Reusable Exploration Vehicle (REV)

Orbital Space Tourism Concept

Ian Clark
Scott Francis
Richard Otero
Grant Wells

Faculty Advisor:
Dr. John Olds

2005
RASC-AL
Design
Competition

Reusable Exploration Vehicle (REV): Orbital Space Tourism Concept

Ian G. Clark[i], Scott Francis[*], Richard Otero[*], Grant Wells[*]
Faculty Advisor: Dr. John Olds

Guggenheim School of Aerospace Engineering
Georgia Institute of Technology, Atlanta, GA 30332-0150
ian_clark@ae.gatech.edu

On the heels of the recent success of the X-Prize, sub-orbital space tourism is nearly a reality. Though the requirements are significantly tougher, orbital space tourism is the next logical step. The Reusable Exploration Vehicle (REV) concept is an economically feasible design capable of making this next step. Centered around a lenticular lifting body, the REV concept relies on commercial launch vehicles to reduce DDT&E expenditures. Capable of ferrying five passengers and one crew member for three orbits, the REV is shown to be capable of keeping maximum debt exposure to less than $250M while attaining an IRR of 70% with an estimated market capture of 66%.

Nomenclature

ACC	=	advance carbon-carbon
AFRSI	=	advanced flexible reusable surface insulation
ECLSS	=	environmental control and life support system
FRCI	=	fibrous refractory composite insulation
FY	=	fiscal year
GPS	=	Global Positioning System
HTPB	=	hydroxyl-terminated polybutadiene
IRR	=	internal rate of return
LES	=	launch escape system
LH2	=	liquid hydrogen
LMNoP	=	Launch Market for Normal People
LOX	=	liquid oxygen
NASA	=	National Aeronautics and Space Administration
NPV	=	net present value
OEC	=	overall evaluation criterion
q-α	=	dynamic pressure times angle of attack
ROI	=	return on investment
PLS	=	Personnel Launch System
PMAD	=	power management and distribution
POST	=	Program to Optimize Simulated Trajectories
REV	=	reusable exploration vehicle
RCC	=	reinforced carbon-carbon
RCS	=	reaction control system
RSE	=	response surface equation
TABI	=	tailorable advanced blanket insulation
TPS	=	thermal protection system

I. Introduction

THE idea that tourism will soon encompass regions beyond Earth's atmosphere is enchanting. With the winning of the Ansari X-Prize, Scaled Composites' Spaceship One has brought enthusiasm and publicity to the idea that people will soon be paying for a trip to space just as they would pay for a ticket on a cruise ship. Although the first space tourists will most likely take suborbital trips, many consider the ultimate goal for space tourism to be the capability to put tourists in orbit. Although the technical knowledge to put people in orbit exists, costs associated with designing, testing, evaluating, and fabricating a vehicle will make it difficult for a space tourism company to be profitable. The purpose of this study is to develop a conceptual design of an orbital space tourism vehicle and to model a space tourism business case which can break even by the end of five years of flight operations, realize a minimum Internal Rate of Return (IRR) of 30% on the business through the end of ten years of flight operations, and stay within a maximum debt exposure of $250M (FY05).

To begin, a design space of alternative concepts was established and an exhaustive down-selection process utilizing a zero-order analysis and Systems Engineering techniques was executed to determine the most appropriate vehicle for the given mission. Many variables were accounted for in the formation of the design space. Some examples include different types of vehicles (i.e. capsule, lifting body, or winged body), purchasing components off the shelf, number of stages, and number of passengers. Once the best configuration was determined, a more in depth study was performed. This study included an analysis of many disciplines including weights, aerodynamics, heating, trajectory, and economics. Finally, the ticket price leading to the best business case was determined.

II. Initial Design Space Exploration

As part of the initial investigation of the overall design space, a zero-order analysis was conducted for the variables and ranges provided in Table 1. To complete the zero-order analysis, eight tools covering the disciplines of aerodynamics, propulsion, trajectory, weights and sizing, operations, safety, cost, and market were integrated into a spreadsheet workbook and 25920 different design cases were run. The tools varied in complexity from simple rocket equation approximations to higher fidelity operations models built around historical shuttle data. Included in the outputs of this analysis were several performance and economic parameters that were used to weight the merits of each design, including

Table 1. Design space for the zero-order analysis.

Variable		Min.	Max.	Levels
Launch Type	0=Vertical 1=Horizontal 2=Ground Assist 3=Aircraft 4=Balloon	0	4	5
COTS Launch Vehicle	0=No, 1=Yes	0	1	2
Vehicle Type	1=Capsule, 2=Lifting Body 3=Winged Body	1	3	3
COTS Manned Vehicle	0=No, 1=Yes	0	1	2
# of Stages		2	3	2
# of Passengers		3	6	4
Type of 1st Stage Prop.	1=LH2/LOX	1	3	3
Type of 2nd Stage Prop.	2=RP/LOX	1	3	3
Type of 3rd Stage Prop.	3=HTPB/LOX	1	3	3
Ticket Price	FY2005 $M	10	15	2
			Cases: 25920	

In order to make the trip affordable to tourists, development and operations costs of the space tourist launch system must be low. To this end, several ground rules and constraints were established. These included the following:

•that $30M had been secured from an investor as starting capital

•the total instantaneous debt could not exceed $250M

•the minimum acceptable Internal Rate of Return (IRR) on the business through the end of 10 years of flight operations was 30%

•the venture must break even by the end of 5 years of flight operations. Of all the design points evaluated, 25 met the IRR, maximum debt exposure, and breakeven year constraints. These are shown plotted against the economic constraints in Figure 1 below.

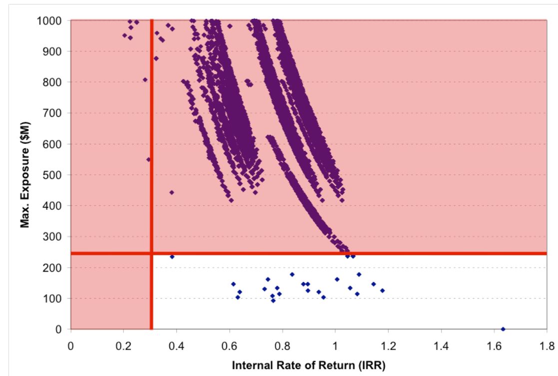

Figure 1. Summary results from zero-order analysis.

The most significant trend from the zero-order analysis was the almost total absence of designs using in-house developed launch vehicles. The two designs that did use an in-house developed launch vehicle used a two-stage booster and capsule launched from a carrier aircraft. Given the constraints on the design space listed above ($30M starting capital and $250M maximum debt), it is not surprising that a launch vehicle cannot be developed in addition to a 4-person (three passengers and one crew member) vehicle for space tourists.

Every feasible case had a two-stage booster, and except for the aircraft launched rocket, all the cases were launched vertically from the ground. No case charged passengers less than $10M for a ticket, which implied that costs cannot be recovered at lower ticket prices.

Since there was a mix of number of passengers and vehicle type (capsule, lifting body or winged body), the Technique for Ordered Preference by Similarity to the Ideal Solution was used to rank each design alternative.

In order to determine what case would be best to use for the conceptual design phase, Monte Carlo runs were performed to determine how many failures could be expected in 1000 cases assuming a reliability of 0.995. Once the Monte Carlo cases were run, the cases were evaluated using an overall evaluation criterion (OEC) incorporating the medians and standard deviations of each design's economic parameters.

From the results of the Monte Carlo runs and subsequent rankings by OEC values, the design shown in Table 2 was judged to be the best design to pursue. It should be mentioned that the output numbers are from the zero-order analysis and do not represent the final economics numbers.

Table 2. Summary of top downselection candidate.

Metric	Feasible Design
Purchase Launch Vehicle?	Y
Launch Type	N/A
Purchase Manned Vehicle?	N
Number of Stages	N/A
Manned Vehicle Type	Lifting Body
Number of Passengers	6
Engine Type	
Stage 1	N/A
Stage 2	N/A
Reusable Launch Vehicle?	N/A
Ticket Price	$15 M
IRR Median	1.145
IRR Std Dev	0.102
NPV Median	$1226 M
NPV Std Dev	$455 M
Exposure Max Median	$146 M
Exposure Max Std Dev	$5455 M
Econ Success Rate	0.944
ROI Median	1.190
ROI Std Dev	13.957
OEC Value	1.258

III. Vehicle Overview

The completion of the design space exploration provided an initial configuration starting point, namely, a lifting body style of crewed vehicle. Three lifting body configurations were chosen for additional downselection: Dyna-Soar, an orbital space plane concept from the early 1960's; the HL-20, a design out of NASA Langley for a space station crew emergency rescue vehicle or a Personnel Launch System (PLS); and the Langley Lenticular Lifting Body, a design examined extensively as a possible lunar return vehicle for the Apollo missions. Each design was examined both quantitatively and qualitatively over a range of metrics. A convenient representation of the results from the analysis is a Pugh Matrix and one is provided in Table 3 below. For comparison purposes, the HL-20 is used as a baseline vehicle. For the metrics considered, the lenticular shape came out the best and was thus selected for the baseline vehicle.

Table 3. Pugh matrix of considered lifting bodies.

	HL-20	Dyna-Soar	Lenticular
Heating	0	-	+
Crossrange	0	+	-
Packaging	0	-	+
Structural Mass	0	-	+
Entry Simplicity	0	0	+
Stability	0	+	-
Previous Studies	0	0	0
Total	**0**	**-1**	**+2**

Provided in Figure 2 are the final vehicle dimensions for the REV concept. Sized to seat five passengers and one crew member, the vehicle has an overall diameter of 18.5 feet and a height of 6 feet. For reference, the diameter of the Apollo capsule was 12.8 feet. Passengers enter through the hatch shown in Figure 3 and sit in their seat to enjoy the ride to orbit. Each seat may in turn be folded down to increase the open internal volume of the vehicle in orbit as shown in Figure 4. In orbit, each passenger and the pilot has a vast window to look out of and enjoy the views of Earth.

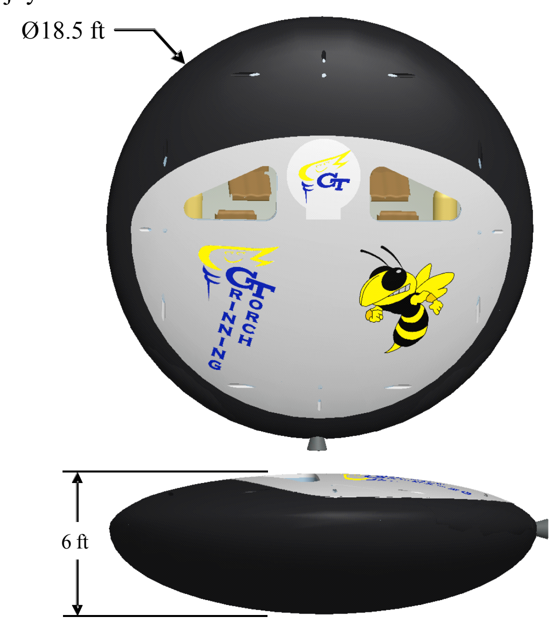

Figure 2. Vehicle dimensions.

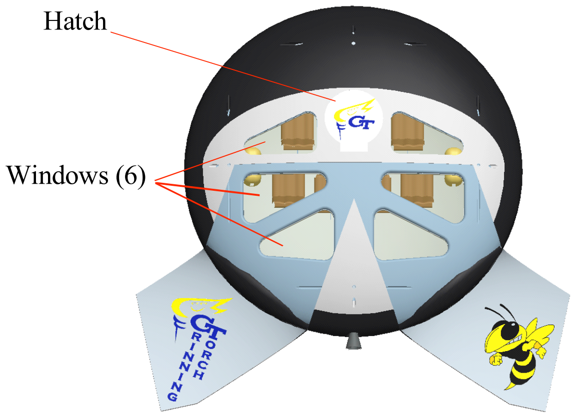

Figure 3. Location of windows and hatch.

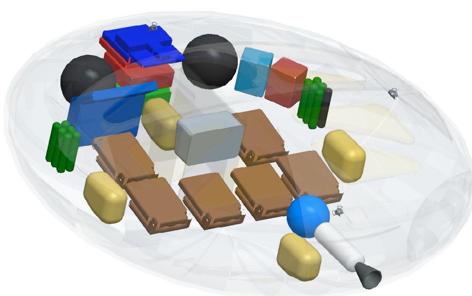

Figure 4. Vehicle with seats folded down.

Table 4. Key aerodynamic performance numbers.

Metric	Value	Units
Planform S_{ref}	268	ft^2
Hypersonic		
L/D_{Max}	0.88	
$\alpha_{L/Dmax}$	25°	
L/D_{Trim}	0.1	
α_{trim}	80°	
Subsonic		
L/D_{Max}	4.86	
$\alpha_{L/Dmax}$	8.2°	
L/D_{Trim}	2.5	
α_{trim}	35°	
Landing speed	153	kts.
c.g. (from nose)	42.8%	

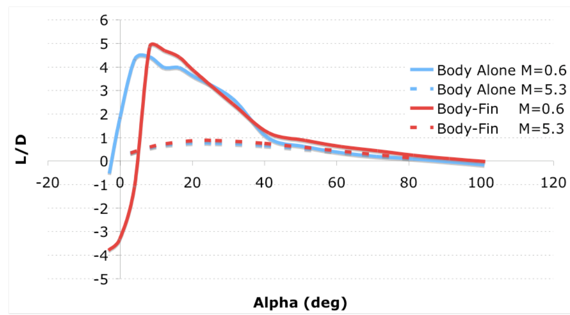

Figure 5. L/D ratios for body and body-fin configurations.

A. Aerodynamics

When evaluating the aerodynamic characteristics of the lenticular shape, two primary configurations are considered, the ballistic entry configuration and the maneuvering configuration with flaps deployed. The ballistic entry configuration is used predominantly for initial atmospheric entry and through peak deceleration and heating. Shortly after peak deceleration the flaps begin deploying and the vehicle transitions to a higher lift and more maneuverable configuration.

Because the vehicle's dimensions are nearly identical to those previously studied in the early 1960's (though scaled), a significant amount of wind tunnel data[2,3,4,5] was available and was utilized for both the subsonic and the supersonic flight regimes. For hypersonic analysis a configuration based aerodynamics code was utilized. A summary of the key aerodynamic performance numbers is presented in Table 4 and lift to drag ratios are provided in Figure 5. It should be mentioned that the hypersonic values correspond to the ballistic configuration without flaps deployed.

B. Propulsion

The on board propulsion system consists of both a de-orbit system and a reaction control system. The de-orbit system uses a hybrid propellant rocket motor system of similar type (though smaller) than the one used by SpaceShipOne. The hybrid motor propellants are hydroxl-terminated polybutadiene (HTPB) (fuel) and nitrous oxide (oxidizer). The cold-gas reaction control system (RCS) has sixteen 20 lbf thrusters to orient the vehicle and provide small translation maneuvers on-orbit. The reaction control system is also used during entry prior to flap deployment to provide control. The de-orbit system is sized to provide 158 ft/sec of ΔV and the RCS system can provide a total of 57 ft/sec of ΔV.

C. Environmental Control and Life Support Systems (ECLSS)

The environmental control and life support system (ECLSS) consists of an atmosphere monitoring and control system, carbon dioxide removal system, and a waste collection system. The galley is also included as part of the system. The atmosphere monitoring and control system maintains sea-level cabin pressure and monitors the oxygen partial pressure. The carbon dioxide removal system consists of simple lithium hydroxide canisters with the required fans and ducting. Lastly, though the orbital duration is only six hours, a waste collection system is provided and is shown in Figure 6.

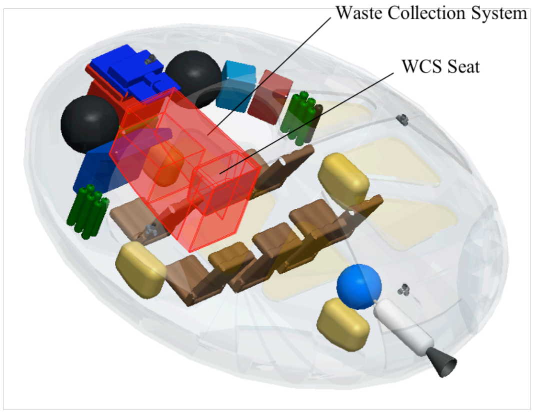

Figure 6. Waste collection system location.

D. Avionics

Avionics includes the following subsystems: (1) navigation & guidance, (2) vehicle health & monitoring, (3) data storage and processing, (4) communications, (5) range safety, and (6) the crew interface systems. The navigation system uses the Global Positioning System (GPS). Operational communications takes place mainly on S-band wavelengths. For passengers, two satellite phones are carried on board to allow them to call home and say "Hello" from orbit.

The vehicle is manually controlled, but entry and landing can be performed with an autopilot should the pilot become incapacitated. Two cameras are mounted on the forward part of the vehicle to provide stereo vision for the pilot. The images from the cameras are fed to goggles the pilot wears while flying the vehicle. Additional cameras provide exterior views for the pleasure of passengers.

E. Power

The electrical power system of the vehicle consists of a lithium-ion battery and a power management and distribution system (PMAD). The details of the power budget are listed below in Table 5.

Table 5. Power budget.

System	Power Consumption (W)	Ascent (hr)	On-orbit (hr)	Descent (hr)	Battery Energy (W-hr)
Command & Data Handling	800	1	5	1	5600
Avionics	475	1	5	1	3325
Communications	350	1	5	1	2450
ECLSS	610	1	5	1	4270
Power	1225	1	5	1	8580
Thermal control	123	1	5	1	860
Lighting	50	1	5	1	350
Amenities	96	1	5	1	672
				Margin	30%
Total power:	5.33 kW			Total energy:	34 kW-hr

F. Thermal Protection System

A key element of any entry vehicle is the design of the thermal protection system, as the heat loads encountered would nearly certainly melt any traditional structural material. However, for the lenticular shape, the intensity of the heating when entering from low-Earth orbit is mitigated because of the large radius of curvature for the windward portion of the vehicle. Using an approximation for the convective stagnation point heat rate[6], the predicted heating environment was calculated along the length of the entry trajectory. The results of those calculations are provided in Figure 7 below.

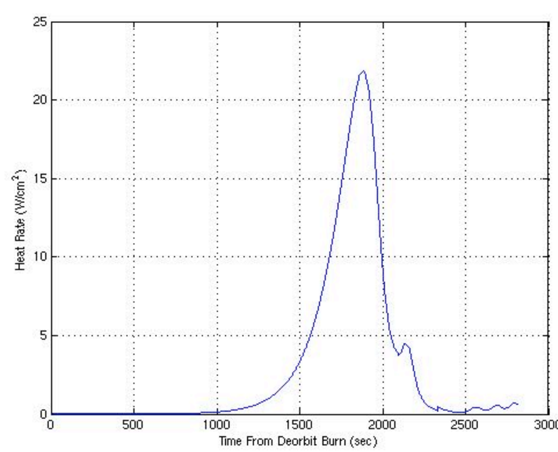

Figure 7. Convective heating for atmospheric entry.

The peak stagnation point heat rate of about 22 W/cm^2 is well within the range of reusable TPS materials. A list of candidate materials is provided in Table 6 below. The list is restricted to those materials that are already flight proven or nearly complete in their development. The TABI and

AFRSI-2200 entries represent thermal blanket style materials.

Table 6. Candidate reusable TPS materials with chosen materials highlighted[7].

Material	Density (kg/m^3)	Multi-use Temperature Limit (K)	Max Heat Rate (W/cm^2)	Purchase Cost ($\$/m^2$)	Reuse Flight Limit
AETB8/TUFI	128	1640	36	$12,500	100
FRCI-12	192	1640	38	$12,500	100
ACC	1600	1870	54	$129,000	100
RCC	1580	1920	60	$129,000	33
TABI	112	1480	24	$11,100	
AFRSI-2200	96.1	1310	14	$3,550	

The underside and perimeter TPS was selected as AETB8/TUFI. While meeting all of the thermal constraints, the AETB8/TUFI tile also provides low density and is a full order of magnitude cheaper than the carbon-carbon style TPS materials. For leeward thermal protection, the Advanced Flexible Reusable Surface Insulation (AFRSI) is baselined. Sizing the primary AETB8/TUFI TPS was based upon limiting the backwall temperature to that allowable for the required RTV-560 adhesive, about 600K.

G. Weight Breakdown

A summary of the weight estimations for the REV tourism concept are provided in Table 7 below.

Table 7. Weight breakdown of REV concept.

1.0	Body Group					3,860	lb
		1.1	Primary Structure	2,220	lb		
		1.2	Secondary Structure	120	lb		
		1.3	Windows	490	lb		
		1.4	Hatch	257	lb		
		1.5	Flaps	760	lb		
2.0	Thermal Protection					710	lb
3.0	Landing Gear					400	lb
4.0	De-orbit Propulsion					110	lb
5.0	RCS Propulsion					290	lb
6.0	Primary Power					500	lb
7.0	Electrical Conversion & Distribution					130	lb
8.0	Surface Control Actuation					50	lb
9.0	Avionics					730	lb
10.0	Environmental Control					480	lb
11.0	Personnel Equipment					490	lb
12.0	Dry Weight Margin					1,550	lb
	Dry Weight					9,280	lb
13.0	Crew and Gear					300	lb
14.0	Passengers					1,500	lb
15.0	Residual Propellants					20	lb
		15.1	De-orbit Propellant Residuals	20	lb		
		15.2	RCS Residuals	0	lb		
	Landed Weight					11,100	lb
16.0	De-orbit Propellants					220	lb
	Weight before De-orbit					11,320	lb
17.0	RCS Propellants (on-orbit)					120	lb
18.0	Inflight Cabin Air Losses					0	lb
	Gross Weight					11,440	lb

In support of the weight estimations provided several historical comparisons were made. Two of these are provided in Figure 8 and Figure 9. In both cases, the mass estimations generated for the REV are well inline with historical trends.

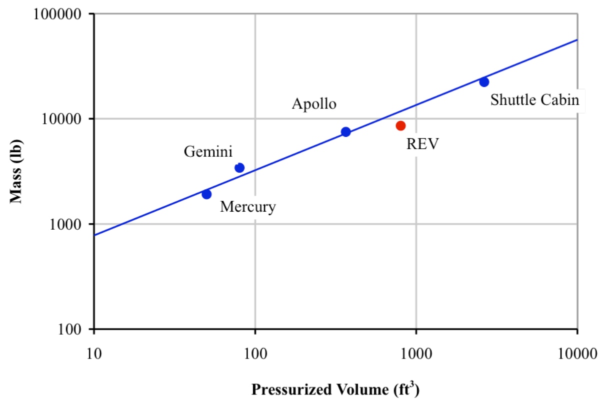

Figure 8. Structure and systems mass (dry mass minus TPS mass) comparison with historical data[1].

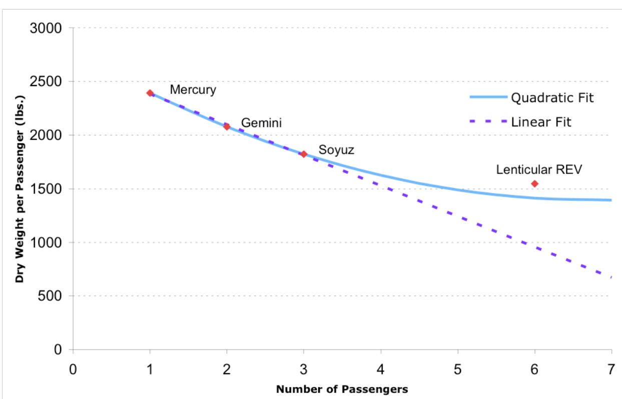

Figure 9. Historical comparison of dry mass per person.

IV. Mission Profile

A. Launch Vehicle

From the initial design space exploration it was determined that on such a limited budget, designing and developing a new launch vehicle concept to launch the lifting body would be unrealistic. Therefore, the launch vehicle will have to be purchased. After comparing the needs of our mission with existing launch vehicles and launch vehicle concepts, SpaceX's Falcon V was determined to be the best suited to achieve the goals. Although not yet in production, the Falcon V has been developed as a concept and has already attained a launch customer[8].

The Falcon V is a two stage, medium lift launch vehicle with the capability of reaching a 28° orbit with 6,020 kg (13,272 lb) of payload. According to SpaceX, the Falcon V has engine out capability and is therefore capable of having one first stage engine fail and still make it to orbit. Another safety feature of the Falcon V is SpaceX's hold-before-release system in which, upon engine ignition, the Falcon is held down until all propulsion and vehicle systems are confirmed to be functioning properly. These safety features greatly increase the reliability of the Falcon V and will help with calming customer concerns about safety.

The primary aspect of the Falcon V that makes it integral in the design of the REV is that at $15.8 million a launch, the Falcon V is much cheaper than other existing launch vehicles. In addition, SpaceX is already planning on human rating the Falcon V by 2010 so as to compete for America's Space Prize[9]. Human rating a vehicle can be a very expensive process and already having that done for the Falcon V means less expense for the space tourism business.

B. Integration

Since the proposed vehicle is larger than the payload fairing of the Falcon V, special considerations will have to be made when integrating the launch vehicle with the REV. The present baseline has the crew vehicle sitting vertically on top of the Falcon V with a specially designed fairing used to mitigate the impact on the aerodynamics. A conceptual image of this is provided in Figure 10, where the red represents the nominal Falcon V payload fairing. It should be noted that although the REV has a larger diameter than the Falcon V, the overall payload fairing is slightly shorter than the nominal fairing. This later becomes relevant when considering the additional loads placed on the Falcon V due to having a non-axisymmetrical payload attached.

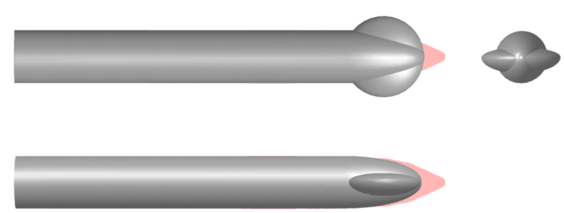

Figure 10. REV integrated with Falcon V.

Flying a launch vehicle with a lifting body on top of it presents some problems. In addition to the complication of having to develop a special fairing, the REV creates lift at the top of the rocket, in essence creating a bending moment and increasing the bending loads on the launch vehicle structure. However, even though flying a lifting body as a payload on a launch vehicle is not trivial, it is nonetheless not an entirely new concept. Several previous incarnations of lifting bodies have almost exclusively relied on standard launch vehicle concepts for orbit insertion. Examples include the early Dyna-Soar concept (Titan III), NASA Langley's HL-20 study (Titan IV), and ESA's Hermes study (Ariane V).

In an effort to analyze the additional loading placed on the Falcon V and insure that only minimal modifications would be required several analyses were performed. To insure that the change in aerodynamics from the original Falcon V would not be a problem, the ascent of the system was modeled in the Program to Optimize Simulated Trajectories (POST). In order to mitigate the torque generated from the added lift at the nose of the Falcon V, the q-α of the ascent was set to a maximum of 80% of the q-α of the nominal ascent. A 20% decrease in maximum q-α was chosen because upon addition of the REV to the Falcon V, the C_L curve increased by roughly 20% over the nominal case.

Finally, an abort system consisting of solid rockets which would carry the REV away from the launch vehicle in the event of an explosion was added to the REV to allow for pad abort capabilities. This 9000 lb system is jettisoned after 48 seconds.

Each of the load mitigation methods performed impacted the payload capability of the Falcon V. However, in the end there still proved to be ample capability for the calculated REV weight. A summary of the payload penalties calculated from POST is provided in Figure 11 below.

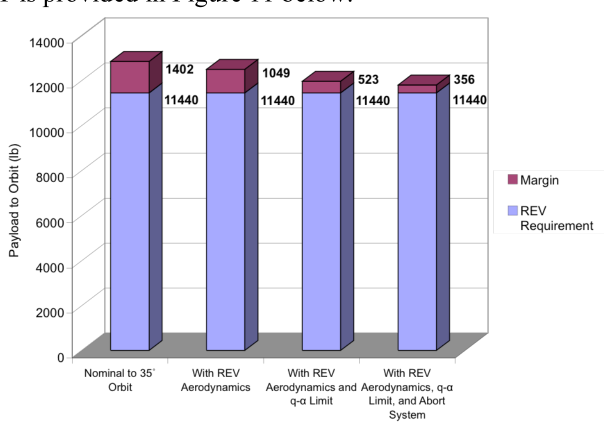

Figure 11. Payload to orbit for each trajectory modification.

C. On Orbit

From launch, the REV is placed into a 35°, 108 nmi. circular orbit. The baseline mission includes roughly three orbits for a total of almost five hours in space. A ground track of the orbit is provided in Figure 12.

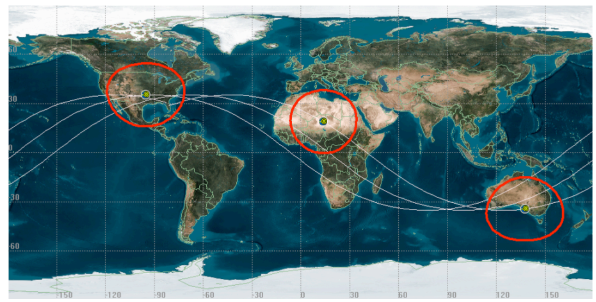

Figure 12. Ground track of baseline orbit.

The red circles in Figure 12 represent the horizon as would be seen from the REV in its orbit. The given orbit was chosen because after three orbits, it passes over central Florida and thus facilitates landing at Kennedy Space Center, minimizing the required crossrange and simplifying the entry trajectory. In the event that a landing at Kennedy is not possible (due to weather, for example), a landing at Mojave Spaceport in California is possible on the next orbit.

Once in orbit, the passengers will have the opportunity to unfasten their seat belts and float about the REV. The seats of the REV fold into the floor for added mobility. The REV will fly with windows pointed towards the Earth for the passengers to view and take pictures. Each passenger will have his or her own 5.3 ft^2 window.

D. Entry

The de-orbit portion begins with the ignition of the hybrid de-orbit engine. Roughly 25 minutes after the completion of the de-orbit burn the spacecraft begins atmospheric entry. In this phase of flight the vehicle is essentially performing a ballistic entry. The flaps are retracted, the vehicle is at an 80º angle of attack, and the only control is through the on-board RCS system. It is during this portion of the trajectory that peak heating and peak deceleration occur. Peak deceleration is limited to 2.88 g's. Roughly a minute after the vehicle passes peak deceleration it initializes flap deployment. With the flaps deployed, the REV transitions from an angle of attack of 80º to a statically stable angle of attack of 25º. The REV then glides to a runway landing at Kennedy Space Center. A summary of the entry profile is provided in below.

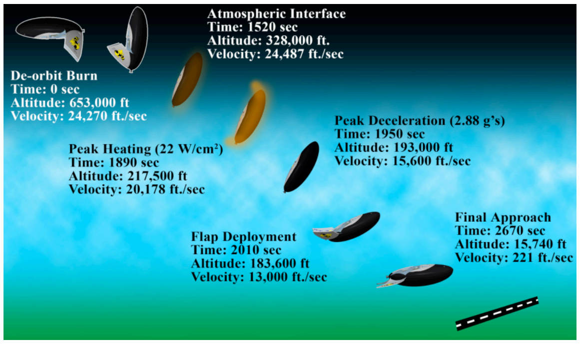

Figure 13. REV entry profile.

V. Concept of Operations

The overall concept of operations is pictured in Figure 14. Once passengers arrive in Florida, they are provided a flight on a commercial zero-gravity aircraft such as Zero Gravity Corporation's Boeing 727-200. There, passengers get a preview of the weightless experience they will have in orbit. The next day, the passengers arrive at Florida Spaceport and board the vehicle. Then, they travel into Earth orbit. Passengers spend 4 to 6 hours in orbit. Entry from orbit takes a little less than one hour.

Once the REV has landed in Florida, a crane loads the REV onto a trailer, and the REV is brought back to the hanger for refit. Several maintenance items need to be accomplished including: thermal protection inspection (80 manhours), de-orbit motor replacement and oxidizer tank fill (16 manhours), waste collection system refurbishment (4 manhours), RCS propellant fill (20 manhours), and atmospheric gas fill (4 manhours). In parallel with the REV refit, SpaceX is preparing the next Falcon V for launch.

If the REV must land in Mojave, California due to inclement weather in Florida, the REV is loaded onto a truck in Mojave and shipped to Florida over 3 to 4 days.

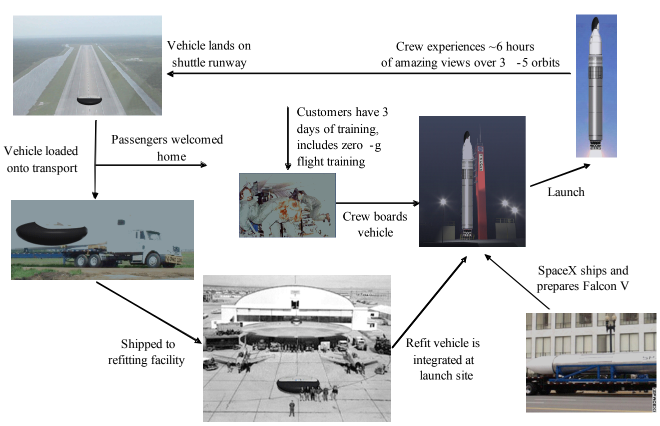

Figure 14. Operations flow chart.

VI. Aborts and Contingencies

Even with the efforts taken to insure launch vehicle reliability, it is impossible to insure the safety of the passengers without accounting for the possibility of a launch vehicle failure. A study of several different abort scenarios was thus performed. These abort scenarios were divided into three segments of the mission: launch pad, ascent, and on-orbit.

To assist in pad and early ascent aborts, a solid rocket based abort system is available for the first 48 seconds of ascent. The system itself is derived from the Apollo Launch Escape System (LES) and consists of four solid motors attached to hard points on the payload fairing, as partially shown in Figure 15. Each of the four motors weighs 2315 lbs and provides 27,500 lbs of thrust. The casings include 810 lbs of solid propellant, with the remaining mass dedicated to structure, avionics, and separation devices.

Figure 15. Early ascent abort engines.

In the event an imminent failure is detected, the launch escape system is activated. The activation sequence consists of first firing separation bolts that will detach the payload fairing and REV from the second stage structure. Nearly simultaneously, the abort engines are ignited and the REV, payload fairing, and abort system begin distancing themselves from the failing launch vehicle. As sized, the abort system provides a minimum separation distance from the launch vehicle of 328 feet at 2 seconds after ignition. In the likely event that the abort system is not utilized, the abort motors are jettisoned 48 seconds into flight to eventually splash down in the Atlantic.

In the event an anomaly is detected during ascent the command sequence to release the payload fairing and the REV will be activated. At this point, the REV will ignite its de-orbit engine to provide a small amount of separation from the launch vehicle. It should be mentioned that this abort situation assumes that the anomaly is not an immediately catastrophic failure. Examples of situations where this abort scenario would be valid would be the loss of several engines or if the launch vehicle began veering off course. A plot of downrange for several aborts is provided in Figure 16.

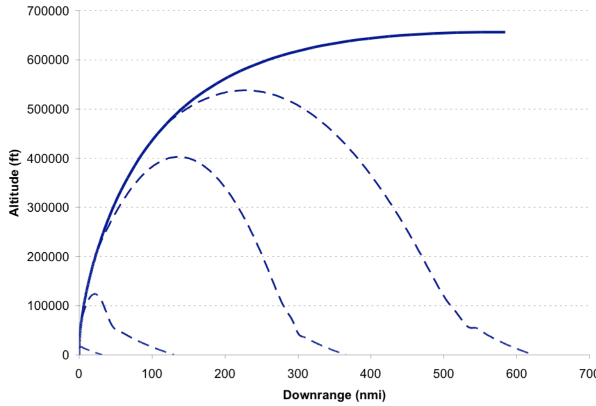

Figure 16. Downrange during an abort on ascent.

Part of the reasoning behind the choice of the number of orbits and the orbital parameters was to provide for the possibility of either an earlier or later de-orbit. In the case of the former, should the REV need to de-orbit ahead of schedule, it will never be more than 90 minutes away from being able to perform a burn that will allow for a landing in the United States. In the case of the latter, the REV has the capacity for two additional orbits beyond the nominal three while again still providing for a landing in the United States. In the case of one additional orbit, the entry trajectory brings the vehicle directly over Mojave spaceport. In the case of two additional orbits, though the orbit track has moved off of Mojave, the REV still has sufficient crossrange to maneuver towards Mojave.

VII. Cost and Economics

Economic analysis was performed using demand curves from the recent Futron space tourism market study[10]. The demand curves were enclosed within a modified version of the Launch Market for Normal People (LMNoP) business analysis tool[11]. Roughly 13.5 million simulations were performed over conservative business cost ranges to develop Response Surface Equations (RSEs) for each business metric. Monte Carlo statistical treatment allowed for RSEs that returned results with a certainty of 90%. Business feasibility and sensitivity were then studied.

For the Design, Development, Test and Evaluation (DDT&E) Cost and the Theoretical First Unit (TFU) cost estimation, the TRANSCOST[12] estimation tool was utilized. TRANSCOST offers well-developed system level Cost Estimation Relationships (CERs) designed for use in the initial conceptual design phase of a project.

Table 8. Bounds for examined economic space.

Variable	Low bound	Nominal	High bound
DDTE ($M)	60	70	100
TFU ($M)	25	30	45
Launch Cost ($M)	20	25	40
Reliability (%)	99	99.5	99.9
# Passengers	3	5	6
Fixed Yearly Costs ($M)	5	10	20
Ticket Price ($M)	5	13	20
Market	33%	100%	100%

The cost breakdown for the estimated recurring cost per flight is shown below in Table 9. The breakdown of the company's projected fixed annual costs is outlined in Table 10. With a vigorous development program, we aim to continually evolve the quality of our product and develop next generation concepts to maintain our lead over the competition. The costs associated with launch pad modifications are neglected under the assumption that the Florida Space Authority will provide the necessary improvements.

Table 9. Cost breakdown for recurring cost per flight.

Item	Cost ($M)
Cost of Falcon V (shipping and rocket support included)	16
Equipment and crew used at landing and abort locations	0.1
Integration to Falcon 5	3
Range fee	1
License fees and legal requirements	1
Insurance Fees	1
Refit at servicing facility	2.65
Analysis of flight for future development	0.25
Total	25

Table 10. Cost breakdown for fixed annual costs.

Item	Cost ($M)
Construction, testing, and refitting facility	2
Total Payroll	3
Insurance	1
Legal fees	1
Promotion of business	2
DDTE on future improvements	1
Total	10

The analysis has led to the conclusion that Grinning Torch should aim for an initial ticket price target of 13 million, in line with the nominal expectations for expenses and revenue. Business results for the nominal business case appear in Table 11.

Table 11. Nominal business case metrics (FY05).

Business Metric	66% Market	100% Market
10 year IRR (%)	70	95
NPV ($M)	402	598
Max Exposure ($M)	239	202
Revenue ($M)	13520	18560
ROI (%)	31	40
Recurring Cost ($M)	8620	12105
Breakeven year (IOC 2010)	2011	2011

The figure below, Figure 17, displays the stochastically most likely cash flow scenario given the expected nominal business variables. This includes the system reliability of 0.995. The statistically expected flight rate, for these nominal values, follows in Figure 18. In Figure 18 the flight rate in 2009 starts at two. This number comes from the company's initial test program that also accounts for the decrease in cash flow seen during that year.

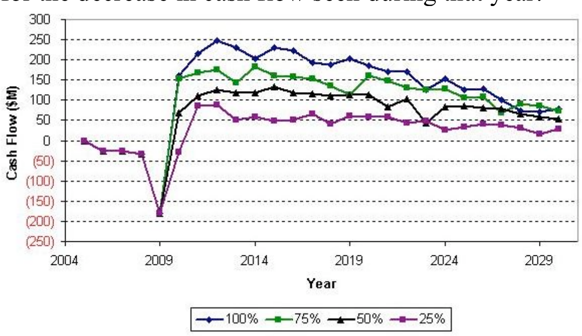

Figure 17. Mean Cash Flow: Yearly Cash Flow by Market Capture (nominal).

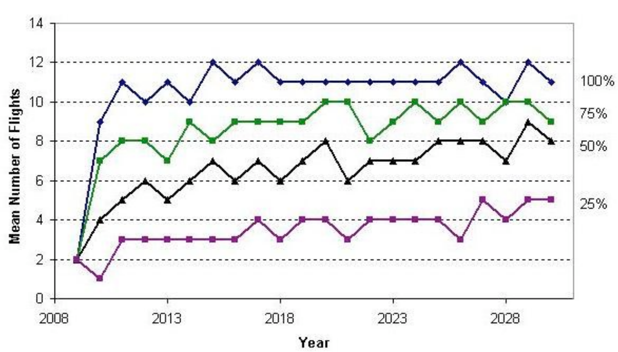

Figure 18. Mean number of flights by Market Capture (nominal).

VIII. Educational Outreach

As part of the outreach efforts of this project, the authors had the privilege of making visits to Northwestern Middle School in Alpharetta, Georgia. During these visits, the authors presented an outlook on space tourism, provided lessons in the environment of space and weightlessness, and introduced the students to basic rocket propulsion concepts. The latter was done through a series of activities with balloons, straws and string and also through demonstrations of a variety of configurations of model rockets.

Figure 19. Introduction to balloon propulsion.

Figure 20. Model rocket propulsion demonstrations.

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