Fisher Space Systems, LLC

Phoenix Rocket Glider

by
Jerry F. Fisher

1.0 Introduction

In the 1960s, the application of the inflatable paraglider concept as a recovery device for expended booster, rockets, manned spacecraft, and instrument payloads from orbital or suborbital flight was considered by NASA and the Air Force. Both experimental and analytical investigations were conducted to evaluate the capabilities of the paraglider as such a device. The results of these studies are reported in references 1 to 8.

It was proposed in the 1960s that the paragliders would use woven metal fabrics impregnated with high temperature silicone rubber as both an ablative coating and as an air barrier. This type of material was suitable for small re-entry spacecraft. However, as the size of the re-entry vehicle and paraglider increased, the weight of the keel and boom reached a limiting value and became impractical. Also, the race was on to reach the moon before Russia and much of the R&D funding for re-entry went to lower risk and proven technologies.

Figure 1. NASA Gemini Paraglider

Years of recreational sport with hang gliders based on the Rogallo flexible wing design, advances in technologies, and improvement in materials has resurrected the inflatable paraglider concept as a recovery device. The time is right for a privately owned, operated, and maintained recreational ground launched rocket glider. This paper presents such a concept.

The Phoenix RG is a single place manned rocket glider. Two advances make the Phoenix possible. First, a standard design pressure fed hydrogen peroxide (HTP)/E85 rocket engine using mixed metal oxides (MMO) as the catalyst. HTP is room temperature storable and high density. As such, the Phoenix can be made to be low cost, simple, and reusable.

The second advance is the flexible thermal protection system (FTPS) developed by NASA (Calomino, A. M., et el, 2012). The 1st generation FTPS can withstand heat fluxes of 25

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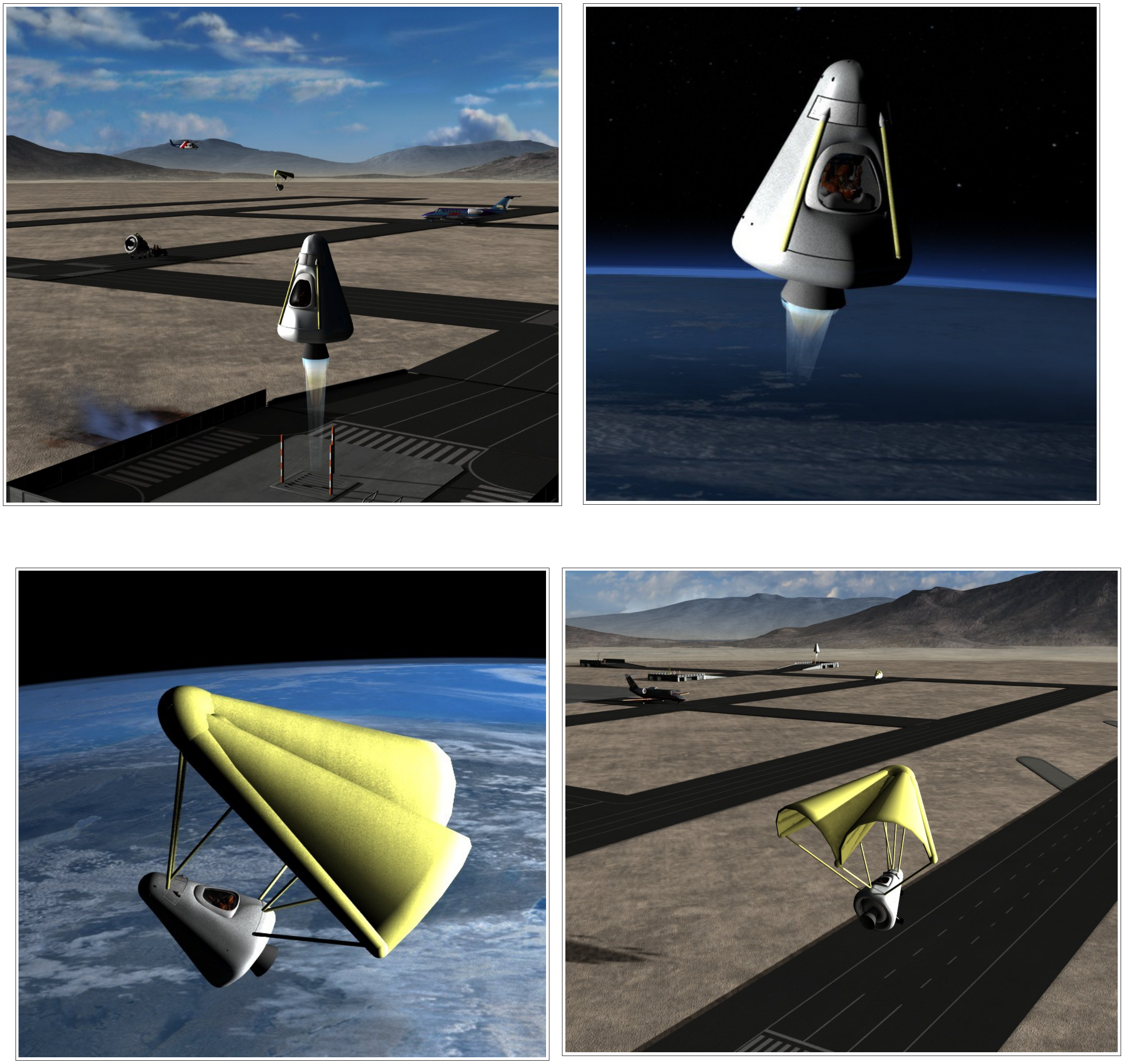
W/cm^2 and the pyrogel insulator layer can absorb energies up to 5000 J/cm^2. For the Phoenix RG, an FTPS type material can be used for an inflatable keel, two inflatable booms, and the canopy for a re-entry paraglider.



Software: Hexagon and Daz3d
Daz3d Renders

Figure 2. Phoenix Launch and Recovery

Also, with HTP/E85 at combustion pressures of 100 psi, the wall temperature is approximately 1200°C with a heat flux of 24 W/cm^2. As such, the rocket engine can be made reusable by using a high melting temperature metal alloy in the combustion chamber, throat, and nozzle which is backed by an FTPS type refractory cloth and insulator. With the energy absorbing capability of the insulator, run times of about 200 seconds are possible before a

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cooling off period is required.

2.0 Phoenix RG

The Phoenix RG is single place manned ground launch rocket glider. The Phoenix RG will thrust for approximately 180 seconds, coast to an altitude of 25-50 km, inflate its paraglider, and glide for about 14 minutes to a conventional runway landing. The oxidizer for the single rocket engine is hydrogen peroxide using a mixed metal oxide to catalyze the HTP and the fuel is E85. The HTP will be produced on site and the pilot will have a choice of HTP blends for the oxidizer: 85/15, 90/10, or 98/2. The 85/15 blend will take the pilot to about 25 km while the 98/2 blend will take the pilot to about 50 km. The flight plan shown below is based on the 98/2 blend. The propellants are room storable and readily available.

A suitable material for the paraglider canopy could be Kevlar or Vectran coated with a sealant. If used for the canopy, the flexible thermal protection system would have an aerial density of about 5 kg/m^2. As such, the mass of the keel, canopy, and two booms for the Phoenix RG is about 740 kg. The gross liftoff mass for the Phoenix RG is about 7143 kg. Landing speed is about 25 m/sec, the landing speed of a Cessna Club. Since piloting a suborbital launch vehicle requires, as a minimum, an instrument rating, the future astronaut can step out of a Cessna and into the Phoenix with minimal training.

2.1 Phoenix RG

The concept design for the Phoenix RG is shown in figure 3 below. The mass estimate for the Phoenix is derived using the mass estimating relationships compiled by Georgia Tech (Rohrschneider, R.R., 2002).

The Georgia Tech database uses a number of different relationships to estimate the mass of individual components for the flight vehicle. Georgia Tech recommends that the user select the relationship that results in the closest comparison to a known flight component. The main fuselage with crew cabin of the Phoenix RG is more closely related to a Mercury space capsule. Therefore, the relationships resulting in the correct weight for the Mercury space capsule will be used for the Phoenix.

Individual components for the Mercury space capsule were selected from the George Tech database, the weights were determined, added together, and compared to the end of reentry weight for the Mercury space capsule No. 20 (Project Mercury, 1963). The weight using the Georgia Tech database shows approximately a 2% difference from the end of reentry weight of the Mercury space capsule (Fisher, J. F., 2012). These relationships, as identified by the reference numbers, were used for the individual components of the Phoenix RG.

The main difference in design between Phoenix and Mercury is the method of launch (i.e. ground launch versus on top of a booster), the addition of the inflated paraglider for re-entry, the propellant tanks, the HTP/E85 rocket engine and nozzle, and the duration of flight and the recovery method.

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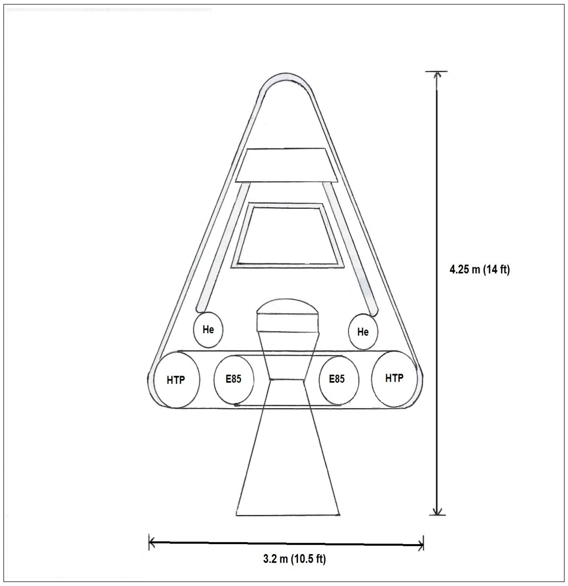

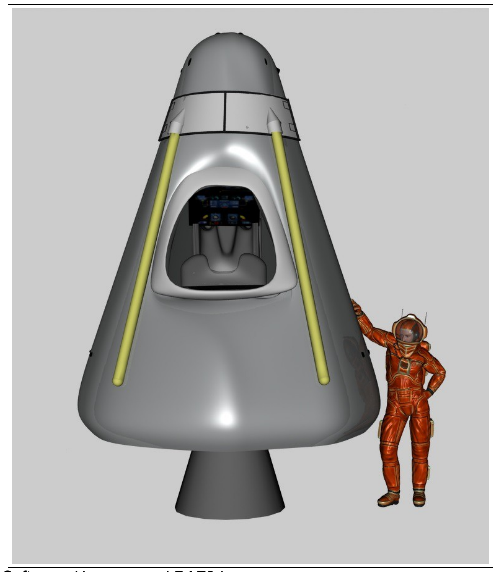

Software: Hexagon and DAZ3d
Astronaut: Mars Explorer for Genesis

Figure 3. Phoenix RG

The Phoenix RG will launch from a ground based launch site, boost under thrust, coast to a peak altitude of about 50 km, deploy the paraglider, return to launch site, and land on a conventional runway. The mass estimates are shown in table I below.

Table I. Phoenix Estimated Weight

Component	Reference	Weight Estimate (lb)
Wing Group	2.2 Paraglider	745
Body (Fuselage & Cabin)	6	2341
Thermal Protection	6	183
Landing Gear, Pwr, & ECL	1	889
Propulsion, ECD, & SCA	3	946
Avionics & PE	4b & 7	437
Total (Sum of 1st 6 rows)	Dry Weight	5541
Dry Weight Margin	10.00%	554
Crew & Gear	4b	256
Propellants	1, 2 & 10	9397
GLOW (GLOM)		15748 (7143 kg)

2.2 Inflated Paraglider

In the 1960s, the application of the inflatable paraglider concept as a recovery device for expended booster, rockets, manned spacecraft, and instrument payloads from orbital or

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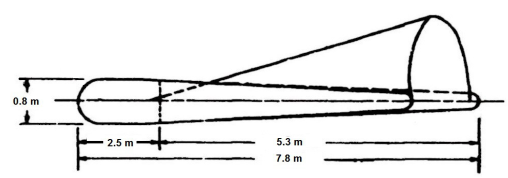
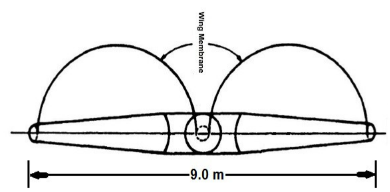
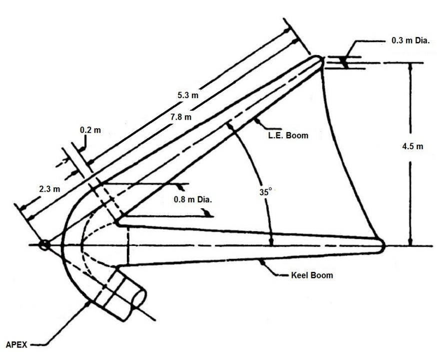
suborbital flight was considered by NASA and the Air Force. The paragliders of the 1960s used woven metal fabrics impregnated with high temperature silicone rubber as both an ablative coating and as an air barrier. This type of material was suitable for small re-entry spacecraft. However, as the size of the re-entry vehicle and paraglider increased, the weight of the keel and boom reached a limiting value and became impractical.

The flexible thermal protection system with its refractory cloth, insulator, and air barrier have an aerial density of about 5 kg/m^2. As such, the mass of the keel, canopy, and two booms for the Phoenix RG is about 740 kg and packs in a volume of less than 0.4 m^3. The keel and two booms are about 5.3 m in length. They are tapered with the forward section 0.8 m in diameter and the rear section 0.3 m in diameter. Concept drawings of the keel, canopy, and two booms are shown in figure 4 below.







Specifications

Length	7.8 m
Wing Span	9.0 m
Sweep Angle	55 deg
Planform Area	45 m^2
Wing Loading	4900 N/m^2

Figure 4. Phoenix RG Canopy

2.3 Aerodynamics

Analysis of the aerodynamic characteristics for the inflated paraglider follows a similar analysis as presented in reference 8, “Semi-Rigid or Non-Rigid Structures for Re-Entry Applications”. Analytical estimates of the pressure distribution at hypersonic speeds were made using Newtonian concepts.

The Newtonian theory assumes that the entire vertical component of stream

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momentum is given up to the glider. As such, the following equation results:

$$C_p = 2\sin^2(\theta) \qquad \text{(Anderson, J. D., 2006)}$$

where θ is the local flow deflection angle. The deflection angle for the canopy is given by $\alpha - \tan^{-1} dz/dx$ (Keville, J. F., 1967). Alpha is the angle of attack and dz/dx is the slope of the curve defined by the canopy in the x, z plane. The local flow deflection angles were computed by assuming that each wing membrane inflated to the shape of half of a right-circular cone, as shown in figure 5 below.

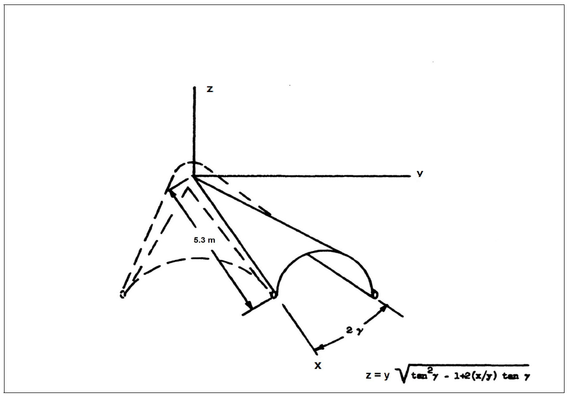

Figure 5. Paraglider Canopy

The equation for the right-circular cone as a function of x and y is given by,

$$z = y\sqrt{\tan^2\gamma - 1 + 2\left(\frac{x}{y}\right)\tan\gamma}$$

where γ is the angle of the right-circular cone. As such, dz/dx is given by,

$$\frac{dz}{dx} = \frac{\tan\gamma}{\sqrt{\tan^2\gamma - 1 + 2\left(\frac{x}{y}\right)\tan\gamma}}$$

The streamwise airfoil sections are parts of parabolas formed by cutting vertical planes parallel to the keel (i.e. planes in the x, z coordinate). The leading and trailing edges of the parabolas are described by the expressions:

$$x_{LE} = \frac{y}{\tan 2\gamma}$$

and

$$x_{TE} = l - y\tan\gamma$$

Taking the derivative, dz/dx, the pressure coefficient at any point on the panel is:

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$$C_p = 2\sin^2(\alpha - \tan^{-1}\frac{dz}{dx})$$

The pressure coefficient was integrated over the periphery of each cross-section by means of a numerical integration. At each incremental area on the parabola defined by the airfoil section, the normal and axial coefficients are calculated using the expression,

$$C_n = dA\,C_p \cos(\tan^{-1}\frac{dz}{dx})$$

$$C_d = dA\,C_p \sin(\tan^{-1}\frac{dz}{dx})$$

where dA is the incremental area equal to 2.3 x 10^{-6} m^2. For each angle of attack, α, the integration began by setting y=0 and calculating x_{LE}, the leading edge, and x_{TE}, the trailing edge. While the leading edge was less than the trailing edge, the normal and axial coefficients for the parabola defined by the vertical plane at y=0 was calculated. The y coordinate was advanced by 1.5 mm and the pressure distribution for the next slice of parabola was determined.

For the concept defined above, y went from 0 to 3.0 m. At the end of the run, the normal and axial coefficients had units of square meter. As such, the coefficients were divided by the canopy area defined by the half-circular cone.

Calculation of the air load distribution on the leading edge booms was made using Newtonian theory which presents the expression (Keville, J. F., 1967):

$$C_p = 2\cos^2(\theta - \alpha)\cos^2(\delta\cos(\alpha))$$

where θ is the local angle of the leading edge boom cross section relative to the x axis, and δ is the leading edge boom sweep angle measured from the y axis. A plot of the pressure distribution for α = 70° is shown in Figure 6. At a given angle of attack, the pressure distribution shown in Figure 6 is applicable to all cross sections of the boom. The pressures were numerically integrated over the periphery of each cross-section.

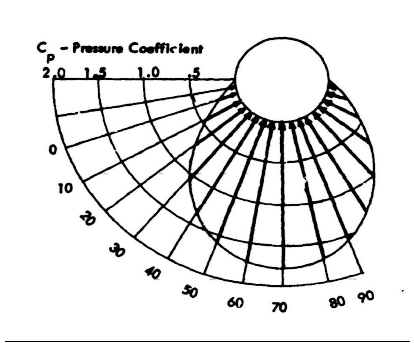

Figure 6. Pressure Distribution on Boom

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With the spanwise pressure distributions established over the canopy and boom, the resulting aerodynamic coefficients and the L/D ratio during hypersonic re-entry were determined as functions of angle-of-attack and are shown in Figure 7 below.

The analysis using the Newtonian theory compares favorably with experimental values obtained at Mach 4.5 using a scale model of a flexible canopy paraglider in the hypersonic facility at the Langley Research Center (Wornom, Dewey E., and Taylor, Robert T., 1963). The variation of these coefficients with Mach number is small in the region of hypersonic re-entry velocities considered. A FORTRAN program handles the numerical integration and is included in the appendix.

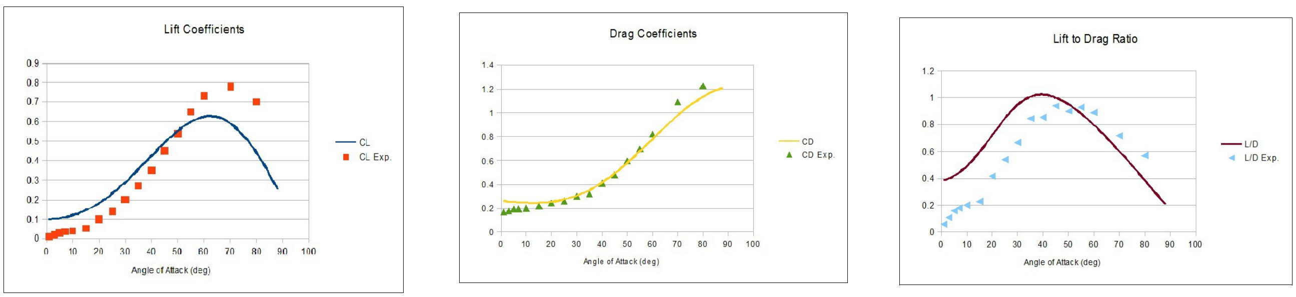

Figure 7. Aerodynamic Coefficients of Phoenix Paraglider

Consideration of a parametric study of the dynamic pressure on the boom indicated that minimum loading occurs near maximum lift coefficient. As a result, a re-entry angle of attack of 70°, which corresponds to a L/D of .6, was selected for the trajectory analysis.

2.4 Trajectory

At an angle-of-attack of 70°, the maximum heat flux and total energy absorbed are approximately 0.2 W/cm^2 and 13 J/cm^2 respectively. For the paraglider, the critical parameter is the dynamic pressure across the boom. It is the dynamic pressure which determines the loading and thus, the inflated pressure for the boom.

2.4.1 Boom Loading

The vertical component of the air load directly on the boom was determined from the boom air loads, the maximum dynamic pressure, and the acceleration for an angle-of-attack of 70°**.** By definition the dynamic pressure is given by,

$$q=\frac{1}{2}\rho v^{\frac{1}{2}}$$

where ρ is the freestream density and v is the freestream velocity. As shown in figure 8 below, the maximum dynamic pressure is about 350 N/m^2. The maximum acceleration is shown in figure 9.

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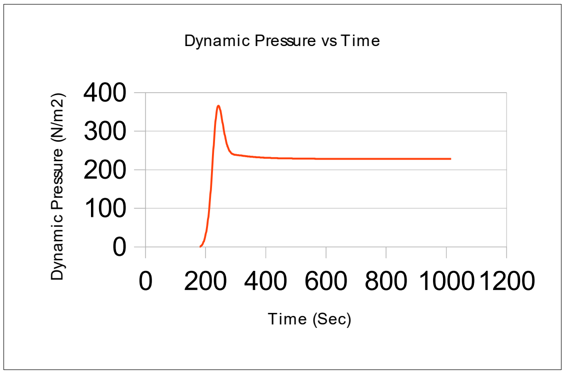

Figure 8. Dynamic Pressure

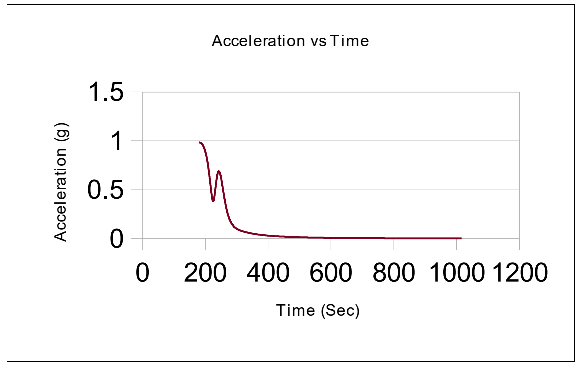

Figure 9. Acceleration

The total boom loading was determined by adding the loading due to membrane, direct air loads, and inertial loads. The maximum load is about 600 N/m and is shown in figure 10 below.

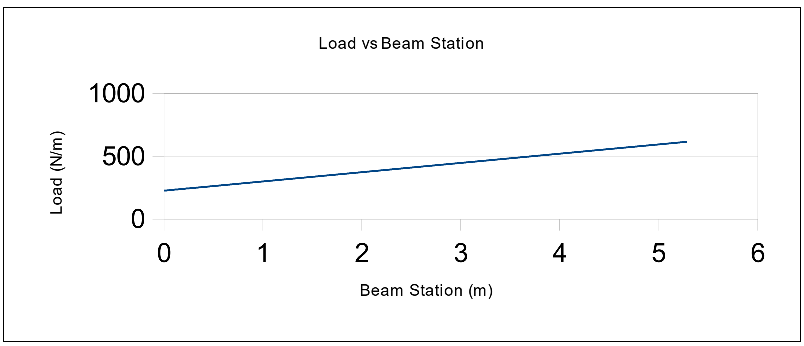

Figure 10. Combined Load

The boom loading was used to determine the maximum deflection at the midpoint of the boom. The boom is supported by metal wires surrounded by inflated beams connecting its front and its end. The uniform load was taken as the maximum loading equal to about 600 N/m. The deflection was modeled by assuming simple support – uniform load which presents the equation (Shigley, J. E. and Mitchell, L. D., 1983) ,

$$y_{max} = \frac{5wl^4}{384EI}$$

where w is the load equal to 600 N/m, l is the length of the boom equal to 5.3 m, E is the modulus of elasticity equal to 1.0 x 10^{10} N/m^2, and I is the area moment of inertia for a thin walled cylinder with an average radius of .34 m and thickness of 3.0 mm equal to 3.8 x 10^{-4} m^4. As a result, the maximum deflection at the midpoint of the boom is about 1.6 mm.

The modulus of elasticity was determined from experiments using a commercially available bi-axial weave Kevlar sleeve with a Mylar air barrier (Fisher, J. F., 2013). The results

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from the in house experiments agree closely with the results presented by Georgia Institute of Technology, Space System Design Lab (Hutchings, Allison L. and Braun, Robert, 2009). The test at Georgia Tech included urethane coated, silicone coated, and kapton coated Kevlar. Both the in-house test and the Georgia Tech test indicate that the Young's modulus for a Kevlar inflated beam is an order of magnitude less that that of the individual fiber. Silicone coated Kevlar was used for the Inflatable Re-Entry Vehicle Experiment (IRVE). IRVE was a test flight of a stacked toroid inflatable decelerator concept. As such, Kevlar or Vectran coated with a sealant could be a suitable material for the Phoenix RG.

2.4.2 Heat Flux and Total Energy

The stagnation point heat flux and total energy absorbed are shown in figures 11 and 12 below. The heat flux is given by (Anderson, J. D., 2006, p.349),

$$q_w = \rho^N v^M C$$

where ρ is the local freestream density, v is the freestream velocity, N = 0.5, M = 3.0, and C = 1.83 x 10^{-8} $R^{-1/2}$. This form of the heat flux is the simplest method for estimating hypersonic aerodynamic heating and is a good approximation for conceptual designs.

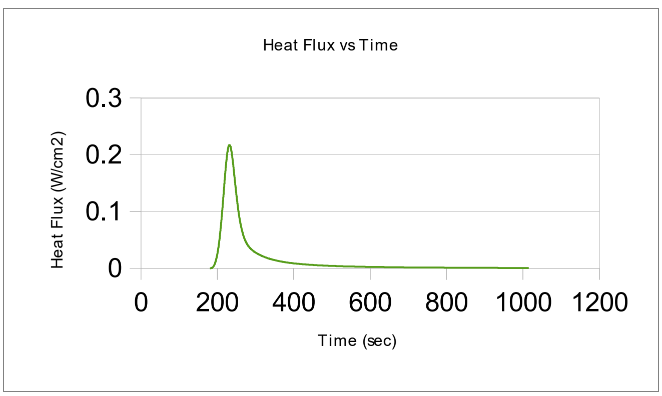

Figure 11. Stagnation Point Heat Flux

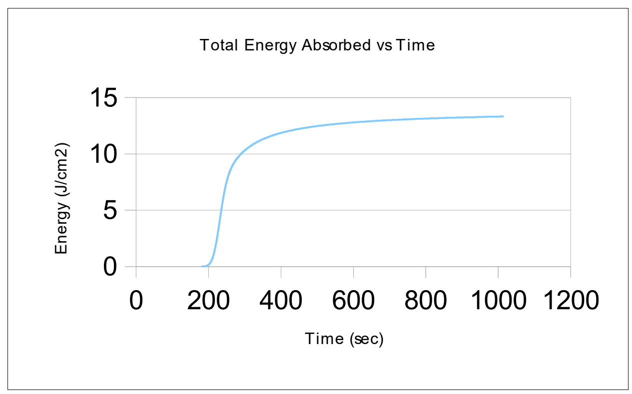

Figure 12. Total Energy Absorbed

The stagnation point heat flux is inversely proportional to the square root of the radius. In other words, the smaller the radius the greater the heat flux. Connecting the Phoenix RG to the paraglider using only small diameter wires could produce heat fluxes and temperatures high enough to weaken or melt the support wires. Therefore, the support wires are enclosed by inflated beams made of the same FTPS as the booms and canopy and are about 10 cm in diameter. As such, the maximum stagnation point heat flux is about 0.2 W/cm^2 and the total energy absorbed is about 13 J/cm^2, well within the capability of the FTPS under investigation by NASA.

2.5 Flight Plan

The Phoenix RG is single place manned ground launch rocket glider. The Phoenix RG

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will thrust for approximately 180 seconds, coast to an altitude of about 50 km, inflate its paraglider, and glide for about 14 minutes to a conventional runway landing. Total flight time is about 17 minutes. The flight plan is shown in figures 13 and 14 below.

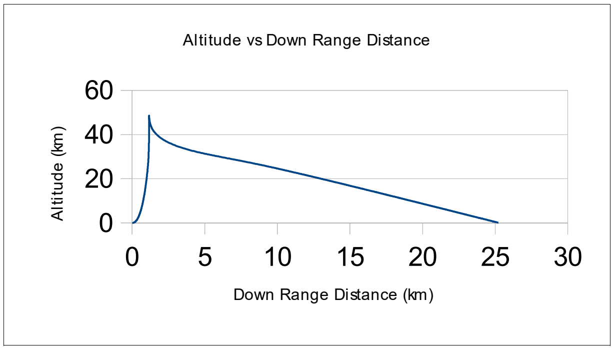

Figure 13. Down Range Distance

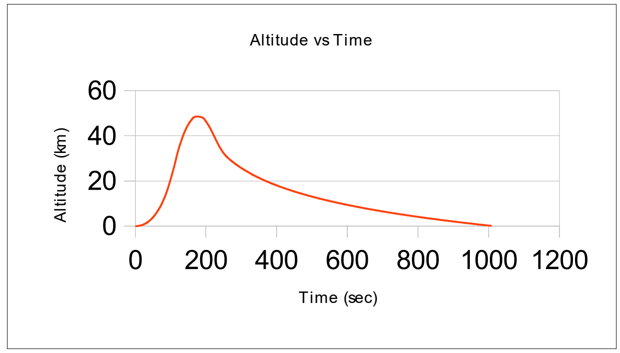

Figure 14. Flight Time

As shown in figure 13, the down range distance at landing is about 25 km. The Phoenix RG can fly to any nearby runway or return to the launch point. To return to the launch point, Phoenix can execute a 2g turn at about 12 km down range and still have enough range to make it back to the launch point. Approach speed is about 50 m/sec. Flaring would reduce the velocity to about 25 m/sec, the landing speed of a Cessna Club. Since piloting a suborbital launch vehicle requires, as a minimum, an instrument rating, the future astronaut can step out of a Cessna and into the Phoenix with minimal training.

2.6 HTP/E85 Rocket Engine

The propellant for the single rocket engine is hydrogen peroxide and E85, both are readily available. The HTP passes through a mixed metal oxide catalyst consisting of cobalt, manganese, and aluminum oxides. Cobalt and manganese are transition metals which act as catalyst for the HTP. The aluminum adds structural support to the MMO. A preliminary design of the HTP/E85 rocket engine is shown in figure 15 below. Shown in figure 16 is the rocket engine surrounded by the toroid propellant tanks. The HTP will be produced on site and the pilot will have a choice of HTP blends for the oxidizer: 85/15, 90/10, or 98/2. The 85/15 blend will take the pilot to about 25 km while the 98/2 blend will take the pilot to about 50 km. The flight plan shown above is based on the 98/2 blend.

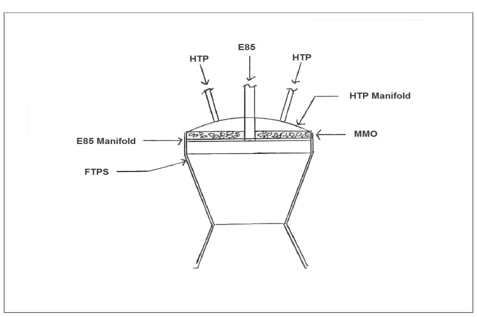

Figure 15. HTP/E85 Rocket Engine

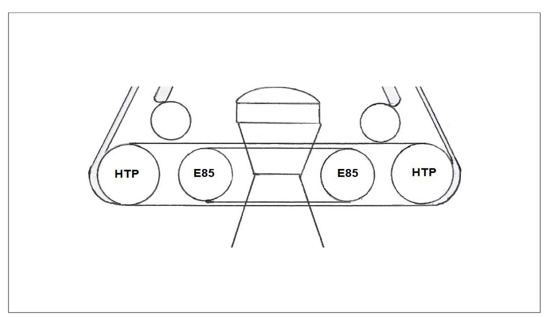

Figure 16. Propellant Tanks

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There are several papers (ref. 17-20) which outline the process for developing the MMO catalyst. The process involves three different oxides each with their own phase diagrams and thermodynamic properties. It can't be done theoretically and must be done experimentally (Kovanda, F., et el, 2006). The process involves the co-precipitation of cobalt, manganese, and aluminum hydrates at the proper mole ratios while keeping the ph level constant. The slurry is then dried, crushed, peptized and calcinated in an oven. The resulting pellets have large surface area to volume ratios and can withstand temperatures over 1100°C. When used as a catalyst for HTP, the MMO pellet result in a shorter combustion chamber and smaller pressure drop across the injector as compared to silver screen catalysts.

The MMO catalyst has been used for durations of up to 5 seconds with 85% HTP. The rocket engine test was at a chamber pressure of about 50 psi and a mass flow rate of about 50 gm/s. Further research and development is required to determine if this holds for durations of 150 to 200 seconds at mass flow rates of 50 kg/s and chamber pressures of 100 psi. A crowd source funding campaign is in the works to continue the research.

The HTP/E85 rocket engine is constructed of a high melting temperature metal alloy (e.g., TZM Molybdenum) and cooled by a flexible thermal protection system wrap. The FTPS surrounds the combustion chamber, throat, and nozzle. At a chamber pressure of 100 psi, the combustion temperature is about 2600 K. The wall temperature is given by,

$$h_{gc} (T_{aw} - T_{wg}) = \varepsilon \, \sigma \, T^4_{wg}$$

where T_{aw} is the combustion gas temperature equal to 2600 K, T_{wg} is the hot side wall temperature, ε is the emissivity equal to .95, and σ is a constant equal to 5.67 x 10^{-8} W/m^2-K^4.

The average heat transfer coefficient, h_{gc}, is a function of the Reynolds number which is itself a function of combustion gas density, velocity, and viscosity. For the combustion parameters under consideration, the average heat transfer coefficient is approximately equal to 218 W/m^2-K. As such, the hot side wall temperature is between 1100–1200°C. This results in a heat flux of 24-26 W/cm^2. The total energy absorbed for a 180 sec run time is about 4320-4680 J/cm^2, within the capabilities of a first generation FTPS.

2.7 HTP/E85 Propellant Tanks

Both the HTP oxidizer tank and the E85 fuel tank are lightweight composite tanks made of glass fiber with a thin walled stainless steel liner. The tanks will be toroidal in design at a pressure of 400 psi for 25+ cycles. The design is based on technologies that have been around since the 1960s (Sanger, M. J., et. El, 1966).

In a 1966 technical report, Aerojet General Corporation under contract with the Air Force, design, fabricated, and tested four, stainless-steel-lined, filament-wound tanks for the containment of nitrogen tetroxide (N_2O_4). The requirement specified storage for 30 days at an operating pressure of 350 psi and at a temperature of 110° F. The tanks were designed for a burst pressure of 761 psi with 25 cycles without failure. With the exception of burst pressure (750 vs 761 psi), all of the design requirements where met. The weight of these tanks is

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consistent with the mass estimating relationships compiled by Georgia Tech (Rohrschneider, R.R., 2002) with a technology reduction factor of 0.38.

3.0 Conclusion

In the 1960s, the application of the inflatable paraglider concept as a recovery device for expended booster, rockets, manned spacecraft, and instrument payloads from orbital or suborbital flight was considered by NASA and the Air Force. The paragliders in the 1960s used woven metal fabrics impregnated with high temperature silicone rubber as both an ablative coating and as an air barrier. As the size of the re-entry vehicle and paraglider increased, the weight reached a limiting value and became impractical.

Years of recreational sport with hang gliders, advances in technologies, and improvement in materials have resurrected the inflatable paraglider concept. The time is right for a privately owned, operated, and maintained recreational ground launched rocket glider.

The two advances that have made the Phoenix possible are the (HTP)/E85 rocket engine using mixed metal oxides as the catalyst and the flexible thermal protection system developed by NASA.

The Phoenix RG is a single place ground launched rocket glider. The Phoenix RG is designed to be a completely reusable manned suborbital launch vehicle consisting of a single pressure fed HTP/E85 engine with a mixed metal oxide catalyst, fiber reinforced stainless steel propellant tanks, a durable fuselage, an inflatable paraglider for controlled reentry, and a tricycle landing gear for a conventional runway landing. The mixed metal oxide catalyst and the inflatable paraglider are the only advanced concepts for the system. The remaining components will follow standard design procedures which require trial and error testing and fine tuning to achieve.

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Daz3d Renders

Dry Mud Desert, Aako, Scene-subset
Streets, MaClean, Scene-subset
Jepes Flame, Jepe, Scene-subset
Pocket Star, Author Unavailable, Scene-subset
Army Vehicles, Bricabrake, Scene-subset
Private Jet C !Classic, Perry-walinga, Scene-subset
Rotor Heads Gdds76 Helicopter, DarMatter, Scene-subset
Mars Explorer for Genesis, Midnight-Stories, Scene-subset

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APPENDIX

```
REAL AOA, GAM, DELT, DTR
!angle of attack, right circular cone angle, sweep angle, and deg to rad conv.
REAL XLE, XTE
!non-dimensional leading & trailing edge.
REAL DZDX, CP, CN, CA, CL, CD, LOD
!dzdx, pressure, normal, axial, lift, drag, & lift over drag coeffiecients.
REAL CNAREA, CAAREA, KNAREA, KAAREA, BNAREA, BAAREA, APNAREA, APAAREA, DEL
!total normal and axial coefficients, integration width.
REAL CSNAREA, CSAAREA
!Canopy planes parallel to the x, z axis.
REAL THT, Y, BLC, L
!angle subtended around boom, y-axis and boom line coordinates, length of boom.
REAL R, TEMP1, TEMP2
!radius of boom at BLC, temporary variables for boom normal and axial areas.
REAL X, Z, S, F, dA, SAC, SAB, SAA
!coordinates for radial distance, arc length on the ellispe, resultant forces.
!incremental area, surface area of canopy and boom.
REAL, PARAMETER:: PI=3.1415927
!OPEN(1,FILE='Phoenix Coefficients.txt')
!WRITE(1,*) 'AOA(deg) ','CL ','CD ','L/D '
OPEN(1,FILE='DATA.txt')
L=17.35
DTR=.0174533
AOA=1*DTR
DELT=55*DTR
GAM=17.5*DTR
SAB=54.
!Surface area of boom.
SAC=185.+.5*SAB
!Surface area of canopy.
SAA=18.
!Surface area of apex.
DO WHILE (AOA .LE. 89*DTR)
        Y=0
        CN=0
        CA=0
        CNAREA=0
        CAAREA=0
        BNAREA=0
        BAAREA=0
        DEL=.005
        DO WHILE (Y .LE. 9.95)
                XLE=Y/TAN(2*GAM)+DEL
! For some odd reason, this only works when I add the +del to initilize the loop.
                XTE=L-Y*TAN(GAM)
                DO WHILE (XLE .LE. XTE)
                        DZDX=TAN(GAM)/SQRT(TAN(GAM)**2-1+2*(XLE/Y)*TAN(GAM))
                        CP=2*(SIN(AOA-ATAN(DZDX)))**2
                        CNAREA=CNAREA+DEL**2*CP*COS(ATAN(DZDX))
                        CAAREA=CAAREA+DEL**2*CP*SIN(ATAN(DZDX))
                        XLE=XLE+DEL
                END DO
! Takes care of canopy at Y.
```

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```
                    IF (Y .EQ. 0) THEN
                              KNAREA=CNAREA/DEL
                              KNAREA=.17*(KNAREA+.1*KNAREA)
                              CNAREA=CNAREA+KNAREA
                              KAAREA=CAAREA/DEL
                              KAAREA=.17*(KAAREA+.1*KAAREA)
                              CAAREA=CAAREA+KAAREA
                    END IF
! .1*KNAREA takes care of 10% greater pressure + .17 CNAREA takes care of the
! keel width.
                    Y=Y+DEL
          END DO
          CNAREA=CNAREA/SAC
          CAAREA=CAAREA/SAC
          Y=0.
          BLC=0.
          DO WHILE (Y .LE. 9.95)
                    THT=AOA-PI/2
                    F=0.0
                    S=(-.046*BLC+1.333)*DTR
                    DO WHILE (THT .LE. AOA+PI/2)
                              CP=2*COS(THT-AOA)**2*COS(DELT*COS(AOA))**2
                              F=F+DEL*S*CP
                              THT=THT+DTR
                    END DO
                    BNAREA=BNAREA+F*SIN(AOA)
                    BAAREA=BAAREA+F*COS(AOA)
! Takes care of tapered boom.
                    IF (Y .EQ. 0.) THEN
                              TEMP1=BNAREA
                              TEMP2=BAAREA
                              BNAREA=BNAREA/DEL
                              BAAREA=BAAREA/DEL
                              BNAREA=.077*(BNAREA+.1*BNAREA)+TEMP1
                              BAAREA=.077*(BAAREA+.1*BAAREA)+TEMP2
                    END IF
! .1*BNAREA takes care of 10% greater pressure & .077 takes care of the
! apex width.

                    Y=Y+DEL
                    BLC=BLC+.0087
          END DO
          BNAREA=BNAREA/(SAB)
          BAAREA=BAAREA/(SAB)
          CN=CNAREA+BNAREA
          CA=CAAREA+BAAREA
          CL=CN*COS(AOA)+CA*SIN(AOA)
          CD=CN*SIN(AOA)+CA*COS(AOA)
          PRINT*,'At angle of attack =', AOA/DTR, CNAREA, CAAREA, BNAREA, BAAREA
          PRINT*, 'CL=',CL,'CD=',CD,'L/D=',CL/CD
!         WRITE(1,*) AOA/DTR, CL ,CD ,CL/CD
          WRITE(1,*) CL , CD
          AOA=AOA+DTR
END DO
END
```

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