AIAA SPACE 2007 Conference & Exposition
18 - 20 September 2007, Long Beach, California

AIAA 2007-9920

An Approach for Calculating the Cost of Launch Vehicle Reliability

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The goal of this paper is to determine the cost of increasing launch vehicle reliability during conceptual design. The launch vehicle mission requirements are held constant while various reliability strategies are evaluated for their affects on different performance and cost metrics. Traditional design disciplines, such as trajectory analysis and propulsion are included within the performance analysis while the cost discipline focuses on launch vehicle development and production cost. The reliability modeling is developed specifically for application to launch vehicles. A design environment is created that integrates the performance, cost, and reliability disciplines for use with optimization. The integrated environment is utilized to determine a set of optimal design configurations based on a specific weighting of cost and reliability. Different design options for the Cargo Launch Vehicle from the Exploration System Architecture Study are considered and the final result is a set of configurations optimized for a particular weighting of cost and reliability.

Nomenclature

APU	Auxiliary Power Unit.
CaLV	Cargo Launch Vehicle.
CCF	Common Cause Failure.
CER	Cost Estimating Relationship.
DDT&E	Design, Development, Testing, and Evaluation.
ESAS	Exploration System Architecture Study.
FTA	Fault Tree Analysis.
LCC	Life Cycle Cost.
MCS	Monte Carlo Simulation.
MER	Mass Estimating Relationship.
MFBF	Mean Flights Between Failure.
MGL	Multiple Greek Letter.
MTTF	Mean Time To Failure.
NAFCOM	NASA/Air Force Costing Model.
NASA	National Aeronautics and Space Administration.
POST	Program to Optimize Simulated Trajectories.
REDTOP-2	Rocket Engine Design Tool for Optimal Performance - 2.
RSE	Response Surface Equation.
SSME	Space Shuttle Main Engine.
SRB	Solid Rocket Booster.
TFU	Theoretical First Unit.

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I. Introduction

This paper describes a process that can be used to quantify the cost of increasing launch vehicle reliability from a baseline concept during the conceptual design phase. The methodology was developed by the author in his Ph.D. dissertation.[1] The results of applying the methodology to the Cargo Launch Vehicle show how different design options can influence the launch vehicle's performance, cost, and reliability metrics.

Reliability is defined as "the probability that an item (component, subsystem, system) will perform a required function under stated conditions for a stated period of time."[2] The reliability of a launch vehicle is the probability that the vehicle will complete its mission successfully. Another metric for evaluating system reliability is the Mean Flights Between Failure (MFBF), which estimates an average number of flights between launch vehicle failures

The different design options under consideration for increasing launch vehicle reliability include: (1) adding a redundant subsystem, (2) reducing the number of engines, (3) adding engine out capability, and (4) decreasing the operating time. Engine out capability is defined as a launch vehicle with the capacity to sustain a benign engine failure and still complete the mission. Decreasing the operating time is accomplished by increasing the vehicle's acceleration and assuming an exponential failure distribution, which will be discussed later in this paper.

The Life-Cycle Cost (LCC) of a launch vehicle is defined by the National Aeronautics and Space Administration (NASA) systems engineering handbook[3] as "the total of the direct, indirect, recurring, nonrecurring, and other related costs incurred, or estimated to be incurred in the design, development, production, operation, maintenance, support, and retirement over its planned life span." In this paper, only the development and production costs are considered while the operations cost is addressed in the future work section.

The development cost is referred to as the Design, Development, Testing and Evaluation (DDT&E) cost. This cost encompasses all of the costs associated with launch vehicle design, prototype creation and all testing and evaluation within the program. The production cost is the cost associated with manufacturing the flight hardware. In the presented results, the Theoretical First Unit (TFU) is the cost of producing each launch vehicle; learning curve effects are not included.

The methodology begins with a baseline launch vehicle configuration created by a top-level system designer, such as the Cargo Launch Vehicle. The mission objectives and constraints of the launch vehicle are constant. The performance disciplines, such as trajectory analysis, propulsion, and mass estimation, are coupled to link cost and reliability. A new vehicle configuration is created by completing the performance analysis while incorporating the different design options. The corresponding system costs are calculated using the subsystem and component mass estimates. System reliability is evaluated for the current launch vehicle configuration. Uncertainty is included in both the cost and reliability disciplines to create confidence bounds. Optimization is applied in the design process to find the best combination of design variables for a weighted value function of cost and reliability. The Cargo Launch Vehicle results illustrate how configuration changes increase system reliability while also showing the resulting affects on performance and cost.

II. Background

Little work has been completed that performs launch vehicle design in an integrated environment while also including the cost and reliability disciplines. An integrated design environment has been used to find the minimum mass launch vehicle.[4] Additionally, studies have optimized a launch vehicle for its lowest subsystem cost[5] by integrating the performance and cost disciplines. A design environment that integrated the performance, cost, and reliability disciplines was created for reusable launch vehicles[6] during the Next Generation Launch Technology program; however, the reliability discipline was recognized for its lack of maturity and there is no mention of optimization capability. The methodology used in this paper builds upon the previous design environments by developing higher fidelity reliability modeling for integration with the performance and cost disciplines to perform integrated launch vehicle analysis.

The Exploration System Architecture Study (ESAS), which selected the architecture for accomplishing the mission objectives created in the Vision for Space Exploration,[7] used an integrated design process to create the vehicles. Launch vehicle concepts were created using"...parametric sizing and structural analysis..." while vehicle lift capability was determined with: "...the generation of three-Degrees-of-Freedom point-mass trajectory designs anchored by the sizing, structural, and subsystem assessment work. Output of the vehicle concept development work was forwarded to the operations, cost, and reliability/safety groups for use in their analysis."[8]

The goal of the process demonstrated in this paper is to augment a design effort such as ESAS by providing the capability to rapidly assess thousands of launch vehicle configurations while maintaining the same level of fidelity. While the present methodology focuses on cost and reliability, qualitative metrics for selected vehicles can still be evaluated once a set of optimized designs is determined. With additional development, other quantitative metrics such as safety and operations cost could be included in the design environment. Additional quantitative metrics are addressed in the future work section of this paper.

The application presented is the original Cargo Launch Vehicle (CaLV), which enables human missions to the Moon. While the current CaLV[9] is different from the ESAS CaLV,[8] the performance models for the ESAS CaLV can be verified to show validation of the performance models. Hence, the ESAS CaLV is referred to only as the CaLV for the remainder of this paper.

The CaLV is composed of the Solid Rocket Boosters (SRBs), the lower booster stage, and the previously mentioned Earth Departure Stage. The SRBs are 5-segment boosters derived from the Space Transportation System but with an additional segment. The booster, or core, stage uses five Space Shuttle Main Engines (SSMEs) from the Shuttle which are modified for use on the CaLV. The Earth Departure Stage uses two J-2S engines, which are a derivative engine of the J-2 used on the Saturn V. The J-2S was an engine in development during the Apollo era but the program was canceled before the J-2S had an opportunity to fly.[10] Like the original J-2 engine, the J-2S is a gas generator engine that uses liquid oxygen and liquid hydrogen as its propellants. An illustration of the CaLV is shown in Figure 1.

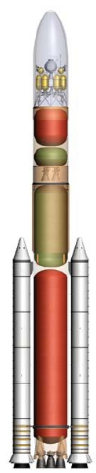

Figure 1: Cargo Launch Vehicle.[8]

Different design options are considered for each core stage of the CaLV while keeping the orbit and payload requirements constant. The SRBs are not altered in the present application. On each stage, the thrust-to-weight ratio, number of engines, and redundancy options are considered to increase CaLV reliability. The redundancy options include engine out capability and/or an additional subsystem. The resultant performance and cost affects are determined using the integrated design environment. The results illustrate a set of CaLV configurations that are optimized for a balance of cost and reliability.

III. Methodology

III.A. Performance Modeling

Integrating the performance models leads to vehicle characteristics, such as the subsystem masses and stage burn time, for use in the cost and reliability disciplines. Mission constraints, such as the payload and the final orbit, are held constant to ensure a fair comparison between different launch vehicle configurations. Additional mission constraints such as maximum dynamic pressure (max q) and maximum acceleration (max gs), are also held constant. These mission constraints are derived from the initial vehicle configuration, such as the baseline CaLV concept.

The mass estimates of a launch vehicle can be calculated using Mass Estimating Relationships (MER). An MER is an equation developed from regression of historical databases that uses vehicle characteristics to estimate the mass of a subsystem or component. Since a launch vehicle is a one of a kind system, there is some error when using MERs because of the reliance upon historical space systems to forecast future subsystem masses.

Another analysis area used to complete a launch vehicle design is the propulsion discipline. The propulsion model calculates the required engine characteristics based upon the design configuration. Using a conceptual design tool, engine characteristics such as the engine thrust, engine thrust-to-weight ratio, and Isp are determined.

A conceptual powerhead design code called Rocket Engine Design Tool for Optimal Performance - 2 (REDTOP-2)[11] is used for the analysis in the methodology. REDTOP-2 requires a user to set a number of design parameters such as the engine thrust level, chamber pressure, and cycle type to calculate the desired output variables. The output variables in this methodology are Isp and engine thrust-to-weight ratio which have excellent agreement with all relevant rockets that have been produced.

The trajectory analysis in this methodology is completed by using the Program to Optimize Simulated Trajectories (POST),[12] which is a three degree of freedom trajectory simulation widely used at NASA. The trajectory is solved by integrating the equations of motion over time for a specific set of mission objectives. POST can be used to find a trajectory that minimizes the amount of propellant required. The vehicle mass will decrease because the mass of the tanks and supporting structure is reduced when the propellant mass decreases. In addition, the burn time of each stage, which is an important parameter for the reliability discipline, is calculated with trajectory analysis.

The vehicle sizing process is completed by integrating the mass estimates from the MERs with the trajectory and propulsion disciplines. In this methodology, an aerodynamics analysis is completed based on the baseline launch vehicle configuration and it is assumed the aerodynamic results will not vary greatly as the configuration changes. Iteration is required within the trajectory, mass, and propulsion disciplines because the MERs cannot create mass estimates of the vehicle without the propellant mass, but the trajectory analysis requires a payload and dry mass for operation. The iteration process is referred to as "closing" a vehicle design and is illustrated in Figure 2.

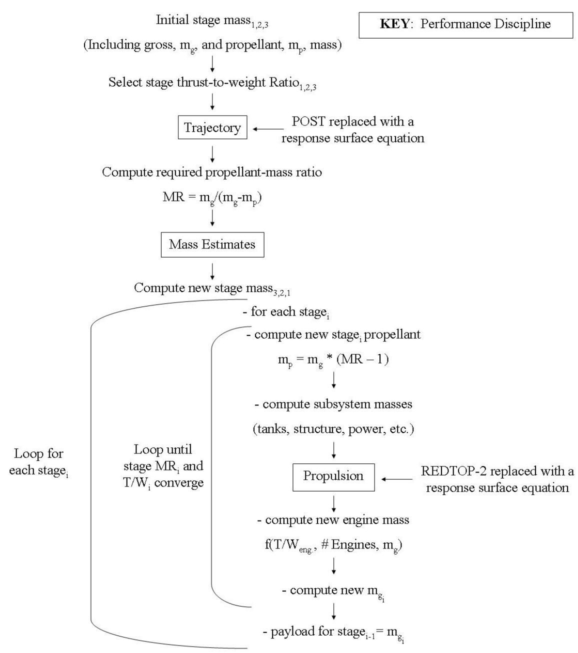

Figure 2: Closing a Launch Vehicle Design.

Figure 2 illustrates closing a three stage launch vehicle as completed in this methodology. A design engineer selects a stage thrust-to-weight ratio and uses trajectory analysis to determine the required mass ratio. The stage propellant mass is calculated from the required mass ratio; the supporting structure and subsystem masses are computed once the propellant mass is known. The required thrust is calculated from the stage thrust-to-weight ratio and leads to the engine thrust-to-weight ratio and engine mass. Iteration may occur because the stage thrust-to-weight ratio and stage mass ratio do not match their initial values.

Therefore, the subsystem masses are calculated in a loop that runs until the stage thrust-to-weight ratio and stage mass ratio equal their initial values.

The process illustrated in Figure 2 is one reason why Response Surface Equations (RSEs) are used in place of the trajectory and propulsion disciplines. RSEs are polynomial equations that speed up the design process[13] by replacing an analysis module. RSEs introduce error but techniques can be used to minimize the error introduced into the design process. Additionally, the time saved by using an RSE instead of an analysis module may be considerable.

To incorporate RSEs, a range of thrust-to-weight ratios for each stage of the CaLV was closed with POST and REDTOP-2 prior to creating an integrated environment. Therefore, the efficiency of closing a launch design, as shown in Figure 2, is improved because the intensive computational programs, such as POST and REDTOP-2, are removed from the design process.

Figure 3 is a design structure matrix that illustrates how the performance disciplines are integrated together to create a launch vehicle design. The black dots in Figure 3 indicate which parameters are used as inputs and outputs for each discipline. The design engineer controls the thrust-to-weight ratio of each stage, the number of engines per stage, and other reliability options that are discussed later in this paper. Based on the configuration selections, the subsystem masses of the launch vehicle are calculated with the process shown in Figure 3. The analysis modules in Figure 3 will be combined with the cost and reliability disciplines to create the integrated environment.

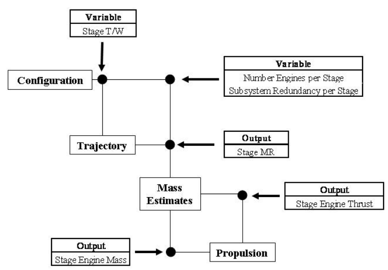

Figure 3: Design Structure Matrix for the Performance Disciplines.

III.B. Reliability Analysis

Reliability is calculated using the survivor function of a governing failure distribution. The survivor function calculates the probability of successful operation at any given time.[14] An example of a survivor function is the exponential distribution shown in Equation 1.

$$R_i(t) = e^{-\lambda_i t} \tag{1}$$

Equation 1 is the formula for calculating the reliability of a component or subsystem. In Equation 1, λ is the failure rate associated with the component and t is the time of operation. The units of the failure rate are failures per unit time. The unreliability of a system is equal to one minus the reliability.

The inverse of the failure rate is the Mean Time To Failure (MTTF), which is an average measure of the amount of time that will elapse before a component fails. For launch vehicles, MTTF is equivalent to MFBF when the exponential distribution is used and the units of MTTF are failures per mission.

The subsystem and component reliability estimates for the CaLV are calculated using the exponential distribution. These estimates are the bottom events in the Fault Tree Analysis (FTA) which calculates system unreliability. FTA is a common reliability analysis technique used to determine the probability of a top level event such as loss of mission,[15].[3]

The FTA will dynamically update as the configuration changes. One option for increasing vehicle reliability is to add a redundant subsystem. For the CaLV application, another power and/or avionics subsystem can be added to each core stage to increase system reliability. The fault tree updates with these configuration changes and a new system reliability value is calculated. Two different options for subsystem redundancy are evaluated: using identical components and functional redundancy.

Identical component redundancy refers to adding an identical string of components to create a one out of two subsystem configuration. While it is assumed this type of redundancy is cheaper, identical components are susceptible to common cause failure.

III.B.1. *Common Cause Failure*

Common Cause Failures (CCFs) can bypass redundancy by causing identical components to fail because of something inherent in all of the components. One example of CCF could be a manufacturing defect that invaded all components in a particular batch.

Common cause failure can be included in a system reliability analysis by using the Multiple Greek Letter (MGL) method.[15] In the CaLV application, a reduced version of the MGL method known as the "Beta" (β) model is used. The full MGL model is shown in Equation 2.

$$Q_k = \frac{1}{\binom{m-1}{k-1}}(1 - rho_{k+1})(\prod_{i=1}^{k} \rho_i)Q_t \tag{2}$$

Only dual components (i.e. a one out of two system) are used for redundancy. This strategy was chosen because a triple redundant system (i.e. one out of three) is almost as likely to fail as a dual redundant system because of common cause failure.[16] Figure 4 illustrates why the benefit from using triple redundancy is minimal.[17]

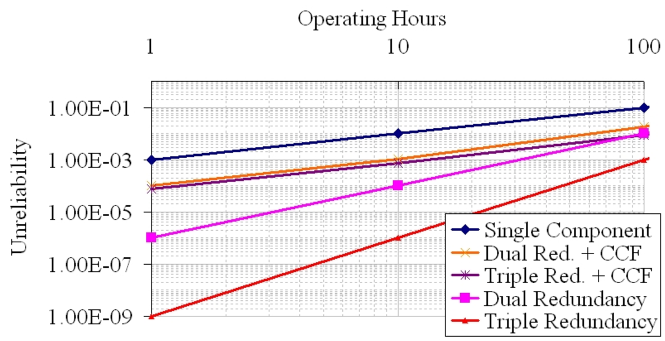

Figure 4: CCF Impacts of Different Component Configurations.[17]

The governing failure distribution in Figure 4 is the exponential distribution; the failure rate for a single component is 5.5E-5. The unreliability of a single component is illustrated by the top line in Figure 4. The bottom two lines, the pink and brown lines, show the benefits of redundancy without considering common cause failure. However, once CCF is considered, as represented by the middle two lines in Figure 4, the reliability benefits of using identical components in a parallel configuration are significantly reduced.

Figure 4 is heavily dependent on the common cause failure probabilities along with the component failure rate, but the graph exhibits the same characteristics until the failure rate for a single component becomes unreasonably high. One mention should be made about the STS-9 mission, which utilized a triple redundant Auxiliary Power Unit (APU) system. During the landing phase, two APUs caught on fire but the orbiter was able to land safely because it had one additional APU.[18]

III.B.2. *Functional Redundancy*

Another type of redundancy is functional redundancy, where two different systems are used to perform the same function. One example of functional redundancy is using batteries and fuel cells for the power subsystem when only batteries are needed. Another example is using two different manufacturers to produce two batteries when only one is required. A famous example of functional redundancy is the ill-fated Apollo 13 mission. When the CSM lost its power and oxygen due to an explosion in the oxygen tanks, the crew was able to use the Lunar Module for life support until re-entry.[19] Functional redundancy is not as susceptible to common cause failure since an inherent flaw in one component may not exist in a separate component. Therefore, functional redundancy will generally provide a higher system reliability benefit compared to using identical components in a parallel configuration.

Table 1 lists a reliability comparison for a one out of two subsystem configuration for the GNC subsystem on the S-II stage. Row one lists the subsystem reliability when functional redundancy is used for full subsystem redundancy. Row two lists the subsystem reliability when two identical components are arranged in a one out of two redundancy configuration. The reliability is higher with functional redundancy because it is assumed that common cause failure does not exist. The mass of each subsystem is also affected when either type of full subsystem redundancy is selected.

Table 1: S-II GNC Redundancy Comparison.

Redundancy Type	**Reliability**	**MFBF**
None	0.958	24
Identical Components	0.994	176
Functional	0.998	561

When subsystem redundancy is used for a subsystem, the mass of the subsystem is doubled. For example, if subsystem redundancy is selected for the power subsystem, the mass of the power subsystem is doubled. This assumption is accurate when two identical sets of components are used to provide full subsystem redundancy, but, when using functionally redundant subsystems, there may be some error in this assumption.

III.B.3. Engine Out Modeling

Engine out capability is another form of redundancy. However, providing complete engine redundancy is typically infeasible; therefore, engine out capability is employed to increase the propulsion subsystem reliability. Due to the failure modes of an engine, engine out capability requires additional modeling.

For this methodology, engine out modeling builds upon an 'm' out of 'n' fault tree with catastrophic failure modes that was proposed by Huang and et al.[20] Huang and et al.[20] propose the percentage of catastrophic failures is between 20 and 40 percent based on historical data. In the original model, common cause failure was not a consideration; the present model includes CCF with the Beta (β) model[15] based upon the assumptions stated earlier. A fault tree representation of this new catastrophic failure model is shown in Figure 5.

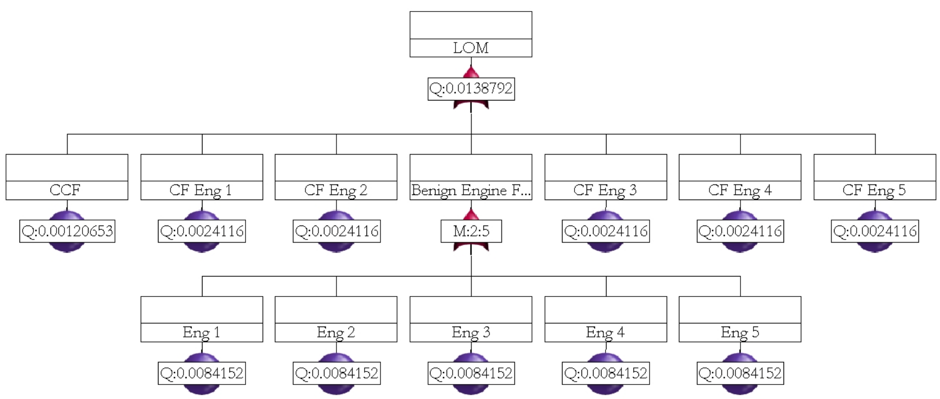

Figure 5: Engine Out Failure Model.

The top calculation in Figure 5 is the unreliability of a five engine configuration with engine out. The model is a propulsion subsystem with a 20 percent catastrophic failure percentage and 10 percent β value for CCF. The middle 'OR' gate signifies a benign engine failure mode where at least two engines must fail before the system will fail. On the same level as the benign engine failure mode are the failure modes that cause immediate system failure: a common cause failure and the catastrophic failure probability for each engine in the configuration.

III.B.4. System Reliability Calculation

The subsystem and component level reliability estimates are combined into their respective stages and the stages are combined into the system level reliability estimate using fault tree analysis. An example of the EDS stage fault tree is shown in Figure 6.

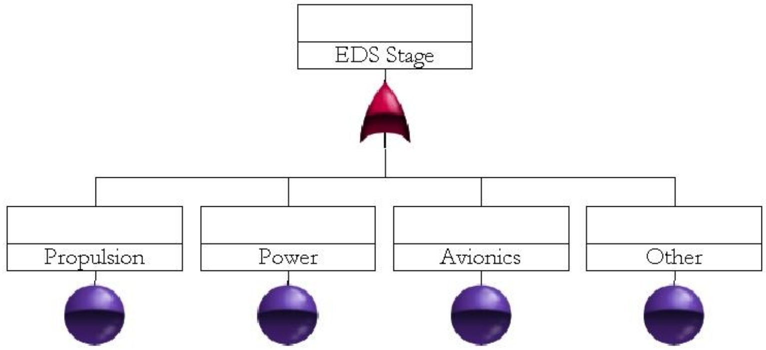

Figure 6: EDS Fault Tree Representation with No Subsystem Redundancy.

The CaLV system reliability is calculated using the reliability estimates for each stage in a series configuration, as illustrated by the fault tree in Figure 7.

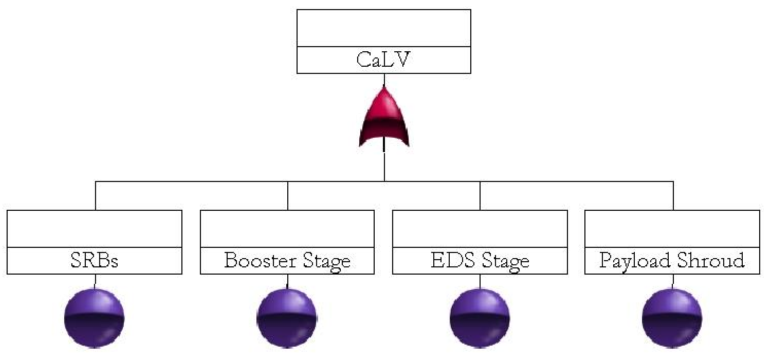

Figure 7: CaLV Reliability Calculation.

III.B.5. CaLV Subsystem Reliability Estimates

Table 2: CaLV Baseline Reliability Estimates.

Booster	**Reliability**	**MFBF**
SSME	0.9935	154
RSRB	0.9993	1370
Shroud	0.9997	3096
Power	0.9998	6250
Other	0.9996	2500
Total	**0.992**	**124**
EDS	**Reliability**	**MFBF**
J-2S	0.996	250
Power	0.995	200
Avionics	0.999	1000
Other	0.9999	10000
Total	**0.989**	**86**

The baseline CaLV subsystem reliability estimates are listed in Table 2. The reliability estimates for the booster stage are from ESAS.[8] The EDS subsystem reliability estimates are determined from examination of historical references[21,22,23,24,25] and engineering judgment.

III.B.6. *Reliability Uncertainty Analysis*

Uncertainty analysis is included with Monte Carlo Simulation (MCS). Triangular distributions are used in place of single failure rates for each individual reliability calculation. The ranges for each reliability estimate are based on historical failure rates from various references[21,22,23,24,25] and engineering judgment.

The final system reliability estimate is the 70 percent certainty value from the Monte Carlo Simulation. The 70 percent certainty value denotes that 70 percent of the values generated will be above this value. The 70 percent certainty value is an assumption and can be modified in this methodology. The system reliability is one of two system metrics required for the optimization scheme.

III.C. Cost Estimation

The cost tool used in this methodology is the NASA/Air Force Costing Model (NAFCOM). NAFCOM uses Cost Estimating Relationships (CERs) to calculate cost by using dry mass as the primary independent variable.[26] NAFCOM also allows the user to select analogous systems to tailor the CER based upon the similarity between the current vehicle design and historical space systems. For example, a user may decide to include only the vehicles from the Apollo era for their cost estimates.

Once the performance disciplines have calculated the subsystem masses of the launch vehicle, the CERs from NAFCOM are used to calculate the DDT&E and production cost. The summation of these subsystem costs results in the vehicle DDT&E and Theoretical First Unit (TFU) cost. The total production cost is calculated by multiplying the number of flights by the TFU cost.

Equation 3 is the formula for calculating the cost metric. The number of flights is equal to twelve for the CaLV application.

$$CostMetric = DDT\&E + No.Flights * TFU \tag{3}$$

The operations and support costs along with the disposal costs are not included within the cost metric for this methodology. It is assumed that the disposal costs are not greatly affected by changing the configuration of a launch vehicle. The operations costs for launch vehicles are difficult to determine because the private space industry guards this data closely.

III.C.1. *Cost and Redundancy*

The cost model treats the subsystem redundancy strategies differently. Table 3 is a notional example that compares the cost of using a one out of two configuration to provide full subsystem redundancy. When a launch vehicle configuration does not include subsystem redundancy, only the baseline subsystem cost estimates are included in the system cost. Using two identical sets of components to provide redundancy is referred to as identical components in Table 3. When two identical sets of components are used to provide full subsystem redundancy, no additional DDT&E cost is required for developing an identical second unit. However, the production cost will double because two units are needed.

Table 3: Full Subsystem Redundancy Notional Cost Comparison.

Redundancy Type	**DDT&E [$M FY '04]**	**Total TFU [$M FY '04]**
None	1	1
Identical Components	1	2
Functional	2	2

When functional redundancy is used, two different sets of components are created; therefore, the DDT&E cost of each set of components must be included in the cost estimate, as listed in Table 3. For example, batteries from two different manufacturers are developed and thus the DDT&E of each battery must be included. It is assumed in this methodology that because the batteries are required to perform the same task, the development cost of two subsystems is equal to double the DDT&E cost of a single subsystem, as listed in Table 3. Furthermore, the production cost for both strings is assumed to equal two times the single subsystem production cost.

III.C.2. Cost Uncertainty Analysis

NAFCOM has the capability of including uncertainty analysis within its cost analysis. Both the mass estimates and CERs are based upon regression of historical databases of space vehicles and error exists because historical data is used to forecast the masses of a new vehicle. Therefore, margins are used on the mass estimates to account for the MER errors while a distribution is assumed for the CERs to incorporate the cost model uncertainty.

To build a distribution around the CER, a minimum, maximum, and most likely mass estimate are required. In the application of the methodology to the Cargo Launch Vehicle, the minimum value is the original subsystem mass estimate from the MER. The most likely estimate adds a 15 percent margin to the baseline while the maximum estimate is a 30 percent margin on the original subsystem mass calculation. Figure 8 illustrates the mass margin assumption.

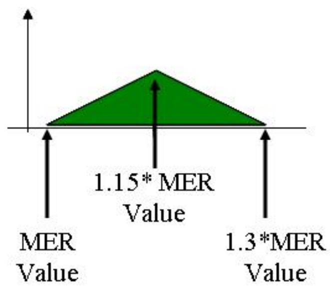

Figure 8: Mass Margin.

Using these three mass estimates, a distribution is built around each subsystem CER. Based on NAFCOM, each subsystem CER uses a lognormal distribution for the uncertainty analysis. The standard deviation is calculated from the range of subsystem masses and then multiplied by a factor to account for the specific CERs used in the cost analysis. The standard deviation multiplier depends upon the NAFCOM analogies selected and will be constant as long as the analogies do not change. Figure 9 illustrates the complete process for using uncertainty with the cost analysis.

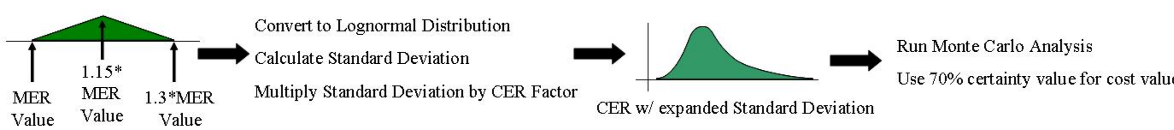

Figure 9: Cost Uncertainty Analysis.

For the single value needed in the objective function for optimization, a 70 percent certainty value is used. For the cost discipline, 70 percent of the calculated costs will be lower than this value. The 70 percent estimate is an assumption and can be changed easily depending upon the engineer's preference. The results will also show the range of cost and reliability estimates based upon the 10 and 90 percent confidence bounds.

III.D. Integrated Model

Once the performance, cost, and reliability models have been created, they can be integrated into one environment to study how cost and reliability are coupled for launch vehicles. A set of design variables will be used to change the launch vehicle configuration and examine the sensitivity of the design variables on the performance, reliability and cost metrics.

The specific design variables used in the CaLV application are listed in Table 4. The only subsystems given the option of full subsystem redundancy (i.e. a one out of two subsystem configuration) are the avionics and power subsystems.

Table 4: CaLV Design Variables.

Design Variable	**Minimum**	**Maximum**	**Type**
CaLV T/W Ratio	1.38	1.5	Continuous
EDS T/W Ratio	0.4	1.1	Continuous
No. of Engines Boos.	2	7	Discrete
No. of Engines EDS	1	4	Discrete
Engine Out Boos.	Yes	No	Discrete
Engine Out EDS	Yes	No	Discrete
Full Power Redundancy Boos.	Yes	No	Discrete
Full Avionics Redundancy Boos.	Yes	No	Discrete
Full Power Redundancy EDS	Yes	No	Discrete
Full Avionics Redundancy EDS	Yes	No	Discrete
Power Redundancy Type Boos.	Identical Components	Functional	Discrete
Avionics Redundancy Type Boos.	Identical Components	Functional	Discrete
Power Redundancy Type EDS	Identical Components	Functional	Discrete
Avionics Redundancy Type EDS	Identical Components	Functional	Discrete

Figure 10 is the design structure matrix for the CaLV created by using this methodology. The discipline tools are listed in the lower right of Figure 10. A genetic algorithm is used to find the optimum combination of design variables for a given weighting of system reliability and cost.

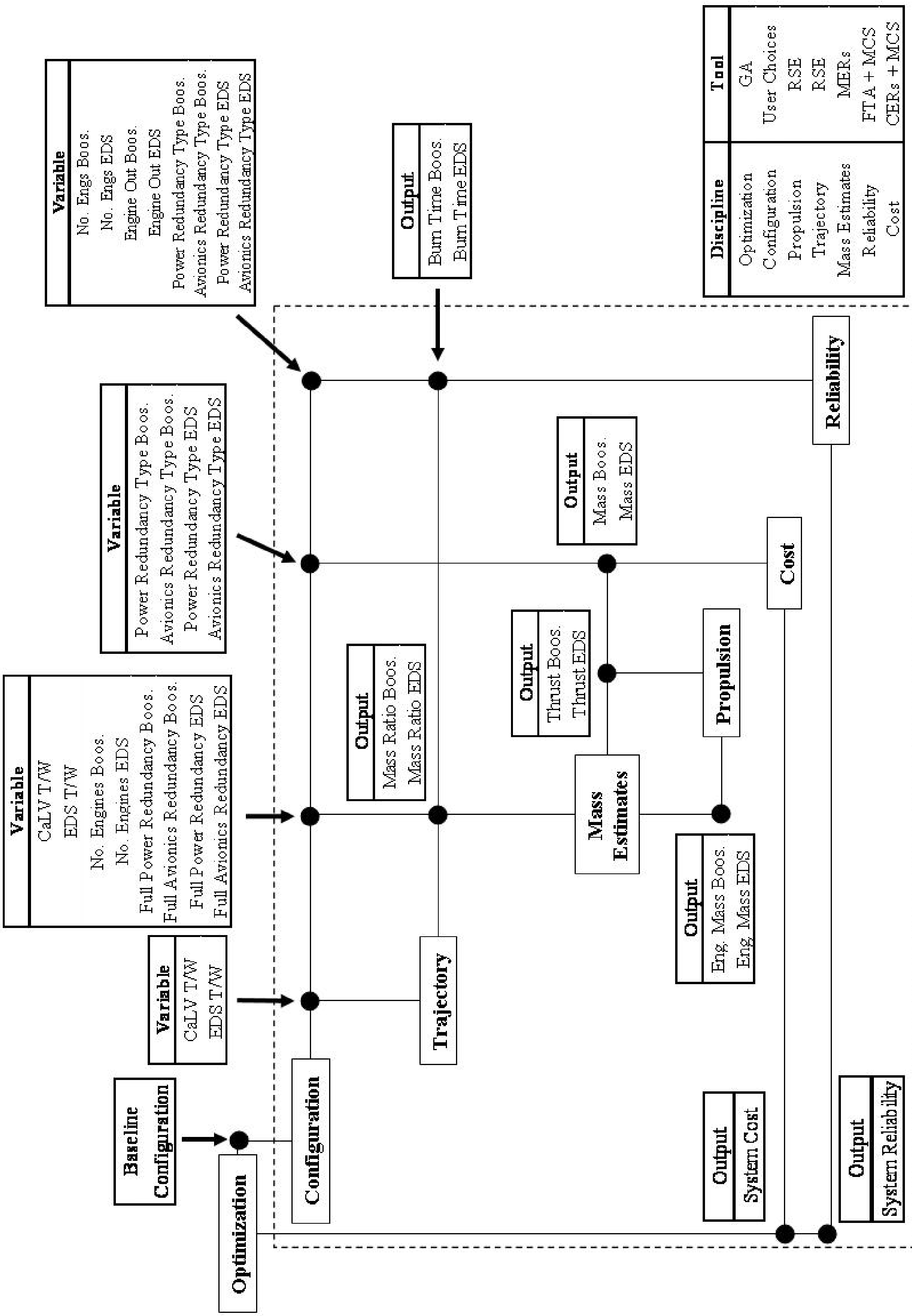

Figure 10: CaLV Design Structure Matrix.

The integrated design environment shown in Figure 10 is used to create the results in the next section. The genetic algorithm optimizes the value function is listed in Equation 4. A set of optimal configurations is created by varying the weighting in Equation 4.

$$OEC = W_R \frac{R_{calc}}{R_{max}} + (1 - W_R)\frac{C_{min}}{C_{calc}} \tag{4}$$

IV. CaLV Results

IV.A. CaLV Model Validation

When modeling a system, validation is important. The development of this methodology required validation due to the performance modeling and the use of response surface equations. The meta-models must be accurate so a design engineer has confidence in using the response surface equations to replace the higher fidelity analysis. The validation presented in this paper compares the numerical results of the methodology with numerical values from ESAS .[8]

The performance models are built upon the work completed by Young and et al.[27] The mass validation is listed in Table 5. There is good agreement between the CaLV model created by Young and et al.[27] and the model used in ESAS.[8]

Table 5: CaLV Mass Comparison Between ESAS and Young [lb].

Booster	**ESAS**	**MERs**	**% Difference**
Dry Mass	194997	194563	0.22
Gross Mass	2428061	2442803	0.61
EDS	**ESAS**	**MERs**	**% Difference**
Dry Mass	42645	42528	0.27
Gross Mass	640171	650816	1.66
CaLV	**ESAS**	**MERs**	**% Difference**
Gross Mass	6393975	6408445	0.23

The trajectory models were also built upon the work completed by Young and et al.[27] POST is used for the trajectory optimization of the CaLV and the results are compared to ESAS.[8] Table 6lists the requirements and assumptions for the trajectory analysis.

Table 6: CaLV POST Assumptions.

Parameter	**Value**
Max. Q [psf]	620
Max. Accel [gs]	3.85
TLI δV [ft/s]	10334
Final Velocity [ft/s]	25707
Perigee Altitude [nmi]	16.2
Perigee Altitude [nmi]	86.4
Lunar Payload [lb]	160112

Figure 11 shows a comparison between the trajectory curve from ESAS[8] and the trajectory curve produced by Young and et al.[27] There is a little error between the curves but the trends and slopes are similar.

The reliability models in the present application are validated with the commercial software RELEX.[28] For any calculations in series, the calculation is simple and no validation is shown; the analysis in the integrated model was cross-checked by hand to ensure the validity of the calculation. When using subsystem redundancy (i.e. a one out two subsystem redundancy configuration) with identical components, common

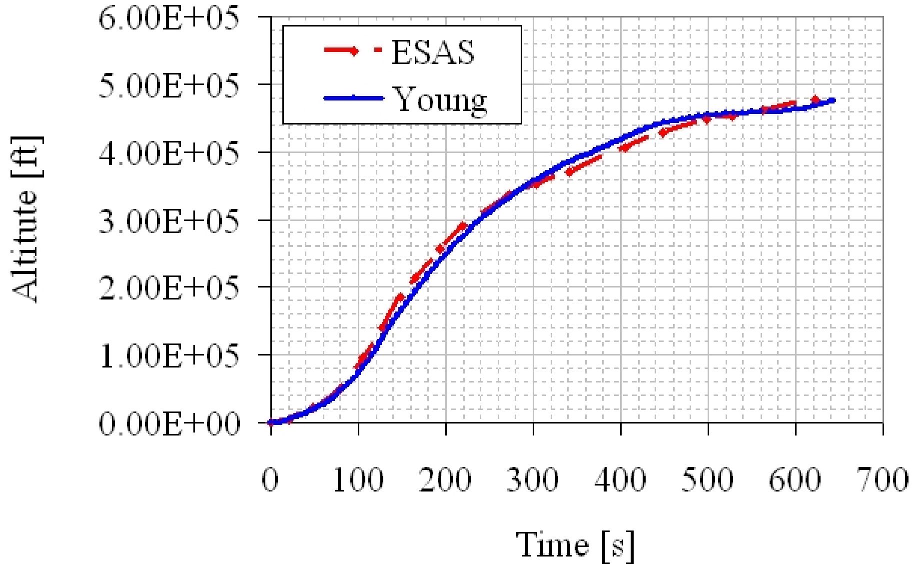

Figure 11: Trajectory Comparison for CaLV.

cause failure is included in the calculations; the validation of this redundancy model is listed in Table 7. The β value is equal to 0.1 and the component reliability is 0.98.

Table 7: Reliability Validation for Component Redundancy.

Calculation	**RELEX**	**Model**	**% Difference**
Reliability	0.9977	0.9977	0.00
MFBF	430.41	430.29	0.03

Since the cost tool is NAFCOM, no validation is required to compare the CERs with the original tool.

IV.B. CaLV RSE Validation

The Cargo Launch Vehicle application also relies upon response surface equations. The propulsion RSE calculates the J-2S engine thrust-to-weight ratio based upon the J-2S thrust. Table 8 lists all of the RSE statistics, such as R^2 and $R^2 - adjusted$, for the propulsion RSE.

Table 8: CaLV Engine T/W Ratio RSE Data.

Indpt. Var.	**J-2S Thrust**
Minimum	**Maximum**
100000	500000
Fit	**Statistics**
R^2	0.968
$R^2 - adj$	0.963

The RSE fit statistics for the mass ratio and burn time are listed in Table 9 through Table 12. The statistics show good agreement between the polynomial equations and the POST software.

Table 9: CaLV Booster Mass Ratio RSE Data.

Indpt. Var.	**CaLV T/W**
Minimum	**Maximum**
1.38	1.5
Fit	**Statistics**
R^2	0.999
$R^2 - adj$	0.998

Table 10: CaLV Booster Burn Time RSE Data.

Indpt. Var.	**CaLV T/W**
Minimum	**Maximum**
1.38	1.5
Fit	**Statistics**
R^2	0.995
$R^2 - adj$	0.993

Table 11: CaLV EDS Mass Ratio RSE Data.

Indpt. Var.	**EDS T/W**
Minimum	**Maximum**
0.4	1.1
Fit	**Statistics**
R^2	0.994
$R^2 - adj$	0.991

Table 12: CaLV EDS Burn Time RSE Data.

Indpt. Var.	**EDS T/W**
Minimum	**Maximum**
0.4	1.1
Fit	**Statistics**
R^2	0.997
$R^2 - adj$	0.996

IV.C. CaLV Results

The Cargo Launch Vehicle results are illustrated in Figure 12. One reason the final CaLV design was selected in ESAS was because it used existing engines. Since the true costs of the ESAS CaLV have not been published, a series of data points are created to represent the ESAS CaLV based upon a percentage of predicted engine DDT&E cost.

In Figure 12, the left most data point is the ESAS CaLV design that does not incur any engine DDT&E costs from modifying the SSME and J-2S. The engine DDT&E cost percentage is applied to the engines on both the booster and Earth Departure Stage. Five different cost values are presented for the ESAS CaLV design in Figure 12 which represent increasing the engine DDT&E cost by 25 percent. The right most data point in the series of ESAS CaLV data points represents a CaLV design that incurs 100 percent of the predicted engine DDT&E cost.

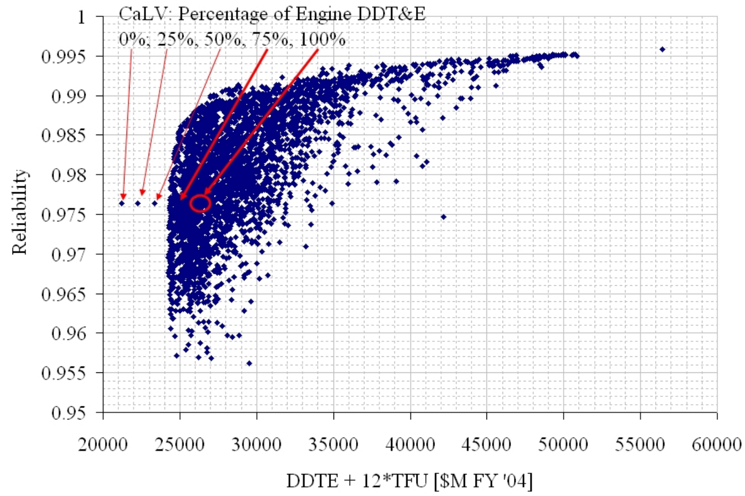

Figure 12: CaLV Design Space.

Figure 12 illustrates that if the full engine DDT&E cost is paid, then the CaLV is a sub-optimal design for the two metrics used in this methodology. In other words, there are different configurations of the CaLV that will result in a lower cost launch vehicle or a design with higher reliability for the same cost. The break-even point is at the 75% cost level for engine DDT&E; if greater than 75% of the engine DDT&E costs are incurred, a different CaLV configuration should be selected.

Figure 13 illustrates the pareto frontier for the Cargo Launch Vehicle. The data points in Figure 13 represent the maximum reliability configurations for a particular cost value. The CaLV from ESAS is shown by the series of five data points representing the percentage of engine DDT&E cost incurred in the design. Figure 13 reinforces that a different CaLV design may be more appropriate if the engine modification costs are greater than 75 percent of the predicted DDT&E cost for a new engine with identical characteristics.

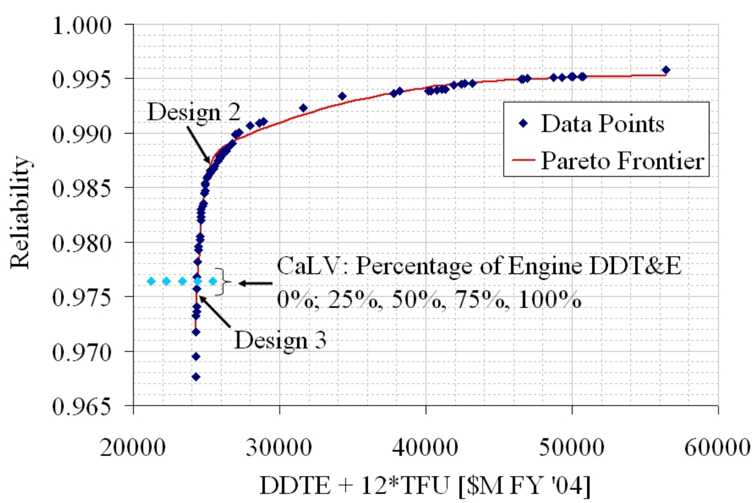

Figure 13: CaLV Pareto Frontier.

The performance, cost, and reliability metrics from four data points in Figure 13 are compared in Table 13. The data points are two baseline CaLV configurations (the minimum and maximum possible cost designs), a configuration with higher reliability (design two), and a design with lower cost compared to the CaLV maximum possible cost (design three).

Table 13: Metric Comparison for Four CaLV Designs.

Parameter	CaLV 0% Eng. DDT&E	CaLV 100% Eng. DDT&E	Design 2	Design 3
Cost [$M FY '04]	21232	25438	25560	24353
Reliability	0.9764	0.9764	0.9870	0.9757
MFBF	42	42	77	41
CaLV Gross Mass [lb.]	6393975	6393975	6352654	6443662
Booster Dry Mass [lb.]	194563	194563	197622	202261
EDS Dry Mass [lb.]	42645	42645	39172	36549
CaLV T/W Ratio	1.38	1.38	1.43	1.41
EDS T/W Ratio	0.84	0.84	0.73	0.54
No. of Engines Booster	5	5	2	4
Engine Out Booster	No	No	No	No
No. of Engines EDS	2	2	1	2
Engine Out EDS	No	No	No	No
Power Redundancy Boos.	None	None	None	None
Avionics Redundancy Boos.	None	None	None	None
Power Redundancy EDS	None	None	None	None
Avionics Redundancy EDS	None	None	None	None
Boos. Eng. Thrust Vacuum [lb.]	469449	469449	1334533	673464
EDS Eng. Thrust Vacuum [lb.]	274500	274500	469782	352748

In Table 13, the significant parameters are the stage thrust-to-weight ratio and the number of engines. The selected vehicles do not use any type of subsystem redundancy. The MFBF of design two is higher compared to the CaLV baseline. The liftoff thrust-to-weight ratio of design two is higher and the number of engines on the booster stage is lower compared to the CaLV baseline. While the EDS thrust-to-weight ratio of design two is lower, the number of engines is also lower. The cost of design two is very close to the CaLV baseline when 100 percent of the engine DDT&E costs are included with the CaLV baseline estimate. Design two would require development of an engine with a thrust level on the order of the F-1 engine, which was used on the first stage of the Saturn V,[29] while having the Isp of an SSME. The EDS stage of design two would use an SSME with air-start capability.

Design three has a lower cost compared to a CaLV baseline that incurs 100 percent of the engine DDT&E cost. Additionally, the MFBF values are close; while the EDS thrust-to-weight ratio is higher on the CaLV baseline, design three has less engines and a higher liftoff thrust-to-weight ratio.

The maximum increase in cost between the baseline CaLV and design two is 20 percent while the maximum difference in cost between the baseline CaLV and design three is 15 percent. This baseline CaLV would incur no engine development cost. The MFBF increase from the baseline CaLV to design two is 81 percent while the difference in MFBF between the CaLV baseline and design three is three percent.

IV.C.1. Uncertainty Results

Figure 14 illustrates the range of possible cost and reliability values around the pareto frontier using the 10 percent and 90 percent confidence bands. The 10 percent band is shorter because the minimum cost configuration will have a higher reliability and the maximum reliability configuration will have a lower system cost. There is a wide range of cost estimates for the maximum reliability configuration; the absolute width is approximately 14 billion dollars [FY '04] (+/- 15%). The absolute cost range is significantly smaller for the lower reliability configurations with the width equal to approximately 6 billion dollars [FY '04] (+/- 11%) for the minimum cost configuration. The reliability estimates also change significantly; the mean flights between failure varies between 197 and 343 (+/- 31%) for the maximum reliability configuration and 25.83 and 46.94 (+/- 34%) for the minimum cost configuration.

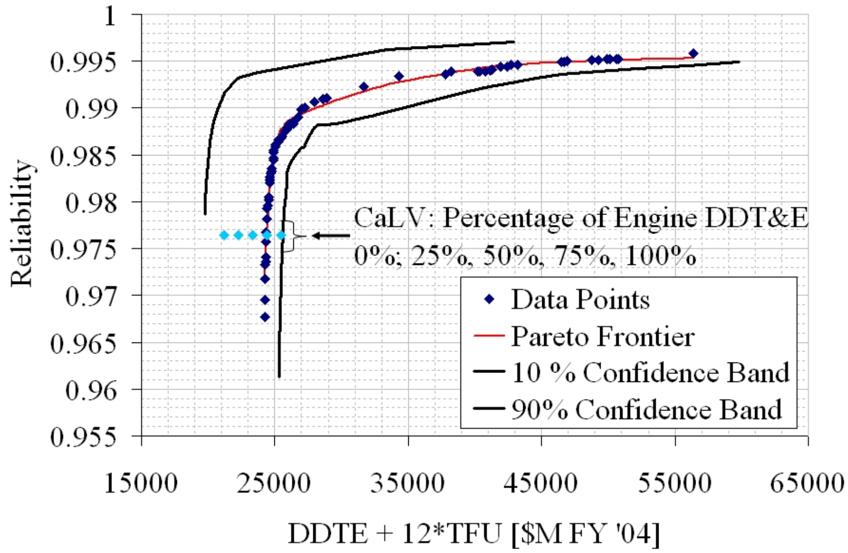

Figure 14: CaLV Pareto Frontier with Uncertainty.

V. Conclusions and Future Work

V.A. Conclusion

The methodology demonstrated in this paper can be used to create an integrated environment to evaluate the system cost and reliability of a launch vehicle. If the baseline CaLV incurs more than 75 percent of the DDT&E cost of a new engine during the current engine development, then an alternative configuration may have provided a better reliability value. Design two had an 81 percent improvement in reliability with a lower cost than the CaLV baseline if the CaLV baseline incurs a full engine development cost. If no engine development cost is incurred, the cost increase is 20 percent. By using an integrated design environment, a set of optimal CaLV configurations were created that illustrate how the CaLV reliability changes as a function of system cost.

V.B. Future Work

This methodology was created with the flexibility to easily make additional enhancements. Therefore, an operations cost model that can discriminate between launch vehicle concepts based on their design characteristics could be added.

Other enhancements could include a quantitative safety model and the cost of unreliability. Both of these inclusions would most likely favor launch vehicle configurations with higher reliability. Furthermore, a learning curve could be applied to calculate the cost of each production vehicle based upon the original first unit value. Then the cost metric would become the summation of these production costs and the total development cost.

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