Simple, Robust Cryogenic Propellant Depot for Near Term Applications

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Abstract[1]— The ability to refuel cryogenic propulsion stages on-orbit provides an innovative paradigm shift for space transportation supporting National Aeronautics and Space Administration's (NASA) Exploration program as well as deep space robotic, national security and commercial missions. Refueling enables large beyond low Earth orbit (LEO) missions without requiring super heavy lift vehicles that must continuously grow to support increasing mission demands as America's exploration transitions from early Lagrange point missions to near Earth objects (NEO), the lunar surface and eventually Mars. Earth-to-orbit launch can be optimized to provide competitive, cost-effective solutions that allow sustained exploration.

This paper describes an experimental platform developed to demonstrate the major technologies required for fuel depot technology. This test bed is capable of transferring residual liquid hydrogen (LH_2) or liquid oxygen (LO_2) from a Centaur upper stage, and storage in a secondary tank for up to one year on-orbit. A dedicated, flight heritage spacecraft bus is attached to an Evolved Expendable Launch Vehicle (EELV) Secondary Payload Adapter (ESPA) ring supporting experiments and data collection. This platform can be deployed as early as Q1 2013.

The propellant depot design described in this paper can be deployed affordably this decade supporting missions to Earth-Moon Lagrange points and lunar fly by. The same depot concept can be scaled up to support more demanding missions and launch capabilities. The enabling depot design features, technologies and concept of operations are described.

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ACRONYMS

ACES	Advanced Common Evolved Stage
AFRL	Air Force Research Laboratory
AR&D	Autonomous Rendezvous and Docking
ARC	Ames Research Center
ATV	Automated Transfer Vehicle
BAC	Broad Area Cooling
BCP	Ball Commercial Platform
BEO	Beyond Earth Orbit
CDR	Critical Design Review
CFM	Cryogenic Fluid Management
COTS	Commercial Orbital Transportation Services
CryoSTAT	Cryogenic Storage and Transfer
CRYOTE	Cryogenic Orbital Testbed
DARPA	Defense Advanced Research Projects Agency
DCSS	Delta Cryogenic Second Stage
DMSP	Defense Meteorological Satellite Program
DOD	Department of Defense
EDS	Earth Departure Stage
EELV	Evolved Expendable Launch Vehicle
ESPA	EELV Secondary Payload Adapter
FTD	Flagship Technology Demonstration
FTINU	Fault Tolerant Inertial Navigation Unit
GH2	Gaseous Hydrogen
GMD	Global Missile Defense
GO2	Gaseous Oxygen
GPS	Global Positioning System
GRC	Glenn Research Center
GSFC	Goddard Space Flight Center
GSO	Geosynchronous Orbit
GTO	Geostationary Transfer Orbit
HLV	Heavy Lift Vehicle
HTV	HII Transfer Vehicle
IMLEO	Initial Mass Low Earth Orbit
IMLI	Integrated Multi Layer Insulation
I_{sp}	Specific Impulse
ISCPD	In-Space Cryogenic Propellant Depot
ISS	International Space Station
IVF	Integrated Vehicle Fluids
JSC	Johnson Space Center
JT	Joule Thompson
JWST	James Webb Space Telescope
KSC	Kennedy Space Center
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LIDAR	Light Detection and Ranging
LN_2	Liquid Nitrogen
LO_2	Liquid Oxygen
MEO	Medium Earth Orbit
MHTB	Multi-Purpose Hydrogen Test Bed
MLI	Multi Layer Insulation
MMOD	Micro-Meteoroid Orbital Debris
MR	Mixture Ratio
MRS	Minimum Residual Shutdown
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
NEO	Near Earth Object
NRO	National Reconnaissance Office
O&C	Operation and Checkout
OSP	Orbital Space Plane
PLF	Payload Fairing
PMD	Propellant Management Device
PRSD	Power Reactant Storage Distribution
RCS	Reaction Control System
SAS	Special Aerospace Services
SBHE	Start Box High End
SOFI	Spray-on Foam Insulation
STP-SIV	Space Test Program-Standard Interface Vehicle
STS	Space Transportation System
TCS	Thermodynamic Cryogen Subcooler
TPS	Thermal Protection System
TRL	Technology Readiness Technology
TVS	Thermodynamic Vent System
ULA	United Launch Alliance
VJ	Vacuum Jacket
WISE	Wide-Field Infrared Survey Explorer

1. INTRODUCTION

Today's space transportation revolves around getting to low Earth orbit (LEO) and beyond. The world's existing large launch vehicles can loft payloads of about 25 mT to LEO, 12 mT to geostationary transfer orbit (GTO) or 9 mT to Earth escape, Table 1. What happens if a mission requires more performance? Moderate performance increases are possible by enhancing existing launch vehicles such as the Delta IV's heavy upgrade program. More pronounced performance increases are possible by enhancing stages for existing launch vehicles such as the Ariane ECB. For substantially larger performance levels one may need to develop entirely new launch vehicles.

For large LEO payloads, mission designers have little flexibility. Either there must be a launch vehicle sufficiently large to loft the payload, or the payload must be split into manageable elements and assembled on-orbit. For the massive 500 mT International Space Station (ISS) NASA chose to assemble the station in orbit with no single element weighing more than ~21 mT, limited by the Space Shuttle's launch capability. For the Constellation programs lunar exploration initiative, needing ~160 mT in LEO, roughly six times the capability of any existing launch vehicle, NASA chose to combine development of a new Saturn V class heavy lift rocket, Ares V, and on-orbit "assembly". The Earth departure stage (EDS) and the lunar lander (Altair) would be launched on Ares V, while the crew would be launched separately in Orion. Orion would rendezvous and mate ("assembly") on-orbit with the EDS and Altair for the trip to the moon.

For demanding missions beyond LEO missions designers have several alternatives to developing larger and larger rockets for each new mission. Once on-orbit, exhaust velocity (I_{sp}) dictates mission performance, so mission planners can choose to use more efficient modes of propulsion such as nuclear thermal, solar thermal, electric ion, etc. These efficient forms of propulsion may bring total system mass within the capability of available launchers or allow a common heavy lift rocket to support ever growing mission requirements. Another option for mission designers is to fuel the propulsion systems on-orbit, basically a form of "simplified" on-orbit assembly. For most missions beyond LEO, the required LEO mass consists primarily of propellant. For example, a mission from LEO to GTO consumes 42% of its initial mass in LEO (IMLEO) as propellant. Higher energy missions consist of even greater propellant mass fractions, Table 2. Thus, as long as launch vehicles have sufficient lift and payload volume to launch the dry spacecraft to LEO, on-orbit fueling can allow very demanding missions, while the launch vehicle can be chosen based on cost, schedule and other critical figures of merit.

America's nascent space exploration initiative is ideally suited to take advantage of on-orbit fueling. These missions require huge amounts of cryogenic propellant and no individual element must be particularly large or heavy. NASA's renewed emphasis on technology development may allow near-term deployment and use of propellant depots.

Table 1. Modern existing large launch vehicles have similar performance.

System	Performance (mT)		
	LEO	GTO	Earth Escape
Delta HLV	28.5	14.2	10.7
Atlas 552[1]	21.1	8.7	6.5
Ariane 5[2]	19.3	10.1	-
Proton	21.0	4.0	-
Sea Launch	14.0	6.0	-
HIIB	10.0	5.8	-
Long March	11.2	5.1	-
Space Shuttle	18-24.4	5.9	-

Table 2. Required propellant mass as a percentage of LEO system mass for various typical missions assuming state-of-the-art LO_2/LH_2 propulsion.

Final Orbit	% propellant mass
GTO	42%
Trans Lunar Inject (C3=-1)	50%
Trans Mars Inject (C3=21)	60%
GSO	61%
Lunar Surface	75%

This paper describes a simple, near-term propellant depot concept that can be fully assembled and tested on the ground followed by launch on a single rocket as an integrated assembly. Also discussed is a low-cost precursor mission platform to demonstrate multiple cryogenic propellant transfer and storage technologies relevant to the simple propellant depot as well as NASA's CryoSTAT mission. The simple propellant depot concept can be scaled from a capacity of 30 mT using existing rockets and manufacturing tooling to 200 mT on enhanced existing rockets to hundreds of tons of capacity flown on proposed heavy lift rockets with larger diameter payload fairings. The technology required to enable efficient cryogenic propellant storage and the processes to control the state of these fluids is also discussed. Both proposed programs provide a method of maturing the technologies and reducing risk for Flagship Technology Demonstration (FTD) class missions as proposed by the NASA ESMD in May of 2010[3] and Crosscutting Capability Demonstrations Technology Demonstration Missions (TDM) proposed by NASA Office of Chief Technologist (OCT) in May 2010[4].

2. DEMONSTRATED STATE-OF-THE-ART IN CRYOGENIC SPACE SYSTEMS

Presently, applications using LH_2 and LO_2 as propellants have been relegated to launch vehicles such as the space shuttle external tank, EELV Centaur upper stage, and the

Delta cryogenic second stage (DCSS), all of which use their cryogens relatively rapidly from earth to orbit and subsequent on-orbit payload delivery (typically <24 hours). The thermal insulation systems used by these launch vehicles and cryogenic upper stages include spray-on foam insulation (SOFI) and thin, limited-layer multilayer insulation (MLI). These systems have been engineered and optimized to manage pre-launch tanking, pad hold, and ascent environments that mitigate liquid air condensation on tank walls, etc. As a consequence, these insulation systems have not been engineered to provide optimum thermal management of cryogens while on-orbit, and in fact, SOFI is virtually useless as a cryogenic insulation system in space (dead weight for long duration or lander missions).

Longer duration storage and utilization of hydrogen and oxygen cryogenic fluids are routinely demonstrated with every space shuttle flight via the power reactant storage and distribution (PRSD) tanks. These tanks store supercritical cryogenic hydrogen and oxygen, the respective gasses from which are provided to the space shuttle's fuel cells for electrical power generation as well as breathing oxygen and drinking water for the crew. The PRSD tanks have limited heat leaks relative to the boil-off rates required to drive reactants from the tanks to meet fuel cell power demands. As such, in-tank heaters are power modulated to achieve required fuel cell reactant flow rates. Here too, the vacuum jacketed (VJ) shell only provides useful insulation during pre-launch filling, pad hold and ascent but no, or limited, benefit while on-orbit. Other than additional Micro-Meteoroid Orbital Debris (MMOD) barrier protection, VJs represent a mass penalty to any space system once on-orbit.

Significantly longer passive storage durations of cryogens have been demonstrated on-orbit without the need to use cryocoolers. Recent examples of this include both the Spitzer and Wide-Field Infrared Survey Explorer (WISE) cryogenically cooled space telescopes cooled with superfluid helium and solid hydrogen cryostat dewars respectively. Both of these missions represent state-of-the-art implementations of cryogenic thermal management technology without the need for active cryocooling. Here Spitzer demonstrated an equivalent average cryogen boil-off rate of 0.05% per day achieving an operational mission duration total of over 5 years (66 months on-orbit). This was accomplished at cryogen temperatures approaching 1.5 Kelvin – 19 to 23 degrees colder than that required to manage liquid hydrogen propellants (depending on tank pressure). As with the previous launch vehicle upper stage and PRSD examples, these systems too suffered from the need to launch with a VJ outer vessel providing limited thermal benefit to their stored cryogens once on-orbit. These flight heritage cryogenic thermal management technologies and new emerging technologies and systems will enable a paradigm shift of how stored cryogens can be used to help transform a new Flexible Path for beyond Earth orbit (BEO) exploration initiative.

3. Exploration Missions and Systems Benefitting From Cryogenic Technologies

In general, all large-scale robotic and crewed missions beyond LEO will benefit from enhanced Cryogenic Fluid Management (CFM) and optimum cryogenic storage of high efficiency LH_2/LO_2 propellants for propulsion. Cryogenically fueled upper stages such as the Constellation EDS will provide the necessary Delta-V to achieve Earth departure trajectories for missions bound to Lagrange points, lunar, NEO and Mars destinations. Additionally cryogenic propellants will provide Delta-V to support orbital capture and allow large down-mass/cargo access into deep gravity wells such as lunar surface missions and Mars. Several of these missions will expend their cryogens rapidly and won't require heroic implementations of low boil-off cryo-thermal management technologies (e.g., lunar surface access and Lagrange missions). Other deep space missions such as human NEO or Mars missions will benefit greatly from demonstrated long duration maintenance of cryogenic propellants at near zero boil-off. Specifically, NASA has identified technology investments to improve cryogenic boil-off as having the single largest impact on normalized mass savings for a human Mars missions, Figure 1[5].

The successful demonstration and adoption of fuel depots in LEO and at Lagrangian points will significantly impact the economies of beyond LEO missions. Depots will enable lower mass Earth departure stages to be propellant-sized for their final destination without concern for using half or more of their propellants to achieve circular LEO orbit prior to departure. Additionally, fuel depots will enable space resident transfer vehicles or tugs capable of cycling cargo and fuel between LEO resident depots and those in GEO or Lagrangian points. Here the savings in terms of dollars per pound will be measured by factors of 2 or greater as compared to missions without a fuel depot infrastructure. The importance to NASA of demonstrating large scale CFM with cryogenic propellants on-orbit is further exhibited with the recent release of the NASA OCT Drafts "Thermal Management Systems Roadmap Technology Area 14" and

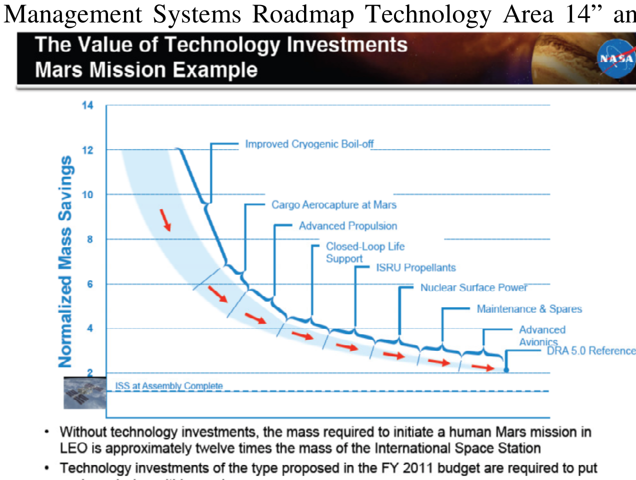


Figure 1 – High priority of improved cryogenic propellant storage recognized by NASA.

"In-space Propulsion Systems Roadmap Technology Area 2" in November, 2011.[6]

Several capability tenets of CFM must be demonstrated and vetted on-orbit prior to a universal implementation and acceptance of long duration use and handling of stored cryogens as mission enabling commodities in space. When implemented with acceptable impacts to exploration space systems resource, these include: 1) microgravity mass gauging of stored cryogenic propellants; 2) high fill efficiency tank-to-tank propellant transfer in low gravity; 3) long duration low boil-off of stored cryogens; and 4) controlled acquisition and flow of cryogenic propellants in low gravity environments.

4. CRYOTE[7,8,9]

The CRYogenic Orbital TEst (CRYOTE) concept, Figure 2, uses a rideshare implementation allowing demonstration of cryogenic propellant transfer, long duration storage and a host of CFM technologies while minimizing the impact to the launch vehicle. CRYOTE rides inside the Atlas V payload adapter. Following payload delivery, residual LH_2 (or LO_2) is transferred from Centaur into the CRYOTE tank.

Rideshare Orbital Flights

The act of transferring residual LH_2 from Centaur requires chilldown of the transfer lines and storage tank as well as demonstrating the ability to fill a receiving vessel. Inside CRYOTE's tank, a multitude of CFM technologies can be demonstrated including propellant management, long-term storage, mass gauging and liquid acquisition. CRYOTE offers an end-to-end, sub-scale demonstration of potentially all propellant depot CFM technologies.

To date the CRYOTE project has involved nearly every NASA center, academia, and several commercial companies. The CRYOTE effort has developed a ground test prototype that will be vacuum tested in early 2011 with LN_2. CRYOTE development is continuing with flight vehicle design and launch vehicle integration on the GeoEye-2 mission for flight in early 2013. CRYOTE can provide critical near-term risk reduction for NASA's CryoSTAT mission, allowing more demanding mission objectives such as supporting an Orion Lagrange mission as early as 2016.

Figure 2 – CRYOTE offers a near term, low cost method to demonstrate in-space propellant depot CFM technologies.

The CRYOTE platform has three levels of implementation: CRYOTE Lite, CRYOTE Pup, and CRYOTE Free Flyer, Figure 3. Both CRYOTE Lite and CRYOTE Pup remain attached to a spent Centaur, while the CRYOTE Free Flyer separates from the Centaur and becomes an independent cryogenic propellant laboratory.

CRYOTE Lite

The CRYOTE missions raise technology readiness levels (TRL) and provide on-orbit demonstrations of critical cryogenic propellant storage and delivery technologies to support the NASA flagship CryoSTAT mission. CRYOTE Lite's 17 hours operational life is limited to battery power available from the Centaur. Avionics to perform experimental control, data collection and data handling is provided by DesignNet of Golden, CO.

Even with the limited lifetime of this concept, significant data on the processes associated with zero-g cryogenic propellant transfer, validation of thermodynamic vent system (TVS) operation, demonstration of fluid phase control with the propellant management device (PMD), and zero-g mass gauging sensors can be generated. On-orbit thermal models for the transient cool down and filling process can be validated.

CRYOTE Pup

Significantly longer life can be demonstrated with the CRYOTE Pup mission concept. Power, communication, attitude control, data acquisition and experimental control are all provided by a Ball Aerospace Space Test Program-Standard Interface Vehicle (STP-SIV) derived spacecraft bus. This bus enables the CRYOTE mission to be extended for over a year with support for multiple experiments. Attitude control would be provided by either hydrazine or cold gas thrusters.

Compared to CRYOTE Lite, this extended lifetime increases the scope of experiments that can be demonstrated. The longer mission duration allows for system steady state thermal equilibrium to be achieved, which in turn provides higher fidelity measurements of the thermal performance of all system components. This additional power, and significantly enhanced experimental data collection and control, enables the demonstration of increasingly complex experiments, such as the use of cryocoolers, in-tank video measurements, the addition of heaters to augment cold gas thruster performance, and other mission-specific experiments. Also, the improvements in overall system thermal performance of various active and passive cooling can be better characterized when multiple equilibrium conditions are achieved.

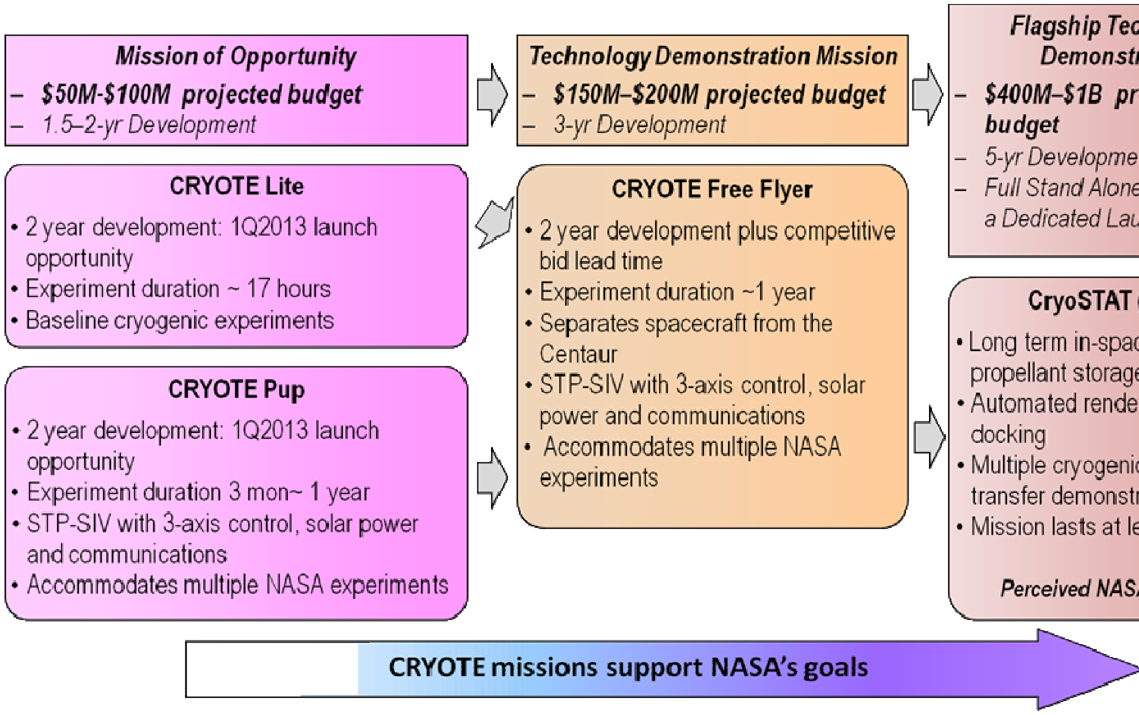


Figure 3 – CRYOTE missions support the demonstration of NASA Flagship missions.

CRYOTE Free Flyer

Both the CRYOTE Lite and Pup missions are performed with the experimental payload attached to the inerted Centaur. The CRYOTE Free Flyer mission separates the experiment from the Centaur and operates as a free-flying spacecraft for up to one year. The Free Flyer has all of the capability of the STP-SIV bus, and, if the mission occurs in LEO, allows for attitude control to be provided by momentum wheels and torque rods. This ability to provide precise or authoritative attitude control of the Free Flyer is the main advantage compared to the CRYOTE Pup mission. For example, forced rotational slosh of the propellants can occur to observe fluid decay properties and effectiveness of PMD's and slosh baffles to mitigate these transient phenomena.

Figure 4 shows the subsystem for all CRYOTE mission variants: the CRYOTE Core. CRYOTE Core employs a custom, 1,000 liter LH_2/LO_2 compatible tank. Solenoid latching valves are employed to facilitate propellant transfer and control of multiple experiments, which are under development by Ball Aerospace and United Launch Alliance (ULA). CRYOTE Core resides in an ESPA ring. Support of the tank is provided by low conductivity, Ball Aerospace flight heritage cryogenic struts, mounted in a hexapod arrangement. For initial missions, standard Ball Aerospace high performance cryogenic MLI, integrated multilayer insulation (IMLI), or a combination of the two will be used to provide tank acreage and penetration insulation. The CRYOTE Core has been designed to directly represent likely configurations of future long duration cryogenic propellant systems, including the mechanical interfaces and geometric shape of the tank. Note that the CRYOTE Core is not penalized with a VJ as the tank is launched warm and empty.

The tank design is based on Ball Aerospace heritage design and manufacturing processes. The tank can accommodate multiple internal experiments, and has external mounting provisions for experimental hardware. Provisions to the tank design have been included for the incorporation of multiple PMD technologies as well as zero-g mass gauging sensors such as NASA Glenn Research Center (GRC) radio frequency (RF)[10,11] technologies and the Sierra Lobo Cryo-Tracker®[12,13,14].

Thermodynamic Vent System (TVS)

A TVS can be used to provide significant cooling to a stored cryogenic fluid. Expansion and pressure drop of liquid (or two-phase fluid) propellant across a Joule Thompson (JT) orifice provides significant cooling capacity the can be used

Figure 4 – CRYOTE Core provides a flexible testbed for short and long term on-orbit cryogenic propellant storage and transfer experiments.

to intercept system heat leaks, or cool the bulk gaseous or liquid stored in the propellant tank. The heat exchange process with the stored fluid can take place within the tank utilizing an internal heat exchanger within the tank, or by using fluid lines coupled to the tank wall.[15] The highest heat transfer efficiency occurs if the TVS line is in contact with the fluid rather than the gas. For the CRYOTE Core, the external TVS heat exchanger line is attached circumferentially to the tank at four places. The CRYOTE PMD guarantees that liquid will always be introduced to the JT orifice, maximizing the available cooling.

Struts

Low conductivity, cryogenic strut technology developed and flown by Ball Aerospace has been tailored for CRYOTE Core. The hexapod arrangement is also based on flight heritage designs, and was examined in detail in the cryogenic propellant storage and delivery contract performed by Ball Aerospace for NASA GRC.[16] The CRYOTE Core platform is very flexible and can allow for the evaluation of multiple strut technologies including disconnect after launch approaches, as well as the use of composite conic sections such as those employed in the CRYOTE ground test article. There is also the opportunity to employ multiple strut technologies on a single flight experiment, assuming the aggregate design meets the launch induced environmental requirements.

Propellant Management Device (PMD)[17,18]

One assumed requirement for all foreseen CRYOTE missions was the implementation of a flight PMD. The cryogenic PMD designed for CRYOTE Core provides access to gaseous cryogen propellant while maintaining wetted walls in a low-g condition. Access to gaseous propellants allows for tank pressure control with minimum loss of propellant. The wetted walls, by controlling the location of the liquid propellant, allows for guaranteed access to liquid for TVS operation as well as direct heat exchange with the propellant at the tank walls. The PMD designed to meet delivery of these two phases of propellants is shown in Figure 5.

If direct heat exchange to the stored cryogenic propellant is desired as part of the CRYOTE Core experiment, revisions to the PMD design can be made to provide an in-tank fluid interface as required, Figure 4. Additionally, a PMD has been designed for CRYOTE Core such that the fluid is retained in the center of the tank with a gas boundary over a significant portion of the tank walls. Specific mission requirements and/or experimental goals will dictate the implemented PMD design.

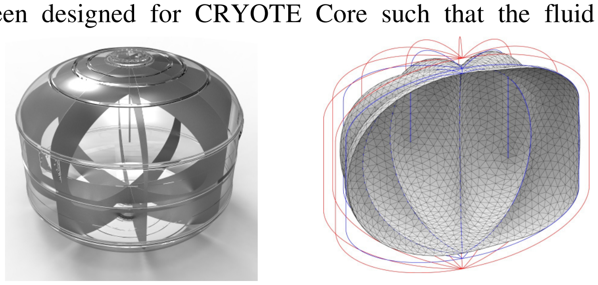


Figure 5 – CRYOTE Core cryogenic PMD guarantees gaseous propellant delivery and wetted walls.

Since PMD designs are such an integral part of the tank internal volume, it is crucial that the system level mission requirements for CRYOTE, CryoSTAT, or operational fuel depots are taken into consideration at the beginning of each program. The integration of a PMD, low-gravity propellant gauging, heat exchangers, mixing pumps, optical sensors, etc. will only result in a robust, mass, schedule and cost effective design if the internal tank components are treated as an integrated subsystem.

STP-SIV Bus Implementation

Both the CRYOTE Pup and Free Flyer mission concepts require a flexible, mature spacecraft bus that can be implemented on an ESPA ring, with customization. In addition, the CRYOTE Core design work finalized in Q4 2010 has a target rideshare launch date in Q1 2013. The Ball Aerospace STP-SIV bus was selected based on cost and schedule compliance, flight heritage, as well as the inherent flexibility of the design.

The STP-SIV platform, Figure 6, was developed by Ball Aerospace for the Air Force to provide a standard payload-to-spacecraft interface for all experiments. In order to provide the greatest flexibility for the end user, this bus is designed for a range of LEO orbits without changes, and includes a set of fixed and articulated solar arrays to operate at any Sun angle.

Mission-tailored MLI blankets are employed to provide proper radiator coverage for specific payload and mission-orbits. The basic STP-SIV bus is compatible with a variety of launch vehicles including ESPA, Figure 7, and is designed/tested to rigorous requirements (e.g., compliant to MIL-STD-1540e). Robust mounting, electrical, and data interfaces are employed to accommodate a wide variety of payloads which reduces non-recurring cost for repeat builds.

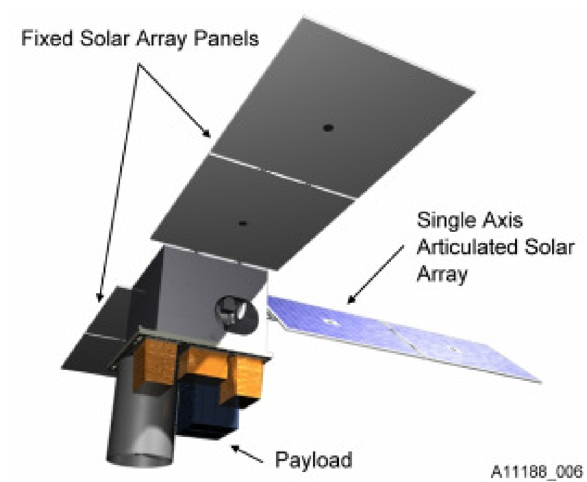


Figure 6 – STP-SIV bus with payload.

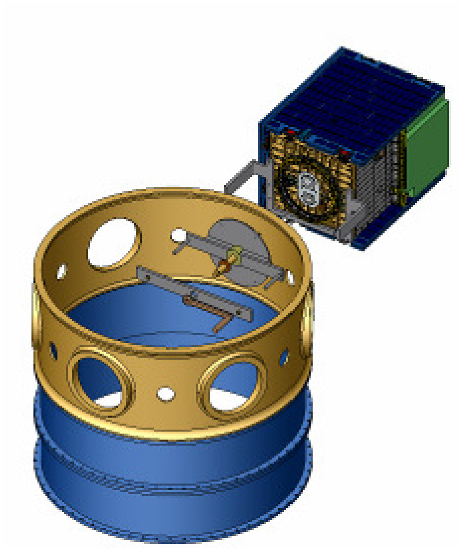


Figure 7 - STP-SIV as secondary payload mounted to ESPA ring.

Currently STP-SIV Sat-2 is on-orbit and operating nominally. The flexibility of this system was demonstrated throughout the program. A third payload was added after critical design review (CDR) without bus design changes. During assembly, three payloads were integrated in four days. Space vehicle integration and test was executed on-schedule to the day. Operationally, STP-SIV Sat-2 was Launched November 19, 2010 on a Minotaur-IV, with bus checkout completed November 27, 2010 and payload checkout completed December 3, 2010.

5. SIMPLE DEPOT[19]

Simple Depot – Overview

The following discussion describes a small, 30 mT capacity, Centaur derived depot, Figure 8, which can be launched by the middle of this decade. By refueling the DCSS upper stage following launch of Orion on Delta IV heavy lift vehicle (HLV) a 30 mT depot can support near-term missions of Orion to the Earth Moon Lagrange points or lunar fly-by missions, Figure 9.[20] The same depot concept lends itself to much larger capacity depots using larger diameter tanks, upper stages and payload fairings. These larger depots can enable missions to NEO, the Lunar surface and Mars.

Storing cryogens, including LH_2, is not particularly difficult as long as one follows a few key cryogenic design principles:

- Minimize penetrations
- Minimize surface area
- Segregate the LO_2, LH_2 and warm mission module
- Use historic cryogenic lessons learned to the greatest practical extent
- Enable full system ground check out

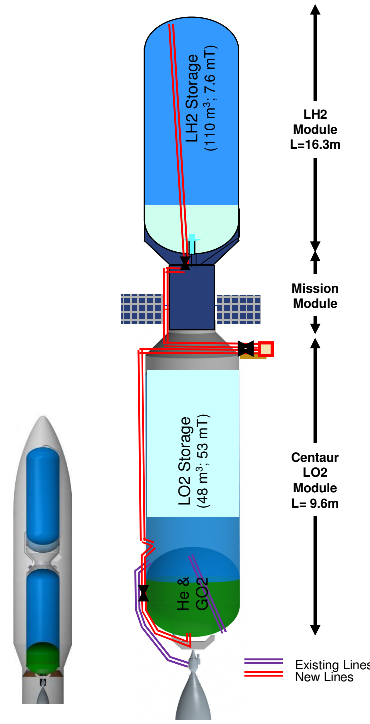


Figure 8 - An Atlas/Centaur derived depot demonstration mission can demonstrate all of NASA's objectives within an FTD $500M budget in less than 5 years.

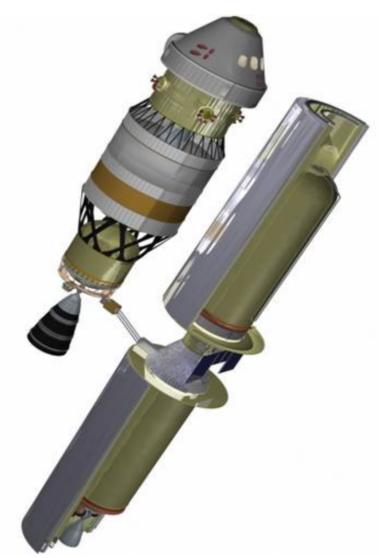


Figure 9 – Orion on a 5m DCSS being refueled prior to departure for L1.

Two additional basic tenants incorporated into the Simple Depot design intended to simplify development, reduce development costs and ensure mission success are:

- The depot should take advantage of existing CFM experience.

- The depot should be built using hardware that is common to the rest of space transportation.

The proposed Simple Depot concept satisfies all of these design principles. Its design employs settled propellant management and predominantly existing flight qualified hardware. The design consists of a large LH_2 tank connected by a warm mission module to the LO_2 tank. This depot concept can be launch on a single Atlas mission requiring no on-orbit assembly allowing for complete system ground check out.

Simple Depot – LH_2 Module

The Simple Depot LH_2 module is composed of a large tank with minimal penetrations. For the "small" 30 mT depot the LH_2 tank is a modified Centaur tank, built on the same tooling, using the same procedures as construction of the Centaur, Figure 10. The LH_2 module is launched with the LH_2 tank filled with ambient temperature helium, not LH_2. This allows the LH_2 and mission modules to be designed primarily for orbital requirements not ground and ascent environments. With these substantially reduced requirements the skin gauge can be reduced from today's 0.020" for Centaur's to 0.013". This is the same gauge as used on early Centaurs. This thinner tank wall allows the tank to be very light weight, (~500 kg). Made of corrosion resistant stainless steel, the thin tank walls reduce the conduction of energy to the liquid and results in a very low thermal mass that must be quenched when the tank is filled or when slosh waves splash warm walls.

All plumbing is connected to the mission module side ("top") of the LH_2 tank, Figure 11. This includes:

- Plumbing to enable liquid hydrogen to be transferred in and out of the tank
- Pneumatic tubes to pressurize the tank to support transferring LH_2 out of the tank
- Vent plumbing that allows tank pressure control as well as a pressure sense line to measure tank pressure. The vent and pressure control valves are closely coupled with the top of the LH_2 tank to allow the lines to be vented to vacuum, reducing heat conduction, when the lines are not in use.

Figure 10 - Centaur tank production.

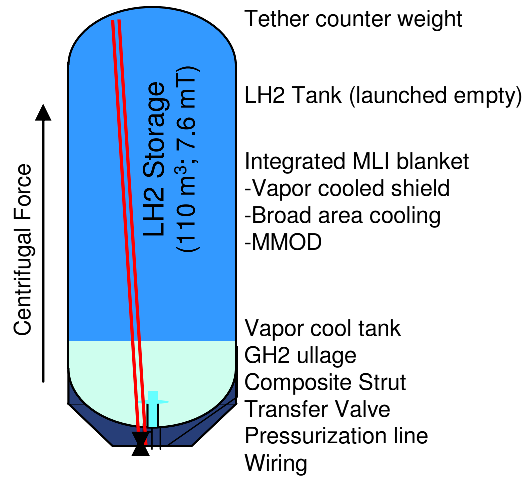


Figure 11 - LH2 module.

All of the instrumentation wiring also enters from the top of the tank. This instrumentation supports mapping the ullage and liquid temperature and temperature gradients in the thermal protection system. All of the plumbing and wiring is routed along a circuitous, spiral path that incorporates vapor cooling to minimize heat conduction.

The LH_2 tank is connected to the mission module by low-conductivity Ball Aerospace heritage cryogenic composite struts. Keeping the entire LH_2 module light weight minimizes the required cross section of these struts. This is critical to minimizing the structural heat transfer from the warm mission module to the very cold LH_2 module. The struts can be vapor cooled to further reduce conductive heat leakage into the LH_2 tank.

LH_2 Module Thermal Protection System

The entire LH_2 tank is encapsulated in a robust, Ball Aerospace IMLI blanket that incorporates radiation barriers, both vapor and active broad area cooling (BAC) as well as MMOD protection. With all the LH_2 tank penetrations concentrated on the front of the tank, the IMLI can approach its theoretical performance.

Since the LH_2 tank is launched warm and only filled on-orbit the tank is not covered with SOFI. Elimination of SOFI reduces weight and any potential for out-gassing that could degrade the IMLI performance.

The described LH_2 tank is 3 m in diameter by 16 m long limited by the existing Atlas payload fairing. The tank is 110 m^3 and can store 5 mT of LH_2. At a useful mixture ratio (MR) of 6:1 this quantity of LH_2 can be paired with 25.7 mT of LO_2, allowing for 0.7 mT of LH_2 to be used for vapor cooling, for a total useful propellant mass of 30 mT. Accounting for the tank weight, plumbing, instrumentation and thermal protection the LH_2 module is anticipated to weigh <2 mT. Based on analysis the described depot will have a boil-off rate of approaching 0.1% per day, consisting entirely of hydrogen.

Simple Depot – LO_2 Module

To conserve volume allowing for a useful sized depot to be fully integrated on the ground and emplaced on-orbit in a single launch, the LO_2 is stored in the upper stages propellant tank, Figure 12. This requires a thermally efficient upper stage that can be completely encapsulated with MLI.

America has three existing LO_2/LH_2 upper stages: the 4 m and 5 m DCSS and Centaur. The DCSS design encapsulates the LO_2 tank in the inter-stage allowing the tank to be wrapped in MLI. The equipment shelf, RL10 engine, feedlines and inter tank struts all attach directly to the tank, however, resulting in thermal shorts. While the DCSS LH_2 tank has fewer attachments it is exposed to atmosphere during ascent preventing application of standard MLI without development of an application-specific aero fairing.

Atlas V fully encapsulates the Centaur inside the 5.4 m payload fairing and is currently flown with either a single or a 4-layer MLI blanket. However, Centaur's LO_2 tank aft bulkhead serves as the equipment shelf with the RL10 engine, feedlines, helium bottles, hydrazine bottles, pneumatics panel and reaction control system loop mounted directly to the bulkhead. This results in substantial tank heating. Centaur's LH_2 tank however is very thermally efficient, especially if there is not a substantial thermal gradient across the common bulkhead.

For these reasons the proposed Simple Depot would be launched on an Atlas and use Centaur's LH_2 tank to store the LO_2. Centaur's LH_2 tank is also relatively large, with a volume of 47 m^3 capable of containing 54 mT of LO_2. At a useful MR of 6:1 this Centaur would optimally be paired with a 5 m diameter LH_2 module tank with over twice the volume possible from a tank derived from the 3 m diameter Centaur described earlier.

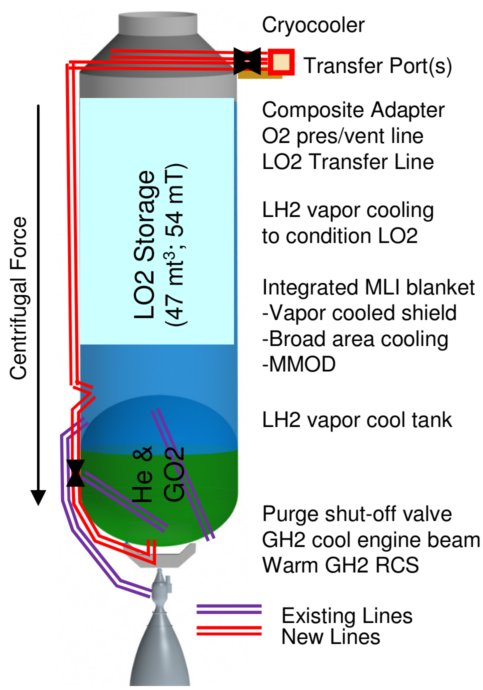


Figure 12 - LO2 module.

Several modifications to the Centaur are required to enable its use as the depots LO_2 module:

- Add valves to close existing Centaur purge lines.
- New plumbing to allow for transfer of propellants.
- Addition of vapor cooling tubing circumnavigating the Centaur to reduce heating and maintain the LO_2 at the desired temperature.
- Cold gas nozzles added to the aft end of Centaur to vent the spent GH2 which is used to maintain the depot control.
- New wiring to support all of the new valves and instrumentation.
- Encapsulation of the entire Centaur tank in a robust IMLI blanket to minimize incident heating and protect the tank from micro meteoroid damage.
- Replaced Centaur's metallic stub adapter with a composite cylinder.

Simple Depot – Mission Module

Between the Centaur LO_2 module and the LH_2 module resides the mission module, Figure 13. The mission module is the brain and control center of the depot. This module includes the flight computer, solar panels, batteries, fluid controls, avionics, remote berthing arm and docking and fluid transfer ports. The mission module would likely be derived from CRYOTE Pup or Free Flyer with a variant of a Ball Commercial Platform (BCP) bus.

Simple Depot - Launch

The combined Centaur, mission module and LH_2 module are integrated at Kennedy Space Center (KSC) prior to launch. Extensive ground checkout validates that all systems are operating nominally. This check out comprises ambient system checks as well as vacuum testing in the Operation and Checkout (O&C) vacuum chamber, Figure 14, to verify mechanical and electrical system performance.

While the Simple Depot is so light that it could be launched on an Atlas 501, it will be launched on an Atlas 551, the same rocket used to initiate Pluto New Horizon's journey,

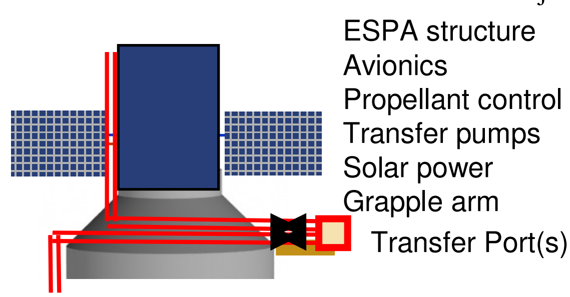


Figure 13 - Mission module.

Figure 14 – NASA KSC's O&C vacuum chamber provides the size required for system check out of the completely integrated depot prior to launch. Courtesy NASA

Figure 15. Launching the depot on an Atlas 551 provides ~12 mT of Centaur residuals (combined LH_2 and LO_2) in a 28.5° by 200 nm circular LEO, Figure 16.

Once safely delivered to orbit the LH_2 module must be chilled prior to transfer of Centaur residual LH_2. Centaur's cold hydrogen ullage gas is vented through the LH_2 mission module tank to chill the tank. This chilldown process has been demonstrated on past Centaur flights to chill the feedlines and RL10 pump housing. The thin, light weight Centaur derived LH_2 module tank minimizes the hydrogen consumed during this chilldown process. The chilldown is conducted under low acceleration to allow the transfer of vapor or liquid as desired. Centaur's existing settling thrusters are fired to provide the necessary axial acceleration.

Figure 15 - Atlas V 551 launch of the Pluto New Horizon spacecraft. Courtesy ULA

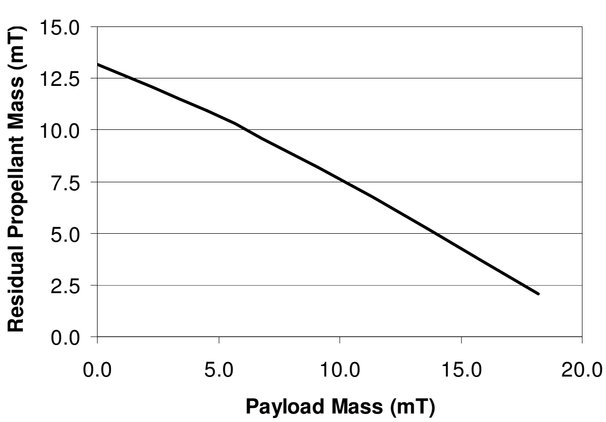


Figure 16 - Atlas 551 provides sufficient performance to allow substantial LO2/LH2 to be transferred on-orbit from Centaur's tanks. Courtesy ULA

Once the LH_2 module is chilled the transfer of Centaur's ~2 mT of residual LH_2 can commence. Once again this is conducted in a settled environment. The LH_2 transfer is pressure fed. LH_2 will enter the LH_2 module tank sub-cooled, quenching the GH2 vapor and sucking in additional LH_2. This "zero-vent fill" transfer process is indifferent to the liquid-gas interface. This zero-vent fill process has been demonstrated to be very effective, attaining nearly 100% fill.[21]

Following completion of the LH_2 transfer, Centaur's LH_2 tank is vented to vacuum, fully evacuating the residual hydrogen gas. Following the Centaur LH_2 tank "safing", the ~10 mT of residual LO_2 is transferred from Centaur's LO_2 tank to Centaur's LH_2 tank, the LO_2 module tank, using the same transfer process. Once Centaur's LO_2 tank is completely drained the tank is locked up trapping the residual helium and GO2. This residual gas must be kept at a higher pressure than Centaur's LH_2 tank (LO_2 module) to avoid reversing Centaur's common bulkhead.

6. TECHNOLOGY FOR FUEL DEPOTS

Successful operation of on-orbit cryogenic propellant depots requires that numerous technologies be synergistically implemented on a scale larger than previously demonstrated, Table 3. To maximize the chance of success the first operational propellant depot should maximize use of already demonstrated technologies and CFM techniques while allowing for demonstration of enhancing, but not first flight critical technologies. This section describes the key technologies that will enable efficient depot operations. Many of the technologies are well understood, while other, less developed technologies can be demonstrated on the depot but are not required for successful operation.

Low Acceleration Settling

Low-level acceleration has been used on all past cryogenic upper stages to separate liquid and gas allowing reliable pressure control through venting as well as efficient propellant acquisition, Figure 17. Historically this acceleration has been provided by small thrusters slowly accelerating the stage forward. For the multi-month/year mission of a propellant depot acceleration through settling is not practical due to the large required reaction mass.

Centrifugal propellant settling provides an alternative mode of acceleration. Spinning the depot provides continuous centrifugal settling and allows similar fluid control as with axial settling. ULA and the Department of Defense (DOD) demonstrated centrifugal propellant control on the Defense

Table 3. Settled cryogenic propellant transfer can benefit from the vast CFM experience used on Centaur and other cryogenic upper stages as well as near term flight demonstrations such as CRYOTE.

Cryo Transfer Technology	Current TRL		TRL Post-CRYOTE Lite		TRL Post-CRYOTE Pup, Free Flier	
	0-g	Stld	0-g	Stld	0-g	10^{-4} g
Transfer System Operation	4	5	4	9	9	9
Pressure Control	4	9	6	9	9	9
Low Acceleration Settling	N/A	9	N/A	9	N/A	9
Tank fill operation	4	5	4	9	9	9
Thermodynamic Vent System	5	5	7	7	9	9
Multi-layer insulation (MLI)	9	9	9	9	9	9
Integrated MLI (MMOD)	6(2)	6(2)	9(7)	9(7)	9	9
Vapor Cooling (H_2 para-ortho)	9(4)	9(4)	9	9	9	9
Passive Broad Area Cooling (active)	9(4)	9(4)	9(4)	9(4)	9	9
Active cooling (20k)	4	4	4	4	9	9
Ullage and Liquid Stratification	3	9	9	9	9	9
Propellant acquisition	2	9	9	9	9	9
Mass Gauging	3	9	9	9	9	9
Propellant Expulsion Efficiency	3	9	9	9	9	9
System Chilldown	4	5	4	9	9	9
Subcooling P>1atm (P<1atm)	9(5)	9(5)	9(5)	9(5)	9(5)	9(5)
Fluid Coupling	3	3	3	3	9	9

Meteorological Satellite Program-18 (DMSP-18) mission (AV-017) October 18, 2009.[22] This flight demonstrated the effectiveness of liquid spin up, transition from axial settling to radial and back to axial settling at low acceleration and while venting. While the DMSP-18 mission demonstrated spinning about the minor axis, Figure 18, the depot will spin about the major axis. This results in a stable rotation that will reduce overall station keeping propellant requirements and increases the rotational radius to the liquid surface which increases the acceleration for a given spin rate.

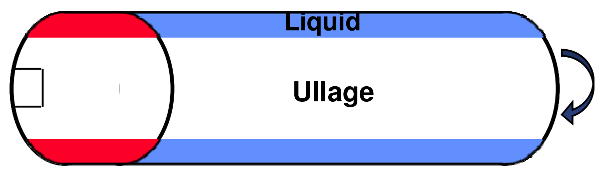


Figure 18 - Centrifugal acceleration can separate the liquid and gas allowing use of existing, flight proven settled cryo-fluid management techniques. Courtesy ULA

Settled operations significantly simplify many aspects of orbital CFM enabling the maximum use of existing, mature CFM technique. With settling, large-scale passive propellant storage and transfer becomes an engineering and demonstration effort, not a technology development endeavor. Beyond greater technical maturity, settled cryogenic fluid management has been flight-proven to actually enhance the cryo-system operation by:

- Reducing liquid heating - much of the heating is absorbed by the ullage.
- Reducing ullage mass – provides a warmer, less dense ullage, Figure 19.
- Improving system reliability - avoids the added complexity of circulation for pumps to distribute cooling in the large tank.

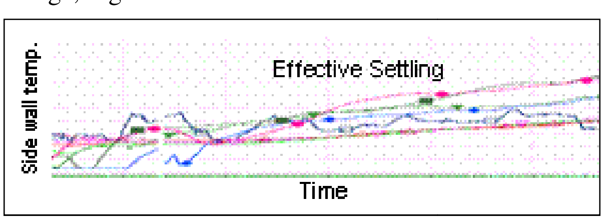


Figure 17 - Centaur has demonstrated effective propellant control at very low accelerations, well below the acceleration required to support settled propellant transfer. Courtesy ULA

Pressure Control

Pressure control of cryogenic propellant tanks is accomplished by thermal management (energy extraction) of the cryogenic propellant. Heating, even if localized,

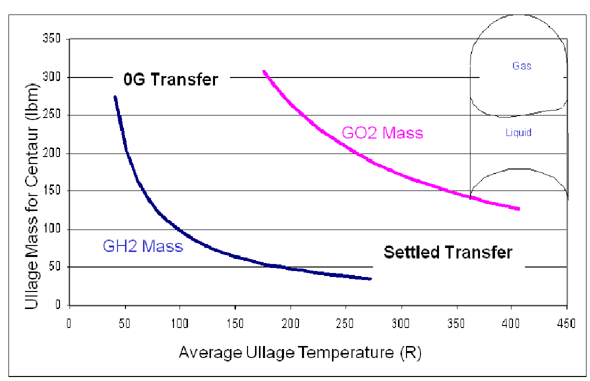


Figure 19 - Low acceleration effectively separates the ullage and liquid enabling pure gas venting while reducing the gaseous residuals. Courtesy ULA

results in propellant boiling that must be controlled to prevent detrimental pressure rises. All forms of pressure control benefit significantly from efficient thermal protection reducing the energy that must be removed from the system.

Ullage venting results in robust tank energy extraction and has been used to control tank pressure for all orbital cryogenic propulsion stages flown, including: 194 Centaur flights, 16 Delta III and IV flights, and 8 Saturn IVB flights. This robustness is due to the fact that ullage venting causes all localized propellant warm spots in the propellant to boil regardless of the location in a tank. This boiling cools the propellant avoiding uncontrolled tank pressure rise that would result if the warm areas were left untended.

The Simple Depot will rely on GH2 venting to control hydrogen tank pressure. Refrigeration will be used to control the oxygen tank pressure with GO2 venting used as a fail safe for oxygen tank pressure control.

Refrigeration provides an alternative method to venting for pressure control. Refrigeration transfers energy from the cryogenic propellant to a separate working fluid through a heat exchanger. This cooling must be distributed through the propellant to prevent hot spots from forming and causing the pressure to climb.

Zero boil-off of LO_2 will be demonstrated in the LO_2 module using GH2 vapor to provide the necessary refrigeration. Cold GH2 will flow through a tube attached to the Centaur tank sidewall above the maximum LO_2 surface. The cold tank wall will result in GO2 condensation, reducing tank pressure. The reduced tank pressure allows GO2 bubbles to form at peak tank heating locations. In the low settled environment these bubbles will rise to the surface and eventually condense back into LO_2 at the cold wall. The flow rate of GH2 will be controlled to maintain a steady oxygen tank pressure.

The Simple Depot will also incorporate active cooling of the LO_2 for redundancy and technology demonstration. Active cooling uses a refrigeration loop to extract energy from the tank, allowing cooling without venting fluid overboard. Active cooling requires electrical power to drive a cryocooler pumping energy from the cold LO_2 to a warm radiator. The required input power grows exponentially with decreasing fluid temperature. Current cryocoolers are quite effective at LO_2 temperatures, while having limited cooling ability at the low temperature of LH_2. For this reason refrigeration on this first depot will only be used for direct LO_2 cooling and broad area cooling, not direct LH_2 cooling.

Para-Ortho Conversion

Para-to-ortho hydrogen conversion is an endothermic process, adsorbing heat. Vapor cooling with a para-to-ortho catalytic converter can generate 500 Joule/gm additional cooling. While para-to-ortho conversion has been demonstrated in commercial LH_2 systems, transition metal catalyst studies with supported iron, nickel, copper, etc., will need to be conducted to optimize the converter performance in terms of percent para-H2 conversion at the calculated GH2 flow rate for the depot system.

Integrated Multi-Layer Insulation (IMLI)[23]

The Simple Depot and CRYOTE Core cryogenic propellant tanks are shrouded in shrouded in IMLI, incorporating radiation barriers and MMOD protection. The various IMLI layers are separated by micro-molded polymer structures, Figure 20. The polymer substructures promise to improve IMLI thermal and MMOD performance by providing precise, engineered layer spacing.

Micro Meteoroid (MMOD) Protection

Standard MLI blankets have a surprisingly high level of protection from the impact of MMOD. The incoming particle is broken apart by the initial impact, and the resulting debris cloud is further retarded and broken up as it progresses through each successive layer. IMLI’s. Precise barrier spacing enhances standard MLI MMOD tolerance by combining increased and controlled spacing between layers and slightly thicker layers than standard MLI blankets. The inclusion of a thicker BAC layer further increases MMOD protection. Increased layer spacing also enhances the protective capability of the IMLI by providing distance over which a debris cloud spreads, thereby dissipating the energy of the cloud before impact onto the next layer.

Broad Area Cooling (BAC)

Enhanced thermal performance of MLI/IMLI systems can be obtained by adding plumbed, cooled shields within the MLI or IMLI blankets. BAC using tank boil-off (passive) or cryocoolers (active) can be used to provide cooling. Passive BAC has flown by Ball Aerospace on LH_2 PRSD tanks, as well as multiple helium flight dewars. Active BAC on a cryogenic propellant tank simulator has been demonstrated in a laboratory environment by NASA and Ball Aerospace[24, 25] Figure 21.

BAC passive cooling can be employed to reduce boil-off losses from cryogenic propellant storage tanks. Vapor cooling intercepts heat flow with vapor vented from the tank. Routing vent plumbing to key penetrations such as structural interfaces or plumbing interfaces allows vapor

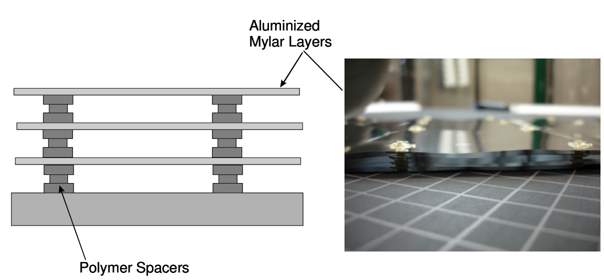


Figure 20 – Integrated MLI provides enhanced thermal and MMOD protection.

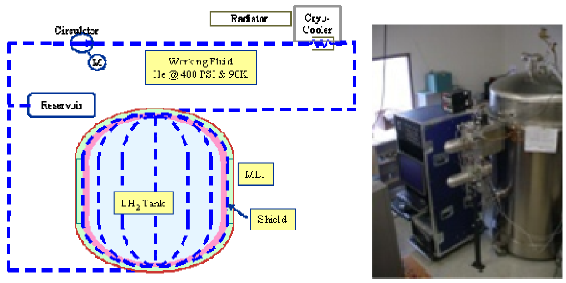


Figure 21 – Schematic of Broad Area Cooling.

cooling to intercept high localized heating of penetrations. Through the use of conductive layers embedded in an MLI blanket, vapor cooling reduces the broad area temperatures reducing heating to the large surface are of tanks.

A significant benefit of employing an active BAC is the change in the operating requirements for the cryocooler. Since the BAC shield is buried within the MLI/IMLI blankets, all of the heat interception occurs at a higher temperature, ~90 K, reducing the performance requirements for the cryocooler. This precludes the heat interception requirement at 20 K.

Hydrogen provides a particularly useful fluid for vapor cooling thanks to its initial cold temperature and high energy absorption potential, Figure 22.

Between 2004 and 2006 NASA, under the In-Space Cryogenic Propellant Depot (ISCPD) Project, investigated the various options by which cryocoolers might be integrated with cryogenic propellant storage tanks. The goal was to substantially reduce or entirely eliminate boil-off losses with a minimal increase of total system mass. For LO_2 storage, the solution is straightforward, as it is possible to integrate existing high capacity 90 K flight coolers, or derivations of them, directly with an LO_2 tank. This is not a viable option for LH_2 because the required 20 K coolers of the required thermal power do not yet exist. However, analysis performed under ISCPD predicted that a large percentage of the heat leak into an LH_2 tank could be intercepted at ~90 K, thus circumventing the need for a high capacity 20 K flight cryocooler. The BAC concept, Figure

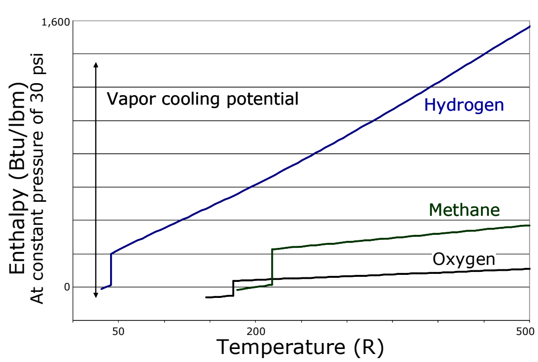


Figure 22 - Hydrogen has 9 times the heat absorption capacity of Oxygen and 4 times that of methane making hydrogen the ideal vapor cooling fluid. Courtesy ULA

21, evolved whereby heat is removed, via a pressurized helium circulation network, from a large but low-mass metal foil thermal radiation shield embedded within an LH_2 tank's passive thermal insulation system. Analytical tools and models were developed and eventually incorporated into a general cryogenic system analysis software package, with which a series of trade studies were conducted. These trades determined practical ranges of values for the circulation loop charge pressure, mass flow rate, tubing diameter, and foil thickness, based on likely mission scenarios.

In order to advance the TRL of this concept, the CFM Project funded, in addition to the ongoing analytical/modeling effort, a series of three experiments at NASA Ames Research Center (ARC) designed to investigate questions of thermal control stability, heat transfer effectiveness, and temperature uniformity in distributed cooling systems. In addition to NASA's investment, the Air Force Research Laboratory has developed a broad area cooling liquid hydrogen zero boil-off ground experiment designed for system integration demonstrations and validations of component technologies. The overall TRL has consequently been raised from 3 to 5.

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 typical, even at the relatively low accelerations encountered during pre-start and end of mission propellant blowdown.

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 warm during the expulsion process. By warming the ullage, there is the potential to increase total expulsion efficiency by ~0.9%, Figure 19.

Mass Gauging

Settling provides a gas/liquid interface that can be measured to establish liquid mass. Thermocouples and gas/liquid diodes internal to the tank have proven very effective in defining the station level of the liquid/gas interface. Sierra Lobo's Cryo-Tracker® concept promises a simple, robust system for accurate internal liquid surface gauging at low acceleration. For systems with a substantial gas-liquid temperature gradient mounting temperature patches directly to thin tank walls has also proven effective, Figure 23.

Less mature flight cryogenic mass gauging techniques such as Pressure-Volume-Temperature, RF and laser offer potential benefits including mass gauging under zero-g conditions. Alternative mass gauging should be included on the depot to provide redundant mass information and demonstrate these promising technologies.

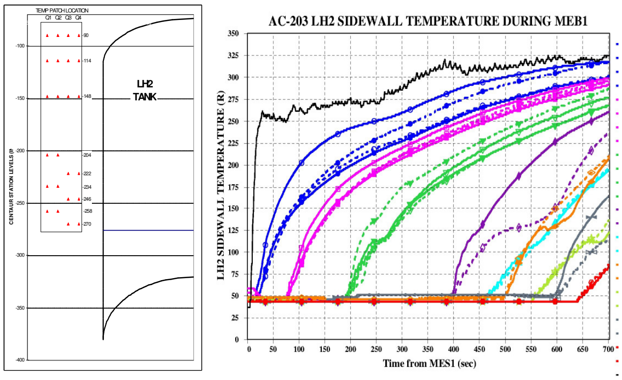


Figure 23 – Centaur externally mounted thermocouples effectively measure liquid level. Courtesy ULA

System Chilldown

Existing upper stages have demonstrated hardware chilldown procedures that are directly applicable to cryogenic transfer. Chilldown of ducting, tank walls and the engines have been demonstrated with multiple alternate chilldown procedures. Chilldown effectiveness using full flow, trickle flow, and pulse flow for both LH_2 and LO_2 flow have been demonstrated in the low g space environment. The pulse chilldown methodology has proven especially effective.

Sun Shield

A sun shield can be used to shadow objects that must be kept very cold. The James Web Space Telescope (JWST) uses an open cavity planer sun shield to ensure that the entire mirror/instrument assembly is maintained at a low temperature, Figure 24. Propellant depots in free space, such as at a Lagrange point, can use this same shielding concept to provide a very cold environment where cryogenic, even LH_2, storage is readily achieved.

For a propellant depot located in LEO radiation and reflection from the Earth result in substantial heating mitigating the benefit of a planer sun shield. For this reason ULA, ILC-Dover and NASA have pursued a conic sun shield that can protect the depots cryogenic tanks from both the Sun and Earth while still providing a view to space to reject heat, Figure 25.[26]

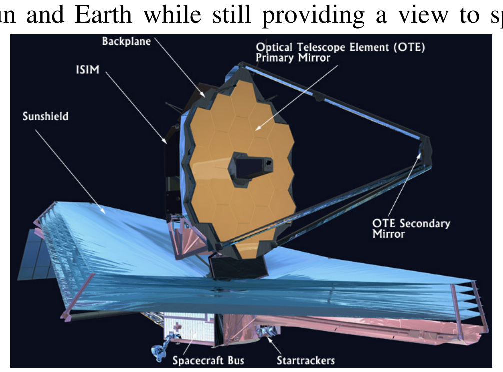


Figure 24 – Depots located distant from a planetary body benefit from a planer sun shield such as being developed for NASA's James Web Space Telescope. Credit: NASA

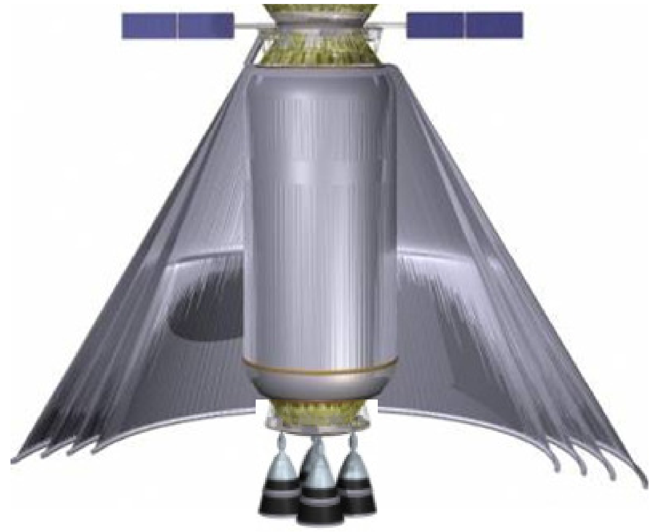


Figure 25 – Conic sun shield shades the cryogenic upper stage to minimize boil-off. Courtesy ULA

For small sun shields it may be possible to erect the sun shield prior to launch. However in most cases the shield will have to be deployed once on-orbit. The JWST uses a mechanical boom to deploy the sun shield. Alternatively a pneumatic boom, inflated with waste GH2, can be used to deploy and support the sun shield, Figure 26.

Structural Heat Load

For cryogenic in-space stages such as the Saturn IVB, Centaur, DCSS as well as the proposed Ares V EDS, structural supports account for a large fraction of the total heat load. This support structure heating is even more critical for long duration cryogenic stages where incorporation of MLI substantially reduces the surface area heating. Substantially reducing the structural load path requires a combination of minimizing the load (e.g., launching tank empty) and providing a structural interface designed for minimum heat transfer (e.g., struts). Ball Aerospace cryogenic dewar strut technology, Figure 27, is readily transferable to large propulsion stage architectures. Key elements include leveraging off Ball Aerospace flight

Figure 26 – ULA, ILC-Dover and NASA have developed a pneumatically deployed sun shield to support long duration cryo-storage. Courtesy ULA

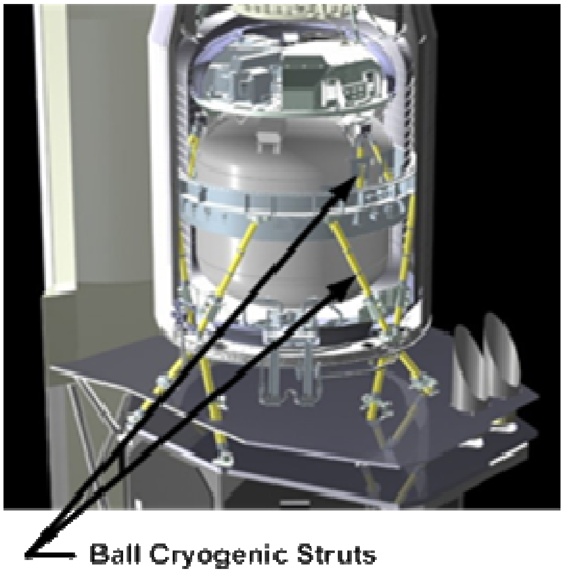


Figure 27 – Cryogenic composite struts on Spitzer.

heritage strut designs, as well as vapor cooling leveraged from flight programs such as Spitzer.

Subcooling Propellant

One powerful technique for easing the challenge of in-space cryogenic fluid storage is to subcool them below the temperature required for their on-orbit use. Propellants such as LH_2 have very large heat capacities. This heat capacity of the chilled hydrogen allows it to absorb the large quantities of energy that leak into the tank over time even with the use of the best thermal protection system (TPS) systems without the need to vent the cryogen, thus extending its in-space vent-free 'hold-time'. This subcooling can represent conditioning the cryogen to a saturation pressure between atmospheric and the engine required saturation pressure as already used on existing upper stages. Or one can subcool the cryogen below the boiling point at atmospheric pressure. Subcooling represents a simple technique that can extend the operational life of a spacecraft, upper stage or an in-space cryogenic depot for months with minimal mass penalty. In fact, subcooling hydrogen to 16 K at 1 atmosphere pressure prior to launch triples its vent-free hold-time over hydrogen loaded into tanks at its normal boiling point of 20 K at 1 atmosphere pressure.[27]

Figure 28 shows the phase diagram for hydrogen. The thermodynamic condition "N" is the normal boiling point of hydrogen (~20.4 K at 1 atm. pressure ~ 101 kPa). Existing cryogenic launch vehicles load LH_2 at a saturation pressure above this to simplify the launch complex and provide structural stability of the tank. For the sake of this explanation, we will use the example of the J2-X engine operation (e.g., EDS-like upper stage) and loading LH_2 at the point "N" thermodynamic condition. The point start box high end (SBHE) is the J2-X start box high end condition (22.2 K at 1.9 atm. pressure ~ 196 kPa). This is the highest thermodynamic condition at which the J2-X engine is rated to start. If the thermodynamic condition of the hydrogen that is being stored increases beyond this condition due to heat

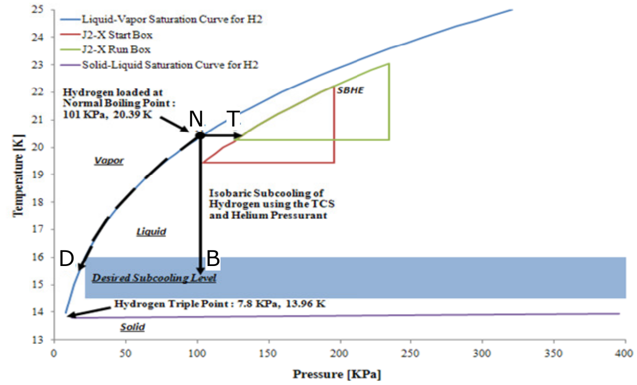


Figure 28 – Thermodynamics of the TCS process and the J2-X engine operating conditions. Courtesy NASA

flow into the tank, the tank will have to be vented to cool the LH_2 (active refrigeration is a possible future alternative) resulting in loss of hydrogen and associated performance degradation. An additional consideration is that the pressure must be kept within the tank allowable pressure limit which will require venting even if engine operation does not have to directly be taken into consideration (e.g., cryogenic propellant depots, although the hydrogen would have to still be brought back down to a thermodynamic condition that would allow the engine to start).

The launched hydrogen can be isobaricly subcooled (following the Arrow "NB", Figure 29) below its normal boiling point prior to launch to about 16 K at 1 atm. The subcooled hydrogen has a very large heat capacity and even though heat will be leaking into the tank while in-space if the hydrogen is still subcooled, there will be no need to vent. In fact, the energy gain required in the stored hydrogen to reach the condition SBHE from the subcooled condition ("B" – 16 K at 1 atm.) is a factor of 3 more than the energy gain required to reach the condition SBHE from the normal boiling point condition ("N"). This means that for the same heat flux the hold time for hydrogen before venting has to be initiated would be tripled if it is launched subcooled instead of launching at its normal boiling point. Depending on tank sizes, subcooling to 16 K at 1 atm could buy weeks to months of hold time margin and tolerance for cryogenic propellant storage and transfer imperfections. Subcooling will dramatically augment all other cryogenic storage technology developments.

Figure 29 shows a notional Thermodynamic Cryogen Subcooler (TCS)[28] that could subcool the cryogenic propellant on the launch pad before launch. Note that the only hardware component that will be launching is the tank. The TCS will isobarically subcool the hydrogen prior to launch. The hydrogen that is being subcooled will be extracted from the tank. Some of this extracted hydrogen will be passed through a JT valve that expands the hydrogen. The hydrogen on the upstream side of the JT valve, at thermodynamic condition 1 (TC1), will be at the temperature of the hydrogen in the tank (initially ~ 20 K). The hydrogen on the downstream side of the JT valve (TC2) will have the same enthalpy as the hydrogen on the

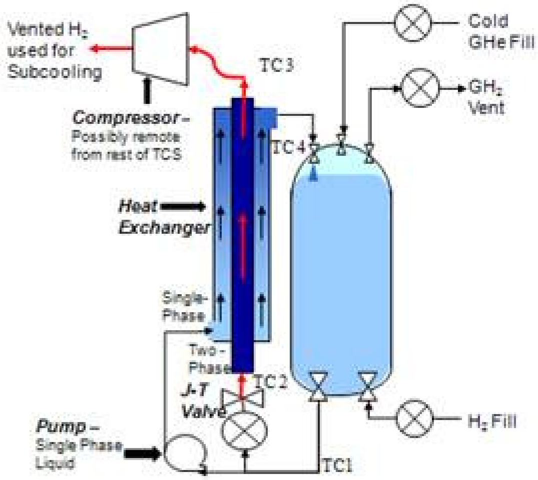


Figure 29 – The TCS concept for subcooling cryogens on the launch pad. Courtesy NASA

upstream side, but at a lower pressure (~0.1 atm.) and substantially lower temperature (~15 K). While going through this expansion the hydrogen at TC2 becomes a two-phase fluid. Most of the liquid hydrogen extracted from the tank at TC1 will be pumped into the outside tube of a concentric tube heat exchanger - the single-phase tube. The two-phase hydrogen at TC2 is passed into the center tube of the concentric tube heat exchanger — the two-phase tube. Since the hydrogen in the two-phase tube is at a lower temperature than the hydrogen in the single-phase tube it can extract heat from the hydrogen in the single-phase tube and thus subcool the propellant. The two-phase hydrogen will increase in vapor quality along the two-phase tube until it totally vaporizes and is vented (TC3) to a flare stack through a compressor system. This compressor system will probably be the heaviest and most power intensive component of the TCS and hence it might be remote from the rest of the TCS. The subcooled hydrogen at the end of the single-phase tube (TC4) is then fed back into the hydrogen tank. As the bulk hydrogen in the tank is subcooled, its density increases so the tank will be backfilled with non-condensable cold helium to prevent the tank from experiencing a compressive atmospheric load. The TCS components will be isolated from parasitic heat inputs by using a vacuum jacket and MLI. By using the cooling enthalpy available in the stored cryogen the need for a high-capacity refrigeration system is diminished. The TCS reduces power, mass and foot-print of the subcooling system easing integration into the launch tower.

Autonomous Rendezvous and Docking

Russia has been performing autonomous rendezvous and docking (AR&D) operations 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 languished in other countries, including the United States, several recent efforts have demonstrated the viability of non-Russian AR&D systems.

NASA's DART, Air Force Research Laboratory's (AFRL) XSS-11, Defense Advanced Research Projects Agency's (DARPA) Orbital Express, Figure 30, Europe's Automated Transfer Vehicle (ATV), and Japan's HII Transfer Vehicle (HTV) missions were all designed to further AR&D capabilities. Dart was the first attempt to demonstrate American autonomous rendezvous technologies. Unfortunately errors in the global positioning system (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, successfully demonstrated numerous autonomous rendezvous and proximity operations during its yearlong mission. Orbital Express launched in March of 2007 demonstrated AR&D, as well as orbital servicing, including the transfer of hydrazine and helium.[29,30] Europe's ATV, launched in February of 2008 and Japan's HTV launched in 2009 successfully demonstrated multiple rendezvous with the ISS before docking to deliver cargo. Robust AR&D development continues with, NASA's Orion crew capsule, along with NASA's two commercial orbital transportation services (COTS) program winners (SpaceX and Orbital Sciences Corporation). Results from these on-going programs will ensure that AR&D is widely available to support the servicing and use of propellant depots.

Integrated Vehicle Fluids (IVF)[31]

Today's DCSS missions are limited to three burns over the course of an eight hour mission due to the limited supply of hydrazine, helium and electrical power. Even to satisfy the existing range of missions from low LEO to medium Earth orbit (MEO), GTO, and geosynchronous orbit (GSO), a variety of mission kits are required. To improve mission flexibility, ULA is developing the integrated vehicle fluids (IVF) system to allow the use of hydrogen and oxygen from the upper stage primary tanks to satisfy the settling, attitude control, pressurization, and power requirements, Figure 31.[32] The IVF will allow the elimination of hydrazine and helium from the vehicle while replacing the existing large capacity batteries with small rechargeable batteries.

Development of the hydrogen/oxygen thruster, engine, pump, alternator, rechargeable battery and cryogenic compatible composite bottles are progressing well with concept testing under way, Figure 32. IVF will be especially

Figure 30 – The depot avionics could be derived from existing spacecraft such as Orbital Express.

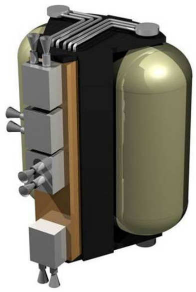


Figure 31 – The integrated vehicle fluids module is designed to support reaction control system (RCS), pneumatic and power requirements. Courtesy ULA

Figure 32 – Testing has demonstrated the functionality of this low cost hydrogen/oxygen thruster. Credit Innovative Engineering Solutions

valuable in a depot based transportation economy by eliminating the need to store and transfer additional commodities such as hydrazine and helium.

7. HISTORIC CRYOGENIC FLIGHT EXPERIENCE

America's existing CFM capability is based on decades of experience storing LH_2/LO_2 in launch vehicle upper stages, Table 4,[33] and numerous cryogens in spacecraft cryostats, Table 5. This historic flight experience suggests that with proper design, a propellant depot can efficiently store large quantities of cryogenic liquid, including LH_2, for years.

Table 4. Titan/Centaur demonstrated 2% per day system (LO_2 and LH_2) boil-off with minimal thermal protection designed for short duration geosynchronous orbit mission. Courtesy ULA

	TC-15		TC-11	
	LO_2	LH_2	LO_2	LH_2
Total Boil-off per day (% of full LO_2 or LH_2)	1.5%	4.1%	1.0%	5.1%
System boil-off per day (%of full LO_2+LH_2)	2.0%		1.6%	

8. TESTING AND DEMONSTRATION

Many features of the advanced thermal protection system required to allow long-term, large-scale cryogenic storage can be demonstrated on the ground. Integrated MLI, vapor cooling, broad area cooling, and light weight tank structures can all be effectively demonstrated on the ground. This ground testing is directly applicable to orbital depots and cryogenic propulsion stages that rely on settling to provide a determinant fluid-gas interface. This same testing is also useful for zero-g cryogenic propellant storage when coupled with orbital testing.

While much has been learned from the multi-purpose hydrogen test bed (MHTB) testing conducted at Marshall Space Flight Center (MSFC), it is crucial that further ground testing use structures that are representative of flight systems, to avoid results that are influenced by MHTB's the ½-in thick aluminum tanks with substantial wall conduction. ULA is working with NASA centers (Plum Brook, GRC, KSC, MSFC, Johnson Space Center [JSC], Ames, and Goddard Space Flight Center [GSFC]) to pursue flight like testing in NASA's Plum Brook B2 vacuum chamber, Figure 33. ULA has offered to loan a Centaur tank to NASA for an extended period, Figure 34. KSC plans on beginning tank modifications for testing in 2011, allowing ground based system testing to begin as early as 2012. The Centaur, with its 0.02-in stainless steel walls, provides the ideal platform for such testing. Centaur's 194 flights provide a vast database, both settled and zero-g, with which to anchor ground test data. Centaur's low thermal/structural mass and low conductivity are critical for long duration flight cryo-storage tanks. It is possible to duplicate thermal protection enhancements such as vapor cooling and IMLI tested at Plum Brook on upcoming flights thanks to the Atlas's ability to encapsulate Centaur in the payload fairing during assent.

Existing Upper Stages as Technology Test Bed

The Centaur, Figure 35, and DCSS can be used to further America's CFM capabilities. ULA has a 50 year history of flight demonstrating new capabilities as summarized in Table 6. This has culminated in the current efficient existing operations of the DCSS and Centaur. Both stages use axial spin (~1 °/sec) to even the effect of solar heating on the vehicle and spacecraft for coast lengths in excess of 20 minutes. Prior to the main engine burn following long coasts Centaur performs a "Maytag" maneuver, reversing roll direction multiple times to mix the propellants and minimize hot spots. Both the DCSS and Centaur continue venting during these roll maneuvers to provide pressure control and propellant conditioning. Fluid acquisition and transfer are initiated while axially settling. Numerous mass gauging techniques have been used on Centaur and DCSS. During high g periods of main engine burn, capacitance probe and ΔP (liquid head) mass gauging have proven successful. During low-g settled coasts, measurement of vehicle acceleration has been used to calculate total propellant remaining, and has been confirmed by multiple sidewall temperature patches used for level sensing.

Even after more than 200 combined DCSS and Centaur missions, experimental flight data is still being gathered. In October 2009, ULA partnered with the Air Force to take advantage of Centaur's 12,000 lb of residual LO_2 and LH_2 on the Defense Meteorological Satellite Program-18 (DMSP-18) mission to demonstrate very low-g axial as well

Table 5. Cryostats already in use or under development for scientific satellites demonstrate the ability to efficiently store cryogens for long durations (public release permission granted by Lockheed Martin and Ball)

Program	Cryogen	Company	Tank capacity (kg)	Operating temp (K)	Life-time (mo)	Flow-rates per day (%)
Gravity probe-B SMD/EDD	Superfluid helium (SFHe)	LMC	334 29	1.8 1.8	16 2	0.17% 1.34%
SHED	SFHe	LMC	29	1.8	30	0.09%
WIRE	Solid hydrogen	LMC	4	12	5.4	0.49%
SPIRIT III	Solid hydrogen	LMC	85	9.5	10	0.264%
CLAES	Solid hydrogen	LMC	85	10.4	21.4	0.15%
CLAES (2-stage) (Post Challenger)	Solid Neon Solid CO2	LMC	449 340	14 125	19.8 25	0.16% 0.13%
Extended life cooler (ELC)	Solid methane Solid ammonia	LMC	150 73	65 145	66 64.9	0.04% 0.07%
Long-life cooler (LLC) (2-stage)	Solid methane Solid ammonia	LMC	91 42	65 145	44.9 40.4	0.07% 0.08%
IRAS	SfHe	Ball	70	1.7	10	0.32%
COBE	SfHe	Ball	83	1.5	49	0.07%
Spitzer	SfHe	Ball	45	1.3	66	0.05%
HTTA	Liquid hydrogen	Beech Ball	446	20	150	0.022%
OTTA	Liquid Oxygen	Beech Ball	7186	90	150	0.022%
Power Reactant Storage Assy (PRSD)1	ScH2 ScO2	Ball	42 354	20-83 90-211	0.5 0.5	~7% w/ heaters
PRSD2	ScO2	Ball	354	90-211	14.8	0.22 % s
PRSD Enhanced	ScO2	Ball	354	90-211	38.4	0.0085 %
HALE	Liquid hydrogen	Ball	426	20	0.17	19.6% SOFI

Figure 33 – NASA's Plum Brook B2 facility is perfect for large scale integrated testing. Courtesy NASA

Figure 34 – Cryo-fluid testing of a Centaur tank can provide invaluable data on the actual performance of proposed thermal protection schemes. Courtesy ULA

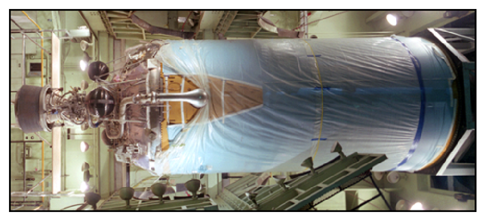


Figure 35 - Existing upper stages can be used to further CFM technologies that do not adversely affect the upper stages primary mission. Courtesy ULA

Table 6. Centaur and Delta's upper stage have conducted numerous CFM flight demonstrations relevant to cryogenic propellant transfer. Courtesy ULA

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 and Liquid Stratification	Pressure Collapse
Propellant Utilization	Bubbler vs. Ullage Pressn.
Mass Gauging	Unbalanced Venting

as centrifugal propellant settling, efficient feedline/RL10 chilldown and two phase RL10 operation.

Creative implementation can actually allow quite advanced demonstrations on the 5 to 10 annual DCSS and Centaur missions. Advanced MLI, deployable sun shields, enhanced chilldown sequences, magnetic liquid positional control and many other promising technologies can all be demonstrated on upcoming flights without compromising mission success. Reasonable modifications can even allow demonstration of vapor cooling, hydrogen/oxygen thrusters, and cryo-transfer.

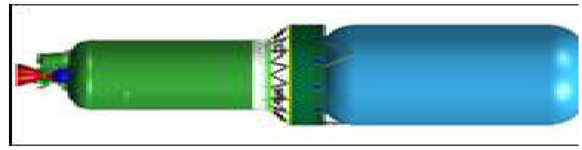
While these advanced demonstrations are designed to benefit future exploration missions, some will provide near term benefit to national security, NASA science, and commercial customers. Ullage-fed H_2/O_2 thrusters, for example, could supplant hydrazine used for settling allowing longer duration coasts and enhanced performance. Such thrusters could also enable performance neutral upper stage de-orbit thereby reducing orbital debris.

9. APPLICABILITY TO LARGER DEPOTS

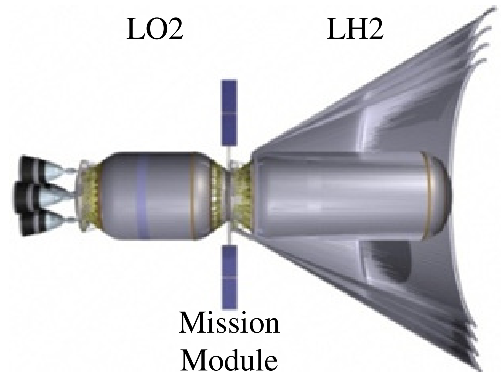
With a 30 mT LO_2/LH_2 capacity the described Centaur derived cryogenic propellant Simple Depot can provide near term operational use supporting large scale robotic missions and even crewed Earth Moon Lagrange point and lunar fly-by missions. By making efficient use of the entire Atlas 5 m payload fairing volume for the LH_2 module the existing Atlas can launch a depot with 70 mT of combined LO_2/LH_2 capacity, Figure 36. With ULA's proposed larger Advanced Common Evolved Stage (ACES) the depot capacity in a single EELV launch increases to 120 mT or even 200 mT with a 6.5m PLF. The same depot concept can be applied to future heavy lift vehicles to allow launch of even larger capacity propellant depots.

Once on orbit the depot will need to be periodically refueled to support beyond LEO missions and make up for boil-off losses. Refueling tanker missions can be launched with propellant mass sized to the selected launch vehicle. The Simple Depot is designed to nominally rotate maintaining settled propellant handling. However, the low thermal mass design enables the depot to also stop rotating for extended periods. This enables two primary options for rendezvous, docking and propellant transfer.

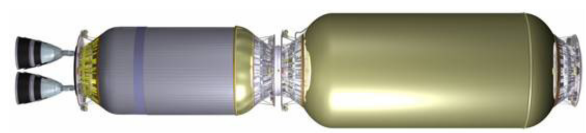
Docking with the rotating depot complicates the actual docking and control of the depot-tank system. However, with a rotational rate of one revolution every 6 minutes and acceleration environment of ~10^{-4} g's it appears that these complications can be accommodated. The benefits of continuing rotation is simplification of propellant acquisition in the tanker, improved propellant transfer efficiency (reduced trapped and ullage mass), reduced boil-off loss in the depot. The tanker to depot propellant transfer will be similar to the LH_2 and LO_2 transfer described in Section 5.



5 m LH_2 tank enables 70 mT depot capacity



5 m ACES and LH_2 tank enables 120 mT capacity



5 m ACES and 6.5 m LH_2 tank enables 200 mT capacity

Figure 36 - The Simple Depot concept provides efficient cryogenic storage and single launch emplacement enabling very exciting exploration missions supporting any launch vehicle NASA chooses to pursue.

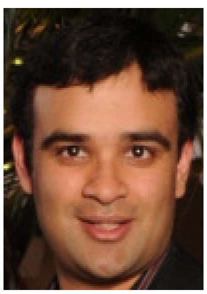
Docking with the depot static simplifies docking and eliminates the system control issues. However, zero-g propellant acquisition and transfer is much less mature, propellant or large momentum wheels are required to support de-spin and re-spin of the depot, the transfer efficiency is less efficient, and the tankers will require added mass and complexity of a zero-g propellant acquisition.

10. SUMMARY

Propellant depots can enhance the mission capability of exploration architectures regardless of the use of small reusable rockets, larger EELV class rockets or much larger heavy lift vehicles. Propellant depots utilizing existing and near term technology can be affordably deployed this decade. These depots can include less mature technology for demonstration purposes without risking mission success. The proposed on-orbit cryogenic propellant mission designs allow for complete ground integration and check out followed by launch on as a secondary payload (CRYOTE), or on a single rocket (Simple Depot). Such demonstrations are sufficiently affordable that mission planners can include flying replacement depots every few years incorporating enhanced techniques and technology.

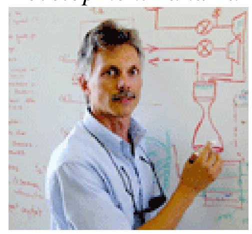
BIOGRAPHY

Christopher McLean *is a Principal Engineer for Ball Aerospace & Technologies Corp. From 2004 he managed Ball Aerospace's advanced technology programs for the long-term storage and delivery of cryogenic propellants for Exploration, as well as leading system design activities for robotic precursor missions. His background is in advanced in-space propulsion technologies including hall effect thruster and arcjet systems. From 1998 to 2004 he was a specialist at Pratt and Whitney Chemical Systems Division managing the development of avionics and components for electric propulsion applications and for the Global Missile Defense (GMD) missile program. From 1996 to 1998 he managed the development of electric propulsion systems and facilities at TRW. From 1991 to 1996 he developed a range of advanced propulsion technologies at Rocket Research Company. He has a BSAA and an MSAA from the University of Washington.*



Shuvo Mustafi *started working at NASA/Goddard Space Flight Center as a co-operative education student in 1997. Mr. Mustafi has worked on the Astro-E cryostat; superconductivity related issues for the Astro E2 spacecraft; cryocooler qualification for the AMS experiment on the International Space Station; and researched a Cryogenic Hydrogen Radiation Shield for protection against solar particle events and galactic cosmic rays. Mr. Mustafi has been working cryogenic propellant storage and transfer related issues since 2006, and since 2009 he has been working on subcooling of cryogenic propellants for long duration in-space storage. Mr. Mustafi has a B.S. in Aerospace Engineering from Purdue University and a M.S. in Applied Physics from Johns Hopkins University*

Mark Wollen *is Vice President of Research and Development and a Founding Principal of Innovative Engineering Solutions (IES). Prior to founding IES in 1994, Mr. Wollen worked for General Dynamics Space Systems Division in San Diego, where he specialized in low-gravity fluid mechanics, fluid and thermodynamic design of launch vehicle cryogenic propellant management and propellant feed systems, and analysis and testing of cryogenic components and instruments. While at General Dynamics, he worked both operational launch vehicle programs (Atlas and Centaur), as well as advanced and conceptual programs (e.g., National Aerospace Plane). Since founding IES, Mr. Wollen has continued to support design, analysis, and testing of launch vehicle propellant feed and propulsion systems for numerous customers. Recent projects include developing novel zero-g cryogenic propellant management devices for NASA, designing and testing new attitude control thrusters to replace existing hydrazine systems with clean propellants, and supporting initial conceptual design and test article fabrication for a liquid hydrogen CRYogenic Orbital TEstbed (CRYOTE).*



Jeffrey Schmidt *is a Senior Manager of Business Development for Ball Aerospace & Technologies Corp. in the Component Technologies Cryogenics Business Unit. He joined Ball Aerospace in 1989 as a Payload Systems Engineer, became a Staff Consultant in 2000 and joined Corporate Business Development as a Senior Manager in 2007. During his career at Ball Aerospace, Dr Schmidt has supported a broad spectrum of programs from NASA planetary science missions to missile defense directed energy and geospatial imaging programs. For over a decade, he was Ball's Principal Fuel Cell Technologist responsible for the design and development of multiple hydrogen- and methanol-based portable fuel cell power solutions for DARPA and the U.S. Army. He holds seven patents in fuel cell and hydrogen storage related technologies. During the last six years Dr. Schmidt has been working to align Ball technology and capability offerings to the present and future needs of NASA Space Exploration. These capabilities include the technologies associated with long duration cryogenic storage of liquid hydrogen, oxygen and methane propellants and 3D Flash Light Detection and Ranging (LIDAR) for relative navigation, rendezvous and docking to name several. Dr Schmidt received his B.S. in Chemistry from North Dakota State University in 1982 and his Ph.D. in Chemistry in 1989 from the Florida State University.*

Laurie K. Walls *began her career in 1982 as a thermal analyst for McDonnell Douglas at the NASA Marshall Space Flight Center (MSFC). In 1985 she transferred to Kennedy Space Center (KSC) and joined NASA in 1986, serving as an Orbiter Structures and Handling engineer in the Space Shuttle Processing Directorate. Ms. Walls moved to the Expendable Launch Vehicles Program at KSC in 1998 where she represented the program for all thermal and aerothermal analytical and integration responsibilities. Since 2008 Ms. Walls has acted as the discipline expert representing the thermal, aerothermal, thermodynamic, and fluids disciplines for the Launch Services Program. Ms. Walls has been involved in many advanced development projects as project and technical lead in the areas of cryogenic fluid management and thermal protection systems, and propellant behaviors in micro-gravity. This work has included experimental testing on the International Space Station (ISS) and design of potential spacecraft mission payloads. Ms. Walls holds a Bachelor of Science in Mechanical Engineering from Seattle University and has published many technical papers related to her work.*

Brian Pitchford *is a Senior Systems Engineer with Special Aerospace Services (SAS) and has over twenty years of experience in the aerospace industry. His experience encompasses many aspects of designing, prototyping, constructing, and testing aerospace flight and ground systems. Mr. Pitchford started his career early in the Spacelab program, worked extensively on the International Space Station (ISS), and has been deeply involved with numerous shuttle and expendable launch vehicle missions during his career. His other activities have included the Orbital Space Plane program, the Jupiter Icy Moons Orbiter program, Evolved Expendable Launch Vehicle (EELV) certification, and additional advanced studies involving space based cryogenic systems and emerging manned and unmanned launch vehicles. Mr. Pitchford has authored papers on subjects including orbital testing of nuclear engines, nullification of asteroids, space-based cryogenic depots, lunar lander concepts, and suborbital applications. He studied Aeronautical and Astronautical Engineering at the Ohio State University, and holds an Associate's Degree from West Virginia State University, a Bachelor's Degree from the University of Central Florida, and a Master's Degree in Space Systems Engineering from the Florida Institute of Technology. He heads up the SAS office at the Kennedy Space Center, and he and his family reside in Titusville, Florida.*

REFERENCE

[1] "Atlas Launch System Mission Planner's Guide", January 2007, http://www.ulalaunch.com/docs/product_sheet/Atlas_Mission_Planner_14161.pdf

[2] "Ariane 5 Data Sheet", 2008, http://www.geocities.com/launchreport/ariane5.html

[3] NASA Solicitation Number NNH10ZTT003L, May 18, 2010, Request for Information Synopsis for The Flagship Technology Demonstrations

[4] NASA Solicitation Number NNH10CC001L, May 27, 2010, NASA Office of The Chief Technologist Crosscutting Capability Demonstrations Division Technology Demonstration Missions Program

[5] Robert D. Braun, NASA Chief Technologist, May 5, 2010 "Investment in the Future: Overview of NASA's Space Technology"

[6] NASA Office of the Chief Technologist, http://www.nasa.gov/offices/oct/home/roadmaps/index.html

[7] Gravlee, M., Vera, C., Wollen, M., McLean, C., Walls, L., Johnson, W; "Micro-gravity Cryogenic Experiment Opportunity", Space 2010, Anaheim, CA, AIAA 2010-8838 September 2010

[8] Gravlee, M., Kutter, B., Wollen, M., Rhys, N., Walls, L.; "CRYOTE (CRYogenic Orbital TEstbed) Concept", Space 2009, Pasadena, CA, AIAA 2009-6440

[9] Wollen, M. A., Gravlee, M., Johnson, W.L.; 'CRYOTE (Cryogenic Orbital Testbed) – Goals, Progress, and Potential', 2009 JANNAF Conference, Colorado Springs

10 Zimmerli, G.A., et al.; Radio Frequency Mass Gauging of Propellants. NASA/TM—2007-214907 (AIAA-2007-1198), 2007

[11] Dodge, Franklin, T.; 'Propellant Mass Gauging: Database of Vehicle Applications and Research and Development Studies', NASA/CR-2008-215281, August, 2008

[12] Haberbusch, Mark S., Lawless, Branden J., Ickes, Jacob C., Walls, Laurie K.; "Reduced Gravity Cryo-Tracker® System," 47th AIAA Aerospace Sciences Meeting, Orlando, FL, AIAA Paper 2009-1599, January 2009

[13] Haberbusch, Mark S.; "Liquid Hydrogen Cryo-Tracker® Systems," 44th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, AIAA-2008-4852, July 2008

[14] Schieb, Daniel J., Haberbusch, Mark S., Yeckley, Alexander J.; "Cryo-Tracker® Mass Gauging System Testing in a Large Scale Expendable Launch Vehicle LOX Tank Simulator," 2006 SPIE Defense and Security Symposium, Orlando, Florida, April 2006

[15] Erickson, R.C.; "Space LOX Vent System – Final Report", NASA-CP-120769, April, 1975

[16] McLean, C., Mills, G., Riesco, M., Meyer, M., Plachta, D., Hurlbert, E.; "Long Term Space Storage and Delivery of Cryogenic Propellants for Exploration", Joint Propulsion Conference, AIAA-2008-4853, Hartford, CT, July 2008

[17] "CRYOTE PMD Trade Study Preliminary Inputs" Don Jaekle, PMD Technology, December 15, 2010

[18] Frank, D. J., Jaekle D. E., Jr.; "Cryogenic Fluid Management for Low-g Transfer". Cryogenic Optical Systems and Instruments II; Proceedings of the Meeting, Los Angeles, CA, Jan. 23, 24, 1986 (A87-26653 10-19), Bellingham, WA, Society of Photo-Optical Instrumentation Engineers, 1986, p. 21-28

[19] Kutter, B.F. et. al.; "A Practical, Affordable Cryogenic Propellant Depot Based on ULA's Flight Experience", AIAA-2008-7644. September 2008

[20] EELV for Use for Demanding Missions, http://spirit.as.utexas.edu/~fiso/telecon/Thronson-Kutter_12-15-10/Kutter%2012-15-10.pdf

[21] Taylor, W.J., Chato, D.J.; "Comparing the Results of an Analytical Model of the No-Vent Fill Process with No-Vent Fill Test Results for a 4.96 m3 Tank", AIAA-92-3078, July 1992

[22] "Successful Flight Demonstration Conducted by the Air Force and United Launch Alliance Will Enhance Space Transportation", http://www.ulalaunch.com/site/docs/publications/SuccessfulFlightDemonstrationConductedbytheAirForceandUnitedLaunchAllianceWillEnhanceSpaceTransportation.pdf

[23] Dye, S.A., Kopelove, A.B., Mills, G.L.; "Integrated and Load Responsive Multilayer Insulation", Advances in Cryogenic Engineering. AIP Conference Proceedings, Volume 1218, pp. 946-953, 2010

[24] Feller, J.R., Plachta, D.W., Mills, G., and McLean, C., "Demonstration of a Cryogenic Boil-Off Reduction System Employing an Actively Cooled Thermal Radiation Shield," Cryocoolers 16, ICC Press, Boulder, CO (2011), pp. 601-609

[25] D.W. Plachta, R.J. Christie, E. Carlberg, J.R. Feller, "Cryogenic Propellant Storage Analyses and Design Tool Evolved From In-Space Cryogenic Propellant Depot Project", Space Cryogenics Workshop, July, 2007, Huntsville, AL

[26] Dew, M., et. al. "Design and Development of an In-Space Deployable Sun Shield for the Atlas Centaur", Space 2008, Anaheim, CA, AIAA 2008-7764 September 2008

[27] Mustafi, S., Canavan, E., Johnson, W., Kutter, B., Shull, J.; "Subcooling Cryogenic Propellants for Long Duration Space Exploration," AIAA SPACE 2009 Conference & Exposition, AIAA 2009-6584

[28] Mustafi, S., Johnson, W., Kashani, A., Jurns, J., Kutter, B., Kirk, D., Shull, J.; "Subcooling for Long Duration In-Space Cryogenic Propellant Storage," AIAA Space 2010 Conference & Exposition. AIAA 2010-8869

[29] Tam ,W., Ballinger, I., Jaekle, D.E. Jr.; "Surface Tension PMD Tank for On Orbit Fluid Transfer", AIAA 2008-5105, 2008

[30] Jaekle, D.E., Jr. Tam, W., Rotenberger, S.; "The Orbital Express Propellant Management Device", 2008 JANNAF MMS/LPW/SPSP Meeting, LPS-111-01, 2008

[31] Zegler, F., Kutter, B.; "Evolving to a Depot-Based Space Transportation Architecture", Space 2010, Anaheim, CA, AIAA-2010-8638, September 2010

[32] Kutter, B.F., et. al. "Ongoing Launch Vehicle Innovation at United Launch Alliance", IEEE Aerospace Conference, Big Sky, MO, IEEE 2010-1020, March 2010

[33] De Kruif, J.S., Kutter, B.F.; "Centaur Upperstage Applicability for Several-Day Mission Durations with Minor Insulation Modifications", Joint Propulsion Conference, AIAA-2007-5845, Hartford, CT, July 2007