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ADVANCED LAUNCH VEHICLE UPPER STAGES USING LIQUID PROPULSION AND METALLIZED PROPELLANTS

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ABSTRACT

Metallized propellants are liquid propellants with a metal additive suspended in a gelled fuel or oxidizer. Typically, aluminum (Al) particles are the metal additive. These propellants provide increases in the density and/or the specific impulse of the propulsion system. Using metallized propellant for volume- and mass-constrained upper stages can deliver modest increases in performance for Low Earth Orbit to Geosynchronous Earth Orbit (LEO-GEO) and other Earth orbital transfer missions. Metallized propellants, however, can enable very fast planetary missions with a single-stage upper stage system.

In this paper, trade studies comparing metallized propellant stage performance with non-metallized upper stages and the Inertial Upper Stage (IUS) are presented. These upper stages are both one- and two-stage vehicles that provide the added energy to send payloads to altitudes and onto trajectories that are unattainable with only the launch vehicle. The stage designs are controlled by the volume and the mass constraints of the Space Transportation System (STS) and Space Transportation System-Cargo (STS-C) launch vehicles. The influences of the density and specific impulse increases enabled by metallized propellants are examined for a variety of different stage and propellant combinations.

INTRODUCTION

With the potential expansion of operations and payload deliveries to Earth orbit, additional payload capability beyond the current IUS and the Titan IV/Centaur G-Prime may be required. Several robotic missions to other planets are planned as precursors to the piloted flights of the NASA Space Exploration Initiative. Also, future planetary missions will be increasingly complex and perform more propulsion-related maneuvers. These maneuvers include multiple orbit changes about the outer planets (as with the Galileo mission to Jupiter and Cassini mission to Saturn). When they require more maneuvering, they also become more propulsion-intensive and, consequently, more massive. Because of the large masses that are needed for these missions, advanced upper stages with high specific impulses (I_{sp}) may be required. Also, because of the limits of the capability of the IUS and potentially limited availability of the Titan IV/Centaur G-Prime for NASA missions, alternatives to these stages should be considered.

In the near future, high-energy upper stages may become scarce (Ref. 1). The Inertial Upper Stage (IUS) is currently the only large STS-compatible upper stage available to NASA. The Centaur G and G-Prime stages are currently no longer candidates for use in the STS and NASA is seeking alternative spacecraft designs and launch strategies (Ref. 2). Titan IV/Centaur G-Prime is a candidate for NASA missions but its availability to NASA may be limited. This is because of the number of payloads that have been delayed due to the STS launch delays, the high priority placed on Air Force missions and the potentially limited total production runs of the Titan IV.

The largest available stage for the STS is the Inertial Upper Stage (IUS). It has the capacity to deliver a 2268 kg payload to GEO. Current Department of Defense (DoD) planning for future missions will require a GEO payload up to 4536 kg. The IUS performance for planetary missions is also limited to low energy missions. The Galileo mission to Jupiter (Ref. 3) was launched on an IUS. Using the STS/IUS, its flight time is 6.5 years. With a high-performance cryogenic upper stage, the flight time would be reduced to 1.5 years. To fully exploit the capabilities of the STS and the planned STS-C, a new upper stage will be needed. Over the last several years, the Air Force and NASA have studied many potential configurations for future upper stages. These studies included stages using cryogenic, Earth- and space-storable propellants.

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The Adaptable Space Propulsion System (ASPS) Study addressed improvements to the current upper stages' capabilities for the Air Force (Refs. 4 and 5). Using the higher density of Earth- and space-storable propellants, a compact stage was designed to fulfill large payload delivery missions to GEO. An ASPS was designed to deliver 4,536 kg (10,000 lb_m) to GEO. Planetary missions were also considered. As a successor to this study, the Upper Stage Responsiveness Study (USRS) Study was conducted (Ref. 5). The U.S. Air Force Systems Command investigated a cryogenic propulsion upper stage for the Titan IV (Refs. 5-7). This stage was designed to send a minimum of 6,123 kg and up to 6,804 kg (13,500 to 15,000 lbm) to GEO.

In the 1980's, NASA embarked on the development of the STS/Centaur G-Prime, an STS-compatible O_2/H_2 stage (Ref. 8). This program was conducted in parallel with the Air Force STS/Centaur G upper stage development. In 1986, both programs were discontinued. These cancellations left the STS with no upper stage that could make the most-effective use of the Space Shuttle cargo capability to LEO.

At NASA, over the last decade, intensive studies of large space-based and ground-based Orbital Transfer Vehicles (OTV, Refs. 9, 10 and 11) and Space Transfer Vehicles (STV) have been conducted (Refs. 12 and 13). None of these studies, however, has been carried to the development of a flight vehicle. With no fixed design under consideration, alternative technologies should be considered to further improve the potential performance of future upper stages.

WHY METALLIZED PROPELLANTS?

One advanced propulsion system that can provide benefits for upper stages is metallized propellants. These propellants offer increases in the overall propellant density and/or the I_{sp} of a propulsion system. These increases can enable significant launch mass reductions or payload increases over conventional chemical propellants. Metallized propellants are propellants with metal added to the fuel or the oxidizer. Typically, the metal is in the form of micron-sized particles. They are suspended in gelled H_2 or other gelled fuel to increase its combustion energy and its density. The I_{sp} of an engine is proportional to:

$$I_{sp} \propto (T / MW)^{1/2}$$

where:

T   Combustion Temperature

MW   Molecular Weight of Combustion Products

Because of a combination of increased combustion temperature, or reductions in the molecular weight of the exhaust products, or both, the I_{sp} of the propulsion system is increased. The increases in propellant density reduce the tankage mass as well as the overall propulsion system dry mass. Because many of the propulsion system elements are dependent on the propellant mass and volume, the propellant density can have a large effect on the overall dry mass.

To increase the payload capability of existing launch vehicles and their upper stages, higher specific impulse (I_{sp}) systems and/or higher density propellants will be needed. Previous studies of Mars and lunar missions (Refs. 14, 15 and 16) determined that metallized propellants are an attractive alternative to O_2/H_2 for future space transportation systems. Higher density metallized Earth- and space-storable propellants were able to enhance the storability of propellants for a Mars ascent vehicle with a minimal increase in the LEO mass over O_2/H_2 propulsion (Ref. 15). For both Mars and lunar missions, the payload delivered to the surface can be increased: 20 to 33 percent added payload for the Mars mission (Ref. 15) and 3 percent for the lunar missions (Ref. 14). Many of these benefits are also directly applicable to upper stages. The STS and STS-C launch capabilities and cargo bay volumes impose strict constraints on an upper stage. Higher I_{sp} and higher density propellants can provide a way to increase the payload capability of a volume- and mass-constrained stage.

Safety is another important advantage of metallized propellants. Because the aluminum is gelled with the fuel, the gel prevents widespread spillage of the propellant if it were released. Cleanup of the spill is easier because the spill is restricted to a more confined area. Also, the gel makes the propellants less sensitive to high-energy particles that penetrate the propellant tank. If a projectile penetrates the propellant tank (such as a wrench dropped during ground assembly, space debris, etc.), the gel propellant will prevent a catastrophic explosion.

To see the benefits of metallized propellant for upper stages, the missions and propulsion system designs must be considered together and analyzed. The succeeding sections will discuss these aspects and the results of the overall systems analysis.

PROPULSION SYSTEMS ANALYSES

In determining the potential performance advantages of metallized propellants, a series of trade studies were performed. These studies used the launch mass and volume constraints of the STS and STS-C to define the capability of future upper stages. After determining the launch vehicle constraints and formulating the missions and generic designs of the stages, these elements can be folded together to find the performance of the stages for the varying mission requirements.

In the analyses presented here, two figures of merit will be considered. These are the payload delivery mass to an Earth orbit and the injected mass onto a planetary trajectory.

To compute the figures of merit, the rocket equation is used:

$$\Delta V = I_{sp}\, g \ln (m_o/m_f)$$

where:

ΔV Velocity Change (m/s)

I_{sp} Specific Impulse (lb_f-s/lb_m)

g Gravitational Acceleration (9.81 m/s^2)

m_o Initial Mass (kg)

m_f Final Mass (kg)

Using the rocket equation, the launch vehicle constraints, the engine performance and the upper stage mass-scaling equations, the payload or the injected mass can be calculated. In the following sections, these constraints on the upper stage designs are discussed.

LAUNCH VEHICLE CONSTRAINTS

The upper stage capability in the results section will be presented for both STS and STS-C launched payloads. Both have significantly different payload capabilities to LEO: 24,950 kg (55,000 lb_m) for the STS and 68,040 kg (150,000 lb_m) for the STS-C. Also, the payload bay lengths are different: 18.3 m (60 feet) for the STS and 25 m (82 feet) for the STS-C. Both have a payload bay diameter of 4.57 m (15 feet).

For both the STS and STS-C, a set of airborne support equipment was included to hold the upper stage within the cargo bay and provide an erection table to elevate the stage for deployment. The mass of the support equipment was 4109 kg (Ref. 17). This mass is subtracted from the payload capability of the launch vehicle when performing the estimates of the upper stage's performance. The total masses available for the upper stages are 20,841 kg and 63,931 kg for the STS and STS-C, respectively.

PROPULSION SYSTEM DESIGN

Engine Performance. Using a computer simulation code (Ref. 18), the engine performance of the metallized propellant combinations was estimated. The expansion ratio (ϵ) for the O_2/H_2 engines was 500:1 and was selected for the stages based on the designs of planned engines. The engine chamber pressure was 1000 psia. This chamber pressure was selected based upon the designs of the various engines under consideration for the upper stage application. The propellants were provided to the combustion chamber in the liquid state.

Table I contrasts the predicted performance of several propulsion systems with and without metallized fuel. The increases in I_{sp} are several lb_f-s/lb_m. Using metallized $O_2/H_2/Al$, an increase in I_{sp} of 5.9 lb_f-s/lb_m is possible over an O_2/H_2 system. An engine I_{sp} efficiency was used to modify the code-predicted I_{sp}. The I_{sp} efficiency (η) is the ratio of the delivered engine performance and the code-predicted I_{sp}. This reduction reflects the losses incurred due to the nozzle boundary layer, engine cycle inefficiencies and other propulsion system losses. The engine efficiencies were derived using the performance estimates from References 19 through 22 and comparisons with the vacuum I_{sp} predicted by the engine code. In this analysis, metallized propellants have the same engine efficiency as the non-metallized systems. There are additional losses that have not been included in this analysis that may potentially penalize the metallized propellant cases, such as two-phase flow losses in the exhaust and the nozzle boundary layer, and nozzle erosion. Numerical modelling, propellant rheology experiments and hot-fire engine testing have been conducted to determine the potential engine efficiency of metallized propellants (Refs. 27 through 30). Without the predicted increases in I_{sp}, the advantages of these propellants are significantly reduced. The effect of lower than predicted I_{sp} efficiency will be discussed later in the paper.

Table I
Metallized Propellant Engine Performance

Propellant	I_{sp} (lb_f-s/lb_m)		I_{sp} Efficiency (η)
	No Metal	Metal-lized	
NTO/MMH	341.2	366.4	0.938
O_2/MMH	381.9	386.2	0.940
O_2/CH_4	382.1	384.3	0.940
O_2/H_2	479.5	485.4	0.984

Expansion Ratio = 500:1
Chamber Pressure = 1000 psia
Aluminum used in the metallized fuel

The mixture ratios and the metal loading for these designs are given in Table II. The metal loading represents the fraction (by mass) of aluminum in the total mass of the fuel. The mixture ratio is defined as it is for traditional chemical propulsion: the ratio of the total oxidizer mass to the total fuel mass. In selecting the "best" metallized system design, the propellant metal loading, its effects on the engine I_{sp} and the propulsion system dry mass must be analyzed. Some of the issues that are important in determining the appropriate design for a metallized propulsion system are discussed below: the propellant density, the performance and the system dry mass.

Table II
Metallized Propellant Engine Design Parameters

Propellant	Mixture Ratio (Metal Loading)	
	No Metal	Metallized
NTO/MMH	2.0 (0.0)	0.9 (50.0)
O_2/MMH	1.7 (0.0)	0.9 (35.0)
O_2/CH_4	3.7 (0.0)	1.8 (45.0)
O_2/H_2	6.0 (0.0)	1.6 (60.0)

Aluminum used in the metallized fuel

Propellant Density. Using the aluminum loadings considered in the engine performance calculations, the propellant density for the H_2 fuel can increase from 70 kg/m^3 to 169 kg/m^3 (H_2 with a 60-percent aluminum loading). The density increase is computed using:

$$\rho_{p,m} = 1 / ([1 - ML]/\rho_p + ML/\rho_m)$$

where:

$\rho_{p,m}$ Density of Metallized Fuel (kg/m^3)

ML Metal Loading (Fraction of Fuel Mass)

ρ_m Density of Metal in the Fuel (kg/m^3)

ρ_p Density of Nonmetallized Fuel (kg/m^3)

Selection of the Best Density-I_{sp} Design Points. To deliver the maximal reduction in LEO mass or the maximal payload increase, trade studies must be conducted to determine the "best" I_{sp} and density for each propulsion system. Figure 1 shows the effect of metal loading on I_{sp} for $O_2/H_2/Al$. The maximal metal loading considered was 60 percent of the fuel mass. A higher I_{sp} is produced at higher metal loadings. The selection of the 60-percent loading performance level was guided by the metal loading experience with solid rocket motors. The total metal loading of all of the propellant (oxidizer and fuel) of the $O_2/H_2/Al$ propulsion system was 23 percent. This loading is comparable to that of existing solid propulsion systems. An I_{sp} of 485.4 lb$_f$-s/lb$_m$ was delivered at a metal loading of 60 percent of H_2 in the H_2/Al fuel, an ϵ of 500:1 and a mixture ratio of 1.60.

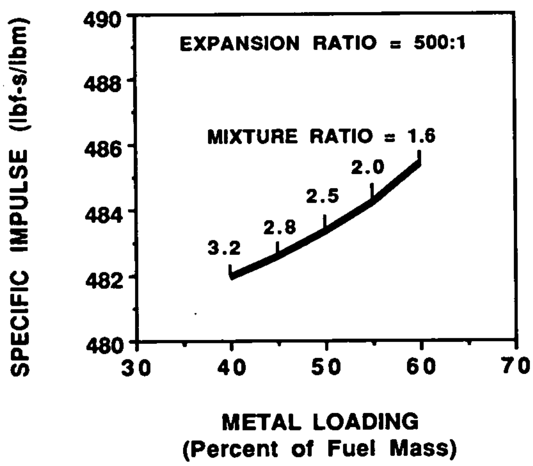

Figure 1. $O_2/H_2/Al$ I_{sp} versus Metal Loading

Because the $O_2/H_2/Al$ bulk density decreases slightly with metal loading over O_2/H_2, the peak I_{sp} design point for $O_2/H_2/Al$, however, may require a heavier propulsion system than the nonmetallized design case. Reference 15 compares the propulsion mass scaling equations for several metal loadings. There is a small variation in the total mass of the propulsion system with the different metal loadings. Based on the trade studies, the highest I_{sp} system of the range in Figure 1 (which has a metal loading of 60 percent) was selected. For all of the remaining metallized combinations, the metal loading was selected to provide the maximal I_{sp} for the propulsion system. The remaining propellant combinations produce an overall density increase. This increase reduces the propellant tank volume and reduces the overall dry mass.

If the benefits of reduced LEO mass or increased payload are not desired or significant, the effects of increased propellant density can still be a benefit to upper stages. Because of the increased density, the propellant tankage size can be reduced, potentially offering better and smaller tank configurations. As an example, for the fixed stage using NTO/MMH/Al, the propellant tank volume is reduced over that for the NTO/MMH case. In the metallized system, the total propellant tank volume was reduced to 41.4 m^3 versus the 49.4 m^3 required for the non-metallized NTO/MMH case.

Although the tankage volume decreased in the NTO/MMH/Al case, other applications of metallized propellants, such as $O_2/H_2/Al$, will show a small tankage volume increase. This is due to the lower mixture ratio of the metallized $O_2/H_2/Al$ system over the O_2/H_2 system. In the fixed $O_2/H_2/Al$ upper stage, the total O_2 tank volume can be reduced from 43.7 m^3 to 31.4 m^3 for the metallized case. The H_2 tank volume, however, increased from 119.6 to 133.7 m_3 with metallized propellants. Overall, the total tank volume increased from 163.3 to 165.1 m^3 (a difference of 1.8 m^3 or 1.1 percent). This example is for the case for the STS-C (sizing the stage for the 160-km^2/s^2 C_3 mission) for both the metallized and the non-metallized O_2/H_2 systems. Though the propellant tank volume increased, the higher I_{sp} enabled by metallized propellants provides 79 percent more payload to a C_3 of 160 km^2/s^2.

Pump-Fed and Pressure-Fed Systems. With the very-high performance O_2/H_2 systems being considered for upper stages, a pump-fed engine is required. Pressure-fed propulsion systems typically require larger masses for propellant tankage and pressurization systems. Using metallized propellants, the propellant feed system must be designed to supply the non-Newtonian, thixotropic metallized propellant with the same reliability as the non-metallized H_2. Currently, metallized propellants are fed to small propulsion systems with positive-displacement propellant expulsion devices (diaphragms, etc., Ref. 31). A positive expulsion system and a pressure-fed system, however, are considered impractical and too massive for large propellant tanks. For the extremely-large propellant loads needed on upper stages, a way of effectively using pump-fed engines will be required.

Mass Scaling Equations. In determining the dry mass of the transfer vehicles, the following general mass-scaling equation was used:

$$m_{dry} = A + B\ m_p$$

where:

A, B Mass Parameters

Table III lists the propulsion mass-scaling parameters for all of the considered systems. These parameters include all of the masses that are required to store and deliver propellants to the main engines. They include tankage, engines, feed system, thermal control, structure, residuals and contingency. The parameter A of the scaling equations varies due to the different configurations of spherical and cylindrical tankage. Only the O_2/H_2 and $O_2/H_2/Al$ stages required special consideration for the use of cylindrical tanks. This is due to the relatively low density of the H_2 and H_2/Al metallized fuels. The B parameter is dependent upon the propellant mixture ratios, the propellant metal loading and hence the propellant density. The specific mixture ratios and the metal loadings are listed in Table II.

Table III
Propulsion Mass-Scaling Parameters

Propellants	A	B	Application
NTO/MMH	440.00	0.1358	STS, STS-C
NTO/MMH/Al	440.00	0.1345	STS, STS-C
O_2/MMH	440.00	0.1396	STS, STS-C
O_2/MMH/Al	440.00	0.1376	STS, STS-C
O_2/CH_4	440.00	0.1458	STS, STS-C
$O_2/CH_4/Al$	440.00	0.1440	STS, STS-C
O_2/H_2	355.12	0.1598*	STS-C
O_2/H_2	373.80	0.1576**	STS
$O_2/H_2/Al$	373.80	0.1584**	STS, STS-C

* Cylindrical O_2 and H_2 Tanks
** Spherical O_2 Tank, Cylindrical H_2 Tank
All Other Tankage is Spherical

All of the tankage configurations considered in the study were based on the ability to package the stage within the STS and STS-C cargo bay volume. For the $O_2/H_2/Al$ and O_2/H_2 stages, cylindrical tankage was required to fit the H_2 and H_2/Al metallized fuel tankage within the 4.3-m diameter cargo bay. A cylindrical tank was also used for the O_2 tank of the O_2/H_2 stage used in the STS-C. All of the remaining tankage for all of the other upper stages was spherical.

The propellant tankage for all of the systems is designed for a 50-psia maximal operating pressure. The propellant is stored at 30 psia. All of the tankage for O_2, H_2 and CH_4 is composed of aluminum alloy (2219-T87). The tanks for NTO and MMH are made of titanium (Ti-6Al-4V). The flange factor and safety factor are 1.4 and 2.0, respectively, for the propellant tanks. The safety factor is based on the tank material ultimate stress. The propellant residuals and holdup mass is 1.5 percent of the total propellant mass. The percentage accommodates a small added propellant mass for cryogenic propellant boiloff. Because the stages are expendable, no large allowance was made for propellant losses due to boiloff.

Each cryogenic O_2/H_2 propulsion system uses autogenous pressurization. The NTO/MMH and the space-storable systems use regulated pressurization. The pressurant is helium. In the pressurant tank, the maximal operating pressure is 3722 psia. The storage pressure is 3444 psia. The flange factor and safety factor for the pressurant tanks are 1.1 and 2.0, respectively. For the autogenous systems, a small helium pressurization system is included. It can pressurize one-tenth of the total propellant tank volume. For thermal control, the cryogenic propellants (O_2, H_2 and CH_4) use a high-performance multilayer insulation (Ref. 8). The storable propellants only require a lower-performance multilayer insulation.

MISSION REQUIREMENTS

The missions under consideration for these large upper stages include two major categories: Earth orbital and planetary. Payload deliveries to GEO and other high Earth orbits are needed (Refs. 1 and 23). The systems to be placed there are communications satellites, observational systems, and other remote sensing satellites, such as those for the Mission to Planet Earth.

One-way LEO-GEO transfers, high-inclination Earth-orbital transfers and planetary mission performance will be considered. Each mission is described by a mission velocity change (ΔV) or an injection energy (C_3).

LEO-GEO Orbit Transfer ΔV. Using the Hohmann orbit-transfer equations (Ref. 24), the ΔV for a minimum energy transfer is computed. The initial altitude for the mission is 241 km. The total one-way ΔV for the LEO-GEO mission is 4.253 km/s and the total plane change is 28.5 degrees. This ΔV must be delivered in two firings. One is the initial firing to place the spacecraft onto an elliptical transfer orbit. The second firing circularizes the orbit at GEO. The orbit transfer equations are:

$$\Delta V = \Delta V_{te} + \Delta V_{circ}$$

$$\Delta V_{te} = V_o \ [(1 + 3\ R)/(1 + R) - C1\ ]^{0.5}$$

$$C1 = 2\ [\ 2\ R\ /\ (1 + R)]^{0.5} \cos\ (\theta_{tot} - \theta_{circ})$$

$$R = r_f/r_o$$

$$V_o = (\mu/r_o)^{0.5}$$

and

$$\Delta V_{circ} = V_o\ [(3 + R)/\ (R\ (1 + R)) - C2\ ]^{0.5}$$

$$C2 = (2\ /\ R)\ [\ 2\ /\ (1 + R)]^{0.5} \cos\ (\theta_{circ})$$

where:

V	orbital velocity (km/s)

r	orbital radius (km)

θ orbital plane change (radians)

μ Earth's Gravitational Constant (398601.3 km^3/s^2)

<u>subscripts</u>

circ circularization

f final

o initial

te transfer ellipse

tot total

The variable θ_{tot} is the total plane change to be conducted during the orbit transfer. Variable θ_{circ} is the plane change performed during the circularization firing. An optimum split between the transfer ellipse and the circularization ΔV was included in the calculation. The ΔV_{te} is 2.468 km/s and includes 2.2 degrees of the plane change. This orbit's apogee will be at the GEO altitude. The second firing (ΔV_{circ}) is performed at GEO. This ΔV is 1.785 km/s. The remaining 26.3 degrees of the plane change are performed during the GEO burn.

<u>Other Earth Orbital Transfer ΔVs</u>. Other Earth orbital missions are under consideration for the Strategic Defense Initiative (SDI) missions: 10000- to 17935-km altitudes with a 65-degree inclination (Ref. 23). The ΔVs for several different orbital transfers are listed in Table IV. These ΔVs were computed using the same equations discussed above. The 36.5 degree inclination change represents a transfer from a 28.5-degree to a 65-degree inclination orbit.

Table IV
Orbit Transfer ΔVs:
Earth Orbital Missions

Mission	ΔV (km/s)	ΔInclination (degrees)
LEO-GEO*	4.253	28.5
LEO-10,000 km	4.293	36.5
LEO-17,935 km	4.367	36.5

* LEO is defined as a 241-km altitude orbit (28.5-degree inclination) and GEO is a 35870-km orbit (0-degree inclination).

<u>Planetary Mission Injection Energy</u>. The performance of an upper stage is described by the delivered injected mass to a specific injection energy (C_3). The C_3 is the hyperbolic excess velocity squared and is defined by:

$$C_3 = ([\ \mu/r_o\ ]^{0.5} + \Delta V\ )^2 - 2\ \mu/r_o = V_\infty^2$$

where:

C_3 Injection Energy (km^2/s^2)

ΔV Velocity Change (km/s)

μ Earth's Gravitational Constant (398601.3 km^3/s^2)

r_o Orbital Radius (km) or 6378.14 km + Orbital Altitude (km)

V_∞ Hyperbolic Excess Velocity

For planetary missions, the figure of merit for the upper stage is the injected mass. This is the total mass (above the upper stage's dry mass) that is placed onto an interplanetary trajectory. It includes the payload and the adapter between the payload and the stage.

Existing upper stages cannot provide the high injection energies for fast missions. Table V lists some past, planned and potential planetary missions (Refs. 3, 25 and 26). The injected masses and the injection energies for the missions are provided. Currently, the Saturn Orbiter/Titan Probe (SOTP) mission is named the Cassini mission. It has an injection energy that is very low: only 28 km^2/s^2. This limit on C_3 is imposed by the Titan IV/Centaur G-Prime capability. The launch vehicle limitation forces the spacecraft to fly a ΔV Earth Gravity Assist (ΔVEGA) trajectory. On such a trajectory, the spacecraft is placed on a flight path that returns to the vicinity of the Earth. This Earth flyby adds the required energy to the spacecraft and sends it on its way to the planet. This adds from 1.5 to 3 years to the flight time of the mission. As is planned with SOTP, the Galileo mission was launched at a low C_3: 17 km^2/s^2. This lower C_3 requires the mission to use a Venus-Earth-Earth Gravity Assist (VEEGA) trajectory: a flyby of Venus and two Earth flybys. Using this flight path adds 5 years to the flight time. A direct trajectory with a C_3 of 80 km^2/s^2 would only require 1.5 years to reach Jupiter. Advanced upper stages can produce a higher C_3, shorten the mission flight times and allow a faster return of the science from the spacecraft.

Table V
Potential Planetary Mission Requirements
(Refs. 3, 25 and 26)

Mission	Injected Mass (kg)	C_3 (km^2/s^2)
Galileo (Direct)	2,550	80.0
Saturn Orbiter/ Saturn Probe (SOTP, Direct)	2,488	109.0
Titan Flyby/ Titan Probe (TFTP)	1,575	136.9
Uranus Flyby/ Uranus Probe (UFUP)	1,298	150.0
Pluto Flyby	700	160.0

RESULTS

In this section, the results of the systems analyses for several different upper stage missions will be discussed. Both one- and two-stage systems were considered. The stage performance for optimized vehicles and fixed stage designs was addressed. Optimized vehicles are those whose stage weights are matched to the specific missions being considered. For example, for differing payloads, the propellant load and the dry mass of the stage were varied such that the entire vehicle's mass was maintained at the limit of the STS or STS-C payload mass. Thus the "optimal" or maximum performance was gained every mission for each payload. This optimal performance over the full range of payloads, however, is only theoretically achievable. In actuality, a fixed stage is used on a launch vehicle. Fixed stages are those that have a fixed dry mass over the range of missions for which it was considered. Because it is has a fixed mass, it will operate "non-optimally" and not deliver the maximum payload for all conditions other than the design point.

The analyses presented here are for optimized and fixed stage designs. Both sets of results are discussed to show the maximal performance potential of the upper stages. The optimized stage results are presented first. After these results are discussed, the fixed stage performance is presented.

OPTIMIZED STAGES

LEO-GEO. A performance comparison of the non-metallized and the metallized stages for the LEO-GEO mission is shown in Figures 2 (for the STS) and 3 (for the STS-C). The $O_2/H_2/Al$ system is able to deliver the highest payload to GEO: 6,226 kg for the STS and 19,211 kg for the STS-C. Using O_2/H_2, however, nearly the same payload can be delivered as with the $O_2/H_2/Al$ system. For the LEO-GEO missions, metallized $O_2/H_2/Al$ provides only a 1.6 percent payload increase over O_2/H_2 with the STS and 1.7 percent increase with the STS-C. Using the space-storable propellants, the payload delivered ranged from 4,300 on the STS to 14,000 kg with the STS-C. The percentage savings with the metallized space-storables were similar to that for metallized $O_2/H_2/Al$: 1.8 to 1.6 percent (for STS and STS-C, respectively) for the $O_2/CH_4/Al$ and 3.0 to 2.6 percent for the $O_2/MMH/Al$ over its non-metallized counterpart.

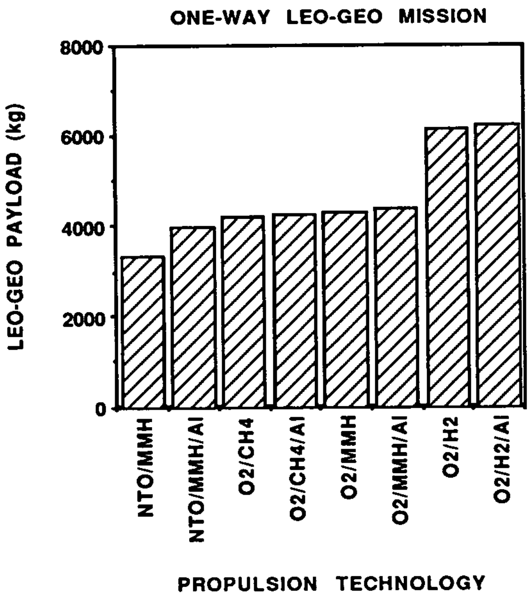

Figure 2. Payload Capability With STS: LEO-GEO

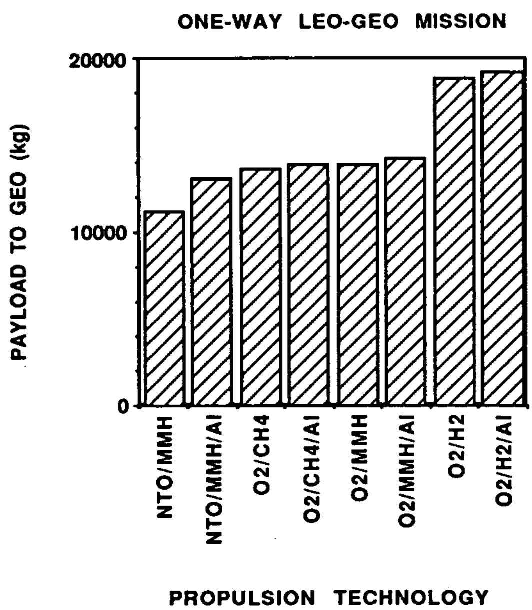

Figure 3. Payload Capability With STS-C: LEO-GEO

The largest percentage gain of any of the metallized combinations is with NTO/MMH/Al. This gain delivers a 17 to 19 percent benefit over NTO/MMH. As a replacement for the IUS, a storable NTO/MMH/Al upper stage can provide a significant increase in delivered payload. An NTO/MMH/Al stage can send 3970 kg to GEO with the STS and 13,090 kg with the STS-C. This is in contrast with the 2268-kg IUS GEO capability with either the STS or STS-C.

All of the systems, both metallized and non-metallized, can enable large payload increases over the IUS. The non-metallized space-storable and cryogenic O_2/H_2 propellants, however, produce a comparable payload performance to their metallized counterparts. In most cases, the benefits of metallized propellants over the non-metallized systems are a modest 1.6 to 3 percent for the LEO-GEO missions. The only exception is the NTO/MMH/Al system. Because of its large percentage gains on GEO payload over NTO/MMH (19 percent on the STS and 17 percent on the STS-C), NTO/MMH/Al is the only metallized propellant combination that produces significant payload gains for this mission class. Non-metallized space-storable and cryogenic propellants are also excellent propulsion options.

Other Earth Orbit Transfers. In Table VI, the payload capabilities of all of the propulsion technologies for two Earth-orbital missions are presented. For these other Earth-orbital missions, the payload gains with metallized propellants are similar to those for the LEO-GEO mission. On the 10,000-km mission (with a 65 degree inclination) and the mission to 17,935 km (65 degree inclination), the payload increases for metallized propellants range from 1.7 to 3.1 percent. This is the performance range for the $O_2/H_2/Al$, $O_2/CH_4/Al$ and the O_2/MMH/Al systems. Again, the NTO/MMH/Al system produced the greatest increase over its non-metallized counterpart: 20 percent (with the STS) and 17 percent (with STS-C). The total payload delivered with NTO/MMH/Al was 3890 on the 10,000 km mission (with the STS) and 12860 kg with the STS-C. As with the GEO missions, metallized NTO/MMH/Al propellants provide a substantial payload gain over the NTO/MMH system. This metallized combination is the only one with large payload benefits for this Earth orbital mission class.

Table VI
Payload Capabilities:
Highly-Inclined Earth Orbit Transfer Missions

Propulsion Technology	Payload Mass (kg)	
Mission Altitude (km) (65 degree inclination)	10,000	17,935
STS Mission:		
NTO/MMH	3267.8	3123.6
NTO/MMH/Al	3892.6	3743.8
O_2/CH_4	4095.6	3943.9
$O_2/CH_4/Al$	4171.9	4020.0
O_2/MMH	4179.6	4028.6
O_2/MMH/Al	4306.5	4154.8
O_2/H_2	6042.8	5886.7
$O_2/H_2/Al$	6142.9	5987.1
STS-C Mission:		
NTO/MMH	10997.3	10559.9
NTO/MMH/Al	12857.0	12420.7
O_2/CH_4	13452.4	13010.0
$O_2/CH_4/Al$	13672.1	13231.5
O_2/MMH	13692.0	13254.3
O_2/MMH/Al	14054.2	13619.0
O_2/H_2	18674.1	18255.7
$O_2/H_2/Al$	18990.3	18575.9

As with the LEO-GEO missions, the space-storable and cryogenic propellants deliver very significant payload masses for these other Earth orbital flights. Their payload capability exceeds that of the NTO/MMH/Al system. This is especially true of the cryogenic O_2/H_2 system. Using these non-metallized space-storable and cryogenic systems are therefore highly beneficial.

<u>Planetary Missions</u>. Using the same upper stage mass-scaling equations used for the LEO-GEO analysis, the performance for planetary injections was determined. These stages' performance were first computed to determine the maximum deliverable injected mass. The stage mass was determined with the propulsion mass-scaling equations of Table III. These optimized stage designs were used later to define a fixed stage and determine the performance differences between it and the optimized system.

The STS-C with a large $O_2/H_2/Al$ upper stage can be an effective tool for conducting fast planetary missions. The STS and STS-C are both able to deliver very significant payloads onto planetary trajectories. The STS-C with a high-energy upper stage, however, delivers a large increment in performance over the STS. Figure 4 compares the STS and STS-C capabilities with $O_2/H_2/Al$ upper stages. The STS can capture one of the missions listed (Galileo) whereas the STS-C can capture all of the missions.

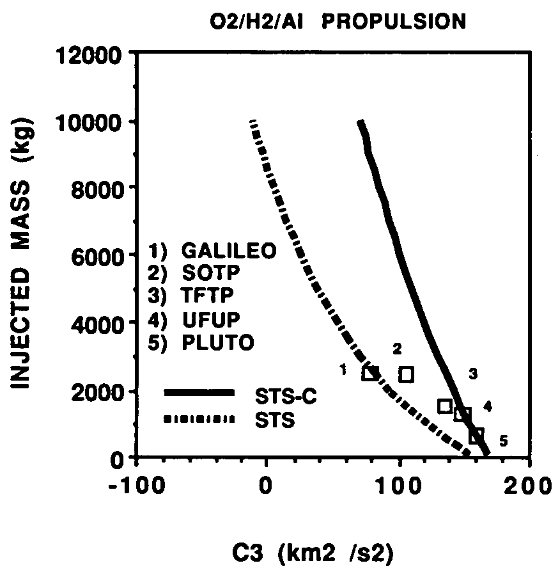

Figure 4. Comparison of $O_2/H_2/Al$ Performance With STS and STS-C

Table VII lists the values of C_3 at which metallized propellants produce a minimum of 10 percent increase in the injected mass (over their non-metallized counterparts) for planetary missions. With metallized propellants, the highest benefit is gained on very-high energy missions. In the comparison of the O_2/H_2 and the $O_2/H_2/Al$ propulsion cases on the missions with a C_3 above 124 km^2/s^2, metallized propellants were able to deliver greater than 10 percent additional payload. At a C_3 of 150 (STS-C), a 28-percent increase is delivered. With the very-high C_3 of 160 km^2/s^2 (STS-C), a 79-percent increase is possible.

Table VII
Minimum Payoff C_3 for High Energy Planetary Missions

Propulsion Technology	Payoff C_3 (km^2/s^2) STS	STS-C
NTO/MMH/Al	-4.5	3.0
O_2/MMH/Al	76.3	82.5
$O_2/CH_4/Al$	83.4	91.5
$O_2/H_2/Al$	127.6	123.8

In general, the payoff C_3 for the STS missions is lower than those using the STS-C. The payoff occurs at a lower C_3 for the STS because the STS stages are smaller than those on the STS-C. An increase in the stage's I_{sp} will improve the stage's performance more rapidly for the smaller stages. Therefore, the payoff occurs at a lower C_3. The only exception to this statement is the case with $O_2/H_2/Al$. This is because of the higher mass penalty paid by the stages with cylindrical tankage. Due to the STS-C cargo bay volume, the O_2/H_2 stage requires the use of cylindrical tankage for both propellants. This places a mass penalty on this stage over the $O_2/H_2/Al$ stage. The metallized stage only uses cylindrical tankage for the H_2/Al fuel.

The $O_2/H_2/Al$ upper stage in the STS-C is the only system that can produce the needed C_3 for all of the fast planetary missions. As an example, in Figure 5, the injected masses for the Uranus Flyby/Uranus Probe (UFUP) mission are contrasted. This mission needs a C_3 of 150 km^2/s^2. With $O_2/H_2/Al$, the margin for the mission is 157 kg. The margin is the injected mass delivered over and above that required for the mission (listed in Table V). The O_2/H_2 system falls short of the required performance. A Pluto Flyby mission (Ref. 26) with a C_3 of 160 km^2/s^2 is also enabled with the $O_2/H_2/Al$ system.

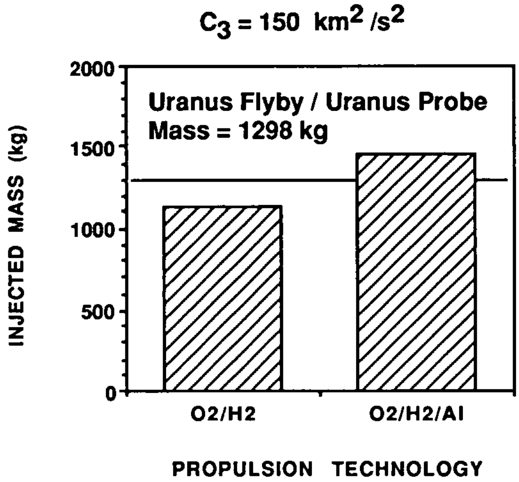

Figure 5. Planetary Mission Performance: UFUP With STS-C

Systems other than $O_2/H_2/Al$ propulsion are capable of capturing some of the planetary missions. Figure 6 compares NTO/MMH and NTO/MMH/Al upper stage performance for a Galileo-class mission. This mission is a high-energy injection to Jupiter with a C_3 of 80 km^2/s^2. Using NTO/MMH/Al (with the STS-C), the upper stage is able to deliver the needed injected mass of 2550 kg with a large 640-kg margin. Without metallized propellants, only 1620 kg could be delivered to this C_3.

An analysis of the influence of the I_{sp} efficiency on the performance of metallized $O_2/H_2/Al$ systems was conducted. Figure 7 shows the injected mass for the UFUP mission versus I_{sp} efficiency. The O_2/H_2 system has a 98.4-percent efficiency. In this example, once the I_{sp} efficiency equals 97 percent, the injected mass for the $O_2/H_2/Al$ and the O_2/H_2 systems are the same. Below an efficiency of 97.7-percent, the UFUP mission is no longer enabled. This shows the critical importance of high I_{sp} efficiency.

Two-Stage Vehicle Performance. Past liquid propulsion upper stage systems, such as the Centaur, Centaur G and G-Prime, have not considered two-stage vehicles. Augmentation of the stages' C_3 with small solid rocket motors has been conducted (as with the Pioneer 10, 11 and Voyager 1 and 2 outer planet spacecraft). Adding these solid rocket motors made effective use of vehicle staging. Many of the high-C_3 missions, however, can gain significant benefits from a specially-tailored two-stage system of high-energy liquid upper stages.

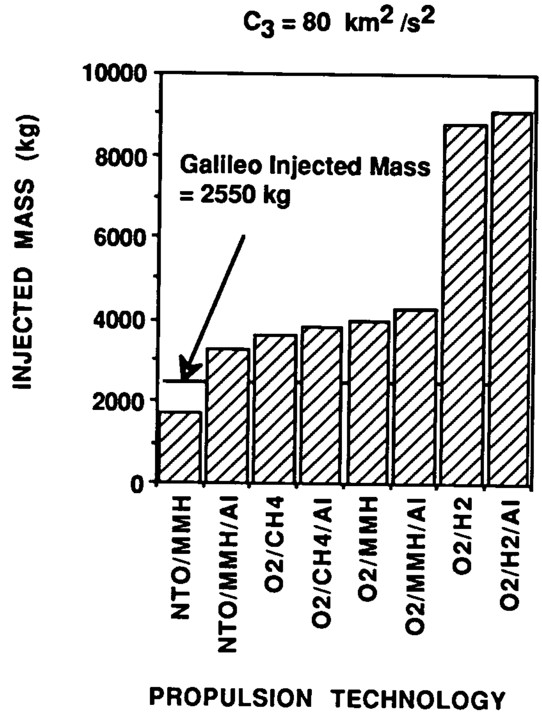

Figure 6. Planetary Mission Performance: C_3 = 80 km^2/s^2 With STS-C

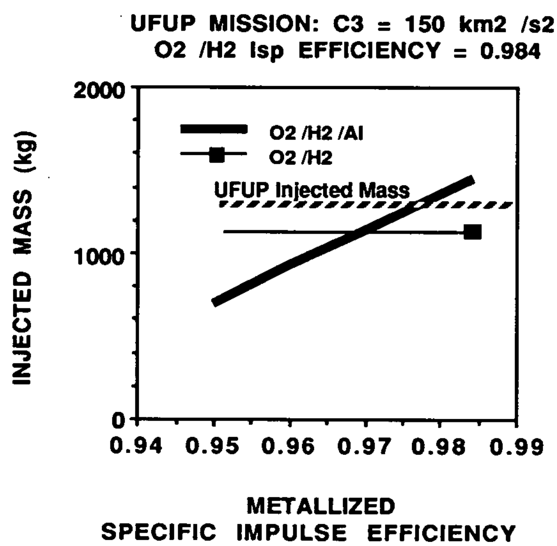

Figure 7. Injected Mass vs. I_{sp} Efficiency

An assessment of the performance differences for two-stage vehicles for the planetary missions was conducted. Tables VIII and IX contrast the performance of single- and two-stage vehicle performance for three missions. In the tables, the STS and STS-C-constrained vehicle performance is significantly enhanced through staging. The planetary capability of such a two-stage vehicle over the single-stage counterpart is considerable. Clearly, the highest performance gains are at a high C_3. With the two-stage $O_2/H_2/Al$ system, potentially all of the advanced planetary missions can be "captured". Similarly, with the STS-C two-stage $O_2/H_2/Al$ system, there is an even higher capability, allowing for even more massive and propulsion-intensive planetary missions. The O_2/H_2 system (STS or STS-C) can deliver sufficient C_3 to capture nearly all of the planetary missions. An important result of this analysis is that the two-stage system can enable a large enough performance increase that the system can capture all of the planetary missions without using metallized propellants.

Table VIII
Payload Capability for Planetary Missions:
Single- and Two-Stage Performance Comparison
STS Launch Mass = 24,950 kg
Total Stage Wet Mass and Injected Mass = 20,841 kg

Propulsion Technology		Injected Mass (kg)	
Number of Stages		One	Two
C_3 (km^2/s^2):	80		
NTO/MMH		221.9	1227.0
NTO/MMH/Al		783.3	1595.9
O_2/H_2		2515.5	3114.0
O_2/H_2/Al		2603.6	3188.0
C_3 (km^2/s^2):	150		
O_2/H_2		121.1	1327.0
O_2/H_2/Al		208.9	1380.0
C_3 (km^2/s^2):	160		
O_2/H_2		--*	1163.0
O_2/H_2/Al		--	1215.0

* Not capable of delivering a payload to this C_3.

Table IX
Payload Capability for Planetary Missions:
Single- and Two-Stage Performance Comparison
STS-C Launch Mass = 68,040 kg
Total Stage Wet Mass and Injected Mass = 63931 kg

Propulsion Technology		Injected Mass (kg)	
Number of Stages		One	Two
C_3 (km^2/s^2):	80		
NTO/MMH		1620.1	4936.0
NTO/MMH/Al		3191.7	6092.2
O_2/H_2		8461.2	10610.0
O_2/H_2/Al		8824.9	10875.0
C_3 (km^2/s^2):	150		
O_2/H_2		1133.9	5140.0
O_2/H_2/Al		1455.1	5240.0
C_3 (km^2/s^2):	160		
O_2/H_2		403.7	4530.0
O_2/H_2/Al		721.0	4717.0

A two-stage system, while it promises very high performance, may not always be considered as a primary option over a single-stage vehicle. The bulk of the planetary and Earth orbital traffic planned for the near and foreseeable future requires relatively low-energy injections. Lower-energy capability vehicles (such as single-stage liquid stages) are, in some cases, decided upon due to the desire fulfill the needs of a broad number of users and constraints of existing stages and launch vehicles.

FIXED STAGES

In the previous discussion, the maximal performance for the one-stage systems was analyzed. Using these data, a performance assessment of a fixed stage was conducted. The fixed stage design point was chosen based on two major factors. The first of these factors is the stage's ability to perform a wide range of planetary missions. The second is the design point where the stage can deliver the maximal payload benefit. Simply put, the fixed stage design point was selected to gain the maximum benefit for the widest variety of missions.

The performance of two types of fixed and optimized stages are compared in Figure 8: NTO/MMH/Al and $O_2/H_2/Al$, both using the STS-C. This analysis was conducted to assure that a fixed stage design could still perform a wide range of the planetary missions. For both systems, the large differences in performance are primarily at the lower injection energies. With the NTO/MMH/Al upper stage, the design point that was selected was a C_3 of 80 km^2/s^2. This C_3 was chosen to capture the Galileo-class mission. This stage is compatible with the STS-C and has a burnout mass of 7589 kg. Table X provides a mass summary for the two systems. For the stage using $O_2/H_2/Al$, the C_3 used for the design point was 160 km^2/s^2. It is also designed for the STS-C and has an 8966-kg burnout mass. At this design point, the stage can still perform all of the desired planetary missions. Using $O_2/H_2/Al$ with the STS-C is the only single-stage propellant combination that will capture all of the missions.

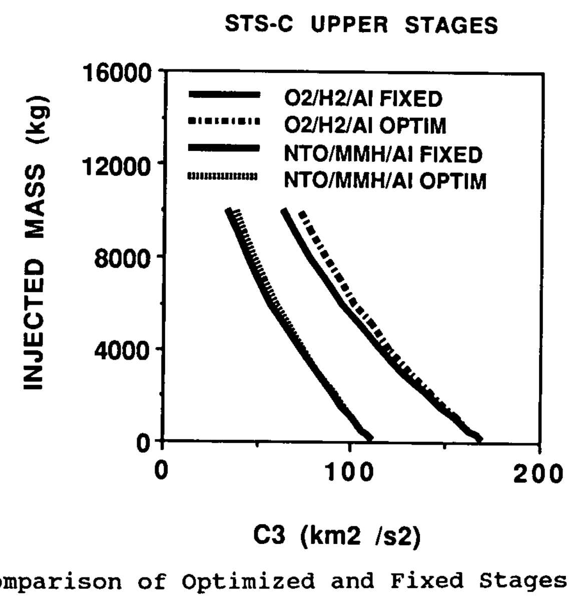

Figure 8. Comparison of Optimized and Fixed Stages With STS-C

Table X
Metallized NTO/MMH/Al and $O_2/H_2/Al$ Upper Stage Mass Summaries:
Fixed Mass One-Stage Design Points
STS-C Launch Mass = 68,040 kg
Total Stage Wet Mass and Injected Mass = 63931 kg

Element	Mass (kg) NTO/MMH/Al	$O_2/H_2/Al$
C_3 Design Point (km^2/s^2)	80	160
Injected Mass	3,192	721
Propellant Tankage	227	1,255
Pressurization	148	246
Engines and Feed System	400	400
Thermal Control	1,595	1,627
Structure	3,720	3,797
Residuals and Holdup	809	826
Contingency (10%)	690	815
Total Burnout Mass	7,589	8,966
Usable Propellant	53,150	54,244
Total	63,931	63,931

CONCLUDING REMARKS

Many technologies are available for increasing the payload capabilities of the STS and STS-C. Earth- and space-storable, cryogenic and metallized propulsion all have the capability to deliver significantly larger payloads than the IUS to GEO. However, the performance benefits of metallized propellants over their non-metallized counterparts are modest. The only exception to this is the NTO/MMH/Al system. Earth-storable NTO/MMH/Al enables a 25-lb_f-s/lb_m I_{sp} increase over NTO/MMH. This increase allows a 17- to 19-percent payload improvement over the non-metallized storable NTO/MMH systems and a 75 percent increase over the STS/IUS. For the GEO mission, NTO/MMH/Al can deliver comparable performance to all of the space-storable propulsion options. Metallized NTO/MMH/Al is therefore recommended as an option for the LEO-GEO transfer mission.

Using the STS-C, a cryogenic stage provides the greatest benefit, but a space-storable stage can deliver many of the performance needs for LEO-GEO missions and very significant improvements over the IUS. The payload delivery benefits of the space-storable systems with the STS-C are superior to the performance of an STS/O_2/H_2 upper stage combination. These significant performance capabilities should not be overlooked.

Metallized $O_2/H_2/Al$ propellants enable a significant performance improvement over non-metallized combinations in several planetary applications. With $O_2/H_2/Al$ propulsion, all of the very-high-energy planetary missions that were once rejected due to launch vehicle constraints are now enabled. The highest gains for the metallized propulsion systems are for planetary injection missions where the upper stage must deliver a C_3 greater than 124 km^2/s^2. At a C_3 below this point, however, the payload advantages are modest. For the Galileo-class mission (80 km^2/s^2), the benefits of metallized $O_2/H_2/Al$ are only 4.3 percent.

Earth-storable, space-storable and metallized propellants also provide attractive options for planetary missions. At a C_3 above 3 km^2/s^2, metallized propellants are able to deliver greater than a 10-percent injected mass increase over the NTO/MMH system. A single-stage NTO/MMH/Al propulsion system can enable a fast Galileo-class mission on the STS-C. At this C_3, the NTO/MMH/Al system can deliver a 97-percent injected mass increase over the NTO/MMH system.

Two-stage systems using non-metallized O_2/H_2 propellants can enable all of the planetary missions. Using a two-stage system tailored to these missions, however, may not be an option for the STS program. Past liquid propulsion upper stages have been almost exclusively single staged (with augmentation from a relatively small solid rocket motor). The capability of the two-stage system should be considered as an important alternative should the need arise for this increased performance level.

The technologies for developing NTO/MMH/Al and O_2/H_2/Al should both be included in future mission studies. These technologies can produce benefits not only for launch vehicle upper stages but also for lunar and Mars missions and launch vehicles themselves. The increased safety benefits offered by metallized propellants (making the propellant less likely to spill and less sensitive to "damaged" propellant tanks) also should not be overlooked.

There are significant potential benefits in using metallized propellants. Metallized propulsion systems performance efficiencies used in these analyses, however, were based on their non-metallized counterparts. The full benefits of metallized propellants will be realized only if these high efficiencies are achieved.

The STS and STS-C both require a high-energy upper stage for effective use by the planetary program and for access to GEO. Development of a high-energy upper stage to gain the maximal advantage from this new vehicle for planetary missions should include the investigation of metallized propellants. Applying these propulsion technologies to future upper stages will make them, the STS and STS-C safer, more productive and more cost-effective.

NOMENCLATURE

Al	Aluminum
ALS	Advanced Launch System
ASE	Airborne Support Equipment
ASPS	Adaptable Space Propulsion System
C_3	Injection Energy (km^2/s^2)
GEO	Geosynchronous Earth Orbit
IUS	Inertial Upper Stage
I_{sp}	Specific Impulse (lb_f-s/lb_m)
LEO	Low Earth Orbit
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
NTO/MMH	Nitrogen Tetroxide/Monomethyl Hydrazine
O_2/CH_4	Oxygen/Methane
O_2/H_2	Oxygen/Hydrogen
O_2/MMH	Oxygen/Monomethyl Hydrazine
STS	Space Transportation System
STS-C	Space Transportation System-Cargo
USRS	Upper Stage Responsiveness Study

Greek Symbols

ΔV Velocity Change (km/s)

ϵ Expansion Ratio

μ Gravitational Constant

η I_{sp} Efficiency

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