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Launch and Deployment Analysis for a Small, MEO, Technology Demonstration Satellite

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A trade study investigating the economics, mass budget, and concept of operations for delivery of a small technology-demonstration satellite to a medium-altitude earth orbit is presented. The mission requires payload deployment at a 19,000 km orbit altitude and an inclination of 55°. Because the payload is a technology demonstrator and not part of an operational mission, launch and deployment costs are a paramount consideration. The payload includes classified technologies; consequently a USA licensed launch system is mandated. A preliminary trade analysis is performed where all available options for FAA-licensed US launch systems are considered. The preliminary trade study selects the Orbital Sciences Minotaur V launch vehicle, derived from the decommissioned Peacekeeper missile system, as the most favorable option for payload delivery. To meet mission objectives the Minotaur V configuration is modified, replacing the baseline 5th stage ATK-37FM motor with the significantly smaller ATK Star 27. The proposed design change enables payload delivery to the required orbit without using a 6th stage kick motor. End-to-end mass budgets are calculated, and a concept of operations is presented. Monte-Carlo simulations are used to characterize the expected accuracy of the final orbit. An optimal launch trajectory is presented.

Nomenclature

a	= semi-major axis of orbit, *km*
A_{ref}	= reference area, m^2
ATK	= Alliant Technology Systems
CCAFB	= Cape Canaveral Air Force Base
CEA	= Chemical Equilibrium with Applications (computer program)
CONOPS	= concept of operations
C3	= specific launch energy, km^2/sec^2
e	= orbit eccentricity
E	= separation spring stored potential energy, *joules*
EELV	= evolved expendable launch vehicle
FAA	= Federal Aviation Administration
F_{spring}	= spring force, *Nt*
GFE	= government furnished equipment
GPS III	= next generation Global Positioning Satellite constellation
GUI	= graphical user interface
g_0	= acceleration of gravity at sea level, $9.8067\ m/sec^2$
I	= orbit inclination, *deg.*
I_{sp}	= specific impulse, *sec*
ITAR	= International Traffic in Arms Regulations
k	= spring constant, *Nt/mm*
KLC	= Kodiak Island Launch Complex
GTO	= geostationary transfer orbit
LEO	= low earth orbit
LMA	= Lockheed Martin Aerospace
MEO	= medium earth orbit
M_{final}	= final mass after insertion burn, *kg*
$M_{payload}$	= mass of payload after kick motor jettison, *kg*
M_{prop}	= propellant mass consumed during insertion burn, *kg*
M_{stage}	= mass of expended stage after jettison, *kg*
NRE	= non-recurrent engineering
N_{spring}	= number of springs in Lightband® separation system

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MTO	= MEO transfer orbit
OSC	= Orbital Sciences Corporation
POST	= Program for the Optimization of Simulation Trajectories (computer program)
R_a	= apogee radius, *km*
R_p	= perigee radius, *km*
$R_\oplus$	= local earth radius, *km*
SDL	= Space Dynamics Laboratory
SLV	= space launch vehicle
SMC	= USAF Space and Missile Systems Center
SSBS	= Space Based Space Surveillance mission
TLV	= target launch vehicle
USU	= Utah State University
VAFB	= Vandenberg Air Force Base
WFF	= Wallops Flight Facility
X_{max}	= spring stroke, *mm*
ΔV	= required velocity change during orbit insertion, *km/sec.*
$\Delta V_{available}$	= available velocity change for a given motor loading, *km/sec.*
$\Delta V_{maneuver}$	= equivalent launch velocity due to maneuvering, *km/sec.*
ΔV_{track}	= equivalent launch velocity along track due to earth's rotation, *km/sec*
η	= spring energy storage efficiency
μ	= planetary gravitational constant for earth, $3.986004418\ 10^5\ km^3/sec^2$
w	= argument of perigee, *deg.*
Ω	= right ascension of ascending node*, deg.*
$\Omega_\oplus$	= angular velocity of earth, $7.292115 \times 10^{5}{}_{rad/\sec}$

I. Introduction

Sandia National Laboratory is investigating advanced technologies required for nuclear explosion monitoring sensors to be deployed with the next-generation of Global Positioning System (GPS III) satellites. The next generation GPS constellation architecture will exploit new sensor and signal processing technologies. These emerging technologies must be developed, matured, and space-qualified prior to deployment with the operational constellation. To meet this need, Sandia has proposed a small, free-flying, satellite with a prototype bus architecture. The mission will deploy in a medium earth orbit (MEO) and have a one- to three-year duration. The program has been dubbed *"SandiaSat."* The Utah State University (USU) Space Dynamics Laboratory (SDL) and several of the academic departments within the College of Engineering at USU are assisting Sandia in developing a conceptual mission plan, together with concept of operations (CONOPS), preliminary satellite and ground station designs, launch options, and associated cost, schedule and other programmatic estimates. This paper will address the available launch and deployment options. Fundamental mission objectives require the payload to be delivered to a circular orbit at 19,000 km altitude at an inclination of 55°. At this altitude, the ascending node will precess at approximately 10 degrees/day; however, the orbit right ascension, argument of perigee, and true anomaly (phasing within the orbit) are not critical to mission. The deployment orbit was selected to be a "junk orbit" and post mission de-orbit of both the payload and expended apogee kick stage is not required. Table 1 summarizes the primary orbit requirements for the MEO mission. It is anticipated that the completed SandiaSat system will be available for launch during the first quarter of calendar year 2012.

Table 1. Required Orbit Parameters.

Parameter	*Requirement (Accuracy)*	*Verification Method*	*Design Compliance*
Orbit Altitude	*The Nominal orbit altitude shall be ~19,000km. (±25 km)*	*Verification by launch vendor*	*Expected compliant based on Monte Carlo simulations*
Orbit Inclination	*The Nominal orbit inclination shall be 55° (±0.25°)*	*Verification by launch vendor*	*Expected compliant based on Monte Carlo simulations*

II. Launch Vehicle Selection

Since the orbit requirements are very general, a wide array of launch options is available for payload delivery. Many factors drive the selection of the best commercial launch systems. Among these factors are:

i) *Mission Needs and Objectives,*
   Dictate performance, trajectory, launch site,
ii) *Mission requirements,*
   Orbit altitude, inclination, and right ascension,
   Satellite weight and size,
iii) *Required Launch Date and Availability of Launch Windows,*
   Dedicated or Shared launches,
iv) *Primary Selection Drivers Include,*
   ITAR / Security restrictions,
   Cost,
   Energy requirements (ΔV, spacecraft mass),
   Launch System Reliability,
   Launch System Availability,
   Operations Support,
   Payload Envelope / Form Factor,
   Launch Environments,
   Launch System Interfaces.

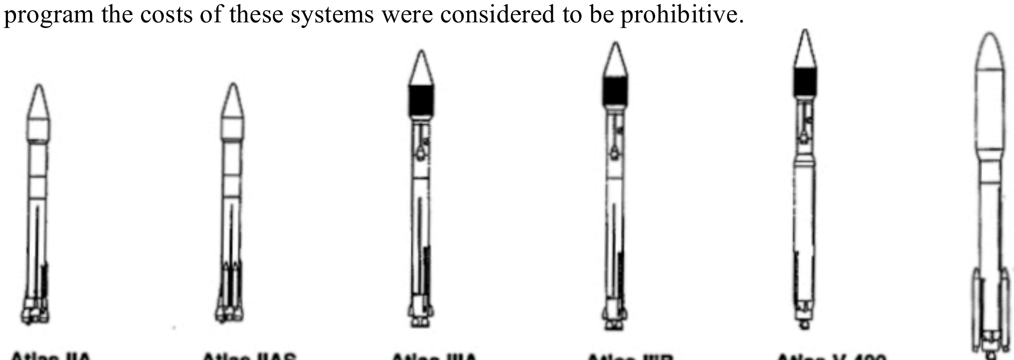
Key in this analysis phase is the selection of a launch system with adequate performance to deliver the payload mass to the required orbit while allowing for sufficient mass margin. The MEO orbit selected for this study is considerably higher than the nominal orbits considered for small launch systems. Typically, for these high altitude orbits, larger medium-lift launch vehicles are the system of choice. However, the small mass of this payload, (< 350 kg) allowed the consideration of small launchers originally intended for Low Earth Orbit (LEO) delivery. For the SandiaSat mission there are three primary constraints on the launch system selection, 1) cost, 2) security, 3) launch site availability. Because the payload is part of a technology demonstration program and not an operational mission, launch and delivery costs are a paramount consideration. The payload includes classified technologies and the use of a USA licensed launch system is mandated. Finally, the high inclination (55°) orbit required for this mission favors a launch from the NASA Wallops Flight Facility *(WFF).*[i] Several launchers can reach this orbit from the Cape Canaveral Air Force Base (CCAFB) test range, but these require a significant inclination change during launch. This plane change results in a decrease in available payload. Additionally, the availability of launch windows for small payloads from WFF is significantly higher than CCAFB, and the cost of launch operations is significantly lower. A preliminary trade analysis was performed to consider every available US launch system. The preliminary trade relied on manufacturers' mission payload charts as well as impulsive ΔV calculations performed using data derived from independently published system information. Systems with launches available from WFF were given priority in the trade analysis. Launches from the west coast test range at Vandenberg Air Force Base (VAFB) are incompatible with the required orbit inclinations, and were not considered for this mission.

A. Preliminary Launch Systems Trade Analysis

Currently, there are 9 Federal Aviation Administration (FAA)-certified launch systems[ii] licensed to carry a classified USA payload. These are (vendor in parenthesis)

- Atlas (United Launch Alliance),
- Athena (Lockheed Martin Aerospace),
- Delta (United Launch Alliance),
- Pegasus (Orbital Sciences Corporation),
- Taurus (Orbital Sciences Corporation),
- Minotaur (Orbital Sciences Corporation),
- Falcon (Space-X),
- Space Shuttle (United Space Alliance, NASA),
- Zenit3SL (Sea Launch Odyssey LTD, Multi-National).

Figure 1 shows the characteristics of the Atlas[iii] family of launch systems including payload to LEO, payload to geostationary transfer orbit (GTO), launch costs in US dollars (*Circa 2002*), date of the first flight, and available launch sites. All of the members of this launch family have medium- or heavy-lift GTO capability, and were primarily designed for large military payloads or for sizeable geostationary communications satellite. They all possess "excess lift" capability and have launch costs that vary from $75-110 million. For this small technology-demonstration program the costs of these systems were considered to be prohibitive.



Vehicle Performance	Atlas IIA	Atlas IIAS	Atlas IIIA	Atlas IIIB	Atlas V 400	Atlas V 500
LEO Maximum	7316 kg	8618 kg	8640 kg	10,718 kg	12,500 kg	Up to 20,050 kg
GTO	3066 kg	3719 kg	4037kg	4477 kg	5000 kg	Up to 8200 kg
Cost	$75–85 million	$90–105 million	$90–105 million	$90–105 million	$75–90 million	$85-110 million
First Flight	1992	1993	1999	2000	2001	2001
Launch Site(s)	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Cape Canaveral	Cape Canaveral	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg

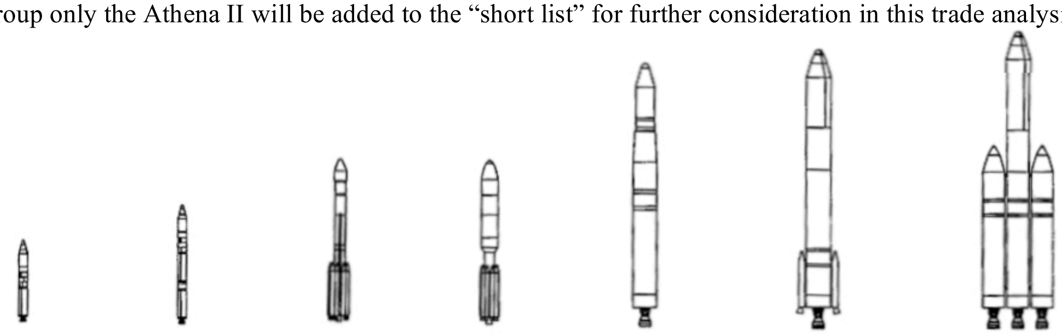
Figure 1. Atlas Series Launch Systems.

Figure 2 shows the characteristics of the Athena[iv,v] and Delta[vi] family of launch systems. Data are shown for payload to LEO, payload to geostationary transfer orbit (GTO), launch costs in US dollars, date of the first flight, and available launch sites. The Delta III and IV systems have excess ΔV capability and launch costs that vary from $85 to 170 million. These costs are considered prohibitive for this mission. Conversely, the Athena I system has insufficient lift capability and was eliminated from consideration. The Athena II and Delta II launch systems have at least minimal lift capacity for the MEO mission, and have moderate costs varying from $22 million (Athena II) to $60 million (Delta II). The main concern with the Delta II launch system is the availability after 2010. An article published by in Wall Street Journal speculates about the fate of the Delta II launch system after U.S. Air Force discontinues its use of the Delta II in 2009[vii] in favor of the Evolved Expendable Launch Vehicle. (EELV).

"The Delta II has been a workhorse of the U.S. space program, which has depended on the rocket and its forerunners since 1960. But the USAF command, confronting mounting war expenses and cuts in space

budgets, have decided they can't afford to continue to help underwrite three Delta II launch pads, associated personnel and other fixed costs. The U.S. National Aeronautics and Space Administration seems unable to shoulder the burden of supporting the launch infrastructure on its own. The retirement of the Delta II system seems likely after the last Air Force payload is delivered in 2009."

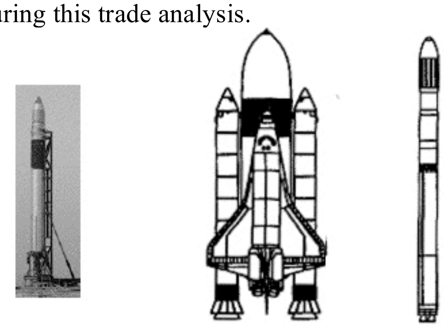
This uncertainly of availability makes the Delta II a high programmatic-risk option. For this single reason the *Delta II, although providing sufficient lift and acceptable costs, will not be considered further in this trade analysis.* From this group only the Athena II will be added to the "short list" for further consideration in this trade analysis.



Vehicle	Athena I	Athena II	Delta II	Delta III	Delta IV Medium	Delta IV Medium Plus	Delta IV Heavy
Performance							
LEO Maximum	820 kg	2065 kg	Up to 5140 kg	8290 kg	8600 kg	Up to 13,600 kg	25,800 kg
GTO	—	590 kg	Up to 1870 kg	3810 kg	3900 kg	Up to 6120 kg	12,400 kg
Cost	$16–17 million	$22–26 million	$45-60 million	$85 million	$75-90 million	$85-110 million	$140-170 million
First Flight	1995	1998	1990	1998	2001	2001	2003
Launch Site(s)	Vandenberg Cape Canaveral Kodiak	Vandenberg Cape Canaveral Kodiak	Cape Canaveral Vandenberg	Cape Canaveral	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg	Cape Canaveral Vandenberg

Figure 2. Athena and Delta Family of Launch Vehicles.

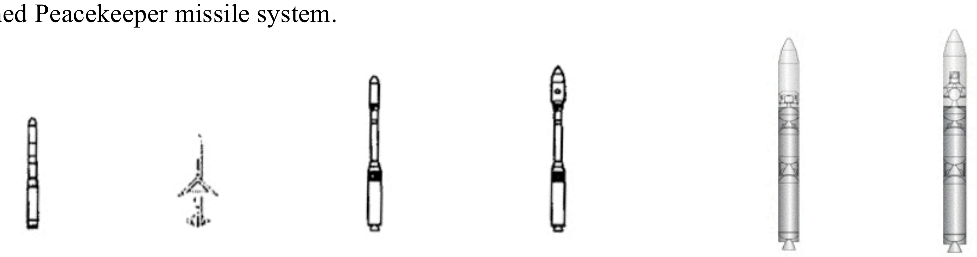
Figure 3 shows the characteristics of the Falcon I[viii], Space Shuttle[ix], and Zenit 3SL[x] launch systems. The Falcon 1 has insufficient lift capacity and is eliminated from consideration. The Zenit costs exceed $75 million, combined with the launch logistics, the Zenit was eliminated from consideration. During the late 1990's, riding along with the Space Shuttle as a secondary payload on one of the ISS supply missions would have been an attractive option. But with the impending retirement of the space shuttle in 2010, the shuttle was eliminated from consideration for this mission. The larger Space-X Falcon 9 medium-lift launch vehicle[xi] is not considered to be of sufficient maturity to be recognized during this trade analysis.



Vehicle	Falcon 1	Space Shuttle	Zenit 3SL
Performance			
LEO Maximum	200-700 kg	24,400 kg	15000 kg
GTO		—	5000 kg
Cost	$7 - 10 million	$300 million	$75–95 million
First Flight	2006	1981	1999
Launch Site(s)	Vandenburg Kwajalein	Kennedy Space Center	Sea Launch Odyssey

Figure 3. Falcon 1, Space Shuttle, and Zenit 3SL Launch Systems.

Finally the family of launch vehicles from the Orbital Sciences Corporation (OSC) was investigated[xii,xiii,xiv]. These launch systems are shown in Figure 4. The Minotaur I, and Pegasus XL launchers have insufficient lift capability. The Taurus XL, Minotaur IV, and Minotaur V Launch systems all have GTO capability with costs varying from $18 million (Taurus) to $28 million (Minotaur V)[‡]. These costs are considered to be within the scope of the allowable programmatic costs and all three systems were added to the "short list" for launch consideration. The Taurus XL is the commercially available option form the Taurus Launch system. The Minotaur IV and V launch vehicle have LEO, GTO, and escape energy configurations and variants of the same vehicle derived from the decommissioned Peacekeeper missile system.



Vehicle	Minotaur I	Pegasus XL	SSLV Taurus	Commercial Taurus (XL)	Minotaur IV	Minotaur V
Performance LEO Maximum	639 kg	443 kg	1320 kg	1590 kg	1735 kg	>2500 kg
GTO	—	—	400 kg	448 kg	GTO Capable with Star48V 4th stage	678 kg
Cost	$12.5 million	$12–15 million	$18–20 million	$18–20 million	$22–25 million	$25-28 million
First Flight	1999	1994	1994	1998	2008 (USAF SSBS Mission)	after 2009
Launch Site(s)	Vandenberg Others	Vandenberg Wallops Cape Canaveral Others	Vandenberg Others	Vandenberg Others	Vandenberg Wallops Cape Canaveral	Vandenberg Wallops Cape Canaveral

Figure 4. OSC Family of Launch Systems.

B. Secondary Launch Systems Trade Analysis

The preliminary trade analysis verified that generally accepted notion traditional high-energy launch systems were not economically feasible for the small satellite community. The acceptable "short list" included only four launch configurations*, 1) LMA Athena II, 2) OSC Taurus XL, 3) OSC Minotaur IV, and 4) OSC Minotaur V*. The comparative positives and negatives of these systems are listed in Table 2. Costs are similar with the Taurus($18-20 million) being least expensive and the Minotaur V ($25-28 million) being most expensive. One of the major additional considerations is that the Athena II can only be launched from CCAFB for east coast launches. Stage impact restrictions limit the maximum inclination to 50^o without a plane change during the stage 3 burn. The required SandiaSat mission inclination of 55^o mandates a 5-degree plane change during launch. This equivalent ΔV loss reduces the payload that can be delivered to the required MTO orbit. This loss must be considered in the "short list" trade analysis.

[‡] Verbal quote from OSC, $35 million first flight, $25-28 million recurrent costs thereafter.

Table 2. Minotaur IV, Minotaur V, Athena II, and Taurus XL Launch System Comparisons.

Minotaur IV

- Maximum LEO Payload: >1735 kg, *possible … GTO capable*
- Successful Launches: no manifests to date, Uses Legacy Peacekeeper stages, 51 launches)
- Launch Site: CCAFB, WFF (28.5° - 55° inclination), VAFB (55-120 deg.)
- Cost: $25 million
- ***Negative Factors: No operational record, limited lift capability***

Athena II

- Maximum LEO Payload: 820 – 2065 kg, *590 Kg to GTO*
- Successful Launches: 6 of 7 since 1995
- Launch Site: CCAFB (28.5° - 50° deg inclination) KLC (64 – 116 deg inclination)
- Cost: $22 –26 million
- ***Negative Factor: Inclination restrictions, no east coast launch to 55° inclination without plane change.***

Minotaur V

- Maximum LEO Payload: >2500 kg, *Hi-Energy version of Minotaur IV, 678 kg to GTO*
- Successful Launches: no manifests to date, Uses Legacy Peacekeeper stages, 51 launches)
- Launch Site: CCAFB, WFF (28.5° - 55° inclination), VAFB (55-120 deg.)
- Cost: $25-28 million
- ***Negative Factors: No operational record, initial development costs incurrent for first operational flight***

Taurus XL

-Maximum LEO Payload: 1590 kg

-Successful Launches: 7 of 8 Since 1995

-Launch Sites: CCAFB, WFF (28.5° - 55° inclination),) VAFB (55 – 120 deg inclination)

-Cost: $18 – 20 million

-Negative Factor: >Limited lift capability, Only 440 kg to GTO

1. *Preliminary Payload Mass Budget Analysis*

Preliminary SDL designs for the SandiaSat resulted in a spacecraft mass of approximately 250 kg. Allowing for a 35% mass contingency on the delivered payload to account for mounting structures and future mass growth, the launch system must be capable of delivering approximately 340 kg to the final MEO orbit. The 4 launch options listed in Table 2 must be capable of delivering to MTO this 340 kg payload plus the mass of the kick motor and propellant required to transfer from MTO to the final MEO orbit. Table 3 shows the required ΔV to transfer from a notional *1000 x 19000 km* altitude transfer orbit to the final *19,000 km* altitude orbit.

Table 3. Required ΔV for MTO-to-MEO Orbit Insertion.

Orbit	Perigee Altitude, *km*	Apogee Altitude, *km*	Velocity, *km/sec*	Inclination, *deg.*	Required ΔV, *km/sec*
MTO	1000	19,000	2.6597 (apogee)	55	--
MEO	19,000	19,000	3.9637	55	1.3041

Using the ΔV value calculated in Table 3 *(1.3041 km/sec),* assuming a kick motor specific impulse of *285 seconds*, and a residual kick-motor motor and inter-stage mass of *100 kg*, Table 4 shows the required mass to be delivered to the MTO orbit by the launch system; approximately 700 kg excluding the mass of the expended 4th stage. [§]

[§] While the optimized transfer orbit, kick stage I_{sp}, and residual mass may differ than these assumed values, these notional parameters were used to benchmark the relative lift capacity of the candidate launch systems.

Table 4. Preliminary Mass to Transfer Orbit.

Itemized Mass, *kg*	
Spacecraft Mass	*340*
Expended Kick Stage +Interstage Mass	*100*
Required Propellant for Apogee Kick	*261.6*
Required Mass to MTO	*701.6*

This mass calculation is performed using a simple rocket equation analysis[xv] assuming impulsive transfer from MTO to MEO

$$M_{prop} = M_{final} \cdot \left[e^{\frac{\Delta V}{g_0 I_{sp}}} - 1 \right]. \quad (1)$$

2. C3 Payload Analysis

The lift capability of the 4 candidate launch systems listed in Table 2 is performed using a *"C3"* launch energy analysis. In this analysis data for lift masses for LEO, GTO, escape velocity, and any other available orbits are curve-fit and interpolated the energy level of the MTO orbit specified in Table 3. [xii,xiii,xiv,xvi] For an *elliptical orbit*, *C3* is always *a negative value*. For an escape trajectory, *C3* is non-negative and equals "excess hyperbolic" velocity.[xvii] *C3* provides a convenient way to compare the required launch energies orbits with widely disparate parameters. The *C3* analysis calculates the total orbital energy based on the specified orbital parameters, where for an elliptical orbit

$$\begin{aligned} C3 = -2\cdot\left[\frac{\mu}{2a}\right] + 2\cdot\left[\frac{\Delta V^2_{maneuvering}}{2} - \frac{\Delta V^2_{track}}{2}\right] = \\ -2\cdot\left(\frac{\mu}{R_p + R_a}\right) + \left(\Delta V_{maneuvering} + \Delta V_{track}\right)\left(\Delta V_{maneuvering} - \Delta V_{track}\right) \approx \\ -2\cdot\left(\frac{\mu}{R_p + R_a}\right) + \left[\Delta V_{maneuvering} + \Omega_{\oplus} R_{\oplus}\cos(i)\right]\cdot\left[\Delta V_{maneuvering} - \Omega_{\oplus} R_{\oplus}\cos(i)\right] \end{aligned} \quad (2)$$

In Eq. 2 μ is the planetary gravitational constant for the earth, a is the orbit semi major axis, R_p is the perigee radius, R_a is the apogee radius, $\Omega_{\oplus}$ is the earth's angular velocity, $R_{\oplus}$ is the local earth radius, and *C3* is the specific launch energy. The term $\Delta V_{maneuvering}$ refers to any additional energy expenditures required to reach the desired orbit. An orbital plane change during launch is a good example of *maneuvering* ΔV losses. The term ΔV_{track} refers to the velocity boost along the track of the orbit provided by the earth's rotation. The final term due to earth rotation is an approximation, but is reasonably accurate as long as the orbit inclination is limited to values greater than or equal to the launch latitude.

The curve-fitted *C3* data is presented in Figure 5 for the Athena II, Taurus XL, Minotaur IV, and Minotaur V Launch Systems. In these figures the payload mass delivered to a given orbit is plotted against the launch energy. The solid lines represent the curve-fit data, and the single data point on each curve represents the payload mass that can be delivered to the MTO orbit at 55^o inclination with *1000 x 19,000 km* perigee and apogee altitudes. The plotted mass does not include the mass of the expended 4th stage. As per the earlier discussion, the plotted Athena II data point includes the effect of having to change orbital planes by *5^o* during the launch.

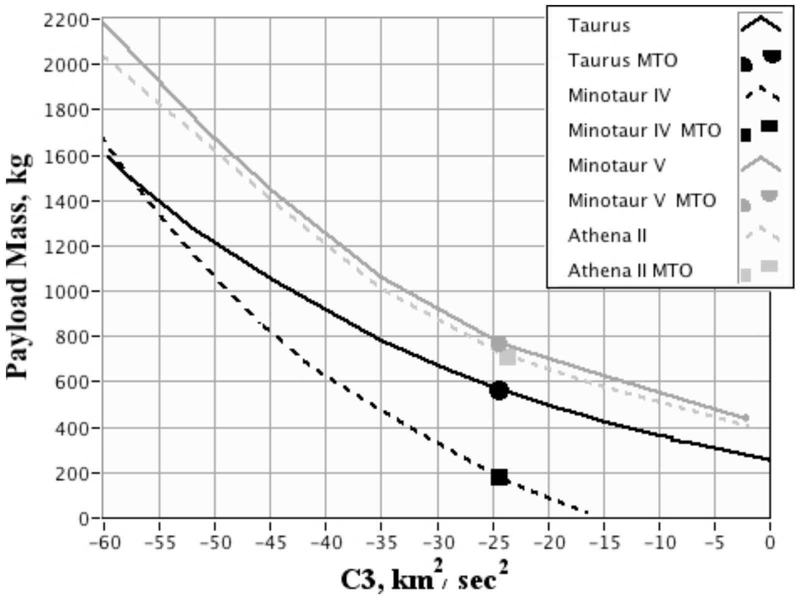


Figure 5. Launch Energy Comparisons for the 4 Candidate Launch Systems.

Table 5 summarizes the total mass that can be delivered to the MTO orbit by each of the launch systems. Only the Athena II and Minotaur V are capable of delivering the required 700^+ kg to the MTO. The other two systems are immediately excluded from further consideration in this analysis. The Minotaur V provides approximately 8% more lift capacity as well as having the option to launch from Wallops providing for greater launch flexibility. As mentioned previously, the availability of small payload launch windows from WFF is significantly higher than from CCAFB, and the comparative cost of launch operations is significantly lower. These factors were considered sufficient to weight in favor of the Minotaur V, and that launch system was selected as the system of choice for the SandiaSat mission.

Table 5. Payload Mass Delivered to SandiaSat MTO.

Launch System	*Mass to 1000 x 1900 km altitude, 55^o inclination, MTO*
Athena II	*715.4 kg*
Taurus XL	*570.1 kg*
Minotaur IV	*183.5 kg*
Minotaur V	*775.4 kg*

III. Detailed Launch Mission Analysis

This section presents a detailed launch mission analysis based on specifications and properties of the Minotaur V launch vehicle selected in the previous section. This subsection details *i)* the baseline Minotaur V configuration, *ii)* recommendations for modifications to the baseline, *iii)* trajectory modeling and optimization, *iv)* mission concept of operations, *v)* final mass budget analysis, *vi)* payload separation and re-contact analysis, *vii)* Monte-Carlo orbital insertion accuracy analysis, *viii)* summary of the design options considered, and *ix)* a rough order of magnitude *(ROM)* cost estimate.

C. Baseline Minotaur V Launch System

The Minotaur family[xviii] includes the Minotaur I, IV, V space launch vehicles (SLV) and the Minotaur II and III suborbital target launch vehicles (TLV). Minotaur vehicles are available from OSC under a contract with the USAF Space and Missile Systems Center (SMC). The Minotaur IV and V SLV's are constructed using decommissioned government-furnished equipment (GFE) Peacekeeper missile stages[xix]. These GFE stages include; stage I (TU-903), stage II (SR119), and stage III (SR120). The Minotaur IV SLV adds a 4th stage based-on the ATK launch Systems (formerly Hercules) Orion 38 motor. The Minotaur V is a 5-stage evolutionary version of the

Minotaur IV, adding propulsive energy need to support high-energy missions such as GTO or escape velocity. In the Minotaur V configuration the larger ATK launch Systems Star 48BV replaces the Orion 38 as the 4th stage motor. The Minotaur V also features an extended payload shroud and support structure for a fifth stage based on the ATK Star 37FM (spin-stabilized) or Star 37FMV (3-axis stabilized) kick-motor. Figure 6 compares the baseline-configurations for the Minotaur IV and Minotaur V launch vehicles.[xx]

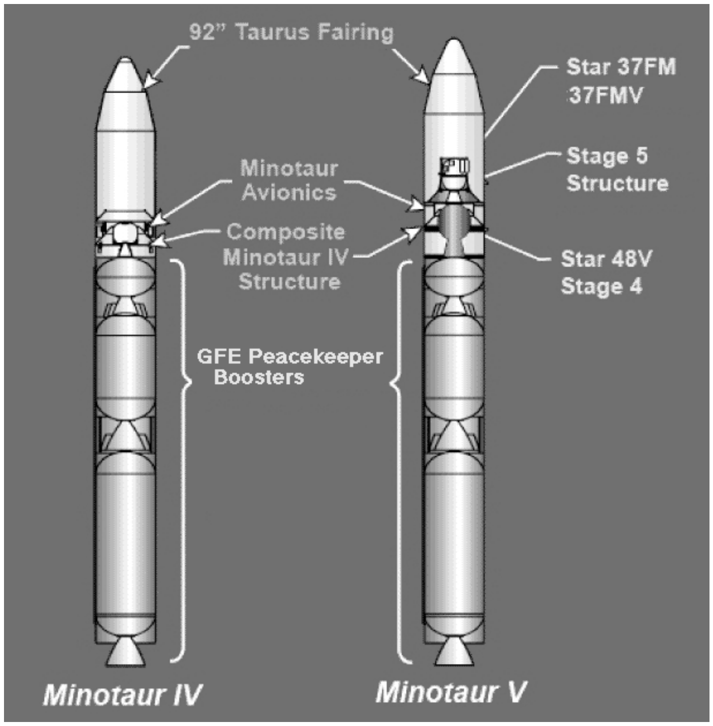


Figure 6. Comparison of the Baseline Minotaur IV and V Launch Vehicles.

There have been thirteen successful launches from the Minotaur family – seven Minotaur 1 SLVs and six Minotaur TLV's. To date neither the Minotaur IV or V systems have been launched. The first scheduled launch of a Minotaur IV in late 2009 will be a USAF payload -- the Space Based Space Surveillance (SSBS) mission. Even though the Minotaur IV and Minotaur V launch systems have no operational history; collectively, the Peacekeeper stages I-III have successfully launched 51 times. So there is a proven flight heritage for the majority of the Minotaur IV and Minotaur V sub-systems

D. Recommended Modifications to the Baseline Minotaur V System

The baseline design for the Minotaur V is intended for high-energy missions with low negative or positive *C3* values. Typically for these high energy missions, stages 1-5 will be used to insert the payload into the required transfer orbit (e.g. trans-lunar injection), and a 6th stage will be integrated with the payload for final orbit trim (e.g. lunar orbit insertion). For the lower energy MEO orbit required for the SandiaSat mission, a 6th stage is unnecessary. A preliminary analysis based on the impulsive burn assumptions showed that the Minotaur V is capable of delivering the required satellite payload to the final MEO orbit in just 5 stages by replacing the large ATK Star 37 motor with the significantly smaller ATK Star 27 motor. This conclusion is supported by data presented in Figure 7 where the payload mass delivered to an elliptical transfer orbit by the Minotaur V stages 1-4 is plotted against apogee altitude. Here the assumed perigee altitude for the transfer orbit is 1000 km, and the orbit inclination is 55°. The plotted mass *includes* the final SandiaSat payload, fully loaded apogee kick motor and any 5th stage inter-stage/separation system mass. The plotted MTO payload mass *does not include* the assumed mass of the 4th stage avionics module/interstage (100 kg) or the mass of the expended Star-48 motor. The mass delivered to an elliptical orbit with an apogee altitude of 19,000 is approximately *688 kg*. This mass is consistent with the required payload mass *(701 kg)* calculated and presented in Table 4. This conclusion is very significant because of the potential for a less complex (and potentially less expensive) launch configuration. Direct launch simulations to be presented later in this paper will verify these preliminary impulsive rocket-equation calculations.

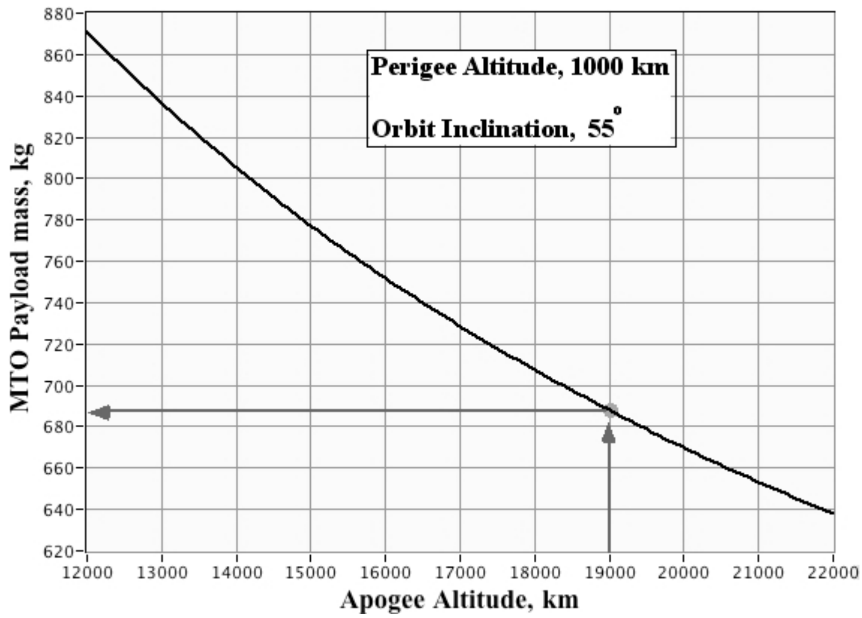


Figure 7. Payload Delivered to MTO by Minotaur V Stages 1-4.

The key to achieving the required MEO orbit in 5-stages is the significantly smaller mass of the Star 27 *(360 kg)* [xxi] motor compared to the Star 37 *(1148 kg).* ,[xxii] Using the smaller Star 27 5th stage motor also provides significantly more working volume within the Minotaur payload fairing. The STAR 27 rocket motor has a proven flight history and was developed and qualified in 1975 for use as the apogee kick motor (AKM) for the Canadian Communications Research Center Communications Technology Satellite. The Star 27 motor has served as the apogee kick motor for various applications including the NAVSTAR, GOES, and GMS series satellites. Although considerably smaller than the baseline 5th stage motor, the Star 27 still provides sufficient ΔV for final MEO orbit insertion. Table 6 compares the Star 37 and Star 27 motors. Recall from Table 3 that *1.3041 km/sec* ΔV is required for final orbit insertion. If one assumes a delivered MEO payload mass of 340 kg (Table 3), then the available ΔV for the Star 27 motor is

$$\Delta V_{available} = g_0 I_{sp} \ln\left[1 + \frac{M_{prop}}{M_{payload} + M_{inert}}\right] = \frac{9.8067}{1000}{}_{\frac{km}{\sec^2}} \cdot 287.9_{\sec} \ln\left[1 + \frac{333.8_{kg}}{340_{kg} + \left(361.3_{kg} - 333.8_{kg}\right)}\right] = 1.824_{km/\sec} \quad (3)$$

In fact the total impulse available from the Star 27 motor is actually excessive by approximately 40% for a 340 kg payload mass, so a propellant off-load may be necessary. Fortunately, the Star 27 motor is designed to accommodate various propellant loadings (9% offload flown) and can be off-loaded up to 20% without recertification. Propellant off-loading has the advantage of allowing and option for more payload mass delivery to the required MEO orbit.

Table 6. Comparison of the ATK Star 37FM and Star 27 Motors.

Motor	Star 37 FM	Star 27
Total Weight	1148.3 kg	361.3 kg
Length	1.689 meters	1.237 meters
manufacturer	0.935 meters	0.693 meters
ATK Thiokol	ATK Thiokol	ATK Thiokol
Case	Titanium	Titanium
Propellant	1066.3 kg TP-H-3340 Solid	333.8 kg TP-H-3135 Solid
Nozzle	48.2: 1 expansion ratio 	48.8: 1 expansion ratio 
Pitch, Roll, Yaw Control	no gimbal	no gimbal
Burn Time	62.7 seconds	34.4 seconds
Thrust	47.276 kNt thrust	25.451 kNt thrust
ISP	289.8-sec	287.9-sec

E. Launch Simulation and Trajectory Optimization

Because the baseline Minotaur V configuration was modified for the SandiaSat mission, it was believed that direct simulation of the system components and the trajectory from launch to MEO insertion were necessary to verify the "back of the envelope" calculations presented in the previous section. The direct simulation also offers the opportunity to optimize the mission-specific endo-atmospheric portion of the launch trajectory. The optimization process was facilitated using an interactive simulation developed at Utah State University. This *three degree-of -freedom* simulation features a graphical user interface (GUI) that provides for operator interaction and direct in-the-loop control. At each data frame the vehicle pitch angle can be prescribed by direct joy-stick input, a pre-defined set of way points, or by a feedback-control loop. The interactive simulation allows for a rapid evaluation of a wide variety of candidate maneuvers and trajectories. Real-time displays allow the user to develop extensive intuition with regard to mission-specific parameters. These "piloted" simulation techniques were pioneered at NASA in early 1970's during the lifting body flight test programs and were paramount to the facilitation of this analysis.[xxiii] This interactive simulation approach was used as a time-saving measure in lieu of more traditional trajectory optimization tools like the Program to Optimize Simulated Trajectories (POST).[xxiv] One of the major drawbacks of POST is the difficulty of setting up the program, and sensitivity of the final solution to the initial trajectory guess. The interactive simulation also allowed perturbed conditions about the optimal trajectory for Monte-Carlo analysis[xxv] of expected orbit insertion accuracies.

For the 1st, 2nd, and 3rd stages, engine mass flow, nozzle exit velocity, and nozzle exit pressure were modeled to enable a thrust calculation as a function of altitude. The 4th and 5th stages were modeled using vacuum-thrust only. These data were collected from a variety of pubic domain sources.[xxvi,xxvii,xxviii] The equilibrium gas-chemistry code *Chemical Equilibrium with Applications* (CEA)[xxix] was used to model the combustion products based on mean properties for the specified propellants. The combustion data were used to develop the engine models for the first three stages. The aerodynamic characteristics of the first 4 vehicles stages were estimated using a panel code for subsonic flight conditions,[xxx] and a USU-developed incidence angle code for the supersonic flight conditions.[xxxi,xxxii] The simulation assumes that while the rocket motor for a particular stage is burning, the base pressure drag is negligible. During stage separation and ballistic coast phases of the launch, plume-off base drag characteristics are estimated using empirical correlations derived from Hoerner.[xxxiii] The launch simulation burned each of the 4 Minotaur stages to exhaustion, assuming a constant engine mass-flow, and depleting mass as a function of time. In this launch profile optimization an "apogee targeting" strategy was used where the 5th stage and payload is inserted directly MTO orbit with a variable perigee altitude and a 19,000 km apogee altitude. After a Keplerian coast period, the 5th state motor was fired just before the transfer orbit apogee to insert the payload into the final orbit. Non-impulsive, continuous-thrust calculations were used throughout the simulation. The launch trajectory optimization considered such factors as pitch profile, MTO perigee altitude, MTO orbit insertion point, and propellant offload required to match the required ΔV for final MEO orbit insertion. The performance metric for optimization was the final mass payload delivered to the MEO orbit. Table 7 presents the engine and stage properties as modeled in the simulation.

Table 7. Minotaur V Stage Propulsion Properties.

Simulation Element	Stage 1	Stage 2	Stage 3	Stage 4	Stage 5
Motor	TU-903	SR119	SR120	Star 48V	Star 27
Designer	Thiokol	Areojet General	Hercules	Thiokol	Thiokol
Structural Mass -Dry Mass *(kg)*	4300	3175.147	635.029	154.584	27.488
Wet Mass *(kg)*	48960	27669.135	7711.07	2164.543	361.15
Propellant	HTPB	HTPB	NEPE	TPH-3340	TPH-3135
Thrust (vac), *kN*	2204.5	1223.261	289.134	68.636	25.444
Thrust (sea level), *kN*	1954.3	1048.34	235.02	24.15	6.7
Exit Area *(m^2)*	2.469	1.7263	0.5341	0.439	0.185
Expansion Ratio	24.911	41.89	22.5	54.8	48.8
Astar *(m^2)*	0.0991	0.0412	0.0237	0.008	0.004
Combustor Pressure *(Mpa)*	12	15.61	6.966	3.985	3.882
Combustor Temperature *(K)*	3059.67	3489.74	2647.32	1894.5	3357.37
Isp (vac), *sec*	282	309	300	292.1	287.9
Isp (sea level), *sec*	250				
Burn Time *(sec)*	56	60.7	72	85	34.4
Ratio of specific heats *(CEA)*	1.878	1.878	1.31	1.142	1.142
Molecular weight, *(CEA)*	27.36	27.36	17.371	22.34	22.34
Mass Flow *(kg/sec)*	797.15	403.68	98.28	23.96	9.01
Exit Mach	3.8544	4.2015	4.3676	3.5667	5.871
Exit Pressure,*(kPa)*	47.88	32.26	20.83	10.99	2.79
Exit Velocity *(m/sec)*	2617.15	2892.33	2828.89	2663.36	2765.85
Exit Temperature *(K)*	1277.52	1313.13	669.05	1510.25	426.25

Figure 8 shows typical simulated launch (a) altitude, (b) downrange, (c) velocity, and (d) acceleration time histories. From the acceleration plot, notice the extended coast time between the 3rd stage motor burnout and the 4th stage motor firing. This coast period was determined as a part of the trajectory optimization process.

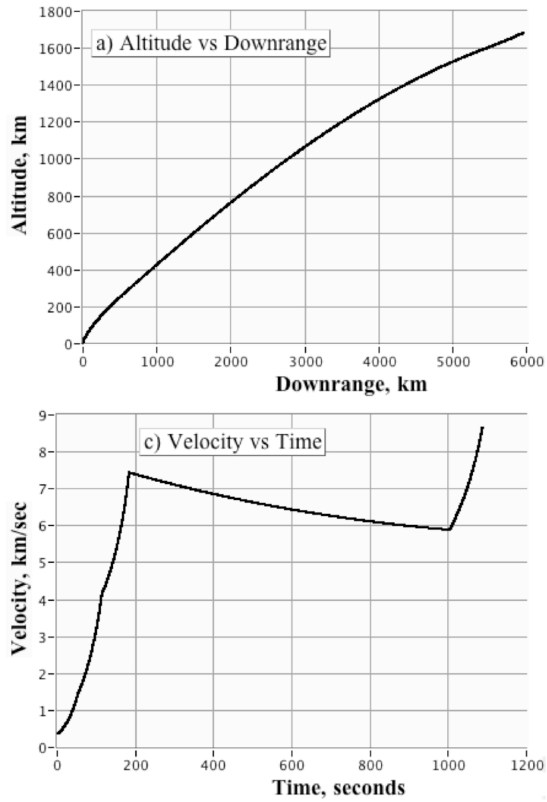


Figure 8. Typical Minotaur V MTO Insertion Launch Trajectory.

As mentioned earlier, the pitch profile and MTO insertion point were optimized to allow the maximum payload delivery. Figure 9 shows the optimized pitch profile compared against a similar ballistic trajectory for the endo-atmospheric portion of the flight; notice that the optimized trajectory initiates the "gravity turn" far sooner than on the ballistic profile and levels out once the endo-atmospheric portion of the flight is completed. This gravity turn was assisted by flying at slightly negative angles of attack to provide a downward lift on the launch stack. Figure 10 shows the lift and drag coefficient and angles-of-attack profile for the optimized launch trajectory. Lateral loads induced on the launch stack by aerodynamic forces were not considered in this optimization, but the larger negative angles-of-attack occur at higher altitudes where dynamic pressures are low and side-loads should not present a problem for the launch stack.

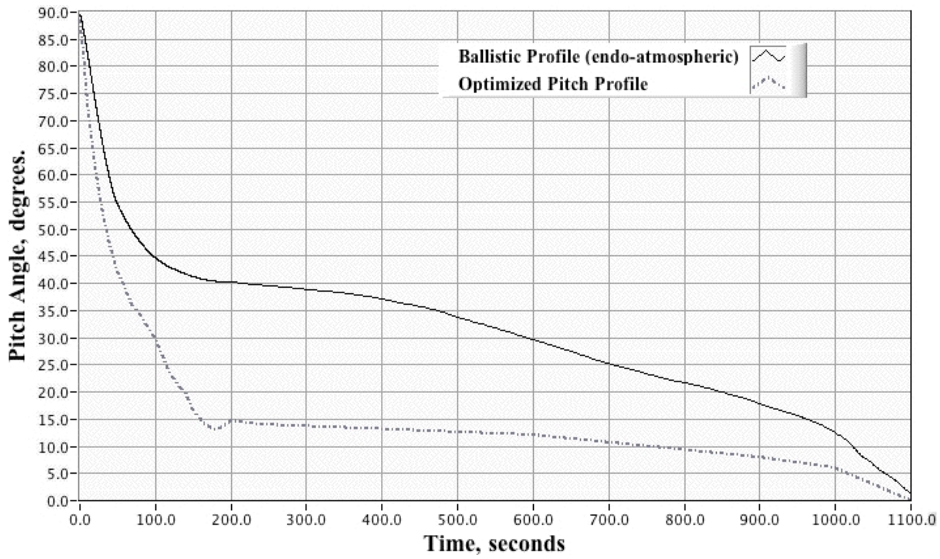
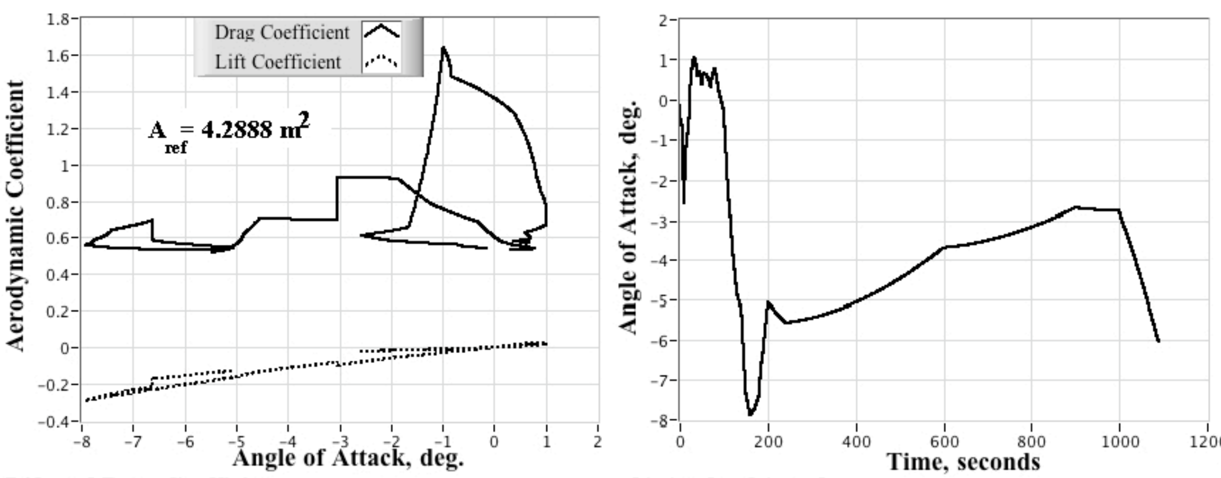


Figure 9. Optimized Launch Pitch Profile.

Figure 11 shows the optimized MTO trajectory plotted in the orbital plane (55^o inclination) and the corresponding ground track showing the launch along the descending node of the orbit. The resulting optimized transfer orbit has an apogee altitude of approximately 19,000 km and a perigee altitude of approximately 1300 km, slightly higher than the perigee altitude assumed during the preliminary trade analysis. Also the actual MTO insertion altitude is approximately 1680 km. This insertion point into the MTO trajectory is noted on Figure 11.



a) Lift and Drag Coefficients **b) Angle of Attack**

Figure 10. Aerodynamic Coefficients and Angle of Attack Along Optimized Launch Trajectory.

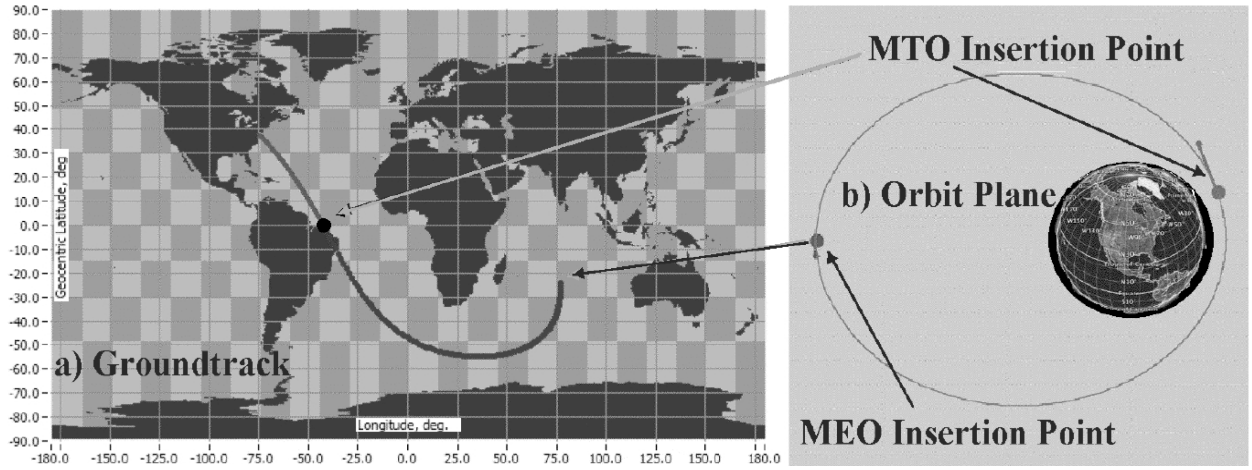


Figure 11. Optimized MTO Trajectory, Ground Track and Orbital Plane.

F. Mission Concept of Operations

Figure 12 shows the end-to-end launch and deployment concept of operations (CONOPS). In this CONOPS the 4th stage (Star 48) inserts the payload and 5th stage (Star 27) into the MTO trajectory. The Minotaur V 4th stage avionics module positions the payload at the required attitude and spins up the system at a rotational rate of approximately 10 revolutions/minute. The coast from MTO insertion to the orbit apogee requires approximately 2.8 hours. Once MTO apogee is reached, the Star 27 fires and inserts the payload into the circular 19,000 km orbit. Once inserted into the final orbit, a cold-gas jet system attached to the payload adapter cone spins down the payload. Following spin down, the expended 5th stage is separated from the SandiaSat. Total mission elapsed time from launch to solar panel deployment is approximately 3.16 hours.

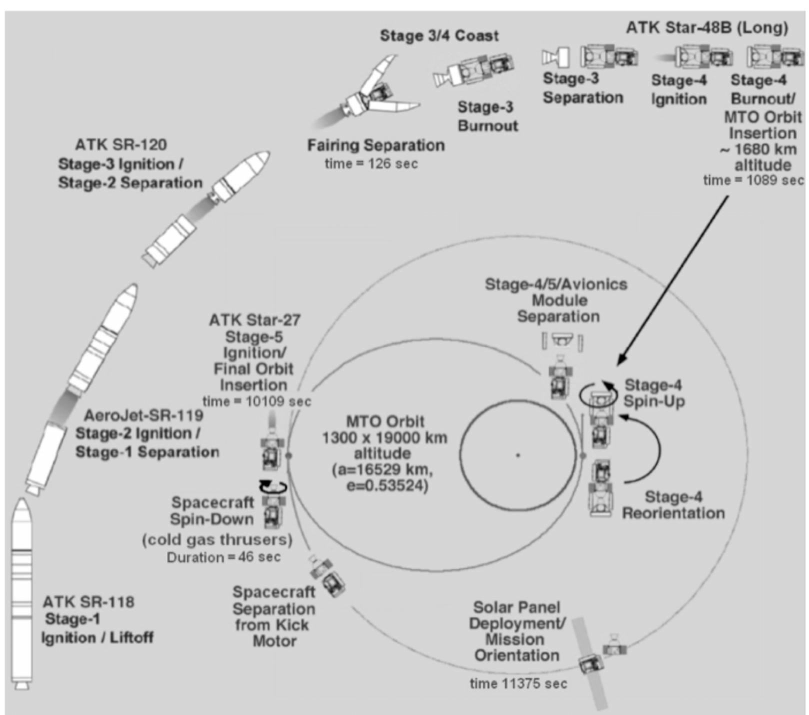


Figure 12. Launch and Deployment Concept of Operations.

G. Mass Budget Analysis

As mentioned in the previous section, the Star 27 kick motor provides excess impulse for the required MTO to MEO orbit transfer. Consequently, off loading propellant allows more payload to be delivered to the final orbit and trajectory optimization balance propellant off load versus the delivered payload mass. The required apogee kick ΔV (and hence the propellant off-load for the Star 27 5th stage) depended on the precise MTO trajectory energy level reached during the launch and MTO insertion phase of the mission. Allowing for the mass of the 4th stage avionics module and the inert 4th stage, the "live payload" delivered to MTO is approximately 682 kg. Table 8 shows the Stage IV mass budget resulting from the optimized MTO launch analysis. The delivered mass of approximately 682 kg into the 55^o inclination MTO trajectory from WFF is comparable to the mass value calculated earlier using the impulsive analysis and presented in Table 5. Notice that a 26.4% propellant offload is required to "tune" the proper MEO orbit insertion ΔV.

Table 8. Mass Budget for Optimized Launch and MTO Insertion.

• ***Stage IV***

-- Star 48 Motor
Inert Motor Weight, *154.85 kg*
Nominal Full Propellant Load, *2008 kg*
Total Wet Mass, 2162.85 kg
-- Stage IV Interstage, Separation Clamp *150 kg*
-- Avionics Module and Spin-up Propellant, *171.14 kg*

• Total Spacecraft delivered to *MTO Perigee*, *1002.4 kg*
• Spacecraft Mass *after Avionics Module Jettison*, *681.3 kg*

Table 9 shows the mass budget for the MEO insertion. The Total Spacecraft mass allocation is approximately 360 kg and is considered to be the maximum that the modified Minotaur V SLV can deliver to the required 19,000 km orbit, 55^o inclination MEO orbit without a 6th stage kick motor. The 360 kg maximum payload offers a 40% contingency for mass growth.

Table 9. Mass Budget for Optimized MEO Insertion.

• ***Stage V***

• Total Spacecraft delivered to *MTO Apogee*, *681.3 kg*
  - -- Lightband Separation System
    - Upper ring, *2.06kg*
    - Lower Ring, *4.11kg*
    - # of Separation Springs, *26*
    - # of Separation Connectors and Switches, *12*
  - -- Star 27 Kick Motor
    - Inert Motor Weight, *27.5 kg*
    - Nominal Full Propellant Load, *333.76 kg*
    - Propellant Offload, *26.4%*
  - -- Kick Motor Adaptor Cone, *16 kg*
  - -- Cold Gas Spin--Down System, *25 kg*

• Total Spacecraft Mass Delivered to MEO, *436.5 kg*
• Spacecraft Mass after Kick Motor Separation, 362.4*kg*
• Delivered MEO Spacecraft Mass Allocation, 360.4*kg*

H. Payload Separation and Re-contact Analysis

The separation system chosen for the SandiaSat was the Planetary Systems Corporation[xxxiv] 38" Motorized Lightband®.[xxxv] The Lightband is stowed with links locking a retaining ring that holds the separation system together. Motors drive a mechanism that allows the retaining ring to contract. The contracted ring allows spring plungers to disengage the payload side of the ring, and separation springs push the two rings apart. The separation springs impart ΔV to the payload, separating it from the expended 5th stage motor. The resulting separation ΔV is determined by the number of separation springs installed in the system. The resulting separation ΔV is described by

$$\Delta V = \sqrt{2\cdot\eta\cdot E\cdot N_{spring}\cdot\left(\frac{M_{payload}+M_{stage}}{M_{payload}\cdot M_{stage}}\right)} = \sqrt{2\cdot\eta\cdot\frac{k\cdot X_{\max}^2}{2}\cdot N_{spring}\cdot\left(\frac{M_{payload}+M_{stage}}{M_{payload}\cdot M_{stage}}\right)} = \sqrt{k\cdot\eta\cdot X_{\max}^2\cdot N_{spring}\cdot\left(\frac{M_{payload}+M_{stage}}{M_{payload}\cdot M_{stage}}\right)} \quad . \tag{4}$$

In Eq. 4 ΔV is the relative velocity between the payload mass *($M_{payload}$)* and the expended stage mass *(M_{stage})*. The parameter E is the stored potential energy of a separation spring, η is the spring potential energy storage efficiency, k is the spring constant, and X_{max} is the full stroke of the separation spring. Table 10 shows the separation spring data published by Planetary Systems.[xxxvi]

Table 10. 38" Lightband Separation Spring Parameters .

Spring Efficiency, η	Spring Constant, k	Full Stroke, X_{max}	Spring Force, F_{spring} Compressed	Spring Energy, E Compressed
~0.9	4.08 Nt/mm	21.06 mm	85.93 Nt	0.9048 J

Because springs store energy inefficiently when compared to pyrotechnic-based separation system, they produce only small separation velocities. The cost of spring mass must be traded against the required separation velocity and recontact after separation is a potential issue. The 38" Motorized Light Band allows a maximum of 94 separation springs. Assuming a spring efficiency (η) of 0.9, the maximum 94 separation springs provides a separation ΔV of *1.58 m/sec*. This system with 94 springs is very stiff and a separate mechanism would be needed to compress the springs during the payload installation. The additional number of springs also takes away from available satellite mass growth. A compromise value of $N_{springs} = 26$ was selected. This number of spring results in a total Lightband resistance force of 2234 *Nt* (227.8 *kgf*). This resistance force can be easily compressed by the weight of the satellite and makes for easier processing during payload installation. The 26 springs provide a separation ΔV of *0.83 m/sec*.

If separation is delayed for 120 seconds after burnout of the Star 27 motor to insure that residual propellant burning is completed, simulations show that the 26-spring ΔV is sufficient to prevent recontact between the payload and exhausted Star 27 upper stage. Figure 13 shows the separation distance between the SandiaSat payload and the expended 5th stage following jettison. This analysis assumes that the Lightband separation ΔV is directed with a 5° degree pitch angle and a 5° out-of -plane (yaw) angle during separation. The absolute separation distance in *km* is plotted with a logarithmic scale on the ordinate and the elapsed time from the MEO insertion burn in *years* is plotted on the abscissa. Notice that the Lightband system provides immediate and positive separation and a potential recontact event occurs approximately every 2 years. During the maximum lifetime of the mission (3 years) the closest approach between the two objects is 20 km. It is possible that somewhere beyond the lifetime of the SandiaSat satellite the two objects could recontact creating an orbital-debris scenario. But since the SandiaSat MEO orbit is considered a "junk orbit" this eventuality is not a great concern.

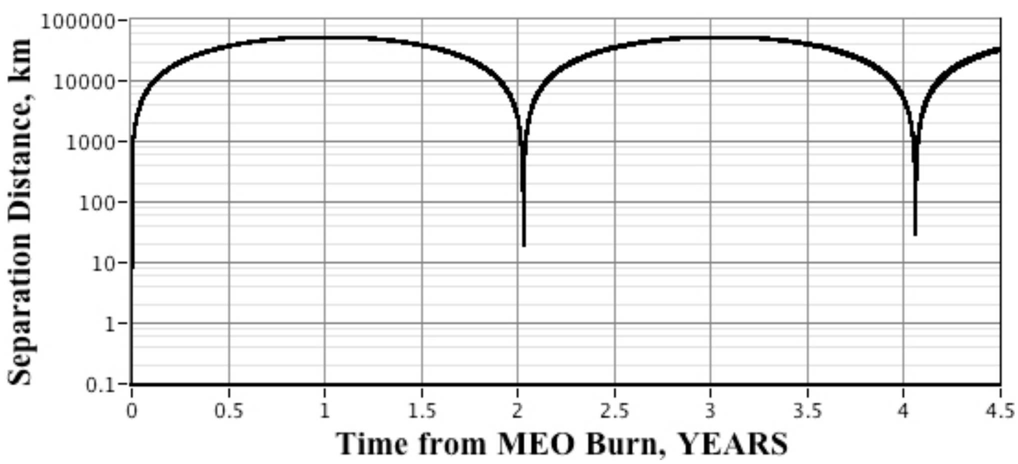


Figure 13. Time History of Separation Distance Between Payload and Expended 5th Stage.

I. Monte Carlo Analysis of Expected Orbit Insertion Accuracy

The end-to-end final orbit insertion accuracy was estimated using a Monte-Carlo simulation analysis. For this analysis, the interactive simulation kernel was run in a batch-mode with rocket and orbit parameters perturbed using Gaussian white noise models. Figure 14 shows the resulting apogee and perigee altitudes for each of the individual Monte Carlo runs. Table 11 shows the 1-σ noise inputs to the Monte-Carlo model. A total of 997 data runs were performed to establish statistical validity (Ref. xxv), and statistics of the final orbital parameters were calculated. Table 12 shows the end-to-end uncertainty estimates in the final MEO parameters. These uncertainties also include the effects of the Lightband separation ΔV. The mean values for the orbit perigee and apogee altitudes derived from the Monte-Carlo simulation are compliant with the values prescribed in Table 1.

Table 11. 1-σ Uncertainty Models in End-to-End Monte Carlo Simulation

- Characteristics Perturbed for Launch and Final Orbit Insertion
- Launch Error Sources, *1-σ*
  - -- Pitch Profile, ±0.05% total error *(deviation from optimal)*
    ... assumes pitch guidance feedback
  - -- Uncertainty in Drag Coefficient, ± *5%*
  - -- Uncertainty in Lift Coefficient, ± *5%*
  - -- Thrust Uncertainty, ± *0.1%*
  - -- Burn Time Uncertainty, ± *0.1%*
  - -- Total Impulse Uncertainty, ± *0.1414%*
- Kick Motor Insertion I (Star 27) Uncertainties (from ATK)
  - -- Thrust Error, *0.16667%*
  - -- Burn Time Uncertainty, *0.084%*
  - -- Total Impulse Uncertainty, *0.1867%*
  - -- Burn Pitch Misalignment Uncertainty, ±*1°*
  - -- Burn Out-of-Plane Misalignment Uncertainty, ±*1°*
- Lightband Separation Uncertainties
  - -- Separation *ΔV* Uncertainty, ± 0.*1 m/sec*
  - -- Separation Pitch Misalignment Uncertainty, ±*1°*
  - -- Separation Out-of-Plane Misalignment Uncertainty, ±*1°*

Table 12. End-to-End Uncertainty Estimates in Final MEO Orbit Parameters

Parameter	*Mean Value*	*1-σ Stnd. Dev.*
a, km	*25370.2*	*±80.8*
e	*0.00124*	*±0.0004*
i, deg	*54.991*	*±0.115*
Ω, deg	*197.71*	*±0.080*
ω, deg	*151.25*	*±0.119*
Perigee Altitude, km	*18975.8*	*±83.9*
Apogee Altitude, km	*19022.9*	*±86.2*

J. Design Change Summary

Table 13 shows the trade options that were considered and the impacts of these design changes on the program for the MEO launch analysis. The major design consideration was the replacement of the Star 38/FMV 5th stage on the Minotaur V with the smaller Star 27 motor. This design change enables the Minotaur V launch system to deliver the SandiaSat payload to the required MEO orbit without using a 6th stage kick motor.

Table 13. Design Changes and Options Considered.

Design Baseline Change Option	Technical Performance Variation	Programmatic Cost Variation	Programmatic Schedule Variation
Minotaur V with Star 37 upper stage	Star 37 too massive for MEO insertion, No residual payload		Minimal
Minotaur V with Star 27 upper stage	With 25-30% offload, 380+ kg deliverable to MEO	$250K increase	Minimal
26 spring Lightband replaces original 92 spring option	None	minimal	Minimal

K. Rough Order of Magnitude Launch Cost Estimate

Since both the Minotaur IV and Minotaur V SLV's do not have an operational flight history, any analysis of the launch costs must be regarded as only a rough order of magnitude (ROM) estimate, And even then this number must be regarded with some skepticism. For this analysis the upper limit of the estimated Taurus Launch Costs, approximately $20 million (Figure 4) was used as the baseline. The non-recurrent engineering (NRE) costs to develop the Minotaur IV SLV from the Peacekeeper stages is estimated at $50 million. Because the Minotaur V avionics, structures, and payload fairing are shared with the Minotaur IV, there is significantly less new development required and the additional NRE is estimated at $10 million. Amortizing these costs across the first 10 Minotaur IV/V flights, the costs of the baseline Minotaur V configuration is estimated as $20M + $60M/10 = $26M. The NRE modifications required for replacing the Star 38/FMV with the smaller Star 27 upper stage motor is estimated at $2.5M. Thus the total launch costs are estimated at $28.5. *OSC estimates the recurrent operational launch costs at $25-28 million (Ref. xviii,) so this estimate is consistent with the vendor's claims.* Table 14 summarizes this ROM calculation.

Table 14. Minotaur V Launch ROM Cost Estimate.

• Taurus Launch (International Launch Guide) $20 M
• Minotaur V Estimate …Taurus + additional NRE $20 M + ($50M+$10M)/10 = **$26M**
• Engineering Star 27 Interface with 5th stage payload adapter … $2.5M
• Total Launch Costs: **$28.5M**

IV. Summary and Concluding Remarks

A trade study investigating the economics, mass budgets, and concept of operation for delivery of a small, technology demonstration satellite to a medium-altitude earth orbit is presented. Sandia National Laboratory has proposed this prototype satellite to space-test and mature emerging technologies required for the next generation of global positioning satellites. A primary objective is the maturation low readiness technologies required for nuclear explosion monitoring. Mission requirements specify the payload to be delivered to a circular orbit at 19,000 km altitude and an inclination of 55^o. The payload includes classified technologies and a USA licensed launch system is mandated. A preliminary trade analysis is performed where all available US launch systems are considered. The initial trade study identifies the Minotaur V launch system is the best launch option. The Minotaur V is a 5-stage evolutionary version of the Minotaur IV constructed using decommissioned government-furnished (GFE) Peacekeeper missile stages for the first three stages. End-to-end mass budgets are calculated, and a concept of operations is presented. Monte-Carlo simulations are used to characterize the expected accuracy of the final orbit. An optimal launch trajectory and an order-of-magnitude cost analysis are presented.

A primary conclusion of this study is that replacing the baseline 5th stage ATK-37FM motor by the significantly smaller ATK Star 27 allows the final orbit to be reached without a 6th stage. This result is very significant in that it offers a less complex and potentially less expensive launch configuration, and provides significantly more working volume within the payload fairing. Other significant conclusions are:

i) Optimized trajectory delivers a total of 681 kg payload to MTO (minus inert 4th stage),
ii) Star 27 kick motor requires ~ 25-30% offload for proper payload insertion ΔV,
iii) Total spacecraft mass after kick motor separation is approximately 360 kg, a 40% mass margin,
iv) Lightband with 26 separation springs provided sufficient ΔV to avoid recontact for mission lifetime,
v) Monte-Carlo analysis shows 1-σ apogee/perigee accuracy of approximately 19000 ± 85 km,
vi) Monte-Carlo analysis shows final orbit inclination of approximately 55° ± 0.1°,
viii) Total mission time line from launch to final orbit is approximately 3.16 hours,
ix) Launch costs including amortization of non-recurrent engineering estimated at $28.5 million.

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