Crew Launch Vehicle (CLV) Independent Performance Evaluation

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The crew launch vehicle is a new NASA launch vehicle design proposed by the Exploration Systems Architecture Study (ESAS) to provide reliable transportations of humans and cargo from the earth's surface to low earth orbit (LEO). ESAS was charged with the task of looking at the options for returning to the moon in support of the Vision for Space Exploration. The ESAS results, announced in September 2005, favor the use of shuttle-derived launch vehicles for the goals of servicing the International Space Station after the retirement of the STS and supporting the proposed lunar exploration program. The first launch vehicle to be developed is the Crew Launch Vehicle (CLV), which will be operational by 2012, and will be derived from a four-segment Shuttle Solid Rocket Booster (SRB) and an upper-stage powered by an expendable version of the Space Shuttle Main Engine (SSME). The CLV will be capable of sending approximately 60,000 lbs to LEO in the form of a Crew Exploration Vehicle (CEV) as well as a Service Module (SM) to support the CEV.

The purpose of this paper is to compare the published CLV numbers with those computed using the design methodology currently used in the Space System Design Laboratory (SSDL) at The Georgia Institute of Technology. The disciplines used in the design include aerodynamics, configuration, propulsion design, trajectory, mass properties, cost, operations, reliability and safety. Each of these disciplines was computed using a conceptual design tool similar to that used in industry. These disciplines were then combined into an integrated design process and used to minimize the gross weight of the CLV. The final performance, reliability, and cost information are then compared with the original ESAS results and the discrepancies are analyzed. Once the design process was completed, a parametric Excel based model is created from the point design. This model can be used to resize CLV for changing system metrics (such as payload) as well as changing technologies.

Nomenclature

CAD	=	computer aided design
CER	=	cost estimating relationship
CES	=	crew escape system
CEV	=	crew exploration vehicle
CLV	=	crew launch vehicle
DDT&E	=	design, development, test, & evaluation
DSM	=	design structure matrix
ESAS	=	Exploration Systems Architecture Study
ETO	=	Earth to orbit
GLOW	=	gross lift-off weight
Isp	=	specific impulse, sec
KSC	=	Kennedy space center
LCC	=	life cycle cost
LEO	=	low earth orbit
LH2	=	liquid hydrogen

LOX	=	liquid oxygen
MECO	=	main engine cutoff
MER	=	mass estimating relationship
MR	=	mass ratio (gross weight / burnout weight)
RSRB	=	reusable solid rocket booster
SSME	=	space shuttle main engine
STS	=	space transportation system
TFU	=	theoretical first unit

I. Introduction

The crew launch vehicle is a new NASA launch vehicle design proposed by the Exploration Systems Architecture Study (ESAS) to provide reliable transportations of humans and cargo from the earth's surface to low earth orbit (LEO). The ESAS results, announced in September 2005, favor the use of shuttle-derived launch vehicles for the goals of servicing the International Space Station after the retirement of the STS and supporting the proposed lunar exploration program. The CLV is a space shuttle derived launch vehicle. The CLV uses shuttle heritage components such as the reusable solid rocket booster (RSRB) and the space shuttle main engine (SSME) to both reduce overall development costs as well as take advantage of the significant effort already spent on increasing the reliability of the shuttle components. The focus on this paper will be the design of the launch vehicle itself including a crew escape system. The crew exploration vehicle and the service module are treated as payload for the CLV and therefore only their weights are considered in this design.

The CLV design is a two stage shuttle derived launch vehicle. The first stage consists of a space shuttle derived RSRB. The second stage is a new stage designed around a single SSME. The second stage will consist of a single LOX tank and a single LH2 tank constructed of Aluminum. The payload of this vehicle is a capsule-style CEV with a supporting service module. The total weight of this payload is approximately 59,900 lbs and it is injected into a 30 X 100 nmi orbit at 60 nmi. The CLV is also designed to improve the reliability of human launch beyond that of the space shuttle. This is accomplished by utilizing the flight proven elements of the shuttle system, and eliminating the potential problems now plaguing the shuttle fleet. This includes eliminating the potential for damage to the reentry heat shield by placing the CEV at the top of the launch vehicle and keeping the thermal protection system shielded through the ascent. The crew escape system further decreases the probability of a loss of crew event. The resulting overall reliability of the system is 0.9988 or 1.19 failures per 1000 flights.

The purpose of this paper is to compare the published CLV numbers with those computed using the design methodology currently used in the Space System Design Laboratory (SSDL) at The Georgia Institute of Technology. This multi-disciplinary conceptual design process is used to create the CLV design. This design process was completed using a disciplinary design tool for each of the following disciplines: external configuration and CAD was completed using ProEngineer, aerodynamic analysis was conducted with APAS[1], trajectory optimization used POST[2], mass estimation and sizing was completed using mass estimating relationships[3] (MERs), Cost estimating was conducted using NAFCOM[4] cost estimating relationships (CERs), and reliability was completed using Relex[5]. Each of these tools was used to analyze their respective disciplines and was iterated to close the CLV design.

II. Crew Launch Vehicle Configuration

The crew launch vehicle is a two stage launch vehicle designed to transport the CEV and service module to low earth orbit. The CLV design utilizes propulsion elements from the current space shuttle. The first stage is a reusable solid rocket booster. This RSRB is the same as the current shuttle solid boosters. The second stage is propelled by one SSME. An SSME was chosen to give the desired thrust to weight (~0.86) on the upper stage, while still providing the efficiency (Isp = 452.1) of a staged combustion LOX/LH2 engine. The SSME design will be modified to start at altitude as well as simplified to limit production costs. These simplifications are thought to limit production costs due to the expendability of the engine without sacrificing reliability.

The CLV is designed to carry a payload of approximately 59,900 lbs into a 30 X 100 nmi orbit injected at 60 nmi. This payload weight was chosen as a result of the ESAS study for the CEV and service module design. This orbit will allow the CEV to rendezvous with the prelaunched Earth departure stage and lunar lander in LEO and continue on to the Moon. The resulting vehicle is 309 ft tall and weights 1.840 million pounds.

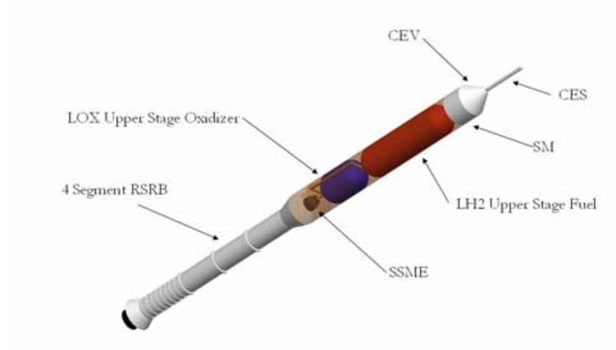

Figure 1. CLV Configuration.

In the design of the CLV reliability and safety are the main concerns. The CLV is designed to provide reliability 10X greater than that of shuttle. This is accomplished by taking reliable shuttle components and eliminating the fault paths discovered in the shuttle program. The main differences between the shuttle and the CLV are that the CLV is a completely inline system. This system eliminated the possibility of ejected pieces from contacting the vital crewed compartment of the vehicle. This eliminates the possibility of insulation damaging the heat shield. The CLV also uses the RSRB on the first stage. This is a highly reliable rocket motor with over 200 successful flights with only one failure. This failure was extensively investigated and resulted in a redesign of the RSRB. The final addition to the CLV to improve safety is the addition of the crew escape system. This system consists of a solid motor placed on top of the CEV. This system will engage if a failure occurs in either stage of the vehicle. It is assumed that if a failure in the launch vehicle occurs the CES has a 90% chance of separating the CEV from the CLV and safely recovering the crew. The resulting calculated reliability of the CLV is 0.9988, which is at least 10X better than the demonstrated shuttle reliability.

As Figure 2 shows the CLV is comparable with other previously existing expendable launch vehicles. The CLV is very similar in overall gross mass with the Titan IV launch vehicle. It is significantly taller than the Titan IV due to the large LOX/LH2 upperstage. The CLV is significantly smaller than the Saturn V in overall height and only a third of the weight. This is due to the limited payload capacity of the CLV (The CLV only launches the capsule and the SM, while the Saturn V launched the lunar module and earth departure stage as well).

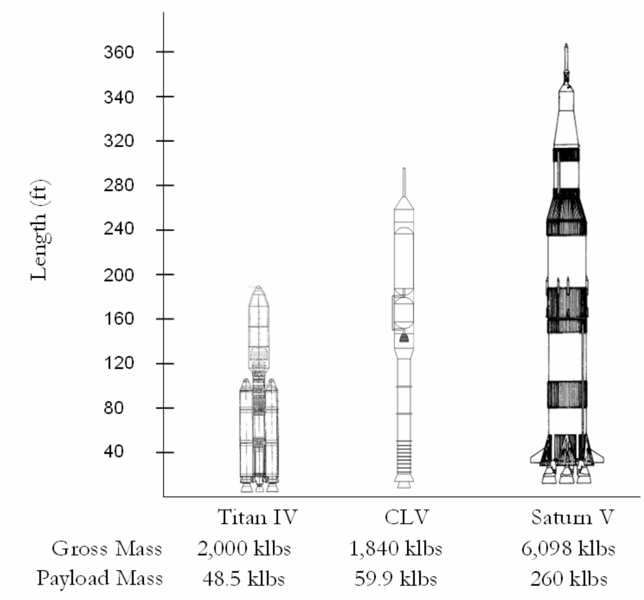

Figure 2. Comparison of CLV with other Expendable Launch Vehicles.

III. Multidisciplinary Design Process

The conceptual design methodology used in the design of the CLV combined analyses from several different disciplines. A different tool was used for each disciplinary analysis, as shown in Table 2. Each tool acts as a contributing analysis to the overall design of the vehicle. In some cases, iteration between two or more analysis tools is required. This coupling can be best visualized as a Design Structure Matrix (DSM) or "N-squared" diagram. Each box along the diagonal represents a contributing analysis, and the lines represent the flow of information. Information that is fed-forward through the design process is represented by lines on the upper right of diagonal, while the lines in the lower left are feed-back.

The DSM for the CLV design is shown in Figure 3. The feedback between the Trajectory analysis and the Weights & Sizing analysis closes the performance and configuration of the launch vehicle. The feedback between Operations, Reliability, and Cost closes the economics of the vehicle.

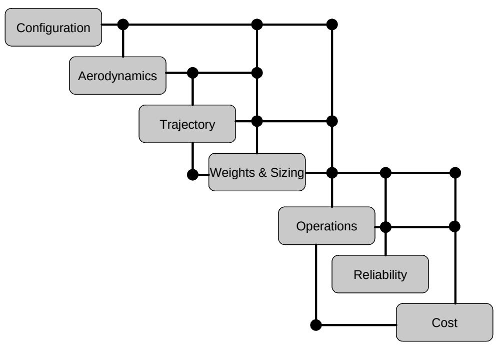

Figure 3. DSM for CLV Design.

Table 1. CLV Design Tools.

Discipline	*Analysis Tool*
Configuration	Pro/E
Aerodynamics	APAS (HABP)
Trajectory	POST 3-D
Weights & Sizing	MS Excel
Operations	AATE
Reliability	Relex
Cost	NAFCOM

IV. Crew Launch Vehicle Closure Results

Each of the design disciplines depicted in Figure 3 and Table 1 are explained below in this section. Each of these design disciplines were iterated and closed to get the final CLV design. In designing and closing the CLV, there were constraints that had to be taken into account. As a human-rated launch vehicle it is important that the maximum dynamic pressure (or "max-q") remain low in order to enable crew escape. It was desired to have a max-q below 740 psf, which was that experienced by the Saturn V rocket[6]. The CLV was closed at a max-q of 700 psf, below that of the Saturn V. However, in order to study what would be required to further reduce the loads on the crew during a crew escape event, certain changes were made to the SRB thrust profile that allowed a maximum dynamic pressure of 600 psf to be obtained. Results from both designs are presented in the following sections.

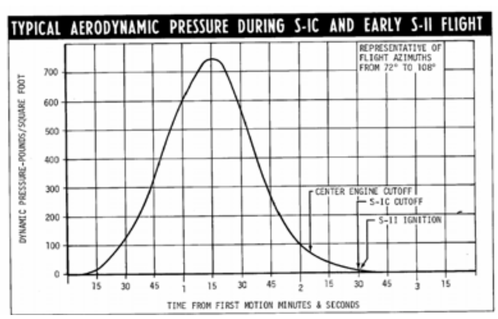

Figure 4 - Saturn V Dynamic Pressure versus Time[6]

A. Internal Configuration and Layout

As noted previously the CLV consists of two flight elements. The first stage is the four segments solid rocket booster from the shuttle program. The second stage is a new LOX/LH2 stage that is 124 feet tall and has the same diameter (18.04 feet, 5.5 meters) as the CEV. This second stage provides a significant portion of the ΔV requirement (74%) to get to LEO. A summary of the individual components follows as Figure 5 and Figure 6.

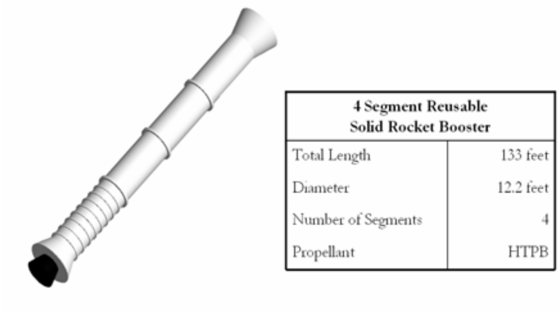

Figure 5. 1st Stage Configuration RSRB.

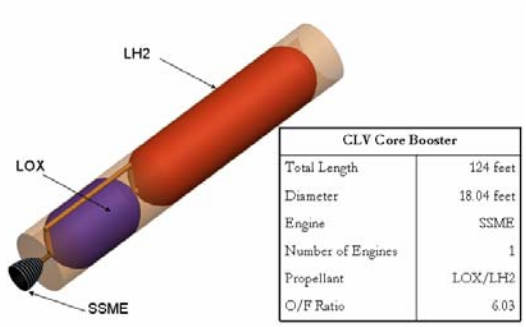

Figure 6. 2nd Stage Configuration (SSME).

B. Propulsion Design

The CLV utilizes two shuttle-derived propulsion systems that both have high demonstrated reliability over the more than two decades of shuttle operation. The first is the Shuttle 4-Segment Solid Rocket Booster, which functions as the sole booster stage of the CLV. During the initial design process, an off-the-shelf 4-segment SRB was used, which has the same performance and thrust profile as the current Shuttle SRBs. However, during design space exploration, it was found that due to the high thrust levels through the lower (and therefore denser) parts of the atmosphere, the vehicle design did not close for a max-q less than 700 pounds per square foot. In order to reduce the acceleration experienced by the crew during an abort, it may be desirable to reduce the max-q to 600 psf. When this constraint was added to the trajectory optimization code used in the CLV analysis, the trajectory code and weights and sizing tools did not converge to a closed vehicle design.

The dynamic pressure versus time plot for the closed vehicle trajectory with a max-q of 700 psf is shown in Figure 7. Also shown on the same plot is the thrust profile of the SRB. The max-q occurs at around 50 seconds, about the time that the thrust reduces to about 2.4 million pounds (point 1). One way of reducing the max-q further would be to reduce the vehicle thrust even further at this point in the trajectory. This could be accomplished by tailoring the grain of the propellant. However, it was desired to keep the total impulse provided by the SRB constant. This is accomplished by increasing the thrust at point 2, while keeping the area under the curve equal to that of the original thrust profile.

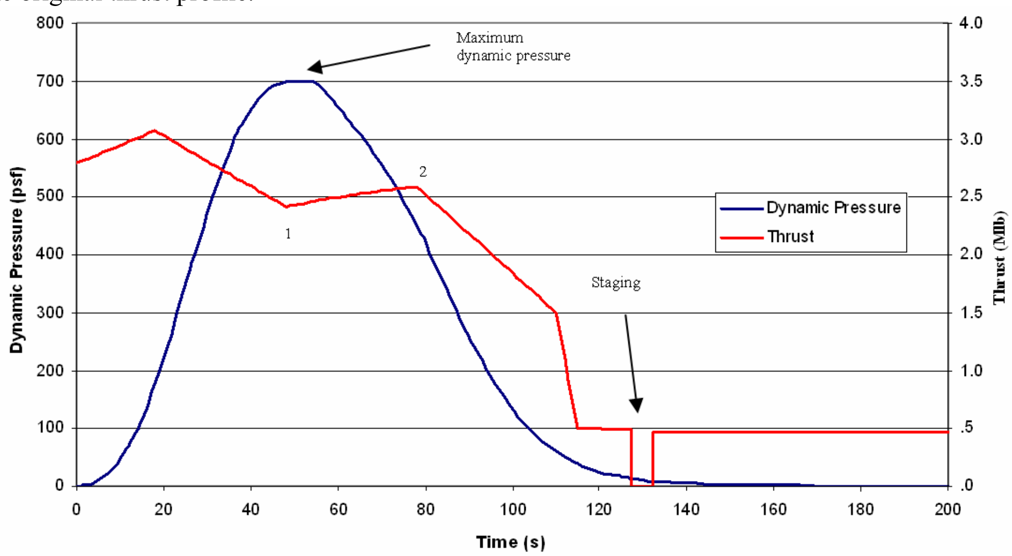

Figure 7. Dynamic Pressure (Max q limited to 700psf) with original SRB Thrust Profile.

The resulting trajectory is shown in Figure 8. With the new thrust profile, the trajectory and sizing analyses were easily able to converge to a closed vehicle design.

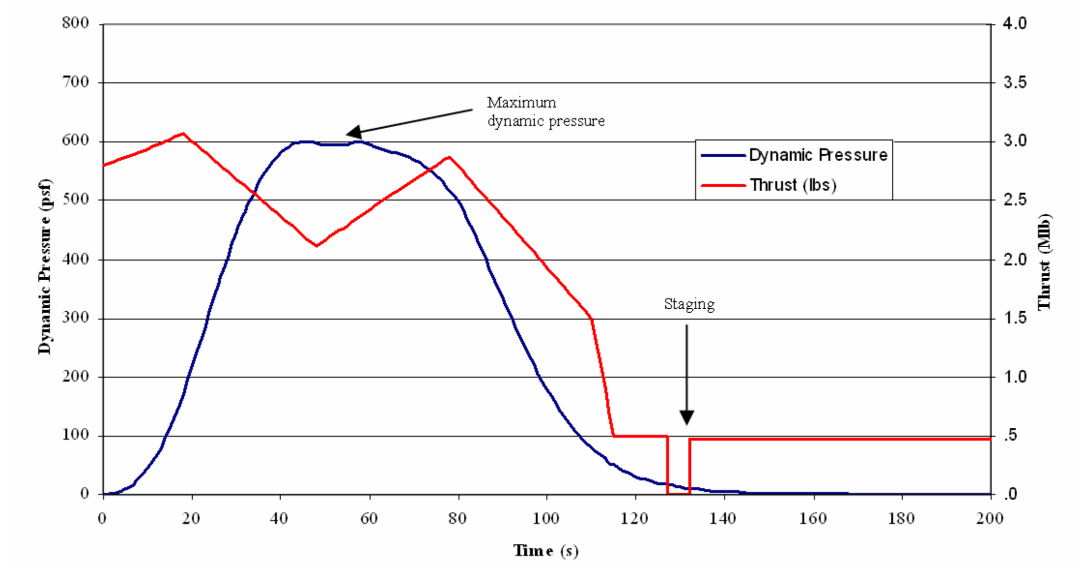

Figure 8. Dynamic Pressure (Max q limited to 600psf) with modified SRB Thrust Profile.

The upper stage propulsion of the CLV is assumed to be a single Space Shuttle Main Engine (SSME), manufactured by Boeing-Rocketdyne. The SSME is staged-combustion engine that runs on liquid oxygen and liquid hydrogen propellants. It has performed very reliably on ever Space Shuttle mission. The SSME used on the Space Shuttle, however, is started using equipment on the launch pad. In order to start the SSME at altitude, modifications must be made to its hardware which adds to the development cost of the propulsion system of the CLV. Other than the air-start capability, however, the same performance characteristics as the current SSME Block II were assumed (Figure 9).

- Staged-Combustion Cycle
- LOX/LH2 Propellants
- Thrust: 469,000 lb (Vacuum)
- Isp: 452.1 s (Vacuum)
- Weight: 7000 lb
- Exit Area: 120 sq. ft
- Expansion Area: 69
- Chamber Pressure: 3000 psi

http://www.boeing.com/defense-space/space/propul/SSME.html

Figure 9. SSME Performance Characteristics.

- 4-segment Solid Rocket Booster
- Propellant: PBAN
- Useable Propellant: 1,108 klb
- Burnout Weight: 180 klb
- Max Thrust: 3,300 klb (Vacuum)
- Isp: 268 s (Vacuum)

http://www.space.com/imageoftheday/image_of_day_031118.html

Figure 10. RSRB Performance Characteristics.

C. Performance

The trajectory of the CLV is optimized using POST 3-D[2]. The simulated trajectory was required to deliver approximately 30 tons to a 30 x 100 nmi transfer orbit, using a SRB booster stage and SSME powered upper stage.

The CLV trajectory is optimized to minimize the gross weight of the CLV by changing the pitch angles during the ascent. The constraints on the trajectory are: the final orbit, the g forces for the ascent must not be greater than 4 g's, the maximum dynamic pressure, and the final payload must be 30 tons. The staging point was not changed, due to the fixed burn time of the SRB first stage.

The trajectory plots for the two closed CLV designs are shown in Figures 11-14.

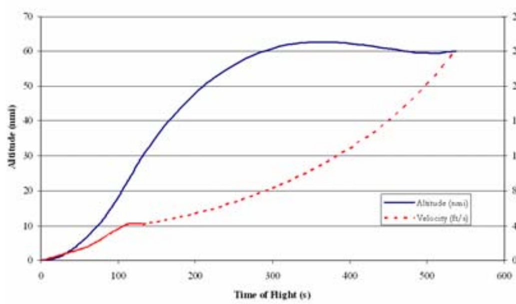

Figure 11. Altitude and Velocity versus Time with original SRB Thrust Profile.

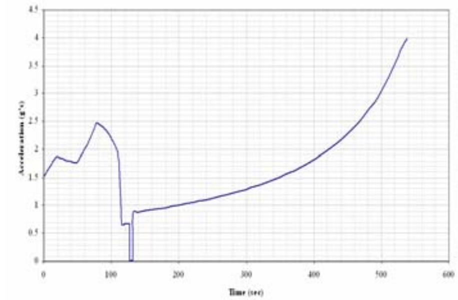

Figure 12. Axial Acceleration Sensed by CEV with original SRB Thrust Profile.

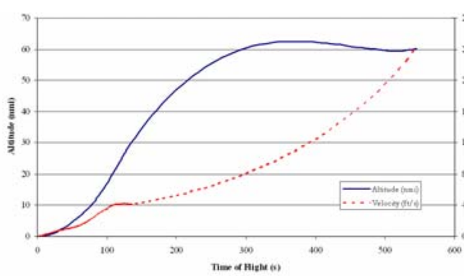

Figure 13. Altitude and Velocity versus Time with modified SRB Thrust Profile.

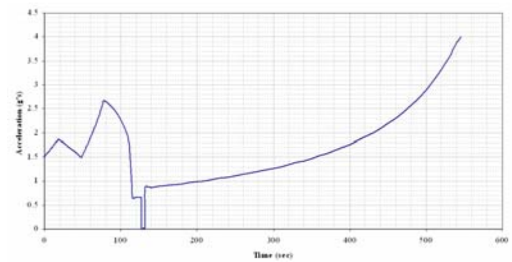

Figure 14. Axial Acceleration Sensed by CEV with modified SRB Thrust Profile.

The differences between the two trajectories can be seen in the acceleration experienced by the CEV Occupants on the two different vehicles. The second peak in acceleration is higher on the 600 psf vehicle, because that corresponds to an increase in thrust of the modified SRB over the original. The two trajectories are similar in most other respects.

Table 2. 700 psf CLV Propellant Breakdown.

Fuel	*Value*
First Stage PBAN	1,108 klb
Second Stage LOX	361 klb
Second Stage LH2	60 klb

Table 3. 600 psf CLV Propellant Breakdown.

Fuel	*Value*
First Stage PBAN	1,108 klb
Second Stage LOX	368 klb
Second Stage LH2	61 klb

Note the difference between the two is only in the upper stage. This is because achieving the lower max-q involves "lofting" the trajectory, which causes the upper stage to take on more of the burden of achieving orbital velocity.

D. Mass Estimation & Structural Design

Mass estimation followed two methodologies. For previously designed elements, such as the booster stage and the upper stage rocket engine, historical masses have been used. The masses of elements of the CLV that will be newly developed, such as the upper stage tanks, interstage adapter, thrust structure, and vehicle subsystems are estimated using MERs[3].

Each subsystem and structural element has a unique MER based on regression of historical data. Tank MERs, for example, are based on the volume of the tanks, the type of propellant being stored inside, and the pressure of the tank.

As each run of POST 3-D is performed, the weights & sizing spreadsheet is updated to match the propellant required to achieve the trajectory. As the propellant weight changes, the CLV tank and structure is appropriately resized, and therefore the dry weight changes. The new dry weights are then inputted into POST 3-D, and the analysis rerun. As this iteration continues, the vehicle design will converge to a closed design. A summary of the closed weights for the 600 and 700 psf closed vehicles are shown in Tables 4 and 5.

Table 4. CLV (max-q = 700 psf) Mass Summary.

Weight Breakdown Structure	*Mass*
Booster Dry Weight	180 klb
Booster Propellant	1,108 klb
Interstage Adapter	5.5 klb
Booster Gross Weight	**1,294 klb**
Upper Stage Structure	25 klb
Upper Stage Subsystems	2.2 klb
Upper Stage Propulsion	8.7 klb
Growth Margin	6.3 klb
Dry Weight	**42 klb**
Reserves and Residuals	4.2 klb
CEV	60 klb
Crew Escape	9.3 klb
Propellant	421 klb
Upper Stage Gross Weight	**538 klb**
CLV Gross Weight	**1,831 klb**

Table 5. CLV (max-q = 600 psf) Mass Summary.

Weight Breakdown Structure	*Mass*
Booster Dry Weight	180 klb
Booster Propellant	1,108 klb
Interstage Adapter	5.5 klb
Booster Gross Weight	**1,294 klb**
Upper Stage Structure	25 klb
Upper Stage Subsystems	2.2 klb
Upper Stage Propulsion	8.7 klb
Growth Margin	6.3 klb
Dry Weight	**42 klb**
Reserves and Residuals	4.2 klb
CEV	60 klb
Crew Escape	9.3 klb
Propellant	429 klb
Upper Stage Gross Weight	**547 klb**
CLV Gross Weight	**1,840 klb**

The gross weight for both vehicle designs is around 1.8 million pounds. This is slightly heavier than the Delta IV Heavy, which has a gross weight of 1.6 million pounds, and a smaller payload to LEO. It should be noted that while the gross weight increased for the second closed vehicle design, the increase in dry weight was less than 300 lb. Most of the extra mass is extra propellant required to fly the more lofted trajectory.

E. Reliability

The reliability for the CLV is calculated using the software RELEX[7]. Among other capabilities, RELEX can create and calculate a fault tree analysis like the one shown below for the CLV. For this analysis, it was assumed that Loss of Vehicle (LOV) and Loss of Crew (LOC) were the same. No distinction is made between the 600 psf CLV and the 700 psf CLV because the only differences are in configuration and weight; the failure modes are assumed to be the same. The majority of the input numbers were taken from a Futron study on launch vehicle reliability[10]. The SSME reliability is referenced from the Boeing Co[8]. There have been no assumptions regarding increases in reliability; all of the numbers used are traced from demonstrated systems. This heritage uses older technology, and thus it could be reasonable to assume that this is a conservative estimate of CLV reliability. Finally, even though the SRB first stage has a demonstrated failure rate of lower than 0.004, the authors of this paper decided to continue with a conservative analysis and calculate the reliability with a full bottoms-up approach. The fault tree is illustrated in Figure 15.

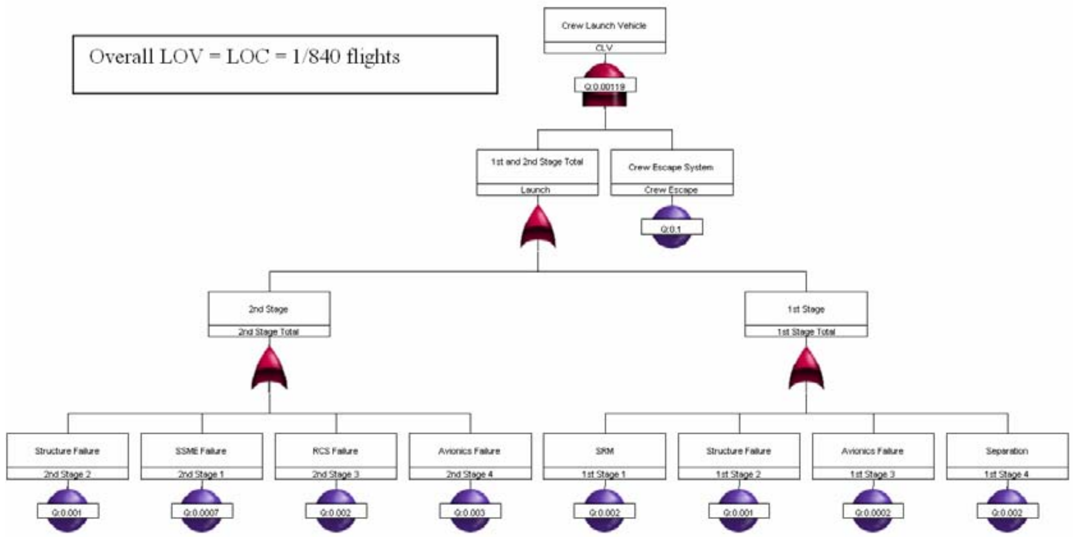

Figure 15. Fault Tree Analysis of CLV

The calculated reliability of the CLV is very high. This can be attributed to two main factors. First, on the SRB stage, the majority of the components are from well demonstrated solid rocket systems. The lack of complexity when using solid rockets will typically manifest itself in reliability calculations. This is one reason why solid boosters are chosen even though they lack the performance that can be achieved with a liquid rocket engine. This first stage also reflects the knowledge gained from the experience with the STS system and thus a high reliability is achieved.

For the second stage, the SSME is expected to be the main driver for LOC. Yet, its reliability is also very high due to its heritage. This engine is one of the most extensively tested rocket engines, and thus an excellent failure rate is achieved. The rest of the second stage inputs are determined from historical liquid rocket systems. With a typical driver of LOC having a high reliability, the whole system will then realize a high reliability. Finally the addition of a crew escape system aids in ensuring a low LOC number.

F. Operations

The operations costs for the CLV are calculated using estimates based on the STS program. The entire first stage is an STS component, which makes this analysis appropriate. In addition, the 2nd stage uses the SSME; this is another reason for why these estimates can be used. However, because these components are derived from the STS program, they will inherit some of its cost structure. The CEV component was not considered in these costs; therefore, additional costs for the turnaround of the CEV, along with its facilities are not included here. Also, the cost of modifying facilities for the CLV are not included in these costs. The lack of reliable data for which to base these estimates upon is the major reason that facility modifications are not included.

The variable cost per flight is estimated at $43.9M FY ’04. The annual fixed costs for operations are estimated at $741.5M FY ’04. These costs are driven from a derivation of shuttle hardware. However, when comparing to the STS program, these costs are a fraction of what the fixed and variable costs used to be. It is acknowledged that the CEV has not been included, nor has any impact of the future Heavy Lift Vehicle been estimated. Yet, with the goal of sustained exploration, these operations costs help the CLV fit within initial budget estimates for achieving sustained access to space with these future systems.

G. Cost & Economics

To estimate the rest of the costs for the CLV design, weight-based cost estimating relationships (CERs) were used. The CERs were used to calculate an overall design, development, testing, and evaluation (DDT&E) cost. Initial production costs are also calculated. Finally, with the use of a 90% learning curve, production costs based upon the number of flights is estimated. These costs are then broken down by stage to see what the main cost drivers are. The CERs are created from data in the NAFCOM[4] model used in cost estimating. An initial summary of the costs are listed in Table 6 (All costs are presented in $M FY 2004 dollars at an undiscounted rate). The margin has already been built in to the costs and is included for more information.

Table 6: CLV DDT&E and TFU Costs ($M FY '04)

			600 Psf	700 Psf
RSRM				
	DDT&E		$589	$417
	Production		$58	$54
	Total RSRM		$647	$471
	Margin (20%)		$107	$78
Upper Stage				
	DDT&E			
		Airframe	$1,465	$1,460
		Engine	$902	$902
	Production			
		Airframe	$283	$282
		Engine	$49	$49
	Total Upper Stage		$2,699	$2,693
	Margin (20%)		$449	$446

While the SRB has been used before, it was assumed that some DDT&E will be needed to prepare it to carry a new upper stage. Additionally, the SRB will now have a different separation sequence, which will require some DDT&E before it can be flight qualified for human travel. The 600 psf will require more DDT&E because that SRB will have to undergo a grain re-design. The 700 psf case uses an off the shelf SRB; however, in order to achieve the lower dynamic pressure, the thrust profile of the SRB must be changed. Hence, there is an increase in the DDT&E of the 600 psf vehicle. The production costs are similar because the grain casting technique should be similar. However, there will be some increase in cost due to a different grain pour.

The second stage will require more development funding than the SRB first stage, due to the fact that it is a new second stage that must be qualified for human flight. Additionally, the "air lighting" of an SSME will require a redesign and re-certification of this engine. Since the SSMEs used on the CLV are expendable, many will be produced for this architecture, and the manufacturing techniques will also be changed. Thus, there will be a learning process during this time which will give the SSME development a cost almost equivalent to a new engine development. However, once this development has taken place, the production cost of the new SSME will be more in line with its original predecessor. The slight differences between costs of the 600 psf CLV and the 700 psf CLV for the 2nd stage can be attributed to weight differences. Since weight based CERs are being used, and these vehicles have slightly different masses, it is reasonable to expect slight differences in their 2nd stage costs. However, since they are using the same "airlight" SSME, these costs will be the same.

The Life Cycle Cost (LCC) for the CLV is based upon a 15 year campaign and a flight rate as illustrated in Figure 16. The calculation includes the DDT&E cost, plus the production costs (dependent upon flight rate), plus the annual fixed cost, plus the variable cost (also dependent upon flight rate). Figure 17 shows the 600 psf case. Illustrated on the graph are both the total costs per flight along with the recurring costs per flight. The two costs are very close; this is because the development cost spread over 15 years does not add greatly to the total cost per flight. The variable and fixed cost per year will easily eclipse the development costs for this vehicle when the total LCC is calculated. This was the trade off made with the decision to use shuttle derived hardware.

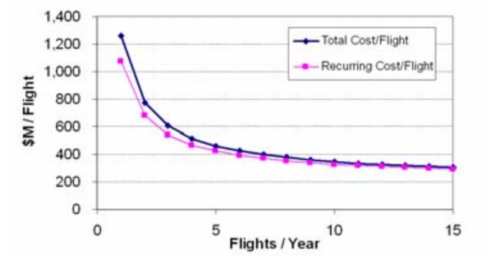

Figure 16: CLV 700 psf Cost Per Flight ($M FY '04), 15 Year Campaign

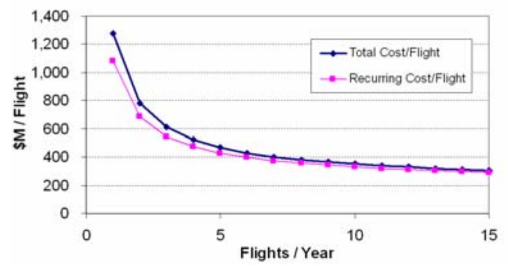

Figure 17: CLV 600 psf Cost Per Flight ($M FY '04), 15 Year Campaign

The next figure shown is Figure 16, which is the 700 psf case. The same trends exist as before: the total cost per flight and the recurring cost per flight are very close. Again this is because the development cost is small compared to the LCC contribution made by the annual fixed and variable costs. The two launch vehicles are very close when compared for overall LCC. The 600 psf vehicle will invariably cost more because it spends more on development for the new grain design. Additionally, more is spent on production of the 600 psf vehicle. Both are comparable in operations costs, thus the difference in development and production costs will account for the slight difference in total LCC. However, both are very close.

The total costs of these vehicles are within projected budgets for exploration. A flight rate of 2 per year will result in a cost of $783M FY '04 for the 600 psf CLV and $773M FY '04 for the 700 psf CLV. With a doubling of flight rate, these costs can be moved into the $500M range. However, this figure can be misleading because it incorporates the DDT&E costs; these costs will already have been paid for by the time of the first flight. Therefore, with a cost of $684M and $680M FY '04 for the 600 and 700 psf vehicles, respectively, the goal of achieving sustained exploration can be reached.

V. Comparison with ESAS Results

It is now appropriate to compare the completed CLV design computed with the GT methodology to the closed ESAS design[9]. This comparison follows as Table 7.

Table 7. Comparison of GT and ESAS Results.

		GT	NASA ESAS
Gross Weight (inc. CEV and CES)		1,840,344 lb	1,775,385 lb
First Stage			
	Dry mass	180,399 lb	180,399 lb
	Gross mass	1,293,517 lb	1,292,655 lb
	Height	133 ft	133 ft
	Diameter	12 ft	12 ft
Second Stage			
	Dry mass	42,084 lb	38,597 lb
	Gross mass	477,629 lb	405,541 lb
	Height	124 ft	105 ft
	Diameter	18.04 ft	16.40 ft

As this table shows the GT results compare very closely with the NASA ESAS results. The major difference was the weight of the second stage. The GT results are indicative of flying a "lofted" profile to limit the maximum dynamic pressure. This limit, as discussed above, was to limit the acceleration sensed by the astronauts on abort (the higher the max dynamic pressure, the larger the abort engines, and the higher the g's on a low dynamic pressure abort). The ESAS studies smaller mass could be a result of further tailoring of the ascent profile, or an introduction of higher technology structures (i.e. Al-Li) to limit the structural weight of the second stage. The resulting extra mass of the GT design still results in a feasible vehicle for the reference payload.

VI. Conclusions

The CLV is a new NASA launch vehicle design proposed by the Exploration Systems Architecture Study (ESAS) to provide reliable transportations of humans and cargo from the earth's surface to low earth orbit (LEO). The CLV uses shuttle heritage components such as the reusable solid rocket booster (RSRB) and the space shuttle main engine (SSME) to both reduce overall development costs as well as take advantage of the significant effort already spent on increasing the reliability of the shuttle components. The CLV design is a two stage shuttle derived launch vehicle. The first stage consists of a space shuttle derived RSRB. The second stage is a new stage designed around a single SSME. The second stage will consist of a single LOX tank and a single LH2 tank constructed of Aluminum. The payload of this vehicle is a 18 ft conical CEV with a supporting service module. The total weight of this payload is approximately 59,900 lbs and it is injected into a 30 X 100 nmi orbit at 60 nmi. The resulting CLV design is over 1.840 million pounds and stands 309 ft tall. These results are just slightly larger then the reference ESAS design.

The CLV is also designed to improve the reliability of human launch beyond that of the space shuttle. This is accomplished by utilizing the flight proven elements of the shuttle system, and eliminating the potential problems now plaguing the shuttle fleet. This includes eliminating the potential for damage to the reentry heat shield by placing the CEV at the top of the launch vehicle and keeping the thermal protection system shielded through the ascent. The crew escape system further decreases the probability of a loss of crew event. The resulting overall reliability of the system is 0.9988 or 1.19 failures per 1000 flights. This increases reliability over shuttle is achieved at a significant savings over the existing shuttle design. The total costs of these vehicles are within projected budgets for exploration. A flight rate of 2 per year will result in a cost of $783M FY '04 for the 600 psf CLV. With a doubling of flight rate, these costs can be moved into the $500M range. However, this figure can be misleading because it incorporates the DDT&E costs; these costs will already have been paid for by the time of the first flight. Therefore, with a cost of $684M for the 600 psf CLV, the goal of achieving a sustainable exploration architecture can be reached.

VII. Acknowledgments

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Full Concept Architecture for Human Lunar Exploration: Tools and Design Methodology

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Abstract

Using NASA's Exploration Systems Mission Directorate Spiral 3 requirements for human lunar surface exploration, a full concept architecture and Crew Transportation System vehicle design was performed using Apollo era architecture as the baseline. The analysis was executed using such systems engineering tools as Solid Edge, Program to Optimize Simulated Trajectories (POST), Aerodynamic Preliminary Analysis System (APAS), NASA/Air Force Costing Model (NAFCOM), Technique for Order Preference by Similarity to the Ideal Solution (TOPSIS) and Space Propulsion Sizing Program (SPSP). These systems engineering tools and techniques allowed the analysis to ensure a robust design that included atmospheric and space performance, system weights and geometries, safety and reliability, and life cycle costing. Implementing these tools provided greater efficiency to create iterative design selections than Apollo era methodology, resulting in a final systems architecture selection that used the multi-attribute decision making tool, TOPSIS.

Nomenclature

APAS	Aerodynamic Preliminary Analysis System
AFRSI	Advanced Flexible Reusable Surface Insulation
CAD	Computer Aided Design
CEV	Crew Exploration Vehicle
c.g.	Center of Gravity
CM	Command Module
CSM	Command and Service Module
CTS	Crew Transportation System
DDT&E	Design, Development, Test, and Evaluation
ECLSS	Environmental Control and Life Support Systems
EDS	Earth Departure Stage
EELV	Evolved Expendable Launch Vehicle
EVA	Extra Vehicular Activity
FOM	Figure of Merit
I_{sp}	Specific Impulse
LEM	Lunar Excursion Module
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LAM	Lunar Ascent Module
LDM	Lunar Descent Module
LSAM	Lunar Surface Access Module
MADM	Multi-Attribute Decision Making
NAFCOM	NASA/Air Force Costing Model
PICA	Phenolic Impregnated Carbonaceous Ablator
PM	Propulsion Module
POST	Program to Optimize Simulated Trajectories
RCS	Reaction Control System
RM	Re-entry Module
RTG	Radioisotope Thermal Generator
SDLV	Shuttle Derived Launch Vehicle
SPSP	Space Propulsion Sizing Program
TOPSIS	Technique for Order Preference by Similarity to the Ideal Solution
TEI	Transearth Injection
TLI	Translunar Injection
TPS	Thermal Protection System
T/W	Thrust to Weight
ΔV	Change in Velocity

1 Introduction

A conceptual design was obtained for a Crew Transportation System (CTS) intended to transport people safely from Earth to the moon and back. The procedure for design began with a validation of engineering tools for analysis of aerodynamics, trajectory performance, weights, propulsion, and cost using the Apollo Command, Service, and Lunar

Exploration Modules as benchmarks. Subsequently, the tools were used to size the Apollo architecture to meet a set of Spiral 3 requirements that are similar to those currently issued by NASA to industry. Today's technologies, unavailable in the 1960's, were analyzed in an effort to reduce the vehicle's mass, decrease total cost, lessen risk, and improve safety.

1.1 Tools Description

To accomplish a systems level design, the engineer must rely upon a variety of tools. Each tool contributes to the accuracy and speed to which results can be produced. Though many engineering tools are available in the market today, there are several that contributed to the development of the CTS. The historical data reference, Solid Edge, Space Propulsion Sizing Program, Aerodynamic Preliminary Analysis System, Program to Optimize Simulated Trajectories, NASA Air Force Costing Model, and the Technique for Order Preference by Similarity to the Ideal Solution, were the primary tools used.

1.1.1 Historical Data

Historical data has been used in engineering analysis as long as the data has been generated to a sufficient degree of validation. It provides the user with a solid understanding of the previous uses of certain technologies as well as a verification data-point for new software tools. In the 1950's, 1960's, and 1970's, a tremendous amount of testing data and flight data was generated on America's first conquest to the moon. Valuable historical data was later provided from the space shuttle, space stations, and various interplanetary missions by NASA. Many tools today base their legitimacy upon historical data because they are valid data points and very difficult to simulate with sufficient accuracy. Such tools as NAFCOM (for costing) and SPSP (for systems sizing) actually draw on this data in their software processing. Historical data can also provide the user with warnings that show less than desirable results, thus saving a project valuable time and money. One drawback to using historical data is that much of it was generated with different technology and different methodology. For example, the main issue during Apollo was to develop new codes and manufacturing methods in a minimum amount of time, while today the issue is cost. As long as the user is able to account for such differences, as is provided for in such tools like TOPSIS, historical data can be valuable data points.

Trajectory, power, mass, cost, and reliability information are a few of the areas where historical data is necessary to prove and sometimes develop a new tool. For this project actual Apollo data was used frequently to provide a baseline for the CTS. In a study performed by Reeves and Scher[1], which explored the decisions made for Apollo, it was determined that the best decisions that could have been made, were indeed made by the Apollo engineers. The system design baseline for this project was thus done so with confidence.

1.1.2 Mass Sizing Tool

In designing a complex architecture, not all subsystems can be accounted for and sized. Simplifying assumptions to these components need to be made in order to move the design process along. For the design of architectures that are closely related to past vehicles, historical data of those past vehicles can be used as design points in sizing new architecture's individual components. The method used in sizing these components relate mostly to five fundamental unit comparisons, mass, distance, area, volume and time.

To describe how the mass sizing tool works, a certain amount of logic needs to be applied to the unit comparisons. First, all the masses are already known for a chosen past vehicle design. The assumption is then made that the new architecture will initially share all the same components as the past design, with other components added and removed as required. The ratio of system increase in distance, area, volume, and time from the new design to the old design is taken, and multiplied by the mass to yield the new increase in mass. This means that systems that only increase logically in one dimension (i.e.-length and time) are only increasing linearly, while systems that increase in two or three dimensions (i.e.-area and volume) are increasing in proportion to either the square or cube of the updated system. From this, an effective mass sizing tool can be created. For the CTS mass sizing tool, Microsoft Office Excel was used. This spread sheet incorporated the Apollo baseline as well as the various revisions that the CTS underwent.

1.1.3 Solid Edge

As computers have progressed, so have their modeling capabilities. Computer Aided Design (CAD) has become commonplace in modern engineering and most engineers have a basic understanding of the capabilities of a CAD software program. The main premise of a CAD drawing is to give the customers and engineers an overall feel of the design in a drawn-to-scale representation of an actual design, and not an artist's perspective. A CAD design should be based off a minimum number of inputs in order to reduce drawing time and maximize results.

The CAD program chosen for this architecture is Solid Edge[2]. When developing the CAD model, specific inputs had to be agreed upon that would drive the dimensions for the entire model. For the design of an entire vehicle, it was good practice to keep a master list of vehicle parameters. Microsoft® Office Excel provided a good medium for this kind of task. With the Excel spreadsheet made readily available to all members of the team, each iteration of the design was easily recorded and exported for modeling in Solid Edge. For the CTS, the driving factors in the Solid Edge model were the vehicle diameter and height. When a design had been agreed upon, specific calculations would first take place in Excel for the simple geometry of the vehicle. These calculations mostly involved tank configuration and habitable volume space, which were two major requirements. These calculations are important because if the tanks were immediately modeled, it may be found that the tanks were too large for the current fairing configuration, and an excess amount of time would have been wasted on modeling an infeasible configuration. Thus, in CAD modeling, it is always a good idea to set up an initial 'bare-bones' calculation sheet that will determine the actual, geometric feasibility of the design.

Once a design has been agreed upon, CAD modeling can begin to take place. Solid Edge does have the feature to make specific dimensions of the model reliant on inputs by calculating dimensions of the vehicle via user-defined equations. If the primary vehicle design is not expected to change drastically over the course of the project, it is a good idea to input these equations from the start. Often, these equations are the same equations used in the 'bare-bones' geometric validation calculations of the design. It should be noted that the majority of CAD software programs have this ability to build-in equations to the model being designed. Furthermore, when designing the parts of a vehicle design, it is good practice to model and save all parts as separate files. The user should never try to create the entire vehicle in one model drawing. Once all parts of the vehicle have been modeled, assembly can take place. Assembling the vehicle is simple enough, but does take a certain amount of finesse that can only come from repetitive use of the program. Once assembly is complete, the model is ready for export.

The advantage in using a CAD software tool is that it allows an overall perspective of the vehicle design. Certain factors can be more easily recognized with a solid model. Feasibility can be determined from common sense analysis of the shape of the design. For the CTS, the CAD modeling allowed the designs to be compared, and improved over the initial designs.

1.1.4 Space Propulsion Sizing Program

The Space Propulsion Sizing Program (SPSP)[3] was developed at NASA Langley through Analytical Mechanics Associates, Inc. in order to obtain quick estimations in propulsion sizing for trade studies and spacecraft design. SPSP produces results through a combination of mass estimating relationships, bottom-up calculations, and historical data to size several vehicle subsystems. Microsoft® Office Excel and Visual BASIC are both utilized in this program and provide a user-friendly workspace.

SPSP performs mass sizing for the following systems: main engines, main tanks, propellant feed, reaction control, power, structures, and avionics. SPSP also provides geometry estimates based upon structure, main tanks, and main engines for the length and diameter of propellant tanks. Besides the numerical outputs of SPSP, Excel also provides a two-dimensional drawing of the vehicle. SPSP also uses Visual Basic, and through the help of Pro-Engineer, can provide detailed three-dimensional drawings. Because of these output options the user can perform top level as well as detailed studies. A visual understanding of the system rounds out the capabilities of SPSP.

The inputs for the overall system in SPSP are payload mass, stage delta-v, propulsion envelope dimensions, upper limit g load, length of mission, number of stages, tank configuration, and maximum distance from the sun. The outputs include mass breakdowns for the entire system as well as sizing of the system, g load, and duration of the engine burn. Additional inputs are required when trying to size a subsystem (e.g. power or engines).

1.1.5 Aerodynamic Preliminary Analysis System

APAS is the Aerodynamic Preliminary Analysis System[4]. APAS was developed to provide a numerical approximation for aerodynamic coefficients in the subsonic, supersonic, and hypersonic flight regimes. An interactive system allows input and analysis of geometry data in an output format similar to wind tunnel testing. The real benefit of APAS is the ability to analyze an arbitrary design in the matter of hours while being able to retrieve a good preliminary concept of the performance characteristics. Numerous different designs can thus be analyzed and exported for analysis allowing many different ideas in the design process to be considered.

Two programs have been merged in APAS to cover all three flight regimes, Unified Distributed Panel (UDP) and Hypersonic Arbitrary Body Program (HABP). For the purposes of this architecture, only hypersonic analysis is necessary because the CTS re-entry module is an Apollo based capsule that does not require highly maneuverable lifting body verification such as the Space Shuttle. Instead, conditions upon re-entry will be more valuable for purposes of heating and trajectory optimization. It is thus prudent to realize that all assumptions and modeling for the CTS re-entry module will be based off the HABP program, and that modeling in the subsonic-supersonic regime with UDP is not necessary for a preliminary design effort that does not involve a maneuverable lifting body.

HABP uses hypersonic analysis based on non-interfering, constant pressure, finite element analysis implemented on a body structure that is defined at the APAS command line. This body structure is input as a surface of revolution about a central axis upon which center of gravity, angle of attack, etc. can be measured. This revolved surface is broken down into a series of panels that is defined from the APAS command line. The program then solves what may be characterized as a purely geometric problem. The most common method, as well as the one used for this analysis, is the Modified Newtonian Flow method, which is a function of the element impact angle. This angle is measured between the direction of flow and the plane of the panel that it is impacting. From this angle, the local impact coefficient of pressure is calculated. These pressures are integrated over the surface to compute lift, impact drag, and pitching moment. In addition, the local viscous drag due to skin friction and the base drag are added to the impact drag to calculate the total drag. Base drag becomes negligible in the hypersonic regime because it is a function of the reciprocal of the square of the Mach number.

The advantage of using APAS is the ability to obtain a good preliminary estimate to the flight characteristics of numerous different designs. Caution needs to be stressed though, that APAS is only an estimate to actual flight conditions and needs to be used only where appropriate. For subsonic-supersonic flight, UDP was found to inaccurately model actual flight characteristics for capsule-based shapes. Thus, the user with minimal knowledge of APAS should not be critically concerned with such data as such analysis is not necessary for this type of architecture analysis.

1.1.6 Program to Optimize Simulated Trajectories

POST is the Program to Optimize Simulated Trajectories[5]. It was initially developed in 1970 as a Space Shuttle Trajectory Optimization program. Specifically, POST is a generalized point mass, discrete parameter targeting and optimization program. It provides the capability to target and optimize point mass trajectories near an arbitrary rotating, oblate planet. The generality of the program is evidenced by its multiple event simulation capability which features generalized planet and vehicle models. Using generalized routines, inputs, and outputs, POST is an excellent tool to simulate a wide variety of vehicle designs for atmospheric re-entry.

In developing a POST simulation, various defaults of POST are very beneficial. For example, POST keeps a very accurate database of Earth's atmosphere and gravity field, allowing such matters to be bypassed for the simulation. POST has acquired other databases for planets besides Earth that can be used if special care is taken to understand the properties defined for those planets. This is the advantage of POST having such a generalized design, that planets can be used in the trajectory simulation at a minimum input effort. This generalization does not extend to just planets though. Other advantages are evident in POST. The inputs do not always have to be in the same format. POST can accept initial states of a spacecraft using Cartesian coordinates, Keplerian coordinates, latitude and longitude, or other common initial states. This generality of different input types is also seen in the output. POST will give various re-entry parameters throughout the simulation for the user's implicit understanding. This includes location of the spacecraft in different coordinate systems, orientation of the spacecraft with respect to quaternions, inertial Euler angles, and much more information that will be more useful depending on the type of output most desired. Overall, POST is a very powerful tool to use because of its generality in its routines, inputs, and outputs.

For this architecture, POST required a certain number of inputs, most notably mass, aerodynamic coefficients, initial state, and entry sequences. The entry sequences relate specifically to the path taken through the atmosphere as understood from historical Apollo data that was described earlier. Note that for architecture design, pinpoint-landing location is not a primary concern to the user, thus the simulation is not required to reach landing, but only perhaps parachute deploy at a low Mach number. This will still allow the user to retrieve the important simulation data such as max g load, heat rate, and heat load.

Although POST is a high fidelity simulation program, various fundamental concepts can be thought of before designing the re-entry simulation. There are fundamental equations for planar flight that will provide a better understanding of POST before using the program. For example, the equation shown below gives an idea of how drag (D), mass (m), and flight path angle (γ) affect the change in velocity with respect to time, or in other words acceleration of the re-entry vehicle.

$$\frac{dV}{dt} = -\frac{D}{m} - g\sin\gamma$$

From the above equation defined in Griffin[6], one can understand that by only changing mass, g-load upon entry will either increase or decrease. Increasing or reducing the total reference area for the re-entry vehicle can also change the drag term in turn affecting g-load as well. G-load can be a driving factor in the design of a spacecraft. This kind of simple understanding shows the ideal solution to such a problem, without having to go through the burden of optimizing g load with POST.

The advantage of using POST is that flight data for such design points as heat shield sizing and g load can be obtained. A user-friendly animation capability of POST is also beneficial to the user in helping to describe the re-entry dynamics and considerations. The disadvantage to POST is the large learning curve for its use. Users will find difficulty in understanding all the capabilities of POST. If used properly though, POST provides a powerful tool in re-entry analysis and design.

1.1.7 NASA Air Force Costing Model

The NASA Air Force Costing Model (NAFCOM) was developed by Space Applications International Corporation for NASA and the Air Force to develop a space-system costing model and is described in Ref. [7]. NAFCOM is based upon previous mission costs, with the primary driver for the costing model being mass and the secondary driver being technology development. NAFCOM is user-friendly, giving relative costing estimates that require little fore knowledge of previous space systems or costing analysis. When a more accurate costing model is desired, some knowledge of the present state of technology is necessary.

To begin creating a costing estimate in NAFCOM, a mission type is selected from the following options: manned launch vehicle, unmanned launch vehicle, Earth orbiting spacecraft, and planetary spacecraft. For the purposes of this architecture, the manned launch vehicle option was selected. The next step is selecting the vehicle type. For manned and unmanned launch vehicle options, the number of stages for the launch vehicle can be selected, or the vehicle can be shuttle derived. For Earth orbiting spacecraft, one of the following types must be selected: scientific, observatory, mapping, communication, reconnaissance, or positioning. For planetary spacecraft, inner planetary or outer planetary options are available with flyby or orbiter and probe as secondary selections. Finally, selections can be made of the type of engines needed, such as liquid or solid, and whether the vehicle should be designated ‘crew and cargo’ or just ‘crew’. Once these options are selected, a vehicle template is generated by NAFCOM. This template includes many subsystems such as thermal control, landing system, and environmental control and life support, which must be accounted for when creating a costing estimate. For each subsystem a weight is required to be input. After the weight is chosen, the technology must be evaluated. There are three slide bars that can be moved depending upon the level of advancement in technology. The three bars are ‘manufacturing methods’, ‘engineering management’, and ‘new design’. An example of how these bars might be implemented is when using a new design with undeveloped manufacturing methods, the ‘new design’ and ‘manufacturing methods’ bars are set to 100. These bars not only affect the flight unit costs, but the development and production costs as well. A useful feature of NAFCOM is that a data table with historical spacecraft information is shown near the bottom of the screen. This allows the retrieval of various weights of spacecrafts without having to go look them up in other sources. When the various subsystems have been filled out to desired completion, a main screen displays four costing estimates. The first is Design, Development and Testing costs. The second is production costs of the total number of spacecrafts. The third is a single flight unit cost and the fourth is the total mission cost.

1.1.8 Technique for Order Preference by Similarity to the Ideal Solution

The Technique for Order Preference by Similarity to the Ideal Solution (TOPSIS) is a tool that aids in decision making. TOPSIS incorporates a user-defined decision matrix (with dimensions based upon ‘criteria’ and ‘alternatives’) that compares each alternative to a preset baseline using a number scale from 1 to 9 with 5 being the same, 9 being much better, and 1 being much worse. This matrix is populated using normalized data from the trade studies, costing analysis and reliability calculations, and qualitative engineering assessment where there is no hard data. TOPSIS then formulates a “most ideal” and “least ideal” solution. The alternatives are then ranked by their Euclidean distance from these “ideals.”[8] Figures Of Merit (FOM) and their relative importance that are fed into TOPSIS come from the Analytical Hierarchy Process (AHP). As described by Saaty[9], AHP uses a full factorial user-defined pair wise comparison to numerically determine a normalized importance for each FOM.

TOPSIS is not user friendly and a good deal of information must be known before the tool can be used correctly. TOPSIS does allow there to be mathematical explanations for complicated decisions that have to be made. Therefore, TOPSIS allows the less experienced user to make a better decision that would be on par with someone who has more experience in decision-making. TOPSIS was implemented when decisions had to be made that involved trade studies. Calculations were made first for Apollo and then for this project.

The tools described above were crucial to the development of the CTS.

1.2 Concept of Operations

The Apollo spacecraft were used as a point of departure for the design resulting in a concept of operations similar to that of Apollo. As shown in Fig. 1, four launches will be required to place a Crew Exploration Vehicle (CEV), a Lunar Surface Access Module (LSAM), and two Earth Departure Stages (EDS), into Low Earth Orbit (LEO) where each vehicle will rendezvous with one EDS. The first EDS will place the uninhabited LSAM on a translunar trajectory approximately 5 days in duration, and the second EDS will transport the CEV with crew to Low Lunar Orbit (LLO) in about 4 days. A rendezvous will occur between the CEV and LSAM, during which the crew will transfer after docking. Then, the two craft will separate from each other prior to a lunar descent performed by the Lunar Decent Module (LDM) portion of the LSAM to a landing site in the polar region of the moon’s surface. Once on the lunar surface, the crew may stay for up to 70 days. In the meantime, the CEV will remain uninhabited in low lunar orbit. The two craft will be rejoined after the Lunar Ascent Module (LAM) portion of the LSAM ascends with the crew from the moon’s surface. After a crew transfer, the CEV and LAM are to be separated a second time before the CEV launches itself into an Earthbound trajectory lasting about 4 days. The LAM is then discarded. The CEV is composed of the Reentry Module (RM) and Propulsion Module (PM). The RM will separate from the PM and protect the crew during direct entry into the Earth’s atmosphere and a subsequent ocean landing.

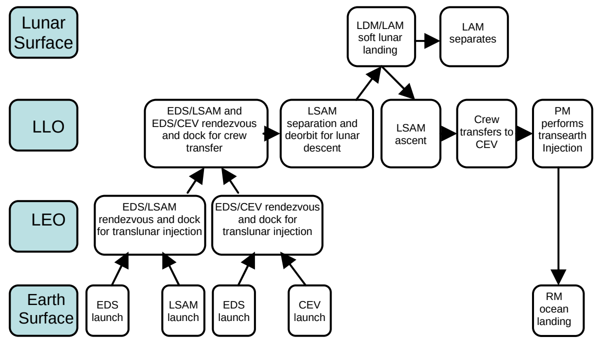

Figure 1: Operational Architectures Flow Chart

2 Requirements

NASA requirements given in Ref. [10-16] were modified to obtain the requirements used in this project. Table I displays how the important requirements varied from the original to the final version.

Table I: Changes in Requirements

Requirements	CSM	CEV	
Crew	3	4	
Diameter	3.9116	4.4958	m
Total ΔV	1951	2588	m/s
Propellant Surplus	5%	10%	
Cargo	138	500	kg
EVA's	0	2	
Uncrewed Days	0	70	days
Crewed Days	14	11.5	days
	LEM	LSAM	
Crew	2	4	
Crew Days	4	7	Days
Landing ΔV	4010	4385	m/s
Return Cargo	0	250	Kg

3 Sizing

In sizing the CTS, two Microsoft Excel workbooks were developed that would size the CTS as a whole. The first spreadsheet developed was for mass sizing. From the mass sizing spreadsheet, geometry sizing was derived for the corresponding masses. The following sections will detail the approach used in sizing with these two spreadsheets.

3.1 Mass Sizing

The mass sizing tool related the various subsystem and component masses to the inputs that would logically affect each of these subsystems and components for the original Apollo elements. For example, the mass of the crew couch in the RM is the Apollo crew couch mass divided by the number of crew on Apollo, 3, and multiplied by the number of crew in the new requirements, 4.

A similar method was used to change the number and types of engines, power supply, and other technologies for each of the elements. The tool was designed to size the Apollo elements when the Apollo requirements and technologies are input. Each of the original Apollo element masses came from Ref. [16]. The underlying assumption of this mass sizing tool is that any architecture designed includes a re-entry module attached to a propulsion module and a two stage lunar landing module. The EDS was sized using the Space Propulsion Sizing Program (SPSP)[3] which allows for detailed conceptual design of in-space stages.

The process of designing the elements for this architecture began with the Apollo spacecraft and lunar module. One at a time, the new requirements were input to get a design point using old 1960's Apollo technology. The effects of each of the requirements on the mass of the CEV can be seen in Fig. 2. The blue bar is the RM and the maroon bar is the PM. A similar effect can be seen in Fig. 3 for the LSAM. The blue bar is the Lunar LAM and the maroon bar is the LDM.

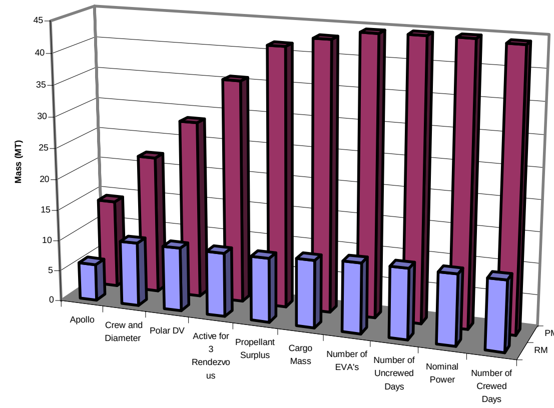

Figure 2: CEV Mass Growth with Requirements

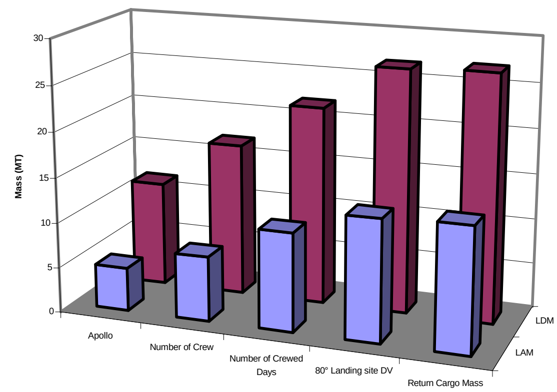

Figure 3: LSAM Mass Growth with Requirements

The most significant increases on the CEV and LSAM masses are the increased crew capacities. Adding one additional astronaut to the CEV two to the LSAM almost doubles the entire system. Additionally, the CEV is greatly affected by increasing the ΔV capabilities.

3.2 Technology

The next step was to modernize the technologies used on the CEV and LSAM. The modernization results required the use of SPSP as well as the mass sizing tool. Figures 4 and 5 show the decrease in mass as each of the new technologies is applied to the CEV and LSAM, respectively.

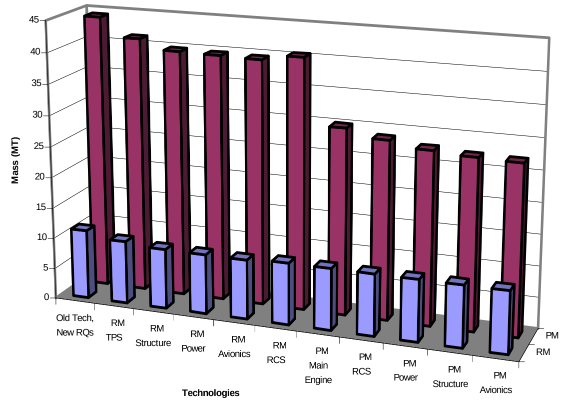

Figure 4: CEV Mass Decrease with Technology

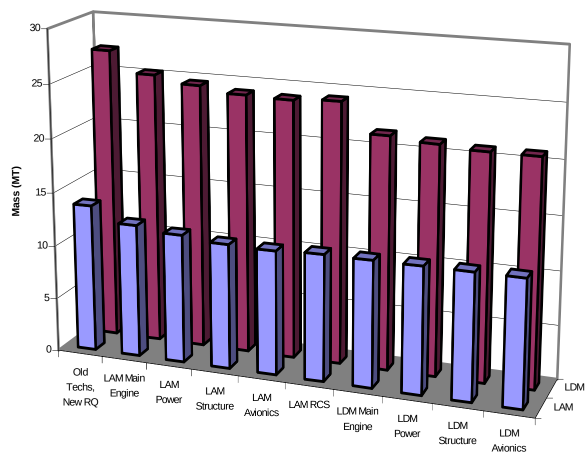

Figure 5: LSAM Mass Decrease with Technology

The RM is impacted most by advances in the Thermal Protection System (TPS) technology, see Table II. Due to the interconnectivity of the RM and PM, mass savings on the RM cause additional savings on the PM. The PM also has significant mass reductions when using a higher performance rocket engine, see Table II. The LAM and LDM have substantial mass savings after switching to better engines. The LDM also has a large reduction when the LAM reduces mass.

The project baseline, with the new requirements and new technologies, is the result of this process. The main features of this baseline are the 5.87 meter diameter, the change to a primary structure of aluminum for all elements, the hydrazine thrusters for the Reaction Control System (RCS) on the RM and LAM, the NK-39 liquid oxygen/kerosene engine on the PM, the gaseous oxygen/kerosene RCS thrusters on the PM, the upgrade to Shuttle Transport System fuel cells, the change to Lithium Ion batteries on all elements requiring batteries, the use of an inflatable airlock for Extra Vehicular Activity (EVA) on the RM, the use of the RD-8 Nitrogen tetroxide/Unsymmetric dimethalhydrazine engine on the LAM and LDM, and modern avionics equipment. Table II outlines the technology changes from the 1960's/Apollo to 2005/final revision.

Table II: Technology for 1960's vs. 2005

Technology	Apollo	Baseline	Benefit (% Mass Savings)
RM TPS	Phenolic Epoxy Resin	AFRSI Blankets with PICA	7.25%
RM Structure	Stainless Steel	Al-Li	3.64%
RM Power	Silver-Zinc	Li-Ion	0.34%
RM Avionics	1960's	2005	0.29%
RM RCS	Storable Bi-Prop	Mono-Propellant	-2.48%
	$I_{sp} = 290$	$I_{sp} = 235$	
PM Main Engine	Storable	Cryogenic	20.37%
	$I_{sp} = 314$	$I_{sp} = 362$	
PM RCS	storable bi-prop	Cryogenic Bi-Prop	3.96%
	$I_{sp} = 290$	$I_{sp} = 265$	
PM Power	1960's fuel cells	Shuttle Fuel Cells	2.15%
PM Structure	Al Honeycomb	Al-Li Honeycomb	1.72%
PM Avionics	1960's	2005	0.39%
LAM Main Engine	Storable	Modern Storable	8.06%
	$I_{sp} = 311$	$I_{sp} = 342$	
LAM Power	Silver-Zinc	Li-Ion	2.88%
LAM Structure	Aluminum	Al-Li	2.33%
LAM Avionics	1960's	2005	0.59%
LAM RCS	Storable Bi-Prop	Mono-Propellant	-1.54%
	$I_{sp} = 290$	$I_{sp} = 235$	
LDM Main Engine	Storable	Modern Storable	7.50%
	$I_{sp} = 311$	$I_{sp} = 342$	
LDM Power	Silver-Zinc	Li-Ion	0.94%
LDM Structure	Aluminum	Al-Li	0.72%
LDM Avionics	1960's	2005	0.01%

3.3 Architecture Change

To accommodate the volume per crew member, as explained in the following geometry section, the diameter was increased, which later the previously generated Solid Edge model. The structure was changed to a lower density material for the majority of each element, but the RM heat shield substructure was left as stainless steel for TPS bonding requirements. The change supported by SPSP to hydrazine thrusters allowed for a simpler system of tanks, although there is a penalty in performance. The change to the NK-39 engine was to increase the Specific Impulse (Isp) while avoiding the problems of liquid hydrogen boil-off; the RCS thruster was changed on the PM to utilize the oxygen from lost boil-off. Each of the power supply systems were updated to more advanced components. The addition of the inflatable airlock decreases the oxygen lost during depressurization to achieve the minimum requirement of two emergency EVA's. For reliability concerns, the LAM and LDM engines needed to be high performance and reliable; the most reliable propellants are hypergolic, and the RD-8 has the highest I_{sp} of that class. The avionics were updated to modern technology standards; the actual masses used were calculated through SESAW. [17]

The EDS's were designed through SPSP to have maximum performance. The EDS for the LSAM, referred to as the LSAM-EDS, was designed for Translunar Injection (TLI) and Lunar Orbit Insertion (LOI) of the LSAM, as in Fig. 1. The EDS for the CEV was only designed for the TLI because the CEV performs its own LOI. Each EDS is capable of an Earth orbit rendezvous since each is launched separately. The CEV EDS with the CEV travels separately from the LSAM EDS with the LSAM. The EDS with the lesser propellant needs was designed to be an off-shoot of the EDS with the larger propellant requirement. As a result, the CEV EDS has tanks that are only 75% filled. The EDS is essentially two tanks, four RL-10B-2 engines, and the necessary structure, power, avionics, and RCS to complete its mission.

The final revisions to the project baseline are based on the geometry, trade studies, and multi-attribute decision making considerations. Table III captures the gross masses, in kilograms, of each element in the evolution of the design process from the original Apollo to Apollo technology with new requirements to the project baseline to the final revision. For the original Apollo, there was only one TLI stage, the S-IVB, and for Apollo with new requirements, a TLI stage was not designed.

Table III: Design Masses in Kilograms

Design	Original Apollo	Apollo w/ new RQs	Baseline	Final Revision
RM	5922	11171	9639	7405
PM	24721	44272	27248	20116
LAM	4795	13737	11748	8785
LDM	11642	27146	21028	12429
CEV EDS	119900	N/A	58765	48614
LSAM EDS	N/A	N/A	75803	52980

3.4 Geometry

The CEV and LSAM were initially sized according to Apollo with the mass sizing tool and Solid Edge. These components had to share similar sizing constraints such as diameter. From this diameter, a link could be established between the PM and RM. This link provided the initial baseline development. From that baseline, the design could then be scaled up or down as dictated by the requirements. Two requirements specifically drove the sizing of the CEV and LSAM, number of crew and habitable volume. This evolution of the design eventually resulted in a 4.5 meter diameter CEV and an equally scaled LSAM.

From Ref. [16], the geometry of the Apollo command module was derived and implemented in a computerized scale model. This geometry was scaled to be a sole function of the diameter of the RM. This diameter then dictated the PM diameter. Reference [16] provided propellant masses for the Apollo service module allowing for the height of the service module to be extracted by sizing the necessary propellant tanks. Apollo used a four tank design, having two fuel and oxidizer tanks and two fuel and oxidizer sump tanks. The four tanks were placed radially in four of six sectors in the service module. The other two sectors were reserved for such components as the RCS fuel tanks and the fuel cells. One pressurant tank was placed directly in the center of the service module. Engine size was taken directly from the actual dimensions of the engine used. The Apollo Lunar Excursion Module sizing was accomplished by using a scaling model provided by Jerry Woodfill from the NASA Johnson Space Center.[19] This scale model allowed for visual manipulation and design of the upcoming revisions.

The baseline was derived from Ref. [10], stating the requirement that, "the CEV shall provide the capability to conduct missions with 1, 2, 3, and 4 crew members onboard." A four crew member CEV had to be designed, thus increasing the radius from 3.91 meters to 5.87 meters. The 5.87 meters was referenced from a study performed by Orbital Sciences Corporation.[20] Using three primary design points, the 5.87 meter diameter, crew size, and mission duration requirement, the masses of the corresponding propellant, pressurant, and RCS consumables were scaled according to Apollo and the accompanying requirements. Tank sizes were then calculated for a variety of different tank configurations (toroidal, four tank, and two tank). The four tank design was selected for center of gravity and packaging space considerations. After a tank model was established, new technology was added to the design reducing propellant mass and tank size. Reduced tank size allowed for changes in the PM height giving a 2.79 meter height for the baseline of the CEV. The LSAM was scaled appropriately to the new masses with no direct changes to the design.

The final revision was derived from Ref. [10] requirements that, “the CEV shall provide a minimum habitable volume of 3.0 cubic meters per crew member,” and “the CEV shall accommodate 1.5 cubic meters of payload.” A model for the RM pressure vessel was then developed through the SPSP tool. The vessel was then loaded with components that matched the estimated stipulation of volume required for a four person crew, as discussed in Ref. [20]. The pressure vessel model was then linked to the RM sizing model so that the diameter could be changed until the habitable volume met the requirement of 13.5 cubic meters. Once this was accomplished, the necessary tank sizing was done using the tank sizing model. This yielded a PM height of 2.71 meters. The evolution from the original Apollo Command and Service Module (CSM) to the final CEV can be seen below.

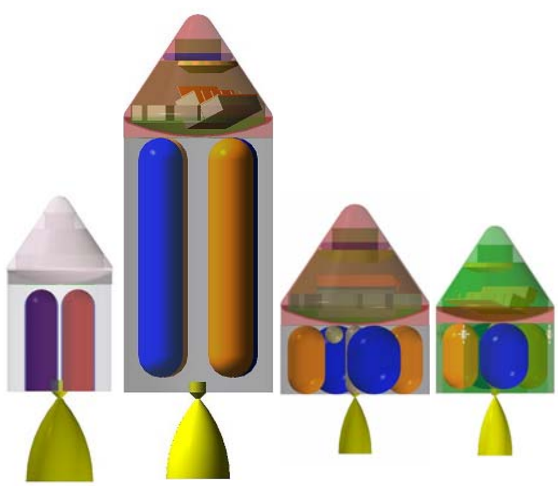

Figure 6: CEV Evolution

The LSAM mass was then recalculated with the mass sizing tool to incorporate the new engine technology of using liquid hydrogen and liquid oxygen instead of kerosene and liquid oxygen. This new propellant required a much larger volume due to the lower density of liquid hydrogen. To have the LSAM remain near the same diameter as the CEV, a new tank design had to be incorporated in the LSAM. Toroidal tanks were then designed for the LSAM because they allow for the largest propellant volume available. Geometry of the LSAM was then adapted to fit these new tanks appropriately. The final dimensions of the LSAM are 4.51 meters wide and 6.56 meters high excluding the landing legs.

Sizing has thus been completed for the CEV and LSAM. All requirements have been met for this design. Also, the current size of this design has been compared to current launch system payload fairings and has been found to fit current day launch systems.

4 Trade Studies

After performing analysis on the baseline, trade studies were conducted, though SPSP and the mass sizing tool, in key areas to improve the baseline. The results of these studies were analyzed through Multi-Attribute Decision Making. The final revision incorporates these results and is detailed in the Summary section.

4.1 Engines

Using the baseline mass sizing tool and SPSP, the PM main engine I_{sp} was varied over a range of 300 to 1000 seconds, while holding the engine thrust-to-weight ratio (T/W) constant at 32.5 and also 65. This operation yielded the following graph of CEV gross mass versus the PM main engine $I_{sp.}$

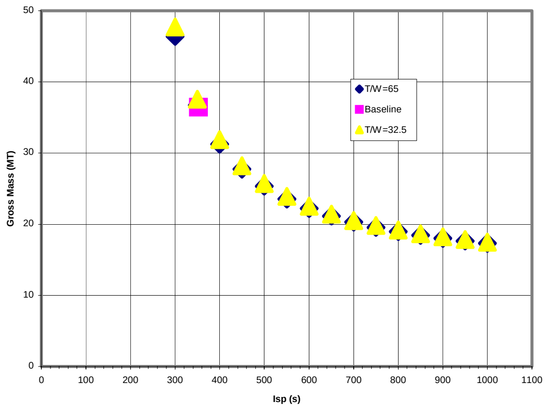

Figure 7: CEV Gross Mass vs. I_{sp}

The curve that resulted was an exponential line as expected due to the use of I_{sp} in the Rocket Equation. The two T/W curves overlap, which demonstrates that the gross mass is not heavily controlled by the T/W ratio. This result is further proven in Fig. 8, which was made by varying T/W from 5 to 150 for I_{sp} values of 350 and 450 seconds.

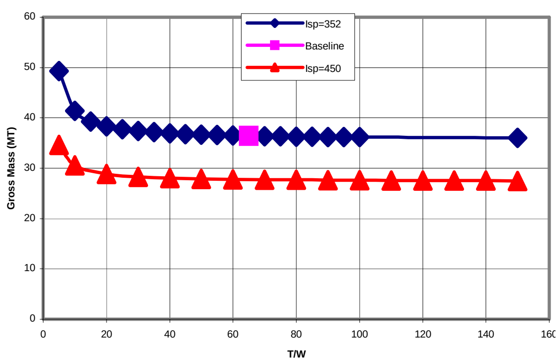

Figure 8: CEV Gross Mass vs. T/W

Changes in engine T/W ratios above 20 have negligible effects; whereas, increases in the I_{sp} reduced mass for all T/W ratios. From these two trades, it is obvious that to minimize gross mass the highest possible I_{sp} should be used with a T/W of at least 20. These two trades were also conducted for the LAM and LDM engines with the same results; it was assumed throughout the trade that the LAM and LDM stages used identical engines. Since nuclear thermal rockets may not be flight ready in time, a liquid oxygen/liquid hydrogen engine in the RL-10 family satisfied the performance criteria outlined in this study. The RL-10B-2 was the best choice because of its high I_{sp} of 462 seconds and T/W of 51.2. This also simplified the design by using of the same engines on multiple elements in the mission.

4.2 *Time of Flight*

The proposed architecture incorporates three notable transfers. First, the LSAM is delivered to LLO where it awaits the crew. The CEV is then delivered to LLO to rendezvous with the LSAM, and finally the crew must return to Earth. By using SPSP and the mass sizing tool, it was seen that these three Times of Flight (TOF) affect both of the EDSs as well as the PM size. As described earlier these stages perform slightly different tasks and carry different payloads. Therefore, by varying the TOF, there was a possibility of evening out the propellant mass required for each EDS.

The first case that was examined was the final TOF in which the PM would perform the TEI in order to return the crew to the Earth. Reducing the TOF reduces the required impulse at injection. This decreases the ΔV requirements on the EDS and PM, exponentially decreasing the mass of the stage through the rocket equation. However, increasing the TOF also increases the crew supplies and Environmental Control and Life Support System (ECLSS) requirements. All of this was taken into account in the mass sizing spread sheet described above. The sizing model could not take into account the psychological and physiological effects that longer TOFs could have on the astronauts. Although the habitable volume requirement is larger than that of Apollo, it is still a small space to be confined for extended periods of time. One solution to this problem would be making habitable volume a function of TOF. This option was not considered because habitable volume was specifically listed as a requirement and as shown in Fig. 9 the relationship quickly levels off at 4 days.

In fact, though it is hard to see in the plot, after 4 days, the mass slowly begins to increase due to the extra crew commodities and ECLSS requirements. From inspection of the plot it can be seen that there is a great mass benefit in a TOF of greater than 3 days with diminishing returns for a TOF of greater than 4 days. The point located at 3.75 days represents a lunarbound TOF of 4 days for which the PM is responsible for the LOI and a 3.5 day earthbound TOF for which the PM performs the TEI. As shown these cross points fit the general trend for half day differences. Differences of greater than a half day were ruled out due to the sharp knee of the plot and the short area of interest (3 to 4 days). As a result of this analysis, combined with engineering judgment to account for psychological and physiological effects it was recommend that since the habitable volume was larger than that of Apollo and the Apollo astronauts did not show signs that an extra half day of travel would have profound negative effects, it was determined feasible to extend both the TLI and TEI to 4 day TOFs. This data and recommendation was then passed on to the Multi-Attribute Decision Making (MADM) process.

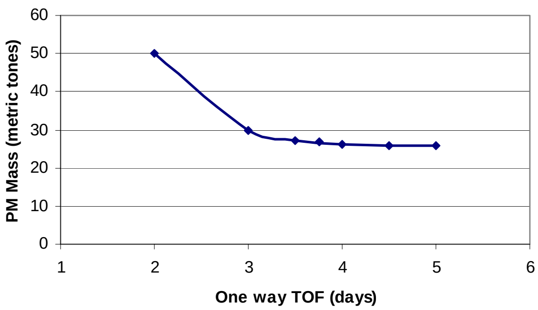

Figure 9: PM Mass as a Function of TOF

Using the Space Propulsion Sizing Program (SPSP) [3] for EDS mass sizing, the ΔVs were varied according to the range of TOF. As seen in the Fig. 10, increases in the TOF have substantial mass savings on the EDS gross mass. The method by which to design the CEV EDS was also evaluated in this study. The first option considered was a CEV EDS identical in every way to the LSAM EDS except with tanks that are not completely full. The other option was a CEV EDS that is completely independent so that the tanks were re-sized for the smaller propellant load. The close proximity of the two curves, varying by about 2 metric tonnes at the longer TOF, and the consideration of the cost of two completely separate EDSs shows that there is not a substantial advantage to designing two completely different vehicles.

For the TOF considerations, it appears that extending the LSAM EDS TOF to the full 5 days is the best choice, particularly since it is not crewed when en route to the moon. However, the CEV EDS must take into account the increase in required consumables for the crew on board, as discussed earlier.

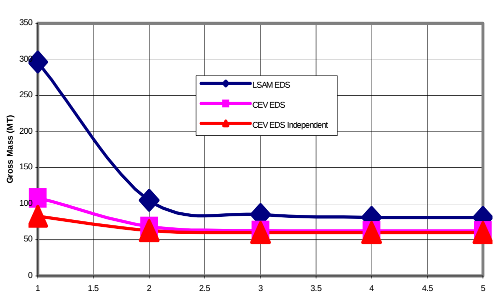

Figure 10: EDS Gross Mass vs. TOF

4.3 Number of EDSs

Using SPSP[3] and the mass sizing tool, the gross mass of the EDSs for 1, 2, 3, and 4 stages for the CEV and LSAM stacks was calculated. The primary assumption was that the same EDS would be used for each stage in the LSAM stack and for the CEV stack. However, for the CEV stack, the EDS will not have full propellant loads due to a decrease in the ΔV requirement of the stack. Figure 11 shows that there is a substantial decrease in gross mass with increases in the number of stages. However, increases in the number of stages logically increase the number of launches. Therefore, a desired effect may be to have a lower gross mass in order to fit the EDS on a particular launch vehicle but at the cost of more launches per mission.

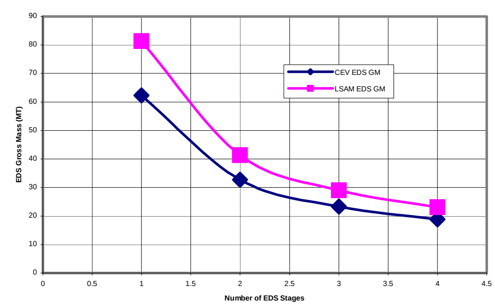

Figure 11: EDS Gross Mass vs. Number of EDSs

4.4 Power

The power analysis was performed using the power module of SPSP. [3] A primary source, which required 7 kilowatts, was selected and a secondary source was added to that, which required 1 kilowatt. Various combinations between the following power sources were considered: Gallium Arsenide solar arrays, Silicon solar arrays, Nickel-Cadmium batteries, Nickel-Hydrogen batteries, Lithium-Ion batteries, Silicon-Zinc batteries, Mercury-Oxide batteries, RTGs, and Shuttle Transportation System grade fuel cells. The driving factor for this design comparison was mass. Based upon this driver, the Gallium-Arsenide body mounted solar arrays and Lithium-Ion battery combination was the best, which resulted in a total mass for the PM and LSAM power systems of 133 kilograms. The second best combination was the body-mounted Gallium-Arsenide solar arrays and fuel cells option. The third most desirable option was the fuel cells and Lithium battery option. These three combinations were considered in further analysis because the fuel cells, though heavier, have the capability to provide drinkable water for the crew, whereas the solar arrays do not, and the water has to be stowed and carried throughout the mission.

The water need for the crew per day is approximately 12 kilograms. The crew is going to be on the PM for 8 days and the LSAM for 7 days, meaning the total water needed is approximately 96 kilograms and 84 kilograms, respectively. Regardless of the power system, 25 kilograms of water will be carried at launch, which drops the PM water requirement down to 71 kilograms. When the water tanks and the total water requirements were added into the equation for total mass, the Gallium-Arsenide solar array- Lithium-Ion battery option still produced the lowest mass.

The power considerations for the RM were calculated differently, mainly because the RM will draw power from the PM until around 3.5 hours before landing. During this short time for re-entry, the RM will rely on three Lithium-Ion batteries.

4.5 Number of Crew

Although the crew size was set by the requirements, the effects of changing crew size was analyzed as a check on the requirements and to determine if a requirement change should be suggested. Again, the mass sizing tool was used to calculate the CEV and the LSAM masses for varying crew size. As shown in Fig. 12, the CEV grows in a quadratic fashion while the LSAM shows a linear relationship. With the required crew size of 4, it is shown that the LSAM is slightly smaller than the CEV. This is ideal for the architecture and sizing the two EDSs. Varying to either 3 or 5 crew would make using the same EDS for both systems impractical. With the extended lunar duration listed in the requirements, having a crew larger than Apollo makes sense in order to allow more specialties to be represented; however, evaluating the benefit gained by adding crew is beyond the scope of this study and for this purpose crew size was assessed on a mass basis only.

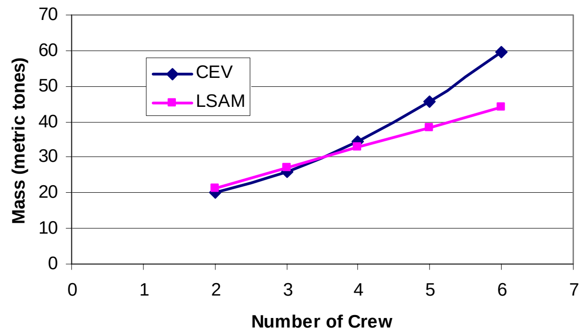

Figure 12: CEV Mass as a Function of Crew Size

5 Trajectory

Aerodynamic Preliminary Analysis System (APAS)[4] was used to determine the primary aerodynamic coefficients of the RM design. These include the coefficients of lift, drag, and pitching moment, which were used in a simulation of the RM's re-entry trajectory into the Earth's atmosphere developed in the Program to Optimize Simulated Trajectories (POST).[5] Various assumptions were implemented in the run of APAS, which will be discussed in further detail below.

For the purposes of this design, only a hypersonic analysis of the RM was required because pinpoint targeting was not necessary to assess heat rate and heat load for heat shield sizing. This is because heat shield size is the most important factor for the design as it comprises almost 30% of the RM's total mass. Thus, APAS was only used to run Mach numbers greater than 4, neglecting all aerodynamics in the supersonic regime and below, which are important for stability and precision landing.

The geometry input into APAS was taken directly from the scaled version of Apollo. For stability and trim, the RM's c.g. location was scaled from the original Apollo Command Module (CM) c.g. location. This was due to the assumption that the packaging of the RM for this preliminary design was imperfect. Thus, the scaled Apollo c.g. location provided a useable estimate that would simulate real world uncertainty. The module was then analyzed for an angle of attack sweep of 110 to 180 degrees measured from the body axis rotated clockwise with the top facing forward.

APAS was initially run for the Apollo CM geometry. From Ref. [18], wind tunnel data for the coefficients of lift, drag, and pitching moment were retrieved and compared to the results given by APAS. The results showed that the use of Modified Newtonian Flow Theory by APAS consistently calculated higher lift and drag coefficient curves than those given by the wind tunnel data. In order to calibrate APAS for application to the new RM design, correction coefficients were introduced for both lift and drag to shift the coefficient curves to match NASA's wind tunnel data. These correction coefficients were 0.93 and 0.94 for the lift and drag coefficients, respectively. An example of this calibration process is illustrated in Figs. 13 and 14.

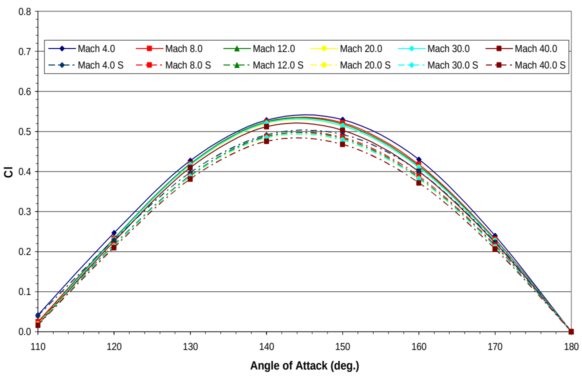

Figure 13: Calibration of APAS Lift Coefficient Curve.

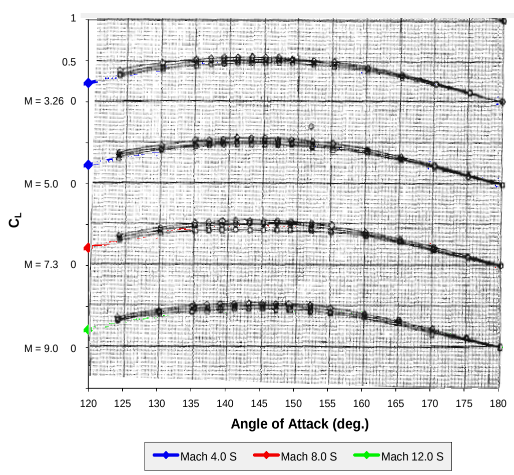

Figure 14: Calibrated APAS Lift Coefficient Plotted Against NASA Wind Tunnel Data.

APAS was then used to calculate aerodynamic coefficients for the 5.87 meter and 4.50 meter diameter designs of the new RM.

With the APAS database complete, all parameters necessary to run the POST simulations were ready for input. In order to provide a valid comparison, a nominal trajectory of the Apollo CM re-entry was first needed. To create a simulation that would model the CM's trajectory as closely as possible, all atmospheric interface conditions as well as the bank angle profile were taken from reconstructed flight data of the Apollo 4 CM,[15] which simulated lunar return conditions. In addition, a generalized acceleration steering algorithm was used within the POST model to control the angle of attack so that the pitching moment would be equal to zero, causing the CM to fly at trim throughout the trajectory. The resulting simulation may be seen in Fig. 15 and is plotted against the actual Apollo 4 flight data.

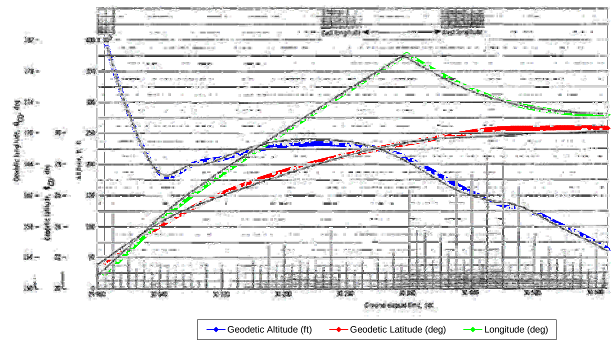

Figure 15: POST Simulated Trajectory of Apollo CM.

With the model complete and validated, the simulation was rerun for the 5.87 meter and 4.50 meter diameter design revisions of the new RM. The input was adjusted to account for the changes in vehicle geometry and aerodynamics, but all other parameters remained the same. The resulting trajectories are shown in Fig. 16.

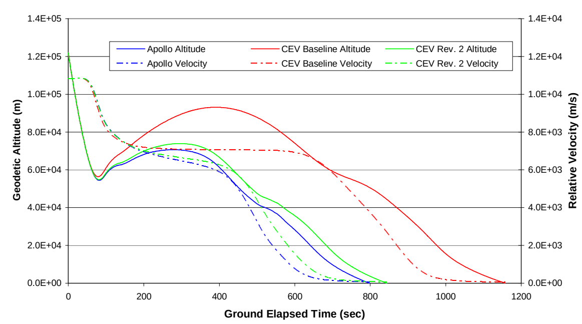

Figure 16: POST Trajectories for Apollo CM and 5.87 meter and 4.50 meter RM Designs.

Using these simulated trajectories, maximum deceleration, maximum heat rate, and total heat load were compared. In reference to the heat rates and heat loads calculated by POST, it should be noted that these calculations are based on Chapman's equations[14] for stagnation point convective heating and do not account for any radiant heating. For these trajectories, the compared parameters are listed in Table IV.

Table IV: Key Parameters from POST Simulations

	Maximum Deceleration (Earth g's)	**Maximum Heat Rate (W/m^2)**	**Total Heat Load (J/m^2)**
Apollo CM	6.9151	2.111×10^6	2.441×10^8
Baseline RM	7.5105	1.513×10^6	1.809×10^8
Final Rev. RM	7.0140	1.925×10^6	2.251×10^8

Based on this trajectory analysis, the final revision of the RM design has been shown to re-enter the Earth's atmosphere at nearly the same conditions as the Apollo CM with only a slight increase in maximum deceleration and reductions in both heat rate and heat load. This RM design was also tested for its maximum and minimum downrange capability. The minimum downrange was restricted by a limit of 20 g's, while the maximum downrange was calculated with bank angle set to zero. The results showed this RM design to have a minimum downrange capability of 1192.3 km and a maximum downrange of 36861 km.

6 Reliability

Using the Excel-based program Quantitative Risk Assessment System: Shuttle Database[22] for Space Shuttle vintage subsystems, the reliability of 39 components, systems, and maneuvers can be determined. Some of the components are batteries, fuel cells, tanks, and displays. Some of the systems described are the propulsion system, electrical system, and communications. The maneuvers covered are rendezvous, docking, and separation. By selecting the applicable components, determining whether reliability is based on cumulative hours or cycles, and indicating the level of redundancy, the program provides the module's reliability. The product of all of the module reliabilities in the mission yields the system reliability. This output is the probability of any failure of the system, and it does not make any assumptions about critical failures.

Using this method, Apollo's reliability was calculated to be 92.5% for the Apollo CSM, LEM and Saturn Upper Stage, S-IVB, which did the TLI maneuver. The Lunar Ascent Module was the element with the lowest reliability at 97.2%, primarily as a result of the rendezvous and docking maneuver with the CSM after lunar ascent. In a close second at 97.9%, the Apollo Service Module had the next lowest reliability; this is also due to the rendezvous and docking maneuvers it had to complete.

This same method yielded 88.6% for the design baseline, which consisted of the CEV, LSAM, and 2 EDSs. At 96.8%, the PM was also the least reliable element of the project baseline; this was again due to rendezvous and docking. The LAM was affected by these maneuvers as the source of low reliability.

For the same number and type of elements, the final revision had slightly lower reliability, of 87.8%, as compared to baseline; although for that miniscule decrease in reliability of 0.8%, the final revision obtained a substantial gain in system performance. It would seem a common trend that the LAM and PM were again the least reliable at 97.1% and 97.6%, respectively. These reliability estimates demonstrated that increased numbers of rendezvous and docking maneuvers enhance the risks of the mission.

Though the reliability calculations did not include any assumptions about the launch vehicle, it is necessary to require that the launch vehicle have a high reliability. To maintain a total mission reliability of at least 85%, the reliability of the launch vehicles for the baseline through the final revision must be 99%. The launcher reliability decreases to 97.5%, if only 80% mission reliability is acceptable. For most launch vehicles, particularly a human-rated vehicle, achieving 97% or more reliability is possible. Table V shows the impact of launch vehicle reliability assumptions on total mission success. Note that the single Apollo launch vehicle approach provides the highest probability of mission success.

Table V: Total Reliability for Various Launchers

	Apollo	Baseline	Final
In Space Reliability	0.925	0.886	0.878
Launchers Reliability	0.99	0.99	0.99
Total	0.92	0.85	0.84
Launchers Reliability	0.98	0.98	0.98
Total	0.91	0.82	0.81
Launchers Reliability	0.96	0.96	0.96
Total	0.89	0.75	0.75

7 Cost

The costing for the CEV, LSAM, and EDS was performed using the NASA/Air Force Costing Model.[7] In addition to mass and technology development, fees, salaries, and other financial considerations may also be taken into account in this tool.

With mass as the primary driver, the costing model provided for a cost comparison between the baseline and final version. Both of these versions were compared to Apollo costs for the same number of missions. Figure 17 displays the comparison between each of these versions for the CEV combination.

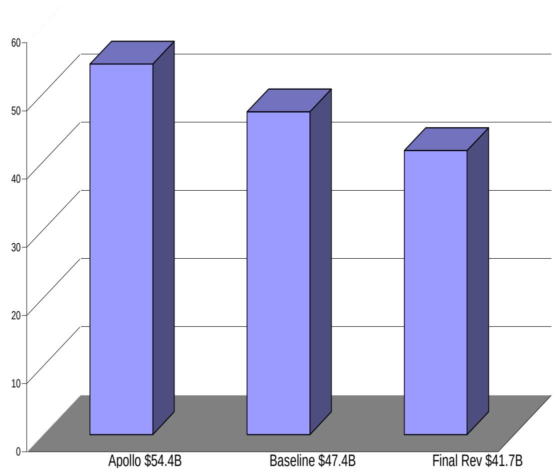

Figure 17: Costing Comparison for CEV

As shown in the figure above, there are drastic differences between Apollo and the final version, even with the extra person that the baseline and final revision included. The Apollo mission, including Design, Development, Test, and Evaluation (DDT&E) as well as 30 flights (design phases A through E), was priced at a total of $54.4 billion. The baseline produced a lower value of $47.4 billion and the final version dropped to a value of $41.7 billion (all dollar amounts are in 2001 US dollars).

The EDS costs were also evaluated. For this project, there were two EDSs taken into account per mission, both were considered and priced as the same amount. The value for the baseline was found to be $10 billion, while the final version was found to cost $9.9 billion. Figure 18 shows the increase in cost for the Apollo S-IVB to the final revision EDS.

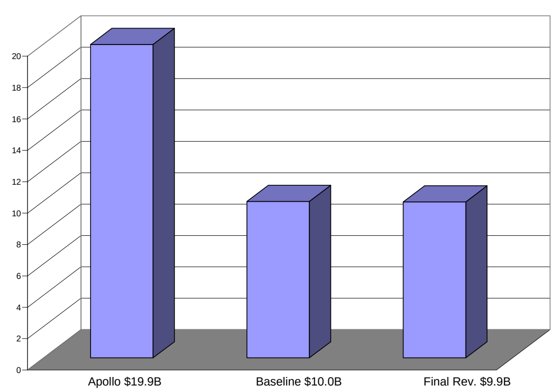

Figure 18: Costing Comparison for EDS

The cost of the LSAM was also obtained through NAFCOM. The baseline price was $32.9 billion, and the final version was found to be $26.2 billion. Figure 19 shows the increase in cost for the LSAM from Apollo to the final revision.

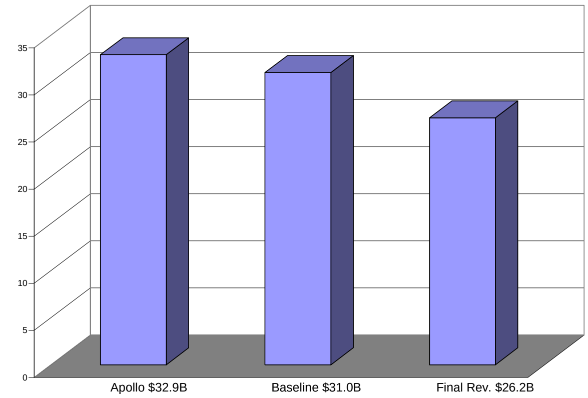

Figure 19: Costing Comparison for LSAM

Another cost consideration was the launch vehicle. The architecture requires 4 launches based on the estimated launcher capability, for either Evolved Expendable Launch Vehicle (EELV) or Shuttle Derived Launch Vehicle (SDLV). An EELV cost is approximately $200 million per launch; whereas, the SDLV has no reported launch cost. For this mission, 4 launches are needed, which would cost approximately $800 million. The costs are significantly lower than the approximate $1.25 billion per launch cost of the Saturn V. This Saturn V launch cost is without the S-IVB and assumes 30 flights and an 80% learning curve.

8 Multi-Attribute Decision Making

With the trade studies completed, the optimum combination of trades was determined using the multi-attribute decision making tool called TOPSIS. This process ensured that the architecture as a whole would be optimized but not necessarily each discipline. In order to do this, a list of Figures of Merit (FOM) was created that looked at each discipline as well as the entire architecture. The list was scrutinized, parsed, and rearranged to ensure that each remaining FOM was independent of all the others. These FOMs were again parsed into a final list of discriminators, throwing out any FOMs that would be too similar over the combination of trades that were considered. This resulted in the six discriminating FOMs in Table VI: Reliability, DDT&E Cost, Production Cost, Flexibility, Extensibility, and Development Risk.

Table VI: Figures of Merit Descriptions

Production Cost	Cost of manufacturing all required element over the lifecycle of the program.
Reliability	Probability of a hardware failure, critical or otherwise.
Extensibility	Applicability and extensibility of technologies, systems, and operations of a lunar mission architecture to other potential exploration missions/destinations.
Development Risk	Applicability and extensibility of technologies, systems, and operations of a lunar mission architecture to other potential exploration missions/destinations.
DDT&E Cost	Cost to design, develop, test, and evaluate all architecture systems to IOC.
Flexibility	Ability of an architecture to increase capabilities to meet evolving mission requirements.

The relative importance of these discriminators was calculated using the Analytical Hierarchy Process (AHP). In the process, AHP calculates a consistency ratio, which is used to ensure that the comparisons are consistent with each other, to eliminate bias, and ensure there were no errors when filling out the inputs. The resulting weightings were then fed into TOPSIS (which is described above) to find the optimum combination of trades based on our FOMs. To ensure a fair calculation, two trials were run. The first used Apollo as the datum point to which all other alternatives were compared, with the second run using the top ranked alternative from the first trial as the datum point.

Both trials gave the same top solution with only one variation between two close rated solutions. With this verification, the optimum combination of trades, as judged by the FOMs, included a liquid oxygen/kerosene propulsion system for the PM with a four day TOF for both the lunar-bound and earth-bound transfers. The LSAM used a liquid oxygen/liquid hydrogen propulsion system and had a lunar-bound TOF of 5 days. Both systems would rely on body-mounted Gallium-Arsenide solar arrays as a primary power source with Li-Ion batteries as a secondary source. The TOPSIS results for the final revision, baseline, and Apollo are shown in Fig. 20.

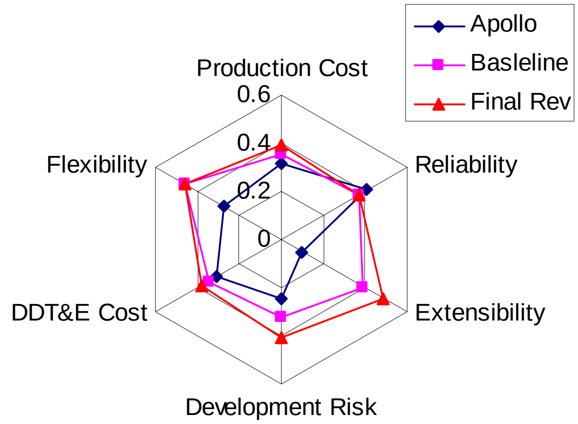

Figure 20: Spider Plot of TOPSIS Results

9 Summary

The final revision of the CTS was created through the use of various aerospace engineering tools and simplifying assumptions. This final revision was a product of trade study results and multi-attribute decision making completed in the areas believed to have significant impact on the design. These results were based primarily on the tools which produced the figures showing mass decrease with technology changes. The main changes in the final revision are the main engines (from SPSP) on the LAM and LDM, power sources (from SPSP) in the PM and LSAM, ΔV requirements and TOF, and CEV diameter (from the mass sizing tool). The RL-10B-2 engine was applied to both stages of the LSAM for increased performance. The PM and LSAM power systems changed (through SPSP) to a combination of solar arrays and Li-ion batteries. The TOF was increased (from historical data) for the LSAM and CEV to decrease ΔV requirements. The CEV diameter was decreased (through the mass sizing tool) due to the substantial excess of habitable volume in the RM. However, due to packaging constraints noted from Solid Edge models, the PM had to revert to the liquid oxygen/kerosene engine since liquid hydrogen occupies a substantial volume.

As has been noted throughout this project, aerospace engineering tools were key to developing a quick, top level design of a human lunar architecture system.

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