

Advanced Chemical Propulsion for Science Missions

Larry Liou
NASA In-Space Propulsion Technology (ISPT) Project Office
NASA John H. Glenn Research Center
Mail Stop 142-2
21000 Brook Park Rd.
Cleveland, OH 44135
216-977-7433
Larry.C.Liou@nasa.gov

Abstract—The advanced chemical propulsion technology area of NASA’s In-Space Technology Project is investing in systems and components for increased performance and reduced cost of chemical propulsion technologies applicable to near-term science missions. Presently the primary investment in the advanced chemical propulsion technology area is in the AMBR high temperature storable bipropellant rocket engine. Scheduled to be available for flight development starting in year 2008, AMBR engine shows a 60 kg payload gain in an analysis for the Titan-Enceladus orbiter mission and a 33% manufacturing cost reduction over its baseline, state-of-the-art counterpart. Other technologies invested include the reliable lightweight tanks for propellant and the precision propellant management and mixture ratio control. Both technologies show significant mission benefit, can be applied to any liquid propulsion system, and upon completion of the efforts described in this paper, are at least in parts ready for flight infusion. Details of the technologies are discussed.^{1 2}

ΔV	=	Delta Velocity
E-Beam	=	Electronic Beam (welding)
GEO	=	Geosynchronous Earth Orbit
GRC	=	NASA Glenn Research Center
GTO	=	Geosynchronous Transfer Orbit
HiPAT™	=	Trade name for Aerojet R-4D-15 engine
Ir/Re	=	Iridium lined Rhenium material system
Isp	=	Specific Impulse
ISPT	=	NASA In-Space Propulsion Technology Project
JPL	=	NASA Jet Propulsion Laboratory
kg	=	Kilogram
MER	=	Mars Exploration Rover
MDP	=	Maximum Design Pressure
MMH	=	Monomethylhydrazine, N ₂ H ₃ CH ₃
MR	=	Mixture Ratio, propellant (synonymous With OF)
MSFC	=	NASA Marshall Space Flight Center
NDE	=	Non-Destructive Evaluation
N ₂ H ₄	=	Hydrazine, N ₂ H ₄
NRA	=	NASA Research Announcement
NTO	=	Nitrogen Tetroxide, N ₂ O ₄
OMG	=	Optical Mass Gauging
OF	=	Oxidizer to Fuel ratio
Pc	=	Chamber pressure
PMD	=	Propellant Management Device
psia	=	Pounds per square inch absolute
Re	=	Rhenium
TLVI	=	Tank Liquid Volume Instrument
T-E	=	Titan-Enceladus

TABLE OF CONTENTS

1. NOMENCLATURE.....	1
2. INTRODUCTION.....	1
3. HIGH TEMPERATURE, STORABLE, BI-PROPELLANT ROCKET ENGINE (AMBR ENGINE).....	2
4. LIGHT WEIGHT TANKS FOR PROPELLANT.....	5
5. PRECISION PROPELLANT MANAGEMENT AND MIXTURE RATIO CONTROL.....	7
6. CONCLUSIONS	9
7. ACKNOWLEDGMENTS.....	9
8. REFERENCES	10
9. BIOGRAPHY	10

1. NOMENCLATURE

AMBR	=	Advanced Materials Bipropellant Rocket
BFM	=	Balanced Flow Meter
COPV	=	Composite Overwrapped Pressure Vessel
CVD	=	Chemical Vapor Deposition

2. INTRODUCTION

Chemical propulsion has provided the basis for rocket system transportation for decades. As NASA prepares for future space exploration, the Agency continues to improve and develop new chemical propulsion systems. The effort ultimately focuses on providing greater capability for space science missions by reducing the launched mass, and cost of spacecraft and operation. Currently, the advanced chemical propulsion projects aim at increasing payload capacity and improving reliability for scientific missions.

Seeking to fulfill these goals, the ISPT Program, which is managed NASA’s Science Mission Directorate in Washington D.C. and implemented by the ISPT Project

¹ IEEEAC Paper # 1482; Version 2, Updated December 6, 2007

² U.S. Government work not protected by U.S. copyright

Office located at NASA John H. Glenn Research Center, Cleveland, Ohio, is investing in systems and components for increased performance and reduced cost of chemical propulsion technologies applicable to near-term science missions. Presently the primary investment in the advanced chemical propulsion technology area is in the AMBR high temperature storable bipropellant rocket engine. The other two technologies also invested include the reliable lightweight tank for propellants and the precision propellant management and mixture ratio control.

All three efforts are discussed in terms of their mission application, mission benefit, and date of availability for flight mission, present status, and technical description.

Not discussed here are the previously invested technologies such as gel propellants, advanced monopropellants, foam core shielding for spacecraft, etc. [1], [2]

3. HIGH TEMPERATURE, STORABLE, BI-PROPELLANT ROCKET ENGINE (AMBR ENGINE)

This task effort is of evolutionary nature and its approach is to improve the present state-of-the-art workhorse engines in order to gain further payload benefit. AMBR engine is the result of such an effort, designed to yield improved performance such as Isp and thrust, and reduced manufacturing cost. AMBR is a new engine based on Aerojet's R-4D-15, dual mode, apogee thruster [3]. This newly designed Ir/Re, NTO/N2H4 and NTO/MMH engine can be ready for science mission development as early as year 2009.

Mission Application

AMBR engine, shown in Figure 1, can be used as a main or secondary propulsion system for a wide range of NASA science missions from Discovery, New Frontiers, to the Flag Ship class. In addition, it can be utilized for orbital maneuvers for a satellite or upper stage rocket.

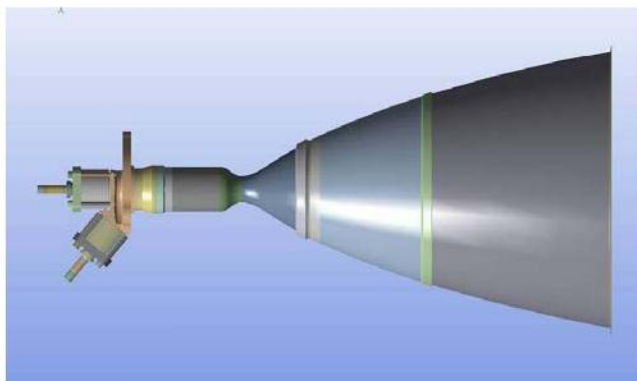


Figure 1 – A rendered image of the AMBR Engine in Development

Mission Benefit

With increased performance (Isp, specific impulse), AMBR engine can reduce the propellant mass required to perform spacecraft maneuvers. See Table 1 and Figure 2 which follow for the mass benefit figure associated with various missions. This propellant mass saving can directly translate to an increase in scientific payload and thus data gathering capability of the spacecraft.

Table 1 – AMBR engine mass saving for various Isp

	Total Propulsion System Mass Reduction (Kg)				
	Isp	320	325	330	332.5
GTO to GEO	0	16	30	37	45
Europa Orbiter	N/A	0	12	16	24
Mars Orbiter	N/A	0	14	22	29
T - E Orbiter	N/A	0	29	45	60

The following bar chart, Figure 2, highlights the payload, or mass benefit at the target Isp of 335 sec for NTO/N2H4 propellant combination.

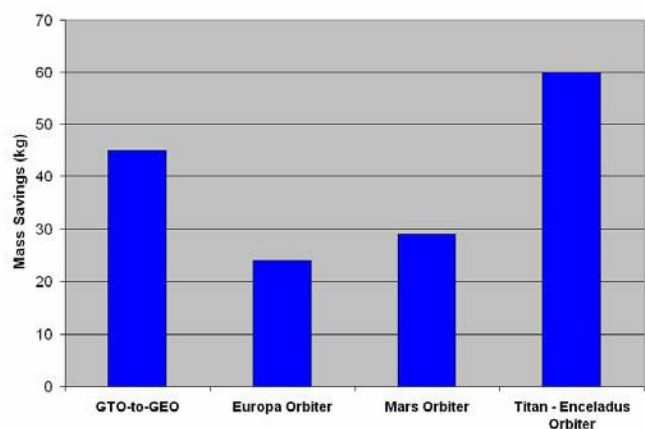


Figure 2 – Mass benefits for various science missions of the AMBR engine at Isp of 335 sec

In addition, because of AMBR's higher thrust, for a mission such as Cassini, the number of thrusters can be reduced, which further reduces the system mass and complexity. The following Table 2 lists the AMBR engine characteristics as compared to the state-of-the-art baseline HiPAT engine.

Table 2 – AMBR Engine Compared with State-of-the-Art HiPAT Dual Mode Engine (NTO.N2H4)

Design Characteristics	AMBR	HiPAT DM
Trust (lbf)	200	100
Specific Impulse (sec)	335	329
Inlet Pressure (psia)	400	310-220
Chamber Pressure (psia)	275	137
Oxidizer/Fuel Ratio	1.2	1.0
Expansion Ratio	400:1	375:1
Physical Envelope	Within existing HiPAT envelope	
Propellant Valves	Existing R-4D Valves	

Availability Date

AMBR engine is scheduled to be ready for mission development beginning year 2009. At that time the engine can be prepared for flight qualification.

Status

Baseline hotfire testing has been completed which yielded data for designing the new AMBR engine. More information related to the hotfire testing is given under the following subheading *Technical Description*.

New engine design is currently underway which includes injector optimization, chamber/nozzle contour optimization, chamber emissivity reduction, and thermal resistance increase between the injector and chamber.

Alternate chamber fabrication processes are being investigated for further cost reduction and performance advantages

Option I engine for NTO/N2H4 is to undergo testing in May 2008.

Technical Description

AMBR engine developmental goal is to increase the Isp of the pressure-fed, earth-storable bipropellant, apogee rocket engine to at least 330 seconds with NTO/MMH propellants and at least 335 seconds with NTO/N2H4 propellants. State of the art Isp are approximately 323 and 328 seconds for the respective propellant combinations.

Increasing engine Isp lengthens the run time of a given amount of propellant, or, inversely, it reduces the amount of propellant required to perform a given spacecraft maneuver. This propellant reduction can be applied to increasing the life or capability of spacecrafts. These gains are discussed

in more detail under the subheading *Mission Benefit* of this section.

The maximum theoretical Isp of the propellant combination NTO/N2H4 has never been reached in a flight engine partially due to the temperature limitation of the rocket chamber material. Therefore, if improvements are made such that propellants are allowed to combust at a higher temperature and pressure, then better engine performance would occur. Certainly, this increase in operating temperature needs to stay within safety margins critical to mission integrity.

On the other hand, although routinely used in space flight, the Ir/Re engine chambers are yet to be operated at temperatures high enough to fully exploit the materials' capability of 4,000 degree F. Higher operating temperature would require further optimization of Ir/Re engine materials, design, and manufacturing—tasks that are the content of this project. This ISPT project aims to push the bi-propellant engine operating temperature higher and thus increase the engine's performance. This is to be achieved by expanding the operating envelope of Aerojet Corporation's flight-proven, Ir/Re, liquid apogee engine R-4D-15--an example of which during hotfire testing is shown in Figure 3.

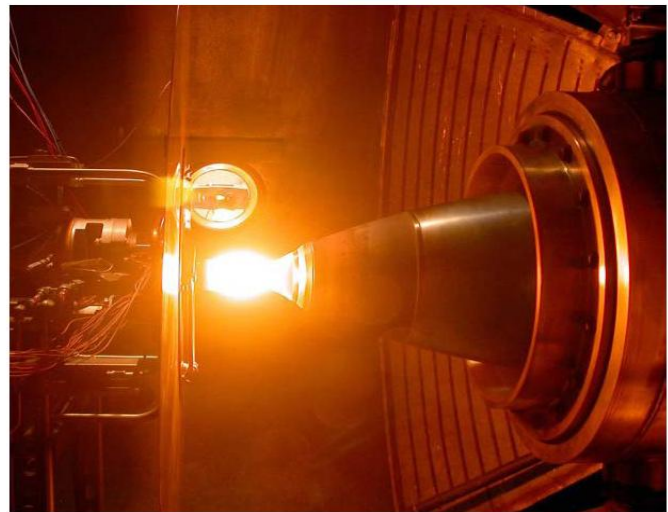


Figure 3 – Aerojet’s Ir/Re, Liquid Apogee Engine Firing

Expansion of the operating envelope is accompanied with design changes made to the baseline engine which are shown in Figure 4. Changes include injector and chamber/nozzle contour optimization, reduced chamber emissivity, and increased thermal resistance between the injector and chamber. Engine operating conditions are also be modified to produce higher combustion gas temperatures. These include higher feed pressure/lower internal pressure drop and higher/optimized mixture ratio.

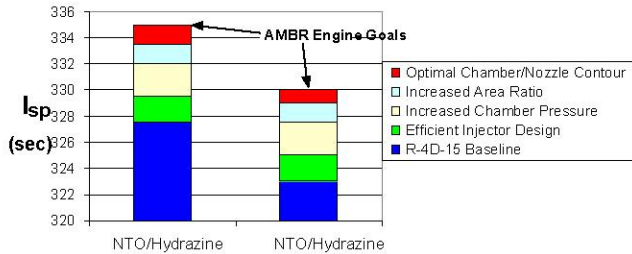


Figure 4 – Design Changes Made to the Baseline Engine for Achieving the AMBR Engine Performance Goals

Again, shown previously in Table 2 is a comparison of the AMBR engine characteristics to those of the Aerojet Corporation’s HiPAT™ engine which is the trade name for the R-4D-15 engine.

As mentioned in the *Status* section, baseline testing was successfully done using the HiPAT™ engine. The purpose of the baseline test was to increase knowledge base prior to design and test the new, high performance engine for this program. And indeed, data collected from the baseline test has been used for the thermal analysis and as design reference for the new engine which was to be built and tested by first half of 2008.

The baseline test covered a range of chamber pressures and mixture ratios, which are plotted as “NASA NRA” points in Figure 5 along with other tested conditions for the HiPAT™ engine. During the 26 hot fire runs totaling 2,909 seconds and consuming more than a thousand pound of NTO/N2H4 propellant, the engine was run purposely both “harder” and “softer” than the design conditions. “Harder” means higher propellant feed pressure or hotter mixture ratio and “softer” means the opposite.

A secondary project goal is to investigate the viability of alternate Ir/Re fabrication processes and other related material systems to determine whether alternate processes offer cost, producibility, or performance advantages over the baseline CVD fabrication process. If one of these alternate processes is found to be of sufficient value and level of development, it would be incorporated into the engine design. [3]

Specifically, the effort investigates better iridium liner forming and rhenium deposition methods, e.g., engineered electroform (EL-Form) with increased allowable tensile stress limits, to further strengthen the chamber materials system for operation at higher temperature and pressure and to reduce production cost [4]. EL-Form process has been demonstrated for fabricating high density rhenium components, and has been applied to fabricate thrust chambers. The process produces high purity material with

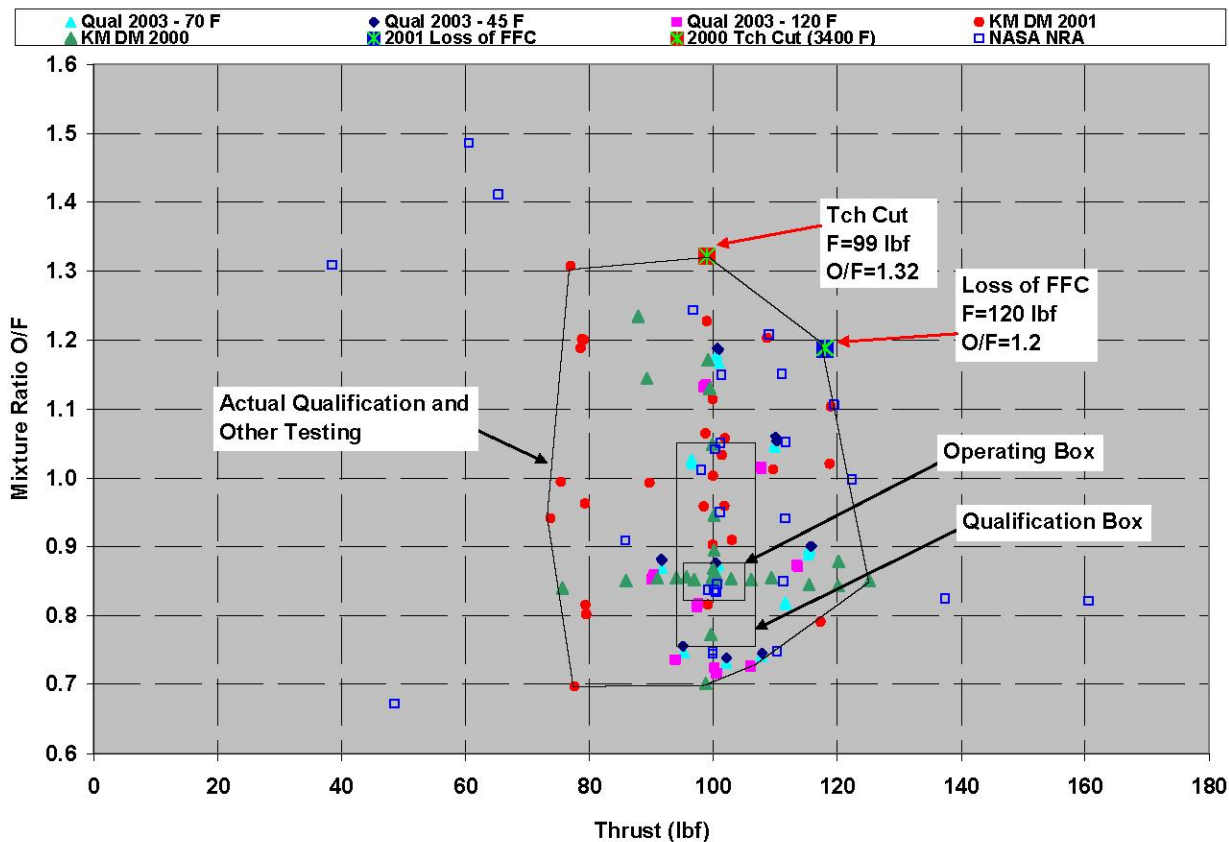


Figure 5 – Baseline test points (NASA NRA) plotted along with other previously tested conditions for the HiPAT™

stable and reproducible properties, but it also decreases the room temperature yield strength. To improve from there, the “engineered” EL-Form process is investigated which deposits multiple layers of rhenium that impede grain growth during deposition and annealing. Figure 6 shows micrographs of the engineered EL-Form Re samples.



Figure 6 – Micrographs of Engineered EL-Form Re in as-deposited (left) and annealed conditions

Presently, the engineered EL-Form process has shown to improve the rhenium room temperature yield strength by 150%. However, the post process, elevated temperature and thermophysical properties of Re were yet to be investigated. More general process development and repeatability demonstration must be performed prior to consistent commercial application.

4. LIGHT WEIGHT TANKS FOR PROPELLANT

On a spacecraft, propellant tanks are often the single largest dry mass component, and thus making them light in weight while maintaining reliability and safety can bring immediate payload benefit. Presently, state-of-the-art light weight tanks are either all-metal or metal-lined and composite-overwrapped--an example for the latter is shown in Figure 7. Further improvement can make the tanks even lighter in weight—such is the goal of our present effort.



Figure 7 – An example of metallic lined propellant tank overwrapped with composite fibers

Mission Application

Light weight tanks can be used in all liquid propulsion systems using either storable or cryogenic propellants. They can be applied to reduce propulsion system dry mass for all classes of science missions.

Mission Benefit

Improvements in the light weight tanks could result in substantial reduction of the overall propulsion systems

weight and allow for more payload and scientific instrumentation, resulting in greater scientific return.

According to an analysis [5], a mass saving of 20 kg, or payload increase, can be realized by using a lightweight, aluminum lined tank with liner thickness of .005” (assuming a tank for 500 kg of NTO). And for a larger mission such as the Titan-Enceladus orbiter mission (assuming 1500 kg of NTO), a payload gain of 50 kg can be achieved [6].

Availability Date

Availability date of the light weight tanks is beyond year 2009, but the component technologies and manufacturing processes and inspection techniques developed for liner welding and for composite over wrapping and bonding to the liner may be available sooner depending on the nature of application.

Status

Present effort focuses on improving the constituent technologies including the Electronic-Beam (E-Beam) welding of the metallic liner and overwrapping and bonding of the composite fibers to the metallic liner. The approach consists of coupon testing and flight-like tank fabrication and testing. Task results are to be documented and reported by end of the task in 2008. Encouragingly, preliminary results indicate strong E-Beam welds with ultimate tensile strength 60% of the parent aluminum material. Also, interim result shows the promising bonding strength of the tested adhesive exceeding 7,000 psi of tensile stress.

At time of this writing, funding for this task is interrupted and there is no plan for continuing development of the technology with the ISPT program beyond March 2008.

Technical Description

A state-of-the-art propellant tank is usually made from titanium alloy—a high strength, low weight metal alloy that is 40 percent lighter than steel and has high resistance to corrosive environments, such as salt air. However, new light weight tanks could offer not only the same level of strength and corrosion resistance, but further reduce propellant tank mass by as much as 50% as compared to all-titanium tank [7]. Light weight tanks can consist of metallic liners with thickness of .01 inch or less, over wrapped with composite material, and the reduced liner thickness coupled with optimized design yields “lightweight” COPV. Taken from [7], Figure 8 shows the dependence of tank mass (normalized to .03 inch thickness) to the liner thickness. In the figure one can see that for an aluminum lined tank, reducing the liner thickness to .005 inch can reduce the tank mass by 50%. This mass reduction translates to tens of kilograms of payload gain for science missions.

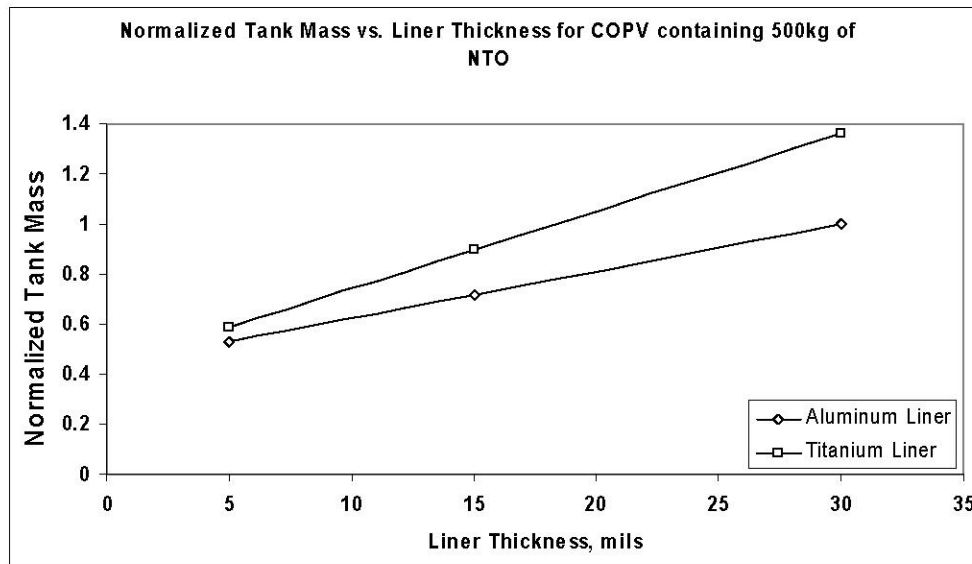


Figure 8 – Tank mass vs. liner thickness for COPV

Currently liquid chemical propellants are often stored in titanium tanks [8]. For example, hydrazine was stored in the titanium tanks, each weighing 5.8 kg dry mass, for the Mars Pathfinder mission. But later, the Mars Exploration Rover (MER) mission wanted a hydrazine tank with reduced mass.

Unfortunately, tank fabrication challenges prevented MER mission from flying these lightweight tanks. However, while JPL subsequently continues to validate this technology for future flight programs, ISPT took the MER tanks through rigorous testing. Then, in the present follow-on task, it continued to address the high risks in liner welding, composite wrapping and bonding, and NDE testing.

Table 3 – MER Propellant Tank Design Requirements

PARAMETER	REQUIREMENTS
MDP	378 psig
Proof Pressure	501 psig (MDP x1.25)
Burst Pressure	624 psig
Pressure Cycle	5 Proof, 1 MDP x1.1, and 7 MDP cycles
Operating Temperature	-20 to 40° C
Non-Operating Temperature	-40 to 55° C
Propellant Load	35 kg max.
Leakage	1 x10 ⁻⁵ scc/s He at MDP
Diameter at MDP	16.4 inches max.
Overall Length at MDP	18.9 inches max.
Minimum Internal Volume	2600 in ³
Maximum Weight (with PMD)	5.10 lbs
Mounting Configuration	Boss Mounted

While state-of-the-art COPV's are frequently used for high-pressure gas storage and they have been flown successfully since the mid-1980's, in order to meet MER's needs for mass reduction, lightweight tanks were designed with very thin liners overwrapped with a high strength, low-density fiber, polybenzoxazole (PBO) and epoxy matrix resin. The resultant tanks would have a dry mass of only about 2.0 kg. These tanks are referred here as the "MER tank." Its design parameters are listed in Table 3.

Reference [8] gives a detailed account of the modeling, analyses, design, fabrication, and testing of MER tanks. Only the validation testing is highlighted here. The test sequence was:

1. Proof test after end-cap welding
2. Helium leak test
3. Volume measurement
4. 4X Pressure Cycle Life Test
5. Post-cycle Test Proof Test
6. Post-cycle Test Helium Leak Test
7. Random Vibration 3 Axes
8. Quasi Static Acceleration (Sine Burst) 2 Axes
9. Post-dynamic Proof Test
10. Post-dynamic Helium Leak Test
11. PMD Bubble-point Test
12. Burst (pressure) Test

Thus far, three (3) tanks have been subject to the above test, but during the Step 6 Post-cycle Test Helium Leak Test, one tank was found to have a leak. The leak was believed to be in the location of the central girth weld of the aluminum liner.

The subsequent task presently is a one-year effort to push forward state-of-the-art of the constituent technologies: liner forming, welding, fiber overwrapping and bonding, and testing. The goal is to develop consistent

manufacturing processes that yield reliable tanks with a maximum liner thickness of .010" and a high degree of manufacturing repeatability with reduced dropout rates.

The liners, which provide a hermetic seal for propellants while offering attractive mechanical properties desirable for manufacturing, are flow formed using 1100-series aluminum instead of the 6061-series aluminum used on the MER tanks. The 1100-series aluminum offers more superior weld quality and other properties that minimize crack growth from flaws. To provide a more consistent weld, E-Beam welding is being investigated for the metallic liners and components. Also, to improve overwrap load bearing capacity, a bonding study is underway to explore alternative adhesives and resins to increase bond strength and decrease the potential for debonds. By providing this multi-pronged risk mitigation approach, the program aims to standardize manufacturing processes and increase confidence in lightweight and ultra-lightweight COPVs for infusion into future missions for science and human exploration.

5. PRECISION PROPELLANT MANAGEMENT AND MIXTURE RATIO CONTROL

Researchers are developing ways to measure and utilize liquid propellants in propulsion systems with higher quantity-precision, in order to ultimately gain payload mass, better engine performance targeting, and improved system safety. This task focuses on testing components and algorithms using mainly a laboratory simulated propellant feed system. Example component technologies are Balanced Flow Meter (BFM), Optical Mass Gauging (OMG), and Tank Liquid Volume Instrument (TLVI). These technologies plus an improved pressurization scheme can help reduce uncertainties and the need for excessive amounts of propellant reserve of which the mass fraction can otherwise be used for science payload. They allow better control of the engine oxidizer-fuel mixture ratio and hence better engine performance targeting. They can also allow detection of small leaks to improve system safety.

Mission Application

This technology is applicable to all liquid propulsion systems, storable or cryogenic, for planetary spacecraft, Earth orbiting satellite, and launch vehicle applications.

Mission Benefit

Study results [9] indicate that science payload mass could be increased by 10% to 50% by accurately controlling propellant flow. This accurate flow control and measurement can be achieved with a combination of the Balanced Flow Meter (BFM), other advanced sensor technologies, and improved pressurization methods. Figure 9 shows an image of the BFM.



Figure 9 – Balanced Flow Meter (BFM) enables precision propellant flow measurement

Availability Date

In part or as whole, variation of the precision propellant management system may become available beginning in year 2010 which can, incrementally through time, offer increasing benefits over state-of-the-art equipment.

Status

At this writing, the task effort investigates a number of component technologies that can be used for precision and active propellant pressurization and mixture ratio control. Accuracy is verified via statistical testing for components such as balanced flow meter (BFM), tank liquid volume instrument (TLVI), and optical mass gauging (OMG).

Using a simulated propellant feed system and benign fluids, new pressurization and flow control schemes are tested. The simulated propellant feed system is shown in Figure 10.

Unfortunately, at time of this writing, the funding for this task is interrupted and there is no plan for continuing development of the technology with the ISPT Program beyond March 2008.

Technical Description

The objective of this task is to achieve substantial payload gains by reducing the 4% to 6% propellant reserves typically carried by spacecraft [10]. It is the desire of this task to minimize the propellant reserves which in turn reduces the vehicle mass. The reserves are provided to protect against propellant mixture ratio variations and gauging inaccuracies which occur in propellant loading, mass gauging, and variations in engine throttle rates in both oxidizers and fuels. Optimizing propellant usage and reducing reserves drive the need for an accurate control of propellant tank ullage pressure and mixture ratio. When propellant amount and consumption are controlled and measured accurately, propellant residuals can be reduced.



Figure 10 – Photograph of simulated propellant feed system.

The motivation for this work is to reduce the amount of propellant reserve for in-flight consumption variance. Precisely controlling the propellant amount in loading and in usage by means of precision tanking and flow rate control and measurement (also known as “mixture ratio control”) can reduce excessive onboard propellant reserve. If this propellant reserve could be cut in half, the mass savings would be significant: It would be equivalent to a 10% reduction in propulsion hardware mass, and this mass reduction translates to a 15 to 50% scientific payload gain. For example, according to a NASA ISPT study performed to determine the potential benefits of implementing an active mixture ratio control system for existing and future missions, payload gain for the Jupiter Polar Orbiter with Probes (JPOP) mission can be 40 kilograms, representing a 13% increase; and for the Mars Sample Return (MSR) mission the payload gain can be 60 kilograms, representing a 48% increase. Propulsion hardware mass reduction of this magnitude, performance-wise, is similar to the highest payoffs in hardware technology for space propulsion mass reduction.

Initial project tasks are to assess the potential of active propellant mixture ratio control to determine whether this is a promising area for technology investment. Propellant pressurization feed system control algorithms are also studied using advanced statistical method to assess their performance and impact on reduction of in-flight propellant reserves.

So far, several technologies have been reviewed, and some now under development, which could potentially achieve precision mixture ratio control. Present components in development are the advanced flow meters and new tank

liquid mass gauging techniques. Effort has been put forth towards advancing the state of the art for the Balanced Flow Meter (BFM) and tank liquid volume instruments. The former can potentially minimize the error in propellant flow measurement down to 0.15%, and the latter can minimize the volume measurement down to less than 0.5%.

Quoted from [11], “The balanced flow meter (BFM) is a thin, multi-hole orifice plate...” shown in Figure 9, “...with holes sized and placed per a unique set of equations. It meets all NASA flow meter requirements, and allows measurements where none were possible before, such as LOX lines upstream of turbopumps. It can condition as well as measure flow while improving velocity, momentum, energy or other profiles. It provides flow measurement, flow conditioning, and controlled flow restriction. It functions with minimal straight pipe run; it simultaneously measures mass flow rate, fluid volumetric flow rate and fluid density. In commercial industry, it is at TRL 6 (Chevron, Texaco, Sloss Industries, etc.) For NASA use it is estimated to be at TRL 2.5...” Figure 11 is a schematic comparison with of it with a conventional orifice flow meter. “...The balanced flow meter has 10X better accuracy, 2X faster pressure recovery (shorter distance), 15X noise reduction, and 2.5X less permanent pressure loss.”

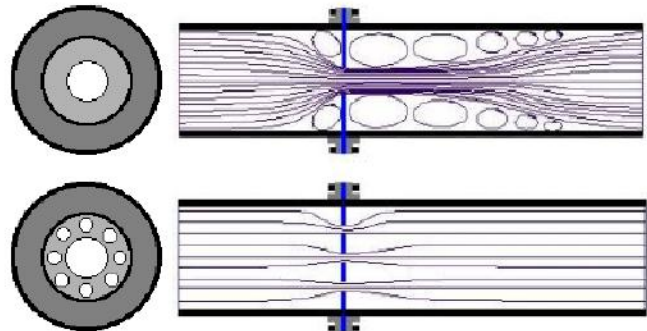


Figure 11 – Comparison of BFM (bottom) with conventional orifice flow meter (top)

Further, according reference [10], BFM has been successfully applied to liquid oxygen (with repeatable accuracy of ~ 0.3%), nitrogen, and methane, gaseous nitrogen, methane, and air; in many other industrial fluids in commercial and harsh military applications.

In order to apply BFM to precision propellant mixture ratio control, present effort statistically characterizes it in a standard spacecraft pressurization system and further compares its performance to that of turbine, venturi, and cavitating venturi flow meters.

Present effort uses a simulated propellant feed system and benign fluids such as water to test variety of commonly used and novel pressurization and flow control schemes. The system is dubbed as the “demonstration rig” for developing the mixture ratio control technology, see Figure

10 for a photograph of it in operation and Figure 12 for it during the assembly phase.

Strategically instrumented for the purpose, the rig will be used to statistically evaluate system and component accuracies, operational issues, equipment capability envelop, etc. Propellant pressurization feed system control algorithms are also to be studied.



Figure 12 – One of the tanks undergoing test prior to mount

Presently, the demonstration system has become operational and has tested 4 BFM's, 2 standard orifices, and calibrated magnetic flow meters in 1.5" & 0.75" lines. It has demonstrated both the tank pressure- and the ratio valve based- flow control to set mixture ratio within approximately 0.5% of the intended value.

The mixture ratio demonstrator is performing within planned tolerances and, in the future, new techniques and components such as pressurization methods, tank level instrument, flow meters, control algorithms, etc., can be rapidly applied and statistically tested to help improve spacecraft precision propellant flow control and measurement.

6. CONCLUSIONS

NASA's In-Space Propulsion Technology Program invests in several technology areas including advanced chemical propulsion. The objective is to enable and enhance the science capabilities in science missions through development of and improvements to the chemical propulsion systems and components with system level benefits. Presently three tasks are ongoing: (1) High temperature thruster, also known as AMBR engine; (2) light weight tanks for propellant; and (3) precision propellant management and mixture ratio control. All three technologies show potential for significant payload benefit

for science missions large or small. For example, AMBR engine can offer a payload gain of 60 kg in a Titan-Enceladus orbiter mission. For the same class of mission, light weight tanks can offer an additional 50 kg of mass advantage. Precisely controlling the propellant amount in loading and in usage by means of precision tanking and flowrate control and measurement (also known as "mixture ratio control") can reduce excessive onboard propellant reserve. For example, payload gain can be 40 kg as result of the propellant reserve reduction for the Jupiter Polar Orbiter with Probes (JPOP) mission. In general, for the technologies discussed here, payload gains are larger for missions requiring larger ΔV . To be incorporated into flight missions, the technology availability dates are year 2009 for the AMBR engine but later for the light weight tank and mixture ratio control. Finally, AMBR engine is scheduled to undergo testing starting May 2008 while light weight tanks and mixture ratio control tasks are to conclude by midyear 2008 due to interruption in funding.

7. ACKNOWLEDGMENTS

Author wishes to gratefully acknowledge the following individuals for their substantial technical input--analytical and informational--and programmatic support. Their effort helps advance the state of the art of in-space chemical propulsion, and they either directly or indirectly contribute to the content of this paper.

John Dankanich, Gray Research, Inc., NASA GRC

Leslie Alexander, Jr., NASA MSFC

Scott Miller, Aerojet General Corporation

Scott Henderson, Aerojet General Corporation

Kimberly Wilson, Aerojet General Corporation

Thomas Delay, NASA MSFC

Anthony Kelly, NASA MSFC

Robert Hickman, NASA MSFC

Jack Chapman, NASA MSFC

Christopher England, NASA JPL

Joseph Lewis, NASA JPL

Lorie Grimes-Ledesma, NASA JPL

Michael P. J. Benfield, University of Alabama in Huntsville

Matthew Turner, University of Alabama in Huntsville

8. REFERENCES

- [1] Gordon Woodcock and Dave Byers, Gray Research, et al., "Advanced Chemical Propulsion Study," 40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 11-14 July 2004, Fort Lauderdale, Florida
- [2] Leslie Alexander, NASA MSFC, "Advanced Chemical Propulsion"—a presentation to the Technology Assessment Group at the JANNAF Propulsion Meetings, Monterey California, January 18, 2005
- [3] Scott Miller, Scott Henderson, Aerojet General Corporation, et al., "Performance Optimization of Storable Bipropellant Engines to Fully Exploit Advanced Material Technologies," 2006 NASA Science and Technology Conference, July 2007
- [4] Robert Hickman and Sandra Elam, NASA MSFC, Scott O'Dell and Anatoly Shchetkovskiy, Plasma Process, Inc., "High Temperature Materials for Chemical Propulsion Applications," 54th JANNAF Propulsion Meeting, Denver Colorado, May 14-14, 2007
- [5] Michael P. J. Benfield, (now of University of Alabama in Huntsville) Science Applications International Corporation (SAIC), "Impact of Selected Propulsion using the ACPS Model", presented to Les Johnson on March 29, 2004
- [6] Chris England, NASA Jet Propulsion Laboratory, "High Temperature Thruster," JPL Propulsion Technology Program FY'07 Year-End Review, October 2007
- [7] Michael P. J. Benfield, J. Belch, Modeling of Spacecraft Advanced Chemical Propulsion Systems, AIAA-2004-4195, 40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Fort Lauderdale, Florida, July 11-14, 2004.
- [8] Joseph Lewis, Lorie Grimes-Ledesma, NASA Jet Propulsion Laboratory, et al., "Development of Ultralight Composite-Overwrapped Propellant Tanks."
- [9] Michael P. J. Benfield, Matthew Turner, University of Alabama in Huntsville, et al., "Advanced Chemical Mixture Ratio Control Study," Power Point presentation for In-Space Propulsion Technology Project Technical Task Agreement (TTA), September 2007
- [10] A. R. Kelley, NASA Marshall Spaceflight Center, "High-Accuracy Mixture Ratio Control and Characterization," JANNAF Propulsion Conference, May 14-18 2007
- [11] L. Alexander and K. Bishop-Behel, NASA Marshall Spaceflight Center, et. al., "Simulation Analysis of Pressurization and Mixture Ratio Control for Spacecraft Propulsion"

9. BIOGRAPHY



Larry Liou is a project manager for the advanced chemical propulsion area in the NASA In-Space Propulsion Technologies Project Office. Previously he performed research in liquid booster rocket engine combustion from 1988 to 1999 during which he evaluated various ignition concepts including laser ignition, and combustion wave ignition for the X-33 Aerospace engine. From 1999 to 2006 he served in various management positions at NASA John H. Glenn Research Center. Larry has a Bachelor of Science degree in mechanical engineering from University of Toledo and a Master's degree in thermal and fluids science from Case Institute of Technology.