

Refueling Orbital Navigator for Active Debris Removal, and Asset Relocation, Recovery and Repair

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Abstract—There are two critical issues for in-space operations that would be addressed and potentially solved by the concept proposed here. These are: Active Debris Removal (ADR) and in-space maneuverability, e.g. providing mobility options to space assets, as well as in-space repair and reconstruction of disabled assets. There are many challenges associated with in-space maneuverability. Due to the high delta-V requirements, conventional chemical or propellant based systems are implausible, and only a reusable system is capable of keeping costs reasonable. It demands a specialized spacecraft that can maneuver between many target objects, rendezvousing, capturing and transferring assets into new orbits, or deorbiting unwanted space debris. Such an Orbital Transfer Vehicle (OTV) requires the ability to perform hundreds of km/s in delta-V over its mission lifetime. An OTV that can utilize in-situ upper atmospheric resources would enable a host of NASA and DoD missions that require extremely high delta-V in a fast, responsive, and repeatable manner. The Refueling Orbital Navigator (RON) spacecraft accomplishes this in the following way: first it employs a highly elliptical orbit to obtain fuel by scooping up and storing the molecular nitrogen and atomic oxygen encountered during the low altitude perigee periods of its orbit. As RON can thrust at apogee, it can achieve the extensive orbit lowering needed for ADR. Additionally, RON can thrust at perigee to provide drag compensation for low perigee refueling, stable non-Keplerian orbits, or rapid phase changes. In a recent effort at MSNW, funded by NASA's NIAC program, the design and the initial testing of the RON concept was successfully performed. It was determined that such a revolutionary system involved three key elements. The first was the adoption of Elliptic Collection Orbits (ECOs). In this way the collection of ambient atmospheric gases for propellant could be decoupled from the constant need for drag make-up that has effectively doomed prior efforts at air-breathing propulsion which typically employed circular orbits. The ECO limits the atmospheric drag to a brief period of time at perigee, but allows for solar energy accumulation and storage over the entire orbital period. This, combined with modern lightweight solar panel technology, make up the second critical element by creating a unique opportunity for lightweight, efficient, and high-power electric propulsion (EP) such as the Hall thruster to be employed. Finally, a highly efficient propellant collection system demanded by a refueling OTV can be achieved by a properly designed low-drag scoop geometry coupled to an active collection and storage system that employs turbo-molecular pumping. Such a collection and storage system has been devised and several key elements have been fabricated and successfully tested. Initial orbital dynamics calculations are presented as well as a sample mission analysis for the removal of a metric ton satellite from orbit.

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1. INTRODUCTION

Successful realization of the RON concept would have a major impact on several NASA mission objectives. An orbital vehicle that could utilize in-situ upper atmospheric resources would enable a host of missions that require extremely high delta-V in a fast, responsive, and repeatable manner. The concept to be described here, the Refueling Orbital Navigator (RON) could provide such capability. RON accomplishes this in the following manner: first it obtains fuel by scooping up and storing the molecular nitrogen and atomic oxygen encountered during low altitude perigee periods of the highly elliptical orbits. As RON can thrust at apogee, it can achieve the extensive orbit lowering needed for Active Debris Removal (ADR). Additionally, RON can thrust at perigee to provide drag compensation for very low perigee refueling, stable non-Keplerian orbits, or rapid phase changes. As such, the RON concept can also find application to other planetary missions.

The ability to capitalize on In Situ Resource Utilization (ISRU) for electric propulsion on the next generation exploration class missions would have a direct impact on potential planetary mission feasibility, scale, and cost. A refuellable orbital navigator employing electric propulsion would enable detailed studies of the planetary bodies from Venus out to Neptune. For the ADR mission, chosen for the focus of this study due to its extreme importance, the requirements for the RON are modest. RON requires only 5 kW of on-board solar power as energy collected during the higher altitude portions of its elliptical orbit can be stored for higher power operation later. Operating in this manner RON would generate 1.2 N of thrust with a specific impulse, I_{sp} ,

of 2500 sec. for ADR. During a 10-year mission life, RON would process 5500 kg of propellant to ferry 80 spacecraft, perform 850 degrees of plane change, with over 100 km/s of delta-V, all with a single spacecraft launch, and requiring no additional onboard propellant. The ADR mission and how RON would enable an affordable method for reducing orbital debris will be outlined in the next section. In addition to ADR, RON's capabilities would not only prove advantageous for near-term ADR, but also open up a whole new system architecture for carrying out endeavors in space from launch and satellite installation, to orbit adjustments and satellite servicing, to post-mission removal. Most significantly however, RON could enable the exploration of other planetary bodies in much greater depth, as well provide a way to utilize planetary resources that would otherwise not be feasible. Section two outlines the requirements and challenges of a Refueling Orbital Navigator. Section three discusses the progress made in the initial effort, and discusses future work to accomplish the next steps in achieving a practical RON.

The Orbital Debris Crises

As of January 2018, an estimated 8,100 tons of space debris has accumulated in low Earth orbit consisting of spent rocket bodies, mission-related debris, and collision fragments. The vast majority of these objects are too small to detect with radar systems, but there are over 29,000 known objects larger than 10 cm. [1]. As of 2020, the United States Space Surveillance Network was tracking more than 14,000 pieces of space debris larger than 10 cm across. Impacts between these objects and operating missions have damaged costly equipment, required expensive collision avoidance maneuvers, and endangered the lives of astronauts on the international space station. In 2009 the accidental collision between the Iridium 33 and Kosmos-2251 satellites created nearly 1500 kg of debris, resulting in a 30% increase in the collision probability to intact objects in low earth orbit (see Fig. 1). This incident illustrates how catastrophic future collisions can be, and has prompted an increased attention to dedicated debris removal missions

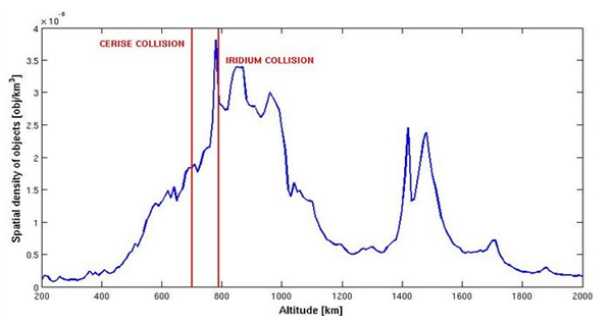


Figure 1. Orbital debris concentration by altitude

In 1979 the NASA Orbital Debris Program Office, in conjunction with Donald J. Kessler, released research on the “Kessler Syndrome” which predicted that collisions would continue to increase. This would lead to an exponential growth in debris that would render access to space impossible within several generations. A partial solution to stabilizing the debris population was also proposed which required new

missions to incorporate post mission disposal measures, as well as missions dedicated to ADR by placing the largest objects into decaying orbits of less than 25 years [2].

There are many issues associated with ADR. Most important is the economics of such a mission. Due to the high ΔV requirements, conventional chemical or propellant based systems are implausible. To remove the significant number of debris needed to stabilize the space environment, a reusable system is necessary to drive costs down. This demands a specialized spacecraft that can maneuver between many target objects, capturing each and transferring them into terminal orbits. Such an Orbital Transfer Vehicle (OTV) [3] requires the ability to perform hundreds of km/s in ΔV over its mission lifetime. However, even with the most fuel efficient high-Isp device, total propellant mass scales prohibitively for long, multi object deorbiting missions. As a solution to this problem, a number of propellantless technologies have been suggested for ADR. Due to the lack of success thus far, and the fact that extensive tests in the actual space environment still need to be performed, it appears that these large structures are not the best near-term solution. Moreover, long range solutions, such as lasers, can easily yield significant amounts of additional debris if target conditions and attitude are not well known.

Proposed Solution to ADR

With regard to ADR, the largest concentrations of high mass debris are found in the inclination band of 81 to 83 degrees and in the altitude range of 800 to 1300 km (see Fig. 2) [4]. The large concentrations of high mass debris in this inclination band, and in the altitude range of 800 to 1000 km, indicate that these regions of space should be the first targeted to reduce the overall collision probability in the space environment. Thus, the initial effort focused on calculations for retrieval with a characteristic mission in which an object is retrieved at 950 km and 82 degrees inclination. It should be noted that the GEO region is also a likely candidate for debris removal as well. A spacecraft design for RON should be flexible enough to be applicable to missions in both regions.

High mass objects are the most critical objects to remove, which leads one to the conclusion that intact satellites and

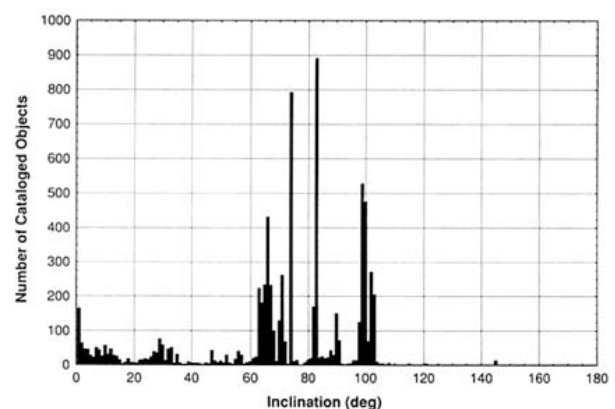


Figure 2. Distribution of trackable debris by inclination.

rocket stages will be the main targets initially. This choice also ensures the ability to track debris targets accurately via the U.S. Space Surveillance Network and the Haystack X-Band Radar facility to ensure efficient and safe rendezvous maneuvers. It is objects in these regions of space that will first be targeted for the RON system. A method of capture on the spacecraft could address a specific type of debris object, but it is recognized that the ability to capture different kinds of debris would significantly increase the mission's repeatability and economy. For this reason, one is driven to find the most all-encompassing capture solution which will be discussed in section four.

The initial study focused on the mission analysis and orbit calculations for the retrieval of the more massive objects at a range of altitudes centered about 950 km and 82 degrees inclination. Work was also carried out to experimentally determine the properties and behavior of a promising scoop design for RON. While the results were adequate to satisfy the necessary conditions for a RON, more optimal designs, as well as a more detailed analysis and further experimental verification tests will be needed in future efforts.

The following section outlines the use of RON to ferry debris objects into terminal orbits for ADR using nothing but ambient gaseous resources.

2. ANALYSIS OF THE REQUIREMENTS FOR THE RON SPACECRAFT

In order to perform a characteristic mission study for the rendezvous and removal of large space debris, the primary physical constraints on the spacecraft demanded by feasibility, as well as the nature of the space environment that is to be utilized by RON, must be taken into account. This can be broken down into the various elements that make up the RON system. In particular, the three key elements of; (1) the fuel collection "scoop" and compression assembly, (2) the propulsion system requirements, and (3) the power system required. These three elements will now be discussed. Based on reasonable assumptions for the performance in these three critical areas, a mission architecture can, and was, formulated to describe how the RON system would work as a solution to ADR.

(1) *Collection Scoop* - The scoop size for RON was chosen to be 10 m² for this preliminary system estimation based on typical fairing sizes assuming no deployment. The other performance metric that was fixed to a reasonable value was the power of the propulsion system. As discussed in the next subsection, the peak power of the propulsion system was assumed to be 30 kW, which is within the range of Electric Propulsion (EP) thrusters already developed for space application. Equation (1) below reflects the relation of Power, P, I_{sp}, and Thrust, Th, of the EP thruster.

$$Th = \frac{2\eta P}{I_{sp}g_0} \quad (1)$$

Here η is the thruster efficiency and g_0 is the gravitational acceleration on Earth of 9.81 m/s. The primary purpose of the

thruster system is to perform drag make-up at perigee. Without enough thrust to counteract these drag forces, RON would assume the same fate as the deorbiting target. To arrive at a reasonable estimate for drag, it is assumed that the scoop stops all incoming atmospheric particles, or

$$F = \dot{m}V, \quad (2)$$

where V is the velocity of the spacecraft and $\dot{m}$ is the mass flow rate of the incoming particles. which can be expressed as:

$$\dot{m} = \rho VA, \quad (3)$$

where ρ is the mass density of the atmosphere and A is the effective spacecraft contact area. By substituting Eq. (1- 3), into Eq. (2), and setting the thrust of the propulsion system equal to the drag force, a mass density can be determined as a function of the thruster parameters, as shown in Eq. (4).

$$\rho = \frac{2\eta P}{I_{sp}g_0V^2A} \quad (4)$$

While both density and orbital velocity are a function of altitude, only density varies significantly with altitude as can be seen in Fig. 3. A good approximation for the RON velocity at perigee is 8 km/s. It will be assumed that the EP thruster, as described in the next subsection has a power of 30 kW with an efficiency of 50% and a specific impulse, I_{sp}, of 2500 s. These are readily achievable parameters for an EP thruster such as the Hall Effect Thruster (HET). While the HET efficiency is higher than 50% when operated on Xenon, it is expected to be lower operating on atmospheric gases, which from Fig. 3 will be primarily N₂ (60%) and O (30%). It will likely take some development to bring the efficiency of the HET to a higher level due to the greater ionization loss per unit mass associated with the lighter ambient gases. Assuming one can achieve these thruster performance characteristics, it can be determined that an atmospheric mass

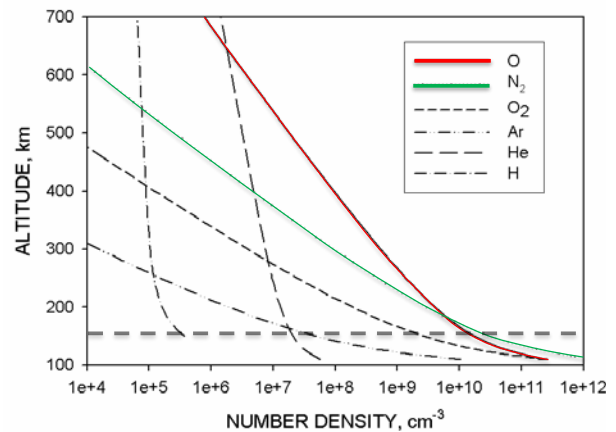


Figure 3. Number density of atmospheric species as a function of altitude. The horizontal dashed line indicates the perigee of the RON spacecraft during propellant collection.

density of $2.3 \times 10^{-9} \text{ kg/m}^3$ is necessary for drag makeup. This density corresponds to an altitude of 151 km.

The collection scoop has one of the lowest Technology Readiness Level (TRL) of any subsystem, but is probably the

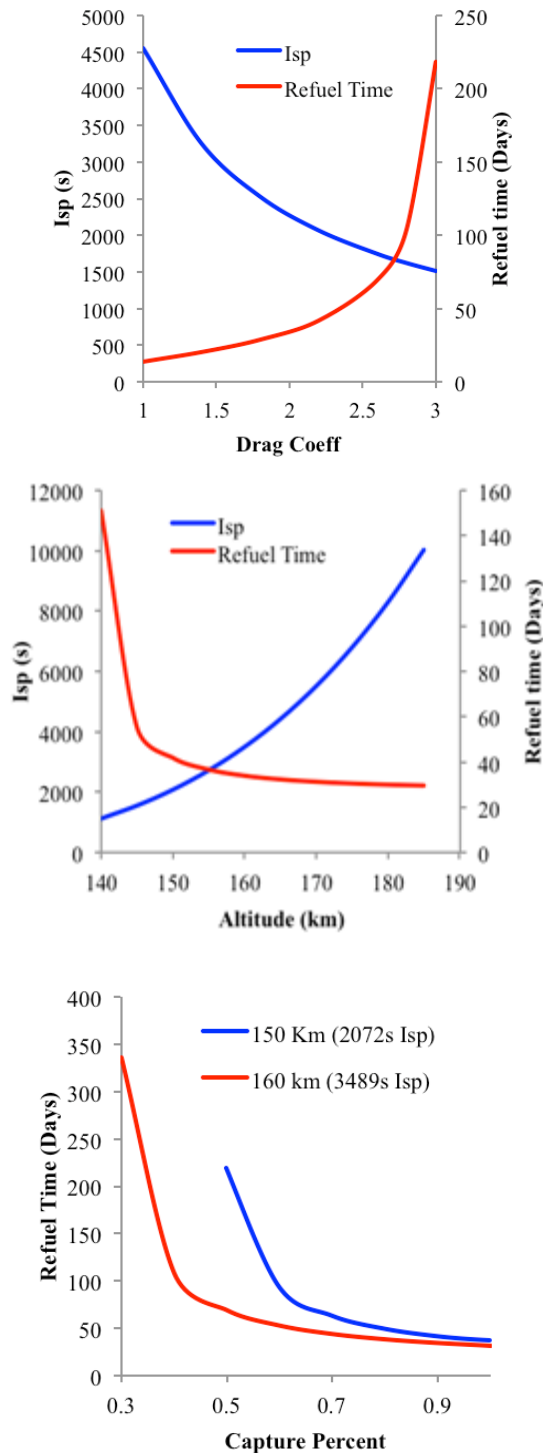


Figure 4. (Top) Impact of the scoop drag coefficient on required Isp and refuel time. (Middle) Isp and Refuel time as a function of altitude. (Bottom) Refuel time as a function of capture percent for 150 km and 160 km perigee orbits.

most critical for a propulsion system like RON. Much more will be said about this later. For the preliminary mission analysis to be considered here, it will be assumed that RON has a typical drag coefficient, C_D , of 2 and that the scoop's capture efficiency is 80%. The significance of these design parameters on the overall mission and spacecraft design is illustrated in Fig. 4 which indicates the impact of C_D and I_{sp} on refueling times. For a fixed power, the higher the C_D , the greater the drag force, and the greater the need to drop I_{sp} to increase the thrust. It also follows that as I_{sp} drops, propellant mass requirements increase. This is problematic because as more propellant is needed to achieve the desired thrust, less is stored and refueling time increases dramatically. It is also important to note that a C_D above 2.5 at the target altitude of 150 km for perigee begins to increase mission time significantly. Figure 4 also shows the relative refueling time for various scoop capture efficiencies. For the assumed drag coefficient of 2 and an altitude of 150 km, the minimum capture percentage is roughly 50%. Below that, RON is less able to take in enough ambient gas to perform the necessary drag makeup. It is also shown that an altitude of 160 km it would be advantageous to have a higher I_{sp} of 3500 s (more on the relationship between I_{sp} and altitude will be found in the next subsection). It is worth noting here that for high capture percentages, refueling times level off and are not reduced significantly with increasing I_{sp} . The opposite is true at lower capture percentages. At 50% capture efficiency and 2000 s design, it would take 220 days to refuel, while a 3500 s design would only take 68 days, a reduction of 69%. A high I_{sp} design would allow for operation at lower capture percentage of 30 to 40%, if need be, but the mission time would increase. In this way, drag can be made up while maximum mass efficiency can be maintained. The plots in Fig. 4 clearly illustrate why optimizing scoop design for minimum drag and maximum capture efficiency is critical for reducing mission time and costs.

(2) - *Propulsion System* As of now there is no clear winner as to the best EP thruster for the RON as most of the current thrusters have been developed to operate on Xenon. There is

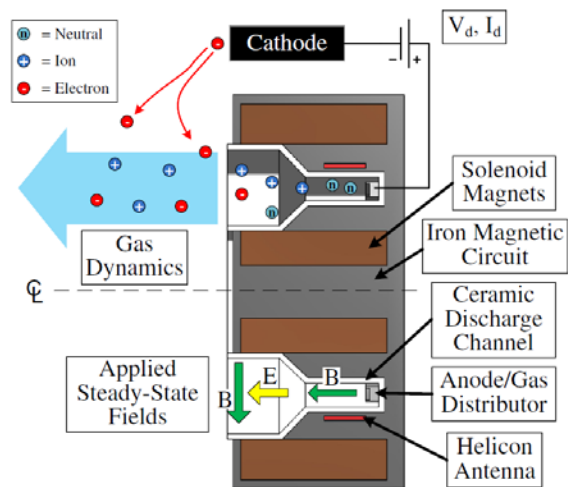


Figure 5. Notional diagram of HHT operation. (From Ref 6.)

in principle no reason that the leading EP candidates – the HET and the RF based Ion Engine (RIT), both of which have been developed to TRL 9, could not be adapted to operate reliably and efficiently on atmospheric propellants. In fact, there has been recent efforts to do just that [5,6]. As the studies are somewhat preliminary, it is difficult to get much in the way of performance data from these devices on atmospheric propellants.

The HET is a device that is of the right size and power so its performance will be considered here. As operation of the device on reactive gases such as Oxygen present problems with rapid cathode erosion [7], it is important to develop alternate methods for initiating the ionization process. One promising candidate for this role is the Helicon discharge. In fact, a Helicon Hall Thruster (HHT) [6] was built and tested

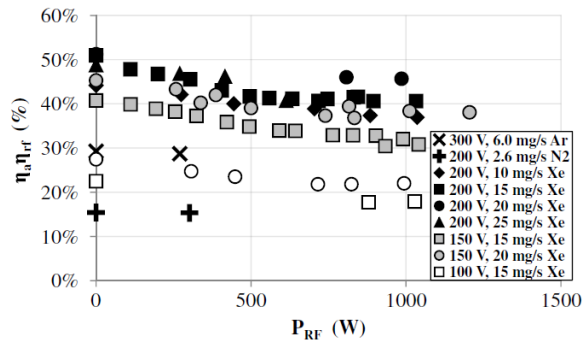


Figure 6. Total anode efficiency vs RF power (from Ref. 6).

on N₂ and Argon as propellants. The principles of operation can be found in Fig. 5. The results from the testing of the thruster with the Helicon addition did not really improve performance as can be attested to in Fig. 6. The RF (Helicon) power actually decreased efficiency as the power was increased, but it should be noted that if the idea is to reduce the damaging effects of oxygen ionization, the HHT could potentially do that.

Operating a standard HET on atmospheric propellants was also performed by a collaborative effort between ESA and Alta [5]. The HET was operated on N₂ as well as a mixture

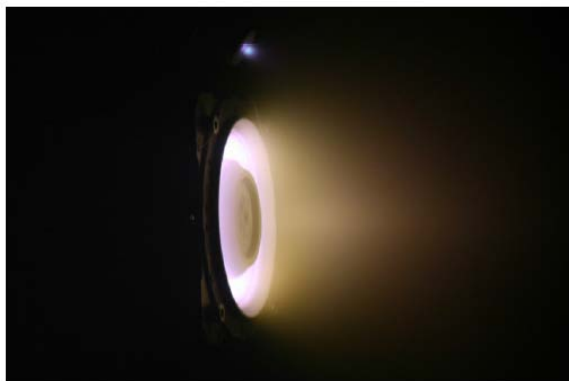


Figure 7. Thruster operating with N₂/O₂ mixture with an anode voltage of 305 V, and a flow rate of 2.75 mg/s. (from Reference 5.)

of N₂ and O₂ with the latter showing somewhat improved performance, which is a positive. A picture of the thruster operating on the N₂-O₂ mixture is shown in Fig. 7. The test results for both the HET and RIT thrusters showed a substantial deterioration of performance with respect to Xe. This is no doubt due to the low ionization efficiency of the atmospheric propellants. However, the results did indicate that both devices were capable of continuous and stable operation on atmospheric propellants. It is difficult to get direct information on the performance as only thrust and electrical operational parameters were published. As can be seen in Fig. 8 for operation of the HET on the N₂-O₂ mixture, the thrust was a strong function of power. The maximum thrust, Th = 30 mN, was obtained at roughly 1 kW input with a flow rate of 2.7 mg/s. This was roughly half of that of the Xenon performance. If one assumes roughly half the efficiency of the Xenon results, one has from Eq. (1-1) that the Isp = 2000s. Since the target thruster power for the RON is 30 kW, these results are far from the operating conditions required for RON. Ionization efficiency and thruster performance quite often get better as the power is scaled up as attested even by the data shown here. It could very well be that that the target of 50% efficiency operating on atmospheric propellants can be achieved at sufficiently high Isp for optimal performance of RON at the power levels desired.

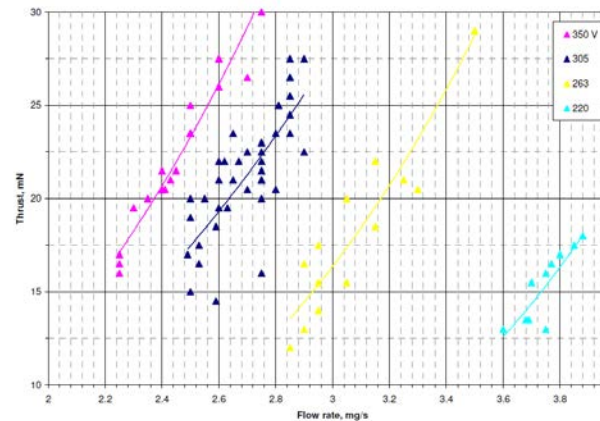


Figure 8. Thrust vs. flow rate with discharge voltage as parameter (from Ref. 5).

(3) - *Power System* One of the major design challenges with air-breathing propulsion systems is the balance between power and drag. Large amounts of power are needed to produce enough thrust to counteract the drag of the spacecraft. The more power is needed, the larger the solar arrays become and the more drag is produced – overall, a losing cycle. However, by remaining in highly elliptic orbits, RON only fires its thruster for relatively short time periods and therefore can use a significantly smaller solar array for power, and thereby reduce drag. The solar panels can be therefore be sized such that the scoop cross section is the largest cross section, and only the scoop will create drag while collecting and storing propellant. In order to employ this technique, the spacecraft will need batteries for perigee operations in the terminal orbit. With an energy density of 900 kJ/kg, a space developed lithium-polymer [9] battery

system would weigh 41 kg, assuming a 50% discharge cycle. This, as will be seen later, is a small fraction of the mass of the RON spacecraft.

The assumed EP system was set at 30 kW, but it will only be operated at that power level for approximately 10% of each orbit, either for drag makeup at perigee or walk-in/walk-out maneuvers at apogee. A 3-kW solar array would therefore be all that would be needed to power the propulsion system. There is however an estimated additional 1 kW of power needed to run spacecraft subsystems such as the turbopump fuel compressor, and the debris deorbiting and control system. At this early stage of design an additional 25% is added as a margin so that the target RON power system will be scaled to a 5-kW solar array.

Mission Architecture

As an example of the characteristic RON mission, a 1000 kg (1 mt) Russian Cosmos class satellite at 950 km will be described. Each retrieval and disposal mission can be divided into four stages. These stages are depicted in Fig. 9, and will now be discussed in more detail.

Stage 1: RON will rendezvous with the target. During this stage RON will assess the dynamics of the debris and the best way to interact with it. Various techniques and options for rendezvous need to be evaluated. It is assumed that there will be extensive demands on the attitude and control systems (ACS), and the propellant mass budget for such maneuvers must be evaluated. In order to move the object in a stable and accurate manner the thrust vector must be applied through the center of mass. Since 70 percent of orbital space debris has no spin, or spins around a nadir pointing axis, it may be advantageous to induce a small spin to the desired target object to aid in stabilizing the motion.

Stage 2: RON will ferry the target to an elliptical termination orbit using onboard fuel. Once there it will release its cargo and start firing at perigee to perform drag make-up. At full power the thruster produces a thrust of 1.5 N, which is equal to the drag force at 151 km for a 1.8 m radius scoop assuming a drag coefficient of 2. The ΔV needed to change from a 950 km circular orbit to an elliptical orbit with a 950 km apogee and 151 km perigee is 223 m/s. For a 1 MT target and an estimated 500 kg RON, this ferrying maneuver will require 15 kg of propellant using a 2,500 sec I_{sp} thruster. At 1.5 N thrust, burning at 10% of the orbit period, RON will take approximately 9 days to ferry to the necessary terminal orbit. The 1 MT satellite example has a lifetime of 76 days in this terminal orbit before atmospheric reentry due to ambient atmospheric drag.

Stage 3: RON will refuel for several orbits using ambient propellant to counteract drag and store fuel for future use in an orbit with an apogee of 950 km and a perigee of 150 km. RON's total mission propellant requirement is 20 kg. With a 151 km perigee it will take 32 days to refuel to 20 kg. Refueling will use a scoop and multi-stage compressors to trap and compress the ambient atmosphere into a storage tank. Reorientation maneuvers can also be done from this

terminal orbit. Inclination changes and right ascension plane changes require high ΔV , and therefore have a high propellant requirement, and this is why RON is so well suited for ADR.

Stage 4: RON will exit the terminal orbit and rendezvous with the next target. As the next target is at a similar altitude, another 223 m/s of ΔV will be required. It is estimated that to move RON into a proper orbit and new position will require 3.4 kg of propellant over 2 days.

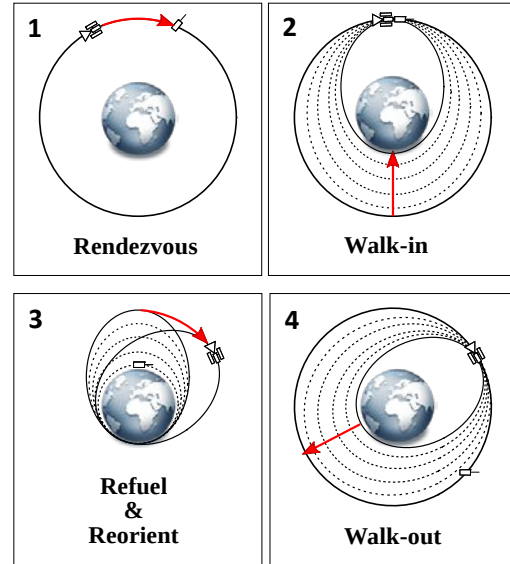


Figure 9. The four mission stages of the RON system

Table I. Stage characteristics

Stage	Operation	Propellant requirement	Time, hrs (days)	Delta-V (m/s)
1	Rendezvous	1.82	24 (1.00)	117
2	Walk in	14.80	222 (9.26)	223
3	Refuel & Realign	NA	771 (32.13)	N/A
4	Walk out	3.41	51 (2.14)	223
TOTAL		20.03	1069 (44.53)	563

Estimates of propellant mass and time requirements for each stage are shown Table I. In total, RON will be able to deorbit the 1000 kg target using 20 kg of propellant in 45 days. Even more impressively, over a 10 year operational period, RON will process greater than 5500 kg of ambient propellant and deorbit 80 debris objects with a high probability for a disastrous collision such as the Iridium 33 and Kosmos-2251 event mentioned earlier.

RON for Exploration and Resource Utilization on Other Planetary Bodies

Virtually all planets, as well as several planetary moons of interest, have a substantial atmosphere. Therefore, it goes without saying that a spacecraft that could be sent to these bodies, that had the ability roam virtually anywhere over the surface of the planet taking data, sending out probes, collecting samples, etc. and limited in time only by the survivability of the spacecraft hardware, would be a game-changing advance [10]. Such would be the case for RON if the requisite science and technology as anticipated by this study can be realized.

The key determining factors in assessing the practicability of using a REfuelable Planetary Orbiter (REPO) as RON is for Earth's ADR are as follows:

(1) the orbital velocity at an atmospheric height characterized by molecular flow (i.e. particle densities $\sim 10^{17} \text{ m}^{-3}$), which is typically in the thermosphere of most planets, must be much less than the Isp of the thruster that will propel the spacecraft, as this will be the inlet flow speed that must be made up by the thruster during particle collection. A plot of the various planetary atmospheric density profiles can be found in Fig. 10. It can be seen that the Earth's nearest neighbors will have a very similar operational perigee with regard to the REPO orbital dynamics. The atmospheric temperature conditions on

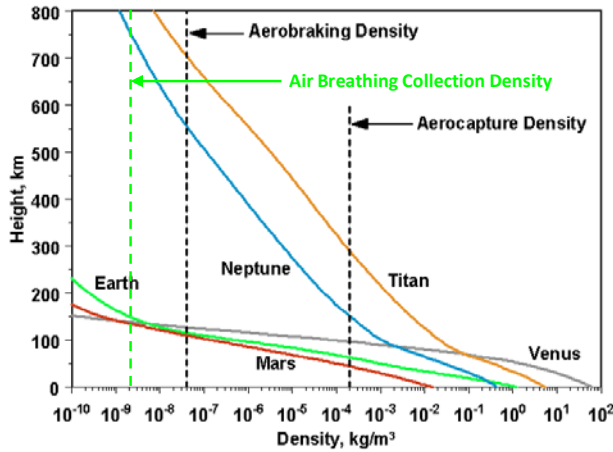


Figure 10. Comparison of density profiles for several planets and Titan.

the planetary bodies of interest are also quite similar as can be seen in Fig. 11 even for the more distant gas giants and their moons. Earth is unusual in that the ambient temperature becomes much higher at high altitude due to the absorption of solar radiation by oxygen. The directional flow experienced by the spacecraft represents a far higher “temperature” especially as this flow is stagnated thereby thermalizing the collected gas with a characteristic thermal velocity similar to the spacecraft orbital speed. There will be more to say about this later in the discussion of the scoop/collector physics. Needless to say, the ambient atmospheric temperatures where the REPO would reside represent no real issues.

(2) The chemical makeup of the atmosphere must be conducive to being used as an EP propellant. A listing of the most likely planetary bodies to be explored and the relevant parameters for a REPO can be found Table II. It can be seen that in terms of drag make up, both Mars and the Jovian moon Titan - two of the most compelling bodies for further detailed exploration - are actually much easier for a REPO to

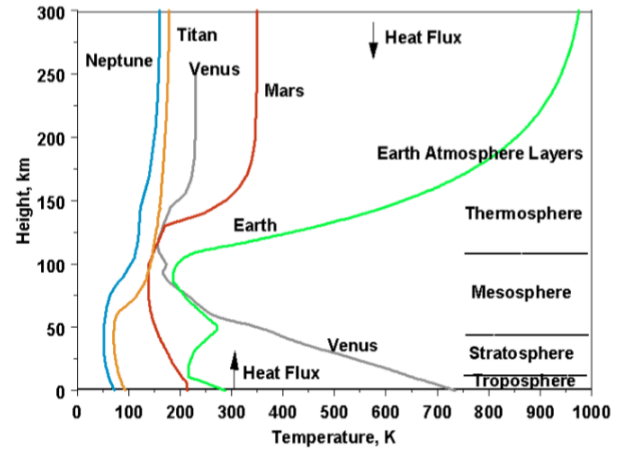


Figure 11. Comparison of atmospheric temperature profiles among the planets and Titan.

negotiate than Earth as the relative atmospheric flow speed is less than half for Mars and a quarter for Titan. Since the drag force scales as V_{flow}^2 , a Martian REPO can operate with a collector and EP system with a significantly lower efficiency threshold than RON requires. In addition, the propellant gases on Titan are more amenable to cathode-based EP as its atmosphere is primarily nitrogen with no oxygen containing molecules.

From the point of view of rare gas retrieval, the gas giants could provide a source of the Helium isotope ^3He which is roughly 0.01% of the total Helium content of the gas giants' atmospheres. It is extremely rare on Earth and is highly desired for several applications such as nuclear detection, cryogenics, medical imaging and most crucially, for use as nuclear fuel to exploit the aneutronic reactions $\text{D} + ^3\text{He} \rightarrow ^4\text{He} + \text{H} + 18.3 \text{ MeV}$ and $^3\text{He} + ^3\text{He} \rightarrow ^4\text{He} + 2\text{H} + 12.86 \text{ MeV}$ for nuclear power generation. In fact, several schemes have been proposed to obtain it from mining the lunar regolith [11] to collecting it from the Jovian atmosphere using

Table II. Candidate planetary and atmospheric data for a REPO mission.

Planet	Radius (10 ⁶ m)	Mass (10 ²⁴ kg)	Orbital Velocity (m/s)	Orbital Period (hr.)	Height where $\rho \sim 1\text{e-}9 \text{ kg/m}^3$ (km)	Principal Gases (%)
Earth	6.38	5.97	7,900	1.4	170	N2 [78] O2 [21] Ar [1]
Mars	3.4	0.642	3,549	1.7	135	CO2 [95] N2 [3] Ar [1]
Venus	6.05	4.87	7,327	1.4	135	CO2 [96] N2 [3]
Jupiter	71.5	1898	42,078	3.0	550	H2 [90] He [10]
Saturn	60.3	568	25,066	4.2	950	H2 [96] He [3]
Uranus	20.6	86.8	16,764	2.1	800	H2 [83] He [15] CH4 [2.5]
Neptune	24.8	102	16,563	2.6	810	H2 [80] He [19] CH4 [1]
Titan	2.57	0.134	1,865	2.4	900	N2 [96] CH4 [3.5] H2 [0.5]

a nuclear-powered spacecraft [12]. When one considers the extreme difficulty for either of these approaches, a REPO mission, where gas collection is the essence of the concept, looks considerably more feasible. Jupiter may not actually be the best planet for collection due its deep gravity well and high thruster Isp requirements. As can be seen from Table II, Uranus and Neptune have the largest Helium fractions at much lower orbital velocity. Finding an EP thruster that can operate on Hydrogen is actually easier than one might think. In fact, there already is an EP propulsion scheme developed at MSNW as part of its fusion research that was designed to operate on Hydrogen isotopes at high efficiency while generating high density plasmoids that were accelerated to velocities that ranged from 200 to 400 km/sec [13,14]. This thruster could supply the high I_{sp} required to explore and refuel in the atmospheres of the gas giants.

Alternate Electric Propulsion Thrusters for REPO and RON

Given the chemically active and corrosive nature of the atmospheric gases found on both Earth and other planetary bodies of interest, it is useful to briefly consider alternative EP thrusters such as the Magnetically Accelerated Plasmoid (MAP) thruster and the Pulsed Inductive Thruster (PIT) that are electrodeless and maintain magnetic separation of the plasma from the thruster body. While these devices are less developed than the Hall or Ion thruster, their exceptional suitability for the REPO or RON would demand further development.

The PIT and MAP thruster are two of a type of electromagnetic propulsion devices that can be described collectively as pulsed inductive accelerators. Inductive plasma accelerators are attractive as propulsive devices for several reasons. The lifetime and contamination issues associated with electrode erosion in conventional plasma thrusters (Ion, Hall, and MPD) do not exist in these devices where the discharge is inductively driven. In addition, a wider variety of propellants (e.g., atmospheric gases, ammonia CO_2 , H_2O) can be used as compatibility with metallic electrodes is no longer an issue. Moreover, pulsed inductive accelerators (indeed, pulsed thrusters in general) can maintain the same performance over a wide range of input power levels by simply adjusting the pulse rate. With pulsed inductive plasma accelerators, energy is stored in a capacitor and then discharged through an inductive coil (or a series of coupled coils as in MAP). These axisymmetric devices are electrodeless, where the large currents are induced in a plasma located some distance away from the induction coil. The plasma is accelerated axially and expelled at a high exhaust velocity through the interaction of the induced azimuthal plasma currents and the radial component of the driver coil (s) magnetic field.

The two concepts considered here, MAP and PIT, have experimentally demonstrated the capability of operating at potentially multi-megawatt power levels. Both the MAP thruster and PIT have been operated at 1 kJ up to several 10s of kJ energy levels for single pulses. In previous experiments the acceleration of the FRC plasmoid was primarily for fusion studies [13,14]. These experiments have produced FRCs that

would translate to thrust powers many times that of the PIT device, and at an $I_{sp} = 30,000$ sec.

The basic physics behind these devices is quite simple. If one places a flux excluding body (e.g. a copper cylinder) inside a flux preserving coil, or near a planer coil, the original “vacuum” magnetic flux created by the coil when pulsed is excluded from the conducting cylinder volume and is compressed in the space between it and the coil thereby increasing the magnetic field there. It takes energy to compress this field and it is this energy that is then released in the form of kinetic energy of projectile motion as it is ejected due to the strong radial magnetic field gradient at the coil exit. This is actually nothing different from what is observed in a magnetically actuated piston or a coil gun.

Magnetically Accelerated Plasmoid (MAP) Thruster—T -For space propulsion this conducting body is a plasmoid, and it is formed *in situ* inside the magnetic coils by inducing a strong azimuthal current that excludes the external coil field. Things are a bit different here as the plasma is compressible, but the results are still governed by energy conservation. The method for creating the plasma in the MAP thruster was to rapidly reverse the voltage applied to the coil and thus the direction of the external magnetic field which, by Lens’ law, generates the necessary currents to exclude the external field. This type of device for plasma initiation and formation is referred to as field-reversed θ -pinch and the plasmoid formed is commonly

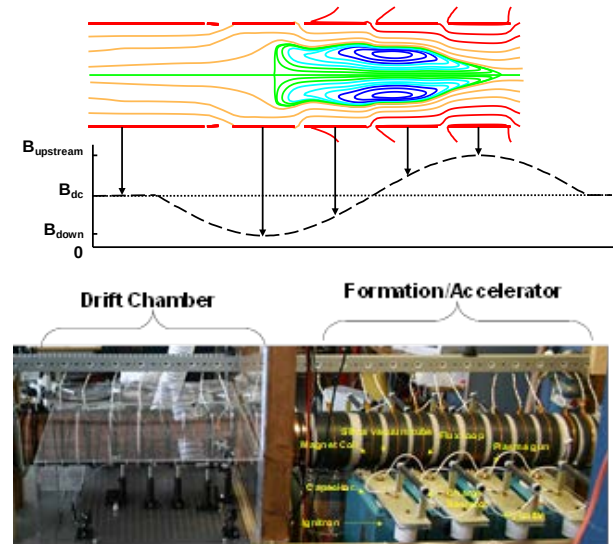


Figure 12. (Top) FRC acceleration method – a propagating magnetic wave. (Bottom) MAP thruster and Drift Chamber at MSNW. The top reflects 2D MagnetoHydroDynamic (MHD) fluid code calculation of the Field Reversed Configuration (FRC) plasmoid acceleration process from an array of cylindrical coils capable of independent actuation. These are arranged axially as in the lower picture of the MAP. The middle plot illustrates phasing of the accelerator coils. Each coil in turn is switched on for one complete cycle. The phase of each coil at time of calculation is indicated by arrow. Energy that is not transferred to FRC is returned to capacitor for the next pulse.

referred to as a Field Reversed Configuration or FRC. The formation of the FRC and the ejection process can actually be combined by sequencing the magnetic field reversal of a series of short coils staged so that the FRC continually experiences a constant and strong radial gradient behind the FRC all along the entire axial length of the coil (see Fig. 12). This B_r crossed with the induced I_θ in the FRC produces the

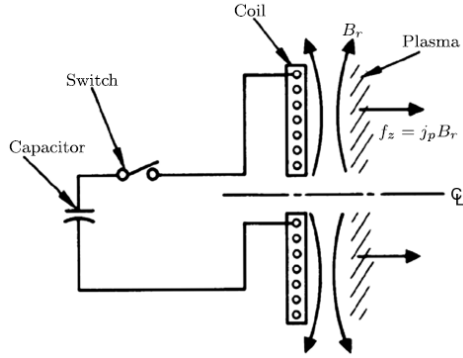
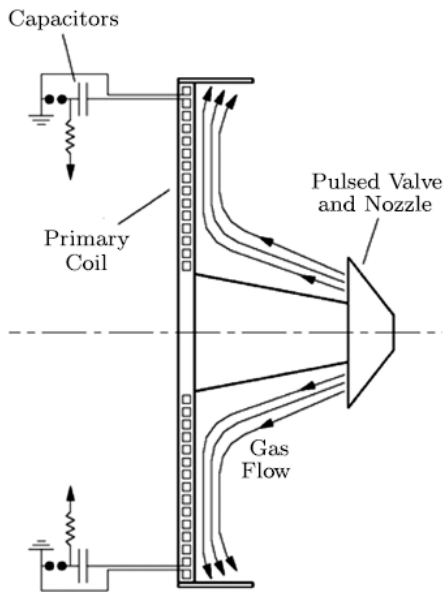
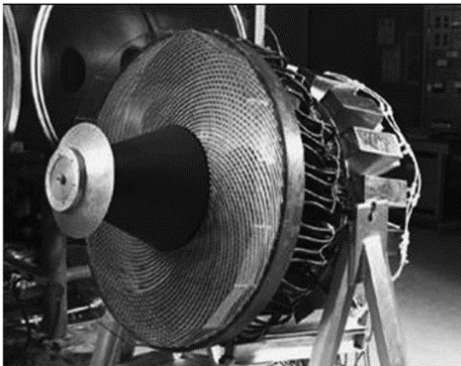


Figure 13. (Top) Schematic showing the basic operation of a planar PIT where j_p is the azimuthal plasma current density. (Middle) Schematic showing a typical propellant injection scheme employed on PITs (Bottom) picture of the PIT MK V. (from Ref. [17]).

strong axial force F_z desired. The MAP thruster concept was validated in experiments at MSNW [15] where a prototype thruster was built and tested. A picture of the test setup can be found in Fig. 12. The terminal velocity of the FRC was found to be 180-240 km/sec ($I_{sp} \sim 18,000-24,000$ s) operating with a Deuterium propellant. The efficiency of the thruster can be quite high ($\eta \sim 0.8-0.9$) despite the low atomic mass of Deuterium (2 amu). There are several reasons for this. (1) there are relatively low ionization losses as the electron temperature is comparable or higher than the ionization energy and is not limited by line radiation loss from the full stripped Hydrogen ions. This is the primary radiation loss mechanism for all moderate to high mass propellants like Xenon. (2) The very high ratio of directed to thermal plasma energy due to the very high directed velocities produced by the multi-stage magnetic accelerator. A magnetic nozzle at the exit to the accelerator also provides for very low frozen flow losses. (3) The unspent magnetic energy not used in accelerating the FRC returns to the storage capacitor after each pulse as the coil and capacitor form a natural L-C oscillator that in principle can be very low loss at only a few percent of the total energy transferred. This recovery and reuse of the driving magnetic field energy can be readily accomplished by using solid-state switches with low-resistance stripline circuitry.

The Hydrogenic fuel employed and the very high I_{sp} achieved in the MAP thruster would not likely be a good choice for a thruster operating in LEO, but it would be ideal in the deep gravity wells and hydrogen rich atmospheres of the gas giants where the REPO would be situated. In fact, it could well be that the MAP thruster is the only thruster capable of powering these missions effectively.

Pulsed Inductive Thruster (PIT)—There are issues for the operation of MAP on heavy mass propellants due to the large gyro-orbits that these more massive ions have ($\rho_{gyro} = m_i v_{th} / e B_{ext}$) that allow for the large mass ions to roam far outside the plasmoid boundary and in possible contact with the thruster wall. One way to avoid this is to have the ion accelerated away from a plate rather than down a cylinder. While this provides only a short acceleration region, it has been shown to be effective for large enough and properly prepared magnetically actuated push-off plates. The device that accomplishes this is referred to as the Pulsed Inductive Thruster, or PIT [16,17]. The PIT employs a large, flat, spiral wound induction coil that is on the order of a meter in diameter (see Fig. 13). Given the inherently small region of strong interaction with the induction coil, the coil current pulse lengths are typically on the order of microseconds. The force accelerating the plasma current sheet formed near the coils at breakdown drops rapidly as the plasma current sheet moves away. In order to achieve efficient pulsed inductive acceleration, the highest possible peak current must be reached before the sheet moves far from $z = 0$. This implies that any attempts to lower the peak current and extend the current pulse length will result in a less efficient acceleration process with a higher fraction of the total energy lost in the circuit through resistive dissipation. These limitations do not apply to the MAP thruster as the current pulse travels along

with the plasmoid in the accelerator, always maintaining close proximity to the plasmoid. The PIT however does not have the charge to mass limitation that the MAP thruster has. In fact, a medium to heavy mass gas with more inertia works much better. With standoff and no electrodes, the PIT is ideally suited for working with reactive atmospheric propellants like those found in the atmospheres of Mars and Venus, and for that matter, the Earth.

The PIT thruster was developed and characterized in the laboratory over many decades. The two most advanced versions of the PIT were the PIT MK V and the MK Va. The MK V had a nominal energy of roughly 2 kJ per pulse at a capacitor charging voltage of 15 kV, yielding a total azimuthal voltage drop of 30 kV across the windings. The second version, the MK Va, had a nominal energy of 4 kJ per pulse, with a capacitor charging voltage of 15 kV. As a clarifying point, the MK Va, which had the best performance and represents the current state of the art for PIT performance, is often referred to in the literature as the MK V. The performance of the MK Va was extensively characterized using thrust balance measurements. While it was noted that the sidewalls of the 1.2-m-diam metallic vacuum chamber could have affected the performance measurements, simulations have shown that the chamber wall

3. RESULTS FROM EXPERIMENTAL TESTING AND ANALYSIS OF THE RON CONCEPT

There were several significant results that have come out of the initial study, both theoretically and experimentally, that justify the continued research into the RON concept. The results of the studies will be presented in the order they were performed.

As previously discussed, the RON's power system requirement was estimated at roughly 5 kW with the propulsion system operating at 30 kW, for approximately 10% of each orbit either for drag makeup at perigee or walk-in/walk-out maneuvers at apogee. As these assumptions depend almost entirely on the propulsion system adopted. As there was no significant thruster development in this initial effort other than to analyze the applicability and performance of various candidate EP devices, which was presented in the last section. For the mission studies as well as further analysis these performance attributes will continue to be assumed.

Mission Study

It is important to note that the numerical values presented in the previous section (see Fig. 4) on the mission architecture were recalculated as part of this task with the Copernicus 4.6 spacecraft trajectory design program for the purpose of feasibility and optimization. The results were essentially unchanged from the original proof of concept calculations performed earlier. The new work included a more detailed orbital model that included effects such as a time dependent calculation of the propellant collection. Finally, it should be noted that the analysis work in this mission study assumed that roughly 10% of the period was spent at or near the perigee altitude, and that collection of ambient nitrogen and oxygen (the dominant species) in the upper atmosphere occurred at a constant rate. Clearly this is an approximation. As the altitude changes, the density will vary significantly over the entire elliptic orbit. A singular orbital model with a power function approximation for density as a function of altitude between 130 and 180 km shows that this estimate was a reasonable one for this preliminary design effort as can be seen in the data shown in Fig. 15.

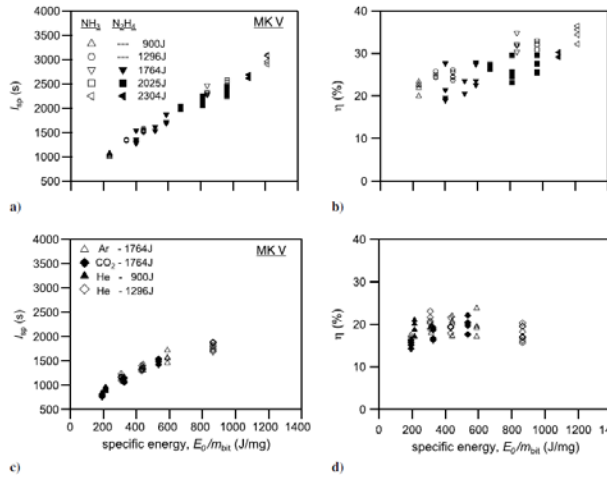


Figure 14. Performance of the PIT MK Va thruster operating on ammonia and hydrazine propellants (from Ref. [17]).

should not have a large effect on performance [18]. Performance data (I_{sp} and η as functions of specific energy, defined as the ratio of the initial charge energy to the mass bit) from the MK Va are shown for operation with ammonia (see Figs. 14a and 14b) and simulated hydrazine propellants (see Figs. 14c and 14d). It is clear that on operation in ammonia at least, the PIT can perform in a range highly relevant to many planetary atmospheres with moderate mass atmospheric gases. It could also be a candidate for our own atmosphere however much more development is required [19,20] as the more mature HET and RIT thrusters have received the lion share of developmental funding. Hopefully that can change if RON and its sister REPO prove out.

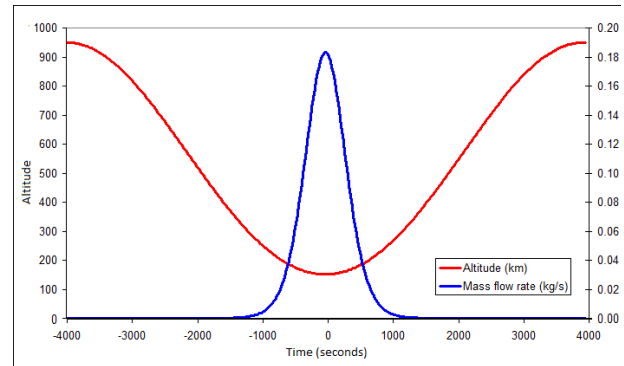


Figure 15. Instantaneous altitude and mass flow rate for the lowest perigee orbit described earlier

Mass Budget—An initial mass estimate was formulated by including all the traditional spacecraft systems and by estimating mission-specific systems using the mission requirements. Some of the unique components include the atmosphere-collecting system and a target capture mechanism. The solar arrays and batteries were sized by requiring 36.8 MJ to run the 30-kW thruster, while including extra energy to operate the other subsystems onboard the spacecraft and account for battery degradation and solar cell loss over time. An extra 25% was added as margin for this early stage of the design. Table III contains the mass and TRL estimates for the various components of the RON.

Table III. Mass Budget for full scale RON

Component	Mass (kg)	Description	TRL
Tank	20	50 kg propellant at 1000 psi, carbon overwrapped titanium alloy tank	5
Scoop	60	Conical silica scoop with 2 mm walls and a 3.5 m diameter	3
Compressor	60	Multi-stage compressor system	3
Thruster	130	30 kW HET thruster	4
ADCS	10	2 gyroscopes and 8 thrusters	8
Comm. and Data	15	Antenna, diplexer, receiver, transmitter, and encoder	8
Power	35	Power conditioning unit	8
Batteries	55	36.8 MJ lithium polymer batteries (900 kJ/kg)	8
Solar arrays	15	XTJ triple junction cells of 84 mg/cm ² and array area of 13.66 m ²	6
Instruments	30	Star tracker, earth sensor, LIDAR unit, and telescopic camera	8
Capture Hardware	20	Estimate	2
Structure	60	Estimate	N/A
Thermal	10	Estimate	N/A
Dry Mass	520		
Margin	130	25% of dry mass	
Propellant	50		
Total Mass	700		

Spacecraft Design

It is not lost on the author that there are several elements of the RON concept that are at a low TRL. The primary focus initially was on the scoop as it is likely the lowest TRL element in the RON system and one that is critical to the success of RON. MSNW possessed a suitable experimental facility for the testing of the key RON scoop parameters. Measurements of the capture efficiency, drag coefficient and neutral compression ratio for a prototype scoop configuration were obtained. While the full range of testing awaits further development time and money, the initial studies were able to shed light on the drag coefficient and collection capability in an experimental setup that was a reasonably accurate subscale version of the relevant parametric conditions that will be found in space. A discussion of these results follows.

Collection Scoop

(1) Scoop Design—It is believed that a scoop geometry much like the Tapered Honeycomb Collector (THC) design sketched in Fig. 16 presents several advantages over the current and past scoop designs considered by other groups [21]. RON for instance will operate with a high Knudsen number (particle mean free path/characteristic channel dimension) throughout the collector so that the compressed flow remains in a condition of free molecular flow at all

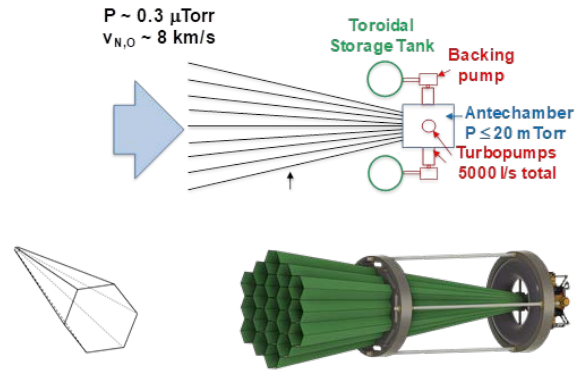


Figure 16. (Top) Schematic of the various CHARON elements. (Bottom) the Tapered Honeycomb Collector (THC).

times. In order to avoid backscatter, the THC wall must be smooth to foster specular reflection and thereby increase collection efficiency and reduce drag. The large duct aspect ratio (duct length/square root of duct areal cross-section) assures that the conduction loss of particles back along the THC from the antechamber/plenum will be negligible. The scoop is designed to have the maximum compression ratio, but will be heavily constrained by the C_D and capture percentage limits imposed by the nature of the wall interaction.

(2) Modeling—This task was carried out with both analytical and numerical efforts. Given the time-consuming and difficult task of trying to adapt existing codes, it was decided to make use of work previously done in the area of molecular flows relevant to the task of analyzing the scoop performance and adapt as needed. While not ideal, it was adequate for interpreting the initial experimental tests that were actually more constrained by what components could be easily purchased or fabricated. The modeling of the molecular flow into a conical tube has been determined by various groups: a

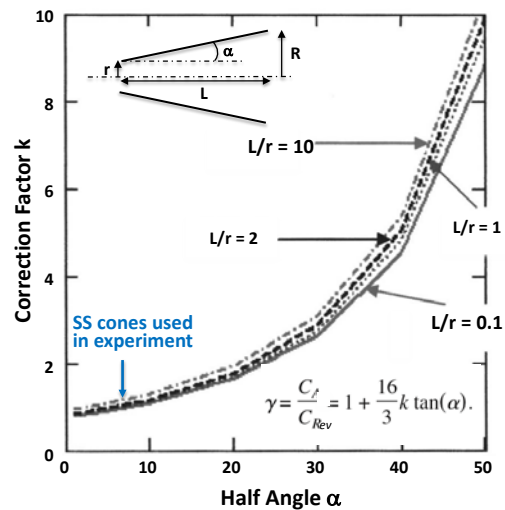


Figure 17. Correction factor k dependence on conical half angle for directional conductance asymmetry ratio. γ .

Monte Carlo calculation coming from CERN, and the Direct Simulation Monte Carlo (DSMC) code of Bird [22]. The results of these calculations are summarized in Fig. 17.

The parameter of interest is the conical conductance ratio, γ , for particles entering from the small radius end of the cone (conductance C) as opposed to entering at the large bore end of the cone (conductance C_{Rev}) as it would be for the atmospheric gases entering into the scoop. As can be seen from the embedded equation in Fig. 17, a large conical pitch angle leads to a greatly diminished conductance of the scoop for particles entering from the large end. In many previous designs proffered as collectors for air breathing propulsion employed scoops with large pitch angles [21]. A pitch angle of 45° results in a conductance $C_{Rev} = 0.02 C$ – a reduction of a factor 50! From Fig. 17, it can be seen that there is only a small dependence on aspect ratio, so it is clear that the scoop should be comprised of many small pitch-angle collectors of sufficient length to provide the desired mass flow and compression.

This is what was done for the scoop used in the experiments. The conical collectors were tapered stainless steel with a shallow pitch angle of 14° , and long enough to provide for a compression ratio of 25 (see Fig. 18). Given the large compression ratio, the conductance reduction was relatively minor, $\gamma = 0.6$. It should be noted however that the conductance ratio derived from the calculations was determined for a molecular flow assuming a Maxwellian gas and diffuse reflection. The “reverse” flow into the cones in space (as well as experimentally) was highly directional with Mach numbers of 10-12 while the flow back out from the plenum will be Maxwellian. Specular reflection also makes a dramatic effect on conductance, so it is possible to make the effective conductance into the plenum much greater than out. It should also be noted that the conductance into the pump is also much larger than that of the collector conductances, and that is also a necessary condition for high collection efficiency.

Key Scoop Physics Issue—Particle Reemission from Surfaces
The behavior of the incoming highly directed, supersonic

atmospheric particles is of critical importance in evaluating the efficiency of collection for the wide array of possible scoop designs. It is generally assumed that the interaction results in a diffuse reflection (see Fig. 19). Many laboratory experiments have found that molecules are reemitted from a contaminated surface in a diffuse, or cosine distribution centered on the normal to the surface. Molecules are reemitted from a clean surface in a narrow beam near the specular angle. As the contamination increases, the quasi-specular fraction is reduced, and the diffuse fraction is increased.

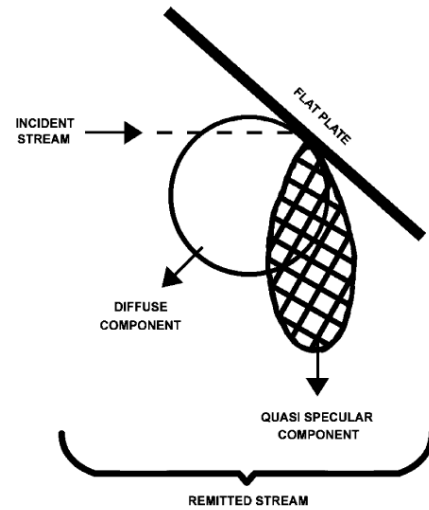


Figure 19. Reemission from a flat plate.

Particles that strike a rough surface, or one that has a large quantity of adsorbed gases, will suffer diffuse reflection as they lose much of the information about their initial energy and momentum in the interaction with the adsorbed gases. The molecules reemitted in the quasi-specular lobe have initially struck a smooth or bare patch of the surface, thereby retaining much of the information about their initial energy and momentum. Most models calculate the drag coefficient under the assumption that all of the particle collisions consist of the diffuse, highly accommodated component of the reemitted molecules. The quasi-specular component is not considered. This is due to the fact that both laboratory and satellite surfaces are typically highly contaminated. The steady flow of highly directed supersonic particles can actually clean the surface if the interaction is at grazing angles as it will be for the scoop designs contemplated for RON. As the surfaces become less contaminated, and the accommodation coefficient falls, the angular distribution approaches the quasi-specular. Conversely, as the surface contamination and mass of the adsorbed molecules increase, the angular distribution progresses from largely quasi-specular to nearly diffuse. For the quasi-specular component, it is useful to employ the model where the accommodation coefficients are derived from the static-lattice, hard-sphere model of Goodman [23], which was constructed to fit the laboratory measurements on clean surfaces. Goodman’s formula for the accommodation coefficient is:

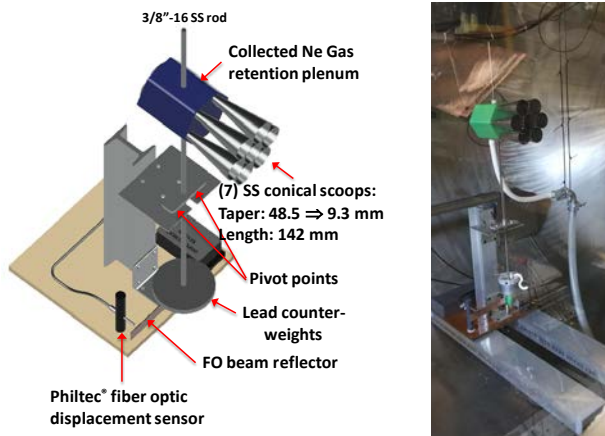


Figure 18. Experimental setup used in initial experiments to determine scoop capture efficiency.

$$\alpha = 3.6 u \sin \theta / (1+u)^2 \quad (5)$$

where u is the ratio of the mass of the incoming molecule to that of the surface molecule, which for the scoop would be $u \sim 1$. For the 14° angle used in the phase I testing, $\alpha = 0.22$ meaning that 78% of the particles will undergo a specular reflection thereby substantially increasing the capture percentage. It is on this basis that the capture percentage is predicted to be as high as 80% for RON. A schematic of the expected propellant flow through the RON system during a

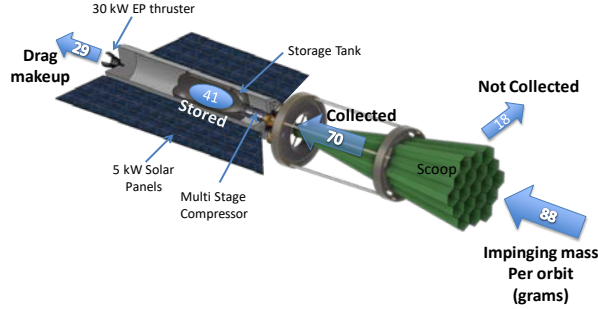


Figure 20. Schematic of RON propellant collection and usage during elliptical orbit. Drag makeup assumes a drag coefficient of 2. A collection efficiency of 80% during perigee is also assumed. Orbital perigee is at 150 km.

single orbit is shown in Fig. 20. This is based on the collection scenario described in section 1 and in particular, Fig. 15 in this section. It is worth pointing out that the drag force is not a strong function of whether the particle interactions with the scoop wall are diffuse or specular as the fate of the gas that is reflected specularly finds its way into the antechamber plenum to be pumped into storage where it stagnates with diffuse collisions against the walls. This assures a final drag coefficient that is never less than 2.

The following equation relates the drag force to the other key parameters.

$$F_d = m \cdot a = 1/2 \rho V_i^2 C_d A_{cs} \quad (6)$$

where F_d is the inline component of the force of atmospheric drag; a is the corresponding in-line component of acceleration; m is the mass of the spacecraft; ρ is the atmospheric density; C_d is the drag coefficient; A_{cs} is the cross-sectional area of the scoop intake projected normal to the velocity vector; and V_i is the speed of the incident air molecules relative to the spacecraft. It is assumed here that the scoop intake cross-section is greater than any other cross-section of the spacecraft, and thereby accounts for all of the drag. This is not strictly true for elongated objects where molecules near the scoop edge can diffuse radially inward and interact with other parts of the spacecraft. This effect should not induce a significant drag as the Mach number of the directed flow is very high and the scoop cross section is likely to be much larger than the cross section of the rest of the spacecraft. The minimum drag force for a particle flow that is completely captured is has a drag coefficient of 2. This merely reflects momentum conservation for a completely

inelastic collision. For this to be true, all the gas must be collected. Any disturbance to the upstream, yet-to-be-collected, inline gas from the gas that was reflected and escapes back out of the scoop can increase the drag coefficient as either as an imparted momentum increase or acting as an effectively greater area than the physical entrance area of the collector. All these corrections are represented as an increase in C_d . More will be said of this in the next subsection. Needless to say, however, the spacecraft drag coefficient is likely to always be somewhat larger than 2.

Scoop Thermal Issues—It is worth at this point to consider the effect of the capture of this supersonic gas on the heating of the scoop/collector system. In order to make this calculation a few simplifying assumptions were made. Orbital velocity calculated at perigee was based on simple orbital mechanics calculations. Atmospheric data was taken from NASA's MSIS-E-90 Atmosphere Model. Scoop area was 10 m^2 in size. Friction power per unit area was based on

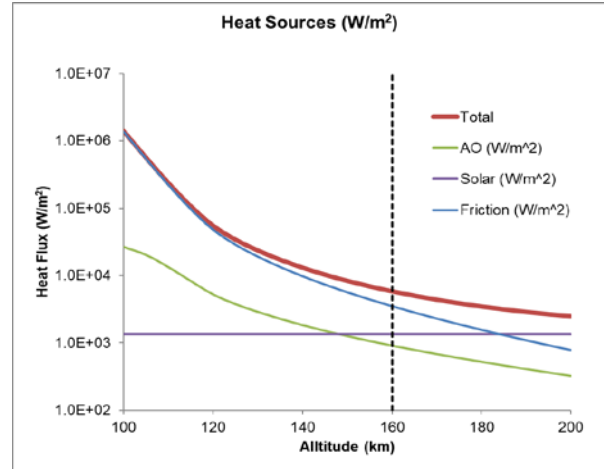


Figure 21. Various heat sources on the scoop/collector system as a function of altitude. AO represents the heat released from recombination of atomic oxygen.

full the stagnation energy of all constituents of the bulk incoming flow. The power per unit area need to recombine all atomic oxygen (AO) was 497 kJ/mol. Full constant solar

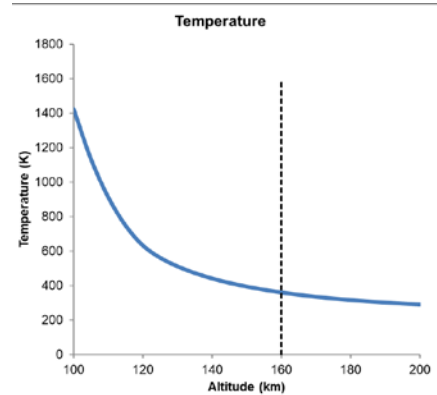


Figure 22. Scoop temperature as a function of altitude. Radiation was the only loss channel considered in the calculation for cooling.

illumination of the scoop was assumed. Scoop surface area radiated with an emissivity of 0.6. The results of the calculations can be found in Fig. 21. It can be seen that at 160 km the majority of the heating is done by flow stagnation (friction). Solar heating is more important than the atomic oxygen recombination. If one assumes that all the heat energy is taken up by the scoop, and that the only loss is through radiation, one can calculate the scoop temperature. The result is shown in Fig. 22. The temperature rise was a modest 90°C at an average altitude of 160 km during gas collection. The temperature rise was a modest 90°C at an average altitude of 160 km during gas collection.

DSMC modeling—As was mentioned in the preceding section, the drag coefficient can be much larger than 2 if factors other than spacecraft cross section come into play. The best method for analyzing these influences on a wide range of molecular flow parameters and physical objects is the use of the Direct Simulation Monte Carlo (DSMC) method developed by Graeme Bird [22] in the early sixties and continually improved into the nineties. The DSMC method is the Monte Carlo method for simulation of dilute gas flows on the molecular level. To date DSMC has become the basic numerical method in the kinetic theory of gases and rarefied gas dynamics.

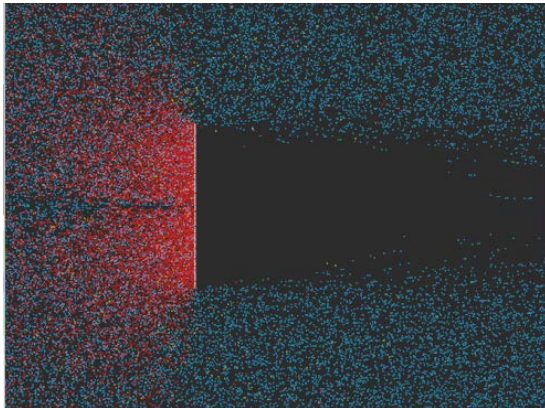


Figure 23. DSMC calculation for normal (left to right) molecular flow onto a planar disk. Input: Mach number = 10, Knudsen number = 100.0, Specular Reflection = 0. Result: Drag Coefficient = 2.19. Red dots represent particles that have struck the surface. Yellow dots represent particles that are indirectly affected. Blue dots represent unaffected free streaming particles.

As an example of the power of this method, and one that has relevance to the experimental testing results to be discussed in the next section, the case of a planar disk normal to a molecular (high Knudsen number) supersonic flow is considered. This would be the case at perigee for the RON spacecraft but hardly the correct shape. The result of the DSMC calculation is shown in Fig. 23. It should be noted that even for this more or less ideal case, the drag coefficient exceeded 2 where it was assumed the interaction with the surface was completely diffuse. If one assumes 100% specular reflection, the results produced a drag coefficient of 3.93 reflecting the elastic collision characteristic of normal specular reflection. It is useful to show the results of a

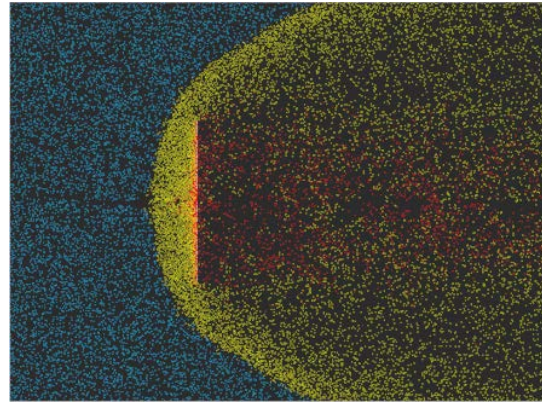


Figure 24. DSMC calculation for normal (left to right) fluid flow onto a planar disk. Input: Mach number = 10, Knudsen number = 0.01, Specular Reflection = 0.0. Result: Drag Coefficient = 1.67. Red dots represent particles that have struck the surface. Yellow dots represent particles that are indirectly affected. Blue dots represent unaffected free streaming particles.

collisional particle flow (low Knudsen number) as the effect of the collisions can actually reduce the drag coefficient to 1.67 (see Fig.24). The particles that execute the primary collisions with the surface almost immediately interact with incoming particles that cause them to diffuse outward and flow around the disk (yellow dots). This of course will not occur in the molecular regime. It is however possible to have a low drag coefficient in this regime for appropriately shaped surfaces. Consider the case that is more relevant to the RON scoop - a conically shaped surface. Figure 25 displays the results for the fully specular case and the fully diffusive case. Even though the pitch angle is large in these two cases, it is clear that the drag coefficient can be less than 2 for the scoop as the particles are deflected with a large fraction of their original inline momentum preserved. Collisions at a more grazing angle will reduce the momentum transfer even more providing for an even lower drag coefficient. As was already indicated, the scoop can thus be constructed to have an effectively high conductance to the directed molecular flow, and thereby provide a high efficiency for capture. The fate of the captured particles however is to ultimately come to rest in the frame of the spacecraft, so that as far as the drag makeup is concerned, the drag coefficient will always be around 2.

The DSMC analysis of the flow into a tapered cone with a small pitch angle is still a work in progress as code convergence in this case appears to be more troublesome. But given the sensitivity of the calculations to the assumed value for the specular reflection, the DSMC method cannot be definitive in predicting capture percent or the drag coefficient for different scoop designs other than provide the limits. While the DSMC analysis is useful for gaining insight as to geometric elements that could enhance particle capture, the most straight forward and valuable way to evaluate the scoop/collector is by obtaining experimental measurements under the spacelike conditions under which the scoop will be employed. That is what was done in the experimental testing, and it will be critical to do that for all candidate scoop designs

in the future with the highest fidelity one can. Ultimately it will have to be confirmed in LEO, but it is believed that a fair idea of the RON performance can be obtained in a laboratory setting. The next section will discuss the first attempt at obtaining the crucial scoop performance data.

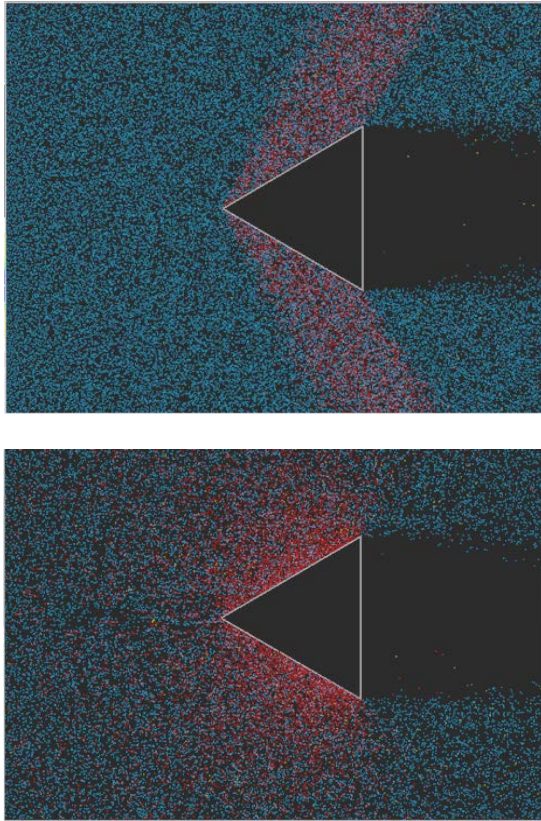


Figure 25. DSMC calculation for normal (left to right) molecular flow onto a conical surface with a pitch angle of 45°. (Top) Input: Mach number = 10, Knudsen number = 100.0, Specular Reflection = 1.0. Result: Drag Coefficient = 0.99.

(Bottom) Input: Mach number = 10, Knudsen number = 100.0, Specular Reflection = 0.0. Result: Drag Coefficient = 2.10.

Red dots represent particles that have struck the surface. Yellow dots represent particles that are indirectly affected. Blue dots represent unaffected free streaming particles.

3. RESULTS FROM EXPERIMENTAL TESTING

A reasonably optimal conical scoop was constructed that consisted of seven close-packed stainless-steel cones. These were mounted to a 3D printed antechamber (plenum) for gas collection. A thrust stand was also fabricated for the key tests of collection efficiency and drag. The experimental setup for both of these measurements is found in Fig. 18 The method for measuring momentum impulse was a counter-weighted inverted pendulum that could measure forces as small as a few μN . In the configuration shown, the gas antechamber was connected via very flexible, thin-wall silicon tubing to a large bore vacuum hose that acted as a secondary plenum. This

hose ran to a port where a sensitive capacitance manometer was mounted (500 mV/mTorr) for an absolute measure of the gas pressure. The entire setup was installed in the MSNW large chamber which was $\sim 2\text{m}$ diameter by 4 m long, with a peak pumping speed of 22,400 l/s. The collector cones were aimed at a port on the side of the chamber which contained a small conical nozzle (see Fig. 26) that produced a supersonic neutral beam of Neon. Neon was chosen as it was a close match to the average molecular mass at 150 km altitude, as well as the fact that it is also nonreactive and readily pumped.

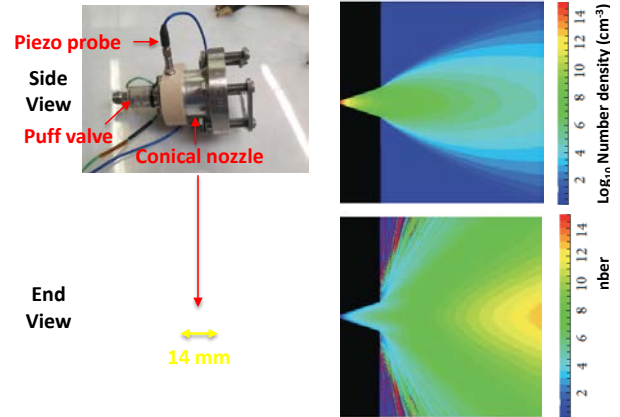


Figure 26. (left) Neon beam hardware. (right) number density and flow Mach number. Range of x axis is 50 nozzle inlet diameters (15 cm).

The 14° conical nozzle had an inlet aperture of 0.3 mm. The flow characteristics were similar to the DSMC calculations also shown in Fig. 26. The gas was supplied by a fast, Parker series-9 puff valve (~ 0.2 msec open/close). The inlet pressure was monitored by a piezo pressure probe inset into the gas feed to nozzle passage. A negative pressure regulator was used to meter the flow (to 50 Torr absolute). This nozzle

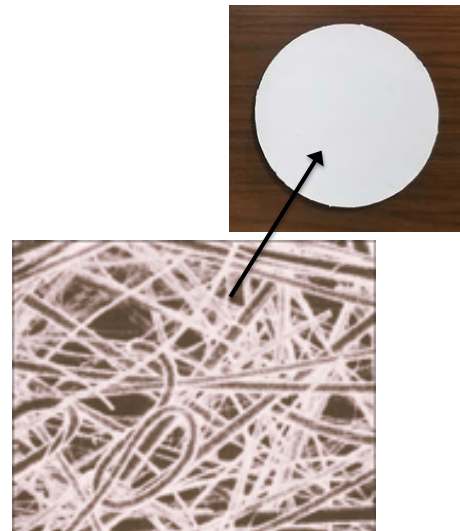


Figure 27. Top left: plastic backbone supporting Quartz Filter Paper (QFP). Top right: Front side of QFP target that faced the Neon neutral beam. Bottom: Electron microscope image of QFP showing microscopic structure.

produced a low-density (1 to $3 \times 10^{17} \text{ m}^{-3}$), supersonic ($\sim$ Mach 10), neutral Neon beam 90 cm downstream at the scoop entrance plane. In this way the Knudsen number in the scoop was always $\gg 1$ (mean free path of 5-12 meters) and thus in the molecular regime.

Neon Neutral Beam Characterization—In order to measure the capture percentage or drag coefficient, it was critical to measure the characteristics of the neutral neon beam emerging from the nozzle. This was accomplished in the following manner. While the DSMC calculations gave a good idea of what one should expect in terms of flow velocity, the angular spread of the beam at the entrance of the scoop as well as the radial profile that far from the nozzle (scoop standoff/nozzle inlet diameter = $900 \text{ mm}/0.3 \text{ mm} = 3,000!$) needed to be quantified. It is difficult to detect neutral particles especially at the very low densities employed here. The objective was to reduce the density to the point of where the collisional mean free path was at least several meters. A collisionless neutral gas is difficult to measure as it emits no light, even a fast ion gauge could not produce a reasonable signal level as densities on the order of $10^{19} \text{ Ne atoms/m}^3$ were required. The target neutral densities were almost two orders of magnitude smaller. The manner that the beam profile and mass flow (hence number density) was determined was to make a series of planer disks of various sizes. These were constructed in a manner that provided a surface that would provide a totally diffuse reemission of the beam particles striking it. It consisted of a quartz fiber filter paper. This paper was adhered to a plastic backbone that was 3D printed to have minimal mass and cross section. A picture of one of the disks as well as a free-electron beam image of the microscopic nature of the Quartz Filter Paper (QFP) can be found in Fig. 27. The fibers were less than 10% of the paper volume and provided for the multiple scattering of the beam particles as they penetrated deeper into the paper resulting in complete absorption of the beam momentum. The DSMC calculation of these disks was shown in Fig. 23 where a drag coefficient of 2.19 was calculated.

The inverted pendulum thrust stand was employed to measure the force (impulse) delivered by the Neon beam. A picture of the testing setup is shown in Fig. 28. As can be seen in Fig. 28, the thrust stand was positioned with the largest QFP disk near the exit of the nozzle so that all of the beam could be captured. The total gas inventory during the pulse was reflected by the rise in the capacitance manometer. Only one pump was employed to give a longer time for the gauge to settle before the chamber pressure changed significantly. Given the volume of the chamber (9.3 m^3) and peak pressure, the total mass of gas emitted by the Neon beam source can be calculated. As the puff duration is well controlled by the fast puff valve and the magnitude of the flow is monitored by the piezo probe, the mass flow rate (dM_{Ne}/dt) from the nozzle is readily and accurately calculated. With the impulse measured from the calibrated thrust stand, the flow velocity is determined. The flow velocity determined in this manner was 630 m/s which is the sound speed for Neon. The flow is quite supersonic though as the expansion of the Neon beam out of the nozzle dramatically cools the beam as attested in the

DSMC calculations shown in Fig. 23. The reduction in the beam thermal velocity (temperature) is further attested to by the relatively smaller beam spreading as it propagates which was measured by the array of disks. The beam profile over the relatively small cross-section of the 7 cones ($\sim 5 \text{ in.}$) is



Figure 28. Left: QFP disk mounted on thrust stand roughly 0.9 m downstream of nozzle. Right: Large disk mounted on extension positioned 10 cm downstream to capture all the Neon neutral beam.

shown in Fig. 29. Based on the measurement of the entire beam inventory as described earlier, the total inventory inside the 10 inch diameter disk when backed off to $\sim 1 \text{ m}$ from the nozzle exit was $1/6$ that measured when the same disk was positioned directly in front of the nozzle as shown in the right side of Fig. 28. The Neon number density at the lowest flow rate which was used for most of the capture and drag measurements was $1.3 \times 10^{17} \text{ m}^{-3}$. The total mass flow from the nozzle was measured to be $2.36 \times 10^{19} \text{ Neon atoms/sec}$.

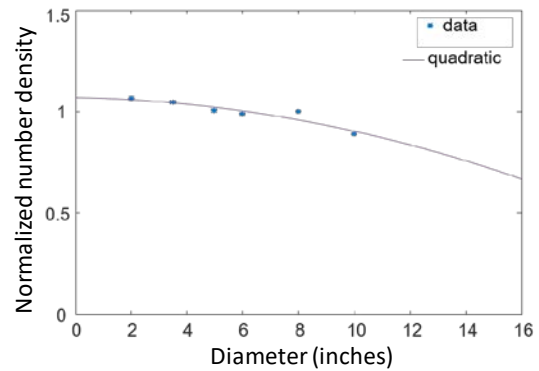


Figure 29. Neutral Neon beam profile determined from array of disks, and the analysis described in the text.

Thrust Stand Calibration—The thrust stand was calibrated in the same manner for all three configurations - the scoop/plenum, the scoop alone, and the disks. The pendulum configurations were calibrated using a simple pendulum with a 2g ball dropped from a known height. Using an SLR with zoom and a video capture feature allowed the ball motion to be freeze framed, the bounce height after impact could be readily measured thus providing for the full measure of the momentum transfer to the QFP. For the measurements involving the scoop, a piece of the QFP was cut out and

placed at the entrance to the central cone. These measurements were conducted with the chamber at atmosphere. A calculation was made of the effect of air drag on the spherical ball. Due to the small size of the ball and the low terminal velocity, air drag made for only a 2% correction to the momentum transfer compared to vacuum.

Measurement of the Scoop Capture and Drag Coefficient—
In the experiments the scoop was operated two ways. Initially, it was operated with a stagnation plenum mounted on the rear of the scoop as in Fig. 18. The plenum was sized to be shadowed by the scoop except for the area between the cones which was roughly 10% of the total area. With no active pumping of the plenum and tubing, only very short duration gas pulses (50-200 msec) could be employed before the plenum pressure would become excessive and create a large backflow. The capacitance manometer signal was corrected for the connecting tubes' volume ballast and constrictive flow to arrive at the plenum pressure as a function of time. It was found to initially be significantly higher than the ambient chamber pressure, and inferred a capture efficiency of 40-60%. There were considerable error bars due to the small signal levels recorded for these short pulses. While this is an adequate capture efficiency, it is hoped that it can be improved in the future by smoother walls, and an even smaller conical pitch angle (and thus many more cones).

The drag coefficient, C_D , was measured in an alternate setup depicted in Fig. 30. As can be seen, only the scoop was



Figure 30. Right: Thrust stand setup for scoop drag force measurement. Left: image of scoop taken through neutral beam vacuum port.

mounted on the thrust stand, and held by a 3D printed array of tapered hoops in a way that provided only a minimal cross section in the direction of the gas jet. In this way the scoop drag was by far the dominant drag. Backflow was reduced as the conical exit was open to the chamber. The drag force of the impinging flow on the candidate scoop was determined again by the pendulum deflection. Results from the drag measurement can be found in Fig. 31. The measured C_D was 2.1.

The importance of these measurements can be gleaned from Fig. 4 which shows the relative refueling time for various scoop capture efficiencies. With a C_D of 2 at an altitude of

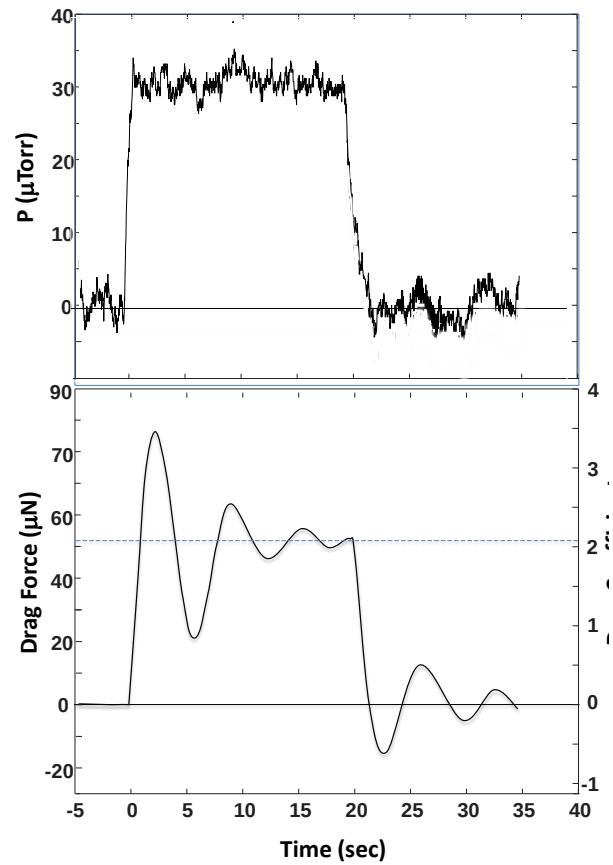


Figure 31. (top) Chamber pressure (bottom) Calibrated thrust stand motion for determination of drag force and C_D .

150 km, the minimum capture percentage would be 50%. It is worth noting that for high capture percentages, refueling times level off and are not reduced significantly with increasing I_{sp} . The opposite is true at lower capture percentages. At a 50% capture efficiency and an I_{sp} of 2000 s, it would take 220 days to refuel, while a 3500 s design would only take 68 days, a reduction of 69%. A high I_{sp} thruster would allow for operation at lower capture percentage of 30 to 40% if need be, but the mission time would increase significantly. A more complete model will be needed to investigate the dependence of I_{sp} and capture efficiency for various operational burn times and configurations. It is clear though from the initial work that a satisfactory scoop can be designed and built, and why optimizing scoop design for minimum drag and maximum capture efficiency is important for reducing mission time.

4. CAPTURE METHODS

Once a quick and effective orbital transfer method has been devised, there still remains an important technological challenge to overcome: the transmission of momentum from the removal system to the space debris in order to carry out the deorbiting maneuver. Several active debris removal concepts have also been proposed in debris mitigation including harpoons, magnetic darts, robotic arms, adhesives, scoops, interface probes, applied fields, nets, and plasma beams. The most obvious way is to dock the removal system with the target before the deorbiting starts. This operation can, however, be technologically complex and very risky. Space debris are noncooperative objects generally characterized by a problematic attitude motion (tumbling motion, flat-spin rotation, large amplitude oscillations, etc.) making docking dangerous and/or unfeasible. There are still several issues with physical connection methods relying on a more remote and therefore safer connection to the target such as harpooning or netting. There is the major difficulty of how to apply the net or where to spear. Besides deployment issues, it would be difficult to reuse said netting or harpoon for multiple removal operations. Debris removal concepts based on pulsed-laser ablation systems do offer a key advantage in this regard as they can be operated potentially a safe distance from the orbiting target, possibly even from the ground. Unfortunately, though, the small impulse obtained from material ablation cannot be effective against targets of size exceeding about 20 cm.

During the initial survey conducted of active debris removal methods, there was one particular approach having special merit. It is referred to as the Ion Beam Shepard (IBS) [25]. It is unique in avoiding many of the problems previously described, but it is especially advantageous when one considers the fact that with RON one has an essentially unlimited fuel resource for this task. A brief overview of this concept will now be given as to its appropriateness to ADR, given the development of a RON, places it heads and shoulders above any ADR capture method of which the author is aware.

The Ion Beam Shepard (IBS) is aptly named as it is what it says it is. The IBS employs a highly collimated, high-velocity ion beam, produced on board the spacecraft orbiting in close proximity to a target debris object, which is directed against the target to modify its orbit and/or attitude with no need for docking. The momentum required to perform these maneuvers is transmitted solely by the ion beam. A considerable advantage of this approach stems from the mature development of such ion beams for propulsion. Ions have been accelerated to 20–50 km/s in ion engines like those that powered the Deep Space 1 and Dawn spacecraft and thus have been successfully flown in space. As was mentioned earlier, ion thrusters have recently been stably operated on atmospheric gases with low erosion although with lower efficiencies than when operated on Xenon. For the same onboard power, the momentum transfer from the high velocity ions to the target satellite is orders of magnitude higher than that obtained using material ablation. The only

serious deficiency of the IBS system for ADR was limited propellant which is completely eliminated with the pairing of the IBS with RON. Figure 32 illustrates how the IBS would work to deorbit a target spacecraft. The shepherd spacecraft

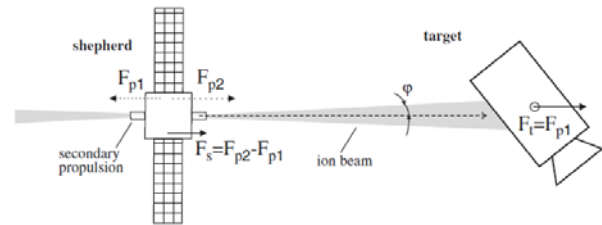


Figure 32. Schematic of the Ion Beam Shepard (IBS) satellite for deorbiting space debris.

(RON) is equipped with a primary propulsion system that emits a beam of accelerated quasi-neutral plasma and points it towards a target object in order to apply a force F_d on the latter through the momentum carried by the plasma ions. It should be pointed out here that the process does not have to be an ion engine. It could just as well be a HET or a PIT as described earlier. It should also be noted that the analysis in Ref. 25 assumes an ion thruster, and uses the terminology of “ion beam” when in fact it is a neutralized and directed plasma beam, and could just as correctly be called a Plasma Beam Shepard (PBS) whose acronym conjures up a more pleasant image than IBS. The ion thruster was chosen as it can have a large I_{sp} and fairly low divergence, and thus a highly efficient deorbiter.

If one neglects the momentum associated with ion sputtering from the target surface and assuming that the beam fully intercepts the target, F_d will be equal and opposite to the force F_{p1} that the primary propulsion system exerts on the shepherd satellite. There is likely to be secondary ions and neutrals that are sputtered back from the target satellite surface increasing, in principle, the net momentum transmitted to the target. But, given that their ejection velocities are generally quite small compared with the ion beam, the effect on the transmitted force is likely negligible. It should be noted that to maintain formation flying, the IBS has to be compensated by an impulse compensation thruster (ICT), which is located on the opposite side of the IBS as shown in Fig. 32. This thruster could just as well be the primary thruster for the RON and a specialized, low-divergence thruster as the Impulse Transfer Thruster (ITT) trained on the target satellite. It is critical for the ITT to have a low plume divergence in order to provide a greater standoff from the debris. At this point it would appear that the ITT would be a RIT as it has the lowest plume divergence and highest potential efficiency among the most advanced EP systems. It is believed that with a RIT that a standoff of greater than 7 m could be maintained [26]. There is certainly no reason at this point to be too concerned with the specifics of the IBS as the availability of copious amounts of propellant will certainly shape the type of propulsion that would be the most optimal. For a RON + IBS it is likely to be a thruster with a much greater thrust to power than the RIT such as the HET and/or PIT.

A final consideration for the IBS is the effects of an off-center plume impact on the debris, which would reduce the momentum transfer efficiency, but it might also induce angular acceleration (i.e. rotation) of the debris target. Even though the associated fragmentation risk is estimated to be quite small [26], it has been suggested the use of a dedicated control strategy would minimize even this risk. The use of applied quadrupole magnetic fields could be used to maintain stability and control the orientation of the target satellite.

5. SUMMARY

As can be seen from the parametric analysis of RON determined in these initial studies, the most critical factor in determining the applicability of RON to ADR and other planetary missions is in the performance of the full system. Detailed mission studies where certain performance levels of both the scoop compression and thruster efficiency are assumed, can be useful for showing the promise of the concept, and the results from the phase I studies clearly support further investigation of the RON concept. There are however critical elements of the RON concept that need to be advanced before the major investment required to develop the full RON system could be justified. It should be clear that does not mean that further mission studies are without merit. Further efforts to delineate the different RON design elements needed for optimal performance on the wide range of planetary missions outlined earlier should also be addressed in future work. At this juncture in RON's

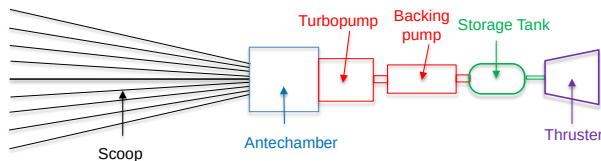


Figure 33. Schematic of the elements in a full RON system.

development, in order to advance to TRL4, the goal must be to demonstrate the key physics of the RON in a laboratory or possible even a LEO subscale demonstration. This would go a long way in determining the technology required for further development. It will be a major effort to carry out the experimental demonstration of the full RON system with a subscale test. This is an ambitious goal as there are many elements that must be brought together to make such a test happen. Fortunately, this is made possible by collaboration with Aerojet Rocketdyne (AR) in Redmond, WA. AR is the major provider of in-space propulsion systems and is ideally situated to provide the assistance required to test the full RON system.

There are four major elements of CHARON to be demonstrated (1) the propellant scoop, (2) the propellant compression (3) propellant storage and metering and (4) thruster performance. This is shown schematically in Fig. 33. Several key elements would be provided by AR. These would include a test Hall thruster as well as the vacuum chamber and thrust stand required to make the evaluation of the subscale CHARON system. To obtain the LEO atmospheric

flow at relevant flow speeds would be provided by an AR arcjet fitted with a diverter magnet to deflect away the plasma component of the primarily neutral flow. This test will be challenging but doable.

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BIOGRAPHY



John Slough received his B.A and PhD in Astrophysics from Columbia University. He has spent over 30 years working in the field of experimental physics. He founded MSNW in 1991, a small business with a long history in the research and development of advanced space propulsion and plasma physics technologies. He was the principal investigator on several large fusion related projects funded by the Dept. of Energy. Experiments performed by Dr. Slough led to advances in the formation and confinement of FRCs as a fusion concept as well as a better understanding of the relevant physics. Recently, he founded a spin-off company, Helion Energy, to develop a commercial fusion reactor which has received both DOE and venture capital funding. Dr. Slough was also on the faculty in the Astronautics and Aeronautics Dept. at the University of Washington for many years.