

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

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A trade study investigation the possibilities of delivering a small technology-demonstration satellite to a medium earth orbit are presented. The satellite is to be deployed in a 19,000 km orbit with an inclination of 55°. This payload is a technology demonstrator and thus launch and deployment costs are a paramount consideration. Also the payload includes classified technology, thus a USA licensed launch system is mandated. All FAA-licensed US launch systems are considered during a preliminary trade analysis. This preliminary trade analysis selects Orbital Sciences Minotaur V launch vehicle. To meet mission objective the Minotaur V 5th stage ATK-Star 37FM motor is replaced with the smaller ATK- Star 27. This new configuration allows for payload delivery without adding an additional 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	= separation spring stored potential energy, $joules$
e	= orbit eccentricity
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
GTO	= geostationary transfer earth orbit
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
k	= spring constant, Nt/mm
KLC	= Kodiak Island Launch Complex
MTO	= medium transfer orbit transfer orbit
LEO	= low earth orbit
LMA	= Lockheed Martin Aerospace
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, kg
N_{spring}	= number of springs in Lightband® separation system
MTO	= MEO transfer orbit
OSC	= Orbital Sciences Corporation
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, $3.986004418 \times 10^5 km^3/sec^2$
w	= argument of perigee, $deg.$
Ω	= right ascension of ascending node, $deg.$
Ω_{\oplus}	= angular velocity of earth, $7.292155 \times 10^5 rad/sec$

I. Introduction

Sandia National Laboratory is investigating advanced technologies required for nuclear explosion monitoring sensors to be deployed with the next-

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generation of Global Positioning System (GPS III) satellites. GPS III is expected to use new sensor and signal processing technologies. In order to verify that these technologies are fully developed and space-qualified, Sandia is developing a small satellite with prototype bus architecture. The satellite is to be deployed in a medium earth orbit (MEO) and have a mission life between one and three years. This satellite has been named SandiaSat. The Utah State University (USU) Department of Mechanical and Aerospace Engineering along with The Utah State University Space Dynamics Laboratory (SDL) are developing the notional mission plan, concept of operations (CONOPS), preliminary satellite and ground station designs, launch options, and associated cost estimates. This paper addresses the launch and deployment options for this mission. Fundamental mission objectives require the payload to be delivered to a circular orbit at 19,000 km altitude at an inclination of 55°. 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. It is anticipated that the completed SandiaSat system will be available for launch during the first quarter of calendar year 2012.

II. Launch Vehicle Selection

A wide array of launchers are available to deliver the payload. Many factors drive the selection of the best commercial launch systems.

In selecting a launcher for this mission the launch vehicle needs to be able 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 usual orbits considered for small launch systems. Usually for MEO orbits a medium-lift launch vehicle is selected, but due to the small payload mass (< 350 kg) launchers originally used for Low Earth Orbit (LEO) were considered.

Primary constraints for the SandiaSat mission are 1) cost, 2) security, 3) launch site availability. The payload is a part of a technology demonstration and not an operational mission, thus launch and delivery costs are a paramount consideration. The payload includes classified technologies, thus the launcher is required to be a USA licensed launch system. Also the high inclination (55°) orbit favors a launch out of NASA Wallops Flight Facility (WFF).ⁱ It is possible to reach this orbit from Cape Canaveral Air Force Base

(CCAFB), but this would require an inclination change during launch, which requires additional propellant, that in turn results in greater cost and less mass available for the payload.

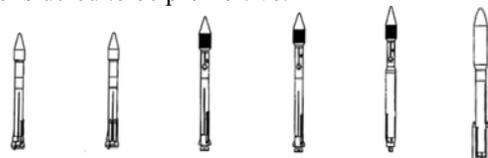
A preliminary trade analysis was performed to consider every available US launch system. The preliminary trade relied on manufacturers’ mission payload charts as well as impulsive ΔV calculations performed using data derived from independently published system information. Systems with launches available 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

The Current Federal Aviation Administration (FAA)-certified launch systemsⁱⁱ licensed to carry a classified USA payload are listed below (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 Atlasⁱⁱⁱ 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 for \$75-110 million. For this small technology-demonstration program the costs of these systems were considered to be prohibitive.



Vehicle	Atlas IA	Atlas IAS	Atlas IIA	Atlas IIB	Atlas V 400	Atlas V 500
Performance						
LEO Maximum	7316 kg	8518 kg	8640 kg	10,718 kg	12,500 kg	Up to 20,050 kg
GTO	3060 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. Payload data to LEO, to geostationary transfer orbit (GTO), launch costs in US dollars, date of the first flight, and available launch sites are all shown. 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. 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 uncertainty of availability makes the Delta II a high programmatic-risk option. For this single reason the Delta II, 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.

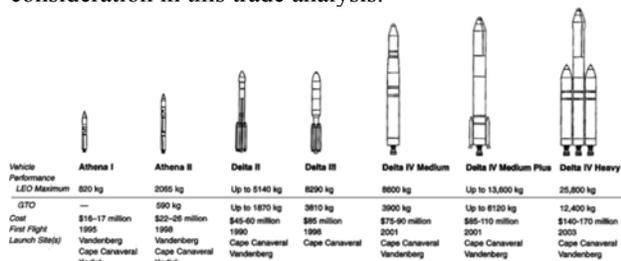


Figure 2. Athena and Delta Family of Launch Vehicles.

Figure 3 shows the characteristics of the Falcon^{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 and combined with the launch logistics eliminated this system 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.

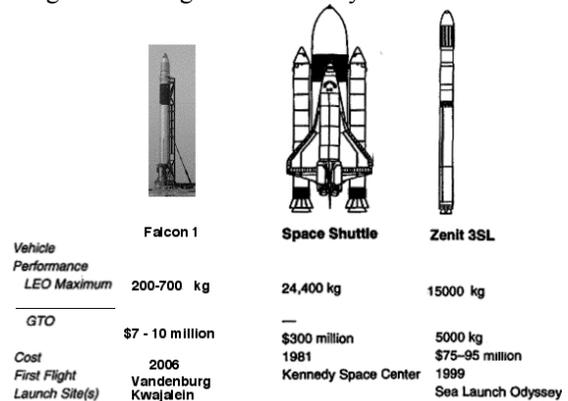


Figure 3. Falcon 1, Space Shuttle, and Zenit 3SL Launch Systems.

The family of launch vehicles from the Orbital Sciences Corporation (OSC) was also considered^{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)³. 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 from 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.

³ Verbal quote from OSC, \$35 million first flight, \$25-28 million recurrent costs thereafter.

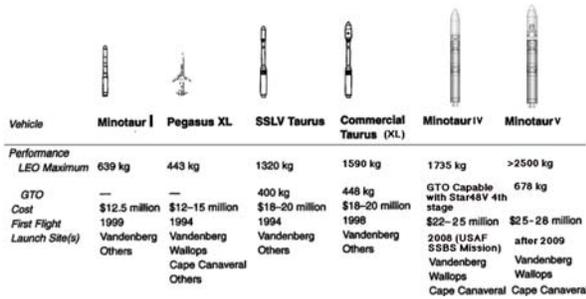


Figure 4. OSC Family of Launch Systems.

B. Secondary Launch Systems Trade Analysis

The four launcher that were placed on the acceptable “short list” included, 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 1. 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° without a plane change during the stage 3 burn. This would require a 5° plane change during launch which as stated earlier adds cost and mass to the system. This loss must be considered in the “short list” trade analysis.

Table 1. Minotaur IV, Minotaur V, Athena II, and Taurus XL Launch System Comparisons.

Minotaur IV	Athena II
<ul style="list-style-type: none"> - 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 	<ul style="list-style-type: none"> - 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	Taurus XL
<ul style="list-style-type: none"> - 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 incurred for first operational flight 	<ul style="list-style-type: none"> -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 1 must be capable of delivering, to an MTO orbit, this 340 kg payload plus the mass of the kick motor and propellant required to transfer from MTO to the final MEO orbit. Table 2 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 2. Required ΔV for MTO-to-MEO Orbit Insertion.

Orbit	Perigee km	Apogee 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 2 (1.3041 km/sec), Table 3 shows that a mass of 700 kg is required to be delivered to the MTO orbit by the launch system; a specific impulse of 285 sec, and an interstage mass of 100 kg are assumed.⁴

Table 3. 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

The propellant mass is calculated using a simple rocket equation analysis^{xv} assuming impulsive transfer from MTO to MEO (Eqn 1)

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

2. C3 Payload Analysis

A C3 launch energy analysis is done on the four launch vehicles listed in Table 1, in order to determine their lift capability. The published data for LEO, GTO and escape velocity are curve-fit and interpolated for the energy level of the MTO orbit specified in Table 2.^{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 (Eqn 2).

⁴ 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.

$$C3 = -2 \left[\frac{\mu}{2a} \right] + 2 \left[\frac{\Delta V_{maneuvering}^2}{2} - \frac{\Delta V_{track}^2}{2} \right] = \quad (\text{Eqn 2})$$

$$-2 \left(\frac{\mu}{R_p + R_e} \right) + (\Delta V_{maneuvering} + \Delta V_{track}) (\Delta V_{maneuvering} - \Delta V_{track}) \approx$$

$$-2 \left(\frac{\mu}{R_p + R_e} \right) + [\Delta V_{maneuvering} + \Omega_e R_e \cos(i)] [\Delta V_{maneuvering} - \Omega_e R_e \cos(i)]$$

In Eqn. 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, Ω_e is the earth's angular velocity, R_e is the local earth radius, and $C3$ is the specific launch energy. The term $\Delta V_{maneuvering}$ refers to any additional energy expenditures required, for example an orbital plane change. 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 for the four launch vehicles selected are presented in Figure 5. In these figures the payload mass delivered to a given orbit is plotted against the launch energy. The solid and dashed 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° inclination with $1000 \times 19,000 \text{ km}$ perigee and apogee altitudes. The plotted mass does not include the mass of the expended 4th stage. The Athena II data point does include the effect of having to change orbital planes by 5° during the launch.

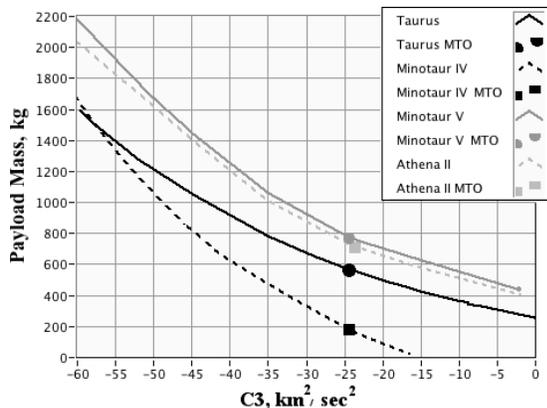


Figure 5. Launch Energy Comparisons for the 4 Candidate Launch Systems.

Table 4 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. 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. Thus the Minotaur V was selected as the optimal system for the SandiaSat mission.

Table 4. Payload Mass Delivered to SandiaSat MTO.

Launch System	Mass to 1000 x 1900 km altitude, 55° 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, and *viii)* summary of the design options considered.

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 the ATK Orion 38 motor as a 4th stage. The Minotaur V is a 5-stage evolutionary version of the Minotaur IV. In the Minotaur V configuration the larger ATK 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. A comparison of the Minotaur IV and V is shown in Figure 6.^{xx}

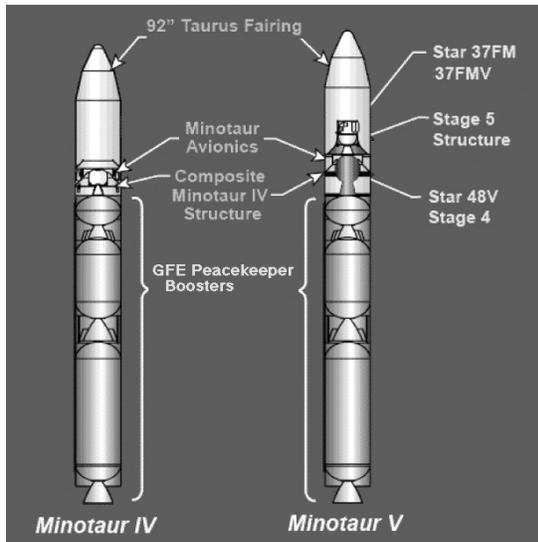


Figure 6. Comparison of the Baseline Minotaur IV and V Launch Vehicles.

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 Minotaur IV’s maiden flight is scheduled for late 2009 and will be a USAF payload. 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 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 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 a perigee altitude of 19,000 is approximately 688 kg. This mass is consistent with the required payload mass (701 kg) calculated and presented in Table 3. 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.

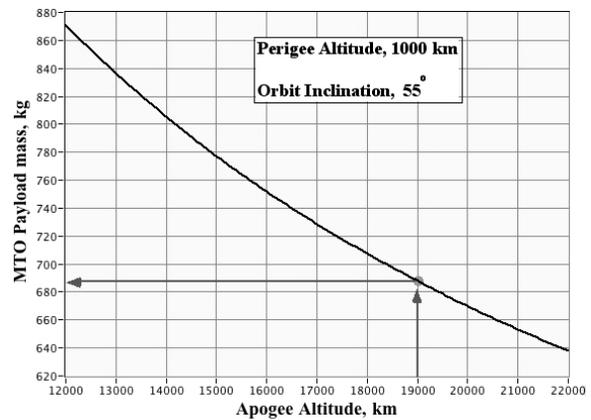


Figure 7. Payload Delivered to MTO by Minotaur V Stages 1-4.

The smaller Star 27 allows the MEO orbit to be achieved in five stages.^{xxi} The smaller 5th stage also allows more working volume within the payload fairing. The Star 27 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. The Star 27 provides a ΔV of 1.63 km/sec which is sufficient for final MEO orbit insertion. Recall from Table 2 a ΔV of 1.3041 km/sec is required for final orbit insertion (Eqn 3).

$$\Delta V = g_o I_{sp} \ln \left(1 + \frac{M_{prop}}{M_{payload} + M_{inert}} \right) \quad (\text{Eqn 3})$$

The ΔV produced by Star 27 motor is actually excessive by approximately 27% for a 340 kg payload mass, thus the motor may need to be off-

loaded. The Star 27 motor can be offloaded up to 20% with out rectification, by offloading the motor the mass of the delivered payload increases. Table 5 compares the Star 37 and Star 27 motors.

Table 5. 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

A direct simulation of the proposed Minotaur V configuration with the trajectory from launch to MEO insertion was deemed necessary to verify the initial 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. This three degree-of-freedom, USU developed, 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.^{xxii} The interactive simulation also allowed perturbed conditions about the optimal trajectory for Monte-Carlo analysis^{xxiii} 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. This data was collected from a variety of public domain sources.^{xxiv,xxv,xxvi} The equilibrium gas-chemistry code Chemical Equilibrium with Applications (CEA)^{xxvii} 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,^{xxviii} and a USU-developed incidence angle code for the supersonic flight conditions.^{xxix,xxx} As the rocket motors burned base drag was assumed negligible. During stage separation and the 4th stage coast, plume-off base drag characteristics were estimated using empirical correlations derived from Hoerner.^{xxxi} 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 stage 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 ΔV necessary for final MEO orbit insertion. The performance metric for optimization was the final mass payload delivered to the MEO orbit.

Figure 8 shows typical simulated launch altitude (a), downrange (b), velocity (c), and acceleration (d) time histories. The extended coast time (acceleration profile) 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.

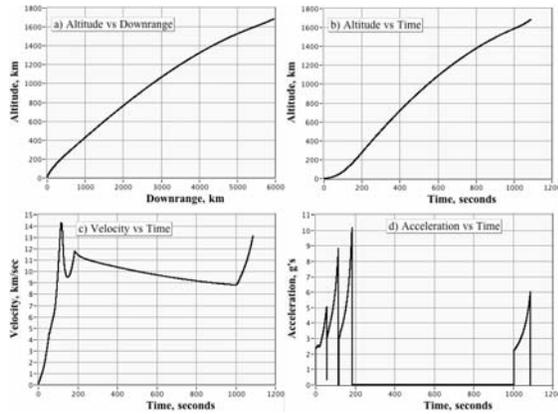


Figure 8. Typical Minotaur V MTO Insertion Launch Trajectory.

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.

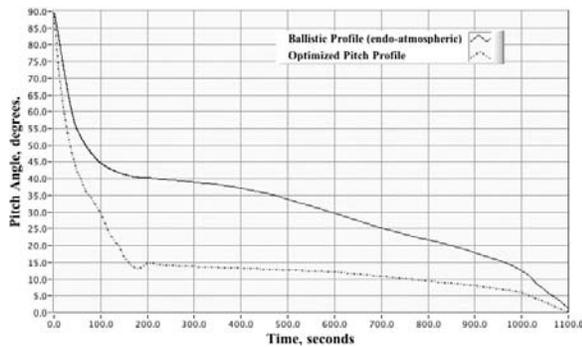


Figure 9. Optimized Launch Pitch Profile.

Figure 112 shows the optimized MTO trajectory plotted in the orbital plane (55° 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 111.

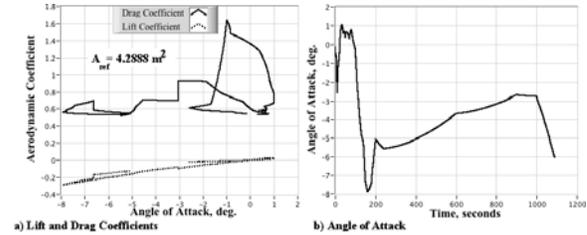


Figure 10. Aerodynamic Coefficients and Angle of Attack Along Optimized Launch Trajectory.

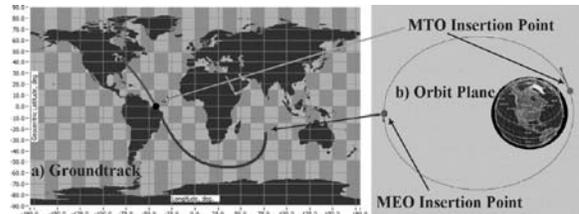


Figure 11. Optimized MTO Trajectory, Ground Track and Orbital Plane.

F. Mission Concept of Operations

An end-to-end launch and deployment concept of operations (CONOPS) is shown in Figure 13. 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 takes 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.

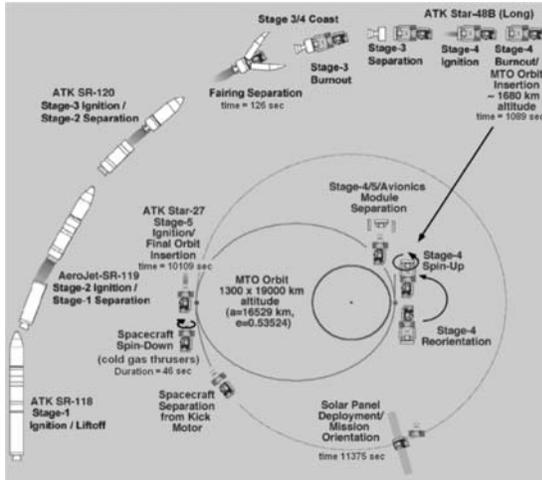


Figure 123. Launch and Deployment Concept of Operations.

G. Mass Budget Analysis

As previously mentioned the Star 27 provides excess ΔV for the orbit insertion. Consequently, off loading propellant allows more payload mass to be delivered to the final orbit. The required apogee kick ΔV (and hence the propellant off-load for the Star 27 5th stage) depends on the precise MTO trajectory energy level reached during the launch and MTO insertion phase of the mission. The 4th stage payload (satellite and 5th stage) mass delivered to MTO is approximately 682 kg. Table 6 shows the Stage IV mass budget resulting from the optimized MTO launch analysis. The delivered mass of approximately 682 kg into the 55° inclination MTO trajectory from WFF is comparable to the mass value presented in Table 4 that was calculated earlier using the impulsive analysis.

Table 6. Mass Budget for Optimized Launch and MTO Insertion.

<ul style="list-style-type: none"> • 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 7 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° inclination MEO orbit without a 6th stage kick motor.

The 360 kg maximum payload offers a 40% contingency for mass growth. A 26.4% propellant offload of the Star 27 is required for MEO orbit insertion.

Table 7. Mass Budget for Optimized MEO Insertion.

<ul style="list-style-type: none"> • Stage V • Total Spacecraft delivered to MTO Apogee, 681.3 kg <ul style="list-style-type: none"> -- Lightband Separation System <ul style="list-style-type: none"> Upper ring, 2.06 kg Lower Ring, 4.11 kg # of Separation Springs, 26 # of Separation Connectors and Switches, 12 -- Star 27 Kick Motor <ul style="list-style-type: none"> 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 Planetary Systems Corporation^{xxxii} 38" Motorized Lightband® was chosen as to be the separation system on SandiaSat.^{xxxiii} 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 Eqn 4.

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

In Eq. 4 ΔV is the relative velocity between the payload mass (M_{payload}) and the expended stage mass (M_{stage}). 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 8 shows the separation spring data published by Planetary Systems.^{xxxiv}

Table 8. 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 26 springs was selected. This number of spring results in a total Lightband resistance force of 2234N. This resistance force can be easily compressed by the weight of the satellite and makes for easier processing during payload installation. The 26 separation springs provide a separation ΔV of 1.05 m/sec.

If the 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 133 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. As shown in Figure 13 a possible recontact happens every two years. Over the maximum lifetime of the mission the closest approach is greater than 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 is not a great concern.

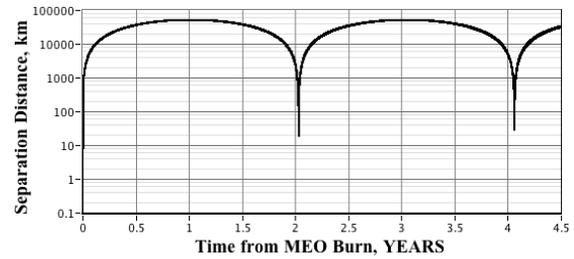


Figure 133. Time History of Separation Distance Between Payload and Expended 5th Stage.

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. Table 9 shows the 1- σ noise inputs to the Monte-Carlo model. A total of 997 data runs were performed to establish statistical validity (Ref. xxiii), and statistics of the final orbital parameters were calculated. Table 10 shows the end-to-end uncertainty estimates in the final MEO parameters. These uncertainties also include the effects of the Lightband separation ΔV .

Table 9. 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, $\pm 0.05\%$ total error (*deviation from optimal*)
... *assumes pitch guidance feedback*
 - Uncertainty in Drag Coefficient, $\pm 5\%$
 - Uncertainty in Lift Coefficient, $\pm 5\%$
 - Thrust Uncertainty, $\pm 0.1\%$
 - Burn Time Uncertainty, $\pm 0.1\%$
 - Total Impulse Uncertainty, $\pm 0.1414\%$
- 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, $\pm 1^\circ$
 - Burn Out-of-Plane Misalignment Uncertainty, $\pm 1^\circ$
- Lightband Separation Uncertainties
 - Separation ΔV Uncertainty, ± 0.1 m/sec
 - Separation Pitch Misalignment Uncertainty, $\pm 1^\circ$
 - Separation Out-of-Plane Misalignment Uncertainty, $\pm 1^\circ$

Table 10. End-to-End Uncertainty Estimates in Final MEO Orbit Parameters

Parameter	Mean Value	1- σ Std. 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, km	18975.8	+83.9
Apogee, km	19022.9	+86.2

J. Design Change Summary

The trade options and the effects of these changes to the MEO launch analysis is shown in Table 11. 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 11. 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 Lighband replaces original 92 spring option	None	minimal	Minimal

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°. 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 is presented.

The 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 1002.4 kg to MTO (681 kg 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 DV 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 $55^\circ \pm 0.1^\circ$,
- viii) Total mission time line from launch to final orbit is approximately 3.16 hours,

ⁱ Chiulli, Roy M., *International Launch Site Guide*, The Aerospace Press, El Segundo, CA, 1994, Chap. 13.

ⁱⁱ Isakowitz, Stephen J., Hopkins, Joseph P., Jr., and Hopkins, Joshua B., *International Reference Guide to Space Launch Systems*, 4th ed., American Institute of Aeronautics and Astronautics, Reston, VA, 2003, Chaps. 2, 5, 6, 8, 15, 25, 28, 32.

ⁱⁱⁱ Atlas, *Encyclopedia Astronautica*, <http://www.astronautix.com/lvfam/atlas.htm>, [retrieved 01 Dec. 2007].

^{iv} Athena 1, *Encyclopedia Astronautica*, <http://www.astronautix.com/lvs/athena1.htm>, [retrieved 01 Dec. 2007].

^v Athena 2, *Encyclopedia Astronautica*, <http://www.astronautix.com/lvs/athena2.htm>, [retrieved 01 Dec. 2007].

^{vi} Delta, *Encyclopedia Astronautica*, <http://www.astronautix.com/lvs/delta.htm>, [retrieved 01 Dec. 2007].

^{vii} DeltaII's Fate Worries Nonmilitary Users, *Wall Street Journal On-line*, http://online.wsj.com/article/SB118039764439516631.html?mod=googlenews_wsj, May 29, 2007, [retrieved 3 Dec. 2007].

^{viii} Falcon 1 Overview, *Space Exploration Technologies*, <http://www.spacex.com/falcon1.php>, [retrieved 02 Dec. 2007].

^{ix} Space Shuttle, *Encyclopedia Astronautica*, <http://www.astronautix.com/craft/spauttle.htm>, [retrieved 02 Dec 2007].

^x Zenit 3SL, *Encyclopedia Astronautica*, <http://www.astronautix.com/lvs/zenit3sl.htm>, [retrieved 02 Dec 2007].

^{xi} Falcon 9 Overview, *Space Exploration Technologies*, <http://www.spacex.com/falcon9.php>, [retrieved 02 Dec. 2007].

^{xii} Taurus Launch System Payload User's Guide, Release 4.0, *Orbital Sciences Incorporated*, March 2006, <http://www.orbital.com/NewsInfo/Publications/taurus-user-guide.pdf>, [Retrieved 15 May, 2007.]

^{xiii} Minotaur IV User's Guide, Release 1.1, *Orbital Sciences Incorporated*, January 2006, http://www.orbital.com/NewsInfo/Publications/Minotaur_IV_Guide.pdf, [Retrieved 15 May 15, 2007.]

^{xiv} Minotaur V Fact Sheet, http://www.orbital.com/NewsInfo/Publications/Minotaur_V_fact.pdf, *Orbital Sciences Incorporated*, [Retrieved 02 Dec. 2007].

^{xv} Sellers, Jerry Jon, *Understanding Space: An Introduction to Astronautics, 2nd Edition*, McGraw Hill, New York, 2000, Chaps. 7, 9.

^{xvi} Mosher, Larry, E., "Solar Terrestrial Relations Observatory (STEREO), Pre-Phase-A Requirements Review, Launch Vehicle," *John Hopkins University*, http://stereo.nrl.navy.mil/orig_stereo/PPA/PPA_RR2_secton_6.pdf, [Retrieved 03 Dec. 2007].

^{xvii} Vallado, David A., *Fundamentals of Astrodynamics and Applications, 2nd ed.*, Microcosm Press, El Segundo, CA, 2001, Chaps. 1,6.

^{xviii} Schoneman, Scott, Amorosi, Lou, Cheke, Dan, and Chadwick, Mark, "Minotaur V Space Launch Vehicle for Small, Cost-Effective Moon Exploration Missions," AIAA SSC07-III-2, *Presented at 21st AIAA/USU Conference on Small Satellites, Logan UT, August, 2007.*

^{xix} Peacekeeper Missile System, http://www.geocities.com/peacekeeper_icbm/specs.htm, [Retrieved August 15, 2007].

^{xx} Minotaur Launch System, *Encyclopedia Astronautica*, <http://www.astronautix.com/engines/peaeper1.htm>, [Retrieved August 15, 2007].

^{xxi} Star 37 Motor, ATK Star Motor Overview, *Alliant Tech Systems*, http://www.mission.com/starmotors/starmotors_star37.asp, [Retrieved: 6 October, 2007].

^{xxii} Evans, M. B., and Schilling, L., J., "The Role of Simulation in the Development and Flight Test of the HIMAT Vehicle," NASA-TM-84912, April 1984.

^{xxiii} Sobol Ilya M., A primer for the Monte Carlo Method, CRC Press, Boca Raton, FL, 1994, Chaps 1-3.

^{xxiv} Peacekeeper Missile, http://www.geocities.com/peacekeeper_icbm/specs.htm, Retrieved: 15 August, 2007.,

^{xxv} Encyclopedia Astronautics, <http://www.astronautix.com/engines/peaeper1.htm>, Retrieved: 15 August, 2007.

^{xxvi} ATK Launch and Mission Systems, *Alliant Tech Systems*, http://www.mission.com/starmotors/starmotors_star48v.asp, Retrieved: 15 August, 2007.

^{xxvii} Gordon, S., and McBride, B. J., "Computer Program for Calculation of Complex Chemical Equilibrium Compositions and Applications", NASA RP-1311, 1994.

^{xxviii} AEROCFD, *Apogee Rockets*, <http://www.apogeerockets.com>, Retrieved: 7 October, 2007.

^{xxix} Whitmore, S. A., "Conical Supersonic Flow Fields," *Utah State University*, URL: http://www.engineering.usu.edu/classes/mae/5420/Compressible_fluids/section11.html, Retrieved: 15 November, 2005.

^{xxx} Anderson, J. D. Jr., *Modern Compressible Flow with Historical Perspective, 3rd ed.*, McGraw-Hill Higher Education, New York, 2003, Chapter 4.

^{xxxi} Hoerner, Sighard F., *Fluid Dynamic Drag*, Self-Published Work, Library of Congress Card Number 64-1966, Washington, D.C., 1965, pp. 3-19, 3-20.

^{xxxii} Lightband Separation Systems, *Planetary Systems Incorporated*, <http://www.planetarysystemscorp.com>, Retrieved: 6 October, 2007.

^{xxxiii} Holemans, Walter, "The Lightband as Enabling Technology for Responsive Space," AIAA Paper RS2-2004-7005, *Presented at the Second Annual Responsive Space Conference, Los Angeles, April, 2004.*

^{xxxiv} Anon., "2000875 Rev – User's Manual for Mark II Lightband," *Planetary Systems Incorporated*, <http://www.planetarysystemscorp.com/download/2000785-UserManual.pdf>, [Retrieved 05 Dec. 2007].