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Design Optimization of a Space Launch Vehicle Using a Genetic Algorithm

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This paper describes an effort to optimize the design of an entire space launch vehicle to low Earth (circular) orbit, consisting of multiple stages using a genetic algorithm with the goal of minimizing vehicle weight and ultimately vehicle cost. The entire launch vehicle system is analyzed using various multistage configurations to reach low Earth orbit. Specifically, three- and four-stage solid propellant vehicles have been analyzed. The vehicle performance modeling requires that analysis from four separate disciplines be integrated into the design optimization process. The disciplines of propulsion characteristics, aerodynamics, mass properties, and flight dynamics have been integrated to produce a high-fidelity system model of the entire vehicle. In addition, the system model has been validated using the existing launch vehicle data. The cost model is mass based and uses extensive historical data to produce a cost estimating relationship for a solid propellant vehicle. For the design optimization, the goal is for the genetic algorithm to minimize the differences between the desired and actual orbital parameters. This ensures that the payload achieves the desired orbit. One final goal is to minimize the overall vehicle mass, thus minimizing the system cost per launch. This paper will represent the first effort of its kind to minimize the solid propellant launch vehicle cost at the preliminary design level using a genetic algorithm.

I. Introduction

THE U.S. Air Force (USAF) continues to seek assured and affordable access to space. The current USAF vision for achieving this capability is called Operationally Responsive Space (ORS). One goal of ORS is to produce a launch vehicle capable of launching a 1000 lb payload into low Earth orbit at a cost of under \$5 million within 24 hours of tasking.

To support ORS, the Defense Advanced Research Projects Agency and the USAF are "jointly sponsoring the Force Application and Launch from CONUS (FALCON) program to develop technologies and demonstrate capabilities that will enable transformational changes in global, time critical strike missions."¶ The goal of this program is to design a launch vehicle with a prompt global strike capability.

Significant work has been done in recent years to advance the design and analysis of entire launch vehicle systems. Numerous efforts have resulted in the development of legacy codes that analyze the performance of solid and liquid propellant rockets. In addition, an aerodynamics model, a mass properties model, and a six-degree-of-freedom (6-DOF) flight dynamics simulator have all been used. In 1998, Anderson [1] assembled these performance codes and created an objective function that could analyze the performance of an entire, single-stage solid propellant rocket vehicle. In addition, Anderson wrote the genetic algorithm (GA) that was used in the design optimization of this single-stage solid propellant rocket vehicle. Using this GA and the objective function, in 2003 and 2006, Burkhalter et al. [2–4] took the design optimization of missile systems further with additional optimizations.

In recent years, evolutionary techniques have been used to solve a myriad of design optimization problems [4–6]. Significant research has been performed in rocket-based vehicle design optimization using various evolutionary techniques [7–18]. The cost has also been considered in some studies [19–21].

Thus, the foundation work necessary to use these performance codes and the GA to pursue the design optimization of space launch vehicles has been completed. In addition to analyzing the performance of each space launch vehicle, a cost model has been written to bring an economic factor into the optimization process.

II. System Modeling and Optimization Algorithm

Modeling of the space launch vehicle consists of employing a suite of performance codes that are based on the physical models for the propulsion system, the mass properties, the aerodynamics, and the vehicle flight dynamics. All critical vehicle performance parameters are calculated using these performance codes. These performance codes form the basis for an objective function that interacts with the GA to produce possible design solutions. The results are used to determine how well each particular space launch vehicle meets the desired goals of the design optimization.

A. Genetic Algorithm

For this study, a tournament-based GA is used to control the design process. Specifically, the IMPROVE 3.1© GA, developed by Anderson [22] is configured to interact with the various physical models that make up the objective function. The IMPROVE 3.1© GA uses the biological concept of generational adaptation to optimize a design problem that may contain numerous local optima

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¶Data available online at System Description: "Force Application and Launch from CONUS (FALCON)," http://www.globalsecurity.org/space/systems/falcon.htm [retrieved 4 August 2007].

[22]. Rather than define the first iteration, the user specifies a range (maximum, minimum, and resolution) for each design parameter, and the GA randomly generates a population of candidate solutions from the parameters within those ranges. After each candidate is analyzed by performance codes, the GA ranks the candidates (members) in order of fitness, or how closely they match the objective function. The tournament method is then used to create the next generation of members, which will possess characteristics of the previous population but in different combinations that may result in better overall fitness. Over the course of many generations, the GA will find solution types that increasingly approach the target fitness.

The performance codes analyze each member generated by the GA to determine the overall performance of the vehicle. This allows the GA to find solution types that not only meet the target fitness but are also realistic.

B. Propulsion Models

After the GA has generated a set of design parameters, the first performance code to be used by the objective function is the solid propellant propulsion system model. This code analyzes the basic thrust characteristics of a solid propellant motor. Specific solid propellant properties are preloaded, and additional properties are generated by the GA. This information is then used to calculate the propulsion characteristics, such as thrust, burn time, grain mass, and combustion pressure. For example, the grain geometry values of a solid propellant motor can be specified by the GA. From these values, the entire thrust profile of the motor is determined in the solid propellant propulsion system model. The development of this model was pioneered in the research performed by Hartfield et al. [23].

For a multistage vehicle, the number of times the solid propellant propulsion system model is used corresponds to the number of stages in the space launch vehicle. The propulsion characteristics of each stage are determined separately and in sequence.

C. Mass and Aerodynamic Properties Models

Upon completion of the propulsion system analysis, the mass and aerodynamic properties are calculated. The process of determining the mass properties of the vehicle involves two sequential steps. First, the mass properties of the individual components of the space launch vehicle are determined. These components can include the nose cone, the payload, an electronics section, the motor casing, the liner, and the nozzle. Next, using the mass properties of the individual components, the overall mass properties of the entire vehicle are determined. These values are then summarized to create a mass properties data file that is used in the 6-DOF flight dynamics simulator. The mass properties model calculates the mass, the location of the centers of gravity, and the moments of inertia for all of the vehicle components along with the entire space launch vehicle itself.

In 1990, Washington [24] developed an aerodynamic prediction package called AeroDesign that is capable of determining the aerodynamic constants of axisymmetric missiles of circular cross section with cruciform wings or fins. Using the AeroDesign package along with the known geometric parameters of the vehicle, the important aerodynamic coefficients are generated. Essential aerodynamic coefficients are determined for a range of flight Mach numbers and angles of attack.

D. 6-DOF Flight Dynamics Simulator

The culmination of the vehicle performance analysis occurs in the 6-DOF flight dynamics simulator. A seventh-/eighth-order Runge–Kutta numerical method is used to integrate the equations of motion. Position, velocity, and orientation of the vehicle are determined at small time intervals throughout the flight. Upon thrust termination, the vehicle flies a ballistic trajectory with the goal of reaching the desired low Earth orbit. Anderson et al. [1,10,11,17,18] performed significant work with this 6-DOF flight dynamics simulator in the design optimization of single-stage, solid propellant tactical missiles.

E. Cost Model

The cost model used in this analysis was derived by Dr. Dietrich E. Koelle [25]. The TRANSCost 7.1 cost model is broken into three submodels. These submodels are the development cost submodel, the recurring cost submodel, and the ground and flight operations submodel. These various submodels employ system-level cost estimation relationships (CERs) to predict cost. In addition, to bring more realism to the cost per launch determination, an insurance cost submodel is included in the cost model. The cost per launch is determined by using the following equation:

$$C_{\mathrm{launch}} = C_{\mathrm{development}} + C_{\mathrm{vehicle}} + C_{\mathrm{flightops}} + C_{\mathrm{insurance}} \tag{1}$$

The CERs are the backbone of the model and provide the cost of a system in a generic unit called the man year (MYr). The reason for using the MYr as the costing unit is that this unit provides firm cost data that are valid internationally and free from annual changes due to inflation and other factors. "For each of the technical systems, a specific CER has been derived that is mostly mass related with the basic form of

$$\mathrm{CER} = aM^x \tag{2}$$

where CER is the cost (MYr), a is the system-specific constant value, M is the mass (kg), and x is the system-specific cost-to-mass sensitivity factor [25]."

The values of a and x in Eq. (2) are determined for specific types of launch vehicle systems using a data fit of the cost-to-mass relationships of a group of similar systems.

III. Model Validation

A. Validation Method

The purpose of model validation is to verify that the results generated by the system model are both reasonable and realistic. How accurately the system model represents a real-world system is very important. The degree of accuracy in the model is directly proportional to the confidence level in the results that the model produces. Thus, model validation helps to provide this confidence in the results.

A method, previously used by Riddle et al. [26], has been employed for this study. This method begins by researching and choosing a launch vehicle that is similar to the system being modeled by the objective function. Next, detailed information on the chosen vehicle is determined and appropriately hard coded into the input locations for the objective function. This information includes the physical size, thrust values, and/or propellant types. The objective function, along with the other design parameters, is then manipulated in an attempt to reproduce the characteristics of the real-world example. Also, to attempt to match the real-world example more closely, a design optimization can be run using the GA. The purpose of this optimization is not to maximize or minimize any particular vehicle performance characteristics but rather to allow the GA to "fine tune" the system model by choosing the remaining unknown parameters so that the resulting vehicle matches, as closely as possible, the real-world example. If the objective function can produce a space launch vehicle that is strikingly similar to the real-world example, given the real-world example's known and GA-determined parameters, then the validity of the model is substantially strengthened.

B. Three- and Four-Stage Solid Propellant Launch Vehicles

For the current study, three- and four-stage solid propellant launch vehicles have been validated against two real-world examples. Given the known parameters for the real-world examples (payload mass, types of propellants, etc.), the system model was manipulated in an attempt to match the physical properties and the performance characteristics of the real-world example. The known parameters used were the payload mass to orbit, the desired altitude and velocity, the individual stage geometry (diameter and length), and the individual stage propellants.

In addition, the use of the propellant mass fraction (f_{prop}) provides an important check on the model validity. The equation for the propellant mass fraction comes from Humble et al. [27] and is given in Eq. (3):

$$f_{prop} = \frac{m_{prop}}{m_{prop} + m_{inert}} \tag{3}$$

where m_{prop} is the mass of the propellant and m_{inert} is the mass of the vehicle or stage minus the mass of the propellant and the mass of the payload (i.e., the dry mass). Also, according to Humble et al. [27], the desired propellant mass fraction for solid propellant vehicles is around 0.90.

The real-world example chosen to validate the three-stage solid propellant launch vehicle was the Minuteman III intercontinental ballistic missile (ICBM). The Minotaur I space launch vehicle (SLV) was chosen to be the real-world example for the validation of the four-stage solid propellant launch vehicle model.

A comparison of the results from the system model validation of the four-stage solid propellant launch vehicle is shown in Table 1. From Table 1, the results show that a good match has been obtained between the validation model and the Minotaur I SLV. The payload was a direct input to the objective function whereas the other parameters were calculated by the various performance models that make up the objective function.

The model produces a space launch vehicle very similar to the Minotaur I SLV. The total weight of the vehicle is about 1700 lb less than the weight of the Minotaur I SLV. Also, the performance parameters for the model closely match the desired orbital velocity and orbital altitude for the Minotaur I SLV.

The subjective nature of vehicle cost is apparent in the cost comparison. Using the published mass values for the Minotaur I SLV and applying the cost model developed for this study, the cost per launch of the Minotaur I SLV is $51.21 million in 2003 dollars. This is roughly about the same as the cost per launch of the vehicle generated by the model ($51.95 million vs $51.21 million). In reality, there is no recurring cost for the first two stages of the Minotaur I SLV because those stages come from surplus Minuteman II ICBMs that have already been built. An adjustment to the cost model to allow for this yields a cost per launch of the Minotaur I SLV to be $29.76 million. Finally, the advertised cost per launch of the Minotaur I SLV is given as $20 million. There is currently no data in an open source format that explain how the $20 million value is determined.

IV. Optimization Results

Design optimization results for three- and four-stage solid propellant launch vehicles are presented here. An initial launch vehicle optimization has been performed to verify that both the optimization process and the problem setup work. After this initial optimization is discussed, the results of the three- and four-stage design optimization are analyzed. For each design optimization that was performed, the GA was configured to run in a non-Pareto format. This format allows for a global parameter set that meets the desired values of the specified design goals.

Table 1 Four-stage solid propellant model vs Minotaur I SLV comparison

Parameter	Validation model	Minotaur I SLV
Payload	738 lbm	738 lbm
Total vehicle weight	78,090 lbm	79,800 lbm
Total vehicle length	64.58 ft	63.02 ft
Total vehicle f_{prop}	0.9185	0.8998
Final altitude	2,425,999 ft	2,430,000 ft
Final velocity	25,002 ft/s	25,004 ft/s
Cost per launch	$51.95 million	$51.21 million
Adjusted cost per launch	——	$29.76 million
Advertised cost per launch	——	$20.00 million

Table 2 Initial launch vehicles mission statistics

Payload mass	1,000 lbm
Launch site	VAFB, CA (34.6° N, 120.6° W)
Launch direction	Due north (0° azimuth or $i = 90°$/polar orbit)
Desired orbital velocity	24,550 ft/s
Desired orbital altitude	2,430,000 ft

A. Initial Launch Vehicle Optimization

The purpose of this initial design optimization was to determine if a three-stage solid propellant rocket could be designed to achieve the desired low Earth orbit. The desired mission statistics for this vehicle are listed in Table 2.

The two goals for this optimization were as follows:

Goal 1: To minimize the difference between the desired orbital velocity (vorb) and the actual velocity of the vehicle (vt1)

Goal 2: To minimize the difference between the desired orbital altitude (altorb) and the actual altitude of the vehicle (alt1)

Note that this first case is merely a demonstration of the feasibility of finding a workable solution using the GA. The design optimization for this three-stage solid propellant rocket uses a population size of 400 members. The optimization is intended to run for 150 generations, but the optimum solution that met both goals was actually achieved by generation 43 (see Figs. 1 and 2). Figures 1 and

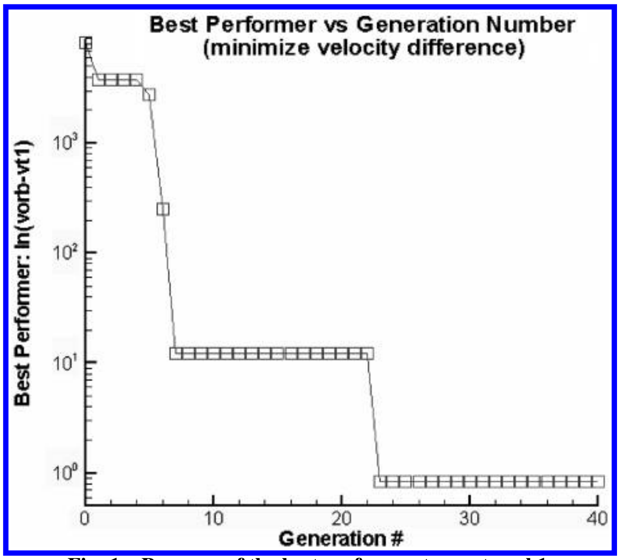

Fig. 1 Progress of the best performer to meet goal 1.

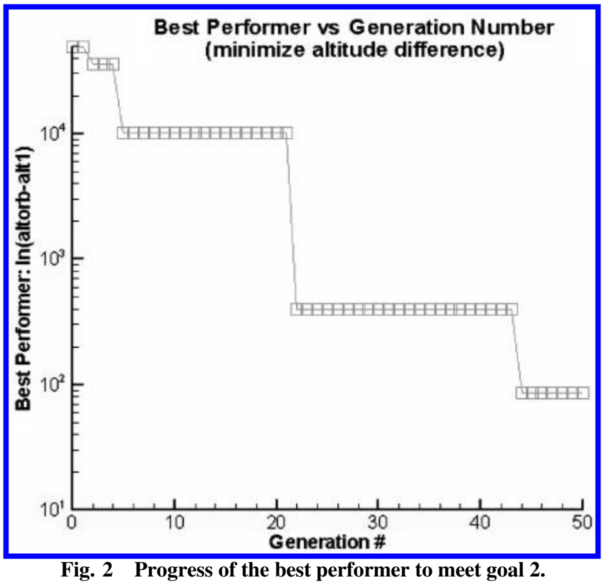

Fig. 2 Progress of the best performer to meet goal 2.

2 show how the best performer improved as each generation proceeded. In Fig. 1, the velocity goal was met by generation 23, whereas in Fig. 2, the altitude goal was met by generation 43.

The best-performing vehicle of the final generation was used to display the actual performance of the optimum design. This vehicle would thus be the three-stage solid rocket that most closely meets the desired performance goals. The design variables (i.e., the 34 GA variables) that created the best performer were run in a single-run format to generate data files, which are summarized in the following paragraphs and figures.

Figures 3–7 show the important performance parameters of this vehicle. Looking at Fig. 3, it appears that the GA selected very similar dimensions for the first and second stages of the rocket and a

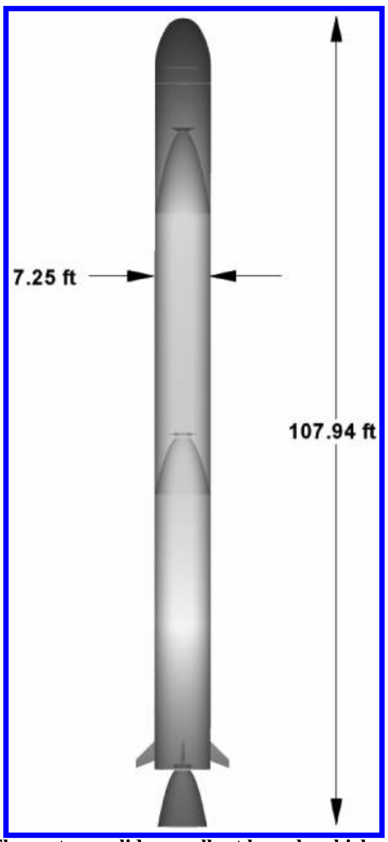

Fig. 3 Three-stage solid propellant launch vehicle schematic.

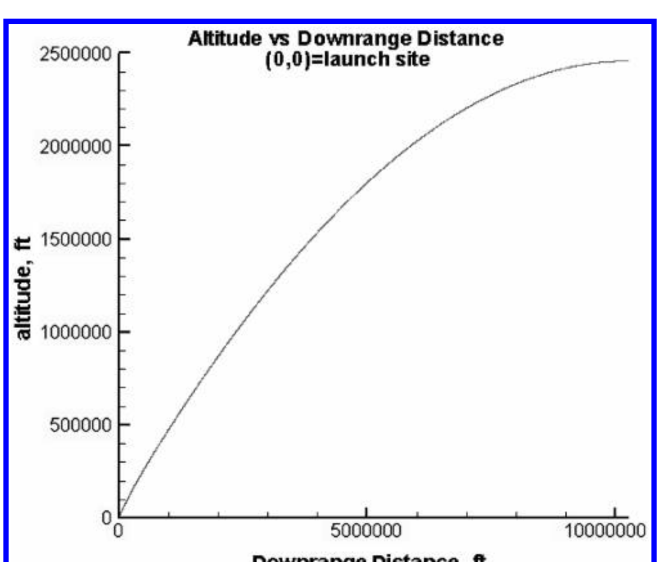

Fig. 4 Altitude vs downrange distance for the best performer.

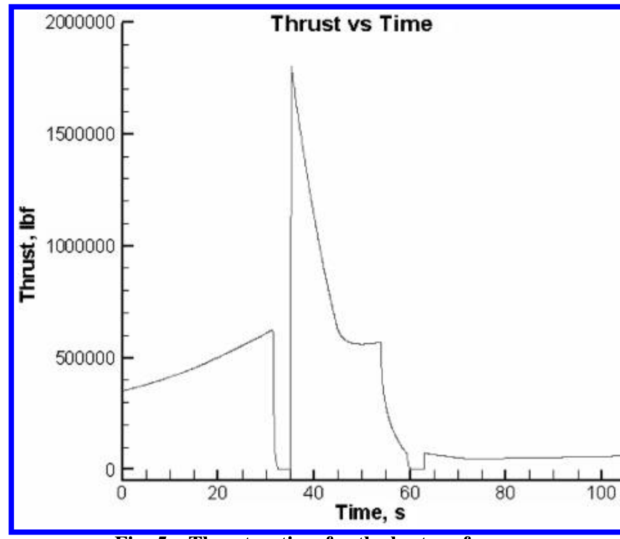

Fig. 5 Thrust vs time for the best performer.

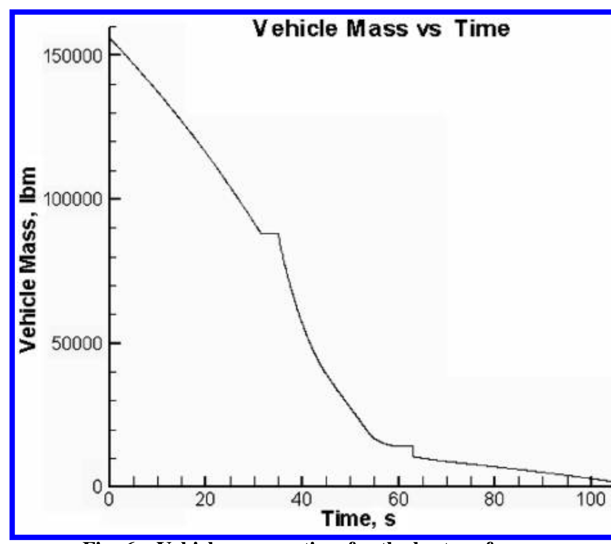

Fig. 6 Vehicle mass vs time for the best performer.

relatively small-sized third stage to meet both design goals. The ballistic flight trajectory of the vehicle is shown in Fig. 4, in which the altitude has reached the orbital altitude at the top of the parabolic trajectory. The next three figures display the changes in the vehicle

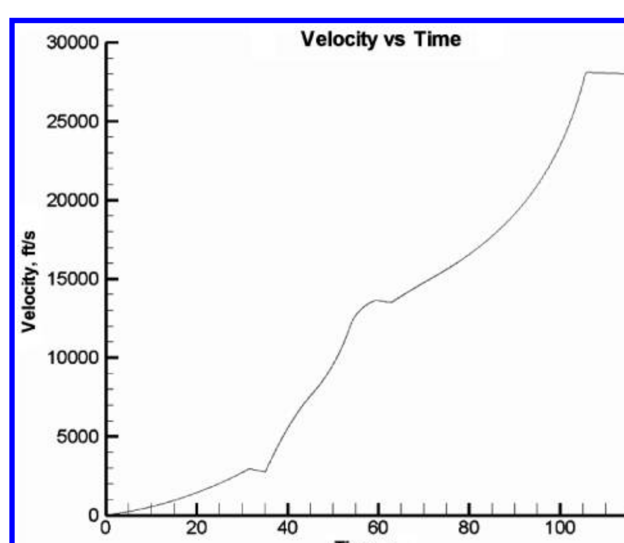

Fig. 7 Velocity vs time for the best performer.

Table 3 Solid propellant launch vehicles mission statistics

	Three-stage solid	Four-stage solid	Minotaur I SLV
Payload	1000 lbm	738 lbm	738 lbm
Launch site	VAFB	VAFB	VAFB
Launch direction	Due north 0° azimuth	Due north 0° azimuth	Slightly northwest
Orbit type	$i = 90°$/polar orbit	$i = 90°$/polar orbit	$i = 97.5°$/sun-sync orbit
Desired orbital altitude	2,430,000 ft	2,430,000 ft	2,430,000 ft
Desired orbital velocity	24,550 ft/s	25,000 ft/s	25,000 ft/s

performance with time. Figure 5 shows the thrust vs time characteristics. Three distinct "humps" in the curve show where the thrust in each stage tails off and then the next stage abruptly starts. There is no delay between the end of one stage and the start of another. Figure 6 (vehicle mass vs time) shows how the mass of the vehicle continuously decreases until the burnout of stage 3 at approximately 107 s. Finally, the velocity vs time plot in Fig. 7 displays the increase in velocity during powered flight and then the slight decrease in velocity during ballistic flight to maximum altitude.

Upon completion of the initial launch vehicle optimization, two additional design optimizations are presented here. These optimizations are for the three- and four-stage solid propellant launch vehicles. Unlike the initial space launch vehicle, the three- and four-stage solid propellant optimizations have built upon the results from the model validation process. The primary focus of these vehicles was to attempt to improve on an existing launch vehicle design. The existing launch vehicle for the three-stage solid propellant optimization was the Minuteman III ICBM. For the four-stage solid propellant launch vehicle, the Minotaur I SLV was used. It is important to note that the Minuteman III ICBM is not an orbital vehicle. As a result, the three-stage solid propellant launch vehicle was optimized to attain a low Earth orbit. Thus, a direct comparison between the two vehicles cannot be made. The desired mission statistics for both vehicles are listed in Table 3. For comparison, the mission statistics of the Minotaur I SLV are also included.

B. Three-Stage Solid Propellant Launch Vehicle

A successful design optimization of a three-stage solid propellant launch vehicle has been performed. Figure 8 shows a schematic of the best performer of the design optimization of the three-stage solid propellant vehicle for a launch out of Vandenberg Air Force Base (VAFB), California.

The important characteristics of the optimized three-stage solid propellant vehicle are shown in Table 4. The final altitude of the vehicle is within 10,000 ft of the desired orbital altitude. The final velocity is slightly above the desired orbital velocity, which ensures that the payload reaches orbit. Thus, the optimized vehicle meets the required orbital parameters, thus reaching orbit and ensuring mission success.

From a mass fraction perspective, the entire vehicle as well as the individual stages closely match the desired value of propellant mass fraction (0.90) given by Humble et al. [27]. The propellant mass fraction (f_{prop}) for the entire vehicle is 0.9072, which is as expected. For each of the three individual stages (stage 1: 0.9139, stage 2: 0.8882, and stage 3: 0.9241), the propellant mass fraction has good values.

The first stage thrust provides the initial high thrust (331,972 lbf) required to lift the vehicle off the ground with a thrust-to-weight ratio of over 3 to 1. The stage 2 and 3 thrust values drop off but still provide the necessary force to get the payload into the desired low Earth orbit.

Finally, the cost per launch of $49.71 million in 2003 dollars is a moderate price for an expendable launch vehicle. It is not exceptionally cheap but it also is not prohibitively expensive. The total vehicle mass of 89,906 lb is the reason for this particular price. This value of the total vehicle mass is an improvement over the initial launch vehicle discussed in the previous section. However, this value is over 10,000 lb higher than the Minuteman III ICBM (79,432 lbm) and the Minotaur I SLV (79,800 lbm). The Minuteman III ICBM is not an orbital vehicle. This would suggest that an additional 10,000 lb is required to enable the launch vehicle to go from suborbital speed (22,000 ft/s) to orbital velocity (24,550 ft/s).

C. Four-Stage Solid Propellant Launch Vehicle

Two design optimizations of the four-stage solid propellant launch vehicle produced optimized vehicles with improved characteristics. The first optimization imposed a limitation on the choice of solid propellants for the GA to use. This limitation involved hard coding the choice of solid propellants to match those used by the Minotaur I SLV. The second design optimization relaxed this restriction and allowed the GA to choose from a variety of solid propellants. Figure 9 shows a schematic comparing the four different resulting space launch vehicles. The upper left-hand image shows the Minotaur I SLV in flight.** The vehicle produced by the validation model is shown in the upper-right-hand image of Fig. 9. An optimized vehicle with the Minotaur I SLV propellants (MP) is shown in the lower-left-hand image and an optimized vehicle with the solid propellant constraint removed is shown in the lower-right-hand image.

A comparison of the different four-stage solid propellant launch vehicles is shown in Table 5. As previously mentioned, the payload was a direct input to the objective function. This optimization had three design goals. The first two goals attempted to minimize the differences between desired orbital parameters and actual performance parameters. As a result, the first goal was to minimize the difference between the desired and actual orbital altitude and the second goal was to minimize the difference between the desired and actual orbital velocity. Finally, the third goal involved minimizing the total vehicle mass. There is significant improvement in both optimized vehicles over the Minotaur I SLV and the validation model. The best performer for optimized vehicle 1 weighs about 7700 lb less than the Minotaur I SLV. This shows that the design can be optimized while restricting the choice of propellants. In addition, the best performer from the design optimization for optimized vehicle 2 weighs 19,000 lb less than the Minotaur I SLV. This is an even greater savings and comes primarily from the difference in the mass of the stage 1 propellants. The stage 1 propellant mass in the optimized vehicle is 30,017 vs 45,371 lb for the Minotaur I SLV. This reduction in the propellant mass was probably also aided by the choice of propellants for the individual stages.

The propellant mass fractions (f_{prop}) for all four vehicles closely match the desired value of 0.90. It should be noted that the both optimized vehicles produce very good propellant mass fractions (1: 0.9185, and 2: 0.8976) while at the same time reducing the total vehicle mass. This adds additional confidence to the system model while at the same time improving the overall vehicle design.

Both optimized vehicles also provide cost savings that result from the reduction in the total vehicle mass due to the design optimization. Optimized vehicle 1 produces a cost savings of $2 million in cost per

**Data available online at System Description: Orbital Sciences Corporation, Space Launch Systems, Minotaur Space Launch Vehicle, http://www.orbital.com/SpaceLaunch/Minotaur/index.html.

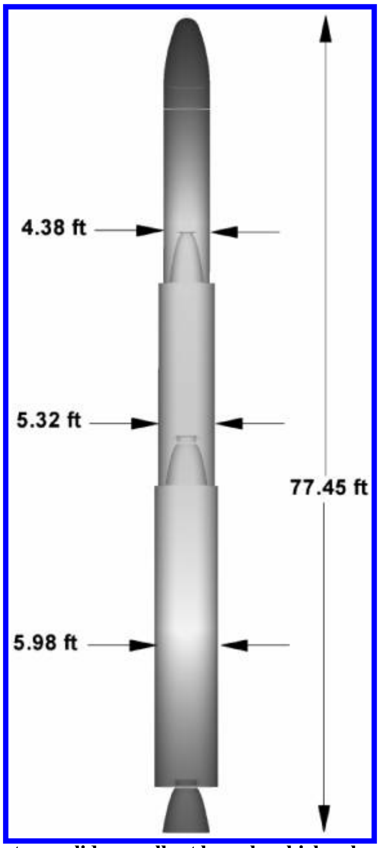

Fig. 8 Three-stage solid propellant launch vehicle schematic (Vandenberg).

launch and optimized vehicle 2 about a $5 million savings in cost per launch over the Minotaur I SLV and the validation model. Using the published mass values of the Minotaur I SLV [28] in the TRANSCost 7.1 cost model resulted in a cost per launch of $51.21 million. The two optimized four-stage solid propellant space launch vehicles yielded a cost per launch of $49.50 and $46.07 million. Although this cost savings is promising, it still does not reach the advertised cost per launch for the Minotaur I SLV of $20 million.

All four vehicles are very similar in terms of physical characteristics. One possible reason for this is that restrictions were placed on the range of values of certain design parameters for the GA. The length and diameter of a particular stage could not be larger than the stage before it. As a result, the GA chose to lengthen the vehicle and slightly reduce the stage diameters. The main difference between the vehicles is that the optimized vehicles have a higher total length than the Minotaur I SLV and the validation model. Because the Minotaur I SLV uses decommissioned Minuteman II ICBMs for its first two stages, the physical characteristics of those two stages are predetermined.

These design optimizations were highly successful in generating improved space launch vehicle designs. Instead of finding a truly global optimum solution (a very difficult task), the GA was able to improve on an existing design. As a result, a good validation of the system model is very important because it lays the groundwork for the design optimization process.

Table 4 Summary of three-stage solid propellant launch vehicle characteristics (Vandenberg Air Force Base)

Entire vehicle		*Stage 1*	
Final altitude	2,439,276 ft	Initial thrust	331,972 lbf
Final velocity	24,595 ft/s	f_{prop}	0.9139
Total vehicle mass	89,906 lbm	*Stage 2*	
Total vehicle length	77.45 ft	Initial thrust	82,093 lbf
Total vehicle f_{prop}	0.9072	f_{prop}	0.8882
Cost per launch	$49.71 million	*Stage 3*	
——	——	Initial thrust	67,192 lbf
——	——	f_{prop}	0.9241

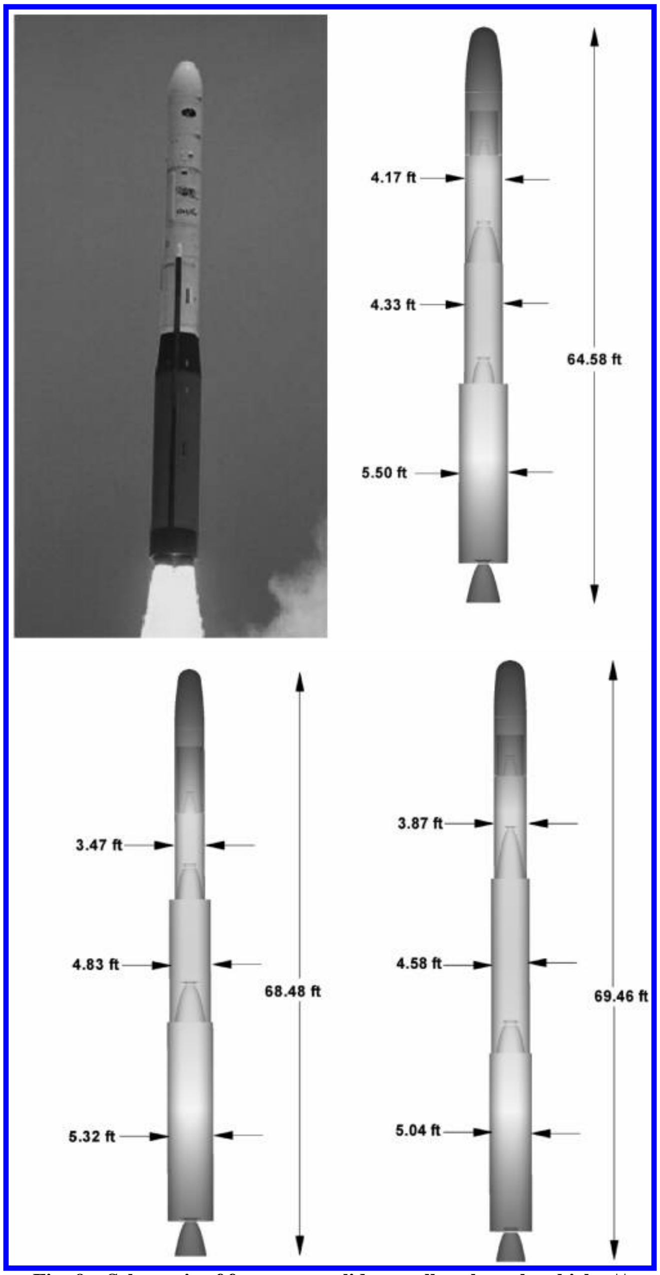

Fig. 9 Schematic of four-stage solid propellant launch vehicles.[**]

V. Conclusions

The design optimization of three- and four-stage solid propellant launch vehicles has been performed. An objective function that modeled the physical and performance characteristics of the launch vehicles was used. A previously developed GA was used to optimize the design. The analysis from these optimizations has shown encouraging results in the ability to model and optimize launch vehicle systems.

For the current study, the GA has been chosen for the design optimization process. One of the main benefits of using a GA is the fact that the algorithm can start without a single point, or guess, to get the optimization running. Direct search and gradient optimization methods all need an initial guess, either a maximum or minimum value, to the problem solution to "march" toward the desired optimized result. Also, derivatives of the functions that describe the system are not required for the GA. This is critical due to the

Table 5 Comparison of four-stage solid propellant launch vehicles

	Minotaur I SLV	Validation model	Optimized vehicle 1 (MP)	Optimized vehicle 2
Payload	738 lbm	738 lbm	738 lbm	738 lbm
Total vehicle weight	79,800 lbm	78,090 lbm	72,069 lbm	60,690 lbm
Total vehicle length	63.02 ft	64.58 ft	68.48 ft	69.46 ft
Total vehicle f_{prop}	0.8998	0.9185	0.9185	0.8976
Final altitude	2,430,000 ft	2,425,999 ft	2,428,909 ft	2,430,505 ft
Final velocity	25,004 ft/s	25,002 ft/s	25,006 ft/s	25,036 ft/s
Cost per launch	\$51.21 million	\$51.95 million	\$49.50 million	\$46.07 million
Advertised cost per launch	\$20.00 million	——	——	——

possibility of a large number of discrete variables being evaluated. The GA does not require that the functions be differentiable in every independent variable. Finally, achieving a global optimum is more likely through the use of this GA. The GA uses a population of guesses that are random and spread throughout the search space. Powerful operators such as selection, crossover and mutation help direct members of each population toward the desired goal(s) of the problem. A binary encoding system allows for a host of variables to be manipulated by the GA and then used in performance codes, called the objective function. The objective function determines the performance of each member of the population, and the GA ranks each member according to how well it meets the desired goal(s).

At the same time, the GA does have some disadvantages. In using the GA, there is a greater likelihood that a global optimum solution will be found. However, finding this global optimum is not guaranteed. Even if the GA is in the neighborhood of the global optimum, there is a possibility through crossover and mutation that the global optimum may not be selected. Also, the GA does not address the robustness of the individual design solutions it creates. The GA simply attempts to meet the desired goals and will adjust the design parameters accordingly. Thus, it is up to the user to ensure the proper operation of the GA and to verify the results it generates. Finally, the satisfactory operation of the GA relies on the accuracy of the system models that make up the objective function.

The entire launch vehicle was modeled using a suite of codes that covered four design disciplines. The propulsion/thrust characteristics were modeled for solid propellant motors. The mass and aerodynamic properties were included in the objective function. Additionally, a 6-DOF flight dynamics simulator was used to fly the vehicle into orbit. Finally, unique to this analysis, a cost model was used to provide the cost per launch information on these vehicles.

The cost model used in this study was the TRANSCost 7.1, which is a mass-based cost model. Because the computations and objective function determine the mass of the various vehicle components, this model was easy to integrate into the optimization process. Also, the TRANSCost 7.1 cost model fits in well with the current effort of the design optimization of space launch vehicles. In the current study, the designs being optimized reflect the optimum solutions at the preliminary design stage. The TRANSCost 7.1 cost model provides a powerful tool to aid in the minimization of vehicle launch costs at this stage of the design process.

Model validation is important in determining the accuracy of the results. For this study, the three- and four-stage solid propellant launch vehicle models were validated against real-world systems. The three-stage model was successfully validated against the Minuteman III ICBM and the four-stage model validated against the Minotaur I SLV. Both validations produced accurate representations of their respective launch vehicles.

The design optimizations have provided good results for three- and four-stage solid propellant launch vehicles. Specifically, the design optimizations of the four-stage solid produced vehicles with some improved characteristics over the Minotaur I SLV. For optimized vehicle 1, the GA used the same propellants as the Minotaur I SLV and produced a mass savings of 7700 lb. With the propellant constraint removed, a mass savings of 19,000 lb was obtained for optimized vehicle 2 while still meeting the desired orbital altitude and orbital velocity parameters.

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References

[1] Anderson, M. B., "Design of a Missile Interceptor Using a Genetic Algorithm," Ph.D. Dissertation, Aerospace Engineering Department, Auburn Univ., Auburn, AL, 1998.

[2] Burkhalter, J. E., Jenkins, R. M., and Hartfield, R. J., "Genetic Algorithms for Missile Analysis—Final Report," Missile and Space Intelligence Center Rept. PAN 20067-02, Redstone Arsenal, AL, Feb. 2003.

[3] Hartfield, R. J., Burkhalter, J. E., Jenkins, R. M., and Metts, J. G., "Genetic Algorithm Upgrade—Final Report," Missile and Space Intelligence Center, HHM402-04-P-0061, Redstone Arsenal, AL, June 2005.

[4] Hartfield, R. J., Jenkins, R. M., and Burkhalter, J. E., "Ramjet Powered Missile Design Using a Genetic Algorithm," AIAA Paper 2004-0451, Jan. 2004.

[5] Budianto, J., and Olds, J., "Design and Deployment of a Satellite Constellation Using Collaborative Optimization," *Journal of Spacecraft and Rockets*, Vol. 41, No. 6, 2004, pp. 956–963. doi:10.2514/1.14254

[6] Lian, Y., and Liou, M., "Multi-Objective Optimization of Transonic Compressor Blade Using Evolutionary Algorithm," *Journal of Propulsion and Power*, Vol. 21, No. 6, 2005, pp. 979–987. doi:10.2514/1.14667

[7] Brown, N., and Olds, J., "Evaluation of Multidisciplinary Optimization (MDO) Technique Applied to a Reusable Launch Vehicle," AIAA Paper 2005-707, Jan. 2005.

[8] Tsuchiya, T., and Mori, T., "Orbital Design of Two-Stage-to-Orbit Space Planes with Airbreathing Engines," *Journal of Spacecraft and Rockets*, Vol. 42, No. 1, 2005, pp. 90–97. doi:10.2514/1.8012

[9] Casalino, L., and Pastrone, D., "Optimal Design and Control of Hybrid Rockets for Access to Space," AIAA Paper 2005-3547, July 2005.

[10] Anderson, M. B., Burkhalter, J. E., and Jenkins, R. M., "Design of an Air to Air Interceptor Using Genetic Algorithms," AIAA Paper 99-4081, Aug. 1999.

[11] Anderson, M. B., Burkhalter, J. E., and Jenkins, R. M., "Intelligent Systems Approach to Designing an Interceptor to Defeat Highly Maneuverable Targets," AIAA Paper 2001-1123, Jan. 2001.

[12] Chernyavsky, B., Stepanov, V., Rasheed, K., Blaize, M., and Knight, D., "3-D Hypersonic Inlet Optimization Using a Genetic Algorithm," AIAA Paper 98-3582, July 1998.

[13] Shelton, J., Frederick, R., and Wilhite, A., "Launch Vehicle Propulsion Design with Multiple Selection Criteria," AIAA Paper 2005-3851, July 2005.

[14] Portz, R., "Launch Vehicle Design Features for Minimum Cost," AIAA Paper 2004-3562, July 2004.

[15] Braun, R. D., Moore, A. A., and Kroo, I. M., "Collaborative Approach to Launch Vehicle Design," *Journal of Spacecraft and Rockets*, Vol. 34, No. 4, 1997, pp. 478–486.

[16] Wollam, J., Kramer, S., and Campbell, S., "Reverse Engineering of Foreign Missiles via Genetic Algorithm," AIAA Paper 2000-0685, Jan. 2000.

[17] Anderson, M. B., Burkhalter, J. E., and Jenkins, R. M., "Multi-Disciplinary Intelligent Systems Approach to Solid Rocket Motor Design, Part I: Single and Dual Goal Optimization," AIAA Paper 2001-

3493, July 2001.
[18] Anderson, M. B., Burkhalter, J. E., and Jenkins, R. M., "Multi-Disciplinary Intelligent Systems Approach to Solid Rocket Motor Design, Part II: Multiple Goal Optimization," AIAA Paper 2001-3494, July 2001.
[19] Wertz, J. R., "Responsive Launch Vehicle Cost Model," AIAA Paper RS2-2004-2004, April 2004.
[20] Unal, R., Braun, R. D., Moore, A. A., and Lepsch, R. A., "Design Optimization for Cost Using Genetic Algorithms," *Proceedings of the 17th Annual Conference of the International Society of Parametric Analysts*, International Society of Parametric Analysts, Chesterfield, MO, May 1995, pp. 183–191.
[21] Olds, J. R., and Lee, H., "Application of a New Economic Analysis Tool to a Two-Stage-to-Orbit RBCC Launch Vehicle," AIAA Paper 1996-4092, Sept. 1996.
[22] Anderson, M. B., "Users Manual for IMPROVE© Ver. 2.8: An Optimization Software Package Based on Genetic Algorithms," Sverdrup Technology, Inc./TEAS Group, Eglin Air Force Base, FL, March 6, 2001.
[23] Hartfield, R. J., Jenkins, R. M., Burkhalter, J. E., and Foster, W., "Analytical Methods for Predicting Grain Regression in Tactical Solid-Rocket Motors," *Journal of Spacecraft and Rockets*, Vol. 41, No. 4, July–Aug. 2004, pp. 689–693. doi:10.2514/1.3177
[24] Sanders, G. A., and Washington, W. D., "Computer Program for Estimating Stability Derivatives of Missile Configurations-Users Manual-Technical Report," Army Missile Command Rept. DRSMI/RD-83-2-TR, Redstone Aresenal, AL, Aug 1982.
[25] Koelle, D. E., *Handbook of Cost Engineering for Space Transportation Systems, Revision 1, with TRANSCost 7.1*, TransCost Systems, Ottobrunn, Germany, 2003.
[26] Riddle, D. B., Hartfield, R. J., Burkhalter, J. E., and Jenkins, R. J., "Genetic Algorithm Optimization of Liquid Propellant Missile Systems," AIAA Paper 2007-362, Jan. 2007.
[27] Humble, R. W., Henry, G. N., and Larson, W. J., *Space Propulsion Analysis and Design*, McGraw–Hill, New York, 1995.
[28] Minotaur I User's Guide, Release 2.1, Orbital Sciences Corporation, Dulles, VA, Jan. 2006.

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1. Mathieu BalesdentNicolas BérendPhilippe Dépincé. 2012. Stagewise Multidisciplinary Design Optimization Formulation for Optimal Design of Expendable Launch Vehicles. *Journal of Spacecraft and Rockets* **49**:4, 720-730. [Citation] [PDF] [PDF Plus]
2. Mathieu Balesdent, Nicolas Bérend, Philippe Dépincé, Abdelhamid Chriette. 2012. A survey of multidisciplinary design optimization methods in launch vehicle design. *Structural and Multidisciplinary Optimization* **45**:5, 619-642. [CrossRef]
3. Francesco CastelliniMichèle Lavagna. 2012. Comparative Analysis of Global Techniques for Performance and Design Optimization of Launchers. *Journal of Spacecraft and Rockets* **49**:2, 274-285. [Citation] [PDF] [PDF Plus]
4. Guobiao Cai, Hao Zhu, Dalin Rao, Hui Tian. 2011. Optimal design of hybrid rocket motor powered vehicle for suborbital flight. *Aerospace Science and Technology* . [CrossRef]
5. 2011. Full issue in PDF / Numéro complet en form PDF. *Canadian Aeronautics and Space Journal* **57**:2, 115-153. [CrossRef]
6. Hanafy M. Omar, Moumen Idres, Raed Kafafy. 2011. Particle swarm optimization of a micro air-launch vehicle trajectory. *Canadian Aeronautics and Space Journal* **57**:2, 115-120. [CrossRef]
7. Roy Hartfield, ; John Burkhalter, ; Mark Carpenter, A Comprehensive Approach to Cataloging Missile Aerodynamic Performance using Surrogate Modeling Techniques and Statistical Learning . [Citation] [PDF] [PDF Plus]
8. Amer Farhan RAFIQUE, Linshu HE, Ali KAMRAN, Qasim ZEESHAN. 2011. Hyper Heuristic Approach for Design and Optimization of Satellite Launch Vehicle. *Chinese Journal of Aeronautics* **24**:2, 150-163. [CrossRef]
9. A. F. Rafique, L. S. He, Q. Zeeshan, A. Kamran, K. Nisar. 2011. Multidisciplinary design and optimization of an air launched satellite launch vehicle using a hybrid heuristic search algorithm. *Engineering Optimization* **43**:3, 305-328. [CrossRef]
10. Amer Farhan Rafique, He LinShu, Ali Kamran, Qasim Zeeshan. 2010. Multidisciplinary design of air launched satellite launch vehicle: Performance comparison of heuristic optimization methods. *Acta Astronautica* **67**:7-8, 826-844. [CrossRef]
11. Mathieu Balesdent, Nicolas Bérend, Philippe Dépincé, Abdelhamid Chriette. 2010. Multidisciplinary Design Optimization of Multi-Stage Launch Vehicle using Flight Phases Decomposition. *International Journal for Simulation and Multidisciplinary Design Optimization* **4**:3-4, 117-125. [CrossRef]
12. Thomas Jazra, ; Michael Smart, Preliminary Design of an Integrated Scramjet-Powered Accelerator . [Citation] [PDF] [PDF Plus]