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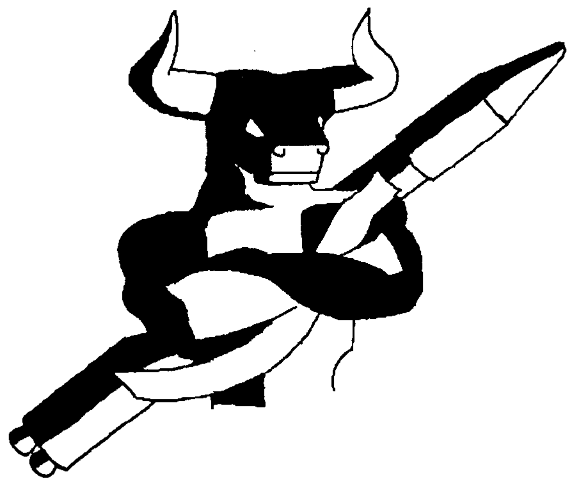

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The University of Texas at Austin
Department of Aerospace Engineering

Final Design Report

on

Converting the Minuteman Missile into a Small Satellite Launch System

Submitted to

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EXECUTIVE SUMMARY

Introduction

Due to the Strategic Arms Reduction Talks (START) treaty between the United States and Ex-Soviet Union, 450 Minuteman II (MMII) missiles were recently taken out of service. Minotaur Designs Incorporated (MDI) intends to convert the MMII ballistic missile from a nuclear warhead carrier into a small-satellite launcher. MDI will perform this conversion by acquiring the Minuteman stages, purchasing currently available control wafers, and designing a new shroud and interfaces for the satellite. MDI is also responsible for properly integrating all systems.

System Description

Figure 1 shows a representation of the MDI missile. The stages that MDI will acquire from the Air Force are the MMII stage I and II, and the MMIII stage III. These stages define the propulsion system of the MDI missile, and an analysis of attainable orbits is performed. This analysis is also performed if the satellite customer requires a STAR 17A or STAR 27 injection stage.

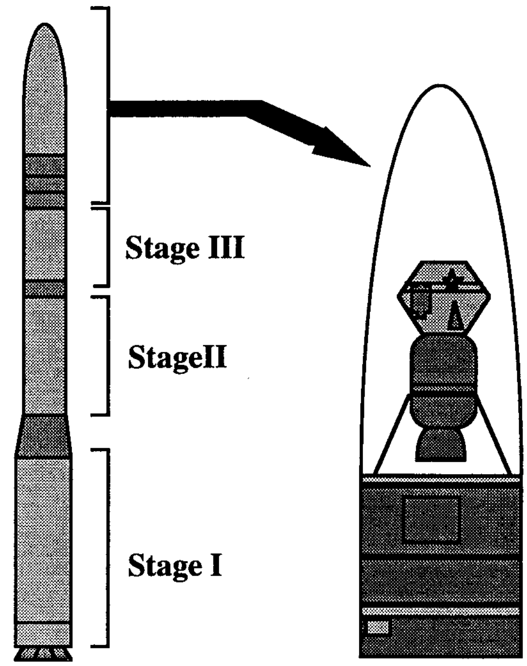

Figure 1. MDI missile

The new MDI system still incorporates the original range-safety raceway and attitude-control actuators. MDI plans to purchase the 52 inch diameter avionics, range-safety, and attitude-control wafers from Martin Marietta's Multi-Service Launch System (MSLS), "D" configuration missile, which is currently under development.

MDI has designed a mechanical interface to the payload, that uses four truss elements and a Marmon clamp, which can wrap around various satellite diameters (see Figures 2 and 3). The support structure is able to support up to 1500 lbm. The payload support bulkhead has an allotted interface for any electrical connections the satellite requires. MDI has also sized a new shroud that fits various small satellites and uses a thrusting-joint-separation mechanism.

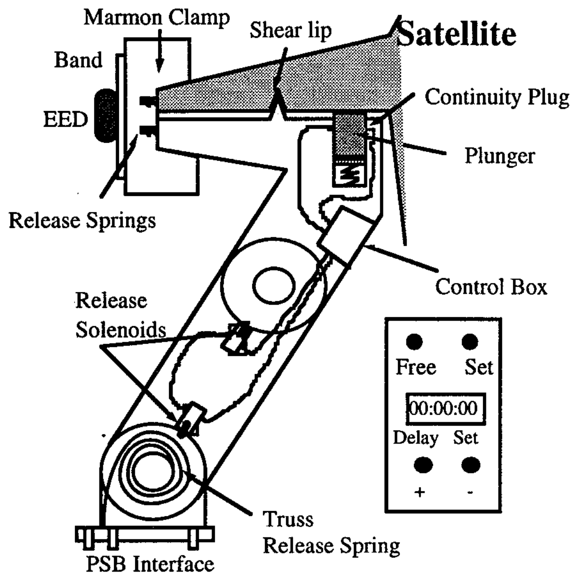

Figure 2: Support Structure

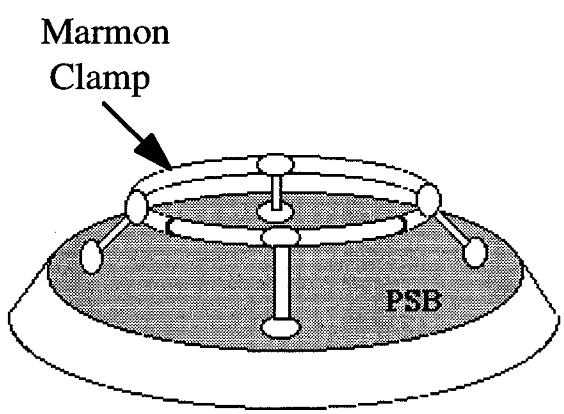

Figure 3: Payload Support Bulkhead and Marmon Clamp

MDI Missile Performance

The primary launch site is Vandenberg Air Force Base (VAFB) in California, and a secondary launch site is also available at Cape Canaveral Air Force Base in Florida. Combined, these two sites provide for a wide range of orbit inclinations (29° - 104°). Figure 4 displays attainable orbits when the missile is launched from VAFB with various injection stages at permissible orbit inclinations. As the figure indicates, the MDI missile can launch payloads a big as 1000 lbm to circular orbits ranging from 100 to 2500 nautical miles (nm), depending on orbit inclination and what injection stage, if any, the customer wishes to use. The expected launch environments (temperatures, pressures, and vibrations) the satellite will be exposed to during launch are carefully studied and taken into consideration in the design of the MDI launch vehicle.

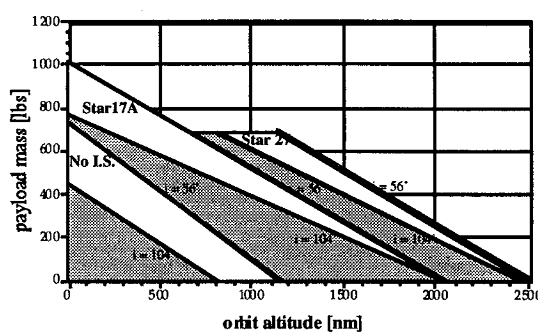

Figure 4: Attainable Orbits from VAFB

Mission Concepts

To stress the feasibility of its launch vehicle, MDI has performed two satellite mission designs as proof of concept. The two payloads chosen are BATSAT and the Asteroid Investigation Microspacecraft (AIM). Each payload has been properly interfaced with the missile, and MDI shows how the orbit needs are attainable with the launch system.

Launch System Competitiveness

To show launch system competitiveness, the MDI missile is compared to other small satellite launch systems (Scout and Pegasus). All systems have comparable attainable orbits, insertion errors, and usable shroud volumes. However, the MDI missile's launch costs ($7.5 million) are considerably less than the other launch systems ($12, $15 million)

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LIST OF ACRONYMS

ACS	Attitude Control
AFB	Air Force Base
AIM	Asteroid Investigation Microspacecraft
ARSS	Airborne Range Safety System
EED	Electrical Explosive Device
GPS	Global Positioning System
GNC	Guidance, Navigation and Control
HSS	Horizontal Separation System
ICBM	Inter Continental Ballistic Missile
IMU	Inertial Measurement Unit
MDI	Minotaur Designs Incorporated
MGS	Missile Guidance Set
MMII	Minuteman II ICBM
MMIII	Minuteman III ICBM
MSLS	Multi-Service Launch System
PAF	Payload Attach Fitting
PSB	Payload Support Bulkhead
PSD	Power Spectral Density
RF	Radio Frequency
R&D	Research and Development
RSLP	Re-entry System Launch Program
START	Strategic Arms Reduction Talks
TVC	Thrust Vector Control
USRA	Universities Space Research Association
VAFB	Vandenberg Air Force Base
VSS	Vertical Separation System

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1.0 INTRODUCTION

With the continuing decrease in funding for space-exploration projects, a greater need for cost-effective launch systems has developed. Recently, 450 Minuteman II (MMII) Intercontinental Ballistic Missiles (ICBM) have been taken out of service as part of the Strategic Arms Reduction Talks (START) peace treaties between the United States and the Ex-Soviet Union [1]. Consequently, converting these missiles from warhead to satellite launchers may provide a cost-effective method for future payload launches.

Two existing programs already make use of MMII stages: the Re-entry Launch System Program (RLSP) and the Multi-Service Launch System (MSLS). The RLSP uses MMIII stages to propel research payloads to sub-orbital trajectories. The MSLS, which is being developed by Martin Marietta, consists of several rocket configurations (A, B, C, and D) that use variations of MMII and MMIII stages. These configurations are displayed in Figure 1.1 . Martin Marietta has also computed the expected launch performance for each rocket configuration, and this data is shown in Figure 1.2 [2].

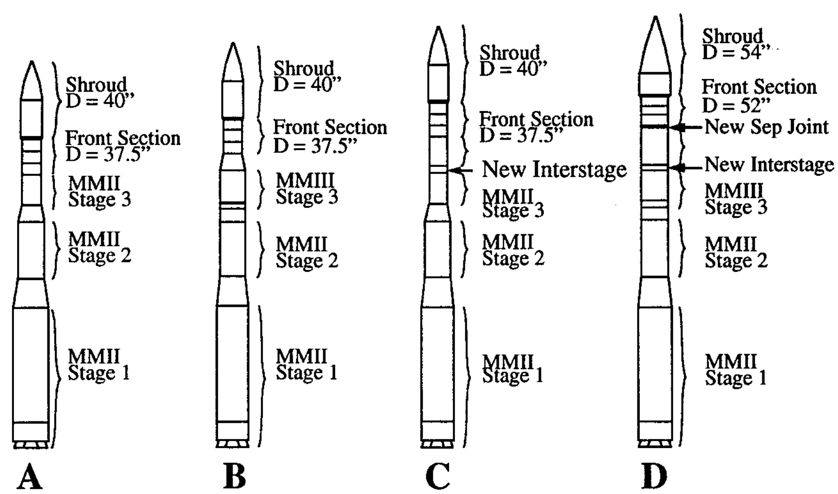


Figure 1.1 MSLS configurations

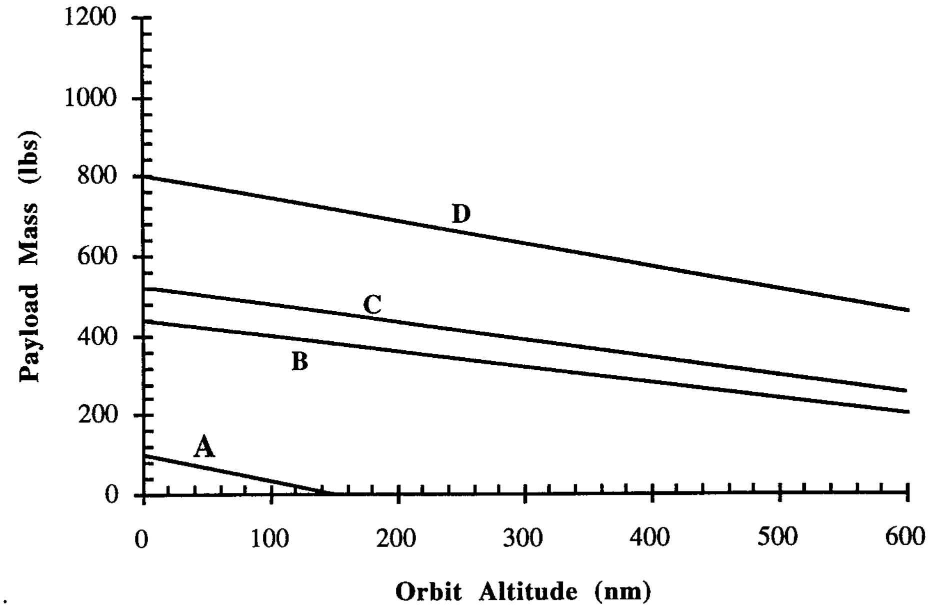


Figure 1.2 Launch Performance for the MSLS Rocket configurations

The Universities Space Research Association (USRA) has taken special interest in converting the MMII rocket into one that can deploy small satellites into orbit. Minotaur Designs Inc.'s (MDI) objective is to produce such a satellite launch system by making use of available MMII systems.

Figure 1.3 shows the design-process that MDI followed to meet its design objective. As the figure indicates, several inputs were considered, such as the mission requirements, the existing satellite market needs, and the expected launch environments. A conceptual design was analyzed and sized through a series of trade studies, which developed into MDI's current preliminary design. To prove system competitiveness, MDI compared its launch vehicle to other small-satellite launch vehicles such as the Pegasus and the Scout. In addition, MDI decided to show a proof of concept by designing two specific satellite mission concepts. The second half of the diagram indicates the future work needed to meet the end goal of a building reliable launch vehicle.

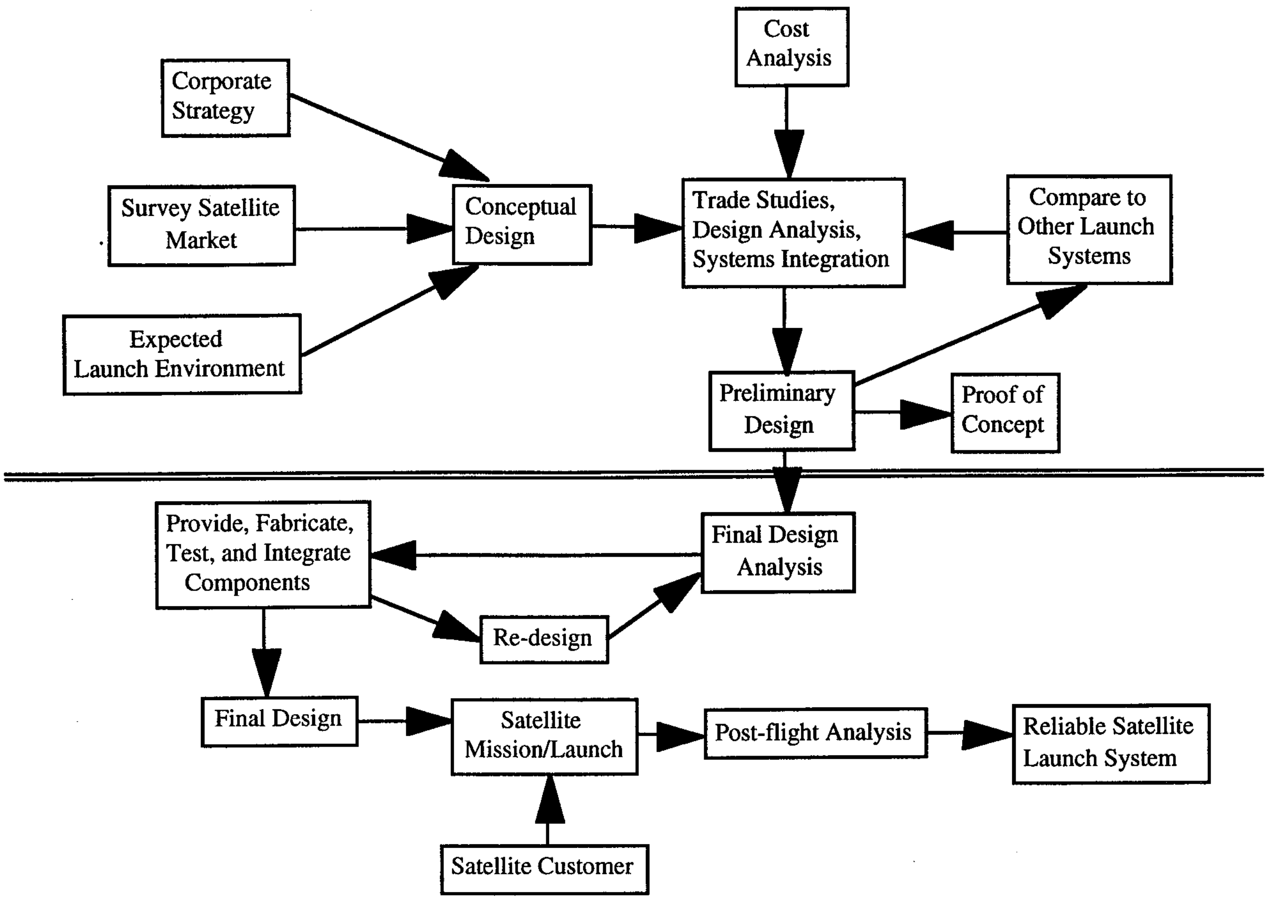


Figure 1.3 Design-process diagram

2.0 MISSION GOALS

MDI's goal is to produce a launch system design that incorporates available MMII systems. The system uses launch pads at Vandenberg AFB as the primary launch site but also can use Cape Canaveral as a secondary launch location. MDI also analyzes the performance gained by using an injection stage to provide a wider range of orbits. Satellite interfaces must be interchangeable and modular so that a variety of satellites can be housed. The launch system should be reliable, competitive, and oriented to the satellite customer. In addition, MDI requires that the launch system be a profitable venture.

3.0 SATELLITE SURVEY

MDI surveyed a number of potential satellite customers. The results of those surveys are included in Appendix A. All of the satellites surveyed fall within the structural and dimensional limits of the MDI design and are launchable with the MDI missile. All of these satellites are viewed as tentative systems for MDI launch. The specific diameters, heights and weights are shown in Figures 3.1 through 3.3.

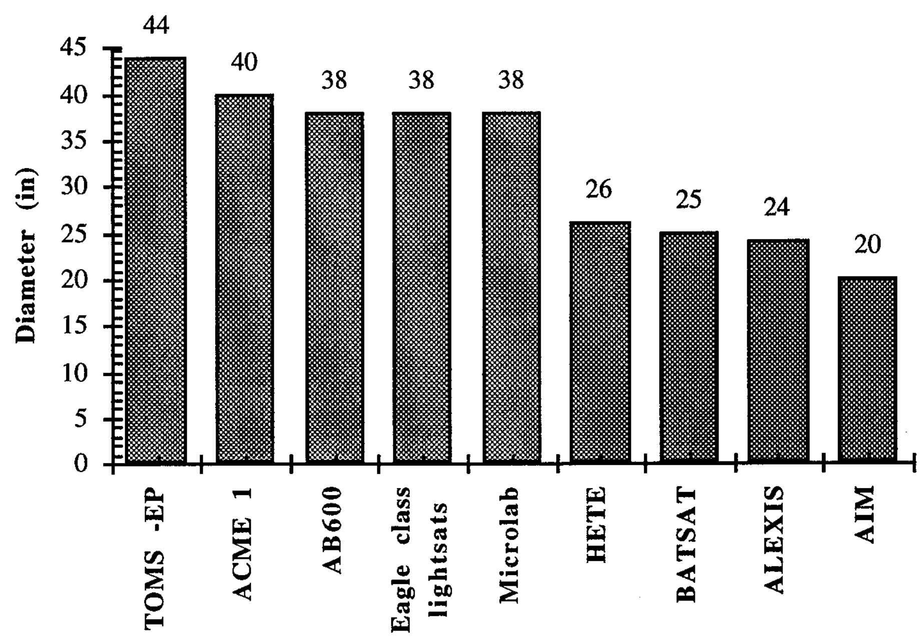


Figure 3.1 Diameters for various satellites.

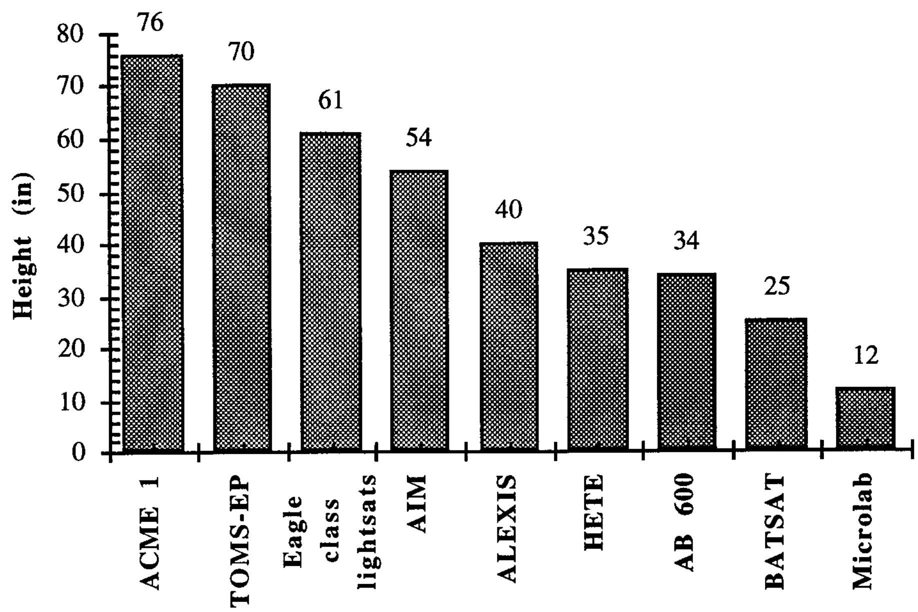


Figure 3.2 Heights for various satellites.

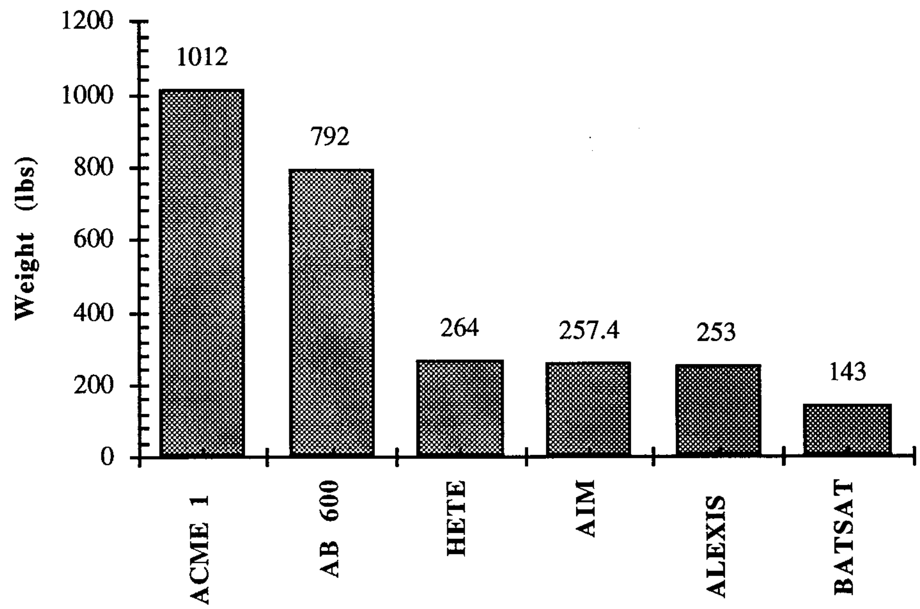


Figure 3.3 Weights for various satellites.

4.0 SYSTEM DESCRIPTION

MDI will acquire MMII stages I and II, MMIII stage III from the U.S. Air Force, and purchase STAR 17A or STAR 27 injection stages from Thiokol. MDI will also purchase previously designed control wafers, all of which are currently available in the market. Furthermore, MDI will also design a missile-to-satellite mechanical and electrical interface as well as a shroud; these designs will then be sub-contracted out in order to minimize cost. MDI will be responsible for the overall integration of all the missile systems. Figure 4.1 shows the launch system components.

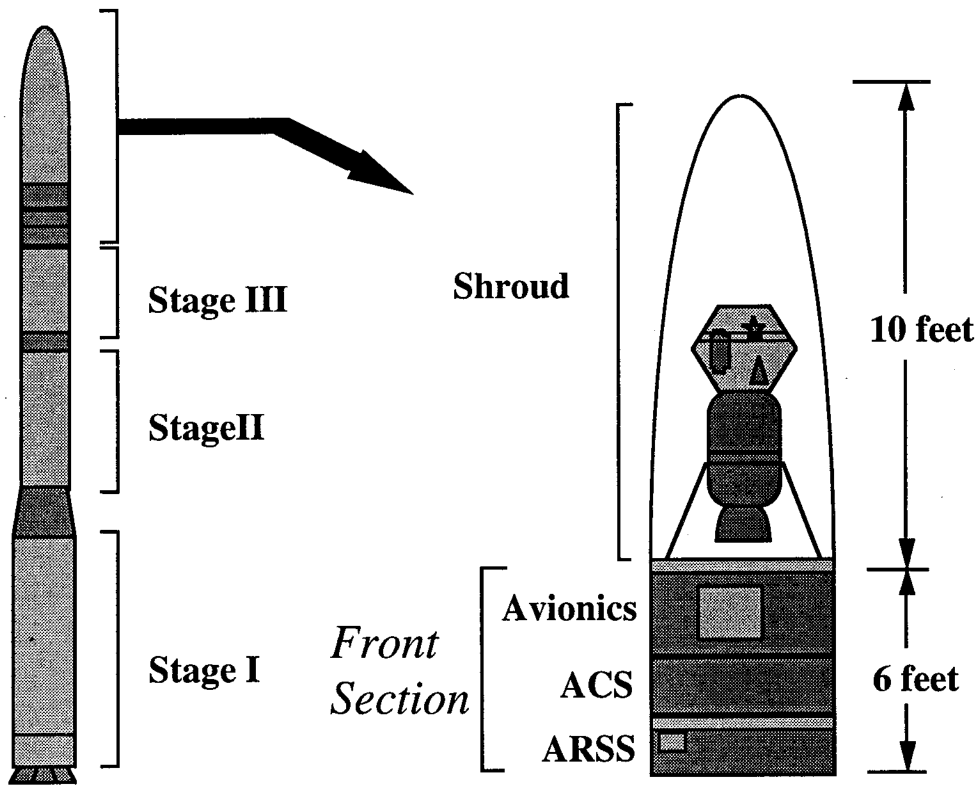


Figure 4.1 MDI missile systems

The Minuteman stages will come with attitude-control actuators that the MDI avionics wafer will command. MDI will also use the range-safety raceway that was originally built with the stages. The MSLS, "D" configuration avionics wafer, attitude-control wafer, and range-safety wafer will be purchased from Martin Marietta. MDI has designed a Marmon clamp, which uses 4 truss elements to connect the payload to the missile. The structure can support a payload of 1500 lbm during the launch. Upon orbit insertion, this mechanical interface will release the satellite. MDI also designed a shroud that uses a thrusting-joint-separation system. All systems are discussed in more detail in the following sections.

4.1 Propulsion

The propulsion system provides the necessary thrust to insert a satellite into a desired orbit. As shown in Figure 4.2, MDI chose stages I and II from the MM II and stage III from the MM III. Also, an optional stage that MDI may use is a Thiokol STAR 17A or a STAR 27 injection stage. These injection stage options provide for a wider range of achievable orbits. This range of orbits is discussed in the "expected performance" section.

MDI originally intended to use a MMII stage III for the third stage, but previous MSLS analysis has shown that the orbit performance of a standard MMII missile (MSLS, configuration A) is limited to low payload masses at low orbital altitudes [2]. The same analysis shows that using a MMIII stage III for the third stage (MSLS B configuration) adds significant attainable orbit performance as shown in Figure 1.2. The MSLS configuration "B" missile's stages are equivalent to the MMIII's, therefore properly mating these stages would require little work if any. Thus MDI opted to slightly increase system complexity by using a MMIII stage instead of a MMII because of the large performance gain.

In choosing the injection stages, MDI considered several constraints: adaptability, reliability, payload weight and volume, and rocket performance. For reliability purposes, MDI's search for injection stages was limited to the most commercially used stages, such as the STAR engine family [3]. The total injection stage plus satellite weight must not exceed a support-structure criteria of 1500 lbm. The injection stages are not to reduce the potential payload mass to less than 200 lbm. Also, the injection stage must fit in the designed shroud. Finally, each injection stage should provide a significantly improved performance, leading to a variety of attainable orbits. After considering all these requirements, MDI chose the STAR 17A and the STAR 27 as the two solid rocket engine options. Table 4.1 shows the solid rocket stage properties of the propulsion system [4].

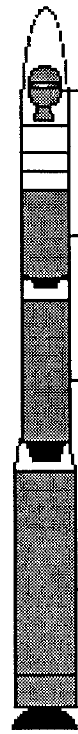


Figure 4.2 Propulsion system

Table 4.1 Rocket stage properties

STAGE	Isp (sec)	MASS Dry (lbm)	Propellant (lbm)
I	268	4680	45730
II	287	1753	13720
III	285	744	7289
STAR 27	286	53	741
STAR 17A	280	30	246

4.2 Avionics

The missile's avionics wafer is responsible for Guidance, Navigation and Control (GNC). GNC consists of computing and controlling the missile's state (position, velocity, attitude, and attitude rates). This wafer sits highest on the front section, as shown in Figure 4.3. The avionics wafer holds the IMU and flight computer. In addition to housing

GNC devices, this wafer houses batteries, receivers, and transmitters and is responsible for electrical power, sequence of events during launch, and communication systems.

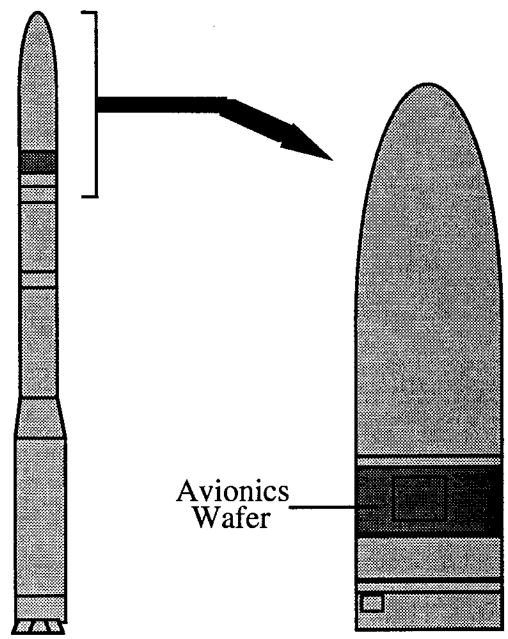


Figure 4.3 Avionics wafer

MDI decided to select and purchase a pre-designed or existing avionics wafer that would meet the design requirements. Consequently, MDI considered the following as potential candidate avionics wafers: existing MMII avionics, RSLP Missile Guidance Set (MGS), and MSLS configuration A, B, C, and D wafers.

Upon careful consideration, MDI chose the MSLS configuration "D" avionics wafer. The driving factor was a desire to keep the shroud and front section flush with the third stage. Of the four MSLS choices, only the "D" configuration has a 52" diameter avionics wafer, as shown in Figure 1.1. (The desire to keep wafer's flush is discussed in the "performance" and "shroud" sections) Additional motivation for not selecting the ICBM avionics is that they are designed for sub-orbital flights and would need to be reprogrammed. Additional motivation against selecting the ICBM or RSLP avionics is that they contain old components that are currently obsolete.

The MSLS D wafer uses a Litton LN-100 Inertial Measurement Unit (IMU) [2]. This IMU uses laser-gyroscopes and accelerators to measure directional accelerations. The LN-100 also includes a flight computer that computes current states and controls launch events (all except the injection stage burn, if any). This computer is programmable, and the launch profile can be modified depending on the mission [5]. The transmitters and receivers are used to communicate with the ground and to relay data regarding the launch. During launch, antennas have line-of-site view with the ground and are attached flush with the outside surface of the wafer.

4.3 Attitude Control

For all phases of the launch, the attitude-control system must maintain the proper orientation of the missile to ensure accurate orbit insertion. The avionics wafer will be responsible for issuing commands to the attitude-control actuators up to the time of the payload separation, and an injection stage will need its own logic to control its proper attitude.

During first-stage flight, the first stage will use differential gimballing of the four nozzles to control roll, and engine gimballing to control pitch and yaw. During second and third-stage flight, the respective stage will use nozzle gimballing to control pitch and yaw, and four tangential thrusters to control roll. For any necessary coasting phases (described in the "performance" section), an attitude-control system (ACS) wafer will maintain proper orientation. This ACS wafer will use four radial thrusters for pitch and yaw and four tangential thrusters for roll. Figure 4.4 shows the actuator locations.

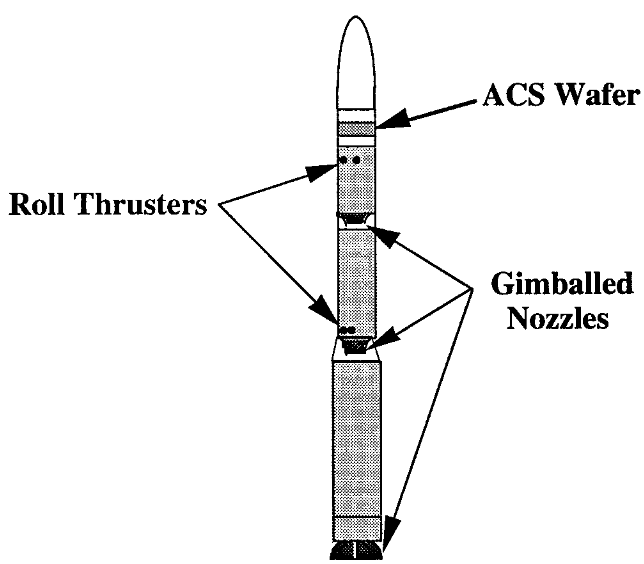


Figure 4.4 Actuator locations

The minuteman stages contain the attitude-control actuators with which the stages were originally designed. Since they are robust enough to control a sub-orbital ICBM trajectory, they should be adequate for MDI's purposes. The ACS wafer will be purchased from Martin Marietta's MSLS program, and as mentioned before, a "D" Configuration wafer will be used to keep the Front section flush with the rocket stage. This MSLS ACS wafer uses cold nitrogen gas to provide the necessary thrusts.

If an injection stage is used, it will also need some form of attitude control. Since it can not receive commands from the avionics wafer, the satellite will have to control the injection stage. Therefore, the satellite customer will be required to purchase and provide command controls for the injection stage. The STAR injection engine will have Thrust Vector Control (TVC), which will use engine gimballing to control pitch and yaw. The TVC actuator used to bend the flexible nozzle is shown in Figure 4.5.

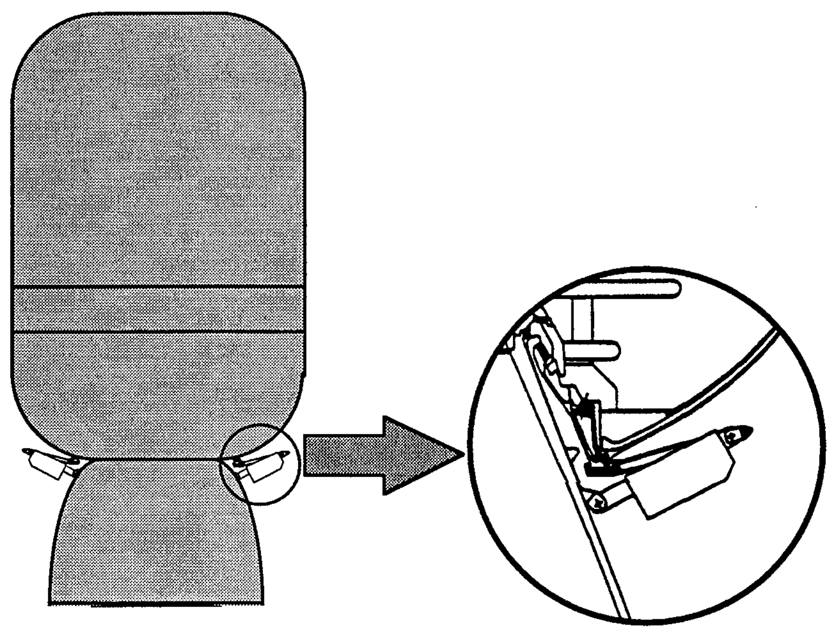


Figure 4.5 Cut-away of TVC actuator

4.4 Range Safety

In the event of a hazardous situation, ground control must be able to destroy the launch vehicle. The Airborne Range Safety System (ARSS) wafer provides the solution for this need. The ARSS uses range transponders, control logic, and a raceway that detonates the stages in case of a major malfunction.

The range transponders are used so that ground stations can track the missile independently of the avionics wafer's computed state. If these transponders fail, a skin track can still be performed. A command logic that resides in this wafer issues a destruct command that will detonate the raceway's explosives in the stage casings (see Figure 4.6).

The stages will be purchased with the originally designed missile raceway. The MSLS, configuration "D", ARSS wafer will be purchased, and MDI will make sure the wafer is properly mated with the raceway. The MSLS ARSS wafer is designed with redundancy to ensure safety. Each system has a one-fault tolerance. Again, configuration "D" wafers were chosen so that the front section and shroud would be flush. This wafer will be discarded with the third stage when it has outlived its purpose.

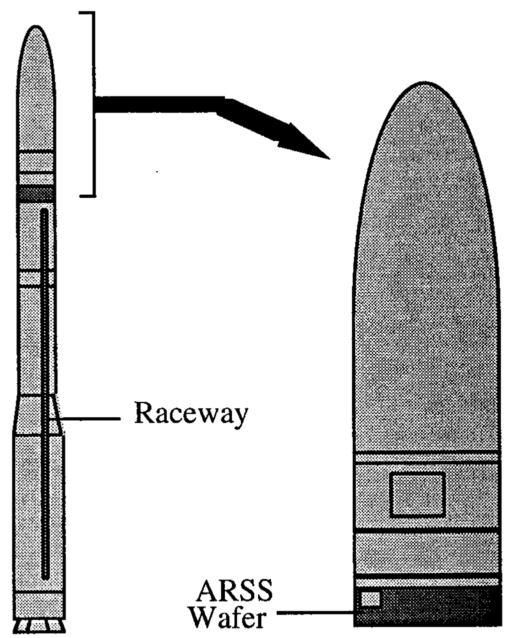


Figure 4.6 Range Safety System

4.5 Support Structure

MDI designed the support structure (shown in Figure 4.7) that attaches the payload to the missile. This support structure is composed of a Payload Support Bulkhead (PSB) and four trusses. The PSB is a standardized structure that can support four trusses. These trusses transfer the satellite loads during launch and are configurable for any size payload up to the maximum diameter of 48 inches. This maximum diameter is dictated by the shroud diameter of 52 inches and a 2 inch safety margin on each side.

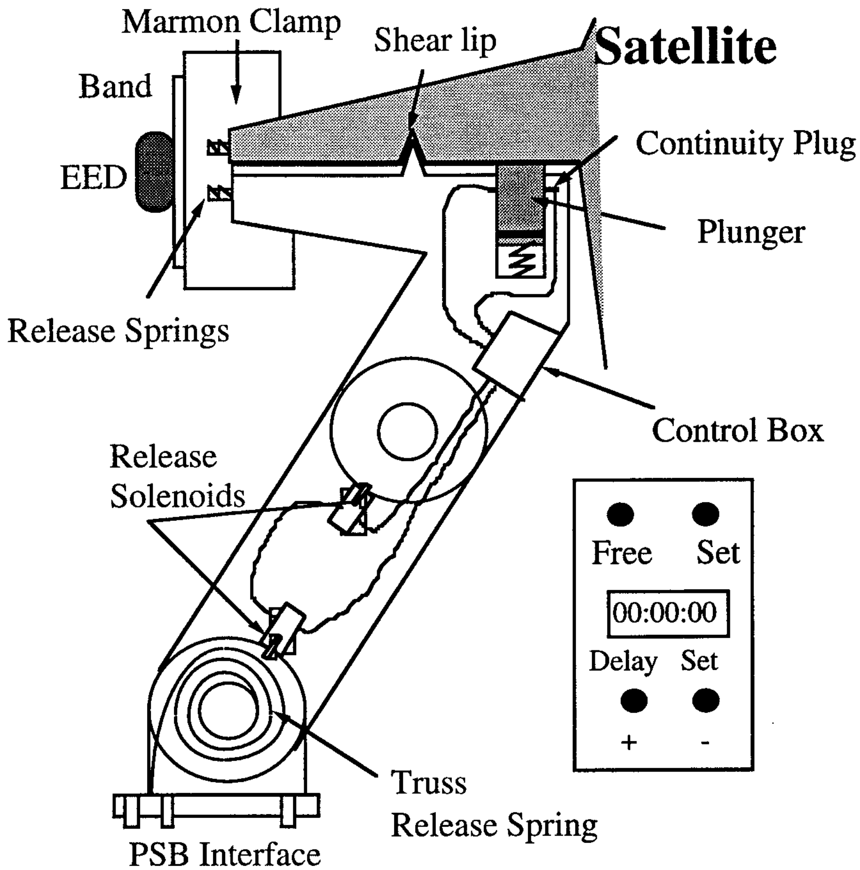


Figure 4.7 Support structure

4.5.1 Design Requirements and Strategy

The support structure can cradle up to a 1500 pound payload and can withstand a sustained 9 g axial and 3.75 g tangential acceleration, the most extreme expected launch environment [6]. MDI determined the upper limit on the payload weight (which includes an injection stage if one is used) by examining the attainable orbits as a function of satellite mass (see "Performance" section). The structure must also accommodate various payload heights and diameters. Finally, the support structure should provide easy access to the payload during ground operations so that the customer can perform any necessary modifications before launch.

4.5.2 Support Structure Design

The support structure consists of a payload-support bulkhead (PSB) mounted on top of the avionics section, four truss elements mounted on the PSB, and a Marmon clamp that connects to the payload.

The PSB, seen in Figure 4.8, is based on the RSLP bulkhead and is designed to mount a variety of load-bearing structures and support the payload electrical interface. Each circle represents a connection point that consists of 5 slots. Each truss is bolted through the slots, and each connection point has a total tension pull-out capability of 15,000 lbm [6, 3-11]. The triangular object represents the electrical- interface connector, but the actual connection point is circular to minimize stress concentrations.

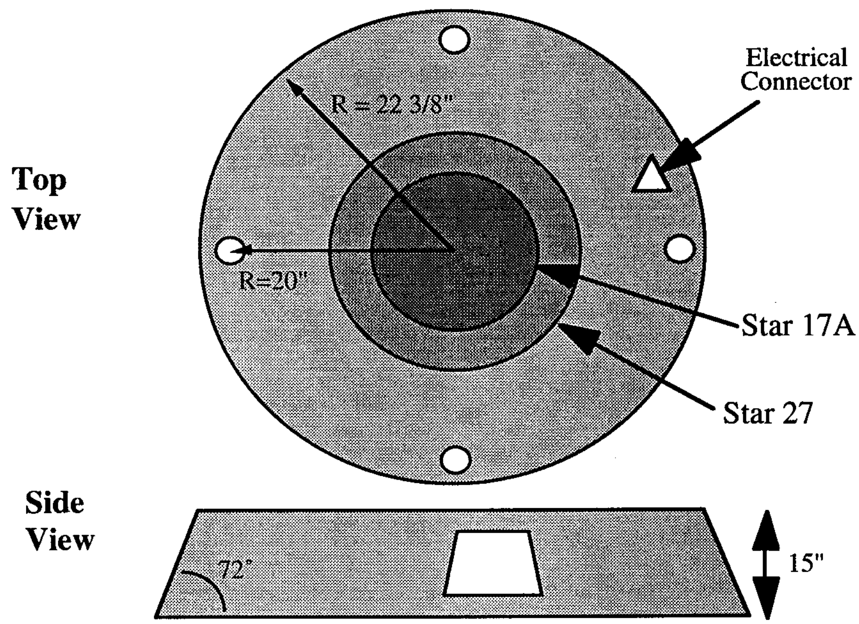


Figure 4.8 Payload support bulkhead

Table 4.2 PSB structural properties

Material	Aluminum 7075
Finish	Alodine 600
Thickness	2.0 inches
Weight	25 lbm
Truss Tension Pullout Capability	15,000 lbm per truss

Additional load-bearing structures consist of truss elements connected to the central clamp. The structure is shown in Figure 4.9, and truss properties are shown in Table 4.3. Each truss element consists of a rod, a clamping end, and two hinges. These hinges can be set to properly fit various payloads. Upon payload separation, the rod elements rotate outward to prevent contact with the payload.

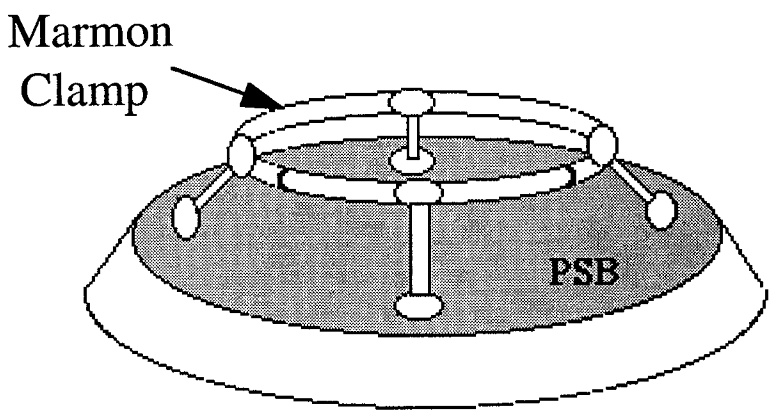


Figure 4.9 Tubular truss structure

Table 4.3 Single truss element properties

Material	Steel
Young's Modulus	30,000 psi
Poisson's Ratio	0.24
Cross Sectional Area	1.00 in
Length	36.00 in
Weight	8 lbm.

The trusses are connected to a payload lip with four wedges and a cinching ring, as seen in Figure 4.7. When a separation discrete is issued, the Electrical Explosive Device (EED) cuts the cinching ring; springs eject the wedges radially, and other springs eject the payload: The clamp is designed with redundant separation initiators that send signals to the EED.

In the case when the satellite needs an injection stage, MDI can provide a payload-attach fitting (PAF). The MDI PAF design uses the same truss element as the support structure does, except two clamping ends are attached to each side of a rod. These clamping ends are attached to the injection stage and satellite as shown in Figure 4.10. If the satellite needs to separate from the injection stage, a Marmon clamp with an EED will be used for the upper attachment, while a permanent clamp will be used for the lower. If the satellite does not need to separate from the injection stage, the two PAF clamps will be permanent.

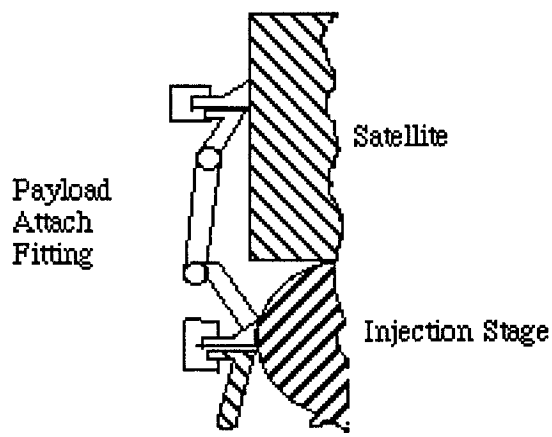


Figure 4.10 Payload-Attach Fitting (PAF) with injection stage

4.6 Electrical Structure

The electrical structure connects all the electrical systems. MDI has designed a wiring harness that connects all required and optional electrical systems. MDI has also defined the satellite interface, the flight data sensor connection and the missile interface.

4.6.1 Design Requirements and Strategy

The electrical structure is required to interface the system components, relay information to ground control, initiate separation mechanisms and confirm separation. The electrical structure is also required to determine the missile attitude and to control the state vector.

MDI has decided that the electrical structure will not be mission specific; therefore MDI has defined a wiring harness that connects all conceived electrical systems. Refitting the wiring harness for mission specific requirements would be the satellite customer's responsibility.

4.6.2 *Electrical Structure Design*

The control-block diagram shown in Figure 4.11 shows the components that are supported by the electrical structure. The flight computer is housed with the IMU inside the avionics wafer. This computer is programmable to accommodate different launch profiles.[5]

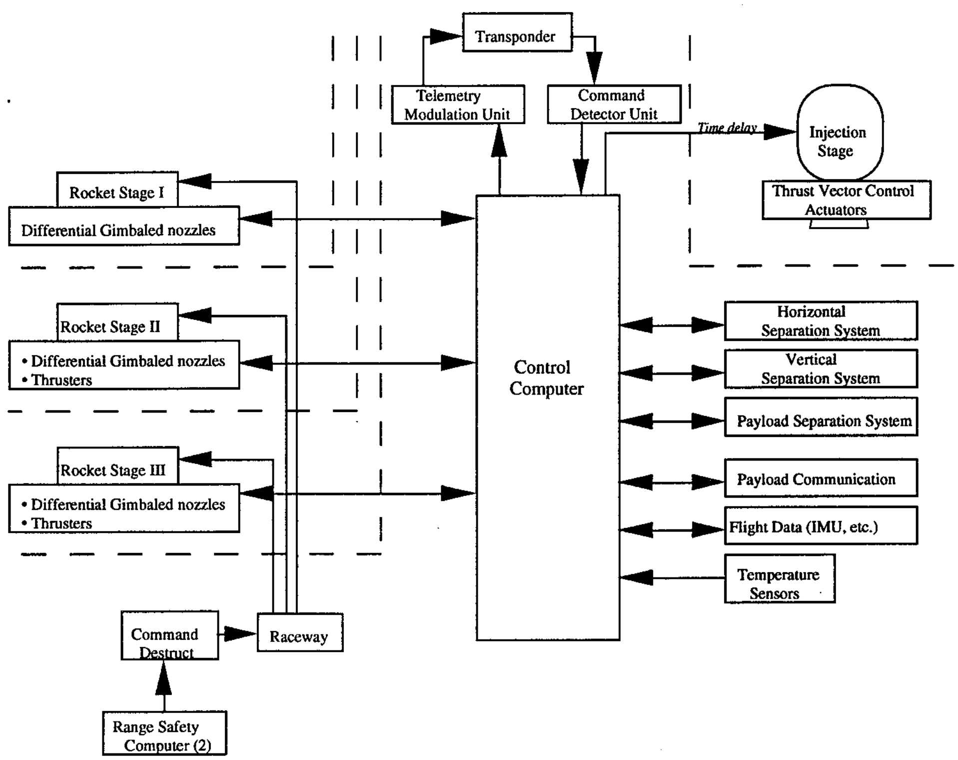


Figure 4.11 Electrical structure

As Figure 4.11 shows, the computer receives data from the IMU's accelerometers, which is housed in the same wafer. The computer electrically interfaces with all the missile's attitude actuators across the stage separation planes. There are also electrical interfaces between the computer and telemetry, and between the range safety wafer and raceway. All the previously mentioned electrical interfaces are under the avionics wafer and have been designed by Martin Marietta.

There are also electrical interfaces from the avionics through the bulkhead. The shroud's thrusting-joint horizontal and vertical separation systems are commanded to separate by the avionics wafer. In addition, flight data from a series of transponders (vibration, acoustic, temperature, pressure) housed in the shroud are sent through the avionics wafer to the ground. Finally, a payload-separation discrete is sent through the bulkhead and causes the clamp to release the satellite or injection stage.

4.7 Shroud

The shroud shown in Figure 4.12 covers the payload and protects it during the atmospheric portion of the flight. The shroud is an expendable structure and is jettisoned after the missile leaves the atmosphere.

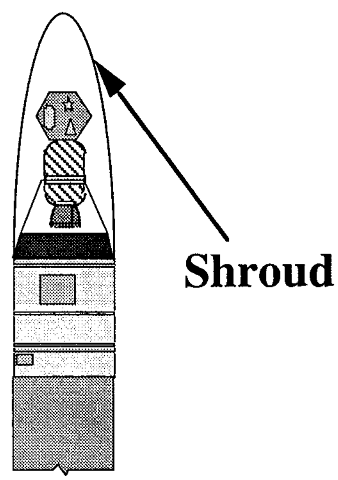


Figure 4.12 Missile shroud

4.7.1 Design Requirements and Strategy

The first requirement for the shroud is to be Radio Frequency (RF) transparent so that satellite vendors can communicate with their payloads at any time during the launch if necessary. Also, the shroud must allow easy access to the payload so that vendors can access their payload easily once the shroud is on. Next, the shroud-separation mechanism must be designed so that minimal shock is imparted to the payload during shroud separation in order to protect the payload. Finally, the shroud should be lightweight and have a simple, redundant mechanism for separation.

As for the design strategy, MDI has taken the initiative to use new technology to design the shroud. The design of the shroud relies on maximizing the usable volume of the payload and minimizing the shroud weight. In addition, the larger diameter of the shroud increases the number of potential satellite customers.

4.7.2 Shroud Design

The MDI shroud is a fiberglass composite two-part structure as shown in Figure 4.13. The composition of the shroud makes it RF transparent. The shroud uses a doubly redundant thrusting-joint separation mechanism with a Horizontal-Separation System (HSS) and a Vertical-Separation System (VSS).

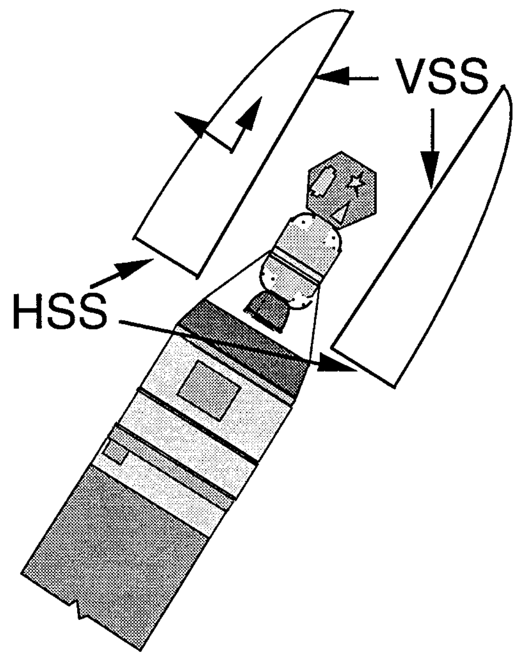


Figure 4.13 MDI shroud

The basis for the MDI shroud design is the MSLS shroud that has a 40 in. outside diameter and a 150 in. height as shown in Figure 4.14. The goals of the MDI shroud are to have comparable drag and weight, as well as a more efficient use of volume than the MSLS shroud.

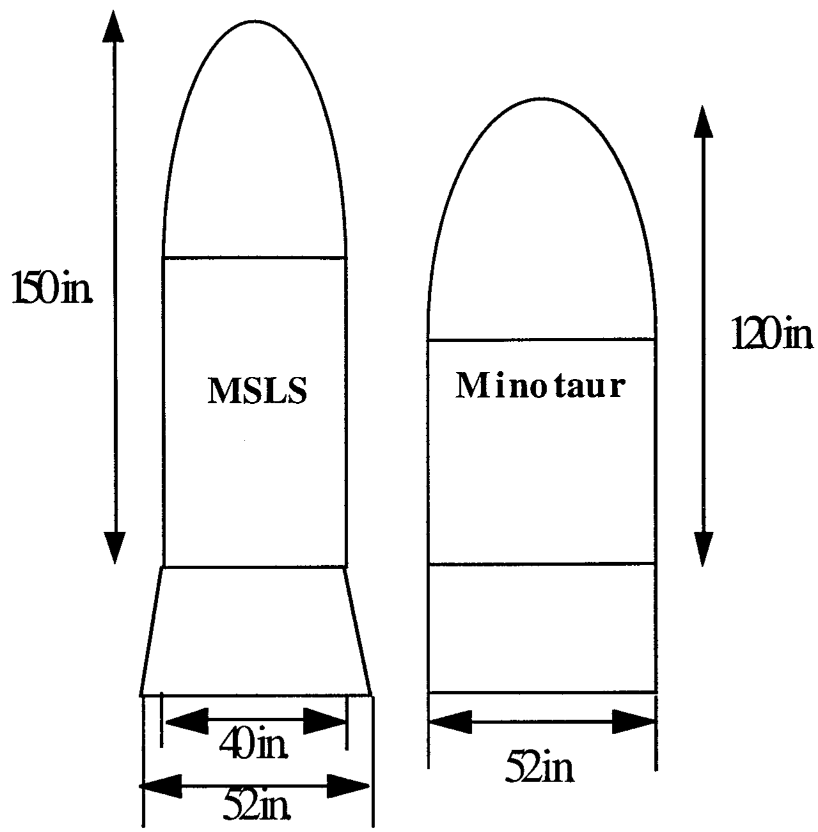


Figure 4.14 MSLS & MDI shroud dimensions

As shown in Figure 4.14, the MDI shroud diameter is 52 in. There are two reasons for using a 52 in. diameter shroud. First, a 52 in. shroud allows the shroud and wafer sections to remain flush with the MMIII third stage, which eliminates the need for a connecting skirt that would increase the drag on the missile. Also, a 52 in. shroud allows MDI to attract a larger portion of the satellite market as shown in the satellite survey (most of the satellites surveyed have diameters between 40 in. and 52 in). Although the MDI shroud has more surface area, the MSLS and MDI shrouds have a similar amount of drag due to the fact that the MSLS shroud has a connecting skirt [6].

Another goal of the MDI shroud design is to have a weight comparable to the MSLS configuration. The MSLS shroud weighs 240 lbm. and uses a hinge-and-thrusters separation mechanism. In an attempt to reduce the weight of the shroud, MDI discovered a lightweight thrusting-joint separation system developed by TRACOR Aerospace. Through data provided by TRACOR, MDI determined that a shroud with MSLS dimensions that uses a thrusting-joint would weight 215 lbm. Therefore, MDI has decided to use the thrusting-joint separation system in order to save weight.

The final goal of the shroud design is to have a more efficient use of volume than the MSLS shroud. The MSLS is 150 in. high with a usable inside-volume height of 111 in. However, since the satellite survey implies that none of the potential satellites is higher than 76 in, then the MSLS shroud would contain up to 35 in. of unused volume for each launch [7]. The MDI shroud design eliminates some of the unused volume inside the shroud. The MDI shroud height is chosen from a plot of total shroud weight against shroud diameter for varying heights, which is shown in Figure 4.15. A shroud weight of 215 lbm. and a diameter of 52 in. would give a shroud height of 120 in., which would have a usable volume height of 90 in. A 90 in. usable volume height would still house all of our potential satellites. Thus, in addition to minimizing unused volume, the MDI shroud would weigh less and can house more potential satellites than the MSLS configuration.

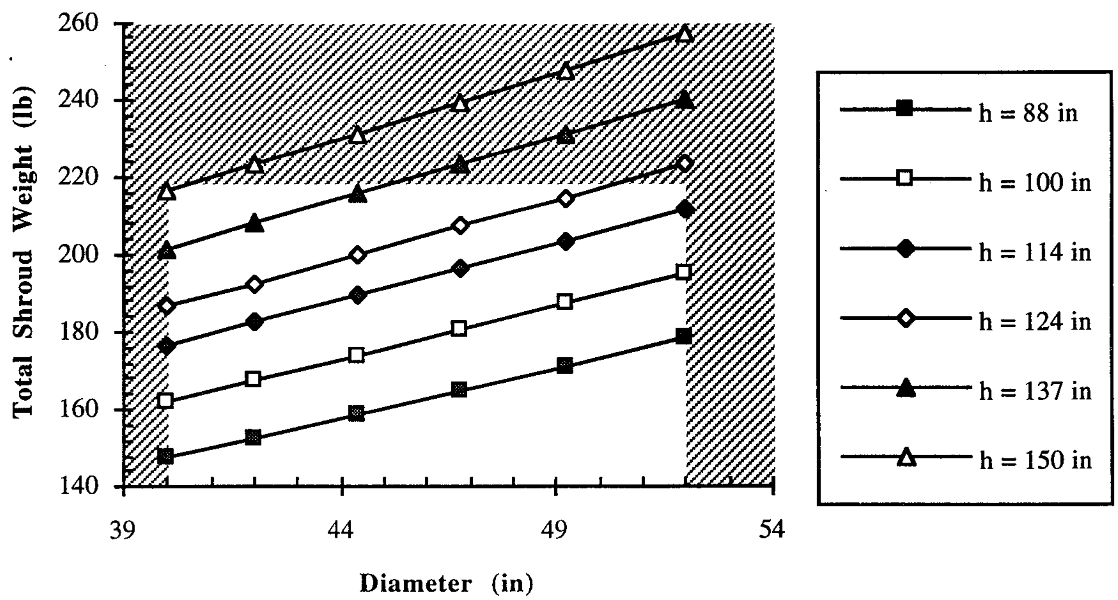


Figure 4.15 Shroud Design Space Chart

4.7.3 Separation Mechanisms

Two thrusting-joint separation mechanisms are employed to jettison the shroud; a Vertical Separation System (VSS) and a Horizontal Separation System (HSS). The separation systems use linear charges that are fired by electrical explosive devices (EED's). Two EED's exist at either end of the linear charge. The EED's are fired simultaneously on each charge. The HSS is fired 20 milliseconds before the VSS. During jettison, the thrusting-joint separation mechanism imparts a lower shock to the payload than a hinge-and-thrusters separation system. Also, the thrusting-joint system is a simple, one-step separation system that is more reliable and has fewer components than a hinge-and-thrusters separation system [7].

The HSS employs a heavy-rail thrusting-joint and a linear charge with dual EED's as shown in Figure 4.16 The VSS uses a light-rail thrusting joint and a linear charge with dual EED's as shown in Figure 4.17 [8]. For both thrusting joints, the explosive chord inflates the bellows, forcing the ends of the joint apart. The separation force is sufficient to shear the rivets holding the joint together. The force can impart a separation velocity of 600 mph to the shroud [6].

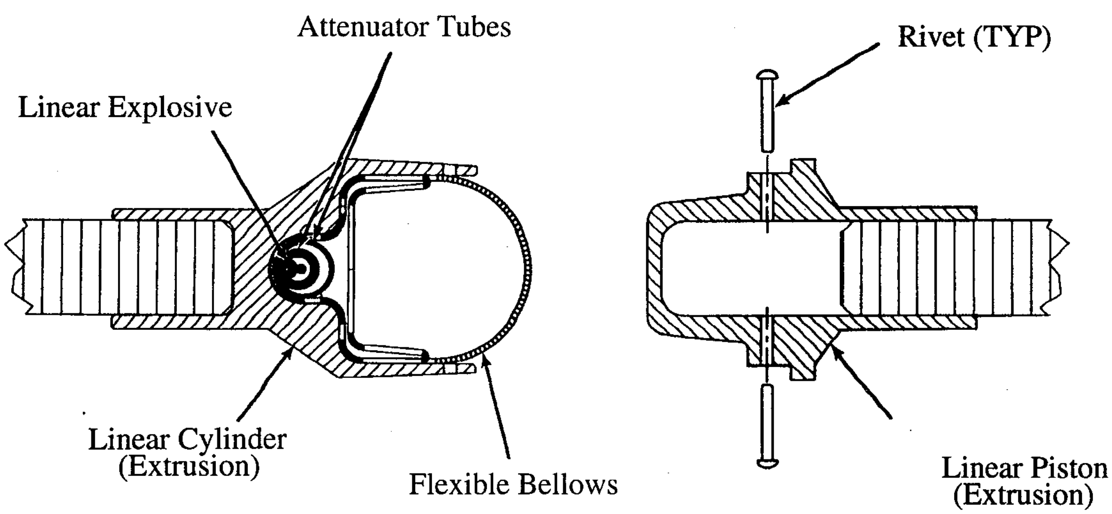
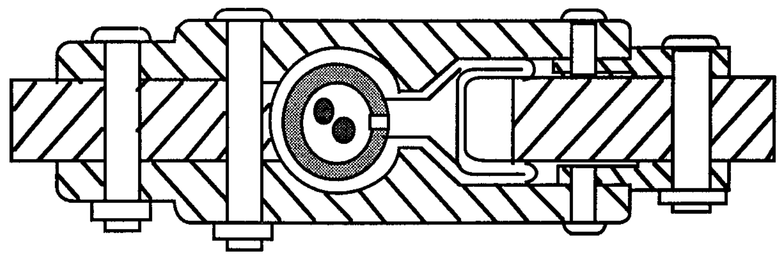
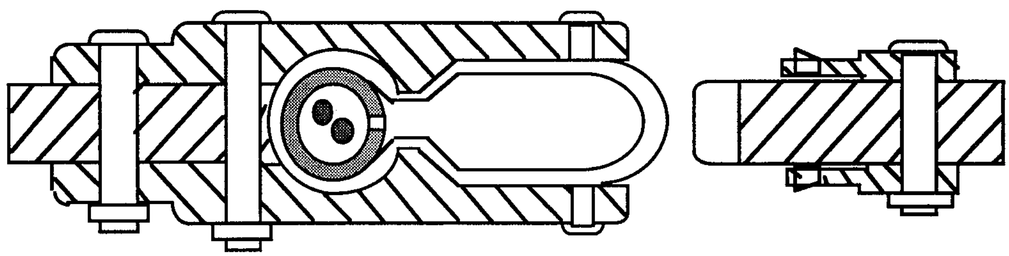


Figure 4.16 Horizontal Separation System (HSS) [8]

Courtesy of TRACOR Aerospace



Installed



Separated

Figure 4.17 Vertical Separation System [8]

Courtesy of TRACOR Aerospace

5.0 LAUNCH VEHICLE PERFORMANCE

5.1 Attainable Orbits

A satellite vendor's primary concern for any launch vehicle is what payload masses can be taken to the desired orbits. The MDI team has performed a preliminary analysis of the MDI launch vehicle to determine the attainable orbits as a function of payload mass. This final orbit altitude is a function of launch location, rocket configuration, payload mass and orbit inclination.

In order to satisfy the wide range of the satellite market, MDI has performed this orbit analysis for VAFB and Cape Canaveral launch sites [9]. In addition, the analysis has been extended to cover payloads with no injection stage, STAR 17A injection stage and STAR 27 injection stage. Details of MDI's analysis are shown in Appendix B

Figures 5.1 and 5.2 summarize the results of MDI's analysis. Figures 5.3 and 5.4 show the restricted launch azimuths for VAFB and Cape Canaveral launch sites. These launch azimuths are restricted for range-safety reasons, and these restrictions bound the range of attainable orbit inclinations.

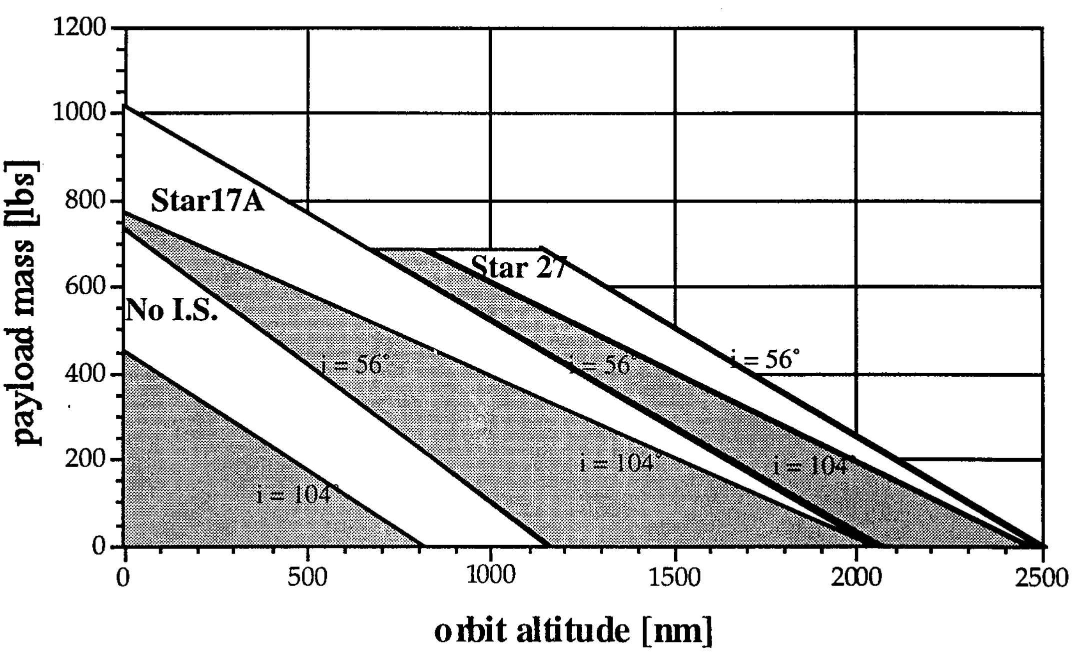


Figure 5.1 Launch vehicle performance at VAFB.

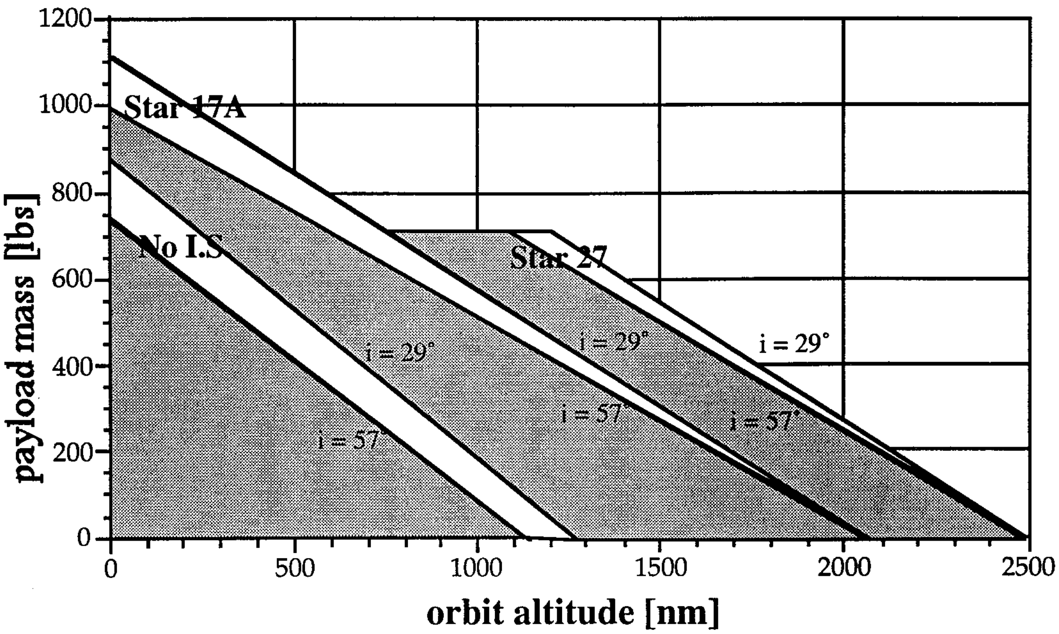


Figure 5.2 Launch vehicle performance at Cape Canaveral.

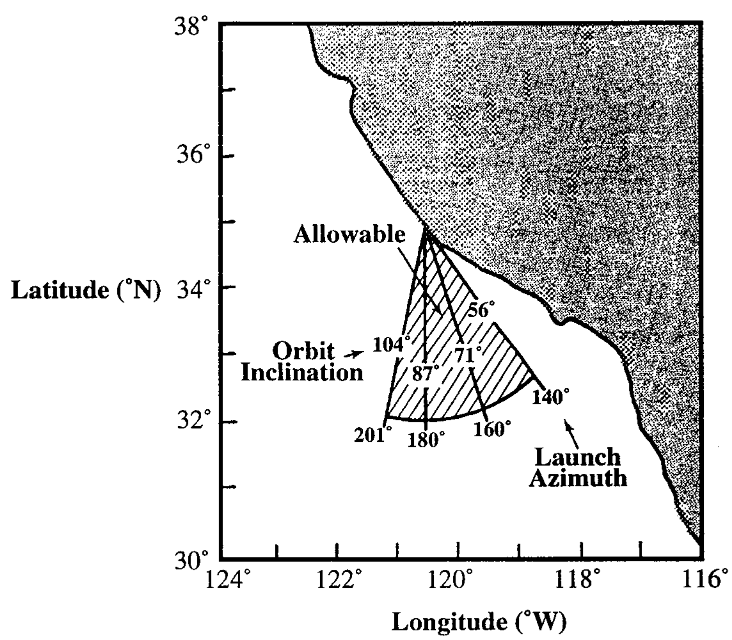


Figure 5.3 VAFB restricted launch angles [8]

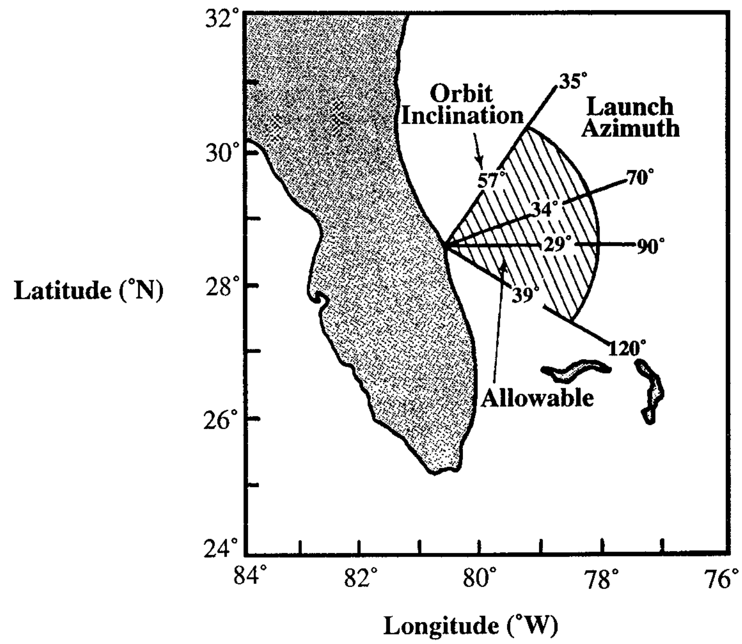


Figure 5.4 Cape Canaveral restricted launch angles [8]

Figures 5.1 and 5.2 show the attainable orbits of MDI's rocket system launched from VAFB and Cape Canaveral, respectively. The figures depict payload mass as a function of circular orbit altitude for 3 rocket configurations:

- No injection stage
- STAR 17A injection stage
- STAR 27 injection stage

As expected, the attainable orbit altitude decreases as the payload mass increases. Also, high orbit inclinations result in lower performance (a decrease in orbit altitude). As the orbit inclination decreases, the launch system's performance increases because of the additional boost from the Earth's angular velocity.

As shown in Figures 5.1 and 5.2, a STAR 17A increases the performance by carrying heavier payloads to higher altitudes. The STAR 27 increases attainable orbit altitudes even more, but the upper mass is bounded at 700 lbm. The payload mass is limited for the STAR 27 because the mass of this injection stage, combined with the mass of the satellite, exceeds the permissible structural support criteria of 1500 lbm.

In addition to attainable orbits, the satellite customer must know the expected orbit-insertion errors for MDI's launch vehicle. Based on previous MSLS analysis of the "B" configuration missile, MDI made the following insertion-error estimates for the final orbit:

- ± 15 NM in orbit altitude
- ± 0.5 ° in orbit inclination. [10]

If the desired orbit is within MDI's launch-system capability, the excess performance must be consumed during launch. The Minuteman stages are equipped with solid rocket propellants and must burn all propellants within the stages. Conventional methods of dissipating this excess performance are ballasting the launch vehicle with more mass or burning out of plane and back [3]. MDI will opt for the latter.

Only circular orbits were considered for this preliminary analysis, since the satellite market survey indicates most satellites require near-circular orbits. Nevertheless, MDI will be able to service satellites with higher eccentricity orbits.

5.2 Launch Profile

MDI's launch system will follow the pre-launch sequence of events shown in Table 5.1. This sequence of events is controlled by the avionics wafer. If an injection stage is included, the onboard logic will be responsible for the final events.

Table 5.1 Pre-launch sequence

Time (min.)	Event
-10	Power decoupled - Batteries (140W max, 5A, 28 ± 2V DC)
-9	Communication check
-7	System verification
-2	Nitrogen gas purge of encapsulated payload
-1	Arm all destructs/disconnects
-0.5	Continuity confirmation

Figure 5.5 and Table 5.2 depict the sequence of events after launch. As seen from the Figure and Table, stages I and II both have a one-minute burn time. Near stage II burnout, the launch vehicle will have reached an altitude of approximately 300,000 ft where the atmospheric forces are negligible [4]. At this altitude, the shroud will be ejected to reduce propelled mass. During shroud separation stage II will continue burning to accelerate the payload past the ejected shroud halves.

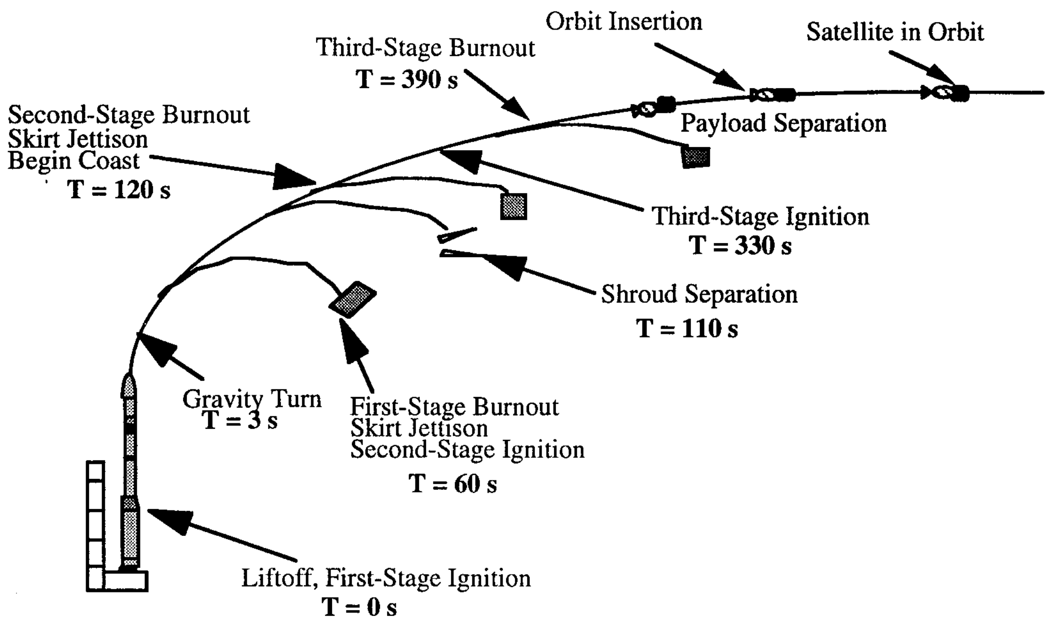


Figure 5.5 Launch sequence of events.

Table 5.2 Launch sequence of events.

Time (sec)	Event
0	1st stage ignition
3	pitch over (gravity turn)
60	1st stage shutdown, 2nd stage ignition
110	shroud separation
120	2nd stage shutdown, begin coast
330	end coast, 3rd stage ignition
390	3rd stage shutdown, orbit insertion, payload separation

A coasting phase will occur after stage II burnout. For the case of no additional injection stage, the coast phase will end when the launch vehicle reaches the targeted orbit altitude. At this altitude, stage III will ignite and burn out in one minute to insert the payload into orbit. For the case of an additional STAR engine, stage III will burn and insert the payload on a transfer trajectory to the desired orbit where orbit insertion will occur with the injection stage burn.

5.3 Expected Launch Environment

MDI has studied the expected launch environments of the launch system by analyzing the MSLS and TRACOR Aerospace data [11]. The main environmental factors of concern are the thermal, pressure, and loading environments.

5.3.1 Thermal Environment

The thermal environment is controlled by an ablative coating of the shroud. The ablative coating heats up and is shed, taking energy with it [8]. This shedding of energy keeps the internal shroud temperature low. The expected temperature profile of MDI's launch vehicle is shown in Figure 5.6. This profile shows internal shroud temperature as a function of time [12].

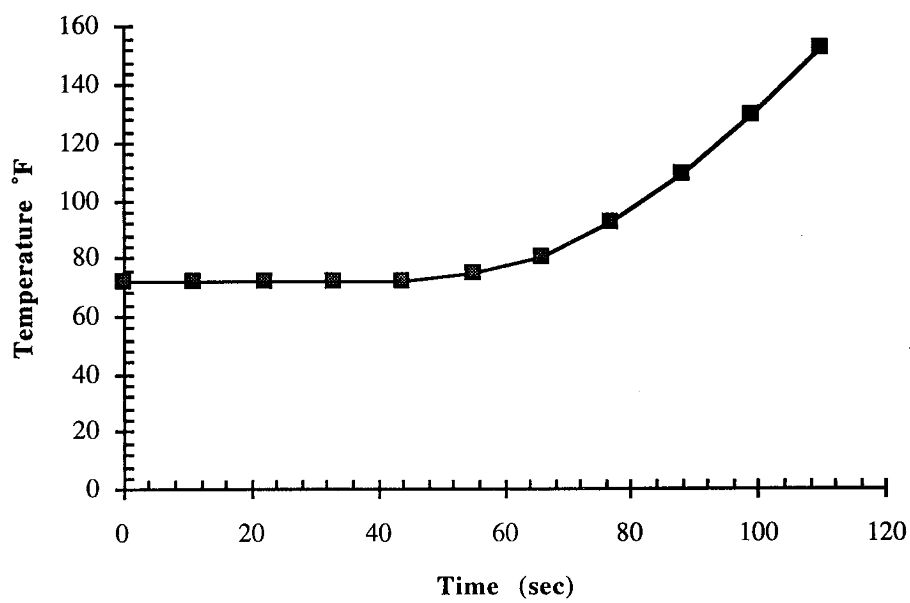


Figure 5.6 Thermal environment

5.3.2 Pressure Environment

The pressure environment is controlled by venting of the payload through the avionics wafer to the outside atmosphere. The internal pressure profile is provided for satellite vendors and they are advised to provide venting in accordance with the pressure profile in Figure 5.7 to avoid structural collapse of their payload [5-6,8].

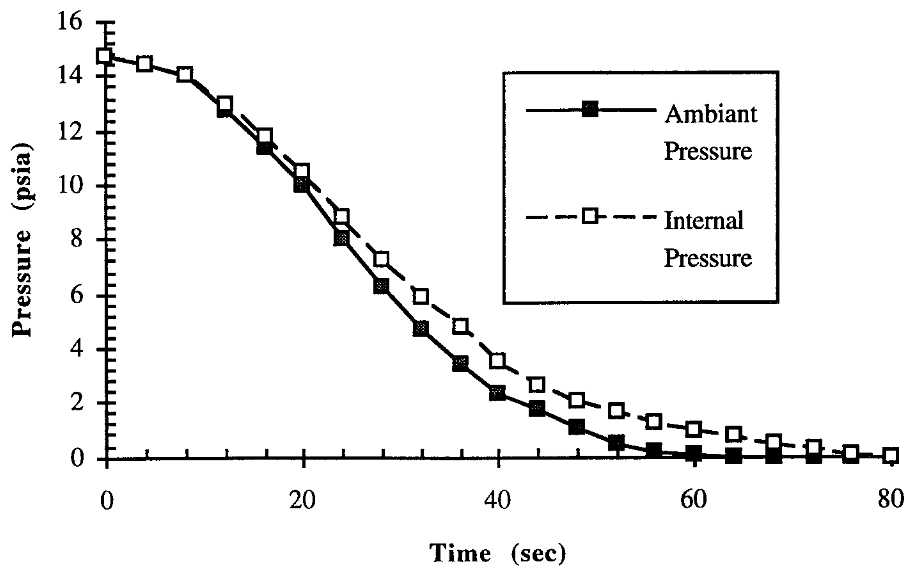


Figure 5.7 Pressure environment

5.3.3 Dynamic Environment

The Minuteman is a solid-fuel rocket and has high-frequency vibration typical of solid-fuel rockets. The vibrational environments are detailed in the Figures 5.8 and 5.9. The power spectral density (PSD) is shown in Figure 5.8 [12:LVS:4.1.5]. This PSD profile is typical of solid-fuel rockets. The maximum accelerations are shown in Figure 5.9. This acceleration profile is extracted from the MSLS project, and similar accelerations are expected on MDI's launch vehicle [11].

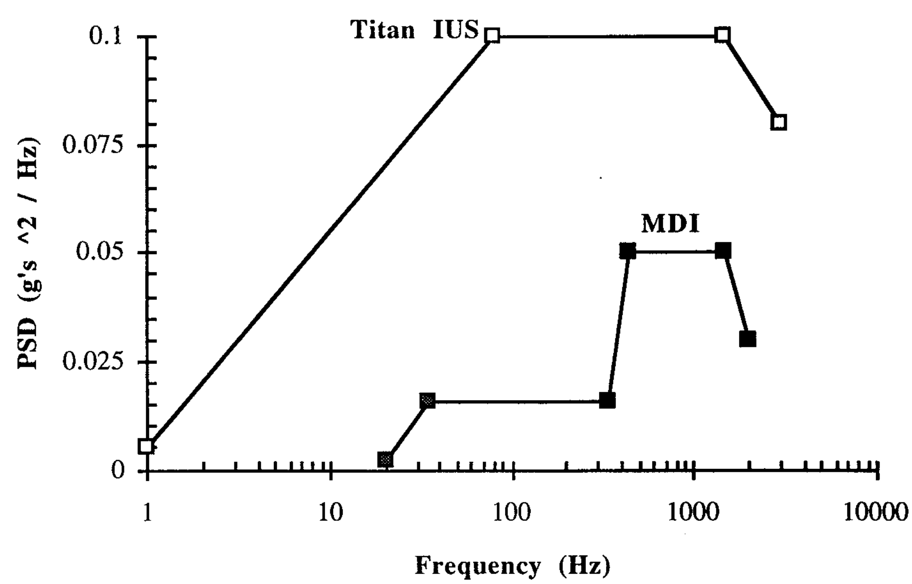


Figure 5.8 Random vibration environment

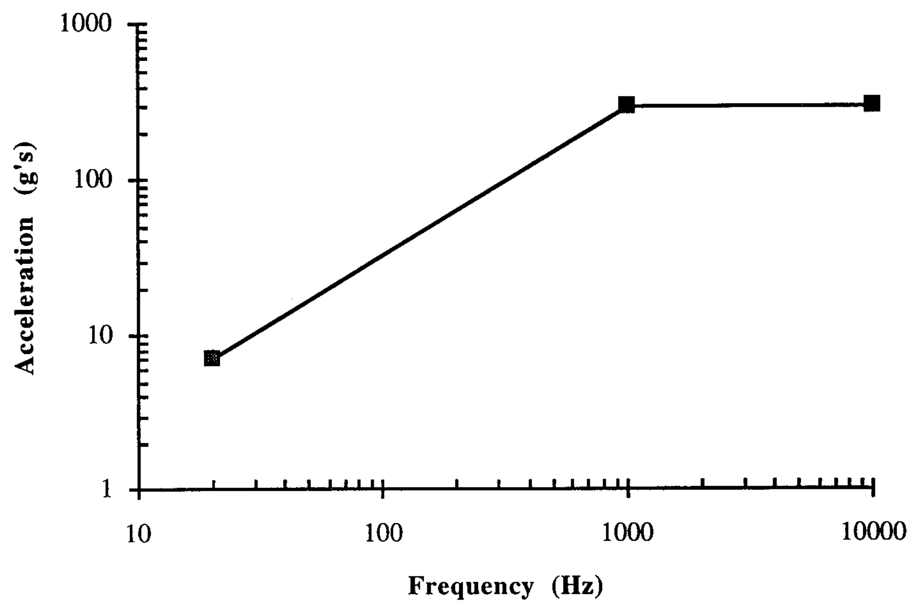


Figure 5.9 Shock environment

6.0 MISSION DESIGNS

MDI has designed two different satellite missions to show proof of concept. These two satellite missions will launch the BATSAT and AIM satellites (see Appendix A for satellite details).

6.1 BATSAT

Figure 6.1 shows a scaled representation of the mechanical interfaces to BATSAT. For mounting purposes, BATSAT requires a lip that fits on one of the structural latitudinal rings. BATSAT cannot mount antennas on this ring. Figure 6.2 shows BATSAT's orbit altitude and payload mass along with the attainable orbits of MDI's launch vehicle. As Figure 6.2 indicates, MDI's vehicle is capable of inserting BATSAT into its desired orbit by using only three stages.

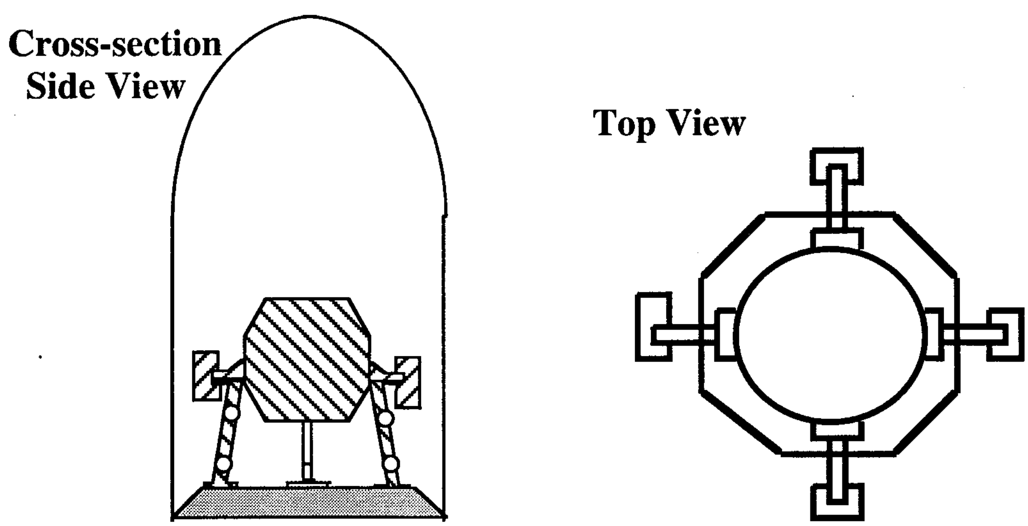


Figure 6.1 BATSAT Mounted in MDI Launch Vehicle.

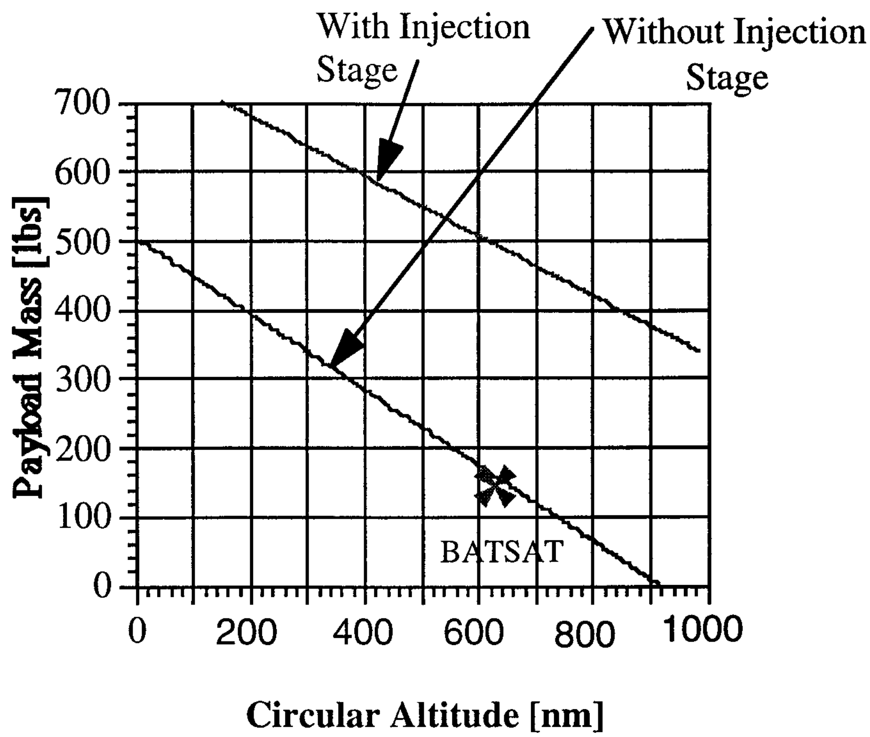


Figure 6.2 BATSAT Orbit Requirement and MDI Vehicle Capabilities.

6.2 AIM

Figure 6.3 shows a scaled representation of the mechanical interfaces to the AIM satellite. The AIM payload includes the STAR 17A injection stage, and MDI does not need to provide the interface between the AIM and STAR 17A. The clamp fits on one of the injection stage's lips since this clamp is designed to interface to both a STAR 17A and STAR 27. The AIM payload will orbit in a circular, 200 nm orbit. Figure 6.4 shows that MDI's launch vehicle can insert AIM into this orbit with the use of three stages. Once in this parking orbit, AIM will use its own injection stage escape velocity to send it to intercept near-earth asteroids.

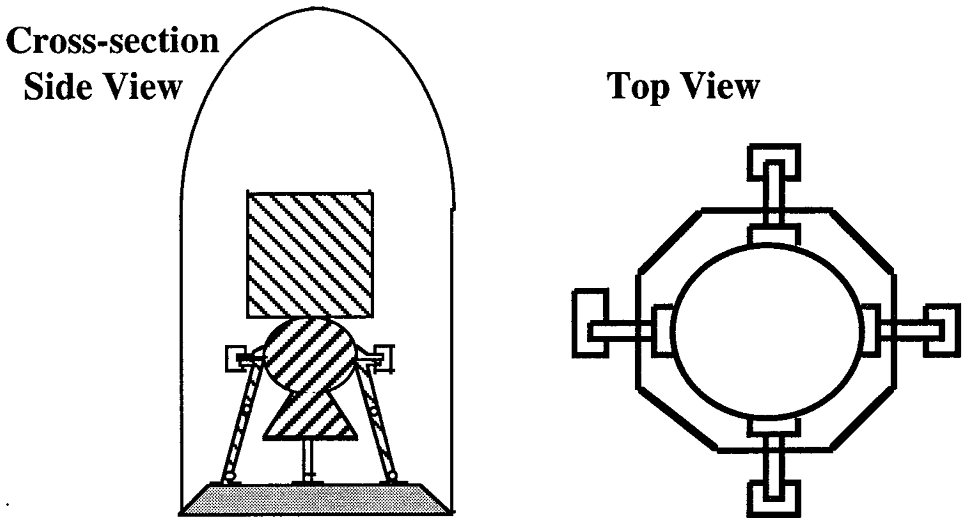


Figure 6.3 AIM mounted in MDI missile

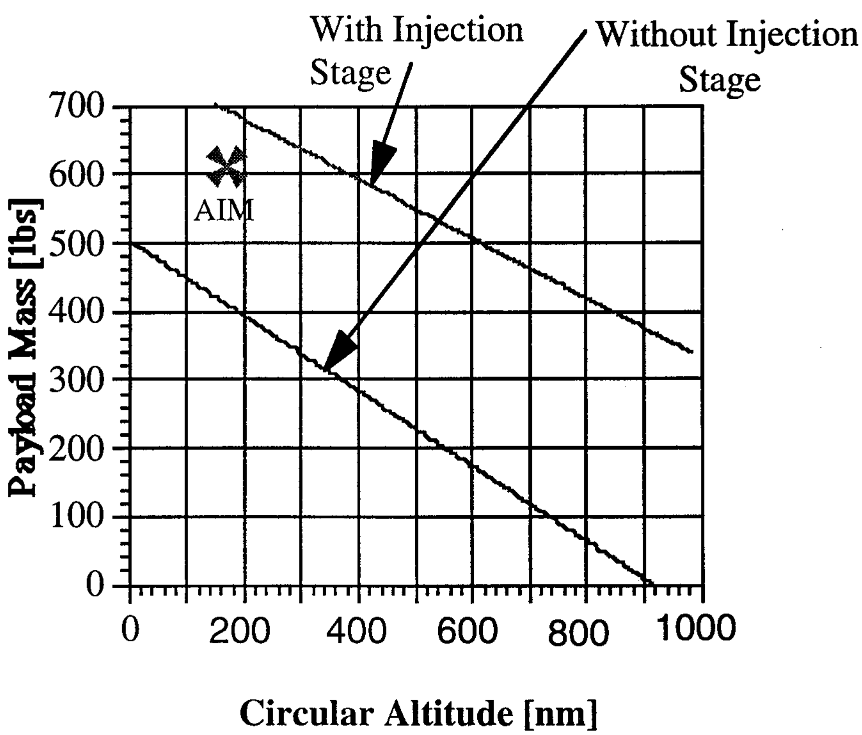


Figure 6.4 AIM orbit requirement and MDI missile capabilities

7.0 TIME LINE AND COST ANALYSIS

The Department of Defense (DoD) issued a memorandum in February 1993 that allows civilian contractors to acquire excess strategic missile assets such as the MMII rocket system cost free upon request for use for orbital satellite-insertion launches. Consequently, MDI plans to take advantage of this DoD policy in order to acquire the MMII rocket stages free of cost.

MDI will purchase the MSLS "D" configuration wafers from Martin Marietta for a unit cost of 2.2 million dollars. A non-recurring cost of 7 million dollars to complete a ground-test program will increase the overall cost of those wafers. MDI will also purchase a complete shroud set-up with a thrusting-joint separation system from TRACOR Aerospace for a unit cost of 0.5 million dollars. The electrical structure and the payload support structure will be built by a subcontractor that will have to meet MDI's quality and cost requirements.

7.1 Research and Development

The research and development (R&D) costs are summarized in Table 7.1. As the table shows, the MDI project will need an initial investment of 11 million dollars for R&D purposes. After all the initial contracts with the satellite customer are finalized, MDI expects to offset the investment cost with the profits achieved. The timeline of the R&D segment of the project is shown in Figure 7.1.

Table 7.1 Research and Development Costs

R&D Area	Cost (USD)
Project Financing	$11,000,000
Shroud R&D	$500,000
Truss R&D	$100,000
PSB and Wafer R&D	$7,000,000
Full Scale Testing	$1,500,000
Finite Element Modeling	$250,000
Flight Data Reduction	$50,000 / Flight

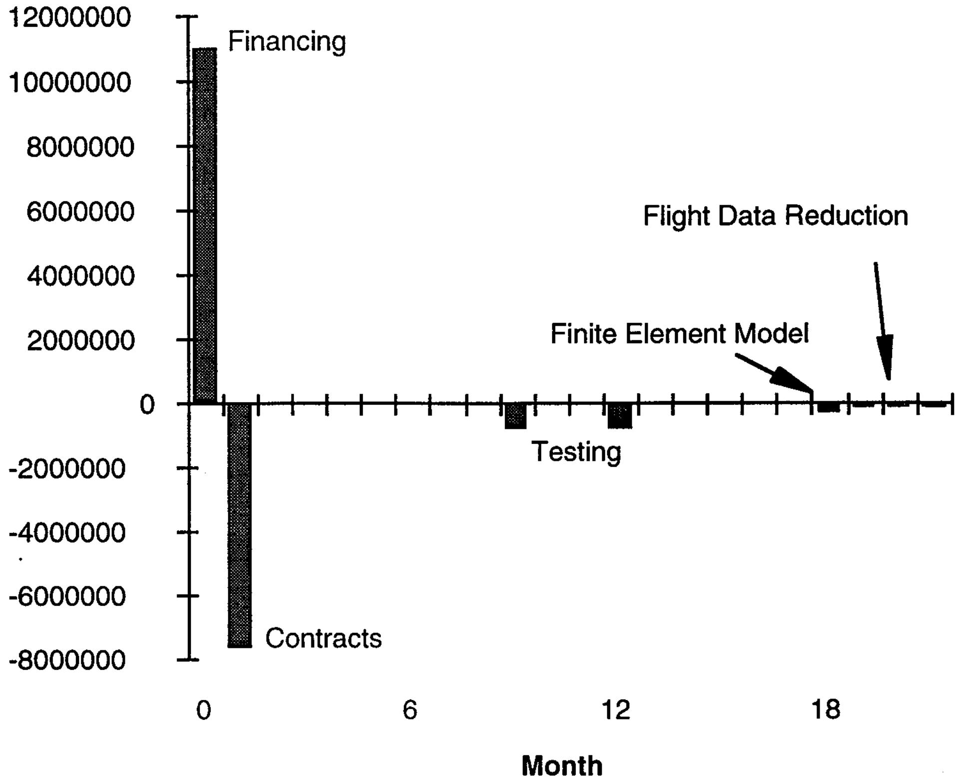


Figure 7.1 Research and development timeline

7.2 Launch Costs

The total launch cost for the MDI rocket will be 7.5 million dollars. The satellite customer will prepay MDI for 2/3 of the launch cost, and the rest of the payment will be made after successful the launch of the rocket. Launch costs are broken down in Table 7.2. A six-month period will be the standard customer waiting time between the contract award and the rocket launch (see figure 7.2).

Item	Cost (USD)
Vendor Prepay	-$5,000,000.
Wafers	$2,400,000
Shroud	$300,000
Trusses	$15,000
Electrical	$65,000
Insurance	$1,000,000
Ground Support	$2,200,000
Vendor Final Pay	-$2,500,000

Table 7.2 Launch costs

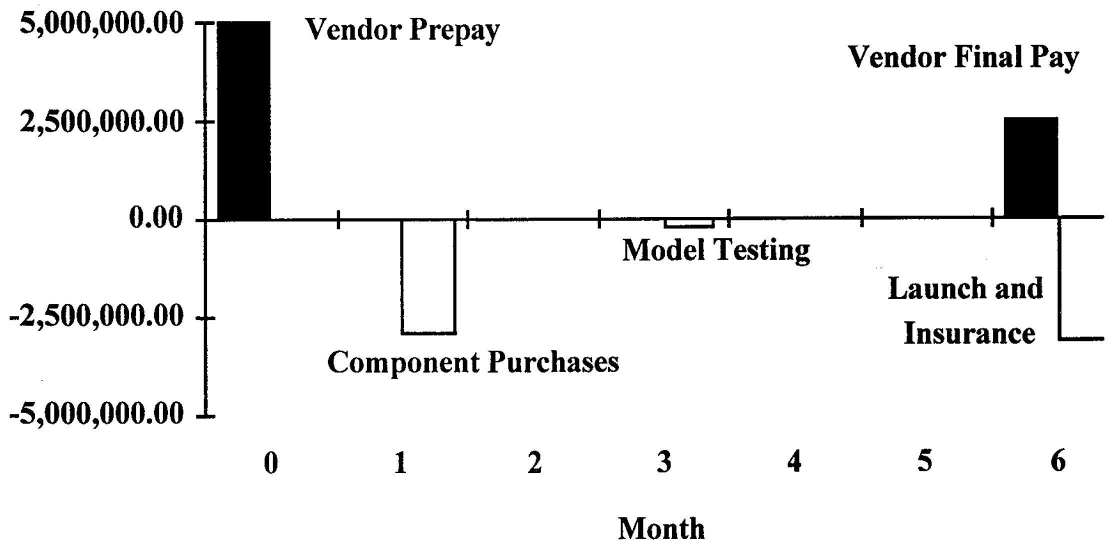


Figure 7.2 Launch timeline

7.3 Long Term Profitability

R&D and launch timelines have been combined to determine the overall net worth of MDI once the project begins. The launch timelines assume one launch per month starting in the 18th month. The net-worth calculation results are plotted in Figure 7.3.

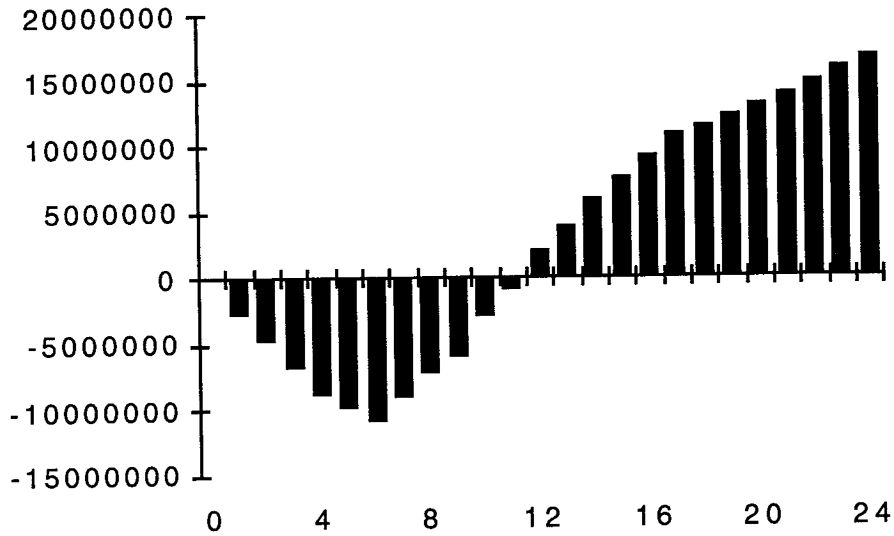


Figure 7.3 MDI projected net worth

8.0 LAUNCH SYSTEM COMPETITIVENESS

In order to show that MDI's launch system is feasible and competitive, a comparison between the MDI launch vehicle and two other available small satellite launch vehicles in the market is performed. The two other launch systems are the Scout and Pegasus. Comparisons of attainable orbits, insertion errors, shroud volumes, and customer launch costs are made, and the results show that MDI's conversion program will yield a profitable, competitive satellite launch system.

8.1 Attainable Orbits Comparison

Figures 8.1 shows the altitude of attainable circular orbits versus payload mass for the MDI missile, Scout, and Pegasus. The figure assumes a VAFB launch at an orbit inclination of 90°. As the figure shows, the MDI rocket performs as well as the other two launch vehicles and can be considered as a competitive system [14].

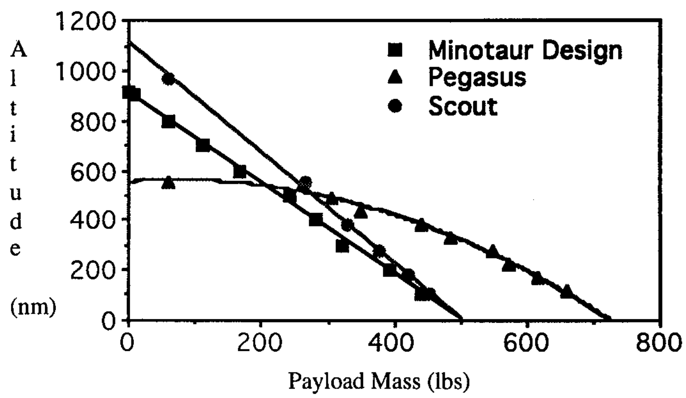


Figure 8.1 MDI missile, Scout, Pegasus attainable orbits.

8.2 Insertion Errors

Table 8.1 shows the insertion errors expected for the MDI missile along with those of the Scout and Pegasus. As the table indicates, the MDI missile is on the same order of error as the other launch systems [14].

Table 8.1 MDI missile, Scout, Pegasus injection errors

	Pegasus 400 nm. circular polar orbit	Scout 300 nm. circular orbit	MDI Project 100 nm. circular orbit
Altitude Δh	± 25 nm.	± 59 nm.	±15 nm.
Inclination Δi	± .03 deg.	± .9 deg.	± .5 deg.
Attitude angles	± 2 deg.	± 1.23 deg.	± 1 deg.
GPS would enhance position accuracy to within an estimated ± 51 cm.			

8.3 Shroud Size Comparison

Figure 8.2 shows the relative size of a variety of launch vehicles, and a comparison of the usable shroud volume for these launch vehicles is shown in Table 8.2 As the table indicates, the usable volume for the MDI missile is greater than that of the other two [2, 14].

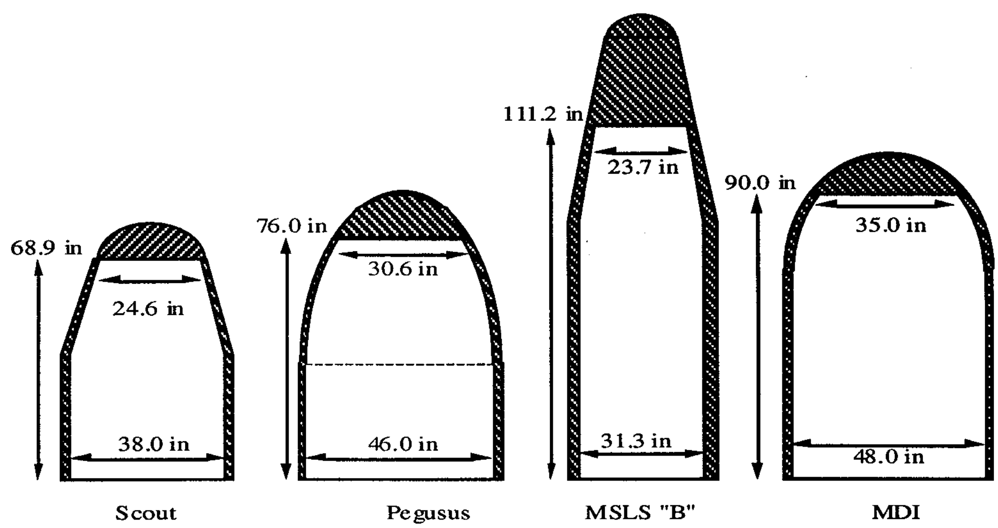


Figure 8.2 Relative shroud volumes

Table 8.2 Usable shroud volumes

Launch Vehicle	Usable Volume
MDI	85 cubic feet
Pegasus	68 cubic feet
Scout	40 cubic feet

8.4 Cost Comparison

The MDI design is a highly competitive launch system based on costs. MDI proposes to charge 7.5 million dollars per launch. As mentioned before, the vendor will be required to post 2/3 of the costs up front and the final 1/3 upon completion of the launch.

A comparison of the launch costs to the vendor for some comparable launch vehicles is given in Table 8.3, and it clearly shows that the MDI design is the most cost effective launch system [14].

Table 8.3 Launch cost comparison

Launch System	Cost
MDI Minuteman II	\$7.5 million
Pegasus	\$12 million
Scout	\$15 million

9.0 MANAGEMENT STATUS

9.1 Management Goals

The primary management goal at Minotaur Designs Inc. was to efficiently use every engineer's talent as best as possible. To achieve this purpose, The management team implemented a dynamic program that promoted efficiency in the group by evenly distributing the work among team members by individual talent and identifying/preventing potential problems before they occurred. Therefore, by establishing a good communication system within the team, the MDI management did not encounter any major problems and was able to achieve the goals it set at the beginning of the project.

9.2 Program Organization

MDI's management team consisted of Rod McHaty as the team leader, Rodolfo Gonzalez as the chief engineer/GNC subsystem manager, and Gordon MacKay as the Structures subsystem manager. The team leader was MDI's primary point of contact, and he kept the contract monitor informed of the team's work progress on a regular basis, in addition to any organizational changes or other potential problems that may have occurred. The team leader set the project goals and milestones, in addition to adjusting manpower within the team in order to conform to any major changes that were encountered over the course of the project.

The chief engineer at the beginning of the project was Vu Pham. However, due to other school obligations, Mr. Pham resigned his position halfway through the course of the project and designated Rodolfo Gonzalez as his successor. Nevertheless, Mr. Pham stayed involved in the management process as the project's chief administrative officer.

Overall, the chief engineer managed the technical aspects of the project and coordinated the efforts of the team members in order to achieve the project goals. Consequently, the chief engineer was the main point of contact for the team members, and all technical developments within the project had to get his approval. Both the chief engineer and the team leader served as assistant administrative officers by organizing the project records, computing program costs, and maintaining the project's notebook.

The supporting organizational structure branched into two separate subsystem groups: Structures and Guidance, Navigation and Control (GNC). The primary subsystem for the project was the Structures subsystem and was managed by Gordon MacKay. The GNC subsystem was managed by Rodolfo Gonzalez who later on became the chief engineer as previously mentioned. The subsystem managers were responsible for dividing/distributing the workload among the respective team members. Figure 9.1 shows the team's organizational chart at the end of the project and the subsystem assignments. The organization was flexible and was changed several times in order to meet new requirements through out the course of the project.

9.3 Program Control and Schedule

Much of the work and designs in the two subsystems were interdependable. Therefore certain subsystem work had to be completed before other work could begin. These constraints made a PERT chart useful in the effective management of this project. The PERT chart (see Figure 9.2) graphically organized the project's major milestones and showed the tasks priority. Another important tool for management control was the GANT chart (see Figure 9.3). These two tools provided a basis for a tight work schedule and helped management foresee problems that would have occurred if a specific task was delayed.

An agenda was prepared by the team leader or the chief engineer for every class meeting. These agendas helped define the work that needed to be achieved and the deadlines that needed to be met. Furthermore, the subsystem managers were required to turn in a progress report memo to the team leader every week. These memos helped the team leader track the progress in each of the two subsystems and also helped the rest of the team members understand the progress made by their colleagues.

9.4 Workload Distribution

The management's primary concern at the beginning of this project was to insure an even distribution of work among the team members. The subsystem managers were responsible for distributing work assignments among the engineers. Workload and costs were monitored through a Progress Planner document that needed to be turned in by all engineers every Monday. Engineers documented their work load on this Planner for specific action items, and estimated other costs such as printing, copying, computer time, and long distance phone calls.

9.5 Project Level of Effort

Figure 9.0 summarizes the cumulative project effort through the end of the project. The chart shows the actual team effort level compared to the effort predicted earlier in the proposal. Due to increased man-hours in the management and engineering sections, MDI's actual level of team effort surpassed the predicted levels. However, the increase in man-hours led to better project results that made the MDI launch system more competitive than the current available small satellite launch systems in the market.

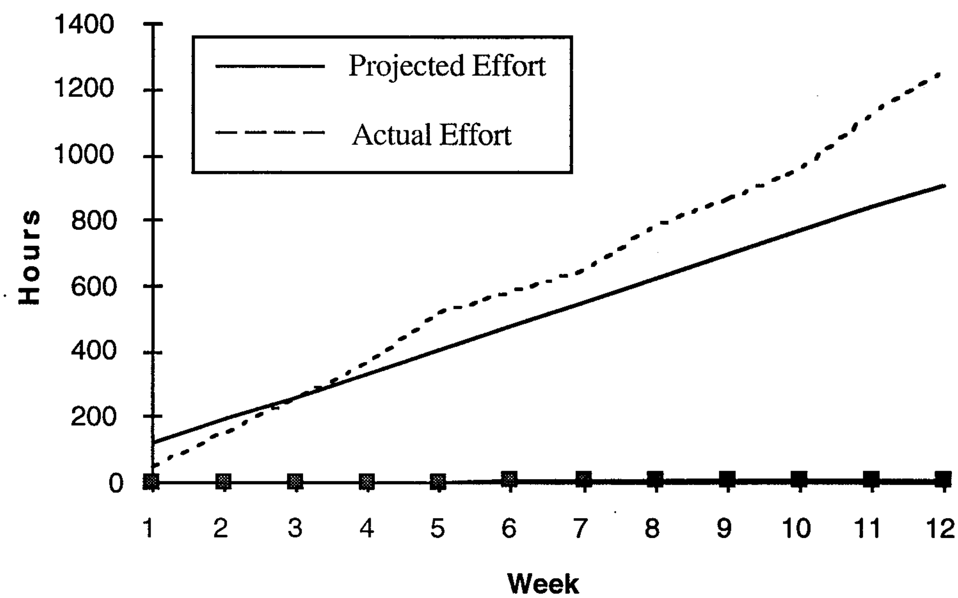
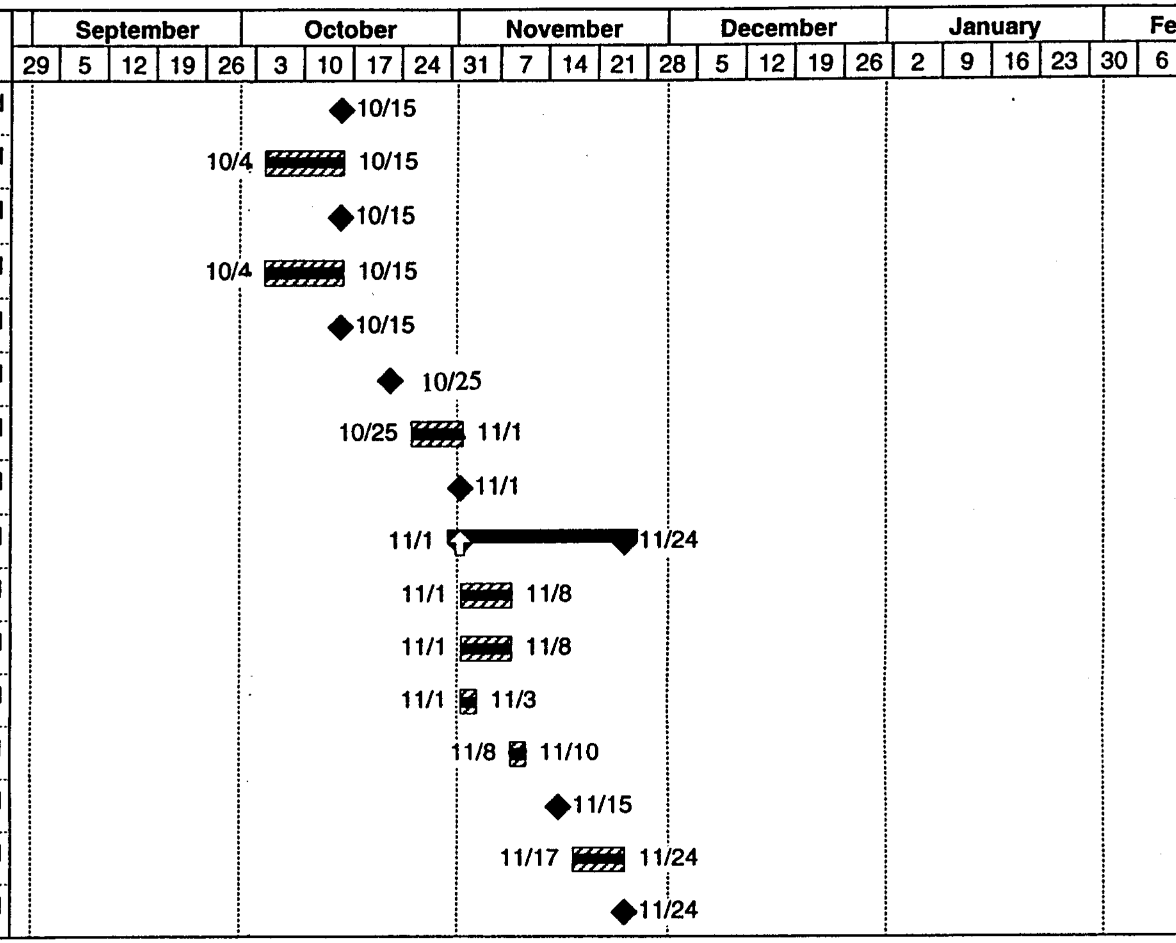


Figure 9.0 Group level of effort

Minotaur Designs Inc.

ID	Name	Duration
17	Electrical Interf. Design	0d
18	Mechanical Interfaces Stud.	10d
19	Mechanical Interf. Design	0d
20	Instrumentation Studies	10d
21	Instrumentation Design	0d
22	PDR1 Oral Report	0d
23	PDR1 Documentation	6d
24	PDR1 Written Report	0d
25	Second Preliminary Design	18d
26	Structure Integration	6d
27	GNC Integration	6d
28	Alternative Studies	3d
29	Project Integration	3d
30	PDR2 Oral Report	0d
31	PDR2 Documentation	6d
32	Final Report	0d



Project: Minuteman II
Date: 11/24/93

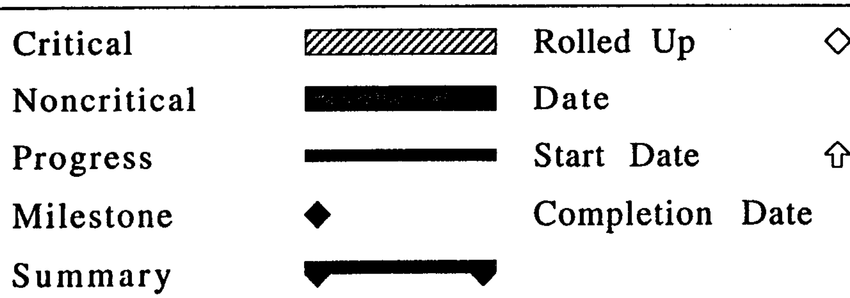


Figure 9.3

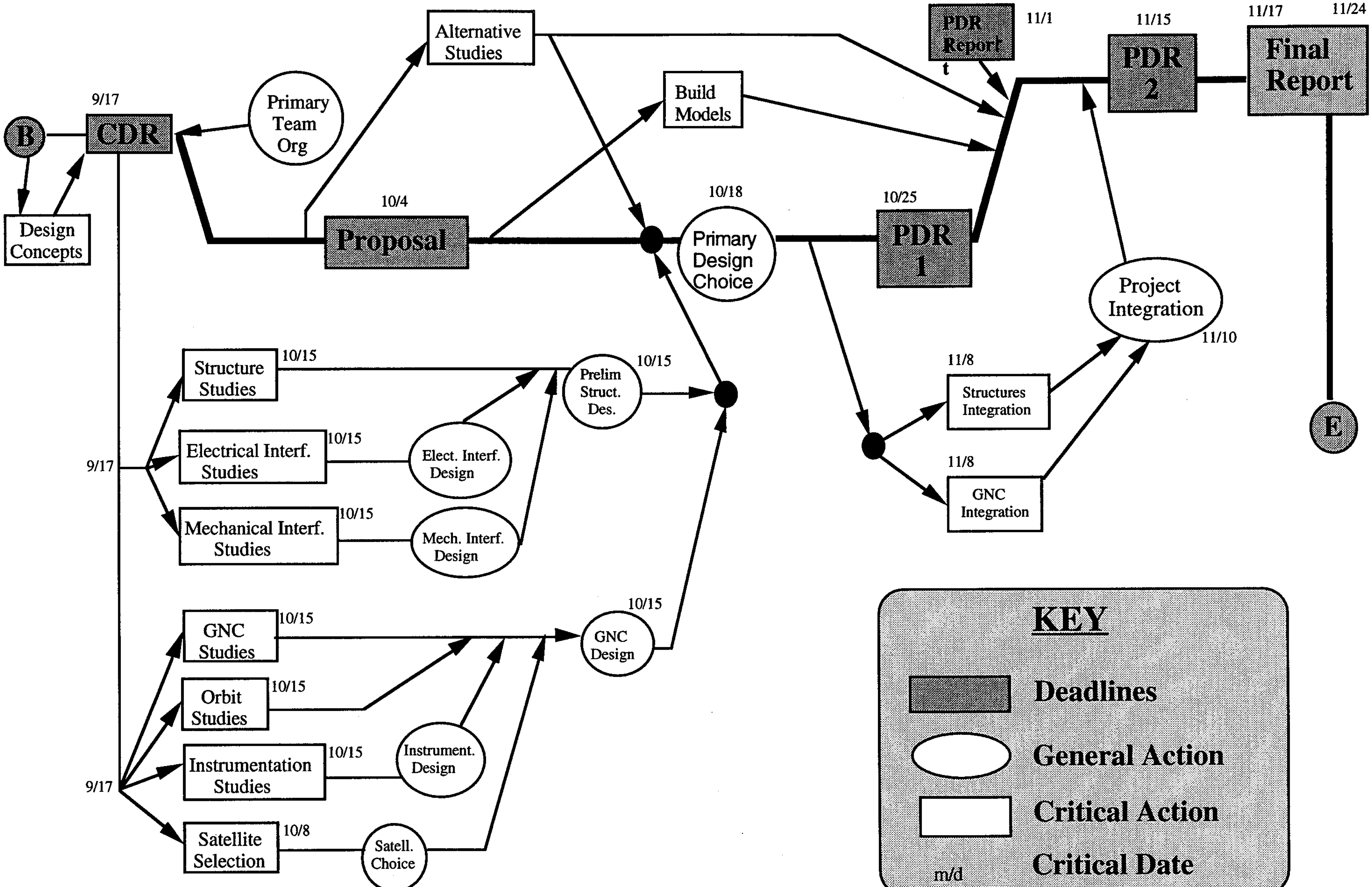


Figure 9.2 Minotaur PERT Chart for the MINUTEMAN II Project

Minotaur Designs Inc.

Minotaur Designs Inc.

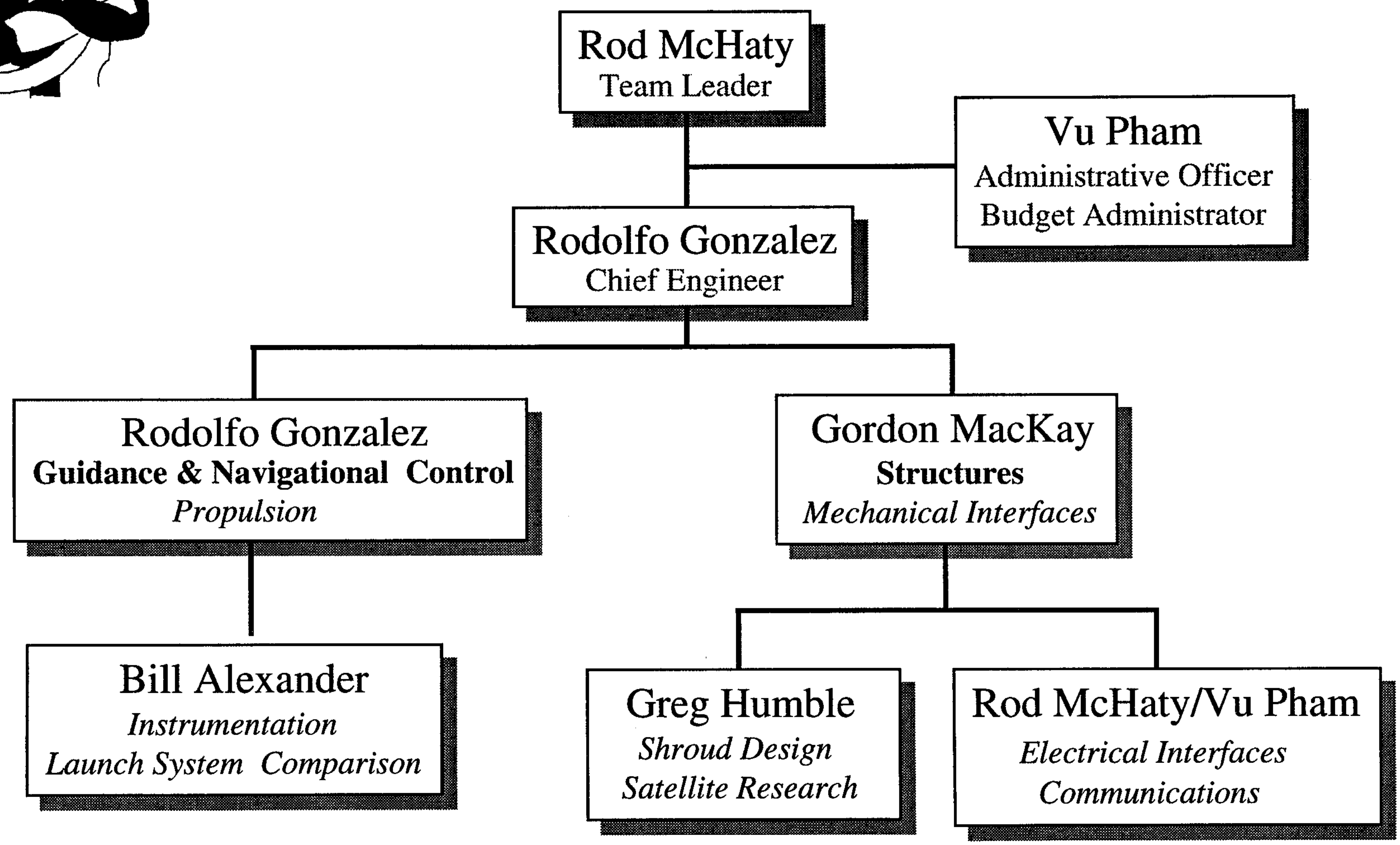
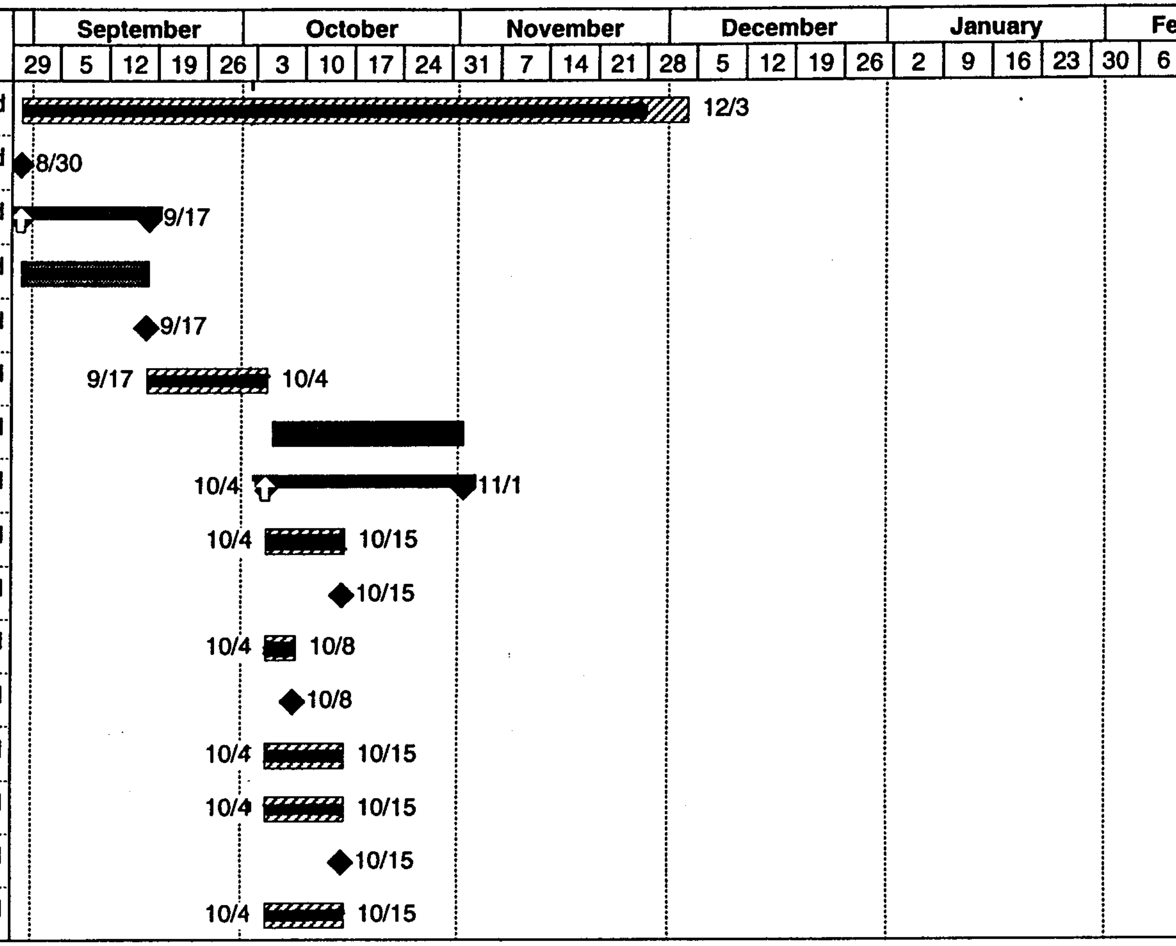


Figure 9.1 Minotaur's Program Organization Structure

Minotaur Designs Inc.

ID	Name	Duration
1	1993 FALL Semester	70d
2	Project Begins	0d
3	Conceptual Design	15d
4	Design Concepts	15d
5	Coneptual Design Review	0d
6	Proposal Development	12d
7	Proposal Delivered	20d
8	First Preliminary Design	21d
9	Structures Studies	10d
10	Prelim. Struct. Design	0d
11	Satellite Selection/Analysis	5d
12	Satellite Choice	0d
13	Orbit Studies	10d
14	GNC Studies	10d
15	GNC Design	0d
16	Electrical Interfaces Stud.	10d



Project: Minuteman II
Date: 11/24/93

Critical		Rolled Up	
Noncritical		Date	
Progress		Start Date	
Milestone		Completion Date	
Summary			

Figure 9.3

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APPENDIX A: PROSPECTIVE SATELLITE DATA

The following sections contain information on candidate satellites that could be flown on a Minuteman II mission. The appendix contains information on missions, dimensions, weight, power, attitude control, and communications for eight satellites. Most of the satellite information in this appendix was provided to Minotaur Designs Inc. by JPL. The following is a Table of Contents for this satellite appendix.

AIM A2
BATSAT A4
ACME 1 .... A5
Eagle class lightsats A6
TOMS-EP A7
HETE A8
ALEXIS A9
Microlab A10

Asteroid Investigation with Microspacecraft (AIM)

The AIM satellite was designed by the Jet Propulsion Lab (JPL) to perform flybys of near earth asteroids for scientific investigation. The spacecraft must be placed in a 200 km parking orbit by the launch vehicle. The spacecraft then uses a STAR 17A injection stage designed by Thiokol to reach the asteroids. Because the AIM satellite itself cannot be easily spin stabilized, the injection stage will have thrust vector control to allow a non-spinning injection. The AIM satellite has a hexagonal shape and is shown in Figure A1.

- Dimensions:
  - AIM - height = 23.6 in, width = 15.7 X 19.7 in.
  - STAR 17A - height = 30.7 in., diameter = 17.3 in.
  - Total height = 54.3 in, Critical width = 19.7 in.
- Weight:
  - S/C wet = 56.6 lbs.
  - SRM dry = 20.5 lbs.
  - Propellant weight = 180.2 lbs.
  - Total satellite weight = 258.4 lbs
- Power:
  - Maximum power generation at 1.2 AU is 55 W.
  - Power generated by an array of multi-bandgap $CuInSe_2$ and GaAs solar cells.
- Attitude Control:
  - 3-axis Attitude Control System with solid state inertial measurement unit, star camera, course sun sensor, and reaction wheel assembly with 3 small reaction wheels.
- Communication:
  - Ka band
  - Includes transponder, command detector unit, telemetry modulation unit, and an antenna.

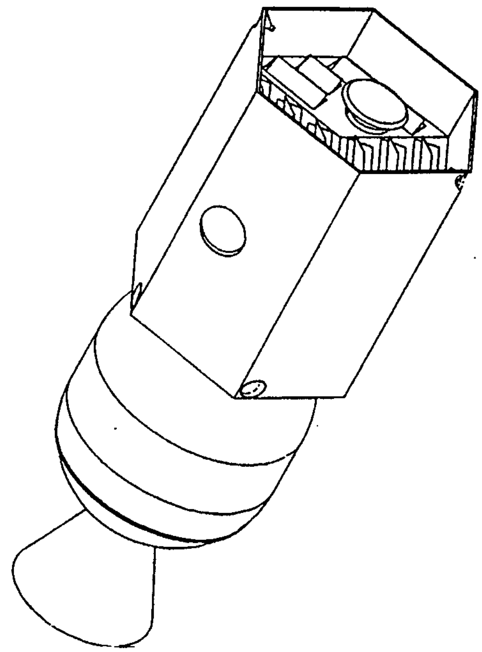

Figure A1. AIM

BATSAT

The BATSAT satellite was designed by Mockingbird Designs at the University of Texas at Austin in the Spring of 1993. The mission of the BATSAT is to study X and Ka band data transmission rates. BATSAT provides simultaneous X and Ka band transmissions to three 34 meter high efficiency Deep Space Network (DSN) receivers. BATSAT will be placed into a orbit of 1163 km and an inclination of 70 degrees. A secondary goal of the satellite is to aid in training DSN personnel. Also, BATSAT has an S-band transmitter which is used to study Doppler shift and blinking techniques. A schematic of BATSAT is shown in figure A2.

• Dimensions:	- height = 24.8 in.
	- diameter = 24.8 in.
• Weight:	- total launch weight = 144.2 lbs.
• Power:	- maximum power consumption = 8.6 W
	- power generated by Silicon solar cells.
• Attitude Control:	- no data at this time.
• Communication:	- Ka, X, or S band.
	- downlink rates: Ka - 31.9 GHz.
	X - 8.45 GHz.
	S - 2.3 GHz.
	- uplink rate = 7.19 GHz.

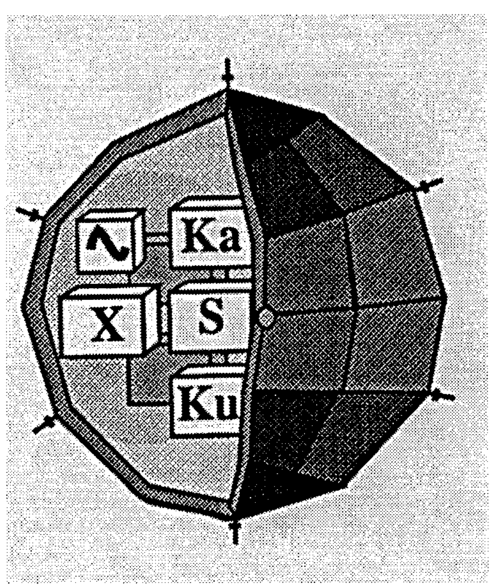

Figure A2. BATSAT

ACME 1

ACME 1 was designed by JPL to use an ultraviolet spectrometer from an elliptical lunar orbit to obtain science data on the Moon's atmosphere. ACME 1 requires a ballistic insertion which means that there will not be a parking orbit around the earth. Therefore, ACME 1 will need a STAR 27 injection stage designed by Thiokol in order to get to a lunar orbit. In addition, the satellite will need to be spin stabilized at 60 rpm before separation from the launch vehicle. The ACME 1 is shown in Figure A3.

• Dimensions:	- ACME 1	- width = approx. 40 in.
		- height = approx. 50 in.
	- STAR 27	- diameter = 27.3 in.
	- Total height = approx. 76 in.	
• Weight:	- ACME 1 = 215 lbs.	
	- STAR 27 = 800 lbs	
	- Total weight = 1015 lbs.	
• Power:	- GaAs solar cells supply 70 W of power.	
• Attitude Control:	- 3-axis stabilized	
	- includes sun sensor, star tracker, and rate sensor.	
• Communication:	- S-band using two low gain antennas.	
	- data rate = 2000 bps.	

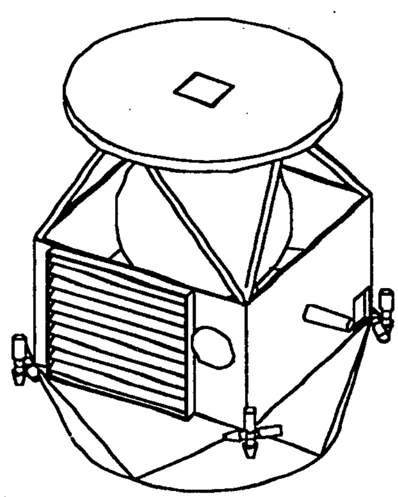

Figure A3. ACME 1

Eagle Class Lightsats

The Eagle class lightsats developed by TRW and Defense Systems Inc. were designed to carry out small, relatively inexpensive scientific investigations, technology demonstrations, and operational missions. The payload and propulsion modules can be custom tailored for each mission. The Eagle class lightsats module is shown in Figure A5.

• Dimensions:	- Core vehicle:	- diameter = 38 in. (12-sided)
		- height = 11.5 in.
	- Payload module:	- diameter = 38 in.
		- height = 14.5 in.
	- Propulsion module:	- diameter = 38 in.
		- height = 23 in.
	- Power module:	- diameter = 38 in.
		- height = 11.5 in.
	- Total possible height	= 60.5 in.
	- Critical diameter	= 38 in.
• Weight:	- No data at this time.	
• Power:	- Solar cells on fixed or articulated arrays.	
	- orbital avg. power up to 450 W.	
	- 30 W power from core vehicle.	
• Attitude Control:	- 3-axis spin or gravity gradient stabilization.	
• Communication:	- S-band and UHF-RF options.	
	- downlink = 1.0 Mbps maximum.	
	- uplink = 2.4 kbps maximum.	

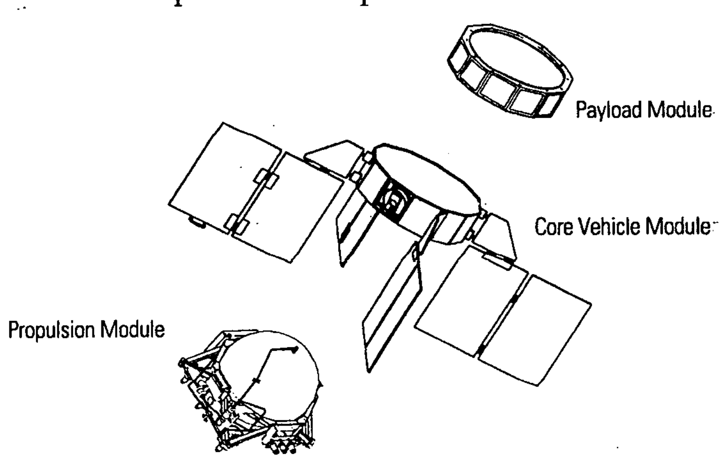

Figure A5. Eagle class lightsats modules

Total Ozone Mapping Spectrometer Earth Probe (TOMS-EP)

The TOMS-EP was designed by TRW for the NASA Goddard Space Center. The satellite mission is to perform daily mapping of the global distribution of Earth's ozone layer. TOMS-EP will be separated from the launch vehicle at an altitude of 355 km and then proceed to a sun synchronous 955 km high polar orbit. The bus is a variation of the Eagle class lightsats. TOMS-EP is shown in Figure A6.

• Dimensions:	- diameter = 44 in.
	- height = 70 in.
• Weight:	- payload = 77 - 132 lbs.
	- propellant = 155 lbs. full tank.
• Power:	- 25 W avg. power demand.
• Attitude Control:	- 3-axis stabilized.
• Communication:	- S-band
	- downlink > 0.25 Mbps.
	- uplink = up to 2 kbps.

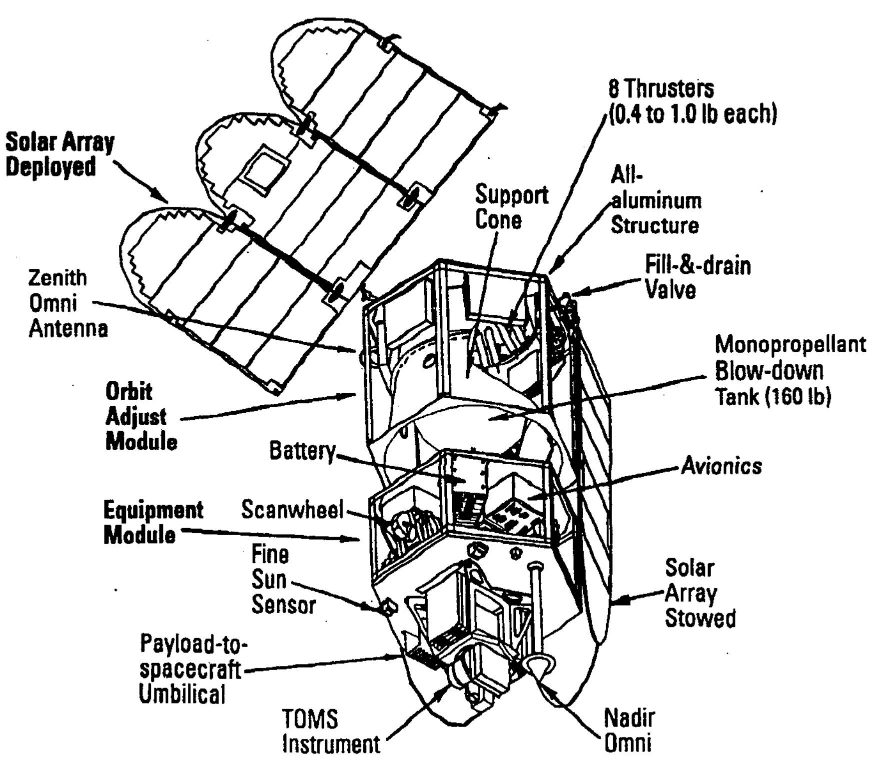

Figure A6. TOMS-EP

High Energy Transient Experiment (HETE)

HETE was designed by AeroAstro for MIT in order to detect and observe high energy events in the gamma ray, X-ray, and UV spectra. The HETE is placed in a 550 km circular orbit with an inclination less than 40 degrees. HETE is shown in Figure A7.

•Dimensions:	- diameter = 26 in.
	- height = 35 in.
•Weight:	- 265 lbs. maximum.
•Power:	- 30 W avg. supplied to payload.
•Attitude Control:	- sun pointing.
•Communication:	- S-band.
	- downlink = 230 kbps.
	- uplink = 7.5 kbps.

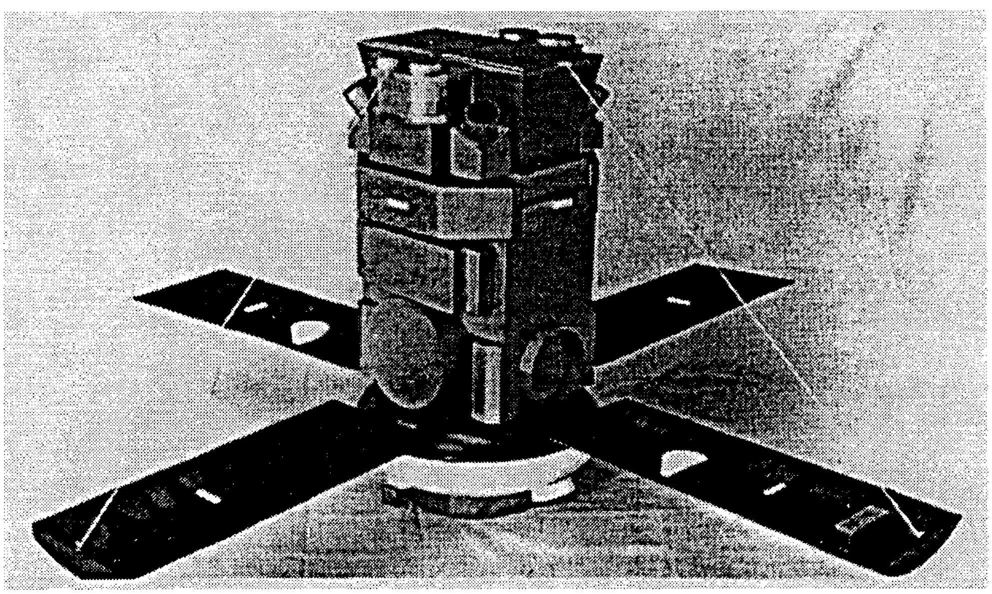

Figure A7. HETE

ALEXIS

ALEXIS was designed by AeroAstro for the Los Alamos National Laboratory to provide high resolution maps of low energy X-ray sources and ionosphere physics. ALEXIS is placed in a 750 km circular orbit at an inclination of 65 degrees. ALEXIS is shown in Figure A8.

• Dimensions:	- diameter = 23.6 in.
	- height = 39.4 in.
• Weight:	- 254 lbs.
• Power:	- supplies an avg. of 50 W to the payload.
• Attitude Control:	- sun pointing with a 2 rpm spin.
• Communication:	- S-band
	- downlink = 750 kbps.
	- uplink = 9.6 kbps.

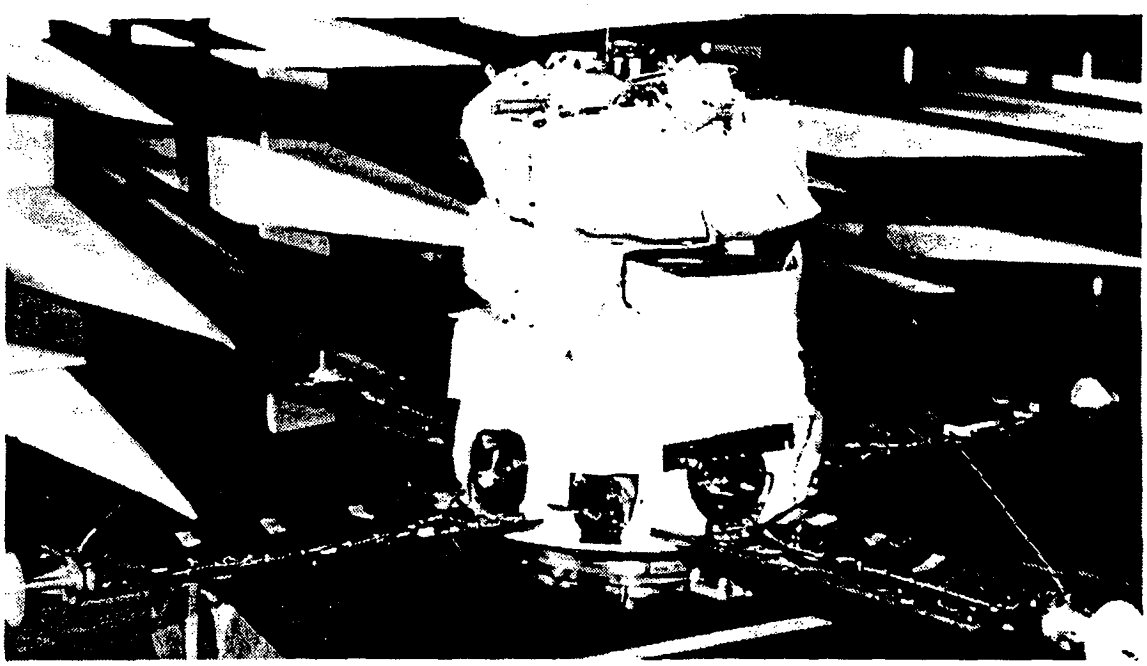

Figure A8. ALEXIS

Microlab

Microlab was designed by Orbital Sciences Corporation in order to support small space based science missions. Some of the applications that Microlab can accommodate are remote sensing, communications, space physics, and materials research. The customer provided payloads will be integrated into the spacecraft bus. The Microlab bus is shown in Figure A9.

- Dimensions: - diameter = 38 in.
  - height = 12 in.
- Weight: - experiment = 100 lbs. maximum.
  - S/C bus - no data at this time.
- Power: - Two deployable solar arrays
  - 100 W provided to payload.
- Attitude Control: - Nadir pointing or sun pointing.
  - spin stabilized.
  - optional 3-axis stabilization.
- Communication: - S-band.
  - Payload data rate = 2.0 Mbps peak.

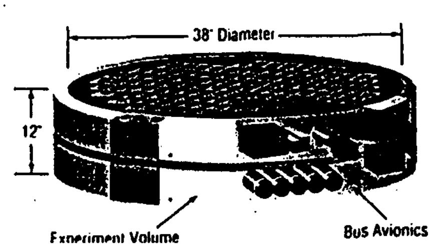

Figure A9. The Microlab Bus

APPENDIX B:
Launch Performance Analysis

In order to compute the attainable orbits for different rocket configurations, MDI set up a TK! Solver Model with simple rocket equations, (see appended TK! SOLVER rule and variable sheet). All stage specific impulses, propellant masses and dry masses are from the MSLS documentation [2]. The launch site latitudes for Vandenberg and Cape Canaveral were found in Wertz[Wertz:]. The wafer and support structure masses were based those used for the MSLS. The shroud mass was computed by MDI. The velocity losses during launch were estimated by matching up the TK! SOLVER calculated altitude with the previous MSLS, configuration B analysis, see Figure B.1.

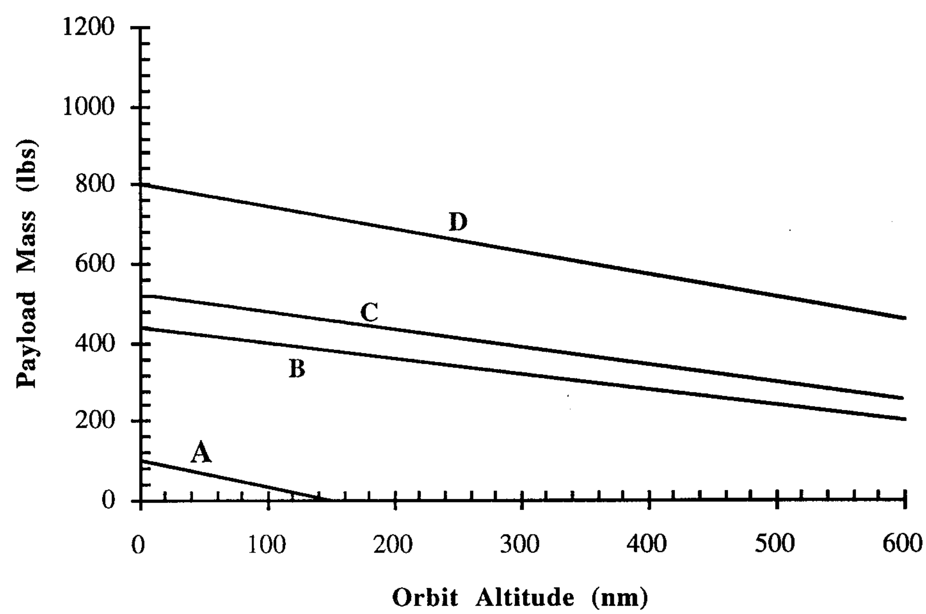

Figure B.1 MSLS configuration B performance.

The simplified model uses the basic rocket equation,

$$\Delta v = g \cdot Isp \cdot \ln\left({m_0}/{m_f} \right)$$

where g is the gravity constant at the surface of the Earth, m0 is the mass before the burn, mf is the mass after the burn, Isp is the specific impulse, and Δv is the total velocity change. Notice in the rule sheet that the various stages are accounted for.

The equations take into account the initial boost due to the rotation of the Earth, and assume that the complicated launch profile can be modeled as two impulsive burns (dv launch and dvins) as indicated in the rules sheet and Figure B.2.

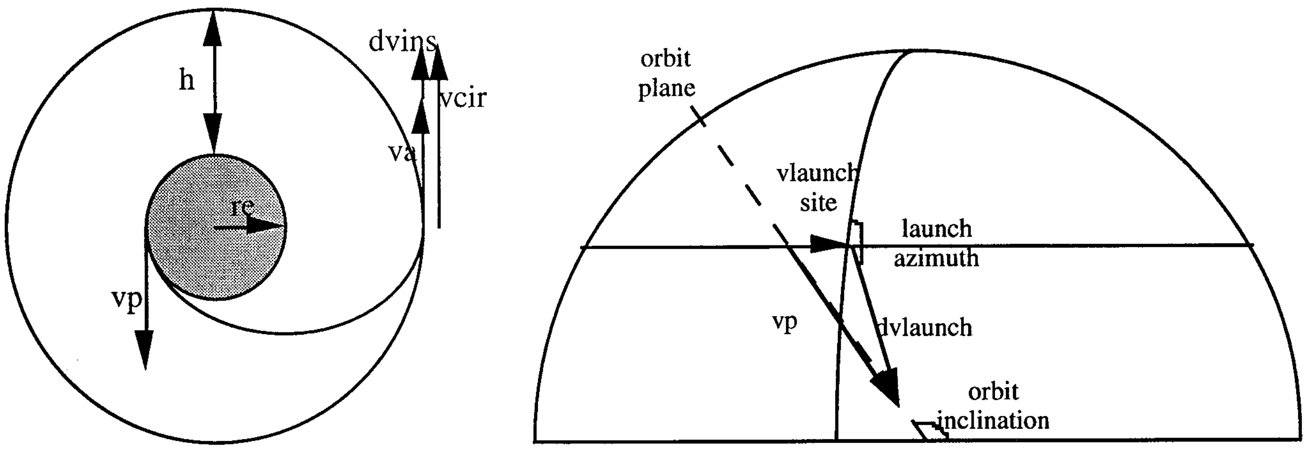

Figure B.2 TK! model equation diagrams

Although the equations are an extreme simplification of the true trajectory (which must be integrated), the model was considered valid for the following reasons. By appropriately calibrating the velocity losses, the MDI derived performance chart very nearly duplicated the MSLS chart, which was derived with a sophisticated integrating program [2]. Also, when a STAR 17A injection stage was added to the model, the improved performance was on the order of the performance for the Pegasus with a STAR 17A. Since the MDI shroud is as wide as the MSLS third stage (but notice it isn't as thick as the MSLS "B" shroud), MDI expects similar drag characteristics as the MSLS, configuration "B" missile.

The TK! model was used to generate the expected launch performances indicated in Figures B.3 and B.4. Notice that when a STAR 27 is used, the permissible payload mass is limited to 706 lbm due to a mass-above-the-bulkhead constraint of 1500 lbm, and the injection stage weighs 794 lbm.

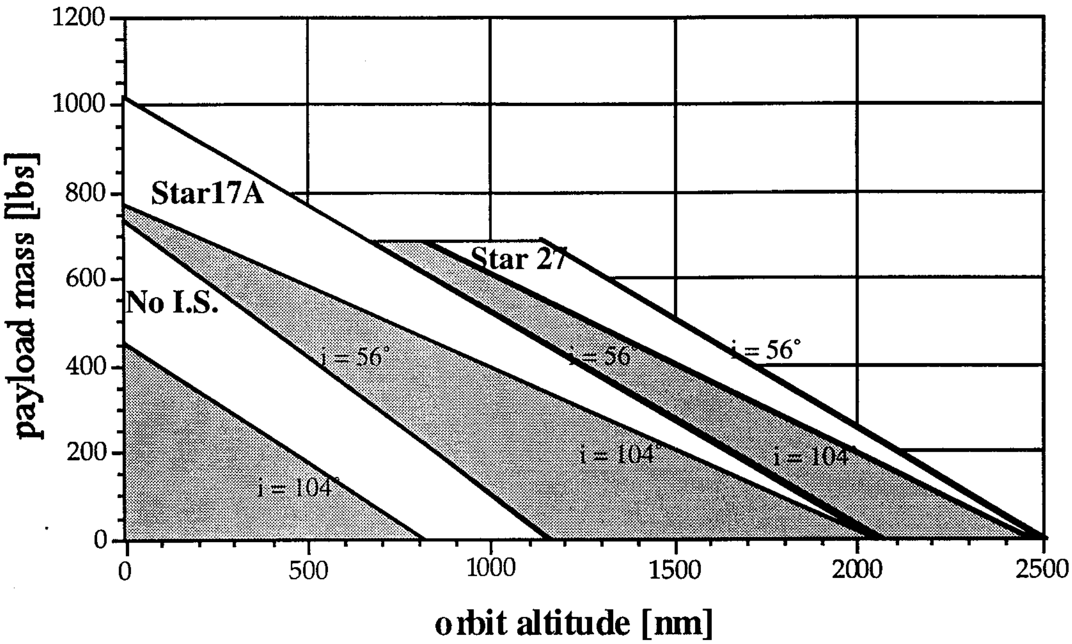

Figure B.3 Launch Performance for Launches from Vandenberg

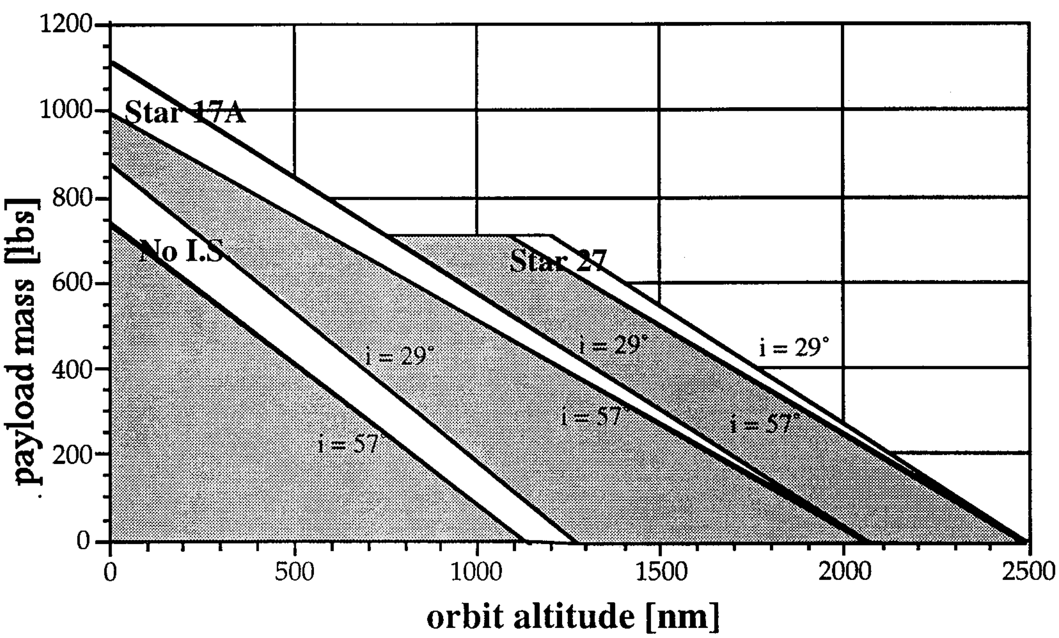

Figure B.4 Launch Performance for Launches from Cape Canaveral

Sample TK! Solver Variable Sheet

St	Input	Name	Output	Unit	Comment
	6378.137	re		km	radius of the Earth
	7.2921E-5	we		rad/s	rotation rate of Earth
	.00981	g		km/s2	gravity const. at Earth surface
	398600.44	mu		km3/s2	gravity parameter
	272	mwaf		kg	combined mass of wafers
	109	mshr		kg	mass of shroud
	20786	mIp		kg	stage I prop. mass
	2129	mId		kg	stage I dry mass
	6237	mIIp		kg	stage II prop. mass
	797	mIId		kg	stage II dry mass
	3313	mIIIp		kg	stage III prop. mass
	338	mIIId		kg	stage III dry mass
	0	mISp		kg	injection stage prop. mass
	1	pmf			prop. mass fraction
		mISd	0	kg	injection stage dry mass
	268	Isp1		s	specific impulse 1
	287	Isp2		s	specific impulse 2
	285	Isp3		s	specific impulse 3
	287.8	Isp4			specific impulse I.S.
	0	dvster		km/s	steering loss
	0	dvdrag		km/s	drag loss
	0	dvgrav		km/s	gravity loss
	1.2708961	dvatpr		km/s	atm. press. loss
		h	277.8	km	circular orbit altitude
		rp	6378.137	km	perigee radius of x-fer orbit (re)
		ra	6655.937	km	apogee radius of x-fer orbit (ins)
		a	6517.037	km	semi-major axis of transfer orbit
		rvb	5257.6505	km	radius of Vandenberg
		vvb	.38339437	km/s	velocity of Vandenberg
		vp	7.9891665	km/s	peri.vel. of x-fer orbit
		va	7.6557213	km/s	apo.vel. of x-fer orbit
		vins	7.7386336	km/s	insertion vel. for cir. orbit
		dvlaunc	8.1099709	km/s	first speed dv.
		dvins	.08291233	km/s	second speed dv.
	34.48	lat		deg	latitude of Vandenberg
L	104	i		deg	orbit inclination
		beta1	-17.06637	deg	inertial launch azimuth candidate
		beta2	197.06637	deg	inertial launch azimuth candidate
		eta	107.06637	deg	local angle b/w vvb and vp
		az1	199.6566	deg	local launch azimuth (i>87)
		az2	160.3434	deg	local launch azimuth (i<87)
		m0	34144.978	kg	mass before burn 1
		m1	13358.978	kg	mass after burn 1
		m2	11229.978	kg	mass before burn 2
		m3	4883.9778	kg	mass after burn 2
		m4	4086.9778	kg	mass before burn 3
		m5	773.97783	kg	mass after burn 3
		m6	163.97783	kg	mass before burn 4
		m7	163.97783	kg	mass after burn 4
		mf	163.97783	kg	mass at orbit ins. (mpay)
		dv1	2.4671993	km/s	first burn
		dv2	2.3442357	km/s	second burn
		dv3	4.6523443	km/s	third burn
		dv4	0	km/s	fourth burn
		dvtot	9.4637793	km/s	total burn
		mpay	163.97783	kg	mass of the payload

TK! Rule Sheet

```
rp = re
ra = re + h
rvb = re * cosd(lat)
vvb = we * rvb
m0 = mIp + mId + mIIp + mshr + mIId + mIIIp + mIIId + mISp + mISd + mpay + mwaf
m1 = m0 - mIp
m2 = m1 - mId
m3 = m2 - mIIp - mshr
m4 = m3 - mIId
m5 = m4 - mIIIp
m6 = m5 - mIIId - mwaf
m7 = m6 - mISp
mf = m7 - mISd
pmf = mISp/(mISp + mISd)
dv1 = g * Isp1 * ln(m0/m1)
dv2 = g * Isp2 * ln(m2/m3)
dv3 = g * Isp3 * ln(m4/m5)
dv4 = g * Isp4 * ln(m6/m7)
dvtot = dv1 + dv2 + dv3 + dv4
a = (ra + rp)/2
vp = sqrt(mu * (2/rp - 1/a))
va = sqrt(mu * (2/ra - 1/a))
vins = sqrt(mu/ra)
dvins = vins - va
sind(beta1) = cosd(i)/cosd(lat)
beta2 = 180 - beta1
eta = beta2 - 90
dvlaunch = sqrt(vvb^2 + vp^2 - 2*vvb*vp*cosd(eta))
sind(eta)/dvlaunch = sind(alpha1)/vp
alpha2 = 180 - alpha1
az1 = 270 - alpha1
az2 = 270 - alpha2
dvlaunch + dvins + dvdrag + dvgrav + dvster + dvatpr = dvtot
hnm = h / 1.852
mlbs = mpay * 2.24
```

APPENDIX C:
Finite Element Model Specifications

Finite element properties are listed in this appendix. MDI felt that time would be best spent coding a finite element model in the same language as the existing Minuteman finite element model. To this end this appendix lists all the parameters needed to create the final finite element model.

C.1 Payload Support Bulkhead

The payload support bulkhead is identical to the MSLS payload support bulkhead with the exception of the holes drilled in the bulkhead. The MDI design relies on a quadrilateral truss design. MDI's design requires the MSLS bulkhead be delivered without payload attach holes. The electrical interface is also different between designs. MSLS has three electrical connector groups. MDI will have only one. The PSB then has the same elemental and dimensional properties as the MSLS. The PSB dimensions are listed in Table C.1.1. Material properties are listed in Table C.1.2. The PSB is shown in Figure C.1.1.

Table C.1.1 PSB dimensions

Height	15 inches
Base Diameter	52 inches
Bulkhead Diameter	47.5 inches
Truss attach hole radius	20 inches
Electrical hole radius	20 inches
Electrical hole off axis	45 degrees

Table C.1.2 PSB material properties

Material	Aluminum 7075
Finish	Alodine 600
Thickness	2.0 inches
Weight	25 lb.
Truss tension pullout capability	15,000 lb. per truss

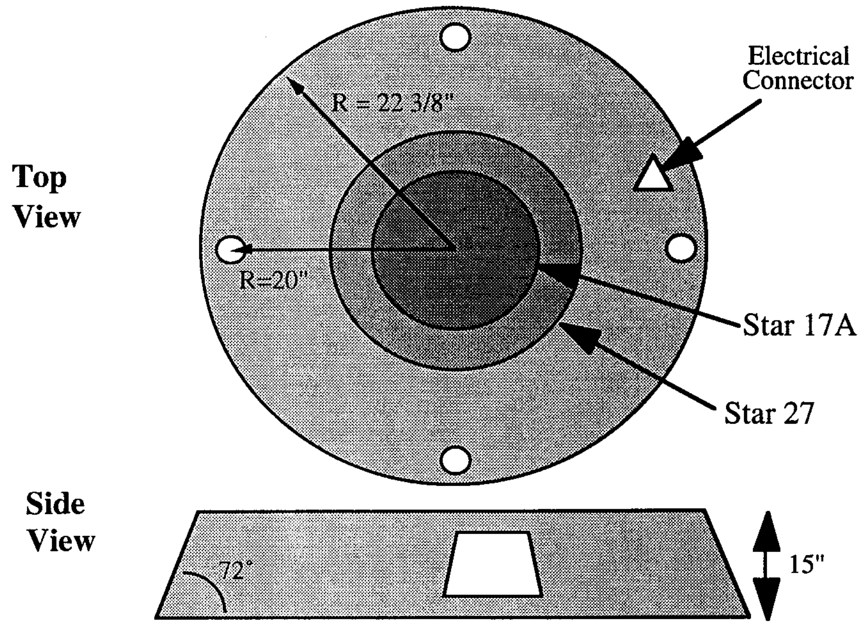

Figure C.1.1 Payload Support Bulkhead

C.2 Truss Elements

The truss elements are composed of three members. There is a top piece which attaches to the lip of the payload vehicle. There is a variable length center section that has a nominal length of 32 inches. There is also a bottom piece which connects to the PSB. The truss is shown in Figure C.2.1. The material properties of the truss support structure is shown in Table C.2.1. Detailed modeling of the truss structure will be concluded during the R&D phase of the project.

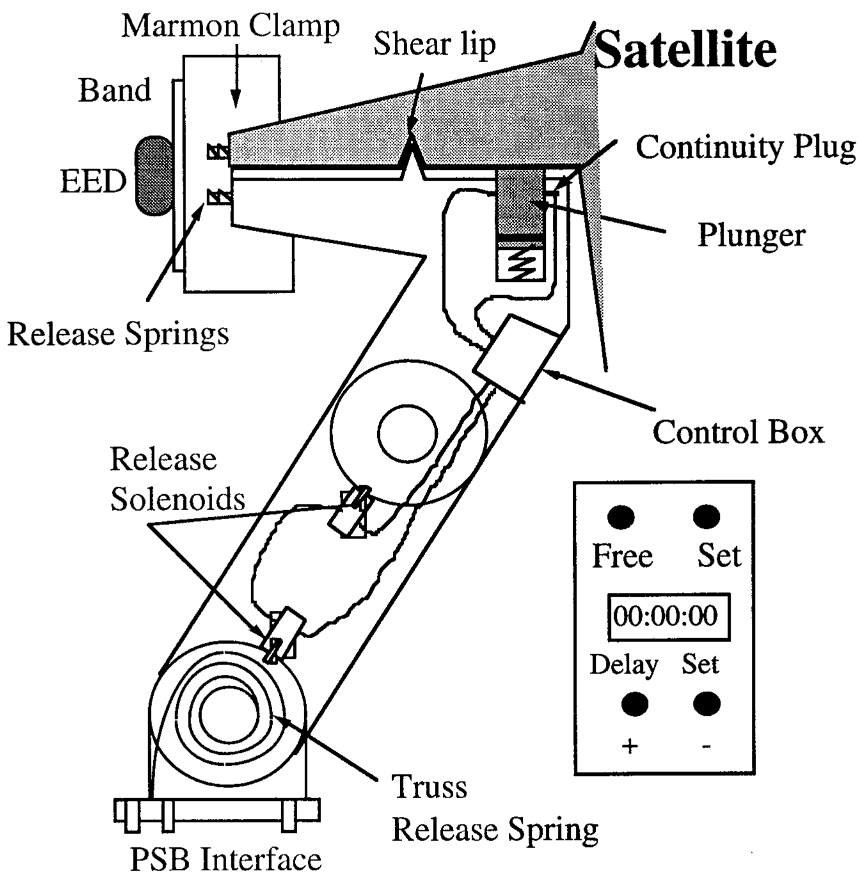

Figure C.2.1 Support Truss

Table C.2.1 Truss material properties

Material	Steel
Nominal Length	32 inches
Nominal Weight	8 lbs
Young's Modulus	30 E + 6 psi
Poisson's Ratio	0.24

C.3 Shroud

The shroud is made of a composite material that is RF transparent. The shroud is a secant ogive cylinder. Dimensions of the shroud are given in Table C.3.1. The shroud is shown in Figure C.3.1.

Table C.3.1 Shroud dimensions

Diameter	52 inches
Height	120 inches
Weight	214 lbs

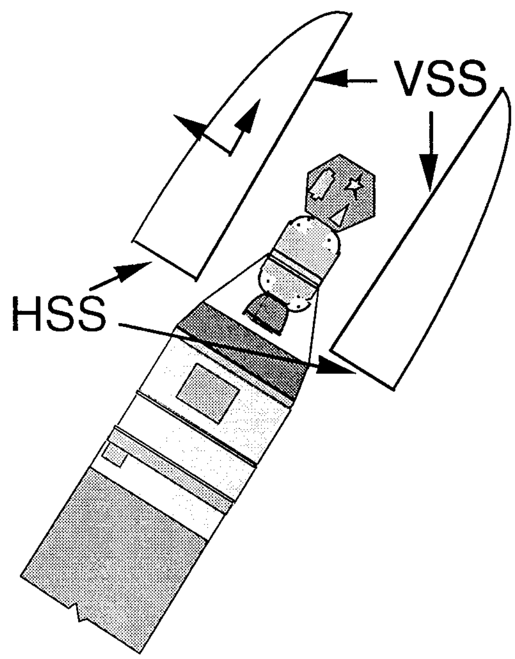

Figure C.3.1 MDI shroud

APPENDIX D:
Payload User's Guide

For the benefit of satellite customers who are considering using the MDI missile as their launch vehicle, MDI has compiled all the satellite concerns in one area. The main points discussed are the attainable orbits of launch system, the mechanical interfaces used to mount the satellite to the missile, electrical interfaces available for the payload, and launch system costs.

D.1 Attainable Orbits

Figures D.1 and D.2 summarize the attainable orbits customers can expect from the MDI launch system. To meet a wide range of orbit needs, MDI can provide orbit inclinations between 29° and 57° by launching from Cape Canaveral and inclinations between 57° and 104° by launching from Vandenberg. The customer may use either a STAR 17A or STAR 27 injection stage if the orbit requirements demand better performance.

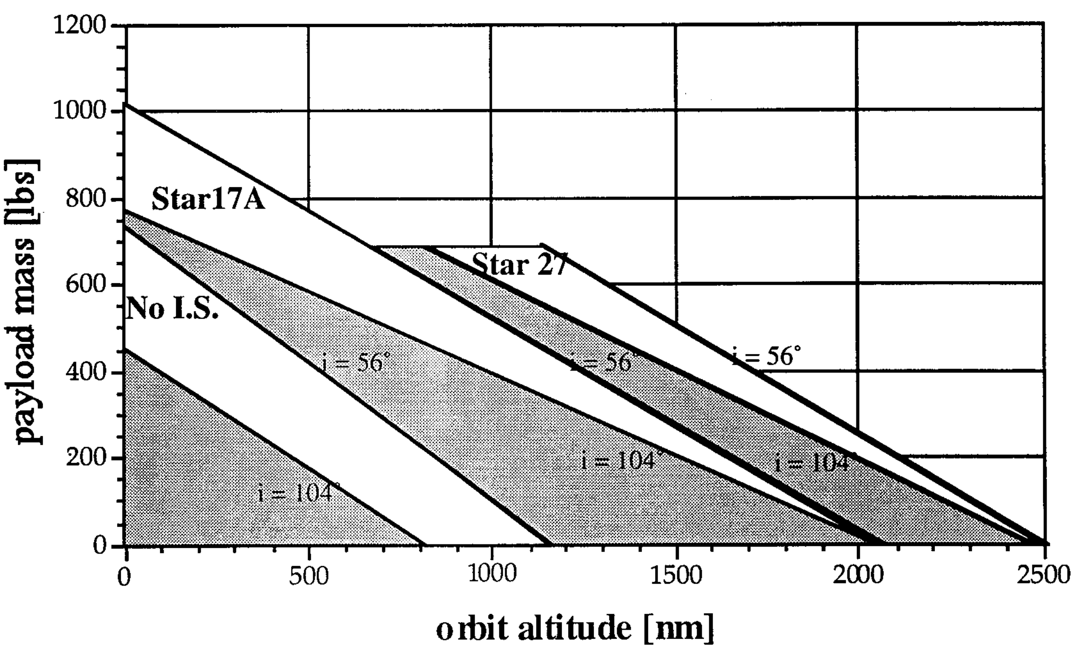

Figure D.1 Launch Performance at VAFB Launch Site

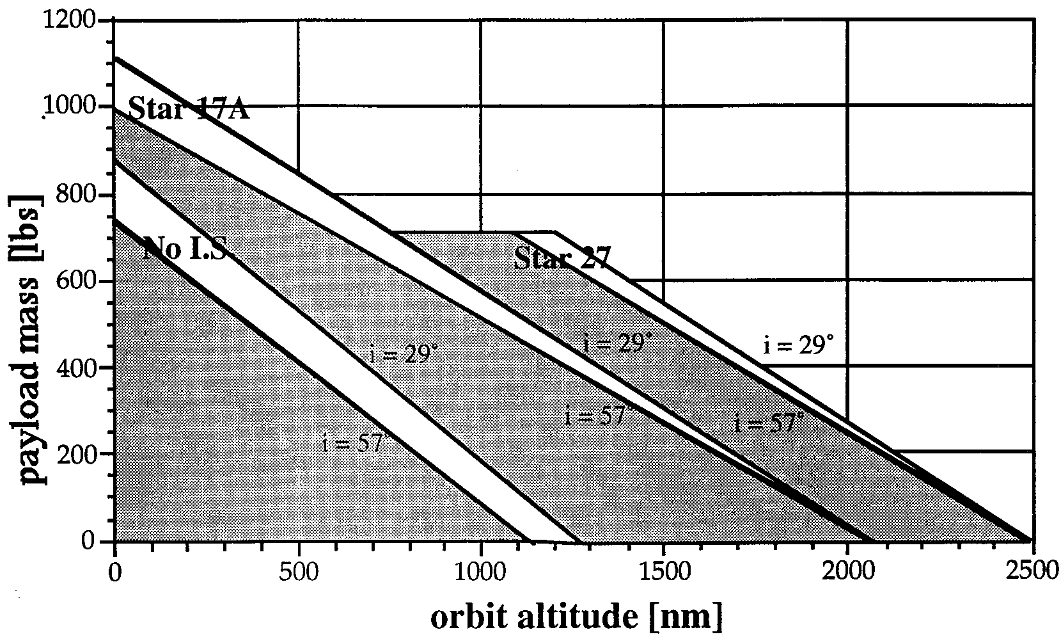

Figure D.2. Launch Performance at Cape Canaveral Launch Site

Like all launch systems, some orbit insertion error that the payload will need to tolerate. MDI expects orbit altitude errors of ± 15 nm and inclination errors of ± 0.5°.

D.2 Mechanical Interfaces

Payload customers may connect directly to the missile bulkhead with use of the flexible MDI marmon clamp, see Figure D.3. If the payload requires an injection stage, MDI will provide for an interface between the injection stage and payload as seen in Figure D.4.

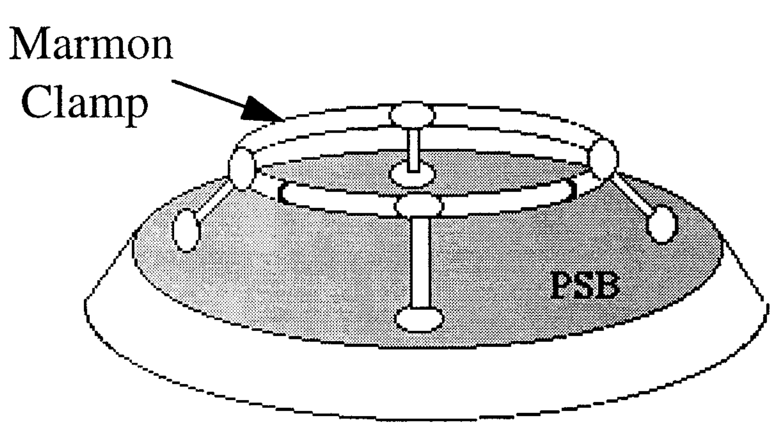

Figure D.3. MDI Marmon Clamp

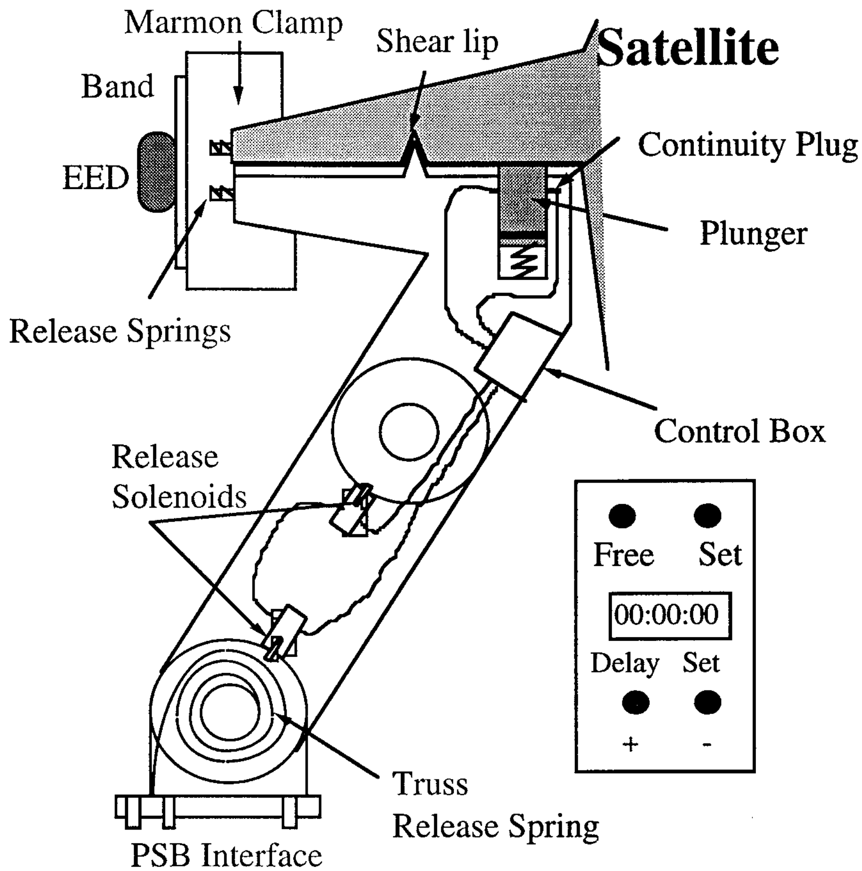

Figure D.4. MDI Payload Attach Fitting

D.3 Launch Time and Costs

Once a customer decides to use the MDI missile as their launch system, they can expect a launch after 6 months and the following launch costs:

Launch Type	Cost
Regular Launch	\$ 7,500,000
Added STAR 17A	+\$500,000
Added STAR 27A	+\$1,000,000

D.4 Expected Launch Environment

Figures D.4 thru D.9 show the anticipated launch environment that a potential satellite will be exposed to.

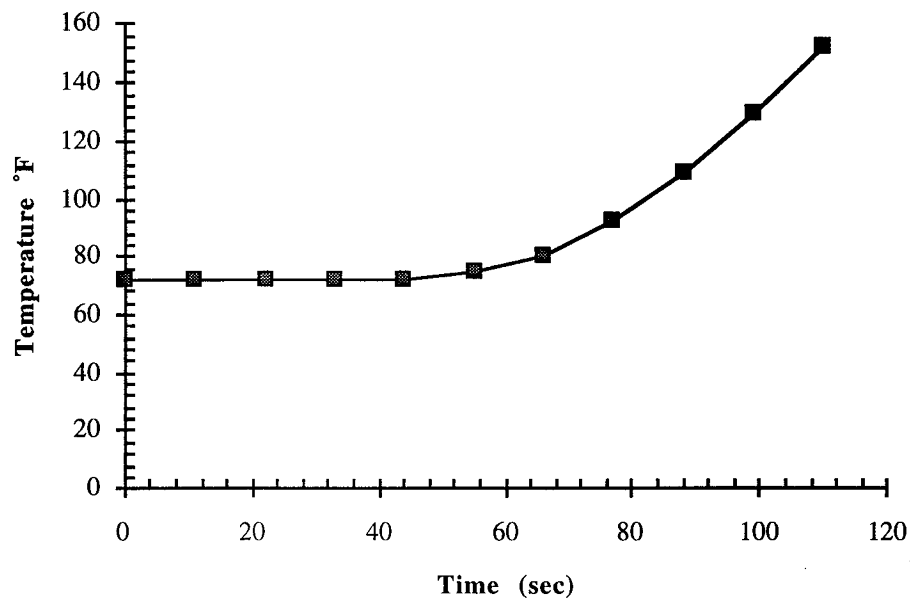

Figure D.5 Thermal environment

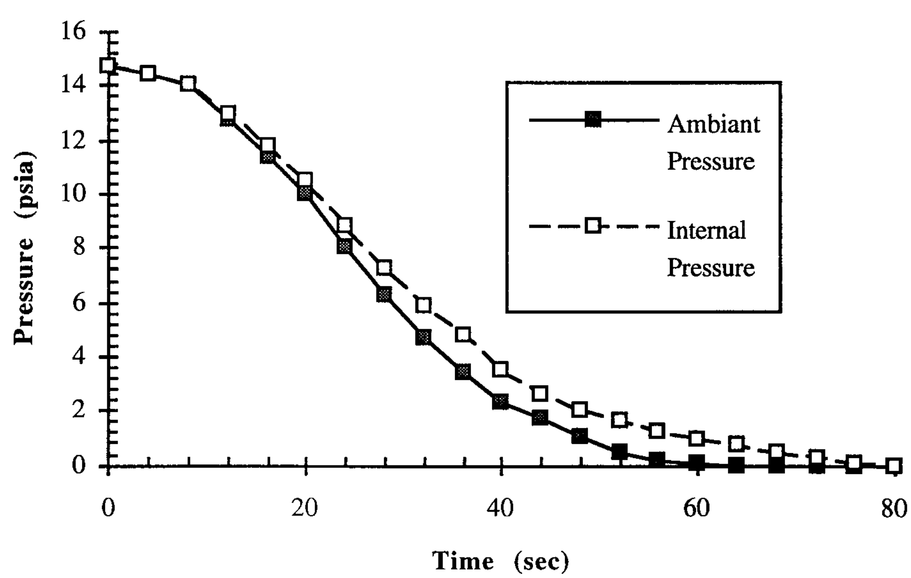

Figure D.6 Pressure environment

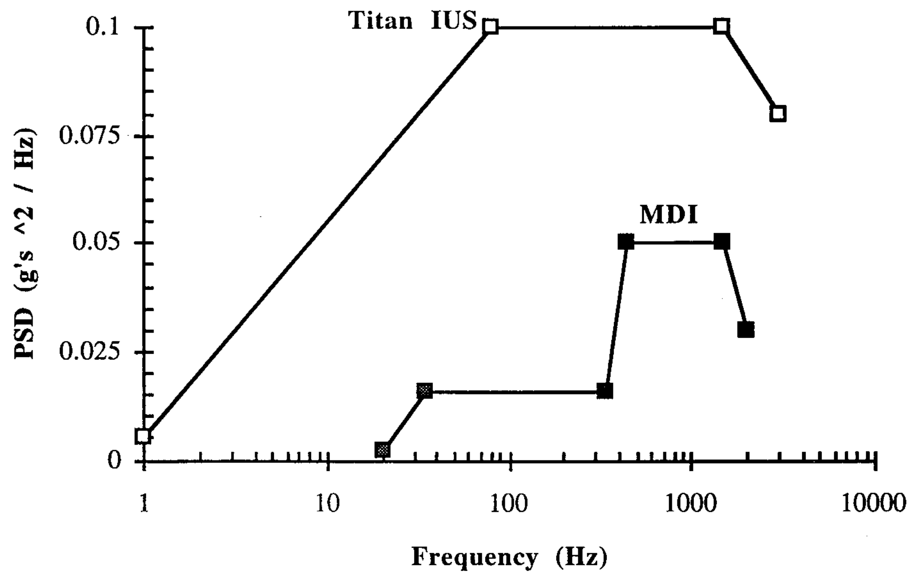

Figure D.7 Random vibration environment

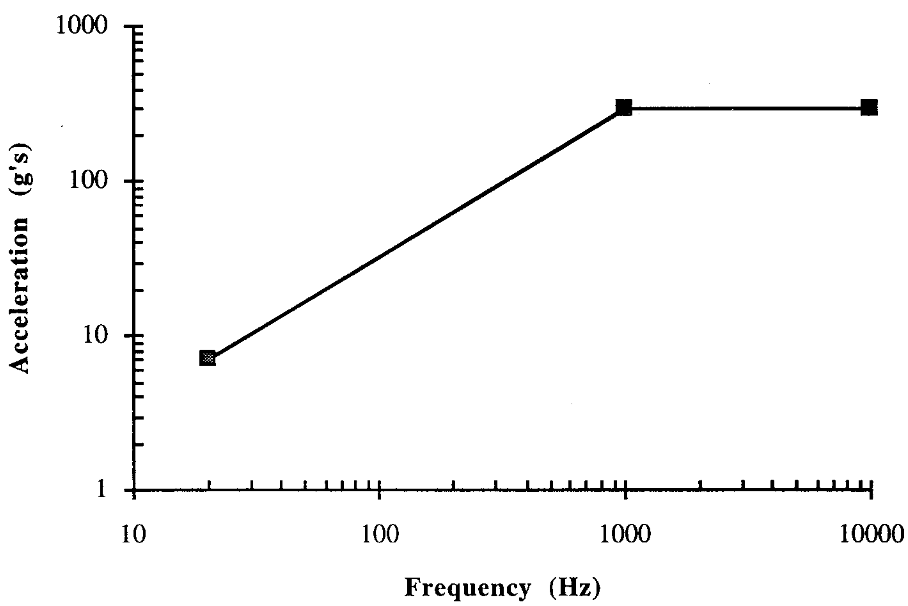

Figure D.8 Shock environment