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On-Board Propulsion System Analysis of High Density Propellants

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Abstract

The impact of the performance and density of on-board propellants on science payload mass of Discovery Program class missions is evaluated. A propulsion system dry mass model, anchored on flightweight system data from the Near Earth Asteroid Rendezvous mission is used. This model is used to evaluate the performance of liquid oxygen, hydrogen peroxide, hydroxylammonium nitrate, and oxygen difluoride oxidizers with hydrocarbon and metal hydride fuels. Results for the propellants evaluated indicate that the state-of-art, Earth storable propellants with high performance rhenium engine technology in both the axial and attitude control systems has performance capabilities that can only be exceeded by liquid oxygen/hydrazine, liquid oxygen/diborane and oxygen difluoride/diborane propellant combinations. Potentially lower ground operations costs is the incentive for working with nontoxic propellant combinations.

Introduction

New NASA planetary exploration missions are being conducted under the Discovery Program[1]. One of the goals of this program is to substantially reduce total mission cost while improving performance, through the use of new technology and the control of design/development and operations costs. Funding constraints require the use of a medium (Delta II) class or smaller launch service. The performance and cost of the on-board propulsion system on these missions can be a significant contributor to the program goal of obtaining the highest possible science value per unit cost.

Hypergolic propellants such as nitrogen tetroxide (NTO), monomethylhydrazine (MMH), and anhydrous hydrazine (N2H4) have been the propellants of choice for on-board propulsion on satellites for more than 30 years. These propellants have a high demonstrated system reliability due to flight heritage, system simplicity, and because they ignite readily on contact with one another (hypergolicity). They are classified as “Earth Storable” due to their room temperature normal boiling points. These propellants are highly toxic, however, and their use can entail high operating costs due to increasingly more stringent environmental regulations making it difficult to store, transport, and handle them.

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The purpose of this paper is to evaluate the mission performance of alternative space storable propellants, especially those fuels selected from among the less toxic, high density, hydrocarbons along with space storable oxidizers. The overall propellant density of each fuel and oxidizer combination is calculated for comparison, in order to draw conclusions on the relative value of density vs. performance of propellant combinations. The mission selected for the comparison of alternative propellants is launched on a Delta II-7925. The propellants chosen have low freezing points to minimize thermal control requirements on the satellite.

Propulsion System Dry Mass Model

In order to evaluate the mission performance of various propellants on a Discovery Program[1] class satellite, a propulsion system dry mass model is assembled and anchored on the Near-Earth Asteroid Rendezvous (NEAR) propulsion system[2]. In the model, the mass injected by the launch vehicle is 805 kg. The NEAR propulsion system supplies an axial ΔV=1176 m/sec at a specific impulse of 313 sec using NTO and N2H4 propellants. A ΔV=184 m/sec for attitude control is supplied by monopropellant N2H4 at a specific impulse of 234 sec. The science payload mass on the mission is reported to be 55 kg in Reference 3. The GN&C, and its associated structure is assumed constant in the model comparisons and only the science payload varies with propulsion system performance. The propulsion system model includes models for the propellant and pressurant tanks plus masses for the feed system components and thrusters.

A generalized dry mass model for conventional propellant tanks was developed in Reference 4 and was compared to spherical tanks in the range of 0.0056 to 0.80 m^3 volume. This model is used in this paper and is given by the following equation:

$$M_{\tan k} = 1.2 + K\rho V\left[\left(\frac{P}{\sigma}+1\right)^3 - 1\right]kg$$

where the constant 1.2 is the minimum mass of reinforced inlet and outlet fittings and K is a tank type factor

K = 1.45 for surface tension tanks
= 1.75 for diaphragm tanks
= 1.1 for shell tanks
ρ = tank material density (kg/m^3)
V = tank volume (m^3)
P = maximum operating pressure (kPa)
σ = tank material ultimate tensile stress (kPa)

Using this equation the weight of a HS 601 propellant tank given in Reference 5 is calculated for comparison. This surface tension tank, fabricated from 6AL-4V titanium alloy has a volume of 0.368 m^3, an operating pressure of 1790 kPa, and a mass of 12.1 kg. The calculated mass by this equation is 12.2 kg, showing excellent agreement for this typical flight type tank.

The pressurant tank is modeled by a carbon composite overwrapped pressure vessel and its mass is given by the following equation:

$$M_{pv} = 1.2 + \rho_{liner} v_{liner} + \rho_{wrap} v_{wrap} \text{ (kg)}$$

where the constant (1.2) again accounts for the reinforced inlet and outlet fittings as in Reference 4. A spherical tank with the wall thickness much less than the tank diameter, can then be put in the form:

$$M_{pv} = 1.2 + 1.5V(SF)\left(\frac{\rho_{liner} P_{liner}}{\sigma_{liner}} + \frac{\rho_{wrap}\left(P - P_{liner}\right)}{\sigma_{wrap}}\right)$$

where

$$P_{liner} = \frac{4\sigma_{liner} t_{liner}}{D(SF)}$$

V = tank volume (m^3)
D = tank diameter (m)
SF = safety factor
ρ_{liner} = tank liner material density (kg/m^3)
P_{liner} = maximum operating pressure of the liner (kPa)
σ_{liner} = tank liner material ultimate tensile stress (kPa)
t_{liner} = tank liner thickness (mm)
ρ_{wrap} = composite overwrap material density (kg/m^3)
σ_{wrap} = composite overwrap material ultimate tensile stress (kPa)

This model is anchored using the tank given in Reference 6. This tank has a volume of 0.0673 m^3, an operating pressure of 31 MPa and a safety factor of 1.5. It has a 0.50 mm thick titanium liner with a 5.61 mm thick carbon composite overwrap. The fiber has a tensile strength of 5.5 MPa and the overwrap has a density of 1800 kg/ m^3. The composite overwrap material ultimate tensile strength is anchored at 0.94 MPa (17% of the fiber tensile strength) using the operating pressure and overwrap thickness of the tank given in Reference 6. Using this equation the weight of the pressurant tank given in Reference 6 is calculated to be 11.1 kg, which compares favorably with the given mass of 10.0 kg.

Earth storable propulsion system component masses are supplied by vendors for the NEAR propulsion system and are given in Table 1. Some of the unit masses given are weighted averages of several sizes of components in order to have a generic mass for the model. The component quantities for the NEAR propulsion system are derived from the propulsion system hydraulic schematic given in Reference 2. Their cumulative mass is given in Table 1 in the column marked Earth Storable System. The propulsion system on NEAR, as given in Reference 2, includes three fuel tanks, two oxidizer tanks, and one pressurant tank along with these feed system components.

In the model, the mass injected by the launch vehicle is 805 kg. The NEAR propulsion system supplies an axial ΔV=1176 m/sec at a specific impulse of 313 sec using NTO and N2H4 propellants. A ΔV=184 m/sec for attitude control is supplied by monopropellant N2H4 at a specific impulse of 234 sec. The helium pressurant is tanked at 30 MPa and is assumed to blow down at a slow rate such that isothermal conditions at 25 ^{o}C exist locally. A final pressure of 1.7 MPa exists when all of the propellants are expended. A summary of the propulsion stage mass as determined by the model is given in Table 2 showing a good comparison with the mass summary given in Reference 2. Note that Reference 2 gives the masses in terms of subassemblies and does not call out a specific mass for lines & fittings. The close agreement between the model and the actual propulsion system is predicated on several constants assumed in the model (as shown in Table 2). These constants serve to anchor the model and will remain fixed in subsequent propellant comparisons.

Propellant Selection

Table 3 lists the propellants selected for evaluation in this study. They include the state-of-art Earth storable propellants, hydrocarbons and metal hydride fuels, and several oxidizers. Hydrocarbon fuels are grouped into the following categories: a) alcohols and ethers, b) amines, c) saturated hydrocarbons, d) unsaturated hydrocarbons, e) ring hydrocarbons, and f) strained ring hydrocarbons.

Hydrocarbon fuels were selected against the criteria that they are liquid at both ground and near-Earth ambient space conditions. That is, those were selected that had a freezing point less than -45 ^{o}C and a boiling point above 20 ^{o}C. Table 3 shows the freezing and boiling points of all the propellants evaluated in this study. Note that the state-of-art NTO and N2H4 are liquid on the ground, but require heating in space to prevent them from freezing.

Along with freezing point and boiling point data given in Table 3, the propellant storage density is also given. The effect of this propellant storage density feeds into propellant storage volume and thus into propulsion system dry mass.

Another criteria that was used in the selection of propellants involved an assessment of its ease of handling. In addition to being stable and insensitive to shock, the propellants had to represent less of a toxicity hazard than the state-of-art propellants. The rationale is that a less toxic propellant will require less costly procedures and apparatus for its handling. But, as discussed in Reference 7, the assessment of a propellant's toxicity is not straightforward. In that study toxicity was based on the time weighted average (TWA) of the vapor concentration exposure limit (threshold level value) of the propellant. Table 3 lists this TWA value for the propellants used in this study, as well as their carcinogenicity. These data were obtained from vendor supplied material safety data sheets, although for some propellants, no TWA data was supplied. Note that the state-of-art propellants are highly toxic and one (monomethylhydrazine) is listed as mutagenic. Note, also, that the metal hydride, diborane, violates both the Earth storable and toxicity criteria. It was selected as a high performance option.

The oxidizers chosen for evaluation are also shown in Table 3. Liquid oxygen (LOX) was chosen because it has no toxicity limit, but it must be handled as a mild cryogen on Earth, as indicated by its low boiling point. In order to give a fair assessment of using LOX as the oxidizer on a spacecraft, an assessment of the need for cryogenic components and their mass is conducted. A survey of available vendor information for cryogenic component masses was conducted in Reference 8. An estimate of the mass of multi-layer tank insulation indicates that it is negligible compared to the tank mass. Therefore, the tank models remain the same in this study. The cryogenic propellants can be loaded on the launch pad and topped off just prior to launch. The fairing can be purged with dry nitrogen to avoid frost buildup on the tanks. Only feed system components in contact with the cryogen on the launch pad are given cryogenic component masses. The feed system component masses along with the quantities used and their cumulative mass are given in Table 1 for a one cryogen type propulsion system on a NEAR type mission. These components have a mass of 28.06 kg compared to the 23.52 kg for a non-cryogen type propulsion system. Similar data for a two cryogen type propulsion system is also given in Table 1, showing a cumulative mass for feed system components of 33.75 kg.

LOX was evaluated with all of the fuels selected. The use of LOX dictates the development of highly reliable ignition systems. Catalyst type ignition systems are especially desirable and fuel cell reformer technology[9] may be applicable. The proposed approach is to reform the fuel into hot gases which auto-ignite with the LOX. This technology remains to be demonstrated for this application, but is assumed to exist without weight penalty over the state-of-art systems in the comparisons of this paper.

The second oxidizer chosen is high concentration 90% hydrogen peroxide[10] (H2O2). There is flight experience using this propellant as a monopropellant dating back to the Mercury spacecraft, but it was subsequently displaced by higher performing hydrazine. Note that H2O2 does not meet the low freezing point selection criteria of this paper and has rather high toxicity (low TWA exposure limit). Its use on spacecraft sets thermal control requirements on the spacecraft similar to NTO. The third oxidizer chosen is 82% hydroxylammonium nitrate[11,12] (HAN). This oxidizer has good thermal properties and low toxicity in tests to date. Both H2O2 and HAN can be decomposed exothermically with a catalyst. The resulting gases can be injected for autoignition with the fuel. In this study, H2O2 and HAN were evaluated with JP-10 only for comparison to LOX.

The final oxidizer chosen for evaluation is oxygen difluoride[13]. This oxidizer does not meet the Earth storability criteria and must be handled as a mild cryogen on the ground. It is also highly toxic and does not improve on the toxicity issue of state-of-art propellants, however, it represents a high performance option. It is chosen for evaluation with diborane because they are hypergolic and have similar thermal and toxicity handling requirements.

Engine Performance

The theoretical performance of the selected propellant combinations were calculated using the JANNAF performance prediction code[14]. To speed the calculation, the one-dimensional kinetics (ODK) performance of each propellant combination was calculated and a percentage of the peak value was used. Test data from a LOX/N2H4 engine[15] was used to determine this percentage. This data was for a 1000 N engine with an area ratio of 200:1, operating at a chamber pressure of 1550 kPa and a mixture ratio of 0.8. This engine has a specific impulse of 351 sec, compared to the ODK peak value of 372 sec. The experimental performance, then, was 94% of the peak ODK value. The losses included in the 94% include nozzle divergence and boundary layer losses along with the combustion efficiency. This performance is shown as a data point on Figure 1 along with the theoretical prediction curves based on one-dimensional equilibrium (ODE), one-dimensional kinetics (ODK), and one dimensional frozen (ODF) nozzle flow. The axial engine performance prediction in this paper, then, is set at 94% of the theoretical ODK peak performance and this peak also establishes the MR.

The theoretical ODE, ODK, and ODF performance for the propellant combinations selected for evaluation in this paper are given in Table 4. These values are calculated for 450 N thrust class engines at a chamber pressure of 650 kPa and an area ratio of 200:1. Recombination kinetics data was not available for boron compounds, so the ODK calculation for LOX/Diborane and OF2/Diborane were omitted. The axial thruster performance estimates, set at 94% of the ODK peak, are given in Table 5. The ACS thruster performance estimates are also shown in the Table 5 and are based on a similar estimate for a 30:1 area ratio nozzle. The ODF peak value is used for the propellant combination with diborane. About 9 sec is subtracted from this peak to be conservative.

Model Results

Model results are summarized in Table 6 and presented in terms of science payload in Figure 2. The model of the SOA system predicts an injected satellite mass of 506.6 kg and a propulsion system dry mass of 120.4 kg. As discussed earlier, this is in good agreement with the actual flight system. The science payload mass is 55 kg leaving 315.4 kg for power, GN&C, and its associated structure. Some metrics on the SOA propulsion system are obtained from the model. For example, a 1 sec increase in the thruster Isp (both axial and ACS) yields a 1 kg increase in the 55 kg science payload. A 7.7% increase in propellant density (both oxidizer and fuel) delivers a 1 kg increase in payload by reducing propulsion system dry mass. In order to increase the axial delta-V by 1%, 2.6 kg are subtracted from the science payload. These metrics vary somewhat with propulsion system.

The model was used first to predict the effect of advanced engine technology on the NEAR propulsion system. Iridium-coated rhenium chamber materials are nearing maturity for use with Earth storable propellants, offering higher performance than state-of-art (SOA) silicide-coated niobium chambers. The use of an advanced rhenium engine[16], with an increase in performance to 328 sec, results in an injected satellite mass of 515.5 kg, a propulsion system dry mass of 117.7 kg, and a science payload of 67.0 kg. This gives a net increase in science payload of 12 kg, that is, a 22% increase over the baseline value of 55 kg.

The model was also used with a SOA NTO/MMH system with a 312 sec axial engine and a 293 sec ACS bipropellant engine. These ACS engines are readily available in the 22 N class but not in the 3.5 N class used for precision pointing. Assuming that there is a GN&C solution to this issue, the use of this propulsion system could have added an additional 10.4 kg to the science payload for a total of 65.4 kg. This increase is due entirely to the increased performance of a bipropellant ACS system. This system must be used as a baseline for comparisons with the other bipropellant ACS systems used in this study.

Advanced NTO/MMH engines[17] have been demonstrated with 321 sec axial engine performance and 305 sec ACS performance. The use of these engines would add an additional 9.2 kg to the science payload for a total of 74.6 kg as noted in Table 6.

A summary of model prediction results for the propellant combinations selected for evaluation in this study is given in Table 7 and also presented in terms of science payload in Figure 2. The first system is LOX/Hydrazine operating at SOA pressures with a monopropellant ACS system. The propulsion system dry mass as shown in Table 7a is 125.0 kg due to the use of cryogenic components. The science payload estimated for this system is 67.8 kg, a 12.8 kg increase over the 55.0 kg SOA baseline. Note, however, that a similar performance is obtained with the advanced NTO/N2H4 system shown in Table 6. Since the performance of the ACS system has a significant impact on mission performance, a LOX/Hydrazine bipropellant ACS system with a performance of 325 sec is evaluated and shown as the second entry of Table 7a. This propulsion system has a dry mass of 121.9 kg and delivers a payload of 83.4 kg.

The third entry in Table 7a is for a LOX/Ethanol system. This is the propellant combination favored in an early LOX/Hydrocarbon auxiliary propulsion study[18]. This system gives a payload capability of only 60.2 kg. It offers little new payload capability, but, is completely nontoxic and may be useful in reducing total mission cost. The fourth propellant combination uses a fuel from the amine group. Methylamine is moderately toxic, as shown in Table 3 and has a boiling point that requires dry ice chilling on the launch pad. It has a fairly high axial engine performance with LOX (337 sec). Chemical equilibrium calculations indicate that it decomposes to methane, hydrogen and nitrogen without soot formation and may be useful as a low performing monopropellant for precision pointing. This combination offers a payload capability of 75.0 kg along with reduced toxicity in comparison with hydrazine.

The system performance of LOX/Pentane is given as 70.8 kg of payload in Table 7a. This is 10.6 kg more than LOX/Ethanol with similar toxicity characteristics. Going to an unsaturated hydrocarbon such as 1-Pentene offers no improvement at 70.2 kg of payload. One of the high density fuels (JP-10) developed for air breathing engines[19] is evaluated to provide 67.9 kg of payload. Its higher density does not offset its lower performance in comparison with Pentane. The second entry in Table 7b is quadricyclane, a strained ring compound identified as a potential rocket propellant in the Air Force high energy density matter (HEDM) program[20]. It offers a payload capability of 72.1 kg, which is the best performance of the room temperature hydrocarbons.

With 90%Hydrogen Peroxide/JP10, a payload of 57.0 kg is possible. It offers the same payload capability as the system which was flown and has much lower propellant toxicity. The 82% HAN/JP-10 system only offers a 10.1 kg payload, making it not a contender for this class of missions without a reduction in the delta-V requirement. The final two propellant combinations are high performance options in which both propellants are cryogens. Significant new performance capability is offered with payloads of 94.3 kg and 107.5 kg for LOX/Diborane and OF2/Diborane, respectively. However, both propellants would require handling procedures and apparatus as elaborate as the SOA propellants, and complicated further by their cryogenic nature.

Conclusions

The performance of the on-board propulsion system is evaluated for its effect on science payload mass in a class of missions such as NEAR in the Discovery Program. The effect of propellant performance and density is evaluated for its ability to increase delivered payload mass. An increase in performance, decreases propellant mass and an increase in propellant density, decreases propulsion system dry mass. A propulsion system dry mass model is developed and anchored on the NEAR flight system. Metrics on the state-of-art propulsion system are obtained from the model. For example, a 1 sec increase in the thruster Isp (both axial and ACS) yields a 1 kg increase in the 55 kg science payload. It takes a 7.7% increase in propellant density (both oxidizer and fuel) to deliver the same 1 kg increase in payload by reducing propulsion system dry mass. In order to increase the axial delta-V by 1%, 2.6 kg is subtracted from the science payload.

This model is used to evaluate the performance of liquid oxygen, hydrogen peroxide, hydroxylammonium nitrate, and oxygen difluoride oxidizers with hydrocarbon and metal hydride fuels. Results of the propellants evaluated indicate that the state-of-art, Earth storable propellants with high perfor-mance rhenium engine technology has performance capabilities that can only be exceeded by liquid oxygen/diborane and oxygen difluoride/diborane propellant combinations. Nontoxic propellant combinations can only offer significantly lower ground operations costs. Propel-lant combinations of LOX/hydrocarbons offer science payload delivery similar to the advanced Earth storable technology using rhenium rockets. The propellant combination, 90% hydrogen peroxide/JP-10 offers science payload delivery similar to that of state-of-art Earth storable propellants (57 kg), while 82% HAN/JP-10 would only deliver a science payload of only 10.1 kg. Propellant combinations which outperform the advanced Earth storable systems include LOX/Hydrazine with a science payload of 83.4 kg, LOX/Diborane with a science payload of 94.3 kg and OF_2/Diborane with a science payload of 107.5 kg.

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Table 1.—Propulsion system component masses used in the model along with quantities used in a NEAR type propulsion system and their cumulative mass.

		Earth Storable System		One Cryogen System		Two Cryogen system	
Component	Unit Mass	Quantity	Total Mass	Quantity	Total Mass	Quantity	Total Mass
	kg		kg		kg		kg
Axial Thruster	4.20	1	4.20	1	4.20	1	4.20
ACS Thruster	0.42	11	4.62	11	4.62	11	4.62
Pyro Valves	0.15	0	0.00	0	0.00	0	0.00
Manual Valves	0.22	4	0.88	2	0.44	0	0.00
Check Valves	0.22	5	1.10	3	0.66	1	0.22
Latch Valves	0.75	9	6.75	7	5.25	4	3.00
Relief Valves	0.34	4	1.36	2	0.68	0	0.00
Filters	0.13	8	1.04	8	1.04	8	1.04
Regulator	1.20	1	1.20	1	1.20	1	1.20
Cryo Pyro Valves	0.45	0	0.00	0	0.00	0	0.00
Cryo Check Valves	0.45	0	0.00	2	0.90	4	1.80
Cryo Manual Valves	0.45	0	0.00	2	0.90	5	2.25
Cryo Latch Valves	1.45	0	0.00	2	2.90	5	7.25
Cryo relief Valves	1.45	0	0.00	2	2.90	4	5.80
Pressure Transducers	0.45	5	2.25	5	2.25	5	2.25
Thermocouples	0.01	12	0.12	12	0.12	12	0.12
Total Component Mass			23.52		28.06		33.75

Table 2.—Comparison of the NEAR propulsion system mass as determined by the model with the flight mass given in Reference 2.

	Model	Flight Mass (Ref. 2)
	kg	kg
Propellant Mass + Contingency (5%)	313.4	315.1
Axial Propellant	256.2	-
ACS Propellant	42.3	-
Contingency (5%)	14.9	-
Propulsion System Dry Mass	120.4	121.0
Residuals (1%)	3.0	3.0
Component Dry Mass Total	117.4	118.0
Fuel Tanks	9.6	23.4
Oxidizer Tanks	4.8	11.9
Pressurant Tanks	4.2	10.1
Feed System Components	23.4	31.3
Propulsion Structure (10%)	35.9	33.1
Lines & Fittings (10%)	35.9	-
Electrical Harness, Heaters, etc. (1%)	3.6	8.2
Pressurant Mass	0.9	1.6
Propulsion System Wet Mass	434.7	437.7

Table 3.—Propellant candidates chosen for evaluation in this mission study.

NAME	FORMULA	F.P.	B.P.	DENSITY	TWA	Carcinogen
		^oC	^oC	kg/m3	ppm	
STATE-OF-ART						
Nitrogen Tetroxide	N2O4	-11	21	1431	3	NO
Anhydrous Hydrazine	N2H4	2	113	1004	0.1	NO
Monomethylhydrazine	CH6N2	-52	88	874	0.2	mutagenic
HYDROCARBONS						
(Alcohols & Ethers) Ethanol	C2H6O1	-114	78	789	1000	NO
(Amines) Methylamine	CH5N	-92	-29	769	5	NO
(Saturated HC) Pentane	C5H12	-130	36	626	600	NO
(Unsaturated HC) 1-Pentene	C5H10	-165	29	640	?	NO
(Ring Compounds) JP-10	C10H16	-79	186	940	?	NO
(Strained Ring HC) Quadricyclane	C7H8	-44	108	985	?	?
METAL HYDRIDES						
Diborane	B2H6	-165	-92	437	0.1	NO
OXIDIZERS						
Liquid Oxygen	O2	-218	-183	1149	None	NO
90% Hydrogen Peroxide	H6.82O5.82	-11	141	1390	1	NO
82% Hydroxylammonium Nitrate	H5.42N1.71O 4.42	-64	124	1520	?	NO
Oxygen Difluoride	OF2	-224	-145	1521	0.1	NO

Table 4.—Theoretical performance of selected propellant combinations at 650 kPa chamber pressure, 450 N thrust class, and 200:1 area ratio.

PROPELLANT COMBINATION	ODE peak		ODK peak		ODF peak	
	MR	Isp	MR	Isp	MR	Isp
		sec		sec		sec
LOX/Anhydrous Hydrazine	1.0	390.9	0.8	371.9	0.7	358.8
LOX/Ethanol	1.9	367.9	1.6	341.1	1.5	329.8
LOX/Methylamine	2.2	386.3	1.8	358.5	1.6	346.5
LOX/JP-10	2.6	379.9	2.2	349.2	2.2	333.2
LOX/Pentane	3.0	383.7	2.6	354.9	2.2	340.7
LOX/1-Pentene	2.8	384.4	2.4	354.7	2.2	339.6
LOX/Quadricyclane	2.2	385.6	2.0	352.8	1.8	335.8
90% Hydrogen Peroxide/JP-10	7.0	333.0	7.0	321.7	6.0	317.9
82% HAN/JP-10	12.0	280.0	11.0	276.2	11.0	276.0
LOX/Diborane	2.0	434.4	-	-	2.0	386.7
OF2/Diborane	4.0	468.5	-	-	3.2	393.0

Table 5.—Axial and ACS thruster performance estimates.

PROPELLANT COMBINATION	Axial Estimate		ACS Estimate	
	MR	Isp	MR	Isp
		sec		sec
LOX/Anhydrous Hydrazine	0.8	343	0.8	325
LOX/Ethanol	1.6	321	1.6	302
LOX/Methylamine	1.8	337	1.8	319
LOX/JP-10	2.2	328	2.2	311
LOX/Pentane	2.6	334	2.6	316
LOX/1-Pentene	2.4	333	2.4	316
LOX/Quadricyclane	2.0	332	2.0	316
90% Hydrogen Peroxide/JP-10	7.0	302	7.0	283
82% HAN/JP-10	11.0	260	11.0	245
LOX/Diborane	2.0	378	2.0	354
OF2/Diborane	3.6	384	3.6	367

Table 6.—Model predictions of state-of-art and advanced Earth storable propulsion system performance on a NEAR type mission in which the mass injected by the launch vehicle is 805 kg.

MISSION PERFORMANCE SUMMARY	NTO/N2H4	NTO/N2H4	NTO/MMH	NTO/MMH
	SOA	Advanced	SOA	Advanced
Axial Isp ~ sec	313	328	312	321
ACS Isp ~ sec	234	234	293	305
Propellant Density ~ kg/m3	1118	1139	1154	1154
End of Life Mass ~ kg	506.6	515.5	514.1	521.1
Propulsion system Dry Mass ~ kg	120.4	117.7	118.0	116.1
Contingency Propellant (5%) ~ kg	14.9	14.5	14.5	14.2
Helium ~ kg	0.9	0.8	0.8	0.8
Power, GN&C, etc., ~ kg	315.4	315.4	315.4	315.4
Science Payload ~ kg	55.0	67.0	65.4	74.6
Propulsion System Wet Mass Fraction of Satellite	0.540	0.525	0.527	0.516
Dry Mass Fraction of Propulsion System	0.277	0.279	0.278	0.280

Table 7a.—Model predictions of propulsion system performance with selected propellants on a NEAR type mission in which the mass injected by the launch vehicle is 805 kg.

MISSION PERFORMANCE SUMMARY	LOX/ N2H4	LOX/ N2H4	LOX/ Ethanol	LOX/ Methyla-mine	LOX/ Pentane	LOX/1-Pentene
Axial Isp ~ sec	343	343	321	337	334	333
ACS Isp ~ sec	234	325	302	319	316	316
Propellant Density ~ kg/m3	1044	1063	978	977	933	932
End of Life Mass ~ kg	523.8	535.7	520.8	531.8	529.8	529.3
Propulsion system Dry Mass ~ kg	125.0	121.9	128.8	125.7	127.6	127.7
Contingency Propellant (5%) ~ kg	14.1	13.5	14.2	13.7	13.8	13.8
Helium ~ kg	1.5	1.6	2.1	2.0	2.2	2.2
Power, GN&C, etc., ~ kg	315.4	315.4	315.4	315.4	315.4	315.4
Science Payload ~ kg	67.8	83.4	60.2	75.0	70.8	70.2
Propulsion System Wet Mass Fraction of Satellite	0.524	0.505	0.533	0.515	0.520	0.521
Dry Mass Fraction of Propulsion System	0.296	0.300	0.300	0.303	0.305	0.304

Table 7b.—Model predictions of propulsion system performance with selected propellants on a NEAR type mission in which the mass injected by the launch vehicle is 805 kg.

MISSION PERFORMANCE SUMMARY	LOX/ JP-10	LOX/ Quadri-cyclane	90% H2O2/ JP-10	82% HAN/ JP-10	LOX/ Dibor-ane	OF2/ Dibor-ane
Axial Isp ~ sec	328	332	302	260	378	384
ACS Isp ~ sec	311	316	283	245	354	367
Propellant Density ~ kg/m3	1074	1089	1312	1446	745	989
End of Life Mass ~ kg	525.9	528.7	506.5	470.2	556.0	560.0
Propulsion system Dry Mass ~ kg	126.5	125.3	118.5	127.2	131.3	123.1
Contingency Propellant (5%) ~ kg	14.0	13.8	14.9	16.7	12.5	12.3
Helium ~ kg	2.1	2.0	0.7	0.7	2.5	1.6
Power, GN&C, etc., ~ kg	315.4	315.4	315.4	315.4	315.4	315.4
Science Payload ~ kg	67.9	72.1	57.0	10.1	94.3	107.5
Propulsion System Wet Mass Fraction of Satellite	0.524	0.519	0.537	0.596	0.491	0.475
Dry Mass Fraction of Propulsion System	0.300	0.300	0.274	0.265	0.332	0.322

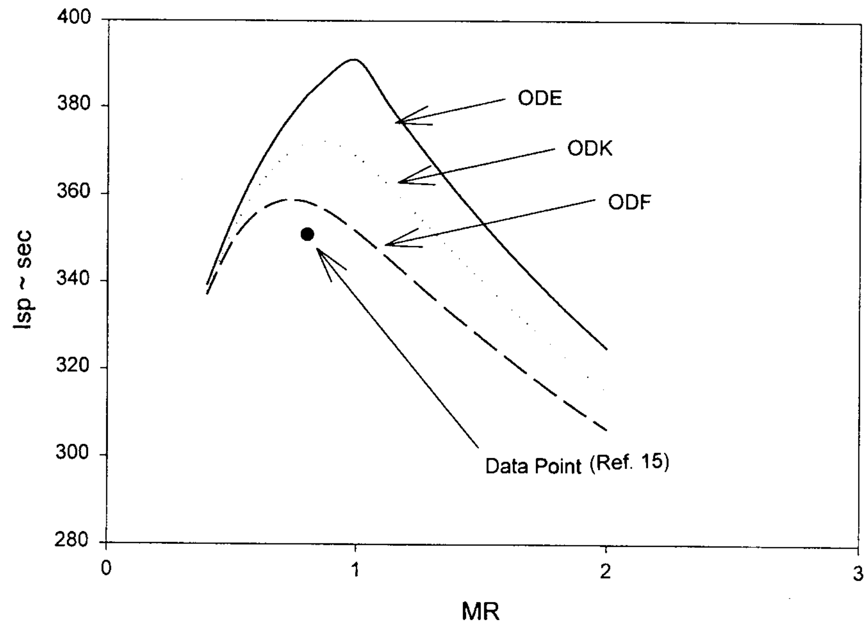

Figure 1.—Comparison of experimental data from Reference 15 with theoretical predictions for LOX/Anhydrous Hydrazine at 1550 kPa chamber pressure, 1000 N thrust class, and 200:1 area ratio. Performance losses are 94% of theoretical ODK peak.

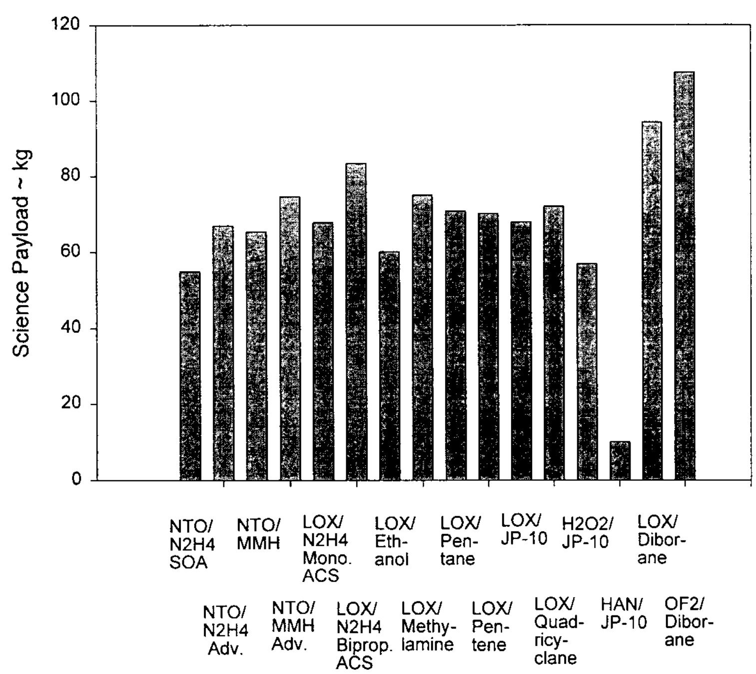

Figure 2.—Comparison of science payload for different propulsion systems on a NEAR type mission

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The impact of the performance and density of on-board propellants on science payload mass of Discovery Program class missions is evaluated. A propulsion system dry mass model, anchored on flightweight system data from the Near Earth Asteroid Rendezvous mission is used. This model is used to evaluate the performance of liquid oxygen, hydrogen peroxide, hydroxylammonium nitrate, and oxygen difluoride oxidizers with hydrocarbon and metal hydride fuels. Results for the propellants evaluated indicate that the state-of-art, Earth storable propellants with high performance rhenium engine technology in both the axial and attitude control systems has performance capabilities that can only be exceeded by liquid oxygen/hydrazine, liquid oxygen/diborane and oxygen difluoride/diborane propellant combinations. Potentially lower ground operations costs is the incentive for working with nontoxic propellant combinations.

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