Mission and System Advantages of Iodine Hall Thrusters

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The exploration of alternative propellants for Hall thrusters continues to be of interest to the community. Investments have been made and continue for the maturation of iodine based Hall thrusters. Iodine testing has shown comparable performance to xenon. However, iodine has a higher storage density and resulting higher ΔV capability for volume constrained systems. Iodine's vapor pressure is low enough to permit low-pressure storage, but high enough to minimize potential adverse spacecraft-thruster interactions. The low vapor pressure also means that iodine does not condense inside the thruster at ordinary operating temperatures. Iodine is safe, it stores at sub-atmospheric pressure, and can be stored unregulated for years on end; whether on the ground or on orbit. Iodine fills a niche for both low power (<1kW) and high power (>10kW) electric propulsion regimes. A range of missions have been evaluated for direct comparison of Iodine and Xenon options. The results show advantages of iodine Hall systems for both small and microsatellite application and for very large exploration class missions.

I. Introduction

Hall thrusters belong to the family of electric thrusters. Electric thrusters use electric energy to heat or ionize and accelerate the injected propellant. In comparison to chemical thrusters, electric thrusters offer a much higher specific impulse capability and thus dramatically reduce the amount of propellant needed to achieve a predetermined ΔV. A large number of flight Hall systems have been developed, flown and continue to fly in space.

Hall thrusters with xenon propellant have excellent mission performance at power levels from 1kW – 10kW. However, small satellites with severe volume constraints have limited potential for high pressure xenon systems. At high power, xenon Hall thrusters >10kW are pushing the limits of ground facility capabilities; with iodine, space-relevant background pressures are much easier to achieve. Additionally, missions requiring large amounts of propellant can have system level advantages due to the higher propellant densities and low storage pressures.

Under the support of the USAF and NASA's Small Business Innovation Research (SBIR) program, Busek has been evaluating the performance of the BHT–200 flight thruster with iodine, collecting performance data for higher power iodine thrusters and maturing a 600W iodine propulsion system. NASA continues its investment in iodine based Hall systems to capture two distinct mission niches, very low power small satellites and high power exploration class electric propulsion. Advantages of a condensable propellant extend beyond flight mission advantages, but also provides ground test advantages; especially at high flow rates. System qualification and demonstration at low power may offer a practical option for a relatively rapid infusion of this new technology and at a fraction of the cost of high power implementation.

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II. Iodine Propulsion Development Background and Status

A. Iodine versus Alternatives

Busek has been leading the exploration of alternative propellants for Hall thrusters. Tests and analyses were completed for magnesium (Mg), zinc (Zn), iodine, and bismuth (Bi). All of the high density options show promise for higher ΔV capability for volume constrained applications. With respect to xenon, magnesium and zinc have lower atomic mass, which allows high specific impulse for constant discharge voltage. Bismuth HET systems have been developed at Busek and elsewhere.[1,2] Soviet era literature indicates thruster efficiencies much greater than 50% have been achieved.[3] With iodine, performance is comparable with SOA xenon, as detailed below, although testing indicates iodine produces slightly higher thrust-to-power, which is desired for many mission applications.

When selecting the propellant for any mission, system level considerations like storage density, pressure, and required temperature should be taken into account. The required power for vaporization limits the applicability magnesium, zinc, and bismuth for small satellite power starved missions. For such spacecraft, the high storage density and high vapor pressure of iodine makes it a superior candidate. Propellant characteristics for selected alternative propellants are shown in **Error! Reference source not found.**.

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

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

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^oC, 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 **Error! Reference source not found.**.

The market has established xenon fueled systems in the 1-9kW power range. Here there is limited justification to supplant incumbents with lower maturity technology. However, both the very small and very large satellite markets have limitations that may be overcome through the implementation of iodine. Small thrusters may be ideal for CubeSats or other small spacecraft where volume is at a premium. Large thrusters and clusters with high throughput requirements can offer system packaging and propellant storage. Also, higher power systems with high flow rates may offer ground test advantages over non-condensable propellants, and therefore mission margin reduction and flight performance risk reduction. Large system applications may include large orbital tugs and human exploration missions.

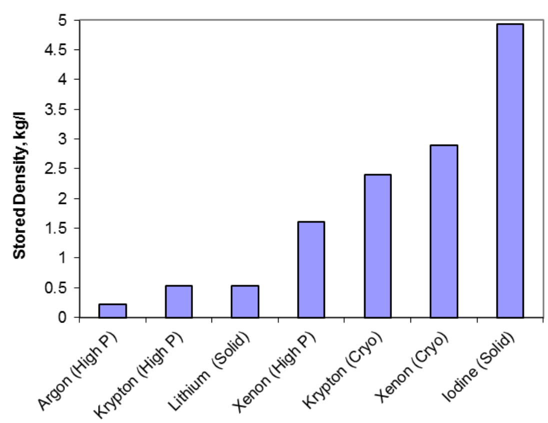

Figure 1. Stored Density of Electric Rocket Propellants in kg/l (high pressure at 14-MPa, 50^oC).

To illustrate the benefits of iodine, it is useful to compute the propellant volume for a high throughput mission. Consider a notional mission requiring 13t of propellant. At typical storage conditions, this implies 8,075-l of high pressure Xe or 4,483-l of cryogenic Xe (2.9-kg/l). With I_2, the propellant volume could drop as low as 2,637-l. Furthermore, the I_2 could be stored at low pressure and tanks not in use could be unregulated and quiescent without heating required and standard manufacturing techniques could be used for tank design and production. Finally, the low operating pressure also allows for conventional or potentially additive manufacturing of propellant tanks to nearly any desired shape without the need for domes or composites.

B. Performance and Plume Testing

Busek first tested the BHT-200[4] with iodine under an Air Force sponsored program and presented results in 2011.[5] Busek subsequently tested the BHT-1000[6] and BHT-8000[7] with iodine. All test data indicate that iodine performance is competitive with xenon performance at all power levels. The iodine operation typically trends to a higher thrust-to-power and slightly reduced specific impulse. Test results also indicate a reduction in plume divergence with iodine.[6]

Thrust is typically measure directly with an inverted pendulum, "Null" type thrust stand.[8] Specific impulse is defined by

$$I_{sp} = T / \dot{m} g_0 . \qquad [1]$$

Here, T is thrust, $\dot{m}$ is the total mass flow rate, and g_0 is the force of gravity the surface of Earth. When I_{sp} is calculated using only the anode mass flow, $\dot{m}_a$, this is known as the anode I_{sp}. The efficiency determines the ratio of thrust to power (T/P) available at a particular I_{sp} through the relation

$$\eta = \frac{T}{P}\frac{I_{sp} g_0}{2} . \qquad [2]$$

Thus, at constant thruster efficiency there is a tradeoff between (T/P) and I_{sp}. At constant voltage, a heavier propellant may yield lower I_{sp} and higher (T/P).

1) BHT-200

With iodine, a BHT-200 demonstrated a nominal anode efficiency of 48%.[6] The thrust peaked at 25mN at a discharge power of 500W, and performed at 12-14mN under the nominal 200W operation. A thrust-to-discharge power ratio of 75mN/kW was demonstrated with 150V discharge and 65mN/kW under nominal conditions. The specific impulse under nominal conditions was approximately 1500s, although the thruster demonstrated more than 2000s I_{SP} with a 400V discharge. The thrust and specific impulse demonstrated at various conditions is provided in Figure 2.

In addition to performance characterization, the iodine plume of the BHT-200 was evaluated for comparison to xenon. Faraday probe measurements of current density $j(\theta)$ showed lower plume divergence with iodine and the strong presence of diatomic ions in the plume.[9] With an experimental version of the BHT-200, the dimer population was found to vary with thruster operating conditions and angular position with respect to the thruster centroid. With the thruster operating at 250-V and 0.6-A to 0.8-A, over 10% of the high energy beam flux by number (over 20% by mass) at some angles was comprised of dimers.[10]

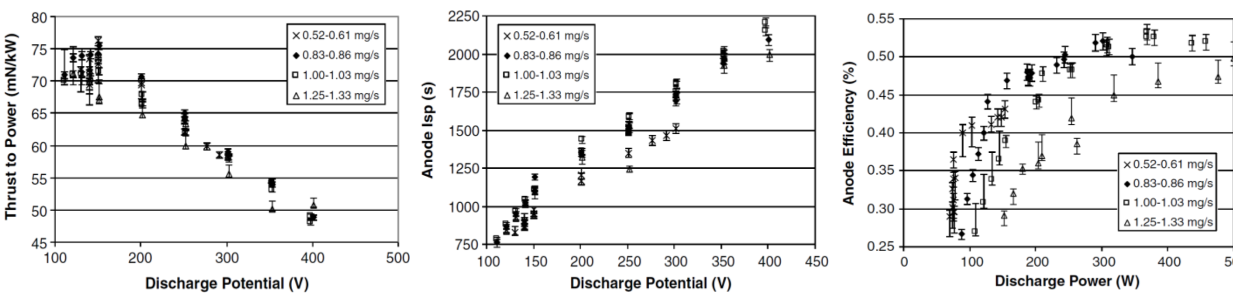

Figure 2. Thrust-to-power (left), anode I_{SP} (center) and anode efficiency (right) from BHT-200 with iodine.

2) BHT-1000 and BHT-8000

Busek continued its iodine experiments with the BHT-1000 and BHT-8000. Performance trends with the BHT-1000 were consistent with the earlier test data; iodine produced slightly lower specific impulse while demonstrating a higher thrust-to-power. Again, the efficiencies were comparable to xenon. The iodine performance of Busek's BHT-8000 was measured between 2kW and 10kW. High power testing showed slightly reduced anode efficiency with iodine, but the estimated uncertainty in these measurements was relatively high due to the iodine flow rate calibration.

Busek also tested hollow cathodes operated on iodine with both lanthanum hexaboride and C12A7 electride emitters. The latter may be preferred for very small spacecraft because it does not require a heater for the insert.

Table 2. BHT-8000 Test Data for Iodine and Xenon

	Discharge Voltage [V]	Discharge Current [A]	Tank Pressure* [Torr]	Discharge Power [W]	Total Power [W]	Thrust [mN]	Discharge Specific Thrust (mN/kW)	Anode Isp [s]	Total Isp [s]	Anode Efficiency [%]	Thruster Efficiency [%]
Xenon	200	9.8	4.2E-05	1962	1972	146	74	1445	1313	53	48
Xenon	201	10.1	2.7E-05	2027	2034	151	74	1423	1293	52	47
Iodine	200	9.1	5.1E-05	1816	2053	129	71	1302	1266	45	39
Xenon	203	15.9	3.8E-05	3224	3235	235	73	1524	1385	54	49
Iodine	199	15.6	5.5E-05	3106	3118	223	72	1344	1321	50	49
Xenon	302	10.0	3.2E-05	3020	3036	199	66	1875	1705	60	55
Iodine	301	9.3	5.1E-05	2787	2855	187	67	1796	1748	59	56
Xenon	300	13.4	3.9E-05	4029	4050	267	66	1927	1752	63	57
Iodine	301	12.3	4.7E-05	3702	3889	243	66	1803	1766	60	56
Xenon	301	16.2	4.6E-05	4863	4893	321	66	1960	1782	63	57
Iodine	301	15.0	5.5E-05	4509	4547	297	66	1791	1760	61	56
Xenon	401	9.9	4.8E-05	3976	4018	226	57	2240	2036	63	56
Iodine	400	9.6	5.1E-05	3856	4011	210	54	2122	2062	57	53
Iodine	400	12.7	4.70E-05	5096	5240	295	58	2254	2206	64	61
Iodine	405	16.94	7.33E-05	6861	6951	407	59	2159	2075	63	60
Iodine	500	12.5	5.60E-05	6260	6360	334	53	2552	2498	67	64
Iodine	505	17.84	8.00E-05	9009	9100	474	53	2519	2420	65	62

*Tank pressure for iodine is indicated value calibrated for xenon (actual pressure for I2 is significantly lower)

Plume measurements were also carried out with the BHT-1000. As seen earlier with the BHT-200, divergence of the ion beam current was less with iodine. For instance, at a discharge potential of V_d = 400-V the 30^o half angle captured 90% of the iodine but only 86% of the xenon plume at a similar power level. For comparable impingement, the solar array and spacecraft plume keep-out half angle can be reduced by 5^o. Testing also showed that the large angle beam could be effectively attenuated by the addition of a simple plume shield.

Based on measured BHT-1000 current densities and known vapor pressures, condensation upon spacecraft surfaces was projected to be very minimal or non-existent.[6] To avoid bulk accumulation of I_2 on surfaces, the removal rate must exceed the arrival rate. The flux away from a surface may be estimated from the I_2 vapor pressure. At typical spacecraft surface temperatures, the vapor pressure of I_2 is two decades higher than that of Hg and four decades higher than that of Cs, both of which have been tested in space with ion engines. The vapor pressures are compared in Figure 4.[11] The comparison with Hg is especially significant because Hg ion engines on the SERT II spacecraft were successfully fired on-orbit for 4000 hours.[12] Most spacecraft surfaces, including solar arrays, were too warm to permit Hg condensation and showed no evidence of condensate. SERT II proved that condensable propellants are a viable option for electric satellite propulsion.

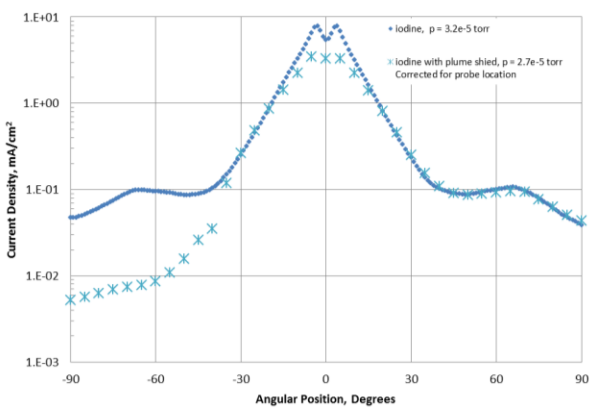

Figure 3. Plume Current Density (uncorrected for charge exchange) with and without Plume Shield, 500V, 2A, Iodine.

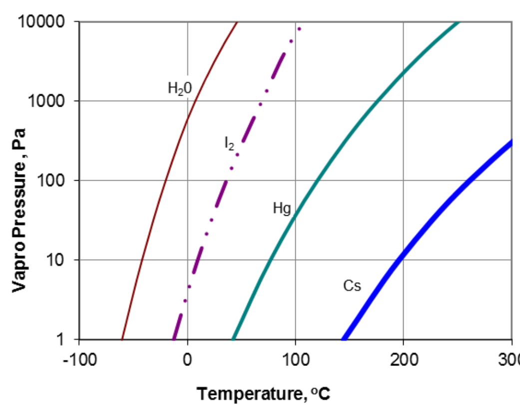

Figure 4. The vapor pressure of I_2 and other electric rocket propellants in Pascals.

C. Development Plans

Development remains prior to implementation of iodine Hall flight systems. NASA MSFC has developed a flight-like iodine feed system and is performing propellant transfer studies. NASA Glenn is leading iodine cathode investigations. Busek is currently working under a Small Business Innovative Research contract to deliver an engineering model (EM) 600W system. NASA MSFC has also purchased an EM model 200W thruster for testing to support a potential near-term demonstration mission.

Under the Space Technology Mission Directorate (STMD), the Game Changing Development (GCD) Program intends to fund the qualification of both the 200W and 600W systems starting in FY15. The GCD effort includes delivery of iodine cathodes, qualification model thrusters and PPUs at both 200W and 600W, system level performance testing and environmental testing necessary. Hardware is planned for delivery in FY16.

III. Mission Applications

A range of mission studies have been completed to identify mission level advantages using iodine as the primary propellant. In addition to iodine based missions for commercial, military, human and robotic exploration, multiple concepts have been developed for a low-cost high value technology demonstration mission.[13] The mission benefits primarily result from two enabling features of iodine, the increased propellant density and low operating pressure.

A. Microsatellites (Mass < 100kg)

The small satellite market continues to see strong growth in mission opportunities and subsystem capabilities. Small satellites offer rapid technology infusion opportunities and are validating the technologies required for interplanetary small satellite missions. The JPL Interplanetary NanoSpacecraft Pathfinder In a Relevant Environment (INSPIRE) was selected for a launch opportunity to become the first interplanetary CubeSat. INSPIRE has mission objectives to demonstrate and characterize nano-spacecraft communication, navigation, command and data handling, and mother-daughter relay communications.[14] CubeSats are also being proposed for high value science; well beyond simple technology demonstrations.

One of the key limitations to small satellite exploration is still the lack of primary propulsion. The challenges are due to limited resources available: power, mass, and volume. Additionally, 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; increasing 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 both low risk to mature and can offer modest attitude control. Miniature ion propulsion has been explored; offering high ΔV potential, but requires high pressure gas. Microfluidic Electrospray Propulsion (MEP) technologies offer 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.

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 4 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. Results in figure 5 do not account for propulsion system mass fraction or supporting subsystem requirements. Figure 5 indicates the potential merit of iodine based propulsion due to the combined benefits of high specific impulse and high storage density.

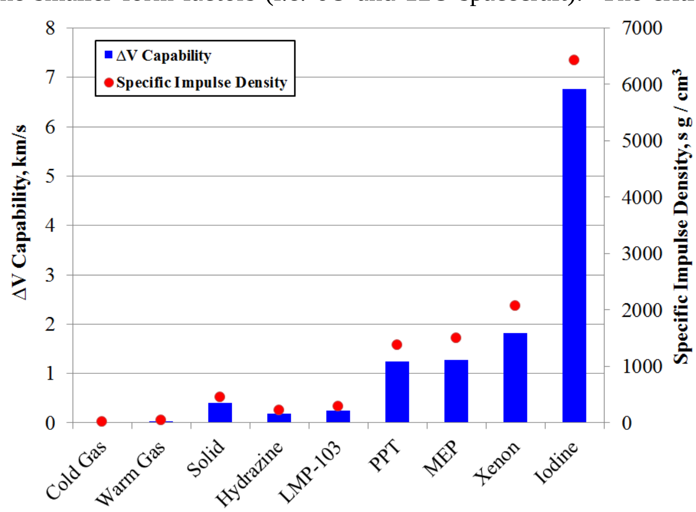

Figure 5. I_{SP}-Density and 1U ΔV capability for a 6U Spacecraft.

1) Geocentric Missions

As discussed previously, the SmallSat market is showing tremendous growth potential, however; SmallSats still have limited capabilties beyond the propulsion limitations. The Department of Defense (DoD), NASA and industry have shown significant interest in LEO SmallSat applications. SpaceWorks Enterprises, Inc. projects between 2,000 and 2,750 nano and microsatellite launches between 2014 and 2020; ranging from 1-50kg in mass.[15] Both military and civil space plans include deployment of large constellations of SmallSats in LEO for communciation and imaging capabilities.[16] Figure 5 provides notional constellations for communcation and imagery. The community continues to investigate responsive capabilities for SmallSats, either through a responsive launch capability or a responsive space deployed maneuverability capability. 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.

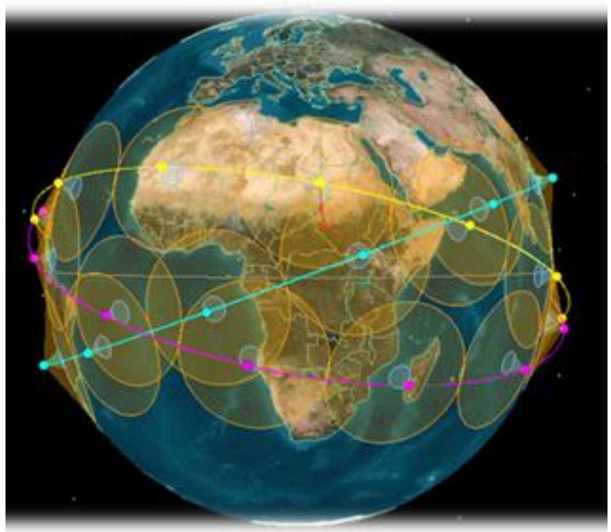

Figure 6. Example constellations achievable from a single starting orbit with iodine propelled SmallSats.

Both 6U and 12U iodine spacecraft have been studied in great detail for LEO applications.[13] A preliminary Mass Equipment List (MEL) for a 12U iodine satellite example with a 2kg payload can carry more than 5kg of iodine and fit within the 20kg standard mass requirement. At 250V discharge voltage, the 12U iSAT is capable of approximately 4km/s of ΔV and also well within the predicted life capability of the BHT-200 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 6 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.

Table 3. Preliminary MEL for 12U iSAT in LEO

iSAT Mass Estimation List - 12U LEO		Basic Mass (kg)	MGA (%)	MGA (kg)	Predicted Mass (kg)
1.0	Structures	1.601	30%	0.480	2.081
2.0	Mechanisms	0.100	30%	0.030	0.130
3.0	Thermal	0.334	30%	0.100	0.434
4.0	Power	2.052	30%	0.616	2.668
5.0	Guidance Navigation & Control	1.518	10%	0.152	1.670
6.0	Communications	0.090	6.00%	0.005	0.095
7.0	Command and Data Handling	0.324	16%	0.053	0.377
8.0	Propulsion	3.846	25%	0.965	4.811
Dry Mass		9.864	24%	2.401	12.265
9.0	Payload	2.000	30%	0.600	2.600
10.0	Non-Propellant Fluids	0.000	0%	0.000	0.000
Inert Mass		11.864	25%	3.001	14.865
11.0	Propellant (Solid Iodine)	5.135		0.000	5.135
iSAT 12U LEO Total Mass		16.999		3.001	20.000

2) Interplanetary Missions

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 opportunties to accelerate exploration from asteroids, comets, Mars, Mars' Moons, outer planets, etc. It is infeasible for NASA to meet it's objectices 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 and potentially 12U volume spacecraft launched into an escape trajectory. However, the mission value is 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 reconnassance 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.

NASA MSFC has assessed the 12U interplnaetary iodine Satellite (iiSAT) for near-Earth asteroid exploration at very low cost. If deployed on an escape trajectory, the iiSAT vehcile has sufficient propulsion capability to redezvous with a wide range of near-Earth targets. The baseline mission is a rendezvous with 2000 SG344, a target of interest as a potential crew rendezvous destination or an Asteriod Redirect Mission target. The iiSAT vehcile with approximtely 2.5km/s of ΔV capability is shown in figure 7.

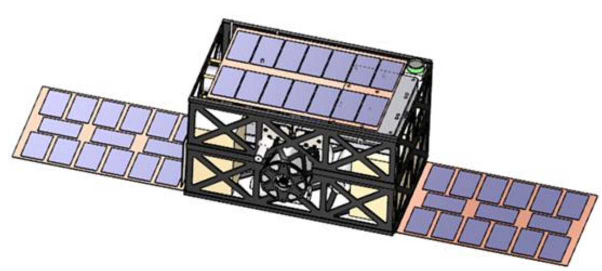

Figure 7. 12U iiSAT preliminary concept design.

B. SmallSats (Mass < 300kg)

Multiple studies have been performed with the NASA GRC COMPASS team and the NASA MSFC Advanced Concepts Office to assess the viability of SmallSats to delivery high value science for low-cost. The goal of the studies included the ability to delivery representative Discovery mission class science payloads to targets of interest. The studies include mission concept development, preliminary spacecraft design and life cycle cost assessment.

1) Near-Earth Asteroid Orbiter

The COMPASS team from NASA GRC evaluated a mission to a near-Earth asteroid using a secondary launch into geosynchronous transfer orbit (GTO). The GTO launch would result in >$100M in launch cost savings over a dedicated interplanetary launch. Using an EELV Secondary Payload Adapter (ESPA) Grande, secondary launch providers can deploy up to 300kg to GTO for an advertised cost of just under $10M.[17] The ESPA Grande with the stowed NEA orbiter is shown in figure 8.

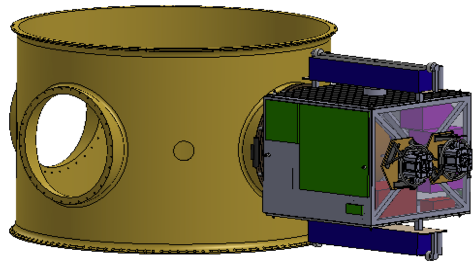

Figure 8. ESPA Grande with stowed vehicle.

The study proposed a solution using two BHT-600 iodine thrusters and a spacecraft power of 1500W at End-of-Life. The mission requires a slow spiral of approximately 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 the same spacecraft. The system carries 13 kg of science and provides a total mission ΔV of 9.6km/s. The science payload was defined by a principal investigator from The Johns Hopkins University Applied Physics Laboratory representative of Discovery class science return including visible and IR imaging, and terrain and gravity field mapping capabilities. The driving constraint for the spacecraft design was the volume limitation.

The desire for efficient power packaging and physical envelope constraints led to a design leveraging two 800W Roll-Out Solar Arrays (ROSA). The preliminary concept design of the near-Earth asteroid explorer is shown in figure 9 in the deployed configuration and the MEL is provided in table 4.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. The authors note that the equivalent xenon system was insufficient to meet the mission needs due to the propellant tank volume requirement.

Table 4. Preliminary MEL for NEO Orbiter.

Main Subsystems	Predicted Mass (kg)	Aggregate Growth (%)
i2Hall Spacecraft	272	
SEP Bus	272	7%
Science Payload	13	0%
Attitude Determination and Control	5	3%
Command & Data Handling	8	28%
Communications and Tracking	7	10%
Electrical Power Subsystem	29	25%
Thermal Control (Non-Propellant)	22	15%
Propulsion (Chemical Hardware)	5	0%
Propellant (Chemical)	1	0%
Propulsion (EP Hardware)	25	10%
Propellant (EP)	131	0%
Structures and Mechanisms	27	17%
Element 1 consumables (if used)	0	
Estimated Spacecraft Dry Mass (no prop,consum)	139	14%
Estimated Spacecraft Wet Mass	272	
. Growth Calculations SEP Bus		Total Growth
Dry Mass Desired System Level Growth	142	30%
Additional Growth (carried at system level)		14%
Total Wet Mass with Growth	287	

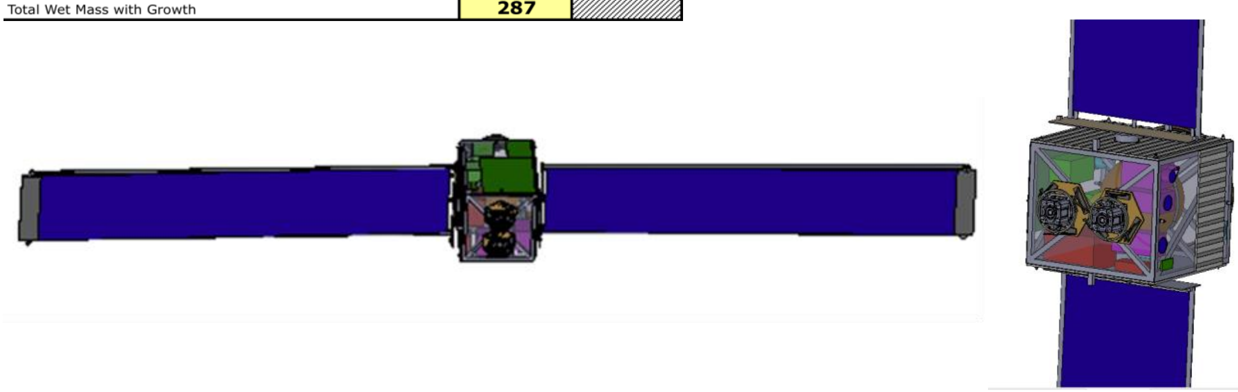

Figure 9. Near-Earth-asteroid orbiter concept design in deployed configuration.

2) Low-Cost Discovery Lunar Orbiter

Another similar study was performed by the MSFC Advanced Concepts Office with mission design support from the Glenn Research Center to assess the potential to achieve Discovery class science at the moon at a cost cap well below the current limitation. Again, 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 was with a mass 367kg wet mass including 84kg of xenon, approximately 20% more than allowable by the ESPA Grande.

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 10 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.

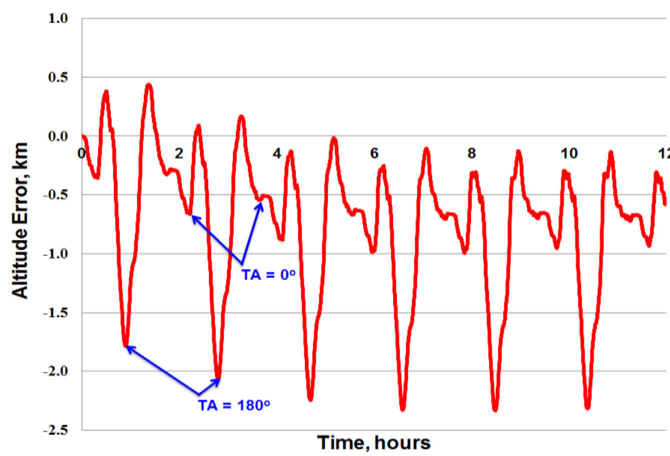

Figure 10. 100km x 15km perilune errors.

The low-cost lunar orbiter has similar attrbutes 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 degredation, 10% margin on the thrusters and 30% power margin for the remainng subsystems. The avionics are strategicaly placed in the spacecraft to increase the effective radiation shielding. The spacecraft uses the heritage Lunar Atmosphere and Dust Envirnonment Explorere (LADEE) communication system in S-band. Similar to the asterid orbiter, the total life cycle cost of the mission, including launch costs, operations and the instrument package is estimated at $149M. The spacecraft MEL and configuration are provided in table 5 and figure 11 respectively.

Table 5. Preliminary MEL for the Lunar Orbiter.

Mass Estimation List (MEL)			Basic Mass (kg)	ave MGA (%)	Predicted Mass (kg)
1.0	Structures		21.2	30%	2756%
2.0	Mechanisms - In Subsystems				0.0
3.0	Thermal		4.8	0.3	6.0
4.0	Power		90.6	0.2	107.2
5.0	Guidance Navigation & Control (GN&C)		8.4	0.1	9.8
6.0	Communications		6.8	0.3	8.5
7.0	Command and Data Handling (C&DH)		7.9	0.3	10.1
8.0	Propulsion		17.3	0.1	17.3
Dry Mass			157.0	16%	186.6
9.0	Insturments		10.1	0.2	12.2
10.0	Non-Propellant Fluids		0.0	0%	0.0
Inert Mass			167.2	16%	198.7
11.0	Propellant				
	11.1	Nitrogen (Cold Gas)	9.4	5%	9.9
	11.2	Iodine	87.0	3%	89.6
Total Mass			263.6		298.2

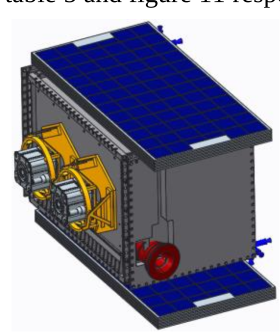
Figure 11. 12U iiSAT preliminary concept design.

3) SmallSat Interplanetary Potential

More than 20 years ago, NASA selected 11 concept studies that evolved into the Discovery mission program. The Discovery program was focused on "costs limited to no more than $150M and acceptance of greater level of risk". The decadal survey emphasizes the Discovery program value and stresses the need to continue frequent opportunities to make progress on key science objectives. The more recent announcement of opportunities request missions capped ~$0.5B, and the selection reviews are shown to have limited tolerance to risk. Small spacecraft continue to increase mission potential through evolutionary and revolutionary technology advancements for subsystems with reduced demands on limited spacecraft resources. The application benefits of the iodine Hall system can enable interplanetary missions for asteroid reconnaissance, missions to the moon, Venus, Mars as secondary payloads on frequent launch opportunities. Additionally, iodine Hall systems can enable high value science on daughter spacecraft or provide strategic deployment of multiple propulsion limited systems. The use of an iodine propulsion with small spacecraft "can deliver high-return missions that are cost-effective, quicker from concept to launch, and responsive to the present budget climate. They promise to revolutionize the way we carry out planetary science in the next century." [18]

C. Orbit Transfer Vehicles

Electric propulsion systems are routinely considered for orbit transfer vehicles (OTV). Long range exploration or even commercial space architectures leverage the use of electric propulsion stages and OTVs. 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 is of course limited by the performance of the on-board propulsion system, notably the specific impulse and packaging efficiency. The near-term focus has been on ESPA based options. 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 SmallSats or a large number of CubeSats to the moons of Mars or into a constellation for communication or observations at Mars.[19] Figure 11 illustrates an ESPA based orbit transfer vehicle using xenon propellant. 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. The change of propellant choice, recall the power processing units are identical and performance is comparable, 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.[20] Concept designs of an ESPA OTV for standard payloads (180kg) or CubeSat P-POD arrays is shown in figure 12.

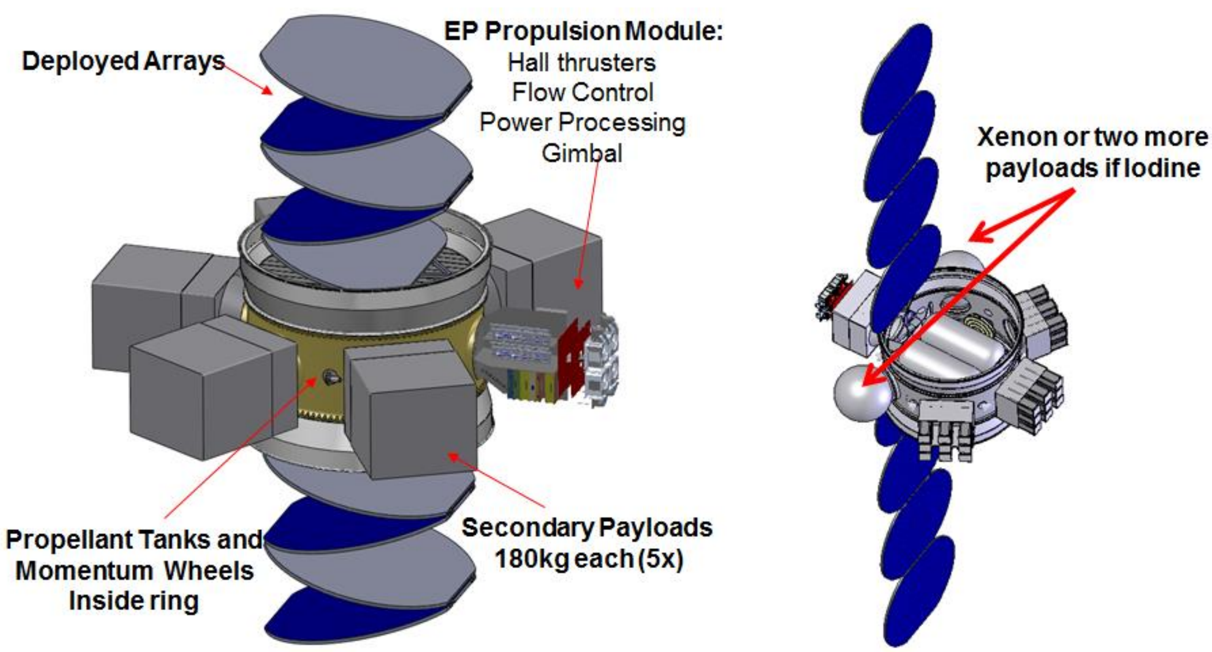

Figure 12. ESPA Based OTV concepts.

D. Exploration Class

Exploration class systems are not as constrained with respect to volume as SmallSats. Although some shroud options and propellant configurations can present packaging challenges; especially with very large >100kW solar arrays. The propellant tanks are expected to be significantly lower cost and lower risk to develop for the very low operating pressures. The propellant can be placed in more packaging efficient cylinders over domed pressure vessels beyond existing manufacturing practices. The GRC COMPASS team performed a large number of propulsion trades for comparison to the Mars Design Reference Architecture 5.0.[20] The results indicated a potential advantage for pairing a solar electric propulsion (SEP) chemical propulsion stage for a bimodal (high thrust / low thrust) option (SEP-Chem) without the need to develop a nuclear thermal rocket propulsion system.[21] The initial studies were based on an 800kW xenon SEP system. The COMPASS team completed a comparison if iodine was used in place of the xenon and several findings were made. First, the increase in storage density was never a driving constraint for the specific mission. The large SEP stage was driven by solar array packaging, but both the xenon and iodine options fit well within the smallest of shroud options. Second, the propellant cost and propellant tank development for iodine was estimated as a ~$400M total cost savings. Finally, the reduced mass of the iodine tank and support structure lead to 3,000kg mass savings, and with system gear-ratio; the total initial mass to LEO was reduced by more than 8t. Based on the system configurations, the savings also resulted in one less launch; assumed to be an automated transfer vehicle for additional cost savings ~$300M. After a detailed design of the crew vehicle and final performance of the Space Launch System is known, it is not clear if an 8t system would become a significant driver. The iodine system was the only propulsion option that fit entirely within two SLS launches; a habitat module and a SEP module. The mass savings between the xenon and iodine systems are provided in table 6, and the SEP crew vehicle concept design is shown in figure 13.

Table 6. Mass savings using iodine.

Mass summaries (kg)	Savings, kg (Xe vs Iodine)
SEP Piloted SLS Launch 1 - HAB Module Totals	
HAB Module Wet Mass	4983
HAB Module Dry Mass	1378
HAB Module Inert Mass	1743
HAB Module Total Chem Prop	199
HAB Module Xe Prop	3406
SEP Piloted SLS Launch 2 - SEP Module Totals	
SSEP Module Wet Mass	3431
SEP Module Dry Mass	1487
SEP Module Inert Mass	1795
SEP Module Total Chem Prop	1112
SEP Module Total Xe Prop	832
Vehicle totals	
Total Vehicle Wet Mass	8414
Total Vehicle Dry Mass	2865
Total Vehicle Inert Mass	3539
Total Chem Prop	1311
Total Xe Prop	4238

Two secondary observations are noted for the applicability of iodine for exploration class systems. First, one of the primary concerns with large throughput and operating time missions for iodine is the cathode. However, for these large scale systems, the small cathode propellant requirement could be a secondary xenon system for just cathode operation and would have minimal total system configuration impact. Second, and potentially a more significant advantage of iodine, is the impact on ground performance testing. Current test capabilities within NASA are already pushing the limits of what may be required for thruster testing and qualification, even for a 10kW class thruster. Operating at higher specific impulse helps significantly, but there are few viable options for ground testing 100kW class xenon thrusters and fully characterizing facility impacts on the system performance. The use of iodine propellant allows for a relatively low-cost option to use cold surfaces for iodine condensation and minimize facility effects. As NASA continues to pursue very high power electric propulsion options, sufficient performance and lifetime ground test conditions should be established for the determination of sufficient ground test capabilities.

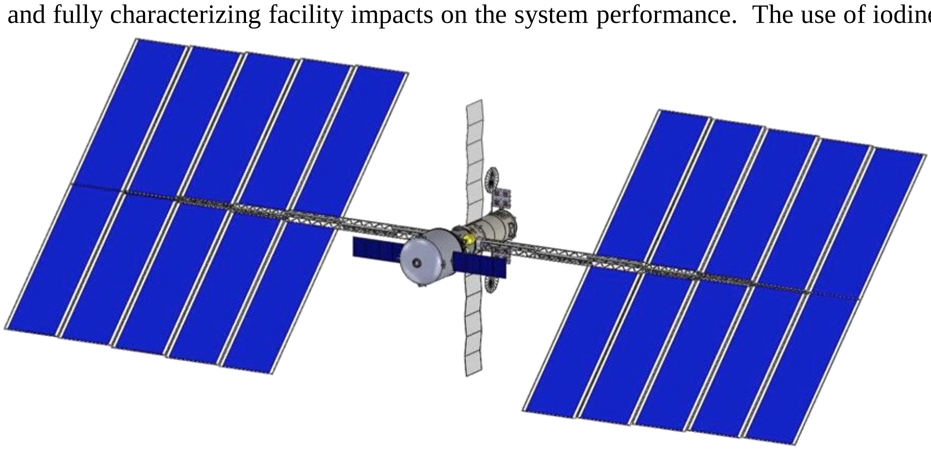

Figure 13. SEP crew nodule for Mars SEP/Chem option.

IV. Summary

Iodine has similar performance as the SOA xenon Hall systems, can leverage xenon PPU investments and has potential system level benefits primarily due to increased storage density and very lower operating pressures. Iodine Propulsion remains one of the key limitations for small satellite missions and iodine may be enabling due to the potential ΔV per unit volume. Additional advantages for secondary payloads include long term quiescence potential, no pre-launch pressurization required, and no special handing requirements. Iodine may enable a new set of planetary and exploration class missions by packaging sufficient ΔV within secondary payload constraints to enable GTO launched secondary spacecraft to transit to the moon, asteroids, and other interplanetary destinations for ~$150M full life cycle cost including the launch. 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. Exploration class electric propulsion systems may also benefit from iodine over xenon due to decreased propellants costs, propellant tank mass costs and system level mass impacts. Finally, iodine may also have a secondary benefit to the ground system performance and qualification approach for high flow rate system.

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