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A Practical, Affordable Cryogenic Propellant Depot Based on ULA's Flight Experience

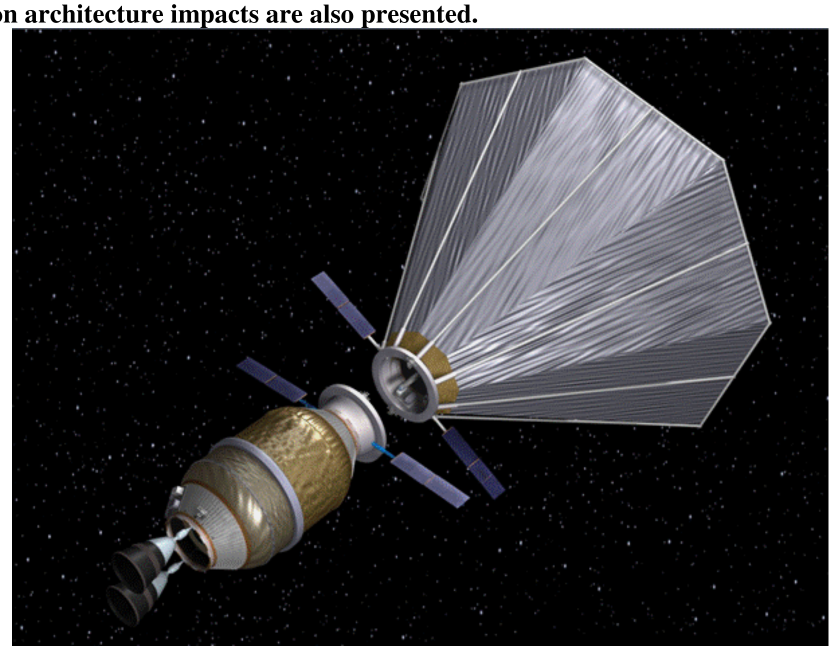
Bernard F. Kutter*, Frank Zegler†,
Gary O'Neil‡, Brian Pitchford§

United Launch Alliance

Denver, CO

Mankind is embarking on the next step in the journey of human exploration. We are returning to the moon and eventually moving to Mars and beyond. The current Exploration architecture seeks a balance between the need for a robust infrastructure on the lunar surface, and the performance limitations of Ares I and V. The ability to refuel or top-off propellant tanks from orbital propellant depots offers NASA the opportunity to cost effectively and reliably satisfy these opposing requirements. The ability to cache large orbital quantities of propellant is also an enabling capability for missions to Mars and beyond.

This paper describes an option for a propellant depot that enables orbital refueling supporting Exploration, national security, science and other space endeavors. This proposed concept is launched using a single EELV medium class rocket and thus does not require any orbital assembly. The propellant depot provides cryogenic propellant storage that utilizes flight proven technologies augmented with technologies currently under development. The propellant depot system, propellant management, flight experience, and key technologies are also discussed. Options for refueling the propellant depot along with an overview of Exploration architecture impacts are also presented.



* Sr. Staff, Manager Advanced Programs, United Launch Alliance, P. O. Box 277005 Denver Co 80127 MS U9115, AIAA Senior Member.
† Sr. Staff, Advanced Programs, United Launch Alliance, P. O. Box 277005 Denver Co 80127 MS U9115, AIAA Member.
‡ Lead Engineer, Thermal & Fluids Analysis Group, NASA Launch Services Program, John F. Kennedy Space Center, AIAA Member.
§ Engineer, Thermal & Fluids Analysis Group, Analex, AIAA Member.

Acronyms

AR&D	Autonomous Rendezvous and Docking
CEV	Crew Exploration Vehicle (Orion)
CFM	Cryogenic Fluid Management
CLV	Crew Launch Vehicle
COTS	Commercial Orbital Transportation Services
CTB	Centaur Test Bed
EDS	Earth Departure Stage
EELV	Evolved Expendable Launch Vehicle
g	Earth's Gravity
LAD	Liquid Acquisition Device
LEO	Low Earth Orbit
LOI	Lunar Orbit Insertion
LSAM	Lunar Surface Access Module (Altair)
LSP	Launch Services Program
MLI	Multi Layer Insulation
mT	Metric Tons (tonnes)
PMD	Propellant Management Device
RCS	Reaction Control System
SM	Service Module
TEI	Trans Earth Injection
TRL	Technology Readiness Level

I. Introduction

IN 2003, President George W. Bush started America on an exciting new era in space exploration where we will return to the moon and eventually extend human exploration to Mars and the rest of the solar system[1]. This journey begins with launches of the Ares I & V, Figure 1, rendezvous in low earth orbit (LEO), and acceleration to Earth escape of Altair and Orion, Figure 2. There currently is a problem with the plan; Ares V does not have enough performance.

Figure 1. NASA's current plans for launching the VSE is composed of the Aries I and V launch vehicles. Credit: NASA

Ares V is capable of delivering 69 mT to Earth escape velocity. However, the lunar missions require a minimum of 77 mT[2]. This 77 mT includes the Orion capsule (20.2 mT), the Altair lunar lander (45 mT), airborne support equipment (3 mT), and L2/L3 margins (9 mT). Even the 77 mT requirement is based on optimistic assumptions, including:

- Altair's current weight estimate is for a minimum functional design. NASA acknowledges that significant enhancements will be required to support an actual mission[3**].
- Back to back Ares V & I launches, with a 3 day orbital loiter for rendezvous and checkout. History suggests that dual launches in quick succession are very unlikely. ESAS assumed a more realistic 90-day maximum interval.

It is therefore very likely that the required lunar mission performance will continue to grow.

Figure 2. The EDS helps loft Altair to LEO and then accelerates the Orion-Altair combination to Earth escape. Credit: NASA

NASA is considering several very significant upgrades to Ares V to increase performance[4,5]. These enhancements include:

- Composite SRB cases
- 5.5 segment SRB's
- A sixth RS68 engine on the booster
- Composite tanks for the Earth Departure Stage (EDS)

Combined, these enhancements almost satisfy the 77 mT earth escape performance. Unfortunately, these upgrades will minimize commonality with the currently planned Ares I launch vehicle, will result in increased development costs and will not provide margins to deal with any additional performance issues that are typical for programs of this maturity.

We propose an alternative for satisfying mission performance needs through the use of on-orbit fueling of the EDS LO2[6]. The use of orbital fueling will allow NASA to maintain the Ares I/Ares V commonality, reduce the architectural cost and speed America's return to the moon while simultaneously stimulating the broad launch industry, benefiting space science, national security space and other space enterprises[7].

Orbital fueling of the EDS provides the opportunity to increase the lunar delivered payload by over 20 mT[8], Figure 3. Other independent studies have found similar results[9,10]. Such a large performance enhancement not only closes

[**] "The LDAC-1 minimum functional design provides the foundation vehicle for safety and reliability trade studies and analysis, that are being performed in LDAC-2."

the current performance gap, but provides a simple path to support future performance issues or enhance mission requirements.

Despite positive comments by Griffin regarding the use of propellant depots to support space exploration[11] their use for near term lunar missions has been assumed to be too technically challenging. This is due in part to the fact that cryogenic propellant transfer has historically been synonymous with zero-g propellant depot space stations, Figure 4. These typical cryogenic depot concepts also assumed zero-g mass transfer, zero boil-off and zero vent fill. Although admirable goals, these depot concepts erect technological barriers that have successfully blocked propellant depot development for 40 years, preventing realization of the enormous benefits that orbital fueling offers to space transportation in general.

This paper describes a concept for economical, near term propellant depots using methods with high Technology Readiness Levels (TRLs). These smaller depots are designed to be launched empty on a single EELV medium class launch vehicle.

NASA's current Exploration transportation architecture is ideally suited to take advantage of propellant depots. 44% of the entire LEO mass is contained in the EDS in the form of LO2. At lift-off, the EDS holds 224 mT of propellant (192 mT of LO2 and 32 mT of LH2)[12]. 60% of this propellant is consumed just getting to LEO, leaving the propellant tanks with 92 mT of propellant (79 mT of LO2 and 13 mT of LH2) for the Earth departure burn. If a depot were to provide the required EDS LO2 on-orbit, NASA could remove as much as 79 mT of the lift-off LO2 from the EDS. This would decrease the Ares V performance requirement while increasing Altair's mass allocation to meet actual needs. The loaded LH2 could also be increased to support boil-off over the desired 90 day LEO stays and providing more LH2 to support increased Earth departure performance. Combined, this will reduce Ares V development time and cost, improve mission reliability and improve lunar delivered performance.

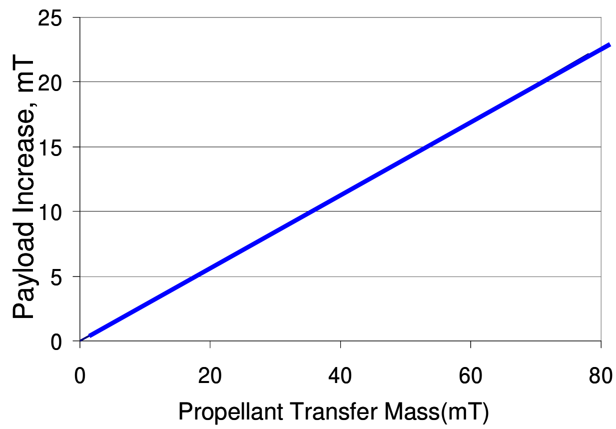


Figure 3. Orbital refueling of the EDS results in a tremendous increase in lunar delivered payload.

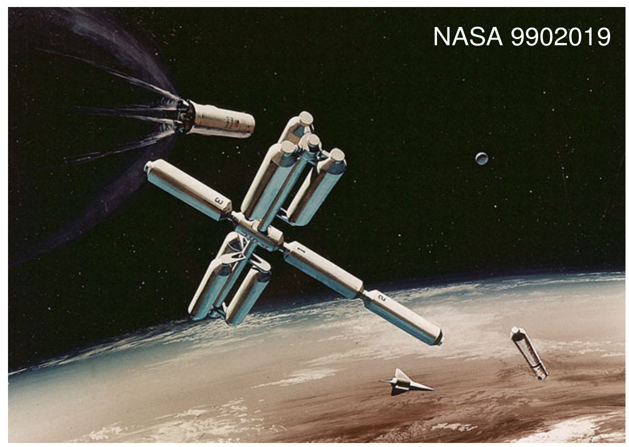


Figure 4. Permanent Space Based Propellant Depot. Credit: NASA

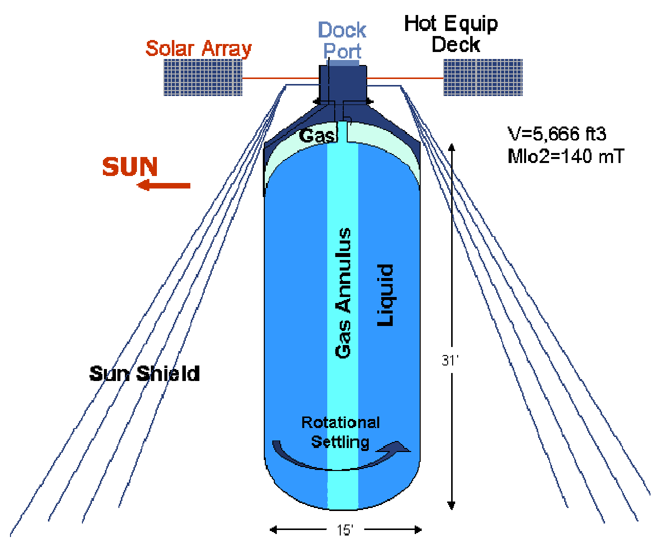


Figure 5. An affordable, near term propellant depot that utilizes existing and in development technologies to provide passive low to zero-boil-off cryogenic propellant storage.

II. Depot concept overview

The proposed depot is composed of a 180" diameter cryogenic tank that can be launched inside of existing 5m diameter payload fairings. This light weight, thermally efficient depot is designed to contain a single fluid, either 140 mT of LO2 or 15 mT of LH2, Figure 5. At the top of the depot is the hot equipment deck which contains the docking collar, avionics, control valves and station keeping propulsion. Sandwiched between the cryo tank and the equipment deck is the vapor-cooled, low conductivity support truss and a thermal isolation gas reservoir providing a torturous thermal path reducing boil-off. To minimize structural mass and maximize the depot propellant capacity within the payload fairing envelopes, this reservoir and the cryo tank share a common, insulated bulkhead. Once on-orbit, a deployable sun shield cocoons the cold structure and cryo tank to minimize heating from solar and Earth sources, while allowing residual heat to radiate to deep space, Figure 6. This system level design utilizes existing, flight proven elements that enable passive, very low boil-off LO2 or LH2 storage in an affordable, reliable package.

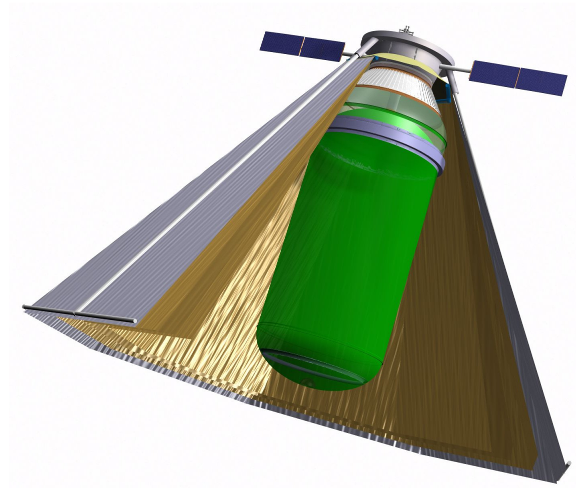


Figure 6. Once on-orbit a pneumatically deployed sun shield protects the cryogenic propellant from solar and Earth radiation.

The entire depot slowly spins about its longitudinal axis to provide centrifugal acceleration. This acceleration provides positive gas/liquid separation by forcing the liquid outward toward the tank sidewall, producing a gaseous annular ullage in the center. This passive gas/liquid separation greatly eases the depot cryogenic fluid management. Pressure control is through the venting of this gaseous core, and is similar to the settled ullage venting of existing cryogenic upper stages. The centrifugal settling also simplifies propellant acquisition, avoiding the need for liquid acquisition devices. Propellant transfer into and out of the depot is accomplished via differential pressure, similar to the way engines are fed on existing cryogenic stages. The well insulated depot can accommodate periods of zero-vent and no rotation to support operational needs, such as docking.

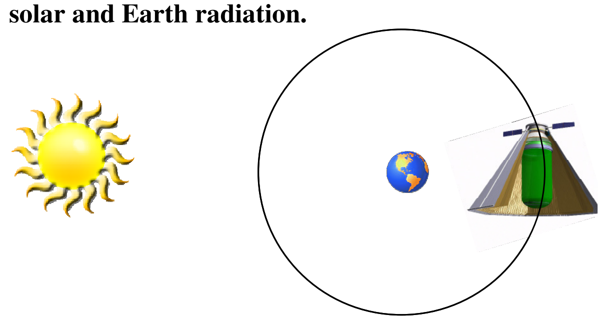


Figure 7. A north ecliptic pointing LEO depot allows the conic sun shield to shield the cryo tank from both the Sun's and Earth's radiation.

The vented gas is stored in a large, cold gas reservoir at the front of the sump. During quiescent operations, the reservoir is maintained at just below tank pressure. This reservoir serves as the last heat sink between the equipment deck and the cryogenic propellant tank. The reservoir also supplies gas for the Reaction Control System (RCS) as well as positive pressure expulsion of liquids during propellant transfer.

For launch, the sun shield is stored on the equipment deck. Following separation from the launch vehicle the multiple layers of the sun shield are deployed. For a LEO depot, the deployed sun shields form concentric cones surrounding the depot. The depot maintains a northern orientation, Figure 7, which enables the sun shield to shadow the tank from both solar and terrestrial heating throughout the LEO orbit. The multiple, concentric conical shield layers are maintained at different angles and provide an open path to direct thermal energy out to deep space, and away from the cryogenic propellant tank. Depots located at LaGrangian points do not encounter significant Earth heating and can use a sun shield similar to the James Webb Space Telescope[13].

III. Depot Mass

The proposed depot builds on existing flight proven elements to minimize risk and uncertainty while still resulting in a light weight system, Table 1. The light weight tank builds on Centaur's 50 years of, monocoque tank construction, updated with modern material advances included in the Delta upper stage. The tank domes are spun aluminum alloy, machined to provide final contours and thin skin gauge. The domes are friction stir welded to the thin monocoque walls constructed of aluminum alloy sheet material. To minimize weight and enhance orbital thermal performance, the depot will be launched empty. With the tank launched empty, foam insulation is not required reducing mass by ~200 Kg. Foam is only required for existing cryo upper stages during atmospheric operations, and is nearly useless as an insulator on-orbit. Launching the depot empty will allow thinner (and lighter) tanks that are designed for the orbital pressure loads, rather than the higher loads associated with a full tank as it launches and traverses the atmosphere. These thinner walls also minimize heat transfer along the tank walls.

The fluid control system is very similar to that already used on current ULA stages, allowing use of existing flight qualified hardware for pressure control and fluid transfer. Redundant, low power draw avionics, similar to those used on- Orbital Express, are assumed for depot command and control, communication, and guidance. The low power not only minimizes the scale of the solar arrays, but also is key to reducing the heat transfer from the warm avionics deck to the cryogenic tank.

Table 1. The proposed simple propellant depot weight is derived from Centaur and Delta US actual weights with allowances for the new hardware. This light weight depot enables launch on an Atlas 501.

System	Mass (mT)
Tank	1.3 mT
Dry Structure	0.2 mT
Avionics & Power	0.4 mT
Deployable Sun shield	0.5 mT
Propulsion and Pneumatics	0.2 mT
Weight Growth Allowance (10%)	0.3 mT
Total Mass	2.9 mT

IV. Thermal Modeling

As alluded to in previous sections, the thermal control scheme for the propellant depot utilizes passive concepts to minimize complexity. Thermal modeling has been developed using analytical tools widely used in the aerospace industry: Thermal Desktop(c) with its components of RadCad and SINDA/Fluint.

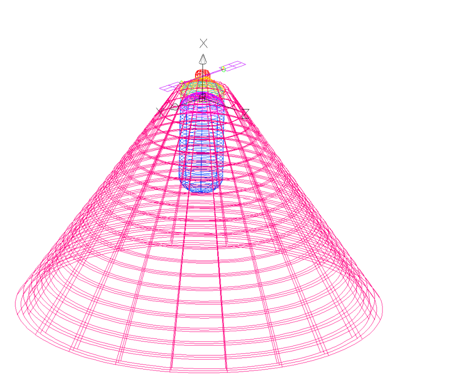


Figure 8. The thermal analysis accounts for the major heat sources, Solar, Earth and avionics and the radiative and conductive flow paths through the depot. Credit NASA

The modeling simulates the depot in LEO with a full load of LO2 and the sun shield deployed, to quantify the absorbed Sun and Earth heat loads and the ability of the conceptual passive thermal control system to minimize parasitic heating to the LO2 tank. The modeling includes reasonable fidelity in the tank structure to capture axial variations in the radiation environment, mainly the varying radiative interaction with the sun shield and deep space, Figure 8. Similarly, the sun shield is nodalized to a reasonable fidelity to capture both circumferential and axial temperature gradients. This fidelity is warranted given that the shield is the primary method for intercepting the significant Sun and Earth heat loads, as well as providing the primary radiative influence to the LO2 tank heat loads.

The structure associated with the docking assembly, avionics support, and solar arrays are included to simulate environmental heat absorption and effectiveness of the GO2 intermediate volume in intercepting these heat loads prior to reaching the LO2 storage tank. The avionics support structure and docking assembly is simulated with a white paint coating for favorable ratio of solar absorptance to infrared emittance. It is assumed that power dissipating avionics will not utilize the support structure as a heat sink, but will locally reject waste heat. For this reason, the sun shield support structure is proposed to be mounted aft of as much of the avionics as possible, such that after its deployment a view to space is preserved for avionics units. Avionics units are not included in this thermal simulation, and given their thermal isolation from the structure, this does not significantly impact parasitic heating to the LO2 volume.

Additional insulation from the warm avionics structure is accomplished via a vacuum space between the LO2 tank dome and the GO2 volume. This vacuum barrier method is used on existing LO2/liquid hydrogen tanks that use a common bulkhead to separate the fluid tanks.

The modeling of the sun shield captures three layers of material in concentric cones differing by 1° in cone angle, and an overall shield half-angle of 34° (this angle is optimized to the planned orbit altitude). The shield layers are closest at the "top" (deployment origin) and widest at the aft end of the LO2 tank. The use of specular shield materials for the tank-facing surface as well as the intervening surfaces allows increased views to the deep space sink via non-diffuse reflection of infrared energy. A Kapton material with vapor deposited aluminum (VDA) surface on one side is proposed. The VDA side is on the inner side of the shield layer (tank side) to take advantage of the low emissivity and minimize transmission of heat to the tank. The Kapton side of the material is considered on the outer side of the shield layer (space side) to utilize the favorable ratio of solar absorptance to infrared emittance facing the incoming solar radiation, minimizing the outermost layer's temperature.

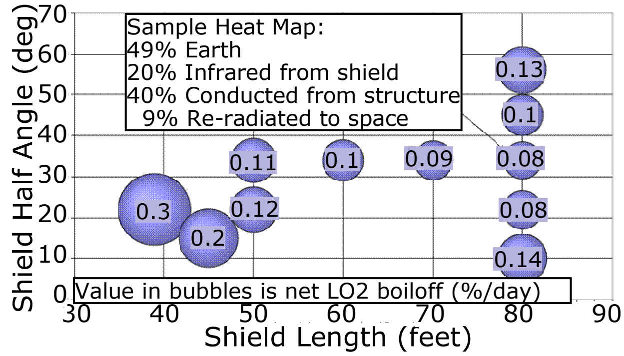


Figure 9. Preliminary thermal results show that this simple depot is capable of supporting passive long duration cryo storage with less than 0.1%/day boiloff with opportunity for further improvement. Credit NASA

The tank was simulated with and without surface multi-layer insulation (MLI). The goal would be to not require tank surface MLI to utilize a view to space for cooling. To achieve this goal, further detailed design of the deployable sun shield would be necessary, coupled with specific orbital information, in order to minimize Earth heat loads into the open end of the conical shield.

The results provided here are from an orbital simulation that uses parameters for a circular orbit at 1300 km altitude and a solar beta angle of 0°. This altitude is chosen to minimize material degradation due to atomic oxygen, potential for impacts to the sun shield, and heating from charged particles. A near zero beta angle results in a maximum Sun eclipse time which is beneficial for keeping the entire system cold.

Several shield configurations were analyzed to optimize the shield length and shield half angle, Figure 9. In LEO, the open end of the sun shield cone tends to collect Earth energy, so making the shield as long as practical helps to minimize these loads. For practical purposes, the length was limited to 80 feet, a length at which all Earth loads received by the LO2 tank are indirect, via reflections off of the inside of the sun shield. Analysis shows that LO2 equivalent side-wall absorbed heat fluxes of approximately 0.5 BTU/hr/ft² can be obtained for a tank with no surface MLI. Note that this is calculated by taking all heat loads, inclusive of conducted heat, into the tank and dividing by the total surface area of the tank. This is roughly equivalent to a boil-off rate of less than 0.1% of full tank volume per day.

Further design and analysis optimization to minimize parasitic heat loads can provide significant further improvement in the cryogenic fluid storage. These improvements include improved thermal isolation of the tank from the warm avionics structure, refined deployable sun shield geometry, and use of LO2 boil-off gas for cooling the sun shield.

V. Technology for depot

Settled operations significantly simplify all aspects of cryogenic fluid management enabling the maximum use of existing, mature upper stage cryogenic fluid management (CFM) techniques[14], Table 2. With settling, large-scale passive propellant storage and transfer becomes an engineering effort, not a technology development endeavor. The key elements enabling efficient, long duration cryogenic storage were refined in concert with NASA KSC[15], Figure 10. Table 3 provides a partial list of relevant CFM capabilities that have been demonstrated on the Centaur and Delta upper stages.

A. Low Acceleration Settling

Over the past 15 years, Centaur has spearheaded the development of ultra-low settling for CFM. Low-g settling provides a reliable method to separate liquid and gas. This settling can be continuous for short durations, or intermittent, separated by periods of zero-G (potentially weeks with adequate tank insulation) for longer missions.

Through improved understanding of low-g fluid behavior Centaur has reduced the standard parking orbit settling from 10^{-3} g to 10^{-4} g realizing a significant performance enhancement while maintaining adequate propellant control. In the quest for even more performance and longer mission duration, Centaur has demonstrated effective propellant control at accelerations down to 10^{-5} g, Figure 11. Similarly, in the 1960's Saturn also demonstrated effective settling at 2×10^{-5} g[16].

Rotational settling promises similar fluid control as with axial settling, figure 12, at potentially lower RCS propellant consumption. Building on the low acceleration fluid control mentioned above, ULA has developed a promising sequence enabling transition to centrifugal acceleration. Thanks to support from our DoD customer community, this centrifugal propellant control will be demonstrated on the DMSP-18 mission (AV-017) flying September 2008. This flight will demonstrate the effectiveness of liquid spin up, transition from axial settling to radial and back to axial settling with low acceleration and while venting.

B. Pressure Control

Pressure control of the depots is accomplished by thermal management of the cryogenic fluid. Heating, even if localized, results in propellant boiling that must be controlled to prevent detrimental pressure rises. Numerous methods of pressure control are available, including: ullage venting; thermodynamic venting; and active cooling.

Settled venting results in extremely robust tank heat rejection. This robustness is due to the fact that any localized propellant warm spots, due to penetration or other high heating sources, causes the propellant to boil regardless of the location in a tank. This liquid/gas separation enables heat rejection via venting for long coasts and has been demonstrated on 185 Centaur flights, 11 Delta III and IV flights, and 8 Saturn S4B flights.

Alternative zero-g vent systems would rely on mechanical mixers to distribute the point cooling during venting. The mixer must ensure complete tank mixing; otherwise localized hot spots will develop resulting in potentially uncontrollable tank pressure.

Similarly, settling allows venting during propellant transfer to maintain pressure in the receiver tank at desired levels. With extremely low acceleration, propellant entering the receiver tank may geyser. To prevent liquid venting, the propellant transfer process may need to be accomplished in pulse mode, where propellant transfer and venting are conducted sequentially.

Table 2. Settled cryogenic propellant transfer can benefit from the vast CFM experience used on Centaur and other cryogenic upper stages.

Cryo Transfer Technology	TRL	
	0-G	Settled
Pressure Control	4	9
Ullage & Liquid Stratification	3	9
Propellant acquisition	3	9
Mass Gauging	3	9
Propellant Expulsion Efficiency	3	8
System Chilldown	8	8
AR&D	7	7
Transfer System Operation	3	6
Fluid Coupling	6	6
Passive Long Duration Storage	5	5

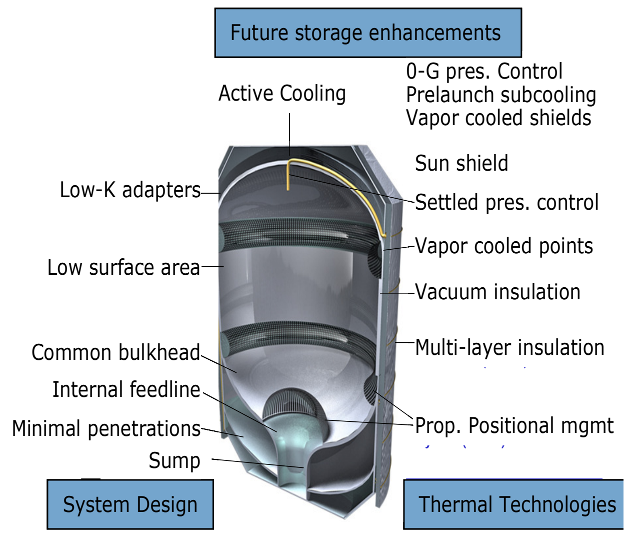


Figure 10. Effective system design combined with key thermal mitigation elements enables passive long duration cryogenic propellant storage.

Table 3. Centaur and Delta's upper stage have conducted numerous CFM flight demonstrations relevant to cryogenic propellant transfer.

Liquid Control (10^{-5} to 6 G's)	Long Coast (to 17 hours)
System Warming & Chilldown	Pressurization Sequencing
Propellant acquisition	Slosh characterization
System Thermal Interaction	Vent Sequencing
Ullage & Liquid Stratification	Pressure Collapse
Propellant Utilization	Bubbler vs. Ullage Pressn.
Mass Gauging	Unbalanced Venting

D. Propellant Acquisition

Propellant acquisition through settling has been used reliably for all large scale cryogenic upper stages. Expulsion efficiencies well in excess of 99.5% of liquids are achieved on Centaur, even at the relatively low accelerations encountered during pre-start and blowdown. Expulsion efficiency at 10^{-5} g is yet to be demonstrated.

With settled operations, expulsion efficiency is further increased by the ability to maintain a warm ullage. Settling effectively separates the liquid and gas in a tank enabling the ullage to remain warm during the expulsion process. By allowing the ullage to remain warm, there is the potential to increase total expulsion efficiency by ~0.9%, Figure 13

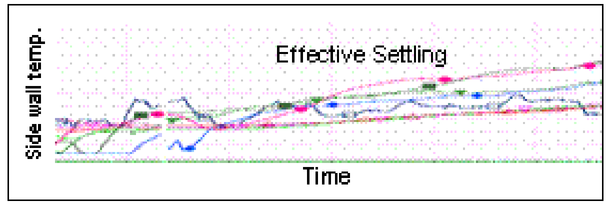


Figure 11. Centaur has demonstrated effective propellant control at 10^{-5} g's, well below the acceleration required to make settled propellant transfer attractive.

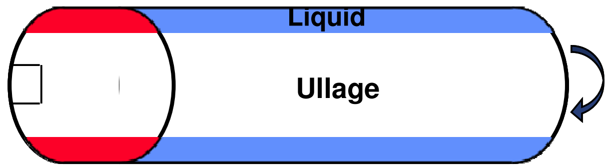


Figure 12. Centrifugal acceleration can separate the liquid and gas allowing use of existing, flight proven settled cryo-fluid management techniques.

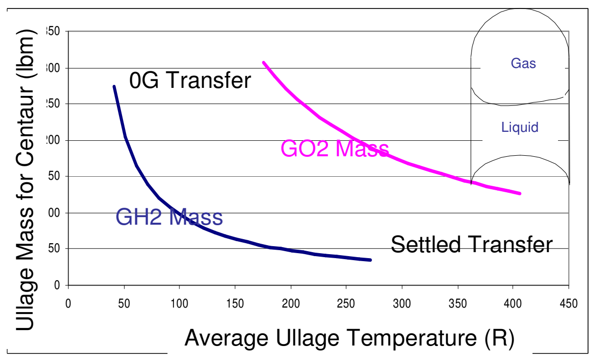


Figure 13. Low acceleration effectively separates the ullage and liquid enabling pure gas venting while reducing the gaseous residuals.

F. Mass Gauging

With settling, mass gauging is accomplished using numerous accurate and reliable techniques. Measuring the acceleration achieved with a known settling thrust provides a simple method that accurately gauges total system mass. Thermal couples and liquid sensors internal to the tank, or mounted to the outside of a thin walled tank, have proven very effective in defining the station level of the liquid/gas interface, Figure 14. The cryo tracker[17] concept promises a simple robust system for accurate liquid surface gauging at low acceleration. At higher accelerations resulting from a burn, tank head pressure has proven to be very effective at measuring liquid mass, ensuring >99.9% relative LO2/LH2 propellant expulsion efficiency for Centaur[18]. All the above methods (other than the cryo tracker) have been successfully used on the Centaur.

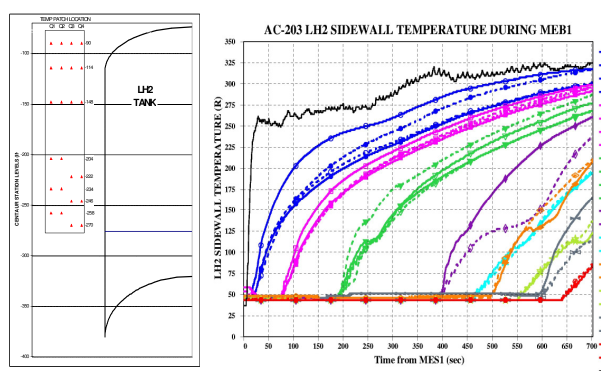


Figure 14. Centaur externally mounted thermal couples effectively measure liquid level.

E. System Chilldown

The Centaur upper stage has demonstrated highly efficient hardware chilldown procedures that are directly applicable to cryogenic transfer. Chilldown of ducting, tank walls and the engine have been demonstrated with multiple alternate chilldown procedures. Chilldown effectiveness using full, trickle, and pulse LH2 & LO2 flow has been demonstrated in the low g space environment. The pulse chilldown methodology has proven especially effective at chilling down the feed lines and the engine.

K. Autonomous Rendezvous and Docking

Russia has been performing autonomous rendezvous and docking (AR&D) for years in support of Salyut, MIR and ISS. Most recently, with the 2.5 year shuttle hiatus resulting from the destruction of Columbia, NASA relied on the Russian Progress vehicle and its AR&D capability for all of the ISS supplies. While development of AR&D has languished in the US, several recent efforts have demonstrated the viability of US-designed AR&D systems. The Dart, XSS-11, and Orbital Express[19] missions were all designed to further this capability. Dart was the first attempt to demonstrate American autonomous rendezvous technologies. Unfortunately errors in the GPS supported guidance algorithms led to excessive propellant consumption and an unplanned "bumping" of the target spacecraft. Incidents such as this provide important lessons and lead to improved capabilities. XSS-11, launched in early 2005, has successfully demonstrated numerous autonomous rendezvous and proximity operations during its year long mission. Orbital Express, launched in March of 2007, demonstrated AR&D as well as orbital servicing, including the transfer of N2H4 and He. It is vitally important that America continue on this path of AR&D development. The Orion vehicle, along with the two commercial orbital transportation services (COTS) program winners (SpaceX and Orbital Sciences Corporation) are also planning to use AR&D for ISS operations.

Figure 15. ULA is developing a pneumatically deployed sun shield to support long duration cryo storage.

J. Passive Long Duration Cryogenic Storage
A recent study[20,21] on the Centaur indicates how robust passive long term LO2/LH2 storage can be accomplished, figure 10. The study shows that efficient passive cryogenic storage for periods up to a year is feasible with proper system design coupled with key thermal isolation technologies. One of these key thermal isolation technologies is a sun shield that reflects the majority of the external radiation environment away from the cold cryogenic system. An open cavity sun shield further minimizes tank heating by allowing some of the transmitted energy to radiate to the cold of deep space. ULA, NASA KSC and ILC-Dover are currently developing a pneumatically deployed conic sun shield that is extremely light weight, can be packaged in minimal space while being scaleable to protect all sizes of cryogenic systems, Figure 15[22]. A cryogenic system is ideally coupled with a pneumatically deployed sun shield where the low boil-off provides the pressurant gas.

K. System Demonstration
Key to enabling programs such as Exploration to include propellant depots and cryo transfer as part of their baseline is end to end cryogenic storage and transfer demonstration in the actual, micro acceleration environment of space. In support of NASA GRC the Atlas program developed a low cost, ride share flight demonstration concept that can demonstrate all aspects of cryo-transfer and CFM technologies at a relevant scale[23]. This Centaur Test Bed concept would modify Centaur to allow transfer of residual propellant into a multi-cubic foot receiver vessel following deployment of the primary satellite. The Centaur Test Bed would enable demonstration of actual propellant transfer, low acceleration fluid acquisition and control, pressure control, thermal containment, mass gauging and fluid mixing.

VI. Open Architecture

A robust propellant depot infrastructure will benefit all aspects of space utilization. Interplanetary science missions will no longer be limited by the launch vehicle performance. National security missions will realize more flexibility in attaining their final orbit and on-orbit maneuvering. Commercial missions will be able to utilize smaller, less costly launch vehicles. Lunar crewed exploration will benefit from robust performance margins while installing the infrastructure to venture to Mars and beyond. Providing propellant to the depots will support a robust, competitive launch market, reducing costs for all aspects of space utilization.

The recent two rounds of COTS competitions demonstrates the huge pent up desire by numerous companies to provide commercial full service space access[24, 25, 26]. Although COTS consists of only a capability demonstration, the promise of a ~15 mT/year ISS servicing market was sufficient to encourage numerous companies to commit to investing hundreds of millions of dollars of private sector money:

- SpaceX
- Boeing
- SpaceDev
- Loral
- Rocketplane Kistler
- Planet Space
- t/Space
- Orbital Sciences
- Spacehab
- Constellation Services

Even the use of propellant depots to only supply LO2 for the lunar missions will result in an annual market requiring 100 to 200 mT, dwarfing the ISS requirements. Such a large market is expected to stimulate much fiercer launch competition, resulting in significant advances in methods of space access, resulting in improved reliability and reduced costs[27]. Some companies are likely to propose very frequent launches of small, potentially reusable launch vehicles, while others may view fewer, much larger launchers as the most cost effective solution. Only time, trial, and competition will decide the success or failure of individual concepts, but NASA, Exploration, and the space utilization market will be assured of continuous, sustained improvement in space access.

ULA is considering multiple options to supply propellant depots, including:

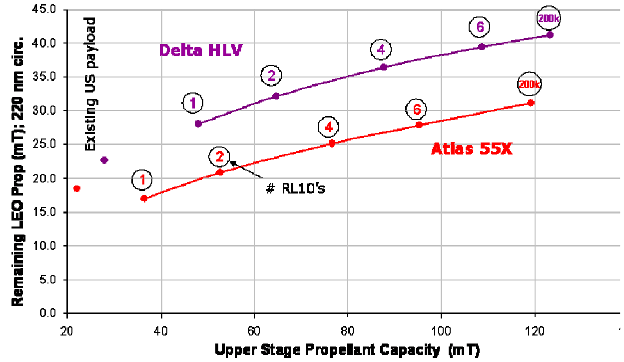


Figure 16. Use of an oversized upper stage, potentially derived from the ACES currently in development, provides significantly enhanced propellant delivery in the existing Atlas and Delta boosters.

1. **Delivery of a fueled propellant transfer vehicle to close proximity of the depot.** Both the Atlas and Delta vehicles can support delivery to orbit of fueled transfer vehicles. Once on-orbit these transfer vehicles would separate from the launch vehicles, autonomously rendezvous with the depot (similar to Progress or ATV), transfer propellants to the depot and then safely deorbit.
2. **Upgrading our upper stages to enable rendezvous and delivery of a fueled propellant tank.** Past studies have shown that with reasonable enhancements to the avionics and RCS systems both the Centaur and the Delta IV upper stages can support orbital rendezvous[20], avoiding the cost of an independent transfer vehicle.
3. **Enlarging ULA's upper stage propellant tanks to store additional LO2 or LH2 for delivery to the depot.** The most cost and mass efficient manner to store cryogens during launch is in the primary propellant tanks, avoiding the cost and mass of a dedicated cryo tank. Lengthening either the LO2 or LH2 tank to support the additional propellant is straight forward and has been done numerous times over the years to support increasing mission requirements
4. **Development of an evolved upper stage with increased thrust and oversized propellant tanks to support the propellant to be delivered.** ULA is currently investigating developing the Advanced Common Evolved Stage (ACES) driven by ULA's desire to realize cost saving while providing enhanced support to our broad customer community. ACES is being designed to allow increased thrust around a modular tank volume. A high thrust ACES stage would nearly double the delivered propellant capability of the existing Atlas and Delta boosters, Figure 16, at no additional cost, resulting in a very cost effective, robust depot servicing system.

VII. Conclusion

The ability to refuel propulsion stages in orbit offers huge benefits to the entire space user community, including science, national security and commercial enterprises. The vast orbital propellant needs of Exploration potentially allow for the most pronounced benefit from orbital refueling. The concentration of the majority of this Exploration propellant in the form of LO2 in the EDS makes it relatively easy for Exploration to take advantage of in-space refueling. Indeed, the orbital fueling of the EDS with 40 mT of LO2 would provide NASA with an attractive alternative to the substantial Ares V upgrades that NASA is currently considering. The current Exploration architecture can readily take advantage of an additional 40 mT of orbital LO2 transfer, supporting an additional 10 mT of lunar delivered payload.

This paper has shown how existing and near term technologies can be used to develop light weight, affordable propellant depots that can be cost effectively launched on single EELV medium class rockets. The proposed depot architecture utilizes an efficient design, coupled with key thermal management technologies (sun shield, settled fluid management and vapor cooling) to enable passive, extended storage of LO2 or even LH2. A proof of concept depot could be flying by 2011, early enough to demonstrate end to end system functionality in support of key Ares V and Altair development decisions.

Propellant delivery to the depot could be by any and all American launch entrants. Indeed, this architecture offers a convenient opportunity for international participation, potentially allowing for more frequent Exploration missions. The propellant could be delivered in any convenient individual quantity; a ton at a time, launched frequently on small low cost launchers, or 25 mT's at a time on EELV class launchers or even in huge 100 mT chunks on Ares V class rockets. Ultimately the realities of the launch business will define the cheapest, most reliable operational concepts, overcoming the current paper analysis debate regarding the best launch vehicle that has plagued the industry for decades.

A significant benefit associated with NASA's use of commercial launch services is NASA's potential to significantly reduce the cost of Exploration. This savings in turn would allow NASA to start the lunar exploration well before the current baseline of 2020. This savings would also allow NASA to fund other high priority elements, such as science and technology development. An added benefit of commercial launch services is that NASA would not be locked into a single launch solution as its needs and priorities change. For Exploration a major benefit of relying on-orbital fuel transfer is the flexibility to support evolving mission needs such as weight growth or Mars exploration without wholesale revamping of the Earth to orbit launch system.

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