N94- 25685

MINOTAUR
MARYLAND'S INNOVATIVE ORBITAL TECHNOLOGICALLY ADVANCED UNIVERSITY ROCKET

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Abstract

Over the past decade, there has been an increasing interest in designing small commercial launch vehicles. Some of these designs include OSC's Pegasus, and AMROC's Aquila. Even though these vehicles are very different in their overall design characteristics, they all share a common thread of being expensive to design and manufacture. Each of these vehicles has an estimated production and operations cost of over $15K/kg of payload. In response to this high cost factor, the University of Maryland is developing a cost-effective alternative launch vehicle, Maryland's Innovative Orbital Technologically Advanced University Rocket (MINOTAUR). A preliminary cost analysis projects that MINOTAUR will cost under $10K/kg of payload. MINOTAUR will also serve as an enriching project devoted to an entirely student-designed-and-developed launch vehicle.

This preliminary design of MINOTAUR was developed entirely by undergraduates in the University of Maryland's Space Vehicle Design class. At the start of the project, certain requirements and priorities were established as a basis from which to begin the design phase: (1) carry a 100 kg payload into a 200 km circular orbit; (2) provide maximum student involvement in the design, manufacturing, and launch phases of the project; and (3) use hybrid propulsion throughout. The following is the list of the project's design priorities (from highest to lowest): (1) safety, (2) cost, (3) minimum development time, (4) maximum use of off-the-shelf components, (5) performance, and (6) minimum use of pyrotechnics.

MINOTAUR Overview

MINOTAUR is a four-stage custom/modular rocket (Figure 1). It stands 30 meters tall, has a gross lift off weight of 30,000 kg, and generates 750 KN of thrust at lift off. Stages 1-3 are composed of modules, each approximately eight meters tall and weighing 2250 kg, in a 7-5-1 configuration respectively. Each module is identical in size and mass except for its nozzle. The nozzle design for each stage module is different to achieve optimal performance from varying ambient pressures during ascent. Stage 4 is a scaled-down (52%) custom version of the modular design.

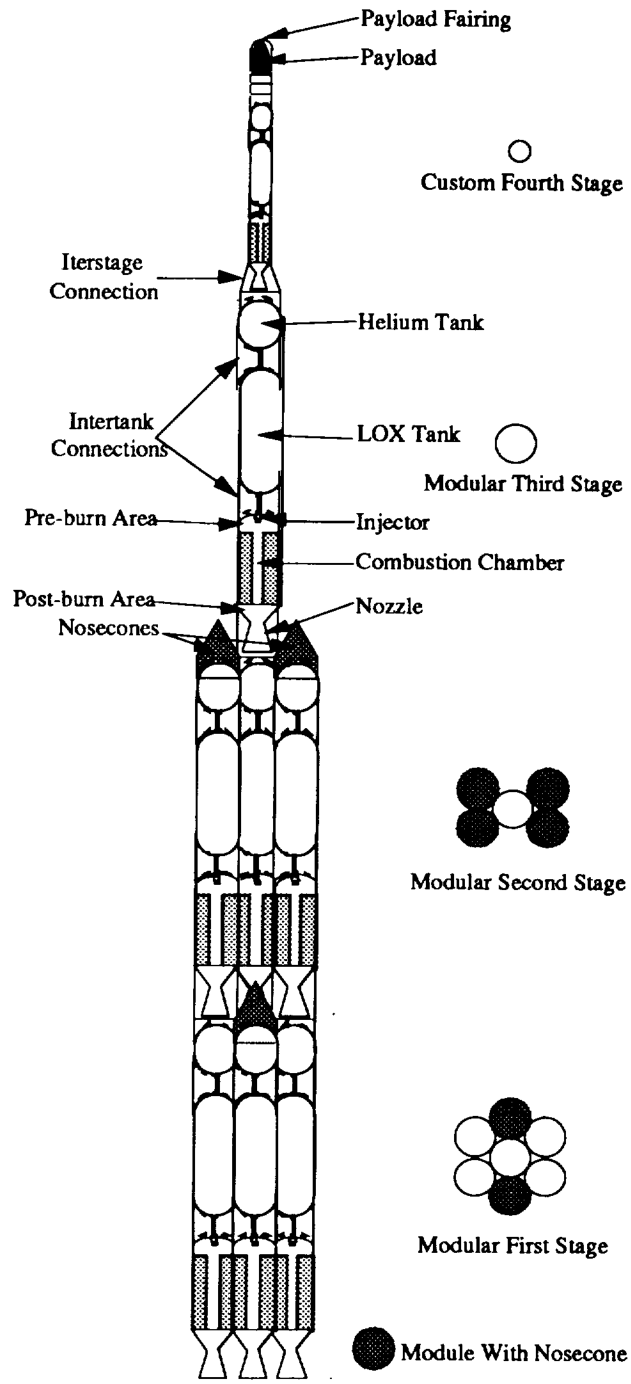

Fig. 1 MINOTAUR side and top view

MINOTAUR has also been developed to explore the possible use of hybrid technology in a viable orbital vehicle. MINOTAUR's propulsion system consists of a liquid oxidizer and a solid grain of fuel, each stored in separate chambers. The oxidizer is introduced to the grain and combined during firing in the holding chamber for the fuel grain. A pressurant is needed to keep the oxidizer at high pressure for blow down to the combustion chamber to occur. The hybrid is considered as the medium between the complexities of liquid systems and the simplistic mechanical operation of a pure solid.

Mass Budget

The mass budget is a detailed component mass list assembled to identify the overall vehicle masses and margins. The current mass budget is divided into six parts: one for each modularized stage, the fourth custom-designed stage, a stage mass summary, and overall conclusive masses. Tables 1 and 2 show the system mass lists for the module and fourth stage. A complete component mass list can be found in the final report. The first, second, and third stage typical module masses do differ slightly even though hey are basically the same module. The difference lies primarily in nozzle mass for reasons already explained.

Margins were added to the vehicle's inert mass to compensate for any mass increase during production. After production is completed, any remaining positive margins can be used to increase payload mass capability or to achieve a higher altitude.

Center of gravity calculations were done to help predict the dynamic stability of the vehicle. The center of gravity of each component was calculated, and then transformed into a module, fourth stage, and vehicle center of gravity. The reference station for all center of gravity calculations was the exit plane of the first stage nozzle. Figures 2 and 3 show mass and center of gravity vs. time of flight, respectively.

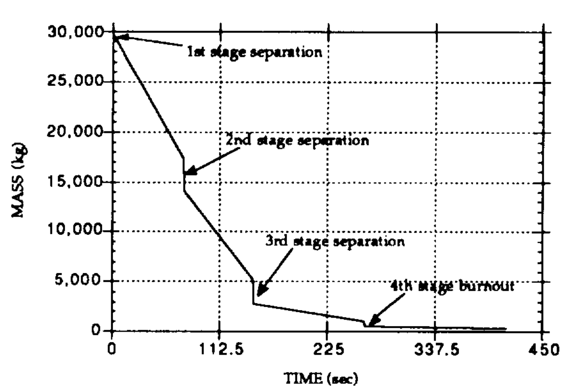

Fig. 2 Mass vs time graph

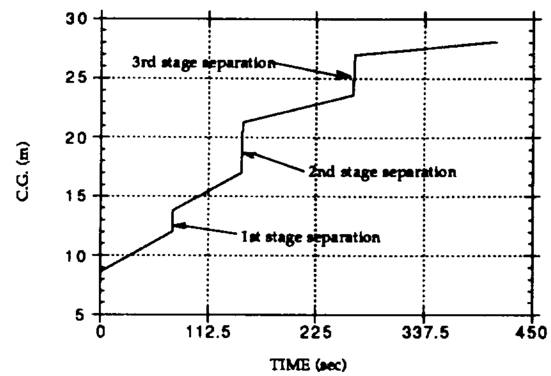

Fig. 3 Center of gravity vs time graph

Table 1 Module mass list

System	Module Mass(kg)
Combustion chamber	78
Oxidizer tank	193
Pressurization tank	59
Ignition	2
Propulsion	61
Destruct	18
Intertank	53
Avionics/electronics	8
Roll control (3rd stage)	5
Subtotal	477
Inert mass margin	7%
LOX	1215
HTPB	552
Total mass	2244

Table 2 Stage 4 mass list

System	Stage 4 Mass(kg)
Combustion chamber	22
Oxidizer tank	30
Pressurization tank	13
Ignition	2
Propulsion	11
Destruct	14
Intertank	27
Avionics/electronics	39
Roll control	5
Power	9
Subtotal	163
Inert mass margin	9%
LOX	180
HTPB	82
Total mass	425

Vehicle Costing

One of the most important requirements of the project is to produce a vehicle at low cost. Therefore, the production effort will have to keep cost in mind at all times. This will mean that elegance and high performance will sometimes have to be sacrificed in order to meet cost requirements.

Cost estimates of the vehicle components have been divided into major vehicle systems. The cost estimates are derived from the following major sources: supplier cost figures for appropriate or similar components, general cost formulas using component mass values (primarily applied for structural elements), and costing formulas from Space Mission Analysis and Design, by Wertz and Larson[3] (primarily applied for electronic components). Table 3 outlines MINOTAUR's systems cost.

Table 3 Vehicle costing

System	Cost($K)
Combustion chamber	132.3
Oxidizer tank	198.3
Pressurization tank	70.7
Intertank	34.9
Propulsion	64.8
Destruct	16.8
Interstage	14.9
Separation	15.4
Power	1.0
Avionics	160
Pad structures	10.8
Flight insurance	50
Margin	30
Total cost	799.9

Trajectory Model

A FORTRAN trajectory model, HYTRAJ (hybrid trajectory), was developed to verify MINOTAUR's capability of meeting the requirement of putting a 100 kg payload into a 200 km orbit. The end condition required for this is a circular orbit at 7784.3 m/sec. HYTRAJ models the kinematics of the rocket. No dynamics are used to compute the rocket's performance in flight. The trajectory is defined using a preset pitchover function. The rocket launches vertically, performs a pitchover maneuver from 3 to 8 seconds into the flight, then is modeled as flying at zero angle of attack, and therefore zero lift, through the region of high dynamic pressure, and finally flies to tangential velocity at 200 km. This program assumes a rotating Earth which is accounted for using a simple vector addition technique. The Earth has a varying gravity based on altitude. It models drag forces and can be used to predict angles of attack. Aerodynamic heating at the nose is computed as is the variation of thrust with altitude and an ablating nozzle. The time step used in HYTRAJ is 0.1 seconds.

HYTRAJ results indicate that the current design can reach the required orbit as shown in Figure 4. The altitude reaches 200 km while the flight path angle goes to 1.57 rad (90 degrees) and the velocity is 7784 m/sec. The sudden increase in velocity around 130 secs is due to a coordinate transformation from Earth-fixed to space-fixed coordinates. This jump accounts for the velocity of the launch site (378 m/sec) due to the Earth's rotation.

Payload Accommodations

To develop payload accommodations for a given payload, the following criteria must be identified early in the mission planning process.

1. structural, electrical and avionics interfaces
2. adapters for separation systems, etc.
3. communications architecture
4. launch system environment before and during launch.

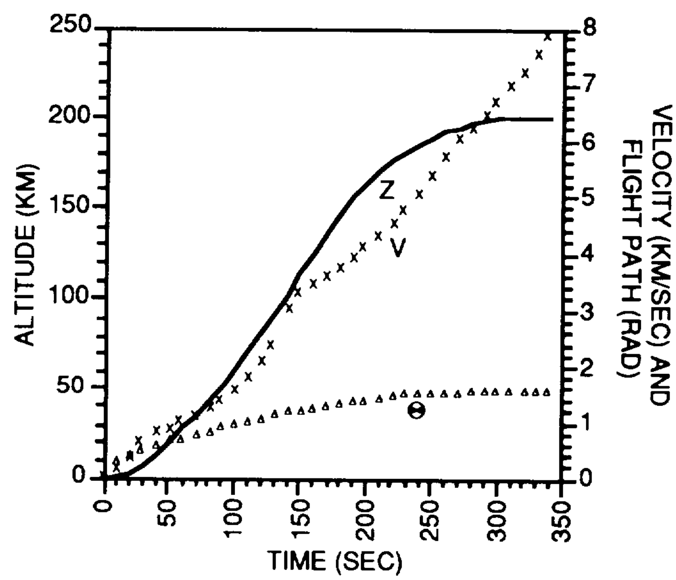

Fig. 4 Trajectory model

Items 1-3 are dependent upon the particular payload chosen for a launch. Once a payload is chosen, all details pertaining to the accommodation requirements need to be analyzed quickly so integration can become a smooth and efficient process. Item 4, however, is dependent upon the launch vehicle. Based on predicted launch characteristics, MINOTAUR's payload environment requirements are shown in Table 4.

As determined by MINOTAUR's trajectory code, payload separation will begin at 200 secs when dynamic pressure = 0.5 N/sq m. This time also corresponds to the approximate point where the heat transfer from the payload fairing to the payload is greater than the heat transfer from the rarefied atmosphere to the unprotected payload. Therefore, the payload fairing is separated to eliminate unnecessary heat transfer to the payload. Another benefit from early fairing separation is that after separation less inert weight is being carried by the rocket. By ejecting the payload fairing before third stage separation, the vehicle's performance is increased.

Table 4. Payload Environment Requirements

Payload Environment	Pre-launch	Ascent
Thermal	Pending	1341 W/sq m
Electromagnetic	Pending	Pending
Contamination	Pending	Pending
Loads		
Venting	NA	1 psi +/-
Aerodynamic	NA	.5 N/sq m
Axial acceleration	NA	4.5 g
Lateral acceleration	NA	1 g
Axial dynamic	NA	30 Hz
Lateral dynamic	NA	15 Hz
Acoustic	NA	Pending

Heating Analysis

Two heating models, one from NACA Rep. 1381[1], the other from Anderson's <u>Hypersonic and High Temperature Gas Dynamics</u>[2], were used to determine the stagnation point heating of MINOTAUR's payload fairing. These models were originally developed for aerodynamic heating analyses of reentry vehicles. Since it was uncertain if the models could be used for anything else but re-entry analysis, they were used as verifiers of each other's validity for fairing heat transfer. Both of these models were used for MINOTAUR's trajectory code to determine the maximum heat transfer to the fairing. Figure 5 shows how the two models compare for the 0.21-meter radius of MINOTAUR's payload fairing.

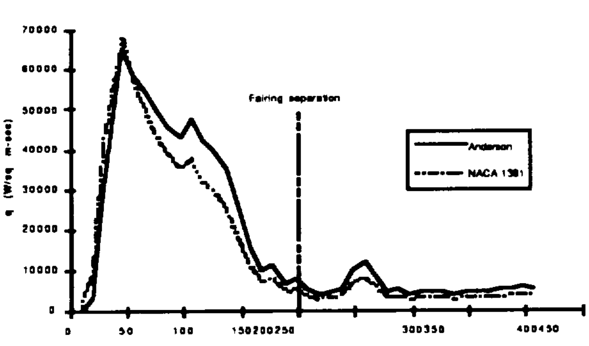

Fig. 5 Stagnation point heat transfer for a 0.21-m sphere

These models were relatively close in their prediction of nose heating, and were therefore considered valid. Anderson's model was chosen to be used for any future analyses because it yielded the highest heat transfer rates. Designing the payload fairing to a lower heat transfer rate than might occur during launch increases the chance of thermal load failure. By integrating the resulting curve from Anderson's model as seen in Figure 5, the maximum stagnation point heat load was determined to be 7200 KJ/sq m.

Separation Systems

The separation systems are divided into two basic types: payload fairing separation systems, and stage separation systems. Both systems require the design and fabrication of a Marmon clamp. A Marmon clamp is a continuous ring held together by an angular clamp. By releasing the clamp tension, the joint is separated and pulled away from the upper stage by means of a spring which is attached to both the lower stage and the Marmon clamp itself. As seen in Figure 6, the clamp is composed of a metal strap and many shoe segments. Marmon clamps were chosen over the shape charge or zipper for three basic reasons: both the shape charge and the zipper require an explosives engineer to design, are very dangerous to handle, and are very expensive. On the other hand, Marmon clamps are student-designed and manufactured, and therefore have a very low cost. Marmon clamps also have a built-in mechanical redundancy because they are releasing at more than one point.

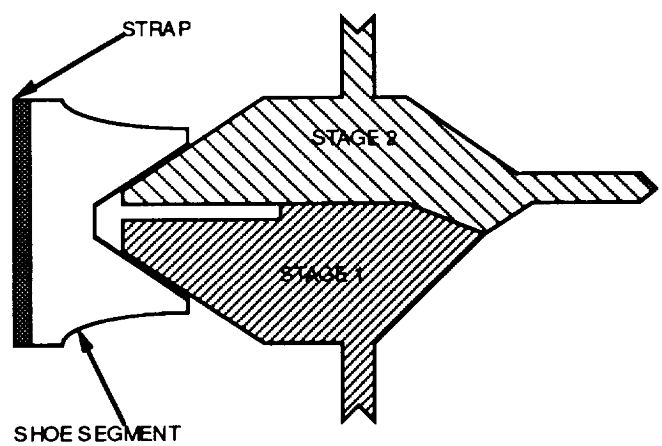

Fig. 6 Marmon clamp design

Destruct System

The destruct system was designed to meet the first and foremost design priority, safety. For this reason the use of pyrotechnics was ruled out. The destruct system was designed to depressurize the rocket and bring it down in a controlled manner. The rocket will receive a signal from the ground which will be distributed to each module. The valves will then be actuated using 6-V Airtronics servos powered by 4 "D" cell batteries. These batteries will be located on top of the modules and will be accessible through the stage fairings. The LOX and helium tanks are depressurized by activating two electrically actuated butterfly valves per tank. The flow of LOX is cut off from the combustion chamber using a modified pressure relief valve. This allows the propellent to burn itself out. The advantage of this system is that solenoid valves and constant pneumatic pressure are not needed. The butterfly valves are constructed from stainless steel because of its high strength to weight attributes, good low temperature material characteristics, and relative low cost.

Propulsion System

The characteristics of MINOTAUR's propulsion system as described in the overview follow.

Hydroxyl-terminated polybutadiene with no additives (save the carbon black) is used for the solid grain. The oxidizer was chosen to be liquid oxygen. A helium pressurant will provide the pressure differential to blow down the oxygen into the combustion chamber. Thrust vectoring uses a liquid injection system of oxygen. Roll control for the upper stages feeds off the main blow down supply of helium.

The valves for release of the helium and oxygen are butterfly and ball valves, respectively. They are electrically actuated with the main oxygen valve being heated. The configuration of the grain is a 3-point rounded star. The ignition system consists of propane injection onto the grain, with ignition provided by a sparking electrical element.

The main LOX injector is a shower head injector. The conical, 15° half angle nozzle is coated with silica phenolic.

Regression model

The requirement for accurately estimating the fuel's regression rate is an important aspect of the hybrid rocket system. In a hybrid, as in a solid, the regression rate of the fuel is the driving factor in the engine's performance and physical characteristics. The rate of regression gives the amount of fuel that is burned from the grain per second. The regression rate in turn determines the fuel's combustion characteristics, which in turn determine the rocket's performance. The regression model is needed to represent the coupling of the various chamber conditions (pressure, temperature, burn area, etc.).

Combustion takes place in the turbulent boundary layer above the solid fuel grain as shown in Figure 7.

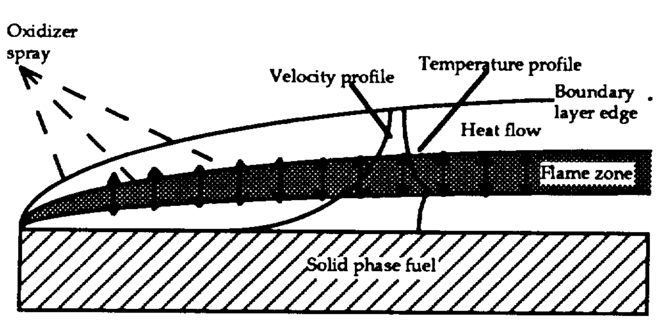

Fig. 7 Hybrid combustion model

The fuel is transported within the boundary layer (crossing the fuel), mixed with the oxidizer, and burned. Due to the fluid mechanics of the situation, regression rate is a function of the local mass flux (which depends upon the regression rate at all points upstream as well as the instantaneous port area) and the local burning perimeter. Pressure can also be a factor if enough radiation is involved. This coupling does not allow the use of a solid rocket regression rate (rate proportional to pressure raised to an empirical constant) for a hybrid system.

Grain configuration

A 3-point rounded star grain configuration was chosen.

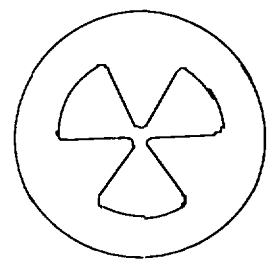

Fig. 8 Grain design

The grain design must provide the constant thrust profile that the trajectory model has established. The star provides a relatively neutral burn and covers the wall during the firing to help protect the chamber. The star configuration that was chosen was based on the assumption of a volumetric loading of 70%. This loading was chosen to allow a large flow of oxidizer through the system to aid in combustion.

Oxidizer pressurizing system

MINOTAUR utilizes liquid oxygen as the oxidizer. To keep the liquid oxygen pressurized and provide the liquid oxygen to the combustion chamber at 300 psi, a pressurizing system is required. A pressure-fed system using helium stored at 3000 psi and 294° K will be used.

Helium has been chosen as the pressurant because of the mass savings over other possible pressurants. The storage conditions of 3000 psi and 294° K have been verified as the best workable design. Figure 9 is a plot of system mass vs. storage pressure which shows 4000 psi to be the optimal pressure. Below and above this pressure, system mass increases; however, there is a high availability of system components built for use at 3000 psi. Due to this high availability and since the mass increase between a 3000 psi and a 4000 psi system is less than 3 kg per module, the 3000 psi system was chosen.

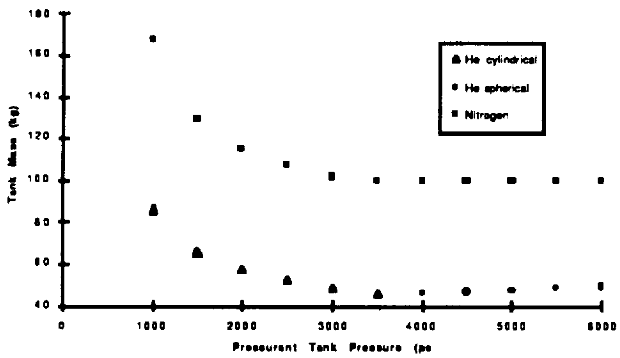

Fig. 9 Pressurant comparison

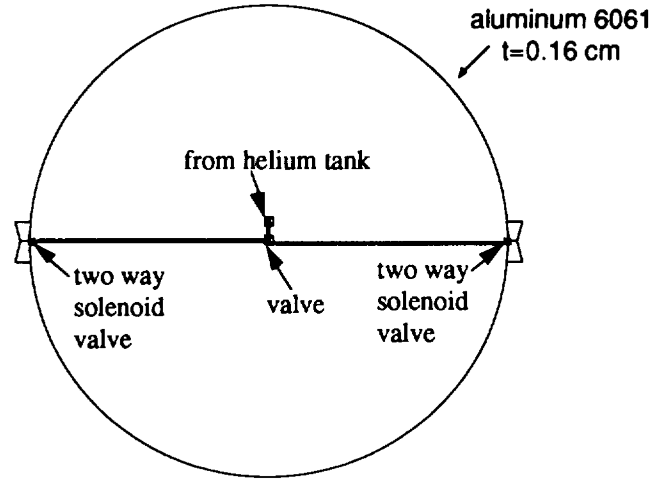

Fig. 10 Roll control system

Roll control system

The upper stage roll control system is a helium propellant cold gas propulsion system with two pairs of opposing thrusters arranged to provide a rolling moment without imparting a pitch/yaw moment. The entire system consists of the helium, piping and valves from the helium supply, four nozzles, and an aluminum circular mounting plate with mounting brackets to mount the system to the intertank structure (see Figure 10).

The amount of helium required is 1.46 kg at 3000 psi, stored in the fourth stage helium tank at 294° K. The helium supply for the roll control system will be regulated down to an operating pressure of 400 psi. Above this pressure, very little mass savings can be realized and the helium supply may not be able to provide the necessary pressure.

Standard 3/8" and 1/4" stainless steel pipe and fittings will be used to provide the necessary flow for the system. An ASCO 7985G2 two-way valve will be used to split the helium supply into two-400 psi flows which will provide the necessary mass flows for each pair of thrusters. Two ASCO 8223G3 two-way solenoid actuated valves will be used for pulsing the helium into the thrusters. The minimum duration thrust pulse for this configuration will be 35 ms. The nozzles will be 0.4 cm stainless steel and will have an area ratio (A_e/A^*) of 8.31 with $A_e = 0.679$ cm^3, $r_e = 0.4649$ cm, $A^* = 0.0817$ cm^3 and $r^* = 0.1613$ cm. The entire system will be fastened to a 0.063" Aluminum plate which will be mounted to the intertank structure between the helium and liquid oxygen tanks of the fourth stage. A small portion of the nozzles will protrude through the skin of the vehicle. Thermal and structural loads due to this have been accounted for in the design of the system.

Nozzle design

In examining the choice of materials to be used for a nozzle, two designs were evaluated for cooling the throat, a heat sink nozzle, and an ablative nozzle. Transient 1D analysis was used to determine the required thickness of the heat sink nozzles. As seen in Table 5, they tended to be very thick and heavy.

Table 5 Nozzle insert comparison

Material	Thickness (cm)	Throat insert weight (kg)
Graphite	18	130
Nickel	10	186

Ablative nozzles were found to be lighter. The composite materials selected were pyrolitic graphite for the throat and silicon-phenolic for the remaining expansion skirt. The pyrolitic graphite has an erosion rate of 0.11 mm/sec and can withstand the high temperatures at the throat; however, the cost of graphite products is five times that of the silica phenolic. Since the heating drops off drastically after two radii down the nozzle, the more economical silica-phenolic of constant thickness will be used in the last half of the expansion/ straightening section of the nozzle. Graphite is brittle at cool temperatures. To add ruggedness and durability to the nozzle, it will be wrapped with two layers of graphite-epoxy as a high strength outer shell. This graphite-epoxy shell will also be wound over a steel flange which will mount to the thrust chamber.

The converging and diverging sections of the nozzle will be made as a single piece. The mold used to make the nozzle will be separable at the throat. Once the phenolic has hardened the mold may then be pulled out of the converging and diverging sections.

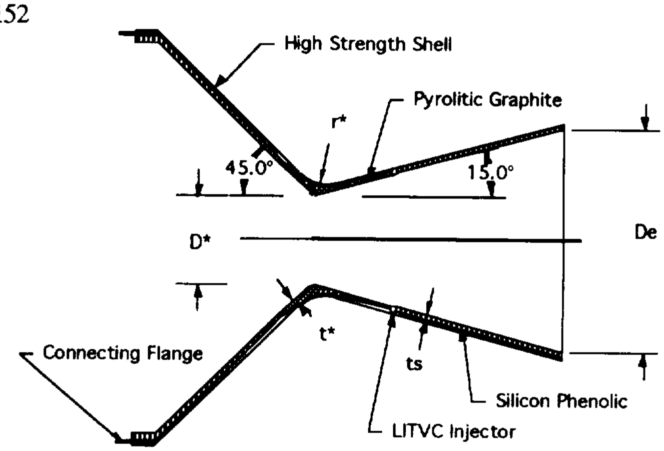

Fig. 11 Nozzle design

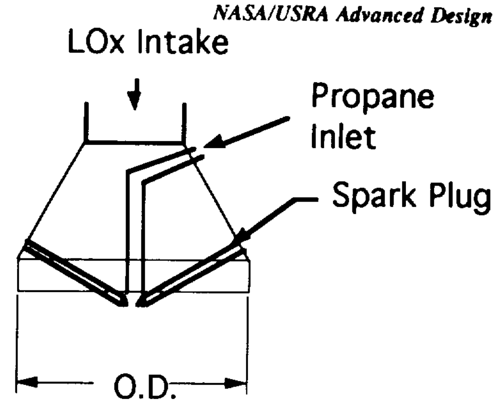

Fig. 12 Injector design

Thrust vectoring design

Liquid injection thrust vectoring (LITVC) was chosen as the thrust vectoring system. LITVC allows for a fixed nozzle which eliminates the need for a flexible bearing. The plumbing involved could be performed in-house for increased student involvement and decreased cost. LITVC utilizes the pressure from shock waves acting against the side of the nozzle and the momentum of the fluid flow to generate a torque on the vehicle.

The thrust vectoring system will consist of eight liquid injectors surrounding each nozzle. Each injector will have 300 psi with which to spray liquid oxygen into the flow. The solenoids will be digitally pulsed to allow for differential thrust vectoring. Typical performance data for oxygen is 300 sec of side Isp. Side Isp is the side force exerted by the injection of the fluid on the nozzle divided by the mass flow of the fluid. The LITVC system was designed to provide a side force equal to 7% of the axial thrust, which is equivalent to 4° of gimbaling .

Injector design

A flat plate injector was chosen to meter the LOX to the combustion chamber. The requirement on the injector is to administer the flow of oxygen to the grain in the smallest droplet size and at the slowest and most turbulent velocity possible to allow for complete combustion inside the chamber. Figure 12 shows the first stage injector.

It utilizes short tube orifices which have a drag coefficient, Cd = 0.88. Note that the ignition system is mounted internally to the injector, this is to save space in the combustion chamber. The injector is mounted flush against the top of the chamber to protect the wires of the ignition system. The thrust chambers insolation may be modified to also reduce vibration which cause chugging and loss of performance. Blocks of propellent may be placed in front of the injector which will also help to mix the flow. The plate design allows for in-house manufacturing.

Table 6 Injector specifications

Stage	1	2	3	4
Mass flow (kg/s)	38.0	12.3	10.7	1.2
Number of holes	513	139	139	27
Outer diameter (cm)	6.3	3.3	3.3	1.5
Mass (kg)	5	3	3	2

Ignition system

The final design of the ignition system is an augmented spark igniter with propane as the ignition fluid and spark plugs as the spark generator.

The propane tank will be situated in the intertank region, attached to the combustion chamber. A pipe 0.5 cm in diameter will lead from the tank into the LOX line and down through the center of the LOX injector, 1.0 cm into the combustion chamber. The exit area of the pipe will be 0.667 cm on the modules and 0.065 cm on the custom stage. Two spark plugs will be used for redundancy. The spark plugs will be located under the exit of the propane pipe. (Refer to Figure 12.)

Off-nominal performance

The off-nominal performance of a rocket deals with the variations in total impulse that occur with either changes in altitude or throat area ratio. As altitude increases, total impulse increases (very rapidly at first, then slowly levels off at a maximum of about 100,000 meters above sea-level). This information was determined from the required thrust and a pressure-altitude table. Also, the total impulse decreases exponentially with a reduction in throat area ratio. This information was determined using a complete thrust calculation over various area ratios for each of the modules based on a varying internal pressure with area. Therefore it is important to maximize thrust near the surface of the Earth and to be aware that the ablation rate on the throat is very important. The effect of the ablation rate depends on the trajectory analysis, and the ablation cannot be greater than the .00011 m/s that was chosen or the rocket will not have enough total impulse to reach orbit. If there is an irregular ablation rate, slow at the beginning but slowly increasing, it might be possible for a larger area to decrease.

Structural Design

Many factors were considered in designing a structure that would not only meet the load requirements, but also be suitable for production and assembly in the University environment. The major aspects for designing a suitable structure for MINOTAUR are outlined in Table 7.

Table 7 Structural design requirements

Structural design concern	MINOTAUR characteristics
Loading criteria (HYTRAJ)	
Axial acceleration	4.5 g's
Lateral acceleration	1 g
Axial natural frequency	30 Hz
Lateral natural frequency	15 Hz
Maximum dynamic pressure	125 kPa
Internal venting pressure	7 kPa
Production and handling	
Safety	
Cost	
Maximum student involvement	
Factors of safety	
Sealed pressure vessels	3.0
Testing on all individual parts	1.25
Proof testing of one unit	1.5
No structural testing	2.0

Intertank structure

There were two main designs that were researched for the design of the intertank structures, monocoque and skin stringer. For this launch vehicle, the skin stringer design was chosen.

Monocoque designs are less capable of withstanding the same loads as the stringer and honeycomb design, but they have the advantage of being easier to manufacture and their load analysis is much simpler. However, a problem arises with monocoque structures when cutouts for access doors, plumbing, and wiring are made. The structure that is removed to allow for the above mentioned elements constitutes a portion of the load-carrying ability of the structure. In order to make up for this loss, the rest of the structure has to be made thicker and consequently heavier.

Skin stringer structures are harder to manufacture, but they utilize the weight of the structure in terms of load carrying capability more efficiently. In this configuration, the lengthwise members are the main axial load carrying members. The primary objective of this monocoque skin is to take the torsional loads incurred on the vehicle. In this configuration, if cutouts were made in the skin, the load carrying capacity would not be appreciably affected as in the simple monocoque design since the majority of the axial loads are taken up by the stringers anyway and the stringers do not have any cutouts in them. Figure 13 shows the analysis between monocoque and skin-stringer for a given aluminum cylinder under loading as prescribed in Table 7.

Material Choice

In choosing the material for the intertank design, there were two main categories from which to pick, metals and composites. Composites have the advantage of being able to supply "tailored" strength and rigidity in the direction needed. However, in order to use composites, all loads must be known with a great degree of precision. More than likely, failure would occur if unaccounted-for-loads were to appear in an undesigned-for direction. Metals, however are isotropic, possessing the same material characteristics in all directions.

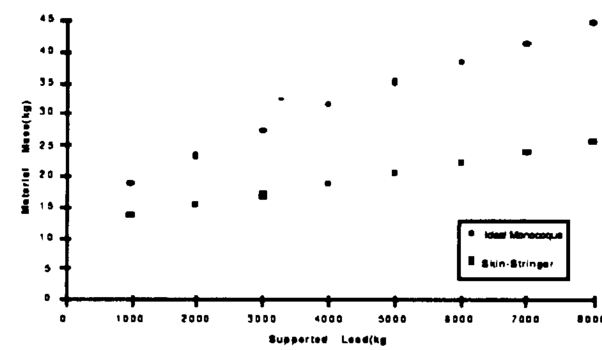

Fig. 13 Intertank structure comparison

Therefore, in designing for maximum loads in one direction, over-design in the other directions automatically occurs, thereby giving an extra degree of safety. Also, the structural analysis and the cost of manufacturing for metals are lower than those for composites. In view of these points, a metal structure was chosen. The candidate metals in these studies were Aluminum 6061(MIL-T-6), Titanium-8Al-1Mo-1V(MIL-T-9046) and Stainless Steel 17-7 PH (MIL-S-25043). The same stress levels are applied to identical structures made from these materials. The mass of structure is obtained by using the above-mentioned theories. Figure 14 shows the results from the material analysis. It is easily seen that Aluminum 6061 is the best choice, and therefore was chosen as the intertank and interstage material.

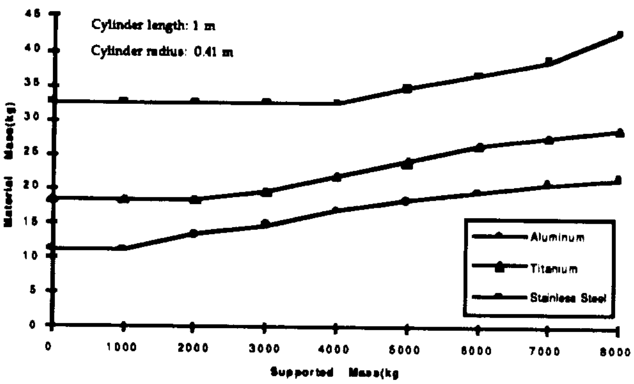

Fig. 14 Intertank material comparison

Interstage connections

A skin stringer structure was used. The buckling stresses can not be predicted using the statistical method mentioned before. The loads analysis for interstage fairings is exactly the same as that for the reinforced section of the intertank structure. Both the first and second stages will be connected by 0.0127-m diameter (0.5 inch) bolts attaching through the intertank connection stringers. The second stage will also have struts connecting the outer modules. This will alleviate torques on the stringers caused from thrust vectoring. On the first stage, there are 10 bolts on each set of stringers connected. The second stage is the same except it uses only four bolts on each set of stringers.

Structural dynamics

The response of the launch vehicle to dynamic loads during launch and pad operations are crucial factors in preparing for flight. Applied loads come from a variety of sources. Transportation, assembly, steady state acceleration from the engines, acoustic noise during transonic regime and liftoff, separation, launch pad wind, payload insertion, engine vibrations, and aerodynamic loads are possible sources of dynamic input into the launch vehicle. These loads will act over various frequency ranges at different periods of the flight.

The launch vehicle will have its own natural frequencies based on the stiffness of the structure. When the applied loads excite the natural frequencies of the vehicle, the loads on the vehicle and corresponding payload will approach extreme values. In order to avoid this resonance phenomena, the design of the vehicle should take the dynamic inputs into account.

In order to minimize the loads that a payload will experience, the payload must be designed to have natural frequencies decoupled from the launch vehicle's natural frequencies. A coupling of responses between the launch vehicle and payload will make the problem worse.

At this stage of the design, the only factors that can be determined are the launch vehicle's natural frequencies. The dynamic loads that will be put into the vehicle during flight are special to the vehicle and can not be determined until after the first flight. Since the payload is currently a 100 kilogram lump of mass at this point, the desired payload frequency characteristics will be designed into the payload after the launch vehicle frequencies are known. Mathematical representations of the launch vehicle were developed using MSC NASTRAN as the finite element software package. These models were used to estimate the natural frequencies of the launch vehicle and one of the modules. The results are as follows:

Table 8 Structural dynamics results

FEM model	Natural frequency
Dry unpressurized module	16.54 Hz
Fueled pressurized module	7.34 Hz
Free-free vehicle in flight	10.08 Hz
Fixed vehicle on launch pad	5.66 Hz

Guidance, Navigation, and Control

The total guidance, navigation, and control system (GN&C) is made up of several elements that must work together to safely get the payload into orbit. The navigation system keeps track of the vehicle's position, attitude, and velocity to give feedback to guidance system. A strapdown inertial navigation system (INS) with an embedded global positioning system (GPS) receiver provides navigation information to the GN&C system (see Table 9). The guidance system uses the navigation information along with the vehicle's equations of motion to make steering commands that will keep the vehicle on its nominal trajectory. The guidance computer carries out all of the guidance functions. The control system carries out the steering commands from the guidance computer. The main control actuator for the vehicle is the liquid injection thrust vector control system (LITVC). Cold gas thrusters (CGT) control roll in the top two stages of the vehicle. Both LITV and CGT are controlled by series of on/off valves connected to controller boards. The controller boards open the proper valves as directed by the guidance computer.

The primary duty of MINOTAUR's GN&C system is to get the payload safely into the desired orbit. The low altitude orbit of 200 Km places very tight requirements on the final burnout conditions. The system must also limit the aerodynamic loadings on the vehicle's structure while the vehicle is flying through the atmosphere. Failure of GN&C in meeting the requirements will insure loss of the vehicle and failure of the program.

Navigation Sensors

A strapdown INS was chosen as the navigation sensor because it can be used as an attitude sensor as well as a position and velocity sensor. The INS selected is the H764-C3 GPS/INS built by Honeywell Inc. Space Systems Group. The system uses a GPS receiver to lower the overall system position and velocity errors. The system is based around Honeywell's GG1320 ring laser gyros.

Table 9 Navigation System Specifications

Integrated GPS/INS	Performance
Position	16 m
Velocity	0.03 m/s rms
Pitch & Roll	0.01 Degrees rms
Yaw	0.02 Degrees rms
Thermal Operating Range:	-54°C to 55°C passive cooling
Power Requirements:	65 W at 28 Vdc
Weight:	9.1 kg

Communications

The communications system includes consideration for ground-based tracking, downlink of telemetry, and uplink of commands.

The first requirement of the communications system is that commands from the ground be accepted onboard. It should be noted that there are no requirements for the spacecraft to accept commands post-launch (except for the destruct system which is separate). The spacecraft will be autonomous after launch. A subordinate requirement is that it must be possible to identify what commands have been received onboard the spacecraft.

The second communications requirement is that telemetry is downlinked so that in the event that there is a catastrophic launch failure, it will be possible to identify the cause of the failure. The communications system must be capable of communicating with Wallop's Island (WFF) or Bermuda (BDA) regardless of spacecraft attitude if it is within line-of-sight of the station.

The third requirement is that sufficient tracking must be collected such that the spacecraft can be reacquired on the second pass of WFF. Tracking data will be collected by WFF and BDA. Ranging data from BDA will be available for 115 to 144 seconds after fourth stage burnout, which is sufficient to generate acquisition data for WFF 80 minutes later.

The fourth requirement is that the system must be compatible with the equipment at WFF. (BDA equipment is equivalent so that compatibility with WFF implies compatibility with BDA).

Commands to the spacecraft will be via an umbilical that disconnects at launch. Details of commands depend on details of procedures required to launch; however, they include tank fill and tank drain and launch commands that cannot be executed while personnel are within the launch area. Critical commands require one command to load the command and one command to execute the command after personnel have verified the command. In the interest of safety (a high mission objective), most, if not all, commands should be denoted as critical.

The downlink from the spacecraft will be via a NASA standard S-Band transponder, using 3 dipole antennas, a solid state amplifier, a data rate of 10 kbps, and will be frequency shift key (FSK) modulated. The use of a NASA standard transponder gives compatibility with WFF. Space-ground link system (SGLS) and Tracking and Data Relay Satellite System (TDRSS) S-Band antennas are not compatible with WFF and so cannot be used in communicating with WFF. The use of an omni antenna is to meet the requirement of spacecraft-to-ground communication regardless of spacecraft attitude. Clearly, this link is at its most important when the attitude is not as expected. The dipole antennas mount to the skin of the fourth stage, under a skin blemish which is integral with the antenna. The use of a solid state amplifier over Travelling Wave Tube Amplifier (TWTA) is recommended by Wertz and Larson[3] (and others) at S-Band frequencies for rf power outputs below 30W. As analysis below shows, MINOTAUR will operate at a fraction of that power level. A data rate of 10 kbps is the current estimate for data volume requirements; however, D. Loveless recommended doubling the data rate to allow for growth and data encoding. In the link margin calculations, data rates of 10 and 20 kbps are shown. The use of fsk modulation is based on its simplicity and the fact that it is not susceptible to phase disturbances. Its chief disadvantage relative to other encoding methods is its poor use of the spectrum, but that is unimportant due to the low data rate.

The Launch Trajectory Acquisition System (LTAS) is a form of C-Band tracking used during launch support. It uses several C-Band antennas to form a single stream of inertial positions (i.e., instead of being angles, ranges, or Doppler shifts, it is x, y, z in Greenwich Rotating (GROT) coordinates). It is available from WFF and BDA. Additionally, S-Band ranging would be possible from WFF and BDA, with the maximum usefulness coming after fourth stage burnout.

System Architecture

The computer system will serve as the "central nervous system" during MINOTAUR's testing and launch. Mission critical activities will be performed and monitored at all system levels of the rocket. The success of MINOTAUR is contingent on the development of a computer system capable of meeting all design requirements.

The design and development of the computer system began by defining the system operating requirements:

• The monitoring of equipment for purposes of collecting telemetry and ensuring the proper functioning of the equipment

• The capability of ground communication to receive commands prior to launch and downlink telemetry

• Computing guidance corrections and "instructing" the liquid injection thrust vector controllers

• Designing the system to be autonomous, thus requiring minimum ground support

• Designing for the highest achievable reliability.

These requirements were used as a baseline as well as a guideline for the development of MINOTAUR's computer system.

Computer configuration

The configuration of the system is detailed in Figure 15. At the heart of the system is the Main Flight Computer. All the time-related functions will be operated from this central unit. It will activate the stage separation systems and will update the other processors when the separation occurs. It will also update the Guidance Computer as the center of gravity shifts with time. The engine start/shut-off systems will also be controlled by the Main Flight Computer.

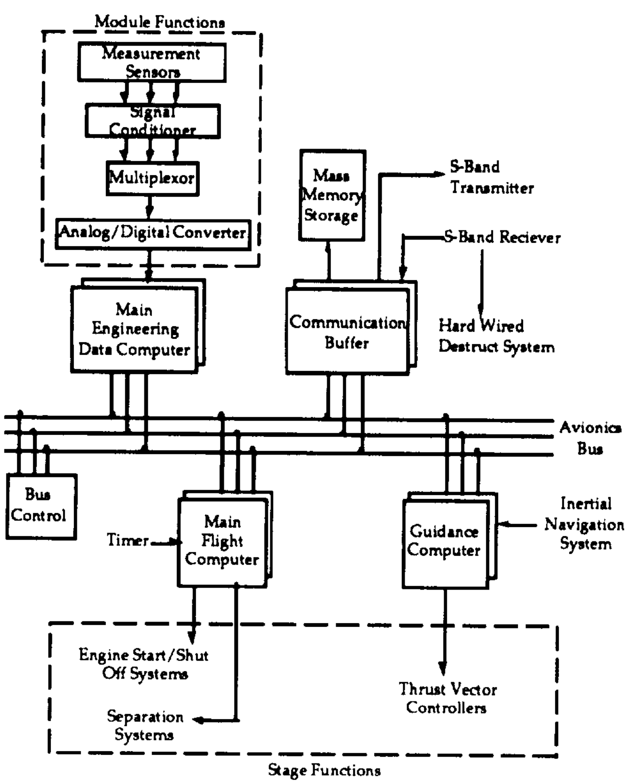

Fig. 15 Computer configuration

The Guidance Computer will receive data updates from the Inertial Navigation System (INS). It will use this data in conjunction with the updates from the Main Flight Computer to perform the guidance correction computations. These corrections will be sent to the liquid injection thrust vector controllers for processing.

The Engineering Data Computer will collect the telemetry from each module and interstage connections. These values will be "validated" against predefined acceptable margins to ensure the proper functioning of the system being monitored. The sensors being utilized include thermocouples, strain gauges, pressure sensors, and flow meters.

Hardware

Before the actual hardware items could be investigated, an estimate of the size and throughput of the computer system needed to be performed. The results are summarized in Table 10. This was based on the method outlined by James R. Wertz and Wiley J. Larson[3]. The analysis was based on a 10-Kbit telemetry data stream. The frequencies were selected based on the restrictions of the liquid injection valves. They could only operate at a rate of 10 Hz and this was used as the baseline for the estimate. This reveals an expected 340 Kwords of memory and a throughput rate of 930.4 thousand instructions per second (KIPS).

Table 10 Computer System Requirement

Component	Frequency (Hz)	Memory (K words)	Throughput (KIPS)
Command Processing	10	5.0	7.0
Telemetry Processing	1	3.5	0.3
Kinematic Integration	10	2.2	15.0
Error Determination	10	1.1	12.0
Thruster Control	10	1.0	6.0
Orbit Propagation	1	17.0	20.0
Complex Autonomy	10	25.0	20.0
Fault Correction	10	12.0	10.0
Application Total		66.8	90.3
Executive		5.5	74.4
Rum Time Kernel		12.0	
I/O Handlers		2.7	67.4
BIT & Diagnostics		1.1	0.5
Math Utilities		1.4	
Subtotal: COTS		12.0	
Subtotal: Non-COTS		10.7	142.3
Operating System Ttl		22.7	142.3
Total		90.8	232.6
Uncertainty Requirement		78.8	232.6
On-Orbit Spare		169.6	465.2
Computer Requirement		239.2	930.4

Electronic sensors

As one of the design requirements, the computer system is responsible for the monitoring of the equipment for purposes of collecting telemetry and ensuring the proper functioning of the systems. If a failure should occur, there must be adequate data to locate and identify the source of the failure. An important aspect of telemetry then is the placement of the sensors. Critical failure modes were identified and sensors were placed correspondingly. For improved reliability, sensors are dual redundant.

Power budget

The first requirement of the system is to calculate power to the rocket for a total of 20 minutes, starting at T minus 6 minutes. This amount of time was chosen because it allowed for pre-launch separation of ground power and a test of on-board avionics system operation as well as extra time at the end of rocket burn for telemetry broadcasting and a safety margin. Table 11 shows MINOTAUR's power requirements.

Batteries

The battery estimate was based on a power-time area integration which yielded estimates for the battery capacity required. The batteries were then rated at 70% efficiency for the purpose of providing a safety margin. Two batteries are required to avoid a possible power spike damaging the avionics. The required battery capacity was computed as 2.53 A*hrs for the avionics battery and 2.63 A*hrs for the propulsion system battery. The batteries were chosen to be sealed lead-acid batteries for three main reasons: (1) Nickel-Cadmium (Ni-Cd) batteries have hysterisis problems and this limits the amount of discharge a battery can take – the lead acid batteries do not have the problem to this degree; (2) the Ni-Cd batteries as a rule cost about four to six times more than the lead acid batteries; and (3) excluding specially manufactured batteries, Ni-Cds use approximately eight times the number of cells, complicating the type and amount of electrical connections. The batteries finally chosen (from among the sealed lead-acid companies on the basis of cost) were PowerSonic batteries. The avionics battery consists of two 12-V and one 6-V cells rated at 3.0A*hrs. The propulsion battery consists of two 12-V cells rated at 3.0A*hrs.

Table 11 Component power requirements

Component	Power (W)	Voltage(V)
INS	68	28
Communications	12	30
CPU	60	6
Sensors	20	9
LITVC (96 @ 10W)	960	24
Intertank valves (28 @ 30W)	840	6
Roll Control (4 @ 10W)	40	24
Igniter tanks (14 @ 10W)	140	24
Igniter plugs (28 @ 25 W)	700	12
Total	2840	
Peak Power @ launch		
Avionics	160	
LITVC(28 @ 10 W)	280	
Intertank valves (14 @ 30 W)	420	
Igniter tanks (7 @ 10 W)	70	
Igniter plugs (14 @ 25 W)	350	
Total	1280	

Testing

The primary launch requirements of this program are to have a suborbital launch in August 1993 followed by the integrated orbital launch in August 1995. To get to the orbital launch, a logical series of launches has been planned to test different aspects of MINOTAUR's design:

Late summer '93 Fourth Stage Sounding Rocket
- Rail-launched at 80 deg and 3.5 g's
- (Wallop's requirement)
- Fin stabilized
- $20K

Winter '93-94 Module Sounding Rocket
- Rail-launched as above
- Thrust vector control (TVC)
- System technology demonstration
- $200K

Summer '94 Top Stage & Module Sounding Rocket
- Vertical launch
- Separation system
- Flight termination system
- TVC
- System technology demonstration
- $340K

Winter '94-95 Top Stage & 3 Module Sounding Rocket
- Ignition of multiple modules
- TVC on multiple modules
- 300 kg payload capability
- Vertical launch
- Separation system
- System technology demonstration
- $390K

Summer '95 Orbital vehicle
- $1M

All of these launches will occur at the existing launch facilities at Wallops Island, Virginia. All production and testing are to be completed at existing and planned campus facilities. After further investigation of the testing requirements of this program, it was concluded that some aspects of testing could not be performed on campus (i.e. static test firings) due to the lack of campus facilities.

University Facilities

Part of the problem of designing a launch vehicle to be constructed at a university location is the lack of available space. This is particularly true at the University of Maryland. Space is limited, but the design requirement to provide maximum student involvement dictates the need for producing a good portion of the launch vehicle on the campus.

Table 12 Production Facilities

Facility	Work to be done
Space Systems Lab Neutral Buoyancy Facility	electronics and milling
Aerospace Laboratory	system testing and construction
Glenn L. Martin Wind Tunnel	aerodynamics testing
Vibrations Lab	vibration testing
Composite Research Lab	fabrication of He tanks
Manufacturing Building	system testing and checkout

Even with these buildings, the size of the modules restricts the available to a greater extent. During production, facilities management will become a crucial factor. The assumption made in this proposal for the use of campus space in the future is based on the notion that the MINOTAUR program will receive all the facility space that is requested.

The requirement to maximize student involvement in the design, production, and operations of the MINOTAUR launch vehicle is a key factor in the decision to use university facilities. Cost is probably even more important. A result of the diversity of the university is the wide range of assets at the design team's disposal. Items that are available include fabrication tools, assembly space, testing space, transportation, computer facilities, and the experience at hand in the professors of the university. Other items which are covered by using campus facilities include reliability, safety, and minimization of development time.

Launch facilities

The primary requirement for the Wallops Island launch site is to provide a launch pad and use of existing service structure, tracking and data acquisition equipment, and safety and emergency equipment. The primary requirements for the University of Maryland at Wallops are to provide assembly and transportation equipment, all launch vehicle communications and control equipment, and all propellant fueling equipment. All necessary personnel for vehicle assembly and integration will be provided by the University of Maryland.

Conclusion

As the complete report explains, with proper, though not excessive, funding, the University of Maryland's Aerospace Engineering Department has the capability to design and launch a low-cost launch vehicle. This launch vehicle will be completely designed and built by the university's students. Outside testing and manufacturing will be kept to a minimum to keep total costs low.

Acknowledgments

The primary editor of this report was T. Rice. The co-editor was W. Vincent. MINOTAUR's preliminary design team consisted of the following nineteen people: P. Adams, L. Boyce, K. Dawson, P. Freeman, L. Gieseker, V. Gowda, A. Jarrah, P. Margoiles, M. Miller, S. Nassau, M. Priztlaff, D. Rabine, T. Rice, A. Shank, R. Snyder, W. Vincent, T. Willard, T. Wilson, and P. Wood. The design team would like to thank Dr. Dave Akin and Dr. Mark Lewis for their guidance and wisdom throughout the entire semester. Thanks are also sent to the USRA for its continued support of this program.

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