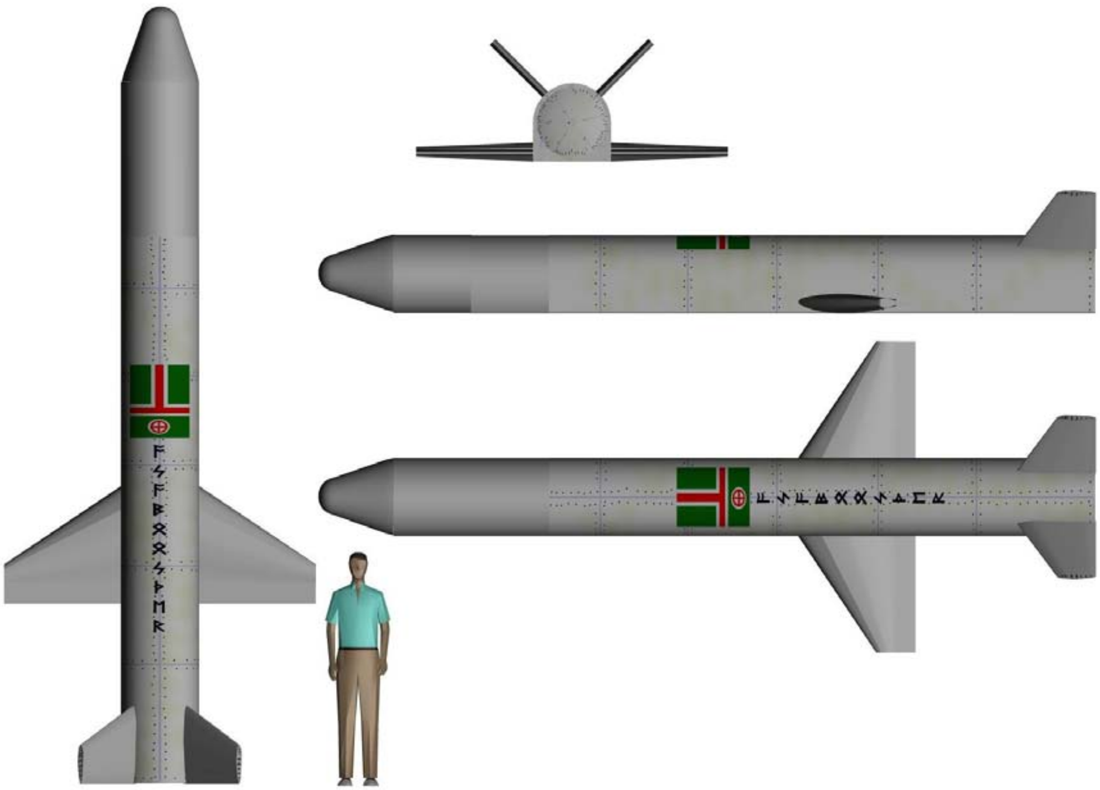
ASABOOSTER CD004

Design Study for an Asaspace Launch Capability

Version 0.13



Ed LeBouthillier

A publication in the ***Asaspace Launch Vehicle Trade Studies Series***

Forward

I am constantly trying to address the issue of "How are Asatruar going to get to space in a timely and cost-effective manner?" As part of the answer to this question, I'm constantly exploring the design space of small rockets and estimating the build complexities, costs and support requirements of different rocket designs.

This report represents one design point which suggests promising capabilities for orbital access within the budgets of amateur space enthusiasts. This is not a small project, but it is probably within the capabilities of well-funded amateurs working part-time.

This report can be accused of being unattainable dreaming. Yes, the project described is large and the costs are high. However, I think that there is benefit in exploring the realm of possible designs, and estimating their costs and complexities.

Additionally, I think that there is benefit in merely presenting a template for approaching the development of these kinds of projects to others. There is benefit in seeing the details worked out and presented in a report format. There is benefit if there is cross-checking and criticism of the ideas. It helps build and maintain an Asatru engineering culture.

There is benefit, also, in merely going through the design process, estimating schedules and costs and getting better at these skills. For, if it can be said that we don't have the skills to do these things now, it can certainly be said that persistently developing the skills surely heads us in the direction of acquiring them.

On this basis, this report is presented for archival purposes to help us learn the process of developing these rockets as well as a part of a growing database on possible designs. I plan on presenting other design points in the future from which more informed decisions of possible future project implementation can develop. Future reports will address cost estimates and project planning for preferred designs. I hope you find this report interesting and educational. I hope it also leads us towards our destiny of Folk space colonization.

Sincerely,
Ed LeBouthillier

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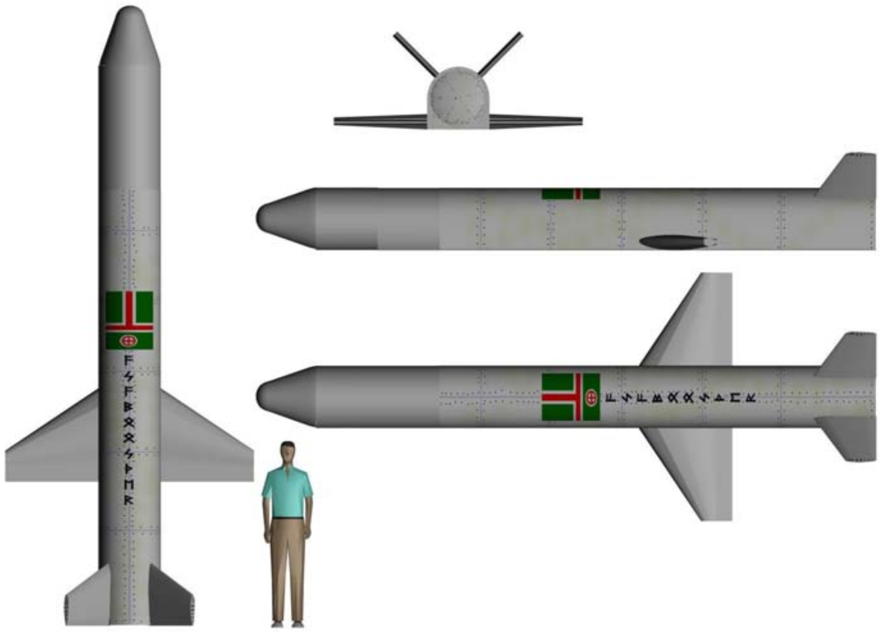
GENERAL

Asabooster CD004 is a conceptual design for a reusable rocket vehicle able to lift 152 lbs with a total delta velocity of 10000 feet per second vertically (not taking gravity and aerodynamic losses into account).



Asabooster CD004		
Oxidizer	Lox	
Fuel	Propane	
Payload	152.154	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	212.000	Seconds
Engine Efficiency	1.000	
Desired DeltaV	10000.000	FPS
Body:Fuel Mass Ratio	0.200	
Mf/Me Ratio	4.327	
Propellant Mass	1513.061	Lbs
Oxidizer Mass	1040.229	Lbs
Fuel Mass	472.832	Lbs
Oxidizer Volume	14.604	cuft
Fuel Volume	13.015	cuft
MT	302.612	Lbs
Me	454.767	Lbs
Mf	1967.828	Lbs

This design is intended to be the lowest-technology development program possible for an orbital-capable rocket. This design rests upon several key design features:

- High body mass to propellant fraction allowing a heavier, easier to build vehicle
- VTHL flight path
- Standard light airplane construction techniques.
- self-pressurized propane with liquid oxygen as propellants
- Low pressure main engines (100 PSI)
- Distributed development by amateurs

The vehicle is intended to be used as a first-stage booster in multi-stage applications leading to placement of payloads into low Earth orbit. It can additionally be clustered with other vehicles to enhance its capabilities.

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

DESIGN JUSTIFICATIONS

SYSTEM DESIGN

From a rocket design standpoint this is a relatively heavy rocket booster. The body to propellant mass fraction is fairly high at about 2:10. This allows a lot of body weight and low-tech construction techniques. Yet an appreciable performance goal of 10,000 feet per second delta velocity was specified so that it could be a useful booster for a fairly sizeable payload.

Reusability is seen by many people as an enabler for low-cost development and operation of rocket vehicles. There are several advantages to this approach. First, system integration can be performed incrementally on a vehicle through flight envelope expansion. This means that an orbital launch attempt can be performed with a vehicle which has demonstrated flightworthiness. Second, the cost to build a vehicle can be amortized over several flights lowering development and operational costs.

VTHL FLIGHT PATH

A VTHL (Vertical Takeoff Horizontal Landing) flight path was chosen over other approaches. This allows the construction of a reusable vehicle that can be developed and tested at different scales and partial flight regimes to validate the system and its subcomponents.

Because of the VTHL flight characteristics, the reusability aspect of the vehicle can be developed independently of its rocket capabilities. For example, the Landing system can be developed without a working propulsion system by airdropping from any number of different lifting approaches (balloons, winged aircraft, rockets, helicopters). In the worst case, these vehicles are only UAV's (unmanned air vehicle) and fall under different FAA guidelines than high-altitude rocket launches.

Useful design work can be done on smaller-scale vehicles. The guidance system components can be developed and validated on smaller-scale vehicles in realistic ways. Problems encountered and solved on smaller scale vehicles will save a lot of headache before trying to do full-scale orbital launches. Aerodynamic characteristics can be identified on smaller-scale vehicles and then scaled up in realistic ways. Work in wind tunnels and scale drop flights can be applied in meaningful ways to a full-scale vehicle.

The availability of large aerodynamic surfaces could be instrumental in this system being able to survive problems. On systems which rely solely on active propulsion systems to land, an engine failure is catastrophic. Engine failures are likely failure modes in under-funded development programs. Winged vehicle systems can be more forgiving and have a long history of surviving many problems.

CONSTRUCTION TECHNIQUES

Because of the specified mass fraction, standard light aircraft construction techniques of riveted monocoque aluminum are viewed as likely realizations of this design. This construction technique was selected in contrast to composite techniques because of the long history it has with small aircraft manufacturers. This makes the construction techniques much more accessible to amateur builders. Additionally, the inherent thermal mass will likely survive unscathed to the expected low levels of aerothermodynamic heating of reentry.

PROPELLANT FEED SYSTEM

The propellant feed system is based on utilizing propane's vapor pressure to feed both the propane and the liquid oxygen into the rocket combustion chamber. Because of propane's low vapor pressure of about 125 PSI at room temperature, it should be possible to use low-tech welded aluminum tanks and readily available pressure lines for carrying the propellants.

LOW PRESSURE ENGINES

Because of the low feed pressures of the propellants, simple construction techniques should be applicable to the rocket engines as well. It is envisioned that three 1000 pound thrust engines will be utilized. This will allow incremental shutdown of the engines to keep ascent acceleration at reasonable levels.

DISTRIBUTED AMATEUR DEVELOPMENT

Due to the nature of this design, it is possible for amateur rocket enthusiasts to develop aspects of the design in a distributed manner. Aerodynamic characteristics of the airframe can be detailed in isolated flight tests of scale vehicles and isolated wind tunnel testing. The landing system can be developed as either a group or individual project through dispersed launches of un-powered scale vehicles. The guidance system also can be developed through isolated scale rocket vehicles.

DESIGN DETAILS

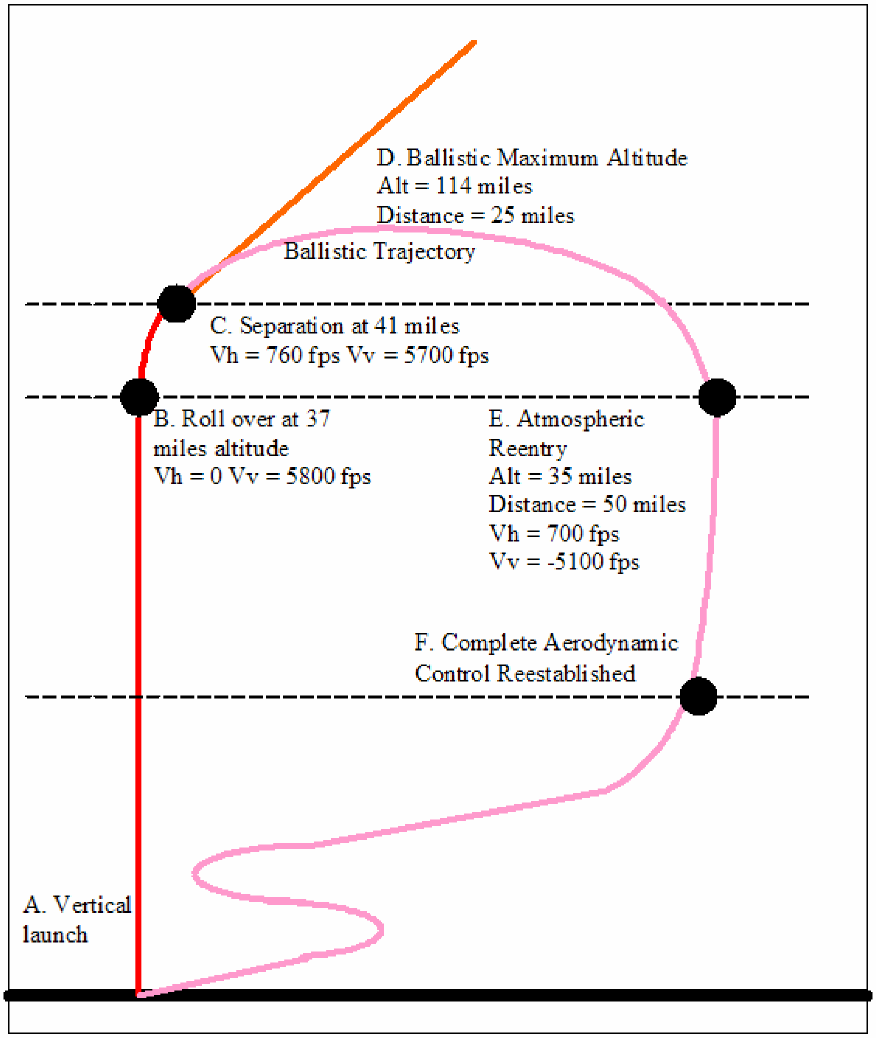
FLIGHT DESIGN FACTORS

TRAJECTORY

The preferred trajectory is one which allows the vehicle to deliver an orbital payload and return to the launch site in an un-powered glide. This is not a fuel-optimal ascent trajectory but it has several benefits. First, one site can be used for both booster launch and recovery. Second, because of the near-vertical flight path, and because of the additional velocity that the booster can impart, orbital upper stages can be launched into either eastward or westward orbital trajectories. Finally, the reduced ground coverage by the booster stage means that it is less likely to go over populated areas. This increases the launch site opportunities.

The booster trajectory design must be considered in relation to possible upper stage capabilities. For this study, the presumed delta Velocity (deltaV) capability of the payload stages is a total of 25000 fps. This means that the booster stage must impart approximately 500 feet per second horizontal velocity.

If we presume a completely vertical ascent trajectory up until a last flight phase during which a roll-over is performed and horizontal velocity is imparted to the payload stages, then the trajectory can be seen to have several phases. In the first phase, purely vertical velocity is imparted. In the second phase, the vehicle rolls over to the horizontal (taking a certain period of time for the roll). In the third phase, as much velocity as necessary is put into the horizontal direction. In the fourth phase, the payload is released and the booster follows a ballistic trajectory leading to atmospheric reentry. In this phase, the vehicle controls via cold-gas thrusters until sufficient control by the aerodynamic surfaces is re-established. In the fifth phase, the aerodynamic surfaces are used to put the vehicle on a glide path back to the launch site. In the seventh stage, the vehicle uses the aerodynamic surfaces to perform an automated or remotely controlled landing.



Worst Case Flight Envelope

Horizontal Velocity Flight Time Estimation

The rocket equation can be used to estimate the amount of propellant required to provide up to 500 fps deltaV horizontally. In the following equation, the Isp of 218 is the expected value at separation altitude.

dV = g Isp ln(Mf/Me)

Mf = Me * e ^ (dV/(g*Isp))

Mf = 454.767 * e ^ (500/(32.2* 218))
Mf = 454.767 * 1.074
Mf = 488.420

Propellant = Mf – Me = 488.420 – 454.767

Propellant = 33.653 lbs

The amount of time to burn that much propellant is:

Tthrust = 218 * 33.653/3000
Tthrust = 2.446 seconds

This means that the vehicle will thrust vertically for 101.8 seconds and then roll over and thrust horizontally for 2.5 seconds.

A simulation of this trajectory has been run to understand the velocity vector at Main Engine Cutoff (MECO). This simulation derived the following trajectory parameters. These represent worst-case values because it's likely that more efficient ascent thrust programs will result in fuel savings that could allow the horizontal velocity to be counteracted after separation, thus resulting in a ballistic trajectory which ends closer to the launch point. These worst case values are:

Flight Phase	X (miles)	Y (miles)	dX (fps)	dY (fps)
Launch	0	0	0	0
Rollover	0	37	0	5800
MECO/Separation	¼	41	760	5700
Atmospheric Reentry	50	35	700	-5100
Flyback Start	50	20	Unknown	Unknown
Landing	0	0	0	0

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 = 212 * 1513.061 / 3000
Tthrust = 107 seconds

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

V = A * T
V = 32.2 * 107 seconds
V = 3445.4 fps

Therefore, the velocity lost due to gravity will be 3445 feet per second on a vertical flight trajectory, which is similar to the expected trajectory.

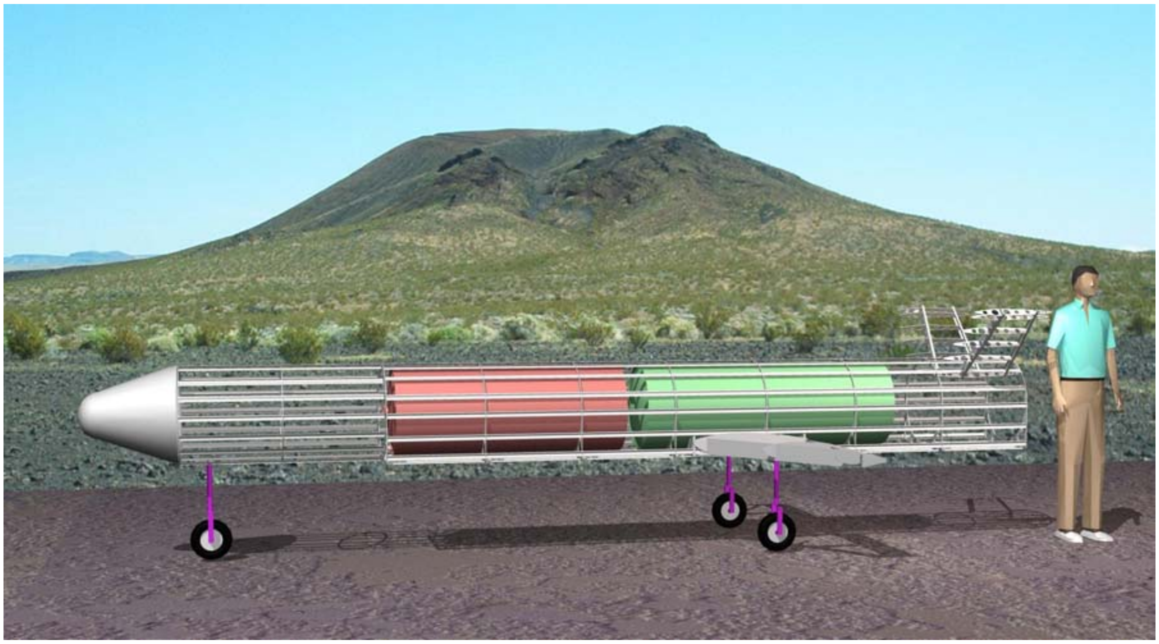
AERODYNAMIC LOSSES

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 10000 fps, simulation results indicate a total vertical delta V of about 6000 fps. Since we already know that gravity losses are on the order of 3500 fps, the resulting loss indicates aerodynamic losses on the order of 500 fps.

SUBSYSTEMS DESCRIPTION

STRUCTURES

The basic structure is a cylindrical flight vehicle of 24 inches in diameter and 20 feet in length. The building technique envisioned for the vehicle's structure is based on standard aircraft riveted aluminum monocoque construction. This is a relatively cheap construction technique requiring few specialized tools. It has been demonstrated as being within the capability for amateurs to build aircraft in the many home-built aircraft built over several decades.



EMPENNAGE

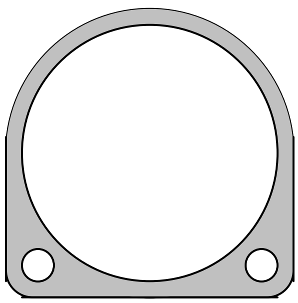
A V-Tail empennage was selected to reduce weight and parts complexity while providing sufficient yaw, pitch and roll control.

WINGS

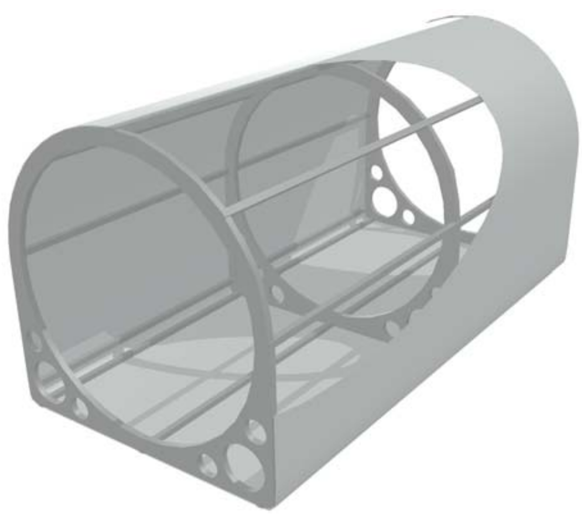
Two clipped delta wings are situated at the center of gravity. They also contain doors and mechanisms for the retractable landing gear. They have a total lifting surface area of about 12 to 15 square feet, producing a wing loading of approximately 25 to 20 pounds per square foot during landing. This is a relatively low wing loading which should support slower landing speeds. A wing spar assembly binds the two wings to the fuselage.

FUSELAGE

The fuselage is a simple shape based on a square-bottom cylinder for holding the empennage, containing the main propulsion engines and their propellant tanks and mounting the wings. The dimensions of this structure are 24” x 24”. It is very similar to shapes built for many small light aircraft and is, thus, a low risk design. Skin stringers are used to strengthen the fuselage as necessary. Rivets hold everything together.

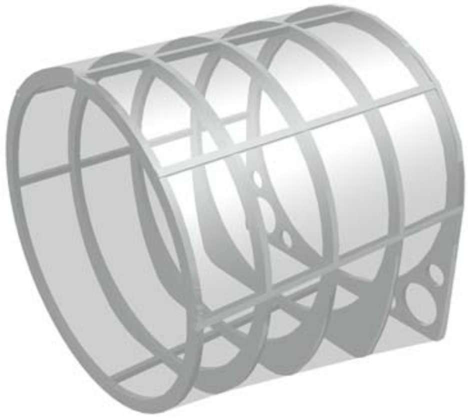


Fuselage frame general cross-section



Basic Fuselage Construction

A transition section is used to merge the squared-off cylinder to the forward landing gear section’s purely cylindrical shape.



Transition Section cross-sections

Finally, a spun aluminum nosecone containing the attitude control nozzles is ahead of the forward landing gear section.

MAIN PROPULSION PROPELLANT TANKS

The main propulsion propellant tanks specified in this design are expected to be cylinders which are 22 inches in diameter and approximately 5 feet long each. Their designed maximum pressure is specified as being 187.5 PSI (125 PSI * 1.5 safety factor). The equation specifying the hoop stress experienced by cylinders is:

Stress = Pressure * Radius / Thickness

Therefore, the stress that the tank construction material must survive is:

Stress = 187.5 * 11 / Thickness
Stress = 2062.5 / Thickness

Oxidizer and Fuel Tank Material Trades

Material	Thickness	Tank Weight	Weight Budget	Hoop Stress	Heat Treated Strength	Welded Strength
Aluminum 6061	0.150”	60 lbs	40 lbs	13750 PSI	40000 PSI	14000 PSI
Aluminum 6061	0.100”	40 lbs	40 lbs	20625 PSI	40000 PSI	14000 PSI
Aluminum 5083-H116	0.100”	40 lbs	40 lbs	20625 PSI	39000 PSI	24000 PSI
Aluminum 5083-H116	0.090	36 lbs	40 lbs	22916 PSI	39000 PSI	24000 PSI
Aluminum 5083-H343	0.064”	26 lbs	40 lbs	32227 PSI	39000 PSI	38000 PSI
Aluminum 5083-H343	0.055”	23 lbs	40 lbs	37500 PSI	39000 PSI	38000 PSI

This table indicates that some aluminum alloys can maintain sufficient strength to be usable in this application after welding. Whether it is cheaper to use some of these other aluminum alloys or to heat treat more common alloys is currently unknown. Additionally, it might be preferable to utilize friction stir welding, which welds aluminum with less loss of strength, while using more commonly available and cheaper aluminum alloys. Future trade studies can indicate a preferable path.

In summary, there is a technical risk associated with production of light strong tanks suitable for use with propane and liquid oxygen. It is not a great risk, but there will need to be some research to identify a suitable cost effective solution. Details presented here indicate that the problem is not insurmountable.

THRUST STRUCTURES

Large dynamic stresses can be expected due to the vertical forces of the engines and the weight of full propellant tanks while accelerating the vehicle during ascent. In order to minimize ascent thrust stresses being applied to the fuselage, two thrust structures convey thrust forces from the engines directly to the tanks rather than to the skin of the vehicle. The lower thrust structure attaches the engines to the bottom of the oxidizer tank. The upper thrust structure attaches the bottom of the propane tank to the top of the oxidizer tank.

ATTITUDE CONTROL PROPELLANT TANKS

A small commercially-available carbon fiber wound tank will be used to contain the gas for the cold-gas attitude control system. This will be a fairly small tank which will fit into the nose of the vehicle with the cold gas valves and nozzles.

LANDING SYSTEM

Extendable tripod landing gear with small wheels constitutes the landing system. If it makes sense, these can probably be released by spring force because they will only be used for landing and can be manually retracted during launch preparation.

PROPULSION

The Propulsion System consists of two independent thruster systems: the main propulsion and the attitude control system. The main propulsion system is used to impart the desired trajectory velocity and the attitude control system is used to maintain proper vehicle orientation when the aerodynamic forces provided by the wings are insufficient.

MAIN PROPULSION

The main propulsion system is responsible for accelerating the vehicle vertically within various acceleration limitations (maximum acceleration < 5 g’s and minimum acceleration > 1.5 g's) up to the specified altitude. To accomplish this, the system utilizes three rocket engines of 1000 pounds thrust each, optimized for sea-level operation.

The specified chamber pressure is 100 PSI with a 1.72:1 nozzle expansion ratio. This will provide a specific impulse of 212 seconds at sea level and about 218 seconds at altitude.

PROPELLANT MANAGEMENT

The Propellant Management system is responsible for containing and feeding the propellants to the Propulsion system. The main propellant is expected to consist of 1040 pounds of liquid oxygen and 472 pounds of pressurized, liquefied propane. The propane vapor pressure is used to feed both the oxidizer and the fuel into the combustion chamber. The major problem that this propellant feed system must face is maintaining the propane vapor pressure as it expands and cools. In order to accomplish this, a fluid heating system extracts thermal energy from the rocket engines and delivers a controlled heat rate into the propane tank.

ATTITUDE CONTROL RCS

The attitude control system is a cold gas propellant system. It is difficult to know beforehand what total impulse is required for adequate control. However, by setting an arbitrary dynamic pressure equivalent to airspeed of 120 mph as the lower limit of controllability, it becomes obvious that for about 20 seconds during ascent the vehicle will not have sufficient dynamic pressure for control. If an additional 10 seconds of control authority is provided for the separation period, then about 30 seconds of control authority is required during ascent. It is not expected that control authority will be provided throughout most of the ballistic flight after payload separation. Another 10 seconds of control authority is provided during the ballistic trajectory, however, to maintain some amount of controlled orientation.

Flight Phase	**Duration of Required Control (seconds)**
Ascent	20
Separation	10
Ballistic Flight Control	10
Reentry Preparation	10
Reentry and high altitude	20
Total Time	70 seconds

During these times, the vehicle will be nearly empty of main engine propellants, minimizing the amount of impulse required to maintain the proper orientation. It is also expected that the nozzles for the attitude control system will be in the nose of the vehicle and/or wing tips, allowing the maximum control moment. If about 5 pounds of thrust is required to perform control for the duration specified, then a total of 350 pound-seconds of impulse is required. Since most cold gas thrusters have a specific impulse greater than

60 seconds, we can calculate an upper bound on the weight of cold gas required. The equation is similar to that used for calculating the Gravity Loss thrust duration:

Wt = Time * Thrust / Isp
Wt = 70 seconds * 5 pounds / 60
Wt = 5.83 pounds

These values will need more explicit experimental determination to be accurate. But, it gives a ballpark figure for the purposes of system design.

GUIDANCE, NAVIGATION AND DATA HANDLING

It is expected that this system have fully autonomous mission control capability with ground supervision and override as necessary. The GNC will be able to control propulsion and aerodynamic surfaces to maintain the desired trajectory, complete the mission and return to the designated runway.

A six degree of freedom navigation system will be required to estimate the vehicle attitude as well as the location relative to the launch site. Modern GPS systems are available which are suitable for this application; the GPS system will be supplemented with other sensor systems.

GUIDANCE COMPUTER

The Guidance Computer is responsible for monitoring and controlling all systems to accomplish the mission function and provide monitored data to the ground station.

GNC SENSORS

The vehicle shall have sufficient on-board sensors to permit it to estimate its position and orientation with sufficient accuracy to accomplish its mission.

Sensor	Sensed Parameter
GPS	Altitude, Latitude, Longitude, Velocity
Barometric Altimeter	Altitude
Inertial Platform	XYZ Acceleration and Rotation Rate
Airspeed Sensor	Airspeed
Sun and Horizon Sensor	Rough Attitude
Temperature	Skin and internal temperature
Height	Height above ground < 30ft

VEHICLE HEALTH AND PEFORMANCE SENSORS

The vehicle shall have sufficient on-board sensors to monitor system health and performance and identify conditions which necessitate an abort or termination.

Sensor	Sensed Parameter
Load Cells x 3	Engine Thrust Forces
Engine Optical Sensors	Engine Combustion
Line Pressure	Propane Pressure in LOX tank
Angle Sensors	Movable air surface positions
Extended Switch	Landing Gear Extended

COMMUNICATIONS SYSTEM

The communication system serves several functions. It provides ground-based control over operations as well as providing telemetry information back to ground control. It also plays an important role in flight termination should the vehicle's path stray from the specified trajectory. Another function it plays is in providing ground-based position and path information to enhance the navigation function.

POWER SYSTEM

This system utilizes a battery-based power system to power the GNC system and various actuators. The total duration of flight and control is approximately 20 minutes from launch to landing. Lightweight batteries able to supply the power for the specified duration will be identified and integrated.

WEIGHT ESTIMATES

WEIGHT BUDGET

The following table details the limits of the weight budget.

Asabooster CD004		
Oxidizer	Lox	
Fuel	Propane	
Payload	152.154	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	212.000	Seconds
Engine Efficiency	1.000	
Desired DeltaV	10000.000	FPS
Body:Fuel Mass Ratio	0.200	
Mf/Me Ratio	4.327	
Propellant Mass	1513.061	Lbs
Oxidizer Mass	1040.229	Lbs
Fuel Mass	472.832	Lbs
Oxidizer Volume	14.604	cuft
Fuel Volume	13.015	cuft
MT	302.612	Lbs
Me	454.767	Lbs
Mf	1967.828	Lbs

The total allowable values for oxidizer weight is 1040.229 lbs (***Oxidizer Mass***), for propellant weight is 472.832 lbs (***Fuel Mass***), the weight limit of the vehicle without payload or propellant is 302.612 lbs (***MT***). The vehicle, without propellants but with payload, is to weigh no more than 454.767 lbs (***Me***). The vehicle weight with payload and propellant is to weigh no more than 1967.828 lbs (***Mf***).

WEIGHT ESTIMATES

Estimating weights is a difficult task. There are so many details which can be easily missed in a preliminary weight estimate. However, some attempt has been made to be complete so that an answer of whether or not the weight budget will be met can be made.

Total Weight Estimate Summary

System	Weight (lb)	Percentage of Total Budget
Structures	65.3	22%
Propulsion	55.0	18%
Main Propellant Handling	89.5	30%
Landing Gear	44.0	15%
GNC, actuators etc	36.0	12%
Total	**289.8**	**97%**

Currently unallocated weight is 13.1 pounds or 4.3%. This means that it is likely that the vehicle will go over its weight budget as additional problems are encountered unless some serious weight control is performed. It is preferable if there were at least a 10% unallocated weight to handle "weight creep," the natural tendency for weight to be added during development. Weight estimates for the Structures and the Propulsion are as accurate as possible.

A likely place for weight savings is in the landing gear. The solution selected utilizes heavy pneumatic wheels. Smaller, lighter wheels or skids might be replaced for the current approach for appreciable weight savings.

Another likely place is the possibility of using heat-treated aluminum tanks. If that were the case, then the tanks might be able to utilize the full yield strength of heat-treated aluminum and the tanks could drop in weight by almost half. This would surely provide the weight margin necessary to ensure successful attainment of the weight budget for the vehicle.

SUBSYSTEM WEIGHT BREAKDOWN

System	Subsystem	Weight (lbs)
Structures	Nose Cone	6.0
	Forward Landing Gear Bay	4.6
	Transition	4.6
	Fuselage	8.5
	Left Wing	4.9
	Right Wing	4.9
	Wing Spar	6.1
	V-Tail	7.4
	Thrust Truss Aft	15.0
	Thrust Ring Forward	3.3
	Total	65.3

System	Subsystem	Weight (lbs)
Propulsion	Engine 1	15.0
	Engine 2	15.0
	Engine 3	15.0
	Cold Gas Propellant	6.0
	Cold Gas Tank	1.0
	Cold Gas Nozzles, Valves and Feedlines	3.0
	Total	55.0

System	Subsystem	Weight (lbs)
Main Propellant Handling	LOX Tank	44.0
	Propane Tank	40.5
	Valves	5
	Total	89.5

System	Subsystem	Weight (lbs)
Landing Gear	Left Landing Gear	14.0
	Right Landing Gear	14.0
	Forward Landing Gear	16.0
	Total	44.0

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
	Wing Actuators	5.0
	Tail Actuators	5.0
	Flight Terminator	3.0
	Wiring	5.0
	Batteries	5.0
	Total	36.0

SYSTEM APPLICATION AND ENHANCEMENTS

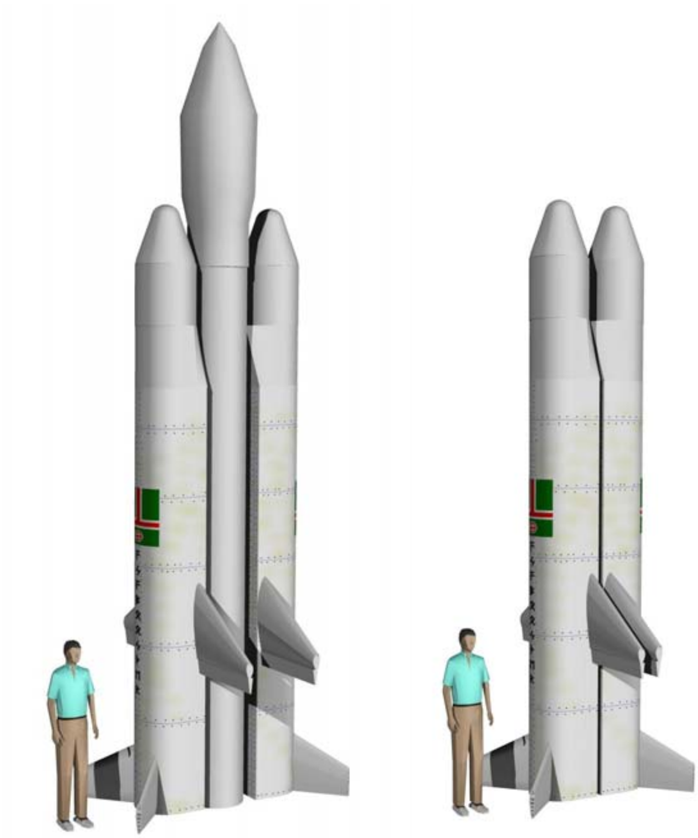
SINGLE BOOSTER THREE STAGE LAUNCH VEHICLE

An early application of this vehicle would be as a booster for a 3 stage launch vehicle. In this configuration, a possible performance capability of placing 5 pounds into low Earth orbit is likely.

CLUSTERED BOOSTER THREE STAGE LAUNCH VEHICLE

By clustering two boosters together to form a first stage, it would be possible to double the payload placed into low Earth orbit.

Additionally, it might be possible to cluster three boosters to place upwards of 15 pounds into orbit. This would probably require that the central booster be expendable without wings.



Clustering Options

SYSTEM ENHANCEMENTS

FLYBACK JET ENGINE

There are a number of small jet engines available on the market which might be suitable in size and weight for inclusion on this vehicle design. Such engines would have to have reliable in-flight restarting capability to be useful for this vehicle. Using one of these could extend the range of the flyback capability should it be desired to place more horizontal velocity on the ascent trajectory.

LIGHTER TANKS

As was seen in the previous section, there are aluminum alloys which can produce significantly lighter tanks, although they may not be as readily available or might require specialized tools to form. Other materials, such as titanium or composites might make very light tanks suitable for this vehicle.

COMPOSITE STRUCTURAL COMPONENTS

There may be structural system components where it makes sense to use composites instead of aluminum. Examples of this could be the wings, tail or other parts.

LIGHTER LANDING GEAR

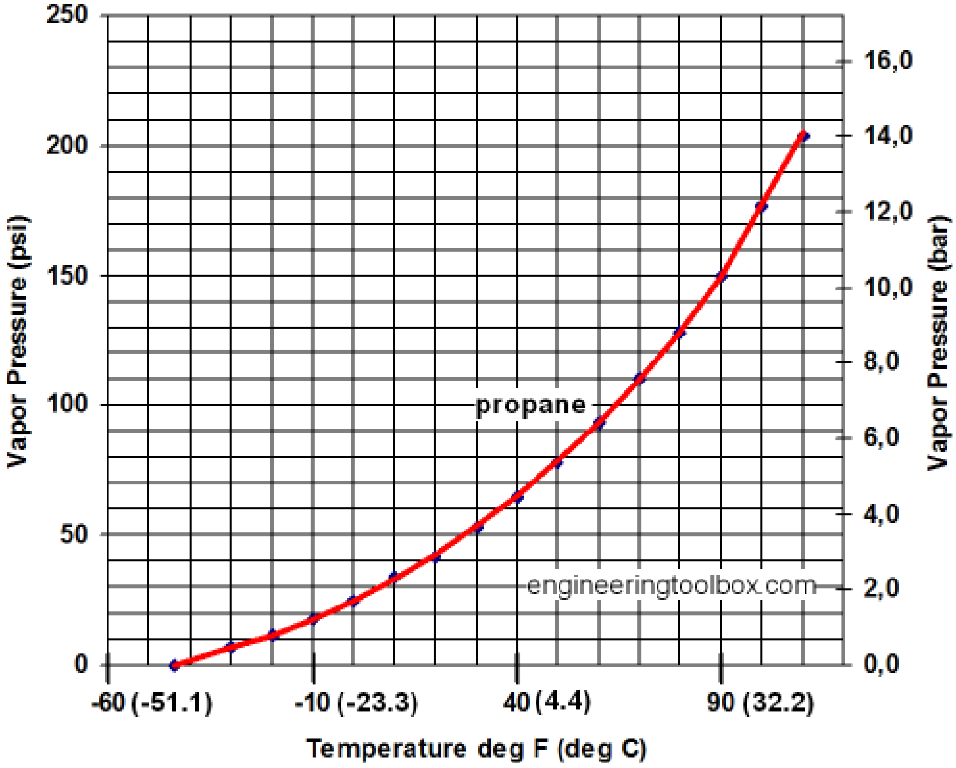
Anyplace that weight savings can be made increases the likelihood of meeting performance objects or even extends beyond the specified objects. The landing gear listed here is very heavy and could probably be lighted appreciably.

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

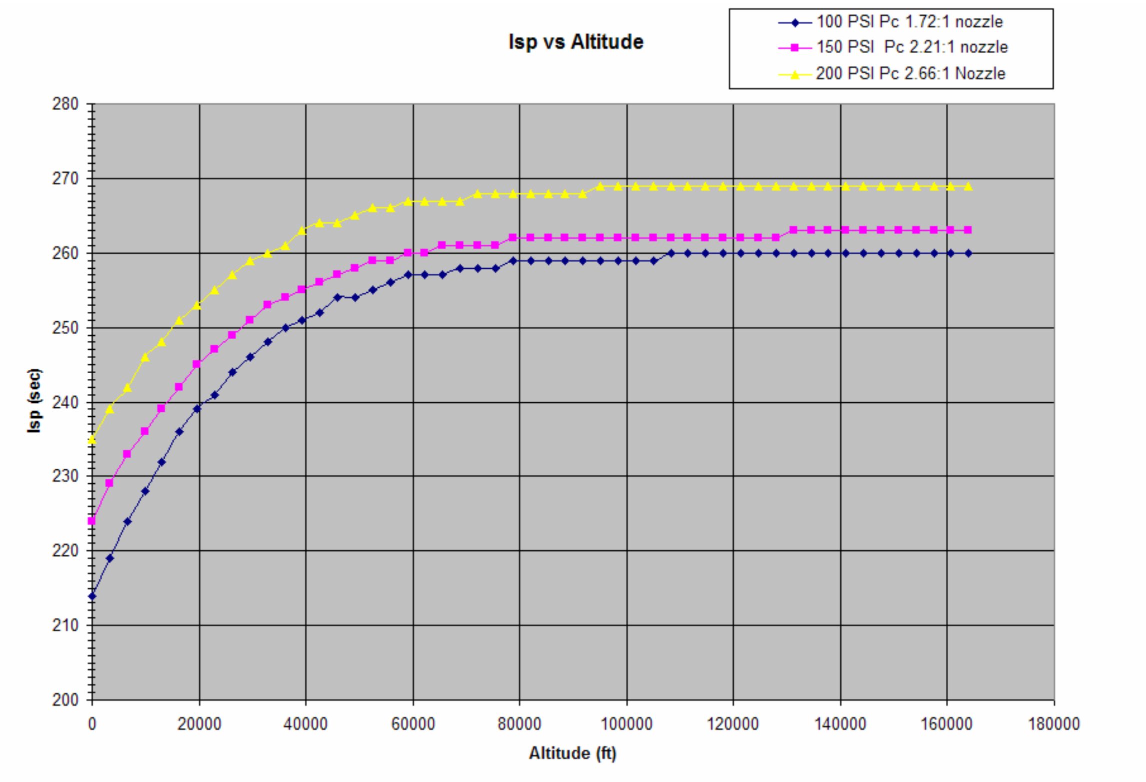
Source: www.engineeringtoolbox.com

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 the 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.

	Propellant 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	5575	degrees Fahrenheit				
	Tc	6034.6	degrees Rankine				
	gamma	1.22					
	Tt	5436.577	degrees Rankine				
Step 3	**Find Throat Pressure**						
	Pc	100	PSI				
	Pt	56.06134	PSI				
Step 4	**Find Throat Area**						
	Gas Molecular Weight	19.415					
	Throat Area	8.830719	square inches				
Step 5	**Find Throat Diameter**						
	Throat Diameter	3.353152	inches				
Step 6	**Find Nozzle Exit Area**						
	Nozzle Expansion Ratio	1.72					
	Nozzle Exit Area	15.18884	square inches				
Step 7	**Find Nozzle Exit Diameter**						
	Nozzle Exit Diameter	4.397618	inches				
Step 8	**Find Chamber Volume**						
	L*	60	inches				
	Chamber Volume	529.8432	cubic inches				
Step 9	**Find Chamber Length**						
	Chamber Scale	2.25	times throat diameter				
	Chamber Diameter	7.544593					
	Chamber Area	44.70552	square inches				
	Chamber Length	11.85185	inches				

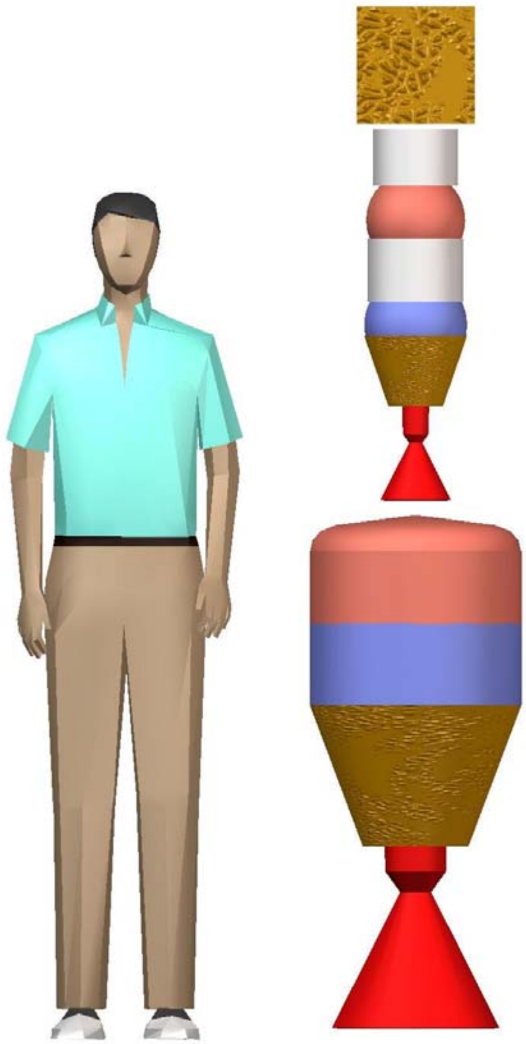
APPENDIX D – NOTIONAL UPPER STAGES

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	5.000	lbs		Payload	25.215	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	12500.000	FPS		Desired DeltaV	12500.000	FPS
Body:Fuel Mass	0.150			Body:Fuel Mass	0.150	
Mf/Me Ratio	3.302			Mf/Me Ratio	3.302	
Propellant Mass	17.578	lbs		Propellant Mass	88.643	lbs
Oxidizer Mass	12.085	lbs		Oxidizer Mass	60.942	lbs
Fuel Mass	5.493	lbs		Fuel Mass	27.701	lbs
Oxidizer Volume	0.170	cuft		Oxidizer Volume	0.856	cuft
Fuel Volume	0.151	cuft		Fuel Volume	0.762	cuft
MT	2.637	lbs		MT	13.297	lbs
Me	7.637	lbs		Me	38.511	lbs
Mf	25.215	lbs		Mf	127.154	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