MSFC-389

Advanced Chemical Propulsion System Study

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Log Number 2007-XXX

A detailed, mission-level systems study has been performed to show the benefit resulting from engine performance gains that will result from NASA's In-Space Propulsion ROSS Cycle 3A NRA, Advanced Chemical Technology sub-topic. The technology development roadmap to accomplish the NRA goals are also detailed in this paper.

NASA-Marshall and NASA-JPL have conducted mission-level studies to define engine requirements, operating conditions, and interfaces. Five reference missions have been chosen for this analysis based on scientific interest, current launch vehicle capability and trends in space craft size:

- **GTO to GEO, 4800 kg, delta-V for GEO insertion only ~1830 m/s**
- **Titan Orbiter with aerocapture, 6620 kg, total delta V ~210 m/s, mostly for periapsis raise after aerocapture.**
- **Enceladus Orbiter (Titan aerocapture) 6620 kg, delta V ~2400 m/s.**
- **Europa Orbiter, 2170 kg, total delta V ~2600 m/s.**
- **Mars Orbiter, 2250 kg, total delta V ~1860 m/s**

The figures of merit used to define the benefit of increased propulsion efficiency at the spacecraft level include propulsion subsystem wet mass, volume and overall cost.

The objective of the NRA is to increase the specific impulse of pressure-fed earth storable bipropellant rocket engines to greater than 330 seconds with nitrogen tetroxide and monomothylhydrazine propellants and greater than 335 seconds with nitrogen tetroxide and hydrazine. Achievement of the NRA goals will significantly benefit NASA interplanetary missions and other government and commercial opportunities by enabling reduced launch weight and/or increased payload. The study also constitutes a crucial stepping stone to future development, such as pump-fed storable engines.

Nomenclature

AR	=	*Area Ratio*
FFC	=	*Fuel Film Cooling*
GEO	=	*Geosynchronous Earth Orbit*
h_g	=	*Convection coefficient*
I_{sp}	=	*Specific impulse*

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LAE	=	*Liquid apogee Engine*
MMH	=	*Monomethylhydrazine,* $N_2H_3CH_3$
MON-X	=	*Mixed oxides of nitrogen, nitrogen tetroxide and X% NO by mass in solution*
N_2H_4	=	*Hydrazine,* N_2H_4
NTO	=	*Nitrogen tetroxide,* N_2O_4
OF	=	*Oxidizer to Fuel ratio*
P_c	=	*Chamber pressure*
psia	=	*Pounds per square inch absolute*
SG	=	*Specific Gravity*
SOA	=	*State of the Art*
TDK	=	*Two Dimensional Kinetic*

I. Introduction

NASA has initiated the Advanced Chemical Technology research announcement with the goal to increase the specific impulse of pressure-fed earth storable bipropellant rocket engines to greater than 330 seconds with nitrogen tetroxide and monomothylhydrazine propellants and greater than 335 seconds with nitrogen tetroxide and hydrazine propellants. State of the art storable rocket engines deliver approximately 323 and 328 seconds I_{sp} for the respective propellant combinations given above. Increased specific impulse has the obvious advantage of reducing the propellant required to perform necessary space craft maneuvers. Figure 1 shows that the top level expected improvement in I_{sp} will result in nearly 1% reduction in the space craft gross mass devoted to propellant. Refining this estimate is one goal of this NRA.

For telecommunications satellites in geosynchronous orbit the mass reduction can be applied to increasing the power generating capability and/or number of transponders, which increases the revenue potential for the satellite's owners and operators. For scientific probes, the propellant mass reduction can be applied to increasing the data gathering capability of the space craft. In some cases, where a direct cost benefit is not relevant, the capacity of the improved technology to enable a useful scientific mission which was not previously viable has been evaluated.

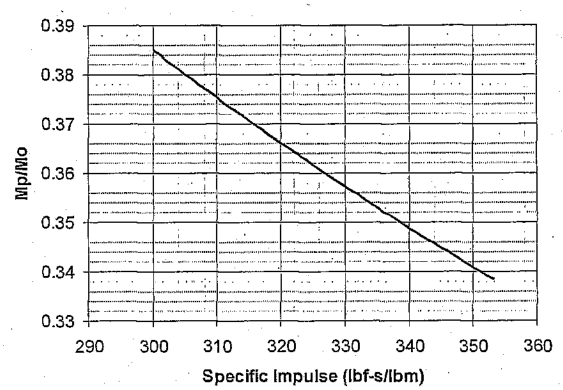

Figure 1. Reduction in Propellant Fraction with Increased I_{sp}

Previous studies have indicated that a sufficient decrease in required propulsion system mass is likely with increased engine performance to justify the further evolution of propulsion technology.

The NRA performance goals will be achieved by expanding the operating envelope of flight-proven iridium/rhenium (Ir/Re) combustion chamber technology currently used for apogee-raising engines. This material system has the capacity to withstand steady-state wall temperatures approaching 2500 K compared to the state of the art usage at less than 1700 K. The temperature increase is expected to be realized by chamber pressure increase and increase of the oxidizer to fuel ratio compared to the state of the art.

Performance will be increased by modifying operating conditions and designing for chamber temperatures to fully exploit the capability of state-of-the-art materials. In addition, chamber fabrication methods for Ir/Re and other promising material systems will be evaluated for potential process and/or performance improvements. Aerojet will also perform analyses to estimate performance improvements and conduct parametric limits testing on an existing Ir/Re development engine to validate potential design approaches.

This study has identified promising chamber materials for further study. Follow-on research will conduct material screening studies, complete detailed design and analysis of selected engine concepts, fabricate new combustion chambers, fabricate new injectors, and assemble and test two engines with the required propellant combinations to verify analytical predictions and demonstrate achievement of program goals.

II. Mission Analysis

Case	Description	Space Craft Mass	Space Craft Delta-V	Comments
1	GTO to GEO	4800 kg	~1830 m/s	Consider GEO insertion only
2	Titan Orbiter with aerocapture	6620 kg	~210 m/s	Δ-V mostly for periapsis raise after aerocapture.
3	Enceladus Orbiter (Titan aerocapture)	6620 kg	~2400 m/s	Propulsive option for comparison to case 2
4	Europa Orbiter	2170 kg	~2600 m/s.	
5	Mars Orbiter	2250 kg	~1860 m/s	

III. Performance Enhancement Roadmap

There are certain "knobs that can be turned" to alter the specific impulse of a given rocket engine. Those most frequently employed are chamber temperature, chamber pressure and expansion ratio. Chamber temperature is a function of the propellants reacting in the chamber, the pressure at which combustion occurs and the amount of fuel film cooling used to protect the chamber walls.

Maximum chamber temperature is limited by the species being reacted. Some propellants are simply more energetic than others and release more energy on combustion. Lightweight reaction products also are favored for increasing specific impulse.

Chamber pressure is limited by the need to minimize the propellant storage pressure while maintaining sufficient injector pressure drop to ensure stable combustion.

A significant fraction of the fuel consumed is used to cool the chamber walls. Reducing the percentage of fuel film cooling (FFC) is another variable that may be used to increase performance, with ramifications that will be discussed in the following section.

IV. Advanced Chamber Materials

Fuel film cooling is essential to prevent hot combustion products from damaging chamber walls. The rhenium/iridium system is limited to maximum wall temperatures less than about 2500 K by the durability of the iridium coating. Even though rhenium has a melting temperature of 3450 K, when exposed to oxidizing species at temperatures in excess of 873 K it will oxidize to rhenium heptoxide, Re_2O_7, with melting temperature of 636 K, resulting in rapid, catastrophic failure. Rhenium is therefore coated with iridium, which provides an oxidation-resistant surface but a melting point of "only" 2683 K. Rocket engine operation duration is limited by iridium erosion which increases as temperature nears iridium's melting point. It is obviously desirable to apply an oxidation resistant coating that will withstand higher temperatures before erosion becomes critical. Some candidate materials selected for evaluation are halfnium oxide, HfO_2.

MSFC-404

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Log Number 2006-032

A detailed, mission-level systems study has been performed to show the benefit resulting from engine performance gains that may result from NASA's In-Space Propulsion ROSS Cycle 3A NRA, Advanced Chemical Technology sub-topic. The technology development roadmap to accomplish NRA goals and the initial development and test results leading to accomplishing these goals are also detailed in this paper.

NASA-MSFC and NASA-JPL have conducted mission-level and system-level studies to extrapolate improved engine performance into spacecraft requirements and performance. Four reference missions have been chosen for this analysis based on scientific interest, current launch vehicle capability and trends in space craft size:

- **GTO to GEO, 4800 kg, ΔV for GEO insertion only ~1830 m/s**
- **Enceladus Orbiter (Titan aerocapture) 6620 kg, ΔV ~2400 m/s.**
- **Europa Orbiter, 2170 kg, total ΔV ~2600 m/s.**
- **Mars Orbiter, 2250 kg, total ΔV ~1860 m/s**

The figures of merit used to define the benefit of increased propulsion efficiency at the spacecraft level include propellant mass, propulsion subsystem wet mass, volume and overall cost.

The objective of the NRA is to increase the specific impulse of pressure-fed earth storable bipropellant rocket engines to at least 330 seconds with nitrogen tetroxide and monomethylhydrazine propellants and at least 335 seconds with nitrogen tetroxide and hydrazine. Achievement of the NRA goals will significantly benefit NASA interplanetary missions and other government and commercial opportunities by enabling reduced launch weight and/or increased payload. The study also constitutes a stepping stone to future developments, such as pump-fed storable engines.

Test of a state-of-the-art rocket engine while varying critical performance parameters, has also been performed in support of this NRA.

Nomenclature

AR	=	*Area Ratio*
*C**	=	*Characteristic Velocity of rocket combustion products*

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ΔV	=	*Delta Velocity*
FFC	=	*Fuel Film Cooling*
GEO	=	*Geosynchronous Earth Orbit*
GTO	=	*Geosynchronous Transfer Orbit*
I_{sp}	=	*Specific Impulse*
JPL	=	*Jet Propulsion Laboratory*
kg	=	*Kilogram*
LAE	=	*Liquid Apogee Engine*
MMH	=	*Monomethylhydrazine, $N_2H_3CH_3$*
MON-X	=	*Mixed Oxides of Nitrogen, nitrogen tetroxide and X% NO by mass in solution*
MSFC	=	*Marshall Space Flight Center*
N_2H_4	=	*Hydrazine, N_2H_4*
NRA	=	*NASA Research Announcement*
NTO	=	*Nitrogen Tetroxide, N_2O_4*
OF	=	*Oxidizer to Fuel ratio*
P_c	=	*Chamber pressure*
psia	=	*Pounds per square inch absolute*
SG	=	*Specific Gravity*

I. Introduction

NASA has initiated the Advanced Chemical Technology research announcement with the goal to increase the specific impulse of pressure-fed earth-storable bipropellant rocket engines to at least 330 seconds with nitrogen tetroxide and monomethylhydrazine propellants and at least 335 seconds with nitrogen tetroxide and hydrazine propellants[1]. State of the art storable rocket engines deliver approximately 323 and 328 seconds I_{sp} for the respective propellant combinations given above[2, 3]. Increased specific impulse has the obvious advantage of reducing the propellant required to perform spacecraft maneuvers. Figure 1 shows that for a spacecraft requiring ΔV of approximately 1430 m/s the rocket equation predicts nearly 1% reduction in the space craft gross mass devoted to propellant if the NRA goals are met. Refining this estimate for some real and projected spacecraft, to show the return on this investment, is one goal of this NRA.

For telecommunications satellites in geosynchronous orbit the mass reduction can be applied to increasing the ACS propellant, hence life of the spacecraft, increasing the power generating capability and/or increasing the number of transponders, which increases the revenue potential for the satellite's owners and operators. For science missions, the propellant mass reduction can be applied to increasing the data gathering capability of the spacecraft. In some cases, where a direct cost benefit is not relevant, the capacity of the improved technology to enable a useful scientific mission which was not previously viable has been evaluated. Preliminary studies have indicated that a sufficient decrease in required propulsion system mass is likely with increased engine performance to justify the further evolution of propulsion technology[4].

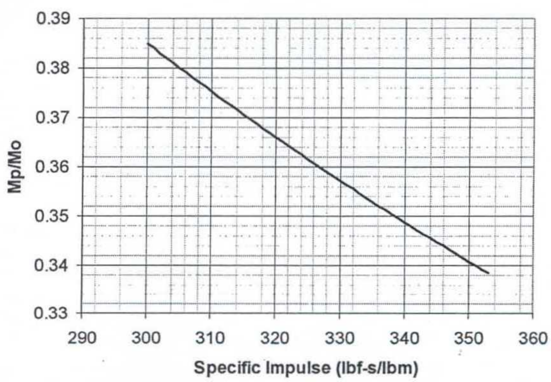

Figure 1. Reduction in Propellant Fraction with Increased I_{sp}. M_o is spacecraft gross mass, M_p is propellant mass

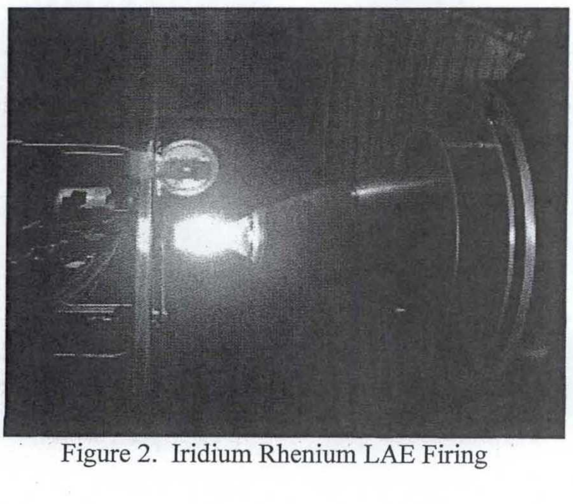

Figure 2. Iridium Rhenium LAE Firing

The NRA performance goals are expected to be achieved by expanding the operating envelope of flight-proven iridium/rhenium (Ir/Re) combustion chamber technology currently used for liquid apogee engines (LAEs), an example of which is shown in Figure 2. This material system has the capacity to withstand steady-state wall temperatures approaching 2470 K[5] compared to the state of the art usage at less than 1700 K. The temperature increase is expected to be realized by chamber pressure increase and increase of the oxidizer to fuel ratio (OF) compared to the state of the art.

Another NRA goal regards improving LAE design for Ir/Re and other promising material systems. The end goal is to expand the body of knowledge available to engine designers to minimize engine cost while maximizing performance and reliability. Aerojet has also performed analyses to estimate LAE performance improvements and conduct parametric limits testing on an existing Ir/Re development engine to validate potential design approaches.

Follow-on research will conduct material screening studies, complete detailed design and analysis of selected engine concepts, fabricate new combustion chambers, fabricate new injectors, and assemble and test two engines with the required propellant combinations to verify analytical predictions and demonstrate achievement of program goals.

II. Mission and Systems Analysis

For this NRA to be worthwhile, the benefits of achieving NRA goals must justify the cost. The benefits of improved system performance may be expressed in increased capability (payload) or improved finances (which usually flows directly from increased capability). In financial terms, there must be a great enough return on investment to amortize the research and development effort. For science missions, where an immediate financial benefit is not easily quantifiable, increased capability can enable a mission, allow use of a less expensive launch vehicle or increase a mission's scientific value.

To build confidence that there is a reasonable return on investment available, mission and system analyses were performed at NASA's Jet Propulsion Laboratory and at NASA's Marshall Space Flight Center. These analyses were intended to determine whether the integrated engine and propulsion system, presuming state of the art and near-term technology, is likely to permit a cost-effective increase in system capability.

A. Mission Analysis

The first step was a mission analysis for each of four reference missions, identified to be of commercial or current scientific interest, conducted primarily at JPL. The reference missions are identified in Table 1 with estimates of the launch mass, cumulative velocity change (ΔV) required of Attitude Control System (ACS) and axial thrust elements of the propulsion system and the mass of any deployed assemblies. The propulsion requirements for a GEO-sat were obtained from The Delft University of Technology[6], extrapolated for a 15 year service life. The ΔV required of a spacecraft's attitude control system can either be minimal, as is the case for many planetary missions, or have a significant effect on

Table 1, Reference Mission Summary Descriptions

Mission	ACS need	Axial Δ-V, m/s	Launch Mass, kg	Deployed/ shed mass, kg
GTO to GEO	797 m/s	1,830	4,800	0
Europa Orbiter	23.4 kg	2,215	2,170	0
Mars Orbiter	20.0 kg	2,064	2,250	0
Titan-Enceladus (T-E) Orbiter	50.0 kg	2,368	6,633	1298, 59.2 & 345

overall propulsion system size, as for the GEO satellite where station-keeping is a major system driver. Frequently, minimum system mass results from a dual-mode system, where the ACS uses hydrazine as a monopropellant and the axial engine burns hydrazine from the same supply system with NTO. Even if minimum mass does not result from a dual-mode system, economy of thrusters and improved reliability may mitigate in favor of such a system. For the purposes of comparison, dual mode axial/ACS propulsion was assumed for all spacecraft.

For each mission, the mass of the spacecraft at launch is estimated based on the expected launch vehicle capability and the terminal velocity which the launch vehicle is obligated to impart. The spacecraft trajectory is planned, in some cases taking advantage of planetary momentum exchange to modify the spacecraft velocity. Main engine burns are an essential part of trajectory planning to keep the spacecraft on course. In one case, the scientific requirements of the mission require deployment of spacecraft elements such as a heat shield or independent landing craft, requiring accounting for the mass decrements. Demands placed on the attitude control system are modeled based on historical data, acceptable limits of spacecraft pointing and statistical distributions of spacecraft attitude perturbations due to internal and external influences. An example of a mission ΔV and mass budget is presented in Table 2. The calculated propellant load is increased by about 1% due to the inability of propellant tanks to completely discharge their contents. Finally, because of the uncertainties inherent in engineering, a degree of margin (about 5%) is added to the propellant load.

Event			Mass	ΔV, m/s.
Earth Launch	1	DV		0
TCM	1	DV		15
TCM	1	DV		10
Venus Flyby	1	DV		0
TCM	1	DV		2
TCM	1	DV		2
Earth Flyby	1	DV		0
TCM	1	DV		1
Earth Flyby	1	DV		0
TCM	1	DV		1
Earth Flyby	1	DV		0
TCM	1	DV		1
TCM	1	DV		1
Titan Aerocapture	1	ACS	59.1676639	
Aeroshell dropoff	1	JET	1298.11178	
TCMs (3 yrs)	1	DV		5
TCMs after Rhea Flybys (>	1	DV		25
Enceladus Capture (i = 50 de	1	DV		2050
Cleanup Maneuver	1	DV		5
Plane Change to release Lar	1	DV		100
Lander dropoff	1	JET	344.802236	
Plane Change to go back to	1	DV		100
Orbit Maintenance	1	DV		50
ACS	1	ACS	50	

Table 2. Sample Mission Analysis Mass and Delta-V Budget

Once the accounting is in place for mass and velocity changes, assumptions are made regarding the efficiency of the propulsion system elements. These assumptions are based on a database of past engine performance or in this case on the goals for improved main engine performance. The propellant mass required to execute the velocity changes required by trajectory planning and ACS analysis are determined by means of the rocket equation or similar calculation. Table 3 summarizes the propellant mass estimates calculated for the reference missions at the current state of the art I_{sp}, assumed to be 320 seconds for GEO missions and 325 seconds for planetary missions, and for main propulsion that achieves the NRA goals. The ACS I_{sp} was assumed to be 230 seconds for monopropellant hydrazine.

Table 3, Total Propellant load (kg) by Main Engine I_{sp} (in seconds)

Mission	I_{sp}	320 sec	325 sec	330 sec	332.5 sec	335 sec
GTO to GEO		2,918	2,901	2,885	2,877	2,869
Europa Orbiter		N/A	1,131	1,120	1,116	1,109
Mars Orbiter		N/A	1,320	1,307	1,300	1,293
T-E Orbiter		N/A	2,969	2,942	2,928	2,914

B. Systems Analysis

High I_{sp} of an engine in isolation is insufficient to assert a system benefit because known means of increasing performance (e.g. higher chamber pressure, more energetic propellants, etc.) may require an increase in system mass or cost that negate the advantages of higher engine performance. Non-recurring costs to redesign and qualify an engine to realize higher performance must also be amortized over engine use unless public funds are available.

MSFC and JPL both maintain databases of propulsion system designs and have each developed a methodology for estimating the mass of components based on performance requirements. The model used by MSFC has been documented in some detail[7]. These models apply correlations for hardware mass based on propellant volume and storage pressure, thrust required and degree of redundancy to estimate the mass of components comprising the propulsion system. MSFC produced a representative system schematic for a single redundant system, shown below as Figure 3, from which the number and size of components was derived.

Propellant tank mass is almost always the largest element of system dry mass. While decreased propellant mass is expected to result in decreased mass of a tank to contain it, an increase in tank pressure required to feed a higher I_{sp} engine may lead to a net mass increase due both to thicker tank walls required as well as more pressurant and a larger/stronger pressurant tank. For the increased performance options presented, it was assumed that the maximum required propellant tank pressure would be 400 psia with a 1.5

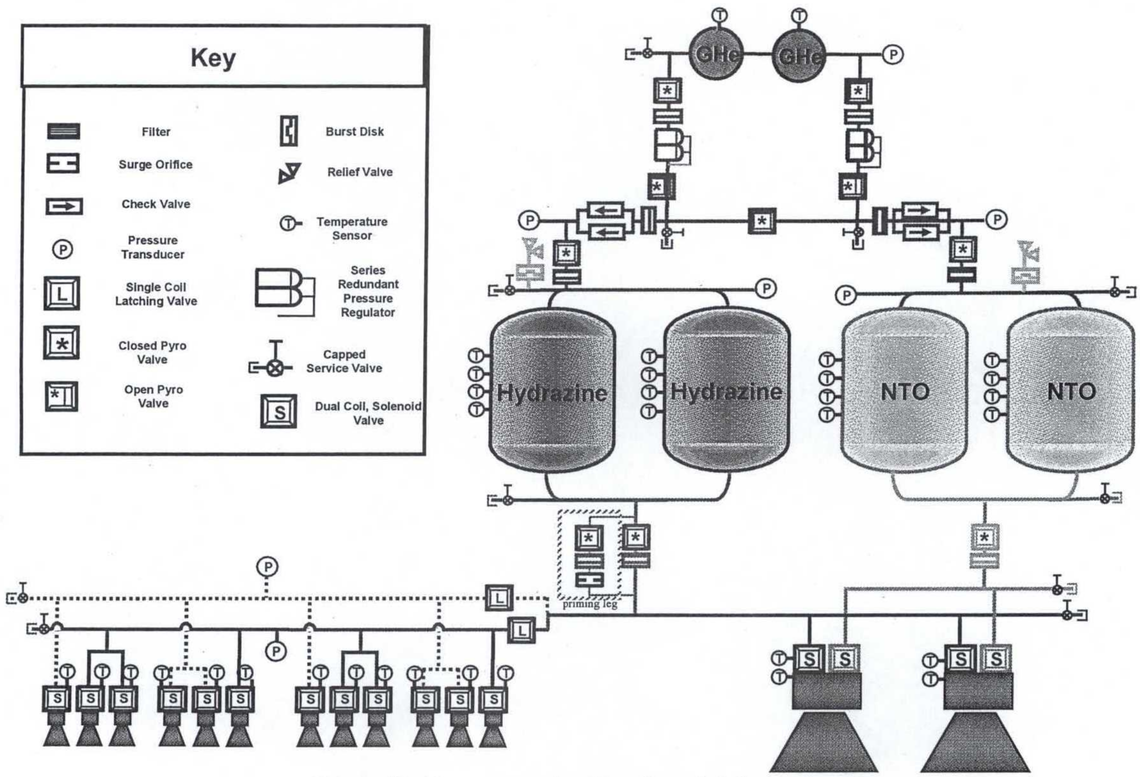

Figure 3. Representative Top-Level Schematic
(Number of Tanks Mission Dependent)

safety factor, compared to a more typical pressure of 300 psia for GEOsats(ref). Tank material was fixed as titanium (6Al-4V), ullage volume at 5% and a surface tension propellant management device (PMD) is assumed to add 10% to tank weight with 1% of the initial propellant load unusable. An assumption is implicit that the tank size may be closely optimized for the amount of propellant the spacecraft requires. In reality, unless many identical spacecraft are intended, which would justify development of an optimized tank, a compromise is sought. A spacecraft integrator usually purchases a qualified tank large enough to contain the required propellant volume, even if that tank is slightly larger than needed.

It is a debatable assumption whether desired engine performance can be obtained with 400 psia maximum propellant tank pressure. To determine the impact of higher tank pressure, the same systems analyzed above were also analyzed with the assumption of 800 psia tank pressure. In each spacecraft case, the net result is that the reduced propellant load is balanced by increased system dry mass for either negligible improvement or slightly worse system performance. So as with any other paper study, caution is in order when relying on mass predictions tied to improved system performance. Real-life system performance is dependent on balancing many mission-specific variables to obtain suitable performance, reliability and safety at an acceptable cost. Regardless of the real bottom line of mass savings within a pressure-fed system, following the path of increased engine performance by increasing chamber pressure is an essential step toward fielding pump fed systems which will reduce not only propellant mass but propellant pressure[8].

Pressurant tanks are often the next largest mass element of a propulsion system. Propellants are pressure fed from the tanks to the engine, so a composite-overwrapped helium pressure vessel was selected with size calculated assuming adiabatic blowdown of gas initially at 4500 psia down to a minimum regulator inlet limit of 800 psia.

For MSFC's system model, component masses are based on the mass of existing hardware that has been flight proven in the space environment (TRL 9[9]) in spacecraft like the Mercury Messenger or Space Shuttle. Additionally, 10% design contingency is applied to give confidence that system mass is not underestimated. Table 4 lists the system burnout masses estimated for each of the reference missions versus main propulsion specific impulse. Burn-out mass includes residual propellants as well as any unused ACS propellant at the time of main engine cut-off.

Table 4. Propulsion System Burn-out Mass (kg) by Main Engine I_{sp} (in seconds)

Mission	I_{sp}	320 sec	325 sec	330 sec	332.5 sec	335 sec
GTO to GEO		390.5	389.3	388.1	387.5	386.9
Europa Orbiter		N/A	161.0	160.2	159.9	159.4
Mars Orbiter		N/A	189.6	188.4	187.8	187.2
T-E Orbiter		N/A	331.5	329.1	327.9	326.7

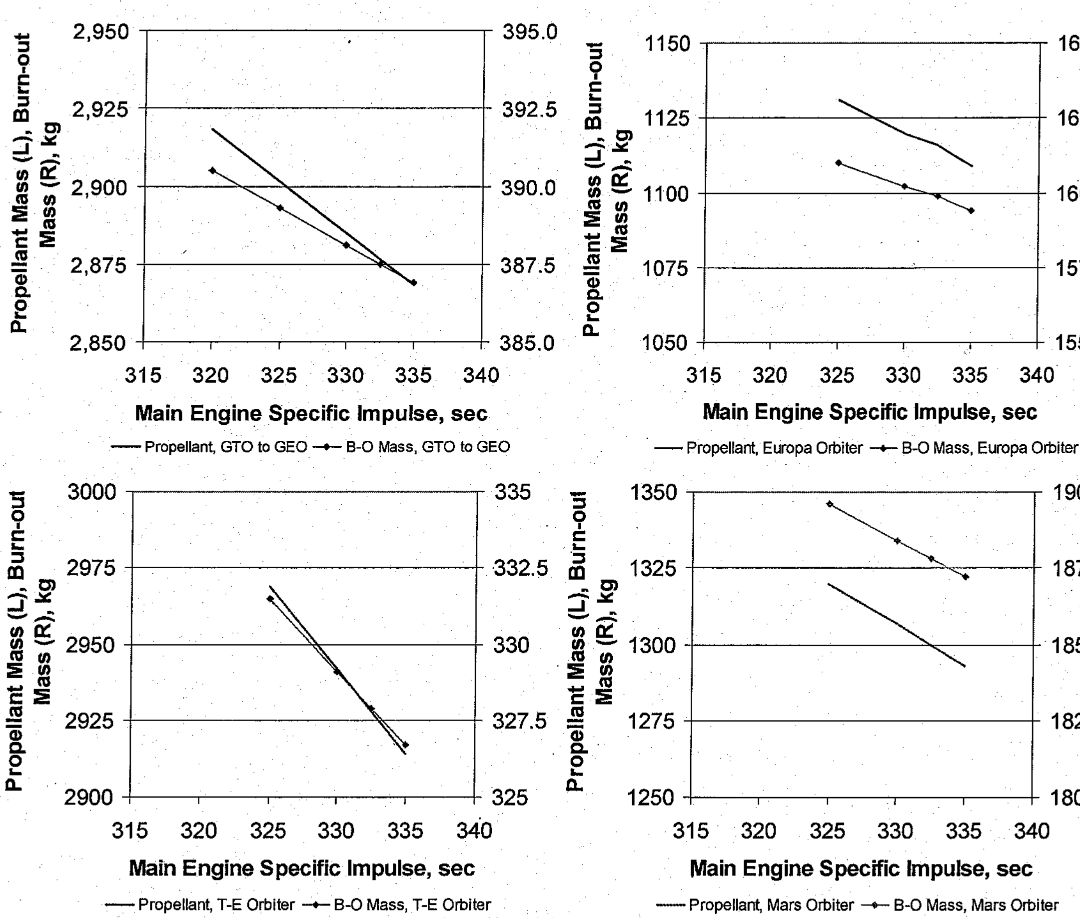

Figure 4, Propellant and System Mass Reductions with Increased Specific Impulse

Figure 4 shows the predicted propellant and system burn-out masses for the four reference missions. As expected the Titan-Enceladus mission, with the highest Δ-V need, has the steepest mass reduction with increased I_{sp}. The GEO mission has a relatively flat mass versus I_{sp} attributable not only to lower axial Δ-V but to a much higher ACS requirement, which is also reflected in a proportionally higher burn-out mass compared to the similarly sized Titan-Enceladus flight. It may be noted that the Mars and Europa orbiters have significantly higher burn-out masses relative to propellant load than the larger spacecraft. It is typical for smaller systems to have higher inert mass fractions due to the fixed mass of components like pressure transducers, regulators and some valves which represent a proportionally smaller fraction of total system weight for larger systems.

It can be concluded from the mission and systems analysis that significant reduction in spacecraft mass may be realized by improving engine specific impulse. A summary of total mass reduction is included in Table 5 compared to the baseline cases of 320 seconds Isp for the GEO mission and 325 seconds for the planetary missions. Compared to the total spacecraft mass these reductions are modest, however as a percentage of useful payload, they can be quite significant. For example, the Mercury Messenger spacecraft instrument payload is approximately 40 kilograms[10]. For a 4800 kg GEOsat, 45 additional kilograms of propellant equates to an increase of useful revenue-earning life of approximately one year, based on the propellant usage and system masses backed out of the system model.

Table 5, Total Propulsion System Mass Reduction

Mission	I_{sp}	320 sec	325 sec	330 sec	332.5 sec	335 sec
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

III. Performance Enhancement Roadmap

There are certain "knobs that can be turned" to alter the specific impulse of a given rocket engine. Those most frequently employed are chamber temperature, chamber pressure and expansion ratio. The performance improvement of increased expansion ratio is well characterized and has a diminishing effect beyond about $\varepsilon = 300$, as shown in figure 5[11].

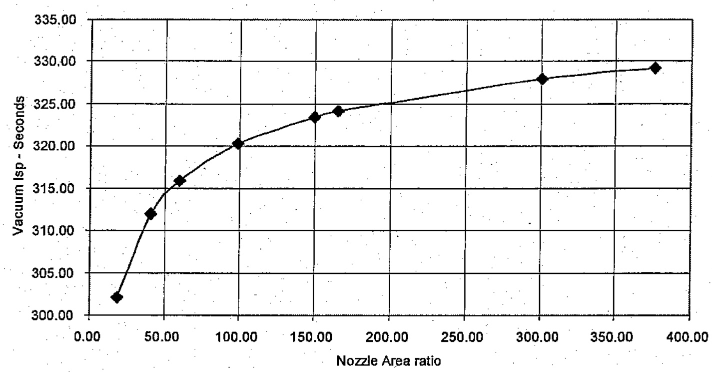

Figure 5. Isp versus Expansion Ratio for Aerojet dual-mode HiPAT LAE

Chamber temperature is a function of the propellants reacting in the chamber, the pressure at which combustion occurs and the amount of fuel film cooling used to protect the chamber walls. At any given OF, efficient mixing and sufficient residence time to achieve high combustion efficiency, the maximum chamber temperature is limited by the species being reacted; some propellants are simply more energetic than others and release more energy on combustion. Lightweight reaction products also are favored for increasing characteristic velocity, C* and in turn specific impulse. One of the ground rules of this NRA is use of NTO, MMH and N_2H_4. With C* efficiency for these propellants already near optimum levels in the core flow of existing engines, the only avenues for increasing temperature is through decreasing fuel film cooling (FFC) or increasing mixing of the FFC layer into the core flow. This implies added thermal stress to engine components that must be addressed by improved design to maintain engine life. For future work, more energetic oxidizers like liquid oxygen and fluorine are desirable but issues of storability, availability and toxicity must be resolved separately[12].

Chamber pressure increase is limited by the need to minimize the propellant storage pressure while maintaining sufficient injector pressure drop to ensure stable combustion. To gain the maximum benefit from increasing chamber pressure it is necessary to iteratively redesign the propellant injector for maximum combustion efficiency and acceptable FFC in a usable P_c/OF design box.

Future work will include investigation of pump-fed engine hardware that will permit increased chamber pressure and thrust with reduced propellant tank pressure.

IV. Test Results

To increase the knowledge base prior to designing and testing a new, higher-performance engine, an experimental investigation was conducted to map the performance of an Aerojet R4D-15, Ir/Re engine over a range of chamber pressures and mixture ratios. This engine is optimized to operate best at feed pressures of 219 to 310 psia and NTO/N_2H_4 OF ratios of 0.716 to 1.188[3], which are both lower than expected for a next generation engine. No effort was made to optimize the test unit for the new conditions. While performance can be expected to increase due to operation at higher P_c and OF the results of this test can not be expected to be as good as if an engine were optimized to operate at these conditions.

Another goal of testing is to determine the minimum delta pressure across the injector required to prevent chug, defined here as low frequency chamber pressure oscillation coupled with feed system pressure. As chamber pressure is increased to improve performance, a correspondingly large feed pressure increase is undesirable for its weight increase on system components. Demonstration of chug delayed to very low feed pressure provides evidence that injector delta pressure may be decreased safely. The test engine was successfully tested by stepping down chamber pressure until sustained chugging was observed at approximately 80 psia as shown in Figure 6. This is much lower than nominal operating P_c indicating that the inlet pressure range of 219 to 310 psia[3] might be reduced, or P_c increased, by lower injector pressure drop without risk of related instability.

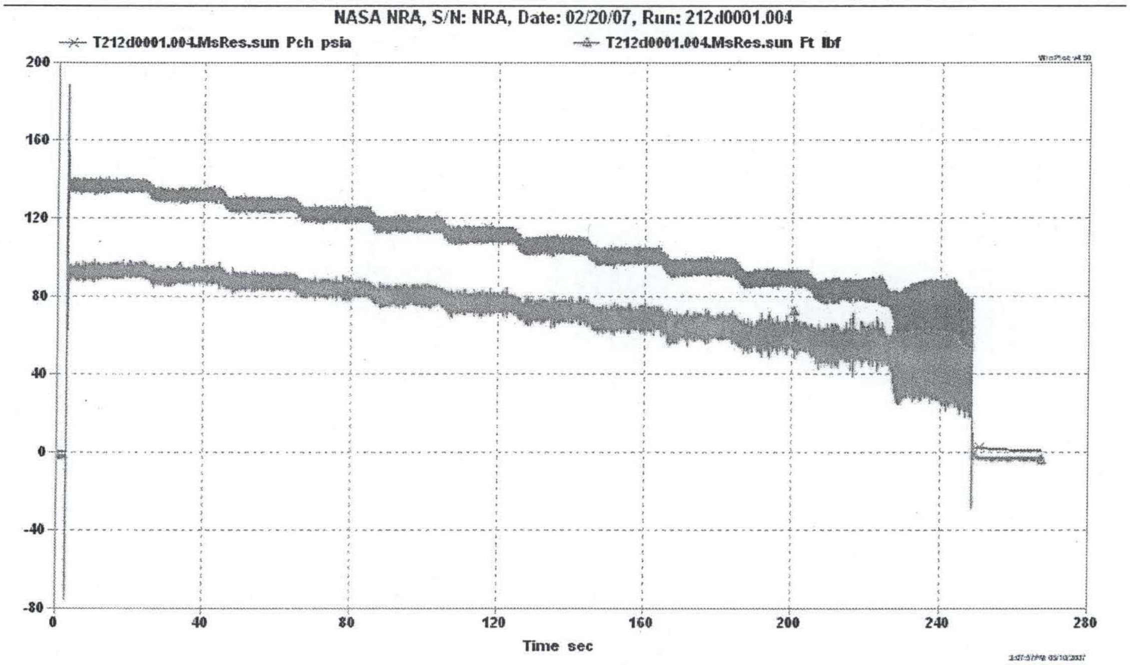

Figure 6. Step Down Chamber Pressure Until Chug Onset

The test engine, shown in figure 6, was instrumented within the walls of an internal step assembly[3] with thermocouples to measure the effectiveness of FFC during firing. The intent was to learn how far OF can be increased before FFC is no longer effective. When the FFC was effective, step temperatures were at approximately the saturation temperature of N_2H_4 at P_c. This indicates that there is normally a two-phase fuel film on the step. Knowledge of the conditions under which FFC breaks down will be used to iterate the injector design and optimize C* while still providing adequate thermal protection.

This test program resulted in high-quality data for engine performance to a maximum chamber temperature near the throat of 2296 K, 217 psia maximum chamber pressure, 53.4 psia minimum chamber pressure and 329 seconds maximum I_{sp} using NTO and N_2H_4 propellants. Engine exterior temperatures were measured using an infrared camera, which produces images like Figure 8 which may be further processed to obtain emissivity data for various external nozzle materials.

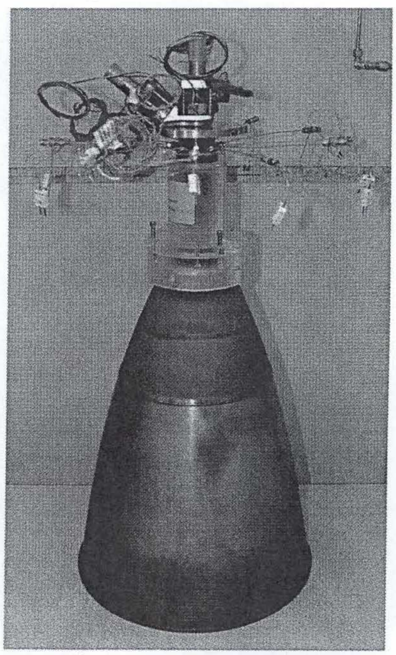

Figure 7. Test Article Instrumented with Thermocouples

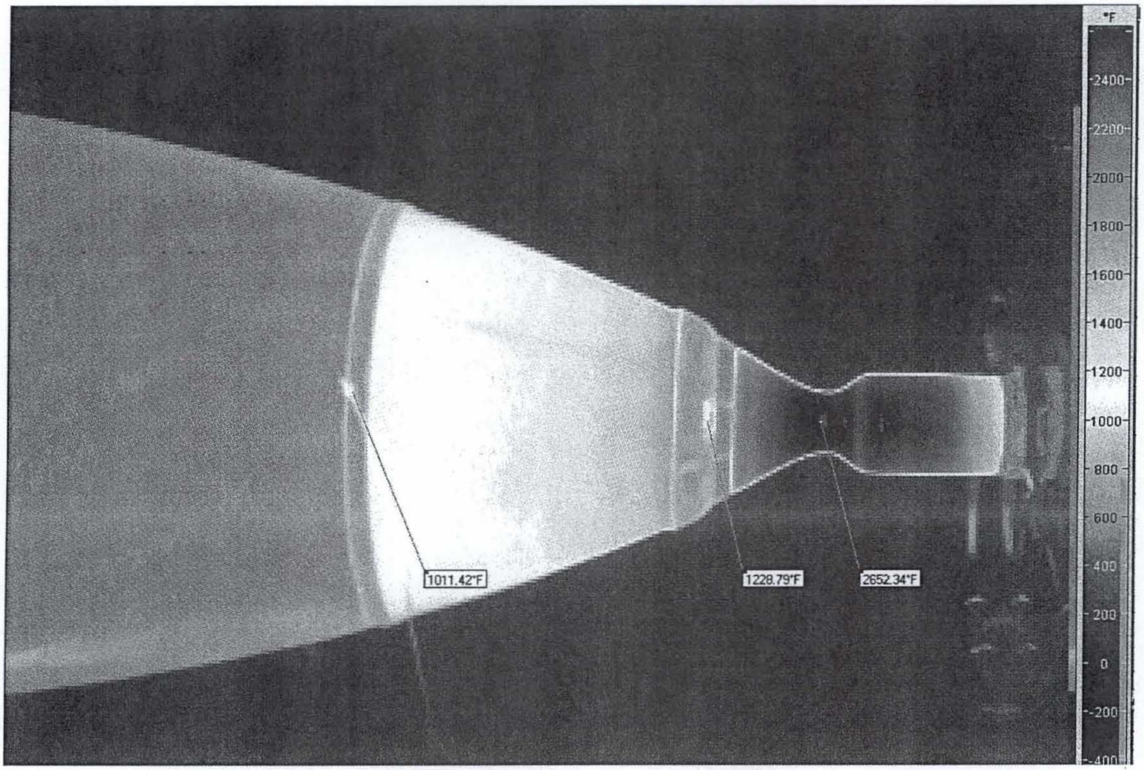

Figure 8. Thermal Image of Firing Engine

V. Advanced Chamber Materials

Fuel film cooling is essential to prevent hot combustion products from damaging chamber walls. The rhenium/iridium system is limited to maximum wall temperatures less than about 2500 K by the durability of the iridium coating[14]. Even though rhenium has a melting temperature of 3450 K, when exposed to oxidizing species at temperatures in excess of 873 K it will oxidize to rhenium heptoxide, Re_2O_7, with melting temperature of 636 K, resulting in rapid, catastrophic failure. For this reason, rhenium is coated with iridium, which provides an oxidation-resistant surface at a melting point of 2683 K. Engine operation duration is limited by iridium erosion and diffusion into the rhenium layer, which exposes rhenium to oxidation. It is obviously desirable to apply a different or thicker oxidation resistant coating that will withstand higher temperatures before diffusion and erosion become critical. Work to identify suitable coating materials is being pursued in parallel with this effort, at MSFC[14].

VI. Conclusions

A mission and systems analysis study has shown that increased combustion chamber pressure and temperature can be expected to yield significantly increased payload for selected spacecraft. Analysis indicates that test data has been obtained that are expected to facilitate design of a next generation LAE capable of meeting NRA goals of 330 seconds I_{sp} using NTO-MMH and 335 seconds I_{sp} using NTO-N_2H_4 propellants.

Acknowledgements

The work described in this paper was funded in whole or in part by the In-Space Propulsion Technologies Program, which is managed by NASA's Science Mission Directorate in Washington, D.C., and implemented by the In-Space Propulsion Technology Projects Office at Marshall Space Flight Center in Huntsville, AL during the period of performance. The program objective is to develop in-space propulsion technologies that can enable or benefit near and mid-term NASA space science missions by significantly reducing cost, mass and travel times.

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