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ADVANCED APS IMPACTS ON VEHICLE PAYLOADS

Steven J. Schneider and Brian D. Reed
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

ABSTRACT

Advanced APS technology has the potential to both, increase the payload capability of ETO vehicles by reducing APS propellant mass, and simplify ground operations and logistics by reducing the number of fluids on the vehicle and eliminating toxic, corrosive propellants. This paper addresses the impact of integrated cryogenic APS on vehicle payloads. In this system, launch propulsion system residuals are scavenged from integral launch propulsion tanks for use in the APS. Sufficient propellant is preloaded into the APS to return to Earth with margin and noncomplete scavenging is assumed. No propellant conditioning is required by the APS, but ambient heat soak is accommodated. High temperature rocket materials enable the use of the unconditioned hydrogen/oxygen in the APS and are estimated to give APS rockets specific impulse of up to about 444 sec.

The payload benefits are quantified and compared with an uprated monomethylhydrazine/nitrogen tetroxide system in a conservative fashion, by assuming a 25.5 percent weight growth for the hydrogen/oxygen system and a 0 percent weight growth for the uprated system. The combination of scavenging and high performance gives payload impacts which are highly mission specific. A payload benefit of 861 kg (1898 lbm) was estimated for a Space Station Freedom rendezvous mission and 2099 kg (4626 lbm) for a "Sortie" mission, with payload impacts varying with the amount of launch propulsion residual propellants. Missions without liquid propellant scavenging were estimated to have payload penalties, however, operational benefits are still possible.

INTRODUCTION

The mass of the auxiliary propulsion system (APS) on an Earth-to-Orbit vehicle is a significant fraction of the mass delivered to orbit by the launch propulsion system. For example, the Shuttle Orbiter APS (which consists of the orbital maneuvering system (OMS) and the primary and vernier reaction control system (RCS)) contains 12 700 kg (28 000 lbm) for the "Due East" mission with ΔV = 258 m/sec (845 ft/sec), increasing to 18 143 kg (40 000 lbm) for a "Sortie" mission with ΔV = 448 m/sec (1470 ft/sec).[1] The propellant mass fraction of these APS systems is approximately 0.70. Technologies which increase the performance of these APS, therefore, have high leverage to increase the vehicle payload by reducing APS propellant mass.

The high specific impulse of hydrogen and oxygen propellants combined with the possibility of utilizing launch propulsion system residuals in the APS and of eliminating the toxic propellants from this system prompted the technology review and assessment of hydrogen/oxygen APS rockets.[2] The development of liquid hydrogen/liquid oxygen (LH/LO) primary reaction control system (RCS) and gaseous hydrogen/gaseous oxygen (GH/GO) vernier RCS was recommended in that study. Vehicle benefits were anticipated both in performance and in simplification of ground operations. This technology was proposed for the Shuttle Orbiter two decades ago, but LH/LO RCS was dropped because of uncertainties with pulse mode ignition of the rockets and the distribution of cryogens throughout the vehicle. Subsequent technology programs revealed that ignition was not a problem[3] and the distribution of cryogens was technically feasible.[4]

A supercritical pressure, cryogenic propellant distribution system was proposed to avoid two-phase fluid technology issues.[4] This system had bellows expansion devices which accommodated thermal soak into the propellant system but the design sought to control the propellant temperature, a severe problem for an intermittently used RCS system. A recent study proposed a propellant management system with the same supercritical pressure, cryogenic propellant distribution system,[5] but did not seek to control the propellant temperature in the distribution system and allowed the temperature to vary depending on propellant usage. The temperature of the propellant delivered to the primary RCS rockets, then, varied from cryogenic to space ambient temperature, so that at supercritical pressures between 3450 to 6900 kPa (500 to 1000 psia), the density of the propellants delivered to the thrusters varied by a factor of 10. Regeneratively cooled thrust chambers could not be designed with adequate cooling at the extreme off design condition posed by this density variation. Therefore, radiation cooled thrust chambers were the only design solution.

Recent developments[6,7] in high temperature materials give rocket chambers an unprecedented material thermal margin and appear to enable the development of radiation cooled LH/LO primary RCS thrust chambers, as well as GH/GO vernier RCS thrusters. This paper seeks to assess the impact of this technology on the payload of advanced Earth-to-Orbit (ETO) vehicles.

ADVANCED APS TECHNOLOGY

The advanced APS proposed for this study was also proposed for a previous study[5] and is shown in simplified schematic in Fig. 1. In this concept, the launch propulsion and auxiliary propulsion systems would be integrated. The scavenging of liquid propellants from the integral launch propulsion tankage of an advanced ETO vehicle after main engine cut off (MECO) was proposed. Scavenging would be accomplished by using a combination of capillary acquisition devices, thermal subcoolers and low g-fields induced by planned orbital maneuvers. The propellants proposed for scavenging would be the launch propulsion residuals which are needed to maintain a net positive suction pressure head on the main engine turbopumps. The heat soak into the launch propulsion tankage would determine the time frame during which the propellants would be scavenged. This propellant transfer would be accomplished by electrically driven liquid pumps within 2 hr after MECO, before the estimated orbital average heat flux of 114.0 W/m^2 (36 Btu/hr-ft^2) vaporized the propellants. The scavenged propellants would be stored in small, low pressure (172 to 207 kPa (25 to 30 psia)), well-insulated liquid APS tanks and would thereby be protected from the environmental heat soak for the remainder of the mission. These liquid APS tanks would be loaded on the ground to their full capacity to provide propellant for fail-return operation of the vehicle. Scavenging would be accomplished during orbit insertion and circularization burns to replenish propellants used for these maneuvers. Hence, the APS tankage would be full when the vehicle reached final orbit.

The propellants would be pumped from the APS tankage into the supercritical pressure (3450 to 6900 kPa (500 to 1000 psia)), cryogenic propellant distribution system by electrically driven pumps. No control of the heat soak into the distribution system is proposed other than circulation pumps to maintain a uniform temperature in the system components and rocket injectors. With heat soak, the propellant would expand into bellows accumulators in the system and would be delivered to the thrusters with variable density and temperature. Likewise, no propellant heating equipment is proposed to condition the propellants for delivery to the thruster. Thus, the sequencing of these components into operation and the associated system performance and mass penalties are all eliminated. The critical component that would be required is a primary RCS thruster, in the 4450 N (1000 lbf) class, which could operate on propellants with inlet densities and temperatures spanning the range from cryogenic to space ambient. The high temperature rocket technology discussed in the next section describes the material thermal margin which enables this rocket development.

Another feature of the proposed APS system is the use of the launch propulsion tankage as low pressure gas accumulators to supply propellants to low pressure vernier RCS thrusters. The propellants remaining in the launch tankage after liquid scavenging was completed, would readily vaporize and be sufficient in quantity for the attitude control requirements of the vehicle while on orbit. No compressors and high pressure gas accumulators are proposed and the weight of such components, along with the problem of sequencing them into operation, is eliminated. The component required is a low pressure (55 to 104 kPa (8 to 15 psia)), vernier RCS thruster in the 112 N (25 lbf) class, the performance of which could be significantly improved by the high temperature rocket technology discussed in the next section.

Another feature of the proposed APS system is the possible elimination of the OMS rockets. The OMS functions could be provided by redundant primary RCS rockets. There is an unassessed vehicle structural impact resulting from the elimination of the large attachment points of the OMS rockets to the structure and the substitution of the multiple small attachment points of the primary RCS rockets. A preliminary evaluation predicts the latter to be less mass. In the analysis, two 26 690 N (6000 lbf) class OMS thrusters would be eliminated and eight additional primary RCS thrusters would be added to the aft end of the vehicle, which, along with the four thrusters needed for the primary RCS function, would provide OMS level thrust. The net effect on the dry mass of the vehicle due to this substitution is near zero. Therefore, the retention of the OMS class of thrusters, if so desired, would not significantly affect the results of this analysis. The main reason for retaining the OMS class of thrusters would be their higher performance in comparison with the primary RCS thrusters. However, as explained in the next section, high temperature rocket technology is driving up the performance of small rockets significantly more than larger rockets and the performance gap between these two classes of rockets is expected to narrow.

Other components in the APS, include the scavenging apparatus, bellows accumulators, cryogenic propellant storage, and electrically driven pumps. These have matured such that breakthrough technology is not required for development. Scavenging technology for the space transportation system (STS) was outlined in Ref. 8. Multiple-cycle bellows accumulators are in common use in aircraft systems as pressure compensation devices. The power reactant storage assembly (PRSA) on the STS provides a basis for cryogenic APS propellant storage technology. Flightweight, high power, electric motors[9] suitable for use with pumps have been developed for electromechanical actuator programs for aircraft systems.

ADVANCED APS ROCKET TECHNOLOGY

Recent efforts by NASA[10,11] to define and demonstrate high temperature materials for rocket chambers has yielded a rhenium structure with an oxidation protective coating of iridium which operates at a temperature of 2478 K (4000 °F). The impetus for this program was the great mission benefits which would accrue to satellite missions if higher performance, longer life apogee, retro, and attitude control rockets were developed. The program goal was to significantly increase the life and performance of these chemical (monomethylhydrazine/nitrogen tetroxide (MMH/NTO)) rockets. This chamber material program was highly successful with the demonstration of over 15 hr of operation on a small 22 N (5 lbf) rocket at temperatures between 2313 K (3700 °F) and 2588 K (4200 °F) and with an increase in specific impulse of 20 sec.

This program was continued[6] to study the problems of interfacing this high temperature rocket chamber with an injector and of forming a metallurgical joint with the material. The thermal interface (soakback) requirements were met by a combination of techniques. First, a thin wall structure was used to minimize the thermal conduction to the injector and was cooled by a fuel film from the injector. The fuel film was then mixed with the core flow downstream of the thin wall. Reaction of this fuel film with the core flow of the rocket then occurred, resulting in a major performance improvement of the rocket. Heat was radiated away from the chamber by a high emissivity surface on the outside of the rocket formed by a highly dendritic (needlelike) rhenium crystal surface. A zirconia thermal barrier inserted between the chamber and injector further prevented heat conduction to the injector. Highly successful operation of a flightweight thruster occurred, with the demonstration of over 1.7 hr of operation on a 22 N (5 lbf) rocket at 2200 K (3500 °F). The demonstrated specific impulse of this rocket was 310 sec, which was an improvement of more than 20 sec over the conventional silicide coated niobium designs constrained to operate at 1588 K (2400 °F). The durability of this hardware was demonstrated with over 100 000 cycles. A metallurgical study for joining rhenium to other metals was also conducted. The use of Nioro (BAu-4) braze filler metal between rhenium, type 304L stainless steel and Hastelloy B2 provided sufficiently strong joints. Inertia welding was the most promising method for joining rhenium to niobium.

This rocket chamber fabrication technology was scaled to the 445 N (100 lbf) class rocket recently.[7] An estimate was made that incorporation of a high performance rocket engine (I_{sp} = 326 sec) into the spacecraft propulsion system of the initial comet rendezvous asteroid flyby (CRAF) mission would have resulted in a reduction of spacecraft wet mass by 600 kg (1323 lbm). The goal of the program was to demonstrate a 5 percent improvement in propulsion efficiency from 308 to 323 sec of specific impulse and more than 12 hr of life with over 100 starts. An average measured specific impulse of 309.4 sec at area ratio of 44:1, which extrapolates to 323.4 sec at the full area ratio of 467:1. Some fabrication problems were encountered in the chemical vapor deposition (CVD) of the iridium coating on the rhenium, which resulted in local blistering of the coating, although it did not result in chamber failure. Over 4 hr of testing at wall temperatures between 2030 K (3200 °F) and 2200 K (3500 °F) and 33 starts were demonstrated before testing was terminated by funding limitations.

The use of this technology for advanced APS rockets would require a demonstration program of the iridium coated rhenium material with hydrogen/oxygen (H/O) propellants. A rhenium thruster in the 67 N (15 lbf) class has been tested[12] with gaseous H/O propellants for 1.9 hr (192 tests) at a mixture ratio of 2.5 and at a temperature of 2633 K (4280 °F). These rhenium chamber tests demonstrated 411 sec of specific impulse at an area ratio of 100:1, which was compared to the 397 sec of specific impulse of a comparable regeneratively cooled chamber.[13] No degradation of the wall occurred due to oxidation. A coating of iridium on rhenium would provide increased protection against oxidation and allow operation at higher mixture ratios and performance levels.

An extensive technology program[3] to demonstrate LH/LO APS rockets was conducted in the early 1970's. This program resolved many of the initial technology issues with LH/LO APS rockets which included reliable ignition over temperatures ranging from cryogenic to ambient, pulse mode operation, combustion stability and heat soakback into the propellant lines. The 5560 N (1250 lbf) class thrust chamber was fabricated from silicide coated niobium with an area ratio of 40:1 and demonstrated a specific impulse of 427 sec at a mixture ratio of 4.5 and a chamber pressure of 3450 kPa (500 psia). Ten tests were conducted at a chamber temperature of 1533 K (2300 °F) for a total run time of 9.3 sec before a streaking injector terminated the tests. This effort, however, gave a performance data point on which to extrapolate the performance of an iridium coated rhenium LH/LO chamber operating at 2478 K (4000 °F).

A theoretical one-dimensional kinetic (ODK) analysis of the performance improvement to rocket chambers achievable through the use of high temperature materials was conducted. For this analysis, the flow in the rocket chamber was split into two zones: a high temperature core flow zone and a cooler zone of gases near the wall to act as a protective barrier (obviously, this cooler zone of gases can become hotter when higher temperature materials are employed). The ODK rocket analysis gave accurate enough results for this comparative evaluation. Boundary layer losses were assumed to be independent of wall temperature.

In this analysis, two separate zones with different mixture ratios as shown in Fig. 2 are assumed to emanate from the rocket injector. The zone near the wall was assumed to operate at a combustion temperature equal to the maximum desired temperature of the wall, since in the low speed flow of the rocket chamber, the adiabatic wall recovery factor was not significant. The one-dimensional kinetic specific impulse ($I_{sp\ ODK}$) of a two-zone rocket was determined by mass averaging the specific impulses of the separate zones. Theoretical increases in specific impulse for LH/LO rockets achievable through the use of iridium coated rhenium materials, operating at higher wall temperatures than the silicide coated niobium materials, were then estimated.

Equations which specify this mass flow split and the zonal mixture ratios were defined as a function of the overall mixture ratio, $(O/F)_T$, the fraction of total fuel in Zone 2, X_F and the fraction of total oxidizer in Zone 2, X_{OX}. This ODK analysis was then conducted for the LH/LO thruster parameters for which test data exists,[3] i.e., overall mixture $(O/F)_T = 4.5$, chamber pressure of 3450 kPa (500 psia), and area ratio of 40:1. A map of the resulting two zone ODK specific impulse as a function of X_F and X_{OX} is given in Fig. 3. An estimate of the achievable increase in ODK specific impulse for any rocket operating with these parameters, due to increasing the operating temperature of the rocket chamber wall from the 1588 K (2400 °F) of silicide coated niobium to the 2478 K (4000 °F) of iridium coated rhenium, was determined by overplotting the dashed lines of constant Zone 2 mixture ratio $(O/F)_2 = 1.4$ and $(O/F)_2 = 2.7$, respectively, on Fig. 3. This contour plot applies to all rockets with the same chamber pressure, area ratio and overall mixture ratio. The flow field split between Zones 1 and 2 can be rather arbitrary in a given analysis, but one generally uses the injector pattern as an indicator. In this case, the fraction of fuel in Zone 2 (X_F) can be closely approximated by the fuel film coolant fraction. The fraction of oxidizer in Zone 2 (X_{OX}) could be placed there by an injector or could be mixed with fuel film coolant by turbulent mixing and diffusion. For the highest estimated performance the fractions of fuel and oxidizer in Zone 2 should be kept small as shown by the increase in performance along the dotted lines $(O/F)_2 =$ 1.4 and 2.7 in Fig. 3. These fractions of fuel and oxidizer in Zone 2 will naturally become smaller as rockets become larger. The fractions required for any given rocket class are generally determined experimentally due to uncertainties in heat transfer calculations, where the purpose of Zone 2 is to provide a blanket of cooler fuel rich gases near the wall. This blanket, however, mixes with oxidizer rich core gases from Zone 1 and stoichiometric combustion occurs along the shear layer between the zones.

The gaseous hydrogen/gaseous oxygen high temperature rhenium thrust chamber tests[12] had an experimentally determined fuel film coolant fraction of 0.58. When X_F was set at 0.58 for Zone 2 and performance was compared between a 1588 K (2400 °F) wall, $(O/F)_2 = 1.4$, and a 2478 K (4000 °F) wall, $(O/F)_2 = 2.7$, a 30 sec increase in specific impulse was indicated as theoretically possible by the analysis shown in Fig. 3. Based on performance improvements to other rocket chambers through the use of high temperature materials, a 4 percent improvement in propulsion efficiency was projected for a LH/LO rocket. The measured performance[3] of a LH/LO rocket fabricated from silicide coated niobium was 427 sec. The 4 percent improvement in propulsion efficiency to 444 sec represents a 17 sec increase in specific impulse, a conservative performance increase at $X_F = 0.58$. The projected performance benefit due to the use of high temperature iridium coated rhenium materials was, therefore, felt to be a conservative estimate, but needs to be verified by an experimental test program.

One can note from Fig. 3 that very large rockets requiring a small fraction of their total fuel for fuel film cooling do not derive significant benefits from the use of high temperature materials. For example, at a $X_F = 0.1$, there is only a 3 sec benefit to specific impulse. In general, this high temperature material technology can be seen as a technology which is driving up the performance of low thrust rockets much more significantly than large rockets. Primary RCS rockets with this technology could become a suitable substitute for OMS rockets from a performance standpoint, eliminating one of the classes of propulsion on an ETO vehicle.

ADVANCED APS IMPACTS

In order to quantify the impacts of advanced APS concepts, the APS mass on an advanced ETO vehicle, such as Shuttle II (Fig. 4), was estimated for a Freedom rendezvous mission, using an integrated LH/LO APS. For comparison, a state-of-the-art, MMH/NTO APS with uprated OMS engines was sized for an advanced ETO vehicle on the same mission. This analysis was similar to one performed by the authors in Ref. 5. The basis for the analyses are expressed in the following equations. The rocket performance equation is written as:

$$\frac{M_p}{M_{orbit}} = 1 - \exp\left(-\frac{\Delta V}{I_{sp} g}\right) \tag{1}$$

where M_p is the total mass of APS propellant required for propulsion and M_{orbit} is the total mass of the vehicle in orbit.

The dry mass fraction of an APS is defined:

$$f_d = \frac{M_d}{M_d + M_p} \tag{2}$$

where M_d is APS dry mass, which includes the APS mission reserves.

The total mass of APS propellant required for propulsion can be expressed as the sum of the loaded APS propellant (M_{pl}) and the APS propellant which is scavenged from the launch tanks (M_{ps}).

$$M_p = M_{pl} + M_{ps} \tag{3}$$

The fraction of APS propellant which is scavenged is defined as:

$$f_s = \frac{M_{ps}}{M_p} \tag{4}$$

Combining these equations, the on-orbit mass fraction required by the APS can be written as:

$$\frac{M_{pl} + M_d}{M_{orbit}} = \left[\frac{1 - f_s + f_s f_d}{1 - f_d}\right]\left[1 - \exp\left(-\frac{\Delta V}{I_{sp} g}\right)\right] \tag{5}$$

A simple thermodynamic model was used to determine the quality of the propellants in the launch tanks throughout the mission. A detailed mass analysis was performed to determine the masses of the integrated H/O APS components. A description of the basis for these masses is given in Ref. 5.

This paper updated the previous analyses in Ref. 5 using the following assumptions:

- Freedom was assumed to be in a 352 km (190 nmi) orbit. The APS mission profile used in this paper is shown in Table 1.

- In the integrated H/O APS, the primary RCS thrusters were assumed to have a specific impulse of 444 sec and the vernier RCS thrusters, a specific impulse of 397 sec. The thrusters were assumed to be made of rhenium with iridium coating.

- The MMH/NTO APS was assumed to have a performance equal to the uprated OMS engine, 331 sec of specific impulse,[14] compared to the current state-of-the-art OMS performance of 314 sec.

- 5443 kg (12 000 lbm) of launch propulsion residual propellants were assumed to be available in the integral launch tanks at MECO, based on the Orbiter external tank mass properties.[1]

- The hydrogen and oxygen APS accumulators were sized to hold 723 kg (1602 lbm) of liquid hydrogen and 3270 kg (7210 lbm) of liquid oxygen, respectively, and were launched full.

- Based on a study on weight growth in space vehicle development,[15] a 25.5 percent margin for weight growth during advanced ETO development was used. No weight growth margin was assumed for the uprated APS.

The results of the analysis are summarized in Table 2 as a propellant inventory. The analysis shows that during the mission, 29.1 percent of hydrogen residuals were scavenged for propulsion and another 12.1 percent were scavenged for a hydrogen APS reserve. Also, 53.1 percent of the oxygen residuals were scavenged for propulsion with an additional 27.2 percent scavenged for oxygen APS reserve. Therefore, a total of 41.2 percent of the hydrogen residuals and 80.3 percent of the oxygen residuals, which would otherwise be vented overboard in a nonscavenging system, were scavenged and utilized for APS.

The integrated H/O APS had a dry mass (including APS mission reserves) of 7266 kg (16 018 lbm). It was necessary to load 3997 kg (8812 lbm) of APS propellant, giving a total APS mass of 11 263 kg (24 830 lbm). This represented 11.9 percent of the total on-orbit mass at MECO. The uprated MMH/NTO APS uses an APS dry mass fraction of 0.3 (based on current shuttle)[1] and with the same on-orbit mass at MECO, the analysis showed an APS dry mass of 3635 kg (8013 lbm) with 8489 kg (18 715 lbm) of loaded propellant. This gave a total uprated APS mass of 12 124 kg (26 728 lbm) or 12.8 percent of the total on-orbit mass at MECO.

In comparing the masses of the integrated H/O APS with the uprated APS, two competing factors were at work - the amount of loaded APS propellant and the APS dry mass. The integrated H/O APS had the higher dry mass than the uprated APS (53 percent compared to 30 percent), primarily because of the addition of well-insulated cryogenic accumulators, expansion bellows accumulators, heavier rhenium thrusters, numerous components dedicated to scavenging, and weight growth margin. The absence of high pressure accumulators and propellant conditioning equipment minimized this gain in APS dry mass. The combination of using the more energetic hydrogen/oxygen propellants and scavenging residual launch propellants reduced the amount of loaded APS propellant to the degree that it offset the higher dry mass, however.

Although the integrated H/O APS dry mass was about two times the uprated APS dry mass, the uprated APS required 2.1 times the mass of loaded APS propellant that the integrated H/O APS does. This resulted in a mass savings (which can be converted to a payload benefit) for the integrated H/O APS of 861 kg (1898 lbm).

The integrated H/O APS in this paper was evaluated at a specific point (ΔV = 305 m/sec (1000 ft/sec) and 5443 kg (12 000 lbm) of launch residuals). If the APS was sized for a different ΔV or if a different amount of launch residual propellant was assumed, one would expect that the mass savings in using integrated H/O APS would vary. In Fig. 5, an estimate of the variance in mass savings gained by integrated H/O over ranges of APS ΔV and launch residuals is shown. The estimate was not meant as a rigorous calculation, but rather as an indicator of the trends in mass savings or penalties obtained in using integrated H/O APS over the uprated APS. In this estimate it was assumed that no more than 80 percent of the hydrogen residuals and 80 percent of the oxygen residuals were available for scavenging. It was also assumed that the APS accumulators were sized for and filled with at least 25 percent of the mission required propellant for primary RCS. The masses of the other APS components were assumed to be the same for each mission.

The estimate showed that at low ΔV, the use of integrated H/O APS would incur large mass penalties. This was a result of the relatively low amounts of APS mission propellant required at these ΔV's, giving high dry mass fractions for the integrated H/O APS and negating the benefit of reducing loaded propellant through scavenging. As ΔV increased, the amount of mission required propellant increased and at a faster rate than APS dry mass, lowering the dry mass fraction and increasing the importance of reducing loaded propellant. Propellant scavenging, then, became more important, alleviating mass penalties incurred by integrated H/O APS. When there was an insufficient amount of launch residuals to meet scavenging requirements, it was necessary to increase the amount of loaded propellant, detracting from the benefit of scavenging and incurring larger mass penalties than APS's assumed to have more launch residuals available.

As the integrated H/O APS was sized for increasing ΔV requirements, a point was reached where the benefit gained by scavenging offsets the APS dry mass to the degree that integrated H/O would start to provide a mass savings over the uprated APS. When 5443 kg (12 000 lbm) of launch residuals was assumed, mass savings began to occur near ΔV = 229 m/sec (750 ft/sec). When 8165 kg (18 000 lbm) of launch residuals was assumed, savings began near ΔV = 198 m/sec (650 ft/sec), while for 2722 kg (6000 lbm) of launch residuals, mass savings were not shown until ΔV = 553 m/sec (1750 ft/sec). Mass savings became significant for both, higher ΔV's and larger amounts of launch residual propellant. For a "Sortie" mission (ΔV = 448 m/sec (1470 ft/sec)), there was a payload benefit of 4491 kg (9900 lbm) when 8165 kg (18 000 lbm) of launch residuals were available. Mass penalties, however, can also become significant for integrated H/O for low ΔV's and smaller amounts of launch residual propellant. For a "Due East" mission (ΔV = 258 m/sec (845 ft/sec)), there was a payload penalty of 1633 kg (3600 lbm) when only 2722 kg (6000 lbm) of launch residuals were available. The payload benefits of integrated H/O APS, then, will be a function of the sizing and missions of the advanced aerospace vehicle.

A LH/LO APS that does not use liquid scavenging was also evaluated for a Freedom rendezvous mission. The amount of loaded propellant was estimated using the same on-orbit mass as the integrated H/O APS. The APS dry mass fraction was then estimated with the APS accumulators resized for the additional propellant and the scavenging components removed. The nonliquid scavenging H/O APS was found to have an APS dry mass of 6955 kg (15 332 lbm) and requires 6453 kg (14 228 lbm) of loaded propellant. The total APS mass for the nonliquid scavenging case was estimated as 13 408 kg (29 560 lbm) or 14.2 percent of the on-orbit mass. This represents a mass penalty of 1284 kg (2832 lbm) when compared to the uprated APS. However, as with the integrated H/O, there is a point where the reduction in loaded propellant offsets the APS dry mass to the degree that there's a mass savings over the state-of-the-art. In this case, the reduction in loaded propellant comes only through the use of more energetic propellant, though the APS dry mass is decreased somewhat by the removal of scavenging equipment. Using an analysis similar to one used to provide Fig. 5, the point where mass savings begins for nonliquid scavenging was estimated at ΔV = 472 m/sec (1550 ft/sec).

SUMMARY OF RESULTS

Advanced APS technology has the potential to both, increase the payload capability of ETO vehicles by reducing APS propellant mass, and simplify ground operations and logistics by reducing the number of fluids on the vehicle and eliminating toxic, corrosive propellants. This paper addresses only the impact of an integrated cryogenic APS on vehicle payloads. The reduction of APS propellant would result from a combination of using the more energetic hydrogen/oxygen propellants for APS, improved performance gained by APS rockets operating at high combustion temperatures, and scavenging launch residual propellants from integral launch propulsion tanks. The integrated H/O APS concept proposed in this paper would integrate the launch and auxiliary propulsion systems and would employ a supercritical pressure, cryogenic propellant distribution system. There would be no propellant conditioning in this APS, but ambient heat soak into the system would be accommodated. Sufficient propellant would be preloaded into the APS for the vehicle to return to earth with margin. Not all of the launch residuals would be scavengable. The key component to the development of this APS is an LH/LO APS rocket capable of operating over a wide range of propellant inlet states. The recent technology demonstration of a small rocket using chamber materials that allowed operation at 2478 K (4000 °F), offers the thermal margin which enables the development of LH/LO APS rockets. The performance of LH/LO APS rockets using high temperature materials is estimated to be 444 sec. This high temperature material technology also has the effect of driving up the performance of low thrust rockets much more significantly than large rockets. This may allow the replacement of OMS class rockets with redundant primary RCS rockets.

The vehicle payload benefits due to the use of this integrated cryogenic APS are quantified in a conservative fashion by assuming a 25.5 percent weight growth for the H/O system and a 0 percent weight growth for an uprated MMH/NTO system. The combination of scavenging and high performance gives payload impacts which are highly mission specific. The performance of an advanced ETO vehicle employing integrated cryogenic APS with a specific impulse of 444 sec was compared to the same vehicle with an uprated APS using MMH/NTO propellants with a specific impulse of 331 sec (compared to the present value of 314 sec). A Shuttle II type vehicle with integral launch propulsion tankage containing 5443 kg (12 000 lbm) of residuals (based on Orbiter external tank mass properties) and a Freedom rendezvous mission with a requirement of $\Delta V = 305$ m/sec (1000 ft/sec) was used for analysis. The results of the analysis were that a total of 41.2 percent of the launch propulsion system residual hydrogen and 80.3 percent of the launch propulsion system residual oxygen was scavenged for APS propulsion and mission reserve. The integrated H/O APS was estimated to have an APS dry mass fraction of 0.53 (including margin for weight growth). The combination of the advanced APS rocket technology and scavenging launch residuals, however, reduced loaded APS propellant to the degree that a payload increase of 861 kg (1898 lbm) resulted for the vehicle with the integrated H/O APS, when compared to the same vehicle using the uprated MMH/NTO APS. The sensitivity of the payload benefit to mission requirements and to launch residual availability was also estimated. Payload benefits are reduced and become negative as the ΔV requirements and the amount of launch residuals available decreased. For example, payload benefit was projected to drop to zero if the mission required ΔV drops from 305 m/sec (1000 ft/sec) to 229 m/sec (750 ft/sec), or if the propellant available for scavenging drops from 5443 kg (12000 lbm) to 4082 kg (9000 lbm). However, payload benefits became significant as the ΔV requirements and the amount of launch residuals available increased. These trends indicate that the payload benefits of integrated H/O APS will be a function of the sizing and missions of the advanced aerospace vehicle. For example, a payload benefit of 254 kg (560 lbm) was estimated for the Shuttle II on a "Due East" mission, while a benefit of 2099 kg (4626 lbm) was estimated for a "Sortie" mission. A nonliquid scavenging H/O APS was also evaluated for the Shuttle II vehicle on a Freedom rendezvous mission and was estimated to have a payload penalty of 1158 kg (2554 lbm). The sensitivity of this payload impact to mission requirements indicated that both the "Due East" and "Sortie" missions have payload penalties but that payload benefits will occur for missions with $\Delta V > 553$ m/sec (1750 ft/sec). The same operational benefits are possible for this non-integrated H/O system as for the integrated H/O system, however.

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TABLE 1. - MISSION EVENTS AND AUXILIARY PROPULSION REQUIREMENTS FOR THE SHUTTLE II VEHICLE ON A SPACE STATION RENDEZVOUS MISSION

Event description	Time for event, sec	Elapsed time, sec	Change in velocity		APS propellant	
			m/sec	ft/sec	kg	lbm
50- by 100-nm insertion, MECO	-------	0	-----	----	------	------
45 min delay	2 700	2 700	0	----	0	------
100- by 100-nm circularization	50	2 750	27.7	91	601.5	1 326
10-min phasing period	600	3 350	0	----	0	
100- by 190-nm insertion	85	3 435	48.8	160	1048	2 311
45-min delay	2 700	6 135	0	----	0	
190- by 190-nm circularization	84	6 219	48.5	159	1030	2 271
Closure and dock	600	6 819	9.1	30	193.2	426
On-orbit attitude control	259 200	266 019	33.5	110	787.4	1 736
De-orbit	-------	-------	137.2	450	2793	6 158
Total	-------	-------	304.8	1000	6453	14 228

TABLE 2. - PROPELLANT INVENTORY FOR INTEGRATED H/O APS

	H_2			O_2		
	kg	lbm	percent	kg	lbm	percent
APS mission required propellant	1255	2767	-----	5199	11 461	-----
Launch tank residual propellant	1814	4000	100	3629	8 000	100
Loaded APS propellant	727	1602	-----	3270	7 210	-----
Total propellant scavenged	747	1647	41.2	2913	6 423	80.3
Propellant scavenged for propulsion	528	1165	29.1	1928	4 251	53.1
APS mission reserve propellant (scavenged)	219	482	12.1	985	2 172	27.2
Launch residuals vented overboard	1067	2353	58.8	715	1 577	19.7
Percentage of APS mission required propellant scavenged	42.1			37.1		

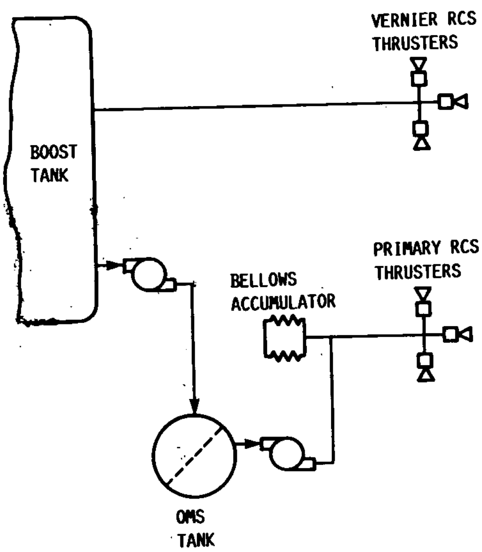

FIGURE 1. - SCHEMATIC OF SCAVENGING H/O RCS.

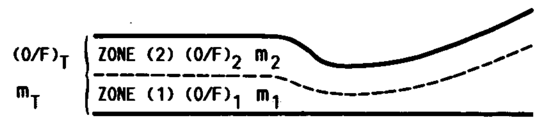

FIGURE 2. - SCHEMATIC OF 2-ZONE ODK ANALYSIS.

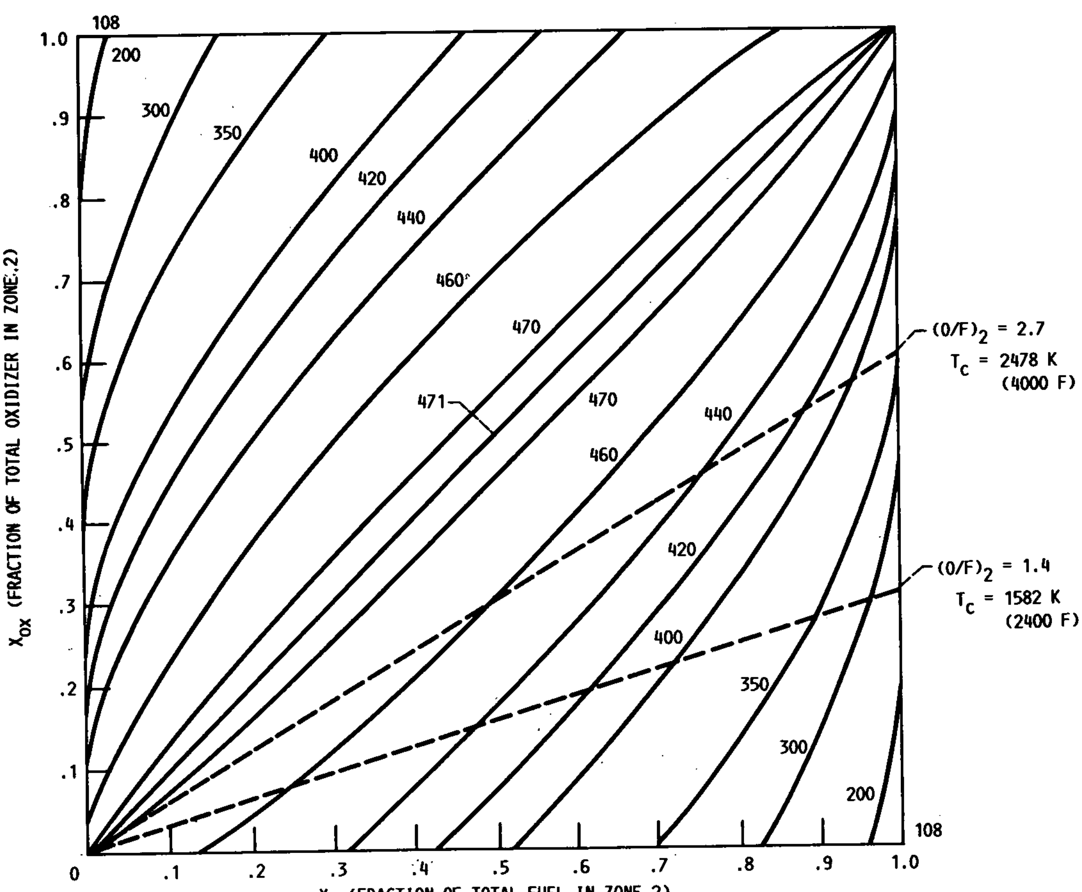

FIGURE 3. - CONTOURS OF ODK VACUUM SPECIFIC IMPULSE AS DETERMINED BY A 2-ZONE ANALYSIS OF AN H/O THRUSTER. $(O/F)_T$ = 4.5, P_c = 500 PSIA AREA RATIO 40:1.

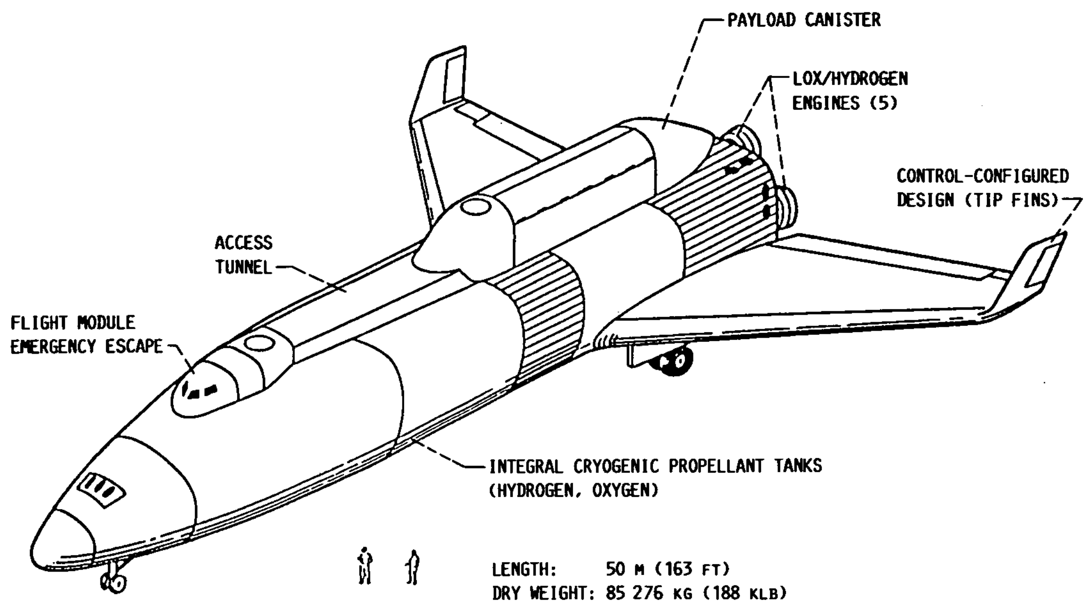

FIGURE 4. - VERSION OF SHUTTLE II ORBITER CHOSEN FOR ANALYSIS IN THIS STUDY.

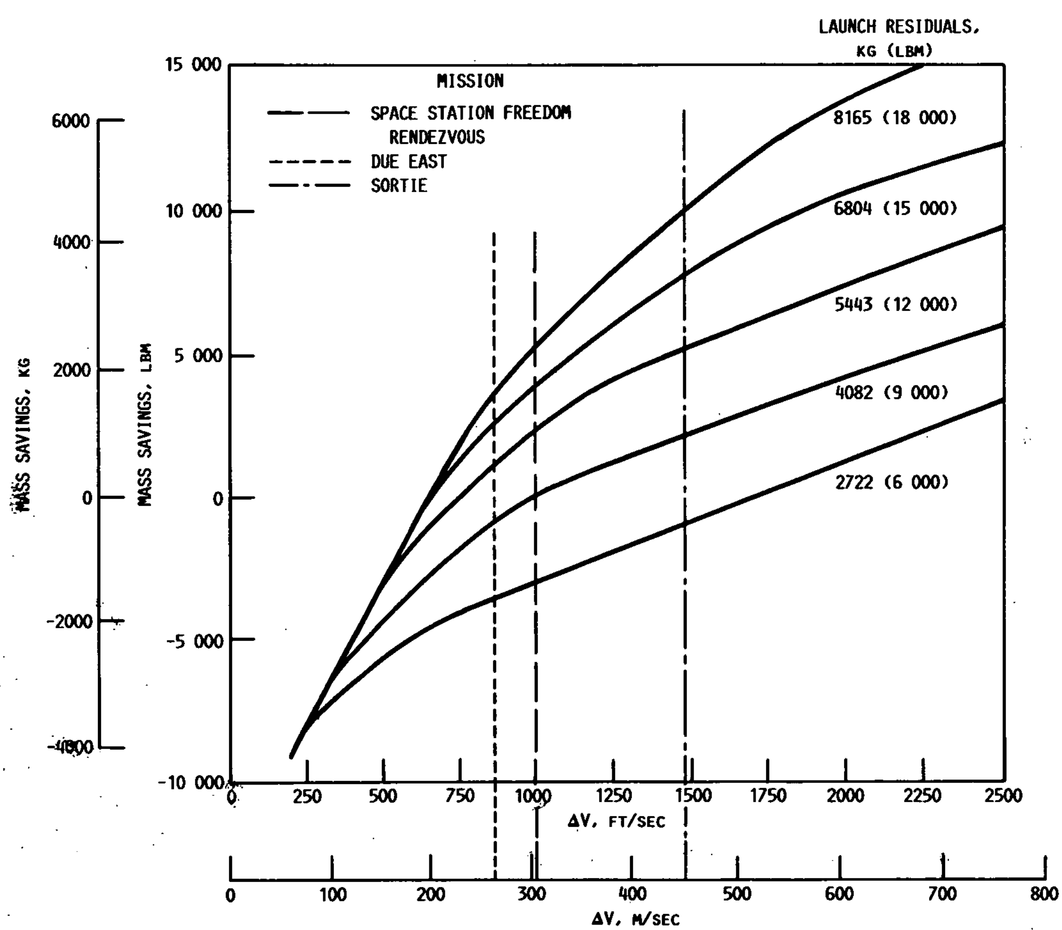

FIGURE 5. - PROJECTED MASS SAVINGS GAINED BY USING INTEGRATED H/O AUXILLIARY PROPULSION INSTEAD OF STATE-OF-THE-ART AUXILLIARY PROPULSION WITH UPRATED OMS ENGINES.