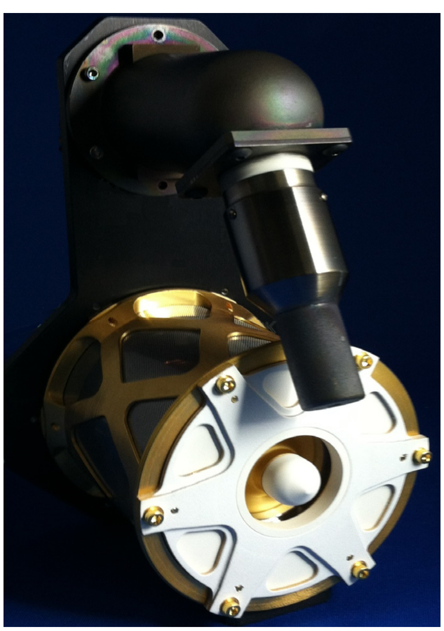

Figure 11: BHT-200-I thruster and cathode assembly.

Thruster

The thruster is a derivative of the BHT-200 flight thruster. The BHT-200 was the first American Hall effect thruster flow in space and launched in 2006 as a part of the TacSat-2 project. The thruster has been studied extensively and provides a good benchmark for comparing performance variances from the iodine version of the thruster; the BHT-200-I. The iodine thruster is distinguished from the nominal BHT-200 by the materials of construction and iodine-resistant coatings. The gas distributor was also redesigned due to material limitations.

Cathode

The BHT-200 flight model is neutralized by a xenon fueled hollow cathode with a BaO`-W emitter. For iSat, the BHT-200-I will be neutralized by an iodine fueled hollow cathode featuring a C12A7 electride emitter.[11] The cathode will leverage the flight heritage design including the mounting structure, but the new emitter enables a significant increase in total system efficiency by reducing the power required for cathode conditioning. The thrust and cathode assembly is shown in figure 11.

Power Processing Unit

The Power Processing Unit is also a major advancement over state of the art. The PPU includes a new topology for compact design. The compact PPU design started under an Air Force SBIR program under the Operationally Responsive Space (ORS) office. A 2nd and 3rd iteration of the compact PPU was funded to support two different NASA SBIR projects. The iSat 200W PPU and a NASA Game Changing Development

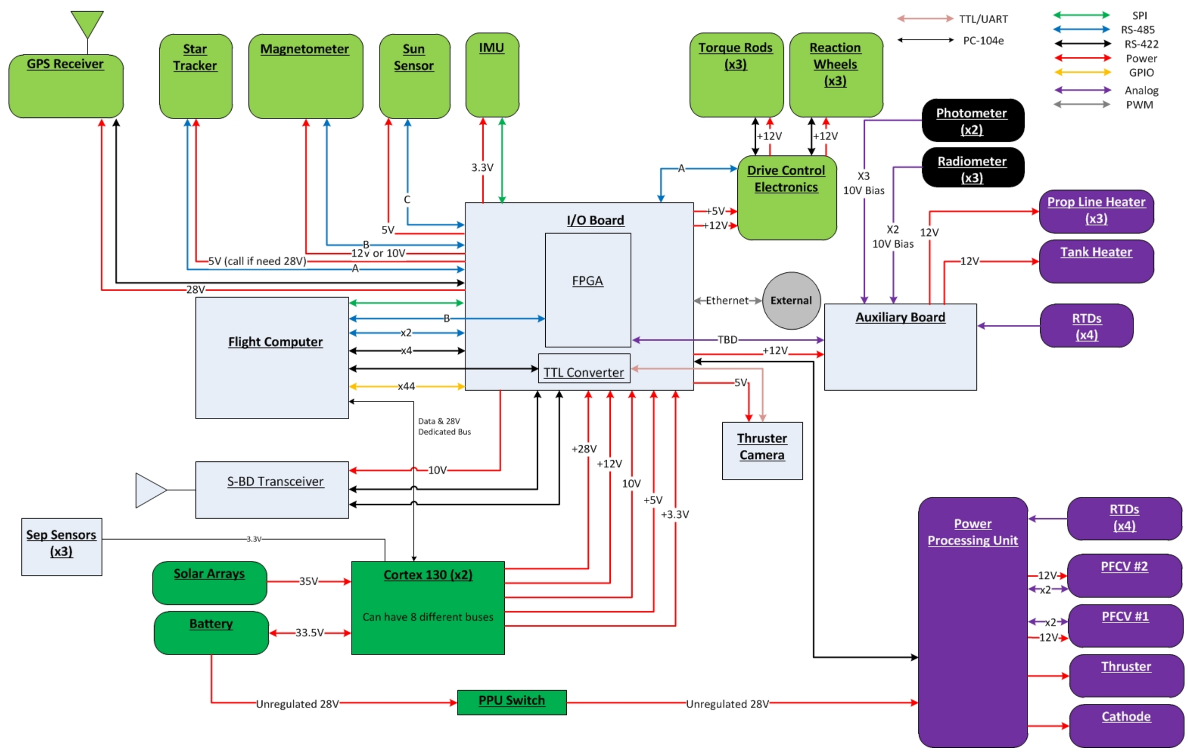

Figure 12: Baseline command and data handling architecture.

600W version are being built in parallel as the 4th and 5th design of the compact PPU. Both the 200W and 600W PPUs are identical form factor with significant commonality between the two. The 200W PPU is essentially a 600W PPU optimized for lower power operation. The PPU has the same power requirements for operating a xenon thruster and can be used for future xenon mission applications. The PPU will use an RS422 interface, accept an input voltage range from 24-36V and leverage FPGA based control of all outputs and telemetry.

Feed System

The feed system is designed similarly to the Advanced Xenon Feed System.[12] The core of the flow control is a pair of parallel flow paths with VACCO proportional flow control valves (PFCVs). The PFVCs have been modified from the xenon qualification valve to reduce the pressure drop, add internal heaters and temperature sensors, and material changes for iodine compatibility.

Traditional systems rely on high pressures to ensure adequate mass flow to the cathode and thruster, and are largely unaffected by gas build-up and small pressure drops along the lines. A low pressure system, however, is very sensitive to all pressure drops which can overwhelm the tank pressure and prevent adequate flow or cause flow reversal. Designing a low pressure sublimation-driven propellant feed system requires careful consideration of the line pressure and design sensitivity to several factors, including temperature, physical line dimensions, filter choice, and tube material. For the feed system design, both modeling and experiment characterization is continuing.

System Architecture

The iSat system is a complete spacecraft with relatively high power density, full three axis attitude control and a significant number of I/O interfaces. The power system is a relatively standard panel deployment of Spectrolab 28.3% efficiency Ultra Triple Junction cells with an approximate performance of 60W power generation capability. The solar arrays provide a voltage boosted to 34V for battery recharge. The system also leverages Spaceflight Industries Cortex 130 boards for power management and distribution.

For command and data handling, the iSat system leverages the Cortex 160 flight computer combined with two custom in-house boards. The Cortex 160 implements a Linux real time operating systems (RTOS) and includes five RS-422, three RS-485, two SPI, two I^2C and two parallel digital camera inputs. The board is designed for a 3 year lifetime and a 15krad total ionizing dose. Despite what it can provide over alternative COTS options, it lacks the additional universal asynchronous receiver/transmitter (UART), pulsed width modulation and RS-232 capabilities required. The custom boards provide these capabilities and an FPGA to do analog to digital conversion, a data bus conversion/bridge, and instrument command and circuitry. The C&DH architecture is shown in figure 12.

Structures and Mechanical Subsystems

The iSat structure is defined to meet the standard interface of a Planetary Systems Corporation 12U CubeSat deployer. The iSat structure maximum outer dimensions are 365mm x 229mm x 212mm. After ejection, the iSat vehicle has spring loaded passive deployment mechanisms for the solar panels. The design will also allow for a commanded deployment mechanism if required by the launch vehicle. The primary structure is fabricated in-house from 7075 aluminum alloy with a hard anodized finish. The design challenges unique to iSat include handling the thermal loads from the thruster, the potential shielding needed for the electromagnetic interference / compatibility (EMI/EMC) environment, the overall power density within the spacecraft and the packaging of a large number of components within the limited volume available while still leaving clearances for standard connections. The design also allows useful viewing angles for the guidance, navigation and control (GN&C) sensors, the S-BD antenna, the GPS antenna, a thruster-imaging camera, and a payload complement of two photometers and three radiometers. The basic layout of the spacecraft is illustrated in figure 13.

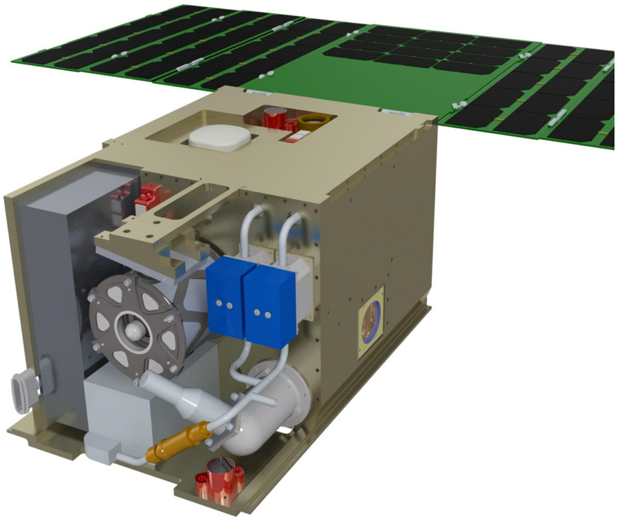

Figure 13: System configuration for iSat.

Technical Performance Metrics

To track the progress of the project towards successfully meeting all mission objectives, the project is using the standard application of technical performance metrics (TPMs). For iSat the TPMS are based on fitting in the 12U package and meeting the mission success criteria. The TPM status and progress from PDR is provided in Table 2. The system is on track for a closed design. The schedule for the project is to deliver the spacecraft for launch in April 2017 and a launch in November of 2017.

Table 2: Technical performance metrics for iSat.

TPM	Metric	Status at PDR	Status at start of DAC-3	Comments
X/Y/Z Margin	Planetary Sys. Envelope	GREEN	GREEN	Fits within envelope
Total ΔV	As determined by sim	GREEN	GREEN	Current ConOps requires 198.4 m/s without margin
Max Duration of Single 200W Burn	Max Duration Burn at 200W assuming one burn an orbit	RED	GREEN	Solar Array redesign, battery capacity increase, and ability to include charging orbits allows for this burn to be accomplished without exceed the 60% DoD
Max Duration of Single 100W Burn	Max Duration Burn at 100W assuming one burn an orbit	RED	GREEN	Solar Array redesign, battery capacity increase, and ability to include charging orbits allows for this burn to be accomplished without exceed the 60% DoD
Thruster Operation Duration	Total Cumulative run time of thruster during mission	GREEN	GREEN	Current ConOps has a runtime of 80+ hours.
Data Bandwidth	Available bits vs. Requested bits per day	GREEN	GREEN	Current S-BD solution exceeds data requests.
Battery SOC	Battery SOC for smallest repeatable interval of each mission phase	RED	GREEN	Battery currently dips to approximately 75% state of charge with current assumptions. Requirement is not to dip below 60% SOC.
Mass Margin	Total S/C mass vs Allocation	YELLOW	GREEN	Given that the spacecraft mass allocation is 24 kilograms, mass is not thought to be a driver. Spacecraft mass update is ongoing to incorporate DAC-2 results.
Available Payload Data	Available bits for the payload vs requested bits for the payload per day	GREEN	GREEN	S-BD solution more than meets the requirements for the data from the diagnostic sensors.
Available Payload Power per Power Budget	Required power on a per orbit basis vs. allocated power	RED	GREEN	The diagnostic sensors do not require much power.

V. BUS UTILITY

The iSat spacecraft is a high performance small spacecraft that fits within a canesterized 12U deployer. While a low-cost "CubeSat", the bus itself has significant capability beyond traditional CubeSats. The spacecraft has full 3-axis attitude control, significant primary propulsion, 60W of power generation and can provide more than 200W at 28V for moderate durations. Addition studies since iSat formulation have shown multi-mission applicable of the spacecraft for either science or additional technology demonstration missions.

VI. SUMMARY

Propulsion remains one of the key limitations for small satellite missions. Iodine may be enabling due to the potential ΔV per unit volume. Iodine enables a new class of planetary and exploration class missions by packaging sufficient ΔV, within secondary payload constraints, for GTO launched spacecraft to transit to the moon, asteroids, and other interplanetary destinations for ~$150M full life cycle cost including the launch. The technology has potential to reduce life cycle mission costs by a factor of 5. ESPA based OTVs are also volume constrained, and a shift from xenon to iodine can significantly increase the transfer vehicle ΔV capability or enable additional secondary payloads for increases revenue potential. The iSat project was initiated to validate the efficacy of iodine propulsion for SmallSat and higher class missions. The project is currently meeting all performance metrics and is still working towards a launch in 2017. Beyond iSat, the bus is also a high performance low-cost option with applicability to future missions.

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