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A CONCEPTUAL DESIGN FOR THE SPACE LAUNCH CAPABILITY OF THE PEACEKEEPER ICBM

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ABSTRACT

This paper presents the results of the concept selection, refinement, and verification phases of the conceptual design for the space launch capability of the Peacekeeper Intercontinental Ballistic Missile (ICBM). The redesigned Peacekeeper Space Launch Vehicle (SLV) is intended to serve primarily as a rapid resupply system for the International Space Station (ISS). Using quality engineering techniques, many potential configurations were determined and evaluated based on performance, cost, availability, reliability, safety, commonality with existing space systems, and compatibility with various launch sites. A single concept and its subsystems were selected. Various alternatives for ISS mating; payload module and shroud design; solid rocket boosters; and guidance, navigation and control systems were examined and chosen. This design of the Peacekeeper SLV was verified probabilistically through a trajectory and orbit transfer analysis, as well as a trajectory optimization tool. Finally, the logistics and costs associated with the Peacekeeper SLV program were assessed.

INTRODUCTION

The continuing use of decommissioned strategic missile systems is a concern for the United States military. The original START II treaty called for the elimination of all multiple-warhead ICBMs. If this treaty or a similar one were ratified, the United States would be required to eliminate all operational Peacekeeper ICBMs, as they are capable of carrying up to twelve independently targeted warheads [1]. However, the Peacekeeper is an important strategic asset with the ability for innovative reuse beyond its original mission. With minimal modifications, the Peacekeeper could be altered to serve as an expendable space launch vehicle for ISS or other low Earth orbit (LEO) missions.

To address the concern for strategic missile reuse, this study evaluated the SLV capability of the Peacekeeper ICBM. Potential missions for this vehicle are light payload delivery and rapid ISS resupply and the delivery of commercial or military payloads into LEO. These missions required a focus on three primary goals: minimization of the time to first launch, minimization of development and production costs, and the maximization of useable payload. The proposed design emphasizes cost effectiveness and rapid development, utilizing as much of the existing Peacekeeper as possible and implementing commercial off-the-shelf components when new hardware was required.

The Peacekeeper SLV is a four stage vehicle with the capability of adding two or more Castor IVA solid rocket boosters to accommodate large payloads. All four of the existing ICBM stages were retained for the space launch vehicle. Two types of payload modules were designed due to the additional complexity required for ISS missions versus standard LEO payloads. The payload module consists of the fourth stage, shroud, reaction control system (RCS), and related hardware. For both module types, the guidance system was updated to replace the Advanced Inertial Reference Sphere (AIRS) system of the ICBM with a significantly lighter, less expensive, more accurate, and commercially available system.

The standard LEO payload module has a hammerhead-shape clamshell fairing. This fairing encompasses the payload and fourth stage and separates upon exit from the atmosphere. The payload is guided to its orbital insertion point using the axial engine of the fourth stage. Upon reaching the orbit placement point, the payload detaches from the fourth stage. The fourth

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stage then performs a reentry burn and disintegrates in the atmosphere.

The ISS payload module serves not only as an aerodynamic shroud during ascent, but as a room of the ISS during berthing operations. The module consists of a removable nose cone, payload can, and primary propulsion system. A grapple fixture is housed on the side of the payload can to provide a mating point for the space station remote manipulator arm during berthing operations. Reaction control maneuvers are performed using twenty-four cold gas thrusters located on the forward and aft periphery of the payload can. These thrusters utilize helium gas in order to minimize the temperature of plume impingement on the ISS. Larger burn maneuvers are accomplished using the axial engine of the primary propulsion system. The primary propulsion system is a modified version of the ICBM fourth stage, with the addition of more helium storage volume to fuel the reaction control system and the removal of the original bipropellant Attitude Control System (ACS) thrusters.

By offering two types of payload modules and the option of adding multiple Castor IVA strap-on boosters, the proposed design can accommodate a wide variety of payloads to the ISS or LEO. By maximizing the use of the existing ICBM hardware and off-the-shelf components, the development costs are minimized. The resulting design potentially offers an affordable alternative to existing launch systems, as well as a means of reutilizing a strategic missile asset.

METHODOLOGY

This study used a comprehensive and robust methodology for the conceptual design of the Peacekeeper SLV. The methodology included an Integrated Product and Process Development (IPPD) approach, coupled with response surface techniques and probabilistic assessments. The methodology was a combination of traditional missile and rocket design and a quality engineering approach. The main goal of this method was to design for the most affordable system possible while ensuring technical feasibility and economic viability.

The first step in the design process was to define the problem by mapping the customer requirements to engineering characteristics. A Quality Function Deployment approach, utilizing a House of Quality, was employed. Possible engine and propellant types, as well as staging arrangements, were organized in a Morphological Matrix of design alternatives. Several vehicle concepts from the Morphological Matrix were then evaluated in terms of performance, cost, availability, reliability, safety, commonality with existing space systems, and compatibility with various launch sites with the use of a Multi-Attribute Decision Making (MADM) technique. A Modeling and Simulation (M&S) environment was created so that the design space could be investigated for technical feasibility. This M&S environment concurrently integrated various disciplines, including propulsion; aerodynamics; flight performance; guidance, navigation, and control (GNC); and structures. Ranges were assigned to several significant design variables, and a sensitivity analysis was performed on the responses to see how small perturbations in the design variables would affect the outcome. A parametric study was also performed on some of the assumptions made in the design process so that the exact effects of the estimates on the vehicle concept could be determined. A Response Surface Methodology (RSM) in conjunction with a Monte Carlo simulation was used for these tasks. This methodology was an iterative process and was repeated until both technical feasibility and economic viability were achieved.

A complete description of the methodology employed in this study and its application is presented in Reference [2]. The concept refinement presented in this paper follows from that methodology.

CONCEPT REFINEMENT

Propulsion and Configuration Selection

Two morphological matrices were created for this study. Such matrices allow for the visualization of all possible technology alternatives for a given engineering characteristic. The first morphological matrix (Table I) contained only the propulsion and configuration alternatives, as these were determined to be the main drivers of the system. Table I resulted in sixty-four possible alternative concepts from which one feasible combination was selected with the use of a MADM technique [2].

Table I: Morphological Matrix of Propulsion and Configuration Alternatives

	Configuration Alternatives			
Primary Boost	Current 3 stages	Current 3 stages + strap-on 1st stage boosters	Current 3 stages + commercial 4th stage	2 current stages
Orbit Insertion	Existing PBS	Modified PBS	Small commercial AKM	Large commercial AKM
Attitude Control	Existing PBS	Modified PBS - monopropellant RCS	Modified PBS - cold gas	Commercial RCS

* AKM – apogee kick motor PBS – post boost system

The second morphological matrix contained all alternatives for the subsystems, including those for structures, GNC, range safety, payload delivery, and logistics which were chosen in this concept refinement phase of the design process. These subsystems were dependent on the propulsion system and configuration of the vehicle and therefore were chosen once the final configuration was determined.

Subsystem Selection

ISS Mating

There are currently three types of visiting vehicle mating systems available for ISS operations: the androgynous peripheral assembly system (APAS), the probe/drogue system, and the common berthing mechanism (CBM). The APAS is currently used for the docking system of the space shuttle and the probe/drogue system is used only on the Russian segment of the station. The CBM is intended to be the standard docking system for smaller vehicles. The Peacekeeper SLV is a light payload delivery system, so the system with the lowest weight was selected for this study. The designed Peacekeeper SLV system utilizes the CBM to mate to the ISS and accommodates the attachment of the ISS remote manipulator arm through the use of a grapple fixture. Required equipment to be installed on the vehicle include running lights, a redundant ranging system, a communications system, the passive end of the common berthing mechanism, targets, a grapple fixture, and a hatch. The estimated total weight of the berthing equipment is 900 pounds.

Payload Module and Shroud

The existing Peacekeeper ICBM has a titanium triconic payload shroud which is approximately fifteen feet long. In order to be able to accommodate payloads similar in size to those carried by existing launch vehicles, four shroud shape alternatives were considered.

A preliminary aerodynamic analysis was conducted on these four alternatives. Three of the alternatives were cone-frustum-cylinder shapes with varying cone angles. The fourth concept was an aerospike design. The four shroud geometries were examined using drag estimation equations provided in Fleeman [3]. These equations include wave drag, base drag for powered flight, and skin friction drag. The results indicate very little difference in the three cone-frustum shapes, and a definite increase in drag for the aerospike design. A cone-frustum-cylinder alternative as shown in Figure 1 was selected for this design, as it produced the least drag and provided the largest internal volume.

The same basic shroud shape was used for both LEO and ISS missions. For LEO payloads, a clamshell type fairing is used. For missions to the ISS, the shroud acts as the outer wall of an integrated payload module. The nose of this module detaches from the main body to expose the passive end of the common berthing mechanism and the entrance hatch. A grapple fixture is located on the side of the module. Upon arrival to the ISS, this grapple fixture rotates ninety degrees (90°) from its stowed position to provide an attach point for the remote manipulator arm of the space station. Figure 1 shows a view of the proposed layout of the ISS variant payload module.

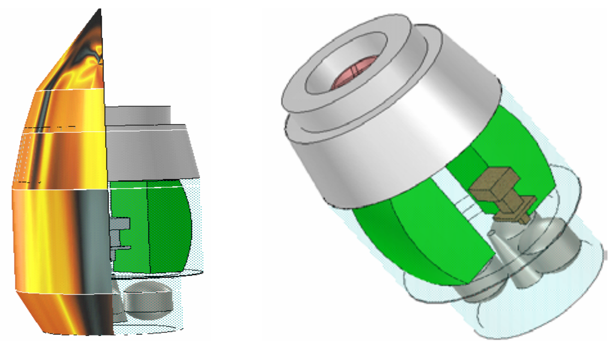

Figure 1: Payload Module Concept

Structural analyses were conducted to attempt to identify the critical failure conditions of the payload module and to provide a preliminary weight estimation of the module. Analysis included the examination of axial and lateral rigidity, critical axial and lateral loads, pressurization conditions, and aerothermal heating effects. At this stage in the design it was difficult to perform a complete structural analysis of the module with adequate fidelity. Lack of data on the classified existing system prevented the use of an in depth structural analysis tool such as a finite element analysis (FEA) code.

Castor IVA Solid Rocket Boosters

In order to maximize the payload weight as well as to minimize the design modifications to the Peacekeeper ICBM, solid rocket boosters (SRBs) were introduced to increase the overall thrust.

By examining existing commercial launch vehicles, two types of boosters were selected as candidates: the Castor IVA and the GEM. The Castor IVA has been used on the Delta II-6925 and the Atlas IIAS. The GEM has been used on the Delta II-7925. Both boosters use HTPB solid propellant, which is the same fuel as the Peacekeeper first stage. Of the two, the GEM is six feet longer and provides more thrust than the Castor IVA. Also, the GEM's graphite-epoxy case is lighter than the steel case used on the Castor. Table II provides a comparison of the two booster types.

Table II: Solid Rocket Booster Comparison

	Gross Weight	Burn Time	Isp_{vac}	Average $Thrust_{vac}$
Castor IVA	25800 lb	56.2 s	265.7 s	108,700 lb
GEM	28600 lb	63 s	273.8 s	110,800 lb

While the GEM has obvious performance advantages, the Castor IVA offers other benefits. The burn time is nearly identical to the Peacekeeper ICBM first stage burn time of 56.5 seconds. Also, due to its shorter length, the Castor IVA booster's upper structural attach point is aligned with the first interstage of the Peacekeeper. The GEM upper attachment point is located near the middle of the second stage motor case. Thus, the Castor offers the advantage of being able to join to the metal interstage structure instead of to a Kevlar motor case. For the above reasons, the Castor IVA was chosen as the SRB for the Peacekeeper SLV.

Guidance, Navigation, and Control

The GNC system in the Peacekeeper ICBM is AIRS, a gyroscopic system with an inertial sphere that floats in a fluid. Although this system is accurate, it is very heavy and would decrease the payload capacity of Peacekeeper SLV. Consequently, a thorough analysis of the different off-the-shelf technologies was performed to reduce the GNC weight while also providing an acceptable level of guidance accuracy to the vehicle.

The most important design constraints regarding the GNC systems were the minimization of cost and the increase in RCS capability as well as increased ISS and payload safety.

The three main functions of the GNC system are to navigate, guide and control. The first of these provides navigation status in terms of position, orientation, velocity and acceleration. This information is transferred to the guidance sub-system, which compares the actual and the planned geometrical variables and sends instructions to the control sub-system. Finally, the control sub-system directs the vehicle to its intended position.

The main goals in the preliminary design of the Peacekeeper GNC system were to size the RCS system and determine the guidance and navigation devices' masses and capabilities, such as maximum acceleration and operating temperature.

Currently, there are two major systems that can be used for navigation in space launch vehicles: platforms and strapdown systems. The former maintains a gimbaled platform in a fixed reference while the vehicle changes its orientation. The strapdown system consists of accelerometers and gyroscopes that move with the vehicle frame and detects rate of change rather than orientation.

Today, off-the-shelf technology is oriented toward strapdown systems mainly because of its lower cost, higher reliability and its compatibility with high accuracy GPS. For this reason, a strapdown system was selected for the Peacekeeper SLV.

There are several different types of strapdown inertial navigation systems (INS) today, including ring laser gyros (RLGs), fiber optic gyros, and micro-electro-mechanical systems. Since an important factor in the design of the Peacekeeper SLV is its minimal time to first launch, ring laser gyros seemed to be the ideal inertial navigation system for the purpose of this project. In addition, Global Positioning Systems (GPS) are the most affordable and readily available technology that can be used for accurate position determination. A GPS was selected for use with the accelerometers and gyroscopes of the RLG to reduce the error of the whole system to less than that of the AIRS system.

Different commercially available strapdown INS systems were evaluated, and the Honeywell GPS/INS was selected. It is made of three RLGs, three accelerometers, and one GPS integrated with a space computer. It is the most accurate system because of the GPS and INS working in concert. It is the best system with regards to the accuracy needed for docking and trajectory guidance.

A cold gas system was chosen for the ISS payload delivery vehicle. In spite of its inferior performance it has a lower cost, greater simplicity, low minimum impulse bit and is significantly safer than other alternatives. All of these advantages justified the choice of a cold gas system for the Peacekeeper SLV RCS. In addition, there was already helium pressurant located in the original post-boost vehicle. Thus, in order to simplify the system, helium was selected for the RCS.

The number of thrusters on the RCS was partially established by the ISS Visiting Vehicle Guide requirements [4]. This document states that all ISS visiting vehicles must have a redundant RCS. A six degree of freedom system was designed to allow translation in all three Cartesian axes without rotation as well as a rotation without translation of the center of mass. Twelve thrusters are required for the six degrees of freedom; however, due to the redundancy requirements for ISS operations the RCS of the

Peacekeeper SLV was designed for twenty-four cold gas thrusters.

The size of the thrusters was determined by calculating the thruster characteristics required to accomplish a pointing window oscillation, a rotation and a translation. A minimum time to achieve each maneuver was established and a thrust curve was created. It was found that the best system to satisfy the requirements was a commercially available cold gas thruster.

The mass comparison between the original and new fourth stages is shown in Table III.

Table III: Final Fourth Stage Mass Breakdown

Old Configuration	Mass
Initial Mass	2,600 lbs
Inert Mass*	1,200 lbs
New Configuration	**Mass**
Initial Mass	2,339 lbs
Inert Mass	939 lbs

* includes AIRS weight of 400 lbs and RCS weight of 50 lbs

CONCEPT VERIFICATION

Trajectory and Orbit Transfer Analysis

A code was developed to simulate a rocket launched from Earth, traveling to ISS orbit. The rocket was forced to follow a predetermined elliptical path and to circularize when a low Earth parking orbit was reached. The input trajectory was based on a quarter of an ellipse, and a flat, non-rotating earth was assumed, which implied that the Coriolis and centrifugal pseudo-forces were negligible. The rocket's motion was evaluated using classical free-body equations of motion.

For the purpose of these calculations, the angle of attack and lift were assumed to be negligible. The thrust and I_{sp} per stage were considered constant. The drag coefficient curve was based on the aerodynamic analysis previously mentioned. Furthermore, no glide time was considered between a stage burnout and its separation.

The change in velocity required to reach the ISS from the parking orbit was computed using the equations governing a Hohmann transfer where the change in speed needed for the transfer is described by Equation (1).

$$\Delta V_{req} = \sum_{n=1}^{\#\text{ of orbits}} \left| V_j - V_i \right|_n \tag{1}$$

In Equation (1), ΔV_{req} is the required impulse for an orbit change, V_j is the speed required for the new orbit and V_i is the actual speed of the vehicle. This expression assumes that the ΔV is an instantaneous burn and that the resulting velocity, once the burn was completed, remained constant throughout the transfer orbit, until an additional burn was initiated. The ΔV that the Peacekeeper SLV can produce was calculated using Equation (2).

$$\Delta V_{available} = -g I_{SP} \ln\left(\frac{m_i}{m_f}\right) - \Delta V_{losses} \tag{2}$$

where m_i is the initial mass, m_f is the final mass and ΔV_{losses} are the total losses in impulse speed (gravity, thrust vector control, drag).

A parametric study was performed by varying four different design variables: structural coefficient of the fourth stage (ε_4), vacuum specific impulse (I_{sp}), drag coefficient multiplier (k_C_d) and ISS final orbit. The structural mass coefficient is defined by Equation (3):

$$\varepsilon_4 = \frac{m_{S_4}}{m_{S_4} + m_{P_4}} \tag{3}$$

where m_{S4} and m_{S4} are the fourth stage structural mass and propellant mass, respectively.

The parameters were varied within a certain range that took into account both degradations and improvements, in order to investigate the complete design space and to observe the tradeoffs between variables. Table IV describes the ranges chosen for each variable in comparison to the baseline value.

Table IV: Variable Ranges

Design Variable	Range	Baseline
ε_4	0.3 - 0.65	0.378
Isp_4 (s)	250 - 308	308
k_Cd	1 - 1.5	1
Final Orbit (nmi)	194.92 – 235.96	235.96

A Design of Experiments (DoE) was constructed using these design variables. A DoE is a mathematically formulated orthogonal table that yields the most information based on a predetermined model with the least number of experiments/simulations. Response surface coefficients and response surface equations (RSEs) are then created based on a multivariate regression analysis for each desired response [5,6,7]. A graphical representation of the RSEs are the prediction profilers. A prediction profiler shows the relative impact of the independent design variables on a given response. The prediction profiler (Figure 2) for this study helped visualize the effects of each design variable on the maximum payload the rocket was able to carry.

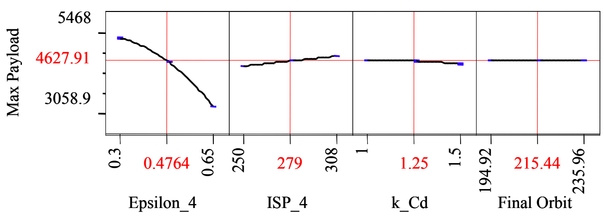

Figure 2: Prediction Profiler

Figure 2 indicates that the structural mass coefficient of the fourth stage (ε_4) was the variable that most affects the payload. The negative slope showed that an increase in ε_4 had a negative effect on the maximum payload, since it reduced the amount of payload the rocket could carry. The I_{sp} of the fourth stage had a minimal positive effect, because most of the propulsive force throughout the mission was performed by the first three stages. The increase in drag had a slight negative effect on the payload, since the Peacekeeper SLV was only in the atmosphere for a very short period of time. Whether the ISS was at apogee or at perigee did not have an appreciable effect.

Using the RSE meta-model, a Monte Carlo simulation [7,8] was performed to determine the probability of carrying a specific payload mass to the ISS and to examine the effects of the design variables on the variability of the response. A uniform distribution was attributed to all of the variables except for the drag coefficient multiplier, which was assigned a triangular distribution with a most likely value of 1.25.

The cumulative distribution function (CDF) in Figure 3 is a result from the Monte Carlo simulation. A CDF is the mathematical function that maps the probability of obtaining a response to the metric within the given range. The variability of the response was large and there was a ninety percent (90%) probability of carrying only 1,250 pounds. Based on the prediction profiler, it was assumed that the range of ε_4 contributed to this wide variability.

Therefore, the range of ε_4 was reduced, and the CDF was plotted on the same graph. The plot was shifted to the right and the base of the curve was narrowed, which means that the variability of the response was reduced. There was a ninety percent (90%) confidence of carrying 2,700 pounds to the ISS and a seventy percent (70%) probability of achieving the deterministic value of 2,800 pounds. In other words, if the structural mass coefficient of the fourth stage is controlled and maintained in the range of 0.3 to 0.35, then the probability of carrying a specific payload can be determined accurately and with high fidelity.

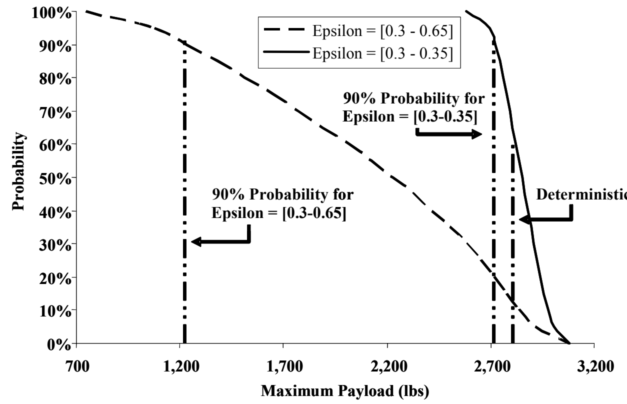

Figure 3: Monte Carlo Simulation Results as Cumulative Distribution Functions

Flight Performance

Optimal Trajectories by Implicit Simulation (OTIS) served as a trajectory optimization tool for the Peacekeeper SLV. OTIS produced higher fidelity results than the previously discussed Trajectory and Orbit Transfer Analysis code; however, OTIS required a larger investment of time and effort. The primary output of OTIS was the useable payload that could be placed in a predefined orbit, but further outputs such as the acceleration and trajectory profiles impacted other areas of the design.

In OTIS, a vehicle is modeled as a group of stages, with each stage having its own aerodynamic and propulsion inputs. Within each stage, there are number of phases. Phases may have no significant physical meaning, but they are necessary because discontinuities can only occur between phases. For example, when the two Castor IVA boosters are separated from the rest of the Peacekeeper SLV, there is an immediate reduction in weight. This weight loss is a discontinuity, as opposed to a continuous loss of mass due to fuel burn, and can only be modeled with a new phase beginning immediately after the Castors are released. There were eight phases in the OTIS model of the Peacekeeper SLV. The mission profile times and events are shown in Table V.

Unlike the model produced in the Trajectory and Orbit Transfer Analysis code, it was assumed the Peacekeeper SLV would proceed directly to the ISS with no parking orbit. This would slightly increase the useable payload but would decrease the size of the launch window. In addition, 100 of the 1,400 pounds of propellant of the fourth stage were kept in reserve for ISS operations. The final mission assumption concerned the three glide phases. Phases 3, 5 and 7 were all assumed to have a duration of five seconds. This number was estimated to be the minimum time necessary to insure the clean separation of the spent stage and the associated interstage; however, because this time was a very rough approximation, a small parametric study was performed to confirm the accuracy of this assumption.

Table V: OTIS Mission Profile

Time (s)	Event
0	Ignition of first stage and 2 Castors
55	Burnout and separation of Castors
60	Burnout and separation of Stage 1
65	Interstage 1 released, Stage 2 ignition
125	Burnout and separation of Stage 2
130	Interstage 2 and shroud released, Stage 3 ignition
205	Burnout and separation of Stage 3
210	Ignition of Stage 4
366	ISS orbit achieved

Deterministic Results

The maximum useable payload was the most important response that resulted from the OTIS optimization. The launch weight of the vehicle was over 250,000 pounds, which included the addition of two Castor IVA SRBs, but after fuel burn and the release of spent stages, only 6,000 pounds were actually placed in a circular orbit at ISS altitude. These 6,000 pounds were not all useable payload because it included the weight of the payload module and docking equipment and fourth stage mass as well as the RCS system. This resulted in a useable payload of over 3,100 pounds that can be delivered to the ISS by the baseline Peacekeeper SLV.

Multi-Booster Analysis

The procedures outlined above were repeated with only slight modifications to analyze the impact of attaching varying numbers of Castor IVA boosters to the Peacekeeper SLV. The results are shown in Figure 4. This plot shows that as the number of boosters increased, the amount of useable payload the Peacekeeper SLV can carry increased as well; however, diminishing returns were seen. For example, the increase in payload between six and eight boosters was significantly less than the increase between zero and two boosters. The other noticeable trend shown here was the difference in useable payload between the ISS (235 nmi) and LEO (108 nmi) missions. This was due in part to the lower final altitude of the LEO mission; however, a larger contributor was the fact that the ISS docking equipment as well as the payload module were not needed for the LEO mission. This led to a significant weight savings which in turn allowed for a much larger payload.

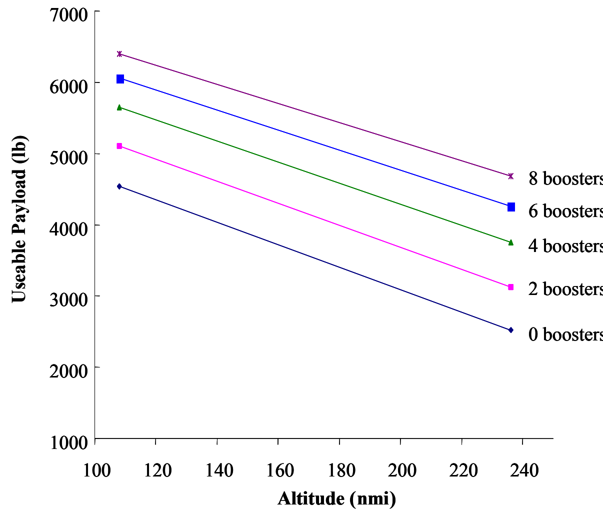

Figure 4: Multi-Booster Results in OTIS

Design Space Exploration

All of the results produced up to this point from the OTIS analysis have been deterministic ones. The vehicle parameters were fixed based on most likely or expected settings. At this stage of the design, it was more likely that certain design parameters would have a range of possible points rather than a single fixed value. In order to evaluate the effect of each of these parameters on the maximum useable payload weight, it was necessary to investigate the design space around these variables. Using the method outlined in Reference 2, a DoE was run in OTIS for a range of settings for six different design variables. These six variables were the structural mass of the fourth stage, the shroud mass, the I_{sp} and mass of propellant of the fourth stage, as well as the ISS orbit and a drag factor. The Peacekeeper SLV with two Castor IVA boosters was used for this study, and the mission was assumed to be the ISS docking alternative.

The results from this DoE are shown in the prediction profiler in Figure 5. The ISS orbit showed almost no impact on the amount of payload. Similarly, the I_{sp} and mass of propellant of the fourth stage had little effect on the amount of payload. These two variables did show the expected trend of increasing payload as the I_{sp} and amount of propellant are increased, but their effect was minimal. This result could have an important influence on any decisions to modify the existing fourth stage to increase performance. Although the maximum payload did decrease as the drag factor was increased, this change was not a large one.

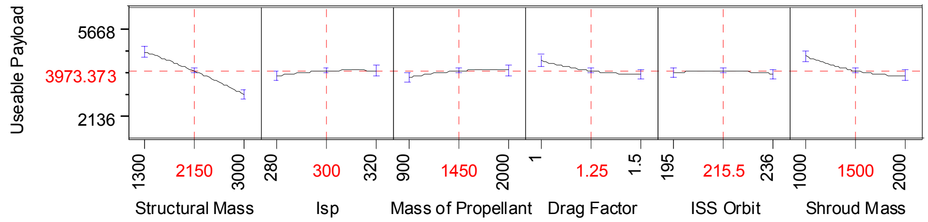

Figure 5: Effect of Design Parameters on Useable Payload

The two design variables that had the most dramatic effect on useable payload were the structural mass above the third stage and the mass of the shroud. Although the Peacekeeper SLV placed the same amount of mass into the ISS orbit, if the structural mass were reduced the amount of useable payload could be increased. A similar trend existed for the shroud mass. In this case the effect was not as pronounced because the shroud was released at the same time as third stage ignition so it was not carried for the entire mission as was the structural mass.

Using a Monte Carlo analysis, a probabilistic assessment of the useable payload was achieved, as shown in Figure 6.

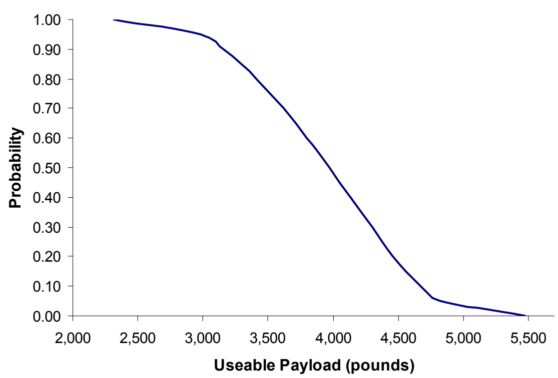

Figure 6: Estimation of Useable Payload

There is a ninety percent (90%) probability that the Peacekeeper SLV will be able to transport at least 3,162 pounds of payload to the ISS. Similarly, there is a ten percent (10%) chance that as much as 4,668 pounds of useable payload could be carried.

<u>Logistics</u>

The logistics analysis examined the details of the operations of the launch system as a whole. This included launch operations, transportation, and range locations. The first of these logistical areas depends upon the launch location. Therefore, the first step was to determine an appropriate launch site for the Peacekeeper SLV program.

Five different types of launch sites, including easterly land launch, westerly land launch, air launch, sea launch, and a "half silo" launch, were investigated. Considerations given to the launch location included environmental constraints, size and weight constraints, ISS orbit inclination, and launch site facilities already in place. In addition, due to the status of the Peacekeeper ICBM as a weapon system, additional considerations and requirements for the launch of the Peacekeeper SLV were necessary.

The first launch possibility investigated was a "half silo" launch. Silos at both F.E. Warren Air Force Base (AFB) and Vandenberg AFB were investigated. The current configuration with two Castor IVA SRBs precludes the Peacekeeper SLV from fitting into any silo currently available. In addition, the removal of the boosters would severely reduce the payload carrying capability of the vehicle. Finally, modification of a silo to fit the Peacekeeper would be expensive and time consuming.

The second launch option considered was a sea launch. The current sea launch platform is in the Pacific Ocean on the equator at 154 degrees West. The current sea launch platform is configured to launch the Zenit-3SL, which is 196 feet tall, over twice the size of the Peacekeeper at 76 feet tall. The Sea Launch User's Guide states that it takes eighteen months for reconfiguration to a new spacecraft type [9]. There is also a ten to twelve day period where the spacecraft is ferried to the launch site from the coast. This would prevent the possibility of this program to function as a rapid ISS resupply. Also, the Peacekeeper is not suited for sea conditions, as it was designed to stay in a climate-controlled environment up through the time of launch.

The third launch method investigated was an air launch. Different launch methods from various aircraft were examined. The most promising was the technique used for the air launch of the Minuteman ICBM. The Minuteman ICBM was released from the back of a C-5 aircraft on October 8, 1974 at 20,000 feet. The maximum payload ever instantaneously released from an aircraft -- 87,320 pounds -- occurred during this Minuteman test. The C-5 holds the record for the most payload ever airdropped -- 190,493 pounds [10]; however, the Peacekeeper SLV weighs about 240,000 pounds so this option was not feasible.

Two locations for a westerly launch were investigated -- Kodiak Island in Alaska and Vandenberg AFB in California. Kodiak Island poses a problem due to the necessity for a controlled climate for the Peacekeeper SLV. Although much of the launch preparation could be done in a controlled climate at Kodiak, there will be a significant amount of time where the Peacekeeper will be exposed to the hostile Alaskan environment.

The Strategic Arms Reduction Treaty (START), which entered into force in December 5, 1994, specifies that any launch vehicle that utilizes the first stage of an ICBM must be launched from a designated space launch facility. Currently, the Vandenberg Space Launch Complex is the only declared space launch facility for the United States. Therefore, Vandenberg AFB is the only available launch site unless other locations are declared to be space launch facilities by the US government. In addition, Vandenberg AFB is well equipped with all infrastructures necessary for the launch operation.

The ideal launch site for the Peacekeeper SLV program would be the Cape Canaveral area, as it can provide an easterly launch. An easterly launch is most beneficial for reaching the ISS orbit, and the facilities at the Cape are more than adequate for the Peacekeeper SLV program; however, at this time Cape Canaveral is not a designated space launch facility, as specified by START.

Cost Analysis

The cost estimation analysis had three main objectives:

1. To model the costs with probabilistic uncertainty;
2. To determine a total program cost and the cost of each of the components;
3. To determine the cost per launch and the cost per pound for comparison to similar systems.

The cost estimation process consisted of a method for the acquisition of numerical estimations for the costs associated with this program. There were four steps to the cost estimation process used for this study. First, an examination of the cost of the entire system and the entire life cycle was conducted. Then, the system level costs were broken down into sub-costs based on historical data of similar sub-costs. Appropriate levels of uncertainty and probabilistic analyses were incorporated into the model. Finally, the system costs were determined using a Monte Carlo analysis.

The first step was to study the costs associated with the life cycle of the Peacekeeper SLV program. Some important considerations included in this step were RDT&E costs, recurring costs, employment costs, disposal costs, overhead, and facilities.

In the second step, the system level costs were broken down into sub-costs based upon reliable data, such as on the current costs of parts or on historical data of similar programs. For example, the Minuteman III Guidance Replacement and Propulsion Replacement Programs (GRP and PRP, respectively) were used as models for the Peacekeeper SLV flight test cost. These two Minuteman Replacement Programs are similar to the Peacekeeper SLV program in that they also performed two flight tests to assess the newly added hardware. Such program-based costing was the foundation of the Peacekeeper SLV program cost analysis.

In addition, the final Peacekeeper SLV configuration, such as the number of Castor IVA SRBs required, was the determining factor for many of the costs. For example, the addition of two Castor IVA SRBs to the Taurus XL launch vehicle increased the cost by about $4 Million USD (all costs were centered to FY 2002 USD for comparison). This was assumed to be comparable to the expected increase in cost for the Peacekeeper SLV system, as well. Other costs were based primarily on an anticipated sixty launches, as there are approximately sixty Peacekeeper ICBMs available for conversion to SLVs. These launches were designed to occur over a six year period, because the Peacekeeper SLV was designed as a short-term solution to the problem of ISS resupply.

Using the above process, deterministic costs were found and then uncertainty was added to the cost estimation model. This uncertainty was modeled with a Monte Carlo analysis. The main types of uncertainty were scheduling and variation from historical program and component cost data.

The ninety percent (90%) confidence values are listed in Table VI. The total cost per launch assumed an expected sixty launches, and the total recurring costs were equal to the price of sixty launches. The RDT&E cost was the least certain estimation in this model, as its standard deviation was the largest of any of the sub-costs. A more certain RDT&E estimation would require a finer breakdown of the subcomponents and additional historical information with which to compare them.

Table VI: Summary of System Costs

One Time Costs		Standard Deviation
RDT&E	$170.70 M	$25.30 M
Launch Setup Equipment	$22.90 M	$8.70 M
Program Closing Costs	$3.20 M	$0.10 M
Total Recurring Costs		
Refurbishment	$479.10 M	$25.25 M
Launch	$180.40 M	$8.70 M
Total Diminishing Costs		
Storage	$9.80 M	$0.76 M
Total Cost	$785.50 M	$36.45 M
Total Cost per Launch	$13.09 M	

CONCLUSION

This paper described the results of the concept selection, refinement, and verification phases of the conceptual design for the space launch capability of the Peacekeeper ICBM. A structured and robust methodology was used to provide an organized approach to this problem. Numerous propulsion configurations were analyzed and a final concept was selected. Appropriate subsystems were examined and chosen, including those for ISS mating; payload module and shroud design; solid rocket boosters; and guidance, navigation and control systems. A confirmation of the Peacekeeper SLV design was performed probabilistically through a trajectory and orbit transfer analysis and the trajectory optimization tool, OTIS. Finally, program costs were assessed with the use of a Monte Carlo simulation and possible launch locations were investigated. The Peacekeeper SLV presented in this paper could offer an economically viable alternative to existing launch systems, while reusing an important strategic missile asset.

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