NASA
Technical
Paper
3191

February 1992

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National Aeronautics and
Space Administration

Office of Management

Scientific and Technical
Information Program

Summary

Metallized propellants are liquid propellants with a metal additive suspended in a gelled fuel. Typically, aluminum particles are the metal additive. These propellants increase the density and/or the specific impulse of the propulsion system. Using metallized propellants for volume- and mass-constrained upper stages can deliver modest increases in performance for low Earth orbit to geosynchronous Earth orbit (LEO–GEO)[1] and other Earth-orbital transfer missions. However, using metallized propellants for planetary missions can deliver great reductions in flight time with a single-stage, upper-stage system.

Tradeoff studies comparing metallized propellant stage performance with nonmetallized upper stages and the Inertial Upper Stage (IUS) are presented. These upper stages, launched from the STS and STS–C, are both one- and two-stage vehicles that provide the added energy to send payloads to high altitude orbits and onto interplanetary trajectories that are unattainable with only the Space Transportation System (STS) and the Space Transportation System—Cargo (STS–C). The stage designs are controlled by the volume and the mass constraints of the STS and 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 Inertial Upper Stage (IUS)[1] 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 will perform more propulsion-related maneuvers. These maneuvers will include multiple orbit changes about the outer planets (as with the Galileo mission to Jupiter and the Cassini mission to Saturn). When spacecraft 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 impulse (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.

The largest available stage for the STS is the Inertial Upper Stage (IUS, ref. 1). It can deliver a 2268-kg (5000-lb_m) payload to geosynchronous Earth orbit (GEO). However, for planetary missions, the IUS is limited to low-energy missions. The Galileo mission to Jupiter (ref. 2) was launched on an IUS. Using the Space Transportation System (STS)/IUS, its flight time will be 6.5 yr. With a high-performance cryogenic upper stage, the flight time could have been reduced to 1.5 yr. There is not, however, any cryogenic upper stage available that is compatible with the STS (refs. 3 and 4).

A new upper-stage system will be needed to fully exploit the capabilities of the STS and the planned Space Transportation System—Cargo (STS–C). An alternative to the STS is the Titan IV. Titan IV/Centaur G–Prime is a candidate for NASA missions, but its availability to NASA may be limited. This limited availability is caused by the number of Air Force payloads that have been deferred because of the STS launch delays and the resulting high priority placed on Air Force missions. Over the last several years, the Air Force and NASA have studied many potential configurations for future upper stages. These studies have included stages using cryogenic, Earth-storable, and space-storable propellants.

Currently planned Department of Defense missions will require large payloads to be delivered to GEO. To accomodate these payloads, the Adaptable Space Propulsion System (ASPS) study addressed improvements to the capabilities of the current upper stages for the Air Force (ref. 5). With 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 4536 kg (10 000 lb_m) to GEO. Planetary missions were also considered. As a successor to this study, the Upper-Stage Responsiveness Study was conducted (ref. 6). The U.S. Air Force Systems Command investigated a cryogenic propulsion upper stage for the Titan IV (refs. 6 to 8). This stage was designed to send a minimum of 6123 kg and a maximum of 6804 kg (13 500 to 15 000 lb_m) to GEO.

Over the last decade, intensive studies of large space-based and ground-based Orbital Transfer Vehicles (OTV, refs. 9 to 11) and Space Transfer Vehicles have been conducted at NASA (refs. 12 and 13). In the wake of the original STS/

[1]Acronyms and symbols are defined in the appendix.

Centaur program, Centaur-derived vehicles have also been analyzed (refs. 14 to 16). Earth-orbital, interplanetary, and lunar transfer missions have been studied to exploit the work conducted in the original Centaur program and to tailor its high-performance potential for future NASA missions, including those of the Space Exploration Initiative.

None of these stage 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. To see the benefits of metallized propellants for upper stages, we must consider the missions and propulsion system designs together and analyze them. This report discusses these aspects and the results of the overall systems analyses.

Why Metallized Propellants?

One advanced propulsion system that can provide benefits for upper stages is a metallized propellant system. The primary benefits of these propellants are increased performance and significantly enhanced safety. These propellants offer increases in the overall propellant density and/or the I_{sp} of a propulsion system, and these increases can enable significant launch mass reductions or payload increases over conventional chemical propellants. Metallized propellants have metal added to the fuel—typically, in the form of micron-sized particles. These particles are suspended in a gelled fuel to increase its combustion energy and its density. The I_{sp} of an engine is proportional to

$$I_{sp} \propto \sqrt{\frac{T}{MW}}$$

where

T combustion temperature

MW molecular weight of combustion products

Because of the increased combustion temperature, or the reduced 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 great effect on the overall dry mass.

Higher specific impulse I_{sp} systems and/or higher density propellants will be needed to increase the payload capability of existing launch vehicles and their upper stages. Previous studies of Mars and lunar missions (refs. 17 to 19) determined that metallized propellants are an attractive alternative to O_2/H_2 for future space transportation systems. 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. 18) and 3 percent added payload for the lunar missions (ref. 17).

Also, higher density metallized Earth- and space-storable propellants were able to enhance propellant storability for a Mars ascent vehicle with a minimal increase in the low Earth orbit (LEO) mass in comparison with O_2/H_2 propellants (ref. 18). A nitrogen tetroxide/monomethyl hydrazine/aluminum (NTO/MMH/Al) system can deliver a 25-$\mathrm{lb_f}$-s/$\mathrm{lb_m}$ I_{sp} increase over NTO/MMH. This performance advantage can make storable propellants an important option over O_2/H_2 propellants for extended stays of several hundred days on Mars.

Many of these benefits for lunar and Mars missions are also directly applicable to upper stages because of the strict constraints on the mass and volume imposed on the upper stages in the STS and STS–C. In addition, the need for upper stages to be able to loiter in Earth orbit because of launch window timing may prove that metallized storable propellants are a useful option.

Safety is another important advantage of metallized propellants (ref. 20). Because the aluminum is gelled with the fuel, widespread spillage of the propellant would be prevented if it were released. The spill would also be easier to clean up because it would be restricted to a more confined area. Also, gelling makes the propellants less sensitive to high-energy particles that penetrate the propellant tank. If a projectile (such as space debris or a wrench dropped during ground assembly) penetrated the propellant tank, the gel would prevent a catastrophic explosion.

Propulsion Systems Analyses

In the process of determining the potential performance advantages of metallized propellants, a series of propulsion systems analyses, or tradeoff 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 the launch vehicle constraints were determined and the missions and generic designs of the stages were formulated, these constraints, missions, and designs could 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:

$$\Delta V = I_{sp} g_o \ln\left(\frac{m_o}{m_f}\right)$$

where

ΔV velocity change

I_{sp} specific impulse

g_o gravitational constant

m_o initial mass

m_f final mass

Using the launch vehicle constraints, the stage designs, the mission requirements, and the rocket equation, we can calculate the payload or the injected mass. In the following sections, these constraints on the upper-stage designs are discussed.

Launch Vehicle Constraints

The upper-stage capability is presented in the results section for both STS and STS–C payloads launched from the Eastern Test Range (28.5° inclination). The launch vehicles have significantly different payload capabilities to LEO: 24 950 kg (55 000 lb_m) for the STS (ref. 5) and 68 040 kg (150 000 lb_m) for the STS–C (ref. 21). The payload bay lengths are also different: 18.3 m (60 ft) for the STS and 25 m (82 ft) for the STS–C. Both systems have a payload bay diameter of 4.57 m (15 ft).

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 (9058 lb_m, ref. 22). This mass was subtracted from the payload capability of the launch vehicle when the performance of the upper stages was estimated. 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.—The engine performance of the metallized propellant combinations was estimated with a computer simulation code (ref. 23). The expansion ratio ϵ for each of the engines, 500:1, was selected on the basis of the planned engine designs. The engine chamber pressure, 1000 psia, was selected on the basis of various engine designs under consideration for the upper stage. 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. An engine I_{sp} efficiency was used to modify the code-predicted I_{sp}. The I_{sp} efficiency η is the ratio of the estimated engine performance and the code-predicted I_{sp}. This reduction reflects the losses due to combustion, the engine flowfield, engine cycle inefficiencies, and other propulsion system losses. The engine efficiencies were derived from the performance estimates of references 24 to 27 and from comparisons with the vacuum I_{sp} predicted by the engine code. In this analysis, metallized propellants have the same engine efficiency as the nonmetallized 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 modeling, propellant rheology experiments, and hot-fire engine testing have been conducted to determine the potential engine efficiency of metallized propellants (refs. 28 to 31). Without the predicted increases in I_{sp}, the advantages of these propellants are significantly reduced. The effect of lower than predicted η will be discussed later in the paper. For the same efficiency for the metallized and nonmetallized engines, the increases in I_{sp} are several lb_f-s/lb_m. With 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. With NTO/MMH/Al, the I_{sp} increase over the nonmetallized case is 25 lb_f-s/lb_m.

TABLE I.—METALLIZED PROPELLANT ENGINE PERFORMANCE

[Expansion ratio, 500:1; chamber pressure, 1000 psia.]

Propellant	Specific impulse, I_{sp} lb_f-s/lb_m		I_{sp}, efficiency, η
	No metal	Metallized (aluminum)	
NTO/MMH	341.2	366.4	0.938
O_2/MMH	381.9	386.2	.940
O_2/CH_4	382.1	384.3	.940
O_2/H_2	479.5	485.4	.984

The mixture ratios and the metal loadings 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, one must analyze the propellant mixture ratio and metal loading and their effects on the engine I_{sp} and the propulsion system dry mass. The "best" system design is chosen to maximize the delivered payload or minimize the initial mass in LEO. 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.

Propellant density.—When the aluminum loadings considered in the engine performance calculations are used, the propellant density for the H_2 fuel can increase from 70 to 169 kg/m^3 (H_2 with a 60-percent aluminum loading). For MMH/Al, the density is 1324 kg/m^3 (50-percent Al loading) in contrast to

TABLE II.—METALLIZED PROPELLANT ENGINE DESIGN PARAMETERS

Propellant	Mixture ratio	
	No metal	Metallized (aluminum loading)
NTO/MMH	2.0	0.9 (50)
O_2/MMH	1.7	.9 (35)
O_2/CH_4	3.7	1.8 (45)
O_2/H_2	6.0	1.6 (60)

870 kg/m^3 for MMH. The density is computed with

$$\rho_{p,m} = \frac{1}{\dfrac{1 - ML}{\rho_p} + \dfrac{ML}{\rho_m}}$$

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

For the maximal reduction in LEO mass or the maximal payload increase, tradeoff 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 results of the systems analyses. The benefits of a metal loading in the fuel above 60 percent were small. This result is discussed later in the paper. The total metal loading of all 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 Al in the H_2/Al fuel, an ϵ of 500:1, and a mixture ratio of 1.60.

Because of the drop in mixture ratio required for the metallized fuels, 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 $O_2/H_2/Al$, therefore, may require a heavier propulsion system than for the nonmetallized case. Reference 18 compares the propulsion masses for several metal loadings. There is a small variation in the total mass of the propulsion system with the different metal loadings. On the basis of the tradeoff studies, the highest I_{sp} system of the range in figure 1 (which has a metal loading of 60 percent) was selected. For all 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.

Even if the benefits of reduced LEO mass or increased payload are not desired or significant, the effects of increased propellant density can benefit 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 would be reduced over that for the NTO/MMH system. In the metallized system, the total propellant tank volume for one mission would be reduced to 41.4 m^3

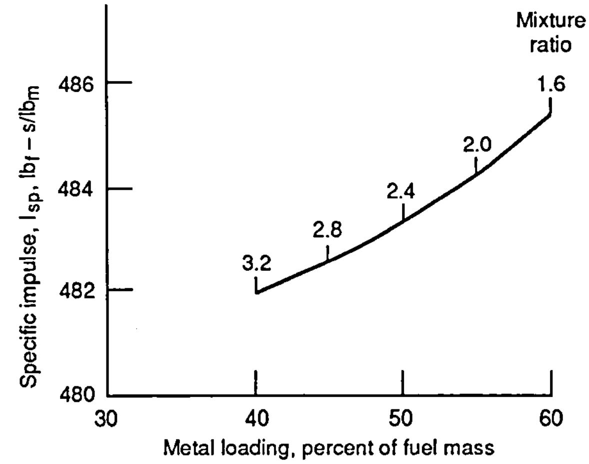

Figure 1.—Specific impulse of $O_2/H_2/Al$ versus metal loading. Expansion ratio, 500:1.

in contrast to the 49.4 m^3 required for the nonmetallized NTO/MMH system.

Although the tankage volume would decrease in the NTO/MMH/Al system, other applications of metallized propellants, such as $O_2/H_2/Al$, would 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 upper stage, the total O_2 tank volume could be reduced from 43.7 to 31.4 m^3 for the $O_2/H_2/Al$ system. The H_2 tank volume, however, would increase from 119.6 to 133.7 m^3 with metallized propellants. Overall, the total tank volume would increase from 163.3 to 165.1 m^3 (a difference of 1.8 m^3, or 1.1 percent). This example is for the fixed stage launched from the STS-C which is described later in the report.

Pump-fed and pressure-fed systems.—To deliver the very high performance being considered in these upper-stage analyses, a pump-fed engine is used. Pressure-fed propulsion systems, while a potential alternative, typically require larger masses for propellant tankage and pressurization systems. For metallized propellants, the propellant feed system must be designed to supply the non-Newtonian, thixotropic metallized propellant with the same reliability as the nonmetallized H_2. Currently, metallized propellants are fed to small propulsion systems with positive-displacement propellant expulsion devices (diaphragms and other devices, ref. 32). 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 may be essential. Past work (ref. 33) has shown that high propellant expulsion efficiencies can be achieved without resorting to positive expulsion devices. Proper design of the tank outlet has allowed normal, predictable outflow with gelled propellants. Turbopumps for metallized engines have also been demonstrated in large-scale tests (ref. 34). These initial tests provided preliminary data on the effectiveness of pump-fed propulsion. Although the pump was somewhat eroded, new methods of pump design may alleviate this.

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

$$m_{dry} = A + B_p$$

where

A,B mass parameters

m_p propellant mass

Table III lists the propulsion mass-scaling parameters for all the systems considered. These parameters include all 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 A parameter of the scaling equations varies because of the different configurations of spherical and cylindrical tankage. Only the O_2/H_2 and $O_2/H_2/Al$ stages required cylindrical tanks. This is due to the relatively low density of the H_2 and H_2/Al fuels. The B parameter depends on the propellant mixture ratio, the propellant metal loading, and hence the propellant density. The specific mixture ratios and the metal loadings are listed in table II.

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

The propellant tankage for all the systems is designed for a 50-psia maximal operating pressure. The propellant is stored at 30 psia. The tanks for O_2, H_2, and CH_4 are made of aluminum alloy (2219-T87), and the tanks for NTO and MMH are made of titanium (Ti-6Al-4V). The flange factor and safety

TABLE III.—PROPULSION SYSTEM MASS-SCALING PARAMETERS

Propellants	Scaling parameter		Application
	A	B	
NTO/MMH	440.00	[a]0.1358	STS, STS-C
NTO/MMH/Al	↓	[a].1345	STS, STS-C
O_2/MMH		[a].1396	STS, STS-C
O_2/MMH/Al		[a].1376	STS, STS-C
O_2/CH_4		[a].1458	STS, STS-C
$O_2/CH_4/Al$	↓	[a].1440	STS, STS-C
O_2/H_2	355.12	[b].1598	STS-C
O_2/H_2	373.80	[c].1576	STS
$O_2/H_2/Al$	373.80	[c].1584	STS, STS-C

[a]Spherical tanks.
[b]Cylindrical O_2 and H_2 tanks.
[c]Spherical O_2 tank, cylindrical H_2 tank.

factor are 1.4 and 2.0, respectively. The safety factor is based on the tank material ultimate stress. The propellant-residual-and-holdup mass is 1.5 percent of the total propellant mass. This percentage accommodates a small added mass for cryogenic propellant boiloff. Because the stages are expendable, no large allowance was made for propellant losses due to boiloff.

The cryogenic propulsion systems use autogenous pressurization. The NTO/MMH and the space-storable systems use regulated helium pressurization. In the pressurant tank, the maximal operating pressure is 3722 psia, and 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. 3). 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. Each mission is described by a mission velocity change ΔV or an injection energy (C_3).

Velocity changes for LEO-GEO orbit transfer.—Many and varied payload deliveries to GEO are planned (refs. 1 and 35). The payloads to be placed there are communications satellites, observation systems, and other remote-sensing satellites, such as those for the Mission to Planet Earth.

The orbit transfer equations are (ref. 36)

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

$$\Delta V_{te} = V_o \sqrt{\frac{1 + 3R}{1 + R} - C1}$$

$$C1 = 2\sqrt{\frac{2R}{1 + R}} \cos\left(\theta_{\mathrm{tot}} - \theta_{\mathrm{circ}}\right)$$

$$R = \frac{r_f}{r_o}$$

$$V_o = \sqrt{\frac{\mu}{r_o}}$$

and

$$\Delta V_{\mathrm{circ}} = V_o \sqrt{\frac{3 + R}{R(1 + R)} - C2}$$

$$C2 = \frac{2}{R}\sqrt{\frac{2}{1 + R}} \cos\left(\theta_{\mathrm{circ}}\right)$$

where

V orbital velocity, km/s

r orbital radius, km (or 6378.14 km + orbital altitude, km)

θ orbital plane change, rad

μ Earth's gravitational parameter, 398601.3 km^3/s^2

Subscripts:

circ circularization

f final

o initial

te transfer ellipse

tot total

By using these Hohmann orbit-transfer equations, we can compute the ΔV for a minimum energy transfer. The initial altitude for the mission is 241 km. The total one-way ΔV for the LEO–GEO mission is 4.25 km/s, and the total plane change is 28.5°. 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 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 2.46 km/s and includes 2.2° 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 1.79 km/s. The remaining 26.3° of the plane change is performed during the GEO burn.

Velocity changes for other Earth-orbital transfers.—Other Earth-orbital missions are under consideration for the Strategic Defense Initiative missions: 10 000- to 17 935-km altitudes with a 65° inclination (ref. 35). The ΔV's for several different orbital transfers are listed in table IV. These ΔV's were computed with the equations discussed previously. The 36.5° inclination change represents a transfer from a 28.5° to a 65° inclination orbit.

Injection energy for planetary missions.—The performance of an interplanetary 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

$$C_3 = \left(\sqrt{\frac{\mu}{r_o}} + \Delta V\right)^2 - 2\frac{\mu}{r_o} = V_\infty^2$$

where

C_3 injection energy, km^2/s^2

ΔV velocity change, km/s

r_o orbital radius, km

V_∞ hyperbolic excess velocity

For planetary missions, the injected mass 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. 2, 37, and 38). The injected masses and the injection energies for the missions are provided. All of the fast missions (except Galileo) use a Jupiter swingby maneuver to shorten the flight time. Currently, the Saturn Orbiter/Titan Probe mission is named Cassini. For a fast mission to Saturn, a C_3 of 109 km^2/s^2 is needed. Currently, its mission is planned with 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 flyby also adds from 1.5 to 3 yr to the flight time of the mission. As is planned for Cassini, the Galileo mission was launched on an IUS at a low C_3 (17 km^2/s^2). This lower C_3 requires a Venus-Earth Earth-gravity-assist (VEEGA) trajectory:

TABLE IV.—ORBIT TRANSFER VELOCITY CHANGES: EARTH-ORBITAL MISSIONS

Mission	Velocity change, ΔV, km/s	Inclination change, deg
LEO to GEO[a]	4.253	28.5
LEO to 10 000 km	4.293	36.5
LEO to 17 935 km	4.367	36.5

[a]Low Earth orbit (LEO) is defined as a 241-km altitude orbit (28.5° inclination), and geosynchronous Earth orbit (GEO) is a 35 870-km orbit (0° inclination).

TABLE V.—POTENTIAL PLANETARY MISSION REQUIREMENTS

[From references 3, 25, and 26.]

Mission	Injected mass, kg	Injection energy C_3, km^2/s^2
Galileo (direct)	2550	80.0
Saturn Orbiter/ Titan Probe (direct)	2488	109.0
Titan Flyby/ Titan Probe	1575	136.9
Uranus Flyby/ Uranus Probe	1298	150.0
Pluto Flyby	700	160.0

one flyby of Venus and two Earth flybys. Using this flight path adds 5 yr to the flight time. A direct trajectory with a C_3 of 80 km^2/s^2 would only require 1.5 yr to reach Jupiter. Advanced upper stages can produce a higher C_3, shorten the mission flight time, and return the science to Earth more quickly.

Results

In this section, the results of the systems analyses for several different upper-stage missions are discussed. Both one- and two-stage systems are considered. The stage performance for optimized vehicles and fixed-stage designs are addressed. Optimized vehicles are those whose stage masses 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 they were considered. Because they have a fixed mass, they will operate "nonoptimally" and not deliver the maximum payload for all conditions other than the design point. The optimized stage results are presented first. After these results are discussed, the fixed-stage performance is presented.

Optimized Stages

LEO-GEO orbit transfer.—A performance comparison of the nonmetallized and the metallized stages for the LEO-GEO mission is shown in figure 2 (for the STS) and figure 3 (for the STS-C). The $O_2/H_2/Al$ system can deliver the highest payload to GEO: 6226 kg for the STS and 19 211 kg for the STS-C. With 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 a 1.7-percent increase with the STS-C. With the space-storable propellants, the payload delivered ranged from 4300 kg on the STS to 14 000 kg with the STS-C. The percentage savings with the metallized space-storables was similar to that for metallized $O_2/H_2/Al$: 1.8 to 1.6 percent over their nonmetallized counterparts (for STS and STS-C, respectively) for the $O_2/CH_4/Al$ propellant, and 3.0 to 2.6 percent for the O_2/MMH/Al propellant.

The largest percentage gain of any of the metallized combinations is with NTO/MMH/Al. An NTO/MMH/Al stage delivers a 17- to 19-percent benefit over NTO/MMH. As a replacement for the IUS, a storable NTO/MMH/Al upper stage can significantly increase the 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 to the 2268-kg IUS GEO capability with either the STS or STS-C.

Other Earth-orbital transfers.—In table VI, the payload capabilities of all of the propulsion technologies for two Earth-

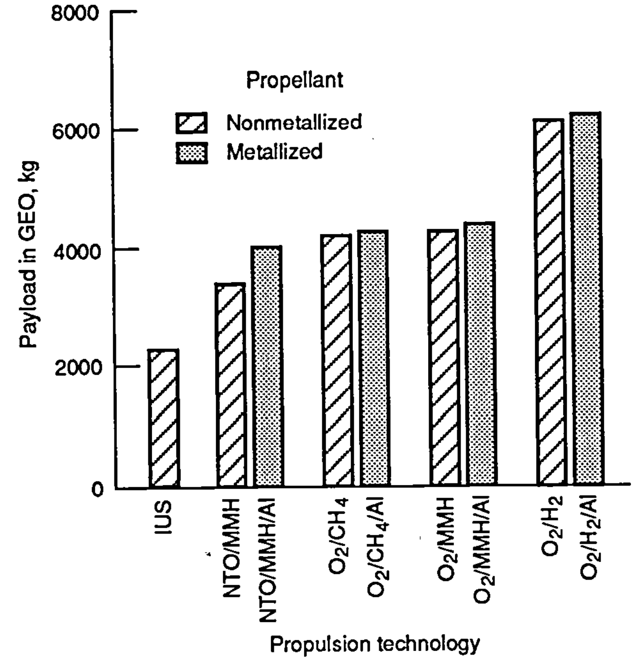

Figure 2.—Payload capability for a one-way STS upper-stage mission from low Earth orbit to geosynchronous orbit (LEO-GEO).

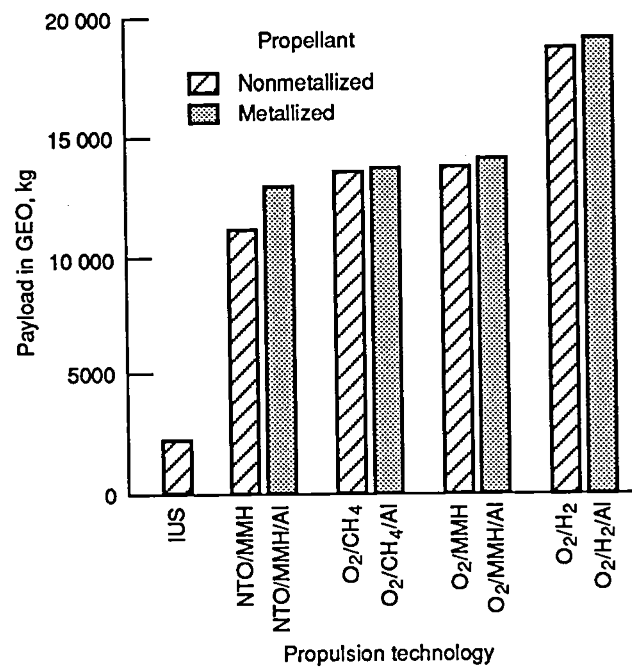

Figure 3.—Payload capability for a one-way STS-C upper-stage mission from low Earth orbit to geosynchronous orbit (LEO-GEO).

TABLE VI.—PAYLOAD CAPABILITIES: HIGHLY INCLINED EARTH-ORBITAL TRANSFER MISSIONS

Propulsion technology	Mission altitude,[a] km	
	10 000	17 935
	Payload mass, kg	
STS mission:		
NTO/MMH	3 267.8	3 123.6
NTO/MMH/Al	3 892.6	3 743.8
O_2/CH_4	4 095.6	3 943.9
$O_2/CH_4/Al$	4 171.9	4 020.0
O_2/MMH	4 179.6	4 028.6
O_2/MMH/Al	4 306.5	4 154.8
O_2/H_2	6 042.8	5 886.7
$O_2/H_2/Al$	6 142.9	5 987.1
STS-C missions:		
NTO/MMH	10 997.3	10 559.9
NTO/MMH/Al	12 857.0	12 420.7
O_2/CH_4	13 452.4	13 010.0
$O_2/CH_4/Al$	13 672.1	13 231.5
O_2/MMH	13 692.0	13 254.3
O_2/MMH/Al	14 054.2	13 619.0
O_2/H_2	18 674.1	18 255.7
$O_2/H_2/Al$	18 990.3	18 575.9

[a]65° inclination.

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. For the 10 000-km mission (with a 65° inclination) and the mission to 17 935 km (65° 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 nonmetallized counterpart: 20 percent (with the STS) and 17 percent (with the STS-C). On the 10 000-km mission, the total payload delivered with NTO/MMH/Al was 3890 kg (with the STS) and 12 860 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.

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 nonmetallized space-storable and cryogenic systems is therefore highly beneficial.

Planetary missions.—With the same upper-stage mass-scaling equations used for the LEO-GEO analysis, the performance for planetary injections was determined. The performance of these stages was first computed to determine the maximum deliverable injected mass. These optimized stage designs were also used to define a fixed stage and to determine the performance differences between it and the optimized system.

Figures 4 to 7 depict the injected mass as a function of C_3 for the STS and STS-C. These plots show the overall performance benefits for metallized propellant for $O_2/H_2/Al$, O_2/MMH/Al, and NTO/MMH/Al. With the STS, the increases are smaller than those enabled with the STS-C: 80 to 100 kg with $O_2/H_2/Al$, 115 to 140 kg with O_2/MMH/Al, and 540 to 650 kg with NTO/MMH/Al. For the STS-C, the injected mass increases enabled with $O_2/H_2/Al$ are in the range of 300 to 360 kg. With O_2/MMH/Al, the STS-C

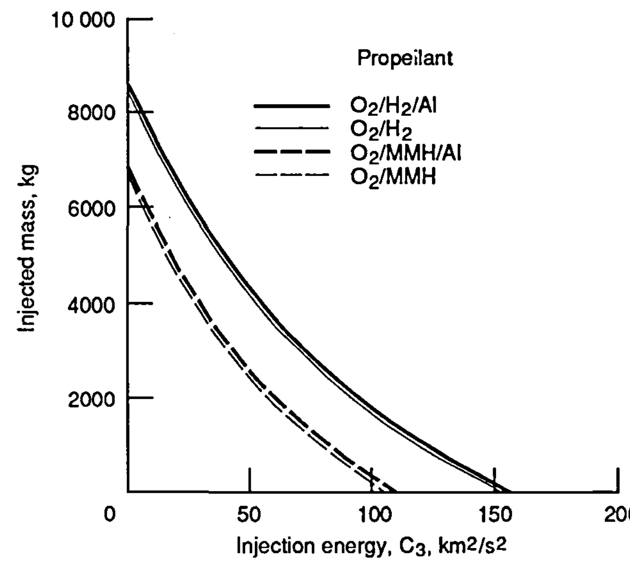

Figure 4.—STS performance with $O_2/H_2/Al$ and O_2/MMH/Al upper stages.

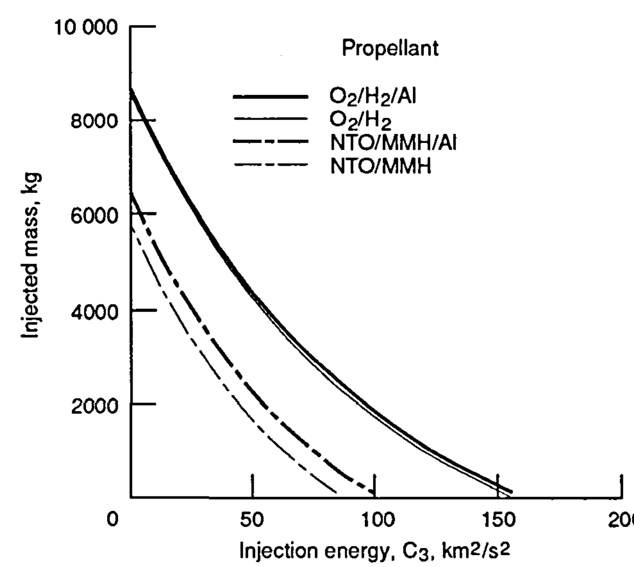

Figure 5.—STS performance with $O_2/H_2/Al$ and NTO/MMH/Al upper stages.

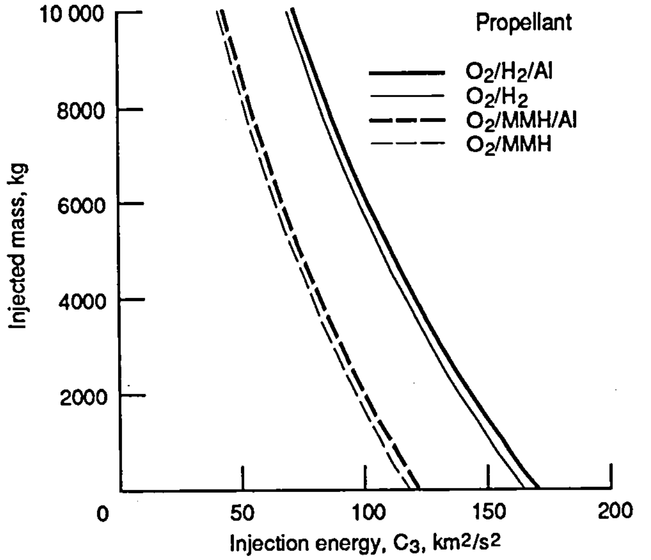

Figure 6.—STS-C performance with $O_2/H_2/Al$ and $O_2/MMH/Al$ upper stages.

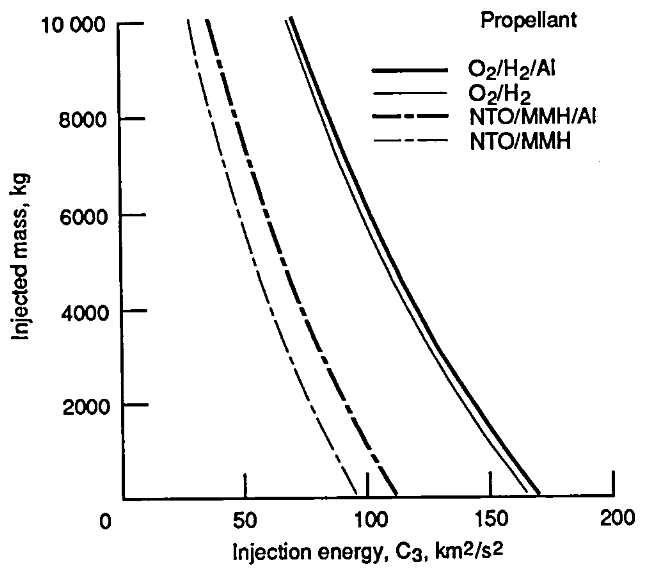

Figure 7.—STS-C performance with $O_2/H_2/Al$ and NTO/MMH/Al upper stages.

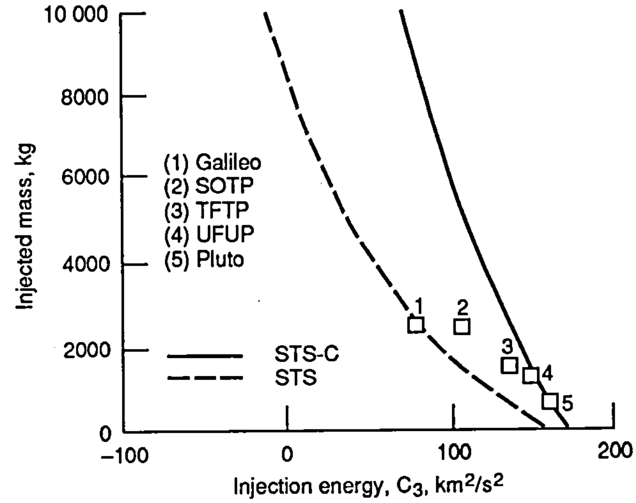

Figure 8.— STS and STS-C performance with $O_2/H_2/Al$ upper stages for different missions: Galileo, Saturn Orbiter/Titan Probe (SOTP), Titan Flyby/Titan Probe (TFTP), Uranus Flyby/Uranus Probe (UFUP), Pluto Flyby.

increases are 300 to 380 kg; and with NTO/MMH/Al, the STS-C injected mass increases were 1510 to 1860 kg.

The STS-C with a large $O_2/H_2/Al$ upper stage can be an effective tool for conducting fast planetary missions. Figure 8 compares the STS and STS-C with $O_2/H_2/Al$ upper stages. The STS with its stage can "capture" one of the missions listed (Galileo), whereas the STS-C and upper stage is the only option that can capture all of the missions. "Capturing" a mission means that the upper stage can deliver the requisite C_3 for the spacecraft injected mass.

Table VII lists the values of C_3 at which metallized propellants increase the injected mass a minimum of 10 percent (over their nonmetallized counterparts) for planetary missions.

TABLE VII.—MINIMUM PAYOFF INJECTION ENERGY C_3 FOR HIGH-ENERGY PLANETARY MISSIONS[a]

Propulsion technology	Injection energy 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

[a]The payoff C_3 is defined as the C_3 at which a 10-percent increase in injected mass is enabled over nonmetallized propellants.

This will be called the payoff C_3. With metallized propellants, larger benefits are gained on very high energy missions. For the STS-C stages with a C_3 of 123.8 km^2/s^2, a comparison of the O_2/H_2 and the $O_2/H_2/Al$ propulsion systems shows that metallized propellants can deliver a 10-percent additional payload. A 28-percent injected mass increase is delivered at a C_3 of 150 km^2/s^2. With the very high C_3 of 160 km^2/s^2, a 79-percent increase is possible. The negative C_3 listed for the NTO/MMH/Al system is representative of an upper-stage mission that delivers a total ΔV that is less than the Earth's escape velocity.

In general, the payoff C_3 in table VII 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 is with $O_2/H_2/Al$ propulsion. This is because of the higher mass penalty paid by the stages with cylindrical tankage. Table III includes a breakdown of the types of tankage used in the different O_2/H_2 and $O_2/H_2/Al$ stages. Because of the STS-C cargo bay volume, its O_2/H_2 stage must use cylindrical tankage for both propellants. This places a mass penalty on this stage over the $O_2/H_2/Al$ stage of the STS-C. The metallized stage only uses cylindrical tankage for the H_2/Al fuel. Because of the added mass penalty for the O_2/H_2 stage of the STS-C, the trend of the STS having the lower payoff C_3 is reversed.

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 the fast planetary missions. As an example, in figure 9, 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. 38) with a C_3 of 160 km^2/s^2 is also enabled with the $O_2/H_2/Al$ system.

Systems other than $O_2/H_2/Al$ propulsion can capture some of the planetary missions. Figure 10 compares metallized and nonmetallized 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.

Influence of specific impulse efficiency on performance.—The influence of η on the performance of the metallized upper stages for various missions was investigated. Because of the two-phase flow of the metallized propellants in the combustion chamber and nozzle, there is a difference between the gas and solid-liquid particle velocities which creates a performance loss. The solid-liquid particles are composed of solid and liquid aluminum oxide (Al_2O_3). Once the potential losses of metallized propellants are introduced into the analysis, the performance may be much lower than that previously predicted. A series of cases showing this influence on the $O_2/H_2/Al$ and NTO/MMH/Al systems were analyzed, and a discussion of the results follows.

Effects of O_2/H_2 Al on specific impulse efficiency: Tables VIII and IX provide the parameters of injected mass and η for $O_2/H_2/Al$ stages in the STS and STS-C, respectively. In these tables, the metallized η varies from 0.934 to 0.984. This range reflects the performance penalties that have been predicted for metallized propellants: up to a 5-percent reduction in η (refs. 28 and 39). The minimum η required for $O_2/H_2/Al$ to equal the performance of the nonmetallized O_2/H_2 drops as the mission C_3 increases. As C_3 increases, the injected mass capability of the higher-I_{sp} metallized system increases, and therefore the metallized system can tolerate a greater η penalty.

With the $O_2/H_2/Al$ stage in the STS-C (table IX), the performance for planetary missions shows a 10-percent benefit

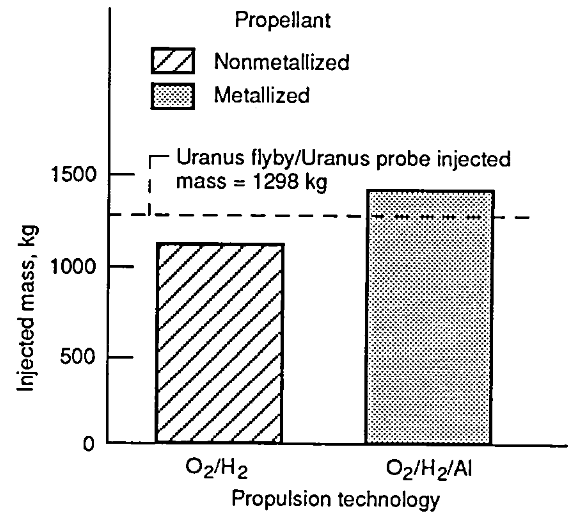

Figure 9.—STS-C mission performance for Uranus Flyby/Uranus Probe (UFUP) with $O_2/H_2/Al$ and O_2/H_2 propulsion. Injection energy, C_3, 150 km^2/s^2.

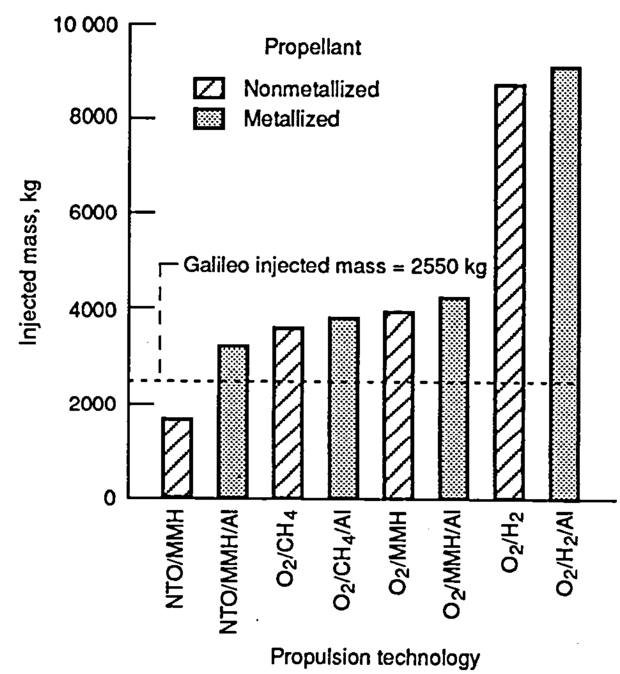

Figure 10.—STS-C mission performance for Galileo mission with various propellants. Injection energy, C_3, 80 km^2/s^2.

over O_2/H_2 once the mission C_3 is greater than 124 km^2/s^2. This is the payoff C_3 discussed in table VII, which assumes that the η for both propulsion systems is equal. In table IX, the payoff C_3 occurs at higher and higher values of C_3 as the η is reduced. For $\eta = 0.974$, the payoff C_3 is in the range of 150 to 160 km^2/s^2. Thus, as the η drops, only the missions with very high injection energies will derive a benefit from metallized propellants.

An example of the influence of reductions in η on the performance of metallized $O_2/H_2/Al$ systems was considered.

TABLE VIII.—SPECIFIC IMPULSE EFFICIENCIES AND INJECTED MASSES FOR STS UPPER STAGES WITH O_2/H_2 AND $O_2/H_2/Al$ PROPELLANTS

(a) Injected mass

Injection energy, C_3, km^2/s^2	Propellants					
	O_2/H_2	$O_2/H_2/Al$				
	Specific impulse efficiency, η					
	0.984	0.984	0.979	0.974	0.964	0.934
	Injected masses of propellants, kg					
0.00	8539.90	8630.90	8589.60	8549.20	8464.10	8203.60
20.00	6475.36	6574.78	6529.03	6484.29	6390.60	6105.97
30.00	5604.78	5705.22	5658.74	5613.30	5518.37	5231.05
40.00	4830.21	4930.04	4883.61	4838.20	4743.59	4458.27
50.00	4142.84	4240.76	4194.99	4150.18	4057.09	3777.26
60.00	3533.81	3628.91	3584.23	3540.43	3449.70	3177.68
70.00	2994.31	3086.02	3042.69	3000.15	2912.24	2649.18
80.00	2515.50	2603.62	2561.73	2520.52	2435.56	2181.40
90.00	2088.54	2173.24	2132.72	2092.74	2010.47	1763.99
100.00	1704.60	1786.40	1747.00	1708.00	1627.80	1386.60
110.00	1354.85	1434.64	1395.94	1357.49	1278.39	1038.88
120.00	1030.46	1109.47	1070.89	1032.39	953.06	710.47
130.00	722.60	802.44	763.21	723.90	642.64	391.03
140.00	422.43	505.07	464.27	423.22	337.97	70.20
150.00	121.11	208.89	165.41	121.53	29.86	-------

(b) Increase of injected mass of $O_2/H_2/Al$ over O_2/H_2

Injection energy, C_3, km^2/s^2	Specific impulse efficiency, η				
	0.984	0.979	0.974	0.964	0.934
	Increase in injected mass, percent				
0.00	1.07	0.58	0.11	−0.89	−3.94
20.00	1.54	.83	.14	−1.31	−5.70
30.00	1.79	.96	.15	−1.54	−6.67
40.00	2.07	1.11	.17	−1.79	−7.70
50.00	2.36	1.26	.18	−2.07	−8.82
60.00	2.69	1.43	.19	−2.38	−10.08
70.00	3.06	1.62	.19	−2.74	−11.53
80.00	3.50	1.84	.20	−3.18	−13.28
90.00	4.06	2.12	.20	−3.74	−15.54
100.00	4.80	2.49	.20	−4.51	−18.66
110.00	5.89	3.03	.19	−5.64	−23.32
120.00	7.67	3.92	.19	−7.51	−31.05
130.00	11.05	5.62	.18	−11.06	−45.89
140.00	19.58	9.91	.19	−19.99	−83.38
150.00	72.47	36.58	.34	−75.34	------

TABLE IX.—SPECIFIC IMPULSE EFFICIENCIES AND INJECTED MASSES FOR STS-C UPPER STAGES WITH O_2/H_2 AND $O_2/H_2/Al$ PROPELLANTS

(a) Injected mass

Injection energy, C_3, km^2/s^2	Propellants					
	O_2/H_2	$O_2/H_2/Al$				
	Specific impulse efficiency, η					
	0.984	0.984	0.979	0.974	0.964	0.934
	Injected masses of propellants, kg					
0.00	25 089.90	25 329.00	25 232.00	25 137.00	24 937.00	24 318.00
20.00	19 821.88	20 126.61	20 008.79	19 893.77	19 652.06	18 909.57
30.00	17 493.24	17 820.29	17 695.61	17 574.00	17 318.73	16 537.29
40.00	15 353.28	15 696.80	15 567.34	15 441.12	15 176.51	14 369.22
50.00	13 391.05	13 745.84	13 613.44	13 484.35	13 214.18	12 392.56
60.00	11 595.59	11 957.07	11 823.36	11 692.95	11 420.48	10 594.52
70.00	9 955.95	10 320.19	10 186.56	10 056.15	9 784.19	8 962.30
80.00	8 461.17	8 824.88	8 692.51	8 563.20	8 294.07	7 483.09
90.00	7 100.31	7 460.82	7 330.67	7 203.34	6 938.88	6 144.09
100.00	5 862.40	6 217.70	6 090.50	5 965.80	5 707.40	4 932.50
110.00	4 736.50	5 085.20	4 961.46	4 839.83	4 588.38	3 835.52
120.00	3 711.64	4 053.01	3 933.01	3 814.67	3 570.59	2 840.35
130.00	2 776.89	3 110.81	2 994.61	2 879.56	2 642.79	1 934.19
140.00	1 921.27	2 248.28	2 135.72	2 023.74	1 793.75	1 104.24
150.00	1 133.85	1 455.11	1 345.81	1 236.45	1 012.23	337.69
160.00	403.66	720.99	614.34	506.93	286.99	-------
170.00	-------	35.59	-------	-------	-------	-------

(b) Increase of injected mass of $O_2/H_2/Al$ over O_2/H_2

Injection energy, C_3, km^2/s^2	Specific impulse efficiency, η				
	0.984	0.979	0.974	0.964	0.934
	Increase in injected mass, percent				
0.00	0.96	0.57	0.19	−0.61	−3.07
20.00	1.54	.94	.36	−.86	−4.60
30.00	1.87	1.16	.46	−1.00	−5.46
40.00	2.24	1.39	.57	−1.15	−6.41
50.00	2.65	1.66	.70	−1.32	−7.46
60.00	3.12	1.96	.84	−1.51	−8.63
70.00	3.66	2.32	1.01	−1.73	−9.98
80.00	4.30	2.73	1.21	−1.97	−11.56
90.00	5.08	3.24	1.45	−2.27	−13.47
100.00	6.06	3.89	1.76	−2.64	−15.86
110.00	7.36	4.75	2.18	−3.13	−19.02
120.00	9.20	5.96	2.78	−3.80	−23.47
130.00	12.03	7.84	3.70	−4.83	−30.35
140.00	17.02	11.16	5.33	−6.64	−42.53
150.00	28.33	18.69	9.05	−10.73	−70.22
160.00	78.61	52.19	25.58	−28.90	------

Figure 11 shows the injected mass for the UFUP mission as a function of η. The O_2/H_2 system has a 98.4-percent efficiency. In this example, once the η equals 97 percent, the injected mass for the $O_2/H_2/Al$ and the O_2/H_2 systems is the same. The UFUP mission is not enabled until η is greater than or equal to 97.7 percent. This shows the critical importance of a high η.

An alternative approach to increase the $O_2/H_2/Al$ injected mass performance and recover some of the losses due to lower η was investigated. If the metal loading of $O_2/H_2/Al$ is increased from 60 percent to 70 percent, the I_{sp} can be increased from 485.4 to 489.8 lb_f-s/lb_m. With the increased I_{sp}, the propulsion system may be able to tolerate a lower η. Figure 12 illustrates the potential I_{sp} increases enabled with increased

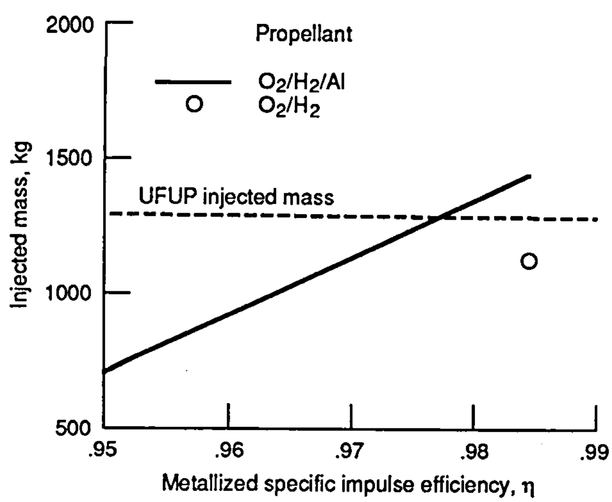

Figure 11.—Injected mass versus specific impulse efficiency η for Uranus Flyby/Uranus Probe (UFUP) mission. Injected energy, C_3, 150 km^2/s^2; specific impulse of O_2/H_2, I_{sp}, 0.984.

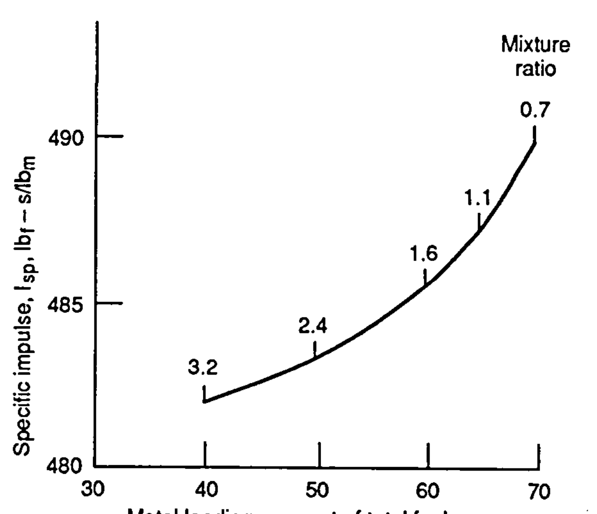

Figure 12.—Specific impulse of $O_2/H_2/Al$ at high metal loadings.

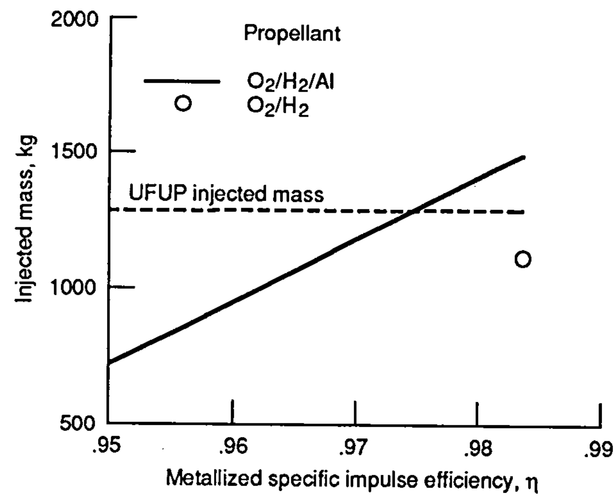

Figure 13.—Uranus Flyby/Uranus Probe (UFUP) mission performance with 70-percent Al loading. Injection energy, C_3, 150 km^2/s^2.

Al loading. If the metal loading is increased to 70 percent, however, the mixture ratio will drop to 0.7. The propellant tankage will change, and a new propulsion system configuration will be required. The new propulsion mass-scaling equation for the 70-percent Al loading case would be

$$m_{dry} = 373.80 + 0.1612\ m_p$$

Figure 13 shows the performance of an $O_2/H_2/Al$ system with 70-percent Al loading in the H_2. The injected mass capability of the new stage is 1500 kg (I_{sp} = 489.8 lb_f-s/lb_m), but only 1455 kg was delivered at a metal loading of 60 percent (I_{sp} = 485.4 lb_f-s/lb_m). This performance level requires an η of 0.984. At an η of 0.974, the metallized system can still perform the UFUP mission. This higher 70-percent metal loading allows the η to be slightly lower than that at a 60-percent loading (η = 0.977) and still perform the mission. Even with the high Al loading, the system can only tolerate a small added 0.3-percent reduction in η. This method of making the metallized system more tolerant of η penalties was, therefore, not considered attractive.

Effects of NTO/MMH/Al on specific impulse efficiency: The overall effect of reduced η is less detrimental for NTO/MMH/Al propellants. With the metallized NTO/MMH/Al, the theoretical I_{sp} increase over NTO/MMH is 25 lb_f-s/lb_m. This large increase is able to "absorb" a larger I_{sp} penalty than the other metallized propellant cases and still enable a large injected mass increase. Tables X and XI provide the parametrics of injected mass and η for NTO/MMH/Al stages in both the STS and STS-C. An η range of 0.888 to 0.938 was used to represent up to a 5-percent penalty on η. As with the results for $O_2/H_2/Al$ in tables VIII and IX, as the NTO/MMH/Al η decreases, the percent increase in injected mass decreases for any given mission C_3.

TABLE X.—SPECIFIC IMPULSE EFFICIENCIES AND INJECTED MASSES FOR STS UPPER STAGES WITH NTO/MMH AND NTO/MMH/Al PROPELLANTS

(a) Injected mass

Injection energy, C_3, km^2/s^2	Propellants					
	NTO/MMH	NTO/MMH/Al				
	Specific impulse efficiency, η					
	0.938	0.938	0.933	0.928	0.918	0.888
	Injected masses of propellants, kg					
0.00	5799.10	6446.40	6401.00	6352.40	6257.10	5963.70
20.00	3676.34	4312.25	4267.03	4218.79	4124.48	3837.68
30.00	2867.12	3478.20	3434.54	3388.03	3297.19	3022.40
40.00	2188.65	2771.01	2729.34	2684.98	2598.39	2337.41
50.00	1612.91	2169.30	2129.63	2087.40	2004.94	1756.56
60.00	1111.86	1651.70	1613.62	1573.06	1493.70	1253.73
70.00	657.56	1196.86	1159.55	1119.73	1041.53	802.75
80.00	221.92	783.32	745.61	705.17	625.28	377.50
90.00	-------	389.80	350.02	307.14	221.82	-------

(b) Increase of injected mass of NTO/MMH/Al over NTO/MMH

Injection energy, C_3, km^2/s^2	Specific impulse efficiency, η				
	0.938	0.933	0.928	0.918	0.888
	Increase in injected mass, percent				
0.00	11.16	10.38	9.54	7.90	2.84
20.00	17.30	16.07	14.76	12.19	4.39
30.00	21.31	19.79	18.17	15.00	5.42
40.00	26.61	24.70	22.68	18.72	6.80
50.00	34.50	32.04	29.42	24.31	8.91
60.00	48.55	45.12	41.48	34.34	12.76
70.00	82.01	76.34	70.29	58.39	22.08
80.00	252.98	235.99	217.76	181.76	70.11

TABLE XI.—SPECIFIC IMPULSE EFFICIENCIES AND INJECTED MASSES FOR STS-C UPPER STAGES WITH NTO/MMH AND NTO/MMH/Al PROPELLANTS

(a) Injected mass

Injection energy, C_3, km^2/s^2	Propellants					
	NTO/MMH	NTO/MMH/Al				
	Specific impulse efficiency, η					
	0.938	0.938	0.933	0.928	0.918	0.888
	Injected masses of propellants, kg					
0.00	18 210.00	19 900.00	19 782.00	19 657.00	19 410.00	18 641.00
20.00	12 219.43	14 069.34	13 939.04	13 801.40	13 530.26	12 694.69
30.00	9 773.82	11 633.67	11 502.01	11 363.12	11 089.98	10 252.41
40.00	7 647.24	9 482.43	9 351.86	9 214.31	8 944.25	8 120.14
50.00	5 804.15	7 589.11	7 461.54	7 327.30	7 064.20	6 264.95
60.00	4 208.97	5 927.23	5 803.98	5 674.45	5 420.96	4 653.91
70.00	2 826.15	4 470.26	4 352.14	4 228.09	3 985.66	3 254.08
80.00	1 620.12	3 191.72	3 078.95	2 960.56	2 729.41	2 032.54
90.00	555.32	2 065.10	1 957.36	1 844.22	1 623.35	956.36
100.00	--------	1 063.90	960.30	851.40	638.60	--------
110.00	--------	161.61	60.72	--------	--------	--------

(b) Increase of injected mass of NTO/MMH/Al over NTO/MMH

Injection energy, C_3, km^2/s^2	Specific impulse efficiency, η				
	0.938	0.933	0.928	0.918	0.888
	Increase in injected mass, percent				
0.00	9.28	8.63	7.95	6.59	2.37
20.00	15.14	14.07	12.95	10.73	3.89
30.00	19.03	17.68	16.26	13.47	4.90
40.00	24.00	22.29	20.49	16.96	6.18
50.00	30.75	28.56	26.24	21.71	7.94
60.00	40.82	37.90	34.82	28.80	10.57
70.00	58.18	54.00	49.61	41.03	15.14
80.00	97.01	90.04	82.74	68.47	25.46
90.00	271.87	252.47	232.10	192.33	72.22

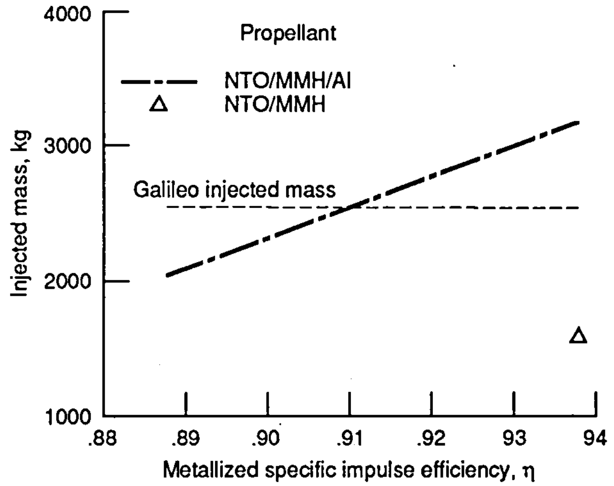

Figure 14.—Sensitivity of Galileo mission performance to specific impulse efficiency. Injection energy, C_3, 80 km^2/s^2.

Figure 14 shows the effect of reduced η on the Galileo-class mission with a C_3 of 80 km^2/s^2. The NTO/MMH η is 0.938. Even if the η is reduced to 0.91, the NTO/MMH/Al stage can still deliver the C_3 required for the mission. Once the η drops to 0.868, the metallized system only delivers the same injected mass performance as the NTO/MMH system.

Clearly, the η will have a very strong influence on reducing the injected mass performance in some of the metallized cases. A penalty of the magnitude predicted for metallized propellants can potentially eliminate their benefits. Small reductions in η, however, can be absorbed with only a small payload penalty. Research on reducing the performance losses of metallized systems has been conducted (ref. 40). Reducing the Al_2O_3 particle size has been shown to reduce the gas and solid-liquid velocity differences, improve the metallized η, and thus improve the delivered payload.

Two-stage vehicle performance.—Past liquid propulsion upper-stage systems, such as the Centaur, Centaur G, and Centaur G-Prime, have not considered two-stage vehicles. Augmentation of the C_3 of the stages with small solid rocket motors has been conducted (as with the Pioneer 10 and 11 and the 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.

An assessment of the performance differences for two-stage vehicles for the planetary missions was conducted. Table XII contrasts the performance of one- and two-stage vehicle performance for three missions. In the tables, the STS and STS-C-constrained vehicle performance is significantly enhanced through staging. 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

TABLE XII.—PAYLOAD CAPABILITY FOR PLANETARY MISSIONS: ONE- AND TWO-STAGE PERFORMANCE COMPARISON

(a) STS launch mass, 24 950 kg; total stage wet mass and injected mass, 20 841 kg

Propulsion technology	Number of stages	
	One	Two
	Injected mass, kg	
Injection energy, C_3, 80 km^2/s^2		
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
Injection energy, C_3, 150 km^2/s^2		
O_2/H_2	121.1	1327.0
$O_2/H_2/Al$	208.9	1380.0
Injection energy, C_3, 160 km^2/s^2		
O_2/H_2	(a)	1163.0
$O_2/H_2/Al$	----	1215.0

[a]Not capable of delivering a payload to this C_3.

(b) STS-C launch mass, 68 040 kg; total stage wet mass and injected mass, 63 931 kg

Propulsion technology	Number of stages	
	One	Two
	Injected mass, kg	
Injection energy, C_3, 80 km^2/s^2		
NTO/MMH	1620.1	4 936.0
NTO/MMH/Al	3191.7	6 092.2
O_2/H_2	8461.2	10 610.0
$O_2/H_2/Al$	8824.9	10 875.0
Injection energy, C_3, 150 km^2/s^2		
O_2/H_2	1133.9	5 140.0
$O_2/H_2/Al$	1455.1	5 240.0
Injection energy, C_3, 160 km^2/s^2		
O_2/H_2	403.7	4 530.0
$O_2/H_2/Al$	721.0	4 717.0

"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.

A two-stage system, although 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, chosen because of the desire to fulfill the needs of a wide range of users and the constraints of existing stages and launch vehicles.

Fixed Stages

In the previous discussion, the maximal performance for the one-stage systems was analyzed. With these data, a performance assessment of a fixed stage was conducted. The fixed-stage design point was chosen on the basis of 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.

Figure 15 compares the performance of two types of fixed and optimized stages: 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.

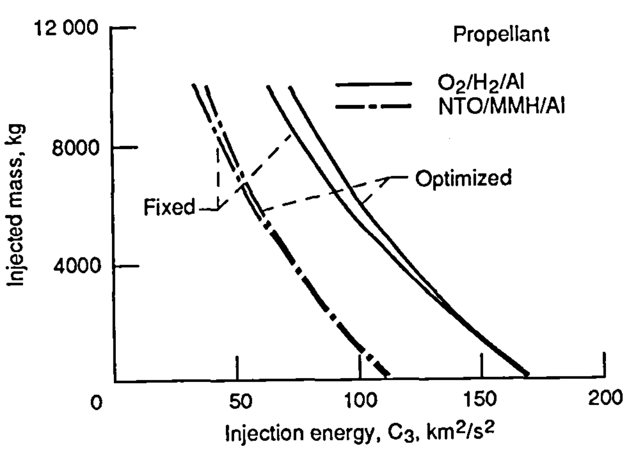

Figure 15.—Comparison of optimized and fixed upper stages for STS-C.

TABLE XIII.—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, 63 931 kg.]

Element	Propellant	
	NTO/MMH/Al	$O_2/H_2/Al$
	Injection energy design point, km^2/s^2	
	80	160
	Mass, kg	
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 percent)	690	815
Total burnout mass	7 589	8 966
Usable propellant	53 150	54 244
Injected mass	3 192	721
Total	63 931	63 931

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 XIII 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. This stage was also designed for the STS-C and has an 8966-kg burnout mass. At this design point, the stage can still perform all the desired planetary missions. The $O_2/H_2/Al$ stage with the STS-C is the only single-stage propellant combination that will capture all the missions.

Concluding Remarks

Many technologies are available to increase 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. In many cases, however, the performance benefits of metallized propellants over their nonmetallized 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 nonmetallized 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 the space-storable propulsion options. Metallized NTO/MMH/Al is therefore recommended as an option for the LEO–GEO transfer mission.

Payload delivery to high Earth-orbital inclinations (65°) will also benefit from metallized fuels. 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 nonmetallized counterpart: 20 percent (with the STS) and 17 percent (with the STS–C). The total payload delivered on the 10 000 km mission with NTO/MMH/Al was 3890 kg (with the STS) and 12 860 kg with the STS–C. This NTO/MMH/Al system is the only combination with large payload benefits for this Earth-orbital mission class. In addition, the potential military need for long-term loitering of these vehicles in orbit may also demand the higher liquid boiling temperatures afforded by Earth- and space-storable propellants. An NTO/MMH/Al system can provide both significantly increased payload performance and easier propellant storability.

Although cryogenic stages provide the greatest payload benefit, a space-storable stage can deliver many of the performance needs for LEO–GEO missions and very significant improvements over the IUS. Both storable and cryogenic propulsion can have a place in future space transfer systems. The use of metallized and nonmetallized space-storable stages in the STS–C can be an effective tool for future missions. In some cases, the payload delivery benefits of the space-storable upper stages with the STS–C are superior to the performance of an O_2/H_2 upper stage in the STS. These significant performance capabilities with only space-storable propellants should not be overlooked.

Metallized $O_2/H_2/Al$ propellants enable a significant performance improvement over nonmetallized combinations in several planetary applications. With $O_2/H_2/Al$ propulsion, all of the very high energy planetary missions that were once rejected because of 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 less than 10 percent. 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- and space-storable metallized propellants also provide attractive options for planetary missions. At a C_3 above 3 km^2/s^2, a stage using metallized NTO/MMH/Al propellants on the STS–C is able to deliver an injected mass increase more than 10-percent greater than that for 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.

Most of the systems considered in these analyses are single-stage systems. Two-stage systems using nonmetallized O_2/H_2 propellants can also enable nearly all of the planetary missions. Using a two-stage system specifically 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 of NTO/MMH/Al and $O_2/H_2/Al$ should both be included in future mission studies. These technologies can benefit launch vehicle upper stages, but, in addition, other studies have shown performance gains for lunar and Mars missions and for the launch vehicles themselves. Potential reductions of tankage volume and mass are also possible with metallized storable propellants. 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 the efficiencies of their nonmetallized 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 should include metallized propellants in the early conceptual design and deployment to gain the maximal advantage from this new vehicle for planetary missions. Applying metallized propulsion options to future upper stages will make them, the STS, and the STS–C safer, more productive, and more cost-effective.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, May 17, 1991

Appendix—Symbols and Acronyms

A,B	mass-scaling parameters
C_3	injection energy (hyperbolic excess velocity squared)
g_o	gravitational constant
I_{sp}	specific impulse
ML	metal loading
MW	molecular weight of combustion products
m	mass
R	radius ratio
r	orbital radius
T	combustion temperature
V_o	initial velocity
V_∞	hyperbolic excess velocity
ΔV	change in orbital velocity
ϵ	expansion ratio
η	specific impulse efficiency
θ	orbital plane change
μ	Earth's gravitational parameter
ρ	density

Subscripts:

circ	circularization
dry	dry
f	final
m	metal in the fuel
o	initial
p	propellant
p,m	metallized propellant
p,n	nonmetallized propellant
te	transfer eclipse
tot	total

Acronyms:

GEO	geosynchronous Earth orbit
IUS	Inertial Upper Stage
LEO	low Earth orbit
NASA	National Aeronautics and Space Administration
NTO/MMH	nitrogen tetroxide/monomethyl hydrazine
STS	Space Transportation System
STS-C	Space Transportation System—Cargo
UFUP	Uranus Flyby/Uranus Probe

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REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY (*Leave blank*)

2. REPORT DATE
February 1992

3. REPORT TYPE AND DATES COVERED
Technical Paper

4. TITLE AND SUBTITLE
Upper Stages Using Liquid Propulsion and Metallized Propellants

5. FUNDING NUMBERS
WU-506-42-72

6. AUTHOR(S)
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7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
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Cleveland, Ohio 44135-3191

8. PERFORMING ORGANIZATION REPORT NUMBER
E-6105

9. SPONSORING/MONITORING AGENCY NAMES(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, D.C. 20546-0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA TP-3191

11. SUPPLEMENTARY NOTES
Responsible person, Bryan A. Palaszewski, (216) 977-7493.

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified - Unlimited
Subject Category 20

12b. DISTRIBUTION CODE

13. ABSTRACT ***(Maximum 200 words)***

Metallized propellants are liquid propellants with a metal additive suspended in a gelled fuel. Typically, aluminum particles are the metal additive. These propellants increase the density and/or the specific impulse of the propulsion system. Using metallized propellants 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. However, using metallized propellants for planetary missions can deliver great reductions in flight time with a single-stage, upper stage system. Tradeoff studies comparing metallized propellant stage performance with nonmetallized upper stages and the Inertial Upper Stage (IUS) are presented. These upper stages, launched from the STS and STS-C, are both one- and two-stage vehicles that provide the added energy to send payloads to high altitude orbits and onto interplanetary trajectories that are unattainable with only the Space Transportation System (STS) and the Space Transportation System-Cargo (STS-C). The stage designs are controlled by the volume and the mass constraints of the STS and 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.

14. SUBJECT TERMS
Metal propellants; Upper stage; Rocket engines; Launch vehicles; Mission planning

15. NUMBER OF PAGES
20

16. PRICE CODE
A03

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	

NSN 7540-01-280-5500

Standard Form 298 (Rev. 2-89)
Prescribed by ANSI Std. Z39-18
298-102

NASA-Langley, 1992