Analytical Assessment of the Reciprocating Feed System

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I. Abstract

A preliminary analysis tool has been created in Microsoft Excel to determine deliverable payload mass, total system mass, and performance of spacecraft systems using various types of propellant feed systems. These mass estimates are conducted by inserting into the user interface the basic mission parameters (e.g., thrust, burn time, specific impulse, mixture ratio, etc.), system architecture (e.g., propulsion system type and characteristics, propellants, pressurization system type, etc.), and design properties (e.g., material properties, safety factors, etc.). Different propellant feed and pressurization systems are available for comparison in the program. This gives the user the ability to compare conventional pressure fed, reciprocating feed system (RFS), autogenous pressurization thrust augmentation (APTA RFS), and turbopump systems with the deliverable payload, inert mass, and total system mass being the primary comparison metrics.

Analyses of several types of missions and spacecraft were conducted and it was found that the RFS offers a performance improvement, especially in terms of delivered payload, over conventional pressure fed systems. Furthermore, it is competitive with a turbopump system at low to moderate chamber pressures, up to approximately 1,500 psi. Various example cases estimating the system mass and deliverable payload of several types of spacecraft are presented that illustrate the potential system performance advantages of the RFS. In addition, a reliability assessment of the RFS was conducted, comparing it to simplified conventional pressure fed and turbopump systems, based on MIL-STD 756B; these results showed that the RFS offers higher reliability, and thus substantially longer periods between system refurbishment, than turbopump systems, and is competitive with conventional pressure fed systems. This is primarily the result of the intrinsic RFS fail-operational capability with three run tanks, since the system can operate with just two run tanks.

II. Propulsion System Mass Estimation Tool

Analytical[1, 2] and experimental[3, 4] work has been conducted in the search for a low-cost high-performance propellant feed system for liquid rocket engines. The nature of this work has been explorative and demonstrative for both in-space[5], launch vehicle, and other applications.[6, 7] There is a need for a tool that can be used for full system trade study comparisons, at least at the conceptual design stage, including mass, reliability, and cost. Such a tool must be able to be validated with data from existing systems. We have developed a trade study code that compares mass and reliability for various propellant feed systems. While the current version of this code does not include cost comparisons, the mass and reliability comparisons allow the primary propellant feed and pressurization systems to be evaluated against each other at the full vehicle system level, and determine the relative ranking of these based on delivered payload, vehicle inert mass, and total propellant mass, for identical mission parameters, such as delta V and engine thrust, for a wide variety of propulsion system architectures (e.g., type of propellant, type of feed and pressurization system, overall flow layout, etc.), material properties (e.g., tensile strength, etc.), design parameters (e.g., safety factors, etc.), characteristics (e.g., propellant mass, chamber pressure, expansion ratio, mixture ratio, specific impulse, etc.), and conditions (e.g., pressurization gas temperatures, etc.).

Our primary objective was to compare the RFS to conventional systems in term of total system performance, especially deliverable payload, but also total system mass and system inert mass. To accomplish this, a computer program based on Excel was developed, which offered ease of use and provided visual representations of the systems being analyzed. The program displays schematics of fifteen propellant feed system and pressurization system configurations including various pressure fed, turbopump, and RFS systems. We also have an advanced version of the RFS that has a form of autogenous pressurization and uses the vented pressurization gases as propellants for small auxiliary thrusters to substantially increase thrust. Currently, four different fuel/oxidizer

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combinations are in the code, and various pressurization systems. An example of a basic sizing code schematic, with selected input and output data, is shown in Figure 1.

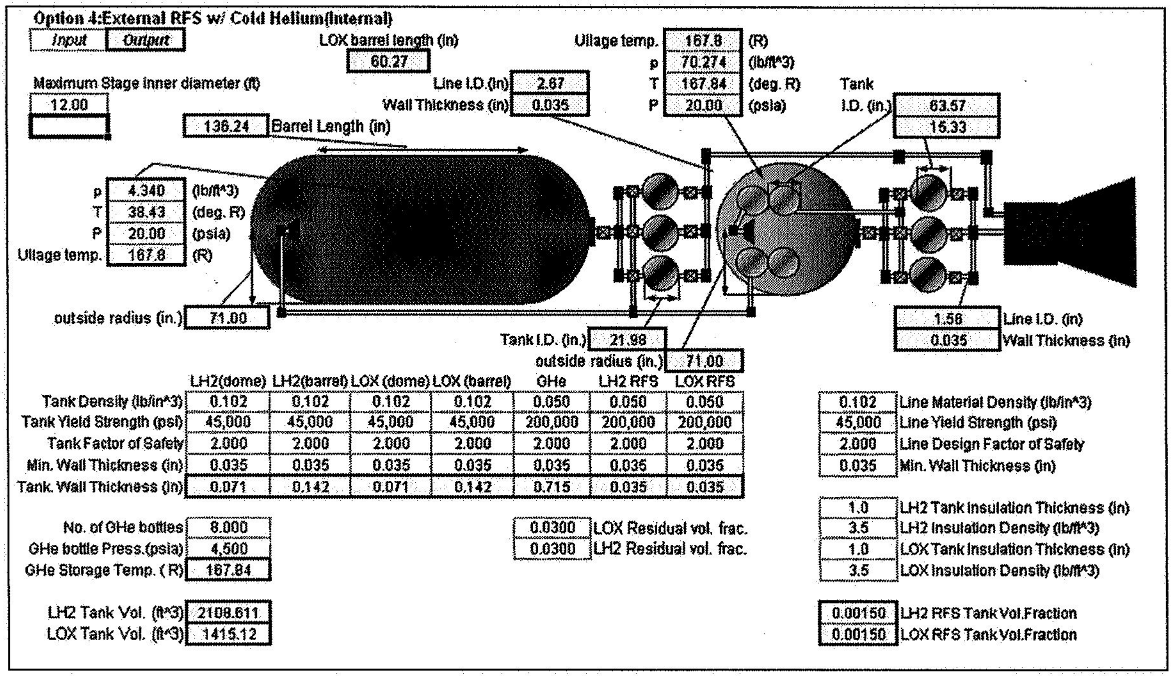

Figure 1 Screenshot of the preliminary system sizing program that shows a schematic of the RFS surrounded by various inputs and outputs

Selections of fuels and oxidizers currently analyzed are LO_2/LH_2, MMH/N_2O_4, LO_2/CH_4, and LO_2/RP-1. The user can input a number of basic mission parameters such as delta V, vehicle inert mass other than payload and propulsion system, thrust, chamber pressure, specific impulse, and mixture ratio. Using these design criteria, the program first calculates the mass of propellant required to provide the average thrust for the given burn time. The mixture ratio determines the masses of fuel and oxidizer; this can be determined separately for a given chamber pressure and expansion ratio. Residual propellants and ullage volumes, typically expressed as percentages, are then used to obtain tank volumes. Tables of properties of propellants and pressurization gases, or closed form expressions, are used to determine liquid and gas densities as a function of temperature and pressure. The pressurant gas mass is then calculated by assuming the pressurant storage vessel is a simple blow down system of an inert gas, stored at high pressure with a constant outflow pressure into the propellant tanks, or variations of this approach, such as with engine-mounted heat exchangers to increase the gas temperature. The amount of pressurant required to displace the volume of propellants, as well as the amount of residual gas, is calculated using basic thermodynamic equations. The pressure drop of the propellant through valves, lines, and ducts as well as the pressure drop across the thrust chamber injector plate is added to the chamber pressure to calculate the total expulsion pressure and pressurization gas mass.

Masses of the storage tanks and pressurization gas are estimated by assuming the main fuel and oxidizer tanks are cylindrical in shape with spherical end caps. The tank diameter is constrained to not exceed the maximum diameter of the vehicle. If the required volume of propellant would fit within a sphere less than or equal to the maximum diameter stipulated by the user, the program would allow the tank to be a sphere. If not, the appropriate sized barrel section would be added to the tank until it was large enough to hold the volume of propellant. Properties of the tank materials (e.g., density, tensile strength, etc.) are provided as inputs so that a user could simulate any material desired (e.g., composite tanks, aluminum, etc.). Thickness of the different tank walls is calculated using the standard pressure vessel design equation involving hoop stress. Safety factors and the minimum gage at which the material is available are accounted for and can be changed by the user. Pressurization gas storage vessels are assumed to be spherical in shape. The volume of gas required can be divided among as many storage vessels as the user desires. The different system options also allow for the storage of these vessels external to the main tanks or submerged within the oxidizer tank or fuel tank, which can be especially beneficial for liquid hydrogen applications. The pressurization gas temperature can be defined by the user, since an engine mounted heat exchanger can be used to heat the gas. The RFS run tanks are assumed to be spherical in shape in the current version and constrained to fit within the vehicle envelope or tank envelope. Configurations of the RFS with the run tanks located external and internal to the main tanks are included as a system architecture option.

The masses of the lines and valves are determined using the flow rates of propellants to the engine as well as line velocities selected by the user. Estimates of the lengths of line required are based on the dimensions of the main tanks. A line length multiplication factor can be used to modify this length. Valves are sized by using a general curve fit of historical data.

Estimates of the conventional pressure fed engine mass for the different systems are based on curve fits from texts[6, 8]. The same curve fit is used for RFS and conventional pressure fed systems. A different curve fit for engine mass is used for turbopump systems, which are typically heavier, for the same chamber pressure.

Estimates of the tank and line insulation are included. The use of spray-on foam insulation is assumed; however, the user can specify the density and thickness of any insulation desired. The default condition has insulation present on all tanks, lines and ducts. However, the user can remove the insulation by specifying an insulation thickness of zero, such as may be the case for storable propellants.

Once propellant, pressurant, and component masses are calculated for all systems, a comparison based on total mass of the system, inert mass, and most importantly, delivered payload, is calculated. It is assumed that any inaccuracies in sizing techniques would be equal across all systems analyzed, regardless of system type. In this way, no unfair advantage was given to any type of system in the analysis. The delta-v input by the user, together with the various design parameters, determines the deliverable payload for each of the systems. To make the comparisons legitimate, it is assumed that the total propellant mass is the same for all options. The user also has the option of varying the chamber pressure so that results for total system mass versus chamber pressure, system inert mass versus chamber pressure, and deliverable payload versus chamber pressure can be generated for all systems. These parameters can be displayed as a function of pressure. The ability to determine the optimum specific impulse and mixture ratio as a function of chamber pressure and nozzle exit diameter limits is typically based on the Cequel™ code. To further ensure that the comparisons are legitimate, it is assumed that the nozzle exit diameter is limited by the stage diameter. A specified nozzle area ratio can also be used.

III. Validation of Propulsion Mass Estimation Tool

A formal validation process of the program's accuracy has not yet been conducted. However, design criteria of Shuttle OMS, Apollo LEM Ascent, and Apollo LEM Descent systems were entered into the program. The calculated results for total system mass and deliverable payload mass were in reasonable agreement to the actual masses in these systems. In most cases, agreement was within a few percent. Formal validation work was initially bypassed to concentrate on higher fidelity system models and to also increase the total number of variations of systems considered. This work and a formal validation process is the topic of on-going research, focused on refining the analyses and extending the comparison to include launch applications as well as in-space propulsion. In the following sections, the predicted values from the analytical tool are compared to the reported values for different propulsion systems.

The following tables show what the inputs and outputs were for the validation tests of the program. Shuttle OMS is given in Tables 1 and 2 (one and two OMS pods, respectively), Apollo LEM Ascent is given in Tables 3 and 4, and Apollo LEM Descent is given in Tables 5 and 6. In all cases the program results were very close to the values for these systems. The discrepancies are typically due to certain assumptions made by the program that are not applicable to these systems. Such assumptions include fuel and oxidizer tanks modeled for the two-pod OMS system as combined tanks instead of separate for each pod, as well as a perhaps over-simplified model of engine weight. In certain cases, available historical data does not specify what constitutes the total engine weight, and thus there is uncertainty as to whether components such as engine mounts, gimbaling mechanisms, cooling jacket lines, or any associated lines or ducts are included in the weight for the basic engine.

Table 1 Major inputs used in the preliminary mass estimation tool for the simulation of a single OMS pod. Major outputs are also shown.

Major Inputs for OMS Pod Simulation		Major Outputs	Simulation	Actual OMS
6,000 lbf	Avg. Thrust	N2O4 Tank mass	249 lb	250 lb
313 sec	Avg. Specific Impulse	MMH Tank mass	251 lb	250 lb
630 sec	Burn Time	GHe Tank mass	261 lb	272 lb
1.65	Mixture ratio	N2O4 propellant mass	7,519 lb	~ 8,000 lb
4,800 psia	GHe storage Pressure	MMH propellant mass	4,557 lb	~ 5,000 lb
125 psia	Chamber Pressure	N2O4 Tank Volume	89.85 ft^3	89.89 ft^3
0.102 lb/in^3	MMH Tank Density	MMH Tank Volume	90.43 ft^3	89.89 ft^3
45,000 psi	MMH Tank Yield Strength	GHe Tank Volume	16.87 ft^3	17.03 ft^3
2.00	MMH Tank Factor of Safety	Engine mass	41 lb	260 lb
0.102 lb/in^3	N2O4 Tank Density			
45,000 psi	N2O4 Tank Yield Strength			
2.00	N2O4 Tank Factor of Safety			
0.075 lb/in^3	GHe Tank Density			
125,000	GHe Tank Yield Strength			
2.00	GHe Tank Factor of Safety			

Table 2 Results from the simulation modeling the two OMS pods as one system

Major Outputs	Simulation	Actual OMS
N2O4 Tank mass	570 lb	250 lb X 2
MMH Tank mass	575 lb	250 lb X 2
GHe Tank mass	523 lb	272 lb X 2
N2O4 propellant mass	15,039 lb	~ 8,000 lb X 2
MMH propellant mass	9,114 lb	~ 5,000 lb X 2
MMH Tank Volume	180.89 ft^3	89.89 ft^3 X 2
N2O4 Tank Volume	179.72 ft^3	89.89 ft^3 X 2
GHe Tank Volume	33.72 ft^3	17.03 ft^3 X 2
Engine mass	72 lb	260 lb X 2
Possible payload	63,265 lb	~ 65,000 lb
Pre-burn Vehicle Mass	255,512 lb	~256,000 lb
Vehicle mass (less consumables)	231,359 lb	~ 230,000 lb

Table 3 Major inputs for the LEM Ascent Stage simulation

Major Inputs	
Thrust	3,500 lbf
Isp	311 sec
Burn time	520 sec
Vehicle weight (less propulsion system)	3,750 lb
Mixture ratio	2.3
Tank material density	0.102 lb/in^3
Tank factor of safety	2
Tank yield strength	45,000 psi
Delta V	7,289 ft/sec

Table 4 Major outputs from the LEM Ascent stage analysis compared to the real system

	Major Outputs	
	From Simulation	LEM Ascent Stage
Total mass	10,879 lb	10,300 lb
Propellant mass	5,852 lb	5,860 lb
Deliverable payload	414 lb	50-300 lb of Rock Samples

Table 5 Major inputs for the LEM Descent Stage simulation

Major Inputs	
Thrust	10,000 lbf
Isp	311 sec
Burn time	560 sec
Vehicle weight (less propulsion system)	3,750 lb
Mixture ratio	2.3
Tank material density	0.102 lb/in^3
Tank factor of safety	2
Tank yield strength	45,000 psi
Delta V	8,110 ft/sec

Table 6 Major outputs from the LEM Ascent stage analysis compared to the real system

Major Outputs	
From Simulation	LEM Descent Stage
22,925 lb	22,375 lb
18,006 lb	18,000 lb
9,427 lb	10,300 lb (LEM Ascent Stage)

IV. Application of Analysis Tool: CEV Analysis

The validation results presented in the last section illustrate that the system design code provides a sufficiently accurate result for at least conceptual design purposes. It should be noted that either referenced data and curve fits, or fundamental engineering equations are used for all mass estimates in our code. Therefore, use of this code should be valid for evaluation of propulsion options for future spacecraft, and, in particular, to assess the RFS relative to conventional pressure fed and turbopump systems. For this purpose, the code was structured such that the system mass, inert mass, and, most importantly, the deliverable payload mass, could be plotted as a function of chamber pressure. As shown below, the RFS allows higher pressure, higher performance engines to be used, compared to conventional pressure fed systems, with very substantial increases in deliverable payload, assuming mission parameters are equal.

To illustrate the use of the analytical tool for conceptual design applications, a sample case consisting of the preliminary mass estimate of an orbital CEV was made. To begin this analysis, a set of design parameters for a CEV[9] were input assuming the use of a conventional pressure fed system. Comparing the results to a RFS is not as simple as using the same design parameters in the RFS portion of the propulsion system mass estimation tool. This is because the RFS allows for the use of a higher chamber pressure engine, which, for the same thrust level and nozzle exit area, will increase the expansion ratio and specific impulse. This will also affect the optimum mixture ratio of the fuel and oxidizer as well as the amount of propellant necessary to fulfill the mission, or alternatively, for the same total mass of propellant, allow for a higher payload. To account for this increase in performance due to the increase in chamber pressure, the Cequel™ computer program was used. Cequel™ calculates the expansion ratio and specific impulse as well as the optimum mixture ratio for a given pressure, nozzle exit area, thrust level, and fuel and oxidizer combination. Using Cequel™, the optimum mixture ratio, vacuum specific impulse, and expansion ratio were found for a thrust level of 15,000 lbf and nozzle exit diameter of 5 feet for the four different fuel and oxidizer combinations available in the system sizing code.

The design requirements for a CEV were obtained[9] or estimated as follows. A single engine with a thrust of 15,000 lbf was assumed. The delta-v was assumed to be 15,400 ft/sec. This was approximately the same as the total delta-v of the LEM ascent and descent stages. The pressure drops across the injector and lines were assumed to be 55 psi and 20 psi respectively. Using the design requirements for total impulse and thrust, a burn time of 724 sec was determined. As with the previous validation examples, standard engineering estimates were used for other design criteria such as tank material properties, ullage volume fractions, residual propellant volume fractions, etc. The design requirements[9] specified a CH_4/LO_2 conventional pressure fed system operating at a chamber pressure of 250 psi. For comparison purposes, chamber pressures of 100, 250, 500, 1,000, and 2,000 psi were used in the Cequel™ code and these results (specific impulse, mixture ratio) then used in the mass estimation tool with conventional pressure fed, turbopump, and reciprocating feed systems. To begin the comparison process, a chamber pressure of 250 psi was used with a conventional pressure fed system, with the pressurant gas stored external to the main propellant tanks. The major outputs, including the amount of propellant and deliverable payload were determined. Next, the same inputs were used with both the RFS and turbopump system. The RFS system model used was the external configuration with the pressurant gas stored internal to the oxidizer tank and routed through an engine mounted heat exchanger before being fed into the different tanks. The turbopump system used was a configuration that stored the small amount of pressurant for the main tank externally to the fuel and oxidizer tanks. The increase in performance due to the increase in chamber pressure by use of the RFS and turbopump was examined by using the data obtained from Cequel™ for chamber pressures of 100, 250, 500, 1,000, and 2,000 psi. To make a comparison on the basis of propellant mass, the burn times of the RFS and turbopump systems were varied until the amount of propellant present in these systems was equal to the amount of propellant present in the conventional pressure fed system operating at a chamber pressure of 250 psi for that propellant combination. This same process was also completed for the conventional pressure fed system for chamber pressures other than 250 psi. Using this process, the difference in the deliverable payload and total system masses of the different types of propulsion systems were found for LH_2/LO_2, CH_4/LO_2, MMH/N_2O_4, and RP-1/LO_2 fuel and oxidizer combinations. As expected, the burn time increased for the higher specific impulse cases, and this resulted in higher payload mass for the given delta-v.

V. Analytical results

The simulations of the Shuttle OMS and LEM ascent and descent stages supported the validity of the mass estimation code. The masses of the Shuttle OMS assemblies and components estimated were within 5% of the actual values. This was also true of the masses estimated for the LEM ascent and descent stages. Tank and propellant masses, which are typically the largest masses in a propulsion system, were determined relatively accurately, with differences of less than 1%. This does not validate the program but it indicates that these mass estimates for other propulsion systems could be reasonably adequate for comparisons of conceptual designs.

Table 7 is based on comparisons of deliverable payload for the same total propellant mass in all cases. The propellant mass baseline assumed was that for the CEV at 250 psi, for CH_4/LO_2. This propellant mass was approximately 28,550 lbs, and therefore the system sizing and performance analysis was conducted such that for all propellant combinations and pressures, the total propellant mass was 28,550 lbs, for the same mission parameters. For the RFS and turbopump systems, the increased I_{sp} at higher pressures resulted in longer burn times for the same amount of propellant. The use of the same total propellant mass provides a more valid comparison than assuming a constant engine burn time, and the increased burn time is relatively small, compared to the substantial increase in payload mass. When the results from Table 7 are considered, several interesting aspects emerge.

Table 7 Summary of the CEV analysis with the same propellant mass of 28,550 lbs, based on the 250 psi conventional pressure fed LO_2/CH_4 case

Propellants	Conventional Pressure fed Chamber Pressure (psia)	Conventional Pressure fed Payload (lbm)	RFS Chamber Pressure (psia)	RFS Payload (lbm)	Conventional Turbopump Chamber Pressure (psia)	Conventional Turbopump Payload (lbm)	Autogeneous Chamber Pressure (psia)	Autogeneous Payload (lbm)
LO_2/CH_4	250	2,919	250	5,830	250	5,822	-	-
	500	570	500	6,193	500	6,217	-	-
	1,000	0	1,000	5,867	1,000	6,141	-	-
LO_2/LH_2	250	0	250	8,726	250	9,277	250	10,311
	500	0	500	8,593	500	9,583	500	10,807
	1,000	0	1,000	7,242	1,000	9,460	1,000	10,873
LO_2/RP-1	250	1,841	250	4,656	250	4,587	-	-
	500	0	500	5,018	500	4,938	-	-
	1,000	0	1,000	5,503	1,000	5,452	-	-
MMH/N_2O_4	250	2,283	250	4,249	250	4,450	-	-
	500	488	500	4,351	500	4,699	-	-
	1,000	0	1,000	3,662	1,000	4,502	-	-

First, as expected, the penalty for operating conventional pressure fed systems at high pressure becomes substantial, typically dropping to zero payload above about 500 psi.

Second, the RFS and turbopump systems provide substantially higher payload masses than conventional pressure fed systems, for all propellant combinations, and for all pressures.

Third, with the conventional pressure fed system, the LO_2/LH_2 propellant combination is not capable of producing a net payload, for the mission conditions assumed, primarily due to the low LH_2 density, relatively large tank volume, and thus tank mass, at 250 psi and above.

Fourth, LO_2/LH_2 with the RFS or turbopump systems provides higher payloads than for LO_2/CH_4, MMH/N_2O_4, and LO_2/RP-1 with conventional pressure fed systems or for RFS or turbopump systems with propellant combinations other than LO_2/LH_2. The LO_2/LH_2 with RFS or turbopump systems provides at least three times as much payload as any of the conventional pressure fed systems, including the baseline 250 psi LO_2/CH_4 system with 28550 lbs of propellant as the common amount for all cases. The RFS and turbopump LO_2/LH_2 systems offer approximately 1.5 to 2 times as much payload as the CH_4/LO_2, LO_2/CH_4, or LO_2/RP-1 RFS and turbopump systems. Clearly, there are substantial payload advantages, as expected, using LO_2/LH_2, but not with the conventional pressure fed system.

Fifth, the maximum amount of payload the LO_2/LH_2 RFS could deliver was within approximately 10% that of the turbopump system. Therefore, the RFS could be an enabling technology for LO_2/LH_2 that would allow for similar increases in payload as a turbopump, but with less development time and costs. It is also shown in the following section that the RFS would provide a higher reliability than for a turbopump system, due to the RFS fail-operational capability.

Sixth, the LO_2/CH_4 conventional pressure fed system provided 2,919 lbs of payload compared to the 2,283 lbs of payload for the MMH/N_2O_4 system, a 28% increase. However, the MMH/N_2O_4 system with the RFS provides 4,351 lbs of payload, which is about 50% greater than that for the LO_2/CH_4 conventional pressure fed system. There could be substantial cost and development time advantages, as well as greater deliverable payload, by using legacy Apollo era storable propulsion systems, combined with the RFS, as opposed to developing a new engine and propulsion system based on LO_2/CH_4.

Seventh, the use of the RFS approximately doubled the amount of payload delivered by the 250 psi conventional pressure system for the LO_2/CH_4, MMH/N_2O_4, and LO_2/RP-1 systems. This is a considerable amount of additional payload that could be gained with little increase in total system complexity, development time and cost, while still employing a low-pressure engine. With the RFS system operated at somewhat higher pressures, even higher payload mass results for the CH_4/LO_2, LO_2/CH_4, and LO_2/RP-1 systems.

Finally, it should be noted that the payload that the APTA RFS for the LO_2/LH_2 propellant combination is roughly 3.7 times as much payload as the baseline CEV LO_2/CH_4 conventional pressure fed case, with 10,873 lbs versus 2,919 lbs, and about 4.7 times that for the CEV MMH/N_2O_4 case. The APTA RFS would sequentially vent the pressurization gases through an auxiliary thruster to produce additional thrust. This system, although more complex than the basic RFS, is far less complicated as a turbopump system, and is less complicated than even the turbopump's gas generators. Gases that would normally be vented are simply combusted. Therefore, the APTA

RFS, which is currently only a conceptual design, should be further investigated. Efforts are now underway at the University of Alabama in Huntsville's Propulsion Research Center to test an APTA engine with gaseous hydrogen and oxygen.

With the RFS payload advantages and relatively simple operation, and the demonstrated ease of control seen in the joint UAH/NASA test program, it is legitimate to ask the question: Is it better to develop a new LO_2/CH_4 propulsion system for the CEV, because it gives a 28% payload increase compared to Apollo-era MMH/N_2O_4 propulsion systems, or is it better to develop the RFS for whatever propellant combination is determined to be preferred, and at least double the payload? Similarly, it appears that the APTA RFS could provide even greater payload mass than a turbopump, with far less development time and costs. Therefore, it is apparent from this study that the use of the RFS deserves consideration in any propulsion system trade study.

VI. Reliability Analysis

Reliability of the RFS as compared to conventional pressure fed and turbopump systems was also considered. The MIL-STD-756B[10] approach was used, with reliability block diagrams based on simplified schematics for typical propellant feed systems, with quad valves, vent valves, pressurization valves, etc. The reliability of the various components was determined for typical operational times and characteristic Mean Time Between Failure (MTBF) values. The results demonstrate that the RFS has an overall system reliability that is comparable to that of the conventional pressure fed system, and higher than that of the turbopump system using a gas generator for the turbine drive. In the turbopump system reliability comparison, it was assumed that the turbopump itself was 100% reliable. It was also assumed that the system had two gas generator inlet valves for the fuel and oxidizer and quad main propellant valves, all with equal MTBF values of 1,000 hours. This type of analysis can only be considered as illustrative of the system reliability comparisons, since there are far higher levels of fidelity required in such analyses, but it does provide support for the RFS with its intrinsic fail-operational mode.

One conclusion drawn from the reliability analysis that is particularly relevant to space based propulsion systems is the difference in the expected time between refurbishment or replacement of the propellant feed system components. The reliability comparison in Figure 2 illustrates this effect. For example, if a reliability of 0.999 is chosen as the point at which the components of the feed system need to be refurbished, then this would occur after roughly a half hour for the turbopump system analyzed. The RFS would not require refurbishment until it had operated approximately 11 hours and the conventional pressure fed system would not require refurbishment until it had operated for approximately 20 hours. This means the turbopump would require maintenance or refurbishment roughly 20 times more often than the RFS and 40 times more often than the conventional pressure fed system.

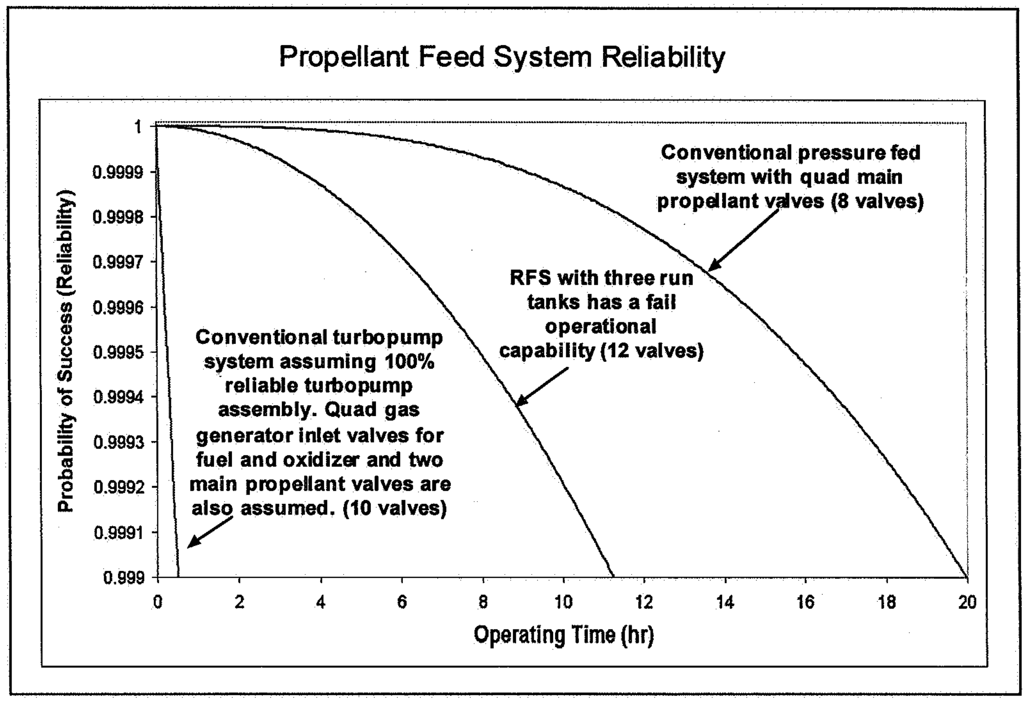

Figure 2 The turbopump system analyzed would require maintenance or refurbishment several times more often than either the RFS or conventional pressure fed systems

Part of the reason for the RFS having a relatively high reliability is that it has a fail-operational mode. The redundancy inherent in the use of three RFS tanks, and the ability of the RFS to operate on only two tanks, allows for a fail-over to the two-tank system. It should be noted that in these comparisons, the same MTBF values were used for all valves and tanks, to be consistent. It should also be noted that the RFS, as a three-tank system, only has one more of each of the valves (main propellant valves, vent valves, pressurization valves, etc.) as compared to the conventional pressure fed systems. In effect, the RFS has the same operational response to the loss of a single valve as the conventional pressure fed system with quad valves. Thus, the number of additional valves required for the RFS is only 50%, not a factor of three, when comparing a three tank RFS with a conventional pressure fed system with quad valves. The RFS meets the same requirement as a conventional pressure fed system with quad valves, in that no credible single failure of the components will result in a mission failure.

VII. Conclusions

Results from the analytical tool for various cases indicate that the RFS offers a substantial increase in payload as compared to conventional pressure fed systems. This payload increase is primarily due to the decrease in the inert mass of the main storage tanks and the increase in specific impulse, for a given nozzle exit diameter, at higher chamber pressures. The RFS also appears to be competitive with a turbopump for systems with chamber pressures as high as 1,500 psi. Above this pressure, the mass of the pressurant gas and gas storage tanks become a significant mass penalty to the system. More advanced pressurization systems with lower mass would further improve the performance of the RFS, relative to conventional propellant feed systems. Other work in propellant feed systems and replacements for turbopumps support these conclusions.[2, 3]

The program does not take into account other important design parameters such as schedule, cost, and only a rudimentary reliability analysis has been conducted. These are aspects that will definitely play a role in the development of a new propulsion system. For example, if an RFS provides slightly less payload than a turbopump system, say 85%, at a much lower cost, say one-tenth that for the turbopump, it may be the most cost-effective design. A situation such as this may allow the RFS to compete with a turbopump even if it is not capable of delivering more payload.

The APTA RFS concepts modeled in the design tool provide a considerable increase in payload as compared to the basic RFS technique, and our results show that that the APTA RFS provides greater payload than the turbopump system. This is primarily due to the fact that additional thrust is created by routing the vented pressurant gas through small thrust augmentation engines. This means the pressurant gas is no longer an inert mass carried along by the propulsion system but a fuel to be consumed. The APTA RFS delivered payload is comparable to, and somewhat higher than, the turbopump system, up to about 2,000 psi.

From the reliability analysis, it appears that a RFS is comparable in terms of reliability to a pressure fed system, but, with its fail-operational capability, appears to be more reliable than a turbopump system with a gas generator. This analysis shows that the reliability of the turbopump is less than that of the pressure fed or RFS systems even when it was assumed that the turbopump assembly components (bearings, shaft, turbines, housing, etc.), other than its main fuel and oxidizer valves, have a reliability of 100%.

It can also be concluded from this analysis that the RFS may be more suitable than a turbopump for long duration missions due to the fact that it would require less maintenance. The turbopump would have to be overhauled approximately 20 times to maintain a reliability as high as the RFS over a given time period. The pressure fed system is more reliable for a long-term mission, but is limited by tank masses when large amounts of propellants are required.

With the RFS payload advantages and relatively simple operation, and the demonstrated ease of control seen in the joint UAH/NASA test program[4], it is legitimate to ask the question: Is it better to develop a new LO_2/CH_4 propulsion system for the CEV, because it gives a 28% payload increase compared to Apollo-era MMH/N_2O_4 propulsion systems, or is it better to develop the RFS for whatever propellant combination is determined to be preferred, and at least double the payload? Similarly, it appears that the APTA RFS could provide even greater payload mass than a turbopump, with less development time and costs. Therefore, it is apparent from this study that the use of the RFS deserves consideration in any propulsion system trade study. We therefore urge that such further consideration be given to this option in trade studies of new propulsion systems.

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