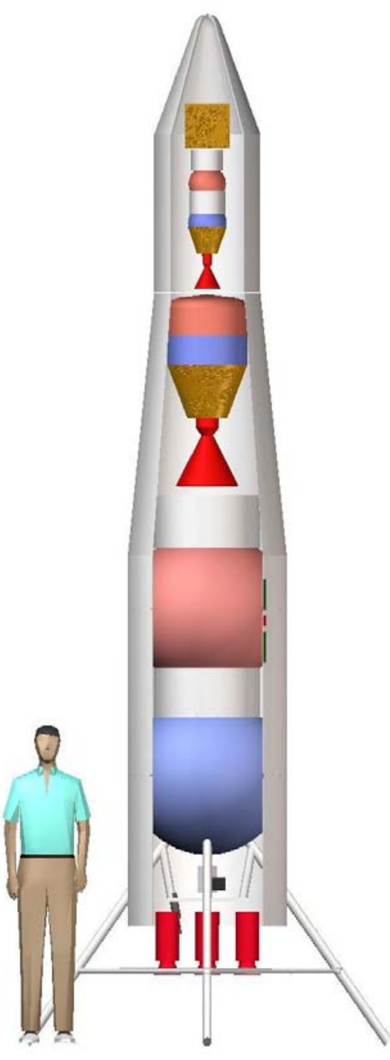
ASABOOSTER CD005

Conceptual Design Study for an Asaspace Launch Capability
Version 0.04



by Ed LeBouthillier

Forward

In a previous conceptual design study, Asabooster CD004, I examined a vertical takeoff horizontal landing design for its performance and weight. In this study, I examine a vertical takeoff vertical landing rocket to understand the same issues.

This design starts with one developed by Armadillo Aerospace. However, in order to meet the performance requirements, an aeroshell was added. This design has several interesting aspects worth consideration.

Sincerely,
Ed LeBouthillier

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GENERAL

Asabooster CD005 is an examination of a Vertical Takeoff Vertical Landing (VTVL) rocket design for a reusable rocket vehicle able to lift two upper stages weighing a total of 163 lbs on a trajectory suitable for a 115 mile altitude orbit. The expected payload delivery to orbit is about 5 pounds.

This design is very simple consisting merely of tanks, an engine, landing legs, and the bare minimum necessities to hold the components together and provide the required functionality. An aeroshell is added to reduce the wind resistance and enhance the performance and altitude capabilities.

The key benefits of this design are:

- Reusability
- VTVL flight profile
- Return to Launch Site
- Self-pressurized propane with liquid oxygen as propellants
- Low pressure main engine

In its intended application, this vehicle would be used as a first-stage booster coupled to upper stages which could deliver a payload to orbit.

This report only examines the flight vehicle design in order to aid in establishing development and operational complexity and cost. Ground support systems are not included.

Asabooster CD005 Design Parameters

ASABOOSTER CD005		
Oxidizer	Lox	
Fuel	Propane	
Payload	163.393	lbs
OF Ratio	2.200	
Oxidizer Density	71.230	lbs/cuft
Fuel Density	36.330	lbs/cuft
Avg Density	60.324	lbs/cuft
Propellant Isp	240.000	Seconds
Desired DeltaV	11250.000	FPS
Body:Fuel Mass	0.200	
Mf/Me Ratio	4.288	
Propellant Mass	1568.580	lbs
Oxidizer Mass	1078.399	lbs
Fuel Mass	490.181	lbs
Oxidizer Volume	15.140	cuft
Fuel Volume	13.492	cuft
MT	313.716	lbs
Me	477.109	lbs
Mf	2045.689	lbs

Symbol Legend

Item	Description	Input/Output
Payload	Weight of the Upper Stages to be lifted to orbit	Input
OF Ratio	Oxidizer to Fuel Mixture Ratio	Input
Avg Density	The Average density of the propellants at the specified OF mixture ratio.	Output – calculated from OF Ratio and individual propellant densities.
Propellant Isp	Average Specific Impulse of the propellant combination.	Input
Desired DeltaV	The target delta velocity according to the rocket equation.	Input
Body:Fuel Mass	The ratio of the body weight to the total propellant weight.	Input
MT	The empty weight of the vehicle without a payload or the propellant	Output
Me	The weight of the vehicle with a payload but without propellant	Output
Mf	The weight of the vehicle with payload and propellant	Output

DESIGN JUSTIFICATIONS

MISSION

The mission this vehicle is designed for is to be the first stage booster of a multistage vehicle able to put 5 pounds into orbit. To accomplish this, the booster lifts two upper stages of up to 163 pounds up to an altitude and velocity suitable for a 115 mile orbit. It then returns to the launch site to allow future reuse.

REUSIBILITY

Reusability is seen to be an enabler for lower cost operations. In order to permit it, though, this design requires sufficient design margins of the propellant to allow a tail-first landing on its landing gear. Although reusability entails greater complexity in the guidance and control system, as well as reliable, restartable engines, reusability can bring cost reductions which are seen as worth attaining.

VTVL FLIGHT PATH

A vertical takeoff vertical landing (VTVL) approach is selected to simplify launch and recovery operations. The takeoff and landing pads for a VTVL vehicle are as minimal of any design. This is contrasted to VTHL vehicles which require a runway for recovery. A vehicle of this size can takeoff or land from a cement pad as small as a few feet across.

RETURN TO LAUNCH SITE

Another significant design approach is to have the vehicle land at the same site from which it takes off. Only one launch/landing site need be manned and resources need be allocated to only one location. This simplifies many aspects of operations and thus is expected to lower costs.

SELF-PRESSURIZED PROPELLANT FEED SYSTEM

Self-pressurized propane and liquid oxygen are used as the propellants. This makes the rocket as simple as possible without additional pressurization devices or pumps. Because of the low feed pressures of the propellants, simple construction techniques should be applicable to the rocket engine as well.

LOW PRESSURE ENGINES

A low-pressure rocket engine allows cheap, simple and light construction. Industrial and commercial parts can be utilized widely in its construction. Because of the low stresses on components, parts can be manufactured using commonly available materials and construction techniques. Third, surprisingly good performance can be achieved.

DESIGN DETAILS

FLIGHT DESIGN FACTORS

TRAJECTORY

The preferred trajectory is one which allows the vehicle to deliver its payload and return to the launch site. This is not a fuel-optimal launch trajectory, but the benefit of returning the vehicle to the launch site is seen as having many benefits.

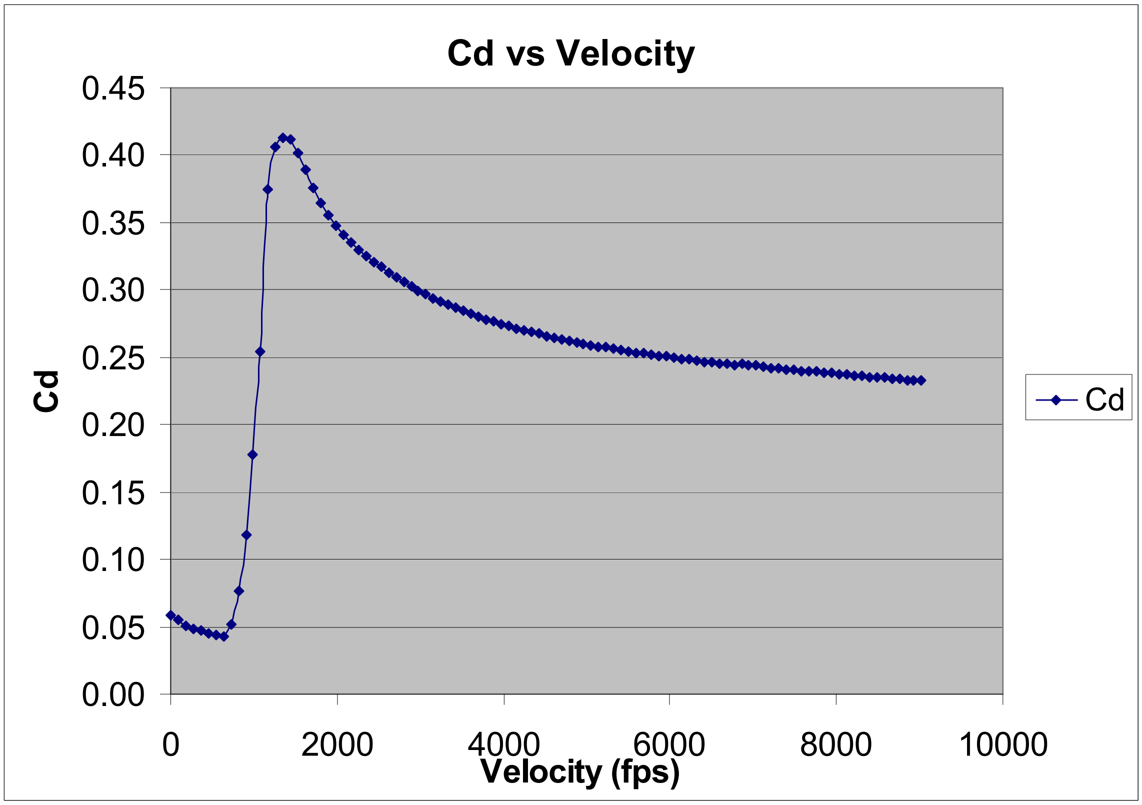
The selected trajectory is a straight-up, straight down flight. In a typical application, the vehicle will thrust vertically for a duration of about 120 seconds, attaining an altitude of 36 miles and a vertical velocity of over 5000 feet per second. This trajectory requires that a portion of the propellants be reserved for landing. The specified flight path will leave about 60 pounds of propellant for landing.

Future studies are needed to determine the actual landing fuel budget needed. This will depend on better estimates of the terminal velocity near the launch site.

COEFFICIENT OF DRAG

A vehicle without any attempt at streamlining has a large Coefficient of Drag (Cd) which makes it highly unlikely to fulfill the performance requirements (terminal altitude and velocity). Repeated searching was performed for design parameters that would result in meeting the performance goals without streamlining and this effort almost invariably failed. In a small vehicle, aerodynamic losses play a significantly larger role because the mass of the vehicle is smaller and thus decelerated faster by aerodynamic forces.

Therefore, an attempt was made to identify an aeroshell that would minimize drag while fulfilling all of the various performance requirements. As a product of this search, a design was selected which resulted in the following estimated Cd.



AERODYNAMIC LOSSES

Simulations of the vehicle flight have indicated approximate and expectable aerodynamic losses on the flight path. Whereas the rocket equation specifies a maximum delta V of 10319 fps for the ascent trajectory (accounting for landing reserve fuel), simulation results indicate a total vertical delta V of about 5155 fps at MECO. Since we already know that gravity losses are on the order of 3890 fps, the resulting value indicates aerodynamic losses on the order of 1274 fps on ascent.

GRAVITY LOSSES

The equation to determine burn duration of a rocket motor is:

Tthrust = isp * propellant_weight/thrust

Presuming a total thrust of 3000 lbs-f at takeoff the duration of thrust is:

Tthrust = 240 * 1507.5 / 3000
Tthrust = 120.6 seconds

The gravity losses, presuming a vertical flight path, are:

V = A * T

V = 32.2 * 120.6 seconds
V = 3883.32 fps

Therefore, the velocity lost due to gravity will be 3883.32 feet per second on a vertical flight trajectory, which is similar to the expected trajectory.

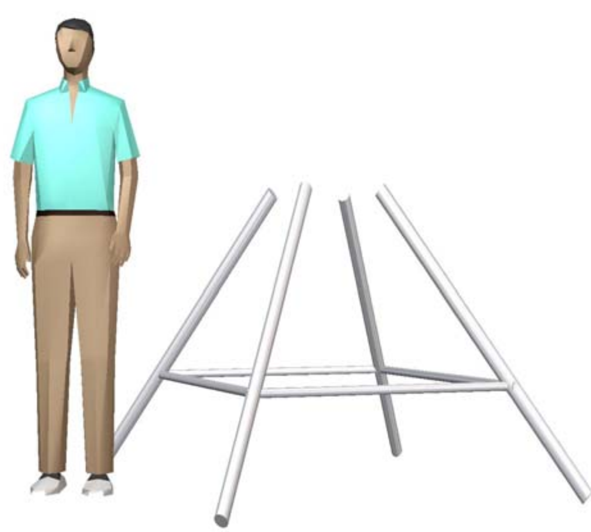
SUBSYSTEM DESCRIPTION

STRUCTURES

LANDING GEAR

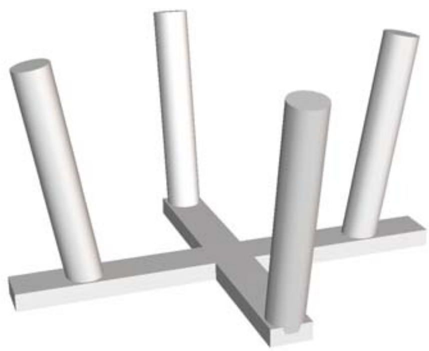
The landing gear is constructed of 2” diameter aluminum pipe which is welded into a frame with 1” diameter aluminum pipe. This is connected to the Thrust Structure to support the entire weight of the vehicle. There were three main goals for the landing gear: to provide a wide enough support base to minimize the likelihood of the vehicle toppling, to be strong enough to support the vehicle during landing and takeoff, and to be light enough to meet the weight budget. The selected landing gear design is not optimal but it should meet these goals. The primary reason for this design over the one used by Armadillo is weight savings. Their landing gear weighed upwards of 100 pounds and these are significantly less at near 15 pounds.

There is a possible problem in this design whereby the structure may not be strong enough against torque forces on the legs. It is recommended that strengthening steel cables in tension be used to strengthen against torque forces. These should provide the necessary strength without adding too much weight.



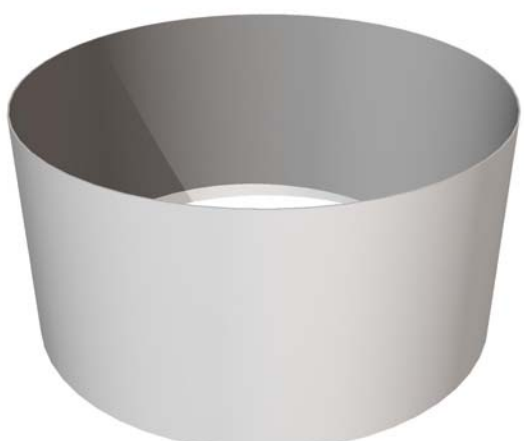
THRUST STRUCTURE

Although there are three engines in this design, only one of them is capable of being gimbaled. The two outboard engines are permanently fixed to thrust along the centerline. The Thrust Structure conveys landing gear and engine forces to the oxidizer tank structure. It is composed of a thick aluminum x-frame bolted to aluminum tubes which are welded to the oxidizer tank.



INTERTANK STRUCTURE

A simple tube structure is used to connect the two tanks together. Constructed of 0.100" aluminum sheet formed into a cylinder 12" in diameter and welded between the tanks, it conveys all mechanical forces between the two tanks. Additionally, it acts as a conduit structure for gas and liquid pipes between the propane and the lox tanks.



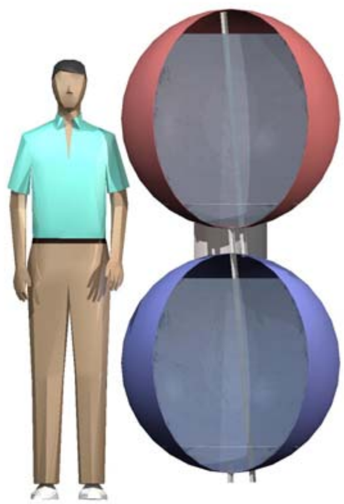
GUIDANCE CONTAINMENT STRUCTURE

A housing is placed on top of the propane tank; this provides a place to protect the electronics of the guidance system.

Structurally, it consists of 0.064" aluminum sheet formed into a cylinder which is 9 inches high and 24 inches in diameter. It is welded to the top of the propane tank. A torispherical cover of the same aluminum material is bolted on top to provide protection.

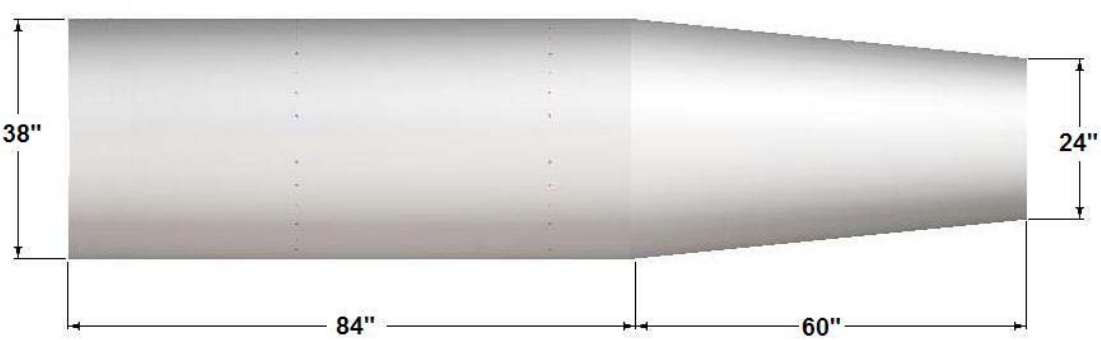
FUEL AND OXIDIZER TANKS

The fuel and oxidizer tanks are both spherical aluminum tanks with a diameter of 37 inches each with a thickness of about 0.080". Most likely they are produced from 5083-H116 aluminum or similar material. Tanks like this will be strong enough to contain the pressurant while being light enough to meet the weight goals. Internal tubing conveys gas pressure and fluids to their destinations. Internal plates are used to reduce the effects of sloshing propellants. Relief and filling valves provide the ability to safely fill and operate these tanks. Each tank is expected to weigh just a little over 33 pounds and have a volume of approximately 15.4 cubic feet. This will allow up to 1078 pounds of liquid oxygen and up to 490 pounds of propane.



AEROSHELL

The aeroshell reduces the vehicle's aerodynamic drag and thus contributes greatly to the vehicle's performance. Structurally, it consists of a cylindrical portion and a truncated cone portion. The major outer diameter is 38" and the cone meets a 24" diameter upper stage. The length of the cylindrical portion is 7 feet and the length of the truncated cone is 5 feet.



The aeroshell is expected to be construction from any one of several different composite materials. Weight savings is important since this consumes a large percentage of the weight budget.

PROPULSION

There are two propulsion systems utilized in CD005: the main propulsion system and an attitude control system.

Main Propulsion System

The main propulsion system is based on utilizing self-pressurizing lox-propane propellants. Although a relatively low pressure is obtained using this combination, it is sufficient for the purposes. It will provide many benefits for low cost construction and operation.

Three combustion chambers are envisioned to have a maximum thrust of 1000 lbs-f each with 1.72:1 nozzle expansion ratios. The sea-level specific impulse (Isp) is expected to be 212 seconds. Vacuum Isp is expected to be upwards of 260 seconds. Each will have a diameter of approximately 10 inches and a length of approximately 15 inches.

Because of the low chamber pressure, it should be possible to utilize a phenolic-lined chamber with a phenolic nozzle. At these combustion pressures, nozzle erosion rates should be about 0.001” per second. During a complete burn time, the nozzle could expect to see about 0.140” erosion. This is approximately 4% of the throat diameter and will have negligible effects on performance.

As currently envisioned, three 1000 lb-f engines are used. The outer two engines are fixed to thrust along the centerline while the center engine is gimbaled to provide pitch and yaw control. During ascent, all three engines are used; during landing, only the center engine is used.

Attitude Control Propulsion System

In a VTVL vehicle, the attitude control system is vital. During powered ascent, it provides roll control since pitch and yaw are provided by main engine gimbal action. After main engine cutoff (MECO), the attitude control system provides pitch, yaw and roll control all the way to the final landing sequence, where again, it provides only roll control.

Conceivably, with a sophisticated-enough guidance system, the vehicle could operate without roll control. But, after MECO, the attitude control propulsion system is vital for proper descent orientation.

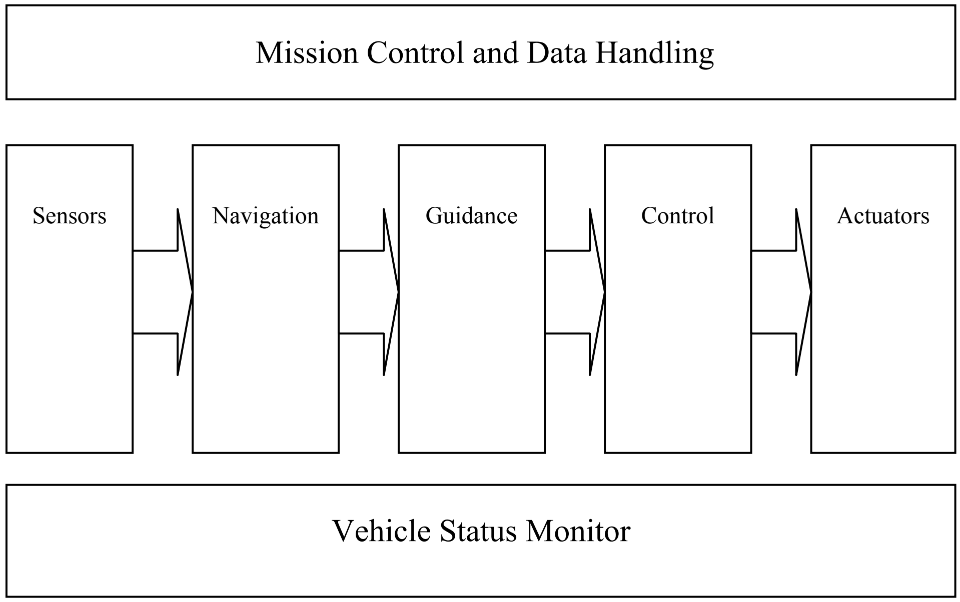
The entire flight sequence, from takeoff to landing, is expected to last upwards of 10 minutes. The following table lists the time durations for each of the flight phases.

Phase	Time (seconds)	Duration (seconds)	Required Control (R = roll, P = pitch, Y = yaw)
Ascent	0 to 121	120	R
MECO	121	0.0	RPY
Ballistic Flight	121 to 371	250	RPY
Reentry Orientation	371 to 470	99	RPY
Descent	470 to 574	104	RPY
Landing	574 to 585	11	R

GUIDANCE, NAVIGATION, CONTROL AND DATA HANDLING

Although a VTVL rocket is structurally simple in design, the guidance and navigation system is more complex. On ascent, the guidance system is functionally identical to that used in expendable rockets. It provides roll, pitch and yaw control to maintain the vehicle on the desired trajectory. During landing, however, a new guidance mode must be utilized which maintains the orientation and trajectory during descent and finally controls the vehicle to a powered landing. Logs and traces of important vehicle statuses and sensors are maintained in non-volatile memory for later analysis.

Functionally, the Guidance, Navigation, Control and Data Handling (GNC&DH) system is composed of the following sub elements:



The GNC&DH element operates on a powerful embedded processor, essentially a PC in a box, to control the vehicle to fulfill its mission. Interface cards allow it to monitor and control the various devices and conditions necessary for understanding where it is in space, where it should be and to control the actuators to cause the vehicle to fly the proper trajectory.

Mission Control and Data Handling

This function is responsible for sending sequenced goals to the GNC functions, extracting telemetry data and communicating with the ground station.

Sensors

There are numerous sensor elements required for proper and safe operation of the vehicle.

Guidance Sensors	Vehicle Status Sensors
IMU	Engine Ignition
GPS	Propellant Level/Mass
Ground Touch	Valve Feedback
Airspeed	Tank Pressure
Air Pressure	Tank Temperature
Ground Distance Rangefinder	Combustion Chamber Pressure

Navigation

The navigation function takes input from several sensor sources to estimate the vehicle latitude, longitude, altitude, roll, pitch, yaw, and time derivatives of each of these. It integrates these components with other sensor data to model the vehicle's flight parameters.

Guidance

The guidance function receives commands from the Mission Controller and parameter estimates from the navigation element and generates commands to bring the vehicle into proper orientation and trajectory.

Control

The control function receives commands from the guidance function and the Mission Controller and then generates control signals for the actuators.

Actuators

Numerous actuators work on valves and other action producing elements to realize the mission goals. The following is a list of the actuators.

Main Lox Valves (3)	Main Fuel Valves (3)
Engine 1 & 3 control Valves (2)	Engine 2 throttle valve (1)
Engine Igniters (3)	Engine 2 yaw actuator (1)
Engine 2 pitch actuator (1)	Cold Gas Attitude Actuators (8)
Payload separation mechanism actuator	

Vehicle Status Monitor

This function monitors various sensors to keep a model of vehicle status.

COMMUNICATIONS SYSTEM

A UHF radio modem provides a ground data link for telemetry and ground commands.

POWER SYSTEM

Batteries provide the electrical power sufficient for all operations.

WEIGHT ESTIMATES

WEIGHT BUDGET

The following table details the weight budget.

ASABOOSTER CD005		
Oxidizer	Lox	
Fuel	Propane	
Payload	163.393	lbs
OF Ratio	2.200	
Oxidizer Density	71.230	lbs/cuft
Fuel Density	36.330	lbs/cuft
Avg Density	60.324	lbs/cuft
Propellant Isp	240.000	Seconds
Desired DeltaV	11250.000	FPS
Body:Fuel Mass	0.200	
Mf/Me Ratio	4.288	
Propellant Mass	1568.580	lbs
Oxidizer Mass	1078.399	lbs
Fuel Mass	490.181	lbs
Oxidizer Volume	15.140	cuft
Fuel Volume	13.492	cuft
MT	313.716	lbs
Me	477.109	lbs
Mf	2045.689	lbs

The total allowable values for oxidizer weight is 1078.399 lbs (***Oxidizer Mass***), for propellant weight is 490.181 lbs (***Fuel Mass***), the weight limit of the vehicle without payload or propellant is 313.716 lbs (***MT***). The vehicle, without propellants but with payload, is to weigh no more than 477.109 lbs (***Me***). The vehicle weight with payload and propellant is to weigh no more than 2045.689 lbs (***Mf***).

WEIGHT ESTIMATES

Because of published values for some of the components, there is a high reliability in their weight estimates. The weight for other items had to be guessed using various analogous weights published by manufacturers of similar components.

System	Weight (lb)	Percentage of Total Budget
Structures	72.4	23.1%
Propulsion	65.8	21.0%
Main Propellant Handling	98.3	31.3%
Landing Gear	12.3	3.9%
GNC, actuators etc	45.0	14.3%
Total	**293.8**	**93.6%**

This budget provides about 20 pounds of currently unallocated weight. This allows for unanticipated growth of subsystems.

SUBSYSTEM WEIGHT ESTIMATES

System	Subsystem	Weight (lbs)
Structures	Sphere Joiner	3.3
	Thrust Structure	1.6
	Thrust Tee	1.7
	GNC Housing	8.4
	Aeroshell	52.4
	Universal Joints	5.0
	Total	72.4

System	Subsystem	Weight (lbs)
Propulsion	Engines (x3)	55.8
	Cold Gas Propulsion	10.0
	Total	65.8

System	Subsystem	Weight (lbs)
Main Propellant Handling	LOX Tank	33.4
	Propane Tank	33.4
	Anti-Slosh	7.9
	Fuel Delivery Pipe	1.6
	Fuel Manifold	3.0
	Lox Manifold	3.0
	Fuel Delivery Hose	3.0
	Lox Delivery Hose	3.0
	Fuel Valve	5.0
	Lox Valve	5.0
	Total	98.3

System	Subsystem	Weight (lbs)
Landing Gear	Leg 1	3.07
	Leg 2	3.07
	Leg 3	3.07
	Leg 4	3.07
	Total	12.28

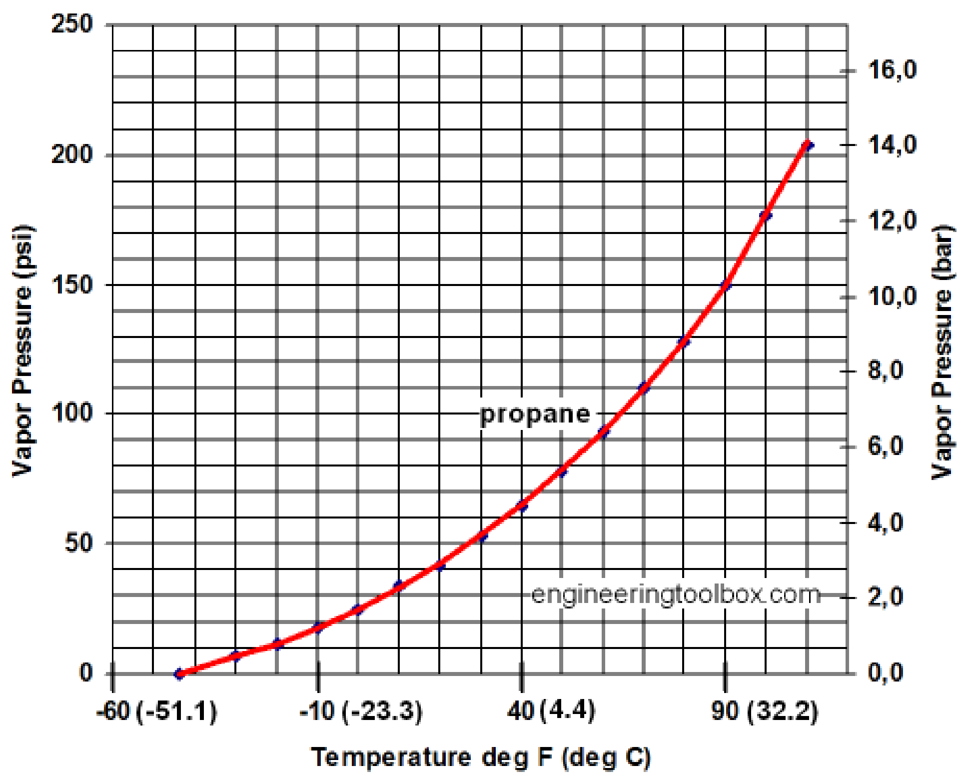
System	Subsystem	Weight (lbs)
GNC, sensors, actuators, etc	GNC Computer	5.0
	Data Radio	3.0
	GPS Receiver	3.0
	Wind Speed & Air Pressure Sensors	1.0
	Propulsion Sensors	1.0
	Lat Actuators	9.0
	Long Actuators	9.0
	Actuator Universals	3.0
	Flight Terminator	3.0
	Wiring	3.0
	Batteries	5.0
	Total	45.0

APPENDIX A – LOX-PROPANE PROPULSION

The propellant system selected for this application utilizes relatively low pressure but self-feeding liquid oxygen with propane fuel/pressurant.

PROPANE VAPOR PRESSURE CHARACTERISTICS

The following chart details the vapor pressure of propane vs temperature.



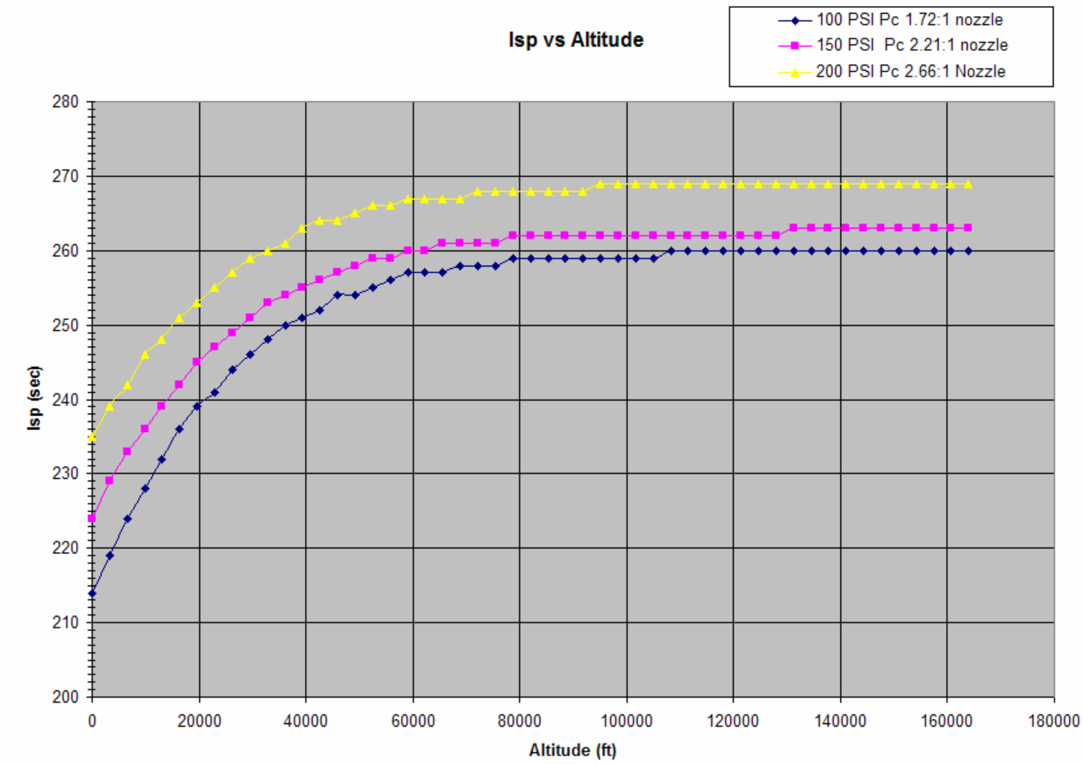
Propane Vapor Pressure vs. Temperature
Source: www.engineeringtoolbox.com

Throughout the range of warm air temperatures (e.g. 70 degrees F to 110 degrees F) the vapor pressure ranges from 110 PSI to 200 PSI. At 80 degrees F, the vapor pressure is about 125 PSI.

It should be possible to feed propane and lox utilizing propane's vapor pressure to obtain a chamber pressure of about 100 PSI. This meets an often-suggested pressure difference of about 20% between the feed system and the internal chamber pressure and thus will help minimize pressure fluctuations during combustion.

EXPECTED ENGINE PERFORMANCE

A nominal combustion chamber pressure of 100 PSI is selected. Although this is a low pressure, it will provide sufficient performance for low-cost operation. The following table lists the specific impulse of propane-lox engines at several pressures at different altitudes.



Isp vs. Altitude for a Propane-Lox engine

As the graph shows, a 100 PSI engine with a 1.72:1 nozzle has a varying Isp throughout its flight envelope. At the ground, it will be relatively low at about 212 seconds. At altitude, the Isp approaches 260 seconds. When a realistic flight trajectory is analyzed, an average Isp of about 240 seconds is observed.

APPENDIX B – STAGE 1 COMBUSTION INFORMATION

The following data is the output of PEP, a combustion estimation program. It is used to estimate combustion characteristics of LOX-PROPANE propellants fed into an engine at an oxidizer to fuel ratio of 2.2 and at a chamber pressure of 100 PSI expanding to atmospheric pressure of 14.7 PSI. This information is used to derive the engine size parameters.

```
Computing case 1
Frozen equilibrium performance evaluation

Propellant composition
Code  Name                    mol    Mass (g)  Composition
686   OXYGEN (LIQUID)         0.0688 2.2000   2O
771   PROPANE                 0.0227 1.0000   8H  3C
Density :  1.659 g/cm^3
3 different elements
O  H  C
Total mass:  3.200000 g
Enthalpy  : -1052.46 kJ/kg

114 possible gazeous species
3 possible condensed species

                       CHAMBER      THROAT        EXIT
Pressure (atm)   :       6.805       3.799       1.000
Temperature (K)  :    3139.062    2814.909    2179.173
H (kJ/kg)        :   -1052.457   -1795.023   -3215.388
U (kJ/kg)        :   -2396.770   -3000.516   -4148.626
G (kJ/kg)        :  -43327.881  -39002.158  -32563.476
S (kJ/(kg)(K)    :      13.468      13.218      13.468
M (g/mol)        :      19.415      19.415      19.415
(dLnV/dLnP)t     :    -1.00000    -1.00000    -1.00000
(dLnV/dLnT)p     :     1.00000     1.00000     1.00000
Cp (kJ/(kg)(K))  :     2.30619     2.27434     2.18746
Cv (kJ/(kg)(K))  :     1.87793     1.84609     1.75921
Cp/Cv            :     1.22804     1.23198     1.24344
Gamma            :     1.22804     1.23198     1.24344
Vson (m/s)       :  1241.53148  1218.66404  1070.54079

Ae/At            :                 1.00000     1.72260
A/dotm (m/s/atm) :               260.40706   448.57609
C* (m/s)         :              1771.96495  1771.96495
Cf               :                 0.68775     1.17377
Ivac (m/s)       :              2207.85647  2528.57059
Isp (m/s)        :              1218.66404  2079.87084
Isp/g (s)        :               124.26915   212.08780

Molar fractions

CO                    3.4126e-01  3.4126e-01  3.4126e-01
CO2                   7.1508e-02  7.1508e-02  7.1508e-02
COOH                  1.8056e-06  1.8056e-06  1.8056e-06
```

H	4.0390e-02	4.0390e-02	4.0390e-02
HCO	6.3187e-06	6.3187e-06	6.3187e-06
HO2	4.4896e-06	4.4896e-06	4.4896e-06
H2	1.9870e-01	1.9870e-01	1.9870e-01
HCHO,formaldehy	1.9357e-07	1.9357e-07	1.9357e-07
HCOOH	2.8379e-07	2.8379e-07	2.8379e-07
H2O	3.1955e-01	3.1955e-01	3.1955e-01
H2O2	7.6773e-07	7.6773e-07	7.6773e-07
O	2.9151e-03	2.9151e-03	2.9151e-03
OH	2.3830e-02	2.3830e-02	2.3830e-02
O2	1.8383e-03	1.8383e-03	1.8383e-03

APPENDIX C – ROCKET ENGINE DESIGN INFORMATION

The results of PEP are placed into a spreadsheet which results in the following table. This is used to size the rocket engine.

Density							
	LOX	71.16	#/cu ft				
	Propane	36.2	#/cu ft				
Step 1	Find Fluid Flow Rates						
	Thrust	1000	#-f				
	O/F Ratio	2.2					
	Isp	212	sec				
	Wt	4.716981	#/sec				
	Wo	3.242925	#/sec	0.335868	GPS	20.15207	GPM
	Wf	1.474057	#/sec	0.300105	GPS	18.00629	GPM
Step 2	Find Throat Temperature						
	Tc	5190	degrees Fahrenheit				
	Tc	5649.6	degrees Rankine				
	gamma	1.22804					
	Tt	5071.363	degrees Rankine				
Step 3	Find Throat Pressure						
	Pc	100	PSI				
	Pt	55.90776	PSI				
Step 4	Find Throat Area						
	Gas Molecular Weight	19.415					
	Throat Area	8.524339	square inches				
Step 5	Find Throat Diameter						
	Throat Diameter	3.29447	inches			0.274539	ft
Step 6	Find Nozzle Exit Area						
	Nozzle Expansion Ratio	1.72					
	Nozzle Exit Area	14.66186	square inches				
Step 7	Find Nozzle Exit Diameter						
	Nozzle Exit Diameter	4.320657	inches			0.360055	ft
Step 8	Find Chamber Volume						
	L*	60	inches				
	Chamber Volume	511.4604	cubic inches				
Step 9	Find Chamber Length						
	Chamber Scale	2.5	times throat diameter				
	Chamber Diameter	8.236176	inches			0.686348	
	Chamber Area	53.27712	square inches				
	Chamber Length	9.6	inches			0.8	ft

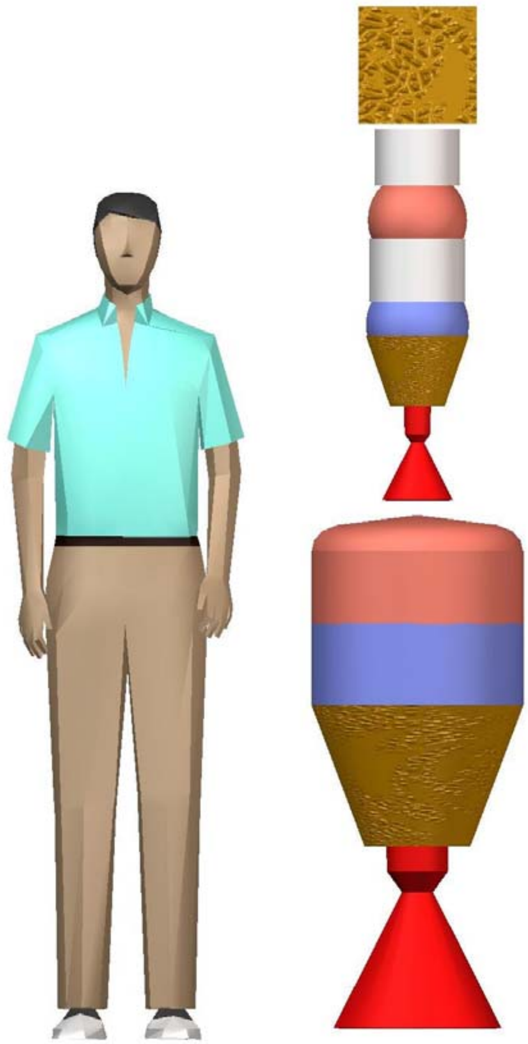
APPENDIX D – NOTIONAL UPPER STAGES

This section describes a likely second and third stage assembly which is based on similar principles to the first booster

The following table illustrates the characteristics of the second and third stages that might be utilized on Asabooster CD004 to place payloads into orbit.

Stage 3			**STAGE 2**		
Oxidizer	Lox		**Oxidizer**	Lox	
Fuel	Propane		**Fuel**	Propane	
Payload	4.600	lbs	Payload	24.201	lbs
OF Ratio	2.200		OF Ratio	2.200	
Oxidizer Density	71.230	lbs/cuft	Oxidizer Density	71.230	lbs/cuft
Fuel Density	36.330	lbs/cuft	Fuel Density	36.330	lbs/cuft
Avg Density	60.324	lbs/cuft	Avg Density	60.324	lbs/cuft
Propellant Isp	325.000	Seconds	Propellant Isp	325.000	Seconds
Engine Efficiency	1.000		Engine Efficiency	1.000	
Desired DeltaV	12750.000	FPS	Desired DeltaV	12750.000	FPS
Body:Fuel Mass	0.150		Body:Fuel Mass	0.150	
Mf/Me Ratio	3.382		Mf/Me Ratio	3.382	
Propellant Mass	17.044	lbs	Propellant Mass	89.670	lbs
Oxidizer Mass	11.718	lbs	Oxidizer Mass	61.648	lbs
Fuel Mass	5.326	lbs	Fuel Mass	28.022	lbs
Oxidizer Volume	0.165	cuft	Oxidizer Volume	0.865	cuft
Fuel Volume	0.147	cuft	Fuel Volume	0.771	cuft
MT	2.557	lbs	MT	13.451	lbs
Me	7.157	lbs	Me	37.651	lbs
Mf	24.201	lbs	Mf	127.321	lbs

Both of these stages, like the booster stage, utilize self-pressurizing propane with liquid oxygen in a low-pressure thrust chamber. With adequate nozzles (about 80:1 expansion ratio) and at altitude, the engine efficiency can be relatively high. Like the booster stage, these stages have relatively low technology requirements for their development.



Notional 2nd and 3rd Stages with payload