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A Reduced Order Lunar Lander Model for Rapid Architecture Analysis

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The conceptual design of an architecture for space exploration involves the evaluation of many concepts. These design spaces may encompass millions or billions of options when each trade is evaluated at the system, vehicle, subsystem, and component level. Various techniques are typically employed to select the configuration of systems that best meets the requirements of the architecture. These include multi-attribute decision making techniques as well as optimization with the use of genetic algorithms and other stochastic methods. In order to speed up the evaluation of these options, a set of reduced-order vehicle models can be used. These models evaluate the gross weight, dry weight, cost, and reliability of a vehicle given a set of programmatic and performance options in less than a second, versus the use of design codes that take on the order of minutes to hours to converge to a vehicle design. The use of such reduced-order models also enables other techniques that would otherwise take too long to run, such as Monte Carlo simulation to model uncertainty, as well as optimization of the vehicle and studies of sensitivities to changes in programmatic and performance inputs.

A reduced-order lunar lander model is presented, utilizing response surface equations (RSEs) in place of detailed disciplinary simulations. While some fidelity is lost in approximating these disciplines with RSEs, this approach can be used to evaluate the relative impact of various trade studies at the subsystem, vehicle, and architecture levels. The propulsion system is modeled using a response surface of the REDTOP-2 code. In a similar manner, the trajectory for lunar descent and ascent is simulated using Program to Optimize Simulated Trajectories (POST), and then approximated with a RSE for use in the reduced-order lunar lander model. The weights and sizing model of the lunar lander is based on a combination of historical mass estimating relationships (MERs), and physics-based mass estimating relationships. Development and production cost modeling is performed using the Cost Estimating Relationships (CERs) from the NASA-Air Force Cost Model (NAFCOM).

Because the reduced-order lunar lander model evaluates rapidly, stochastic optimization methods such as genetic algorithms can be used to find the performance inputs (such as thrust-to-weight ratios, propellant choices, and expansion ratios) that optimize the vehicle for smallest mass, highest reliability, or smallest development cost. A user-customizable Overall Evaluation Criterion (OEC) can be used to optimize the vehicle for a weighted combination of multiple criteria. Within an architecture analysis, this quick turn-around is useful for rapidly designing the lunar lander to meet the mass constraints of the launch vehicles, and the cost and reliability constraints of the programmatics.

Nomenclature

CaLV	=	Cargo Launch Vehicle
CER	=	Cost Estimating Relationship
CEV	=	Crew Exploration Vehicle

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DDT&E	=	Design, Development, Testing, & Evaluation
DOE	=	Design of Experiments
DSM	=	Design Structure Matrix
EDS	=	Earth Departure Stage
EOR	=	Earth Orbit Rendezvous
ESAS	=	Exploration Systems Architecture Study
LH_2	=	Liquid Hydrogen
LOC	=	Loss of Crew
LOM	=	Loss of Mission
LOR	=	Lunar Orbit Rendezvous
LOX	=	Liquid Oxygen
MER	=	Mass Estimating Relationship
MMH	=	Monomethyl Hydrazine
NAFCOM	=	NASA-Air Force Cost Model
NASA	=	National Aeronautics and Space Administration
NTO	=	Nitrogen Tetraoxide
OEC	=	Overall Evaluation Criterion
POST	=	Program to Optimize Simulated Trajectories
REDTOP-2	=	Rocket Engine Design Tool for Optimal Performance
ROSETTA	=	Reduced Order Simulation for Evaluation of Technologies and Transportation Architectures
RSE	=	Response Surface Equation
T/W	=	Thrust-to-Weight ratio
TFU	=	Theoretical First Unit

I. Introduction and Background

The conceptual design of complex systems frequently involves the use of simplified models to enable the quick evaluation of complicated analyses. This is especially useful when designing architectures for space exploration. Optimization of systems within the architecture, such as transfer vehicles, capsules, habitats, and launch vehicles, often requires optimization within non-linear, multi-modal, and discrete design spaces. Optimization of these problems is commonly performed with stochastic, non gradient-based optimizers such as genetic algorithms. Performing these optimizations, which utilizes hundreds to thousands of function calls, requires a model that evaluates quickly.

The renewed focus of NASA on transitioning to lunar exploration requires the study of a wide array of exploration system options for returning to the moon[1]. A huge number of design decisions must be made in a relatively short period of time. Each design decision has an impact on the overall life cycle cost, safety, and reliability of the system. Therefore, it is necessary to understand the relative impact of each design variable option on these parameters early in the design phase. The use of a Reduced Order Simulation for Evaluation of Technologies and Transportation Architectures (ROSETTA) model allows designers to gain important knowledge of the design space early in the design process.

The ROSETTA modeling process was developed in the Space Systems Design Lab at Georgia Tech as a way of creating meta-models that are Excel-based, and are thus quick to execute. Whereas in a traditional design process multiple team members individually use their disciplinary codes (such as POST for trajectory analysis), in the ROSETTA modeling process, experiments are conducted using the disciplinary codes to fit RSEs, which execute in under a second. While some fidelity is lost in this process, if the important design variables and responses for each discipline are known ahead of time, the result is a quickly executing model that has acceptable accuracy for use at the conceptual stage of design. In the creation of a ROSETTA model for a lunar lander, RSEs of trajectory and propulsion are used. Weight estimation is performed using historical and physics-based mass estimating relationships. Cost estimation is performed using the NAFCOM cost estimating relationships, which are curve-fits derived from previous missions with the addition of a complexity factor multiplier. Vehicle reliability is modeled in the conceptual design phase using a fault tree of subsystem reliabilities. Subsystem reliabilities are derived from historical analysis of vehicle subsystems. This methodology can yield an estimate of loss of mission and, given abort system reliabilities, loss of crew probability.

In this paper the process for creating a ROSETTA model for an in-space transportation system is described, with the lunar lander as an example. Optimization is performed with various objective functions: minimizing mass, minimizing cost, and maximizing reliability.

II. Design Methodology & Trade Space

The first step in creating a ROSETTA model is bounding the design space, for which a Morphological Matrix is employed. In a Morphological Matrix, each row represents a design variable that in some way impacts the configuration or performance of the system being modeled. Each column represents either a discrete option for that design variable, or the upper or lower bounds on a continuous design variable. A Morphological Matrix for the configuration of the lunar lander vehicle is shown in Table 1.

Table 1. Top-Level Lunar Lander Morphological Matrix.

Discrete Variables	**Option 1**	**Option 2**	**Option 3**	**Option 4**	**Option 5**	**Option 6**
Number of Stages	1	1.5‡	2	2.5‡		
Number of Crew	1	2	3	4	5	6
Continuous Variables	**Min**	**Max**				
Payload to Lunar Surface (kg)	0 kg	5000 kg				
Number of Days on Lunar Surface	3 days	14 days				
Number of Days Loiter in Low Earth Orbit	0 days	180 days				
Volume/Crew Member	3 m^3	14 m^3				

The next step is to determine how the information flows between subsystems. In a traditional group design, this involves the flow of information between disciplinary specialists, each employing his own code or analysis. In creating a ROSETTA model, these disciplinary specialists are replaced with a RSE of their code, which is applicable within the relevant design space determined with the help of the specialist. The flow of information between these disciplines is laid out using the Design Structure Matrix (DSM), shown in Figure 1.

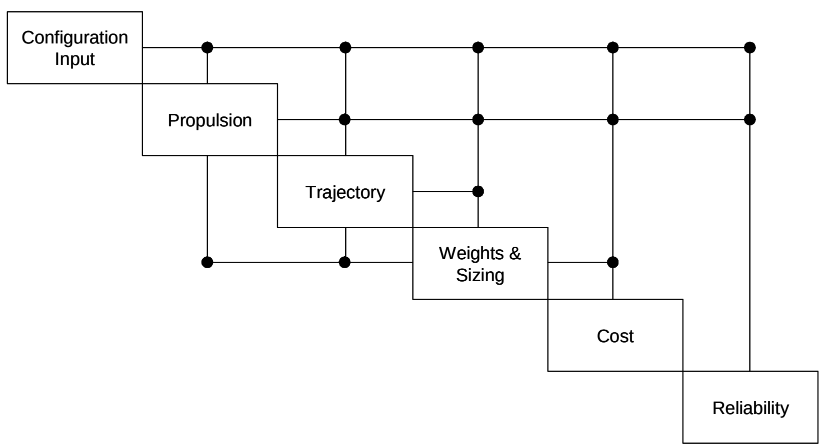

Figure 1. Lunar Lander Design Structure Matrix.

In a DSM, forward-fed information is shown using a link on the upper right of the disciplines; feedback is shown with a link on the bottom left of the disciplines. The disciplines should be ordered so as to minimize the amount of feedback between disciplines; this is to minimize the time required for convergence between the disciplines, and to reduce the possibility of mathematically divergent design. The tools used for each discipline are shown in Table 2.

Table 2. Disciplinary Tools.

Discipline	**Analysis Tool**	**Implementation in ROSETTA Model**
Configuration Inputs	Morphological Matrix Options	Inputs Page
Propulsion	REDTOP-2[2]	RSE of Standalone Program
Trajectory	POST[3]	RSE of Standalone Program
Weights & Sizing	Historical MERs[4]	MERs implemented in Excel
Cost	NAFCOM[5]	CERs implemented in Excel
Reliability	Fault Tree	Fault Tree implemented in Excel

‡ Half stages indicate a propellant drop tank is utilized.

In this lunar lander ROSETTA model, the trajectory and propulsion analyses are replaced with RSEs. These quadratic equations execute extremely quickly compared to the legacy analysis codes they replace. The form of a general 2nd order RSE is as shown in Eq. 1 and might appear as shown in Figure 2 for a RSE with two independent variables[6].

$$R = b_0 + \sum_{i=1}^{k} b_i x_i + \sum_{i=1}^{k} b_{ii} x_i^{\,2} + \sum_{i=1}^{k-1} \sum_{j=i+1}^{k} b_{ij} x_i x_j + \varepsilon \tag{1}$$

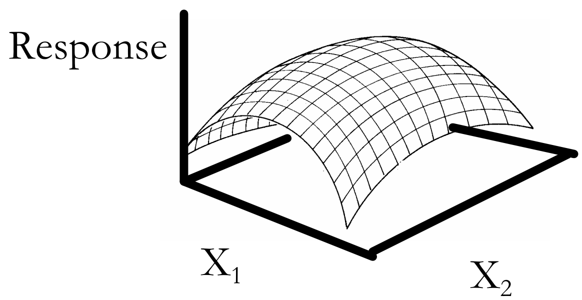

Figure 2. Quadratic Response Surface[6].

In order to create the RSE in an efficient manner, Design of Experiments (DOE) is used. The goal of DOE is to generate the maximum amount of information with the minimum expenditure of effort, where the effort in this case is the time needed for the runs of the disciplinary tools being approximated by the RSEs. A face-centered central composite design is selected to generate the runs of the disciplinary tools. The RSE is created by fitting a least squares quadratic model to the data. The quality of the RSE fit to the data is checked with the "goodness of fit" procedure, which includes testing for a random error and residuals distribution[6].

III. Disciplinary Analyses

A. Propulsion

The propulsion system is modeled using a response surface of the REDTOP-2 code[2]. When given inputs such as propellant type, mixture ratio, expansion ratio, and thrust chamber pressure, REDTOP-2 returns values of thrust, specific impulse, and engine weight. Examples for two potential engines for a lunar lander are shown in Table 3 and Table 4. The first is a pump fed liquid hydrogen and liquid oxygen engine using the expander cycle, similar to a RL-10. The second is a pressure fed hypergolic (MMH/NTO) engine, similar to that used in the Apollo program.

Table 3. LOX/LH_2 Engine Example Inputs/Outputs (RL-10 Derivative)[2].

Input Parameter	**Value**
Engine Cycle	Expander
Propellant Combination	LOX/LH_2
Thrust at 100% Throttle	99 kN
O/F Ratio	5.5
Expansion Ratio	84
Output Parameter	**Value**
Specific Impulse (Vacuum)	452 s
Engine Weight	173 kg
Engine Length	2.12 m

Table 4. MMH/NTO Engine Example Inputs/Outputs (TR-201 Derivative)[2].

Input Parameter	Value
Engine Cycle	Pressure Fed
Propellant Combination	MMH/ NTO
Thrust at 100% Throttle	44 kN
O/F Ratio	1.8
Expansion Ratio	50
Output Parameter	**Value**
Specific Impulse (Vacuum)	308 s
Engine Weight	137 kg
Engine Length	2.03 m

Using a face-centered central composite DOE, the function of the REDTOP-2 code is replaced by a RSE that, replicates the functionality of the engine design code within certain ranges of the independent variables. While REDTOP-2 takes several minutes to run, a RSE can be evaluated in less than a second. The morphological matrix for the propulsion ROSETTA model is shown in Table 5.

Table 5. Propulsion Analysis Morphological Matrix.

Discrete Variables	Option 1	Option 2	Option 3	Option 4	Option 5	Option 6
Propellant Combination	LOX/Kerosene	LOX/RP1	LOX/LH_2	LOX/Methane	MMH/NTO	UDMH/NTO
Cycle Type	Pressure Fed	Expander				
Continuous Variables	**Min**	**Max**				
Engine Isp Multiplier	95%	108%				
Engine T/W Multiplier	80%	110%				
Area Ratio	30	180				

B. Weights and Sizing

An important part of the performance sizing loop is the weights and sizing analysis. The inputs for this analysis come from the vehicle configuration, propulsion options, and trajectory. The propellant weights required for descent and ascent are fed into the weights and sizing spreadsheet, which then calculates the dry weight of the vehicle parametrically. This calculation is based on a database of mass estimating relationships from several sources[4,7]. These mass estimating relationships are typically parametric regressions of historical data, as shown in Figure 3.

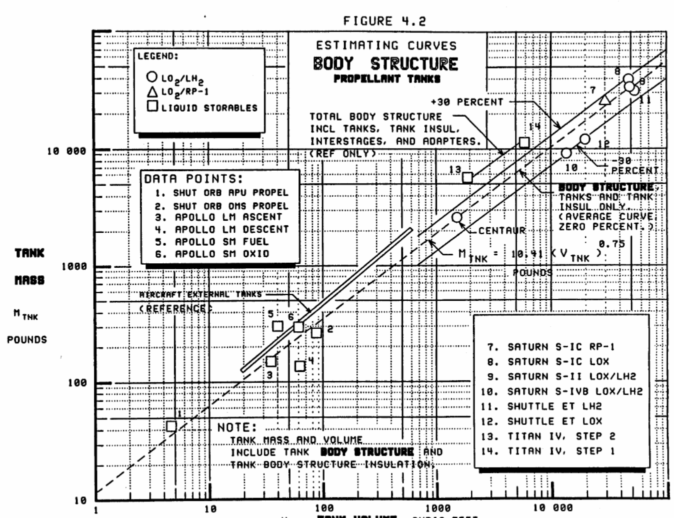

Figure 3. Regression of a MER from Historical Data[7].

To account for the advance of technology, or uncertainty in weight estimation, the user has the option to modify the value of each subsystem mass by a certain percentage. The user also has control over tank, habitat, and structure materials. Also included in the weight estimation is the dry weight margin. These options are shown in Table 6.

Table 6. Weight Estimation Morphological Matrix.

Discrete Variables	**Option 1**	**Option 2**	**Option 3**	**Option 4**	**Option 5**	**Option 6**
Tank Material	Al	Al-Li	Ti	Gr-Ep	Steel	MMC
Crew Compartment Material	Al	Al-Li	Ti	Gr-Ep	Steel	MMC
Structure Material	Al	Al-Li	Ti	Gr-Ep	Steel	MMC
Continuous Variables	**Min**	**Max**				
Primary Structure Weight Multiplier	0%	400%				
Secondary Structure Weight Multiplier	0%	400%				
Propellant Tank Weight Multiplier	0%	400%				
Pressurant Tank Weight Multiplier	0%	400%				
Crew Pressure Vessel Weight Mutliplier	0%	400%				
Landing Leg Weight Multiplier	0%	400%				
Dry Weight Margin Multiplier	0%	50%				

C. Trajectories

Trajectory simulation is performed in POST[3]. POST finds trajectories that originate from a set of initial conditions (position on the surface of a planetary body, or in orbit) and satisfy a set of final conditions. In this process, it finds the optimal path between the initial and final points and is capable of incorporating additional constraints, such as the maximum sensed acceleration and the maximum dynamic pressure. An example of an optimal versus non-optimal trajectory is shown in Figure 4.

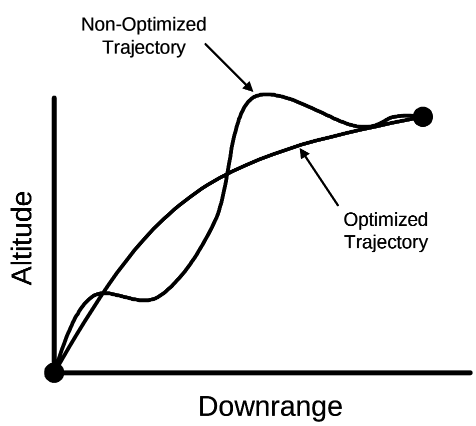

Figure 4. POST Trajectory Optimization.

Once an optimal trajectory is found, POST returns the overall flight delta-V, as well as the delta-Vs for gravity loss and thrust vector loss. These can be used to find the required propellant mass to fly the trajectory using Equations 2-4[7].

$$\Delta V_{total} = \Delta V_{flight} + \Delta V_{gravity} + \Delta V_{thrust} \tag{2}$$

$$\frac{M_{initial}}{M_{final}} = \exp\left(\frac{\Delta V_{total}}{g_0 Isp}\right) \tag{3}$$

$$M_{propellant} = M_{initial} - M_{final} \tag{4}$$

Ascent and descent trajectory simulations are performed separately. An initial guess is made of the stage dry weights; the trajectory is then run using a projected gradient optimization. The trajectory analysis passes the required mass of propellant to the weights and sizing code, which calculates the size of the tanks and structure required to hold that much propellant. New gross and dry weights are then passed back to the trajectory analysis. Because of this feedback loop, iteration is required to allow the performance of the vehicle to converge to a final solution (as is shown in Figure 1). The ascent trajectory optimization is constrained to meet the final conditions specified by the desired orbit at engine cut-off, and the desired mass at the end of the launch trajectory. The constraints on altitude, flight path angle, and inertial velocity place the ascent stage on an elliptical transfer orbit, which is chosen to match either Apollo or ESAS concept of operations for descent, ascent, and rendezvous based on user input.

A full factorial design of experiments is used to create a RSE of the trajectory analysis for inclusion into the reduced-order analysis. The independent variables are stage gross weight, thrust, and specific impulse. The response to model is the stage delta-v for an optimal trajectory between the initial and final conditions. The morphological matrix for the trajectory inputs are shown in Table 7.

Table 7. Trajectory Analysis Morphological Matrix.

Continuous Choices	Min	Max
Descent T/W in Lunar Orbit	1.5	3
Descent Contingency ΔV	0%	25%
Ascent T/W on Lunar Surface	1.5	3
Ascent Contingency ΔV	0%	25%

D. Cost Estimation

Cost estimation is performed within the ROSETTA model using relationships from NAFCOM[5]. NAFCOM uses Cost Estimation Relationships (CERs) that are based primarily on subsystem masses. CERs are similar to MERs in that they are based upon regressions of historical data. In addition to mass, NAFCOM considers complexity factor as a multiplier in many CERs. An example of a CER for the cost of a rocket engine is shown in Equation 5 where *CF* is the complexity factor, and *a* and *b* are constants derived from historical data. The data from which such a CER might be derived is shown in Figure 5.

$$EngineCost = CF \cdot a \cdot (Weight)^b \tag{5}$$

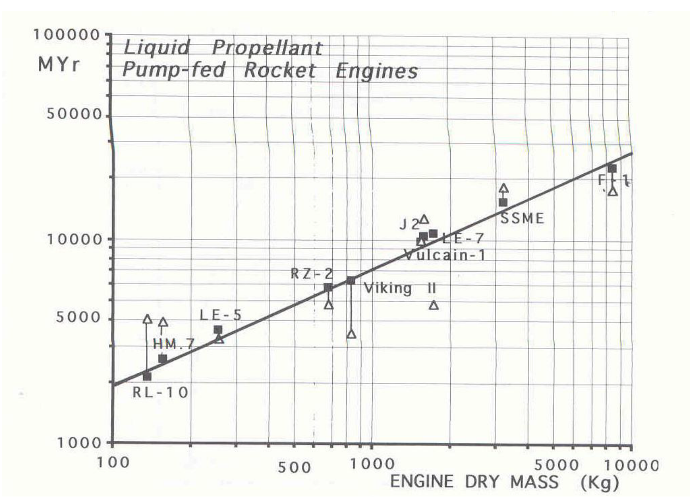

Figure 5. Regression of a CER from Historical Data[9].

NAFCOM is designed to calculate the non-recurring cost of an aerospace project. This can be broken into Design, Development, Testing, & Evaluation (DDT&E) and Theoretical First Unit (TFU) costs. Separate CERs are used for DDT&E and TFU, both of which are included in the lunar lander ROSETTA model.

E. Reliability

Lunar lander reliability is calculated using a fault tree analysis within the ROSETTA model. Subsystem reliabilities are based on historical reliabilities, with user-adjustable multiplicative factors to account for technological advances. Engine reliabilities are calculated in REDTOP-2 and are inputs into the model. They are based on historical reliabilities of the components within the engine and are dependent on the engine cycle selected.

IV. Sample Trades and Optimization

Because the lunar lander ROSETTA model is intended to allow the execution of rapid lunar architecture analysis and optimization, it is intended to model the lunar landers from the Apollo and ESAS architectures as closely as possible. The user can select between an Apollo and ESAS concept of operations with a top level input flag. For the purposes of architecture studies currently ongoing, the Earth Orbit Rendezvous/Lunar Orbit Rendezvous (EOR/LOR) architecture that ESAS uses is of interest. The ESAS architecture calls for separate launches of crew (the CEV) and cargo (the lunar lander) which rendezvous in earth orbit (the EOR component of the mission). After injection to the moon, the lunar lander descent stage inserts the vehicle into lunar orbit. The crew descends to the surface, performs their mission, and then returns to the CEV in the ascent stage of the lunar lander (the LOR component of the mission). The CEV then provides the propulsion to return to the Earth[1].

The nominal ESAS architecture calls for a crew of four astronauts to go to the surface. One interesting trade study is to examine the impact of changing the number of crew that the lunar lander can accommodate. In the event that the nominal four astronaut system cannot be realized due to cost or mass concerns, it is useful to quickly analyze the impact of reducing the crew complement. Conversely, some decision makers might want to know how much it would cost to "buy" an extra crew member or two onto the lander. Using the reference ESAS inputs (pump fed LOX/LH_2 descent stage, pressure fed MMH/NTO upper stage) the number of crew is varied from 2 to 6. The esults for dry weight, gross weight, development and unit cost, and vehicle reliability are shown in Table 8.

Table 8. Lunar Lander Sensitivity to Number of Crew.

Number of Crew	Dry Weight	Gross Weight	DDT&E ($M FY04)	TFU ($M FY04)	LOM every X flights	LOC every X flights
2	9,287 lb	38,961 lb	$10,567	$1,114	29	344
3	10,186 lb	42,184 lb	$11,093	$1,179	27	154
4	*11,041 lb*	*45,253 lb*	*$11,568*	*$1,237*	*24*	*99*
5	11,914 lb	48,392 lb	$12,041	$1,297	22	73
6	12,495 lb	51,508 lb	$12,495	$1,354	21	58

The results show the potential mass, cost, and reliability savings of reducing the number of crew from the baseline of 4 (shown in italics). It may also be important to note that in adding an additional two crew, which may be desirable in later long term lunar campaigns, will add over three tons to the gross weight of the vehicle, add a billion dollars to the development costs, and reduce the mission safety and reliability by a significant amount.

Another important trade study that can be performed is the selection of a propulsion system. The ascent stage propulsion system has a very high impact on the mass of the rest of the lunar architecture because it is carried along for so much of the mission. As such, every other stage must be sized to accommodate its mass, including the lunar descent stage, the Earth Departure Stage (EDS), and the first stage of the Cargo Launch Vehicle (CaLV, also known as the Ares V). Using the lunar lander ROSETTA model, a trade can be performed of the upper stage propulsion system in just a few minutes. The results are shown in Table 9.

Table 9. Lunar Lander Ascent Stage Trade Study.

Ascent Stage Propellants	Ascent Stage Engine Cycle	Dry Weight	Gross Weight	DDT&E ($M FY04)	TFU ($M FY04)	LOM every X Flights	LOC Every X Flights
LOX/Kerosene	Pump Fed Expander	11,156 lb	45,537 lb	$10,663	$1,115	24	92
LOX/Hydrogen	Pump Fed Expander	10,804 lb	41,325 lb	$11,109	$1,114	24	89
LOX/Methane	Pump Fed Expander	10,629 lb	42,470 lb	$10,643	$1,093	24	92
MMH/NTO	Pressure Fed	11,041 lb	45,253 lb	$11,568	$1,237	24	99

As shown above, the solution that minimizes the gross weight of the lunar lander is the LOX/Hydrogen system, although it is noticeably less reliable than the pressure fed hypergolic (MMH/NTO) system, which is about two tons heavier, and thus more expensive, as the NAFCOM CERs are primarily weight based.

The lunar lander ROSETTA is also useful for performing optimization. Using the ESAS requirements (four crew to the surface, on a two stage lunar lander), six separate optimizations are performed using the genetic algorithm: minimize gross weight, minimize dry weight, minimize DDTE&E cost, minimize TFU cost, maximize flights before LOM, and maximize flights before LOC. The independent variables for each optimization are:

- Tank Material
- Crew Compartment Material
- Structural Element Material
- Ascent Fuel Type
- Ascent Engine Cycle
- Ascent Engine Area Ratio
- Descent Fuel Type
- Descent Engine Cycle
- Descent Engine Area Ratio
- Ascent T/W at Lunar Surface
- Descent T/W in Lunar Orbit
- Ascent Engine Out
- Descent Engine Out

The results are quite interesting, and show that each objective function produces a unique vehicle. The optimization results are shown in Table 10.

Table 10. Single Objective Optimization Results.

Variable	Min Gross Weight	Min Dry Weight	Min DDT&E	Min TFU	Max LOM	Max LOC
Tank Material	4	4	5	5	4	4
Crew Compartment Material	4	4	4	4	4	4
Structure Material	4	4	4	4	4	4
Ascent Fuel Type	3	6	6	6	6	6
Ascent Engine Cycle	2	2	2	2	1	1
Ascent Engine Area Ratio	146.79	177.89	31.69	179.72	46.06	46.06
Descent Fuel Type	3	6	6	6	6	6
Descent Engine Cycle	2	2	2	2	1	1
Descent Engine Area Ratio	162.47	30.83	30.83	179.09	115.18	115.18
Ascent T/W at Lunar Surface	2.47	1.52	1.50	1.50	1.50	1.50
Descent T/W in Lunar Orbit	1.51	1.51	1.50	1.50	1.50	1.50
Ascent Engine Out	0	0	0	0	2	2
Descent Engine Out	0	0	0	0	2	2
Gross Weight	37,427 lb	46,001 lb	49,834 lb	46,268 lb	63,102 lb	63,102 lb
Dry Weight	9,401 lb	7,340 lb	8,285 lb	8,210 lb	11,234 lb	11,234 lb
DDT&E ($M FY04)	$10,940	$8,672	$8,242	$8,274	$12,097	$12,097
TFU ($M FY04)	$1,071	$964	$943	$941	$1,322	$1,322
LOM	24	25	25	25	26	26
LOC	89	104	104	104	136	136

Minimizing gross weight produces a vehicle that uses LOX/LH_2 on both the ascent and descent stage, because it is the propellant combination that yields the highest specific impulse. For a constant delta-V requirement, a higher specific impulse uses less propellant. Other interesting results are the comparison between minimizing development and first unit costs. In either case, the dry mass should be reduced because the CERs are mainly weight based. However, minimizing TFU results in an area ratio as high as allowed by the model, and minimizing DDT&E results in an area ratio as low as allowable. This is because the larger an area ratio is, the more massive the engine is, and the more expensive it is to develop; since engine development is a major percentage of total development costs, the engine should be as small as possible for the minimum DDT&E case. However, engine production cost is relatively small compared to the rest of the vehicle, so the added performance gained by using a large engine nozzle makes up for the additional cost of producing a heavier engine.

Optimization can also be performed using some combination of cost, weight, and reliability objectives, combined into a single objective function through an OEC or a Utility Function. When this is done, the user should be careful to pick the weighting factors on each separate objective so that they accurately his opinions about the comparative importance of each.

V. Conclusions

Because of the large size of the trade space for the analysis of exploration architectures, picking the best configuration of each element can be difficult. Traditional design methodologies have relied on comparative methods that are subjective, and as such are sensitive to the biases of the decision makers. Through the use of a ROSETTA model in the conceptual stage of design, the configuration of each element can be chosen so that it meets a set of performance and programmatic requirements. An optimized vehicle design can rapidly be produced, and compared to various competing designs to show the differences in mass, cost, and reliability.

The most important advantage of using a ROSETTA model is the speed in which it executes. This added speed enables quick optimization and modeling of uncertainty that would otherwise take hours to days to complete. Through the use of ROSETTA meta-models and probabilistic optimization techniques, decision makers can make the element configuration choices early in the conceptual design process that meet the mission requirements while being robust to uncertainty. Additional capabilities include the ability to model the relative impacts of technology investment decisions on vehicle mass, cost, and reliability. Future work will include mapping specific technologies to the multiplicative factors on subsystem mass, cost, and reliability, so the performance advantages of technological advances can be traded against the cost and risk of developing new systems.

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