Low-Cost Propellant Launch to LEO from a Tethered Balloon – Economic and Thermal Analysis

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Abstract—As we have previously reported [1-3], it may be possible to launch payloads into low-Earth orbit (LEO) at a per-kilogram cost that is one to two orders of magnitude lower than current launch systems, using only a relatively small capital investment (comparable to a single large present-day launch). [1] [2] An attractive payload would be large quantities of high-performance chemical rocket propellant (e.g. LO2/LH2) that would greatly facilitate, if not enable, extensive exploration of the moon, Mars, and beyond. The concept is to use small, mass-produced, two-stage, LO2/LH2, pressure-fed rockets (e.g. without turbopumps, which increase performance but are costly). These small rockets could reach orbit with modest atmospheric drag losses because they are launched from very high altitude (e.g. 22 km). They reach this altitude by being winched up a tether to a balloon that is permanently stationed there. The drag losses on a rocket are strongly related to the ratio of the rocket launch mass to the mass of the atmospheric column that is displaced as the vehicle ascends from launch to orbit. By reducing the mass of this atmospheric column to a few percent of what it would be if launched from sea level, the mass of the rocket could be proportionately reduced while maintaining drag loss at an acceptably small level.

The system concept is that one or more small rockets would be launched to rendezvous on every orbit of a propellant depot in LEO. There is only one orbital plane where a depot would pass over the launch site on every orbit – the equator. Fortunately, the U.S. has two small islands virtually on the equator in the mid-Pacific (Baker and Jarvis Islands). Launching one on every orbit, approximately 5,500 rockets would be launched every year, which is a manufacturing rate that allows significantly reduced manufacturing costs, especially when combined with multiyear production contracts, giving a projected propellant cost in LEO of \$400/kg or less. The configuration of the proposed propellant depot and the manner in which the propellant would be utilized has already been reported [1]. The launch processing facility (a small, modified container ship) and cable-car that moves the rocket on the tether have also been reported [2]. This paper provides new analysis of the economics of low-cost propellant launch coupled with dry hardware re-use, and of the thermal control of the liquid hydrogen once on-orbit. One conclusion is that this approach enables an overall reduction in the cost-per-mission by as much as a factor of five as compared to current approaches for human exploration of the moon, Mars, and near-Earth asteroids.

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1. INTRODUCTION

Many mission studies of alternative approaches for extending human reach into the solar system have shown that the lofting of propellant mass into Earth orbit is a dominant cost of any such effort [4]. The Saturn V moon rocket was typical of exploration missions in that 85% of the mass put into Earth orbit was propellant, as needed to leave Earth orbit, enter lunar orbit, land on the moon, return to lunar orbit, and depart back to Earth. Reducing the cost of lofting that propellant is key to the affordability of any sustainable exploration architecture, at least until the infrastructure is so advanced that extraterrestrial resources could provide the needed propellant at a lower effective cost. With current launch costs of ~\$10,000 for every kilogram that is delivered to LEO, and with all the exploration architectures that are capable of captivating the imagination and support of the public and the Congress requiring one-to-several thousand tons of propellant mass to be lofted into LEO each year, the cost of propellant launch could easily consume the majority of the NASA budget (currently ~17 billion dollars (G\$) per year). The pace of space exploration is almost completely limited by the rate at which propellant mass can be launched into LEO, since advanced exploration architectures envision reusable vehicles that could make multiple round-trips beyond LEO so long as they have sufficient propellant. Thus any affordable and sustainable exploration strategy involves

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[2] IEEEAC paper #1074, Version 2, Updated Dec 14, 2009

reducing the launch cost for propellant by a large factor.

Reducing the cost of launch into orbit has been extensively studied [5]. Unfortunately, credible proposals to achieve significant reductions in operational costs have very large up-front investment costs. Such systems include various combinations of new large reusable chemical rocket stages (with or without air-breathing 1st stages and/or runway launch and/or recovery), high speed "guns" that fire payloads through the atmosphere, and orbital towers or tethers that could be used as elevators to space. Advocates typically maintain that reduction of launch cost by a factor of 3 would require a capital investment at least equal to one year of the total NASA budget [5]. This is almost certainly unaffordable and unrealistic. This paper expands on previous analysis [1-3] to study a means to reduce propellant launch costs by a factor of about 30, using a capital investment that is small (<<1G$) and that is demonstrable within a few years. Furthermore, in the "propellant-rich" architecture envisioned here, all other space hardware could be made lower-performance (e.g. somewhat more massive but much lower cost per kilogram), and especially **more reusable** than it otherwise would, since there would be plenty of propellant available to move it back-and-forth beyond LEO. Thus this approach may potentially reduce the overall cost of exploration by an order of magnitude or more when reuse of dry hardware allows 95-99% of the total mass put into LEO to be low-cost propellant.

As this paper describes concepts that are not yet approved as missions by NASA, all topics covered here are "for planning and discussion purposes only."

2. Summary of Previously-Reported Results

Think of the tethered balloon as a "flagpole" (Figure 1). The balloon supports a pulley that has the tether looped over it, and winches at the surface could lift a rocket launcher up to the stationary balloon, much as a flag is hoisted up a flagpole. A reloaded launcher goes up to the balloon from one winch as an empty launcher is lowered from the balloon to the other winch. By physically separating the two winches on the surface, the tether lines would not get tangled and the rocket could be dropped freely between the two tether lines for a few seconds before firing so that neither the balloon nor the tether are put at risk of being incinerated by the rocket. The flagpole architecture has the disadvantage that the tension in the tether acts on both sides of the pulley at the balloon, thereby doubling the required lift of the balloon. A superior alternative (described in detail in [2]) is to have a self-powered cable-car that winches itself up a single "large" tether (about 3 cm diameter), and then have a "small" tether to provide the required geometry at launch, and which also carries aircraft warnings such as radar reflectors, strobe lights, etc.

Launching from the equator, the fuel depot passes over the launch site on every orbit – a key to the volume manufacturing approach. Any non-equatorial launch site would pass through the orbital plane of a fuel depot only twice each day, but even then it would be rare for the fuel depot to happen to be passing over the launch site at those times as needed for direct rendezvous. Although ship-launch from the equator is possible, it is fortunate that the U.S. has two territories south and southwest of Hawaii that are within a few kilometers of the equator: Baker and Jarvis Islands. This allows the balloon to be tethered to buoys anchored to the shallow ocean bottom, and allows the resupply ship to conduct operations in the prevailing wind and ocean current "lee wake" of the island. A special benefit of launching from the equator is that it lies in the inter-tropical convergence zone, where powerful winds are almost non-existent. Indeed, because there are no Coriolis forces at the equator, there is no tendency for unstable air to organize into cyclones. Hurricanes do not occur on or cross the equator [6] and there are no jet streams [7], both of which make tethering a balloon at high altitude on the equator much easier than at other places. Another advantage of launching from an extremely remote location in the Pacific Ocean, far from normal shipping lanes, is that the most economical system would use rockets that are less than 100% reliable. The lowest-cost overall system may well have ~10% launch failure rate. "3-sigma" or greater reliability is not needed, greatly reducing the overall cost.

In [2] we showed, based on "textbook analysis," that a small LO_2/LH_2 pressure-fed rocket could be manufactured that is capable of delivering ~200 kg of LO_2/LH_2 into LEO

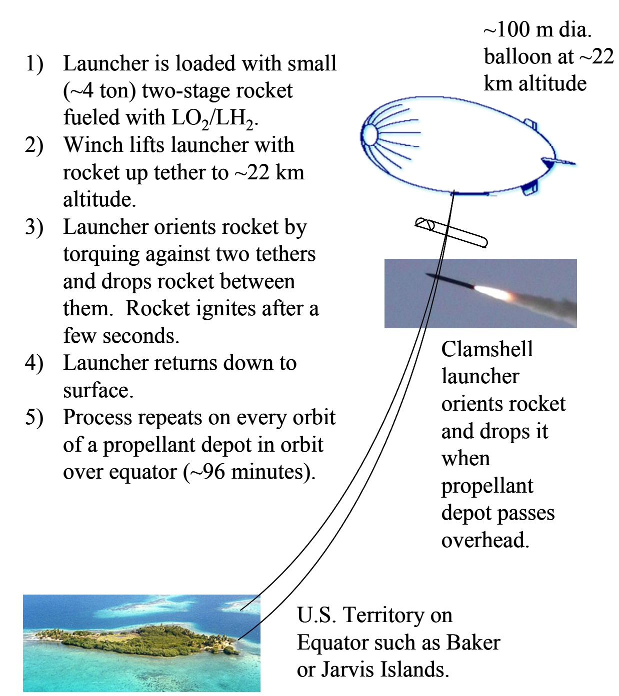

Figure 1: Concept for Low-Cost Propellant Launch to LEO from a Tethered Balloon

and that has reasonable performance and expected cost. In this case, the textbook is *Modern Engineering for Design of Liquid-Propellant Rocket Engines*, by Dieter K. Huzel and David H. Huang; 1992 [8]. The expected manufacturing cost was minimized based on the assumed relative cost per kilogram of the pressurized propellant tanks and the rocket engines. We showed that the cost function has a broad optimum at an operating thrust chamber pressure of about 2 MPa (300 PSI), over a wide range of assumed thrust chamber material costs and chamber pressures. We concluded that an approximately cost-optimal second-stage engine could be built that has a specific impulse of 441 s, 41kN of thrust, a nozzle throat diameter of 11.6 cm, a nozzle exit diameter of 73.1 cm, and consumes about 9.5 kg/s of propellant. We speculated that the lowest-cost manufacturing approach would be to use the well-known "channel-wall" thrust chamber design [8, p96], where the entire thrust chamber, including throat and nozzle, is cast as a monolithic thick-walled copper-alloy structure, with many channels then machined into the outside of the casting to form passages for the cryogenic hydrogen that is both coolant and fuel. A structural backing supports the channel-wall liner, but that backing never sees high temperatures and so could be made out of inexpensive materials. Propellant pressure is maintained by routing the coolant from the channels into a heat exchanger in the hydrogen tank, where a small fraction of its thermal energy is used to boil the hydrogen. The hydrogen coolant is then injected into the engine as fuel. The hydrogen gas boiled by the heat exchanger pressurizes both the hydrogen tank and the oxygen tank, with a relief valve to prevent over-pressurization. The tank pressure is assumed to be 20% higher than the thrust chamber pressure to ensure stable injection, following the guidelines given in [8, p115].

The production cost in volume manufacturing is estimated based on the experience of the automotive industry. The "learning curve" data that we used in [2] is summarized in Figure 2 [9]. A curve-fit to this data follows the traditional learning curve with free parameter 0.75 (e.g. the cost of unit 2^N is $(0.75)^N$ times the cost of the first unit). Perhaps the most important part of the volume-manufacturing approach is the "LOX Post" co-axial fuel injector, shown in Figure 3. The optimal injector design dimensions, derived in [2], are only a function of chamber pressure and so are common to both the first and second stages. As a result, this particular component would be needed in quantities of literally millions per year. As can be seen in Figure 2, even for complex products such as automobiles, the manufacturing cost per kilogram is only asymptotically greater than the bulk materials cost at these high production rates. This fuel injector is probably the most complex element of the launch system, so it is fortunate that it would be manufactured in such large volumes.

We assumed in [1,2] that the rocket is controlled by two small head-end "vernier" rocket engines that are attached via gimbals to the payload assembly and that draw

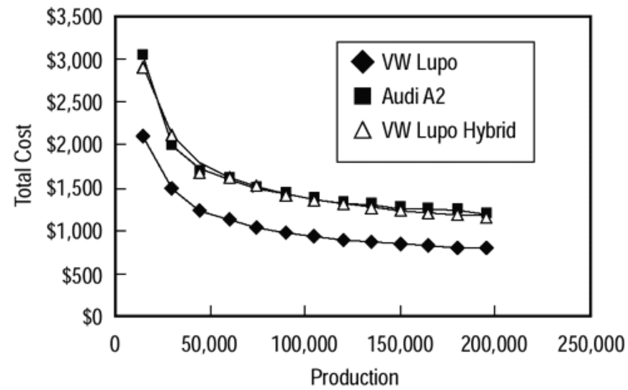

Figure 2: Manufacturing cost of automobiles in mass production (originally published in 2001 as Figure 7 in [9], reprinted with permission of The Minerals, Metals & Materials Society).

propellant from the payload, which is slightly oversized to account for this loss. The two main stages themselves have no thrust-vector or throttle modulation control so as to keep their cost as low as possible. The first stage ignites shortly after the rocket is dropped at the base of the balloon, with the second stage firing just after the first stage is spent. The rocket is oriented so as to emerge quickly from the atmosphere and coast to the desired LEO orbital altitude, where the vernier thrusters complete the orbit injection. The discarded first stage drops into the Pacific Ocean within a few hundred km of the launch site; the spent second stage almost reaches orbital velocity and burns up over the ocean after a partial orbit. The vernier control rockets are used to complete the orbit injection of the payload and to accomplish rendezvous with the propellant depot as described in [1]. A relatively small amount of extra propellant is carried by the payload module at launch to provide for these maneuvers. Only the nominal control propellant is provided – off-nominal consumption would reduce the payload delivered to orbit. The cable-car/rocket-launcher insulates the rocket to reduce boiloff (beyond that needed to bring the tanks up to working pressure) while the assembly is being winched up the tether; the tanks may also

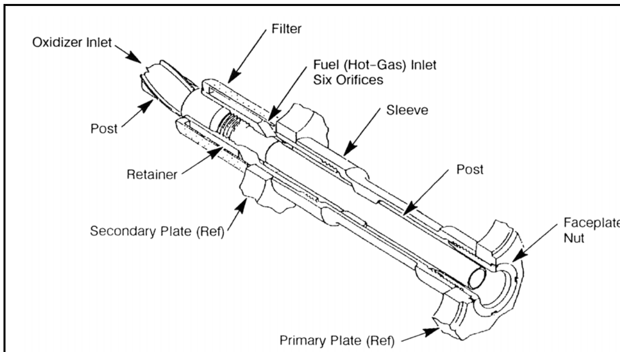

Figure 3: LO_2/LH_2 fuel injector, which would be manufactured in quantities of ~1.4M/y under this concept (originally published as Figure 4-61 in [8]; reprinted with permission of the American Institute of Aeronautics and Astronautics, Inc.).

be topped off by the launcher just before it is dropped.

As described in [1], we envision the only permanent part of the propellant depot to be a robot that images arriving propellant modules (using visible light and thermal infrared radiation) so as to give radio commands for vernier thrusting to accomplish a precise rendezvous. On final approach, the robot grapples the incoming propellant module, and secures it by clips or similar means to extend a row of identical propellant modules, forming an "arm" of the "star" propellant depot (Figure 4). The star depot, once complete, is spun using synchronized thrusting of the many gimbaled vernier engines in the propellant modules, and accelerated by similar thrusting to transport itself to an inclined orbit, to the Earth-moon L1 libration point, to low lunar orbit, etc. With an assumed dry mass of 35 kg for each propellant module, and an assumed specific impulse of 420 s for each of the vernier thrusters, then starting with 200 kg of propellant in equatorial LEO, about 73 kg would remain in each module after transit to L1, or 58 kg would remain in each module after performing a propulsive plane-change maneuver to reach the 28.5^o LEO inclination required to rendezvous with hardware launched from the Kennedy Space Center in Florida. From the L1 position (where the thermal equilibrium temperature allows indefinite storage of LH_2), it could be used to fuel or refuel vehicles making round-trips to the lunar surface or to Mars or beyond. To minimize propellant boil-off, star depots are only delivered to the 28.5^o Earth orbit or to low lunar orbit as needed to promptly rendezvous with and refuel vehicles that are already there. It is assumed that one propellant module arrives in LEO every 96 minutes (the 90 minute orbital period, plus 6 minutes to catch up with the launch site moving with the rotating Earth). It is also assumed that one star depot departs equatorial LEO every 2 weeks (at the launch windows to L1), implying that each arm of the star has 36 modules. This means that a star depot could deliver about 16 metric tons of propellant to L1 or 12.5 tons to the 28.5^o LEO orbit inclination every 2 weeks. All star depots that stay in LEO are eventually de-orbited over the ocean. Propellant is reserved so that all star depots that leave the Earth's gravity well (and the refueled Earth-departure stages) could eventually be mothballed at one of the stable Earth-moon Lagrange points (L4 or L5) to reduce orbital debris and for use as a long-term resource. This would also avoid polluting the moon with the volatiles that escape from crashed vehicles, and the risk those impacts and their ejecta pose to surface astronauts and assets.

Each star is formed around a "hub" module which is of a slightly different configuration than all the rest of the propellant modules. It has six sets of clips around its circumference so that the six radial arms could be connected to it. It has the same vernier thrusters and GPS-augmented inertial navigation and radio-commandable control system as the other modules so that it could be launched in the same way. However, it has smaller propellant tanks so that the remaining mass and volume could be devoted to fuel-transfer apparatus, including a concentric fluid slip-ring that allows the rotating star to refuel a non-rotating vehicle. Included in this hardware is a relatively long coaxial hose that extends from the fluid-slip-ring along the spin axis of the star and permits safe refueling of expensive (and perhaps crewed) vehicles from the rotating star. The vehicle to be refueled could carry additional hose extensions if desired. The star hub also carries additional computing, communications and navigation equipment (e.g. a star tracker) so that it could function as the command computer for the depot, issuing thrust commands to individual propellant modules by radio. Further analysis may indicate that capillary forces are sufficient to collect the liquids in the relatively large surface-to-volume-ratio tanks of the small propellant modules, eliminating the need to spin the star depot, simplifying the plumbing and operations.

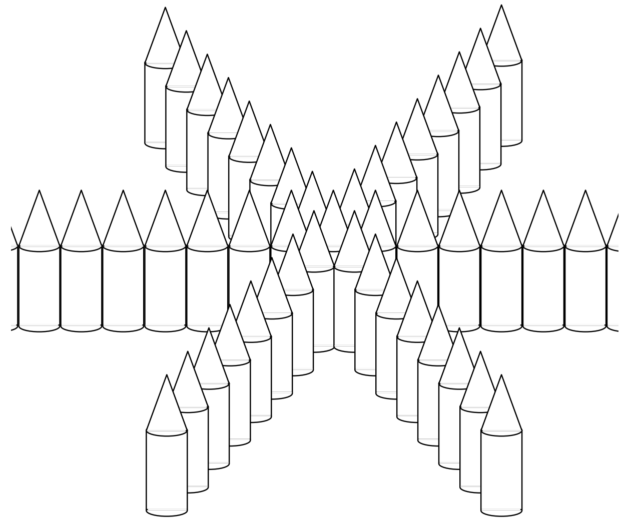
Figure 4: The Star Propellant Depot – a spin-stabilized, self-propelled array of small propellant modules.

The propellant depot robot connects flexible hoses between adjacent modules during assembly of the star so that the propellant could be transferred along the length of each arm of the star, and ultimately through the hub to the target vehicle. A total of four hoses need to be connected when each module arrives: one each for vapor and liquid for both hydrogen and oxygen. It is assumed that the dexterous manipulation required for this comes from a humanoid robot such as "Robonaut" [10] using an appropriate control methodology that is insensitive to the speed-of-light latencies implicit in control from Earth via satellite relay [11]. The drains for liquid in each tank are arranged so that either axial or rotational acceleration delivers the liquid to the drain, and similarly the vapor ports are arranged to ensure that the vapor bubble is at the port under either axial or rotational acceleration. Similar to the LO_2/LH_2 upper stage engine on the Saturn V launch vehicle, a small acceleration is required to keep the fuel settled at the drain ports to prevent ingestion of vapor when the engines are restarted. Prior to rotation of the star, and again similar to the Saturn V upper stage, this acceleration is provided by directing the boil-off vents to the rear [12]. Each propellant module has a small solar array, battery and electronics module at the rear (between the vernier thrusters) that provides long-term power and command interface – forward of that is a thermal shield with foil wings that pop out at the time of payload separation to ensure that, when the back end of the star depot is pointed toward the sun, the heat load into the cryogenic payload is minimized while the avionics stay within a reasonable temperature range. The payload tanks are extremely well insulated, and appropriate surface coatings on the payload are optimized for indefinite passive propellant storage at L1.

The economics of manufacturing the rocket suggest that the cost would be low. The dry mass of stages 1 and 2 combined is about the same as the smallest car, and most of that mass is made of relatively thick aluminum panels that may be friction-stir welded together. Aerospace aluminum alloys typically command a 20% price premium over basic aluminum, but bulk aluminum only costs about $3.50/kg [9]. Nothing in stages 1 or 2 is remotely as complex as an automobile engine, transmission or dashboard. In our case, there are only a relatively few components in stages 1 and 2. The only moving parts in stages 1 and 2 are motorized ball valves that need to actuate only once. Based on manufacturing data such as that plotted in Figure 2, it seems quite possible that the manufacturing cost of stages 1 and 2 (combined) could be less than $10,000 - perhaps very much less. The current cost of liquid hydrogen and liquid oxygen, purchased in bulk, is under $2 /kg. So the total cost of the wet vehicle, less payload, is expected to be under $13,000.

The payload assembly, however, would have a relatively high specific cost. As discussed in [2], it is probably best to make this a highly-reliable piece of spacecraft-quality hardware. It represents the "brain" of the launch vehicle, having the inertial measurement unit and the avionics that trigger ignition and staging events, and control the vernier thrusters. Weight and reliability improvements on the payload assembly would pay handsome dividends in the overall system because the payloads have to change orbits while preserving as much propellant as possible and are expected to operate for many weeks or months, performing on command. So it seems reasonable that the assumed 35 kg of dry mass in the payload assembly might have a specific cost of $1,000/kg, even manufactured in annual production volumes of 5,500. So we estimate that the completed rocket costs 13k$ for stages 1 and 2 (wet) and 35k$ for the payload assembly, for a total of about 50k$ per launch.

Launch operations would be managed by a small cargo-container-type ship that could be leased for 100k$/day (including crew, estimated based on quotations for a large oceanographic research vessel for a preliminary balloon deployment experiment at Baker Island [13]). Two such ships are needed, one on-station and one going back and forth to port, loading in port, and providing shore leave for the crew. Launching every 96 minutes, the total cost of the rockets is 750k$/day. So the total cost of ships and rockets is about 350M$/y. If lithium-ion batteries that have a life of 200 cycles are used to power the cable-car that runs up and down the tether, then even at current laptop battery prices that is only 20M$/y. Even if the balloon and tether needed to be replaced every 100 days, that only adds a few M$ to the total cost, based on the cost of the NASA Ultra-Long Duration Balloon and the catalog price of the advanced rope that would be used (e.g. PBO). Replacing every 10 years a 200M$ on-orbit robot that manages the arrival and rendezvous of the payloads (as described in [1]) would add only 20M$/y to the system cost. Thus we expect the total system cost to be about 400M$/y, while launching 1100 tons of propellant into orbit, for a specific launch cost of 360$/kg. This is a reduction by a factor of almost 30 compared to current launch services. Note that the total annual system cost is about the same as only a few present-day expendable launches, or less than the cost of a single projected heavy-lift launch vehicle. As discussed in [1], the system could be expanded by launching more than one rocket on every orbit of the propellant depot, increasing the amount of propellant delivered in integer multiples while further driving down the per-kilogram costs via mass production [9]. Perhaps most importantly, these savings could be achieved with very low capital investment. Certainly a vendor who configures a factory to manufacture the rockets or the payload assemblies would need to be assured of a multi-year contract with an appropriate early-termination clause, but given that, there is no particular reason that the government should make a large up-front investment, or to contract on anything but a fixed-price basis. Presumably the government would have to pay to develop proof-of-principle rocket/payload prototypes. One obvious procurement strategy is to get multiple prototypes built by competing prospective full-production bidders in a "shoot-out".

Figure 5 shows the launch configuration. The overall length of the vehicle at launch is about 12 m (fitting in a standard 40' shipping container). As described in [3] we derived detailed design parameters for the required rocket engines for the 2-stage launch vehicle. We adopted that 2nd stage engine design as a baseline, and considered the case where the first stage consists of a cluster of four engines identical to the single 2nd stage engine. Each of the four is canted so that their thrust vector passes through the nominal center-of-mass of the launch stack, so that variations in performance between the engines don't have an undue effect on the control authority requirements. The manufacturing volume of this engine would be almost 30,000 per year, making relevant the use of the manufacturing cost data shown in Figure 2. The only other somewhat complex components (beyond the fuel injector described previously) needed for each engine are precision, highly-polished cryogenic ball valves (although low-leakage is not particularly crucial on the launch stages

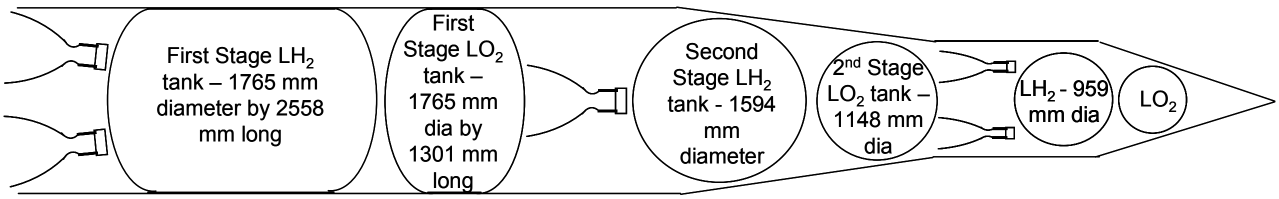

Figure 5: Launch configuration: overall length is ~12 m, 1.8 m diameter, 4439 kg at launch; both first stage tanks have wall thickness 6.9 mm; second stage hydrogen tank wall thickness is 2.5 mm and oxygen tank thickness 1.3 mm; payload assembly hydrogen tank wall thickness is 0.9 mm and oxygen tank thickness 0.5 mm. There will be a single 41 kN engine for the second stage with cluster of 4 identical engines for the first stage.

because the propellant is in the stage for a very short period). Again the high production volume should make the cost similar to that of modern automotive components having similar precision and reliability.

It is presumed that there are large cut-outs in the skin adjacent to the two head-end vernier engines so that they could gimbal out at possibly large radial and tangential angles to provide transverse thrust and roll control during ascent. Fortunately, the low atmospheric density and dynamic pressures associated with launch from high altitude make this strategy viable.

Spherical tanks are used in the second stage and the payload assembly to reduce mass. The tank mass per unit volume for a spherical tank is only 75% that of a long cylindrical tank, at fixed tank pressure and wall stress. Cylindrical tanks are used in the first stage at the same diameter as the cluster of four engine nozzles to minimize the frontal area and hence aerodynamic drag. The tank mass in the payload is only about 9.5 kg of the allocated 35 kg for the dry payload assembly. The low mass of the spherical tanks means that the thrust chamber pressure of the vernier thrusters could be the as high as the other engines (2 MPa), so all fuel injectors are identical and the O_2/H_2 injection velocity ratio is not a concern, as discussed in [8, p116].

The results of a simple spreadsheet-based dynamic simulation of the launch are shown in Figure 6. Aerodynamic lift and drag are modeled (as was done in [1]) using published wind-tunnel data for a cylindrical body with a sharp conical nose at Mach 3.12 and Reynold's numbers of $8x10^6$ and $14x10^6$ [14]. For the current model, a $\sin(2\alpha)$ function of angle of attack α was fit to the lift-to-drag ratio of the published measurements, giving multiplicative coefficient 17.96 (e.g. the lift-to-drag ratio is $17.96*\sin(2\alpha)$). The drag coefficient is similarly fit to the data as $1-0.8*\cos(2\alpha)$. Note that the vehicle speed is very low except when the angle of attack is small (e.g. <0.1 radians), so this simple model (based only on frontal area) should be adequate. The Mach number of the published data reflects a speed of about 1 km/s, which roughly corresponds to the peaks of lift and drag in Figure 6. The Reynolds numbers for the published data are also in rough agreement with the flight conditions near the peaks of lift and drag. Atmospheric density is modeled as an exponential with scale height 7.3 km, based on a sea-level density of 1.293 kg/m^3. This gives reasonable agreement with empirical data over the regime of interest represented in Figure 6.

The initial conditions of the launch simulation results shown in Figure 6 are that the rocket is dropped from an altitude of 22 km 6 seconds before ignition, at which time it has an elevation of 40.2 degrees above the horizontal. (The launch altitude is optimized based on wind speed data [15] as discussed in [2].) A GPS-augmented inertial navigation system is assumed to command the vernier thrusters to control the angle-of-attack to follow the lower curve in Fig. 6, aligning the vehicle axis with a ballistic trajectory after an exponential decay. (This control law is representative but not optimized.) The launch mass of the rocket is 4439 kg. The first stage consumes 2,329 kg of propellant in 61 seconds, accelerating the vehicle to a horizontal velocity of 2,587 m/s and a vertical velocity of 1,157 m/s at a point 68 km downrange and an altitude of 53 km. The inert mass of

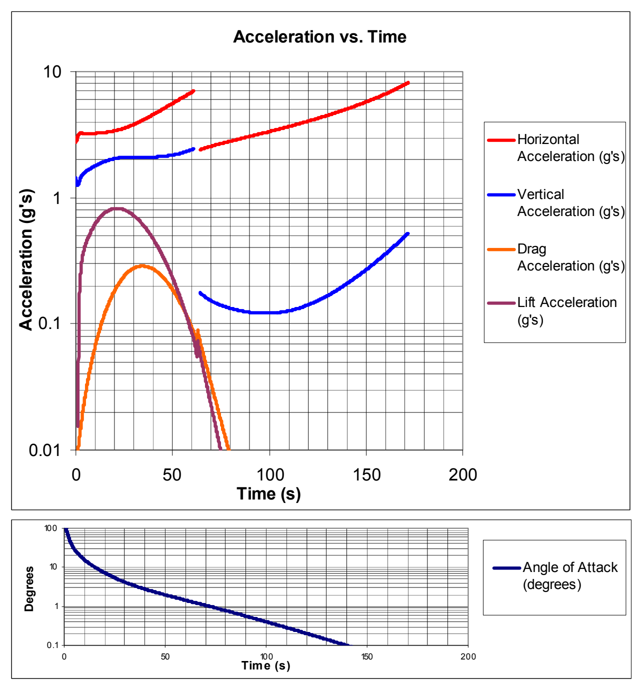

Figure 6: Acceleration and angle of attack versus time during launch. Lift and drag accelerations are expressed in air-relative-velocity-centered coordinates (not gravity-relative) but their resolved effects are included in the horizontal and vertical acceleration plots.

the first stage (582 kg) is jettisoned, and the second stage ignites a few seconds later. The second stage consumes 1,024 kg of propellant in 108 seconds, accelerating the vehicle to a horizontal velocity of 7,196 m/s and a vertical velocity of 1,339 m/s at a point 555 km downrange with an altitude of 187 km. The inert mass of the second stage (249 kg) is jettisoned, and the vehicle coasts for another 147 seconds up to an altitude of 278 km (150 nautical miles). When the horizontal velocity is added to the original speed of the launch site (463 m/s with respect to the center of the Earth) the inertial velocity is 7,659 m/s, compared to a required circular orbit velocity of 7,748 m/s. The vernier thrusters add the necessary 89 m/s of delta-V to circularize the orbit at 278 km altitude, and then to rendezvous with the propellant depot by radio command as described above. The drag and gravity losses of the first stage are 384 m/s. The drag and gravity losses of the second stage are 189 m/s. The spent second stage burns up over the ocean.

In [3] we concluded that, when aerodynamic torques are considered (not just thrust misalignment torques), the benefits of spin-stabilization disappear. Control propellant usage is minimized when the center of mass starts out about 29 cm behind the center of pressure (which could be pre-set by attaching small trim tabs to the structure). As the propellant from stage 1 is expelled, the center of mass moves forward. At 24 seconds after ignition, the center of mass passes through the center of pressure. By first stage burnout, the center of mass has moved 126 cm ahead of the center of pressure. The first peak in aerodynamic torque occurs 10 seconds after ignition at 4.8 kNm of torque. After the center of mass passes through the center of pressure, the second peak in aerodynamic torque occurs 42 seconds after ignition, with 5.0 kNm of torque (in the opposite direction). Without spin stabilization, about 13 kg of propellant would be expelled by the vernier thrusters to counter this torque over the 1st stage burn. We concluded earlier that each vernier engine needs to deliver about 800 N of thrust. Since the two vernier thrusters are located at least 3 meters forward of the center of mass during the first stage burn, the maximum counterbalancing torque that the vernier thrusters could exert is about 5 kNm, or about the same as is required. To further reduce the propellant consumed for control during the aerodynamic portion of the flight, one possibility is to affix small canard fins on the exterior of the nozzles of the vernier thrusters. We speculated in [3] that it might be good to attach small fins to the vernier rockets that protrude out of the side of the vehicle, deflecting the air stream and providing control authority without firing the engines. The mass impact of these small fins would be traded against the mass savings in tankage needed to carry the offset propellant.

We have assumed that each vernier engine has a specific impulse of 420 s, slightly lower than the main engines because of the somewhat adverse surface-to-volume ratio of the small-diameter thrust chamber. However, we assume that the design of the vernier engines is based on the same methodology given in [8]. Because it is a regenerative engine (e.g. the coolant for the thrust chamber is dumped back in as fuel), it retains a relatively high specific impulse. To generate ~800 N of thrust would require a propellant flow rate of about 0.2 kg/s. As previously mentioned, the propellant injectors in the vernier engines are expected to be the same as those in the first and second stage, and to operate at the same chamber pressure. The 41 kN second stage engine derived in [2] has 52 injectors, meaning that each injector accounts for about 790 N of thrust. So each vernier engine could achieve approximately the desired thrust and propellant flow rate using exactly one injector as seen in Figure 3.

Another issue addressed in [3] is possible ice build-up on the tether. The equatorial climate of the launch site has high humidity at low altitudes, but the tether spans the hot, humid conditions at the surface and the frigid, dry conditions at the balloon. There would be a transition zone where sustained ice buildup on the tether is possible. Unchecked, this could drag down the balloon. Fortunately, our concept involves a cable-car that runs both up and down the tether every 96 minutes. This cable-car would flex the tether around wheels, which would shatter any ice buildup. The cable-car could also carry special implements to remove excess ice. In our concept, there would also be a stationary, "lightweight" tether that carries aircraft warnings and also pulls the main tether off-vertical to facilitate safe rocket launch. This stationary tether may also need to have some means for clearing ice build-up. One possibility would be to have many small solar-powered cable cars that each carries an aircraft warning (strobe light and radar reflector). Each small cable-car would move up and down its assigned section of the tether, clearing the ice.

The significant aerodynamic forces due to winds at the launch altitude would necessitate use of a superpressurized balloon that could maintain the balloon shape and avoid fatigue-induced failure of balloon material that is flagging in the wind. There are two basic design options: a spherical balloon that uses a high strength film plus fabric laminate material, or a pumpkin balloon, which is a tendon reinforced polyethylene film balloon. Fabric plus film laminates are commonly used materials for blimps and aerostats, while the pumpkin balloon is being developed by NASA for ultra long duration scientific balloon flights at very high stratospheric altitudes (~36 km). At the present time, it is unclear which design option would be best for the rocket launch application. The spherical fabric balloon is likely to be more robust and have a longer lifetime, but with the probable disadvantages of being more expensive to build and more massive than a pumpkin balloon. Prototypes would need to be built and tested to quantify the cost, mass and lifetime issues, after which a final balloon design option could be selected.

3. ECONOMIC ANALYSIS

In [16] Mike Griffin and Bill Claybaugh describe a simplified economic model for estimating the cost of access to space using parameters such as the specific cost of dry hardware, the fraction of hardware re-use, etc. They discuss production volume, pointing out that the specific cost of transportation system hardware drops by two orders of magnitude as the production volume increases by three orders of magnitude for products such as airplanes, boats and cars. But they conclude that "really significant cost reductions, to below $1000/lb-payload... require expendable launch vehicles to be built for prices similar to those for boats [$50-$100/lbm when manufactured at 1k-10k/y]. This seems unattainable with any presently foreseeable unit production rates." The thesis of this paper is that production volumes comparable to boats or even automobiles (10k-100k/y) are indeed feasible.

We can make our own simplified economic analysis that addresses the effect of low-cost propellant launch and hardware re-use. Assume that dry hardware costs $100k/kg to prepare for launch, which is roughly the cost of the Apollo dry hardware inflated to current prices. Furthermore, assume that 85% of the total mass needed in LEO for each human mission beyond LEO is propellant (same as Apollo). Also let us assume that dry hardware is launched at $10k/kg (e.g. the conventional launch approach). Now dry hardware that is single-use could be built lighter-weight than hardware that is designed for multiple re-uses. Let us assume a power-law relationship, so that each doubling of dry mass gives some number "k" of possible reuses, at fixed dry hardware cost. (Clearly mass, cost, and number of reuses are somewhat independent - in principal we could get more reuses at fixed mass by increasing cost. But in this case, we fix cost and explore the relationship between mass and number of reuses.)

The results of this analysis are plotted in Figure 7. The relative cost of the mission is plotted vertically, with the cost of the single-use mission (e.g. Apollo) where propellant is launched at the same specific cost as the dry hardware having a relative cost of one. In each case, we plot two curves where we assume that each doubling of dry mass would allow either 4 uses of each piece of hardware, or 10 uses, again at fixed dry hardware cost. This range of reuses seems to bound what is reasonable. The number of actual uses is plotted along the horizontal axis, with the dry hardware mass increasing according to the assumed power-law. Each pair of mass-reuse power-laws is plotted for different costs of delivered propellant - $10k/kg (conventional launch), $3k/kg, $1k/kg, and $0.3k/kg.

The conclusions of Figure 7 are striking - note that the benefits of reuse reach the point of diminishing returns after about 5-7 uses, and the overwhelming majority of the benefit of reuse has been achieved within 3-4 uses of the dry hardware. Note further that the overwhelming majority

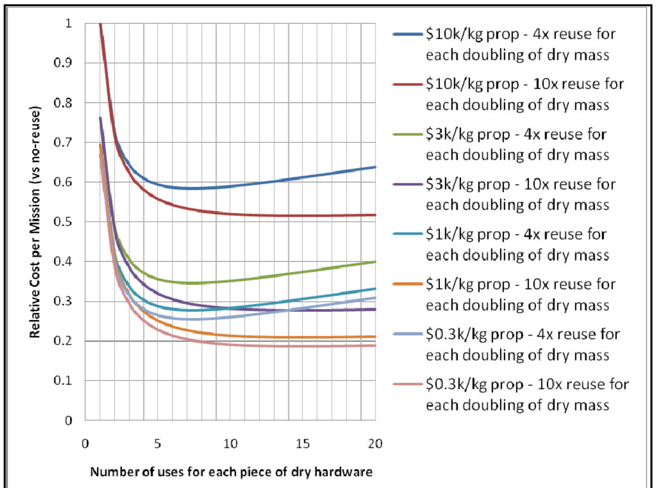

Figure 7: Effect of hardware reuse and low-cost propellant launch on mission cost.

of the benefits of low-cost propellant comes from reducing the launch cost of propellant from $10k/kg to $3k/kg, and that there is virtually no benefit to driving the cost of propellant down from $1k/kg to $0.3k/kg. Note that it doesn't really matter very much whether a doubling of dry hardware mass enables 4 uses, or 10 uses. Based on this analysis, it is reasonable to project that mission costs could be reduced by a factor between 3 and 5 compared to the conventional approach.

We must emphasize that this analysis is simplistic in that it assumes that all hardware is available for refueling after each use. With Apollo, for example, most of the hardware was discarded at various points along the mission in a way such that it would have been very difficult to make it available for refueling. So a detailed analysis of each particular mission concept would be required to do an accurate assessment of the benefits of hardware reuse and refueling. However, as discussed in [1-3], if we assume that all hardware is propulsively returned to LEO, L1, or LLO as appropriate, there are huge benefits available if the propellant is cheap.

Figure 8 gives a summary of the estimated costs for propellant launch from a tethered balloon, based on the component cost estimates given here and in [1-3].

4. PRELIMINARY THERMAL ANALYSIS

In [1] we assumed that the "hub" of the "star" propellant depot (Figure 4) would include a cryocooler that reduces or eliminates boiloff of the cryogenic propellant stored in the depot. In [2] we presented refined analysis indicating that each payload module would need its own cryocooler for boiloff control, since the radiator area requirements would be too great for the hub module alone. Boiloff of the cryogenic payload while in LEO is a major concern. As described in [2], the equilibrium temperature behind a sun-shield at the Earth-moon L1 libration point is only about 40K, so it should be possible to maintain the liquid hydrogen indefinitely at 20K with a single-stage cryocooler operating between the hydrogen and oxygen tanks. The LO_2 tank, operating at 89K, could be passively-cooled by radiating into space at the L1 point. But our operational concept calls for the star propellant depot to remain in LEO for about two weeks during its construction and prior to its self-propelled transit to L1. We now further examine the issue of boiloff during this two-week interval.

As described in [2-3], each payload assembly arrives on-orbit with approximately 200 kg of cyrogenic propellant as the payload (29 kg of LH_2 and 271 kg of LO_2). The dry mass of the payload assembly is estimated at 35 kg, including 9.5 kg of spherical cryogenic tanks. Two gimbaled, pressure-fed "vernier" rocket engines burn hydrogen and oxygen from the payload to provide thrust and control of the payload assembly as needed prior to and after integration with the "star" depot. A solar panel at the back of the payload assembly (between the two vernier engines) provides power for batteries and electronics co-located with the solar array.

Once on-orbit, the payload assembly is separated into a "hot" assembly and a "cold" assembly. The hot assembly includes the solar array, batteries, electronics, and vernier engines. The cold assembly includes the cryogenic tanks. Separating the hot assembly (kept pointed always at the sun) and the cold assembly (pointed away from the sun) is a thermal barrier. This consists of multi-layer blanket material, at least some of which is "spring-loaded" to expand beyond the nominal skin of the rocket fairing so that it completely shadows the cold assembly from the sunlight. This expanded sunshield may deploy at the time of separation of the payload from the second stage - perhaps the second stage could pull off the cylindrical skin of the payload as part of separation, revealing the thermal shield for spring deployment. It may be desirable to "fan-out" the foil elements so that they have extra space between layers so extra radiative cooling could occur out the sides of the blanket, similar to what is planned for the James Webb telescope sunshield. Within the cold assembly, the liquid hydrogen tank (at 20K) needs to be thermally isolated from the liquid oxygen tank (at 89K).

Reference [17] provides test data showing that multilayer insulating blankets leak heat between a cryogenic side and a room-temperature side at the rate of about 1W/m^2. Let us assume that stainless steel flex lines are used for the cryogenic fluids. Reference [18] indicates that stainless steel has a thermal conductivity of about 2 W/m-K at 20K. A 25 mm diameter stainless steel flex line with 1 mm wall thickness has a metal cross-section of 80 mm^2, and so with a thermal gradient of 100K/m it leaks heat at only 0.016 W. The heat of vaporization of LH_2 is about 450kJ/kg. So if the thermal blanket around the hydrogen tank has an area of 5 m^2 and a leak rate of 1W/m^2, and if all the stainless flex lines and mechanical linkages have a leak of 2 W, then the total heat leak into the tank is 7 W and the boiloff rate

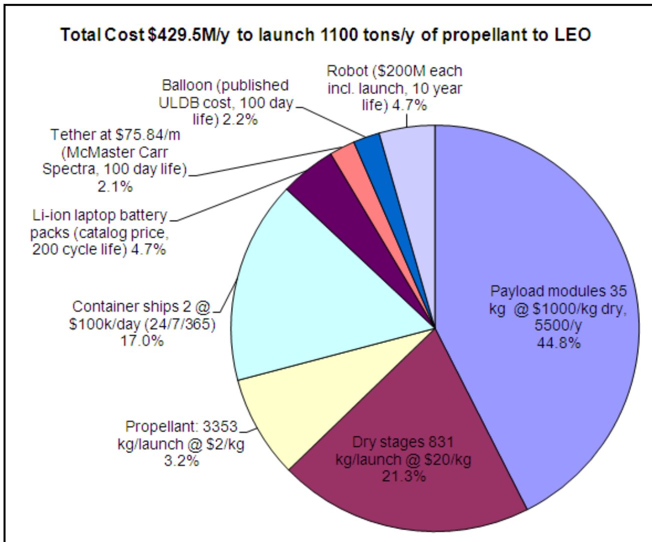

Figure 8: Estimated cost of propellant launch from a tethered balloon - $390/kg.

(without cryocooler) would be 0.056 kg per hour. This would boil off 19 kg in 14 days - about 65% of all the hydrogen. One possibility is to over-size the hydrogen tank and simply allow this much hydrogen to boil away. Since hydrogen is a small fraction of the total payload, this is not much of a mass penalty (~10% of payload mass).

It is clear that valves need to prevent the cryogenic liquids and gasses from transiting down the stainless steel flex lines between the cold and hot assemblies - otherwise our estimate of the heat leak of the flex lines would be far too low. This means that the valves, presumably highly-polished metallic ball valves, need to be on the "cold side" and mechanically actuated through linkages from the "hot side". This could be accomplished with cables or pushrods. Based on the thermal conduction of the metallic flex lines, we estimate that the heat leak through such mechanical linkages would be negligible.

Reference [19] describes an extremely detailed boil-off analysis of several interplanetary spacecraft, of which the Mars Sample Return Earth Return Vehicle (configuration M3) is most similar to our application. That vehicle carries 5148 m/s of LH_2/LO_2 propellant, is solar powered, and operates in the relatively "hot" environment of Mars orbit for a long period of time. They assume the Passive Orbit Disconnect Strut (PODS) technology, which provides stiff load paths during launch that are separated on-orbit to reduce the heat leak to "1/10th that of state-of-the-art struts," with the penalty of reducing the natural frequency by a factor of about 3 (the thermal conductivity is about proportional to the square of the frequency). Despite the fact that this spacecraft carries >10 times as much LH_2 as the vehicle considered here, it only had a heat leak of ~0.8W during interplanetary cruise at 1 A.U, requiring about 300W of power from the main bus for cooling. In equatorial Mars orbit, where the conical shield axis could be aligned with the polar axis to continuously shield both from the sun and from Mars, the bus power requirement for cooling was only about 400W. Since the radiative temperature of Mars is not that much different from Earth (at least compared to 20K) we can infer that it is possible that we could maintain zero-boiloff in LEO with substantially less than 400W of bus power devoted to a cryocooler. This suggests that a similar alignment between the conical shield axis and the polar axis of the Earth may be worth considering, despite the changes required to the mating of adjacent payload modules in forming the "star" and the necessity of gimbaling the solar arrays. Perhaps the solar array gimbals could be combined with the thruster gimbals. Presumably the conical heat shield would be open toward the north star in the northern winter and be pointed oppositely in northern summer to optimize the view angles of the solar array.

The overall conclusion is that a cryocooler may not be necessary to manage boiloff in LEO, if the hydrogen tank could be somewhat oversized, but a small cryocooler is desirable anyway to manage boiloff for indefinite periods at L1, as described in [2]. It seems likely that this cryocooler, operating between the hydrogen and oxygen tanks, could perform a useful function during the loiter in LEO, but this is not required for the integrated system to be effective.

5. EXPLORATION ARCHITECTURE IMPLICATIONS

By refueling the Exploration hardware now in development, significant extra mass could be delivered to the moon with every mission. For example, by refueling the Earth Departure Stage (EDS) in LEO so that it could perform both the Trans-Lunar Injection and the Lunar Orbit Injection (LOI) burns, and by launching the Altair lunar lander with empty descent tanks, then approximately 80 metric tons of extra payload (including possible extra structure needed to take the launch loads) could be put into LLO as compared to the non-refuelable baseline architecture. This approach also solves the "loiter boil-off problem," where either the EDS or the Orion (crew exploration vehicle) may have to wait for the other to launch and rendezvous. Because the Altair descent stage has been envisioned as performing the LOI burn for both the Altair and the Orion/Service Modules, it has extra-large hydrogen and oxygen tanks. This is particularly advantageous for our concept. A "block upgrade' that adds refueling capability, modest extra structure to take redistributed launch loads, and additional restarts to the EDS and Altair would enable quantum improvements in the architecture. The Altair descent stage could make multiple trips back and forth between lunar orbit and the lunar surface. In this configuration, a trip down to the surface could carry about 5600 kg of payload in addition to the fully-fueled ascent vehicle carrying a crew of four. This represents about 90 days of provisions for the crew. The ascent stage would be carried on a cargo pallet and offloaded and transported away in the same manner as other cargo from the top deck of the descent stage [20]. (The 5,600 kg of payload includes the mass of the cargo pallet.) At that point there would be enough propellant remaining in the tanks for the descent stage to return to LLO. Because of boil-off issues, presumably the descent stage would remain on the surface for as little time as possible. One means to utilize this capability is for the previous crew to take off for LLO at about the same time as the new crew lands. Before the old crew departs for Earth in their Orion vehicle, they could perform EVAs to attach additional cargo and provision modules (delivered into LLO by the refueled EDS) onto the refueled Altair descent stage that has returned-to-orbit. The Altair could then robotically land those extra provisions onto the surface, landing almost 12 metric tons on each round-trip. This process could repeat several times. The current strawman concept for the Altair is envisioned to use the venerable Pratt & Whitney RL10 LO_2/LH_2 engine. The data sheet for a current model of the RL10 (the RL10B-2) lists a thrust of 110 kN, a service life of 3,500 seconds and 15 total starts [21]. Each trip down to the lunar surface and back to orbit would consume about 900 seconds of life and 3-4 starts, so that each Altair could make three round trips to the lunar surface without any engine upgrades. Enough rated life and restarts remain for a fourth one-way trip down to the lunar surface, so that a total of almost 41 tons of cargo is put on the lunar surface over the four landings. In this way, a single Ares-V launch could put onto the lunar surface about the same payload as the first 10 missions combined under the non-refuelable architecture. With a modest increase in the life and number of restarts of the Altair descent engine, the amount of cargo delivered per launch of each Ares-V could double again, so that each mission puts as much cargo on the surface as had been envisioned over the entire first decade of a non-refuelable architecture.

For future Mars exploration, a large quantity of propellant could be stockpiled at the Earth-moon L1 point, where the thermal environment is conducive to long-term storage of liquid hydrogen. A single Ares-V launch, refueled in LEO and again at L1, could land a crewed vehicle with a dry mass of about 55 tons onto the surface of Mars. The refueled EDS would launch the stack away from L1 (~900m/s) and perform the Trans-Mars Injection at the top of the Earth's atmosphere (~1200m/s, starting almost at escape velocity). The zero-boiloff, two-stage Mars lander (launched dry and fueled at L1) would perform an all-propulsive "stop and drop" maneuver at Mars, with the first stage doing the "stop" from hyperbolic entry velocity (~6400m/s) and the last stage performing a subsonic constant-velocity "drop" through the atmosphere from an altitude of ~125 km. The resulting landed mass of ~55 tons is greater than that envisioned by most crewed Mars mission studies, even those requiring ten or more heavy-lift launches. This propellant-intensive approach "solves" the difficulties associated with landing heavy payloads on Mars using aeroshells [22]. While such an approach closes with respect to mass, the volume of such a Mars vehicle would be substantially greater than the current launch shroud envisioned for the Ares-V. Using two Ares-V launches, this difficulty would be eliminated by using the second EDS as the first stage of the Mars Lander.

6. SUMMARY AND CONCLUSIONS

This paper describes a system of small rockets (performance summary given in Table 1) that are launched from an equatorially-tethered balloon at an altitude of 22 km. The purpose of this paper is to further elaborate the engineering and economic issues associated with this proposed system for low-cost propellant launch from a tethered balloon. We believe that this system could deliver propellant for under $1000/kg to either the Earth-moon L1 point (where it could be stored indefinitely by passive means) or to a 28.5º inclination LEO orbit where it could refuel dry stages arriving from Florida.

Modest initial steps toward demonstrating the feasibility of this approach include prototyping the small vernier thrusters needed for the payload module, tethering a balloon for perhaps a week at the nominal altitude near Baker Island, prototyping key components of the propellant-depot robot, demonstrating the feasibility of building the "cable car" launcher that goes up and down the tether, and continued system engineering and analysis. An important following step would be to conduct a longer-duration proof of concept experiment at Baker Island using a subscale balloon and tether system to demonstrate stable, long duration tethered balloon flight at an altitude of 22 km. This would include a buoy with tether-management spool, multiple solar/battery cable cars to clear ice and provide aircraft warning, and the balloon. Preliminary analysis indicates that a sub-scale balloon in the range of 20 to 30 m in diameter would be required for such an experiment, with a likely choice of a high strength fabric-based balloon material and a spherical superpressure balloon architecture. Such balloons could be made robust enough to survive the expected wind speeds both during ascent and float, while the superpressure would help preserve their shape at float and thereby minimize aeroelastic deformations and attendant material fatigue. Important secondary outcomes of the sub-scale experiment would be to obtain long-term direct measurements of the wind, icing, and lightning conditions at Baker Island and to gain operational experience with the cable-car and tether management subsystems.

Using on-orbit refueling of currently-planned Exploration architecture hardware, with only modest block upgrades, a total of 40 to 80 tons of provisions and equipment could be delivered to lunar orbit along with a crew of four by each Ares-V/EDS/Altair and Ares-I/Orion launch. A refueled Altair descent stage could land on the surface and still have enough propellant to lift the fully-fueled ascent stage and crew back to LLO, with the ascent stage used only as an abort contingency. Ultimately, the EDS and Altair could make multiple all-propulsive round-trips from LLO to LEO so that most lunar missions require only a single Orion/Ares-I launch. In this concept, the Altair could serve as an “Apollo-13” lifeboat on both the forward and return legs of the trip. Over time, many fully-fuelled ascent stages would accumulate on the surface to provide a wealth of emergency abort options. If 1-2G$/y were spent on launching propellant at 1k$/kg (as delivered into the target spacecraft), and another 1-2G$/y were spent launching reusable dry hardware and provisions at a cost of 10k$/kg (e.g. two Ares-V launches per year), then lunar missions involving only Ares-I/Orion launches could probably be afforded every two months, marking an exploration program that would truly excite and captivate both the public and the Congress. A crewed Mars mission could be mounted with as few as two Ares-V launches.

ISP=441s	Dry mass (kg)	Propellant (kg)	Stack at ignition (kg)	Horizontal ΔV (m/s)	Drag/gravity/control losses (m/s)
Stage 1	582	2329	4439	2587	631
Stage 2	249	1024	1528	4609	189
Payload assembly	35	220	255	89	248
Launch site velocity				463	
Total	866	3573	4439	7748	

Table 1: Summary masses and delta-Vs

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REFERENCES

[1] Wilcox, Brian H., "Low-Cost Propellant Launch to Earth Orbit from a Tethered Balloon", proc. IEEE Aerospace Conference, 5-9 March 2006, Big Sky, MT, paper 0-7803-9546-8/06 (1240).

[2] Wilcox, Brian H., Schneider, Evan G., Vaughan, David A., Hall, Jeffrey L. "Low-Cost Propellant Launch to Earth Orbit from a Tethered Balloon – An Update", proc. IEEE Aerospace Conference, 2-7 March 2008, Big Sky, MT, paper 1-4244-1488-1/08 (1377).

[3] Wilcox, Brian H., Schneider, Evan G., Vaughan, David A., Hall, Jeffrey L. "Low-Cost Propellant Launch to LEO from a Tethered Balloon - Recent Progress", proc. IEEE Aerospace Conference, 7-13 March 2009, Big Sky, MT, paper 978-1-4244-2622-5/09 (1418).

[4] A good discussion of the need for vast amounts of propellant in LEO to support the exploration vision is contained in the plenary talk "CA-1: Exploration Super System (ESS)" by Ralph Ewig of Andrews Space at the NASA Exploration Systems Technology & Systems Integration Technical Interchange Meeting, 13-17 December 2004, Ritz-Carlton, Pasadena, CA, CD proceedings disk1:Tuesday Plenary/6_Andrews Space CE&R.ppt (available from conference organizer Neville Marzwell of the Jet Propulsion Laboratory).

[5] See http://www.spacefuture.com/archive/designing_reusable_launch_vehicles_for_future_space_markets.shtml. Last accessed 14 Dec 2009.

[6] See http://www.solar.ifa.hawaii.edu/Tropical/summary.html. Last accessed 14 Dec 2009.

[7] See http://www.physicalgeography.net/fundamentals/

7q.html. Last accessed 14 Dec 2009.
[8] Huzel, Dieter K. and Huang, David H.; *Modern Engineering for Design of Liquid-Propellant Rocket Engines*, 1992, published by the AIAA, ISBN 1-56347-013-6. Figure 3 of this paper was originally published as Figure 4-61 in this reference, reprinted with permission from the American Institute for Aeronautics and Astronautics, Inc.
[9] Kelkar, Anish; Roth, Richard; and Clark, Joel; "Automobile Bodies: Can Aluminum Be an Economical Alternative to Steel?", *JOM* (Journal of The Minerals, Metals & Materials Society), 53 (8) (2001), pp. 28-32. Available at http://www.tms.org/pubs/journals/JOM/ 0108/Kelkar-0108.html (last accessed 14 Dec 2009). Figure 2 of this paper originally published as Figure 7 in this reference reprinted with permission from The Minerals, Metals & Materials Society (TMS), Warrendale, Pennsylvania USA.
[10] Ambrose, Robert O.; Aldridge, Hal; Askew, R. Scott; Robert, R. Burridge; Bluethmann, William; Diftler, Myron; Lovchik, Chris ; Magruder, Darby; Rehnmark, Fredrik; "Robonaut: NASA's Space Humanoid", IEEE Intelligent Systems Volume 15 , Issue 4 (July 2000) Pages: 57 – 63, 2000, ISSN:1541-1672
[11] Wilcox, Brian; "An Evolutionary Strategy for Telerobotic Operation Via Geosynchronous Relay", IEEE International Conference on Robotics and Automation, Workshop on Integration of AI with Robotics, Scottsdale, AZ, May 1989.
[12] Gershman, Robert (a propulsion engineer on the Saturn S-IVB upper stage, now at JPL) personal communication, May 2008.
[13] See http://www.soest.hawaii.edu/UMC/RVKOK.htm, last accessed 14 Dec 2009.
[14] Jack, John R. and Moskowitz, Barry, NACA Research Memorandum E53L11b, National Advisory Committee for Aeronautics, Lewis Flight Propulsion Research Laboratory (now NASA Glenn Research Center), Apr. 2, 1954. Available at http://naca.central.cranfield.ac.uk/reports/1954/naca-rm-e53l11b.pdf (last accessed 14 Dec 2009).
[15] Jiang, Jonathan H.; Wang, Bin; Goya, Kenshi; Hocke, Klemens; Eckermann, Stephen D.; Ma, Jun; Wu, Dong L.; Read1, William G.; "Geographical distribution and interseasonal variability of tropical deep convection: UARS MLS observations and analyses", JOURNAL OF GEOPHYSICAL RESEARCH, VOL. 109, D03111, doi:10.1029/2003JD003756, 2004, available at http://mls.jpl.nasa.gov/library/2003JD003756.pdf (last accessed 14 Dec 2009). See figure 1a and 1b for wind speed distribution in Western Pacific.
[16] Griffin, Michael D. and Claybaugh, William R. II, "The Cost of Access to Space," proc. British Interplanetary Society 60th Anniversary Meeting, Hastings, England, 15-16 October 1993.
[17] Fesmire James E., Augustynowicz Stan D., Darve C. "Performance characterization of perforated multilayer insulation blankets," proc. 19th International Cryogenic Engineering Conference, ICEC 19, Narosa Publishing House, New Delhi, 2003, pp 843-846. Available at http://cdarve.web.cern.ch/cdarve/Publications_CD/ICEC19_MLI.pdf, last accessed 14 Dec 2009.
[18] Marquardt E.D., Le, J.P., and Radebaugh, Ray "Cryogenic Material Properties Database", National Institute of Standards and Technology, available at http://cryogenics.nist.gov/Papers/Cryo_Materials.pdf, last accessed 14 Dec 2009.
[19] Guernsey, Carl S., Baker, Raymond S., Plachta, David, and Kittel, Peter, "Cryogenic Propulsion with Zero Boil-Off Storage Applied to Outer Planetary Exploration", 41st AIAA/ASME/SAE/ASEE Joint Propulsion Conference, 10-13 July 2005, Tucson AZ AIAA paper 2005-3559.
[20] Wilcox, Brian H., "ATHLETE – A Cargo and Habitat Transporter for the Moon", proc IEEE Aerospace Conference, 7-14 Mar 2009, Big Sky, MT, paper 1417.
[21] Pratt & Whittney RL10B-2 data sheet (P&W TS 6.04).
[22] Nancy Atkinson, "The Mars Landing Approach: Getting Large Payloads to the Surface of the Red Planet", Universe Today, 17 July, 2007 (http://www.universetoday. com/2007/07/17/the-mars-landing-approach-getting-large-payloads-to-the-surface-of-the-red-planet/, last accessed 14 Dec 2009).

BIOGRAPHIES

Brian H. Wilcox is a principal member of the technical staff in the Autonomous Systems Program Development Office at JPL. He was the supervisor of the JPL Robotic Vehicles Group for over 20 years, during which the group was responsible for planetary rover development leading up to the Sojourner and Mars Exploration Rovers. The group was responsible for development of the electronics, control, flight software, ground software, and mission operations of the Sojourner rover that explored part of Mars in 1997. Brian was personally responsible for the imaging and hazard detection sensors and the hazard avoidance algorithms on Sojourner. More recently, he is now the Principal Investigator for ATHLETE – the All-Terrain, Hex-Limbed, Extra-Terrestrial Explorer (see http://www-robotics.jpl.nasa.gov/tasks/ index.cfm). He has a B.S. in Physics and a B.A. in Mathematics from the University of California at Santa Barbara, and an M.S. in Electrical Engineering from the University of Southern California.

Evan G. Schneider is a junior at MIT where he is majoring in Mechanical Engineering and furthering his lifelong interest in the sciences. He is a graduate of The College Preparatory School, located in Oakland, California, where he won the Science Department prize at the end of his senior year. He designs, builds, and flies model rockets, airplanes, and helicopters, machines and welds, and races remote controlled cars in his free time. He is an avid tinkerer and has dismantled, reassembled, and occasionally improved objects ranging in complexity from clocks to hydrogen fuel cells. He worked as a summer intern at JPL under Brian Wilcox and David Vaughan in 2006 and 2008.

David A. Vaughan is the supervisor of the Propulsion Flight Systems Group at JPL. Prior to joining JPL, he was manager of the Analytical Engineering Department at Lockheed-Martin Michoud Space Systems in Huntsville, AL. He has over 30 years experience in cryogenic and storable propulsion systems. He has been involved with various technology developments including LH_2/LOX propellant densification, the X-33 LH_2/LOX propulsion system, no-vent fill, advanced LH_2/LOX turbo-pumps for the Space Shuttle main engine, and electric ion propulsion. He has a BSc. in Mechanical Engineering from Bath University, England, and a MSc. in High Speed Aeronautics from the von Karman Institute for Fluid Dynamics, Belgium.

Jeffrey L. Hall is a group leader and senior engineer at JPL who leads its technology development and mission application efforts in aerocapture and aerobots. His work includes design of aerocapture vehicles and missions, development of ballutes, Earth flight testing of balloons and airships, and design of advanced planetary aerobots. He has a Bachelors degree in Engineering Science from the University of Toronto, and Masters and Ph.D. degrees in Aeronautics from the California Institute of Technology.