Distributed Launch - Enabling Beyond LEO Missions

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Mission planners currently are limited by the mass rockets can launch to the desired destination. Mir and the International Space Station (ISS) have bypassed this limitation by transporting hardware into orbit across numerous launches. So far this tact has not been employed for destinations beyond Low Earth Orbit (LEO). This paper describes the use of multiple launches (potentially of different rockets) and propellant transfer to enable missions that are impossible today. Such missions include very large National Security Geosynchronous Orbit (GSO) satellites that are beyond the Delta Heavy Launch Vehicle capability, Commercial resupply of Cis-Lunar stations, or missions to the Moon and Mars. The method of launching cryogenic propellant, storing it for weeks or months in a disposable Drop Tank, and then transferring the liquid hydrogen and liquid oxygen into a cryogenic upper stage such as Centaur, Delta Cryogenic Second Stage, or Advanced Cryogenic Evolved Stage (ACES) to propel the mission to the desired destination, is coined "Distributed Launch". This paper describes the Distributed Launch concept and summarizes relevant Atlas/Centaur and Delta flight results and on-going testing at ULA, Yetispace and National Aeronautics and Space Administration (NASA) supporting the required enabling capabilities. The current technology readiness level of enabling technologies will be described along with the on-going technology development at ULA that will enable distributed launch in the near future.

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Nomenclature

A552C = *Atlas with 5m Payload fairing, five strap-on solid rocket boosters, two RL10 engines and the Centaur upper stage*
ACES = *Advanced Cryogenic Evolved Stage*[1]
CFM = *Cryogenic Fluid Management*
CRYOTE = *Cryogenic Orbital Testbed*
DCSS = *Delta Cryogenic Second Stage*
FOSS = *Fiber Optic Sensing System*
GH2 = *Gaseous Hydrogen*
GO2 = *Gaseous Oxygen*
GSO = *Geosynchronous Orbit*
GTO = *Geostationary Transfer Orbit*
IMLEO = *Initial Mass in Low Earth Orbit*
Isp = *Specific Impulse*
ISS = *International Space Station*
IVF = *Integrated Vehicle Fluids*
LEO = *Low Earth Orbit*
LH2 = *Liquid Hydrogen*
LN2 = *Liquid Nitrogen*
LO2 = *Liquid Oxygen*
MHTB = *Multi-Purpose Hydrogen Test Bed*
MLI = *Multi Layer Insulation*
MR = *Mixture Ratio*
NASA = *National Aeronautics and Space Administration*
NRO = *National Reconnaissance Office*
SLS = *Space Launch System*
TRL = *Technology Readiness Level*
ULA = *United Launch Alliance*
V564A = *Vulcan*[2] *with 5m Payload fairing, six strap-on solid rocket boosters, four RL10 engines and the ACES upper stage*

I. Introduction

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 than is available with the existing rockets? 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 replacing stages for existing launch vehicles such as the Ariane ECA. For substantially larger performance levels one will need to develop entirely new launch vehicles or be more creative.

For large LEO payloads, mission designers have little flexibility. Either there must be a launch vehicle sufficiently large to loft the payload, for example Saturn V launching Skylab, or the payload must be split into manageable elements and assembled on-orbit. For the massive 500 mT International Space Station (ISS) the National Aeronautics and Space Administration (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 NASA's space exploration program, NASA is developing the Space Launch System (SLS) rocket that will increase the launch mass capability to 120 mT to LEO, or 40 mT to Earth escape. Even so, crewed missions to the Lunar Surface will require two SLS launches, and crewed missions to Mars will require seven or more SLS launches.

For demanding beyond-LEO missions, designers have several alternatives to developing larger and larger rockets for each new, more demanding mission. Once on-orbit, exhaust velocity (Isp) dictates mission performance, so mission planners can choose to use more efficient modes of propulsion such as nuclear thermal or solar electric propulsion to reduce the required Initial Mass in Low Earth Orbit (IMLEO). These efficient forms of propulsion may bring total system mass within the capability of available launchers at the cost of trip time or substantial technology investment. Another option for mission designers is to assemble the mission on-orbit, launched by a number of launches. This can be accomplished by joining multiple propulsion stages and payload elements as proposed in NASA's Exploration Systems Architecture Study[3]. Alternatively, the payload can be launched on a single rocket and its upper stage be fueled by pre-deployed propellant, basically a form of "simplified" on-orbit assembly. Distributed Launch is the method of launching propellant on one launch and the payload on a subsequent launch, using on-orbit propellant transfer to refuel the payload launches' upper stage to enable demanding, beyond LEO missions.

Table 1 Modern existing large launch vehicles have similar performance which limits the types of missions that can be accomplished[4].

System	Performance (mT)				
	LEO	GTO	C3=0 km^2/sec^2	C3=20 (GSO) km^2/sec^2	C3=80 km^2/sec^2
Atlas 551 or 552[5]	21	9	6.1	4.1	1.2
Delta HLV[6]	28	14	11	7.4	2.0
Falcon 9 V1.1[7]	17	5.7	3.6	1.8	-
Falcon 9 HLV	53	21	13	6.4	-
Proton[8]	21	6	5	3	-
Ariane 5[9]	21	10	7.2	5	0.6
Vulcan Centaur	22	11	7.5	5.1	1.3
Vulcan ACES	35	17	12	8	2

II. Distributed Launch

Distributed Launch is a method to efficiently combine the lift capability of two or more launches to enable beyond LEO mission capability in excess of what a single launch can accomplish. The Liquid Oxygen (LO2) and Liquid Hydrogen (LH2) propellant is launched in a disposable "Drop Tank" on the first launch, Fig. 1. This Drop Tank waits in orbit for the second launch that can be days, weeks, or even months later. The payload is launched on the second launch. Some to all of the propellant from the payload launches' upper stage is consumed to achieve orbit. The upper stage/payload combination loiters in LEO to allow orbit phasing for rendezvous with the Drop Tank. Following rendezvous, the Drop Tank and payload/upper stage fly in formation. The transfer line is robotically coupled enabling transfer of the LO2 and LH2 propellants to refuel the payload's upper stage. Once all of the propellants have been transferred the two systems separate. The upper stage/payload perform one or more main engine burns to achieve the desired trajectory. The upper stage/Drop Tank perform a deorbit burn to dispose of the Drop Tank system in Earth's atmosphere.

The performance of the distributed launch is substantially greater than that provided by today's large launch vehicles, Table 2. Distributed Launch is expected to affordably more than double the performance of available launch vehicles without the cost burden of additional fixed infrastructure, such as the cost of a permanent orbital propellant depot. Distributed Launch is ideally suited to support infrequent missions such as NRO's heavy Geosynchronous Orbit (GSO) payloads, or the support of a permanent Cis-Lunar station needing logistics missions to re-supply consumables and science experiments. These re-supply functions are very similar to Cargo Resupply Service provided to the International Space Station by Cygnus. The launch of a Cygnus to Cis-Lunar Space will require a different launch solution than to the ISS with substantially more performance than provided by Antares or Atlas V. Distributed Launch can support Cygnus resupply missions anywhere in Cis-Lunar space using commercial launch vehicles.

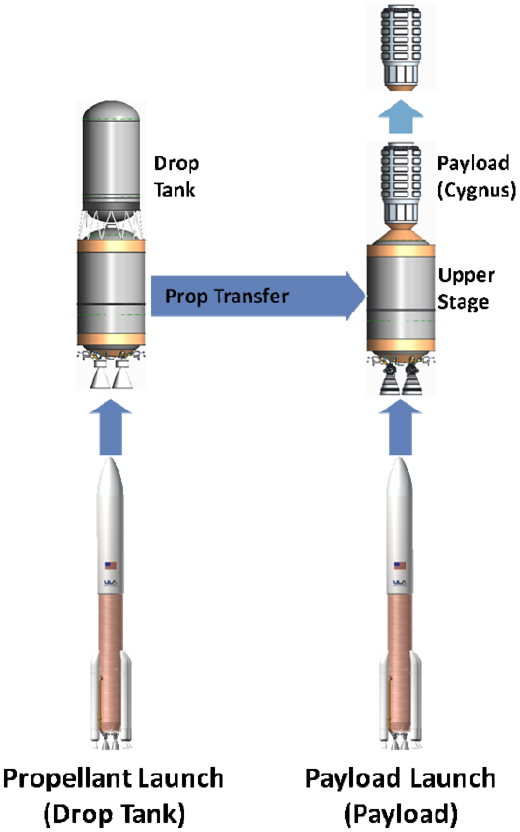

Figure 1 Distributed launch lofts propellant in a disposable "Drop Tank" on the first launch followed by launch of the payload and propulsion stage on the second launch.

Table 2 Distributed Launch can dramatically increase the throw capability to high energy trajectories as compared to single launch on equivalent class rockets.

Propellant Launch	Payload Launch	Performance (mT)		
		C3=0 km^2/sec^2	C3=20 (GSO) km^2/sec^2	C3=80 km^2/sec^2
	Delta HLV	11	7.4	2.0
A552C	A501C	8.0	7.8	2.6
A552C	A522C	14	8.6	2.9[5]
V564A	V504A	18	14	6.4
V564A	V564A	26	20	8.6

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[5] Propellant launch on an Atlas 532C provides sufficient propellant to fill the Payload launches' Centaur upper stage.

III. Launch #1: Propellant

The first launch of a Distributed Launch campaign is designed to maximize the available propellant that can be transferred to refuel the upper stage of the payload launch. To achieve this, the Drop Tank system mass and the propellant loss to boiloff during the LEO loiter must be minimized. To enable near term, affordable, Distributed Launch implementation, the described Drop Tank system is designed to take advantage of existing flight experience and production capability.

A. Drop Tank Launch

The Drop Tank is launched on a rocket with sufficient lift to provide the propellant required by the mission. The Drop Tank itself is shrouded in the payload fairing during launch. The rockets second stage expends its propellant to achieve LEO. The orbit can be tailored for each mission, but use of repeating ground track orbits will enable frequent launch opportunities with direct rendezvous of the Payload launch with the Drop Tank, Fig. 2. A low orbit provides better mission performance than a higher orbit, however, the orbit altitude must be sufficiently high to avoid atmospheric drag from causing the Drop Tank to reenter Earth's atmosphere while waiting weeks for the Payload launch. For launches out of Cape Canaveral a launch orbit of 180 nm balances these issues, providing every other day direct rendezvous and limits orbital drag to 20 nm orbit altitude loss over the course of a month.

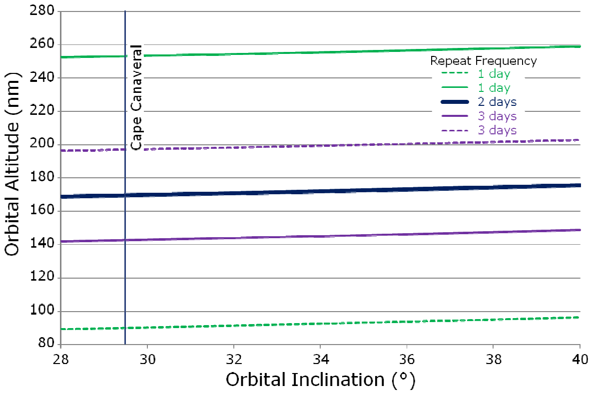

Figure 2 Launching the Drop Tank from Cape Canaveral to 180 nm provides for every-other-day Payload launch with direct rendezvous, results in less than 20 nm orbit loss due to drag, and provides efficient Earth departure performance.

B. Drop Tank Design

The Drop Tank includes the LH2 and LO2 tanks separated by a vacuum insulated common bulkhead to reduce mass and optimize thermal performance, Fig. 3. The tanks are supported above the launch vehicle upper stage via low conductivity struts to minimize heat conduction from the spent upper stage. Once on-orbit the Drop Tank remains attached to the upper stage and is placed in a slow (1^{0}/sec) transverse spin. The transverse spin provides centrifugal acceleration during the multi-week LEO loiter that allows settled Cryogenic Fluid Management (CFM) of the LH2. Settled CFM has been successfully used on all of the over 300 cryogenic upper stage missions flown globally. Keeping the Drop Tank attached to the spent upper stage pulls the system Cg aft toward the middle of the LO2 tank, providing a moment arm to the LH2 tank to enable the centrifugal LH2 settling.

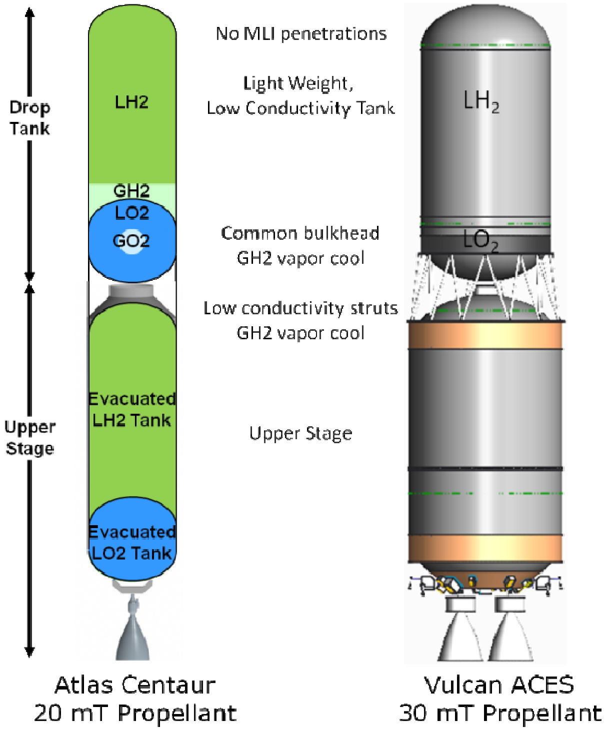

Figure 3 The Drop Tank launch is designed to maximize the propellant available for transfer to the Payload upper stage after loitering for weeks or longer on-orbit.

To keep the Drop Tank light weight, the Drop Tank is constructed of thin stainless steel and is pressure stabilized like the Centaur tank[10]. For a 20 mT class propellant load (Atlas 552 launch) the Drop Tank is a simplified Centaur tank, 10 feet in diameter, 30 feet long, built on Centaur tooling, Fig. 4. For the 30 mT propellant load (Vulcan 564A launch) the Drop Tank will be built using the 17 foot diameter Advanced Cryogenic Evolved Stage (ACES) tooling. Structural loads on the Drop Tank are very low since the Drop Tank only has to contain the LO2 and LH2 propellant without supporting additional payload weight. This allows the tank skin gauge to be reduced substantially compared to Centaur's 0.020" providing for a very light weight cryogenic storage system.

Figure 4 The Drop Tank, like Centaur's tank, will be produced from thin walled stainless steel resulting in a very light weight tank with low thermal conductivity for enhanced cryogenic storage.

The Drop Tank is specifically designed to minimize LH2 boiloff and store the LO2 without any boiloff. The system is designed to isolate elements at different temperatures, Fig. 3. The LH2 tank is the coldest element (36^0 R) and is isolated at the top of the Drop Tank system stack. The LH2 tank's only structural connection is to the moderately warm (180^0 R) LO2 tank. The LO2 tank is connected to the hot (~500^0 R) spent second stage using low conductivity struts.

Table 3 The Drop Tank is designed to efficiently store the LH2 and LO2 propellant while being lightweight and robust (mT).

	Atlas Centaur (mT)	Vulcan ACES (mT)
Launch Capability	21	35
Tank, Insulation & Plumbing Mass	0.7	1.0
Payload Truss	0.2	0.5
Mission Module	0.9	0.9
Transfer Loss (Chilldown & Residuals)	0.2	0.3
One Month LH2 Boiloff	0.6	0.9
Margin	0.3	0.8
Useable Transfer Propellant	18.1	30.5

The LH2 and LO2 tanks are separated by a vacuum insulated common bulkhead to minimize system mass and boiloff. The common bulkhead is constructed from two nested domes held apart with structural insulation, similar to Centaur construction. The thin stainless steel tank walls connecting the LH2 and LO2 tanks result in low structural heat conduction. The entire Drop Tank is shrouded in a 40 layer Multi Layer Insulation (MLI) blanket with no penetrations other than on the side facing the upper stage. The net heating of the LH2 tank results in LH2 boil-off under 0.1%/day of the total propellant load or 900 kg/month for a 30 mT propellant load launched on the Vulcan ACES.

To enable zero boil-off LO2 storage the external heating of the LO2 from solar absorption and structural heat conduction through the truss adapter must be offset with heat loss to the hydrogen. The common bulkhead separating the LO2 and LH2 tanks is designed to passively transfer about 80% of the LO2 thermal load to the hydrogen tank. The hydrogen that is vented to maintain hydrogen tank pressure is still cold and will be used to vapor cool the exterior of the LO2 tank to control the LO2 tank heating to a net zero, allowing zero boil-off LO2 storage.

Table 3 summarizes the Drop Tank system mass and available propellant to refuel the rendezvousing upper stage. The Drop Tank system mission module includes a robotic arm to connect the transfer plumbing with the Payloads upper stage and associated avionics. The propellant transfer will be pressure fed resulting in some unusable propellant due to chilldown of the propellant transfer line as well as gaseous and liquid residuals remaining in the Drop Tank.

IV. Payload Launch

Once the Propellant Drop Tank has been launched and is waiting in LEO the Payload will be launched. If the Payload is launched on a different type of rocket and from a different launch complex than the Drop Tank launch, this can take place within days of the Drop Tank launch. If the Payload and Drop Tank are launched on the same type of rocket then time must be allowed to ready the next launch. For Atlas V, launches can occur from LC-41 on 24 day centers. The Payload launch timing will be set to allow direct ascent rendezvous with the Drop Tank. For a 168 nm Drop Tank orbit, Fig. 2, the Payload launch can recycle every two days in case of weather or technical delays.

Once on-orbit the upper stage will need to perform numerous small maneuvers to support rendezvous with the Drop Tank. This maneuverability is not currently possible with any existing upper stage, but will be enabled with the Integrated Vehicle Fluids (IVF) system. Existing cryogenic stages are limited to hours of operation because the power is provided by heavy batteries. Existing stage maneuverability is limited by the available hydrazine or storable reaction control propellant. The number of main engine burns on existing cryogenic stages is limited to two or three burns by the helium available to pressurize the tanks to prevent engine cavitation.

IVF replaces the traditional upper stage batteries, helium and hydrazine systems that are dedicated to providing power, reaction control and pressurization, with an integrated system that is fed by the hydrogen and oxygen from the primary tanks. IVF consumes hydrogen and oxygen to power a small internal combustion engine. This internal combustion engine drives an alternator to provide power. IVF feeds pressurized Gaseous Hydrogen (GH2) and Gaseous Oxygen (GO2) to gimbaled thrusters to provide reaction control. IVF uses waste heat from the internal combustion engine to warm recirculated ullage gasses to pressurize the LH2 and LO2 tanks for each burn. This IVF enabled flexibility allows the Payload upper stage to maneuver and rendezvous with the Drop Tank.

Once the Payload upper stage is approaching the Drop Tank, the Drop Tank will initiate axial settling, powered by the Drop Tank upper stage's IVF GH2/GO2 thrusters and terminate its transverse roll. Axial settling at 10^{-3} g's will support cryogenic fluid management during the rendezvous maneuvers. The Payload upper stage will then maneuver alongside the Drop Tank and fly in low settled acceleration formation, Fig. 6. A robotic arm on the Drop Tank mission module will connect the flexible LO2 and LH2 transfer lines to couplings on the Payload upper stage. Formation flying prevents large, concentrated loads from needing to be reacted structurally as propellant is being transferred from the large (80 foot) Drop Tank system to the equally large Payload upper stage system. The actual cryogenic propellant transfer will be pressure fed, enabled by the IVF on the Drop Tank launch.

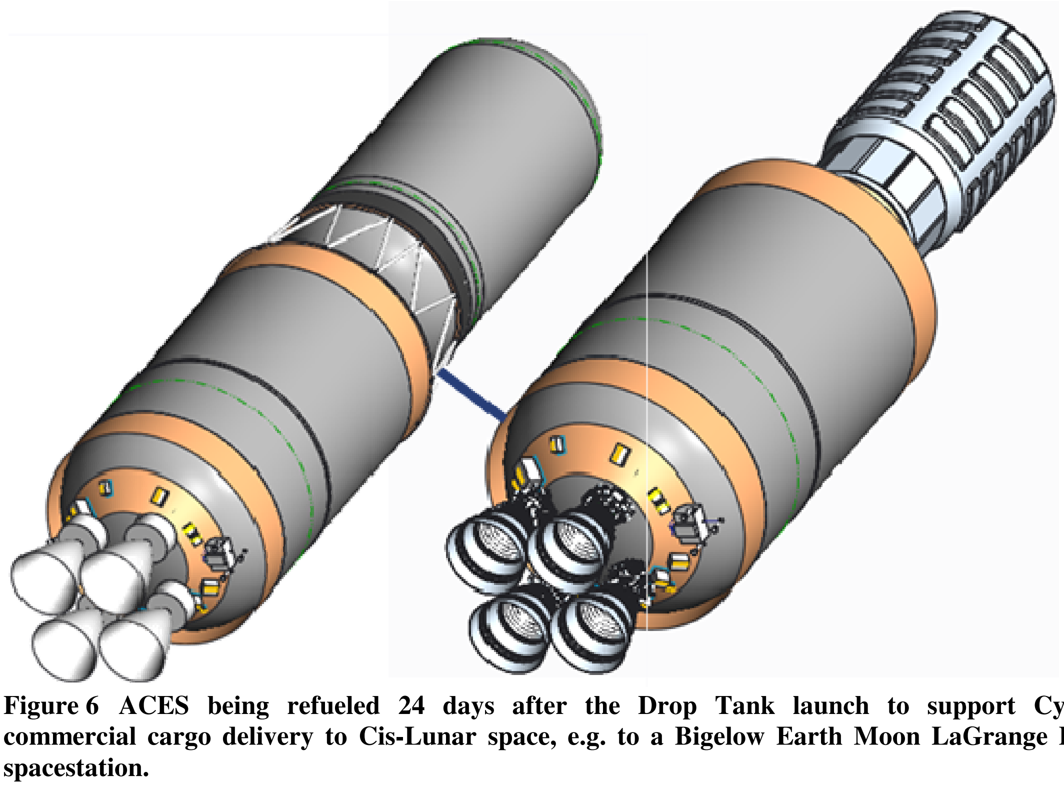

Figure 6 ACES being refueled 24 days after the Drop Tank launch to support Cygnus commercial cargo delivery to Cis-Lunar space, e.g. to a Bigelow Earth Moon LaGrange Point spacestation.

Once the propellant has been transferred, the transfer coupling will be released and the two systems will slowly maneuver away from each other. Following separation the Payload upper stage system will align for the Earth departure burn. Depending on mission needs, the Payload upper stage may continue to operate to perform midcourse corrections and deliver the payload, such as Cygnus to any location in Cis-Lunar space. The Drop Tank will align for safe disposal in Earth's atmosphere over an ocean and use IVF to consume the hydrogen and oxygen residuals for the disposal burn.

V. Key Enabling Capabilities for Distributed Launch

While distributed launch is a simple concept, it does require the advancement of a few key enabling capabilities. These primary enabling capabilities are Integrated Vehicle Fluids, Cryogenic Storage, and Cryogenic Transfer, Table 4. ULA has been working with NASA, a number of universities and several companies to mature these enabling capabilities.

Table 4 Key enabling technologies for Distributed Launch are either mature or under development.

Enabling Capabilities and Technology	Current TRL
Integrated Vehicle Fluids System	**4**
- Internal Combustion Engine	4
- H2/O2 Thruster	6
- Generator	4
- Compressor	3
- Heat exchanger	4
- Advanced computer	6
Cryogenic Storage System	**7**
- Pressure Control	9
- Low Acceleration Settling	9
- Multi-layer insulation (MLI)	9
- Vapor Cooling	9
- Mass Gauging	9
- Transfer System Chilldown	9
Cryogenic Transfer System	**5**
- Cryogenic Fluid Coupling	5
- Robotic Hose Connection	5
- Propellant Acquisition	9
- Tank Fill Operation	5
- Propellant Expulsion Efficiency	9

C. Integrated Vehicle Fluids

Integrated Vehicle Fluids[11, 12] is a breakthrough technology that transforms an upper stage into a very efficient service module capable of precise maneuvering, large ΔV's and supporting multi-day or longer mission durations. The IVF system is composed of numerous subsystems that work synergistically. IVF will replace the batteries, helium and hydrazine systems on an upper stage, simplifying the stage, reducing the cost and importantly reducing the multitude of physical mountings on the stage that drive cryogenic propellant boil-off, Fig. 7.

ULA, working with Roush, has been systematically developing the underlying technologies of IVF for 6 years. This started with an attempt by Innovative Engineering Solutions to develop a catalytic GH2/GO2 thruster. While the catalytic ignition didn't perform as desired, the thruster performed well, eventually resulting in the existing, electronically ignited, regeneratively cooled, low cost GH2/GO2 thruster, Fig. 8. Similarly, after experimenting with fuel cells, micro turbines, and Wankel engines, ULA decided to pursue an internal combustion piston engine as the power source for IVF due to the low cost and mature industrial base, Fig. 9. Early testing of the GH2/GO2 internal combustion engine powering the alternator demonstrated incredibly repeatable combustion cycle strokes compared to gasoline engines, Fig. 10.

This development culminated in a complete system cryogenic test in March of 2015. Much was learned from this testing and is being used in development of the on-going flight IVF system.

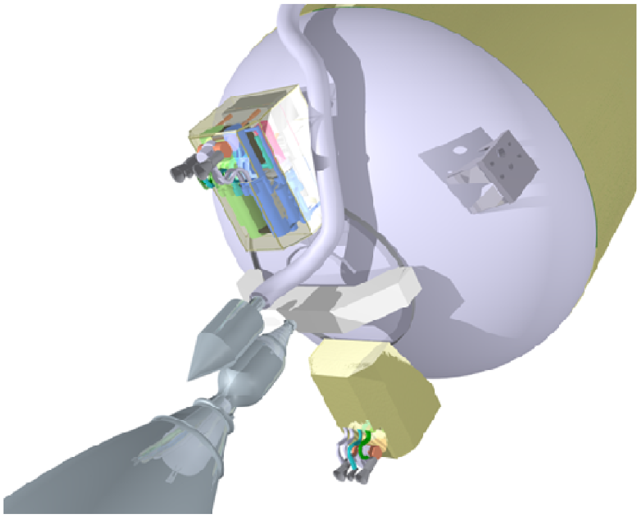

Figure 7 Twin IVF Modules mounted to Centaur's aft bulkhead provides Centaur with service module like mission capability enabling rendezvous.

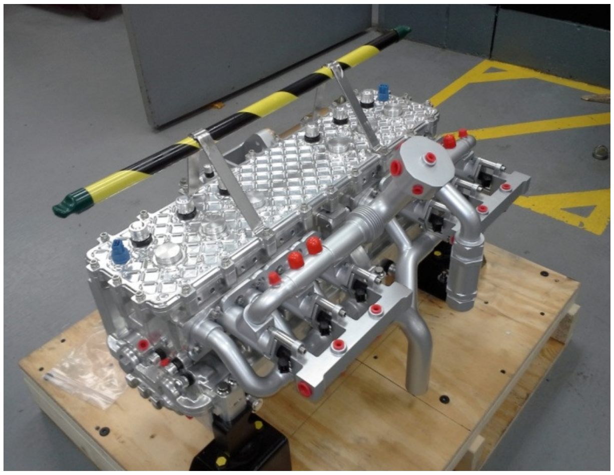

Figure 9 GH2/GO2 6 cylinder internal combustion engine developed by Roush has accumulated over 300 hours of successful run time.

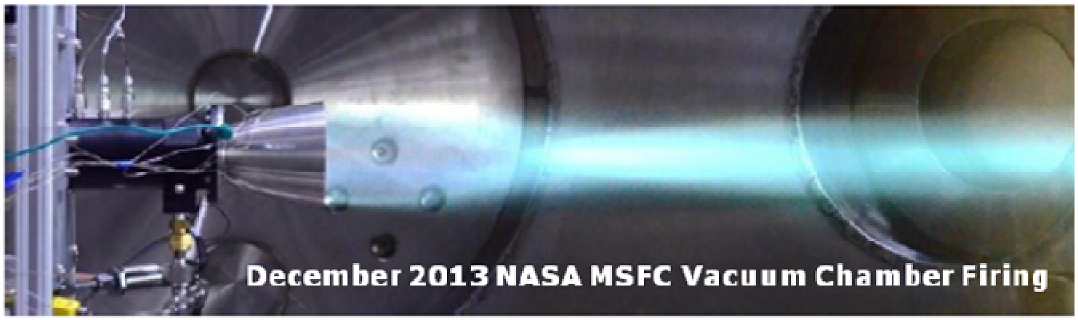

Figure 8 Testing IES's GH2/GO2 thruster in NASA MSFC's vacuum chamber.

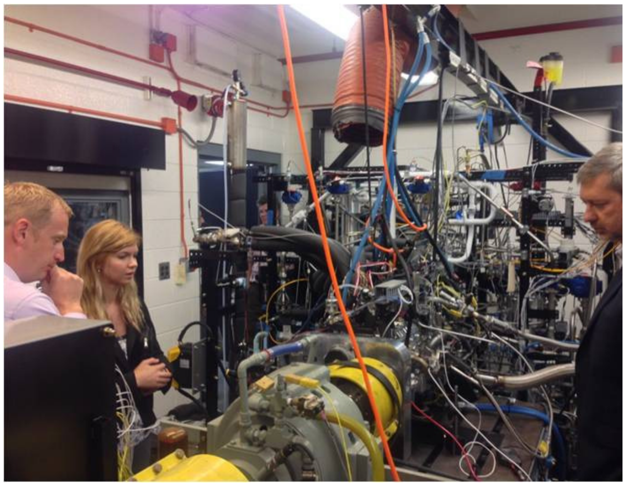

Figure 10 Testing IVF's internal combustion engine at Roush in 2013, Courtesy Roush.

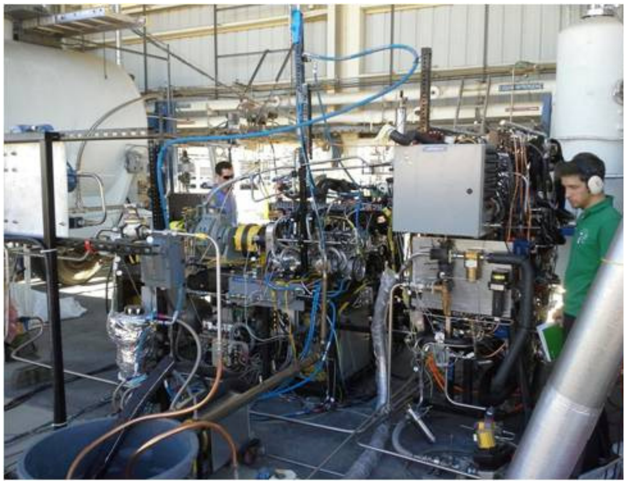

Figure 11 Integrated Cryogenic testing of the IVF system at Innovative Engineering Solutions.

D. Cryogenic Storage

Distributed Launch needs to efficiently and reliably store LH2 and LO2 for weeks in the Drop Tank. The Drop Tank design relies on lessons learned from ULA's and its heritage companies 259 cryogenic upper stage flights[13], flight cryogenic dewars, and ground testing.

Titan/Centaur was designed to support eight-hour GSO missions and demonstrated 2% per day boiloff, Table 5. By increasing Titan/Centaur's MLI blanket from 3 to 20 layers the boil-off can be reduced to 1% per day[14]. Centaur's LO2 aft bulkhead acts like an equipment shelf with hundreds of attachment brackets to support the hydrazine bottles, lines and thrusters, the helium bottles, pressurization valves, instrumentation and one or two RL10 engines. IVF consolidates most of these attachments into roughly a dozen. This reduced LO2 tank penetration, coupled with enhancing Centaur's common bulkhead insulation, is expected to reduce the boiloff to 0.3% per day. The Drop Tank eliminates the payload attach fitting on the front of the LH2 tank and eliminates the RL10 attachments and large feedlines further reducing the boiloff to less than 0.1% per day, Table 6.

To demonstrate this reduced boiloff, ULA has been working with NASA and Yetispace on the Cryogenic Orbital Testbed (CRYOTE)[15, 16] program. CRYOTE 1 included a small 30 inch diameter tank installed in an ESPA ring that would allow potential low cost rideshare flight demonstration, Fig. 12. While the flight demonstration has not occurred, substantial cryogenic testing at NASA MSFC's Exploration Flight Test vacuum chamber in 2011 provided valuable lessons regarding operations of a low boiloff system with vapor cooling. Results from this initial CRYOTE 1 testing were used to modify the vent and vapor cooling system resulting in CRYOTE 2. The CRYOTE 2 testing in 2012 provided invaluable insight into low boiloff operations with vapor cooling that are applicable to the large scale Drop Tank and cryogenic upper stages.

Table 5 Titan/Centaur demonstrated 2% per day system (LO2 and LH2) boil-off with minimal thermal protection designed for short duration geosynchronous orbit mission.

	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%	

Table 6 The Drop Tank can store LH2 and LO2 with less than 0.1% per day combined boiloff.

	TC-15		
	LO_2	LH_2	LO_2 + LH_2
Titan/Centaur experience	1.5%	4.1%	2.0 %
Minimum modifications: 20 layer MLI blanket	0.8%	2.5%	1.1%
Centaur & Reduce penetrations & Improve common bulkhead insulation	0.3%	0.7%	0.4%
Drop Tank	0%	0.7%	0.1%

Figure 12 CRYOTE 1 provided affordable cryogenic storage testing of an integrated system.

Building on this experience, ULA, NASA and Yetispace are currently outfitting CRYOTE 3 to demonstrate large scale, low boiloff cryogenic storage. CRYOTE 3 is a 10 foot diameter, Centaur derived, monocoque stainless steel, flight weight tank, Fig. 13. CRYOTE 3 will provide the scale, tank thermal mass and low conductivity wall that will allow high fidelity cryogenic testing applicable to upper stages and the Drop Tank, Table 7. CRYOTE 3 is supported by the tank's lower sidewall flange in a manner that thermally replicates the Drop Tank – Upper Stage structural connection, Fig. 14. CRYOTE 3 instrumentation will provide unprecedented clarity into the tank/cryogenic fluid interaction, heating and fluid stratification through the use of Fiber Optic Sensing System (FOSS), Fig. 15. This enhanced knowledge of the behavior within the cryogenic system will help mature ULA's analytic models that will be used to support actual Distributed Launch mission modeling.

Parameter	Value
Material	Stainless steel
Diameter	10 ft
Length	10 ft
Wall Thickness	0.02 in
Dome Geometry	Elliptical
Surface Area	330 ft2
Volume	580 ft3
Empty Mass	300 lbm
Full Mass – LN2	32,500 lbm (14,700 kg)
Full Mass – LO2	46,000 lbm (20,900 kg)
Full Mass – LH2	3,100 lbm (1,400 kg)
Structural Interface	Cylinder Rings
Support equipment	Cylinder structure
Penetrations	Minimal top and bottom
Manufacturer	ULA

Figure 13 CRYOTE 3 is a 10 foot diameter, Centaur derived flight qualified tank designed to demonstrate high fidelity low boiloff cryogenic storage and transfer.

Table 7 CRYOTE 3 is designed to provide the scale and cryogenic storage properties similar to upper stages and the Drop Tank to enable high fidelity ground cryogenic testing applicable to Distributed Launch.

Test Bed	Material	Liquid	Diameter (m)	Length (m)	Wall	Volume (m3)	Chill Energy (MJ)	Energy per Volume (MJ/m3)
MHTB	Aluminum	LN2	3.0	3.0	Thick smooth	18	220	12
CTB2	Stainless Steel	LN2	1.8	2.0	Thick smooth	4.0	120	30
VATA	Stainless Steel	LN2	1.2	1.6	Thick smooth	1.4	24	17
CRYOTE 1	Titanium	LN2	0.76	0.76	Thin smooth	0.23	1.3	5.4
CRYOTE 3	Stainless Steel	LN2 LH2	3.0	3.0	Thin smooth	16	11 14	0.69 0.87
Centaur LH2	Stainless Steel	LH2	3.0	7.7	Thin smooth	49	42	0.85
Centaur LO2	Stainless Steel	LO2	3.0	2.9	Thin smooth	16	27	1.7
5m DCSS LH2	Aluminum	LH2	5.0	4.3	Isogrid	64	193	3.0
5m DCSS LO2	Aluminum	LO2	3.2	3.6	Isogrid	22	80	3.7

Figure 14 CRYOTE 3 is supported by a ring connected to the bottom tank flange, equivalent to the interstage adapter connecting the upper stage and booster or the Drop Tank truss adapter.

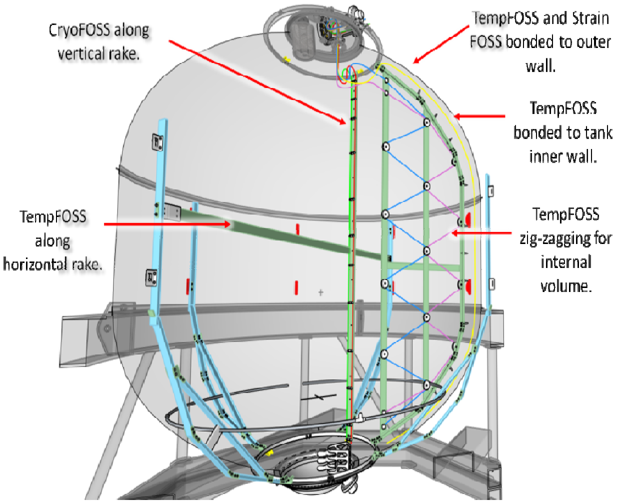

Figure 15 CRYOTE 3 will provide unprecedented resolution of the heat transfer process thanks to the fiber optic sensing system. This will help anchor analytic tools that will be used for flight missions.

E. Cryogenic Transfer

Efficient, reliable cryogenic propellant transfer is the corner stone of the Distributed Launch concept. NASA and others pioneered methods for transferring cryogenic fluids in space in the 1990's, including no-vent-fill[17, 18]. The challenge of cryogenic transfer in space revolves around efficient fluid acquisition in the donor tank and minimum loss injection into the receiving tank.

While liquid acquisition can use propellant management devices that employ surface tension to collect liquid at the tank outlet, these propellant management devices tend to be heavy, especially for large tanks. Cryogenic liquids also have very low surface tension making the use of propellant management devices a challenge. Low level settling is an alternative that has been used very effectively on all cryogenic upper stage flights to feed liquid to the main engines and will be used for the fluid acquisition in the Drop Tank during liquid transfer[19].

Liquid injection into a tank typically requires venting of the gas that is in the tank and any evolved gas during the fill process. Under low acceleration in space, the momentum of the entering liquid will result in a liquid gas mixture from which liquid free venting is a challenge. Venting liquid results in a substantial performance penalty and unbalanced forces that can make the transfer of liquid between two vehicles very challenging. No-vent-fill avoids these difficulties by utilizing the thermodynamic properties of cold liquids to condense the vapor in the tank enabling filling the tank without venting.

Yetispace developed a low cost apparatus to investigate the driving properties behind no-vent-fill using the CRYOTE 2 and CPST VATA hardware, Fig. 16. During 2014 Yetispace transferred Liquid Nitrogen (LN2) back and forth between the VATA and CRYOTE 2 tanks using multiple vent schemes, liquid injection "shower heads", and tank pressures. From this testing Yetispace determined that the most important parameter governing no-vent-fill liquid transfer was the heat transfer between the liquid entering the receiving tank and the ullage gas. This heat transfer is driven by the incoming liquid stream pattern. With this knowledge Yetispace has reliably demonstrated the transfer of cryogenic liquid under flight like conditions (less the zero g) with 100% fill of the receiving tank. Though this testing is being conducted in a 1-g environment here on Earth, the results are conservative relative to space. This is because at 1-g, gravity quickly settles the incoming stream to the liquid pool at the bottom of the tank.

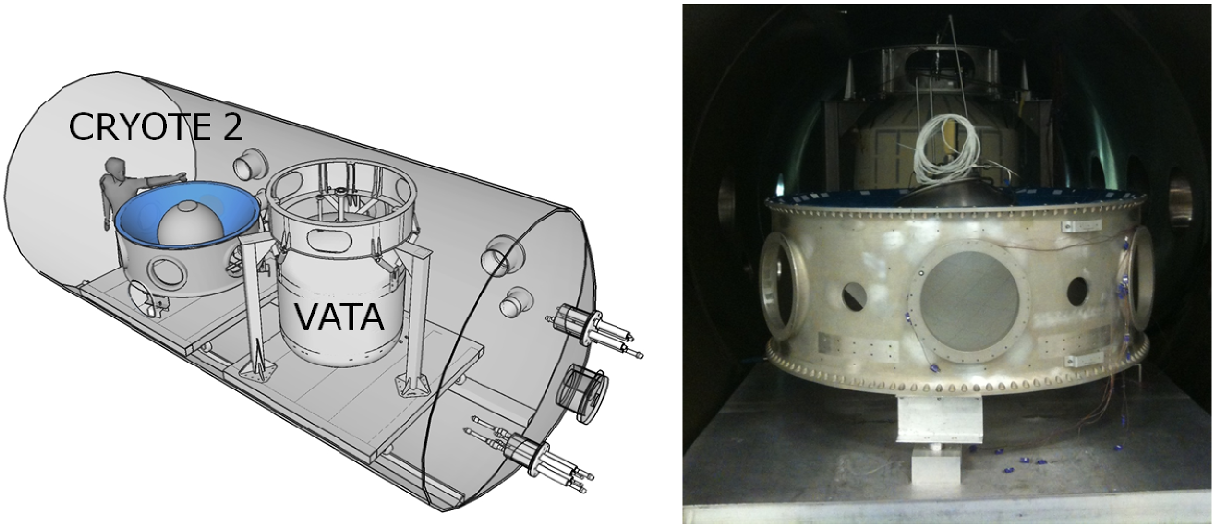

Figure 16 CRYOTE 2 and CPST's VATA have been used to develop techniques for transferring cryogenic liquids using No-Vent-Fill. This testing has shown reliable 100% fill using flight like transfer conditions.

Under low acceleration, in space, the momentum of the incoming stream will tend to cause the liquid/gas to continue to move around the tank and interact, resulting in greater liquid/ullage heat transfer.

These results will be applied to cryogenic filling of the CRYOTE 3 tank to anchor transfer models. These models will be used to support the flight cryogenic propellant transfer supporting Distributed Launch.

VI. Summary

Distributed Launch efficiently combines the launch capability of two rockets to enable beyond LEO performance over double what a single rocket can accomplish. Distributed Launch can provide the National Reconnaissance Office with the ability to emplace huge national security payloads in geosynchronous orbit. The use of Distributed Launch can enable Cygnus to provide Commercial Resupply Services supporting space stations, such as proposed by Bigelow, anywhere in the Cis-Lunar system. Distributed Launch can also enable commercial human missions to the Lunar Surface as proposed by Golden Spike.

ULA is currently developing the critical technologies (Integrated Vehicle Fluids, Cryogenic Storage and Cryogenic Transfer) that will enable Distributed Launch to support national security missions and the migration of humans beyond Earth.

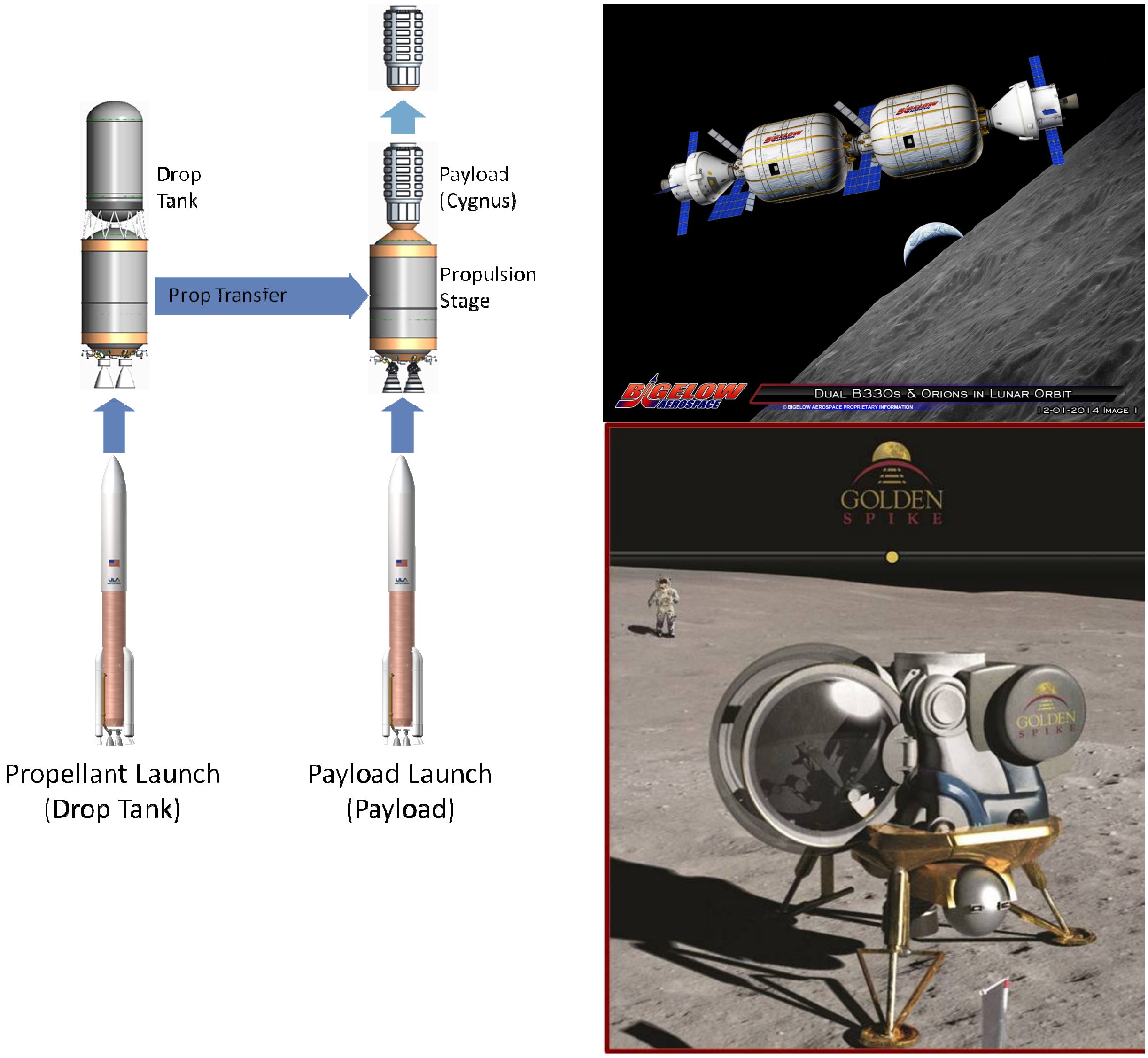

Figure 17 Distributed Launch opens the heavens to help dreams become reality.

VII. References

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