Bifrost: A 4th Generation Launch Architecture Concept

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Abstract

A 4th generation launch architecture is studied for the purpose of drastically reducing launch costs and hence enabling new large mass missions such as space solar power and human exploration of other planets. The architecture consists of a magnetic levitation launch tube placed on the equator with the exit end elevated to approximately 20 km. Several modules exist for sending manned and unmanned payloads into Earth orbit. Analysis of the launch tube operations, launch trajectories, module aerodynamics, propulsion modules, and system costs are presented. Using the hybrid logistics module, it is possible to place payloads into low Earth orbit for just over $100 per lb.

Introduction

Humanity has dreamed of expanding their realm to include space and other planetary bodies and to use space to improve our own planet. Most of these goals require a large mass in Earth orbit. However, before this becomes practical the cost of access to space must be reduced drastically. *Bifrost* is one of many 4th generation launch concepts designed to reduce the cost of access to space, and hence enable projects such as space solar power and human exploration of other planets. The overall architecture is based on a concept developed by Powell et al [1].

Bifrost consists of a magnetic levitation launch tube with the exit end elevated to approximately 20 km. A common hybrid logistics module (HLM) is designed to attach to an array of propulsion modules that accommodate different missions. This paper focuses on the trajectory analysis for placing the HLM into LEO, and the solar electric propulsion module for circularizing payloads in Geosynchronous Earth orbit (GEO). A brief description of the remaining components is also provided for completeness.

Concept Overview

The architecture consists of an evacuated magnetic levitation launch tube, with one end elevated to 20 km located on the equator, which sends a vehicle into an elliptic Earth orbit. The apogee of the orbit depends on the desired destination. The launch tube is capable of launching either a HLM and associated hardware, or the deep space space shuttle (DSSS). The HLM is a payload canister with a common interface for the propulsion module. It can contain several different internal configurations for launching different types of payload. Three propulsion modules exist for three different in-space purposes. For circularizing in low Earth orbit (LEO) there is a solid apogee kick motor. For unmanned operations in near Earth space there is a module with a liquid rocket engine. For circularizing payloads in Geostationary Earth orbit (GEO) there is a low thrust electric propulsion module. The DSSS has a passenger cabin and uses a propulsion system based on the liquid rocket engine module. It can be used for manned missions to the moon, and beyond with its refueling capability. Each component of the architecture is described in the following sections in more detail. Appendix A shows a summary of each component in the architecture.

Launch Tube

The launch tube uses magnetic levitation technology to accelerate a vehicle through an evacuated tube to orbital speeds. Each vehicle is loaded into the 5 meter diameter launch tube through an airlock to maintain the vacuum. Due to acceleration limits on human cargo, the tube must be about 1400 km long. Track acceleration can be varied to change the departure velocity for different missions and to accommodate human payloads. The tube is elevated using large magnets to repel the tube from the ground, and it is held down using adjustable length cables to vary the exit angle.

The launch tube is located on the equator to take advantage of the Earth's rotation, and the exit end is elevated to about 20 km to reduce drag on the vehicle being launched. The launch tube concept is based on the Startram concept developed by Powell et al [1]. Figure 1 shows an artist's rendering of a vehicle being launched using the *Bifrost* launch tube. The cables that hold the tube down and make adjustment of the departure angle possible are clearly visible in the sketch.

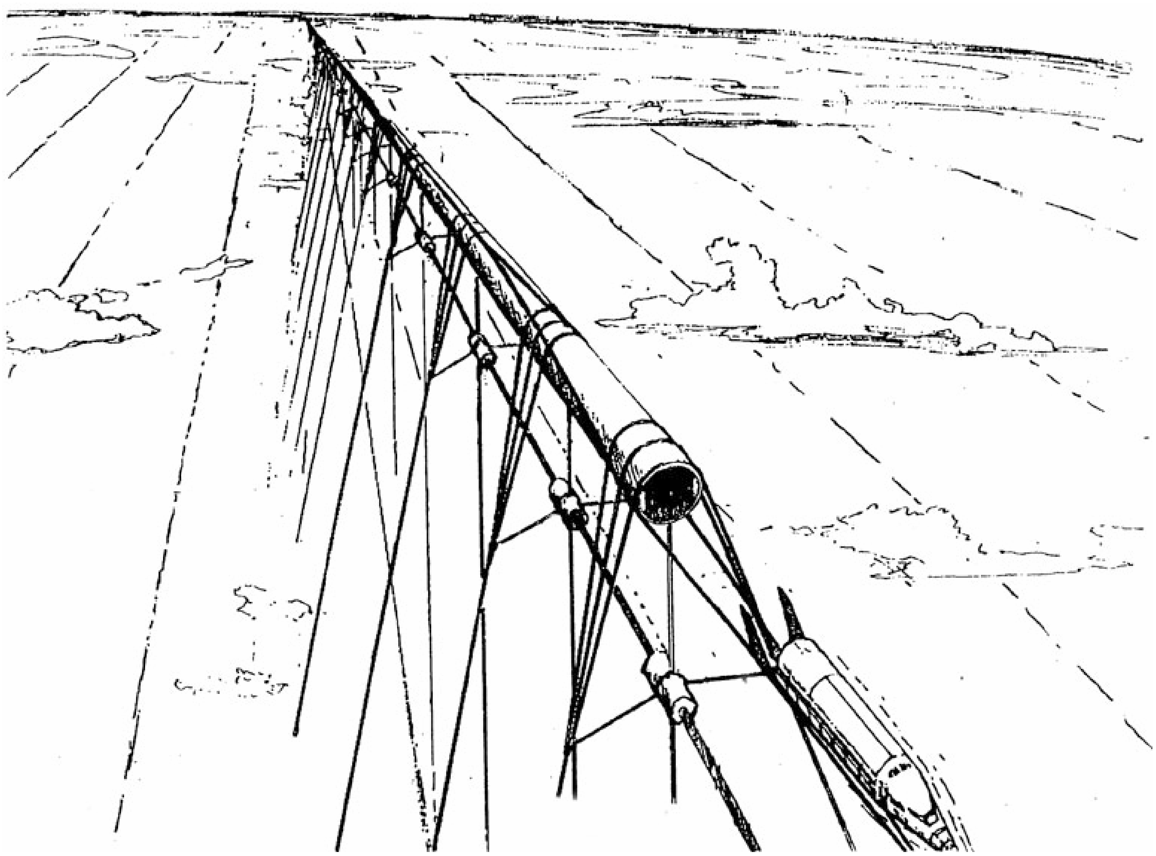

Figure 1: Artist's rendering of a vehicle being launched using the *Bifrost* launch tube. [1]

DSSS

The deep space space shuttle is designed to be a refuel-able spacecraft which will be used for a variety of manned missions to both earth based as well as interplanetary space. Two different DSSS configurations will be analyzed and sized. Both variants of the DSSS will consist of large expansion ratio liquid rocket engines to impart 6,000 m/s of velocity to the spacecraft. The differences in the configurations will be in the fuels analyzed (LH2 and CH4). These different propulsion types are summarized in Table 1. Figure 2 shows a possible configuration for the DSSS which is capable of carrying 8 passengers and 250 kg of payload.

Table 1: Propulsion Characteristics For DSSS

Type	Liquid	Liquid
Propellants	LOX-LH2	LOX-CH4
Cycle	Expander	Expander
Mixture Ratio	5.5	3.5
Expansion Ratio	175	175
Isp_{vac} (sec)	463	368

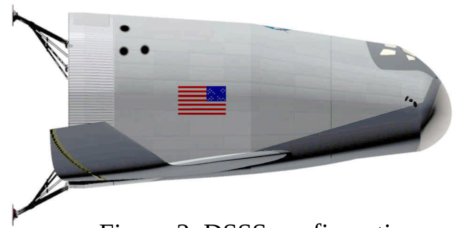

Figure 2: DSSS configuration.

DSSS Missions

As aforementioned the DSSS module was sized to accomplish a ? V maneuver of 6,000 m/s. This sizing allows the DSSS module to complete many different missions. An analysis was conducted to find the required ? V to accomplish different missions including space tourism, Lunar Landing missions, L1 missions, as well as different Mars trajectories. A summary of the missions is included in Table 2.

Table 2: Trajectory Analysis for the DSSS.

	Space Tourism	**Lunar Lander***	**Mars Hohmann**	**Mars 120 day**$^{+}$	**Mars Lander***
Circularization	300 m/s				
Leaving LEO		3106 m/s	3590 m/s	5139 m/s	3590 m/s
Arrival		1445 m/s	2093 m/s	4847 m/s	2093 m/s
DeOrbit	275 m/s	2070 m/s			160 m/s
Ascent		2070 m/s			2280 m/s
Total Delta V	**575 m/s**	**8691 m/s**	**5683 m/s**	**9986 m/s**	**8123 m/s**

Table 2 (Continued): Trajectory Analysis for the DSSS.

	L1 Hohmann	**L1/Mars Transfer***
Circularization		
Leaving LEO	3089 m/s	3089 m/s
Arrival L1	224 m/s	224 m/s
Depart L1		224 m/s
TMI		501 m/s
Mars Insertion		2093 m/s
Total Delta V	3312 m/s	6130 m/s

* Refueling Required

$^{+}$ Not Possible with current configuration

As this table depicts, the DSSS will be able to accomplish space tourism, Mars Hohmann, and L1 Hohmann missions without refueling. It should be noted that the Mars Hohmann mission is very slow and would not be feasible with this size manned spacecraft. The other missions such as Lunar Lander, L1/Mars Transfer and Mars Lander are possible if the infrastructure exists to facilitate refueling of the DSSS. Currently missions to Mars in 120 days are not possible with the propellant available on the 6,000 m/s sized DSSS. Additional propellant volume and therefore ? V can be added to the DSSS, but it would result in a larger more costly spacecraft.

HLM

The hybrid logistics module is designed to have a common interface for attaching to various propulsion modules, aero-shells, and the launch tube. Four HLM internal configurations were analyzed for this project. They include a logistics configuration, a water transport configuration, a space solar power configuration, and a communications satellite configuration. Packaging for each module is shown in Figure 3. The HLM measures 5 meters in diameter, 12 meters long, and has an empty mass of 11.1 metric tons. For simplicity of the propulsion modules, the RCS system is located on the HLM.

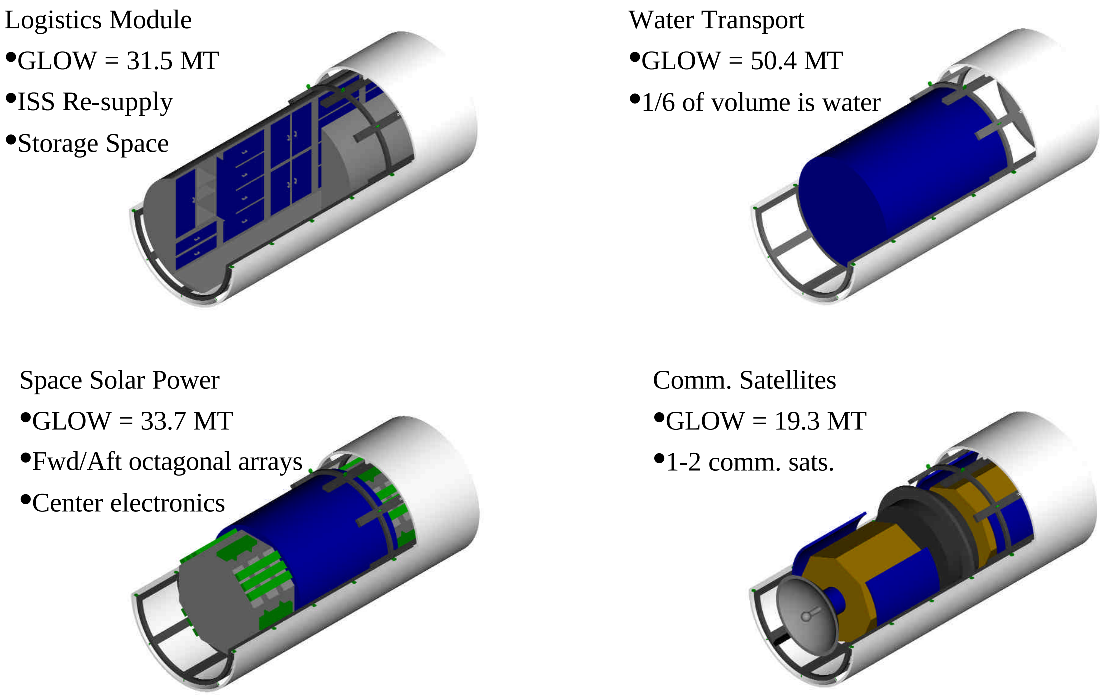

Figure 3: Four configurations for the HLM showing packaging and total mass.

Propulsion Modules

Three different propulsion modules were designed to integrate with the HLM. They include a solid apogee kick motor for circularizing in LEO, a liquid rocket engine for near Earth operations and further with refueling, and a solar electric propulsion module for circularizing in GEO.

Solid Module

For circularizing in LEO, a solid propellant motor was chosen due to the low velocity change required, and the low cost of solid rocket motors. About 300 m/s is required to circularize in a 400 km circular orbit assuming the launch trajectory has its apogee at the same altitude. Parameters for the solid motor are shown in Table 3. Figure 4 shows an artist's conception of the solid propulsion module.

Table 3: Parameters for the solid rocket propulsion module.

Type	Solid
Propellants	AP, Al, HMX, HTPB Binder
Expansion Ratio	85
Ispvac (sec)	297

Figure 4: Schematic of the solid propulsion module.

Liquid Module

The liquid propulsion module is designed to carry payload to accompany manned missions using the DSSS and to deliver payload to Mars. The 120 day Mars mission is the highest delta V of the possible missions for the liquid propulsion module at 10 km/s. A payload of the most massive HLM, the 50.4 ton water transport module is assumed. Rocket engine performance was calculated using SCORES [2], an in-house liquid rocket engine analysis code, using the parameters shown in Table 4. A schematic of the liquid propulsion module configuration is shown in Figure 5.

Table 4: Parameters for the liquid rocket engine.

Type	Liquid
Propellants	LOX-LH2
Cycle	Expander
Mixture Ratio	5.5
Expansion Ratio	175
Ispvac (sec)	463

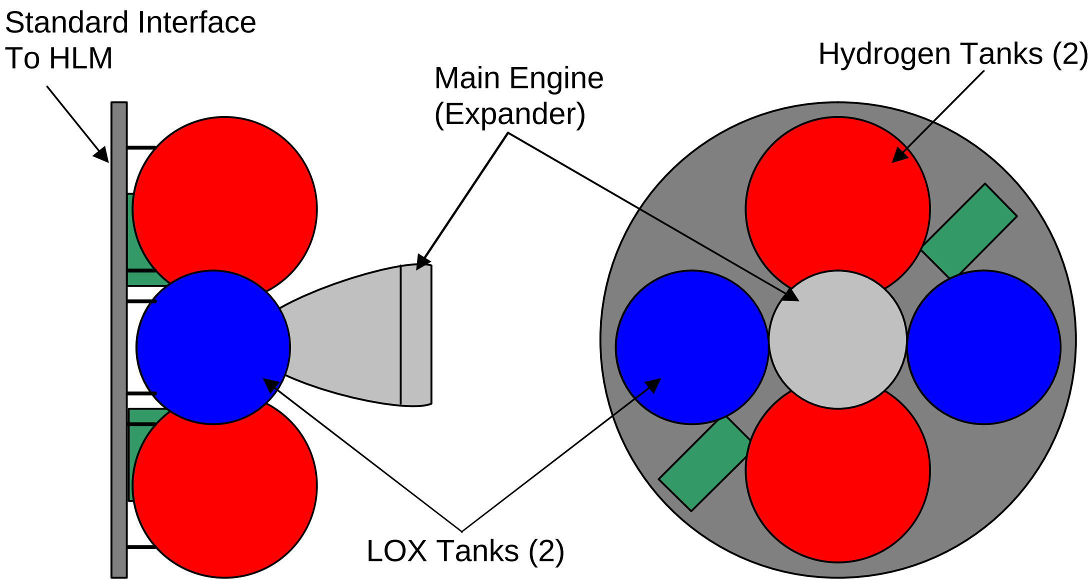

Figure 5: Schematic of liquid propulsion module configuration.

SEP Module

The SEP module makes use of the high efficiency of electric propulsion to circularize a payload in GEO from a highly elliptic transfer orbit. The module first circularizes by thrusting for approximately 60º centered around apogee, and then thrusting continuously to slowly spiral out to GEO. Due to the low thrust of the propulsion, a solid apogee kick motor is fired on the first orbit to raise the perigee to 100 km altitude. Figure 6 shows a schematic of the trajectory.

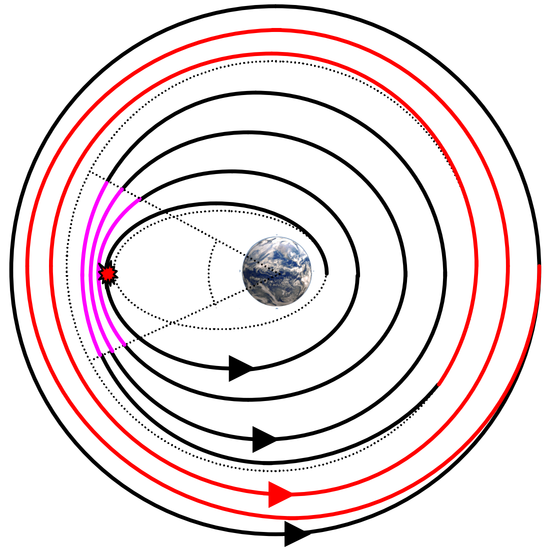

Figure 6: Notional spiral trajectory showing circularization, and the orbit rising to GEO.

The vehicle makes use of inflatable technology to increase the specific power of the solar collection system. Two inflatable concentrating reflectors are attached to opposite sides of the vehicle, and are supported by inflatable struts. Light is reflected from the inflatable reflectors through a lens, off a mirror, and onto a solar array. This complicated light path allows for two key features: only light passes through the rotating joint of the reflectors, and the solar arrays are close to the electric propulsion. The complexity and weight of the system is minimized by only passing light (and not electricity) through the rotating joint of the reflectors. Proximity of the power source to the load reduces the line losses by about 4%, and reduces the mass of wiring. Due to the size of the inflatable reflectors, a deployment system is necessary for the rotating mechanism and the lens. The configuration of the vehicle is shown in Figure 7.

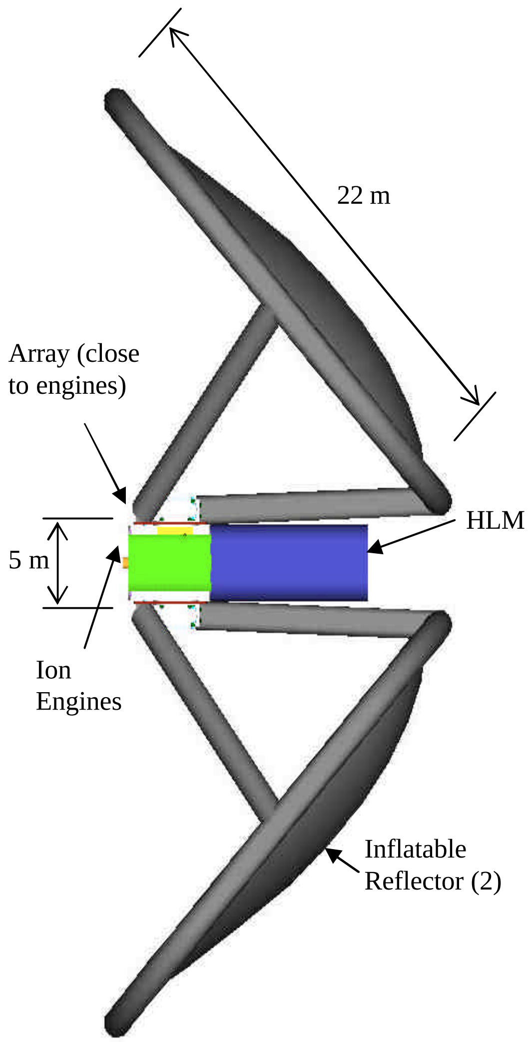

Figure 7: Configuration of the SEP module. HLM attaches on the right.

The power generation system uses the inflatable reflectors to concentrate sun onto a solar array. The array is composed of triple junction $GaInP_2/GaAs/Ge$ solar cells operating at 30% efficiency. To prevent the cell efficiency from dropping due to heat, a thermal control system is necessary. Mass per area of the inflatable structure was assumed to be similar to hardware produced by L'Gard [3]. Total mass of the power system comes to 900 kg to produce 86.8 kW of electric power.

The propulsion system consists of a Xenon ion engine. The analysis was based on propellant type, exhaust velocity and available power. Using the available power, the Isp is calculated. A curve fit of thruster efficiency as a function of Isp from [4] is used to get the thrust after reducing the input power to account for the power processing unit. Engine mass was also calculated using curve fits from [4], and accounts for the engine and the power processing unit.

Trajectory analysis was performed using an in-house numerical integration code based on Cowell's method. Two steps were required for the trajectory from the launch tube to GEO. The first step used eccentricity as the stopping condition and commanded thrust for about 60º of true anomaly centered on apogee. The next phase used continuous thrust to increase the orbital radius until GEO was reached. Both phases maintained the thrust parallel to the velocity vector. The Earth shades the vehicle for only 4.8% of the orbital period, and so was not accounted for. Since the SEP module only thrusts for 60° of the orbit, launch timing can be used to determine the argument of periapsis to keep the thrust segment out of the shadows during circularization.

Use of a computational framework enabled system-level numerical optimization. Each of the disciplinary analyses were wrapped and added to the model. In addition to the disciplinary analysis, several built in optimization methods are available. The design was converged using a script component that performs fixed point iteration.

After several trial runs, sequential quadratic programming was chosen as the most effective optimization scheme of those available in ModelCenter®. The optimization process used the normalized initial mass for the objective function, with the goal to minimize this quantity. Table 5 lists the constraints and the design variables with their upper and lower bounds. q, the angle of the rear reflector support strut is limited to prevent shading from the strut. The rear strut length, D_1, and the overall reflector length, L_{refl}, are limited in size to keep the structural dynamics problems to a minimum. Side constraints were placed on the parabola constant, X_{refl}, exhaust velocity, and magnification to keep them within physically reasonable bounds. The design variables are the parabola constant, the position of the lower edge of the reflector, X_{refl}, the exhaust velocity, and the magnification. All other quantities were fixed due to the *Bifrost* launch architecture.

To aid the optimizer, the objective function, the constraints, and the design variables were all normalized. The default settings were used for finite difference gradients and convergence.

Table 5: Constraints and design variables with their bounds.

Variable	Lower Bound	Upper Bound
q (deg.)	3	None
D_1 (m)	None	60
L_{refl} (m)	None	45
Parabola Const.	0.001	0.1
X_{refl} (m)	0.01	1.0
Exhaust Vel. (m/s)	14,000	40,000
Magnification	40	800

Results

This project focused on analysis of the launch trajectory, and the design of the SEP module. Results from the analysis and optimization from those two activities are presented below. Additional work was done on the DSSS and solid propulsion module and the cost of launch using the DSSS and HLM.

LEO Launch Trajectory

Trajectory analysis of the HLM was performed using the Program to Optimize Simulated Trajectories (POST) [5]. Due to the preliminary stage of the study, a parametric representation of the data was desired. Launch tube release velocity, exit angle, altitude, and aero-shell geometry were varied, and the required spacecraft velocity increment to reach LEO (400 km circular orbit) was recorded. Dynamic pressure, heat rate, and acceleration were also recorded for use as constraints.

Aerodynamic analysis was performed using the Aerodynamic Preliminary Analysis System II (APAS) tool [6]. Three different aero-shells were analyzed for the HLM, to determine the sensitivity of the trajectory to the aerodynamic performance of the HLM. The three shapes analyzed are shown in Figure 8 along with their respective drag numbers. Figure 9 shows the variation in drag coefficient as a function of Mach number for each length of aero-shell. The reference area is the cross-section area of the vehicle.

The aerodynamic data from above was then used in POST to create velocity increment charts. Figure 10 shows a representative plot of the results of the trajectory trade study using the lowest drag aero-shell with the acceleration, heat rate, and dynamic pressure constraints marked on each curve. Values of all constraints increase as the release velocity increases.

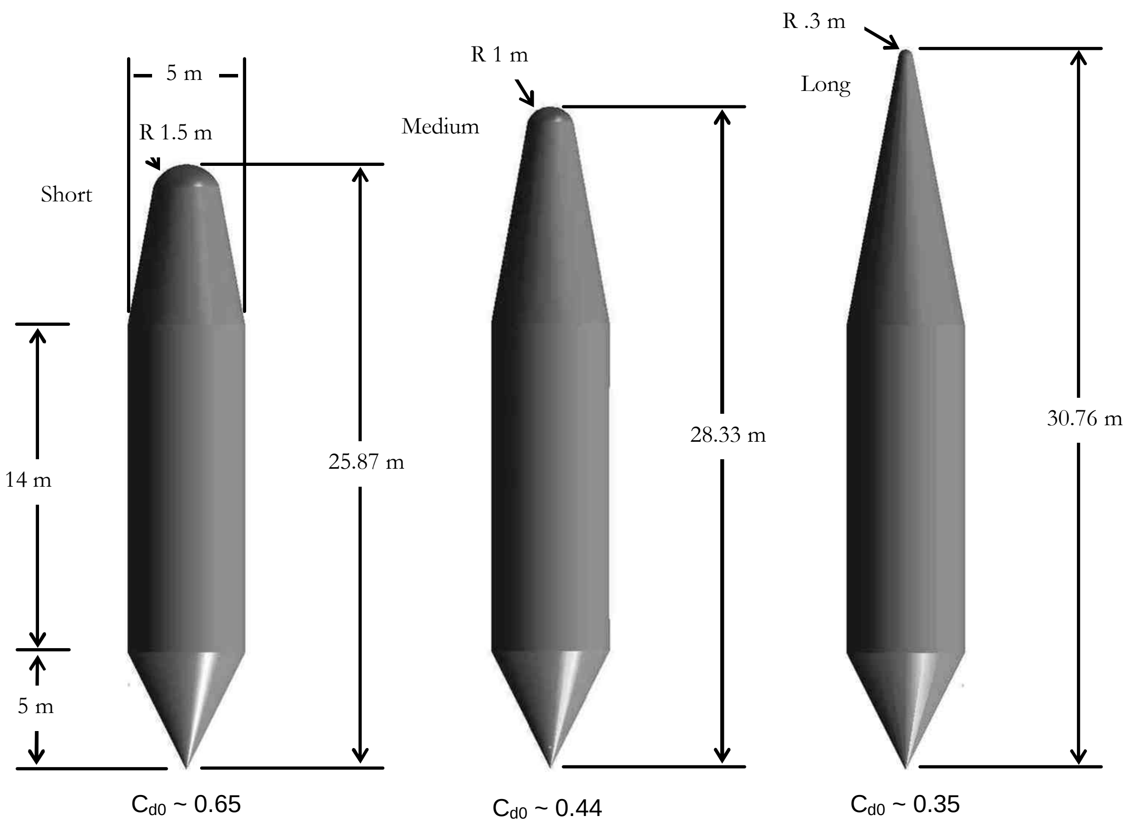

Figure 8: Geometry for the three different aero-shells studied.

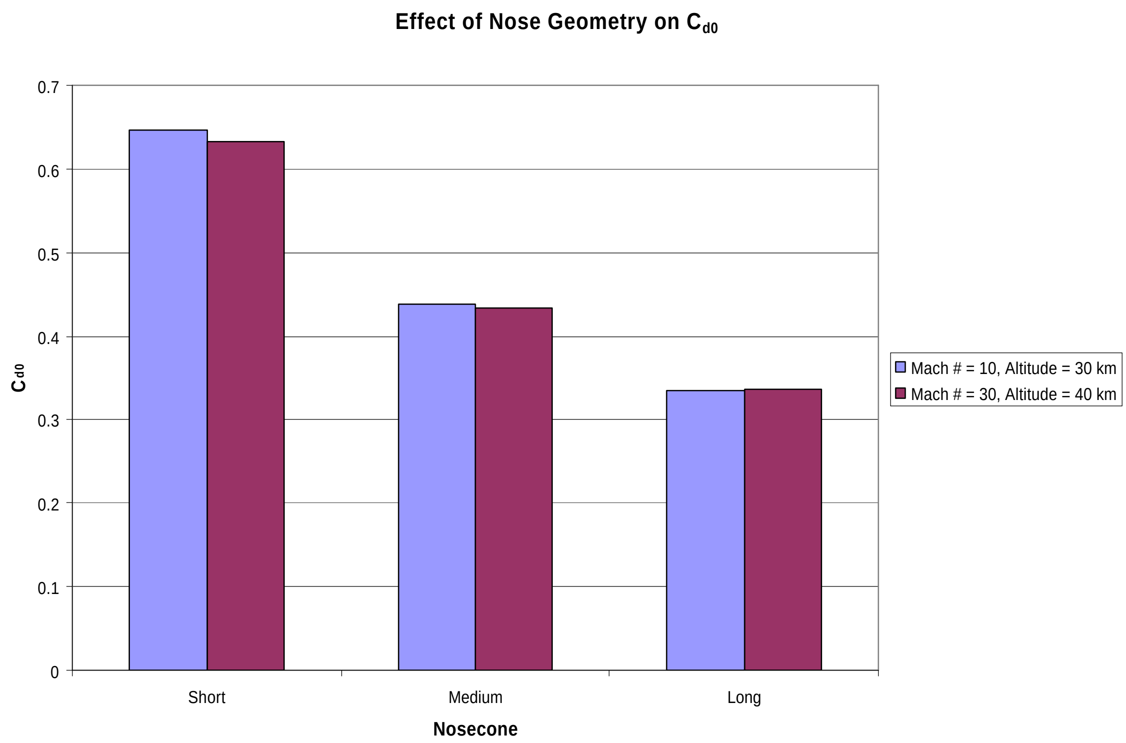

Figure 9: Comparison of drag at Mach 10 and Mach 30 for the three different aero-shell lengths.

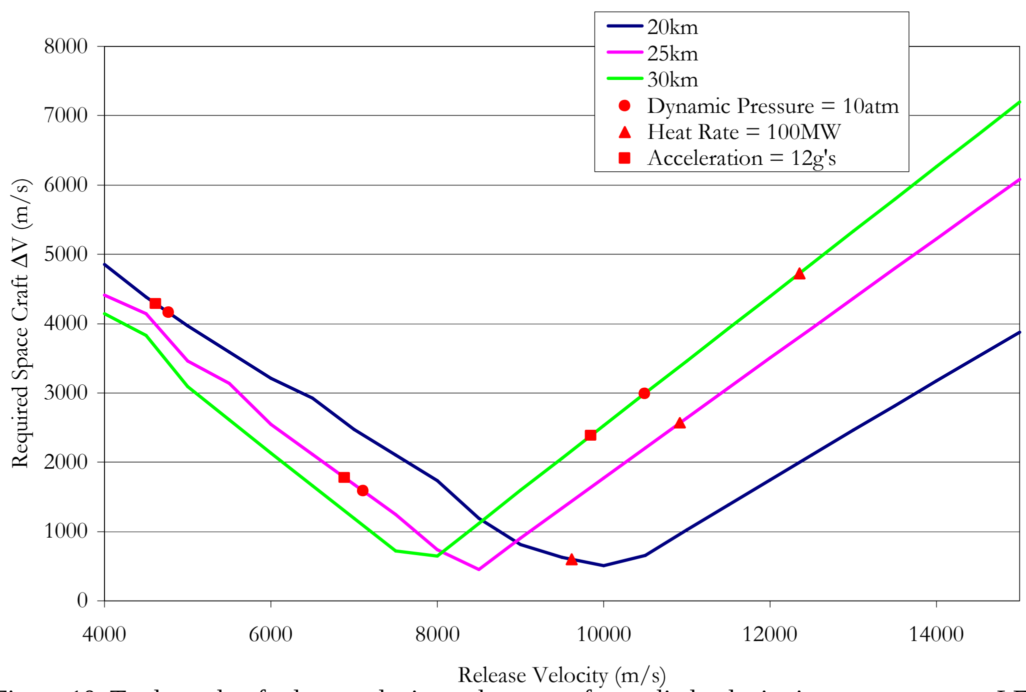

Figure 10: Trade study of release velocity and spacecraft supplied velocity increment to enter LEO using the lowest drag aero-shell at a 6º release angle. Dynamic pressure, heat rate, and acceleration increase as release velocity increases.

DSSS

As described in the preceding sections two fuels were analyzed for the DSSS, CH4 and LH2. Both variants of the DSSS use LOX as an oxidizer and are sized for the ? V requirements of 6,000 m/s. Since the launch tube inner diameter is set at 5 meters only the length of the overall shuttle was changed to meet the propellant requirements. A set cargo (manned and cargo) volume of 80 m^3 was used for both variants of the DSSS to accommodate the eight passengers and 250 kg of cargo. Table 6 is a sizing summary of the DSSS.

From the sizing analysis it is shown that the hydrogen configuration, although having a higher Isp, results in approximately a 45% increase in volume and length. This of course does not include any technology enhancements such as slushed hydrogen, which may be technologically viable when *Bifrost* becomes operational. Table 6 shows a sizing summary of the DSSS.

Table 6: Sizing summary of the DSSS.

Propulsion:	**LH2/LOX**	**CH4/LOX**
Landing Isp (sea level)	440 sec	350 sec
In-space Isp	462 sec	368 sec
Installed Eng. T_sl/We	55	75
Overall T/W at Exit	0.8	0.8
O/F Ratio (weight)	5.5	3.5
Geometry:		
Total Propellant Mass	64,185 kg	96,857 kg
Oxidizer Volume	51 m^3	65 m^3
Fuel Volume	148 m^3	50 m^3
Non-Cabin Volume	56 m^3	39 m^3
Total Est. Volume	335 m^3	234 m^3
Est. Length	17.0 m	11.9 m
Sizing:		
Mass Ratio	3.76	5.27
Initial Mass	87,459 kg	119,541 kg

Solid Module

Using the above trajectory analysis and a simple sizing routine, a mass was determined for the solid propulsion module for each of the four HLM configurations. In order to produce only four different solid rocket motors it will be necessary to ballast the HLM to one of the masses listed in Figure 3 above. Table 7 shows the mass of the solid propulsion module for each HLM configuration.

Table 7: Solid propulsion module mass for each HLM configuration.

HLM Configuration	HLM Mass (metric tons)	Solid Propulsion Module Mass (kg)
Water Transport	50.4	6,150
Space Solar Power	33.7	4,120
Logistics Module	31.5	3,850
Communications Satellites	19.3	2,357

SEP Module

On completion of the optimization, the vehicle had lost significant mass from the initial guesses for the design variables. The final values of the design variables and select outputs are shown in Table 8. All variables are up against constraints except for the parabola constant, and the reflector dimensions L_{refl} and D_1. Since no time constraint was specified the exhaust velocity is at the maximum allowed for ion engines to maximize the engine Isp, and hence reduce mass. The reflector dimensions are primarily derived from the exhaust velocity since this determines the power required and the reflector size. Trip time came to 575 days due to the lack of a time constraint. Depending on the cargo, this trip time may not be acceptable. Earlier in the design process optimization was performed with a minimum thrust constraint of 13 Newtons, resulting in a trip time of 190 days, but a much higher initial mass of 49,530 kg. The full 12 point mass breakdown structure is shown in Table 9.

Operations and Cost

Operations analysis was also performed as a contributing analysis to determine the cost of the launch architecture. The Architecture Assessment Tool – enhanced (AATe) written at Kennedy Space Center was used in the development of the operational model for *Bifrost*. The analysis is based on data taken from Space Shuttle operations but uses aggressive assumptions in terms of automation and required maintenance. Figure 11 shows the results of the site layout and analysis.

The operational model and costs were added to a weight based costing model to determine overall system costs. The DSSS is designed to last for 500 flights with a unit cost of $2.5B, while the HLM is expendable with a unit cost of $6M including a propulsion module. The ground facility was assumed to be paid for by a government, but the operational and maintenance costs are covered by the launch operator. All non-recurring costs were distributed evenly over the 40 year program life. Table 10 shows the remaining assumptions for each vehicle, the life cycle cost for each vehicle, and the total cost per unit mass to orbit using the HLM. This architecture shows drastic improvement over current launch technology, and enables launch for just over $100/lb.

Table 8: Design variables and selected outputs after optimization.

Variable	Value
Parabola Constant	0.03476
X_{refl} (m)	0.9779
Magnification	40.0
Exhaust Velocity (m/s)	40,000
q (deg.)	3.0
D_1 (m)	18.7
L_{refl} (m)	22
A_{refl} (m^2)	305
Mass Ratio	1.04622
M_{init} (kg)	40,610
Propellant Mass (kg)	1,790
Engine Thrust (N)	4.22
Engine Isp (sec)	4,077
Engine Propellant	Xenon

Table 9: Mass breakdown statement for the optimized SEP module.

Component	Mass (kg)
Structure	1,450
Power Generation	830
Power Distribution	50
Thermal Control	190
Propulsion	480
Control and Avionics	110
Margin (20%)	620
Dry Mass	**3,730**
Reserves and Residuals	90
Pressurant	1
Payload	35,000
GEO Mass	**38,820**
Boost Propellant Mass	1,790
M_{init}	**40,610**
AKM Mass	170
Initial On Orbit Mass	**40,780**

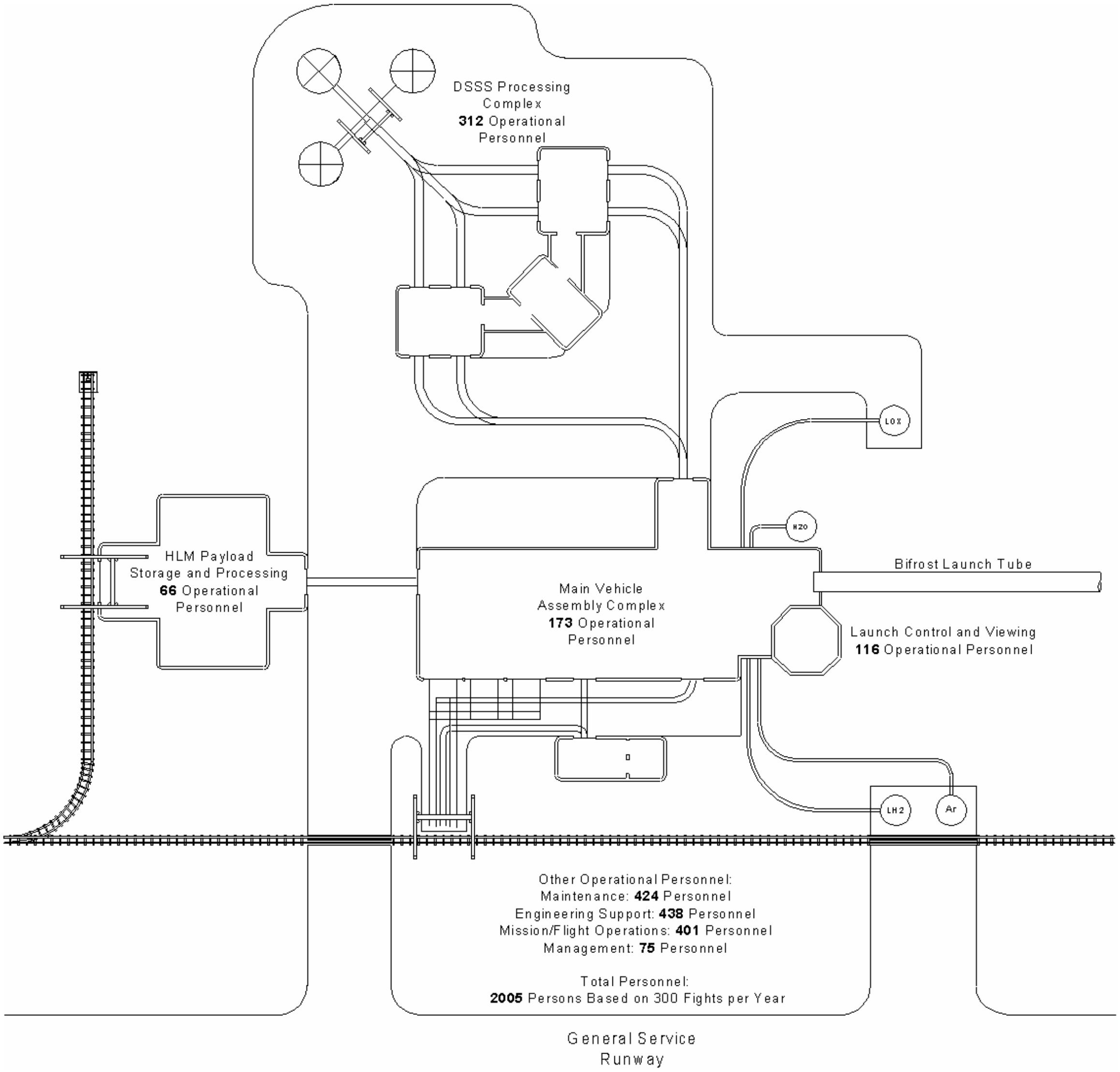

Figure 11: Operational layout showing number of required personnel to operate *Bifrost*.

Table 10: Cost of *Bifrost* architecture in FY02$.

DSSS			**HLM**		
Flights Per Year	50	flights	Flights Per Year	250	flights
DDT&E	10,000	M	DDT&E	1,000	M
Fixed Cost per Year	519	M	Fixed Cost per Year	1,145	M
Variable Cost per Flight	9.73	M	Variable Cost per Flight	6.63	M
Total Cost per Flight	20.10	M	Total Cost per Flight	11.21	M
Total Life Cycle Cost	40,207.41	M	Total Life Cycle Cost	112,087.08	M
			Total Cost Per Payload kg	224.17	$/kg
			Total Cost Per Payload lb	**101.67**	**$/lb**

Conclusions

Detailed analysis was performed for the launch trajectory to LEO, and for the SEP module. The launch trajectory to LEO was explored and a parametric model is now available for use in overall architecture optimization. Input parameters include aero-shell, release velocity, and exit angle. Heat rate, acceleration, and dynamic pressure were also recorded for use as constraints. Optimization was performed on the SEP module which resulted in an initial mass of 40,780 kg and a trip time of 575 days to GEO carrying a 35,000 kg payload. By increasing the thrust to 13 N a more reasonable trip time of 190 days to GEO was achieved, but at the much higher mass of 49,530 kg.

Preliminary analysis of the DSSS and solid propulsion modules was also performed. Two fuels were explored for the DSSS, CH4 and LH2. As expected the hydrogen fueled vehicle had a lower initial mass of 87,460 kg but was bulkier, measuring 17m long. The methane fueled vehicle had an initial mass of 119,540 kg at a length of only 12m. The solid propulsion module required different mass modules for each different HLM configuration. Module mass varied from 6,150 kg to 2,360 kg for payloads ranging from 50.4 mt to 19.3 mt.

The *Bifrost* architecture is designed to reduce the cost of access to space and to enable a large number of missions. The analysis presented in this paper shows that the *Bifrost* architecture is successful in reducing the cost of launch. If the government pays for the facility, the cost of placing one pound of payload into orbit using the HLM is just over $100.

Future Work

Though much analysis has been performed on the *Bifrost* architecture, there is still room for improvement. There are several components that could benefit from more detailed analysis including the liquid and solid propulsion modules and the DSSS. Generation of a complete mass breakdown structure for these modules would result in better cost and maintenance analyses. The launch trajectory to GEO also requires further analyses and could validate some of the assumptions made in the design of the SEP module.

Once a complete model of each component of the architecture is complete, optimization of the overall system based on the predicted launch market could be performed. This analysis could show the correct number of each module to manufacture in order to minimize the cost of payload, manned or unmanned, to orbit. Care would need to be taken in the choice of market model since the Bifrost concept has the potential, itself, to change the market model.

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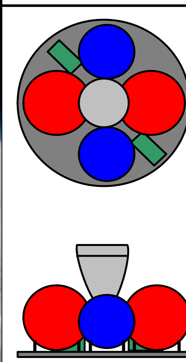
Appendix A

Summary of *Bifrost* architecture components.

Component Name	Picture	Initial mass (kg)	Cost per Flight (FY02$ Millions)	Payload (type, kg)	DV (m/s)
DSSS		87,460 (LH2) 119,540 (CH4)	20.1	8 passengers 250 kg payload	6,000
HLM		50,400 (water trans.) 33,700 (space solar power) 31,500 (logistics module) 19,300 (Comm. Sats.)	11.21 Including propulsion module	Water Space solar power Equipment & Inst. Communication Sat.	
SEP Module		40,780		HLM Assumed to weight 35,000 kg to GEO	1,805
Liquid Propulsion Module				HLM	10,000

Component Name	Picture	Initial mass (kg)	Cost per Flight (FY02$ Millions)	Payload (type, kg)	DV (m/s)
Solid Propulsion Module		6,150 (water trans.) 4,120 (space solar power) 3,850 (logistics module) 2,360 (Comm. Sats.)		Water Space solar power Equipment & Inst. Communication Sat.	300
Launch Tube				HLM + Propulsion Module + Aero-shell or DSSS	>4,000