

The Space Gas Station

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Abstract

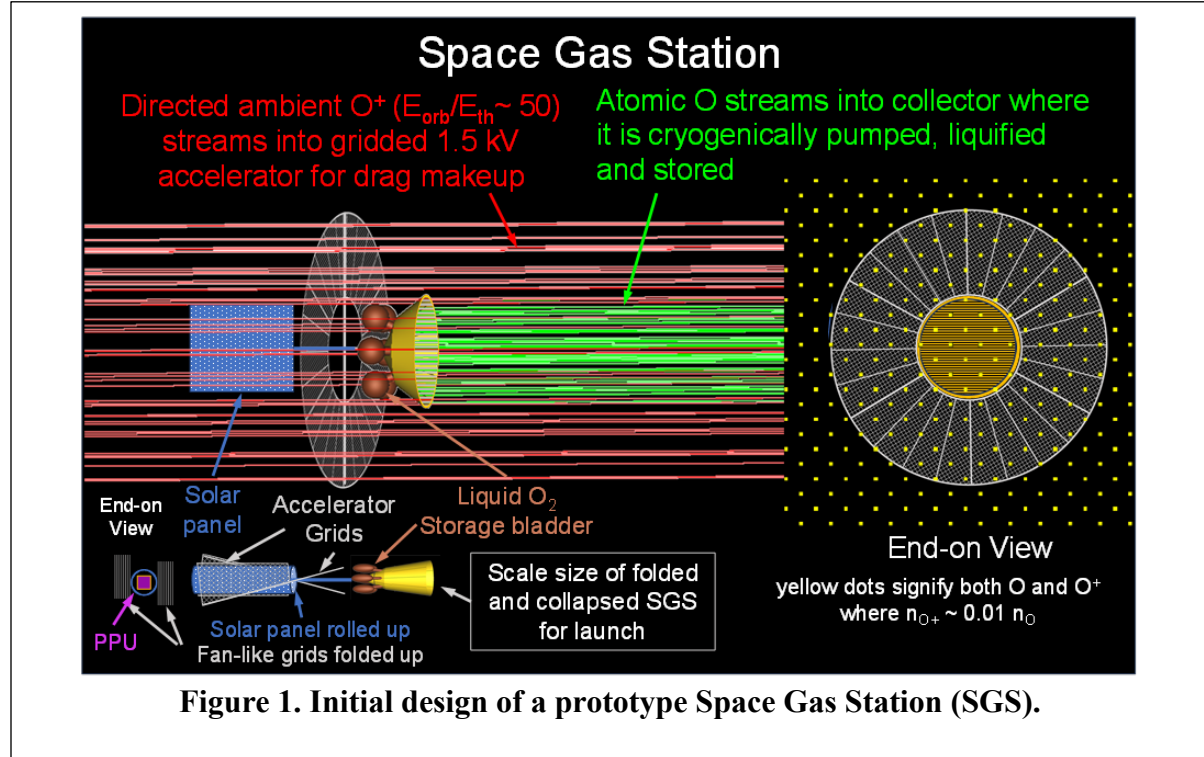
Liquid oxygen depots in low Earth orbit could transform space logistics, however every kilogram of stored propellant must itself be launched at a cost of ~22 kg of launch propellant. The Space Gas Station (SGS) removes this penalty by harvesting propellant in situ. A 5 m radius collector captures the ambient atomic oxygen stream at 300 km and stores it as liquid oxygen. An Ambient Ion Accelerator (AIA) comprised of a two-grid electrostatic accelerator exploits the pre-existing ionospheric O^+ population to provide drag makeup. The AIA requires no pre-ionization stage, neutralizer, or onboard propellant. High gas collection efficiency is achieved with a cryogenic system that avoids the direct collection of the incoming stream by employing a louvered cold-head lining a radiatively cooled conical wall. A 96% capture rate was determined from particle-trajectory calculations with only 7% of the hyperthermal flux reaching the cold-head, thereby keeping required cooling power to 2–4 kW. The reference SGS collects 490 kg of oxygen per year for a launch mass of 170 kg and 17 kW, repaying its launch mass in four months. The analysis identifies the opposed altitude gradients of the neutral and ion populations as the constraint governing orbit selection, motivating an elliptical orbit that collects at perigee and thrusts at apogee. A sample mission analyzes propellant-neutral removal of a one-tonne spent rocket body from 950 km. Also, a small fleet is shown sufficient to supply the oxidizer for the Mars Ascent Vehicle and the many control thrusters needed for human Mars exploration.

Keywords: in-situ resource utilization; VLEO; atmosphere-breathing electric propulsion; propellant depot; active debris removal; cryogenic collection

1. Introduction

Liquid oxygen (LOX)-based propellant depots in Low Earth Orbit (LEO) have been identified as a potential method for reducing launch costs, with NASA and various aerospace contractors investing in fuel storage and transfer technologies [1,2]. Fuel depots could be the answer to one of the greatest challenges in creating a sustainable cislunar economy: refueling a rocket's upper stage from an orbital depot eliminates the need to launch everything simultaneously. However, the fuel depot itself, and all the propellant it carries must still be launched from Earth at the cost of approximately 22 kg of launch propellant for every kilogram placed into orbit. For instance, SpaceX's concept of operations for the Starship lunar lander requires one launch to place a propellant depot in orbit, followed by approximately 19 tanker launches to transfer methane and liquid oxygen before the lander version can rendezvous and refuel.

The Space Gas Station (SGS) would eliminate this massive launch penalty by capturing propellant in situ in Very Low Earth Orbit (VLEO). The SGS goes beyond the current airbreathing propulsion concepts [3]. While it captures and stores atomic oxygen found at approximately 300 km altitude, it then converts it directly to liquid oxygen (LOX) for storage. Drag make-up at this altitude is performed by the Ambient Ion Accelerator (AIA), a gridded ion accelerator that exploits the directed ambient O^+ ions that are already present in the ionosphere.



It operates similarly to an ion thruster but has fundamental differences that make it simpler, more efficient, and less massive than any conventional electric propulsion (EP) system.

The AIA eliminates the primary subsystems that define a conventional ion thruster: there is no discharge chamber or ionization stage, no propellant feed system or flow controller, no external cathode neutralizer, and no magnetic barrier fields (see Fig. 1). Because the ambient electrons have thermal velocities (~ 190 km/s at 0.1 eV) far exceeding the orbital drift velocity (7.7 km/s), they redistribute freely to maintain plasma quasi-neutrality as ions are extracted and accelerated, and the beam is thus readily neutralized by ambient plasma. The AIA requires only two grids and a (high) voltage source. This paper presents a refined analysis of the AIA ion optics, an updated force balance, and a scale optimization framework that quantitatively establishes the SGS design space in Section 6.

Beyond the primary function of LOX production for deep-space propulsion, the SGS enables a class of high-delta-V missions that are currently impractical because of propellant costs. With the SGS as a refueling depot, a Refueling Orbital Navigator (RON) spacecraft can perform hundreds of km/s of cumulative delta-V over its mission lifetime without returning to Earth for propellant [4]. The most impactful near-term application is Active Debris Removal (ADR):

Earth's orbital environment from VLEO to GEO faces a growing threat from large defunct satellites and rocket bodies, and only a reusable, in-space-refueled system can address this at the required scale. The RON, fueled by the SGS and employing the Plasma Beam Shepherd (PBS) technique [5,6], is uniquely suited to this mission and is examined in detail in Section 7.

An advanced version of the SGS could ultimately operate at 800–900 km in the upper ionosphere, where an inflatable magnetic helical quadrupole lens would collect both H⁺ and O⁺ ions and store them as water ice. This ionospheric extension shares the concept of in-situ propellant harvesting but differs significantly in implementation; its description will be deferred to a separate paper. The present work focuses on the VLEO oxygen collection system, which alone would substantially reduce the propellant launch mass for all missions that use LOX-based propellants, i.e. virtually every liquid-fueled rocket and spacecraft.

The SGS consists of three integrated subsystems: (1) a deployable parabolic collector cone that intercepts and cryogenically captures ambient atomic oxygen; (2) an Ambient Ion Accelerator (AIA), a gridded electrostatic accelerator that uses the pre-existing ambient O⁺ ion population to provide drag make-up thrust; and (3) a solar power system that energizes the AIA. These three subsystems are described in turn, followed by an analysis of the AIA design space.

2. The SGS collector

2.1 Ambient gas collection physics

To collect a significant quantity of O, or for that matter any of the gases in VLEO, a large collector is required. The number density of the principal neutral species at relevant altitudes, obtained from NRLMSISE-00 and IRI for solar minimum and maximum at mid-latitude noon [7], is shown in Fig. 2. The nominal operating altitude of 300 km represents the midpoint between high and low solar activity, with a mean atomic O density of 10^{15} m^{-3} , which is chosen as the reference number for the SGS. Lower altitudes provide substantially higher densities, increasing by a factor of five at 250 km, but introduce increasing nitrogen concentration and drag, a trade-off that is explored further in Section 4.3 as, for some missions, gaseous species does not matter, or nitrogen may actually be preferable.

The nominal inlet radius of the collector is $R_{\text{col}} = 5 \text{ m}$. The net drag force exerted by the directed neutral flow striking the inlet at orbital speed is:

$$F_{\text{drag}} = (n_{\text{O}} \cdot m_{\text{O}} \cdot v_{\text{orb}} \cdot A_{\text{inlet}}) \cdot v_{\text{orb}} = [(10^{15})(2.67 \times 10^{-26})(7700)(\pi)(5^2)] \cdot (7700) = 0.12 \text{ N}, \quad (1)$$

where the expression in brackets is the oxygen mass flow, $\text{d}m_{\text{O}}/\text{d}t$. Thus, the structural requirements for the collector are minimal. It is envisioned that the collector could be a relatively thin membrane supported by staves, much like an umbrella, and like an umbrella, be contracted to a much smaller footprint for launch than the 5 m radius (see Fig. 1). Integrating the mass flow, the yearly oxygen mass flow into the collector is approximately 510 kg at 300 km altitude. As will be shown in Section 2.3, the efficiency of collection, though quite high, reduces the amount collected to 490 kg/yr.

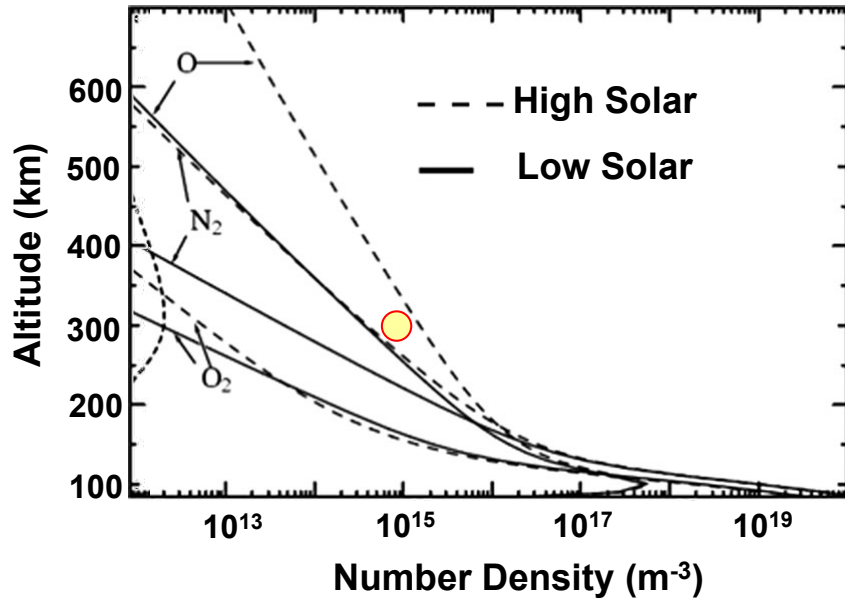


Figure 2. Number density of the principal neutrals O, O₂, and N₂ verses altitude obtained from NRLMSISE-00 and IRI for solar min. and max. at mid-latitude noon.[7] The circle represents the altitude corresponding average range of O number density assumed in the SGS preliminary analysis.

Surface accommodation of the incoming O atoms at the collector wall is well characterized in the satellite drag literature. Fifty years of orbital drag measurements on dozens of satellites have established that surfaces in the 150–300 km altitude range are coated with a monolayer of adsorbed atomic oxygen, driving the energy accommodation coefficient to greater than 0.95. For the SGS collector the inner surface would be ~ 110 °K, which is the temperature of a surface shaded from the sun. The adsorption is thus even more favorable than for a typical satellite exposed to the sun (~ 290 °K), and the exothermic recombination of atomic O to O₂ at the wall provides an additional energy dissipation channel. Therefore, the assumption of full thermal accommodation expressed in Eq. (1) as well as Eq. (2) is well-supported.

Thermal analysis indicated that the heating of the collector by the O atoms was negligible compared to the solar heat flux of 1.38 kW/m^2 onto the surface of the collector. The orbital directed kinetic energy per O atom is 5 eV with an additional $\frac{1}{2} \cdot 6.4 \text{ eV}$ from the exothermic recombination to O₂ and an additional 0.1 eV from the O thermal energy. The energy deposition rate per unit area onto the collector surface is thus $(8.3 \text{ eV} \cdot k_B)$ times the particle flux ($v_{\text{orb}} \cdot n_{\text{O}}$), or

10.2 W/m², which is quite small and miniscule compared to the solar heat flux. For the reference design ($A_{\text{inlet}} = 78.5 \text{ m}^2$) the total heating power is only 800 W. For a conical collector with a height to base radius ratio of 2, the collector surface area $A_{\text{col}} = 175.6 \text{ m}^2$ which results in a further reduction of the heat flux to 4.56 W/m².

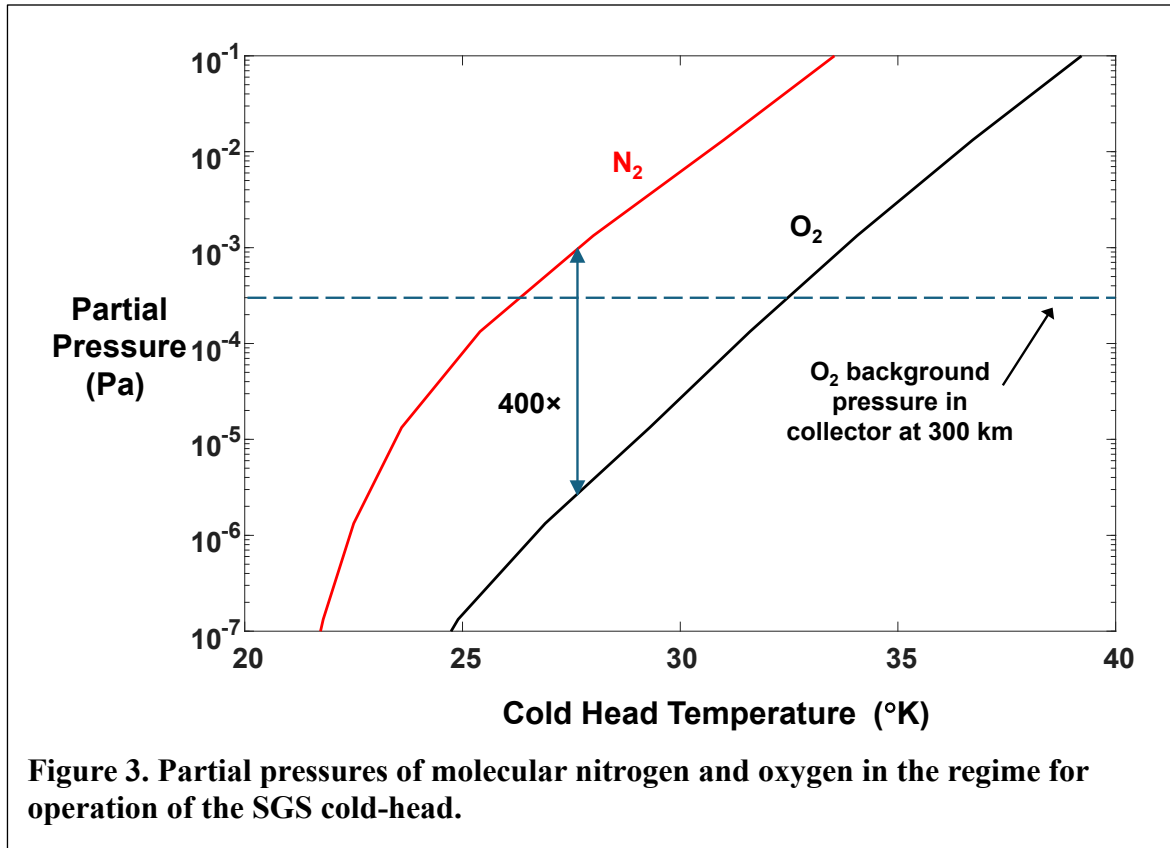
Because the force is also so small, the structural demands on the collector are minimal. It is envisioned as a thin membrane supported by staves, much like an umbrella that can be contracted to a small footprint for launch and re-expanded on orbit. There are other advantages to this type of deployment, as will be discussed later in Section 2.5, where propellant collection and storage methods are discussed. As mentioned, the thermal accommodation provided by the wall interaction with the O atoms decreases their energy to that of the collector wall. With solar shading, the equilibrium temperature of the wall (0.0095 eV), substantially diminishes the backflow energy and thus velocity of the O₂ molecules after wall interaction. In fact, the power flow of the O₂ gas off the conical back wall into the chamber is miniscule:

$$P_{O_2} = \frac{dN_{O_2}}{dt} E_{O_2} = n_{O_2} v_{orb} A_{\text{inlet}} k_B T_{O_2} = \frac{1}{2} (10^{15}) (7700) \pi (5^2) (1.6 \times 10^{-19}) (0.0095) = 0.46 \text{ W}, \quad (2)$$

The factor of a half is due to the doubling of the particle mass after recombination. The backflow velocity compared to the inflow at orbital speed, even for an unshaded collector, is also significantly reduced [$(\frac{1}{2} \cdot (0.025)/5.1)^{1/2} = 0.03$]. At an altitude of 300 km, the standard atmospheric pressure is approximately $8.8 \times 10^{-6} \text{ Pa}$, so that the pressure inside the collector will rise to $\sim 3 \times 10^{-4} \text{ Pa}$.

2.2 Collector architecture and energy management

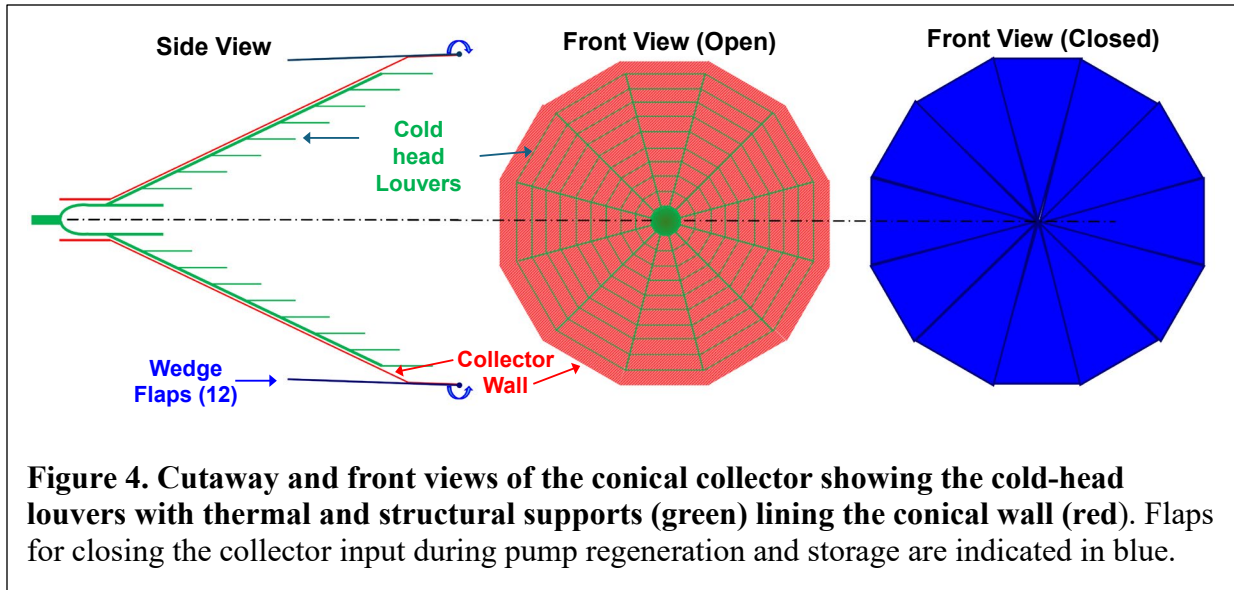
Two requirements govern the collector design, and in the most direct embodiment they are in tension. First, the captured oxygen must be retained and not allowed to flow out of the collector. A cryogenic surface is needed to condense it, and for the local pressures inside the collector (of order 10^{-4} Pa) the saturation vapor pressure of O₂ only begins to fall below the local ambient pressure at around 32 °K as shown in Fig. 3. At surface temperatures greater than this the surface re-evaporates as fast as it collects and provides no net pumping. Cold-head temperatures below 28 °K are therefore desired to reduce the partial pressure by at least a factor of 100 below ambient to achieve virtually complete cryo-condensation of the O₂ gas. Second, the directed flow arrives at a hyperthermal energy of 5 eV per O atom at orbital speed. If this flux is captured on the cryogenic surface, its kinetic energy is deposited there as heat. For the reference point conditions, assuming that all of the incoming directed O energy is captured by the cold-head surface on first pass, the total power absorbed is 484 W. Removing this load at 28 °K, where space cryocoolers exhibit a specific power of order 80 W_e/W_{th}, would require nearly 40 kW of electrical cooling power, which would dominate the entire vehicle power budget.



The collector design resolves this tension by separating the surface that absorbs kinetic energy from the surface that condenses oxygen. The collector consists of a relatively warm conical wall and a cryogenic, louvered cold-head that is placed along the wall as shown schematically in Fig. 4. The collector wall is allowed to float at its radiative equilibrium temperature. Given that it is in shadow and that the cold-head is held at 25 to 30 °K, the condensation occurs without the large heat load. The geometry is arranged such that the incoming directed beam strikes the warm wall thereby depositing its kinetic energy where it can be radiated away, while the cold-head intercepts the slow, thermalized gas that the warm wall subsequently re-emits.

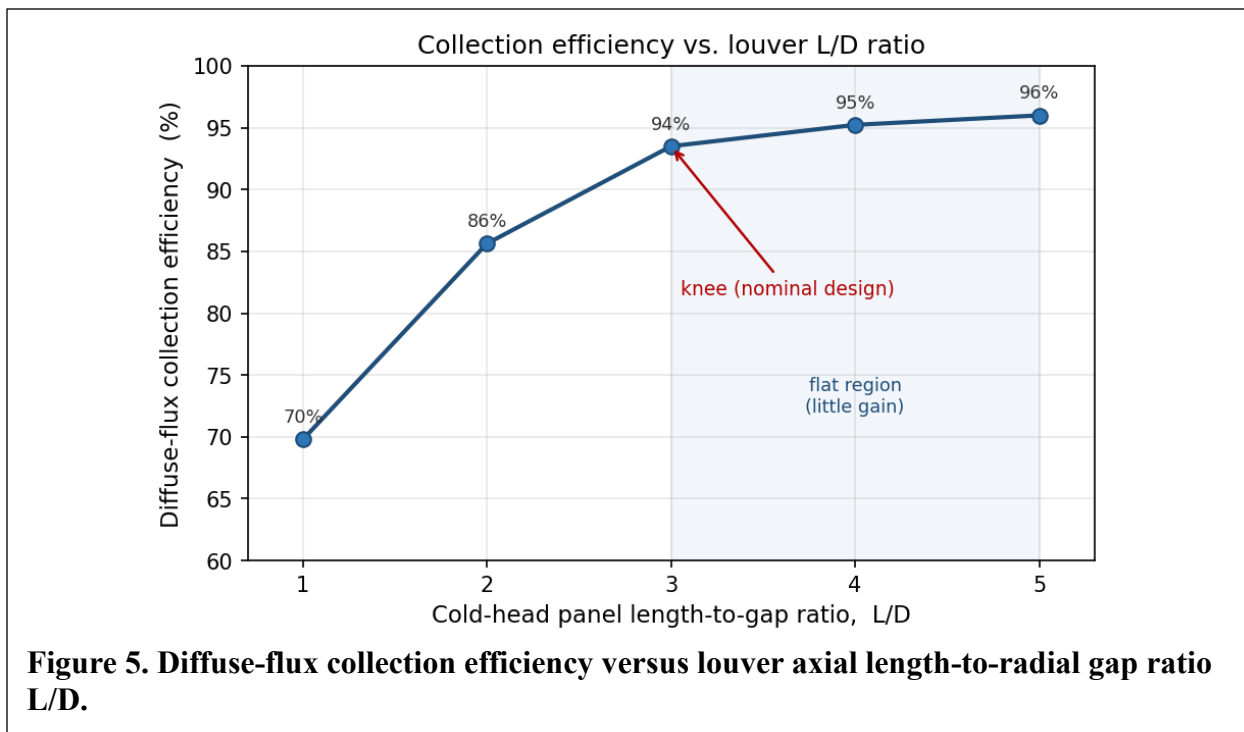
2.3 Velocity-selective louvered cold-head

The cold-head is an array of nested, multisided, nearly cylindrical louvers that are staggered axially to align with the conical wall. The louvers are thermally connected to the cryopump, as well as held structurally in place, by flat plate staves in the r - z plane. Both louvers and staves are set edge-on to the O flow and thus represent minimal adsorption targets. Each louver has the same axial length-to-radial gap ratio L/D , which is a key design parameter. The incoming flow is strongly collimated: with a directed-to-thermal speed ratio $S = v_{\text{orb}}/v_{\text{th}} \approx 7.5$ where the thermal spread corresponds to $\sim 1,000$ °K. The beam has an angular half-width of only 8° . The channel



with a ratio of $L/D = 5$ accepts angles up to $\arctan(1/5) \approx 11^\circ$. Because the beam is narrower than the channel acceptance, the directed flow passes through the louvers with little interaction and continues on to strike the warm conical wall behind them.

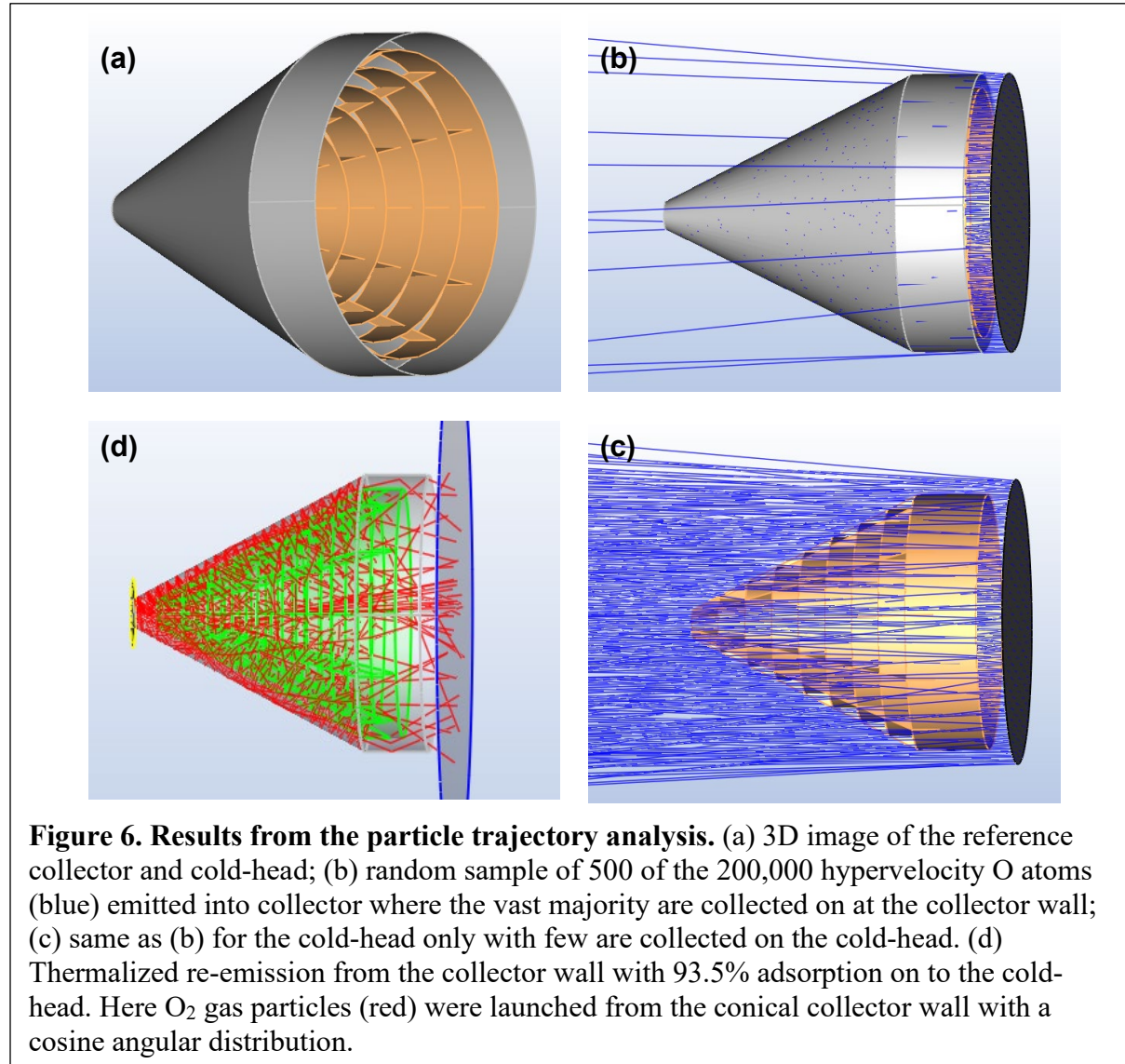
At the collector wall, the atoms accommodate, deposit their 5 eV of directed energy and any recombination energy, and are re-emitted as O_2 diffusively at the wall temperature. This thermalized population has a broad (cosine) angular distribution, so that the great majority of the particles arrive at the louvers at angles well beyond the 11° acceptance, are then intercepted and condensed. The louver array is thus transparent to the hyperthermal beam but nearly opaque to the diffuse return flux acting as a velocity- and angle-selective filter that routes the kinetic



energy to the warm wall and the mass to the cold-head. The condition $L/D < 5$ places the selectivity threshold between the directed and thermal populations. A scan of L/D (Fig. 8) shows the collected fraction is relatively flat from $L/D = 5$ down to $L/D = 3$ and then falls off rapidly. The $L/D = 3$ ratio is adopted as the nominal design as it retains essentially the full collection efficiency while minimizing louver length, material, and most critically heating from the direct interception of the incoming hypervelocity O atoms.

2.3 Collection efficiency from particle-trajectory calculations

The collection performance was evaluated by employing a three-dimensional particle-trajectory code of the conical collector and louvered cold-head. To characterize the collection



efficiency and obtain a measure of the direct heating of the cold-head approximately 200,000 particles with the mass of atomic oxygen were launched from a planar emitter somewhat larger than the collector inlet to ensure full coverage. The particles were given a directed velocity along the axis of symmetry equal to the orbital velocity at an altitude of 300 km (equivalent directed

energy of 5 eV) with a Maxwell–Boltzmann thermal distribution at $T_0 = 1000$ °K. At this altitude the collisionless, free-molecular regime (Knudsen number $\gg 1$) makes the capture fractions independent of the absolute scale, so the results for the 5 m reference collector apply equally well to any practical scale for the SGS collector. Figure 6 displays the several aspects of the various calculations that were carried out. A 3D image of the reference collector and cold-head is shown in Fig. 6a. Twelve staves were employed here as in Fig. 4, however the louvers in this case were made cylindrical to simplify comparison with differing numbers of staves. This simplification was justified as the difference in the results of the trajectory analysis was negligible compared to a twelve-sided louver.

On the first (directed) pass, the great majority of the entering atoms strike the warm conical wall as can be seen in in Fig. 6b where a random sample of 500 of the 200k trajectories is shown. As can be seen, there is a small fraction of particles that escape the collector on first pass due to the small gap between the emitter and the SGS collector. In order to get a measure of the fraction of particles striking the cold-head louvers and staves, the same calculation can be performed with only the cold-head as a collector. The resulting sample of 500 trajectories can be found in Fig. 6c where the result is essentially indistinguishable from the no collection case with only $\sim 6.8\%$ intercepted directly by the cold-head. Roughly 0.6% of the particles pass out through the small pump-out aperture at the apex. The worst case was actually employed here that had axially the longest louvers tested ($L/D = 5$). Thus, the directed beam deposits the large majority of its kinetic energy on the warm wall, as intended, with only a small fraction reaching the cold-head on first contact.

The second simulation modeled the thermalized re-emission of the particles from the collector wall and adsorption on to the cold-head. Here O_2 gas particles were launched from the conical collector wall with a cosine angular distribution to the normal at $T_{O_2} \sim 100$ °K (no directed velocity). The wall was also specified as a reflector for incoming particles so that particles missing the cold-head remain confined in the collector until captured or lost out the entrance to the collector. This process is depicted in Fig. 6d where a random sample of 1,000 particle trajectories of the 150,000 particles emitted from the wall are shown in red. Of this population, $\sim 93.5\%$ was collected by the louvered cold-head at the nominal $L/D = 3$. In Fig. 6d the cold-head is shown in green wire frame to give more transparency to the particle trajectories. Combining the two passes, the overall fraction η_c of incoming atoms captured is:

$$\eta_c = 0.068 + (0.926)(0.935) \approx 0.93. \quad (3)$$

This is a conservative value, because the calculation treats the conical wall as a specular reflector for the small fraction of re-emitted atoms that miss the cold-head on the first encounter. With a diverging cone a specular wall reflection assists in the escape, with each reflection routing an outward-bound trajectory progressively toward the axis and out the exit. As has been discussed, the actual situation is an almost purely diffuse reflecting wall, which randomizes the direction at each bounce, returning roughly half of these atoms inward for another opportunity to be captured. Since $\sim 90\%$ of the re-emitted flux is collected on first encounter, only the residual

~10% is affected by the purely reflective assumption. Accounting for the diffuse re-reflection raises the total capture by an estimated 2–3%, to approximately 96%. The corresponding oxygen collection rate is ~490 kg/yr at the 300 km reference point.

The directed beam thus carries the oxygen atom energy to the collector wall, and the diffuse return delivers the mass to the cold-head. Should the direct first-pass interception (and its associated cold-head heating) prove limiting, it can be reduced substantially by shortening the louvers: cutting the panel length by a factor of two from $L/D = 5$ to 2.5, lowered the first-pass cold-head strikes from ~12% to 5.5%, which more than halved the direct kinetic load at the cost of a modest reduction in collection efficiency (Fig. 5). Thus, the panel length is a functional handle that trades collection efficiency for cold-head heat load.

2.4 Thermal analysis

The collector wall absorbs the kinetic energy of ~87-94% of the beam that reaches it along with the oxygen recombination energy released there and radiates this load to space. For the reference collector it was previously found that total power release into the collector wall was $P_{\text{wall}} = 800 \text{ W}$ spread over a wall area A_{wall} of 176 m^2 . Assuming an emissivity $\varepsilon = 0.85$, the radiative equilibrium temperature is

$$T_{\text{wall}} = \left(\frac{P_{\text{tot}}}{\varepsilon \sigma A_{\text{wall}}} \right)^{1/4} = 98.5 \text{ }^\circ\text{K}. \quad (4)$$

This rises to ~105 °K once the external Earth's IR loading is included. The collector wall therefore floats near 100 °K which is well above the ~30 °K condensation threshold, so oxygen does not accumulate on it, yet cold enough that the re-emitted gas velocity is low (a few hundred meters per second) minimizing the heat load with its subsequent capture on the cold-head.

The cryogenic heat load on the cold-head has four contributions: (i) the kinetic energy of ~7% of the beam intercepted directly, which is at most 32 W at full accommodation, and roughly half that under the grazing-incidence accommodation discussed below ; (ii) the condensation latent heat, ~ 4 W; (iii) the thermal energy of the captured re-emitted gas, ~1 W; and (iv) radiative exchange with the collector warm wall and the entrance. This is controlled to 5-50 W by low-emissivity external surfaces and avoidance of direct and indirect solar radiation. The dominant uncertainty is the disposition of the recombination energy (6.4 eV per molecule). If the collector wall is catalytic for recombination, this energy is released on to the radiating wall. If the wall is inert and atoms reach the cold-head still atomic, up to ~240 W is deposited there. To keep the recombination heat off the cold-head, a catalytic wall coating will be an important design element if it becomes an issue.

There is a further reduction related to contribution (i) concerning the direct interception of the hypervelocity oxygen atoms by the cold-head louvers. The directed O beam strikes the louvers at near-grazing incidence, such that the energy accommodation coefficient governing heat deposition is well below unity, and the perpendicular energy fraction is small. Moreover, atoms striking the thin physisorbed oxygen-ice layer (binding energy ~0.1 eV) at grazing

incidence can sputter that loosely bound layer, carrying directed energy off the cold-head with the ejected molecules. Because the collector is a nearly closed cold cavity, such sputtered and reflected particles do not represent lost collection as they will then encounter the collector wall or an adjacent louver and are accommodated and pumped, and energy-shedding does not reduce the capture. The effective accommodation coefficient and the recombination-energy partition are the two quantities that most warrant experimental measurement, for example by calorimetry of a hyperthermal-oxygen beam on a cold, ice-covered louver coupon at the relevant grazing angle.

Table 1. Cold-head cryogenic load and corresponding 20 K cooling power for the 5 m reference collector at the nominal $L/D = 3$ design point (~ 490 kg O_2 /yr).

Cold-head load case	Cold load (W)	Cryocooler (kW)	Governing factor
Optimistic (grazing + catalytic wall, low- ϵ)	$\sim 15\text{--}25$	$\sim 1\text{--}2$	recomb. on wall
Most likely range	$\sim 25\text{--}50$	$\sim 2\text{--}4$	partial recomb.
Conservative (inert wall, full accommodation)	$\sim 250\text{--}300$	$\sim 20\text{--}24$	Recomb. on cold-head

Table I indicates the various possible cryocooler power demand across the entire range of possibilities with a cold-head cooling power from 1 to 24 kW, with a most-probable value of $\sim 2\text{--}4$ kW. Under all circumstances the SGS collector can achieve near-unity capture at a cryogenic cost compatible with the overall power budget.

2.5 Storage duty cycle and regeneration

Because the cold-head also serves as the primary store, the collector operates in long collection intervals punctuated by brief regeneration. The surface area of the reference cold-head shown in Fig. 5 including both louvers and staves is 400 m^2 . The areal deposition rate at the reference point is thus approximately 3.4 g/m^2 per day, so the frost grows at a few μm per day. At $25\text{--}30\text{ }^\circ\text{K}$, solid oxygen exists in its β -phase. This phase is stable between $24\text{ }^\circ\text{K}$ and $44\text{ }^\circ\text{K}$. In this temperature range the density of solid oxygen is approximately $1,400\text{ kg/m}^3$. The standard operational limit of roughly a 1.0 cm thick frost layer for cryo-panels is assumed here before there is a need to regenerate. Beyond this thickness, given the thermal conductivity of the solid oxygen, the increasing temperature gradient would begin to allow the outer layer temperature to rise above the threshold desired to maintain the partial pressure to at least $100\times$ less than that of the ambient collector pressure during collection (see Fig. 3) thereby effectively decreasing the pumping efficiency. At a thickness of 1 cm , the accumulated mass on the cold-head would be $5,600\text{ kg}$ prior to regeneration. As this could take over 10 years at 300 km altitude, regeneration is only performed when and where it is desired to pump and transfer the oxygen based on whatever the mission requirements might dictate.

Regeneration is accomplished by closing the collector to confine the gas and warming the louvers and staves using a thin resistive element bonded to the wall. This would drive the

condensed oxygen off to a mechanical pump and storage. Regeneration and cooldown would thus take minutes to hours, against a collection interval of possibly years. The duty cycle exceeds 0.999 and the effective capture is essentially equal to the instantaneous value. The amortized regeneration heating power would be on the order of watts.

There are two feasible options for closing the collector for regeneration. The most straightforward method is to rotate lightweight wedge-shaped flaps as depicted in Fig. 4. These flaps have been stowed behind the conical opening to avoid any drag during collection. The flaps have a slight overlap and electro-magnetic catches to ensure that no significant gas escapes during pump out. The second method would be to reclose the collector to its original launch configuration which would minimize the size of the closure curtain as well as significantly reduce drag during the regeneration process. The downside is that the process is much more involved and risks damage or misalignment of the cold-head louvers. The flaps are simple and have far less risk but do create a small mass penalty.

It is clearly advantageous to have a collapsible collector at launch, and this can also work for the fixed collector and flap option. The analogy to an umbrella is appropriate here. Because there are no significant forces acting on the collector or cold-head, a key operational characteristic of the staves for both the collector and cold-head deployment is to go from the collapsed state of a thin, membranelike structure to one where it is held in taught alignment as in the umbrella. For the cold-head louvers this can be achieved by employing copper wire cloth or metalized film



with passive tensioners.

The collector is envisioned as a thin silvered Teflon membrane ($\sim 50 \text{ g/m}^2$) supported by 12 composite staves similar to those of the Advanced Composite Solar Sail System (ACS3) [8], which fit an 84 m^2 sail into a 12U CubeSat. An interesting analog to what is desired here is the

Godox light box shown in Fig. 7. The staves here are external to the light box with no internal supports. The close-packed square ‘louvers’ here produce a parallel beam of light going out rather than allowing a parallel beam of atoms to come in but functionally the same as the louvers.

3. SGS propulsion – the Ambient Ion Accelerator (AIA)

Drag make-up is provided by the Ambient Ion Accelerator (AIA), a gridded electrostatic accelerator positioned as an annular ring downstream of the collector (see Fig. 1). The AIA accelerates the ambient O^+ ions that are already present in the ionosphere with a thermal energy of roughly 0.1 eV which drift with respect to the SGS at the orbital velocity (directed energy ~ 5 eV). The AIA exploits several intrinsic advantages of the ambient ion population which fundamentally distinguish it from a conventional ion thruster. In a conventional ion thruster, the extracted ion beam carries a net positive charge, and an external cathode neutralizer is required to maintain spacecraft potential. In the SGS, the ambient electrons have a thermal velocity of approximately 190 km/s at $T \approx 0.1$ eV which greatly exceeds the 7.7 km/s orbital drift, so they redistribute freely and maintain plasma quasi-neutrality without an external neutralizer cathode. The ambient O ions are created by solar UV radiation in the ionosphere; so that the ionization energy ‘tax’, the dominant loss channel in any EP thruster, is completely absent here. This also means that no discharge chamber, ionizer, or propellant feed system is required. The highly directed orbital flow also simplifies the ion optics significantly compared with an isotropic plasma source. As can be seen in Fig. 8, the ambient plasma density at 300 km is $n \approx 10^{13} \text{ m}^{-3}$

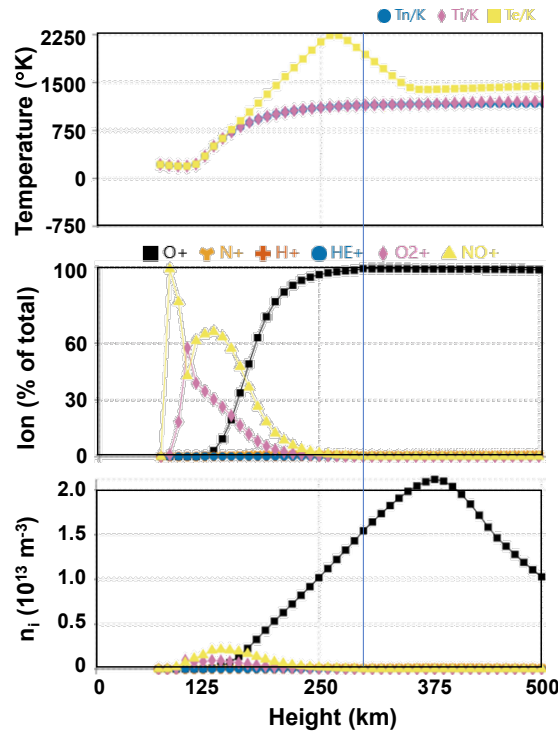


Figure 8. Density, temperature and ion fraction for principal ion species in the ionosphere on 3/21/2023. (IRI 2020 model) [7].

which is approximately 4 orders of magnitude lower than the number densities in NSTAR/NEXT thrusters. Because the SGS operates in the molecular flow regime, the grid transparency, α , is a free design parameter unconstrained by charge-exchange, grid erosion or space-charge optics. The only hardware required for the AIA is a pair of grids and a voltage source.

3.1 AIA force balance

The drag make-up condition equates the neutral O momentum flux intercepted by the collector to the thrust produced by the ion beam:

$$n_O v_{\text{orb}}^2 A_{\text{inlet}} = \alpha n_{O^+} v_{\text{acc}} v_{\text{orb}} A_{\text{AIA}} \quad (5)$$

where $n_O = 10^{15} \text{ m}^{-3}$ is the neutral O density, $n_{O^+} = 10^{13} \text{ m}^{-3}$ is the ambient O^+ density, $v_{\text{orb}} = 7.7 \text{ km/s}$ is the orbital velocity, $A_{\text{inlet}} = \pi R_{\text{col}}^2 = 78.5 \text{ m}^2$ is the collector inlet area, $A_{\text{AIA}} = k \cdot A_{\text{inlet}}$ is the AIA grid area ($k = 5$ for the design reference), and $\alpha = 0.975$ is the grid open-area fraction. Solving for the required exit velocity one has:

$$v_{\text{acc}} = \frac{n_O}{n_{O^+}} \cdot \frac{A_{\text{inlet}}}{\alpha A_{\text{AIA}}} \cdot v_{\text{orb}} = 158 \text{ km/s}. \quad (6)$$

Only the wire-blocked fraction $(1-\alpha)$ receives the momentum of the incoming neutral O atoms on the AIA structure. The total drag to be compensated is therefore:

$$F_{\text{drag}} = m_O n_O v_{\text{orb}}^2 [A_{\text{inlet}} + (1-\alpha) A_{\text{AIA}}]. \quad (7)$$

For the design reference ($\alpha = 0.975$, $k = 5$), the collector contributes 0.124 N and the AIA wire drag contributes only 0.015 N, resulting in a total drag of 0.139 N. This is balanced by an AIA operating voltage of 2606 V. Operating at $V = 3,000 \text{ V}$ produces a thrust of 0.149 N and a net positive force of 0.010 N, providing a reasonable margin for other structural drag sources.

Above break-even, the net force and required power both scale as R_{col}^2 (collector radius squared), so the thrust-to-power ratio is to some extent independent of scale. The break-even voltage is also independent of the collector radius, depending only on α , the ambient density ratio $n_{\text{neut}}/n_{\text{ion}}$, and the area ratio k .

3.2 Child-Langmuir grid gap

The separation between the screen (high-potential) and accelerator (ground) grids is set by the Child-Langmuir space-charge-limited current condition. The relevant current density is the conserved beam current density, which is constant throughout the gap owing to current continuity:

$$J = e n_{O^+} v_{\text{orb}} = 0.0123 \text{ A/m}^2 \quad (8)$$

As ions accelerate through the gap, their number density falls as $n(z) = n_{O^+} v_{\text{orb}}/v(z)$ by continuity, so the product $n(z) \cdot v(z)$ remains constant. Substituting J into the Child-Langmuir formula:

$$d = \left[\frac{4\epsilon_0}{9J} \sqrt{\frac{2e}{m_0}} \right]^{1/2} V_{grid}^{3/4} = 42.7 \text{ cm} \quad \text{at} \quad V_{grid} = 3000 \text{ V} . \quad (9)$$

A large gap is advantageous across every design metric. The electrostatic pressure between grids scales as $E^2 \propto 1/d^2$. The electric field at 3,000 V operating voltage is 7.0 kV/m which is three orders of magnitude below the Kilpatrick breakdown threshold for clean aluminum (~ 5 MV/m), providing a 714 $\times$ safety margin. The grid aspect ratio, R_{gg} (gap-to-grid size ratio) for the reference design is $R_{gg} \sim 23$, well below the R_{gg} on the order of 300 that is characteristic of the NSTAR and NEXT ion thrusters. Carbon composite standoffs at 50 $\times$ 50 cm spacing can be used to maintain the gap under launch loads and thermal cycling with negligible mass penalty.

3.3 Ion optics, aperture geometry and grid design

A dedicated 2D cylindrical Poisson solver coupled with a leapfrog particle tracker was developed to characterize the AIA ion optics. The solver finds the electrostatic potential $\phi(r,z)$ in the gap between the two grids using finite differences with the screen grid at $+V$ and the accelerator grid at ground and then integrates the equations of motion for a large ensemble of O^+ test particles launched at the inlet. The code was validated against the analytical Child-Langmuir solution (Eq. 8) and confirmed energy conservation to within 0.13% across the full gap.

The ambient O^+ population follows a drifting Maxwellian: axial velocity $v_z = v_{orb} + \delta v_z$ where δv_z is drawn from a Gaussian of width $\sigma = (kT/m_0)^{1/2} = 776$ m/s, and transverse velocity v_r is drawn from a Rayleigh (2D Maxwell-Boltzmann) distribution with the same width σ . As before, the resulting inlet angular half-spread is 8.1° , corresponding to $v_{orb}/v_{th} = 7.0$ where $v_{th} = (2kT/m_0)^{1/2} = 1,098$ m/s is the thermal speed.

The beam perveance, $P = I/V_{grid}^{3/2}$ can be compared to the Child-Langmuir maximum perveance $P_{max} = (\pi\epsilon_0/9)(2e/m_0)^{1/2}(D/d)^2$ per aperture of diameter D . At the design point,

$$\frac{P}{P_{max}} = \frac{I / V_{grid}^{3/2}}{(\pi\epsilon_0 / 9)\sqrt{2e / m_0} (D / d)^2} \approx 0.975 . \quad (10)$$

This places the AIA in the geometric optics regime where space charge contributes negligibly to beam divergence. This means divergence is set entirely by the inlet angular spread and the electrostatic lens action at the aperture rims.

As ions accelerate from 5 eV (orbital KE) to 3,005 eV (orbital + 3,000 V), the transverse-to-axial velocity ratio is compressed by the factor $(E_{inlet}/E_{exit})^{1/2} = (5/3005)^{1/2} = 0.04$. The mean exit half-angle from the simulation using the drifting Maxwellian distribution one has:

$$\theta_{exit} = \theta_{inlet} \sqrt{\frac{E_{inlet}}{E_{exit}}} = 8.1^\circ \times \sqrt{\frac{4.92}{3005}} = 0.33^\circ \quad (11)$$

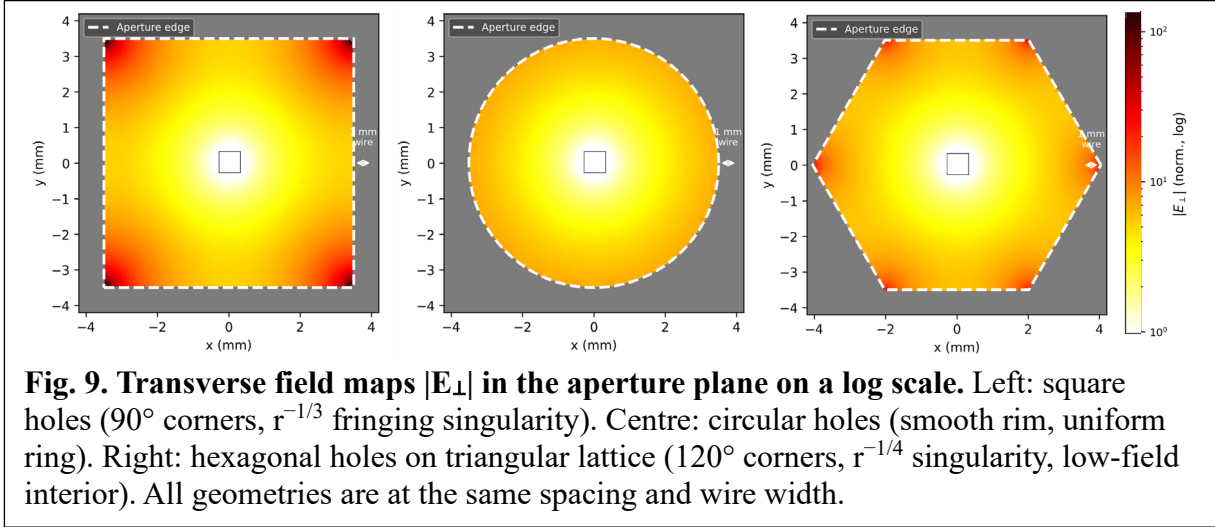
This is essentially perfect collimation: the thrust vector loss is less than 0.003%. Varying the aperture size or grid gap by a factor of ten has no measurable effect on this result, confirming the geometric optics interpretation.

The AIA grid can in principle employ any aperture shape be it circular, square, or hexagonal. Conventional ion thrusters use circular apertures because their smooth rim produces a uniform azimuthal electric field at the aperture edge, minimizing beam deflection and thus wall erosion from the ions. The radial component of the electric field becomes exponentially large near the corners of the noncircular lattices with E_r given by:

$$E_r \propto r^{(\pi/\alpha_w - 1)} \quad (12)$$

where α_w is the inner wire angle. For the AIA, where the beam is already highly directed and the divergence is compression-limited, the primary selection criteria are the open-area fraction and manufacturing practicality.

The transverse (radial) electric field $|E_\perp|$ within the aperture plane was computed for each geometry by solving the 2D Laplace equation with the wire grid boundary at the grid potential and the aperture interior free to relax with a reference potential of half the grid voltage at the aperture center. Figure 4 shows the resulting $|E_\perp|$ maps on a logarithmic scale for all three geometries at the same center-to-center aperture spacing (8 mm) and wire width (1 mm) used for



the comparison study. The boundary of each aperture is indicated by a dashed white line.

The field topology of each geometry is apparent in Fig. 9. The square aperture shows strong field concentration at its four corners, a direct consequence of the $r^{-1/3}$ singularity associated with a 90° re-entrant corner: ions passing near a corner receive a transverse impulse that slightly increases beam divergence. With no corners, the circular aperture shows a smooth, azimuthally uniform ring of field at its rim. While there is no preferred deflection direction, the lattice packing limits the open-area fraction. The hexagonal aperture presents the optimal solution: 120° interior angles at all six corners produce the weaker $r^{-1/4}$ fringing singularity, the 6-fold

symmetry ensures no net transverse bias in any radial direction, and the triangular lattice provides the maximum possible open-area fraction for any regular tessellation.

For equal open-area fraction, hexagonal holes on a triangular lattice require smaller wire width than the alternatives, resulting in lighter grids. For an equal wire width, hexagonal holes provide the highest open-area fraction and lowest ion interception. The hexagonal geometry is therefore optimal for the AIA on all relevant criteria.

The AIA grid uses hexagonal holes with a center-to-center spacing of 32 mm and a wire width of 0.4 mm. These dimensions were selected to satisfy two constraints simultaneously. The wire width must be mechanically adequate for launch loads and long-duration operation, and the open-area fraction must be as close to unity as practicable. A 0.4 mm wire width in 1 mm thick expanded aluminum sheet can support a 50 cm $\times$ 50 cm panel against a 10g launch load with a safety factor of 5. The resulting grid parameters are found in Table 2.

Table 2. Design Reference AIA Grid Parameters

Parameter	Value
Aperture shape	Hexagonal, flat-top, triangular lattice
Center-to-center spacing	32 mm
Wire width	0.4 mm (1.25% of spacing)
Grid transparency α	0.9752
Hex circumradius s	18.2 mm
Hex flat-to-flat width	31.6 mm
Grid material	Expanded aluminum sheet, 0.4 mm thick
Grid areal density (two grids)	0.054 kg/m ²
Standoff spacing	50 cm $\times$ 50 cm (4 per square meter)
Standoff mass	10 g each (CF tube, 8 mm OD, 43 cm long)
Total AIA areal density	0.094 kg/m ² (grids + standoffs)
Operating voltage V_a^g	3,000 V
Grid gap d (at 3,000 V)	42.7 cm

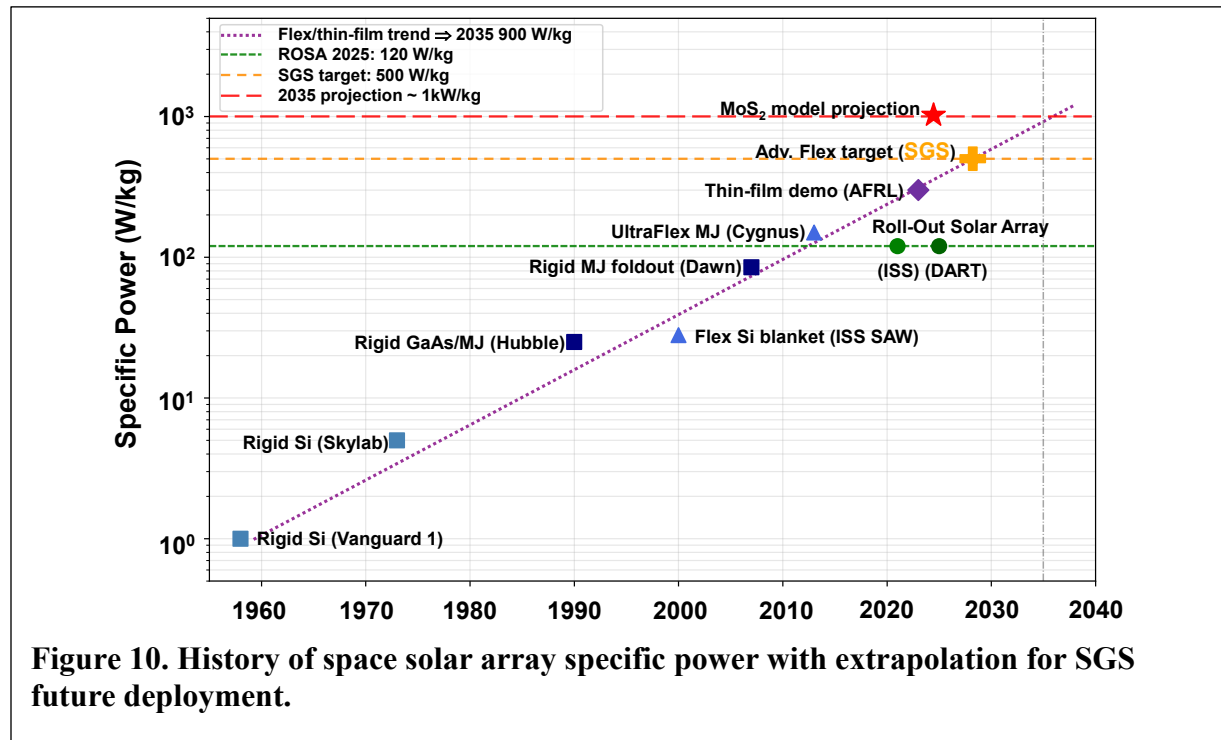
4. Solar panels and SGS orbit

The AIA and cryocooler are the only continuous power loads on the SGS. The collector and its regeneration draw negligible average power (Section 2.5). The drag-make-up beam power at the reference operating point ($V_{op} = 3$ kV, see Section 3.1) is 14.2 kW, and the cryocooler draws ~ 3 kW (80 W_c/W_{th} times the most-probable cold-head load of ~ 38 W, see Section 2.4). The total electrical demand is therefore ~ 17 kW for the reference design. Both loads, the beam power with

the captured momentum flux and the cooling load with the captured mass flux, scale as R^2 so the power-to-collection ratio is essentially scale-invariant.

4.1 Solar array

The history of space-based solar power is one which the specific power has shown exponential growth, as Fig. 10 attests. From this logarithmic plot, one could reasonably assume that given the time required to develop and begin to deploy a fleet of Space Gas Stations, the projected solar array specific power, $P_{sol} = 500 \text{ W/kg}$ will have been achieved for a large, thin-film roll-out array of the class now being developed. The roughly 17 kW power demand would then correspond to $\sim 34 \text{ kg}$ of solar array. Such arrays can be stowed compactly for launch and



deployed in orbit, consistent with the membrane-and-stave deployment philosophy of the collector itself. The array is the second-largest mass item in the budget to be discussed in Section 5. The specific power will thus be a primary factor on total system mass and on the payoff time to be examined in Section 6.

4.2 Sun-synchronous dawn–dusk orbit

A sun-synchronous orbit (SSO) is selected for the SGS, and the dawn–dusk variant in particular offers several advantages. Because the orbit plane remains nearly perpendicular to the Sun line throughout the year, the spacecraft is continuously illuminated and experiences essentially no eclipse. This removes the need for large battery storage to carry the AIA and cryocooler through the earth’s shadow resulting in a substantial mass saving for a continuously-powered system. This orbit also provides a stable thermal environment avoiding the cyclic heating of a terminator-crossing orbit.

The dawn–dusk geometry is also favorable for the collector thermal design. With the orbit plane along the terminator, the collector axis (aligned with the ram direction) lies nearly perpendicular to the Sun line, so the collector interior and the cold-head are shaded from direct sunlight. With the sun in a fixed direction, this shading, together with the addition of a simple, fixed orientation sunscreen allows the collector wall to float near 100 K rather than the ~290 K of a sunlit surface, and it minimizes the solar load on the cold-head that would otherwise dominate the cryogenic budget (Section 2.4). The SSO further aligns the accelerated ion beam favorably with the geomagnetic field. While the transverse deflection of the low-energy ambient ions is small under any circumstance, with the SSO the residual deflection is insignificant.

4.3 Elliptical Collection Orbit (ECO)

The collection rate scales linearly with ambient atomic-oxygen density, which rises by roughly a factor of five between the 300 km reference altitude and 250 km (see Fig. 2). A circular orbit at the lower altitude would collect gas at a proportionally faster rate but at a continuous drag and orbital-decay penalty. An Elliptical Collection Orbit (ECO) decouples these by lowering perigee into the dense region while keeping apogee higher in a low-density region. The SGS collects in concentrated bursts near perigee where density, and hence both capture rate and drag, are greatest while spending the majority of each orbit at higher altitude where drag is negligible. In addition, as can be seen in Fig. 8, the ion density can actually increase at higher altitudes compensating for the increased drag at lower altitudes. The time-averaged drag is thereby held to a level that the AIA can readily compensate, while the time-averaged collection rate is raised well above the 300 km circular value.

The ECO also turns the increased nitrogen fraction at low altitude from a liability into an asset. The higher N_2 content co-collected at perigee can be separated from the oxygen during regeneration. As indicated in Fig. 3, the two species have distinct condensation temperatures and can thus be stored in separate propellant reservoirs. The N_s can be used for on-board electrothermal thrusters for a range of operations such as drag make-up, spacecraft control and orientation maintenance, as well as orbital maneuvers which will be discussed more thoroughly in Section 7. A full optimization of perigee altitude, eccentricity, and the resulting collection-versus-drag balance is developed in the scale and orbit study of Section 6.

5. Design reference mass budget

The details of the mass budget for the design-reference SGS ($R_{col} = 5$ m, $R_{AIA} = 12$ m, $V_{op} = 3$ kV, $\alpha = 0.975$) for the SGS can be found in Table 3, which is also shown graphically in the pie chart in Fig. 11. While only the point design is given in Table 3, the dimensional scaling for each major subsystem is listed in the final column and will be used in the next section for the optimization study. Component areal densities are taken independent of scale as there is no structural reason to thicken the membrane, grids, or louvers with size, since the aerodynamic loads remain minute (0.14 N total drag). The area-dependent items scale as R^2 and the fixed overheads (cryocooler base, power electronics) become a vanishing fraction at larger scale.

Table 3. SGS design-reference mass budget ($R_{col} = 5$ m, $V_{op} = 3$ kV, $\alpha = 0.975$, $P_{sol} = 500$ W/kg, cryocooler sized to the most-probable cold-head load).

Component	Mass (kg)	% of total	Scaling
Collector membrane (silvered Teflon 50 g/m ²)	8.8	6%	R ²
Collector staves (12× composite booms, 65 g/m)	3.9	3%	R
Cold-head louvers + staves (metallized film, 50 g/m ²)	10.0	7%	R ²
AIA grids ×2 (expanded Al, 0.054 kg/m ²)	21.2	16%	R ²
AIA standoffs (4/m ² , 10 g each)	15.7	12%	R ²
Solar array (17.2 kW at 500 W/kg)	34.4	25%	R ²
Cryocooler (Stirling/pulse tube, ~ 40 W at < 28 °K)	20.0	15%	R ²
O ₂ storage bladder + compressor	6.9	5%	R ²⁺
PPU + electronics	15.0	11%	~ R ²
Total Dry Mass	136	100%	
Misc. components + reserve (25% of total dry mass)	34.0		
TOTAL	170		≈ R²

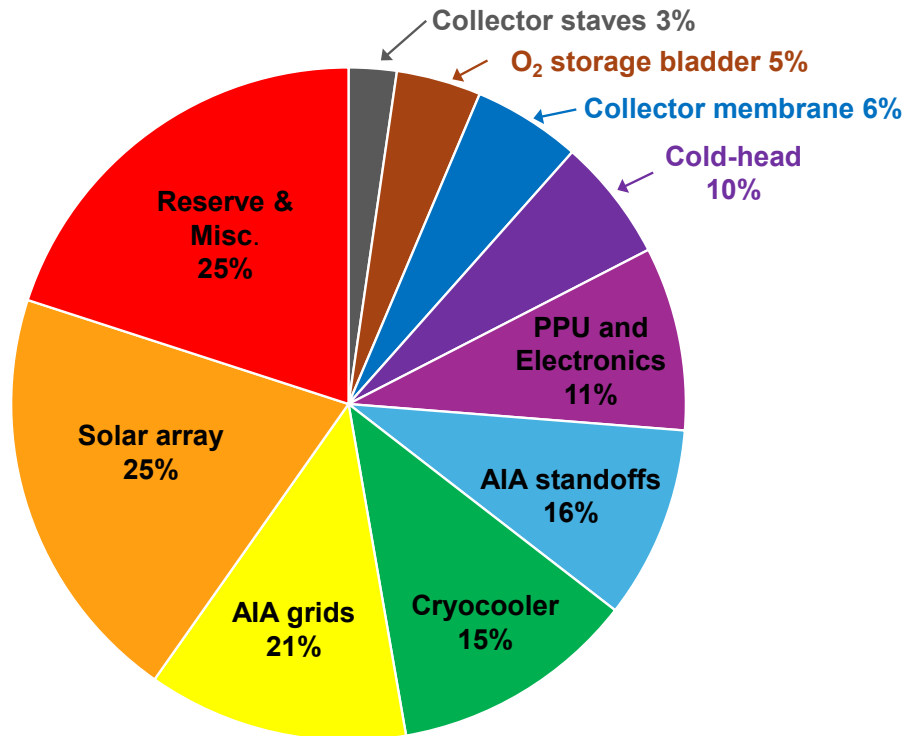


Figure 11. Fractional percentage of the total dry mass for the major components of the Space Gas Station for the nominal reference point design.

The dominant dry mass components are the solar array (25%), the AIA grids and standoffs (37% combined), and the cryogenic system comprised of the cryocooler plus cold-head structure (25% combined). The collector membrane itself remains light at 8.8 kg for 176 m² of wall area, a reflection of the negligible aerodynamic loading. This is consistent with the ACS3 solar-sail (see Fig. 12), in which 84 m² of sail and four 7 m long composite booms (0.9 kg each) together have a mass of only 12 kg. It is claimed that the composite boom technology used for this ACS3 technology demonstration could be used in future missions for solar sails up to 500 m² [8], which is roughly the size of a basketball court. Follow-on composite boom technologies now in development could enable solar sails as large as 2,000 square meters.

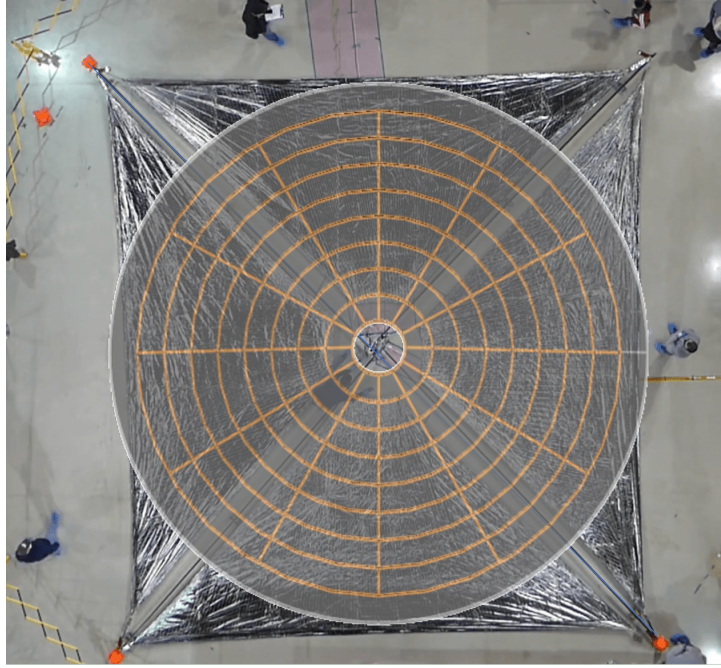


Figure 12. End-on view of the SGS at the design reference scale ($R_{\text{inlet}} = 5$ m) overlaid on an overhead picture of the Advanced Composite Solar Sail System's solar sail (ACS3).

The cryogenic system is a key subsystem of comparable mass to the collector and AIA. This stems primarily from the thermal analysis of Section 2.4. Due to its somewhat uncertain impact on the mass budget, as well as other component masses such as additional structures, thrusters, heat shields, etc. not yet determined, an additional mass reserve of 25% of the of the total dry mass is added for growth allowance in Table 3, which is appropriate for a low TRL concept. The total launch mass of ~ 170 kg compares favorably with the annual collection of ~ 490 kg of O_2 at the 300 km reference point. The mass payoff time, the interval over which the SGS collects its own launch mass in propellant, can be stated as:

$$\tau_{\text{payoff}} = M_{\text{SGS}} / (dm_{\text{O}_2}/dt) = 170/490 \approx 0.35 \text{ yr}, \quad (13)$$

which is about four months for the prototype scale of $R_{\text{col}} = 5$ m. Because the collected gas mass scales as R^2 while the fixed overheads do not, the payoff time shortens somewhat further with increasing collector size. This dependence, together with the trade against solar-array specific power and orbit selection, is the subject of the scale optimization in the next section.

6. Scale and orbit optimization

It must be said at the outset that there is significant variation in both the neutral and ion particle number densities in in the ionospheric region of LEO. As indicated in Fig. 2 the neutral density can vary considerably with altitude. The plasma (O^+) density can also vary widely as well. In the dawn-dusk sun-facing orbit there will be a recurring oscillation in density with each

orbit. Due to the exposure to the ionizing radiation of the sun, the plasma density rapidly increases during the day peaking at dusk, conversely it drops at night with a minimum at dawn. The same behavior also characterizes the neutral gas in the thermosphere as the sun heats the lower, denser regions of VLEO which expand to increase the density at ~ 300 km, helping to maintain the amount of neutral collection and ion propulsion in balance. There are, however, several other solar phenomena that change the magnitude of both the collection and propulsion aspects of the SGS. The analogy to a sailing vessel is apt here. The sailing vessel operates more efficiently when following the trade winds. For the SGS the critical system is the ion propulsion system as the ion density peaks in the F2 layer which extends over a fairly narrow range in altitude and decreases as one goes to higher or lower altitude, whereas the neutral density increases exponentially at lower altitude. To keep the balance between drag and thrust make-up, the SGS must follow the ‘trade wind’ of the F2 layer.

A sail vessel is also affected by changing seasonal weather patterns. Space weather changes not only with the seasons due to the orientation of the magnetic axis of the earth respect to the sun, but there are also the changes due to the daily sun cycle, as well as the 27-day rotation of the sun’s atmosphere. Here, solar sunspots can greatly enhance ion production in the F2 layer as the sunspots rotate and come into alignment with the earth. In addition, there is the eleven-year cycle of sunspot activity itself that can significantly affect both the ionosphere and thermosphere ion and neutral populations as Fig. 2 attests. So much like the sailing vessel, one needs to be able to adapt to the weather conditions. For the SGS there are several ways in which this can be accomplished and will now be discussed.

Starting with the dawn-dusk variation; the optimal altitude for ion propulsion is at the peak of the F2 layer which lies between 250-300 km at dawn and then rises to 350-400 km at dusk. The optimal orbit is thus an Elliptical Collection Orbit (ECO) with the perigee of 275 km at dawn and the apogee of 375 km at dusk. The ECO allows for the spacecraft to follow where its resource is most abundant in both altitude and in local time. Placing apogee on the dusk leg therefore compounds two ion enhancements, the F2-peak altitude and the dusk local-time leads to excess thrust as the neutral drag is least; perigee falls on the dawn leg, where the collector harvests the dense neutrals while the reduction in thrust in the ion-lean environment has been compensated for at apogee.

The seasonal variation is small as the SGS dawn-dusk orbit experiences both the local winter and summer on each orbit. The effect of sunspot activity, on both short (sun rotation) and long (sunspot cycle) time scales are more significant. Solar activity makes for an increased plasma and neutral density along with a higher F2 layer (400-450 km). With increased ion thrust power and collection, the power dependent components may not be able to take full advantage unless spare capacity is built in from the start. The peak power required can be factored into the SGS build and launch as the phase in the sunspot cycle will be known when the SGS will be deployed.

Variability in the match between thrust and drag will mainly be affected by sporadic events and geomagnetic storms. For the sailing vessel, the solution is to drop or add sails as the needed.

For the SGS the most straightforward way to address any significant thrust/drag imbalance, whether it arises due to remaining imbalances from the anticipated variances such as the dawn-dusk and sunspot activity is adding or decreasing the voltage on the AIA ‘sail’. The AIA thrusts continuously at constant electrical power, matched to the constant solar supply, with the voltage as the modulating knob. The thrust $F = I \times (2m_{O^+} \cdot V/e)^{1/2}$ where $P = IV$. As the area of the AIA is fixed, one has for the thrust at fixed power, $F \propto (n_{O^+} P)^{1/2}$. Thus, the voltage rises where ions are lean and falls where they are plentiful. For the reference design point and the dusk-dawn orbit the voltage swings between ~ 1.3 kV (dusk apogee) and ~ 16 kV (dawn perigee) over each revolution. The accelerator gap is sized for the apogee arc, which delivers the great majority of the per-orbit impulse, and the AIA is throttled through the perigee pass. It should be noted that the cryocooler's collection load claims the power on this arc (the two loads breathe in opposition, as noted in Section 4). Momentum balance is thus maintained per orbit.

If the sporadic events are too large or persist too long, one must do as the sailing vessel would do in a fierce storm and drop the sails or seek safe harbor. There are several ways for the SGS to also handle this issue. Again, like a sailing vessel, it can collapse the collector thereby reducing the area cross section and thus drag by a factor of roughly 70. This of course has the temporary disadvantage of eliminating collection. It could minimize this by a partial collapse, but this too can complicate deployment and jeopardize collector and cold-head recovery. The SGS can also seek safe harbor. The SGS can raise its orbit somewhat (+50-100 km) to balance drag by reduced collection. This can be complicated as the plasma density also drops with altitude as well. The safest option would be to substantially raise the orbit by up to 300 km which drops the drag by a factor of 10^3 .

When a sailing vessel gets stalled by the doldrums (in space this would be the solar minimum) the best solution is to deploy the ‘iron sail’ (i.e. outboard motor). The space equivalent would be to deploy an EP thruster that can operate on the copious, on-board store of N_2 and O_2 . This capability must also be added to the SGS as any lengthy period of uncompensated drag can quickly deorbit a gossamer-like craft like the SGS.

6.1 Backup EP thruster for SGS risk reduction

To assure the uninterrupted operation of the SGS under any space weather conditions a backup electric thruster of the appropriate power and Isp which can operate using the captured propellant. This would most likely be performed by a conventional Hall-Effect Thruster (HET) or an RF Ion Thruster (RIT) currently being designed and tested for Air Breathing Electric Propulsion (ABEP) in VLEO on the ambient O_2/N_2 mixture. Ignoring for the moment the efficiency of the EP thruster, the propellant economics for drag make-up are governed by a single ratio. The deficit thrust, $\delta \cdot F_{drag}$, will be made up by the thruster which consumes propellant at a rate $\dot{m}_p = \delta F_{drag} / v_{exh}$, while the collector captures mass at a rate $\dot{m}_c = \eta_c \cdot (1 - f_{AIA}) F_{drag} / v_{orb}$, where it was found in Section 2.3 that $\eta_c \cong 0.93$ with the AIA making up a relatively small fraction $f_{AIA} = 0.121$ of the total drag (see Section 3.1). The captured mass flux and the drag are the same incident momentum flux divided by different velocities. Assuming that the auxiliary thruster supplies all the required make-up deficit, the fraction of the collection production is, therefore:

$$\frac{\dot{m}_p}{\dot{m}_c} = \frac{\delta}{\eta_c (1 - f_{AIA})} \frac{v_{orb}}{v_{exh}} = 1.22 \delta \frac{v_{orb}}{v_{exh}}, \quad (14)$$

which is a small fraction if the EP exhaust velocity considerably exceeds the orbital velocity. As can be seen from Table 4, a RIT has been operated on N₂/O₂ with a $v_{exh} = 24.5$ km/s. Full drag make-up ($\delta = 1$) with AIA turned off would consume 38% of the gas collected but would ensure that the SGS remain fully orbit-stable under any circumstance. From Table 4 it can be seen that the RIT operates, not surprisingly, with roughly the same thrust efficiency as the AIA (~ 10 mN/kW). The issue with the RIT is that a 14-kW thruster that would be required for the full SGS reference design drag, would add additional system mass of 110 kg which is close to the mass budget for the entire SGS.

Table 4. Summary Comparison on Atmospheric Air (N₂, O₂ or Mixed)

Engineering Metric	Hall Thruster (HET)	RF Ion Thruster (RIT)	Pick for Best ABEP Thruster
Thrust Eff. (mN/kW)	15–20	8–12	Hall Thruster Higher thrust for overcoming drag
V_{exh} (km/s)	12 - 16	25	Ion Thruster Lower propellant consumption
Component Lifespan	wall/cathode erosion w O ₂	Excellent (RF designs isolate grids)	Ion Thruster for O ₂ Hall Thruster for N ₂
Thruster Sys. Mass (kg) @ 5 kW & 15 kW	25 & 65	45 & 110	Hall Thruster

From Table 4, the HET can operate at up to twice the thrust efficiency as the RIT, but with a lower exhaust velocity. It would be capable of only operating on N₂ due to rapid erosion issues using O₂. This would add some complication to its use but not necessarily a showstopper. Assuming a mean value for the thrust efficiency (17.5 mN/kW) and v_{exh} (14 km/s), the fraction consumed of the propellant collected would be 0.67 for full drag make-up and require roughly 11 kW. From Table 4 this would require a HET system mass of 50 kg, while smaller than the RIT is still a significant increase in SGS mass. However, it must be stated that it is highly unlikely that the entire AIA thrust ($\delta = 1$) would be required short of a complete failure of this system as the space weather events that would cause a significant decrease in plasma density have a similar corresponding impact on the neutral density so that the drag make-up would never be as severe as this worst case. It is also clear that the mass penalty of the RIT only matters in the time to it will

take to collect enough propellant to offset the increase in launch mass, which is a subject that will be discussed in the next section. As will be seen in Section 7, several missions enabled by the SGS will employ the use of high-power EP thrusters linked to the SGS in order to facilitate the transfer of the propellant to other spacecraft or carry out other missions in space.

6.2 Scaling with collector size

As it is not clear at this point what size or type of EP backup thruster for the SGS reference point design is appropriate, or even required, given the various operational modalities available to deal with space weather variability. It is worthwhile to understand the scaling of the SGS with size in order to begin to optimize the SGS given the various mission trade-offs and demands. The base analysis employs the mass budget of Section 5 that was summarized in Table 3 with each component evaluated for its most likely scaling with size. This permits the scale design space to be mapped. Separating the Table 3 entries by class, the SGS mass scaling with size, along with the dependence on particle density, takes the following form:

$$M_{SGS}(R, n) = 1.25 [a_2 R^2 + bR + c(n/n_0)R^2 + m_{fix}], \quad (15)$$

where $a_2 = 2.50 \text{ kg/m}^2$ reflects the components that scale with area (membrane, cold-head, grids, standoffs, storage), $b = 0.78 \text{ kg/m}$ for the stave booms, and $c = 2.46 \text{ kg/m}^2$ reflecting components that scale with power (solar array, cryocooler, and most of the PPU). These all reflect the change in drag make-up and cooling power with changing scale and neutral particle density from nominal. Here $m_{fix} \approx 8 \text{ kg}$ the fixed avionics. The leading multiplier of 1.25 reflects the 25% mass reserve of Table 3.

The collection rate is given by: $dm/dt = k(n/n_0)R^2$ with $k = 19.6 \text{ kg yr}^{-1} \text{ m}^{-2}$ at the 300 km reference density. The figure of merit is the mass payoff time $\tau_{sgs} = M_{SGS}/(dm/dt)$, the interval

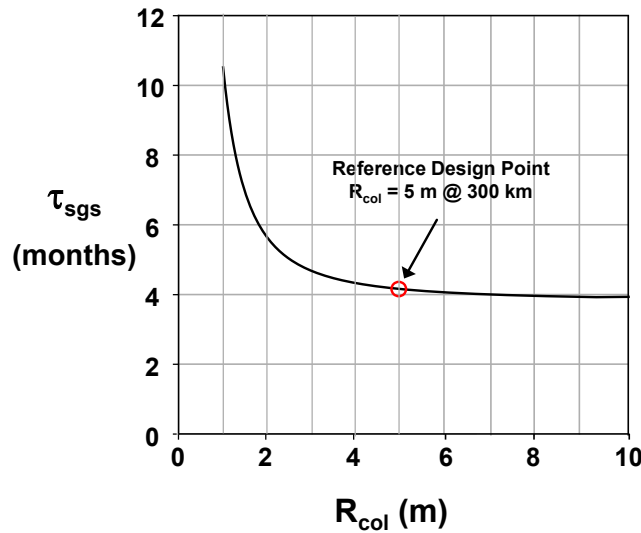


Figure 13. Mass payoff time in months versus collector radius for a circular orbit at 300 km altitude based on scaling found in Table 3 for the various SGS components.

over which the SGS takes to collect its own launch mass. This time represents the how long the SGS must operate to accumulate the same fuel mass as what could have been delivered to orbit as a fuel depot without the SGS.

Figure 13 shows τ_{sgs} against collector radius at the reference design point altitude of 300 km. Because both the dominant masses and the collection scale as R^2 , the payoff time is nearly scale-invariant. It falls from 5.6 months at $R = 2$ m to 4.2 months at the $R = 5$ m reference and approaches an asymptote of 3.8 months as the fixed overheads become negligible. The reference design is thus already within 10% of the asymptotic payoff. The practical conclusion is that scale buys throughput, not time to payoff: a larger SGS collects proportionally more oxygen but repays its own mass no faster or slower. A larger collector size should therefore be chosen to meet the desired production rate, not to optimize payoff time. Conversely, a smaller collector will be easier to assemble, put in orbit and deploy. Meeting the desired gas production with a fleet of SGSs could be a better alternative for missions such as satellite refueling and end-of-life deorbiting which would not require large quantities of gas in a single depot. Making the collector radius as small as two meters would allow for meaningful tests to further develop and validate the concept.

6.4 Design points for development of the SGS

Table 4 assembles three design points spanning the development path: the 300 km prototype covered in Sections 2 thru 5; a subscale test unit at $R = 2$, and a upscaled SGS unit at $R = 10$ m employing the 250×300 km ECO. The payoff time improves only modestly along the sequence from 5.6 to 3.8 months for the circular orbit. The ECO orbit makes a more significant difference in collection only at large scale, as expected from the near-invariance discussed in Section 6.1,

Parameter	Subscale Prototype	Reference Point	Upscaled SGS	Scaling
Collector radius R	2 m	5 m	10 m	—
Orbit	300 km circular	300 km circular (250×300 km ECO)	250 × 300 km ECO (300 km circular)	—
Mean density $\bar{n}/n_0$	1.0	1.0	1.8	—
Launch mass	37 kg	170 kg	870 kg	Eq. (12)
Electrical power	17 kW	17 kW	120 kW	$\propto \bar{n}R^2$
O ₂ + N ₂ collection	80 kg/yr	0.5 t/yr	3.5 t/yr	$\propto \bar{n}R^2$
Mass payoff time	5.6 months	4.2 months (3.8 months)	3.0 months (3.8 months)	

Table 4. SGS design points along the development path. Payoff is nearly scale-invariant; production rate grows as $\bar{n}R^2$.

while the production rate grows by a factor of forty, from 0.08 to 20.5 t/yr. The upscaled SGS power of ~ 120 kW is substantial (240 kg of array at 500 W/kg, the dominant single mass), but

presents no new physics. Every subsystem scales along with the laws already established, and the total drag of 1.0 N remains trivially small for the structure.

7. Mission Study

The mission selected to benchmark the SGS is one that would demonstrate the utility of the SGS in enabling high mobility operations in LEO, and in particular Active Debris Removal or ADR. Several active debris removal concepts have been proposed for debris mitigation that included 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. 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, 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) [5,6]. It is unique in avoiding many of the problems previously described, but it is especially advantageous when one considers the fact that with a RON spacecraft refueled by an SGS, one has an essentially unlimited fuel resource for this task. A brief overview of this concept will now be given with regard to its appropriateness to ADR. Given the successful development of a SGS, this approach would become the leading ADR approach particularly for massive, high altitude debris targets.

This approach to ADR will be referred to as the Plasma Beam Shepard (PBS) which more accurately describes what is essentially an EP thruster plasma plume impinging upon the target debris object. The PBS employs a highly collimated, high-velocity plasma beam, produced on board the RON spacecraft orbiting in close proximity to a target debris object, and 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 plasma beam. A considerable advantage of this approach stems from the mature development of several plasma-based thrusters for propulsion. In Ion Engines, like those that powered the Deep Space 1 and Dawn spacecraft, and which have been successfully flown in space, the plasma ions have been accelerated to 20–50 km/s. As was indicated earlier and elsewhere [3], EP 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 laser induced material ablation. The only serious deficiency of a PBS system for ADR was propellant limitations due to the launch expense as pointed out in Section 1. The limitation due to lack of

propellant resources is of course eliminated with the combination of the PBS (RON) with the SGS.

Figure 14 illustrates how the PBS would work to deorbit a target spacecraft. The RON spacecraft is equipped with a primary propulsion system that emits a beam of accelerated plasma towards a target object in order to apply a force 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 Hall Effect Thruster HET or a Pulsed Inductive Thruster (PIT). It is important to have the primary thruster to have a low plume divergence in order to allow for a greater standoff from the debris. A thruster such as the RF based Ion Thruster (RIT) is favored as it has a large Isp and fairly low divergence and can thus be 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, the force will be equal and opposite to the force that the primary propulsion system exerts on the RON spacecraft. There is likely to be primary ions that are scattered back as well as 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 reflection and ejection velocities along the line of force are generally quite small compared with the plasma beam, the effect on the total directed force is likely to be small. It should be noted that to maintain formation flying, the PBS has to be compensated by a force compensation thruster, which is located on the opposite side of the RON spacecraft as shown in Fig. 14. This thruster could just as well be the same as the primary thruster for the

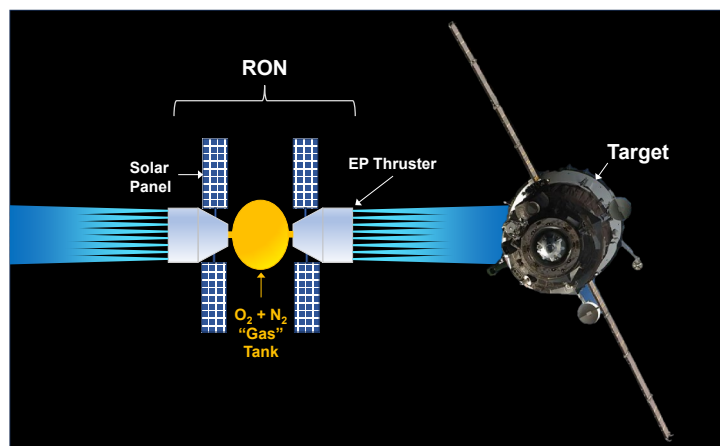


Figure 14. Illustration of the RON spacecraft employing the Plasma Beam Shepherd (PBS) technique for deorbiting space debris. A second, oppositely directed EP thruster is used to maintain RON in close proximity to, and aligned with the target debris. Note: spacecraft and other components are not to scale.

RON and also act as the low-divergence primary thruster trained on the target satellite and thereby provide for some redundancy. At this point it would appear that the primary thruster would likely be the RIT as it has the lowest plume divergence and highest potential efficiency among the most advanced EP systems. It is believed with a RIT that a standoff of greater than 7 m could be maintained [6]. There is certainly no reason at this point to be too concerned with the

specifics of the PBS as the availability of copious amounts of propellant will certainly shape the type of propulsion that would be the most optimal for reactive force compensation. For this role the RON spacecraft propulsion alone it is likely to be a thruster with a 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 [6], 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.

7.1 Scenario for large debris removal with RON and SGS

As an example of an ambitious ADR mission for the RON spacecraft, would be the deorbiting of a 1000 kg (1 tonne) Russian Cosmos class satellite at an altitude of 950 km which will now be detailed. Each retrieval and disposal mission can be divided into four stages. These stages are depicted in Fig. 15, as well as and summarized in Table 5. Each stage will now be discussed in more detail.

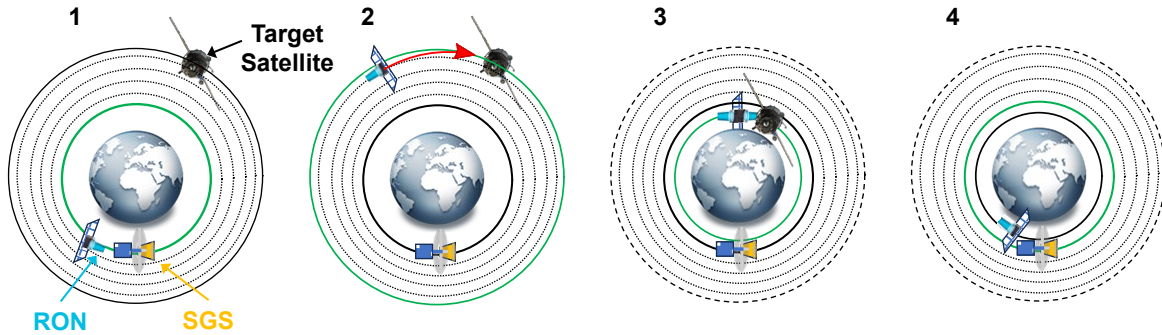


Figure 15. The four mission stages of the Refueling Orbital Navigator (RON) spacecraft for sample ADR of a 1 t satellite at 950 km . Green orbit is the RON orbit at end of each mission stage. Bold line indicates SGS stationary circular orbit at 300 km. The characteristics of each stage can be found in the text.

Stage 1: A Refueling Orbital Navigator will be fueled by and launched from the SGS in order to rendezvous with a large satellite that will be referred to as the target. As there are several viable options for the RON propulsion system, a listing of the desired performance criteria for the thrusters will be presumed to be consistent with what has been achieved by both ion and Hall thrusters, albeit not necessarily on the mix of O_2 and N_2 that would be used on the mission. Even though the fuel supply is essentially unlimited, consuming more fuel limits the number of missions that can be accomplished in a given time as the rate of production of propellant is on the order of 0.5 t/yr (2.74 kg/day) for a single SGS operating at a mean altitude of 300 km. It will be assumed that an EP thruster exhibits a modest I_{sp} of 2.5 ksec with either the O_2 or N_2 propellant, or both. The RON would likely be equipped with two 10 kW thrusters placed in

Stage	Operation	Altitude (km)	Propellant (kg)	Time hrs (days)	Delta-V (m/s)
1	Fuel and launch RON from SGS	300	-	4 (0.125)-	NA
2	RON raises orbit to align with target and rendezvous	300-950	1.89	32.9 (1.37)	350
3	Using thrusters as beam shepherd, move target into low orbit	250	16.85 (2)	293 (12.2)	210
4	Target orbit left to decay (<15 days) RON raises orbit and realigns w SGS	250-300	0.12	2.1 (0.088)	140+
5 (1)	Dock and Refuel	300	-	4 (0.125)	NA
	TOTAL		35.7	336 (14)	700+

Table 4. Stage characteristics of the ADR using a RON fueled by an SGS.

apposition and oppositely directed as depicted in Fig. 15. It will also be assumed that they operate at 50% efficiency yielding a jet power of 5 kW per thruster. The rate of propellant usage for each thruster is thus:

$$\dot{m}_p = \frac{2P_{jet}}{v_{exh}^2} \cong \frac{10^4}{(2.5 \times 10^4)^2} = 1.6 \times 10^{-5} \frac{kg}{s}, \quad (16)$$

where $\dot{m}_p$ is the propellant mass and $v_{exh} \cong 10 \cdot I_{sp}$. A zero-order mass budget for the RON is as follows: The thruster mass is estimated at ~ 20 kg each which is consistent with the mass of current masses of both 10-kW ion and hall thrusters. The PPU for both is likely to add another 20 kg. An Attitude Determination and Control System (ADCS) will need to be more extensive due to the maneuvers required on both ends of the mission with rendezvous and steering the target, as well as docking and refueling at the SGS. At a minimum two gyroscopes and at least 8 electrothermal thrusters will be required with a mass of ~ 10 kg. Communication and instrumentation equipment would add another 10 kg. The solar panels with some battery backup will add 15 kg and structural hardware and tankage another 5 kg. This brings the total to around 100 kg for the dry mass. As can be seen from Table 5, the ΔV s for operations involving only the RON requires very little in the way of propellant as well as time so that an increase in the RON mass would make for only a small increase to the total propellant and time requirements. This lack of sensitivity is accommodating in that there will likely be more propellant than estimated in Table 5 for the mission as significant changes in inclination between the SGS and the target will require additional propellant for the RON target rendezvous as well as the return to the SGS. Fortunately, there is no need for an inclination maneuver to deorbit the target.

Stage 2: During this stage the RON will assess the dynamics of the debris and the best way to interact with it. The Plasma Beam Sheppard (PBS) for remote, non-contact interaction with the target is likely the best and certainly most appropriate given that propellant availability is not the driving issue. It is assumed that there will be extensive demands on the ADCS, and the propellant

mass budget for such maneuvers needs to be more critically evaluated. In order to move the object in a stable and accurate manner the thrust vector must be applied through the center of mass of the target. 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. This will naturally occur once plasma beam breaking ensues with a subsequent adjustment in aim to the target center of mass

Stage 3: RON spacecraft will now ferry the target from 950 km altitude to a termination orbit of 250 km. At this lower altitude, the atmospheric drag will be sufficient to deorbit the target debris in less than 15 days. It is clear from Table 5 that this maneuver will require by far the largest expenditure of propellant (~ 34 kg). It is certainly possible that there are other orbital strategies that might reduce the propellant usage such as inducing the target into an elliptical orbit with a very low perigee (e.g. 100-150 km). Even with the simpler walk-in approach assumed here, the mission does not require an unreasonable propellant usage or transport time.

Stage 4: Once the target has reached the terminal orbit the RON spacecraft will leave the target and raise its orbit, adjusting its inclination and right ascension for a rendezvous with the SGS. After rendezvous and docking with the SGS, the RON spacecraft can then be refueled for its next mission. It is worth noting that during the 14-day ADR mission the SGS would have captured 38.4 kg of new propellant. From Table 1 this would replenish what was expended with a small surplus. A less massive or lower altitude target would yield an even larger surplus and allow for additional debris to be targeted before refueling.

Success in demonstrating the SGS in LEO could lead to more ambitious missions such as a Human Mission to Mars (HMM). The SGS could be used to produce oxidizer for the Reaction

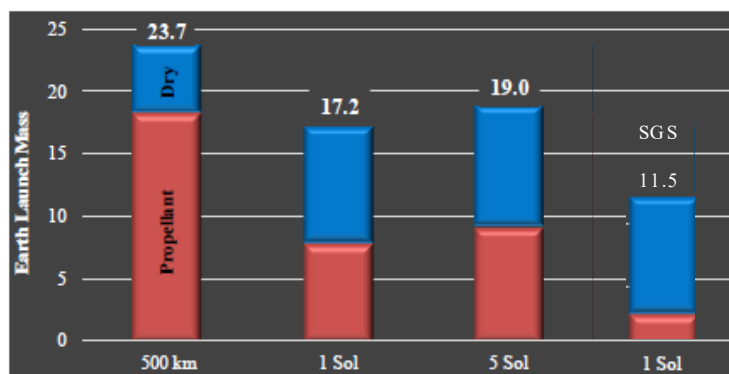


Figure 16. MAV Masses at Earth Launch. For the 1 Sol and 5 Sol spacecraft orbit, the MAVs are based on Mars ISRU for LOX propellant production. The SGS reflects LOX production for both the EDL and MAV.

Control System thrusters for aerocapture on the Entry, Descent and Landing (EDL) vehicle for separation from the transportation stage using ~ 5 t LOX [10]. The Mars Ascent Vehicle (MAV) requiring 30 t of chemical propellant can also be considered [11]. As illustrated in Table 4, the analysis of the SGS depicted in Fig. 1 can harvest 0.5 to 3.5 t of O₂ per year depending on scale and altitude. Given a few years, multiple SGS's or larger SGS's, could then meet the goal for

both phases of the HMM. The current mission plan is to attempt to produce the MAV LOX on the Marian surface with a precursor mission launched several years earlier (see Fig. 16). The SGS has been added to this plot from Ref. 11, where the small remaining propellant mass reflects the hydrogen that is needed for the LOX+LH2 propellant. The advantage of the SGS is clear.

8. Summary and Conclusions

The successful development of a Space Gas Station would open up a new era of space development where the SGS could service a large variety high ΔV missions in LEO starting with ADR on the largest scale. Matched with the appropriately equipped Refueling Orbiting Navigator (RON) spacecraft, smaller targets could be eliminated as well with the smallest employing other techniques such as laser-ablation. By employing multiple SGS's paired with various RON spacecraft, hundreds of missions could be performed without the need to refuel from earth, and in this manner rapidly eliminate up most of the space debris of concern. The SGS+RON could be used to facilitate refueling, repair, repositioning or capture of a large range of space assets as well.

The SGS alone could provide the fuel oxidizer (LOX) for many if not most space transport missions which would have a significant impact on the cost, schedule, and capability for any deep space mission that relies on LOX as part of the propellant fuel mix, which is virtually all such missions. It would include unmanned exploration missions as well as sample and return, With the primary use of LOX in both landing and ascent, it would provide for much larger payloads. Recognition of the advantages of ISRU fuel generation is demonstrated by inclusion in the plans for several future missions. Given the remote and hostile environments of the outer planets and their moons, along with the completely autonomous processing required to perform the mining, chemistry, storage, etc. this particular approach becomes daunting. The SGS would be far simpler as well as far easier to implement, maintain and access in VLEO, not to mention it can be used for launch and return mission transits.

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