Spacecraft Propulsion System Impacts When Incorporating Advanced Chemical Propulsion System Technologies

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Abstract— A study was performed for the NASA MSFC In-Space Propulsion Technology Projects Office to assess the propulsion system wet mass impact of incorporating candidate chemical propulsion system technologies (active mixture ratio control, ultra light weight tank, high temperature and pressure thrust chambers, LOX/N2H4 and advanced monopropellants) into previously flown spacecraft missions (MESSENGER, Cassini, MRO, and MGS).

Combining the technologies, three proposed propulsion systems were developed. The advanced monopropellant propulsion system incorporated ultra light weight tank technology with a higher performance monopropellant. The advanced earth storable propulsion system incorporated the ultra light weight tank technology, the active mixture ratio control technology, and the high temperature and pressure thrust chamber technology. The advanced space storable propulsion system incorporated the ultra light weight tank technology, the active mixture ratio control technology, and the LOX/N2H4 technology.

Utilizing a spacecraft propulsion system sizing tool, the proposed propulsion systems were modeled and the impacts assessed. Results indicate that advanced monopropellant propulsion systems provide benefit to low spacecraft propulsive energies and advanced space storable propulsion systems provide benefit to high spacecraft propulsive energies, while the advanced earth storable propulsion systems have the greatest overall benefit to all ranges of spacecraft propulsive energy.

1. INTRODUCTION

The objective of this study was to assess the benefits of advanced chemical technologies to actual flown missions to benchmark the potential benefits to future missions. System impacts to address included wet spacecraft mass, decreased mass fraction, increase payload, increase propulsion performance, reduced power, propulsion subsystem dimensions, significant deltas to other spacecraft subsystems, significant deltas to spacecraft and launcher/fairing integration, and significant deltas to ground operations. It was assumed that the first three system impacts were the most important and thus received the focus during the study.

2. STUDY APPROACH AND ASSUMPTIONS

The approach for this study is outlined in Figure 1. Mission parameters (Mo, ΔV, and propulsion system payload) were found for the four reference missions as shown in Table 1. These parameters were input into the Advanced Chemical Propulsion (ACPS) model. The output of the model, the calculated propulsion system payload, was then used as the baseline for the mission. Concurrently to the reference mission information being gathered, a literature review was conducted on the candidate technologies to understand the state of the art for each technology under consideration. These technologies were then modeled for input into ACPS. The technology evaluation incorporated the modeled technologies with the baseline ACPS configuration to determine the effect of the candidate technologies. In order to determine the amount of propulsion system payload that was gained or deducted from the mission, the difference between the ACPS baseline and the output of the technology incorporation was added to the mission defined propulsion system payload. This payload is the amount of mass that the propulsion system has delivered to the destination, not the scientific payload for the mission.

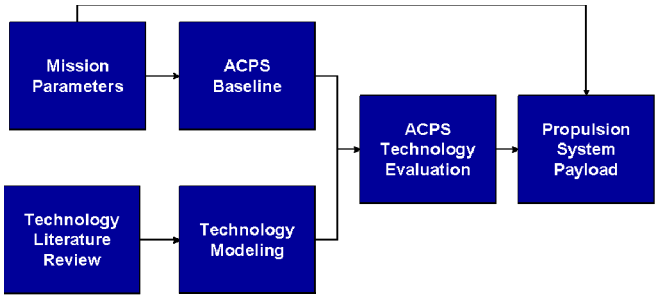

Figure 1. Study Approach

Table 1. Selected Mission Parameters

Mission	Mo (kg)	ΔV (m/sec)	Propulsion System Payload (kg)
MRO	2000	1395	655.5
MESSENGER	1089	2045	273.8
MGS	1060	1280	473.9
Cassini	5609	2039	2102.8

The assumptions for this study outlined the baseline conditions for the ACPS model for each mission. Each mission's engine(s) thrust level was kept constant. The engine conditions for the bipropellant system (NTO/N2H4) were a chamber pressure of 140 psia, and mixture ratio of 0.9, and a specific impulse of 326.9 seconds. For the monopropellant system, the chamber pressure was assumed to be 150 psia and the specific impulse of the engine was 229 seconds. The composite tank baseline was to have a stress factor of 1.0 (meaning that the strength of the over wrap was considered to be SOA) and a liner thickness of 30 mils. Residuals for the bipropellant system were assumed to be 5% of the useable propellant load.

3. SELECTED MISSIONS

Four missions were selected for evaluation purposes during this study. The four missions represented a cross-section of spacecraft propulsion system energies (ΔV and Mo) to allow for understanding the impact of potential technologies at different energy levels. Figure 2 depicts the energy levels of the selected missions.

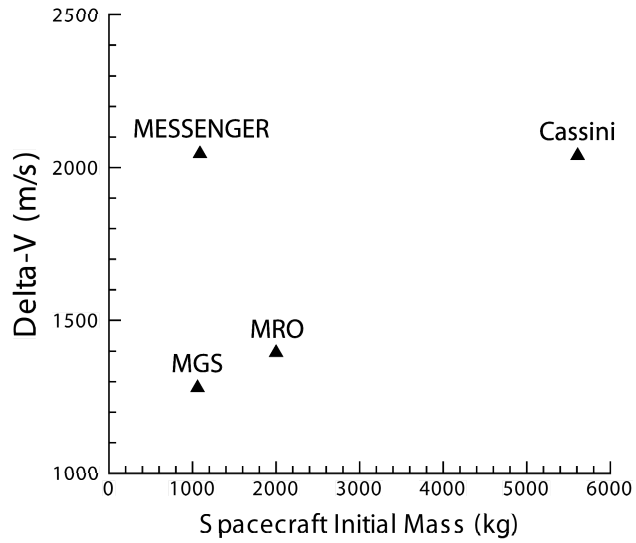

Figure 2. Selected Mission Propulsion Energies

The Mars Reconnaissance Orbiter (MRO) was launched in 2005 to search for evidence that water persisted on the surface of Mars for a long period of time. The MRO mission was a medium energy mission with a total propellant load of 1070.9 kg (N2H4). The total spacecraft propulsion system wet mass was 1164.4 kg. The propulsion system consisted of one fuel tank and six 170N thrusters. For more information on the MRO propulsion system see Reference 1.

The Mercury Surface, Space Environment, Geochemistry, and Ranging (MESSENGER) mission was a Discovery-class mission to study the planet Mercury. The MESSENGER mission was a medium energy mission with a total propellant load of 587.6 kg (N2O4 and N2H4). The total spacecraft propulsion system wet mass was 681 kg. The propulsion system consisted of one 667N thruster and three (two fuel and one oxidizer) 6AL-4V titanium propellant tanks. For more details on the MESSENGER propulsion system see Reference 2.

The Cassini mission placed a spacecraft in orbit around Saturn and delivered the probe Huygens to the Saturn moon Titan. Being a flagship mission, the Cassini propulsion subsystem was large with 3000 kg of total propellant (N2H4, MMH, and N2O4) on board. The total spacecraft propulsion system wet mass was 3632.6 kg. The propulsion system had two main engines (one operational and one spare) delivering 445N of thrust and two titanium propellant tanks (one for N2O4 and one for MMH). For more details on the Cassini propulsion system see Reference 3.

The Mars Global Surveyor (MGS) spacecraft was a replacement for the Mars Observer mission that entered Mars orbit in 1997. The MGS was a relatively low energy mission with only 387.4 kg of total propellant (N2O4, N2H4). The total spacecraft propulsion system wet mass was 451.5 kg. The propulsion system had one 655N main engine and three (two fuel and one oxidizer) 6AL-4V

titanium propellant tanks. For more details on the MGS propulsion system see Reference 4.

4. TECHNOLOGY BACKGROUND

Six advanced chemical technologies were chosen for evaluation during this study: ultra light weight tank, high temperature thrust chamber, high pressure thrust chamber, mixture ration control, LOX/N2H4, and advanced monopropellant. The high temperature thrust chamber and the high pressure thrust chamber were combined due to their integrated effects. Each technology is summarized below.

Ultra Light Weight Tank

Typically, the propulsion tank is the single largest highest dry mass item of an in-space propulsion system. Ultra Light Linerless Composite Tanks (ULLCT) promise to deliver the efficiencies that will make future propulsion systems viable and minimize propulsion system mass growth. ULLCTS may offer up to a 25% weight reduction compared to conventional metal lined composite over-wrapped tanks, allowing increased reactant storage and/or reduced launch mass.

Monolithic titanium construction is the current state-of-the-art for chemical propellant tanks, while composite over wrapped pressure vessels (COPV) with a metallic liner are the current state-of-the-art for solar electric propulsion (SEP) tanks. With a switch to ULLCT, significant mass, cost, and fabrication time savings can be realized.

The characteristics that make composite materials so effective in tank applications are their high strength and stiffness along the fiber direction, and their significantly lower density than metals. Current COPV incorporate a thin metallic or polymeric inner liner, which serves as an inert permeation barrier. The liner prevents the contained gases from leaking through the composite laminate that tends to form microcracks and leak paths at high strain levels. Most traditional composite over wrapped pressure vessels with metallic or polymeric liners are designed to safeguard against structural failure by rupture, since the liner is trusted to take care of the containment of the fluids. In essence the structural design of the tank is decoupled from the fluid containment requirement of the design. In contrast, the linerless composite tanks depend on the composite shell itself to serve as a permeation barrier in addition to carrying all pressure and environmental loads. However, metal liners are difficult to fabricate and can constitute up to 50% of the tank's total mass and a significant percentage of the total cost and time to fabricate the tank. Linerless composite tanks have been identified by both NASA and DOD as an enabling technology for future reusable launch vehicles (RLV), where they may offer up to a 25% weight reduction compared to conventional tanks, allowing increased reactant storage and/or reduced launch mass. Structural weight reductions will translate directly to additional payload margins, and thus improved mission capabilities and reduced cost. A linerless all-composite tank like an ULLCT can reduce total tank mass, and hence increases efficiency and therefore provides the most efficient storage vessels for in-space propulsion systems.

High Temperature and Pressure Thrust Chambers (HTPTC)

Reference 6 describes a study performed on the application of temperature resistant materials for use in propulsion system thrust chambers. This study incorporated the study of the use of Iridium-Coated Rhenium Radiation-Cooled Rockets. The study showed that a system utilizing rhenium (Re) substrate and an iridium (Ir) coating provides higher temperature operation and increased lifetime. By utilizing the iridium-coated rhenium (Ir/Re) reduces or eliminates the need for fuel film cooling. The improvement introduced with the use of Ir/Re system allows chamber wall temperatures of 2200°C. This increased chamber wall temperature ability allows for higher fuel combustion temperatures. This increase in fuel combustion temperature can be directly related to increased oxidizer to fuel mixture ratio. The advantage of increasing the chamber walls ability to resist higher temperatures allows a propulsion system to burn a higher mixture ratio. The higher mixture ratio will provide a higher specific impulse for the propulsion system which in-turn makes the spacecraft propulsion system more efficient.

Reference 7 describes a study conducted to compare various high-pressure thrust chambers. The study incorporated various aspects of the combustion chamber design under extreme temperature and pressure scenarios. The results of this report show a positive increase in chamber pressure to maximize the specific impulse. Using NTO/N2H4 an Isp of 330 lbf-sec/lbm was achieved at a mixture ratio (O/F) of 1.0 and a chamber pressure of 500psia.

Reference 8 describes a comparative study performed on High Pressure Earth Storable (HIPES) Rocket Technology. The study consisted of various high pressure and mixture ratio (O/F) scenarios that were tested to determine maximum performance characteristics. The test data shows that the use of NTO/N2H4 provides a maximum theoretical Isp of 343 lbf-sec/lbm at a mixture ratio of 1.25 and a chamber pressure of 500psia. However this data is only theoretical due to leakage. Also because the 500psia data was not determined from testing, the extrapolated chamber temperatures indicate the possibility of reaching or exceeding the maximum allowable for Ir-Re.

Reference 9 describes a trade study performed to evaluate the spacecraft-level performance increase attainable by using high pressure bipropellant engine technology. Parameters were varied to understand the effect of engine performance on overall spacecraft mass including chamber pressure (150-700 psia) and mixture ratio (0.8 to 1.5). Data

was extrapolated from this report to be used in this study. Figure 3 below depicts the data extrapolated. Data was not provided for chamber pressures below 300 psia.

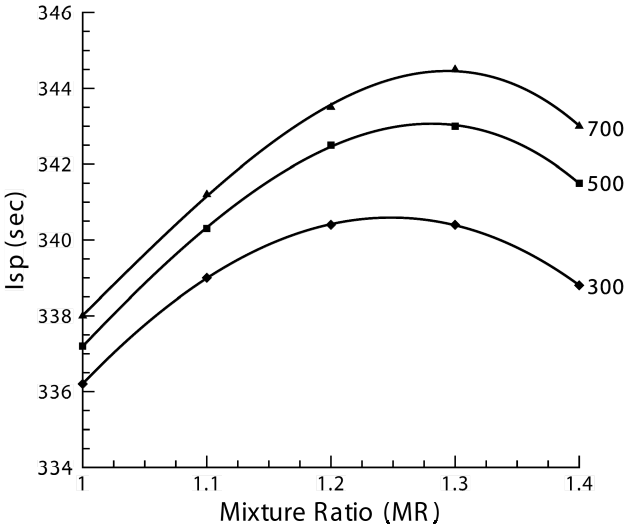

Figure 3. Engine Performance for a Regeneratively Cooled Engine (reference 9)

Mixture Ratio Control

Mixture ratio control is a concern on any liquid bi-propellant rocket stage. Poor mixture ratio control can result in relatively large percentages of one of the propellants remaining on board when the other propellant has been burned to depletion. These residual propellants can significantly reduce the performance. Proper mixture ratio control can be accomplished in a number of ways. As in nearly any control problem, control can be accomplished by a closed loop or open loop system. The most common open loop mixture ratio control technique is the simple approach of trying to load the same ratio of propellant masses as you expect to consume during the flight. The closed loop control is typified by an in-flight propellant mass measuring system which feeds an error signal to some type of engine mixture ratio controller.

The closed loop system is comprised of continuous capacitance-type level measurement transducers feeding an error signal to a tank pressure controller. Mixture ratio is controlled by the modulation of one of the tank pressures. The requirements on a mixture ratio control system are really very simple. The system must minimize the difference between loaded and consumed propellant mass ratios and keep the mixture ratio within safe engine operating limits. In a pressure-fed stage either throttling or tank pressure modulation is suitable mixture ratio control methods. The interfaces of the mixture ratio control system are very important. Interfaces for the closed loop are the propellant loading system, the propellant feed and engine system, the pressurization system, the electrical system, and the tankage system.

Propellant loading requirements with a closed loop mixture ratio control system are not stringent. Loading errors can either be corrected out in flight or the propellant measuring portion of the control system can be used to load to the proper initial levels. The propellant feed and engine systems place certain requirements on the mixture ratio control system. That is, the control system must not be able to vary the mixture ratio beyond safe engine operational limits. The mixture ratio control system requires that at least one of the tank pressures be controlled over a wider range of pressures than would otherwise be required.

The open loop mixture ratio control is not a system but rather a technique. It is also not new to liquid rockets. The technique is simply to try to match the loaded mixture ratio as closely as possible to the mixture ratio which is expected to be consumed. Actually, this is also done in the case of the closed loop system but not nearly as much care is required. The "system", therefore, is comprised of an engine whose mixture ratio consumption versus inlet pressure is accurately known and an accurate propellant loading procedure. For this system, the engine consumption ratio will be facilitated by propellant tank volume calibration. This system interfaces with generally the same systems as the closed loop except that there are no flight electrical requirements. The pressurization system interface is a critical one. Any tolerances around the desired tank pressures will alter the consumed mixture ratio. However, the pressurization system and its control concept strongly affect the degree of mixture ratio sensitivity to tank pressure errors.

Tankage interfaces with the open loop mixture ratio control are due to loaded propellant level variations and the resulting initial ullage volume variations. The total tank volume must be somewhat larger than nominal if there is an initial minimum ullage volume requirement which cannot be violated. This approach can keep loaded propellant mass constant. The performance of this open loop control method is dependent on many variables. These variables are anything which would cause a propellant loading error or cause the engine to consume a mixture ratio different than that predicted. For more information on mixture ratio control, see reference 10.

Reference 11 provides a source of information on the state of the art in mixture ratio control devices. According to the report, from simulation results the residuals in the propulsion system can be reduced from 5 to 2% with an additional 4.9 kg of control equipment added.

LOX/N2H4

LOX/N2H4 is not a common propellant combination. Though through theoretical calculations it proves to have a substantially higher Isp than standard NTO/N2H4. Reference 12 predicts that the Isp of a LOX/N2H4 engine could be as high as 353 seconds. Other studies suggest the Isp achievable is closer to 340 seconds (reference 13). This make for a slight increase in performance to typical propellants. Reference 14 provides for previous analysis where an Isp of 345 seconds was used as reference point.

Monopropellant
A monopropellant is the most common and reliable propellant for spacecraft used today. Monopropellants, like the name suggests uses only one chemical for combustion. Monopropellants usually come in a liquid form. Some current state of the art (SOA) monopropellants are catalytic decomposed N2H4 and hydroxylammonium nitrate (HAN).

N2H4 has been most commonly used for many attitude controls and even in a small engine for multiple space crafts. Reference 15 provides that the typical Isp associated with N2H4 is generally around 230s. The temperatures associated with N2H4 are low. But this chemical does have a high hazard risk associated with it. The resulting vapor from the burns is a toxic vapor making it difficult to handle.

HAN propellants are more commonly used for military applications for its safer handling. HAN propellants generally run at higher temperatures to achieve the same performance as the N2H4. HAN has more desirable qualities than N2H4 for instance; the hazardous risks associated with this propellant are greatly reduced to a hazard of just skin exposure. It has characteristics of stored gases which makes then non-flammable and non-explosive (reference 16).

5. TECHNOLOGY MODELING

Each technology was modeled in the ACPS according to the data that were found during the literature review. The method of modeling each technology is described below.

Ultra Light Weight Tank
The assumption for an ultra light weight tank is that the tank is made of composite material with a composite over wrap for strength and a metallic liner to prevent propellant leakage. For the ultra light weight tank, the engine parameters were kept at baseline conditions (Pc of 140 psia, MR of 0.9, and Isp of 326). The liner thickness of the composite tank was varied from 5 mils to 30 mils, with 30 mils being the baseline condition for the Chandra X-ray Observatory (baseline tank reference). The over wrap composite strength was varied from one times the baseline strength (again, Chandra) to four times the baseline strength. The basic assumption is that when the strength of the composite increases the mass of the over wrap decreases for a given tank pressure.

High Temperature and Pressure Thrust Chambers (HTPTC)
For the modeling of the high temperature and pressure thrust chambers, data from reference 9 was extrapolated to be used in the model as shown in Figure 3. Parameters that were varied included mixture ratio, chamber pressure, and the resulting specific impulse (Isp) that was given in the reference data.

Mixture Ratio Control
Reference 11 provided data concerning the method for incorporating the mixture ratio control technology into the reference missions. From the report, the propellant residuals could be reduced from 5% of the useable propellant mass to 2% with an additional 4.9 kg of mass being included in the system for measurement.

LOX/N2H4
Few sources of information were available on data concerning LOX/N2H4 test firings. Data gathered from reference 14 provided that the specific impulse of 345 seconds was achievable with engine conditions of 200 psia for chamber pressure and a mixture ratio of 0.85. These conditions were input to the model for the LOX/N2H4 technology. The LOX was assumed to be passively cooled and therefore did not need to have a zero boil-off system onboard or additional propellant for boil-off.

Monopropellant
Parametric modeling was used to understand the effect of increasing the Isp of the monopropellant. Reference data produced specific impulses in the range of 230-240 seconds, which was used as the baseline conditions for the monopropellant system. The specific impulse was varied in the monopropellant from 230-300 seconds at 10 second intervals. The density of N2H4 was used throughout the analysis due to the fact that no other information was found on other monopropellants. It is assumed that the density of any other propellants would be comparable to the density of N2H4.

6. INDIVIDUAL TECHNOLOGY ASSESSMENT RESULTS

The candidate technologies were assessed individually for the four selected missions. Overall results indicate that the high temperature and pressure thrust chamber and the mixture ratio technologies have the greatest increase in propulsion system payload for the selected missions. The results from the individual technology evaluations are provided below.

Ultra Light Weight Tank
The evaluation of the ultra light weight tank indicates that the propulsion system payload change was between 1 and 4% of the baseline mission. In general, decreasing the thickness of the liner yields higher propulsion system payloads than increasing the strength of the composite over wrap material (and thus decreasing the mass of the material needed) as shown in Figure 4 which is representative of the four missions. The energy of the mission does not seem to have an effect on the increase in propulsion system payload. Overall, the conditions yielding the highest payload amounts were a liner thickness of 5 mils and an over wrap strength of four times the baseline.

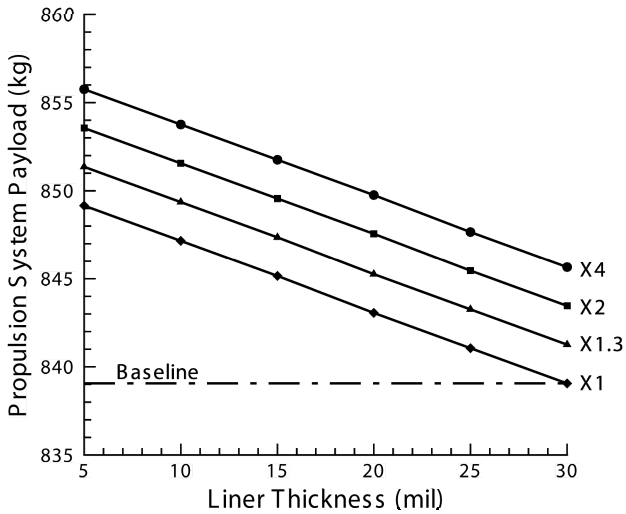

Figure 4. ULWT Impact on the MRO Mission

High Temperature and Pressure Thrust Chambers (HTPTC)

The evaluation of the high temperature and pressure thrust chambers indicates that as the chamber pressure increases, the amount of propulsion system payload decreases. Furthermore, the optimal mixture ratio for a NTO/N2H4 propulsion system tends to be 1.2 to 1.3. Figure 5 is provided as an example of the data trend for the MGS mission. Excluding the MRO mission due to it being a monopropellant system, the effect of incorporating the high temperature and pressure thrust chamber can vary from -6% (for MESSENGER) to 6% (for Cassini). This technology tends to yield higher propulsion system payloads for higher energy missions. Overall the highest propulsion system payload conditions were found to be with a mixture ratio of 1.2 and a chamber pressure of 300 psia. Further analysis should be undertaken to determine if chamber pressures below 300 psia produce higher propulsion system payloads.

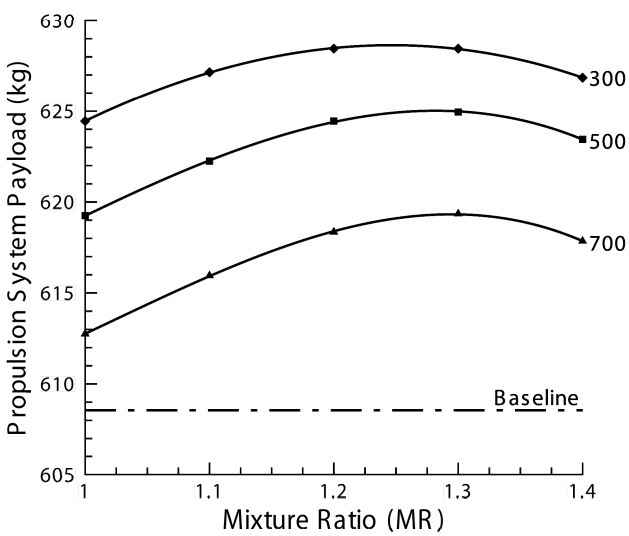

Figure 5. Effect of HTPTC on the MGS Mission

To illustrate the effect of decreasing the propulsion system payload while increasing the chamber pressure, the follow example is provided. As Figure 6 shows, taking the MGS mission and holding the mixture ratio at 1.0 and varying the chamber pressure from 300 to 700 psia, the propellant decreases slightly (1 kg) while the propellant tanks, helium, and helium tanks increase. The net result of this action is that the overall propulsion system wet mass has increased and thus reduced the available propulsion system payload delivered to the destination (assuming that the overall spacecraft mass is fixed).

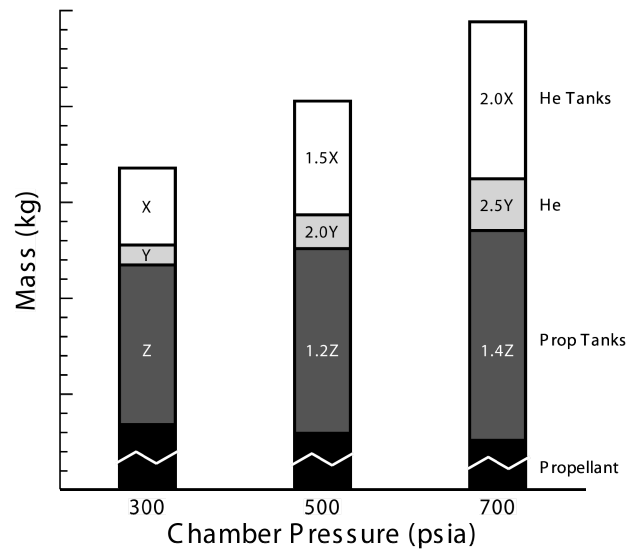

Figure 6. Chamber Pressure Increase Effect Example

Mixture Ratio Control

The MRO mission was excluded from this technology evaluation because it was a monopropellant system. The other three missions were evaluated for the mixture ratio control technology. As shown in **Figure 7**, the propulsion system payload increased from 4% to 6% for the various missions. Mission energy seemed to have no effect on the amount of increase in the propulsion system payload.

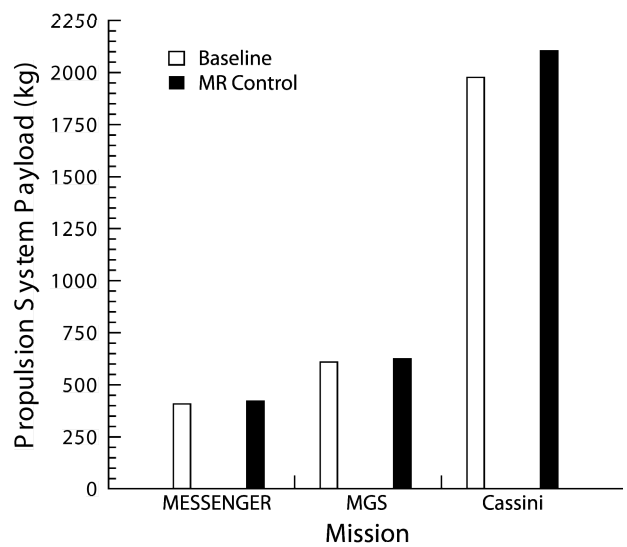

Figure 7. MR Control Effect on Selected Missions

LOX/N2H4

The LOX/N2H4 propellant combination yielded payload changes of -5% to 4% (excluding MRO due to it being a monopropellant system) as shown in Figure 8. Lower energy missions saw no increase in propulsion system payload and MESSENGER actually saw a decrease. From the missions selected, only Cassini (high energy) had an increase in propulsion system payload. Although the Isp of LOX/N2H4 is higher than a standard NTO/N2H4 propellant combination (by 19 seconds) the mass of all of the major propulsion subsystems (tanks, helium, helium tank, and components) is larger than the reduction in the propellant load due to the higher Isp as shown in Figure 9.

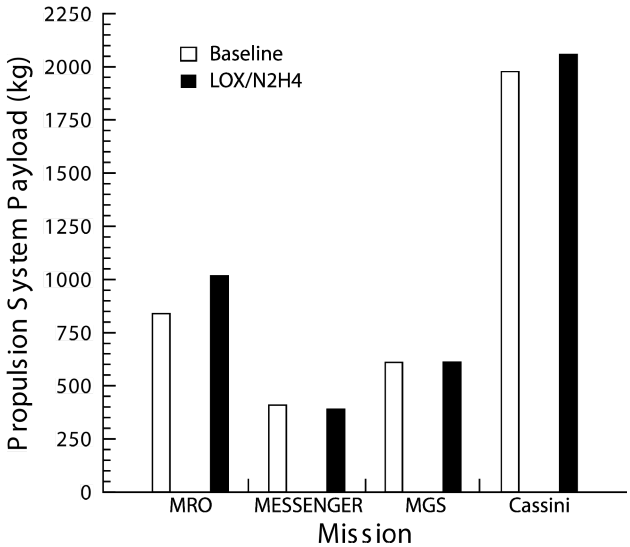

Figure 8. Effect of LOX/N2H4 on Selected Missions

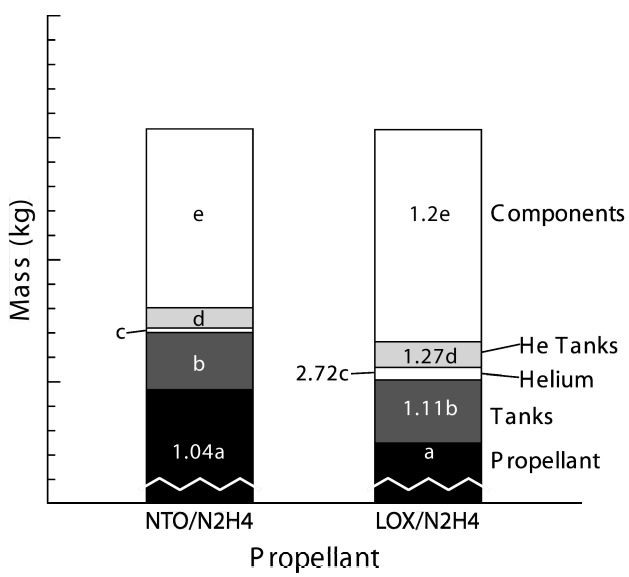

Figure 9. Propellant Comparison for the MESSENGER Mission

Monopropellant

The advanced monopropellant technology yielded large payload changes from -53% to 24%. As the Isp of the monopropellant approached 300 seconds, the baseline payload could be attained. As shown in Figure 10, lower energy missions, like MESSENGER, could produce their baseline payload with a monopropellant Isp of approximately 290 seconds.

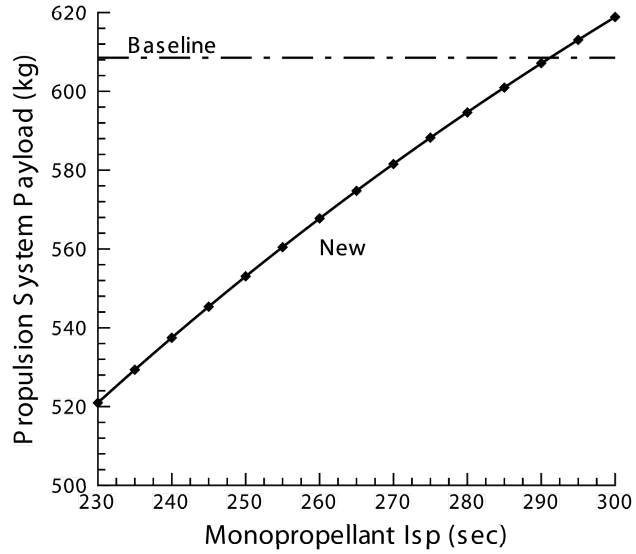

Figure 10. Impact of an Advanced Monopropellant on the MESSENGER Mission

Due to the monopropellant system's simplicity of one tank, one helium tank, and simple feed system, the Isp of the system can be lower than that of a conventional bipropellant system (NTO/N2H4) and still achieve the same propulsion system payload as depicted in **Figure 11**.

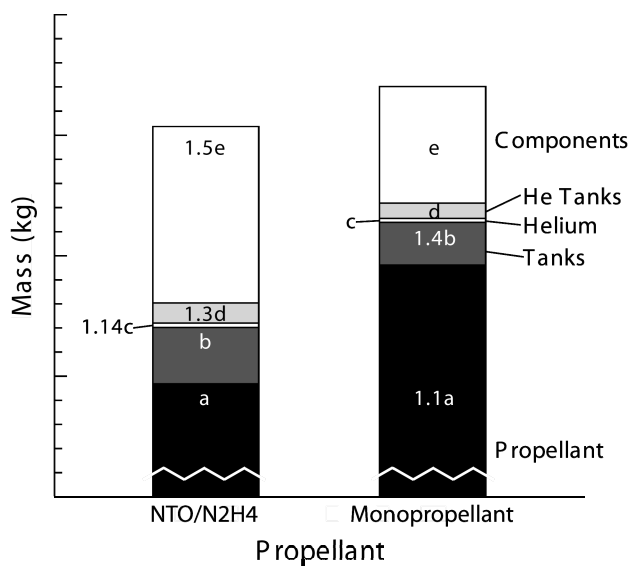

Figure 11. Propulsion Subsystem Mass Comparison for the MESSENGER Mission

7. Technology Combination Results

Three propulsion systems were evaluated when the technologies were combined. The advanced earth storable propulsion system incorporated the high temperature and high pressure thrust chamber, the ultra light weight tank,

and the mixture ratio control technologies. The advanced space storable propulsion system incorporated the LOX/N2H4, the ultra light weight tank, and the mixture ratio control technologies. The advanced monopropellant propulsion system incorporated the advanced monopropellant and the ultra light weight tank technologies. Two cases were evaluated; a midpoint of the technology improvement (MPT) and a total technology capability (TTC) whereby the ability of the technology is pushed to its limits. Table 2 outlines the parameters for each propulsion system at each case.

Table 2. Combined Technology Combinations

Propulsion System	Parameter	MPT	TTC
Earth storable	MR	1.1	1.2
	Pc (psia)	500	300
	Isp (sec)	340.3	340.4
	Tank liner thickness (mil)	15	5
	Overwrap strength	2	4
	MR control	Yes	Yes
Space storable	MR	0.85	0.85
	Pc (psia)	200	200
	Isp	345	345
	Tank liner thickness (mil)	15	5
	Overwrap strength	2	4
	MR control	Yes	Yes
Monopropellant	Isp (sec)	260	300
	Tank liner thickness (mil)	15	5
	Overwrap strength	2	4

Results indicate that the advanced earth storable propulsion system seems to have the most promise for all mission sets. The propulsion system payload is generally higher in all of the evaluated missions. The ULWT technology yields the highest payloads except for the high energy mission. For that mission the MR control technology yielded the highest results. The advanced space storable propulsion system does not show significant payload increase over the advanced earth storable. The LOX/N2H4 technology does not provide any increase in payload except for the high energy mission. The mixture ratio control technology tends to yield higher payload than the other technologies. The advanced monopropellant propulsion system shows promise for lower energy missions. The ULWT technology provides little benefit.

Figure 12 depicts the combined technology results for the advanced earth storable propulsion system. As can be seen in the figure, all of the missions have increased propulsion system payload except the MESSENGER mission at the midpoint technology of the high temperature and pressure thrust chamber. Further analysis shows that as the propulsion system energy of the mission increases the more benefit obtained by the advanced earth storable propulsion system technologies.

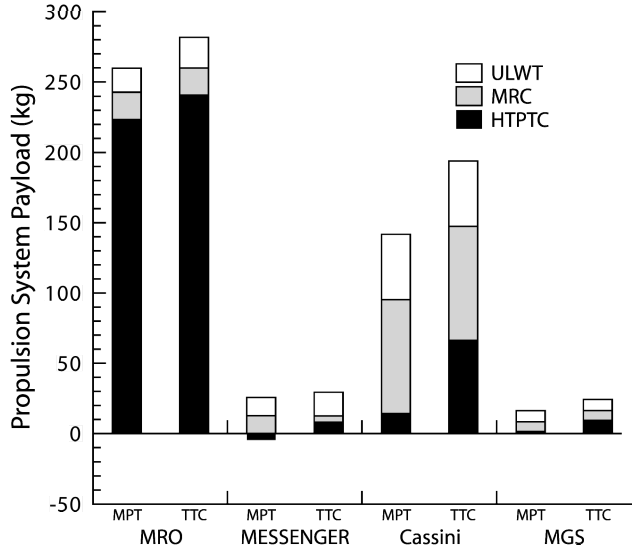

Figure 12. Advanced Earth Storable Propulsion System Payload Comparison

As shown in Figure 13, only a large spacecraft (like Cassini) with a significant propulsion system requirement would have an increase in propulsion system payload when using a space storable propulsion system. Utilizing a space storable propulsion system for low energy missions does not provide increased propulsion system payload except when technologies are combined.

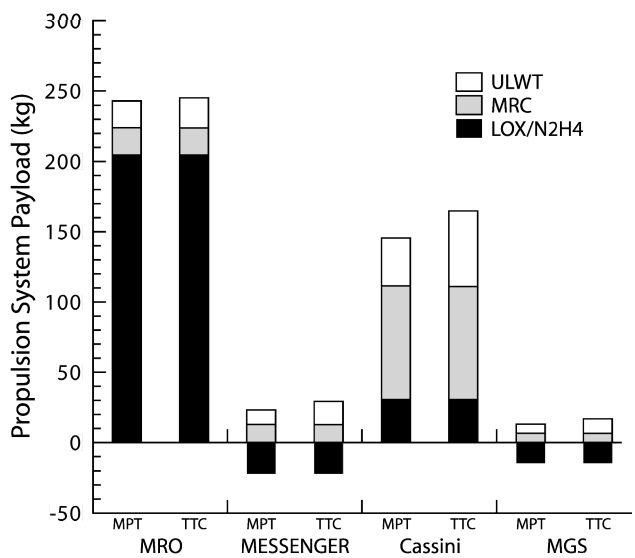

Figure 13. Space Storable Propulsion System Payload Comparison

Figure 14 depicts the results of the advanced monopropellant system. When the propulsion system requirements are small for a mission, an advanced monopropellant system could be used to meet the payload requirements instead of a bipropellant system.

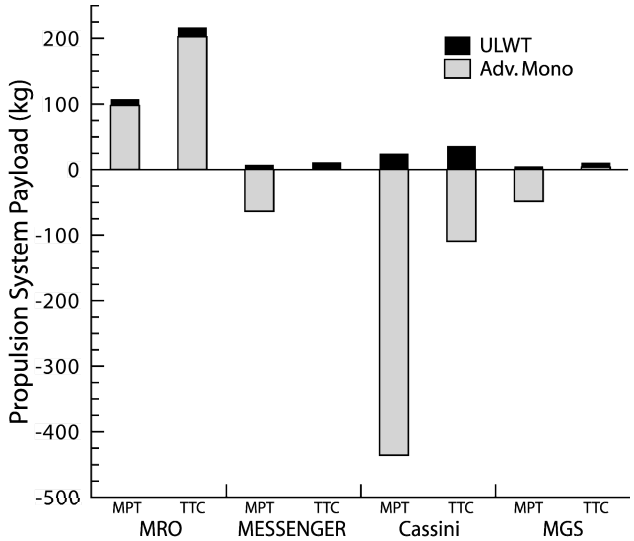

Figure 14. Advanced Monopropellant Propulsion System Payload Comparison

8. SCIENTIFIC PAYLOAD INCREASE EVALUATION

All data shown to this point have been the technology effect on the propulsion system payload, the amount of mass the propulsion system delivers to its destination. To understand the impact of a technology on the scientific payload of the mission, data were obtained from three of the missions evaluated. MESSENGER had a scientific payload of 40 kg, MRO had 139 kg, and Cassini had 356 kg. For purposes of this study it was assumed that 50% of the propulsion system mass savings could be allocated to additional scientific payload. Using this metric and the results of the combined technology propulsion system evaluations, promising propulsion systems (ones that showed increased propulsion system payload) were assessed to determine the increase in scientific payload. Data were plotted as an increase (or decrease) to the baseline payload. For the MRO mission all three propulsion systems were evaluated. As Figure 15 depicts, with MRO being a monopropellant system most advanced propulsion systems can double the scientific payload of the mission. The exception is the MPT case for the advanced monopropellant propulsion system due to its lower Isp.

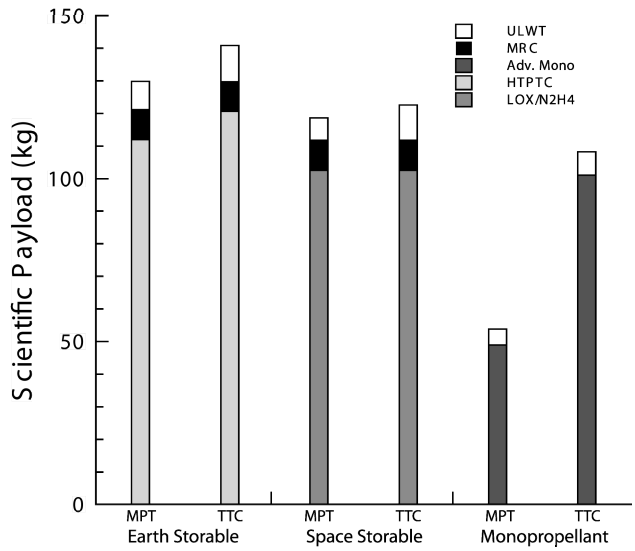

Figure 15. Scientific Payload Comparison for the MRO Mission

For the MESSENGER mission the advanced earth storable and the advanced monopropellant propulsion systems were evaluated. The advanced earth storable propulsion system can provide an increase in the scientific payload of the mission by 25% to 50% as shown in Figure 16. Only the TTC case of the advanced monopropellant propulsion system provides an increase.

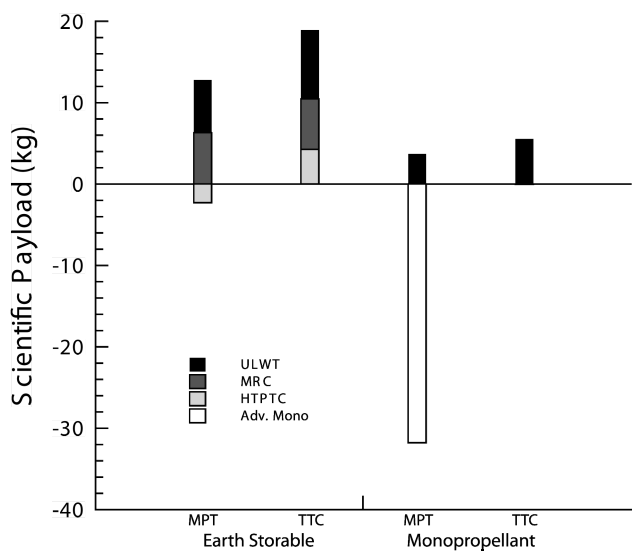

Figure 16. Scientific Payload Comparison for the MESSENGER Mission

For the Cassini mission the advanced earth storable and the advanced space storable propulsion systems were evaluated. As shown in Figure 17, both propulsion systems provide approximately a 25% increase in scientific payload for the mission.

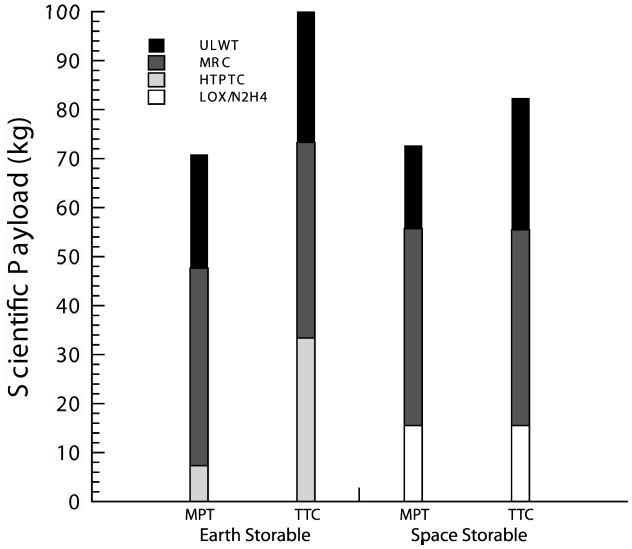

Figure 17. Scientific Payload Comparison for the Cassini Mission

9. CONCLUSIONS

This study evaluated five candidate technologies to assess their impact of incorporation into existing spacecraft missions. The results of the study indicate that for lower energy missions 1) advanced monopropellants provide the baseline payloads due to their simplicity, 2) high temperature and pressure thrusters tend to yield higher payloads, however the chamber pressure needs to be kept low (<300 psia), 3) the ultra light weight tank and the mixture ratio control technologies tend to yield similar results, and 4) LOX/N2H4 provides no benefit. The results further indicate that for higher energy missions 1) advanced monopropellants are of no use – they provide no increase in propulsion system payload, 2) LOX/N2H4 provides some payload increase, 3) the mixture ratio control technology tends to yield higher payloads than the ultra light weight tank technology due to the larger propellant loads, and 4) high temperature and pressure thrusters tend to yield higher payloads, however the chamber pressure needs to be kept low (<300 psia). Overall, the study results indicate that combining technologies can provide significant increases in propulsion system and scientific payload.

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